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Liberty
Technical
Document

Document Number: 135A-970-100
Revision: E

Liberty XL-2 Airplane Maintenance Manual

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Preface

Liberty Aerospace provides Instructions for Continued Airworthiness based on the design, testing, and certification of the XL-2 aircraft which Liberty Aerospace is the holder of the Type Certificate (TC) issued by the Federal Aviation Administration (FAA). Instructions in this Liberty manual, which include maintenance, repair limits, overhaul, and installation, are applicable only to the Liberty XL-2 model aircraft.

This manual, applicable service documents, and other related publications constitute the Instructions for Continued Airworthiness (ICA) prepared by Liberty Aerospace and reviewed by the FAA. Pursuant to Federal Aviation Regulation (FAR) § 43.13, each person performing maintenance, alteration, or preventive maintenance on the airframe, engine or accessories must use methods, techniques, and practices prescribed in the Instructions for Continued Airworthiness.

Except for FAR part 43.3 authorized owner maintenance, Liberty Aerospace has written the ICA for exclusive use by FAA (or equivalent authority) licensed mechanics or FAA (or equivalent authority) certified repair center employees working under the supervision of an FAA licensed mechanic. Information and instructions contained in this manual anticipate the user possesses and applies the knowledge, training, and experience commensurate with the requirements to meet the prerequisite FAA license and certification requirements. No other use is authorized.

Installation of parts deviating from Liberty Aerospace approved type-design criteria is not allowed. Liberty Aerospace accepts no liability for the suitability, durability, longevity, or safety of such parts installed on the XL-2 aircraft. Installation of parts deviating from type design must be performed using Instructions for Continued Airworthiness prepared by the part manufacturer and approved by the FAA for the subject installation. Do not use Liberty Aerospace ICA for such parts.

Service Documents may contain advance changes to the ICA. It is the responsibility of the organization/person maintaining or operating the aircraft to verify that current and complete information, including Service Bulletins, FAA Airworthiness Directives (AD), and publications are used.

To facilitate the use of current data, Liberty Aerospace provides information via the Internet on www.libertyaircraft.com. The information available includes a listing of the latest manual versions, service bulletins, FAA ADs, and other information applicable to the Instructions for Continued Airworthiness. This information is free of charge to owners of Liberty Aerospace aircraft by registering through Owner Support.

Table of Contents

PREFACE	III
TABLE OF CONTENTS	IV
SUPERSEDE NOTICE	V
LIST OF EFFECTIVE PAGES	V
HIGHLIGHT OF REVISIONS	VII
RECORD OF REVISION	X
LIST OF SERVICE BULLETINS	XI
SERVICE BULLETINS RELEASED AFTER PUBLICATION	XIII
TABLE OF CHAPTERS	XV

Supersede Notice

This manual supersedes and renders obsolete Liberty Aerospace Maintenance Manual 135A-970-006 revision A dated October 2008.

List of Effective Pages

Use this page(s) to determine current revision and effective date for each page in the Maintenance Manual. As revisions are issued this list of effective pages will be amended.

PAGE	REVISION	DATE	PAGE	REVISION	DATE
Preface i - xvi	E	07/14/11	12-01 thru 12-02	E	07/14/11
01-01 thru 01-20	E	07/14/11	12-03 thru 12-13	~	07/31/09
01-05 thru 01-06	~	07/31/09	12-13 thru 12-14	E	07/14/11
01-07 thru 01-18	E	07/14/11	12-15 thru 12-16	~	07/31/09
01-18 thru 01-20	~	07/31/09	20-01 thru 20-02	D	02/16/11
03-01 thru 03-10	E	07/14/11	20-03 thru 20-06	~	07/31/09
04-01 thru 04-2	D	02/16/11	20-07 thru 20-16	D	02/16/11
04-03 thru 04-6	~	07/31/09	20-17 thru 20-26	~	07/31/09
04-07 thru 04-20	D	02/16/11	21-01 thru 21-40	~	07/31/09
04-21 thru 04-22	~	07/31/09	23-01 thru 23-04	D	02/16/11
05-01 thru 05-02	D	02/16/11	23-05 thru 23-08	~	07/31/09
05-03 thru 05-08	~	07/31/09	23-09 thru 23-38	D	02/16/11
05-09 thru 05-34	D	02/16/11	23-39 thru 23-44	~	07/31/09
05-35 thru 05-42	~	07/31/09	24-01 thru 24-02	E	07/14/11
05-43 thru 05-52	C	06/21/10	24-03 thru 24-04	E	07/14/11
06-01 thru 06-08	~	07/31/09	24-05 thru 24-08	E	07/14/11
07-01 thru 07-02	E	07/14/11	24-09 thru 24-16	~	07/31/09
07-03 thru 07-04	~	07/31/09	24-17 thru 24-44	E	07/14/11
07-05 thru 07-14	E	07/14/11	24-45 thru 24-62	D	02/16/11
08-01 thru 08-16	~	07/31/09	25-02 thru 25-04	C	06/21/10
09-01 thru 09-12	~	07/31/09	25-05 thru 25-10	~	07/31/09
10-01 thru 10-12	~	07/31/09	25-11 thru 25-66	D	02/16/11
11-01 thru 11-2	E	07/14/11	27-01 thru 27-02	E	07/14/11
11-03 thru 11-12	~	07/31/09	27-03 thru 27-06	~	07/31/09
11-13 thru 11-34	E	07/14/11	27-07 thru 27-34	D	02/16/11
11-34 thru 11-38	~	07/31/09	27-35 thru 27-60	E	07/14/11
			27-61 thru 27-76	D	02/16/11
			27-77 thru 27-84	D	02/16/11
			27-84 thru 27-100	D	02/16/11
			27-101 thru 27-120	~	07/31/09

PAGE	REVISION	DATE	PAGE	REVISION	DATE
28-01 thru 28-02	E	07/14/11	70-01 thru 70-06	~	07/31/09
28-03 thru 28-17	B	01/30/10	71-01 thru 71-02	E	07/14/11
28-17 thru 28-36	E	07/14/11	71-03 thru 71-64	~	07/31/09
28-37 thru 28-56	~	07/31/09	71-65 thru 71-76	E	07/14/11
30-1 thru 30-20	C	06/21/10	71-77 thru 71-74	~	07/31/09
31-01 thru 31-54	~	07/31/09	72-01 thru 72-30	~	07/31/09
32-01 thru 32-02	E	07/14/11	73-01 thru 73-38	~	07/31/09
32-03 thru 32-18	~	07/31/09	74-01 thru 74-02	B	01/30/10
32-19 thru 32-78	E	07/14/11	74-03 thru 74-12	~	07/31/09
32-79 thru 32-88	~	07/31/09	74-13 thru 74-20	B	01/30/10
33-01 thru 33-02	C	01/30/10	75-01 thru 75-08	~	07/31/09
33-03 thru 33-06	~	07/31/09	76-01 thru 76-18	~	07/31/09
33-07 thru 33-22	B	01/30/10	77-01 thru 77-02	C	06/21/10
33-23 thru 33-40	C	06/21/10	77-03 thru 77-06	~	07/31/09
34-01 thru 34-02	B	01/30/10	77-07 thru 77-34	C	06/21/10
34-03 thru 34-52	~	07/31/09	78-01 thru 78-16	~	07/31/09
34-53 thru 34-78	B	01/30/10	79-01 thru 79-28	~	07/31/09
34-79 thru 34-116	~	07/31/09	80-01 thru 80-04	B	01/30/10
51-01 thru 51-74	~	07/31/09	80-05 thru 80-08	~	07/31/09
52-01 thru 52-30	~	07/31/09	80-09 thru 80-14	E	07/14/11
53-01 thru 53-04	B	01/30/10	91-01 thru 91-02	C	06/21/10
53-05 thru 53-08	~	07/31/09	91-03 thru 91-12	~	07/31/09
53-09 thru 53-22	E	07/14/11	91-13 thru 91-80	C	06/21/10
53-23 thru 53-24	~	07/31/09			
53-25 thru 53-30	B	01/30/10			
53-31 thru 53-32	~	07/31/09			
55-01 thru 55-02	D	02/16/11			
55-03 thru 55-04	B	01/30/10			
55-05 thru 55-06	~	07/31/09			
55-07 thru 55-28	D	02/16/11			
55-29 thru 55-30	~	01/30/10			
56-01 thru 56-04	B	01/30/10			
56-05 thru 56-10	~	07/31/09			
56-11 thru 56-36	E	07/14/11			
56-36 thru 56-56	~	07/31/09			
57-01 thru 57-34	D	02/16/11			
61-01 thru 61-12	~	07/31/09			

Highlight of Revisions

Revision	Revision Date	Chapter	Description
IR	07/31/2009	ALL	Initial release of the entire manual 135A-970-100.
A	10/28/2009	04	Incorporated approved firewall blanket life extension.
B	01/30/2010	00	Updated the chart to indicate the inclusion of SID-09-001. Stepped through all chapters and matched the table on page x of the Preface (Chapter 00). Added Highlight of Revisions table.
		05	Updated exhaust system inspection procedure.
		23	Updated antenna mounting hardware callout in compliance with current design.
		25	Updated ELT antenna remove/install procedures.
		27	Added drawing and procedure steps to clarify control yoke ground installation.
		28	Updated fuel boost pump remove/install and test procedure. Added procedure steps for boost pump control switch remove/install. Added fuel system operational check procedure steps. Added Fuel System Troubleshooting chart. Updated the flow of the procedures to match other chapter is the manual.
		33	Updated strobe power supply circuit breaker call out in compliance with current design and SID-09-001.
		34	Added note to compass swing procedure for identification of northern and southern hemisphere compass units.
		53	Added belly panel remove/install procedure steps.
		55	Added horizontal stabilizer inspection steps.
		56	Removed procedure steps duplicated elsewhere and directed reader to the steps.
		74	Added pointer to FADEC system checks procedure.
		75	Chapter removed as content is redundant to another chapter.
		77	Deleted note incorporating hardware call out in work step.
		80	Added cold start steps in accordance with TCM operations procedures.
		91	Drawings updated in accordance with current approved design changes.

Revision	Revision Date	Chapter	Description
C	06/21/2010	00	Updated the List of effected Pages to include new chapter 30.
		04	Added Life Limits for Fire Extinguisher. Re-formatted the tables for readability .
		05	Added inspection points for fire extinguisher. Added inspection points for bolts associated with the stabilizer. Added inspection points for the alternate air valve cabling. Clarified the 500 hour engine inspection.
			Added definitions to Hard Landing and Crash.
		30	All new chapter.
		33	Added procedures for the LED Wing Tip Light Fixtures.
		55	Add inspection information and inspection notes for stabilizer bolt length.
		77	Removed procedures for the Pitot/Static Blade and heater. Procedures were moved to Chapter 30.
		91	Updated Schematic Figure 91-18 to include the LED Wing Tip Light Fixture.
D	02/16/2011	00	Added information about incorporation of SSI-10-001.
		04	Added Life Limits for the exhaust.
			Added part numbers for the replacement exhaust. muffler to table 04-3.
			Added mandatory inspection for the aft weldment clevis pins for both port and starboard.
			Added additional mandatory inspection for the aft weldment clevis pins for both port and starboard on SN 0009, 0116, 0117, and 0119.
		05	Redefined the inspections for the batteries in the ELT and the ELT Remote Switch.
			Split the inspections requirements for the wing pins, wing box, and aileron quick disconnect assemblies to account for the 100-hour inspections that does not remove the wings and the annual inspection that does remove the wings.
			Breaking out inspections for the spar attachment lugs, pins, flap-spigot, and the locking mechanisms.
		20	Updated the procedures to torque castellated nuts.
		23	Added reference to Chapter 27 for the remove and installation of the push-to-talk switch/button.
		24	Added information on the F3 Alternator Fuse (SSI-10-001).
			Added a procedure to replace the F3 Fuse.
			Added information to clarify what instruments will continue to operate in the event of a primary battery failure.

Revision	Revision Date	Chapter	Description
		25	Correctly point to the battery limits for the ELT batteries to Chapter 05 (was Chapter 04).
		27	Updated the procedures to torque castellated nuts.
			Added procedures to the section for the yoke control to remove and install the push-to-talk switch/button.
		55	Updated the procedures to torque castellated nuts.
		57	Rewrote the entire chapter with a new layout of the sections.
			Added information in the removal and installation of the anti-chafing tape for the wing root fairing.
E	07/14/2011	01	Updated Information in the Revision Tracking paragraph.
			Updated table with new AFM part numbers.
			Corrected Figure 01-1.
			Rewrote Section 01-40.
		03	Rewrote entire chapter.
		07	New images, graphics and photos.
			Updated the procedure for lifting airplane.
		11	Added the part numbers for the Japanese placards.
			Figure 11-23 – Updated figure.
			Figure 11-25 – Corrected the S/N for factory gross weight.
		12	Added wing pins to Table 12-3.
		24	Figure 24-4 Redrew figure.
			Figure 24-12 Corrected battery callout in figure.
			Page 38 – Reworded paragraph
		27	Pages 52 and 56 corrected battery callout for secondary battery.
		28	Correct relay callout on page 34
		32	Added procedures to cover new parking brake.
			Updated graphics.
		53	Corrected battery callout for secondary battery.
		56	Page 16 – removed broken link.
			Page 34 – New step added for correct callout of post install inspections.
		71	Added new procedures for alternate air valve.
		80	Pages 12 and 14 Corrected callouts for the primary and secondary batteries.

Record of Revision

Use this checklist to record and control of all of the revisions you put in this Airplane Maintenance Manual (AMM). Put the affected pages of the revision into the AMM as soon as you get them. Remove and destroy the pages which are superseded. Complete the table below when you have put the revision in the AMM.

Record of Revisions							
Revision	**Date Issued**	**Inserted On**	**Inserted By**	**Revision**	**Date Issued**	**Inserted On**	**Inserted By**
A	10/28/2009	10/28/2009	LAI				
B	01/30/2010	01/30/2010	LAI				
C	06/21/2010	06/21/2010	LAI				
D	02/16/2011	02/16/2011	LAI				
E	07/14/2011	07/14/2011	LAI				

List of Service Bulletins

Service documents and technical information are incorporated in this edition of the Liberty Aerospace Maintenance Manual 135A-970-100 and are listed in the chart below. Information relevant to the XL-2 model aircraft from these service documents has been incorporated in this manual. The full content of all Liberty Aerospace service documents is available on www.libertyaircraft.com , support section.

Service Documents Incorporated in this Manual			
Service Bulletin	**Subject**	**Section Title**	**Affected Chapter**
SSI-10-001	Alternator Fuse F3	Electric Power	Chapter 24
SID-09-001	Strobe Power	Lights Wiring Diagrams and Schematics	Chapter 33 Chapter 91
AD2009-08-05	Exhaust Muffler Cracks	Exhaust System Troubleshooting	Chapter 78 Chapter 05
CSB-08-003	Aileron pushrod	Flight Controls Inspections	Chapter 27 Chapter 05
CSB-08-002	Aileron pushrod	Flight Controls Inspections	Chapter 27 Chapter 05
RKI-SIL-08-001	Gross weight increase compliance	Fuselage Wing Navigation & Pitot static Flight Controls Leveling & Weighing	Chapter 53 Chapter 57 Chapter 34 Chapter 27 Chapter 08
SB-08003, Rev B	Rudder Pushrod clearance	Flight Controls	Chapter 27
SB-08-002	Toe brake rudder clearance adjustment	Landing Gear	Chapter 32
SB-08-001	Possible nose wheel shimmy	Landing Gear	Chapter 32
CSB-08-001	Fuel tank vibration	Fuel System Inspection	Chapter 28 Chapter 05
SIL-08-002	Footstep placard installation	Placards and Markings	Chapter 11
CSB-07-002 REV B	Aileron Skeleton Inspection and Installation Clearance	Inspection	Chapter 05
RKI-CSB-07-003 REV B	Replacement of Aileron Ribs and Gusset reinforcement	Flight Controls	Chapter 27
CSB-07-002 REV C	Flap rib 2 & wing rib ‘1’ aft inspection	Inspection	Chapter 05
RKI-CSB-07-002 REV C	Inboard flap rib reinforcement	Flight Controls	Chapter 27
SIL-O7-012 *UPDATED REV	Wheel fairing installation	Landing Gear	Chapter 32

Service Documents Incorporated in this Manual			
Service Bulletin	**Subject**	**Section Title**	**Affected Chapter**
SIL-07-011 *UPDATED REV	Nose gear wheel bearing jam nut	Landing Gear	Chapter 32
CSB-07-001 REV C	Rear spar aileron closeout	Wing Inspection	Chapter 57 Chapter 05
RKI-CSB-07-001, Rev B	Rear spar aileron closeout & mass balance enclosure flange	Wing	Chapter 57
SIL-07-008	EDI-200 revision upgrade	Indicators & Recording Equipment	Chapter 31
SIL-07-003 REV B	Footstep installation	Fuselage	Chapter 53
SIL-07-006	Nose landing gear leg inspection	Inspection Landing Gear	Chapter 05 Chapter 32
SIL-07-002 REV C	Mt propeller installation	Propeller	Chapter 61
SIL-06-008	Increase life limit of the Liberty XL-2 from 3,333 hours to 5,000 hours.	Airworthiness Limitations	Chapter 04
SIL-06-007	Cabin Heat and Defog System	Environmental Systems	Chapter 21
SIL-06-002	IOF-240-B4B engine conversion	Power Plant	Chapter 71
SIL-06-006	TCM EDI-200 light weight FADEC data recorder installation	Indicators & Recording Equipment	Chapter 31
SIL-06-005	Increase life limit of the Liberty XL-2 from 1,666 hours to 3,333 hours.	Airworthiness Limitations	Chapter 04
SIL-06-001	Reports from the field that cracks are being found in certain P/N 653924, Teledyne Continental Motors (TCM), Revision "F" or earlier.	Inspections	Chapter 05
SIL-06-004	Increase life limit of the Liberty XL-2.	Airworthiness Limitations	Chapter 04
SIL-06-003	Aircraft Weight and Balance Check	Leveling and Weighing	Chapter 08
SIL-06-002	FADEC system health status annunciator indications	Engine Indicating	Chapter 77
SB-06-001	135A-910-105 rev E, VM1000FX electronic engine display – Software upgrade	Engine Indicating	Chapter 77
SB221-006	Potential Cracks in XL-2 Acrylic Windshield.	Inspection	Chapter 05

Service Documents Incorporated in this Manual			
Service Bulletin	**Subject**	**Section Title**	**Affected Chapter**
CSB-06-007	Routing of electrical wiring from fuse holder to emergency battery terminal	Inspection Electrical Power	Chapter 05 Chapter 24
CSB-06-002	Potential fatigue cracks of the propeller extension Liberty PN: 4ARP.	Propeller	Chapter 61
CSB-06-001	135A-10-145 rev C control stick boot installation	Flight Controls	Chapter 27

Service Bulletins Released After Publication

Liberty Aerospace strives to provide clear, concise, and accurate information and instructions based on the best available engineering data at the time of publication. Ongoing process improvements at Liberty may change specification or procedure after manual release. Service documents expedite customer notification and serve as the prevailing instruction over conflicting information until the new information is incorporated in the manual text. As bulletins are received, note the bulletin's number, title and applicable section affected by the bulletin in the blank cells provided below. Insert a copy of the Service Bulletin behind the last page of this section.

The following bulletins, released after this manual, affect and supplement the procedures herein. When performing procedures affected by the bulletins, review the bulletin content prior to commencing the task to ensure compliance with the most current information. Review of service documents is required prior to start of any procedure in this manual.

Service Documents Published After this Manual			
Service Bulletin	**Subject**	**Section Title**	**Affected Chapter**

Table of Chapters

Chapters	Description	Page
01	Introduction	01-01
03	General Description	03-01
04	Airworthiness Limitations	04-01
05	Time Limits/Maintenance Checks/Inspection Intervals	05-01
06	Dimensions and Areas	06-01
07	Lifting and Jacking	07-01
08	Leveling and Weighing	08-01
09	Towing and Taxiing	09-01
10	Parking and Mooring	10-01
11	Markings and Placards	11-01
12	Servicing	12-01
20	Standard Practices Airframe	20-01
21	Environmental Systems	21-01
23	Communications	23-01
24	Electrical Power	24-01
25	Equipment and Furnishings	25-01
27	Flight Controls	27-01
28	Fuel System	28-01
30	Ice and Rain Protection	30-01
31	Indicators and Recording Systems	31-01
32	Landing Gear	32-01
33	Lights	33-01
34	Navigation and Pitot/Static	34-01
51	Standard Practices – Structures	51-01
52	Doors	52-01
53	Fuselage	53-01
55	Stabilizers	55-01
56	Windows	56-01
57	Wings	57-01
61	Propellers	61-01
70	Standard Practices – Engine	70-01
71	Power Plant	71-01
72	Engine	72-01
73	Engine Fuel Control	73-01
74	Engine Ignition	74-01
75	Air Induction	75-01
76	Engine Controls	76-01
77	Engine Indicating	77-01
78	Engine Exhaust	78-01
79	Engine Oil	79-01
80	Starting	80-01
91	Wiring Diagrams	91-01

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CHAPTER 01

INTRODUCTION

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Table of Contents

SECTION 01-00	GENERAL	5
SECTION 00-01	THE MAINTENANCE MANUAL STYLE	5
SECTION 01-10	REVISION SYSTEMS	7
SECTION 10-01	REFORMATTED CHAPTERS REVISION TRACKING	7
SECTION 01-20	PUBLICATIONS	9
SECTION 20-01	SERVICE DOCUMENTS	9
SECTION 20-02	RELATED PUBLICATIONS	10
SECTION 01-30	CHAPTER FORMAT AND LAYOUT	13
SECTION 30-01	CHANGES TO TEXT	13
SECTION 30-02	BLANK PAGES	13
SECTION 30-03	NOTES, CAUTIONS AND WARNINGS	13
SECTION 30-04	SECTION HEADINGS AND SECTION TEXT	14
SECTION 30-05	TABLE AND FIGURE CAPTIONS	14
SECTION 30-06	HEADERS AND FOOTERS	14
SECTION 30-07	PROCEDURES	15
SECTION 01-40	MANUAL ARRANGEMENTS	17
SECTION 01-50	MAIN CHAPTER GROUPS	19

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Section 01-00 General

Liberty Aerospace, Inc. is releasing this manual as a brand new manual. The purpose of the Maintenance Manual is to furnish maintenance personnel with all data required for normal maintenance of the Liberty XL-2 airplane. It comprises detailed system and subsystem descriptions, troubleshooting tables, component removal, installation procedures, and detailed maintenance procedures.

Unless otherwise specified, This Maintenance Manual contains only removal and installation instructions for specific components for example: alternator, fuel system components, instruments, etc. Obtain the detailed maintenance and service instructions for individual components from the applicable component manufacturer's maintenance manual, website, or service department. Contact Liberty Aircraft, Customer Support to request special assistance.

The performance of all maintenance work including inspections, routine maintenance, and repairs will be in accordance with the procedures set forth in this Maintenance Manual.

IT IS THE RESPONSIBILITY OF THE OWNER AND MAINTENANCE TECHNICIAN TO CHECK THAT OPERATING PROCEDURES, INSPECTION, REPAIR OR MAINTENANCE PRACTICES, ARE CORRECTLY FOLLOWED, USING THE MOST UP-TO-DATE FAA APPROVED VERSIONS OF THE COMPONENT MANUFACTURERS MANUALS NOTED ABOVE.

Section 00-01 The Maintenance Manual Style

This new release of the maintenance manual contains a new style and a new look and feel for the Liberty Aerospace, Inc. Maintenance Manual. The new style should make the manual more user-friendly.

One of the most obvious changes came to the header and footer on each page.

The header contains the chapter title and Liberty Aerospace, Inc. logo. The footer contains the current section number (for example 01-00), the part number of the maintenance manual, the revision of that particular section (for example Rev. ~), the page number and total pages. The tilde symbol, ~, indicates an initial release of the section. A revision provided on the manual cover page indicates the latest revision package incorporated. Details of the other changes to the maintenance manual and the chapters are in the remaining sections of this chapter.

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Section 01-10 Revision systems

This section details the methods used to track revisions and changes to the chapters in this manual. In the initial release of the manual and the chapters, revisions were tracked in the front matter sections of the manual. With the changes to the format of this manual, the tracking of changes and revisions moved to the end of the front matter behind the Table of Contents and Table of Chapters.

Section 10-01 Reformatted Chapters Revision Tracking

Revisions to this manual are applied on a section by section basis. The Footer of each page of a section provides a revision letter representing the latest revision package applied to that section. On initial release each page will show as Revision ~. Thereafter, each section shows the latest revision by letter that has been applied to that section. Maintenance manual revisions are released as packages of changed sections that can span a number of chapters. Each of these packages and the sections in them are marked with the same revision package letter. Revision packages are labeled with letters that are sequential (A, B, C and so on). A new maintenance manual cover page marked with the latest revision package letter is issued with each release. Each book owner must replace pages or add pages and the cover page provided by the revision package and log incorporation in the Record of Revision provided in the manual Preface. To assist in this process an amended List of Effective Pages table is provided with each revision package and must be incorporated in the manual Preface.

When amended the List of Effective Pages table provides a breakdown of which pages changed with each Revision package. It is important to review this list any time work is performed to assure the latest information is being used. Figure 01-1 below provides an example of a List of Effective Pages amendment.

PAGE	REVISION	DATE	PAGE	REVISION	DATE
Preface i - xiv	C	06/21/10	27-01 thru 27-02	B	01/30/10
01-01 thru 01-20	~	07/31/09	27-03 thru 27-76	~	07/31/09
03-01 thru 03-10	~	07/31/09	27-77 thru 27-84	B	01/30/10
04-01 thru 04-2	C	06/21/10	27-84 thru 27-120	~	07/31/09
04-03 thru 04-6	~	07/31/09	28-01 thru 28-36	B	01/30/10
04-07 thru 04-20	C	06/21/10	28-37 thru 28-56	~	07/31/09
04-21 thru 04-22	~	07/31/09	30-1 thru 30-20	C	06/21/10
05-01 thru 05-02	C	[illegible]	[illegible]	~	07/31/09
05-03 thru 05-08	~	[illegible]	[illegible]	~	07/31/09
05-09 thru 05-34	C	[illegible]	[illegible]	C	01/30/10
05-35 thru 05-42	~	07/31/09	33-03 thru 33-06	~	07/31/09
05-43 thru 05-52	C	[illegible]	[illegible]	B	01/30/10
06-01 thru 06-08	[illegible]	[illegible]	[illegible]	[illegible]	06/21/10
07-01 thru 07-14	~	07/31/09	34-01 thru 34-02	B	01/30/10
08-01 thru 08-16	~	07/31/09	34-03 thru 34-52	~	07/31/09
09-01 thru 09-12	~	07/31/09	34-53 thru 34-78	B	01/30/10
10-01 thru 10-12	~	07/31/09	34-79 thru 34-116	~	07/31/09
11-01 thru 11-38	~	07/31/09	51-01 thru 51-74	~	07/31/09
12-01 thru 12-16	~	07/31/09	52-01 thru 52-30	~	07/31/09

This graphic is provide
to show an example of
of the List of Effective Pages.
See the Preface Section for
the current List of Effective Pages.

Figure 01-1 – List of Effective Pages Example

Changes that are on any particular page will be indicated by a revision bar along the outside edge, such as those that are evident on this page.

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Section 01-20 Publications

This section describes the different documents that are published.

Section 20-01 Service Documents

Liberty Aerospace, Inc. has adopted a service document format which will provide the user with an explanation of the compliance criticality of the document. This format was developed by the General Aviation Manufacturers Association (GAMA) to standardize service documents used by aviation manufacturers. The documents can be presented in six categories as defined below. Each revision to the service document will be noted in the information block located at the bottom of each page. This information will include the date of original issuance, revision date, page number and revision number. The most recent revisions to the document will be identified by a solid bold vertical line on the outside edge of the page, adjacent to the revised sentence, paragraph, drawing, etc. As applicable, each service document's content is incorporated into the maintenance manual in the appropriate chapter and section. Release packages incorporating service documents will include an amended List of Service Documents. This list is incorporated in the manual Preface by the book owner as part of the update process.

SERVICE DOCUMENT CATEGORY DEFINITION

CATEGORY 1: "MANDATORY SERVICE BULLETIN" (MSB)

Service documents, relating to known or suspected hazards to safety, that have been incorporated in whole or in part in an Airworthiness Directive (AD) issued by the FAA or have been issued, at the direction of FAA, by the manufacturer, in order to allow compliance with an already issued AD (or an equivalent issued by another country's airworthiness authority).

CATEGORY 2: "CRITICAL SERVICE BULLETIN" (CSB)

Service documents (not included in Category 1) that have been determined by the product manufacturer to constitute a threat to continued safe operation of an aircraft or to persons or property on the ground unless some specific action (inspection, repair, replacement, etc.) is taken by the product owner or operator. Documents in this category are candidates for incorporation in an Airworthiness Directive issued by the FAA (but may not be).

CATEGORY 3: "SERVICE BULLETIN" (SB)

Service documents (not included in Categories 1 and 2) considered by the product manufacturer to constitute a substantial improvement to the inherent safety of an aircraft or component of an aircraft. This "Service Bulletin" category also includes the most recent updates of instructions for continued airworthiness.

CATEGORY 4: "SERVICE INFORMATION DIRECTIVE" (SID)

Serviced documents (not included in Categories 1, 2 or 3) that have been determined by the manufacturer to be of value to an owner/operator in the use of a product by enhancing safety, maintenance or economy.

CATEGORY 5: "SERVICE INFORMATION LETTER" (SIL)

This category includes all information (not included in Category 1-4) that may be of use to the owner, operator or maintainer of the aircraft

CATEGORY 6: "SPECIAL SERVICE INSTRUCTION" (SSI)

This category is used to address an issue on specific aircraft serial numbers. Liberty Aerospace will distribute SSI notification directly to the owners of the affected aircrafts. SSI's will not be included in the general service bulletin set but will be made available through Liberty Customer Service to owners of the affected aircrafts only. An SSI may contain updates to the Instructions for Continued Airworthiness applicable only to the listed aircrafts.

CATEGORY 7: "REWORK KIT INSTRUCTION" (RKI)

Detailed written technical procedures stating how to conduct and complete a repair required by a Service Document. A Rework Kit Instruction may also be part of optional upgrades and installation.

Section 20-02 Related Publications

The chart below lists related publications, source and accessibility relevant to maintaining the Liberty XL-2 model aircraft.

WARNING

USE ONLY THE LATEST REVISION OF ALL PUBLICATIONS. USING INFORMATION THAT HAS BEEN SUPERSEDED, JEOPARDIZES AIRCRAFT AIRWORTHINESS.

Publication Name	Publication Number	Supplied With Aircraft	Internet via Liberty Aircraft web site	Order From Liberty Aircraft Inc.	Available From Manufacturer
Liberty Aircraft					
XL-2 Maintenance Manual	135A-970-100	YES	YES	YES	N/A
XL-2 Airplane Flight Manual – Standard	135A-970-300	YES	NO	YES	N/A
XL-2 Airplane Flight Manual – Gross Weight	135A-970-200	YES	NO	YES	N/A
XL-2 Airplane Flight Manual – EASA	135A-970-500	YES	NO	YES	N/A
XL-2 Airplane Flight Manual – EASA Gross Weight	135A-970-600	YES	NO	YES	N/A
Engine Documents					
IOF-240-B Engine Installation and Operation Manual	OI-22	YES	NO	NO	TCM
IOF-240-B Overhaul Manual	OH-22	YES	NO	NO	TCM

Publication Name	Publication Number	Supplied With Aircraft	Internet via Liberty Aircraft web site	Order From Liberty Aircraft Inc.	Available From Manufacturer
IOF-240-B Maintenance Manual	M-22	YES	NO	NO	TCM
Accessory Documents[1]					
GNS 430 User Guide	190-00140-00	YES	N0	NO	GARMIN
GNS 430 Installation Manual	190-00140-05	YES	NO	NO	GARMIN
GNS 530 User Guide	190-00181-00	YES	NO	NO	GARMIN
GNS 530 Installation Manual	190-00181-02	YES	NO	NO	GARMIN
GNS 430W User Guide	190-00356-00	YES	NO	NO	GARMIN
GNS 430W Installation Manual	190-00356-02	YES	NO	NO	GARMIN
GNS 430W ICA Manual	190-00356-65	YES	NO	NO	GARMIN
GNS 530W User Guide	190-00357-00	YES	NO	NO	GARMIN
GNS 530W Installation Manual	190-00357-02	YES	NO	NO	GARMIN
GNS 530W ICA Manual	190-00357-65	YES	NO	NO	GARMIN
SL30 User Guide	190-00846-00	YES	NO	NO	GARMIN
SL30 Installation Manual	560-0404-03a	YES	NO	NO	GARMIN
SL40 User Guide	190-06488-00	YES	NO	NO	GARMIN
SL40 Installation Manual	560-0956-03	YES	NO	NO	GARMIN
GMA 340 Installation Manual	190-00149-01	YES	NO	NO	GARMIN
GMA 340 User Guide	190-00149-10	YES	NO	NO	GARMIN
SAE-35 Installation Manual	305186-00	YES	NO	NO	SANDIA
M803 Installation Manual	M803	YES	NO	NO	DAVTRON
GTX 327 User Guide	190-00187-00	YES	NO	NO	GARMIN
GTX 327 Installation Manual	190-00187-02	YES	NO	NO	GARMIN
Gi-106A Installation Manual	190-00180-00	YES	NO	NO	GARMIN
MD200-306 Installation Manual	8017972	YES	NO	NO	GARMIN
W69EK7-63G Propeller Operation & Maintenance	WOOD-CF-REV-A	YES	NO	NO	Sensenich Wood Propeller
MT175R127-2Ca Propeller – Operation & Installation	E-112	YES	NO	NO	MT-Wood-Composite Propeller
RG-25 Maintenance Manual	5-0142	YES	NO	NO	Concord Battery

Table 01-1 Related Publications

1 Documents supplied with aircraft as equipped.

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Section 01-30 Chapter Format and Layout

Along with the changes to the style of the chapters, there were changes to the format and layout of pages. This section describes the different changes to Changed Text, Blank Pages, Notes, Cautions, and Warnings Callouts, Section Headings and Section Text, Headers and Footers, Figure and Table Captions, and Procedures.

Section 30-01 Changes to Text

Because the new look and feel made changes that affected the entire chapter, the first release of the chapters will not indicate the changes by the use of a revision bars. However, subsequent releases of the reformatted chapters will indicate any changes (other than grammatical or spelling) with revision bars located in the outside margin of the page (left margin on even pages and right margin on odd pages).

Section 30-02 Blank Pages

Identification of a blank page is by the legend, "**PAGE LEFT INTENTIONALLY BLANK**", in the center of each page.

Section 30-03 Notes, Cautions and Warnings

In the reformatted release of the maintenance manual, Notes, Cautions, and Warnings are depicted with the appropriate icon, with the text on the next line. Below are examples of these. The appearance of the message is such to highlight the importance of the message.

AN OPERATING PROCEDURE, INSPECTION, REPAIR, OR MAINTENANCE PRACTICE, WHICH IF NOT CORRECTLY FOLLOWED, COULD RESULT IN PERSONAL INJURY OR LOSS OF LIFE.

An operating procedure, inspection, repair or maintenance practice, which if not strictly observed could result in damage or destruction of equipment.

An operating procedure, inspection, repair, or maintenance condition, etc, which is deemed essential to highlight.

Section 30-04 Section Headings and Section Text

The section headings were given a specific font size and weight (Bold and Italics), and were left justified to the margin. The primary heading will have the chapter number followed by the section number. The primary heading will start on an odd page. A subsection heading will have the section number followed by the subsection number.

Section information is also in the footer of each page. The footer contains the chapter and section number on the outside margin. See the section on footer information.

The text within a section is fully justified and indented by 0.4 inches. This will aid the reader to scan quickly the document for headings and for Notes, Cautions, and Warnings, which run from margin to margin.

Section 30-05 Table and Figure Captions

Figures will have the designation that includes the chapter number, a serial number that restarts with each chapter and a description of what is in the figure. This designation appears below the figure. For example:

Figure 01-1 This is an example of a figure caption

Likewise, tables will have a similar formatted callout and caption and will appear below the table. For example:

Table 01-1 This is an example of a table caption

Section 30-06 Headers and Footers

The headers and footers have changed to aid the reader in identifying the page and the information on that page. This change allows the document to be more scanning friendly to the reader.

The headers contain the logo for Liberty Aerospace, Inc. along the inside edge of each page. The outside edge identifies the chapter by title (for example *INTRODUCTION*) and the subject airplane (for example XL-2 Airplane).

The footer has additional information for the reader. On the inside margin, is the top-level part number for the maintenance manual, P/N 135A-970-100. Below the part number is the revision applied to this page.

On the outside edge, the footer has the chapter number and the section of the chapter covered on that page. The subsection does not appear, as a single page can contain more than a single subsection. The page number and the page count are presented directly below chapter and section number.

Looking at Figure 01-1, the page that has the upper footer is part of Chapter 01, Section 20 and this is page 10 of 16. Directly below the manual part number is the latest revision applied to this page which in this case is “~” indicating initial release. The page that has the lower footer is part of Chapter 01, Section 30, and this is page 11 of 16. Below the manual part number, is the revision of the page, in this case revision A is the latest revision applied to this page.

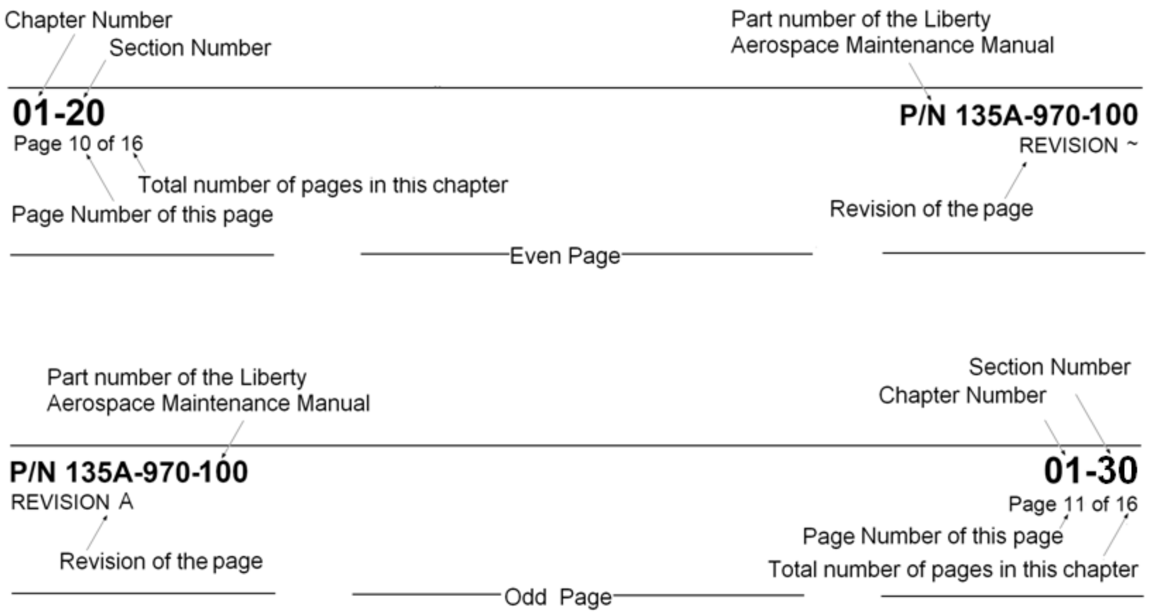

Figure 01-1 Examples of the Maintenance Manual Footer

Section 30-07 Procedures

Procedures will have an unnumbered heading that is in the center on the page. Each procedure will begin on a new page to distinguish it from a preceding procedure. Long or involved procedures will come as a series of smaller procedures with a table showing a list of the procedures that are involved. The end of a procedure will have a sentence stating the procedure has been completed successfully.

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Section 01-40 Manual Arrangements

The overall arrangement of this Maintenance Manual is in accordance with the Air Transport Association of America (ATA) Specification number 100. In addition, the manual complies with the General Aviation Manufacturers Association (GAMA) Specification number 2 issue in January 1978 - *Specification for Manufacturers Maintenance Data*.

In the general Table of Contents, located in the front of the manual, is the title of each chapter. Each chapter covers a specific main system; sub-sections describe relevant subsystems.

Changes to the format, as noted in the above sections of this chapter, are such to comply with most of provisions of GAMA Specification number 9 (Version 2.0), dated March 1999 – *Electronic Publication Standard*.

In addition, the preparation of this manual starts the process of moving the publications for the airplane towards electronic documentation. Within the pages of this manual are software links such that when a reader is viewing the manual with a computer, the reader can quickly move to the information that is needed. When viewing this manual on a computer, the title of each chapter in the general Table of Contents is a link to the first page of that chapter. From there, the reader can go to the chapter table of contents (always starts on page 3 of the chapter) and find links to the sections and subsections within that chapter.

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Section 01-50 Main Chapter Groups

The maintenance manual combines the chapters in to five groups. This grouping brings together chapters that cover similar information about the airplane. Shown in Table 01-2 are the five groups, and the chapters that are in that group.

Table 01-2 Group Chapters

Manual Group	Chapters in Group
General	01 – 12
Equipment	20 – 34
Airframe	51 – 57
Propeller	61
Power Plant (Engine)	71 – 91

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CHAPTER 03

GENERAL DESCRIPTION

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Table of Contents

SECTION 03-00	GENERAL	5
SECTION 00-01	DESCRIPTION	5
SECTION 03-01	VENDER DOCUMENTATION	7
SECTION 03-02	AIRCRAFT MEASUREMENTS	9
SECTION 02-01	REFERENCE PLANES	9
SECTION 02-02	MEASUREMENTS FROM THE BUTT LINE	10
SECTION 02-03	MEASUREMENTS FROM THE WATERLINE AND STATION	11
SECTION 03-03	AIRCRAFT ACCESS PANELS AND COWLINGS	13

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Section 03-00 General

The Liberty XL-2 airplane is manufactured by Liberty Aerospace, Inc. and is approved in the normal airworthiness category. The company address is listed below.

Liberty Aerospace, Inc.
100 Aerospace Drive
Melbourne, Florida, USA, 32901
Phone: 321-752-0332

Section 00-01 Description

The Liberty XL-2 airplane is a low-wing two-place aircraft with tricycle landing gear. Its central structural element is a welded steel tubing frame or "chassis" which has the following components attached:

- Engine and Propeller
- Wings (conventional aluminum construction)
- Main Landing Gear (aluminum construction)
- Nose Landing Gear (steel construction)
- Fuselage (composite construction)
- AFT Fuselage with Vertical Fin (composite construction)
- Rudder and Stabilator (aluminum construction)
- Center Fuselage including crew/passenger cabin, fuel tank, and baggage compartment
- Engine Cowlings

The wings are riveted aluminum semi-monocoque structures which are secured to the chassis by means of heavy steel pins in attach fittings. With the wing or wings removed, the fuselage remains supported by the landing gear. The aluminum horizontal tail is a stabilator type in which the entire unit moves to change pitch on the aircraft. The only movable surfaces on the horizontal tail are the anti-servo tabs that have both pitch trim and anti-servo functions. Integrated into the fuselage is the composite vertical fin; the rudder is an aluminum structure. Ailerons and stabilators operate by pushrods from cockpit control sticks.

The rudder also operates by pushrods attached to adjustable cockpit rudder pedal assemblies. Aluminum trailing edge flaps operate electrically.

The power for the Liberty XL-2 is from an air-cooled Teledyne Continental Motors IOF-240B four cylinder, horizontally opposed, fuel injected engine. The rating of the engine is 125 continuous SAE horsepower at 2800 RPM. The engine is equipped with a Full Authority Digital Engine Control (FADEC) system to control ignition timing and fuel mixture without pilot intervention.

Fuel for the engine is from a single 28-gallon (106-liter) usable capacity aluminum tank. The tank is installed in the fuselage between the pilot/passenger seatbacks and the baggage compartment. The preferred propeller is the MT Propeller, which is a two-blade, fixed pitch, wood composite propeller with spinner assembly (including spacer, and supporting hardware) is model MT175R127-2Ca and manufactured by MT-Propeller Entwicklung GmbH. The permissible propeller is the Sensenich propeller, which is a fixed pitch, constructed of wood and fiberglass, part number: W69EK7 63G.

Section 03-01 Vender Documentation

This manual may refer to the following vendors, and their documentation.

Engine	IOF-240 B Teledyne Continental Motors 2039 Broad Street, Bldg. # 96 Mobile, AL. 36601 Tel: 251-438-8291 Fax: 251-432-7352 Website#: http://www.tcmlink.com
Propeller	MT175R127-2Ca MT-Propeller Entwicklung GmbH Flugplatzstr.1 D-04348 Atting, Germany Tel: +49-(0)9429-9409-0 Fax: +49-(0)9429-8432 Website: http://www.mt-propeller.com
Propeller	W69EK7-63G Sensenich Wood Propeller 2008 Wood Court Plant City. FL 33563 Tel: 813-752-3711 Fax: 813-752-2818 Website http://sensenichprop.com
Main Wheel Brakes	Parker Hannifin, Aircraft Wheel & Brakes 1160 Center Road Avon, OH. 44011-0158 Tel: 440-937-6211 Fax: 440-937-6416 Website: http://www.parker.com
Navigation/Communication Eq.	Garmin International Inc. 1200 East 151st Street Olathe, KS. 66062 Tel: 913-397-8200 Fax: 913-397-8282 Website: http://www.garmin.com

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Section 03-02 Aircraft Measurements

This section contains the measurements of the Liberty XL2 airplane.

Section 02-01 Reference Planes

As a standard, the measurements are from three reference planes. These planes are the Waterline, Station, and the Butt Line, also known as the centerline. Figure 03-1 shows the location of the three planes. Two of these planes (Waterline and Station) can be hard to define and/or impossible to measure from.

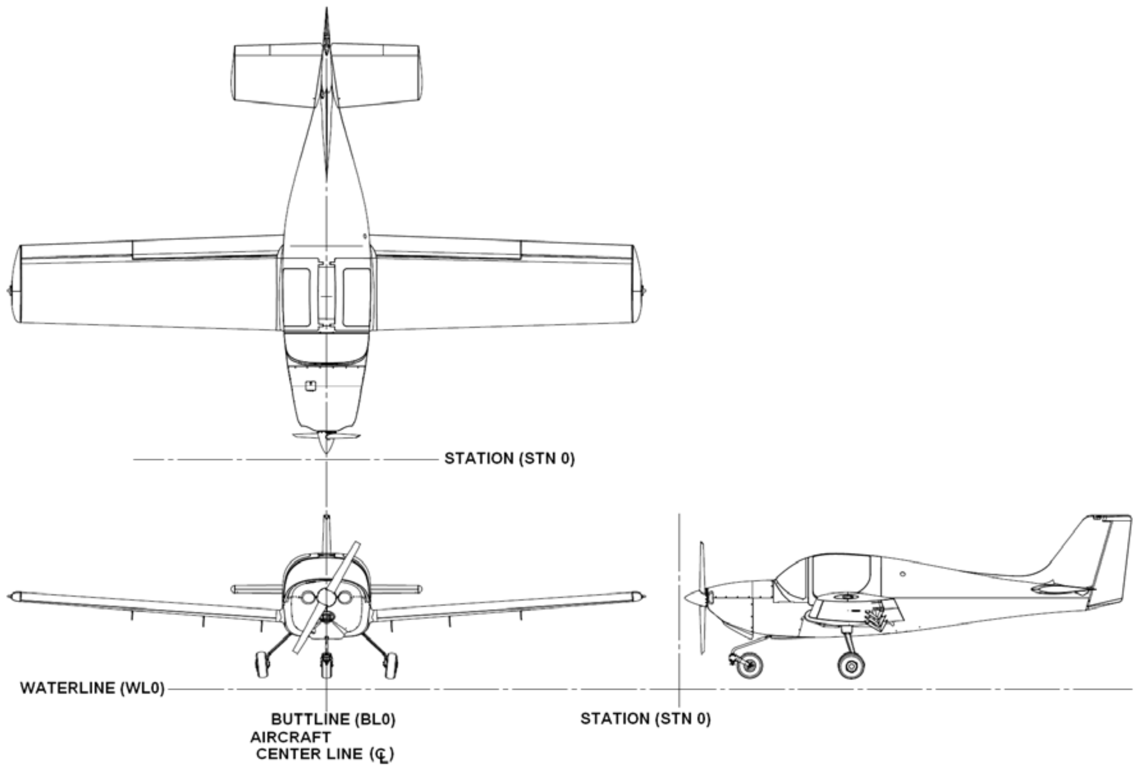

Figure 03-1 Location of the Waterline and Station Planes

There are various reference points on the airplane that may be used to locate other points on the airplane from either the Waterline or Station. Figure 03-2 shows the location and description of two of these reference points.

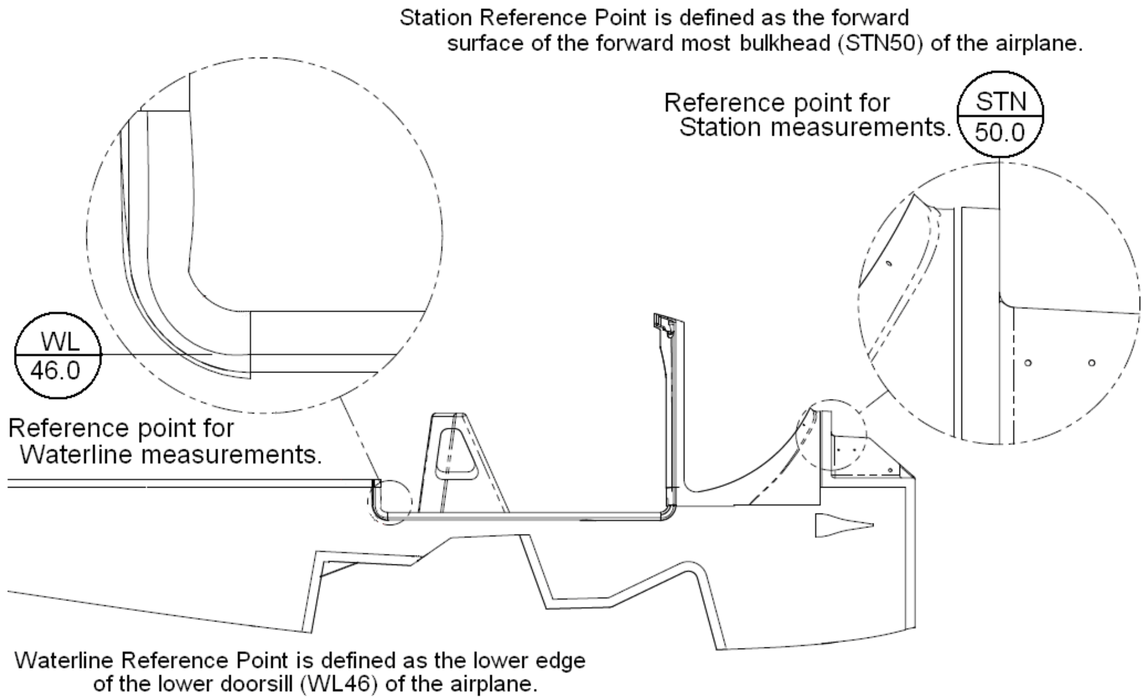

Figure 03-2 Waterline and Station Reference Points

Section 02-02 Measurements from the Butt Line

Measurements from the Butt Line are measurements port or starboard from the center of the airplane. See Figure 03-3 for various measurements that are left and or right of the butt line. This figure also has that span from the left side of the airplane to the right side.

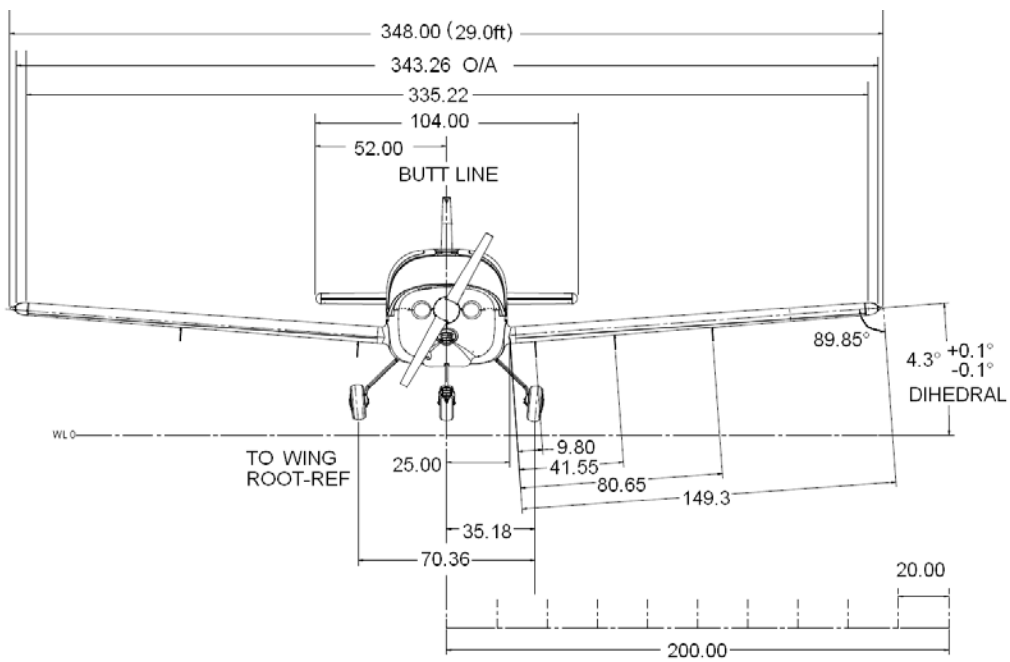

Figure 03-3 Measurements from the Butt Line

Section 02-03 Measurements from the Waterline and Station

Measurements for Waterline and Station are from an imaginary plane below the landing gear and forward of the propeller. That is the reason that on the airplane there are points defined as the Waterline and Station reference points. See Section 02-01 Reference Planes in this chapter. Figure 03-4 shows various Waterline and Station points on the airplane. These points are location of critical items (bulkheads, control surface axis, weight and balance, etc.) The numbers in the figure are aft of the Station 0 plane or above the Waterline 0 plane.

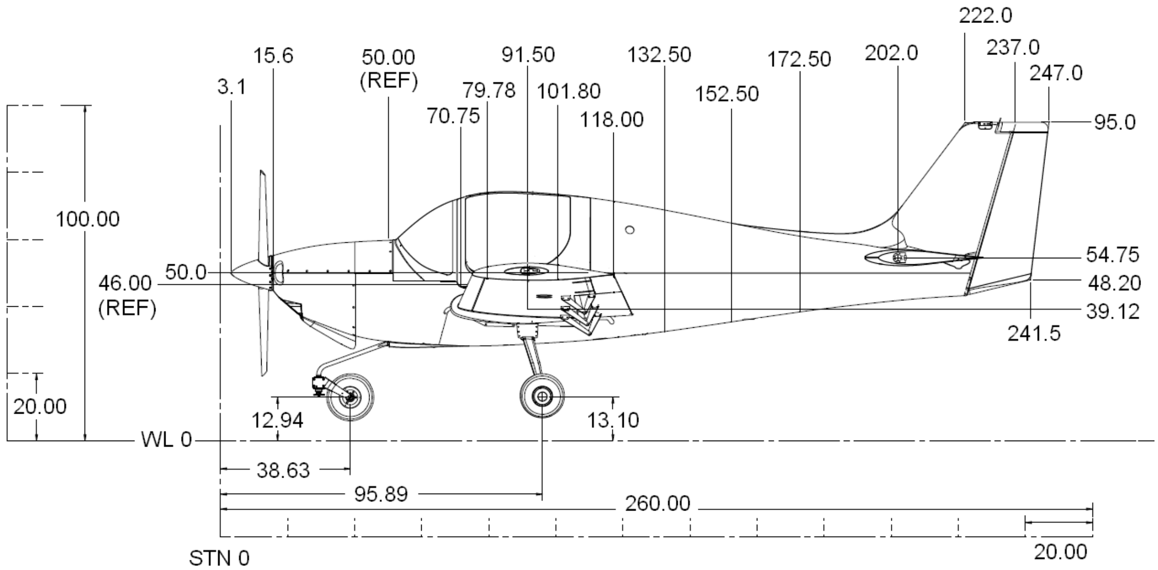

Figure 03-4 Measurements from the Waterline and Station

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Section 03-03 Aircraft Access Panels and Cowlings

This section describes the type and location of the various cowlings, panels and access panels on the exterior and interior of the airplane. Figure 03-5, Figure 03-6, Figure 03-7, and Figure 03-8 show the detail of the location of the access panels. There are three major cowlings or panels. These are:

- Upper Engine Cowling
- Lower Engine Cowling
- Belly Panel

The exterior of the airplane has the following access panels

- Fuel Filler Access Panel
- Wing Access Panels (two panels on the underside of each wing)
- Three Access Panels on the starboard side by the tail
  - Upper Torque Tube Access Panel
  - Lower Torque Tube Access Panel
  - Trim Motor Access Panel
- Engine Oil Filler Access Panel
- Port and starboard wing root fairings

The interior of the airplane has the following access panels:

- Fuel Tank Sender Access Panel (located in the pilot's seat back)
- Baggage Floor Access Panel (center of the baggage compartment)
- Baggage Bay Aft Bulkhead Access Panel

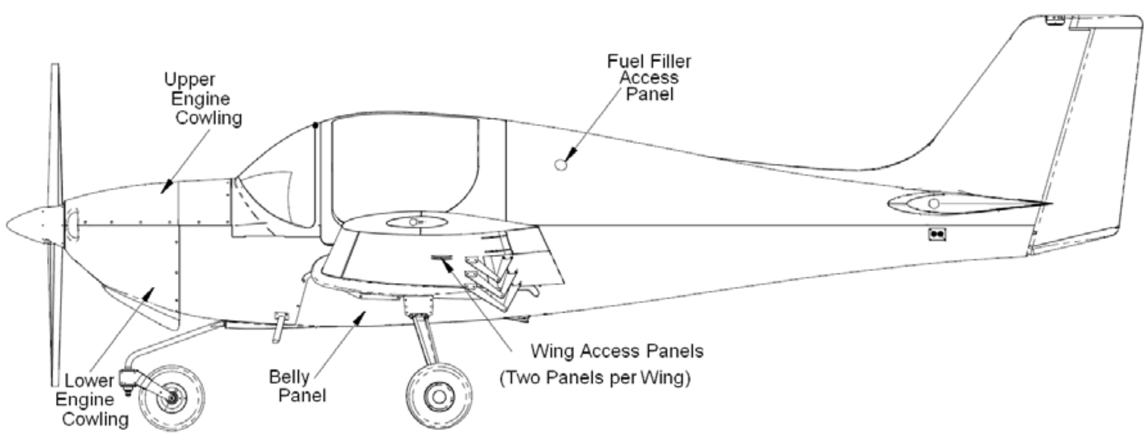

Figure 03-5 Access Panels Port

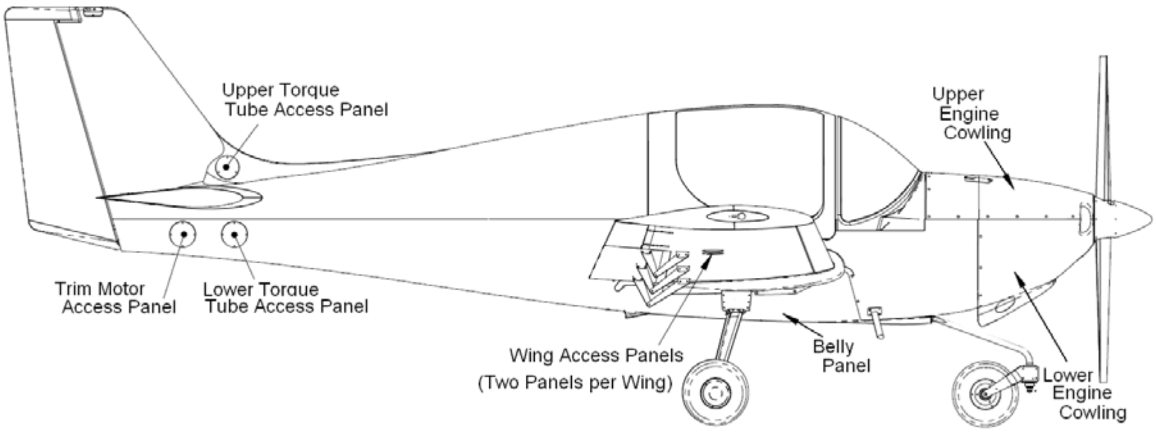

Figure 03-6 Access Panels Starboard

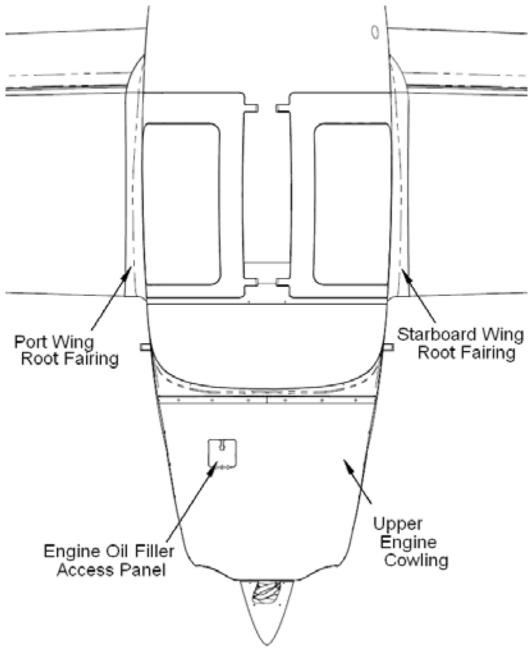

Figure 03-7 Wing Root and Engine Oil Filler Access Panel

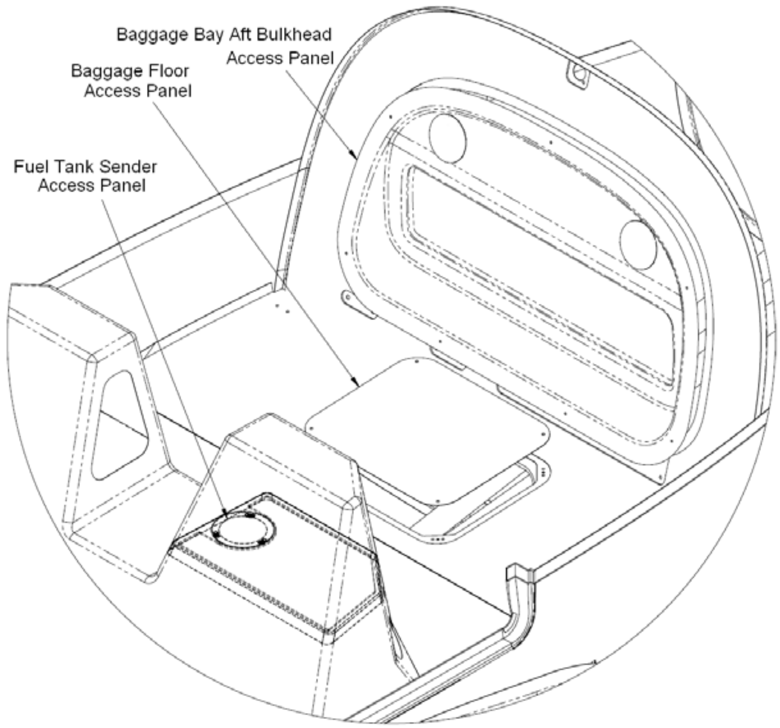

Figure 03-8 Interior Access Panels Aft of the Cockpit

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CHAPTER 04

AIRWORTHINESS LIMITATIONS

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Table of Contents

SECTION 04-00	GENERAL	5
SECTION 04-01	AIRWORTHINESS LIMITATIONS	7
SECTION 04-02	LIFE LIMITS AND MANDATORY INSPECTIONS	9
SECTION 02-01	ITEMS SUBJECT TO LIFE LIMITS	9
SECTION 02-02	ITEMS SUBJECT TO MANDATORY INSPECTIONS	10
SECTION 02-03	BONDED COMPOSITE LAMINATES	13
SECTION 02-04	LIBERTY XL-2 COMPLIANCE AT A GROSS WEIGHT OF 1750 LBS.	17
SECTION 02-05	BATTERY INSPECTION AND REPLACEMENT REQUIREMENTS	19
SECTION 04-03	EXTERNAL SURFACE PAINT REQUIREMENTS	21

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Section 04-00 General

The Airworthiness Limitations section contains the following requirements:

Life Limits and Mandatory Inspections:

- Structure and Components Subject to Life Limits
- Mandatory Inspections
- Battery Replacement Requirements
- External Surface Paint Requirements

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Section 04-01 Airworthiness Limitations

The Airworthiness Limitations Section is FAA approved and specifies inspections and other maintenance required under paragraphs 43.16 and 91.403 of the Federal Aviation Regulations unless an alternative program has been FAA approved.

This section is also EASA approved by the FAA on behalf of the European Aviation Safety Agency (EASA) using the bilateral agreement between the FAA and the DGAC-France. This section was prepared in accordance with the EASA requirements and any variations must be EASA approved in accordance with CS23.1529.

REV	DATE	APPROVED
~	07/31/09	For: Melvin D Taylor, Manager Atlanta Aircraft Certification Office Federal Aviation Administration Central Region
A	10/28/09	For: Melvin D Taylor, Manager Atlanta Aircraft Certification Office Federal Aviation Administration Central Region
B	02/02/2010	For: Melvin D Taylor, Manager Atlanta Aircraft Certification Office Federal Aviation Administration Central Region
C	06/21/2010	For: Melvin D Taylor, Manager Atlanta Aircraft Certification Office Federal Aviation Administration Central Region

REV	DATE	APPROVED
D	02/16/2011	For: Melvin D Taylor, Manager Atlanta Aircraft Certification Office Federal Aviation Administration Central Region

Table 04-1 FAA Approvals

Section 04-02 Life Limits and Mandatory Inspections

This section details items that are subject to life limits and mandatory inspections.

Section 02-01 Items Subject to Life Limits

The items listed in Table 04-2 are life-limited items with mandatory replacement times. Remove these items from service at the replacement times indicated. Render the out-of service items unserviceable and discard them to prevent re-use.

NOTE

Date of removal and replacement of life-limited components, including flight hours, must be recorded in aircraft maintenance records to ensure correct observation of the stated interval(s).

Component	P/N	Mandatory Replacement Time
Composite Fuselage	135A-10-105	5000 hours
Wing Skin and Internal Structure	135A-20-005[1] 135A-20-006[1] 135A-20-007[1] 135A-20-008[1]	15000 hours
Wing Root Fitting (Spar Tang)	135A-20-501	3220 hours
Space Frame Assembly	135A-10-075	15000 hours
Nose Lock Pin	135A-40-565	500 hours
Tail Plane Attachment Lug	135A-30-615	500 hours
Firewall Blanket	33M1154-301	2000 hours
Induction Air Filter	R-1260 [2]	500 hours
Dawley Exhaust	DEL-200201-005, DEL-200201-006	1000 hours
Fire Extinguisher	RT A600[3]	12 years

[1]This does not include Wing Root Fitting (Spar Tang), 135A-20-501.
[2] Reference TCM - Installation and Operation Manual OI-22, Chapter 5
[3]Reference H3R, Inc. Service Manual for the RT A1200, RT A600, RTA400 Fire Extinguisher

Table 04-2 Structure and Components Subject to Life Limits

Section 02-02 Items Subject to Mandatory Inspections

Table 04-3 details those components that are subject to mandatory inspection.

Component	P/N	Inspection Type	Inspection Interval
Throttle Control System	135A-50-017	Refer to Chapter 05 Engine (Firewall Forward) Checklist	Annual for all aircraft and 100 hour as applicable per Part 91
Tail Plane Mass Balance Drive Arm	135A-45-291	Visual Inspection: Inspect the control circuit connection for the elevator from the forward, lower trim motor access panel. Refer to Chapter 05.	Annual for all aircraft and 100 hour as applicable per Part 91
Wing Skin and Internal Structure	135A-20-005 135A-20-006 135A-20-007 135A-20-008 This does not include Wing Root Fitting (Spar Tang), 135A-20-501.	Refer to Chapter 05 Control Surfaces Inspection Checklist	Annual for all aircraft and 100 hour as applicable per Part 91
Wing Root Fittings	135A-10-085 135A-10-086 135A-20-501	Refer to Chapter 05 Control Surfaces Inspection Checklist	Annual for all aircraft and 100 hour as applicable per Part 91
Space Frame Assembly	135A-10-075	Refer to Chapter 05 Control Surfaces Inspection Checklist	Annual for all aircraft and 100 hour as applicable per Part 91
Composite Fuselage	135A-10-105	Visual Inspection of All: Fuselage Bond Lines and Bonded Composite Laminate Structures to Fuselage as listed in Table 04-4 through Table 04-7 Give attention to joints between Baggage Bay Floor Supports and Fuselage. See Figure 04-5 for the location of these supports Tap Test and/or other Inspection method(s) delineated in Chapter 51 on All Bonded Composite Laminated Structures: Horizontal Center Line Between Upper and Lower Fuselage Between Upper Fuselage Headline Structure and Lower Fuselage Hoop Reinforcement Structure All Bonded Composite Laminate Structures to Fuselage See Figure 04-4 through Figure 04-9 Refer to Chapter 05.	Annual for all aircraft and 100 hour as applicable per Part 91

Component	P/N	Inspection Type	Inspection Interval
For airplanes equipped with a standard exhaust system and the optional bypass scat tube has not been installed	Dawley Drawing 200201-002	Refer to Chapter 05 Engine (Firewall Forward) Checklist	Inspect every 25 hours Time-In-Service or every 12 months, whichever occurs first, see FAA AD 2009-08-05
For airplanes equipped with a standard exhaust system and the optional bypass scat tube has been installed	Dawley Drawing 200201-002	Refer to Chapter 05 Engine (Firewall Forward) Checklist	Inspect every 50 hours Time-In-Service or every 12 months, whichever occurs first, see FAA AD 2009-08-05
For airplanes equipped with a reduced sound exhaust system and the optional bypass scat tube has been installed	Dawley Drawing 200201-003	Refer to Chapter 05 Engine (Firewall Forward) Checklist	Inspect every 50 hours Time-In-Service or every 12 months, whichever occurs first, see FAA AD 2009-08-05 for placard requirements for the cabin heater
For airplanes equipped with a reduced sound exhaust system and the optional bypass scat tube has not been installed	Dawley Drawing 200201-003	Refer to Chapter 05 Engine (Firewall Forward) Checklist	Inspect within 10 hours Time-In-Service after April 20, 2009, see FAA AD 2009-08-05
For airplanes equipped with a reinforced standard or reduced sound exhaust system and the optional bypass scat tube has not been installed	Standard Dawley Drawing 200201-006 Or Reduced Sound Dawley Drawing 200201-005	Refer to Chapter 05 Engine (Firewall Forward) Checklist	Inspect every 50 hours Time-In-Service or every 12 months, whichever occurs first Report inspection results to Liberty Customer Service
For airplanes equipped with a reinforced standard or reduced sound exhaust system and the optional bypass scat tube has been installed	Standard Dawley Drawing 200201-006 Or Reduced Sound Dawley Drawing 200201-005	Refer to Chapter 05 Engine (Firewall Forward) Checklist	Inspect every 50 hours Time-In-Service or every 12 months, whichever occurs first Report inspection results to Liberty Customer Service
Nose Lock Pin	135A-40-565	Refer to Chapter 05 Landing Gear Inspection Checklist	Annual for all aircraft and 100 hr as applicable per Part 91

Component	P/N	Inspection Type	Inspection Interval
Tail Plane Attachment Lug	135A-30-615	Refer to Chapter 05 Control Surfaces Inspection Checklist	Annual for all aircraft and 100 hour as applicable per Part 91
Firewall Blanket	33M1154-301	Refer to Chapter 05 Engine (Firewall Forward) Checklist	Annual for all aircraft and 100 hour as applicable per Part 91
For All Airplanes, Port-side Clevis Aft Spar Weldment	135A-10-235 (Part of Space Frame Assy., 135A-10-075)	Refer to Chapter 05 Control Surfaces Inspection Checklist	Inspect every 100 hours of time in service.
For All Airplanes, Starboard-side Clevis Aft Spar Weldment	135A-10-236 (Part of Space Frame Assy., 135A-10-075)	Refer to Chapter 05 Control Surfaces Inspection Checklist	Inspect every 100 hours of time in service.

[1]This does not include Wing Root Fitting (Spar Tang), 135A-20-501.

Table 04-3 Components Subject to Mandatory Inspections

Section 02-03 Bonded Composite Laminates

An uncured "pre-preg" cloth is a fiber, either carbon or glass, impregnated with an epoxy resin. Layers of pre-preg will determine the thickness. The increases in thickness can come from the insertion of foam between the layers to create sandwich constructions. Cured composite pre-preg plies produce laminates, which are Solid, or Sandwich. Different thicknesses of laminates are used to form a structural support to distribute stresses through the structure. The majority of the Liberty XL-2 fuselage is sandwich construction. There are areas of the structure like the "rollover" hoop structure that are solid laminate. Bond lines are areas of solid laminates joined together with a permanent two-part epoxy paste adhesive.

Figure 04-1, Figure 04-2, and Figure 04-3 shows the three types of joints.

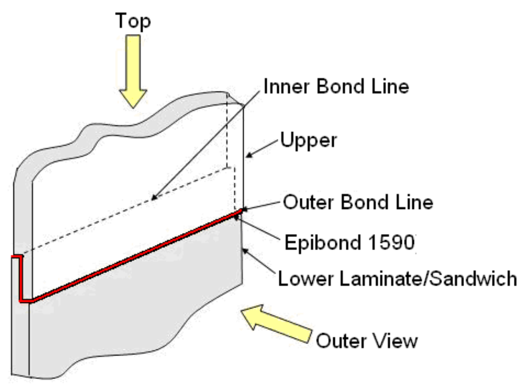

Figure 04-1 Lap Joint

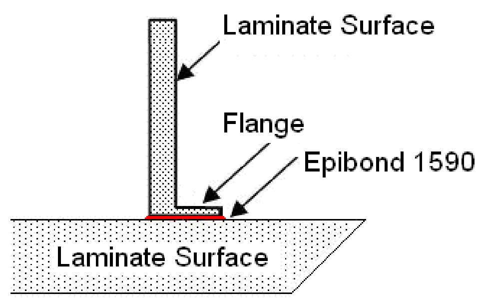

Figure 04-2 Flange Joint

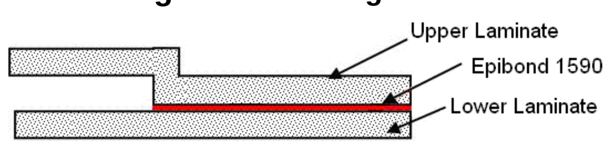

Figure 04-3 Face-to-Face Joint

Table 04-4 through Table 04-6 details the fuselage bond lines and the fuselage composite laminates. Table 04-7 gives the details of the additional fuselage bond lines for gross weight 1750 lbs. compliance. See Figure 04-4 through Figure 04-9 for details of the location of the bond joints.

Figure Callout	Description	Bond Joint
1	Horizontal Center Line between Upper and Lower Fuselage	Lap Joint
2	Bulkhead Baggage Bay between Upper and Lower Fuselage	Flange Joint
3	Bulkhead Mid Fuselage and the Upper and Lower Fuselage	Flange Joint
4	Battery Braces (Main and Backup) and Lower Fuselage	Face-to-Face Joint
5	Fin Spar and Upper and Lower Fuselage	Flange Joint
6	Fin Rib 1 and Upper Fuselage	Flange Joint
7	Fin Rib 2 and Upper Fuselage	Flange Joint
8	Fin Rib 3 and Upper Fuselage	Flange Joint
9	Fin Closeout-Vertical Closeout and Upper and Lower Fuselage	Flange Joint
10	Bond Line between Headliner and Upper & Lower Fuselage	Face-to-Face Joint

Table 04-4 Bond Line Callouts for Figure 04-4

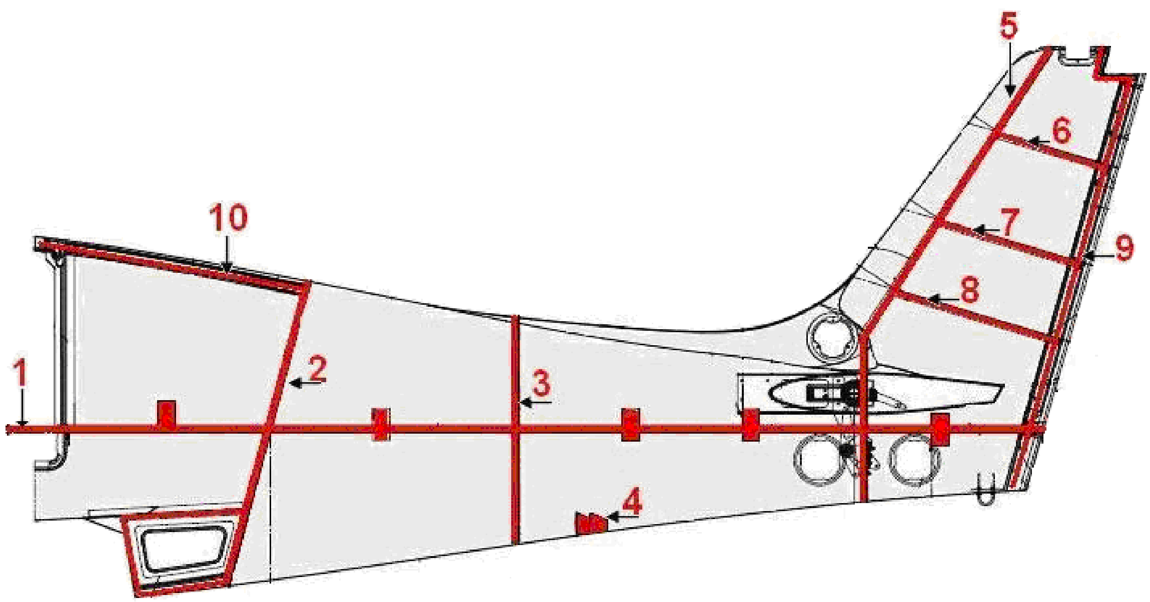

Figure 04-4 Upper and Lower Fuselage Bond Lines

NOTE

During inspections, give attention to joints between the fuselage and baggage bay floor supports (item number 12 in Figure 04-5); which can be viewed through the floor access panel. There are two baggage bay floor supports (Port and Starboard), which are bonded to the fuselage. See Figure 04-5 for the location of the baggage bay supports.

Figure Callout	Description	Bond Joint
2	Bulkhead Baggage Bay between Upper and Lower Fuselage	Flange Joint
3	Bulkhead Mid Fuselage and the Upper and Lower Fuselage	Flange Joint
5	Fin Spar and Upper and Lower Fuselage	Flange Joint
6	Fin Rib 1 and Upper Fuselage	Flange Joint
7	Fin Rib 2 and Upper Fuselage	Flange Joint
8	Fin Rib 3 and Upper Fuselage	Flange Joint
9	Fin Closeout-Vertical Closeout and Upper and Lower Fuselage	Flange Joint
11	Baggage Bay Floor and Laminate Structures	Flange Joint
12	Baggage Bay Floor Supports (Port and STBD) and Laminate Structures	Flange Joint
18	Fin Horn Closeout	Flange Joint

Table 04-5 Bond Line Callouts for Figure 04-5

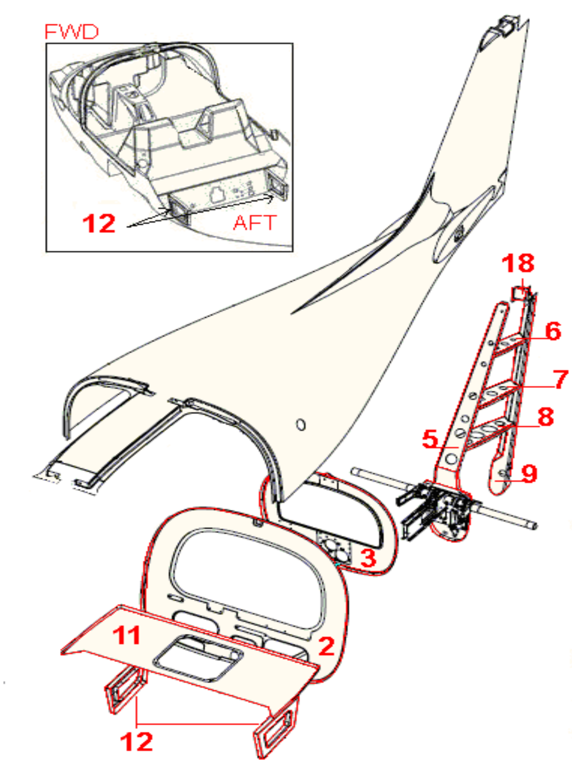

Figure 04-5 Upper Fuselage Bond Lines Detailing the Location of the Baggage Bay Supports

Figure Callout	Description	Bond Joint
4	Battery Braces (Main and Backup) and Lower Fuselage	Face-to-Face Joint
13	Bond Line between Hoop Reinforcement and Upper & Lower Fuselage	Face-to-Face Joint
14	Ducts NACA Cabin Air (Port and STBD) and Lower Fuselage	Flange Joint
15	Closeout Seatbacks (Port and STBD)	Flange Joint
16	Horizontal Center Line Reinforcement Strap between Upper & Lower Fuselage (10 quantity)	Face-to-Face Joint
17	Reinforcement Strap for Rollover Hoop and Lower & Upper Fuselage	Face-to-Face Joint

Table 04-6 Bond Line Callouts for Figure 04-6

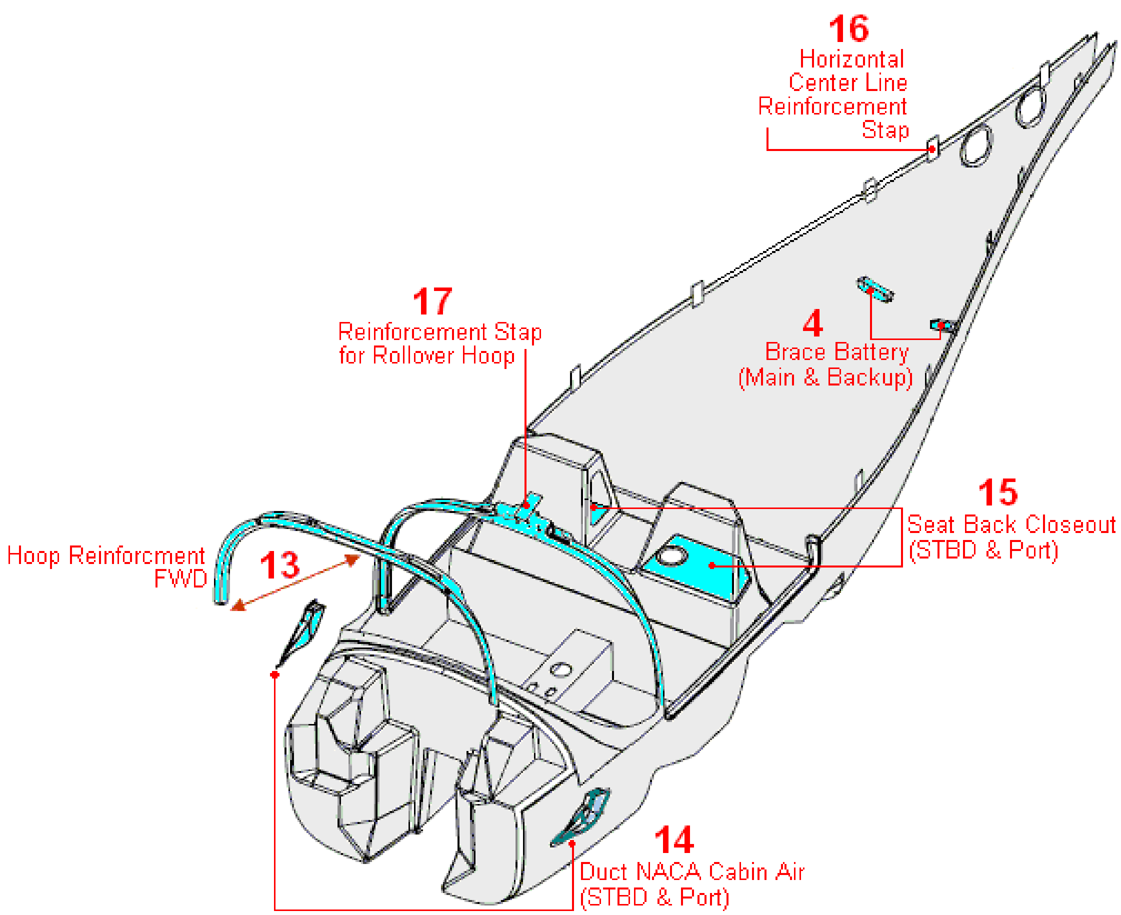

Figure 04-6 Lower Fuselage Bond Lines

Section 02-04 *Liberty XL-2 Compliance at a Gross Weight of 1750 Lbs.*

Table 04-7 and Figure 04-7 through Figure 04-9 details the location of the various stiffeners and bond lines for compliance for the gross weight of 1750 lbs. upgrade.

LIBERTY XL-2 COMPLIANCE AT A GROSS WEIGHT OF 1750LBS

- For SN0007 and SN0009 thru SN0115 if modified in accordance with RKI-SIL-08-001 for a maximum gross weight of 1750 lbs.
- For SN0116 and subsequent, these modifications are installed at the factory for a gross weight of 1750 lbs.

Figure	Description	Bond
Figure 04-7	Bond Line between Seat Base Stiffener and Lower Fuselage (port and starboard)	Face-to-Face Joint
Figure 04-8	Bond Line between FWD Bulkhead Reinforcement and Lower Fuselage	Face-to-Face Joint
Figure 04-9	Bond Line between Seat Back Stiffener and Lower Fuselage (starboard)	Face-to-Face and Lap Joint

Table 04-7 Compliance at a Gross Weight of 1750 lbs.

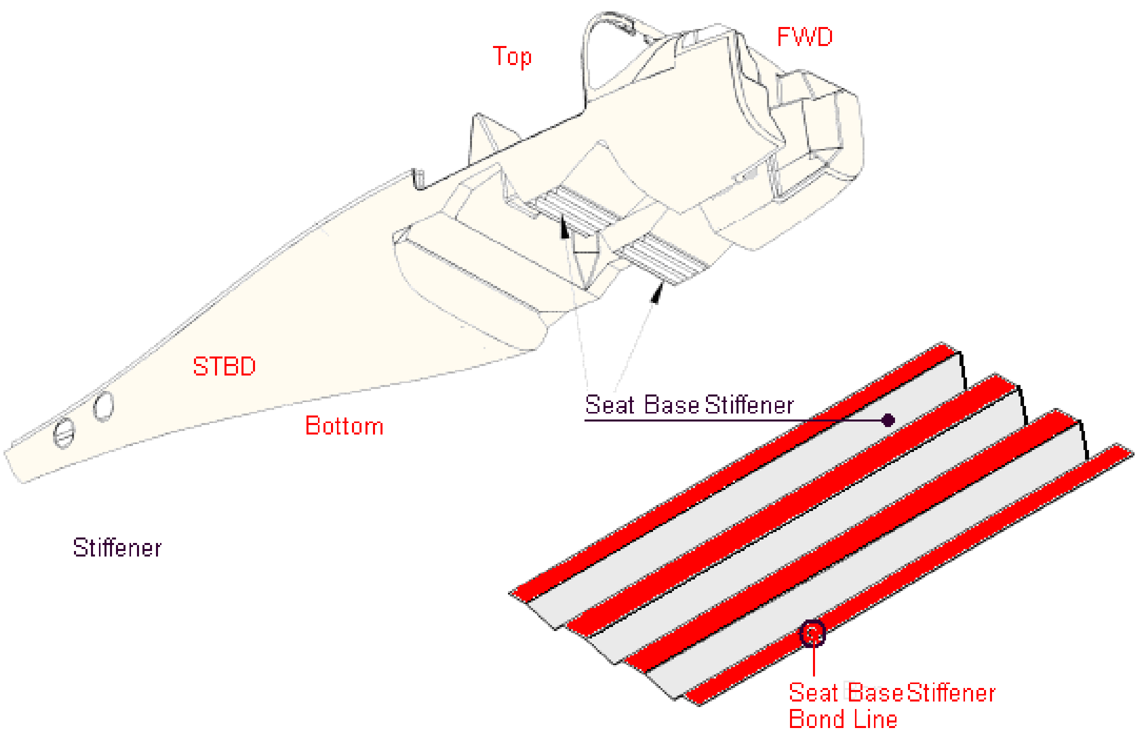

Figure 04-7 Seat Base Stiffener and Lower Fuselage Bond Lines

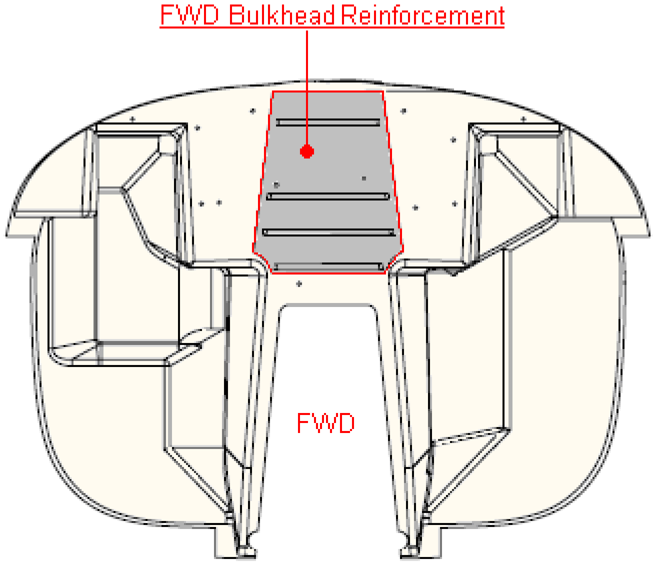

Figure 04-8 FWD Bulkhead and Lower Fuselage Bond Lines

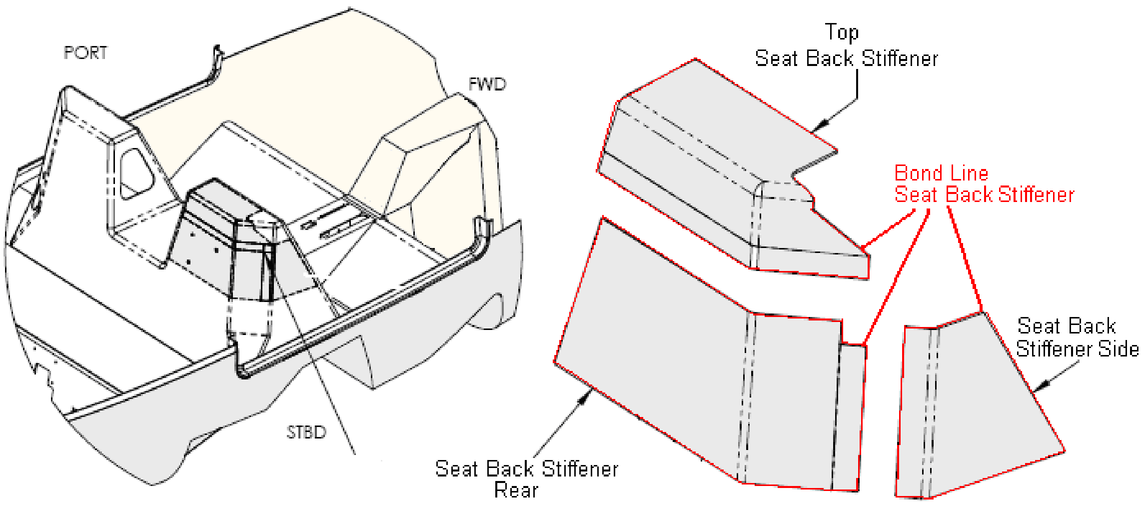

Figure 04-9 Seat Back Stiffener and Lower Fuselage Bond Lines

Section 02-05 Battery Inspection and Replacement Requirements

Inspect, and replace as required, the batteries listed in this table with new batteries at the intervals shown or at the recommended interval by the battery's vendor or at the interval prescribed by applicable local regulations, which ever occurs first.

Component	OEM	P/N	Inspection Interval	Mandatory Replacement Time
Primary Battery	Concord Battery	RG-25XC or RG-25	Annual for all aircraft	• 1800 hours or 3 years, whichever occurs first
Secondary Battery	Teledyne Continental Motors	6560701.	Annual for all aircraft	• 12 calendar months after date of installation. • If the secondary battery has been used for more than one (1) hour (emergency operations). • If the secondary battery is severely depleted • If the EBAT FAIL light stays illuminated for more than 5 minutes on the HSA panel

Table 04-8 Battery Inspection and Replacement Requirements

1. The TCM part number provides a Power-Sonic Corporation model PS-12120 12vdc 12 A.H. battery..

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Section 04-03 External Surface Paint Requirements

To ensure the temperature of the load-bearing composite structure is kept below the structural temperature limit, the outer surface of the composite components must be painted white except for areas of registration marks, placards, and minor trim, Figure 04-10 defines the zones.

NOTE

In Figure 04-10, zone I are areas of composite painted surfaces and zone II are areas of metallic painted surfaces.

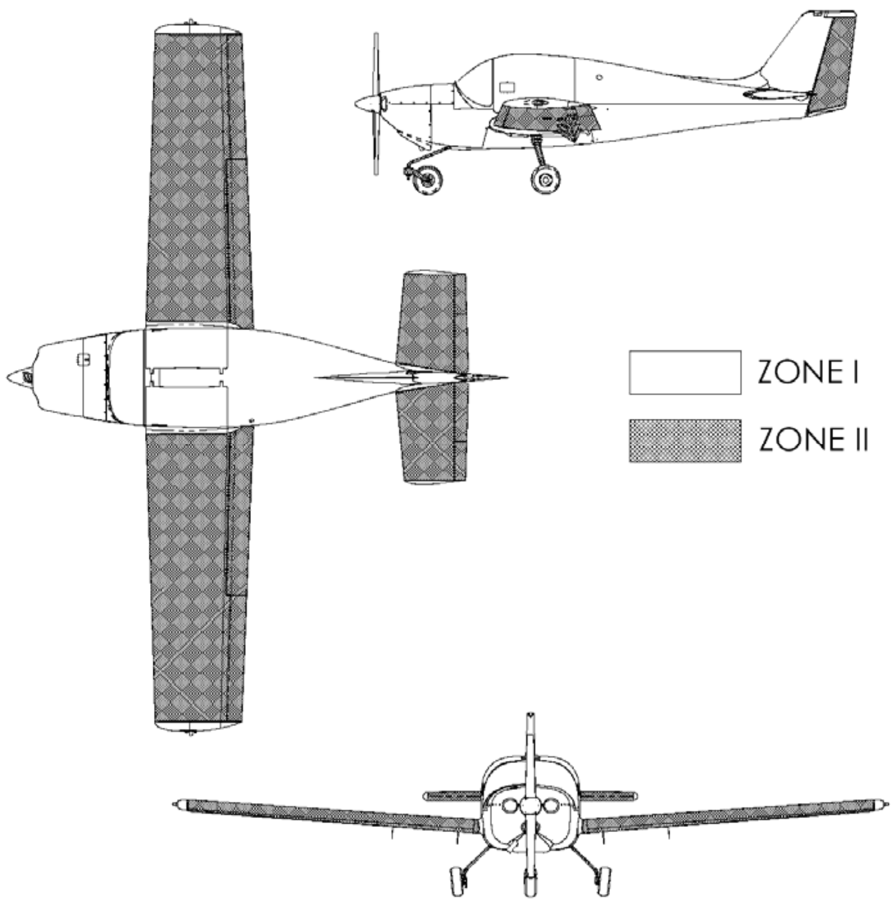

Figure 04-10 Paint Zones

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CHAPTER 05

TIME LIMITS/MAINTENANCE CHECKS/INSPECTION INTERVALS

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Table of Contents

SECTION 05-00	GENERAL	5
SECTION 05-10	TIME BETWEEN OVERHAUL LIMITS	7
SECTION 05-20	SCHEDULED MAINTENANCE CHECKS	9
	PRE-INSPECTION CHECKLIST	10
	ENGINE OPERATIONAL CHECKLIST	11
	ENGINE (FIREWALL FORWARD) CHECKLIST	12
	PROPELLER INSPECTION CHECKLIST	16
	CABIN AREA INSPECTION CHECKLIST	17
	AVIONICS INSPECTION CHECKLIST	19
	SPACE FRAME AND AFT FUSELAGE AREA (FIREWALL AFT) INSPECTION CHECKLIST	21
	CONTROL SURFACES INSPECTION CHECKLIST	23
	LANDING GEAR INSPECTION CHECKLIST	27
	CABIN HEAT INSPECTION CHECKLIST	29
	BATTERIES INSPECTION CHECKLIST	31
	GENERAL INSPECTION CHECKLIST	33
SECTION 05-30	MAINTENANCE CHECK FLIGHT	35
SECTION 30-01	PREFLIGHT INSPECTIONS	35
SECTION 30-02	LIBERTY XL-2 POST INSPECTION CHECK FLIGHT	35
	LIBERTY XL-2 POST INSPECTION CHECK FLIGHT	36
SECTION 05-40	SUPPLEMENTAL INSPECTIONS	43
SECTION 40-01	REPLACEMENT OF THE PROPELLER	43
SECTION 40-02	REPLACEMENT OF ENGINE, IGNITION, OR INSTRUMENTS:	43
SECTION 05-50	UNSCHEDULED MAINTENANCE CHECKS	45
SECTION 50-01	SUDDEN ENGINE STOPPAGE OR PROP STRIKE	45
SECTION 50-02	FOREIGN OBJECT DAMAGE TO PROPELLER	45
SECTION 50-03	NON-OPERATION OF MORE THAN 30 DAYS	45
SECTION 50-04	HARD LANDING INSPECTION	45
SECTION 50-05	HARD LANDING PRELIMINARY ASSESSMENT	47
SECTION 50-06	HARD LANDING DETAILED INSPECTION	49

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Section 05-00 General

Maintenance personnel should refer to Chapters 04 and 05 as guidelines for proper scheduling and execution of inspections and maintenance. Inspections and intervals described in these two chapters are considered the minimum required to maintain airplane in airworthy condition.

WARNING

TO PREVENT INJURY TO PERSONNEL CHEECK PROPELLER AREA IS CLEAR OF ANY PERSON IN THE EVENT THAT INSPECTION OR MAINTENANCE MUST BE PERFORMED WITH: ALT/BAT (MASTER) SWITCH "ON" AND EITHER FADEC "A" OR "B" SWITCH IS "ON," BATTERY CONNECTED AND PROPELLER BEING MOVED. IF IGNITION SWITCH OR GROUND WIRE IS DEFECTIVE, UNINTENTIONAL ENGINE FIRING OR RUNNING IS POSSIBLE EVEN WITH THE IGNITION SWITCH "OFF".

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Section 05-10 Time Between Overhaul Limits

Table 05-1 shows the time between overhauls for various components.

Component	OEM	P/N	Specified Time Between Overhaul
Engine, Including Accessories	Teledyne Continental Motors	IOF-240B	• 2000 operating hours or 12 years, whichever occurs first
Propeller	MT-Propeller	MT175R127-2Ca	• There is no specified overhaul time • Remove the propeller from service when it does not meet MT OEM Continued Airworthiness Requirements
Propeller	Sensenich	W69EK7-63G	• There is no specified overhaul time • Remove the propeller from service when it does not meet Sensenich OEM Continued Airworthiness Requirements
Induction Air Filter	K&N	R-1260	• 500 hours
Primary Battery	Concord Battery	RG-25XC or RG-25	• 1800 hours or 3 years, whichever occurs first
Secondary (FADEC Backup) Battery	Teledyne Continental Motors	6560701[1].	• 12 calendar months after date of installation • If the backup battery has been used for more than one (1) hour (emergency operations) • If the backup battery is severely depleted • If the EBAT FAIL light stays illuminated for more than 5 minutes on the HSA panel
Ameri-King[3] ELT Main Alkaline Manganese Dioxide Batteries	Duracell	MN1300	• Replace prior to expiration date printed on battery or annually whichever occurs first • After an emergency use • Whenever the ELT is activated for an unknown period of time the main batteries must be replaced [2] • On indication of damage or corrosion

Component	OEM	P/N	Specified Time Between Overhaul
Ameri-King[3] ELT Remote Unit Lithium Cell Battery	Duracell	DL 1/3 NB	• Every 8 years • After an emergency use • Whenever the ELT is activated for an unknown period of time[2] • On indication of damage or corrosion
Artex[3] ELT Main Battery Lithium 6 volt 7.5 Ah	Artex 3.	452-6499 (Order by Kit) (455-0012)	• Every 5 years • After an emergency use • After an operation of unknown duration • On indication of damage or corrosion
Artex[3] ELT Remote Switch Lithium 6 volt 170 mAh	Artex 3	131-0001	• Every 5 years • After an emergency use • After an operation of unknown duration • On indication of damage or corrosion

Table 05-1 Table Showing Time Between Overhaul Limits For Various Components

[1] The TCM part number provides a Power-Sonic Corporation model PS-12120 12vdc 12 Ah battery.
[2.] All batteries installed must have the same expiration date.
[3.] The Liberty XL-2 will be fitted with either an Ameri-King model AK-450 or Artex model ME406 ELT. Refer to airplane equipment list for model installed.

Section 05-20 Scheduled Maintenance Checks

The tasks contained in this section include requirements for performing scheduled inspections. Perform all inspection tasks in the following Inspection Checklists at the specified intervals. Intervals are measured in hours of operation and, as applicable, calendar months. Care must be taken to assure the correct hours and calendar month intervals noted are used for each checklist item. Liberty Aerospace, Inc. recommends photocopying this checklist, use to compile a record of work performed and retain with the airplane records for each inspection completed.

Check lists present scheduled inspection requirements at intervals of every 100 hours of operation and annually in accordance with applicable Federal Aviation Regulations. Where applicable, additional maintenance intervals are identified and noted next to the affected maintenance item.

Annual inspections are required for any Liberty XL-2 airplane that is flown less than 100-hours during any 12 calendar-month period. Do a 100-hour inspection to meet the requirements of an annual inspection.

INSPECTION CHECKLIST FOR LIBERTY XL-2 AIRPLANE			
Airplane Serial Number:		**Airplane Registration:**	
Inspection Date:		**Engine Tach Time:**	
Type Of Inspection:		**Airframe Total Time:**	

Inspection Checklist	Page
Pre-Inspection Checklist	10
Engine Operational Checklist	11
Engine (Firewall Forward) Checklist	12
Propeller Inspection Checklist	16
Cabin Area Inspection Checklist	17
Avionics Inspection Checklist	19
Space Frame and AFT Fuselage Area (Firewall AFT) Inspection Checklist	21
Control Surfaces Inspection Checklist	23
Landing Gear Inspection Checklist	27
Cabin Heat Inspection Checklist	29
Batteries Inspection Checklist	31
General Inspection Checklist	33

PRE-INSPECTION CHECKLIST

Table 05-2 details the items to check during the pre-inspection.

Pre-Inspection Checklist						
Item #	**Item Description**	**INTERVAL**				
		50 Hr	**100 Hr**	**Annual**	**Other**	**Initials**
1	Comply with Applicable Airworthiness Directives	X	X	X		
2	Comply with all of the following applicable Liberty Critical Service Bulletins: [1] • Category 1: Mandatory Service Bulletin/Airworthiness Directive (AD) *Associated "Rework Kit Instruction" (RKI) may accompany a Service Document.*	X	X	X		
3	Check the following applicable Liberty Critical Service Bulletins:[1] as maybe required to maintain warrantee • Category 2: Critical Service Bulletin (CSB) *Associated "Rework Kit Instruction" (RKI) may accompany a Service Document.*	X	X	X		
4	Check, as required, for all applicable Liberty Service Documents:[1] • Category 3: Service Bulletin (SB) • Category 4: Service Information Directive (SID) • Category 5: Service Information Letter (SIL) • Category 6: Special Service Instruction (SSI) *Associated "Rework Kit Instruction" (RKI) may accompany a Service Document.*	X	X	X		
5	Check Logs for Time in Service	X	X	X		
6	Check Logs for Discrepancies	X	X	X		
7	Wash Airplane (such that visibility of imperfections are apparent)	X	X	X		
8	**Complied with ALL requirements above**	X	X	X		

Table 05-2 Pre-Inspection Checklist

[1] Liberty Service Documents are available at the Liberty Aerospace, Inc. website http://www.libertyaircraft.com

ENGINE OPERATIONAL CHECKLIST

Table 05-3 details the items to check during engine operational check.

Engine Operational Check						
Item #	**Item Description**	**Inspection Interval**				
		50 Hr	**100hr**	**Annual**	**Other**	**Initials**
1	Perform engine operational performance test in accordance with TCM IOF-240-B Maintenance Manual, M-22 Chapter 6, latest revision	X	X	X		
2	Run engine until oil temperature reaches 75° F or 21° C minimum	X	X	X		
3	Check engine instruments for operation within normal limits	X	X	X		
4	Perform FADEC A and B power transfer check (See TCM Installation and Operation Manual, OI-22, Chapter 7)	X	X	X		
5	Check ignition L and R at 1700 RPM (a minimum RPM drop should be observed; maximum RPM drop 150 RPM)	X	X	X		
6	Check for RPM drop when ALT AIR control is pulled.	X	X	X		
7	Perform alternator, alternator belt and alternator control unit operation and inspection in accordance with Liberty Maintenance Manual Chapter 24 – *Electrical Power*	X	X	X		
8	**Complied with ALL requirements above**	X	X	X		

Table 05-3 Engine Operational Checklist

ENGINE (FIREWALL FORWARD) CHECKLIST

Table 05-4 details the items to check during the engine (firewall forward) check.

WARNING

OPERATING A MALFUNCTIONING ENGINE CAN RESULT IN ADDITIONAL ENGINE DAMAGE, BODILY INJURY, OR DEATH.

WARNING

WORN SPARK PLUGS THAT ARE ALLOWED TO CONTINUE IN SERVICE MAY CAUSE EXCESSIVE LOADING OR PREMATURE BREAKDOWN OF THE FADEC HIGH VOLTAGE COILS.

CAUTION

Failure to properly install and maintain engine baffles and baffle seals will adversely affect cylinder service life.

Engine (Firewall Forward) Checklist						
Item #	**Item Description**	**Inspection Interval**				
		50 Hr	**100 Hr**	**Annual**	**Other**	**Initials**
1	Remove upper and lower cowling, check the cowling for cracks, overheated areas, deformation, delaminating, loose or missing fasteners, chafing or abnormal condition and oil, reference Liberty Maintenance Manual Chapter 71 – *Power Plant*.	X	X	X		
2	Inspect engine baffles for cracks and signs of metal fatigue in accordance with TCM IOF-240-B Maintenance Manual, M-22 Chapter 7.		X	X		
3	Inspect engine mount for cracks. Check engine mount bushings and supporting hardware for signs of deterioration, proper assembly and security, reference Liberty Maintenance Manual Chapter 71 – *Power Plant*.		X	X		
4	Check all engine fasteners for integrity and security.		X	X		
5	Visually inspect condition of firewall blanket for tears or discoloration of the facing sheet materials or surrounding sealant materials. Tears or discoloration is unacceptable. Refer to Liberty Maintenance Manual Chapter 71 "*Power Plant*" for inspection details.		X	X		

Engine (Firewall Forward) Checklist						
Item #	**Item Description**	**Inspection Interval**				
		50 Hr	**100 Hr**	**Annual**	**Other**	**Initials**
6	Inspect engine case, cylinders and accessory case for evidence of leaking fuel or oil. Correct any discrepant items in accordance with TCM IOF-240-B Maintenance Manual, M-22 Chapter 7 prior to release of airplane to service.	X	X	X		
7	Thoroughly wash engine with approved cleaning solution and visually inspect in accordance with TCM IOF-240-B Maintenance Manual, M-22; Chapter 9-1.	X	X	X		
8	Perform all engine inspections, and maintenance in accordance with TCM IOF-240-B Maintenance Manual, M-22 Chapter 7, and other manuals as maybe applicable.	X	X	X		
9	Check brake fluid level reference Liberty Maintenance Manual Chapter 12 - *Servicing*.	X	X	X		
10	Check engine frame rubber mounts for “sag”.	X	X	X		
11	Remove, inspect and service induction air filter; inspect 3/8 inch clearance between lower alternate air tube and #4 engine cylinder exhaust riser. Clean lower alternate air tube of dust and debris reference Liberty Maintenance Manual Chapter 75 - *Engine Air*.	X	X	X		
12	Check the induction air filter for time in service, per Table 05-1 of this chapter. If time limit has expired, replace the filter in accordance with Chapter 75 – *Engine Air*		X	X		
13	Inspect the threads for the alternate air valve arm for wear or damage; reference Chapter 71- *Power Plant*.		X	X		
14	Inspect induction for leaks, cracks, deterioration, broken, missing, or loose brackets, clamps, and hardware. Clean and oil air filter, replace, and re-safety wire.		X	X		
15	Remove, inspect, and replace oil filter. Install new filter element in accordance with TCM IOF-240-B Maintenance Manual, M-22; Chapter 9-3.	X	X	X		
16	Inspect air/oil separator, oil cooler, oil sump, oil filler neck, oil lines and fittings for cracks, leaks, deterioration, loose and missing hardware.	X	X	X		

Engine (Firewall Forward) Checklist						
Item #	Item Description	Inspection Interval				
		50 Hr	100 Hr	Annual	Other	Initials
17	All airplanes with standard exhaust without the SCAT bypass tube installed (Dawley Drawing 200201-002). Remove external shroud and inspect exhaust system for cracks, excessive leakage, deterioration, loose and missing brackets, clamps, and hardware. Visually inspect exhaust system components for evidence of exhaust leakage, reference Liberty Maintenance Manual Chapter 78 – *Engine Exhaust*.	X	X	X	X 25 Hours	
18	All airplanes with a standard exhaust with the SCAT bypass tube installed (Dawley Drawing 200201-002). Remove external shroud and inspect exhaust system for cracks, excessive leakage, deterioration, loose and missing brackets, clamps, and hardware. Visually inspect exhaust system components for evidence of exhaust leakage, reference Liberty Maintenance Manual Chapter 78 – *Engine Exhaust*.	X	X	X		
19	All airplanes with standard reinforced exhaust without the SCAT bypass tube installed (Dawley Drawing 200201-002 or 200201-006 standard and Dawley Drawing 200201-003 or 200201-005 quiet). Remove external shroud and inspect exhaust system for cracks, excessive leakage, deterioration, loose and missing brackets, clamps, and hardware. Visually inspect exhaust system components for evidence of exhaust leakage, reference Liberty Maintenance Manual Chapter 78 – *Engine Exhaust*.	X	X	X	X 25 Hours	
20	All airplanes with standard reinforced exhaust with the SCAT bypass tube installed (Dawley Drawing 200201-002 or 200201-006 standard and Dawley Drawing 200201-003 or 200201-005 quiet). Remove external shroud and inspect exhaust system for cracks, excessive leakage, deterioration, loose and missing brackets, clamps, and hardware. Visually inspect exhaust system components for evidence of exhaust leakage, reference Liberty Maintenance Manual Chapter 78 – *Engine Exhaust*.	X	X	X		
21	Clean, inspect, gap, and test all spark plugs.		X	X		

Engine (Firewall Forward) Checklist						
Item #	Item Description	Inspection Interval				
		50 Hr	100 Hr	Annual	Other	Initials
22	Check all engine controls, control cables, control rod ends, and levers for security, wear, improper assembly, routing, and freedom of movement through entire range of travel.		X	X		
23	Inspect forward and aft throttle cable rod end swage for excessive wear such as frayed or broken strands. Inspect rod end bearing for excessive side to side play. Check that a split pin is installed properly with no excessive wear. Check large flat washer between bearing rod end and castellated nut is installed; reference Liberty Maintenance Manual Chapter 76-*Engine Controls* for installation and adjustment procedures.		X	X		
24	Visually inspect nut-exhaust manifold flange (size .31-24) and verify torque value in accordance with TCM IOF-240-B Maintenance Manual, M-22, Appendix B.		X	X		
25	Inspect FADEC high and low voltage wire harnesses for chafing, or damage in accordance with TCM IOF-240-B Maintenance Manual, M-22.		X	X		
26	Remove and clean gascolator. Replace safety-wire; reference Liberty Maintenance Manual Chapter 28 – *Fuel System*.		X	X		
27	Check for proper function of “fuel shutoff” and “fuel return & check valve”.	X	X	X		
28	At the completion of any engine maintenance event the engine must be given a complete and thorough Operational Run-up. A “Test Flight” maybe required by FAR 91.407 if any engine adjustments have been made which might appreciably affect flight characteristics, unless ground checks can determine airworthiness.	X	X	X		
29	Inspect/replace any accessories required to be replaced at 500-hour check in accordance with TCM IOF-240-B Maintenance Manual, M-22 and Liberty Maintenance Manual Chapter 04 – *Airworthiness Limitations*.		X	X		
30	**Correct all ENGINE discrepancies noted.**	X	X	X		

Table 05-4 Engine (Firewall Forward) Checklist

PROPELLER INSPECTION CHECKLIST

Table 05-5 details the items to check during the propeller check.

WARNING

IMPROPER PROPELLER BOLT TORQUE MAY CAUSE IN FLIGHT BOLT, PROPELLER, OR BLADE FAILURE, ANY OF WHICH WILL RESULT IN DEPARTURE OF PROPELLER FROM THE AIRPLANE.

CAUTION

If the operating humidity and/or temperature environment will change significantly for a long period of time, check the propeller bolt torque shortly after arrival at the new location. Then recheck the bolt torque at intervals of no more than 10 hours until the propeller has stabilized and two or more successive 10-hour checks show no change in bolt torque.

Propeller Inspection Checklist						
Item #	**Item Description**	**Inspection Interval**				
		50 Hr	**100 Hr**	**Annual**	**Other**	**Initials**
1	Remove spinner; check spinner and spinner bulkheads for cracks, dents, run-out, loose or missing fasteners, reference Liberty Maintenance Manual Chapter 61 - *Propeller*.	X	X	X		
2	If the airplane has an MT propeller, check propeller mounting bolt torque in accordance with latest revision of the MT Propeller Maintenance Manuals. See the CAUTION with this table.		X	X	After first flight 25 hr Post Install	
3	If the airplane has a Sensenich propeller, check propeller mounting bolt torque in accordance with latest revision of the Sensenich Maintenance Manuals. See the CAUTION with this table.	X	X	X	After first flight 25 hr Post Install	
4	Check propeller, hub, and blades for damage, abrasions, or other abnormalities in accordance with the MT-Propeller or Sensenich Maintenance Manuals.	X	X	X		
5	Reinstall spinner, reference Liberty Maintenance Manual Chapter 61-*Propeller*.	X	X	X		
6	**Correct all PROPELLER discrepancies noted.**	X	X	X		

Table 05-5 Propeller Inspection Checklist

CABIN AREA INSPECTION CHECKLIST

Table 05-6 details the items to check during the cabin area check.

Cabin Area Inspection Checklist						
Item #	**Item Description**	**Inspection Interval**				
		50 Hr	**100 Hr**	**Annual**	**Other**	**Initials**
1	Confirm all required instrument, control knob and handle markings, and interior and exterior placards are properly displayed and correct, see Liberty Maintenance Manual Chapter 11 – *Placards and Markings*	X	X	X		
2	Check door locking mechanisms for proper "over center" operation; reference Liberty Maintenance Manual Chapter 52 - *Doors*	X	X	X		
3	Check door hinges for cracks or signs of fatigue, reference Liberty Maintenance Manual Chapter 52 - *Doors*	X	X	X		
4	Check that doors seal tight against the rubber seal that runs around the door cutout in the fuselage	X	X	X		
5	Check that door release pins are fully seated	X	X	X		
6	Check condition of gas springs and gas springs attachments; reference Liberty Maintenance Manual Chapter 52 - *Doors*	X	X	X		
7	Check door window to doorframe bond for signs of deterioration, check window for cracks, crazing, and condition of vent windows (if installed), reference Liberty Maintenance Manual Chapter 56 - *Windows*	X	X	X		
8	Check windscreen for crazing, cracks, leakage around windshield fastener and bondline, reference Liberty Maintenance Manual Chapter 56 - *Windows*	X	X	X		
9	Check for temporary repairs to the windscreens. If temporary repairs are present, replace the windscreen see Liberty Maintenance Manual Chapter 56 – *Windows*.		X	X		
10	Check safety belts/harnesses for condition, proper operation of buckles reference Liberty Maintenance Manual Chapter 25 – *Equipment and Furnishings*	X	X	X		

Cabin Area Inspection Checklist						
Item #	Item Description	Inspection Interval				
		50 Hr	100 Hr	Annual	Other	Initials
11	Check rudder pedals and adjustment mechanism for condition, operation. If the rudder movement is beyond rigging values, it is excessive wear, contact the factory. Reference Liberty Maintenance Manual Chapter 27 – Flight Controls	X	X	X		
12	If installed, check the operation of the left and right finger brake; reference Liberty Maintenance Manual Chapter 32 – Landing Gear.	X	X	X		
13	If installed, check the operation of the toe brakes on the starboard side rudder pedals and on the port side rudder pedals, reference Liberty Maintenance Manual Chapter 32 – Landing Gear.	X	X	X		
14	Check parking brake operation; reference Liberty Maintenance Manual Chapter 32 – Landing Gear	X	X	X		
15	Check all interior and exterior lights for proper operation, reference Liberty Maintenance Manual Chapter 33 - Lights	X	X	X		
16	Remove seat cushions, remove any dirt or debris	X	X	X		
17	Check yoke weldment assembly (pilot's and copilot's control stick). There should be free movement from one stop to the other and in all directions. There is to be no interference to this freedom of movement throughout this range. There is to be no movement of either stick beyond any one stop	X	X	X		
18	Check Fire Extinguisher see Chapter 25 – *Equipment and Furnishings*		X	X		
19	Check all control circuits	X	X	X		
20	Check compass for proper operation and for current correction card, reference Liberty Maintenance Manual Chapter 34 – Navigation and Pitot/Static	X	X	X		
21	Check instruments for functional operation, reference Liberty Maintenance Manual Chapter 31- Indicators and Recording System.		X	X		
22	**Correct all discrepancies**	X	X	X		

Table 05-6 Cabin Area Inspection Checklist

AVIONICS INSPECTION CHECKLIST

Table 05-7 details the items to check during the avionics check.

NOTE

Perform the following inspections as applicable for the avionics configuration installed. For inspection list equipment not installed, mark N/A in the initials column.

Avionics Inspection Checklist						
Item #	**Item Description**	**Inspection Interval**				
		50 Hr	**100 Hr**	**Annual**	**Other**	**Initials**
1	Audio Panel Perform operation check in accordance with Liberty Maintenance Manual Chapter 23 - *Communications* for audio panel model installed.		X	X		
2	COM/NAV Receiver #1 Perform operation check in accordance with Liberty Maintenance Manual Chapter 23 - *Communications* for receiver model installed in the COM1 avionics panel position.		X	X		
3	NAV Receiver #1 Perform VOR accuracy check in accordance with 14CFR Part 91.171 for receiver model installed in the COM1 avionics panel position.				30 days	
4	COM/NAV Receiver #2 Perform operation check in accordance with Liberty Maintenance Manual Chapter 23 - *Communications* for receiver model installed in the COM2 avionics panel position.		X	X		
5	NAV Receiver #2 Perform VOR accuracy check in accordance with 14 CFR Part 91.171 for receiver model installed in the COM2 avionics panel position.				30 days	
6	Course Deviation Indicator #1 Perform operation check in accordance with Liberty Maintenance Manual Chapter 34 – *Navigation and Pitot/Static* for model installed in the CDI1 flight panel position.		X	X		

Avionics Inspection Checklist						
Item #	**Item Description**	**Inspection Interval**				**Initials**
		50 Hr	**100 Hr**	**Annual**	**Other**	
7	Course Deviation Indicator #2 Perform operation check in accordance with Liberty Maintenance Manual Chapter 34 – *Navigation and Pitot/Static* for model installed in the CDI2 flight panel position.		X	X		
8	Transponder Perform operation check in accordance with Liberty Maintenance Manual Chapter 34 – *Navigation and Pitot/Static* for transponder model installed in the avionics panel.		X	X		
9	Transponder Perform test and inspection of transponder in compliance with 14 CFR 91.413 requirements for the transponder model installed.				24 month	
10	Altitude Encoder Perform operation check in accordance with Liberty Maintenance Manual Chapter 34 – *Navigation and Pitot/Static* for altitude encoder model installed.		X	X		
11	Altitude Encoder Perform test and inspection of transponder in compliance with 14 CFR 91.413 and 91.217 requirements for the model altitude encoder model installed.				24 month	
12	Emergency Locator Transmitter Perform operation check and inspection in accordance with Liberty Maintenance Manual Chapter 25 – *Equipment and Furnishings* for ELT model installed.		X	X		
13	**Correct all <u>AVIONICS</u> discrepancies noted.**		X	X	X	

Table 05-7 Avionics Inspection Checklist

SPACE FRAME AND AFT FUSELAGE AREA (FIREWALL AFT) INSPECTION CHECKLIST

Table 05-7 details the items to check during the avionics check.

Space Frame and AFT Fuselage Area (AFT Firewall)Inspection Checklist						
Item #	**Item Description**	**Inspection Interval**				
		50 Hr	**100 Hr**	**Annual**	**Other**	**Initials**
1	Remove belly panel assembly; reference Liberty Maintenance Manual Chapter 53 - *Fuselage*.		X	X		
2	Inspect space frame assembly *(chassis)* for damage, peeling of painted surfaces or external corrosion		X	X		
3	Inspect yoke weldment assembly mount *(under fuselage*) for any signs of damage. Check the copilot's control stick mount for damage or signs of excessive wear, such as dents, cracks or lost paint or any free play. *Note: Free play of the system is considered excessive wear. Excessive wear is evident if the yoke goes beyond the rigging limits as defined in the TCDS.*		X	X		
4	Check flight control hinges and linkage for excessive wear. If the pin is visible between the teeth of the hinge, it is excessive wear. Refer to Liberty Maintenance Manual Chapter 27 – *Flight Controls* for details on inspection criteria.		X	X		
5	Inspect control throttle linkage		X	X		
6	Check brake master cylinders and brake lines for condition and attachment, reference Liberty Maintenance Manual Chapter 32 – *Landing Gear*		X	X		
7	Check elevator and rudder hanger bellcranks in center fuselage for condition and proper connection *(access panel in floor of baggage compartment)* reference Liberty Maintenance Manual Chapter 27 – *Flight Controls*		X	X		
8	Check elevator and rudder bellcranks in AFT fuselage for condition and proper connection *(access panels on STB, lower AFT fuselage)* reference Liberty Maintenance Manual Chapter 27 – *Flight Controls*		X	X		

Space Frame and AFT Fuselage Area (AFT Firewall)Inspection Checklist						
Item #	Item Description	Inspection Interval				
		50 Hr	100 Hr	Annual	Other	Initials
9	Check fuel lines in space frame assembly (*chassis*) for leakage, reference Liberty Maintenance Manual Chapter 28 – *Fuel Systems*		X	X		
10	Inspect the bottom and sides of the fuel tank for evidence of leakage such as blue witness lines or 'liquid' marks. Reference Liberty Maintenance Manual Chapter 28 – *Fuel Systems*		X	X		
11	Check all antennas and pitot/static head for condition		X	X		
12	Check operation of stall warning vane		X	X		
13	Check fuselage for damages and defects: fuselage bond lines- (between upper & lower fuselage) and (between headliner and lower fuselage). Check fuselage laminate: fuselage-vertical stabilizer, baggage bay floor and floor supports, hoop reinforcement, reference: Liberty Maintenance Manual Chapter 04 - *Airworthiness Limitations*, Chapter 51 – *Standard Practices – Structures*, and Chapter 53 – *Fuselage*.	X	X	X		
14	Check footstep as described in Chapter 53- *Fuselage*		X	X		
15	Irnspect Tail Plane Attachment Lugs for wear. Remove pins and inspect internal boes and the external of all lugs for scratches dents or signs of stress.		X	X		
16	**Correct all SPACE FRAME AND AFT FUSELAGE discrepancies noted**	X	X	X		

Table 05-8 Space Frame and AFT Fuselage Inspection Checklist

CONTROL SURFACES INSPECTION CHECKLIST

Table 05-9 details the items to check during the control surfaces check.

Control Surfaces Inspection Checklist						
Item #	**Item Description**	**Inspection Interval**				
		50 Hr	**100 Hr**	**Annual**	**Other**	**Initials**
1	Remove belly panel assembly; reference Liberty Maintenance Manual Chapter 53 - *Fuselage*.		X	X		
2	Verify control surfaces operating deflection limits, reference Liberty Maintenance Manual Chapter 06 – *Dimensions and Areas*.		X	X		
3	Check electrical bonds and grounds; reference Liberty Maintenance Manual Chapter 20 – *Standard Practices*.		X	X		
4	Check AFT fuselage rudder and elevator push rods for condition and deformation, reference Liberty Maintenance Manual Chapter 27 – *Flight Controls*.		X	X		
5	Check skins of wings, and tail surfaces for any signs of damage such as cracks or buckling in the skin. Pay close attention to the wing root surface ribs. There is to be no evidence of rivets working loose (exhibited by a black ring around the outside edge of any one rivet, also known as a smoking rivet).See Liberty Maintenance Manual Chapter 57 – *Wings*.		X	X		
6	Check all exposed areas of the wing pin ends for evidence of wear (such as gouges, cracks, bent pins, or peeling of the nickel-plating). Inspect wing pins for full seating (1/16-inch of exposed shaft, beyond the chamfered surface). Inspect all exposed areas of the area around the wing box, aft tang weldment (attachment lugs), flap spigot receiver, forward and aft wing tangs and flap spigot for evidence of wear (such as gouges, cracks, metal shavings or deformed metal), reference Liberty Maintenance Manual Chapter 57 – *Wings*.		X			
7	Remove wings; reference Liberty Maintenance Manual Chapter 57 – *Wings*.		X	X		

Control Surfaces Inspection Checklist						
Item #	**Item Description**	**Inspection Interval**				
		50 Hr	**100 Hr**	**Annual**	**Other**	**Initials**
8	Inspect the two 1.00 diameter holes in the port and starboard main wing tangs (two holes on each wing tang), the two 1.00 diameter holes in the port and starboard chassis-side wing box, the 0.625 inch hole in the port and starboard aft wing spar attachment lug, and chassis-side port and starboard aft wing weldment for gouges, cracks, metal shavings, and wear, such as free-play between parts or out-of-round condition in the lugs Report any free-play between parts or out-of-round condition to Liberty Customer Service.		X	X		
9	Inspect the six wing pins for evidence of wear, such as gouges, cracks, metal shavings, bent pins, peeling of the nickel plating, or free-play between mating surfaces. Determine free-play by measuring the internal bore of the lugs and the external diameter of the pins. Report any differences greater than 0.005in to Liberty Customer Service. In addition, moving the wing fore and aft carefully at the wing tip may highlight any free-play.		X	X		
10	Inspect the flap spigot and flap spigot receiver bearings for gouges, crack, metal shavings or bent pins		X	X		
11	Inspect the mechanical linkage between the wing pins and wing locking mechanism. Confirm proper operation of both the mechanical and electrical mechanism. Refer to Liberty Maintenance Manual Chapter 57 – *Wings*		X	X		
12	Check aileron quick-connect assemblies for condition. There shall be no free play between mating surfaces while holding the control surface(s) and attempting to move the yoke control, or holding the yoke control and attempting to move the control surface. Reference Liberty Maintenance Manual Chapter 57 – *Wings*		X	X		

Control Surfaces Inspection Checklist

Item #	Item Description	Inspection Interval				
		50 Hr	100 Hr	Annual	Other	Initials
13	Check aileron quick-connect assemblies for condition. There is to be no evidence of wear on the mating surfaces of the quick connect assemblies, such as gouges or cracks.		X	X		
14	Inspect the wing root fairing and wing skin attachment for any Epibond® voids or any cracks, reference Liberty Maintenance Manual Chapter 57 – *Wings*.		X	X		
15	Install wings; reference Liberty Maintenance Manual Chapter 57 – *Wings*.		X	X		
16	Check operation of stall warning system.		X	X		
17	Check for movement of aileron and elevator control circuits with the control yokes or sticks immobilized, reference Liberty Maintenance Manual Chapter 27 – *Flight Controls*		X	X		
18	Check ailerons, flaps, and rudder hinges for corrosion and inspect for cracks using 10x magnifying lens reference Liberty Maintenance Manual Chapter 27– *Flight Controls*.		X	X		
19	Check aileron bell-cranks in wings for condition and proper connection (access panels in underside of wings) reference Liberty Maintenance Manual Chapter 27– *Flight Controls*.		X	X		
20	Inspect inboard aileron pushrod, check for chafing at rib 0, reference Liberty Maintenance Manual Chapter 27– *Flight Controls*.		X	X		
21	Inspect hinges and anti-servo tabs for condition. If the pin is visible between the teeth of the hinge, it is excessive wear. Verify proper deflection and direction of anti-servo tabs using electric trim system; reference Liberty Maintenance Manual Chapter 27– *Flight Controls*.		X	X		

Control Surfaces Inspection Checklist						
Item #	Item Description	Inspection Interval				
		50 Hr	100 Hr	Annual	Other	Initials
22	Tail Plain Mass Balance – Inspect welded region around attachment for cracks. Inspect fastener holes for elongation (on torque tube attachment) mass balance slider block for wear (0.010 max) before replacement, or wear of the neoprene stops beyond 0.010”		X	X		
23	Check trim system, actuator motor and trim position indicator for proper operation, reference Liberty Maintenance Manual Chapter 27– *Flight Controls*.		X	X		
24	Check trim actuator for condition, secure attachment, and electrical connections, reference Liberty Maintenance Manual Chapter 27– *Flight Controls*.		X	X		
25	Check flap actuator and position sensor, reference Liberty Maintenance Manual Chapter 27– *Flight Controls*.		X	X		
26	Remove horizontal stabilizers; inspect anti-servo/trim tab drive plates and links, and attachments reference Liberty Maintenance Manual Chapter 27– *Flight Controls*.		X	X		
27	Inspect the attachment bolts for the correct size, type, length, and torque. The bolts must be an AN3-7 – 10-32 X 29/32 inch bolt. The torque on the bolt shall be measured and be within the range of 25-40 in-lbs (32.5 in-lbs nominal). For location, details, and procedures, refer to Chapter 55 – *Stabilizers*.		X	X		
28	Inspect for cracks in rear spar aileron closeout and inspect for cracks in mass balance enclosure flange.		X	X		
29	Inspect the FLAP RIB 2 and the wing RIB ‘1’ AFT for any damage or crack.		X	X		
30	Inspect aileron structure and installation clearances.		X	X		
31	**Correct all <u>CONTROL SURFACES</u> discrepancies noted.**		X	X		

Table 05-9 Control Surfaces Inspection Checklist

Landing Gear Inspection Checklist

Table 05-10 details the items to check during the landing gear check.

Landing Gear Inspection Checklist						
Item #	Item Description	Inspection Interval				
		50 Hr	100 Hr	Annual	Other	Initials
1	Check tires for condition, wear, proper inflation (50 psi -0/+2 psi).		X	X		
2	Clean wheels.		X	X		
3	Clean and grease wheel bearings.		X	X		
4	Check wheel bearings for play, corrosion, smooth running; reference Liberty Maintenance Manual Chapter 32 – *Landing Gear*.		X	X		
5	Check Main and Nose wheel rims for cracks. If cracks are found, replace the wheel rims. Reference Chapter 32 – *Landing Gear*		X	X		
6	Check Main landing gear legs for damage and deformation; reference Liberty Maintenance Manual Chapter 32 – *Landing Gear*.		X	X		
7	Check Main landing gear attachment bolts for condition, security; condition of bushings; reference Liberty Maintenance Manual Chapter 32 – *Landing Gear*.		X	X		
8	Check Nose wheel steering bearing for smoothness of operation, proper resistance to motion (shimmy), reference Liberty Maintenance Manual Chapter 32 – *Landing Gear*.		X	X		
9	Inspect and clean nose wheel shimmy dampening components, reference Liberty Maintenance Manual Chapter 32 – *Landing Gear*		X	X		
10	Check Nose wheel assembly for secure attachment, cracks, deformation, and straightness with airplane centerline datum.		X	X		
11	Check Nose landing gear for damage or deformation; reference Liberty Maintenance Manual Chapter 32 – *Landing Gear*.		X	X		

Landing Gear Inspection Checklist						
Item #	**Item Description**	**Inspection Interval**				
		50 Hr	**100 Hr**	**Annual**	**Other**	**Initials**
12	Check Nose landing gear attachment bolts for condition and security. (Fuselage belly fairing must be removed for access to landing gear bolts). Reference Liberty Maintenance Manual Chapter 32 – *Landing Gear*, and Chapter 53 - *Fuselage*		X	X		
13	Check Nose lock pin for excessive wear: larger inner diameter of nose lock pin barrel may not be less than 0.240 inches.		X	X		
13	Check brake linings for condition and wear, reference Liberty Maintenance Manual Chapter 32 – *Landing Gear*.		X	X		
14	Check brake caliper for condition, leaks, reference Liberty Maintenance Manual Chapter 32 – *Landing Gear*.		X	X		
15	Inspect wheel fairings fasteners must have high strength Loctite Number 270 on threads and safety-wire all AN502-10-8 machine screws.	X	X	X		
16	Check wheel fairings are rigidity installed. CAUTION: If parts are not thread-locked and wire-locked, vibration may occur in flight.	X	X	X		
17	**Correct all LANDING GEAR discrepancies noted.**	X	X	X		

Table 05-10 Landing Gear Inspection Checklist

CABIN HEAT INSPECTION CHECKLIST

Table 05-11 details the items to check during the cabin heat check.

Cabin Heat Inspection Checklist						
Item #	**Item Description**	**Inspection Interval**				
		50 Hr	**100 Hr**	**Annual**	**Other**	**Initials**
1	Remove cabin heat shroud and inspect muffler for corrosion, holes, and cracks, reference Liberty Maintenance Manual Chapter 21 – *Environmental Systems*.			X		
2	Check fit of cabin heat shroud around muffler and tail pipe; no air gaps larger than .02” permissible, reference Liberty Maintenance Manual Chapter 21 – *Environmental Systems*.			X		
3	Examine hoses for holes, cracking, and crushing; remove and replace if necessary, reference Liberty Maintenance Manual Chapter 21 – *Environmental Systems*.			X		
4	Inspect cabin heat valve box flapper for proper operation; lubricate linkage, reference Liberty Maintenance Manual Chapter 21 – *Environmental Systems*.			X		
5	Check control cable for damage, nicks, or chaffing and check that it operates through full range of motion, reference Liberty Maintenance Manual Chapter 21 – *Environmental Systems.*			X		
6	Inspect firewall blanket around cabin heat valve box for tight fit or any settling of the blanket, reference Liberty Maintenance Chapter 21 – *Environmental Systems*.			X		
7	Inspect carbon around both cabin heat foot-well air valve assemblies for damage due to heating cycles; reference Liberty Maintenance Manual Chapter 21 – *Environmental Systems*.			X		

Cabin Heat Inspection Checklist						
Item #	**Item Description**	**Inspection Interval**				
		50 Hr	**100 Hr**	**Annual**	**Other**	**Initials**
8	Inspect fiberglass instrument panel around both louvers for damage due to heating cycles; reference Liberty Maintenance Manual Chapter 21 – *Environmental Systems*.			X		
9	Verify round insert on cabin heat push/pull knob is secure.	X	X	X		
10	**Correct all <u>CABIN HEAT</u> discrepancies noted.**	X	X	X		

Table 05-11 Cabin Heat Inspection Checklist

BATTERIES INSPECTION CHECKLIST

Table 05-12 details the items to check during the batteries check.

Batteries Inspection Checklist						
Item #	**Item Description**	**INTERVAL**				
		50 Hr	**100 Hr**	**Annual**	**Other**	**Initials**
1	Use the Operational Check and Inspection procedure for the primary battery in Chapter 24 – *Electrical Power* to check primary battery for cleanliness, corrosion of connections, security of mounting and state of charge in accordance with Liberty Maintenance Manual Chapter 24 – *Electrical Power*		X	X		
2	Inspect primary battery tray and battery cable terminals for cracks, corrosion or signs of fatigue. Repair or replace as required. Reference Liberty Maintenance Manual Chapter 24 – *Electrical Power*		X	X		
3	Check primary battery for time in service. Reference Liberty Maintenance Manual Chapter 04 – *Airworthiness Limitations* If time limit has expired, replace the primary battery. Reference Liberty Maintenance Manual Chapter 24 – *Electrical Power*		X	X		
4	Use the Operational Check and Inspection procedure for the secondary battery in Chapter 24 – *Electrical Power* to check secondary battery for cleanliness, corrosion of connections, security of mounting and state of charge. Replace as required. Reference Liberty Maintenance Manual Chapter 24 – *Electrical Power*	.	X	X		
5	Inspect secondary battery tray and battery cable terminals for cracks, corrosion or signs of fatigue. Repair or replace as required. Reference Liberty Maintenance Manual Chapter 24 – *Electrical Power*		X	X		
6	Check secondary battery for time in service. Reference Liberty Maintenance Manual Chapter 04 – *Airworthiness Limitations* If time limit has expired, replace the secondary battery. Reference Liberty Maintenance Manual Chapter 24 – *Electrical Power*		X	X		

Batteries Inspection Checklist						
Item #	**Item Description**	**INTERVAL**				
		50 Hr	**100 Hr**	**Annual**	**Other**	**Initials**
7	Inspect ELT battery and battery compartment for the following in accordance with Liberty Maintenance Manual Chapter 25 – *Equipment and Furnishings*: • After use in an emergency • When the transmitter has been in use for more than 1 cumulative hour • After an inadvertent activation of unknown duration • On any evidence of corrosion or leakage of any cell			X		
8	Check ELT battery for time in service. If time limit has expired or is about to expire, replace the ELT battery. Reference Liberty Maintenance Manual Chapter 25 – *Equipment and Furnishings*		X	X		
9	Inspect ELT remote switch battery and battery compartment for the following in accordance with Liberty Maintenance Manual Chapter 25 – *Equipment and Furnishings*: • After use in an emergency • When the transmitter has been in use for more than 1 cumulative hour • After an inadvertent activation of unknown duration • On any evidence of corrosion or leakage of any cell			X		
10	Check ELT remote switch battery for time in service. If time limit has expired or is about to expire, replace the ELT battery. Reference Liberty Maintenance Manual Chapter 25 – *Equipment and Furnishings*		X	X		
11	**Correct all BATTERIES discrepancies noted.**		X	X		

Table 05-12 Batteries Inspection Checklist

GENERAL INSPECTION CHECKLIST

Table 05-13 details the items to check during the general check.

General Inspection Checklist						
Item #	Item Description	INTERVAL				
		50 Hr	100 Hr	Annual	Other	Initials
1	Drain/purge pitot/static system; reference Liberty Maintenance Manual Chapter 34 – Navigation and Pitot/Static.		X	X		
2	Check pitot/static system for leaks and cleanliness, reference Liberty Maintenance Manual Chapter 34 – Navigation and Pitot/Static.		X	X		
3	Lubricate in accordance with lubrication schedule, reference Liberty Maintenance Manual Chapter 12 - Servicing.	X	X	X		
4	At the tail ballast assembly located at approximately STN 217, (gain access through the exterior rear access panels beneath the horizontal stabilizer). Inspect the weight and balance records for the correct ballast weight. This applies only to non gross weight upgrade aircraft operating at a weight of 1653 lbs. Airplanes upgraded to a gross weight of 1750 lb are not approved to operate with tail ballast weight.	X	X	X		
5	Check for foreign objects or tools; re-install all access plates and covers.	X	X	X		
6	Re-install belly panel assembly, reference Liberty Maintenance Manual Chapter 53 - Fuselage.	X	X	X		
7	Re-install engine cowlings.	X	X	X		
8	Re-install wheel fairing, if applicable.	X	X	X		
9	Perform “Liberty XL-2 Post Inspection Check Flight” using the Liberty XL-2 Post Inspection Check Flight form confirming all items on the form have been checked.	X	X	X		
10	Enter Inspection and All Reports in Airplane Maintenance Logbook.	X	X	X		
12	**Correct all GENERAL discrepancies found that affect airworthiness.**	X	X	X		

Table 05-13 General Inspection Checklist

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Section 05-30 Maintenance Check Flight

Preflight Inspection and Liberty XL-2 Post Inspection Check Flight are part of the maintenance procedures.

Section 30-01 Preflight Inspections

This inspection should be performed before all maintenance check flights by the pilot that will be making the flight. This inspection is to determine the general condition of the airplane and the engine. Check airplane flight logs for any discrepancies, and recording of all landings and flight hours. Procedures for the Preflight Inspection are contained in the approved Airplane Flight Manual (AFM).

Section 30-02 Liberty XL-2 Post Inspection Check Flight

This section lists the specific flight checks that must be successfully accomplished (when applicable) following a regularly scheduled 50-hour, 100-hour, or Annual inspection. or other regularly scheduled maintenance checks. A pilot qualified in the Liberty XL-2 should fly the check flight

The pilot is responsible for reviewing the airplane's logbook and other pertinent documentation to ensure the airplane is ready for a flight. At a minimum, the pilot should conduct a preflight briefing with the mechanic responsible for the maintenance check. This is to determine the status of the airplane and if there are any special considerations for the flight.

If required, the check flight items can be tailored based on the scope of work accomplished. The flight check items are listed in a sequence that would normally allow accomplishment of the required events while minimizing risk to the pilot and airplane. The check pilot is ultimately responsible to complete the flight check in a safe and efficient manner based on the status of the airplane, mission requirements, weather, time constraints and any other conditions that affect the mission.

Upon completion of the mission, the pilot shall fill out the appropriate areas (date, name, flight time) and indicate which items passed or failed. Document all items that did not pass on the section form and debrief with the responsible mechanic. Retain completed forms as part of the airplane's record.

LIBERTY XL-2 POST INSPECTION CHECK FLIGHT

Registration:		Takeoff time:		Takeoff fuel:	
Date:		Land time:		Takeoff weight:	
Pilot:		Flight time:		Landing fuel:	
Location:		Engine time:			

Post Inspection Check Flight Procedure				
Step	**Step Description**	**Pass**	**Fail**	**Initials**
1	**Preflight – Exterior** Complete the following items and accomplish an exterior preflight check per the AFM			
1.1	Documents – Verify required documents are onboard – current or temporary AFM, current weight & balance, airworthiness certificate, and registration.			
1.2	Airplane Preflight – Normal checklist items accomplished and no anomalies noted.			
2	**Preflight – Interior** Complete the following & accomplish a normal interior preflight.			
2.1	Instruments & gauges – Check all instruments installed with proper markings.			
2.2	Throttle – Verify smooth unrestricted operation with proper cushioning at ‘stops’			
2.3	Flight controls – Verify proper control surface movement (ailerons, rudder, stabilator), smooth operation, with no binding or excessive forces.			
2.4	Electric stabilator trim			
2.4.1	Check for full, continuous travel in both directions			
2.4.2	Verify pitch trim circuit breaker stops trim motion			
2.4.3	Verify correct direction of trim tab motion			
2.4.4	Verify trim tab approximately faired to ¼” up (relative to stabilator) with the trim set in the takeoff position and stabilator in the faired position.			
2.5	Flaps – Check for proper operation and lights for each flap position.			
2.6	Fuel gauge – Verify that the fuel gauge reads maximum indicated capacity with the fuel tank topped off.			
2.7	Interior lighting – check proper operation of all cockpit lighting			
2.8	Preflight Inspection checklist – accomplished with no discrepancies noted.			

Post Inspection Check Flight Procedure				
Step	**Step Description**	**Pass**	**Fail**	**Initials**
3	**Engine Start** Complete the following & accomplish the normal Engine Start checklist.			
3.1	Starter Annunciator – Verify light is illuminated when starter engaged & extinguished when released			
3.2	Alternator – after engine start, confirm alternator operating by verifying that electrical system voltage is greater than 12 volts.			
3.3	At 1000 +/-100 RPM, verify that the ammeter load increases/decreases with increased/decreased load (landing light, strobe, beacon, pitot heat).			
3.4	Engine start checklist accomplished with no discrepancies noted.			
4	**Before Taxi, Taxi and Engine Run-up** Complete the following items & accomplish normal checklists per the AFM			
4.1	Brakes – Verify proper operation without excessive binding or chatter and check forces required to brake and steer are reasonable.			
4.2	Heading systems – Verify magnetic compass & Directional Gyro (DG) are functioning normally.			
4.3	Altimeter – Verify altimeter reads ± 50 feet of field elevation with local altimeter.			
4.4	VSI – Verify zero rate of climb.			
4.5	Electronic Control Unit (ECU) – With engine at idle power, momentarily turn the ignition switch to the OFF position and then back to BOTH (before engine stops). Verify engine attempts to turn off.			
4.6	Checklists accomplished with no discrepancies noted.			
5	**Before Takeoff, Takeoff and Climb out** Complete the following items and accomplish normal checklists per the AFM.			
5.1	During takeoff roll and climb out, verify:			
5.1.1	WOT light on the HSA light panel illuminated with throttle at MAX power.			
5.1.2	Proper airspeed indications			
5.1.3	Flight controls command proper airplane response.			
5.1.4	Normal longitudinal control forces during rotation.			
5.1.5	RPM stays within limits (< 2800) and engine instruments within the green arc.			
5.1.6	Airplane noise levels are within tolerable levels.			
5.1.7	Checklists accomplished with no discrepancies noted.			

Post Inspection Check Flight Procedure				
Step	**Step Description**	**Pass**	**Fail**	**Initials**
6	**Air work** Conduct at safe altitude (normally ≥ 2,500ft AGL)			
6.1	Controllability and Stability			
6.1.1	Verify proper airplane response and control force harmony.			
6.1.2	At cruise airspeed (110±10kts), wings level, trimmed straight & level, verify airplane maintains coordinated flight (yaw axis).			
6.2	Lateral trim – Verify airplane does not require excessive lateral forces to maintain wings level, coordinated flight.			
7	**Instrumentation & Controls**			
7.1	Vertical Speed Indicator (VSI) – Verify VSI is operating normally.			
7.2	Attitude indicator – Verify ADI is operating normally.			
7.3	At cruise power (60-70%), wings level, trimmed straight & level, verify the following:			
7.3.1	Engine instruments indicate normal (in green) to include percent power, RPM, manifold pressure, fuel pressure, fuel flow, oil pressure, oil temperature, voltage, amperes, cylinder head temperature and EGT.			
7.3.2	Cabin vents and airplane heating system work normally.			
7.3.3	Doors are sealed and air does not come up through center console.			
7.3.4	Observe engine, airframe, and instrument panel for excessive vibration. Verify vibration levels are within tolerable levels.			
7.3.5	Observe cabin noise levels due to engine noise, airflow, door seals, etc. Verify noise level is within tolerable levels.			
7.3.6	Directional Gyro (DG) works normally and processes < 4° in 10 min of flight with normal attitudes.			
7.3.7	Verify operation of alternate static source.			
7.3.8	Monitor magnetic compass and verify proper operation.			
8	**VM 1000FX** Verify proper operation of the VM1000FX to include:			
8.1	Percent Power (Normal 0% to 100%)			
8.1	RPM (Normal 825 to 2800 RPM)			
8.2	Cylinder Head Temperature (Normal 240° to 420°F)			

Post Inspection Check Flight Procedure				
Step	**Step Description**	**Pass**	**Fail**	**Initials**
8.3	Exhaust Gas Temperature (Normal 1000° to 1675°F)			
8.5	Fuel Pressure (Normal 25 to 98 PSIA)			
8.6	Oil Pressure (Normal 30 to 60 PSIG)			
8.7	Bus Voltage (Normal 12.0 to 14.3 Volts)			
8.7	Manifold Pressure (Normal 15 to 29.5 In.Hg.)			
8.8	Oil Temperature (Normal 75° to 220° F)			
8.9	Main Bus Current (Normal 3 to 48 Amps)			
9	**Stall Warning Indicator**			
9.1	Verify that the audible stall warning actuates 5-10 knots above charted stall airspeed.			
10	**Communication and Avionics**			
10.1	Garmin GMA 340 Audio Panel			
10.1.1	Verify proper operation of the audio control panel.			
10.1.2	Verify ability to transmit, receive, and monitor radio/intercom/navaids from both headset/microphone jacks.			
10.2	COM1			
10.2.1	Verify the proper operation of COM1.			
10.3	COM2 (if installed)			
10.3.1	Verify the proper operation of COM2			
10.4	NAV1 (if installed) - Garmin GNS 430/530 (or installed navigation receiver)			
10.4.1	Verify proper operation of NAV1. VOR/ILS/GPS as applicable to radio equipment option installed.			
10.5	NAV2 (if installed) - Garmin GNS 430/530 (or installed navigation receiver)			
10.5.1	Verify proper operation of NAV2 VOR/ILS/GPS as applicable to radio equipment option installed			
10.6	Transponder – Garmin GTX 327 (or installed transponder)			

Post Inspection Check Flight Procedure				
Step	**Step Description**	**Pass**	**Fail**	**Initials**
10.6.1	Verify proper operation of the transponder.			
10.6.2	Verify proper operation of the altitude encoder			
11	**Descent, Approach and Landing** Note the following items and accomplish normal checklists per the AFM.			
11.1	Accomplish a normal approach and landing and verify normal handling qualities.			
11.2	Verify controllability during landing phase and lack of nose wheel shimmy.			
11.4	Checklists accomplished with no discrepancies noted			
12	**After Landing and Engine Shutdown** Accomplish the following items and accomplish checklists per the AFM.			
12.1	Idle RPM - Verify idle RPM with fuel pump on (ON or AUTO) is 850±50.			
12.2	Fuel shutoff valve – Verify engine stops or begins to stop when the fuel shutoff valve is in the OFF position. Reset the fuel shutoff valve to the ON position once complete.			
12.3	Checklists accomplished with no discrepancies noted.			

Table 05-14 Post Inspection Check Flight Procedure

Date	#	Comments / Squawks	Action
	1		
	2		
	3		
	4		
	5		
	6		
	7		
	8		
	9		
	10		
	11		
	12		

Acceptance Date: ____________________

Acceptance Pilot: ______________________ ______________________

Print Name Signature

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Section 05-40 Supplemental Inspections

This section details other supplemental inspections that are in addition to those inspections shown in the prior sections of this chapter.

Section 40-01 Replacement of the Propeller

For complete propeller inspection requirements see manufactures requirements below.

- Maintenance Manual for MT-Propeller, MT175R127-2Ca Propeller

  Operation and Installation Manual
  E-112 (ATA 61-01-12)
  MT-Wood-Composite
  Fixed Pitch Propellers
  November 21, 2006

- Maintenance Manual for Sensenich W69EK7-63G Propeller

  Sensenich Wood Propeller Company Inc
  Wood Propellers: Installation Operation, & Maintenance
  Integral Flange Crankshafts WOOD-CF-REV-A.doc 5-20-04

Propeller bolt torques must be checked at regular intervals depending on the manufacturer of the propeller.

If installing an MT-Propeller, check the bolt torque after the first flight and after 25-hours of operation. Then check the bolt torque every 100-hours of operation or annually, which ever occurs first.

If installing a Sensenich propeller, check the bolt torque after the first flight and after 25-hours of operation. Then check the bolt torque every 50-hours of operation and annually. Additionally, if the operating humidity and/or temperature environment will change significantly for a long period of time, it is required to check propeller bolt torque at 10-hour intervals until successive torque values do not change.

Section 40-02 Replacement of Engine, Ignition, or Instruments:

If replacing the engine, engine ignition or airplane's electrical components, or cockpit instruments, check the magnetic compass for correct and accurate operation. If necessary, prepare a new correction calibration card.

For complete engine inspection requirements: Refer to Teledyne Continental Motors IOF-240-B, Maintenance Manual; M-22 (outlined below).

25-Hour Initial Operation Inspection Frequency

Perform the 25-hour Inspection under the following four circumstances:

- Twenty-five (25) hours or six months (whichever occurs first) after
- A new, rebuilt or overhauled engine is placed in service
- Replacing one or more new engine cylinders and/or piston rings
- After every 25 hours of operation until oil consumption has stabilized

50-Hour Engine Inspection Frequency

Perform the 50-hour Inspection under the following two circumstances:

- 25 hours after the 25-hour Initial Operation Inspection
- After every 50 hours of operation

100-Hour (Annual) Engine Inspection Frequency

Perform the 100-Hour Inspection under the following two circumstances:

- After each 100 hours of accumulated engine operation
- Annually, if the engine did not accumulate 100 hours of operation during the calendar year since the last 100-hour inspection

500-Hour Engine Inspection Frequency

Perform a 500-Hour Inspection under the following circumstances:

- After each 500 hours of accumulated engine operation
- After installation of any engine components, or other optional engine accessories

Additionally, during the 500-Hour Inspection, check or perform the following:

- Complete all 100-hour inspection items.
- Replace paper-type induction air filter.

Section 05-50 Unscheduled Maintenance Checks

Unscheduled inspections are only performed in the event that during flight or ground operations an incident occurs which is not part of normal flight or ground operation, and might have caused a defect to the airplane or affected airworthiness of the airplane. These inspections are primarily done in response to reports made by the pilot. Below are examples of unscheduled inspections.

Section 50-01 Sudden Engine Stoppage or Prop Strike

Refer to TCM IOF-240-B Installation and Operation Manual, OI-22, Chapter 05 and TCM IOF-240-B Maintenance Manual, M-22 Chapter 7.

Section 50-02 Foreign Object Damage to Propeller

If there is foreign object damage to the propeller, refer to the following information for the appropriate actions.

- Maintenance Manual for Sensenich W69EK7-63G Propeller

  Sensenich Wood Propeller Company Inc
  Wood Propellers: Installation Operation, & Maintenance
  Integral Flange Crankshafts WOOD-CF-REV-A.doc 5-20-04

- Maintenance Manual for MT-Propeller

  MT175R127-2Ca Propeller
  Operation and Installation Manual
  E-112 (ATA 61-01-12)
  MT-Wood-Composite
  Fixed Pitch Propellers
  November 21, 2006

- Maintenance Manual for TCM IOF-240-B, Engine

  Teledyne Continental Motors Inc.
  Continental Aircraft Engine
  Maintenance Manual Publication M-22
  latest FAA approved revision.

Section 50-03 Non-Operation of More than 30 Days

If the airplane has not been operated for 30 days or more, refer to Liberty Maintenance Manual, Chapter 10 – *Parking and Mooring* for details for long-term storage and return to service.

Section 50-04 Hard Landing Inspection

A qualified A & P mechanic, following an incidence of a hard landing, should perform this inspection.

When a Liberty XL-2 has a hard landing, the airplane is subjected to stress levels beyond those anticipated during normal operation. The component might deform plastically. This component may be identified easily by the naked eye, but still there may be some unseen deformations to the structure. The job of a qualified A & P mechanic is to disassemble the airplane to check for deformation that may have an effect on the airworthiness of the airplane.

NOTE

Distinguishing between a hard landing and a crash –

- *A "hard landing" - results in the denting, bending or buckling of airframe components*
- *A "crash" – results in the rupturing, tearing or breaking of airframe components*

The following hard landing checklists are applicable to aircraft subjected to a "hard landing" only. If the airplane has experienced a "crash", the damage maybe too extensive to make the airplane airworthy

The following are components that Liberty has identified as critical after a hard landing and need a detailed inspection. Inspection methods must be used, which are sufficient to determine whether an unsafe condition exists. As Liberty Aerospace may not know the severity of the impact, it is the responsibility of the A & P mechanic to inspect the airplane using methods that he/she considers acceptable to determine whether the airplane is airworthy. Table 05-15 shows the components that require inspection after a Hard Landing. The table divides the items into a Preliminary Assessment and a Detailed Inspection.

	Components	**Preliminary Assessment**	**Detailed Inspection**
1	Upper and Lower Cowl	✓	
2	Engine Mount Frame	✓	✓
3	Belly Panel Assembly	✓	✓
4	Propeller	✓	✓
5	Fuselage	✓	✓
6	Space Frame Assembly *(Chassis)*		✓
7	Yoke Assembly	✓	✓
8	Fuel Tank		✓
9	Landing Gear: Nose U.C. Assembly G.A.	✓	✓
10	Landing Gear: Main U/C Port and STBD Assembly	✓	✓
11	Wings	✓	✓
12	Flaps	✓	✓
13	Ailerons	✓	✓
14	Aileron Torque Tube-Weldment		✓
15	Aileron Push Rod Ladder		✓
16	Tail Ballast, if any	✓	✓
17	Rudder	✓	✓
18	Mass Balance-Horn	✓	✓
19	Tailplanes	✓	✓
20	Doors	✓	✓
21	Seats	✓	✓
22	Windscreen	✓	✓
23	Instrument Panel	✓	✓

Table 05-15 Table of Inspection Components

Section 50-05 Hard Landing Preliminary Assessment

This section highlights the areas of the Liberty XL-2 for further Detailed Inspection. There are many aspects to evaluate when a Liberty XL-2 goes through a hard landing. Before making a detail inspection, the complete airplane should be inspected from Forward to Aft, and Starboard to Port. You may or may not come across deviations while performing a Preliminary Assessment; still the Detail Inspection must be performed. Components tabled earlier are discussed in detail below:

1. **Upper and Lower Cowl:** Lower cowl may be subjected to scratches or local dents from the nose leg hitting the clearance hole. Inspect any paint removal. Inspect the same on upper cowl. Look for any loose camlocs and clearance/interference between propeller hub and cowls.
2. **Engine Mount Frame:** Check for any loose, bent, cracked, or broken mounting bolts on the engine frame, along with paint removal. Inspect the rubber engine mounts on the top of the engine for any deformations, cracks, or breaks in the engine mounts. Also, inspect the engine mounting frame for any bent, cracked, or broken frame members.
3. **Belly Panel Assembly:** Inspect vapor zone hood assembly if applicable or vent in the belly panel, check if it is intact.
4. **Propeller:** Inspect propeller blades and look for any indentation marks due to impact. Look for any paint removal along with foreign object deposited on blades. (During hard landing blades are more vulnerable to damage and susceptible to ground impact.) Check for ‘water marks’ on blades that show an impact has taken place.
5. **Fuselage**: Inspect **all bondlines** along upper and lower fuselage, inspect for any paint removal or flaking, any dents, cracks, or damage (lower AFT portion of fuselage may hit ground. If there is a reverse movement during hard landing, lower aft portion is more vulnerable to hit hard on the ground after nose).
6. **Space Frame (Chassis):** See Detail Inspection.
7. **Yoke Assembly:** Inspect for any cracks in yoke weldment and yoke-passenger-side stick along with boot.
8. **Fuel Tank**: See Detail Inspection.
9. **Landing Gear: Nose Under Carriage Assembly G. A.:** Check for any scrub marks on tire along with its treads or castor assembly. Also look for any tire marks on the underside of nose leg. (If tire marks are found, it suggests that during hard landing tire has touched the nose leg and there might be a plastic deformation in the bend region.) Inspect nose gear leg, inspect castor weldment for any paint removal or flaking. (If paint has flaked, this suggests that it was subjected to excessive stress and it implies that the plate could be bent in that region.)

10. **Landing Gear: Main Under Carriage Port and Starboard Assembly:** Inspect geometry of main gear legs. Check for any scrub marks on the tire along with its treads. Inspect main gear leg, inspect for any paint removal or flaking.
11. **Wings:** Inspect for any scratches, dents or paint removal on wing and also check for any loose or black working ‘smoking’ rivets.
12. **Flaps:** Inspect for any scratches, dents or paint removal on flaps and check for any loose or black working ‘smoking’ rivets.
13. **Ailerons:** Inspect for any scratches, dents or paint removal on aileron and check for any loose or black working ‘smoking’ rivets.
14. **Aileron Torque Tube-Weldment:** See Detail Inspection.
15. **Aileron Push Rod Ladder:** See Detail Inspection.
16. **Tail Ballast (if any):** Inspect tail ballast assembly to see if weights have shifted or holding bracket has become loose.
17. **Rudder**: Inspect for any scratches, dents or paint removal on rudder and check for any loose or black working ‘smoking’ rivets.
18. **Mass Balance-Horn:** Inspect all mass balance structures for deformation along with scratches or paint removal.
19. **Tailplanes:** Inspect for any scratches, dents or paint removal on tailplanes and also check for any loose or black working ‘smoking’ rivets.
20. **Doors:** Inspect for any loose parts along with paint removal or flaking.
21. **Seats:** Inspect for any buckling of seat base and inspect seating to see if it has been submarined or is spongy.
22. **Windscreen**(windshield)**:** Check for any cracks on windshield. Inspect fasteners: (rivets or screws), and check for any de-bonding of the adhesive between windshield and fuselage.
23. **Instrument Panel:** Check for any cracks on instrument panel console material or instrument panels.

Section 50-06 Hard Landing Detailed Inspection

At the discretion of the Airframe & Powerplant Mechanic, (A&P Mechanic), or , if there is damage noted during the hard landing preliminary assessment, the following hard landing detail inspection is mandatory. Perform a detailed inspection by dismantling the airplane and **inspecting all components** mentioned before. An A&P Mechanic must record all discrepancies and report these to Liberty Aerospace, Inc. Customer Service.

1. **Upper and Lower Cowl:** See Preliminary Assessment.
2. **Engine Mount Frame**: After removal of engine from engine mount frame, it is easy to visually inspect the engine, firewall and engine mount frame. Look for any paint removal on the frame or any structural damages. Inspect the engine separately, and remove engine mount frame from Space Frame Assembly. Remove the bolts connecting engine mount frame to the Space Frame Assembly. Then insert the bolts. If the bolts insert smoothly, this suggests there is no bend in the bolts. Check all engine block mount lugs and inspect them for any bend that could be caused due to hard landing. Inspect each engine mount frame tube for any bends and paint removal. Paint removal suggests a plastic deformation in that area/section.
3. **Belly Panel Assembly:** Inspect vapor zone hood assembly. If applicable, or vents in the belly panel, verify they are intact.
4. **Propeller**: Inspect rotation of propeller blades. Rotation should be in a vertical plane. Any off-plane rotation indicates damage to propeller.
5. **Fuselage**: Inspect upper and lower fuselage for any damages and paint removal. Perform a tap test; (the area with any discrepancy should have a hollow sound). Check all 16-point contacts of chassis to fuselage for any deformation.
6. **Space Frame Assembly (Chassis):** After removal of fuselage and landing gear, do a visual inspection of the chassis; inspect that the chassis main bars are parallel to the ground without any bends. Dismantle chassis and reinsert the bolts. (A smooth process suggests that there is no bend in the bolts, this process should be smooth.) Check for bends and color removal in each tube. The nose leg should pass through the chassis connection, smoothly suggesting no bend in the concerned area. The main legs should pass through the chassis connection, smoothly suggesting no bend in the concerned area.
7. **Yoke Assembly:** (weldment and yoke-passenger side stick)**:** Inspect for any cracks in yoke and passenger weldment along with boot. Inspect control column motion. (After a hard landing, it has an ability to impact the instrument panel.) Movement of control column should be free, full and in correct sense.
8. **Fuel Tank**: Inspect for any dents/damages to fuel tank. Also, check the tank supports on the AFT portion of Space Frame Assembly. Check the tank for leaks or sweating.

9. **Landing Gear: Nose Under Carriage Assembly G. A.**: Visually look into the structural symmetry of the nose gear along with castor weldment and bearing housing. In a standing position, upper and lower faces of the bearing housing should be parallel to the ground and side faces should be perpendicular. The nose leg should assemble with the bearing housing in a vertical direction. Check tire rotation. It should rotate smoothly without touching castor weldment in pitch axis, inspect for yaw motion. Tire pressure should be 50(-0/+2) psig (cold inflation pressure). Remove axle and reinsert it. (This procedure should be smooth, if there is any kind of obstruction, it suggest that there is a bend in the axle.) Perform same procedure for bolts in bearing housing. In addition, nose leg should pass vertically through the bearing housing without any obstruction. Inspect weldment of nose leg with chassis. Paint flaking suggests that weldment has been affected. Check nose lock pin has not been sheared off and is in place. Also check nose wheel shimmy damper and see if it is tight.
10. **Landing Gear: Main Under Carriage Port and Starboard Assembly:** As with the nose gear, perform an inspection on the main gear legs and assembly. Detach and attach the gear. This process should be smooth. Similarly inspect for main gear under carriage attachments along with the nuts and the bolts, and confirm the tire pressure. Tire pressure should be 50(-0/+2) psig (cold inflation pressure).
11. **Wings:** Inspect the wing structure for loose rivets on upper and lower wing skin surfaces. Inspect for paint removal. Inspect the linkage mechanism for damage, verify proper operation.
12. **Flaps:** Inspect for any scratches, dents or paint removal on flaps and also check for any loose or black working ‘smoking’ rivets.
13. **Ailerons:** Inspect the linkage mechanism along with the rivets. Check for the position of mass balance. Verify proper operation. Check for excessive aileron hinge wear.
14. **Aileron Torque Tube-Weldment**: Inspect for any bends in the torque tube. Inspect the alignment of torque tube with the spigot bearing and the drive arm. Also inspect for any elongation of control circuit holes.
15. **Aileron Push Rod Ladder:** Inspect the alignment with the aileron drive arm. Inspect its functionality. Also, inspect for any elongation of control circuit holes.
16. **Tail ballast (if any):** Inspect the tail ballast assembly to be sure weights have not shifted and the holding bracket bolts are tight. Align weights and tighten bolts if necessary.
17. **Rudder**: Along with rivets, inspect the linkage mechanism. Due to the shock created during hard landing, rivets may come loose. Verify proper operation.
18. **Mass Balance Horn:** Remove the mass balance horn and inspect the assembly for any shift of the mass balance. Inspect the linkage mechanism along with the rivets. Verify proper operation.

19. **Tailplanes:** Inspect for any shift in the tailplanes mass balance. Inspect every aspect of this part. Verify proper operation including the external and internal components.
20. **Doors:** Verify doors lock and unlock properly. Remove hinges and re-install them. Removal and installation procedure should be smooth. Do a visual inspection on door screens for cracks.
21. **Seats**: Remove all cushions, and check for any dents and buckling of seat base or seat back.
22. **Windscreen** (windshield)**:** Check for any cracks on windshield. Inspect fasteners: (rivets or screws) and check for any de-bonding of the adhesive between windshield and fuselage.
23. **Instrument Panel:** Inspect to see if all controls are performing to standards. Check for any loose and or damaged wiring connections to instrument panel.

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CHAPTER 06

DIMENSIONS AND AREAS

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Table of Contents

SECTION 06-00 GENERAL 5

SECTION 00-01 THREE VIEWS OF AIRPLANE 5
SECTION 00-02 AIRPLANE DIMENSIONS 7

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Section 06-00 General

This chapter has the overall dimensions of the airplane. Dimensions given in this chapter are in feet and inches (and tenths of inches if necessary for precision).

Section 00-01 Three Views of Airplane

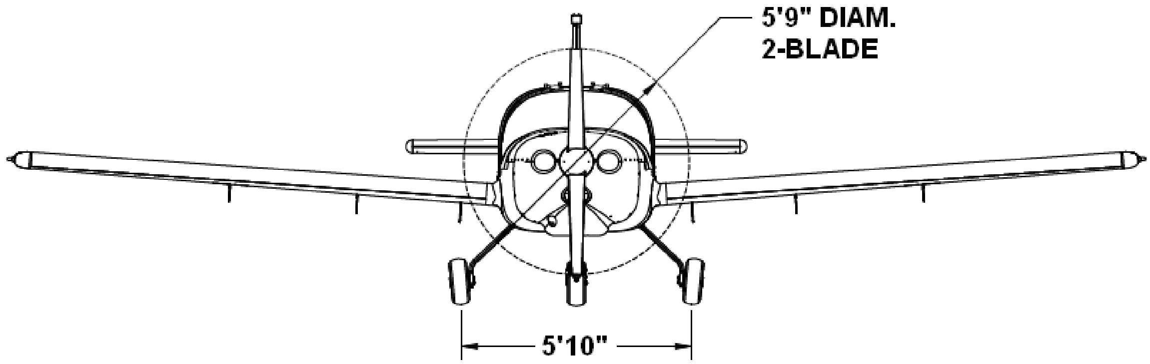

Figure 06-1 (Reference Only) Front View of Airplane Dimensions *

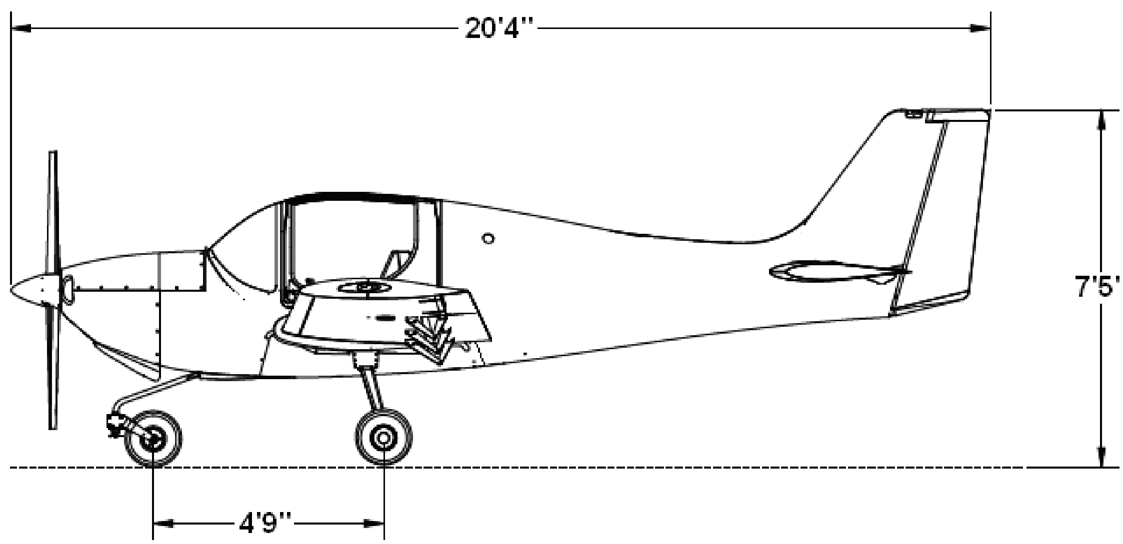

Figure 06-2 (Reference Only) Side View of Airplane Dimensions *

* - Landing gear measurements are nominal figures. Refer to Chapter 08 for procedure to measure wheelbase for a particular airplane.

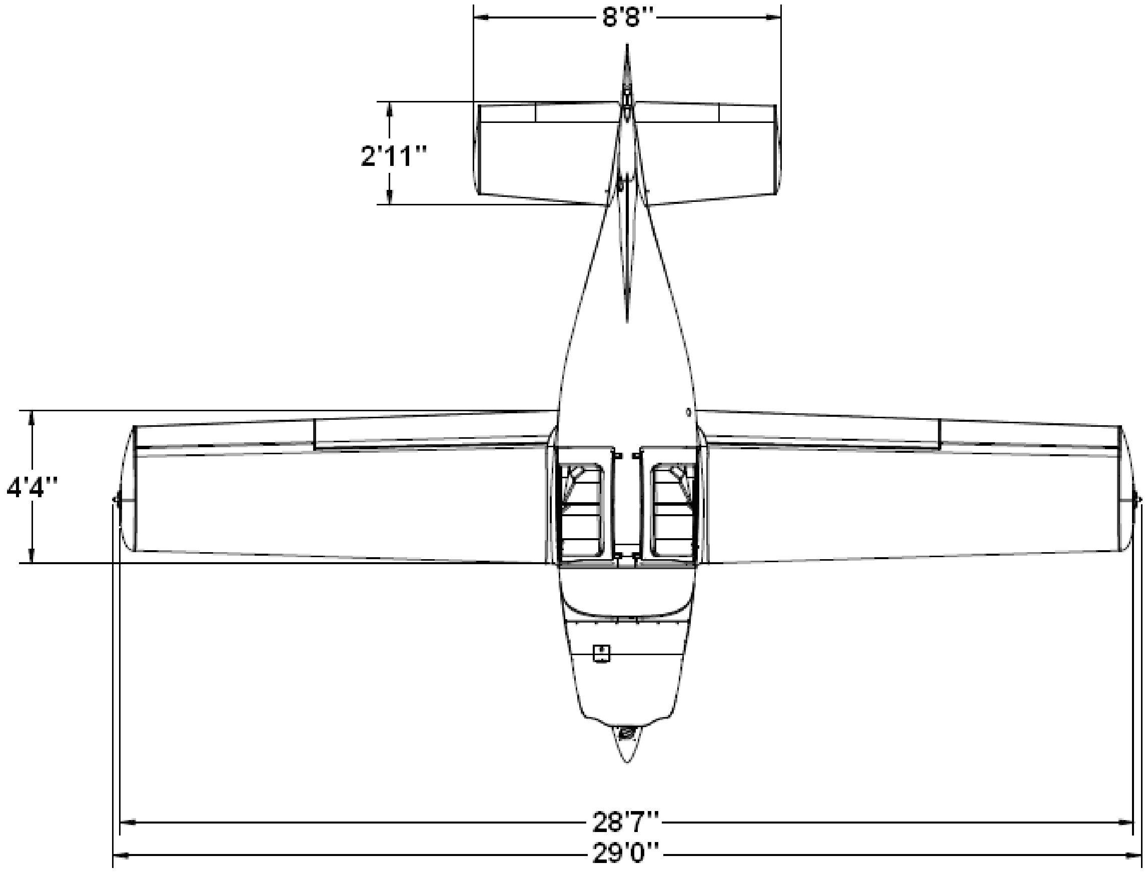

Figure 06-3 (Reference Only) Top View of Airplane Dimensions

Section 00-02 Airplane Dimensions

ITEM	DIMENSION
Overall Dimensions	
Wingspan (without anti-collision lights)	28 ft. 7 inches
Wingspan (including anti-collision lights)	29 ft. 0 inches
Airplane	
Length	20 ft. 4 inches
Height	7 ft. 5 inches to top of Vertical Stabilizer
Ailerons:	
Area	3.89 ft^2
Deflection	24° ± 1° Up, 19° ± 1° Down
Wings	
Airfoil	Liberty Custom Airfoil
Wing Area	112 ft^2
Mean Aerodynamic Chord (MAC)	4.0 ft.
Aspect Ratio	7:1
Dihedral	4.25° ± 0.1°
Sweep of leading edge	1.14°
Wing Flaps	
Area	8.78 ft^2
Deflection	0° Up, 29° ± 1° Down
Horizontal Tail Surfaces (Stabilators)	
Area	23.7 ft^2
Deflection (Nose Down)	13° ± 0.5°
Deflection (Nose Up)	5° ± 0.5 °
Anti Servo/Trim Tab Area	2.88 ft^2
Anti Servo/Trim Tab Deflection (Up)	5° ± 0.5°
Anti Servo/Trim Tab Deflection (Down)	5° ± 0.5°
Vertical Tail Surfaces (Fin and Rudder)	
Fin Area	11.2 ft^2
Rudder Area	5.65 ft^2
Rudder Deflection (Left)	30° + 0.5°/-1.5°
Rudder Deflection (right)	30° + 0.5°/-1.5°
Landing Gear Wheelbase (Measured from STA 0)	
Main wheels	7 ft. 11.9 inches*
Nose wheel	3 ft. 2.7 inches*

* - Landing gear measurements are nominal figures. Refer to Chapter 08 for procedure to measure wheelbase for a particular airplane.

Table 06-1 Airplane Dimensions

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CHAPTER 07

LIFTING AND JACKING

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Table of Contents

SECTION 07-00 GENERAL 5
SECTION 00-01 REQUIRED EQUIPMENT 5
SECTION 07-10 LIFTING AND JACKING 7
SECTION 10-01 LIFTING AND LOWER THE AIRPLANE 7
LIFTING THE AIRPLANE 8
LOWERING THE AIRPLANE 9
SECTION 10-02 JACKING THE AIRPLANE: 10
JACKING THE AIRPLANE (ALL THREE WHEELS) 11
JACKING THE AIRPLANE (MAIN GEAR ONLY) 12
LOWERING THE AIRPLANE 13

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Section 07-00 General

The airplane can be raised from the ground by lifting from above or by jacking from below. The preferred method is to jack the airplane. Lifting or jacking may be necessary for purposes of leveling and weighing the airplane (see Chapter 08 – *Leveling and Weighting*), servicing, or replacing the landing gear or components (see Chapter 32 – *Landing Gear*).

NOTE

While jacking or lifting, it may be necessary to support the nose landing gear area, with a padded block assembly, placed under the frame and aft of the attach point for the nose gear.

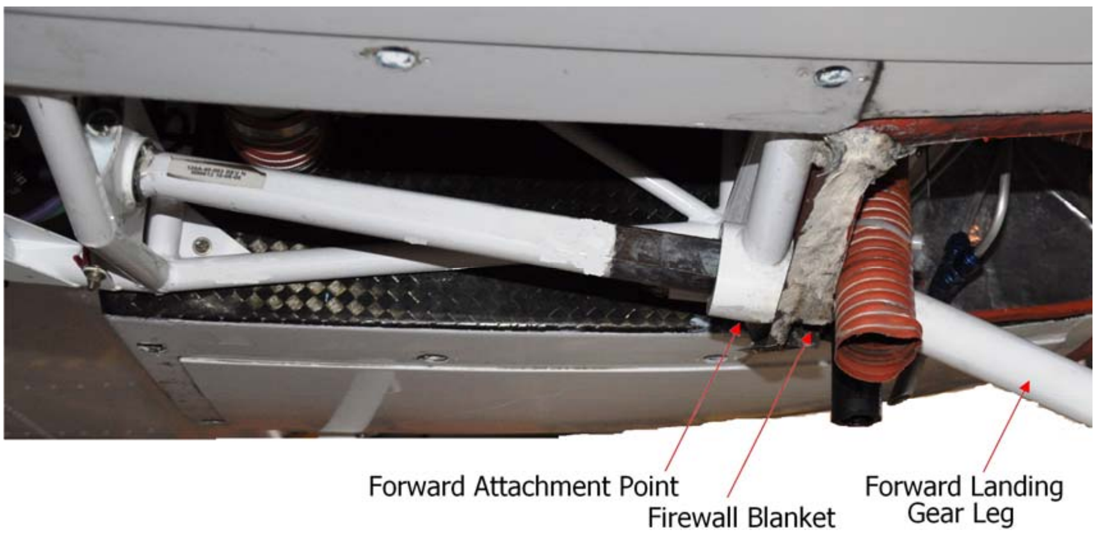

Figure 07-1 Location of the Forward Attachment Point

Section 00-01 Required Equipment

This section details the equipment required to either lift or jack the airplane.

The following is a list of required equipment needed for lifting the airplane

- A crane or overhead hoist, with the ability to lift and support 2,000 lbs
- Two nylon webbing straps, 2,000 lb capacity, and a minimum 10 - ft length; adjusting buckles are preferred
- Two tie off straps
- An assortment of felt or foam padding to avoid damage to airplane

The following is a list of required equipment needed for jacking the airplane

- Two or three hydraulic aircraft jacks with conical depression, capable of supporting 1,000 lbs each
- An aircraft tail support or padded sawhorse of the appropriate height with a weight capacity of a minimum of 300 lbs

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Section 07-10 Lifting and Jacking

This section details the procedures for lifting and jacking the airplane.

Section 10-01 Lifting and Lower the Airplane

This section details the procedures for lifting and lowering the airplane.

CAUTION

When lifting by the fuselage, as described below, the airplane is not supported laterally, and can sway. Therefore, all lifting should be performed indoors only, (to stay out of the wind), and personnel should be available to steady the airplane at its wings and tail during the entire time it is lifted.

CAUTION

Check the nylon strap placed around the fuselage does not lay across either the OAT probe, belly antenna, or front landing gear assembly.

NOTE

Most landing gear components, can be replaced while the airplane is raised on jacks. Lifting the airplane should be used only when jacks are not available. If replacing a main landing gear, the gear to be replaced can be used to jack up the airplane, but the airframe must be blocked using padded blocks, and the jack removed from that side in order to remove that landing gear.

LIFTING THE AIRPLANE

Perform this procedure to lift the airplane

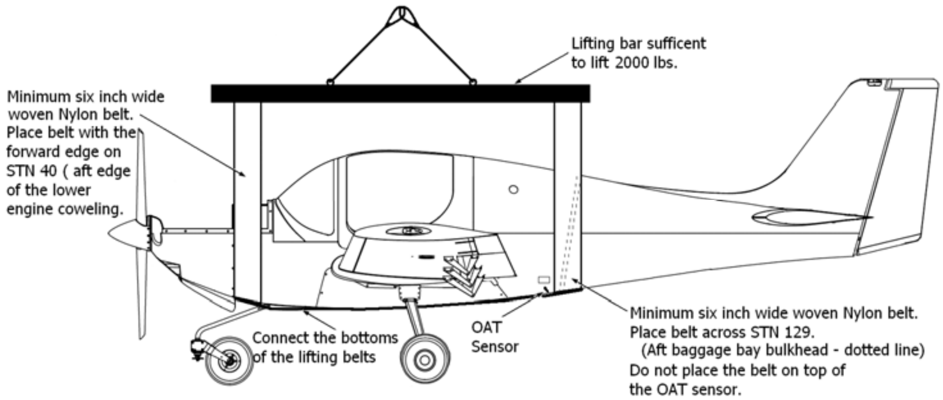


Figure 07-2 Location of the Straps to Lift the Airplane

1. Remove the upper engine cowling (see Chapter 71 – *Power Plant*).
2. The forward nylon wide-webbing strap is best placed by looping it around the lower fuselage and securing it at the top, with the front edge of the strap at Station 40. The location of Station 40 is along the aft edge of the lower engine cowling. See Figure 07-2.

NOTE

When placing the strap, be sure to locate it slightly forward of where the nose landing gear attaches to the space frame assembly.

3. The aft strap is placed around the fuselage with the center of the strap across Station 129. This will align the strap across the bulkhead at the rear of the baggage compartment. See Figure 07-2.

CAUTION

Check the nylon strap placed around the fuselage does not lay across either the OAT probe, belly antenna, or front landing gear assembly. Tie both sides of the bottom lifting strap to the main landing gear legs, as shown in Figure 07-2.

4. Tie off the bottom of the straps, securing them to both main landing gear legs. Then tie together the top of both straps, with the center of this tie off being the point for lifting the aircraft.
5. Secure the top tie off strap to the hook of the crane or hoist to be used. Adjust the length of the straps so that the fuselage remains level as the airplane is lifted.
6. Lift the airplane from the ground, employing additional personnel to steady the wings and tail.

LOWERING THE AIRPLANE

Perform this procedure to lower the airplane.

1. Check the area under the airplane that it is clear.
2. Be sure to station personnel to steady the wings and tail of the airplane.
3. Lower the airplane to the ground.
4. Complete disconnecting of all straps used to lift the aircraft.

Section 10-02 Jacking the Airplane:

For purposes of changing or servicing a single main landing gear wheel, the airplane may be jacked up on one side only, using a single jack at the applicable main gear jack point. For purposes of changing or servicing the nose landing gear wheel, the airplane may either be jacked up using a single jack at the nose gear chassis interface, or the tail may be held down and secured using a weighted tail stand attached to the tail tie-down ring. In either of the two latter cases, the main wheels remain on the ground.

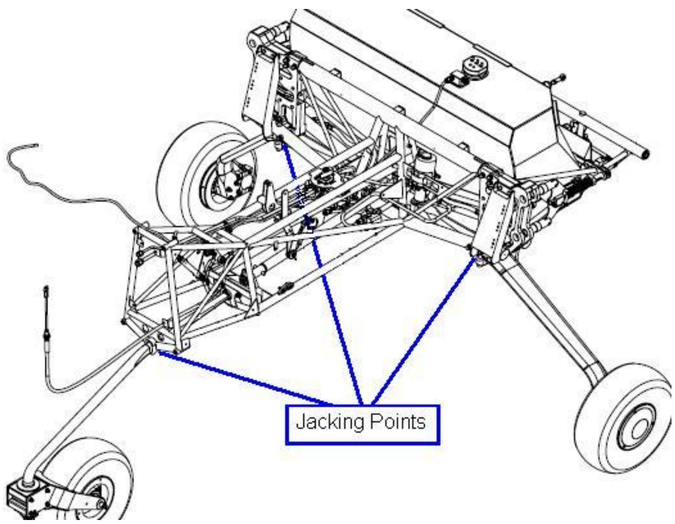


Figure 07-3 Jacking Points on the Space Frame

CAUTION

The airplane should only be jacked up indoors, in an area free from major air currents.

CAUTION

If fewer than all three wheels are to be jacked up, all wheels remaining on the ground must be securely chocked.

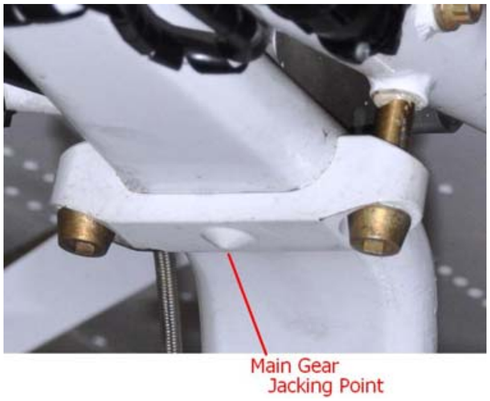
NOTE

The fuselage belly panel must be removed (see Chapter 53 - Fuselage) to gain access to the main and nose gear jack points.

JACKING THE AIRPLANE (ALL THREE WHEELS)

Perform this procedure to jack all three wheels of the airplane.

1. Remove the fuselage belly panel Refer to Chapter 53- *Fuselage* for the procedure to remove the belly panel.
2. Place a suitable jack under each main gear jack point.



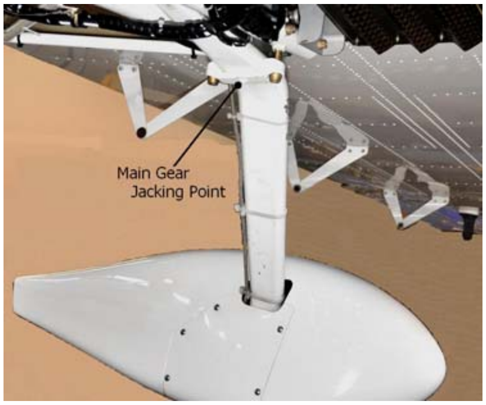


Figure 07-4 Main Gear Jacking Points

3. Place a suitable jack under the nose gear jack point.

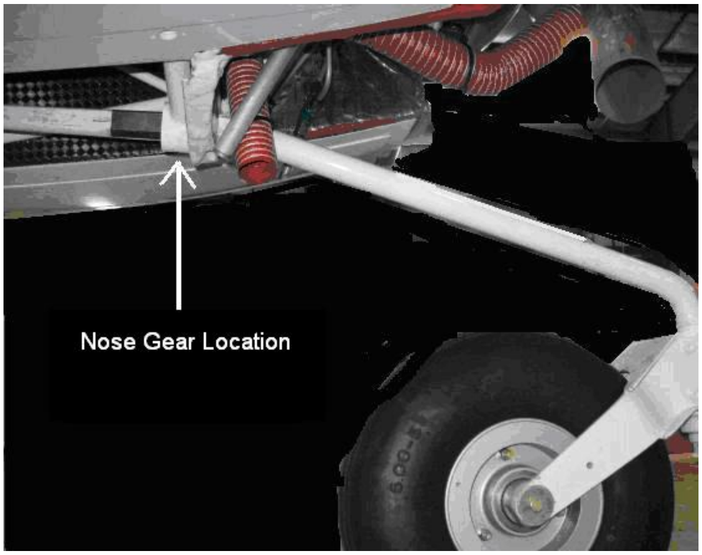


Figure 07-5 Jacking Point for the Nose Gear

4. Operate all three jacks simultaneously to raise airplane from the ground.

JACKING THE AIRPLANE (MAIN GEAR ONLY)

Perform this procedure to jack the airplane's main gear only.

1. Place a suitable jack under each main gear jack point.

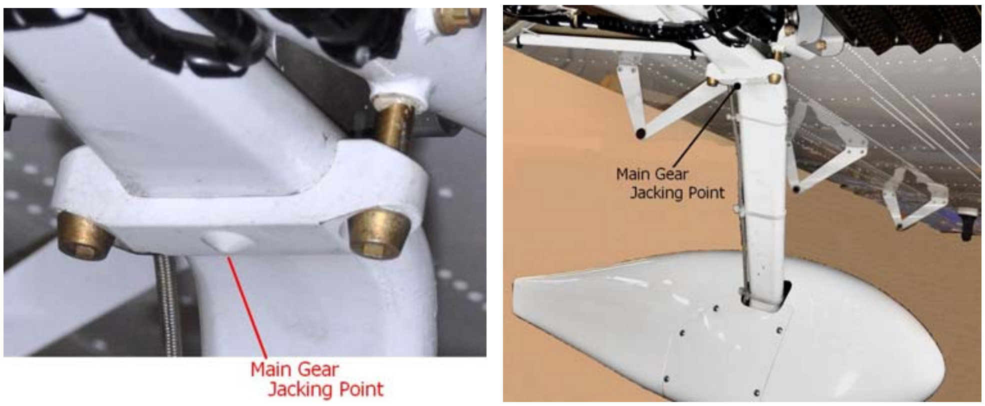


Figure 07-6 Main Gear Jacking Points

2. Secure a weighted tail stand to the tail tie-down point.
3. Operate both main gear jacks simultaneously to raise airplane from the ground.

LOWERING THE AIRPLANE

Perform this procedure to lower the airplane.

1. Check the area below the airplane that it is clear.
2. Lower all jacks simultaneously until all wheels are on the ground.
3. Remove all jacks and/or tail stand.
4. Replace fuselage belly panel. See Chapter 53 – *Fuselage*.

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CHAPTER 08

LEVELING AND WEIGHING

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Table of Contents

SECTION 08-00 GENERAL 5

SECTION 00-01 WEIGHT AND BALANCE AND THE CALCULATION FOR THE CENTER OF GRAVITY 5

SECTION 08-10 WEIGHING AND BALANCING PROCEDURES 7

SECTION 10-01 EQUIPMENT REQUIRED 7
EMPTY WEIGHT AND PREPARATION 8
LEVELING 9
MOMENT ARM DETERMINATION 10
AIRPLANE WEIGHING 12

SECTION 08-20 LEVELING 13

LONGITUDINAL LEVELING 14
ALTERNATE METHOD OF LONGITUDINAL LEVELING 15
LATERAL LEVELING 16

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Section 08-00 General

This chapter includes the information necessary to properly level the aircraft for any of the various maintenance, overhaul or major repairs which may be necessary during the life of the aircraft.

The information in this chapter comes from Liberty Aerospace, Inc. document number 135A-995-254 (currently revision A). Use the most current revision of this document as the definitive source for information when weighing and leveling the airplane and calculating the center of gravity.

It also includes those units or components that are specifically dedicated to record, store or compute weight and balance data.

The chapter includes the practices necessary to prepare the aircraft for weighing, leveling, and calculating the center of gravity. The information from this chapter needs to match the information in the Airplane Flight Manual for each airplane.

Section 00-01 Weight and Balance and the Calculation for the Center of Gravity

The result of the weight and balance calculation is a critical piece of information and needs to be very accurate for the safe operation of the airplane. The result of the weight and balance calculation is the location of the center of gravity of the airplane, or the point at which an aircraft would balance if it were possible to suspend it at that point.

NOTE

The procedures that are in Section 08-10 - Weighing and Balancing Procedures on page 7 of this chapter will provide the data that goes into the following tables.

Certain information is needed to find the center of gravity (CG or CoG) of the airplane. This is the total weight of the airplane and the moment (in in-lbs). The airplane needs weighing on three scales (adding the weights) and calculating the individual moments.

The example shown here uses data from an actual XL-2 airplane. Use this information with the data derived from the procedures to calculate the CG of a specific serial number airplane.

Table 08-1 shows the weights taken from the three scales used to weigh the airplane. This data is from the Airplane Weighing procedure on page 12 of this chapter. From these weights, it is required to subtract any items used to level the airplane (such as shims, chocks, etc.). This will give you the empty weight at each point and the total of these gives you the total empty weight of the airplane.

Data Point (lbs) / Location of Weight	Scale Reading	Weight of non-airplane items	Net Weight (W_n, W_p, W_s)
Nose Wheel (N)	305	15	290
Port Main Wheel (P)	457	22	435
Starboard Main Wheel (S)	455	18	437
		Total Weight (W_e)	1162

Table 08-1 Table Showing the Calculations for the Empty Weight of the Airplane

Table 08-2 shows the weight data from Table 08-1, the station point for a given weight, and the calculation of the moment. The moment is the weight (lbs) times the station number (in). The station points are from the Moment Arm Determination procedure on page 10 of this chapter.

The data entered into the table is calculated based on the reference station data point of 48.1. Therefore, the station number (Moment Arm) for the nose wheel (9.935 inches forward of station 48.1) is 38.2 inches (Station Reference – Distance forward of the Reference). The station for the port and starboard main wheels is aft of station 48.1, therefore, the station number (47.973 inches aft of station 48.1) is 96.1(Station Reference + Distance aft of the Reference).

Data Point / Location of Moment	Weight (lbs) (W_n, W_p, W_s)	Station (in)	Moment (in-lbs) (M_n, M_p, M_s)
Nose Wheel (N)	290	38.2	11078.0
Port Main Wheel (P)	435	96.1	41803.5
Starboard Main Wheel (S)	437	96.1	41995.7
Total Weight (W_e)	1162	**Total Moment(M_e)**	94877.2

Table 08-2 Table Showing the Calculations for the Empty Weight of the Airplane

The calculation of the center of gravity or CG is the division of the Total Moment (Me) by the total Weight (We). In this case, the CG is located at Station 81.6 or 33.5 inches aft of the Station Reference (Station 48.1).

Section 08-10 Weighing and Balancing Procedures

The following procedures are given to determine the aircraft's weight and balance. Weighing and balancing of the airplane is done in four procedures, Empty Weight and Preparation, Leveling, Determining the Moment Arm, and Weighing.

Section 10-01 Equipment Required

The following is a list of required equipment to perform the procedures in this chapter.

- Aircraft Scales
- Level – Spirit type 4 to 5 feet long
- Levels - Digital Torpedo type
- Aileron yoke blocks – P/N 135A-02-511 (requires two blocks)
- Plumb bob with string
- Measuring Tape
- Metal yard stick
- Carpenter's combination square

NOTE

The Aileron yoke blocks, P/N 135A-02-511, are available from Liberty Aerospace, Inc. Customer Service only.

EMPTY WEIGHT AND PREPARATION

The aircraft must be brought to its "empty" status to obtain the aircraft empty weight. This is accomplished as follows:

1. Verify equipment installation is in accordance with aircraft equipment records.
2. Run engine to 'wet' fuel system.
3. Open fuel tank sump drain fitting and fuel gascolator drain valve to flush system of all fuel. Close drains and add 1.5 gallons of 100 LL Avgas to represent unusable fuel.
4. Service engine oil to maximum mark on the dip stick.
5. Service the hydraulic brake fluid to top of clamp holding brake reservoir.
6. Inflate tires (3) to recommended operating pressures (50 -0/+2 psi) and align nose wheel with aircraft centerline.
7. Place all control surfaces in neutral position, with flaps fully retracted (0°).
8. Verify no baggage within baggage bay and seat back. Place the Airplane Flight Manual in aircraft right seat back.
9. Check the fire extinguisher is current and full and mounted in its carrier.
10. Check for the safety hammer and micro-fine cleaning cloth are in their proper place and secure.
11. Place all seat belts and shoulder straps within seat base.
12. Check that all interior cushions and carpets are installed. Check foot wells, seats, baggage bay, carpets and cushions are clean, and in good condition.
13. Install and secure all cowls (3) (upper engine cowl, lower engine cowl, and belly panel) and all access panels (8).
14. Rotate propeller until it is parallel to horizontal and close doors.

This completes the Empty Weight and Preparation procedure.

LEVELING

Perform this procedure to level the airplane in preparation for weighing the airplane.

1. Perform Longitudinal Leveling procedure (on Page 14) or Alternate Method of Longitudinal Leveling (on Page 15) procedure of this chapter.
2. Perform Lateral Leveling procedure (Page 16) of this chapter.

Moment Arm Determination

To calculate the center of gravity (CG) location with respect to the reference datum, one must calculate each wheel moment with respect to the reference datum. To calculate the moment, the weight and the arm is needed. The weights are provided by the scales; the arms are determined as follows;

1. Check the nose wheel that it is aligned with aircraft centerline.
2. Level the airplane per the Leveling procedure on page 9 of this chapter.
3. Drop a plumb bob from both port and starboard lower engine frame pickups. Draw reference line between these points on floor. This trace marks station 48.1 (STA 48.1). Use this reference to measure wheel locations.

Figure 08-1 Plumb Bob Dropped at Reference Location (STA 48.1)

Measure from each side of nose wheel axle and average the measurements.

4. Measure distance between each side of nose wheel axle and STA 48.1.
5. Average the measurements from step 4. Enter the number in to the cell for the nose wheel station in a table similar to Table 08-2

$$\frac{\text{Measurement 1 + Measurement 2}}{2} = \text{Average}$$

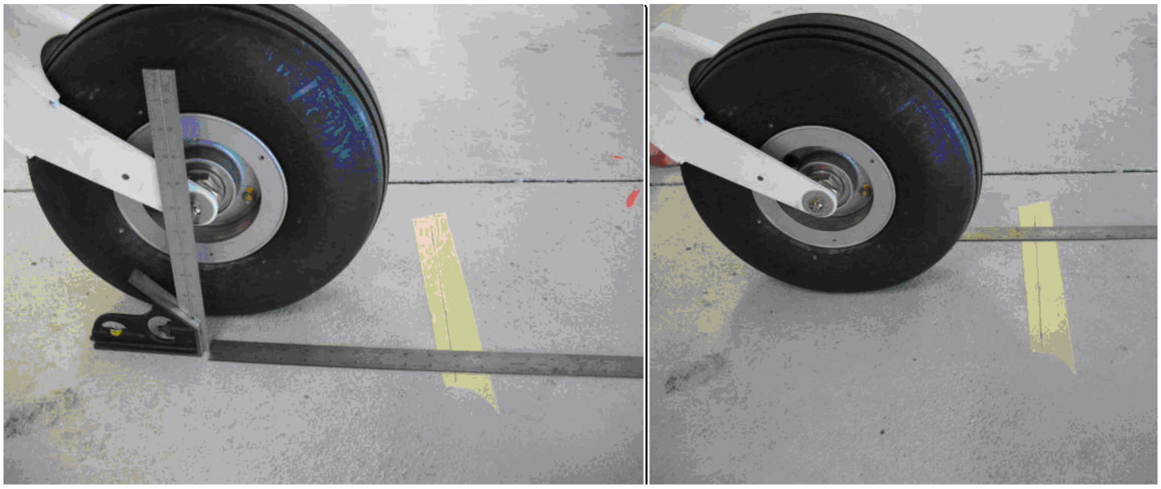

Figure 08-2 Measurement of Distance Between STA 48.1 and Both Sides of Nose Gear Axle

6. Mark floor at inside face of each main gear leg to represent port and starboard main wheels axle.
7. Measure distance between port and starboard main wheels axle and mark representing STA 48.1. Enter the number in to the cell for the port and starboard main wheel station in a table similar to Table 08-2.

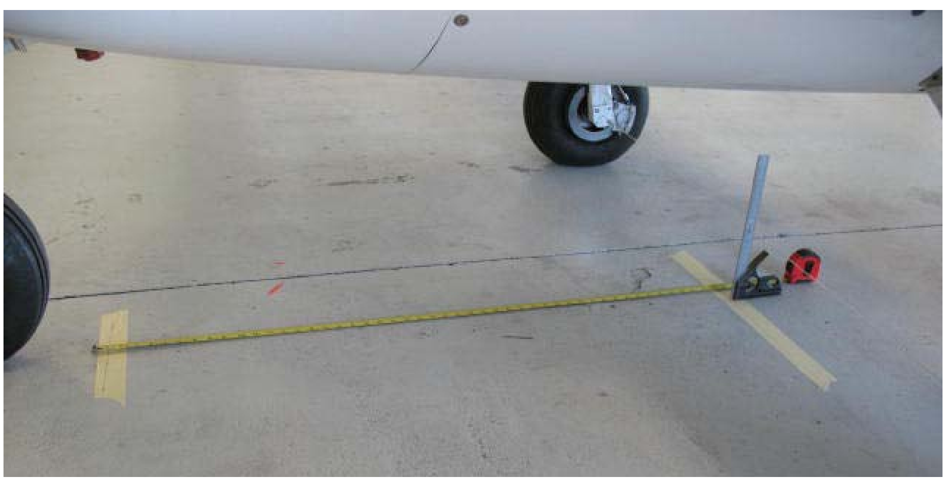

Figure 08-3 Measurement of Distance Between STA 48.1 and Main Gear Axles

This completes the Moment Arm Determination procedure.

AIRPLANE WEIGHING

Perform this procedure to weigh the airplane.

NOTE

Perform the weighing of the airplane indoors, in a location free from air currents.

1. Bring airplane to empty condition per Empty Weight and Preparation procedure on page 8 of this chapter.
2. Level aircraft on scales see Section 08-20 - Leveling on page 13 of this chapter.
3. Roll aircraft off scales.
4. Zero all scales. After all scales read zero, roll aircraft back onto scales. Check that wheels are centered on scales and apply parking brake.
5. Align nose wheel tire with aircraft centerline and verify aircraft is level.
6. Enter the weights indicated by all scales in to the cells of Table 08-1 for the Nose Weight, Port Wheel Weight, and the Starboard Wheel Weight.
7. Roll aircraft off scales.
8. Weigh the shims, wheel chocks or other items used to weigh the airplane but are not part of the airplane and were associated with the scale. Enter these weights in to the appropriate cells of Table 08-1.
9. Subtract this weights entered in to Table 08-1 from step 8 from the numbers entered in step 6.
10. Transfer these numbers from Table 08-1 in to the appropriate cells in Table 08-2

This completes the Airplane Weighing procedure.

Section 08-20 Leveling

Leveling of the aircraft for the weight and balance determination must be conducted on aircraft scales.

The port and starboard doorsills are to be used as a longitudinal support for the spirit level or digital protractor.

LONGITUDINAL LEVELING

Figure 08-4 Port Doorsill

1. Place spirit level or digital protractor on top of port or starboard doorsill.

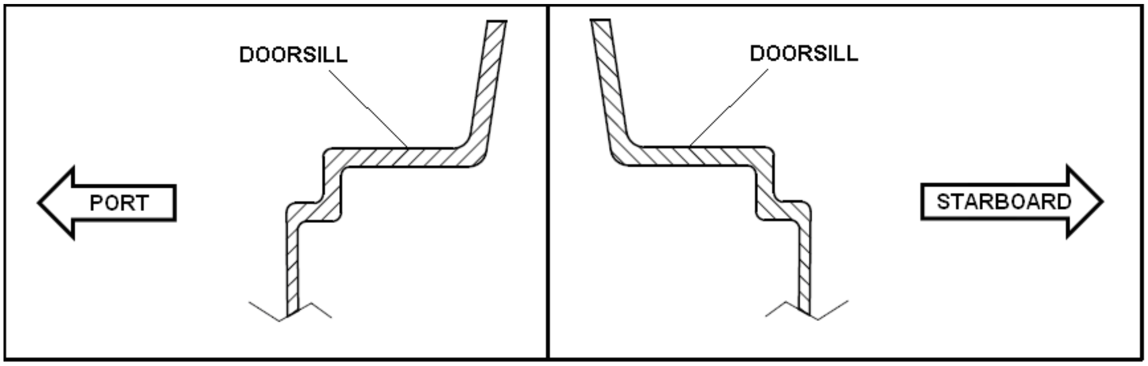

Figure 08-5 Doorsills Used in Longitudinal Leveling

2. Shim under tires to achieve longitudinal level.
3. Repeat on opposite side of aircraft to confirm level has been achieved.

This completes the Longitudinal Leveling procedure.

ALTERNATE METHOD OF LONGITUDINAL LEVELING

This alternate method of longitudinal leveling is presented as a matter of being possibly more convenient than the previous method. Both methods are equally acceptable.

1. Place spirit level lengthwise along aircraft centerline on flat area between two seat backs.
2. Shim under tires to achieve longitudinal level.

Figure 08-6 Alternate Method of Longitudinal Leveling

LATERAL LEVELING

Perform this procedure to do the lateral leveling of the airplane.

NOTE

To determine that level is placed correctly, place the level such that equal distance exists between forward end of doorsill and end of level.

1. Refer to Figure 08-7. Place a 4-ft length (minimum) beam level at approximately center of the door opening 19 to 20 inches from forward edge of door.

NOTE

Remove a portion of the door seal from the lower edge of the door opening

Place the beam level perpendicular to the aircraft's centerline.

Liberty Aerospace, Inc. recommends the use of aileron yoke blocks, P/N 135A-02-511, placed on the doorsill to support the beam level. The doorsill is a better surface for checking the lateral level of the airplane

Figure 08-7 Aircraft Laterally Leveled (Aileron Yoke Blocks Are Not Shown)

2. Shim under main gear tires to achieve lateral level.
3. After leveling is achieved, remove level. Do not disturb balance.
4. Roll aircraft off scales and shims.
5. If aircraft is to be weighed, weigh shims so their weight may be subtracted from the overall weight obtained.

This completes the Lateral Leveling procedure.

CHAPTER 09

TOWING AND TAXIING

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Table of Contents

SECTION 09-00 GENERAL 5
SECTION 00-01 EQUIPMENT AND ACCESSORIES 5
SECTION 09-10 TOWING 7
POSITIONING BY HAND 8
SECTION 09-20 TAXIING 9
SECTION 20-01 TAXIING PROCEDURES 9
TAXIING PROCEDURES (FINGER BRAKES) 10
TAXIING PROCEDURES (TOE BRAKES) 11

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Section 09-00 General

The airplane can be moved on the ground by taxiing under its own power or by towing either by hand. In view of its small size and light weight, the airplane can be moved easily by one person. For short distances, moving the airplane by hand is the preferred method. For longer distances, taxiing is the preferred method whenever feasible. Due to the possibility of damage to the nose landing gear, extreme caution should be exercised when towing the airplane with a vehicle.

Section 00-01 Equipment and Accessories

Third party sources have developed a special accessory for towing the XL-2 airplane. This accessory allows for safe and secure towing by hand. Other tow bars can cause damage to the airplane. Liberty Aerospace, Inc. does not make any recommendations on the use of any tow bar from a third party source. Use of a tow bar is at the owner’s risk.

CAUTION

A tow bar designed for the airplane is available from third party sources. Use of any locally fabricated tow bars can cause damage to the airplane. Liberty Aerospace, Inc. does not make any recommendations on the use of any tow bar from a third party source. Use of a tow bar is at the owner’s risk. If there is a need for additional information concerning a tow bars, contact the Customer Service department of Liberty Aerospace, Inc.

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Section 09-10 Towing

This section details the procedures for towing the airplane by hand.

CAUTION

If attempting to tow or move the aircraft, caution should be exercised to assure the nose landing gear is not damaged. The best way of moving the aircraft is by positioning a person, or people, in safe positions around the aircraft, making sure not to apply excessive pressure to aircraft components. In addition, use care when turning the aircraft.

CAUTION

When moving the airplane backward, the nose landing gear will tend to caster to a "hard over" left or right position. Maintain a firm grip on the tow bar to prevent the nose landing gear from contacting limit stops at 80-degree left or right position.

POSITIONING BY HAND

Perform this procedure to position the airplane by hand.

1. Remove forward and aft nose gear faring from nose gear assembly. See Figure 09-1 for an exploded view of the forward wheel fairing.

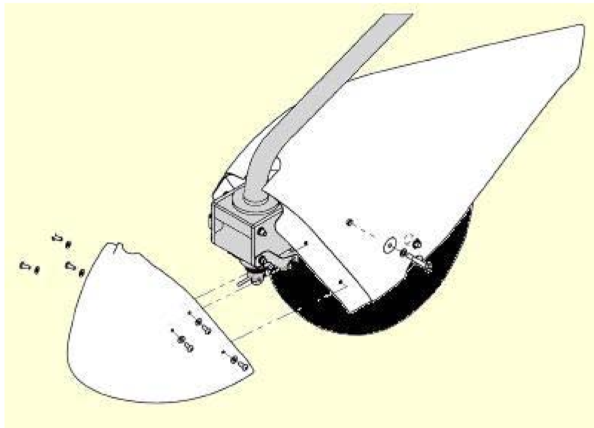

Figure 09-1 Exploded view of the Forward Wheel Fairing

2. If using a tow bar, attach the tow bar to the nose landing gear. See the note on page 5 of this chapter.
3. Remove chocks and release the parking brake.
4. Move airplane to desired position.

CAUTION

If attempting to tow or move the aircraft, caution should be exercised to assure the nose landing gear is not damaged. The best way of moving the aircraft is by positioning a person, or people, in safe positions around the aircraft, making sure not to apply excessive pressure to aircraft components. In addition, use care when turning in the aircraft.

CAUTION

When moving the airplane backward, the nose landing gear will tend to caster to a "hard over" left or right position. Maintain a firm grip on the tow bar to prevent the nose landing gear from contacting limit stops at 80-degree left or right position.

5. Chock main wheels.
6. If using a tow bar to move the airplane, remove tow bar.
7. Replace the forward and aft nose gear faring.
8. Set the parking brake.

Section 09-20 Taxiing

On the ground, control of the Liberty XL-2 is from the brakes on the main gear. The left and right finger brakes (located on the cockpit center console) or the toe brakes (mounted on the rudder controls) control the brakes. Very tight turn radii can be achieved by operation of an individual brake. A parking brake lever is also located on the cockpit center console.

Figure 09-2 Pilot's (Left) and Co-pilot's (Right) toe brakes

Figure 09-3 Parking Brake in an Airplane with Toe Brakes (Left) and with Finger Brakes (Right)

Section 20-01 Taxiing Procedures

This section contains the individual procedures for taxiing the airplane. One procedure uses the finger brakes the other uses the toe brakes.

WARNING

QUALIFIED AND PROPERLY TRAINED PERSONNEL SHOULD ONLY TAXI THE AIRPLANE.

TAXIING PROCEDURES (FINGER BRAKES)

Perform this procedure to taxi the airplane using the finger brakes.

1. Check the area around and in front of airplane that it is clear of all obstacles and foreign objects (personnel, tools, work stands, toolboxes or carts, etc).
2. Remove chocks, tie-down ropes, and tow-bar if attached.
3. Enter cockpit; secure seat belt; release parking brake; check brake operation by pulling aft on both brake levers.
4. Set the parking brake.
5. Start engine (see Liberty XL-2 Approved Airplane Flight Manual).
6. Release the parking brake.
7. Use minimum required power to taxi the airplane to desired position. Operate left brake to steer airplane left, or operate the right brake to steer right. Operate both brakes simultaneously to reduce speed or bring airplane to a stop.

NOTE

Both brake levers must have firm and equal resistance within first 2 inches of travel.

8. Stop engine (see Liberty XL-2 Approved Airplane Flight Manual).
9. Secure the airplane as necessary see Chapter 10 - *Parking*.

TAXIING PROCEDURES (TOE BRAKES)

Perform this procedure to taxi the airplane using the finger brakes.

1. Check the area around and in front of airplane that it is clear of all obstacles and foreign objects (personnel, tools, work stands, toolboxes or carts, etc).
2. Remove chocks, tie-down ropes, and tow-bar if attached.
3. Enter cockpit; secure seat belt; release parking brake; check brake operation by pushing on the individual toe brake pedals.
4. Set the parking brake.
5. Start engine (see Liberty XL-2 Approved Airplane Flight Manual).
6. Release the parking brake.
7. Use minimum required power to taxi the airplane to desired position. Operate left toe brake to steer airplane left, or operate the right toe brake to steer right. Operate both brakes simultaneously to reduce speed or bring airplane to a stop.

NOTE

Both brake levers must have firm and equal resistance within first 2 inches of travel.

8. Stop engine (see Liberty XL-2 Approved Airplane Flight Manual).
9. Secure the airplane as necessary, see Chapter 10 - *Parking*.

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CHAPTER 10

PARKING AND MOORING

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Table of Contents

SECTION 10-00 GENERAL 5
SECTION 00-01 EQUIPMENT REQUIRED 5
SECTION 10-10 PARKING 7
SECTION 10-01 PARKING PROCEDURES 7
SHORT-TERM PARKING 8
LONG-TERM PARKING 9
SECTION 10-20 MOORING (TYING DOWN) 11

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Section 10-00 General

This chapter describes parking and mooring of Liberty XL-2 airplane. Due to its small size and lightweight, the Liberty XL-2 airplane requires mooring (or tying down) when not in use. Pilots and maintenance personnel should familiarize themselves with the procedures described below to help prevent damage from nature's wind or turbulence caused by other aircraft.

Section 00-01 Equipment Required

The following is a list of equipment need to park and/or moor (tie down) the Liberty Aerospace, Inc. XL-2 airplane.

• 4 wheel chocks

• 3 screw-in mooring rings (left and right wing, and tail)

• 3 ropes (nylon or other non-shrinking/non-stretching synthetic material)

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Section 10-10 Parking

This section gives the details in parking the airplane.

Section 10-01 Parking Procedures

This section gives has the procedures for parking the airplane.

CAUTION

If wheel brakes are hot from prolonged taxi, allow brakes to cool before setting parking brake.

NOTE

Controls may be secured with ailerons neutral and horizontal stabilizers leading edge down by pulling the control stick aft as far as possible and fastening seat belt snugly around it.

SHORT-TERM PARKING

Perform this procedure for short-term parking of the airplane.

1. Taxi or tow airplane to desired parking position.
2. Align nose of airplane into the wind.
3. Ensure nose wheel is centered.
4. In windy or gusty weather, moor (tie down) the airplane, see Section 10-20 Mooring (Tying Down) on page 11 of this chapter.
5. Set the parking brake.

CAUTION

If wheel brakes are hot from prolonged taxi, allow brakes to cool before setting parking brake.

6. Place chocks in front of and behind main wheels.
7. Release the parking brake.
8. Secure flight controls in neutral position; retract flaps.

NOTE

Controls may be secured with ailerons neutral and horizontal stabilizers leading edge down by pulling the control stick aft as far as possible and fastening seat belt snugly around it.

9. Close and lock the doors.

LONG-TERM PARKING

Perform this procedure for long-term parking of the airplane.

1. Perform the steps for short-term: parking.
2. Moor (tie down) the airplane, see Section 10-20 Mooring (Tying Down) on page 11 of this chapter..
3. Install external rudder lock if available.

WARNING

ALL GUST LOCKS MUST BE REMOVED FROM THE AIRCRAFT PRIOR TO TAXI AND FLIGHT. CARE SHOULD BE TAKEN NOT TO DEFORM OR DAMAGE THE STRUCTURE DURING INSTALLATION AND REMOVAL OF THESE LOCKS. ALL DEFOMRATION, DAMAGE AND INTERFERENCE MUST BE REVIEWED BY A QUALIFIED MECHANIC OR TECHNICIAN PRIOR TO FLIGHT.

4. Install pitot/static, canopy, and propeller covers as applicable.
5. Refer to engine, electrical, and fuel system chapters of this manual for information on required servicing for long-term storage.

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Section 10-20 Mooring (Tying Down)

The airplane has three mooring points: one under each wing, and one under the tail. Mooring rings are provided to secure tie down ropes into the mooring points.

Park the airplane, see the procedures for short and long term parking of the airplane.

Attach tie-down ropes to ground tie-downs and aircraft mooring rings. Leave sufficient play or looseness in the ropes to prevent inadvertent loading of the structure. Also, if using a rope, tie a bowline knot to allow tension freedom.

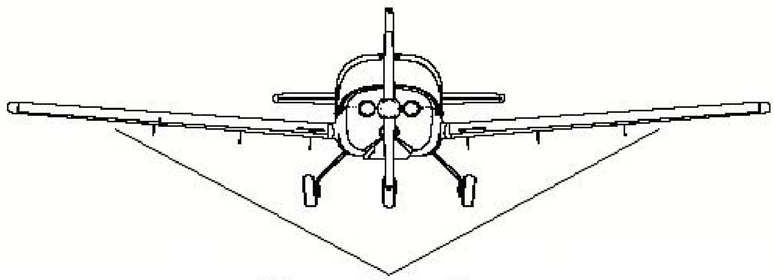

Figure 10-1 Mooring Points on the Wings

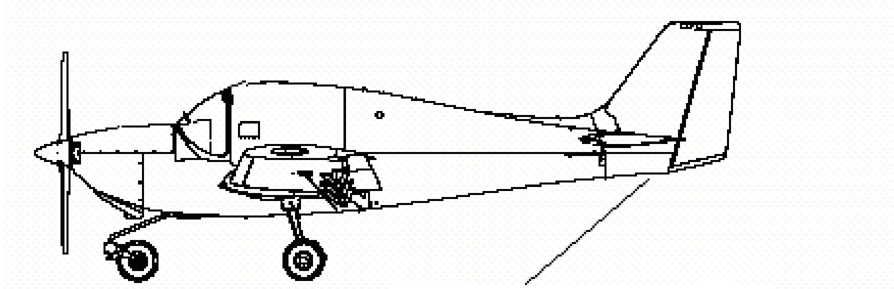

Figure 10-2 Mooring Point on the Tail

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CHAPTER 11

MARKING AND PLACARDS

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Table of Contents

SECTION 11-00	GENERAL	5
SECTION 00-01	MEASUREMENTS	5
SECTION 00-02	REMOVAL AND INSTALLATION	6
	PLACARD REMOVAL	7
	PLACARD INSTALLATION	8
SECTION 00-03	INTERNATIONAL LABELING	9
SECTION 11-10	EXTERIOR COLOR SCHEMES AND MARKINGS	11
SECTION 10-01	EXTERIOR GRAPHICS	11
SECTION 11-20	EXTERIOR PLACARDS, LABELS AND DATA PLATE	13
SECTION 20-01	LOCATION OF EXTERIOR PORT SIDE PLACARDS	13
SECTION 20-02	LOCATION OF EXTERIOR STARBOARD SIDE PLACARDS	15
SECTION 20-03	LOCATION OF EXTERIOR TOP SIDE PLACARDS	16
SECTION 20-04	LOCATION OF EXTERIOR BOTTOM SIDE PLACARDS	18
SECTION 20-05	LOCATION OF EXTERIOR DATA PLATE	19
SECTION 11-30	LOCATION OF INTERIOR PLACARDS AND LABELS	21
SECTION 30-01	MID-FUSELAGE PLACARD	21
SECTION 30-02	BAGGAGE BAY BULKHEAD PLACARDS	21
SECTION 30-03	BAGGAGE AREA	22
SECTION 30-04	INTERIOR DOOR	22
SECTION 30-05	CENTER CONSOLE PLACARDS	25
SECTION 11-40	INTERIOR PANELS	27
SECTION 40-01	INSTRUMENT PANEL	27
SECTION 40-02	AVIONICS PANEL	28
SECTION 40-03	CB (CIRCUIT BREAKER) PANEL	29
SECTION 40-04	UPPER CENTER CONSOLE PANEL	30
SECTION 40-05	LOWER CENTER CONSOLE	31
SECTION 11-50	MISCELLANEOUS PLACARDS AND LABELS	35
SECTION 50-01	BALL DRIVE ACTUATOR LABEL	35
SECTION 50-02	STEPPING SURFACES	35
SECTION 50-03	COMPASS CORRECTION CARD FILLING	36

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Section 11-00 General

This chapter describes exterior color schemes, markings, and placards. Also described are the location of placards on the exterior surfaces of the airplane and the location of placards and labels on interior surfaces. The use of placards is for identification and indication purposes. They describe function, operation, and operating limitations of various systems and equipment.

NOTE

If there is damage to a placard or the placard is unreadable, replace the placard. Most placards use a self-adhesive vinyl foil.

Section 00-01 Measurements

As a standard, the measurements for the location of placards are from three reference planes. These planes are the waterline, station, and the butt line, also known as the centerline. These planes can be hard to define or impossible to measure from. Therefore, this chapter of the manual uses measurements taken from certain key points on the airplane called the waterline reference point and the station reference point. See Figure 11-1 for the location of the waterline and station planes.

For the waterline measurements, this chapter references measurements as above or below the lower doorsill of the airplane. For station measurements, this chapter references measurements as forward or aft the forward surface of the forward bulkhead (behind the firewall assembly). See Figure 11-2 for the location of these two reference points.

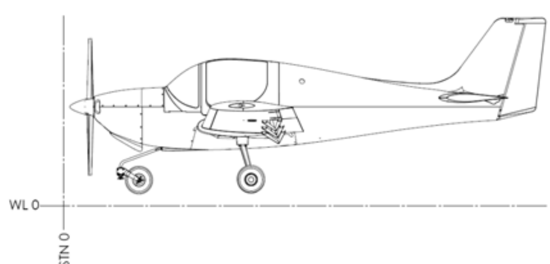
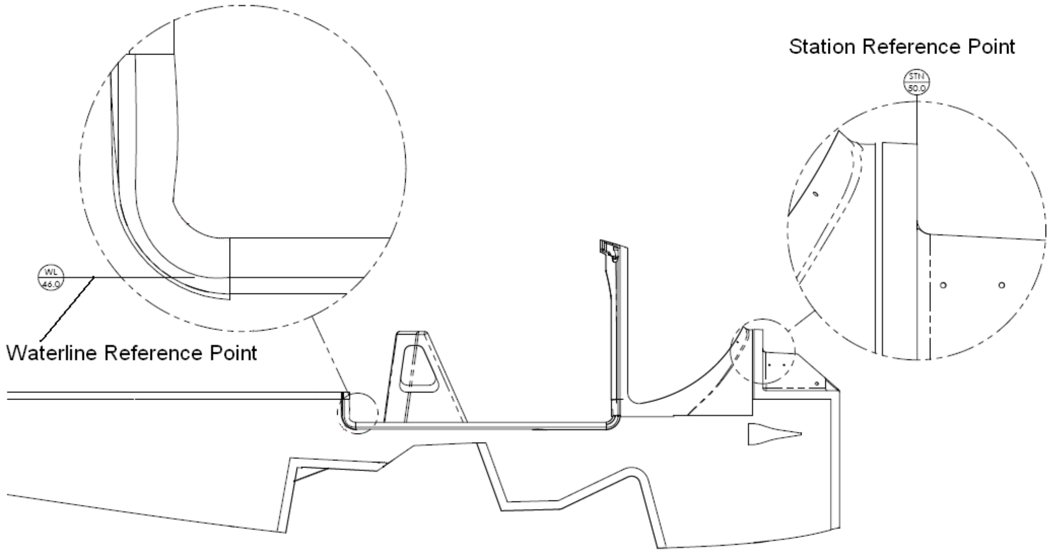


Figure 11-1 Location of the Waterline and Station Planes



Measurements in Chapter 11 -*Placards and Markings* that would normally reference the Waterline are shown as above or below of the Waterline Reference Point. This point is defined as the lower edge of the lower doorsill (WL46) of the airplane.

Measurements in Chapter 11 -*Placards and Markings* that would normally reference Station are shown as forward or aft of the Station Reference Point. This point is defined as the forward surface of the forward most bulkhead (STN50) of the airplane.

Figure 11-2 Location of the Waterline Reference Point and Station Reference Point

Section 00-02 Removal and Installation

The following are the procedures for removing and installing a placard from the airplane's surface.

PLACARD REMOVAL

Perform the following procedure to remove a placard from the airplane.

1. Remove the damaged placard.

NOTE

On the wings and horizontal stabilizers (metal surfaces), remove the old placard from any metal surfaces, such as the wings, by gently heating the old placard with a hot air blower.

On the fuselage and the vertical stabilizer, and all interior surfaces (composite or polymer surfaces), remove the old placard from any composite surface, such as the fuselage, by using a suitable plastic scraper.

2. Clean the area of sufficient size removing dirt, soil, and other foreign material with isopropyl alcohol.

NOTE

When removing any placard from the exterior surface, do not remove any of the surface finish. Remove any adhesive residual by using a surface cleaner, such as isopropyl or denatured alcohol on a soft cloth. Do not use any type of chemical solvent, such as Methyl-Ethyl-Ketone (M.E.K.) or acetone, on any of the composite or finished surfaces.

This completes the Placard Removal procedure.

PLACARD INSTALLATION

Perform this procedure to install a placard.

1. Clean an area of sufficient size removing dirt, soil, and other foreign material with isopropyl alcohol.
2. Peel off backing from self-adhesive vinyl placard.
3. Mount placard in appropriate position and appropriate orientation.
4. Use a plastic spatula to remove air bubbles from under the placard.

This completes the Placard Installation procedure.

Section 00-03 International Labeling

Some placards may be available in different languages depending on National Aviation Authorities' requirements. Figure 11-3 shows an example of one of the placards that are available with different languages. The difference of language maybe printed directly on the placard (such as the door placards) or printed on an overlay applied to another placard (such as the center console placard denoted in Figure 11-3). Table 11-1 only shows some examples of the different labels that are currently available.

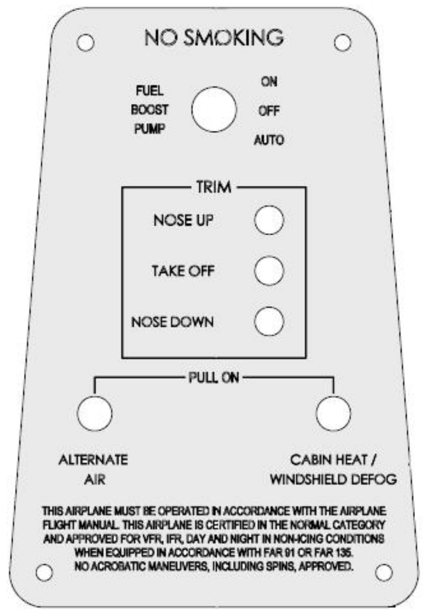


Figure 11-3 Example of one of the placards (center console placard) that is available with different languages

THIS AIRPLANE MUST BE OPERATED IN ACCORDANCE WITH THE AIRPLANE FLIGHT MANUAL. THIS AIRPLANE IS CERTIFIED IN THE NORMAL CATEGORY AND APPROVED FOR VFR, IFR, DAY AND NIGHT IN NON-ICING CONDITIONS WHEN EQUIPPED IN ACCORDANCE WITH FAR 91 OR FAR 135. NO ACROBATIC MANEUVERS, INCLUDING SPINS, APPROVED.	FAA approved in normal category based on FAR 23.
THIS AIRPLANE MUST BE OPERATED IN ACCORDANCE WITH THE AIRPLANE FLIGHT MANUAL. THIS AIRPLANE IS CERTIFIED IN THE NORMAL CATEGORY AND APPROVED FOR DAY/NIGHT VFR AND IFR IN NON-ICING CONDITIONS WHEN EQUIPPED IN ACCORDANCE WITH NATIONAL OPERATING REQUIREMENTS. NO ACROBATIC MANEUVERS, INCLUDING SPINS, APPROVED.	The FAA approved this label on behalf of the European Aviation Safety Agency (EASA). This label is also used for airplanes delivered to China.

Table 11-1 Examples of the currently available languages

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Section 11-10 Exterior Color Schemes and Markings

This section details the exterior color schemes and markings on the airplane.

Section 10-01 Exterior Graphics

The exterior of the airplane is white with distinctive color graphics. The color graphics, including the airplane's tail number, are a self-adhesive decal. Because of the design of the airplane, the location of these decals can only be in specific locations. None of the decals can cross the main horizontal bond line of the fuselage. The main horizontal bond line for the fuselage goes from the aft edge of the doorframe to the vertical stabilizer. This main horizontal bond line runs parallel to the ground from 4 inches above the waterline reference point to 5 inches above the waterline reference point. See Figure 11-4 for details on the main horizontal bond line.

The exterior of the airplane comes in one of three FAA approved graphics styles. These styles are the Vanguard design, see Figure 11-5, the Flow design, see Figure 11-6, and the Razor design, see Figure 11-7. Because of the design of the airplane, exterior graphics and the white base color of the airplane must use Liberty approved colors and decals. No other exterior graphic schemes are FAA approved and shall not be applied to the airplane.

Install registration number decals and placards on painted prepared surface, smooth away any air pockets using vinyl applicator. When ordering individual registration numbers specify 10 inch height FAA specified roman font, slant style.

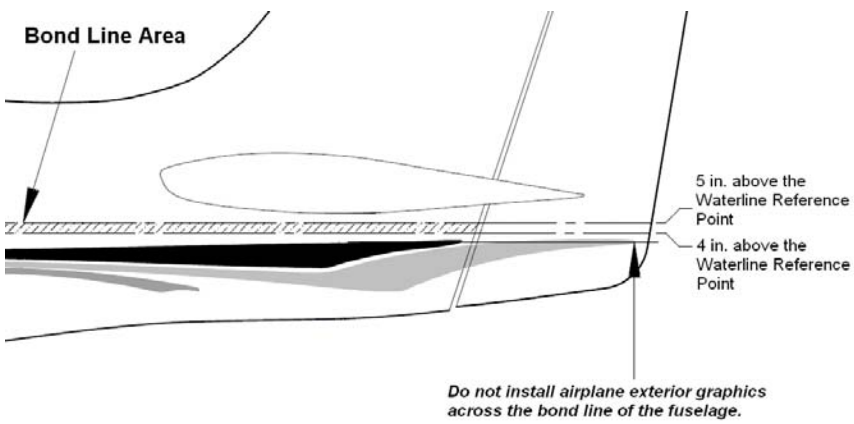


Figure 11-4 Bond Line of the Fuselage – Do not Install Exterior Graphics or Placards Across the Bond Line

NOTE

Do not install airplane exterior graphics across the main horizontal bond line of the fuselage. This bond line of the airplane goes from the aft edge of the door to the vertical stabilizer, and is in an area between 4.0 inches above the waterline reference point and 5.0 inches above the waterline reference point. See Figure 11-4 for location of the bond line.

Figure 11-5 **Exterior Graphics – Vanguard Design**

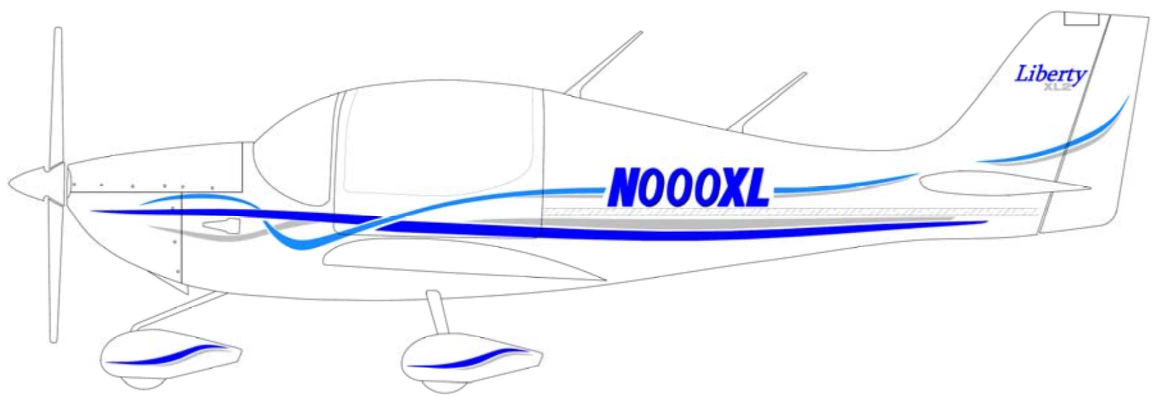


Figure 11-6 **Exterior Graphics - Flow Design**

Figure 11-7 **Exterior Graphics - Razor Design**

Section 11-20 Exterior Placards, Labels and Data Plate

This section describes the type and location of exterior placards. Located along both sides of the airplane, are various exterior placards. These placards provide information or instructions for proper operation of the airplane. This section has the following sections: Exterior Port Side, Exterior Starboard Side, Exterior Top Side, Exterior Bottom Side, and Exterior Data Plate.

Section 20-01 Location of Exterior Port Side Placards

In Figure 11-8 is the location of placards that are along the port side of the airplane. Table 11-2 is a detail explanation and examples of each of the exterior placards that are on the port side of the airplane.

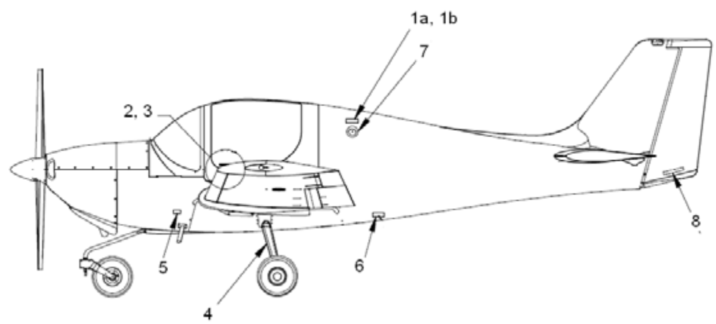


Figure 11-8 Exterior Placards – Port Side of Airplane

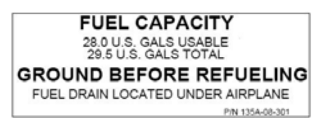
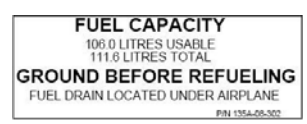
Figure 11-8 Item Number	Location	Part Number	Details
1a	Align fuel capacity placard 1.0" above round AVGAS placard	135A-08-301 Fuel capacity placard determined by aircraft purchase order	 *US delivery only*
1b	Align fuel capacity placard 1.0" above round AVGAS placard	135A-08-302 Fuel capacity placard determined by aircraft purchase order	 *Non-US delivery*
2	Located aft of the door handle	135A-08-349-* * indicates language, -1 is English only, -2 is English and Chinese -3 is English and Japanese	

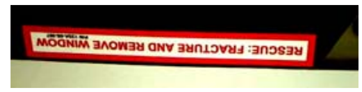
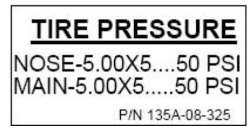
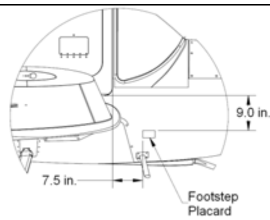
Figure 11-8 Item Number	Location	Part Number	Details
3	Located on the exterior side of the window Label is mounted such that it is inverted	135A-08-307-* * indicates language, -1 is English only, -2 is English and Chinese -3 is English and Japanese	
4	Located on the main under carriage leg 10 inches away from under carriage leg cover plate	135A-08-325	
5		135A-08-340	
6	Located aft of the wing, above the OAT Sensor	135A-09-301	
7	Located such that it surrounds the fuel filler cap	CUST-FUEL	
7	Located such that it surrounds the fuel filler cap	135A-09-303	 *Delivery to China only*
8	Located on the vertical stabilizer and below horizontal stabilizer	135A-09-315-* * indicates color, -1 is black -2 is blue	NO PUSH *This is not a single placard but a series of letters.*

Table 11-2 Exterior Placards – Port Side of Airplane

Section 20-02 Location of Exterior Starboard Side Placards

In Figure 11-9, is the location of placards that are along the starboard side of the airplane. Table 11-3 is the detail explanation and examples of each of the exterior placards that are on the starboard side of the airplane.

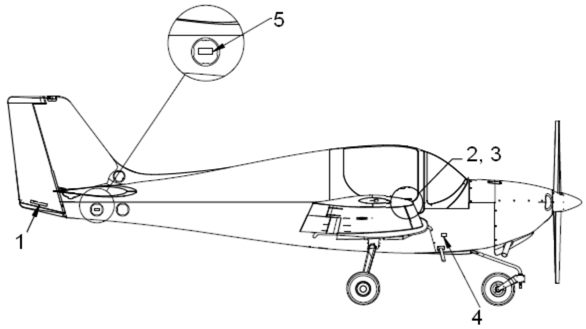


Figure 11-9 Exterior Placards – Starboard Side of Airplane

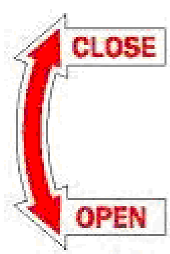
Figure 11-9 Item Number	Location	Part Number	Placard Details
1	Located on the vertical stabilizer and below horizontal stabilizer	135A-09-315-* * indicates color, -1 is black -2 is blue	NO PUSH *This is not a single placard but a series of letters.*
2	Located aft of the door handle	135A-08-350-* * indicates language, -1 is English only, -2 is English and Chinese -3 is English and Japanese	
3	Located on the exterior side of the window Label is mounted such that it is inverted	135A-08-307-* * indicates language, -1 is English only, -2 is English and Chinese -3 is English and Japanese	 *Inverted-red* 

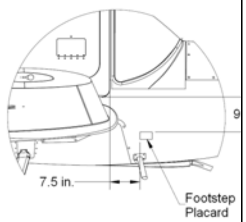
Figure 11-9 Item Number	Location	Part Number	Placard Details
4	 	135A-08-340	**WARNING:** **DO NOT ENTER OR EXIT THE AIRCRAFT WITH THE PROPELLER RUNNING** **P/N 135A-08-340**
5	Located centered in view through access hole on starboard side of the fuselage and is mounted on the inside surface of the port side of the fuselage	135A-08-397	DO NOT REMOVE OR ADD TO TAIL BALLAST WEIGHT *Only on airplanes with a gross weight of 1653 lbs.*

Table 11-3 Exterior Placards – Starboard Side of Airplane

Section 20-03 Location of Exterior Top Side Placards

In Figure 11-10, is the location of placards that are along the top of the airplane. Table 11-4 is the detail explanation and examples of each of the exterior placards that are on the top of the airplane.

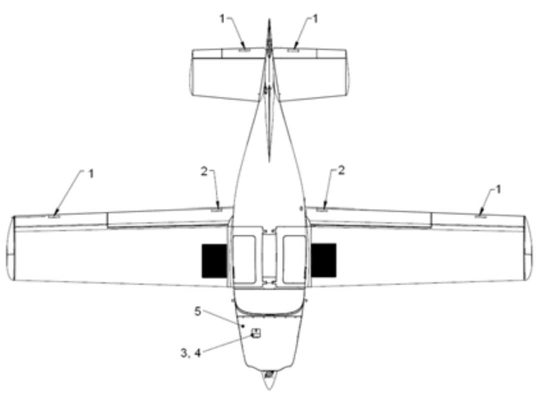


Figure 11-10 Exterior Placards – Top of Airplane

Figure 11-10 Item Number	Location	Part Number	Placard Details
1	Located on horizontal stabilizers and the ailerons	135A-09-315-* * indicates color, -1 is black -2 is blue	NO PUSH *This is not a single placard but a series of letters.*

Figure 11-10 Item Number	Location	Part Number	Placard Details
2	Located on the wing flaps near the fuselage	135A-09-313-* * indicates color, -1 is black -2 is blue	NO STEP *This is not a single placard but a series of letters.*
3	Located on the inside surface of the access hatch on the upper engine cowling	135A-08-306	
4	Located on the inside surface of the access hatch on the upper engine cowling	UXP-034	
5	Located in the engine compartment on the starboard side next to the brake fluid reservoir	135A-08-317	*original*  *updated* 

Table 11-4 Exterior Placards – Top Side of Airplane

Section 20-04 Location of Exterior Bottom Side Placards

In Figure 11-11, is the location of placards that are along the bottom of the airplane. Table 11-5 is the detail explanation and examples of each of the exterior placards that are on the bottom of the airplane.

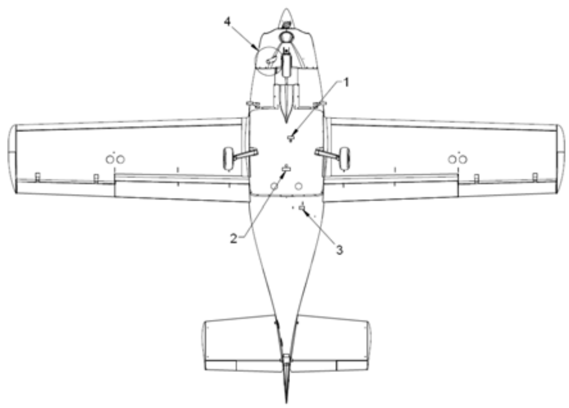


Figure 11-11 Exterior Placards – Bottom of Airplane

Figure 11-11 Item Number	Location	Part Number	Placard Details
1	Located on the belly panel assembly positioned 1 in. forward of the gascolator drain	09-43917	
2	Located on the belly panel assembly positioned 1 in. aft of the fuel tank drain	09-43917	
3	Center position 1in. aft of the fuel tank vent	135A-08-323	
4	Located on the lower engine cowling near the engine exhaust	74-1	

Table 11-5 Exterior Placards – Bottom of Airplane

Section 20-05 Location of Exterior Data Plate

The aircraft data plate, P/N 135A-00-501, is located on the port side of the airplane, below the horizontal stabilizer. See Figure 11-12 and Figure 11-13 for the location and an example of the aircraft data plate.

CAUTION

The aircraft data plate must be located below the main horizontal bond line. The bond line of the airplane goes from the aft edge of the door to the vertical stabilizer, and is in an area between 4.0 inches above the waterline reference point and 5.0 inches above the waterline reference point. See Figure 11-4 for location of the bond line.

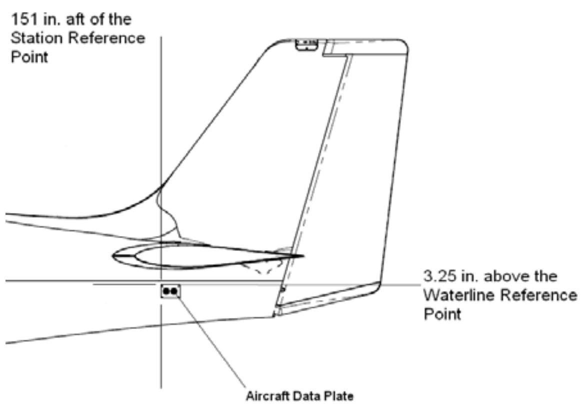


Figure 11-12 Location of the Aircraft Data Plate

Figure 11-13 Example of the Aircraft Data Plate

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Section 11-30 Location of Interior Placards and Labels

This section details the location of interior placards and labels. The interior has the following subsections: Mid-Fuselage, Baggage Bay Bulkhead, Baggage Area, Interior Door, and Center Console.

Section 30-01 Mid-Fuselage Placard

There is a single placard located on the aft side of the mid-fuselage bulkhead. See Figure 11-14 for the location of the placard, and see Figure 11-15 for a detail of the placard.

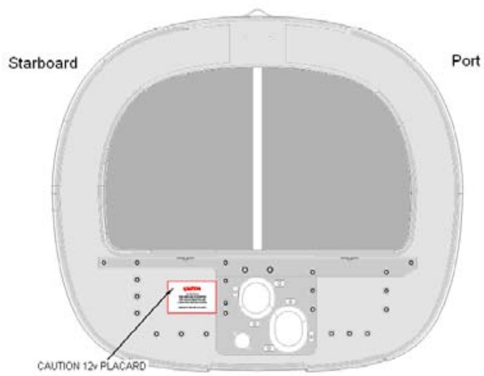


Figure 11-14 Aft side of the Mid Fuselage Bulkhead showing location of placard

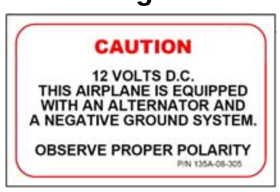


Figure 11-15 Caution 12V Placard P/N 135A-08-305

Section 30-02 Baggage Bay Bulkhead Placards

There are two placards located on aft side of the baggage bay bulkhead. These placards identify the two power relays that switch the batteries. See Figure 11-16 for the location and details of the relay placards.

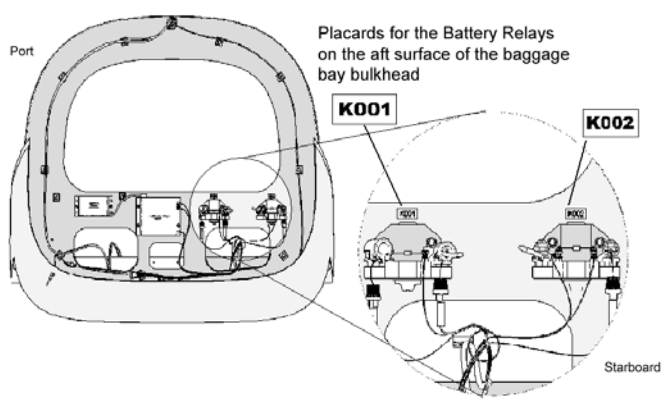


Figure 11-16 Aft side of the Baggage Bay Bulkhead showing the location and details of the relay placards

Section 30-03 Baggage Area

The baggage area has three placards or labeling. See Figure 11-17 for location of the placards. See Table 11-6 for details and a description of the location of the placards.

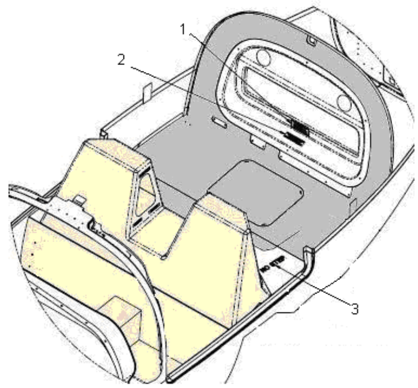


Figure 11-17 Baggage Area Showing Location of Placards

Figure 11-11 Item Number	Location	Part Number	Placard Details
1	Located on the Baggage Bay Closeout panel	135A-08-315	MAXIMUM BAGGAGE CAPACITY 100 POUNDS SEE AIRPLANE FLIGHT MANUAL WEIGHT AND BALANCE DATA FOR DETAILED LOAD INSTRUCTIONS AND APPROVED CARGO RESTRAINTS P/N 135A-08-315
2	Located on the Baggage Bay Closeout panel	135A-08-319-* * indicates language, -1 is English only, -2 is English and Chinese -3 is English and Japanese	EMERGENCY LOCATOR TRANSMITTER INSTALLED INSIDE THIS COVER MUST BE SERVICED IN ACCORDANCE WITH FAR PART 91.207 P/N 135A-08-319
3	Located on the Fuel Filler Hose Cover	135A-09-313-* * indicates color, -1 is black -2 is blue	NO STEP *This is not a single placard but a series of letters.*

Table 11-6 Baggage Compartment Placards

Section 30-04 Interior Door

The next set of figures and tables show the location of the placards on the interior side of the doors.

Figure 11-18 shows the location of the placards that are on the interior of the starboard side door. Table 11-7 gives the details of the placards and their location on the starboard side door.

NOTE

Figure 11-18 and Table 11-7 show the optional door windscreen vent. If there is no vent in the door windscreen (window), items 1 and 2 in Figure 11-18 and Table 11-7 are not included.

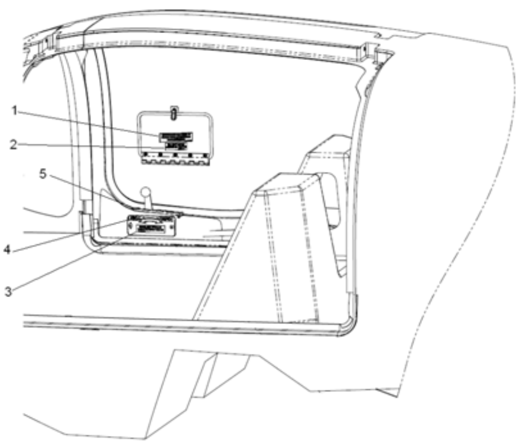


Figure 11-18 Interior of the Starboard Door Showing the Location of the Placards

Figure 11-18 Item Number	Location	Part Number	Placard Details
1	Located on the optional door screen vent If there is no vent in the door windscreen (window), this item is not included.	135A-08-373-* * indicates language -1 is English only -2 is English and Chinese -3 is English and Japanese	ENSURE DOOR VENT PANEL IS CLOSED PRIOR TO OPERATING DOOR MECHANISM 135A-08-373-1
2	Located on the optional door screen vent If there is no vent in the door windscreen (window), this item is not included.	135A-08-321-* * indicates language -1 is English only -2 is English and Chinese -3 is English and Japanese	DO NOT OPEN ABOVE 50 KNOTS 135A-08-321-1
3	Located on the door latch access plate	135A-08-341-* * indicates language -1 is English only -2 is English and Chinese -3 is English and Japanese	CANOPY RELEASE DO NOT OPEN IN FLIGHT P/N 135A-08-341
4	Located on the door latch access plate	135A-08-505-* * indicates language -1 is English only -2 is English and Chinese -3 is English and Japanese	OPEN CLOSED

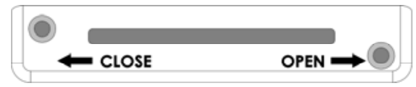
Figure 11-18 Item Number	Location	Part Number	Placard Details
5	Located on the door under the handle	135A-10-386	 *The lettering is reverse color for clarity.*

Table 11-7 Placards on the Interior of the Starboard Side Door

Figure 11-18 shows the location of the placards that are on the interior of the port side door. Table 11-8 gives the details of the placards and their location on the port side door.

NOTE

Figure 11-19 and Table 11-8 show the optional door windscreen vent. If there is no vent in the door screen (window), items 1 and 2 in Figure 11-19 and Table 11-8 these items are not included.

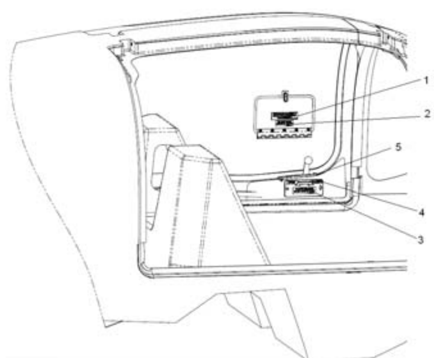


Figure 11-19 Interior of the Port Door Showing the Location of the Placards

Figure 11-19 Item Number	Location	Part Number	Placard Details
1	Located on the optional door screen vent, if there is no vent in the door windscreen (window), this item is not installed.	135A-08-373-* * indicates language -1 is English only -2 is English and Chinese -3 is English and Japanese	ENSURE DOOR VENT PANEL IS CLOSED PRIOR TO OPERATING DOOR MECHANISM 135A-08-373-1

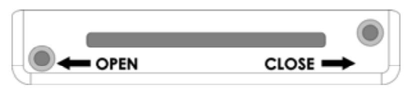
Figure 11-19 Item Number	Location	Part Number	Placard Details
2	Located on the optional door screen vent, if there is no vent in the door windscreen (window), this item is not included.	135A-08-321-* * indicates language -1 is English only -2 is English and Chinese -3 is English and Japanese	
3	Located on the door latch access plate	135A-08-341-* * indicates language -1 is English only -2 is English and Chinese -3 is English and Japanese	
4	Located on the door latch access plate	135A-08-506 -* * indicates language -1 is English only -2 is English and Chinese -3 is English and Japanese	
5	Located on the door under the handle	135A-10-385	*The lettering is reverse color for clarity.*

Table 11-8 Placards on the Interior of the Port Side Door

Section 30-05 Center Console Placards

Figure 11-20 and Figure 11-21 shows the locations of the placards on the center console of the cabin. Table 11-9 gives the details and a description of the location and the placards.

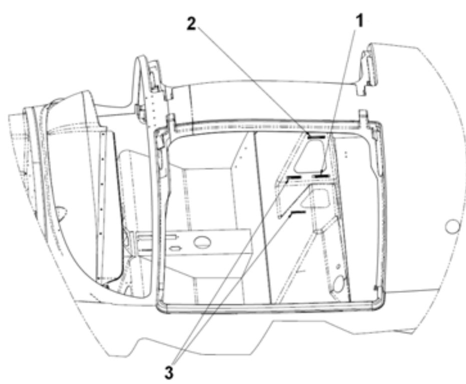


Figure 11-20 Location of the Placards on the Center Console Seat Back

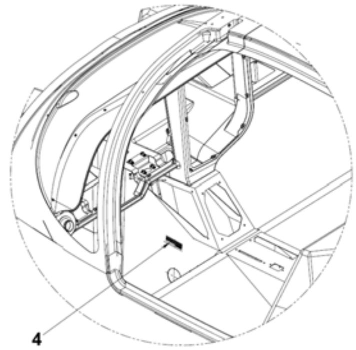


Figure 11-21 Location of the Ventilation Placard on the Center Console

Figure 11-18 Item Number	Location	Part Number	Placard Details
1	Located on the center console seat back starboard side	135A-08-346	AIRPLANE FLIGHT MANUAL LOCATED IN SEAT BACK
2	Located on the center console seat back starboard side	135A-08-327-* * indicates language -1 is English only -2 is English and Chinese -3 is English and Japanese	ACCESS TO EMERGENCY SAFETY HAMMER 135A-08-327-1
3	Located on the center console seat back starboard side	135A-08-343-* * indicates language -1 is English only -2 is English and Chinese	SOFT GOODS ONLY 2 LBS MAX 135A-08-343-1
4	Located on the center console in foot well one on each side (total of two)	135A-08-393	CLOSE BOTH CENTER TUNNEL VENTS TO DEFOG WINDSHIELD P/N 135A-08-393

Table 11-9 Interior Placards Mounted to the Center Console

Section 11-40 Interior Panels

In the interior of the airplane, there are five panels with controls and/or indicators installed in them. These panels are the instrument panel, avionics panel, the CB (circuit breaker) panel, upper center console panel, and the lower center console.

Section 40-01 Instrument Panel

The instrument panel is located on the port side of the instrument panel console assembly (dashboard), in front of the pilot's seat. Mounted to this panel are the flight indicators, an indicator for airplane engine health, ignition switch and other related controls and indicators. The instrument panel has a placard that has the callouts for the controls and indicators. Figure 11-22 shows a typical Instrument panel placard. Figure 11-23 shows the details of an instrument panel.

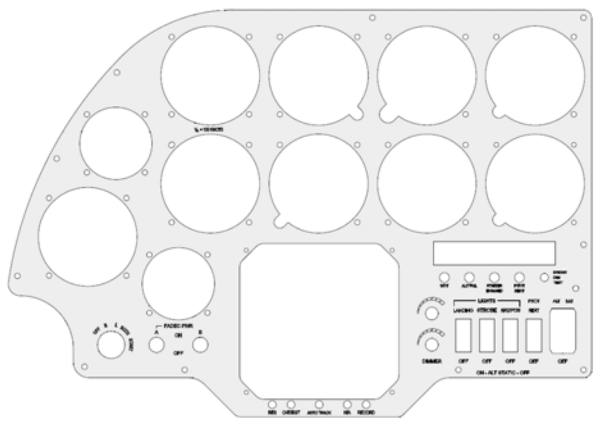


Figure 11-22 Placard for the Typical Instrument Panel

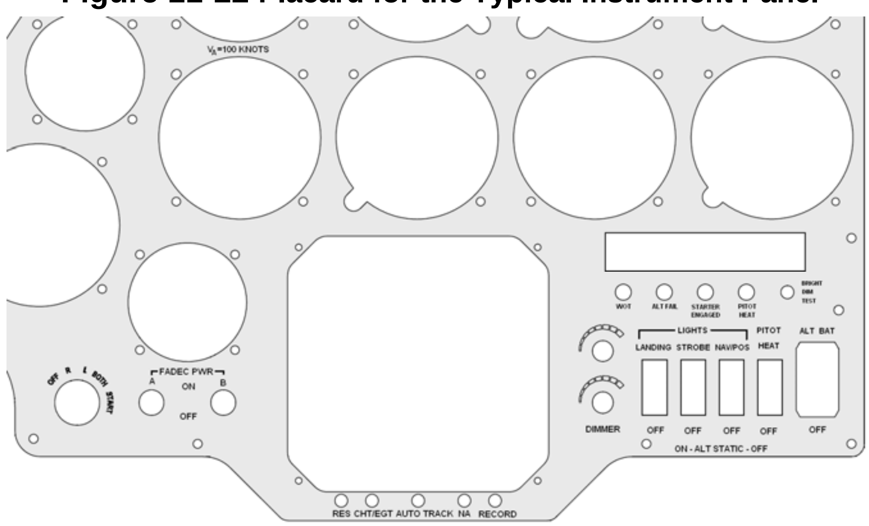


Figure 11-23 Enlarge View of the Instrument Panel P/N 135A-80-311

NOTE

The lettering in the above pictorials is in reverse color for clarity. The lettering is normally white on either a grey or a beige placard.

An engine RPM limitation placard is applied to the VM1000FX engine display bezel as shown in Figure 11-24. This placard instructs the pilot to avoid prolonged engine operation at speeds between 850 and 900 RPM. In the event of a VM1000FX display replacement this placard, Liberty Part Number 135A-80-325, must be applied to the replacement display bezel.

Figure 11-24 RPM Limitation Placard

Section 40-02 Avionics Panel

The avionics panel is located in the center of the instrument panel console assembly. Covering the front surface of the panel is a placard. There are two different placards for the avionic panel depending on the gross weight of the airplane. See Figure 11-25 for details of the avionics panel placard and the differences depending on the gross weight. Also mounted in this panel are the various avionics and controls. In addition, mounted on this panel are the avionics master switch, flaps position indicators, flaps up/down rocker switch, and the ELT control.

The ME406 ELT also adds an overlay label to the avionics placard. Figure 11-25 shows the location of this overlay label. The previous ELT remote panel had the labeling information on the panel part and was not a separate placard or label.

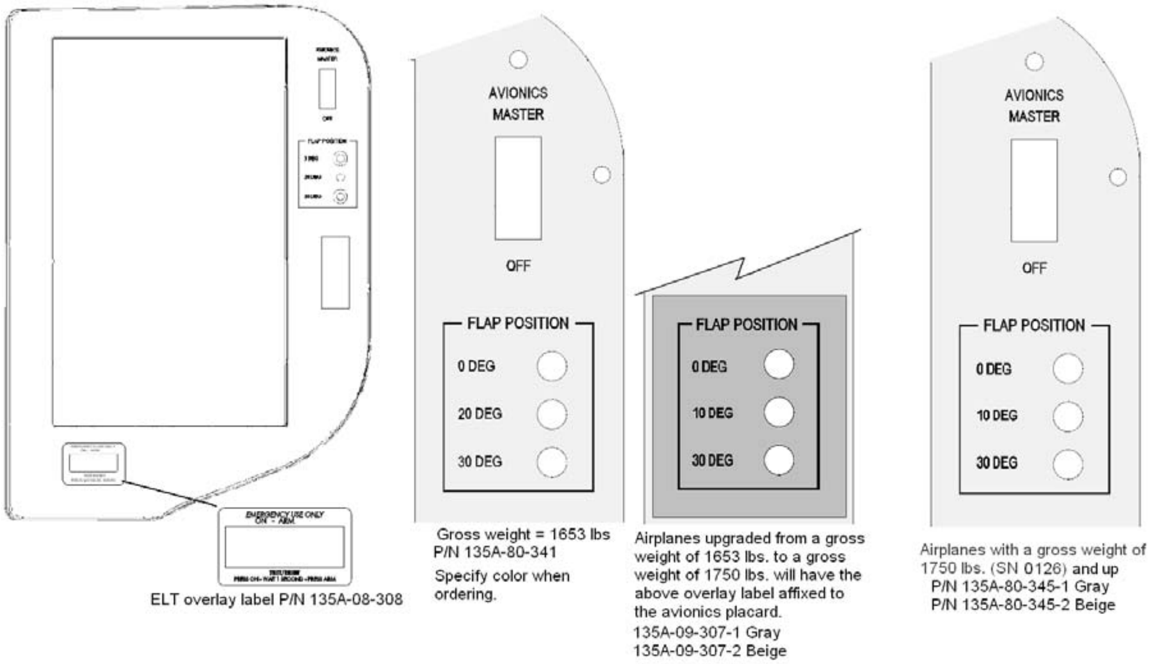


Figure 11-25 Placard for the Avionics Panel

NOTE

The above pictorials are in reverse color for clarity. They are normally grey or beige with white lettering.

Section 40-03 CB (Circuit Breaker) Panel

The CB or Circuit Breaker panel is located on the starboard side of the instrument panel console, in front of the passenger's seat. Mounted to this panel are the airplanes circuit breakers, the Hobbs meter, and a 12-volt receptacle. The placard covers the surface of the CB panel and identifies the various circuit breakers mounted to the panel. The panel shown in Figure 11-26 may differ from the panel that is in the airplane. This difference will depend on the avionics installed in the airplane.

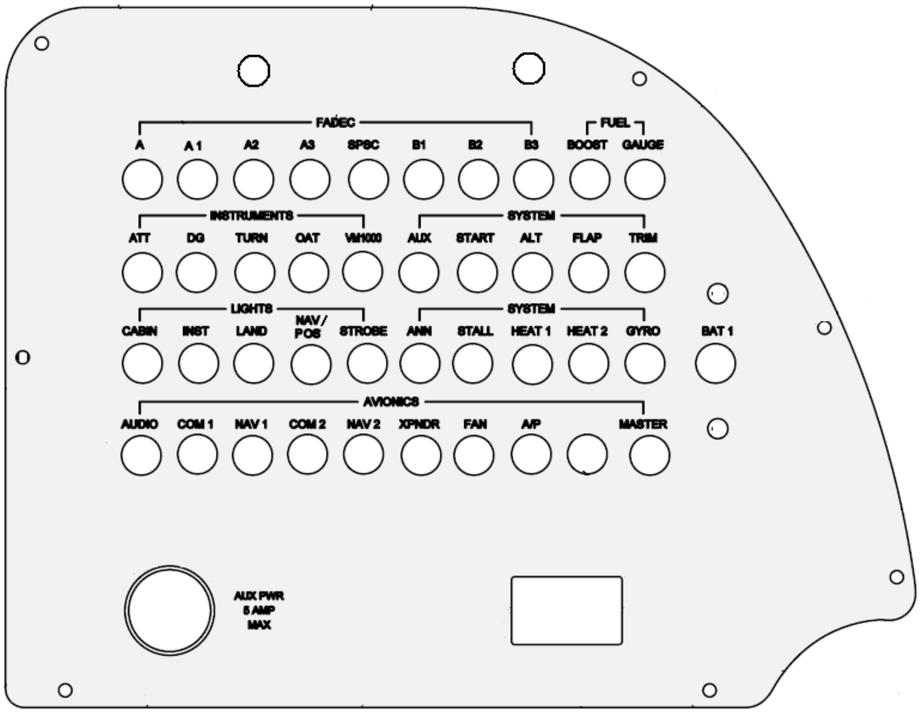


Figure 11-26 Placard for the CB (Circuit Breaker) Panel P/N 135A-80-303

NOTE

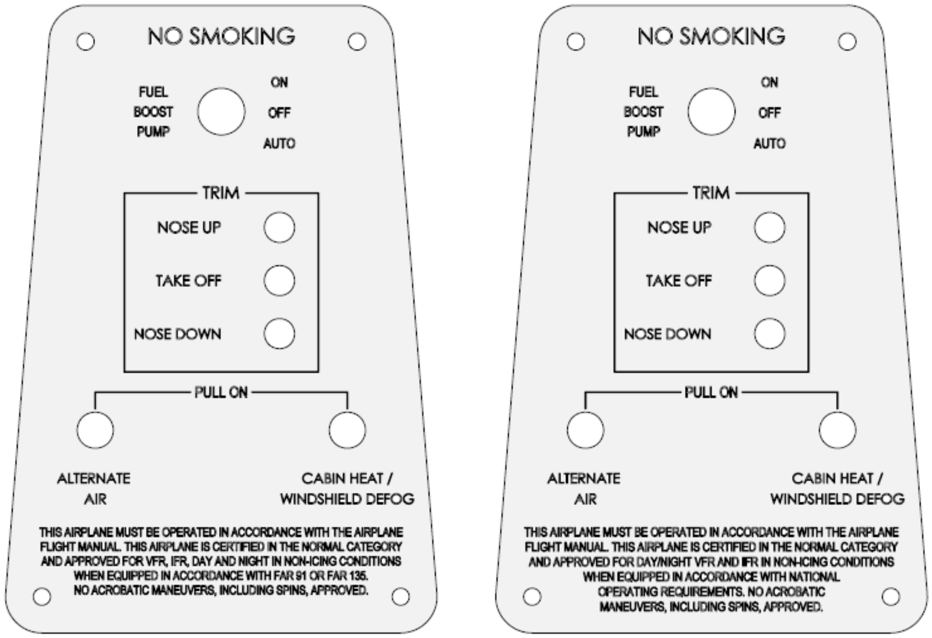
The above pictorial is in reverse color for clarity. The CB panel placard is normally grey or beige with white lettering.

NOTE

In the above pictorial, the placard shown reflects the current version of the CB placard. In earlier placards, the NAV/POS circuit breaker in the lights section only said NAV. In either case, this circuit breaker controls both the navigation and position lights

Section 40-04 Upper Center Console Panel

The upper center console panel is located below the avionics panel. This panel has the fuel boost pump switch, trim indicators, and cabin environmental controls. See Figure 11-27 for details of the upper center console panel. For airplanes delivered to China, a decal covers the No Smoking portion of the placard. The decal has No Smoking in English and Chinese. See Figure 11-28 for details of the decal.



FAA Version
-1 Gray, -2 Beige

EASA Version
-3 Gray, -4 Beige

The part number of the placard is 135A-80-323-*. The * indicates color and agency approval statement.

Figure 11-27 Placard for the Upper Center Console

Figure 11-28 No Smoking Decal, P/N 135A-09-305, for Airplanes Delivered to China

NOTE

The above pictorials are in reverse color for clarity. They are normally grey or beige with white lettering.

Section 40-05 Lower Center Console

The lower center console area has four placards. The placards are for the throttle control, airplane brakes, rudder trim switch, and the fuel shut-off. See Figure 11-29 for details of the lower center console area. Table 11-10 will show the details of the placards or plates and their location.

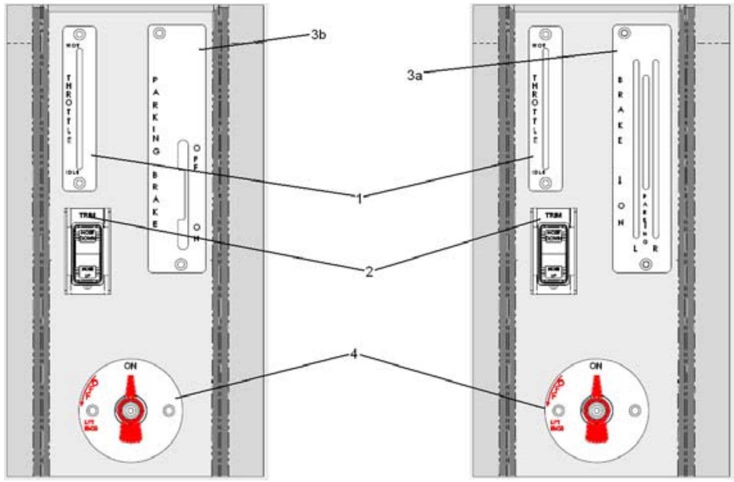
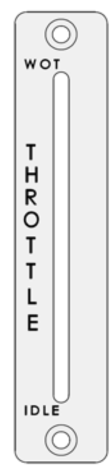


Figure 11-29 Placards on the Lower Center Console

NOTE

The pictorials of Items 1 – 3a in Figure 11-29 and Table 11-10 are in reverse color for clarity. They are normally black with white lettering.

Item Number	Location	Part Number	Placard Details
1	Located on the port area of the lower center console	135A-10-329	
2	Located on the port area of the lower center console	135A-10-347	

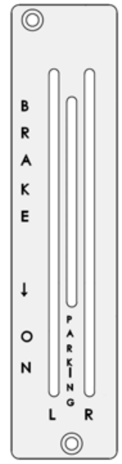
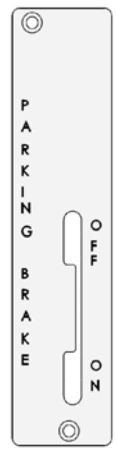
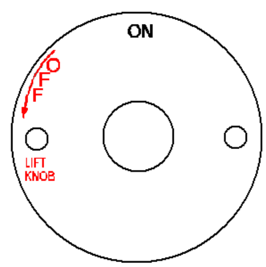
Item Number	Location	Part Number	Placard Details
3a	Located on the starboard area of the lower center console, this plate is for airplanes equipped with hand brakes.	135A-10-330	
3b	Located on the starboard area of the lower center console, this plate is for airplanes equipped with toe brakes.	135A-11-331	
4	Located in the Aft area of the lower center console	135A-50-627	

Table 11-10 Lower Center Console Placards

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Section 11-50 Miscellaneous Placards and Labels

This section is for other labels, placards, and other markings not associate with the above categories.

Section 50-01 Ball Drive Actuator Label

There is a caution label on the wiring of the ball drive actuator on the wing lock mechanism located on the port and starboard side of the space frame. See Figure 11-30 and Figure 11-31 for the location and details of the label.

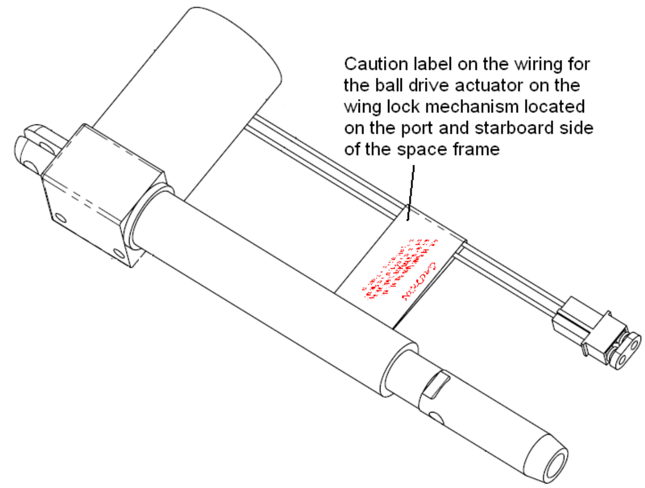

Figure 11-30 Ball Drive Actuator Showing the Location of the Caution Label

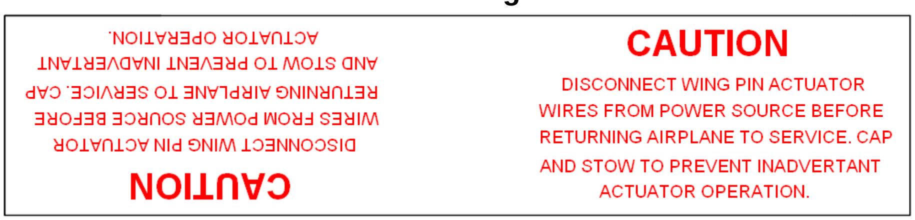

Figure 11-31 Details of the Caution Label on the Ball Drive Actuator

Section 50-02 Stepping Surfaces

Stepping surfaces have a strip of anti-skid tape applied to the surface. If the airplane is equipped with toe brakes, there is a strip of anti-skid tape applied to the brake and the rudder pedals.

On the footsteps there is the black anti-slip tape with an adhesive backing, P/N 6970T63, see Figure 11-32 for the location of the tape. Apply directly to the surface of the step.

If the airplane has toe brakes, then the rudder pedals and the toe brake pedals, have the black anti-slip tape with an adhesive backing, P/N 6970T63, see Figure 11-33 for the location of the tape. Apply directly to the surface of the pedal.

Figure 11-32 Location of the anti-skid strip tape on the foot step

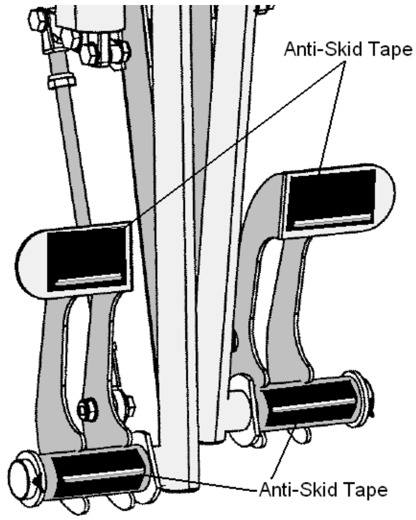

Figure 11-33 Location of the anti-skid strip tape on the rudder pedals and the toe brake pedals

Section 50-03 Compass Correction Card Filling

The correction card, supplied with the compass, mounts to the underside of the compass. Withdraw the lower housing as shown and clip the card into place. Replace the lower housing when finished.

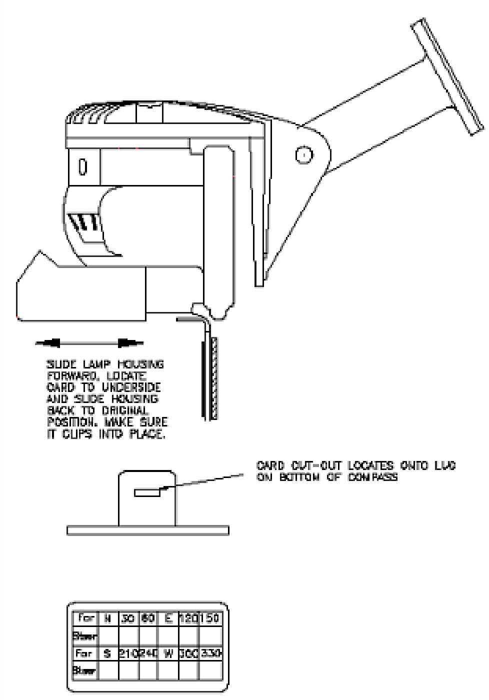

Figure 11-34 Example of the Compass Correction Card and Its Location on the Magnetic Compass

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CHAPTER 12

SERVICING

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Table of Contents

SECTION 12-00	GENERAL	5
SECTION 12-10	REPLENISHING	7
SECTION 10-01	FUEL SYSTEM	7
SECTION 10-02	DEFUELING	7
SECTION 10-03	OIL SYSTEM	8
SECTION 10-04	RECOMMENDED VISCOSITY FOR TEMPERATURE RANGE	9
SECTION 10-05	OIL SERVICING	9
SECTION 10-06	OIL DRAINING	10
SECTION 10-07	APPROVED REPLACEMENT OIL	10
SECTION 10-08	BRAKE SYSTEM	11
SECTION 10-09	TIRES:	11
SECTION 10-10	BATTERIES:	11
SECTION 12-20	SCHEDULED SERVICING	13
SECTION 20-01	LUBRICATION SCHEDULE	13
SECTION 20-02	NO LUBRICATION COMPONENTS	13
SECTION 12-30	UNSCHEDULED SERVICING	15
SECTION 30-01	EXTERIOR CLEANING	15
SECTION 30-02	INTERIOR CLEANING	15
SECTION 30-03	ENGINE CLEANING	15
SECTION 30-04	SNOW AND ICE REMOVAL	16

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Section 12-00 General

This chapter describes normal servicing procedures for the Liberty XL-2 airplane.

These procedures include replenishment of consumables and periodic lubrication of certain components. Procedures for servicing other components at certain intervals are also included in this chapter.

Detailed procedures for preventive and corrective maintenance of individual systems are provided in applicable specific chapters of this Maintenance Manual.

NOTE

Required inspection and maintenance intervals for the airplane as a whole, and for various systems and subsystems, are listed in Chapter 05 of this manual.

Figure 12-1 Liberty Aerospace, Inc. XL-2 Airplane

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Section 12-10 Replenishing

This section contains the information on replenishing the airplane.

Section 10-01 Fuel System

Refueling: The Liberty XL-2's fuel supply is contained in a single tank of 28 US gallons (106 liters) usable capacity. The tank is a welded aluminum tank located in the center fuselage. The tank cover forms the seatback structure for the left and right seats.

There is a filler port located near the top of the fuselage on the Port side used for refueling the airplane

Figure 12-2 Fuel Specification Labeling

CAUTION

Observe the following precautions during refueling:

- *Turn off all aircraft electrical power before fueling. All power must be OFF and must remain off until completion of fueling and the fuel filler cap is securely closed.*
- *A fire extinguisher must be available for immediate use.*
- *The airplane and fuel truck must be grounded via a suitable ground cable connected to the engine exhaust pipe.*

Only fuel types approved in the Airplane Flight Manual (AFM) may be used. These are Aviation Gasoline (AVGAS), Grade 100LL (colored blue), and Aviation Gasoline (AVGAS), Grade 100 (colored green).

Section 10-02 Defueling

The airplane can be defueled via the fuel sump drain valve on the bottom of the fuselage.

Figure 12-3 Location of Fuel Tank Sump Drain

CAUTION

Use approved metal container(s) for defueling. Ground container to the engine exhaust pipe prior to starting defueling, and maintain ground connection until defueling is complete.

Section 10-03 Oil System

The engine oil is defined by the following references:

- Teledyne Continental Motors Inc, Continental Aircraft Engine, Maintenance Manual Publication M-22, latest FAA approved revision.
- Teledyne Continental Motors Inc, Continental Aircraft Engine, Installation and Operation Manual Publication OI-22, latest FAA approved revision.

The airplane comes from the factory with a straight mineral oil approved specifically for engine break-in. This oil should be drained, and the filter replaced after the first 25 hours of engine operation.

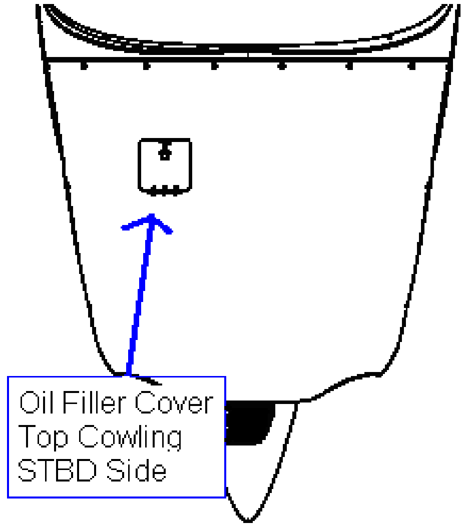

Figure 12-4 Location of Oil Filler Cover

Thereafter, only mineral oil, MIL-C-6529, TYPE II, or Ashless Dispersant oils specifically approved by Teledyne Continental Motors for the IOF-240-B engine should be used. Liberty Aerospace, Inc. recommends recording the brand, type, and grade of oil currently used. Keep this information on board the airplane for reference by pilots and maintenance personnel.

Section 10-04 Recommended Viscosity for Temperature Range

Multi-viscosity or straight grade oil may be used year round for engine lubrication. Refer to Table 12-1 for temperature vs. viscosity ranges:

TEMPERATURE RANGE	MIL-C-6529 TYPE LL SAE GRADE	ASHLESS DISPERSANT GRADE
ABOVE 4°C/ 40° F	SAE 50	SAE 50
BELOW 4° C/ 40° F	SAE 30 OR 15W50	SAE 30 OR 15W50
ALL TEMPERATURES	SAE 20W50 OR 20W60	SAE 20W50 OR 20W60

Table 12-1 Approved Oil Viscosities and Grades for the XL-2 Airplane

Section 10-05 Oil Servicing

The engine oil system has an oil sump capacity of 6.0 US quarts. Oil is added through a filler tube on the upper right side of the engine. The filler tube cap incorporates an oil dipstick.

Section 10-06 Oil Draining

Drain the engine oil by way of a drain plug at the bottom of the engine oil sump. For detailed engine oil draining procedures, see Teledyne Continental Motors Installation and Operation Manual OI-22, Chapter 05 section 2.2.3.

Section 10-07 Approved Replacement Oil

Table 12-2 shows those manufacturers of grades of ashless dispersant engine oil that are approved for the XL-2 airplane:

Manufacturer	Oil Grade and Type
BP Oil Corporation	BP Aero Oil
Castrol	Castrol AD Aero Oil
Castrol Limited (Australia)	Castrol AD Aero Oil
Chevron, U.S.A., Inc.	Chevron Aero Oil
Continental Oil	Conoco Aero S
Delta Petroleum Company	Delta Avoil Oil
Exxon Company, U.S.A.	Exxon Aviation Oil EE
Gulf Oil Company	Gulfpride Aviation AD
Mobil Oil Company	Mobil Aero Oil
NYCO, S.A	TURBONYCOIL 3570
Pennzoil Company	Pennzoil Aircraft Engine Oil
Phillips Petroleum Company	Phillips 66 Aviation Oil, Type A X/C Aviation Multi-viscosity Oil, SAE 20W50 or 20W60
Quaker State Oil & Refining Co	Quaker State AD Aviation Engine Oil
Red Ram Limited (Canada)	Red Ram X/C Aviation Oil 20W50
Shell Australia	Aeroshell® W
Shell Canada Limited	Aeroshell Oil W Aeroshell W 15W50, Anti-Wear Formulation, Aeroshell Oil W 15W50
Sinclair Oil Company	Sinclair Avoil
Texaco, Inc	Texaco Aircraft Engine Oil Premium AD
Total France	Total Aero DM 15W50
Union Oil Co. of California	Union Aircraft Engine Oil HD
Approved mineral oil	MIL-C-6529 TYPE II

Table 12-2 Table of Engine Oil Manufacturers Approved for the XL-2 Airplane

Section 10-08 Brake System

Routine servicing of the brake system is limited to checking and refilling the brake fluid reservoir with MIL-PRF-5606A hydraulic fluid. The brake fluid reservoir is located on the starboard front firewall. Access the reservoir by removing the upper cowling; see Chapter 71- *Power Plant*.

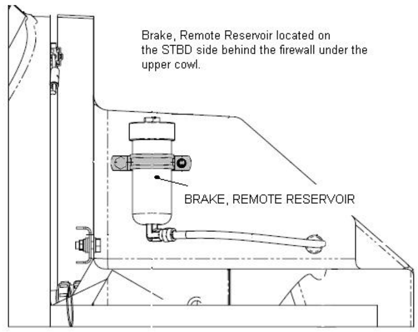

Figure 12-5 Location of Brake Reservoir

NOTE

Pack absorptive material under and around the reservoir while adding fluid. Wipe up any spilled brake fluid immediately, as it rapidly damages painted surfaces.

Section 10-09 Tires:

Main and nose landing gear tires are 5.00 x 5 and inflated to 50 (-0, +2) psi. A visual tire inspection should be performed each time pressure is checked.

There is a red slip marking (a red dot on the tire associated with the valve stem) on the tires to indicate weather or not the tire is slipping on the rim. Check this marking at this time to make sure the tire is not slipping on the rim. The inspection should include a thorough inspection of the tire’s tread. If the slip marking has moved from the area of the valve stem, or there is unusual wear at one point on the tire’s tread, replace the tire.

Section 10-10 Batteries:

The Liberty XL-2 has two recombinant-gas type (RG) batteries. Both batteries mount to the airplane aft of the baggage compartment in the fuselage. Access to the batteries is through the baggage compartment’s rear closeout bulkhead. The primary battery supplies aircraft electrical services such as engine starting, emergency power, and electrical system surge suppression. The secondary battery is dedicated as a backup power source for the Full Authority Digital Engine Control (FADEC) system, and is isolated from the rest of the aircraft electrical system except for charging purposes. In the event of a primary power failure (both primary battery and alternator) the secondary battery will support FADEC, Attitude Gyro, and Turn Coordinator for 60 minutes.

Both batteries are of a sealed maintenance-free type. Servicing is limited to replacement of batteries. Detailed procedures for battery checking and replacement are in Chapter 24 – *Electrical Power*.

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Section 12-20 Scheduled Servicing

This chapter details specific areas and components that require periodic lubrication. In addition, it identifies areas that must not be lubricated.

Section 20-01 Lubrication Schedule

Table 12-3 shows the schedule for lubrication of different parts on the XL-2 airplane.

Lubrication Schedule		
Aircraft Part	**Time Interval**	**Lubricant**
Aileron hinges/Flap hinge points/trim servo tab hinge	100 hours/ Annual Inspection	Corrosion X
Wing pins	100 hours/ Annual Inspection	Corrosion X
Tailplane Torque Tube Bearing	100 hours/ Annual Inspection	Corrosion X
Main/Nose Wheel Bearings	100 hours/ Annual Inspection	SHC100 Mobil Aviation Grease

Table 12-3 Table Showing the Lubrication Schedule for the XL-2 Airplane

NOTE

The time interval is a guideline. Lubricate components at whichever interval occurs first.

Section 20-02 No Lubrication Components

The following items do not require lubrication and never should be lubricated.

- Nose Wheel Friction Shimmy Damper

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Section 12-30 Unscheduled Servicing

This section details those areas of the airplane that need periodic servicing; however, these areas are not on a specific schedule.

Section 30-01 Exterior Cleaning

Wash the exterior of the airplane with generous amounts of water and, if desired, a mild soap. Small amounts of common cleaner/degreaser products may be applied by hand (rag, sponge) to remove resistant oil or exhaust stains.

WARNING

CHECK THAT PITOT AND STATIC TUBES ARE COVERED BEFORE WASHING AIRPLANE.

CAUTION

Use generous amounts of water, mild soap, and, if desired, cleaning products specifically formulated for acrylic transparencies to clean the windshield and windows. Check that no dirt remains on the plexiglas before rubbing with a soft cloth. The best method for initial cleaning is to flush the transparent surfaces with generous amounts of water while rubbing gently with the palm of a hand (do not wear gloves). Remove rings and watches before cleaning transparency.

CAUTION

Never rub or polish dry acrylic transparency material.

CAUTION

Do not use high pressure spray washers to clean any part of aircraft exterior.

Section 30-02 Interior Cleaning

Floor coverings, carpeted sidewalls, and (fabric) seat covers should be vacuumed at regular intervals. Leather seats may be treated with standard automotive leather upholstery conditioners as necessary.

Instrument panels, cockpit center console, and instrument panel glare shield may be wiped with a dampened soft cloth (water only). Check that no solvents or strong cleaning agents are used on interior surfaces.

Section 30-03 Engine Cleaning

The engine may be cleaned with standard cold solvents (Mineral Spirits or Stoddard solvent, etc.) using a low pressure spray only.

CAUTION

Check that all engine and accessory breathers and vent openings are sealed (tape, plastic bags, etc.) Before cleaning engine. Do not spray solvent directly on or near any electric or electronic (FADEC) accessories or connectors when cleaning engine.

CAUTION

Check that all solvent has evaporated or dried before starting engine.

Section 30-04 Snow and Ice Removal

If snow has collected on the airplane, it should be removed as soon as possible (ideally, before it has a chance to melt) to prevent melt-water from refreezing on the airplane surface or in control surface gaps. Do not use sharp objects or scrapers to remove snow or ice accumulations from airplane. The best method for snow and/or ice removal is to place the airplane in a heated hangar.

CAUTION

To avoid melted snow or ice refreezing on or in the aircraft, do not remove aircraft from heated hangar until at least one half hour after all melt-water has drained (all dripping has stopped). Check all drain holes for free draining capability. In the event that ice is observed between the flight control surfaces and adjacent structures, remove ice prior to cycling flight controls.

CHAPTER 20

STANDARD PRACTICES AIRFRAME

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Table of Contents

SECTION 20-00 GENERAL 5
SECTION 20-10 STANDARD PRACTICES MECHANICAL 7
SECTION 10-01 BOLT TYPES 7
SECTION 10-02 TORQUES 7
SECTION 10-03 SPECIAL TORQUES 9
SECTION 10-04 TORQUE MEASURING DETAILS 9
SECTION 10-05 SAFETY WIRING 9
SAFETY WIRE PROCEDURE 11
SECTION 10-06 COTTER PIN/SPLIT PIN 13
COTTER PIN/ SPLIT PIN PROCEDURES 14
SECTION 10-07 TAB WASHERS 15
SECTION 10-08 WET ASSEMBLY 15
WET ASSEMBLY: (NON-PERMANENT FASTENER INSTALLATION) 16
SECTION 20-20 STANDARD PRACTICES ELECTRICAL 17
SECTION 20-01 WIRING STANDARDS 17
SECTION 20-02 WIRE NUMBERING 17
SECTION 20-03 WIRE COLOR CODE DESIGNATION 18
SECTION 20-04 SPLICING 18
SECTION 20-05 CRIMP SPLICES 19
SECTION 20-06 SPLICING PROCEDURES 19
ONE-WIRE TO ONE-WIRE IN-LINE SPLICES, OR TWO-WIRE STUB SPLICES 20
MULTIPLE-TO-ONE IN-LINE SPLICES AND MULTIPLE-WIRE STUB SPLICE 22
MULTIPLE-WIRE TO MULTIPLE-WIRE IN-LINE SPLICES 24
INSPECTION OF THE SPLICE 26

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Section 20-00 General

This chapter lists maintenance related information such as prescribed torque values, measurement techniques, and safety wiring. This chapter defines Wet Assembly. This chapter also describes standard practices for the electrical wiring, such as wire numbering and sizing, wire termination, splicing, and soldering.

Specific fastener torque values appearing in individual Liberty Maintenance Manual chapters and sections may supersede the general torque values given in this chapter.

For Propeller Installation consult individual manufacturer's maintenance manuals for torque values.

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Section 20-10 Standard Practices Mechanical

This section details the information for the hardware used on the Liberty Aerospace, Inc. XL-2 airplane.

Section 10-01 Bolt Types

Bolts used in the Liberty XL-2 conform to AN (Air Force-Navy) specification. In general, cadmium plating protects against corrosion. See Figure 20-1 for details on bolt markings.

- Corrosion Resistant steel bolts are marked with a dash (—) on bolt head.
- Non-Corrosion Resistant (un-plated) steel bolts are marked with an (X) on the bolt head.

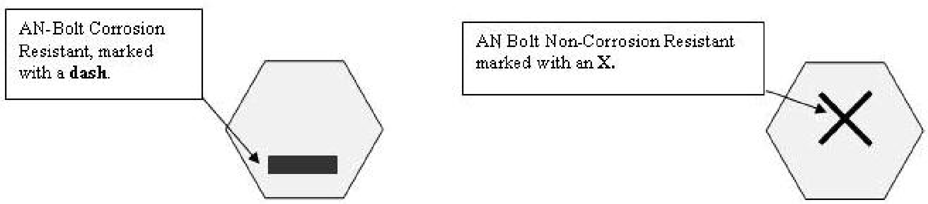

Figure 20-1 Bolt Markings to Indicate Corrosion Resistance

WARNING

THE CADMIUM PLATING ON CORROSION-RESISTANT AIRCRAFT FASTENERS HAS TOXIC PROPERTIES AND, IF ABSORBED IN THE BODY, MAY LEAD TO SYMPTOMS OF HEAVY METAL POISONING. NEVER PLACE ANY CADMIUM-PLATED FASTENER OR COMPONENT IN YOUR MOUTH.

Nuts used with these bolts are in accordance with AN3 through AN20 specifications. Some specialized lock nuts may conform to differing specifications

Section 10-02 Torques

The permissible torque values for bolts and nuts are given below. To avoid crushing underlying composite layers remember locknut torque values are lower when fasteners are used in composite structures.

Installation Torque for Bolt/Nut Combinations		
Nut installation torque values (lubricant free, cadmium plated		
Nominal Fastener Diameter (inches)	Mid-Range Values (in-lbs)	Range (in-lbs) Min-Max (90 KSI in bolts)
0.1640	17.5	15-20
0.1900	32.5	25-40
0.250	95	90-100
0.3125	202.5	180-225
0.3750	345	300-390

Table 20-1 Torque Values for Bolt and Nut Combinations

Installation Torque for Bolt/Nut Combinations		
Hex nuts-self locking (NAS1291XX)		
Nominal Fastener Diameter (inch)	Mid-Range Values(in-lbs)	Range(in-lbs) Min-Max
0.1875	30	25-35
0.250	70	60-80
0.3125	145	130-160
0.375	220	200-240

Table 20-2 Torque Values for Self Locking Hex Nuts

Installation Torque for Bolt/Nut Combinations in Composite Structures		
Nut installation torque values		
Nominal Fastener Diameter (inch)	Mid-Range Values(in-lbs)	Range(in-lbs) Min-Max
0.1640	17.5	15-20
0.1900	32.5	25-40
0.2500	95	90-100
0.3125	202.5	180-225
0.3750	345	300-390
Locknut torque values		
Nominal Fastener Diameter (inch)	Mid-Range Values(in-lbs)	Range(in-lbs) Min-Max
0.1562	20	15-25
0.1875	30	25-35
0.2500	70	60-80
0.3125	145	130-160
0.3750	220	200-240

Table 20-3 Torque Value for Hardware in Composite Structures

Section 10-03 Special Torques

For specific torque requirements relating to Propeller Installations, please refer to manufacturer's maintenance manual

- Maintenance Manual for Sensenich W69EK7-63G Propeller

  Sensenich Wood Propeller Company Inc.
  Wood Propellers Installation Operation, & Maintenance
  Integral Flange Crankshafts WOOD-CF-REV-A.doc 5-20-04

- Maintenance Manual for MT175R127-2Ca Propeller

  Operation and Installation Manual
  E-112 (ATA 61-01-12)
  MT-Wood-Composite
  Fixed Pitch Propellers
  November 21, 2006

Section 10-04 Torque Measuring Details

CAUTION

Self-locking nuts of any type must be replaced if there is any indication of diminished frictional torque upon removal or reinstallation.

If using self-locking nuts, the torque required to overcome the locking feature (friction torque) must be determined by measuring indicated torque while nut is being tightened and adding the indicated torque to table torque values. Determine friction torque before it has seated against washer or other seating surface.

If using castellated nuts and split pins, torque to the lowest value of torque as shown in the tables. Continue to torque to nearest castellated slot for split pin. Do not exceed the highest value of torque as shown in the tables. Install a new split pin.

It is always preferable to tighten nut, rather than bolt, since rotation of bolt in the hole may damage the corrosion-resistant cadmium plating.

If tightening from bolt head side is unavoidable (e.g., the nut is inaccessible to a torque wrench), the turning torque of bolt in the hole must be determined, by measuring indicated torque before nut is placed on bolt. This torque must be added to both the torque value from tables above and torque value, if any, from any self-locking nut that is installed.

Section 10-05 Safety Wiring

CAUTION

Safety wire must never be reused; always discard wire immediately upon removal.

Figure 20-2 Main wheel caliper safety wire

The purpose of safety wiring fasteners is to prevent fasteners from becoming loose during use. Thus, standard safety wiring patterns should be used to apply clockwise force to fasteners and secure them against inadvertent rotation in a counterclockwise direction. Safety wiring can be secured either from fasteners to adjacent aircraft structure or, if necessary, between adjacent fasteners such that counterclockwise rotation is mutually avoided.

SAFETY WIRE PROCEDURE

Safety wiring or lockwiring is the securing together of two or more parts with lock-wire which shall be installed in such a manner that any tendency for a part to loosen will be counteracted by additional tightening of the wire.

1. If using castellated nuts, torque to the lowest value of torque as shown in the tables. Continue to torque to nearest castellated slot for split pin. Do not exceed the highest value of torque as shown in the tables.
2. Wire shall be pulled taut while being twisted and caution must be exercised during the twisting operation to keep the wire tight without over stressing. See Figure 20-10-03-1-1 and 20-10-03-1-2 for steps in applying lockwire.
3. Lockwire shall be new at each application.
4. Torque all items to be safety wired to the proper value. Applying torque above or below specified limits to obtain alignment of holes is not permitted.
5. Lockwire shall be installed in such a manner that the strand through the hole will have a tendency to pull in the tightening direction.
6. Insert half of the required length of wire through the first unit and bend around the head of the unit. Direction of wraps and twist of strands shall be such that the loop around the unit comes under the strand protruding from the hole so that the loop will stay down and will not tend to slip up and leave slack loop. Twist strands while taut until twisted part is just short of a hole in the next unit. Twisted portion should be within 1/8 inch from hole in either unit.
7. Insert uppermost strand through hole in second unit and follow instructions in the previous paragraph.
8. After lockwiring last unit, continue twisting wire to form a pigtail providing a minimum of four twists. This will ensure the pigtail stays secure. Cut off excess lockwire and bend pigtail toward the part and against bolt head flats. Do not allow the pigtail to extend above the bolt head.

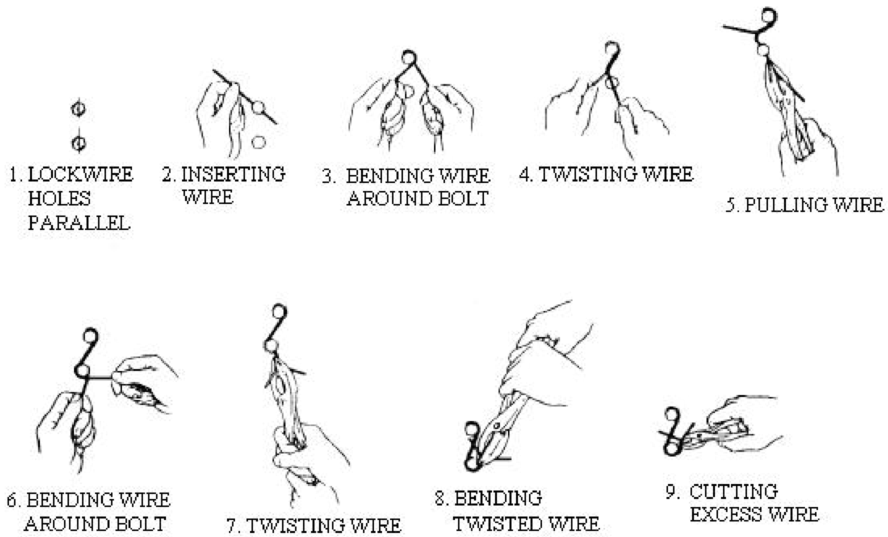

Figure 20-3 Illustration of the Procedure to Use Lockwire

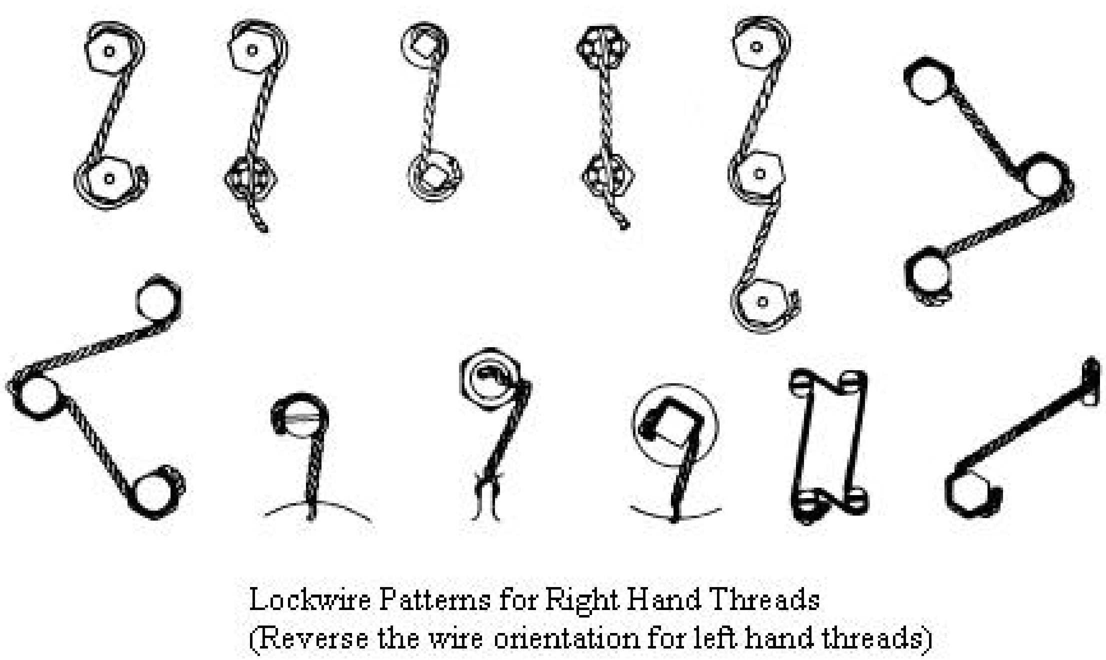

Figure 20-4 Different Methods of Using Lockwire

Section 10-06 Cotter Pin/Split Pin

CAUTION

Never reuse cotter pins/split pins; always discard immediately upon removal.

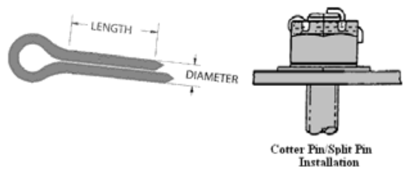

Figure 20-5 Split Pin

Cotter pins/split pins are typically used to secure castellated (slotted) nuts against inadvertent rotation. The following procedure outlines the two methods of installing cotter/split pins.

COTTER PIN/ SPLIT PIN PROCEDURES

There are two methods to install a cotter or split pin. Both methods are shown here.

Method 1

1. Torque the castellated nut to the lowest value of torque as shown in the tables. Continue to torque to nearest castellated slot for split pin. Do not exceed the highest value of torque as shown in the tables. Install a new split pin.
2. Insert cotter pin/split pin through castellated nut with “eye” of cotter pin/split pin vertical to allow seating to maximum depth in slot of nut.
3. Using side or flush cutters cut off excess length of cotter pin/split pin.
4. Bend one leg of cotter pin/split pin down along side of nut. Check the end of bent leg stops short of bottom edge of nut.
5. Bend other leg of cotter pin/split pin back across top of nut and bolt; cut off excess length.

Method 2 (An Alternate Method)

1. Torque the castellated nut to the lowest value of torque as shown in the tables. Continue to torque to nearest castellated slot for split pin. Do not exceed the highest value of torque as shown in the tables. Install a new split pin.
2. Insert cotter pin/split pin as far as possible through castellated nut with “eye” of cotter pin/split pin horizontal.
3. Trim both legs of cotter pin/split pin to appropriate length to allow ends to be bent back and pushed into slots in castellated nut adjacent to slot occupied by body of cotter pin/split pin.

Section 10-07 Tab Washers

CAUTION

Discard tab washer immediately upon removal. Never re-use a tab washer.

Tab washers are used to prevent inadvertent fastener rotation by securing facets to fasteners to adjacent material or structure. A hole must be provided adjacent to the fastener location into which at least one tab of the tab washer can be bent. Remaining tab(s) can be bent up against the facets (flats) of the fastener. Use a light hammer and a soft (brass) drift to bend tabs. Check that tabs are not cracked during bending process.

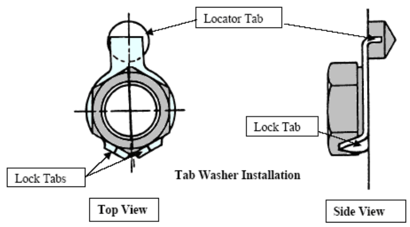

Figure 20-6 Tab Washer

Section 10-08 Wet Assembly

Technical Data CA 1000 is a non-chromate corrosion inhibitive jointing compound PRC® aerospace sealant from PRC-DeSoto International, Inc.

"This material acts as an effective barrier against the common causes of corrosion on aluminum alloys or between dissimilar metals. The compound remains permanently mastic after prolonged exposure to aircraft fuels, both jet fuel and aviation gas. CA 1000 is a one part, epoxy capped, Permapol® polysulfide compound. The material is a thixotropic paste suitable for application by brush or spatula."

Material Description	Material Identification	Recommended Supplier
CA 1000	non-curing, non-chromate, corrosion inhibitive jointing compound	Bergdahl Assoc., Inc., Reno, NV

Table 20-4 Identification of the Compound CA1000

WET ASSEMBLY: (NON-PERMANENT FASTENER INSTALLATION)

1. Prepare surfaces, if required, by cleaning with solvent to remove dirt, greases and other possible contaminants.
2. Apply the compound to the surface(s) per manufacturer's instructions. Install fasteners while the jointing compound is still tacky such that it will flow sufficiently under pressure.
3. Clean excess compound from the joint.
4. Non-curing corrosion inhibitive compounds used should allow for easy removal, with compound still tacky

Section 20-20 Standard Practices Electrical

This section details the information for the electrical wiring used on the Liberty Aerospace, Inc. XL-2 airplane.

Section 20-01 Wiring Standards

Wiring used in the Wiring Diagrams for non-shielded wires are per MIL-W-22759. All non-shielded wires are 22 AWG unless otherwise noted. Wiring used in the Wiring Diagrams for shielded wires are per MIL-W-27500.

Section 20-02 Wire Numbering

Each wire in the XL-2 airplane has a unique number. The number defines what circuit or system the wire belongs to, the route for the wire, the route subsection, the gauge of the wire, and if the wire is a travelling ground.

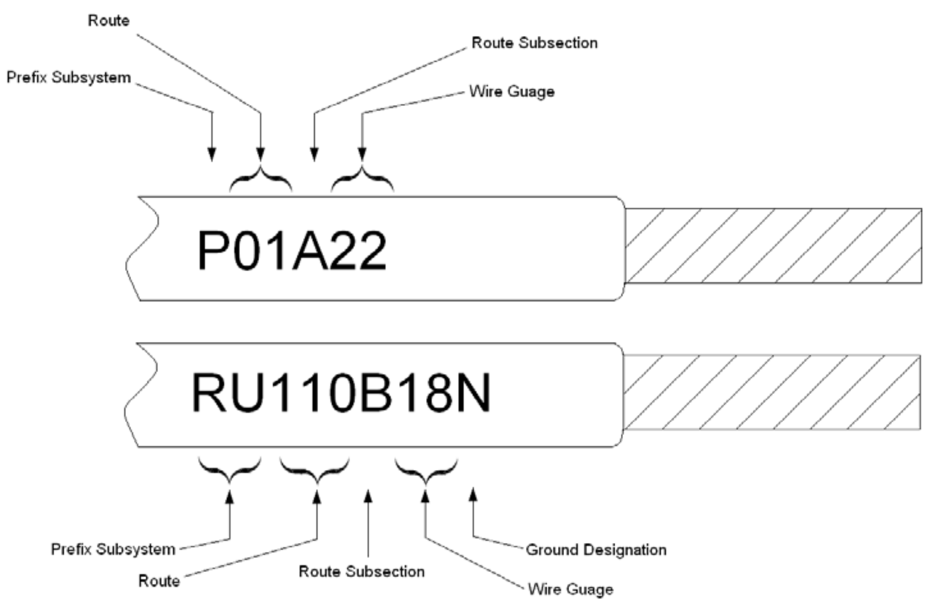

Figure 20-7 Wire Number Coding Standard

Table 20-5 gives the system description for the subsystem designation on the wire number. The route subsection designation for the route from source to termination.

Sub System Designation	System Description
A	Antenna
AP	Autopilot
C	Clock and Facility socket
E	Fuel System
EGND	Earth Ground – ground to aircraft frame
F	FADEC
G	Gyro Instrumentation
H	Pitot Heat, Hour Meter, Outside Air Temperature
IGND	Instrumentation Ground – ground path isolated from Earth Ground
JPR	Jumper Wire

Sub System Designation	System Description
L	Lighting
P	Power Distribution
R	Avionics General
RD	Avionics Display Systems (glass cockpit)
RG	Avionics Garmin
RU	Avionics UPS Aviation Technologies
SP	Splice
ST	Solder Sleave
T	Trim Flaps
V	VM1000FX
W	Annunciation, Stall Warning

Table 20-5 Subsystem Prefix Designation for Wire Numbers

Section 20-03 Wire Color Code Designation

Table 20-6 defines the abbreviations used in the schematics for various wire colors. If a wire has multiple colors, the different colors are separated by a forward slash (/) between each color. The base color is first, followed by the second significant color then the third significant color and so on.

For example, a wire that has a blue base color, then red then green, would have the designation BLU/RED/GRN.

Color Code	Color
BLK	Black
BRN	Brown or Tan
RED	Red
ORG	Orange
YEL	Yellow
GRN	Green
BLU	Blue or Azure
VIO	Violet or Purple
SLT	Grey or Slate
WHT	White
SLD	Shield
TIP	Center conductor of a Coax Cable

Table 20-6 Wire Color Code Abbreviations

Section 20-04 Splicing

Splicing is permitted on wiring as long as it does not affect the reliability and the electromechanical characteristics of the wiring. Keep splicing of electrical wire to a minimum. Avoid splicing of electrical wire entirely in locations subject to extreme vibrations.

There should not be more than one splice in any one wire segment between any two connectors or other disconnect points, except; when attaching to the spare pigtail lead of a potted connector, to splice multiple wires to a single wire, to adjust wire size to fit connector contact crimp barrel size, and to make an approved repair.

Splices in bundles must be staggered so as to minimize any increase in the size of the bundle, preventing the bundle from fitting into its designated space, or cause congestion that will adversely affect maintenance.

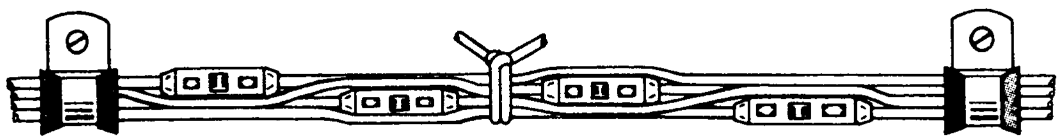

Figure 20-8 Staggered Splices in Wire Bundle

Splices should not be used within 12 inches of a termination device, except when attaching to the pigtail spare lead of a potted termination device, or to splice multiple wires to a single wire, or to adjust the wire sizes so that they are compatible with the contact crimp barrel sizes.

Section 20-05 Crimp Splices

Crimp splices normally come in two parts, a wire crimp sleeve, see Figure 20-9, and a sealing sleeve. The sealing sleeve can have either two integral one-hole seals, one integral one-hole seal and one integral multiple-hole seal, or with one integral multiple-hole seal and a separate multiple-hole seal, see Figure 20-10.

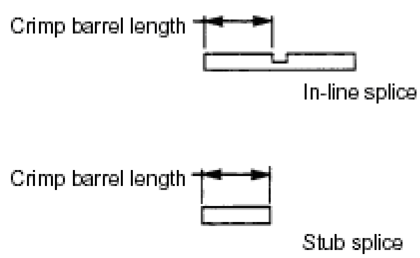

Figure 20-9 Wire Crimps

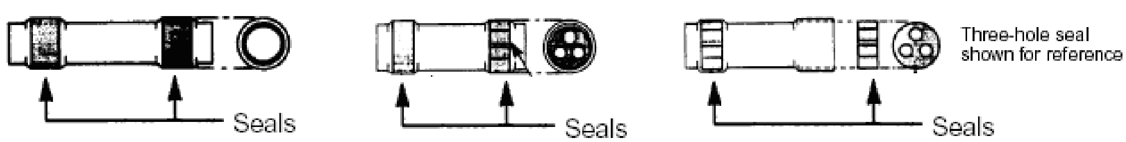

Figure 20-10 Sealing Sleeves

Section 20-06 Splicing Procedures

This section contains general splicing procedures for one-wire-to-one-wire or two-wire stub splicing, multiple-to-one wire or multiple-wire stub splicing, multiple-wire-to-multiple-wire splice. After the splicing procedures is the procedure to inspect the splice for compliance with Liberty Aerospace, Inc. specifications.

ONE-WIRE TO ONE-WIRE IN-LINE SPLICES, OR TWO-WIRE STUB SPLICES

Perform the following procedure to make a one-wire-to-one-wire or two-wire stub splice

1. Strip the wires to be spliced. If any conductor is to be folded back (to increase the effective cross-sectional area), strip the wire to twice the specified strip length.
2. The strip length for a particular crimp splice equals the length of the crimp barrel plus 1/32–1/16 inch (0.8–1.6 mm) as shown in Figure 20-11.

NOTE

For most AWG 12 and smaller crimp splices, the correct strip length is 5/16–11/32 inch (7.9–8.7 mm).

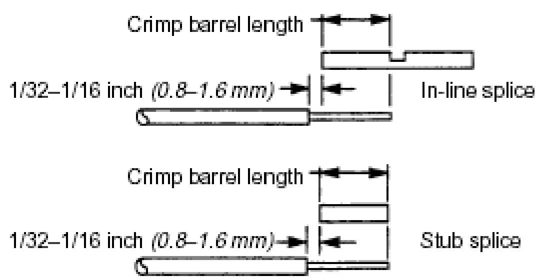

Figure 20-11 Strip Length

3. If required to fold any conductors back, fold the appropriate conductor(s) as shown in Figure 20-12.

Figure 20-12 Fold Back

4. If making an in-line splice, slide the sealing sleeve onto either wire. For stub splices, the sealing sleeve will be installed later.

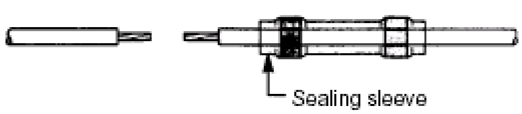

Figure 20-13 In-line Sealing Sleeve

5. Crimp the wires in the crimp splice as shown in Figure 20-14. The gap between the end of the wire insulation and the crimp splice must be 1/32–1/16 inch (0.8–1.6 mm).

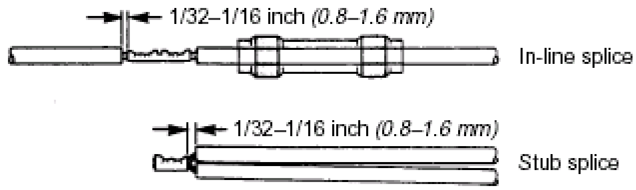

Figure 20-14 Insulation Gap

6. Center the sealing sleeve over the splice area as shown in Figure 20-15.

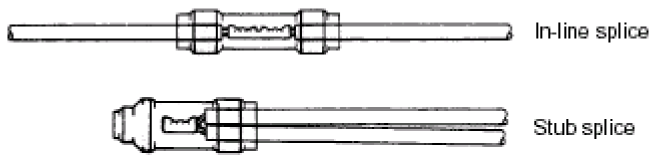

Figure 20-15 Center Sealing Sleeve

7. Heat sleeve from one end toward the other. For stub splices, start heating the splice at the end from which the wires exit.
8. Continue heating until the seals melt and flow along the wire insulation. For stub splices, heat until one seal melts and closes the end.
9. Allow the assembly to cool undisturbed.

MULTIPLE-TO-ONE IN-LINE SPLICES AND MULTIPLE-WIRE STUB SPLICE

Perform the following procedure to make a multiple-to-one-wire or multiple-wire stub splice.

1. Strip the wires to be spliced. If any conductor is to be folded back (to increase the effective cross-sectional area), strip the wire to twice the specified strip length..
2. The strip length for a particular crimp splice equals the length of the crimp barrel plus 1/32–1/16 inch (0.8–1.6 mm) as shown in Figure 20-16.

NOTE

For most AWG 12 and smaller crimp splices, the correct strip length is 5/16–11/32 inch (7.9–8.7 mm).

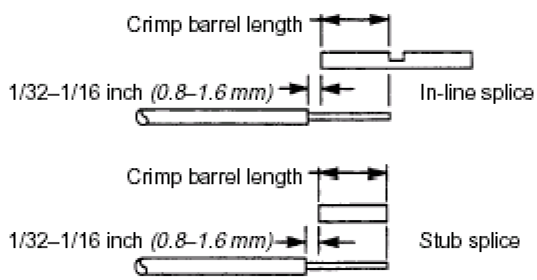

Figure 20-16 Insulation Strip Length

3. If required to fold any conductors back, fold the appropriate conductor(s) as shown in Figure 20-17.

Figure 20-17 Fold Back

4. Insert one of the multiple wires through each hole of the multiple-hole seal, and slide the sealing sleeve back over the multiple wires as shown in Figure 20-18.

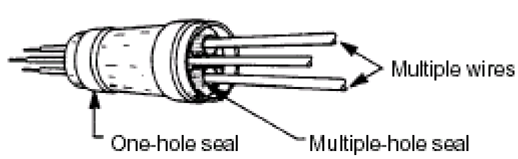

Figure 20-18 Multiple Wires

5. Crimp the wires in the crimp splice as shown in Figure 20-19.

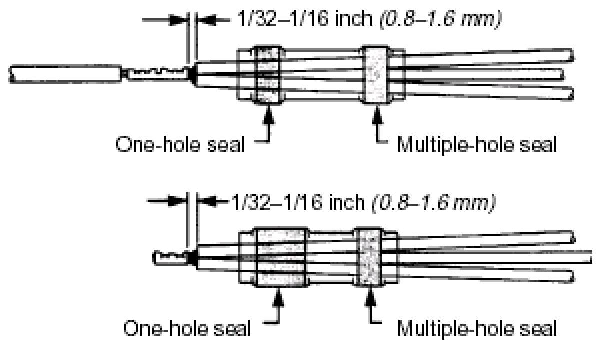

Figure 20-19 Multiple Wire Crimp Splice

6. Make sure that the multiple wires do not become crossed between the multiple-hole seal and the crimp splice.
7. Center the sealing sleeve over the splice area as shown in Figure 20-20.

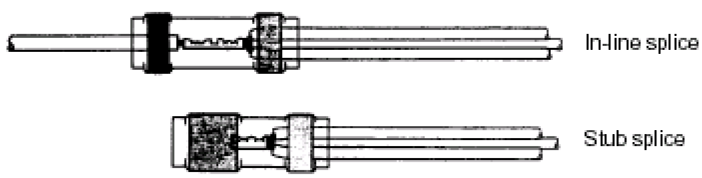

Figure 20-20 Center Sealing Sleeve

8. Heat the multiple-wire end first, until the multiple-wire seal melts and flows around and along the wire insulation.
9. Move the heat toward the single-wire or open end, shrinking the sleeve along the way.
10. Heat the second end until the seal melts and flows along the wire insulation or closes the open end.
11. Allow the assembly to cool undisturbed.

MULTIPLE-WIRE TO MULTIPLE-WIRE IN-LINE SPLICES

Perform the following procedure to make a multiple-to-one-wire or multiple-wire stub splice.

1. Strip the wires to be spliced. If any conductor is to be folded back (to increase the effective cross-sectional area), strip the wire to twice the specified strip length..
2. The strip length for a particular crimp splice equals the length of the crimp barrel plus 1/32–1/16 inch (0.8–1.6 mm) as shown in Figure 20-21.

NOTE

For most AWG 12 and smaller crimp splices, the correct strip length is 5/16–11/32 inch (7.9–8.7 mm).

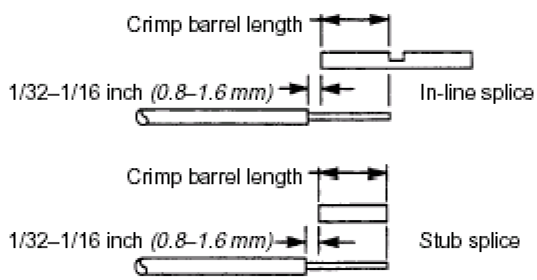

Figure 20-21 Insulation Strip Length

3. If required to fold any conductors back, fold the appropriate conductor(s) as shown in Figure 20-22.

Figure 20-22 Fold Back

4. Insert the multiple wires coming from one direction through the integral seal and sleeve, one wire per hole, as shown in Figure 20-23.

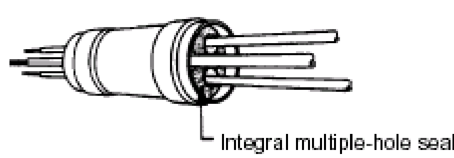

Figure 20-23 Multiple Wires

5. Insert the multiple wires coming from the other direction through the separate seal, one wire per hole, as shown In Figure 20-24.

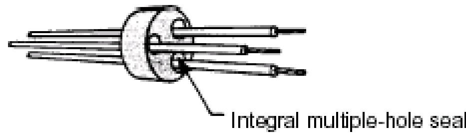

Figure 20-24 Multiple Hole Seal

6. Crimp the wires in the crimp splice as shown below.
7. Make sure that the multiple wires do not become crossed between either one of the multiple-hole seals and the crimp splice.

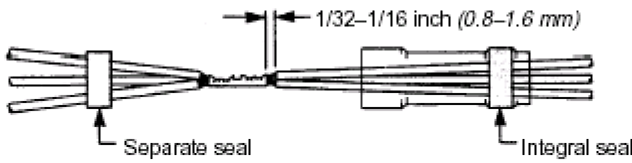

Figure 20-25 Crimp Splice Multiple Wires

8. Push the separate multiple-hole seal up against the crimp splice.
9. Position the sealing sleeve over the splice and over the separate multiple-hole seal so that the separate seal is fully seated within the end of the sealing sleeve as shown in Figure 20-26.
10. Squeeze the wires together to hold the separate seal against the splice while moving the sealing sleeve into position.

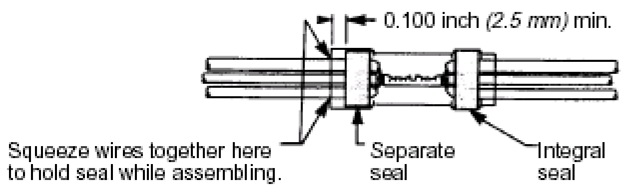

Figure 20-26 Center Sealing Sleeve

11. Heat the end with the separate seal first, until the seal melts and flows around and along the wire insulation.
12. Move the heat toward the end with the integral seal, shrinking the sleeve along the way.
13. Heat the end with the integral seal until the seal melts and flows along the wire insulation.
14. Allow the assembly to cool undisturbed.

INSPECTION OF THE SPLICE

Perform this procedure to inspect the splice after completing the splice. The following set of figures illustrate acceptable and unacceptable splices.

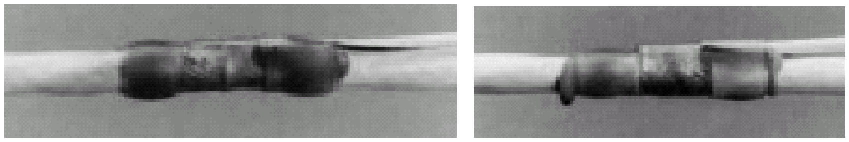

Figure 20-27 Acceptable

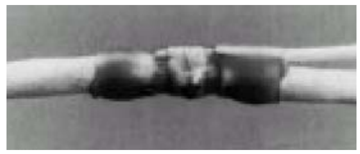

Figure 20-28 Unacceptable – Insufficient Heat

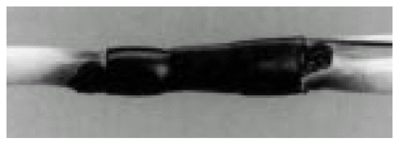

Figure 20-29 Unacceptable - Overheated/ Burnt

1. The sealing sleeve must be centered over the splice area so that the melted seal overlaps the wire insulation by at least 1 1/2 times the diameter of the wire insulation.
2. The wire insulation must end at a point 1/32–1/16 inch (0.8–1.6 mm) from the crimp splice.
3. The sealing sleeve must be completely shrunk onto the splice area and wire insulation.
4. If sealing a single wire, seal must have flowed along the wire insulation.
5. If sealing multiple wires, melted sealing material must be visible between the wires where they exit from the sealing sleeve. Look between the wires to make sure that the center portion of the seal has melted.
6. The free end of a stub splice must be completely sealed.
7. The sealing sleeve must not be so discolored it prevents visual inspection of the splice (overheated condition).

NOTE

Sealant can be re-flowed, if necessary, by applying additional heat with the heating tool

8. Inspection for Damage
9. The sealing sleeve must not be cut or split.
10. The wire insulation must not show signs of mechanical damage or overheating, such as cuts, tears, melting, or charring.

CHAPTER 21

ENVIRONMENTAL SYSTEMS

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Table of Contents

SECTION 21-00 GENERAL 5

SECTION 21-20 DISTRIBUTION SYSTEMS 7

SECTION 20-01 FRESH AIR DISTRIBUTION SYSTEM 7
SECTION 20-02 PERIODIC MAINTENANCE 7
SECTION 20-03 FRESH AIR DISTRIBUTION SYSTEM PROCEDURES 7
FRESH AIR DISTRIBUTION SYSTEM INSPECTION 8
SECTION 20-04 FRESH AIR DISTRIBUTION SYSTEM TROUBLESHOOTING GUIDE 8
SECTION 20-05 HEATING AND DEFOGGING DISTRIBUTION SYSTEM 9
SECTION 20-06 PERIODIC MAINTENANCE 9
SECTION 20-07 HEATING AND DEFOGGING DISTRIBUTION SYSTEM PROCEDURES 9
AERODUCT REMOVAL PROCEDURE 10
AERODUCT INSTALLATION PROCEDURE 11
FOUR WAY TUBE CONNECTOR REMOVAL 12
FOUR WAY TUBE CONNECTOR INSTALLATION 13
HEATING AND DEFOGGING DISTRIBUTION SYSTEM OPERATIONAL CHECK AND INSPECTION 14
SECTION 20-08 HEATING AND DEFOGGING DISTRIBUTION SYSTEM TROUBLESHOOTING GUIDE 15

SECTION 21-40 HEATING AND DEFOGGING SYSTEM 17

SECTION 40-01 HEATING SYSTEM 18
SECTION 40-02 PERIODIC MAINTENANCE 18
SECTION 40-03 HEATING SYSTEM PROCEDURES 18
HEAT EXCHANGER SHROUD REMOVAL 19
HEAT EXCHANGER SHROUD INSTALLATION 20
HEATING SYSTEM OPERATIONAL CHECK AND INSPECTION 21
SECTION 40-04 HEATING SYSTEM TROUBLESHOOTING GUIDE 23
SECTION 40-05 DEFOGGING SYSTEM 24
SECTION 40-06 PERIODIC MAINTENANCE 24
SECTION 40-07 DEFOGGING SYSTEM PROCEDURES 24
DEFOGGING DUCT REMOVAL 25
DEFOGGING DUCT INSTALLATION 26
DEFOGGING SYSTEM OPERATIONAL CHECK AND INSPECTION 27
SECTION 40-08 DEFOGGING SYSTEM TROUBLESHOOTING GUIDE 28

SECTION 21-50 TEMPERATURE CONTROL: 29

SECTION 50-01 PERIODIC MAINTENANCE 30
SECTION 50-02 TEMPERATURE CONTROL PROCEDURES 30
CABIN HEAT VENT VALVE REMOVAL 31
CABIN HEAT VENT VALVE INSTALLATION 32
HEAT BOX REMOVAL 33
HEAT BOX INSTALLATION 34
HEATER CABLE REMOVAL 35
HEATER CABLE INSTALLATION 37
TEMPERATURE CONTROL OPERATIONAL CHECK AND INSPECTION 38
SECTION 50-03 TEMPERATURE CONTROL TROUBLESHOOTING GUIDE 39

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Section 21-00 General

The heating and cooling system of the XL-2 is by means of a passive system, (no thermostatic control). Air circulation controls are located in the instrument console on the port (pilot) and starboard (co-pilot) side of the cockpit.

The heat exchanger provides heated air. The location of the heat exchanger is adjacent to the exhaust muffler with air then regulated by a heat control valve. Heated air for the passenger compartment is through the control valve, then to a hose connected to manually adjustable leg vents located on the in-board face of the foot well. Additionally, defogging capability automatically activates by pulling on the heat control knob. Partial airflow to the defogging system will occur if either of the two leg vents is open. Full airflow to the system occurs when both leg vents are closed.

Outside Air is through a vent system channeling air to directional outlets at the left and right lower corners of the instrument panel. The outlets are adjustable for pilot or co-pilot/passenger comfort. During idle conditions, there is very little airflow; therefore, cabin temperature (flow rates) will vary with aircraft movement, altitude, airspeed, and attitude. (See Section 21-00 - Distribution on page 7 of this chapter).

Additionally, a fresh air window vent is an optional item in each cabin door screen. The air window vent will aid in air circulation while in operation on the ground. Secure the window from opening prior to taxiing or flight.

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Section 21-20 Distribution Systems

The Liberty XL-2 provides a fresh air system and a cabin heat system incorporating a wind screen defogging feature. Fresh air and heated air are distributed by means of separate systems. This section addresses each of these systems.

Section 20-01 Fresh Air Distribution System

The fresh air distribution system is passive incorporating no boost fan or other active component. Airflow is achieved by forward motion of the aircraft. In operation outside air enters the passenger compartment through a NACA-type flush inlet on each side of the forward fuselage, connected, through molded, bonded duct to directional outlets at the left and right lower corners of the instrument panel. The outlet can be directed for pilot or passenger comfort. Rotating the outlet sleeve regulates the amount of outside air delivered.

Figure 21-1 Fresh Air Distribution System

Section 20-02 Periodic Maintenance

Periodic Distribution maintenance entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the inspection procedure in this section.

Section 20-03 Fresh Air Distribution System Procedures

The following procedures inspect components of the fresh air distribution system. This system integrated into composite structures of the fuselage and instrument console assembly. Corrective actions are limited to cleaning and repair of ventilation duct structures on condition and in accordance with standard practices referenced in the fresh air distribution system troubleshooting guide.

FRESH AIR DISTRIBUTION SYSTEM INSPECTION

The following procedure performs Fresh Air Distribution System Inspection. Refer to troubleshooting guide at the end of the section for corrective action.

1. Inspect port NACA duct for cracks, chipped paint or debris.
2. Inspect starboard NACA duct for cracks, chipped paint or debris.
3. Inspect port instrument console duct for cracks or debris.
4. Inspect starboard instrument console for cracks or debris.

This completes the Fresh Air Distribution System Inspection procedure.

Section 20-04 Fresh Air Distribution System Troubleshooting Guide

This section contains a guide to troubleshooting issues with the fresh air distribution system.

Complaint	Possible Cause	Remedy
NACA Duct Cracks	• Impact damage	• Perform standard repair in accordance with Chapter 51 Standard Practices – Structures.
NACA Duct Chipped Paint	• Impact damage	• Perform standard repair in accordance with Chapter 51 Standard Practices – Structures.
NACA Duct Debris	• Insects • Undeveloped field operations • Cleaning operations	• Remove debris.
Instrument Console Duct Cracks	• Impact damage	• Perform standard repair in accordance with Chapter 51 Standard Practices – Structures.
Instrument Console Duct Debris	• Insects • Undeveloped field operations • Cleaning operations	• Remove debris

Section 20-05 Heating and Defogging Distribution System

Heating and defogging distribution system components consist of Aeroduct flexible tubing secured with hose clamps to rigid interconnecting elements providing airflow distribution. Component installations are divided between the engine compartment and cabin of the XL-2 aircraft. In order to service the system access to the engine compartment, cabin and lower fuselage areas are required. Refer to the chapters referenced in procedures to follow for access.

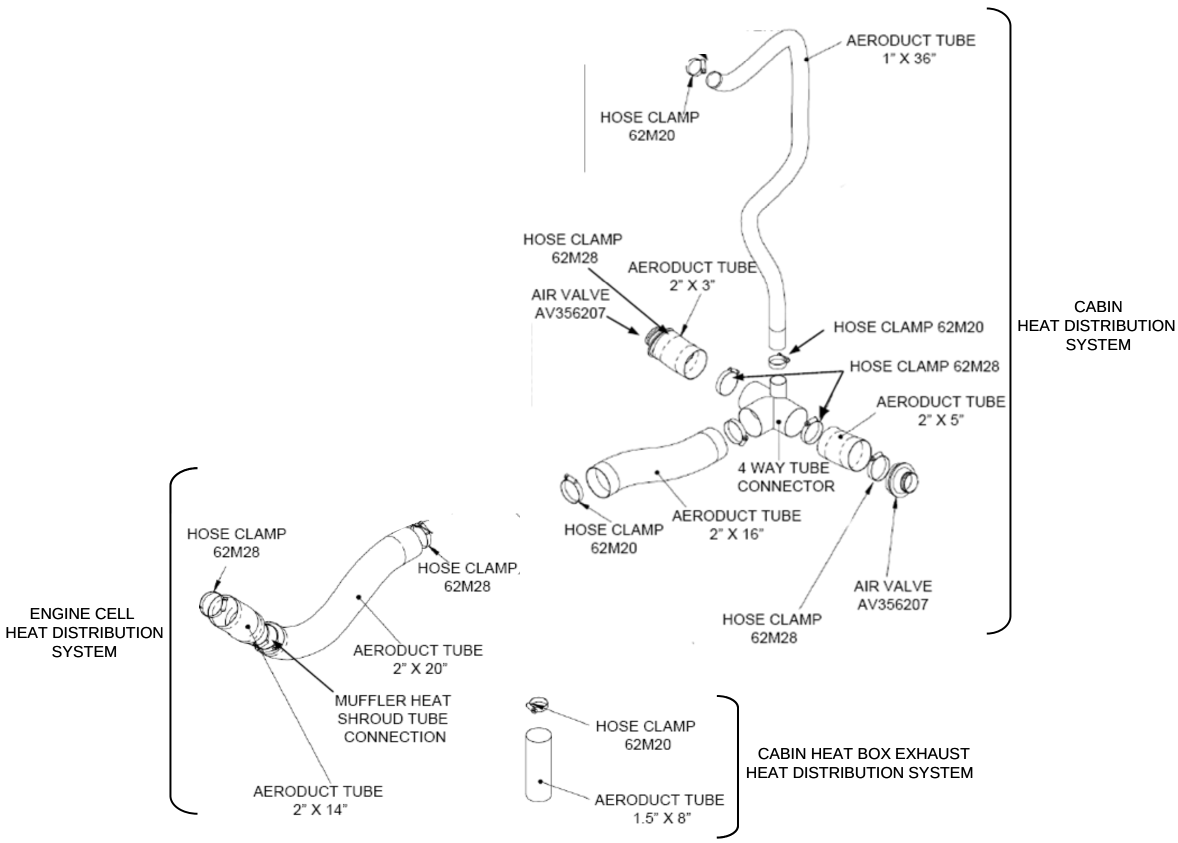

Figure 21-2 Heating and Defogging Distribution System

Section 20-06 Periodic Maintenance

Periodic Distribution maintenance entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the inspection procedure in this section. Distribution system component removal is done only on condition mandating replacement.

Section 20-07 Heating and Defogging Distribution System Procedures

AERODUCT REMOVAL PROCEDURE

This removal procedure applies to all Aeroduct installations in the engine cell and cabin. Repeat the procedure for each section of the distribution system removed.

1. Position the master switch OFF.
2. Position FADEC Power Switch A and B - OFF –
3. For each section of the distribution system requiring access perform the applicable sections from the maintenance manual chapters indicated below.

- Engine System: Upper and lower cowl removal, Chapter 71 – *Power Plant.*
- Cabin Heat System: Belly panel removal, Chapter 53 - Fuselage.
- Defog System: Avionics Panel Removal, Chapter 23 - Communications

4. Loosen the hose clamp on each secured end of the Aeroduct tube to be removed.
5. Slide each end of the Aeroduct tube back off of the attachment fitting and remove the tube.

This completes the Aeroduct Removal Procedure.

AERODUCT INSTALLATION PROCEDURE

This installation procedure applies to all Aeroduct installations in the engine cell and cabin. Repeat the procedure for each section of the distribution system installed.

1. Position the master switch OFF.
2. Position FADEC Power Switch A and B - OFF –
3. Slide each end of the Aeroduct tube onto the attachment fitting.
4. Tighten each ends hose clamp to secure the Aeroduct tube.

CAUTION

Do not over tighten the hose clamps. Damage to attachment fittings will result causing system air leakage in operation.

5. For each section of the distribution system that required access perform the applicable sections from the maintenance manual chapters indicated below.

- Engine System: Upper and lower cowl installation, Chapter 71 – *Power Plant.*
- Cabin Heat System: Belly panel installation, Chapter 53 - Fuselage.
- Defog System: Avionics Panel installation, Chapter 23 - *Communications*

6. Perform heating and defog distribution system inspection in accordance with Chapter 21 – *Environmental Systems*

This completes the Aeroduct Installation Procedure.

FOUR WAY TUBE CONNECTOR REMOVAL

This procedure performs removal of the four way tube connector assembly located below the center console.

1. Position the master switch OFF.
2. Position FADEC Power Switch A and B - OFF –
3. Remove belly panel in accordance with Chapter 53 – *Fuselage*

NOTE

For the following steps, refer to Figure 21-2 for location of components

4. Locate and loosen the forward 2 inch Aeroduct hose clamp. Slide the clamp forward clear of the four way tube connector fitting area
5. Locate and loosen the port 2 inch Aeroduct hose clamp. Slide the clamp outboard clear of the four way tube connector fitting area.
6. Locate and loosen the starboard 2 inch Aeroduct hose clamp. Slide the clamp outboard clear of the four way tube connector fitting area.
7. Slide the three, 2 inch Aeroduct tubes clear of the four way tube connector assembly.
8. Gently guide the four way tube connector assembly downward until the 1 inch Aeroduct tube located on top of the unit is accessible.
9. Loosen the 1 inch Aeroduct hose clamp. Slide the clamp upward clear of the four way tube connector assembly fitting area.
10. Slide the 1 inch Aeroduct tube clear the four way connector assembly and guide the assembly down and out of the aircraft.

This completes the Four Way Tube Connector removal procedure.

FOUR WAY TUBE CONNECTOR INSTALLATION

This procedure performs removal of the four way tube connector assembly located below the center console.

1. Position the master switch OFF.
2. Position FADEC Power Switch A and B - OFF –

NOTE

For the following steps, refer to Figure 21-2 for location of components

CAUTION

Do not over tighten the hose clamps. Damage to attachment fittings will result causing system air leakage in operation.

3. Locate and slide the 1 inch Aeroduct tube onto the four way connector and secure with hose clamp.
4. Guide the four way tube connector assembly up into location beneath the center console.
5. Position hose clamps on each of three, 2 inch Aeroduct tubes half way down their lengths.
6. Slide the forward, port and starboard 2 inch Aeroduct tubes over the corresponding four way tube connector fittings and secure with hose clamps.
7. Perform heating and defog distribution system inspection in accordance with Heating and Defogging Distribution System Operational Check and Inspection on page 14 of the chapter.
8. Install belly panel in accordance with Chapter 53 – *Fuselage*

This completes the Four Way Tube Connector Installation procedure.

HEATING AND DEFOGGING DISTRIBUTION SYSTEM OPERATIONAL CHECK AND INSPECTION

The following procedure may be performed on condition or in support of maintenance tasks called for in Chapter 05. For corrective actions refer to the heating and defogging distribution system troubleshooting guide section.

1. Position the master switch - OFF.
2. Position FADEC Power Switch A and B - OFF
3. Remove upper and lower cowl in accordance with applicable sections of Chapter 71 – *Power Plant.*
4. Remove belly panel in accordance with applicable sections of Chapter 53 – *Fuselage*.
5. Inspect heat exchanger connecting Aeroduct tubes for damage and security of attaching hardware.
6. Inspect the cabin heat control box connected Aeroduct tubing for damage and security of attaching hardware.
7. Close cabin heat vent valves located on the port and starboard sides of the center console.
8. Pull to open the cabin heat control located on the center console panel.
9. Apply a 150 to 200 CFM clean air flow to the forward engine baffle inlet.
10. Verify air is not flowing out of the overboard vent tube located beneath the cabin heat box assembly.
11. Inspect each connection point of the distribution system for indication of leaks.
12. Remove air flow from forward engine baffle inlet.
13. Install upper and lower cowl in accordance with applicable sections of Chapter 71 – *Power Plant.*
14. Install belly panel in accordance with applicable sections of Chapter 53 – *Fuselage.*

This completes the Heating and Defogging Distribution System Operational Check and Inspection procedure.

Section 20-08 Heating and Defogging Distribution System Troubleshooting Guide

This section contains a guide to troubleshooting issues with the heating and defogging distribution system.

Complaint	Possible Cause	Remedy
Aeroduct tube leaks	• Tube abraded or holed	• Replace Aeroduct Tube
	• Hose clamp loose	• Tighten
	• Four way tube connector damaged	• Replace
No air flow through tubes	• Collapsed Aeroduct tube	• Replace
	• Tube obstructed	• Remove obstruction
	• Tube disconnected from system • Adjacent tube disconnected	• Reconnect

Table 21-1 Heating Defogging Distribution System Troubleshooting Guide

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Section 21-40 Heating and Defogging System

When so equipped, the Liberty XL-2 heating and defogging system provides warm air to the aircraft cabin for crew comfort and to the wind screen condensation removal. The system as depicted in Figure 21-3 is passive containing no boost fan components. Airflow is achieved by ram air entering a forward engine baffle located port. Air is then routed to a muffler mounted heat shroud. Shroud warmed air is directed to a firewall mounted valve that mixes cool air to achieve the desired temperature. Mixture control is manual by means of a cable routed to the cabin center console within crew reach. (see Figure 21-4)

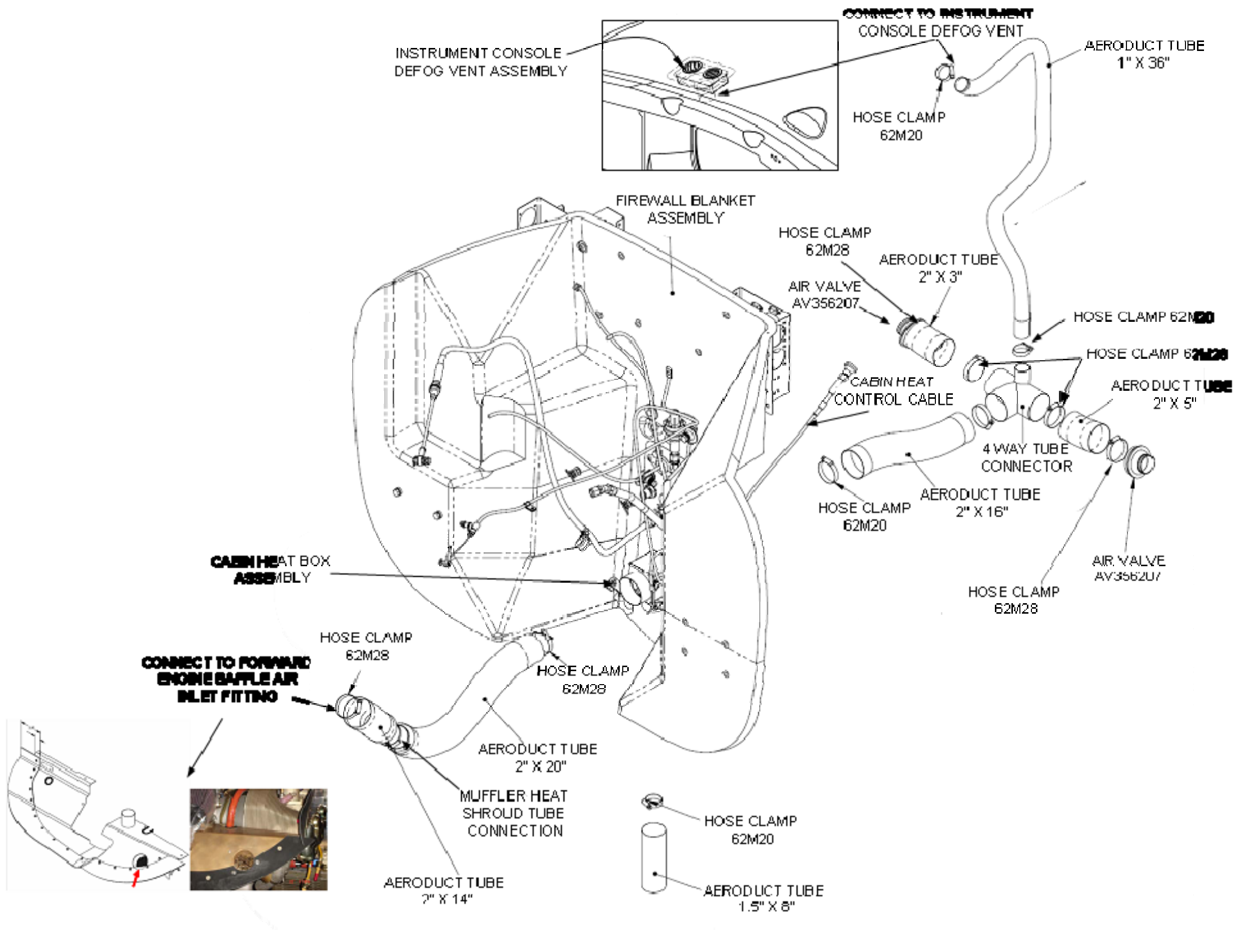

Figure 21-3 Heating and Defogging System

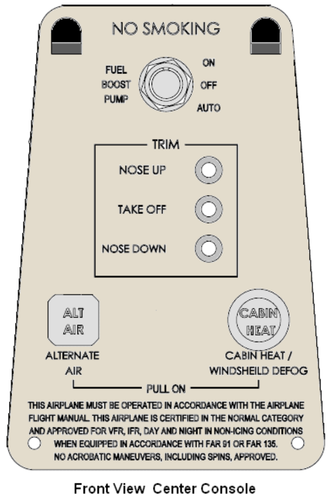

Figure 21-4 Center Console Cabin Heat Control

Section 40-01 Heating System

Heating system components consist of Aeroduct flexible tubing secured with hose clamps to rigid interconnecting elements providing airflow control and distribution. Component installations are divided between the engine compartment and cabin of the XL-2 aircraft by a cabin heat box assembly affixed to the firewall blanket assembly. In order to service the system access to the engine compartment, cabin and lower fuselage areas are required. Refer to the chapters referenced in procedures to follow for access.

Section 40-02 Periodic Maintenance

Periodic Heating System maintenance entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 40-03 Heating System Procedures

This section has the procedures to remove, install and inspect the components of the heating system. All of the air ducts on the airplane use standard screw type hose clamps to secure the duct to the heating system component.

HEAT EXCHANGER SHROUD REMOVAL

Perform this procedure to remove the heat exchanger shroud. Refer to Figure 21-5 during this procedure.

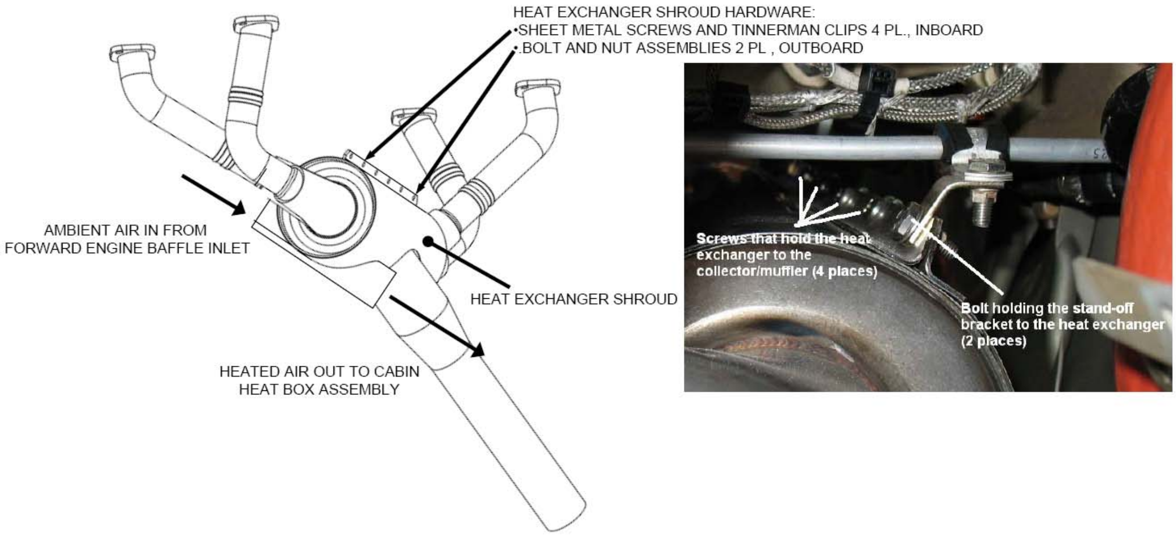

Figure 21-5 Heat Exchanger Hardware

1. Position the master switch - OFF.
2. Position FADEC Power Switch A and B - OFF.
3. Remove upper and lower cowling in accordance with applicable section of Chapter 71 – *Power Plant*.
4. Remove hose clamp securing the lower Aeroduct tube to the heat exchanger and slide the tube off of the fitting.
5. Remove hose clamp securing the upper Aeroduct tube to the heat exchanger and slide the tube off of the fitting.
6. Remove the two bolts that secure the stand-off brackets to the collector/muffler.
7. Remove the four screws that hold the heat exchanger shroud on to the collector/muffler.
8. Slowly work the heat exchanger shroud down the exhaust pipe to remove the cover from the collector/muffler.

This completes the Heat Exchanger Shroud Removal procedure

HEAT EXCHANGER SHROUD INSTALLATION

Perform this procedure to install the heat exchanger shroud.

WARNING

BEFORE INSTALLING THE HEAT EXCHANGER SHROUD, INSPECT THE ENTIRE SURFACE OF THE COLLECTOR/MUFFLER FOR CRACKS. USE A MIRROR TO INSPECT ALL SURFACES ON THE COLLECTOR/MUFFLER. IF THERE ARE ANY CRACKS FOUND IN THE COLLECTOR/MUFFLER, EITHER REPLACE THE COLLECTOR/MUFFLER OR FIX THE CRACKS BY WELDING. IF REPAIRING THE CRACKS BY WELDING, THE WELDS MUST COMPLY WITH MIL-STD-1595A USING AMS 5680 FILLER.

1. Position the master switch - OFF.
2. Position FADEC Power Switch A and B - OFF.
3. Inspect the collector/muffler for cracks. Use a mirror to inspect all surfaces on the collector/muffler.
4. Slide the heat exchanger shroud up the exhaust pipe and attach it to the collector/muffler.
5. Install the four screws removed in the Heat Exchanger Shroud Removal procedure of this chapter.
6. Install the two bolts removed in the Heat Exchanger Shroud Removal procedure of this chapter.
7. Slide a hose clamp over the lower Aeroduct tube.
8. Slide the lower Aeroduct tube over the lower heat exchanger fitting and secure with the hose clamp.
9. Slide a hose clamp over the upper Aeroduct tube.
10. Slide the upper Aeroduct tube over the heat exchanger fitting and secure with the hose clamp.
11. Perform the heating system operational check and inspection in accordance with the applicable sections of Chapter 21 – *Environmental Systems.*
12. Install the lower and upper cowling in accordance with applicable section of Chapter 71 – *Power Plant*.

This completes the Heat Exchanger Shroud Installation procedure.

HEATING SYSTEM OPERATIONAL CHECK AND INSPECTION

The following procedure may be performed on condition or in support of maintenance tasks called for in Chapter 05. For corrective actions refer to the heating system troubleshooting guide section.

1. Position the master switch - OFF.
2. Position FADEC Power Switch A and B - OFF
3. Remove upper and lower cowl in accordance with applicable sections of Chapter 71 – *Power Plant.*
4. Remove belly panel in accordance with applicable sections of Chapter 53 – *Fuselage*.
5. Visually inspect the heat exchanger shroud. Fit of heat shroud around muffler and tail pipe must have no air gaps larger than .02”.
6. Inspect heat exchanger connecting Aeroduct tubes for damage and security of attaching hardware.
7. Inspect the cabin heat control box for damage and security of connected Aeroduct tubing.
8. Cycle the cabin heat control cable and verify full range of motion can be achieved with no binding or interruption.
9. Close cabin heat vent valves located on the port and starboard sides of the center console.
10. Pull to open the cabin heat control located on the center console panel.
11. Apply a 150 to 200 CFM clean air flow to the forward engine baffle inlet.
12. Verify air is not flowing out of the overboard vent tube located beneath the cabin heat box assembly.
13. Cycle open the port cabin heat vent valve. Verify smooth and complete vent operation and a steady steam of air in the open position.
14. Close the port cabin heat vent valve.
15. Cycle open the starboard cabin heat vent valve. Verify smooth and complete vent operation and a steady steam of air in the open position.
16. Push to close the cabin heat control located on the center console panel.
17. Verify airflow discontinues from the starboard cabin heat valve.
18. Remove air flow from forward engine baffle inlet.
19. Install upper and lower cowl in accordance with applicable sections of Chapter 71 – *Power Plant.*
20. Install belly panel in accordance with applicable sections of Chapter 53 – *Fuselage.*
21. Position the aircraft in a safe run up area.

22. Start the aircraft engine in accordance with Aircraft flight manual procedures.
23. Pull the center console located cabin heat control to the open position.
24. Open the cabin heat vent valves.
25. Run up the engine to 2000 RPM and verify warm air flow out of the cabin heat vent valves.
26. Reduce engine speed to idle.
27. Close cabin heat vent valves.
28. Run up the engine to 2000 RPM and verify warm air flow out of the upper instrument console defogging vents.
29. Reduce RPM to idle and shut down in accordance with Airplane Flight Manual procedures.

This completes the Heating System Operational Check and Inspection procedure.

Section 40-04 Heating System Troubleshooting Guide

This section contains a guide to troubleshooting issues with the heating system.

Complaint	Possible Cause	Remedy
None or restricted air flow from cabin heating vent valves.	• Damaged or collapsed Aeroduct tube	• Replace
	• Disconnected heating system Aeroduct tube	• Connect Aeroduct tube
	• Faulty cabin heat box assembly	• Replace
	• Faulty cabin heat control cable	• Replace
	• Faulty cabin heat control cable connection to cabin heat box assembly	• Repair connection
	• Cabin heat valves obstructed	• Clear cabin heat valve obstruction
Cold cabin heat vent valve air temperature	• Disconnected Aeroduct tube forward of cabin heat box assembly	• Reconnect Aeroduct tube
	• Damaged Aeroduct tube forward of the cabin heat box assembly.	• Replace damaged Aeroduct
	• Cabin heat box flapper range of travel limited	• Inspect and repair flapper • Adjust cabin heat control cable connection to achieve full travel
	• Heat exchanger shroud penetration	• Repair shroud
	• Heat exchanger shroud loose	• Reinstall in accordance with applicable sections of Chapter 21 – *Environmental Systems*
Engine fumes detected in cabin with heat on.	• Exhaust leak beneath heat exchanger shroud	• Inspect, repair or replace in accordance with applicable sections of Chapter 78 – *Exhaust.*
	• Aeroduct tube forward or the cabin heat box damaged	• Replace Aeroduct tube
	• Aeroduct tube disconnected forward of cabin heat box.	• Reconnect.

Table 21-2 Heating System Troubleshooting Guide

Section 40-05 Defogging System

Air is provided to defog louvers automatically when the heat control knob is pulled. Full airflow for defogging occurs when the two leg vents are closed (see Figure 21-6), note placard above leg vents. There are two vents installed in the upper center of the instrument consoled, see Figure 21-7.

Figure 21-6 Cabin Heat Placard

Partial air-flow for defogging is available when the vents are open. Since very little air flow occurs during idle conditions, before taxiing and flight, the interior and exterior of the windshield must be cleaned of any ice, precipitation, fog or mist.

Figure 21-7 Defog Louvers

The Liberty XL-2 airplane has a demist cloth which must be used during ground operations over the cabin heat defogging system.

Section 40-06 Periodic Maintenance

Periodic Defogging System maintenance entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 40-07 Defogging System Procedures

This section has the procedures to remove, install and inspect the components of the defogging system. Replacement of the components is performed on condition only. Periodic maintenance procedures do not require removal of these components. Inspection procedures included in this section support periodic maintenance called for in Chapter 05.

DEFOGGING DUCT REMOVAL

Perform this procedure to remove the defogging duct.

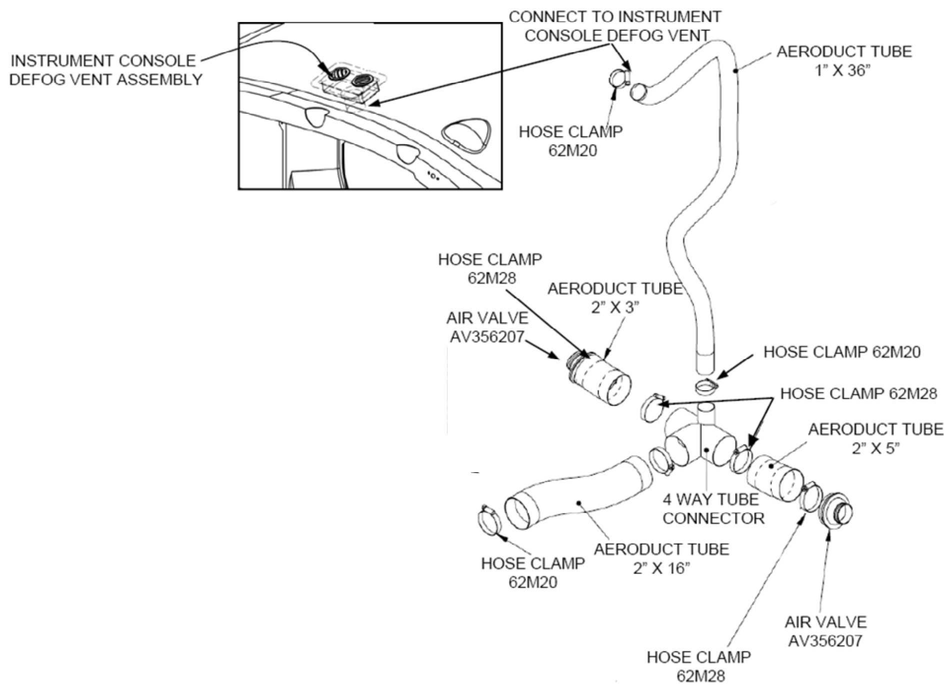

Figure 21-8 Defogging System

1. Position the master switch - OFF.
2. Position FADEC Power Switch A and B - OFF
3. Remove belly panel in accordance with the applicable section of Chapter 53 – *Fuselage*.
4. Remove avionics panel in accordance with applicable section of Chapter 23 – *Communications*.
5. Loosen and slide hose clamp from 1 inch Aeroduct defogging tube end located on top of the four way tube connector assembly as shown in Figure 21-8.
6. Slide the defogging tube off of the four way tube connector assembly.
7. Loosen and slide hose clamp from 1 inch Aeroduct defogging tube end located on the instrument console vent assembly as shown in Figure 21-8.
8. Slide Aeroduct defogging tube along with related hose clamps down and out of the fuselage.
9. Inspect instrument console vent assembly for damage and debris. Remove any debris found.

This completes the Defogging Duct Removal procedure.

DEFOGGING DUCT INSTALLATION

Perform this procedure to install the defogging duct.

1. Position the master switch - OFF.
2. Position FADEC Power Switch A and B - OFF.
3. Slide the 1 inch Aeroduct tube up from the four way tube connector assembly into the instrument console. Continue routing up to the defogging vent assembly as shown in Figure 21-8.

CAUTION

On completion of tube routing, check for interference with adjacent structures, wiring and avionics panel assembly chassis clearances. Reposition defogging tube as required to assure the tube will not be compressed or damaged. Improper routing will prevent avionics panel installation and diminished defogging performance.

4. Slide hose clamp on to the Aeroduct tube end connecting to the instrument console vent assembly.
5. Slide the Aeroduct tube over the instrument console vent assembly fitting and secure with hose clamp.
6. Slide hose clamp on to the Aeroduct tube end connecting to the four way tube connector assembly.
7. Slide the Aeroduct tube over the four way tube connector assembly fitting and secure with hose clamp.
8. Install avionics panel in accordance with applicable section of Chapter 23 – *Communications*.
9. Perform defogging system operational check and inspection in accordance with Chapter 21 – *Environmental Systems*.
10. Install belly panel in accordance with the applicable section of Chapter 53 – Fuselage.

This completes the Defogging Duct Installation procedure.

DEFOGGING SYSTEM OPERATIONAL CHECK AND INSPECTION

The following procedure may be performed on condition or in support of maintenance tasks called for in Chapter 05. For corrective actions refer to the defogging system troubleshooting guide section.

1. Position the master switch - OFF.
2. Position FADEC Power Switch A and B - OFF.
3. Remove belly panel in accordance with the applicable section of Chapter 53 – *Fuselage*.
4. Visually inspect the defogging tube running from the four way tube connector assembly to the instrument console vent assembly. Verify there are no penetrations wear points or deformations.
5. Close cabin heat vent valves located on the port and starboard sides of the center console.
6. Pull to open the cabin heat control located on the center console panel.
7. Apply a 150 to 200 CFM clean air flow to the forward engine baffle inlet and verify air flow out of the defogging vents is occurring with no indication of air leaks.
8. Close cabin heat control and verify air flow discontinues from defogging vents.
9. Remove air flow from forward engine baffle inlet.
10. Install belly panel in accordance with Chapter 53 – Fuselage.

This completes the Defogging System Operational Check and Inspection.

Section 40-08 Defogging System Troubleshooting Guide

This section contains a guide to troubleshooting issues with the defogging system.

Complaint	Possible Cause	Remedy
No or restricted air flow from defogging vents	• Damaged or collapsed defogger Aeroduct tube	• Replace
	• Disconnected defogger Aeroduct tube	• Connect Aeroduct tube
	• Faulty cabin heat box assembly	• Replace
	• Faulty cabin heat control cable	• Replace
	• Faulty cabin heat control cable connection to cabin heat box assembly	• Repair connection
	• Cabin heat valves opened	• Close cabin heat valves
Cold defogging vent air temperature	• Disconnected Aeroduct tube forward of cabin heat box assembly	• Reconnect Aeroduct tube
	• Cabin heat box flapper range of travel limited	• Inspect and repair flapper • Adjust cabin heat control cable connection to achieve full travel

Table 21-3 Defogging System Troubleshooting Guide

Section 21-50 Temperature Control:

Cabin temperature control in the XL-2 is provided by means of manual controls incorporated in the fresh air vent system and the cabin heat system.

Fresh air is provided by the opening of vent valves located in the port and starboard sides of the instrument console as shown in Figure 21-9 below.

Figure 21-9 Starboard Instrument Console Vent Valve

Heated Air is provided by pulling the heat control knob on the center console, with air flow then regulated by rotating the leg vents located on the in-board face of the foot well.

Figure 21-10 Center Console Cabin Heat Temperature Control

Figure 21-11 Cabin Heat Vent Valve Installation

Control of external air is by means of opening or closing the vent valve located on each lower corner of the instrument panel (see Figure 21-10 and Figure 21-11).

Section 50-01 Periodic Maintenance

Periodic Heating System maintenance entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 50-02 Temperature Control Procedures

The following procedures remove and replace temperature control components of heating system. Remove and replace procedures are used on condition mandating removal and replacement of components in accordance with guidance provided by this sections troubleshooting guide.

CABIN HEAT VENT VALVE REMOVAL

This procedure performs removal of the heat system vent valve assemblies. This procedure may be applied to either heat vent valve removal required.

1. Position the master switch - OFF.
2. Position FADEC Power Switch A and B - OFF.
3. Perform belly panel removal in accordance with Chapter 53 - *Fuselage*.
4. Loosen Aeroduct hose clamp attached to the vent valve to be removed and slide it back clear of the vent valve fitting.
5. Slide the 2 Inch Aeroduct tube clear of the vent valve assembly
6. Loosen and remove the vent valve assembly locking ring as shown in Figure 21-12.

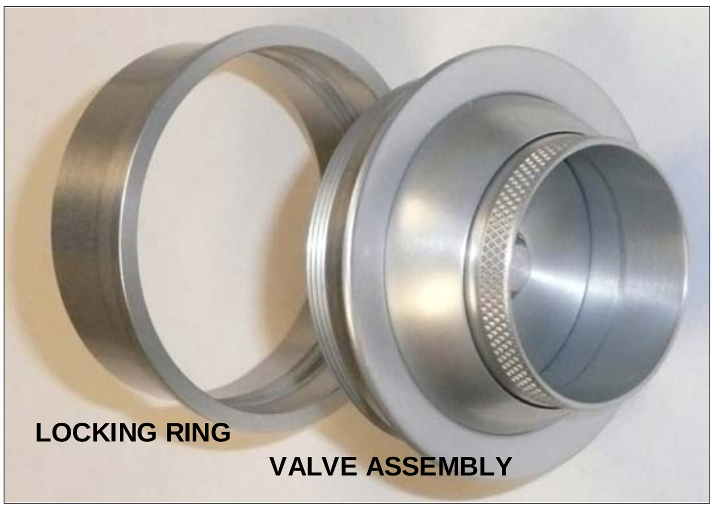

Figure 21-12 Vent Valve Assembly

7. Slide the vent valve assembly out of the panel.

This completes the Cabin Heat Vent Valve Removal procedure.

CABIN HEAT VENT VALVE INSTALLATION

This procedure performs installation of the heat system vent valve assemblies. This procedure may be applied to either heat vent valve installation required

1. Position the master switch - OFF.
2. Position FADEC Power Switch A and B - OFF
3. If not already performed, remove the vent valve assembly locking ring.
4. Slide the vent valve assembly into the console and secure with the locking ring.
5. Position the hose clamp over the 2 inch Aeroduct.
6. Slide Aeroduct over the vent valve assembly flange followed by the hose clamp. Ensure the clamp is over the vent valve flange area.
7. Tighten the hose clamp.
8. Perform temperature control system operation check and inspection at the end of this section.
9. Install the belly panel in accordance with Chapter 53 – *Fuselage*.

This completes the Cabin Heat Vent Valve Installation procedure.

HEAT BOX REMOVAL

Perform this procedure to remove the heat box from the airplane. During this procedure, refer to Figure 21-13.

1. Position the master switch - OFF.
2. Position FADEC Power Switch A and B - OFF.
3. Remove upper and lower cowling in accordance with applicable sections of Chapter 71 – *Power Plant*.
4. Remove the belly panel in accordance with applicable sections of Chapter 53 –*Fuselage*.

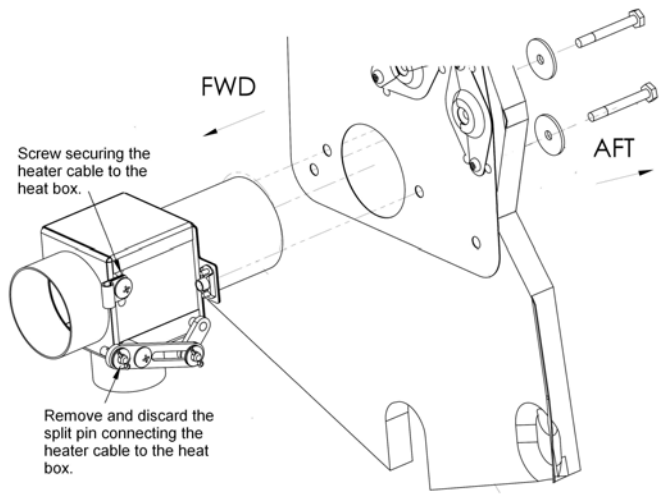

Figure 21-13 Cabin Heat Box Assembly Installation

5. Disconnect clamp next to the heat box that secures the heat exchanger to cabin heater box hose.
6. Disconnect clamp next to the heat box that secures the heat box to cabin heating vent.
7. Remove the two bolts that secure the heat box to the firewall.
8. Carefully remove enough of the fire caulk to enable the removal of the heat box.
9. Loosen heat box from the firewall of the airplane.
10. Cut and discard the cotter or split pin securing the heater cable to the heat box.
11. Loosen the screw securing the heater cable to the heat box and slide cable clear of control arm.

This completes the Heat Box Removal procedure.

HEAT BOX INSTALLATION

Perform this procedure to remove the heat box from the airplane. During this procedure, refer to Figure 21-13.

1. Position the master switch - OFF.
2. Position FADEC Power Switch A and B - OFF.
3. Ensure all electrical switches are off.
4. If install a heat box after removing the previous heat box go to step 7.
5. Remove upper cowling. Refer to Chapter 71 – *Power Plant*.
6. At the port side of lower cowling, disconnect landing light; remove lower cowling. Refer to Chapter 71 – *Power Plant*.
7. Remove the belly panel. Refer to Chapter 53 –*Fuselage*.
8. Insert the heater cable into the clamp on the heat box.
9. Attach the cable to the heat box control. Do not secure at this time with cotter keys.
10. Insert the heat box into the firewall and secure the heat box with the bolts removed in step of the procedure. Torque the bolts in accordance with Chapter 20 – *Standard Practices*.
11. Apply a generous amount of Fire Caulk around the opening in the firewall where the heat box comes through.
12. Attach the ducts to the heat box and secure with the hose clamps removed.
13. Adjust heat control cable for full travel and secure with cotter keys.
14. Install the engine cowlings. Refer to Chapter 71 – *Power Plant*.
15. Install the belly panel. Refer to Chapter 53 –*Fuselage*.

This completes the Heat Box Installation procedure.

HEATER CABLE REMOVAL

Perform this procedure to remove the heater cable from the airplane.

1. Position the master switch - OFF.
2. Position FADEC Power Switch A and B - OFF.
3. Remove upper cowling. Refer to Chapter 71 – *Power Plant*.
4. At the port side of lower cowling, disconnect landing light; remove lower cowling. Refer to Chapter 71 – *Power Plant*.
5. Remove the belly panel. Refer to Chapter 53 –*Fuselage*.
6. Cut and discard the cotter or split pin securing the heater cable to the heat box.
7. Loosen the screw securing the heater cable to the heat box.
8. Remove the two screws securing the firewall grommet shield to the firewall.
9. Remove enough of the fire caulk to allow the heater cable to pass through the firewall.
10. From inside the cockpit, remove the two screws holding in the center console panel assy. Refer to Figure 21-14 for the location of the two screws.
11. Remove the access panel on the starboard side of the center console.
12. Reach through the access hole and disconnect the wires leading to the two post lights.
13. Reach through the access hole and using a ¼-inch socket to loosen and remove the nuts and lock washers on the underside of the post lights.

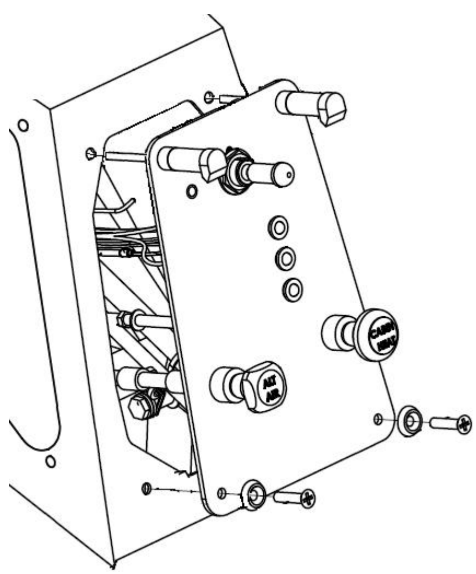

Figure 21-14 Center Console Panel

14. Carefully pull the center console panel out.

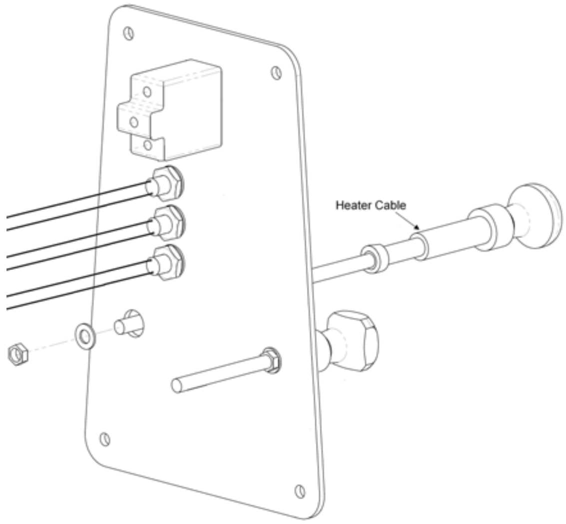

Figure 21-15 Heater Cable Assembly

15. Remove the nut and washer as shown in Figure 21-15.
16. Carefully work the cable up and out of the center console of the airplane.
17. If installing the cable later, secure the center console, install the belly panel, and the engine cowlings.

This completes the Heater Cable Removal procedure.

HEATER CABLE INSTALLATION

Perform this procedure to install a heater cable.

1. Position the master switch - OFF.
2. Position FADEC Power Switch A and B - OFF
3. Remove upper cowling. Refer to Chapter 71 – *Power Plant*.
4. At the port side of lower cowling, disconnect landing light; remove lower cowling. Refer to Chapter 71 – *Power Plant*.
5. Remove the belly panel. Refer to Chapter 53 –*Fuselage*.
6. Open up the Center Console.
7. Insert the heater cable into the center console
8. Secure the cable with the washer and nut as shown in Figure 21-15.
9. Feed the cable down through the center of the space frame towards the firewall.
10. Feed the cable through the firewall.
11. Secure the cable to the firewall with the firewall grommet shield with the two screws removed in step 8 in the Heater Cable Removal procedure.
12. Apply sufficient quantity of Fire Caulk to the aft side of the firewall blanket to seal the cable to the firewall.
13. Secure the cable to the heat box using the clamp and screw removed in step 7 in the Heater Cable Removal procedure.
14. Connect the heater cable to the heat box lever. Secure the cable using a new cotter or split pin.
15. Install the center console using the hardware removed previously in the Heater Cable Removal procedure.

This completes the Heater Cable Installation procedure.

TEMPERATURE CONTROL OPERATIONAL CHECK AND INSPECTION

The following procedure may be performed on condition or in support of maintenance tasks called for in Chapter 05. For corrective actions refer to the defogging system troubleshooting guide section.

1. Position the master switch - OFF.
2. Position FADEC Power Switch A and B - OFF
3. Remove upper and lower cowl in accordance with applicable sections of Chapter 71 – *Power Plant.*
4. Visually inspect the firewall mounted cabin heat control assembly for damage and secure connection of the cabin heat control cable.
5. Cycle the cabin heat control cable and verify full range of motion can be achieved with no binding or interruption.
6. Close cabin heat vent valves located on the port and starboard sides of the center console.
7. Pull to open the cabin heat control located on the center console panel.
8. Apply a 150 to 200 CFM clean air flow to the forward engine baffle inlet.
9. Verify air is not flowing out of the overboard vent tube located beneath the cabin heat box assembly.
10. Cycle open the port cabin heat vent valve. Verify smooth and complete vent operation and a steady steam of air in the open position.
11. Close the port cabin heat vent valve.
12. Cycle open the starboard cabin heat vent valve. Verify smooth and complete vent operation and a steady steam of air in the open position.
13. Push to close the cabin heat control located on the center console panel.
14. Verify airflow discontinues from the starboard cabin heat valve.
15. Remove air flow from forward engine baffle inlet.
16. Install upper and lower cowl in accordance with applicable sections of Chapter 71 – *Power Plant.*

This completes the Temperature Control Operational Check and Inspection procedure.

Section 50-03 Temperature Control Troubleshooting Guide

This section contains a guide to troubleshooting issues with temperature control system. The fresh air, heating and defog systems temperature control are addressed here.

Complaint	Possible Cause	Remedy
None or restricted air flow from heating vent valves	• Damaged or collapsed heating Aeroduct tube	• Replace
	• Disconnected heating Aeroduct tube	• Connect Aeroduct tube
	• Faulty cabin heat box assembly	• Replace
	• Faulty cabin heat control cable	• Replace
	• Faulty cabin heat control cable connection to cabin heat box assembly	• Repair connection
	• Cabin heat valves not opened	• Open cabin heat valves
Cold heat vent valve air temperature	• Disconnected Aeroduct tube forward of cabin heat box assembly	• Reconnect Aeroduct tube
	• Cabin heat box flapper range of travel limited	• Inspect and repair flapper • Adjust cabin heat control cable connection to achieve full travel
Heating vent valve will not close or will not open	• Faulty vent valve • Corroded vent valve	• Replace vent valve
	• Debris in the vent valve plate	• Clear debris

Table 21-4 Temperature Control Troubleshooting Guide

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CHAPTER 23

COMMUNICATIONS

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Table of Contents

SECTION 23-00	GENERAL	5
SECTION 23-10	SPEECH COMMUNICATIONS	9
SECTION 10-01	GARMIN GNS430 COM/NAV/GPS	9
SECTION 10-02	GARMIN GNS 530 COM/NAV/GPS	10
SECTION 10-03	GARMIN GNS 430W AND GNS 530W COM/NAV/GPS WAAS	11
SECTION 10-04	GARMIN SL30 COM/NAV RADIO	11
SECTION 10-05	GARMIN SL40 COM RADIO	12
SECTION 10-06	PUSH-TO-TALK CONTROL OF THE COMMUNICATIONS	12
SECTION 10-07	PERIODIC MAINTENANCE	12
SECTION 10-08	VHF TRANSCEIVER PROCEDURES	12
TRANSCEIVER REMOVAL		13
TRANSCEIVER INSTALLATION		14
AVIONICS PANEL REMOVAL		15
AVIONICS PANEL INSTALLATION		17
TRANSCEIVER OPERATION CHECK AND INSPECTION		18
SECTION 10-09	TROUBLESHOOTING GUIDE	19
SECTION 10-10	COM/NAV/GPS ANTENNA	19
SECTION 10-11	PERIODIC MAINTENANCE	20
SECTION 10-12	COM/NAV/GPS ANTENNA PROCEDURES	21
MODEL CI 2480-200 ANTENNA REMOVAL		22
MODEL CI 2480-200 ANTENNA INSTALLATION		25
MODEL CI 2580-200 ANTENNA REMOVAL		28
MODEL CI 2580-200 ANTENNA INSTALLATION		31
COM/NAV/GPS ANTENNA INSPECTION		34
SECTION 10-13	TROUBLESHOOTING GUIDE	36
SECTION 23-50	AUDIO INTEGRATION	37
SECTION 50-01	GMA 340 AUDIO/ICS SYSTEM	37
SECTION 50-02	AUDIO SYSTEM PROCEDURES	38
GMA 340 AUDIO PANEL REMOVAL		39
GMA 340 AUDIO PANEL INSTALLATION		40
GMA340 OPERATION CHECK AND INSPECTION		41
SECTION 50-03	TROUBLESHOOTING GUIDES	42

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Section 23-00 General

The XL-2 has a Garmin VHF radio system, used for communication between the airplane and a ground station or between airplanes. The system operates in the frequency ranges of 118.000 to 136.975 MHz, and has 760 channels with a separation of 25 kHz. Up to two communications radios may be available for use depending on the avionics options package installed. Options packages are comprised of Garmin COM/NAV, COM/NAV/GPS and Audio Panel units of the following model types:

- GMA 340 Audio Panel
- GNS 530 COM/NAV/GPS
- GNS 430 COM/NAV/GPS
- GNS 430W COM/NAV/GPS WAAS
- GNS 530W COM/NAV/GPS WAAS
- SL 30 COM/NAV
- SL 40 COM

Table 23-1 shows the different option packages built from the units above; permit the installation of one of the following Garmin communication radio configurations. All of these options come with a GMA340 audio panel. See Figure 23-1 for a view of typical installations of the communication transceivers.

COM 1 Position		COM 2 Position	
Model	Description	Model	Description
GNS 530 (or W)	COM/NAV/GPS		
GNS 530 (or W)	COM/NAV/GPS	GNS 430 (or W)	COM/NAV/GPS
GNS 530 (or W)	COM/NAV/GPS	SL30	COM/NAV
GNS 530 (or W)	COM/NAV/GPS	SL40	COM/NAV
GNS 430 (or W)	COM/NAV/GPS		
GNS 430 (or W)	COM/NAV/GPS	GNS 430 (or W)	COM/NAV/GPS
GNS 430 (or W)	COM/NAV/GPS	SL30	COM/NAV
GNS 430 (or W)	COM/NAV/GPS	SL40	COM
SL30	COM/NAV		
SL30	COM/NAV	SL30	COM/NAV
SL30	COM/NAV	SL40	COM
SL40	COM		

Table 23-1 Table Showing the Various Avionics Packages Available for the XL2 Airplane

Figure 23-1 Typical Radio Installation on the left, is the GNS530/430 typical installation, on the right, is the GNS 430W/SL30 typical installation

An avionics upgrade option is available to configurations listed above that replaces GNS 430 and GNS 530 COM/NAV/GPS units with GPS WAAS (Wide Area Augmentation System) capable GNS 430W and GNS 530W COM/NAV/GPS units. WAAS capable GNS units look identical to their Non-WAAS predecessors. Refer to the aircraft equipment list. This list has the information on the versions of the radios installed in the airplane. On power up of the GNS430/530 transceivers will display a screen similar (depending on model) to the screen shown in Figure 23-2. Verify the unit identification and software revisions from this display.

GNS Model Identifier GNS 430W or GNS 530W

- Main Software Revision
- GPS Software Revision
  - If not 3.0 or greater perform Garmin Ltd. Software Service Bulletin 0740 Revision A, dated November 29, 2007

Figure 23-2 GNS Model and Software Verification

In addition to the communication radio stack fitted to the center instrument console, there are two antennas fitted in a forward and aft position as shown in Figure 23-3. The forward antenna is typically for the COM1 radio and the aft antenna for the COM2 radio for all radio installation options. Audio connections for the pilot and co-pilot are made by an overhead audio jack panel, see Figure 23-19, located center and aft of the seat backs. There is no cabin speaker installed in the airplane. Transmit switches (PTT) are located at the tip of the pilot and co-pilot control sticks.

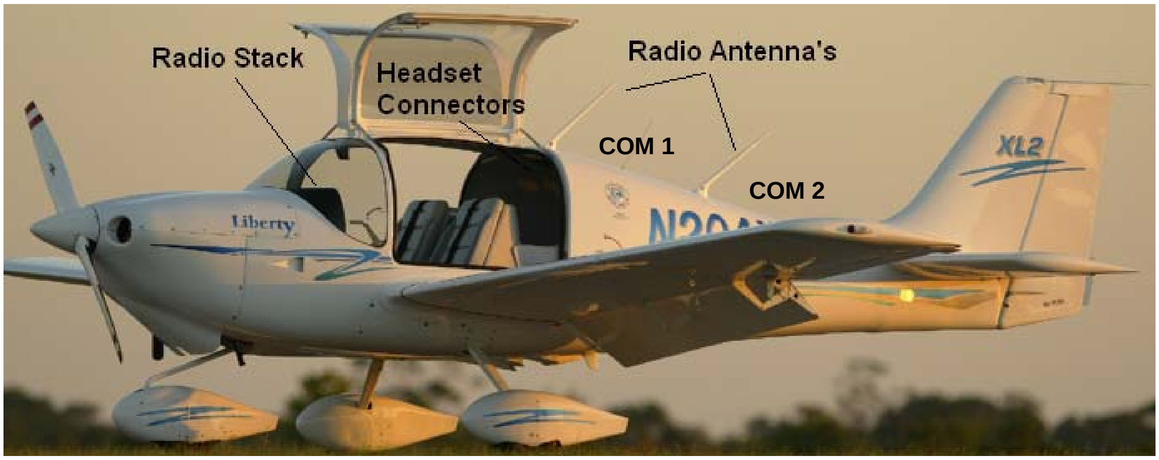

Figure 23-3 Communications Equipment Placement

The alternator, and/or battery provide power for the avionics. The power goes through the Avionics Master switch and the COM circuit breaker. There is a dedicated circuit breaker for each radio. The size of circuit breaker matches the radio option installed.

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Section 23-10 Speech Communications

On the Liberty Aerospace, Inc. XL-2 airplane, speech communications is by VHF transceivers mounted in the avionics panel. There can be several models of VHF transceiver(s) installed in the airplane. Unless otherwise ordered, airplanes come equipped with transceivers from Garmin. The models installed in the airplane may include the GNS 530, GNS 430, SL 30, SL 40, GNS 530W, and/or the GNS 430W transceiver.

Section 10-01 Garmin GNS430 COM/NAV/GPS

The GNS 430 COM/NAV/GPS transceiver provides radio communication between pilot and ground stations, and VOR/LOC/GS/GPS reception. The installation of the GNS 430 transceiver can be in either the COM 1 or COM 2 stack position of the avionics panel depending on the radio option package installed.

NOTE

The description of the NAV and GPS subsystem may be found in Liberty Maintenance Manual Chapter 34-Navigation and Pitot/Static.

The GNS 430 series units are 6.25 inch wide by 2.66 inch high. The display is a 128 by 240-pixel color LCD. The units include two removable data cards, one with a Jeppesen database, and the second an optional custom data card.

CAUTION

The display of the GNS 430 has a special anti-reflective coating, which is very sensitive to skin oils, waxes and abrasive cleaners. It is very important to clean the lens with a lint-free cloth using an eyeglass lens cleaner specified safe for antireflective coatings.

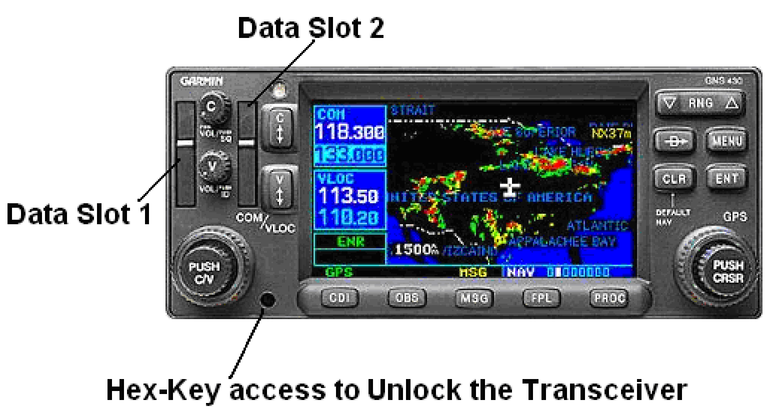

Figure 23-4 GNS 430 Face Plate Key Slot

The GNS 430 GPS receiver is certified for IFR en route, terminal and non-precision approach operations. The VHF communications transceiver is an IFR certified airborne unit. VOR/Localizer and Glideslope receivers are also IFR certified.

The Garmin user guide and installation manual for the GNS 430 COM/NAV/GPS receiver has additional information on the control functions and features.

Section 10-02 Garmin GNS 530 COM/NAV/GPS

The GNS 530 COM/NAV/GPS transceiver provides radio communication between pilot and ground stations. This radio also provides VOR/LOC GS and GPS reception. The typically installation of the GNS 530 is in the COM 1 stack position of the avionics panel depending on the radio option package installed.

NOTE

The description of the NAV and GPS subsystem is in Chapter 34.

The GNS 530 series units are 6.25 inch wide by 4.60 inch high. The display is a 234 by 320-pixel color LCD. The radio includes two removable data cards, one with a Jeppesen database, and the second an optional custom data card.

CAUTION

The display of the GNS 530 has a special anti-reflective coating, which is very sensitive to skin oils, waxes and abrasive cleaners. It is very important to clean the lens with a lint-free cloth using an eyeglass lens cleaner specified safe for antireflective coatings.

Figure 23-5 GNS 530 Face Plate Key Slot

The VHF communications transceiver is an IFR certified airborne unit. The VOR/Localizer and Glideslope receivers are also IFR certified. The GNS 530 GPS receiver is IFR certified for en route, terminal, and non-precision approach operations.

The Garmin user guide and installation manual for the GNS 530 COM/NAV/GPS receiver has additional information on the control functions and features.

Section 10-03 Garmin GNS 430W and GNS 530W COM/NAV/GPS WAAS

The GNS 430W and 530W represent upgraded versions of the GNS 430 and GNS 530 units described above. Physical appearance, control appearance, and operation are the same as the GNS 430 and GNS 530. Enhancements installed in the GNS 430W and GNS 530W units permit precision GPS navigation utilizing WAAS technology.

The GNS 430W series units are 6.25 inches wide 2.66 inches high. The display is a 128 by 240-pixel color LCD. The units include two removable data cards, one with Jeppesen database inserted in the left card slot and the second being a terrain database inserted in the right card slot.

The GNS 530W series units are 6.25 inches wide 4.60 inches high. The display is a 234 by 320-pixel color LCD. The units include two removable data cards, one with Jeppesen database inserted in the left card slot and the second being a terrain database inserted in the right card slot.

The GNS 430W and GNS 530W are WAAS GPS units that meet the requirements of Technical Standard Order (TSO) C146a and may be approved for IFR en-route, terminal, non-precision and precision approach operations when installed in reference to the OEM instructions as referenced in AML STC SA01933LA.

Both the GNS 430W and GNS 530W units include a VHF communications transceiver, VOR/Localizer and glideslope receiver. These are the same features and capabilities offered by the non-WAAS GNS 430 and GNS 530 versions.

The Garmin user guides and installation manuals for the GNS 430W COM/NAV/GPS GNS 530W COM/NAV/GPS receivers have additional information on the control functions and features. See Section 10-09 Troubleshooting Guide on page 19 of this chapter.

Section 10-04 Garmin SL30 COM/NAV Radio

The SL30 COM/NAV transceiver provides radio communication between pilot and ground stations, and VOR/LOC/GS reception. Install the transceiver in either the COM1 or COM2 avionics panel position depending on the radio option package installed. Typical installations pair the SL30 as COM2 and a GNS series radio as COM1.

NOTE

The description of the NAV subsystem is in Chapter 34.

The SL30 series radio is a slim line chassis design measuring 1.3 inches by 6.26 inches. The SL30 contains a 760-channel VHF communications transceiver with 200 channel VOR, Localizer, and Glide slope receivers. The Garmin user guide and installation manual for the SL30 COM/NAV receiver has additional information on the control functions and features.

Figure 23-6 SL30 Face Plate Key Slot

Section 10-05 Garmin SL40 COM Radio

The SL40 COM transceiver provides radio communication between pilot and ground stations. Installation can be in either the COM1 or COM2 stack position of the avionics panel depending on the radio option package installed. Typical installations pair a GNS series radio in the COM1 position with the SL40 radio in the COM2 position. The SL40 series radio is a slim line chassis design measuring 1.3 inches by 6.26 inches. The Garmin user guide and installation manual for the SL30 COM receiver has additional information on the control functions and features.

Figure 23-7 SL40 Face Plate Key Slot

Section 10-06 Push-to-Talk Control of the Communications

Push-to-Talk control of the communication receivers is from a push-to-talk push button mounted to the yoke control. Because the button is part of the yoke control, procedures to remove and install the push-to-talk button are in Chapter 27 – *Control Surfaces*.

Section 10-07 Periodic Maintenance

Periodic maintenance of this unit entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 10-08 VHF Transceiver Procedures

This section contains the removal and installation procedures.

NOTE

Only an appropriately equipped and certificated avionics maintenance facility can perform maintenance operations beyond simple removal or replacement of a communications transceiver.

TRANSCEIVER REMOVAL

Perform the following procedure to remove the transceiver. Use this procedure for any of the radios installed in the avionics panel.

1. Check that both aircraft battery switch and avionics master switch are OFF.
2. Insert hex key ("3/32" Allen wrench") into hole in faceplate of the radio and rotate counterclockwise until the retaining claw is released. (See illustration on previous page).
3. Slide the radio out of the mounting tray.

This completes the Transceiver Removal procedure.

TRANSCEIVER INSTALLATION

Perform the following procedures to install the transceiver. Use this procedure for any of the radios installed in the avionics panel.

1. Check that both aircraft battery switch and avionics master switch are OFF.
2. Check mating connectors on rear of radio and in mounting tray to check the integrity of all connections. Verify pins are straight and at the proper depth.
3. Verify that radio-retaining claw is in released position. If necessary, insert hex key into the hole in the radio faceplate and turn CCW.
4. Slide the radio into the mounting tray until connectors engage.
5. Insert hex key into the hole in the radio faceplate and turn counterclockwise until the radio firmly seats into the mounting tray.
6. Perform operation check of the COM/GPS system described in Transceiver Operation Check and Inspection on page 18 of this chapter.

This completes the Transceiver Installation procedure.

AVIONICS PANEL REMOVAL

Perform the following procedure to remove the avionics panel from the airplane. Liberty Aerospace, Inc. recommends performing this procedure while sitting in the pilot's seat. Have the block of soft foam cushioning for the avionics panel in your lap that it covers the yoke assembly.

NOTE

This procedure describes the removal of all possible equipment available for the airplane. Some of the equipment and/or options called out in this procedure may not be in the airplane.

1. Remove instrument panel from airplane (see Chapter 31 – *Indicators and Recording Systems*).
2. Remove circuit breaker (CB) panel from airplane (see Chapter 24 – *Electrical Power*).
3. Remove the avionics radios from the avionics panel. To remove the radios, insert a hex key ("3/32" Allen wrench") into the hole in faceplate of the radio and rotate counter-clockwise until the retaining claw is released. Slide the radio towards you and out of the radio tray.

NOTE

Removal of the avionics radios reduces the weight of the avionics panel. This will make the panel easier to handle and less likely to damage the Instrument Panel Console assembly surfaces when removing the panel from the console.

4. Place the radio outside of the airplane.
5. Remove the eight screws holding in the avionics panel assy. Refer to for the location of the eight screws.

This completes the avionics panel removal procedure.

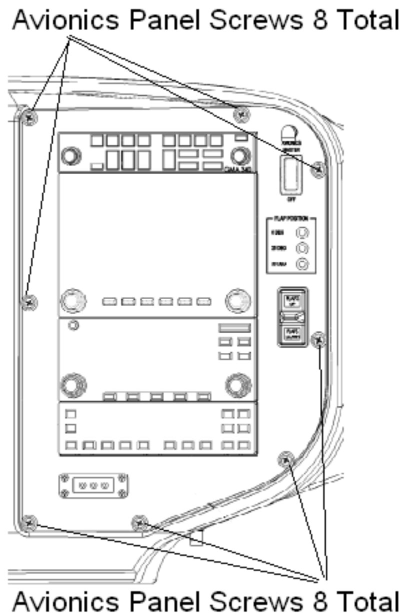

Figure 23-8 Avionics Panel Showing the Location of the Screws

6. Disconnect the connectors P5354 and P5355 from the altitude encoder.
7. Disconnect the transponder antenna cable ATXP01 from the transponder tray.
8. Disconnect the marker beacon antenna cable AMB01B from the cable AMB01A.
9. Disconnect the RJ13 type (telephone jack) connector from the back of the Emergency Locator Transmitter, ELT, and remote unit.
10. Disconnect the COM-01 cable ACOM01A from the cable ACOM01B.
11. Disconnect the COM-02 cable ACOM02A from the cable ACOM02B.
12. Disconnect J41 from the connector P/J41on the bracket on the power distribution harness.
13. Disconnect J07 from the connector P/J07on the bracket on the power distribution harness.
14. Disconnect the NAV/LOC/GLS antenna cable, AVLOC01A from the antenna/signal splitter mounted on the side of the chassis of the avionics panel.
15. Remove the avionics panel and place face down in to a block of soft foam cushioning. Remove the avionics panel from the airplane, keeping the panel on its face into a block of soft foam cushioning.

This completes the avionics panel removal procedure.

AVIONICS PANEL INSTALLATION

Perform the following procedure to install the avionics panel.

1. Retrieve the avionics panel removed in step 5 in the procedure Avionics Panel on Page 15.
2. Connect J41 to P41 on the bracket on the right side of the power distribution harness.
3. Connect J07 to P07 on the bracket on the right side of the power distribution harness.
4. If the airplane has one or two Course Deviation Indicators, CDI, route their respective cable(s) from the avionics panel out the opening for the instrument panel.
5. Connect the COM-02 cable ACOM02A to the cable ACOM02B.
6. Connect the COM-01 cable ACOM01A to the cable ACOM01B.
7. Connect the RJ13 type (telephone jack) connector to the back of the Emergency Locator Transmitter remote unit.
8. Connect the marker beacon antenna cable AMB01B to the cable AMB01A.
9. Connect the transponder antenna cable ATXP01 to the transponder tray.
10. Connect the connectors P5354 and P5355 to the altitude encoder.
11. Connect the NAV/LOC/GLS antenna cable, AVLOC01A to the antenna/signal splitter mounted on the side of the chassis of the avionics panel.
12. Retrieve the avionics radios removed in step 4 in the procedure Avionics Panel on Page 15.
13. Check mating connectors on rear of radio and in the mounting tray to check the integrity of all connections. Check that pins are straight and at the proper depth.
14. Verify that radio-retaining claw is in released position. If necessary, insert hex key (“3/32” Allen wrench”) into the hole in the radio faceplate and turn counter-clockwise.
15. Slide the radio into the mounting tray until connectors engage.
16. Insert hex key (“3/32” Allen wrench”) into the hole in the radio faceplate and turn counter-clockwise until the radio is firmly seated in the mounting tray.
17. Install the circuit breaker panel in accordance with Chapter 24 –*Electrical Power.*
18. Install instrument panel in accordance with Chapter 31–*Indicators and Recording Systems*

This completes the avionics panel installation procedure.

TRANSCEIVER OPERATION CHECK AND INSPECTION

Upon completing the installation of the transceiver, perform an operational check of the radio per the Garmin Ltd. OEM installation manual Post Installation Checkout procedure. Refer to Table 23-2 for applicable manual and section. In all cases, use the latest manual version.

MODEL	DOCUMENT	TITLE	SECTION	NOTES
SL40	560-0956-03	INSTALLATION MANUAL	2.0	Contact GARMIN Ltd. Dealer
SL30	560-0404-03	INSTALLATION MANUAL	2.0	Contact GARMIN Ltd. Dealer
GNS 430	190-00140-02	INSTALLATION MANUAL	5.0	Contact GARMIN Ltd. Dealer
GNS 430W	190-00356-02	INSTALLATION MANUAL	5.0	Contact GARMIN Ltd. Dealer
GNS 530	190-00181-02	INSTALLATION MANUAL	5.0	Contact GARMIN Ltd. Dealer
GNS 530W	190-00357-02	INSTALLATION MANUAL	5.0	Contact GARMIN Ltd. Dealer

Table 23-2 Table Showing the Documents Required to Perform the Operation Check and Inspection

This completes the Transceiver Operation Check and Inspection procedure.

Section 10-09 Troubleshooting Guide

Table 23-3 gives a basic troubleshooting guide for the Garman radios.

Complaint	Possible Cause	Remedy
Unable to transmit or receive; communications transceiver display dark	Defective power supply wiring	• Repair
	Defective COM circuit breaker	• Replace in accordance with Liberty Maintenance Manual Chapter 24- *Electrical Power*
	If other avionics are also without power: defective avionics master switch or relay	• Repair/replace
	Defective COM unit	• Repair/replace
Unable to transmit from pilot side	Defective PTT switch on pilot side	• Replace
	Defective wiring between pilot PTT switch and audio/intercom panel	• Repair
Unable to transmit from copilot side	Defective PTT switch on Copilot side	• Replace
	Defective wiring between copilot PTT switch and audio/intercom panel	• Repair
No modulation when transmitting from pilot side	Defective headset / microphone	• Replace
	Defective wiring	• Repair
No modulation when transmitting from copilot side	Defective headset / Microphone	• Replace
	Defective wiring	• Repair
Weak transmission	Antenna fault	• Replace
	Microphone fault	• Replace
	Defective COM unit	• Repair/replace
Weak reception	Antenna fault	• Replace
	Headphone fault	• Replace
	Defective COM unit	• Replace
	Audio panel fault	• Repair/Replace

Table 23-3 Troubleshooting Guide for the Garman Radios

Section 10-10 COM/NAV/GPS Antenna

This section details the information about the COM/NAV/GPS antennas installed on the airplane. Antenna installations are found on the top of the airplane aft of the doors. The forward antenna position is for COM/NAV/GPS 1 antenna, and the aft antenna position is for the COM/NAV/GPS 2 antenna.

The Liberty XL-2 uses one of two model COM/NAV/GPS antennas. The first is a model CI 2480-200 manufactured by Comant Industries. This model will support the GNS 430 and GNS 530 model NAV/COM/GPS (non-WAAS). This model antenna is designed for standard GPS radios. It is not able to support the higher precision requirements of GNS 430W and GNS 530W WAAS capable NAV/COM systems. GNS 430W and GNS 530W NAV/COM/GPS systems are supported by Comant Industries ComDat COM/NAV/GPS antenna model CI 2580-200 designed specifically to meet GPS WAAS Gamma 3 specifications. Both models of antenna are physically identical in appearance as show in Figure 23-9 but are not identical in performance.

Figure 23-9 CI 2480-200 and CI 2580-200 Common Appearance

CAUTION

Care must be taken during maintenance operations to verify the type of GPS navigation radio installed. WAAS and Non-WAAS GPS antenna models must match the radio they are connected to. Use of the incorrect antenna model will result in significant degradation of GPS navigation accuracy.

Some radio installation options on the XL-2 include the Garmin SL30 COM/NAV or SL40 COM radios. These units do not use the GPS portion of the antenna; therefore, they can connect to either model Comant COM/NAV/GPS antenna.

CAUTION

Prior to antenna removal and installation operations, verify avionics options and antennas installed. Connecting a non-WAAS radio to WAAS antenna or connecting a WAAS radio to a non-WAAS antenna will cause degraded GPS performance.

Section 10-11 Periodic Maintenance

Periodic maintenance of this system entails inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the inspection procedure in this section.

Section 10-12 COM/NAV/GPS Antenna Procedures

For the two models of COM/NAV/GPS antennas that may be installed on the Liberty XL-2 inspection procedures are performed to determine if physical damage or deterioration has occurred that could degrade antenna performance. Antenna removal and replacement procedures are provided in the event replacement of damaged antenna is required.

MODEL CI 2480-200 ANTENNA REMOVAL

Perform this procedure to remove the model CI 2480-200 Antenna. Refer to Figure 23-10 and Figure 23-11 during this procedure.

This section has removal and installation procedures for the COM/NAV/GPS antennas. Perform the following removal and installation procedures in the event there is an issue with the COM/NAV/GPS antenna. The procedure applies to both the forward and the aft antenna positions.

CAUTION

Failure to follow the correct removal and installation procedure will result in degraded antenna performance.

CAUTION

Before starting these procedures, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

CAUTION

GPS sections of the antennas are sensitive to Electrostatic Discharge (ESD). This procedure requires the use of ESD protection procedures to reduce the possibility of damage due to ESD.

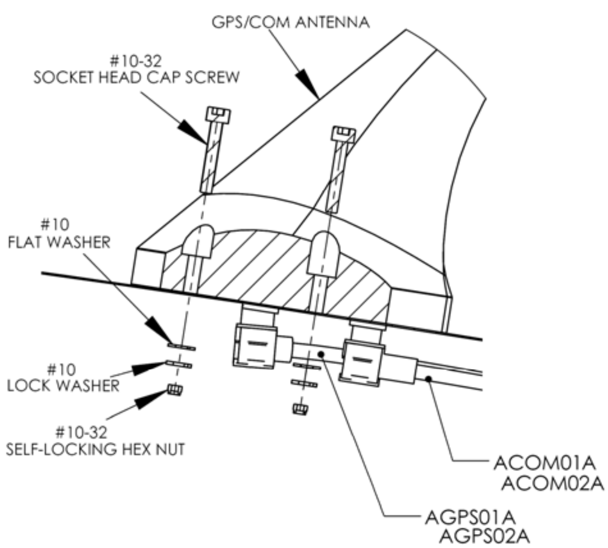

Figure 23-10 CI 2480-200 COM/GPS Antenna Installation details

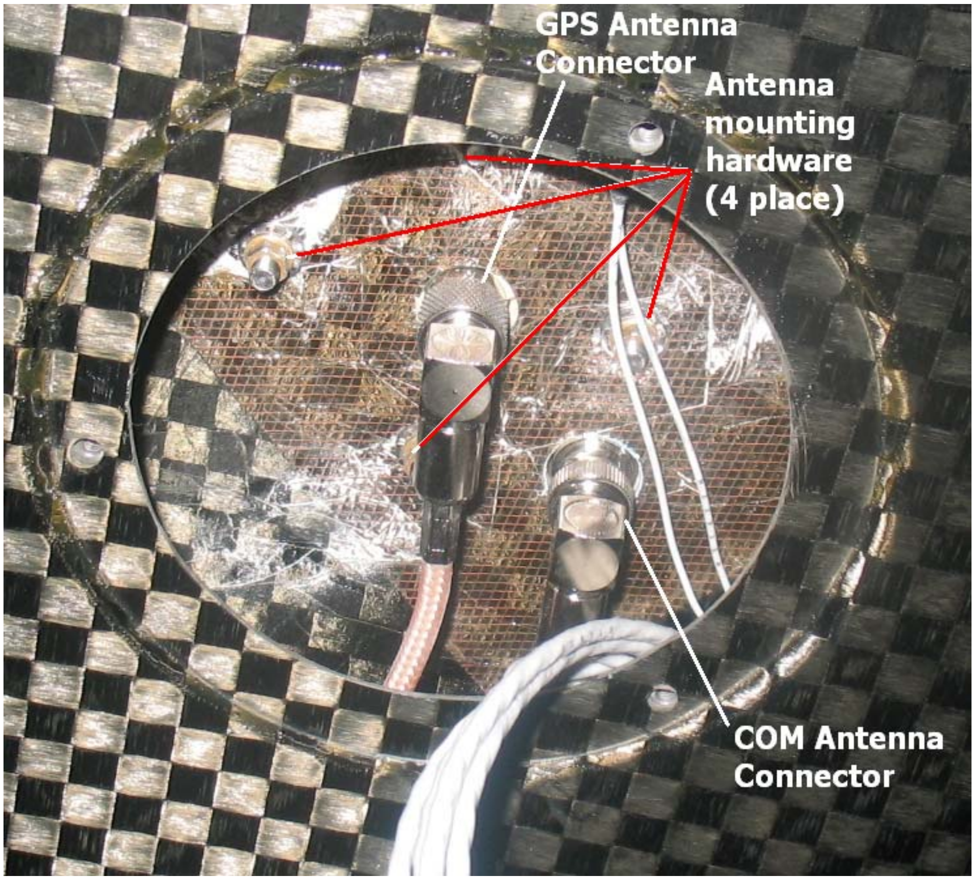

Figure 23-11 **View of the Underside of the Forward COM/NAV/GPS Antenna**

1. Position aircraft master switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Connect an ESD wrist straps to a convenient chassis ground.
4. If removing the aft COM/NAV/GPS antenna, remove the cabin aft bulkhead access panel, by removing securing screw hardware, then go to step 6 of this procedure.
5. If working on the forward most COM/GPS antenna, remove the three screws securing the audio jack panel.
6. Disconnect COM antenna cable (ACOM01B or ACOM02B) from the COM connector.
7. Disconnect GPS antenna cable (AGPS01B or AGPS02B) from the GPS connector.
8. Remove four (4) antenna mounting screws and hardware.

CAUTION

Do not use any type of prying device to remove the antenna from the fuselage as this may cause damage to the fuselage.

9. Apply a gentle rocking motion and twisting motion to the antenna body to release bonding material and remove the antenna from the upper fuselage.
10. Install the ESD covers on to the connectors of the antenna.

This completes the Model CI 2480-200 Antenna Removal procedure

MODEL CI 2480-200 ANTENNA INSTALLATION

Perform this procedure to install the model CI 2480-200 Antenna.

This section has removal and installation procedures for the COM/NAV/GPS antennas. Perform the following removal and installation procedures in the event there is an issue with the COM/NAV/GPS antenna. The procedure applies to both the forward and the aft antenna positions.

CAUTION

Failure to follow the correct removal and installation procedure will result in degraded antenna performance in the form of weak COM radio reception or transmission.

CAUTION

GPS sections of the antennas are sensitive to Electrostatic Discharge (ESD). This procedure requires the use of ESD protection procedures to reduce the possibility of damage due to ESD.

CAUTION

Before starting these procedures, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position aircraft master switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Mask-off an area of the fuselage that will be covered by the antenna.

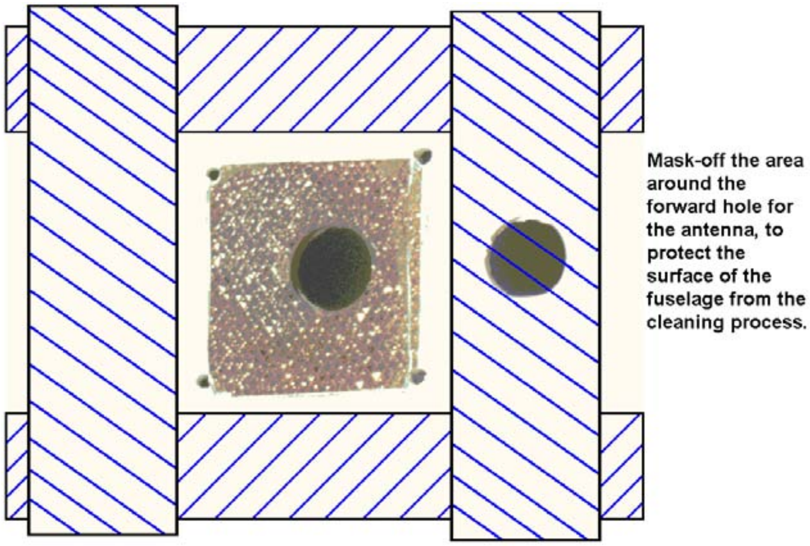

Figure 23-12 Area of the Fuselage to Mask-Off to Protect the Fuselage

4. Using fine (220-grit) sandpaper and/or a Scotch-Brite™ 7447 (maroon) hand pad, carefully clean residual sealant and debris within the rectangular area defined by the mounting holes. The goal here is to clean the carbon surface of debris without removing the carbon fibers.
5. Clean the surface with acetone and clean, lint free rags. Do not allow the acetone to air dry on the bonding surfaces. Dry the solvent using clean, lint free rags. Liberty Aerospace, Inc. recommends the use of two hands, one with a solvent dampened rag, and one following with a dry rag.
6. Continue wiping operation until the drying rag comes up clean.
7. Using de-natured alcohol, repeat the cleaning operations in steps 5 and 6.
8. Remove masking material.
9. Apply a bead of electrically conductive, MMS-040 Silver Coated Nickel Electrically Conductive RTV Silicone (Moreau Marketing & Sales, http:// http://www.rmoreau.com) along the inner edge of the gelcoat/carbon interface, taking care to stay a safe distance from the antenna cable hole. The goal here is to apply just enough RTV to get good contact between the antenna base and the carbon, but not so much that the RTV creeps out from under the antenna base when the antenna is installed. See Figure 23-13 for details of the location for the RTV sealant.

CAUTION

Do not allow the conductive RTV Silicone to come in contact with either the BNC or TNC RF connectors on the base of the antenna. The conductive RTV Silicone can short out the connector and can cause damage to the radio and/or the antenna.

CAUTION

Silastic, RTV or Silicone-Based sealing/caulking compounds are not to be used around the base or over the screw fillets. The high dielectric content of these materials distort satellite reception at low angles of elevation.

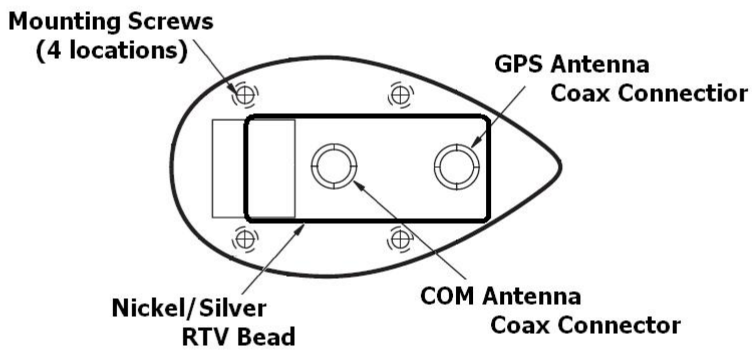

Figure 23-13 COM/GPS Antenna Conductive Sealant (bottom view)

10. Remove the protective ESD plugs.
11. Place the antenna through the fuselage.
12. Connect an ESD wrist straps to a convenient chassis ground.
13. If installing the aft COM/NAV/GPS antenna, remove the cabin aft bulkhead access panel, by removing securing screw hardware, then go to step 15 of this procedure.
14. If installing the forward COM/NAV/GPS antenna, remove the three screws securing the audio jack panel.
15. Mount the antenna using screws, washers, nuts. Gently tighten the mounting hardware so that uniform stress is placed on each side of the antenna. #10-32 screws DO NOT exceed 23 in-lbs of torque.
16. Connect the GPS coax antenna lead.
17. Connect the COM coax antenna lead
18. Check continuity between a mounting screw and the chassis for < 0.5 ohms resistance (0.003 ohms is ideal).
19. If working on the aft most COM/NAV/GPS antenna, install cabin aft bulkhead access panel then go to step 21 of this procedure.
20. If working on the forward most COM/NAV/GPS antenna, use the three screws to install the audio jack panel.
21. Perform operation check of the COM/GPS system described in Transceiver Operation Check and Inspection on page 18 of this chapter.

This completes the Model CI 2480-200 Antenna Installation procedure.

MODEL CI 2580-200 ANTENNA REMOVAL

Perform this procedure to remove the model CI 2580-200 Antenna. Refer to Figure 23-14 and Figure 23-15 during this procedure.

This section has removal and installation procedures for the COM/NAV/GPS antennas. Perform the following removal and installation procedures in the event there is an issue with the COM/NAV/GPS antenna. The procedure applies to both the forward and the aft antenna positions.

CAUTION

Failure to follow the correct removal and installation procedure will result in degraded antenna performance in the form of weak COM radio reception or transmission.

CAUTION

GPS sections of the antennas are sensitive to Electrostatic Discharge (ESD). This procedure requires the use of ESD protection procedures to reduce the possibility of damage due to ESD.

CAUTION

Before starting these procedures, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

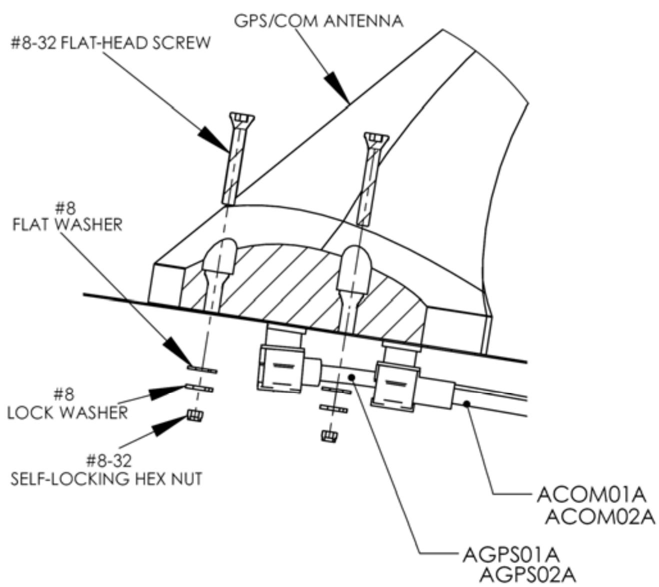

Figure 23-14 CI 2580-200 COM/GPS Antenna Installation details

NOTE

Examine the screws used to mount the antenna to the fuselage. If the screws are the socket cap screw type, then replace the screws with #8-32 flat head screws.

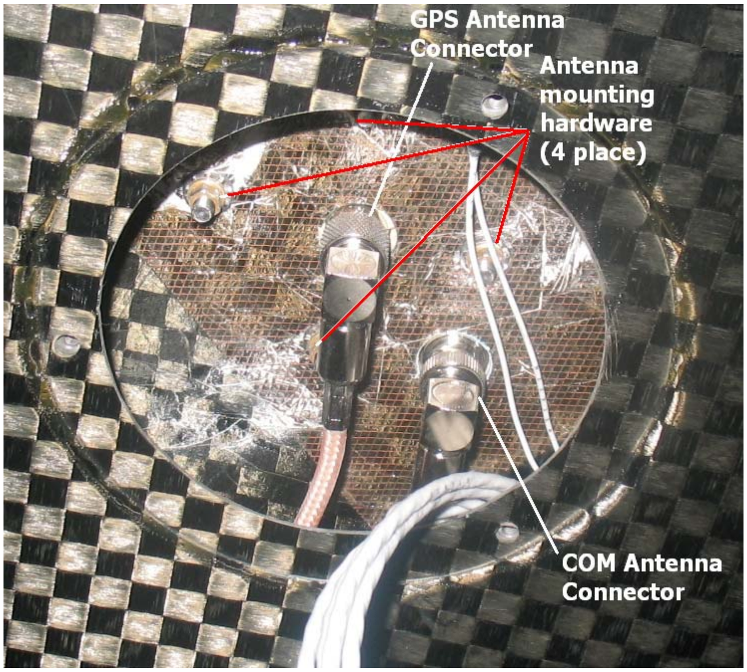

Figure 23-15 View of the Underside of the COM/GPS Antenna

1. Position aircraft master switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. .Connect an ESD wrist straps to a convenient chassis ground.
4. If working on the aft most COM/GPS antenna, remove the cabin aft bulkhead access panel, then go to step 6 of this procedure.
5. If working on the forward most COM/GPS antenna, remove the three screws securing the audio jack panel.
6. Disconnect COM antenna cable (ACOM01B or ACOM02B) from the COM connector.
7. Disconnect GPS antenna cable (AGPS01B or AGPS02B) from the GPS connector.
8. Remove four (4) antenna mounting screws and hardware.

CAUTION

Do not use any type of prying device to remove the antenna from the fuselage as this may cause damage to the fuselage.

9. Apply a gentle rocking motion and twisting motion to the antenna body to release bonding material and remove the antenna from the fuselage.
10. Install the ESD covers on to the connectors of the antenna.

This completes the Model CI 2580-200 Antenna Removal procedure.

MODEL CI 2580-200 ANTENNA INSTALLATION

Perform this procedure to install the model CI 2580-200 Antenna.

This section has removal and installation procedures for the COM/NAV/GPS antennas. Perform the following removal and installation procedures in the event there is an issue with the COM/NAV/GPS antenna. The procedure applies to both the forward and the aft antenna positions.

CAUTION

Failure to follow the correct removal and installation procedure will result in degraded antenna performance in the form of weak COM radio reception or transmission.

CAUTION

GPS sections of the antennas are sensitive to Electrostatic Discharge (ESD). This procedure requires the use of ESD protection procedures to reduce the possibility of damage due to ESD.

CAUTION

Before starting these procedures, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment

1. Position aircraft master switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Mask-off an area of the fuselage that will be covered by the antenna.

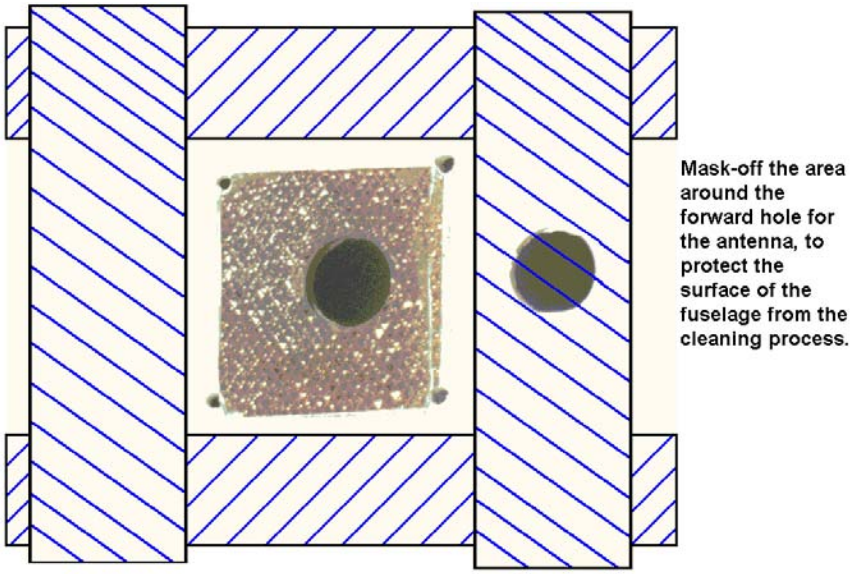

Figure 23-16 Area of the Fuselage to Mask-Off to Protect the Fuselage

4. Using fine (220-grit) sandpaper and/or a Scotch-Brite™ 7447 (maroon) hand pad, carefully clean residual sealant and debris within the rectangular area defined by the mounting holes. The goal here is to clean the carbon surface of debris without removing the carbon fibers.
5. Clean the surface with acetone and clean, lint free rags. Do not allow the acetone to air dry on the bonding surfaces. Dry the solvent using clean, lint free rags. Liberty Aerospace, Inc. recommends the use of two hands, one with a solvent dampened rag, and one following with a dry rag.
6. Continue wiping operation until the drying rag comes up clean.
7. Using de-natured alcohol, repeat the cleaning operations in steps 5 and 6.
8. Remove masking material.
9. Apply a bead of electrically conductive, MMS-040 Silver Coated Nickel Electrically Conductive RTV Silicone (Moreau Marketing & Sales, http://http://www.rmoreau.com) along the inner edge of the gelcoat/carbon interface, taking care to stay a safe distance from the antenna cable hole. The goal here is to apply just enough RTV to get good contact between the antenna base and the carbon, but not so much that the RTV creeps out from under the antenna base when the antenna is installed. See Figure 23-13 for details of the location for the RTV sealant.

Do not allow the conductive RTV Silicone to come in contact with either the BNC or TNC RF connectors on the base of the antenna. The conductive RTV Silicone can short out the connector and can cause damage to the radio and/or the antenna.

Silastic, RTV or Silicone-Based sealing/caulking compounds are not to be used around the base or over the screw fillets. The high dielectric content of these materials distort satellite reception at low angles of elevation

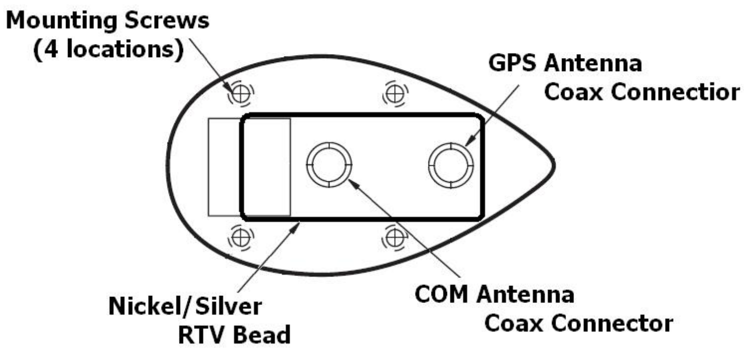

Figure 23-17 COM/GPS Antenna Conductive Sealant

10. Remove the protective ESD plugs and place the antenna through the fuselage.
11. Connect an ESD wrist straps to a convenient chassis ground.
12. If installing the aft COM/NAV/GPS antenna, remove the cabin aft bulkhead access panel, by removing securing screw hardware, then go to step 14 of this procedure.
13. If installing the forward COM/NAV/GPS antenna, remove the three screws securing the audio jack panel.
14. Mount the antenna using screws, washers, nuts. Examine the screws used to mount the antenna to the fuselage. If the screws are the socket cap screw type, then replace the screws with #8-32 flat head screws. Gently tighten the mounting hardware so that uniform stress is placed on each side of the antenna. DO NOT exceed 23 in-lbs of torque.
15. Connect the GPS coax antenna lead.
16. Connect the COM coax antenna lead
17. Check continuity between a mounting screw and the chassis for < 0.5 ohms resistance (0.003 ohms is ideal).
18. If working on the aft most COM/GPS antenna, install cabin aft bulkhead access panel then go to step 20 of this procedure.
19. If working on the forward most COM/GPS antenna, use the three screws to install the audio jack panel.
20. Perform operation check of the COM/GPS system described in Transceiver Operation Check and Inspection on page 18 of this chapter.

This completes the Model CI 2580-200 Antenna Installation procedure.

COM/NAV/GPS ANTENNA INSPECTION

This procedure performs antenna inspections. Use this procedure in support of inspections identified in Liberty Maintenance Manual Chapter 05.

1. Position the aircraft on a level surface
2. Install a tail stand

CAUTION

The following steps require access to the aft fuselage section. Failure to install a tail stand prior to accessing this area could result in damage to the aircraft and possible injury.

3. Locate and remove the cabin aft access panel
4. Locate and remove the audio jack panel in the cabin ceiling
5. Inspect COM/NAV/GPS antenna 1, forward and 2 aft for the following:
   - Loose or missing mounting hardware
   - Corrosion of mounting hardware
   - worn or loose connectors
   - Damaged coaxial antenna cable

NOTE

The following steps will inspect antenna exteriors for condition including presence of contamination around the antenna base. Antennas must not have any foreign material on the sides of the antenna base. GPS reception performance can be degraded by presence of foreign material. If material is found in this area remove it.

6. Inspect COM/NAV/GPS antenna 1, forward and 2, aft exterior installation for the following:
   - Looseness
   - Evidence if cracking
   - Corrosion of mounting hardware
   - Missing mounting hardware
   - Inspect antenna and antenna base for presence of foreign material including dirt, oil, and wax and sealing material.

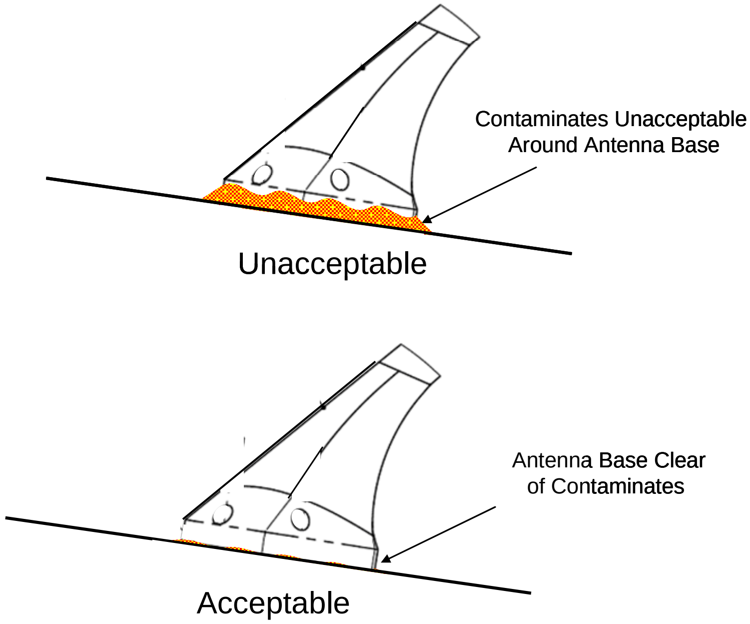

Figure 23-18 Antenna Contamination Inspection

7. Check continuity between a mounting screw and the chassis for < 0.5 ohms resistance (0.003 ohms is ideal).
8. Install audio jack panel in the cabin ceiling
9. Install cabin aft access panel
10. Remove tail stand

This completes the COM/NAV/GPS antenna inspection.

Section 10-13 Troubleshooting Guide

Complaint	Possible Cause	Remedy
Antenna cracked	In service wear	• Replace
	Impact damage	• Replace
Poor antenna performance	Physical antenna damage	• Replace
	Coax cable connection fault	• Repair
	Excess bonding resistance	• Remove, inspect and reinstall antenna in accordance with Liberty Maintenance Manual Chapter 23
Loose antenna	Incorrect installation	• Remove and reinstall in accordance with Liberty Maintenance Manual Chapter 23

Section 23-50 Audio Integration

The XL2 airplane comes with an integrated audio selector panel/intercom system installed. The transmitter selector switch routes audio from the pilot and copilot microphones to the #1 or #2 communications transmitters. The receiver selector switches allow audio from the #1 or #2 communications receivers, the VHF navigation receiver(s), and the marker beacon receiver to be routed to the crew headsets.

The intercom system is of the voice actuated "hot mike," type which means no additional action is required from the pilot or co-pilot for in-cockpit communications. Pressing the push-to-talk switches located on either control stick actuates the #1 or #2 communications transmitter as selected by the COM switch.

Headphones, including types with an integrated boom microphone, connect via standard headphone (1/4") and aircraft microphone (3/16") jacks. The location of these jacks is overhead between the pilot and co-pilot seats.

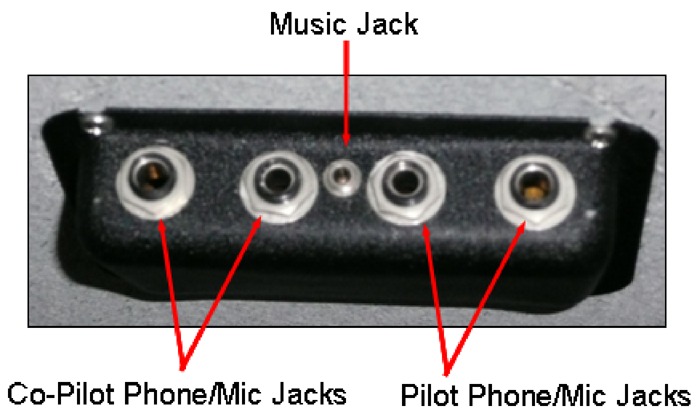

Figure 23-19 Audio Jack Panel

Section 50-01 GMA 340 AUDIO/ICS System

The Liberty XL-2 is equipped with a Garmin model GMA 340 audio panel as standard equipment. This unit provides central audio management for all possible avionics options. LED illuminated push buttons allow audio selection for both NAV and COM audio. A large single button activation of the COM microphone and audio for up to three COM transceivers is provided. A fail-safe feature is incorporated that will connect the pilots headset directly to COM1 in the event power is interrupted or the unit is turned off. In the event of a power loss a fail-safe warning audio tone will be heard.

Figure 23-20 GMA 340 Face Plate Key Slot

The audio panel includes a six position intercom system (ICS) with electronic cabin noise de-emphasis, two stereo music inputs and independent pilot and co-pilot volume controls. One stereo music input is available at the audio jack panel located in the cabin ceiling.

The XL-2 has audio panel installation is configured to make use of the pilot and co-pilot audio inputs only. Full intercom capability between these two stations is available.

Additional audio GMA340 audio panel control functions and feature may be found in the Garmin user guide and installation manual for this unit. See troubleshooting guide section below.

Section 50-02 Audio System Procedures

This section has the procedures to remove and install the audio equipment.

NOTE

Only an appropriately equipped and certificated avionics maintenance facility can perform maintenance operations beyond simple removal or installation of a communications transceiver.

GMA 340 AUDIO PANEL REMOVAL

This procedure performs audio pane until removal.

1. Check that both aircraft battery switch and avionics master switch are OFF.
2. Insert hex key ("3/32" Allen wrench") into the hole in the faceplate of the audio/intercom panel. Rotate counterclockwise until the retaining claw is released.
3. Slide the unit out of the mounting tray.

This completes the GMA 340 Audio Panel Removal procedure.

GMA 340 AUDIO PANEL INSTALLATION

This procedure performs audio panel unit installation.

1. Check that both the aircraft battery switch and avionics master switch are OFF.
2. Check the mating connectors on the rear of the unit and in the mounting tray to check that no pins are bent or pushed back into the connector housing.
3. Verify that the retaining claw is in the released position. If necessary, insert a hex key into the hole in the faceplate and turn CCW.
4. Slide unit into the mounting tray until the connectors engage.
5. Insert a hex key into the hole in the faceplate and turn CW until the unit is firmly seated in the mounting tray.
6. Perform operation check of the audio system described in GMA340 Operation Check and Inspection on page 41 of this chapter.

This completes the GMA 340 Audio Panel Installation procedure.

GMA340 OPERATION CHECK AND INSPECTION

Perform an operational check of the audio panel per the Garmin Ltd. OEM installation manual Post Installation Checkout procedure. Refer to Table 23-4 for applicable manual and section. Use the latest manual version.

Model	Document	Title	Section	Notes
GMA 340	190-00149-01	Installation Manual	2.6	Contact Garmin Ltd. Dealer

Table 23-4 Manual to Use to Perform the Operation Check and Inspection

This completes the GMA340 Operation Check and Inspection procedure.

Section 50-03 Troubleshooting Guides

Refer to Table 23-5 and Table 23-6 to resolve issues with the Garmin audio/intercom system. The manuals in Table 23-5 are reference to the latest versions of the Garmin Ltd. Manuals.

MODEL	DOCUMENT	TITLE	NOTES
GMA 340	190-00149-10	USER GUIDE	Available at WWW.GARMIN.COM
	190-00149-01	INSTALLATION MANUAL	Contact Garmin Dealer

Table 23-5 Garmin Troubleshooting Guide Reference

Complaint	Possible Cause	Remedy
Unable to transmit or receive; communications transceiver display dark	Defective power supply wiring	• Repair
	Defective COM circuit breaker	• Replace
	If other avionics are also without power: defective avionics master switch or relay	• Replace
	Defective radio unit	• Repair/replace
Unable to transmit from pilot side	Defective PTT switch on pilot side	• Replace
	Defective wiring between pilot PTT switch and audio/intercom panel	• Repair
Unable to transmit from copilot side	Defective PTT switch on copilot side	• Replace
	Defective wiring between copilot PTT switch and audio/intercom panel	• Repair
No modulation when transmitting from pilot side	Defective headset or microphone.	• Replace
	Defective wiring	• Repair
No modulation when transmitting from copilot side	Defective headset or microphone.	• Replace
	Defective wiring	• Repair
Transmission reported weak by other stations, receive mode OK	Defective COM radio	• Repair/Replace
	Headset microphone fault	• Replace
	Audio jack fault	• Repair/replace
Receive mode weak / noisy	Defective COM radio	• Repair/replace
	Headphone fault	• Replace
	Audio jack fault	• Repair/replace
Transmit and receive both weak or noisy	Defective COM antenna, coaxial cable, or connectors	• Repair/replace
	Headset fault	• Replace
	Audio panel fault	• Repair/replace
Intercom system inoperative, transmit mode functional	Defective audio/ICS panel	• Repair/replace

Table 23-6 Troubleshooting Guide for the XL2 Airplane COM/GPS Systems

CHAPTER 24

ELECTRICAL POWER

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Table of Contents

SECTION 24-00 GENERAL 5

SECTION 24-10 ALTERNATOR BELT DRIVE 9

SECTION 10-01 PERIODIC MAINTENANCE 10
SECTION 10-02 ALTERNATOR BELT PROCEDURES 10
ALTERNATOR BELT REMOVAL 11
ALTERNATOR BELT INSTALLATION 12
ALTERNATOR BELT TENSIONING 13
ALTERNATOR BELT OPERATION CHECK AND INSPECTION 14
SECTION 10-03 ALTERNATOR BELT TROUBLESHOOTING GUIDE 15

SECTION 24-30 DC GENERATION 17

SECTION 30-01 ALTERNATOR 17
SECTION 30-02 PERIODIC MAINTENANCE 17
SECTION 30-03 ALTERNATOR PROCEDURES 17
ALTERNATOR REMOVAL 18
ALTERNATOR INSTALLATION 19
ALTERNATOR FUSE F3 REPLACEMENT 20
ALTERNATOR OPERATION CHECK AND INSPECTION 22
SECTION 30-04 ALTERNATOR TROUBLESHOOTING GUIDE 23
SECTION 30-05 ALTERNATOR CONTROL UNIT (ACU) 24
SECTION 30-06 PERIODIC MAINTENANCE 24
SECTION 30-07 ALTERNATOR CONTROL UNIT (ACU) PROCEDURES 24
ALTERNATOR CONTROL UNIT (ACU) REMOVAL 25
ALTERNATOR CONTROL UNIT (ACU) INSTALLATION 26
ALTERNATOR CONTROL UNIT (ACU) OPERATION CHECK AND INSPECTION 27
SECTION 30-08 ACU TROUBLESHOOTING GUIDE 29
SECTION 30-09 BATTERIES 30
SECTION 30-10 PRIMARY BATTERY 30
SECTION 30-11 PERIODIC MAINTENANCE 31
SECTION 30-12 PRIMARY BATTERY PROCEDURES 31
PRIMARY BATTERY REMOVAL 32
PRIMARY BATTERY INSTALLATION 33
PRIMARY BATTERY OPERATION CHECK AND INSPECTION 35
SECTION 30-13 PRIMARY BATTERY TROUBLESHOOTING GUIDE 37
SECTION 30-14 SECONDARY BATTERY 38
SECTION 30-15 PERIODIC MAINTENANCE 39
SECTION 30-16 SECONDARY BATTERY PROCEDURES 39
SECONDARY BATTERY REMOVAL 40
SECONDARY BATTERY INSTALLATION 41
SECONDARY BATTERY OPERATIONAL CHECK AND INSPECTION 42
SECTION 30-17 SECONDARY BATTERY TROUBLESHOOTING GUIDE 43

SECTION 24-60 DC ELECTRICAL LOAD DISTRIBUTION 45

SECTION 60-01 DC POWER DISTRIBUTION 45
SECTION 60-02 PERIODIC MAINTENANCE 45
SECTION 60-03 CIRCUIT BREAKER AND MASTER RELAY PROCEDURES 45
CIRCUIT BREAKER REMOVAL 46
CIRCUIT BREAKER INSTALLATION 48
MASTER RELAY REMOVAL 49
MASTER RELAY INSTALLATION 51
CIRCUIT BREAKER AND MASTER RELAY OPERATION CHECK AND INSPECTION 53
SECTION 60-04 MASTER RELAY TROUBLESHOOTING GUIDE 55
SECTION 60-05 AVIONICS POWER DISTRIBUTION 57
SECTION 60-06 CIRCUIT BREAKER REMOVAL AND INSTALLATION 58
SECTION 60-07 PERIODIC MAINTENANCE 58
SECTION 60-08 AVIONICS POWER DISTRIBUTION PROCEDURES 58
AVIONICS MASTER RELAY REMOVAL 59
AVIONICS MASTER RELAY INSTALLATION 61
AVIONICS POWER DISTRIBUTION OPERATION CHECK AND INSPECTION 62
SECTION 60-09 AVIONICS POWER SUBSYSTEM TROUBLESHOOTING GUIDE 64

Section 24-00 General

The airplane is equipped with a 14-volt (nominal) single-wire ground-return electrical system. A belt-driven alternator mounted at the front of the engine generates primary power. Two 12-volt maintenance-free recombinant-gas (RG) type lead-acid batteries are mounted in the aft fuselage as in Figure 24-1.

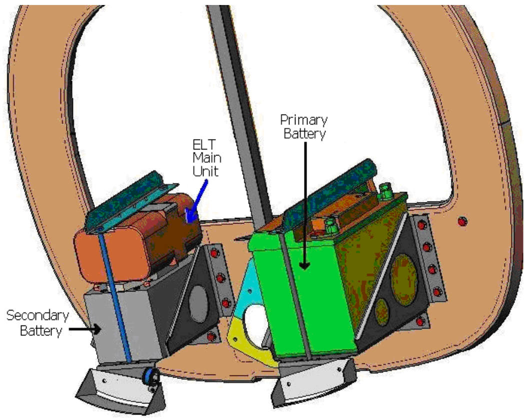

Figure 24-1 Battery Location

In the event of alternator failure, the primary battery will power all aircraft loads, including the FADEC, for up to 30 minutes depending on load. In the event of total electrical failure and/or discharge of the primary battery, the secondary battery or FADEC secondary battery will power the FADEC maintaining engine, Attitude Indicator (AI) gyro and Turn Coordinator (TC) gyro operation for up to 60 minutes.

In the event of primary power failure (alternator failure and primary battery failure), the secondary battery powers only the engine FADEC system and the Attitude Indicator gyro and Turn Coordinator gyro. All other electrical systems on the airplane (such as flap and trim systems, communication and radio navigation system, and electrically driven flight instrument) will be off line. Pneumatic systems (such as altitude indicator and airspeed indicator) will continue to operate in the event of electrical system failures.

Electrical system voltage and alternator output amperage are displayed on the integrated engine instrument system model VM1000FX (see Chapter 77 – *Engine Indicating* for details).

Circuit breakers and fuses protect electrical circuits in the airplane. A two-sided Master Switch allows the alternator to be switched on or off independently from the battery. See Figure 24-2 for the location of the master switches for the battery and the alternator.

Figure 24-2 Instrument panel showing the location of the ALT FAIL (Alternator Fail) indicator and the ALT/BAT switch

However, the battery must be in the ON position to allow the alternator switch to be moved to the ON position.

An electronic Alternator Control Unit (ACU) provides both voltage regulation and over-voltage protection functions. An Alternator Failed (ALT FAIL) warning light on the instrument panel driven by ACU control logic illuminates to warn the pilot when the ACU has detected an alternator system malfunction.

Illumination of the ALT FAIL warning light is normal prior to engine start. Illumination of the ALT FAIL light after engine start may be due to one or more of the following causes:

- Alternator switch not on
- Broken alternator drive belt
- Alternator failure
- Alternator over-voltage trip
- ACU failure
- Wiring fault

Primary and secondary power system cable runs are routed separately through the aircraft structure. This is a safety feature of the aircraft preventing harness failure of one power system affecting the other. Separation of primary and secondary power wire harnesses is maintained from each battery to the bus bar array located in the instrument panel. Maintenance or repair of the harness system must maintain this separation as shown in Figure 24-3 and Figure 24-4 below.

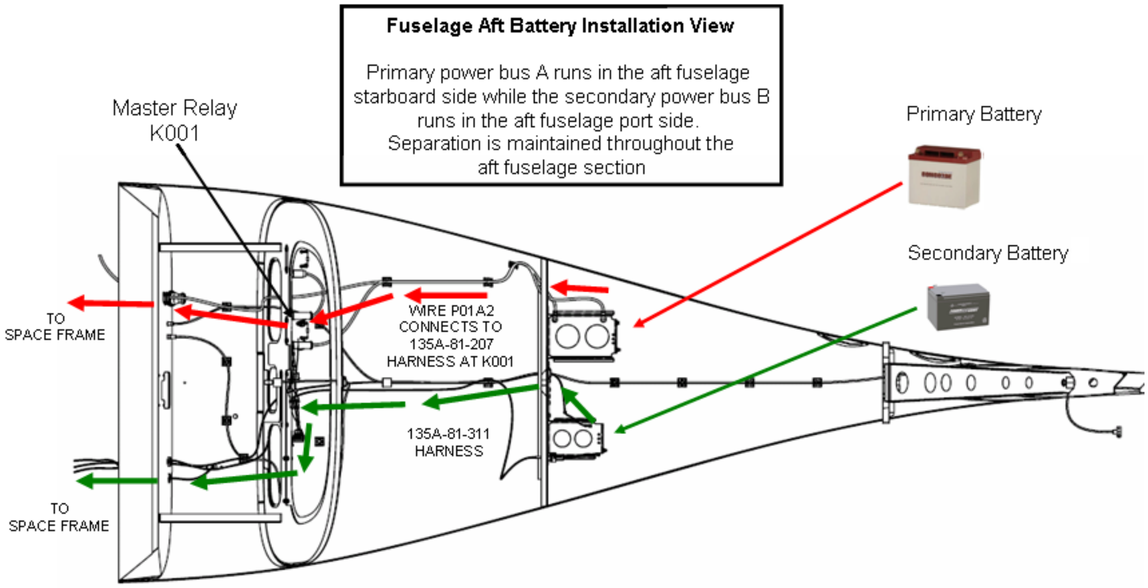

Figure 24-3 Fuselage Aft Battery Installation View

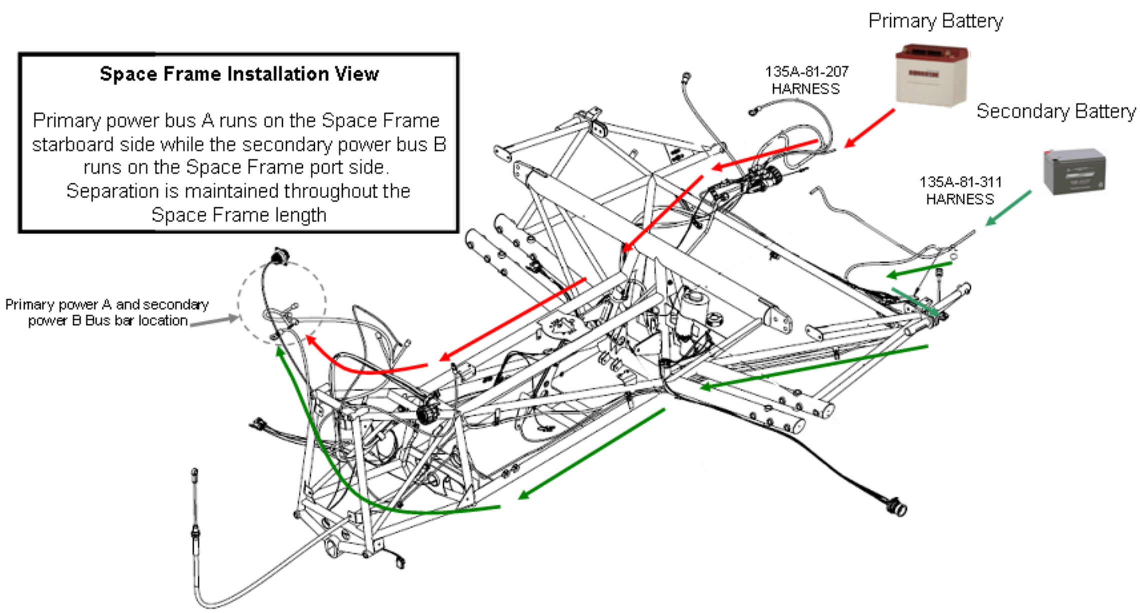

Figure 24-4 Space Frame Installation View

Maintenance operations performed by this chapter require adherence to procedures set forth in the chapter and with applicable sections of 14 CFR Part 43 Maintenance, Preventive Maintenance, Rebuilding, and Alterations to remain in compliance.

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Section 24-10 Alternator Belt Drive

The alternator is driven by a V-belt using pulleys on the alternator and on the engine crankshaft. Belt tension should be checked as part of the regular preflight inspection by application of a force to the midpoint of the belt, halfway between the pulleys. Belt deflection of approximately 1/2 in. is normal.

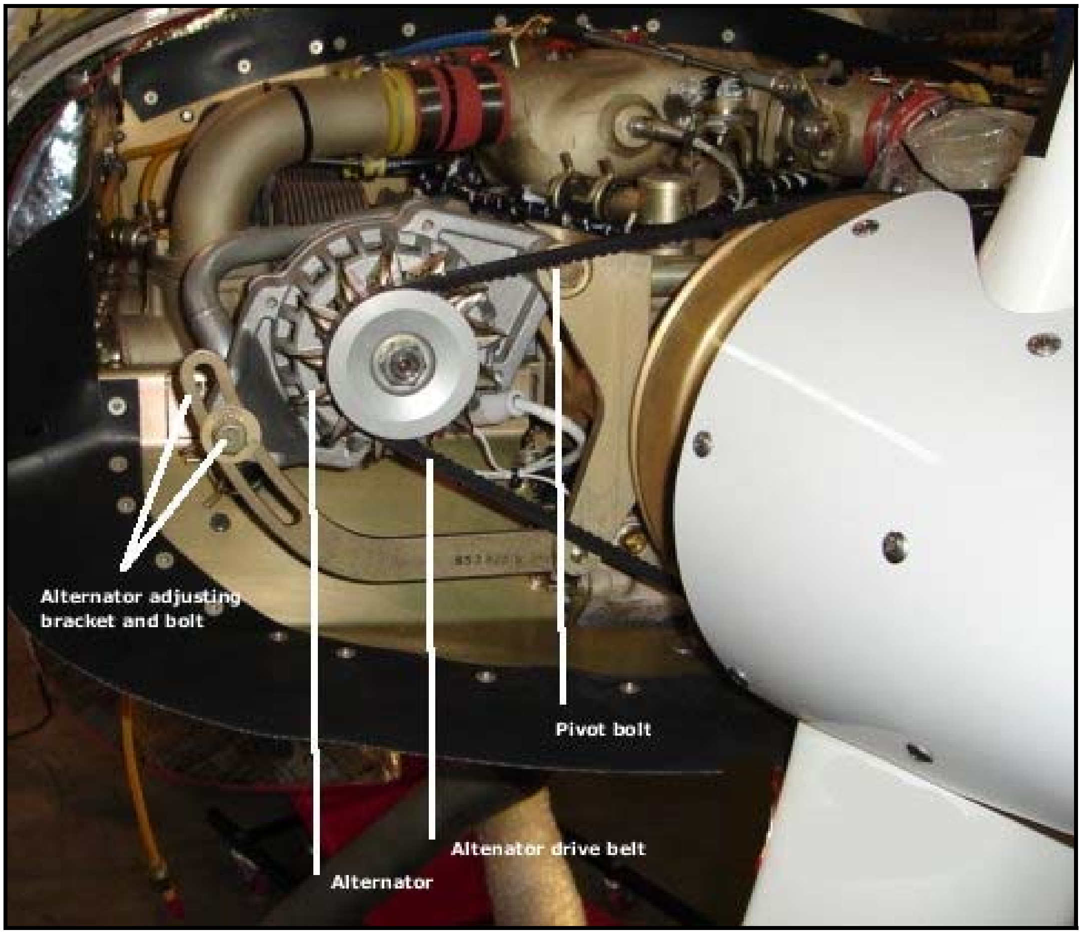

Figure 24-5 View of Front of Aircraft Engine

WARNING

ALTERNATOR BELT REPLACEMENT REQUIRES REMOVAL AND REPLACEMENT OF THE PROPELLER. IMPROPER TORQUE OF PROPELLER BOLTS MAY CAUSE BOLT OR PROPELLER FAILURE AND RESULT IN INJURY OR DEATH.

NOTE

See Liberty Maintenance Manual Chapter 61 for propeller removal / replacement procedures and correct bold torque values.

Section 10-01 Periodic Maintenance

Alternator belt inspections take place at 100 hour and annual intervals. Perform the procedure Alternator Belt Operation Check and Inspection on page 14 of this chapter.

Section 10-02 Alternator Belt Procedures

This section details the procedures to remove, install, and inspect and check the alternator belt.

ALTERNATOR BELT REMOVAL

Perform the following procedure to remove the alternator belt.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Remove upper and lower engine cowling.
3. Remove spinner and propeller in accordance with Liberty maintenance manual Chapter 61 procedures.
4. Loosen nuts on alternator mounting bracket and alternator belt tension adjustment bolts. See Figure 24-5 for the location of the alternator tensioning bolts.
5. Move (rotate) alternator toward engine until belt can be lifted off alternator pulley.
6. Lift belt from engine crankshaft pulley and remove belt.

This completes the Alternator Belt Removal procedure.

ALTERNATOR BELT INSTALLATION

Perform the following procedure to install the alternator belt.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Move (rotate) alternator as far as possible toward engine case.
3. Place replacement belt around engine crankshaft and alternator pulleys.
4. Secure alternator attachment and belt tension adjustment nuts; but do not tighten fully.
5. Replace propeller and spinner using Maintenance Manual Chapter 61 procedures
6. Tension the alternator belt in accordance with Alternator Belt Tensioning procedure on page 13 of this chapter.
7. Verify alternator and engine pulley alignment.
8. Replace upper and lower cowlings.
9. Perform the procedure Alternator Belt Operation Check and Inspection on page 14 of this chapter.

This completes the Alternator Belt Installation procedure.

ALTERNATOR BELT TENSIONING

Perform this procedure to tension the alternator belt to the proper tension.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Remove upper cowling.
3. Loosen pivot sufficiently to permit the alternator to move as the adjustment arm is moved.
4. Loosen the adjustment arm sufficiently to permit alternator rotation along adjustment arm slot.
5. Tension belt to 40 to 45 lbs (½-inch mid-span deflection) and tighten adjustment arm bolt with a torque of 240 to 260 in-lbs. Safety wire bolt head.
6. Tighten pivot bolt with a torque of 400 to 450 in-lbs.
7. Replace upper cowling
8. Run engine over a range of 850 to 2000 RPM for a period of 20 minutes.
9. Return engine to idle (850 to 950 RPM) and shut down.
10. Verify belt tension is 40 to 45 pounds (½-inch mid-span deflection). Adjust as required by repeating steps 1 through step 10.
11. Perform the procedure Alternator Belt Operation Check and Inspection on page 14 of this chapter.

This completes the Alternator Belt Tensioning procedure.

ALTERNATOR BELT OPERATION CHECK AND INSPECTION

Perform this procedure to do an operational check and inspection of the alternator belt.

NOTE

Refer to the Alternator Belt Troubleshooting Guide for corrective actions as required.

1. Inspect alternator pulley, belt and engine pulley for wear or damage. Do not proceed if damage is present.
2. Verify alternator belt tension is 40 to 45 lbs. (½-inch mid-span deflection).
3. Position the aircraft in the run up area.
4. Perform a standard start and warm up procedure.
5. Set engine to 1700 RPM.
6. Cycle engine throttle from 1000 to 2000 RPM and verify:
   a. Main bus voltage remains within the 14.1 to 14.3 Vdc range of operation
   b. Main bus amp meter reads above zero
7. ALT FAIL lamp remains OFF.
8. With the engine set for 1700 RPM, position the split Master Switch ALT side to OFF.
9. Verify the ALT FAIL lamp is ON.
10. Position the split Master Switch ALT side to ON.
11. Verify ALT FAIL lamp is OFF.
12. Reduce engine RPM to idle (850 to 950 RPM) and perform an engine shutdown.
13. Verify there is no indication of slippage or damage to the belt, alternator pulley, or engine pulley.

This completes Alternator Belt Operation Check and Inspection.

Section 10-03 Alternator Belt Troubleshooting Guide

This section contains a guide to troubleshooting issues with the alternator belt.

Complaint	Possible Cause	Remedy
Alternator belt slips	Tensioning arm bolt loose	Set belt tension, tighten and safety tension arm bolt and safety wire.
	Belt tension fault	Set belt tension
	Excess belt wear	Replace belt
Belt damage	Excess operation hours	Replace belt
	Alternator pulley damage	Replace alternator and belt
	Engine pulley damage	Replace engine pulley and belt

Table 24-1 Alternator Belt Troubleshooting Guide

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Section 24-30 DC Generation

This section details the generation of DC voltage from the airplane's engine.

Section 30-01 Alternator

The alternator (nominal capacity 60 amps maximum at 14.2 Vdc maximum) is mounted on the right side of the front of the engine. Maintenance operations are normally limited to removal and installation of the alternator for condition. For further alternator service information, refer to applicable portions of Teledyne Continental Motors IOF-240-B Overhaul Manual, Teledyne Continental Motors no. OH-22.

Section 30-02 Periodic Maintenance

Alternator inspections take place at 100 hour and annual intervals. Perform the procedure Alternator Operation Check and Inspection on page 22 of this chapter.

Section 30-03 Alternator Procedures

This section details the procedures to remove, install, and inspect and check the alternator.

ALTERNATOR REMOVAL

Perform this procedure to remove the alternator from the airplane.

1. Pull the BAT 1 (CB001) circuit breaker to OPEN.
2. Pull the ALT Field (CB002) circuit breaker to OPEN.
3. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
4. Remove upper and lower engine cowling, in accordance with Chapter 71 – *Power Plant*.

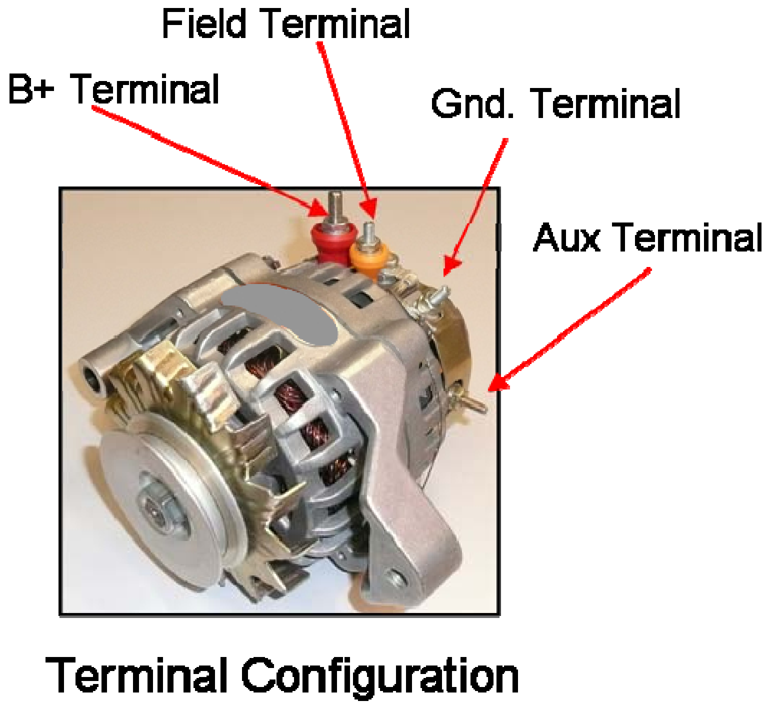

Figure 24-6 View of Alternator Connections

5. Disconnect wire ID: P10B6 and EMI Filter Lead from the “B+” alternator terminal.
6. Disconnect wire ID: P15D16 from “Field” alternator terminal.
7. Disconnect wire ID: P26E16 from “AUX” alternator terminal.
8. Disconnect wire ID: P28A14N from “Ground” alternator terminal
9. Relieve alternator belt tension by removing alternator tension bar bolt.
10. Remove alternator pivot bolt and remove alternator from mounting bracket.

This completes the Alternator Removal procedure.

ALTERNATOR INSTALLATION

Perform this procedure to install the alternator on to the airplane.

1. Pull the BAT 1 (CB001) circuit breaker to OPEN.
2. Pull the ALT Field (CB002) circuit breaker to OPEN.
3. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
4. Install alternator using nut and bolt hardware removed previously. The hardware is to remain finger tight only at this time.
5. Perform the Alternator Belt Tensioning on page 11.
6. Connect wire ID: P10B6 and EMI Filter Lead to “B+” terminal on alternator.
7. Connect wire ID: P15D16 to “Field” terminal on alternator,
8. Connect wire ID: P26E16 to “AUX” terminal on alternator
9. Connect wire ID: P28A14N to Ground terminal on alternator
10. Replace upper and lower cowlings.
11. Push in the ALT Field (CB002) circuit breaker to CLOSE.
12. Push in the BAT1 (CB001) circuit breaker to CLOSE.
13. Perform Alternator Operation Check and Inspection on page 22 of this chapter.

This completes the Alternator Installation procedure.

ALTERNATOR FUSE F3 REPLACEMENT

Perform this procedure to remove the alternator from the airplane.

1. Pull the BAT 1 (CB001) circuit breaker to OPEN.
2. Pull the ALT Field (CB002) circuit breaker to OPEN.
3. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
4. Remove upper and lower engine cowling, in accordance with Chapter 71 – *Power Plant*.

NOTE

Inspect the fuse holder in accordance with SSI-10-001 Alternator Fuse Holder F3 Installation. If the fuse holder installation was correct, continue with this procedure. If the fuse holder installation was incorrect, perform the procedures of SSI-10-001.

PUSH IN ON THE SMALL FREE END OF THE FUSE HOLDER AND TWIST COUNTER-CLOCKWISE TO SEPARATE THE F3 FUSE HOLDER.

Figure 24-7 Fuse Holder F3 Disassembly

5. Holding on to the free end of the fuse, push in on the fuse holder and twist in a counter-clockwise direction.
6. Pull the fuse holder apart.

NOTE

There is a small rubber "O" ring gasket inside the fuse holder. This gasket should remain in side the large portion of the fuse holder, this is normal. If gasket come out or comes out on the smaller portion of the fuse, insert the gasket back in to the larger portion of the fuse holder.

7. Replace the fuse inside. The fuse is a 5.0 AMP ceramic shell fuse (ABC-5).
8. Insert the small section of the fuse into the larger section and twist in a clockwise direction until the two lock together.
9. Install the lower and upper cowlings, in accordance with Chapter 71 – *Power Plant*.
10. Close the alternator field circuit breaker ALT Field (CB002).
11. Close the battery circuit breaker BAT 1 (CB001).

This completes the Alternator Fuse F3 Replacement procedure.

ALTERNATOR OPERATION CHECK AND INSPECTION

Perform this procedure to do an operational check and inspection of the alternator.

NOTE

Refer to the Alternator Troubleshooting Guide for corrective actions as required.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF
2. Remove upper cowling.
3. Inspect and verify alternator terminal connections are secure and free from damage or corrosion.
4. Inspect alternator belt and pulley systems for damage in accordance with Alternator Belt Operation Check and Inspection on page 14 of this chapter.
5. Install upper cowl.
6. Position the airplane in the run up area.
7. Perform a standard start and warm up procedure.
8. Set engine to 1700 RPM.
9. Cycle engine throttle from 1000 to 2000 RPM and verify:

- Main bus voltage remains within the 14.0 to 14.2 Vdc range of operation
- Main bus amp meter reads above zero
- ALT FAIL lamp remains OFF

10. With the engine set for 1700 RPM, place the split Master Switch ALT side to OFF.
11. Verify the ALT FAIL lamp is ON.
12. Place the split Master Switch ALT side ON.
13. Verify ALT FAIL lamp is OFF.
14. Return the engine to idle (850 to 950 RPM) and shut down.

This completes the Alternator Operation Check and Inspection.

Section 30-04 Alternator troubleshooting guide

This section contains a guide to troubleshooting issues with the alternator.

Complaint	Possible Cause	Remedy
No power supplied by alternator with engine running; ALT FAIL warning light illuminated, voltmeter <13.7v	Alternator failed	Replace
	ACU failed	Replace
	Cabling, connectors, electrical connections	Repair
	Failed BAT / ALT master switch	Replace
	Open alternator fuse F3	Replace fuse F3, investigate reason for fuse F3 failure.
No power supplied by alternator with engine running, ALT FAIL warning light illuminated, voltmeter indicates >15 v (over-voltage trip)	Momentary over-voltage	Cycle ALT portion of BAT/ ALT master switch off, then on adjust ACU voltage down
	Defective ACU	Replace
No power supplied by alternator with engine running, ALT FAIL warning light NOT illuminated, voltmeter indicates <13.7 v.	Defective ACU	Replace
	Defective wiring or connectors	Repair
	Defective ALT FAIL Annunciator	Replace
	Defective BAT / ALT master switch	Replace
	Defective ALT FIELD circuit breaker	Replace
RPM related noise (whine) in headset	Defective ground connections in alternator or ACU wiring	Repair
	Defective ground connections in shielding of radio / audio wiring	Repair
	Defective diode(s) in alternator	Replace

Table 24-2 Alternator Troubleshooting Guide

Section 30-05 Alternator Control Unit (ACU)

The Alternator Control Unit (ACU) is located in the aft fuselage adjacent to the strobe light power supply. The unit does not require periodic maintenance. Adjustments are limited to a single regulation level control set on installation.

NOTE

In this section the Alternator Control Unit (ACU) is the same as Voltage Regulator, and maybe used interchangeably.

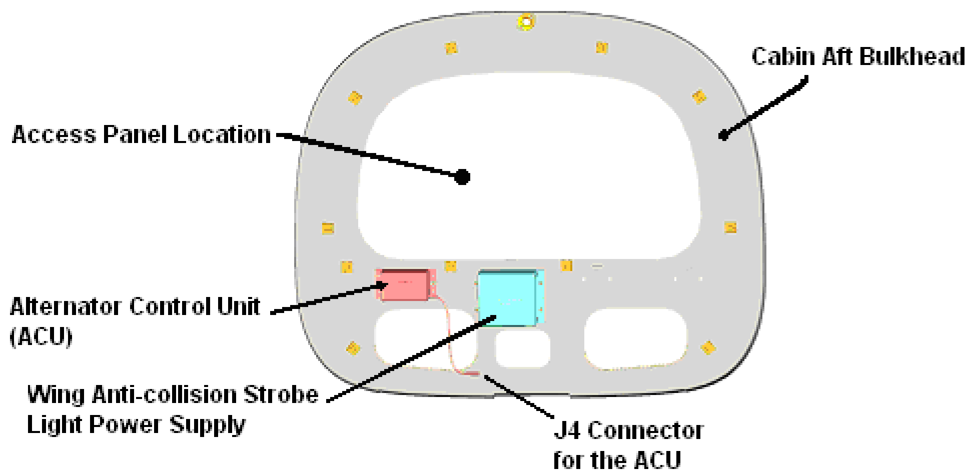

Figure 24-8 Alternator Control Unit (ACU) or Voltage Regulator Location

Section 30-06 Periodic Maintenance

ACU inspections take place at 100 hour and annual intervals. Perform the procedure Alternator Control Unit (ACU) Operation Check And Inspection on page 27 of this chapter.

Section 30-07 Alternator Control Unit (ACU) Procedures

This section contains the procedures to remove, install, and inspect and check of the ACU.

ALTERNATOR CONTROL UNIT (ACU) REMOVAL

Perform this procedure to remove the alternator control unit.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Remove the cabin aft bulkhead access panel, by removing securing screw hardware.
4. Disconnect connector J4 from ACU.
5. Remove ACU bracket 10-32 mounting bolts and lift the ACU out of the fuselage.

This completes the Alternator Control Unit (ACU) Removal procedure.

ALTERNATOR CONTROL UNIT (ACU) INSTALLATION

Perform this procedure to install the alternator control unit.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Remove the cabin aft bulkhead access panel, by removing securing screw hardware.
4. Position the ACU on the aft fuselage bulkhead adjacent to the strobe light power supply and secure with 10-32 bolts hardware removed previously.
5. Connect J4 turn locking ring until detent is felt to be engaged.
6. Perform the Alternator Control Unit (ACU) Operation Check And Inspection on page 27.
7. Install the cabin aft bulkhead access panel.
8. Remove the tail support.

This completes the Alternator Control Unit (ACU) Installation procedure.

ALTERNATOR CONTROL UNIT (ACU) OPERATION CHECK AND INSPECTION

Perform this procedure to do an operational check and inspection of the alternator control unit.

NOTE

Refer to the ACU Troubleshooting Guide for corrective actions as required.

1. Install a tail stand
2. Remove the cabin aft bulkhead access panel by removal of retaining screws.
3. Inspect the ACU installation for the following:

- Secure mounting
- Proper electrical connection
- Corrosion or physical damage
- Correct deficiencies found by repair or replacement of the ACU.

4. Remove the tail stand
5. Position aircraft in the run up area
6. Set the aircraft parking brake
7. Perform a standard start and warm up procedure
8. Set engine to 1700 RPM
9. Verify flight instrument panel mounted “Alt Fail” annunciator is NOT on. If the annunciator is on refer to troubleshooting guide for next action.
10. Verify the ACU mounted status annunciator is NOT on. If the annunciator is on refer to troubleshooting guide for next action.
11. Note the VM1000FX voltage reading. If the reading is 14.1 ± 0.1 Vdc proceed with step 19.
12. Return the engine to idle (850 to 950 RPM) and shut down.
13. Install a tail stand
14. Access the ACU adjustment by removing the access hole protective plug, as shown in Figure 24-9.

Figure 24-9 ACU Voltage Adjustment

15. Using a jeweler's screwdriver, make small adjustments to the ACU voltage control. Turn the control clockwise (CW) to increase voltage and counterclockwise (CCW) to decrease voltage.
16. Remove tail stand
17. Start the engine and set the throttle to 1700 RPM
18. While monitoring main bus voltage on the flight instrument panel mounted VM1000FX engine instrument display verify main bus voltage reads between 14.1 ± 0.1 Vdc. If the voltage is out of range, repeat steps 12 to step 18.
19. Cycle engine throttle from 1000 to 2000 RPM and verify main bus voltage remains within the 14.1 ± 0.1 Vdc range of operation.
20. Return the engine to idle (850 to 950 RPM) and shut down.
21. Install tail stand.
22. Install ACU voltage adjustment access hole protective plug.
23. Install aft cabin aft bulkhead access panel.
24. Remove tail stand.

This completes the Alternator Control Unit (ACU) Operation Check And Inspection.

Section 30-08 ACU Troubleshooting Guide

This section contains a guide to troubleshooting the Alternator Control Unit.

Complaint	Possible Cause	Remedy
No power supplied by alternator with engine running; ALT FAIL warning light illuminated, voltmeter <13.7VDC	Alternator failed	Replace
	ACU failed	Replace
	Cabling, connectors, electrical connections	Repair
	Failed BAT / ALT master switch	Replace
ACU does not regulate to specification	ACU not set correctly	Perform ACU voltage adjustment
	ACU fault	Replace ACU
	Alternator fault	Replace alternator and perform ACU voltage adjustment
	Faulty regulator sense wire	Inspect and repair wiring
No power supplied by alternator with engine running, ALT FAIL warning light illuminated, voltmeter indicates >15 VDC (over-voltage trip)	Momentary over-voltage	Cycle ALT portion of BAT/ ALT master switch off, then on adjust ACU voltage down
	Defective ACU	Replace
No power supplied by alternator with engine running, ALT FAIL warning light NOT illuminated, voltmeter indicates <13.7 VDC	Defective ACU	Replace
	Defective wiring or connectors	Repair
	Defective ALT FAIL Annunciator	Replace
	Defective BAT / ALT master switch	Replace
	Defective ALT FIELD circuit breaker	Repair
RPM related noise (whine) in headset	Defective ground connections in alternator or ACU wiring	Repair
Case Integrity Fault	ACU has been dropped or has been exposed to unknown forces or environmental conditions	Replace

Table 24-3 Alternator Control Unit Troubleshooting Guide

Section 30-09 Batteries

This section details information about the battery systems for the airplane.

Section 30-10 Primary Battery

The Liberty XL-2 may be fitted with the Concorde model RG-25 or the Concord model RG-25XC as the primary aircraft battery. The Concorde RG-25, has a C1 rated capacity of 22 amp/hours while the Concorde RG-25XC offers a higher rated C1 of 24 amp/hours cranking power. Both batteries are a Recombinant Gas RG® battery technology. The RG series are low resistance valve regulated lead acid (VRLA) batteries. Either model installed provides the function of engine starting, generating system stabilization, and emergency backup power for the airplane electrical system as a whole. Both batteries are of identical size using the same mounting hardware and maintenance procedures.

Figure 24-10 Primary Battery Models RG-25 (left) and RG-25XC (right)

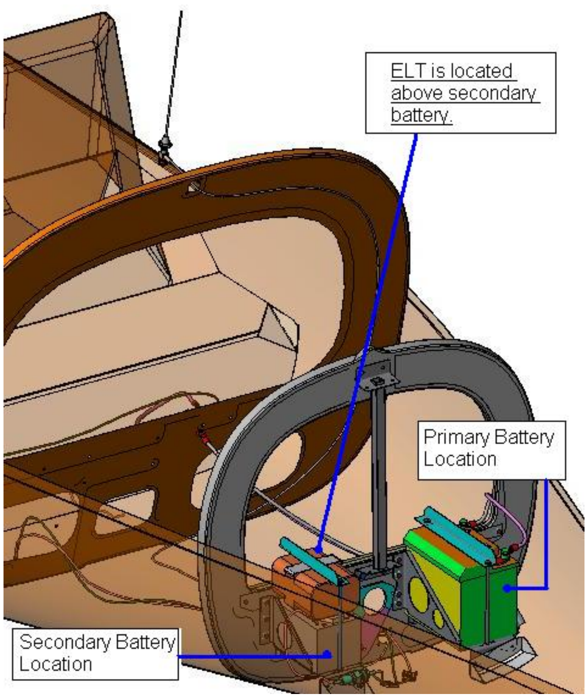

Figure 24-11 Location of Batteries Aft of Baggage Closeout Area

Section 30-11 Periodic Maintenance

Primary battery inspections take place at 100 hour and annual intervals. Perform the procedure Primary Battery Operation Check and Inspection on page 35 of this chapter.

CAUTION

During removal and replacement operations care must be taken to assure separation of primary and secondary battery leads is maintained. At no time are primary and secondary battery leads to be bundled or routed together.

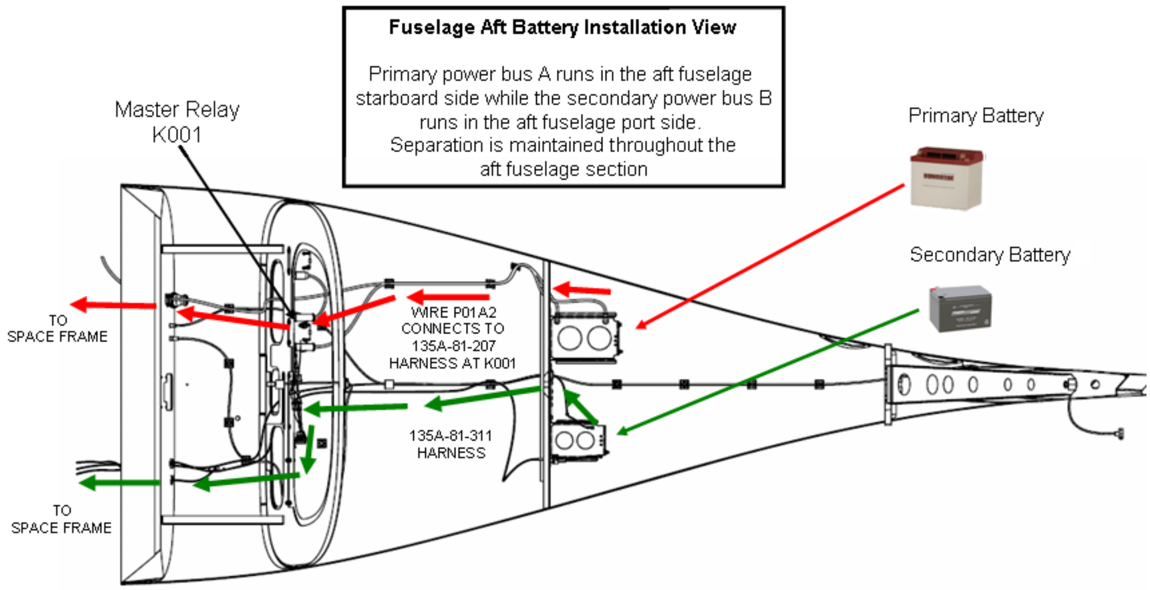

Figure 24-12 Battery Locations

Section 30-12 Primary Battery Procedures

This section contains the procedures to remove, install, check, and inspect the primary battery.

PRIMARY BATTERY REMOVAL

Perform this procedure to remove the primary battery from the airplane.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Pull the BAT1 (CB001) circuit breaker to OPEN.
3. Install a tail stand underneath the tail section of the airplane.
4. Remove the cabin aft bulkhead access panel, by removing securing screw hardware.
5. Disconnect the negative then the positive leads from the primary battery. Isolate the negative terminals on the batteries to prevent accidental connection.
6. Remove primary battery hold-down bolts.

CAUTION

The primary battery is very heavy. Obtain assistance while removing the primary battery from the airplane.

7. Remove primary battery
8. If the primary battery is to be installed later, perform Primary Battery Operation Check and Inspection on page 35 of this chapter.

This completes the Primary Battery Removal procedure.

PRIMARY BATTERY INSTALLATION

Perform this procedure to install the primary battery into the airplane.

1. Verify that replacement battery is correct type.
2. Perform the procedure Primary Battery Operation Check and Inspection on page 35 of this chapter.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

3. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
4. Pull the BAT1 (CB001) circuit breaker to OPEN.
5. Install a tail stand underneath the tail section of the airplane.
6. Remove the cabin aft bulkhead access panel, by removing securing screw hardware.
7. Clean battery terminals with a stiff brush and contact cleaner.
8. Clean battery cable terminals with stiff brush and contact cleaner

CAUTION

The primary battery is very heavy. Obtain assistance while installing the primary battery to the airplane.

9. Position battery in battery tray with terminals oriented outboard
10. Install and secure battery hold down bar with two AN4 ¼-28 bolts. Safety wire bolts heads.

CAUTION

Always connect negative battery cable last.

11. Install positive battery lead P01A2 with terminal bolt. Apply a torque of 70 in-lbs ± 5 in-lbs.
12. Install negative battery lead P07A2N with terminal bolt, Apply a torque of 70 in-lbs ± 5in-lbs.

13. Apply a thin coating of Dow Corning 4 (DC4) electrical insulating compound over the terminal/bolt assembly.
14. Push in the BAT1 (CB001) to CLOSE.
15. Position the split master switch BAT side to ON
16. Verify OAT voltage meter function indication is within +/- 1 volt of operational check reading performed in step 2 above.
17. Position the split master switch to OFF
18. Install cabin aft bulkhead access panel.
19. Remove the tail support.

This completes the Primary Battery Installation procedure.

PRIMARY BATTERY OPERATION CHECK AND INSPECTION

Perform this procedure to do an operational check and inspection of the primary battery.

NOTE

Refer to the Primary Battery Troubleshooting Guide for corrective actions as required.

1. Perform primary battery removal as described in Primary Battery Removal shown on page 32 of this chapter.
2. Inspect battery terminals for corrosion, damage or signs of fatigue.
3. Inspect battery cable terminals for corrosion, damage or signs of fatigue.
4. Inspect battery case for cracks or other physical damage.
5. Inspect battery tray and hold down hardware for corrosion, structural damage or signs of fatigue. Replace corroded, damaged or fatigued components.
6. Inspect battery for time in service.
   a. A primary battery exceeding life limits provided in maintenance manual Chapter 04 must be replaced.
   b. A primary battery in service for 24 months or 1200 hours of operation, whichever occurs first, requires an initial capacity test in accordance with Concord Battery procedure 5-0142. The battery is acceptable for use with a capacity of 85% or greater of nominal rated capacity (C1).
   c. A primary battery in service for 12 months or 200 hours of operation, whichever occurs first since an initial capacity test and not in excess of Chapter 04 replacement limitations, requires a capacity test in accordance with Concord Battery procedure 5-0142. The battery is acceptable for use with a capacity of 85% or greater of nominal rated capacity (C1).
7. Using a DC voltmeter measure voltage across the battery terminals and note the state of charge per Table 24-4:

State of Charge	12 Volt Open Circuit Voltage
100%	12.9 Vdc
75%	12.7 Vdc
50%	12.4 Vdc
25%	12.0 Vdc
0%	≤ 11.7 Vdc

Table 24-4 Table showing the Open Circuit Voltage indicating state of charge

8. Charge the battery using a constant potential capable charger at 14.1 volts until the charge current stabilizes for 1 hour
9. Verify the battery did not become very hot, 55°C (130°F) or greater, during the charge process. Replace battery is temperature exceeds limit.
10. Measure and note the voltage across the battery terminals, using a DC voltmeter, and note the state of charge per Table 24-4 above. If the battery has not achieved a 100% threshold of 12.9 Vdc state of charge, perform battery capacity test in accordance with Concord Battery maintenance manual 5-0142. A battery that fails to achieve 80% capacity must be replaced.
11. Perform the procedure Primary Battery Installation shown on page 33 of this chapter.
12. Operation check and inspection complete.

This completes the Primary Battery Operation Check and Inspection procedure.

Section 30-13 Primary Battery Troubleshooting Guide

Use Table 24-5 as a guide in troubleshooting issues with the primary battery.

Complaint	Possible Cause	Remedy
Unable to connect the primary battery to the airplane's buses	Battery discharged	charge battery
	Defective master switch	Replace
	Defective battery contactor	Replace
	Defective diode X1-X2	Replace
	Defective wiring	Repair
	Defective master relay	replace
	Terminal corrosion	Clean battery and wire terminals with stiff brush and contact cleaner
Battery does not hold charge	Battery beyond serviceable life	Replace battery
Battery gets hot during charge, 55^0C (130^0F) or greater	Battery beyond serviceable life	Replace battery
Ammeter indicates zero whether battery is charging or discharging	defective VM1000FX indicator system	replace (see chapter 77)
	Defective ammeter sensor	Replace (see chapter 77)
Battery capacity consistently low ("does not hold a charge")	charging voltage too low	check, replace alternator or ACU
	Battery fault	replace
	Terminal corrosion	• Clean battery and wire terminals with stiff brush and contact cleaner • Apply a thin coating of Dow Corning 4 (DC4) electrical insulating compound over the terminal/bolt assembly.
Starter motor cranks very slowly or not at all	battery discharged	recharge battery
	bad ground between engine and airframe electrical system ground	Replace ground strap
	Diminished battery capacity	Test capacity using Concord OEM procedure 5-0142.
Open Circuit Voltage Low	Battery discharged	Charge battery
	Diminished battery capacity	Test capacity using Concord OEM procedure 5-0142.
Time in service meets or exceeds chapter 04 limits	Battery has reached end of useful life	Replace

Complaint	Possible Cause	Remedy
Terminal corrosion	Loss of corrosion insulating compound	• Clean battery and wire terminals with stiff brush and contact cleaner • Apply a thin coating of DC4, Dow Corning 4, electrical insulating compound over the terminal/bolt assembly
	Leaking battery terminal	Replace battery
Battery case is deformed, bulging, or damaged	Battery fault	Replace battery

Table 24-5 Primary Battery Troubleshooting Guide

Section 30-14 Secondary Battery

The secondary battery (referred to as FADEC backup battery in some documentation), TCM P/N 656070, (Power-Sonic Corporation model PS-12120 12vdc 12 A.H. battery) has a rated capacity of 12 amp-hours. The secondary battery is dedicated solely to emergency backup operation of the engine's Full Authority Digital Engine Control system (FADEC), Attitude Indicator (AI) gyro and Turn Coordinator (TC) gyro operation for up to 60 minutes. This battery is charged any time the alternator is in operation, but has no other connection to the primary electrical system.

Figure 24-13 FADEC Secondary Battery

Section 30-15 Periodic Maintenance

Secondary battery inspections take place at annual intervals. Perform inspection in accordance with Secondary Battery Operational Check and Inspection on page 42 of this chapter.

The secondary battery, providing backup power for the engine FADEC system, must be replaced at specific service intervals regardless of condition. See Liberty Maintenance Manual Chapter 04 limitations.

CAUTION

During removal and replacement operations care must be taken to assure separation of primary and secondary battery leads is maintained. At no time are primary and secondary battery leads to be bundled or routed together.

Section 30-16 Secondary Battery Procedures

This section contains the procedures to remove, install, and check and inspect the secondary battery.

SECONDARY BATTERY REMOVAL

Perform the following procedure to remove the secondary battery from the airplane.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Pull the SPSC (CB024) circuit breaker to OPEN.
3. Install a tail stand underneath the tail section of the airplane.
4. Remove the cabin aft bulkhead access panel, by removing securing screw hardware.

CAUTION

Take care when removing the ELT from its mounting above the secondary battery. Failure to properly remove the ELT can trigger a false emergency signal. Possible fines may be imposed if there is a false activation of an ELT.

5. Remove Emergency Locator Transmitter by Maintenance Manual Chapter 25 procedure.
6. Disconnect the negative then the positive leads from the secondary battery. Isolate the negative terminal on the batteries to prevent accidental connection.
7. Remove battery hold-down bracket bolts and remove bracket.
8. Remove battery.

This completes the Secondary Battery Removal procedure.

SECONDARY BATTERY INSTALLATION

Perform this procedure to install the secondary battery.

1. Verify that replacement battery is correct type and that it is properly charged.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

2. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF. Pull the SPSC (CB024) circuit breaker to OPEN.
3. Install a tail stand underneath the tail section of the airplane. Remove the cabin aft bulkhead access panel, by removing securing screw hardware.
4. Position battery in battery tray with terminals oriented aft.
5. Connect secondary battery cable positive and negative quick disconnects.
6. Apply a thin coating of Dow Corning 4 electrical insulating compound, DC4, over the terminal/bolt assembly.
7. Install battery hold down bracket with two bolts. Safety wire bolt heads.

CAUTION

Take care when installing the ELT to its mounting position. Failure to properly install the ELT can trigger a false emergency signal. Possible fines may be imposed if there is a false activation of an ELT.

8. Install Emergency Locator Transmitter in accordance with Chapter 25 – *Equipment and Furnishings*.
9. Push in the SPSC (CB024) circuit breaker to CLOSE.
10. Position the FADEC PWR B switch ON.
11. Verify the FADEC HSA is illuminated.
12. Verify Attitude Indicator (AI) and Turn Coordinator (TC) gyros are spinning.
13. Position the FADEC PWR B switch OFF.
14. Install baggage compartment closeout panel.
15. Remove the tail support.

This completes the Secondary Battery Installation procedure.

SECONDARY BATTERY OPERATIONAL CHECK AND INSPECTION

Perform the following procedure to do an operational check and inspection of the secondary battery.

NOTE

Refer to the Secondary Battery Troubleshooting Guide for corrective actions as required.

1. Remove the secondary battery from the airplane per the procedure Secondary Battery Removal as shown on page 40 of this chapter.
2. Inspect battery terminals for corrosion, damage or signs of fatigue. Inspect battery cable terminals for corrosion, damage or signs of fatigue. Inspect battery case for cracks or other physical damage.
3. Inspect battery tray and hold down hardware for corrosion, structural damage or signs of fatigue. Inspect battery for time in service. A secondary battery exceeding life limits provided in maintenance manual Chapter 04 must be replaced.
4. Using a DC voltmeter measure voltage across the battery terminals and note the state of charge per the chart below:

State of Charge	12 Volt Open Circuit Voltage
100%	12.9 Vdc
75%	12.7 Vdc
50%	12.4 Vdc
25%	12.0 Vdc
0%	≤ 11.7 Vdc

Table 24-6 Table showing the Open Circuit Voltage indicating state of charge

5. Using a Constant Potential (CP) charger set to 14.1 volts, charge the battery until battery current drawn reaches 0.120 amps. Discontinue charging once this current has been reached.
6. Verify the battery case temperature did not reach, 55°C (130°F) or greater, during the charge process. Replace battery if temperature exceeds limit.
7. Using a DC voltmeter, measure the open circuit voltage across the battery terminals and note the state of charge per Table 24-6 above. If the battery has not achieved a 100% state of charge threshold of 12.9Vdc, replace the battery.
8. Install the secondary battery per the procedure Primary Battery Installation as shown on page 41 of this chapter.

This completes the Secondary Battery Operational Check and Inspection procedure.

Section 30-17 Secondary Battery Troubleshooting Guide

Refer to Table 24-7 for a guide to troubleshooting issues with the secondary battery.

Complaint	Possible Cause	Remedy
Battery will not power FADEC "B"	Battery discharged	Charge battery
	Defective FADEC B switch	Replace
	Defective wiring	Repair
Ammeter indicates zero whether battery is charging or discharging	Defective VM1000FX indicator system	Replace (see chapter 77)
	Defective current sensor	Replace (see chapter 77)
Battery capacity consistently low ("does not hold a charge")	Charging voltage too low	Check, replace alternator or ACU
	Battery fault	Replace
Excessive battery temperature during charge (55^0C / 130^0F) or greater)	Battery fault	Replace
Battery case is deformed, bulging, or damaged.	Battery fault	Replace

Table 24-7 Secondary Battery troubleshooting Guide

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Section 24-60 DC Electrical Load Distribution

This section details the DC electrical load distribution for the airplane.

Section 60-01 DC Power Distribution

DC power from the primary battery is routed to the main distribution bus through a 70-amp circuit breaker. Power from the alternator is connected directly to the main distribution bus, with control and protection provided by the ACU. Circuit breakers and fuses protect individual circuits powering various airplane subsystems and components from the main distribution bus.

The airplane uses an entirely separate “FADEC power B bus” in emergencies. This bus, powered by the secondary battery, provides backup power for the engine FADEC system, Attitude Indicator (AI) gyro, and Turn Coordinator (TC) gyro operation for up to 60 minutes. In normal operation, this bus maintains the secondary battery in a fully charged condition at all times.

Section 60-02 Periodic Maintenance

The DC power distribution system does not require periodic maintenance cycles. In the event that a fault condition is noted perform the procedure Circuit Breaker and Master Relay Operation Check And Inspection on page 53 of this chapter.

Section 60-03 Circuit Breaker and Master Relay Procedures

This section details the removal, installation and inspections of the DC power distribution system. These procedures cover the removal and installation of an individual circuit breaker and the removal and installation of the master relay, K001.

CIRCUIT BREAKER REMOVAL

Perform this procedure to remove an individual circuit breaker from the circuit breaker panel.

1. Pull circuit breaker BAT 1 (CB001) to OPEN.
2. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
3. Locate two post lamps installed near the upper panel edge and remove hoods. Set them aside for later installation.
4. Locate and remove seven (7) 8-32 machine screws securing the circuit breaker panel to the instrument console. Retain these screws for later installation.

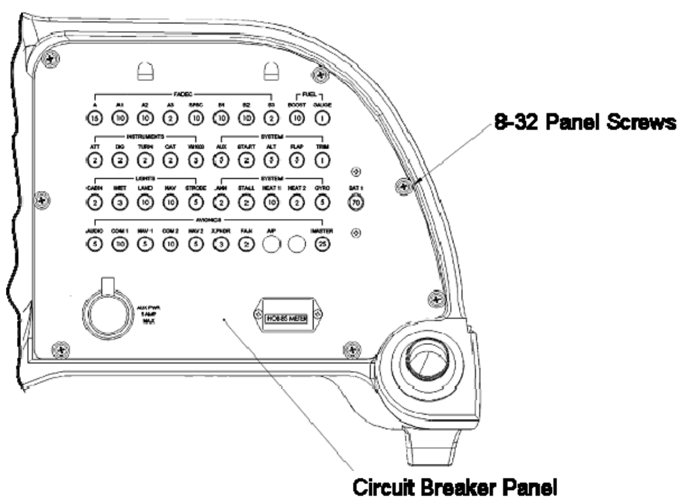


Figure 24-14 Circuit Breaker Panel

NOTE

The following step will remove the circuit breaker panel. There are a number of electrical wire harness connections to the instrument console and ground plane. Care must be taken to guide wire harnesses during the removal process in order to prevent damage.

5. Prepare a protective cushion surface just below and in front of the circuit breaker panel console position.
6. Gently pull the circuit breaker panel out of the instrument console and pivot face down in front of the console on the protective cushion surface.
7. Disconnect engine data processing unit ribbon cable connector PVM03A from DPU connector PVM03
8. Disconnect engine data processing unit cable connector PVM01 from DPU connector PVM01.

9. Disconnect engine data processing unit cable connector PVM04 from DPU connector PVM04.

10. Remove four (4) 6-32 screws holding the DPU mounting bracket to the circuit breaker panel assembly. See Figure 24-15 for the location of the DPU mounting screws.

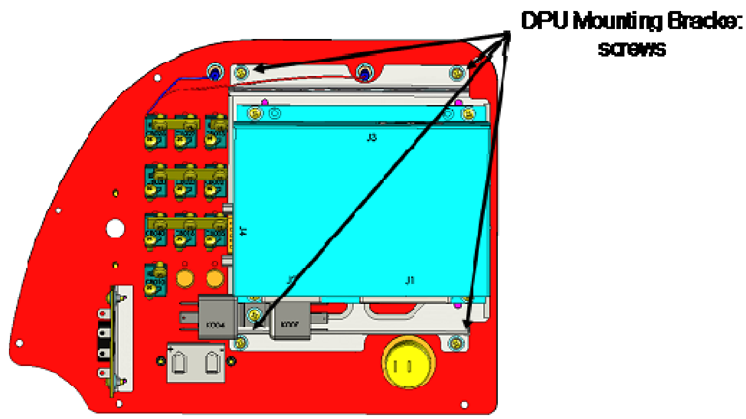


Figure 24-15 DPU Bracket Mounting Screws

11. Remove the DPU and mounting bracket as an assembly. Retain for later installation.

12. Remove the line ring terminal from the circuit breaker. If ganged together with other circuit breakers, remove the bus bar connecting the gang of circuit breakers. Removing the bus bar from the circuit breakers will facilitate the removal of the circuit breaker. Remove load ring terminal and remove circuit breaker from the panel. Retain these screws for later installation.

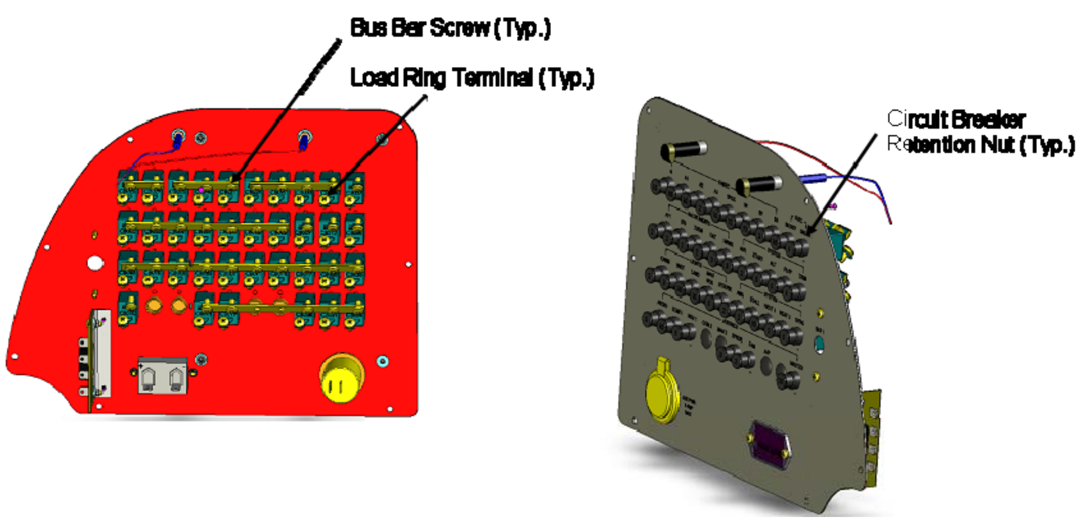


Figure 24-16 Circuit Breaker Mounting Hardware

13. Loosen circuit breaker retention nut and remove circuit breaker from panel assembly.

This completes the Circuit Breaker Removal procedure.

CIRCUIT BREAKER INSTALLATION

Perform this procedure to install an individual circuit breaker from the circuit breaker panel.

1. Position replacement circuit breaker in panel location and install retention nut finger tight.
2. Install circuit breaker bus bar screw and tighten to a torque of 9-12 in-lbs. If installing CB001, tighten the terminal bolt to a torque of 50 to 100 in-lbs.
3. Install circuit breaker load wire ring terminal(s) screw and tighten to a torque of 9-12 in/lbs. If installing CB001, tighten the terminal bolt to a torque of 50 to 100 in-lbs.
4. Tighten circuit breaker retention nut.
5. Position the DPU and the DPU mounting bracket assembly. Secure with four (4) 6-32 machine screws.
6. Connect engine data processing unit ribbon cable connector PVM03A to DPU connector PVM03
7. Connect engine data processing unit cable connector PVM01 to DPU connector PVM01.
8. Connect engine data processing unit cable connector PVM04 to DPU connector PVM04.

NOTE

The following steps install the circuit breaker panel into the instrument console. To avoid damage to the wiring harness, use care during the installation of the circuit breaker panel.

9. Gently pivot circuit breaker panel to vertical and slide into the instrument console.
10. Install seven (7) 8-32 machine screws securing the circuit breaker panel to the instrument console.
11. Install two post lamp hoods removed previously.
12. Push circuit breaker BAT 1 (CB001) to CLOSED
13. Perform operation check and inspection in the procedure Circuit Breaker and Master Relay Operation Check And Inspection on page 53 of this chapter.

This completes the Circuit Breaker Installation procedure.

MASTER RELAY REMOVAL

Perform this procedure to remove the master relay.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Pull the BAT1 (CB001) circuit breaker to OPEN.
3. Install a tail stand underneath the tail section of the airplane.
4. Remove the cabin aft bulkhead access panel, by removing securing screw hardware.
5. Locate and disconnect primary battery terminal cable P07A2N (Negative) from the battery.
6. Locate and disconnect primary battery terminal cable P01A2 (Positive) from the battery.

CAUTION

Failure to disconnect the battery can cause damage to the electrical circuitry of the airplane.

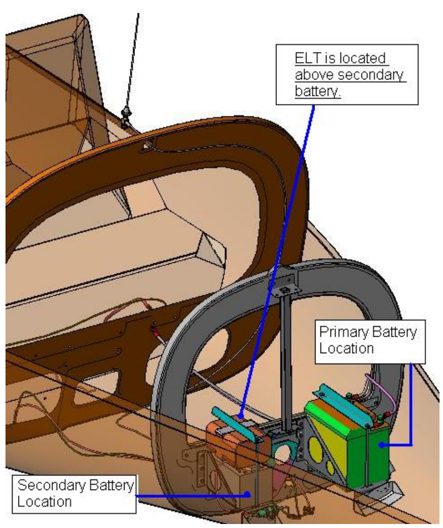


Figure 24-17 Primary Battery Location

7. Locate relay K001 (master relay) on the aft baggage bay bulkhead.

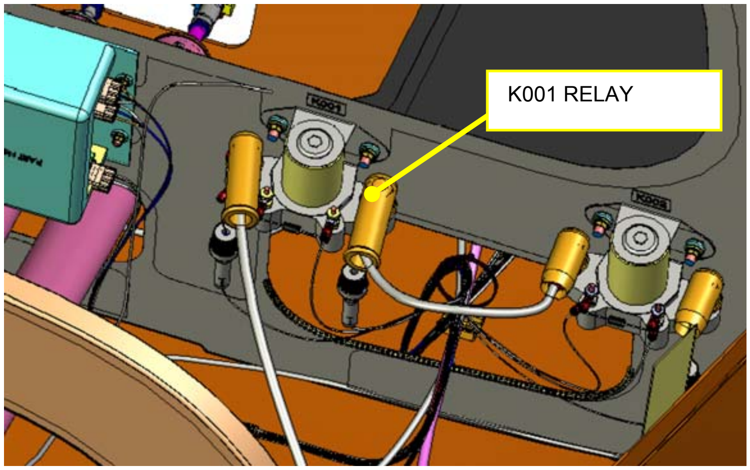


Figure 24-18 Master Relay Location

Figure 24-19 Master Relay Showing the Relay's Connections

8. Remove A1 terminal nut and slide ring terminals off the relay stud. Refer to Figure 24-19 for the location of the A1 terminal.
9. Remove A2 terminal nut and slide ring terminals off the relay stud. Refer to Figure 24-19 for the location of the A2 terminal.
10. Remove terminal X1 nut and slide ring terminals off the relay stud. Refer to Figure 24-19 for the location of the X1 terminal.
11. Remove terminal X2 nut and slide ring terminals off the relay stud. Refer to Figure 24-19 for the location of the X2 terminal.
12. Remove two (2) ¼-28 bolts, nuts and washers from either side of the K001 relay assembly and remove the relay.

This completes the Master Relay Removal procedure.

MASTER RELAY INSTALLATION

Perform this procedure to install the master relay.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Pull the BAT1 (CB001) circuit breaker to OPEN.
3. Install a tail stand underneath the tail section of the airplane.
4. Remove the cabin aft bulkhead access panel, by removing securing screw hardware.
5. Disconnect the negative then the positive leads from the primary battery. Isolate the negative terminal on the battery to prevent accidental connection.

CAUTION

Failure to disconnect the battery can cause damage to the electrical circuitry of the airplane.

6. Position master relay on baggage bay bulkhead below a placard marked K001 and install two (2) ¼-28 bolts, self-locking nuts and washers from either side of the relay assembly.
7. Install wire assemblies as indicated in Table 24-8 and secure with terminal nuts removed previously.

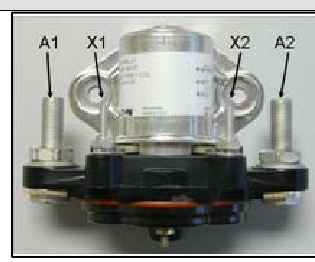
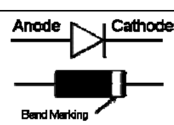
WIRE ID	RELAY TERMINAL	WIRE ID	RELAY TERMINAL	K001 Master Relay
C01E22	A1	P21A20	X1	
P01A2	A1	D3 (Cathode)	X1	
P21A20	A1	D3 (Anode)	X2	
P02A4	A2	P05A20	X2	
P33A2	A2			
P17C16	A2			

Table 24-8 Table showing the wiring on the master relay

8. Reconnect positive then the negative cable to airplane's primary battery. Apply a torque of 70 in-lb ± 5 in-lbs.
9. Install the aft baggage compartment closeout.
10. Press circuit breaker BAT 1 (CB001) to CLOSE.
11. Remove tail stand under the aircraft.
12. Perform operation check and inspection in the procedure Circuit Breaker and Master Relay Operation Check And Inspection on page 53 of this chapter.

This completes the Master Relay Installation procedure.

CIRCUIT BREAKER AND MASTER RELAY OPERATION CHECK AND INSPECTION

Perform the following procedure to do an operational check and inspection of the circuit breakers and master relay.

NOTE

Refer to the Master Relay Troubleshooting Guide for corrective actions as required.

1. Verify primary battery is fully charged, Refer to Primary Battery Operation Check and Inspection procedure on page 35 of this chapter.
2. Position the aircraft in a designated run up area and set the parking brake. Push circuit breaker BAT 1 (CB001) to CLOSE.
3. Inspect circuit breakers for evidence of damage, open or "tripped" condition.

NOTE

In the following steps, power will be applied to the DC power distribution system. If a circuit breaker trips when power is applied, discontinue the operation check procedure. Identify and correct cause of the trip prior to continuing. Refer to Table 24-10 for a guide to troubleshooting the DC power distribution.

4. Position the following distribution system circuit breakers to ON.

SYSTEM	SUB-SYSTEM	SYSTEM	SUB-SYSTEM
FADEC	A	LIGHTS	CABIN
	A1		INST
	A2		LAND
	A3		NAV
	SPSC		STROBE
	B1	SYSTEM	ANN
	B2		STALL
	B3		HEAT 1
FUEL	BOOST		HEAT 2
	GAGE		GYRO
INSTRUMENTS	ATT	SYSTEM	AUX
	DG		START
	TURN		ALT
	OAT		FLAP
	VM1000		TRIM

Table 24-9 Table indicating circuit breaker systems and sub systems

5. Position the aircraft master switch to ON.
6. Verify the master relay engages.
7. Verify there is an indication of the correct battery voltage on the VM1000FX engine status display.
8. Power on each of the systems and subsystems listed in the table above. Verify that each of the systems or subsystems power on and the associated circuit breaker remains set.
9. Power off each of the systems and subsystems activated.
10. Using standard flight manual procedure, start the engine.
11. Set engine RPM to idle (850 to 950 RPM).
12. Power on each of the systems and subsystems listed in the table above not already activated. Verify that each of the systems or subsystems power on and the associated circuit breaker remains set.
13. After the oil temperature reaches 75 degrees, set throttle to 1700 RPM.
14. Verify each system and subsystem activated previously remains powered on and associated circuit breaker remains set.
15. Set throttle for idle (850-950 RPM).
16. Using standard flight manual procedures, secure selectable loads and shut down the engine.
17. Position the aircraft master switch to OFF.

This completes the Circuit Breaker and Master Relay Operation Check And Inspection procedure.

Section 60-04 Master Relay Troubleshooting Guide

Refer to Table 24-10 for a guide to troubleshooting issues with the master relay.

Complaint	Possible Cause	Remedy
No power to system components	Master switch fault	Replace
	Master relay fault	Replace
	BATT circuit breaker fault	Replace
	Wiring fault	Repair
Circuit breaker will not set	System component fault	Isolate, repair or replace faulty system component
	Circuit breaker fault	Replace
	System wiring fault	Repair

Table 24-10 Master Relay Troubleshooting Guide

Section 60-05 Avionics Power Distribution

Power is provided to avionics components from a separate avionics power bus. The avionics bus receives power from the main power distribution bus via a single 25-amp circuit breaker, and is controlled by an avionics power relay operated by an instrument panel avionics master switch.

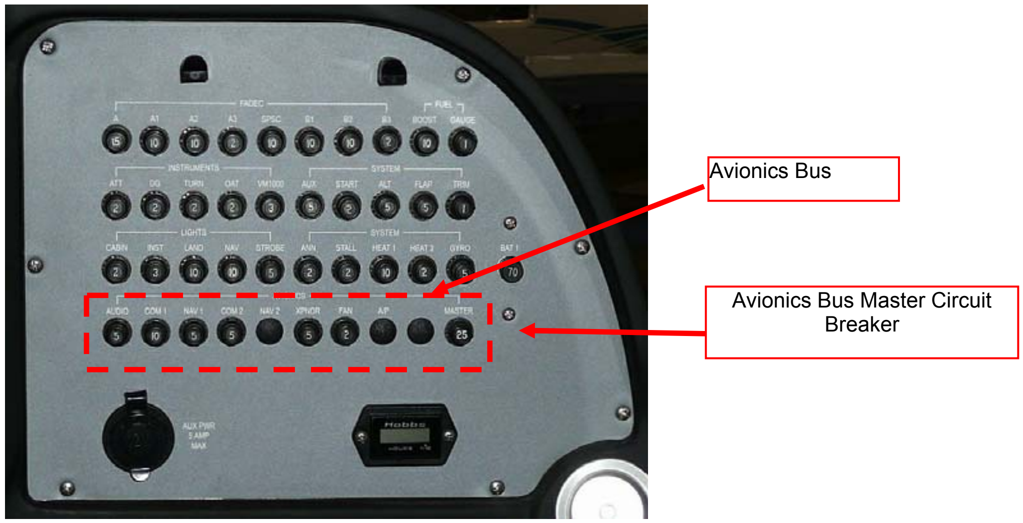


Figure 24-20 Circuit Breaker Panel – Avionics Bus

The avionics master relay is located behind the circuit breaker panel and is configured for normally-closed operation to provide a fail-operational installation. Thus, when the entire airplane electrical system is unpowered, the avionics power relay will be closed. When power is applied to the airplane electrical system with the avionics master switch in it's placarded OFF position, the avionics power relay will be energized to its open position, and the avionics bus will be unpowered.

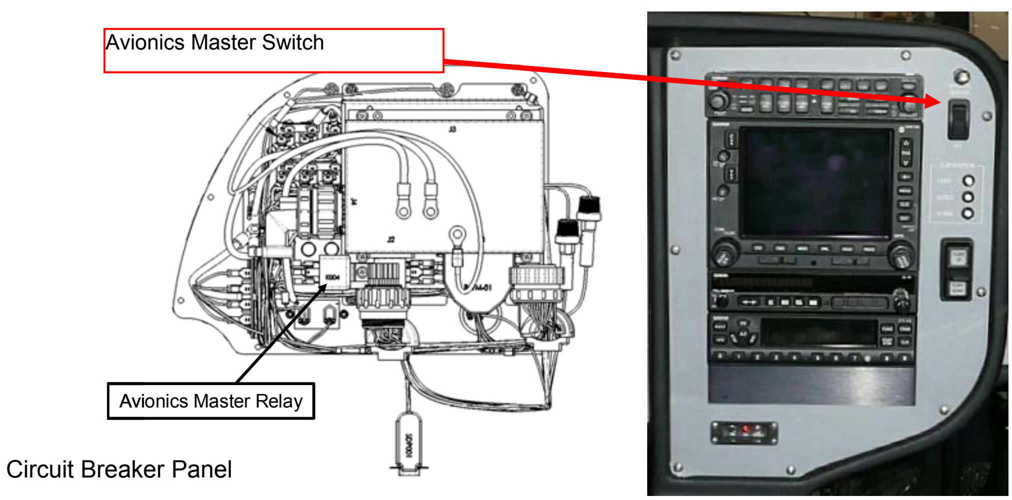


Figure 24-21 Avionics Panel – Avionics Master Switch

When the avionics master switch is moved to its placarded ON position, the avionics power relay relaxes to its normally closed (NC) position and aircraft power is applied to the avionics bus.

Section 60-06 Circuit Breaker Removal and Installation

The circuit breakers for the avionics are the same as the other circuit breakers mounted to the CB (Circuit Breaker) panel. The procedures to remove and install the circuit breakers for the avionic panel are the same as the other circuit breakers. To remove any of the circuit breakers for the avionics panel, refer to the procedure Circuit Breaker Removal on page 46 of this chapter. To install any of the circuit breakers for the avionics panel, refer to the procedure Circuit Breaker Installation on page 48 of this chapter.

NOTE

After installing a new avionics circuit breaker, perform an operational check and inspection per the procedure Avionics Power Distribution Operation Check And Inspection on page 62 of this chapter.

Section 60-07 Periodic Maintenance

The system for the avionics power distribution does not require periodic maintenance cycles. In the event a fault condition is noted, perform an operational check and inspection per the procedure Avionics Power Distribution Operation Check And Inspection on page 62 of this chapter.

Section 60-08 Avionics Power Distribution Procedures

This section details the removal, installation and inspections of the avionics master relay, K004, located on the rear side of the CB (Circuit Breaker) panel.

AVIONICS MASTER RELAY REMOVAL

Perform this procedure to remove the avionics master relay.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Pull the BAT1 (CB001) circuit breaker to OPEN.
3. Locate two post lamps installed near the upper panel edge and remove hoods. Set them aside for later installation.
4. Locate and remove seven (7) 8-32 machine screws securing the circuit breaker panel to the instrument console. Retain these screws for later installation.

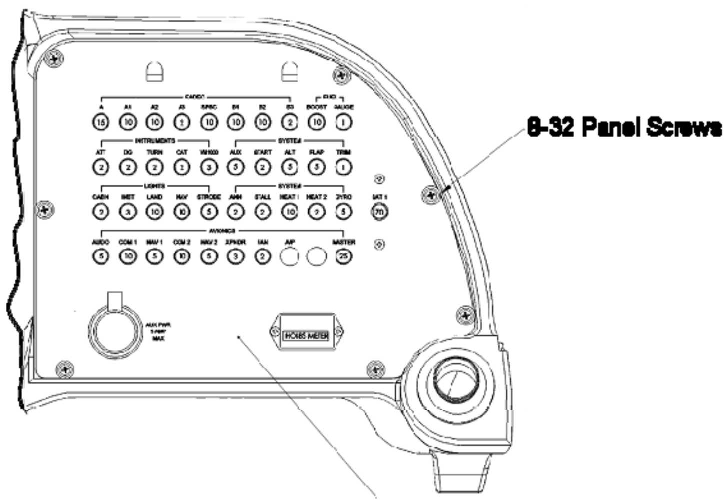


Figure 24-22 Circuit Breaker Panel

NOTE

The following step will remove the circuit breaker panel. There are a number of electrical wire harness connections to the instrument console and ground plane. Care must be taken to guide wire harnesses during the removal process in order to prevent damage.

5. Prepare a protective cushion surface just below and in front of the circuit breaker panel console position.
6. Gently pull the circuit breaker panel out of the instrument console and pivot face down in front of the console on the protective cushion surface.
7. Locate avionics master relay, K004, on the circuit breaker panel rear assembly.

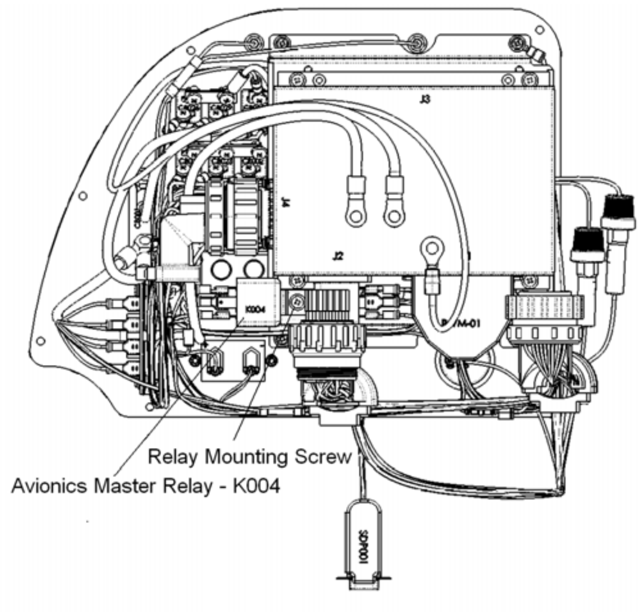


Figure 24-23 Avionics Master Relay Location

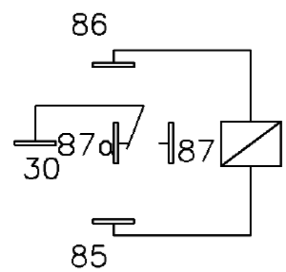
8. Remove quick disconnects attached to avionics master relay K004 terminals
9. Loosen and remove the 6-32 screw securing the relay to the circuit breaker panel. Retain this hardware for later installation.

This completes the Avionics Master Relay Removal procedure.

AVIONICS MASTER RELAY INSTALLATION

Perform this procedure to install the avionics master relay.

1. Position replacement K004 relay on the circuit breaker panel rear assembly as show above.
2. Secure relay K004 with one (1) 6-32 screw and washer removed previously.
3. Attach the following disconnects to relay K004 terminals.

<table>
<tr><th>Wire ID</th><th>Relay Terminal</th><th>K004 Avionics Master Relay</th></tr>
<tr><td>R04A12</td><td>87A</td><td rowspan="5">

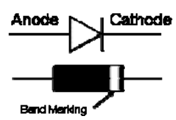
Schematic Drawing
(Botton views)</td></tr>
<tr><td>R03A12</td><td>30</td></tr>
<tr><td>R06A22</td><td>85</td></tr>
<tr><td>R07A22</td><td>86</td></tr>
<tr><td colspan="2">
</td></tr>
</table>

Table 24-11 Wiring chart for the Avionics Master Relay

4. Gently pivot circuit breaker panel to vertical and slide into the instrument console.
5. Install seven (7) 8-32 machine screws securing the circuit breaker panel to the instrument console.
6. Install two post lamp hoods removed previously.
7. Push circuit breaker BAT 1 (CB001) to CLOSED
8. Perform operation check and inspection in the procedure Circuit Breaker and Master Relay Operation Check And Inspection on page 62 of this chapter.

This completes the Avionics Master Relay procedure.

AVIONICS POWER DISTRIBUTION OPERATION CHECK AND INSPECTION

Perform this procedure to do an operational check and inspection of the avionics power distribution.

NOTE

Refer to the Avionics Power Subsystem Troubleshooting Guide for corrective actions as required.

1. Verify primary battery is fully charged. To verify the charge on the primary battery, refer to the procedure Primary Battery Operation Check and Inspection on page 35 of this chapter.
2. Position the airplane in a designated run up area and set the parking brake.
3. Push circuit breaker BAT 1 (CB001) CLOSED.
4. Inspect circuit breakers for evidence of damage, open or “tripped” condition.

NOTE

In the following steps, power will be applied to the avionics power distribution system. If a circuit breaker trips when power is applied, discontinue the operation check procedure. Identify and correct cause of the trip prior to continuing. Refer to the Avionics Power Distribution Troubleshooting Guide below for possible cause and remedy.

5. Position the following distribution system circuit breakers to ON.

System	Sub-System	System	Sub-System
FADEC	A	LIGHT	CABIN
	A1		INST
	A2		LAND
	A3		NAV
	SPSC		STROBE
	B1	SYSTEM	ANN
	B2		STALL
	B3		HEAT 1
FUEL	BOOST		HEAT 2
	GAGE		GYRO
INSTRUMENT	ATT	SYSTEM	AUX
	DG		START
	TURN		ALT

System	Sub-System	System	Sub-System
	OAT		FLAP
	VM1000		TRIM
AVIONICS	AUDIO		
	COM 1		
	NAV 1		
	COM 2		
	NAV 2		
	XPNDR		
	FAN		
	MASTER		

Table 24-12 Table indicating circuit breaker systems and sub systems

6. Position the aircraft master switch to ON
7. Position the avionics master switch to ON
8. Place each of the systems and subsystems listed in the table above in the ON condition and verify each power up and associated circuit breaker remains set.
9. Place each system not required for starting in the OFF condition
10. Position the Avionics Master Switch OFF
11. Verify radios are OFF
12. Using standard flight manual procedure, start the engine.
13. Set engine RPM to idle (850 to 950 RPM)
14. Position the Avionics Master Switch to ON
15. Place each of the systems and subsystems listed in the table above in the ON condition if not already activated. Verify each power up and associated circuit breaker remains set.
16. After the oil temperature reaches 75 degrees, set throttle to 1700 RPM
17. Verify each system and subsystem activated previously remains powered on and the associated circuit breaker remains set.
18. Set throttle for idle (850-950 RPM)
19. Position the avionics master switch OFF
20. Verify the radios are OFF
21. Shut down the engine using standard flight manual procedure.
22. Position the aircraft master switch OFF.

This completes the Avionics Power Distribution Operation Check And Inspection procedure.

Section 60-09 Avionics Power Subsystem Troubleshooting Guide

Refer to Table 24-13 for a guide in troubleshooting issues with the avionics power subsystem.

Complaint	Possible Cause	Remedy
No power to avionics components	Avionics master switch fault	Replace
	Avionics master relay fault	Replace
	Avionics master circuit breaker fault	Replace
	Defective wiring	Repair
Avionics circuit breaker will not set	Avionics component fault	Replace faulty avionics component
	Circuit breaker fault	Replace
	System wiring fault	Repair

Table 24-13 Avionics Power Subsystem Troubleshooting Guide

CHAPTER 25

EQUIPMENT AND FURNISHINGS

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Table of Contents

SECTION 25-00	GENERAL	5
SECTION 25-10	FLIGHT COMPARTMENT EQUIPMENT	7
SECTION 10-01	CASES AND CONTAINERS	7
SECTION 25-50	CARGO COMPARTMENT	9
SECTION 25-60	EMERGENCY EQUIPMENT	11
SECTION 60-01	FIRE EXTINGUISHER	11
SECTION 60-02	FIRE EXTINGUISHER MOUNTING	11
SECTION 60-03	FIRE EXTINGUISHER INSPECTION	12
SECTION 60-04	SAFETY HAMMER	12
SECTION 60-05	EMERGENCY LOCATOR TRANSMITTER	13
SECTION 60-06	ELT ANTENNA	13
SECTION 60-07	ELT ANTENNA PROCEDURES	13
ELT ANTENNA REMOVAL		14
ELT ANTENNA INSTALLATION		16
ELT ANTENNA CABLE REMOVAL		17
ELT ANTENNA CABLE INSTALLATION		18
SECTION 60-08	121.5/243 MHZ EMERGENCY LOCATOR TRANSMITTER (ELT)	19
SECTION 60-09	PERIODIC MAINTENANCE	21
SECTION 60-10	121.5/243 MHZ ELT SERVICE PROCEDURES	22
ELT REMOVAL		23
ELT BATTERY REMOVAL		25
ELT BATTERY INSTALLATION		26
ELT INSTALLATION		27
ELT REMOTE CONTROL SWITCH REMOVAL		28
ELT REMOTE SWITCH BATTERY CELL REMOVAL AND INSTALLATION		29
ELT REMOTE SWITCH INSTALLATION		31
ELT OPERATIONAL CHECK AND INSPECTION		32
ELT G-SWITCH CHECK		35
ELT 121.5 MHZ TRANSMITTER TEST		36
RESET ELT FROM REMOTE SWITCH		37
RESET ELT FROM LOCAL CONTROLS		38
SECTION 60-11	121.5/243 MHZ ELT TROUBLESHOOTING GUIDE	39
SECTION 60-12	406/121.5 MHZ EMERGENCY LOCATOR TRANSMITTER (ELT)	40
SECTION 60-13	PERIODIC MAINTENANCE	42
SECTION 60-14	406/121.5 MHZ ELT SERVICE PROCEDURES	43
ELT REMOVAL		44
ELT BATTERY REMOVAL		46
ELT BATTERY INSTALLATION		49
ELT INSTALLATION		50
ELT REMOTE CONTROL SWITCH REMOVAL		51
ELT REMOTE SWITCH BATTERY CELL REMOVAL AND INSTALLATION		52
ELT REMOTE CONTROL SWITCH INSTALLATION		54
ELT OPERATION CHECK AND INSPECTION		55
G-SWITCH CHECK		58
ELT 121.5 MHZ TRANSMITTER TEST		60
ELT 406 MHZ TRANSMITTER TEST		61
RESET ELT FROM THE REMOTE SWITCH		63
RESET ELT FROM LOCAL CONTROLS		64
SECTION 60-15	ELT REGISTRATION	65
SECTION 60-16	406/121.5 MHZ ELT TROUBLESHOOTING GUIDE	66

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Section 25-00 General

The airplane's accommodation includes the flight compartment, for the pilot and copilot or passenger and a contiguous baggage area immediately to the rear of the flight compartment.

Figure 25-1 View of Aircraft Interior

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Section 25-10 Flight Compartment Equipment

Furnishings in the flight compartment include removable carpets, removable seat and back cushions, and four-point harnesses for both occupants.

The carpets and seat cushions uses Velcro® hook and loop fasteners to hold them in place.

Seat belt and shoulder harness components are removable. To remove them from the airplane, remove the bolts and washers securing them to their attachment points. Replace any worn seat belt or harness components.

CAUTION

Use only water and mild soap or detergents to clean seat belt and shoulder harness components. Do not use solvents of any type. Ensure water does not enter harness buckle assemblies.

Section 10-01 Cases and Containers

There is a small storage area inside the headrest section of the port and starboard side seat back structures. These storage areas can hold small items that are easily accessible to either the pilot or passenger during flight. The starboard-side storage area holds the flight manual, micro-fiber cleaning cloth, computer interface cable and safety hammer. The portside storage area can hold such items as a customer supplied bungee cord that would hold the rudder in the locked position, headsets, charts, and other small items.

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Section 25-50 Cargo Compartment

The cargo (baggage) compartment is immediately aft of, and contiguous with, the flight compartment. Hook and loop fasteners secure the carpet to the floor of the cargo compartment. An access plate of composite material is in the cargo compartment floor. It provides access for maintenance of flight control components in the center fuselage area.

NOTE

Tie-downs points in the baggage or cargo area are provided to allow cargo to be secured to the airplane during flight.

WARNING

IT IS REQUIRED TO SECURE ANY BAGGAGE OR CARGO AGAINST UNINTENDED MOVEMENT IN FLIGHT. MAXIMUM BAGGAGE/CARGO = 100 LBS. DO NOT EXCEED AIRCRAFT WEIGHT AND BALANCE LIMITATIONS.

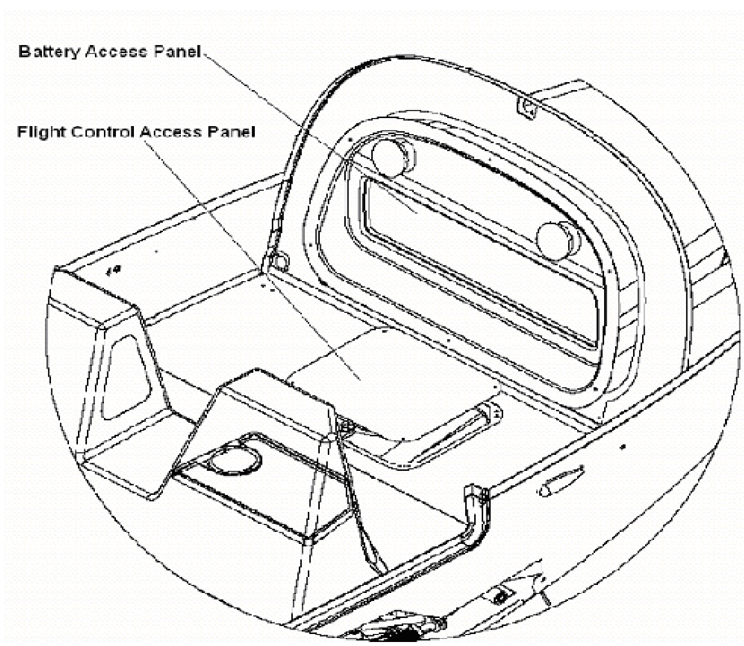

Figure 25-2 Battery Access and Flight control Access Panels

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Section 25-60 Emergency Equipment

The airplane comes equipped with a fire extinguisher, safety hammer, and an Emergency Locator Transmitter, ELT. The sections that follow discuss each of these items.

Section 60-01 Fire Extinguisher

Secured behind the co-pilot/passenger seat in the baggage compartment is the portable fire extinguisher. The fire extinguisher is a Halon 1211-1301 blend fire extinguishers and is a one-time use, disposable-type, non-refillable aviation fire extinguisher. Figure 25-3 shows the location of the fire extinguisher.

Figure 25-3 Location of Fire Extinguisher Behind the Passenger's/Co-Pilot's Seat

Section 60-02 Fire Extinguisher Mounting

The fire extinguisher comes with an integral holding bracket. This bracket mounts the back of the Passenger's/Co-Pilot's seat using two screws, washers and nuts. Use only the bracket that comes with the fire extinguisher.

CAUTION

Before replacing or changing the fire extinguisher, check the local airworthiness authority for Airworthiness Directives or service documents specific to the make and model of the fire extinguisher.

Section 60-03 Fire Extinguisher Inspection

Inspect the fire extinguisher on the intervals as shown in Chapter 5 – *Time Limits* of this manual. The following items are inspected.

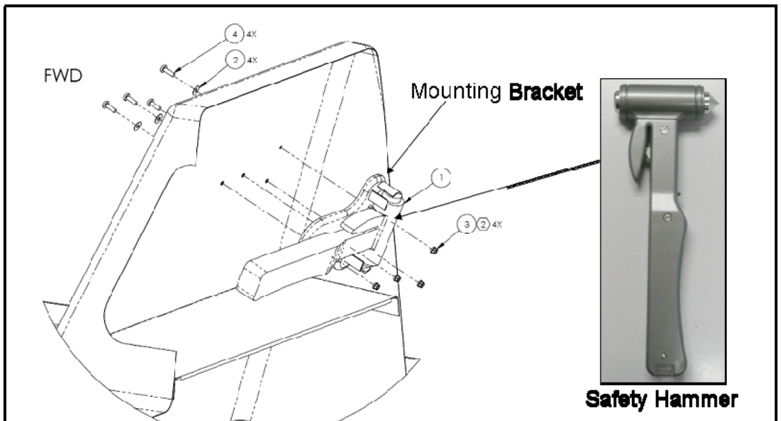
- Extinguisher is in its designated place
- No obstruction to access or visibility
- Operating instructions on nameplate legible and facing outward
- Safety seal (plastic wire) is not broken or missing
- No obvious physical damage, corrosion, leakage, or clogged nozzle
- HMIS label is in place
- Weight the fire extinguisher. The weight must 686 grams (24.2 oz.) or greater

NOTE

When an inspection reveals a deficiency in any of the conditions listed above, it shall be returned from service, not discharged, and returned to H3R, a fire equipment dealer or distributor to permit recovery of the Halon.

Section 60-04 Safety Hammer

Mounted in the passenger seat back storage compartment, within easy reach of the pilot in command, is the safety hammer. In the event of an emergency and the door or doors will not open, use this hammer to break the door windscreen (window).



ITEM NO.	QTY	PART NUMBER	DESCRIPTION
1	1	21-919U	EMERGENCY SAFETY HAMMER
2	4	AN960C-6L	WASHER, No. 6
3	4	MS21043-06	NUT, SELF LOCKING 6-32, CORROSION RESISTANT
4	4	MS51957-29	PAN HEAD SCREW 6-32 CORROSION RESISTANT

Figure 25-4 Safety Hammer Location and Mounting

Section 60-05 Emergency Locator Transmitter

The Liberty XL-2 is fitted with an Emergency Locator Transmitter in the aircraft's aft fuselage section above the secondary battery, with a fuselage-mounted antenna located on the top of the fuselage between the two COM/NAV/GPS antennas. Currently certified for the airplane are two ELT types, the Ameriking AK450 and the Artex ME406. Maintenance requirements for these two model units differ. Consult the aircraft equipment list to determine which unit is in the airplane.

The Ameriking AK450 is complaint with TSO-C91a to transmit on 121.5 MHz and 243 MHz.

The Artex ME406 is compliant with TSO-C126 to transmit on 121.5 MHz and 406 MHz.

Section 60-06 ELT Antenna

The ELT transmitters use different ELT antenna. The 121.5/243 MHz ELT uses Ameriking P/N 4056017; the 121.5/406MHz ELT uses Artex P/N 110-773 The location and mounting are the same for both antennas. The ELT antenna mounts on top of the fuselage aft of the doors between the two COM/NAV/GPS antennas. Access to the connector and cable for the ELT antenna is through the round access panel located just forward of the aft cabin bulkhead.

Figure 25-6 AK450 Antenna Placement

Figure 25-7 ME406 Antenna Placement

Section 60-07 ELT Antenna Procedures

This section details the removal and installation of the ELT antenna.

ELT ANTENNA REMOVAL

Perform this procedure to remove the ELT antenna.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Remove the three panel access screws from the access panel that is just forward of the aft cabin bulkhead. Retain screws and access cover for later installation.
4. Disconnect the cable leading to the antenna. See Figure 25-8 .

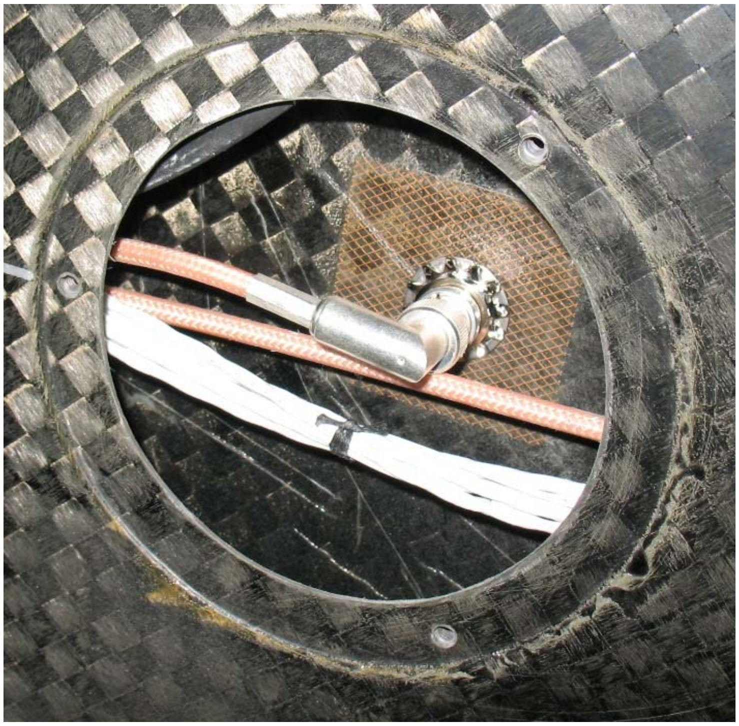


Figure 25-8 ELT Antenna Cable and Connection

5. Remove the nut and two washers. See Figure 25-9.

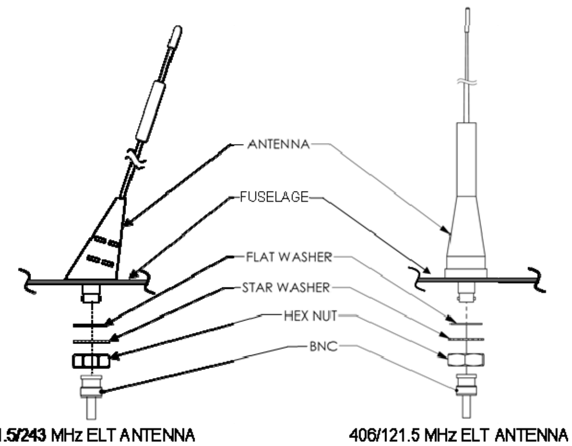


Figure 25-9 Exploded View of ELT Antenna Installations

6. Remove the ELT antenna.
7. If installing a replacement antenna later, cap and secure the cable end, cover the ELT antenna-mounting hole in the fuselage, and install the ELT antenna access cover.

This completes the ELT Antenna Removal procedure.

ELT ANTENNA INSTALLATION

Perform this procedure to install the ELT antenna.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. If installing the ELT antenna immediately after removing the previous antenna, go to step 5.
2. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
3. Install a tail stand underneath the tail section of the airplane.
4. Remove the three panel access screws from the access panel that is just forward of the aft cabin bulkhead. Retain screws and access cover for later installation.
5. Remove any cap or covering installed during the removal procedure.
6. Insert the replacement antenna through the hole in the fuselage.
7. Install the two washers and the nut to secure the antenna to the fuselage. See Figure 25-10.

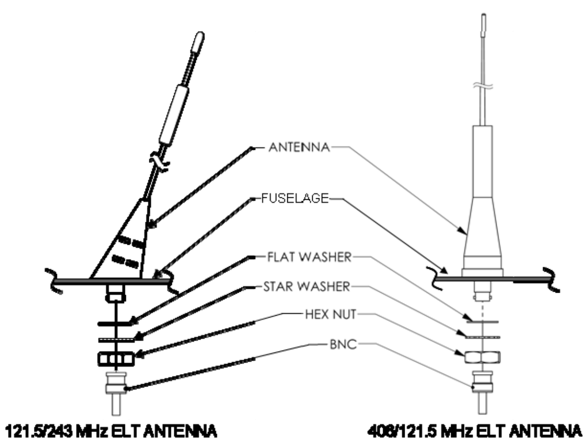


Figure 25-10 Exploded View of ELT Antenna Installations

8. Connect the coax cable AELT01A22 to the antenna.
9. Install the ELT antenna access cover.
10. Secure the cover with screws for the access cover.
11. Remove the tail stand.

This completes the ELT Antenna Installation procedure.

ELT ANTENNA CABLE REMOVAL

Perform this procedure to remove the ELT antenna cable.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Remove the three panel access screws from the access panel that is just forward of the aft cabin bulkhead. Retain screws and access cover for later installation.
4. Disconnect the cable leading to the antenna. See Figure 25-8.
5. Remove the two bolts, washers, and nuts securing the split nylon cable feed-thru to the baggage bay aft bulkhead. See Figure 25-10.

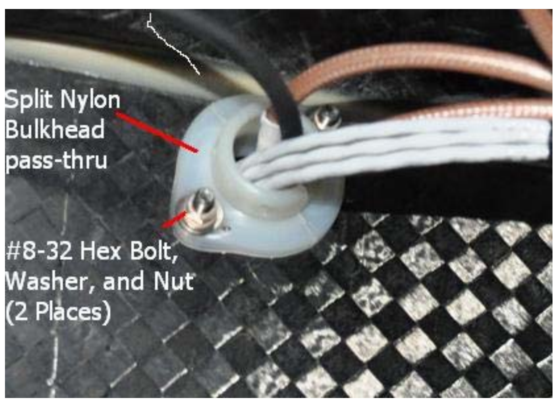


Figure 25-11 Split Nylon cable pass-thru

6. Remove the feed-thru.
7. Carefully feed the cable and connector back through the access hole behind the ELT antenna.
8. Remove sufficient cable ties along the path of the cable to free up the cable from the surface of the fuselage.
9. Carefully disconnect the antenna cable from the ELT.
10. Feed the cable back through aft fuselage bulkhead and out of the airplane.

This completes the ELT Antenna Cable Removal procedure.

ELT ANTENNA CABLE INSTALLATION

Perform this procedure to install the ELT antenna cable.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Remove the three panel access screws from the access panel that is just forward of the aft cabin bulkhead. Retain screws and access cover for later installation.
4. If installing the cable immediately after removing the existing cable, go to step 7 of this procedure.
5. Disconnect the cable leading to the antenna. See Figure 25-7.
6. Remove existing cable as detailed in the ELT Antenna Cable Removal procedure on page 16 of this manual.
7. Feed one end of the replacement cable through the aft bulkhead feed-thru.
8. Connect the cable to the ELT.
9. Run the cable along the same path as was the existing cable.
10. Add sufficient cable ties to secure the cable to the other cables and to the fuselage.
11. Feed the bundle of cables through the split nylon feed-thru then through the baggage bay aft bulkhead.
12. Secure the split nylon feed-thru to the bulkhead using the bolts, washers, and nuts removed in step 5 of the ELT Antenna Cable Removal procedure.
13. Connect the cable connector to the ELT antenna.
14. Install the ELT Antenna access cover.
15. Install the baggage bay aft bulkhead access panel to the bulkhead.

This completes the ELT Antenna Cable installation procedure.

Section 60-08 121.5/243 MHz Emergency Locator Transmitter (ELT)

The Liberty XL-2 can be equipped with an Ameriking model AK450, 121.5/243 MHz Emergency Locator Transmitter (ELT) like that shown in Figure 25-12. The location of the unit is in the rear fuselage directly above the secondary battery as shown in Figure 25-13. Two hold down bolts and a hold down bracket secure the ELT as shown in Figure 25-15. Installed on the upper fuselage, as shown in Figure 25-6, is an external, crash resistant, fuselage mounted whip antenna dedicated to the ELT.

Operation of the unit is automatic in the event of sudden impact or excessive deceleration forces. Upon activation, the unit will broadcast an internationally recognized distress signal on frequencies of 121.5 MHz and 243.0 MHz for a minimum of 72 hours after activation. Manual activation, status monitoring, and reset of the ELT may be accomplished either from an avionics panel located remote switch or from controls mounted on the ELT itself. Refer to Figure 25-12 and Figure 25-14 for location of manual controls.

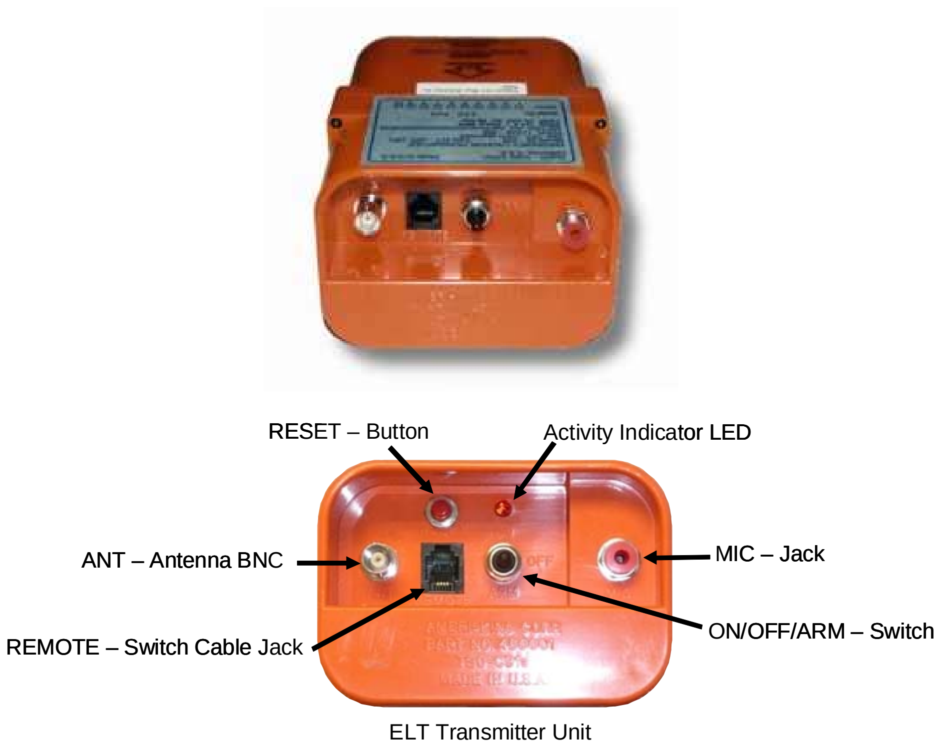


Figure 25-12 Emergency Locator Transmitter – 121.5/243 MHz – TSO-C91a

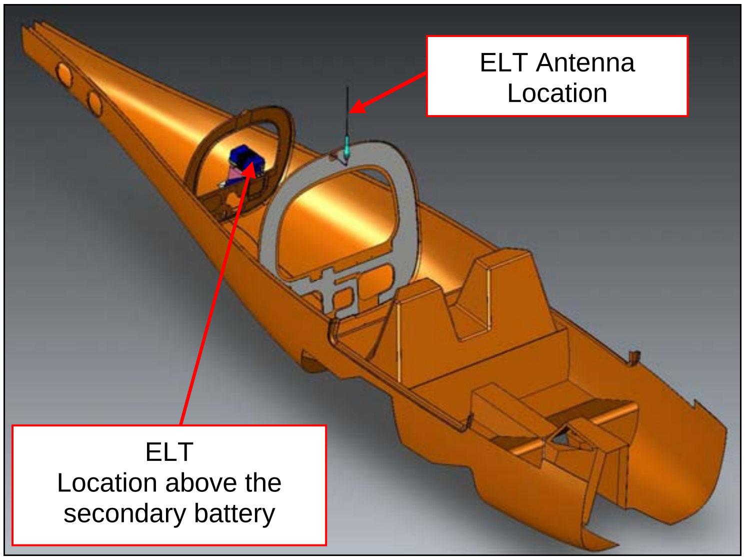


Figure 25-13 Location of the Emergency Locator Transmitter and Antenna

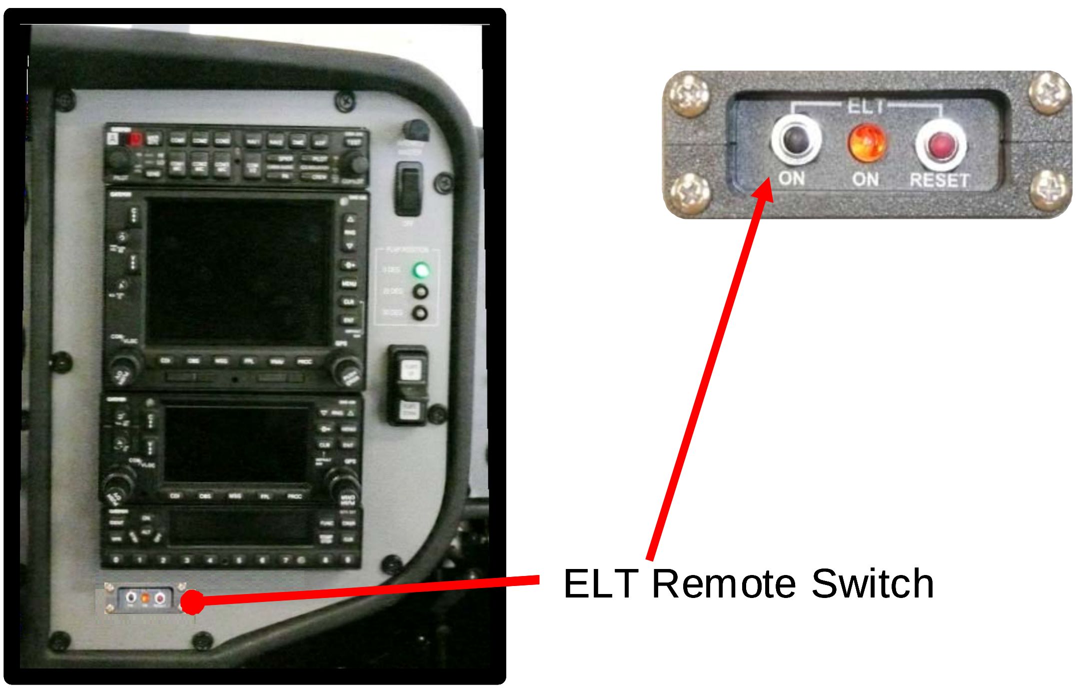


Figure 25-14 ELT Remote Switch Location

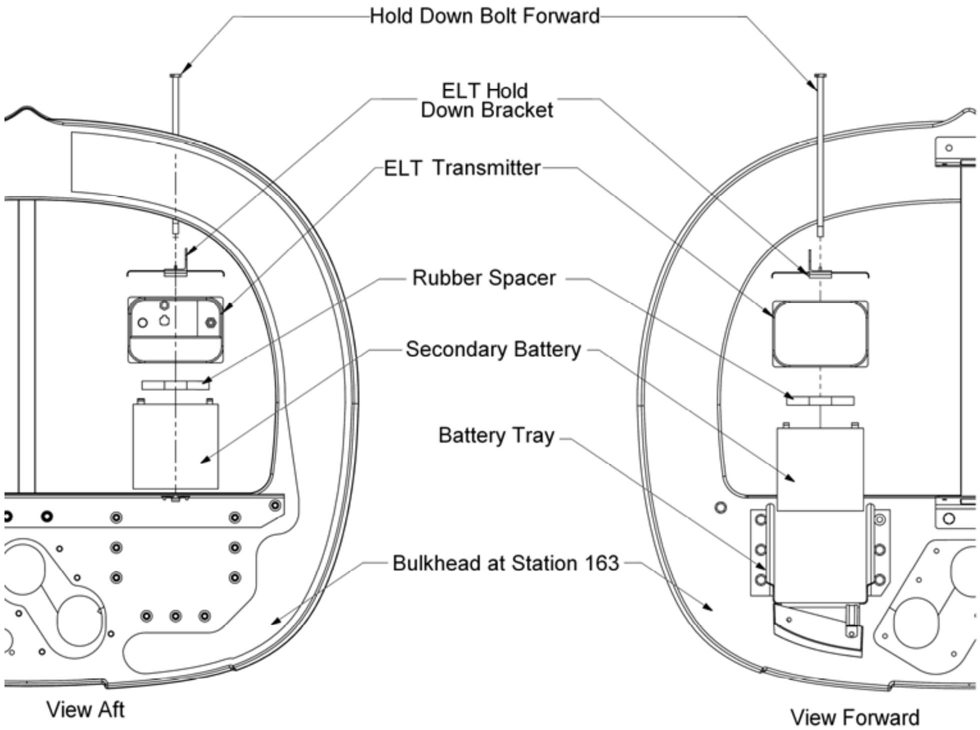


Figure 25-15 ELT Installation

NOTE

Periodic maintenance requires an operational check of the ELT, however, avoid unnecessary ELT operational checks, as any operation of the ELT reduces its battery life.

Section 60-09 Periodic Maintenance

In the United States, minimum annual maintenance requirements for an ELT are in paragraph (d) of 14 CFR Part 91.207. That paragraph states:

(d) Each emergency locator transmitter required by paragraph (a) of this section must be inspected within 12 calendar months after the last inspection for--
(1) Proper installation;
(2) Battery corrosion;
(3) Operation of the controls and crash sensor; and
(4) The presence of a sufficient signal radiated from its antenna.

In addition, to ensure continued reliability and airworthiness, inspect the ELT for damage and wear caused by age, exposure to the elements, vibration, or other causes. Inspections are to take place annually per FAR part 91.409. FAR 43, Appendix D(i). The appendix states, in part that each 100-hour or annual inspection shall inspect the following components of the ELT:

(1) (ELT unit and mount) for improper installation and insecure mounting.

(2) Wiring and conduits - for improper routing, insecure mounting, and obvious defects.

(3) Bonding and shielding - for improper installation and poor condition.

(4) Antenna, including trailing antenna-for poor condition, insecure mounting, and improper operation.

NOTE

All references to maintenance for the United States shall also apply to all ELT users outside of the US unless otherwise required by relevant national regulations.

Compliance with these requirements for the 121.5/243 MHz ELT is part of Section 60-10, 121.5/243 MHz ELT Service Procedures, of this chapter. Refer to the ELT Operational Check and Inspection procedure on page 32 for additional information.

Section 60-10 121.5/243 MHz ELT Service Procedures

This section details the ELT Service Procedures. The following is a list of procedures:

Procedure Title	Page
ELT Removal	23
ELT Battery Removal	25
ELT Battery Installation	26
ELT Installation	27
ELT Remote Control Switch Removal	28
ELT Remote Switch Battery Cell Removal and Installation	29
ELT Remote Switch Installation	31
ELT Operational Check and Inspection	32
ELT G-Switch Check	35
ELT 121.5 MHz Transmitter Test	36
Reset ELT From Remote Switch	37
Reset ELT From Local Controls	38

ELT REMOVAL

Perform this procedure to remove the ELT from the airplane.

CAUTION

Take care when removing the ELT from its mounting. Removed improperly, the ELT can trigger a false emergency signal. Possible fines may be imposed if there is a false activation of an ELT.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position aircraft master switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Remove the cabin aft bulkhead access panel, by removing securing screw hardware.
4. Access the ELT location above the secondary battery on the aircraft port side.
5. Position the ELT manual control switch to OFF.
6. Remove cable connection from ELT jack marked REMOTE.
7. Inspect remote cable connector for damage or evidence of corrosion.
8. Disconnect the BNC RF antenna connector.
9. Inspect BNC cable connector for damage or evidence of corrosion.
10. Remove the forward and aft ELT hold down bolts followed by the hold down bar.
11. Remove ELT unit from its tray.
12. Remove rubber spacer for ELT installation use.

NOTE

Evidence of damage requires inspection, repair, or replacement by a repair station qualified for TSO-C91a ELT units. Do not install an ELT with evidence of damage.

13. Inspect ELT remote jack pins for damage or evidence of corrosion.
14. Inspect ELT case and ELT Tray for cracks or other obvious damage.
15. Inspect all installation metal parts for signs of corrosion.

This completes the ELT Removal procedure.

ELT BATTERY REMOVAL

Perform this procedure to remove the ELT Battery.

CAUTION

Take care when removing the ELT from its mounting. Removed improperly, the ELT can trigger a false emergency signal. Possible fines may be imposed if there is a false activation of an ELT.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Remove ELT from the aircraft. Refer to the ELT Removal procedure on page 23 of this chapter.
2. Using a 3/32" Hex Driver, evenly loosen and remove four (4) retaining screws and split lock washers attaching the battery case to the ELT Transmitter Assembly.
3. Remove batteries form the battery case. There are six (6) D sized batteries snug fit to the case.
4. Verify the battery expiration date. Refer to Chapter 05 – *Time Limits/Maintenance Checks/Inspection Intervals* for limitations. If the battery pack has not expired, it is permissible to use the battery. If any of the conditions shown in Table 25-1 exist, install a new battery pack. Refer to ELT Battery Installation procedure on page 26 of this chapter to install a new battery pack.

Condition Requiring Battery Replacement
After use in an emergency
When the transmitter has been in use for more than 1 cumulative hour
After an inadvertent activation of unknown duration
On or before the battery expiration date
On any evidence of corrosion or leakage of any cell

Table 25-1 Conditions Requiring Battery Replacement

5. Examine batteries and battery case contacts for any dirt or corrosion. Clean the contacts as necessary using electrical contact cleaner and a stiff brush.

This completes ELT Battery Removal procedure.

ELT BATTERY INSTALLATION

Perform this procedure to install the ELT battery.

NOTE

Replacement batteries are Duracell MN1300 model cells. Each of the installed cells must all have the same replacement date markings.

1. If installing new cells, record the battery replacement data.
2. Install batteries, Duracell MN1300, to match battery case installation placard.
3. Check polarity of the installed battery set as shown in Figure 25-16.

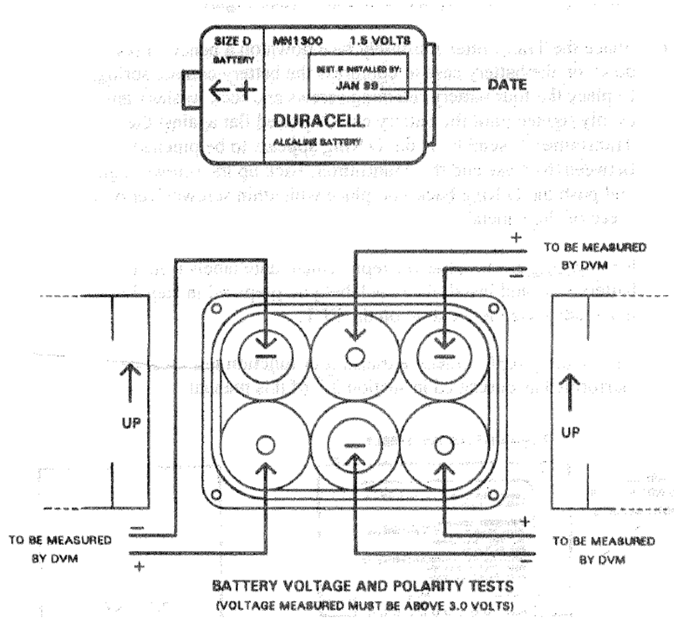


Figure 25-16 Battery Polarity Check

4. Install the battery case to the Transmitter Assembly making sure all ± polarities arrow markings located on the battery case, the battery separator and transmitter assembly are in the same direction. Secure with four (4) Hex Drive screws removed previously.
5. Enter battery replacement/inspection information in the aircraft log book as applicable.
6. If installed, remove expired battery label from the ELT case.
7. If installing new batteries, place a new expiration data label on the ELT case in a location that will be visible after installation.

This completes the ELT Battery Installation procedure.

ELT INSTALLATION

Perform this procedure to install the Emergency Locator Transmitter.

CAUTION

Take care when removing the ELT from its mounting. Removed improperly, the ELT can trigger a false emergency signal. Possible fines may be imposed if there is a false activation of an ELT.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position aircraft master switch to OFF
2. Install a tail stand underneath the tail section of the airplane.
3. Remove the cabin aft bulkhead access panel, by removing securing screw hardware.
4. Position the ELT manual control switch to OFF.
5. Position rubber spacer removed previously on top and center of the secondary battery.
6. Connect remote switch cable to ELT jack marked REMOTE.
7. Connect the BNC RF ELT antenna connector.
8. Position the ELT with data plate showing and arrow pointing forward. Locate the ELT at center of secondary battery and on top of rubber spacer.
9. Install hold down bar followed by the forward and aft ELT hold down bolts. Safety wire bolt heads.
10. Position the ON/OFF/ARM switch to ARM.
11. Install the cabin aft bulkhead access panel with hardware removed previously.
12. Remove tail stand.
13. Prior to return to service, perform ELT 121.5 MHz Transmitter Test procedure as shown on page 36 of this chapter.

This completes the ELT Installation procedure.

ELT REMOTE CONTROL SWITCH REMOVAL

Perform this procedure to remove the ELT remote control switch.

1. Position aircraft master switch to OFF.
2. Verify the ELT remote switch activity LED is off.
3. Remove four (4) 4-40 machine screws securing remote switch to the avionics panel. Refer to Figure 25-17 for screw locations.
4. Slide remote switch assembly out of the avionics panel.
5. Locate and disconnect switch extension cable connector.

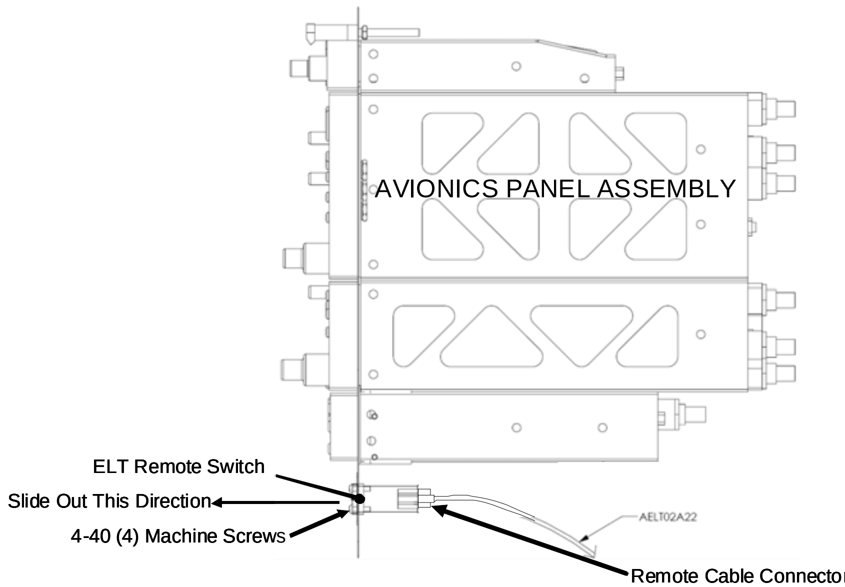


Figure 25-17 Location of the Remote ELT Switch and its Removal

6. Inspect remote switch extension cable connector for damage or evidence of corrosion
7. Inspect remote switch assembly for damage or evidence of corrosion.

This completes the ELT Remote Control Switch Removal procedure.

ELT Remote Switch Battery Cell Removal and Installation

Perform this procedure to remove and install the ELT remote switch battery cell.

1. Referring to Figure 25-18, remove three (3) retaining screws from the remote switch body to access the battery compartment.

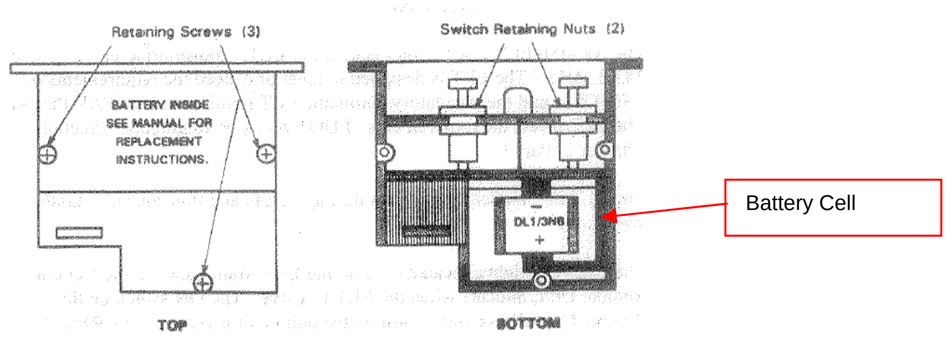


Figure 25-18 Remote Switch Battery Access

2. Loosen two switch-retaining nuts located on the front of the unit as shown in Figure 25-18.
3. Remove the top half of the unit, exposing the battery compartment.
4. Remove the battery and inspect the battery compartment for corrosion or damage.
5. Verify the battery expiration date. Refer to Chapter 05 – *Time Limits/Maintenance Checks/Inspection Intervals* for limitations. If the battery pack has not expired it may be reinstalled. . If any of the conditions shown in Table 25-2 exist, install a new battery pack.

Condition Requiring Battery Replacement
After use in an emergency
When the transmitter has been in use for more than 1 cumulative hour
After an inadvertent activation of unknown duration
On or before the battery expiration date
On any evidence of corrosion or leakage of any cell

Table 25-2 Conditions Requiring Battery Replacement

CAUTION

The battery compartment contacts are nickel and gold plated spring steel. Abrasive material will remove this plating. If the contacts appear pitted with corrosion they must be replaced.

6. Clean the contacts with electrical contact cleaner and stiff brush

7. Insert battery, Duracell model DL 1/3NB, with the polarity shown in Figure 25-18 above.
8. Replace the top half of the remote unit and replace the three (3) retaining screws removed previously.
9. Tighten two switch-retaining nuts loosened earlier.
10. Annotate aircraft logbook with cell replacement/inspection date information as applicable.
11. Install label on the switch body in a visible location with the cell replacement date annotated.

This completes the ELT Remote Switch Battery Cell Removal and Installation procedure.

ELT Remote Switch Installation

Perform this procedure to install the ELT remote switch.

1. Place aircraft master switch in the OFF position.
2. Locate and connect remote cable to switch jack. Refer to Figure 25-17.
3. Slide switch assembly into the avionics panel
4. Install four (4) 4-40 machine screws securing the remote switch to the avionics panel.
5. Verify remote switch ELT activity LED is off.
6. Prior to return to service, perform ELT 121.5 MHz Transmitter Test procedure as shown on page 36 of this chapter.

This completes the ELT Remote Switch Installation procedure.

ELT OPERATIONAL CHECK AND INSPECTION

Perform this procedure to perform an operational check and inspection.

NOTE

Procedure to follow provides inspection and maintenance required to comply with 100 hour and annual ELT inspection interval requirements, specified in Liberty maintenance manual Chapter 05 – Time Limits/Maintenance Checks/Inspection Intervals*.*

NOTE

Refer to the Section 60-11 121.5/243 MHz ELT Troubleshooting Guide shown on page 39 of this chapter for corrective actions.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position aircraft master switch to OFF
2. Install a tail stand underneath the tail section of the airplane.
3. Remove the cabin aft bulkhead access panel, by removing securing screw hardware.
4. Access the ELT location above the secondary battery on the aircraft port side.
5. Position the ELT manual control switch to OFF.
6. Inspect ELT unit and installation hardware for security, wear or physical damage.
7. Inspect remote switch wire harness and antenna cable for proper routing, insecure mounting or obvious defects.
8. Inspect antenna installation as shown in Figure 25-19 for condition, evidence of damage or other factors affecting proper operation.
9. Verify ELT antenna bonding between the mounting nut and the chassis is less than 0.5 ohms resistance (0.003 ohms is ideal).

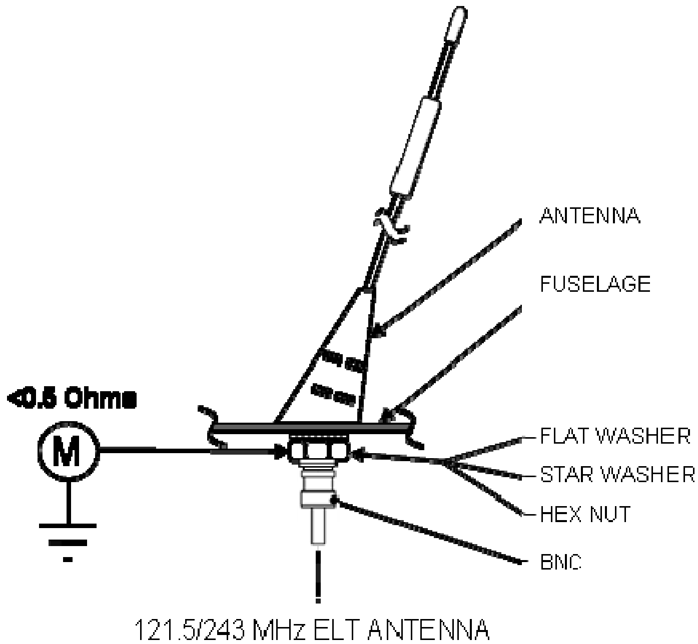


Figure 25-19 121.5/243 Mhz Antenna Inspection

10. Inspect the avionics panel mounted ELT remote switch face for condition, evidence of damage or other factors affecting proper ELT operation.
11. If performing the 100-hour inspection proceed with step 20 else proceed with step 12 to complete an annual inspection.

NOTE

The following steps will call for remove and install procedures. It is necessary to adhere to all of the inspection points that are in each of these to remain in compliance. These procedures and the 121.5/243 MHz ELT Troubleshoot Guide on page 39 provide corrective action to any issues found.

12. Perform the ELT Removal procedure shown on page 23 of this chapter.
13. Perform the ELT Battery Removal procedure shown on page 25 of this chapter.
14. Perform the ELT Battery Installation procedure shown on page 26 of this chapter.
15. Perform the ELT G-Switch Check procedure shown on page 35 of this chapter.
16. Perform the ELT Installation procedure shown on page 27 of this chapter.
17. Perform the ELT Remote Control Switch Removal procedure shown on page 28 of this chapter.

18. Perform the ELT Remote Switch Battery Cell Removal and Installation procedure shown on page 29 of this chapter.
19. Perform the ELT Remote Switch Installation procedure shown on page 31 of this chapter.
20. Perform the ELT 121.5 MHz Transmitter Test procedure shown on page 36 of this chapter.
21. Position the ELT manual control switch to ARM.
22. Close the cabin aft bulkhead access panel and secure with screws removed previously.
23. Remove tail stand.

This completes the ELT Operational Check and Inspection procedure.

ELT G-SWITCH CHECK

Perform this procedure to check the function of the G-switch in the ELT.

NOTE

The G-switch is a simple electromechanical switch intended to operate when an abrupt change of direction occurs. Test the switch by applying a force of 3 G's (Gravity) or more in a rearward direction. The following procedure will test the G-switch operation.

1. Perform the ELT Removal procedure shown on page 23 of this chapter to remove the ELT from the airplane.

CAUTION

Start and complete the following test between the top of the hour and five (5) minutes past the hour. This is the only time during each hour to perform this test without alerting monitoring agencies. Conclude this test after no more than three (3) sweeps of the transmitter (approx. one second) or monitoring agencies may consider the signal a valid distress call and initiate rescue action.

2. With the control switch in the arm position, rapidly move the ELT forward then reverse as shown in Figure 25-20 and verify LED status lamp is on steady.

Figure 25-20 G-switch Test Motion

3. Press and release the RESET switch.
4. Verify ELT Status LED is OFF.
5. Move the ON/OFF/ARM switch to OFF.
6. Perform the ELT Installation procedure shown on page 27 of this chapter.

This completes the ELT G-Switch Check procedure.

ELT 121.5 MHz Transmitter Test

Perform this procedure to test the ELT 121.5 MHz transmitter.

CAUTION

Start and complete the following test between the top of the hour and five (5) minutes past the hour. This is the only time during each hour to perform this test without alerting monitoring agencies. Conclude this test after no more than three (3) sweeps of the transmitter (approx. one second) or monitoring agencies may consider the signal a valid distress call and initiate rescue action.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Install a tail stand underneath the tail section of the airplane.
2. Remove aft cabin bulkhead access panel and locate the ELT installation above the secondary battery. Verify the ELT activity indicator is visible.
3. Turn on BAT/ALT MASTER and AVIONICS MASTER switches.
4. Tune COM1 receiver to 121.5 MHz and check receiver audio is feeding the headphones.
5. During the period from 00 to 05 minutes of each hour, press ELT remote switch "ON" button and monitor headphones for no more than three sweeps of ELT tone. Verify both the remote switch LED activity indicator and the ELT transmitter LED activity indicator are ON during sweeps.
6. Press ELT remote switch "RESET" button. Verify the tone ceases and both LED activity indicators extinguish.
7. Turn off BAT / ALT MASTER and AVIONICS MASTER switches.
8. Install the aft baggage compartment access panel.
9. Remove the tail support from the airplane.
10. Make required entry in aircraft records.

This completes the ELT 121.5 MHz Transmitter Test Procedure.

RESET ELT FROM REMOTE SWITCH

Perform this procedure to reset the ELT from the remote switch.

1. To reset the ELT, locate the remote switch located in the avionics panel below the transponder. See Figure 25-21 for the location of the reset button.
2. On the remote switch, press the reset button.
3. Verify the ELT activity led is off.

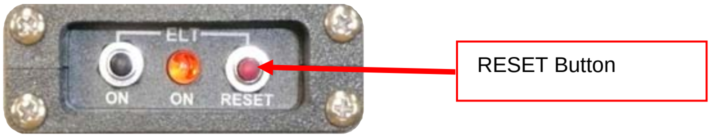


Figure 25-21 ELT Controls and Indicators

This completes the Reset ELT From Remote Switch procedure

RESET ELT FROM LOCAL CONTROLS

Perform this procedure to reset the ELT from the local controls.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Install a tail stand underneath the tail section of the airplane.
2. Open cabin aft access panel and locate the ELT mounted above the secondary battery
3. To reset the ELT, locate the RESET button as shown in Figure 25-22.

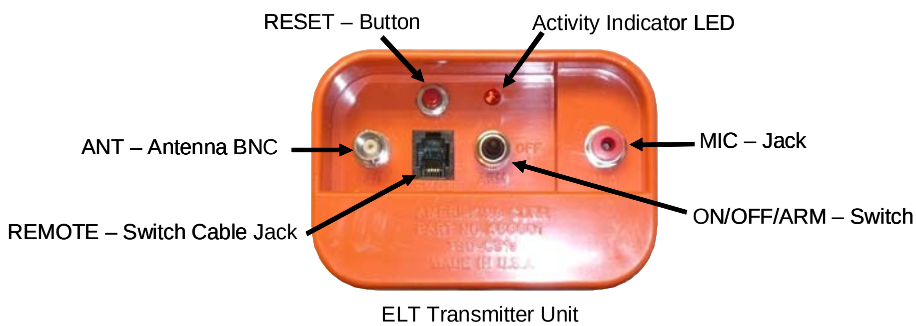


Figure 25-22 ELT RESET Button

4. On the ELT, press the RESET button.
5. Verify the ELT activity LED is OFF.
6. Verify the connection of the ELT antenna and remote switch cable to the ELT.
7. Place the ON/OFF/ARM switch in the ARM position.
8. Install the aft baggage compartment access panel.
9. Remove the tail support.

This completes the Reset ELT From Local Controls procedure.

Section 60-11 121.5/243 MHz ELT Troubleshooting Guide

Use Table 25-3 as a guide to troubleshoot issues with the ELT.

Symptom	Likely Cause	Action
Remote ELT switch LED always on	Wiring error or frayed wires shorting out pins in back of Remote Switch	Inspect wiring and repair as required. Inspect and repair, solder or crimp joints as required.
LED of the remote switch is not ON when either the ELT or the remote switch is turned on	The remote switch battery cell is weak.	Replace the remote switch battery cell
	Battery compartment corrosion	Clean battery compartment contacts Inspect for serviceability Replace battery
	The remote switch extension wire harness is not secured to the remote switch	Check the modular jack connector is fully inserted in the back of the remote switch
	The remote switch cable is not connected to the ELT	Check the connection
	Switch battery cell is not installed correctly.	Check switch cell installation polarity for correct orientation
Beacon tones not heard during test	ELT batteries are weak or discharged.	Replace batteries
	Battery compartment corrosion	Clean battery compartment contacts Inspect for serviceability Replace batteries
	Antenna damaged	Inspect antenna for damage and replace if damage is found. Inspect antenna cable for damage and replace if damage is found.
	Faulty ELT transmitter	Replace ELT
G-Switch does not trigger ELT transmitter	Faulty G switch	Replace ELT
	ELT batteries are weak or dead	Replace ELT batteries
	ELT manual control switch set to OFF	Place control on the ARM position and test the G switch.
Triggered ELT will not turn off	Faulty manual control switch	Remove ELT batteries to deactivate Replace ELT
ELT battery leaking	Faulty battery	Replace
Remote switch battery leaking	Faulty battery	Replace
Case Integrity Fault	ELT Dropped or ELT exposed to unknown forces or environment	Return the ELT to an ELT OEM depot for repair or replacement

Table 25-3 121.5/243 MHz ELT Troubleshoot Guide

Section 60-12 406/121.5 MHz Emergency Locator Transmitter (ELT)

The Liberty XL-2 can be equipped with an Artex model ME406 Emergency Locator Transmitter (ELT) like that shown in Figure 25-23. This unit is qualified under TSO-C126 for ELT's capable of transmitting identification data on 406 MHz. This unit is also qualified to transmit an emergency sweep tone on 121.5 MHz.

The unit is self contained, utilizing internal batteries for power. Table 25-4 shows a list of external interfaces and controls.

External Interfaces and Controls
Remote control switch D-subminiature (D-Sub) connection
Antenna cable BNC connection
Status indication LED
Control switch

Table 25-4 External Interfaces and Controls

ELT installation may be found in the rear fuselage as shown in Figure 25-24. A bracket and tray installation, shown in an exploded view in Figure 25-26, fits directly above the secondary battery, supporting the ELT, warning buzzer and interconnecting wiring. An exterior dual band whip antenna (Figure 25-24 and Figure 25-25) connects directly to the ELT by means of a single RG400 coax cable. Pilot control and monitoring of the ELT is provided by a self powered remote switch assembly located in the avionics panel directly below the transponder as shown in Figure 25-25

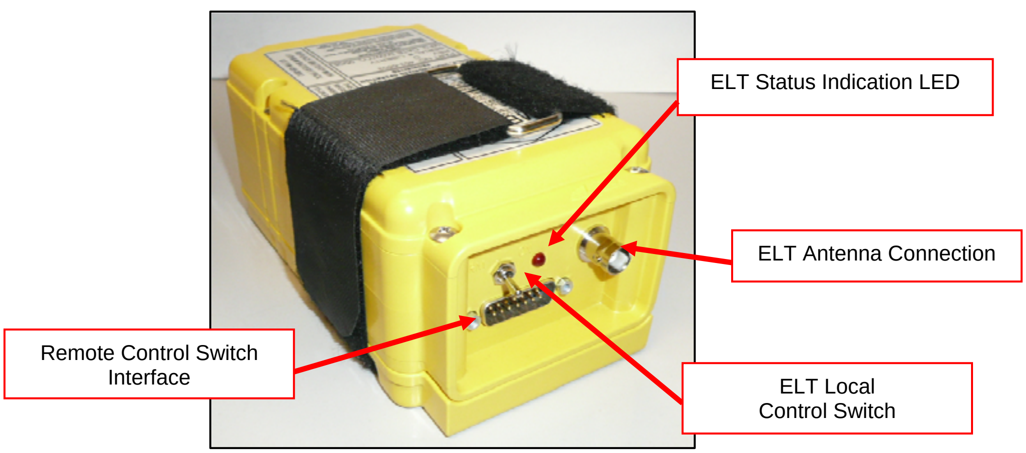


Figure 25-23 Emergency Locator Transmitter – 406/121.5 MHz TSO-C126

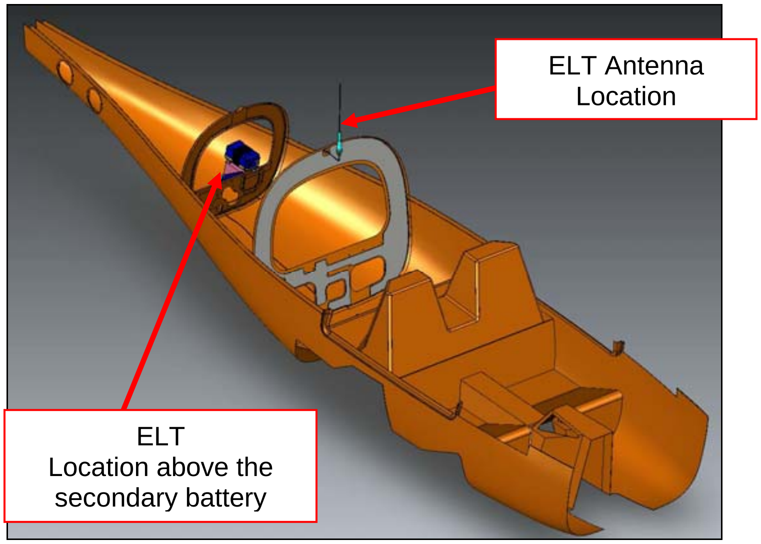


Figure 25-24 ELT Location

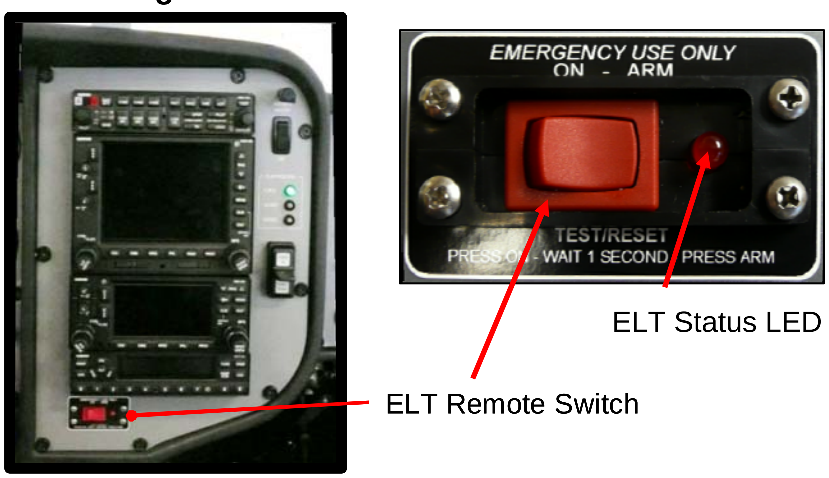


Figure 25-25 ELT Remote Switch Location

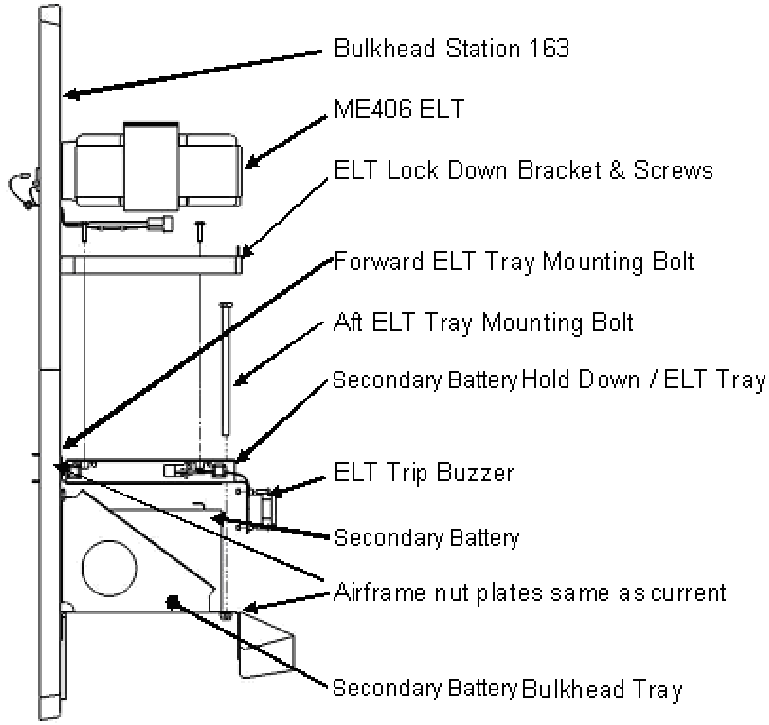


Figure 25-26 ELT Installation

ELT function is automatic. In the event of sudden impact or excessive deceleration, it will broadcast an internationally recognized distress signal on frequencies of 121.5 MHz and 406 MHz. Transmission of a standard sweep tone on 121.5 MHz continues until battery power is exhausted. In addition, for the first 24 hours of operation, the ELT transmits a 406 MHz signal containing an identification code at 50-second intervals. This transmission lasts 440ms and is received by the Cospas-Sarsat satellites for retransmission to search and rescue agencies. The transmitted identification code is referenced in a database (maintained by the national authority responsible for ELT registration) and is used to further identify the beacon and owner.

Section 60-13 Periodic Maintenance

In the United States, minimum annual maintenance requirements for an ELT are in paragraph (d) of 14 CFR Part 91.207. That paragraph states:

(d) Each emergency locator transmitter required by paragraph (a) of this section must be inspected within 12 calendar months after the last inspection for--
(1) Proper installation;
(2) Battery corrosion;
(3) Operation of the controls and crash sensor; and
(4) The presence of a sufficient signal radiated from its antenna.

In addition, to ensure continued reliability and airworthiness, inspect the ELT for damage and wear caused by age, exposure to the elements, vibration, or other causes. Inspections are to take place annually per FAR part 91.409. FAR 43, Appendix D(i). The appendix states, in part that each 100-hour or annual inspection shall inspect the following components of the ELT:

(1) (ELT unit and mount) for improper installation and insecure mounting.

(2) Wiring and conduits - for improper routing, insecure mounting, and obvious defects.

(3) Bonding and shielding - for improper installation and poor condition.

(4) Antenna, including trailing antenna-for poor condition, insecure mounting, and improper operation.

NOTE

All references to maintenance for the United States shall also apply to all ELT users outside of the US unless otherwise required by relevant national regulations.

Compliance with these requirements for the 406/121.5 MHz ELT is part Section 60-14, 406/121.5 MHz ELT Service Procedures, of this chapter. Refer to the ELT Operation Check And Inspection procedure on page 55 for additional information.

Section 60-14 406/121.5 MHz ELT Service Procedures

This section details the ELT Service Procedures. The following is a list of procedures:

Procedure	Page
ELT Removal	44
ELT Battery Removal	46
ELT Battery Installation	49
ELT Installation	50
ELT Remote Control Switch Removal	51
ELT Remote Switch Battery Cell Removal and Installation	52
ELT Remote Control Switch Installation	54
ELT Operation Check And Inspection	55
G-Switch check	58
ELT 121.5 MHz Transmitter Test	60
ELT 406 MHz Transmitter Test	61
Reset ELT from the Remote Switch	63
Reset ELT From Local Controls	64

ELT REMOVAL

Perform this procedure to remove the ELT from the airplane.

CAUTION

Take care when removing the ELT from its mounting. Removed improperly, the ELT can trigger a false emergency signal. Possible fines may be imposed if there is a false activation of an ELT.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position aircraft master switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Remove the cabin aft bulkhead access panel, by removing securing screw hardware.
4. Access the ELT, which is co-located with the secondary battery. See Figure 25-26 for the location of the ELT. Loosen the D-Subminiature connector's thumbscrews as shown in Figure 25-27and disconnect the D-Subminiature connector.

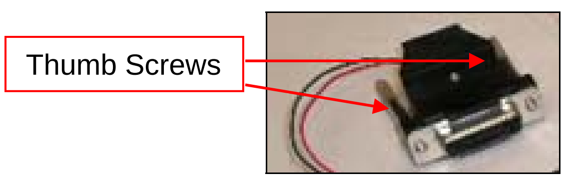


Figure 25-27 D-Sub Connector

5. Inspect D-Sub cable connector and remote switch extension cable attached for damage or evidence of corrosion.
6. Disconnect the BNC RF antenna connector.
7. Inspect BNC cable connector for damage or evidence or corrosion.
8. Loosen "Velcro" ELT restraining strap.
9. Remove ELT unit from its tray and the airplane.

NOTE

Evidence of damage requires inspection, repair, or replacement by a repair station qualified for TSO C126 ELT units. Do not install an ELT with evidence of damage.

10. Inspect ELT case and ELT Tray for cracks or other obvious damage.
11. Inspect all installation metal parts for signs of corrosion.

This completes the ELT Removal procedure.

ELT BATTERY REMOVAL

Perform this procedure to remove the ELT battery.

CAUTION

Take care when removing the ELT from its mounting. Removed improperly, the ELT can trigger a false emergency signal. Possible fines may be imposed if there is a false activation of an ELT.

CAUTION

The battery pack and ELT contain electro-statically sensitive parts. Take ESD precautions before handling. Damage to the exposed electronic parts may occur preventing correct operation of the ELT.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position aircraft master switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Remove ELT from the aircraft. Refer to the ELT Removal procedure on page 44 of this chapter.
4. Remove eight (8) securing screws from the battery-side cover as shown in Figure 25-28. Identify the battery pack by the embossed text "BATTERY ACCESS ON THIS SIDE".

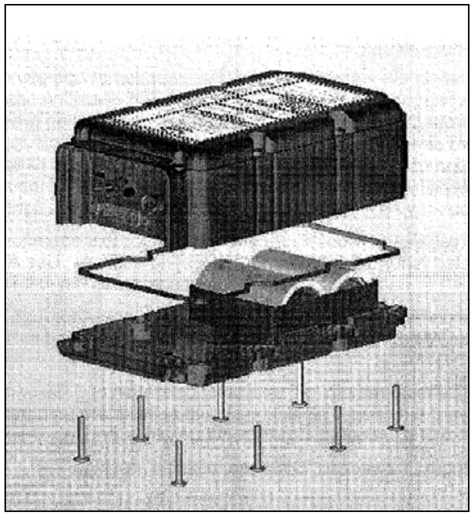


Figure 25-28 ELT Battery Replacement

5. Carefully lift the cover (battery pack) away from the ELT and unplug the flex-cable connected to the battery pack. Do not pull on the flexible portion of the cable – use the rigid section of the flex circuit at the connector as a handle.

6. Inspect the battery pack and ELT chassis. The battery cells, components, and connectors should be free of corrosion. Inspect flex-circuit for broken connections or damage. Check the battery housing is free of cracks or other visible damage.

NOTE

Evidence of damage requires inspection, repair or replacement by a qualified repair station. Do not install an ELT with evidence of damage. Contact the ELT's OEM for recommended repair or replace action.

7. Verify the battery expiration date. Refer to Chapter 05 – *Time Limits/Maintenance Checks/Inspection Intervals* for limitations. If the battery pack has not expired it may be reinstalled. If any of the conditions shown in Table 25-5 exist, install a new battery pack.

Condition Requiring Battery Replacement
After use in an emergency
When the transmitter has been in use for more than 1 cumulative hour
After an inadvertent activation of unknown duration
On or before the battery expiration date
On any evidence of corrosion or leakage of any cell

Table 25-5 Conditions Requiring Battery Replacement

8. Examine batteries and battery case contacts for any dirt or corrosion. Clean the contacts as necessary using Electrical Contact cleaner and a stiff brush.

CAUTION

The battery case contacts are nickel and gold plated spring steel. Abrasive material will remove this plating. If the contacts appear pitted with corrosion, replace the contact before returning the ELT to service.

This completes ELT Battery Removal procedure.

ELT BATTERY INSTALLATION

Perform this procedure to install the batteries into an ELT.

NOTE

This procedure is done out of the airplane.

1. Use only an Artex part number 452-6499 battery. Replacement batteries are obtained in kits containing all parts and labels that must be replaced with the battery. See Artex battery replacement kit 455-0012.

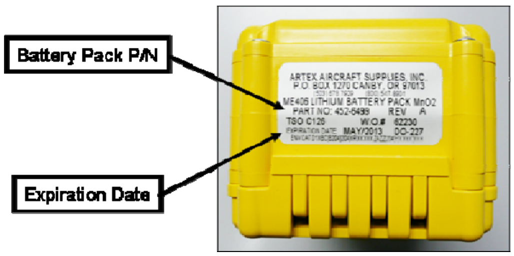


Figure 25-29 Battery Identification Placard

2. Lay the battery pack on the work surface.
3. Install a replacement seal that came with the replacement battery kit.
4. Leaving the battery assembly as it is, position the ELT over the battery pack with one hand and plug the flex-cable connector into the battery assembly using the other hand. Properly installed the cable should not be twisted and the connector should “click” into place.

NOTE

The connector is keyed to prevent incorrect installation

5. Mate the ELT assembly with the battery assembly taking care to assure the seal is positioned correctly in the process.
6. Replace eight (8) securing screws removed previously and torque 10-12 inch-lbs.
7. Enter battery replacement information in the aircraft log book.
8. If installed, remove the expired battery label from the ELT case.
9. Place the new expiration data label on the ELT case in a location that will be visible after installation.

This completes the ELT Battery Installation procedure.

ELT INSTALLATION

Perform this procedure to install the Emergency Locator Transmitter.

CAUTION

Take care when removing the ELT from its mounting. Removed improperly, the ELT can trigger a false emergency signal. Possible fines may be imposed if there is a false activation of an ELT.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position aircraft master switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Remove the cabin aft bulkhead access panel, by removing securing screw hardware.
4. Install the ELT into the mounting tray at an angle so that the locking ears at the end fit into the mounting tray locking slots. Fasten the Velcro strap around the ELT so that it is held in place.
5. Position ARM / ON switch to ARM
6. Connect the D-Sub connector adapter and tighten thumb screws.
7. Connect the BNC RF antenna connector.
8. Connect the connector from the remote switch wire harness to the D-Sub connector adapter.
9. Connect buzzer to D-Sub adapter lead connector. See Figure 25-26 for detail.
10. Install cabin aft bulkhead access panel by installing securing screw hardware removed previously.
11. Prior to return to service, perform the ELT 121.5 MHz Transmitter Test procedure as shown on page 60 of this chapter.
12. Prior to return to service, perform the ELT 406 MHz Transmitter Test procedure as shown on page 61 of this chapter.

This completes the ELT Installation procedure.

ELT REMOTE CONTROL SWITCH REMOVAL

Perform this procedure to remove the ELT remote control switch.

1. Position aircraft master switch to OFF.
2. Verify remote switch ELT activity LED is OFF.
3. Remove four (4) 4-40 machine screws securing the remote switch to the avionics panel. Refer to Figure 25-30 for screw locations.
4. Slide remote switch assembly out of the avionics panel.
5. Locate and disconnect switch extension cable connector (P76) and ground connector (P79).

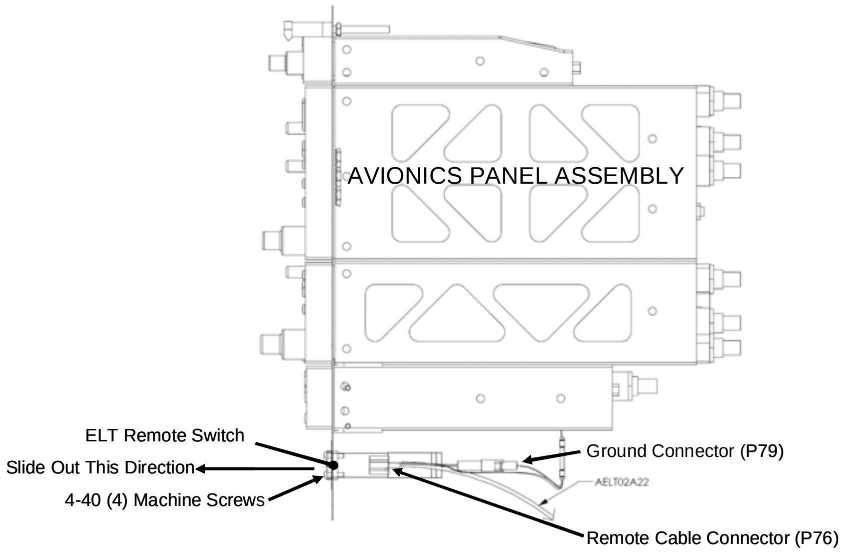


Figure 25-30 Remote Switch Removal

6. Inspect connector for the remote switch extension cable for damage or evidence of corrosion.
7. Inspect remote switch assembly for damage or evidence of corrosion.

This completes the ELT Remote Control Switch Removal procedure.

ELT REMOTE SWITCH BATTERY CELL REMOVAL AND INSTALLATION

Perform this procedure to remove and install the ELT remote switch battery cell.

1. Referring to Figure 25-31, remove four (4) retaining screws from the remote switch body to access the battery compartment.

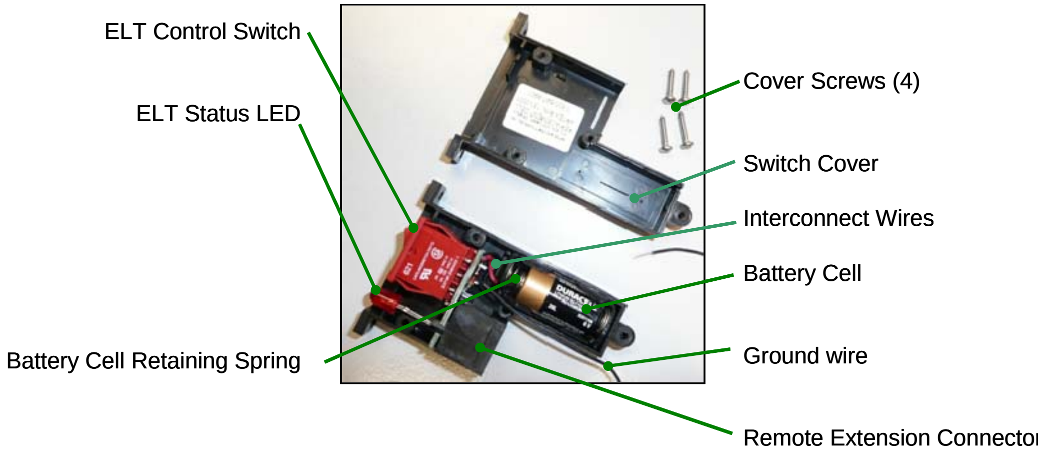


Figure 25-31 Remote Switch Battery Access

2. Remove protective cover opposite the product label.
3. Carefully remove the cell from the plastic holder.
4. Verify the battery expiration date. Refer to Chapter 05 – *Time Limits/Maintenance Checks/Inspection Intervals* for limitations. If the battery has not expired, it may be reinstalled. If any of the conditions shown in Table 25-6 exist, install a new battery pack.

Condition Requiring Battery Replacement
After use in an emergency
When the transmitter has been in use for more than 1 cumulative hour
After an inadvertent activation of unknown duration
On or before the battery expiration date
On any evidence of corrosion or leakage of any cell

Table 25-6 Conditions Requiring Battery Replacement

5. Inspect remote switch battery compartment for damage or evidence of corrosion. Clean battery contacts with a stiff brush and contact cleaner.
6. Carefully slide the negative end of the replacement cell inside the plastic holder following the polarity mentioned inside the holder and on the cell, by matching the positive (+) end of the cell to the red wire side of the holder.
7. Pull the built in spring away from the end of the cell and slide the cell inside the holder.
8. Release the spring and check that it seats properly against the cell end.

9. Gently route black and red wire leads from the plastic cell holder away from the leads of the rocker switch as necessary.
10. Position the ground wire (black) along the side of the plastic cell holder and away from the screw boss.
11. Install the mating cover to the assembly by matching the cutouts according to the switch components.
12. Check the 26 AWG ground wire is properly fed through the mating cover notch and is not pinched by the switch cover.
13. Secure the covers together using retaining screws previously removed.
14. Notate in the aircraft logbook the installation date of the cell.
15. Install label on the switch body in a visible location with the replacement date annotated.

This completes the ELT Remote Switch Battery Cell procedure.

ELT Remote Control Switch Installation

Perform this procedure to install the ELT remote control switch.

1. Position aircraft master switch to OFF.
2. Locate and connect switch extension cable connector (P76) and ground connector (P79). Refer to Figure 25-30 for the location both connectors.
3. Slide switch assembly into the avionics panel.
4. Secure the switch assembly with four, 4-40 machine screws.
5. Prior to return to service, perform the ELT 121.5 MHz Transmitter Test procedure as shown on page 60 of this chapter.
6. Prior to return to service, perform the ELT 406 MHz Transmitter Test procedure as shown on page 61 of this chapter.

This completes the ELT Remote Control Switch Installation procedure.

ELT OPERATION CHECK AND INSPECTION

Perform this procedure to perform an operational check and inspection of the ELT.

NOTE

Procedures to follow provide inspection and maintenance required to comply with 100 hour and annual ELT inspection requirements, specified in Liberty maintenance manual Chapter 05 – Time Limits/Maintenance Checks/Inspection Intervals*.*

NOTE

Refer to the 406/121.5 MHz ELT Troubleshooting Guide Troubleshooting Guide for corrective actions.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Place aircraft master switch in the OFF position.
2. Install a tail stand and verify aircraft is stable.
3. Remove the cabin aft bulkhead access panel by removing securing screw hardware.
4. Access the ELT location above the secondary battery on the aircraft port side.
5. Inspect ELT unit and installation hardware for security, wear or physical damage.
6. Inspect remote switch wire harness and antenna cable for damage and proper connection.
7. Inspect antenna installation for evidence of damage and proper installation
8. Verify ELT antenna continuity between the mounting hex nut and the chassis is < 0.5 ohms resistance (0.003 ohms is ideal). Refer to Figure 25-32

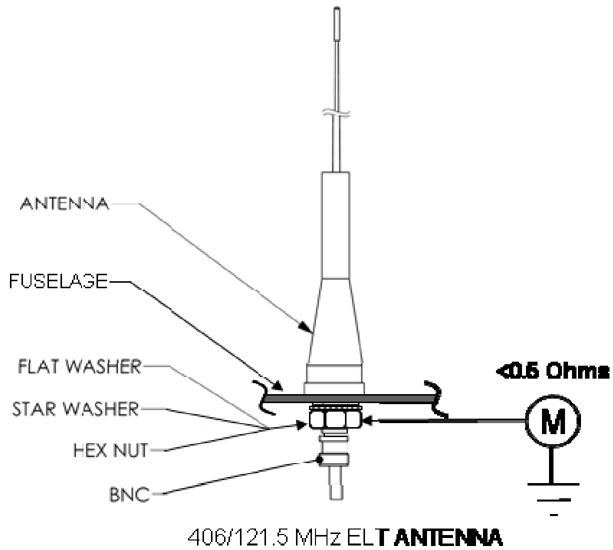


Figure 25-32 406/121.5 ELT Antenna Inspection

9. Inspect the avionics panel mounted ELT remote switch face for condition, evidence or damage or other factors affecting proper ELT operation.

10. If a 100 hour inspection is being performed proceed with step 19 else proceed with step 11 to complete an annual inspection.

NOTE

The following steps will call for remove and install procedures. It is necessary to adhere to all of the inspection points that are in each of these to remain in compliance. These procedures and the 406/121.5 MHz ELT Troubleshooting Guide on page 66 provide corrective action to any issues found.

11. Perform the ELT Removal procedure shown on page 44 of this chapter.

12. Perform the ELT Battery Removal procedure shown on page 46 of this chapter.

13. Perform the ELT Battery Installation procedure shown on page 49 of this chapter.

14. Perform the G-Switch check procedure shown on page 58 of this chapter.

15. Perform the ELT Installation procedure shown on page 50 of this chapter.

16. Perform the ELT Remote Control Switch Removal procedure shown on page 51 of this chapter.

17. Perform the ELT Remote Switch Battery Cell Removal and Installation procedure shown on page 52 of this chapter.

18. Perform the ELT Remote Control Switch Installation procedure shown on page 54 of this chapter.
19. Verify current ELT registration. As required, prior to returning the airplane to service, perform the ELT Registration procedure shown on page 65 of this chapter.
20. Prior to return to service, perform the ELT 121.5 MHz Transmitter Test procedure as shown on page 60 of this chapter.
21. Prior to return to service, perform the ELT 406 MHz Transmitter Test procedure as shown on page 61 of this chapter.
22. Position the ELT manual control switch to ARM
23. Close the cabin aft bulkhead access panel and secure with screws removed previously.
24. Remove tail stand.

This completes the ELT Operation Check And Inspection procedure.

G-SWITCH CHECK

Perform this procedure to check the G-switch.

NOTE

The G-switch is a simple electromechanical switch intended to operate when an abrupt change of direction occurs. Test the switch by applying a force of 3 G's (Gravity) or more in a rearward direction. The following procedure will test the G-switch operation.

1. Perform the ELT Removal procedure shown on page 44 of this chapter.
2. Using a 15-pin D-sub connector fabricate a shorting plug connecting pins 5 and 12 together.
3. Connect shorting plug to the ELT remote switch interface connector shown in Figure 25-23.

CAUTION

Start and complete the following test between the top of the hour and five (5) minutes past the hour. This is the only time during each hour to perform this test without alerting monitoring agencies. Conclude this test after no more than three (3) sweeps of the transmitter (approx. one second) or monitoring agencies may consider the signal a valid distress call and initiate rescue action.

4. With the control switch in the ARM position rapidly move the ELT forward then reverse as shown in Figure 25-33 and verify LED status lamp is flashing.

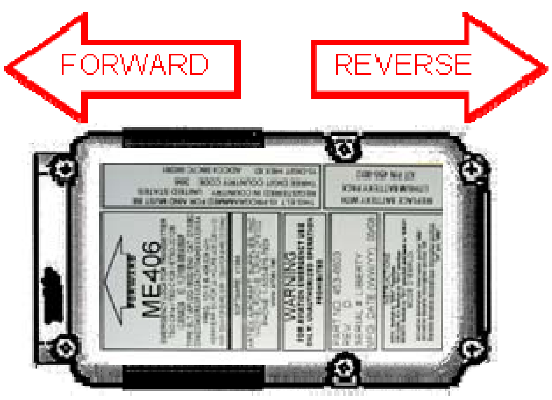


Figure 25-33 G-Switch Test Motion

5. Deactivate the ELT from the local control switch by moving the ON/ARM switch to – ON then to ARM.

6. Verify ELT Status LED is OFF.
7. Remove shorting plug from remote control switch interface connector.
8. Perform the ELT Installation procedure on page 50 of this chapter.

This completes the G-Switch check procedure.

ELT 121.5 MHz Transmitter Test

Perform this procedure to test the 121.5MHz transmitter.

CAUTION

Start and complete the following test between the top of the hour and five (5) minutes past the hour. This is the only time during each hour to perform this test without alerting monitoring agencies. Conclude this test after no more than three (3) sweeps of the transmitter (approx. one second) or monitoring agencies may consider the signal a valid distress call and initiate rescue action.

NOTE

Complete test steps 3 through 6 in rapid succession to prevent more than three sweeps of the ELT transmitter.

1. Turn on the BAT and ALT master switches, and the Avionics master switch.
2. Tune the Com 1 receiver to 121.5 MHz. Check that receiver audio is feeding the headphones.
3. On the ELT remote switch located in the avionics panel, press ON

Figure 25-34 Remote Switch – 121.5 MHz Transmitter Test

4. Verify the remote switch red LED illuminates and the ELT buzzer sounds.
5. Verify the transmitter sweep is heard on the Com 1 radio.
6. After the third sweep, on the ELT remote switch located on the avionics panel, press ARM
7. Verify the alarm and LED gives only one flash. If more than one LED flash is seen record the number of flashes and refer to Table 25-7 in the ELT Troubleshooting Guide procedure on page 67 of this chapter for corrective action.

This completes the ELT 121.5 MHz Transmitter Test procedure.

ELT 406 MHz Transmitter Test

Perform this procedure to test the ELT 406 MHz transmitter.

NOTE

The following test requires use of a suitable test set capable of reading and decoding 406 MHz ELT transmitted signal messages. Artex test set part number 453-1000 or Aeroflex model number IFR 4000 NAV/COM test set are recommended for this purpose

CAUTION

Start and complete the following test between the top of the hour and five (5) minutes past the hour. This is the only time during each hour to perform this test without alerting monitoring agencies. Conclude this test after no more than three (3) sweeps of the transmitter (approx. one second) or monitoring agencies may consider the signal a valid distress call and initiate rescue action.

NOTE

Test steps 3 through 7 must be completed in rapid succession to prevent more than three sweeps of the ELT transmitter.

1. Turn on the BAT and ALT master switches, and the Avionics master switch.
2. Tune the Com 1 receiver to 121.5 MHz. Check that receiver audio is feeding the headphones.
3. Configure test set to receive a 406 MHz ELT transmission. Refer to test set OEM manual for specific set up instructions.
4. On the ELT remote switch located in the avionics panel, press ON

Figure 25-35 Remote Switch – 406 MHz Transmitter Test

5. Verify the remote switch red LED illuminates and the ELT buzzer sounds.
6. Verify the transmitter sweep is heard on the Com 1 receiver.

7. After the third sweep, on the ELT remote switch located on the avionics panel, press ARM
8. Verify the alarm and LED gives only one flash. If more than one LED flash is seen record the number of flashes and refer to troubleshooting guide section for corrective action.
9. On the test set, verify message information matches the ELT registration information and 15 digit Hex ID code shown on the ELT data plate:

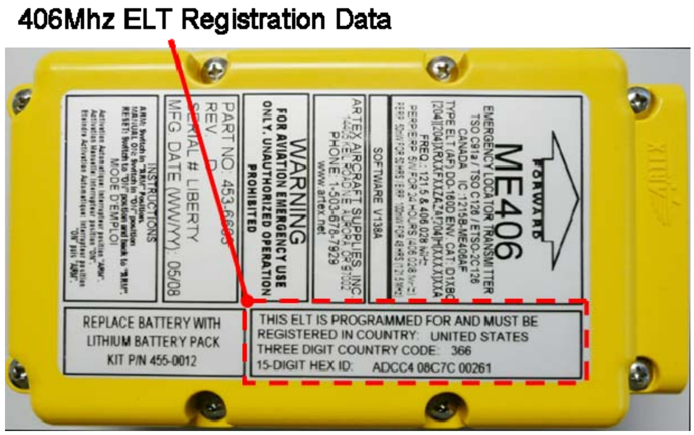


Figure 25-36 The 406 MHz ELT Registration Data

10. In the event read data does not match the data plate remove the unit from service and route to a facility qualified to test and program C126 compliant ELT's.

This completes the ELT 406 MHz Transmitter Test procedure.

RESET ELT FROM THE REMOTE SWITCH

Perform this procedure to reset the ELT from the remote switch.

1. To “RESET” the ELT, locate the remote switch located in the avionics panel below the transponder. See Figure 25-37.
2. On the remote switch, press and release – ON.
3. Wait one (1) second.
4. Press and release – ARM.
5. Verify the ELT activity LED is OFF and ELT buzzer is silent.

Figure 25-37 Remote Switch – ELT Reset

This completes the Reset ELT from the Remote Switch procedure.

RESET ELT FROM LOCAL CONTROLS

Perform this procedure to reset the ELT from the local controls.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Place aircraft master switch in the OFF position.
2. Install a tail stand and verify aircraft is stable.
3. Remove the cabin aft bulkhead access panel by removing securing screw hardware.
4. To "RESET" the ELT, locate the RESET button as shown in Figure 25-38.
5. On the ELT, move the ON/ARM switch to the ON position then back to the ARM position.
6. Verify the ELT activity LED is OFF and ELT buzzer is silent.
7. Verify ELT antenna and remote switch cable with its adapter are connected to the ELT.

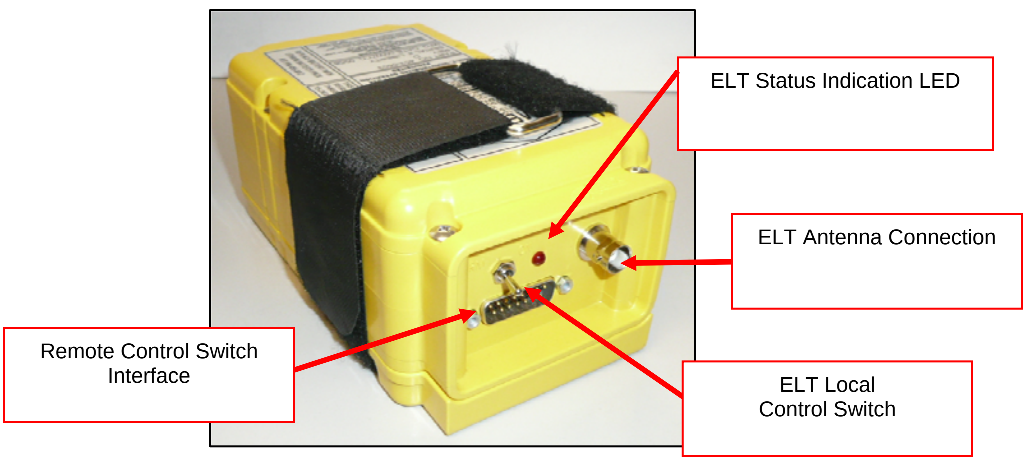


Figure 25-38 ELT Reset

8. Close cabin aft access panel.

This completes the Reset ELT From Local Controls procedure.

Section 60-15 ELT Registration

All ELTs operating on 406 MHz must be registered. In the United States, the National Oceanic and Atmospheric Administration (NOAA) is the agency that administrates beacon registration. Upon registration, NOAA will issue a “Proof of Registration” label that must be affixed to the ELT registered in a visible location.

Figure 25-39 Proof of Registration Label Example

To register a 406 MHz ELT in the U.S., contact NOAA at:

NOAA/NESDIS

SARSAT Operations Division

Code E/SP3

Federal Building 4

Washington, DC 20233

HTTP://www.sarsat.noaa.gov

NOTE

For ELTs that will have a country of registration other than the USA, contact the appropriate Civil Aviation Authority in that country for guidelines and documentation needed to assure proper registration. Always follow the national procedures.

Section 60-16 406/121.5 MHz ELT Troubleshooting Guide

Use Table 25-7 as a guide to troubleshoot issues with the ELT.

SYMPTOM	LIKELY CAUSE	ACTION
1 Flash after performing self-test	Indicates system is operational and that no error conditions were found	None
3 Flashes after performing self-test	Bad RF load detected. Possible open or short on the antenna output or cable	Check that the RF cable is connected and in good condition Perform the antenna cable center conductor and shield continuity test. Check for shorts or open conditions Check for intermittent connection in the RF cable Check antenna for open, short and ground bonding
4 Flashes after performing self-test	Low power detected. Occurs if output power is below 33dBm (2 watts) for the 406 MHz signal or 17dBm (50mw) for the 121.5MHz. May also indicate the 406 signal is off frequency. Also can indicate ELT battery voltage <5.6 Vdc.	Replace ELT battery Return unit to OEM depot for repair or replacement
5 Flashes after performing self-test	Indicates the ELT has not been programmed. Does not indicate erroneous or corrupted programming	Program ELT per OEM specification
6 Flashes after performing self-test	Indicates G-switch loop between pins 5 and 12 of the D-Sub connector is not installed. ELT will not activate in a crash	Measure the resistance between pins 5 and 12 of the D-sub connector. Replace the adapter if reading is not less than 1 ohm.
7 Flashes after performing self-test.	Indicates ELT battery has accumulated operating time in excess of 1 hour.	Replace the battery
Remote ELT switch LED always on	Wiring error or frayed wires shorting out pins in back of Remote Switch	Inspect wiring and repair as required. Inspect and repair, solder or crimp joints as required.
LED of the remote switch is not flashing when either the ELT or the remote switch is turned on	The remote switch battery cell is below 3.3 volts.	Replace the remote switch battery cell
	The remote switch extension wire harness is not secured to the remote switch	Check the modular jack connector is fully inserted in the back of the remote switch
	The remote switch ground wire not connected to the module interface ground	Check the P79 ground wire connection
	The module interface is not connected to the ELT	Check the connection
	Switch battery cell is not	Check switch cell installation polarity

SYMPTOM	LIKELY CAUSE	ACTION
	installed correctly.	for correct orientation
Buzzer fails to sound on ELT activation	Buzzer failed	Replace buzzer
	Buzzer disconnected	Check ELT tray connections to the buzzer. Connect as required.
ELT battery leaking	Faulty battery	Replace
Remote switch battery leaking	Faulty battery	Replace
Case Integrity Fault	ELT Dropped or ELT exposed to unknown forces or environment	Return the ELT to an ELT OEM depot for repair or replacement

Table 25-7 ELT Troubleshooting Guide

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CHAPTER 27

FLIGHT CONTROLS

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Table of Contents

SECTION 27-00 GENERAL 5
SECTION 00-01 PUSH ROD LENGTHS 5
SECTION 27-10 AILERON 7
SECTION 10-01 AILERON PROCEDURES 8
AILERON REMOVAL 9
AILERON INSTALLATION 11
AILERON WING BELL-CRANK REMOVAL 15
AILERON WING BELL-CRANK INSTALLATION 16
AILERON OUTBOARD WING PUSHROD REMOVAL 18
AILERON OUTBOARD WING PUSHROD INSTALLATION 19
INBOARD WING PUSHROD REMOVAL 20
INBOARD WING PUSHROD INSTALLATION 21
WING-MOUNTED QUICK-DISCONNECT DEVICE REMOVAL 22
WING-MOUNTED QUICK-DISCONNECT DEVICE INSTALLATION 23
INBOARD QUICK-DISCONNECT DEVICE REMOVAL 25
INBOARD QUICK-DISCONNECT DEVICE INSTALLATION 26
INBOARD AILERON PUSH ROD REMOVAL 27
INBOARD AILERON PUSHROD INSTALLATION 28
AILERON TORQUE TUBE REMOVAL 30
AILERON TORQUE TUBE INSTALLATION 32
SECTION 10-02 AILERON INSPECTION 33
SECTION 10-03 TROUBLESHOOTING 33
SECTION 27-20 RUDDER 35
SECTION 20-01 RUDDER PROCEDURE 37
RUDDER REMOVAL 38
RUDDER INSTALLATION 40
RUDDER MID-FUSELAGE IDLER-PLATE REMOVAL 41
RUDDER MID-FUSELAGE IDLER-PLATE INSTALLATION 43
FORWARD RUDDER BELL-CRANK REMOVAL 45
FORWARD RUDDER BELL-CRANK INSTALLATION 47
RUDDER INPUT PUSHROD REMOVAL 49
RUDDER INPUT PUSHROD INSTALLATION 50
RUDDER PEDAL ASSEMBLY REMOVAL 52
RUDDER PEDAL ASSEMBLY INSTALLATION 55
EXTENDING THE CLOSEOUT HOLE FOR THE RUDDER 57
SECTION 20-02 RUDDER INSPECTION 59
SECTION 20-03 TROUBLESHOOTING 60

SECTION 27-30 STABILATOR AND TAB 61
SECTION 30-01 STABILATOR SYSTEM PROCEDURES 62
STABILATOR REMOVAL 63
STABILATOR INSTALLATION 64
MID-FUSELAGE PITCH CONTROL IDLER-PLATE REMOVAL 66
MID-FUSELAGE PITCH CONTROL IDLER-PLATE INSTALLATION 69
ANTI-SERVO/TRIM TAB REMOVAL 70
ANTI-SERVO/TRIM TAB INSTALLATION 71
PITCH TRIM ACTUATOR REMOVAL 72
PITCH TRIM ACTUATOR INSTALLATION 73
SECTION 30-02 STABILATOR AND TAB INSPECTION 75
SECTION 30-03 TROUBLESHOOTING 75
SECTION 27-40 CONTROL YOKE 77
SECTION 40-01 CONTROL YOKE PROCEDURES 77
CONTROL YOKE BOOT REMOVAL 78
CONTROL YOKE BOOT INSTALLATION 79
STARBOARD (CO-PILOT) CONTROL STICK REMOVAL 80
STARBOARD (CO-PILOT) CONTROL STICK INSTALLATION 81
PUSH-TO-TALK SWITCH REMOVAL 82
PUSH-TO-TALK SWITCH INSTALLATION 83
CONTROL YOKE ASSEMBLY REMOVAL 84
CONTROL YOKE ASSEMBLY INSTALLATION 86
SECTION 27-50 FLAPS 87
SECTION 50-01 FLAP SYSTEM PROCEDURES 88
FLAP REMOVAL 89
FLAP INSTALLATION 90
FLAP ACTUATOR REMOVAL 93
FLAP ACTUATOR INSTALLATION 94
FLAP POSITION SENSOR REMOVAL 95
FLAP POSITION SENSOR INSTALLATION 96
FLAP INSPECTION 97
SECTION 50-02 FLAP REPAIR 100
SECTION 50-03 FLAP DAMAGE 100
SECTION 50-04 FLAP TROUBLESHOOTING 102
SECTION 27-90 RIGGING PROCEDURES 103
SECTION 90-01 PRELIMINARY INSPECTION AND SETUP 103
VERIFICATION OF FLYING SURFACES RELATIVE MOVEMENT & RESPONSE 104
FLAPS RIGGING 107
AILERON RIGGING 109
HORIZONTAL STABILIZERS & TRIM TABS 115
RUDDER RIGGING 120

Section 27-00 General

The flight controls of the airplane include conventional aileron, rudder and stabilator with anti-servo/trim tab.

All flight controls are operated by pushrods, with dual controls operating the aileron and the stabilator, and dual rudder pedals operating the rudder. The left (pilot) and right (co-pilot) rudder pedals are individually adjustable to accommodate different pilot heights and leg lengths.

An anti-servo/trim tab is installed on both port and starboard stabilator. The anti-servo/trim tab attitude is electrically adjustable in flight (with respect to the stabilator) by the pilot, thus providing a pitch trim function. Airplane pitch trim attitude is indicated on the center console by a “Nose Up”, “Takeoff,” and “Nose Down” indicator.

Ground adjustable fixed trim tabs are installed on the port aileron and on the rudder.

A single common actuator controlled by a three-position switch on the avionics panel electrically operates a conventional flap on each wing. The flap position is indicated on the control panel as:

- 0º, 20º or 30º for airplanes certified at 1653 lbs.
- 0º, 10º or 30º for airplanes certified at 1750 lbs.

Section 00-01 Push Rod Lengths

The airplane uses different length push rods throughout the airplane to control the aileron, rudder, and stabilizer. The finished length of the push rod is set by manufacturing during the assembly of the airplane. Push rods that are sent to the field (as service spares) are set to an initial length. Table 27-1 shows the initial settings length that push rods sent from manufacturing to replace damaged push rods. The final length is determine by either matching the length of the existing push rod (in the airplane) or by adjustment during a rigging procedure. The distance shown in Table 27-1 is measured between the centers of the two end bearings, see Figure 27-1.

Location	Part Number	Length inches
Aileron Ladder	135A-45-233	21.5
Aileron Push Rod	135A-45-235	81.2
Aileron Push Rod to Aileron	135A-45-237	16.4
Rudder Push Rod Input	135A-45-243	9.2
Rudder Push Rod Forward	135A-45-245	58.9
Rudder Push Rod Intermediate	135A-45-247	83.0
Rudder Push Rod Aft	135A-45-249	22.5
Horizontal Stabilizer Push Rod Forward	135A-45-251	48.4
Horizontal Stabilizer Push Rod Aft	135A-45-253	84.0

Table 27-1 Torque Tube Initial Length

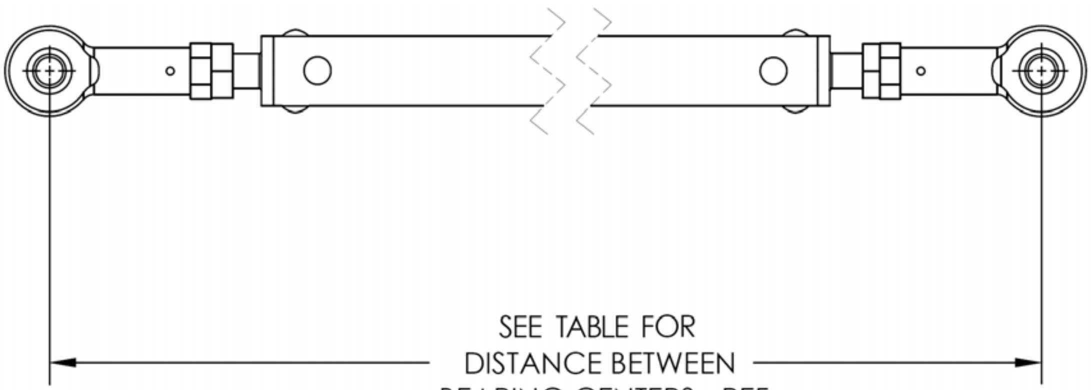

Figure 27-1 Torque Tube Distance

Section 27-10 Aileron

The airplane aileron is of conventional aluminum construction and is hinged to the lower wing surface by conventional piano hinges. The port aileron is equipped with a ground adjustable control surface trim tab, located at the outboard rear of the aileron. The trim tab is riveted to the underside of the aileron. Each aileron incorporates two mass balance weights. See Figure 27-2 for details of the aileron drive system.

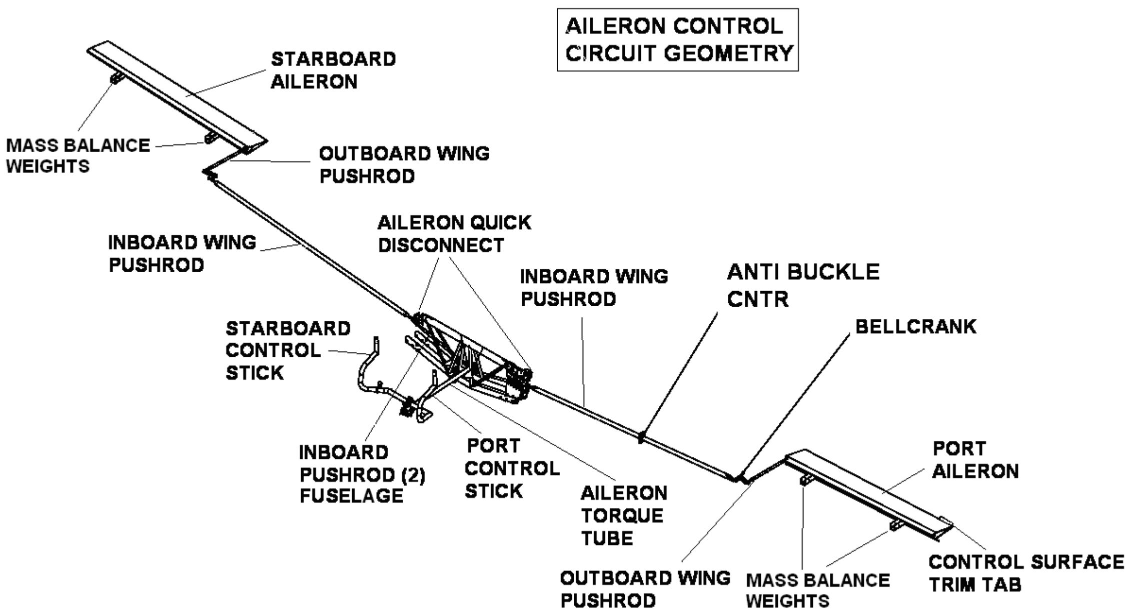

Figure 27-2 Aileron Control Circuit Geometry

Both (pilot and copilot) control sticks are attached to a common yoke tube pivoted on a bearing on the centerline of the airplane (through the main chassis). Left/right movement of the control sticks tilts this yoke tube in the corresponding direction. The common yoke is fastened to a torque tube running aft between the pilot and copilot; tilting of the control stick yoke rotates this torque tube in a clockwise or counterclockwise direction.

At its aft end, the torque tube actuates two pushrods leading to aileron quick-connect assemblies mounted on the chassis wing box assembly. When either wing is assembled to the fuselage, a similar quick-connect device in the wing root mates with the corresponding unit in the center section to complete the quick-connect assembly and transmit aileron movement to the linking components in the wing.

In the wing, a single pushrod passes from the quick-connect assembly at the root through a push rod anti-buckling support approximately midway along its length to a bell-crank located forward and just inboard of the aileron root rib. Another pushrod transmits motion from the bell-crank to the inboard end of the aileron.

Access to bearings in the chassis center section is gained by removing the fuselage belly panel (see Chapter 53 - *Fuselage*). Access to both the fuselage and wing components of the aileron quick-connect fittings is gained by removal of the wing(s) (see Chapter 57 - *Wings*). Access to the aileron bell-crank in each wing is gained via removable access panels in the wing lower surface.

Section 10-01 Aileron Procedures

This section contains the procedures for the ailerons systems. These procedures include the following:

Procedure	Page
Aileron Removal	9
Aileron Installation	11
Aileron Wing Bell-Crank Removal	15
Aileron Wing Bell-Crank Installation	16
Aileron Outboard Wing Pushrod Removal	18
Aileron Outboard Wing Pushrod Installation	19
Inboard Wing Pushrod Removal	20
Inboard Wing Pushrod Installation	21
Wing-Mounted Quick-Disconnect Device Removal	22
Wing-Mounted Quick-Disconnect Device Installation	23
Inboard Quick-Disconnect Device Removal	25
Inboard Quick-Disconnect Device Installation	26
Inboard Aileron Push Rod Removal	27
Inboard Aileron Pushrod Installation	28
Aileron Torque Tube Removal	30
Aileron Torque Tube Installation	32

AILERON REMOVAL

Perform this procedure to remove the aileron.

1. Fully extend wing flap.
2. Temporarily mark (using masking tape) on the aileron and on the aileron hinge, alignment marks such that when installing the aileron, the aileron will be in the same position, see Figure 27-3. If installing a different aileron, contact Liberty Aerospace, Inc. Customer Service.

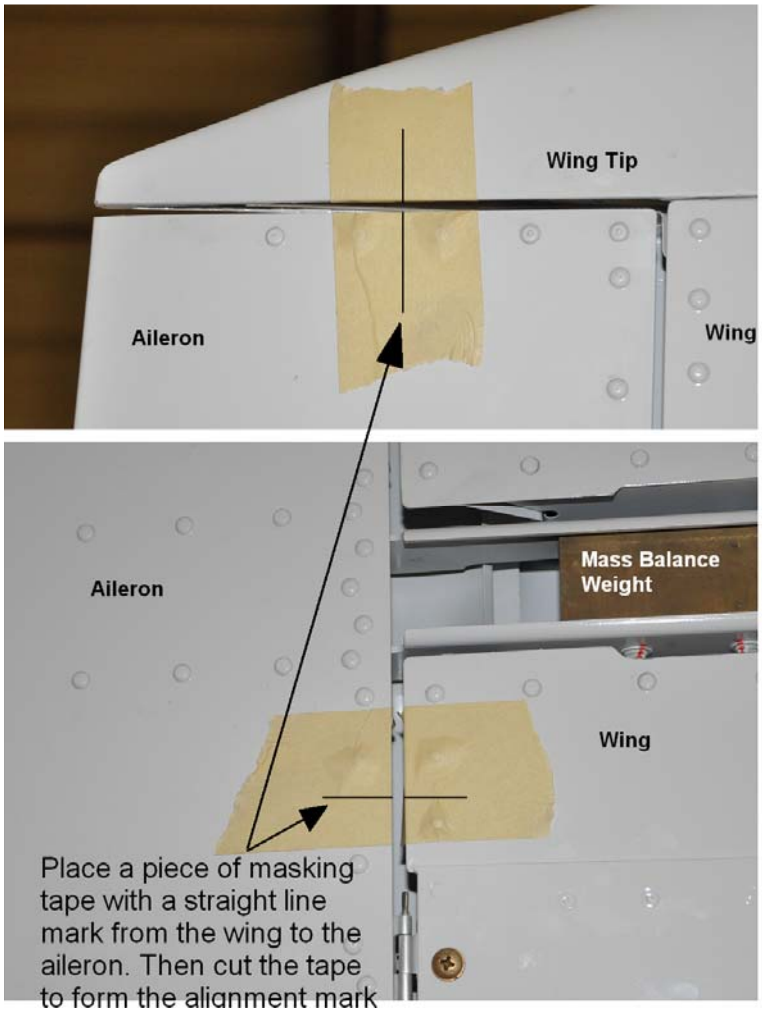

Figure 27-3 Alignment Marks Using Masking Tape

3. Remove split pin (1), nut (2), washer (3) and bolt (4) that attach pushrod to inboard end of aileron (nearest to flap) to disconnect pushrod. See Figure 27-4. Separate pushrod from aileron.
4. Remove screws securing aileron hinges to wing lower surface. Hinges remain attached to aileron.

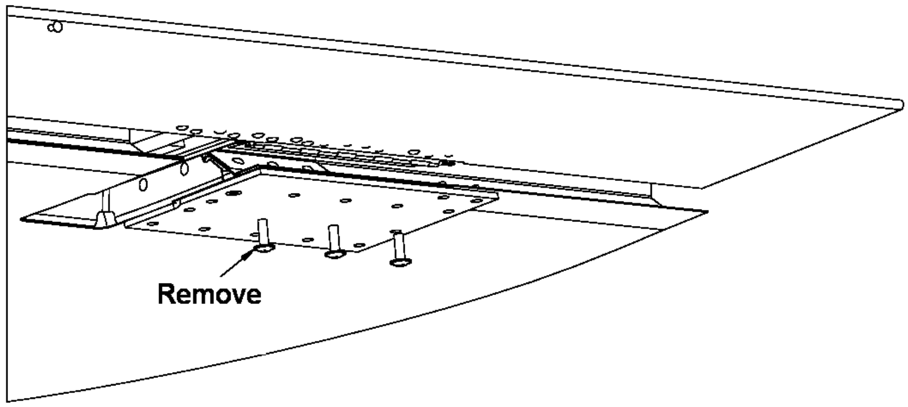

Figure 27-4 Screws Removal on Aileron Hinge

5. While supporting pushrod, remove aileron.

WARNING

AILERONS ARE MASS-BALANCED CONTROL SURFACES. IF MASS BALANCE WEIGHTS ARE REMOVED, THEY MUST BE MARKED FOR RE-INSTALLATION IN PROPER POSITION. ANY REPAIRS MADE TO AILERONS (INCLUDING REPAINTING) REQUIRE THAT BALANCE OF THE CONTROL SURFACE BE CHECKED BEFORE IT IS INSTALLED. CHECK THE BALANCE OF THE AILERON IN ACCORDANCE WITH CHAPTER 51 – STANDARD PRACTICES STRUCTURES.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

AILERON INSTALLATION

Perform this procedure to install the aileron.

WARNING

AILERONS ARE MASS-BALANCED CONTROL SURFACES. IF MASS BALANCE WEIGHTS ARE REMOVED, THEY MUST BE MARKED FOR RE-INSTALLATION IN PROPER POSITION. ANY REPAIRS MADE TO AILERONS (INCLUDING REPAINTING) REQUIRE THAT BALANCE OF THE CONTROL SURFACE BE CHECKED BEFORE IT IS INSTALLED. CHECK THE BALANCE OF THE AILERON IN ACCORDANCE WITH CHAPTER 51 – STANDARD PRACTICES STRUCTURE.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

1. Fully extend wing flap.
2. Align the aileron to the wing, using the alignment marks made in step 2 of the Aileron Removal procedure on page 9 of this chapter
3. Loosely install all of the screws that secure aileron hinges to wing lower surface.
4. Tighten one screw at a time, rechecking the alignment with each screw tighten. Make sure the push rod end fitting location is unchanged.
5. Torque each screw in accordance with Chapter 20 - Standard Practices for torque requirements.
6. Refer to Figure 27-5 during the step. Install bolt (4), washer (3), and nut (2) securing aileron pushrod end to inboard end of aileron. Torque to a value of at least 50 in-lbs. Continue to torque to nearest castellated slot for split pin (1). Do not exceed 70 in-lbs. of torque. Install a new split pin. Refer to Chapter 20 – *Standard Practices*.

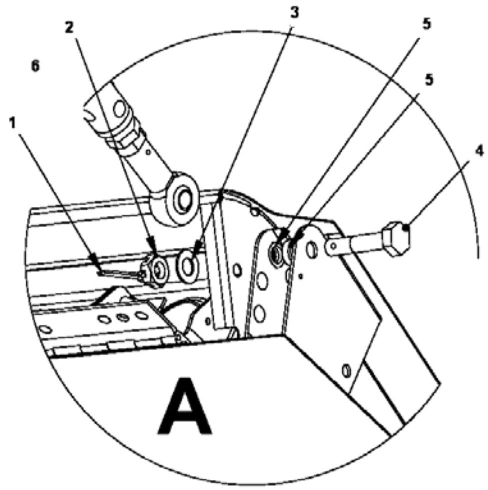

Figure 27-5 Details of the Pushrod Assembly

7. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
8. Check clearances between the top of the aileron skin and the wing trailing edge skin (T/E) as follows per Table 27-2 and Figure 27-6.

Clearance Description (Top Aileron Skin to Wing T/E)	Clearance
Aileron full up position (A)	0.080”
Aileron neutral position (B)	0.090”
Aileron full down position (C)	0.080”

Table 27-2 Table of Measurements for the Aileron

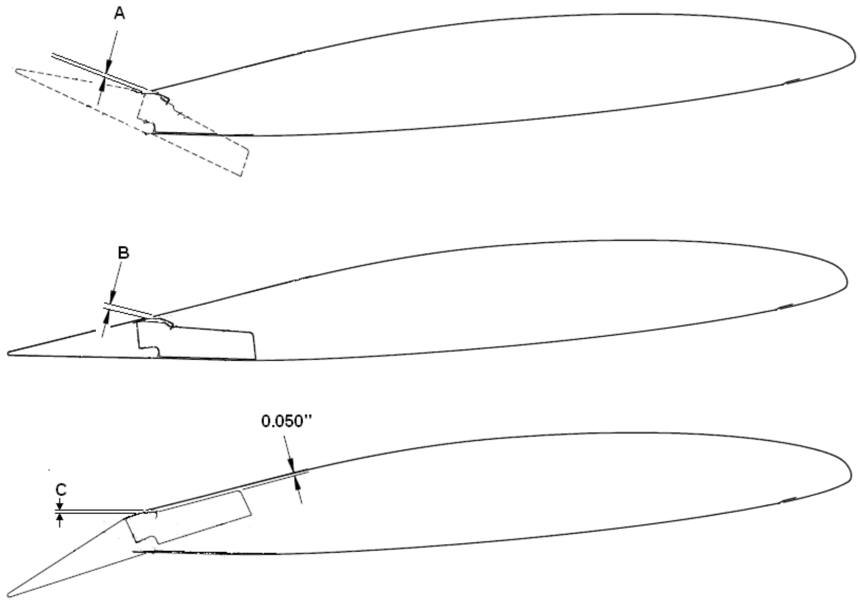

Figure 27-6 Details for the Measurement of the Aileron

9. Check clearance of 0.050" (minimum) between aileron mass balance arm and internal surface of wing skin, with aileron in the down position (2 places port/2 places starboard).

10. Check clearance of 0.050" (minimum) between each end of bolts on mass balance arms and each mass balance mass balance enclosure (2 places X 2 enclosures port/ 2 places X 2 enclosures (starboard).

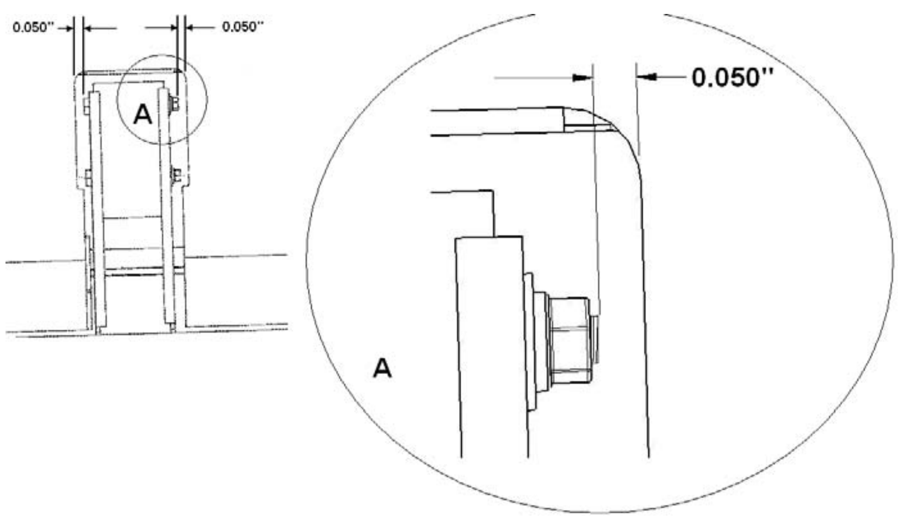

Figure 27-7 Clearance Between Balance Weight and Wing

NOTE

For airplanes certified at 1750 lbs, it may be necessary to open the top of wing-aileron closeout to obtain minimum clearances. Perform the following in order:

11. Remove aileron.
12. Clamp wing trailing edge uniformly and continuously (see Figure 27-8)

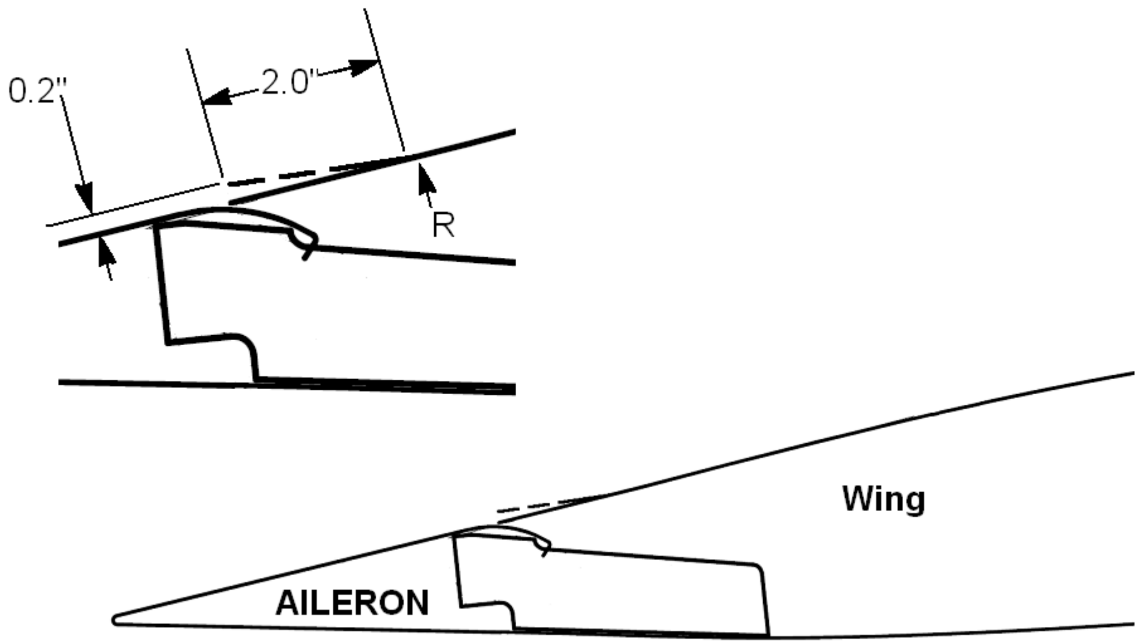

Figure 27-8 Opening "C" Channel of Wing Aileron Closeout.

CAUTION

When bending trailing edge up, do not exceed 0.2" over a distance of 2.0" measured along wing chord. Bend radius 'R' must be gradual, continuous and not leave tool marks, significant surface impression, or any surface cracks.

13. Bend upward slowly in small increment.
14. Re-attach aileron and check clearances.
15. Repeat steps 8 thru 14 until minimum clearances are obtained. Record Clearances.
16. Check for correct operation and freedom of movement.
17. Verify rigging in accordance with Section 27-90 Rigging Procedures on page 103 this chapter.

NOTE

Bending of wing trailing edge to achieve clearances may have loosened rivets. Rivets must be inspected for looseness and, if found, tighten (by squeezing rivet) as required. Inspect trailing edge rivets for looseness; tighten as required

AILERON WING BELL-CRANK REMOVAL

Perform this procedure to remove the bell-crank. Refer to Figure 27-9 during this procedure.

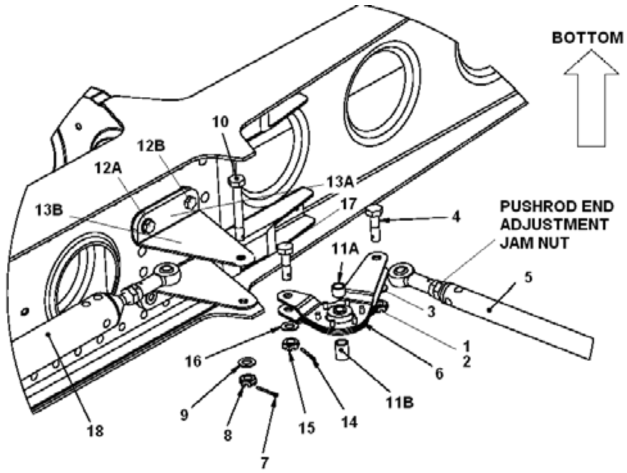

Figure 27-9 Aileron Wing Bell-crank Assembly

1. Remove two access panels on lower wing surface to gain access to bell-crank.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

2. Remove split pin (1), nut (2), washer (3), and bolt (4) that fasten outboard pushrod (5) to bell-crank (6). Separate pushrod from bell-crank.
3. Remove split pin (7), nut (8), washer (9), and bolt (10) from bell-crank center, noting placement of bushings (11a and 11b).
4. Remove 2 lower bolts (12) of lower bell-crank bracket (13) to wing and remove lower bell-crank bracket and "cuddle" plate (13a).
5. Remove split pin (14), nut (15), washer (16) and bolt (17) that fasten inboard aileron pushrod (18) to bell-crank and separate pushrod from bell-crank.
6. Remove bell-crank and bearing.

AILERON WING BELL-CRANK INSTALLATION

Perform this procedure to install the bell crank. Refer to Figure 27-10 during this procedure.

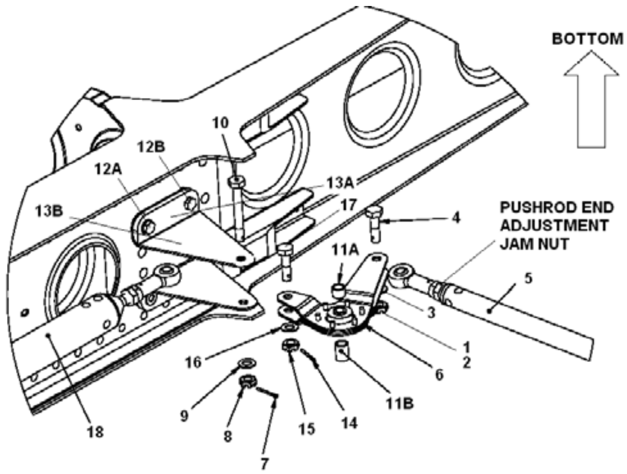

Figure 27-10 Aileron Wing Bell-crank Assembly

1. Using bolt (10), temporarily position bell-crank (6) and bushings (11).

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut..

2. Reconnect inboard pushrod (18) to bell-crank using bolt (17), washer (16) and castellated nut (15). Torque to a value of at least 90 in-lbs. Continue to torque to nearest castellated slot for split pin. Do not exceed 100 in-lbs. of torque. Install a new split pin. Refer to Chapter 20 – *Standard Practice*.
3. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
4. Remove bolt (10) from bell-crank and bushings.
5. Install lower bell-crank bracket to wing using bolt 12A (AN3-16A - long) and bolt 12B (AN3-5A short). Bolt 12A goes towards the wing root; bolt 12B goes toward the wing tip. Torque to a value of at least 30 in-lbs. Continue to torque to nearest castellated slot for split pin. Do not exceed 40 in-lbs. of torque. Install a new split pin. Refer to Chapter 20 – *Standard Practice*.
6. Install bell-crank and bearing ensuring bushings are in place and torque to 80-100 in-lbs. Torque to a value of at least 80 in-lbs. Continue to torque to nearest castellated slot for split pin. Do not exceed 100 in-lbs. of torque. Install a new split pin. Refer to Chapter 20 – *Standard Practice*.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

7. Reconnect outboard aileron pushrod (5) using bolt (4), washer (3) and castellated nut (2). Torque to a value of at least 50 in-lbs. Continue to torque to nearest castellated slot for split pin. Do not exceed 70 in-lbs. of torque. Install a new split pin. Refer to Chapter 20 – *Standard Practice*.
8. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
9. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.
10. Install access panels.

AILERON OUTBOARD WING PUSHROD REMOVAL

Perform this procedure to remove the outboard wing pushrod.

1. Remove two access panels on lower wing surface to gain access to bell-crank.
2. Fully extend wing flap.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

3. Refer to Figure 27-11. Remove split pin (1) and discard. Remove nut (2), washer (3) and bolt (4) securing end of pushrod (6) to inboard end of aileron, noting position of spacers (5).

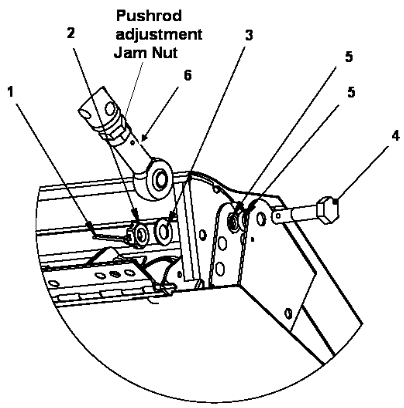

Figure 27-11 Pushrod to Aileron Connection

4. Refer to Figure 27-12. Remove split pin (7) and discard. Remove nut (8), washer (9) and bolt (10) securing end of pushrod to bell-crank (6).

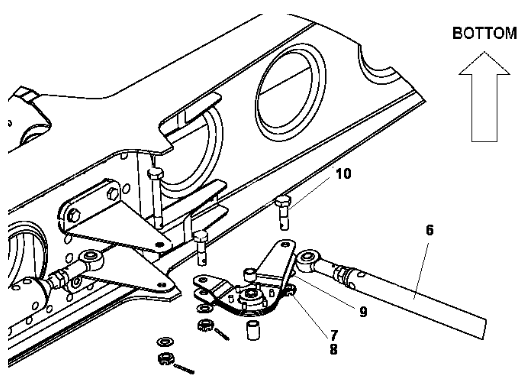

Figure 27-12 Outboard Aileron Push Rod Connection

5. Remove pushrod; mark wing (bell-crank) and aileron ends.

AILERON OUTBOARD WING PUSHROD INSTALLATION

Perform this procedure to install the aileron outboard wing pushrod.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

1. If installing a different pushrod than the one removed in the removal procedure, adjust the replacement pushrod to the length shown in Table 27-1.
2. Insert pushrod into wing, ensuring correct orientation of wing and aileron ends.
3. Install bolt (4), washer (3), and nut (2) securing pushrod bearing end to inboard end of aileron. Torque to a value of at least 50 in-lbs. Continue to torque to nearest castellated slot for split pin. Do not exceed 70 in-lbs. of torque. Install a new split pin. Refer to Chapter 20 – *Standard Practices*.
4. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
5. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.
6. Install bolt (10), washer (9) and nut (8) securing pushrod bearing end to wing bell-crank. Torque to a value of at least 50 in-lbs. Continue to torque to nearest castellated slot for split pin. Do not exceed 70 in-lbs. of torque. Install a new split pin. Refer to Chapter 20 – *Standard Practices*.
7. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
8. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.
9. If installing a new pushrod, Verify rigging in accordance with Section 27-90 Rigging Procedures on page 103 this chapter.
10. Install access panels.

INBOARD WING PUSHROD REMOVAL

Perform this procedure to remove the inboard wing pushrod.

NOTE

Wing(s) must be removed from airplane (See Chapter 57 - Wings).

1. Remove 2 access panels on lower wing surface to gain access to bell-crank.
2. Remove bell-crank in accordance with Aileron Wing Bell-Crank Removal on page 15 of this chapter.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

3. Refer to Figure 27-13. Remove split pin and discard. Remove nut, washers and bolt (1) securing inboard end of aileron inboard-wing pushrod to wing portion of aileron quick-connect device (2).

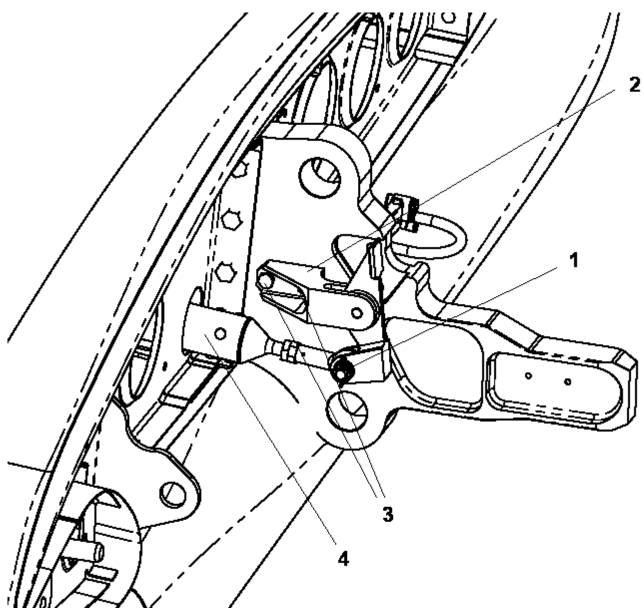

Figure 27-13 Pushrod to Quick-connect Device Connection

4. Remove aileron quick connect device in accordance with the Inboard Quick-Disconnect Device Removal procedure on page 25 of this chapter.
5. Remove pushrod (4); mark inboard and outboard ends.

INBOARD WING PUSHROD INSTALLATION

Perform this procedure to install the inboard wing pushrod.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

1. If installing a different pushrod than the one removed in the removal procedure, check the length of the replacement pushrod against the length shown in Table 27-1.
2. Insert pushrod (4) in wing. Check proper orientation of inboard and outboard ends and check that pushrod passes through anti-buckling support approximately halfway along its length.
3. Install quick-connect in accordance with the Wing-Mounted Quick-Disconnect Device Installation procedure on page 23 of this chapter.
4. Install bolt, washer and castellated nut securing inboard pushrod end bearing to aileron quick-connect device. Torque to a value of at least 50 in-lbs. Continue to torque to nearest castellated slot for split pin. Do not exceed 70 in-lbs. of torque. Install a new split pin. Refer to Chapter 20 – *Standard Practices*.
5. Install bell-crank in accordance with Section Aileron Wing Bell-Crank Installation procedure on page 16 of this chapter.
6. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
7. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.
8. If installing a new pushrod, Verify rigging in accordance with Section 27-90 Rigging Procedures on page 103 this chapter.
9. Replace access panels.

WING-MOUNTED QUICK-DISCONNECT DEVICE REMOVAL

Perform this procedure to remove the wing mounted quick-disconnects. Refer to Figure 27-14.

NOTE

Wing(s) must be removed from airplane (See Chapter 57 - Wings).

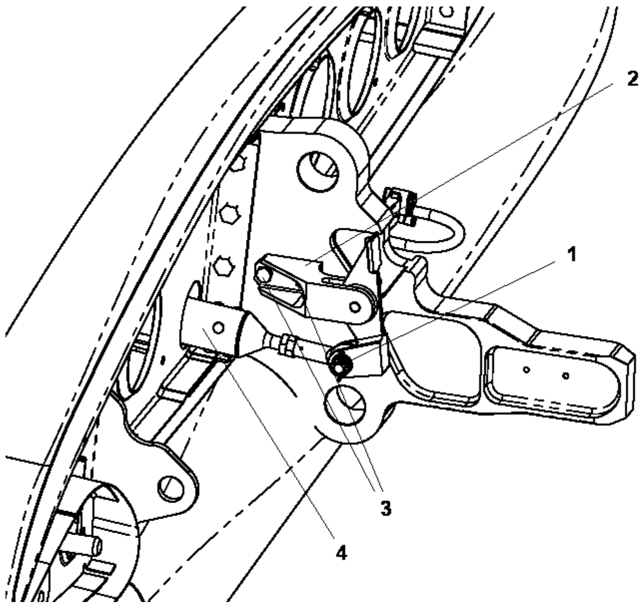

Figure 27-14 Wing Root Assembly

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

1. Remove split pin and discard. Remove nut, washers and bolt (1) securing inboard end of aileron inboard-wing pushrod to wing portion of aileron quick-connect device (2).
2. Remove two bolts (3).
3. Remove quick-disconnect device (2).

WING-MOUNTED QUICK-DISCONNECT DEVICE INSTALLATION

Perform this procedure to install the wing mounted quick-disconnects. Refer to Figure 27-15.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

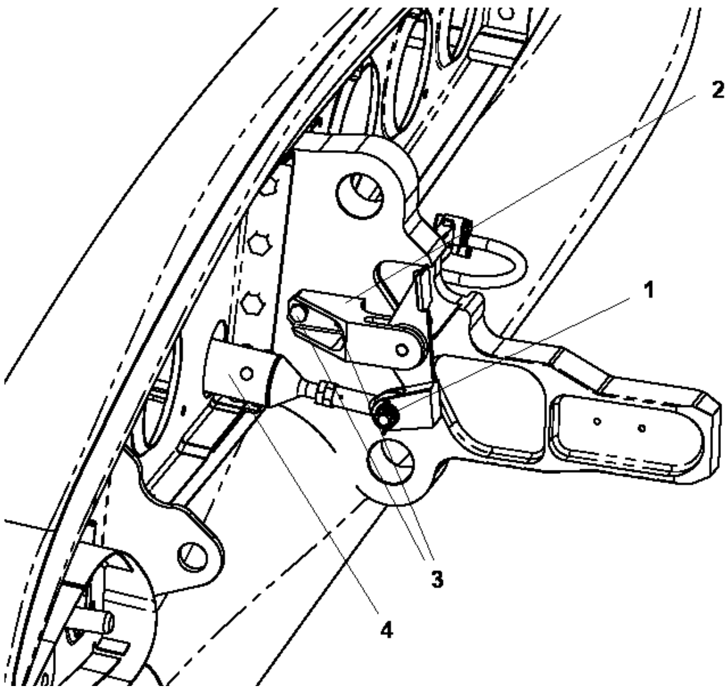

Figure 27-15 Wing Root Assembly

1. Install quick-disconnect device (2).
2. Install two bolts (3). Torque to 30-60 in-lbs.
3. Install bolt, washer and castellated nut securing inboard pushrod end bearing to aileron quick-connect device. Torque to a value of at least 50 in-lbs. Continue to torque to nearest castellated slot for split pin. Do not exceed 70 in-lbs. of torque. Install a new split pin. Refer to Chapter 20 – *Standard Practices*.

NOTE

For correct operation of the quick-disconnect for the aileron, the center pivot of the wing mounted quick-disconnect device must match the center pivot of the fuselage mounted quick disconnect device when the wing is installed on the airplane. The clearance *between the two mating surfaces must only be between 0.000" (minimum) and 0.006" (maximum).*

4. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
5. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.

INBOARD QUICK-DISCONNECT DEVICE REMOVAL

Perform this procedure to remove the inboard quick-disconnect device.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

1. Remove belly panel (see Chapter 53 - *Fuselage*).
2. Remove appropriate wing in accordance with Chapter 57 - *Wings*.

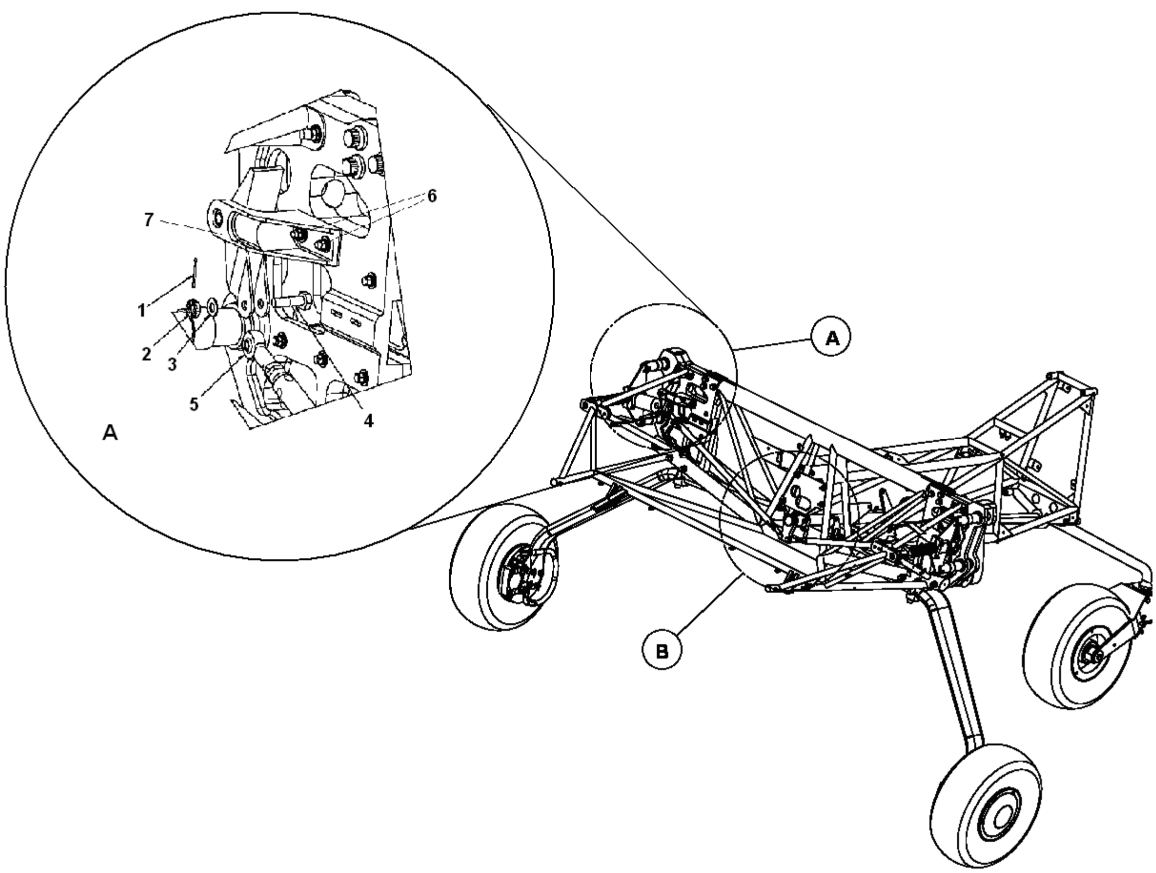

Figure 27-16 Aileron Inboard Quick-disconnect Device

3. Remove split pin (1) and discard. Remove nut (2), washer (3), and bolt (4) that fasten inboard aileron pushrod (5) to quick-disconnect device (7) and separate pushrod from quick-disconnect device.
4. Remove 2 nuts, 4 washers, and 2 bolts (6) and remove quick-disconnect device.

INBOARD QUICK-DISCONNECT DEVICE INSTALLATION

Perform this procedure to install the inboard quick-disconnect device.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

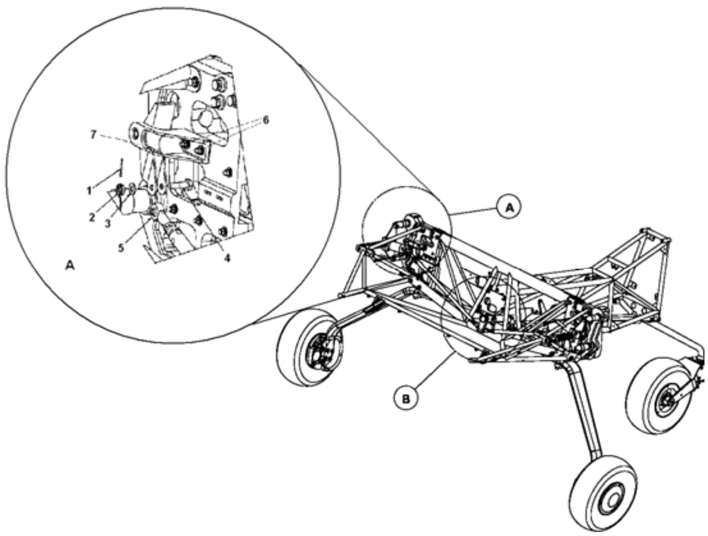

Figure 27-17 Aileron Inboard Quick-disconnect Device

1. Install quick-disconnect device (7) using two bolts, four washers, and two nuts (6). Torque the nuts in accordance with Chapter 20 – *Standard Practices*.
2. Connect inboard aileron pushrod (5) to quick-disconnect device using bolt (4), washer (3), and castellated nut (2). Torque the nuts in accordance with Chapter 20 – *Standard Practices*. Install a new split pin (1).
3. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
4. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.

NOTE

For correct operation of the quick-disconnect for the aileron, the center pivot of the wing mounted quick-disconnect device must match the center pivot of the fuselage mounted quick disconnect device when the wing is installed on the airplane. The clearance between the two mating surfaces must only be between 0.000" (minimum) and 0.006" (maximum).

INBOARD AILERON PUSH ROD REMOVAL

Perform this procedure to remove the inboard aileron push rod.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

1. Remove belly panel (see Chapter 53 - Fuselage).
2. Remove appropriate wing in accordance with Chapter 57 - *Wings* and place upside down on workbench.
3. Disconnect inboard aileron pushrod from quick-disconnect device in accordance with the Inboard Quick-Disconnect Device Removal procedure on page 22 of this chapter.
4. Remove split pin (1) and discard. Remove the nut and washer (2), and bolt (3) from appropriate aileron inboard pushrod (4) and separate pushrod from torque tube.

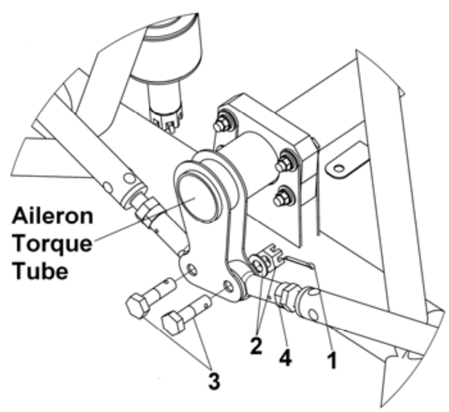

Figure 27-18 Aileron Inboard Pushrod to Torque Tube

5. Remove inboard aileron pushrod.

INBOARD AILERON PUSHROD INSTALLATION

Perform this procedure to install the inboard aileron pushrod.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

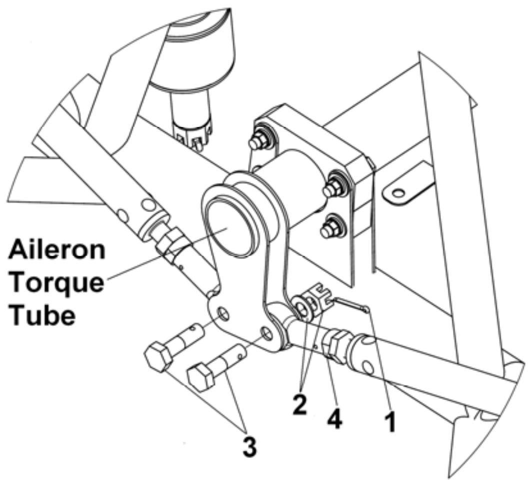

Figure 27-19 Aileron Inboard Pushrod to Torque Tube

1. If installing a different pushrod than the one removed in the removal procedure, adjust the replacement pushrod to the length shown in Table 27-1.
2. Insert aileron inboard pushrod into position.
3. Attach inboard pushrod to aileron torque tube using bolt (3), washer, and nut (2). Torque the nut in accordance with Chapter 20 – *Standard Practices*.
4. Insert a new split pin (1).
5. Attach aileron inboard pushrod to quick-disconnect device in accordance with the Wing-Mounted Quick-Disconnect Device Installation procedure on page 23 of this chapter.
6. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
7. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.

.

NOTE

When installed correctly, the components should mate together without slippage between parts.

8. Verify rigging in accordance with Section 27-90 Rigging Procedures on page 103 this chapter.
9. Install belly panel (see Chapter 53 – *Fuselage*).

AILERON TORQUE TUBE REMOVAL

Perform this procedure to remove the aileron torque tube

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

1. Remove belly panel (see Chapter 53 – *Fuselage*).
2. Remove inboard aileron pushrods (Port and Starboard) in accordance with the Inboard Aileron Push Rod Removal procedure on page 27 of this chapter.
3. At rear of aileron torque tube, remove 4 nuts (3), 4 washers (2), and 4 bolts (1) from aileron torque tube bearing halves (4). Remove bearing halves. See Figure 27-20.

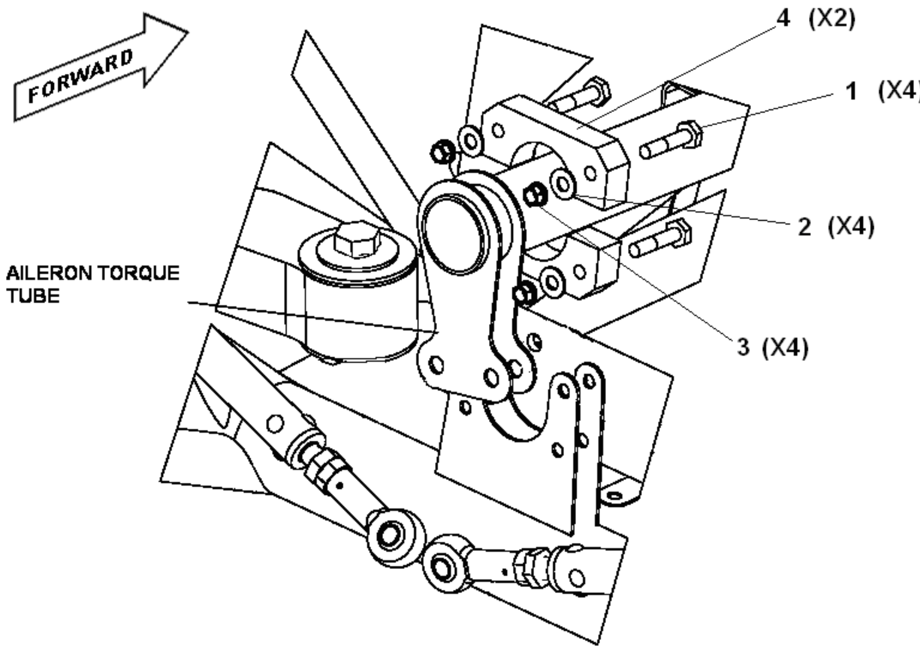

Figure 27-20 Aileron Torque Tube (Rear)

4. Disconnect yoke assembly from aileron torque tube in accordance with the procedures in Section 27-40 - *Control Yoke* on page 77 of this chapter.
5. Disconnect forward pitch control pushrod from aileron torque tube in accordance with the procedures in Section 27-30 - *Stabilator and Tab* on page 61 of this chapter.
6. See Figure 27-21. At forward end of aileron torque tube, remove split pin (1) and discard. Remove nut (2), washers (3 and 4).

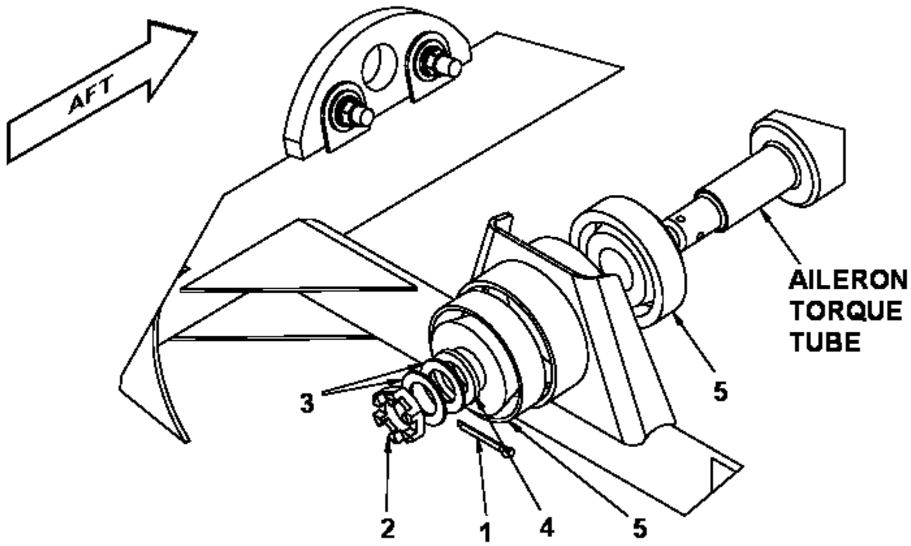

Figure 27-21 Aileron Torque Tube (Front)

7. Pull torque tube aft, retaining bearings (4), noting their positions and orientation on torque tube.
8. Work aileron torque tube around, down, and out through belly opening.

AILERON TORQUE TUBE INSTALLATION

Perform this procedure to install the aileron torque tube.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

1. Work aileron torque tube around, up, and into position through belly opening.
2. At forward end of aileron torque tube, install bearings (4), in position and orientation on torque tube as noted in step 7 of the Aileron Torque Tube Removal procedure on page 30 of this chapter.
3. Install washers (3 and 4), and castellated nut (2). Torque the nut in accordance with Chapter 20 – *Standard Practices*. Install a new split pin (see Figure 27-21)
4. At rear of aileron torque tube, install bearing halves (4) using 4 bolts (1), 4 washers (2), and 4 nuts (3) (Figure 27-20).
5. Install inboard aileron pushrods (Port and Starboard) in accordance with the Inboard Aileron Pushrod Installation procedure on page 28 of this chapter.
6. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
7. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.
8. Connect yoke assembly to aileron torque tube in accordance with the procedures in Section 27-40 - *Control Yoke* on page 77 of this chapter.
9. Connect forward pitch control pushrod to aileron torque tube in accordance with the procedures in Section 27-30 - *Stabilator and Tab* on page 61 of this chapter.
10. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
11. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.
12. Check control yoke for proper operation and ease of movement.
13. Verify rigging in accordance with Section 27-90 Rigging Procedures on page 103 this chapter.
14. Install belly panel.

Section 10-02 Aileron Inspection

Inspect the following items on the aileron for abnormalities:

- Inspect aileron skin for dents, scratches, malformations, bubbled paint, and foreign object debris.
- Inspect for “smoking rivets”.

NOTE

“Smoking Rivets” – Term used to describe a loose or working rivet whose vibration causes a black streak trailing aft. This could indicate a failure of the riveted joint. Black streaks may also indicate excessive wear of the piano hinge pin. If black streaks are noted, then inspect both the rivets and the hinge pin for excessive wear.

Section 10-03 Troubleshooting

Table 27-3 has troubleshooting information for use in resolving issues with the ailerons.

Complaint	Possible Cause	Remedy
Aileron control binding or stiffness	“Dry” or defective bearings	Replace as required
	Misaligned aileron hinge line	Replace as required
	Pushrod hindrance	Verify all pushrods are clear
Airplane rolls left or right (wing heaviness)	Aileron linkage adjusted incorrectly	Adjust (see Chapter 06 for correct deflections)
	Aileron fixed trim tab adjusted incorrectly	Adjust aileron tab
Looseness (“play” or “slop”) in aileron controls. Play or slop is no greater than .050 inches of movement of a surface’s trailing edge, when surface is fixed to adjacent structure, with control circuit locked in neutral position.	Worn or loose attachments or bearings	Check entire aileron control circuit, replace as necessary
	Play in aileron quick-connect assemblies	Reset quick connect mount bolting to remove free play

Table 27-3 Aileron System Troubleshooting

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Section 27-20 Rudder

The airplane rudder is of conventional aluminum construction. It is attached to the starboard side of the fuselage vertical fin by a full-length piano hinge. A fixed ground-adjustable trim tab is attached to the trailing edge of the rudder

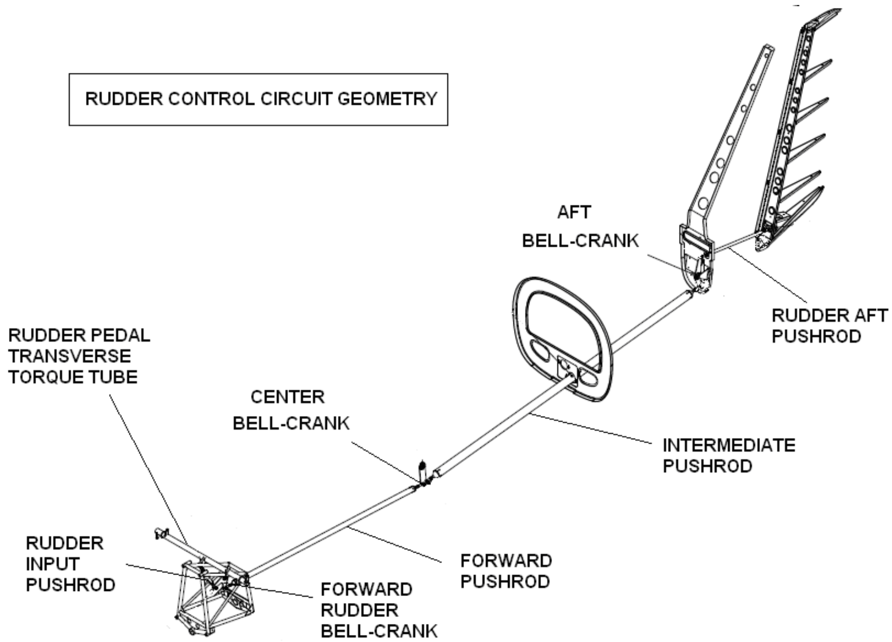


Figure 27-22 Rudder Control Circuit Geometry

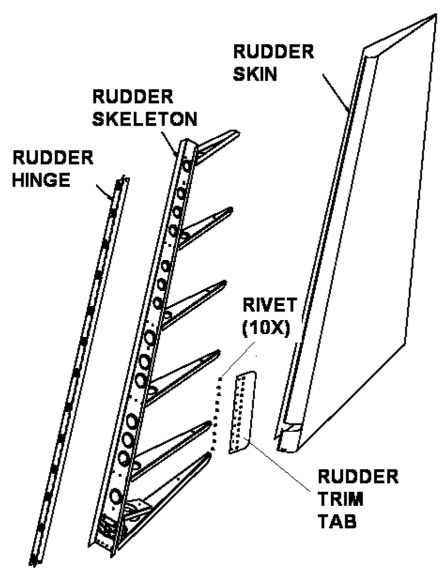


Figure 27-23 Rudder

Port side (pilot) and starboard (copilot) rudder pedal assemblies are connected to a transverse torque tube located behind the instrument panel. Movement of the left/right pedals rotates the torque tube. This rotation deflects the rudder via a series of bell-cranks, idler bearings, and pushrods which pass through the cockpit center console and aft fuselage. The aft-most pushrod terminates at a rudder drive arm which protrudes slightly through the left side of the rudder.

Each rudder pedal assembly incorporates an adjustment feature to accommodate different pilot heights and leg lengths. Rotating the adjustment crank clockwise moves the pedals aft; rotating it counterclockwise moves them forward. The total range of travel for rudder pedals on airplanes equipped with finger brakes is 4.5 inches. The total range of travel for rudder pedals on airplanes equipped with finger brakes is 1.5 inches. This adjustment may be made on the ground or in-flight (not recommended) without affecting rudder control.

One of two options of rudder pedal assemblies is on the airplane, dependant upon the braking system used on the airplane. Rudder pedal assembly P/N 135A-45-011 is fitted to airplanes that utilizing finger brakes and does not have toe brakes. Rudder pedal assembly P/N 135A-45-013 provides toe brake control.

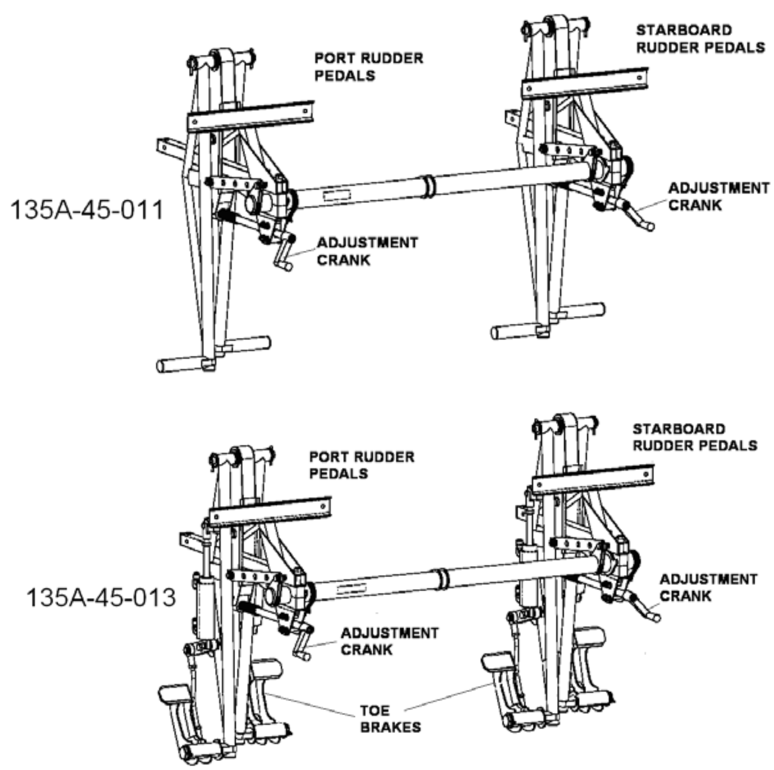


Figure 27-24 Rudder Pedal Assemblies

For lubrication purposes, access to the rudder pedal assemblies is gained from inside the cockpit. Access to the upper rudder pedal mount bushings may require that the instrument panel or circuit breaker panel be removed. For panel removal, refer to Chapter 53 - *Fuselage*.

The rudder input pushrod assemblies are attached with castellated nuts and split pinned bolts. Refer to Chapter 20 – *Standard Practice* for installation torque. Care must be taken to avoid over tightening these nuts and binding the pushrod.

Access to the forward rudder pushrods and idler bearings is gained by removing the fuselage belly panel.

Access to pushrods and idler bearings in the center fuselage area is gained by removing an access panel in the floor of the baggage compartment.

Access to the aft-most bell-cranks and pushrod are gained by removing the rudder itself and/or access panels on the starboard side of the lower aft fuselage.

Section 20-01 Rudder Procedure

This section contains the procedures for the rudder system. These procedures include the following:

Procedure	Page
Rudder Removal	38
Rudder Installation	40
Rudder Mid-fuselage Idler-plate Removal	41
Rudder Mid-fuselage Idler-plate Installation	43
Forward Rudder Bell-crank Removal	45
Forward Rudder Bell-crank Installation	47
Rudder Input Pushrod Removal	49
Rudder Input Pushrod Installation	50
Rudder Pedal Assembly Removal	52
Rudder Pedal Assembly Installation	55
Extending the Closeout Hole for the Rudder	57

NOTE

Removal and replacement of the forward and intermediate rudder pushrods is not anticipated during routine maintenance. In the event that forward and forward and intermediate rudder need to be removed from the airplane, contact Liberty Customer Support for the recommended procedure.

RUDDER REMOVAL

Perform this procedure to remove the rudder.

NOTE

Two persons are required for this procedure. Rudder hinge remains with rudder.

1. Remove 13 washer-head screws securing rudder hinge to right side of vertical fin. See Figure 27-25.

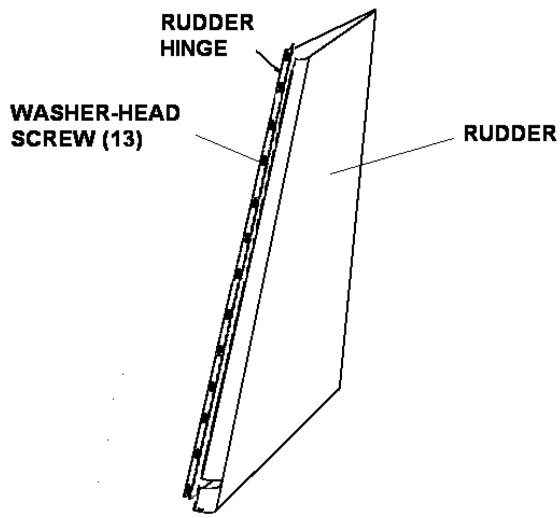


Figure 27-25 Rudder Hinge Screws

CAUTION

Support rudder to prevent damage to rudder or vertical stabilizer.

2. Deflect rudder fully to starboard.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

3. Remove split pin (1) and discard. Remove the nut (1), washers (3) and bolt (4) securing pushrod end bearing to rudder drive brackets, noting placement of washers.

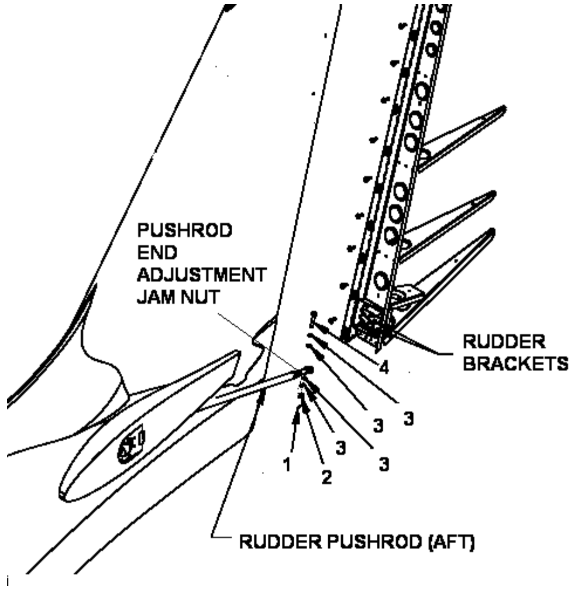


Figure 27-26 Rudder Pushrod Aft Connection

4. Remove rudder.

CAUTION

Do not operate rudder pedals while rudder is removed. Operating rudder pedals while rudder is removed may damage the composite vertical stabilizer-rudder closeout.

RUDDER INSTALLATION

Perform this procedure to install the rudder.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

NOTE

Two persons are required for this procedure.

1. If installing a different pushrod than the one removed in the removal procedure, adjust the replacement pushrod to the length shown in Table 27-1.
2. Install bolt, washers, and castellated nut securing rudder drive arm to rudder pushrod. Torque to a value of at least 80 in-lbs. Continue to torque to nearest castellated slot for split pin. Do not exceed 100 in-lbs. of torque. Install a new split pin. Refer to Chapter 20 – *Standard Practices*.
3. Install new split pin.
4. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
5. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.
6. Wet assemble 13 washer-head screws with RTV 100, then install screws securing rudder hinge to right side of vertical fin and torque to 30 - 40 in-lbs.
7. Verify correct operation and freedom of movement.
8. Verify rigging in accordance with Section 27-90 Rigging Procedures on page 103 this chapter.

.

RUDDER MID-FUSELAGE IDLER-PLATE REMOVAL

Perform this procedure to remove the rudder mid-fuselage idler-plate.

CAUTION

Before starting this procedure, the tail of airplane requires support. Failure to support tail of airplane may cause damage to tail section while accessing any area aft of passenger compartment.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

1. Support the tail of the airplane.
2. Remove the four screws for the access panel from floor of baggage bay.

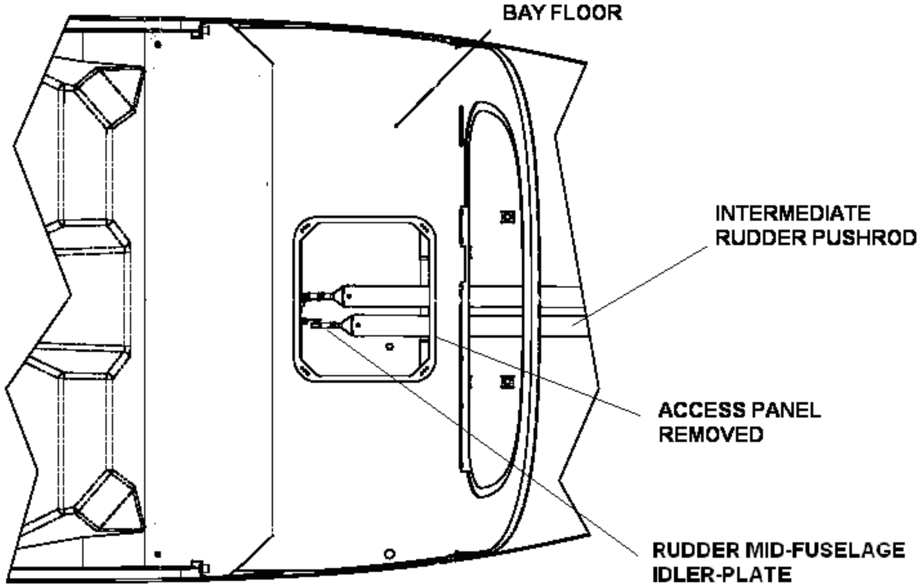


Figure 27-27 Center Idler-plate Access Panel

3. Refer to Figure 27-27. Remove split pin (1) and discard. Remove the nut (2), washers (3 and 4), and bolt (5) from center idler plate (7) and separate forward rudder pushrod (6) from the idler-plate.

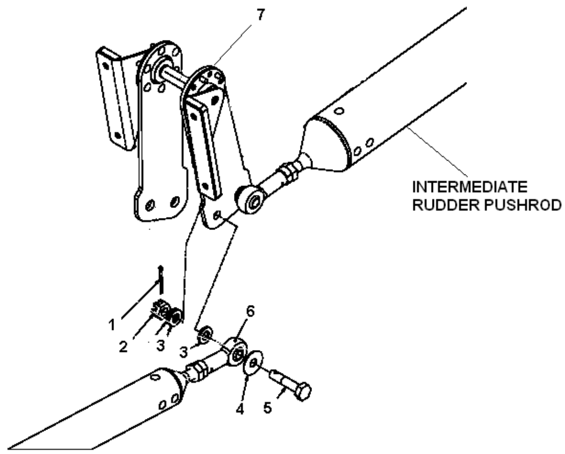


Figure 27-28 Disconnecting Fwd. Rudder Pushrod

4. Refer to Figure 27-28. Remove split pin (1) and discard. Remove the nut (2), washers (3 and 4), and bolt (5) from idler plate (6) and separate intermediate rudder pushrod from idler-plate.

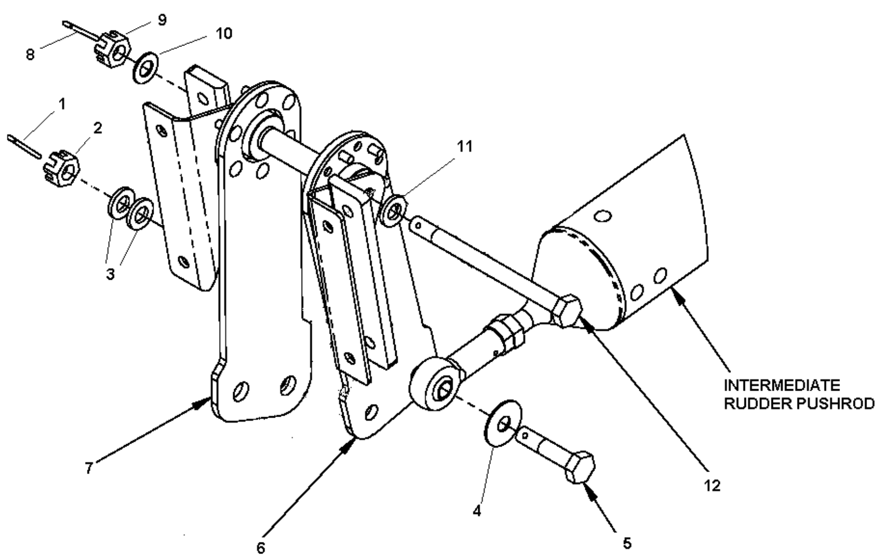


Figure 27-29 Intermediate Rudder Pushrod and Center Idler-plate

5. Remove split pin (8) and discard. Remove the nut (9), washers (10 and 11), and bolt (12). While supporting horizontal stabilizer idler plate (7), remove rudder idler-plate (6) from pivot point (not shown).

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

RUDDER MID-FUSELAGE IDLER-PLATE INSTALLATION

Perform this procedure to install the rudder mid-fuselage idler-plate.

CAUTION

Before starting this procedure, the tail of airplane requires support. Failure to support the tail of airplane may cause damage to tail section while accessing any area aft of passenger compartment.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

1. Support tail of airplane.

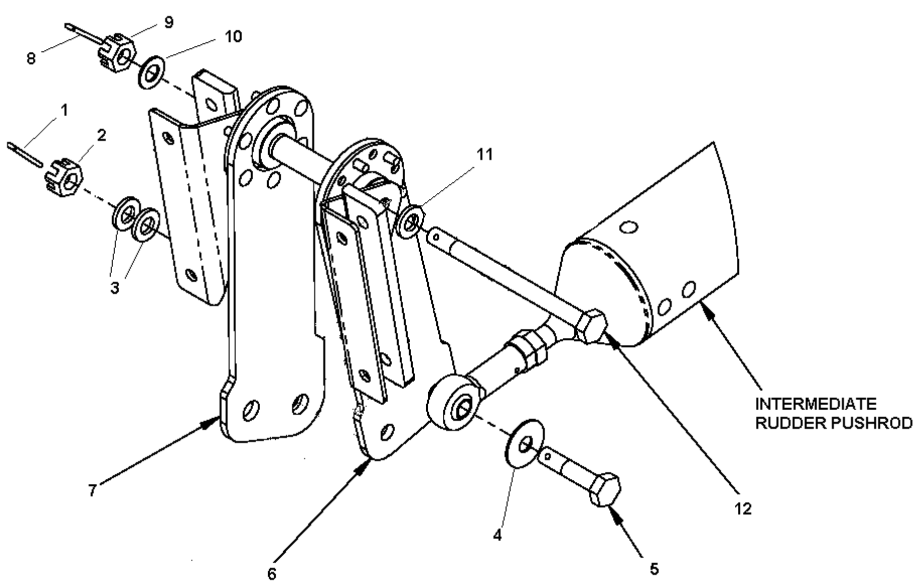


Figure 27-30 Intermediate Rudder Pushrod and Center Idler-plate

2. See Figure 27-30. Place washer (11) onto bolt (12) and insert through rudder idler-plate (6), stabilizer idler-plate (7), and pivot point (not shown).
3. Install washer (10) and castellated nut (9). Torque the nut in accordance with Chapter 20 – *Standard Practices*.
4. Insert a new split pin (8).
5. Attach intermediate rudder pushrod to rudder idler-plate using bolt (6), washers (3 and 4), and castellated nut (2). Torque the nut in accordance with Chapter 53 - *Fuselage*.
6. Insert a new split pin (8).

7. See Figure 27-31. Attach forward rudder pushrod to rudder idler-plate using bolt (6), washers (3 and 4), and castellated nut (2). Torque the nut in accordance with Chapter 53 - *Fuselage*.
8. Insert a new split pin (8).

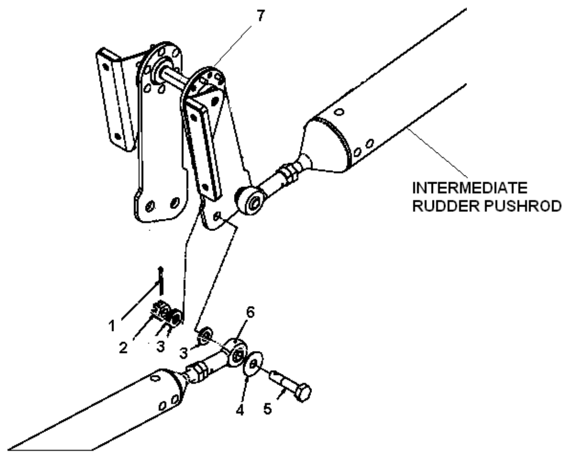


Figure 27-31 Connecting Fwd. Rudder Pushrod

9. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
10. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.
11. Verify rigging in accordance with Section 27-90 Rigging Procedures on page 103 this chapter.
12. Install access plate in baggage bay floor.

FORWARD RUDDER BELL-CRANK REMOVAL

Perform this procedure to remove the forward rudder bell-crank.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

1. Remove belly panel (See Chapter 53 - Fuselage).
2. See Figure 27-32. Remove split pin (1) and discard. Remove the nut (2), washer (3), spacer (4), and bolt (5). Separate the forward rudder pushrod from forward rudder bell-crank.

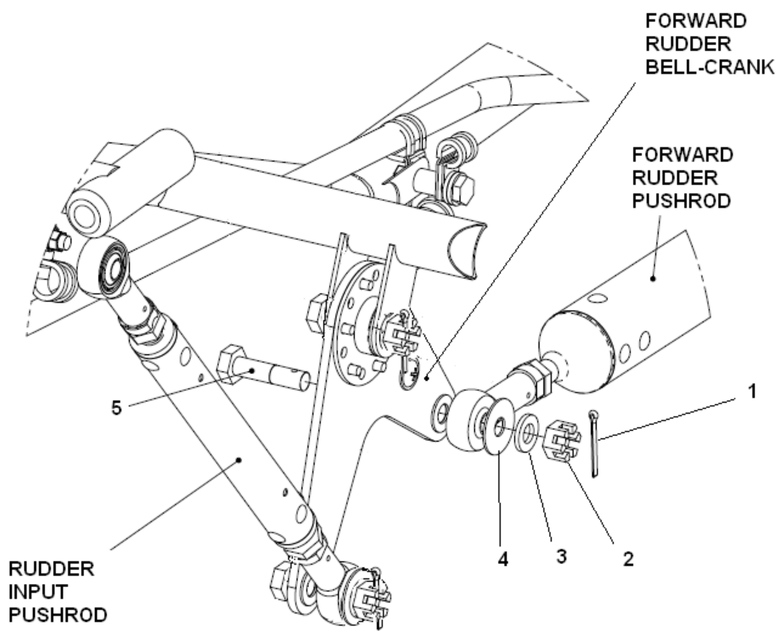


Figure 27-32 Forward Rudder Bell-crank to Forward Rudder Pushrod Connection

3. See Figure 27-33. Remove split pin (1) and discard. Remove the nut (2), washer (3), spacer (4 and 5), and bolt (6). Note positions of washers. Separate the forward rudder pushrod from forward rudder bell-crank.

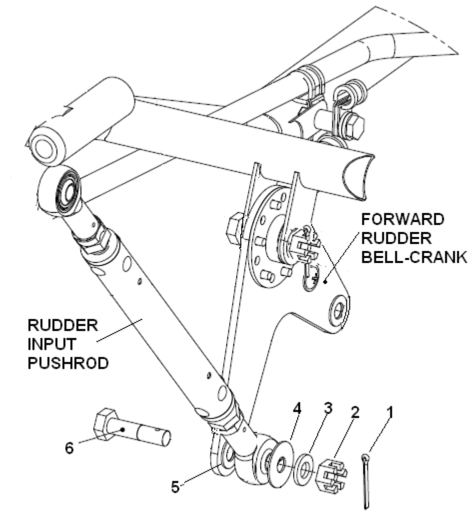


Figure 27-33 Exploded View of the Input Pushrod to Forward Bell-crank

4. Remove split pin (1) and discard. Remove the nut (2), washer (3), and bolt (4) that fasten forward rudder bell-crank to space frame and remove bell-crank.

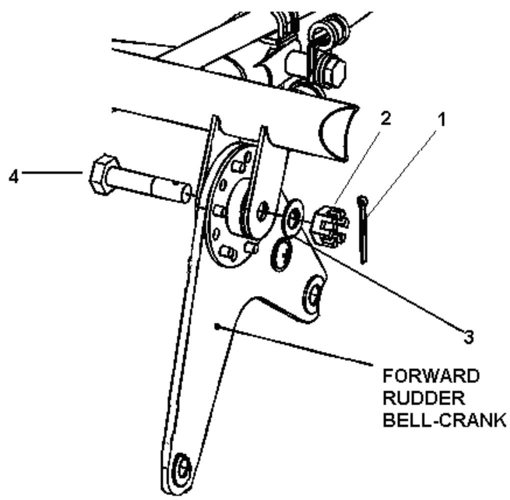


Figure 27-34 Forward Rudder Bell-crank

FORWARD RUDDER BELL-CRANK INSTALLATION

Perform this procedure to install the forward rudder bell-crank.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

1. Install forward rudder bell-crank using bolt (4), washer (3), and castellated nut (2). Torque the nut in accordance with Chapter 20 - Standard Practices.

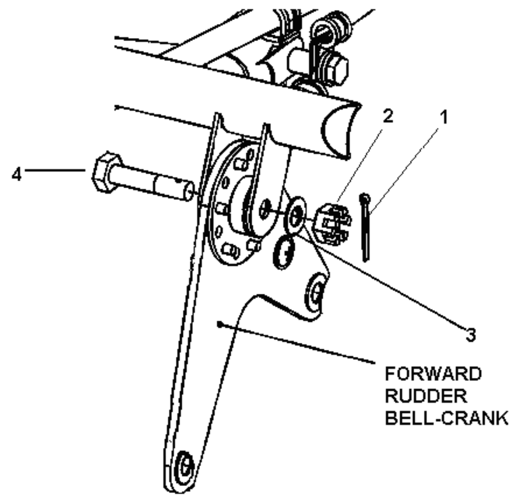


Figure 27-35 Forward Rudder Bell-crank

2. Install a new split pin (1).
3. Install rudder input pushrod to forward rudder bell-crank using bolt (6), spacer (4 and 5), washer (3), and nut (2). Torque the nut in accordance with Chapter 20 - *Standard Practices*. Install new split pin (1).

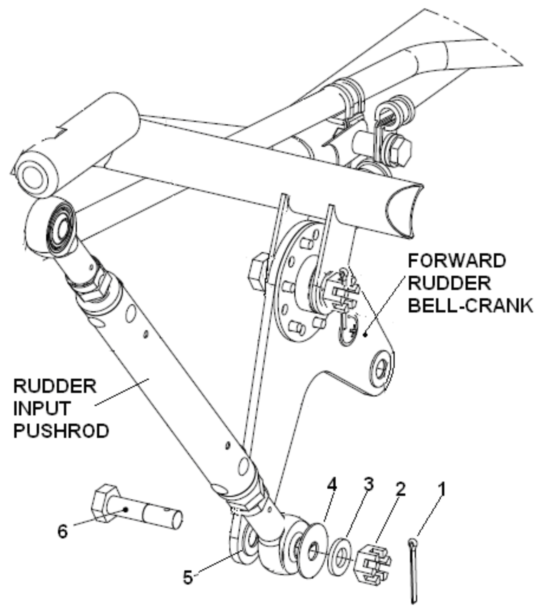


Figure 27-36 Exploded View of the Input Pushrod to Forward Bell-crank

4. Install forward pushrod to forward rudder bell-crank using bolt (5), spacer (4), washer (3), and nut (2). Torque the nut in accordance with Chapter 20 - *Standard Practices*. Install new split pin (1).

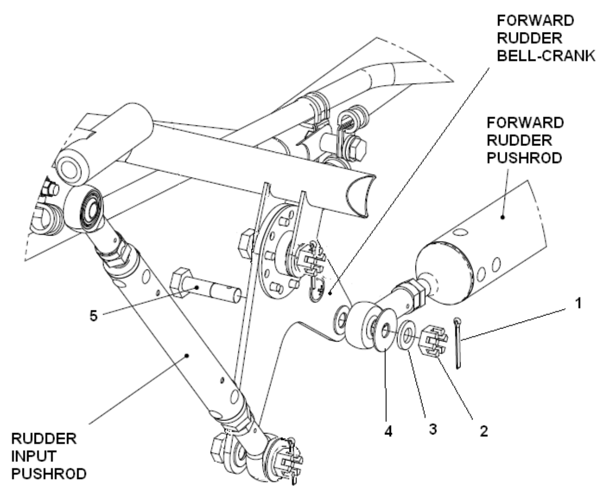


Figure 27-37 Forward Rudder Bell-crank to Forward Rudder Pushrod Connection

5. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
6. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.
7. Verify rigging in accordance with Section 27-90 Rigging Procedures on page 103 this chapter.
8. Install belly panel (See Chapter 53 - *Fuselage*).

RUDDER INPUT PUSHROD REMOVAL

Perform this procedure to remove the rudder input pushrod.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

1. Remove belly panel (see Chapter 53 - *Fuselage*).
2. See Figure 27-38. Remove split pin (1) and discard. Remove the nut (2), washer (3), and bolt (4) that connects the rudder input pushrod to the rudder pedal assembly transverse torque tube. Separate pushrod from torque tube.
3. Remove split pin (5) and discard. Remove the nut (6), washers (7 -9), and bolt (10). Note positions of washers. Separate input the pushrod from forward rudder bell-crank, and remove the pushrod.

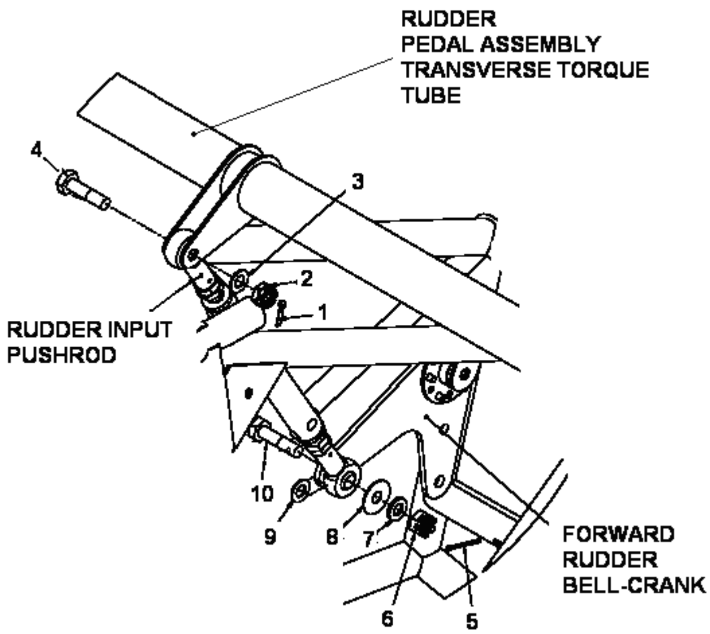


Figure 27-38 Exploded View of the Rudder Input Pushrod

RUDDER INPUT PUSHROD INSTALLATION

Perform this procedure to install the rudder input pushrod. Refer to Figure 27-39during this procedure.

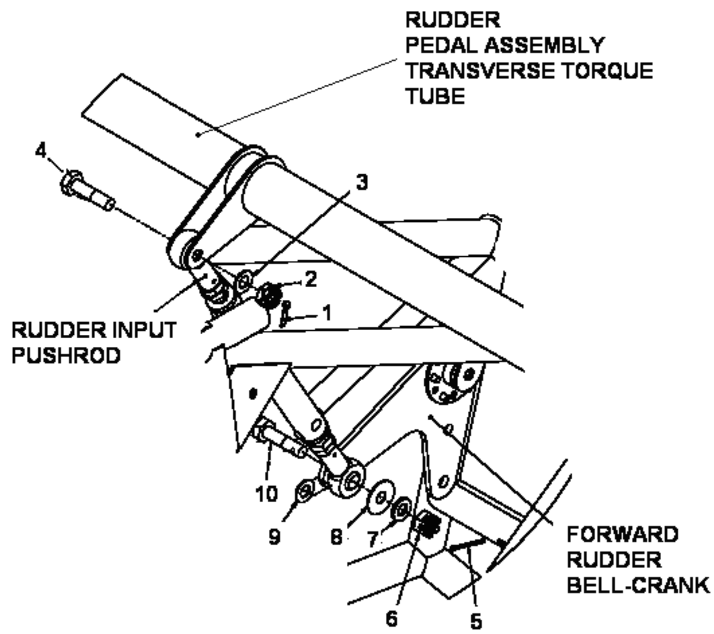


Figure 27-39 Exploded View of the Rudder Input Pushrod

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

1. If installing a different pushrod than the one removed in the removal procedure, adjust the replacement pushrod to the length shown in Table 27-1.
2. Install the input pushrod to the forward rudder bell-crank using bolt (10), washers (7-9) and nut (6). Place washers in positions noted during removal. Torque the nut per Chapter 20 - *Standard Practices*.
3. Install new split pin (5).
4. Install rudder input pushrod to rudder pedal assembly transverse torque tube using bolt (4), washer (3), and nut (2). Torque the nut per Chapter 20 - *Standard Practices*.
5. Install new split pin (1).
6. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.

7. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.
8. Check rudder pedal assembly for normal unrestricted operation.
9. Verify rigging in accordance with Section 27-90 Rigging Procedures on page 103 this chapter.
10. Install Remove belly panel (see Chapter 53 - *Fuselage*).

RUDDER PEDAL ASSEMBLY REMOVAL

Perform this procedure to remove the rudder pedals.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support tail of airplane may cause damage to tail section while accessing any area aft of passenger compartment.

CAUTION

Failure to disconnect batteries may cause damage to airplane electrical circuitry.

NOTE

1. Cover forward windscreen internally to protect from scratching. Liberty recommends using Shrink-Wrap plastic.

2. Two people are required for the removal procedure.

1. Position aircraft master switch to OFF.
2. Install a tail stand underneath tail section of airplane.
3. Remove cabin aft bulkhead access panel, by removing securing screw hardware.
4. Disconnect negative then positive leads from both primary battery and secondary battery. Isolate terminals on batteries to prevent accidental connection.
5. Remove instrument panel from airplane (see Chapter 31 – *Indicators and Recording Systems*).
6. Remove circuit breaker (CB) panel from airplane (see Chapter 24 – *Electrical Power*).
7. Remove avionics panel from airplane (see Chapter 23 – *Communications*).
8. Remove instrument panel console in accordance with Chapter 53 - *Fuselage*.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

9. Disconnect rudder input pushrod from rudder pedal assembly transverse torque tube in accordance with the Rudder Input Pushrod Removal procedure on page 49 of this chapter.

CAUTION

When removing nuts from bolts, check inboard bolts are not pushed out of their mounting holes into engine compartment. Failure to check this may cause damage to firewall when retrieving bolts.

10. Remove four nuts from bolts securing rudder pedal assembly.

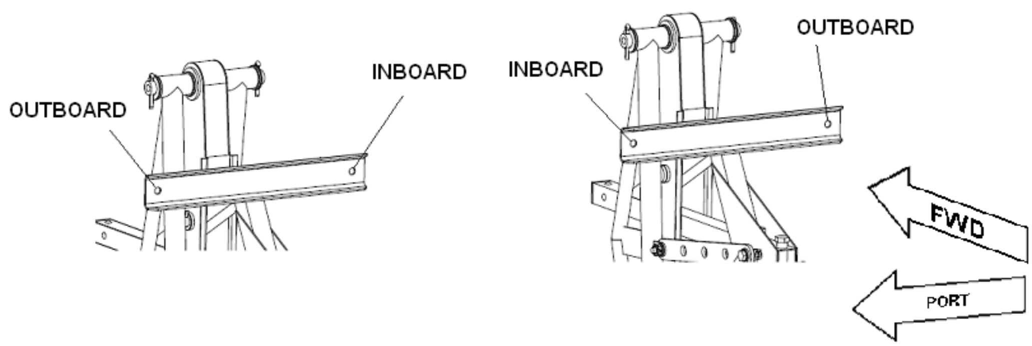


Figure 27-40 Rudder Pedal Assembly Mounting Bolt Locations

NOTE

Washers/spacers may be installed on outboard bolts behind rudder pedal assembly. Note their position and retain for installation.

11. If rudder pedal assembly is equipped with toe brake hydraulic cylinders, disconnect eight hydraulic line fittings from rudder pedal assembly.

NOTE

Use any available means to minimize or clean up any spilled hydraulic fluid.

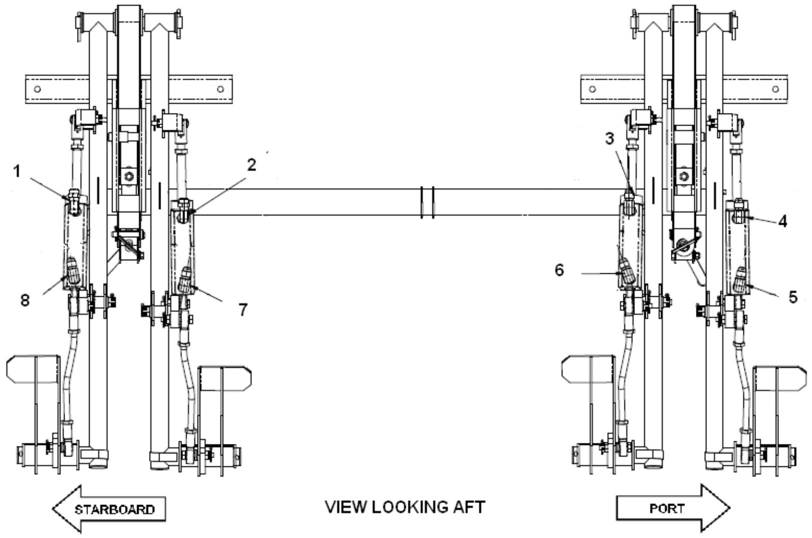


Figure 27-41 Brake Pedal Assembly Hydraulic Line Fittings

12. Lift rudder pedal assembly straight out and away from fuselage and out of airplane.

RUDDER PEDAL ASSEMBLY INSTALLATION

Perform this procedure to install the rudder pedal assembly.

CAUTION

Before starting this procedure, the tail of airplane requires support. Failure to support tail of airplane may cause damage to tail section while accessing any area aft of passenger compartment.

CAUTION

Failure to disconnect batteries may cause damage to airplane electrical circuitry.

NOTE

1. Cover forward windscreen internally to protect from scratching. Liberty recommends using Shrink-Wrap plastic.

2. Two people are required for installation procedure.

1. Prior to installation, check that rudder pedal assembly moves freely.
2. Mount the washers/spacers on outboard mounting bolts in position noted during removal.

CAUTION

When installing rudder pedal assembly and nuts to bolts, check inboard bolts are not pushed out of their mounting holes into engine compartment. Failure to check this may cause damage to firewall when retrieving bolts and re-inserting them into bolt holes.

3. Set rudder pedal assembly into position, with rear end mounts fitting into mating holes in fuselage. Install the nuts (4) on mounting bolts.
4. Torque the bolts in accordance with Chapter 20 - *Standard Practices*.

NOTE

If rudder pedal assembly does not move freely after apply the correct torque to the nuts, additional washers/spacers may be required to eliminate binding.

5. Check rudder pedal assembly moves freely.

NOTE

Upper fittings on starboard rudder pedals (1 and 2) are compression fittings. Compression sleeves must be replaced, necessitating removal of portion of tubing end. If resultant tubing length prohibits proper connection, tubing must be replaced.

6. If rudder pedal assembly is equipped with toe brake hydraulic cylinders, connect eight hydraulic line fittings to rudder pedal assembly. Replace compression sleeves (P/N 260P04) of fittings (1 and 2) as required.
7. Bleed brake system in accordance with Chapter 32 – *Landing Gear*.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

8. Connect rudder input pushrod to rudder pedal assembly transverse torque tube in accordance with the Rudder Input Pushrod Installation procedure on page 50 of this chapter.
9. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
10. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.
11. Install instrument panel console in accordance with Chapter 53 - *Fuselage*.
12. Install avionics panel in accordance with Chapter 23 – *Communications*.
13. Install the circuit breaker panel in accordance with Chapter 24 – *Electrical Power*.
14. Install instrument panel in accordance with Chapter 31 – *Indicators and Recording Systems*.
15. Connect negative then positive leads to both primary battery and secondary battery.
16. Install cabin aft bulkhead access panel using securing screw hardware.
17. Remove protective cover from inside of front windscreen.

EXTENDING THE CLOSEOUT HOLE FOR THE RUDDER

If during the inspection, or if binding is detected when using the rudder, perform this procedure to extend the closeout hole in the fuselage.

WARNING

SANDING OR GRINDING THE FUSELAGE, OR OTHER COMPOSITE SURFACES, WILL CREATE A FINE DUST. USE A BREATHING MASK WHILE SANDING ANY COMPOSITE SURFACE. DO THE SANDING IN A WELL-VENTILATED AREA. ALSO, USE A VACUUM SYSTEM WITH A MICRO-FILTER TO REMOVE DUST PARTICLES DURING THE GRINDING PROCESS.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

1. Perform the Rudder Removal procedure on page 38 of this chapter.
2. Remove split pin (1) and discard. Remove the nut (2), washers (3 and 4), and bolt (5) and separate aft rudder pushrod (6) from aft idler-plate.

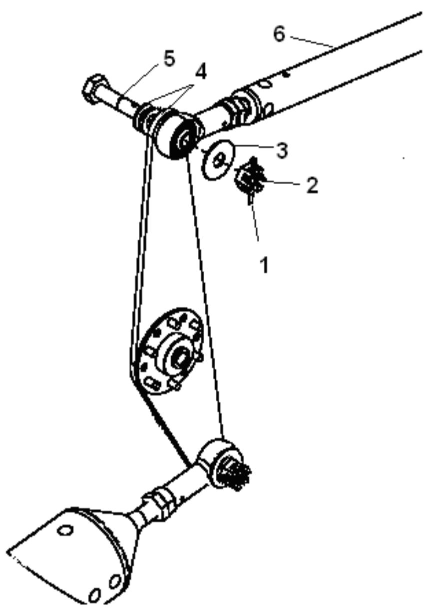


Figure 27-42 Aft Pushrod to Aft Idler Plate Connection

3. Use a 1-inch diamond wheel to cut and remove composite material as shown in Figure 27-43. Figure 27-44 shows the pass-through hole as seen before and after the trimming.

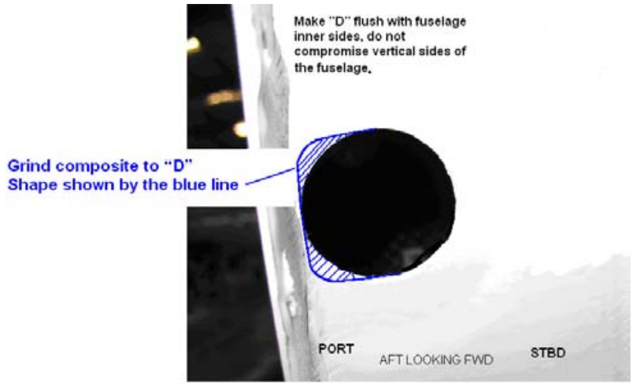


Figure 27-43 Showing Area of Fuselage

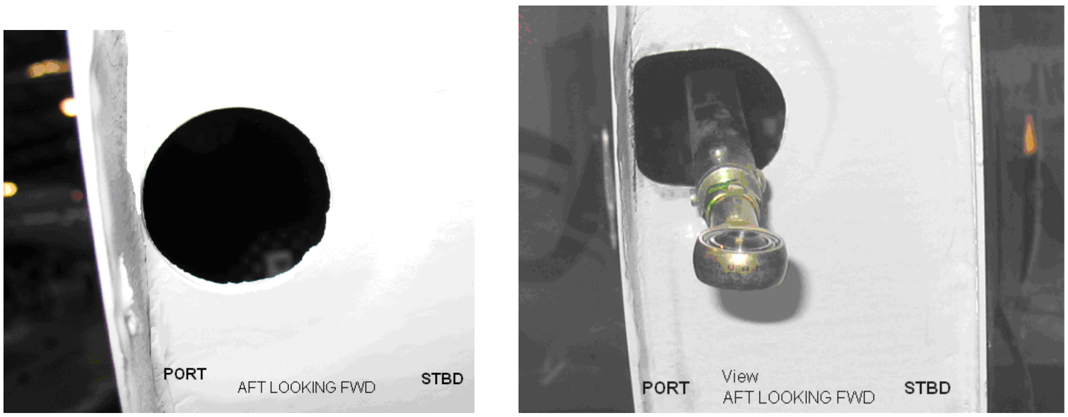


Figure 27-44 Before and After Extending the Pass-thru Hole

4. Finish with a diamond tip barrel grinder. Check that the new boundary of the pass through hole, does not have any delaminations or sharp edges.

NOTE

Ensure that there are no delaminations or sharp edges after trimming. Ensure the inner vertical sides of the fuselage are not compromised by grinding.

5. Install aft rudder push rod to aft idler plate. Install the nut (2), washers (3 and 4), and bolt (5) and separate aft rudder pushrod (6) from aft idler-plate.
6. Install new split pin.
7. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
8. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.
9. Perform the Rudder Installation procedure on page 40 of this chapter.
10. Verify rigging in accordance with Section 27-90 Rigging Procedures on page 103 this chapter.

Section 20-02 Rudder Inspection

Inspect the following items on the rudder for any abnormalities.

- Check the hinge pin for freeplay between the pin and teeth. There is to be no freeplay between the pin and the teeth.
- Check attachment to fin and make sure the pushrod attachment meets clearance requirements. Clearance requirements are met if there are no visible signs of chafing or binding between the push rod and the composite surface. See Figure 27-45 for details. If chafing and/or binding is detected, go to the Extending the Closeout Hole for the Rudder procedure on page 57 of this chapter.

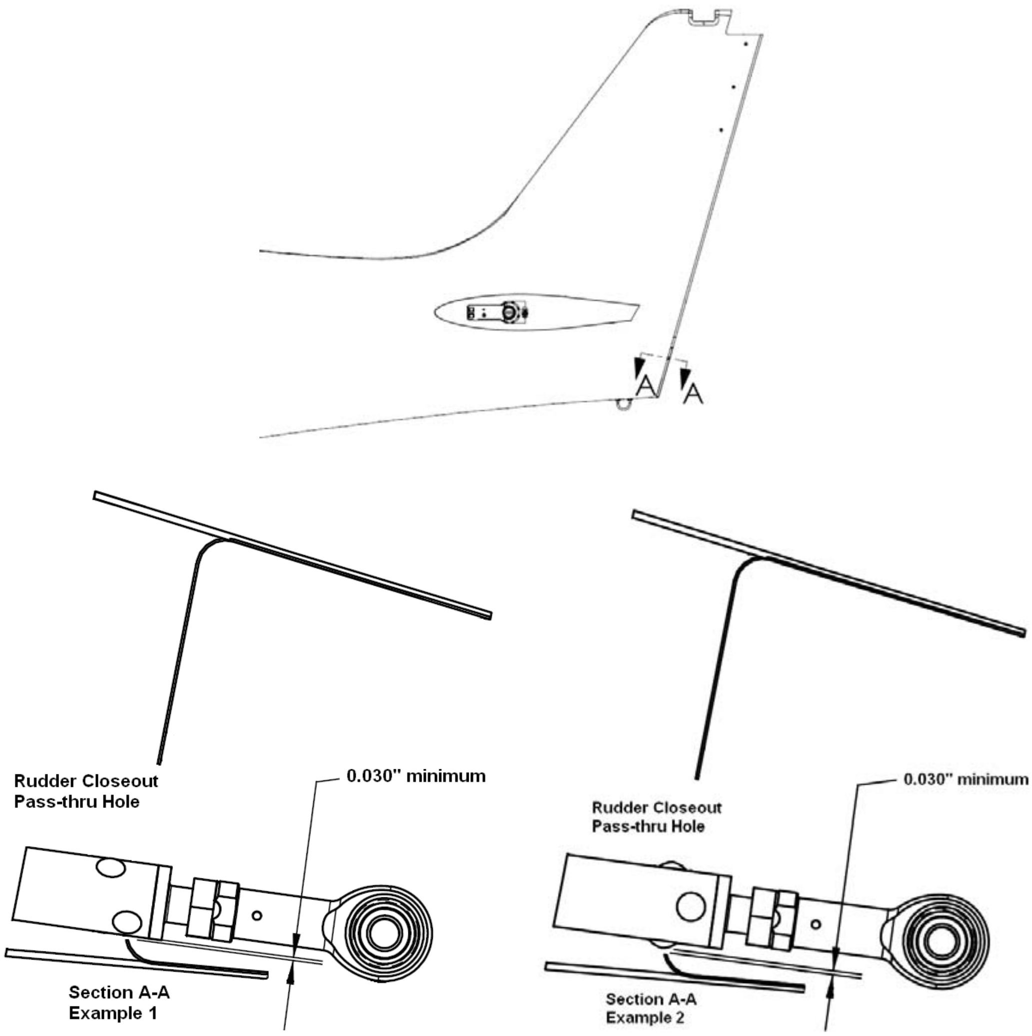


Figure 27-45 Rudder Pushrod Clearance

- Check lower composite surface for cracking. If the cracking is from over rotation of the airplane during flare and landing, the composite fairing needs replacing. Inspect the control surface and control circuit for damage that may be hidden that resulted from the impact. Contact Liberty Aerospace, Inc. for inspection details.

Section 20-03 Troubleshooting

Table 27-4 contains information to aid in troubleshooting issues with the rudder system.

Complaint	Possible Cause	Remedy
Stiffness or binding in rudder system	Dirty, “dry,” or defective rod end bearings in rudder linkage	Lubricate with Corrosion X or replace as necessary
	Dirty or dry components in rudder pedal assy.	Lubricate with Corrosion X
Airplane requires constant rudder pressure for coordinated flight	Incorrectly adjusted rudder tab	Readjust rudder tab
	Improper rigging	Verify rigging in accordance with Section 27-90 Rigging Procedures on page 103 this chapter
Rudder pedal position (leg length) difficult to adjust	Dirty or inadequately lubricated threaded adjustment rod or adjustment nut	Clean and lubricate with Corrosion X as necessary
Binding or chaffing of the rudder pushrod against the composite fuselage.	Pushrod out of alignment	Check the alignment, check the pushrod, if pushrod is damaged, replace pushrod.
	Pushrod opening in the fuselage is too narrow	Perform the Extending the Closeout Hole for the Rudder procedure on page 57 of this chapter.

Table 27-4 Rudder System Troubleshooting

Section 27-30 Stabilator and Tab

The Liberty XL-2 airplane uses an all-flying stabilator with an anti-servo/trim tab on the trailing edge. Figure 27-46 shows the various parts in the stabilator system.

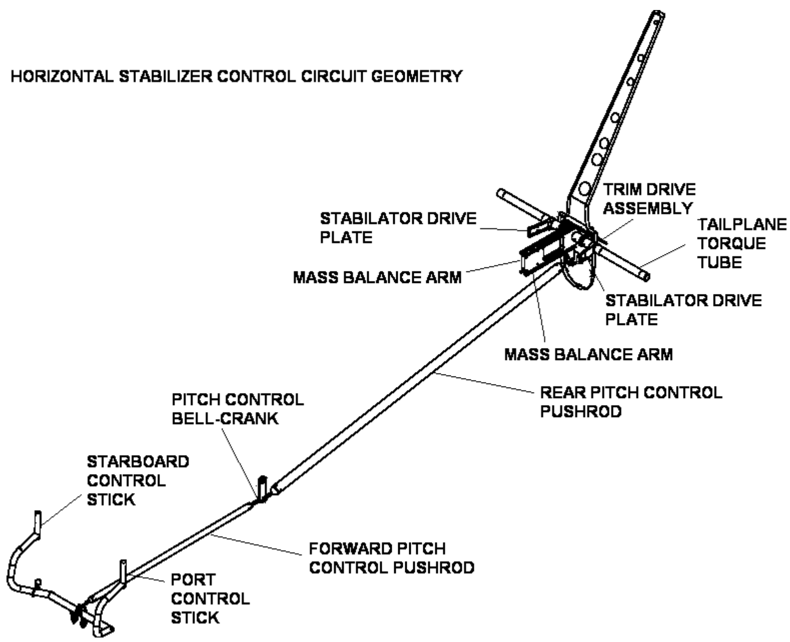


Figure 27-46 Horizontal Stabilizer Control Circuit Geometry

The tab has the following functions:

- To stabilize the stabilator and provide stick force feedback to the pilot.
- Longitudinal (pitch) trim.

A fore and aft deflection of the cockpit control stick(s) causes an up/down deflection of the entire stabilator. An anti-servo/tab on the trailing edge of each stabilator deflects in the same direction as stabilator movement.

The zero-deflection point of the tab is the point at which it is faired with the trailing edge of the stabilator. Trim tab positions that may be set are as follows:

- Nose Up 5-degrees upward deflection
- Takeoff 0-degrees deflection
- Nose Down 5-degrees downward deflection

A fore/aft movement of the control stick moves the transverse yoke tube between the control sticks forward and aft. This motion is transmitted via a pushrod in the cockpit center console. Idler bell-cranks mounted in the fuselage center section, and a second pushrod in the aft fuselage operates a torque tube mounted in bearings in the aft fuselage. The left and right stabilator are mounted on this torque tube. A mass balance for the stabilator is mounted to the torque tube, and extends forward from it, inside the aft fuselage.

Deflection of the anti-servo/trim tab is affected by movable plates secured to the inboard rib of each horizontal stabilizer and connected via a link to the tab. Holes in these plates mate with drive pins extending from the fuselage, which are positioned by a linkage to a gear-head motor (trim actuator) installed in the aft fuselage.

Access to the control stick pitch bearing and forward pitch pushrod is gained by removal of the fuselage belly panel. Access to the idler bell-crank and forward end of the aft pushrod is gained by removing the panel in the floor of the baggage compartment. A removable access panel is provided on the starboard side of the lower aft fuselage to allow access to the trim servo.

Section 30-01 Stabilator System Procedures

This section contains the procedures for the stabilator system. These procedures include the following:

Procedure	Page
Stabilator Removal	63
Stabilator Installation	64
Mid-fuselage Pitch Control Idler-plate Removal	66
Mid-fuselage Pitch Control Idler-plate Installation	69
Anti-Servo/Trim Tab Removal	70
Anti-Servo/Trim Tab Installation	71
Pitch Trim Actuator Removal	72
Pitch Trim Actuator Installation	73

NOTE

Removal and replacement of the forward and rear pitch control pushrods is not anticipated during routine maintenance. In the event that forward and rear pitch control pushrods need to be removed from the airplane, contact Liberty Customer Support for the recommended procedure.

STABILATOR REMOVAL

Perform this procedure to remove the stabilator.

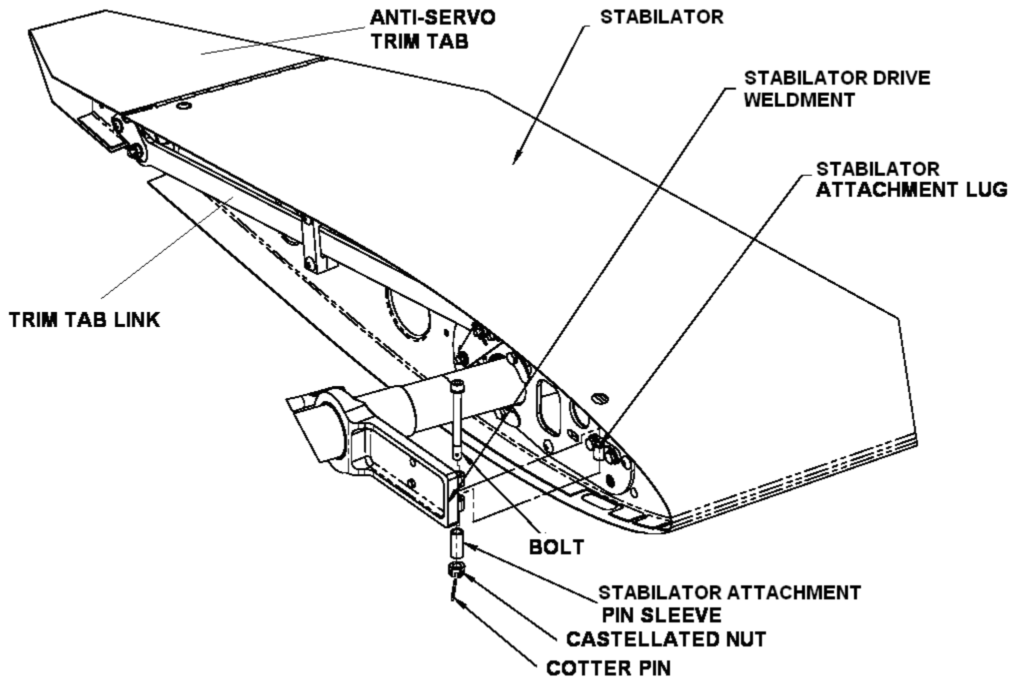


Figure 27-47 Stabilator Assembly

CAUTION

Take precautions to prevent damage to the surface airfoil when removing bolt assembly.

1. Remove and discard split pin from castellated nut on stabilator attach bolt.
2. Remove castellated nut and sleeve.
3. Remove bolt (it may be necessary to drive out from below by tapping gently with appropriately sized drift punch).
4. Remove stabilator by pulling straight away from fuselage.

STABILATOR INSTALLATION

Perform this procedure to install the stabilator.

1. Prior to installation, inspect the torque tube for any type of corrosion or oxidation. If the torque tube does not have a primer coat refer to Chapter 55 – *Stabilators* for the procedure to apply a primer coat to the torque tube.
2. Clean any foreign materials from torque tube surfaces. Apply Corrosion X Aviation (or corrosion inhibitor conforming to MIL-C-81309E, Type II) to the areas as shown in Figure 27-48. Apply LPS 3 inhibitor (or corrosion preventive compound conforming to MIL-PRF-16173E Grade 2) to bearing surfaces.

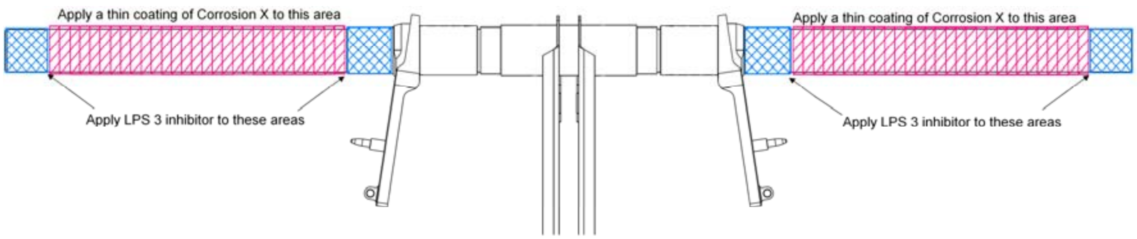


Figure 27-48 Location for the Corrosion Inhibitors

3. Place stabilator over torque tube and slide inboard until torque tube engages stabilator internal structure.
4. Position anti-servo/tab so that hole on tab drive plate (aft of torque tube) engages tab drive spigot on fuselage.

NOTE

Anti-servo/trim tab hinge is located on stabilator top surface.

5. Continue to slide stabilator inboard parallel with drive arm, so that hole on stabilator root rib (forward of torque tube) engages spigot on stabilator drive plate on fuselage and stabilator attachment bolt lug fits between corresponding lugs on drive plate.

NOTE

Wet assemble the securing bolt assembly using CA-1000.

6. Install the stabilator securing bolt from top to bottom through hole in inboard stabilator skin.
7. Install sleeve and castellated nut; hand tight, do not torque above 10 in-lbs, replace split pin.

CAUTION

No lateral, fore/aft, or up/down movement should be detected. If any lateral movement greater than .006” is detected, contact Liberty Aerospace Customer Support for additional instructions. If any fore/aft or up/down movement is detected, replace stabilator bearing(s) in accordance with procedures in Chapter 55 - Stabilizers.

8. Verify rigging in accordance with Section 27-90 Rigging Procedures on page 103 this chapter.

MID-FUSELAGE PITCH CONTROL IDLER-PLATE REMOVAL

Perform this procedure to remove the mid-fuselage pitch control idler plate.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

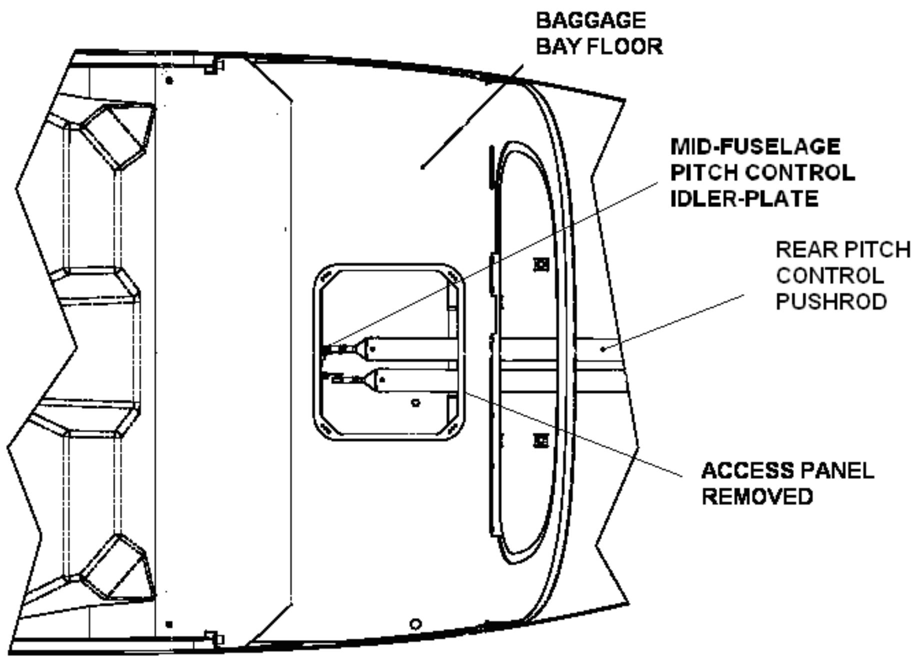


Figure 27-49 Mid-fuselage Pitch Control Idler-plate Location

1. Remove belly panel (see Chapter 53 - *Fuselage*).
2. Refer to Figure 27-50. Remove split pin (1) and discard. Remove the nut (2), washers (3), spacer (4), and bolt (5). Separate rear pitch control pushrod from mid-fuselage pitch control idler-plate.

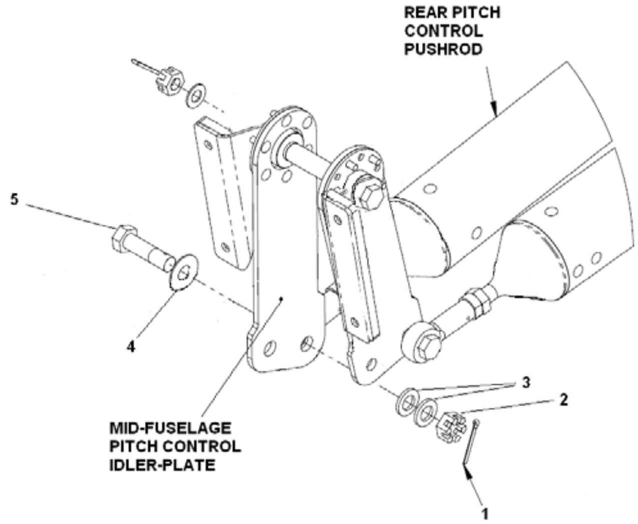


Figure 27-50 Rear Pitch Control Pushrod to Mid-fuselage Pitch Control Idler-plate

3. Refer to Figure 27-51. Remove the split pin (6) and discard. Remove the nut (7), washers (8 & 9), and spacer (10). Note the locations of washers and spacer.
4. Remove bolt (12) and washer (11) and disconnect forward pitch control pushrod from mid-fuselage pitch control idler-plate.

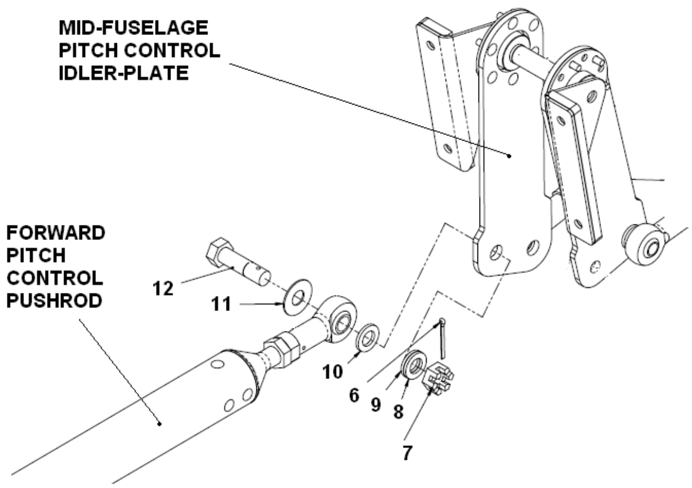


Figure 27-51 Forward Pitch Control Pushrod to Mid-fuselage Pitch Control Idler-plate

5. Refer to Figure 27-52. Remove split pin (13) and discard. Remove the nut (14), and washer (15) from bolt (17). Slide bolt out part way from mid-fuselage pitch control idler-plate enough to allow center spacer and idler plate to separate. Remove mid-fuselage pitch control idler-plate.

6. Check the center space remains on bolt and temporarily re-install the bolt into space frame connector and loose-fit nut for temporarily support to the rudder pushrods and idler plate.

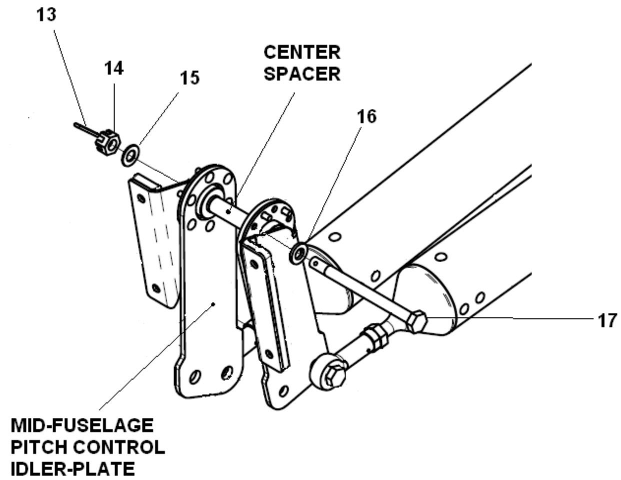


Figure 27-52 Mid-fuselage Pitch Control Idler-plate Removal and Installation

MID-FUSELAGE PITCH CONTROL IDLER-PLATE INSTALLATION

Perform this procedure to install the mid-fuselage pitch control idler plate.

CAUTION

To ensure that rigging is not disturbed, do not alter or adjust pushrod end adjustment jam nut.

1. Refer to Figure 27-52. Remove nut (14) that was placed on bolt (17) to temporarily support rudder pushrods and idler plate.
2. Partially withdraw bolt enough to position mid-fuselage pitch control idler-plate and re-insert bolt.
3. Refer to Figure 27-52. Install washer (15) and nut (14).
4. Torque per Chapter 20 - *Standard Practices*. Install a new split pin (13).
5. Refer to Figure 27-51. Connect forward pitch control pushrod to mid-fuselage pitch control idler-plate using bolt (12), washer (11), spacer (10), washers (8 & 9) - installing washers and spacers as noted during removal - and nut (7).
6. Torque per Chapter 20 - *Standard Practices*. Install a new split pin (6).
7. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
8. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.
9. Refer to Figure 27-50. Connect rear pitch control pushrod to mid-fuselage pitch control idler-plate using bolt (5), spacer (4), washers (3), and nut (2).
10. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
11. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.
12. Verify rigging in accordance with Section 27-90 Rigging Procedures on page 103 this chapter.
13. Install belly panel (see Chapter 53 - *Fuselage*).

ANTI-SERVO/TRIM TAB REMOVAL

Perform this procedure to remove the anti-servo/trim tab.

NOTE

Anti-servo/trim tab hinge is located on stabilator top surface.

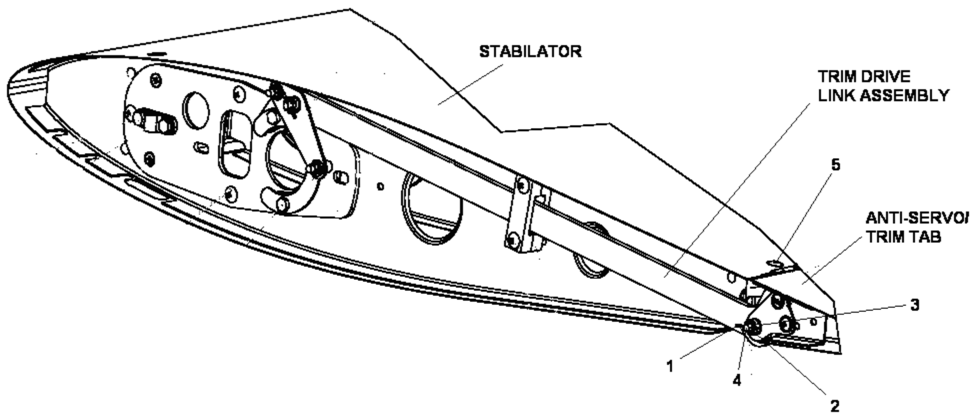


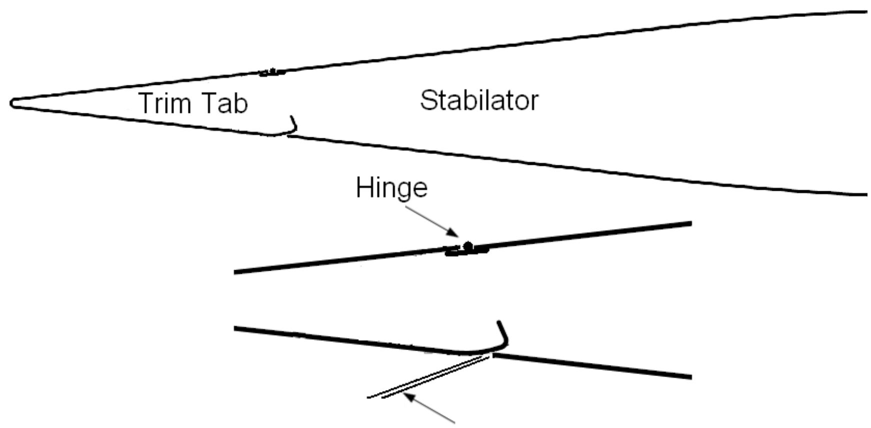
Figure 27-53 Stabilator and Anti-servo/Trim Tab

1. Remove Stabilator from airplane in accordance with the Stabilator Removal procedure on page 63 of this chapter. Place on workbench in inverted position.
2. Remove split pin (1) and discard. Remove the nut (2), washers (3), and screw (4). Disconnect link from anti-servo/trim tab. See Figure 27-53.
3. Remove five screws (5) and remove anti-servo/trim tab from stabilator.

ANTI-SERVO/TRIM TAB INSTALLATION

Perform this procedure to install the anti-servo/trim tab.

1. Install anti-servo/trim tab onto stabilator. Take care to position hinge of tab between outer skin of stabilator and underlying structure.
2. Install five screws (5) that fasten anti-servo/trim tab hinge to stabilator.
3. Connect link to anti-servo/trim tab using screw (4), washers (3), and castellated nut (2). Torque the nut in accordance with Chapter 20 - *Standard Practices*. Install a new split pin (1). See Figure 27-53.
4. Check minimum clearance 0.070 in to lower stabilator trailing edge and tails of rivets immediately forward of lower trailing edge at all ranges of travel on both trim tab and stabilator. See Figure 27-54 for details.



Minimum clearance 0.070in between the lower surface of the leading edge of the trim tab and the rivet tails in the lower surface of the trailing edge of the stabilizer.

Figure 27-54 Trim Tab Clearances

5. Install stabilator to airplane in accordance with the Stabilator Installation procedure on page 64 of this chapter.

PITCH TRIM ACTUATOR REMOVAL

Perform this procedure to remove the pitch trim actuator. Refer to Figure 27-55 during this procedure.

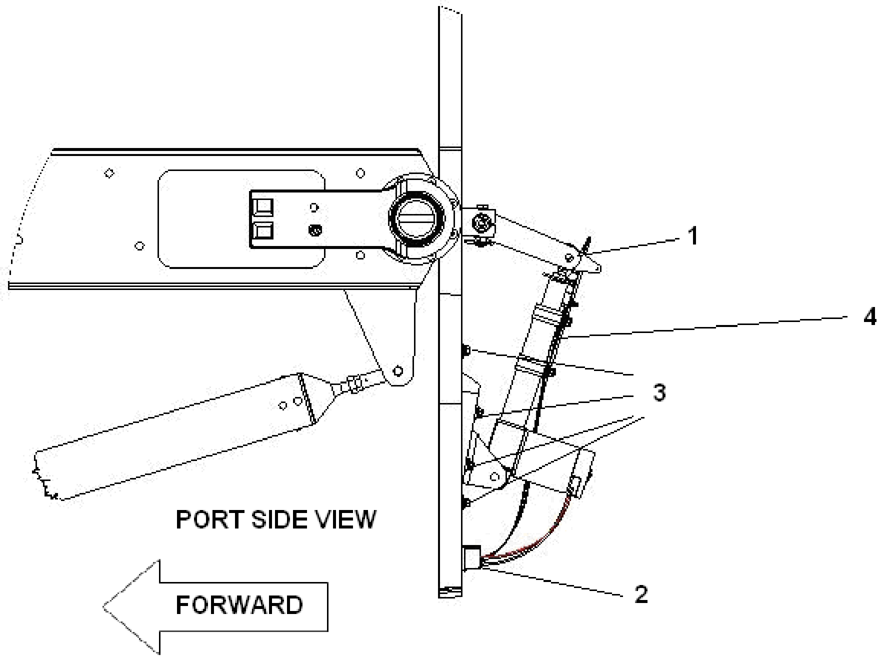


Figure 27-55 Pitch Trim Actuator and Linkage

1. Place the BAT master switch to ON.
2. Position the pitch trim to full nose-down position.
3. Place the BAT master switch is OFF.
4. Remove trim actuator access panel (rear-most panel on lower right side of aft fuselage).
5. Disconnect trim actuator electrical connector – P21 (1).
6. Remove bolt securing bearing end of trim actuator to trim linkage (2).
7. Remove bolt and any hardware securing motor end of trim actuator to airframe noting placement of bushings (3).
8. Carefully remove wire link between trim position sensors and trim linkage.
9. Remove actuator (4).

PITCH TRIM ACTUATOR INSTALLATION

Perform this procedure to install the pitch trim actuator.

1. Clean the hardware used to secure motor end of trim actuator to airframe.
2. Temporarily install trim actuator assembly in position.
3. Check the stabilator. Make sure that it is locked in neutral position.
4. Apply 12 Vdc to pins 1 and 2 of actuator electrical connector. Reverse polarity to reverse direction of motion.

NOTE

An alternate method is to temporarily connect electrical connector and operate cockpit trim switch.

5. Cycle trim motor so that trim tab travels fully in both directions and align trim tab with Stabilator (neutral).
6. Verify rigging in accordance with Section 27-90 Rigging Procedures on page 103 this chapter.
7. Adjust length of bearing end of servo by screwing or unscrewing bearing end as required. Tighten jam nut.

CAUTION

Use a carbide drill bit for drilling to prevent drill bit from breaking inside servo column.

8. If installing a new trim actuator, drill hole as shown in Figure 27-56 and drive safety pin in to keep column from moving out of place if jam nut loosens.

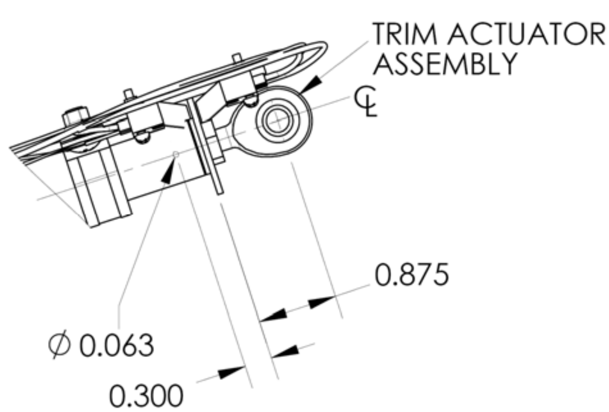


Figure 27-56 Trim Actuator Installation

9. Wet assemble bolt securing motor end of trim actuator to airframe bracket. Verify position of bushings and rod. See Figure 27-57.

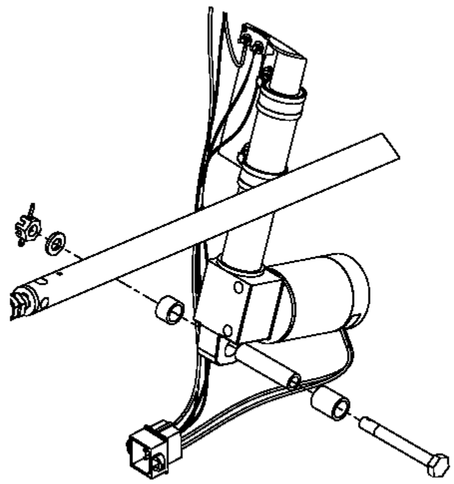


Figure 27-57 Trim Actuator to Airframe Installation

10. Re-install bolt securing bearing end of trim actuator to trim linkage and hand torque bolt, replace split pin.
11. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
12. Check for correct operation and freedom of movement. When alignment is correct, pushrod will freely rotate approximately 1/10th of a turn.
13. Connect electrical connector.
14. Apply power to airplane electrical system. Operate pitch trim over entire range, observing that anti-servo/tab moves down when trim switch is positioned in nose up direction. Verify correct indication of cockpit trim indicator. Move trim switch to nose down position and verify that anti-servo/tab moves in up direction. Verify correct indication of cockpit trim indicator. See Table 27-5

Trim Tab Switch Input	Expected Response
Forward Half Labeled 'NOSE DOWN'	Trim Tab Trailing Edge Deflects Upward
Aft Half Labeled 'NOSE UP'	Trim Tab Trailing Edge Deflects Downward

Table 27-5 Trim Tab Input / Response

CHECK DIRECTION OF ANTI-SERVO/TRIM TAB IS IN CORRECT RESPONSE FOR A GIVEN INPUT.

15. Install trim actuator access panel.

Section 30-02 Stabilator and Tab Inspection

Inspect the following items on the stabilator and tab for any abnormalities.

- Check for movement of the stabilator (with the yoke fixed). No lateral, fore/aft, or up/down movement should be detected. If any lateral movement greater than .006” is detected, contact Liberty Aerospace Customer Support for additional instructions. If any fore/aft or up/down movement is detected, replace stabilator bearing(s) in accordance with procedures in Chapter 55 - *Stabilizers*.
- Check the tab to horizontal stabilizer hinge pin and attachment, there shall be no visible free play

If either of these inspections fails, contact Liberty Aerospace, Inc. for further instruction.

Section 30-03 Troubleshooting

This section contains the troubleshooting information.

Issue	Possible Cause	Remedy
Binding or stiffness in pitch trim circuit	Rod-end bearings	Replace as necessary
	Torque tube not lubricated	Lubricate as necessary
Looseness in pitch control	Bearings or mounting bolts	Replace as necessary
	Scoring on trim drive to stabilator torque tube	Replace as necessary
Pitch trim inoperative	Pitch trim circuit breaker	Replace as necessary
	Pitch trim switch	Replace as necessary
	Pitch trim actuator	Replace as necessary
	Wiring	Repair
Un-commanded pitch trim movement	Pitch trim switch	Inspect full circuit to determine root cause
		Repair or replace as needed.
Pitch trim moves in one direction only	Pitch trim switch	Replace
Pitch trim indicator does not show correct trim position	Indicator	Replace indicator
	Sensor in trim actuator	Replace trim actuator
	Wiring	Repair

Table 27-6 Stabilator Troubleshooting Chart

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Section 27-40 Control Yoke

The control yoke provides pilot, and/or copilot, input to the ailerons for roll control, and stabilators for pitch control. The yoke assembly consists of a weldment providing physical attachments for the forward pitch control pushrod and aileron torque tube. The pilot control stick is part of the weldment. The removable co-pilot control stick attaches to the yoke to complete the assembly.

The control yoke also has the push-to-talk switch(es) that control the communication receivers. The wires for the push-to-talk switch threads through the yoke controls to individual connectors (J26 for the pilot and P28 for the copilot).

Section 40-01 Control Yoke Procedures

This section contains the procedures for the yoke control.

CONTROL YOKE BOOT REMOVAL

Perform this procedure to remove the yoke boot. Refer to Figure 27-58 during this procedure.

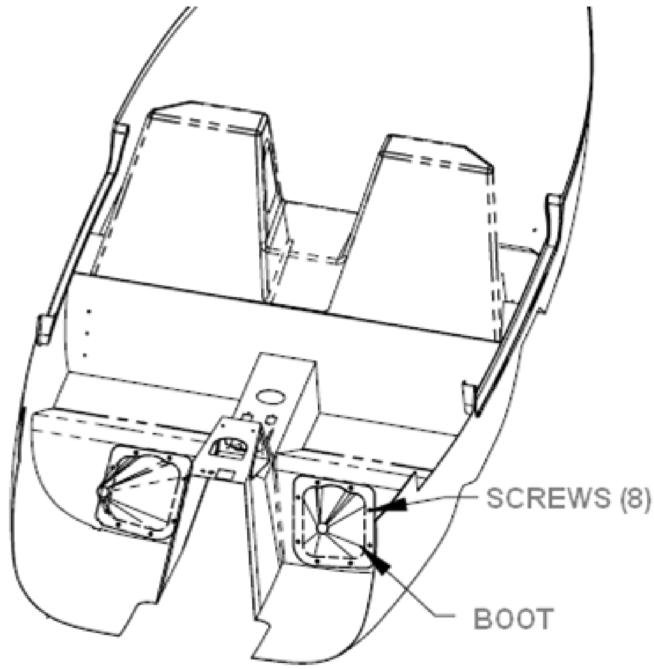


Figure 27-58 Control Yoke Boot

1. Remove eight bolts fastening control yoke boot to fuselage.
2. Carefully cut the two tie wraps that secure the boot to the control yoke. Remove and discard tie wraps.
3. Maneuver control boot over and off control stick.

CONTROL YOKE BOOT INSTALLATION

Perform this procedure to install the control yoke boot. Refer to Figure 27-58 during this procedure.

1. Maneuver control boot over and onto control stick.
2. Install eight bolts to fasten control yoke boot to fuselage.
3. Carefully thread two tie warps through the top of the control yoke boot.
4. Thread the tie wraps through the hole in the control yoke. See Figure 27-59.
5. Close the tie wraps to secure the control yoke boot to the control yoke.

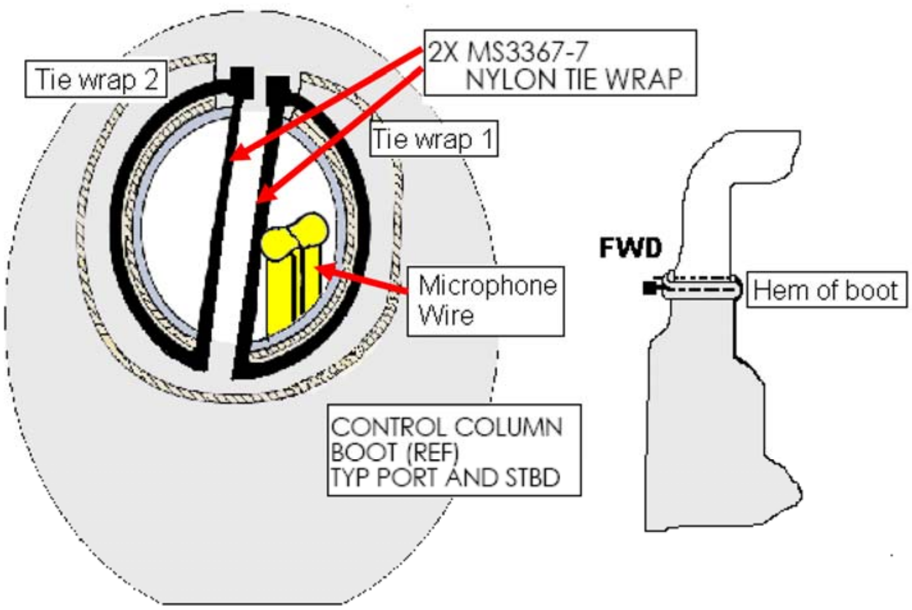


Figure 27-59 Installation of the Tie Wraps to Secure the Boot to the Control Yoke

STARBOARD (CO-PILOT) CONTROL STICK REMOVAL

Perform this procedure to remove the starboard (co-pilot) control stick. Refer to Figure 27-60 during this procedure.

NOTE

If control stick is to remain removed from airplane during flight, install the ground cable, bolt (4), washer (3), nut (2), and new split pin. Also, remove the boot from control stick and install the boot back on to the airplane.

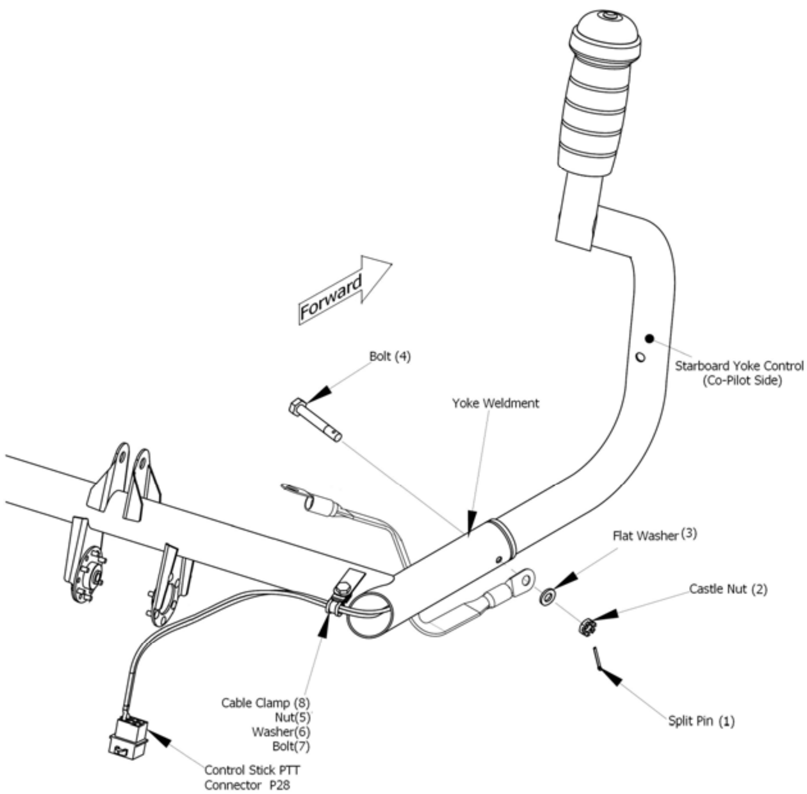


Figure 27-60 Starboard Control Stick Removal and Installation

1. Remove belly panel (see Chapter 53 - *Fuselage*).
2. Remove split pin (1) and discard. Remove the nut (2), washer (3), and bolt (4) and ground cable from starboard control stick.
3. Disconnect starboard control stick push-to-talk connector – P28.
4. Remove self-locking nut (5), washer (6), and bolt (7) from cable clamp (8). Remove clamp.
5. Remove starboard control yoke boot in accordance with the Control Yoke Boot Removal procedure on page 78 of this chapter.
6. Taking care not to damage push-to-talk cable, remove control stick.

STARBOARD (CO-PILOT) CONTROL STICK INSTALLATION

Perform this procedure to install the starboard (co-pilot) control stick. Refer to Figure 27-60 during this procedure.

1. Maneuver control stick into position.
2. Feed PTT cable through weldment.
3. Install control stick using bolt (4), washer (3), ground cable, and castellated nut (2). Torque the nut in accordance with Chapter 20 - Standard Practices. Install a new split pin (1).
4. Place cable clamp (8) around PTT cable and install clamp using bolt (7), washer (6), and self-locking nut (5).
5. Connect PTT connector P28.
6. Check for proper operation of control stick.
7. Check for proper operation of push-to-talk switch.
8. Install control stick boot in accordance with the Control Yoke Boot Installation procedure on page 79 of this chapter.
9. Install belly panel in accordance with Chapter 53 - *Fuselage*.

PUSH-TO-TALK SWITCH REMOVAL

Perform this procedure to remove the push-to-talk switch assembly. This procedure can be used for either the pilot's or the copilot's push-to-talk switch assembly. Refer to Figure 27-61 for details of this procedure.

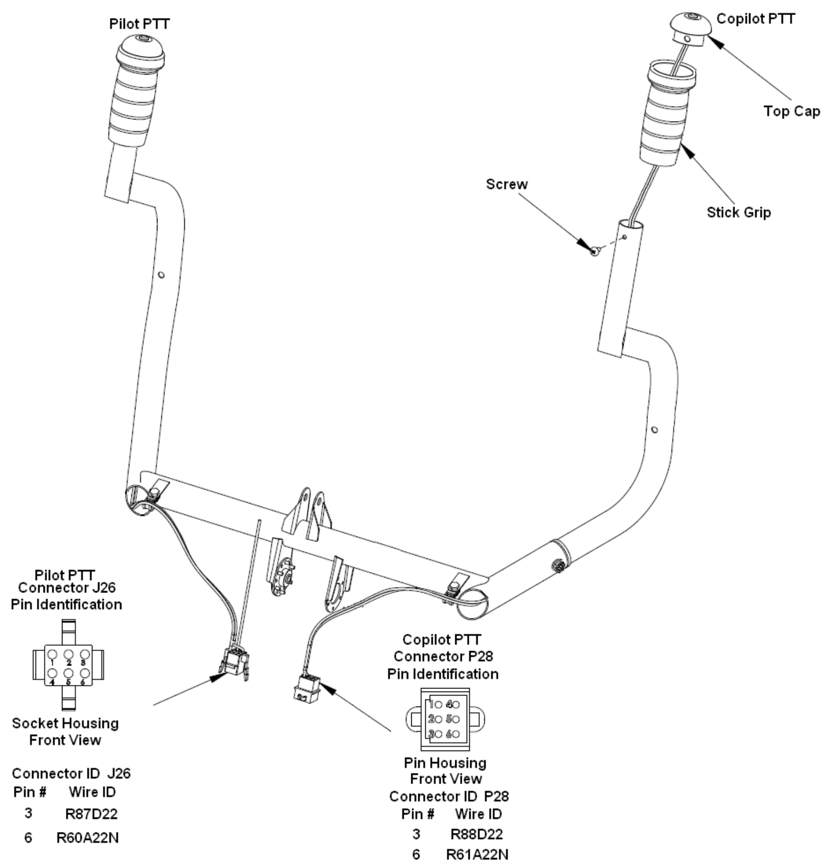


Figure 27-61 Details of the Pilot's/Copilot's Push-To-Talk Switch Assembly

1. Remove belly panel (see Chapter 53 - *Fuselage*).
2. Disconnect the appropriate connector.
3. Remove the wire and pins from the connector shell.
4. Attach a fish wire or string to the wire and pins.
5. From the cockpit, gently slide the stick grip down at the top.
6. Remove the screw securing the top cap.
7. Feed the wires up through the yoke control, pulling the fish wire through at the same time.

This completes the Push-To-Talk Switch Removal procedure.

PUSH-TO-TALK SWITCH INSTALLATION

Perform this procedure to install the push-to-talk switch assembly. This procedure can be used for either the pilot's or the copilot's push-to-talk switch assembly. Refer to Figure 27-61 for details of this procedure.

1. Remove belly panel (see Chapter 53 - *Fuselage*).
2. Attach the wire and pins from the replacement top cap switch assembly to the fish wire pulled during the removal procedure.
3. Using the fish wire, pull the wire and pins through the yoke control.
4. Gently slide the stick grip down at the top.
5. Install the screw removed in step 6 of the Push-To-Talk Switch Removal procedure to secure the top cap.
6. Insert the wire and pins in to the connector shells as shown in Figure 27-61.
7. Install belly panel (see Chapter 53 - *Fuselage*).

This completes the Push-To-Talk Switch Installation procedure.

CONTROL YOKE ASSEMBLY REMOVAL

Perform this procedure to remove the control yoke assembly.

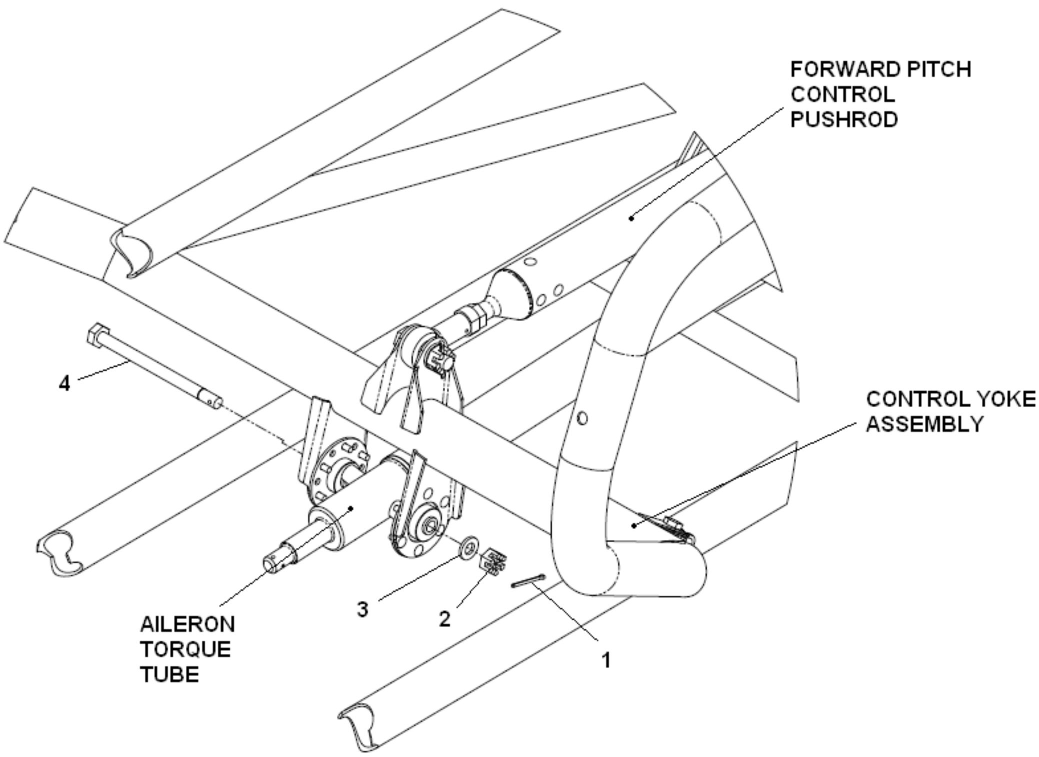


Figure 27-62 Control Yoke Removal and Installation

1. Remove starboard control stick in accordance with the Starboard (Co-pilot) Control Stick Removal procedure on page 80 of this manual.
2. Disconnect port control stick PTT connector – P26.
3. Remove port control yoke boot in accordance with the Control Yoke Boot Removal procedure on page 78 of this manual.
4. Remove split pin (1) and discard. Remove the nut (2), washer (3), and bolt (4) from forward pitch control pushrod. Separate pushrod from yoke assembly.

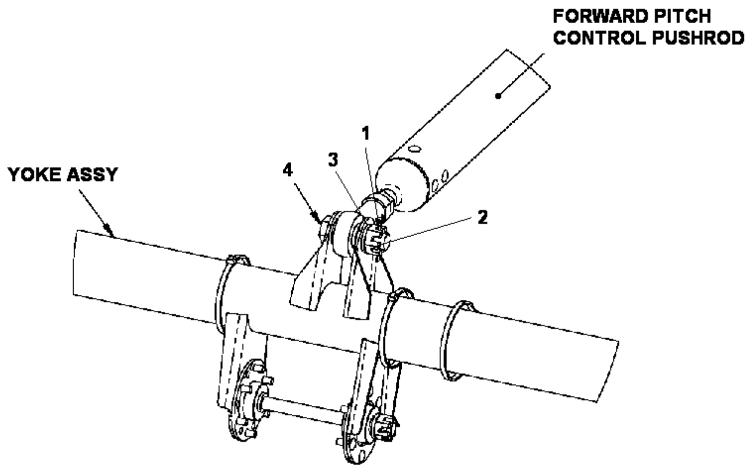


Figure 27-63 Forward Pitch Control Pushrod to Yoke Assembly

5. Remove split pin (1) and discard. Remove the nut (2), washer (3), and bolt (4) connecting aileron torque tube to control yoke assembly and separate control yoke from torque tube.

CAUTION

Extreme care must be taken while maneuvering yoke assembly out of airplane so as not to damage instrument panel or fuselage.

6. Carefully maneuver yoke assembly down through fuselage opening and out of airplane.

CONTROL YOKE ASSEMBLY INSTALLATION

Perform this procedure to install the control yoke assembly.

Extreme care must be taken while maneuvering yoke assembly into airplane so as not to damage instrument panel, fuselage, or control yoke assembly.

1. Carefully maneuver yoke assembly up into airplane and through fuselage opening
2. Install bolt (4), washer (3), and castellated nut (2) to connect aileron torque tube to control yoke assembly. Torque the nut in accordance with Chapter 20 - *Standard Practices*.
3. Install a new split pin (1).
4. Place forward pitch control pushrod into position on yoke assembly and install using bolt (4), washer (3), and nut (2).
5. Torque the nut in accordance with Chapter 20 - *Standard Practices*. Install new split pin (1).
6. Install port control yoke boot in accordance with the Control Yoke Boot Installation procedure on page 79 of this manual.
7. Connect port control stick push-to-talk connector – P26.
8. Install starboard control stick in accordance with the Starboard (Co-pilot) Control Stick Installation procedure on page 81 of this manual.
9. Install starboard control stick boot in accordance with the Control Yoke Boot Installation procedure on page 79 of this chapter.

Section 27-50 Flaps

The airplanes single-slotted wing flaps are of conventional aluminum construction. Each flap attaches to the wing by three hinges set approximately six inches below the wing lower surface.

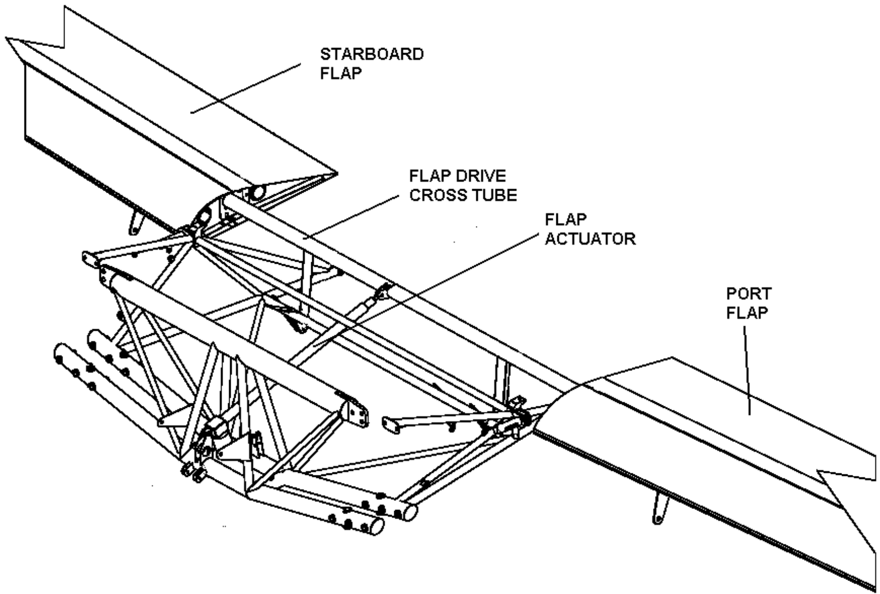


Figure 27-64 Flaps Control Circuit Geometry

A single electrical flap actuator operating a flap drive cross-tube between the port and starboard flap drives the flaps. The "spigot" in the root rib of each flap engages with the spherical bearing at each end of the flap drive cross tube to provide automatic disconnection/connection of the flap drive when the wing is removed and installed. The available off-axis play of the bearings accommodates changes in geometry as the flaps extend and retract. A separate flap position sensor provides flap position indication. Removal of the fuselage belly panel provides access to the flap actuator, flap position sensor, and flap drive cross-tube.

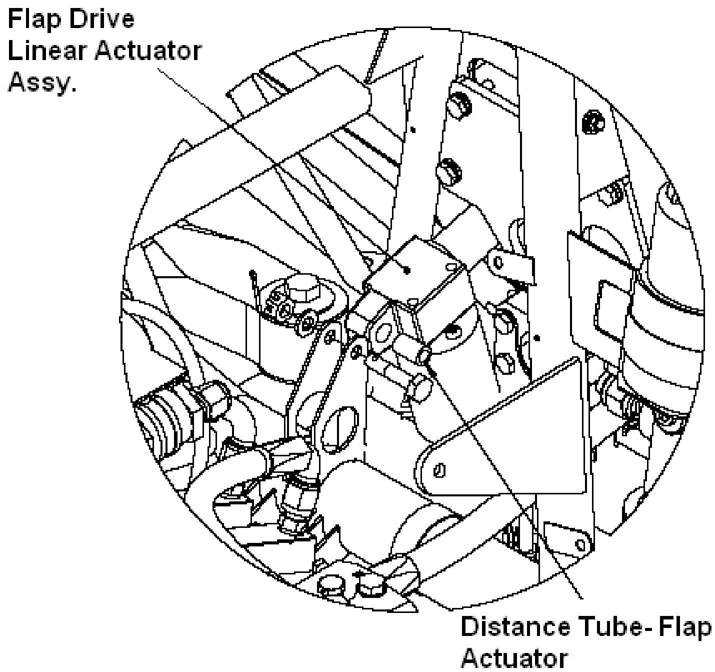


Figure 27-65 Flap Drive Linear Actuator Assembly

Section 50-01 Flap System Procedures

This section contains the procedures for the flap systems.

FLAP REMOVAL

1. Fully extend flap.
2. Support flap at each end. Remove split pins and discard.
3. Remove the nuts, and washers from three hinge bolts.
4. Remove hinge bolts, and spacers, noting their locations.

NOTE

It is necessary to move aileron to full up or neutral position to allow flap to be removed.

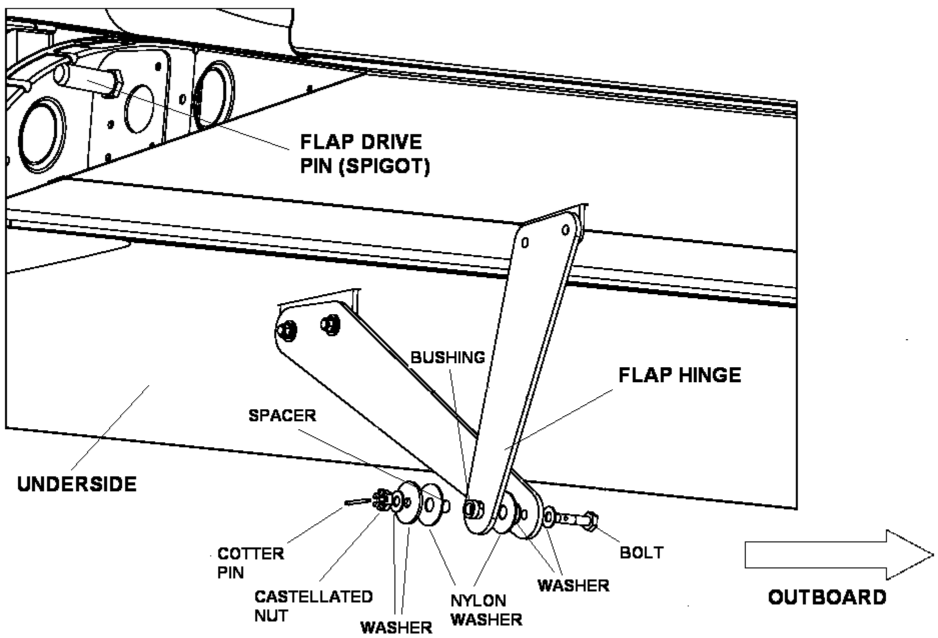


Figure 27-66 Flap Hinge Assembly

NOTE

Note the orientation of the hinge plates. The wing side of the hinge plate is outboard of the flap side of the hinge plate.

5. The flap now rotates freely around the point at which the flap spigot engages the flap drive cross tube bearing. Rotate flap slightly to separate hinges then remove flap.

FLAP INSTALLATION

1. With power applied to electrical system hold flap switch in DOWN position until flap indicator shows full down and flap actuator stops moving to check that the flap drive tube is fully extended.
2. Support flap at both ends. Align drive spigot in flap root rib with bearing in flap drive cross tube and slide flap inboard to engage spigot.

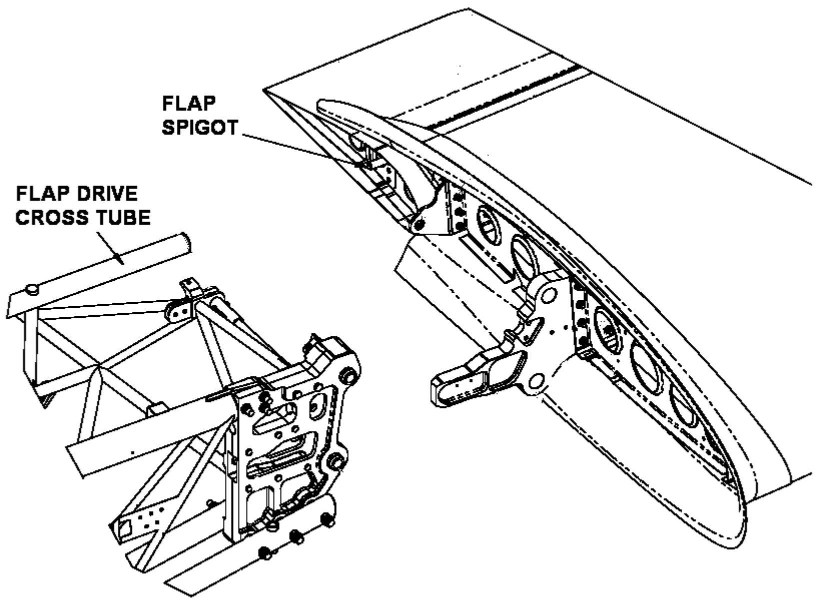


Figure 27-67 Flap Drive Cross Tube and Spigot

3. Rotate flap around drive tube axis until flap hinge halves align with corresponding wing hinge halves.

NOTE

Hinge half attached to flap must be inboard of hinge half attached to wing structure).

4. Install bolts, spacers, and washers in three hinges. Install castellated nuts on hinge bolts and torque the nuts in accordance with Chapter 20 - *Standard Practices*.
5. Install new split pins in three hinge bolts.
6. Check minimum tolerances as defined in Table 27-7 and Figure 27-68.

Clearance Description	1653-lbs	1750-lbs
Top of flap skin to wing T/E Flap 0º and in transition (A)	0.050”	0.070”
Flap Rivet tails to wing T/E Flap 0º and in transition (B)	0.080”	0.100”
Rivet heads (Rib 0 thru Rib 7) and Flap leading edge Flap 0º (C)	0.050”	0.070”
Flap lower skin to wing (T/E-lower) Flap 0º and in transition (D)	0.025”	0.025”

Table 27-7 Flap Clearance Tolerances

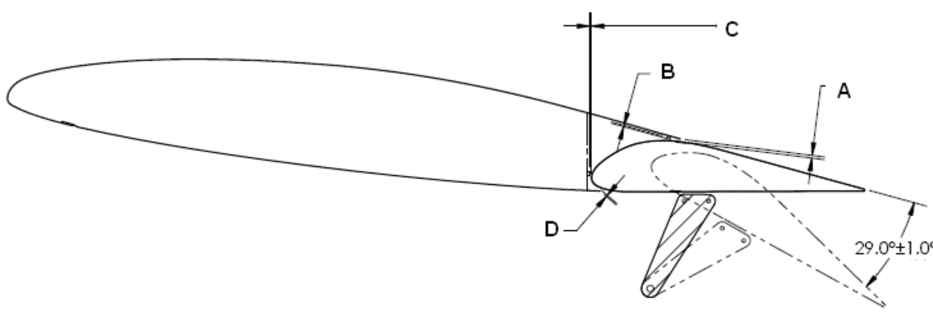


Figure 27-68 Wing-Flap Closeout

NOTE

For aircraft certified at a gross weight of 1750-lbs – It may be necessary to open top of wing-flap closeout to obtain minimum clearances. Perform the following in order:

7. Remove flap.
8. Clamp flap trailing edge uniformly and continuously (see 0).

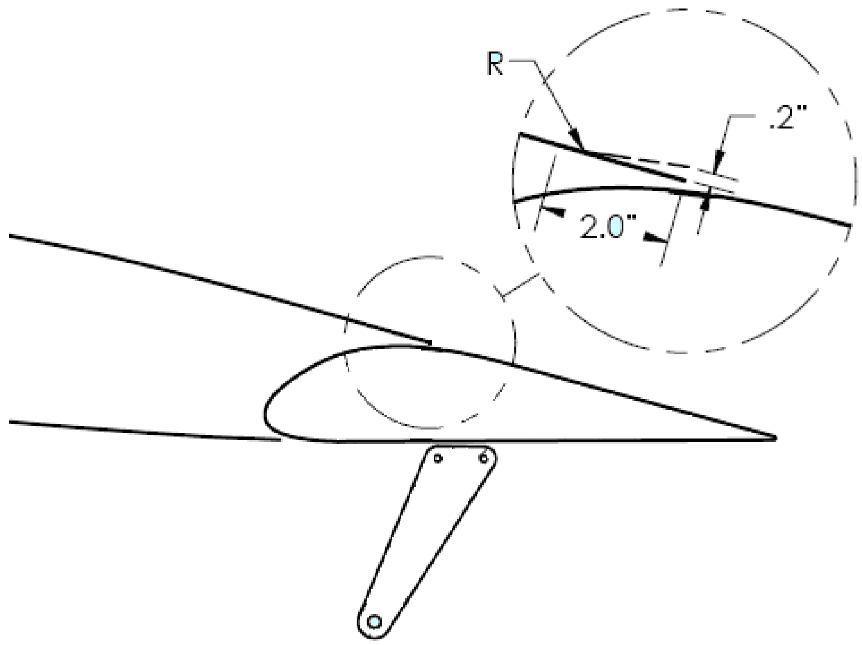


Figure 27-69 Opening "C" Channel of Wing-Flap Closeout

When bending trailing edge up, do not exceed 0.2" over a distance of 2" measured along wing chord. Bend radius 'R' must be gradual, continuous and not leave tool marks, significant surface impression, or any surface cracks.

9. Bend the trailing edge upward slowly in small increments.
10. Re-attach flap and check clearances.
11. Repeat steps 6 thru 10 until minimum clearances are obtained.
12. Record the clearances.
13. Operate flap over full range; verify correct operation and freedom of movement.
14. Verify rigging in accordance with Section 27-90 Rigging Procedures on page 103 this chapter.

FLAP ACTUATOR REMOVAL

Perform this procedure to remove the flap actuator. See Figure 27-70.

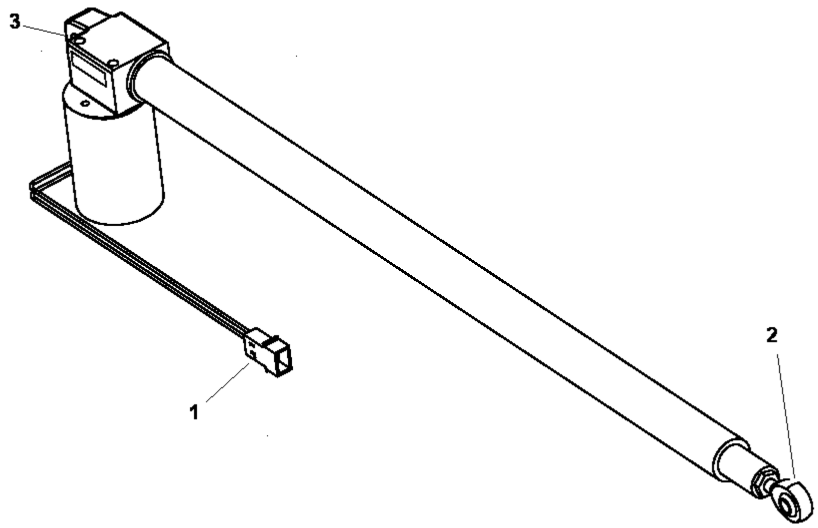


Figure 27-70 Flap Actuator

1. Fully extend flap.
2. Remove belly panel (see Chapter 53 - *Fuselage*).
3. Disconnect flap actuator electrical connector (1).
4. Remove bolt securing bearing end of actuator (2) to flap cross tube.

NOTE

Flap will move slightly beyond fully extended position when actuator is disconnected from cross tube. Support flap as required.

5. Remove bolt securing forward end of actuator (3) to airplane space frame structure; remove actuator.

FLAP ACTUATOR INSTALLATION

Perform this procedure to install the flap actuator. See Figure 27-70

1. Position flap actuator to fully extended position by applying 12 Vdc to pins 1 and 2 of electrical connector; reversing polarity will reverse motion.

NOTE

An alternate method is to temporarily connect electrical connector and operate cockpit trim switch.

2. Install bolt securing forward end of actuator (3) to airplane space frame structure.
3. Support flap so that bearing end of actuator (2) can be positioned in fitting on flap cross tube, and install bolt and split pin.
4. If installing a new flap actuator, drill hole as shown in Figure 27-71 and drive safety pin in to keep column from moving out of place if jam nut loosens. Orient the new pin left to right (horizontally). Do not mount the pin vertically (up to down).

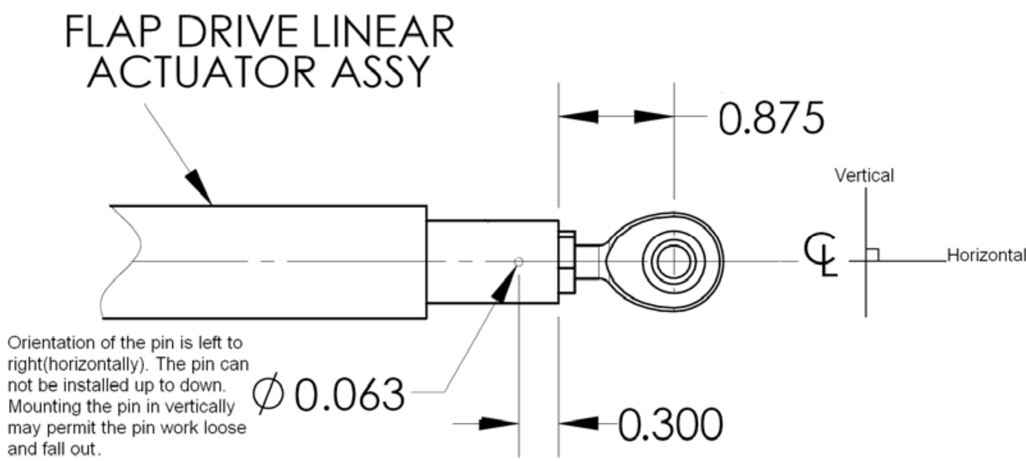


Figure 27-71 Flap Actuator Installation

5. Connect flap actuator electrical connector (1).
6. Operate flap over full range; verify correct operation and freedom of movement.
7. Verify inner bearing race is clamped, and outer race is free to rotate through its full range.
8. Install belly panel (see Chapter 53 - *Fuselage*).

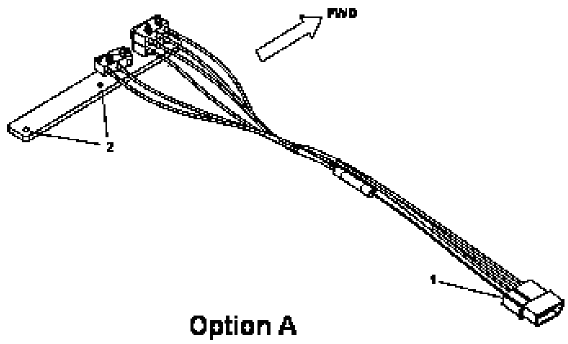
FLAP POSITION SENSOR REMOVAL

Perform this procedure to remove the flap position sensor.

NOTE

For adjustment of flap sensor indicator refer to 135A-911-050 Liberty XL-2 Rigging Procedure for appropriate steps.

1. Remove belly panel (see Chapter 53 - *Fuselage*).
2. Disconnect electrical connector (1) at flap position sensor.



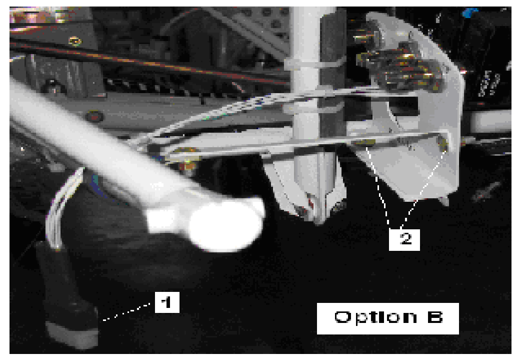


Figure 27-72 Flap Position Sensor

3. Remove bolts securing sensor to airframe (2); remove sensor.

FLAP POSITION SENSOR INSTALLATION

Perform this procedure to install the flap position sensor.

1. Install sensor and install bolts securing sensor to airframe (2).
2. Connect electrical connector to flap position sensor (1).
3. Install belly panel.

FLAP INSPECTION

Scheduled inspections as outlined in Chapter 05 - *Time Limits/Maintenance Checks/Inspection Intervals* of this manual include the scheduled maintenance procedures for the flaps. Those inspection and maintenance procedures are mandatory to support the continued airworthiness of the Liberty XL-2. In addition, the following inspections are recommended when performing flap inspection:

1. With the wings installed on the airplane, check the flap for lateral movement. The flap should freely move laterally without coming in to contact with either the fuselage or the aileron.

2. With the flap pulled fully away from the fuselage, check the clearance of the flap to the aileron. This clearance must be a minimum of 0.20 in (0.30 in maximum) between flap and aileron. If the clearance between the flap and the aileron is less than 0.20 in (with the flap pulled fully away from the fuselage), adjust the separation by adding a shim. This gap must be maintained throughout the travel of the flap.

CAUTION

If it is required to add shims to get the clearance between the flap and the aileron, after adding the shim, recheck the lateral movement of the flap. Do not put in any shims if there is no lateral movement of the flap. Contact the Customer Service department of Liberty Aerospace, Incorporated.

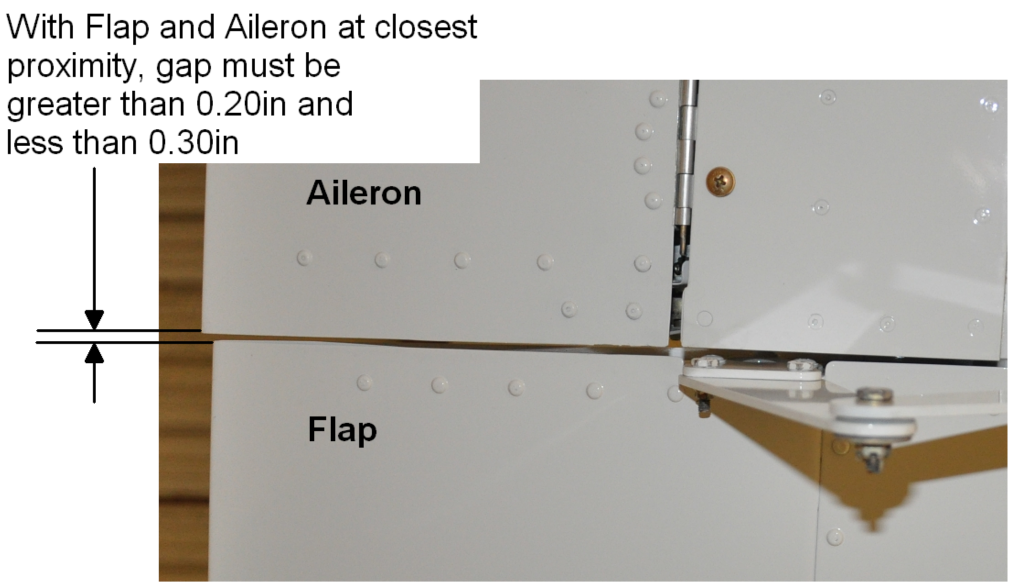


Figure 27-73 Clearance between Flap and Aileron

3. Check there is a clearance fit between flap drive pin, located on inboard rib of flap, and flap drive bearing, located on chassis. Pin fit should allow no binding through normal movement of flap travel in inboard/outboard direction. Keep area greased for smooth opportunity.

NOTE

Any binding may cause lateral bending of flap hinges under load as stated in (CSB-07-002).

4. With the flap pulled fully towards the fuselage, check the clearance of the flap to the fuselage. This clearance must be a minimum of 0.20 in (0.30 in maximum) between flap and fuselage.

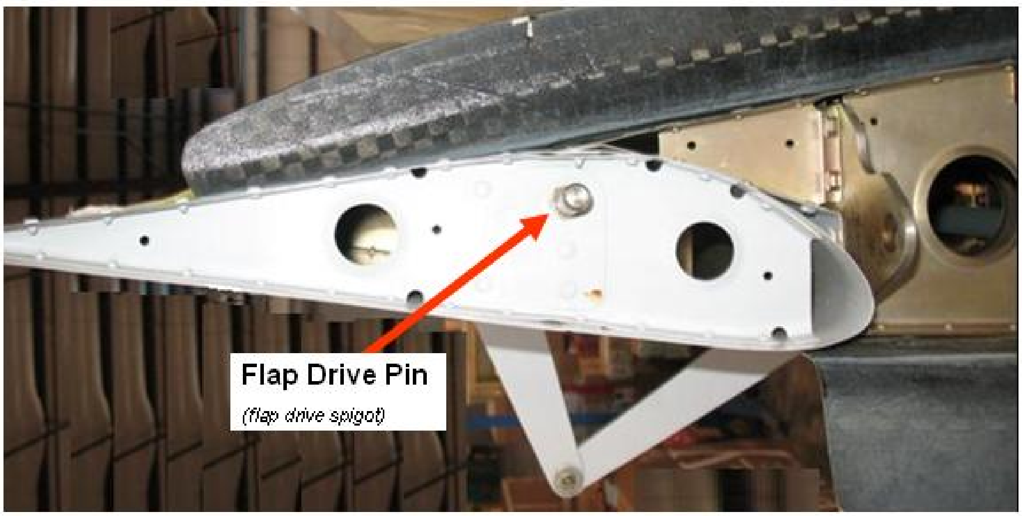


Figure 27-74 Flap Drive Pin

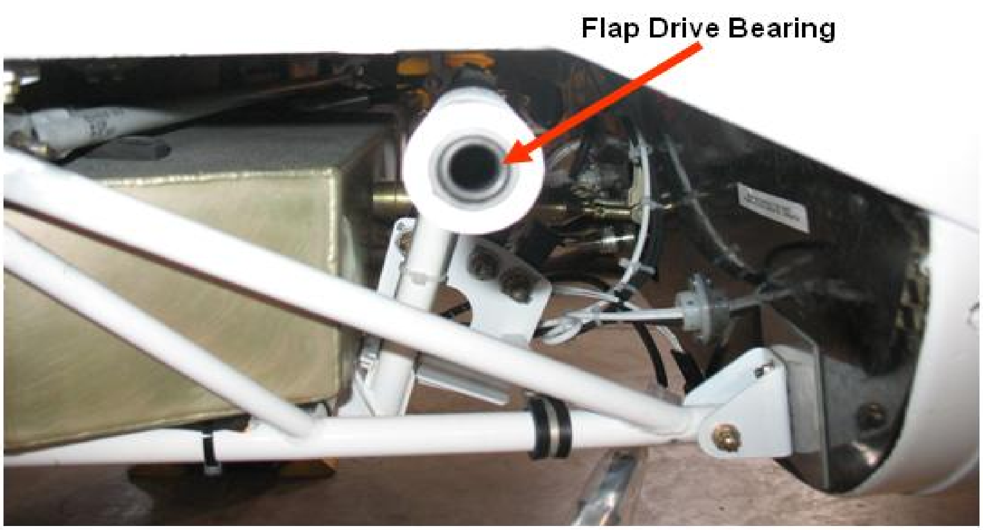


Figure 27-75 Flap Drive Bearing

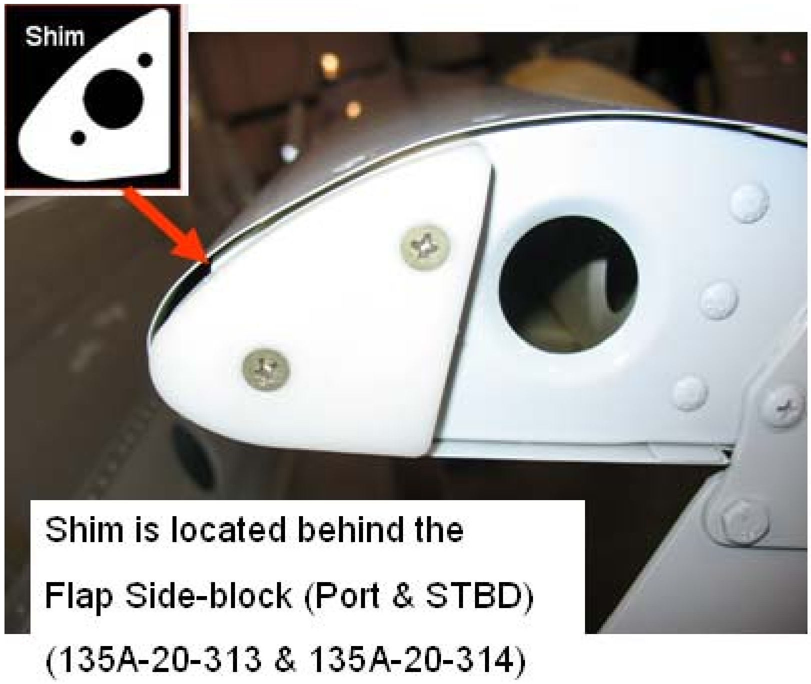


Figure 27-76 **Flap Shim**

Section 50-02 Flap Repair

At the time of this printing, Liberty Aerospace has not published a repair procedure. When this procedure is compiled and approved, it will be come a part of this manual. If damage is incurred to the airplane before that time, contact Customer service at Liberty Aerospace for assistance. Damage is listed as:

- Negligible – Any smooth dents in the flap surface, free from cracks, abrasions, and sharp corners. No stress wrinkles and does not interfere with any internal structure. Further definition is available from Liberty Aerospace Customer Service.
- Repairable – Any damage that is minor in nature i.e.: small holes (<0.1" diameter), creases, or abrasions that require the surface to be treated but do not require replacement as deemed by a certified Airframe and Powerplant (A&P) mechanic..
- Replacement – If damage is extensive or deemed major by an A & P, and/or repair costs exceed that of replacement costs, replace the flap. Contact the Customer Service Department of Liberty Aerospace.

Section 50-03 Flap Damage

The following are some example conditions when flap has undergone a load above limit load. When performing maintenance inspection, it is advised to pay special attention to these types of damage.

1. Multiple observable metal scores that remove paint, on upper flap surfaces

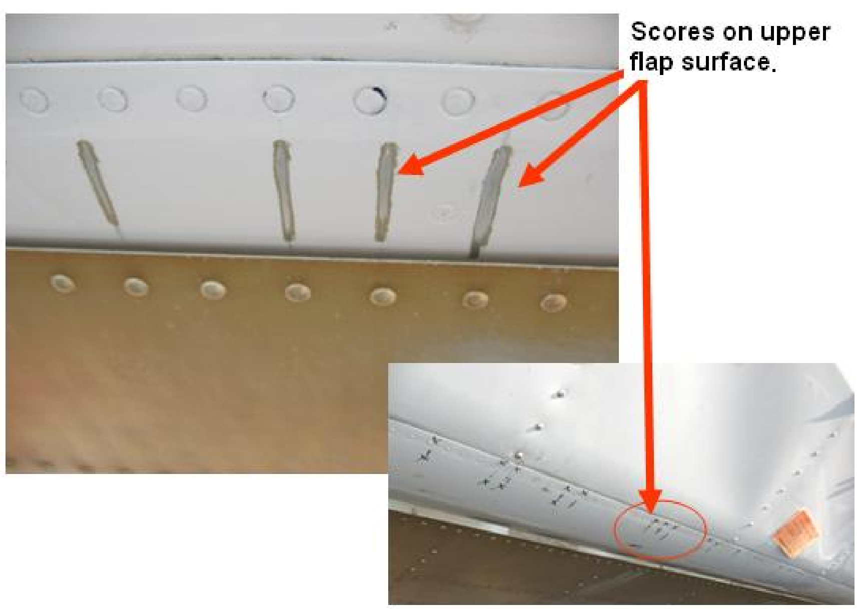


Figure 27-77 Metal Scores on Flap Surfaces

2. Permanent deformation on flap drive arm

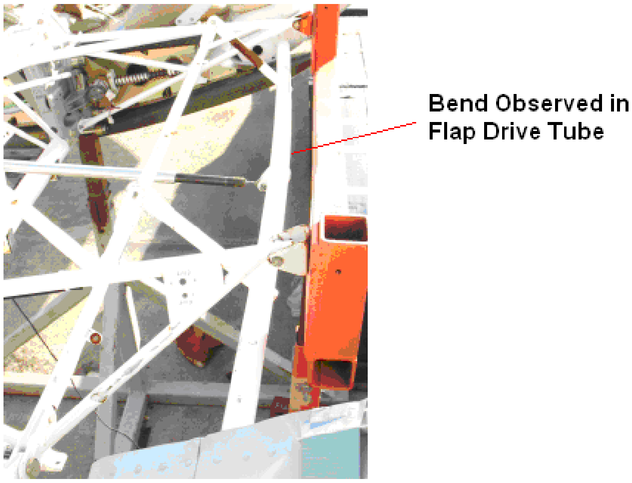


Figure 27-78 Bent Flap Drive Tube

If this deformation is suspected, welded areas highlighted below should be inspected for any cracks by Magnetic Particle Inspection (MPI) or by qualified technician and the result should be reported to Liberty Customer Support.

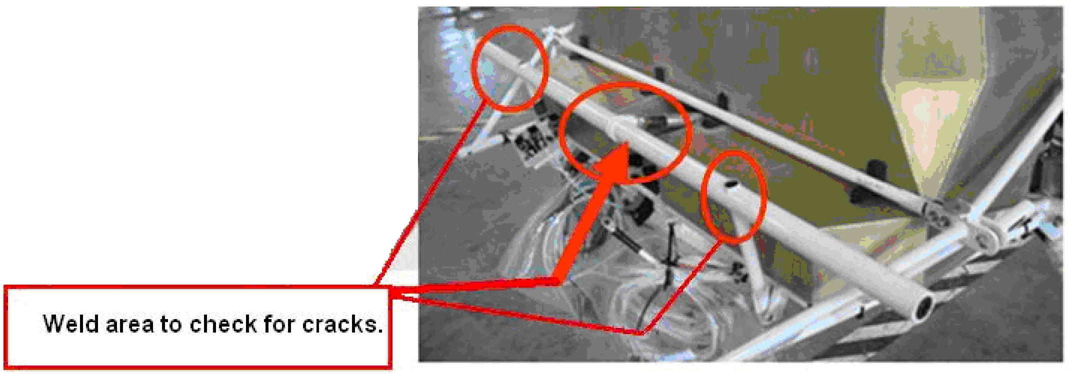


Figure 27-79 Cracks Inspection Area

Section 50-04 Flap Troubleshooting

Table 27-8 will aid in troubleshooting issues with the flaps.

Complaint	Possible Cause	Remedy
Flap inoperative	Defective flap circuit breaker	Replace
	Defective flap switch	Replace
	Defective flap actuator	Replace
	Defective wiring	Repair
Un-commanded flap motion	Defective flap switch	Replace
	Faulty wiring	Inspect and test wiring repair
Flap moving in one direction only	Defective flap switch	Replace
	Faulty wiring	Inspect and test wiring repair
Flap indicator does not correctly indicate flap position	Defective indicator	Replace indicator
	Defective position sensor	Replace sensor
	Defective wiring	Repair

Table 27-8 Flap System Troubleshooting

Section 27-90 Rigging Procedures

This section contains the procedures to check the rigging of the airplane's control surfaces. These procedures should not be attempted without contacting Liberty Aerospace, Inc. Customer Service before starting.

Section 90-01 Preliminary Inspection and Setup

Perform this procedure to do a preliminary inspection and setup prior to doing the rigging procedures. These procedures require two to three people to complete. The aircraft must be in a hanger or similar structure during the rigging procedures. The aircraft must not be subject to air movements or wind.

NOTE

The SmartTool TM Digital Inclinometer & bevel protractor referred to in this section is available for purchase from Liberty Aerospace, Inc. or through the internet

NOTE

Always zero out the SmartTool TM Digital Inclinometer by the inbuilt calibrating procedure prior to use, so that when moving from location to location & possibly reversing the inclinometer the reading will not be compromised. In addition when placing on a control surface subject to actuation, either tape the device to the surface for security with double sided masking tape or hold against the surface with out gripping and potentially creating an incorrect reading.

NOTE

These procedures consider that rod end fittings are undisturbed or have remained undisturbed since the airplane left the factory. If new push rods have been installed, the A&P mechanic should verify the new pushrod distance between the two end fittings and that it is consistent with the push rod being removed. If this is not the case, contact Liberty Aerospace, Inc. Customer Service.

VERIFICATION OF FLYING SURFACES RELATIVE MOVEMENT & RESPONSE

Perform the following to verify that control surfaces move and respond correctly to inputs from the cockpit. Start with the Ailerons and Horizontal Stabilizers then move to the rudder and finally with the power controlled flaps and trim tabs.

1. With an observer free to move around the airplane, gently move the pilot then co-pilot control column, progressively increase range until full movement of ailerons and horizontal stabilizer has been met. The operator is to note any undue stiffness and the observer is to note any sounds, binding, or interference.
2. Verify correct response of ailerons and horizontal stabilizer according to the control column input defined in Table 27-9.

Control Column Input	Expected Response
To Port	Port aileron up; starboard aileron down
To Starboard	Starboard aileron up; port aileron down
Forward	Horizontal stabilizer trailing edge down
Aft	Horizontal stabilizer trailing edge up

Table 27-9 Yoke Control Response

3. With an observer free to move around the airplane, gently move the pilot then co-pilot rudder pedals, progressively increase range until full movement of the rudder is achieved by hitting the stops. The operator is to note any undue stiffness and the observer is to note any sounds, binding, or interference.
4. Verify correct response of rudder according to the rudder pedal input defined in the following table:

Rudder Pedal Input	Expected Response
Port Rudder Pedal Full Forward	Rudder Trailing Edge Deflects to Port
Starboard Rudder Pedal Full Forward	Rudder Trailing Edge Deflects to Starboard

Table 27-10 Rudder Control Response

5. Place the Master Power Switch for the Battery to ON. All other switches are OFF
6. To deploy the flap, push the flap switch’s tongue downward or push on the bottom half of the switch labeled ‘FLAPS DOWN’ and hold (with normal finger pressure to fully deploy the flaps with the observer monitoring travel at all times. The operator should be prepared to release the flap switch at anytime if any anomalies occur or the observer notes any sounds, or indications of binding, or visually observes non-symmetrical deployment of the flaps.
7. After driving the flaps all the way out, the actuator should freely spools. This indicates the end of travel.

8. As the flap deploys, the flap spigot must slide freely within the bearing of the flap drive cross bar assembly. If the flap spigot does not slide within the bearing smoothly, clean the flap drive spigot and oil spigot and bearing using MIL-PRF-32033D. See Chapter 57 – Wings for the procedure to clean and lubricate the drive spigot.
9. With the flaps fully deployed, push the flap switch's tongue upward (with normal finger pressure) or push on the top half of the switch labeled 'FLAPS UP' and hold to fully retract the flaps with the operator and an observer monitoring travel at all times. The operator should be prepared to release the flap switch at anytime if any anomalies occur. Verify symmetrical retraction of the flaps.
10. After driving the flaps all the way in, the actuator free spools, this is the end of travel.
11. As the flap retacts, the flap spigot must slide freely within the bearing of the flap drive cross bar assembly. If the flap spigot does not slide within the bearing smoothly, clean the flap drive spigot and oil spigot and bearing using MIL-PRF-32033D. See Chapter 57 – Wings for the procedure to clean and lubricate the drive spigot.
12. With control column held in the neutral position, momentarily press the trim switch in any direction and verify the system responds accordingly. Continue pressing the switch until full travel has been established in both directions. After driving the trim tabs to the extremes of travel, the actuator free spools.
13. The actuation of the trim tab must not induce control column motion or force at the hand grip. Ensure lubrication of all moving parts using lubricant that complies with Mil-C-81309E type 2 (Corrosion-X for aviation) to prevent the trim tab moving the horizontal stabilizer.
14. An observer must monitor the actuation of the trim tab mechanism in the lower aft portion of the fuselage (access via aft lower access panel aft starboard fuselage). The observer must ensure that the trim tab drive mechanism stays clear of the fuselage and the rudder push rod during operation.
15. Repeat previous step 14 while moving the control column through its entire range of motion.
16. Check all combinations of trim tab / horizontal stabilizer movement against full travel of rudder to ensure a minimum clearance of 0.375in between inboard aft corner of trim tab (worst condition tail plane and trim tab extended fully down) and rudder skin at maximum rudder travel see Figure 27-80.

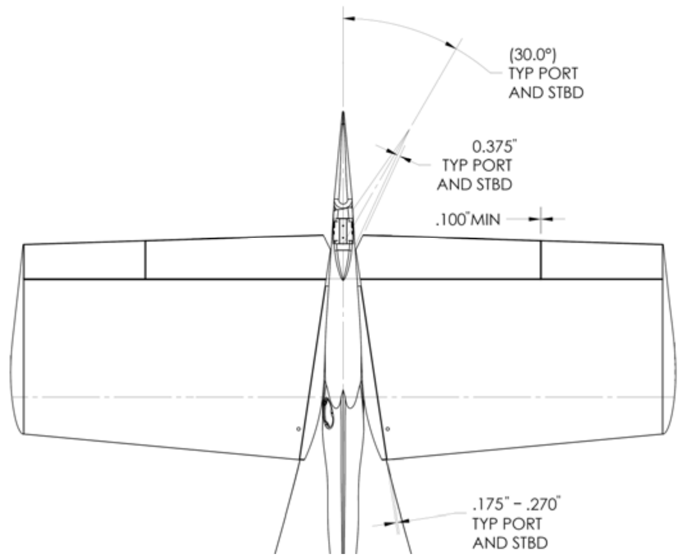


Figure 27-80 Tail Section Measurements

17. Run the trim tab fully in both directions to verify proper movement of control surface. If the following conditions are not met then an investigation into switch orientation and wiring must be initiated.

Trim Tab Switch Input	Expected Response
Forward Half Labeled 'NOSE DOWN'	Trim Tab Trailing Edge Deflects Upward
Aft Half Labeled 'NOSE UP'	Trim Tab Trailing Edge Deflects Downward

Table 27-11 Trim Tab Input / Response

18. Trim tab movement must be symmetrical and must not cause movement of the horizontal stabilizer circuit or surface.

19. Place the Master Power Switch to OFF.

FLAPS RIGGING

Perform this procedure to check, inspect and correct any issues with the rigging for the flaps. These procedures require two to three people to complete.

NOTE

This procedure considers that rod end fittings are undisturbed or have remained undisturbed since the airplane left the factory. If new push rods have been installed, the A&P mechanic should verify the new pushrod distance between the two end fittings and that it is consistent with the push rod being removed. If this is not the case, contact Liberty Aerospace, Inc. Customer Service.

1. Place the Master Power Switch for the Battery to ON. All other switches are OFF.
2. Check the flaps are fully retracted by pushing the flap switch's tongue upward or pushing on the top half of the switch (with normal finger pressure) labeled 'FLAPS Up' and hold to fully retract the flaps.

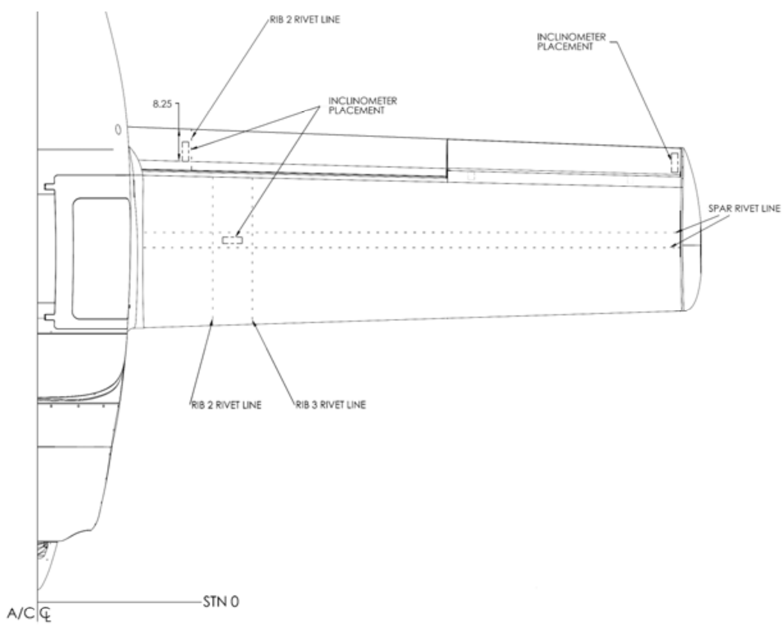


Figure 27-81 Location inclinometer for flap incidence true Port & Starboard

3. Locate a SmartTool™ digital inclinometer as shown in Figure 27-81.
4. Record the attitude angle of the port flap upper surface outboard of the second flap rib as shown in Figure 27-81 with the flap fully retracted.

5. Place the flap in the fully deployed position. With flap fully deployed, record the attitude angle of flap at the same position on flap surface. Subtract the first reading from the second reading and record. The recorded flap deflection must fall within the range of 29° ±1.00° as shown in Figure 27-82.
6. Repeat full flap deployment measurement steps for starboard flap.
7. Asymmetry between port and starboard measurement must not exceed 1°.
8. If the measurement of the asymmetry is within 1°, go to step 13.
9. If the measurements violate the asymmetry specification, contact Liberty Aerospace, Inc. Customer Service before making any adjustments. If adjustments are needed and Customer Service has directed the adjustments be made, adjustment of the port and / or starboard flap may be achieved by loosening some or all twelve (12) bolt assemblies (adjacent to the lower skin of the wing or flap) associated with the flap pivot arms.
10. Place the ailerons in the neutral position by inserting the aileron neutral blocks.
11. Align the top surface and trailing edge of the flap with the top surface and the trailing edge of the associated aileron.
12. Tighten the bolt assembly on the flap that were loosened in step 9.
13. Verify that the flap, as rigged, does not violates the minimum clearances as shown in Figure 27-82.

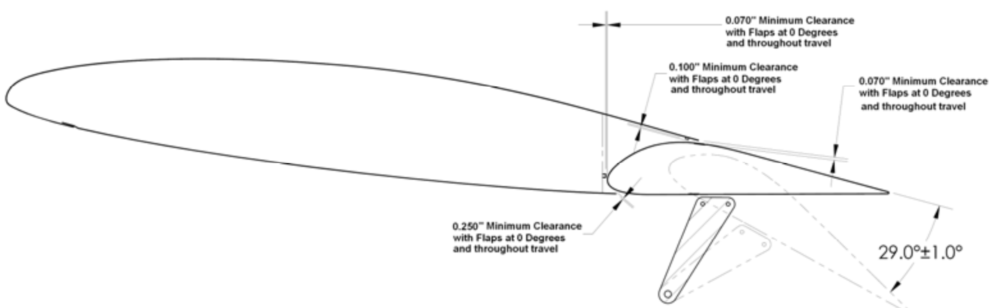


Figure 27-82 Clearances for the Flap

14. Place the Master Power Switch to OFF.

AILERON RIGGING

Perform this procedure to check, inspect and correct any issues with the rigging for the aileron. These procedures require two to three people to complete.

NOTE

This procedure considers that rod end fittings are undisturbed or have remained undisturbed since the airplane left the factory. If new push rods have been installed, the A&P mechanic should verify the new pushrod distance between the two end fittings and that it is consistent with the push rod being removed. If this is not the case, contact Liberty Aerospace, Inc. Customer Service.

NOTE

This procedure calls for the use of rigging blocks, part number 135A-02-511. These rigging blocks are available. Contact the Customer Service department of Liberty Aerospace, Inc.

1. Remove the belly panel (see Chapter 53 – Fuselage).
2. Set the ailerons to their neutral position by locking the aileron control circuit within the chassis by securing two (2) Aileron Neutral blocks of an equal height of 1.0"   between the port and starboard control stop aileron and the yoke assembly, see Figure 27-6. Shim between the blocks and the yoke equally as required to eliminate play using 3M 425 Aluminum Foil Tape (5mil nominal thickness).

NOTE

Both blocks plus added shim must be of equal height.

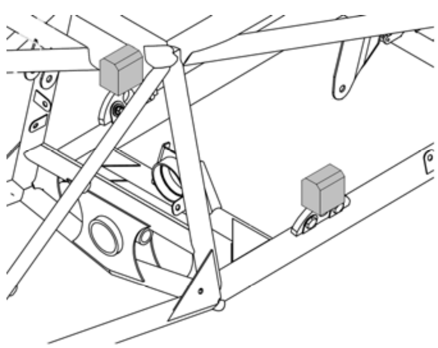


Figure 27-83 Aileron Neutral Blocks Placement

3. Verify using feeler gages that the parallel gap between the mating faces of the aileron quick connect channel and the port aileron quick connect bearing assembly falls between 0.001 inches and 0.006 inches.

4. When using the filler gage to measure the gap between the mating faces of the quick disconnect, the gage must be fully inserted such that it is making contact along the entire surface area between the disconnects. For additional information, see Figure 27-84 and Figure 27-85.

CAUTION

Contact between the mating surfaces of the Quick Disconnect device must across the entire surface. Contact at just the 'heel' or 'toe' of this device is not acceptable.

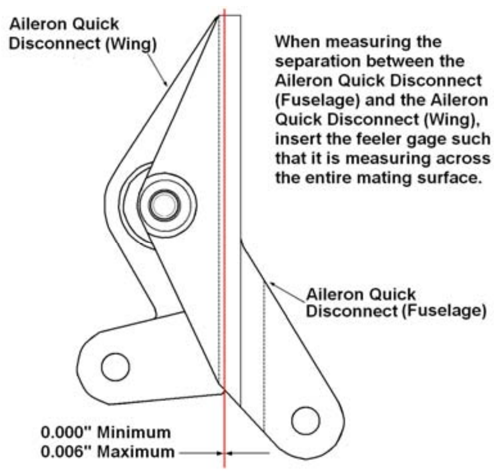


Figure 27-84 Aileron Quick Disconnect Clearances

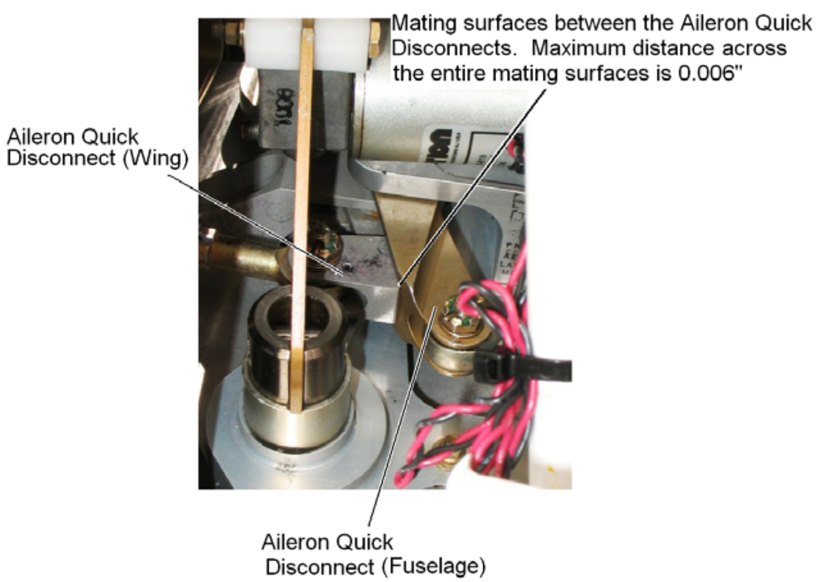


Figure 27-85 Aileron Quick Disconnect

5. If the gap is greater than 0.006in, reduce the gap by bringing the two assemblies closer together by adjusting the aileron quick connect assembly mounted to the wing. Adjusting the aileron quick connect assembly requires removing the wing from the airplane. Refer to Chapter 57 – Wings.
6. Repeat steps 4 and 5 for the starboard aileron quick connect assembly.

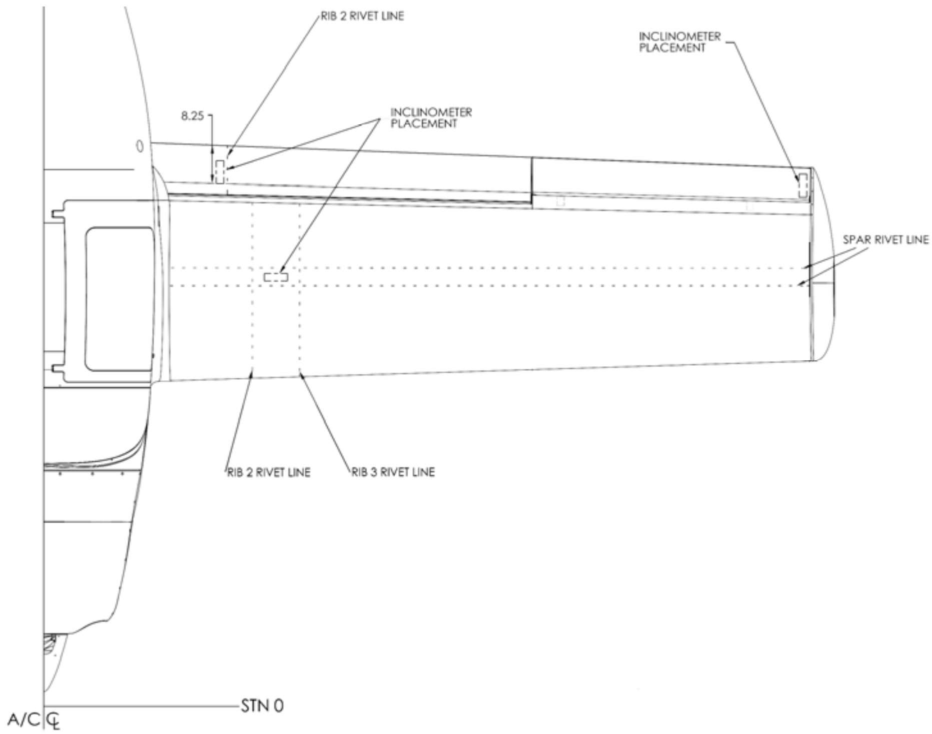


Figure 27-86 Location Aileron Inclinometer true Port & Starboard

7. With the rigging block holding the yoke in the neutral position, place a SmartTool™ digital inclinometer positioned on the top aileron skin surface of the port wing as shown in Figure 27-86.
8. Record the angle reading on the inclinometer on line 1 for port Aileron in Table 27-4 below.
9. Repeat Steps 7 & 8 for the starboard wing.

NOTE

The flaps do not provide an accurate reference and may not be used to set the aileron neutral attitude.

	Record all angles as an absolute value (disregarding any + or - sign)	**Port Aileron**	**Starboard Aileron**
1	Inclinometer reading at neutral		
2	Inclinometer reading at aileron fully up		
3	Inclinometer reading at aileron fully down		
4	Aileron Deflection Up (Add Lines 1 and 2) 24.0°± 1.0°		
5	Aileron Deflection Down (Subtract Line 1 from Line 3) 19.0°± 1.0°		
6	Total Deflection (Range) (Add Lines 4 and 5) 43.0°± 1.0°		

Table 27-12 Aileron Angle Measurements

10. Remove both (2) aileron neutral blocks.
11. Move the control column to port, fully to the stop, and hold.

NOTE

It is recommended to use a good quality bungee cord to hold the yoke control at the extremes of movement. The bungee cord can be hooked to either the throttle control or the brake handle.

12. Record the up angle on a SmartTool™ digital inclinometer on the port aileron in line 1 of Table 27-12 above for the port aileron.
13. Record the down angle on the inclinometer on the starboard aileron on line 2 of Table 27-12 above for the starboard aileron.
14. Move the control column to starboard, fully to the stop, and hold.
15. Record the down angle on a SmartTool™ digital inclinometer on the port aileron in line 2 of Table 27-12 above for the port aileron.
16. Record the up angle on the inclinometer on the starboard aileron on line 1 of Table 27-12 above for the starboard aileron.
17. Calculate the up angle deflection for the port and starboard aileron. The angle should be 24.0°± 1.0°.
18. Calculate the down angle deflection for the port and starboard aileron. The angle should be 19.0°± 1.0°.
19. Calculate the total angle of travel (range) for the aileron (adding the numbers from steps 17 and 18. The angle should be 43.0°± 1.0°.
20. If the range for the port aileron is not within the range of 43.0°± 1.0°, adjust the port aileron deflection range by adjusting the port aileron push rod ladder. To reduce the range, lengthen the push rod. To increase the range, shorten the push rod.

21. If the range for the starboard aileron is not within the range of 43.0°± 1.0°, adjust the starboard aileron deflection range by adjusting the starboard aileron push rod ladder. To reduce the range, lengthen the push rod. To increase the range, shorten the push rod.

NOTE

The total range angel (43.0°± 1.0°) is necessary for the neutral alignment of the aileron.

The adjustment of the port and starboard deflection range is independent of each other. Adjusting the port range does not affect the starboard range. Adjusting the starboard range does not affect the port range.

22. If there have been any adjustments to the ranges of the ailerons, repeat steps 7 through 19 to recalculate the new range. When the range of the port and starboard ailerons is in the range of 43.0°± 1.0° and within 0.5° of each other, go to step 24.
23. If there have been no adjustments made to the aileron control circuit (port and/or starboard) go to step 28.
24. To reset ailerons neutral, lock the control circuit within the chassis by securing two (2) aileron neutral blocks between the port and starboard control stop aileron and the yoke assembly. Shim between the blocks and the yoke equally as required to eliminate play using 3M 425 Aluminum Foil Tape (0.005"nominal thickness).

NOTE

Both blocks plus added shim must be of equal height. A maximum of 2 shims may be used.

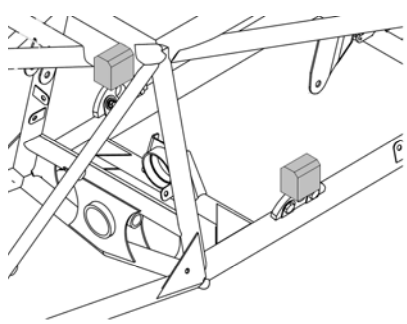


Figure 27-87 Aileron Neutral Block Placement

25. Adjust the port aileron to 0.5° of the port wing tip, by adjusting the port aileron pushrod.

26. Adjust starboard aileron as necessary by adjusting the starboard wing pushrods for the aileron control circuits until both aileron trailing edges are within 0.5° of each other.

NOTE

The flaps do not provide an accurate reference and may not be used to set the aileron neutral attitude.

27. Remove both (2) aileron neutral blocks.

28. Record these final values.

29. When satisfied with the rigging of both ailerons:

30. Visually check pushrods rod ends, six (6) places port and six (6) places starboard for approximately equal amount of thread showing.

31. Check the rod end bearings are not over extended by trying to pass safety wire through the witness hole in the shank of the rod end bearings. The wire should not pass through the hole.

32. Verify push rod end jam nuts within the wing, four (4) places port and four (4) places starboard, and within the chassis, two (2) places port and two (2) places starboard tightened.

33. Check that the aileron, as rigged, does not violate the minimum clearances as shown in Figure 27-88.

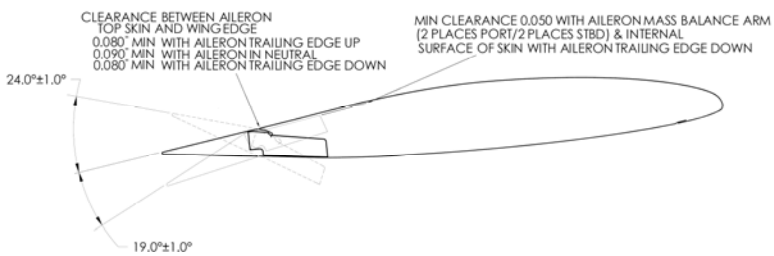


Figure 27-88 Aileron Measurements

34. Install belly panel, see Chapter 53 – Fuselage.

HORIZONTAL STABILIZERS & TRIM TABS

Perform this procedure to check, inspect and correct any issues with the horizontal stabilizer. These procedures require two to three people to complete.

NOTE

This procedure considers that rod end fittings are undisturbed or have remained undisturbed since the airplane left the factory. If new push rods have been installed, the A&P mechanic should verify the new pushrod distance between the two end fittings and that it is consistent with the push rod being removed. If this is not the case, contact Liberty Aerospace, Inc. Customer Service.

1. Remove the belly panel, see chapter 53 – Fuselage.
2. Verify that the minimum distance between the center of the co-pilot yoke handle (center of microphone switch) and the instrument console is not less than 1.5”. If less than 1.5” adjust the horizontal stabilizer control by decreasing the length of the forward horizontal stabilizer push rod.
3. Verify that the minimum distance between the body of either the pilot’s or co-pilot’s yoke handle, and the forward upper edge of the seat pan is not less than 2.0”. If less than 2.0” adjust the horizontal stabilizer control by increasing the length of the forward horizontal stabilizer push rod.
4. If there has been any adjustments to the push rods, verify all adjusted jam nuts fully tightened.
5. Set the horizontal stabilizer to neutral, temporarily tape horizontal stabilizer inboard skin to horizontal stabilizer profile stub on fuselage, Port & Starboard. This will effectively hold the horizontal stabilizer in neutral.

NOTE

Masking tape is satisfactory & will not damage paint surface.

NOTE

The incidence of the starboard horizontal stabilizer must be within 0.1° of the incidence of the port horizontal stabilizer.

A maximum of 0.5° of asymmetry between the port and starboard horizontal stabilizer neutral position is acceptable.

6. Place on top of the Port horizontal stabilizers SmartTool™ digital inclinometer as shown in Figure 27-89. Record the port horizontal stabilizer upper skin neutral angle.

7. Place on top of the Starboard horizontal stabilizers SmartTool™ digital inclinometers as shown in Figure 27-89. Record the starboard horizontal stabilizer upper skin neutral angle.
8. The incidence of the starboard horizontal stabilizer must be within 0.1° of the incidence of the port horizontal stabilizer.
9. Check the control column; it should be in the neutral position.
10. Place the Master Power Switch for the Battery to ON. All other switches are OFF.
11. Place on top of the port trim tab SmartTool digital inclinometer as shown in Figure 27-89. Actuate the trim tab 'NOSE UP' or 'NOSE DOWN' until the trailing edge of the port trim tab is within 0.2° of the angles recorded for the port horizontal stabilizer. Transfer SmartTool digital inclinometer to starboard or utilize a second SmartTool digital inclinometer on the starboard location. The trim tabs are to be within 0.5° of each other.
12. Record the port and starboard trim tab upper skin neutral angle.

NOTE

A 0.5° of maximum asymmetry between the port and starboard trim tab at neutral is considered acceptable

13. Fully deflect the trim tab trailing edge up by pushing the 'NOSE DOWN' labeled section of the trim tab switch until the actuator is free spooling.
14. Use the SmartTool™ digital inclinometer located on the top of the port trim tab skin as shown in Figure 27-89. Record the trailing edge up deflection angle. The angle must fall between 4.5° and 5.5° (5.0° ±0.5°). See Figure 27-90.
15. Fully deflect the trim tab trailing edge up by pushing the 'NOSE UP' labeled section of the trim tab switch until the actuator is free spooling.
16. Use the SmartTool™ digital inclinometer located on the top of the port trim tab skin as shown in Figure 27-89. Record the trailing edge down deflection angle. The angle must fall between 4.5° and 5.5° (5.0° ±0.5°). See Figure 27-90.

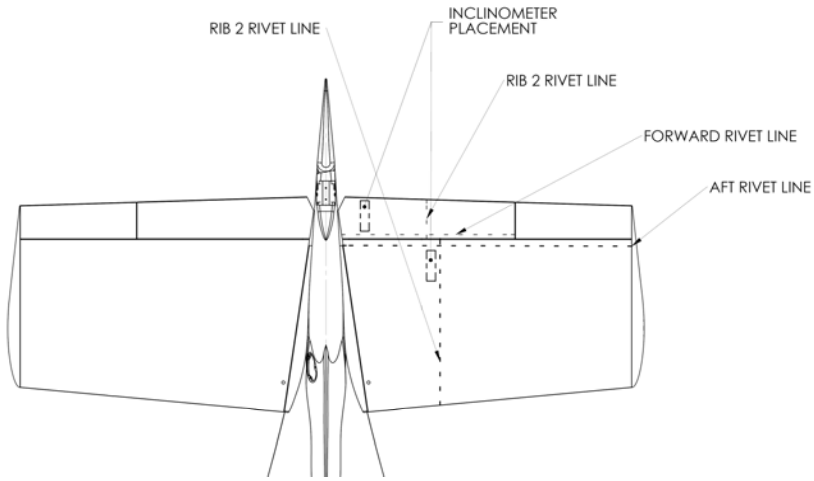


Figure 27-89 Location Inclinometer for horizontal stabilizer

17. Remove tape restraining horizontal stabilizer to fuselage stub profile.
18. Move the control column aft, fully to the stop and hold.
19. Record the port horizontal stabilizer pitch up (trailing edge up) attitude from a SmartTool™ digital inclinometer located on the top of the tailplane skin as shown in Figure 27-89. This measurement must fall between 12.5° and 13.5° (13.0°±0.5°) of the horizontal stabilizer neutral attitude angle recorded in Step 6 See Figure 27-90.

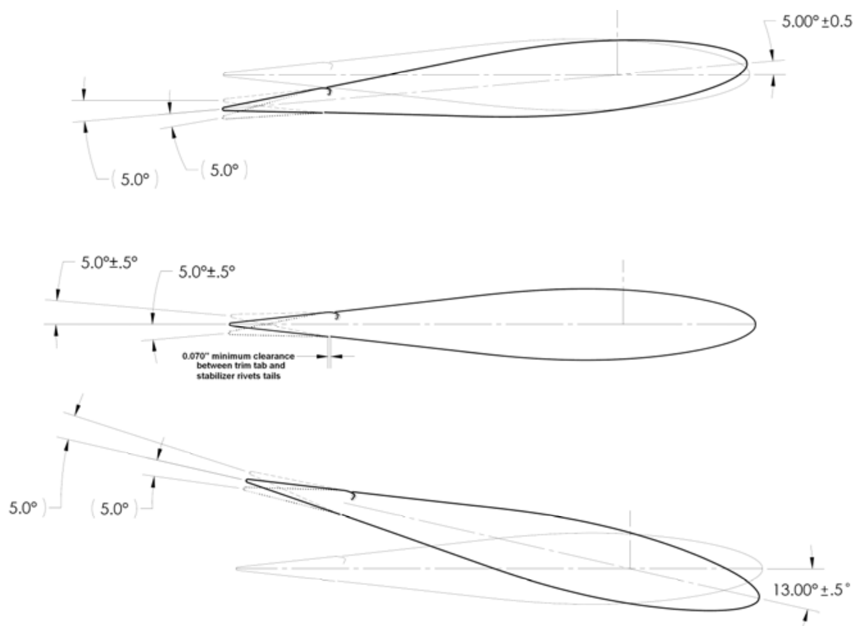


Figure 27-90 Stabilizer Deflections

20. Move the control column forward, fully to the stop and hold.
21. Record the port horizontal stabilizer pitch down (trailing edge down) attitude from a SmartTool™ digital inclinometer located on the top of the tail plane skin as shown in Figure 27-89. This measurement must fall between 4.5° and 5.5° (5.0° ±0.5°). See Figure 27-90.

22. Check the clearance between the trim tab and the stabilizer. The clearance should be 0.100in minimum. See Figure 27-92.

23. Measure the clearance between the horizontal stabilizer and the fuselage. The clearance should be more than 0.175in and less than 0.270in. See Figure 27-92.

24. Check the clearance between the co-pilot's control yoke handle and the instrument console. The clearance between the center of the yoke handle (the push-to-talk button) and the instrument console must be greater than 1.5in., see Figure 27-91. If less the 1.5in, adjust the horizontal stabilizer control by decreasing the length of the forward horizontal stabilizer push rod.

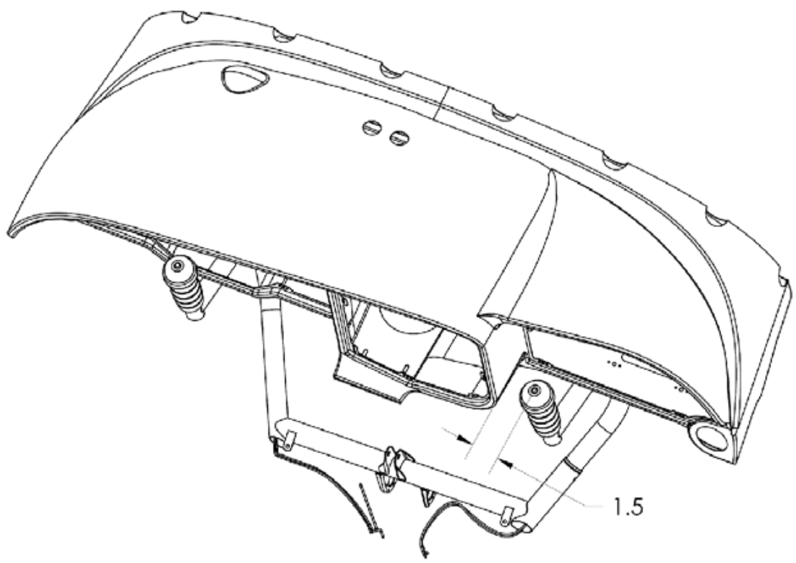


Figure 27-91 Clearance Between the Co-pilot's Yoke and the Instrument Console

25. Check all combinations of trim tab / horizontal stabilizer movement against full travel of rudder to ensure a minimum clearance of 0.375in between inboard aft corner of trim tab (worst condition tail plane and trim tab extended fully down) and rudder skin at maximum rudder travel see Figure 27-92.

26. Check the rod end bearings are not over extended by trying to pass safety wire through the witness hole in the shank of the rod end bearings. The wire should not pass through the hole.

27. Place the Master Power Switch to OFF.

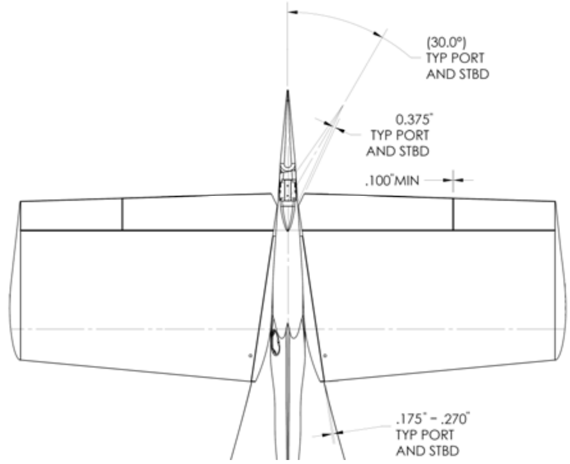


Figure 27-92 Tail Section Clearance Measurements Between Horizontal Stabilator and the Vertical Stabilizer

RUDDER RIGGING

Perform this procedure to check, inspect and correct any issues with the rigging for the rudder. These procedures require two to three people to complete.

NOTE

This procedure considers that rod end fittings are undisturbed or have remained undisturbed since the airplane left the factory. If new push rods have been installed, the A&P mechanic should verify the new pushrod distance between the two end fittings and that it is consistent with the push rod being removed. If this is not the case, contact Liberty Aerospace, Inc. Customer Service.

NOTE

The Bevel protractor referred to in this section is a scaled or none scaled protractor capable of measuring obtuse angles, which is available via package through Liberty Aerospace, Inc. or through the internet. For none scaled protractors set bevel protractor to rudder angle & use a second scaled protractor to record actual angle.

NOTE

Pilots & co-pilots rudder stops are on a common drive so input from the pilots position is acceptable for the following sequence.

1. Press and hold the pilot's port rudder pedal.
2. An observer to locate bevel protractor normal to hinge line port at location defined, measure and record the angle of deflection of the rudder if scaled, or transfer angle to a scaled protractor and record angle.
3. The angle of the rudder shall be 30.0° +0.5°/-1.5°, in the range of 28.5° to 30.5°, to the Port side. See Figure 27-94 for details.
4. Press and hold the pilot's starboard rudder pedal.
5. The observer to locate bevel protractor normal to hinge line starboard at location defined, measure and record the angle of deflection of the rudder if scaled, or transfer angle to a scaled protractor and record angle.
6. The angle of the rudder shall be 30.0° +0.5°/-1.5°, in the range of 28.5° to 30.5°, to the starboard side. See Figure 27-94 for details.
7. If necessary, adjust the rudder control circuit pushrods to bring the rudder trailing edge within the range of 28.5° to 30.5° Port & 28.5° to 30.5° starboard.

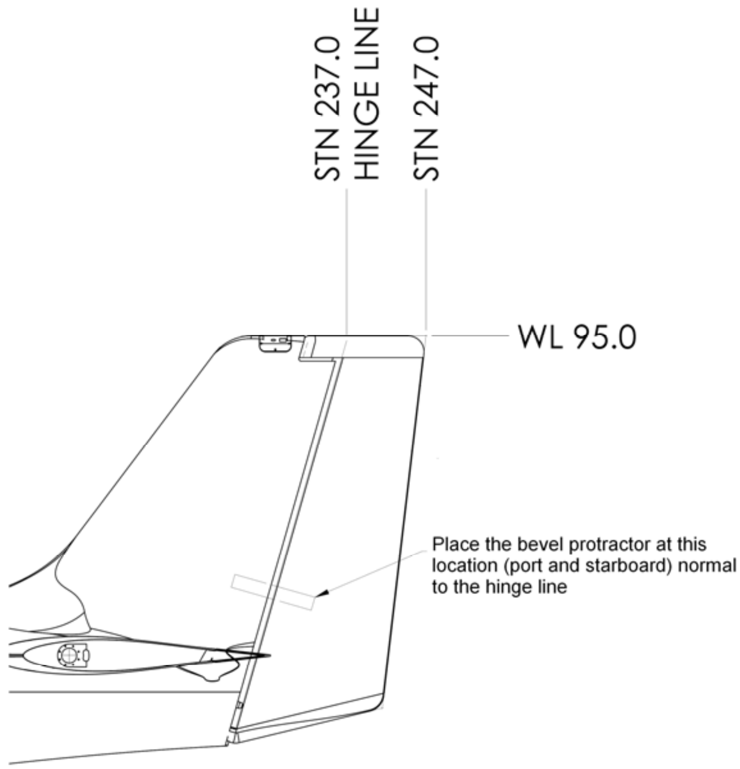


Figure 27-93 Rudder Measurements

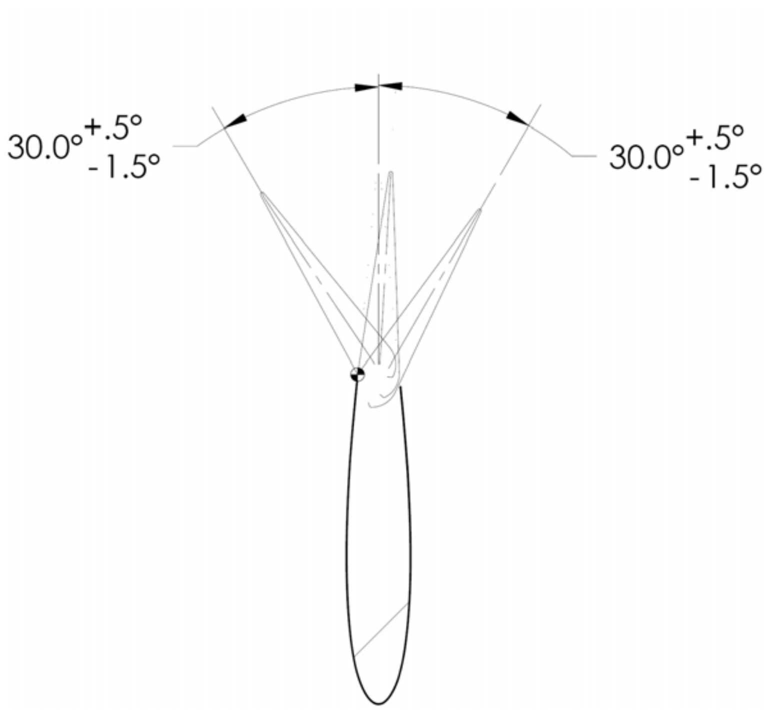


Figure 27-94 Rudder Deflection Measurements

8. Check the clearances of the rudder to the stabilizers as shown in Figure 27-95.

9. Only if disturbed verify all jam nuts on each of the rod ends within the control circuit, eight (8) places have been fully tightened.
10. Visually check pushrods rod ends (8) for approximately equal amount of thread showing.
11. Check the rod end bearings are not over extended by trying to pass safety wire through the witness hole in the shank of the rod end bearings. The wire should not pass through the hole.

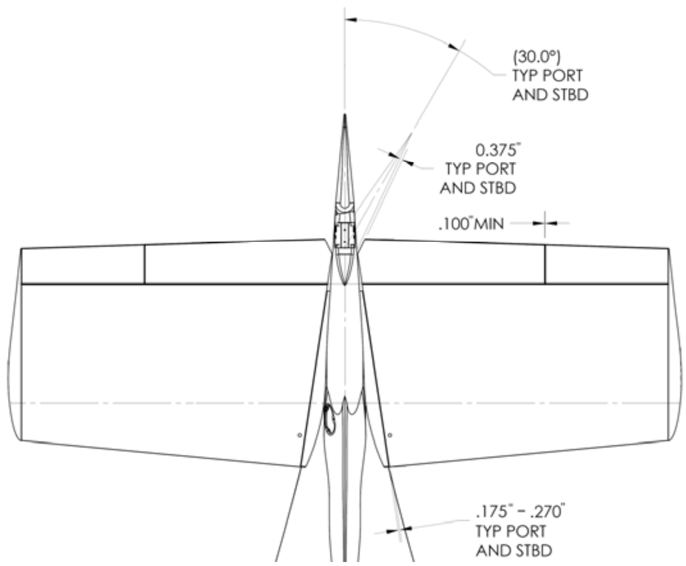


Figure 27-95 Tail Section Measurements

NOTE

If the rudder deflection is adjusted, check all combinations of trim tab / horizontal stabilizer movement against full rudder travel to ensure a minimum clearance of 0.375in between inboard aft corner of trim tab (worst condition tail plane and trim tab extended fully down) and rudder skin at maximum rudder travel see Figure 27-95.

CHAPTER 28

FUEL SYSTEM

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Table of Contents

SECTION 28-00 GENERAL 5

SECTION 28-10 STORAGE 7

SECTION 10-01 FUEL TANK PROCEDURES 7
FUEL TANK AND SYSTEM INSPECTION 8
SECTION 10-02 DAMAGE CLASSIFICATION 9
SECTION 10-03 REPAIRS TO THE FUEL TANK 10
REPAIRING THE FUEL TANK BY WELDING 11
REPAIRING THE FUEL TANK BY TITANIUM PUTTY 12
SECTION 10-04 FUEL TANK DRAINING PROCEDURE 14
DRAINING THE FUEL TANK 16

SECTION 28-20 DISTRIBUTION 17

SECTION 20-01 FUEL SYSTEM MAINTENANCE ACCESS 17
SECTION 20-02 FUEL SYSTEM PROCEDURES 17
GASCOLATOR BOWL REMOVAL 18
GASCOLATOR BOWL INSTALLATION 19
GASCOLATOR REMOVAL 20
GASCOLATOR INSTALLATION 21
ELECTRIC FUEL PUMP REMOVAL 22
ELECTRIC FUEL PUMP INSTALLATION 23
ELECTRIC FUEL PUMP CHECKOUT 24
COCKPIT FUEL VALVE REMOVAL 25
COCKPIT FUEL VALVE INSTALLATION 26
CHECK VALVE REMOVAL 27
CHECK VALVE INSTALLATION 28
FUEL LINE REMOVAL 29
FUEL LINE INSTALLATION 30
SECTION 20-03 FUEL SYSTEM ELECTRICAL 31
SECTION 20-04 FUEL SYSTEM ELECTRICAL PROCEDURES 31
FUEL BOOST PUMP SWITCH REMOVAL 32
FUEL BOOST PUMP SWITCH INSTALLATION 33
FUEL BOOST RELAY REMOVAL 34
FUEL BOOST RELAY INSTALLATION 35
SECTION 20-05 AIRFRAME FUEL SYSTEM OPERATIONAL CHECK AND INSPECTION 36
AIRFRAME FUEL SYSTEM OPERATIONAL CHECK AND INSPECTION 37
SECTION 20-06 FUEL SYSTEM TROUBLESHOOTING 39

SECTION 28-40 FUEL INDICATION 41

SECTION 40-01 FUEL LEVEL SENDER REMOVAL AND REPLACEMENT 41
FUEL SENDER REMOVAL 42
FUEL SENDER INSTALLATION 43
SECTION 40-02 CAPACITIVE FUEL SENDER 44
SECTION 40-03 LEVEL FLIGHT ATTITUDE 45
SECTION 40-04 UNUSABLE AND TOTAL FUEL QUANTITY 45
SECTION 40-05 FUEL INDICATION CALIBRATION 45
SECTION 40-06 FUEL QUANTITY BY WEIGHT TOOL 45
SECTION 40-07 FILL FUEL TANK WITH 100LL BLUE OR 100 GREEN AVIATION FUEL ONLY. 47
SECTION 40-08 CALIBRATION TEST 48
CALIBRATION TEST – CONFIGURATION 49
CALIBRATION TEST – GALLONS FUEL GAUGE 51
CALIBRATION TEST – LITERS FUEL GAUGE 53
SECTION 40-09 ADJUSTMENTS 57
SECTION 40-10 TROUBLESHOOTING 59

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Section 28-00 General

This chapter covers the airframe fuel system from the fuel tank to the inlet of the engine fuel system and includes the following:

• Fuel Tank

• Fuel Boost Pump

• Gascolator

• Associated Tubing

The location of the fuel tank is in the space frame in the area of the seat back for the pilot and passenger. The fuel boost pump is located just forward of the fuel tank on the port side. The gascolator is located in the space frame with in the center console. Mounted along the structure of the space frame is the associated metal tubing that connects these items to the engine.

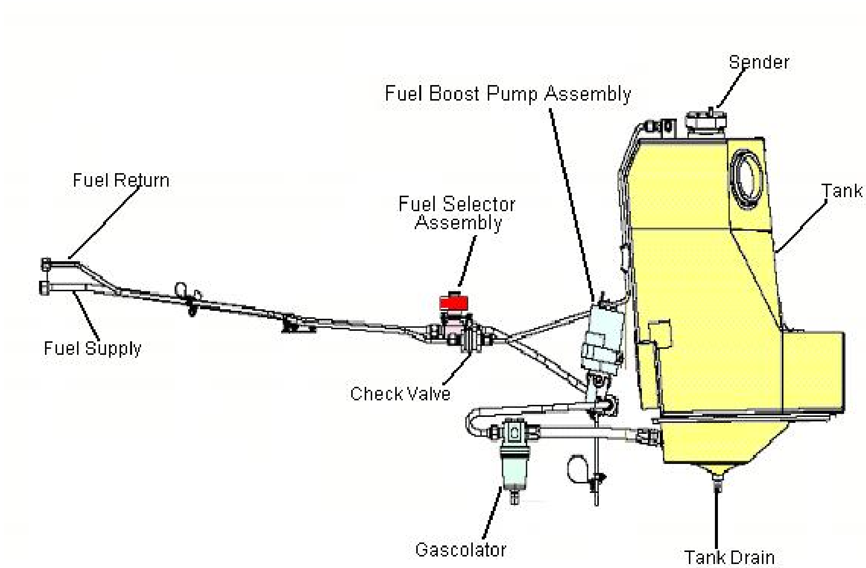

Figure 28-1 Fuel System Components

NOTE

The use of "fuel quantity indicator" and "fuel gauge" are used throughout this document and are interchangeable. In addition, the word liter and litre are used interchangeable.

The terms "gallons" and "ounces" are used throughout this document and relate to the United States fluid measurement system where one US gallon equals 128 fluid ounces.

The term "Liter" is used throughout this document and relates to the basic unit of measurement for volume in the metric system; equal to 61.025 cubic inches or 1.0567 liquid quarts. One Liter is equal to 0.2642 gallons (US).

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Section 28-10 Storage

The airplane fuel supply is contained in a single tank of 28 US Gallons (106 liters). The tank is of welded aluminum construction and is located in the fuselage directly below the seats. The tank is vented to the atmosphere. This section will also contain information to resolve issues with the fuel tank including minor damage, including the classification of different types of damage.

A fuel filler cap near the top of the fuselage on the port side, see Figure 28-2 connects to the fuel tank via a large diameter tube. A molded sump at the bottom of the tank is provided to trap water and contaminants. A standard fuel drain valve is installed at the bottom of the sump and is accessible through an opening in the fuselage belly fairing.

Figure 28-2 Fuel Filler

CAUTION

After a hard or crash landing, inspect the fuel tank in accordance with Chapter 5 - Time Limits/Maintenance Checks/Inspection Intervals. Also, check the tank's internal baffles for damage or unusual deformities.

Section 10-01 Fuel Tank Procedures

The fuel tank is located within the space frame and as such is not removable once the airplane has left Liberty Aerospace. If it becomes necessary to remove the fuel tank from the airplane, contact Liberty Aerospace, Inc. The only procedure presented here is for a visual inspection of the fuel tank.

FUEL TANK AND SYSTEM INSPECTION

Perform this procedure to inspect the fuel tank.

WARNING

REMOVE FROM THE AREA ALL SOURCES OF IGNITION. EVEN IF THE FUEL TANK IS DRAINED OF FUEL, VAPORS FROM THE FUEL TANK CAN IGNITE. WORK IN A WELL-VENTILATED AREA AWAY FROM ALL FLAMES OR EQUIPMENT THAT CAN CREATE A SPARK.

1. Connect a grounding wire to the airplane's ground, the exhaust pipe on the engine. If the exhaust pipe from the engine is not available, attach the grounding wire to an un-painted section of the airplanes main frame.
2. Remove the belly panel from the airplane. For the procedure to remove the belly panel, see Chapter 53 – *Fuselage*.
3. Use flashlight and small hand mirror to inspect all accessible surfaces of the fuel tank. Inspect closely all seams and fittings of the tank for leaks.

NOTE

Leaks are evident by the presence of the blue marker from the fuel. This marker may leave blue witness lines or marks that indicate the presence of a leak. If there is any evidence of leakage, regardless of size, the airplane is classified as un-airworthy. Repair or replacement of the part or parts that are shown to be leaking is required before returning the airplane to service.

4. Inspect the Gascolator, Fuel Boost Pump, Fuel Cut-off, and Check Valve in the same manner. Pay close attention to the connectors and tubing that connects to each of these devices.
5. Inspect all the tubing, paying close attention to areas of stress such as bends in the tubing. Look for an unusual bending or creasing in the tubing.

WARNING

If there are any abnormalities in the fuel system, the airplane is classified as un-airworthy. All discrepancies need resolution before the airplane is once again airworthy. An abnormality is defined as any leakage, evidence of leakage, unusual bending or flexing, or creasing of the tubing or fuel tank.

Section 10-02 Damage Classification

Fuel tank failure or leakage points may be hairline cracks (L) or circles (D) as shown in the Figure 28-3. Classification is shown in Table 28-1.

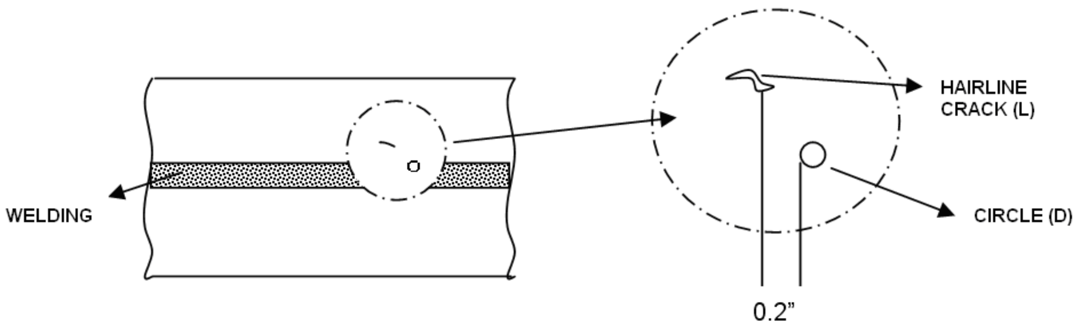

Figure 28-3 Damage Types

Maximum Damage Size (in)	Allowable Diameter (in)	Minimum Allowable Distance Between Two Damage/Leakage Points (in)	Maximum Number Of Damage/Leakage Points (in)	Repair Patch Dimensions On Each Leakage (in)
D- 0.2	4	0.2	3	1x1
L- 0.40	4	0.2	3	1x1

Table 28-1 Damage Classification

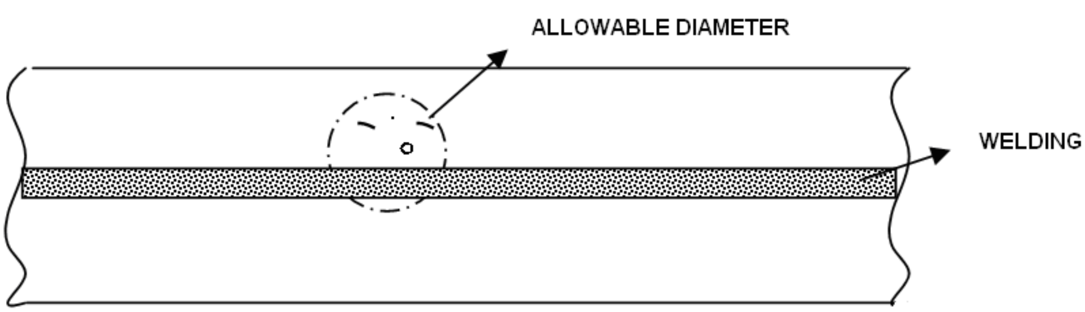

Figure 28-4 Damage Classification

The maximum length (L) or diameter (D) of damage or leakage points is 0.2". A maximum of three damages or leakage points are allowed within a 2" radius circle centered at one of the damage or leakage points, and the three points are not in a line (as shown in the Figure 28-4).

Table 28-2 shows damage classification where the proximity of the damage points is small. Such damages are considered to be one damage with a summation of the maximum length (L) or diameter (D). Figure 28-5 shows examples for each type presented in the table.

Damage Type	Damage Type	Minimum Allowable Distance Between two Damages (in)	Maximum Allowable Summation of Damage (in)	Repair Patch Dimensions (in)
I	Circle (D)	0.1	0.25	1.5X1.5
II	Hairline Crack (L)	0.1	0.50	1.5X1.5
III	Both	0.1	0.55	2.0X2.0

Table 28-2 Damage and Repair Classification

Figure 28-5 shows the special cases of failure/leakage types.

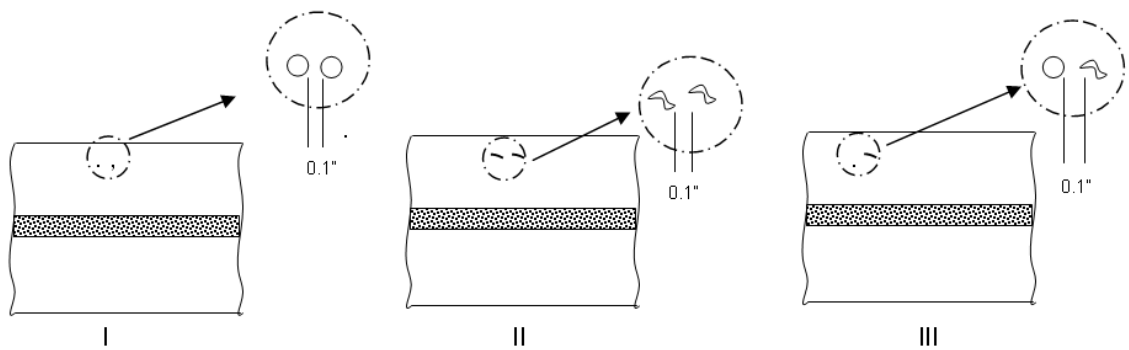

Figure 28-5 Damage Types

Section 10-03 Repairs to the Fuel Tank

There are two methods to repairing a leaking fuel tank: Welding and using Titanium Putty. This section contains the procedures for both. This section also contains the needed information to work with the Titanium putty.

The Titanium putty allows repair to the fuel tank without using heat or flame and therefore less chance of causing damage from igniting any residual fuel. Titanium putty is a high-tech, titanium-reinforced epoxy putty engineered for making critical repairs to machinery and precision parts. The Devcon Corporation's part number for the one pound kit is 10760; for the two pound kit, the part number is 10770. A properly mixed batch will have a pot life of 21 minutes (@75°F), and cures to 75% in 16-hours. A full cure takes seven days.

NOTE

Safety instructions for the Devcon® Titanium putty can be found at this link: http://www.devcon.com/techinfo/10760.PDF.

REPAIRING THE FUEL TANK BY WELDING

WARNING

METAL FUEL TANKS MUST NOT BE WELDED OR SOLDERED UNLESS THEY HAVE BEEN ADEQUATELY PURGED OF FUEL FUMES. KEEPING A TANK FILLED WITH CARBON DIOXIDE WILL PREVENT EXPLOSION OF FUEL FUMES.

Fuel tanks may be repaired by welding. Welding shall be by AWS/SFA A5.10 using filler wire ER 4043. See AC 43-13 Section 5 for information on welding.

After repair fuel tank must be tested for leaks.

REPAIRING THE FUEL TANK BY TITANIUM PUTTY

Perform this procedure to repair the fuel tank by using Titanium putty.

NOTE

Safety instructions for the Devcon® product can be found at this link: http://www.devcon.com/techinfo/10760.PDF

1. Thoroughly clean the surface, using Devcon® Cleaner Blend 300 and a clean lint free cloth soaked with alcohol to remove all oil, grease, and dirt.
2. Hand sand around the failure/leakage point with a course wheel or Bright Buff Quick-lok Discs (Very fine grit size is recommended), to create increased surface area for better adhesion. Surface preparation must be of similar size as that of the repair patch size.

NOTE

An abrasive disc pad can only be used provided white mesh is revealed. For metals exposed to sea water or other solution, hand sand with fine abrasive pad and high-pressure-water-blast the area, then leave overnight to allow any salts in the metal to "sweat" to the surface. Rinse to "sweat out" all soluble salts. Perform chloride contamination test to determine soluble salt content (should be no more than 40ppm).

3. Clean surface again with cleaner blend 300/Clean lint free cloth soaked in alcohol to remove all traces of dust or other substances from the blasting.
4. Repair surface as soon as possible to eliminate any changes or surface contaminants.

NOTE

The successful application of Titanium putty repair depends on proper surface preparation. Dust, dirt, oil, grease, rust and dampness can all adversely affect the adhesion of epoxies causing the entire repair to chip, crack, or break away under stress. A clean, dry, slightly roughened surface will ensure maximum adhesion of Titanium putty repair.

5. Add Titanium putty resin to Titanium putty hardener. See Table 28-3 for the correct mixing ratios.

Mixing ratio	Resin to Hardener Ratio
Mix ratio by Volume	3.1:1
Mix ratio by Weight	4.3:1

Table 28-3 Mixing Ratio's of Repair Material

6. Mix thoroughly using square edged mixing sticks (continuously scrape material away from sides and bottom of container) until a uniform; streak-free consistency is obtained.
7. Spread mixed material on repair area and work firmly into substrate to ensure maximum surface contact. Apply the Titanium putty material such that the damage or leakage is centered and repair material completely covers it. In order to avoid sharp edges, repair material must be applied with rounding. The repair thickness must be between 0.0625"to 0.125".

NOTE

Functional (75%) strength is achieved in 16 hrs @75 °F.

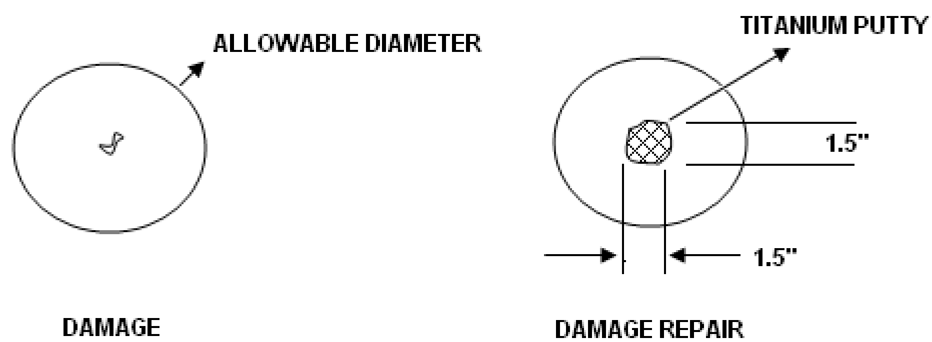

Figure 28-6 Putty Application

This completes the Repairing the Fuel Tank by Titanium Putty procedure.

Section 10-04 Fuel Tank Draining Procedure

This section provides the information and procedure to drain the fuel tank. Listed below is the required equipment to properly drain or defuel the fuel tank. Figure 28-7 and Figure 28-8 show the two drains for the fuel system.

- Approved fire extinguisher
- Approved sealable METAL containers sufficient for expected volume of fuel to be drained
- Fuel resistant hose of appropriate diameter for snug “push fit” on fuel
- Drain valve
- Low-resistance ground wire with alligator clips on both ends

Figure 28-7 Fuel Tank Sump Drain

Figure 28-8 Gascolator Drain

Perform fuel draining procedure in well-ventilated area free of any source of ignition, including electrical switches or self-starting electrical equipment such as shop air compressors.

Disposal of all fuel is to be performed in accordance with local standard practices and procedures.

The airplane fuel tank must be fully drained up to and including the fuel shut off valve and all standard precautions for open fuel tanks and possible uncontained fuel must be taken, prior to any maintenance operation. See below for draining procedure and standard precautions.

NOTE

The use of "fuel quantity indicator" and "fuel gauge" are used throughout this document and are interchangeable. Also, the word liter and litre are used interchangeable.

The terms "gallons" and "ounces" are used throughout this document and relate to the United States fluid measurement system where one US gallon equals 128 fluid ounces.

The term "Liter" is used throughout this document and relates to the basic unit of measurement for volume in the metric system; equal to 61.025 cubic inches or 1.0567 liquid quarts. One Liter is equal to 0.2642 gallons (US).

DRAINING THE FUEL TANK

Perform this procedure to drain the fuel tank.

1. Check that all airplane electrical switches are OFF and will remain OFF for duration of procedure
2. Check the airplane that it is in a well-ventilated location free of any ignition source.
3. Check the fire extinguisher that it is available for immediate use.
4. Place the fuel shutoff valve in the cockpit to the OFF position.

NOTE

Lift the red button in center of valve handle to move the handle to the OFF position.

5. Attach hose to fuel tank sump drain valve.
6. Position approved sealable metal container below airplane with drain hose inside mouth of container.
7. Ground the airplane in accordance with Chapter 51 – *Standard Practices - Structures*.
8. Open fuel drain valve and drain fuel into container. If procedure requires more than one container, change containers as follows:
9. Close the fuel drain valve when the container is nearly full. Allow remaining fuel to drain from hose into container.
10. Close and seal container.
11. Disconnect ground wire from closed container and remove container to safe location.
12. Insert hose into mouth of next container and connect ground wire.
13. Continue draining fuel as described above.
14. When all fuel has been drained from fuel sump drain valve, transfer drain hose and container to Gascolator drain. Leave ground wire connected between airplane and fuel container.
15. Drain fuel from Gascolator. See Figure 28-8.

This completes the Draining the Fuel Tank procedure.

Section 28-20 Distribution

A finger strainer is installed in the tank at a level slightly above the bottom of the fuel sump. A fuel line connects the finger strainer to the Gascolator assembly, mounted approximately below the crew seats and accessible for draining through an opening in the fuselage belly fairing. From the Gascolator, fuel is plumbed to the input of the electric fuel boost pump, which is secured to a bracket on the fuselage center section space frame. Operation of this pump can be selected automatically by the engine FADEC system, or manually by the pilot using the Boost Pump Mode Switch (BPMS) on the instrument panel. Automatic (FADEC controlled) operation is controlled via a fuel pump relay. Manual operation (BPMS “ON” position) bypasses the relay and powers the electric fuel pump directly.

From the pump, fuel flows through an ON/OFF fuel selector valve in the cockpit center console, Figure 28-9, to a “through-hull” fitting on the firewall. A fire-protected fuel line in the engine compartment completes the connection to the input of the engine-driven fuel pump.

Figure 28-9 Fuel Switch Center Console

The engine-driven fuel pump includes a centrifugal separator to remove air or vapor bubbles from the fuel and return them, together with excess fuel not required for combustion at the current power setting, to the fuel tank. This return occurs via a single fuel line that passes aft through the cockpit center console to a one-way check valve and into the fuel tank.

Section 20-01 Fuel system maintenance access

Access to the Gascolator and electric fuel pump is gained by removing the fuselage belly fairing (see Chapter 53 - Fuselage). Maintenance access for engine mounted fuel system components is done by removing the engine cowlings (see Chapter 71 – *Power Plant*).

Section 20-02 Fuel System Procedures

This section contains the procedures to remove and install the gascolator bowl, the gascolator, the fuel boost pump, the fuel valve, and the check valve. Procedures for engine mounted fuel system components are in the current TCM IOF-240-B Maintenance Manual, P/N: M- 22.

GASCOLATOR BOWL REMOVAL

Perform this procedure to remove the gascolator bowl.

1. Defuel airplane as described in the Draining the Fuel Tank procedure on page 16 of this chapter.
2. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
3. Remove fuselage belly fairing. For the procedure to remove the belly panel, see Chapter 53 – *Fuselage*.
4. Place cockpit fuel valve in OFF position. See Figure 28-9.
5. Check that all fuel has been drained from Gascolator.
6. Remove safety wire and unscrew bowl retaining nut and remove Gascolator bowl.
7. Clean Gascolator bowl and filter.

This completes the Gascolator Bowl Removal procedure.

GASCOLATOR BOWL INSTALLATION

Perform this procedure to install the gascolator bowl.

1. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
2. Position bowl on Gascolator body and tighten bowl-retaining ring.
3. Install the safety-wire on to the bowl nut.
4. Check the Gascolator drain valve that it is closed.

CAUTION

During the next set of steps, be watchful for any leaks. If a leak is observed, stop the procedure and investigate the source of the leak and resolve.

5. Place small amount (1 gallon) of fuel in airplane fuel tank and verify there are no leaks.
6. Install fuselage belly fairing. For the procedure to remove the belly panel, see Chapter 53 – *Fuselage*.
7. Refuel airplane as required.
8. Return cockpit fuel valve to ON position.

This completes the Gascolator Bowl Installation procedure.

GASCOLATOR REMOVAL

Perform this procedure to remove the gascolator.

1. Defuel airplane as described in the Draining the Fuel Tank procedure on page 16 of this chapter.
2. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
3. Remove fuselage belly panel. For the procedure to remove the belly panel, see Chapter 53 – *Fuselage*.
4. Place cockpit fuel valve in OFF position.
5. Check that all fuel has been drained from Gascolator.

NOTE

A small amount of residual fuel may drain from fuel lines; check that all necessary precautions are taken.

6. Remove fuel line fittings from Gascolator inlet and outlet.

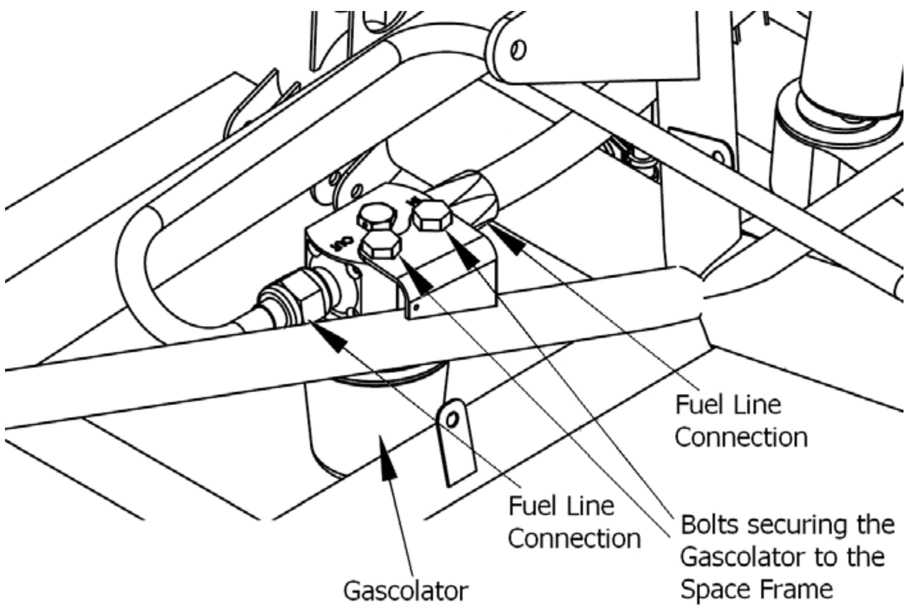

Figure 28-10 Gascolator Mounting

7. Remove bolts securing Gascolator body to airframe.emove bolts securing Gascolator body to airframe;
8. Remove Gascolator.

This completes the Gascolator Removal procedure.

GASCOLATOR INSTALLATION

Perform this procedure to install the gascolator.

1. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
2. Install Gascolator using the two bolts and washers removed in step 7 of the Gascolator Removal procedure. Torque the bolts to 95 in/lbs.
3. Install fuel line fittings to Gascolator inlet and outlet. Torque the fittings to 100 in/lbs.
4. Check the Gascolator drain valve. If it is open, closed the drain valve.

CAUTION

During the next set of steps, be watchful for any leaks. If a leak is observed, stop the procedure and investigate the source of the leak and resolve.

5. Place small amount (1 gallon) of fuel in airplane fuel tank and verify there are no leaks.
6. Install fuselage belly panel. For the procedure to install the belly panel, see Chapter 53 – *Fuselage*.
7. Refuel airplane as required.
8. Return cockpit fuel valve to ON position.

This completes the Gascolator Installation procedure.

ELECTRIC FUEL PUMP REMOVAL

Perform this procedure to remove the electric fuel pump.

1. Defuel airplane as described in the Draining the Fuel Tank procedure on page 16 of this chapter.
2. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
3. Remove fuselage belly panel. For the procedure to remove the belly panel, see Chapter 53 – *Fuselage*.
4. Place cockpit fuel valve in OFF position.
5. Pack the area under the pump with dry absorbent material to catch any fuel that may be in the fuel lines.
6. Remove the three fuel line fittings from fuel pump.

NOTE

A small amount of residual fuel may be spilled from fuel lines; take necessary precautions.

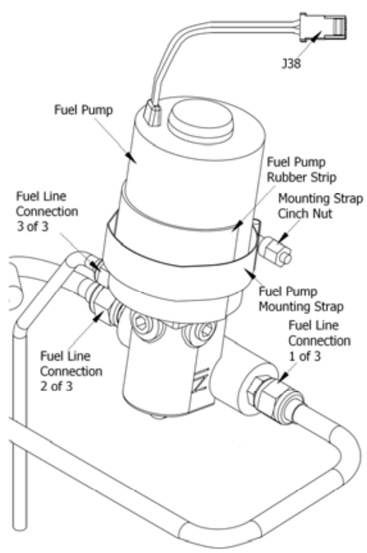

Figure 28-11 Fuel Pump Mounting

7. Separate P/J38 to disconnect the fuel pump electrical connection. Secure P38 away from the fuel pump.
8. Loosen the nut on the fuel pump-mounting strap until the fuel pump is free from the strap.
9. Remove fuel pump.

This completes the Electric Fuel Pump Removal procedure.

ELECTRIC FUEL PUMP INSTALLATION

Perform this procedure to install the electric fuel pump.

1. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
2. Install fuel pump. Position the fuel pump such that the rubber strip is against the space frame.
3. Assemble the pump-mounting strap around the pump.
4. Tighten the nut just enough to hold pump. Check that no wires or tubing are under the strap or pinched by the strap.
5. Torque the nut to 40 inch-pounds.
6. Install the three fuel line fittings to fuel pump inlet and outlet.
7. Torque the fittings to 100 inch-pounds.
8. Connect J38 to P38.
9. Place small amount (1 gallon) of fuel in airplane fuel tank to verify there are no leaks.
10. Install fuselage belly panel. For the procedure to install the belly panel, see Chapter 53 – *Fuselage*.

This completes the Electric Fuel Pump Installation procedure.

ELECTRIC FUEL PUMP CHECKOUT

Perform this procedure to checkout the fuel pump system after installation.

CAUTION

During this procedure, be watchful for any leaks. If a leak is observed, stop the procedure and investigate the source of the leak and resolve.

1. If installed, remove the belly panel. For the procedure to remove the belly panel, see Chapter 53 – *Fuselage*.
2. Apply power to airplane electrical system.
3. Check the cockpit fuel valve that it remains in the OFF position.
4. Place cockpit Boost Pump Mode Switch in ON position. Check the pump that it operates and there are no fuel leaks.
5. Check the propeller area that it is clear of all personnel. Turn FADEC 'A' power on.
6. Place cockpit fuel valve in ON position.
7. Verify indicated fuel pressure of 32 psia ±2 psia on the VM1000 Integrated Engine Instrument Display. (This may take up to 30 seconds).
8. Return the Boost Pump Master Switch to OFF or AUTO position.
9. Place the FADEC A switch in the OFF position.
10. Place the Master Power switch in the OFF position.
11. Replace fuselage panel. For the procedure to install the belly panel, see Chapter 53 – *Fuselage*.
12. Refuel airplane as required.

This completes the Electric Fuel Pump Checkout procedure.

COCKPIT FUEL VALVE REMOVAL

Perform this procedure to remove the cockpit fuel valve.

1. Defuel airplane as described in the Draining the Fuel Tank procedure on page 16 of this chapter.
2. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
3. Check that the tank and Gascolator drain valves are closed.
4. Remove belly panel. For the procedure to remove the belly panel, see Chapter 53 – *Fuselage*.
5. Position cockpit fuel valve to ON, this will allow as much fuel as possible to return to fuel tank.
6. Move selector between ON and OFF.
7. Push down on the edge of the detent knob to remove access cover, remove “C” clip.
8. Remove detent knob.
9. Remove Allen screw from selector handle and lift off.
10. Remove fuel selector placard.
11. Position an absorbent material below the fuel valve to contain any spillage.
12. Remove fuel lines from inlet and outlet of fuel valve.
13. Remove screws and nuts holding fuel valve to airframe.
14. Remove the valve.

This completes the Cockpit Fuel Valve Removal procedure.

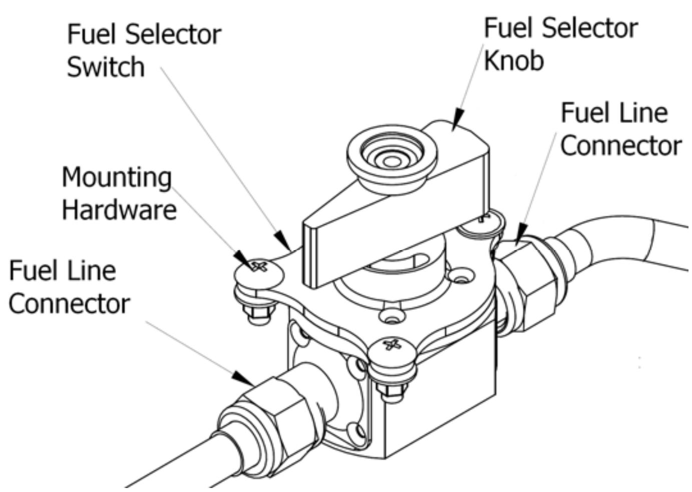

Figure 28-12 Fuel Selector Valve

COCKPIT FUEL VALVE INSTALLATION

Perform this procedure to install the cockpit fuel valve.

1. If installed, remove the belly panel. For the procedure to remove the belly panel, see Chapter 53 – *Fuselage*.
2. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
3. Secure screws and nuts holding fuel valve to airframe. Torque the nuts to 95 inch-pounds.
4. Connect fuel lines on inlet and outlet of fuel valve. Torque the fuel connectors to 100 inch-pounds.
5. Install fuel selector placard.
6. Install selector handle. Secure handle to switch with Allen screw.
7. Install detent knob. Install the "C" clip. Install detent access cover.
8. Move selector through on and off to check for operation.

CAUTION

During the next set of steps, be watchful for any leaks. If a leak is observed, stop the procedure and investigate the source of the leak and resolve.

9. Place small amount (1 gallon) of fuel in airplane fuel tank.
10. Position the cockpit fuel valve to ON.
11. Place the Master Power Switch in the ON position.
12. Place the Boost Pump Master Switch in ON position. Verify that pump operates and that no fuel leaks from inlet or outlet of fuel valve.
13. Return the Boost Pump Master Switch to OFF or AUTO position.
14. Place the Master Power Switch in the OFF position.
15. Remove and discard absorbent material from under valve.

NOTE

Lift the red button in the center of the Fuel Switch handle. Then move the handle to the OFF position.

16. Return cockpit fuel valve to OFF position.
17. Install the belly panel. For the procedure to install the belly panel, see Chapter 53 – *Fuselage*.

This completes the Cockpit Fuel Valve Installation procedure.

CHECK VALVE REMOVAL

Perform this procedure to remove the check valve.

1. Defuel airplane as described in the Draining the Fuel Tank procedure on page 16 of this chapter.
2. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
3. Remove belly panel. For the procedure to remove the belly panel, see Chapter 53 – *Fuselage*.
4. Position absorbent material below fuel/vapor return check valve to prevent spillage.
5. Remove fuel lines from inlet and outlet of check valve.
6. Remove the clamp securing the check valve to the space frame.
7. Remove check valve from airframe.

This completes the Check Valve Removal procedure.

CHECK VALVE INSTALLATION

Perform this procedure to install the check valve.

1. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
2. If installed, remove the belly panel. For the procedure to remove the belly panel, see Chapter 53 – *Fuselage*.
3. Position replacement check valve between fuel line fittings. Check the arrow on check valve housing that it points aft. See Figure 28-13.
4. Attach check valve to the space frame using the clamp removed in step 6 of the Check Valve Removal procedure.
5. Connect fuel lines to inlet and outlet of check valve and torque to 100 in/lbs.
6. Remove and discard absorbent material.
7. Install belly panel. For the procedure to install the belly panel, see Chapter 53 – *Fuselage*.

This completes the Check Valve Installation procedure.

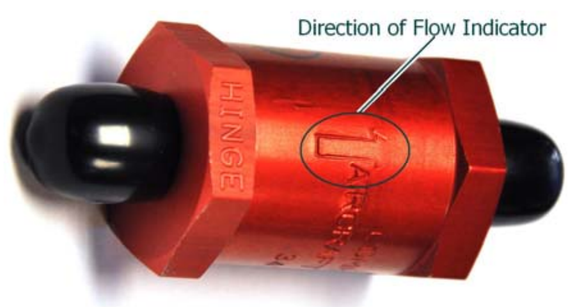

Figure 28-13 Check Valve Showing The Arrow That Indicates The Direction Of The Flow – Arrow Points Aft

FUEL LINE REMOVAL

Perform this procedure to remove any of the fuel lines. If any step is unique to a particular fuel line, that difference will be noted in the step. Use Figure 28-14 as a guide in locating the fuel line and mounting hardware. Figure 28-14 is not a scaled drawing of the fuel system and should be used as a reference only.

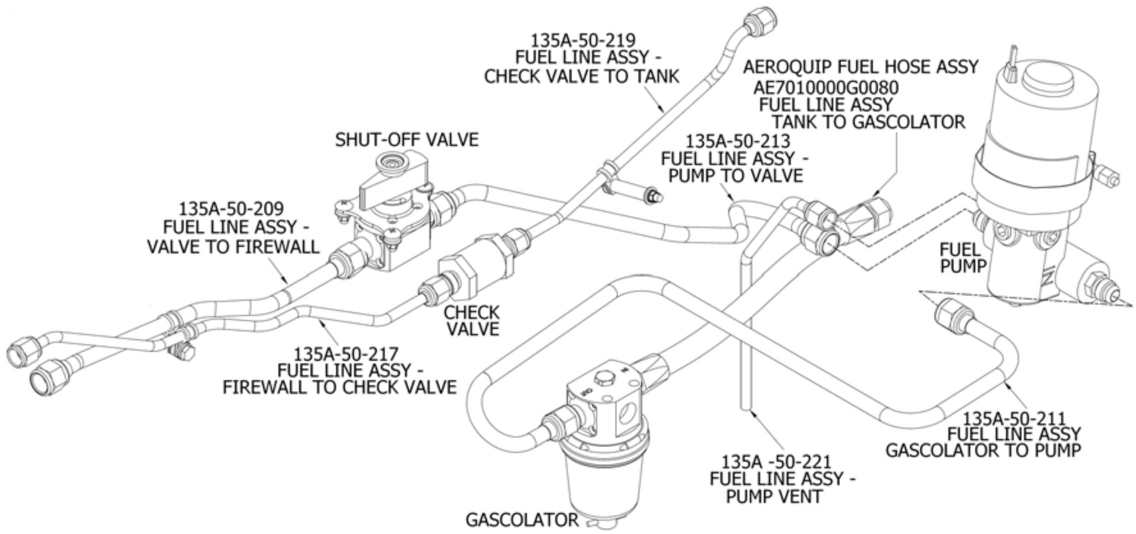

Figure 28-14 Fuselage Fuel Lines

1. Defuel airplane as described in the Draining the Fuel Tank procedure on page 16 of this chapter.
2. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
3. Remove belly panel. For the procedure to remove the belly panel, see Chapter 53 – *Fuselage*.
4. Position absorbent material below the connectors for the fuel line being replaced.
5. Place one wrench on the fuel line connector and a second wrench on the item connected to the fuel line. Disconnect the fuel line on each end of the fuel line.
6. On the Check Valve to Tank fuel line, there is a clamp holding the fuel line to the space frame. Disassemble the clamp holding the fuel line to the space frame.
7. On the Valve to Firewall fuel line, there is a clamp holding the fuel line to the space frame. Disassemble the clamp holding the fuel line to the space frame.
8. On the Firewall to Check Valve fuel line, there is a clamp holding the fuel line to the space frame. Disassemble the clamp holding the fuel line to the space frame.
9. Carefully remove the fuel line from the space frame.

This completes the Fuel Line Removal procedure.

FUEL LINE INSTALLATION

Perform this procedure to install any of the fuel lines in the space frame. If any step is unique to a particular fuel line, that difference will be noted in the step. Use Figure 28-14 as a guide in locating the fuel line and mounting hardware. Figure 28-14 is not a scaled drawing of the fuel system and should be used as a reference only.

1. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
2. If installed, remove the belly panel. For the procedure to remove the belly panel, see Chapter 53 – *Fuselage*.
3. Carefully feed the fuel line into the space frame.
4. On the Check Valve to Tank fuel line, there is a clamp holding the fuel line to the space frame. Loosely assemble the clamp holding the fuel line to the space frame. Do not tighten the clamp at this point.
5. On the Valve to Firewall fuel line, there is a clamp holding the fuel line to the space frame. Loosely assemble the clamp holding the fuel line to the space frame. Do not tighten the clamp at this point.
6. On the Firewall to Check Valve fuel line, there is a clamp holding the fuel line to the space frame. Loosely assemble the clamp holding the fuel line to the space frame. Do not tighten the clamp at this point.
7. Connect the fuel line to the item associated with that fuel line.
8. Use two wrenches, one on the connector and the second on the item.
9. Torque the fuel line connector to 100 inch-pounds.
10. If there is a clamp on the fuel line, tighten the clamp at this point.
11. Torque the clamp hardware to 95 inch-pounds.
12. Place small amount (1 gallon) of fuel in airplane fuel tank to verify there are no leaks.
13. Remove and discard absorbent material.
14. Install fuselage belly panel. For the procedure to install the belly panel, see to Chapter 53 – *Fuselage*.

This completes the Fuel Line Installation procedure.

Section 20-03 Fuel System Electrical

This section contains information on the electrical system for the fuel system within the space frame.

The electrical system consists of circuit breaker, CB029, fuel boost pump switch, SW007, fuel boost pump relay, K003, the recirculating valve, mounted on the engine, and the fuel boost pump. For a schematic of the fuel boost pump circuit, see Chapter 91 – *Wiring Diagrams*.

The fuel boost pump switch, SW007, mounts to the center console, below the Avionics panel. The fuel boost relay, K003, mounts to the power distribution panel, behind the Instrument panel, see Figure 28-15. The recirculating valve is mounted on the engine. The fuel boost pump mounts to the space frame, behind the pilot's seat.

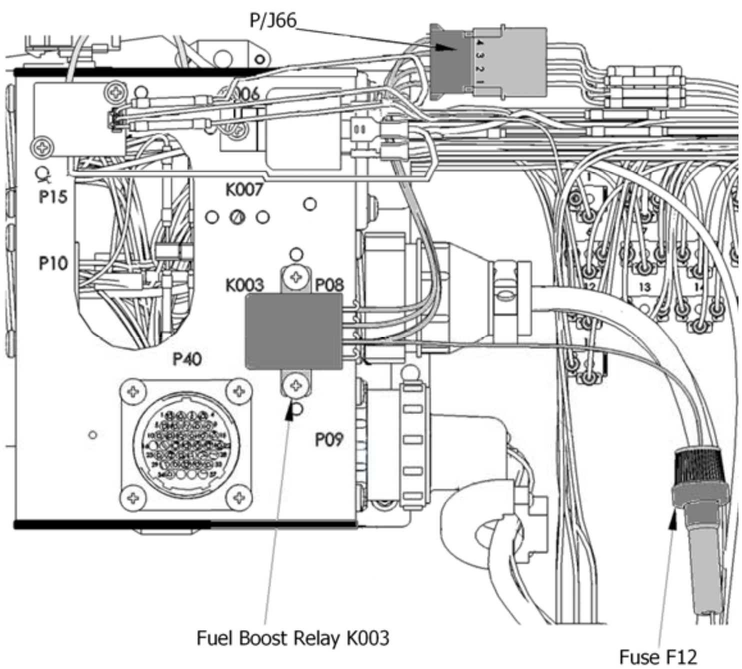

Figure 28-15 Fuel Boost Relay Location

With the switch in the center OFF position, there is no power available for the fuel boost pump or the recirculating valve. With the switch in the ON position, this will supply power directly to the fuel boost pump and the recirculating valve. Applying power to the recirculating valve will cause the valve to open. With the switch in the AUTO position, this will supply power to the fuel boost relay. Control of the relay comes from the FADEC ECU. When the FADEC ECU calls for fuel boost, the relay contacts close. The contacts supply power to the fuel boost pump and the recirculating valve.

Section 20-04 Fuel System Electrical Procedures

This section contains the procedures to remove and install the fuel boost pump switch and the fuel boost relay. The procedures for removing and installing the electric Fuel Boost Pump is in Section 20-02 Fuel System Procedures on page 17 of this chapter. The procedures for removing and installing the recirculation valve are in Chapter 73 – *Engine Fuel and Control*.

FUEL BOOST PUMP SWITCH REMOVAL

Perform this procedure to remove the fuel boost pump switch.

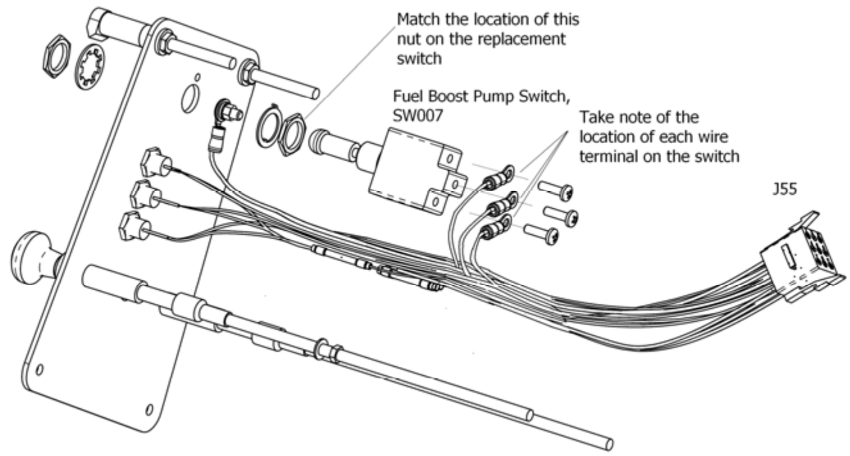

Figure 28-16 Fuel Boost Pump Switch in Center Console

1. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
2. Remove the four screws securing the side access panel on the center console.
3. Remove the nut and washer securing the switch to the center console.
4. Reaching through the side access, guide the switch back through the center console panel and out the side access.
5. Note the location of the three wires going to the switch.
6. Remove the three screws securing the wires to the switch.
7. Remove the switch from the airplane.

This completes the Fuel Boost Pump Switch Removal procedure.

FUEL BOOST PUMP SWITCH INSTALLATION

Perform this procedure to install the fuel boost pump switch.

1. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
2. Attach the three wire removed in step 6 of the Fuel Boost Pump Switch Removal procedure on page 32 of this chapter.
3. Carefully guide the switch through the side access and into the center console panel.
4. Secure the switch using the supplied washer and nut.
5. Test the switch for proper operation.

This completes the Fuel Boost Pump Switch Installation procedure.

FUEL BOOST RELAY REMOVAL

Perform this procedure to remove the fuel boost relay.

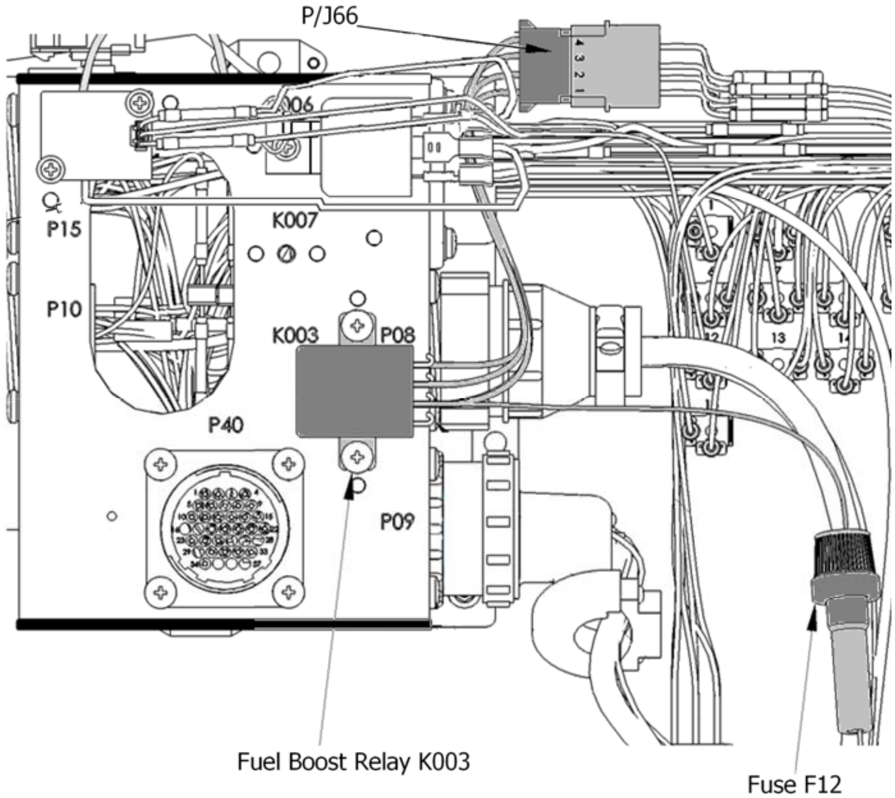

Figure 28-17 Fuel Boost Relay Removal

1. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
2. Remove the Instrument Console panel. For the procedure to remove the panel, see Chapter 31 – *Indicators and Recorders*.
3. Remove the two screws securing the fuel boost relay, K003, to the Power Distribution Panel. Retain screws for installation.
4. Separate connector P/J66.
5. Disassemble fuse F12.
6. Remove relay from airplane.
7. If installing the relay later, temporally secure the instrument console panel.

This completes the Fuel Boost Relay Removal procedure.

FUEL BOOST RELAY INSTALLATION

Perform this procedure to install the fuel boost relay.

1. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
2. If the instrument console panel is place, remove the instrument console panel.
3. Install the relay, using the hardware removed.
4. Assemble connector P/J66.
5. Assemble the fuse F12.
6. Test the relay's operation.
7. Install the instrument panel console. For the procedure to install the instrument console panel, see Chapter 31 – *Indicators and Recorders*.

This completes the Fuel Boost Relay Installation procedure.

Section 20-05 Airframe Fuel System Operational Check and Inspection

This section contains the airframe fuel system operational check and inspection procedure. Perform this procedure anytime after performing any work of the fuel system.

Refer to Figure 28-18 during this procedure.

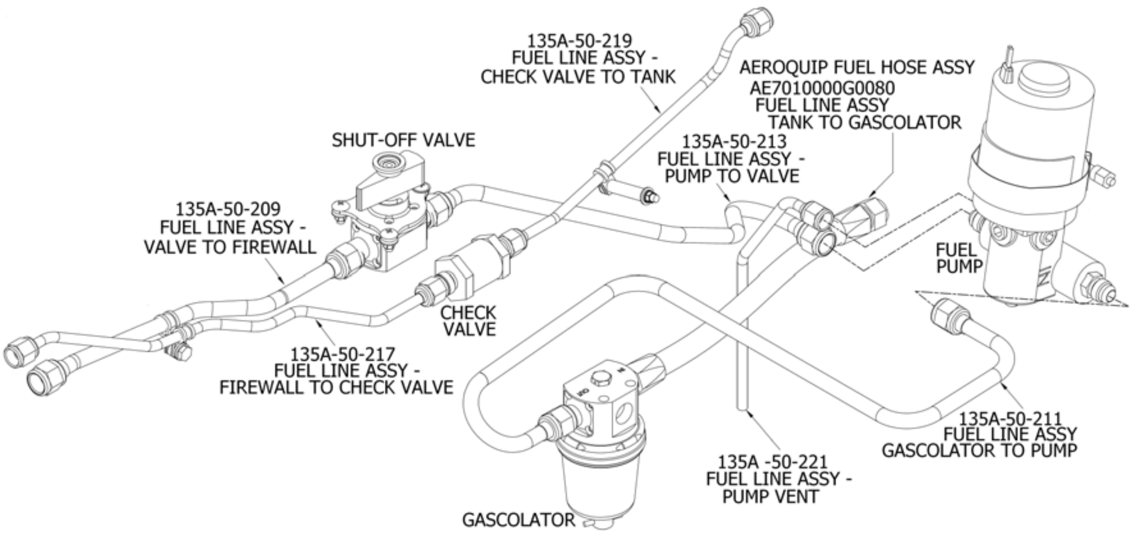

Figure 28-18 Fuselage Fuel Lines

NOTE

Steps in the procedure to follow require an engine run up. Ensure an appropriate safe engine run up area is used while performing these steps.

AIRFRAME FUEL SYSTEM OPERATIONAL CHECK AND INSPECTION

Perform this procedure to inspect and check the fuel system in the airframe.

NOTE

This procedure requires a minimal amount of fuel to be in the system.

1. Remove the belly panel from the airplane in accordance with Chapter 53 – *Fuselage, Belly Panel Removal.*
2. Inspect each fuel system connection system shown in Figure 28-18 for evidence of leaks.
3. Inspect along each of the fuel lines looking for evidence of wear, chafing, unusual bending, or flexing, or creasing of any of the fuel lines.
4. Inspect around the fuel tank for evidence of leaking, wear, chafing, unusual bending, flexing, or creasing.

CAUTION

Perform the following steps from the cockpit with an exterior observer watching for leaks. Discontinue testing in the event leaks are seen and correct condition in accordance with Fuel System Troubleshooting table at the end of this section before proceeding.

5. Position the split master switch ON. Position FADEC A and B power switches ON. Locate circuit breaker CB003, START, and pull to OPEN.
6. Position cockpit fuel selector OFF.
7. Position the boost pump switch to ON. Listen for the pump running and verify the VM1000FX engine display indicates a nominal 15 psig (bottom of scale)
8. Inspect each fuel system connection as shown in Figure 28-18 for evidence of leaks.
9. Inspect along each of the fuel lines looking for evidence of leaking of any of the fuel lines. Inspect around the fuel tank for evidence of leaking.
10. Position cockpit fuel selector to ON and verify the VM1000FX engine display indicates a nominal 30 psig +/- 2 psig (green arc) within 10 seconds.
11. Inspect each fuel system connection as shown in Figure 28-18 for evidence of leaks.
12. Inspect along each of the fuel lines looking for evidence of leaking of any of the fuel lines. Inspect around the fuel tank for evidence of leaking.
13. Position the boost pump switch to AUTO. Verify the electric boost pump stops running.

14. Position the boost pump switch to OFF. Position the Master Switch to OFF. Push in circuit breaker CB003, START, to CLOSE the breaker.
15. Move airplane to a safe designated run up area and set the parking brake.
16. Perform a normal engine start-up in accordance with the current Liberty Aerospace Airplane Flight Manual. Run engine until the oil temperature is at or above 75°.
17. Cycle the engine throttle through 1200 RPM and verify the boost pump should stops running between 1200 and 1250 RPM.
18. Cycle the engine throttle back below 1200 RPM and verify the boost pump starts running between 1200 and 1150 RPM.
19. Continue to cycle the engine throttle to IDLE and verify the boost pump continues to operate.
20. Shut down the engine in accordance with the current Liberty Aerospace Airplane Flight Manual. Pull circuit breaker CB003, START, to OPEN.
21. Verify Master Switch is OFF.
22. Inspect each fuel system connection shown in Figure 28-18 and the fuel tank for evidence of leaks.
23. Install the belly panel on the airplane in accordance with Chapter 53 – *Fuselage*, *Belly Panel Installation*.

This completes the Airframe Fuel System Operational Check and Inspection procedure.

Section 20-06 Fuel System Troubleshooting

This section contains information that will aid in troubleshooting issues with the fuel system located in the fuselage. For troubleshooting information on the fuels systems associated with the engine, see Chapter 73 – *Engine Fuel and Control*.

Complaint	Possible Cause	Remedy
Smell of fuel in airplane	Loose or defective fuel line fittings	Tighten or replace
Evidence of fuel leaks (stains on bottom of airplane aft of sump or Gascolator drains)	Defective drain valves	Tighten or replace
	Defective fuel line fittings	Tighten or replace
Electric fuel pump does not operate with engine not running and fuel boost pump switch ON (manual).	Defective fuel pump circuit breaker	Replace
	Defective fuel boost pump switch	Replace
	Defective electric fuel pump	Replace
	Defective wiring	Repair
	Defective fuel pump relay	Replace
Electric fuel pump does not operate with engine running and fuel boost pump switch in AUTO. Engine RPM ≤1200 rpm	Fuel boost pump switch faulty	Replace
	Defective fuel pump relay	Replace
	Defective wiring	Repair
	Defective electric fuel pump	Replace
	FADEC fails to command pump to turn on	Troubleshoot in accordance with the TCM OI-22
Boost pump continues to operate above 1200 RPM	Faulty fuel pump relay	Replace relay
	Faulty FADEC fuel pump relay control wiring	Repair
	Engine driven pump failure	Replace engine driven pump
	FADEC fault	Troubleshoot in accordance with the TCM OI-22
Low VM1000FX displayed boost pump operating pressure with selector in the ON position.	Insufficient voltage	Check, charge battery
	Foreign object in fuel system	Clean fuel system
	Defective electric pump	Replace pump
	Gascolator contaminated	Clean Gascolator
	Fuel system leak	Inspect for leaks and repair
	Faulty fuel pressure transducer	Replace transducer(s)
	Faulty transducer wiring	Repair

Complaint	Possible Cause	Remedy
Low VM1000FX displayed fuel pressure with engine running and fuel boost pump switch OFF.	Fuel filters clogged	Replace (see Chapter 73 – *Engine Fuel and Control*)
	Engine driven fuel pump pressure set incorrectly.	Adjust engine driven fuel pump in accordance with the TCM OI-22 Chapter 6
	Engine driven fuel pump failure	Replace
	Fuel pressure transducer fault	Run Level I Diagnostic to confirm fault
		Replace transducer(s)
Unusually high fuel consumption, evidence of fuel stains aft of fuel pump drain, lower cowling	Fuel/vapor return line or check valve may be blocked	Check line, replace check valve
Fuel pressure above ambient displayed on VM1000FX display with fuel selector in the OFF position, boost pump ON (manual) and engine not running.	Fuel selector valve leaking	Replace selector valve
	Defective fuel pressure transducer	Replace transducer(s)
High boost pump operating pressure displayed on VM1000FX with selector in the ON position.	Faulty boost pump internal relief valve.	Replace pump
	Defective fuel pressure transducer	Replace transducer(s)
	Faulty fuel pressure transducer wiring	Repair wiring

Table 28-4 Fuel System Troubleshooting Chart

Section 28-40 Fuel Indication

A capacitive probe installed in the fuel tank provides a signal for the instrument panel mounted fuel contents gauge.

The FADEC system senses the fuel pressure (see Chapter 72 - Engine) and displays the information on the integrated engine instrument system VM1000FX (see Chapter 77 – *Engine Indicating*).

Access to the capacitive fuel sender is gained by removing a cover plate in the left seatback/headrest structure. See Figure 28-19.

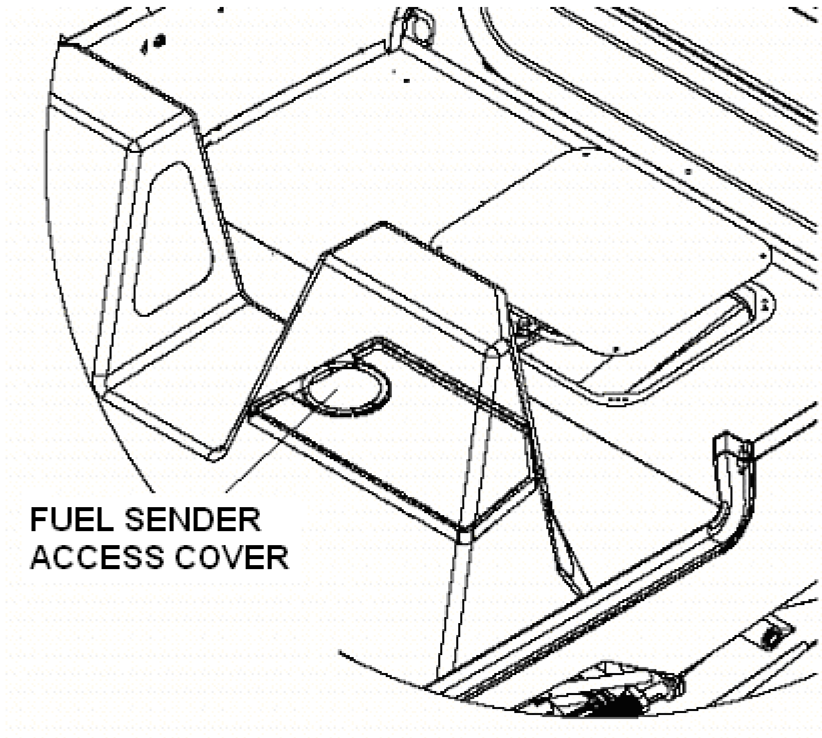

Figure 28-19 Fuel Sender Access Cover

Access to the cockpit fuel shutoff valve and fuel/vapor return check valve is gained by removing the fuselage belly panel.

Section 40-01 Fuel Level Sender Removal and Replacement

Unless replacement sender unit is available for immediate installation, use heavy aluminum foil, "Ty-wrap," tape, etc., to make temporary vapor-tight closure of fuel level sender opening. Do not use plastic sheeting ("baggies," "Visqueen," etc.).

FUEL SENDER REMOVAL

Perform this procedure to remove the fuel sender.

CAUTION

Comply with fire prevention precautions as described in the Draining the Fuel Tank procedure on page 16 of this chapter.

CAUTION

Check that all airplane electrical switches are OFF and will remain OFF for the duration of procedure.

1. Remove fuel level sender access plate.
2. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
3. Disconnect wires from fuel level sender; mark to ensure correct reconnection.
4. Loosen and unscrew fuel level sender to mounting flange.
5. Remove fuel level sender. If necessary, immediately close fuel sender opening with appropriate temporary materials (see above).

This completes the Fuel Sender Removal procedure.

FUEL SENDER INSTALLATION

Perform this procedure to install the fuel sender.

CAUTION

Comply with fire prevention precautions as described in the Draining the Fuel Tank procedure on page16 of this chapter.

CAUTION

Check that all airplane electrical switches are OFF and will remain OFF for the duration of procedure.

NOTE

This procedure calls for the use of Loctite P/N: 39901, anti-seize compound. Take care not to allow any of this compound to contaminate the interior of fuel tank. .

1. Position the Main Battery Switch, Alternator Switch, FADEC A and B and the ignition switch in the OFF position.
2. Remove temporary covering from fuel level sender mounting flange.
3. Apply Loctite P/N: 39901, anti-seize compound to the first three threads of fuel sender.
4. Reinstall fuel level sender by threading back into threaded flange on the top of the fuel tank.
5. Reconnect wires to fuel level sender.
6. Perform the fuel system functional check as described in Section 40-08 - Calibration Test on page 48 of this chapter.
7. Replace fuel sender access plate.

This completes the Fuel Sender Installation procedure.

Section 40-02 *Capacitive Fuel Sender*

The Liberty XL-2 uses a Mitchell Aircraft Product capacitive fuel sender for measuring fuel quantity when mounted inside of the fuel tank. The probe has a measurable range of 16.380¬-inches over which the entire tank contents may be read. Correct probe installation will have 3 of the 1.5 NPT threads fully engaged with the tank boss. A nominal 0.300-inch space will remain between the tank boss and the probe head flange as shown in Figure 28-20.

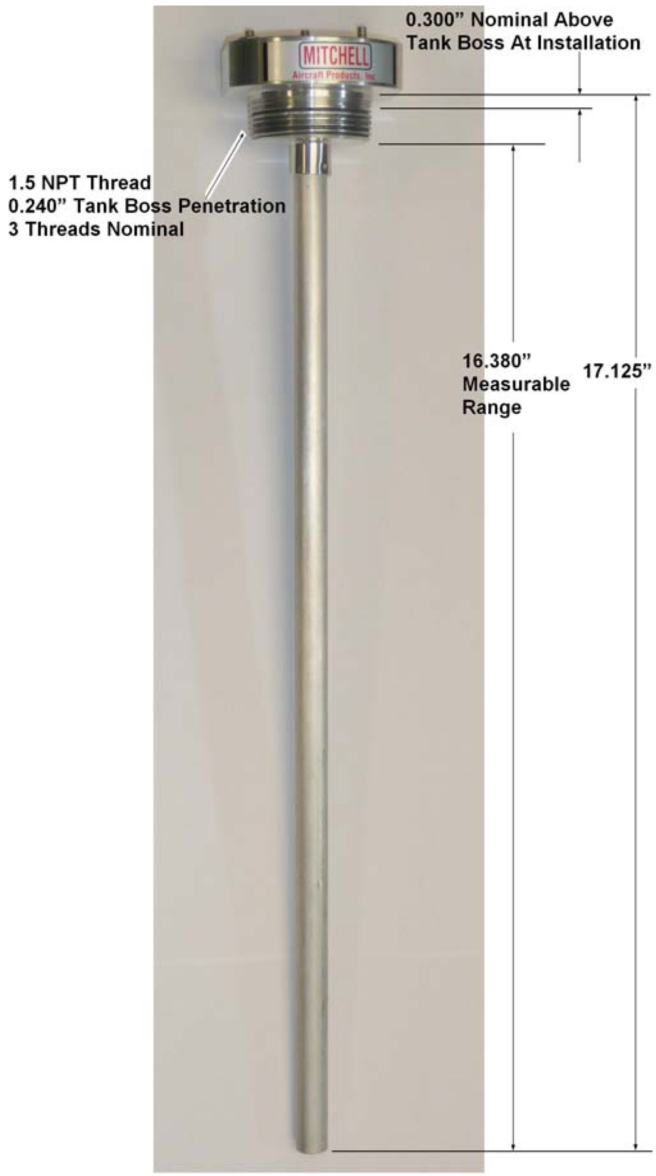

Figure 28-20 Capacitive Fuel Sender

Section 40-03 Level Flight Attitude

The fuel quantity gauge system accuracy is directly affected by airplane pitch and roll attitudes. The fuel quantity gauging system is designed to be the most accurate in level flight cruise attitude (–1.7° nose up in pitch and 0° roll). The fuel indication check is conducted in this attitude in accordance with FAR 23.1337(b)(1).

Section 40-04 Unusable and Total Fuel Quantity

Table 28-5 shows total, usable, and unusable fuel quantities for the airplane.

Unusable	Usable	Total
1.5 Gallons	28.0 gallons	29.5 gallons
5.7 Liters	106 Liters	111.7 Liters

Table 28-5 Unusable, Usable, and Total Fuel Quantities

Section 40-05 Fuel Indication Calibration

Fuel gauges calibrated by this procedure are configured in one of two engineering unit standards as shown in Figure 28-21. Gauges calibrated in Liters are marked in increments from 0-106 liters. Gauges calibrated in gallons are marked in increments from 0-28 US GALS. The gauges are electrically identical; therefore, either gauge may be installed with no change to the fuel sender or related interconnect wire harnesses. The gauges are fitted with dashpots to suppress erratic needle movement while in flight. In the static ground test environment dashpot drag will slow needle response during the calibration process. A slight tap on the gauge face will correct needle position. Settle times of 30 seconds are typical and do not represent faulty operation.

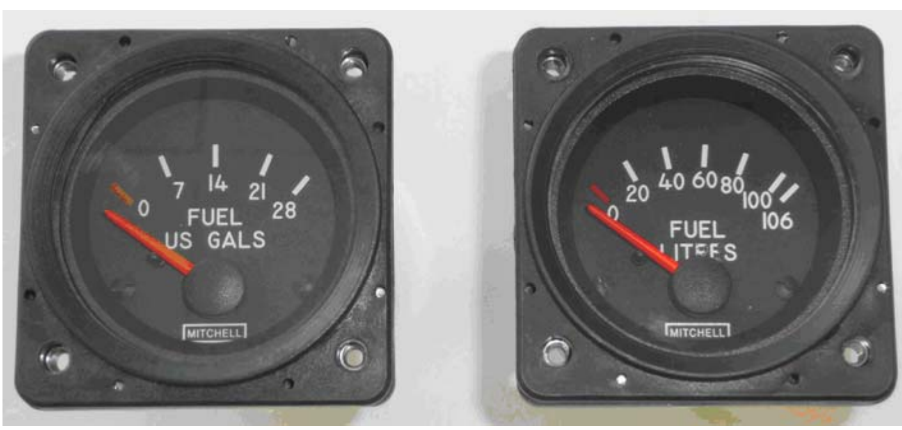

Figure 28-21 Gallon and Liter Fuel Gage

Section 40-06 Fuel Quantity by Weight Tool

Calibration accuracy is directly dependent upon accurately tracking the amount of fuel added to or removed from the system. Tracking fuel quantity by weight (compensated for ambient temperature) affords the most correct and repeatable results. Figure 28-22 provides a graph to aid in the calculation of Aviation fuel grade 100LL weight per Gallon / Liter over a wide ambient temperature range.

Aviation fuel grade 100LL has a specific expansion rate. Weight per unit of volume compensated for temperature is determined by the formula:

$$(-0.008 \times \text{Temp } ^\circ C) + 6.00 = \text{Pounds per Gallon (Lbs)}$$

$$1 \text{ Gallon (G)} = 3.785 \text{ Liters (L)}$$

Table 28-8 shows the list of required equipment and recommended suppliers to perform the fuel calibration test. Figure 28-24 and Figure 28-25 shows some of the equipment used to calibrate the fuel system.

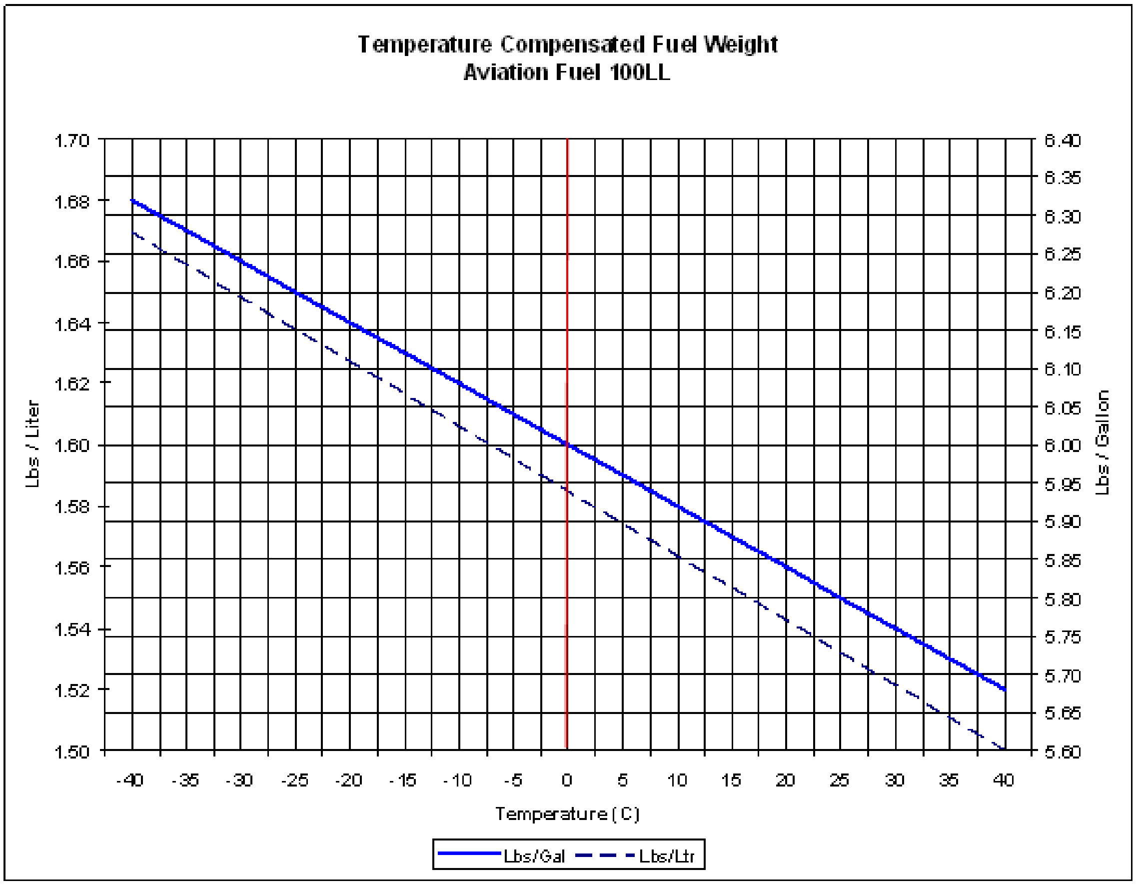

Figure 28-22 Temperature Compensated Fuel Weight - 100LL

Part Number	Part Description	Supplier	Qty
HS-30 (OR EQ.)	Scale 3 digit accuracy – 60 lb capacity (min)	ASI	1
50841 (OR EQ)	Fuel Can – 6 to 8 Gal capacity – removable nozzle	Blitz USA Inc.	3
66490 (OR EQ.)	Fuel Pitcher – 4 liter capacity	US Plastics	1
GC-30 (OR EQ.)	Fuel Caddy system	Handy Products Inc.	1
SHOP AID	Fuel Tank Drain Valve Assembly	Liberty	1
SHOP AID	Safety Ground Wire Assembly	Liberty	1
SE1275A (OR EQ.)	Battery Charger	Schumacher Elec.	1
92346 (OR EQ.)	Digital Level	SmartTool	1

Table 28-6 Required Equipment

Section 40-07 Fill Fuel Tank With 100LL Blue Or 100 Green Aviation Fuel Only.

In the calibration steps to follow, adjust the fuel level sender by means of controls located on top of the unit. See Figure 28-23 Prior to making any adjustment, verify the correct control has been identified for the calibration step being performed. Make only small adjustments allowing sufficient time for the gauge to settle between adjustments.

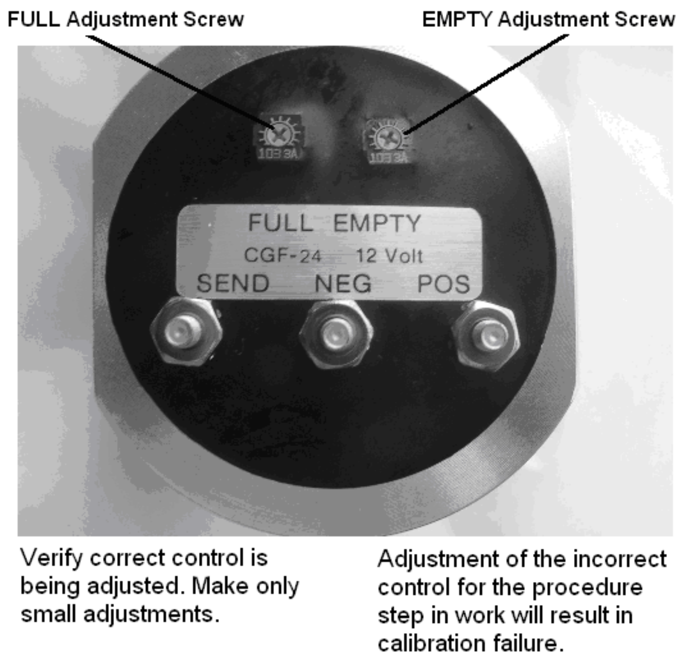

Figure 28-23 Capacitive Fuel Sender Adjustment

Fuel Quantity Indicated by Fuel Gauge (US GALLONS)	Allowable Fuel Gage Error Tolerance (US GALLONS)
1.5[1]	± 0.2
0	± 0.5
7	± 0.8
14	± 1.1
21	± 1.4
28	± 1.7

Fuel Quantity Indicated by Fuel Gauge (LITERS)	Allowable Fuel Gage Error Tolerance (LITERS)
5.7[2]	± 0.7
0	± 1.9
20	± 2.8
40	± 3.6
60	± 4.5
80	± 5.3
100	± 6.2
106	± 6.4

[1] Unusable fuel in tank, Gallons
[2] Unusable fuel in tank, Liters

Table 28-7 Allowable Fuel Quantity Error Tolerance

Section 40-08 Calibration Test

This section contains the calibration tests for the fuel system.

Table 28-8 shows a list of required items to perform the calibration tests. Figure 28-24 and Figure 28-25 give an illustration of some the items need to perform the calibration tests.

Part Number	Part Description	Supplier	Qty
HS-30 (OR EQ.)	Scale 3 digit accuracy – 60 lb capacity (min)	ASI	1
50841 (OR EQ)	Fuel Can – 6 to 8 Gal capacity – removable nozzle	Blitz USA Inc.	3
66490 (OR EQ.)	Fuel Pitcher – 4 liter capacity	US Plastics	1
GC-30 (OR EQ.)	Fuel Caddy system	Handy Products Inc.	1
SHOP AID	Fuel Tank Drain Valve Assembly	Liberty	1
SHOP AID	Safety Ground Wire Assembly	Liberty	1
SE1275A (OR EQ.)	Battery Charger	Schumacher Elec.	1
92346 (OR EQ.)	Digital Level	SmartTool	1

Table 28-8 Required Equipment

Figure 28-24 Test Fuel Containers

Figure 28-25 Tank Drain Valve Assembly (Typical)

CALIBRATION TEST – CONFIGURATION

1. Attach battery charger to airplane battery in accordance with approved practices and charge battery during fuel indication check
2. Check the airplane tires for correct air pressure and drain the airplane's fuel storage system completely empty of fuel in accordance with approved practices.
3. Record airplane serial number and serial numbers of capacitive fuel sender and fuel indicator installed in airplane.
4. Position the airplane on a level surface.
5. Referring to Figure 28-26, place a digital level (SmartTool 92346 or equivalent) between the two XL-2 seat backs and parallel with the centerline of airplane. Digital level should indicate a pitch of 1.7 ± 0.2° nose up. If the level does not indicate 1.7 ± 0.2°, adjust the nose gear by means of tire inflation or shim as required to achieve 1.7 ± 0.2°.
6. Rotate the digital level 90° so that it is parallel with the airplanes' wings. Digital level should indicate 0 ± 0.2°. If the level does not indicate a roll axis position of 0 ± 0.2°, adjust the main landing gear by means of under wheel shim or tire inflation to achieve 0 ± 0.2°.

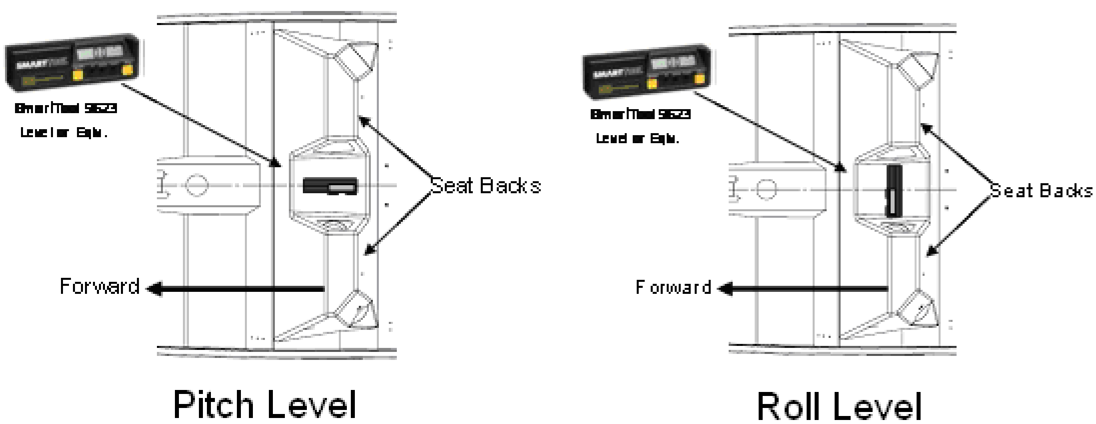

Figure 28-26 Level Placement

7. Rotate digital level 90° until it is parallel with the centerline of airplane. Periodically check airplane attitude while performing indication check to ensure 1.7 ± 0.2° nose up and 0 ± 0.2° wings level is maintained during fuel indication check.

WARNING

TO GUARD AGAINST THE POSSIBILITY OF A SPARK IGNITING FUEL FUMES, A GROUND WIRE MUST BE ATTACHED TO THE AIRPLANE'S GROUND POINT (EXHAUST PIPE) BEFORE THE FUEL CAP IS REMOVED.

8. Ground the airplane in accordance with Chapter 51 – *Standard Practices Structures*.
9. Drain the airplane fuel tank and Gascolator.
10. Close Gascolator valve on completion.
11. Remove the fuel tank drain valve and install the valve assembly shown in Figure 28-25. Verify the valve lever is in the CLOSED position.
12. Weigh the fuel can without nozzle and record for reference. Set the scale TARE feature to “zero” the fuel can weight. (Refer to scale OEM manual for TARE feature information).

CALIBRATION TEST – GALLONS FUEL GAUGE

Perform the following procedure to calibrate the fuel gauge for gallons.

1. Start with a dry tank.
2. Turn the Master Switch – ON.
3. On the airplane's OAT read the current ambient temperature in degrees Celsius.
4. Turn the Master Switch – OFF.
5. Add 1.5 gallons by weight.
6. Turn the Master Switch – ON.
7. Verify the fuel gauge reads zero ± 0.5 Gals. See Table 28-7.
8. As required, make small adjustments to the fuel sender "Empty" pot so the fuel gauge reads 0 gallons.
9. Turn the Master Switch – OFF.
10. Add 21 gallons of 100LL Blue or 100 Green fuel by weight.
11. Turn the Master Switch – ON.
12. Verify fuel gauge reads 21 ± 1.4 Gals. See Table 28-7.
13. As required, make small adjustments to the fuel sender "Full" pot until the gauge reads 21 gallons.
14. Turn the Master Switch – OFF.
15. Add 7 gallons of 100LL Blue or 100 Green fuel.
16. Turn the Master Switch – ON.
17. Verify the fuel gauge reads 28 ± 1.7 Gals. See Table 28-7.
18. Record gauge reading in Table 28-9 space provided.
19. Turn the Master Switch – OFF.
20. Prepare for tank draining by configuring the fuel can as shown in Figure 28-16.

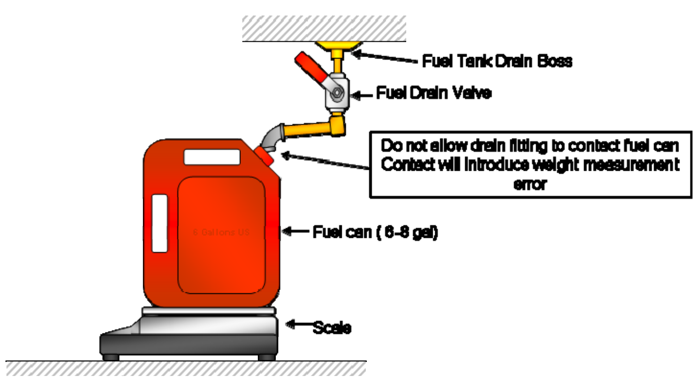

Figure 28-27 Fuel Drain Equipment Set Up

21. Drain 7 gallons of fuel by weight.
22. Turn the Master Switch – ON.
23. Verify gauge reads 21 ± 1.7 Gals. See Table 28-7.
24. Record gauge reading in Table 28-9 space provided.
25. Turn the Master Switch – OFF.
26. Drain 7 gallons of fuel by weight.
27. Turn the Master Switch – ON.
28. Verify gauge reads 14 ± 1.1 Gals. See Table 28-7.
29. Record gauge reading in Table 28-9 space provided.
30. Turn the Master Switch – OFF.
31. Drain 7 gallons of fuel by weight.
32. Turn the Master Switch – ON.
33. Verify gauge reads 7 ± 0.8 Gals. See Table 28-7.
34. Record gauge reading in Table 28-9 space provided.
35. Turn the Master Switch - OFF.
36. Drain 7 gallons of fuel by weight.
37. Turn the Master Switch – ON.
38. Verify gauge reads 0 ± 0.5 Gals. See Table 28-7.
39. Record gauge reading in Table 28-9 space provided.
40. Turn the Master Switch – OFF.
41. Drain remaining fuel and measure weight of the fuel removed.
42. Verify remaining fuel measures 1.5 ± 0.2 Gals. by weight.
43. Record fuel remaining in Table 28-9 space provided.
44. Remove fuel drain valve assembly installed for calibration.
45. Install airplane fuel tank drain valve removed for calibration.
46. Remove test equipment installed for calibration procedure.

This completes the Calibration Test – Gallons Fuel Gauge procedure.

CALIBRATION TEST – LITERS FUEL GAUGE

Perform the following procedure to calibrate the fuel gauge for gallons.

1. Start with a dry tank.
2. Turn the Master Switch – ON.
3. On the airplane OAT read the current ambient temperature in degrees Celsius.
4. Turn the Master Switch – OFF.
5. Add 5.7 Liters (1.5 gallons) by weight.
6. Turn the Master Switch – ON.
7. Verify the fuel gauge reads zero ± 1.9 Liters. See Table 28-7.
8. As required, make small adjustments to the fuel sender "Empty" pot so the fuel gauge reads 0 Liters.
9. Turn the Master Switch – OFF.
10. Add 100 Liters of 100LL Blue or 100 Green fuel by weight.
11. Turn the Master Switch – ON.
12. Verify fuel gauge reads 100 ± 6.2 Liters See Table 28-7.
13. As required, make small adjustments to the fuel sender "Full" pot until the gauge reads 100 Liters. Figure 28-18 identifies the "FULL" pot location.
14. Turn the Master Switch – OFF.
15. Add 6 Liters of 100LL Blue or 100 Green fuel by weight.
16. Turn the Master Switch – ON.
17. Verify the fuel gauge reads 106 ± 6.4 Liters. See Table 28-7
18. Record gauge reading in Table 2 space provided.
19. Turn the Master Switch – OFF.
20. Prepare for tank draining by configuring the fuel can as shown in Figure 28-16.
21. Drain 6 Liters of fuel by weight.
22. Turn the Master Switch – ON.
23. Verify gauge reads 100 ± 6.2 Liters See Table 28-7.
24. Record gauge reading in Table 28-10 space provided.
25. Turn the Master Switch – OFF.
26. Drain 20 Liters of fuel by weight.
27. Turn the Master Switch – ON.
28. Verify gauge reads 80 ± 5.3 Liters See Table 28-7.

29. Record gauge reading in Table 28-10 space provided.
30. Turn the Master Switch – OFF.
31. Drain 20 Liters of fuel by weight.
32. Turn the Master Switch – ON.
33. Verify gauge reads 60 ± 4.5 Liters See Table 28-7.
34. Record gauge reading in Table 28-10 space provided.
35. Turn the Master Switch - OFF.
36. Drain 20 Liters of fuel by weight.
37. Turn the Master Switch – ON.
38. Verify gauge reads 40 ± 3.6 Liters See Table 28-7.
39. Record gauge reading in Table 28-10 space provided.
40. Turn the Master Switch – OFF.
41. Drain 20 Liters of fuel by weight.
42. Turn the Master Switch – ON.
43. Verify gauge reads 20 ± 2.8 Liters See Table 28-7.
44. Record gauge reading in Table 28-10 space provided.
45. Turn the Master Switch – OFF.
46. Drain 20 Liters of fuel by weight.
47. Turn the Master Switch – ON.
48. Verify gauge reads 0 ± 1.9 Liters See Table 28-7.
49. Record gauge reading in Table 28-10 space provided.
50. Turn the Master Switch – OFF.
51. Drain remaining fuel and measure weight of the fuel removed on completion.
52. Verify remaining fuel measures 5.7 Liters (1.5 gallons) See Table 28-7.
53. Remove fuel drain valve assembly installed for calibration
54. Install airplane fuel tank drain valve removed for calibration
55. Remove test equipment installed for calibration.

This completes the Calibration Test – Liters Fuel Gauge procedure.

Fuel Indication Check – Gallons

Airplane Serial Number: ______________

Airplane Registration: _______________

Fuel Indicator Serial Number: _________________

Test Date: _______________

Capacitive Fuel Sender Serial Number: ______________

Fuel Gauge Quantity	Fuel Gauge Needle Position	Fuel Quantity Error Tolerance	Pass/Fail Circle One
27 gallons		27 ±1.7 gallons Upper Limit = 28.7 Gals. Lower Limit = 25.3 Gals.	Pass / Fail
21 gallons		21 ±1.4 gallons Upper Limit = 22.4 Gals. Lower Limit = 19.6 Gals.	Pass / Fail
14 gallons		14 ±1.1gallons Upper Limit = 15.1 Gals. Lower Limit = 12.9 Gals.	Pass / Fail
7 gallons		7 ± 0.8 gallons Upper Limit = 7.8 Gals. Lower Limit = 6.2 Gals.	Pass / Fail
0 gallons		0 ± 0.5 gallons Upper Limit = 0.50 Gals. Lower Limit = -0.50 Gals.[1]	Pass / Fail
1.5 gallons[2]	Unusable fuel in tank[3]	1.5 ± 0.2 gallons Upper Limit = 1.7 Gals. Lower Limit = 1.3 Gals.	Pass / Fail

[1] Gauge indication can go below 0 in a properly operating gauge system. This is measuring into unusable fuel.
[2] Represents unusable fuel remaining in the tank.
[3] Enter unusable fuel drained from the tank as determined by weight of fuel removed compensated for temperature.

Table 28-9 Fuel Indication Check – Gallons

Fuel Indication Check – Liters

Airplane Serial Number: ___________

Airplane Registration: _______________

Fuel Indicator Serial Number: __________________

Test Date: _______________

Capacitive Fuel Sender Serial Number: ______________

Fuel Gauge Quantity	Fuel Gauge Needle Position	Fuel Quantity Error Tolerance	Pass/Fail Circle One
106 Liters		104 ± 6.4 Liters Upper Limit = 110.4 Liters Lower Limit = 97.7 Liters	Pass / Fail
100 Liters		100 ± 6.2 Liters Upper Limit = 106.2 Liters Lower Limit = 93.8 Liters	Pass / Fail
80 Liters		80 ± 5.3 Liters Upper Limit = 85.3 Liters Lower Limit = 74.7 Liters	Pass / Fail
60 Liters		60 ± 4.5 Liters Upper Limit = 64.5 Liters Lower Limit = 55.5 Liters	Pass / Fail
40 Liters		40 ± 3.6 Liters Upper Limit = 43.6 Liters Lower Limit = 36.4 Liters	Pass / Fail
20 Liters		20 ± 2.8 Liters Upper Limit = 22.8 Liters Lower Limit = 17.2 Liters	Pass / Fail
0 Liters		0 ± 1.9 Liters Upper Limit = 1.9 Liters Lower Limit = -1.9 Liters[1]	Pass / Fail
5.7 Liters[2]	Unusable fuel in tank[3]	5.7 ± 0.7 Liters Upper Limit = 6.4 Liters Lower Limit = 5.0 Liters	Pass / Fail

[1] Gauge indication can go below 0 in a properly operating gauge system. This is measuring into unusable fuel.
[2] Represents unusable fuel remaining in the tank.
[3] Enter unusable fuel drained from the tank as determined by weight of fuel removed compensated for temperature.

Table 28-10 Fuel Indication Check – Liters

Section 40-09 Adjustments

To adjust the fuel sender, refer to Figure 28-28, turn screw as required to increase or decrease distance between 28-gallon (106-Liters) fuel gauge mark and needle location. Note that this adjustment will have an equal effect to all other indication marks. Table 28-11 shows a summary of tolerances with respective plots shown in Figure 28-29 and Figure 28-30.

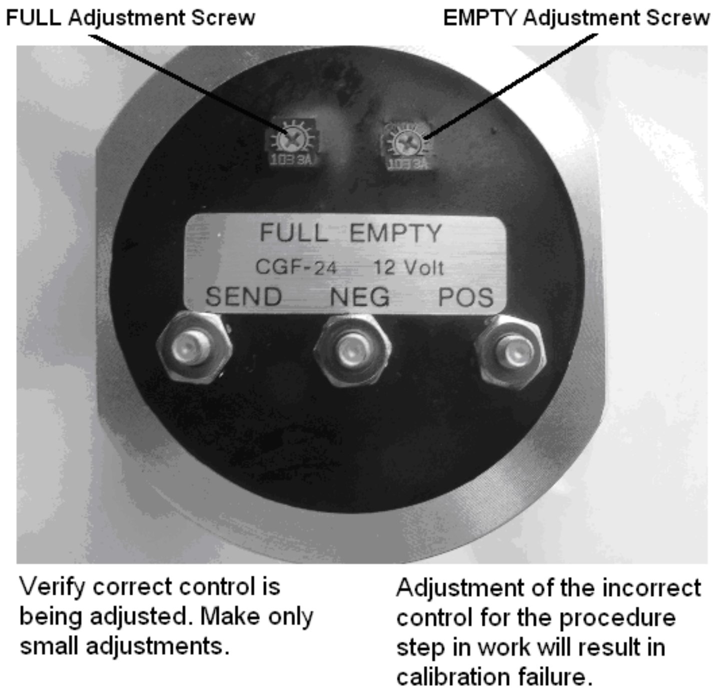

Figure 28-28 Capacitive Fuel Sender Adjustment

Fuel Quantity Indicated by Fuel Gauge (US GALLONS)	Allowable Fuel Gage Error Tolerance (US GALLONS)
1.5[1]	± 0.2
0	± 0.5
7	± 0.8
14	± 1.1
21	± 1.4
28	± 1.7

Fuel Quantity Indicated by Fuel Gauge (LITERS)	Allowable Fuel Gage Error Tolerance (LITERS)
5.7[2]	± 0.7
0	± 1.9
20	± 2.8
40	± 3.6
60	± 4.5
80	± 5.3
100	± 6.2
106	± 6.4

[1] Unusable fuel in tank, Gallons
[2] Unusable fuel in tank, Liters

Table 28-11 Allowable Fuel Quantity Error Tolerance

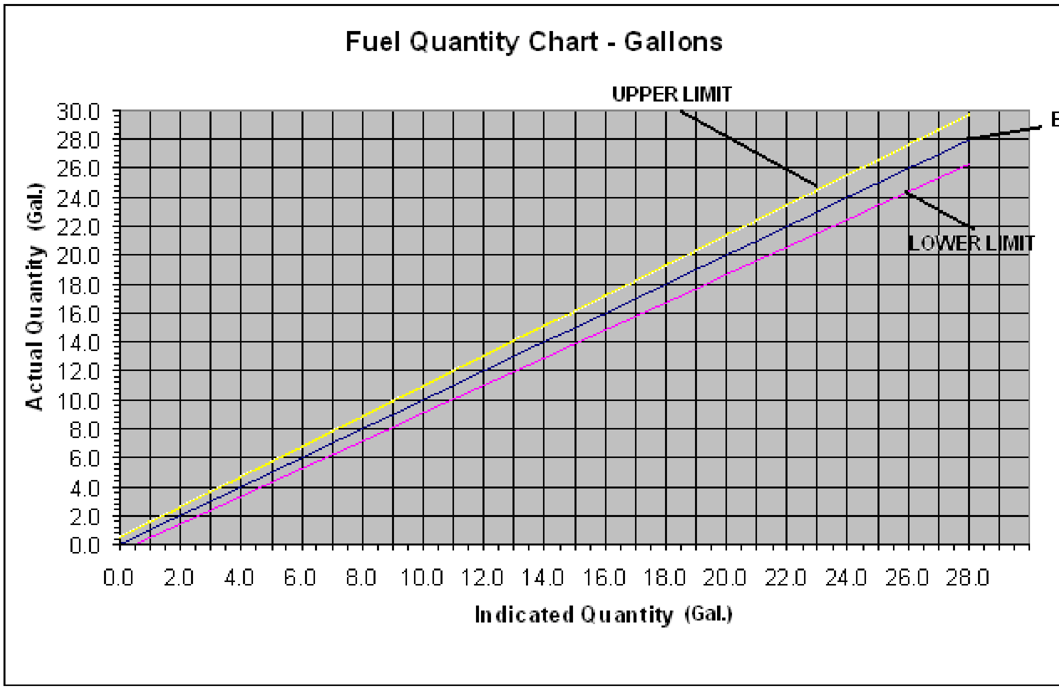

Figure 28-29 Fuel Quantity Tolerance Chart – Gallons

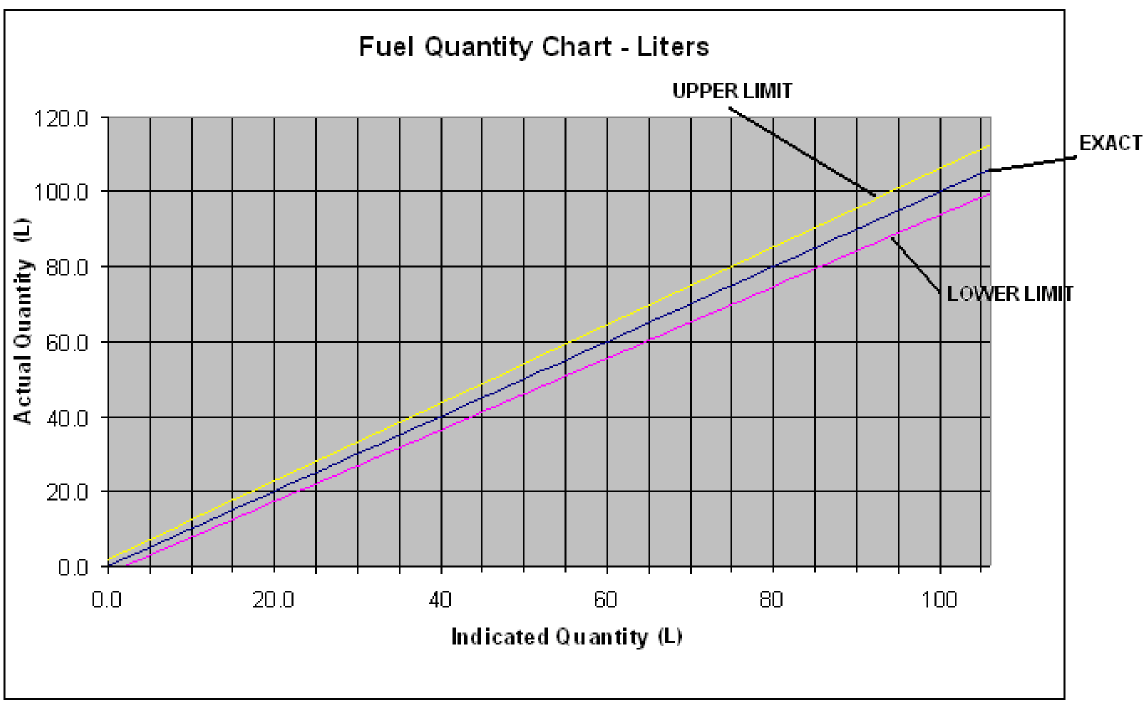

Figure 28-30 Fuel Quantity Tolerance Chart – Liters

Section 40-10 Troubleshooting

This section has information to aid in troubleshooting issues with the fuel system indiicators.

Complaint	Possible Cause	Remedy
Fuel level indicator inoperative or inaccurate	defective fuel level circuit breaker	replace
	defective fuel level indicator	replace
	defective fuel level sender	replace
	defective wiring	replace

Table 28-12 Fuel System Troubleshooting

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CHAPTER 30

ICE AND RAIN PROTECTION

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Table of Contents

SECTION 30-00 GENERAL 5

SECTION 30-30 PITOT-STATIC SYSTEM 7

SECTION 30-01 PITOT-STATIC PROCEDURES 7
PITOT-STATIC BLADE REMOVAL 8
PITOT-STATIC BLADE INSTALLATION 9
PITOT HEAT CURRENT SENSOR REMOVAL 11
PITOT HEAT CURRENT SENSOR INSTALLATION 13
PITOT HEAT RELAY REMOVAL 15
PITOT HEAT RELAY INSTALLATION 17
PITOT-STATIC PNEUMATIC INSPECTION AND CHECK OUT 19
PITOT-STATIC ELECTRICAL INSPECTION AND CHECK OUT 20
SECTION 30-02 TROUBLESHOOTING 21

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Section 30-00 General

This chapter describes systems, on the Liberty Aerospace, Inc. XL2 airplane that protects the airplane during rain and icing conditions. Although the airplane is not certified for known icing conditions, systems are provided to prevent ice formation in the wing mounted Pitot-Static Blade assembly.

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Section 30-30 Pitot-Static System

This section details the Pitot-Static system. This section contains the procedures to remove, install, and check the wing mounted pitot-static blade assembly, the heating circuit, and associated pitot-static tubing.

The pitot-static system consists of the heated pitot-static blade assembly, PN 135A-20-267, two pitot-static spacers, PN 135A-20-329 and 135A-20-331, the pitot-static heater circuit and the two pitot-static tubes that go from the blade to the instrument panel.

The pitot-static blade mounts to the port wing near the aileron. The pitot-static lines and heater circuit run up through the wing to the fuselage. At the point where the wing meets the fuselage, there are quick disconnects for both the pitot-static lines and the wire harness for the heater circuit.

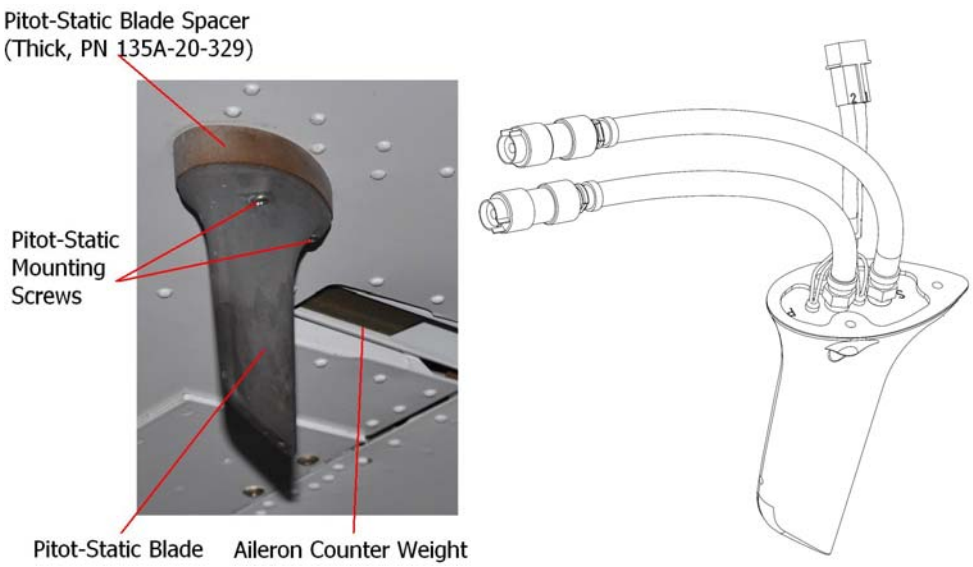

Figure 30-1 Pitot-Static Blade

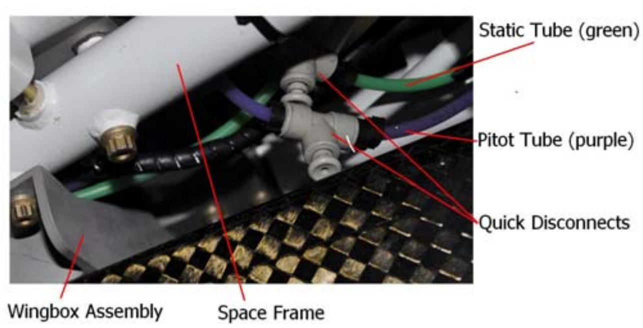

Figure 30-2 Pitot and Static Quick Disconnects

Section 30-01 *Pitot-Static Procedures*

This section details the procedures for the pitot-static system.

PITOT-STATIC BLADE REMOVAL

Perform this procedure to remove the Pitot-Static Blade.

Figure 30-3

1. Pull the circuit breaker for the pitot heat.
2. Remove the three screws securing access cover closest to the pitot-static blade.
3. Using a 3/16” deep-socket to hold the nut on the inside, remove the three screws securing the pitot-static blade to the wing.
4. Lower the blade from the wing.
5. Carefully feed the pitot, static, and heater wiring down through the hole in the wing.
6. Disconnect the pitot-static lines and the heater wiring.
7. Remove the two spacers
8. If not installing a replacement blade immediately, cover the hole in the wing and install the access cover.

This completes the Pitot-Static Blade Removal procedure.

PITOT-STATIC BLADE INSTALLATION

Perform this procedure to install the Pitot-Static Blade.

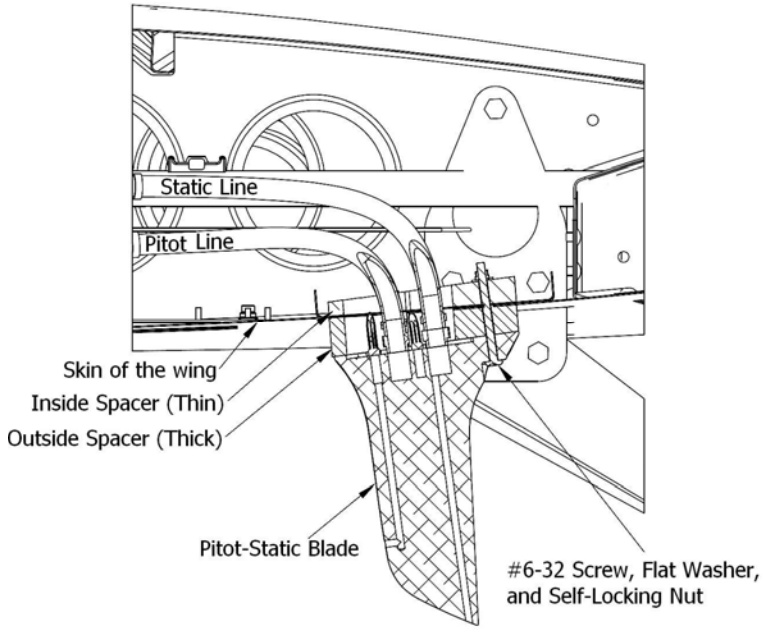

Figure 30-4 Pitot-Static Blade Installation Detail

1. Pull the circuit breaker for the pitot heat.
2. Remove the three screws securing access cover closest to the pitot-static blade.
3. Insert the inside spacer in to the wing, aligning it with the cutout for the pitot-static blade. See Figure 30-4 for details on installing the inside washer.

NOTE

The thinner portion of the spacer points forward.

4. Assemble the outside spacer (thick) to the pitot-static blade as shown in Figure 30-4.

NOTE

The thicker portion of the spacer points forward.

5. Feed the pitot-static tubing through the inside space and out of the wing. Feed the cable for the pitot heat through the space and out of the wing.

6. Connect the purple tube to the tube coming from the forward connector marked “P”.
7. Connect the green tube to the tube coming from the aft connector marked “S”.
8. Connect the electrical cable to the electric cable coming from the pitot-static blade.
9. Insert one of the #6-32 screws removed in step 3 in the Pitot-Static Blade Removal procedure on page 8 of this chapter.
10. Loosely thread the screw into the self-locking nut.
11. Insert the other two screws and loosely thread the screws in to the self-locking nut.
12. Snug each to draw the two spacers together evenly.
13. Torque the screws as defined in Chapter 20 – *Standard Practices*.
14. Perform the Pitot-Static Purge procedure in Chapter 34 – *Navigation and Pitot-Static*.
15. Perform the Pitot-Static Pneumatic Inspection and Check Out on page 19 of this chapter.

This completes the Pitot-Static Blade Installation procedure.

PITOT HEAT CURRENT SENSOR REMOVAL

Perform this procedure to remove the Pitot Heat Current sensor.

CAUTION

This device contains electro-statically sensitive parts. Take ESD precautions before handling this or any other electronic device on the airplane.

1. Ensure all electrical switches are off.
2. Pull circuit breaker BAT 1 (CB001) to OPEN.
3. Place a cover over the pilot yoke control.
4. Have a large block of soft foam on your lap to place the instrument panel on.
5. Remove the ten screws securing the instrument panel to the instrument console. See for Figure 30-5 location of the screws.

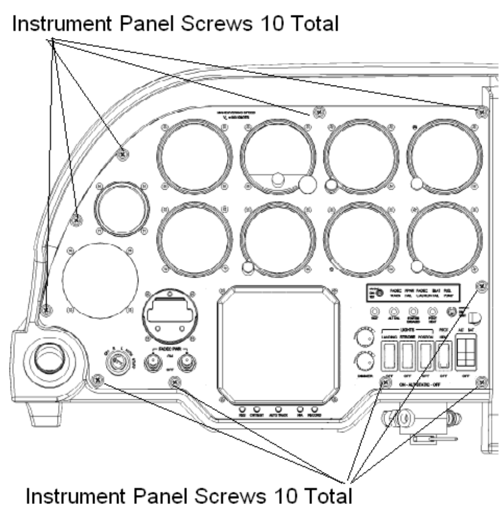

Figure 30-5 Location of the Screws Securing the Instrument Panel

6. Gently pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
7. Do not to disturb electrical connections, or pitot-static lines.
8. Connect an anti-static strap from you body to the metallic surface of the power distribution harness.
9. Disconnect wire H03A14 from terminal 87 of relay 006. See Figure 30-6.
10. Feed the wire back through the current sensor. Secure the wire or connect it back to terminal 87 of relay K006.

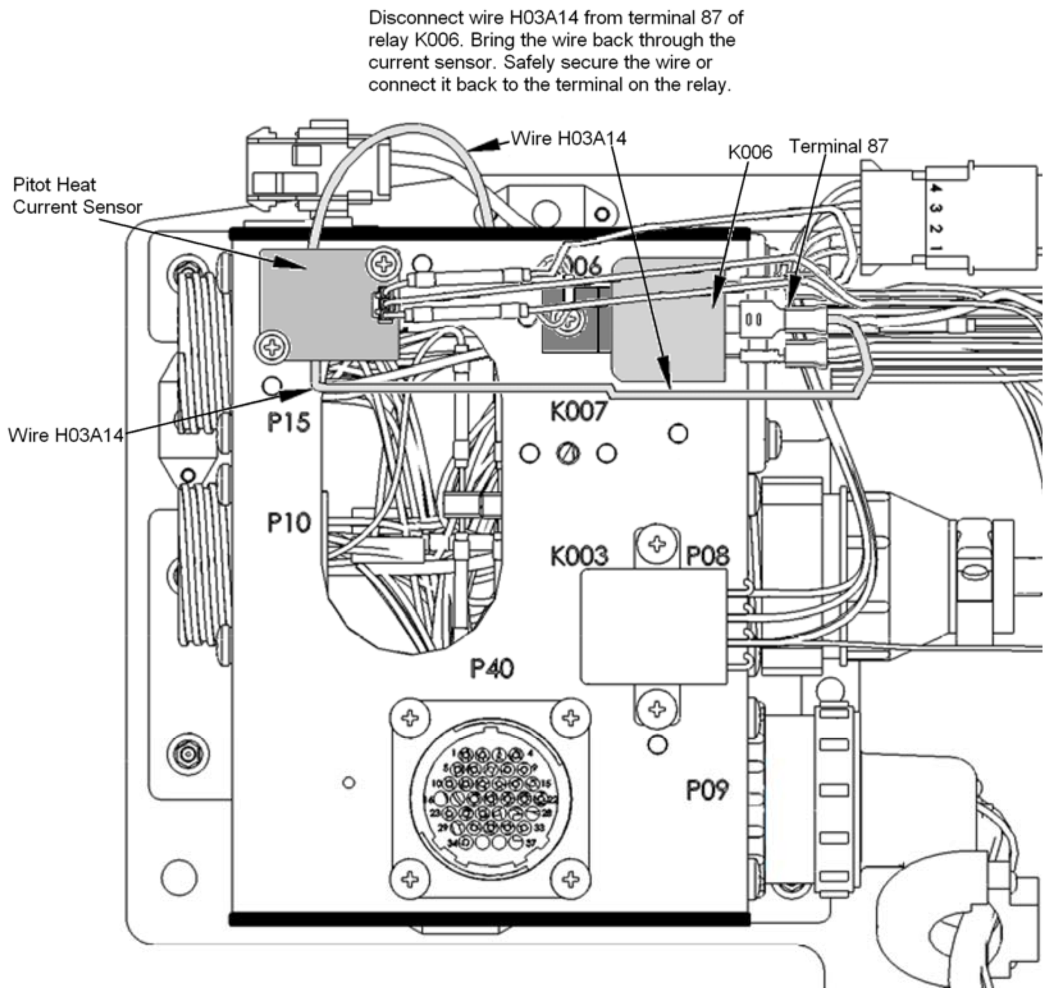

Figure 30-6 Location of the Pitot Heat Current Sensor Showing Wire H03A14

11. Remove the two screws securing the pitot heat current sensor to the bracket.

12. Disconnect the three-wire cable leading to the current sensor, and secure.

This completes the Pitot Heat Current Sensor Removal procedure.

Pitot Heat Current Sensor Installation

Perform this procedure to install the Pitot Heat Current sensor.

CAUTION

This device contains electro-statically sensitive parts. Take ESD precautions before handling this or any other electronic device on the airplane.

1. Ensure all electrical switches are off.
2. If installing a replacement the pitot heat current sensor immediately after removing an existing pitot heat current sensor, then proceed to step 7 below.
3. Place a cover over the pilot yoke control.
4. Have a large block of soft foam on your lap to place the instrument panel on.
5. Remove the ten screws securing the instrument panel to the instrument console. See Figure 30-5 for location of the screws.
6. Gently pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
7. Connect an anti-static strap from you body to the metallic surface of the power distribution harness.
8. Attach the replacement pitot heat current sensor to the bracket using two 4-40 X 1.0 pan head Philips corrosion resistant screws.
9. Connect the current sensor cable to the current sensor. The connector has an orientation key for correct positioning.
10. Route wire H03A14 through the current sensor from top to bottom.
11. Connect wire H03A14 to terminal 87 on relay 006. See Figure 30-7 for the details of the location of terminal 87 on relay K006.

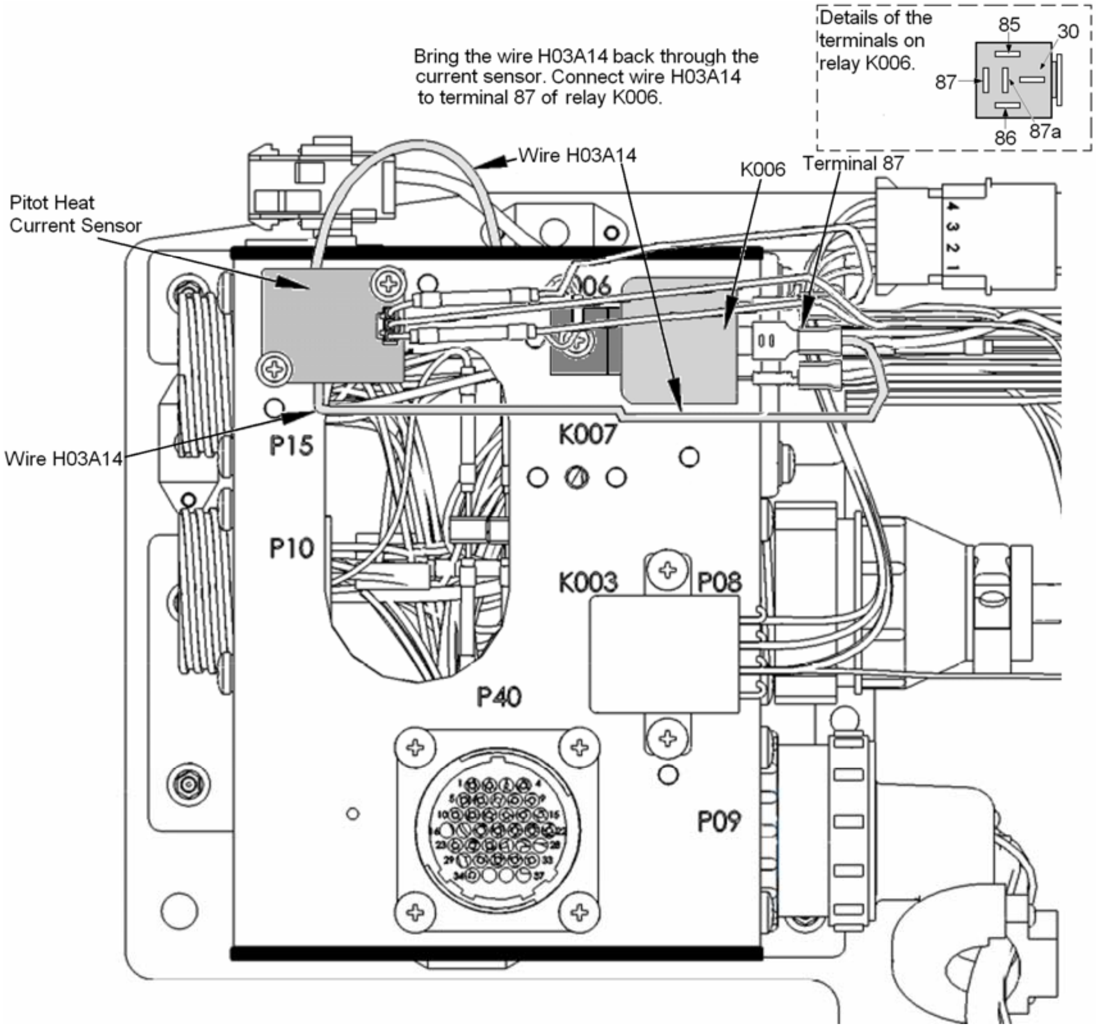

Figure 30-7 Location of the Pitot Heat Current Sensor with Details of Relay K006

12. Secure any wiring with wire ties as needed.
13. Install the instrument panel into the Instrument console.
14. Secure the instrument panel with the ten screws removed earlier.
15. Perform the Pitot-Static Electrical Inspection and Check Out on page 20 of this chapter.

This completes the Pitot Heat Current Sensor Installation procedure.

PITOT HEAT RELAY REMOVAL

Perform this procedure to remove the Pitot Heat Relay.

CAUTION

This device contains electro-statically sensitive parts. Take ESD precautions before handling this or any other electronic device on the airplane.

1. Ensure all electrical switches are off.
2. Pull circuit breaker BAT 1 (CB001) to OPEN.
3. Place a cover over the pilot yoke control.
4. Have a large block of soft foam on your lap to place the instrument panel on.
5. Remove the ten screws securing the instrument panel to the instrument console. See for Figure 30-5 location of the screws.

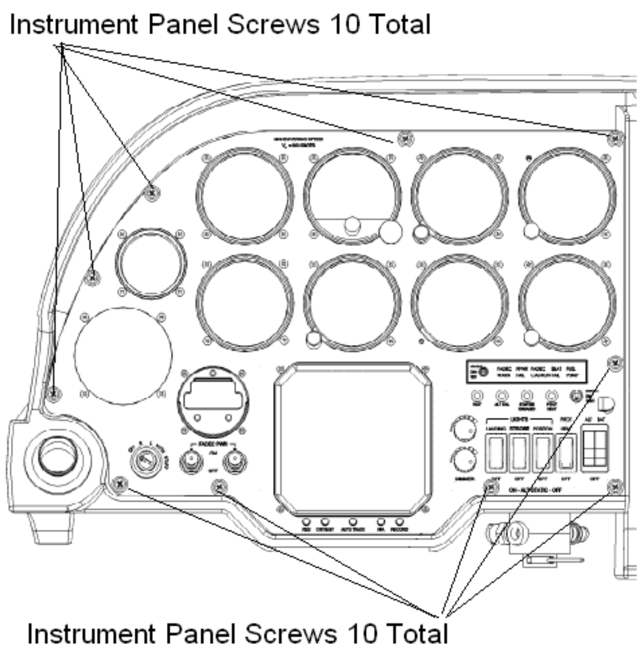

Figure 30-8 Location of the Screws Securing the Instrument Panel

6. Gently pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
7. Do not to disturb electrical connections, pitot, or static lines.

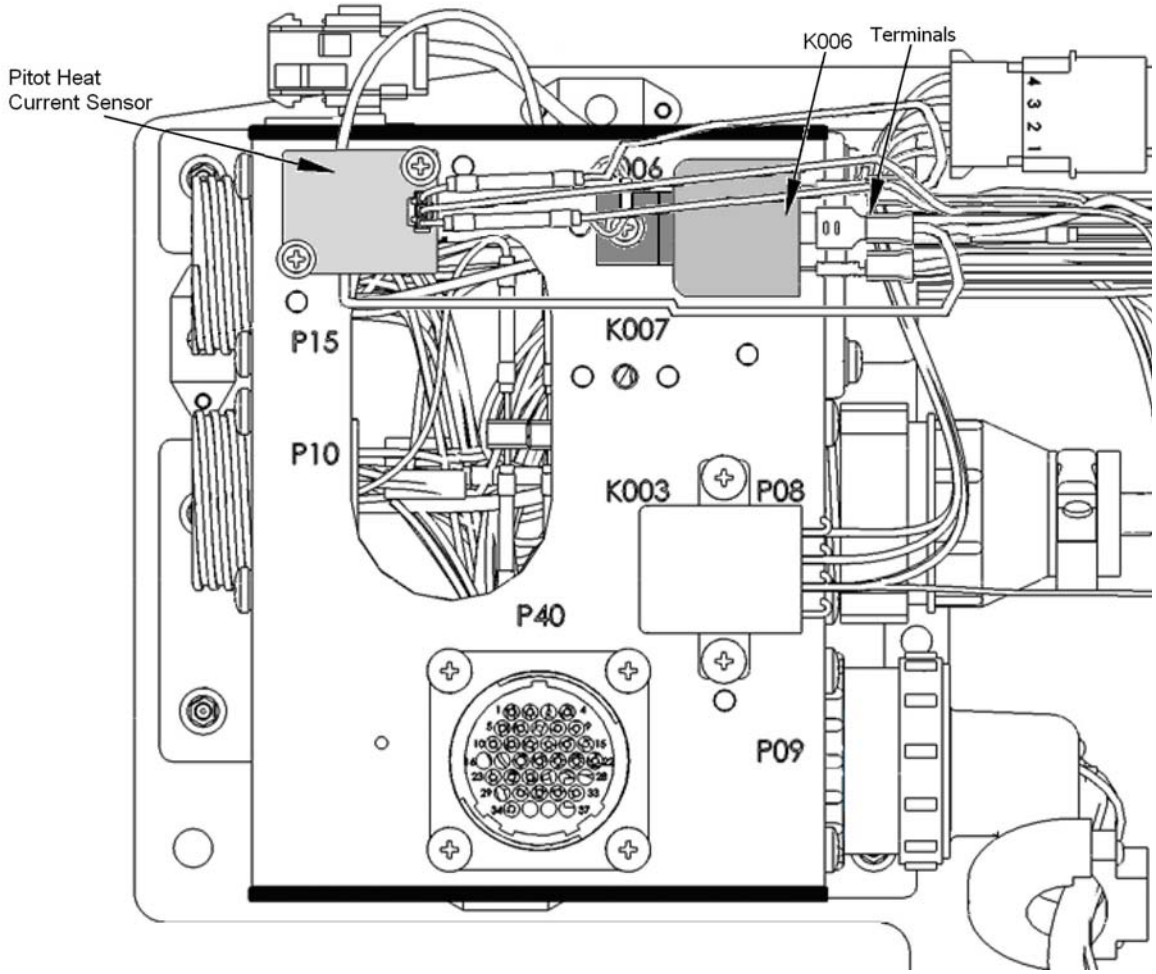

Figure 30-9 Location of the Pitot Heat Relay K006

8. Connect an anti-static strap from you body to the metallic surface of the power distribution harness.
9. Disconnect the wires connected to relay 006. See Figure 30-9.
10. Remove the screw securing the relay to the power distribution panel.

This completes the Pitot Heat Relay Removal procedure.

PITOT HEAT RELAY INSTALLATION

Perform this procedure to install the Pitot Heat Relay.

CAUTION

This device contains electro-statically sensitive parts. Take ESD precautions before handling this or any other electronic device on the airplane.

1. Ensure all electrical switches are off.
2. If installing a replacement the pitot heat relay immediately after removing an existing pitot heat relay, then proceed to step 7 below.
3. Place a cover over the pilot yoke control.
4. Have a large block of soft foam on your lap to place the instrument panel on.
5. Remove the ten screws securing the instrument panel to the instrument console. See Figure 30-5 for location of the screws.
6. Gently pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
7. Connect an anti-static strap from you body to the metallic surface of the power distribution harness.
8. Attach the replacement pitot heat relay to the bracket using one 4-40 X 1.0 pan head Philips corrosion resistant screws.
9. Connect the current sensor cable to the current sensor. The connector has an orientation key for correct positioning.
10. Connect wires to terminals on relay 006. See Figure 30-10 for the details of the location of terminals on relay K006.

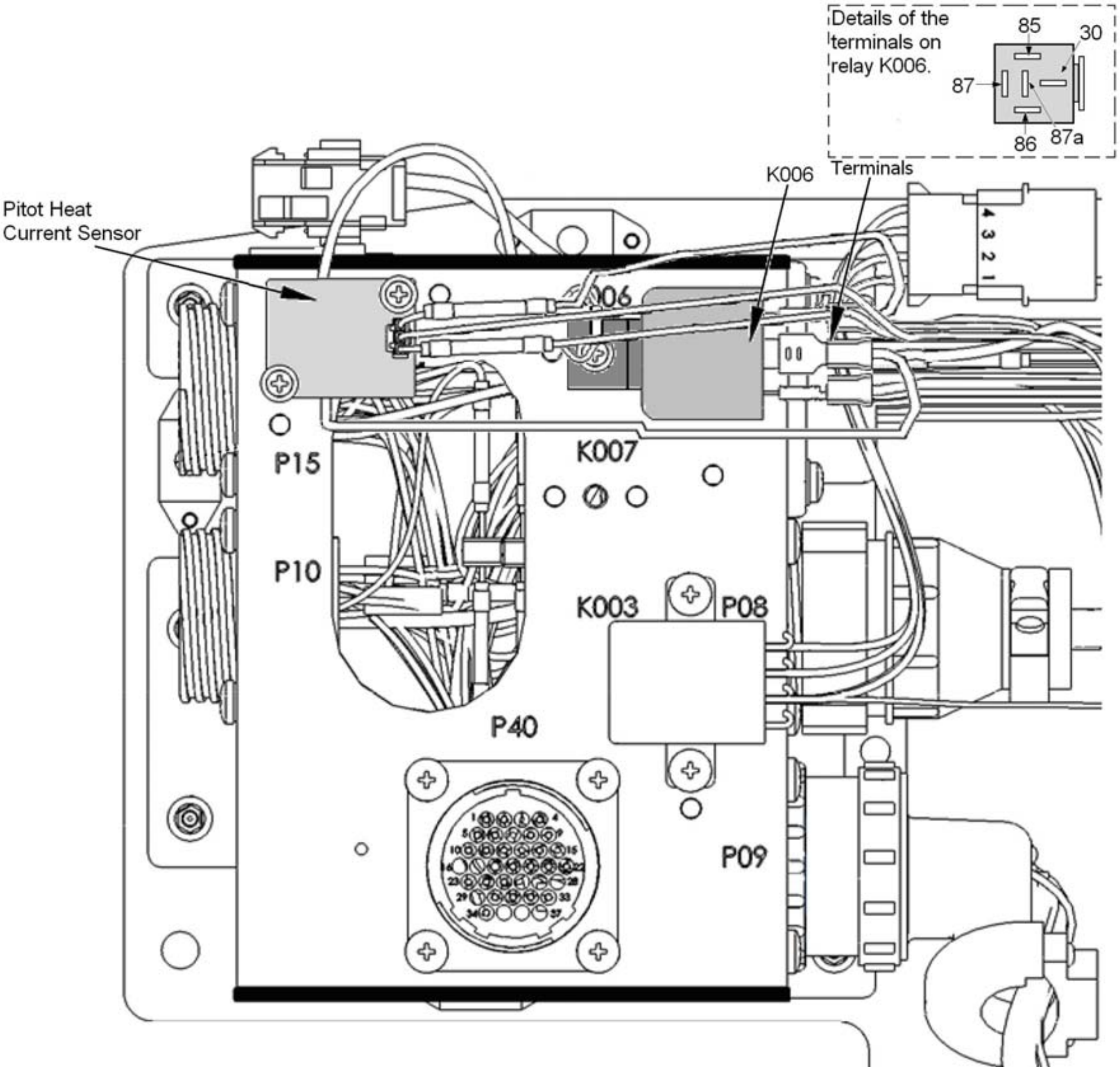

Figure 30-10 Location of the Pitot Heat Relay K006 Terminals

11. Secure any wiring with wire ties as needed.
12. Install the instrument panel into the Instrument console.
13. Secure the instrument panel with the ten screws removed earlier.
14. Perform the Pitot-Static Electrical Inspection and Check Out on page 20 of this chapter.

This completes the Pitot Heat Relay Installation procedure.

PITOT-STATIC PNEUMATIC INSPECTION AND CHECK OUT

Perform this procedure to inspect and check the Pitot-Static Blade.

1. Check all hardware for proper torque. For the correct torque values, see Chapter 20 – *Standard Practices*.
2. Check the two inlets on the blade are clear and free of foreign material.
3. Perform the Static System Operational Check in Chapter 34 – *Navigation and Pitot-Static*.
4. Perform the Pitot System Operational Check in Chapter 34 – *Navigation and Pitot-Static*.
5. Perform the Pitot-Static Electrical Inspection and Check Out on page 20 of this chapter.

This completes the Pitot-Static Pneumatic Inspection and Check Out procedure.

PITOT-STATIC ELECTRICAL INSPECTION AND CHECK OUT

Perform this procedure to inspect and check the pitot-static electrical system.

NOTE

If the pitot-static tubing has been changed or altered, perform the Pitot-Static Pneumatic Inspection and Check Out on page 19 of this chapter before performing this procedure.

1. Place the Master switch ON.
2. Place the Pitot Heat switch OFF. The light in the Pitot Heat switch must be ON. The panel annunciator for Pitot Heat must be OFF.
3. Place the Pitot Heat switch ON. The light in the Pitot Heat switch must be OFF. The panel annunciator for Pitot Heat must be OFF.
4. After a few seconds (< 10 seconds), pull the System - Heat 1 circuit breaker. The switch-mounted indicator must be OFF, and the instrument panel annunciator must be ON.

CAUTION

The Pitot-Static blade can get very hot and could burn. When checking the blade heater, use care when touching the blade.

5. Carefully touch the blade. The blade should feel very warm.
6. Reset the System – Heat 1 circuit breaker. The instrument panel annunciator should go OFF.
7. Place the Pitot Heat switch OFF. Place the Master switch OFF.

This completes the Pitot-Static Electrical Inspection and Check Out procedure.

Section 30-02 *Troubleshooting*

For troubleshooting information, see Chapter 34 – *Navigation and Pitot-Static*.

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CHAPTER 31

INDICATORS AND RECORDING SYSTEMS

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Table of Contents

SECTION 31-00 GENERAL 5
SECTION 31-10 INSTRUMENT PANEL 7
SECTION 10-01 INSTRUMENT PANEL INDICATORS 7
SECTION 10-02 INSTRUMENT PANEL PROCEDURES 7
INSTRUMENT PANEL REMOVAL 8
INSTRUMENT PANEL INSTALLATION 11
GENERAL INDICATOR REMOVAL 13
GENERAL INDICATOR INSTALLATION 14
SECTION 10-03 FUNCTIONAL CHECKS 15
SECTION 31-20 INDEPENDENT INDICATORS 17
SECTION 20-01 FLAP POSITION SWITCH 17
SECTION 20-02 FLAP POSITION INDICATORS 17
SECTION 20-03 TRIM POSITION SWITCH 17
SECTION 20-04 TRIM POSITION INDICATORS 17
SECTION 20-05 WIDE OPEN THROTTLE (WOT) INDICATOR 18
SECTION 20-06 ALTERNATOR FAIL (ALT FAIL) INDICATORS 18
SECTION 20-07 STARTER ENGAGED INDICATORS 18
SECTION 20-08 PITOT HEAT INDICATOR 18
SECTION 20-09 HEALTH STATUS ANNUNCIATOR INDICATOR 18
SECTION 20-10 INDEPENDENT INDICATOR PROCEDURES 18
HEALTH STATUS INDICATOR REMOVAL 19
HEALTH STATUS INDICATOR INSTALLATION 20
INSTRUMENT PANEL LED INDICATOR REMOVAL 21
INSTRUMENT PANEL LED INDICATOR INSTALLATION 22
FLAP POSITION SWITCH REPLACEMENT 23
FLAP POSITION LED INDICATOR REMOVAL 26
FLAP POSITION LED INDICATOR INSTALLATION 28
TRIM POSITION SWITCH REPLACEMENT 30
TRIM POSITION LED INDICATOR REMOVAL 32
TRIM POSITION LED INDICATOR INSTALLATION 34
SECTION 31-30 RECORDERS 37
SECTION 30-01 VM-1000FX ENGINE INSTRUMENT DISPLAY 37
SECTION 30-02 ENGINE DATA INTERFACE 38
SECTION 30-03 ENGINE DATA INTERFACE PROCEDURES 38
ENGINE DATA INTERFACE REMOVAL 39
ENGINE DATA INTERFACE INSTALLATION 41
ENGINE DATA INTERFACE CHECKOUT 42
ENGINE DATA INTERFACE INITIALIZATION AND TEST 43
SECTION 31-50 STALL WARNING 45
SECTION 50-01 STALL WARNING PROCEDURES 45
STALL WARNING REMOVAL 46
STALL WARNING INSTALLATION 48
STALL WARNING SENSOR REMOVAL 49
STALL WARNING SENSOR INSTALLATION 50
STALL WARNING SYSTEM CHECK 51
SECTION 50-02 STALL WARNING SYSTEM TROUBLESHOOTING 52
SECTION 31-60 CENTRAL DISPLAY SYSTEMS 53

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Section 31-00 General

With the exception of the trim position indicator, installed on the upper surface of the cockpit center console, all instruments and displays in the airplane are installed on the instrument console.

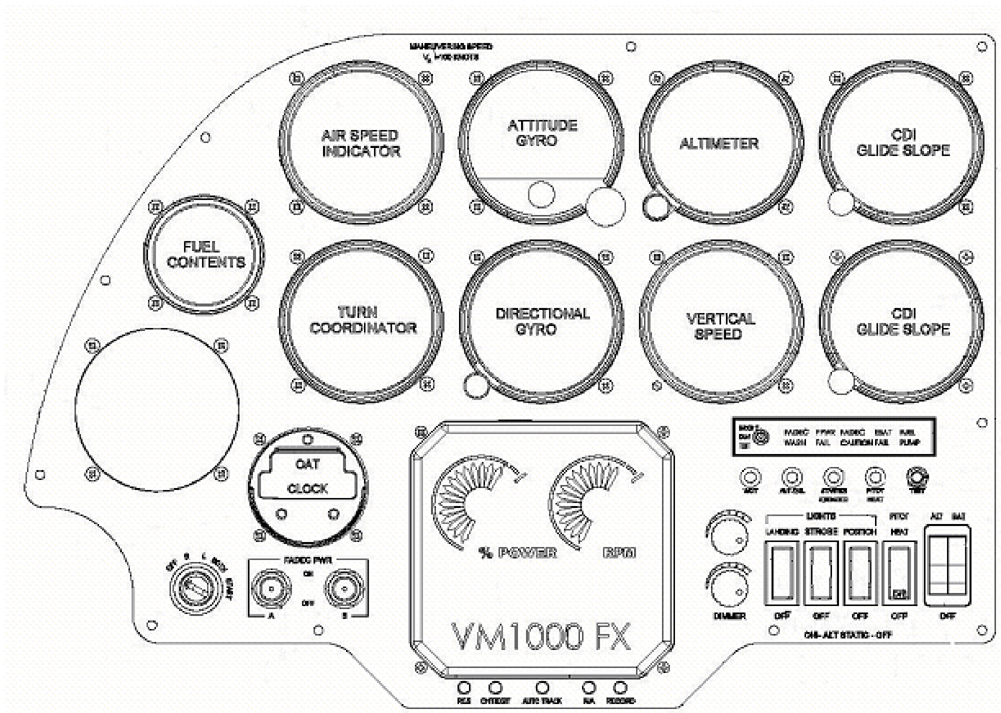

Figure 31-1 Left Side Instrument Panel

Flight and navigation instruments, conforming to a standard layout, are part of the instrument panel mounted on the left side of the instrument console. Information pertaining to the instrument panel is described in this chapter

All avionics control panels (and optional avionics displays) are installed in a vertical stack in the center of the instrument console. Information pertaining to the avionics panel is in Chapters 23 – *Communications* and Chapter 34 – *Navigation and Pitot/Static*.

Electrical system circuit breakers are part of the CB Panel on the right side of the instrument console. Information pertaining to the CB panel and the electrical circuit breakers is in Chapter 24 - *Electrical Power*.

Individual instruments are discussed in detail in the applicable chapters for each instrument or related subsystem.

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Section 31-10 Instrument Panel

The primary navigation instruments are in two horizontal rows in the upper portion of the instrument panel. The smaller fuel contents gauge and electric clock are to the left of the flight instruments.

A single integrated electronic display provides all engine instrumentation functions and is installed at the lower center of the left instrument panel. The engine ignition switch and FADEC A & B switches are installed to the left of the engine instrumentation display. Aircraft electrical and subsystem switches are installed to the right of the engine instrumentation display.

Section 10-01 Instrument Panel Indicators

The XL-2 airplane comes equipped with the following indicators mounted to the instrument panel:

- Air Speed Indicator
- Attitude Gyro
- Altimeter (barometric)
- Turn Coordinator
- Directional Gyro
- Vertical Speed
- Fuel Contents
- Clock/Outside Air Temperature
- CDI Glide Slope (optional) X2

Specific information on each of these indicators, except for the Fuel Contents, is in *Chapter 34 – Navigation*. The Fuel Contents indicator is described in *Chapter 28 – Fuel*.

Section 10-02 Instrument Panel Procedures

This section contains the procedures to remove and install the instrument panel. Also included in this section is a general procedure to remove and install individual instruments from the instrument console. Specific removal or installation procedure pertaining to a specific instrument, display, or gauge will be in the *Chapter 34 - Navigation*. Some procedures may require the splicing of wires. If a splice is needed refer to Chapter 20 – *Standard Practices Airframe* for details on making splices.

INSTRUMENT PANEL REMOVAL

Perform this procedure to remove the instrument panel from the airplane. Liberty Aerospace, Inc. recommends performing this procedure while sitting in the pilot's seat. Have the block of soft foam cushioning in your lap that it covers the yoke assembly.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Pull the BAT1 (CB001) circuit breaker to OPEN.
3. Remove the ten screws that secure the Flight Instrument Panel assembly to the console. Refer to Figure 31-2 for the location of these screws.

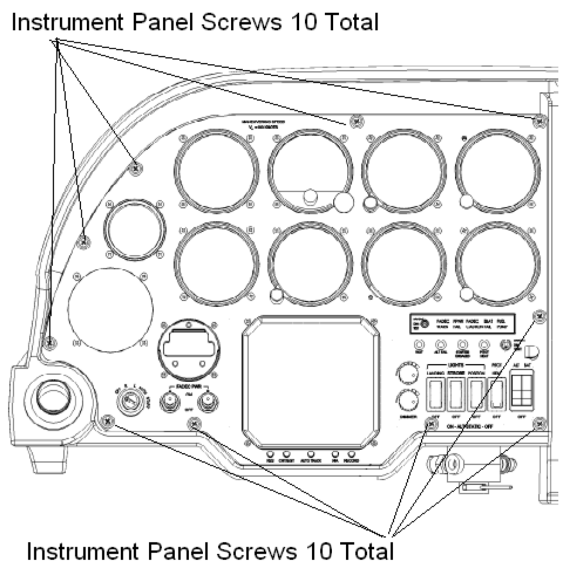

Figure 31-2 Instrument Panel Assembly Showing the Location of the Screws

4. Bring this panel towards you and place it on its face in to the block of soft foam cushioning that is in your lap.
5. If the instrument panel has one or two Course Deviation Indicators (CDI), disconnect the electrical connectors on the back of the CDI. (P/J106-1 and/or P/J106-2) See Figure 31-3 for the location of P/J106-1 and/or P/J106-2. Disconnect the ground wire from the CDI cable(s) from the instrument panel grounding point.

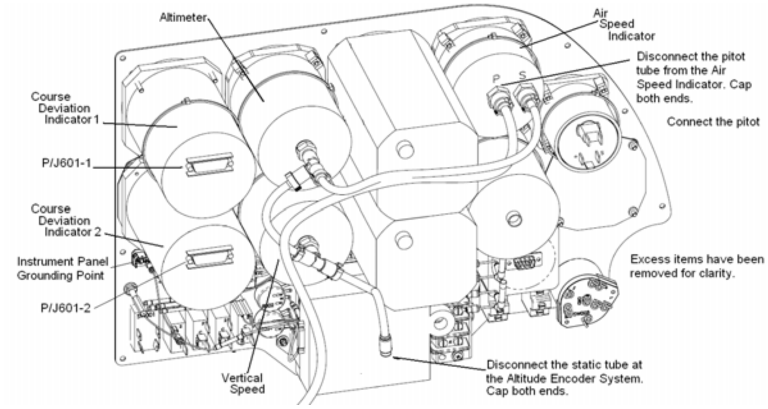

Figure 31-3 Rear View of the Instrument Panel

6. Disconnect the pitot line (violet tubing) from the Airspeed indicator. Disconnect the static line (green tubing) at the altitude encoder system. Cap all lines and/or indicators. For the location of the pitot and static lines, and where to disconnect these lines, see Figure 31-3 and Figure 31-4. Disconnect the single wire connector at P/J69. See Figure 31-6 for the location of P/J69.

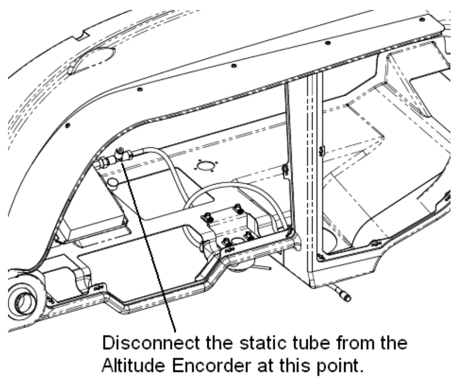

Figure 31-4 Location of the Altitude Encoder System

7. Disconnect the ribbon cable from the VM1000FX display. See Figure 31-5 for the location of the connector for the VM1000FX.

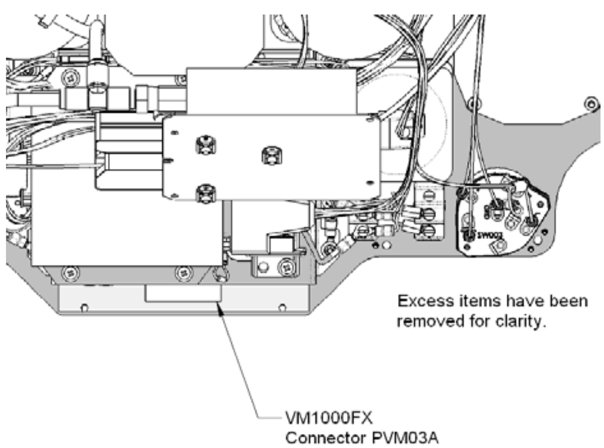

Figure 31-5 Location of the Connector on the VM1000FX

8. Disconnect P03, P10, and P15 from their mating connectors on the power distribution bracket. See Figure 31-6 for location of P/J03, P/J10, and P/J15.

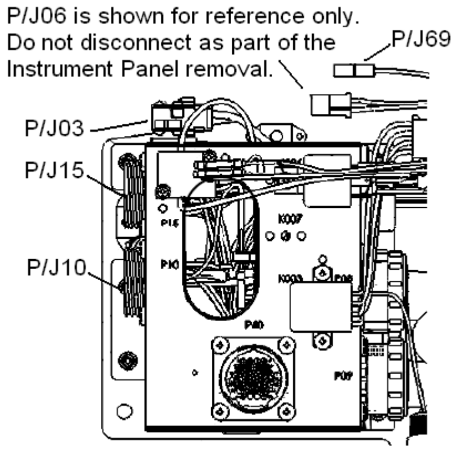

Figure 31-6 Left Hand Side of the Power Distribution Harness Assembly

9. Remove the instrument panel from the airplane keeping the instrument panel on its face into a block of soft foam cushioning.

This completes the Instrument Panel Removal procedure.

INSTRUMENT PANEL INSTALLATION

Perform this procedure to install the instrument panel from the airplane. Liberty Aerospace, Inc. recommends performing this procedure while sitting in the pilot's seat. Have the block of soft foam cushioning in your lap that it covers the yoke assembly.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Pull the BAT1 (CB001) circuit breaker to OPEN.
3. Retrieve the Instrument panel removed in step 9 of the procedure to Instrument Panel Removal on page 8 of this chapter.
4. Connect P03, P10, and P15 to their mating connectors on the power distribution bracket. See Figure 31-7 for location of P/J03, P/J10, and P/J15.

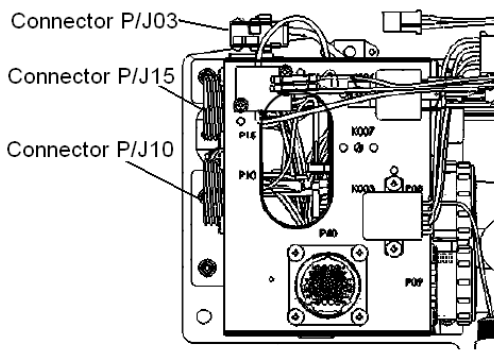

Figure 31-7 Left Hand Side of the Power Distribution Harness Assembly

5. Connect the ribbon cable to the VM1000FX display. See Figure 31-8 for the location of the connector on the VM1000FX.

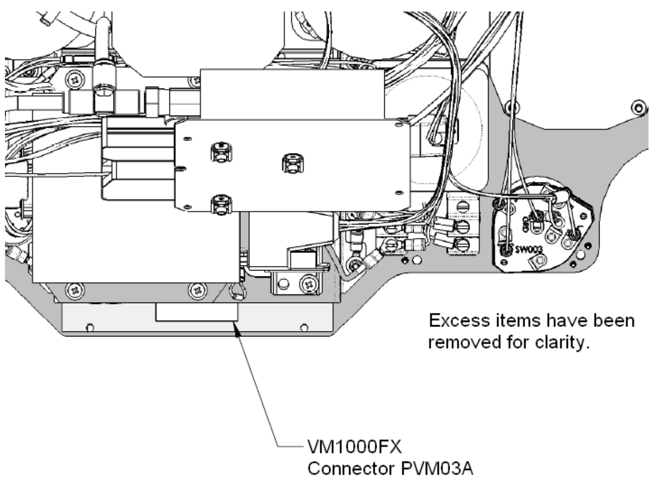

Figure 31-8 Location of the Connector on the VM1000FX

6. If the instrument panel has one or two Course Deviation Indicators (CDI), connect the electrical connectors on the back of the CDI. (P/J106-1 and/or P/J106-2) See Figure 31-9 for the location of the connectors P/J106-1 and/or P/J106-2. Connect the ground shields from the CDI cables to the grounding point on the instrument panel.
7. Remove the caps from the pitot and static lines and the Airspeed indicator. Connect the pitot line to the Airspeed indicator. Connect the static line to the Altitude Encoder system.

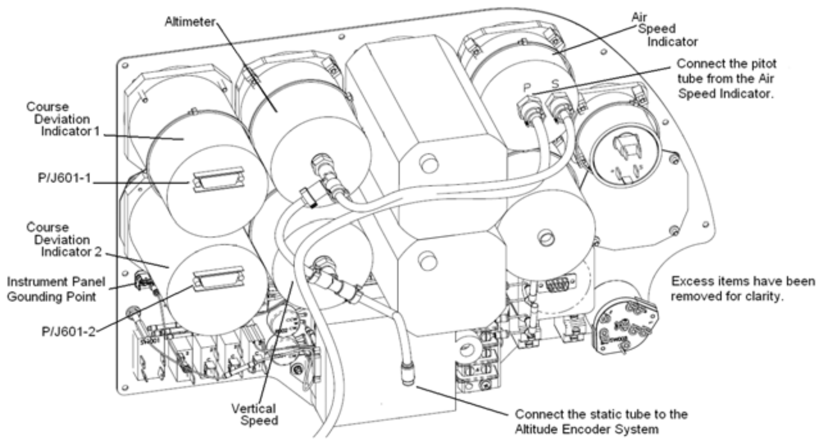

Figure 31-9 Rear of Instrument Panel Showing the Locations for Connecting the Pitot and Static Lines

8. Install the panel into the opening in the instrument panel console assembly.
9. Install the ten screws to secure the instrument panel to the console. See Figure 31-10 for the location of the ten screws.

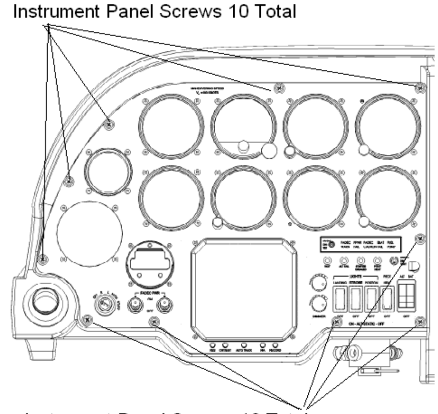

Figure 31-10 Location of the Ten Screws Securing The Instrument Panel

This completes the Instrument Panel Installation procedure.

GENERAL INDICATOR REMOVAL

Perform this procedure to remove any of the indicators, gauges, or displays from the instrument panel. If it is necessary to remove any of the switches or controls from the instrument panel, Liberty Aerospace, Inc. recommends removing the instrument panel from the airplane and working on it at a bench.

1. Perform steps 1 through 4 of the Instrument Panel Removal procedure on page 8 of this chapter.
2. Remove individual gauges by unplugging and capping any electrical connectors. If removing Airspeed indicator, Altitude, or Vertical speed indicator, unplugging and capping any pitot/static lines.
3. Lift panel so that instrument-retaining screws are accessible.
4. Loosened and removed the screws that secure the indicator or gauge to the instrument panel.
5. Remove individual indicator or gauge from back side of panel.

NOTE

The indicators and gauges mounted to the instrument panel have a cutout (bezel) light assembly. This is a separate component from the instrument or gauge.

This completes the General Indicator Removal procedure.

GENERAL INDICATOR INSTALLATION

Perform this procedure to install any of the indicators or gauges into the instrument panel.

1. Ensure all electrical switches are off.
2. If installing a replacement indicator or gauge immediately after removing an previous indicator or gauge, then proceed to step 8 below.
3. Place a cover over the pilot yoke control.
4. Have a large block of soft foam on your lap to place the instrument panel on.
5. Remove the ten screws securing the instrument panel to the instrument console. See Figure 31-2 for location of the screws.
6. Gently pull the instrument panel towards you, placing it face down on to the block of soft rubber.
7. Prepare the cutout (bezel) light assembly to allow for the installation of the indicator or gauge.
8. Install indicator or gauge, and cutout (bezel) light assembly to the instrument panel and secure them to panel with the instrument screws.
9. Remove any caps or covers from the indicator or gauge.
10. Connect any cabling or tubing to the indicator or gauge, ensuring correct connection.
11. Install instrument panel into the instrument console.
12. Secure the instrument panel with the ten instrument panel screws.
13. Perform the Function Checks shown in Section 10-03 Functional Checks on page 15 of this chapter.

This completes the General Indicator Installation procedure.

Section 10-03 Functional Checks

Perform a functional check to Functional Check table. Perform a functional check on all lights and dimming features.

FUNCTIONAL CHECKS		
Indicator	**Test**	**Reference Chapter**
Air speed indicator	Leak test	Chapter 34 – *Navigation and Pitot/Static*
Turn Coordinator	Power on / Gyro audible	Chapter 34 – *Navigation and Pitot/Static*
Attitude Gyro	Power on / Gyro audible	Chapter 34 – *Navigation and Pitot/Static*
Directional Gyro	Power on / Gyro audible	Chapter 34 – *Navigation and Pitot/Static*
Altimeter	Leak test	Chapter 34 – *Navigation and Pitot/Static*
Vertical speed indicator	Leak test	Chapter 34 – *Navigation and Pitot/Static*
CDI Glide scope	AV power on / Check for needle movement during CDI BIT (BIT occurs within 5 sec.)	Chapter 34 – *Navigation and Pitot/Static*
Fuel contents gauge	Power on / Check indication	Chapter 28 – *Fuel System*
OAT/Clock	Power on / Check indication	Chapter 34 – *Navigation and Pitot/Static*
VM1000FX	Engine ground run / Perform a functional check	Chapter 77 – *Engine Indicating*
HSA	Functional check per TCM OI-22 Ch. 6	Chapter 77 – *Engine Indicating*

Table 31-1 Functional Checks for the Instrument Panel

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Section 31-20 Independent Indicators

This section describes other indicators that are independent of other systems in the airplane or are not described in other chapters in the maintenance manual.

Section 20-01 Flap Position Switch

The flap position switch is co-located below the flap position indicators on the avionics panel. The switch is a three-position Momentary ON/OFF/ Momentary ON paddle switch. The switch has internal lighting to indicate the position of the switch. When the switch is in the flaps up position, the green light within the switch for Flaps Up is illuminated. When the switch is in the flaps down position, the green light within the switch for Flaps Down is illuminated. When the switch is in the neutral (OFF) position, both green lights within the switch are not illuminated.

Section 20-02 Flap Position Indicators

Flap position indicators are on the avionics panel right hand side. Three Light Emitting Diodes (LEDs) activated by position switches within the flap deployment mechanism are utilized to report fully retracted, fully deployed and an intermediate flap positions. A placard surrounding the indicators identifies flap angle of deflection from fully retracted in degrees. Airplanes certified for a gross weight of 1653 lbs have a placard that indicates flap deflections of 0° +1°/-0°, 20° ±1°, and 30° ±1°. Airplanes certified for a gross weight of 1750 lbs have a placard that indicates flap deflections of 0° +1°/-0°, 10° ±1°, and 30° ±1°. A green coded LED indicates zero degrees of flap, while the indication of the remaining degree settings of flap has amber coded LEDs. If the flaps are at any angle between these points, none of the indicators are illuminated. For more information on the flap indicator system, is in *Chapter 27 – Flight Controls*.

Section 20-03 Trim Position Switch

The trim position switch is located on the lower center console panel. The switch is a three-position Momentary ON/OFF/ Momentary ON rocker switch. The switch has internal lighting to indicate the position of the switch. When the switch is in the trim up position, the green light within the switch for Trim Up is illuminated. When the switch is in the trim down position, the green light within the switch for Trim Down is illuminated. When the switch is in the neutral (OFF) position, both green lights within the switch are not illuminated.

Section 20-04 Trim Position Indicators

Trim position indicators are on the center console directly below the avionics panel. Three Light Emitting Diodes (LEDs) activated by position switches within the trim tab deflection mechanism are utilized to report trim tab full up (13° ± 0.5°), full down (5.0° ± 0.5°) and a midpoint take-off position (0° ± 0.5°). A placard surrounding the indicators identifies the tab nose up, nose down and takeoff indicators. A green coded LED indicates takeoff position, while the indication of Nose Up and Nose Down positions has amber coded LEDs. For more information on the trim indicator, system is in *Chapter 27 – Flight Controls*.

Section 20-05 Wide Open Throttle (WOT) Indicator

The Wide Open Throttle, WOT, indicator is part of the flight instrument panel indicator row above and left of the airplane split master switch. When the throttle control is in the fully forward, wide-open position, an engine mounted switch closes and illuminates this green-coded LED.

Section 20-06 Alternator Fail (ALT FAIL) Indicators

The Alternator Fail, ALT FAIL, indicator is part of the flight instrument panel indicator row above and left of the airplane split master switch. The amber coded LED illuminates when the alternator system has failed, and is no longer providing power to airplane buses or battery charging systems.

Section 20-07 Starter Engaged Indicators

The Starter Engaged, indicator is part of the flight instrument panel indicator row above and left of the airplane split master switch. The amber coded LED illuminates whenever the starter contactor (relay) is closed and the starter solenoid is engaged.

Section 20-08 Pitot Heat Indicator

The Pitot Heat, indicator is part of the flight instrument panel indicator row above and left of the airplane split master switch. The amber coded LED illuminates whenever there is power applied to pitot heaters but no current is flowing through the pitot heater indicating a failed pitot heat system.

Another pitot heat indicator is part of the flight instrument panel within the pitot heat power switch located left of the airplane split master switch. The amber coded LED illuminates whenever the pitot power switch is in the OFF position.

Section 20-09 Health Status Annunciator Indicator

The Health Status Annunciator (HSA) is an indicator module mounted above the flight instrument panel indicator row. The HSA indicator module contains five LED annunciations providing FADEC health and status information as follows:

- FADEC WARN - Illuminates red upon a potential total failure of the FADEC system
- FADEC CAUTION - Illuminates amber when there is a fault in the FADEC system
- PPWR FAIL - Illuminates red upon failure of the airplane primary power system
- EBAT FAIL – Illuminates amber in the event of a secondary battery failure or in the event primary power fails and FADEC is running exclusively from the secondary battery.
- FUEL PUMP - Illuminates green when FADEC calls for boost pump operation.

Section 20-10 Independent Indicator Procedures

This section has the removal and installation procedures for the Health Status annunciator, the various LED indicators, illuminated switches.

HEALTH STATUS INDICATOR REMOVAL

Perform this procedure to remove the health status indicator. Liberty Aerospace, Inc. recommends removing the instrument panel from the airplane and working on it at a bench.

1. Perform the Instrument Panel Removal procedure on page 8 of this chapter.
2. Carefully cut the cable ties around the bundle that includes the cable for the health status indicator.
3. Separate the D-Sub connector that runs to the health status indicator.
4. Loosened and removed the two 3/16 inch nuts that secure the indicator to the instrument panel.
5. Remove the indicator from backside of panel.

This completes the Health Status Indicator Removal procedure.

HEALTH STATUS INDICATOR INSTALLATION

Perform this procedure to install the health status indicator. Liberty Aerospace, Inc. recommends removing the instrument panel from the airplane to do this procedure.

1. If installing a replacement indicator immediately after removing a previous indicator, then proceed to step 3 below.
2. Perform the Instrument Panel Removal procedure on page 8 of this chapter.
3. Install the health status indicator using the two 3/16 inch nuts removed in step 4 of the Health Status Indicator Removal procedure on page 19 of this chapter.
4. Connect the D-Sub connector from the indicator to the mating connector from the health status annunciator.
5. Secure the wire bundle with the necessary cable ties.
6. Install the instrument panel per the Instrument Panel Installation procedure on page 11 of this chapter.
7. Move the battery switch to the ON position.
8. Move either of the FADEC Power Switches to ON.
9. Move the Bright/Dim/Test switch on the health status indicator to the test position. Verify that all indicators on the health status indicator and the WOT indicator are illuminated.
10. Move the Bright/Dim/Test switch on the instrument panel to the test position. Verify that ALT FAIL, STARTER ENGAGED, and PITOT HEAT indicators are illuminated and the WOT indicator is not illuminated.
11. Move FADEC Power Switches to OFF.
12. Move battery switch to the OFF position.

This completes the Health Status Indicator Installation procedure.

INSTRUMENT PANEL LED INDICATOR REMOVAL

Perform this procedure to remove any of the LED indicators on the instrument panel. Liberty Aerospace, Inc. recommends removing the instrument panel from the airplane and working on it at a bench.

1. Perform the Instrument Panel Removal procedure on page 8 of this chapter.
2. Carefully cut the cable ties around the bundle that includes the cable for the LED Indicator.
3. Mark for identification and cut the wires ahead of the splice.
4. Use a 10mm open-end wrench to loosen and remove the nut.
5. Slide the indicator out through the front of the instrument panel.

This completes the Instrument Panel LED Indicator Removal procedure.

INSTRUMENT PANEL LED INDICATOR INSTALLATION

Perform this procedure to install any of the LED indicators mounted to the instrument panel.

NOTE

If replacing the WOT LED indicator, the LED must be GREEN. If replacing the ALT FAIL, STARTER ENGAGED, or PITOT HEAT LED indicator, the LED must be YELLOW.

1. If installing a replacement indicator immediately after removing a previous indicator, then proceed to step 3 below.
2. Perform the Instrument Panel Removal procedure on page 8 of this chapter.
3. Install the LED indicator through from the front of the instrument panel.
4. Secure the LED indicator with the nut removed in step 4 of the Instrument Panel LED Indicator Removal procedure on page 21 of this chapter.
5. Connect the wires by splicing them to the wires that were cut in step 3 in the Instrument Panel LED Indicator Removal procedure on page 21 of this chapter. See Chapter 20 – *Standard Practices* for wire splicing procedures and requirements.
6. Secure the wire bundle using wire ties.
7. Install the instrument panel per the Instrument Panel Installation procedure on page 11 of this chapter
8. Move the battery switch to the ON position.
9. Move either FADEC Power Switch to ON.
10. Move the Bright/Dim/Test switch on the health status indicator to the test position. Verify that all indicators on the health status indicator and the WOT indicator are illuminated.
11. Move the Bright/Dim/Test switch on the instrument panel to the test position. Verify that ALT FAIL, STARTER ENGAGED, and PITOT HEAT indicators are illuminated and the WOT indicator is not illuminated.
12. Move FADEC Power Switches to OFF.
13. Move battery switch to the OFF position.

This completes the Instrument Panel LED Indicator Installation procedure.

FLAP POSITION SWITCH REPLACEMENT

Perform this procedure to replace a defective flap position switch.

1. Remove the avionics radios from the avionics panel. To remove the radios, insert a hex key (“3/32” Allen wrench”) into the hole in faceplate of the radio and rotate counter-clockwise until the retaining claw is released. Slide the radio towards you and out of the radio tray.

CAUTION

Removal of the avionics radios reduces the weight of the avionics panel. This will make the panel easier to handle and less likely to damage the Instrument Panel Console assembly surfaces when removing the panel from the console.

2. Place the radio outside the airplane.
3. Remove the eight screws holding in the avionics panel assy. Refer to Figure 31-11 for the location of the eight screws.

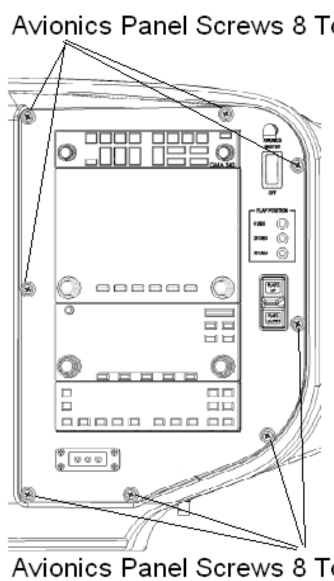


Figure 31-11 Avionics Panel Showing the Location of the Screws

4. Carefully pull the avionics panel out of the instrument console.

NOTE

The avionics panel will not come complete out of the instrument console. However, the panel will come out sufficiently to remove the flap position switch.

5. Tag and remove the six wires from the back of the switch by gently pull on the quick connect terminals. Refer to Figure 31-12 for wire numbers for wires connecting to the flap switch.

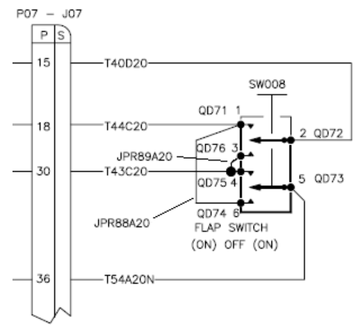


Figure 31-12 Partial Schematic of Flap Switch Circuit

6. Press in on the spring tabs on the top and bottom of the switch. See Figure 31-13 for details of the spring tabs.

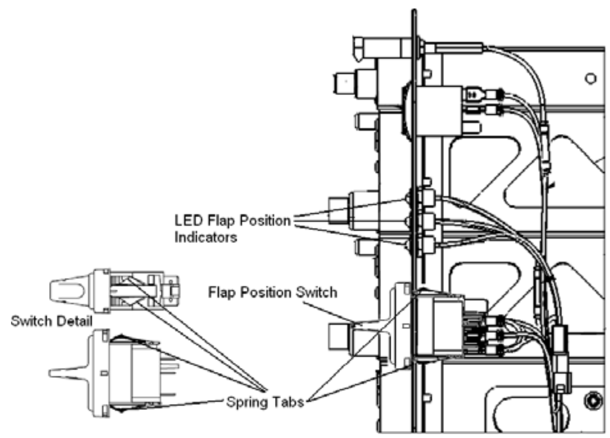


Figure 31-13 Flap Position Switch Showing the Spring Tabs

7. While pressing in on the spring tabs, rock the switch out from the front of the avionics panel.
8. Carefully insert the switch into the panel. Continue pushing the switch until it seats securely in the avionic panel.
9. Connect the wire removed in step 5 above. See Chapter 20 – *Standard Practices* for wire splicing procedures and requirements.
10. Position the master battery switch to the ON position.
11. Move the flap switch to Flaps Down position. Verify the correct switch indicator illuminates, the flap actuator motor runs and the flaps are moving to the down position.
12. As the flaps are moving from the fully up position (0°) to the fully down position (30°), verify the flap position indicators illuminate to correspond to the flap position.
13. When the flaps are fully down, release the switch. Verify the switch moves back to the neutral position, the switch illumination is out, and the flap actuator motor stops running.

14. Move the flap switch to Flaps Up position. Verify the correct switch indicator illuminates, the flap actuator motor runs and the flaps are moving to the up position.
15. As the flaps are moving from the fully down position (30°) to the fully up position (0°), verify the flap position indicators illuminate to correspond to the flap position.
16. When the flaps are fully up, release the switch. Verify the switch moves back to the neutral position, the switch illumination is out, and the flap actuator motor stops running.
17. Position the master battery switch to the OFF position.
18. Insert the avionics panel back in to the instrument console.
19. Secure the avionic panel with the screws removed in step 3 above.
20. Insert the avionics radios in to the avionics panel.

This completes the Flap Position Switch Replacement procedure.

FLAP POSITION LED INDICATOR REMOVAL

Perform this procedure to remove the LED indicators associated with the Flap Indicator.

1. Remove the avionics radios from the avionics panel. To remove the radios, insert a hex key ("3/32" Allen wrench") into the hole in faceplate of the radio and rotate counter-clockwise until the retaining claw is released. Slide the radio towards you and out of the radio tray.

CAUTION

Removal of the avionics radios reduces the weight of the avionics panel. This will make the panel easier to handle and less likely to damage the Instrument Panel Console assembly surfaces when removing the panel from the console.

2. Place the radio outside the airplane.
3. Remove the eight screws holding in the avionics panel assembly. Refer to Figure 31-14 for the location of the eight screws.

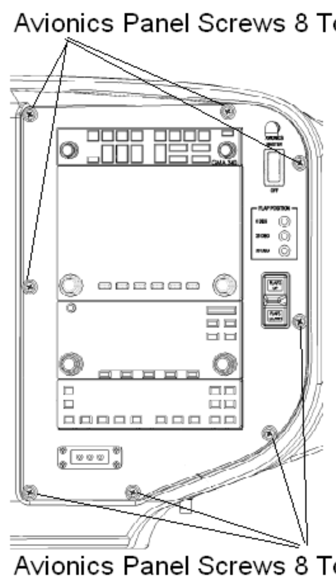


Figure 31-14 Avionics Panel Showing the Location of the Screws

4. Carefully pull the avionics panel out of the instrument console.

NOTE

The avionics panel will not come complete out of the instrument console. However, the panel will come out sufficiently to remove the flap position LEDs.

5. Carefully cut the cable ties around the bundle that includes the cable for the LED Indicator.

6. Mark for identification and cut the black wires ahead of splice SP019. Refer to Figure 31-15 for the schematic location of the splice.

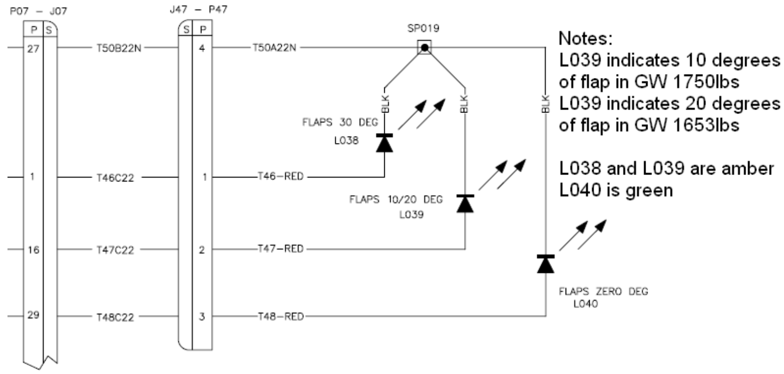


Figure 31-15 Partial Schematic of the Flap Indicator Circuit

7. Separate P/J47.
8. Remove the appropriate connector pin from P47. Refer to Figure 31-15 for which pin in P47 to remove for which LED indicator.
9. Cut the connector pin from the red wire.
10. Use a 10mm open-end wrench to loosen and remove the nut.
11. Slide the indicator out through the front of the avionics panel.
12. If a replacement indicator is not available, install the avionics panel; otherwise proceed to the Flap Position LED Indicator Installation procedure.

FLAP POSITION LED INDICATOR INSTALLATION

Perform this procedure to install any of the LED indicators mounted to the instrument panel.

1. If installing a replacement indicator immediately after removing a previous indicator, then proceed to step 3 below.
2. Perform step 1 through step 4 of the Flap Position LED Indicator Removal procedure on page 26 of this chapter.
3. Install the LED indicator through from the front of the avionics panel.
4. Secure the LED indicator with the nut removed in step 4 of the Instrument Panel LED Indicator Removal procedure on page 21 of this chapter.
5. Attach and crimp a connector pin to the red wire. See Chapter 20 – *Standard Practices* for wire splicing procedures and requirements.
6. Insert the connector pin into the correct location on P47. Refer to Figure 31-16 for the correct pin location.

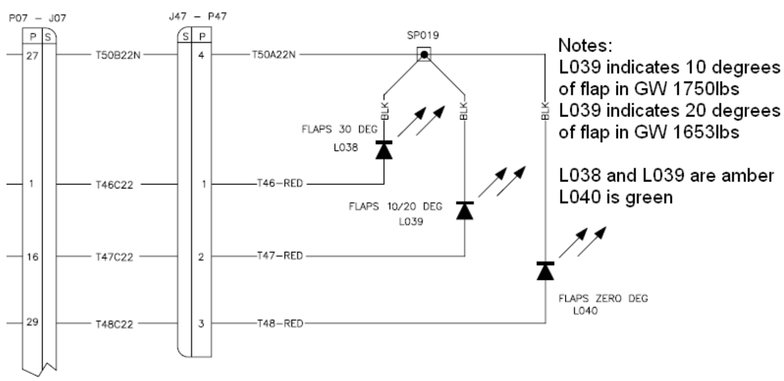


Figure 31-16 Partial Schematic of the Flap Indicator Circuit

7. Strip and crimp a new splice SP019 to wire T50A22N.
8. Strip the three black wires going to the LED indicators. Insert and crimp the three black wires in to the other side of the splice SP019. See Chapter 20 – *Standard Practices* for wire splicing procedures and requirements.
9. Secure the wire bundle using wire ties.
10. Install the avionics panel in to the instrument console. Secure the panel with the screws removed in step 3 of the Flap Position LED Indicator Removal procedure on page 26 of this chapter.
11. Install the avionics radios in to the avionics panel.
12. Move the battery master switch to the ON position.
13. Verify the flaps are fully retracted. If the flaps are not fully retracted, move the flap switch to bring the flaps up to the fully retracted position.
14. If the flaps are fully retracted, verify the Flaps 0 Degrees green LED indicator is illuminated.

15. On the instrument console, move the Bright/Dim/Test switch to the Bright position. Verify the flaps 0 Degrees LED indicator is illuminated and bright.
16. On the instrument console, move the Bright/Dim/Test switch to the Dim position. Verify the flaps 0 Degrees LED indicator is illuminated and dimmer than the Bright setting.
17. On the instrument console, move the Bright/Dim/Test switch to the Test position. Verify the flaps 0 Degrees LED indicator is illuminated and is brighter than Dim and dimmer that Bright.
18. With the Bright/Dim/Test switch in the Test position, verify the other flap indicators are not illuminated.
19. Using the flaps switch, move the flaps to until the Flaps 10 (GW = 1750lbs.) or 20 (GW = 1653lbs.) Degrees amber LED indicator is illuminated.
20. On the instrument console, move the Bright/Dim/Test switch to the Bright position. Verify the LED indicator is illuminated and bright.
21. On the instrument console, move the Bright/Dim/Test switch to the Dim position. Verify the LED indicator is illuminated and dimmer than the Bright setting.
22. On the instrument console, move the Bright/Dim/Test switch to the Test position. Verify the LED indicator is illuminated and is brighter than Dim and dimmer that Bright.
23. With the Bright/Dim/Test switch in the Test position, verify the other flap indicators are not illuminated.
24. Using the flaps switch, move the flaps to until the Flaps are fully extended (30 degrees); verify the flaps 30 Degrees amber LED indicator is illuminated.
25. On the instrument console, move the Bright/Dim/Test switch to the Bright position. Verify the 30 Degrees LED indicator is illuminated and bright.
26. On the instrument console, move the Bright/Dim/Test switch to the Dim position. Verify the 30 Degrees LED indicator is illuminated and dimmer than the Bright setting.
27. On the instrument console, move the Bright/Dim/Test switch to the Test position. Verify the 30 Degrees LED indicator is illuminated and is brighter than Dim and dimmer that Bright.
28. With the Bright/Dim/Test switch in the Test position, verify the other flap indicators are not illuminated.
29. Using the flaps switch, return the flaps to the fully retracted position.
30. Move the battery master switch to the OFF position.

This completes the Flap Position LED Indicator Installation procedure.

TRIM POSITION SWITCH REPLACEMENT

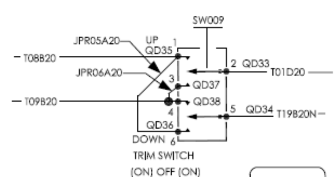
Perform this procedure to remove the flap position switch.

NOTE

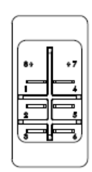
This procedure is best done with two people. One person positioned under the airplane to start the switch removal and feed the wires to the switch up through the hole in the center console. The other one is in the cockpit to pull the switch up and out of the console.

The rocker switch guard is not part of the switch. Do not attempt to remove the switch by removing the switch guard.

1. Remove the belly panel from the airplane. Refer to *Chapter 53 - Fuselage*, Section 53-40 for the procedure to remove the belly panel.
2. From underneath the airplane, reach up through the space frame and press in on the spring tabs on either side of the switch.
3. While pressing in on the spring tabs, gently push up on the switch to bring it through the center console.
4. After the switch is free of the hole in the fuselage, carefully feed the wiring to the switch up through the hole.
5. Tag and remove each of the wires going to the trim switch. Refer to Figure 31-17 for the wire numbers and location of the terminals on the switch.



SW009CONNECTION CHART	TERMINATION
1	QD35
2	QD33
3	QD37
4	QD38
5	QD34
6	QD36



SWITCH AS VIEWED FROM BOTTOM

Figure 31-17 Partial Schematic of the Trim Switch Circuit

6. Connect the wires to the replacement switch. Refer to Figure 31-17 when connecting the wires to the terminals on the switch. If wire splices are required, see Chapter 20 – *Standard Practices* for wire splicing procedures and requirements.
7. Position the master battery switch to the ON position.
8. Press and hold the trim switch to Nose Down position. Verify the correct switch indicator illuminates, the trim actuator motor runs and the flaps are moving to the down position.

9. As the trim tabs are moving from the neutral position to the nose down position, verify the trim tab position indicators illuminate to correspond to the trim tab position.
10. When the trim tabs reach the nose down position, release the switch. Verify the switch moves back to the neutral position, the switch illumination is out, and the trim tab actuator motor stops running.
11. Press and hold the trim switch to Nose Up position. Verify the correct switch indicator illuminates, the trim tab actuator motor runs and the trim tabs are moving to the nose up position.
12. As the flaps are moving from the nose down position to the nose up position, verify the trim position indicators illuminate to correspond to the trim tab position.
13. When the flaps reach the fully nose up position, release the switch. Verify the switch moves back to the neutral position, the switch illumination is out, and the flap actuator motor stops running.
14. Press and hold the trim switch to move the trim tabs back to the neutral position.
15. Position the master battery switch to the OFF position.
16. Carefully feed the wires down through the hole in the console.
17. Carefully insert the trim switch in to the hole in the console. Push on the center top of the switch until the switch is fully seated in to the center console.
18. From underneath the airplane, verify the wiring from the switch is not interfering with any of the controls or actuators in the space frame.
19. Install the belly panel. Refer to Chapter 53 - *Fuselage* for the procedure to install the belly panel.

This completes the Trim Position Switch Replacement procedure.

TRIM POSITION LED INDICATOR REMOVAL

Perform this procedure to remove the LED indicators associated with the Trim Indicator.

1. Remove the two screws holding in the center console panel assy. Refer to Figure 31-18 for the location of the two screws.
2. Remove the access panel on the starboard side of the center console.
3. Reach through the access hole and disconnect the wires leading to the two post lights.
4. Reach through the access hole and using a ¼-inch socket to loosen and remove the nuts and lock washers on the underside of the post lights.

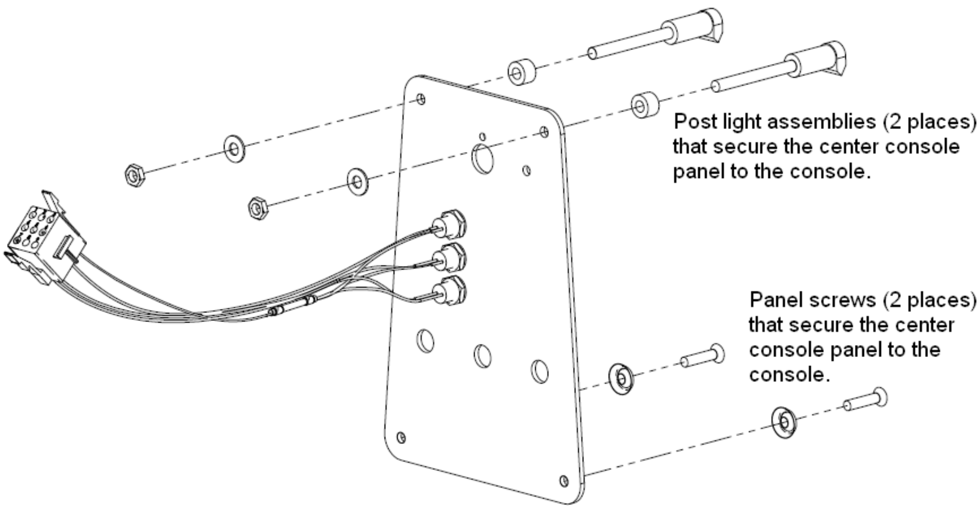


Figure 31-18 Center Console Panel

5. Carefully pull the center console panel out.

NOTE

You will not be able to remove the center console panel totally; however, it will come out enough to work on the LED indicators.

6. Disconnect the connector, P/J55, leading to the center console.
7. Carefully cut the cable ties around the bundle that includes the cable for the LED Indicator.
8. Mark for identification and cut the red wires going to splice SP030. Discard the splice.
9. If there is a splice on the black wire, mark for identification and cut the black wire going to splice. Discard the splice. If there is no splice on the black wire, remove the black wire and the associated pin from the connector shell for J55.

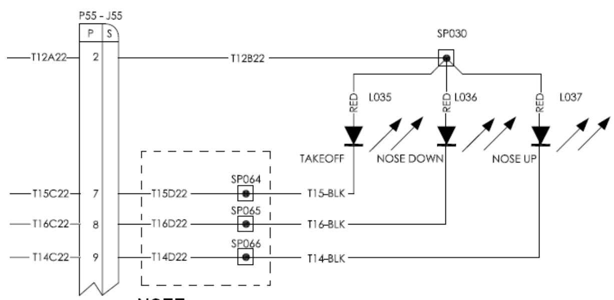
10. Use a 10mm open-end wrench to loosen and remove the nut.
11. Slide the indicator out through the front of the center console panel.
12. If a replacement indicator is not available, install the upper center console panel; otherwise proceed to the Trim Position LED Indicator Installation procedure.

This completes the Trim Position LED Indicator Removal procedure.

TRIM POSITION LED INDICATOR INSTALLATION

Perform this procedure to install any of the LED indicators mounted to the instrument panel.

1. If installing a replacement indicator immediately after removing an previous indicator, then proceed to step 3 below.
2. Perform step 1 through 6 of Trim Position LED Indicator Removal procedure on page 30 of this chapter.
3. Install the LED indicator through from the front of the center console panel.
4. Secure the LED indicator with the nut removed in step 10 of Trim Position LED Indicator Removal procedure on page 30 of this chapter.
5. Connect the red wires from all LED indicators and wire number T12B22 by splicing them together using a splice with then number SP030. See Chapter 20 – *Standard Practices* for wire splicing procedures and requirements.
6. If there was a splice on the black wire before, connect the black wire to its associated white wire using an appropriately numbered splice. If the black wire connected directly to the connector shell for J55, mount a connector pin to the black wire and insert the pin into the correct hole in J55. See Figure 31-19 for more information. See Chapter 20 – *Standard Practices* for wire splicing procedures and requirements.



NOTE:
Early airplanes had splices SP064, SP065, and SP066, and wires T15D22, T16D22, and T14D22

Later airplanes eliminated these wires and splices

Figure 31-19 Trim Indicator Schematic

7. Install the upper center console panel in to the center console. Secure the panel with the screws removed in step 1 of procedure on page 32 of this chapter.
8. Install the lock washer and nuts on to the post lights removed in step 4 of the Trim Position LED Indicator Removal procedure on page 32 of this chapter.
9. Connect the post light wires to the post lights.
10. Move the battery master switch to the ON position.

11. Verify the trim tabs are in the Takeoff position. If the trim tabs are not in the taker-off position, use the trim switch to bring the trim tabs to the take-off position.
12. If the trim tabs are in the Takeoff position, verify the take-off green LED indicator is illuminated.
13. On the instrument console, move the Bright/Dim/Test switch to the Bright position. Verify the Takeoff LED indicator is illuminated and bright.
14. On the instrument console, move the Bright/Dim/Test switch to the Dim position. Verify the Takeoff LED indicator is illuminated and dimmer than the Bright setting.
15. On the instrument console, move the Bright/Dim/Test switch to the Test position. Verify the Takeoff LED indicator is illuminated and is brighter than Dim and dimmer that Bright.
16. With the Bright/Dim/Test switch in the Test position, verify the other Trim indicators are not illuminated.
17. Using the trim switch, move the trim tabs to the nose up position. Verify the amber Nose Up LED indicator is illuminated.
18. On the instrument console, move the Bright/Dim/Test switch to the Bright position. Verify the Nose Up LED is illuminated and bright.
19. On the instrument console, move the Bright/Dim/Test switch to the Dim position. Verify the Nose Up LED is illuminated and dimmer than the Bright setting.
20. On the instrument console, move the Bright/Dim/Test switch to the Test position. Verify the Nose Up LED is illuminated and is brighter than Dim and dimmer that Bright.
21. With the Bright/Dim/Test switch in the Test position, verify the other trim indicators are not illuminated.
22. Using the trim switch, move the trim tabs to the nose down position. Verify the amber Nose Down LED indicator is illuminated.
23. On the instrument console, move the Bright/Dim/Test switch to the Bright position. Verify the Nose Down LED is illuminated and bright.
24. On the instrument console, move the Bright/Dim/Test switch to the Dim position. Verify the Nose Down LED is illuminated and dimmer than the Bright setting.
25. On the instrument console, move the Bright/Dim/Test switch to the Test position. Verify the Nose Down LED is illuminated and is brighter than Dim and dimmer that Bright.
26. With the Bright/Dim/Test switch in the Test position, verify the other trim indicators are not illuminated.
27. Using the trim switch, return the trim tabs to the take-off position.
28. Move the battery master switch to the OFF position.

This completes the Trim Position LED Indicator Installation procedure.

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Section 31-30 Recorders

The XL-2 airplane has two recording systems. These are the VM-1000FX and the Engine Data Interface. This section describes both systems.

Section 30-01 VM-1000FX Engine Instrument Display

The VM-1000FX integrated engine instrument display system can record, and display, maximum and minimum engine operating information about each flight.

Figure 31-20 VM-1000FX Engine Instrument Display

Information from the VM-1000FX system can be accessed by successive presses of the rightmost button at the bottom of the engine instrument display panel ("button 5", the rightmost button at bottom of display).

When electrical power is applied before engine start or after engine shutdown, the digital display in the center of the RPM display indicates total engine hours since installation. Hours are counted any time the engine is operating at greater than 1500 RPM.

When button 5 is pressed once, the first set of data are flight minimums encountered (i.e., lowest oil pressure, lowest voltage, amperage, etc.). Also the RPM digital display now shows the actual flight hours in hours and tenths of hours.

Press button 5 again. The next set of data are flight maximums encountered (i.e., max CHT, max Oil Temp, max RPM, etc.).

Press button 5 a third time. The Flight Data Recorder is shut off. The recorder data will automatically shut off in approximately 20 seconds if no button is pressed. Recorded data will automatically be overwritten on the next flight.

Section 30-02 Engine Data Interface

The aircraft has an Engine Data Interface (EDI). The purpose of the system is to acquire engine data from the Serial Bus Controller (SBC-100), process data for storage, compress data, store data, and provide for ground retrieval of the stored data. It communicates with the SBC-100 and ground test equipment.

The EDI-200 is a computer-based engine data acquisition, recording, retrieval, and communication system intended for installation in a General Aviation aircraft powered by a reciprocating engine. The EDI-200 interfaces with external and internal hardware, and processing elements, systems or equipment in order to perform its specified functions.

The EDI-200 components are designed for easy installation and replacement, consistent with the location of the component. It is removable without use of special tools, taking about 15 minutes. All mounting fasteners are accessible without disassembling the unit. Connectors attach via size and shape. Two connector screws act as locking screws. Fasteners are re-useable and the act of removing or replacing the unit does not damage the mating interface on the engine or airframe.

Table 31-2 EDI-200 Unit

Section 30-03 Engine Data Interface Procedures

This section details the procedures to remove and install the Engine Data Interface.

ENGINE DATA INTERFACE REMOVAL

Perform this procedure to remove the engine data interface annunciator.

CAUTION

Do not attempt to remove the EDI without taking the CB panel out of the instrument console. The cabling and connectors are tie-wrapped to the EDI that must be removed before removing the EDI from the instrument console.

CAUTION

Although this procedure has the ALT and BAT master switches, the FADEC PWR A & B switches, and the ignition switch in the OFF position, there is still voltage from the secondary battery on the terminal barrier strip mounted to the power distribution harness assembly behind the CB panel.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Pull the BAT1 (CB001) circuit breaker to OPEN.
3. From below the instrument console, under the EDI, remove the four screws that secure the EDI to the instrument console. Do not pull on the EDI in an attempt to remove the EDI at this point.
4. Remove the seven screws holding the CB (Circuit Breaker) Panel assembly. Refer to Figure 31-21 for location of the screws.

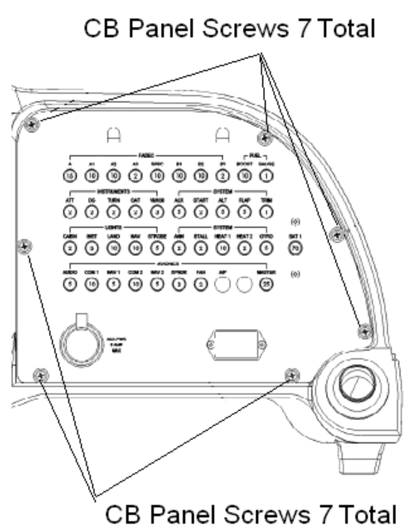

Figure 31-21 CB Panel showing the location of the Screws

5. Remove the CB panel from the instrument console by gently pulling the panel toward you, swing the panel down.
6. Place the CB panel on its face in to the block of soft foam cushioning.

7. Disconnect PVM03A ribbon cable connector from the DPU PVM03 connector. See Figure 31-22 for the location of PVM03A connector.

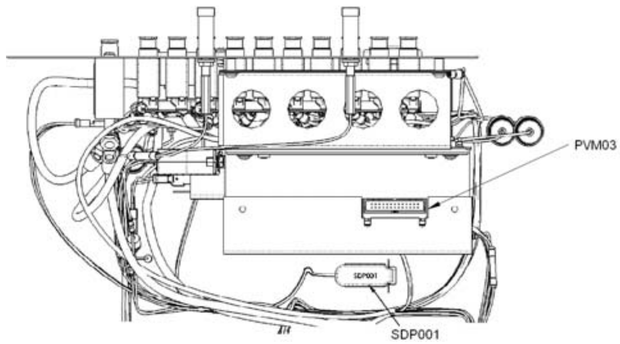

Figure 31-22 Top View of the CB Panel Showing the Connectors

8. Disconnect the Engine Data Interface, EDI, at the connector EDIJ1/SPD001. See Figure 31-23 for the location of the EDI and its associated connector.

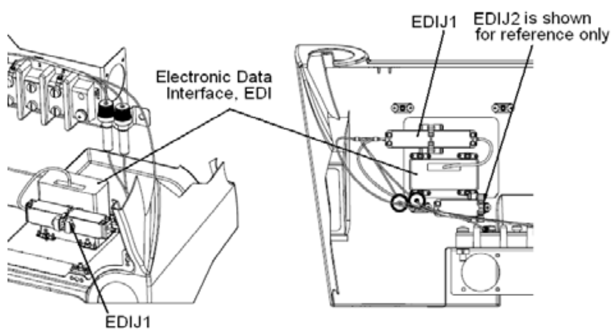

Figure 31-23 Showing the Location of the EDI Connector EDIJ1

9. Remove the EDI from the airplane.
10. If installing a replacement EDI immediately after removing an previous EDI, then proceed to step 5 of the Engine Data Interface Installation procedure on page 41 of this chapter.
11. Connect PVM03A ribbon cable connector from the DPU PVM03 connector. See Figure 31-22 for the location of PVM03A connector.
12. Insert the CB panel back into the instrument console.
13. Secure the CB panel to the instrument console using the seven screws removed in step 3 of this procedure.

This completes the Engine Data Interface Removal procedure.

ENGINE DATA INTERFACE INSTALLATION

Perform this procedure to remove the engine data interface annunciator.

CAUTION

Although this procedure has the ALT and BAT master switches, the FADEC PWR A & B switches, and the ignition switch in the OFF position, there is still voltage from the secondary battery on the terminal barrier strip mounted to the power distribution harness assembly behind the CB panel.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Pull the BAT1 (CB001) circuit breaker to OPEN.
3. If installing a replacement EDI immediately after removing a previous EDI, then proceed to step 5 below.
4. Perform step 1, 2, 4, and 6 of the Engine Data Interface Removal procedure on page 39 of this chapter.
5. Install the EDI up through the bottom of the Instrument console assembly.
6. Connect the Engine Data Interface, EDI connector EDIJ1 to the cable connector SPD001. See Figure 31-23 for the location of the EDI and its associated connector.
7. Apply sufficient tie wraps or lacing cord to secure the EDI connector to the EDI.
8. Connect PVM03A ribbon cable connector from the DPU PVM03 connector. See Figure 31-22 for the location of PVM03A connector.
9. Insert the CB panel back into the instrument console.
10. Secure the CB panel to the instrument console using the seven screws removed in step 4 of the Engine Data Interface Removal procedure on page 39 of this chapter.
11. Secure the EDI with the screws removed in step 3 of the Engine Data Interface Removal procedure on page 39 of this chapter.
12. Push the BAT1 (CB001) circuit breaker to CLOSE.

This completes the Engine Data Interface Installation procedure.

ENGINE DATA INTERFACE CHECKOUT

Perform this procedure to checkout the Engine Data Interface unit.

1. Position the aircraft in a safe run up area.
2. Perform a normal start of the engine and set the throttle to idle (850 to 950 RPM).
3. Using inspection mirror, observe the EDI-200 LEDs and verify expected status listed below:

Green LED Status	Red LED Status
On Steady – Receiving Data	On Steady - Recording

Table 31-3

4. Verify EDI monitor program displays data scrolling.
5. Perform an engine shutdown leaving power applied through the master switch, FADEC PWR A and FADEC PWR B Switches.
6. Using an inspection mirror observe the EDI-200 LEDs and verify expected status listed below.

Green LED Status	Red LED Status
On Steady – Receiving Data	OFF – Not Recording

Table 31-4

7. Turn off the power to the aircraft by toggling the FADEC A and B switches to the OFF position followed by a toggling of the master switch to the OFF position.

Engine data recording indication test complete.

ENGINE DATA INTERFACE INITIALIZATION AND TEST

The EDI-200 begins and completes a self-test as completely as practical depending on the aircraft operational state when power is applied to the EDI. This test will power up the EDI-200 and verify a valid initialization state is achieved.

NOTE

In the following steps EDI-200 operation will be verified by reading status LEDs located on the EDI-200 face plate. When verifying correct status, refer to the following table for a description of each LED condition observed.

EDI-200 Indicating Lamp Code	
Green LED	**Red LED**
Flashing, Alternately With Red - Initializing	Flashing Alternately With Green - Initializing
OFF - Unit Fault Or Unit Powered Off	Off - Not Recording Or Unit Powered Off
On Steady – Receiving Data	On Steady - Recording
Flashing – Unit OK, Not Receiving Data	Flashing - Disk Error (CF disk full, file corrupted)

1. Apply power to the aircraft by closure of the split master switch.
2. Using an inspection mirror, view the EDI-200 LEDs and verify expected status listed below:

Green LED Status	Red LED Status
Flashing, Alternately With Red – Initializing - Process will take approximately 30 seconds	Flashing Alternately With Green – Initializing - Process will take approximately 30 seconds
Flashing – Unit OK, Not Receiving Data	OFF – Not Recording

3. Power up FADEC A by closing FADEC PWR Switch A. Using inspection mirror, view the EDI LEDs and verify expected status listed below:

Green LED Status	Red LED Status
On Steady – Receiving Data	Off - Not Recording

4. Power up FADEC B by closing FADEC PWR switch B. Using inspection mirror, view the EDI LEDs and verify expected status listed below:

Green LED Status	Red LED Status
On Steady – Receiving Data	Off - Not Recording

5. Place FADEC PWR A and FADEC PWR B switches in the OFF state. Using inspection mirror, view the EDI LEDs and verify expected status listed below:

Green LED Status	Red LED Status
Flashing – Unit OK, Not	Off - Not Recording

Receiving Data	

6. Remove power from the aircraft by deactivation of the master switch. Using inspection mirror, view the EDI LEDs and verify expected status listed below:

Green LED Status	**Red LED Status**
OFF - Unit Fault Or Unit Powered Off	Off - Not Recording Or Unit Powered Off

This completes the Engine Data Interface Initialization and Test procedure.

Section 31-50 Stall Warning

The XL-2 airplane comes with a voice alert stall warning annunciator. The annunciator connects to a lift switch mounted in the port wing. The annunciator mounts on the left side of the avionics stack, behind the avionics panel in the instrument panel console. See Figure 31-24 for the location of the stall warning annunciator.

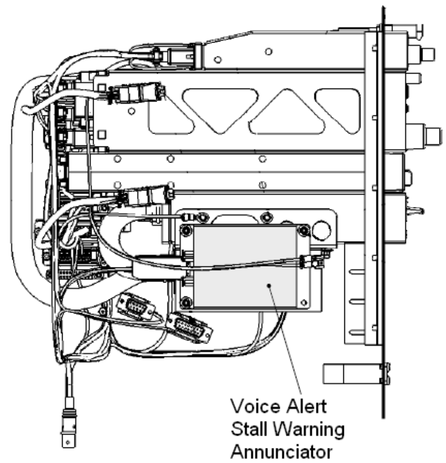

Figure 31-24 Stall Warning Annunciator

The sensor for the stall warning annunciator mounts in the middle of the leading edge of the port wing.

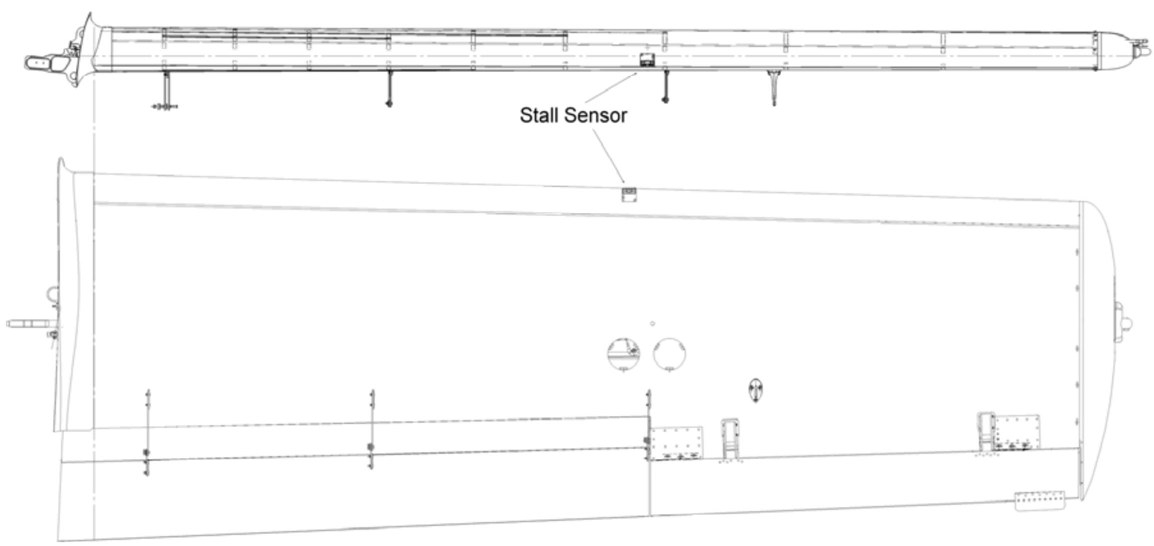

Figure 31-25 Stall Sensor Location

Section 50-01 Stall Warning Procedures

This section details the procedures to remove and install the stall warning annunciator and the stall warning sensor.

STALL WARNING REMOVAL

Perform this procedure to remove the stall warning annunciator.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Pull the BAT1 (CB001) circuit breaker to OPEN.
3. Remove the avionics radios from the avionics panel. To remove the radios, insert a hex key ("3/32" Allen wrench") into the hole in faceplate of the radio and rotate counter-clockwise until the retaining claw is released. Slide the radio towards you and out of the radio tray.

CAUTION

Removal of the avionics radios reduces the weight of the avionics panel. This will make the panel easier to handle and less likely to damage the Instrument Panel Console assembly surfaces when removing the panel from the console.

4. Place the radios outside the airplane.
5. Remove the eight screws holding in the avionics panel assy. Refer to Figure 31-26 for the location of the eight screws.

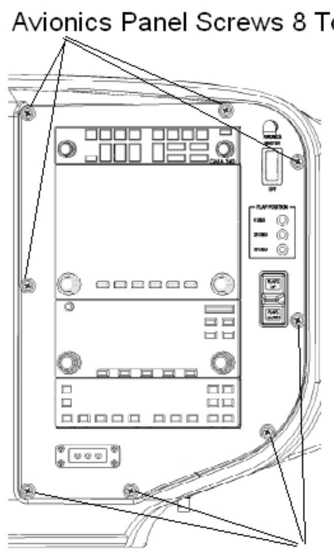

Figure 31-26 Avionics Panel Showing the Location of the Screws

6. Carefully pull the avionics panel out of the instrument console.

NOTE

The avionics panel will not come complete out of the instrument console. However, the panel will come out sufficiently to remove the stall warning annunciator.

7. Remove the connector on the side of the stall warning annunciator.
8. Remove the three 3/16 inch nuts that secure the stall warning annunciator to the avionics panel.
9. Remove the stall warning annunciator.
10. If immediately installing a replace stall warning annunciator go to step of the procedure on page of this chapter.
11. Carefully insert the avionics panel in to the instrument console.
12. Install the screws removed in step 5 of this procedure.
13. Install the radios removed in step 3 of this procedure.

This completes the Stall Warning Removal procedure.

STALL WARNING INSTALLATION

Perform this procedure to install the stall warning annunciator.

1. If installing a replacement stall warning annunciator immediately after removing a previous stall warning annunciator, then proceed to step 5 below.
2. Perform step 1 through 6 of the Stall Warning Removal procedure on page 46 of this chapter.
3. Mount the stall warning annunciator to the side of the avionics panel on the studs provided.
4. Secure the stall warning annunciator with the 3/16 inch nuts removed in step 8 of the Stall Warning Removal procedure on page 46 of this chapter
5. Connect the stall warning cable connector to the stall warning annunciator and secure the connector with the captured screws on the connector.
6. Carefully insert the avionics panel in to the instrument console.
7. Secure the avionic panel with the screws removed in step 5 of the Stall Warning Removal procedure on page 46 of this chapter.

This completes the Stall Warning Installation procedure.

STALL WARNING SENSOR REMOVAL

Perform this procedure to remove the stall sensor.

1. Check that all power is off on the airplane.
2. Use a sharp permanent felt tip marker to mark the location on the plate of the switch the location of the four mounting screws.
3. Remove four screws securing the lift switch.
4. Slowly remove the switch from the wing.
5. Disconnect the switch from the cable.
6. If a replacement sensor switch is not immediately available for installation, secure the cable in the wing and place a cover over the sensor location. Mark the airplane as non-flyable.

This completes the Stall Warning Sensor Removal procedure.

STALL WARNING SENSOR INSTALLATION

Perform this procedure to install the stall sensor.

1. If installing a replacement stall warning sensor immediately after removing a previous stall warning sensor, then proceed to step 3 below.
2. Perform step 1 through 6 of the Stall Warning Sensor Removal procedure on page 49 of this chapter.
3. Check that all power is off on the airplane.
4. If installing the switch removed in the Stall Warning Sensor Removal procedure above, go to step 6.
5. Transfer the alignment marks from the previous switch assembly to the switch assembly that will be installed.
6. Connect the cable from the switch to the cable in the wing.
7. Carefully install the switch into the wing.
8. Loosely install all four screws that secure the switch sensor to the wing.
9. Align the switch using the marks on the plate for the switch.
10. Loosely tighten the four screws checking/correcting the alignment of the marks to the screws until all four screws are tight and the switch is securely fasten to the wing.

STALL WARNING SYSTEM CHECK

Perform this procedure to check the stall warning system.

1. Apply power to the airplane by pressing the BAT Master switch to ON.
2. Wait for 15 seconds to allow all system to come on line.
3. While someone monitors the systems in the cockpit, a second person needs to gentle lift the stall sensor vane. With the vane in the up position, the annunciator must report a stall condition.
4. Remove power from the airplane by pressing the BAT Master switch to OFF.

This completes the Stall Warning System Check procedure.

Section 50-02 Stall Warning System Troubleshooting

This section provides information to trouble shoot the stall warning system. See Table 31-5 for information to troubleshoot the stall warning system.

Issue	Cause	Repair
Stall warning annunciator does not sound when lift switch is lifted	No Power	Check that power is ON.
	Annunciator is inoperable	Replace annunciator
	Lift switch is faulty	Replace lift switch
Lift switch does not move freely	Dirt or debris in switch	Remove switch, clean, install switch

Table 31-5 Stall Warning System Troubleshooting

Section 31-60 Central Display Systems

All engine instruments, as well as voltage and amperage of the airplane electrical system, are combined in the VM1000FX Integrated Engine Instrument Display system. For details, refer to Chapter 77 – *Engine Indicating*.

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CHAPTER 32

LANDING GEAR

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Table of Contents

SECTION 32-00	GENERAL	5
SECTION 32-10	MAIN LANDING GEAR	7
SECTION 10-01	PERIODIC MAINTENANCE	7
SECTION 10-02	MAIN LANDING GEAR PROCEDURES	7
	MAIN LANDING GEAR ASSEMBLY REMOVAL	8
	MAIN LANDING GEAR ASSEMBLY INSTALLATION	10
	MAIN LANDING GEAR OPERATIONAL CHECK AND INSPECTION	11
SECTION 10-03	MAIN LANDING GEAR TROUBLESHOOTING GUIDE	12
SECTION 32-20	NOSE LANDING GEAR	13
SECTION 20-01	PERIODIC MAINTENANCE	13
SECTION 20-02	NOSE LANDING GEAR PROCEDURES	13
	NOSE LANDING GEAR ASSEMBLY REMOVAL	14
	NOSE LANDING GEAR ASSEMBLY INSTALLATION	16
	NOSE LANDING GEAR OPERATIONAL CHECK AND INSPECTION	17
SECTION 20-03	NOSE LANDING GEAR TROUBLESHOOTING GUIDE	18
SECTION 32-40	WHEELS AND BRAKES	19
SECTION 40-01	WHEEL FAIRINGS	19
SECTION 40-02	PERIODIC MAINTENANCE	22
SECTION 40-03	WHEEL FAIRING PROCEDURES	22
	MAIN WHEEL FAIRING INITIAL INSTALLATION	23
	NOSE WHEEL FAIRING INITIAL INSTALLATION	25
	MAIN WHEEL FAIRING REMOVAL	26
	NOSE WHEEL FAIRING REMOVAL	27
	MAIN WHEEL FAIRING INSTALLATION	28
	NOSE WHEEL FAIRING INSTALLATION	29
	WHEEL FAIRING OPERATIONAL CHECK AND INSPECTION	31
SECTION 40-04	WHEEL FAIRING TROUBLESHOOTING GUIDE	32
SECTION 40-05	WHEELS	33
SECTION 40-06	PERIODIC MAINTENANCE	33
SECTION 40-07	WHEEL PROCEDURES	33
	NOSE WHEEL REMOVAL	34
	NOSE WHEEL INSTALLATION	35
	MAIN WHEEL REMOVAL	36
	MAIN WHEEL INSTALLATION	37
	WHEEL OPERATIONAL CHECK AND INSPECTION	39
SECTION 40-08	WHEEL TROUBLESHOOTING GUIDE	40
SECTION 40-09	TOE BRAKE SYSTEM	42
SECTION 40-10	PERIODIC MAINTENANCE	44
SECTION 40-11	TOE BRAKE PROCEDURES	44
	TOE BRAKE MASTER CYLINDER REMOVAL	45
	TOE BRAKE MASTER CYLINDER INSTALLATION	47
	TOE BRAKE OVER-CENTER STOP REMOVAL AND INSTALLATION	49
	PARKING BRAKE VALVE REMOVAL	50
	PARKING BRAKE VALVE INSTALLATION	51
	TOE BRAKE OPERATIONAL CHECK AND INSPECTION	52
SECTION 40-12	TOE BRAKE TROUBLESHOOTING GUIDE	53
SECTION 40-13	FINGER BRAKE SYSTEM	55
SECTION 40-14	PERIODIC MAINTENANCE	56
SECTION 40-15	FINGER BRAKE SYSTEM PROCEDURES	56

Finger Brake Master Cylinder Removal		57
Finger Brake Master Cylinder Installation		59
Finger Brake Operational Check and Inspection		60
Section 40-16	Finger Brake Troubleshooting Guide	61
Section 40-17	Main Wheel Brake Caliper System	63
Section 40-18	Periodic Maintenance	63
Section 40-19	Main Wheel Brake Caliper System Procedures	63
Main Wheel Brake Caliper Removal		65
Main Wheel Brake Caliper Installation		67
Brake Caliper Pad/Liner Replacement		69
Brake Bleeding Procedure		74
Brake Caliper Operational Check and Inspection		76
Section 40-20	Main Wheel Caliper System Troubleshooting Guide	78
Section 32-50	Steering	79
Section 50-01	Periodic Maintenance	79
Section 50-02	Steering Procedures	79
Nose Gear Leg Serial Number Verification		80
Castor Stop/Friction Base Removal		82
Castor Stop/Friction Base Installation		83
Nose Gear Shimmy Damper Torque Test		85
Nose Gear Steering Operational Check and Inspection		87
Section 50-03	Steering Troubleshooting Guide	89

Section 32-00 General

The Liberty XL-2 landing gear is a fixed tricycle landing gear. All of the three landing gear wheels employ 5.00 x 5 tires (with inner tubes) inflated to a pressure of 50 +2 /-0 psi for all three tires.

The main undercarriage (U/C) landing gear legs are fabricated from single billets of heat-treated aluminum. Machined surfaces at the top of the gear leg provide a bearing surface for the inboard attachment bolt, one of the two points at which each gear leg is secured to the center fuselage space frame. A specific arrangement of fasteners, including a Belleville spring washer and a flexible bushing, is required to secure the inboard end of the gear leg. At the outer portion of the space frame, the gear is secured in a saddle with two additional bolts. Self-locking castellated nuts and split pins are used to secure all three bolts for each landing gear leg. Disc brakes are installed on the Port and Starboard main landing gear wheels. Steering is effected by differential application of the main landing gear brakes.

At its lower extremity, each main landing gear leg employs machined surfaces to locate and accommodate the wheel brake caliper mounting plate and the wheel axle.

The nose undercarriage leg is fabricated from solid heat-treated spring steel. The assembly is free to caster (steer) left / right up to 85º. A caster stop prevents rotation beyond this point to prevent interference between the nose landing gear and the propeller.

Access to the nose attachment fitting, main gear attachment fittings, and the cockpit components of the Finger Brake System are obtained by removal of the fuselage belly fairing, cover-plate upper undercarriage legs, and wheel fairing as necessary.

The Toe Brake System's master cylinders are accessible from inside the cockpit. The parking brake valve is accessible by removal of the belly fairing, cover-plate upper undercarriage legs, and wheel fairing as necessary.

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Section 32-10 Main Landing Gear

Section 10-01 Periodic Maintenance

Periodic Main Landing Gear maintenance entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 10-02 Main Landing Gear Procedures

This section contains the procedures for the removal and installation of the main landing gear.

MAIN LANDING GEAR ASSEMBLY REMOVAL

Perform this procedure to remove the main landing gear assembly. This procedure applies to both the port and starboard main landing gear assembly.

NOTE

Minor spillage of brake fluid is unavoidable. Place absorbent material below fitting area as required; and have appropriate plastic caps or plugs available for both fuselage brake line and landing gear fitting.

1. Securely chock opposite main undercarriage landing gear and nose landing gear.
2. Remove fuselage belly fairing, cover-plate upper undercarriage legs, and wheel fairing if applicable.
3. If removing “PORT” main landing gear, disconnect electric fuel boost pump from chassis attachment. in accordance with Liberty Maintenance Manual Chapter 28 - *Fuel System*.
4. Cut and remove safety wire. Remove two bolts securing brake caliper. Cut away three (3) tie-wraps around brake line and undercarriage leg. Tie back brake caliper and hose back to structure.
5. Remove and discard split pin from castellated nuts on all three main landing gear attachment bolts.
6. Jack the airplane, using jack point on undercarriage restraint lower (saddle) on gear to be removed, until wheel of gear to be removed is clear of floor. Place a padded sawhorse of sufficient height under second rib of wing, on jack side, to keep wheel off ground. Make sure wing flaps are retracted and sawhorse support’s main wing spar. Remove center main gear attachment bolt, making careful note of position and order of all washers and flexible bushing.
7. Remove the jack from under undercarriage restraint lower (saddle) jack point.
8. Remove outboard main gear attachment bolts and main gear undercarriage lower restraint (saddle). Slide the main gear leg outboard clear of the space frame structure.

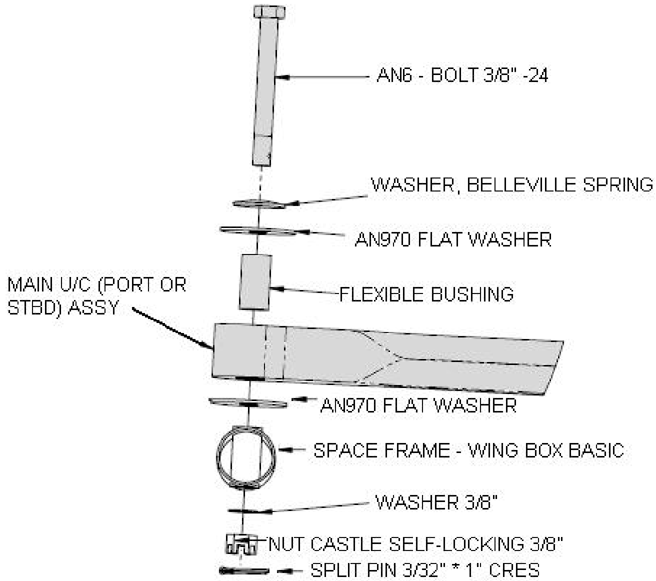


Figure 32-1 Exploded View of Main Landing Gear Attachment Bolt

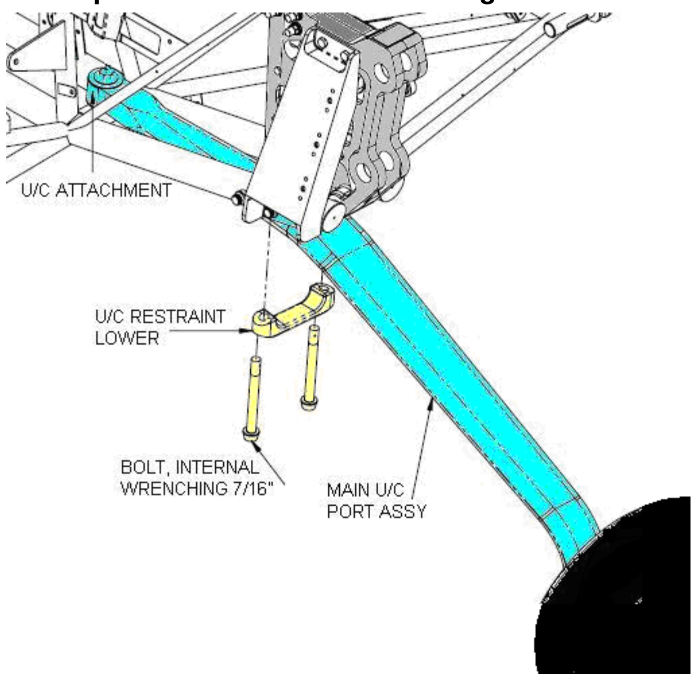


Figure 32-2 Main Landing Gear Port Attachment Bolt and undercarriage Restraint Lower (Saddle)

This completes the Main Landing Gear Assembly Removal procedure.

MAIN LANDING GEAR ASSEMBLY INSTALLATION

Perform this procedure to install the main landing gear assembly. This procedure applies to both the port and starboard main landing gear assemblies.

NOTE

Any misalignment or tracking issue with the main landing gear may be an indicator of a hard landing; if this is suspected, perform the hard landing checklist in Chapter 05 - Time Limits/Maintenance Checks/Inspection Intervals.

1. Install flexible bushing in inboard end of main gear leg.
2. Place main landing gear leg in position.
3. Install outboard main gear attachment undercarriage lower restraint (saddle) and bolts; tighten "hand tight" only at this time. Verify routing of hose.
4. Install inboard main gear attachment bolts with all hardware in correct order shown in Figure 32-1.
5. Tighten inboard main gear attachment bolt to 50 +0/-10 in-lbs rotate nut counter clockwise to align nearest slot and hole to install spit pin.
6. Tighten outboard main gear attachment bolts to 475-± 5 in-lbs rotate nut counter clockwise to align nearest slot and hole to install spit pin.
7. Bleed brake (if brake system has been disturbed); reference Brake Bleeding Procedure on page 74 of this chapter.
8. Replace jack under gear undercarriage restraint lower (saddle) and jack airplane to remove sawhorse from under wing.
9. Lower the airplane from jack.
10. If the fuel pump was removed, install the fuel pump.
11. Reinstall fuselage belly fairing, cover- plate upper undercarriage legs, and wheel fairing if applicable.

This completes the Main Landing Gear Assembly Installation procedure.

MAIN LANDING GEAR OPERATIONAL CHECK AND INSPECTION

The following procedure performs main landing gear operational check and inspection. This procedure applies to both the port and starboard main gear assemblies.

1. Position the aircraft master switch OFF.
2. Chock main wheels.
3. Remove the belly panel in accordance with Liberty Maintenance Manual Chapter 53-*Fuselage*.
4. Check Main landing gear legs for damage and deformation.
5. Check Main landing gear port and starboard attachment bolts for condition, security and condition of bushings.
6. Install the belly panel in accordance with Liberty Maintenance Manual Chapter 53-*Fuselage*.

This completes the main landing gear operational check and inspection procedure.

Section 10-03 Main Landing Gear troubleshooting Guide

Complaint	Possible Cause	Remedy
Main gear leg bent	Hard Landing	Perform a hard landing inspection in accordance with Liberty Maintenance Manual Chapter 05 Replace main gear leg
Negative camber	Main gear leg bent	Replace main gear leg
Gear leg cracks	Hard Landing	Perform a hard landing inspection in accordance with Liberty Maintenance Manual Chapter 05 Replace main gear leg
Attach bolts loose	Hard Landing	Perform a hard landing inspection in accordance with Liberty Maintenance Manual Chapter 05 Replace damaged hardware as identified on inspection. Install main landing gear hardware in accordance with Liberty Maintenance Manual Chapter 32
Worn bushings	Hard landing	Perform a hard landing inspection in accordance with Liberty Maintenance Manual Chapter 05 Replace worn bushings Install main landing gear hardware in accordance with Liberty Maintenance Manual Chapter 32

Section 32-20 Nose Landing Gear

Section 20-01 Periodic Maintenance

Periodic maintenance entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 20-02 Nose Landing Gear Procedures

This section contains the procedures for the removal and installation of the nose landing gear.

NOSE LANDING GEAR ASSEMBLY REMOVAL

Perform this procedure to remove the nose landing gear assembly.

CAUTION

Check that all electrical switches are OFF before rotating propeller.

NOTE

Some aircraft have a thin stainless shim (P/N 135A-04-757) located on notched AFT section of nose gear leg. This shim is required to allow proper engagement of nose gear tapered pin.

If nose gear is removed, verify that a shim is or is not in place in notched AFT section of leg prior to removal and reinstallation of leg. If shim is used, check the shim that it is reinstalled with leg. If shim is worn contact Liberty Customer Support. Verify no more than two washers are needed under nut during installation of tapered pin. If another washer is required, it may indicate shim (P/N 135A-04-757) not applied, Contact Liberty Customer Support.

1. Securely chock both main wheels.
2. Check the propeller blades are in horizontal position.
3. Remove fuselage belly panel, cover plate, and (nose wheel fairing if applicable).
4. Secure tail stand weighing at least 300 lbs. to AFT tie-down fitting; lower tail of airplane until nose wheel is just clear of floor. If a tail stand is not available, place a pad under tail ensuring rudder is free to move. Place 50 – 75 lbs. shot or sand bags on stabilizer root centerline on each side and carefully lower tail to floor. Aircraft should rest on tie-down fitting – do not perform this if tie-down fitting is not installed, as damage to rudder surface could occur.
5. Remove split pin from 7/16” taper pin securing AFT end of nose gear leg to fuselage center section space frame.
6. Using a pneumatic tool (protect threads from damage during removal), remove 7/16” taper pin.
7. To remove nose gear leg pull forward and out of mountings in fuselage center section space frame.

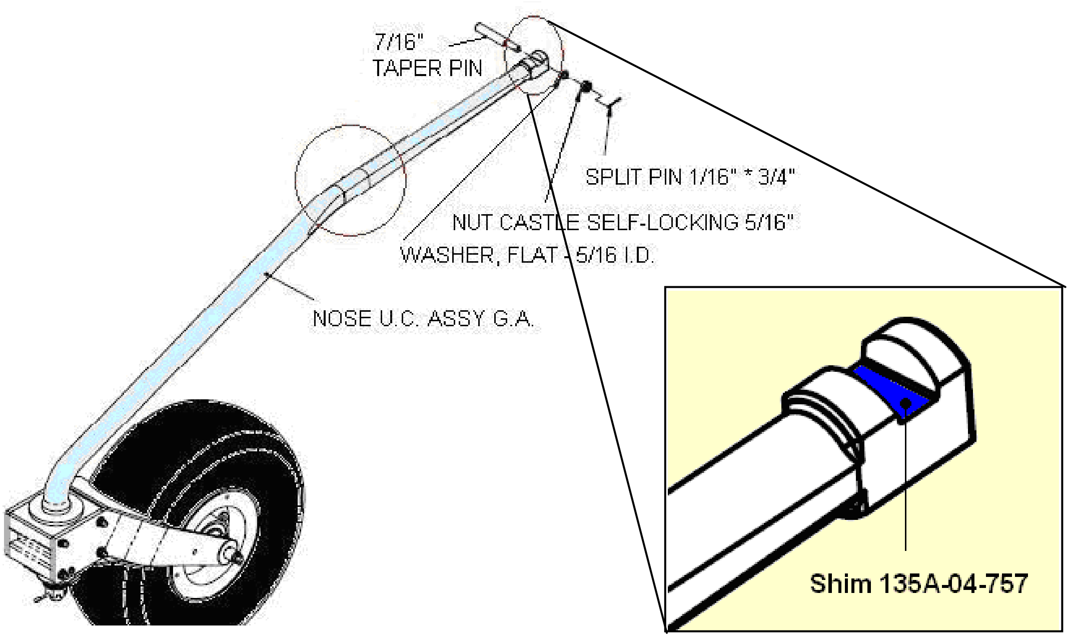


Figure 32-3 Nose Gear Installation

This completes the nose gear assembly removal procedure.

NOSE LANDING GEAR ASSEMBLY INSTALLATION

Perform this procedure to install the nose landing gear assembly.

CAUTION

Check that all electrical switches are OFF before rotating propeller.

NOTE

Some aircraft have a thin stainless shim (P/N: 135A-04-757) located on notched AFT section of nose gear leg. This shim is required to allow proper engagement of nose gear tapered pin.

If nose gear is removed, verify that a shim is or is not in place in notched AFT section of leg prior to removal and reinstallation of leg. If shim is used, check the shim that it is reinstalled with leg. If shim is worn contact Liberty Customer Support. Verify no more than two washers are needed under nut during installation of tapered pin. If another washer is required, it may indicate shim (P/N 135A-04-757) not applied, Contact Liberty Customer Support.

1. Slide nose gear through mountings in the fuselage center section space frame until rear end of nose gear is aligned with the taper pin holes.
2. Set taper pin shim back in position if previously install and secure nose gear leg by wet installing tapered pin with non-chromate CA 1000 with flat tapered section facing down (if applicable install shim (P/N 135A-04-757), tighten nut until pin is fully engaged into AFT notch in leg (no play or movement between end of nose gear and attachment fittings).
3. Install self-locking castle nut and install split pin in castle nut.
4. Verify end of tapered pin is recessed no more than 3/16" from end of welded tube when fully installed and no more than two washers are needed under nut. If another washer is required it may indicate shim (P/N 135A-04-757) not applied, re-inspect. If additional washer are necessary contact Liberty Customer Support.
5. Replace fuselage belly fairing, cover plate, and (wheel fairing if applicable).
6. Remove tail stand, and weights and lower nose wheel to floor.

This completes the nose landing gear assembly installation procedure.

NOSE LANDING GEAR OPERATIONAL CHECK AND INSPECTION

This procedure performs nose landing gear operational check and inspection.

1. Position the aircraft master switch OFF.
2. Chock main wheels.
3. Remove the belly panel in accordance with Liberty Maintenance Manual Chapter 53-*Fuselage*.
4. Check Nose landing gear for damage or deformation.
5. Check Nose landing gear attachment bolts for condition and security.
6. Check nose gear for lateral play.
7. Check Nose taper lock pin for excessive wear: larger inner diameter of nose lock pin barrel may not be less than 0.240 inches.
8. Install the belly panel in accordance with Liberty Maintenance Manual Chapter 53-*Fuselage*.

This completes nose landing gear operational check and inspection.

Section 20-03 Nose Landing Gear Troubleshooting Guide

Complaint	Possible Cause	Remedy
Nose gear leg bent	Hard Landing	Perform a hard landing inspection in accordance with Liberty Maintenance Manual Chapter 05 Replace nose gear leg
Nose gear leg crack	Hard Landing	Perform a hard landing inspection in accordance with Liberty Maintenance Manual Chapter 05 Replace nose gear leg
Nose gear lateral play	Taper pin wear	Replace taper pin
	Nose gear shim wear	Replace nose gear shim

Section 32-40 Wheels and Brakes

All three-wheel assemblies on the airplane are similar. However, only the two main wheels have disc brakes installed as shown in Figure 32-4. The main axles and nose wheel axle assemblies are different. Wheels must be disassembled for normal maintenance such as tire and tube replacement or wheel bearing service.

NOTE

Additional specifications on wheels and brakes can be obtained from the manufacturer, please contact vender for details:

Parker Hannifin, Aircraft Wheel & Brakes
1160 Center Road
Avon, OH. 44011-0158
Tel: 440-937-6211
Fax: 440-937-6416
Website: http://www.parker.com

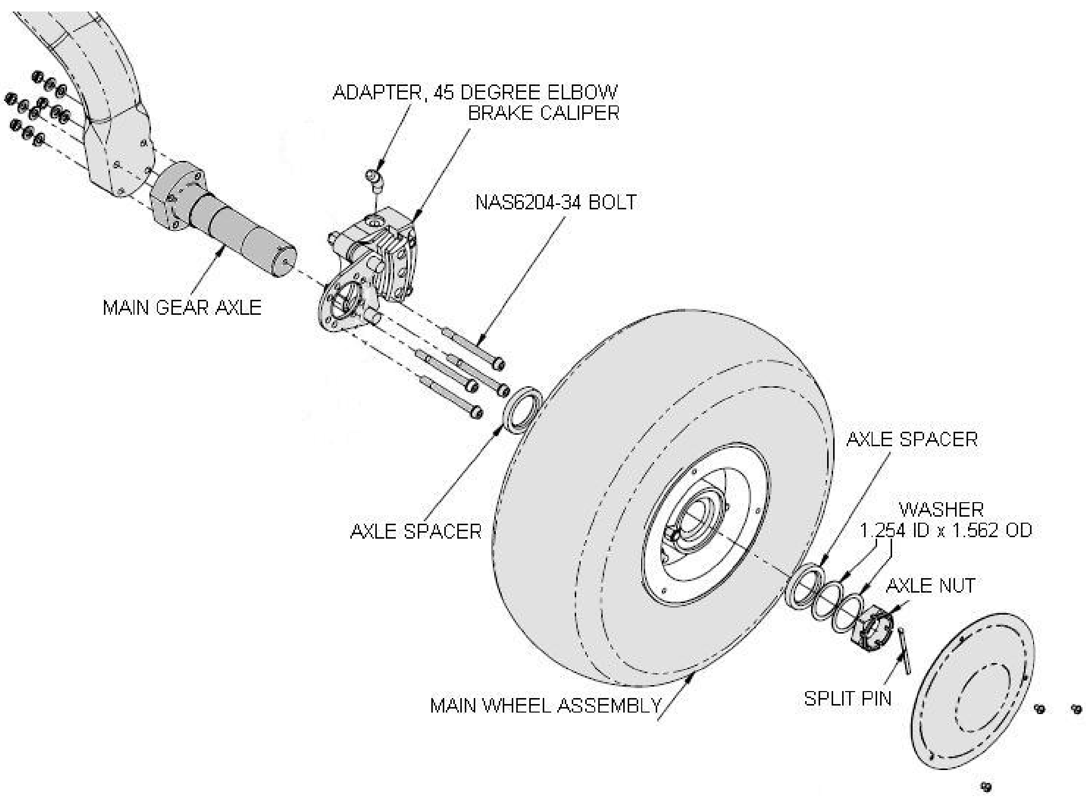


Figure 32-4 Main undercarriage Assembly

Section 40-01 Wheel Fairings

Wheel fairing installations consist of four major assemblies as shown in Figure 32-5. Assemblies include spacers, and supporting hardware. When referring to these assemblies the following descriptions apply:

- Nose Gear Fairing Forward (1)
- Nose Gear Fairing Assembly Aft (2)
- Main Wheel Fairing Assembly Port (3) with Port access panel (4 - not shown)
- Main Wheel Fairing Assembly starboard (5) with starboard access panel (6)

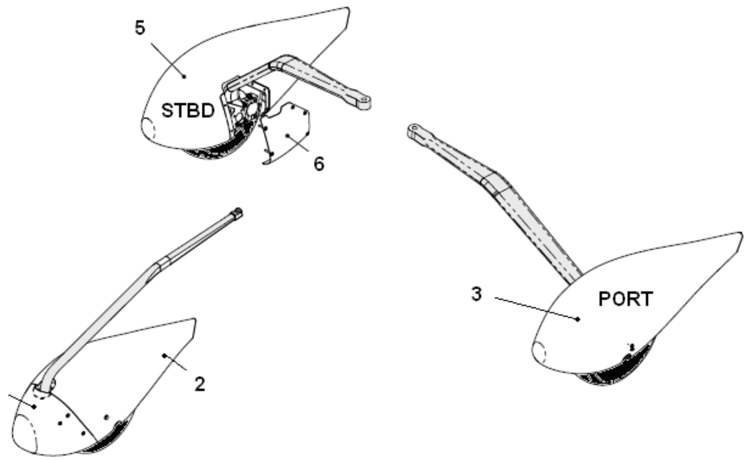


Figure 32-5 Wheel Fairing Installation Assembly

CAUTION

For installation of main wheel fairing the AN502-10-8 fasteners must be installed with high strength Loctite® No. 270 on threads, safety wire (MS20995-C32, CRES, 0.32-in diameter) to prevent loosening from in-flight vibration.

NOTE

Make sure wheels are always installed with air valve accessible through wheel fairing cut-out. Pay particular attention to nose wheel orientation, with respect to nose fairing after maintenance.

Install nose and main wheel fairing brackets, main spacers outboard and supporting hardware onto aircraft first to allow wheel fairings to be installed and removed with minimal effort.

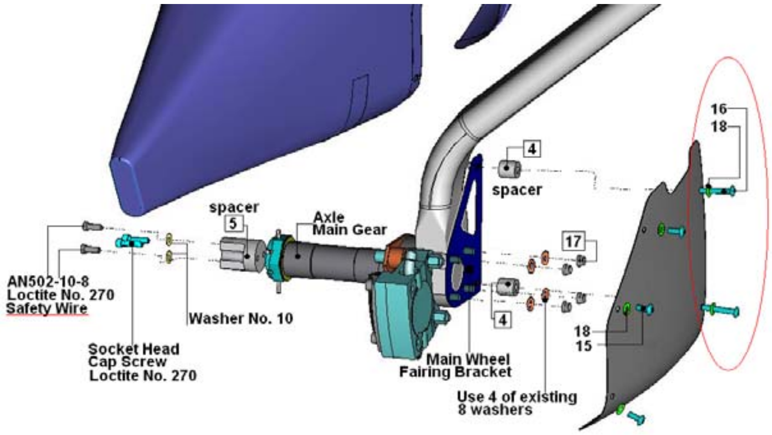


Figure 32-6 Exploded View of Main Wheel Fairing, Port

Wheel Fairing Component List			
Item	**Qty**	**Part Number**	**Description**
1	1	135A-40-021	Main Wheel Fairing Assembly, Port
2	1	135A-40-022	Main Wheel Fairing Assembly, Starboard
3	2	135A-40-305	Spacer, Nose Wheel Fairing
4	4	135A-40-307	Spacer, Main Wheel Fairing, Inboard
5	2	135A-40-308	Spacer, Main Wheel Fairing, outboard
6	1	135A-40-401	Nose Gear Fairing, Forward
7	1	135A-40-402	AFT Nose Gear Fairing Assembly
8	1	135A-40-405	Main Wheel Fairing Panel, Port
9	1	135A-40-406	Main Wheel Fairing Panel, Starboard
10	2	135A-40-515	Nose Gear Fairing Bracket
11	1	135A-40-585	Main Wheel Fairing Bracket, Port
12	1	135A-40-586	Main Wheel Fairing Bracket, Starboard
13	2	AN4C-15A	An4 - Bolt1/4" – 28
14	4	AN502-10-8	Machine Screw 10-32 (Loctite® 270 & Safety Wire)
15	12	AN526C-1032R8	Machine Screw 10-32
16	4	AN526C-1032R20	Machine Screw 10-32
17	10	MS21042-4	Self Locking Nut, 1/4-28
18	20	NAS1149C0332R	Washer No.10
19	4	NAS1149C0432R	Asher 1/4" Id 1/2" OD 0.032 Thick, Cad Plated
20	4	SCSH010C0012STBL	Socket Head Cap Screw (Loctite® 270)
21	As Req.	LCT270	LOCTITE® 270 Thread-locker Consumable Items
22	As Req.	MS20995-C32	Safety Wire MS20995-C32, CRES, 0.32-in dia.

Table 32-1 Wheel Fairing Component List

NOTE

Wheel fairing hardware installation must comply with following mandatory requirements:

1. AN502-10-8 - use Loctite® No. 270 on threads and are safety wired.

2. Socket head cap screws SCSH010C0012STBL - use Loctite® No. 270 on threads.

3. Torque in accordance with Liberty Maintenance Manual Chapter 20:

Installation Torque for Bolt/Nut Combinations Nut installation torque values (lubricant free, cadmium plated).		
Nominal Fastener Diameter (inch)	**Mid-Range Values (in-lbs)**	**Range (in-lbs) Min-Max (90 KSI in bolts)**
0.250	95	90-100
Installation Torque for Bolt/Nut Combinations in Composite Structures Locknut torque values:		
Nominal Fastener Diameter (inch)	**Mid-Range Values (in-lbs)**	**Range(in-lbs) Min-Max**
0.2500	70	60-80

Table 32-2 Installation Torque for Bolt/Nut Combinations

Section 40-02 Periodic Maintenance

Periodic Wheel Fairing maintenance entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 40-03 Wheel Fairing Procedures

This section contains the procedures to install the main wheel fairings and the nose wheel fairing. These procedures are performed as a part of landing gear, wheel, and break periodic maintenance operations. In each of those sections, a referral to perform this section will be made when fairing removal or installation is required.

MAIN WHEEL FAIRING INITIAL INSTALLATION

Perform this procedure to install main wheel fairings on an aircraft not previously so equipped.

NOTE

Once initial installation of nose and main wheel fairing brackets, outboard main spacers and additional supporting hardware are installed with the aircraft wheel fairings. These features permit later removal and re-installation with minimal effort.

1. Apply brakes to opposite wheel not being used and securely chock opposite main undercarriage landing gear and nose landing gear.
2. Check and to ensure tire pressure is 50 +2 /-0 psi.
3. Remove hubcap (3 screws each) on both Port and starboard side; and do not reuse.
4. Jack main landing gear in accordance with Liberty Maintenance Manual Chapter 07.

NOTE

Install FIRST: (Port) main brackets, spacer outboard, and supporting hardware.

When brackets have been installed on both main wheels, lower aircraft and remove jack. Install wheel fairings while all three wheels are on ground.)

5. Take main Port wheel off and split brake caliper allowing installation of main wheel fairing bracket. Wheel must be removed to access bolt head for proper torque. Torque bolts per the Main Wheel Installation procedure on page 37 of this chapter.
6. Verify axles have two drill holes. Drill holes should appear at outboard axle end. If no drill holes, then drill and tap two holes as specified by Figure 32-7.

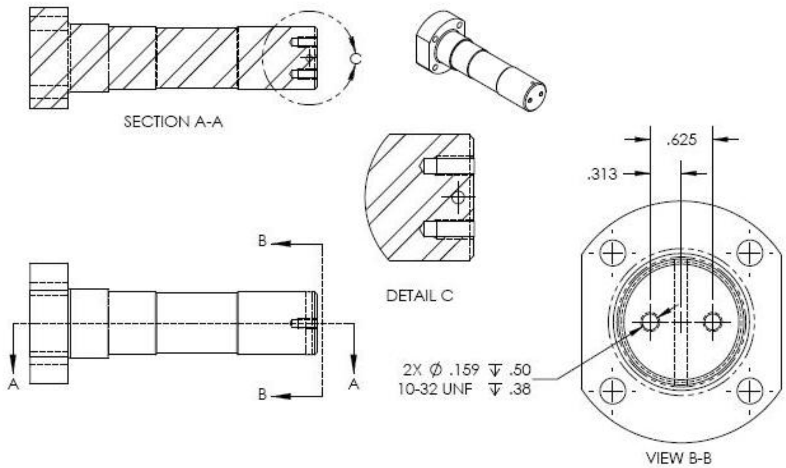


Figure 32-7 Drill Hole Location on Axle

7. Install wheel fairing bracket using four (4) of existing eight (8) washers.
8. Check the brake caliper that it is properly re-installed.
9. Install spacer outboard using a socket head cap screw and apply Loctite® No. 270 on threads.
10. Lower aircraft, remove jack and repeat for starboard main wheel, making sure to apply brakes and chock aircraft.
11. Slide main wheel fairing over wheel; align drill holes with spacer outboard. Apply high strength Loctite® No. 270 to threads on AN502-10-8 screws and a number 10 washer, safety wire and torque accordingly.
12. Install two (2) inboard spacers, align main wheel fairing panel and supporting hardware (washer no. 10 and AN526C-1032R20).
13. Repeat for other main wheel fairing.
14. Verify minimum 0.50 inch clearance exists between wheel fairing cutout in fairing and tire. Trim if necessary.
15. Verify wheel fairing ground clearance is minimum 7 inches.

This completes the Main Wheel Fairing Initial Installation procedure.

NOSE WHEEL FAIRING INITIAL INSTALLATION

Perform this procedure to install a nose wheel fairing on an aircraft not previously so equipped.

NOTE

Once initial installation of nose and main wheel fairing brackets, outboard main spacers and additional supporting hardware are installed with the aircraft wheel fairings. These features permit later removal and re-installation with minimal effort.

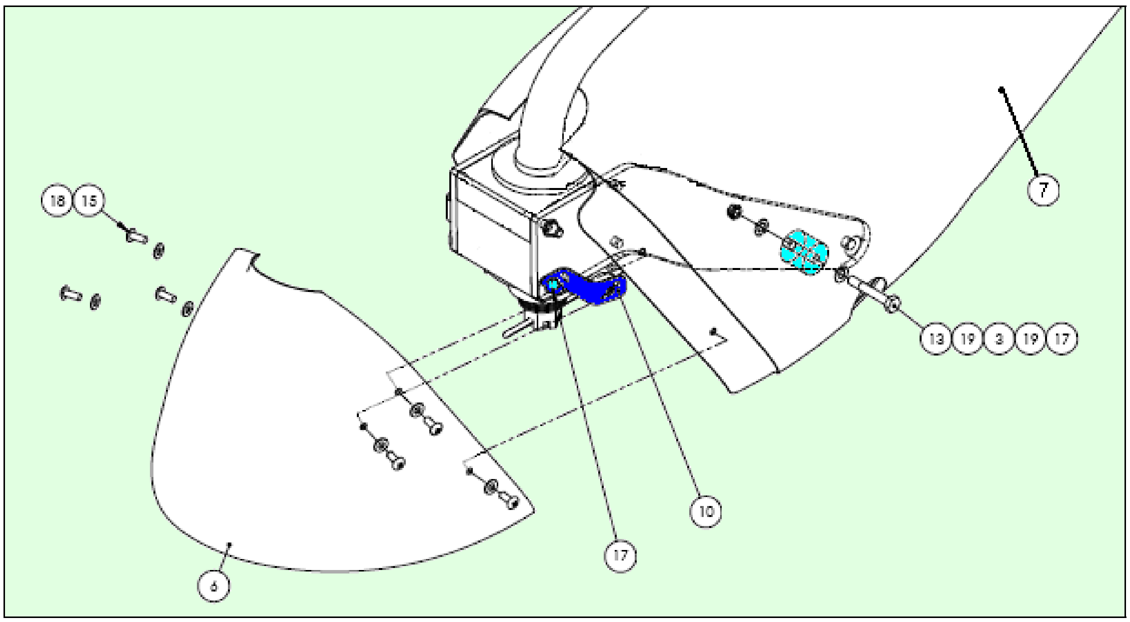


Figure 32-8 Exploded View of Nose Wheel Fairing

1. Securely chock main undercarriage landing gear.
2. Check and to ensure tire pressure is 50 +2 / -0 psi.
3. Install 2 nose gear fairing brackets.
4. Slide AFT nose gear fairing assembly on to nose landing gear assembly.
5. Align drill holes, nose wheel spacer, and supporting hardware (P/N: NAS1149C0432R washer and AN4C-15A Bolt).
6. Install nose gear fairing FWD and supporting hardware.
7. Verify minimum 0.50 inch clearance exists between wheel fairing cutout in fairing and tire. Trim if necessary.
8. Verify wheel fairing ground clearance is minimum 7 inches in accordance with Section 40-01Wheel Fairings on page 26 of this chapter.

This completes the Nose Wheel Fairing Initial Installation procedure.

MAIN WHEEL FAIRING REMOVAL

Perform this procedure to remove the main wheel fairings on an aircraft so equipped. Procedure is applicable to both the port and starboard main wheel fairings.

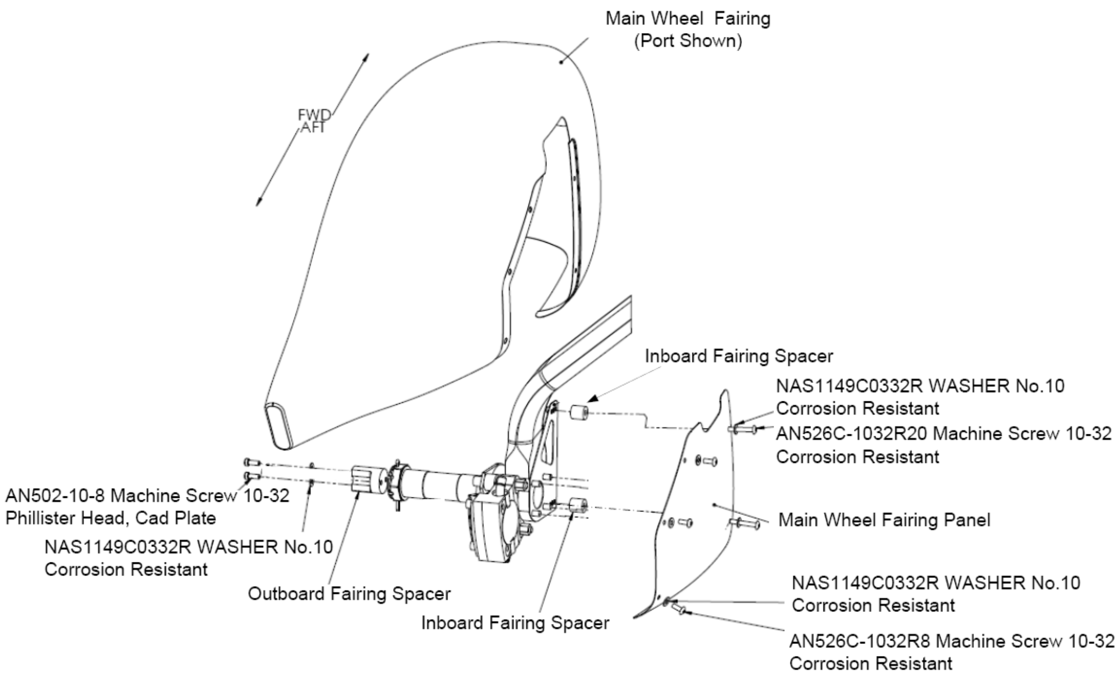


Figure 32-9 Main Wheel Fairing Installation

1. Position aircraft on a level surface and set the parking brake
2. Remove main wheel fairing panel screws (5) and remove the wheel fairing panel as shown in Figure 32-9.
3. Remove the main wheel fairing screws (2) and washers (2) remove the main wheel fairing upward clear of the wheel assembly as shown in Figure 32-9.

This completes the main wheel fairing removal procedure.

NOSE WHEEL FAIRING REMOVAL

Perform this procedure to remove the nose wheel fairings on an aircraft so equipped.

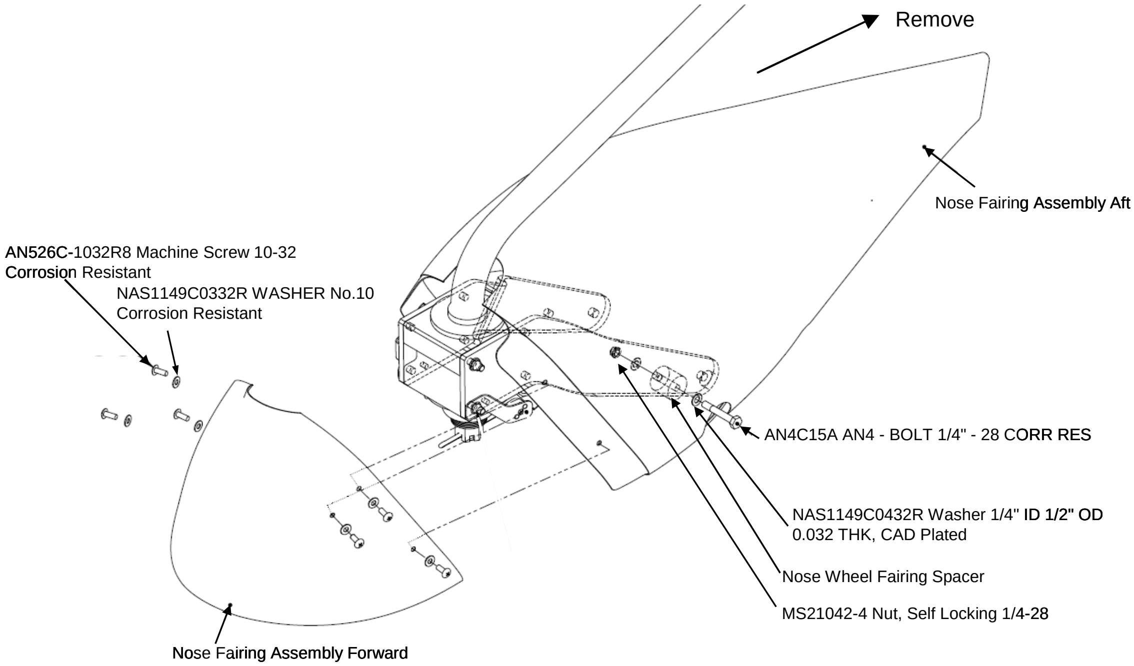


Figure 32-10 Nose Wheel Fairing Installation

1. Position aircraft on a level surface and set the parking brake
2. Remove the nose wheel fairing assembly forward screws (6) and washers (6). Remove the nose fairing assembly forward as shown in Figure 32-10.
3. Remove the nose wheel fairing assembly aft bolts (2), spacers (2), and washers (4). Remove the fairing aft clear of the nose wheel assembly as shown in Figure 32-10.

This completes the nose wheel fairing removal procedure.

MAIN WHEEL FAIRING INSTALLATION

Perform this procedure to install the main wheel fairings on an aircraft so equipped. Procedure is applicable to both the port and starboard main wheel fairings.

1. Position the aircraft on a level surface and engage the parking brake.
2. Inflate main wheel tires to 50 +2/-0 PSI.

NOTE

Outboard fairing axle fasteners must have high strength Loctite Number 270 on threads and safety-wire all AN502-10-8 machine screws.

3. Slide the main wheel fairing over the wheel assembly and install screws (2) and washers (2) as shown in Figure 32-9. Do not torque the screws at this time.
4. Install main wheel fairing panel with screws (5) and washers (5) as shown in Figure 32-9. Do not torque screws at this time.
5. Verify wheel fairing ground clearance is a minimum of 7 inches as shown in Figure 32-11. Small adjustments can be made prior to application of final torque.

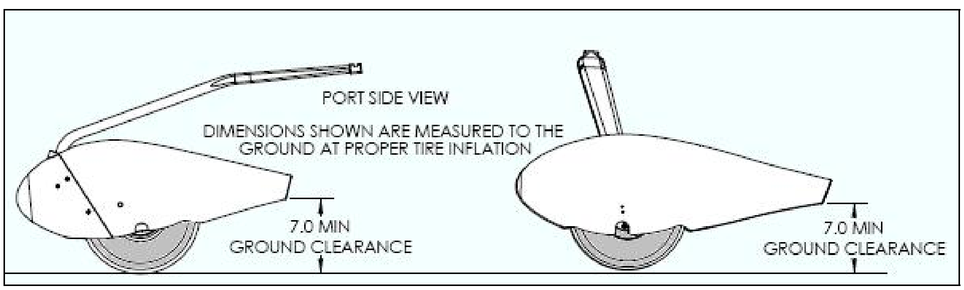


Figure 32-11 Ground Clearance

6. Torque the fasteners in accordance with Liberty Maintenance Manual Chapter 20 – *Standard Practices Airframe*.
7. Release the parking brake.
8. Move the aircraft forward through a full tire rotation and verify the tires do not scuff against the fairing.

This completes the main wheel fairing installation procedure.

NOSE WHEEL FAIRING INSTALLATION

Perform this procedure to install the nose wheel fairings on an aircraft so equipped.

1. Position the aircraft on a level surface and engage the parking brake.
2. Inflate nose wheel tire to 50 +2/-0 PSI.
3. Install the fairing aft nose wheel assembly as shown in Figure 32-10. Install the nose wheel fairing assembly aft bolts (2), spacers (2), and washers (4). Do not torque hardware at this time.
4. Install the nose fairing assembly forward as shown in Figure 32-10. Install the nose wheel fairing assembly forward screws (6) and washers (6). Do not torque hardware at this time.

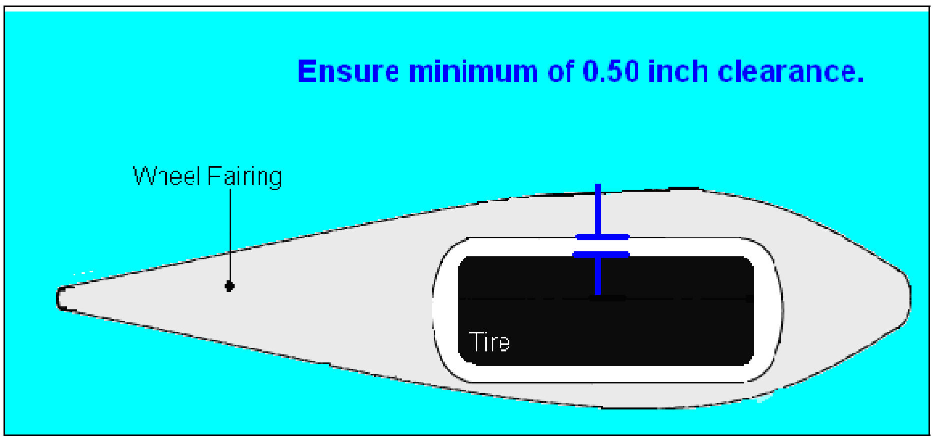


Figure 32-12 Nose Fairing Tire Clearance

5. Check the nose wheel tire to fairing clearance is a minimum 0.50 inches.
6. Verify nose wheel fairing ground clearance is a minimum of 7 inches as shown in Figure 32-13. Small adjustments can be made prior to application of final torque.

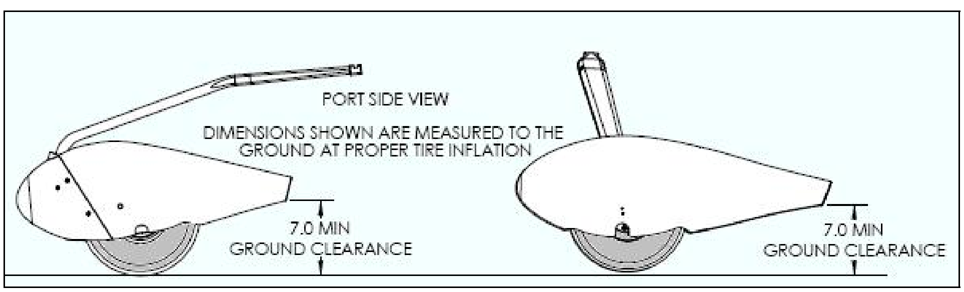


Figure 32-13 Ground Clearance

7. Torque the fasteners in accordance with Liberty Maintenance Manual Chapter 20 – *Standard Practices Airframe*.
8. Release the parking brake.

9. Rotate the nose wheel through its entire turning range of motion. Verify the nose wheel fairing and installation is free of binding, chaffing, or other interference.
10. Roll the aircraft forward through a full tire rotation and verify the tire does not scuff on the fairing.

This completes the nose wheel fairing installation procedure.

WHEEL FAIRING OPERATIONAL CHECK AND INSPECTION

This procedure performs wheel fairing operational checks and inspection.

1. Inspect fairings for cracks or deformations.
2. Inspect fairings for proper installation alignment as shown in Figure 32-14 and Figure 32-15.

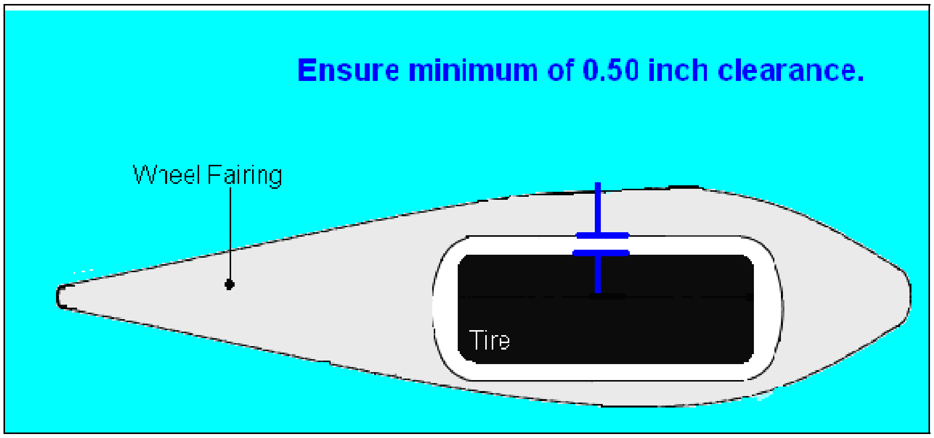


Figure 32-14 Wheel Fairing Clearances

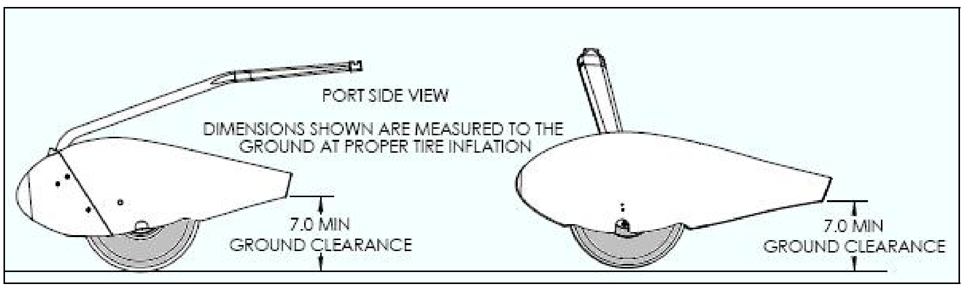


Figure 32-15 Wheel Fairing Clearance

3. Inspect main wheel fairing fasteners. Outboard fasteners must have high strength Loctite Number 270 on threads and safety-wire all AN502-10-8 machine screws.

CAUTION

If parts are not thread-locked and wire-locked as indicated, vibration may occur in flight.

4. Check wheel fairings are rigidity installed.

This completes the wheel fairing operational check and inspection procedure.

Section 40-04 Wheel Fairing Troubleshooting Guide

Complaint	Possible Cause	Remedy
Cracked fairing	Hard Landing	Perform a hard landing inspection in accordance with Liberty Maintenance Manual Chapter 05 Replace
	Ground handling	Replace
Fairing out of alignment	Ground Handling	Align in accordance with Liberty Maintenance Manual Chapter 32.
	Hard Landing	Perform a hard landing inspection in accordance with Liberty Maintenance Manual Chapter 05 Align in accordance with Liberty Maintenance Manual Chapter 32.
	Spacer fault	Replace fairing spacer(s)
	Loose fairing mounting screws	Adjust fairing alignment in accordance with Liberty Maintenance Manual Chapter 32.
Loose hardware	Improper securing of fasteners	Secure fasteners in accordance with Liberty Maintenance Manual Chapter 32.

Section 40-05 Wheels

Aircraft wheels fitted to the Liberty XL-2 are designed and qualification tested in accordance with TSO C26 for the tire type and size matching the aircraft requirements. Operating the wheel assembly with unapproved tires, improper inflation pressures or subjected to loads in excess of design is a violation of the wheel certification basis and is prohibited.

Wheels are made from aluminum castings, magnesium castings, or aluminum forgings. The wheel as shown in Figure 32-16 is of the divided type, incorporating inner wheel half (10) and outer wheel half (8), Ref. Figure 3, which are fastened together with tie bolts (7), washers (6), and nuts (5). An o-ring (9) fitted between the two wheel halves provides the air seal for wheels designed to operate with tubeless tires. The wheel rotates on two tapered roller bearings (4) which seat in bearing cups, shrink fitted into the hubs. Grease seals (3) provide protection and lubricant retention for the bearings. Hubcaps, when used on the main wheels, are secured to the outboard wheel half by a snap ring (1) or three attachment screws. Full wheel covers are fastened by three attachment screws. Main wheels differ from the nose wheel in that they are fitted with brake disk assemblies on one wheel half.

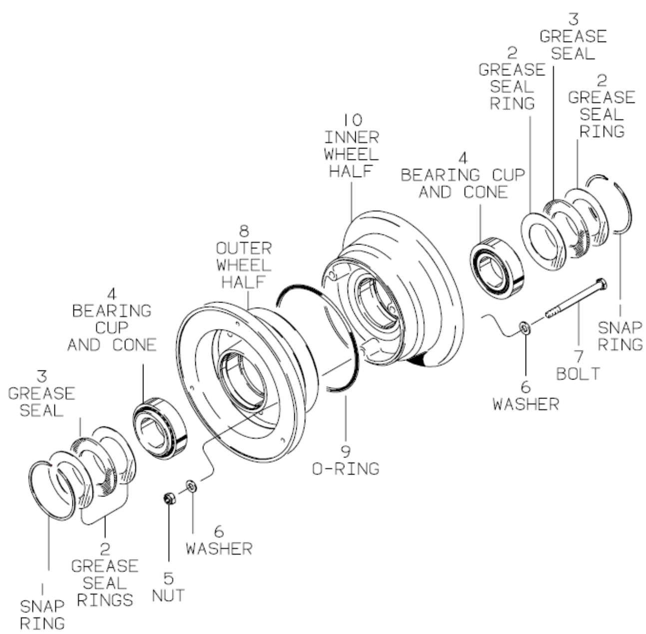


Figure 32-16 Wheel Assembly

Section 40-06 Periodic Maintenance

Periodic Wheel maintenance entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 40-07 Wheel Procedures

This section contains the procedure to remove, install checkout and inspect the main wheels and the nose wheel.

NOSE WHEEL REMOVAL

Perform this procedure to remove the nose wheel.

1. Chock two wheels that will not be removed. Do not apply parking brake if removing/replacing a main wheel.
2. Jack airplane to lift wheel to be removed just clear of floor, see Chapter 07 – *Lifting and Jacking*.
3. Remove securing bolts and screws to remove wheel fairing, if installed.
4. Deflate tire slowly and check that it is fully deflated before removing tire valve core.

NOTE

Check that wheel halves will not separate when axle nut is removed, if this happens-then wheel bolts are defective and wheel assembly must be replaced.

5. Withdraw 5/16” axle bolt from nose gear caster to remove wheel.
6. To remove bearings, remove two jam nuts from nose wheel axle. Note orientation of jam nuts and spacers in Figure 32-17.

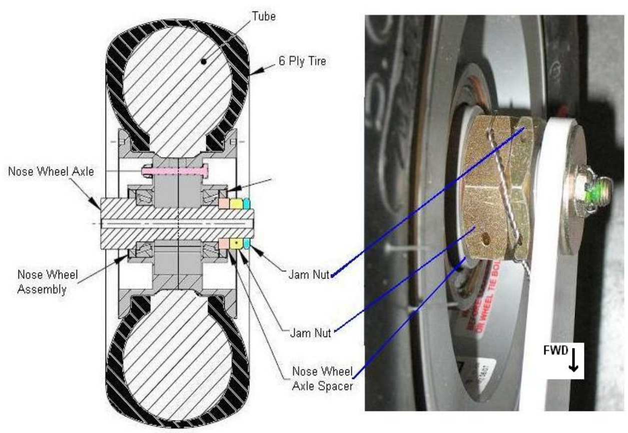


Figure 32-17 Nose Jam Nut Orientation

This completes the nose wheel removal procedure.

NOSE WHEEL INSTALLATION

Perform this procedure to install the nose wheel.

1. Check that wheel bearings are greased and installed.
2. Install nose wheel axle and spacer; torque jam nut (NAS509-17) to 25 in-lbs to seat bearing.
3. Back jam nut off to 0 in-lbs; re-torque to 15-25 in-lbs.
4. Torque jam nut (NAS1423-16) to 60 ft-lbs while securely holding inner jam nut (NAS509-17) in place with open-end wrench. Take care to install and torque jam nuts in proper sequence. Bearings and seals are supplied with nose wheel assembly.
5. Safety wire jam nut and inflate tire to 50 +2 / -0 psi.
6. Install nose wheel assembly into caster assembly with 5/16" axle bolt washer, castellated nut and split pin. Torque to the lowest value of torque as shown in Chapter 20 – *Standard Practices*. Continue to torque to nearest castellated slot for split pin. Do not exceed the highest value of torque as shown in Chapter 20 – *Standard Practices*. Install a new split pin.
7. Reinstall wheel fairing if present.

Figure 32-18 Nose Wheel

This completes the nose wheel installation procedure.

MAIN WHEEL REMOVAL

Perform this procedure to remove the main wheel

NOTE

It is unnecessary to disconnect medium pressure hose to caliper. Do not apply parking brake or operate any brake system when removing or installing a main wheel.

1. If fitted, remove and retain main wheel fairing. If no main wheel fairing, remove and retain hub cap and 3 associated screws.
2. Remove safety wire, and remove bolts securing brake pad to caliper; separate and remove brake caliper.
3. Remove and discard split pin from castellated axle nut; remove castellated axle nut.
4. Remove wheel and bearing spacers from axle.

WARNING

CHECK THE TIRE THAT IT IS FULLY DEFLATED TO PREVENT PERSONAL INJURY OR DEATH FROM SEPARATING WHEEL HALVES UNDER PRESSURE.

5. Deflate tire slowly and check the tire that it is fully deflated; remove tire valve core.

NOTE

Check that wheel halves will not separate. If wheel halves will separate, wheel bolts are defective and wheel assembly must be replaced.

This completes the main wheel removal procedure.

MAIN WHEEL INSTALLATION

Perform this procedure to install the main wheel.

1. Check the wheel that it has been properly reassembled after service. Inflate tire to 50 +2 / -0 psi (all wheels).

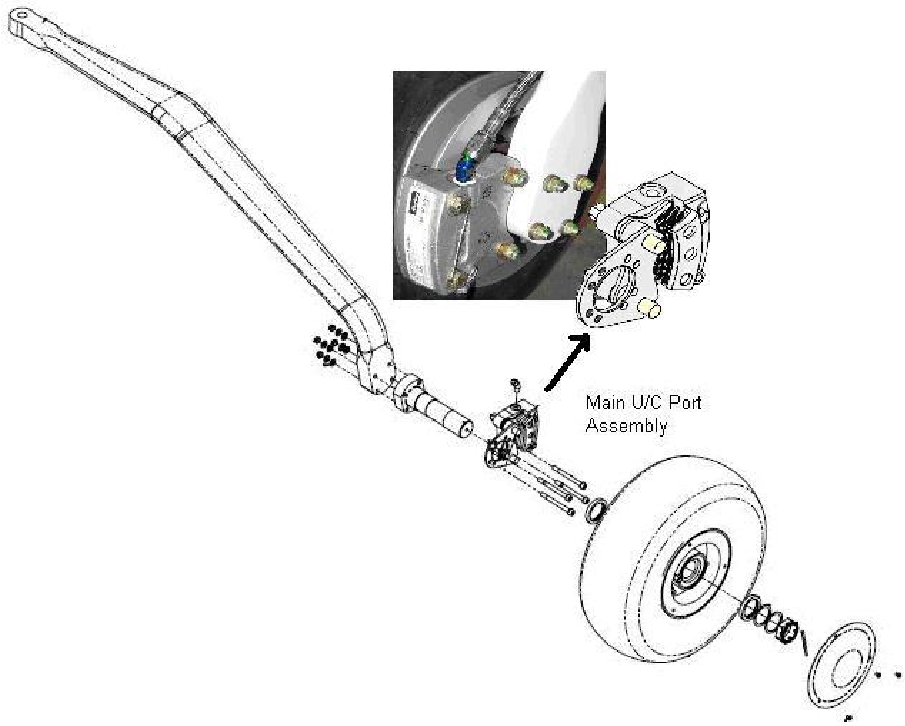


Figure 32-19 Main Landing Gear Wheel

1. Spray dry film lubricant on main gear axle and axle spacers.
2. Saturate felt seals on main wheel assembly with SHC Mobil Aviation grease (or equivalent).
3. Slide one axle spacer onto axle.
4. Slide main wheel assembly onto axle.
5. Slide second axle spacer onto axle butting it up against main wheel assembly.
6. Install two Seastrom washers and main axle nut.
7. Remove floating brake pad unit from main body of caliper. Spray brake caliper slide pins with dry film lubricant. Re-assemble floating brake pad to main caliper body and slide assembly into caliper plate on main undercarriage. Locate brake pad on back side of disc and refit 2 caliper bolts. Torque caliper bolts 75-80 in-lbs. and install safety wire.
8. Install safety wire onto brake pad mounting bolts.

9. Rotate wheel/tire while tightening axle nut to 150 – 200 in-lb to seat bearing. Back off axle nut to zero torque. Tighten axle nut to 30 – 40 in-lb while rotating wheel/tire. Rotate axle nut (clockwise or counterclockwise) to nearest slot and split pin hole.

NOTE

Wheel must rotate freely without perceptible play.

10. Install split pin and bend around nut to avoid contact with hub cap if fitted.
11. Reinstall hubcap or wheel fairing as applicable.

This completes the main wheel installation procedure.

WHEEL OPERATIONAL CHECK AND INSPECTION

This procedure performs main and nose wheel operational and checks and inspections.

1. Position aircraft in a level location and chock wheels not undergoing operation check or inspection.
2. Check tires for condition, wear, proper inflation (50 psi -0/+2 psi).
3. Clean wheels.
4. Clean and grease wheel bearings.
5. Check wheel bearings for play, corrosion, smooth running.
6. Check Main and Nose wheel rims for cracks. If cracks are found, replace the wheel rims.

CAUTION

Do not attempt to weld or repair cracks in wheel halves. If cracks are found in the wheel or wheel half replacement is mandatory.

This completes the main and nose wheel operational check and inspection.

Section 40-08 Wheel Troubleshooting Guide

Complaint	Possible Cause	Remedy
Cracked or distorted wheel or wheel half	Hard landing	Perform a hard landing inspection in accordance with Liberty Maintenance Manual Chapter 05 Replace wheel
	Hitting rocks or other hard objects during landing or takeoff	Inspect wheel using Zyglo to determine condition Replace wheel
	Use of sharp object to break tire bead.	Replace wheel
	Landing with flat tire	Replace wheel
	Landing in crabbing position in crosswind causing excess side force.	Replace wheel
	Normal fatigue failure when used beyond expected wheel life	Replace wheel
Tire wear	Service life wear	Replace if worn to tire manufacturers tread limit
	Main landing gear alignment	Inspect landing gear in accordance with Liberty Maintenance Manual Chapter 32. Replace tire if worn to tire manufacturers tread limit.
	Nose wheel shimmy	Inspect nose wheel assembly in accordance with Liberty Maintenance manual Chapter 32. Replace tire if worn below tire manufacturers tread limit
	Landing in crabbing position in crosswind causing tire cross tread wear	Replace tire if worn below tire manufacturers tread limit
Damaged wheel bearing	Foreign matter in bearing	Replace bearing and repack with grease Replace bearing seals
	Incorrect axle nut torque	Replace bearing and repack with grease Install wheel on landing gear with correct torque in accordance with Liberty Maintenance Manual Chapter 32

Complaint	Possible Cause	Remedy
	Lack of bearing grease	Replace bearing and repack with grease.
Damaged bearing cone	Misalignment of bearings	Replace bearing cone being sure it is properly seated in bearing bore
	Axle nut torque improper	Replace bearing cone being sure it is properly seated in bearing bore Replace and torque axle nut to correct torque in accordance with Liberty Maintenance Manual Chapter 32 –*Landing Gear*
	Foreign matter in bearing grease	Replace bearing cone being sure it is properly seated in bearing bore Check grease seals for damage
	Lack or bearing grease	Replace bearings and repack with grease
Worn or damaged grease seals	Normal wear or improper installation	Replace grease seals

Section 40-09 Toe Brake System

The airplane uses independent disc brakes on the port and starboard main landing gear for both deceleration and low-speed (taxi) steering.

In a toe brake installation, a master cylinder is mounted on each of the four rudder legs and actuated by a toe pad at the bottom of each leg. A pressure hose joins the pilot and co-pilot port side rudder leg/master cylinders (Master Cylinders 2, 4), through the parking brake to port brake caliper for left hand braking. A separate pressure hose joins the starboard side pilot and co-pilot rudder leg/master cylinders (Master Cylinders 1, 3), through the parking brake to the starboard caliper for right hand braking. For straight line braking, apply equal pressure to left and right-hand toe brake, (either pilot or co-pilot). With left and right-hand toe brake pressed, the parking brake may be applied, or apply parking brake and press the left and right-hand toe brake set together.

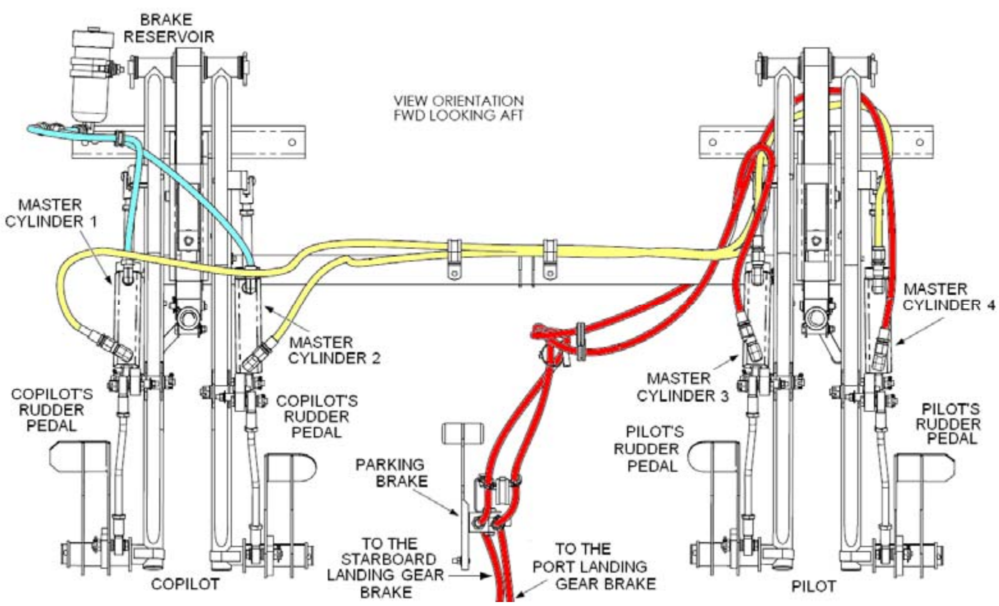


Figure 32-20 Toe Brake System

Figure 32-21 Toe Brake System Parking Brake Lever

The Parking Brake Lever shown Figure 32-21 is located to the right of the throttle lever and operates a simple hydraulic valve. When open, hydraulic fluid is free to flow in both directions. When closed hydraulic fluid can only flow to the caliper under pressure produced by actuating the brake toe pad. To apply the parking brake depress both right and left toe brake pads and move the parking brake lever to the ON position then release the pressure on the toe brakes. To release the parking brake, move the parking brake lever to the OFF.

Section 40-10 Toe Brake Parking Brake Valve

The airplane can come with one of two different valve assemblies (valve, adapters, mounting bracket, and hardware) for the parking brake. These assemblies are functionally interchangeable. However, do not replace the later valve assembly with the earlier assembly. The earlier valve is no longer available. The mounting bracket for the valve is also different. When replacing the original valve with a replacement valve, the mounting bracket must also change. Figure 32-22 shows the difference between the two valves, adapters, associated brackets, and hardware. Figure 32-23 shows the differences between the two mounting brackets.

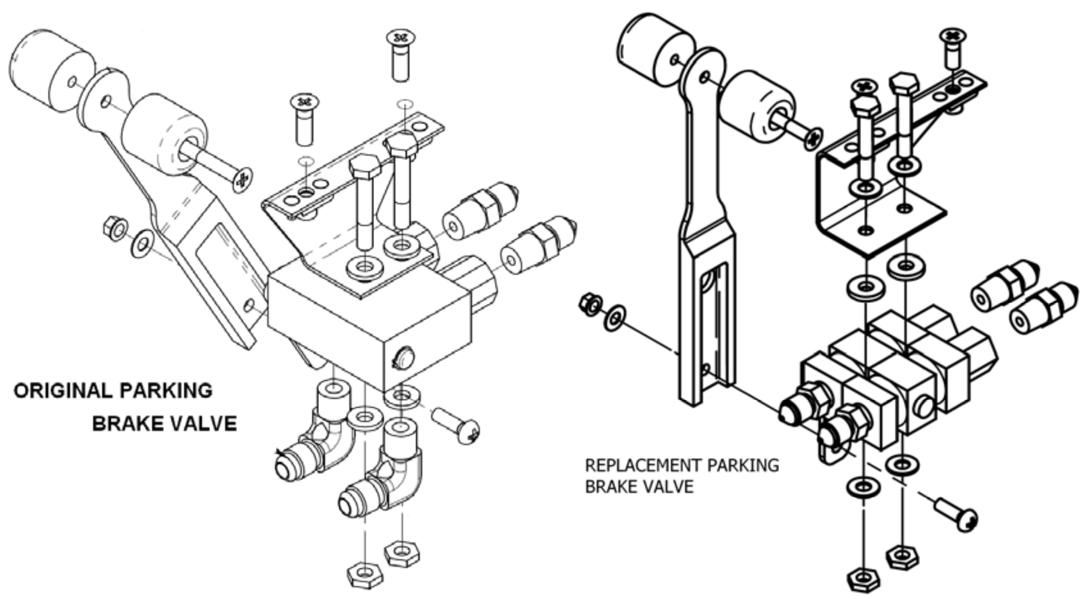


Figure 32-22 Parking Brake Valve

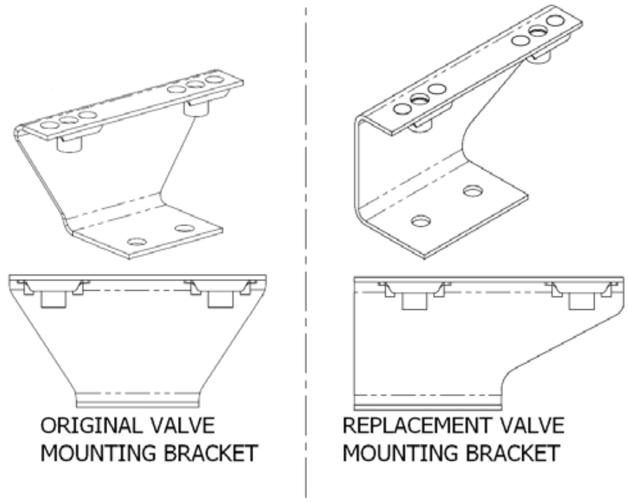


Figure 32-23 Differences Between the Original and Replacement Mounting Bracket

Section 40-11 Periodic Maintenance

Periodic Toe Brake System maintenance entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 40-12 Toe Brake Procedures

The following procedures are for removal and installation of the toe brake master cylinders.

CAUTION

Four toe brake master cylinders are inside cockpit, below instrument console. Care must be taken to reduce spillage of brake fluid when disconnecting brake line.

For the low-pressure brake line (translucent), it is permissible to disconnect and re-assemble compression nuts at “TEE Fitting” and three “Elbow Fittings”, but separation in part or totally of plastic compression sleeve from tube requires tube replacement and new plastic compression sleeve part number 260P04.

Any suspect low pressure clear brake line (Paraflex NN-4-040) must be replaced, length for length.

NOTE

Minor spillage of brake fluid is unavoidable when master cylinders are removed. Since both master cylinders are supplied by a reservoir, it is recommended that this reservoir be emptied (siphoned out or drained) when any master cylinder is removed or replaced. Have caps, plugs and absorbent material ready to contain spillage. Do not operate any toe brake lever or parking brake lever while any part of brake system is disturbed.

TOE BRAKE MASTER CYLINDER REMOVAL

Perform this procedure to remove the master cylinder mounted to the rudder pedals. The reservoir is located engine bay, starboard under upper engine cowl aft of firewall. Access to four master cylinders located on each of four rudder pedal weldments is via cockpit:

NOTE

Minor spillage of brake fluid is unavoidable when master cylinders are removed. Since both master cylinders are supplied by a reservoir, it is recommended that this reservoir be emptied (siphoned out or drained) when any master cylinder is removed or replaced. Have caps, plugs and absorbent material ready to contain spillage. Do not operate any toe brake lever or parking brake lever while any part of brake system is disturbed.

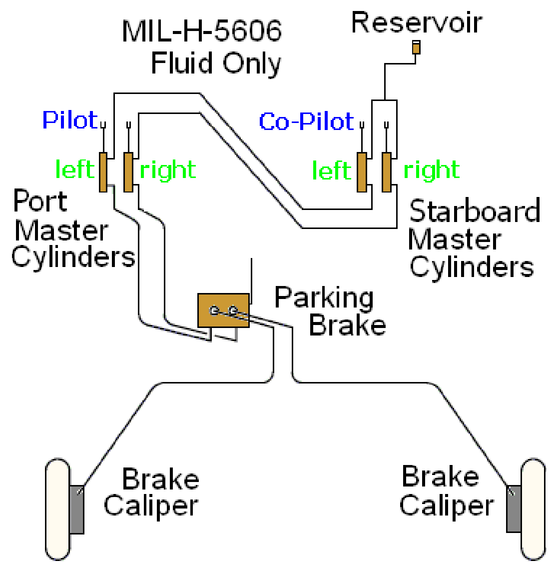


Figure 32-24 Schematic Toe Brake System

1. Chock all wheels. For a starboard master cylinder, drain reservoir and disconnect clear brake line at master cylinder, then the brake line hose that leads to the port toe brake system. Cap the end of the lines removed.

2. For a port master cylinder, disconnect the upper brake line from the master cylinder. Then disconnect the lower brake line that leads to the parking brake valve. Cap the end of the lines removed.

3. Remove split pin from self-locking castellated nuts and discard both nuts and split pins securing upper and lower bolts of master cylinder to rudder leg weldment and bellcrank.

NOTE

Note the clocking angle of the elbow fittings from the master cylinder removed. See Figure 32-25 for the correct clocking angles for the brake line fittings.

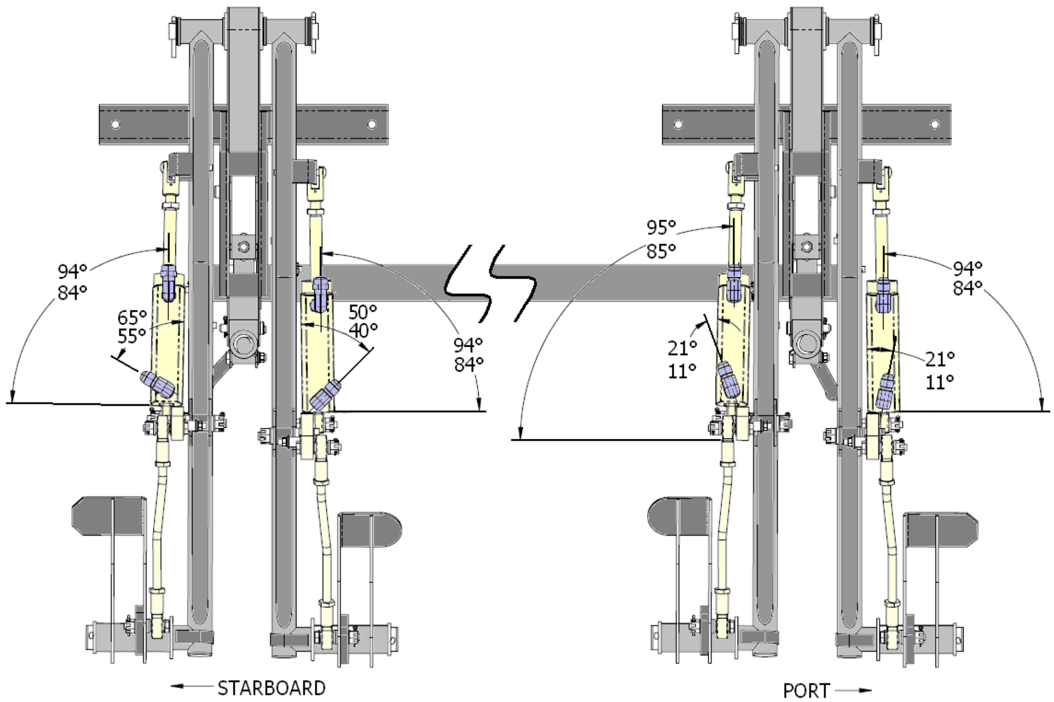


Figure 32-25 Clocking of the Brake Line Fittings

4. Remove and discard master cylinder and associated elbow fittings.

This completes the toe brake master cylinder removal procedure.

TOE BRAKE MASTER CYLINDER INSTALLATION

Perform this procedure to install the master cylinder for the toe brakes.

NOTE

Vendor for brake master cylinder P/N 10-55A is Parker Hannifin Corp., Ravenna, OH. During assembly, check that mounting bolts are normal to longitudinal centerline of brake master cylinder; check that brake lines do not cause side loading or interference with master cylinder assembly.

1. Apply adequate PTFE tape or Loctite® 545 to pipe thread and pre-fit 90º elbows flared or compression adapters as appropriate.

NOTE

See Figure 32-25 for the correct clocking angles for the brake line fittings.

2. Adjust clevis of replacement master cylinder to match dimension between center of lug mounting hole and clevis mounting hole. This dimension is 7.88” ± 0.02”, lock jam nut to clevis. See Figure 32-26 for details on adjusting the replacement master cylinder.

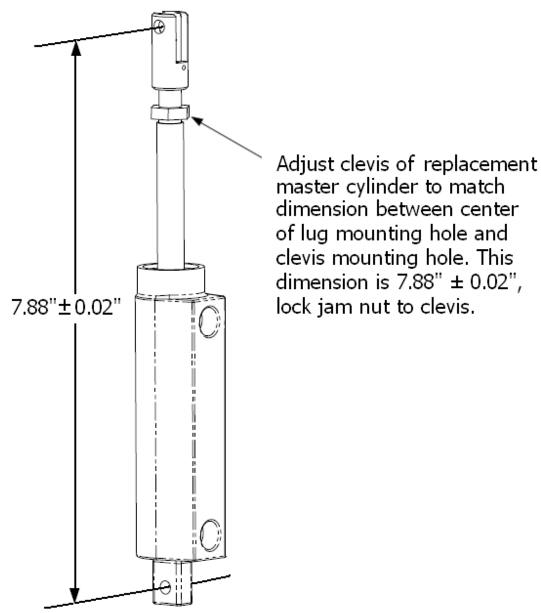


Figure 32-26 Master Cylinder Clevis Adjustment

CAUTION

Fully engage self-locking castellated nut 10-32, then back off half turn to nearest castellated slot. Check the bolt (AN3-11, 10-32) that it freely spins by hand so as not to hinder movement of bell crank.

3. Install adjusted master cylinder with original bolts (AN3-11, 10-32), washers, new self-locking castellated nuts and fit new split pins, 2 places.
4. Master cylinders Port brake system, reconnect medium pressure hoses from starboard system and from parking brake valve, apply adequate PTFE tape or Loctite® 545 to pipe thread.
5. Master cylinders starboard system, reconnect medium pressure hose from Port system and connect low pressure clear brake line from hydraulic reservoir, apply adequate PTFE tape or Loctite® 545 to pipe thread.
6. Refill brake fluid reservoir, allowing room for bleeding procedure. Bleed brake system per the Brake Bleeding Procedure on page 74 of this chapter and perform a functional check.
7. Replace wheel fairing if applicable.

This completes the toe brake master cylinder installation procedure.

TOE BRAKE OVER-CENTER STOP REMOVAL AND INSTALLATION

Perform this procedure to remove and then install the toe brake Over-Center stop. The item numbers called out are shown in Figure 32-28.

1. Remove (item #1) P/N: MS24694S59 screw 10-32 from bellcrank, remove two washers (item #2) P/N: NAS1149F0332P washers, (item #3) P/N: 135A-30-595 spacer and (item 4) P/N: MS21042-3 self-locking nut which secures over-center stop to rudder leg, see Figure 32-28.
2. For replacement install P/N: MS24694S59 screw and apply high strength Loctite® No. 270 on threads. Install 2 P/N: NAS1149F0332P washers, P/N: 135A-30-595 spacer and P/N: MS21042-3 self locking nut.

Figure 32-27 Toe Brake Over-Center Stop

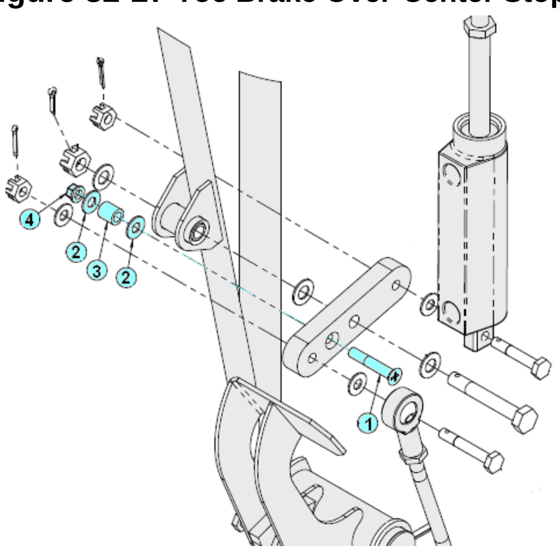


Figure 32-28 Toe Brake Over-Center Stop Hardware

This completes the toe brake over center stop removal procedure.

PARKING BRAKE VALVE REMOVAL

Perform this procedure to remove the parking brake valve used with the toe brake system. There are two different valves used for the parking brake. The valves are functionally interchangeable; however, do not replace a later valve with the earlier valve. The procedure in this section and the one to follow are for the removal and installation of the parking brake valve in airplanes equipped with toe brakes.

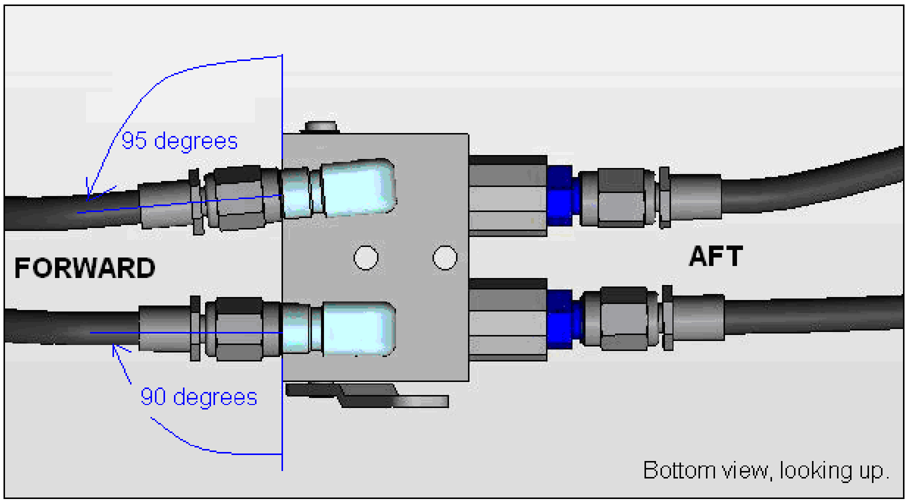


Figure 32-29 Clocking of Parking Brake Valve Fittings (Earlier Brake Valve ONLY)

NOTE

Minor spillage of brake fluid is unavoidable when removing parking brake valve, it is recommended that reservoir is emptied (siphoned or drained) when parking brake valve is removed or replaced. Have caps, plugs, and absorbent material ready to contain spillage. Do not operate any toe brake pedal or parking brake lever with any part of brake system is disturbed.

1. Chock aircraft wheels.
2. Remove fuselage belly panel. For the instructions to remove the belly panel, see Chapter 53 – *Fuselage*.
3. Disconnect all brake lines to parking brake valve and cap/plug as appropriate.
4. Dismantle and retain, except for lock nut, 8-32 UNC washer head bolt assembly securing custom parking brake lever directly to lever on parking brake.
5. Dismantle and retain, except for lock nuts, two 10-32 UNF bolt assemblies securing parking brake valve to “C” Bracket.
6. Remove parking brake valve and discard along with two straight and two 90º elbow adapters.

This completes the Parking Brake Valve Removal procedure.

PARKING BRAKE VALVE INSTALLATION

Perform this procedure to install the parking brake valve used with the toe brake system.

NOTE

Vender Matco. Device is not a serviceable item. Check disturbed brake lines to make sure they are recovered back to original alignment.

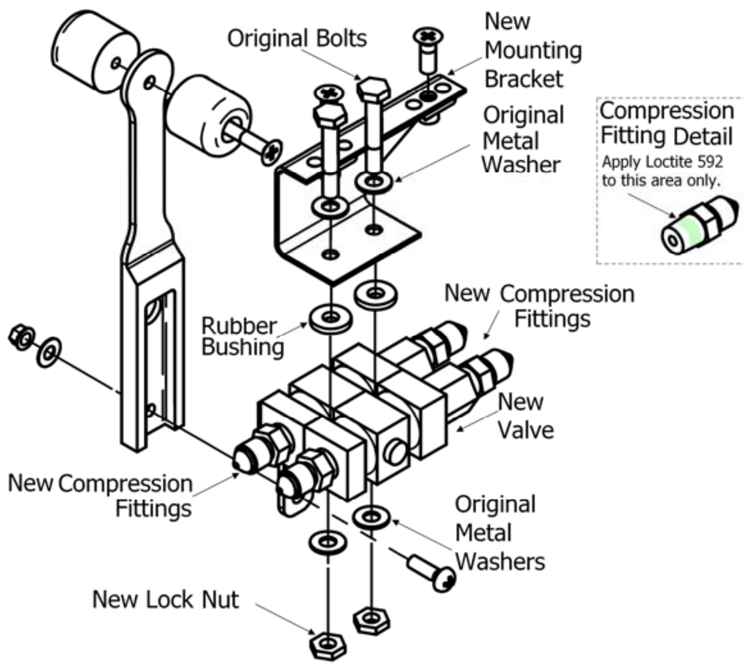


Figure 32-30 Parking Brake Valve Parts Assembly

1. Build up parking brake unit with four straight adapters; apply adequate Loctite® 545 to pipe thread, reference detail above see Figure 32-23. Do not allow the Loctite to go onto the last two threads of the adapter.
2. Install parking brake assembly using original bolt, metal washers and rubber bushings with new lock nut, 2 places. For proper part assembly order, see Figure 32-30.
3. Install custom parking brake lever to lever on parking brake using original bolt washer and new lock nut.
4. If installing the newer valve, install two AN816-3D pipe to tube straight adapters.
5. Reconnect all brake lines to parking brake valve.
6. Bleed and perform functional check per the Brake Bleeding Procedure on page 74 of this chapter.

This completes the parking brake valve installation procedure.

TOE BRAKE OPERATIONAL CHECK AND INSPECTION

This procedure performs toe brake operational check and inspection.

1. Remove the upper cowl in accordance with Liberty Maintenance Manual Chapter 71-*Powerplant*.
2. Remove the belly panel in accordance with Liberty Maintenance Manual Chapter 53-*Fuselage*.
3. Fill the brake reservoir with MIL-H-5606 hydraulic fluid to the indicated full line.

NOTE

The following step is best accomplished with assistance. One person applies pressure on the brake system while another inspects the pressurized brake system for leaks.

4. Apply brake pressure by pressing on the toe brake pedals. Verify the brake pedals are firm and do not drift forward as pressure is applied to the pedals.
5. Verify brake calipers are fully engaged with both the port and starboard wheel brake disks.
6. With pressure still applied at the toe brake pedal, inspect brake lines from reservoir connections to the brake calipers for indication of leaks.
7. While holding pressure with the toe brake *pedal*s engage the parking brake lever as shown in Figure 32-21. Release pressure from the toe brakes once the parking brake is set.
8. Verify both the port and starboard brake caliper is fully engaged.
9. Inspect the parking brake valve and brake lines from the valve to the port and starboard brake calipers for leaks.
10. Verify pressure holds on the system while the parking brake is engaged.
11. Release the parking brake and verify the port and starboard calipers release.
12. Verify hydraulic fluid in the brake reservoir is still full and free of air bubbles.
13. Install the belly panel in accordance with Liberty Maintenance Manual Chapter 53-*Fuselage*.
14. Install the upper cowl in accordance with Liberty Maintenance Manual Chapter 71-*Powerplant*.

This completes the toe brake operational check and inspection procedure.

Section 40-13 Toe Brake Troubleshooting Guide

Complaint	Possible Cause	Remedy
One or both toe brakes fail to hold	Air in the brake system	Inspect system for leaking fitting and repair. Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Leak in the brake system fittings	Repair leak Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Brake pedal master cylinder fault	Replace master cylinder Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Brake pads worn to limits or contaminated	Replace brake pads Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Brake disc worn below limit	Replace disc
Parking brake fails to hold	Air in the brake system	Inspect system for leaking fitting and repair. Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Parking brake valve fault	Replace valve Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Leak in the parking brake system fittings	Repair leak Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Brake pads worn to limits	Replace brake pads Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.

Complaint	Possible Cause	Remedy
Brake reservoir air bubbles after system test	Air in the brake system	Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
Brake drag	Piston cocked in cylinder, resulting in overheating brake and/or excessive lining wear.	Remove and repair cylinder or piston, or replace brake.
	Foreign matter wedged in brakes.	Locate and remove
	Back pressure due to malfunction of master cylinder or parking brake system	Inspect and bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Water or ice in hydraulic system.	Thaw ice Inspect and bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Excessive bolt torque has caused back plate to crush cylinder, evidenced by depressions around bolt holes.	Replace cylinder
	Piston does not retract	Remove and inspect piston for damage Replace damaged piston Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Warped pressure plate	Replace or straighten to within 0.010 inch (0.254 mm)
	Corroded anchor bolts and/or torque plate bushings	Clean and lubricate Replace if damaged
	Cocked anchor bolts and/or torque plate bushings	Replace

Complaint	Possible Cause	Remedy
	Bent/cracked torque plate	Replace
	Restriction in hydraulic line	Isolate and remove restriction
	Lining not firmly seated flush against pressure/back plate	Debur rivet hole on surface adjacent to lining

Section 40-14 Finger Brake System

For Finger Brake installation: The brakes are actuated by two master cylinders installed in the chassis center section and controlled by finger levers in the center console, see Figure 32-32. The levers are positioned for easy one-hand operation. Parking Brakes are applied using a center lever, located between the two brake levers. When pulled simultaneously with the brake levers, a ratchet mechanism will hold the left and right brake levers. To release the parking brakes, the brake levers are subsequently pulled without pulling the center lever, a spring disengages it and releases the parking brake.

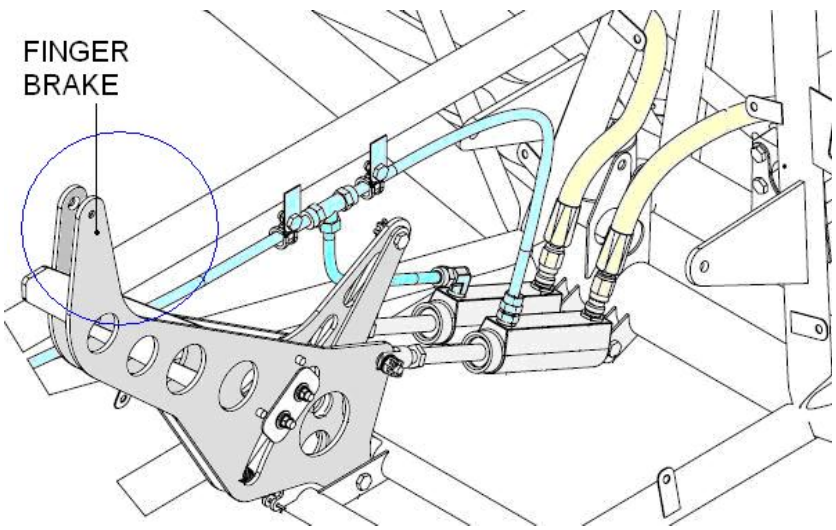


Figure 32-31 Finger Brake Location

The reservoir supplies both the Port and starboard brake system with MIL-H-5606 hydraulic fluid. It can be serviced by removing the upper engine cowl. It is located on the AFT starboard side behind the firewall.

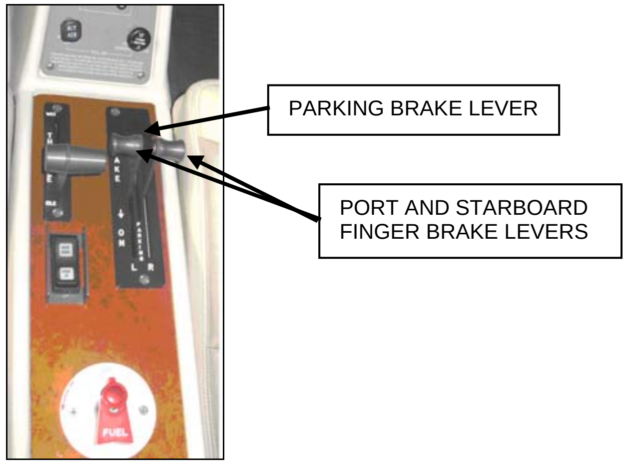


Figure 32-32 Finger Brake System Parking Brake Lever

Section 40-15 Periodic Maintenance

Periodic Finger Brake System maintenance entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 40-16 Finger Brake System Procedures

This section contains the procedures to remove, install, checkout and inspect the finger brake system and its components.

FINGER BRAKE MASTER CYLINDER REMOVAL

Perform this procedure to remove the master cylinder for the finger brakes.

CAUTION

For the low-pressure brake line (translucent), it is permissible to disconnect and re-assemble compression nuts at the “TEE Fitting” and the three “Elbow Fittings”. Separation in part or totally of plastic compression ferrule from tube requires tube replacement and new plastic compression ferrules P/N 260P04. Any suspect low pressure clear brake line (Paraflex NN-4-040) must be replaced, length for length.

NOTE

Minor spillage of brake fluid is unavoidable when master cylinders are removed. Since both master cylinders are supplied by a reservoir, it is recommended that this reservoir be emptied (siphoned out or drained) when either master cylinder is removed or replaced. Have caps, plugs and absorbent material ready to contain spillage. Do not operate finger brake lever or parking brake lever while any part of brake system is disturbed.

1. Remove fuselage belly fairing, cover-plate upper undercarriage legs, and wheel fairing if applicable.
2. Chock wheels not being serviced.
3. Disconnect Finger Brake medium pressure hose from master cylinder, cap or plug.

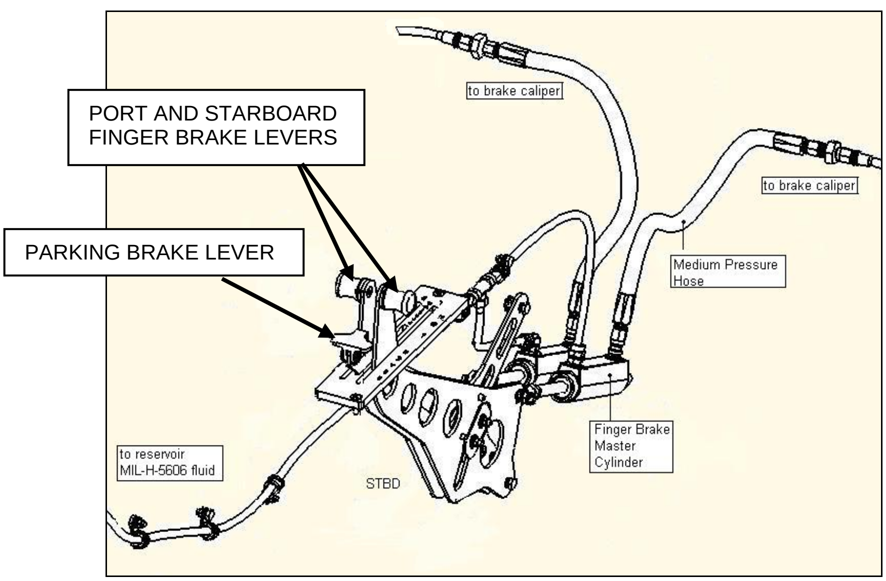


Figure 32-33 Finger Brake System

4. Disconnect line from reservoir to master cylinder and cap or plug.
5. Remove and discard split pin from castellated nuts securing front and rear bolts of master cylinder to cockpit operating lever and to fuselage center section space frame.
6. Remove and discard master cylinder, complete with 90º elbow or straight compression adapter and straight flared adapter.

This completes the finger brake master cylinder removal procedure.

FINGER BRAKE MASTER CYLINDER INSTALLATION

Perform this procedure to install the master cylinder fro the finger brakes.

NOTE

Parker Hannifin Corporation, Ravenna, OH, master cylinder P/N 10-55A must be in fully extended position to ensure correct operation. And all mounting bolts centers normal to longitudinal centerline of master cylinder. In addition, check that hoses and tubing do not cause side loading or interference with master cylinder.

1. Apply adequate PFTE tape or Loctite® 545 to pipe thread on 90º or straight compression fitting and Loctite® 545 to pipe thread on straight flared fitting.
2. Adjust clevis of replacement master cylinder such that dimensions between center of lug mounting hole and center clevis mounting hole is 7.850 inch ± .020 inch and install with original bolts and washers, and new castellated lock nuts, fit new split pins 2 places.
3. Reconnect brake line from brake reservoir, ensuring compression fitting correctly installed.
4. Reconnect medium pressure brake line.
5. Refill brake fluid reservoir, allowing room for bleeding procedure. Bleed brake system per the Brake Bleeding Procedure on page 74 of this chapter and perform a functional check.
6. Replace fuselage belly fairing, cover-plate, and wheel fairing if applicable.

This completes the finger brake master cylinder installation procedure.

FINGER BRAKE OPERATIONAL CHECK AND INSPECTION

This procedure performs toe brake operational check and inspection.

1. Remove the upper cowl in accordance with Liberty Maintenance Manual Chapter 71-*Powerplant*.
2. Remove the belly panel in accordance with Liberty Maintenance Manual Chapter 53-*Fuselage*.
3. Fill the brake reservoir with MIL-H-5606 hydraulic fluid to the indicated full line.

NOTE

The following step is best accomplished with assistance. One person applies pressure on the brake system while another inspects the pressurized brake system for leaks.

4. Apply brake pressure by pulling finger brake levers aft. Verify the brake levers are firm, even and do not drift backward (AFT) as pressure is applied to the levers.
5. Verify brake calipers are fully engaged with both the port and starboard wheel brake disks.
6. With pressure still applied at the finger brake levers, inspect brake lines from reservoir connections to the brake calipers for indication of leaks.
7. While holding pressure with the finger brake levers, engage the parking brake lever located just ahead (forward) of the finger brake levers as shown in Figure 32-32. Release the finger brake levers once the parking brake is set.
8. Verify both the port and starboard brake caliper is fully engaged.
9. Inspect the parking brake valve and brake lines from the valve to the port and starboard brake calipers for leaks.
10. Verify pressure holds on the system while the parking brake is engaged.
11. Release the parking brake and verify the port and starboard calipers release.
12. Verify hydraulic fluid in the brake reservoir is still full and free of air bubbles.
13. Install the belly panel in accordance with Liberty Maintenance Manual Chapter 53-*Fuselage*.
14. Install the upper cowl in accordance with Liberty Maintenance Manual Chapter 71-*Powerplant*.

This completes the toe brake operational check and inspection procedure

Section 40-17 Finger Brake Troubleshooting Guide

Complaint	Possible Cause	Remedy
One or both finger brakes fail to hold	Air in the brake system	Inspect system for leaking fitting and repair. Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Leak in the brake system fittings	Repair leak Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Brake master cylinder fault	Replace master cylinder Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Brake pads worn to limits or contaminated	Replace brake pads Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Brake disc worn below limit	Replace disc
Parking brake fails to hold	Air in the brake system	Inspect system for leaking fitting and repair. Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Master cylinder fault	Replace master cylinder Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Parking brake lever fault	Replace parking brake lever assembly.
	Leak in the parking brake system fittings	Repair leak Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.

Complaint	Possible Cause	Remedy
	Brake pads worn to limits	Replace brake pads Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
Brake reservoir air bubbles after system test	Air in the brake system	Bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
Brake drag	Piston cocked in cylinder, resulting in overheating brake and/or excessive lining wear.	Remove and repair cylinder or piston, or replace brake.
	Foreign matter wedged in brakes.	Locate and remove
	Back pressure due to malfunction of master cylinder or parking brake system	Inspect and bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Water or ice in hydraulic system.	Thaw ice Inspect and bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Excessive bolt torque has caused back plate to crush cylinder, evidenced by depressions around bolt holes.	Replace cylinder
	Piston does not retract	Remove and inspect piston for damage Replace damaged piston Inspect and bleed brake system in accordance Brake Bleeding Procedure on page 74 of this chapter.
	Warped pressure plate	Replace or straighten to within 0,010 inch (0.254 mm)
	Corroded anchor bolts and/or torque plate bushings	Clean and lubricate Replace if damaged

Complaint	Possible Cause	Remedy
	Cocked anchor bolts and/or torque plate bushings	Replace
	Bent/cracked torque plate	Replace
	Restriction in hydraulic line	Isolate and remove restriction
	Lining not firmly seated flush against pressure/back plate	Debur rivet hole on surface adjacent to lining

Section 40-18 Main Wheel Brake Caliper System

The following sections provides information on the main wheel brake caliper system.

Section 40-19 Periodic Maintenance

Periodic Main Wheel Brake Caliper System maintenance entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 40-20 Main Wheel Brake Caliper System Procedures

This section has the procedures to remove and install the brake caliper system.

Figure 32-34 Main Wheel Brake Caliper

Access to main wheel caliper requires removal of main wheel faring if fitted.

NOTE

For Toe Brake System: move parking brake lever to "ON." Do not operate toe brakes, there is no requirement to drain reservoir.

For Finger Brake System, do not apply parking brake or operate finger brakes. It is recommended to siphon or drain reservoir. Minor spillage of brake cylinder is unavoidable when removing or replacing the brake caliper.

Main Wheel Brake Caliper Removal

Perform this procedure to remove the main wheel caliper. Removal of the entire caliper assembly is performed on condition mandating replacement of the caliper assembly. For caliper pad replacement is done in accordance with Liberty Maintenance Manual Chapter 32-*Landing Gear* section titled Brake Caliper Pad/Liner Replacement.

NOTE

Vendor – Parker Hannifin Corp. Aircraft Wheel and Brake Division, Avon, Ohio 44011

1. Chock nose wheel and main wheel not subject to caliper replacement.
2. Release parking brake and verify calipers are not gripping wheel disk.
3. If so equipped, remove the affected main wheel faring in accordance with Liberty Maintenance Manual Chapter 32.
4. Disconnect medium pressure brake hose from 45º elbow adapter plug and cap as appropriate.
5. Cut and remove safety wire and remove tie bolts, securing back plate to main caliper body.
6. Carefully slide the brake cylinder out of the torque plate.

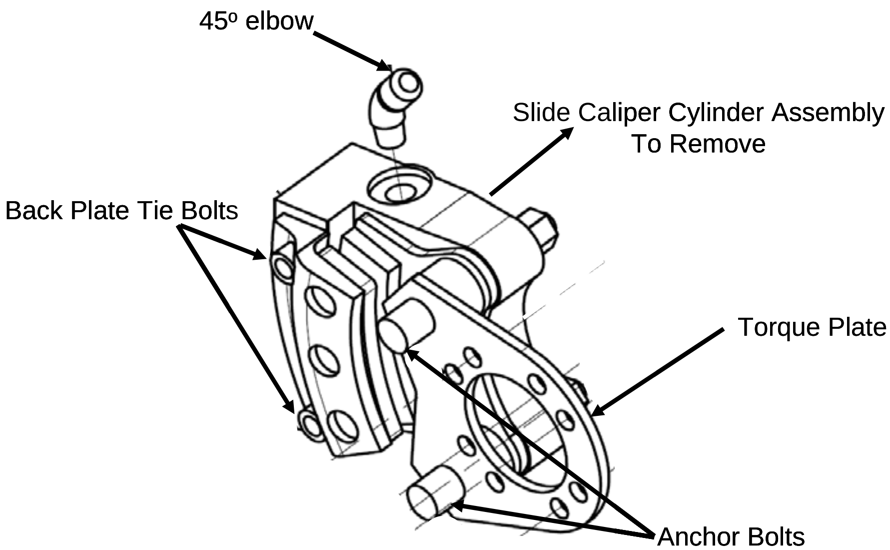


Figure 32-35 Caliper Assembly

7. Remove wheel in accordance with Liberty Maintenance Manual Chapter 32 – *Landing Gear*.

NOTE

The following step is not required if the torque plate is serviceable. Perform this step if a condition requiring replacement of the caliper torque plate exists.

8. Remove four (4) NAS6204-30 internally wrenching bolts, nuts and washers securing the axle and caliper torque plate to the main landing gear leg.

This completes the main wheel brake caliper removal procedure.

MAIN WHEEL BRAKE CALIPER INSTALLATION

Perform this procedure to install the main wheel caliper.

1. Chock nose wheel and main wheel not subject to caliper replacement.
2. Fit and clock a new 45º adapter to main body of caliper using PFTE tape or Loctite® 592 to pipe thread, see Figure 32-25.

NOTE

The following step is not required if the torque was not removed previously. Perform this step if a condition requiring replacement of the caliper torque plate exists.

3. Install the caliper torque plate and main leg axle assembly with four (4) NAS6204-30 internally wrenching bolts, nuts and washers as shown in Figure 32-36.

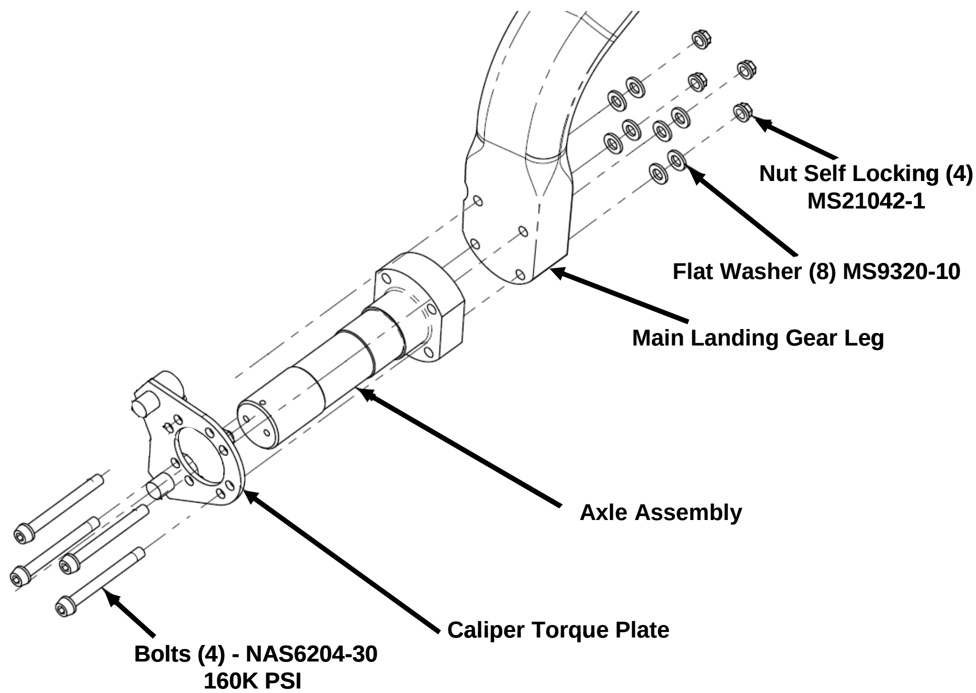


Figure 32-36 Caliper Torque Plate Installation

4. Install main wheel in conjunction with new caliper torque plate in accordance with Liberty Maintenance Manual Chapter 32-*Landing Gear* Set tire inflation to 50 + 2 / -0 psi all wheels.
5. Reconnect medium pressure brake hose line.

6. Install caliper back plate tie bolts and washers allowing removal of back plate. Torque tie bolts to 75-80 in-lb. Safety wire bolt heads using 0.032 safety wire MS20995 Type 203/204.
7. Bleed and perform operational check per Liberty Maintenance Manual Chapter 32-*Landing Gear*.
8. If so equipped, install the affected main wheel fairing in accordance with Liberty Maintenance Manual Chapter 32-*Landing Gear*.

This completes the main wheel brake caliper installation procedure.

BRAKE CALIPER PAD/LINER REPLACEMENT

Perform this procedure to replace the caliper brake pads. The procedure is applicable to both the port and starboard brake calipers.

NOTE

The brake caliper assembly manufacturer is Parker Hannifin Corp. Aircraft Wheel and Brake Division, Avon, Ohio 44011. Refer to this vendor for internal details and specifications.

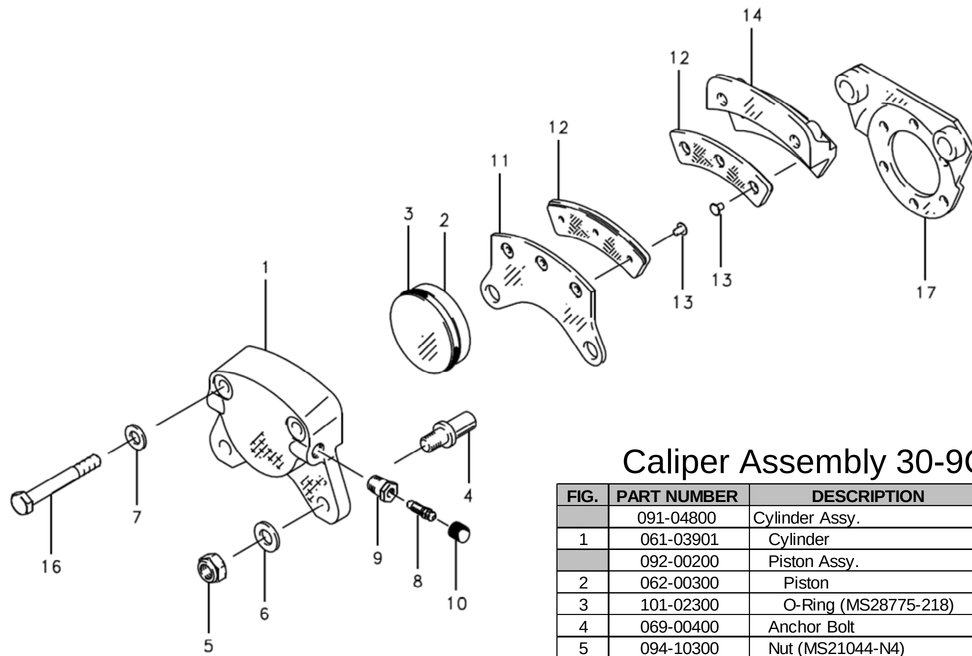
NOTE

It is not necessary to remove a main landing gear wheel from aircraft, or to disconnect medium pressure hose between landing gear leg and brake caliper, for brake pad liner replacement. Do not operate any toe brake pedal, finger brake lever or parking brake lever while brake calipers are dismantled.

1. Chock all wheels
2. Remove main landing gear wheel fairing in accordance with Liberty Maintenance Manual Chapter 32-*Landing Gear.*
3. Cut and remove safety wire and remove tie bolts, securing back plate to main caliper body.
4. Carefully slide the brake cylinder out of the torque plate.
5. Referring to Figure 32-37, inspect the caliper for the following:
   - Corrosion of any part of the assembly.
   - Bent anchor bolts.
   - Cracks around bolts.
   - Cracks around anchor bolt lugs and inlet fittings, and other visible damage.
   - Inspect the torque plate assemblies for corrosion around anchor bolt bushings and excessive wear in bushings.

CAUTION

Correct all discrepant conditions prior to proceeding with this procedure. Uncorrected conditions may result in poor brake performance or brake failure.



Caliper Assembly 30-9C

FIG.	PART NUMBER	DESCRIPTION	QTY
	091-04800	Cylinder Assy.	1
1	061-03901	Cylinder	1
	092-00200	Piston Assy.	1
2	062-00300	Piston	1
3	101-02300	O-Ring (MS28775-218)	1
4	069-00400	Anchor Bolt	2
5	094-10300	Nut (MS21044-N4)	2
6	095-10400	Washer (AN960-4-16)	2
7	095-10200	Washer (AN960-416L)	2
8	079-00300	Screw-Bleeder	1
9	081-00100	Seat-Bleeder	1
10	183-00100	Cap-Bleeder	1
	073-00300	Pressure Plate Assy.	1
11	063-00500	Pressure Plate	1
12	066-10600	Lining	1
13	105-00200	Rivet	3
	074-00200	Back Plate Assy.	1
14	064-00500	Back Plate	1
12	066-10600	Lining	1
13	105-00200	Rivet	3
16	103-12100	Bolt (AN4H15A)	2
17	075-00800	Torque Plate Assy.	1

Figure 32-37 Brake Caliper Subassembly Details

NOTE

The organic brake lining used in this caliper assembly is identified by its semi hard composition and rivets used to attach the lining to the pressure plate or back plate. The rivet holes are visible on the lining. Non-asbestos, lead free lining material is also being used as a replacement for the old style organic lining and is removed and installed in the same manner as the organic lining.

6. Old organic lining may be removed by using a small drift pin or carefully drilling out the rivets with a 1/8-inch diameter drill. Use care to prevent elongating the rivet holes. Deburr the surface adjacent to the lining to allow lining to set flush.
7. Clean pressure plate and back plate surfaces of dirt, grease, etc. before installing new linings.

8. Inspect pressure plate and back plate for excessive corrosion, visible damage, or excessive warping. The pressure plate should not be used if warped in excess of 0.010 inch (0.254 mm) flatness. Correct by using draw flattening or straightening technique shown in Figure 32-38.

NOTE

Excessive warping can result in brake drag. This drag will be most pronounced when a new disc and set of linings are installed.

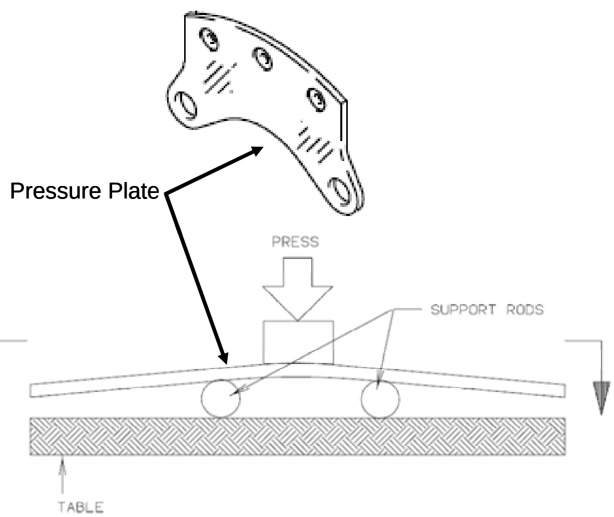
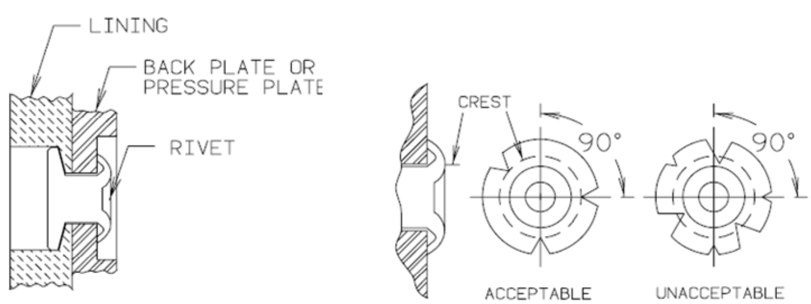


Figure 32-38 Pressure Plate Straightening

9. Align new factory authorized replacement lining segments on pressure plate/back plates and install rivets of corresponding part number, using Cleveland's Rivet Set, P/N 199-1, or appropriate riveting tools. Refer to Figure 32-39 for acceptable rivet set criteria.



- The split shall not occur inside the crest of the clenched surface.
- No more than two splits shall occur in a 90° area
- A total of no more than three splits shall be allowed.

Figure 32-39 Brake Lining Rivet Set Acceptance Criteria

10. Check to be sure lining is tight and movement free with no distortion of parts.
11. Carefully wipe dirt, grease, etc. from cylinder, pressure plate, and portions of piston extending beyond cylinder face, and push piston back into cylinder.
12. Slide pressure plate with new lining over anchor bolts and install brake caliper into torque plate. For equipment that is operated in extremely wet climates, lubricate the anchor bolt with Lubriplate. For equipment used in a non-amphibious environment, lubricate anchor bolt with a dry film lubricant (silicone spray). Torque caliper bolts 75-80 in-lbs. and install safety wire.

CAUTION

Do not use grease or oil to lubricate caliper parts. These materials will attract dirt and enhance the wear of the anchor pins.

13. Locate brake pad back plate on backside of disc and secure with 2 caliper bolts. Torque caliper bolts 75-80 in-lbs. and install safety wire.
14. Bleed brake system in accordance with Liberty Maintenance Manual Chapter 32-*Landing Gear*

NOTE

The following steps perform brake lining conditioning. All new brake lining installations require this conditioning procedure in order to assure maximum braking performance.

15. Taxi aircraft for 1500 feet with engine at 1700 rpm applying brake pedal force as needed to develop a 5-10 mph taxi speed.
16. Allow the brakes to cool for 10 to 15 minutes.
17. Apply brakes and check for restraint at high static throttle. If brakes hold, conditioning is complete.
18. If brakes cannot hold aircraft during static run up, allow brakes to completely cool, and repeat the previous three steps.

NOTE

This conditioning procedure will wear off high spots and generate sufficient heat to create a thin layer of glazed material at the lining friction surface. Normal brake usage should generate enough heat to maintain the glaze throughout the life of the lining.

Properly conditioned linings will provide many hours of maintenance free service. A visual inspection of the brake disc will indicate the lining condition. A smooth surface, one without grooves, indicates the linings are properly glazed. If the disc is rough (grooved), the linings must be conditioned again. The conditioning procedure should be performed whenever the rough disc condition is observed. Light use, such as in taxiing, will cause the glaze to be worn rapidly.

19. Replace main wheel fairing if previously fitted.

This completes the brake caliper pad/liner replacement procedure.

BRAKE BLEEDING PROCEDURE

Perform this procedure to bleed the brake system.

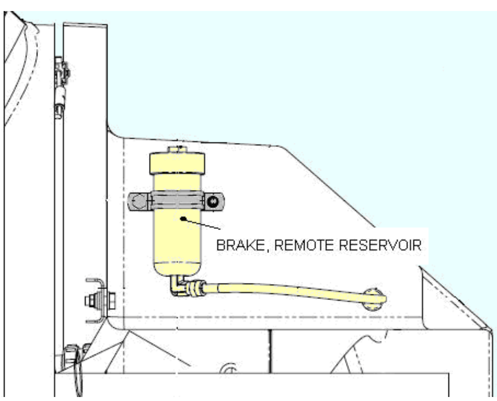
NOTE

This procedure requires the use of a pressure bleeder: either a hand-pump pressure accumulator or a 12 volt pump, both compatible with MIL-H-5606 hydraulic fluid. It is necessary to pump hydraulic fluid through the Port and starboard brake bleeder simultaneously. Therefore the supply line from the pressure bleeder must have a "TEE" fitting in it and one line from the "TEE" fitting attached to each brake bleeder port.

Check the pressure bleeder and supply of MIL-H-5606 hydraulic fluid to make sure they are COMPLETELY clean and free from contaminants.

During this procedure, hydraulic fluid will be returned from the brake system to the fluid reservoir. To avoid overflow and spillage, start the procedure with an empty reservoir, and a clean catch can. With a 6 foot clear tube attach to a substitute hydraulic reservoir cap.

Check that each brake pedal is not in any way in the depressed or actuated position during the bleeding process.



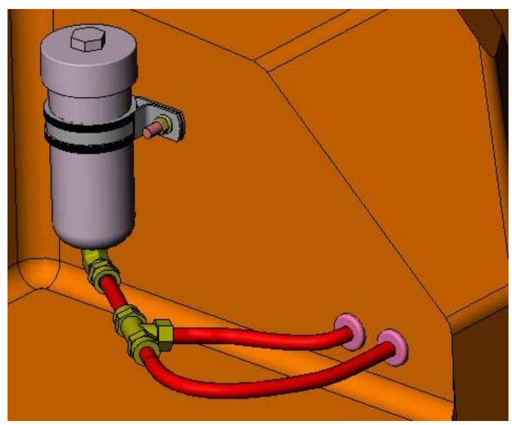


Figure 32-40 Remote Reservoir for the Finger Brake System (Left) and the Toe Brake System with T-Fitting (Right)

1. Chock all wheels.
2. If fitted remove main wheel fairing cover plate from either Port and starboard main wheel as appropriate or both main wheels if bleeding full system.
3. Remove and retain reservoir cap.
4. Check the brake fluid reservoir that it is no more than half-full with MIL-H-5606 hydraulic fluid.
5. Check the parking brake that it is released.

6. Open "ON/OFF" valve and check the hydraulic fluid in feed line from pressure bleeder that it is flowing freely with no air bubbles and no bubbles in line. Fit over Port and starboard bleed nipple with minimum fluid loss.
7. Open bleed fitting ½ to 1 turn.
8. Allow hydraulic fluid to be pumped through (left and right) brake system lines until no air bubbles are present in clear tubing connected to reservoir.
9. Close bleeder Port and starboard pressure bleeder.
10. With reservoir properly serviced and parking brakes in "OFF" position, cycle toe brakes several times from left seat first and then right seat.
11. Check the reservoir that it is properly serviced during brake cycling procedure.
12. Repeat step 8, 9 until a solid stream of hydraulic fluid is being pumped into reservoir (no visible air bubbles in 2 clear tubes which are connected to reservoir).
13. Close bleed fitting prior to discontinuing pressure bleed; disconnect pressure bleeder.
14. Stop pressure source.
15. Check and add fluid (as required) to the reservoir to top-off the fluid to the top of the bracket clamp.
16. Clean the filler cap. Inspect the breather hole that it is clear and unobstructed. Re-cap the reservoir with filler cap.
17. Refit main wheel fairing if previously removed.
18. Perform taxi test of both pilot and co-pilot brakes. Check the system that it operates correctly, prior to flight.

This completes the brake bleeding procedure.

BRAKE CALIPER OPERATIONAL CHECK AND INSPECTION

This procedure performs toe brake operational check and inspection.

1. Remove the upper cowl in accordance with Liberty Maintenance Manual Chapter 71-*Powerplant*.

NOTE

The following procedure applies to either a toe brake fitted aircraft or a finger brake fitted aircraft. Apply brake pressure in accordance with the system installed.

2. Fill the brake reservoir with MIL-H-5606 hydraulic fluid to the indicated full line.

CAUTION

The following steps will inspect brake caliper, lining and disc installations. Correct all discrepant conditions found prior to proceeding with this procedure. Uncorrected conditions may result in poor brake performance or brake failure.

3. Inspect port and starboard caliper assemblies for evidence of the following conditions:
   - Corrosion of any part of the assembly.
   - Bent anchor bolts.
   - Cracks around bolts.
   - Cracks around anchor bolt lugs and inlet fittings, and other visible damage.
   - Inspect the torque plate assemblies for corrosion around anchor bolt bushings and excessive wear in bushings.
4. Inspect the brake disk for evidence of the following conditions:
   - Corrosion in excess of surface coating.
   - Grooves, deep scratches or excessive general pitting. Single or isolated grooves up to 0.030 deep should not be cause for replacement, although general grooving of the disc faces will reduce lining life.
   - Coning beyond 0.015 inch (0.381 mm) in either direction.
   - Brake disc thickness equal to or less than 0.167 inch (4.242 mm) minimum allowed.

5. Inspect brake linings for the following conditions:
    - Loose, cracked, broken, or improperly glazed liner material.
    - Lining thickness equal to or less than 0.100 inch (2.54 mm) minimum allowed.

NOTE

The following steps are best accomplished with assistance. One person applies pressure on the brake system while another inspects the pressurized brake system for leaks.

6. Apply pressure to both the port and starboard brake control. Verify the brake controls are firm and do not drift as pressure is applied.
7. Verify brake calipers are fully engaged with both the port and starboard wheel brake disks.
8. With pressure still applied, inspect caliper and attached brake lines indication of leaks.
9. While holding pressure with the toe brake *pedal*s engage the parking brake lever as shown in Figure 32-21. Release pressure from the toe brakes once the parking brake is set.
10. Verify both the port and starboard brake caliper is fully engaged.
11. Inspect the parking brake valve and brake lines from the valve to the port and starboard brake calipers for leaks.
12. Verify pressure holds on the system while the parking brake is engaged.
13. Release the parking brake and verify the port and starboard calipers release.
14. Verify hydraulic fluid in the brake reservoir is still full and free of air bubbles.
15. Install the belly panel in accordance with Liberty Maintenance Manual Chapter 53-*Fuselage*.
16. Install the upper cowl in accordance with Liberty Maintenance Manual Chapter 71-*Powerplant*.
17. Taxi aircraft for 1500 feet with engine at 1700 rpm applying brake pedal force as needed to develop a 5-10 mph taxi speed.
18. Allow the brakes to cool for 10 to 15 minutes.
19. Apply brakes and check for restraint at high static throttle.
20. If brakes cannot hold aircraft during static run up, allow brakes to completely cool, and repeat the previous three steps. If brakes cannot hold after repeating these steps refer to main wheel caliper troubleshooting guide for corrective action.

This completes the brake caliper operation check and inspection procedure.

Section 40-21 Main Wheel Caliper System Troubleshooting Guide

Complaint	Possible Cause	Remedy
Calipers do not hold	Liner worn below limits Liner damaged	Replace liners
	Liners not conditioned	Perform liner conditioning procedure in accordance with Liberty Maintenance Manual Chapter 32-*Landing Gear*
	Disc worn below limits	Replace disc
	Leaking hydraulic fitting	Tighten fitting
	Air in brake system	Bleed brake system in accordance with Liberty Maintenance Manual Chapter 32-*Landing Gear*
Disc below limits for thickness or coning	In service wear Corrosion	Replace Disc
Caliper frozen	Anchor pins binding	Remove caliper and clean assembly
	Piston binding	Remove caliper and overhaul in accordance with manufactures directions.

Section 32-50 Steering

The airplane is steered on the ground by differential braking and/or rudder inputs. The nose landing gear is free to caster 85º to either side. Caster stops prevent it from turning further, thus preventing contact between the nose landing gear or its wheel fairing and the propeller.

The caster stop is incorporated into the friction plate located below the nose gear bearing housing.

Two ¾ inch tapered roller bearings, at the top and bottom of the housing, secure the nose gear assembly to the (vertical) end of the nose gear leg. Clean and lubricate these bearings in accordance with wheel bearing maintenance and extra care must be taken not to contaminate the friction surfaces of the shimmy damper assembly.

At the bottom of the nose gear bearing housing, a castellated nut presses on a set of six Belleville spring washers to set the nose gear's resistance to turning, thus providing a shimmy dampening function. Orientation of the six washers is critical to prevent shimmy, see Figure 32-29.

Section 50-01 Periodic Maintenance

Periodic Steering maintenance entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 50-02 Steering Procedures

This section has the removal and installation procedures for the castor stop/friction base.

NOSE GEAR LEG SERIAL NUMBER VERIFICATION

This procedure performs nose gear leg serial number verification.

1. Remove the belly panel in accordance with Chapter 53-*Fuselage*.
2. Record the nose gear has the serial number etched on the flat aft face of the leg (see Figure 32-28).
3. Contact Liberty Aerospace Customer Support for a the correct nose under carriage friction base as follows:
   - NOSE under carriage FRICTION BASE replacement part (P/N 135A-04-727), if your nose gear serial number is within (8940-6750 and 8940-6769).
   - NOSE under carriage FRICTION BASE replacement part (P/N 135A-40-555), if your nose gear leg serial number is not within (8940-6750 and 8940-6769) these numbers.

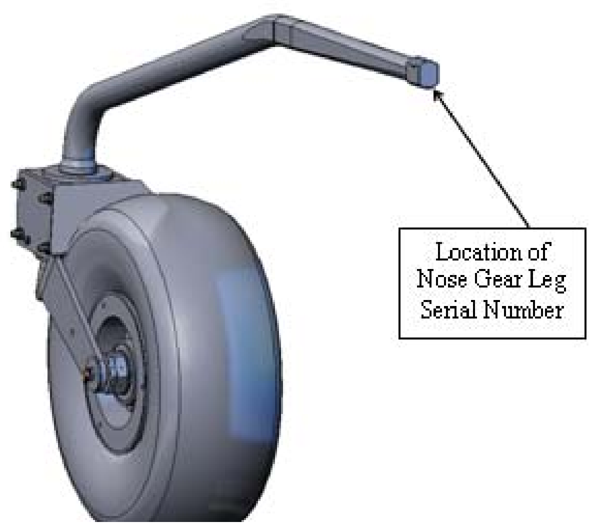


Figure 32-41 Location of Nose Gear Leg Serial number

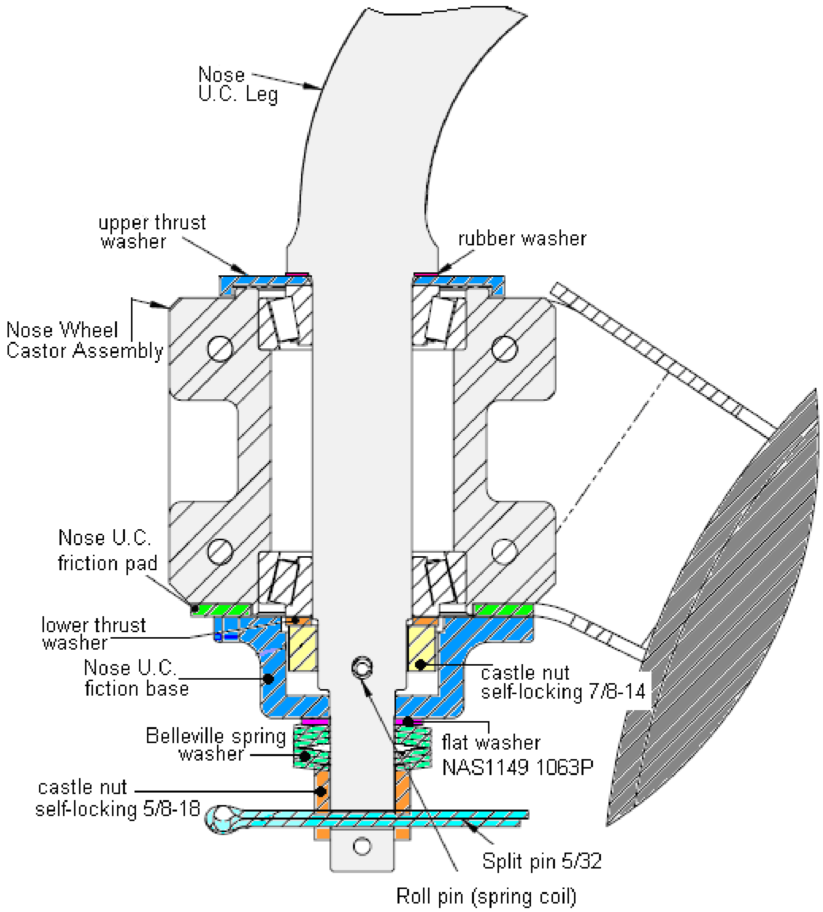


Figure 32-42 Nose Wheel Steering Assembly

4. Install the belly panel in accordance with Chapter 53-*Fuselage - Fuselage*.

This completes the nose gear leg serial number verification procedure.

CASTOR STOP/FRICTION BASE REMOVAL

Perform this procedure to remove the castor stop/friction base.

1. Chock both main landing gear wheels.
2. Fasten tail tie-down to tail stand weighing at least 300 lbs. If a tail stand is not available, place a pad under tail ensuring rudder is free to move. Place 50 – 75 lbs shot or sand bags on stabilizer root centerline on each side and carefully lower tail to 6” above floor.
3. Remove nose wheel fairing, if installed.
4. Remove and discard split pin from castellated nut below nose wheel bearing housing.
5. Remove castellated nut (MS17825-10, 5/8-18) and 6 Belleville spring washers, flat washers, and nose under carriage friction base.

This completes the castor stop/friction base removal procedure.

CASTOR STOP/FRICTION BASE INSTALLATION

Perform this procedure to install the castor stop/friction base.

NOTE

1. Orientation of washers: lower three with cone shape pointing down, upper three with cone shape pointing up.

2. Nose gear legs with a serial number 8940-6750 through 8940-6769 indicated on aft end of gear leg requires a matching castor stop (steel friction base P/N 135A-04-727). In event that replacement of piece part, P/N 135A-04-727 is required, this should be ordered to match nose gear legs in noted serial number range.

1. Replace nose undercarriage friction base, flat washer and 6 Belleville spring washers: upper three with cone shape pointing up, lower three with cone shape pointing down.
2. Perform Nose Gear Shimmy Damper Torque Test on page 85 of this chapter.
3. Install castellated nut (MS17825-10, 5/8-18) and tighten until a 30–35 lb-force is achieved at nose wheel axle to move nose wheel castor assembly. Rotate nut counter clockwise to align nearest slot and hole and install split pin. Verify that nose wheel cannot be turned past 85º left or right of center.

NOTE

Six Belleville spring washers, a friction base, and a friction pad constitute key elements of mechanical shimmy damper mechanism. Torque lower castellated nut (P/N MS17825-10, 5/8-18) spring loads Belleville washers and presses friction base on to friction pad. That torque must be sufficient to generate a set resistance to shimmy at nose wheel axle.

4. Install wheel fairing if previous installed per Main Wheel Fairing Removal on page 26 of this chapter.
5. Remove tail stand and lower nose wheel to floor.

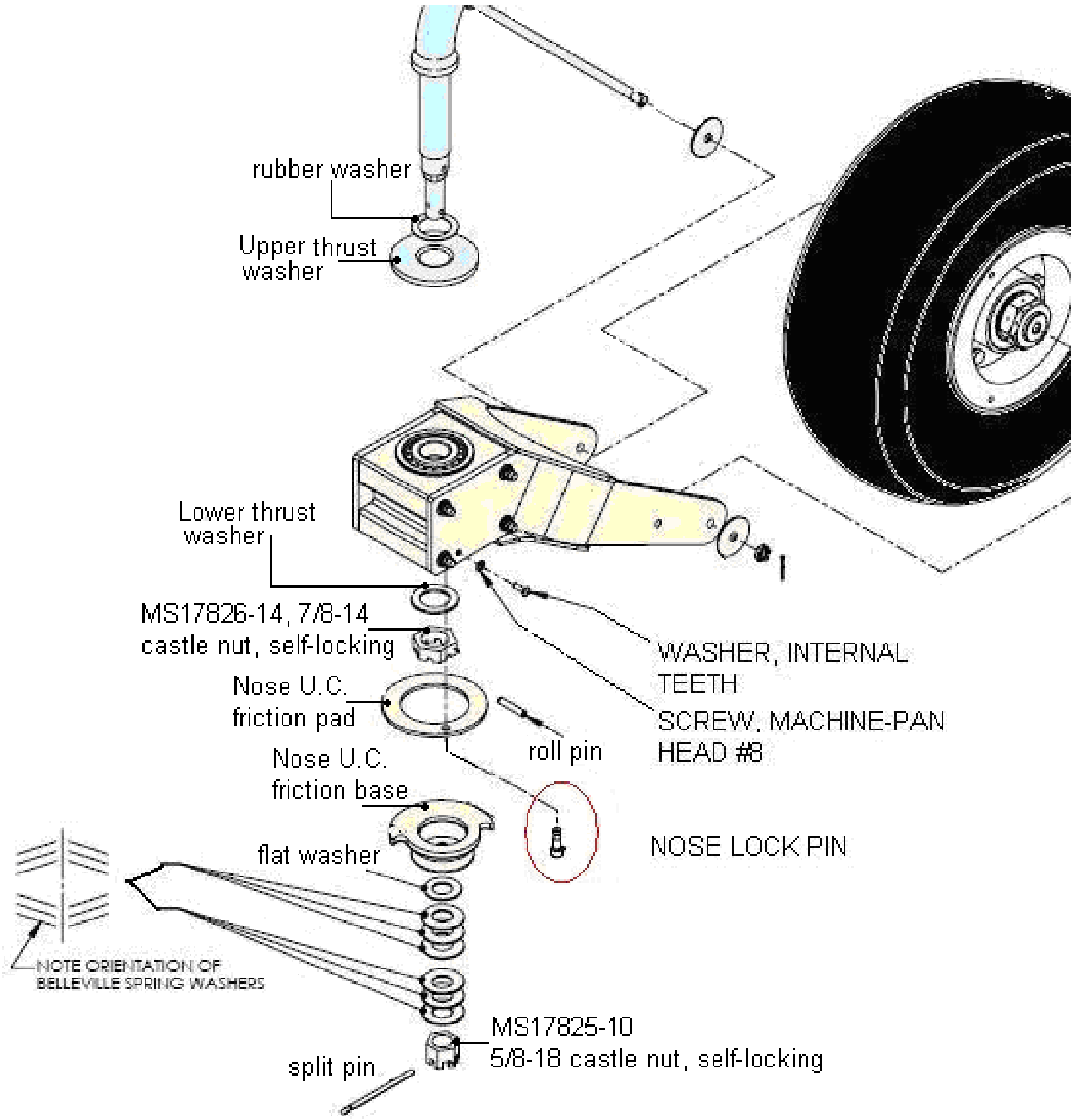


Figure 32-43 Exploded View of Nose Wheel Steering Assembly

This completes the castor stop/friction base installation procedure.

NOSE GEAR SHIMMY DAMPER TORQUE TEST

Perform this procedure to test the torque damper for nose gear shimmy.

1. Temporarily install split pin (P/N MS24665-428) to prevent castle nut (MS17825-10) from backing off during nose wheel axle pull testing.
2. Place a small strip of masking tape on nose wheel castor upper thrust washer and another one on top of nose wheel castor assembly. Make a mark on masking tape across upper thrust washer (see Figure 32-44).
3. Rotate nose wheel castor assembly until shimmy damper mechanism experiences friction drag. This is one end of free play; mark this position on tape located on nose wheel castor assembly. Rotate nose wheel castor assembly in other direction until shimmy damper mechanism experiences friction drag. This is other end of free play; mark this position on tape located on nose wheel castor assembly and measure distance between two marks. (See Figure 32-45).

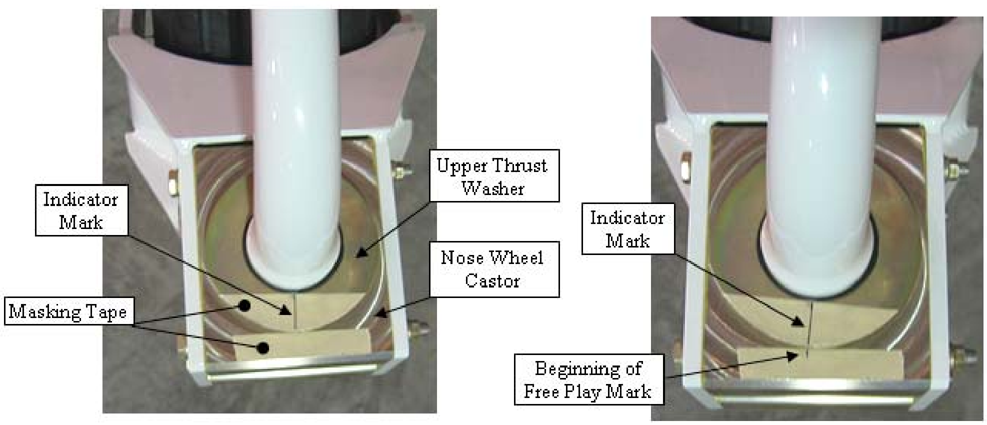


Figure 32-44 Rotational Free Play Measurement - Indicator Mark

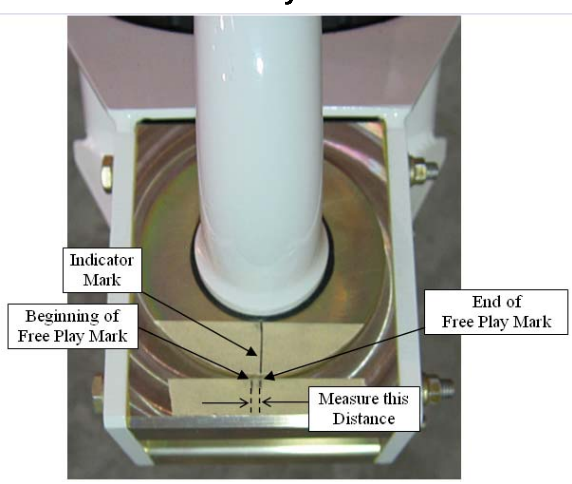


Figure 32-45 Maximum Rotational Free Play

4. Using a calibrated force gauge, perform Nose Gear Shimmy Damper Torque Test. Nose wheel tire must be off ground. For reference, Liberty's Quality Assurance personnel use a calibrated Mark-10 model MG100 load cell for this inspection.
5. Install a split pin (MS24665-428) to lock nut in position.
6. Remove, service, and install castor stop/friction base per Castor Stop/Friction Base Installation in accordance with Liberty Maintenance Manual Chapter 32-*Landing Gear*.
7. Verify tire pressure on all three (3) tires is set to 50 +2 / -0 psi cold.

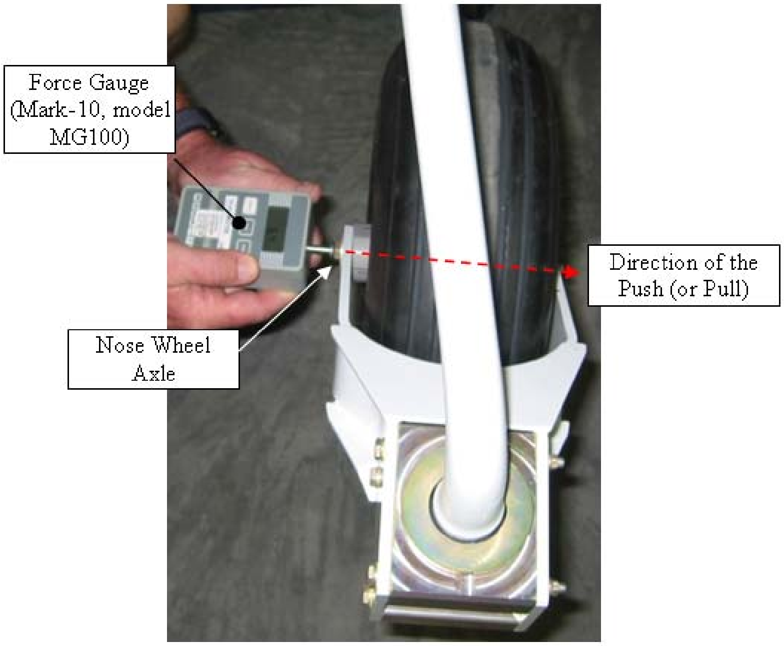


Figure 32-46 Nose Wheel Shimmy Load Test

This completes the nose gear shimmy dampener torque test procedure.

NOSE GEAR STEERING OPERATIONAL CHECK AND INSPECTION

Perform this procedure to inspect and maintain the nose gear steering.

1. Remove castor stop/friction base in accordance with procedure on page 82 of this chapter.

NOTE

There are six Belleville spring washers; a friction base and a friction pad constitute key elements of mechanical shimmy damper mechanism. Application of torque to the lower castellated nut (P/N MS17825-10, 5/8-18) spring loads Belleville washers and presses friction base on to friction pad. That torque must be sufficient to generate a set resistance to shimmy at nose wheel axle. This resistance was evaluated on worn assemblies and on assemblies with contaminated friction surfaces.

The following details steps to evaluate nose castor assembly and details service for shimmy damper mechanism.

2. Inspect Nose under carriage Friction Base for excess wear and rotational slop. Friction base is designed to wear against nose gear leg.

CAUTION

If free play between friction base and nose gear leg is greater than 0.15-inch, friction base must be replaced at next maintenance event, not exceeding 25 flight hours, or before next flight if nose wheel shimmy is currently being experienced.

3. Remove Nose under carriage Friction Pad (135A-40-303).
4. Remove nose lock pin.

NOTE

Inspect every 100 hour inspection and at every annual inspection. Replace every 500 hour inspection in accordance with Chapter 04.

5. Inspect for excessive wear-dimensional measurements not consistent with drawing (Nose Lock Pin 135A-40-565).
6. Remove roll pin from bearing retaining self-locking castle nut (P/N MS17826-14, 7/8-14). Remove castle nut and washer.
7. Remove Nose Wheel Castor Assembly from nose gear leg.
8. Inspect Nose Wheel Castor Assembly for cracks, deformations or other damage. Refer to the Steering Troubleshooting Guide at the end of this section for corrective action.

NOTE

If necessity clean and pack nose wheel castor bearings using SHC Mobil aviation grease. Check that all excess grease is removed form exterior of bearings and nose gear bearing housing.

9. Inspect nose gear leg spindle and if necessary clean and reapply a thin layer of grease (SHC Mobil aviation grease or equivalent).
10. Inspect under carriage Friction Pad (135A-40-303). Inspect for grease contamination and excessive wear. Inspect friction pad for deep grooves, replace as necessary. Clean friction base and pad from all contaminations (grease, oil, dirt, dust, etc.) using acetone before reinstalling.
11. Check that grease does not come in contact with any friction faces of shimmy damper components during installation. Friction pad moves with nose wheel castor and provides friction against friction base fixed with nose gear leg. Friction pad nominal thickness should be 0.125-inch.

NOTE

Liberty recommends replacement at next maintenance event, not exceeding 25 flight hours, if part has deep grooves or more than a 10% loss of thickness (minimum thickness of 0.112 in).

12. Reinstall Nose Wheel Castor Assembly. Install washer and self locking nut. Torque to 20-25 ft-lbs, and then back off to 0 ft-lb, then torque again to 20-25 ft-lbs. Rotate nut Counter Clockwise to align nearest slot and roll pin hole and insert roll pin.
13. Install friction pad. Install nose lock pin using Loctite® 270.
14. Install Castor Stop/Friction Base per Castor Stop/Friction Base Installation on page 83 of this chapter.

This completes the nose gear steering operational check and inspection procedure.

Section 50-03 Steering Troubleshooting Guide

Complaint	Possible Cause	Remedy
Nose tire irregular wear	Shimmy dampener fault	Perform nose shimmy dampener test in accordance with Liberty Maintenance Manual Chapter 32-*Landing Gear*
	Low tire pressure	Inflate tires to standard 50 +2/-0 PSI
Poor directional control	Loose castor assembly	Tighten castor assembly in accordance with Liberty Maintenance Manual Chapter 32-*Landing Gear*
	Castor bearing fault	Replace castor bearings.
	Low tire pressure	Inflate tires to standard 50 +2/-0 PSI
Cracked castor weldment	Hard landing	Perform hard landing inspection in accordance with Liberty Maintenance Manual Chapter 05. Replace castor
	Corrosion	Replace castor

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CHAPTER 33

LIGHTS

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Table of Contents

SECTION 33-00	GENERAL	5
SECTION 33-10	FLIGHT COMPARTMENT	7
SECTION 10-01	INTERNAL INSTRUMENT LIGHTING	7
SECTION 10-02	POST LIGHTING	7
SECTION 10-03	OVERHEAD LIGHTING	8
SECTION 10-04	LIGHTING PROCEDURES	8
	DC TO AC INVERTER REMOVAL	9
	DC TO AC INVERTERS INSTALLATION	10
	PANEL LIGHT DIMMER CONTROLS REMOVAL	12
	PANEL LIGHT DIMMER CONTROLS INSTALLATION	13
	FLIGHT INSTRUMENT BEZEL LUMINARIES REMOVAL	15
	FLIGHT INSTRUMENT BEZEL LUMINARIES INSTALLATION	16
	FLIGHT COMPARTMENT LIGHTING FUNCTIONAL CHECK	18
SECTION 10-05	TROUBLESHOOTING GUIDE	20
SECTION 33-40	EXTERIOR LIGHTING	21
SECTION 40-01	POSITION LIGHTS	21
SECTION 40-02	ANTI-COLLISION LIGHTS	22
SECTION 40-03	LANDING LIGHT	22
SECTION 40-04	EXTERIOR LIGHTING PROCEDURES	22
	POSITION LIGHT REMOVAL (INCANDESCENT ONLY)	23
	POSITION LIGHT INSTALLATION (INCANDESCENT ONLY)	24
	WING TIP LIGHT FIXTURE REMOVAL (LED ONLY)	25
	WING TIP LIGHT FIXTURE INSTALLATION (LED ONLY)	27
	ANTI-COLLISION FLASH TUBE REMOVAL (INCANDESCENT ONLY)	28
	ANTI-COLLISION FLASH TUBE INSTALLATION (INCANDESCENT ONLY)	30
	ANTI-COLLISION FLASH TUBE REMOVAL (LED ONLY)	31
	ANTI-COLLISION FLASH TUBE INSTALLATION (LED ONLY)	33
	LANDING LIGHT REMOVAL	34
	LANDING LIGHT INSTALLATION	35
	EXTERIOR LIGHTING FUNCTIONAL CHECK	36
SECTION 40-05	TROUBLESHOOTING GUIDE	37

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Section 33-00 General

The airplane lighting system includes interior (flight compartment) and exterior (position, anti-collision, and landing) lights.

The interior lighting consists of energy efficient super bright Light Emitting Diodes (LEDs) or LED panels and electro-luminescent panels. These devices provide sufficient light to allow the pilot to see instruments and to read charts without being too bright as to blind the pilot at night.

Mounted to panels and consoles are individual area-lighting LEDs post lamp assemblies. These device shine their light out and down so the light the area of the panel without shining in the eyes of the pilot. Individual instruments mounted to the instrument console have a bezel mounted light source. This light source is an electro-luminescent panel mounted to the instrument. This type of light source provides a softer light and illuminates the instruments faces to allow the pilot to see and read the various instruments. Both the LED post lights and the bezel lights have individual dimmer controls that allow the pilot to set the level of illumination.

The ceiling mounted overhead LED light panel provides general lighting to the cabin. It is meant to allow the pilot the ability to read charts or the flight manual at night. A panel mounted switch turns the overhead light on and off.

The exterior lighting consists of the forward-looking landing light (mounted to the lower engine cowl) and the airplane, position, navigation, and anti-collision lighting mounted to each of the wing tips.

The landing light is to allow the pilot to illuminate the immediate area ahead of the airplane. This is particularly suitable while taxing on the ground or during approach. The light source is a standard sealed-beam incandescent light bulb. Control of the light is from an instrument panel light switch.

The position, navigation, and anti-collision lights allow the airplane to be seen at night by other aircraft in the area. These lights are mounted to the airplane's wing tips. The position lights each face looking aft from the wing and are white in color. The navigation lights face looking forward and are red for the left wing and green for the right wing. Control of the position and navigation lights is from a panel-mounted switch in front of the pilot's position.

The anti-collision lights are high-energy light strobes that face looking out to the side from the wing. Control of the anti-collision lights is from a panel-mounted light switch in front of the pilot's position.

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Section 33-10 Flight Compartment

Interior (flight compartment) lighting in the Liberty XL-2 includes internal lighting for most instruments, integral backlighting for the Integrated Engine Instrument System display, instrument bezel lights for all installed avionics components, post lighting for instrument panel switches and some indicators, and an overhead light for general illumination and map-reading purposes.

All interior lighting is powered from the main distribution bus via a 3-amp circuit breaker labeled “LIGHTING.”

Section 10-01 Internal Instrument Lighting

Separate bezel lights illuminate the following instruments: Airspeed indicator, Attitude Indicator, Altimeter, Turn Coordinator, Direction Indicator, Vertical Speed Indicator, NAV/ILS Indicator(s), Clock, and Fuel Quantity.

AC power from a solid-state inverter provides power to illuminate bezel lights. A dimmer control, labeled INSTRUMENT LIGHTS, provides variable DC voltage to control the AC power, and therefore the amount of light from the bezel lighting system.

Maintenance for the bezel instrument lighting system is limited to identification and repair of defective wiring or replacement of the dimmer and inverter. Replacement of bezel lights installed inside instruments must be carried out by an authorized instrument repair facility.

Section 10-02 Post Lighting

Six low-drain LED (Light Emitting Diodes) post lights illuminate various panels in the airplane. There is one post light on the instrument panel, one on the avionics panel, two on the CB (circuit breaker) panel, and two on the center console panel. There is internal illumination for the magnet compass (mounted to the forward windscreen).

Control of the DC voltage to all of the post lights, except the instrument panel post light, is through a panel mounted dimmer control (R001), located to the right of the VM1000FX display. Control of the instrument panel post light is on the same dimmer control for the internal instrument lighting (R002).

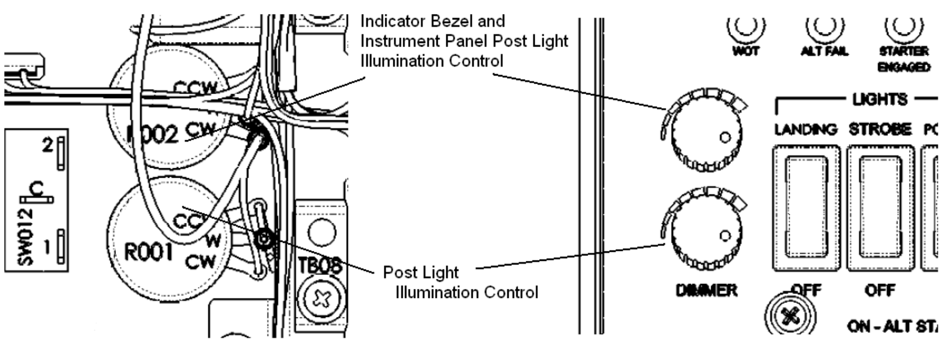

Figure 33-1 Location of the Dimmer Controls for the Post Lights

Maintenance for the post light system is limited to identification and repair of defective wiring or replacement of the dimmer or of individual post light units.

Section 10-03 Overhead Lighting

Maintenance for the overhead light is limited to identification and repair of defective wiring or replacement of the overhead LED light unit.

Section 10-04 Lighting Procedures

This section has the removal and installation procedures for the following assemblies:

- DC to AC invertors
- Panel Light Dimmer controls
- Bezel Luminaries

DC TO AC INVERTER REMOVAL

Perform this procedure to remove the DC to AC inverter. Liberty Aerospace, Inc. recommends removing the instrument panel from the airplane to do this procedure.

1. Remove the instrument panel from the airplane. See *Chapter 31- Indicators and Recording Systems* (Revision D), Section 31-10-02, for the procedure to remove the instrument panel.
2. Disconnect the three connectors on the health status annunciator.
3. Remove the four screws securing the Health Status Annunciator box to the terminal board bracket. Lay the Health Status Annunciator to one side.

NOTE

It is not necessary to totally remove the Heath Status Annunciator. However, it is necessary to disconnect all three of the connectors to be able to move the annunciator from the instrument panel enough to access the terminal barrier strips.

4. Remove the two screws securing the DC to AC inverter to the bracket on the rear of the instrument panel.
5. Cut sufficient cable ties to free up the wiring from the inverter.
6. Remove ring terminal with the violet wire from terminal 4 of terminal barrier TB07.
7. Remove the other ring terminal with the violet wire from terminal 4 of terminal barrier TB08.
8. Remove the ring terminal with the black wire from the chassis ground.
9. Identify and tag the wires going to splice SP056.
10. Cut the wires that go to the splice SP056. Discard the used splice.
11. Remove the DC to AC inverter from the instrument panel.
12. If installing a replacement DC to AC inverter now, proceed to the DC to AC Inverters Installation procedure.
13. If installing the replacement DC to AC inverter later, secure the Health Status Annunciator using the hardware removed in step 3 of this procedure.
14. Connect the three connectors that go to the Health Status Annunciator.
15. Install the instrument panel in to the instrument console. See *Chapter 31- Indicators and Recording Systems* (Revision D), Section 31-10-02, for the procedure to install the instrument panel.

This completes the DC to AC Inverter Removal procedure.

DC TO AC INVERTERS INSTALLATION

Perform this procedure to install the DC to AC inverter. Liberty Aerospace, Inc. recommends removing the instrument panel from the airplane to do this procedure. For a schematic of this circuit, see Chapter 91 – *Wiring Diagrams*.

1. If installing a replacement DC to AC inverter immediately after removing a previous DC to AC inverter, then proceed to step 5 below.
2. Remove the instrument panel from the airplane. See *Chapter 31- Indicators and Recording Systems* (Revision D), Section 31-10-02, for the procedure to remove the instrument panel.
3. Disconnect the three connectors on the health status annunciator.
4. Remove the four screws securing the Health Status Annunciator box to the terminal board bracket. Lay the Health Status Annunciator to one side.

NOTE

It is not necessary to totally remove the Heath Status Annunciator. However, it is necessary to disconnect all three of the connectors to be able to move the annunciator from the instrument panel enough to access the terminal barrier strips.

5. Using the two screws removed in step 4 of the DC to AC Inverter Removal procedure; secure the DC to AC inverter to bracket on the rear surface of the instrument panel.
6. If the DC to AC inverter does not already have ring terminals on the wires, install the appropriately sized ring terminals to the two violet wires and the black wire of the inverter. The green wire does not need to have a ring terminal.
7. Connect the ring terminal from one of the violet wires to terminal 4 on terminal barrier TB07.
8. Connect the ring terminal from other of the violet wires to terminal 4 on terminal barrier TB08.
9. Connect the ring terminal from the black wire to a convenient chassis ground stud on the instrument panel.
10. Connect the green wire from the DC to AC inverter, and wire numbers JPR50A22, JPR49A22, L85C22 with a splice labeled SP056.
11. Secure the loose wires with cable ties.
12. Secure the Health Status Annunciator using the hardware removed in step 3 of the DC to AC Inverter Removal procedure.
13. Connect the three connectors that go to the Health Status Annunciator.
14. Secure the loose wires with cable ties.

15. Install the instrument panel in to the instrument console. See *Chapter 31-Indicators and Recording Systems* for the procedure to install the instrument panel.
16. Perform the Flight Compartment Lighting Functional Check on page 18 of this chapter.

This completes the DC to AC Inverters Installation procedure.

PANEL LIGHT DIMMER CONTROLS REMOVAL

Perform this procedure to remove the panel light dimmer controls. Both dimmer controls are located in the same area of the instrument panel and are removable using this procedure. Liberty Aerospace, Inc. recommends removing the instrument panel from the airplane to do this procedure.

1. Remove the instrument panel from the airplane. See *Chapter 31- Indicators and Recording Systems* for the procedure to remove the instrument panel.
2. Disconnect the three connectors on the health status annunciator.
3. Remove the four screws securing the Health Status Annunciator box to the terminal board bracket. Lay the Health Status Annunciator to one side.

NOTE

It is not necessary to totally remove the Heath Status Annunciator. However, it is necessary to disconnect all three of the connectors to be able to move the annunciator from the instrument panel enough to access the dimmer controls.

4. Tag and remove the wires from the dimmer control.
5. Use a 0.048mm Allen-key to remove the knob from the dimmer control.
6. Use a ½-inch nut driver to remove the panel nut securing the dimmer control to the instrument panel.
7. Remove the dimmer control from the rear of the instrument panel,
8. If installing a replacement dimmer control now, proceed to the Panel Light Dimmer Controls Installation procedure.
9. If installing the replacement dimmer control later, secure the Health Status Annunciator using the hardware removed in step 3 of this procedure.
10. Connect the three connectors that go to the Health Status Annunciator.
11. Install the instrument panel in to the instrument console. See *Chapter 31- Indicators and Recording Systems* for the procedure to install the instrument panel.

This completes the Panel Light Dimmer Controls Removal procedure.

PANEL LIGHT DIMMER CONTROLS INSTALLATION

Perform this procedure to install the panel light dimmer control. Both dimmer controls are located in the same area of the instrument panel and maybe installed using this procedure. For a schematic of this circuit, see Chapter 91 – *Wiring Diagrams*.

1. If installing a replacement dimmer control immediately after removing a previous dimmer control, then proceed to step 5 below.
2. Remove the instrument panel from the airplane. See *Chapter 31- Indicators and Recording Systems* for the procedure to remove the instrument panel.
3. Disconnect the three connectors on the health status annunciator.
4. Remove the four screws securing the Health Status Annunciator box to the terminal board bracket. Lay the Health Status Annunciator to one side.

NOTE

It is not necessary to totally remove the Heath Status Annunciator. However, it is necessary to disconnect all three of the connectors to be able to move the annunciator from the instrument panel enough to access the dimmer controls.

5. Install the dimmer control in to the instrument panel. Using a ½-inch nut driver, secure the dimmer control using the ½-inch nut removed in step 6 of the Panel Light Dimmer Controls Removal procedure on page 12.
6. Using a 0.048mm Allen-key, secure the knob for the dimmer control removed in step 5 of the Panel Light Dimmer Controls Removal procedure on page 12.
7. Connect the dimmer control wiring removed in step 4 of the Panel Light Dimmer Controls Removal procedure on page 12. See Figure 33-2 for a schematic of the wiring for an individual dimmer control.

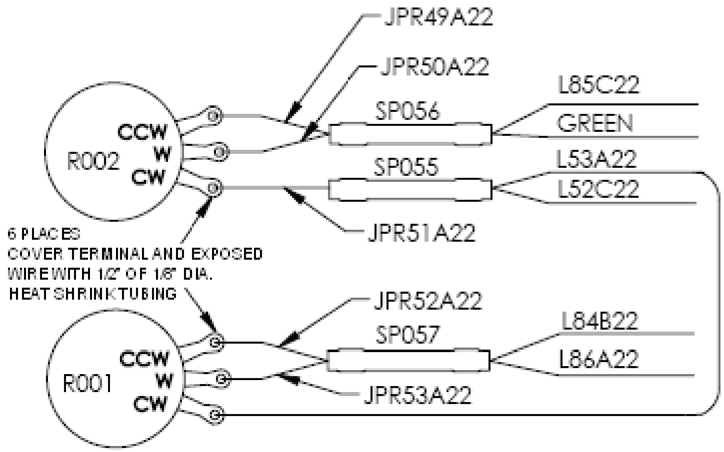

Figure 33-2 Light Dimmer Wiring

8. Secure the loose wires with cable ties.
9. Secure the Health Status Annunciator using the hardware removed in step 3 of the DC to AC Inverter Removal procedure.
10. Connect the three connectors that go to the Health Status Annunciator.
11. Secure the loose wires and cables with cable ties.
12. Install the instrument panel in to the instrument console. See *Chapter 31-Indicators and Recording Systems* for the procedure to install the instrument panel.
13. Perform the Flight Compartment Lighting Functional Check on page 18 of this chapter.

This completes the Panel Light Dimmer Controls Installation procedure.

FLIGHT INSTRUMENT BEZEL LUMINARIES REMOVAL

Perform this procedure to remove the flight instrument bezel luminaries. All of the flight instruments on the instrument panel have an external bezel luminary for instrument lighting. The removal of the luminaries is the same for all of the instruments. Liberty Aerospace, Inc. recommends removing the instrument panel from the airplane to do this procedure.

1. Remove the instrument panel from the airplane. See *Chapter 31- Indicators and Recording Systems* for the procedure to remove the instrument panel.
2. Disconnect the three connectors on the health status annunciator.
3. Remove the four screws securing the Health Status Annunciator box to the terminal board bracket. Lay the Health Status Annunciator to one side.

NOTE

It is not necessary to totally remove the Heath Status Annunciator. However, it is necessary to disconnect all three of the connectors to be able to move the annunciator from the instrument panel enough to access the dimmer controls.

4. Remove the four screws that secure the instrument and bezel luminary to the instrument panel.
5. Remove the defective bezel luminary from the instrument panel.
6. If not replacing the luminary immediately, secure the flight instrument to the instrument panel using the screws.
7. Carefully cut and remove sufficient cable ties to free the wiring for the luminary.
8. Remove the ring terminals attached to the wires for the luminary from terminal barriers TB07 and TB08.
9. If replacing the luminary immediately, proceed to the Flight Instrument Bezel Luminaries Installation procedure on page 9 of this chapter.
10. Secure the Health Status Annunciator to the instrument panel using the screws removed in step 3 above.
11. Connect the Health Status Annunciator cables to their respective connector.
12. If not replacing the luminary immediately, install the instrument panel. See *Chapter 31- Indicators and Recording Systems* for the procedure to install the instrument panel.

This completes the Flight Instrument Bezel Luminaries Removal procedure.

FLIGHT INSTRUMENT BEZEL LUMINARIES INSTALLATION

Perform this procedure to install the flight instrument bezel luminaries. Liberty Aerospace, Inc. recommends removing the instrument panel from the airplane to do this procedure.

1. If installing a replacement instrument bezel luminary immediately after removing a previous instrument bezel luminary, then proceed to step 5 below.
2. Remove the instrument panel from the airplane. See *Chapter 31- Indicators and Recording Systems* for the procedure to remove the instrument panel.
3. Disconnect the three connectors on the health status annunciator.
4. Remove the four screws securing the Health Status Annunciator box to the terminal board bracket. Lay the Health Status Annunciator to one side.

NOTE

It is not necessary to totally remove the Heath Status Annunciator. However, it is necessary to disconnect all three of the connectors to be able to move the annunciator from the instrument panel enough to access the terminal barriers for the instrument bezel luminaries.

5. Position the bezel luminary between the instrument and the instrument panel.

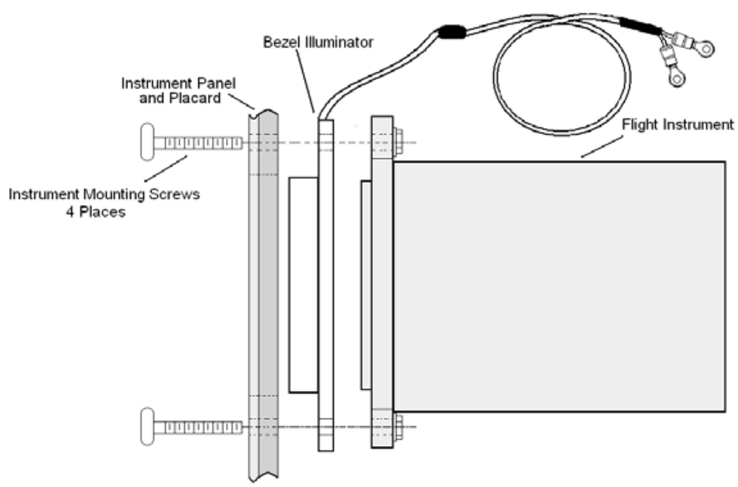

Figure 33-3 Showing the Assembly of the Bezel Luminary into the Instrument Panel

6. Secure the instrument and bezel luminary with the screws removed in step 4 of the Flight Instrument Bezel Luminaries Removal procedure on page 15 of this chapter.
7. Secure the loose wires with cable ties. Use one cable tie around the indicator securing the bezel wiring to the indicator.

8. Secure the Health Status Annunciator using the hardware removed in step 3 of the DC to AC Inverter Removal procedure.
9. Connect the three connectors that go to the Health Status Annunciator.
10. Secure the loose wires with cable ties.
11. Install the instrument panel in to the instrument console. See Chapter 31 - *Indicators and Recording Systems* for the procedure to install the instrument panel.
12. Perform the Flight Compartment Lighting Functional Check on page 18 of this chapter.

This completes the Flight Instrument Bezel Luminaries Installation procedure.

FLIGHT COMPARTMENT LIGHTING FUNCTIONAL CHECK

Perform the following procedure any time interior lighting systems have been serviced. Refer to the troubleshooting guide to follow for corrective actions as required.

1. Position BAT1 circuit breaker – CLOSED
2. Position LIGHTS, CABIN circuit breaker – CLOSED
3. Position LIGHTS, INST circuit breaker – CLOSED
4. Position aircraft split master switch – ON
5. Position Avionics Master Switch – ON
6. Radio power switches - ON
7. Rotate upper DIMMER control clockwise to full bright
8. Rotate lower DIMMER control clockwise to full bright
9. Verify LIGHTS, INST circuit breaker remains - CLOSED
10. Verify each of the following instrument lights are on bright. Note any deviation in the space provided in Table 33-1
11. Rotate upper DIMMER control to ~50%
12. Rotate lower DIMMER control to ~50%
13. Verify each of the following instrument lights are ~ 50% bright. Note any deviation in the spare provided in Table 33-1

Light	Verified Bright	Verified Dim	Light	Verified Bright	Verified Dim
Airspeed Indicator			VM1000FX Engine Display		
Attitude Indicator			Annunciator Post Light		
Courser Deviation Indicator 1 (If Installed)			Avionics Panel Post Light		
Turn Coordinator			Audio Panel Back Lighting		
Directional Gyro			Radio Back Lighting		
Vertical Speed Indicator			Center Console Post Lights		
Course Deviation Indicator 2 (If Installed)			Circuit Breaker Panel Post Lights		
OAT/Clock					

Table 33-1 Instrument Lighting Check Table

14. Rotate upper DIMMER control fully counter clockwise

15. Rotate lower DIMMER control fully counter clockwise
16. Position Avionics Master Switch - OFF
17. Position cabin light switch – ON
18. Verify cabin light is on.
19. Verify LIGHT, CABIN circuit breaker remains – CLOSED
20. Position cabin light switch - OFF
21. Position aircraft split master switch – OFF

This completes Flight Compartment Lighting Functional Check procedure.

Section 10-05 Troubleshooting Guide

The following table has a guide to troubleshooting the flight compartment lighting.

Complaint	Possible Cause	Remedy
All interior lights except overhead light inoperative	defective LIGHTING circuit breaker	replace circuit breaker
	defective wiring	repair
All internal instrument lights inoperative, post lights OK	defective INSTRUMENT LIGHTS dimmer	replace dimmer
	defective wiring	repair
Post lights inoperative, internal instrument lights OK	defective POST LIGHTS dimmer	replace dimmer
	defective wiring	repair
Lights remain at full brightness or dimmer action uneven (Instrument or Post Lights)	defective dimmer	replace dimmer
Overhead light inoperative	defective 2 amp fuse	replace fuse
	cabin “CB”	reset or replace circuit breaker
	defective wiring	repair
	defective overhead LED light	replace LED assembly
Individual bezel instrument light inoperative	defective bezel light assembly	replace bezel light assembly
	defective wiring	repair
Individual post light inoperative	defective LED post light	replace LED assembly
	defective wiring	repair

Table 33-2 Compartment Lighting Troubleshooting Guide

Section 33-40 Exterior Lighting

Exterior lighting on the airplane includes two combination position / strobe (anti-collision) lights installed on the left and right wingtips and a single landing light located in the lower engine cowl.

Section 40-01 Position Lights

Each wingtip position/anti-collision light includes a colored position light visible from the front and side of the airplane (red on the left wingtip, green on the right wingtip) and a white light visible from the rear of the airplane.

The light source for the wingtip position/anti-collision lights can be either conventional incandescent bulbs or new generation super-bright LED bulbs. Externally, incandescent and LED light assemblies are very similar. Each are mounted the same wing tip foot print. Identification of which type of assembly is installed requires a brief visual examination.

NOTE

The word "bulb" is used to refer to either an incandescent or super-bright LED light source.

Inspect the navigation light lens. If the light has a red lens on the left (port) side and a green lens on the right (starboard) side incandescent assemblies are fitted to the aircraft. If each navigation light lens is clear, super-bright LED assemblies are installed. In addition, examine the aircraft main circuit breaker panel. If the LIGHTS, NAV circuit breaker is 10-Amp the airplane most likely has incandescent fixtures. If the circuit breaker is 2-Amp the airplane has super-bright LED fixtures installed. With a 10 amp breaker always check the lens color to be to verify an incandescent light fixture is installed.

Figure 33-4 Wingtip Lights LED and Incandescent

Power for all four of incandescent fixture bulbs is through the 10-amp LIGHTS, NAV circuit breaker. Power for all four of the super-bright LED bulbs is through the 2-amp LIGHTS, NAV circuit breaker. The LIGHTS, POSITION switch located on the flight instrument panel controls the power going to all of the NAV and POSITION lights.

CAUTION

Do not pair incandescent light fixtures with LED light fixtures on the same airplane. Incandescent fixtures fitted to an LED fixture configured aircraft will result in a LIGHTS, NAV circuit breaker trip.

Section 40-02 Anti-collision Lights

Each wingtip position/anti-collision light includes a strobe flash tube. A single high-voltage power supply (installed in the fuselage) powers the left and right strobes. The high voltage power supply is configured to flash strobes alternately.

DC power to the strobe power supply is provided through a 10-amp circuit breaker, (LIGHTS, STROBE). Control of the strobe lights is through a LIGHTS, STROBE switch located on the flight instrument panel.

Section 40-03 Landing Light

The single landing light is installed in the lower engine cowling.

Power for the landing light is through the 10-amp LIGHTS, LAND circuit breaker on the main distribution bus. Control is provided by means of a LIGHTS, LANDING switch located on the flight instrument panel.

Section 40-04 Exterior Lighting Procedures

This section contains the removal and installation procedures for the position lights, anti-collision lights, and the landing light. There are separate procedures depending on the type of wing tip light fixture installed on the aircraft.

POSITION LIGHT REMOVAL (INCANDESCENT ONLY)

Perform this procedure to remove the position light bulbs. Procedure applies to port or starboard light installations as required. This procedure is for wingtip light assemblies that have a standard incandescent bulb.

NOTE

Skin oils on the surface of position light bulbs or anti-collision light flash tubes will significantly shorten bulb/flash tube life. Wear latex or vinyl gloves when handling bulbs/flash tubes. Bulbs/flash tubes can be cleaned using 70% isopropyl alcohol swabs ("medical prep" swabs).

1. Position aircraft split master switch - OFF
2. Position circuit breaker BAT1 - OPEN
3. Remove screw securing position light assembly cover.
4. Remove lens covering position light bulb.
5. Remove defective position light bulb.

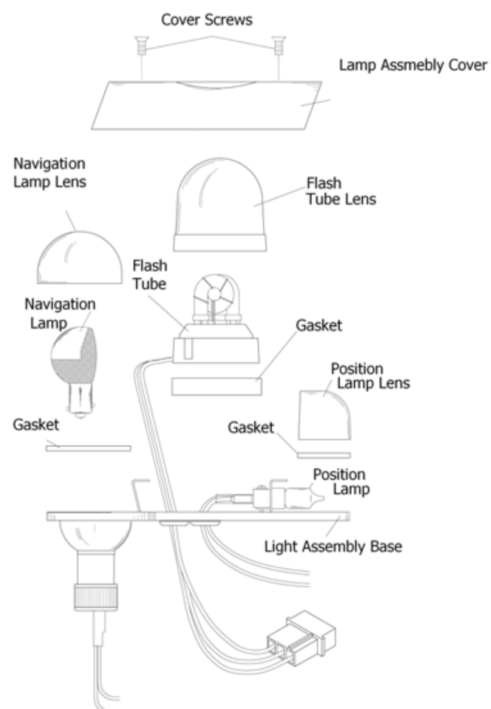

Figure 33-5 Position Light Assembly

This completes the Position Light Removal (Incandescent Only) procedure.

POSITION LIGHT INSTALLATION (INCANDESCENT ONLY)

Perform this procedure to install the position light. Procedure applies to port or starboard installations as required. This procedure is for wingtip light assemblies that have a standard incandescent bulb.

NOTE

Skin oils on the surface of position light bulbs or anti-collision light flash tubes will significantly shorten bulb/flash tube life. Wear latex or vinyl gloves when handling bulbs/flash tubes. Bulbs/flash tubes can be cleaned using 70% isopropyl alcohol swabs ("medical prep" swabs).

1. Position aircraft split master switch - OFF
2. Position circuit breaker BAT1 - OPEN
3. Install replacement position light bulb as required.
4. Clean the bulb using 70% isopropyl alcohol.
5. Clean the inside surfaces of the lens using 70% isopropyl alcohol.
6. Replace position light lens.
7. Secure lens in place by installation of light assembly cover with retaining screw.
8. Position circuit breaker BAT1 – CLOSED
9. Perform functional test as described in the Exterior Lighting Functional Check procedure on page 36 of this chapter.

This completes the Position Light Installation (Incandescent Only) procedure.

WING TIP LIGHT FIXTURE REMOVAL (LED ONLY)

Perform this procedure to remove the LED wing tip light fixture. Procedure applies to port or starboard light installations as required.

NOTE

LED elements are integrated in the fixture base. Navigation and position LED lights are replaced by replacement of the fixture base or the entire fixture assembly.

NOTE

Skin oils on the surface of position light bulbs or anti-collision light flash tubes will significantly shorten bulb/flash tube life. Wear latex or vinyl gloves when handling bulbs/flash tubes. Bulbs/flash tubes can be cleaned using 70% isopropyl alcohol swabs (“medical prep” swabs).

1. Position aircraft split master switch - OFF
2. Position circuit breaker BAT1 – OPEN

WARNING

Anti-collision (strobe) light systems can generate dangerous or lethal voltages. Ensure that anti-collision light system has been turned off for at least 30 minutes before attempting to disconnect flash tubes.

3. Remove wing tip assembly in accordance with applicable sections of the AMM Chapter 57, *Wings*. Retain screws for installation.

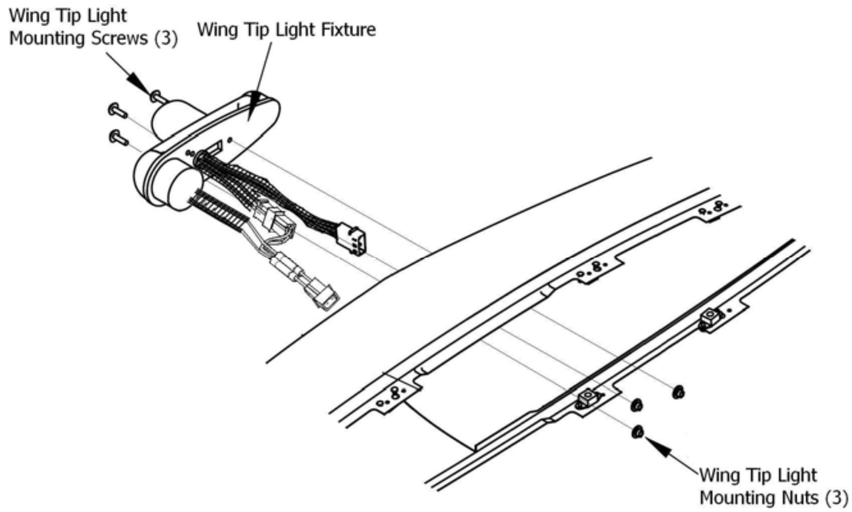

Figure 33-6

4. For the starboard light fixture, disconnect the connectors P/J 50 and P/J 64. For the portside light fixture, disconnect connector P/J-49, and P/J-59.
5. Remove screws securing lens retainer (cover).
6. Remove the three screws securing the light fixture
7. Feed the cable assembly out of the wing tip.

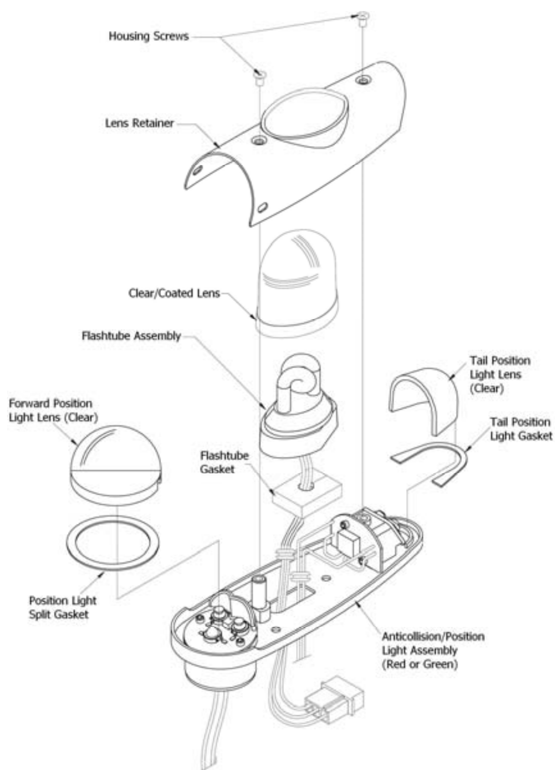

Figure 33-7

This completes the Wing Tip Light Fixture Removal (LED Only) procedure.

WING TIP LIGHT FIXTURE INSTALLATION (LED ONLY)

Perform this procedure to install the wing tip light fixture. Procedure applies to port or starboard light installations as required.

NOTE

LED elements are integrated in the fixture base. Navigation and position LED lights are replaced by replacement of the fixture base or the entire fixture assembly.

NOTE

Skin oils on the surface of position light bulbs or anti-collision light flash tubes will significantly shorten bulb/flash tube life. Wear latex or vinyl gloves when handling bulbs/flash tubes. Bulbs/flash tubes can be cleaned using 70% isopropyl alcohol swabs ("medical prep" swabs).

1. Position the aircraft's split master switch – OFF.
2. Position the circuit breaker BAT1 – OPEN.

WARNING

Anti-collision (strobe) light systems can generate dangerous or lethal voltages. Ensure that anti-collision light system has been turned off for at least 30 minutes before attempting to connect flash tubes.

3. Install the light fixture to the wing tip using the hardware removed the Wing Tip Light Fixture Removal (LED Only) procedure on page 25 of this chapter.
4. Install the wing tip to the wing in accordance with applicable sections of the AMM Chapter 57, *Wings* using the hardware removed in the Wing Tip Light Fixture Removal (LED Only) procedure on page 25 of this chapter.
5. Clean the inside surfaces of the lens using 70% isopropyl alcohol.
6. Install position light lens.
7. Secure lens in place by installation of light assembly cover with retaining screw.
8. Position circuit breaker BAT1 – CLOSED
9. Perform functional test as described in the Exterior Lighting Functional Check procedure on page 36 of this chapter.

This completes the Wing Tip Light Fixture Installation (LED Only) procedure.

ANTI-COLLISION FLASH TUBE REMOVAL (INCANDESCENT ONLY)

Perform this procedure to remove anti-collision flash tubes. This procedure is for wingtip light assemblies that have a standard incandescent bulb. Procedure applies to port or starboard installations as required.

WARNING

ANTI-COLLISION (STROBE) LIGHT SYSTEMS CAN GENERATE DANGEROUS OR LETHAL VOLTAGES. ENSURE THAT ANTI-COLLISION LIGHT SYSTEM HAS BEEN TURNED OFF FOR AT LEAST 30 MINUTES BEFORE ATTEMPTING TO REMOVE OR REPLACE FLASH TUBES.

NOTE

Skin oils on the surface of position light bulbs or anti-collision light flash tubes will significantly shorten bulb/flash tube life. Wear latex or vinyl gloves when handling bulbs/flash tubes. Bulbs/flash tubes can be cleaned using 70% isopropyl alcohol swabs ("medical prep" swabs).

1. Position aircraft split master switch - OFF
2. Position circuit breaker BAT1 - OPEN
3. Verify the anti-collision light system has been off for a minimum of 30 minutes.
4. Remove screw securing position light assembly cover.
5. Remove lens covering anti-collision flash tube.
6. Remove wing tip assembly. Retain screws for reinstallation.

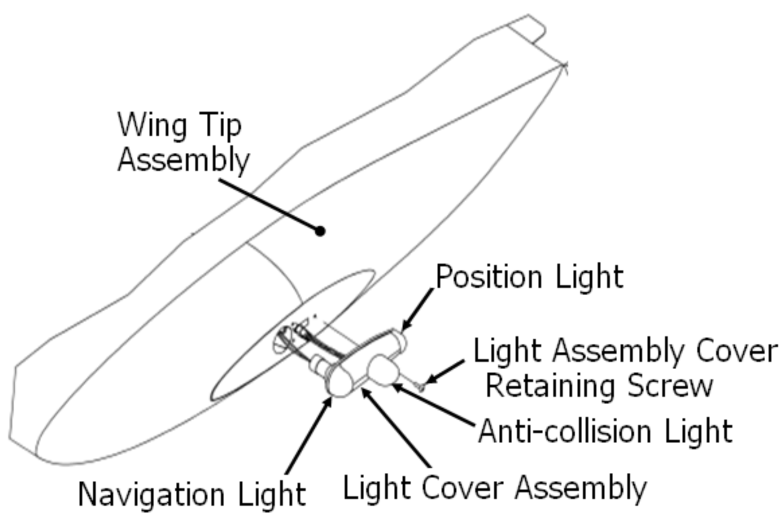

Figure 33-8 Wing Tip Assembly

7. Locate and disconnect P49 (Port) or P50 (Starboard)
8. Remove pins from P49 (Port) or P50 (Starboard)
9. Remove braided sleeve and retain for later use.
10. Slide flash tube wires out of the light assembly through a rubber grommet located at the center of the assembly.

This completes the Anti-Collision Flash Tube Removal (Incandescent Only) procedure.

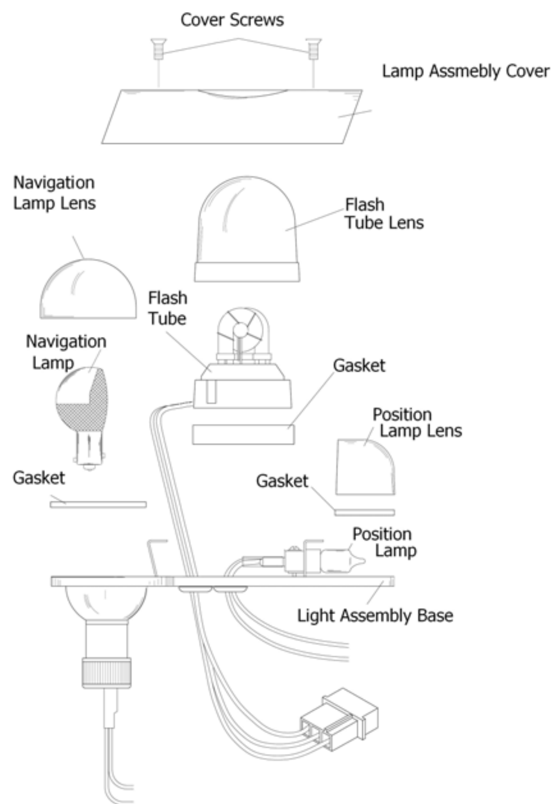

Figure 33-9 Anti-collision light installation

ANTI-COLLISION FLASH TUBE INSTALLATION (INCANDESCENT ONLY)

Perform this procedure to install anti-collision flash tubes. This procedure is for wingtip light assemblies that have a standard incandescent bulb. Procedure applies to port or starboard installations as required.

WARNING

ANTI-COLLISION (STROBE) LIGHT SYSTEMS CAN GENERATE DANGEROUS OR LETHAL VOLTAGES. ENSURE THAT ANTI-COLLISION LIGHT SYSTEM HAS BEEN TURNED OFF FOR AT LEAST 30 MINUTES BEFORE ATTEMPTING TO REMOVE OR REPLACE FLASH TUBES.

NOTE

Skin oils on the surface of position light bulbs or anti-collision light flash tubes will significantly shorten bulb/flash tube life. Wear latex or vinyl gloves when handling bulbs/flash tubes. Bulbs/flash tubes can be cleaned using 70% isopropyl alcohol swabs ("medical prep" swabs).

1. Position aircraft split master switch - OFF
2. Position circuit breaker BAT1 - OPEN
3. Verify the anti-collision light system has been off for a minimum of 30 minutes.
4. Slide the replacement flash tube wire through the light assembly base grommet as shown above.
5. Install braided sleeve over wires.
6. Insert replacement flash tube wire pins in P49 (Port) or P50 (Starboard) as shown above.
7. Connect P49 (Port) to J49 (Port) or P50 (Starboard) to J50 (Starboard) as applicable.
8. Attach wing tip assembly to wing using screws removed previously.
9. If necessary, use 70% isopropyl alcohol to clean flash tube.
10. Secure lens in place by installation of light assembly cover with retaining screws.
11. Position CIRCUIT BREAKER BAT1 – CLOSED
12. Perform functional test as described in the Exterior Lighting Functional Check procedure on page 36 of this chapter.

This completes the Anti-Collision Flash Tube Installation (Incandescent Only) procedure.

ANTI-COLLISION FLASH TUBE REMOVAL (LED ONLY)

Perform this procedure to remove anti-collision flash tubes. This procedure is for wingtip light assemblies that have a super-bright LED bulb. Procedure applies to port or starboard installations as required.

WARNING

ANTI-COLLISION (STROBE) LIGHT SYSTEMS CAN GENERATE DANGEROUS OR LETHAL VOLTAGES. ENSURE THAT ANTI-COLLISION LIGHT SYSTEM HAS BEEN TURNED OFF FOR AT LEAST 30 MINUTES BEFORE ATTEMPTING TO REMOVE OR REPLACE FLASH TUBES.

NOTE

Skin oils on the surface of position light bulbs or anti-collision light flash tubes will significantly shorten bulb/flash tube life. Wear latex or vinyl gloves when handling bulbs/flash tubes. Bulbs/flash tubes can be cleaned using 70% isopropyl alcohol swabs ("medical prep" swabs).

1. Position aircraft split master switch - OFF
2. Position circuit breaker BAT1 - OPEN
3. Verify the anti-collision light system has been off for a minimum of 30 minutes.
4. Remove screw securing position light assembly cover.
5. Remove lens covering anti-collision flash tube.
6. Remove wing tip assembly. Retain screws for reinstallation.

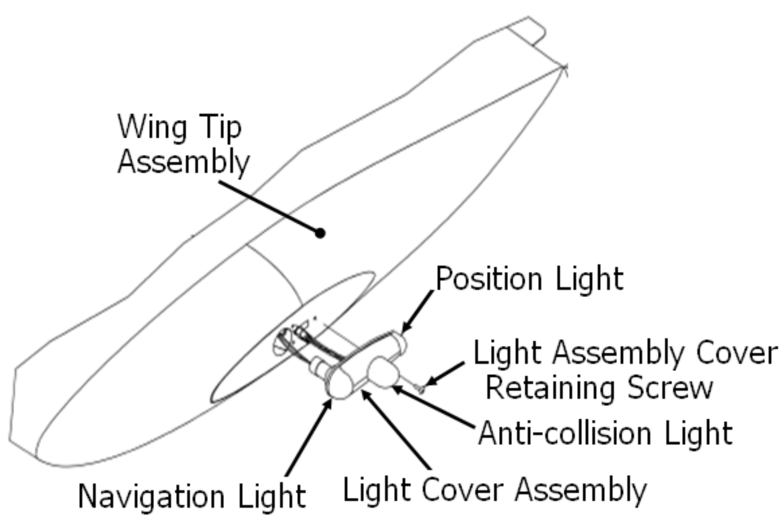

Figure 33-10 Wing Tip Assembly

7. Locate and disconnect P49 (Port) or P50 (Starboard)
8. Remove pins from P49 (Port) or P50 (Starboard)
9. Remove braided sleeve and retain for later use.
10. Slide flash tube wires out of the light assembly through a rubber grommet located at the center of the assembly.

This completes the Anti-Collision Flash Tube Removal (LED Only) procedure.

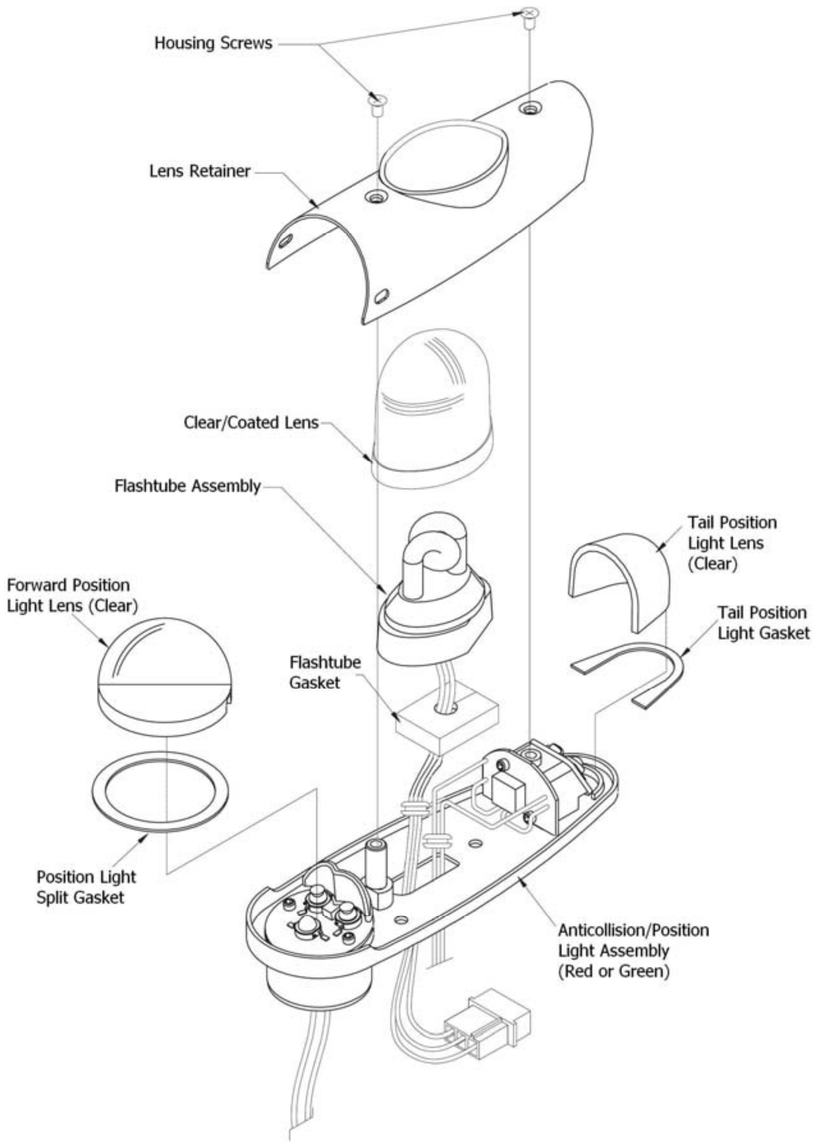

Figure 33-11 Anti-collision light installation

ANTI-COLLISION FLASH TUBE INSTALLATION (LED ONLY)

Perform this procedure to install anti-collision flash tubes. This procedure is for wingtip light assemblies that have a super-bright LED bulb. Procedure applies to port or starboard installations as required.

WARNING

ANTI-COLLISION (STROBE) LIGHT SYSTEMS CAN GENERATE DANGEROUS OR LETHAL VOLTAGES. ENSURE THAT ANTI-COLLISION LIGHT SYSTEM HAS BEEN TURNED OFF FOR AT LEAST 30 MINUTES BEFORE ATTEMPTING TO REMOVE OR REPLACE FLASH TUBES.

NOTE

Skin oils on the surface of position light bulbs or anti-collision light flash tubes will significantly shorten bulb/flash tube life. Wear latex or vinyl gloves when handling bulbs/flash tubes. Bulbs/flash tubes can be cleaned using 70% isopropyl alcohol swabs ("medical prep" swabs).

1. Position aircraft split master switch - OFF
2. Position circuit breaker BAT1 - OPEN
3. Verify the anti-collision light system has been off for a minimum of 30 minutes.
4. Slide the replacement flash tube wire through the light assembly base grommet as shown above.
5. Install braided sleeve over wires.
6. Insert replacement flash tube wire pins in P49 (Port) or P50 (Starboard) as shown above.
7. Connect P49 (Port) to J49 (Port) or P50 (Starboard) to J50 (Starboard) as applicable.
8. Attach wing tip assembly to wing using screws removed previously.
9. If necessary, use 70% isopropyl alcohol to clean flash tube.
10. Secure lens in place by installation of light assembly cover with retaining screws.
11. Position CIRCUIT BREAKER BAT1 – CLOSED
12. Perform functional test as described in the Exterior Lighting Functional Check procedure on page 36 of this chapter.

This completes the Anti-Collision Flash Tube Installation (LED Only) procedure.

LANDING LIGHT REMOVAL

Perform this procedure to remove the landing light.

1. Position aircraft split master switch - OFF
2. Position circuit breaker BAT1 - OPEN
3. Ensure all electrical switches are OFF.
4. Remove the upper and lower engine cowling in accordance with applicable sections of the AMM Chapter 71, *Powerplant.*
5. Disconnect electrical connection P62 from landing light harness.
6. Cut tie wraps that secure the landing light wire harness.
7. Remove screws securing landing light assembly to cowling.
8. Remove landing light assembly.

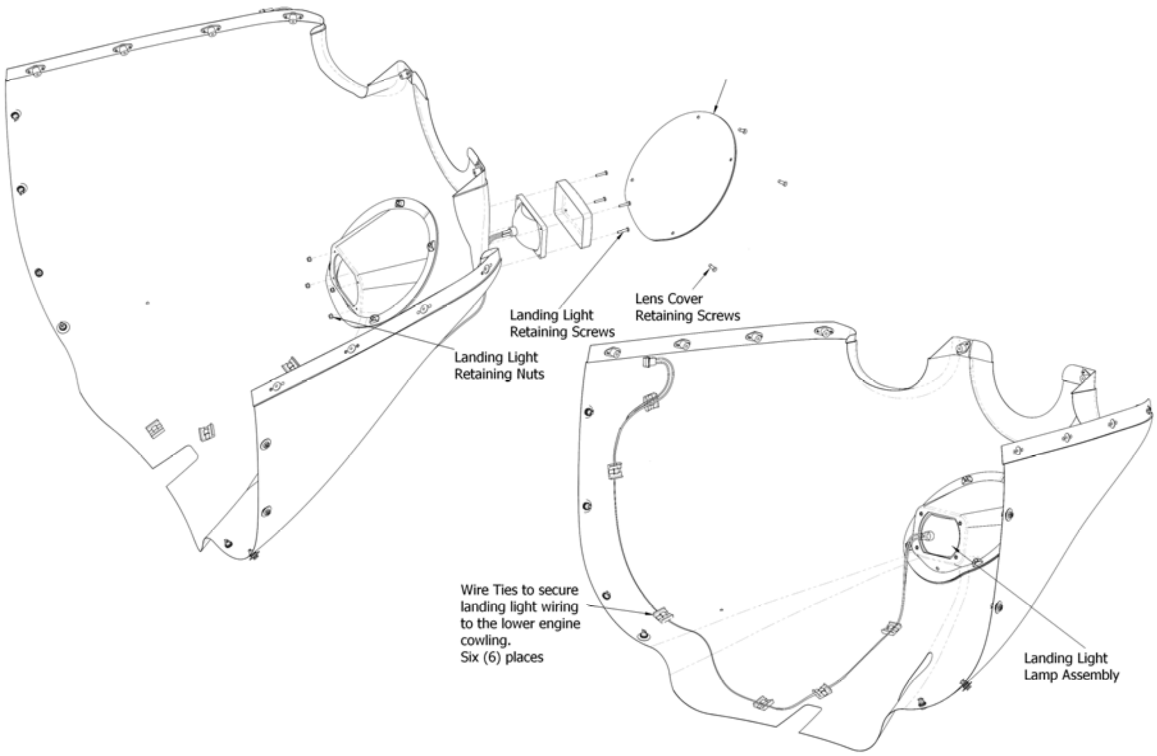

Figure 33-12 Landing Light Installation

This completes the Landing Light Removal procedure.

LANDING LIGHT INSTALLATION

Perform this procedure to install the landing light.

1. Position aircraft split master switch - OFF
2. Position circuit breaker BAT1 - OPEN
3. Install landing light assembly into the lower cowling.
4. Secure with screws and nuts removed previously.
5. Secure landing light wire harness to lower cowl using wire ties in cowl mounted wire anchors.
6. Install lower cowling
7. Connect electrical connection P62 to J62 of the landing light harness
8. Install upper engine cowling in accordance with applicable sections of the AMM Chapter 71, *Powerplant.*
9. Position circuit breaker BAT1 - CLOSED
10. Perform functional test as described in the Exterior Lighting Functional Check procedure on page 36 of this chapter.

This completes the Landing Light Installation procedure.

EXTERIOR LIGHTING FUNCTIONAL CHECK

Perform the following procedure any time exterior lighting systems have been serviced. Refer to the troubleshooting guide to follow for corrective actions as required.

1. Position BAT1 circuit breaker – ON
2. Position LIGHTS, NAV circuit breaker – CLOSED
3. Position LIGHTS, LAND circuit breaker – CLOSED
4. Position LIGHTS STROBE circuit breaker - CLOSED
5. Position aircraft split master switch – ON
6. Position aircraft NAV/POS light switch – ON
7. Verify the LIGHTS, NAV circuit breaker is and remains - CLOSED
8. Verify the port wing tip navigation and position lights are on and steady
9. Verify the starboard wing tip navigation and position lights are on a and steady
10. Position aircraft STROBE light switch – ON
11. Verify the LIGHTS, STROBE circuit breaker is a remains – CLOSED
12. Verify the port wing tip anti-collision light is flashing at a steady rate
13. Verify the starboard wing tip anti-collision light is flashing at a steady rate
14. Position aircraft STROBE light switch – OFF
15. Position aircraft NAV/POS light switch – OFF
16. Position aircraft LANDING light switch – ON
17. Verify the LIGHTS, LAND circuit breaker is and remains – CLOSED
18. Verify landing light is on and steady
19. Position aircraft LANDING light switch to – OFF
20. Position aircraft split master switch - OFF

This completes the Exterior Lighting Functional Check procedure

Section 40-05 Troubleshooting Guide

Table 33-3 is a troubleshooting guide to resolve issues with the position/anti-collision/landing light systems.

Complaint	Possible Cause	Remedy
Any single position light inoperative	defective bulb	replace bulb (incandescent)
		replace the wingtip light assembly (LED)
	defective wiring or connector	repair
Both navigation and position lights on one wing inoperative	2 defective bulbs	replace bulbs (incandescent)
		replace the wingtip light assembly (LED)
	defective wiring or connector	repair
All navigation and position lights inoperative	defective LIGHTS, NAV circuit breaker	replace circuit breaker
	defective LIGHTS, POSITION switch	replace switch
	defective wiring	repair
LIGHTS. NAV circuit breaker tripped and will not reset	defective circuit breaker	replace
	defective wiring	repair
	shorted fixture	replace
	LED fixture paired with Incandescent fixture (2A CB)	replace incandescent fixture with LED fixture
One anti-collision light inoperative	defective flash tube	replace flash tube
	defective high voltage wiring or connector	repair
	defective power supply	replace
Both anti-collision lights inoperative	defective STROBE LIGHTS circuit breaker	replace circuit breaker
	defective power supply	replace power supply
	defective STROBE LIGHTS switch	replace switch
	defective wiring	repair
Landing light inoperative	defective bulb	replace landing light assembly
	defective LANDING LIGHT circuit breaker	replace circuit breaker
	defective LANDING LIGHT switch	replace switch
	defective wiring	repair

Table 33-3 Troubleshooting Guide

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CHAPTER 34

NAVIGATION AND PITOT/STATIC

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Table of Contents

SECTION 34-00 GENERAL 7

SECTION 34-10 FLIGHT ENVIRONMENT DATA 9

SECTION 10-01 PITOT STATIC SYSTEM 9
SECTION 10-02 PERIODIC MAINTENANCE 9
SECTION 10-03 PITOT/STATIC SYSTEM PROCEDURES 9
PITOT/STATIC SYSTEM PURGE 10
STATIC SYSTEM OPERATIONAL CHECK 13
PITOT SYSTEM OPERATIONAL CHECK 15
SECTION 10-04 TROUBLESHOOTING GUIDE 16
SECTION 10-05 ALTIMETER 17
SECTION 10-06 PERIODIC MAINTENANCE 17
SECTION 10-07 ALTIMETER PROCEDURES 17
ALTIMETER REMOVAL 18
ALTIMETER INSTALLATION 20
OPERATION CHECK 21
SECTION 10-08 TROUBLESHOOTING GUIDE 22
SECTION 10-09 TRANSPONDER ALTITUDE ENCODER 23
SECTION 10-10 PERIODIC MAINTENANCE 23
SECTION 10-11 TRANSPONDER ALTITUDE ENCODER PROCEDURES 23
TRANSPONDER ALTITUDE ENCODER REMOVAL 24
TRANSPONDER ALTITUDE ENCODER INSTALLATION 27
TRANSPONDER ALTITUDE ENCODER OPERATIONAL CHECKOUT 30
SECTION 10-12 TROUBLESHOOTING GUIDE 32
SECTION 10-13 VERTICAL SPEED INDICATOR 32
SECTION 10-14 PERIODIC MAINTENANCE 33
SECTION 10-15 VERTICAL SPEED INDICATOR PROCEDURES 33
VERTICAL SPEED INDICATOR REMOVAL 34
VERTICAL SPEED INDICATOR INSTALLATION 36
VERTICAL SPEED INDICATOR OPERATIONAL CHECK 37
SECTION 10-16 TROUBLESHOOTING GUIDE 38
SECTION 10-17 AIRSPEED INDICATOR 38
SECTION 10-18 PERIODIC MAINTENANCE 38
SECTION 10-19 AIRSPEED INDICATOR PROCEDURES 38
AIRSPEED INDICATOR REMOVAL 39
AIRSPEED INDICATOR INSTALLATION 41
AIRSPEED INDICATOR OPERATION CHECKOUT 42
SECTION 10-20 TROUBLESHOOTING GUIDE 43

SECTION 10-21 OUTSIDE AIR TEMPERATURE INDICATOR/SENSOR 43

SECTION 10-22 PERIODIC MAINTENANCE 43
SECTION 10-23 OUTSIDE AIR TEMPERATURE INDICATOR/SENSOR PROCEDURES 43
OUTSIDE AIR TEMPERATURE INDICATOR REMOVAL 44
OUTSIDE AIR TEMPERATURE INDICATOR INSTALLATION 45
OUTSIDE AIR TEMPERATURE SENSOR REMOVAL 46
OUTSIDE AIR TEMPERATURE SENSOR INSTALLATION 47
UNIVERSAL TIME CLOCK SET UP 48
LOCAL TIME SET UP 49
OUTSIDE AIR TEMPERATURE INDICATOR/SENSOR OPERATIONAL CHECK 50
SECTION 10-24 TROUBLESHOOTING GUIDE 51

SECTION 34-20 ATTITUDE AND DIRECTION 53

SECTION 20-01 MAGNETIC COMPASS 53
SECTION 20-02 PERIODIC MAINTENANCE 53
SECTION 20-03 MAGNETIC COMPASS PROCEDURES 53
MAGNETIC COMPASS REMOVAL 54
MAGNETIC COMPASS INSTALLATION 55
MAGNETIC COMPASS AND BRACKET INSTALLATION 56
MAGNETIC COMPASS COMPENSATION (“SWING”) PROCEDURE 57
COMPASS CORRECTION CARD FILLING 59
SECTION 20-04 TROUBLESHOOTING GUIDE 60
SECTION 20-05 ATTITUDE GYRO 61
SECTION 20-06 PERIODIC MAINTENANCE 62
SECTION 20-07 ATTITUDE INDICATOR PROCEDURES 62
ATTITUDE INDICATOR REMOVAL 63
ATTITUDE INDICATOR INSTALLATION 64
ATTITUDE INDICATOR OPERATIONAL CHECK OUT 65
SECTION 20-08 ATTITUDE INDICATOR TROUBLESHOOTING 66
SECTION 20-09 DIRECTIONAL GYRO 67
SECTION 20-10 PERIODIC MAINTENANCE 67
SECTION 20-11 DIRECTIONAL GYRO PROCEDURES 67
DIRECTIONAL GYRO REMOVAL 68
DIRECTIONAL GYRO INSTALLATION 69
DIRECTIONAL GYRO OPERATIONAL CHECKOUT 70
SECTION 20-12 DIRECTIONAL GYRO TROUBLESHOOTING 71
SECTION 20-13 TURN RATE GYRO (TURN COORDINATOR) 73
SECTION 20-14 PERIODIC MAINTENANCE 73
SECTION 20-15 TURN RATE GYRO PROCEDURES 73
TURN RATE GYRO REMOVAL 74
TURN RATE GYRO INSTALLATION 75
TURN RATE GYRO OPERATIONAL CHECKOUT 76
SECTION 20-16 TURN COORDINATOR GYRO TROUBLESHOOTING 77

SECTION 34-30 LANDING AIDS 79

SECTION 30-01 VOR/LOCALIZER RECEIVER 79
SECTION 30-02 PERIODIC MAINTENANCE 79
SECTION 30-03 VOR/LOCALIZER PROCEDURES 79
NAVIGATION RECEIVER REMOVAL 80
NAVIGATION RECEIVER INSTALLATION 81
NAVIGATION ANTENNA REMOVAL 82
NAVIGATION ANTENNA INSTALLATION 84
NAVIGATION SYSTEM OPERATIONAL CHECKOUT 85
SECTION 30-04 TROUBLESHOOTING GUIDE 86
SECTION 30-05 GLIDESLOPE RECEIVER 87
SECTION 30-06 PERIODIC MAINTENANCE 87
SECTION 30-07 GLIDESLOPE RECEIVER PROCEDURES 87
GLIDESLOPE OPERATIONAL CHECKOUT 88
SECTION 30-08 TROUBLESHOOTING GUIDE 89
SECTION 30-09 MARKER BEACON RECEIVER 89
SECTION 30-10 PERIODIC MAINTENANCE 90
SECTION 30-11 MARKER BEACON PROCEDURES 90
MARKER BEACON ANTENNA REMOVAL 91
MARKER BEACON ANTENNA INSTALLATION 92
MARKER BEACON OPERATIONAL CHECKOUT 94
SECTION 30-12 TROUBLESHOOTING GUIDE 95

SECTION 34-50 DEPENDENT POSITION DETERMINING 97

SECTION 50-01 GPS RECEIVER 97
SECTION 50-02 PERIODIC MAINTENANCE 98
SECTION 50-03 GPS RECEIVER PROCEDURES 99
GPS SYSTEM OPERATIONAL CHECKOUT 100
SECTION 50-04 TROUBLESHOOTING GUIDE 101
SECTION 50-05 TRANSPONDER 102
SECTION 50-06 PERIODIC MAINTENANCE 103
SECTION 50-07 TRANSPONDER PROCEDURES 103
TRANSPONDER ANTENNA REMOVAL 104
TRANSPONDER ANTENNA INSTALLATION 105
TRANSPONDER SYSTEM OPERATIONAL CHECKOUT 107
SECTION 50-08 TROUBLESHOOTING GUIDE 108
SECTION 50-09 COURSE DEVIATION INDICATORS 108
SECTION 50-10 PERIODIC MAINTENANCE 110
SECTION 50-11 COURSE DEVIATION INDICATOR PROCEDURES 110
COURSE DEVIATION INDICATOR REMOVAL 111
COURSE DEVIATION INDICATOR INSTALLATION 113
COURSE DEVIATION INDICATOR OPERATIONAL CHECKOUT 114
SECTION 50-12 TROUBLESHOOTING GUIDE 115

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Section 34-00 General

Navigation instruments include those operated by pitot and/or static pressure (airspeed indicator, altimeter, vertical speed indicator); those operated by gyroscopes (attitude and directional gyros, turn rate indicator); and various avionics subsystems used to determine aircraft position, perform enroute navigation, and assist in approach and landing operations.

Figure 34-1 View of the Cockpit Navigation Instruments

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Section 34-10 Flight Environment Data

The Liberty Xl-2 is equipped with a pitot static system providing air data to an airspeed indicator, altimeter, and vertical speed indicator. Additional air data temperature data is provided by means of a solid-state OAT probe and display that provides clock functions. The following sections describe these systems in more detail and provide maintenance procedure for each component.

Section 10-01 Pitot Static System

A combination pitot/static probe installed on the underside of the port wing senses both dynamic (pitot) and static air pressure.

In the event the primary (under-wing) static source becomes blocked, a secondary (cockpit) static source may be selected by moving the alternate static valve, located under the left instrument panel just left of the cockpit center console, to the ON position. See Figure 34-2 for the location of the alternate static valve.

Figure 34-2 Location of Alternate Air Valve

Both dynamic (pitot) and static pressure lines are routed from the under-wing probe; through disconnect fittings near the left wing root, to the rear of the instrument panel. These disconnect fittings can be considered a low point from which any water can be drained from the pitot / static system. In addition, internal water trap loops are provided to ensure that any moisture entering the system does not block lines or reach the instrument panel.

Dynamic (pitot) pressure is plumbed to the airspeed indicator. Static pressure is plumbed to the airspeed indicator, altimeter, vertical speed indicator, and transponder altitude reporting system (altitude encoder).

Section 10-02 Periodic Maintenance

Periodic maintenance of this system entails operational checks and inspections performed at intervals specified in the Chapter 05 – *Time Limits/Maintenance Checks/Inspections Interval* and in accordance with the operational check and inspection procedure in this section.

Section 10-03 Pitot/Static System Procedures

This section contains the procedures for the Pitot/Static System Purge, and Pitot/Static System Leak Check.

PITOT/STATIC SYSTEM PURGE

Perform this procedure to purge the Pitot/Static system.

CAUTION

Use only dry nitrogen (not "shop" compressed air) to purge pitot and/or static lines. To avoid damage to instruments, check the pitot line that it is disconnected from airspeed indicator and that nitrogen purge supply is connected only to static line from instrument panel to left wing static source (i.e., not to static manifold line leading to instruments).

1. Check that all electrical switches are OFF.
2. Place a cover over the pilot yoke control.
3. Have a large block of soft foam on your lap to place the instrument panel on.
4. Remove the ten screws securing the instrument panel to the instrument console. See Figure 34-3 for location of the screws.

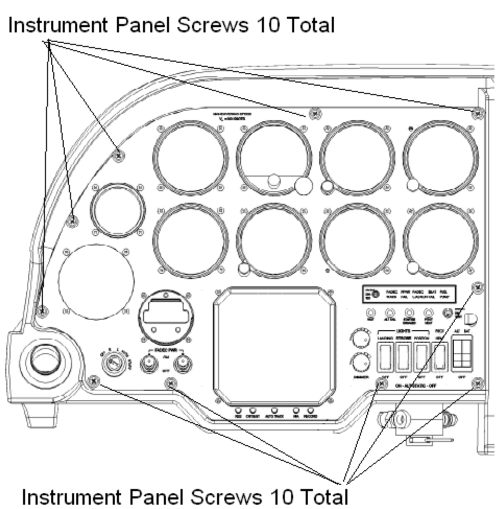


Figure 34-3 Location of the Screws Securing the Instrument Panel

5. Gently pull the instrument panel towards you, placing it face down on to the block of soft foam rubber. The pitot connection (rear of airspeed indicator) and static connections ("T" or "L" junction) connect to the airspeed indicator, altimeter, vertical speed indicator, and altitude encoder). See Figure 34-4 for location of the pitot/static lines.

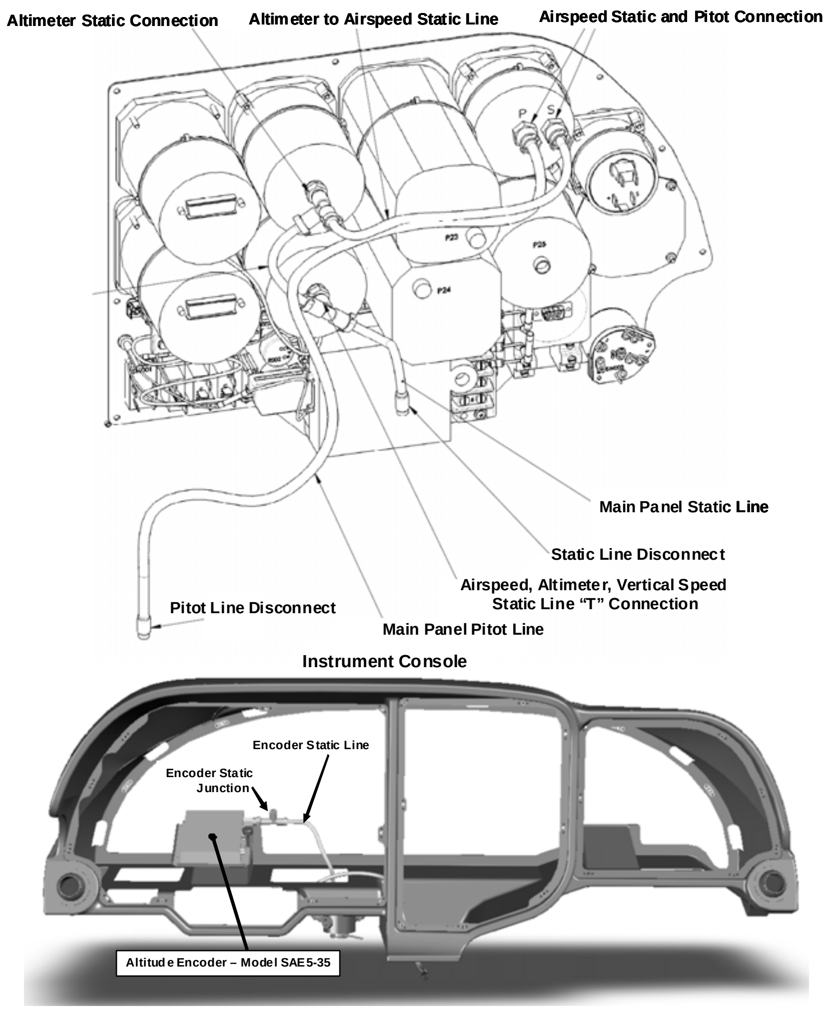


Figure 34-4 Instrument Panel Pitot/Static Connections

6. Disconnect pitot connection from airspeed indicator.
7. Disconnect static connection from junction.
8. Verify proper identification and location of single static connection leading directly to under-wing static probe.
9. Cap the Airspeed Indicator (ASI) port.
10. Cap the Static ports to Altimeter and Vertical Speed Indicator (VSI) encoder.
11. Connect regulated dry nitrogen source to pitot line. Set pressure to 5 psi and allow to flow for at least 3 minutes. Check under-wing pitot tube and verify gas flow.
12. Move regulated dry nitrogen source to static line. Set pressure to 5 psig and allow to flow for at least 3 minutes. Check at static openings on under-wing probe to verify gas flow.
13. Disconnect nitrogen source.
14. Reconnect pitot line to airspeed indicator.

15. Reconnect static line to “T” or “L” junction.
16. Reinstall instrument panel.

Perform the static and pitot leak checks as described in Static System Operational Check on page 13 and This completes the static system operational check procedure.

17. Pitot System Operational Check on page 15 of this chapter.

This completes the Pitot/Static System Purge procedure.

STATIC SYSTEM OPERATIONAL CHECK

This procedure performs an operational check of the static system by testing for system leaks.

CAUTION

The following are precautions that should be taken when performing the static system operational checks:

- *Verification that the pressure inside the pitot system is equal to or greater than the static system*
- *Reversal of tubes can cause damage to the air data instruments they are attached to, check tube orientation*
- *The applied pressure (rate of change of pressure) must not exceed the design limits of the equipment under test*
- *After performing the leak test, check the system that it returns to its normal operating condition*

1. Check the Static lines are free entrapped moisture and restrictions.
2. Evacuate the static pressure system to a pressure differential of approximately 1 inch of mercury or to a reading on the altimeter, 1,000 feet above the aircraft elevation at the time of the test.
3. Without additional pumping for a period of 1 minute, monitor the loss of indicated altitude. To pass the FAR requirements (23.1325 (b)(2)(i)), the loss of altitude must not exceed 100 feet on the altimeter.

CAUTION

Perform steps 5 through 7 in less than 30 seconds. Applying heat to the pitot/static blade heater for longer than 30 second can cause damage to the pitot/static blade or heater.

4. If installed, remove any type of pitot/static blade cover.
5. Apply power to the pitot/static blade heater.
6. Check the operation of the static port heater.
7. Remove power from the pitot/static blade heater.
8. If there is any type of pitot/static blade cover, install cover on to the pitot/static blade.

9. Check for any alterations or deformations of the airframe surface that have been made that would affect the relationship between air pressure in the static pressure system and true ambient static air pressure for any flight condition.

This completes the static system operational check procedure.

PITOT SYSTEM OPERATIONAL CHECK

This procedure performs an operational check of the pitot system by testing for system leaks.

CAUTION

The following are precautions that should be taken when performing the pitot system operational check:

- *Verification that the pressure inside the pitot system is equal to or greater than the static system*
- *Reversal of tubes can cause damage to the air data instruments they are attached to, check tube orientation*
- *The applied pressure (rate of change of pressure) must not exceed the design limits of the equipment under test*
- *After performing the leak test, check the system that it returns to its normal operating condition*

NOTE

Failure to hold pressure for the time interval specified in the following test represents a pitot system fault. Do not proceed until all faults are isolated, corrected and a successful leak test has been accomplished.

1. Check the pitot lines are free entrapped moisture and restriction.
2. For this step, use the AISS Calibrated Pitot-Static Test Set or equivalent; introduce a pressure to the pitot system such that the airspeed indicator registers within the cruise range. (100 Knots Typ.)
3. Hold that pressure for at least one minute. Check for any drop in indicated airspeed. A drop in airspeed indicates a leak in the system.
4. If there are no leaks detected, slowly release the pressure applied to the system allowing the pressure within the instrument to slowly return to ambient.

This completes the pitot system operational check procedure.

Section 10-04 Troubleshooting Guide

This section has the troubleshooting guide. Refer to Table 34-1 for the troubleshooting guide.

Complaint	Possible Cause	Remedy
Airspeed indicator inaccurate or erratic (other static instruments OK)	Blocked or contaminated pitot connection	• Drain / purge
	Pitot or static leak	• Repair
	Defective instrument	• Replace
Other static air operated instrument(s) inaccurate or erratic	Blocked or contaminated static connection	• Drain / purge
	Defective instrument(s)	• Replace

Table 34-1 Troubleshooting Guide for the Pitot/Static Lines

Section 10-05 Altimeter

The altimeter is a sensitive absolute pressure gauge referenced to standard sea level pressure. It can be adjusted to compensate for local changes in barometric pressure. It has three pointers: the large pointer indicates hundreds of feet above sea level, the smaller pointer reads thousands of feet above sea level, and the narrow line pointer (with a triangle at the outer periphery of the altimeter dial) reads tens of thousands of feet above sea level.

The adjustment knob and barometric pressure-setting window allow the pilot to set the altimeter to correspond with local (sea level equivalent) barometric pressure. The aircraft is fitted with a dual scale altimeter permitting display or barometric setting in Inches of Mercury (IN Hg) and millibars (mb).

Figure 34-5 Altimeter

Section 10-06 Periodic Maintenance

Periodic maintenance of this unit entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 10-07 Altimeter Procedures

This section contains the altimeter removal and installation procedures. This section also includes the operational checkout and troubleshooting of the altimeter.

ALTIMETER REMOVAL

Perform this procedure to remove the altimeter.

1. Check that all electrical switches are OFF.
2. Place a cover over the pilot yoke control.
3. Have a large block of soft foam on your lap to place the instrument panel on.
4. Remove the ten screws securing the instrument panel to the instrument console. See Figure 34-6 for location of the screws.

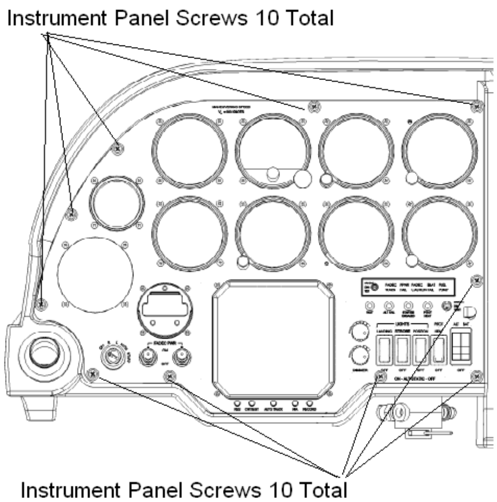


Figure 34-6 Location of the Screws Securing the Instrument Panel

5. Gentle pull the instrument panel towards you, placing it face down on to the block of soft rubber.
6. Disconnect static line (green). See Figure 34-7 for location of the Altimeter and the green static line.

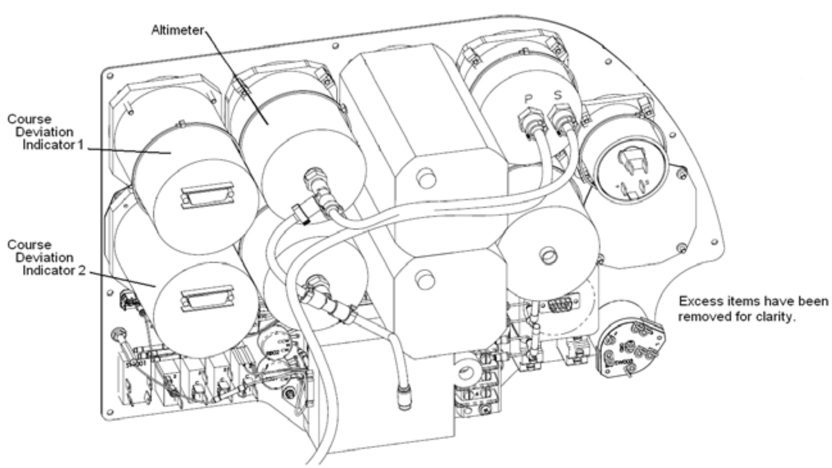


Figure 34-7 Rear View of the instrument Panel Showing Location of the Altimeter

7. Install caps on to the end of the static line and on to the static port on the altimeter to prevent moisture from entering into the static line or altimeter.
8. Remove (4) panel screws and remove instrument.
9. Secure the cutout (bezel) light assembly to the panel to prevent damage to the assembly.
10. If the installation of the altimeter will take place later, install the instrument panel in to the instrument console. Secure the instrument panel with the ten screws removed in step 4 above.

This completes the Altimeter Removal procedure.

ALTIMETER INSTALLATION

Perform this procedure to install the altimeter into the instrument panel.

1. Check that all electrical switches are OFF.
2. If installing a new altimeter immediately after removing an old altimeter, then proceed to step 8 below.
3. Place a cover over the pilot yoke control.
4. Have a large block of soft foam on your lap to place the instrument panel on.
5. Remove the ten screws securing the instrument panel to the instrument console. See Figure 34-6 for location of the screws.
6. Gentle pull the instrument panel towards you, placing it face down on to the block of soft rubber.
7. Prepare the cutout (bezel) light assembly to allow for the installation of the altimeter.
8. Install altimeter and cutout (bezel) light assembly to the instrument panel and secure them to panel with the four instrument screws.
9. Remove the static line caps from the altimeter and green static line.
10. Connect static line (green) to the altimeter, ensuring correct connection.
11. Install instrument panel into the instrument console.
12. Secure the instrument panel with the ten instrument panel screws.
13. Perform operation check Operation Check procedure on page 21.

This completes the Altimeter Installation procedure.

OPERATION CHECK

Perform the following procedure to check the operation of the altimeter.

1. Check the Static lines are free entrapped moisture and restrictions.
2. Evacuate the static pressure system to a pressure differential of approximately 1.0 inch of mercury or to a reading on the altimeter of 1,000 feet above the aircraft elevation at the time of the test.
3. Without additional pumping for a period of 1 minute, monitor the loss of indicated altitude. To pass the FAR requirements (23.1325 (b)(2)(i)), the loss of altitude must not exceed 100 feet on the altimeter.

This completes the Operation Check procedure.

Section 10-08 Troubleshooting Guide

Table 34-2 is a troubleshooting guide to resolve issues the altimeter.

Complaint	Possible Cause	Remedy
Altimeter indicator inaccurate or erratic (other static instruments OK)	Blocked or contaminated static connection	• Drain / purge
	Static leak	• Repair
	Defective instrument	• Replace

Table 34-2 Altimeter Troubleshooting Guide

Section 10-09 Transponder Altitude Encoder

The altitude encoder is an altimeter device that provides an electronic signal ("gray code") to the transponder for use in its Mode C altitude reporting function. The signal provided by the encoder, and radiated by the transponder when responding to Mode C interrogations, is "pressure altitude," i.e., it is not corrected for local variations in barometric pressure. Instead, its output is electronically adjusted for local conditions before being displayed to Air Traffic Controllers.

The SAE5-35 is a solid state -1000 to 35,000-foot altitude data system that converts pressure altitude into a digital output as set forth in the International Standard for SSR Pressure Altitude Transmission. The data output of the SAE5-35 is referenced to 29.92 inch HG (1013 Milli-bars). The SAE5-35 has been designed to provide altitude data to GPS and Terrain Awareness Systems in addition to Mode C Transponders. The SAE5-35 uses RS-232-C serial ports to output altitude data to the transponder and to two other systems requiring this format, such as a GPS receiver.

Figure 34-8 Altitude Encoder

Section 10-10 Periodic Maintenance

Periodic maintenance of this unit entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 10-11 Transponder Altitude Encoder Procedures

This section contains the removal and installation procedures for the transponder altitude encoder. This section also contains operational checkout procedures and troubleshooting for the transponder altitude encoder.

TRANSPONDER ALTITUDE ENCODER REMOVAL

Perform the following procedure to remove the transponder altitude encoder from the airplane. Liberty Aerospace, Inc. recommends to remove the instrument panel from the airplane while working on the transponder altitude encoder. Have the block of soft foam cushioning in your lap that it covers the yoke assembly.

1. Move all electrical switches to OFF.
2. Remove the ten screws that secure the Flight Instrument Panel assembly to the console. Refer to Figure 34-9 for the location of these screws.

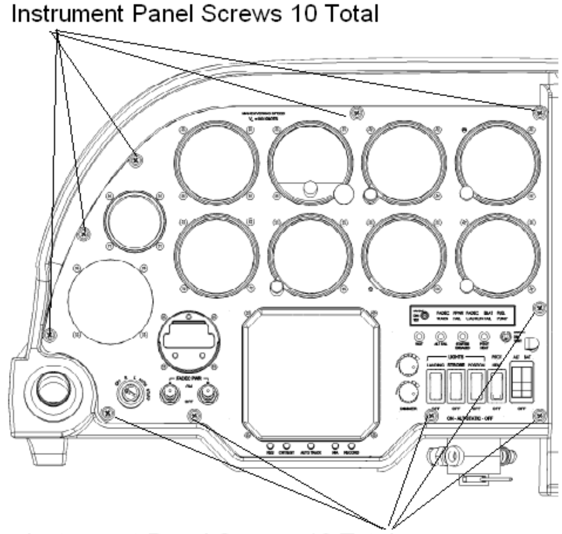


Figure 34-9 Instrument Panel Assembly Showing the Location of the Screws

3. Bring this panel towards you and place it on its face in to the block of soft foam cushioning that is in your lap. If the instrument panel has one or two Course Deviation Indicators (CDI), disconnect the electrical connectors on the back of the CDI. (P/J106-1 and/or P/J106-2) See Figure 34-10 for the location of P/J106-1 and/or P/J106-2. Disconnect the ground wire from the CDI cable(s) from the instrument panel grounding point.

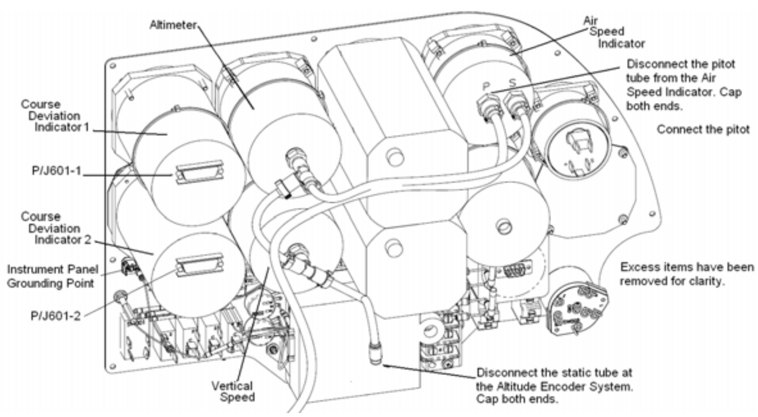


Figure 34-10 Rear View of the Instrument Panel

4. Disconnect the pitot line (violet tubing) from the Airspeed indicator. Disconnect the static line (green tubing) at the altitude encoder system. Cap all lines and/or indicators. For the location of the pitot and static lines, and where to disconnect these lines, see Figure 34-10 and Figure 34-11. Disconnect the single wire connector at P/J69. See Figure 34-13 for the location of P/J69.

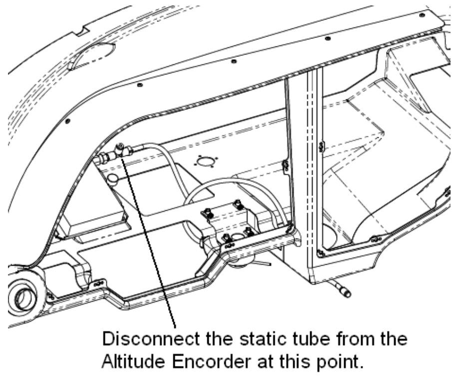


Figure 34-11 Location of the Altitude Encoder System

5. Disconnect the ribbon cable from the VM1000FX display. See Figure 34-12 for the location of the connector for the VM1000FX.

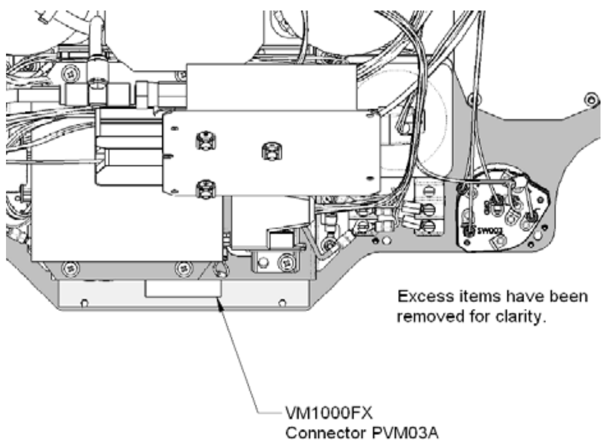


Figure 34-12 Location of the Connector on the VM1000FX

6. Disconnect P03, P10, and P15 from their mating connectors on the power distribution bracket. See Figure 34-13 for location of P/J03, P/J10, and P/J15.

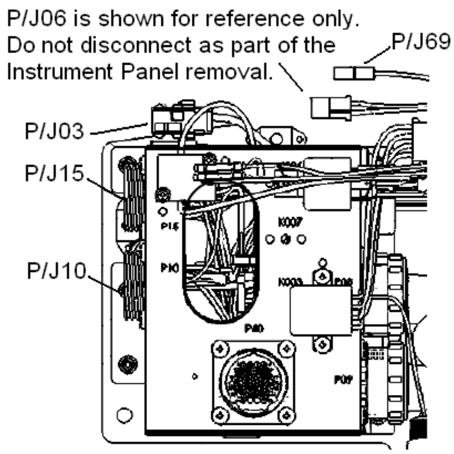


Figure 34-13 Left Hand Side of the Power Distribution Harness Assembly

7. Remove the instrument panel from the airplane keeping the instrument panel on its face into a block of soft foam cushioning.

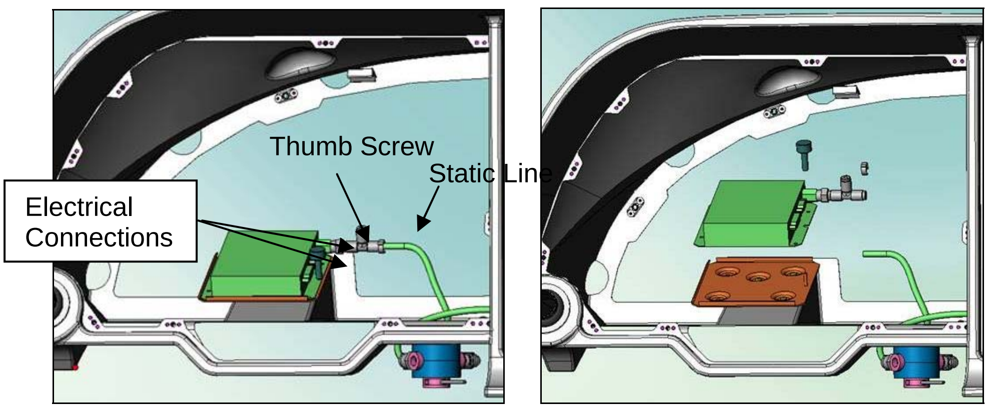


Figure 34-14 Encoder Removal

8. Disconnect static (green) line.
9. Disconnect the 9-pin D-Sub connector, P5355, and the 15-pin D-Sub connector, P5354, from the encoder box.
10. Remove encoder hold down thumbscrew and remove unit from mounting tray assembly.

This completes the Transponder Altitude Encoder Removal procedure.

TRANSPONDER ALTITUDE ENCODER INSTALLATION

Perform this procedure to install the transponder altitude encoder.

1. If the instrument panel is in the instrument console, then do step 1 through step 7 of the Transponder Altitude Encoder Removal procedure on page 24 of this chapter.
2. Install encoder by engaging hold down rail and tightening the thumbscrew.
3. Connect static (green) line to the static line port.
4. Connect the 9-pin cable connector, P5355, from the avionics panel to the 9-pin connector on the encoder box.
5. Connect the 15-pin cable connector, P5354, from the avionics panel to the 15-pin connector on the encoder box.

NOTE

Verify connector screws are fully engaged with the encoder body. Do not over torque.

6. Retrieve the Instrument panel removed in the procedure to Transponder Altitude Encoder Removal on page 24 of this chapter.
7. Connect P03, P10, and P15 to their mating connectors on the power distribution bracket. See Figure 34-15 for location of P/J03, P/J10, and P/J15.

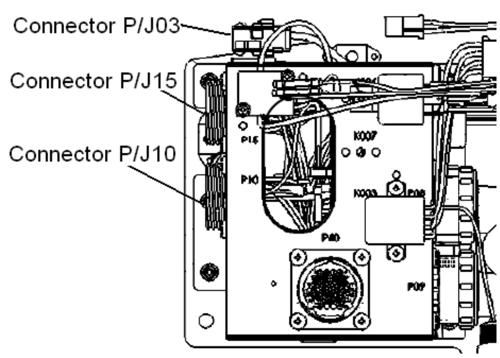


Figure 34-15 Left Hand Side of the Power Distribution Harness Assembly

8. Connect the ribbon cable to the VM1000FX display. See Figure 34-16 for the location of the connector on the VM1000FX.

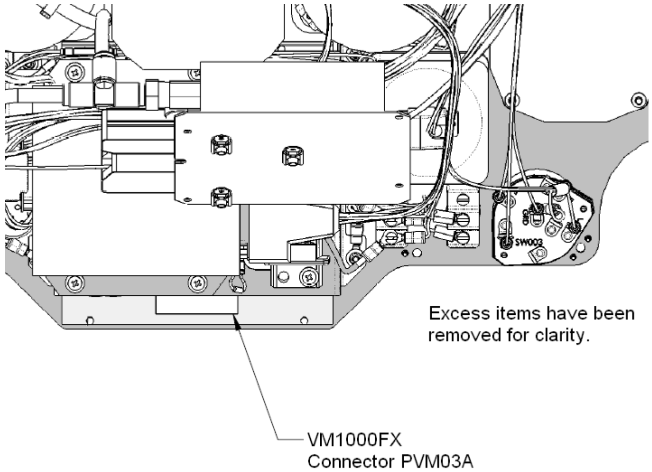


Figure 34-16 Location of the Connector on the VM1000FX

9. If the instrument panel has one or two Course Deviation Indicators (CDI), connect the electrical connectors on the back of the CDI. (P/J106-1 and/or P/J106-2) See Figure 34-17 for the location of the connectors P/J106-1 and/or P/J106-2. Connect the ground shields from the CDI cables to the grounding point on the instrument panel.
10. Remove the caps from the pitot and static lines and the Airspeed indicator. Connect the pitot line to the Airspeed indicator. Connect the static line to the Altitude Encoder system.

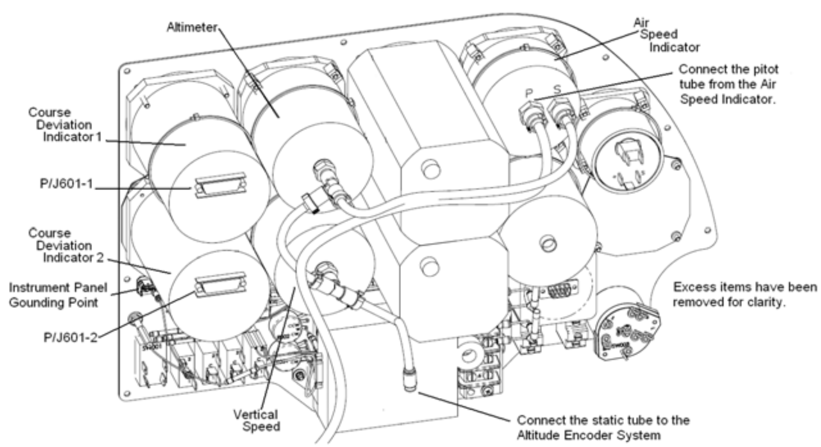


Figure 34-17 Rear of Instrument Panel Showing the Locations for Connecting the Pitot and Static Lines

11. Install the panel into the opening in the instrument panel console assembly.
12. Install the ten screws to secure the instrument panel to the console. See Figure 34-18 for the location of the ten screws.

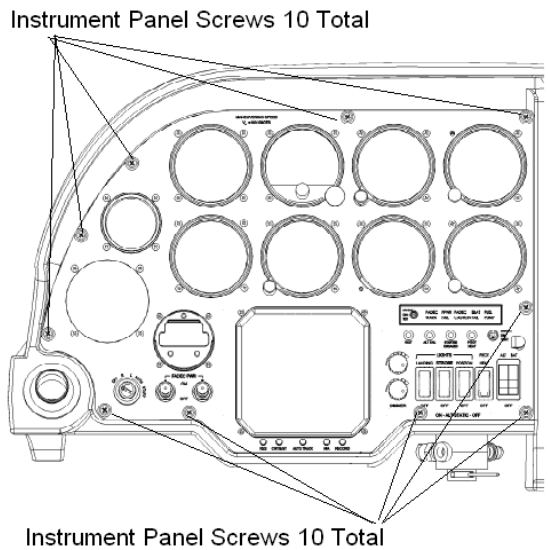


Figure 34-18 Location of the Ten Screws Securing The Instrument Panel

13. Perform Transponder Altitude Encoder Operational Check on page 30 of this chapter.

NOTE

The factory calibrates the SAE5-35 to a pressure datum traceable to the National Bureau of Standards. However, when installing the encoder in the aircraft, a certified repair station must recalibrate the encoder to correspond to the primary flight altimeter before returning the airplane to service. Further, every 24 calendar months both, the encoder and primary altimeter requires recalibration. This calibration ensures the altitude code generated from the SAE5-35 is within 125 feet of the altitude displayed to the pilot. Refer to 14 CFR 91.217 and 91.413 for details.

14. Perform calibration of the encoder system by means of a certified repair station in compliance with 14 CFR 91.217 and 91.413.

This completes the Transponder Altitude Encoder Installation procedure.

TRANSPONDER ALTITUDE ENCODER OPERATIONAL CHECKOUT

Perform this procedure to do an operational checkout of the Transponder Altitude Encoder.

1. Perform Static System Operational Check procedure starting on page 13.
2. Perform Pitot System Operational Check procedure starting on page 14.

NOTE

The following operation check may be performed at any time for the purpose of verifying the encoder is functioning. The procedure does not replace calibration by a certified repair station as required in 14 CFR Part 91.413.

3. Connect an AISS Calibrated Pitot-Static Test Set or equivalent to the aircraft static system.
4. Position the aircraft master switch to the ON position.
5. Position the avionics master switch to the ON position.
6. Apply power to the altitude encoder and the ATC transponder by pressing the transponder ON button as shown in Figure 34-19 below.

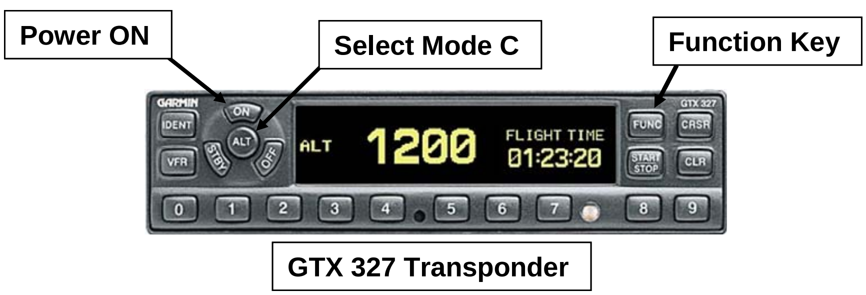


Figure 34-19 GTX327 Transponder

7. Set the transponder for Mode C operation by pressing the ALT button until "ALT" is displayed in the status window as shown in Figure 34-19.
8. Using the function key, configure the transponder to display "PRESSURE ALT" as shown in Figure 34-20 below. Altitude is presented in 100s of feet.

Figure 34-20 Transponder Pressure Altitude Display

NOTE

The Primary Flight Altimeter must be calibrated per part 43 section E. Verify compliance with AC-43.13 as applicable.

9. Set primary flight altimeter to 29.92 inches of mercury.
10. Set the TKM Michel 3300 ATC Transponder Test Set or equivalent, to read altitude (ALT) from the transponder system.
11. Apply vacuum from the pitot-static test set to obtain an altimeter reading of 19,900 feet.
12. Verify the transponder displays “FL 199” +/- 100 feet.
13. Using the Transponder Test Set, verify difference between altimeter and digital encoder reported altitude is within 125 feet.
14. Slowly decrease altitude stopping at 5000 foot intervals to verify encoder and altimeter remain within the 125 foot tolerance. Continue to decrease altitude until field elevation is reached.
15. Position the avionics master switch to the OFF position.
16. Position the aircraft master switch to the OFF position.
17. Disconnect an AISS Calibrated Pitot-Static Test Set or equivalent from the aircraft static system.

This completes the Transponder Altitude Encoder Operational Check procedure.

Section 10-12 Troubleshooting Guide

Table 34-3 and Table 34-4 have the troubleshooting information for the transponder altitude encoder. The instruction manual noted in Table 34-3 must be the latest versions of the OEM manuals listed below:

MODEL	DOCUMENT	TITLE	NOTES
SAE5-35	305186-00	Installation Manual	Available at www.sandia.aero

Table 34-3 Installation Manual for the Transponder Altitude Encoder

Complaint	Possible Cause	Remedy
Altitude not reported	defective altitude encoder defective encoder cable Encoder not powered	replace repair or replace Power on encoder
Unit fails to power up	Transponder circuit breaker open	Investigate cause of trip and reset on resolution
Reported altitude drift or erratic	Static line leak	Seal leak and perform static system leak test
Reported altitude steady but incorrect	Calibration fault	Calibrate encoder

Table 34-4 Troubleshooting Guide for the Transponder Altitude Encoder

Section 10-13 Vertical Speed Indicator

The vertical speed indicator is a sensitive differential pressure gauge referenced to an air chamber with a small controlled leak to static pressure. It indicates the rate of ascent or descent of the airplane through the atmosphere.

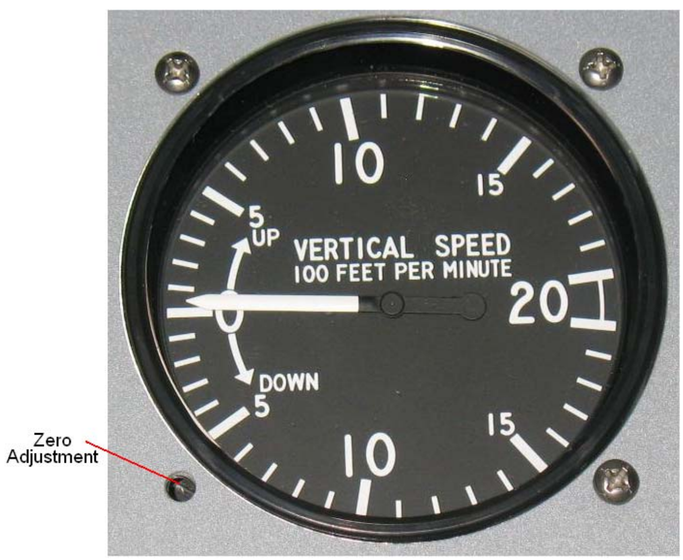


Figure 34-21 Vertical Speed Indicator

Section 10-14 Periodic Maintenance

Periodic maintenance of this unit entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 10-15 Vertical Speed Indicator Procedures

This section contains the removal and installation procedures for the vertical speed indicator. This section also contains operational checkout and troubleshooting for the vertical speed indicator.

VERTICAL SPEED INDICATOR REMOVAL

Perform this procedure to remove the vertical speed indicator.

1. Check that all electrical switches are OFF.
2. Place a cover over the pilot yoke control.
3. Have a large block of soft foam on your lap to place the instrument panel on.
4. Remove the ten screws securing the instrument panel to the instrument console. See Figure 34-22 for location of the screws.

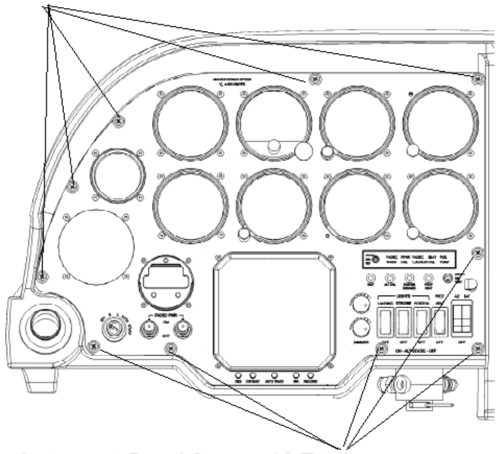


Figure 34-22 Location of the Screws Securing the Instrument Panel

5. Gentle pull the instrument panel towards you, placing it face down on to the block of soft rubber.
6. Disconnect static line (green) from the vertical speed indicator. See Figure 34-23 for location of the vertical speed indicator and the green static line.

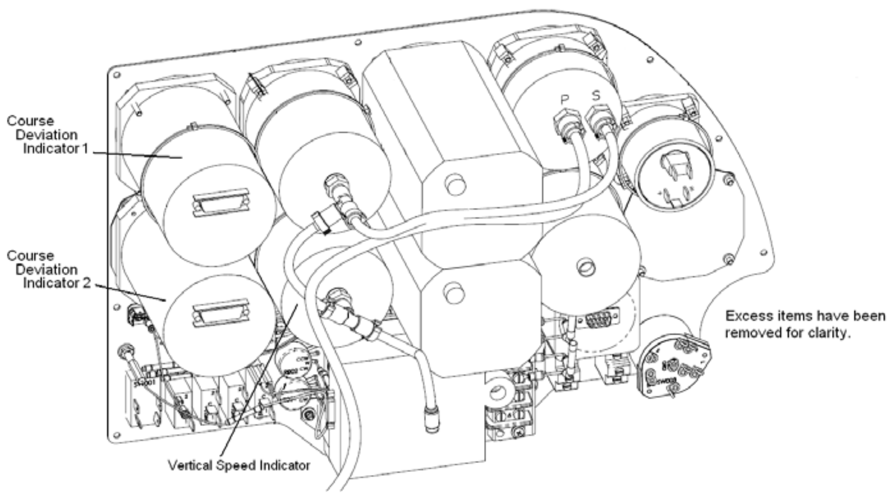


Figure 34-23 Rear View of the instrument Panel Showing Location of the Vertical Speed Indicator

7. Install caps on to the end of the static line and on to the static port on the vertical speed indicator to prevent moisture from entering into the static line or vertical speed indicator.
8. Remove (4) panel screws and remove instrument.
9. Secure the cutout (bezel) light assembly to the panel to prevent damage to the assembly.
10. If the installation of the vertical speed indicator will take place later, install the instrument panel in to the instrument console. Secure the instrument panel with the ten screws removed in step 4 above.

This completes the Vertical Speed Indicator Removal procedure.

VERTICAL SPEED INDICATOR INSTALLATION

Perform this procedure to install the altimeter into the instrument panel.

1. Check that all electrical switches are OFF.
2. If installing a new airspeed indicator immediately after removing an old vertical speed indicator, then proceed to step 8 below.
3. Place a cover over the pilot yoke control.
4. Have a large block of soft foam on your lap to place the instrument panel on.
5. Remove the ten screws securing the instrument panel to the instrument console. See Figure 34-22 for location of the screws.
6. Gentle pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
7. Prepare the cutout (bezel) light assembly to allow for the installation of the vertical speed indicator.
8. Install vertical speed indicator and cutout (bezel) light assembly to the instrument panel and secure them to panel with the four instrument screws.
9. Remove the static line caps from the vertical speed indicator and green static line.
10. Connect static line (green) to the vertical speed indicator, ensuring correct connection.
11. Install instrument panel into the instrument console.
12. Secure the instrument panel with the ten instrument panel screws.
13. Perform operation check Vertical Speed Indicator Operational Check procedure on page 37.

This completes the Vertical Speed Indicator Installation procedure.

VERTICAL SPEED INDICATOR OPERATIONAL CHECK

Perform the following procedure to check the operation of the vertical speed indicator, VSI.

1. Referring to Figure 34-21 above, set VSI zero indication with adjustment screw shown.
2. Check that static lines are free entrapped moisture and restrictions.
3. Evacuate the static pressure system to a pressure differential of approximately 1.0 inch of mercury or to a reading on the altimeter of 1,000 feet above the aircraft elevation at the time of the test.
4. Without additional pumping for a period of 1 minute, monitor the loss of indicated altitude. To pass the FAR requirements (23.1325 (b)(2)(i)), the loss of altitude must not exceed 100 feet on the altimeter.
5. On leak check completion, verify and adjust VSI zero indication.

This completes the Vertical Speed Indicator Operational Check procedure.

Section 10-16 Troubleshooting Guide

This section presents the troubleshooting guide.

Complaint	Possible Cause	Remedy
VSI indicator inaccurate or erratic (other static instruments OK)	blocked or contaminated static connection	drain / purge
	static leak	repair
	defective instrument	replace
VSI fails to indicate zero vertical speed in level flight	Indicator out of zero adjustment	On the ground, set instrument to zero vertical speed.

Table 34-5 VSI Troubleshooting Guide

Section 10-17 Airspeed Indicator

The airspeed indicator is a sensitive pressure gauge, which reads the difference between dynamic (pitot) and static pressure. This information (indicated airspeed) must be corrected for deviations from standard pressure and temperature (29.92 in. Hg. sea level pressure, 15 deg. C) to yield true airspeed.

Figure 34-24 Airspeed Indicator

Section 10-18 Periodic Maintenance

Periodic maintenance of this unit entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 10-19 Airspeed Indicator Procedures

This section contains the removal and installation procedures for the airspeed indicator. This section also contains operational checkout and troubleshooting information for the airspeed indicator.

AIRSPEED INDICATOR REMOVAL

Perform this procedure to remove the vertical speed indicator.

1. Check that all electrical switches are OFF.
2. Place a cover over the pilot yoke control.
3. Have a large block of soft foam on your lap to place the instrument panel on.
4. Remove the ten screws securing the instrument panel to the instrument console. See Figure 34-22 for location of the screws.

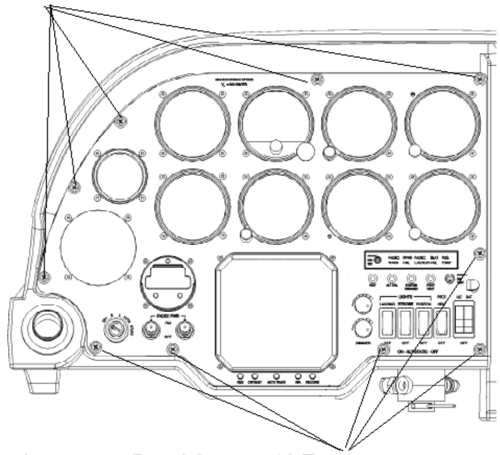


Figure 34-25 Location of the Screws Securing the Instrument Panel

5. Gentle pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
6. Disconnect static line (green) and the pitot line (purple), from the airspeed indicator. See Figure 34-26 for location of the airspeed indicator and the green static line.

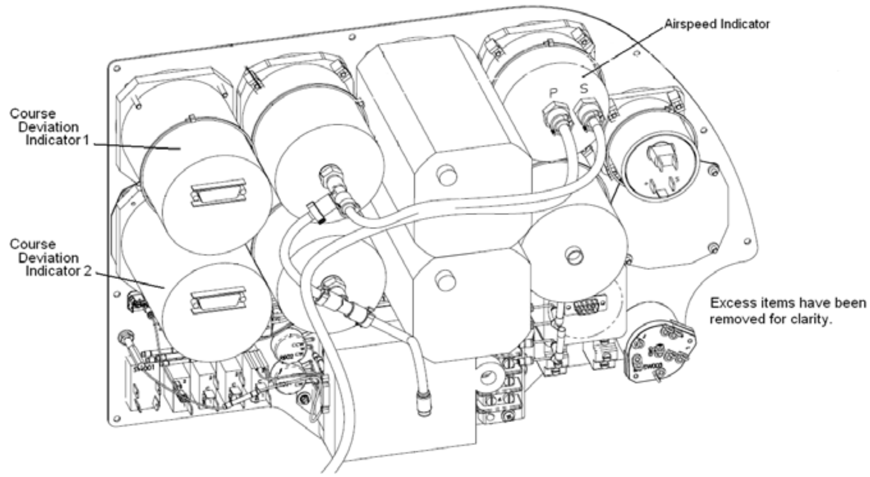


Figure 34-26 Rear View of the instrument Panel Showing Location of the Airspeed Indicator

7. Install caps on to the end of the pitot and static line and on to the pitot and static port on the airspeed indicator to prevent moisture from entering into the pitot and static line or airspeed indicator.
8. Remove (4) panel screws and remove instrument.
9. Secure the cutout (bezel) light assembly to the panel to prevent damage to the assembly.
10. If the installation of the airspeed indicator will take place later, install the instrument panel in to the instrument console. Secure the instrument panel with the ten screws removed in step 4 above.

This completes the Airspeed Indicator Removal procedure.

AIRSPEED INDICATOR INSTALLATION

Perform this procedure to install the altimeter into the instrument panel.

1. Check that all electrical switches are OFF.
2. If installing a new airspeed indicator immediately after removing an old airspeed indicator, then proceed to step 8 below.
3. Place a cover over the pilot yoke control.
4. Have a large block of soft foam on your lap to place the instrument panel on.
5. Remove the ten screws securing the instrument panel to the instrument console. See Figure 34-25 for location of the screws.
6. Gentle pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
7. Prepare the cutout (bezel) light assembly to allow for the installation of the vertical speed indicator.
8. Install airspeed indicator and cutout (bezel) light assembly to the instrument panel and secure them to panel with the four instrument screws.
9. Remove the pitot and static line caps from the airspeed indicator, purple pitot and green static line.
10. Connect static line (green) to the airspeed indicator, ensuring correct connection.
11. Connect pitot line (purple) to the airspeed indicator, ensuring correct connection.
12. Install instrument panel into the instrument console.
13. Secure the instrument panel with the ten instrument panel screws.
14. Perform operation check Airspeed Indicator Operation Checkout procedure on page 42.

This completes the Airspeed Indicator Installation procedure.

AIRSPEED INDICATOR OPERATION CHECKOUT

Perform this procedure to check the operation of the airspeed indicator.

1. Check the Static lines are free entrapped moisture and restrictions.
2. Check the pitot lines are free entrapped moisture and restriction.
3. Evacuate the static pressure system to a pressure differential of approximately 1.0 inch of mercury or to a reading on the altimeter of 1,000 feet above the aircraft elevation at the time of the test.
4. Without additional pumping for a period of 1 minute, monitor the loss of indicated altitude. To pass the FAR requirements (23.1325 (b)(2)(i)), the loss of altitude must not exceed 100 feet on the altimeter.
5. For this step, use the AISS Calibrated Pitot-Static Test Set or equivalent; introduce a pressure to the pitot system such that the airspeed indicator registers within the cruise range. (100 Knots Typ.)
6. Hold that pressure for at least one minute. Check for any drop in indicated airspeed. A drop in airspeed indicates a leak in the system.
7. If there are no leaks detected, slowly release the pressure applied to the system allowing the pressure within the instrument to slowly return to ambient.

This completes the Airspeed Indicator Operation Checkout procedure.

Section 10-20 Troubleshooting Guide

This section presents the troubleshooting guide.

Complaint	Possible Cause	Remedy
Airspeed indicator inaccurate or erratic (other static instruments OK)	Blocked or contaminated pitot connection	• Drain / purge
	Pitot or static leak	• Repair
	Defective instrument	• Replace

Table 34-6 Airspeed Indicator Troubleshooting Guide

Section 10-21 Outside Air Temperature Indicator/Sensor

The airplane is equipped with a Davtron model M803 electric Outside Air Temperature indicator, consisting of a temperature-sensing element and an instrument panel indicator. The sensor is positioned in the main fuselage directly aft of the left wing flap. Outside air temperature indicator is located under the avionics stack in the center instrument panel. The instrument also provides clock, stopwatch and bus voltage features.

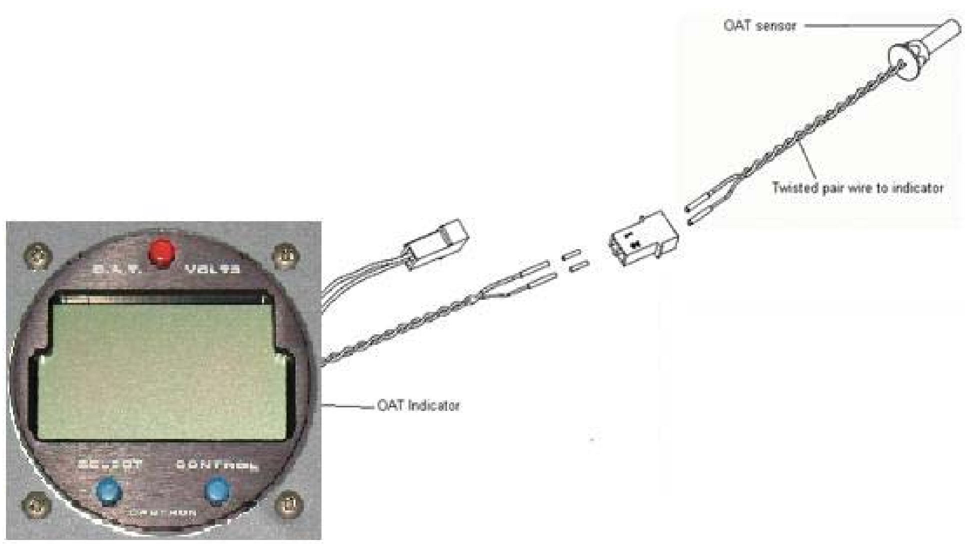


Figure 34-27 OAT Indicator and OAT Sensor

Section 10-22 Periodic Maintenance

Periodic maintenance of this unit entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 10-23 Outside Air Temperature Indicator/Sensor Procedures

This section contains the removal, installation, and operational check procedures for the Outside Air Temperature indicator and sensor.

OUTSIDE AIR TEMPERATURE INDICATOR REMOVAL

Perform this procedure to remove the Outside Air Temperature indicator.

1. Check that all electrical switches are OFF.
2. Place a cover over the pilot yoke control.
3. Have a large block of soft foam on your lap to place the instrument panel on.
4. Remove the ten screws securing the instrument panel to the instrument console. See Figure 34-28 for location of the screws.

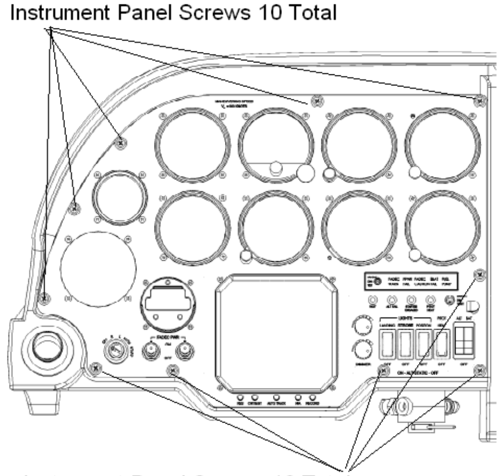


Figure 34-28 Location of the Screws Securing the Instrument Panel

5. Gentle pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
6. Do not to disturb electrical connections, pitot, or static lines.
7. Remove (4) panel screws and remove instrument and disconnect electrical connectors.
8. If the installation of the OAT/Clock indicator will take place later, install the instrument panel in to the instrument console. Secure the instrument panel with the ten screws removed in step 4 above.

This completes the Outside Air Temperature Indicator Removal procedure.

OUTSIDE AIR TEMPERATURE INDICATOR INSTALLATION

Perform this procedure to install the Outside Air Temperature/Clock, OAT/Clock, indicator.

1. Check that all electrical switches are OFF.
2. If installing a new OAT/Clock indicator immediately after removing an old OAT/Clock indicator, then proceed to step 7 below.
3. Place a cover over the pilot yoke control.
4. Have a large block of soft foam on your lap to place the instrument panel on.
5. Remove the ten screws securing the instrument panel to the instrument console. See Figure 34-28 for location of the screws.
6. Gentle pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
7. Install OAT/Clock indicator to the instrument panel and secure them to panel with the four instrument screws.
8. Connect J56 of cable harness to OAT indicator receptacle
9. Install instrument panel into the instrument console taking care not to disturb electrical connections, pitot or static lines.
10. Secure the instrument panel with the ten instrument panel screws.
11. Perform operation check and clock set up located in Outside Air Temperature Indicator/Sensor Operational Check procedure on page 50, the Universal Time Clock Set Up procedure on page 48, and the Local Time Set Up on page 49 of this chapter.

This completes the Outside Air Temperature Indicator Installation procedure.

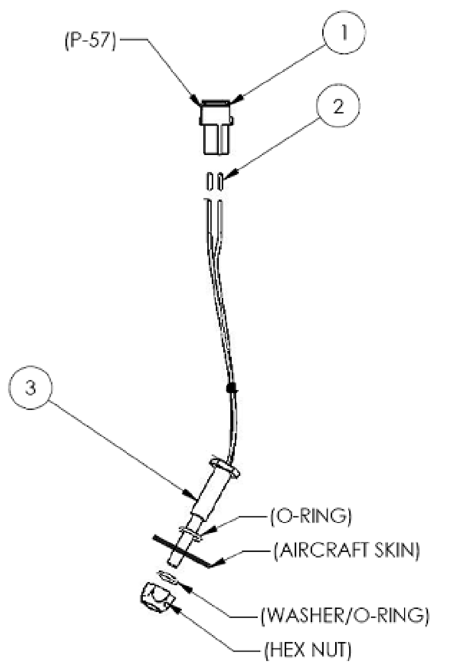
OUTSIDE AIR TEMPERATURE SENSOR REMOVAL

Perform this procedure to remove the Outside Air Temperature sensor.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position aircraft master switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Remove cabin baggage compartment floor access panel.
4. Disconnect temperature probe connector P57
5. Remove the retaining nut and washer on the outside of the fuselage.



ITEM NO.	QTY.	PART NO.	DESCRIPTION
1	1	1-480319	2 CIRCUIT MATE-N-LOK PIN HOUSING
2	2	60618-1	PIN CONTACT
3	1	AD590	DIGITAL CLOCK/OAT SENSOR

Figure 34-29 OAT Sensor Installation

6. Remove the OAT sensor.

This completes the Outside Air Temperature Sensor Removal procedure.

OUTSIDE AIR TEMPERATURE SENSOR INSTALLATION

Perform this procedure to remove the Outside Air Temperature sensor.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position aircraft master switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Remove cabin baggage compartment floor access panel.
4. Insert OAT sensor in fuselage hole.
5. Replace the washer and retaining nut on the outside fuselage.
6. Connect OAT electrical connector P57
7. Replace aft cargo bay bulkhead closeout.
8. Perform operation check and clock set up located in Outside Air Temperature Indicator/Sensor Operational Check procedure on page 50, the Universal Time Clock Set Up procedure on page 48, and the Local Time Set Up on page 49 of this chapter.

This completes the Outside Air Temperature Sensor Installation procedure.

UNIVERSAL TIME CLOCK SET UP

Perform this procedure to set up the Universal Time on the clock. Refer to Figure 34-30 during this procedure.

1. Press SEL button to select and display Universal Time (UT)
2. Simultaneously press SEL and CTL buttons to enter set mode
3. Verify tens of hours digit is flashing
4. Using the CTL button, set the current time digit
5. Press SEL button for the next digit
6. Repeat steps d an e for each digit
7. Press SEL to exit the clock set mode of operation
8. Verify lighted annunciator is flashing indicating normal operation

This completes the Universal Time Clock Set Up procedure.

LOCAL TIME SET UP

Perform this procedure to set the local time on the clock. Refer to Figure 34-30 during this procedure.

1. Press SEL button to select and display Local Time (LT)
2. Simultaneously press SEL and CTL buttons to enter set mode
3. Verify tens of hours digit is flashing
4. Using the CTL button, set the current time digit
5. Press SEL button for the next digit
6. Repeat steps 4 and 5 for each digit
7. Press SEL to exit the clock set mode of operation
8. Verify lighted annunciator is flashing indicating normal operation
9. Local Time clock set up complete

This completes the Local Time Set Up procedure.

OUTSIDE AIR TEMPERATURE INDICATOR/SENSOR OPERATIONAL CHECK

Perform this procedure to perform an operational check of the OAT Indicator and sensor. Refer to Figure 34-30 during this procedure.

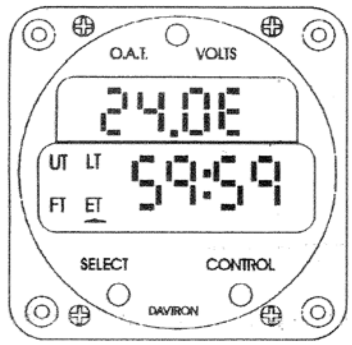


Figure 34-30 OAT / CLOCK Controls

1. From the flight instrument panel position the aircraft master switch ON
2. Verify display illuminates.
3. Press and hold the SEL button for three (3) seconds.
4. Verify display indicates 88:88 and all four annunciators are active.
5. Press the OAT/VOLTS button.
6. Verify temperature is displayed in Fahrenheit and agrees with known independent source.
7. Press the OAT/VOLTS button.
8. Verify temperature is displayed in Centigrade and agrees with known independent source.
9. Press the OAT/VOLTS button.
10. Verify bus A voltage is displayed.

This completes the Outside Air Temperature Indicator/Sensor Operational Check procedure.

Section 10-24 Troubleshooting Guide

This section has the troubleshooting information on the OAT/Clock indicator and OAT sensor.

MODEL	DOCUMENT	TITLE	NOTES
M803	M803	Installation Manual	Available at www.davtron.com

Table 34-7 Installation Manual for the OAT/Clock System

Complaint	Possible Cause	Remedy
Temperature reading in error	Temperature probe connection fault Defective temperature probe Defective indicator	Repair or replace connection Replace probe Replace indicator
Blank display	OAT CB open Defective indicator	Check for fault correct and close breaker Replace indicator
No voltage reading	Defective indicator	replace
Clock error	Fuse F1 open	Inspect for short, repair and replace fuse

Table 34-8 Troubleshooting Guide for the OAT/Clock System

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Section 34-20 Attitude and Direction

The airplane's standard equipment includes a gyroscopic attitude indicator (gyro horizon), a gyroscopic directional indicator (directional gyro), a gyroscopic turn rate indicator (turn coordinator), and a magnetic compass. All of the gyroscopic instruments are electrically operated and are powered by individual 2-amp circuit breakers on the main distribution bus. The magnetic compass requires no power for operation. If replacing one of these instruments, consult the data plate for correct type and setting.

Section 20-01 Magnetic Compass

The magnetic compass is mounted at the top of the windshield and consists of a rotating magnetic element immersed in damping liquid. It requires no power for operation (although an internal electric light is provided for night flying).

The compass is equipped with internal adjusting magnets to allow it to be compensated for local magnetic distortions (installation of equipment, electromagnetic fields from onboard equipment, etc.).

NOTE

There are two types of magnetic compasses available. One is for the northern hemisphere and the other is for the southern hemisphere. The part numbers for both compasses are the same, except the compass for the southern hemisphere has a –SH suffix added to the part number.

Section 20-02 Periodic Maintenance

Periodic maintenance of this unit entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 20-03 Magnetic Compass Procedures

This section contains the removal and installation of the magnetic compass. This section also contains the procedure to "swing" the compass.

MAGNETIC COMPASS REMOVAL

Perform this procedure to remove the magnetic compass.

1. Check that all aircraft electrical power is OFF
2. Disconnect J06 compass light connector from P06 and remove wire from windshield hoop.

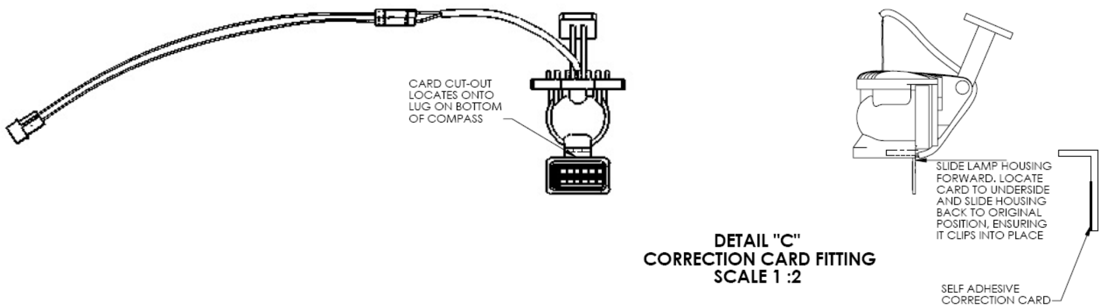

Figure 34-31 Compass Installation

3. Loosen and remove compass mounting screw and nut located where the compass body and mounting post meet.

This completes the Magnetic Compass Removal procedure.

MAGNETIC COMPASS INSTALLATION

Perform this procedure to install ONLY the magnetic compass installation.

NOTE

Ensure that the compass being installed is suitable for the hemisphere of operation. A compass equipped for the southern hemisphere will have a part number ending in-SH

1. Install compass body to mounting post by means of mounting screw and nut. Level the compass body and tighten the hardware until the compass body is secure from movement. Do not over tighten the hardware.
2. Install compass card as shown in Figure 34-31 above.
3. Connect J06 compass light to P06 in the instrument panel
4. Dress out compass light wire to windscreen hoop and secure with wire tie.
5. Swing compass Magnetic Compass Compensation ("Swing") Procedure on page 57 of this chapter.

This completes the Magnetic Compass Installation procedure.

MAGNETIC COMPASS AND BRACKET INSTALLATION

Perform this procedure to install the magnetic compass and compass bracket installation.

1. Check that the mounting surface is not less than 10 degrees C.
2. Using the cleaning cloth contained in the sachet provided, clean the area of the windshield where the compass is to be mounted. (do not touch this area after cleaning)
3. Carefully remove the plastic film from the base of the mounting bracket. (Do not touch the adhesive pad).
4. Position the compass and bracket on to the mounting surface, making sure that it is square and true to the aircraft.
5. Press very lightly to the mounting surface and check that the location is correct.
6. When you are sure the location is correct, press hard and maintain pressure for approximately 30 seconds.
7. Do not stress the mounting for 24 hours. This will allow the adhesive bond to fully cure.
8. Install compass card as shown in Figure 34-31 on page 54 of this chapter.
9. Connect J06 compass light to P06 in the instrument panel
10. Dress out compass light wire to windscreen hoop and secure with wire tie.
11. Swing compass Magnetic Compass Compensation ("Swing") Procedure on page 57 of this chapter.

This completes the Magnetic Compass and Bracket Installation procedure.

MAGNETIC COMPASS COMPENSATION ("SWING") PROCEDURE

Perform this procedure to perform the magnetic compass compensation. This procedure should be carried out if erratic compass function is suspected, or any time significant changes are made in the airplanes installed electrical equipment.

Required Equipment:

- Aircraft with compass installed
- Approved compass rose
- Corrector key
- One maintenance technician outside the aircraft
- One maintenance technician inside the aircraft

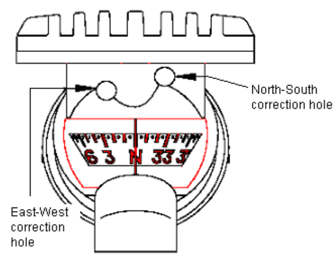

Figure 34-32 View of Compass Showing Correction Holes

NOTE

Ensure that the compass being calibrated is suitable for the hemisphere of operation. A compass equipped for the southern hemisphere will have a part number ending in-SH

1. Start the engine and taxi the aircraft to the north (0º) radial on the compass rose (master compass). With the aircraft facing north and the person in the cockpit running the engine at 1000 rpm, and with radios "ON", a maintenance technician, standing approximately 30 feet in front of the aircraft and facing south, aligns the master compass with the aircraft center line.

2. Using hand signals, the maintenance technician signals the person in the cockpit to make additional adjustments to align the aircraft with the master compass. Once aligned on the heading, the person in the cockpit runs the engine to approximately 1,700 rpm to duplicate the aircraft's magnetic field in flight and then he/she reads the compass.

3. If the aircraft compass is not in alignment with the magnetic north of the master compass, insert the corrector key into the right hand hole at the front of the compass and rotate the key in either direction so that the north point is directly under the index line. Align the aircraft on the compass rose facing east. If the aircraft compass is not in alignment with magnetic east, insert the corrector key, into the left hand hole at the front of the compass and rotate the key in either direction so that the east point is directly under the index line. Continue by aligning the aircraft on the compass rose to180º and adjust the right hand screw to remove one-half of the south's heading error. This will throw the north off, but the total north-south error should be divided equally between the two headings. Turn the aircraft until it is heading west (270º) and adjust the left hand screw on the compensator to remove one-half of the west error. This should divide equally the total east-west error.
4. With the aircraft heading west, start calibration data. Make recordings to be transferred to the compass correction card when finished. Record the magnetic heading of 270º and the compass reading with the avionics systems "ON" then "OFF". If there is a significant difference (>9º) between the two readings at each heading, two compass cards will need to be installed, one marked "Radios "ON"', the other marked "Radios 'OFF'". Turn the aircraft to align with each of the lines on the compass rose and record the compass readings every 30º. There should be not more than a plus or minus 10º difference between any of the compass rose headings and the magnetic headings of the aircraft.
5. If the compass cannot be adjusted to meet requirements, replace the compass.
6. The compass correction card is graduated in 30-degree increments with cardinal points at N, S, E & W. Because of the limited space available, the markings between the cardinal points should be multiplied by a factor of 10. i.e.: 3 = 30º degrees, 24 = 240º degrees etc.
7. When the compass is satisfactorily swung, fill out the compass correction card completely and install it.

This completes the Magnetic Compass Compensation ("Swing") Procedure.

COMPASS CORRECTION CARD FILLING

The correction card, supplied with the compass, is to be fitted to the underside of the compass. Withdraw the lower housing as shown and clip the card into place. Replace the lower housing when finished. Make sure it clips into place.

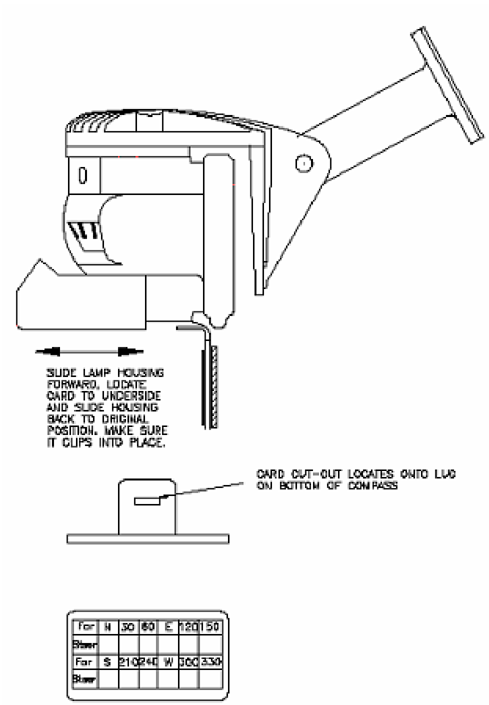

Figure 34-33 Compass Correction Card Installation

Magnetic Compass Calibration Form			
Aircraft Tail Number:		**Date:**	
Radios ON		**Radios OFF**	
Master Compass	**Aircraft Compass**	**Master Compass**	**Aircraft Compass**
360		360	
030		030	
060		060	
090		090	
120		120	
150		150	
180		180	
210		210	
240		240	
270		270	
300		300	
330		330	

Table 34-9 Magnetic Compass Calibration Form

Section 20-04 Troubleshooting Guide

This section has the troubleshooting information.

Complaint	Possible Cause	Remedy
Compass error	Magnetic interference	• Remove sources of magnetic interference
	Compass not level	• Level Compass
	Defective compass	• Replace
	Compass not swung	• Swing Compass
Low compass oil	Leak in compass seal	• Replace seal • Swing compass

Table 34-10 Magnetic Compass Troubleshooting Table

Section 20-05 Attitude Gyro

The attitude gyro is installed in the top center of the instrument panel.

Figure 34-34 Attitude Gyro

Movement of the brown and blue “earth and sky” attitude display behind the fixed pointer displays aircraft pitch and roll attitude changes to the pilot. A red “off” flag appears in the display to alert the pilot of insufficient electrical power to the attitude gyro or insufficient gyro wheel speed; appearance of this flag is normal before the airplane electrical system is powered on for engine start, for up to one minute after initial gyro startup, or after aircraft shutdown.

Cage/pitch adjustment knobs are provided on the instrument. Rotating the pointer knob adjusts the vertical position of the pointer to accommodate changes in aircraft cruise pitch attitude and/or pilot eye position. Pulling the cage knob away from the panel temporarily “cages” the attitude gyro in a wings level/level pitch attitude position. This is normally necessary only when the gyro is initially energized in order to rapidly synchronize it with the airplane’s normal ground attitude. The knob is spring-loaded and will return to its normal position, thus un-caging the gyro, when it is released.

NOTE

An automatic erection mechanism in the attitude gyro will synchronize it with the local vertical after initial startup. However, this process may require more time than manually caging the gyro after initial power-up.

Maintenance procedures for the attitude gyro are limited to removal and replacement of a defective instrument.

NOTE

When replacing the attitude gyro, ensure that the replacement unit is the exact same part number or determine that any replacement attitude gyro is designed for installation in an instrument panel tilted six degrees in level flight. Installation of other attitude gyros, or units designed for a different panel tilt, will result in inaccurate indication of aircraft pitch attitude.

Section 20-06 Periodic Maintenance

Periodic maintenance of this unit entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 20-07 Attitude Indicator Procedures

This section contains the removal, installation, and operational check procedures for the attitude indicator.

ATTITUDE INDICATOR REMOVAL

Perform this procedure to remove the attitude indicator.

1. Check that all electrical switches are OFF.
2. Place a cover over the pilot yoke control.
3. Have a large block of soft foam on your lap to place the instrument panel on.
4. Remove the ten screws securing the instrument panel to the instrument console. See Figure 34-35 for location of the screws.

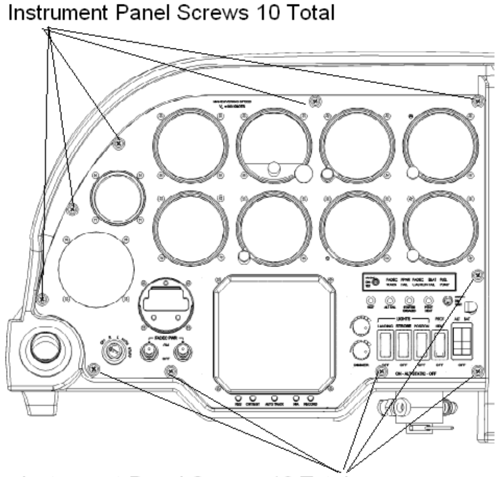

Figure 34-35 Location of the Screws Securing the Instrument Panel

5. Gentle pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
6. Do not to disturb electrical connections, pitot or static lines.
7. Remove (4) panel screws and remove attitude indicator and disconnect electrical connector.
8. Secure the cutout (bezel) light assembly to the panel to prevent damage to the assembly.
9. If the installation of the attitude indicator will take place later, install the instrument panel in to the instrument console. Secure the instrument panel with the ten screws removed in step 4 above.

This completes the Attitude Indicator Removal procedure.

ATTITUDE INDICATOR INSTALLATION

Perform this procedure to install the attitude Indicator.

1. Check that all electrical switches are OFF.
2. If installing a new attitude gyro indicator immediately after removing an old attitude gyro indicator, then proceed to step 7 below.
3. Place a cover over the pilot yoke control.
4. Have a large block of soft foam on your lap to place the instrument panel on.
5. Remove the ten screws securing the instrument panel to the instrument console. See Figure 34-35 for location of the screws.
6. Gentle pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
7. Prepare the cutout (bezel) light assembly to allow for the installation of the attitude indicator.
8. Install attitude indicator to the instrument panel and secure them to panel with the four instrument screws.
9. Connect attitude indicator cable.
10. Install instrument panel into the instrument console taking care not to disturb electrical connections, pitot or static lines.
11. Secure the instrument panel with the ten instrument panel screws.
12. Perform operation check described in Attitude Indicator Operational Check Out on page 65 of this chapter.

This completes the Attitude Indicator Installation procedure.

ATTITUDE INDICATOR OPERATIONAL CHECK OUT

Perform this procedure to do an operation check of the attitude indicator.

1. Place the following aircraft circuit breakers in the position indicated.

Circuit Breaker Section	Circuit Breaker	Position
SYSTEM	GYRO	CLOSED
INSTRUMENTS	ATT	CLOSED
INSTRUMENTS	DG	OPEN
INSTRUMENTS	TURN	OPEN
BAT1		CLOSED

Table 34-11 Circuit Breaker Condition Pre-Checkout

2. Verify/position the aircraft level in pitch and roll axis.
3. Position the aircraft master switch to ON.
4. Run attitude gyro for a period of 3 minutes and verify the following:

- Gyro flag pulls clear
- Gyro indicates level in pitch and roll axis
- Gyro motor operation is smooth and free of chatter or vibration

5. Position the aircraft master switch to OFF.
6. Place the following aircraft circuit breakers in the position indicated.

Circuit Breaker Section	Circuit Breaker	Position
SYSTEM	GYRO	CLOSED
INSTRUMENTS	ATT	CLOSED
INSTRUMENTS	DG	CLOSED
INSTRUMENTS	TURN	CLOSED
BAT1		CLOSED

Table 34-12 Circuit Breaker Condition Post Checkout

This completes the Attitude Indicator Operational Check Out procedure.

Section 20-08 Attitude Indicator Troubleshooting

This section has the attitude indicator troubleshooting information.

Complaint	Possible Cause	Remedy
Attitude gyro inoperative (OFF flag visible)	Defective attitude gyro circuit breaker	• Replace circuit breaker
	Defective instrument	• Replace instrument
	Defective wiring	• Repair
Attitude gyro indications inaccurate or erratic (OFF flag not in view)	Defective attitude gyro	• Replace
	Low bus voltage	• Isolate electrical fault and repair

Section 20-09 Directional Gyro

The directional gyro is installed directly below the attitude gyro and indicates the airplane's heading.

Figure 34-36 Directional Gyro Indicator

Due to the primary trait of gyroscopic rigidity in space, the directional gyro provides much more stable indications in accelerated or turning flight, or in turbulence, than the magnetic compass. However, the directional gyro does not sense the Earth's magnetic field, but simply displays changes in direction relative to the position in which the gyro element first stabilized (became rigid) upon initial power application.

Therefore, an adjustment knob is provided. When pushed in and rotated, the knob manually adjusts the directional gyro to correspond with the magnetic heading indicated by the magnetic compass. The knob is spring-loaded to return to its initial position, and disengage the adjustment mechanism, when it is released.

Over a period, the directional gyro will exhibit precession, or "drift," away from its initial setting. Excessive rates of drift, i.e., >3 degrees in 15 minutes, indicates that the gyro should be replaced.

A red "off" flag appears in the directional gyro display to alert the pilot of insufficient electrical power to the attitude gyro or insufficient gyro wheel speed; appearance of this flag is normal before the airplane electrical system is powered on for engine start, for up to one minute after initial gyro startup, or after aircraft shutdown.

Section 20-10 Periodic Maintenance

Periodic maintenance of this unit entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 20-11 Directional Gyro Procedures

This section details the instructions to remove and install the directional gyro, DG. This section also provides the operational checkout of the DG post installation.

DIRECTIONAL GYRO REMOVAL

Perform this procedure to remove the directional gyro.

1. Check that all electrical switches are OFF.
2. Place a cover over the pilot yoke control.
3. Have a large block of soft foam on your lap to place the instrument panel on.
4. Remove the ten screws securing the instrument panel to the instrument console. See Figure 34-37 for location of the screws.

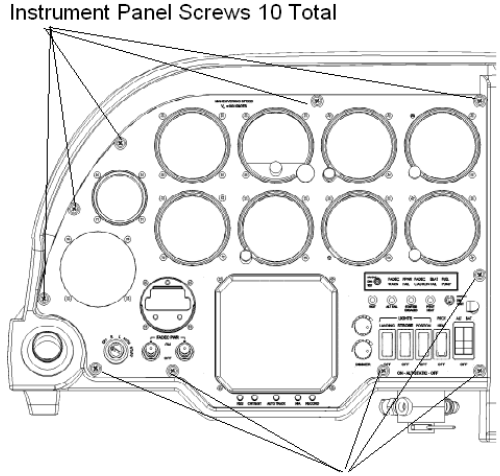

Figure 34-37 Location of the Screws Securing the Instrument Panel

5. Gentle pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
6. Do not to disturb electrical connections, pitot or static lines.
7. Remove (4) panel screws and remove directional gyro indicator and disconnect electrical connector.
8. Secure the cutout (bezel) light assembly to the panel to prevent damage to the assembly.
9. If the installation of the directional gyro will take place later, install the instrument panel in to the instrument console. Secure the instrument panel with the ten screws removed in step 4 above.

This completes the Directional Gyro Removal procedure.

DIRECTIONAL GYRO INSTALLATION

Perform this procedure to install the directional gyro.

1. Check that all electrical switches are OFF.
2. If installing a new directional gyro indicator immediately after removing an old directional gyro indicator, then proceed to step 7 below.
3. Place a cover over the pilot yoke control.
4. Have a large block of soft foam on your lap to place the instrument panel on.
5. Remove the ten screws securing the instrument panel to the instrument console. See Figure 34-37 for location of the screws.
6. Gentle pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
7. Prepare the cutout (bezel) light assembly to allow for the installation of the directional gyro indicator.
8. Install directional gyro indicator to the instrument panel and secure them to panel with the four instrument screws.
9. Connect directional gyro indicator cable.
10. Install instrument panel into the instrument console taking care not to disturb electrical connections, pitot or static lines.
11. Secure the instrument panel with the ten instrument panel screws.
12. Perform the Directional Gyro Operational Checkout procedure on page 70 of this chapter.

This completes the Directional Gyro Installation procedure.

Directional Gyro Operational Checkout

Perform this procedure to do an operational checkout of the directional gyro.

1. Place the following aircraft circuit breakers in the position indicated.

Circuit Breaker Section	Circuit Breaker	Position
SYSTEM	GYRO	CLOSED
INSTRUMENTS	ATT	OPEN
INSTRUMENTS	DG	CLOSED
INSTRUMENTS	TURN	OPEN
BAT1		CLOSED

Table 34-13 Circuit Breaker Condition Pre Checkout

2. Verify/position the aircraft level in pitch and roll axis
3. Position the aircraft master switch to ON.
4. Run directional gyro for a period of 3 minutes and verify the following:

- Gyro flag pulls clear
- Gyro holds set heading without drifting
- Gyro motor operation is smooth and free of chatter or vibration.

5. Position the aircraft master switch to OFF.
6. Place the following aircraft circuit breakers in the position indicated.

Circuit Breaker Section	Circuit Breaker	Position
SYSTEM	GYRO	CLOSED
INSTRUMENTS	ATT	CLOSED
INSTRUMENTS	DG	CLOSED
INSTRUMENTS	TURN	CLOSED
BAT1		CLOSED

Table 34-14 Circuit Breaker Condition Post Checkout

This completes the Directional Gyro Operational Checkout procedure.

Section 20-12 Directional Gyro Troubleshooting

This section contains the troubleshooting guide for the directional gyro.

Complaint	Possible Cause	Remedy
Directional gyro inoperative (OFF flag visible)	Defective directional gyro circuit breaker	• Replace circuit breaker
	Defective instrument	• Replace instrument
	Defective wiring	• Repair
Directional gyro precession ("drift") rate excessive (>3 deg. In 15 minutes)	Defective instrument	• Replace instrument

Table 34-15 Troubleshooting Guide for the Directional Gyro

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Section 20-13 Turn Rate Gyro (Turn Coordinator)

The turn coordinator is mounted to the left of the directional gyro. This instrument employs a tilted gyro rotor mounted with one degree of freedom. Motions associated with turns of the aircraft (both the initial bank to start a turn and the steady yaw rate of an established turn) cause the gyro to precess, displacing a calibrated rate spring and tilting the pointer visible to the pilot. The displacement is proportional to the rate of aircraft heading change, with a reference mark on the instrument face indicating a standard rate turn (3 degrees per second) in either direction. No pitch information is provided.

Figure 34-38 Turn Coordinator Indicator

A red flag appears in the turn coordinator display (just above the right side of the pointer) to alert the pilot of insufficient electrical power to the turn coordinator; appearance of this flag is normal before the airplane electrical system is powered on for engine start, or after aircraft shutdown. A bank indicator (“inclinometer”), consisting of a metal ball moving in a curved glass tube filled with damping liquid, indicates the airplane bank attitude relative to the resultant “local vertical” determined by the airplane’s rate of turn. No power is required for operation of the inclinometer. An expansion chamber above one end of the glass tube accommodates any air bubbles that may form in the damping liquid. In the unlikely event that severe turbulence or rapid taxi over rough ground causes an air bubble to appear in the inclinometer tube, it can be removed by removing the instrument from the instrument panel and rotating it 90 degrees about its longitudinal axis to move the bubble to the expansion chamber.

Section 20-14 Periodic Maintenance

Periodic maintenance of this unit entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 20-15 Turn Rate Gyro Procedures

This section contains the removal and installation procedures for the turn rate gyro. This section also contains the operational checkout procedure.

TURN RATE GYRO REMOVAL

Perform this procedure to remove the turn rate gyro.

1. Check that all electrical switches are off.
2. Place a cover over the pilot yoke control.
3. Have a large block of soft foam on your lap to place the instrument panel on.
4. Remove the ten screws securing the instrument panel to the instrument console. See Figure 34-39 for location of the screws.

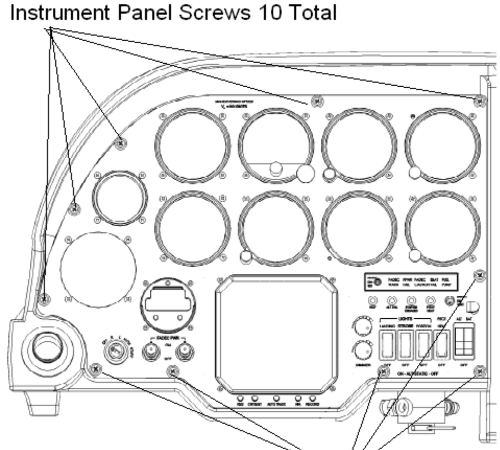

Figure 34-39 Location of the Screws Securing the Instrument Panel

5. Gentle pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
6. Do not to disturb electrical connections, pitot or static lines.
7. Remove (4) panel screws and remove turn rate gyro indicator and disconnect electrical connector.
8. Secure the cutout (bezel) light assembly to the panel to prevent damage to the assembly.
9. If the installation of the turn rate gyro indicator will take place later, install the instrument panel in to the instrument console. Secure the instrument panel with the ten screws removed in step 4 above.

This completes the Turn Rate Gyro Removal procedure.

TURN RATE GYRO INSTALLATION

Perform this procedure to install the turn rate gyro indicator.

1. Check that all electrical switches are off.
2. If installing a new turn rate gyro indicator immediately after removing an old turn rate gyro indicator, then proceed to step 7 below.
3. Place a cover over the pilot yoke control.
4. Have a large block of soft foam on your lap to place the instrument panel on.
5. Remove the ten screws securing the instrument panel to the instrument console. See Figure 34-37 for location of the screws.
6. Gentle pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
7. Prepare the cutout (bezel) light assembly to allow for the installation of the turn rate gyro indicator.
8. Install turn rate gyro indicator to the instrument panel and secure them to panel with the four instrument screws.
9. Connect turn rate gyro indicator cable.
10. Install instrument panel into the instrument console taking care not to disturb electrical connections, pitot or static lines.
11. Secure the instrument panel with the ten instrument panel screws.
12. Perform operation check described in Turn Rate Gyro Operational Checkout on page 76 of this chapter.

This completes the Turn Rate Gyro Installation procedure.

TURN RATE GYRO OPERATIONAL CHECKOUT

Perform this procedure to do an operational checkout of the turn rate gyro.

1. Place the following aircraft circuit breakers in the position indicated.

Circuit Breaker Section	Circuit Breaker	Position
SYSTEM	GYRO	CLOSED
INSTRUMENTS	ATT	OPEN
INSTRUMENTS	DG	OPEN
INSTRUMENTS	TURN	CLOSED
BAT1		CLOSED

Table 34-16 Circuit Breaker Condition Pre Checkout

2. Verify/position the aircraft level in pitch and roll axis.
3. Position the aircraft master switch to ON.
4. Run turn rate gyro for a period of 3 minutes and verify the following:

- Gyro flag pulls clear
- Gyro holds wings level, no slip indications
- Gyro motor operation is smooth and free of chatter or vibration.

5. Position the aircraft master switch to OFF.
6. Place the following aircraft circuit breakers in the position indicated.

Circuit Breaker Section	Circuit Breaker	Position
SYSTEM	GYRO	CLOSED
INSTRUMENTS	ATT	CLOSED
INSTRUMENTS	DG	CLOSED
INSTRUMENTS	TURN	CLOSED
BAT1		CLOSED

Table 34-17 Circuit Breaker Condition Post Checkout

This completes the Turn Rate Gyro Operational Checkout procedure.

Section 20-16 Turn Coordinator Gyro Troubleshooting

Table 34-18 Section 34-00contains the troubleshooting guide for the turn rate gyro.

Complaint	Possible Cause	Remedy
Turn coordinator inoperative (“OFF” flag visible)	Defective turn coordinator circuit breaker	• Replace circuit breaker
	Defective instrument	• Replace instrument
	Defective wiring	• Repair
Turn coordinator erratic or inoperative	Defective instrument	• Replace instrument
Inclinometer (“skid ball”) damping liquid lost	Cracked inclinometer tube	• Replace instrument

Table 34-18 Troubleshooting Guide for the Turn Rate Gyro

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Section 34-30 Landing Aids

Landing aids in the airplane include Localizer, Glideslope, and Marker Beacon receivers and their associated displays. Localizer and glide-slope receivers are integrated with Course Deviation Indicator (CDI) displays providing Localizer and Glideslope position information to the pilot.

Section 30-01 VOR/Localizer Receiver

The Liberty XL-2 offers five avionics units with localizer reception capability:

- Garmin Model GNS 430 COM/NAV/GPS Receiver
- Garmin Model GNS 530 COM/NAV/GPS receiver
- Garmin Model GNS 430W COM/NAV/GPS WAAS Receiver
- Garmin Model GNS 530W COM/NAV/GPS WAAS Receiver
- Garmin Model SL30 COM/NAV Receiver.

In each case localizer signals are received from a "V" Dipole VOR/Glideslope Antenna located at the top of the aircraft's vertical fin. The antenna is capable of receiving the full 108-118 MHz band of frequencies required for both the VOR and localizer.

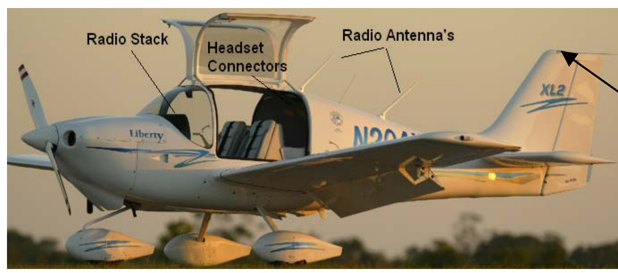


Figure 34-40 VOR/Glideslope Antenna

This single antenna services both the NAV 1 and NAV 2 radios by means of antenna splitters and diplexers configured to support the receiver option installed. Refer to aircraft schematics providing interconnect details for the option package installed. The antenna, diplexer and antenna require no specific maintenance unless a problem with the antenna system is suspected.

Section 30-02 Periodic Maintenance

Periodic maintenance of this system entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 30-03 VOR/Localizer Procedures

This section contains the information to remove and install any of the VOR/Localizer receivers and the VOR/Localizer antenna. This section also contains information troubleshooting issues with the VOR/Localizer system.

NAVIGATION RECEIVER REMOVAL

Perform this procedure to remove any of the navigation receivers.

1. Check that all electrical switches are off.
2. Locate small installation/removal hole on front face of affected component (location varies).
3. Insert hex key (size varies with avionics manufacturer) into hole and turn counterclockwise as many turns as possible without force. Avionics component should move slightly out of mounting tray.
4. Remove component.

This completes the Navigation Receiver Removal procedure.

NAVIGATION RECEIVER INSTALLATION

Perform this procedure to install any of the navigation receivers.

1. Check that all electrical switches are off.
2. Locate small installation/removal hole on front face of component (location varies).
3. Insert hex key and ensure that installation screw has been “backed out” (turned CCW) as many turns as possible to pre-position securing claw for installation.
4. Check rear of avionics component to ensure nothing is blocking any of connectors and no pins are bent.
5. Using a flashlight, inspect mounting tray to ensure nothing is blocking any connectors and no pins are bent.
6. Carefully slide avionics component into mounting tray. Use only gentle pressure; component should stop while still protruding slightly from instrument panel.
7. Insert hex key and turn clockwise to engage mounting claw, pull unit into panel, and seat connectors. Stop when component is seated in mounting tray or at any time if excessive resistance is encountered.
8. Perform operation check described in Navigation System Operational Checkout on page 85 of this chapter.

This completes the Navigation Receiver Installation procedure.

NAVIGATION ANTENNA REMOVAL

Perform this procedure to remove the navigation antenna.

1. Remove all power from the aircraft
2. Remove VOR/Glideslope antenna cover screws (6)
3. Carefully slide the cover and attached antenna assembly vertically until access to the antenna cable BNC connector is obtained. Prior to disconnection of the BNC connector inspect assembly for damage.
4. Disconnect the antenna BNC connector and secure so that it will not slip down into the vertical fin.
5. Loosen antenna element 8-32 set screws located forward of the element sufficiently to permit un-threading of the antenna element.

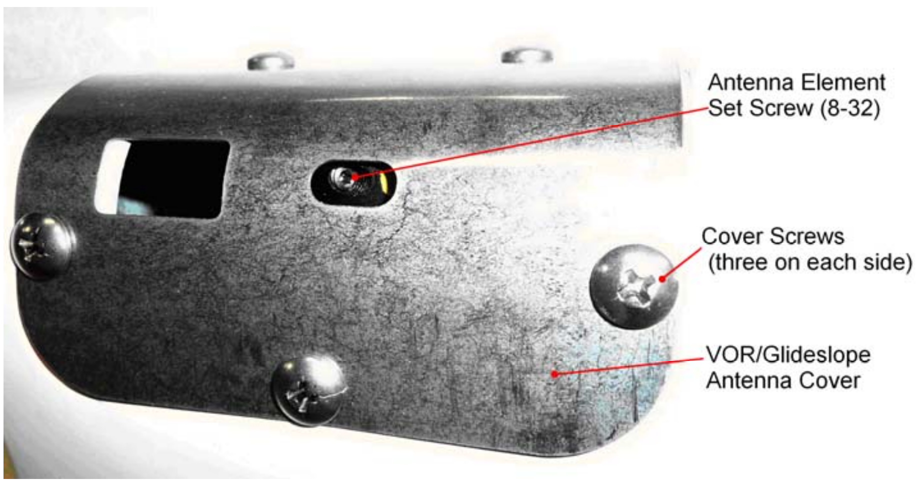


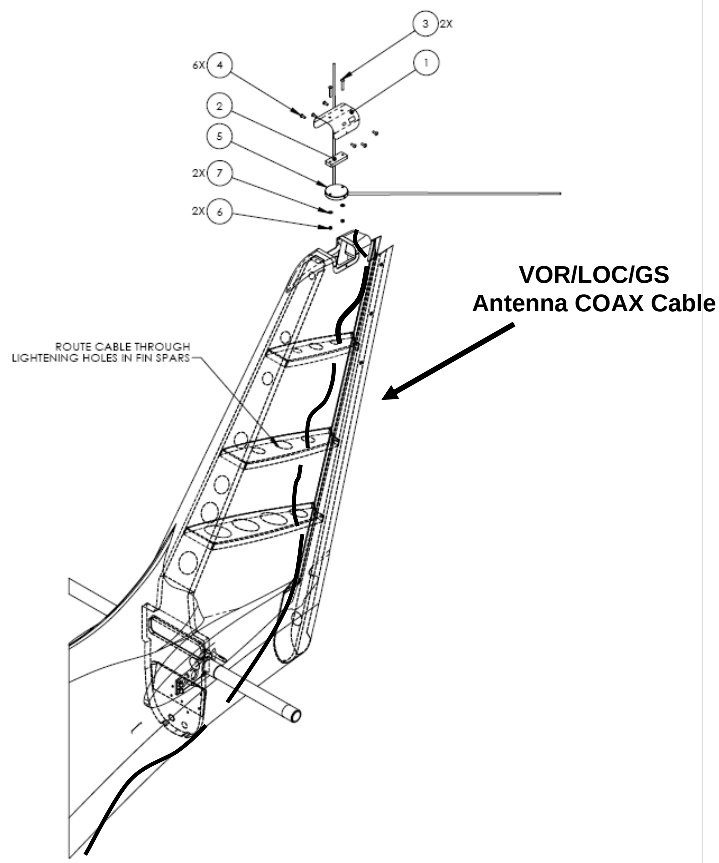
Figure 34-41 VOR/Glideslope Antenna Mount

CAUTION

The following step removes the antenna element from its mounting base. The element is coated in an antistatic finish. Do not use any tool that could damage this finish. Degraded antenna performance will result.

6. Remove the antenna element by rotating counterclockwise.
7. Repeat the process for the opposite side element
8. Remove antenna base element from cover assembly by removal of retaining hardware refer to diagram below for sequence.

This completes the VOR/Glideslope Antenna removal procedure.



ITEM NO.	QTY	PART NUMBER	DESCRIPTION	REMARKS
1	1	135A-10-821	FIN SPAR TOP COVER	
2	1	135A-82-301	VOR SPACER BLOCK	
3	2	AN526C-1032R20	MACHINE SCREW 10-32 CORROSION RESISTANT	
4	6	AN526C-1032R8	MACHINE SCREW 10-32 CORROSION RESISTANT	
5	1	CI 159C	VOR/LOC/GLIDE SLOPE ANTENNA	COMANT
6	2	MS21042-3	NUT, SELF LOCKING 10-32	
7	2	NAS1149C 0332R	FLAT WASHER #10 .203 ID X .438 OD X .032 THK CR	
8	2	NMC S45A-13	SEAL, ELECTRIC WIRE HARNESS	NYLON MOLDING CORP
9	1	P8AX-25BW-FF	COAXIAL SURGE PROTECTOR	CITEL

Figure 34-42 VOR/LOC/GS Slope Antenna Installation (135A-82-103)

This completes the Navigation Antenna Removal procedure.

NAVIGATION ANTENNA INSTALLATION

Perform this procedure to install the navigation antenna.

1. Remove power from the aircraft
2. Install antenna base element in the cover assembly using retaining hardware. Refer to diagram above for assembly sequence.
3. Position the cover over the vertical fin and connect BNC connector to antenna base element.
4. Slide cover over the vertical fin and secure with six (6) retaining screws.
5. Verify antenna element set screw is backed out sufficiently to permit antenna element installation without binding.

CAUTION

The following step installs the antenna element to its mounting base. The element is coated in an antistatic finish. Do not use any tool that could damage this finish. Degraded antenna performance will result.

6. Install antenna element by threading into the base clockwise.
7. Repeat for the opposite side element
8. Tighten the antenna element (#8-32) set screw to 15 in-lb. Do not over tighten.
9. Repeat for the opposite element
10. Perform operation check described in Navigation System Operational Checkout on page 85 of this chapter.

This completes the Navigation Antenna Installation procedure.

NAVIGATION SYSTEM OPERATIONAL CHECKOUT

Perform the post installation procedure using the Garmin Ltd OEM installation manual to do an operation checkout of the navigation system. Use Table 34-19 for the applicable Garmin manual to use.

MODEL	DOCUMENT	TITLE	SECTION	NOTES
SL30	560-0404-03	INSTALLATION MANUAL	2.0	Contact GARMIN Ltd. Dealer
GNS 430	190-00140-02	INSTALLATION MANUAL	5.0	Contact GARMIN Ltd. Dealer
GNS 430W	190-00356-02	INSTALLATION MANUAL	5.0	Contact GARMIN Ltd. Dealer
GNS 530	190-00181-02	INSTALLATION MANUAL	5.0	Contact GARMIN Ltd. Dealer
GNS 530W	190-00357-02	INSTALLATION MANUAL	5.0	Contact GARMIN Ltd. Dealer

Table 34-19 Garmin Ltd Installation Manuals to Use for Post Installation Checkout

Section 30-04 Troubleshooting Guide

This section has the information to troubleshoot issues with the navigation system.

MODEL	DOCUMENT	TITLE	NOTES
GNS 430	190-00140-00	USER GUIDE	Available at www.garmin.com
	190-00140-02	INSTALLATION MANUAL	Contact Garmin Dealer
GNS 530	190-00181-00	USER GUIDE	Available at www.garmin.com
	190-00181-02	INSTALLATION MANUAL	Contact Garmin Dealer
GNS 430W	190-00356-00	USER GUIDE	Available at www.garmin.com
	190-00356-02	INSTALLATION MANUAL	Contact Garmin Dealer
	190-00356-65	ICA MANUAL	Contact Garmin Dealer
GNS 530W	190-00357-00	USER GUIDE	Available at www.garmin.com
	190-00357-02	INSTALLATION MANUAL	Contact Garmin Dealer
	190-00357-65	ICA MANUAL	Contact Garmin Dealer
SL30	190-00486-00	USER GUIDE	Available at www.garmin.com
	560-0404-03a	INSTALLATION MANUAL	Contact Garmin Dealer

Table 34-20 Garmin Manuals to Use in Troubleshooting Issues With the Navigation System

Complaint	Possible Cause	Remedy
No Localizer Reception	NAV radio unit powered off	• Check unit switch is in to ON position • Check avionics master switch is in the ON position
	CB trip	• Check for trip cause correct and reset CB
	Defective NAV unit	• Replace
Erratic localizer needle	Defective antenna	• Replace
	Defective splitter or diplexer	• Replace
	Defective coax cable	• Replace
	Defective indicator (CDI)	• Replace

Table 34-21 Troubleshooting Chart to Use in Troubleshooting Issues With the Navigation System

Section 30-05 Glideslope Receiver

The Liberty Aerospace, Inc. XL-2 airplane offers five avionics units with an integrated Glideslope reception capability:

- Garmin Model GNS 430 COM/NAV/GPS Receiver
- Garmin Model GNS 530 COM/NAV/GPS Receiver
- Garmin Model GNS 430W COM/NAV/GPS WAAS Receiver
- Garmin Model GNS 530W COM/NAV/GPS WAAS Receiver
- Garmin Model SL30 COM/NAV Receiver

In each case Glideslope signals are received from a "V" Dipole VOR/Localizer/Glideslope Antenna located at the top of the aircraft's vertical fin. The antenna is capable of receiving the full 329-335 MHz bandwidth required for Glideslope reception.

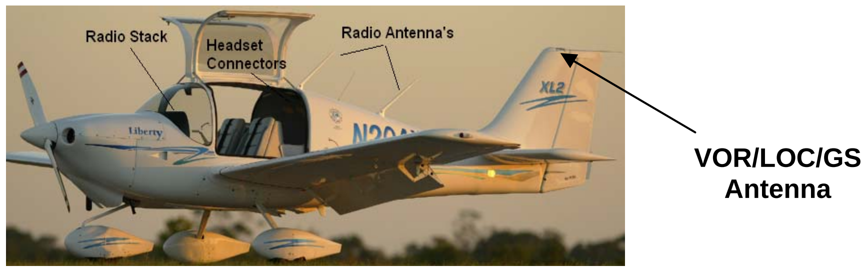


Figure 34-43 VOR/Glideslope Antenna Installation

This single antenna services both the NAV 1 and NAV 2 radios by means of antenna splitters and diplexers configured to support the receiver option installed. Refer to aircraft schematics providing interconnect details for the option package installed. The antenna, diplexer and antenna require no specific maintenance unless a problem with the antenna system is suspected.

Section 30-06 Periodic Maintenance

Periodic maintenance of this system entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 30-07 Glideslope Receiver Procedures

Because the glideslope system is totally integrated with the navigation system, the removal and installation procedures are the same. Therefore, the removal and installation procedures are the same as given in Liberty Maintenance Manual Chapter 23- *Communications.*

GLIDESLOPE OPERATIONAL CHECKOUT

Perform the post installation procedure using the Garmin Ltd OEM installation manual to do an operation checkout of the navigation system. Use Table 34-22 for the applicable Garmin manual to use.

MODEL	DOCUMENT	TITLE	SECTION	NOTES
SL30	560-0404-03	INSTALLATION MANUAL	2.0	Contact GARMIN Ltd. Dealer
GNS 430	190-00140-02	INSTALLATION MANUAL	5.0	Contact GARMIN Ltd. Dealer
GNS 430W	190-00356-02	INSTALLATION MANUAL	5.0	Contact GARMIN Ltd. Dealer
GNS 530	190-00181-02	INSTALLATION MANUAL	5.0	Contact GARMIN Ltd. Dealer
GNS 530W	190-00357-02	INSTALLATION MANUAL	5.0	Contact GARMIN Ltd. Dealer

Table 34-22 Garmin Ltd Installation Manuals to Use for Post Installation Checkout

Section 30-08 Troubleshooting Guide

This section has the information to troubleshoot issues with the navigation system.

MODEL	DOCUMENT	TITLE	NOTES
GNS 430	190-00140-00	USER GUIDE	Available at www.garmin.com
	190-00140-02	INSTALLATION MANUAL	Contact Garmin Dealer
GNS 530	190-00181-00	USER GUIDE	Available atwww.garmin.com
	190-00181-02	INSTALLATION MANUAL	Contact Garmin Dealer
GNS 430W	190-00356-00	USER GUIDE	Available at www.garmin.com
	190-00356-02	INSTALLATION MANUAL	Contact Garmin Dealer
	190-00356-65	ICA MANUAL	Contact Garmin Dealer
GNS 530W	190-00357-00	USER GUIDE	Available at www.garmin.com
	190-00357-02	INSTALLATION MANUAL	Contact Garmin Dealer
	190-00357-65	ICA MANUAL	Contact Garmin Dealer
SL30	190-00486-00	USER GUIDE	Available at www.garmin.com
	560-0404-03a	INSTALLATION MANUAL	Contact Garmin Dealer

Table 34-23 Garmin Manuals to Use in Troubleshooting Issues With the Glideslope System

Complaint	Possible Cause	Remedy
No Glideslope Reception	NAV radio unit powered off	• Check unit switch is in to ON position
	CB trip	• Check for trip cause correct and reset CB
	Defective NAV unit	• Replace
	NAVAID inoperative	• Check NOTAMs
Erratic Glideslope needle	Defective antenna	• Replace
	Defective splitter or diplexer	• Replace
	Defective coax cable	• Replace
	Defective indicator (CDI)	• Replace

Table 34-24 Troubleshooting Chart to Use in Troubleshooting Issues With the Navigation System

Section 30-09 Marker Beacon Receiver

The Marker Beacon receiver and three-light display are incorporated in the Garmin audio selection and switching panel model GMA 340 standard on the XL-2. All marker signals are received on a single common frequency of 75 MHz.

Figure 34-44 Garmin GM340

Outer marker signals are indicated by a 400 Hz audio tone and illumination of the blue marker light. Middle marker signals are indicated by a 1300 Hz audio tone and illumination of the amber marker light. Inner or fan marker signals are indicated by a 3000 Hz audio tone and illumination of the white marker light.

The Garmin GMA-340 Marker Beacon receiver has a switch for high or low sensitivity and may have a test switch to illuminate all three marker lights. A “mute” function, which suppresses Marker Beacon audio in the crew headsets while maintaining marker light capability, may also be installed.

A dedicated Marker Beacon reception antenna is located on the lower fuselage surface. The antenna is capable of receiving the 75 MHz frequency required for Marker Beacon reception.

Figure 34-45 Marker Beacon Antenna Installation

Section 30-10 Periodic Maintenance

Periodic maintenance of this system entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 30-11 Marker Beacon Procedures

The marker beacon antenna procedures are presented in this section. Also, this section has the procedure to perform an operational check of the marker beacon system. The market beacon receiver is integrated in the audio panel. Removal and replacement procedures for the audio panel can be found in Liberty Maintenance Manual Chapter 23 – *Communications.*

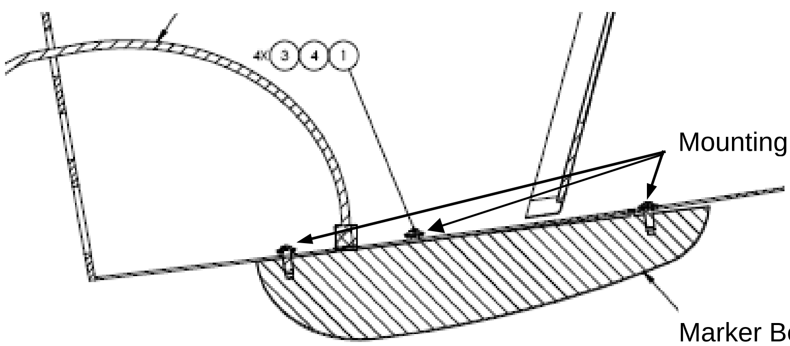
MARKER BEACON ANTENNA REMOVAL

Perform this procedure to remove the marker beacon antenna.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position aircraft master switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Connect an ESD wrist straps to a convenient chassis ground.
4. Remove cabin baggage compartment floor access panel.
5. Locate and disconnect Marker Beacon antenna BNC connector.
6. Locate ar (4) antenna mounting screws and hardware.



ITEM NO.	QTY.	PART NO.	DESCRIPTION
1	4	AN526C-1032R8	MACHINE SCREW 10-32 CORROSION RESISTANT
2	1	CI-102	MARKER BEACON ANTENNA
3	4	MS35335-60	EXTERNAL STAR LOCK WASHER NO. 10, CORROSIVE RESISTANT
4	4	NAS1149C 0332R	FLAT WASHER #10 .203 ID X .438 OD X .032 THK CR

Figure 34-46 Marker Beacon Antenna Mounting Hardware

7. Apply a gentle rocking motion to the antenna body to release bonding material and remove the antenna from the lower fuselage.

This completes the Marker Beacon Antenna Removal procedure.

MARKER BEACON ANTENNA INSTALLATION

Perform this procedure to install the marker beacon antenna.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position aircraft master switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Remove cabin baggage compartment floor access panel
4. Using fine (220-grit) sandpaper and/or a Scotch-Brite™ 7447 (maroon) hand pad, carefully clean residual sealant and debris within the rectangular area defined by the mounting holes. The goal here is to clean the carbon surface of debris without removing the carbon fibers.
5. Clean the surface with acetone and clean, lint free rags. Do not allow the acetone to air dry on the bonding surfaces. Dry the solvent using clean, lint free rags. Liberty Aerospace, Inc. recommends the use of two hands, one with a solvent dampened rag, and one following with a dry rag.
6. Continue wiping operation until the drying rag comes up clean.
7. Using de-natured alcohol, repeat the cleaning operations in steps 5 and 6.
8. Apply a bead of electrically conductive, MMS-040 Silver Coated Nickel Electrically Conductive RTV Silicone (Moreau Marketing & Sales, http:// http://www.rmoreau.com) along the inner edge of the gelcoat/carbon interface, taking care to stay a safe distance from the antenna cable hole. The goal here is to apply just enough RTV to get good contact between the antenna base and the carbon, but not so much that the RTV creeps out from under the antenna base when the antenna is installed. See Figure 34-47 for details of the location for the RTV sealant.

CAUTION

Do not allow the conductive RTV Silicone to contact the BNC RF connectors on the base of the antenna. The conductive RTV Silicone can short out the connector and can cause damage to the radio and/or the antenna.

CAUTION

Silastic, RTV or Silicone-Based sealing/caulking compounds are not to be used around the base or over the screw fillets. The high dielectric content of these materials distort satellite reception at low angles of elevation.

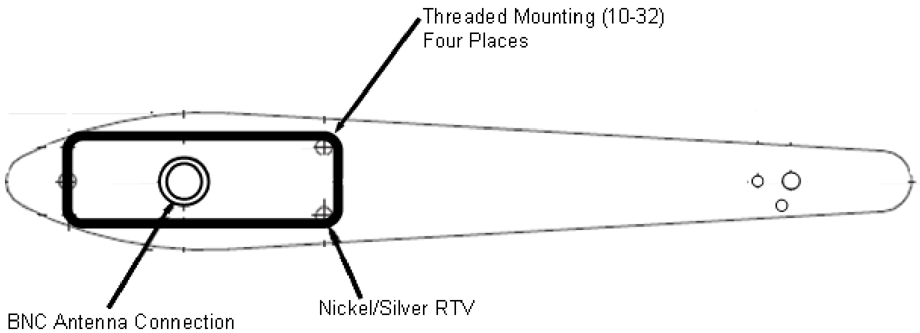


Figure 34-47 Marker Beacon Antenna Conductive Sealant

9. Mount the antenna using screws, washers, nuts and backing plate. Gently tighten the mounting hardware so that uniform stress is placed on each side of the antenna. #10-32 screws DO NOT exceed 23 in-lbs of torque.
10. Run a bead of white, non-corrosive RTV (Dow Corning 748, or equivalent) around the base of the antenna. Fillet the edges for a neat installation.
11. Connect the Marker Beacon antenna lead.
12. Check continuity between a mounting screw and the chassis for < 0.5 ohms resistance (0.003 ohms is ideal).
13. Install cabin baggage compartment floor access panel.
14. Remove tail support.
15. Perform the Marker Beacon Operational Checkout procedure on page 94 of this chapter

This completes the Marker Beacon Antenna Installation procedure.

MARKER BEACON OPERATIONAL CHECKOUT

Perform the post installation procedure using the Garmin Ltd OEM installation manual to do an operation checkout of the navigation system. Use Table 34-25 for the applicable Garmin manual to use.

MODEL	DOCUMENT	TITLE	SECTION	NOTES
GMA 340	190-00149-01	INSTALLATION MANUAL	2.6	Contact GARMIN Ltd. Dealer

Table 34-25 Garmin Ltd Installation Manuals to Use for Post Installation Checkout

Section 30-12 Troubleshooting Guide

This section has the information to troubleshot issues with the navigation system.

MODEL	DOCUMENT	TITLE	NOTES
GMA 340	190-00149-10	USER GUIDE	Available at www.garmin.com
	190-00149-01	INSTALLATION MANUAL	Contact Garmin Dealer

Table 34-26 Garmin Manuals to Use in Troubleshooting Issues With the Marker Beacon System

Complaint	Possible Cause	Remedy
Marker Beacon inoperative	Defective receiver in audio panel	• Replace audio panel
Marker Beacon reception poor	Defective antenna or connections	• Repair or replace as required
Marker lights function but no audio signals	Defective “mute” function (some models)	• Replace audio panel
	Defective audio panel	• Replace audio panel
One or more marker lights fail to illuminate (signal received or “test” function)	Defective marker light bulb	• Replace / repair marker lights (to be performed by avionics shop)

Table 34-27 Troubleshooting Chart to Use in Troubleshooting Issues With the Marker Beacon System

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Section 34-50 Dependent Position Determining

The airplane may be equipped with avionics system installations that provide dependent position determination as well as approach and landing guidance based on signals from ground stations (VOR, ILS) and the satellite-based Global Positioning System (GPS).

Due to the technologically complex nature of avionics equipment, field maintenance is generally limited to removal and replacement of components. In the event an internal fault is suspected refer troubleshooting and repair operations to a certified avionics maintenance facility.

Installations include avionics components themselves (accommodated in mounting trays in the center instrument panel) and indicator instruments (mounted in the instrument panel to the left of the center avionics stack).

Section 50-01 GPS Receiver

GNS 430 and GNS 530 position and velocity determination is made using signals transmitted by Global Positioning System (GPS) Satellites. Position data received is graphically presented to the pilot against a stored navigation and map database by means of a color LCD screen. The GPS Receiver of the GNS 430 or GNS 530 radio is a twelve channel parallel receiver that is capable of tracking and using up to twelve visible satellites for position, velocity, and time calculations.

GNS 530 GPS DISPLAY

GNS 430 GPS DISPLAY

Figure 34-48 GNS Radio GPS Display

An avionics upgrade option is available to configurations listed above that replaces GNS 430 and GNS 530 COM/NAV/GPS units with GPS WAAS capable GNS 430W and GNS 530W COM/NAV/GPS units. WAAS capable GNS units look identical to their Non-WAAS predecessors. Refer to the aircraft equipment list to identify which versions of these units are installed. On power up of the GNS units the display shown in Figure 34-49 will be seen. Verify the unit identification and software revisions from this display.

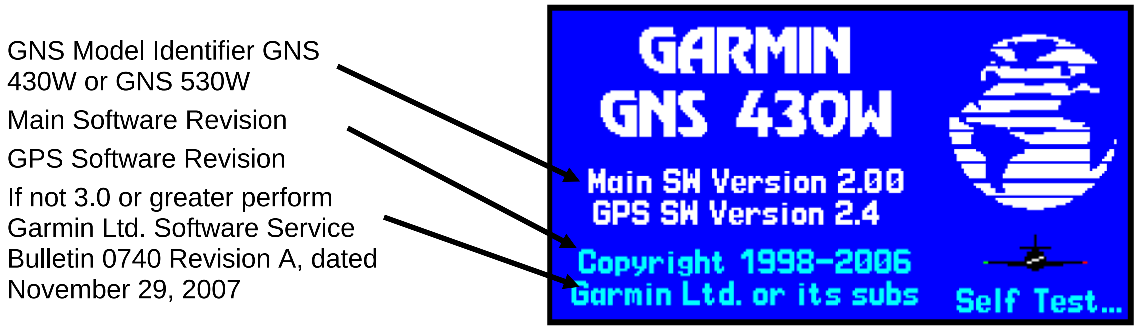
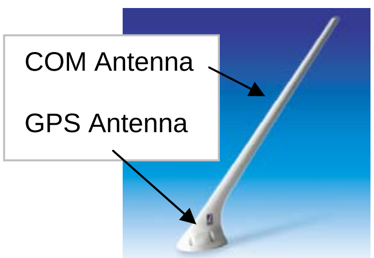


Figure 34-49 GNS Model and Software Verification

Two GPS antennas are installed on the XL-2. A compound antenna installation is utilized that contains both a GPS reception antenna and a COM antenna. A forward antenna is assigned to the GPS receiver in the COM/NAV 1 avionics stack position. The aft antenna is assigned the COM/NAV 2 stack position. These antennas required no specific maintenance unless a fault is detected indicating an antenna problem.



CI-2480-200 GPS/COM

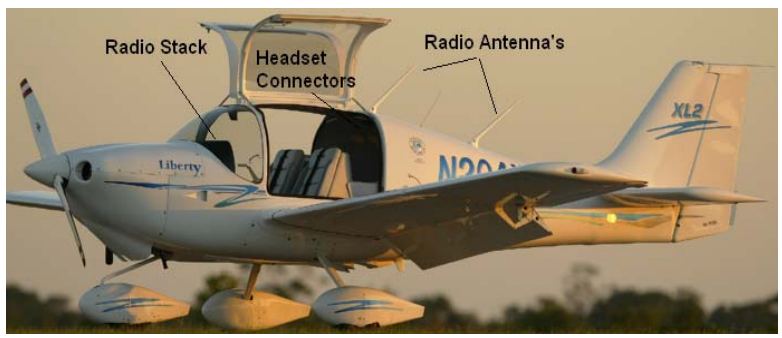


Figure 34-50 GPS Antenna Placement

CAUTION

Two models of COM/GPS antennas may be installed on the aircraft depending on the avionics options installed. Prior to antenna remove and replace operations verify avionics options and antennas installed

Aircraft verified as equipped with GNS 430 or GNS 530 radios will be fitted with Comant model CI 2480-200 model antennas installed forward for COM/NAV/GPS 1 and aft COM/NAV/GPS 2. Aircraft verified as equipped with GNS 430W or GNS 530W units will be fitted with Comant model CI 2580-200 antenna. Use the correct procedures below for each antenna model.

Section 50-02 Periodic Maintenance

Periodic maintenance of this system entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 50-03 GPS Receiver Procedures

The GPS receivers and antennas are integral to the COM/NAV receivers and antennas. The procedures for removing and installing the receivers and/or antennas are in Chapter 23 – *Communications*.

GPS SYSTEM OPERATIONAL CHECKOUT

Perform the post installation procedure using the Garmin Ltd OEM installation manual to do an operation checkout of the navigation system. Use Table 34-28 for the applicable Garmin manual to use.

MODEL	DOCUMENT	TITLE	SECTION	NOTES
SL30	560-0404-03	INSTALLATION MANUAL	2.0	Contact GARMIN Ltd. Dealer
GNS 430	190-00140-02	INSTALLATION MANUAL	5.0	Contact GARMIN Ltd. Dealer
GNS 430W	190-00356-02	INSTALLATION MANUAL	5.0	Contact GARMIN Ltd. Dealer
GNS 530	190-00181-02	INSTALLATION MANUAL	5.0	Contact GARMIN Ltd. Dealer
GNS 530W	190-00357-02	INSTALLATION MANUAL	5.0	Contact GARMIN Ltd. Dealer

Table 34-28 Garmin Ltd Installation Manuals to Use for Post Installation Checkout

Section 50-04 Troubleshooting Guide

This section has the information to troubleshoot issues with the navigation system.

MODEL	DOCUMENT	TITLE	NOTES
GNS 430	190-00140-00	USER GUIDE	Available at www.garmin.com
	190-00140-02	INSTALLATION MANUAL	Contact Garmin Dealer
GNS 530	190-00181-00	USER GUIDE	Available at www.garmin.com
	190-00181-02	INSTALLATION MANUAL	Contact Garmin Dealer
GNS 430W	190-00356-00	USER GUIDE	Available at www.garmin.com
	190-00356-02	INSTALLATION MANUAL	Contact Garmin Dealer
	190-00356-65	ICA MANUAL	Contact Garmin Dealer
GNS 530W	190-00357-00	USER GUIDE	Available at www.garmin.com
	190-00357-02	INSTALLATION MANUAL	Contact Garmin Dealer
	190-00357-65	ICA MANUAL	Contact Garmin Dealer
SL30	190-00486-00	USER GUIDE	Available at www.garmin.com
	560-0404-03a	INSTALLATION MANUAL	Contact Garmin Dealer

Table 34-29 Garmin Manuals to Use in Troubleshooting Issues With the GPS System

Complaint	Possible Cause	Remedy
No display	GNS radio unit powered off	• Check unit switch is in to ON position • Check avionics master switch is in the ON position
	CB trip	• Check for trip cause correct and reset CB
	Defective GPS unit	• Replace
Poor reception of GPS signals	Defective GPS receiver	• Replace
	Defective GPS antenna cable	• Replace
	Defective GPS antenna	• Replace
	Possible local GPS signal interference	• Replace
	Possible local GPS signal interference	• Check GPS-NOTAMs
Poor reception of low angle GPS satellite signals (WAAS)	GPS antenna contamination	• Inspect and clean GPS antenna of all contamination including excess sealant.

Table 34-30 Troubleshooting Chart to Use in Troubleshooting Issues With the GPS System

Section 50-05 Transponder

The Garmin GTX 327 is a panel-mounted transponder with the addition of altitude reporting and timing functions. The transponder is a radio transmitter and receiver that operates on radar frequencies, receiving ground radar or TCAS interrogations at 1030 MHz and transmitting a coded response of pulses to ground-based radar on a frequency of 1090 MHz.

As with other Mode A/Mode C transponders, the GTX 327 replies with any one of 4,096 codes, which differ in the position and number of pulses transmitted. By replying to ground transmissions or TCAS interrogations, the GTX 327 enables ATC to display aircraft identification, altitude and groundspeed on ATC radar screens or TCAS traffic indicators. The GTX 327 is equipped with IDENT capability that activates the Special Position Identification (SPI) pulse for 18 seconds.

The GTX 327 is configured with all key controls. The layout of the front panel keys and displays segregates the transponder's primary functions from the secondary timing functions. The unit can be configured so the aircraft avionics master bus can turn the unit on.

Figure 34-51 GTX 327 Transponder

The transponder uses a dedicated antenna located on the bottom fuselage surface in a location below the cabin baggage compartment. The antenna support standard Mode C transponder frequency band of 1030-1090 MHz. The antenna requires no maintenance unless a fault is detected indicating an antenna fault.

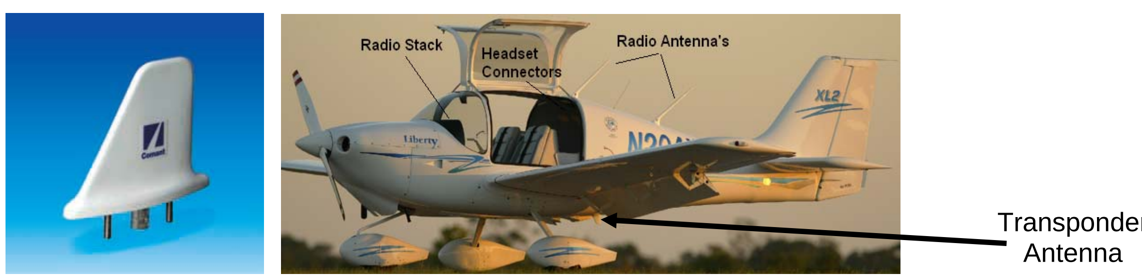


Figure 34-52 Transponder Antenna Placement.

Section 50-06 Periodic Maintenance

Periodic maintenance of this unit entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Additional calibration testing is required at intervals to remain in compliance with 14 CFR Part 91 requirements.

Section 50-07 Transponder Procedures

The transponder transceiver is part of the avionics stack, and therefore the procedures are that same as the other avionics transceivers. To remove or install the transponder, see Section 30-02 page 79 of this chapter. The antenna mounts to the underside of the fuselage, aft of the belly panel. This section contains the removal and installation procedures for the transponder antenna.

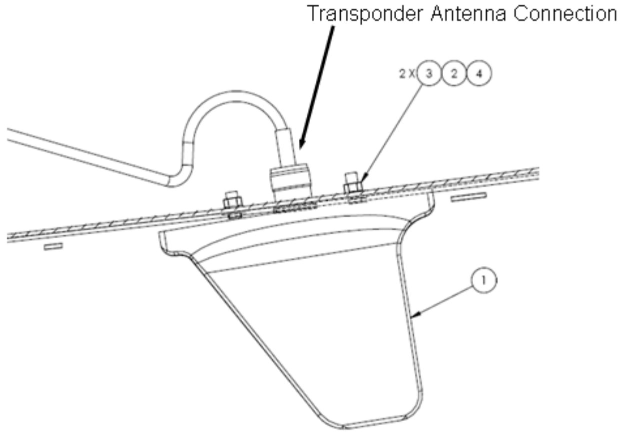
TRANSPONDER ANTENNA REMOVAL

Perform this procedure to remove the transponder antenna.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position aircraft master switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Remove cabin baggage compartment floor access panel
4. Locate and disconnect transponder antenna connectors.



ITEM NO.	QTY	PART NUMBER	DESCRIPTION
1	1	CI 105	ANTENNA, DME TRANSPONDER
2	2	MS35335-31	EXTERNAL TOOTH LOCK WASHER #8
3	2	MS21042-08	NUT, SELF LOCKING 8-32
4	2	NAS1149C N816R	FLAT WASHER #8 .174 ID X ..375OD X .016THK CR

Figure 34-53 Transponder Antenna Installation Details

5. Locate and remove two (2) antenna mounting nuts and hardware.
6. Apply a gentle rocking motion to the antenna body to release bonding material and remove the antenna from the upper fuselage.

This completes the Transponder Antenna Removal procedure.

TRANSPONDER ANTENNA INSTALLATION

Perform this procedure to install the transponder antenna.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position aircraft master switch to OFF.
2. Install a tail stand underneath the tail section of the airplane.
3. Remove cabin baggage compartment floor access panel
4. Using fine (220-grit) sandpaper and/or a Scotch-Brite™ 7447 (maroon) hand pad, carefully clean residual sealant and debris within the rectangular area defined by the mounting holes. The goal here is to clean the carbon surface of debris without removing the carbon fibers.
5. Clean the surface with acetone and clean, lint free rags. Do not allow the acetone to air dry on the bonding surfaces. Dry the solvent using clean, lint free rags. Liberty Aerospace, Inc. recommends the use of two hands, one with a solvent dampened rag, and one following with a dry rag.
6. Continue wiping operation until the drying rag comes up clean.
7. Using de-natured alcohol, repeat the cleaning operations in steps 5 and 6.
8. Apply a bead of electrically conductive, MMS-040 Silver Coated Nickel Electrically Conductive RTV Silicone (Moreau Marketing & Sales, http://www.rmoreau.com) along the inner edge of the gelcoat/carbon interface, taking care to stay a safe distance from the antenna cable hole. The goal here is to apply just enough RTV to get good contact between the antenna base and the carbon, but not so much that the RTV creeps out from under the antenna base when installing the antenna. See Figure 34-54 for details of the location for the RTV sealant.

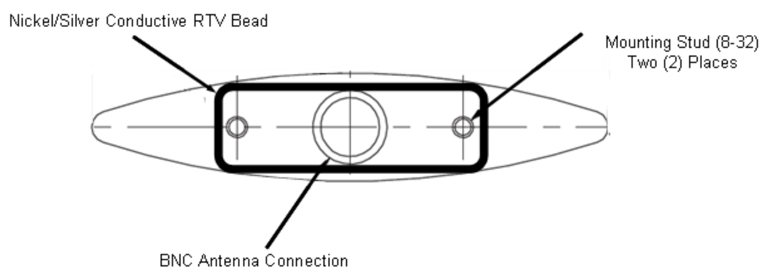


Figure 34-54 Transponder Antenna Conductive Sealant

9. Mount the antenna using #8-32 screws, washers, nuts and backing plate. Gently tighten the mounting hardware so that uniform stress is placed on each side of the antenna. Torque the screws to a maximum of 20 in-lbs of torque.

10. Run a bead of white, non-corrosive RTV (Dow Corning 748, or equivalent) around the base of the antenna. Fillet the edges for a neat installation.
11. Connect the Transponder coax antenna lead.
12. Check continuity between a mounting screw and the chassis for <0.5 ohms resistance (0.003 ohms is ideal).
13. Install cabin baggage compartment floor access panel
14. Remove tail support.
15. Perform the Transponder System Operational Checkout on page 107 of this chapter.

This completes the Transponder Antenna Installation procedure.

TRANSPONDER SYSTEM OPERATIONAL CHECKOUT

Perform an operational check using Garmin Ltd. OEM installation manual Post Installation Checkout procedure. Refer to the table below for applicable manual and section. In all cases, the latest manual version is to be used.

MODEL	DOCUMENT	TITLE	SECTION	NOTES
GTX 327	190-00187-02	INSTALLATION MANUAL	3.0	Contact GARMIN Ltd. Dealer

Table 34-31 Garmin Ltd Installation Manuals to Use for Post Installation Checkout

Section 50-08 Troubleshooting Guide

Use the information in Table 34-32and Table 34-33 as a guide to trouble shooting.

MODEL	DOCUMENT	TITLE	NOTES
GTX 327	190-00187-00	USER GUIDE	Available at www.garmin.com
	190-00187-02	INSTALLATION MANUAL	Contact Garmin Dealer

Table 34-32 Garmin Manuals to Use in Troubleshooting Issues With the Transponder System

Complaint	Possible Cause	Remedy
Poor ATC reception	Defective transmitter	Replace
	Defective antenna	Replace
	Defective antenna cable	Repair
Altitude not reported	Defective altitude encoder	Replace
	Defective encoder cable	Repair or replace
	Encoder not powered	Power on encoder
Unit fails to power up	Circuit breaker open	Investigate cause of trip and reset on resolution
	Power switch fault	Replace unit
Unit not seen by GNS series radios	Data connection lost	Repair wiring
	GNS radio not configured for transponder bus	Configure bus per Garmin procedure

Table 34-33 Troubleshooting Chart to Use in Troubleshooting Issues With the Transponder System

Section 50-09 Course Deviation Indicators

The XL-2 can be fitted with two models of Course Deviation Indicator (CDI) depending on which radio options have been installed. Refer to the configuration table below for identification of CDI models with radios installed.

NOTE

The CDI models have a similar appearance. Take care to assure the correct CDI model is installed for the radio supported.

NAV 1 POSITION			NAV 2 POSITION		
RADIO	DESCRIPTION	CDI	RADIO	DESCRIPTION	CDI
GNS 430 (or W)	COM/NAV/GPS	GI-106A	NONE	NONE	NONE
GNS 430 (or W)	COM/NAV/GPS	GI-106A	GNS 430	COM/NAV/GPS	GI-106A
GNS 530 (or W)	COM/NAV/GPS	GI-106A	NONE	NONE	NONE
GNS 530 (or W)	COM/NAV/GPS	GI-106A	GNS 430	COM/NAV/GPS	GI-106A
GNS 530 (or W)	COM/NAV/GPS	GI-106A	SL30	COM/NAV	MD200-306
GNS 530 (or W)	COM/NAV/GPS	GI-106A	SL40	COM	NONE
GNS 430 (or W)	COM/NAV/GPS	GI-106A	SL30	COM/NAV	MD200-306
GNS 430 (or W)	COM/NAV/GPS	GI-106A	SL40	COM	NONE
SL30	COM/NAV	MD200-306	NONE	NONE	NONE
SL30	COM/NAV	MD200-306	SL30	COM/NAV	MD200-306
SL30	COM/NAV	MD200-306	SL40	COM	NONE
SL40	COM	NONE	NONE	NONE	NONE

Table 34-34 Navigation Receiver Model/Indicator Chart

The Garmin GI-106A Course Deviation Indicators are designed to operate with VHF and GPS navigational equipment to provide VOR, Localizer (LOC) and Glideslope (GS) information to the pilot. XL-2 radio configurations utilizing a Garmin GNS 430 or GNS 530 COM\/NAV/GPS radio us the GI-106A CDI.

The GI-106A is designed to accept DC Left-Right course deviation signals as well as To-From and Left-Right warning flag signals from the GNS 430 and GNS 530 radios. The unit will accept DC signals from the GNS 430 or GNS 530 Glideslope receivers which will drive the Up-Down needle along with an Up-Down warning flag. The unit provides NAV, GPS and VOR/LOC annunciators with photocell dimming.

When GPS is selected for display, the GI-106A receives inputs from the GNS 430 or GNS 530 to provide a visual presentation to the pilot of GPS tracking.

Figure 34-55 GI-106A CDI

The MD200-306 CDI is designed to operate with VHF, and GPS navigational equipment to provide OMNI (VOR), GPS, LOCALIZER (LOC) and Glideslope (GS) information.

The MD200-306 is designed to accept DC Left-Right course deviation signals as well as To-From and NAV warning flag signals from an SL 30 COM/NAV unit. Additionally, the MD200-306 will accept DC signals from the SL 30 Glideslope receiver which will drive the UP-DOWN needle along with a GS warning flag. The unit incorporates NAV, GPS and BC annunciation with photocell dimming.

Figure 34-56 MD200-306 CDI

Section 50-10 Periodic Maintenance

Periodic maintenance of these units entails operational checks and inspections performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 and in accordance with the operational check and inspection procedure in this section.

Section 50-11 Course Deviation Indicator Procedures

This section contains the removal and installation procedures for the course deviation indicators (CDI). This section also has an operational checkout procedure and a troubleshooting guide for the CDI.

COURSE DEVIATION INDICATOR REMOVAL

Perform this procedure to remove the course deviation indicator.

1. Check that all electrical switches are OFF.
2. Place a cover over the pilot yoke control.
3. Have a large block of soft foam on your lap to place the instrument panel on.
4. Remove the ten screws securing the instrument panel to the instrument console. See Figure 34-57 for location of the screws.

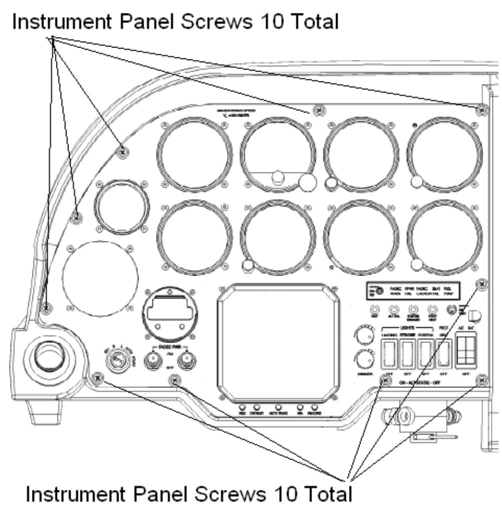


Figure 34-57 Location of the Screws Securing the Instrument Panel

5. Gentle pull the instrument panel towards you, placing it face down on to the block of soft rubber.

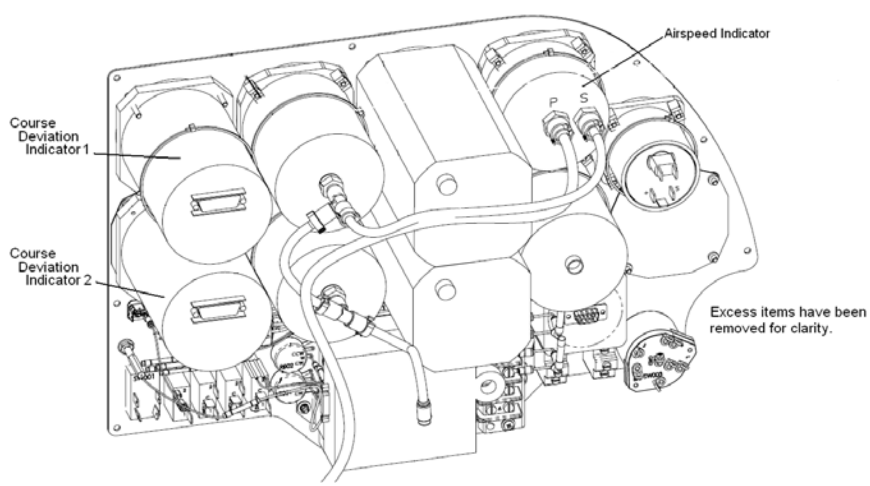


Figure 34-58 Rear View of the instrument Panel Showing Location of the Course Deviation Indicator(s)

6. Disconnect the cable from the defective CDI.
7. Remove (4) panel screws and remove instrument.

8. Secure the cutout (bezel) light assembly to the panel to prevent damage to the assembly.
9. If the installation of the course deviation indicator will take place later, install the instrument panel in to the instrument console. Secure the instrument panel with the ten screws removed in step 4 above.

This completes the Course Deviation Indicator Removal procedure.

COURSE DEVIATION INDICATOR INSTALLATION

Perform this procedure to install the course deviation indicator or CDI.

1. Check that all electrical switches are OFF.
2. If installing a new CDI immediately after removing an old CDI, then proceed to step 8 below.
3. Place a cover over the pilot yoke control.
4. Have a large block of soft foam on your lap to place the instrument panel on.
5. Remove the ten screws securing the instrument panel to the instrument console. See Figure 34-57 for location of the screws.
6. Gentle pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
7. Prepare the cutout (bezel) light assembly to allow for the installation of the vertical speed indicator.
8. Install CDI and cutout (bezel) light assembly to the instrument panel and secure them to panel with the four instrument screws.
9. Connect the cable from the NAV receiver to the CDI, ensuring correct connection.
10. Install instrument panel into the instrument console.
11. Secure the instrument panel with the ten instrument panel screws.
12. Perform the Course Deviation Indicator Operational Checkout on page 114 of this chapter.

This completes the Course Deviation Indicator Installation procedure.

COURSE DEVIATION INDICATOR OPERATIONAL CHECKOUT

Perform an operational check using Garmin Ltd. OEM installation manual Post Installation Checkout procedure. Refer to the table below for applicable manual and section. In all cases, use the latest manual version.

MODEL	DOCUMENT	TITLE	SECTION	NOTES
GI-106A	190-00180-00	INSTALLATION MANUAL	5.0	Contact GARMIN Ltd. Dealer
MD200-306	8017972	INSTALLATION MANUAL	4.0	Contact GARMIN Ltd. Dealer

Table 34-35 Garmin Ltd Installation Manuals to Use for Post Installation Checkout

Section 50-12 Troubleshooting Guide

Troubleshooting of the CDI's is by reference to the latest versions of the Garmin manuals listed below.

MODEL	DOCUMENT	TITLE	NOTES
GNS 430	190-00140-00	USER GUIDE	Available at www.garmin.com
	190-00140-05	INSTALLATION MANUAL	Contact Garmin Dealer
GNS 530	190-00181-00	USER GUIDE	Available at www.garmin.com
	190-00181-02	INSTALLATION MANUAL	Contact Garmin Dealer
GNS 430W	190-00356-00	USER GUIDE	Available at www.garmin.com
	190-00356-02	INSTALLATION MANUAL	Contact Garmin Dealer
	190-00356-65	ICA MANUAL	Contact Garmin Dealer
GNS 530W	190-00357-00	USER GUIDE	Available at www.garmin.com
	190-00357-02	INSTALLATION MANUAL	Contact Garmin Dealer
	190-00357-65	ICA MANUAL	Contact Garmin Dealer
SL30	190-00846-00	USER GUIDE	Available at www.garmin.com
	560-0404-03a	INSTALLATION MANUAL	Contact Garmin Dealer
GI-106A	190-00180-00	INSTALLATION MANUAL	Contact Garmin Dealer
MD200-306	8017972	INSTALLATION MANUAL	Available at www.mcico.com

Table 34-36 Garmin Manuals to Use in Troubleshooting Issues With the Course Deviation Indicators

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CHAPTER 51

Standard Practices - Structures

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Table of Contents

SECTION 51-00	GENERAL	5
SECTION 00-01	QUALIFIED FACILITIES AND DOCUMENTATION	8
SECTION 51-10	STRUCTURAL COMPOSITE REPAIRS	9
SECTION 10-01	AN EXTENSIVE FUSELAGE OR FLYING SURFACE SKIN REPAIR	9
SECTION 10-02	FITTINGS REQUIRING JIGGING FOR POSITIONAL LOCATION	9
SECTION 51-11	COMPOSITES - SOLID LAMINATE AND SANDWICH CONSTRUCTION	11
SECTION 51-12	COMPOSITES DAMAGE CLASSIFICATION	13
SECTION 12-01	TYPE 1 - DAMAGE	14
SECTION 12-02	TYPE 2 - DAMAGE	14
SECTION 12-03	TYPE 3 - DAMAGE	15
SECTION 12-04	TYPE 4 - DAMAGE	15
SECTION 51-13	LAMINATED COMPOSITES INSPECTION CRITERION	17
SECTION 13-01	COMPOSITE LAMINATES	17
SECTION 13-02	TAP TESTING	18
SECTION 51-14	LAMINATED COMPOSITES REPAIR PROCEDURES	19
	TYPE 1 OR 2 DAMAGE TO ONE SIDE OF SANDWICH (CO-CURE REPAIR)	21
	TYPE 1 OR 2 DAMAGE TO ONE SIDE OF SANDWICH (ALTERNATIVE SECONDARY BONDING REPAIR)	24
	TYPE 1 OR 2 DAMAGE TO BOTH SIDES OF SANDWICH STRUCTURE (FLAT SURFACE)-SECONDARY BONDING	26
	TYPE 1 OR 2 DAMAGE TO BOTH SIDES OF SANDWICH STRUCTURE (BOTH FLAT SURFACE – ALTERNATE CO-CURE REPAIR)	29
	TYPE 1 OR 2 DAMAGE TO BOTH SIDES OF SANDWICH STRUCTURE (CURVED SURFACE)	32
	TYPE 3 SINGLE SKIN REPAIR ONE SIDE ACCESSIBLE (CO-CURE REPAIR)	35
	TYPE 3 SINGLE SKIN REPAIR ONE SIDE ACCESSIBLE (ALTERNATIVE SECONDARY REPAIR)	37
	TYPE 3 SINGLE SKIN REPAIR BOTH SIDES ACCESSIBLE (SECONDARY BONDING REPAIR)	39
SECTION 51-15	ADHESIVE JOINT STRUCTURAL REPAIRS:	41
SECTION 51-16	ADHESIVE JOINT DAMAGE CLASSIFICATION:	43
SECTION 16-01	TYPE 1- DAMAGE:	45
SECTION 16-02	TYPE 2- DAMAGE:	45
SECTION 16-03	TYPE 3- DAMAGE:	45
SECTION 16-04	TYPE 4- DAMAGE:	45
SECTION 51-17	ADHESIVE JOINT INSPECTION CRITERION:	47
SECTION 17-01	TAP TESTING:	48
SECTION 51-18	ADHESIVE JOINT REPAIR METHODS:	49
SECTION 18-01	TYPE 1 DAMAGE:	49
SECTION 18-02	TYPE 2 DAMAGE:	49
SECTION 18-03	TYPE 3 DAMAGE:	49
SECTION 18-04	TYPE 4- DAMAGE:	49

TYPE-1 OR 2 DISBOND OF ADHESIVE JOINT 50
TYPE-1 OR 2 REPAIR TO BOTH SIDES OF ADHESIVE JOINT LAMINATE TO LAMINATE 52
TYPE-1 OR 2 REPAIR OF ADHESIVE JOINT SANDWICH TO SANDWICH 54
TYPE-3 DAMAGE REPAIR PROCEDURE 57

SECTION 51-20 MINOR STRUCTURAL METALLIC REPAIRS 59

SECTION 20-01 GENERAL CORROSION INSPECTION AND METAL COMPONENT PROTECTION: 59
METAL COMPONENT INSPECTION FOR CORROSION OR RUST: 60
STRUCTURAL ALUMINUM REPAIRS: 61
STRUCTURAL STEEL REPAIRS 62

SECTION 51-60 CONTROL SURFACES BALANCING 63

SECTION 60-01 MASS BALANCE VERTICAL CLAMP JIG 63
SECTION 60-02 BALANCING PROCEDURES 63
AILERON BALANCING 64
RUDDER BALANCING 66
HORIZONTAL STABILIZER BALANCING 68

SECTION 51-80 GROUND AND BONDING 69

SECTION 80-01 GROUND STUDS 69
SECTION 80-02 BULKHEAD CONNECTORS 69
SECTION 80-03 METALLIC PIPES, TUBES AND HOSES 70
SECTION 80-04 CARBON FIBER COMPOSITE TO ALUMINUM BONDS 70
SECTION 80-05 ANTENNA BONDS 70
SECTION 80-06 INSPECTION PROCEDURES FOR GROUNDING AND BONDING 71
GROUND AND BONDING INSPECTION 72

Section 51-00 General

The Liberty XL-2 airplane uses a mixture of structural technologies including:

- Conventional aluminum construction for flying and control surfaces
- Welded steel tube construction for the fuselage center section space frame which accepts and consolidates loads from the wings, fuselage, landing gear, and engine mount
- Aluminum and steel elements for the landing gear
- Composite (fiber/epoxy laminate) constructions for the fuselage, belly fairing and engine cowlings.

NOTE

Prior to contacting Liberty Aerospace Inc customer support it is imperative that the mechanic has made a major/minor repair decision or has a view to the significance of the repair. If the mechanic is unable to make the decision due to a lack of data, Liberty Aerospace Inc. customer support will furnish this data to help the mechanic make the major/minor determination.

As it is impossible to define criteria and catalogue every type of damage, inspection or repair permutation, please contact Liberty Aerospace Inc customer support for further clarification if any doubt exists with interpretation of this document.

Minor repairs to the aluminum flying surfaces and controls may be carried out using standard materials and techniques in accordance with FAA Advisory Circular 43.13-1B. These repairs should be limited to patching of small holes < 0.1in or tears in aluminum skins, replacement of moving parts such as hinges, etc. Any repairs involving damage to underlying structural components of flying surfaces (significant skin damage, any damage to underlying structures such as spars, ribs, stringers, etc.) should be referred to Liberty Aerospace Inc. Customer Support.

Composite components for the Liberty XL-2airplane are manufactured from specialized pre-impregnated (“Pre-Preg”) materials and structural foam. The solid (no foam core) and sandwich (with a foam core) composite laminate are closely controlled in the fabrication of the Liberty XL-2 aircraft, in which the mixture of fiber reinforcing materials (fiberglass, carbon fiber, etc.) and resin matrix is very closely controlled.

The definitions of the different composite components are:

- **Primary Structures**: "The structure that carries flight, ground, loads, and whose failure would reduce the structural integrity of the airplane or may result in injury or death to passengers or crew is defined as primary structure." Table 51-1 shows a list of components that are Primary Structures.
    - o Examples of primary structures made of composite materials on the XL-2 are the seats which carry crash loads, the majority of the fuselage structure which carries flight and ground loads, and the vertical tail which carries flight loads.

- o Interior structures that carry crash loads, as required by 14 CFR, part 23, FARs 23.561 and 23.562 are primary structure. These are the primary load carrying members. Their failure would reduce the structural integrity of the airframe. These components are an integral portion of the fuselage.

Part Number	Description
135A-10-105	Fuselage Assembly*
135A-10-109	Lower Fuselage
135A-10-411	Upper Fuselage
135A-10-413	Fin Close Out (Vertical Close Out)
135A-10-423	Bulkhead Baggage Bay
135A-10-425	Bulkhead Mid Fuselage
135A-10-427	Fin Spar
135A-10-431	Fin Rib #1
135A-10-433	Fin Rib #2
135A-10-435	Fin Rib #3
135A-10-437	Baggage Bay Floor
135A-10-439	Duct. NACA - Port Cabin Air
135A-10-440	Duct. NACA - Stbd Cabin Air
135A-10-444	Hoop Reinforcement, Fwd
135A-10-445	Baggage Bay Floor Support, Port
135A-10-446	Baggage Bay Floor Support, Starboard
135A-10-465	Fuselage Bond Line Reinforcement Strap
135A-10-481	Closeout, Seat Back, Port
135A-10-482	Closeout, Seat Back, Starboard
135A-10-483	Headliner
135A-11-107	Seat Back Stiffener Installation*
135A-11-187	Seat Base Stiffener Installation*
135A-11-403	Fuselage Bond Line Reinforcement Strap for Rollover Hoop
135A-11-409	Seat Base Stiffener
135A-11-422	Seat Back Stiffener, Rear
135A-11-424	Seat Back Stiffener, Top
135A-11-426	Seat Back Stiffener, Side
135A-50-215	Bulkhead Reinforcement installation*
135A-50-413	Reinforcement, Fwd Bulkhead

Table 51-1 Primary Structures

- **Secondary Structures**: These are not primary load carrying members AND their failure would not reduce the structural integrity of the airframe. These components do not form an integral portion of the fuselage, for example access panels. Table 51-2 shows a list of components that are Secondary Structures.

Part Number	Description
135A-10-123	Door Frame Assembly, Port*
135A-10-124	Door Frame Assembly, Starboard*
135A-10-407	Belly Fairing
135A-10-487	Inner Door Frame Port
135A-10-488	Inner Door Frame Starboard
135A-11-401	Footstep Hard point - Port
135A-11-402	Footstep Hard point – Starboard
135A-11-407	Outer Door Shell, Port
135A-11-408	Outer Door Shell, Starboard
135A-50-401	Upper Cowl
135A-50-403	Lower Cowl
135A-80-413	Instrument Console Untrimmed

Table 51-2 Secondary Structures

- **Tertiary Structures:** These are not primary or secondary load carrying members. These components do not form an integral portion of the fuselage. Table 51-3 shows a list of components that are Tertiary Structures.

Part Number	Description
135A-10-419	Baggage Bay Close Out
135A-10-418	Fin Horn Closeout
135A-10-447	Baggage Bay Floor Access Panel
135A-10-449	Access Panel, Fuel Sender
135A-10-456	Access Panel Torque Tube, Upper
135A-10-458	Access Panel Torque Tube, Lower
135A-10-460	Access Panel, Trim Motor
135A-10-469	Access Cover, Port Door Actuator
135A-10-470	Access Cover, Stbd Door Actuator
135A-20-415	Wing Root Fairing, (Port)
135A-20-416	Wing Root Faring, (Stbd)
135A-20-421	Wing Tip Port
135A-20-422	Wing Tip Stbd
135A-30-401	Tailplane Tip (Port & Stbd)
135A-30-407	Rudder Horn
135A-30-409	Rudder Tip - Lower
135A-40-021	Main Wheel Fairing Assy, Port*
135A-40-022	Main Wheel Fairing Assy, Stbd*
135A-40-401	Nose Gear Fairing, Fwd
135A-40-402	Aft Nose Gear Fairing Assembly
135A-40-403	Main Wheel Fairing, Port
135A-40-404	Main Wheel Fairing, Stbd
135A-40-405	Main Wheel Fairing Panel, Port

Part Number	Description
135A-40-406	Main Wheel Fairing Panel, Stbd
135A-40-411	Wheel Fairing Bulkhead, Port
135A-40-412	Stbd Wheel Fairing Bulkhead, Aft
135A-50-406	Access Panel, Oil
135A-50-415	Spinner Fwd Plate
135A-50-417	Spinner Back Plate
135A-50-423	Spinner

Table 51-3 Tertiary Structures

If the A&P technician is unable to determine the nature of whether the structure us primary or secondary, or whether additional clarification is required prior to the mechanic making the determination whether a repair is major or minor, the mechanic must contact Liberty customer support for further clarification.

If the A&P technician considers that the damage is extensive and/or that the impact may warrant additional Non Destructive Inspection beyond the definition provided within this document, the mechanic must contact Liberty customer support for further clarification.

Section 00-01 Qualified Facilities And Documentation

Repairs to the Liberty XL-2 composite components (sandwich and solid laminate) or metal components may be carried out by any appropriately rated composite repair facility familiar with primary structural composite repairs. Example repair processes are delineated in the later section of this chapter.

Documentation of approved repairs should go into the aircraft maintenance logbook and documented in accordance with FAR part 43.9.

NOTE

The following materials and processing specifications are required to be reviewed prior to performing the composite repairs listed herein. These material and processing specifications are available from Liberty Aerospace, Inc. Customer Service upon request.

Document number	Title
135A-911-042	Liberty aircraft inspection plan
135A-925-001	Airex Core material specification
135A-925-985	Epibond paste adhesive material spec
135A-925-997	Toray plain weave material specification
135A-926-012	Prime and paint process specification
135A-926-994	Secondary bonding of composite materials
135A-926-998	Processing of composite materials

Section 51-10 Structural Composite Repairs

The fuselage of the Liberty XL-2 is composed of structural composite materials. The fuselage aft of the engine cowling is fabricated from carbon fiber reinforced fabrics that are used as facing plies adhered to core materials to form a structural sandwich. As the fuselage is a load bearing monocoque structure, care must be taken when performing repairs to the fuselage. The processes used to perform the structural repair must be followed and the fabrication steps must be adhered with in accordance with the FAA approved technical documents noted on page 6 and per instructions delineated in the later section of this chapter.

WARNING

LIBERTY AEROSPACE INC CUSTOMER SUPPORT MUST BE CONTACTED PRIOR TO PERFORMING THESE REPAIRS.

MODIFICATIONS TO THE FUSELAGE STRUCTURE THAT ARE TO BE PERFORMED ON FORM 337 OR VENDOR STC' MUST BE COORDINATED WITH LIBERTY AEROSPACE INC. CUSTOMER SUPPORT. IF COORDINATION DOES NOT OCCUR, ANY FIELD REPAIRS OR INSTALLATION MAY LEAD TO A NON-AIRWORTHY FUSELAGE STRUCTURE.

Section 10-01 An Extensive Fuselage Or Flying Surface Skin Repair

In the event that large areas > 1.0" of the aircraft skin require repair, it may be difficult to reform the correct surface profile without proper rigid tooling. In addition, the structure may be weakened by the extensive removal and replacement of a load bearing skin. For performing structural repair of such type, contact Liberty Aerospace Inc Customer Support for additional instructions.

Section 10-02 Fittings Requiring Jigging For Positional Location

In the event that fittings have been torn from their original location, this may require special jigging to ensure that they are correctly re-located relative to neighboring components. In the event that a repair of this nature is required, contact Liberty Aerospace Inc. Customer Support for additional instructions.

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Section 51-11 Composites - Solid Laminate And Sandwich Construction

The majority of the fuselage of the Liberty XL-2 is carbon fiber reinforced sandwich construction. Different thicknesses of core material are used to form a structural sandwich in order to support the distribution of stresses through the structure. There are areas of the structure, like the rollover hoop structure for example that are solid laminate. Care must be taken to inspect and determine the nature of the composite base materials prior to performing the repair. Solid laminate has the ability to support concentrated bearing and clamp-up loads better than sandwich construction. Prior to performing any structural repair, the base material should be reviewed and compared with the appropriate Liberty XL-2 drawings to ensure that the correct repair method is performed.

The repair methods noted below make the distinction between solid laminate and sandwich construction repairs.

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Section 51-12 Composites Damage Classification

Damage to the fuselage can occur because of accident, negligence, or corrosion. These may comprised of scratches, dents, tears, holes or cracks. Such damage must be classified to ensure the adequacy of the repairs.

Damage to the fuselage can be classified into four distinct type of damages based on the following criteria:

- Load-path type (primary/secondary structure)
- Location on fuselage ((any sandwich/solid laminate/cosmetic)
- Damage size (length/diameter)
- Damage number (per unit length / area)

Critical damage dimensions have been determined for the Liberty XL-2 fuselage. The acceptable limit for the quantity of 'damage' per unit length or area has also been determined. This allows the fuselage damage to be specified as one of four with each types (1 – 4) having two classes (I and II). Damage should therefore be recorded by the maintenance facility as Type 1, Class II etc.

Laminate classifications are defined in Table 51-4 below. These classifications are used solely to define allowable laminate defects.

Laminate Classification	Maximum Cumulative Defect Size	Defect Accumulation Threshold (DAT)
I	1.00”	1.0 ft^2
II	2.00”	1.0 ft^2

Table 51-4 Laminate Defect Classifications

Laminate defects include:

- Impact damage (such as obvious fracture or penetration of matrix or fibers)
- Inclusions (foreign matter in laminate)
- Extreme porosity (frequent small , 0.050in voids in laminate)

Shallow (less than 0.005 in) scratches and mild porosity (less than 0.050 sq in within a 1sq inch region) do not count as structural defects. When there is doubt regarding classifying a potential defect as a defect, treat it as a defect.

The greatest linear dimension of a defect defines the severity of the defect. For example, circular defects are defined by their diameter, crack defects are defined by their length, and other defects are defined by their largest measurement.

All defects within the Defect Accumulation Threshold, or DAT, are cumulative. For example, if the DAT allows for 1.0” within a 1.0 ft^2 area, then add up all defect sizes within a given square foot. If the total exceeds 1.0 inch, then submit the component to Liberty Aerospace Inc. for determination.

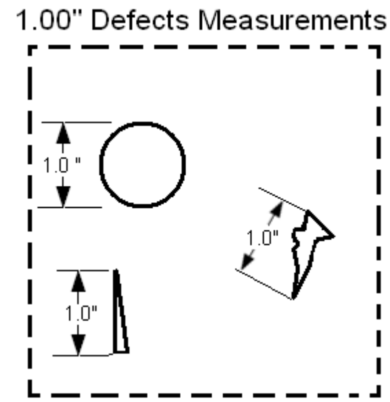

Figure 51-1 Laminate Defect Size Definition Examples

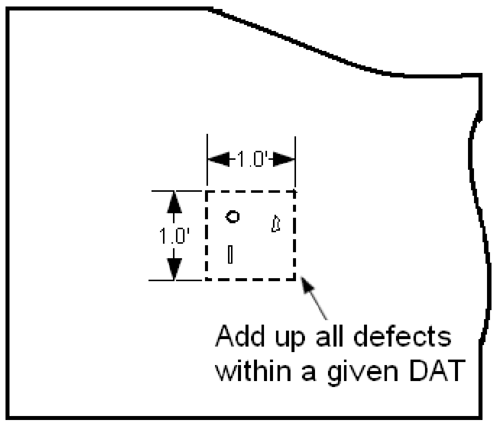

Figure 51-2 Example Application of Laminate DAT

The examples of laminate defect size definition show how the DAT is used to determine cumulative defects. After all defects are identified, move the DAT “window” around to determine which defects must be treated as cumulative.

Section 12-01 Type 1 - Damage

The definition of Type 1-damage is critical damage inflicted to primary or secondary structure at locations such as highly stressed (contact Liberty Aerospace Customer support for clarification) regions and underlying structural elements.

Damage is classified as Type 1 when the size, location, and number of damages per unit length or area endanger the structural integrity of the aircraft.

This type of damage requires partial or complete reconstruction of parts or repairs of large areas.

Section 12-02 Type 2 - Damage

The definition of Type 2-damage is damage inflicted to primary or secondary structure involving complete penetration of the sandwich or laminate materials. Damage is classified as Type 2 if it has the potential to affect the structural integrity of the aircraft in flight.

Section 12-03 Type 3 - Damage

The definition of Type 3-damage is damage limited to outer skin only (no damage to internal facing plies or core material). Damages are classified Type 3 when the size, location and number of damage per unit length/area does not endanger the structural integrity of the aircraft.

Section 12-04 Type 4 - Damage

The definition of Type 4-damage is damage that is inflicted to parts of minimal structural importance. Type 4 damage includes light surface erosion, scratches, grooves, dents (no ply and core damage), etc. that do not penetrate the composite outer skin. This includes damage to replaceable access covers, etc.

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Section 51-13 Laminated Composites Inspection Criterion

Table 51-5 below defines laminate composite inspection classifications.

Laminate Inspection Classifications

Laminate Classification	Minimum Required Inspection
NS	Inspection type not specified
A	100% visual + 100% NDI ultrasound
B	100% visual + 100% NDI tap testing
C	100% visual only

Table 51-5 Inspection Classifications

NOTE

The classification “NS” means that the inspection method is not specified. Any practical method(s) may be used if the method chosen demonstrates itself to the A&P Mechanic that the defect criterion may be met for the component, i.e. the defect can be picked up by the method.

“Visual” means optical. Actual inspection may be done via the naked eye, a bore scope, or other imaging means.

A component may always be inspected more intensively than its classification requires. For example, if a laminate is classified as “C” (visual only), it may still be inspected with ultrasound, tap testing, or other means in addition to the visual inspection as deemed necessary to make a major or minor determination by the A&P Mechanic. If question exists over the methods to employ for inspection or over the need for a more intensive inspection technique, Liberty Aerospace Inc Customer Support must be contacted to obtain clarification.

The specified inspection type is only prescribed for the initial inspection. If defects are found, the defects must be thoroughly mapped using whatever additional inspection means may be deemed necessary by the A&P Mechanic. For example, if tap testing is required and a defect is found, it may be necessary to use ultrasound to fully define the defect. Liberty Aerospace Inc Customer Support must be contacted to obtain these additional inspection instructions.

Section 13-01 Composite Laminates

All composite laminates in the fuselage and vertical stabilizer must be inspected for defects or to evaluate any structural damage. The classification is I-B for all fuselage and vertical stabilizer laminates. Laminate classification defines acceptable defect criteria.

Section 13-02 Tap Testing

The fuselage has been inspected prior to release, the methods used were visual and tap testing. At the time of aircraft release no ultrasound/radiographic testing was performed, but should be considered by an A&P Mechanic as an alternate means of evaluation if doubt exists as to whether damage exists in the structure that requires repair. The use of ultrasound must be coordinated with Liberty Aerospace Inc Customer Support.

Tap testing is widely used to detect the presence of delaminations or debonding. The tap testing procedure consists of lightly tapping the surface of the part with a coin, or light special hammer with a maximum of 2 ounces (see figure below) or any other suitable object.

A flat or dead response is considered unacceptable. The acoustic response of a good part can vary with changes in geometry and laminate thickness. Care should be taken to compare good solid laminate with solid laminate of interest. Similarly good adhesive joints should be compared with the adhesive joint of interest and good sandwich structure of comparable lay-up and thickness should be compared with sandwich structure of interest.

By removing internal access panels, access to both sides of the laminate can be obtained. A review of the fuselage illustration located later within this chapter highlights the bonded areas of the structure for clarity.

The entire area of interest must be tap tested. The surface should be dry and free from oil, grease, and dirt.

NOTE

The accuracy of this test depends on the subjective interpretation of the test response; therefore, only A&P technicians familiar with composite tap testing should perform this test.

As it is impossible to define criteria for all permutations of damage that could require inspection, Liberty customer support should be contacted in the event that a good comparison cannot be obtained from the initial tap test.

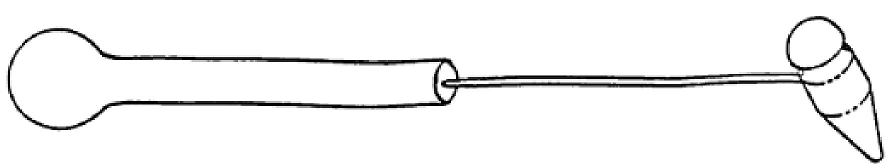

Figure 51-3 Sample Of Special Tap Test Hammer For Tap Testing

Section 51-14 Laminated Composites Repair Procedures

All repairs must be coordinated with Liberty Aerospace, Inc. Customer Support. Separate FAA approval will be obtained for aircraft specific repair instructions. Liberty Customer Support will provide these instructions to the customer.

WARNING

Repair methods provided herein are by way of example only. They are intended to help the maintenance technician to determine the nature of the repair required for the damage and to elaborate on the accepted process that will be specified to perform the repair. For specific FAA approved instructions on the repair of composite damage, contact Liberty Aerospace, Inc. Customer Support.

As it is impossible to define criteria and catalogue every damage, inspection or repair permutation, please contact Liberty Aerospace Inc. customer support for further clarification if any doubt exists with interpretation of this document.

This section has the procedures to effect repairs to the four types of damage. There are different repair procedures depending on the severity and location of the damage.

Type 1 - damage is critical damage inflicted to primary or secondary structure at locations such as highly stressed (contact Liberty Aerospace Customer support for clarification) regions and underlying structural elements.

Type 2 - damage is damage inflicted to a primary or secondary structure involving complete penetration of the sandwich or laminate materials. Employ the same repair procedures as Type 1 –damage.

The following is a list of procedure to effect repairs for Type 1 and Type 2 damage.

Procedure Title	Page
Type 1 or 2 Damage To One Side Of Sandwich (Co-Cure Repair)	21
Type 1 or 2 Damage To One Side Of Sandwich (Alternative Secondary Bonding Repair)	24
Type 1 or 2 Damage To Both Sides Of Sandwich Structure (Flat Surface)-Secondary Bonding	26
Type 1 or 2 Damage To Both Sides Of Sandwich Structure (Both Flat Surface – Alternate Co-Cure Repair)	29
Type 1 or 2 Damage To Both Sides Of Sandwich Structure (Curved Surface)	32

Type 3 - damage is damage limited to outer skin only. No damage is caused to the underlying foam core. This damage may be of following types:

- Delamination between two plies
- Fibers damage to the top plies

- No foam core is damaged for TYPE 3 damage inflicted to the XL-2 composite structure.

In any of the above cases, there is to be no damage to the underlying foam core.

The follow is a list of procedures for all Type 3 case.

Procedure Title	Page
Type 3 Single Skin Repair One Side Accessible (Co-Cure Repair)	35
Type 3 Single Skin Repair One Side Accessible (Alternative Secondary Repair)	37
Type 3 Single Skin Repair Both Sides Accessible (Secondary Bonding Repair)	39

Type 4-damage is defined as damage that is inflicted to parts of minimal structural importance. Rework to the outer surface of the fuselage can be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.

NOTE

If rework/repair is done on the outer surface of the fuselage, it is to be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.

TYPE 1 OR 2 DAMAGE TO ONE SIDE OF SANDWICH (CO-CURE REPAIR)

This procedure covers the repair of crushed foam core for a single skin of a sandwich structure. The belly panel, upper and lower cowl comes under this category.

Perform the following initial preparation as follows:

- Minimum 2-plies of Plain Weave, or PW, (135A-925-997) will be laid up with same orientation as of the plies removed such that the covering repair patch equals or exceeds the lay-up in the deviant region. Thickness of one cured ply is between 0.0083-0.0089 inches. Repair patch (cured or uncured) must replicate the same number of plies (n) removed plus one i.e. (n+1) from the discrepant/deviation section (For example if 2 plies of carbon PW are present prior to the 3mm /5 mm foam, then the patch repair must be n+1 i.e. 3 plies)
- During core removal or surface preparation, no damage should be caused to the underlying ply.
- The size of the repair is to be such that the crushed foam core region is removed and the patch extends minimum +1” all around the deviant region.
- The Plies need to be staggered such that each ply extends 0.5”-1” beyond the previous ply with the innermost ply of the patch being the smallest.
- Crushed/indented foam core must be removed with chamfered edges 30º-45º.
- Shape and size of the repair foam core must be same as that of the core removed.
- The Patches should be rectangular (rounded edge) or elliptical or circular in shape.

NOTE

This covers both flat and curved surfaces. Flat surface is shown for clarity.

Perform this procedure to affect a repair to the composite.

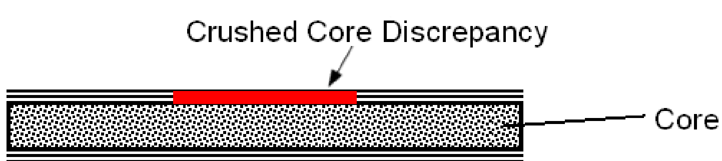

Figure 51-4 Original View of the Damage

1. Completely grind out the crushed foam from the deviant region without damaging the underlying ply.

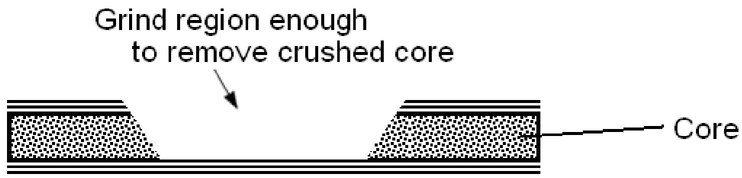

Figure 51-5 Section of the Composite After Grinding

2. Clean and prep the surface +1” of the core removed and apply two plies of Hysol EA9696 (0.03psf) Film adhesive (135A-925-992) such that it extends +1” beyond the exposed foam core region.

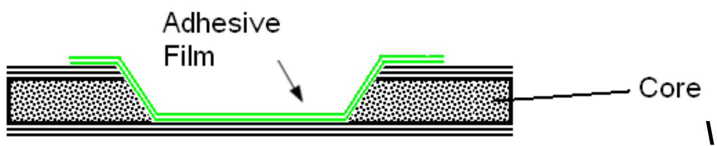

Figure 51-6 Application of the Adhesive Film

3. Create a chamfered 3mm or 5mm Airex foam core piece such that it matches the sanded/prepped region and locate it in the repair section as shown in Figure 51-7.

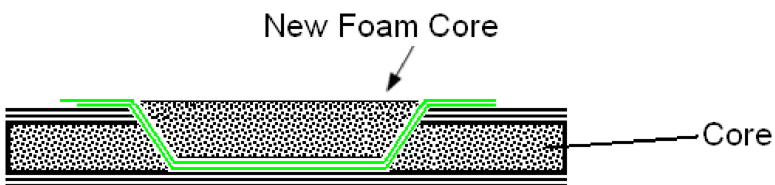

Figure 51-7 Inserting a New Piece of Foam Core

4. Lay up at least two plies of Carbon Plain weave (135A-925-997) at +/- 45° such that it covers the film adhesive. (Stagger Plies)

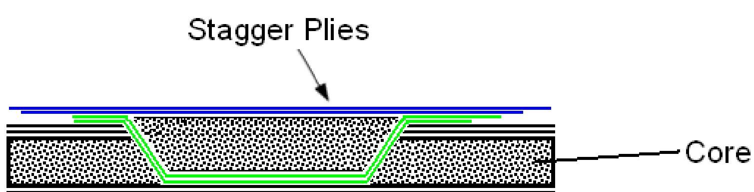

Figure 51-8 Application of the New Staggered Plies of Plain Weave

NOTE

The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).

5. Vacuum bag the repair and perform a standard AGATE/TCA cure cycle in accordance with 135A-926-998 specifications.

6. Upon completion of the cure, inspect the following aspects:

- The cure cycle is within process spec limits (135A-926-998)
- DMA coupon was created and tested TG passed the requirement detailed within 135A-925-997
- The consolidation of the repair is good with no void/discontinuity as observed by inspection procedures in accordance with 135A-911-042

Based on a successful post repair inspection the part may be released to service.

TYPE 1 OR 2 DAMAGE TO ONE SIDE OF SANDWICH (ALTERNATIVE SECONDARY BONDING REPAIR)

Perform this procedure to affect a repair to the composite material.

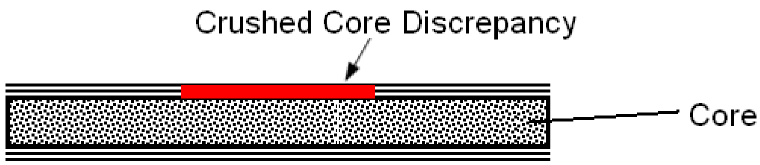

Figure 51-9 Original View of the Damage

1. Completely grind out the crushed foam core region without damaging the underlying ply. Clean and prep the surface +1" all around the discrepant/deviant section as shown:

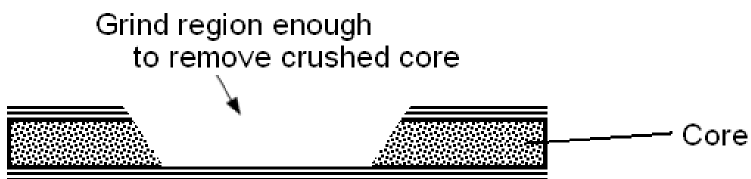

Figure 51-10 Section of Composite After the Grinding

2. Secondarily bond a pre-cured patch of at least 2 ply PW@45°/ 3mm or 5mm Foam Core using Epibond 1590 adhesive (135A-925-985) such that it extends minimum +1" beyond the repair region. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994-G until handling strength, 6hr per 135A-926-994-H.

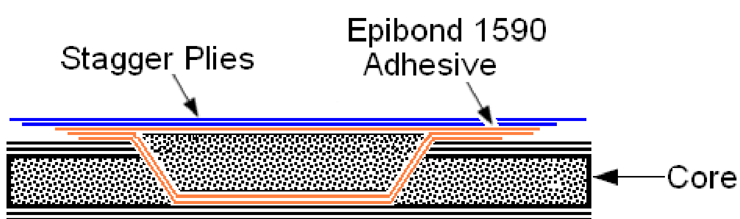

Figure 51-11 Application of the Epibond Adhesive and the Staggered Plies of Plain Weave

NOTE

The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).

3. Elevated temperature post cured the patch in accordance with 135A-926-994.

4. After completion of the cure, inspect the following aspects:

- Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
- Check that a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
- The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.

TYPE 1 OR 2 DAMAGE TO BOTH SIDES OF SANDWICH STRUCTURE (FLAT SURFACE)-SECONDARY BONDING

This procedure covers the repair of crushed foam core from both sides of a sandwich structure

NOTE

If the rework/repair is done on the outer surface of the fuselage, it is to be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.

Damaged regions under this section: Type 1 and 2 damages are categorized under this section. A critical damage inflicted to primary or secondary structures from both sides, at locations such as highly stressed (contact Liberty Aerospace Customer support for clarification) regions, bonding areas etc. The belly panel, upper and lower cowl comes under this category.

The initial preparation will be done as follows:

- Minimum two Plies of Plain weave (PW) (135A-925-997) will be laid up with same orientation as of the plies removed such that the covering repair patch equals or exceeds the lay-up in the deviant region. Thickness of one cured ply is between 0.0083-0.0089 inches. Repair patch (cured or uncured) must replicate the same number of plies (n) removed plus one i.e. (n+1) from the discrepant/deviation section (For example if 2 plies of carbon PW are present prior to the 3mm /5 mm foam, then the patch repair must be n+1 i.e. 3 plies)
- The size of the Repair is to be such that the crushed foam core region is removed and the patch extends minimum +1” all around the deviant region.
- Crushed/indented foam core must be removed with chamfered edges 30º-45º.
- Shape and size of the repair foam core must be same as that of the core removed.
- The Patches should be rectangular (rounded edge) or elliptical or circular in shape.

Perform this procedure to affect repair of the composite.

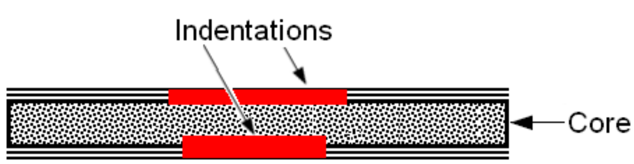

Figure 51-12 Cross Section of Damaged Composite Sandwich

1. Completely grind out the crushed foam core region as follows:

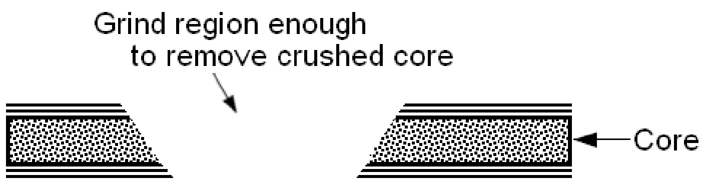

Figure 51-13 Section of the Composite after Grinding to Remove Damaged Area

2. Clean and surface prep the Outer and the inner surface +1" all around the discrepant section.

3. Create a secondarily bond of a pre-cured patch of at least two ply PW@45 using Epibond 1590 adhesive (LAI 135A-925-985) such that it extends minimum +1" beyond the repair region all around. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994-G until handling strength, 6hr per 135A-926-994-H.

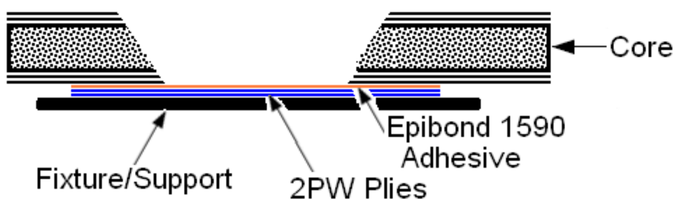

Figure 51-14 Application of the Epibond Adhesive and the Staggered Plies of Plain Weave with a Backing Support

NOTE

If damage is accessible from both sides, backing fixture/support (Aluminum caul plate or similar) can be used after this step.

NOTE

The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).

4. Secondarily bond pre-cured 2 ply PW@45°/ 3mm or 5mm Foam Core using Epibond 1590 adhesive (LAI 135A-925-985) such that it extends minimum +1" beyond the repair region all around. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994-G until handling strength, 6hr per 135A-926-994-H.

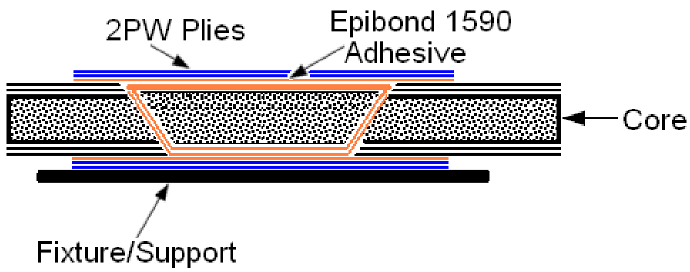

Figure 51-15 Insertion of the New Foam Core and Application of the Epibond Adhesive and Staggered Plies of Plain Weave

5. Elevated temperature post cure the repair in accordance with 135A-926-994.

6. After completion of the cure, inspect the following aspects:

- Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
- Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
- The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.

Based on a successful post repair inspection the part may be released to service.

TYPE 1 OR 2 DAMAGE TO BOTH SIDES OF SANDWICH STRUCTURE (BOTH FLAT SURFACE – ALTERNATE CO-CURE REPAIR)

Damaged regions under this section: Co-cured repair is an alternative method to repair the major damage inflicted to both sides of the sandwich structure. This covers both flat and curved surfaces. Flat surface is shown for clarity only.

NOTE

If the rework/repair is done on the outer surface of the fuselage, it is to be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.

The initial preparation will be done as follows:

- Minimum two Plies of Plain weave (PW) (135A-925-997) will be laid up with same orientation as of the plies removed such that the covering repair patch equals or exceeds the lay-up in the deviant region. Thickness of one cured ply is between 0.0083-0.0089 inches. Repair patch (cured or uncured) must replicate the same number of plies (n) removed plus one i.e. (n+1) from the discrepant/deviation section (For example if 2 plies of carbon PW are present prior to the 3mm /5 mm foam, then the patch repair must be n+1 i.e. 3 plies)
- The Plies need to be staggered such that each ply extends 0.5”-1” beyond the previous ply with the innermost ply of the patch being the smallest.
- Crushed/indented foam core must be removed with chamfered edges 30º-45º.
- Shape and size of the repair foam core must be same as that of the core removed.
- The Patches should be rectangular (rounded edge) or elliptical or circular in shape.

Perform the following procedure to repair the composite

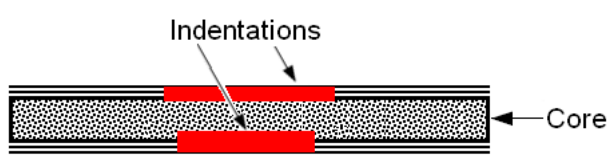

Figure 51-16 Cross Section of Damaged Area

1. Completely grind out the crushed foam core region on the outer and the inner side as shown below:

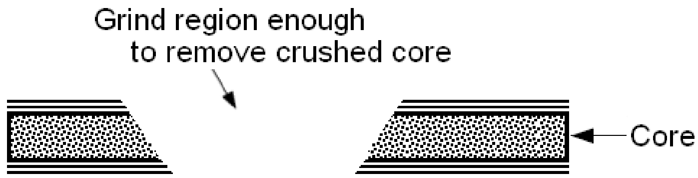

Figure 51-17 Damaged Area after Grinding

2. Apply two plies of Hysol EA9696 (0.03psf) Film adhesive (135A-925-992) such that it extends +1" beyond the exposed foam core region as shown:

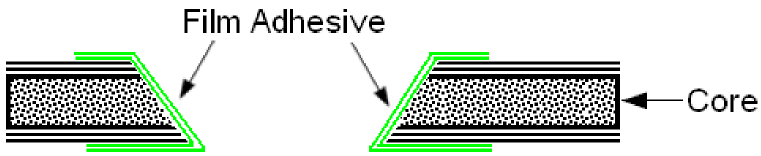

Figure 51-18 Application of the Film Adhesive

3. Lay up at least two plies of Carbon Plain weave (135A-925-997) at +/- 45° such that it covers the film adhesive. (Stagger Plies)

NOTE

The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).

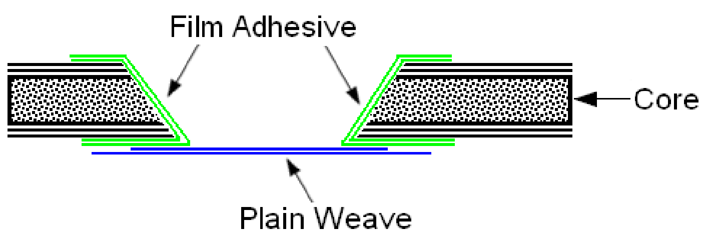

Figure 51-19 Application of Staggered Plies of Plain Weave

4. Apply two plies of Hysol EA9696 (0.03psf) Film adhesive (135A-925-992) such that it extends +1" beyond the exposed foam core region.

NOTE

If damage is accessible from both side, backing fixture/support (Aluminum caul plate or similar) can be used after this step.

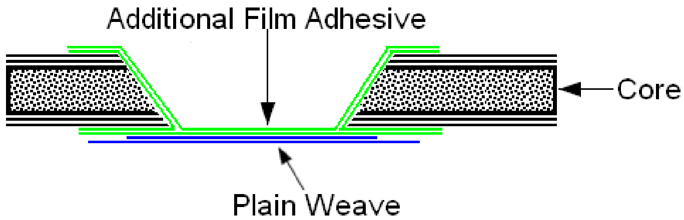

Figure 51-20 Application of Additional Layers of Adhesive Film

5. Create a chamfered 3mm or 5mm Airex foam core piece such that it matches the sanded region and locate it in the repair as shown below

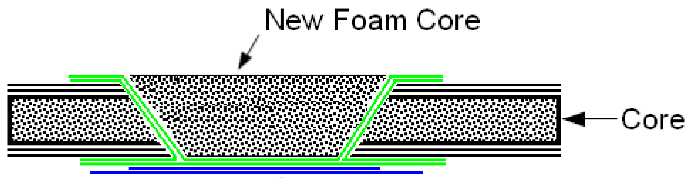

Figure 51-21 Insertion of the New Core

6. Lay up at least two plies of Carbon Plain weave (135A-925-997) at +/- 45° such that it covers the film adhesive. (Stagger Plies)

NOTE

The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).

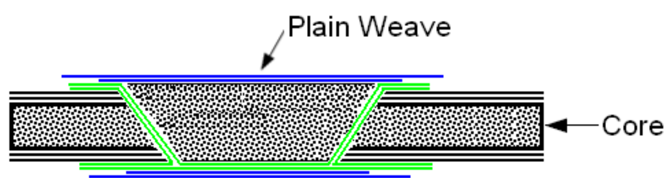

Figure 51-22 Application of the Staggered Plies of Plain Weave

7. Vacuum bag the repair and perform a standard AGATE/TCA cure cycle in accordance with 135A-926-998 specifications.
8. Upon completion of the cure check the following aspects are inspected:

- The cure cycle is within process spec limits (135A-926-998)
- DMA coupon was created and tested TG passed the requirement detailed within 135A-925-997
- The consolidation of the repair is good with no void/ discontinuity as observed by inspection procedures in accordance with 135A-911-042

Based on a successful post repair inspection the part may be released to service.

TYPE 1 OR 2 DAMAGE TO BOTH SIDES OF SANDWICH STRUCTURE (CURVED SURFACE)

Damaged regions under this section: This type of damage repair will be performed when the damaged region is curved.

NOTE

If the rework/repair is done on the outer surface of the fuselage, it is to be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.

NOTE

The two-ply outer skin and 2-ply inner skin are adhered to the fuselage, thus the 'cored' insert will be shorter in overall dimensions such that the two-ply outer skin and inner skin overlay on both the inside and outside of the structures.

Perform the following procedure to repair curved surfaces.

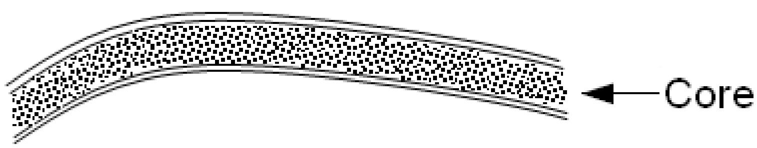

Figure 51-23 Original Shape Before Damage

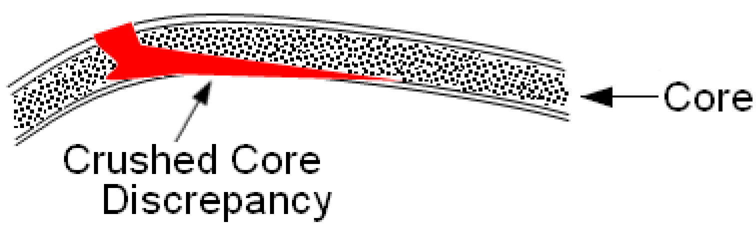

Figure 51-24 Same Area Indicating the Damaged Area

1. Completely grind out the crushed foam core region as follows

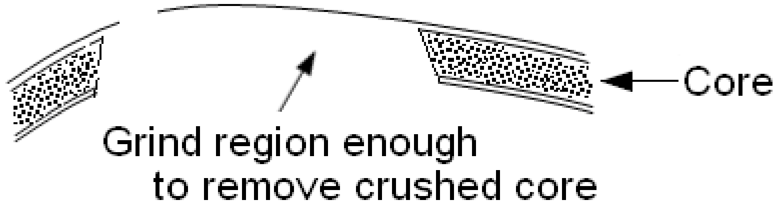

Figure 51-25 Same Region After Grinding to Remove the Damaged Core

2. The 'cored' insert must be cut from a lower fuselage with good cure and good DMA.

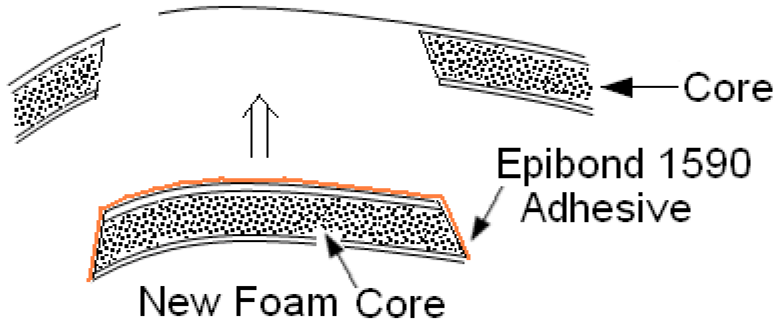

Figure 51-26 The Core Insert Ready for Insertion into Damaged Area

3. This patch is to be bonded using Epibond 1590 adhesive (135A-925-985). The facing sheets (2 ply outer skin and 2-ply inner skin) must overlay the cored insert by at least 1.50" all the way around the cored insert edge. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994-G until handling strength, 6hr per 135A-926-994-H.

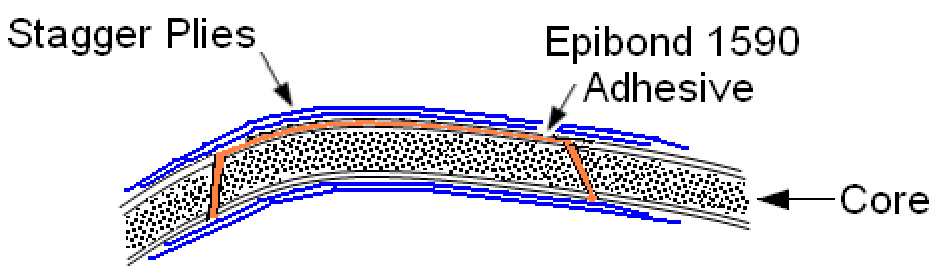

Figure 51-27 Finished Repair Showing the Layers of Plain Weave

NOTE

The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).

4. After completion of the cure, inspect the following aspects:

- Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
- Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
- The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.

Based on a successful post repair inspection the part may be released to service.

NOTE

The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).

TYPE 3 SINGLE SKIN REPAIR
ONE SIDE ACCESSIBLE (CO-CURE REPAIR)

Perform the following procedure to repair Type 3 – damage.

NOTE

If the rework/repair is done on the outer surface of the fuselage, it is to be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.

The initial preparation is done as follows

- Prior to rework, a visual determination needs to be made to check if there is any damage to the underlying foam core
- It must be ensured that the covering repair patch exceed the lay-up in the deviant region per print post repair and with same orientation of ply or plies removed. Thickness of one cured ply is between 0.0083-0.0089 inches. (For example if 2 plies of carbon PW were removed to get to the foreign object, then the repair patch must have 3 plies of Carbon PW)
- The size of the Repair is to be such that the patch extends minimum +1" all around the deviant region. No damage should be caused to the underlying foam core while performing repair process.
- The Plies need to be staggered such that each ply extends 0.5"-1" beyond the previous ply with the innermost ply of the patch being the smallest.
- The Patch should be rectangular (with rounded edges), elliptical, or circular in shape.

The steps described below illustrate the repair techniques for single skin damage, where the damage is accessible from one side only.

The size of the Repair is to be such that the damage region with the foreign object or Delamination must be completely covered with the patch & extends minimum +1" all around the deviant region.

Perform this procedure to repair the composite surface

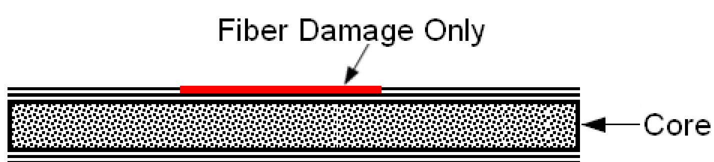

Figure 51-28 Cross Section of Damaged Area

1. Carefully grind out the inner/outer ply (or plies) until the damaged area is completely removed as noted below. No damage should be inflicted to the underlying foam core.

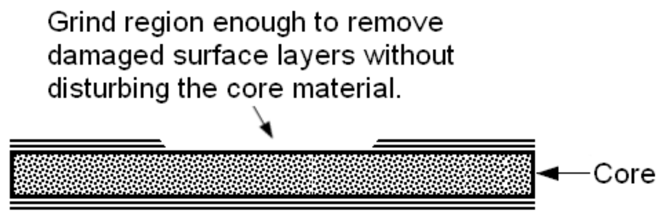

Figure 51-29 Same Area After Careful Grinding to Remove the Damage but Not the Underlying Core Material

2. Clean and surface prep the outer surface and apply TWO plies of Hysol EA9696 Film adhesive (135A-925-992) such that it extends +1" beyond the repair region.

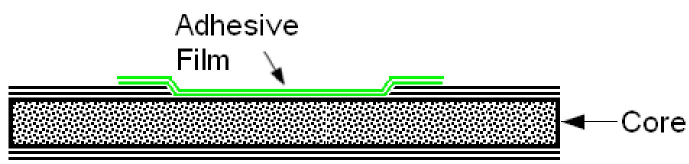

Figure 51-30 Application of the Adhesive Film

3. Lay up at least 2 plies of Carbon Plain weave (135A-925-997) at +/- 45° such that it covers the film adhesive. (Stagger Plies)

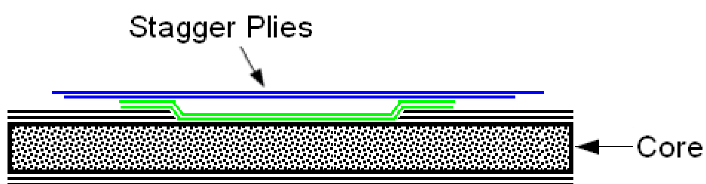

Figure 51-31 Application of the Staggered Layers of Plain Weave

NOTE

The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).

4. Vacuum bag the repair and perform a standard AGATE/TCA cure cycle in accordance with 135A-926-998 specifications.
5. After completion of the cure, inspect following aspects.

- The cure cycle is within process spec limits (135A-926-998)
- DMA coupon was created and tested TG passed the requirement detailed within 135A-925-997
- The consolidation of the repair is good with no void/ discontinuity as observed by inspection procedures in accordance with 135A-911-042

Based on a successful post repair inspection the part may be released to service.

TYPE 3 SINGLE SKIN REPAIR
ONE SIDE ACCESSIBLE (ALTERNATIVE SECONDARY REPAIR)

Perform the following procedure to repair the composite surface.

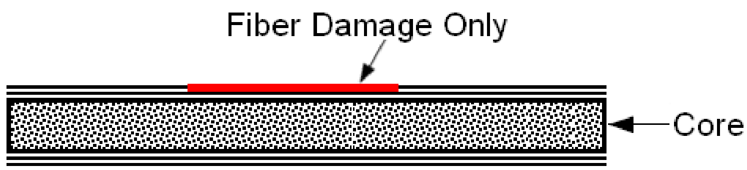

Figure 51-32 Cross Section of the Damaged Area

1. Carefully grind out the inner/outer ply (or plies) until the damaged area is completely removed as noted below. No damage should be inflicted to the underlying foam core.

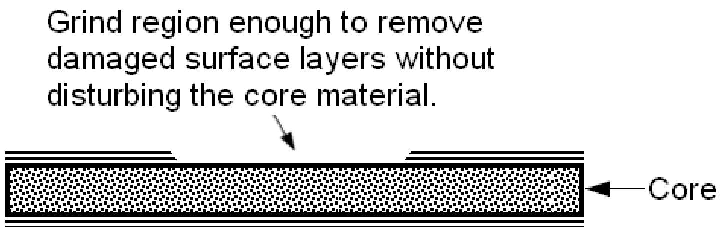

Figure 51-33 Same Area after Grinding the Damaged Layers

2. Clean and prep the surface +1" all around the discrepant/deviant section.

3. Create a secondarily bond of a pre-cured patch of at least 2 ply PW@ +/-45° using Epibond 1590 adhesive (135A-925-985) such that it extends 1" beyond the discrepant region. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994-G until handling strength, 6hr per 135A-926-994-H.

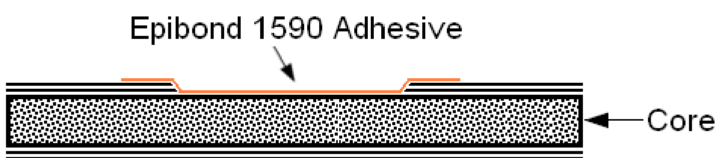

Figure 51-34 Application of the Epibond Adhesive

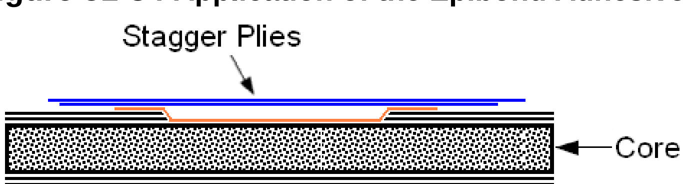

Figure 51-35 Application of the Staggered Plies of Plain Weave

NOTE

The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).

4. Elevated temperature post cured in accordance with 135A-926-994.
5. After completion of the cure, inspect the following aspects.

- Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
- Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
- The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.

Based on a successful post repair inspection the part may be released to service.

TYPE 3 SINGLE SKIN REPAIR
BOTH SIDES ACCESSIBLE (SECONDARY BONDING REPAIR)

This procedure describes the repair of single skin damage when it is readily accessible from both sides. The effective bonding of the repair may be achieved by this process.

Damaged regions under this section: If an outer region is damaged, a satisfactory repair could be made from outside; however, because of the easy accessibility, the repair may be performed from outside against a support that presses hard on the repair until it's hardened/cured. This method of repair can give a smooth molded finish to the external surface.

Perform this procedure to repair the composite surface.

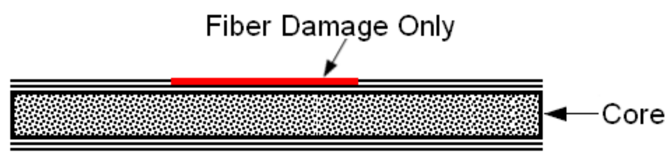

Figure 51-36 Cross Section of the Damaged Area

1. Carefully grind out the inner/outer ply (or plies) the damaged area is completely removed as shown in Figure 51-37.

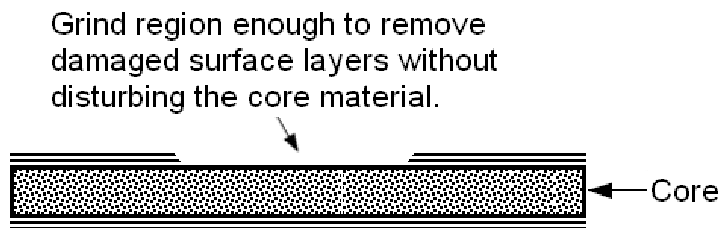

Figure 51-37 Same Area after Grinding the Damaged Layers

2. Clean and prep the surface +1" all around the discrepant/deviant section.
3. Create a secondarily bond with the pre-cured patch of at least 2 ply PW@+/- 45° with Epibond 1590 adhesive (135A-925-985) such that it extends 1.0 in beyond the repair region. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994-G until handling strength, 6hr per 135A-926-994-H.

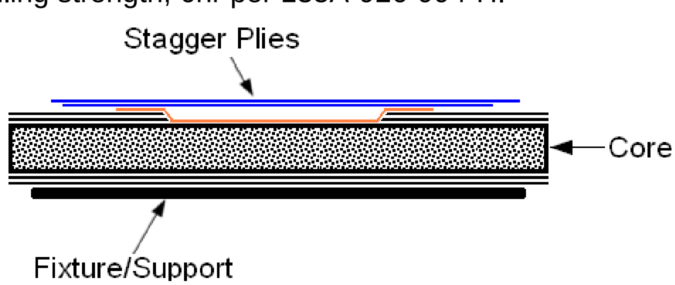

Figure 51-38 Application of the Epibond adhesive and the Staggered Plies of Plain Weave

NOTE

The number of plies and orientation shown are for illustration purpose only. The number of plies laid-up (n+1) depends upon the number of plies removed (n).

4. Elevate the temperature post cured in accordance with 135A-926-994.
5. After completion of the cure, inspect the following aspects:

- Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
- Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
- The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.

Based on a successful post repair inspection the part may be released to service.

Section 51-15 Adhesive Joint Structural Repairs:

The fuselage of the Liberty XL-2 is composed of structural composite materials. The fuselage aft of the engine cowling is fabricated from carbon fiber reinforced fabrics that are used as facing plies adhered to core materials to form a structural sandwich. The main load bearing members of this structure are secondarily bonded together using an epoxy paste adhesive. Care must be taken when performing repairs to the fuselage adhesive joints.

NOTE

Modifications to the fuselage structure including adhesive joints that are performed on form 337 must be coordinated with Liberty Aerospace Inc. Customer Support.

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Section 51-16 Adhesive Joint Damage Classification:

Adhesive joint damage classifications are defined by the following table. These classifications are used solely to define allowable ‘bondline’ defects.

Bond defects include:

- Disbonds (separation between bonded adherents)
- Inclusions (foreign matter in bondline)
- Porosity (frequent small voids in bondline)
- Lack of adhesive (visible due to no squeeze-out)

When there is a doubt regarding classifying a potential defect as a defect, it should be treated as a defect. For example, if a bondline has several air bubbles but it is not clear if it should be treated as a porosity defect, treat it as a porosity defect.

Bond Classification	Maximum Cumulative Defect Size	Defect Accumulation Threshold (DAT)
I	1.0 in^2	12.0 linear inches
II	2.0 in^2	12.0 linear inches
III	3.0 in^2	12.0 linear inches
IV	4.0 in^2	12.0 linear inches
Reserved	Reserved	Reserved

Table 51-6 Bond Defect Classifications

Bond defects are cumulative within the DAT (Defect Accumulation Threshold). In other words, if 1.0 in^2 of defect is allowed and the DAT is 12.0 linear inches, then all bond defects within a given 12 inch length of bondline must be added.

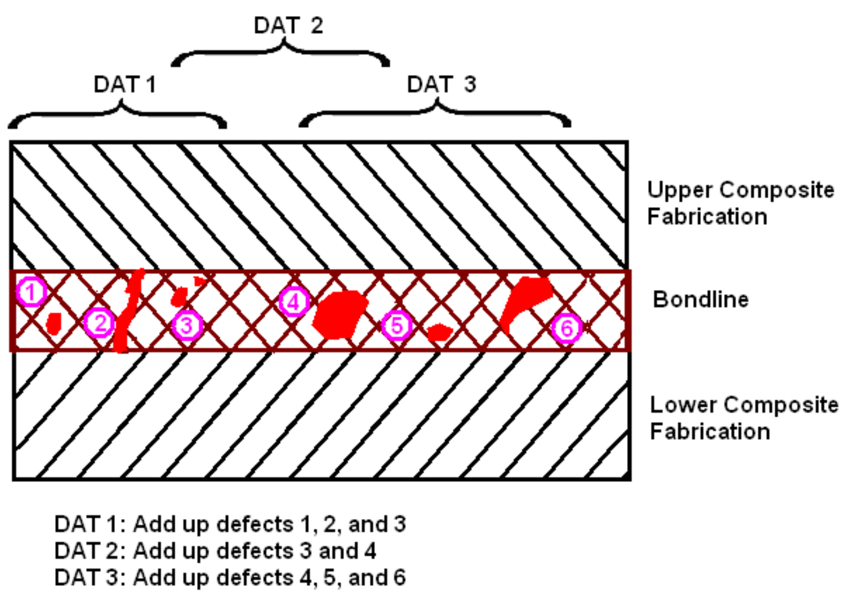

Figure 51-39 Example Application of Bondline DAT

Figure 51-39 above shows how the DAT is used to determine cumulative defects. After all defects are identified, measure out the DAT on either side of each defect to determine what other defects must be treated as cumulative. Damage should therefore be recorded by the maintenance facility as Type 1, Class II etc.

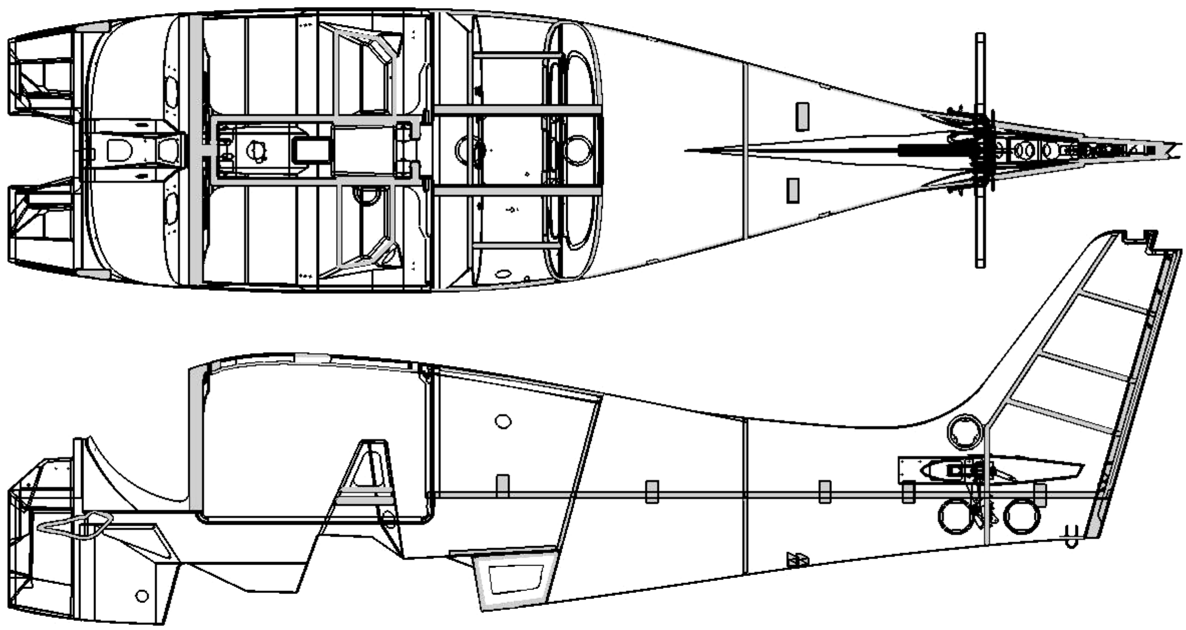

Figure 51-40 The Bond-Line Locations (in Gray) On The Airplane's Fuselage

The table below gives the bond inspection classifications to be used for determining the bondline defects.

Bond Classification	Minimum Required Inspection
NS	Inspection type not specified
A	100% visual + 100% NDI ultrasound
B	100% visual + 100% NDI tap testing
C	100% visual only
Reserved	Reserved

Table 51-7 Bond Inspection Classifications

NOTE

The classification "NS" means that the inspection method is not specified. Any practical method(s) may be used if the method chosen demonstrates that the acceptable defect *criterion may be met for the component.*

"Visual" means optical. Actual inspection may be done via the naked eye, a bore scope, or other imaging means.

This document does not define detailed procedures for specified inspection methods.

A component may always be inspected more intensively than its classification requires. For example, if a bond is classified as "B" (visual and tap test), it may still be inspected with ultrasound testing, or other means in addition to the visual inspection.

Section 16-01 Type 1- Damage:

Type 1-damage is defined as critical damage inflicted to primary or secondary structural joints at locations such as highly stressed (contact Liberty Aerospace Customer support for clarification) regions and underlying structural elements.

Damage is classified as Type 1 when the size, location, and number of damages per unit length or area endanger the structural integrity of the aircraft.

This type of damage requires partial or complete reconstruction of parts or repairs of large areas.

Section 16-02 Type 2- Damage:

Type 2-damage is defined as damage inflicted to primary or secondary structure involving complete penetration of the joint materials.

Damage is classified as Type 2 if it has the potential to affect the structural integrity of the aircraft in flight.

Section 16-03 Type 3- Damage:

Type 3-damage is defined as damage limited to outer skin only (no damage to internal facing plies or adhesive).

Damages are classified Type 3 when the size, location and number of damage per unit length/area does not endanger the structural integrity of the aircraft.

Section 16-04 Type 4- Damage:

Type 4-damage is defined as damage that is inflicted to parts of minimal structural importance.

Type 4 damage includes light surface erosion, scratches, grooves, small dents, etc. that do not penetrate the composite outer skin. This includes damage to replaceable access covers, etc.

Rework to the outer surface of the fuselage can be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.

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Section 51-17 Adhesive Joint Inspection Criterion:

All Liberty XL-2 adhesive joints must meet the requirements of the applicable engineering drawing and material and process specifications.

In addition, bonds must be inspected for defects according to their classification as presented in the table below. The classification defines both the inspection method required and the standard acceptable defect sizes and defect frequencies.

Bond	Classification
Upper to lower fuselage bond	II-B
Roll Over Hoop to upper fuselage bond	II-B
Bulkhead to fuselage bond	II-B
Windscreen to fuselage bond	II-B
Door transparency to door frame bond	II-B
Vertical stabilizer skin to rib bonds	II-B
Vertical stabilizer skin to spar bonds	II-B
Vertical stabilizer closeout spar bond	II-B

Table 51-8 FUSELAGE BOND CLASSIFICATIONS

For the purposes of this inspection, bonds shall be classified by combining the defect and inspection classifications listed in the previous sections. For example, a bond may be classified as “II-B”, or “I-NS”.

The liberty fuselage has a Bond Classification of “B”, as listed in the table above. Qualified personnel using a coin (penny) to find out any damage or defect (hollow sound) can also perform tap testing as defined in Section 51-13 - Laminated Composites Inspection Criterion on page 17 of this chapter. The tester will tap an area small or large enough to ensure that no voids or disbondings exist in the tested component. A void or disbonding is heard to have a “dull” or empty sound. To ensure the area is in fact a void or disbond, testing of the surrounding area is recommended.

NOTE

The classification “NS” means that the inspection method is not specified. Any practical method(s) may be used if the method chosen demonstrates that the acceptable defect criterion may be met for the component.

“Visual” means optical. Actual inspection may be done via the naked eye, a bore scope, or other imaging means.

This document does not define detailed procedures for specified inspection methods.

A component may always be inspected more intensively than its classification requires. For example, if a bond is classified as “B” (visual and tap test), it may still be inspected with ultrasound testing, or other means in addition to the visual inspection.

The specified inspection type is only the initial inspection required. If defects are found, the defects must be thoroughly defined using whatever inspection means are required to do so. For example, if tap testing is required and a defect is found, it may be necessary to use ultrasound to define the defect.

Section 17-01 Tap Testing:

The aircraft fuselage and all adhesive joints have been inspected prior to release, the methods used were visual and tap tests. At the time of aircraft release no ultrasound/radiographic testing was performed, but should be considered, as an alternate means of evaluation if doubt exists as to whether any disbonding exist in structures that require repair.

Section 51-18 Adhesive Joint Repair Methods:

Below is a general representation of the laminates/Sandwich adhesive joint. Refer to this orientation when reading next sections.

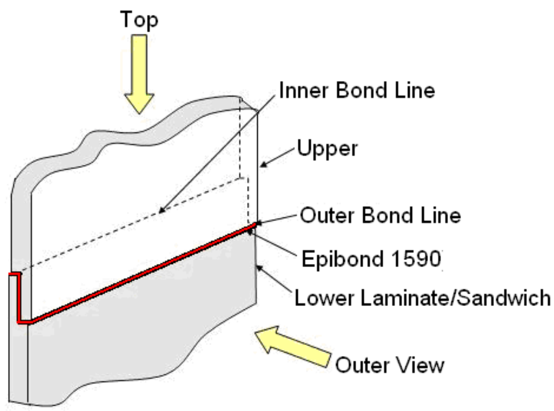


Figure 51-41 Representation of the Laminate/ Sandwich adhesive joint

Section 18-01 Type 1 Damage:

Type 1-damage is defined as critical damage inflicted to primary or secondary structural joints at locations such as highly stressed (contact Liberty Aerospace Customer support for clarification) regions and underlying structural elements.

Type-1 damage primarily consists of regions where:

• Sandwich structure/laminate-fuselage skin disbond

• Upper and lower fuselage bondline disbond

Section 18-02 Type 2 Damage:

Type-2 damage is defined as damage inflicted to primary or secondary structure involving complete penetration of the adhesive joint. Repair methodologies should be consistent with those employed for Type 1-DAMAGE above.

Section 18-03 Type 3 Damage:

Type-3 damage is defined as no damage to the fuselage structure. Type-3 damage is limited to presence of a foreign object and of porosity in the Epibond adhesive.

NOTE

This repair work is performed from inside of the fuselage (IML).

Section 18-04 Type 4- Damage:

Type 4-damage is defined as damage that is inflicted to parts of minimal structural importance.

TYPE-1 OR 2 DISBOND OF ADHESIVE JOINT

Perform this procedure to repair an adhesive joint.

6. This is a general representation of the bonds described earlier under fuselage bond classification.

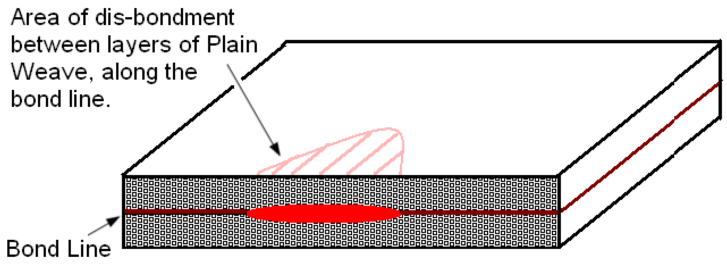


Figure 51-42

7. Clean and surface prep the disbond along the bondline, in accordance with 135A-926-994. All adhesive material needs to be removed down to the bare carbon. Prep the bare carbon material at the removed Epibond adhesive section for good surface adhesion of Epibond adhesive.

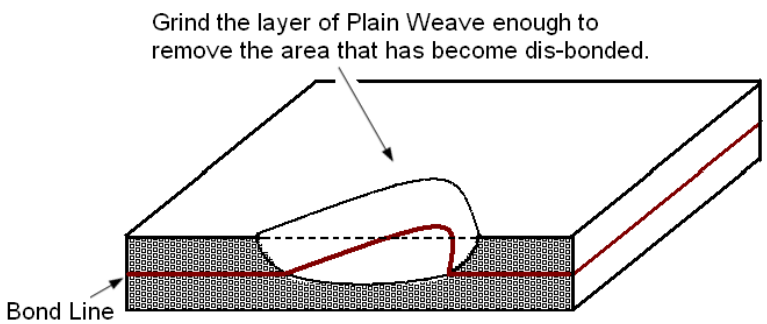


Figure 51-43

NOTE

Efficient bonding of the substrate and Epibond adhesive depends upon clean, moisture free and good prep surface. Underlying carbon ply must NOT be damaged during surface prepping.

8. Fill the gap with Epibond 1590 adhesive (135A-925-985).

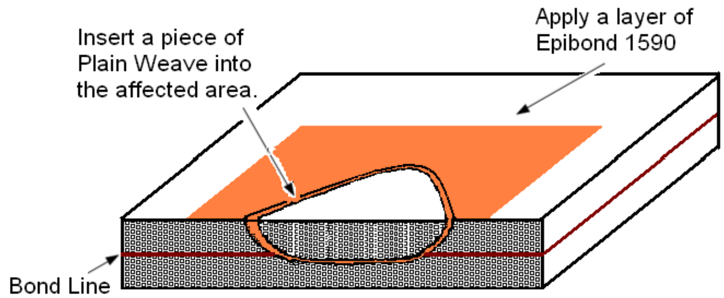


Figure 51-44

9. Secondarily bond a 4-ply reinforcement patch of Carbon Plain Weave (135A-925-997), using Epibond 1590 adhesive in accordance with 135A-926-994. The patch should be elliptical or rectangular (rounded edges) and should cover the disbond and +1in all around. The repair patch should be laid with orientation of ±45 / 0-90 / 0-90 / ±45.

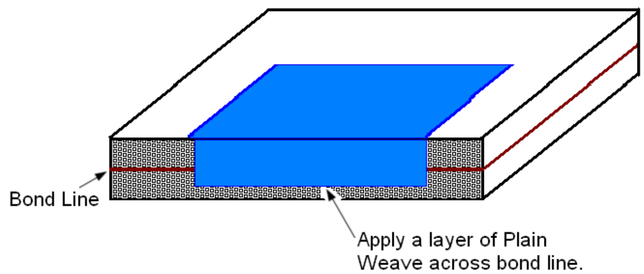


Figure 51-45

10. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994 until handling strength, 6hr per 135A-926-994.
11. Elevate temperatures post cured in accordance with 135A-926-994.
12. When the repair work is done from outside, the surface of, the fuselage is to be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.

NOTE

Flush the outer repair patch to blend it with the profile of the Liberty XL-2.

13. Upon completion of the cure check the following aspects are inspected:

- Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
- Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
- The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.

TYPE-1 OR 2 REPAIR TO BOTH SIDES OF ADHESIVE JOINT LAMINATE TO LAMINATE

Perform this procedure to repair both sides of an adhesive joint

1. Original Configuration:

This is a general representation of the bonds described earlier under fuselage bond classification.

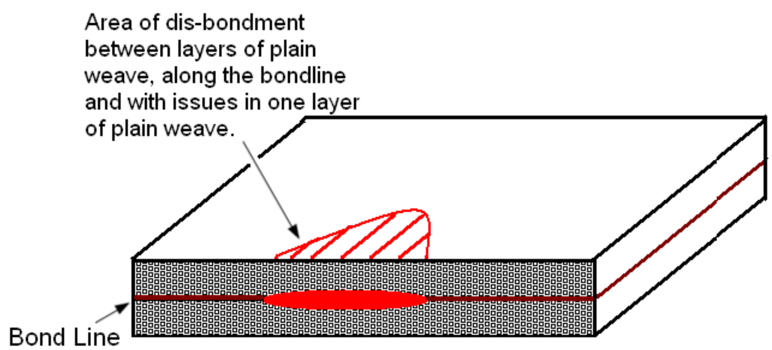


Figure 51-46

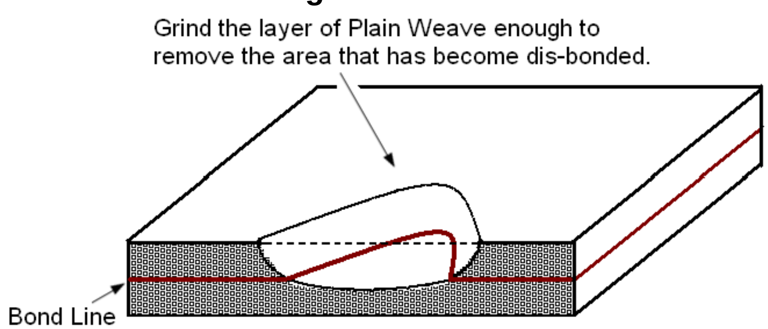
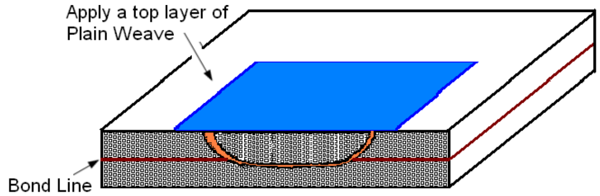
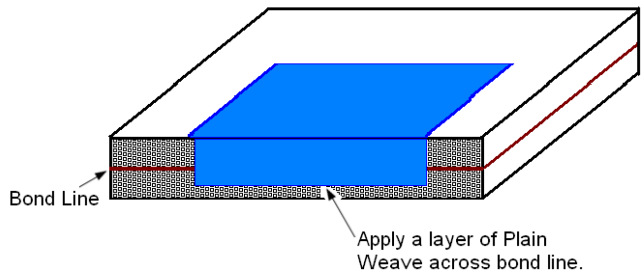


Figure 51-47

2. Clean and prep the discrepancy along the bondline on both inner and outer surfaces, in accordance with 135A-926-994. All adhesive material needs to be removed down to the bare carbon. Also, prep the bare carbon material at the removed Epibond adhesive section.
3. Fill the gap with Epibond 1590 adhesive (135A-925-985).
4. Secondarily bond a 4-ply reinforcement patch of Carbon Plain Weave (135A-925-997), using Epibond 1590 adhesive in accordance with 135A-926-994. The patch should be elliptical or rectangular with rounded edges and should cover the disbond and 1.0 inches on all sides. In addition, it should be laid in the same schedule and orientation of the Bond Line strap i.e. ±45 / 0-90 / 0-90 / ±45.



a) Outer (Front) View



b) Back (Inner) View

c) Top View showing both discrepancies

5. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994 until handling strength, 6hr per 135A-926-994.
6. Elevate temperature post cured in accordance with 135A-926-994.
7. The outer surface of the fuselage is to be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012.
8. After completion of the cure, inspect the following aspects:

- Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
- Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
- The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.

TYPE-1 OR 2 REPAIR OF ADHESIVE JOINT SANDWICH TO SANDWICH

The sandwich-to-sandwich joints in Liberty XL-2 can be observed in the area of Fin Spar where fin ribs are joined to the spar by means of adhesive and fin spar is joined to the upper fuselage by means of adhesive. Any discrepancy with the adhesive joint is hard to access in such areas. Hence, make a through circular hole on a non-discrepant surface near to the discrepant area so that this joint can be made accessible for work and inspection.

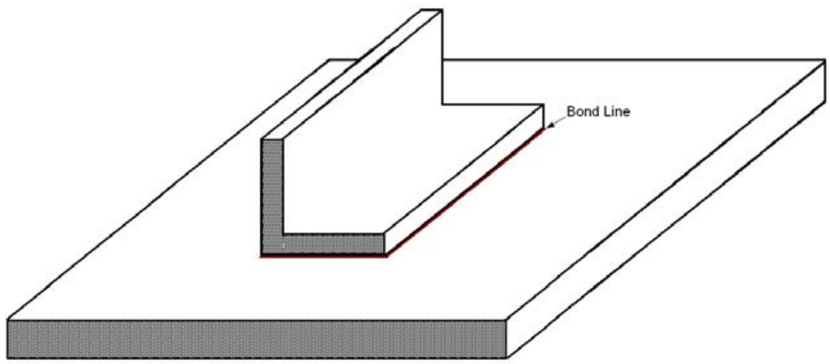


Figure 51-48 Original Configuration:

NOTE

Record the number of plies removed. One (01) cured ply thickness of carbon plain weave is 0.0083in-0.0089in. Repair patch (cured or uncured) must replicate the same number of plies (n) removed plus one (n+1) from the discrepant/deviation section (For example if 2 plies of carbon PW are present prior to the 3mm /5 mm foam, then the patch repair must be n+1or 3 plies)

The size of the Repair is to be such that the patch extends minimum 1" all around the deviant region.

The Plies need to be staggered such that each ply extends 0.5"-1" beyond the previous ply with the innermost ply of the patch being the smallest.

Shape and size of the repair foam core must be same as that of the core removed.

The Patches should be rectangular (rounded edge) or elliptical or circular in shape.

Perform this procedure to repair on the adhesive joint once the joint is made accessible.

1. Locate the effected area. If necessary, an access hole can be cut in the fuselage, only large enough to complete the task.

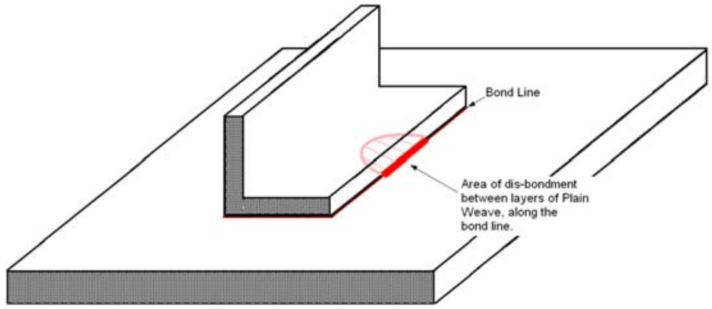


Figure 51-49

2. Clean and surface prep the discrepancy along the bondline, in accordance with 135A-926-994-G. All adhesive material needs to be removed down to the bare carbon. Also, prep the bare carbon material at the removed Epibond adhesive section.

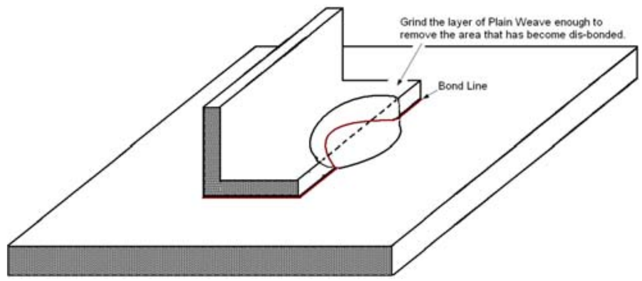


Figure 51-50

NOTE

Efficient bonding of the substrate and Epibond adhesive depends upon clean, moisture free and good prep surface. Underlying carbon ply must NOT be damaged during surface prepping.

3. Fill the gap with Epibond 1590 adhesive (135A-925-985).

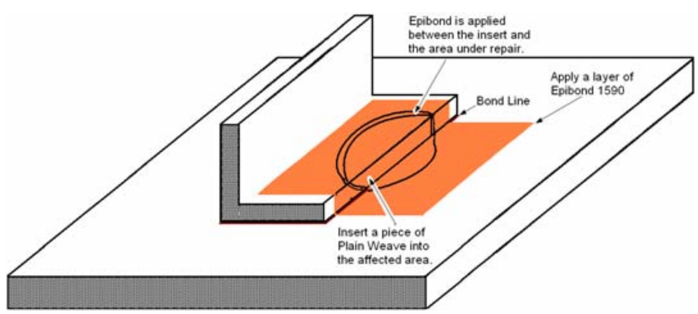


Figure 51-51

4. Secondarily bond a 4-ply patch of Carbon Plain Weave (135A-925-997), using Epibond 1590 adhesive in accordance with 135A-926-994. The patch should be elliptical or rectangular (rounded edges) and should cover the disbond and +1.0in all around. The repair patch should be laid in the orientation of ±45 / 0-90 / 0-90 / ±45.

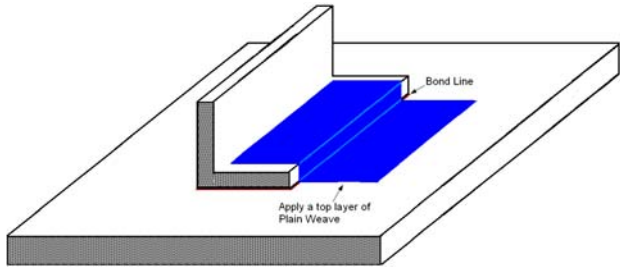


Figure 51-52

5. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994 until handling strength, 6hr per 135A-926-994.
6. Elevate temperature post cured in accordance with 135A-926-994.
7. After completion of the cure, inspect the following aspects:

- Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
- Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
- The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.

8. Once the adhesive repair is complete, repair any hole that used to access the area. Refer to the following procedure Type 1 or 2 Damage To Both Sides Of Sandwich Structure (Flat Surface)-Secondary Bonding on page 26 of this chapter.
9. Elevated temperature post cure the repair in accordance with 135A-926-994.
10. The outer surface of the fuselage is to be smoothed using aerodynamic filler during final painting in accordance with 135A-926-012
11. After completion of the cure, inspect the following aspects:

- Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
- Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
- The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.

Based on a successful post repair inspection the part may be released to service.

TYPE-3 DAMAGE REPAIR PROCEDURE

Below are mentioned two is minor damages defined under type-3 in other words foreign object deposition and porosity. Lack of adhesive also falls under this category. Foreign objects can be visually inspected, but tap testing is performed to confirm the porosity present in an adhesive.

NOTE

The repair for single damage (foreign object or porosity) may also be performed as steps defined below.

Perform this procedure to repair type 3 damage.

1. Original Configuration:

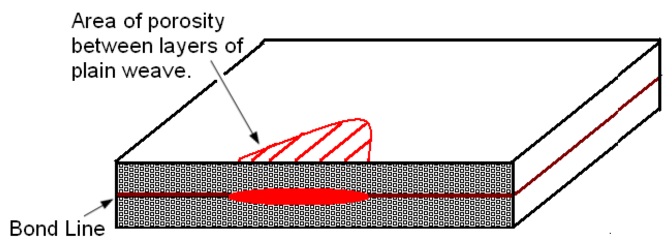


Figure 51-53

2. Remove the foreign object completely. Clean and surface prep the area along the discrepancy (around foreign object and porosity) in accordance with 135A-926-994. All adhesive material needs to be removed down to the bare carbon. Also, prep the bare carbon material at the removed Epibond adhesive (porosity) section.

NOTE

Efficient bonding of the substrate and Epibond adhesive depends upon clean, moisture free and good prep surface. Underlying carbon ply must not be damaged during surface prepping.

3. Fill the discrepant area with Epibond 1590 adhesive (135A-925-985).

4. Secondarily bond a 4-ply reinforcement patch of Carbon Plain Weave (135A-925-997), using Epibond 1590 adhesive in accordance with 135A-926-994. The patch should be elliptical or rectangular (rounded edges) and should cover the foreign object and porosity +1.0 in all directions. The repair patch should be laid in the following sequence i.e. ±45 / 0-90 / 0-90 / ±45.

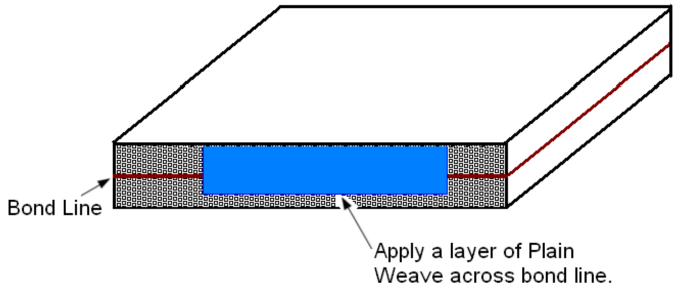


Figure 51-54

5. Mechanically apply a distributed load on the back of the patch to ensure the Epibond adhesive joint is 0.015in (+0.025, -0.010) thick in accordance with 135A-926-994 until handling strength, 6hr per 135A-926-994.
6. Elevated temperature post cure the repair in accordance with 135A-926-994.
7. Upon completion of the cure check the following aspects are inspected:

- Pre cured patch must have good Material traceability (135A-925-997 & 135A-925-001) and must be processed in accordance with 135A-926-998 with good cure and good DMA
- Check for a good SHORE D and DMA for the mixed batch of Epibond adhesive in accordance with 135A-925-985 & 135A-926-994.
- The consolidation of the repair is good with no voids/discontinuities observed during tap testing of the repaired region.

Based on a successful post repair inspection the part may be released to service.

Section 51-20 Minor Structural Metallic Repairs

Minor repairs to the aluminum flying surfaces and controls may be carried out using standard materials and techniques in accordance with FAA Advisory Circular 43.13-1B. These repairs should be limited to patching of small holes < 0.1in or tears in aluminum skins, replacement of moving parts such as hinges, etc. Any repairs involving damage to underlying structural components of flying surfaces (significant skin damage, any damage to underlying structures such as spars, ribs, stringers, etc.) should be referred to Liberty Aerospace Inc Customer Support.

The wings and flying surfaces of the Liberty XL-2 is composed of structural aluminum alloys. These materials are riveted or mechanically fastened to form structural stringer stiffened structures.

NOTE

Major repairs on aileron, flap, horizontal stabilizer, tab and rudder surfaces is by replacement. Removal and replacement or mass balanced flight controls requires flight control rebalancing in accordance with Section 51-60 Control Surfaces Balancing on page 63 of this chapter. Major repairs to the fixed, non-moving, portion of the wing surface must be coordinated with Liberty Aerospace Inc Customer Support.

The chassis and engine mount frame of the Liberty XL-2 is composed of carbon steel tubing materials. These materials are welded to form structural trusses.

Section 20-01 General Corrosion Inspection And Metal Component Protection:

All metal parts of the airplane have been inspected and corrosion protected prior to leaving the manufacturing facility. Aluminum components have been treated with Alodine® EC^2™ Electro-Ceramic Coating, a corrosion inhibitor, a paint primer, and paint. Steel components have been treated with Zinc Chromate a rust inhibitor, a paint primer and paint. Any repair to metal components should be made by qualified personnel in an environment where once paint, primer, and the corrosion/rust inhibitor have been removed the exposed metal will not incur prolonged exposure to the elements to incur more damage.

NOTE

When applying Alodine or Zinc Chromate, care should be taken, as these chemicals are hazardous to humans and the environment in liquid form.

METAL COMPONENT INSPECTION FOR CORROSION OR RUST:

Metal components of the airplane include the rolling chassis, engine mount frame, and wing and tail plane surfaces. The rolling chassis and the engine mount frame are constructed of steel, and the wing and tail plane surfaces are aluminum. These components should be inspected for signs of rust or corrosion.

1. Visually inspect all painted metal surfaces for cracking, pitting, or corrosion or rust.
2. Corrosion or rust will appear as a bulge or lifting of the painted surface.
3. Corrosion will appear as a white or light grey chalky powder.
4. Rust will have a reddish brown appearance and appear grainy.
5. Repair rust or corrosion as stated below.

STRUCTURAL ALUMINUM REPAIRS:

Perform this procedure to effect repairs to the aluminum structure.

NOTE

All precautions should be taken as to local requirements for the handling of chemicals and the disposal of chemically soaked rags, wiping cloths, or materials used in the preparation and repair of the surface.

1. Identify the area to be repaired, isolating it from surrounding areas.
2. Remove paint down to a bare metal surface.
3. Remove any foreign materials.
4. Re-treat the exposed metal surface with Alodine in accordance with the product manufacturer handling procedure.
5. Re-prime the surface for painting.
6. Paint the surface and blend to match the surrounding area.

NOTE

Structural repair for aluminum fuel tanks is mention in chapter 28.

STRUCTURAL STEEL REPAIRS

Perform this procedure to effect repairs to the steel structure.

NOTE

All precautions should be taken as to local requirements for the handling of chemicals and the disposal of chemically soaked rags, wiping cloths, or materials used in the preparation and repair of the surface.

1. Identify the area to be repaired, isolating it from surrounding areas.
2. Remove paint down to a bare metal surface.
3. Remove any foreign materials.
4. Re-treat the exposed metal surface with Zinc Chromate in accordance with the product manufacturer handling procedure.
5. Re-prime the surface for painting.
6. Paint the surface and blend to match the surrounding area.
7. The repair methods prescribed by FAA Advisory Circular 43.13-1B must be used to prevent and repair corrosion.

If tube wall thickness is reduced in diameter by 0.005 inches due to corrosion, or is visibly marred by corrosion or damage, or leads to free-play (movement) between adjacent mating components (for example: tail-plane surface to torque tube interface, or torque tube to fuselage bearings), Liberty Customer support should be contacted such that specific structural repair procedures can be provided.

Section 51-60 Control Surfaces Balancing

This section details the information on balancing the aileron and rudder control surfaces. If the balance on the horizontal stabilizer is suspect, contact Liberty Aerospace, Inc. Customer Service.

Section 60-01 Mass Balance Vertical Clamp Jig

The aileron and rudder mass balance procedure call for a vertical clamp jig. This jig is fabricated locally as needed for these procedures. The jig is made from a 2X4 mounted to the side of a table or bench. Mounting plates attach to the 2X4, and either the aileron or rudder attach to the plates. Once attach, the aileron or rudder should be able to move freely on its hinge. Figure 51-55 shows the mass balance vertical test jig.

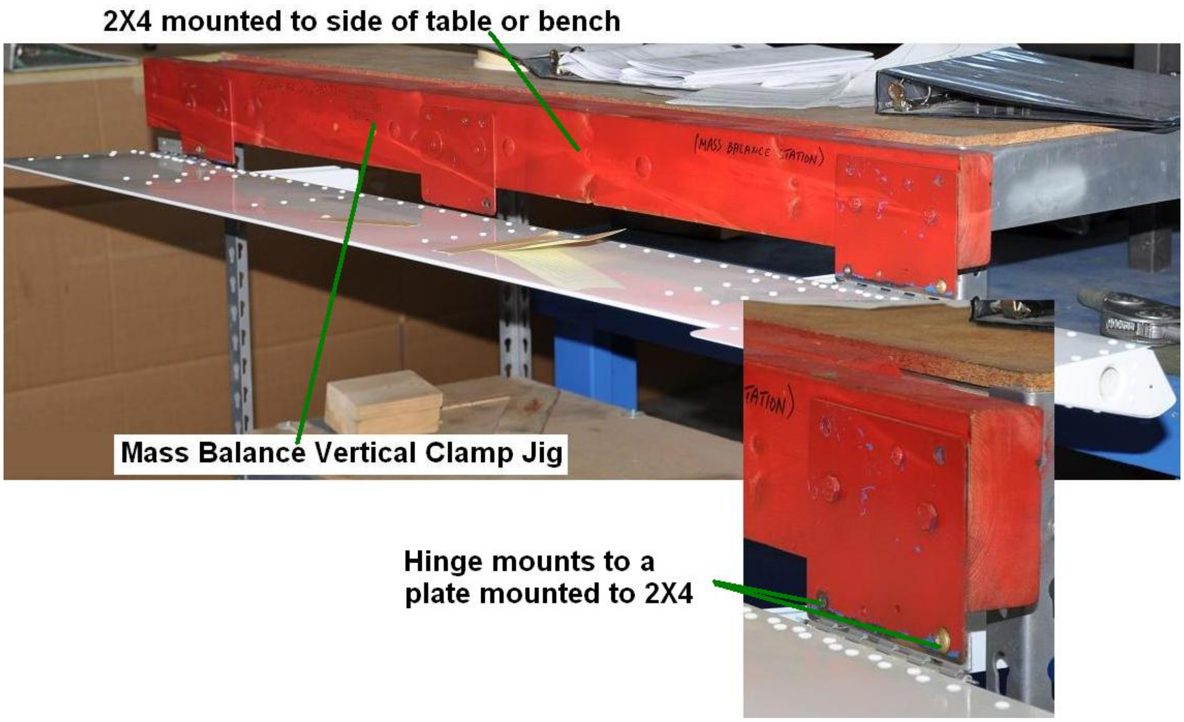


Figure 51-55 Mass Balance Vertical Clamp Jig

Section 60-02 Balancing Procedures

This section details the procedures to balance the aileron, rudder and horizontal stabilizer control surfaces.

AILERON BALANCING

Perform this procedure to check and correct the balance of the aileron. The vertical clamp jig referred to in this procedure is any method that will allow the aileron to swing freely.

NOTE

This procedure can not be done with the aileron mounted on the airplane.

1. If the aileron is mounted to the airplane, remove the aileron as shown in Chapter 27 – *Control Surfaces*.
2. Thoroughly clean all surfaces of the aileron.
3. Hold the two hinges in a vertical clamp jig.
4. Lubricate the hinges so that they are free to operate.
5. Operate the aileron in the vertical clamp jig to verify that the hinges are in the same plane and operate about the collinear axis without binding or galling through their full range of movement.
6. The weight of the mass balance is engineered to be heavy on the balance arm, nose heavy (NH).
7. Locate a load cell (such as a Dillon/Quality Plus, Inc. Force Gauge CFG-50N) under one of the mass balance, 5.0 in from the hinge line. The load recorded on the load cell to bring the lower surface of the aileron horizontal must be between 0 and 0.05 lb. For details, see Figure 51-56.

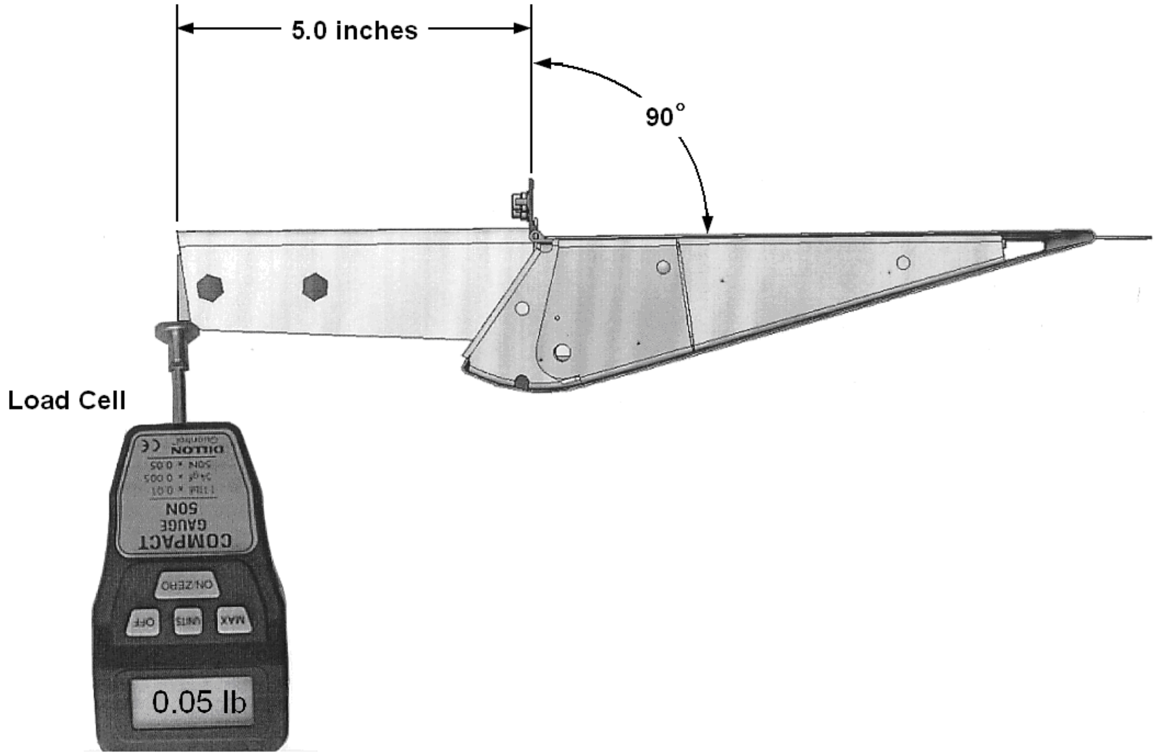


Figure 51-56 Aileron Mass Balancing Setup

8. If the load recorded is higher than 0.05 lb, weight must be subtracted by sanding from both mass balances.
9. In case the aileron is trailing edge heavy, check the surfaces (internal and external) of the aileron for foreign material or other matter. Also, if the aileron has been painted (since leaving manufacturing) remove all finishes (back to the original paint on the aileron) and clean. Repeat step 7 above.
10. Re-attach the surface to the wing, rig assembly per Chapter 27 – *Control Surfaces*.

RUDDER BALANCING

Perform this procedure to check and inspect the mass balance on the rudder. The vertical clamp jig referred to in this procedure is any method that will allow the rudder to swing freely.

1. If the rudder surface is attached to the vertical stabilizer, remove the rudder from the vertical stabilizer. See Chapter 27 – *Control Surfaces*.
2. Thoroughly clean all surfaces of the rudder.
3. Attach the test article to the vertical clamp jig.
4. Lubricate the hinge so that it is free to operate.
5. Operate the rudder in the vertical clamp jig to verify that the hinges are in the same plane operate about the collinear axis without binding or galling through its full range of movement.

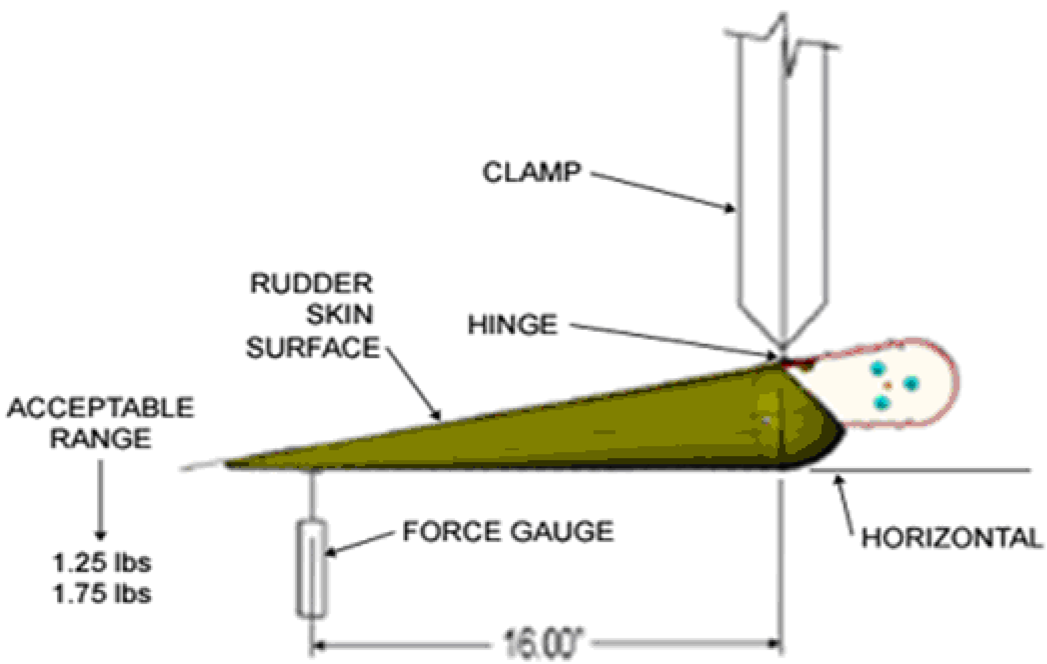


Figure 51-57 Location of Load Cell on Rudder

6. To measure balance, use a load cell positioned beneath the rudder surface root, adjacent to the drive, at a point 16.0 in aft of, and perpendicular to, the hinge center. This point lies almost coincident with the location of the rudder skin to rudder cap joint.
7. Record the load required to hold rudder level.
8. If a load of 1.50 to 1.56 lb is noted, proceed to step 14 below.

NOTE

The intent is to verify that the rudder hinge moment lies between 24 in-lbs and 25 in-lbs.

9. If the load is outside of the range of 1.5 to 1.56 lbs., check all surfaces (internal and external) for foreign material or other matter. Re-clean the surfaces of the rudder and go back to step 6 above.
10. If the upward load is more than 1.56lb, it will require the removal of the rudder horn from the rudder assembly. Remove the rudder horn from the rudder assembly by de-riveting. Gain access to the mass balance inside the rudder horn and add mass by means of adding washers (NAS 1149FO 332P or equivalent), as needed, to the mass balance mounting screws. If washer stack exceeds safe thread length of screws, increase screw length as required. Install the rudder horn temporarily to the rudder assembly, and repeat step 6 above.
11. If the upward load is less than 1.5 lb, it will require the removal of the rudder horn from the rudder assembly. Remove the rudder horn from the rudder assembly by de-riveting. Gain access to the mass balance inside the rudder horn and remove washers, as needed for required weight, from mass balance mounting screws. It may be necessary to remove/shave off a portion of the mass balance (from the aft side). Fine adjustment may be made on the assembled rudder by drilling the Mass Balance – Horn (135A-30-643) thru the 0.201in tooling hole using a #7-drill bit. If removed from the rudder, install the rudder horn temporarily to the rudder assembly, and repeat step 6 above.
12. After completing step 9 or 11, permanently attach the rudder horn to the rudder assembly using rivets.
13. After assembling the rudder horn and rudder together (permanently), repeat steps 3 through 11 until rudder is made level within the stated tolerance.
14. Attach the rudder to the vertical stabilizer.
15. Check the rigging; see Chapter 27 – *Control Surfaces*.

This completes the Rudder Balancing procedure.

HORIZONTAL STABILIZER BALANCING

At this current time, Liberty Aerospace, Inc. has not defined a field procedure to check the balance on the horizontal stabilizer. If the balance on the horizontal stabilizer is suspect, contact Liberty Aerospace, Inc. Customer Service.

Section 51-80 Ground and Bonding

This section details the information on grounding and bonding on the airplane that are specific to the Liberty XL-2 airplane.

Section 80-01 Ground Studs

Ground studs are permanently installed on the airplane metallic structure and shall be treated as permanent bonds. See Figure 51-58.

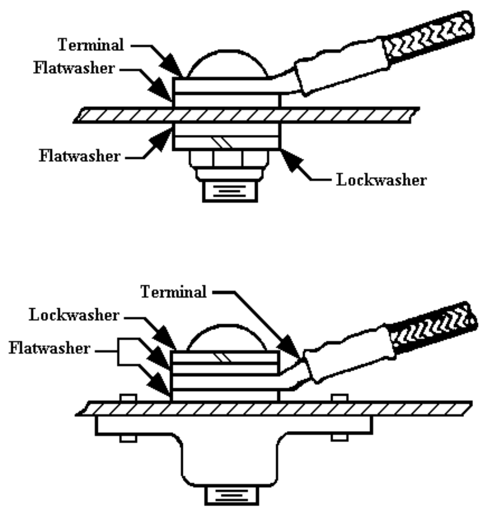


Figure 51-58 Ground Studs

Section 80-02 Bulkhead Connectors

Bulkhead connectors that are used for termination of cable shields shall be bonded to the airplane structure with a maximum resistance of 0.003 ohm. See Figure 51-59.

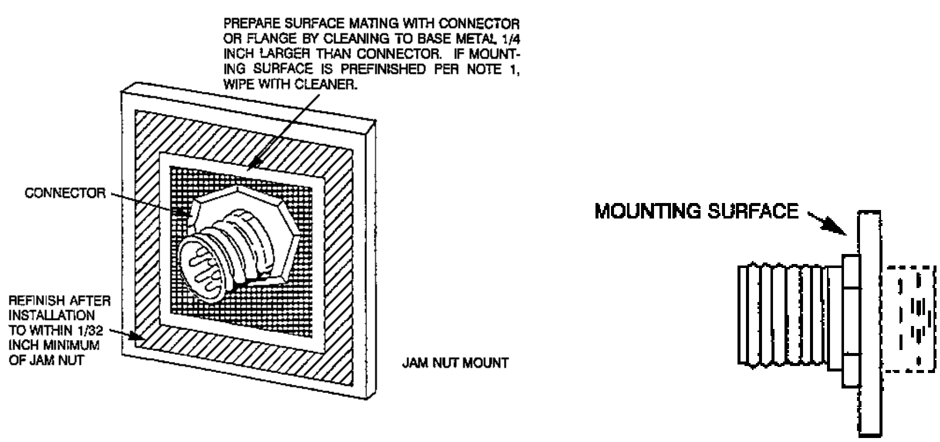


Figure 51-59 Bulkhead Connectors

Section 80-03 Metallic Pipes, Tubes and Hoses

Metallic pipes, tubes, hoses and etc. that carry fluids in motion shall be bonded to structure as shown in Figure 51-60.

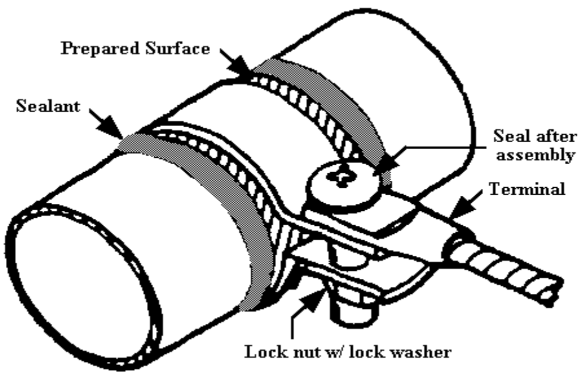


Figure 51-60 Bonding to a Metal Pipe/Tube/Hose

Section 80-04 Carbon Fiber Composite to Aluminum Bonds

Aluminum components cannot be placed in direct contact with Carbon Fiber Composites (CFC) components due to corrosion that occurs with such contact. Direct contact between aluminum and CFC components shall be prevented by the use of insulation (i.e. fiberglass, Mylar, paint, etc.) between these components. Bonding between these materials shall be performed by use of Corrosion Resistant (CRES) steel or titanium fasteners between the components. See Figure 51-61 for details.

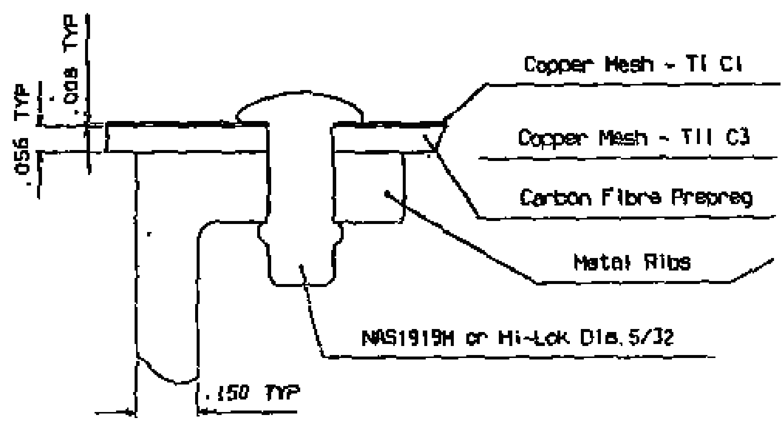


Figure 51-61 Composite to Metal Bonding

Section 80-05 Antenna Bonds

Antenna mounting plates shall be mounted to the carbon fiber composite skins to enhance the ground plane and for structural support. Refer to Figure 51-62 for antenna installation.

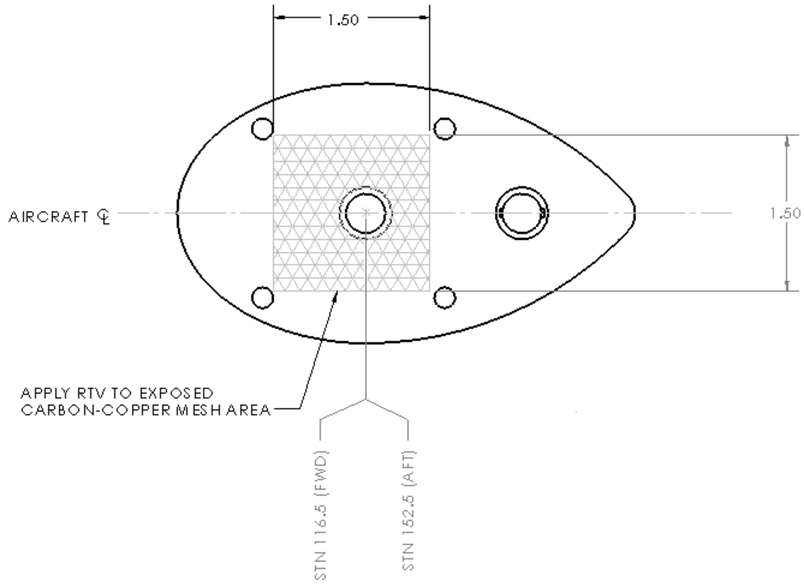


Figure 51-62 Antenna Bounding

Section 80-06 Inspection Procedures for Grounding and Bonding

This section details the procedures for checking and inspecting the airplane's bonding and grounding systems.

GROUND AND BONDING INSPECTION

Perform this procedure to check and inspect the airplane's ground and bonding system. Although a standard digital voltmeter will suffice, Liberty Aerospace, Inc. recommends the use of a calibrated milliohm meter.

1. Remove the belly panel and engine cowlings and inspection covers. See Chapter 53 – *Fuselage*.
2. Open inspection covers in the wings. See Chapter 57 – *Wings*.
3. Refer to Figure 51-63 for Test points.

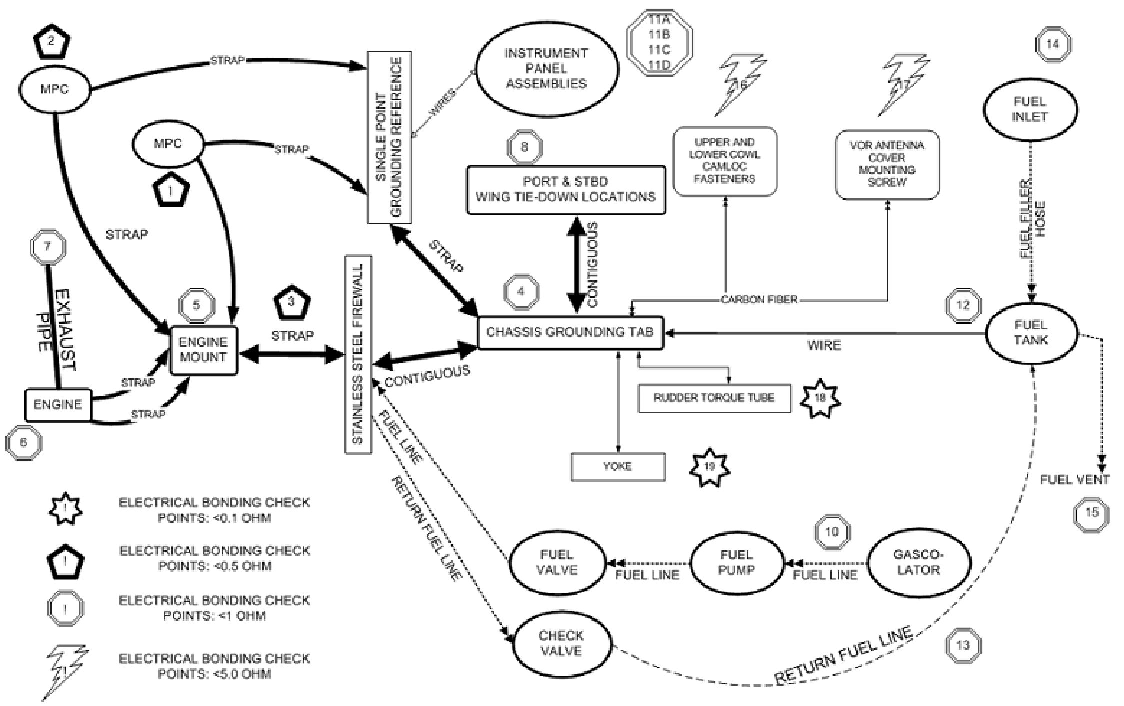


Figure 51-63 Diagram of the Ground Checkpoints for Testing

4. Refer to Table 51-9 for testing points. Use copies of the table during the inspection process.
5. Test each checkpoint to the nearest chassis ground point.
6. Indicate the resistance reading in the table.

Aircraft Serial Number:			Date:	
Checkpoint	**Location**	**Maximum Value**	**Value Received**	**Pass?**
1	Starboard MPC Ground Bolt	< 0.5 Ω	Ω	
2	Port MPC Ground Bolt	< 0.5 Ω	Ω	
3	Stainless Steel Firewall	< 0.5 Ω	Ω	
4	Chassis Grounding Tab	< 0.5 Ω	Ω	
5	Engine Mount Ground Tab	<1.0 Ω	Ω	
6	Engine Block	<1.0 Ω	Ω	
7	Exhaust Pipe	<1.0 Ω	Ω	
8	Starboard and Port Wing Tie-downs	<1.0 Ω	Ω	
10	Fuel Line Between Gascolator and Fuel Pump	<1.0 Ω	Ω	
11A	Aluminum Panel: Circuit Breakers	<1.0 Ω	Ω	
11B	Aluminum Panel: Avionics	<1.0 Ω	Ω	
11C	Aluminum Panel: Instruments	<1.0 Ω	Ω	
11D	Aluminum Panel: Console	<1.0 Ω	Ω	
12	Fuel Tank	<1.0 Ω	Ω	
13	Fuel Line Between Tank and Check Valve	<1.0 Ω	Ω	
14	Verify resistance between refueling inlet and composite fuselage and between refueling inlet and chassis	<5.0 Ω	Ω	
15	Fuel Vent Line on Fuel Tank	<1.0 Ω	Ω	
16	Camloc Fastener on Upper Cowl	<5.0 Ω	Ω	
17	VOR Antenna Cover Mounting Screw	<5.0 Ω	Ω	
18	Verify resistance between rudder pedal control circuit and chassis across braided connectors	<0.10 Ω	Ω	
19	Verify resistance between aileron and elevator control circuit and chassis across braided connectors	<0.10 Ω	Ω	

Table 51-9 Table of Ground Checkpoints for Testing

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CHAPTER 52

DOORS

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Table of Contents

SECTION 52-00	GENERAL	5
SECTION 52-01	PASSENGER / CREW DOORS	7
SECTION 01-01	OPENING THE DOOR - EXTERIOR	7
SECTION 01-02	OPENING THE DOOR - INTERIOR	8
SECTION 01-03	DOOR REMOVAL AND INSTALLATION PROCEDURES	8
DOOR REMOVAL		9
DOOR INSTALLATION		10
DOOR SEAL REMOVAL		11
DOOR SEAL INSTALLATION		12
SECTION 01-04	DOOR LATCH REMOVAL AND INSTALLATION:	13
DOOR LATCH REMOVAL		14
DOOR LATCH INSTALLATION		16
SECTION 01-05	DOOR LOCK REMOVAL AND INSTALLATION	18
DOOR LOCK REMOVAL		19
DOOR LOCK INSTALLATION		20
SECTION 01-06	DOOR STRAP REMOVAL AND INSTALLATION	21
DOOR STRAP REMOVAL		22
DOOR STRAP INSTALLATION		23
SECTION 01-07	GAS SPRING CYLINDER REMOVAL AND INSTALLATION:	24
GAS SPRING CYLINDER REMOVAL		25
GAS SPRING CYLINDER INSTALLATION		26
SECTION 01-08	DOOR TROUBLESHOOTING	27
SECTION 52-02	EMERGENCY EXIT	29

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Section 52-00 General

The airplane has two large "gull wing" or swing-up doors to provide pilot and passenger access to the flight compartment. See Figure 52-1 for details on the location of the two doors. These doors also provide access to the baggage area behind the seats; there is no provision for a separate baggage door.

Figure 52-1 View of Aircraft Doors

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Section 52-01 PASSENGER / CREW DOORS

A large door on each side provides passenger and crew access. The doors are top-hinged to the upper fuselage and open upward. In the open position, a gas cylinder (“gas spring”) provides support for each door. The gas spring contains oil, which provides a dampening function as the door approaches the fully opened position.

Each door incorporates a transparent or clear Acrylic door screen, which forms the cockpit side window (windscreen). As an option, there is an auxiliary vent window mounted in the door screen. A rubber seal around the entire periphery of each door prevents air or water from entering the cabin.

Over-center geometry of the gas spring provides a small closing force to assist door latching and sealing when the door nears its fully closed position. There is no provision to hold the door in intermediate positions between fully open and fully closed.

NOTE

The airplane doors can be in the open or up position while the airplane taxis (maximum 1100 rpm) or during surface winds of less than 10 knots. However, if winds are more than 10 knots, or before applying more than 1100 rpm, close and latch both doors.

Forward and rear pins installed in the door fit into pin bushings in the fuselage on each doorframe. There is a hanging hand strap located along the bottom, near the center of the door. Use this strap when opening and closing the door.

Section 01-01 Opening the Door - Exterior

Each door has an exterior door latching mechanism on the forward, lower area of the door. See Figure 52-2 for the location of the exterior door handles.

To use the handle to open door, press the recessed end of the handle, this will extend the main handle outward from the recessed position. To open the door, rotate the handle fully downward. To close the door, rotate the handle fully upward. When finished, rotate the handle to the center position and return the handle to the original recessed location.

Figure 52-2 Location of the Exterior Door Handle (Port Side Shown)

Section 01-02 Opening the Door - Interior

Each door has an interior door latching mechanism located on the forward, lower area of the door. See Figure 52-3 for the location of the interior door latching mechanism.

To open the door, pull the handle to its opened position (towards the aft of the airplane). To close and lock the door, move the handle to the closed position (towards the front of the airplane). When opening or closing the door, if the handle is not in the fullest extended closed position, the mechanism of the door latch will return the handle to the opened position.

Figure 52-3 Interior Door Latching Mechanism (Starboard Side Shown)

WARNING

PRIOR TO FULL ENGINE RUN-UP AND FLIGHT, BE SURE TO CLOSE BOTH DOORS. CHECK THE DOORS FOR PROPER SEATING AND CHECK THAT LATCHES ARE SECURE BY PUSHING OUTWARD ON THE FORWARD AND AFT AREAS OF THE DOOR. THIS WILL PREVENT DOOR VIBRATION OR THE DOORS FROM OPENING DURING RUN-UP OR FLIGHT.

SECURELY LATCH BOTH DOORS BEFORE TAKEOFF. UNLATCHING A DOOR DURING FLIGHT MAY RESULT IN THE DOOR BEING JETTISONED FROM THE AIRPLANE, WITH POTENTIAL DAMAGE TO OTHER AREAS OF THE AIRPLANE.

Section 01-03 Door Removal and Installation Procedures

This section has the procedures to remove and install the doors on the airplane. To access to the door hinges and hinge pins, gain access by removing interior access panels from the cabin headliner assembly. To access the door latches and linkages, gain access by removing the door access cover.

DOOR REMOVAL

Perform the following procedure to remove the door from the airplane. This procedure requires a Phillips screwdriver.

1. Fully open the door.
2. Using the Phillips screwdriver, remove the two upholstery access covers from cabin headliner to reveal the door hinge pins.
3. Support the door in open position and disconnect gas cylinder end from door. The cylinder end incorporates a locking ring. After removing the locking ring, snap the cylinder off the ball fitting in door. The piston may extend slightly farther than normal door open position.
4. Carefully lower the door to the closed position.
5. Push the blue button in the center of each door hinge pin to withdraw the pin from the hinge. See Figure 52-4 for the location of an installed hinge pin, and see Figure 52-5 for details of the hinge pin.
6. Remove the door from the airplane.

Figure 52-4 Typical installation of the door hinge pin

Figure 52-5 Detailed view of the door hinge pin

DOOR INSTALLATION

Perform the following procedure to install the door on the airplane. There are no tools required for this procedure.

1. Hold door in closed position.
2. Align hinges and re-insert hinge pins.
3. Slightly compress gas spring and snap cylinder end of gas spring to fitting in door.
4. Insert the locking ring.
5. Replace the two cabin headliner upholstery access covers.

DOOR SEAL REMOVAL

Perform the following procedure to remove the door seal.

1. Fully open the door.
2. Grasp the one-piece door seal at the break.
3. Pull the door seal up and out. The seal will separate from the fuselage.
4. Clean any silicon rubber sealant from the edge of the doorframe.

NOTE

When removing the silicon rubber from the exterior surface, try not to remove any of the surface finish. Remove any adhesive residue by using a surface cleaner, such as isopropyl or denatured alcohol on a soft cloth.

CAUTION

Do not use any type of chemical solvent, such as M.E.K. (Methyl Ethyl Ketone) or acetone on any of the composite surfaces.

DOOR SEAL INSTALLATION

Perform the following procedure to install the door seal. This installation requires the use of a rubber mallet and a razor knife.

1. Cut an 11 foot 6 inch length of door seal. Do not trim the door seal until step 5.
2. Apply a bead of RTV100 Silicon sealer to the forward, top, and aft edges of the doorframe.
3. Install the one-piece door seal starting at the bottom, 5 inches forward of the aft doorframe. Work in a clockwise direction, all around the frame, ending at the same point. See Figure 52-6 for details.

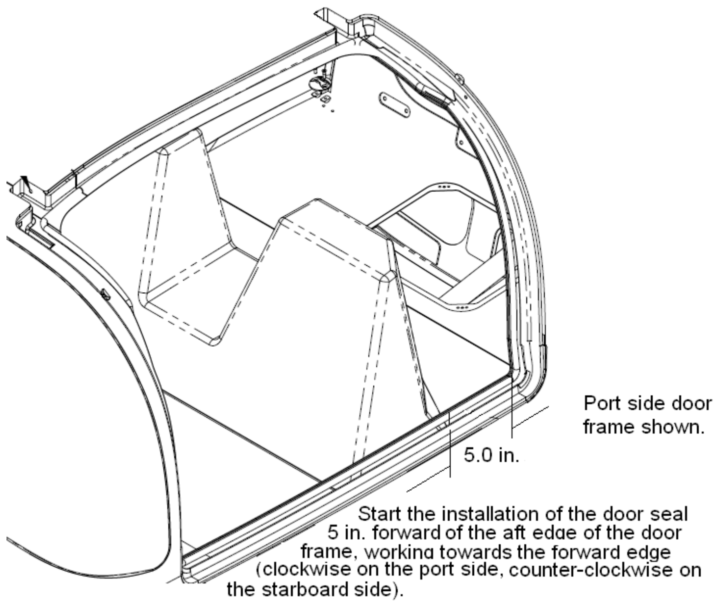

Figure 52-6 Location of the Starting Point to Install the Door Seal

4. Use a small rubber mallet to tap the seal into place. Pay close attention to the corner areas and all curved areas of the fuselage.

NOTE

If the installation of the door seal is correct, there should be only about a 1 to 2 inch overage. If there is more than a 2-inch of overage, recheck the seal to insure installation is complete in each of the four corners.

5. Once the silicon rubber has cured, carefully trim any excess seal such that each end of the seals comes together without a gap. Close the door, there should be no other adjustments required.
6. Test the door seal for proper installation. To perform a seal evaluation, pouring water through door gaps. In the event the door does not seal properly, contact Liberty Aerospace Customer Service.

Section 01-04 Door Latch Removal and Installation:

This section details the procedure to remove and install the door latch mechanism. See Figure 52-7 for an exploded view of the door latch and lock mechanisms.

Some parts shown in Figure 52-7 are for reference only and are not available as an individual field replaceable part. Those parts are part of an assembly with other associated parts. See notes in the associated procedures.

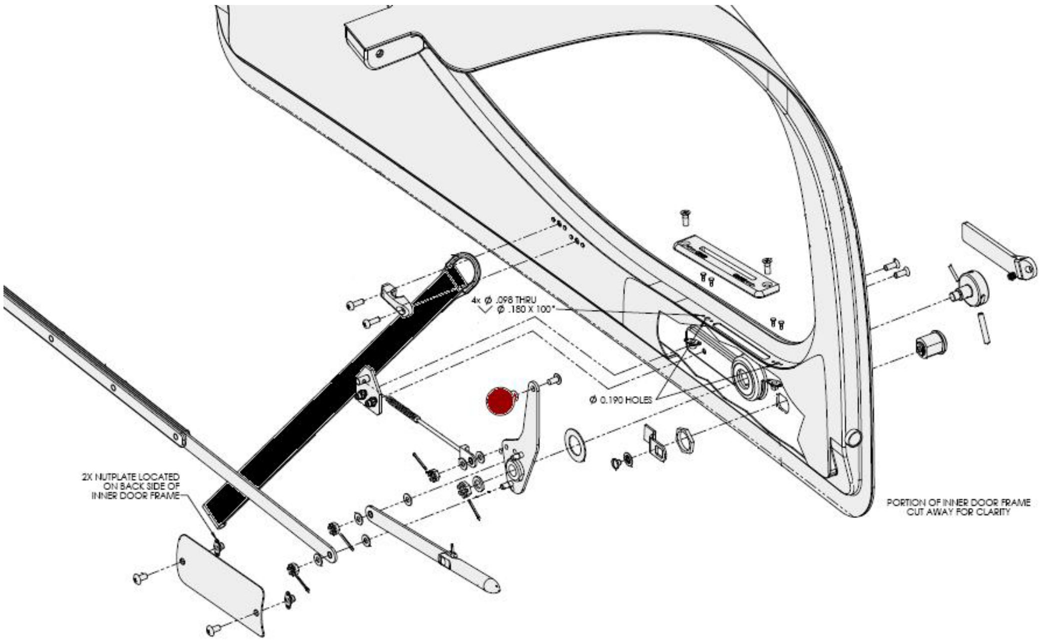

Figure 52-7 Exploded View of the Door Latch and Lock Mechanisms

DOOR LATCH REMOVAL

Perform the following procedure to remove the door latch mechanism. See Figure 52-8 for an exploded view of the parts available for field replacement. The following tools are required to work on the door latch: Phillips screwdriver, standard screwdriver, right angle Phillips screwdriver, side cutters, pointed alignment tool, ⅜-inch socket, ½-inch socket, 2 pieces of 18 - 22 gauge wire (.032 Max.) 30 inches long.

Make the pointed alignment tool from an ice pick. Put a bend 1-inch from the end of the ice pick. The bend should be approximately 45°.

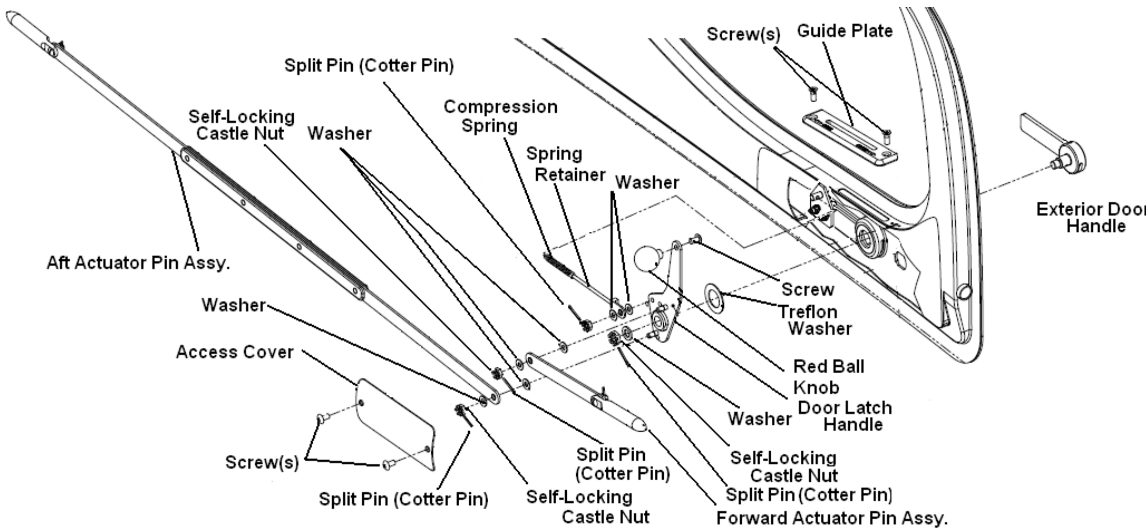

Figure 52-8 Door Latch Assembly (Port Side)

NOTE

Perform this procedure with the door on or off the airplane. However, the recommendation is to remove the door from the airplane.

1. If the door is on the airplane, fully open the door. If removing the door from the airplane, perform the procedure outlined in subsection 0 Door Removal at this point.
2. Remove the two screws securing the access cover and remove the access cover.
3. Remove the split pins from the castle nuts for the forward and aft actuator pin assemblies.
4. Use a ⅜-inch socket to remove the castle nuts. Remove the first washer. Do not remove the actuator pin assembly from the stud on the handle.
5. Thread a wire through the hole in the actuator pin.

6. Lift the actuator pin assembly from the stud on the door latch handle. If removing the actuator pin from the door, use the wire to assist in guiding the actuator pin through the interior of the door. Leave the wire in place (threaded through the door) after removing the actuator pin assembly from the door. If the actuator pin assembly will stay in the door, move the actuator pin assembly to position that will not interfere with the removal of the door latch handle. Remove the second washer from the door latch handle.
7. Use the right angle Phillips screwdriver to remove the red ball knob from the door latch handle.

NOTE

The recommendation is to protect the surface of the door screen (window) with a paper back tape. The door screen is Acrylic.

8. Remove the split pin from the castle nut securing the door latch handle to the exterior door handle.
9. Use a ½-inch socket to remove the castle nut.
10. Remove the door latch handle from the door. When removing the door latch handle from the door, the door latch handle will be under some tension from the compression spring. Make sure the compression spring comes with the door latch handle, and does not fall in to the interior of the door.
11. Remove the exterior door handle. Remove the Teflon washer.
12. Remove the split pin from the castle nut securing the spring retainer to the door latch handle.
13. Use a ⅜-inch socket to remove the castle nut. Remove the washer.
14. Remove the spring retainer from the stud on the door latch handle. Remove the second washer from the door latch handle.

DOOR LATCH INSTALLATION

Do the following procedure to install the door latch mechanism. . See Figure 52-8 for an exploded view of the parts available for field replacement. The following tools are required to work on the door latch: Phillips screwdriver, standard screwdriver, right angle Phillips screwdriver, side cutters, pointed alignment tool, ⅜-inch socket, ½-inch socket, 2 pieces of 18 - 22 gauge wire 30 inches long.

Make the pointed alignment tool from an ice pick. Put a bend 1-inch from the end of the ice pick. The bend should be approximately 45°.

NOTE

Perform this procedure with the door on or off the airplane. However, the recommendation is to remove the door from the airplane. This procedure will assume the door is already off the airplane.

1. Install a washer on to the stud for door latch handle for the spring retainer.
2. Install the spring retainer on to the door latch handle.
3. Install the second washer on top of the spring retainer.
4. Thread on the ⅜-inch castle nut until snug. Back the nut off enough to align the castle nut with the hole in the stud. The castle nut is on with the correct torque if the spring retainer can easily move on the stud, but is not overly loose.
5. Install the split pin. The recommendation is to use a new 1/16-inch diameter split pin.
6. Use the pointed alignment tool to move the hole in the canopy latch pivot to ease the installation of the spring retainer.
7. Slide the compression spring on to the spring retainer.
8. Carefully install the door latch handle and inserting the spring retainer in to the hole in the canopy latch pivot.
9. Install the Teflon washer.

NOTE

Install the door latch handle without disturbing the Teflon washer. The door latch handle can cut and/or crush the Teflon washer. If damage occurs to the Teflon washer, the door latch will not work properly.

10. Install the exterior door handle through the door.
11. Thread on the ½-inch castle nut until snug. Back the nut off enough to align the castle nut with the hole in the exterior door handle stud.

12. Install the split pin. The recommendation is to use a new 1/16-inch diameter split pin.
13. Install the forward and aft actuator pin assemblies using the guide wire that from steps 5 and 6 in the procedure in subsection Figure 52-7 Door Latch Removal.
14. Install one washer on to each of the remaining studs on the door latch handle.
15. Install the aft actuator pin assembly on to the stud that is closest to the bottom of the door. Install the forward actuator pin assembly on to the stud that is closest to the door screen (window).
16. Install the second washer on top of the forward and aft actuator pin assemblies.
17. Thread on the ⅜-inch castle nut until snug. Back the nut off enough to align the castle nut with the hole in the stud. The castle nut is on with the correct torque if the forward or aft actuator pin assemblies can easily move on the stud, but is not overly loose.
18. Install the split pin. The recommendation is to use a new 1/16-inch diameter split pin.
19. Remove the alignment wire installed in steps 5 and 6 in the procedure in subsection Figure 52-7 Door Latch Removal.
20. Apply Loctite #243 to the threads of the screw for the red ball knob. Install the red ball knob to the door latch handle.

NOTE

The recommendation is to protect the surface of the door screen (window) with a paper back tape. The door screen is Acrylic.

21. Install the access cover and secure the access cover with two screws.
22. If the door is not on the airplane, install the door on to the airplane.
23. Check the door for proper operation. If there remains an issue with door operation, contact Liberty Aerospace Customer Service.

Section 01-05 Door Lock Removal and Installation

This section details the procedure to remove and install the door lock mechanism. See Figure 52-7 for an exploded view of the door latch and lock mechanisms.

Some parts shown in Figure 52-7 are for reference only and are not available as an individual field replaceable part. Those parts are part of an assembly with other associated parts. See notes in the associated procedures.

DOOR LOCK REMOVAL

Perform the following procedure to remove the door lock mechanism. The door lock is part of a three-lock set. See Figure 52-9 for an exploded view of the door lock assembly. The following are the tools required: Philips Screwdriver, Flat Blade Screwdriver, 7/8 inch deep socket.

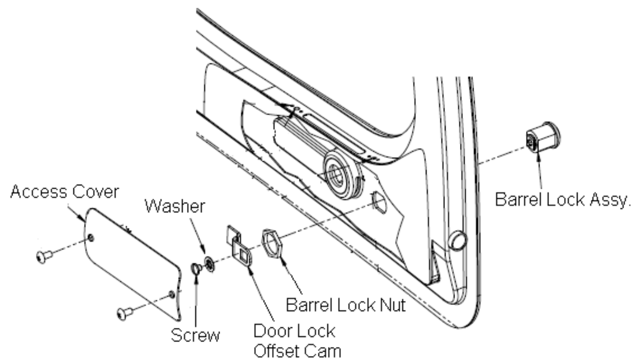

Figure 52-9 Exploded View of Door Lock Assembly(Port Side Shown)

NOTE

Perform this procedure with the door on or off the airplane.

1. If the door is on the airplane, fully open the door.
2. Remove the two Phillips screws holding the access cover.
3. Remove the access cover.
4. Remove the screw and washer holding on the door lock offset cam.
5. Remove the door lock offset cam.
6. Remove the barrel lock nut.
7. Remove the barrel lock assembly.

Door Lock Installation

Perform the following procedure to install the door lock mechanism.

NOTE

Perform this procedure with the door on or off the airplane.

1. Insert the barrel lock assembly through the hole in the door.
2. Apply Loctite #243 to the threads of the barrel lock assembly. Thread on the barrel lock nut. Tighten the nut.
3. Install the door lock offset cam.
4. Apply Loctite #243 to the threads of the screw for the lock cam. Thread on the screw and washer.
5. Install the access cover, securing the cover with the two Phillips screws.

Check the door for proper operation. If there remains an issue with the door lock mechanism, contact Liberty Aerospace Customer Service.

Section 01-06 Door Strap Removal and Installation

This section details the procedure to remove and install the door strap mechanism. See Figure 52-7 for an exploded view of the door latch and lock mechanisms.

Some parts shown in Figure 52-7 are for reference only and are not available as an individual field replaceable part. Those parts are part of an assembly with other associated parts. See notes in the associated procedures.

DOOR STRAP REMOVAL

Perform the following procedure to remove the door strap. The tools required for this procedure are a Phillips screwdriver.

NOTE

Perform this procedure with the door on or off the airplane.

1. If the door is on the airplane, fully open the door.
2. Remove the two screws holding the door strap to the door.
3. Remove the strap.

DOOR STRAP INSTALLATION

Perform the following procedure to install the door strap.

NOTE

Perform this procedure with the door on or off the airplane.

1. Hold the door strap up to the door.
2. Install the two screws.
3. Check the door for proper operation. If there remains an issue with door strap installation, contact Liberty Aerospace Customer Service.

Section 01-07 Gas Spring Cylinder Removal and Installation:

This section details the procedure to remove and install the door latch mechanism. See Figure 52-10 for location of the gas spring. There are no tools required for this procedure. Use a fingernail to remove the locking pins.

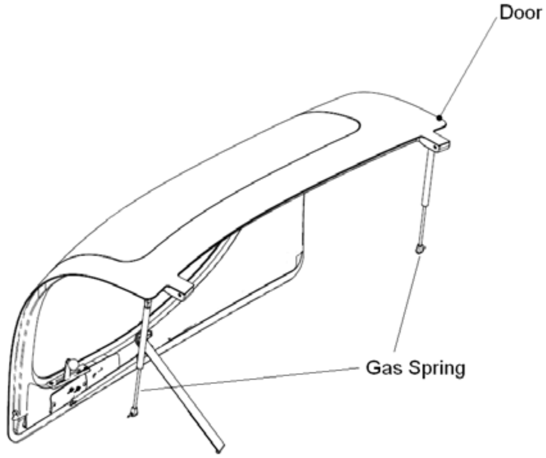

Figure 52-10 Location of the Gas Spring Cylinder (Typical for Both Doors, Starboard Side Shown)

GAS SPRING CYLINDER REMOVAL

Perform the following procedure to remove the gas spring cylinder.

NOTE

Perform this procedure with the door on or off the airplane.

1. If the door is on the airplane, fully open the door.
2. Remove locking pins from the socket mounts of the gas spring.
3. Unsnap the end of the gas spring piston from doorframe (fuselage).
4. Unsnap the end of the gas spring piston from door.

GAS SPRING CYLINDER INSTALLATION

Perform the following procedure to install the gas spring cylinder.

CAUTION

Replace gas spring only with exact duplicate part number obtained from liberty aerospace, inc. Use of "generic" replacement gas springs may cause damage to door or doorframe structure and prevent doors from closing correctly.

NOTE

Install the gas spring with cylinder portion uppermost (attached to door).

NOTE

Perform this procedure with the door on or off the airplane.

1. If the door is on the airplane, support the door in the open position.
2. Snap the end of the gas spring piston onto fitting on doorframe.
3. Snap the end of the gas spring piston onto fitting on door.
4. Install locking rings.

Check the door for proper operation. If there remains an issue, contact Liberty Aerospace Customer Service

Section 01-08 Door Troubleshooting

Use Table 52-1 to troubleshoot issues with the door.

Complaint	Possible Cause	Remedy
Door does not latch properly; door latches difficult to engage	latch(es) or linkage out of adjustment latches need lubrication	adjust Lubricate with Corrosion X
Door latches OK but difficult to unlatch	excessive pressure from door seal material	check door seal, replace if necessary
Excessive wind noise in flight; evidence of water in cabin after rain	defective door seal	replace
Door will not remain in fully open position	defective gas spring	replace
Door difficult to move from open to closed position	dirt on gas spring piston defective gas spring	clean replace

Table 52-1 Door Troubleshooting

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Section 52-02 Emergency Exit

The airplane doors have an acrylic door screen or window. The design of the window is such that it will fracture when stuck with sufficient force. This is a safety feature in the event that crewmembers need to escape the airplane and the doors will not or cannot open properly.

There is a placard affixed to the exterior of the door screen that instructs rescuers to fracture and remove the window. See Figure 52-11 for proper orientation of the placard.

NOTE

For emergency exit, normal door operation maybe used or a safety hammer is provided in the passenger seatback headrest.

Figure 52-11 Placard Instructing Rescuers to fracture and Remove Window in the Event of an Emergency (Placard is properly installed when it appears up side down when the door is in the closed position).

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CHAPTER 53

FUSELAGE

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Table of Contents

SECTION 53-00	GENERAL	5
SECTION 00-01	FUSELAGE INSPECTION AND MAINTENANCE	6
SECTION 53-10	FUSELAGE SECTIONS	9
SECTION 10-01	AFT FIREWALL (CABIN AREA)	9
SECTION 10-02	INSTRUMENT PANEL CONSOLE	10
SECTION 10-03	INSTRUMENT PANEL PROCEDURES	10
INSTRUMENT PANEL CONSOLE REMOVAL		11
INSTRUMENT PANEL CONSOLE ASSEMBLY INSTALLATION		15
SECTION 10-04	AFT CABIN AREA AND FWD BAGGAGE BAY CLOSEOUT	19
SECTION 10-05	BAGGAGE BAY FLOOR ACCESS PANEL	19
SECTION 10-06	BAGGAGE BAY CLOSEOUT	20
SECTION 10-07	AFT BULKHEAD MID FUSELAGE	20
SECTION 53-20	EXTERIOR FUSELAGE ACCESS PANEL	23
SECTION 53-30	REMOVABLE PANELS	25
SECTION 30-01	ENGINE COWLINGS	25
SECTION 30-02	BELLY PANEL ASSEMBLY	25
SECTION 30-03	BELLY PANEL PROCEDURES	26
BELLY PANEL REMOVAL		27
BELLY PANEL INSTALLATION		28
BELLY PANEL INSPECTION		29
SECTION 53-31	FOOTSTEP	31
SECTION 31-01	FOOTSTEP PROCEDURES	31
FOOTSTEP REMOVAL		32
FOOTSTEP INSTALLATION		33
FOOTSTEP INSPECTION		34

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Section 53-00 General

The Liberty XL-2 fuselage is a load bearing monocoque structure composed of structural composite materials (solid and sandwich laminates). The fabrication of the fuselage is from carbon fiber reinforced fabrics used as a facing plies that adhere to core materials to form a structural sandwich. There are different thicknesses of core material used to form the structural sandwich in order to support the distribution of stresses through the structure.

The composite airframe (fuselage) includes the engine compartment firewall, the cabin area, the baggage compartment, aft baggage bay closeout, and aft bulkhead mid fuselage, which supports the empennage. Internal fuselage components (bulkheads, stiffeners, cockpit center console, seatbacks, baggage bay floor, etc.) are bonded in position during manufacture. The composite airframe (fuselage) is secured to the space frame assembly (chassis) by attachment fittings. The space frame assembly is made of welded steel tubing as shown in Figure 53-1. The space frame assembly carries all the weights and receives all the forces from aircraft movement or operation.

NOTE

Removal and replacement of the fuselage to the rolling chassis is not anticipated during routine maintenance. In the event that the fuselage needs to be removed from the rolling chassis contact Liberty Aerospace, Inc .Customer Support for the recommended procedure.

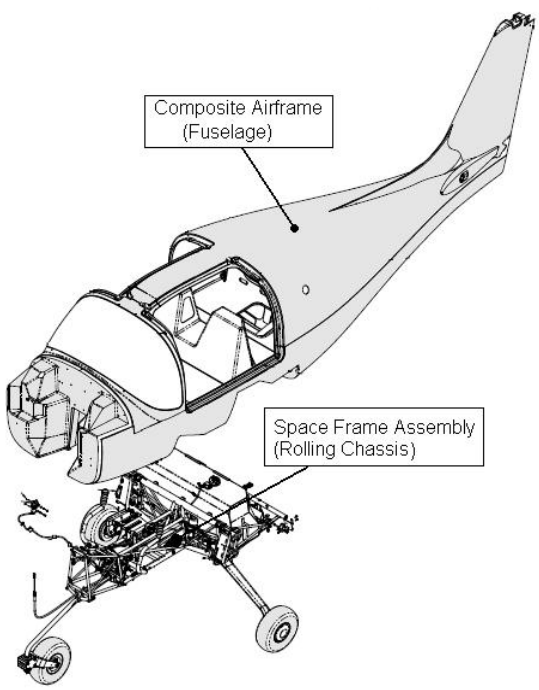


Figure 53-1 Composite Airframe (fuselage) and Space Frame Assembly (rolling chassis)

Section 00-01 Fuselage Inspection And Maintenance

Inspect the composite laminate structure (fuselage and bonded laminates) in accordance with inspection techniques delineated in Chapter 51 – *Standard Practices – Structures* of this service manual. Inspect the composite laminate structures in the Liberty XL-2 aircraft per Chapter 04 – *Airworthiness* at the inspection intervals shown in Chapter 05 - *Time Limits/Maintenance Checks/Inspection Intervals*. The inspections, intervals, and maintenance procedures are mandatory to support the continued airworthiness of the Liberty XL-2.

In addition to the inspections and maintenance practices outlined within Chapter 04 – *Airworthiness* and Chapter 05 - *Time Limits, Maintenance Checks, and Inspection Intervals*, on the composite laminate (fuselage and bonded laminates), the following inspections are recommended when performing other schedule and unscheduled maintenance for regions of high stress.

Figure 53-2 shows the Inspection location that is between the bonded section of upper and lower fuselage just aft of the doorsill. The tap test and/or other Non Destructive Inspection (NDI) test prescribed within chapter 51 – *Standard Practices – Structures*, must be performed from outside on both Port and STBD sides of the fuselage.

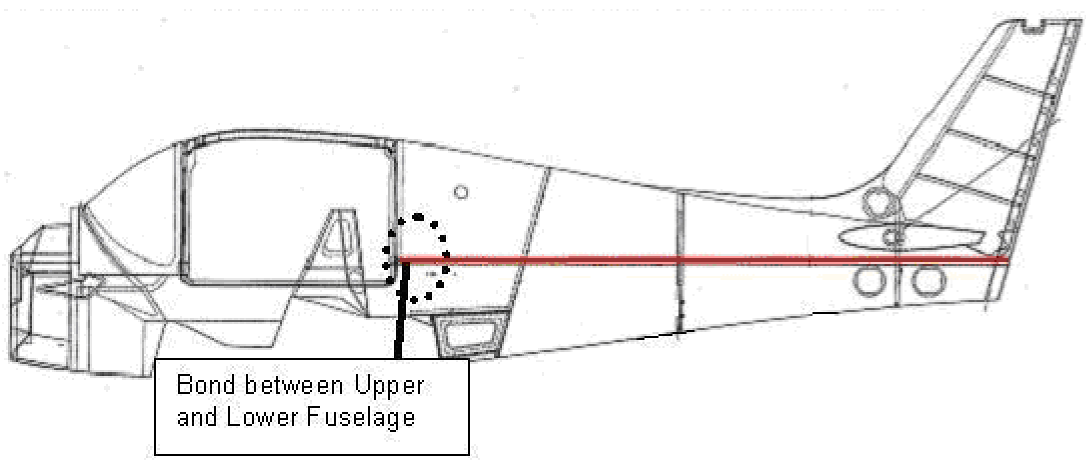


Figure 53-2 Location for Inspection

Figure 53-3 shows the inspection location between the bonded sections of the support for the baggage bay floor and the lower fuselage. The tap test and/or other non-destructive inspection test must be performed from outside on both port and starboard. Access to the supports for the baggage bay floor (port and starboard) and laminate structures is by removing the access panel in the baggage bay floor, refer Section 10-06 - Baggage Bay Closeout on page 20 of this chapter.

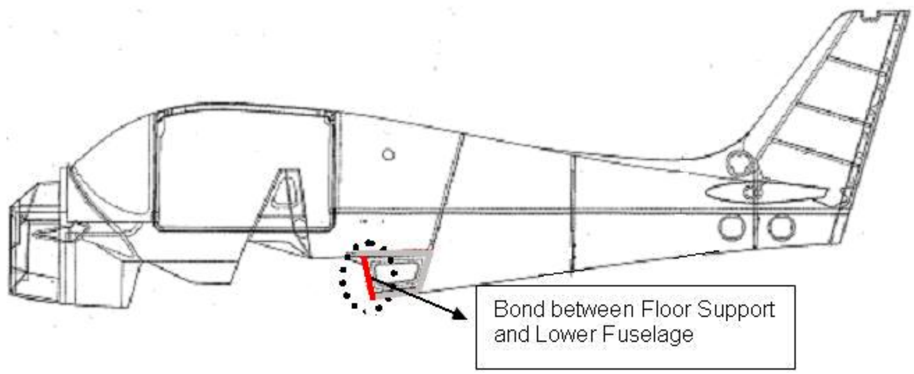


Figure 53-3 Inspection Location Between The Bonded Sections Of The Baggage Bay Floor

Figure 53-4 shows the inspection location between the bonded section of the baggage bay closeout and the upper fuselage. The tap test and/or other non-destructive inspection test prescribed within chapter 51 – *Standard Practices – Structures*, must be performed from outside on both port and starboard.

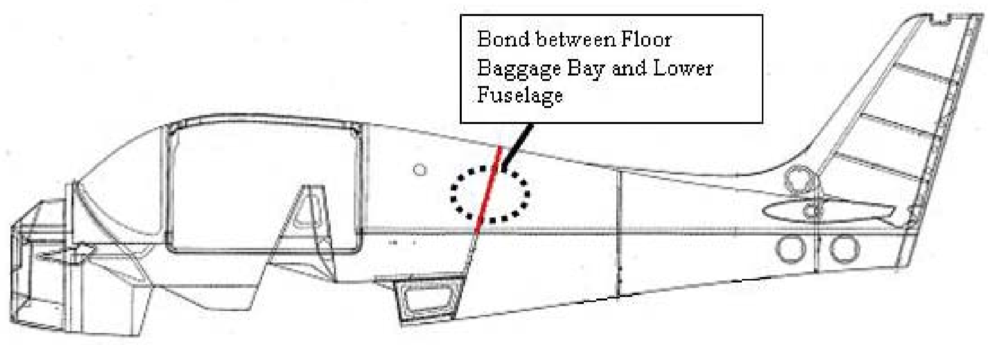


Figure 53-4 Inspection Location Between The Bonded Section Of The Baggage Bay Closeout

Figure 53-5 shows the location is just upward of the flange and at butt-line (BL 9.15, approximately). This is typical on both port and starboard side of the forward bulkhead. The tap test and/or other non-destructive inspection test must be performed to probe any possible delamination (facing plies separation) from outside during inspection. This inspection requires the removal of the upper engine cowl.

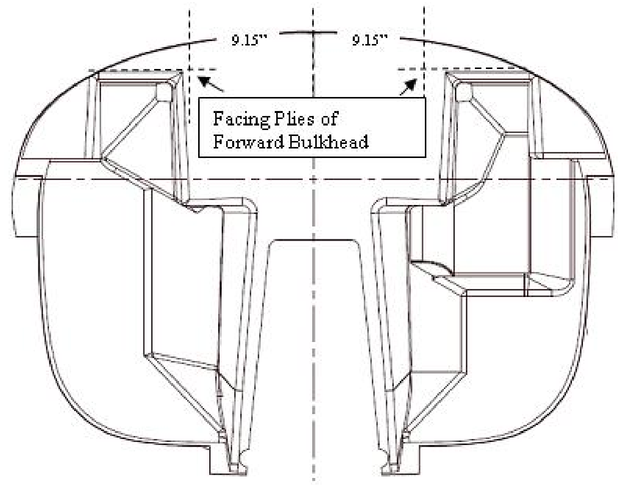


Figure 53-5 Location: Just Upward Of The Flange And At Butt-Line

Figure 53-6 shows the location of the windshield attachment with the fuselage. Perform the tap test and/or other non-destructive inspection test all around the periphery of the attachment of windshield with the fuselage. Inspect the windshield to probe for any damage such as surface cracks (within close proximity of its attachment to the fuselage).

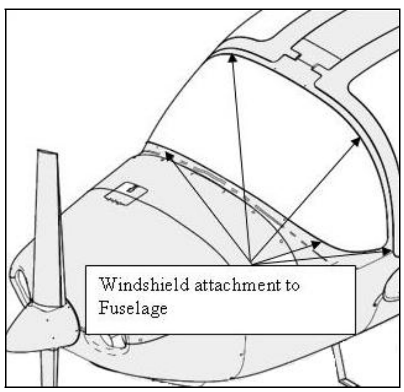


Figure 53-6 Location Of The Windshield Attachment With The Fuselage

Section 53-10 Fuselage Sections

This section deals with the different fuselage sections.

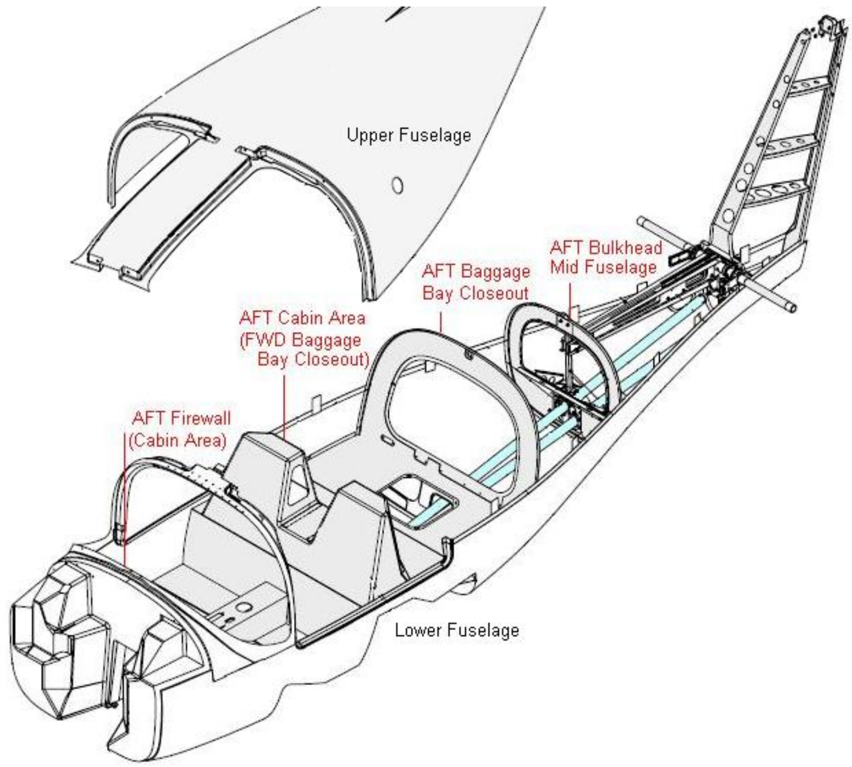


Figure 53-7 Composite Fuselage Sections

Section 10-01 Aft Firewall (Cabin Area)

The cabin area houses all the avionics and flight control panels to operate the Liberty XL-2 aircraft. The seat cushions are installed into the seat back and seat base of the fuselage at both port and starboard side.

The seats in the Liberty XL-2 aircraft are integral to the composite fuselage. There are recesses on the inside faces of both seat backs as shown in Figure 53-7 Figure 53-8 and Figure 53-9. The starboard section of the seat back recess is a storage compartment that carries the airplane flight manual and safety hammer.

Figure 53-8 Seat Back Closeout Access Panels

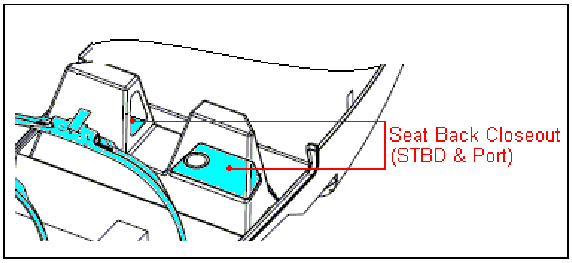


Figure 53-9 Seat Back Closeout Access Panels

Section 10-02 Instrument Panel Console

Located in the cabin area is the instrument panel console. This console houses the instrument panel, avionics panel, and the circuit breaker panel. Behind these panels, the instrument panel console also houses the altitude encoder, the engine data interface, air ventilation for the forward windscreen (windshield), and access to the airplane's main electrical harness and mounting hardware for the rudder pedal assembly.

Section 10-03 Instrument Panel procedures

This section contains the procedures to remove and install the instrument panel console.

INSTRUMENT PANEL CONSOLE REMOVAL

Perform the following procedure to remove the instrument panel console assembly from the airplane.

CAUTION

Before starting this procedure, the tail of the airplane requires support. Failure to support tail of airplane may cause damage to tail section while accessing any area aft of passenger compartment.

CAUTION

Failure to disconnect batteries may cause damage to airplane electrical circuitry.

NOTE

1. Cover forward windscreen internally to protect from scratching. Liberty recommends using Shrink-Wrap plastic.

1. Position aircraft master switch to OFF.
2. Install a tail stand underneath tail section of airplane.
3. Remove cabin aft bulkhead access panel, by removing securing screw hardware.
4. Disconnect negative then positive leads from both primary battery and secondary battery. Isolate terminals on batteries to prevent accidental connection.
5. Remove instrument panel from airplane (see Chapter 31 – Indicators and Recording Systems).
6. Remove circuit breaker (CB) panel from airplane (see Chapter 24 – Electrical Power).
7. Remove avionics panel from airplane (see Chapter 23 – Communications).
8. Start on the starboard side of the instrument panel console. Remove the battery wire from the lower terminal of Terminal 2 on the terminal strip. See Figure 53-10 for the location of the battery wire.
9. Remove the Alternator wire from the lower terminal of Terminal 3 on the terminal strip.

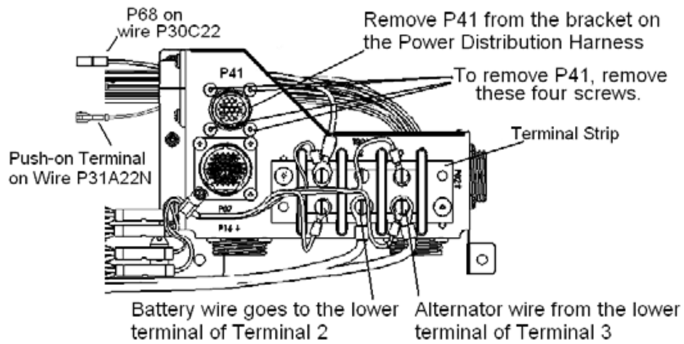


Figure 53-10 Starboard Side of the Power Distribution Harness Showing the Location of the Battery and Alternator Wires and Connector P41

10. Remove the four screws that hold P41 to the bracket on the power distribution harness.
11. Remove any tie-wraps that hold the wires going to the connector P41.
12. Disconnect the single wire connector P68 on wire P30C22.
13. Carefully pull the push-on terminal on wire P31A22N to disconnect the terminal from the ground plate that forms the base of the power distribution harness.
14. Disconnect the Engine Data Interface, EDI, at the connector EDIJ1. See Figure 53-11 for the location of the EDI and its associated connector.

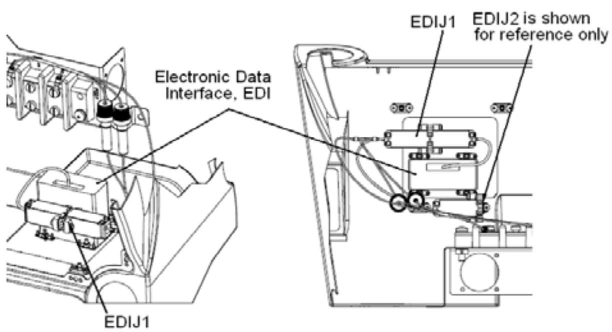


Figure 53-11 Showing the Location of the EDI Connector EDIJ1

15. Move to the pilot’s seat. Fine the in-line fuse connected to relay K003. Disconnect the fuse from the wiring harness. You do not need to disconnect the connector P66. See Figure 53-12 for the location of the fuse associated with K003.
16. Disconnect magnetic compass at the connector P/J06. Secure the cable leading to the compass using a small sealable plastic bag.

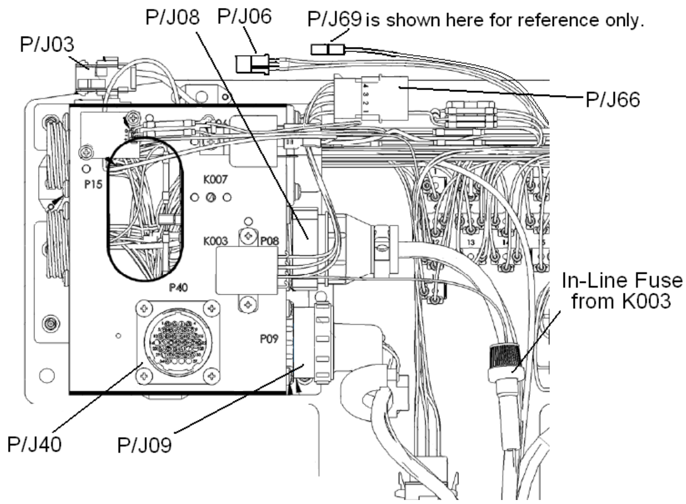


Figure 53-12 Port Side of the Power Distribution Harness

17. Disconnect P08, P09, and P40 from their connectors on the left hand bracket on the power distribution harness. See Figure 53-12 for the location of P/J08, P/J09, and P/J40.

18. Remove the wire P25A10 from the connector P03. The connector shell holds the wire in. Gently pull the connector apart to release the wire.

19. Remove the static line (green tube) that comes up from the space frame and goes to the alternate air valve on the under surface of the instrument panel console assembly. See Figure 53-13 for the location of the alternate air valve and the static line.

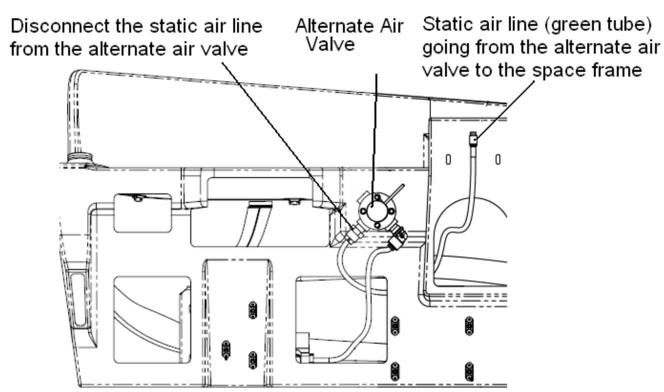


Figure 53-13 Location of the Alternate Air Valve

20. Disconnect the air vent tube from the air vents mounted in the top of the instrument panel console.

21. Bundle all of the cables, tubes and other items together. Insert this bundle into a plastic bag, sealing the bag with tape to protect the items in the bundle from damage and/or contamination.

22. Remove the top engine cowling.

23. From the engine compartment, remove the four screws that secure the instrument panel console assembly. See Figure 53-14 for the location of the four screws.

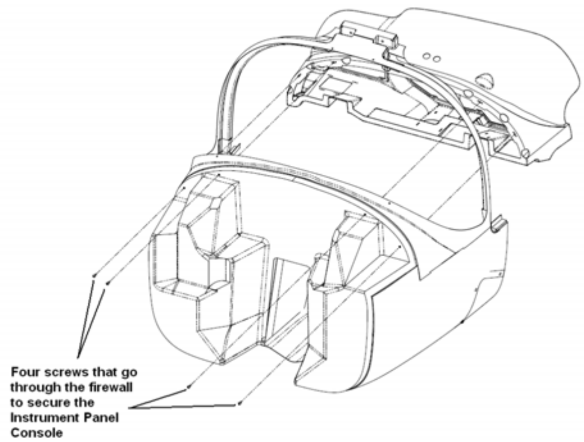


Figure 53-14 Location of the Four Screws the Secure the Instrument Panel Console to the Firewall of the Airplane

24. Remove the two screws that hold the instrument panel console assembly to the center console. See Figure 53-15 for the location of the two screws.

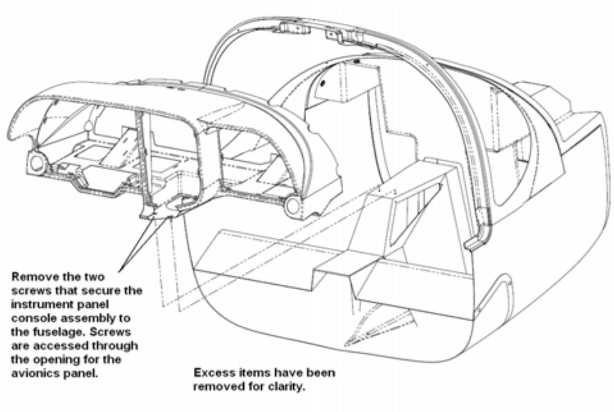


Figure 53-15 Location of the Two Screws that Secure the Instrument Panel Console to the Center Console

25. Carefully remove the instrument panel console assembly from the airplane while threading the bundle of cables and tubes through the opening in the bottom of the instrument panel console assembly.

INSTRUMENT PANEL CONSOLE ASSEMBLY INSTALLATION

Perform the following procedure to install the instrument panel console assembly.

CAUTION

Before starting this procedure, the tail of airplane requires support. Failure to support tail of airplane may cause damage to tail section while accessing any area aft of passenger compartment.

CAUTION

Failure to disconnect batteries may cause damage to airplane electrical circuitry.

1. Install a tail stand underneath tail section of airplane.
2. Disconnect negative then positive leads from both primary battery and secondary battery. Isolate terminals on batteries to prevent accidental connection.
3. Carefully insert the bundle of cables and tubes through the opening in the instrument panel console assembly while installing the instrument panel console assembly from the airplane.
4. Install the two screws that hold the instrument panel console assembly to the center console. See Figure 53-16 for the location of the two screws.

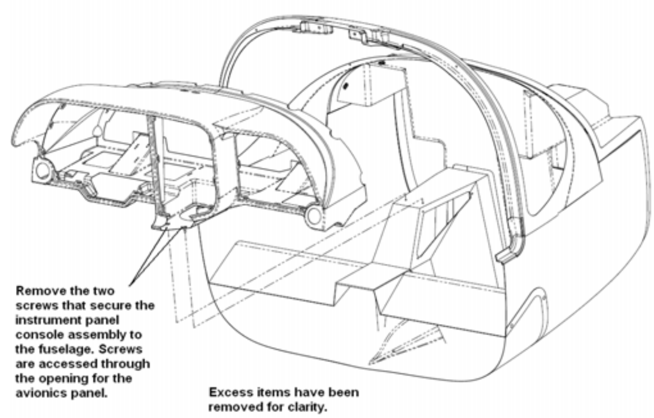


Figure 53-16 Location of the Two Screws that Secure the Instrument Panel Console to the Center Console

5. From the engine compartment, install the four screws that secure the instrument panel console assembly. See Figure 53-17 for the location of the four screws.

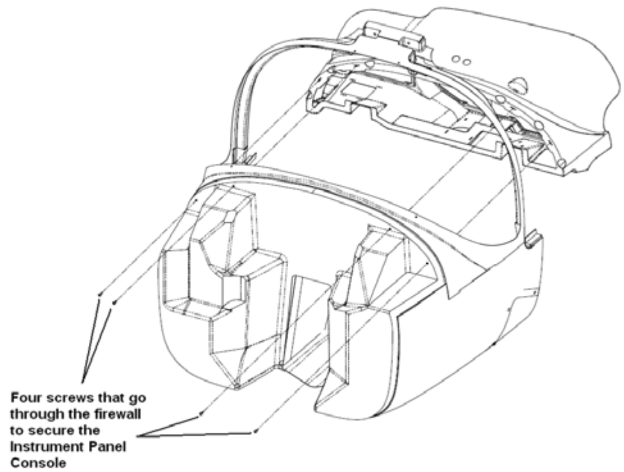


Figure 53-17 Location of the Four Screws the Secure the Instrument Panel Console to the Firewall of the Airplane

6. Install the top engine cowling.
7. Remove the protective covering from the bundle of cables and tubes.
8. Connect the air vent tube to the air vents mounted on the top surface of the instrument panel console.
9. Connect the static line (green tube) that comes up from the space frame and goes to the alternate air valve on the under surface of the instrument panel console assembly. See Figure 53-18 for the location of the alternate air valve and the static line.

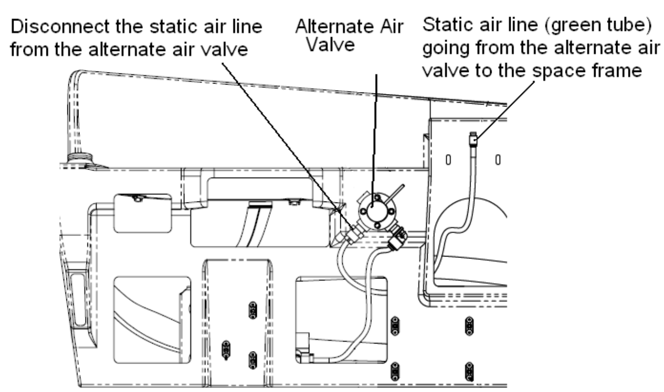


Figure 53-18 Location of the Alternate Air Valve

10. While sitting in the pilot's seat, connect the wire leading to the magnetic compass at the connector P/J06.
11. Insert the wire P25A10 in to the connector for P03. Push the connector shell together to seat and capture the wire in to the connector.

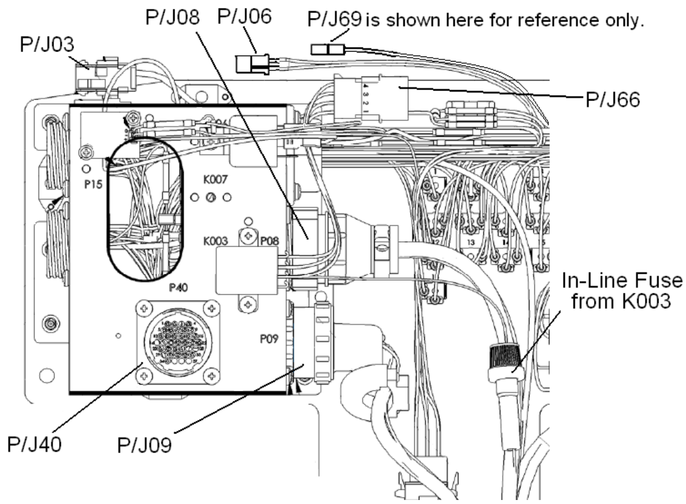


Figure 53-19 Port Side of the Power Distribution Harness

12. Connect P08, P09, and P40 from their connectors on the left hand bracket on the power distribution harness. See Figure 53-19 for the location of P/J08, P/J09, and P/J40.

13. Find the in-line fuse connected to relay K003. Connect the fuse from the wiring harness. See Figure 53-19 for the location of the fuse associated with K003.

14. Move to the passenger's seat. Connect the Engine Data Interface, EDI, at the connector EDIJ1. See Figure 53-20 for the location of the EDI and its associated connector.

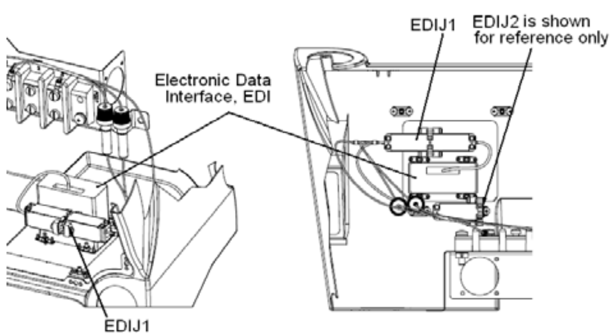


Figure 53-20 Showing the Location of the EDI Connector EDIJ1

15. Carefully push the push-on terminal on wire P31A22N on to a terminal on the ground plate that forms the base of the power distribution harness.

16. Connect the single wire connector P68 on wire P30C22.

17. Install P41 into the bracket on the power distribution harness. Secure P41 with the four screws removed in step 10 of the procedure to Instrument Panel Console .

18. Install tie-wraps as required to hold the wires going to the connector P41.

19. Connect the battery wire, P33A2, to the lower terminal of Terminal 2 on the terminal strip. See Figure 53-21 for the location of the battery wire.

20. Connect the Alternator wire, P10A6, to the lower terminal of Terminal 3 on the terminal strip.

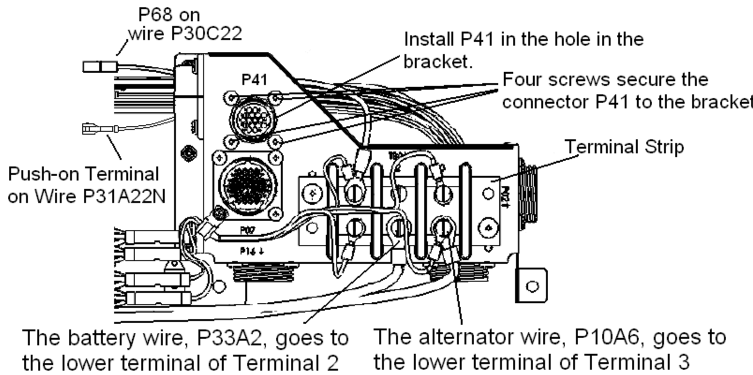


Figure 53-21 Starboard Side of the Power Distribution Harness Showing the Location of the Battery and Alternator Wires and Connector P41

21. Install avionics panel in accordance with Chapter 23 – *Communications*.
22. Install the circuit breaker panel in accordance with Chapter 24 – *Electrical Power*.
23. Install instrument panel in accordance with Chapter 31 – *Indicators and Recording Systems*.
24. Connect negative then positive leads to both primary battery and secondary battery.
25. Install cabin aft bulkhead access panel using securing screw hardware.

Section 10-04 Aft Cabin Area And Fwd Baggage Bay Closeout

This fuselage section gives access to the baggage bay compartment and fuel filler hose. The Baggage Bay Floor Access Panel is located in the floor under the carpet. Be careful, do not damage interior fabric and/or carpet (if applicable) when accessing floor panel.

NOTE

To view the forward Horizontal Stabilizer Pushrod and/or Rudder Push-Rod Intermediate remove carpet, remove access panel by removing hardware. Retain hardware for access panel install.

This area is designated for storage of items limited by weight. This area houses two (2) access panels as shown in Figure 53-22:

- 1. Baggage Bay Floor Access Panel
- 2. Baggage Bay Closeout

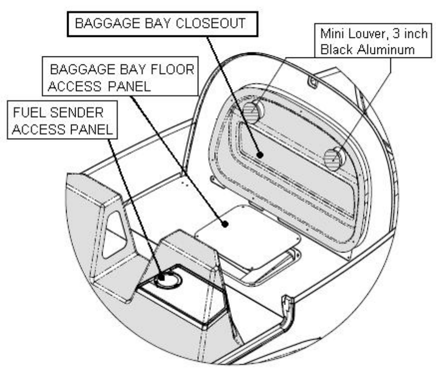


Figure 53-22 Baggage Bay Closeout

Section 10-05 Baggage Bay Floor Access Panel

The access panel in the floor of the baggage bay compartment allows access to the view: floor supports, electrical cabling, stabilizer, rudder pushrod, bell-cranks, and their bearings as shown in Figure 53-23.

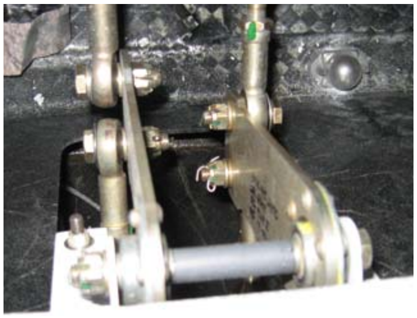


Figure 53-23 FWD stabilizer and rudder pushrod and their bearings

Section 10-06 Baggage Bay Closeout

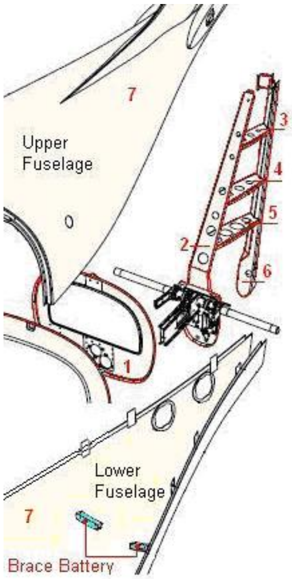
Access panels are provided as internal structural elements to facilitate maintenance. The Baggage Bay Closeout is a removable component to allow access to the interior of the AFT fuselage and components. To access the interiors of the aft fuselage, temporarily remove the baggage bay closeout. To remove the baggage bay closeout remove and retain the mounting hardware. See Figure 53-22

Section 10-07 Aft Bulkhead Mid Fuselage

The access panel in the aft mid fuselage bulkhead, allows access to the aft most section of the fuselage and to access other installed items such as the primary battery, secondary batteries, emergency locator transmitter (ELT), electrical system components, tail plane drive torque assembly, stabilizer, and rudder pushrod.

Figure 53-24 AFT Bulkhead Mid Fuselage

The fuselage aft most tail section houses fin spar, ribs which is bonded to the fuselage structure. Figure 53-25 below shows items installed at the tail of the fuselage. The fuselage tail structure is a permanent adhesive composite laminate structure consisting of fin spar, ribs and upper & lower fuselage. Noted below are the tail structure bondlines.



No.	Component
1	AFT Bulkhead Mid Fuselage
2	Fin Spar
3	Rib 1
4	Rib 2
5	Rib 3
6	Fin Closeout-Vertical Closeout
7	Upper and Lower Fuselage

Figure 53-25 Tail of Fuselage

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Section 53-20 Exterior Fuselage Access Panel

An access panel on the starboard side of the upper aft fuselage allows access to the bearings for the stabilator torque tube and mass balance. See Figure 53-26 for the location of the access panel.

Two access panels on the starboard side of the lower aft fuselage. These panels allow maintenance of the pitch trim servo, bell-cranks, and pushrods of the elevator and rudder control systems. See Figure 53-26 for the location of these panels.

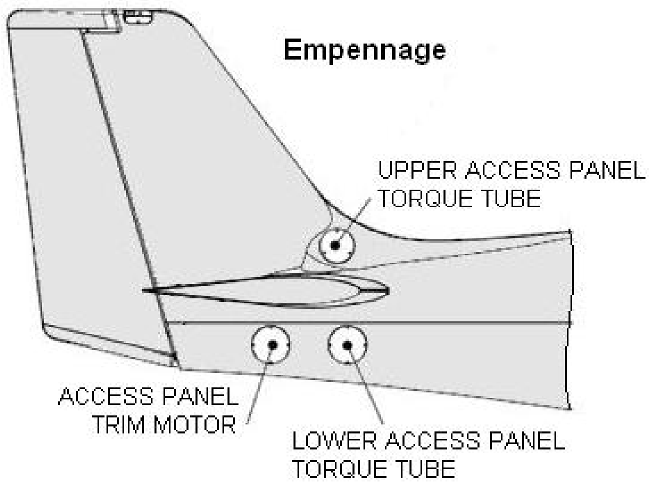

Figure 53-26 Starboard Access Panels

The fuel tank filler access or gas cap is on the Port side upper fuselage. See Figure 53-27. The gas cap has a locking lever that locks the cap in place for flight. To open the gas cap, pull the lever our and give it a quarter to the left, and pull the cap away from the airplane. To install the gas cap, put the cap in to the filler opening, press in on the cap while turn the lever a quarter turn to the right.

Figure 53-27 Fuel Tank Filler Access or Gas Cap

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Section 53-30 Removable Panels

This section details information about removable panels that attach to the fuselage.

Section 30-01 Engine Cowlings

Although the engine cowlings on the XL-2 airplane attach to the fuselage and are an integral part of the fuselage, they are part of the engine. Therefore, information concerning the engine cowlings is in Chapter 71 - *Power Plant*.

Section 30-02 Belly Panel Assembly

The Belly Panel or Fairing Assembly is an access panel, which is a cover for the space frame assembly. The belly panel is also an aerodynamic structure for increase flying efficiency. The belly panel allows maintenance and inspection of other installed units/components.

NOTE

The term fairing and panel are used interchangeably and have the same meaning.

The belly panel attaches to the fuselage under the space frame assembly (rolling chassis) by means of a series of CAMLOC® fasteners. The belly panel has three (3) access holes for the gascolator, fuel boost pump assembly and fuel tank assembly (see Figure 53-28). Two (2) vapor zone hoods are integral design features of the current belly fairing assembly. Two cutouts on port and starboard side of the belly fairing are for the installation of under carriage cover for main gear legs as shown in Figure 53-29.

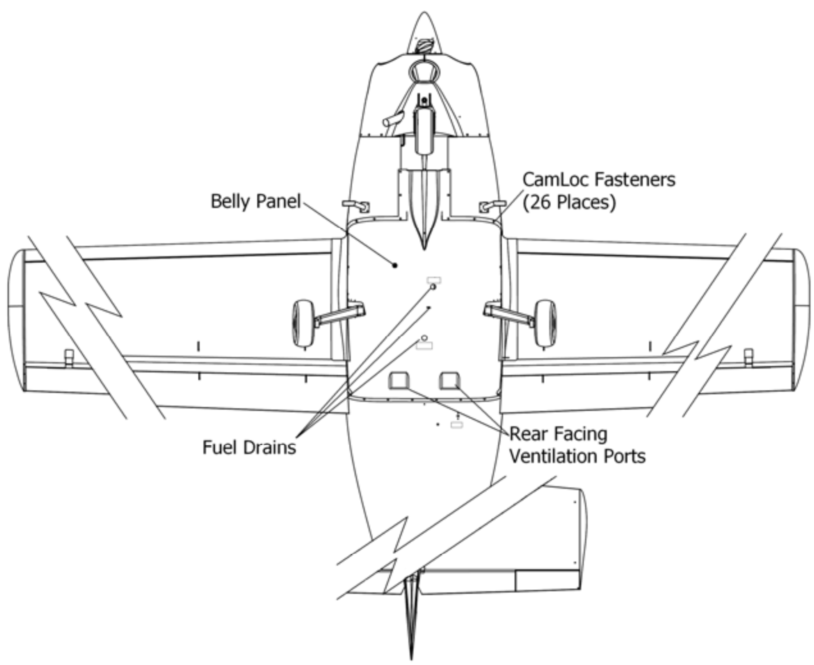

Figure 53-28 Underside of Airplane Showing the Belly Panel

Two added cutout are match drilled to the middle-sides port and starboard main leg AFT bolts in the belly. Remove the belly panel to gain access to the main and nose gear jack points.

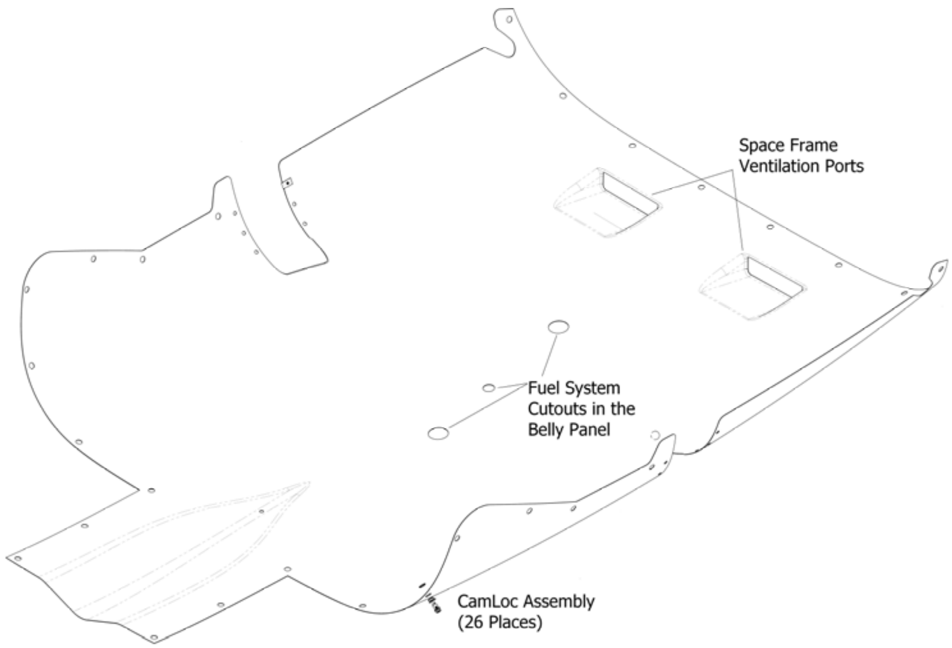

Figure 53-29 Belly Panel

Section 30-03 Belly Panel Procedures

This section contains the procedures to remove and install the belly panel.

BELLY PANEL REMOVAL

Perform this procedure to remove the belly panel. A small padded stand capable of supporting the weight of the belly panel is the recommended equipment. A stand with adjustable height is also a desirable utility.

CAUTION

No person, their hands or extremities should be near the wing flaps during operation, due to the extreme power of closing/opening the wing flap units.

1. First, extend (deploy) the wing flaps to gain access to the belly panel fasteners.
2. Remove and retain hardware.
3. Remove the Cover Plate Upper under carriage leg (both port and starboard).
4. Place a padded support under the approximate center of the belly panel.
5. Except for the four (4) fasteners at the four corners of the fuselage belly panel, disengage all CAMLOC® fasteners.
6. While ensuring that the belly panel remains supported, unfasten the remaining four cam loc fasteners.
7. Remove belly panel.

This completes the Belly Panel Removal procedure.

BELLY PANEL INSTALLATION

Perform this procedure to install the belly panel

CAUTION

Failure to install the belly panel properly can result in the buckling of the belly panel, which will damage the panel and may cause it to separate from the airplane during flight.

When installing belly panel assembly, it is important to place the belly panel properly, and secure it loosely with the four (4) cam loc fasteners at the corners, then loosely insert the remaining CAMLOC® fasteners. Begin tightening each fastener, insuring that the belly panel is seating properly and flush with the fuselage face.

1. Place a padded support under the approximate center of the fuselage opening.
2. Place belly panel on the support. (Be careful when reassembling: do not inadvertently open the fuel line cock).
3. Align the belly panel with all the opening.
4. Insert and fasten the four (4) CAMLOC® fasteners at corners of the fuselage belly panel.
5. Fasten the remaining fasteners.
6. Retract the flaps.

This completes the Belly Panel Installation procedure.

BELLY PANEL INSPECTION

Perform this procedure to inspect the belly panel after installation.

1. Inspect each of the 26 CamLoc fasteners. All of the fasteners must be flush with the surface and tight.
2. Inspect along the interface between the fuselage and the belly panel. Look for any buckling or separation between the surfaces.
3. Check the belly panel is being held tight to the surface of the fuselage.
4. Check the fuel system drains. They should be coming through the belly panel.
5. Check the ventilation ports. They should be clear of any foreign matter.

This completes the Belly Panel Inspection procedure.

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Section 53-31 Footstep

An optional accessory installs a footstep on the port and starboard sides of the fuselage just ahead of the wings of the airplane. If the footstep is installed, it installs to a hard-point that is glued to the interior surface of the fuselage. Figure 53-30 shows a typical installation of the footstep.

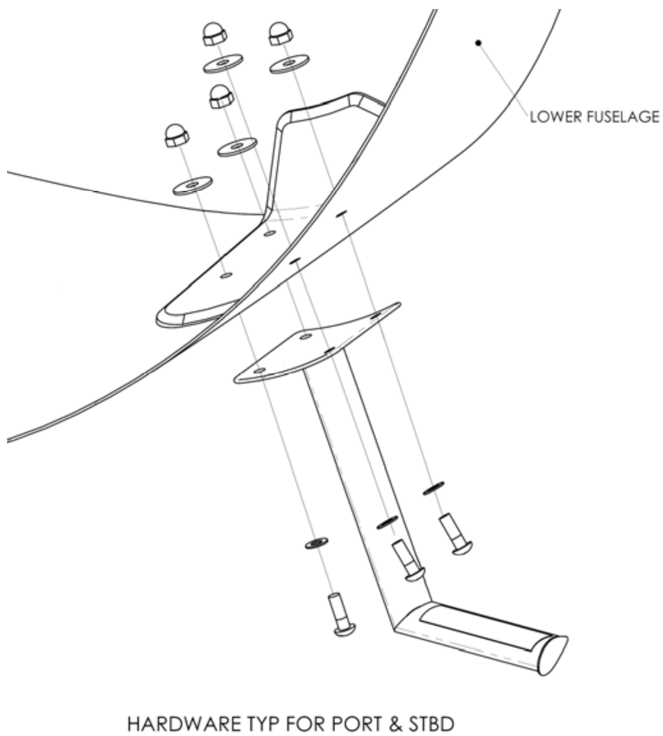

Figure 53-30 Footstep Installation

Section 31-01 Footstep Procedures

This section contains the procedures to remove, install, and inspect the footstep.

FOOTSTEP REMOVAL

Perform this procedure to remove the footstep from the fuselage.

1. Remove four screws securing the footstep to the fuselage.
2. Remove the footstep.

CAUTION

Do not use any type of pry to remove the footstep. It may damage the composite in the fuselage.

3. Use a small die grinder to remove any remaining Epi-Bond adhesive left behind on the fuselage.
4. Do not grind in to the finish of the fuselage.
5. If installing a footstep later, insert the screws back into the holes and tighten enough to hold. Then cover the area with VAC-PAK® A6200, ETFE/Fluoro-polymer Release Film to keep any contaminants from entering the holes in the fuselage.

This completes the Footstep Removal procedure.

FOOTSTEP INSTALLATION

Perform this procedure to install the footstep on to the fuselage.

1. If installing the footstep immediately after removal, go to step 3 below.
2. Remove the footstep as instructed in the Footstep Removal procedure; omitting the final step. Then go to step 4 below.
3. If the four footstep screws are mounted in the fuselage, remove the four screws.
4. Use 120-220 grit sandpaper or Scotch-BriteTM 7447 (maroon) Hand Pads to scuff the inner surface of the footstep plate.
5. Apply a small amount of release film around the area of the footstep installation.
6. Mix a small batch of Epibond 1590 adhesive.

NOTE

Epibond 1590 for this purpose is used as a gap-filler and not as an adhesive between fuselage and footstep.

7. Apply the Epibond to the inner surface of the footstep plate.
8. Install the footstep on the fuselage.
9. Insert the four screws and tighten sufficiently to secure the footstep to the fuselage.
10. Let the Epibond cure for a minimum of 6 hours, before using the foot control.
11. Remove the release film and clean any excess Epibond adhesive.

This completes the Footstep Installation procedure.

FOOTSTEP INSPECTION

Perform this procedure to inspect the footstep installation.

1. Check each of the screws to make sure they are not loose.
2. Check the footstep weldment for any cracks.
3. Check the footstep anti-skid tape that it is not peeling.
4. Check the acorn nuts (cap nuts) on the inside of the fuselage to make sure the nuts present and not loose.
5. Check the composite hardpoint for any cracks or other signs of stress (such as delamination).

This completes the Footstep Inspection procedure.

CHAPTER 55

STABILIZERS

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Table of Contents

SECTION 55-00 GENERAL 5

SECTION 55-10 HORIZONTAL STABILIZERS 7

SECTION 10-01 STABILATOR PROCEDURES 7
STABILATOR ATTACHMENT LUG REMOVAL 8
STABILATOR ATTACHMENT LUG INSTALLATION 9
STABILATOR DRIVE PLATE REMOVAL 10
STABILATOR DRIVE PLATE INSTALLATION 11
STABILATOR MASS BALANCE EXTENSION ARM REMOVAL 12
STABILATOR MASS BALANCE EXTENSION ARM INSTALLATION 15
STABILATOR DRIVE TORQUE TUBE REMOVAL 17
STABILATOR DRIVE TORQUE ASSEMBLY INSTALLATION 20
TORQUE TUBE COATING 23
STABILATOR BEARING REMOVAL 24
STABILATOR BEARING INSTALLATION 25
SECTION 10-02 STABILATOR INSPECTION 27

SECTION 55-30 VERTICAL STABILIZER AND RUDDER 29

SECTION 30-01 VERTICAL STABILIZER AND RUDDER INSPECTION 29

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Section 55-00 General

The stabilizers of the airplane are of conventional aluminum construction and carbon composite. The all-flying horizontal stabilizers (stabilators) provide pitch control. A rudder, hinged to the rear spar of the vertical stabilizer (fin), provides yaw control.

This chapter describes the construction and attachment methods of the stabilizers. Chapter 27 – *Flight Controls*, provides most instructions for removal and installation of stabilizer components. Detailed instructions for removal, installation, and/or inspection of particular focused components are given in this chapter, including:

- Stabilator attachment lug
- Stabilator drive plate
- Stabilator mass balance extension arm
- Stabilator drive torque assembly
- Vertical stabilizer

NOTE

While performing the procedures in this chapter, the following are applicable:

Apply LPS 3 inhibitor (or equivalent) conforming to MIL-PRF-16173 grade 2 to all bearing surfaces, prior to installation.

Treat unpainted surfaces using MIL-C-81309E Type 2 (Corrosion-X Aviation).

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Section 55-10 Horizontal Stabilizers

The tailplane uses all-flying stabilators rather than the more traditional arrangement of a fixed horizontal stabilizer and hinged elevator. The stabilators are of conventional aluminum construction incorporating forward and rear spars, ribs, and skins. Anti-servo/trim tabs are hinged to the top rear of the stabilators.

Fittings attached to the stabilator root rib transmit pitch torque loads to the structure. A linkage attached to the root rib connects the anti-servo/trim tab to the fuselage-mounted trim servo, which provides its deflection.

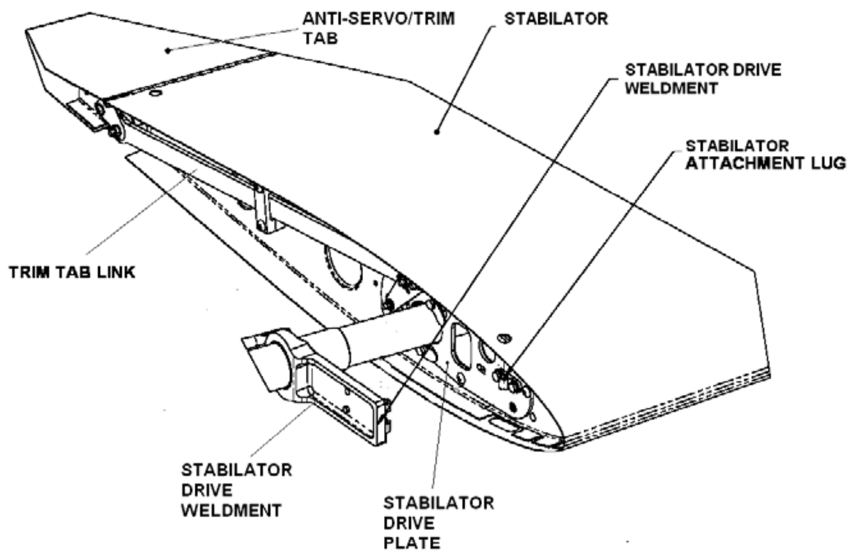

Figure 55-1 Stabilizer Drive Components

Routine horizontal stabilator removal and installation procedures are provided in Chapter 27 – *Flight Controls*. The focus of this chapter is on those items that interface between the pushrods and bell-cranks in the fuselage and the horizontal stabilizers themselves.

Section 10-01 Stabilator Procedures

This section contains the procedures to remove, install and/or inspect those components that connect the horizontal stabilators to the pushrods and bell-cranks in the fuselage. These procedure include the following:

- Stabilator Attachment Lug
- Stabilator Drive Plate
- Stabilator Mass Balance Extension Arm
- Stabilator Drive Torque Assembly
- Stabilator Bearings
- Stabilator Torque Tube Coating

STABILATOR ATTACHMENT LUG REMOVAL

Perform this procedure to remove the stabilator attachment lug.

1. Remove stabilator in accordance with Chapter 27 – *Flight Controls*.

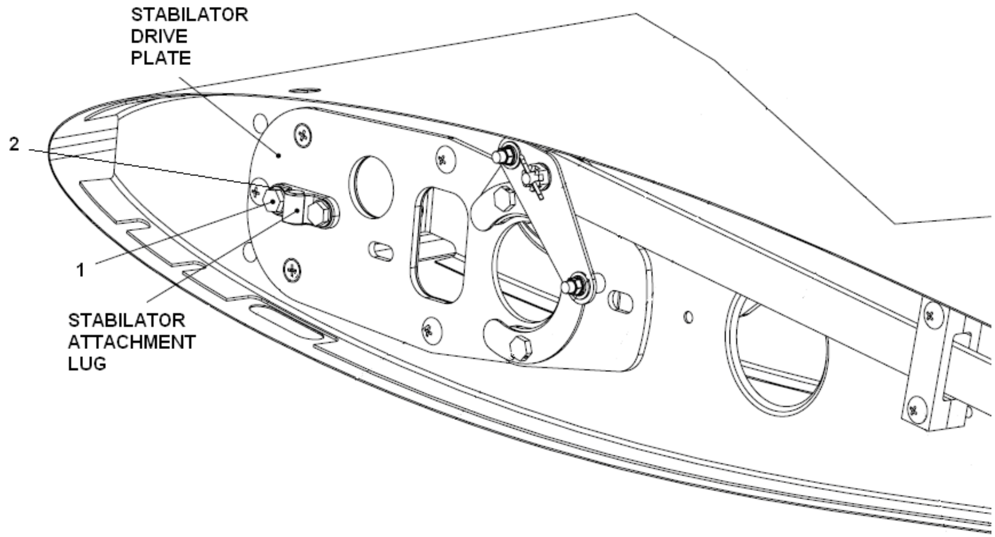

Figure 55-2 Stabilator (Tailplane) Attachment Lug

2. Remove two bolts (1) and washers (2) from stabilator attachment lug and remove attachment lug.
3. Inspect the hardware for proper size and type. The hardware securing the lug must be AN3-5A, #10-32X21/32 bolts with an NAS1149C0332R No. 10 corrosion resistant washer. If the hardware is not this type or size, discard the bolts.

This completes the Stabilator Attachment Lug Removal procedure.

STABILATOR ATTACHMENT LUG INSTALLATION

Perform this procedure to install the stabilator attachment lug to the stabilator.

1. Inspect the hardware that attaches the attachment lug to the stabilator. The hardware securing the lug must be AN3-5A, #10-32X21/32 bolts with an NAS1149C0332R No. 10 corrosion resistant washer. If the hardware is not this type and size, do not use them. Acquire the correct size and type bolts.
2. Install stabilator attachment lug using 2 bolts (1) and washers (2). See Figure 55-2.
3. Torque the bolts to a torque of 32.5 in-lbs, and in accordance with Chapter 20 – *Standard Practices*.
4. Install stabilator on to airplane a shown in Chapter 27 – *Control Surfaces*.
5. This completes the Stabilator Attachment Lug Installation procedure.

STABILATOR DRIVE PLATE REMOVAL

Perform this procedure to remove the Stabilator Drive Plate.

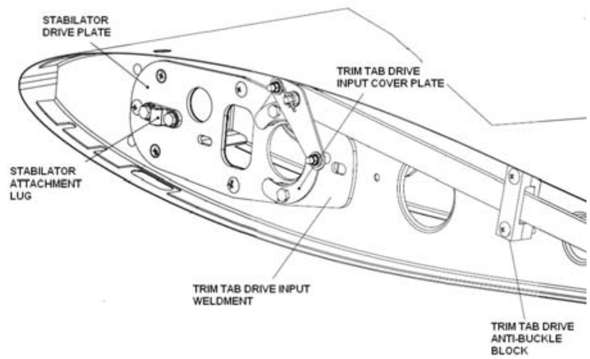

Figure 55-3 Stabilator Drive Plate

1. Remove stabilator in accordance with Chapter 27 – *Flight Controls*.
2. Remove the stabilator attachment lug in accordance with the Stabilator Attachment Lug Removal procedure on page 8 of this chapter.
3. Remove two screws (4) from trim tab drive anti-buckle block. Remove anti-buckle block (top portion).

NOTE

Note the routing of safety wire.

4. Remove safety wire from three bolts (3) securing trim tab drive input cover plate and trim tab drive input weldment to stabilator drive plate.

CAUTION

Each bolt securing trim tab drive input weldment to stabilator drive plate retains an idler bushing located beneath weldment. Use extreme caution to save idler bushings for reuse when removing bolts and trim tab drive input weldment form stabilator drive plate.

Use caution so as to not distort trim tab drive link while separating trim tab drive input weldment from stabilator drive plate.

5. Remove three bolts (3) and separate cover plate and tab drive input weldment from stabilator drive plate, retaining idler bushings.
6. Remove two counter-sunk screws (1) that attach drive plate to stabilator root rib.
7. Remove three screws (2) and remove stabilator drive plate from stabilator root rib.

STABILATOR DRIVE PLATE INSTALLATION

Perform this procedure to install the drive plate on the stabilator.

NOTE

This is a wet assemble with jointing compound to LIB PS 135A-926-004, Section 3.2 Type III.

1. Install stabilator drive plate to stabilator root by installing three screws (2).
2. Install two counter-sunk screws (1) that attach drive plate to stabilator root.
3. Install trim tab drive input weldment and trim tab drive input cover plate using three bolts (3) and idler bushings. Place idler bushings in the locations noted during removal. Tighten bolts for snug fit, ensuring no binding of trim drive.

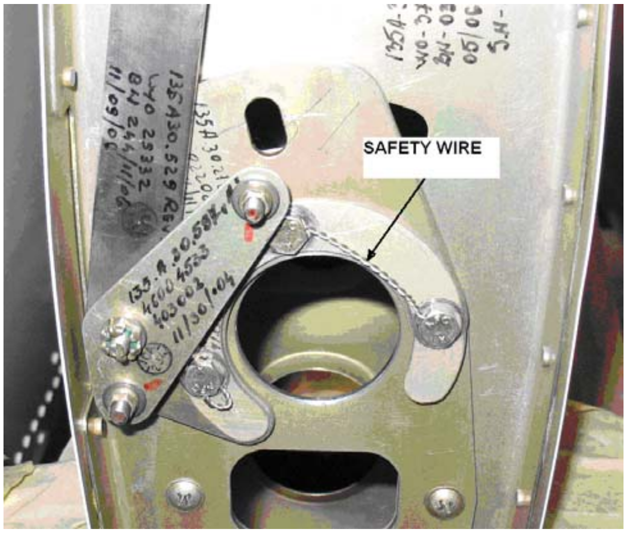

Figure 55-4 Safety Wire Placement

4. Check the safety wire that it is routed so as not to interfere with operation of trim tab operation or interfere with stabilator torque drive assembly hole in root rib.
5. Refer to Figure 55-4. Install safety wire into three bolts (3) that secure trim tab drive input cover plate; check the wire that it does not restrict movement of trim tab operation or interfere with stabilator torque drive assembly hole in root rib.
6. Install trim tab drive anti-buckle block (top) using two screws (4).
7. Check for smooth operation of trim tab, with no binding.
8. Install stabilator attachment lug in accordance with the Stabilator Attachment Lug Installation procedure on page 9 of this chapter.
9. Install stabilator in accordance with Chapter 27 – *Flight Controls*.

STABILATOR MASS BALANCE EXTENSION ARM REMOVAL

Perform this procedure to remove the stabilator mass balance extension arm.

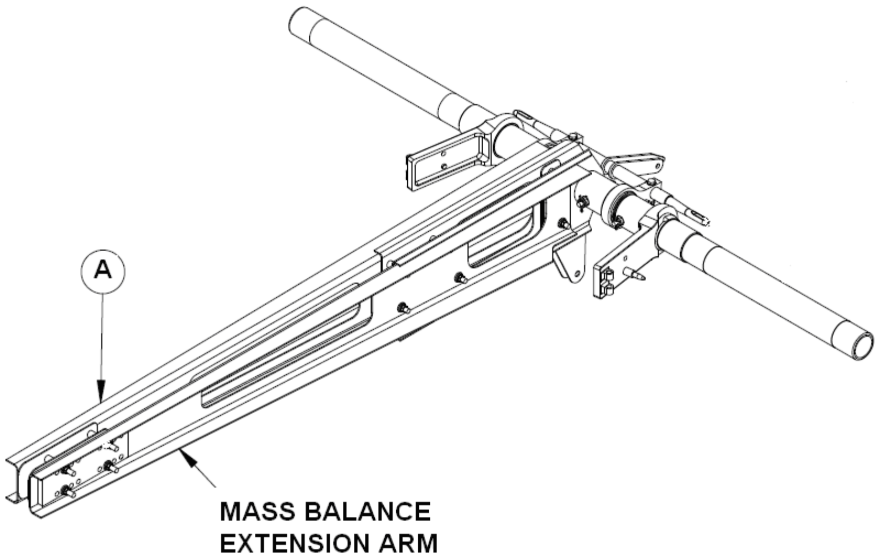

Figure 55-5 Stabilator Mass Balance Arm

CAUTION

Before starting this procedure, tail of airplane requires support. Failure to support tail of airplane may cause damage to tail section while accessing any area aft of passenger compartment.

1. Support tail of airplane.
2. Remove baggage bay closeout panel.
3. Cover batteries with non-conductive material.

NOTE

Some airplanes are outfitted with short spacers and adjustment washers. Note position of all components being removed. Note bolt holes used to mount Tailplane balance mass plates.

4. Refer to Figure 55-6. From inside airplane, remove mass balance plates from mass balance extension arm as follows:

- Nuts (7) – 4 ea
- Washers (6) – 4 ea
- Tailplane Balance Mass (3) – 2 ea
- Mass Balance Bearing Surface (5) – 2 ea

- Spacers (4) – 4 ea (and washers*, if required)
- Washers (2) – 4 ea
- Bolts (1) – 4 ea

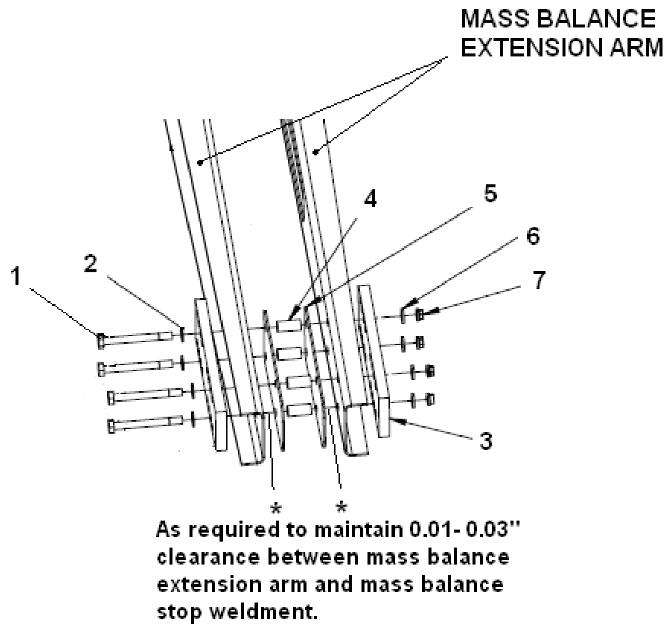

Figure 55-6 Mass Balance Weights

5. Refer to Figure 55-7. From inside aft portion of airplane, remove nut (1), washer (2), and bolt (3) that retain top leg of mass balance stop weldment.
6. Remove nut (4), washer (5), and bolt (6) from lower leg of mass balance stop weldment.

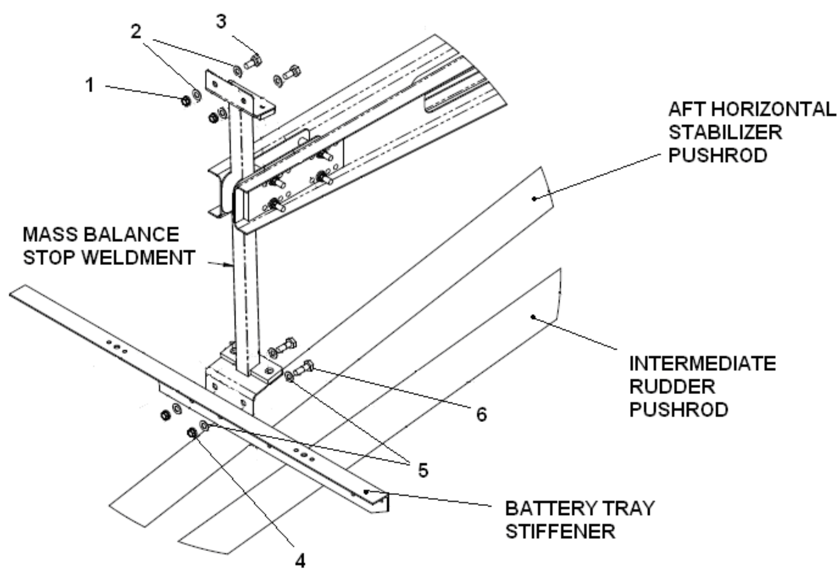

Figure 55-7 Mass Balance Stop Weldment

7. Maneuver weldment out of position and forward out of airplane.

CAUTION

With stabilators installed, removal of stop weldment will allow mass balance extension arms to elevate and damage the fuselage. Take care not to allow extension arms to damage fuselage.

With stabilators removed, removal of stop weldment will allow mass balance extension arms can drop and damage underlying pushrods. Take care not to allow extension to damage underlying pushrods.

8. Refer to Figure 55-8 and Figure 55-9. Remove the following from mass balance extension arms and remove extension arms from airplane:

- Locking Nuts (8) – 6 ea
- Washers (7) – 6 ea
- Spacers (6) – 4 ea
- Spacers (5) – 2 ea
- Washers (4) – 6 ea
- Bolts (3) – 6 ea

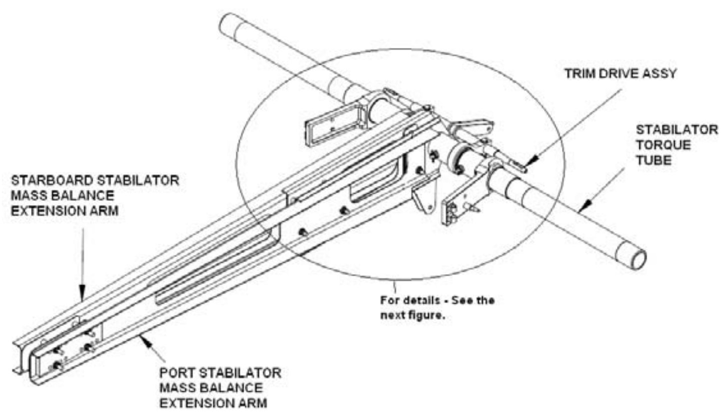

Figure 55-8 Stabilator Mass Balance Extension Arm

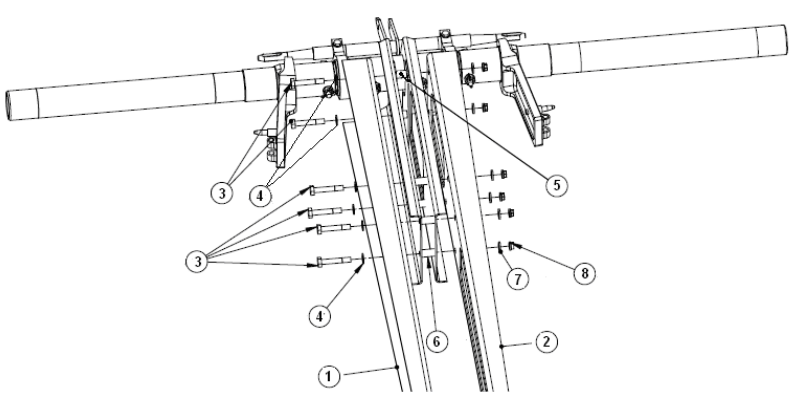

Figure 55-9 Stabilator Mass Balance Extension Arm Detail

STABILATOR MASS BALANCE EXTENSION ARM INSTALLATION

Perform this procedure to install the stabilator mass balance extension arm.

CAUTION

Before starting this procedure, tail of airplane requires support. Failure to support tail of airplane may cause damage to tail section while accessing any area aft of passenger compartment.

With stop weldment removed, mass balance extension arms can drop and damage underlying pushrods. Take care not to allow extension arms to drop.

1. Support tail of airplane.
2. Remove baggage bay closeout panel.
3. Cover batteries with non-conductive material.

NOTE

While installing mass balance extension arms, check that arms are parallel and in line with mass balance weldment arms.

4. Refer to Figure 55-8 and Figure 55-9. From inside the airplane, install mass balance extension arms using the following:

- Bolts (3) – 6 ea
- Washers (4) – 6 ea
- Spacers (5) – 2 ea
- Spacers (6) – 4 ea
- Washers (7) – 6 ea
- Locking Nuts (8) – 6 ea

5. Refer to Figure 55-7. Install lower leg of mass balance stop weldment using bolt (6), washer (5) and nut (4).
6. Install top leg of mass balance stop weldment using bolt (3), washer (2), and nut (1).
7. Torque all locking nuts in accordance with Chapter 20 – *Standard Practices*.

NOTE

Some airplanes are outfitted with adjustment washers. Position all components as noted during removal. Mount tailplane balance mass plates using bolt holes noted during removal.

8. Refer to Figure 55-8 and Figure 55-9. From inside airplane, install mass balance plates to mass balance extension arm as follows:

- Bolts (1) – 4 ea
- Washers (2) – 4 ea
- Spacers (4) – 4 ea (and washers*, if required)
- Mass Balance Bearing Surface (5) – 2 ea
- Tailplane Balance Mass (3) – 2 ea
- Washers (6) – 4 ea
- Nuts (7) – 4 ea

NOTE

Stabilator Mass Balance Extension Arm must move up/ down unrestricted and without any binding.

9. Check mass balance extension arm for smooth up/down movement, clearance between extension arm and stop weldment of .01 - .03-inch. Use washers on spacers (4) to achieve proper clearance.
10. Remove cover from batteries.
11. Install baggage closeout panel.

STABILATOR DRIVE TORQUE TUBE REMOVAL

Perform this procedure to remove the drive torque tube for the stabilator. The Stabilator drive torque assembly is a custom-fit assembly that consists of the following:

- Stabilator Drive Weldments (Port/Starboard)
- Stabilator Torque Tube
- Stabilator Mass Balance Drive

Individual components may not be replaced. The entire assembly must be replaced as a unit.

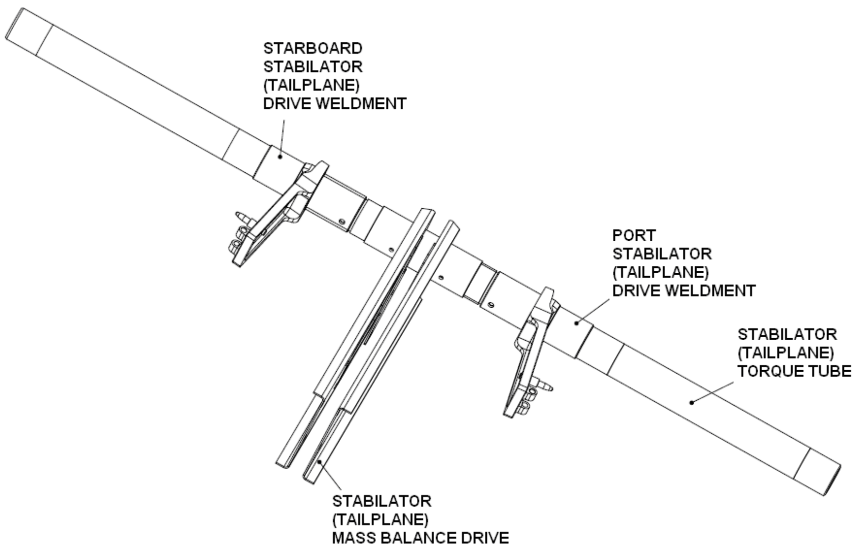

Figure 55-10 Stabilator Torque Tube

CAUTION

When dealing with the stabilator (Tailplane) drive torque assembly, take extreme care not to damage the torque tube.

CAUTION

Before starting procedure tail of airplane requires support. Failure to support tail of airplane may cause damage to tail section while accessing any area aft of passenger compartment.

CAUTION

As each item is removed, mark the item as to placement and orientation so items can be returned to original positions and aspects.

1. Remove starboard aft access panels (see Chapter 53 - *Fuselage*).
2. Remove trim actuator in accordance with Chapter 27 – *Flight Controls*.
3. Remove stabilator mass balance extension arm in accordance with the Stabilator Mass Balance Extension Arm Removal procedure on page 12 of this chapter.
4. Refer to Figure 55-11. Remove cotter pins, nuts, washers, and bolt (1) from port and starboard stabilator drive weldments and torque tube. Note the positions of the washers and remove stabilator drive weldments.

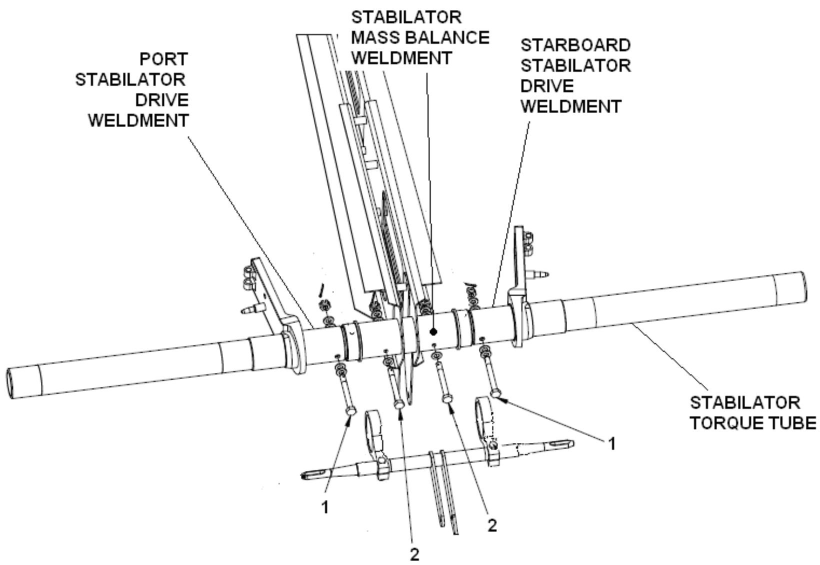

Figure 55-11 Stabilator Torque Tube

5. Remove cotter pins, nuts, washers, and bolts (2) from stabilator mass balance weldment and torque tube.
6. Refer to Figure 55-12. Remove two cotter pins, nuts (3), washers (2), and bolts (1) from trim drive assembly and slide trim drive plates from trim drive bearings and stabilator mass balance weldment.

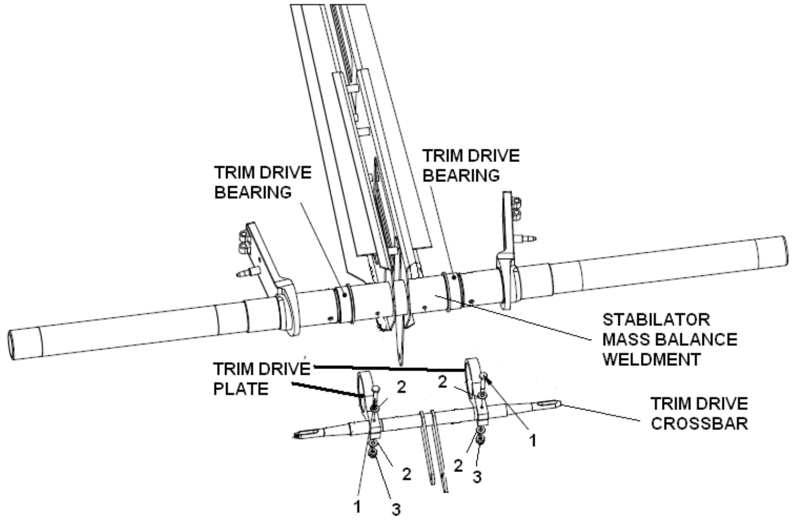

Figure 55-12 Stabilator Mass Balance Weldment

7. Slowly withdraw torque tube from port side of airplane and remove the starboard stabilator drive weldment.
8. Continue to slowly withdraw torque tube until stabilator mass balance weldment is free and can be lifted out from inside airplane. Remove mass balance weldment from airplane.
9. Continue to slowly withdraw torque tube and remove port stabilator drive weldment.

STABILATOR DRIVE TORQUE ASSEMBLY INSTALLATION

Perform this procedure to install the stabilator drive torque assembly.

NOTE

If torque drive torque assembly is being replaced, a dot is inscribed on end of new torque tube. The dot indicates port end of torque tube, and is to be aligned facing forward.

1. If new stabilator torque drive assembly is being installed, note dot on port end of torque tube.
2. Place new port stabilator drive plate into port stabilator bearing.
3. Working from the port side of the airplane, insert torque tube through the port stabilator drive plate.
4. Refer to Figure 55-12. Slide trim drive plates into position on bearing surfaces on mass balance weldment.
5. Refer to Figure 55-11. Insert torque tube through stabilator mass balance weldment.
6. Continue inserting torque tube and place starboard stabilator drive plate onto torque tube and into starboard stabilator bearing.

NOTE

If new stabilator drive torque assembly is being installed, use bolts supplied with new assembly for the following. If old assembly is being re-installed, use old bolts.

7. Temporarily, insert bolts into stabilator mass balance weldment and stabilator drive weldments (P/S) and stabilator torque tube.

CAUTION

No lateral, fore/aft, or up/down movement should be detected. If any lateral movement greater than .006" is detected, contact Liberty Aerospace Customer Support for additional instructions. If any fore/aft or up/down movement is detected, replace stabilator bearings in accordance with chapter.

8. From outside airplane, attempt to move the entire assembly from port to starboard, forward to aft, and up/down on port and starboard sides of airplane. Check for any movement.

9. Refer to Figure 55-11. When temporary installation is satisfactory, install washers and nuts (2) on bolts to attach stabilator mass balance weldment to torque tube.

CAUTION

Check the freedom of movement of trim drive plates. If trim drive plate bearings do not rotate freely, the bearing must be burnished with lapping compound until freedom of movement is attained.

10. Lubricate trim drive bearing surfaces with LPS 3 inhibitor (or equivalent) conforming to MIL-PRF-16173 grade 2 and install trim drive plates. Check the freedom of movement of the trim drive plates.
11. Torque to the lowest value of torque as shown in Chapter 20 – *Standard Practices*. Continue to torque to nearest castellated slot for split pin. Do not exceed the highest value of torque as shown in Chapter 20 – *Standard Practices*. Install a new split pin.
12. Remove each stabilator drive weldment and lubricate bearing surfaces with LPS 2 inhibitor (or equivalent) conforming to MIL-PRF-16173 grade 2; install each stabilator drive weldment to torque tube using close tolerance bolt, washer, and nut.
13. Torque to the lowest value of torque as shown in Chapter 20 – Standard Practices. Continue to torque to nearest castellated slot for split pin. Do not exceed the highest value of torque as shown in Chapter 20 – Standard Practices. Install a new split pin.

CAUTION

No lateral or fore/aft movement should be detected. If any lateral movement is detected, contact Liberty Aerospace Customer Support for additional instructions. If any fore/aft movement is detected, replace stabilator bearing(s) in accordance with chapter.

14. Check the drive torque assembly of the stabilator for movement in lateral, forward or aft for any discernable movement.
15. Check rotational movement of drive torque assembly is smooth and without binding.

CAUTION

No discernable differential movement is allowed between the port and starboard stabilator drive plates after installation of the assembly.

16. While holding the port stabilator drive plate, attempt to rotate the starboard stabilator drive plate in both directions; if any movement is detected, tighten stabilator drive plate bolts and return to step 15.
17. Refer to Figure 55-12. Reassemble trim drive assembly by inserting trim drive crossbar into trim drive plates, installing bolts, washers, and nuts. Torque to the lowest value of torque as shown in Chapter 20 – Standard Practices. Continue to torque to nearest castellated slot for split pin. Do not exceed the highest value of torque as shown in Chapter 20 – Standard Practices. Install a new split pin.
18. Check rotational movement of drive torque assembly is smooth and without binding.
19. Install trim actuator in accordance with Chapter 27 – *Flight Controls*.
20. Check rotational movement of drive torque assembly is smooth and without binding.
21. Install starboard aft access panels.

TORQUE TUBE COATING

This procedure applies a primer coat to the torque tube to prevent oxidation of the torque tube as required. This procedure needs to be done once. After which the torque tube is protected from further oxidation. All that is needed is an inspection of the surface of the torque tube for any other corrosion or damage as indicated in Chapter 5 – *Time Limits/Maintenance Checks /Inspection Intervals*.

NOTE

If the Torque tube already has a primer coat do not apply additional coats of primer. As an added protection, apply a very thin layer of Corrosion X to these areas.

1. If installed, remove the horizontal stabilators from the airplane as shown in Chapter 27 – *Control Surface*.
2. Carefully inspect the entire surface of the torque tube for any corrosion or other oxidation.
3. Mechanically remove any oxidation from the torque tube using the minimal means of removal as possible.

NOTE

Products such as navel jelly, that contain Phosphoric Acid, can be employed to help remove oxidation.

4. Use a clean cloth that has been soaked in alcohol to wipe down the surface of the torque tube to remove any residual foreign material or surface oil.
5. Use masking tape to mask off the tip ends of the tube and the inboard area of the boss. If the torque tube is out of the airplane, mask the area of the stabilizer bearing as shown in Figure 55-13

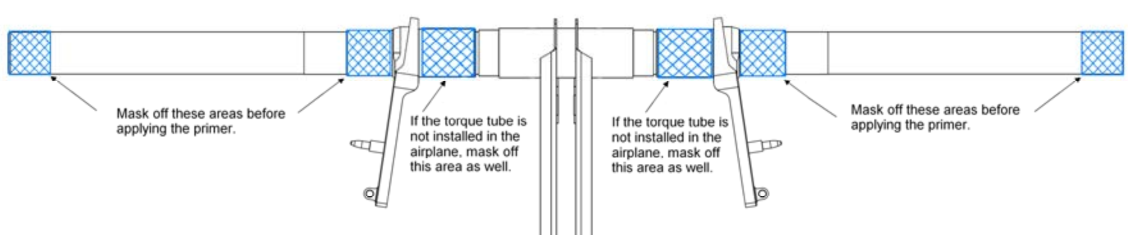

Figure 55-13 Torque Tube Areas Requiring Masking

6. Apply an even coat of Skilcraft Primer coating, T7-P-1757B, Type 1, Class C to the unmasked areas of the torque tubes. Allow sufficient time for the coating to cure and dry.

STABILATOR BEARING REMOVAL

Perform this procedure to remove the stabilator bearing.

1. Remove stabilator drive torque assembly in accordance with Stabilator Drive Torque Tube Removal procedure on page 17 of this chapter.
2. Refer to Figure 55-14. Remove eight screws (1), washers (2), and nuts (3) from stabilator bearing and fuselage, noting positions of washers.

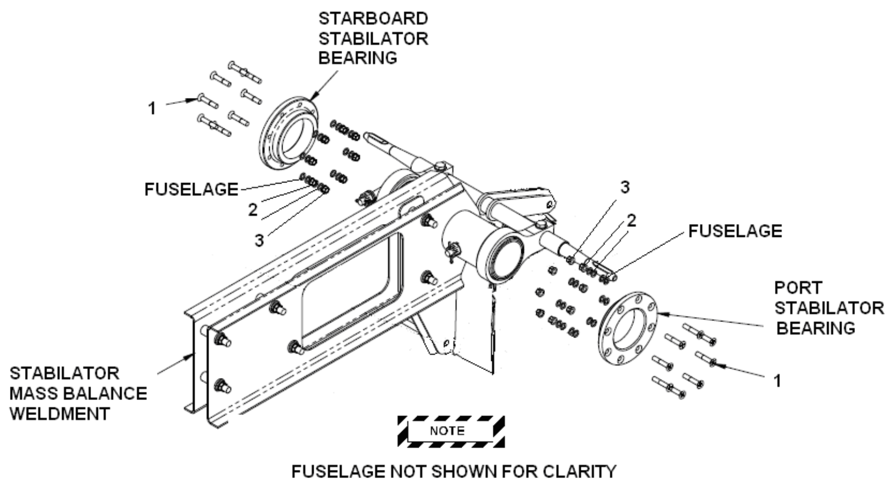

Figure 55-14 Stabilator Bearing Removal and Installation

CAUTION

Bearing removal requires two people – use second person to ensure that bearing, when removed, does not fall and become damaged.

Use extreme care to not damage Epibond®-1590 formed shim while removing bearing from its seat in fuselage.

3. Using a malleable (brass or wood) rod and working from the opposite side of the fuselage, carefully tap the inner portion of the bearing being removed, working around the perimeter of the bearing, and remove bearing from fuselage.

CAUTION

A formed Epibond® shim is used to position bearing and prevent binding of stabilator drive assembly. Check the Epibond® shim is not damaged and is ready to accept new bearing. If shim appears damaged, or if unsure if shim is acceptable, contact Liberty Aerospace Customer Support for additional instructions

4. Inspect the Epibond® shim for damage. If damaged, contact Liberty Aerospace Customer Support for additional instructions.

STABILATOR BEARING INSTALLATION

Perform this procedure to install the stabilator bearing.

CAUTION

A formed Epibond® shim is used to position bearing and prevent binding of stabilator drive assembly. Check the Epibond® shim is not damaged and is ready to accept new bearing. If shim appears damaged, or if unsure if shim is acceptable, contact Liberty Aerospace Customer Support for additional instructions

1. Check the inner surface of bearing mounting flange is clean.
2. Prior to inserting bearing into mounting hole, insert all bolts into bolt holes of bearing and check that bolts are countersunk to a depth of flush to .020”.
3. Coat the inner surface of bearing mounting flange with light coat of beeswax (or equivalent) as a release agent.
4. From outside fuselage, insert bearing into bearing mounting hole.
5. Refer to Figure 55-14. Install bearing using eight screws (1), washers (2), and nuts (3). Place washers according to positions noted during removal.
6. Insert applicable stabilator drive plate into bearing; check the freedom of movement without binding.

CAUTION

Check the freedom of movement of stabilator drive plate. If drive plate does not rotate freely, the bearing must be burnished with lapping compound until freedom of movement is attained.

7. Install stabilator drive torque assembly in accordance with Stabilator Drive Torque Tube Removal procedure on page 17 of this chapter.

CAUTION

Check the freedom of movement of stabilator drive torque assembly. If drive torque assembly does not rotate freely, determine if width of stabilator drive weldments are restricting movement, contact Liberty Aerospace Customer Support for additional instructions.

Check the smoothness of movement of stabilator drive torque assembly. If bearing is dragging, bearing must be burnished with lapping compound until freedom of movement is attained.

8. Check the rotational movement of drive torque assembly is smooth and without binding.

CAUTION

No lateral, fore/aft, or up/down movement should be detected. If any lateral movement greater than .006" is detected, contact Liberty Aerospace Customer Support for additional instructions. If any fore/aft or up/down movement is detected, replace stabilator bearing(s) in accordance with procedures in this chapter.

9. Check lateral, fore/aft, and up/down movement stabilator drive torque assembly movement for any discernable movement.

Section 10-02 Stabilator Inspection

As required, inspect the following areas of the stabilator.

- Visually inspect stabilator skin for dents, scratches, malformations, bubbled paint, or other foreign objects.
- Visually inspect all attachments for loose bolts.

NOTE

"Smoking Rivets" – Term used to describe a loose or working rivet whose vibration causes a black streak trailing aft. The presence of smoking rivets indicates a failure of the riveted joint.

- Inspect for "smoking rivets". If discovered, refer to Chapter 51 – *Standard Practices – Structure*.

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Section 55-30 Vertical Stabilizer and Rudder

The vertical stabilizer incorporates composite (carbon fiber) skins contiguous with the composite upper fuselage molding and composite internal structure including forward and rear spars and ribs.

The rudder is of conventional aluminum construction and includes a forward spar, ribs, and skin. A single full-length piano hinge on the right side of the forward spar secures the rudder to the rear spar of the vertical stabilizer. Fittings attached to the rudder forward spar transmit operating loads from the rudder control pushrod.

Rudder and rudder system components removal and installation procedures are provided in Chapter 27 – *Flight Controls*.

Section 30-01 Vertical Stabilizer and Rudder Inspection

As required, inspect the following areas of the vertical stabilizer and rudder.

- Visually inspect vertical stabilizer skin for dents, scratches, malformations, bubbled paint, or other foreign objects.
- Visually inspect vertical stabilizer rudder closeout for any cracking or sign of wear.
- Visually inspect all attachments for loose bolts.
- Inspect rudder skin for dents, scratches, malformations, bubbled paint, or other foreign objects.

NOTE

"Smoking Rivets" – Term used to describe a loose or working rivet whose vibration causes a black streak trailing aft. The presence of smoking rivets indicates a failure of the riveted joint.

- Inspect for "smoking rivets". If discovered, refer to Chapter 51 – *Standard Practices – Structure*.

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CHAPTER 56

WINDOWS

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Table of Contents

SECTION 56-00 GENERAL 5
SECTION 00-01 NORMAL MAINTENANCE PROCEDURE 5
SECTION 56-10 REPAIRS 7
SECTION 10-01 MINOR REPAIRS 7
SECTION 10-02 TEMPORARY REPAIRS 7
SECTION 10-03 PERMANENT REPAIRS 8
SECTION 10-04 POLISHING AND FINISHING 8
SECTION 56-20 CLEANING 9
SECTION 20-01 WAXING 9
SECTION 20-02 PROTECTION 9
SECTION 56-30 REPLACEMENT 11
SECTION 30-01 TOOLS 11
SECTION 30-02 MODIFIED PUTTY KNIFE 11
SECTION 30-03 DOOR WINDSCREEN PROCEDURES 12
DOOR WINDSCREENS REMOVAL 13
DOOR WINDSCREEN INSTALLATION 15
SECTION 30-04 REPLACING THE FORWARD WINDSCREEN 24
PREPARING THE AIRPLANE FOR WINDSCREEN REPLACEMENT 25
REMOVE THE FORWARD WINDSCREEN FROM THE AIRPLANE 26
SURFACE PREPARATION 27
APPLYING THE EPI-BOND® 1590-B ADHESIVE TO THE AIRPLANE 29
INSTALLING THE FORWARD WINDSCREEN SEAL 31
COCKPIT INSTALLATION, PROTECTIVE COVERINGS REMOVAL, AND INSPECTION OF AIRPLANE 33
SECTION 56-40 CRACKS IN THE FORWARD WINDSCREEN 35

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Section 56-00 General

The airplane windscreens are integral to the flight compartment; there is no separate passenger area. The forward windscreen (windshield) and door windscreens (windows) on the airplane are an acrylic plastic material and are an integral part of the fuselage or doors see Figure 56-1.

Figure 56-1 View of XL-2 Airplane Forward Windscreen and Door Windscreens
Sub Section Title

Section 00-01 Normal Maintenance Procedure

Employ normal maintenance procedures and precautions as appropriate for acrylic material. The forward windscreen manufacturer recommends the use of Plexies™ cleaner to clean all acrylic plastic materials.

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Section 56-10 Repairs

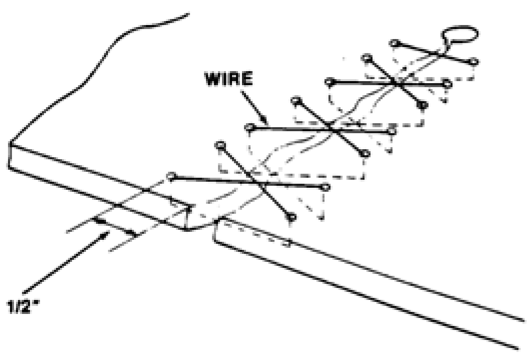
Repairs are applicable to plastic forward windscreens, enclosures, and door windscreens in non-pressurized airplanes. When there is damage to forward windscreens and side door windscreens made of acrylic plastics, replace the forward windscreen or side door windscreens unless the damage is minor and a repair would not be in the line of vision. Repairs usually require a great deal of labor. Replacement parts are readily available, so replacement is normally more economical than repair.

Section 10-01 Minor Repairs

There are times, however, when a small crack has developed in a forward windscreen or door windscreen. If there is no effect on safety, repair the forward windscreen or door windscreen by stop-drilling the ends of the crack with a # 30 drill (1/8 inch) to prevent the concentration of stresses causing the crack to continue. Drill a series of #40 holes a half-inch from the edge of the crack about a half-inch apart, and lace through these holes with brass safety wire (see Figure 56-2) and seal with clear silicone to waterproof. If the crack continues to grow beyond the holes, or additional cracks appear, replace the windscreen.

Section 10-02 Temporary Repairs

Temporary repairs are permissible, so long as the repair does not affect safety. Although a temporary repair will allow the windscreen to remain in service, replace the temporarily repaired windscreen by the next annual or 100-hour inspection. One way to make a temporary repair is to stop-drill the ends of the crack and then drill number 27 holes every inch or so in the crack, see Figure 56-2. Use AN515-6 screws and AN365-632 nuts with AN960-6 washers on both sides of the plastic. This will hold the crack together and prevent further breakage. The windscreen needs replacing by the next annual or 100-hour inspection.



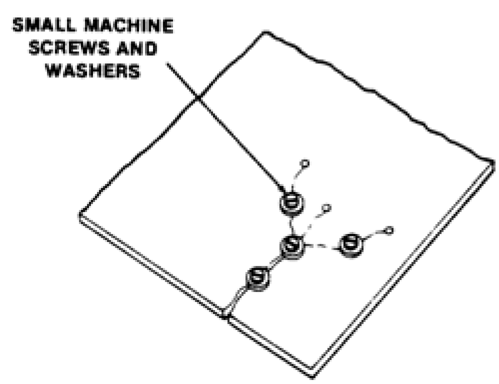


Figure 56-2 Temporary Repair of Cracked Forward Windscreen or Door Windscreen

Section 10-03 Permanent Repairs

Forward windscreens or side door windscreens with small cracks that affect only the appearance rather than the airworthiness of a sheet, may be repaired by first stop-drilling the ends of the crack with a # 30 or a 1/8-inch drill. Then use a hypodermic syringe and needle to fill the crack with polymerizable cement such as PS-30 or Weld-On 40, and allow capillary action to fill the crack completely. Soak the end of a 1/8-inch acrylic rod in cement to form a cushion and insert it in the stop-drilled hole. Allow the repair to dry for about 30 minutes, and then trim the rod off flush with the sheet.

Section 10-04 Polishing and Finishing

Within certain limitation, scratches and repair marks are removable from acrylic plastic. Do not do any sanding that could adversely affect the plastic's optical properties and distort the pilot's vision on any portion of a forward windscreen. If there are scratches or repair marks in an area that can be sanded, remove them by first sanding the area by using 320- or 400-grit abrasive paper that is wrapped around a felt or rubber pad.

Use circular rubbing motions, light pressure, and a mild liquid soap solution as a lubricant. After the sanding is complete, rinse the surface thoroughly with running water. Then, using a 500-grit paper, continue to sand lightly. Keep moving to higher grit paper. Continue sanding and rinsing the area, until there are no more marks from the sanding or repair.

After using the finest abrasive paper, use a rubbing compound and buff in a circular motion to remove all traces of the sanding.

Section 56-20 Cleaning

Clean Acrylic forward windscreens and door windscreens by washing them with mild soap and running water. Rub the surface with your bare hands in a stream of water. Follow with the same procedure but with soap and water. After flushing away the soap and dirt dry the surface with a soft, clean cloth or tissue and polish it with a forward windscreen cleaner especially approved for use on aircraft transparent plastics. Purchase these cleaners through aircraft supply houses.

Section 20-01 Waxing

A thin coating of wax will fill any minute scratches that may be present and will cause rain to form droplets that easily blow away.

Section 20-02 Protection

Acrylic forward windscreens sometimes called “lifetime windshields”, to distinguish them from those made of the much shorter-lived acetate material. However, even acrylic must be protected from the elements.

Parking the airplane in direct sunlight, will cause the forward windscreen to absorb heat and will actually become hotter than either the inside of the aircraft or the outside air. The sun will cause the inside of a closed aircraft to become extremely hot, and the plastic forward windscreen absorbs this heat.

To protect against this damage, it is wise to keep the aircraft in a hangar. If this is not possible, provide some type of shade to keep the sun from coming in direct contact with the forward windscreen.

CAUTION

Never use any cleaning product containing solvents of any kind. Never use any cleaning product containing abrasives of any kind.

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Section 56-30 Replacement

In the event, that a forward windscreen or a door windscreen needs replacing (scratched, cracked, crazed, etc.), obtain a replacement forward windscreen or door windscreen from Liberty Aerospace or an authorized supplier. Liberty Aerospace may ship replacements that are slightly oversized and may require trimming for correct fit.

Section 30-01 Tools

The following is a list of tools and/or components needed to replace the door windscreen. Assemble this list before starting the replacement procedure

- Replacement windscreen
- Phillip screwdriver
- Rubber Mallet
- Modified Putty Knife (see Section 30-02 Modified Putty Knife)
- Air Die Grinder
- 80-grit Die Grinder Disk (several)
- 180/200-grit Sand Paper
- Denatured Alcohol
- Epi-Bond® 1590-B Two-part Adhesive
- Pro-Seal Two-part Sealant
- Rags Several
- Soft Blanket to Cover Work Table
- Masking Tape
- 2-inch Cloth Ratchet Type Hold Down Straps
- Metal straps ½ to 1 inch wide, ¼ inch thick, 4-5 feet long
- Mold Release Paper
- Tongue Depressors
- Mixing Cups (several 200ml)
- Rubber Gloves

Section 30-02 Modified Putty Knife

The removal procedure requires the use of a modified putty knife to aid in breaking the adhesive bond between the windscreen and the doorframe.

Make the tool from a 1” to 1-½” wide putty knife. Round both corners as shown in Figure 56-3. The finished leading edge of the putty knife should be about 60% to 70% of the starting width. After cutting the corners, grind a shallow bevel back from the leading edge.

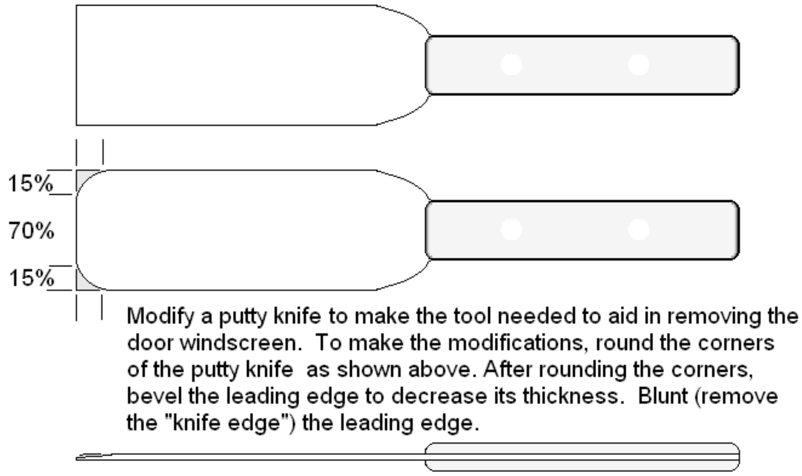


Figure 56-3 Details of the Tool to Aid in Removal of the Door Windscreen

Section 30-03 Door Windscreen Procedures

This section contains the removal and replacement procedures for the door windscreens.

DOOR WINDSCREENS REMOVAL

Perform the following procedure to remove the door windscreens. This procedure is somewhat involved and only a person experienced in working with acrylic windscreens and composite surfaces should attempt this procedure.

1. Remove the door from the airplane. For the procedure to remove the door, see Chapter 52 – *Doors*.
2. If the replacement door windscreen does not already have a covering to protect its surface, use the shrink-wrap plastic to protect the interior and exterior surface of the door and windscreen.
3. Use a MODIFIED putty knife, along the interior surfaces of the windscreen, start working the edge of the putty knife between the composite doorframe and the windscreen. Do not use any kind of prying motion.

CAUTION

Do not use any type of prying motion on the windscreen or composite surface. Attempting to pry the windscreen from the doorframe can cause non-repairable damage to the composite doorframe.

4. As you are working around the edge of the door windscreen with the modified putty knife, the windscreen should "pop" out of the doorframe.

NOTE

When the windscreen comes out of the doorframe, some of the exterior finish on the door will come off with the windscreen. This is normal. After replacing the windscreen in to the doorframe, the exterior finish will require repair.

5. Using an air die grinder fitted with 80-grit sandpaper, go around the entire opening in the doorframe and remove any of the old Epi-Bond® 1590-B two-part adhesive.

WARNING

SANDING OR GRINDING THE FUSELAGE, DOORS OR WINDSCREENS WILL CREATE A FINE DUST. USE A BREATHING MASK WHILE SANDING ANY COMPOSITE SURFACE. DO THE SANDING IN A WELL-VENTILATED AREA. ALSO, USE A VACUUM SYSTEN WITH A MICRO-FILTER TO REMOVE DUST PARTICALLS DURING THE GRINDING PROCESS.

CAUTION

Do not allow the grinder to come into contact with the black composite portion of the fuselage or door.

6. Using a rag damped with de-natured alcohol, remove any of the dust from the door and the windscreen opening.
7. Using 180 or 200-grit sandpaper, smooth the carbon surface. See Figure 56-4.

Figure 56-4 Sanding the Carbon Surface to Prep the Surface

8. Using a rag damped with de-natured alcohol, remove any of the dust from the door and the windscreen opening.
9. Visually inspect the composite surfaces of the doorframe to check for any cracking in the composite. If cracking is seen, call Liberty Aerospace, Inc. customer support for information.
10. Using a mist sprayer, mist the composite surface of the doorframe. Inspect the composite surface. If the water forms beads, that area of the doorframe needs a little more sanding. If the water forms "sheets", that area is ready for the Epi-Bond® adhesive. See Figure 56-5.
11. Wipe the doorframe again with alcohol.

This completes the procedure.

DOOR WINDSCREEN INSTALLATION

Perform the following procedure to install the door windscreens. This procedure is somewhat involved and only a person experienced in working with acrylic windscreens and composite surfaces should attempt this procedure.

CAUTION

The surface of the windscreen can be easily damaged or scratched. Cover the surface of the work area with a soft blanket or thick cloth.

1. Remove the wing from the airplane on the side where the windscreen is being installed. Refer to Chapter 57 – *Wings*, for the correct procedures to remove the wing.
2. Remove two inches of the outer edge of the interior covering from the replacement windscreen.
3. Remove ½ inch of the outer edge of the interior covering from the replacement windscreen.
4. Clean along the exposed surfaces with a rag dipped into alcohol.
5. Apply two-inch wide masking tape around the exterior edge of the windscreen. Apply the tape such that the edge of the tape is 1/8-inch back from the edge of the windscreen. Also, use masking tape to cover an voids or tares in the covering on the exterior surface of the windscreen.
6. Place the replacement windscreen in to the doorframe. Start in the corner above the door handle.
7. Apply sufficient pressure to the windscreen and working out from the corner, dry fitting the windscreen into the opening. Note any portion of the edge that does not fit flush into the doorframe. In addition, the windscreen surface should be just below the surface of the doorframe.
8. Tape the windscreen to the door sufficiently to hold the windscreen to the door.
9. Turn the door over. Using a semi-permanent marker, mark the surface of the interior windscreen where the doorframe comes up on the windscreen.
10. Remove the windscreen from the door.
11. Tape the interior surface of the windscreen between the edge of the protective covering and the line drawn by the marker.
12. Using 80-grit sandpaper, hand sand the exposed edge of the windscreen, sufficient to scuff the surface of the windscreen. Do not sand above the mark into the masking tape.

Figure 56-5 Sanding the Edge of the Door Windscreen to Prep for the Adhesive

13. Wipe the surface of the scuffed area of the windscreen.

NOTE

The next several steps require three people, one person inside the airplane, and two people outside the airplane. Do not attempt this procedure without having three people.

CAUTION

Mask-off and cover the interior surfaces of the cockpit to prevent damage to the interior during the next several steps.

14. Prepare three 2-inch wide cloth straps that will be used to hold the windscreen to the door. The go from the space frame to the openings in the upper fuselage that provide access to door hinge pins.

15. Install the door on to the airplane. For the procedure to install the door on to the airplane, see Chapter 53 – *Doors*. It is not required to install the gas cylinders at this time. Installing the door into the airplane at this point will assure the door will fit correctly after the adhesive cures. Position the three straps as shown in Figure 56-8.

16. Close and securely latch the door.

17. Mix a batch of the Epi-Bond® 1590-B two-part adhesive. Use 100 ml of part A and 55 ml of part B. Mix for six minutes before apply to the doorframe or windscreen.

18. Carefully apply a modest amount of the Epi-Bond adhesive to the surface of the doorframe and the windscreen. See Figure 56-6.

Figure 56-6 Appling a Modest Amount of Adhesive to the Door Frame and the Windscreen

19. With one person on the inside of the airplane and one person on the outside of the airplane, carefully set the windscreen into the doorframe from the outside. With the person inside the airplane supporting the windscreen, start by placing the windscreen in the corner just above the door latch. Working out from the corner, press the windscreen in to the doorframe. See

Figure 56-7 Fitting the Windscreen into the Doorframe

20. After the windscreen is completely in the inserted into the doorframe and while one of the outside people holds the windscreen in the opening, the third person covers the entire door and windscreen with a sheet of mold release paper.

21. Place the cloth straps over the mold release paper and draw them up snug but not tight. The two outside straps need to run along the vertical seam between the doorframe and the windscreen. The center strap needs to go across the middle. See Figure 56-8.

Figure 56-8 Placing the Straps Over the Mold Release Paper

22. Place metal straps under the cloth straps and along the top and bottom horizontal seams. See Figure 56-9 and Figure 56-10

Figure 56-9 Place the Metal Bands Under the Cloth Straps Over the Horizontal Upper Seam of the Doorframe and the Windscreen

Figure 56-10 Place the Metal Bands Under the Cloth Straps Over the Horizontal Lower Seam of the Doorframe and the Windscreen

23. Tighten down the cloth straps enough to hold the window in place for 24-hours.
24. After the straps a firmly in place, the person on the inside of the airplane needs to start removing any access adhesive along the in side edge. See Figure 56-11.

Figure 56-11 Use a Tongue Depressor to Remove the Excess Adhesive from the Interior Edge Between the Doorframe and the Windscreen

25. After removing the excess adhesive, wipe the entire edge of the window and doorframe with a fresh rag moistened with alcohol. See

Figure 56-12 Clean the Interior of the Joint Between the Doorframe and the Windscreen with a Clean Cloth Moistened with Alcohol

26. Allow the Epi-Bond® 1590-B two-part adhesive to cure for 24-hours.

27. After 24-hours, remove the door from the airplane. Place the door on a padded work surface.

28. Inspect along the entire edge of the windscreen and look for any voids in the adhesive, see Figure 56-13. If the adhesive has not come all the way to the edge and there is good adhesion between the doorframe and windscreen, fill the void area with a small amount of the Epi-Bond® adhesive.

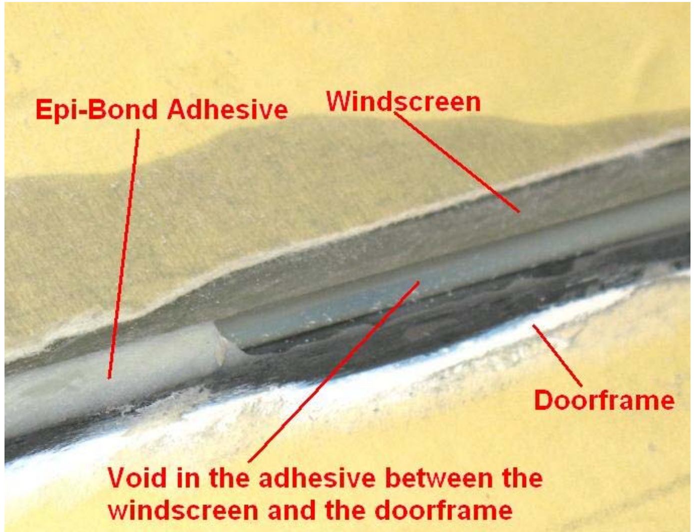


Figure 56-13 Photograph Showing a Void in the Epi-Bond

WARNING

SANDING OR GRINDING THE FUSELAGE, DOORS OR WINDSCREENS WILL CREATE A FINE DUST. USE A BREATHING MASK WHILE SANDING ANY COMPOSITE SURFACE. DO THE SANDING IN A WELL-VENTILATED AREA. ALSO, USE A VACUUM SYSTEN WITH A MICRO-FILTER TO REMOVE DUST PARTICALLS DURING THE GRINDING PROCESS.

CAUTION

Do not allow the grinder to come into contact with the black composite portion of the fuselage or door.

29. Use an 80-grit sanding wheel on an air die-grinder, carefully remove any excess cured adhesive from around the edge of the windscreen. Do not grind into masking tape. See Figure 56-14.

Figure 56-14 Grinding the Cured Epi-Bond on the Outside Surface of the Doorframe and Windscreen

30. Using a blue Scotch-Brite® pad that has a beveled edge, work along the interior edge of the windscreen, and carefully remove any cured adhesive that remains. See 22Figure 56-15.

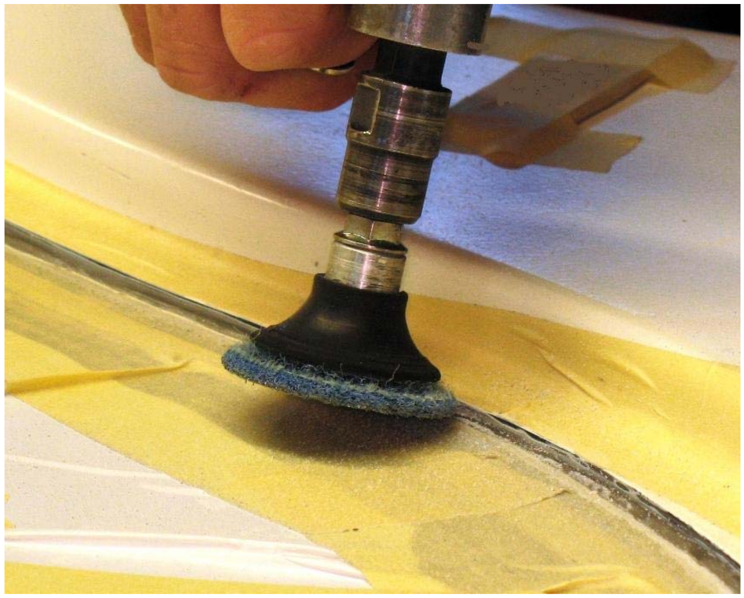


Figure 56-15 Grinding the Cured Epi-Bond on the Inside Surface of the Doorframe and Windscreen

31. Wipe all surfaces of the door and windscreen with a cloth that is moistened alcohol.

32. Remove the masking tape.

33. Mask-off an area along the edge of the doorframe around the windscreen. Mask-off an area on the windscreen approximately ¼-inch from the edge of the doorframe. Mask-off an area on the doorframe around the edge of the windscreen a maximum of ⅛-inch from the edge of the fuselage.

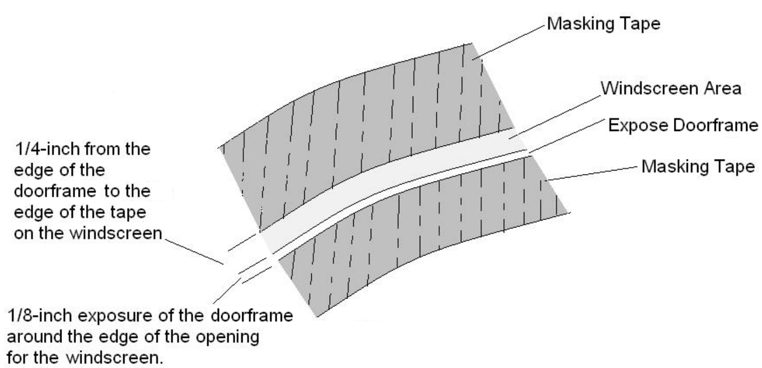


Figure 56-16 Diagram Showing the Area to Mask-off Before Applying the Pro-Seal Sealant

34. Mix-up an amount of Pro-Seal two-part sealant and apply it along the area of the windscreen masked-off.

35. Use an applicator tool to spread the Pro-Seal sealant around the edge of the windscreen, forcing the sealant into the gap between the windscreen and the doorframe.
36. After spreading the Pro-Seal sealant, and before it starts to cure, carefully remove the masking tape from the windscreen and the doorframe.
37. Allow the Pro-Seal to cure for about 4-hours.
38. Install the door into the opening in the fuselage.
39. Hold door in closed position.
40. Align hinges and re-insert hinge pins.
41. Open the door and have someone support the door.
42. Slightly compress gas spring and snap cylinder end of gas spring to fitting in door.
43. Insert the locking ring.
44. Replace the two cabin headliner upholstery access covers.
45. Install the wing removed in step 1 of this procedure.

This completes the Door Windscreens procedure

Section 30-04 Replacing the Forward Windscreen

Replacing the forward windscreen involves a number of procedures. The procedure to replace the forward windscreen is very involved, only personnel experienced in working with acrylic windscreens, and composite surfaces should attempt this procedure.

Step	Procedure	Page
1	Preparing The Airplane For Windscreen Replacement	25
2	Remove The Forward Windscreen From The Airplane	26
3	Surface Preparation	27
4	Applying The Epi-Bond® 1590-B adhesive To The Airplane	29
5	Installing The Forward Windscreen Seal	31
6	Cockpit Installation, Protective Coverings Removal, And Inspection Of Airplane	33

PREPARING THE AIRPLANE FOR WINDSCREEN REPLACEMENT

Perform the following procedure to prepare the airplane for the replacement of the forward windscreen.

1. Around the edge of the forward windscreen, install a plastic or paper covering over the exterior surface of the airplane, sealing the edge with tape. Liberty Aerospace recommends using Shrink-Wrap plastic, as it will cling to the surface of the airplane.

CAUTION

Before starting these procedures, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

2. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
3. Pull the BAT1 (CB001) circuit breaker to OPEN.
4. Install a tail stand underneath the tail section of the airplane.
5. Remove the cabin aft bulkhead access panel, by removing securing screw hardware.
6. Disconnect the negative then the positive leads from the primary battery. Isolate the negative terminals on the batteries to prevent accidental connection.

CAUTION

Failure to disconnect the batteries can cause damage to the electrical circuitry of the airplane.

7. Grinding on the airplane's surfaces during this procedure will produce a fine dust or particulate. Cover the interiors of the airplane to protect these areas from the dust and particulate.
8. If the replacement forward windscreen does not already have a covering protecting the interior and exterior surface, install a Shrink-Wrap plastic in these surfaces to protect them during this procedure.
9. Remove the Instrument Panel Console from the airplane. For the procedure to remove the Instrument Panel Console, see Chapter 53 – *Fuselage*.
10. Remove the Magnetic Compass from the forward Windscreen. For the procedure to remove the Magnetic Compass, see Chapter 34 – *Navigation and Pitot/Static.*

REMOVE THE FORWARD WINDSCREEN FROM THE AIRPLANE

Perform the following procedure to remove the forward windscreen from the airplane.

1. If there is a foam sound proofing material covering on the interior surface, carefully peel this foam back from the edge of the forward windscreen.
2. Remove the nuts and washers from around the edge of the forward windscreen.
3. Remove the screws. RTV silicon holds the screws in place.

NOTE

There are two different length screws used in the forward windscreen. The shorter screws (½-inch long) are along the front edge of the forward windscreen. The longer screws (⅝-inch long) are along the top edge of the forward windscreen

4. Use a very sharp razor knife and carefully cut around the edge of the forward windscreen to cut the Pro-Seal (black sealant around the edge of the forward windscreen).

NOTE

Use care when cutting around the forward windscreen. Do not cut into the composite fuselage.

5. Using a rubber mallet, gently tap around the edge of the forward windscreen to aid in breaking the adhesive bond between the forward windscreen and the fuselage.

NOTE

Do not use any type of prying device to separate the forward windscreen from the fuselage. Doing so may cause damage to the composite around the edge of the forward windscreen opening. Also, do not strike the fuselage with the rubber mallet.

6. Remove the forward windscreen from the airplane.

SURFACE PREPARATION

Perform the following procedure to prepare the different surfaces for the installation of the forward windscreen.

1. Carefully scrape around the edge of the fuselage to remove all remaining Pro-Seal sealant.

NOTE

DO NOT use any type of solvent based cleaner on the surface of the fuselage.

2. Using an 80-grit sanding wheel mounted on a die grinder, carefully sand the inside surface of fuselage around the opening for the forward windscreen to remove the layer of Epi-Bond® 1590-B adhesive.

WARNING

SANDING OR GRINDING THE FUSELAGE, DOORS OR WINDSCREENS WILL CREATE A FINE DUST. USE A BREATHING MASK WHILE SANDING ANY COMPOSITE SURFACE. DO THE SANDING IN A WELL-VENTILATED AREA. ALSO, USE A VACUUM SYSTEN WITH A MICRO-FILTER TO REMOVE DUST PARTICALLS DURING THE GRINDING PROCESS.

CAUTION

Do not allow the grinder to come into contact with the black composite portion of the fuselage or door..

3. Apply a strip of tape around the edge of the forward windscreen. See Figure 56-17 for location of the tape.

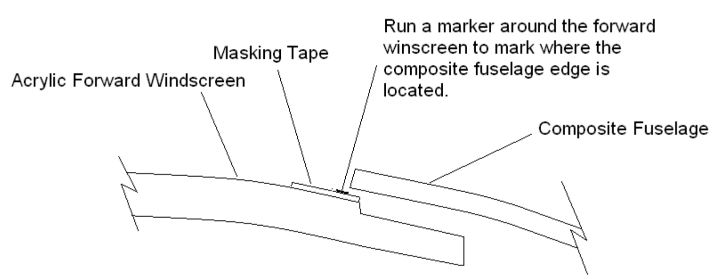


Figure 56-17 Forward Windscreen Showing Location of the Masking Tape

4. Dry fit the forward windscreen in the opening in the fuselage. Check along the entire edge of the forward windscreen to make sure the forward windscreen sits centered in to the opening and has even coverage along the under edge of the fuselage.

NOTE

To locate and mark the top edge of the forward windscreen, place the forward windscreen on the ground along one edge. The forward windscreen will sit squarely on the top edge. If it sits at all on the forward edge, without tipping over, it will sit only with a small portion of the forward edge touching the ground.

5. Use small to medium sized vice-grip type "C" clamps to hold the forward windscreen in place.
6. Locate the lower center screw hole in the fuselage. Use a #27-drill bit and drill through this hole and in to the forward windscreen.

NOTE

When drilling Acrylic, only use a drill bit designed for drilling Acrylic. Other drill bits will damage the Acrylic.

7. Use a sharp permanent marker, and mark the tape on the forward windscreen along the edge of the opening in the fuselage, See Figure 56-17.
8. Remove the forward windscreen from the airplane.
9. Use a die grinder, fitted with an appropriate bit, to grind away the top surface of the forward windscreen that was under the edge of the composite fuselage. See Figure 56-18 for details.
10. After grinding down the forward windscreen, ease the edge by using 80-grit sandpaper. Dry fitting the forward windscreen into the fuselage, there should be a small gap (⅛ – ¼ inch) between the edge of the fuselage opening and the top surface of the forward windscreen.

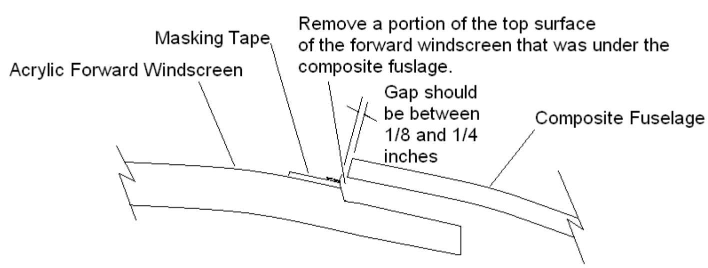


Figure 56-18 Forward Windscreen Showing the Area Under the Edge of The Fuselage of the Top Surface Removed

APPLYING THE EPI-BOND® 1590-B ADHESIVE TO THE AIRPLANE

Perform the following procedure to apply the Epi-Bond® 1590-B adhesive layer to the fuselage in preparation for the replacement windscreen.

1. Refit the forward windscreen into the opening in the fuselage, aligning the hole drilled in step 6 in the procedure Preparing The Airplane For Windscreen Replacement on Page 27, and clamping the forward windscreen with the vice grips.
2. Starting in the lower center of the forward windscreen and working out from this point, drill every other hole in the fuselage, and drill through the existing hole in the fuselage and into the forward windscreen. Use a drill specifically designed for drilling into Acrylic. Drill only five holes on either side of the lower center. It is not necessary to drill the holes along the top edge of the forward windscreen at this time.

NOTE

When drilling Acrylic, only use a drill bit designed for drilling Acrylic. Other drill bits will damage the Acrylic forward windscreen.

3. Remove the forward windscreen.

CAUTION

The usable "pot life" of Epi-Bond® 1590-B adhesive is 30 minutes. Complete steps 4, 5, and 6 within this 30-minute pot life window.

4. Mix a batch of Epi-Bond® 1590-B adhesive. Apply a thin layer to both the inside surface of the fuselage around the opening for the forward windscreen and along the edge of the forward windscreen.
5. Clamp the forward windscreen using the vice-grip "C-clamps".
6. Use ⅛ inch spring loaded Clecos, or 6-32 screws, washers, and nuts, to hold temporarily the forward windscreen to the fuselage, using in the holes drilled in step 2 of this procedure. Coat Clecos or the screws with wax or mold release agent; do not use any type of oil or grease. If you are using screws to hold the windscreen to the fuselage, tighten the screws only enough to secure the forward windscreen to the fuselage. Torque the screws and nuts to a maximum of 10 inch-pounds. Do not tighten the screws in an attempt to close any gap between the forward windscreen and fuselage.

NOTE

Always coat whatever fastener used to hold the windscreen to the fuselage with wax or a mold release agent. This will prevent the Epi-Bond® 1590-B adhesive from adhering to the fastener.

NOTE

If you use screws washers and nuts, it is a recommendation not to use the screws that originally came on the airplane. Use the originally screws that came on the airplane to secure the forward windscreen to the fuselage after the Epi-Bond® 1590-B adhesive has cured. The screws used here do not need to be aircraft rated screws. The screws used here are only to hold the forward windscreen to the fuselage while the Epi-Bond® 1590-B adhesive is curing.

CAUTION

Over tightening of the screws at this point, may cause the acrylic forward windscreen to crack.

7. Clean any excess Epi-Bond® 1590-B adhesive that comes out of the holes in the fuselage and from the surface of the forward windscreen.
8. The Epi-Bond® 1590-B adhesive takes about 24-hours to cure.

INSTALLING THE FORWARD WINDSCREEN SEAL

Perform the following procedure to install the windscreen seal.

1. Starting with the upper center of the forward windscreen and working out from this point, use a #27-drill to drill through the existing hole in the fuselage and into the forward windscreen. Use a drill specifically designed for drilling into Acrylic.
2. Drill the remaining holes along the front edge of the forward windscreen.
3. If the forward windscreen has screws holding it to the fuselage, remove the screws installed in step 6 and recycle. If the forward windscreen has Clecos holding it to the fuselage, remove the Clecos installed in step 6. Remove the clamps.

NOTE

The forward windscreen should hold in place with just the Epi-Bond® 1590-B adhesive, however, if the forward windscreen is bumped before completing step 4, the forward windscreen can pop out of the fuselage.

4. Apply a small amount of RTV 100 silicon sealer to each of the holes. Starting in the upper and lower centers screws and working out from there, install the screws, washers, and nuts. Use the shorter screws (½-inch long) along the front edge of the forward windscreen and the longer screws (⅝-inch long) along the top edge of the forward windscreen. The torque on the nuts can be no more than a maximum of 10 inch-pounds.

CAUTION

Applying a torque on the screw/nuts to a torque of greater than 10 inch-pounds, can cause damage to the forward windscreen.

5. After the silicon sealer on the screws has completely cured, clean off any excess from the fuselage. Remove the tape and covering from the forward windscreen.

NOTE

Before applying the masking tape in the next step, inspect the entire length of the edge of the opening for any remaining Epi-Bond® 1590-B adhesive or silicon sealer. Remove, clean or minimize any remaining surface imperfections or materials.

6. Mask-off an area along the edge of the fuselage around the forward windscreen. Mask-off an area on the forward windscreen approximately ¼-inch from the edge of the fuselage. Mask-off an area on the fuselage around the edge of the forward windscreen a maximum of ⅛-inch from the edge of the fuselage.

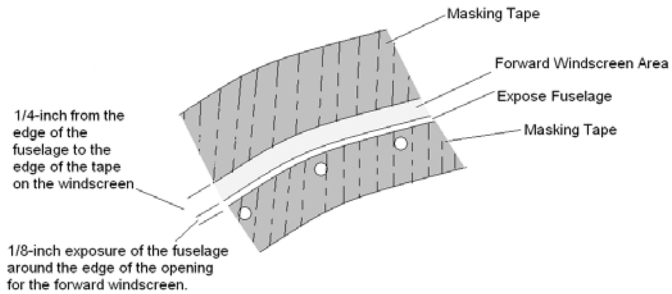


Figure 56-19 Detail showing the Masked-off Area of the Forward Windscreen

7. Mix-up an amount of Pro-Seal two-part sealant and apply it along the area of the forward windscreen masked-off.
8. Use an applicator tool to spread the Pro-Seal sealant around the edge of the forward windscreen, forcing the sealant into the gap between the forward windscreen and the fuselage.
9. After spreading the Pro-Seal sealant, and before it starts to cure, carefully remove the masking tape from the forward windscreen and the fuselage.
10. Remove the covering from the power distribution harness assembly. Use compressed air to blow out any particulate that may have gotten on to the power distribution harness assembly.

COCKPIT INSTALLATION, PROTECTIVE COVERINGS REMOVAL, AND INSPECTION OF AIRPLANE

Perform this procedure to remove the protective coverings and to inspect the airplane prior to a return to fight.

1. Install the Magnetic Compass. For the procedure to install the magnetic compass, see Chapter 34 – *Navigation and Pitot/Static*.
2. Install the Instrument Panel. For the procedure to install the instrument panel, see Chapter 53 – *Fuselage*.
3. Remove the protective coverings from the interior and exterior areas of the airplane.
4. Clean any areas using a vacuum and a solution of mild soap and water.
5. Inspect around the edge of the forward windscreen for any cracks or scratches.
6. Inspect for any extra Epi-Bond® adhesive, RTV sealant, or Pro-Seal Sealant. Carefully remove any excess adhesive or sealant. Do not use any solvent-based cleaners on the surface of the fuselage or forward windscreen.
7. Position the ALT and BAT master switches in the OFF position, Position the FADEC PWR A and B switches, and the ignition switch to OFF.
8. Pull the BAT1 (CB001) circuit breaker to OPEN.
9. Install a tail stand underneath the tail section of the airplane.
10. Clean battery and battery cable terminals with a stiff brush and contact cleaner.

CAUTION

Always connect negative battery cable last.

11. Install positive battery lead P01A2 with terminal bolt.
12. Apply a torque of 70 in-lbs ± 5 in-lbs.
13. Install negative battery lead P07A2N with terminal bolt.
14. Apply a torque of 70 in-lbs ± 5in-lbs.
15. Apply a thin coating of Dow Corning 4 (DC4) electrical insulating compound over the terminal/bolt assembly.
16. Install cabin aft bulkhead access panel.
17. Push in the BAT1 (CB001) to CLOSE.
18. Remove the tail support.

19. Perform the following tests as defined by Chapter 05 – Time Limits/Maintenance Checks/Inspection Intervals:
    - Magnetic Compass Compensation (“Swing”) Procedure in Chapter 34 – *Navigation and Pitot/Static*
    - Check instruments for functional operation in Chapter 31 – *Indicators and Recording Systems*
    - Check Avionics for functional operation in Chapter 23 – *Communications*
    - Check the Cabin Heat ducts as in Chapter 21 – *Environmental Systems*

Section 56-40 Cracks in the Forward Windscreen

During an earlier Liberty XL-2 full-scale damage tolerance and fatigue test, Liberty Aerospace, Inc. observed a crack in the forward windscreen of the test article.

As a result, during periodic aircraft inspections it is important to inspect for forward windscreen cracks. Pay close attention to the forward windscreen attachment area around the perimeter of the forward windscreen local to the screw line. (See drawings on the following page). If cracks are observed in the forward windscreen acrylic adjacent to the top headliner region of the fuselage, please contact Liberty Aircraft, Technical Support for appropriate guidance. If cracks form outside of this region, a qualified A&P Mechanic in accordance with previous pages of this Chapter must repair the aircraft.

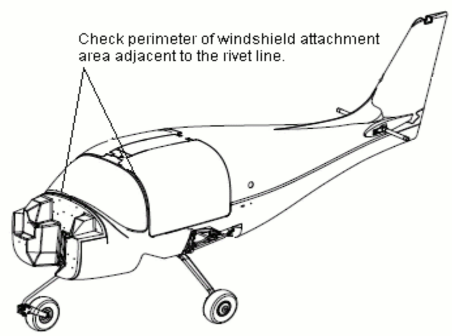
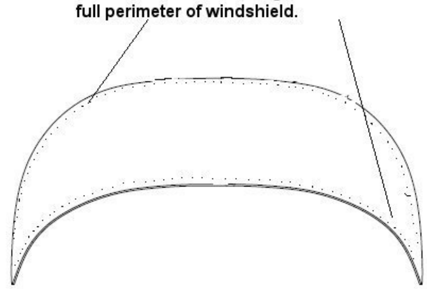


Figure 56-20 Check Perimeter of Forward Windscreen for Cracks



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CHAPTER 57

WINGS

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Table of Contents

SECTION 57-00 GENERAL 5
SECTION 00-01 WING REPAIR 5
SECTION 57-10 WINGS MAIN FRAME 7
SECTION 10-01 DIFFERENT WING TYPES 7
GROSS WEIGHT 1750 LBS OPTION ONLY INTRODUCTION OF REAR SPAR TANG, REINFORCEMENT (BOTH PORT AND STARBOARD WINGS) 8
SECTION 10-02 ATTACHMENT FITTINGS 10
SECTION 10-03 MANUAL OPERATED WING PINS 11
MANUAL WING REMOVAL 12
MANUAL WING INSTALLATION 14
SECTION 10-04 ELECTRICAL OPERATED WING PINS 16
SECTION 10-05 ELECTRICAL WING PROCEDURES 16
WING REMOVAL (WITH ELECTRICALLY OPERATED WING PINS) 17
WING INSTALLATION (ELECTRICALLY OPERATED WING PINS) 19
SECTION 10-06 WING ATTACHMENT FITTINGS INSPECTION 21
SECTION 10-07 WING LIMIT LOAD CONDITION 23
SECTION 57-11 WING SPARS, RIBS, AND STRINGERS 25
SECTION 57-14 WING ATTACHMENT 27
SECTION 14-01 WING BOX PROCEDURES 28
SECTION 14-02 AFT WING CLEVIS PIN WELDMENT 28
SECTION 14-03 AFT WING CLEVIS PIN WELDMENT PROCEDURES 28
AFT WING CLEVIS PIN WELDMENT REPAIR 29
SECTION 57-20 WING SKIN AND ACCESS PANELS 33
SECTION 57-30 WING TIP 35
WING TIP REMOVAL 36
WING TIP INSTALLATION 37
SECTION 57-60 FLIGHT CONTROL SURFACES 39
SECTION 60-01 AILERONS 39
SECTION 60-02 FLAPS 39
SECTION 57-61 WING ROOT FAIRING 41
SECTION 61-01 WING ROOT FAIRING PROCEDURES 41
ANTI CHAFING TAPE REMOVAL 42
ANTI CHAFING TAPE INSTALLATION 43

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Section 57-00 General

The airplane wings are of conventional aluminum construction and include main, leading edge, and rear spar, ribs, stringers, and wing skins. The wing's main spar slides within the wing box located in the Space Frame Assembly for both port and starboard. Two, 1-inch wing pins secure wing main spar within the wing box. The wing lock aft pin secures the wing aft spar to the Space Frame Assembly.

Aileron and flap controls are designed to be a quick connect when wing assemblies are installed.

Ailerons and flaps are of conventional aluminum construction and secured to wing rear spar by three flap hinges and two aileron hinges. The hinge line of the flap is located below the wing lower surface for proper flap extension geometry. The aileron hinges are flush with the wing lower surface.

Wing root aerodynamic fairing and wing tip assemblies are both fabricated from composite material. Only the wing tips are removable.

Internal conduits within each wing accommodate wiring for wing tip mounted position and anti-collision lights and facilitate replacement or maintenance of wiring, as well as the pitot/static blade that is installed on the lower surface of the port wing only. Wing removal is accomplished by removal of three wing pins, either by electronic method or by manually wing pin removal. Access to wing root electrical and pneumatic connections, and the electrical connectors for the wing pin actuators, is gained by removal of the fuselage belly panel.

Maintenance procedures covered in this manual are limited to removal and replacement of entire wing(s).

Section 00-01 Wing Repair

Liberty Aerospace, at this time has not published a repair procedure. When this procedure is compiled and approved, it will become a part of this manual. If any damage is incurred to the aircraft before that time, contact Customer Service at Liberty Aerospace for assistance. Damage is listed as:

- Negligible – Any smooth dents in the wing surface, free from cracks, abrasions, and sharp corners. No stress wrinkles and does not interfere with any internal structure and control circuit. Further definition is available from Liberty Aerospace Customer Service.
- Repairable – Any damage that is minor in nature i.e. small holes, creases, or abrasions that require the surface to be treated but do not require replacement as deemed by Liberty Aerospace Customer Service or qualified service personnel.
- Replacement – If any damage that is extensive or major enough, where repair costs exceed that of replacement costs, Contact Liberty Aerospace Customer Service.

The wings of the Liberty XL-2 must be inspected per Chapter 04 – *Airworthiness Limitations* and Chapter 05 – *Time Limits/Maintenance Checks/Inspection Intervals*.

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Section 57-10 Wings Main Frame

This section describes the main frame of the wing as it interfaces and connects to the fuselage. This section includes procedures to remove and install the wings, remove and install the anti-chafing tape on the wing root fairing, and the inspection of the wings

Section 10-01 Different Wing Types

There are two different wing types.

The first type of wings was used on the early 1653 lbs gross weight airplanes. The part number of the wings is:

- Main Wing Starboard Assembly Aileron and Flap Heavy Wings, 135A-20-006
- Main Wing Port Assembly Aileron and Flap (Heavy Wing), 135A-20-005

The second type of wings is used on the early 1653 lbs gross weight airplanes that have the 1750 lbs. gross weight modification and on all 1750 lbs. gross weight airplanes. The part number of the wings is:

- Main Wing Starboard Assembly Aileron and Flap Heavy Wings, 135A-20-007
- Main Wing Port Assembly Aileron and Flap (Heavy Wing), 135A-20-008

CAUTION

For airplane installed with Main Wing Starboard Assembly Aileron and Flap Heavy Wings, 135A-20-006, and Main Wing port Assembly Aileron and Flap (Heavy Wing), 135A-20-005, must be removed for Gross Weight Compliance and replaced with Wing and Aileron and Flap Assembly, port (135A-20-007) and starboard (135A-20-008).

GROSS WEIGHT 1750 LBS OPTION ONLY INTRODUCTION OF REAR SPAR TANG, REINFORCEMENT (BOTH PORT AND STARBOARD WINGS)

The following steps describe the installation procedures (both port and starboard wings) for the Gross Weight 1750 lbs Option for Rear Spar Tang, Reinforcement:

1. Deploy flaps fully down.
2. Support the wing.
3. Actuate the aft wing pin out of the aft tang on the wing.

NOTE

This procedure does not involve Liberty XL-2 wings removal.

4. Position one (1 qty.) reinforcement (P/N: 135A-20-713) on the aft side of the aft wing tang and one (1 qty.) blank reinforcement (P/N: 135A-20-863) on the forward side of the aft wing tang, as shown in Figure 57-1 and Figure 57-2.
5. Engage the aft wing pin back into the aft wing tang and the two reinforcements.
6. Line up the reinforcements with the aft wing tang edge and match drill using the pilot holes of the aft reinforcement. Step drill using 90-degree drill and short drill bits up to a final size of the hole is 0.250” +0.005 / -0.001.

WARNING

IF THIS STEP IS PERFORMED INCORRECTLY, REPORT TO THE LIBERTY CUSTOMER SUPPORT FOR FURTHER DISPOSITION.

7. Install hardware as shown in Figure 57-2. For the port side secure the reinforcements with bolt head facing aft. For the starboard side secure the reinforcements with bolt head facing aft. Torque to 50 to 60 in-lbs.
8. Repeat procedure for the opposite wing. Then push the wing pins back in making sure they have gone back all the way. The installed spar tang reinforcement is shown in Figure 57-1.

NOTE

Verify wings are securely installed with all wing pins fully engaged.

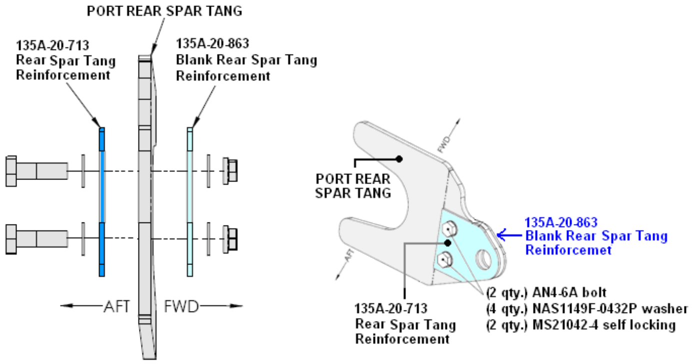

Figure 57-1 Port Rear Spar Tang Reinforcement Installation

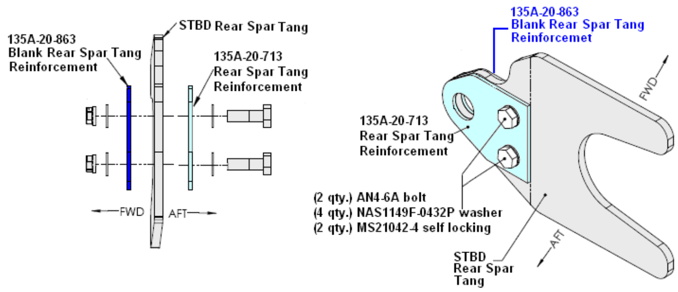

Figure 57-2 Starboard Rear Spar Tang Reinforcement Installation

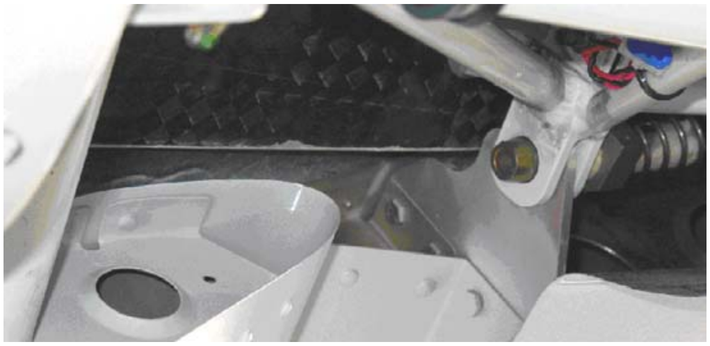

Figure 57-3 Location of Aft Wing Tang, Prior to Reinforcements

Figure 57-4 Tang Reinforcements Installed

Section 10-02 Attachment Fittings

Three fittings secure each wing to the fuselage. Upper and lower main spar fittings transmit primary lift and secondary drag loads to the fuselage center section chassis. A fitting on the rear spar transmits primary drag and secondary lift loads to the fuselage center section chassis. The aircraft may be installed with the mechanical wing lock mechanism or the electrical wing lock mechanism.

The following section describes the removal and installation procedure is the section dealing with the inspection of the attachment fittings.

CAUTION

Wing removal requires three people; one extra person is utilized for flap control. Each wing panel is heavy.

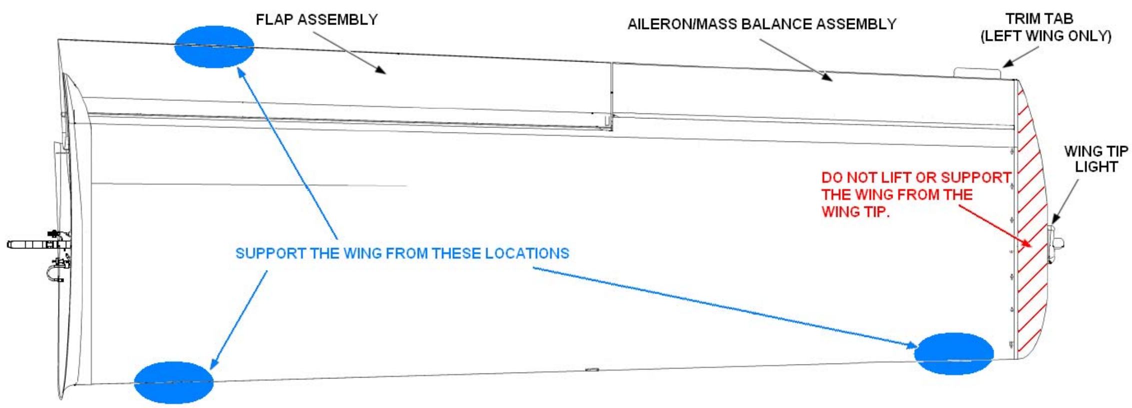

Figure 57-5 Wing Handling Positions

Section 10-03 Manual Operated Wing Pins

This section details the procedures to remove and install the wings with the manually operated wing pins.

CAUTION

The wing pins must move fluidly in and out of the wing lock mechanism. There shall be no grinding, scraping or interference to the movement of the pin. If there is any interference or grinding sensed during pin movement, do not cycle the pin in and out of the mechanism. Retract the pin, remove the wing, and investigate to find the source of the interference.

To prevent wing pin gouging, the wing pins require inspection and cleaning every time the wings are removed from the airplane. Also, inspect the pins for any damage to the chrome plating on the surface of the wing pins.

MANUAL WING REMOVAL

Perform this procedure to remove wings that have the manually operated wing pins.

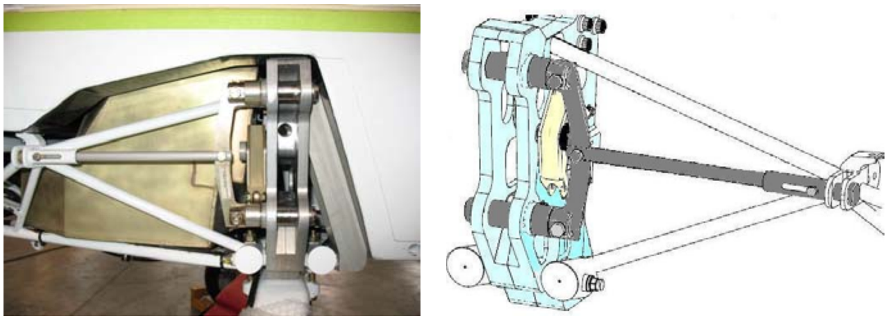

Figure 57-6 Mechanical Wing Lock Mechanism

Required Equipment:

- Padded wing stand(s).
- Extractor Wing Pin Tool Kit (135A-02-011) which included 135A-02-551 and 135A-02-552, see Figure 57-7. This is only used for mechanical removal of wing.
- Air ratchet wrench or ratcheting wrench

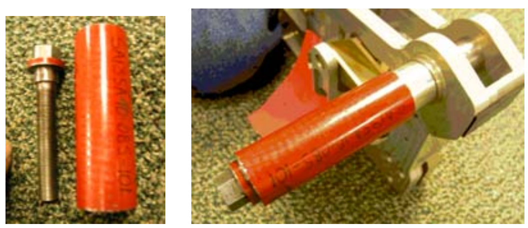

Figure 57-7 Extractor Wing Pin Tool Kit

1. Set parking brake, or chock the wheels.
2. Turn power on. Fully retract flaps to the up position. Turn power off.
3. Remove the belly panel. See Chapter 53 - *Fuselage*.
4. Support opposite wing on padded support.

CAUTION

Ensure support is placed below wing internal structure, rather than below unsupported wing skin areas between ribs.

5. Disconnect electrical connector in fuselage center section for position/anti-collision light(s) located in the fuselage near the wing root.
6. If removing port wing, disconnect pneumatic connections for pitot and static sources. Remove black retaining ring, push down on locking clip, and pull tubing out located in the fuselage near the wing root.

7. User will not be able to see internal mechanism; mechanic must reach into the underbody of aircraft. Remove wing locking: aft pin, key rod, plate, and hardware.
8. Remove both sleeve wing pin.
9. Remove lower wing pin by putting extractor tool over wing pin and unscrewing extractor pin from sleeve.
10. Use extractor tool to remove upper wing pin. The use of an air-ratchet wrench is recommended to remove upper wing pin from wing box.

CAUTION

Wing removal requires three people; one extra person is utilized for flap control. Each wing panel is heavy.

11. Wing is held in place by the spar tang. The wing will not drop once the wing pins are removed. The recommended three people must separate the wing from the wing box. Station one person at wing leading edge at wing root, one at wing trailing edge at wing root, and one person at leading edge near the wing tip.

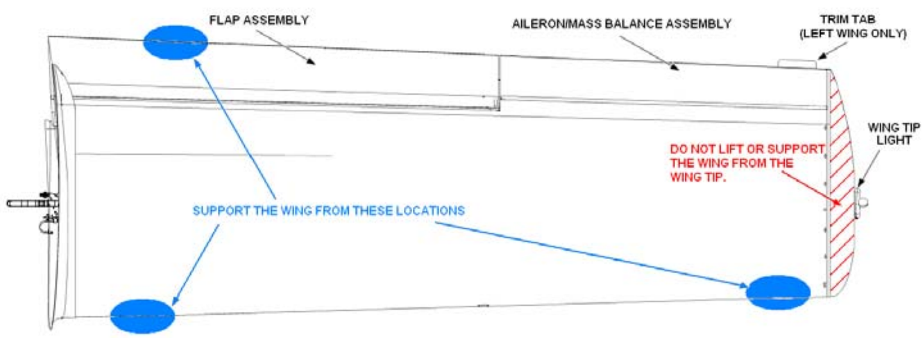

Figure 57-8 Wing Handling Positions

12. Pull wing away from fuselage and place on padded supports for further maintenance as required.
13. Inspect wing pins. If corrosion is visible contact Liberty Aerospace Customer Service for instruction or correction.

CAUTION

Person at wing root trailing edge must be ready to support flap as it disengages from flap actuator cross tube in fuselage during wing removal. It is recommended to temporarily tape flaps before separating wing from wing box to prevent them from swinging down and damaging the wing.

MANUAL WING INSTALLATION

Perform this procedure to install wings that have the manually operated wing pins.

CAUTION

If both wings have been removed, support the first wing to be installed on padded support.

1. Turn power on. Fully retract flaps to the up position. Turn power off.
2. Inspect the wing pins for dirt, debris or other contaminants. Also, inspect for any gouges in the pins or a flaking of the chrome-plated surfaces. Clean all surfaces of the pins and associated mating surface.
3. Prior to reassembly, remove any corrosion or foreign particles from spar tunnel. Apply LPS 3 inhibitor activation to appropriate areas and the wing pins.

CAUTION

Wing removal requires three people; one extra person is utilized for flap control. Each wing panel is heavy.

4. Station one person at wing leading edge at wing root, one at wing trailing edge at wing root, and one person at leading edge near the wing tip. Place main spar fittings just inside fuselage wing root fitting. Support flap so that flap drive spigot in flap root rib engages bearing in flap actuator cross tube in fuselage center section. Ensure that the flap is properly engaged. Slide wing fully into fuselage.

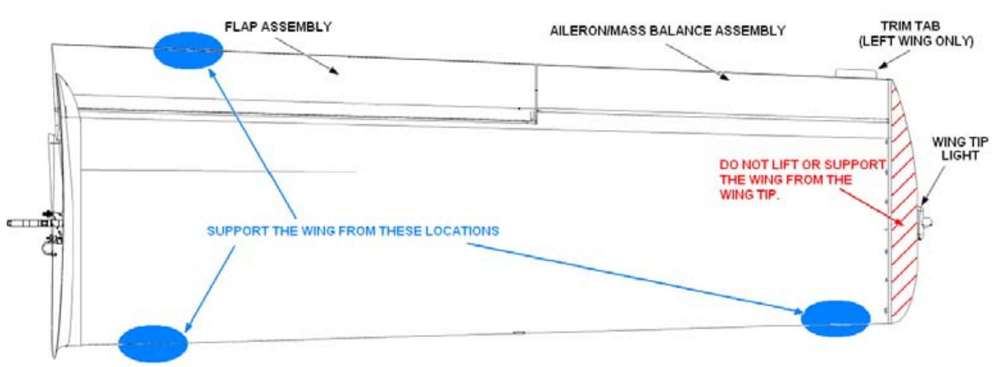

Figure 57-9 Wing Handling Positions

5. Ensure that all three pins (two in main spar, one in aft spar) are fully seated. If necessary, apply slight upward pressure to align the wing to the wing box.

6. When wing pins are fully seated ensure that the chamfered surface and approximately 1/16” of the shank are visible beyond the pin housing.
7. Reconnect electrical connector for position/anti-collision lights. If installing port wing, reconnect pneumatic connections for pitot and static sources.
8. Perform functional check of ailerons, flaps, and lights.
9. Perform Pitot/Static checks, reference: Chapter 34 – *Navigation and Pitot/Static*.
10. Re-rig in accordance with Chapter 27 – *Control Surfaces*.
11. Reinstall the belly fairing, reference: Chapter 53 - *Fuselage*.
12. Perform lateral stability flight test.

Section 10-04 Electrical Operated Wing Pins

This section has the procedures to remove wing that have the electrically operated wing pins.

Section 10-05 Electrical Wing Procedures

Figure 57-10 shows the assembled unit. The pin assembly on each side of the fuselage center section chassis secures the wing fittings to their counterparts in the fuselage.

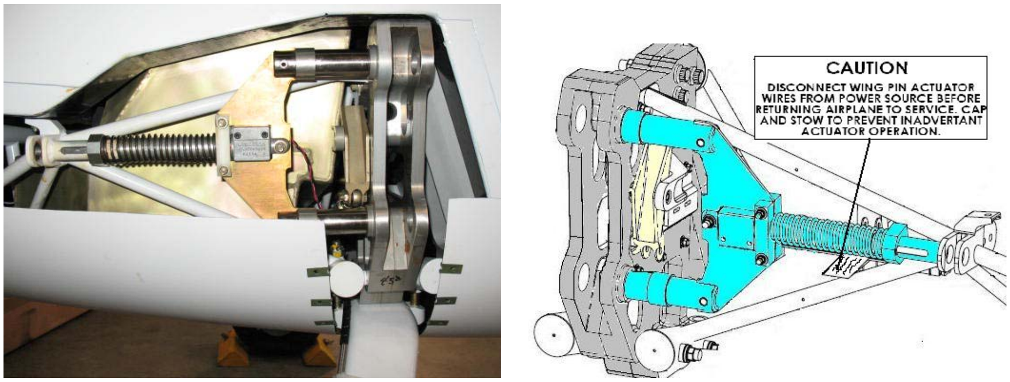

Figure 57-10 Electrical Wing Lock Mechanism

WARNING

If the airplane has an electrically operated wing pin system, to avoid inadvertent in-flight activation of the system, do not connect the wing pin system to the airplane electrical system. Instead, connectors are provided for temporary connection to a ground power source when necessary for wing removal and/or installation.

CAUTION

Wing removal requires three people; one extra person is utilized for flap control. Each wing panel is heavy.

CAUTION

The wing pins must move fluidly in and out of the wing lock mechanism. There shall be no grinding, scraping or interference to the movement of the pin. If there is any interference or grinding sensed during pin movement, do not cycle the pin in and out of the mechanism. Retract the pin, remove the wing, and investigate to find the source of the interference. To prevent wing pin gouging, the wing pins require inspection and cleaning every time the wings are removed from the airplane. Also, inspect the pins for any damage to the chrome plating on the surface of the wing pins.

WING REMOVAL (WITH ELECTRICALLY OPERATED WING PINS)

Perform the following procedure to remove wings that have the electrically operated wing pins.

Required Equipment

- Padded wing stand(s).
- Source of 12vdc electrical power (min. 10 amp capacity).

1. Set parking brake, or chock the wheels.
2. Fully extend flaps.
3. Remove belly panel see Chapter 53 - *Fuselage*.
4. Support opposite wing on padded support.

CAUTION

Ensure support is placed below wing internal structure, rather than below unsupported wing skin areas between ribs.

5. On electrically controlled pin assemblies, disconnect electrical connector in fuselage center section for position/anti-collision light(s). If removing port wing, disconnect pneumatic connections for pitot and static sources. Remove black retaining ring, push down on locking clip, and pull tubing out.
6. Ensure switch on power source is in OFF position; connect cable to 12vdc power source. Connect wing pin actuator cable to actuator connection in fuselage center section.
7. Station a person at the leading edge near the wing tip and place switch on cable to RETRACT position. Wing pins will retract. If necessary, apply slight upward pressure to reduce load on wing pin actuator.

NOTE

Listen for actuator operation; actuator motor will continue to operate (slip clutch) even after pins are fully retracted.

8. Return actuator cable switch to OFF position and disconnect. Confirm visually that all three wing pins (two in main spar, one in aft spar) are fully retracted.

CAUTION

Wing removal requires three people; one extra person is utilized for flap control. Each wing panel is heavy.

9. Station one person at wing leading edge at wing root, one at wing trailing edge at wing root, and one person at leading edge near the wing tip.

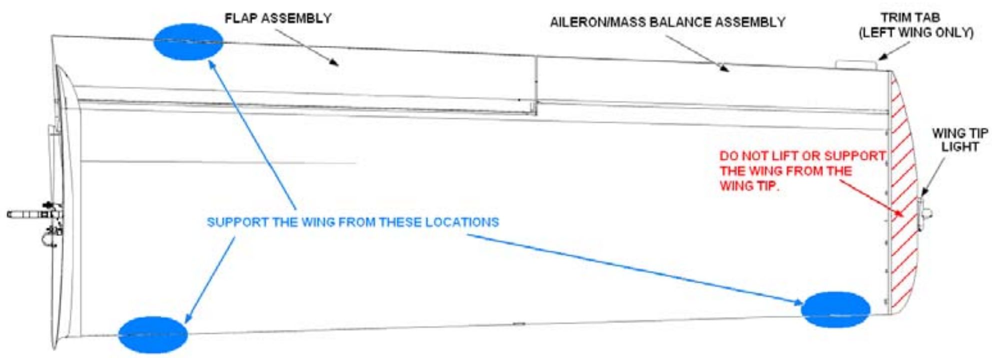

Figure 57-11 Wing Handling Positions

10. Pull wing away from fuselage and place on padded supports for further maintenance as required.

11. Inspect the wing pins. If corrosion is visible contact Liberty Aerospace Customer Service for instruction or correction.

CAUTION

Person at wing root trailing edge must be ready to support flap as it disengages from flap actuator cross tube in fuselage during wing removal. It is recommended to temporarily tape flaps before separating wing from wing box.

CAUTION

The wing pins must move fluidly in and out of the wing lock mechanism. There shall be no grinding, scraping or interference to the movement of the pin. If there is any interference or grinding sensed during pin movement, do not cycle the pin in and out of the mechanism. Retract the pin, remove the wing, and investigate to find the source of the interference.

To prevent wing pin gouging, the wing pins require inspection and cleaning every time the wings are removed from the airplane. Also, inspect the pins for any damage to the chrome plating on the surface of the wing pins

WING INSTALLATION (ELECTRICALLY OPERATED WING PINS)

Perform this procedure to install the wings with the electrically operated Wing pins. It is necessary to support the first wing installed with a padded support while installing the second wing.

CAUTION

The wing pins must move fluidly in and out of the wing lock mechanism. There shall be no grinding, scraping or interference to the movement of the pin. If there is any interference or grinding sensed during pin movement, do not cycle the pin in and out of the mechanism. Retract the pin, remove the wing, and investigate to find the source of the interference.

To prevent wing pin gouging, the wing pins require inspection and cleaning every time the wings are removed from the airplane. Also, inspect the pins for any damage to the chrome plating on the surface of the wing pins.

1. Turn power on.
2. Fully retract flaps to full up position.
3. Turn power off.
4. Inspect the wing pins for dirt, debris or other contaminants. Also, inspect for any gouges in the pins or a flaking of the chrome-plated surfaces. Clean all surfaces of the pins and associated mating surface.
5. Ensure switch on cable is in OFF connection; connect cable to 12vdc power source. Connect the cable for the wing pin actuator.
6. If required, place cable switch in RETRACT position and confirm visually that all three pins are fully retracted.
7. Position the switch to the off position.
8. Prior to reassembly, remove any corrosion or foreign particles from springs, motor areas, and spar tunnel. Apply LPS 3 to appropriate areas and the wing pins. Verify prior to flight that motor-armatures and wiring is not affected by application of LPS 3.

CAUTION

Wing removal requires three people; one extra person is utilized for flap control. Each wing panel is heavy.

9. Station one person at wing root leading edge, one person at wing root (flap) trailing edge, one person at wing tip. Place main spar fittings just inside fuselage wing root fitting. Support flap so that flap drive spigot in flap root rib engages bearing in flap actuator cross tube in fuselage center section. Ensure that the flap is properly engaged. Slide wing fully into fuselage.
10. Place cable switch in EXTEND position to engage wing pins. Observe successful operation of wing pin actuator from below airplane. If necessary, apply slight upward pressure to wing tip to reduce load on wing pin actuator.

NOTE

Listen for actuator operation; actuator motor will continue to operate.

11. Return actuator cable switch to OFF position; disconnect cable from actuator connector and 12vdc power source.
12. When wing pins are fully seated, ensure that the chamfered surface and approximately 1/16” of the shank are visible beyond the pin housing.
13. Reconnect electrical connector for position/anti-collision lights. If installing left wing, reconnect pneumatic connections for pitot and static sources.
14. Perform functional check of ailerons, flaps, and lights.
15. Perform Pitot/Static checks, reference: Chapter 34 – *Navigation and Pitot/Static*.
16. Re-rig in accordance with Chapter 27 – *Control Surfaces*.
17. Reinstall belly panel see Chapter 53 - *Fuselage*.
18. Perform lateral stability flight test

CAUTION

Person at wing root trailing edge must be ready to support flap as it disengages from flap actuator cross tube in fuselage during wing removal. It is recommended to temporarily tape flaps before separating wing from wing box to prevent them from swinging down and damaging the wing.

Section 10-06 *Wing Attachment Fittings Inspection*

This section details the inspection of the fittings used to attach the wings to the airplane. Inspect the attachment fittings of the wing each time the wing is removed from the airplane for any reason.

The following items require a visual inspection, as shown in Figure 57-12 and Figure 57-13 :

- Exterior surfaces of the forward wing tang (wing)
- Interior and exterior surfaces of the forward wing box (space frame)
- Exterior surfaces of the aft wing tang (wing)
- Interior and exterior surfaces of the aft wing tang attachment to the space frame (space frame)
- The wing pins (space frame)
- Flap spigot (wing – flap)
- Flap actuator tube (space frame)
- Aileron quick disconnect (both)

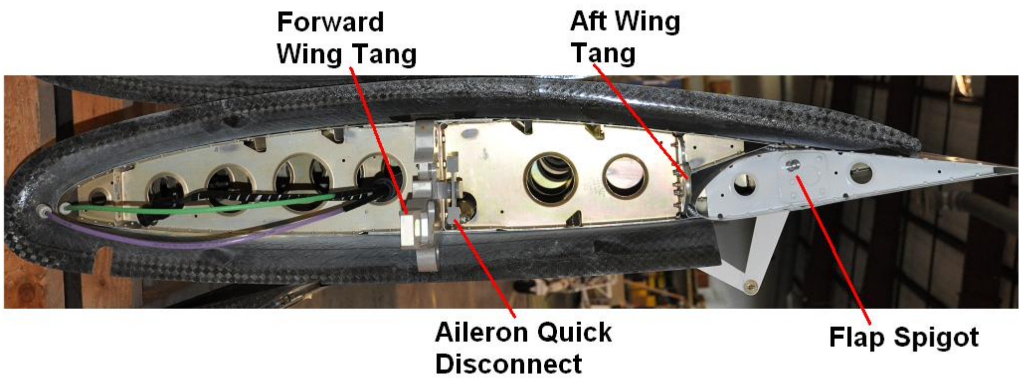

Figure 57-12 Wing Inspection Points

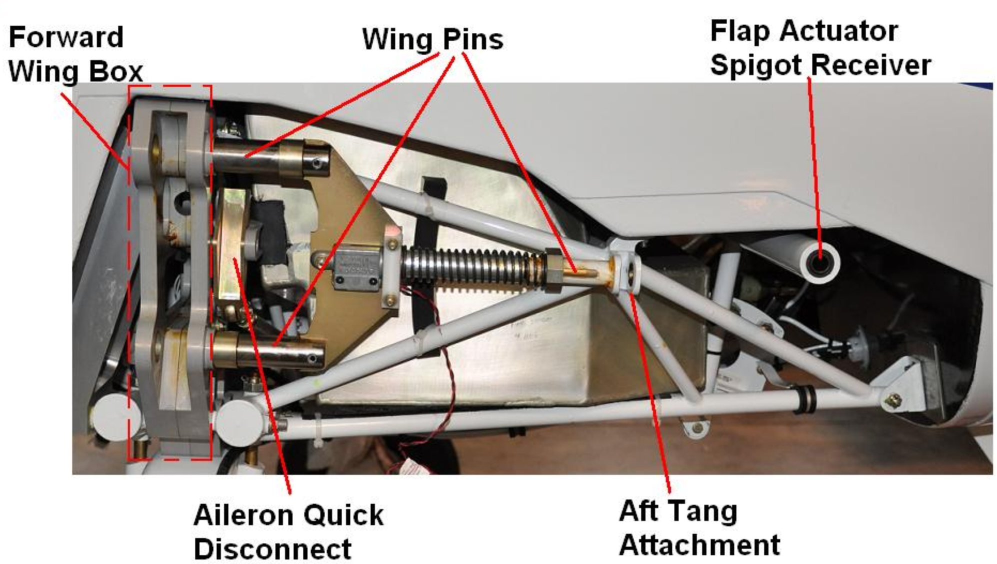

Figure 57-13 Fuselage Inspection Points

Items	Inspect For	Resolution
Forward Wing Tang, Forward Wing Box, Flap Spigot, and Flap Spigot Receiver	Corrosion or rust	Light surface rust – clean then lubricate See Chapter 12 – *Servicing* All else, Contact Liberty Aerospace.
	Metal Shavings and Flakes	Contact Liberty Aerospace.
	Gouges or grooves in the metal	Contact Liberty Aerospace.
	Metal fatigue (Deformed metal and cracks)	Contact Liberty Aerospace.
Pins	Corrosion or rust	Light surface rust – clean then lubricate See Chapter 12 – *Servicing* All else, Contact Liberty Aerospace.
	Metal Shavings and Flakes	Contact Liberty Aerospace.
	Gouges or grooves in the metal	Contact Liberty Aerospace.
	Metal fatigue (Deformed metal and cracks)	Contact Liberty Aerospace.
	Grinding noise while moving	Clean the pins. Lubricate the pins see Chapter 12 – *Servicing*
Aft Wing Tang and Aft Wing Tang Attachment	Corrosion or rust	Light surface rust – clean then lubricate See Chapter 12 – *Servicing* All else, Contact Liberty Aerospace.
	Metal Shavings and Flakes	Contact Liberty Aerospace.
	Gouges or grooves in the metal	Contact Liberty Aerospace.
	Metal fatigue (Deformed metal and cracks)	Contact Liberty Aerospace.
	Elongation of Aft Wing Tang	Contact Liberty Aerospace.
Aileron Quick Disconnect	Corrosion or rust	Light surface rust – clean then lubricate See Chapter 12 – *Servicing* All else, Contact Liberty Aerospace.
	Metal Shavings and Flakes	Contact Liberty Aerospace.
	Gouges or grooves in the metal	Contact Liberty Aerospace.
	Metal fatigue (Deformed metal and cracks)	Contact Liberty Aerospace.
	A gap of greater than 0.006 in between the mating surfaces of the aileron quick disconnect	See Chapter – 27 *Flight Controls*

Table 57-1 Inspection and Resolution Chart

Section 10-07 Wing Limit Load Condition

In addition, the inspection for the following limit load conditions are recommended while performing 100-hour Inspection. The following are examples of result conditions due to wing structures experiencing a load greater than limit load. If any of these conditions are observed structural replacement is mandatory. Contact Liberty Aerospace Customer Service for assistance.

Figure 57-14 Outer And Inner Panels Buckling On The Topside Skin Of Wing (Wing is Mounted With The Topside Down)

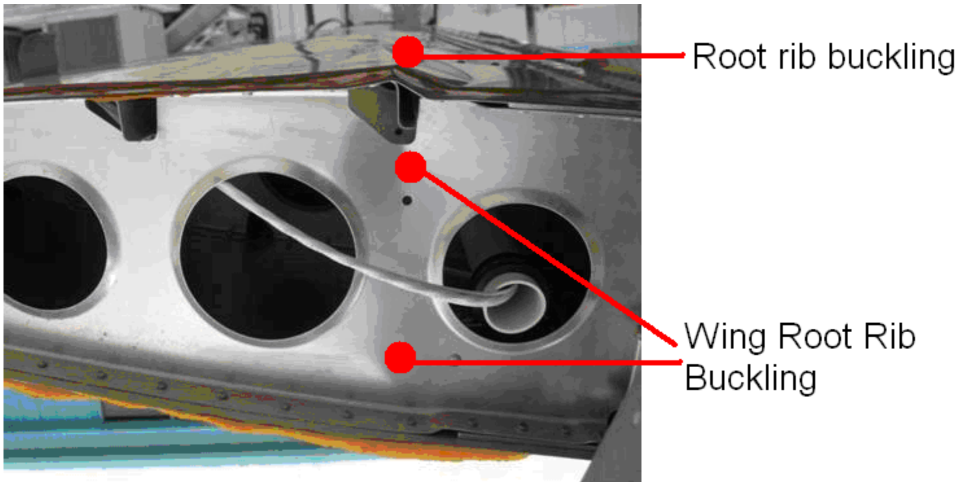

Figure 57-15 Buckling on Root Rib Aft of Main Spar and Wing Skin

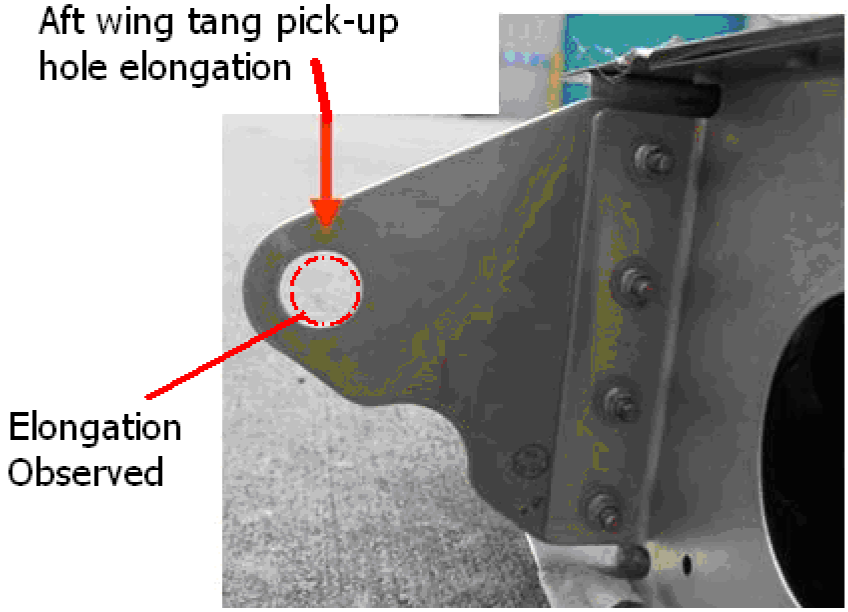

Figure 57-16 Aft Wing Pick-Up Elongation

Section 57-11 Wing Spars, Ribs, and Stringers

The wing skeleton includes a built-up aluminum main forward & aft spar, forward spar (leading edge spar), and rear spar flap shroud.

Eleven ribs (including root and tip ribs) are secured to the spars. There are eight ribs between the root and the junction of the flap and the aileron, with three further ribs between this point and the outboard end of the aileron. The ribs are non-continuous, i.e., each rib is comprised of a leading edge rib forward of the leading edge spar, a section between the leading edge spar and the main spar, and a section between the main spar and the rear spar.

Six stringers, three each between the leading edge spar and the main spar and between the main spar and the rear spar, extend span-wise to strengthen the inboard wing area. The stringers nearest the main spar extend from the root to the eighth rib; those forward and aft of these are progressively shorter.

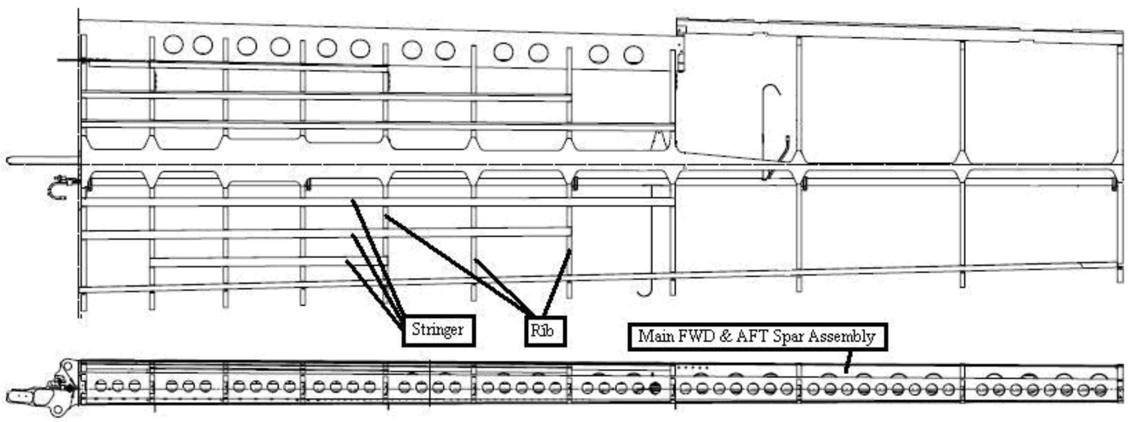

Figure 57-17 Wing Skeleton

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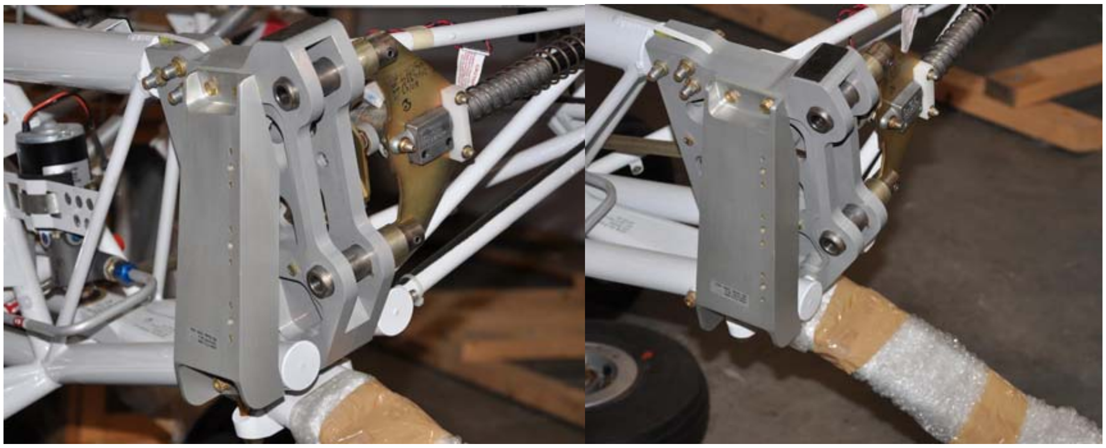
Section 57-14 Wing Attachment

This section describes the wing attachment, which includes the wing box, wing pins and associate structures. The wing box is the attachment point on the fuselage for the wing to connect and transfer its forces to the fuselage. The aft wing pin weldment completes the attachment of the wing to the fuselage.



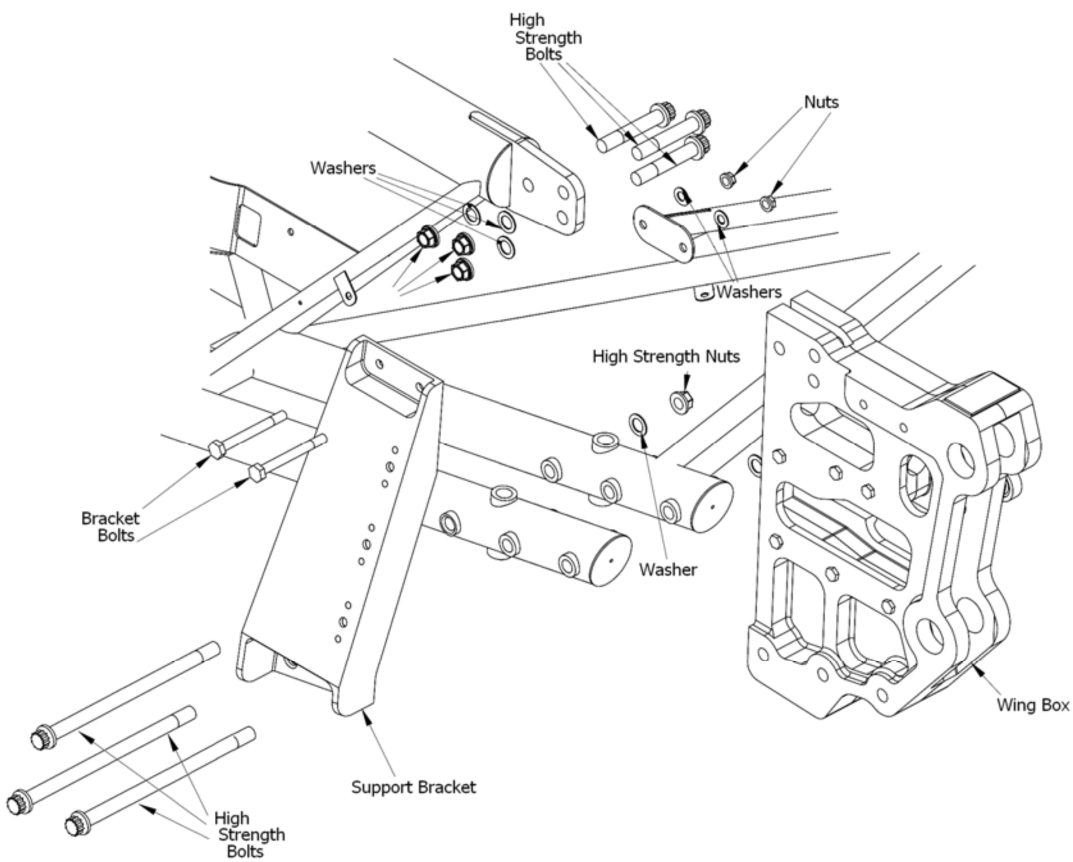


Figure 57-18 Wing Box Explodes View

Section 14-01 Wing Box Procedures

This section will contain the information to remove, install, and inspect the wing box. However, Liberty Aerospace, Inc. has not published field procedures to remove, install, and inspect the wing box. If the wing box requires replacing, contact Liberty Aerospace, Inc. Customer Service.

Section 14-02 Aft Wing Clevis Pin Weldment

Some Liberty XL-2 airplane's aft clevis components may have been supplied with lug bores slightly out of tolerance, leading to wear of the components later. Liberty Aerospace, Inc. has addressed this issue by making production changes to this weldment. For all airplanes in the field, there are increased inspections and life limits for this area, as defined in Chapter 4 – Airworthiness Limitations and Chapter 5 – *Time Limits and Maintenance Inspections*. Procedures in this section are to address this issue on airplanes found to be out of tolerances.

Section 14-03 Aft Wing Clevis Pin Weldment Procedures

This section contains the procedures to address wear from the slight increase in tolerance of the aft wing clevis pin weldment. If during the 100-hour/annual inspection an out-of-tolerance condition exists, please contact Liberty Aerospace, Inc. Customer Service with the information on the airplane and the amount of out of tolerance observed.

AFT WING CLEVIS PIN WELDMENT REPAIR

Perform this procedure to repair the aft wing clevis pin weldment.

1. Remove the wings in accordance with the Wing Removal (With Electrically Operated Wing Pins) procedures on page 17 of this chapter.
2. Using a calibrated micrometer, measure and record the diameters vertically, horizontally and at 45° to both chassis lug holes. See Figure 57-19.

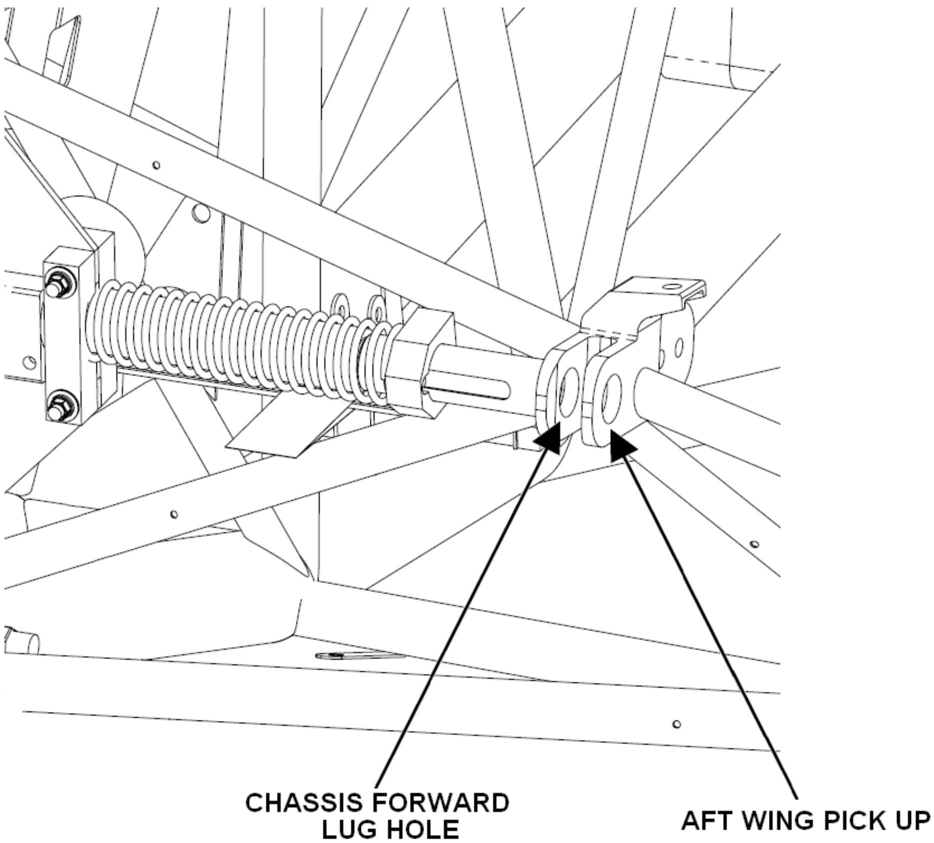


Figure 57-19 Measurement of the Aft Wing Clevis Pin Weldment

3. If the measurement is in tolerance (0.650 – 0.000 to +0.002 inches) and there is no observable free play, that completes this procedure. If the measurement is outside this tolerance range and/or free play is observed, continue with this procedure.
4. Activate the ball drive actuator to move the pins out.
5. While someone holds the spring and spring support back, use a drift punch to drive the actuator bar out of the aft pin. See Figure 57-20.
6. After the actuator bar is out of the pin, carefully release the tension on the spring.

7. Remove the three bolts, nuts and washers securing the ball drive actuator to the forward pin assembly.
8. Actuate the ball drive actuator to retract the aft pin and to disconnect the ball drive from the forward pin assembly.
9. Remove the ball drive actuator.
10. Remove the aft pin from the ball drive actuator and substitute the oversize aft pin, P/N 135A-04-793 assembly with Loctite LCT 270.

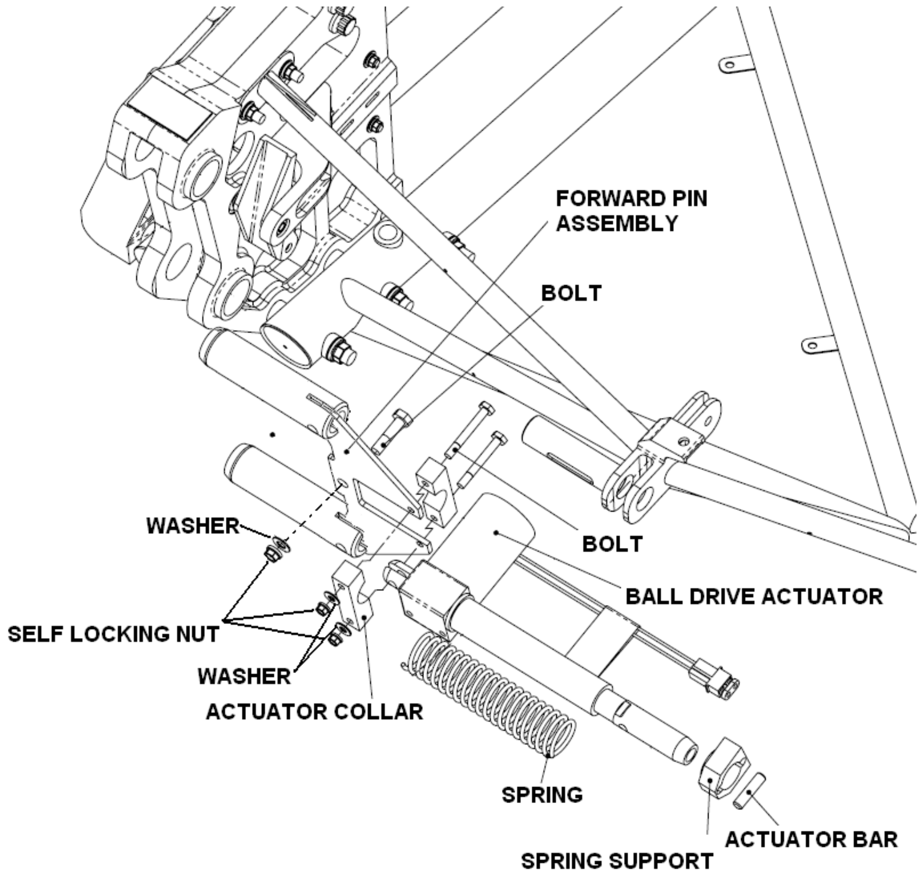


Figure 57-20 Drive Actuator Assembly

11. Ensure the fuselage is supported
12. Ream the aft wing pick hole and the chassis aft lug hole, leading from the forward face of the clevis weldment. The final diameter of both the chassis lugholes shall be 0.6562in ±0.0005 after reaming.
13. Visually inspect all surfaces for cracks or created flaws with a magnifying glass. Ensure that components are clean, and free from burrs, turnings, chips, or filings, and are lubricated in accordance with Chapter 12 – *Servicing* prior to installation.

14. Install the spring and spring support. Install the ball drive actuator in to the aft weldment. Protect from corrosion using Corrosion X or ACF 50 aerosol spray.
15. Compress the spring and insert the actuator bar.
16. Carefully guide the ball drive actuator in to the forward pin assembly.
17. Install the actuator collar and two bolts washers and self-locking nuts. If there is place for a split pin, install a new split pin.
18. Activate the ball drive to move the pins in to the weldment.
19. Verify by inspection that all lug faces make contact with the pin without gaps or free-play. With the pins fully extended in to the weldment, inspect wing pins for full seating (1/16-inch of exposed shaft beyond the chamfered surface.
20. Retract the wing pins.
21. Install the wings in accordance with the Wing Installation (Electrically Operated Wing Pins) procedures on page 19 of this chapter.
22. Record the alteration in the aircraft log book.

Section 57-20 Wing Skin and Access Panels

The wing skins are wrapped around the internal structure and are riveted to the ribs, stringers, and spars.

Two round access panels are provided on the lower surface of each wing, to provide access to the aileron bell-crank and pushrods for maintenance. The access panels are secured by three screws each to nut-plates attached to doublers around the access panel openings.

Openings and internal structure in the lower wing skin forward of the ailerons accommodate movement of the aileron mass balance assemblies.

The pitot/static tube is attached to the lower surface of the port wing. For the procedures to service the pitot/static blade and associated tubing, see Chapter 34 – *Navigation and Pitot/Static*.

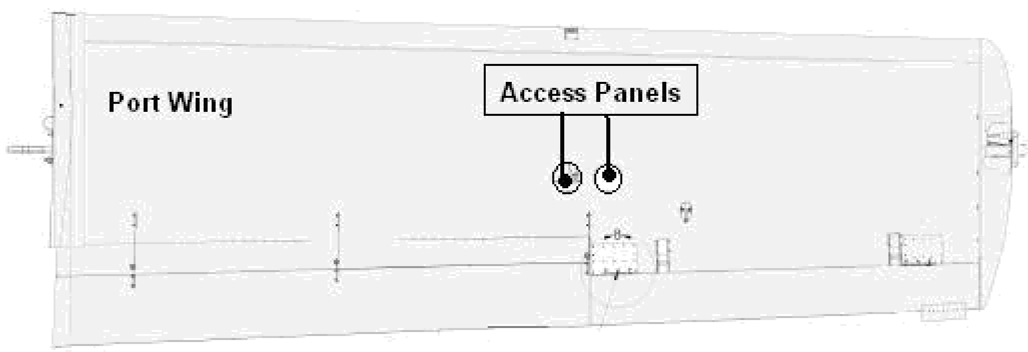


Figure 57-21 Wing Access Panel

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Section 57-30 Wing Tip

The composite wing tip is secured to the outermost wing rib by screws and nut-plates. The composite wing tip is removable. This section contains the procedures to remove and install the wing tips.

WING TIP REMOVAL

Perform this procedure to remove the wing tips.

23. Ensure all electrical switches are off and that at least 30 minutes have elapsed since most recent operation of anti-collision lights.
24. Remove position and anti-collision light lenses, bulbs, and flash tube (see Chapter 33 – *Lights*) for procedure.
25. Remove screws securing position/anti-collision assembly light to internal bracket inside wing tip.
26. Remove screws securing wing tip to tip rib. Remove and support wing tip to avoid damage to position/anti-collision light wiring harness.
27. Disconnect position/anti-collision light electrical connector and ground strap. Remove wing tip.

WING TIP INSTALLATION

Perform the following procedure to install the wing tips.

CAUTION

Ensure position/anti-collision light wiring harness is not pinched between bracket and inside of wing tip.

1. Support wing tip near end of wing and connect position/anti-collision light electrical connector.
2. Place wing tip in position and secure screws.
3. Replace screws securing position/anti-collision light assembly to internal bracket.
4. Replace position/anti-collision light bulbs, flash tube, and lenses and perform functional test (see Chapter 33 - *Lights*).

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Section 57-60 Flight Control Surfaces

This section describes the structure of flight control surfaces which are parts of the wings only. For the complete information on the flight control surfaces, refer to Chapter 27 – Flight Controls.

Section 60-01 Ailerons

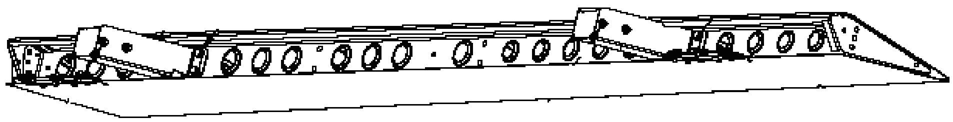

Figure 57-22 Ailerons

Each aileron comprises a front spar, eight ribs, a one-piece skin, two hinges, and two mass balance assemblies.

The mass balances extend forward of the aileron hinge line and serve to prevent aerodynamic flutter. If the aileron is repaired or refinished, its balance must be checked and adjusted to remain within limits. See Chapter 51 – *Standard Practices Structure*.

Hardware fastened to the aileron root rib (most inboard rib) secures the bearing end of the aileron pushrod and transmits pilot control deflections to the aileron.

Section 60-02 Flaps

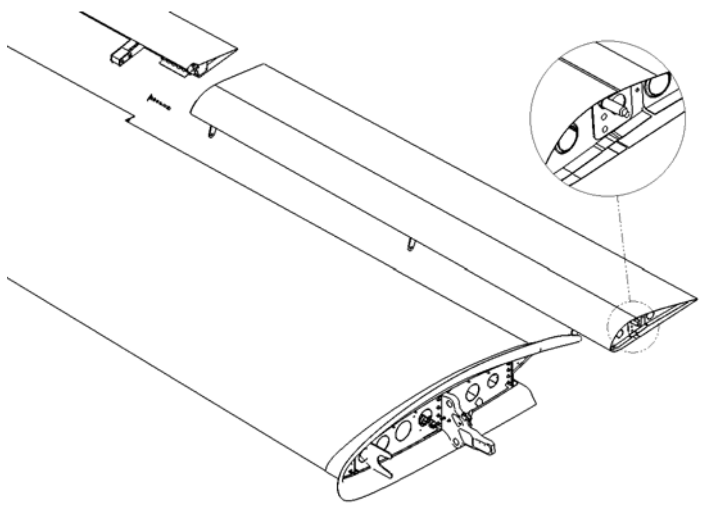

Figure 57-23 Flaps

Each flap comprises a spar, six ribs, and two skins. One skin forms the flap's leading edge. The skin is then wrapped from the spar around the leading edge of the ribs. The other skin forms the remainder of the flap. It is folded at the flap trailing edge, and is secured at top and bottom to the flap spar. Flap hinge arms are secured to the second, fourth, and six ribs.

A flap drive spigot protrudes inboard from the flap root rib and engages a bearing in the flap actuator cross tube in the fuselage center section. This bearing allows sufficient off-axis alignment to accommodate changes in geometry throughout the flap's operating range.

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Section 57-61 Wing Root Fairing

The composite wing root fairing, shown in Figure 57-24, adds strength to the overall wing structure. It is essential that the wing root fairing remains properly attached to the wing skin. This ensures that the wing skin is structurally reinforced.

Figure 57-24 Properly Secured Wing Root Fairing Attached to Wing Skin

If the wing root fairing is not installed or is installed incorrectly, internal structures of the wing can buckle and cause further damage to the wing. See Figure 57-25 for an example of buckling that can occur if the wing root fairing is not in place.

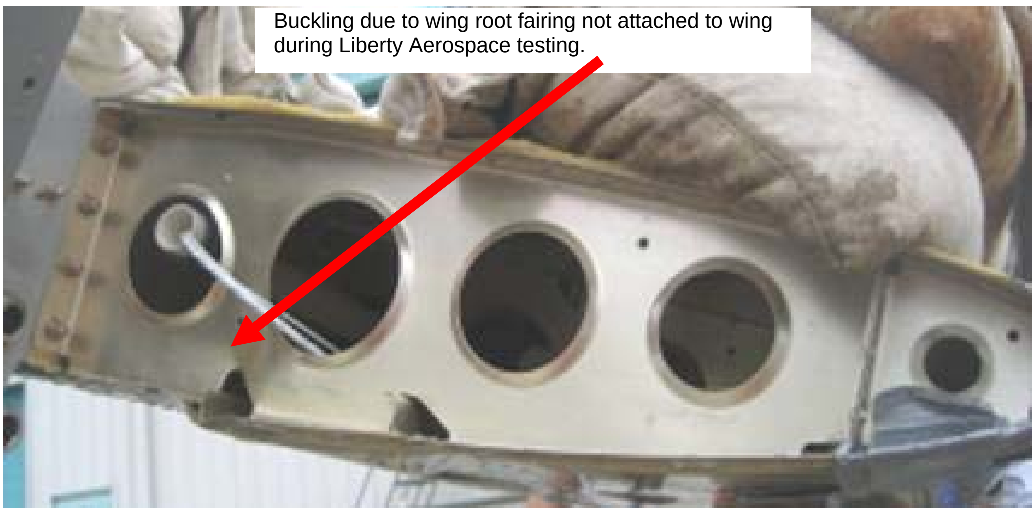

Figure 57-25 Buckling Observed During Strength Test

A Wing Root Fairing Inspection is required during the 100-hour Inspection. Remove wing from fuselage and inspect the wing root fairing and wing skin attachment for any Epibond® voids or any cracks.

Section 61-01 Wing Root Fairing Procedures

This section contains the procedures to remove and install the anti-chaffing tape that attaches to the wing root fairing.

ANTI CHAFING TAPE REMOVAL

Perform this procedure to remove the anti chafing tape from the wing root fairing.

1. Remove the wing(s) from the airplane in accordance with the procedures in Section 57-10 on page 7 of this chapter.
2. Remove any remaining tape from the wing root fairing.
3. Solvent wipe with acetone using clean, lint free rags.
4. Dry solvent from surface using clean, lint free rags. Do not allow the acetone to air dry on the surface. The use of two hands, one with a solvent dampened rag and one following with a dry rag, is recommended.
5. Repeat above operations until the drying rag shows no signs of contamination.
6. Abrade surface using wet/dry sandpaper.

This completes the Anti Chafing Tape Removal procedure.

ANTI CHAFING TAPE INSTALLATION

Perform this procedure to remove the anti chafing tape from the wing root fairing.

1. Solvent wipe with acetone using clean, lint free rags.
2. Dry solvent from surface using clean, lint free rags. Do not allow the acetone to air dry on the surface. The use of two hands, one with a solvent dampened rag and one following with a dry rag, is recommended.
3. Repeat above operations until the drying rag shows no signs of contamination.
4. Abrade surface using wet/dry sandpaper.
5. Apply a single layer of the anti-chaffing tape along the edge of the wing root fairing that interfaces with the fuselage
6. Install the wing(s) on to the airplane in accordance with the procedures in Section 57-10 on page 7 of this chapter.

This completes the Anti Chafing Tape Installation procedure.

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CHAPTER 61

PROPELLERS

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Table of Contents

SECTION 61-00 PROPELLERS ON THE LIBERTY XL-2 AIRPLANE 5

SECTION 00-01 MT PROPELLER 5
SECTION 00-02 SENSENICH PROPELLER 5
SECTION 00-03 PROPELLER PROCEDURES 5
PROPELLER REMOVAL 6
PROPELLER INSTALLATION (BALANCING OR CLOCKING) 7
PROPELLER TRACKING INSPECTION 10
PROPELLER AND BOLT TORQUE INSPECTION 11
SECTION 00-04 TROUBLESHOOTING CHART 12

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Section 61-00 Propellers on the Liberty XL-2 Airplane

The currently approved (and supplied on new XL-2 airplanes) and preferred propeller is the MT Propeller, Model MT175R127-2Ca. Some airplanes can have the Sensenich Wood Propeller, Model W69EK7-63G. Although both propellers are approved, the preferred propeller is the MT Propeller. No other propeller may be installed without contacting Liberty Aerospace, Inc. for approval through the FAA.

Section 00-01 MT Propeller

All descriptions and maintenance activities for the MT propeller are given in the most recent FAA approved MT Propeller Manual:

ATA-61-01-12 (E112)
MT-WOOD COMPOSITE
FIXED PITCH PROPELLERS
MT-Propeller USA, Inc.
1180 Airport Terminal Dr.
DeLand, FL 32724
http://www.mt-propeller.com/pdf/manuals/e-112.pdf

Section 00-02 Sensenich Propeller

All descriptions and maintenance activities for the Sensenich propeller are given in the most recent FAA approved Sensenich Propeller Manual:

Installation, Operation, and Maintenance
DOC# WOOD-CF-REV-A DOC 5-20-04
Sensenich Wood Propeller Company
2008 Wood Court
Plant City, FL 3356
http://www.sensenichprop.com/sen.html/aircraft_cet/install/cf-a.pdf

Section 00-03 Propeller Procedures

This section contains removal, installation, tracking inspection, and torque inspection procedures that are applicable to both propellers.

PROPELLER REMOVAL

Perform this procedure to remove the propeller from the airplane.

NOTE

Take note of the starting orientation of blade #1 of the propeller in relationship to the engine hub. If the propeller is not installed back onto the airplane in the same orientation, it could cause a vibration in the airplane.

1. Ensure all electrical switches are OFF and park the airplane in accordance with Chapter 10 – *Parking and Mooring*.
2. Remove the upper and lower engine cowls in accordance with Chapter 53 – *Fuselage*.
3. Remove the propeller spinner.
4. Use a permanent marker, to place an alignment mark on the bulkhead for the propeller spinner and the engine hub.
5. Remove the propeller, and the bulkhead for the propeller spinner from the engine hub.

PROPELLER INSTALLATION (BALANCING OR CLOCKING)

Perform this procedure to install and balance the propeller.

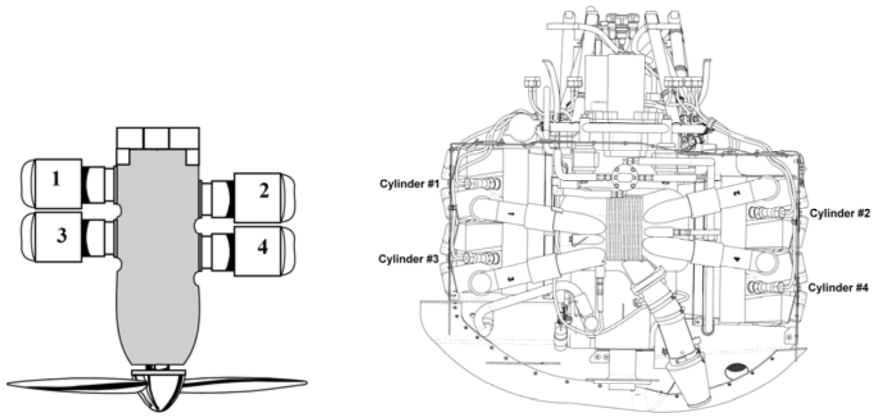

Figure 61-1 TCM IO-240-B Cylinder Assignment

1. Install the propeller and bulkhead assembly on the propeller extension horizontally, with the number one (#1) propeller blade at the 9 o'clock position as viewed from the front, see Figure 61-2.

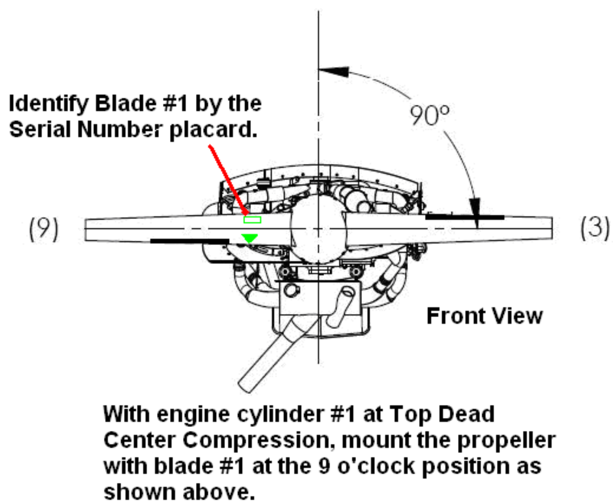

Figure 61-2 Initial MT Propeller Position

NOTE

The propeller assembly is statically balanced. The #1 propeller blade must be aligned with the #1 stamping on the flange of the spinner bulkhead. The #1 blade is marked by a number one (1) stamped on the back of the propeller hub near the #1 blade; also, the #1 propeller blade bears the MT serial number placard.

2. If installing an MT Propeller, torque propeller stop nuts (6) in accordance with Section 4.5 and 4.6.1 of MT Propeller Maintenance Manual E-112 Issue 14 (03/19/2009) or later approved issue. If installing a Sensenich propeller, torque the stop nuts (6) in accordance the Sensenich Propeller Manual, DOC# WOOD-CF-REV-A DOC 5-20-04.
3. Install the spinner. When installing the spinner, match the number one (1) marked on the inside of the spinner with the #1 propeller blade, and the number one (1) stamped in the spinner bulkhead flange.
4. Perform flight test to evaluate the propeller-induced vibration.
5. If the vibration has been reduced to an acceptable level, this completes the clocking procedure. Check the torque on the propeller nuts; re-torque as required.
6. If the vibration is equal to or greater than previously recorded, remove the spinner, the propeller and the spinner bulkhead and re-install the entire assembly 180° from previous. In other words, with the #1 blade at the 3 o'clock position as viewed from the front, see Figure 61-3.

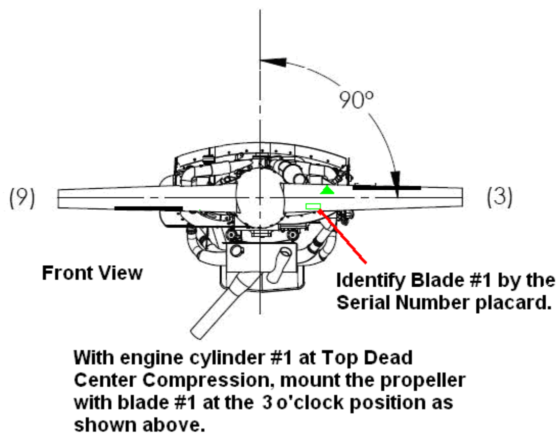

Figure 61-3 Second MT Propeller Position

7. Verify the #1 propeller blade match the #1 stamped on the spinner bulkhead.
8. If installing an MT Propeller, torque propeller stop nuts (6) in accordance with Section 4.5 and 4.6.1 of MT Propeller Maintenance Manual E-112 Issue 14 (03/19/2009) or later approved issue. If installing a Sensenich propeller, torque the stop nuts (6) in accordance the Sensenich Propeller Manual, DOC# WOOD-CF-REV-A DOC 5-20-04.
9. Install the spinner. When installing the spinner, match the number one (1) marked on the inside of the spinner with the #1 propeller blade, and the number one (1) stamped in the spinner bulkhead flange.
10. Perform flight test to evaluate the propeller-induced vibration.
11. If the vibration has been reduced to an acceptable level, this completes the clocking procedure. Check the torque on the propeller nuts; re-torque as required.

12. If the vibration is equal to or greater than previously recorded, remove the spinner, the propeller and the spinner bulkhead and return to the original configuration, aligning the marks made in step 4 of this procedure. This is the reference position.
13. Reposition the spinner bulkhead and the propeller to the next drive bushing counter-clockwise (ref. Section 4.4 of MT E-112 Issue 14). Verify #1 blade / spinner alignment and torque in accordance with Section 4.6.1 of MT E-112 Issue 14.
14. Perform flight test to evaluate the propeller induced vibration.
15. Repeat Steps 13 and 14 of this procedure (walking the propeller assembly around the propeller extension) until a vibration-less mounting position has been determined.
16. Once the vibration has been reduced to an acceptable level, verify the propeller nut torque, re-torque as required.
17. In the event that the balancing procedure detailed so far does not abate the vibration to an acceptable level, the propeller assembly must be reset to its original configuration (step 4 of this procedure) and be dynamically balanced. A peak velocity of 0.07 inch per second (IPS) or less recorded during dynamic balancing presents vibration levels that will not be detected by pilot or passengers. 0.15 IPS is the maximum acceptable level after dynamic balancing.

NOTE

The propeller, spinner, spinner bulkhead, spacer flange and propeller extension assembly are provided by MT to Liberty Aerospace as a statically balanced assembly. If any component supplied as part of the original assembly becomes damaged or modified, either the full assembly must be replaced with another statically balanced assembly and installed in accordance with this Service Document, or the modified assembly must be dynamically balanced on the airplane. Do not remove and replace propeller assembly components without dynamically balancing the assembly on the aircraft (ref. Section 4.6.3 of MT E-112 Issue 14).

PROPELLER TRACKING INSPECTION

A propeller out of track will cause undue vibration and stress to the airframe. Verify propeller tracking in accordance with Paragraph 8.108 of FAA AC 43.13-1B

1. Ensure all electrical switches are OFF and park the airplane in accordance with Chapter 10 – *Parking and Mooring*.
2. Remove the upper and lower engine cowls in accordance with Chapter 53 – *Fuselage*.
3. Set engine cylinder number one (#1) at top dead center (TDC) compression in accordance with Teledyne Continental Motors Maintenance Manual for IOF-240-B (TCM M-22). For the location of the number one (#1) engine cylinder, see Figure 61-4).

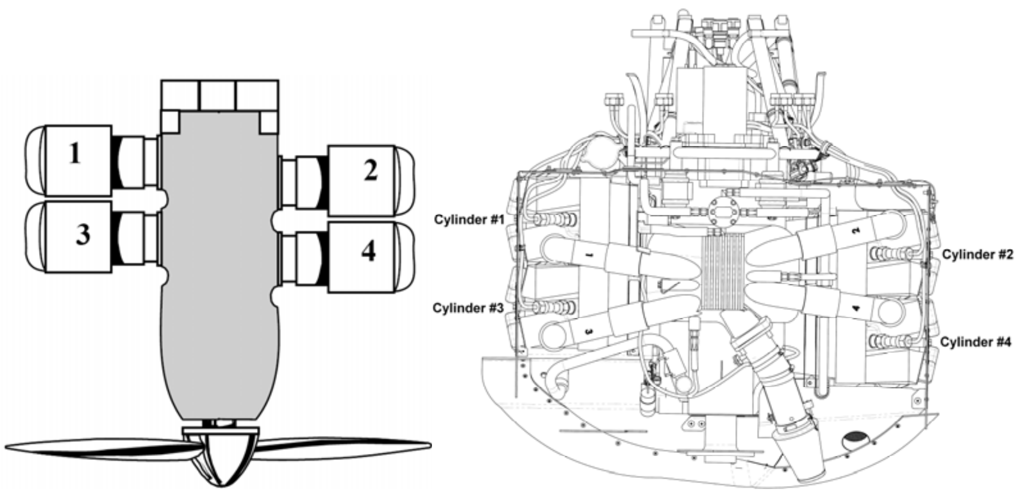

Figure 61-4 Cylinder Numbering

4. Check the orientation of the propeller to the TDC of the number one (#1) engine cylinder. The normal orientation is to have blade #1 of the propeller at the 9 o'clock position.
5. Measure the torque of the propeller bolts. The torque values should be as shown in the latest issue of the MT manual E-112 - Section 4.6 or Sensenich manual DOC# WOOD-CF-REV-A DOC 5-20-04.

PROPELLER AND BOLT TORQUE INSPECTION

Perform this procedure to inspect the torque on the propeller bolts. Perform this procedure on the schedule as defined in Chapter 5 – *Time Limits/Maintenance Checks/Inspection Intervals*.

1. Ensure all electrical switches are OFF and park the airplane in accordance with Chapter 10 – *Parking and Mooring*.
2. Remove the upper and lower engine cowls in accordance with Chapter 53 – *Fuselage*.
3. Remove the propeller spinner.
4. Set a calibrated torque wrench to 310 in-lbs.
5. Check the torque on each of the bolts uniformly and crosswise. Avoid crushing of the front plate in the hub.
6. Visually inspect the spinner for damage such as cracks or abrasions.
7. Visually inspect the propeller hub (spinner bulkhead, spacer flange and propeller extension assembly) for damage such as cracks, abrasions, or other damage.
8. Inspect the propeller for scratches, dings, voids or blistering of the surface of the propeller. As applicable to the installation, refer to the latest issue of the of the MT manual E-112 - Section 4.6 or Sensenich manual DOC# WOOD-CF-REV-A DOC 5-20-04
9. Install the spinner to the propeller, align the spinner to blade one on the propeller.
10. Torque the spinner bolts to 35 to 44 in-lbs.

NOTE

The propeller, spinner, spinner bulkhead, spacer flange and propeller extension assembly are provided by MT to Liberty Aerospace as a statically balanced assembly. If any component supplied as part of the original assembly becomes damaged or modified, either the full assembly must be replaced with another statically balanced assembly and installed in accordance with this Service Document, or the modified assembly must be dynamically balanced on the airplane. Do not remove and replace propeller assembly components without dynamically balancing the assembly on the aircraft (ref. Section 4.6.3 of MT E-112 Issue 14).

Section 00-04 Troubleshooting Chart

Table 61-1 provides information to aid in troubleshooting issues with the propeller.

Complaint	Possible Cause	Remedy
Vibration	Propeller out of balance	Do the Propeller Installation (Balancing or Clocking) procedure on page 7 of this chapter
Dents in the erosion sheath larger than 0.24” x 0.24”	Minor impact	Replace erosion strip
Dents in the erosion sheath smaller than 0.24” x 0.24”	Minor impact	Repair or Replace per the MT Manual E112 or the Sensenich manual DOC# WOOD-CF-REV-A DOC 5-20-04 (As applicable to the installation)
Cracking in the erosion Sheath	Various	Replace erosion sheath
Cracking in the lacquer	Various	Refer to the MT Manual E112 or the Sensenich manual DOC# WOOD-CF-REV-A DOC 5-20-04 (As applicable to the installation)
Cracking that goes down to or into the composite wood core	Over the rated speed or prop-strike	Replace

Table 61-1 Troubleshooting guide for the Propeller

CHAPTER 70

STANDARD PRACTICES - ENGINE

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Table of Contents

SECTION 70-00 GENERAL 5

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Section 70-00 General

The IOF-240B engine is a four-cylinder, four-stroke reciprocating aircraft engine. It is a horizontally-opposed, air-cooled, naturally-aspirated engine. The IOF-240B engine has a wet oil sump with a high pressure lubrication system. The engine also has an overhead plenum intake manifold, a side-mounted accessory drive pad and provisions for a downdraft exhaust system.

The engine is equipped with a Full Authority Digital Engine Control (FADEC) System for continuous monitoring and control of ignition timing, fuel injection timing, and fuel mixture. The microprocessor-based FADEC System monitors engine operating conditions and automatically sets the fuel mixture and ignition timing for any given power setting. The FADEC engine does not require a fuel mixture control lever. In the FADEC-controlled engine a cylinder can be leaned or enriched individually without affecting the other cylinders.

The FADEC System controls the fuel supplied to each cylinder using solenoid-actuated sequential port fuel injectors. The engine-driven, positive displacement fuel pump, which is driven at the same speed as the crankshaft, supplies fuel to the injectors. Fuel flow and fuel pressure vary directly with engine speed.

Fuel passes through a fuel filter to the fuel distribution block, and is distributed to each fuel injector. An electric boost pump is used for starting the engine and during low RPM operation. Ignition spark, variable from cranking speed to 2,000 RPM, is timed to the engine’s crank position. Spark energy varies with respect to engine load.

The FADEC System is electrically powered from the aircraft’s primary electrical bus and a secondary battery that is used to supply power to the FADEC System independently from the aircraft’s primary bus.

Maintenance of the TCM IOF-240B engine is contained in the current Teledyne Continental Motors Maintenance Manual for IOF-240B series engines, P/N M-22.

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CHAPTER 71

POWER PLANT

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Table of Contents

SECTION 71-00 GENERAL 5

SECTION 00-01 ENGINE MODEL DESCRIPTION 5
SECTION 00-02 ENGINE PROCEDURES 6
ENGINE REMOVAL 7
ENGINE INSTALLATION 14
ENGINE OPERATIONAL CHECK 21
ENGINE LEVEL I DIAGNOSTIC 24
SECTION 00-03 ENGINE TROUBLESHOOTING GUIDE 25

SECTION 71-10 COWLING 28

SECTION 10-01 COWLING PROCEDURES 28
COWLING REMOVAL 29
COWLING INSTALLATION 30
COWLING INSPECTION 31
SECTION 10-02 COWLING TROUBLESHOOTING GUIDE 32

SECTION 71-20 ENGINE MOUNTS 33

SECTION 20-01 ENGINE MOUNT PROCEDURES 33
ENGINE MOUNT REMOVAL 34
ENGINE MOUNT INSTALLATION 35
ENGINE MOUNT FRAME REMOVAL 36
ENGINE MOUNT FRAME INSTALLATION 37
ENGINE MOUNT AND FRAME INSPECTION 38
SECTION 20-02 ENGINE MOUNT TROUBLESHOOTING GUIDE 39

SECTION 71-30 FIREWALL BLANKET 41

SECTION 30-01 FIREWALL BLANKET PROCEDURES 42
FIREWALL BLANKET REMOVAL 43
FIREWALL BLANKET INSTALLATION 46
FIREWALL BLANKET INSPECTION 50
SECTION 30-02 FIREWALL BLANKET TROUBLESHOOTING GUIDE 51

SECTION 71-50 ENGINE ELECTRICAL HARNESS 53

SECTION 50-01 FADEC LOW VOLTAGE HARNESS DESCRIPTION 53
SECTION 50-02 FADEC IGNITION SYSTEM HARNESS 54
SECTION 50-03 FADEC HARNESS ASSEMBLY PROCEDURES 55
IGNITION SYSTEM HARNESS REMOVAL 56
IGNITION SYSTEM HARNESS INSTALLATION 57
IGNITION SYSTEM HARNESS INSPECTION PROCEDURE 59
LOW VOLTAGE HARNESS REMOVAL 60
LOW VOLTAGE HARNESS INSTALLATION 61
LOW VOLTAGE HARNESS INSPECTION 62
SECTION 50-04 ENGINE ELECTRICAL HARNESS TROUBLESHOOTING GUIDE 64

SECTION 71-60 AIR INTAKES 65

SECTION 60-01 AIR INTAKE PROCEDURES 65
AIR INTAKE REMOVAL 66
AIR INTAKE INSTALLATION 68
ALTERNATE AIR FLAPPER VALVE ARM REMOVAL 69
ALTERNATE AIR FLAPPER VALVE ARM PROCEDURE 70
AIR INTAKE INSPECTION 72
SECTION 60-02 AIR INTAKE TROUBLESHOOTING GUIDE 76

SECTION 71-70 ENGINE DRAINS 77

SECTION 70-01 MANIFOLD FUEL DRAIN PROCEDURES 77
MANIFOLD FUEL DRAIN CHECK PROCEDURE 78
SECTION 70-02 ENGINE DRAINS TROUBLESHOOTING GUIDE 79

Section 71-00 General

This chapter provides a descriptive overview of the IOF-240-B engine installed on the airplane. Detailed information for routine line maintenance for each engine subsection or system is provided in the appropriate chapter. More detailed information for repairs and maintenance on systems and components specific to the IOF-240-B engine (in particular, the FADEC system) are provided in the current release of the Teledyne Continental Motors Maintenance Manual for IOF-240-B series engines, TCM p/n: M-22.

Section 00-01 Engine Model Description

A Teledyne Continental Motors IOF-240-B engine rated at 125 bhp maximum continuous power powers the airplane. Recommended cruise power is 90 BHP. The engine drives a two-blade wooden fixed pitch propeller.

The IOF-240-B-X is a four-cylinder air-cooled engine. Its type designation reflects the following:

I – the engine is fuel injected

O – Cylinder layout - horizontally opposed

F– The engine uses a Full Authority Digital Engine Control system (FADEC)

240 – Engine displacement is 240 cubic inches

B – Model designation indicating that the engine is designed for use with a fixed-pitch propeller, with a doweled six bolt hole propeller flange and no provision for a hydraulic propeller governor

X – Specific configuration of accessories as supplied to Liberty Aerospace, Inc., for use on the airplane.

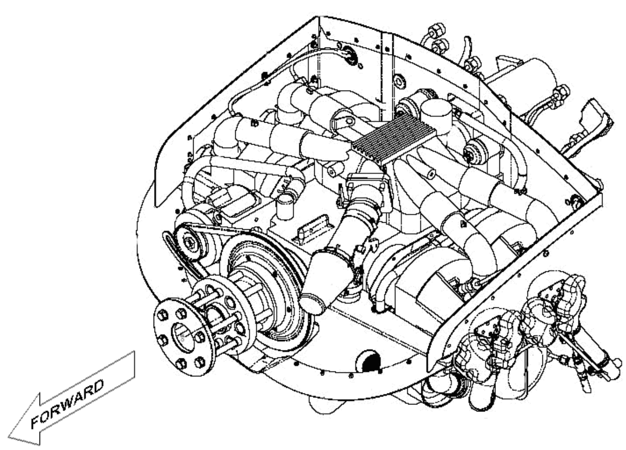

Figure 71-1 Engine Assembly

Because of the nature and complexity of the FADEC system, procedures covered in this maintenance manual include routine line servicing, replenishment of operating fluids, and replacement of “consumable” components such as spark plugs only. Detailed information and procedures covering maintenance and troubleshooting of the FADEC system and related components are provided in the Teledyne Continental Motors Maintenance Manual for IOF-240-B series engines,

A cylinder head temperature (CHT) sensor is installed in each cylinder head to provide required data to the FADEC system computers. An exhaust gas temperature (EGT) sensor is installed in each cylinder’s exhaust pipe, approximately two inches from its attachment to the cylinder exhaust port, to provide required data to the FADEC system computers.

Section 00-02 Engine Procedures

The following sections provide procedures for removal, installation, and operational check of the engine. Operational checks are performed in accordance with Chapter 05 – *Time Limits/Maintenance Checks/Inspection Intervals*.

ENGINE REMOVAL

Perform this procedure to remove the engine.

CAUTION

Engine removal requires use of an engine hoisting system of sufficient capacity to suspend the full weight of the engine and attached accessories. Using a hoist of insufficient capacity may result in personal injury, engine damage or both.

CAUTION

Before starting these procedures, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section while accessing any area aft of the passenger compartment.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Pull the BAT1 (CB001) circuit breaker to OPEN.
3. Install a tail stand underneath the tail section of the airplane.
4. Remove the cabin aft bulkhead access panel, by removing securing screw hardware.
5. Disconnect the negative then the positive leads from the primary battery. Isolate the negative terminals on the batteries to prevent accidental connection.

CAUTION

Failure to disconnect the batteries can cause damage to the electrical circuitry of the airplane.

6. Turn cockpit fuel selector OFF.
7. Remove upper and lower cowlings in accordance with the Cowling Removal procedures on page 29 of this chapter.
8. Remove propeller and spinner in accordance with Chapter 61 - *Propellers*.
9. Disconnect alternate air operating linkages referring to Page 66 of this Chapter for linkage details.
10. Disconnect throttle linkage referring to Chapter 76 – *Engine Controls* for linkage removal details.

CAUTION

The aircraft can be equipped with one of two types of crankcase breather systems a direct breather line and an air/oil separator Breather system. Refer to step 11 for the air/oil separator system or step 13 for the direct breather system.

11. If so equipped, disconnect the crankcase breather line from the air/oil separator at the location shown in Figure 71-2. The case breather line will be removed with the engine.
12. If so equipped, disconnect the air/oil separator oil sump return line from the engine fitting as shown in Figure 71-2. Cap the line and engine fitting after disconnection.

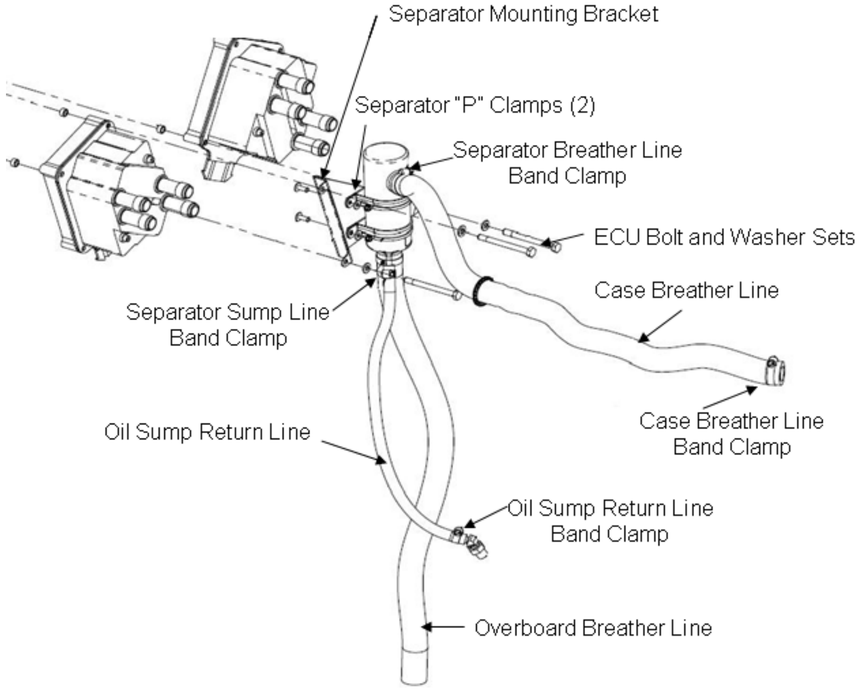

Figure 71-2 Air/Oil Separator Engine Disconnections

13. If so equipped, remove crankcase direct breather line clamp assembles from the lower engine mount frame assembly as shown in Figure 71-3. The crankcase breather line is removed with the engine.

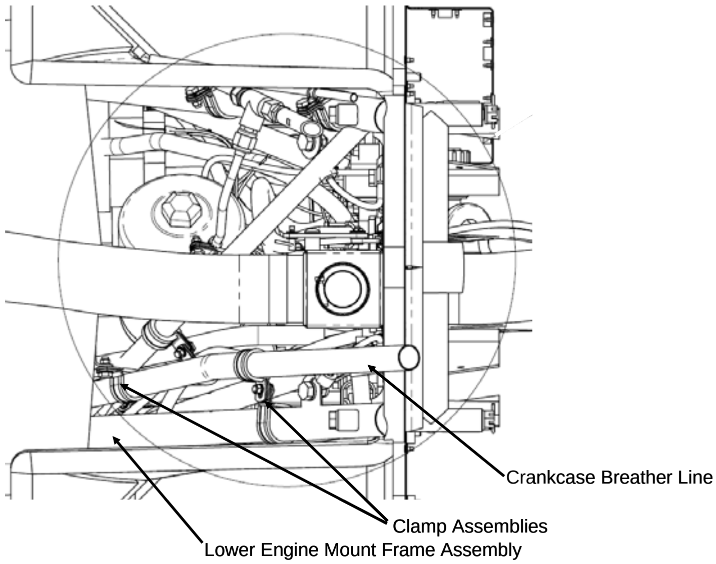

Figure 71-3 Crankcase Direct Breather Line Clamp Removal

NOTE

In the following steps wire harnesses and ignition leads from the engine to the airframe will be disconnected. Harness routing is secured by means of clamps and wire ties. Remove all securing clamps and hardware required to free harnesses permitting engine removal. Refer to Figure 71-4 for location of components to be disconnected.

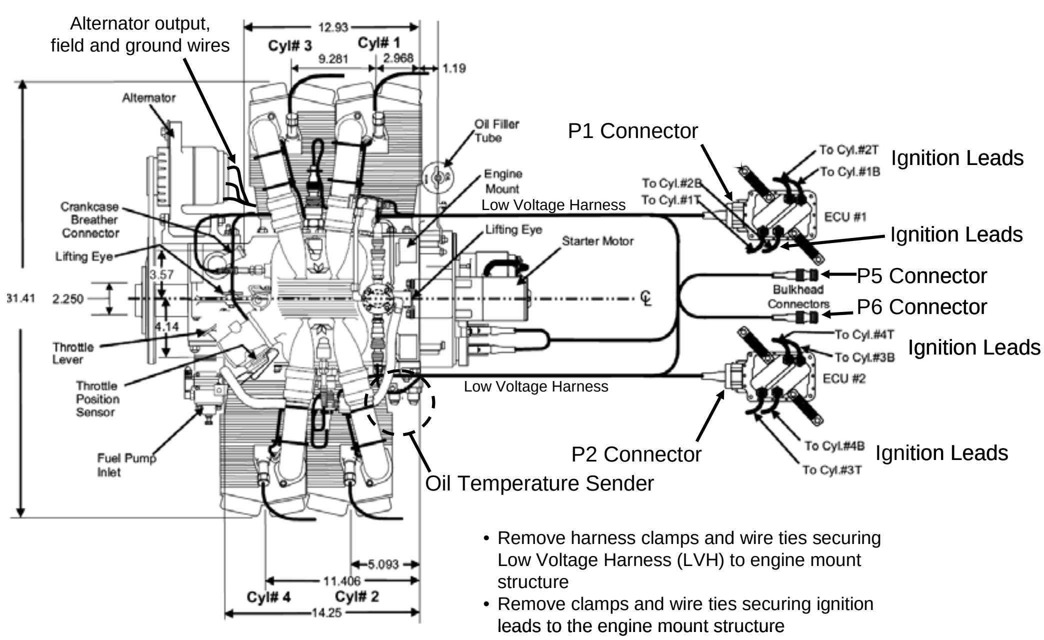

Figure 71-4 Engine Disconnections

14. Disconnect ignition leads from each spark plug. Cap and stow the leads aft of the engine mounts. Remove ignition lead clamps and insulated baffle plates as required to permit lead removal from the engine. Ignition leads will remain with the aircraft connected to ECU-1 and ECU-2. Ignition lead clamps aft of the engine mounts are to remain in place.
15. Disconnect power lead P04A4 from starter. This lead remains with the airframe.
16. Disconnect ground strap from the engine as shown in Figure 71-5. Do not disconnect strap from the engine mount frame grounding tab.

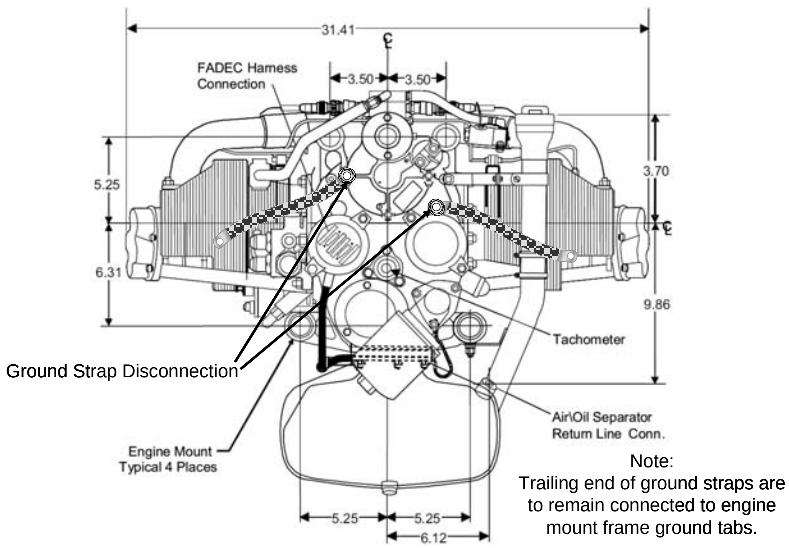

Figure 71-5 Engine Ground Strap Disconnection

17. Remove clamps and wire ties securing low voltage wire harnesses to the engine mount frame. Ignition lead clamps are to remain in place.
18. Disconnect cabin heat duct from Cabin heat box assembly and muffler shroud assembly. Remove duct and set aside for engine installation.
19. Disconnect P5 From P32 on the tunnel firewall assembly as shown in Figure 71-6.
20. Disconnect P6 From P32 on the tunnel firewall assembly as shown in Figure 71-6.
21. Disconnect J33 from P33 on the tunnel firewall assembly as shown in Figure 71-6.

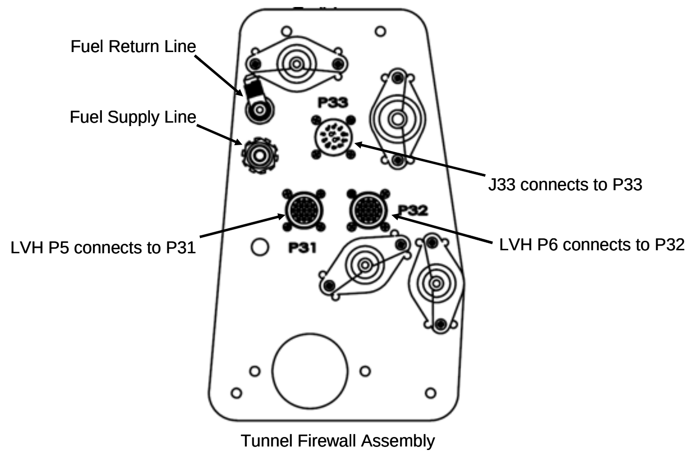

Figure 71-6 Tunnel Firewall Assembly Electrical Disconnections

22. Disconnect Low Voltage Harness connector P1 from ECU-1.
23. Disconnect Low Voltage Harness connector P2 from ECU-2.
24. Disconnect J34 from the oil pressure transducer located as shown in Figure 71-7

NOTE

Minor fuel spillage may occur; position containers and/or absorbent material below firewall connections before disconnecting.

25. Disconnect fuel supply and fuel return lines at firewall location shown in Figure 71-7. Cap lines and fittings after disconnection.

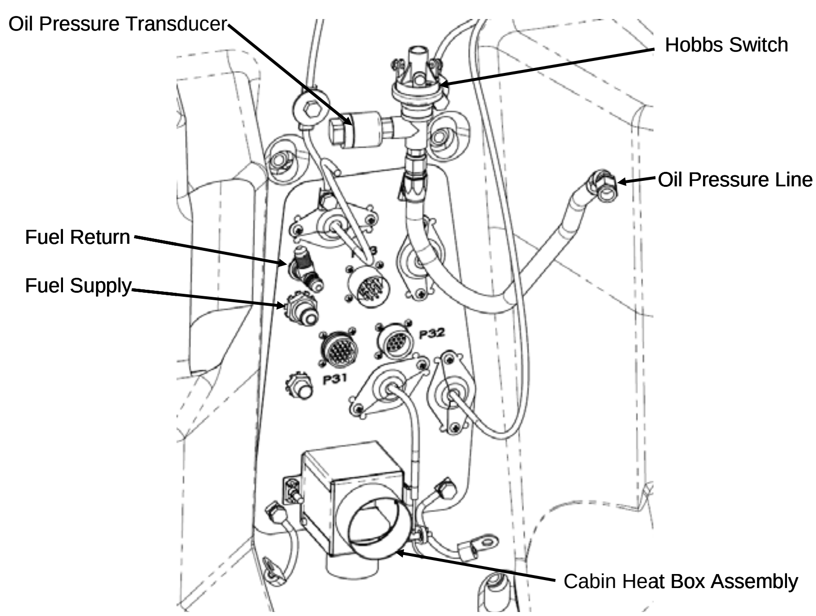

Figure 71-7 Tunnel Firewall Fuel Line Disconnections

26. Disconnect oil pressure line from the Oil Cooler Adapter mounted Oil Restrictor as shown in Figure 71-7. Cap open connections.

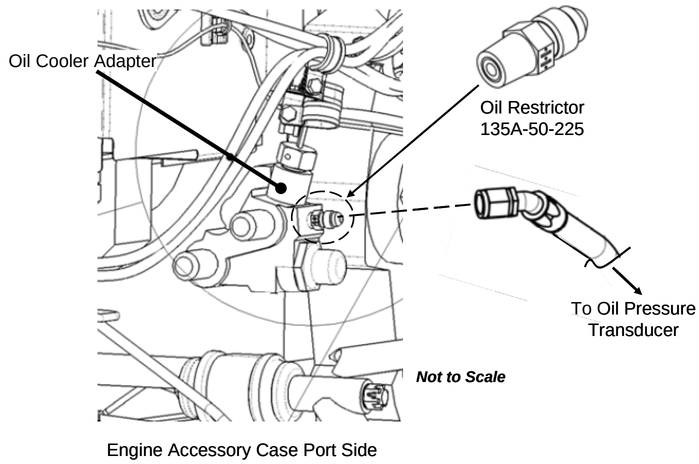

Figure 71-8 Oil Pressure Line Disconnection

27. Remove clamp assemblies connecting fuel drain lines to the lower engine mount frame assembly as shown in Figure 71-9. Fuel drain lines are removed with the engine.

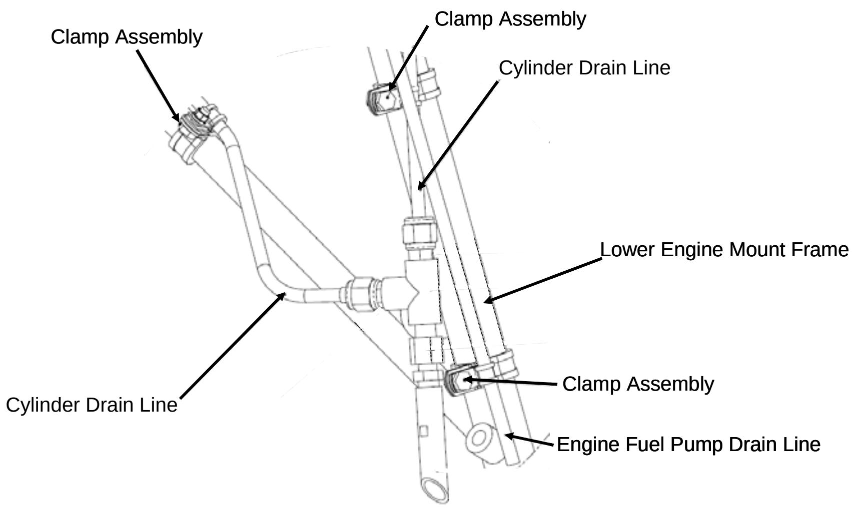

Figure 71-9 Fuel Drain Line Clamp Assemblies

28. Disconnect oil cooler from firewall Oil Cooler Bracket Assembly as shown in Figure 71-10. The Oil Cooler Assembly is removed with the engine. Secure the assembly to engine structure with temporary support relieving oil line strain.

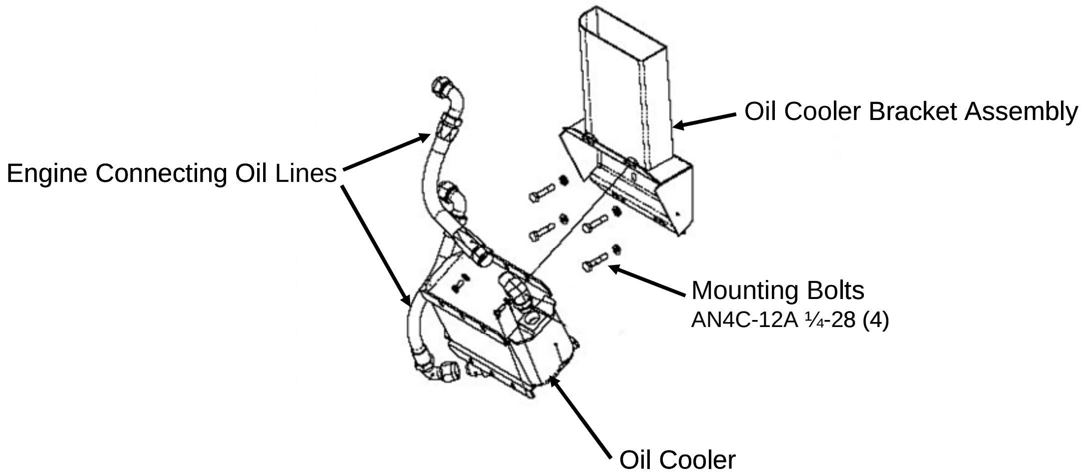

Figure 71-10 Oil Cooler Removal

29. Attach hoist to lifting eye at top of crankcase.

WARNING

IF ENGINE HOIST IS MOVABLE, THE POWER-PLANT PACKAGE CAN BE MOVED AWAY FROM AIRPLANE. PRIOR TO ANY MOVEMENT, VERIFY A TAIL STAND HAS BEEN INSTALLED. FAILURE TO DO SO MAY RESULT IN PERSONAL INJURY, DAMAGE TO THE AIRCRAFT OR BOTH.

30. Raise hoist until load on engine mounts is relieved.
31. Remove four (4) engine mounts in accordance with Page 34 of this Chapter.
32. Remove entire power-plant package including exhaust system, oil cooler, baffles, etc. as one assembly.

This completes the Engine Removal procedure.

ENGINE INSTALLATION

Perform this procedure to install the engine.

1. Verify electrical power is removed from the aircraft and that primary and secondary batteries have been disconnected at their terminals.
2. Using hoist, place dressed engine assembly in correct position to allow (4) engine mount bolts to be inserted.
3. Install (4) engine mounts in accordance with Page 35 of this Chapter.
4. Connect oil cooler to oil cooler bracket assembly as shown in Figure 71-11. Torque ¼-28 mounting bolts in accordance with torque tables provided in Chapter 20 *Standard Practices Airframe*.

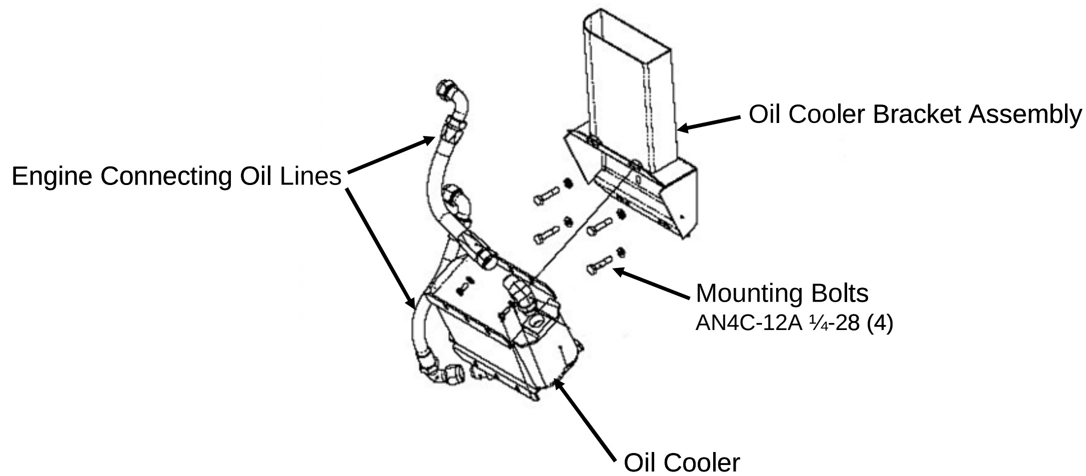

Figure 71-11 Oil Cooler Installation

5. Install clamp assemblies connecting fuel drain lines to the lower engine mount frame assembly as shown in Figure 71-12. Fuel drain lines are removed with the engine.

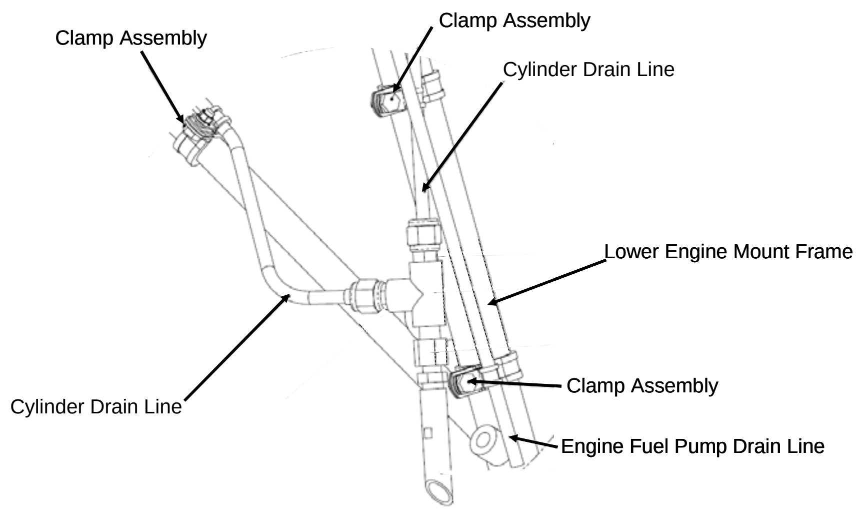

Figure 71-12 Fuel Drain Line Clamp Assembly Installation

6. Connect oil pressure line from the Oil Cooler Adapter mounted Oil Restrictor as shown in Figure 71-13. Cap open connections.

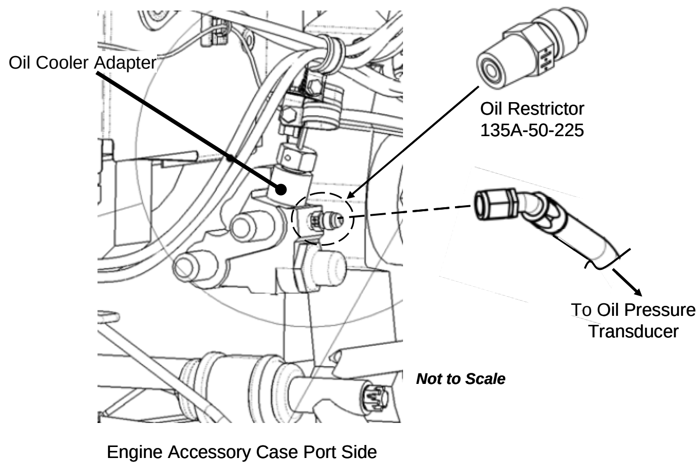

Figure 71-13 Oil Pressure Line Connection

7. Connect fuel supply and fuel return lines at firewall location shown in Figure 71-14.

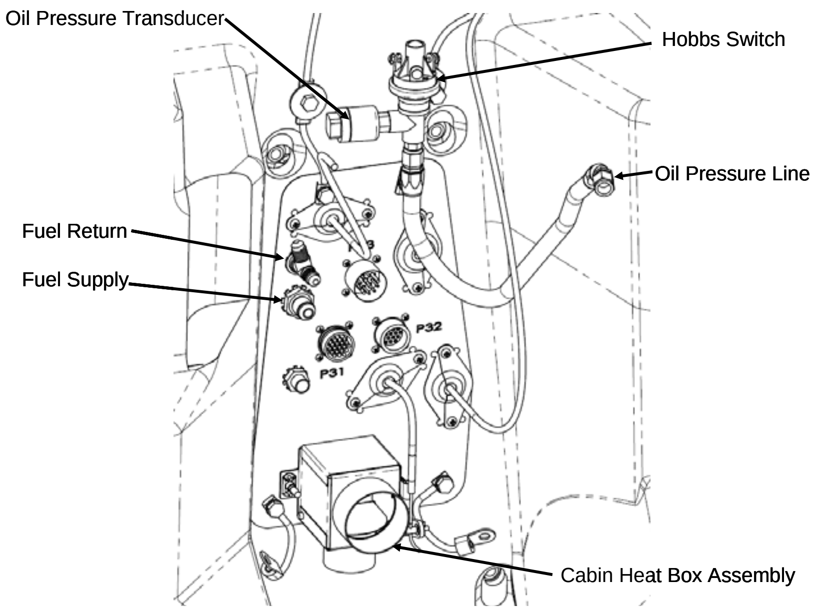

Figure 71-14 Tunnel Firewall Fuel Line Connections

NOTE

In the following steps wire harnesses and ignition leads from the engine to the airframe will be connected. Harness routing is to be secured by means of clamps and wire ties. Install harness securing clamps and hardware removed previously. Refer to Figure 71-15 for the location of components to be connected.

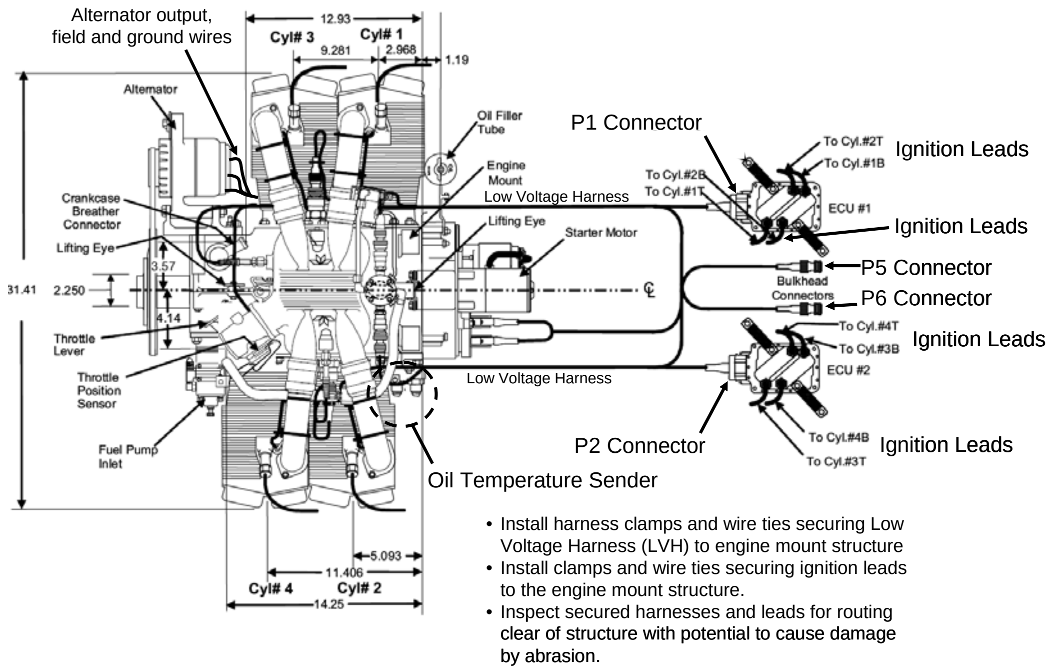

Figure 71-15 Engine Connections

8. Connect ignition leads to respective ECU-1 and ECU-2 ignition towers in accordance with Teledyne Continental Motors Installation and Operations manual OI-22 Chapter 5.
9. Connect Low Voltage Harness connector P1 from ECU-1 in accordance with Teledyne Continental Motors Installation and Operations manual OI-22 Chapter 4.
10. Connect Low Voltage Harness connector P2 from ECU-2 in accordance with Teledyne Continental Motors Installation and Operations manual OI-22 Chapter 4.
11. Connect J34 to the oil pressure transducer located as shown in Figure 71-14.
12. Connect P5 To P32 on the tunnel firewall assembly as shown in Figure 71-16.
13. Connect P6 To P32 on the tunnel firewall assembly as shown in Figure 71-16.
14. Connect J33 To P33 on the tunnel firewall assembly as shown in Figure 71-16.

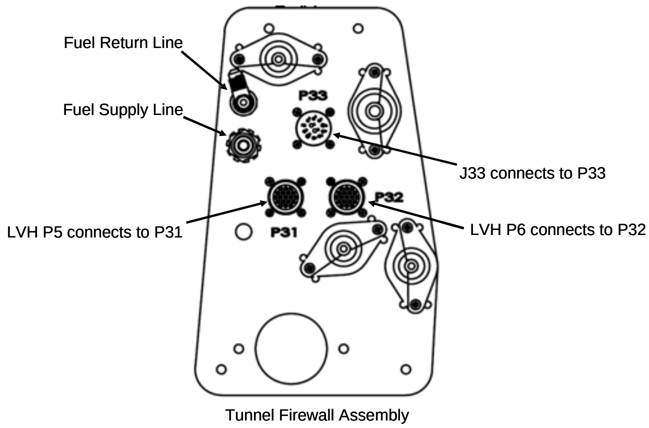

Figure 71-16 Tunnel Firewall Assembly Electrical Connections

15. Connect Low Voltage Harness connectors P5 and P6 at firewall (Cannon plugs) in accordance with Teledyne Continental Motors Installation and Operations manual OI-22 Chapter 4.

16. Connect cabin heat duct to Cabin heat box assembly and muffler shroud assembly as show in Figure 71-17.

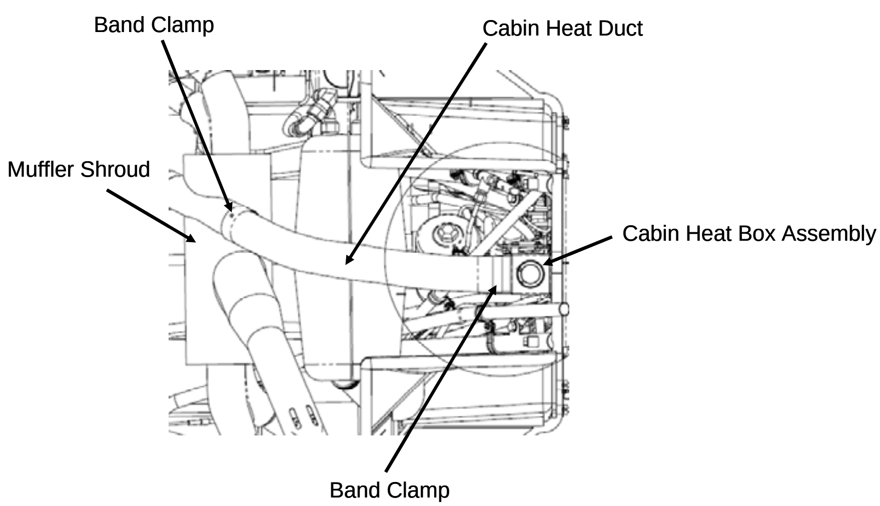

Figure 71-17 Cabin Heat Duct Installation

17. Connect ground strap between engine and airframe as shown in Figure 71-18. Torque strap nuts in accordance with Teledyne Continental Motors Installation and Operations manual OI-22 Appendix B.

18. Perform ground strap bond check in accordance with Teledyne Continental Motors Installation and Operations manual OI-22 Chapter 4.

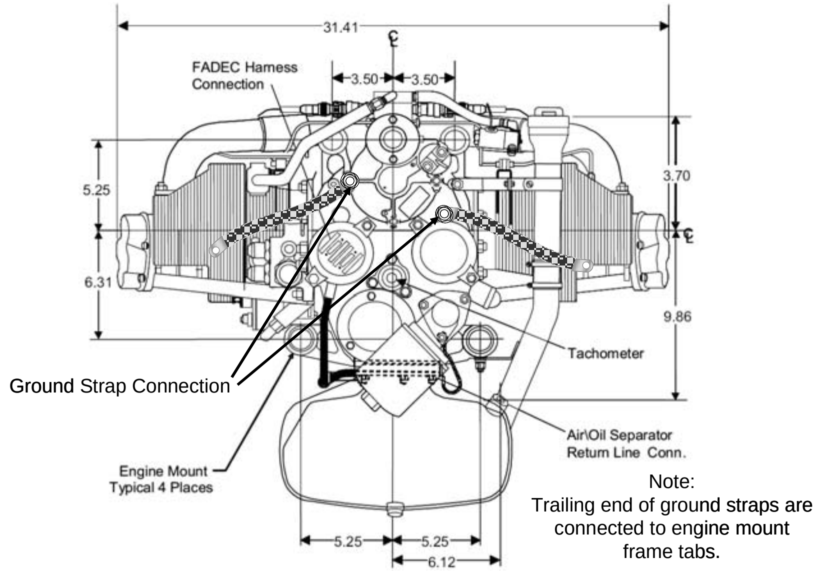

Figure 71-18 Engine Ground Strap Connections

19. Connect power lead P04A4 to the starter. Torque terminal nut to 100 +/- 5 in/lbs.

NOTE

The aircraft can be equipped with one of two types of crankcase breather systems a direct breather line and an air/oil separator Breather system. Refer to step 11 for the air/oil separator system or step 13 for the direct breather system.

20. If so equipped, Connect the crankcase breather line from the air/oil separator at the location shown in Figure 71-19. The case breather line will be removed with the engine.
21. If so equipped, disconnect the air/oil separator oil sump return line from the engine fitting as shown in Figure 71-2. Cap the line and engine fitting after disconnection.

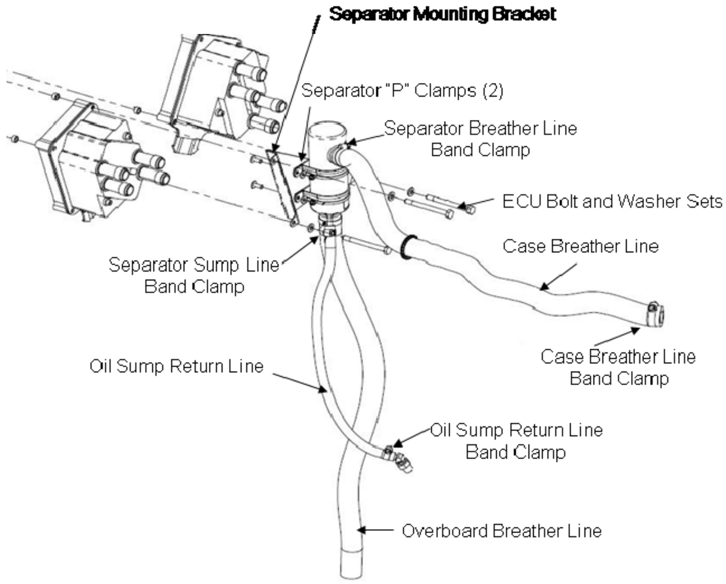

Figure 71-19 Air/Oil Separator Engine Connections

22. If so equipped, Install crankcase direct breather line clamp assembles to the lower engine mount frame assembly as shown in Figure 71-20. Breather line is installed on the engine prior to airframe installation.

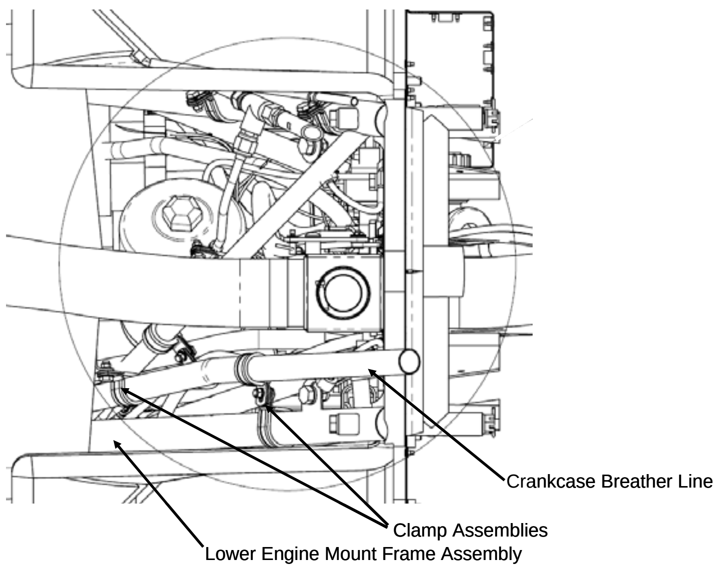

Figure 71-20 Crankcase Direct Breather Line Clamp Installation

23. Connect throttle linkage referring to chapter to Chapter 76 – *Engine Controls* for linkage installation detail.
24. Install air intake ducts and alternate air operating linkages referring to Page 68 of this Chapter for installation details.
25. Install propeller and spinner in accordance with Chapter 61 - *Propellers*.
26. Install upper and lower cowlings in accordance with the Cowling Installation procedures on page 29 of this chapter.

27. Perform Engine Operational Check in accordance with TCM Installation and Operation Manual OI-22, Chapter 5.

This completes the Engine Installation procedure.

ENGINE OPERATIONAL CHECK

Perform this procedure to run an operational check of the engine prior to 50 and 100-hour engine inspections as called out in Chapter 5 of this manual or at any time an engine suspect condition exists.

For the following conditions perform engine operational checks in accordance with Teledyne Continental Motors Installation and Operations Manual OI-22 Chapter 5:

- 50-Hour Inspection
- 100-Hour inspection
- Engine Installation
- Maintenance and Troubleshooting
- Engine Overhaul
- Return From Storage

CAUTION

The following step will start the engine. Monitor oil pressure indication. In the event oil pressure does not rise within 30 seconds shut down the engine and proceed to Engine Troubleshooting Guide at the end of this section.

1. Start the engine following POH/AFM procedures. Allow engine to warm up until oil temperature is 75 deg. F or higher.
2. Check and record the following engine status data on start:

Check	Result	Notes
Engine Hours	HOBBS: Tach:	• Record Hobbs • Record Tachometer
Fuel Boost Pump	Auto: Manual: PSI:	• Runs in Auto mode • Runs in Manual mode • 30 psi minimum
Starter Operation	Starter: Indicator:	• Verify starter operates • Verify indicator lights.
Oil Pressure	PSI: Rise Time:	• Idle 850-900 RPM • Pressure rise < 30 sec.
Oil Temperature	F:	• 75 – 220 Normal
FADEC Both RPM	RPM:	• Record RPM drop for each FADEC channel • Check at 1700 RPM • Maximum allowable drop is 150 RPM,
FADEC Left RPM	RPM:	

Check	Result	Notes
FADEC Right RPM	RPM:	• Maximum allowable difference is 75 rpm.
FADEC Power B OFF FADEC Power A ON	EBAT FAIL: ______ Engine Op: ______	• Verify EBAT FAIL is ON • Verify normal engine operation
FADEC Power B ON FADEC Power A OFF	PPWR FAIL: ______ EBAT FAIL: ______ Engine Op: ______	• Verify PPWR FAIL is ON • Verfiy EBAT FAIL is ON • Verify engine operation is normal
FADEC Power B ON FADEC Power A ON	PPWR FAIL: ______ EBAT FAIL: ______ Engine Op: ______	• Verify PPWR FAIL is OFF • Verfiy EBAT FAIL is OFF • Verify engine operation is normal

3. Increase engine to full throttle and record:

Check	Result	Notes
WOT Light	WOT:	• On at full throttle • Off below full throttle
Manifold Pressure	InHg:	• 15 – 29.5 Normal
RPM at Full Throttle	RPM:	•
Fuel Pressure:	PSI:	• 25 – 98 PSI – Normal
Fuel Boost Pump:		• Verify boost pump cycles off above 1200 RPM.
Oil Pressure	PSI:	• 30 – 60 PSI Normal
Oil Temperature	F:	• 75 – 220 Normal
Cylinder Head Temperature (CHT)	CY1: ______ CY2: ______ CY3: ______ CY4: ______	• 240 – 420 F Normal
Exhaust Gas Temperature (EGT)	CY1: ______ CY2: ______ CY3: ______ CY4: ______	• 1000 – 1675 F Normal
Alternator Output	Volts: ______ VM1000 Amps: ______ VM1000 Volts: ______ OAT	• 12.0 – 14.3 Normal, 14.1 +/- 0.1 Optimum • Expect current < 60A

Check	Result	Notes
HSA Status	FADEC WARN: ______ FADEC CAUTION: ______ PPWR FAIL: ______ EBAT FAIL: ______ FUEL PUMP: ______	• Verify all HSA indicators are OFF.

4. Reduce engine to idle and record:

Check	Result	Notes
Manifold Pressure	InHg:	• 15 – 29.5 Normal
RPM		• Idle 850-900 RPM
Oil Pressure	PSI:	• Idle 850-900 RPM
Oil Temperature	F:	• 75 – 220 Normal
Cylinder Head Temperature (CHT)	CY1: ______ CY2: ______ CY3: ______ CY4: ______	• 240 – 420 F Normal
Exhaust Gas Temperature (EGT)	CY1: ______ CY2: ______ CY3: ______ CY4: ______	• 1000 – 1675 F Normal
Fuel Pressure	PSI: Boost Pump:	• 25 – 98 PSI – Normal • Verify boost pump runs at 1200 and below with switch in Auto

5. Shut down the engine following Liberty POH/AFM procedures.

This completes the Engine Operational Check procedure

ENGINE LEVEL I DIAGNOSTIC

The following procedure performs a Engine Level I Diagnostic procedure. A test computer loaded with the latest revision of Teledyne Continental Motors PowerLink™ Maintenance Manual Suite MMS-22 diagnostic software is required. Data cables delivered with the aircraft are also required. This procedure may be performed at any time on condition as well as during 50 and 100-hour inspections called out in Chapter 5 of this manual.

1. Verify aircraft split master switch is in the off position.
2. Locate the Engine Data Interface (EDI-200) unit beneath the circuit breaker panel and connect data cables as shown in Figure 71-21 with the test computer.

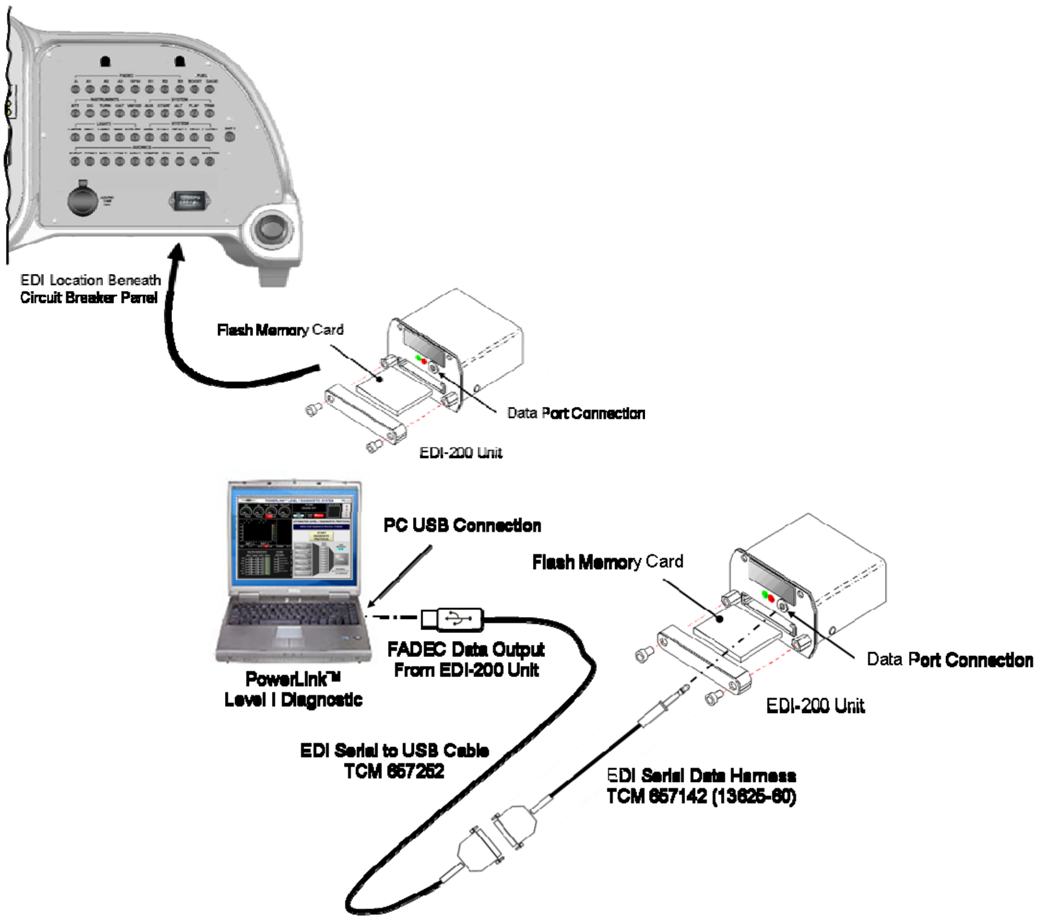

Figure 71-21 EDI Connections

3. Launch the Level I Diagnostic application in accordance with Teledyne Continental Motors PowerLink™ FADEC Advanced Power Plant Diagnostic System User Guide PLG-22.

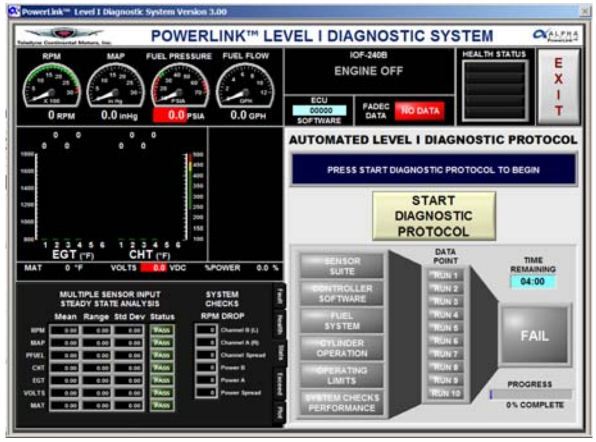

Figure 71-22 PowerLink™ Level I Diagnostic Display

4. Following instructions in the user guide and instructions provide by the application complete a Level I diagnostic and print the results for aircraft records. Perform engine start and shutdown procedures in accordance with Liberty POH/AFM procedures.

5. On completion, remove data cables and test computer.

This completes the Engine Level I Diagnostic procedure

Section 00-03 Engine Troubleshooting Guide

Use this troubleshooting guide to resolve issues with the engine. This guide provides initial troubleshooting guidance. Detailed procedure may be required in other chapters of this manual and in Teledyne Continental Motors maintenance manual M-22. Contact TCM for current manual information.

Complaint	Possible Cause	Remedy
Engine will not start	• Primary battery	• Inspect battery in accordance with chapter 24
	• Secondary battery	• Inspect battery in accordance with chapter 24
	• FADEC fault	• Perform troubleshooting procedure in accordance with TCM Maintenance Manual M-22 Chapter 8
	• Fuel fault	• Inspect fuel system in accordance with chapter 73
Oil pressure does not rise on start	• No oil in the engine	• Add oil
	• Oil pressure relief valve fault	• Refer to TCM Overhaul Manual M-22 Chapter 8 for corrective action
	• Oil pump failure	• Refer to TCM Overhaul Manual M-22 Chapter 8 for corrective action
Engine runs rough	• Induction system obstruction	• Inspect and clear induction system • Replace air filter

Complaint	Possible Cause	Remedy
	• FADEC fault	• Perform troubleshooting procedure in accordance with TCM Maintenance Manual M-22 Chapter 8
	• Propeller imbalance	• Check propeller torque in accordance with chapter 61
	• Engine mounts	• Inspect engine mounts in accordance with chapter 71
Poor engine performance	• Induction system obstruction	• Inspect and clear induction system • Replace air filter
	• FADEC fault	• Perform troubleshooting procedure in accordance with TCM Maintenance Manual M-22 Chapter 8
	• Throttle rigging	• Inspect and adjust throttle rigging in accordance with chapter 76
Level I diagnostic Fault	• Refer to diagnostic generated report for probable cause	• Take corrective action in accordance with diagnostic report guidance.

Table 71-1 Engine Troubleshooting Guide

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Section 71-10 Cowling

The airplane uses a two-piece (upper and lower) cowling of composite construction. The upper cowling incorporates a small door to allow oil level checks and oil replenishment without the need to remove the cowling.

Air exits from the engine compartment via an opening at the rear of the lower cowling.

Both cowlings are secured to the airframe and to each other with cam lock fasteners.

Section 10-01 Cowling Procedures

This section contains the procedures to remove, install, and inspect upper and lower engine cowl assemblies.

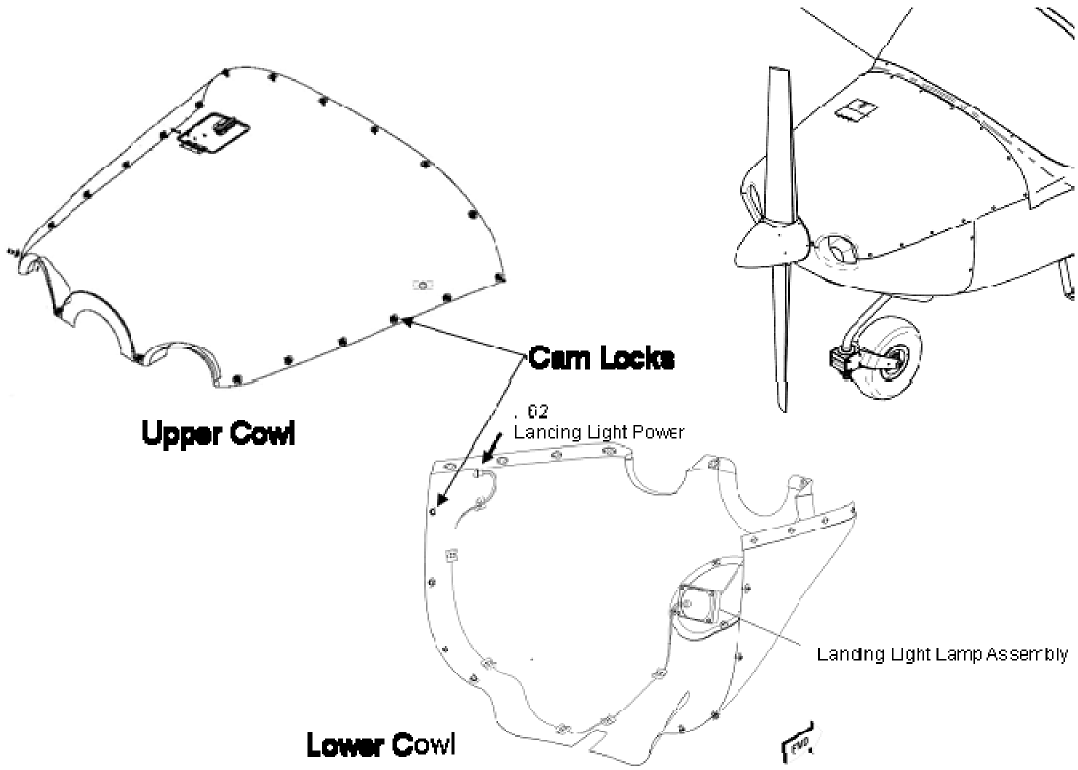

Figure 71-23 Guidance for Re-attaching Cowling

COWLING REMOVAL

Perform this procedure to remove the upper and lower cowling.

CAUTION

When working around the cowling and engine exhaust areas, allow time for cooling of components to avoid maintenance personnel burning themselves.

1. Ensure all electrical switches are OFF, and the engine has had adequate cool down time and the exhaust is cool.
2. Unfasten all Camloc® fasteners securing upper cowling to fuselage and upper cowling to lower cowling.
3. Lift off upper cowling.
4. Disconnect landing light connector P62 located near L/H engine baffle as shown in Figure 71-23.
5. Unfasten all Camloc® fasteners securing lower cowling except top rear fasteners on left and right side.
6. Support lower cowling while removing remaining fasteners.
7. Remove lower cowling taking care to remove cowl around exhaust.

This completes the upper and lower Cowling Removal procedure.

COWLING INSTALLATION

Perform this procedure to install the upper and lower cowling.

CAUTION

When re-installing cowling, it is important to place cowling properly and secure it loosely with (4) Camloc® fasteners at corners. Loosely insert remaining Cam-Locks. Tighten each Camloc® fasteners, insuring cowling is seating properly. Failure to re-install cowling properly can result in improper bending or "scooping".

1. With engine cool, raise lower cowling into position and fasten top rear Camloc on left and right side, taking care to maneuver cowl around exhaust.
2. Fasten remaining Camloc fasteners to secure lower cowling to fuselage.
3. Reconnect landing light connector P62 as shown in Figure 71-23.
4. Place upper cowling into position.

CAUTION

Operate engine only with both upper and lower cowling installed and secured, or with both upper and lower cowling removed (do not operate with either upper or lower cowling installed alone). Do not operate engine at high power unless both upper and lower cowling are installed and secured.

5. Fasten all cam locks securing upper cowling to fuselage and upper cowling to lower cowling.

This completes Cowling Installation procedure

COWLING INSPECTION

Perform the following Cowling Inspection in accordance with Chapter 05 – *Time Limits/Maintenance Checks/Inspection Intervals* any time cowling is removed for maintenance operations.

1. Inspect clearance/interference between propeller hub and cowls
2. Remove upper and lower cowling in accordance with the Cowling Removal procedure on page 29 of this chapter.
3. Inspect the cowling for cracks
4. Inspect for overheated areas, deformation or delamination
5. Inspect for loose or missing fasteners
6. Inspect for chafing or abnormal condition and oil,
7. If a hard landing is suspected, inspect the lower cowl for scratches or local dents from the nose leg hitting the clearance hole.
8. Inspect any paint removal.
9. Install lower and upper cowl in accordance with the Cowling Installation procedure on page 30 of this chapter.

This completes the Cowling Inspection procedure.

Section 10-02 Cowling Troubleshooting Guide

Use this troubleshooting guide to resolve issues with the engine.

Complaint	Possible Cause	Remedy
Missing or damaged Cam Lock	• In service Wear • Hard landing	• Replace
Cracks	• Hard Landing • In service wear • Impact damage	• Repair/replace in accordance with Chapter 51
Scratched Paint	• Impact damage	• Inspect for subsurface damage • Repaint
Delamination	• Excess heat exposure	• Inspect for heat source and correct • Repair/replace in accordance with Chapter 51

Table 71-2 Cowling Troubleshooting Guide

Section 71-20 Engine Mounts

A welded steel tube structure secures the engine to the fuselage center section space frame. It is attached to the space frame by four bolts with nuts and washers. If necessary, the engine mount can be removed from the space frame.

Four vibration-absorbing rubber mounts, with washers and bushings, are located at the rear of the engine crankcase. They are secured to the airplane engine mount by bolts, washers, and castellated nuts with cotter pins.

Adequate clearance to replace individual engine mount components, such as vibration absorbers, can be gained by withdrawing the engine mount bolts and moving the engine slightly forward (approximately one inch). The airplane tail must be supported and the engine must be raised with an engine hoist sufficiently to relieve loads on the engine mounts, but sufficient slack is available in most engine connections to minimize the amount of disconnection and disassembly required.

Section 20-01 Engine Mount Procedures

The following procedures perform engine mount removal, installation and inspection. Inspection is performed in accordance with the maintenance schedules in . Removal and replacement procedures are performed on condition.

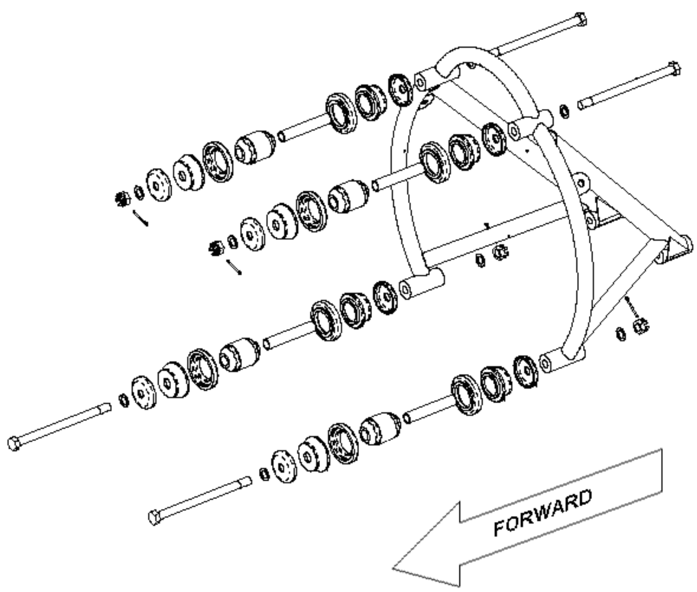

Figure 71-24 Exploded View of Engine Mount Installation

ENGINE MOUNT REMOVAL

Perform this procedure to remove the engine mount.

CAUTION

Before starting these procedures, the tail of the airplane requires support. Failure to support the airplane's tail may cause damage to the airplane's tail section.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Pull the BAT1 (CB001) circuit breaker to OPEN.
3. Install a tail stand underneath the tail section of the airplane.
4. Remove the cabin aft bulkhead access panel, by removing securing screw hardware.
5. Disconnect the negative then the positive leads from the primary battery. Isolate the negative terminals on the batteries to prevent accidental connection.

CAUTION

Failure to disconnect the batteries can cause damage to the electrical circuitry of the airplane.

6. Remove upper and lower engine cowling in accordance with the Cowling Removal procedure on page 29 of this chapter.
7. With an engine hoist raise engine, enough to relieve loading, but leaving sufficient slack to not bind mount bolts.
8. Loosen engine mount bolt and remove engine mount, noting sequence of parts, refer to engine mount exploded view for correct positioning of parts.

This completes the Engine Mount Removal procedure.

ENGINE MOUNT INSTALLATION

Perform this procedure to install the engine mount. Procedure applies to one or all four installations.

1. Ensure all isolators are flexible and not damaged.
2. Visually inspect all engine attachments and ensure they are not damaged.
3. Install engine mount(s) by reversing the above removal procedure.
4. Torque engine mount bolts to 175 ±15 in-lbs.

This completes the Engine Mount Installation procedure

ENGINE MOUNT FRAME REMOVAL

Perform this procedure to remove the engine mount frame.

1. Refer to Figure 71-25. Remove engine in accordance with the Engine Removal procedure on page 7 of this chapter.

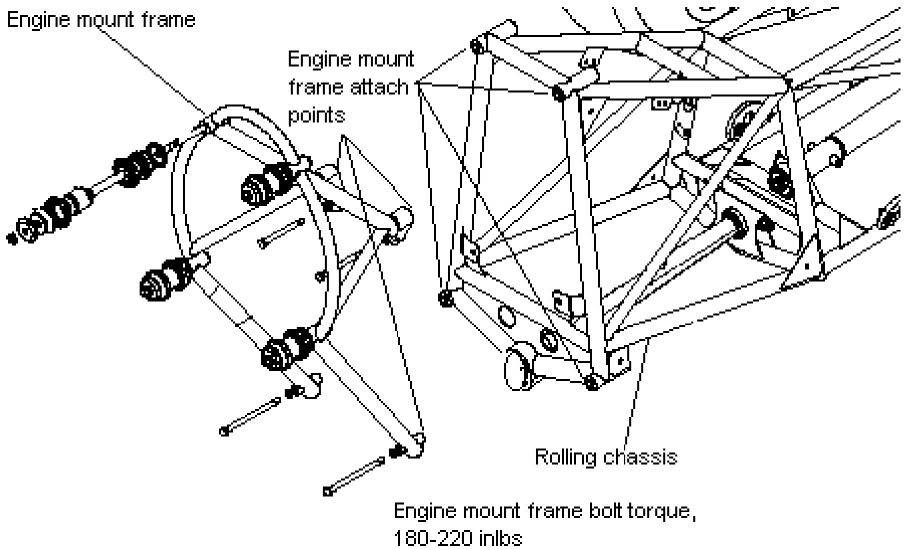

Figure 71-25 Engine Mount Frame Removal and Installation

2. Remove harness clamps securing ignition leads to the engine mount frame assembly. Retain for installation procedure.
3. Remove ground straps shown in Figure 71-26 connecting the engine mount frame to ECU-1 and ECU-2

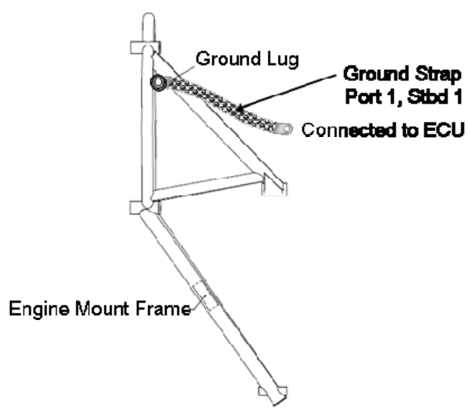

Figure 71-26 Engine Mount Frame Ground Straps

4. Remove four bolts, washers, and nuts securing engine mount frame to rolling frame and remove engine mount frame.
5. Remove engine isolators from engine mount frame in accordance with the Engine Mount Removal procedure on page 7 of this chapter.

This completes the Engine Mount Frame Removal procedure.

ENGINE MOUNT FRAME INSTALLATION

Perform this procedure to install the engine mount frame.

1. Refer to Figure 71-4. Install engine mount frame to rolling frame using 4 bolts, washers, and nuts. Torque the nuts in accordance with Chapter 20 – *Standard Practices*.
2. Back the nuts off to first castellated slot for split pins. Install new split pins.
3. Install engine mounts to engine mount frame in accordance with the Engine Mount Installation procedure on page 35 of this chapter.
4. Install engine in accordance with the Engine Installation procedure on page 7 of this chapter.

This completes the Engine Mount Frame Installation

ENGINE MOUNT AND FRAME INSPECTION

Perform Engine Mount and Frame Inspection in accordance with chapter 05 and on condition.

1. Visually inspect engine mount frame for any damage, bent tubes, bubbled paint, or damaged mounting hardware.
2. Visually inspect engine mounts for any bent or damaged isolators and attachments.
3. Ensure engine mounts are free of cracks, corrosion, or damage.
4. Check for proper alignment and fastening of the engine and airframe mounting points.
5. Sufficient clearance exists between engine and aircraft/engine mount.
6. All engine-to-airframe connections are flexible and correctly supported to prevent vibration transmission, chafing, and breakage.

This completes the Engine Mount and Frame Inspection procedure.

Section 20-02 Engine Mount Troubleshooting Guide

Use this troubleshooting guide to resolve issues with the engine.

Complaint	Possible Cause	Remedy
Isolator damage	• In service wear, age • Hard landing	• Remove and replace isolator
Mount frame tube cracks	• Hard landing	• Removal and replace frame
Paint scratched	• In service wear	• Refinish
Frame corrosion	• Surface finish deterioration	• Inspect, repair or replace in accordance with chapter 51
Engine mount misaligned	• Hard landing	• Inspect for damage • Reinstall IWA chapter 71 if serviceable.

Table 71-3 Engine Mount Troubleshooting Guide

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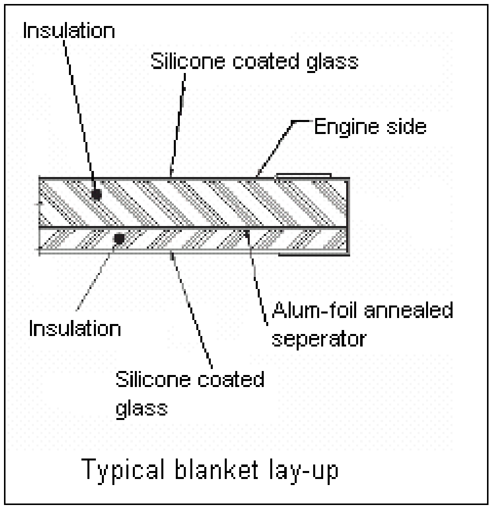
Section 71-30 Firewall Blanket

The removable engine firewall blanket is a fireproof barrier made specifically for the XL2 airplane. The blanket is constructed of silicone coated, aluminized, glass cloth. The layers are sealed on the edges with high temperature red RTV.



Typical blanket lay-up

Figure 71-27 Firewall Cutaway

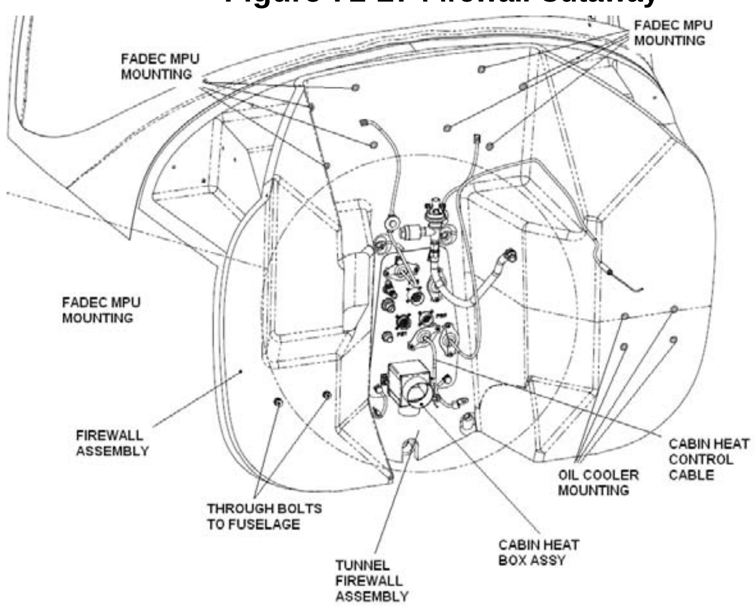


Figure 71-28 Engine Firewall Blanket Installation

NOTE

The firewall blanket is to be inspected for general condition. Any abnormalities such as tears, discoloration, or evidence of damage will be referred to Liberty Aerospace Customer Service for the corrective action procedure applicable to the condition.

Section 30-01 Firewall Blanket Procedures

This section contains the procedures to remove, install, and inspect the firewall blanket. Inspection is performed in accordance with the maintenance schedules in Chapter 05 – *Time Limits/Maintenance Checks/Inspection Intervals*

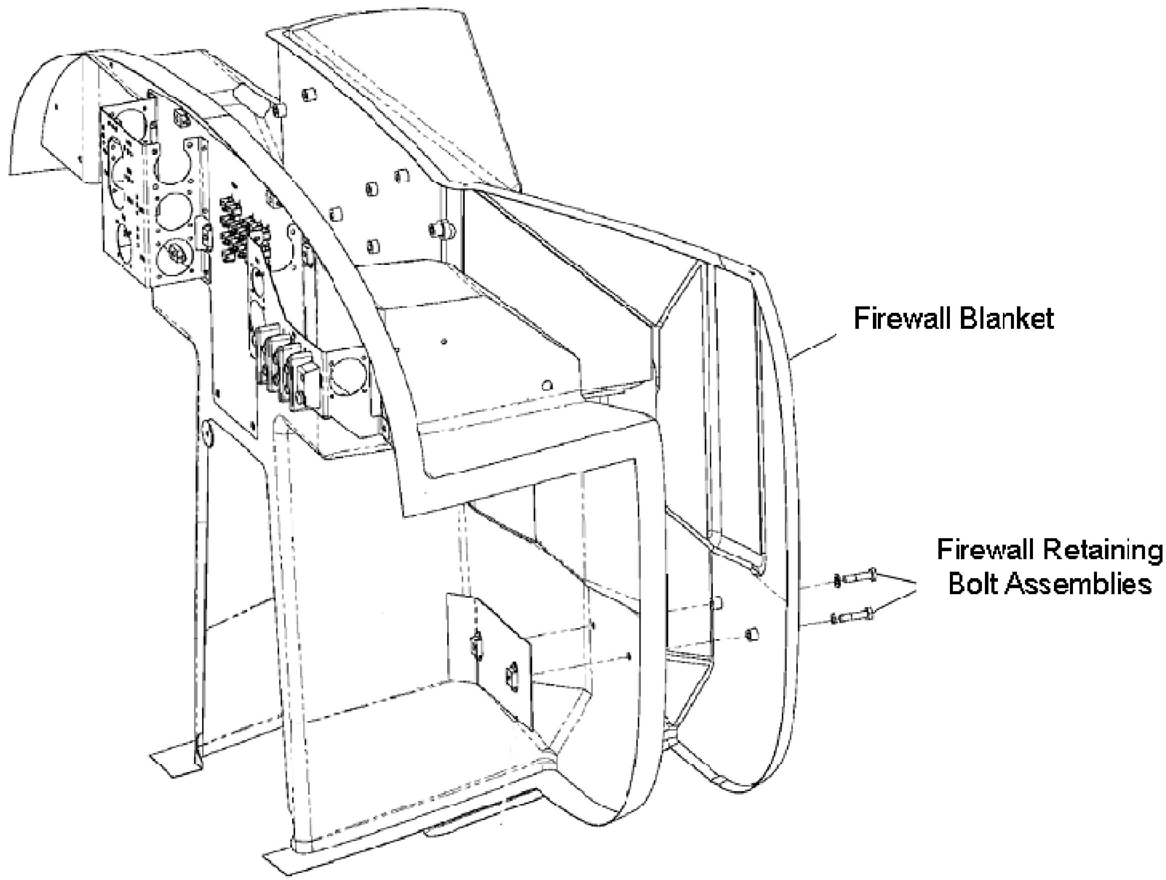


Figure 71-29 Firewall Blanket Installation

FIREWALL BLANKET REMOVAL

Perform this procedure to remove the firewall blanket.

1. Remove upper and lower cowlings in accordance with the Cowling Removal procedure on page 29 of this chapter.
2. Remove engine in accordance with the Engine Removal procedure on page 7 of this chapter.
3. Disconnect ignition leads from ECU-1 and ECU-2. Cap exposed lead ends and ECU terminal towers.
4. Perform Engine Mount Removal in accordance with the procedure on Page 34 of this chapter.
5. Disconnect the following from the Tunnel Firewall Assembly engine side as shown in Figure 71-30.

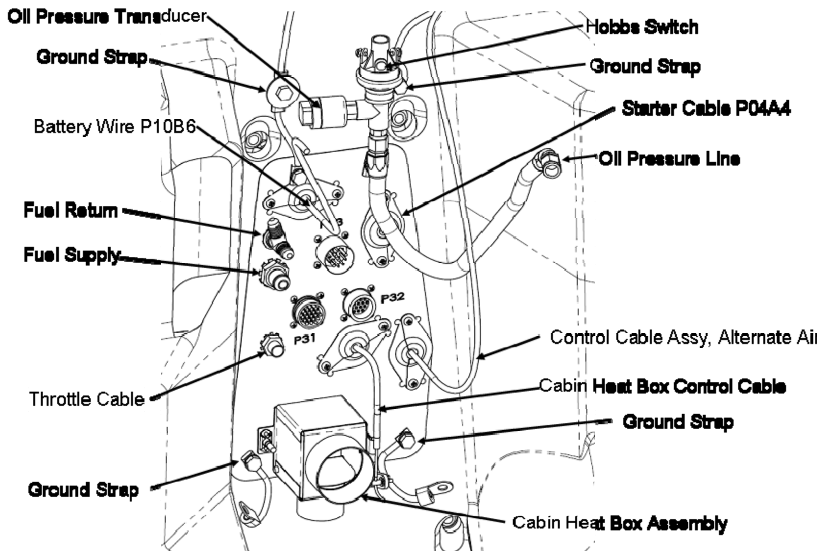


Figure 71-30 Tunnel Firewall Assembly Disconnects - Engine

6. Remove two bolts (1) and two washers (2) from cabin heat box assembly.
7. Disconnect cabin heat control cable from cabin heat box assembly, and remove heat box from firewall.

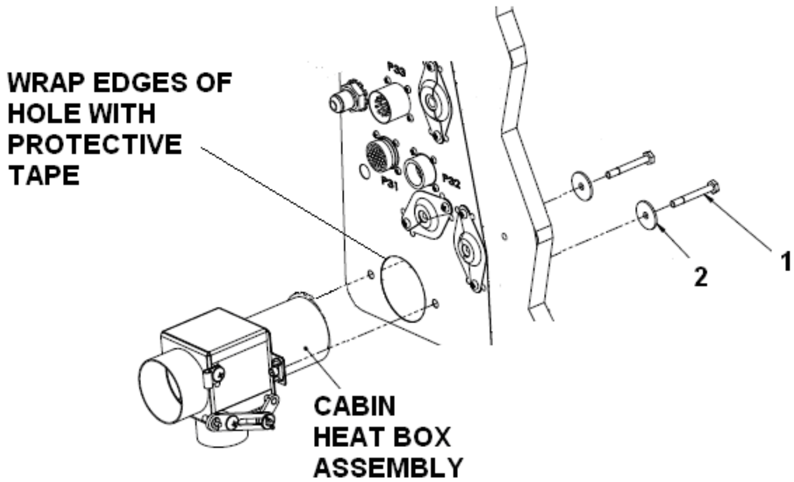


Figure 71-31 Cabin Heat Box Assembly Removal and Installation

8. Disconnect the Tunnel Firewall Assembly fuselage side components as shown in Figure 71-32.

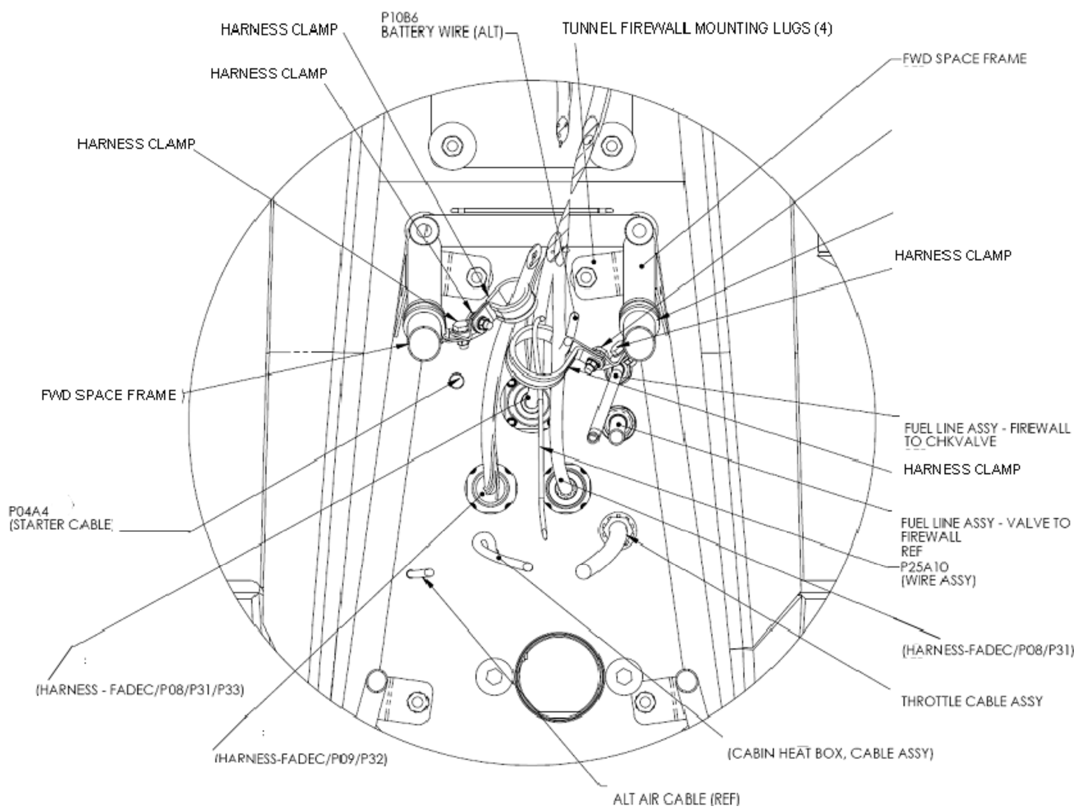


Figure 71-32 Tunnel Firewall Assembly Disconnects - Fuselage

9. Remove the Tunnel firewall Assembly from the Space Frame mounting lugs (4)
10. Remove oil cooler Bracket Assembly.

11. Remove FADEC ECU-1 and ECU-2.

12. Loosen and remove firewall-retaining bolts as shown in Figure 71-29. Note bolt and washer stack-up.

13. Remove firewall from fuselage.

This completes the Firewall Blanket Removal procedure

FIREWALL BLANKET INSTALLATION

Perform this procedure to install the firewall blanket.

NOTE

Ensure all washers placed in contact with surface of firewall blanket are coated with thin layer of high temperature RTV; P/N GE 106, Red High Temperature RTV.

1. Position the Firewall Blanket as shown in Figure 71-33 and secure with through bolts. Torque bolts in accordance with Chapter 20 – *Standard Practices.*

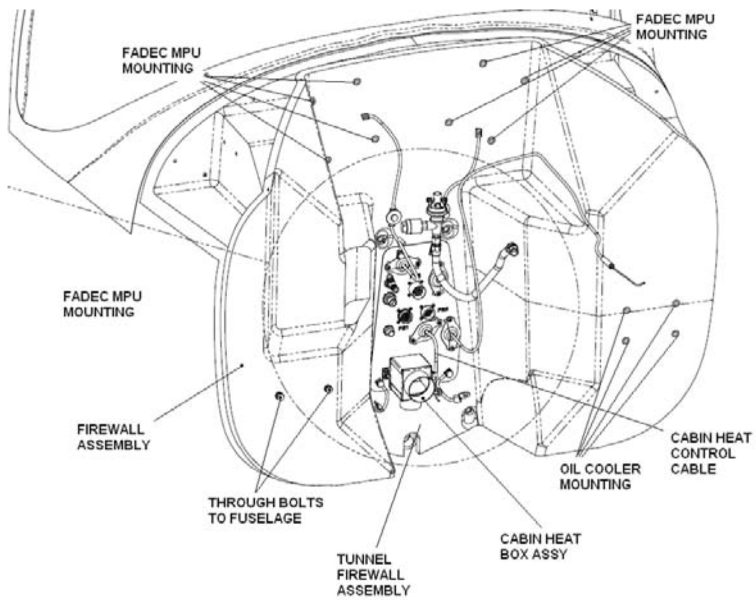


Figure 71-33 Firewall Blanket Installation

2. Install oil cooler bracket assembly. Torque bolts in accordance with Chapter 20 – *Standard Practices.*
3. Install FADEC ECU-1 and ECU-2 as shown in Figure 71-34. Torque bolts in accordance with Chapter 20 – *Standard Practices.*

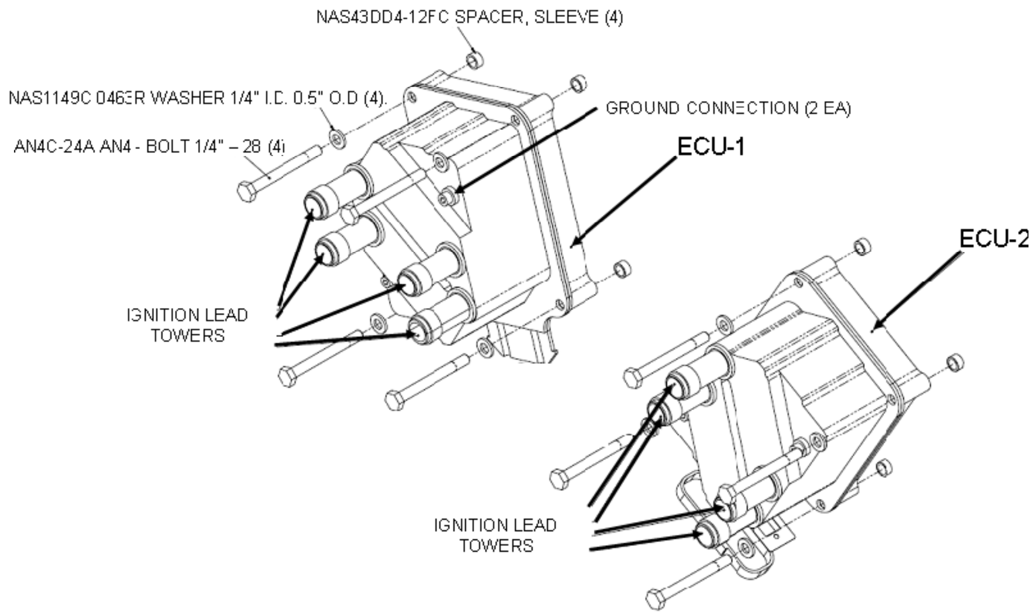


Figure 71-34 ECU Firewall Installation

NOTE

Caulk all fuel lines with high temperature RTV; P/N GE106, Red High Temperature RTV.

4. Connect the Tunnel Firewall Assembly fuselage side components as shown in Figure 71-35.

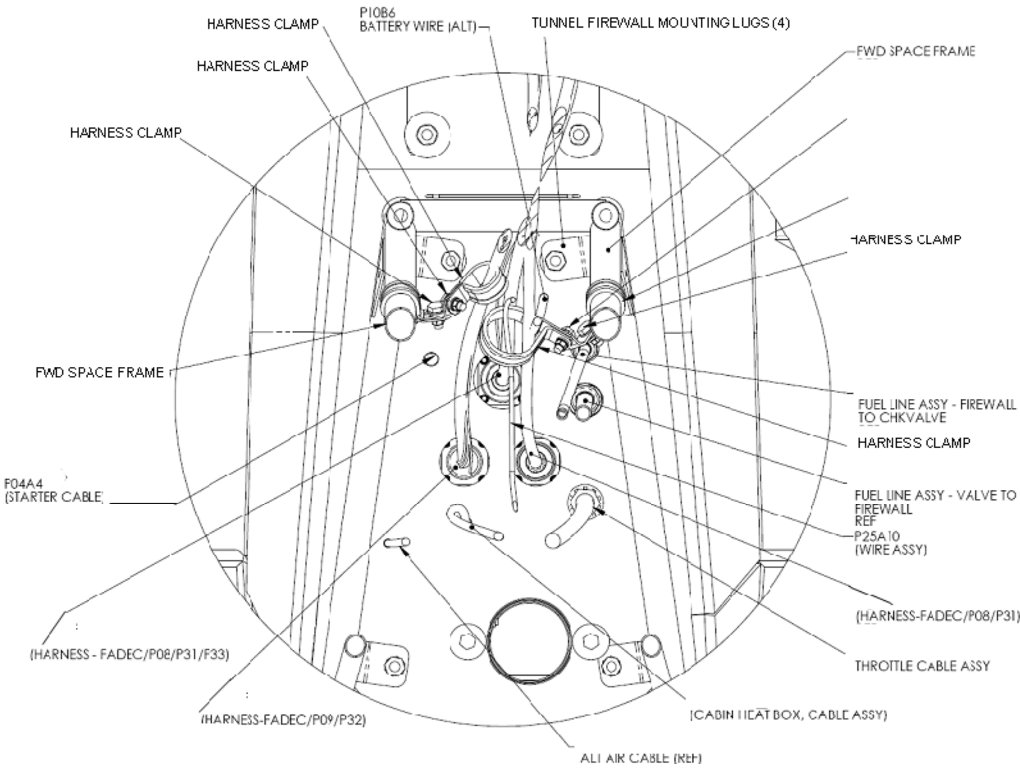


Figure 71-35 Tunnel Firewall Assembly Connects - Fuselage

5. Connect cabin heat control cable to cabin heat box assembly.

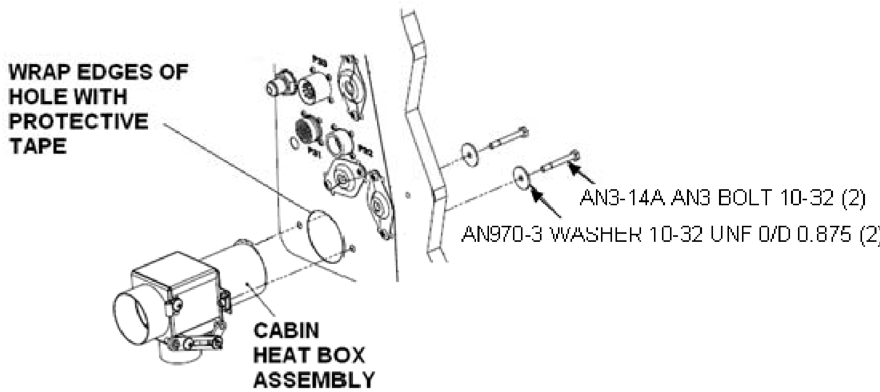


Figure 71-36 Cabin Heat Box Assembly Installation

6. Install two bolts and two washers as shown in Figure 71-36 to secure cabin heat box assembly.

7. Perform Engine Mount Installation in accordance with the procedure on Page 35 of this chapter.

NOTE

In the following step Tunnel Firewall Assembly connection on the engine facing side will take place. Connect only those components that do not require an installed engine to accomplish. Remaining connections will be made at engine installation.

8. Connect the following from the Tunnel Firewall Assembly engine side as shown in Figure 71-37. Torque all bolts to specifications for individual component installations. See Chapter 20 – *Standard Practices*.

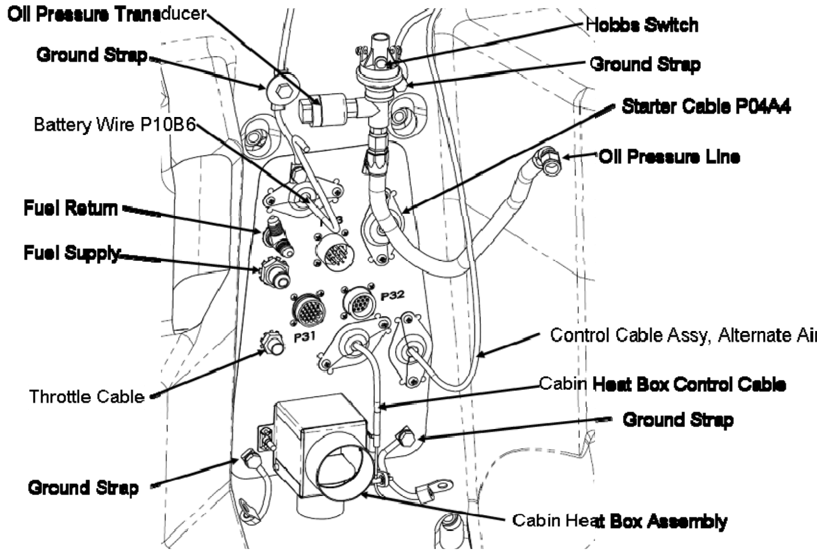


Figure 71-37 Tunnel Firewall Assembly Connects - Engine

9. Install engine in accordance with the Engine Installation procedure on page 14 of this chapter.
10. Install upper and lower cowlings in accordance with the Cowling Installation procedure on page 30 of this chapter.

This completes the Firewall Blanket Installation procedure.

FIREWALL BLANKET INSPECTION

Perform Firewall Blanket Inspection in accordance with Chapter 05 intervals or as required by condition.

1. Remove upper and lower cowlings in accordance with the Cowling Removal procedure on page 29 of this chapter.
2. Visually inspect condition of firewall blanket for tears or discoloration indicating excess heat exposure of the facing sheet materials or surrounding sealant materials as shown in Figure 71-38. Tears or discoloration is unacceptable.

NOTE

The firewall blanket is to be inspected for general condition. Any abnormalities such as tears, discoloration, or evidence of damage will be referred to Liberty Aerospace Customer Service for the corrective action procedure applicable to the condition.

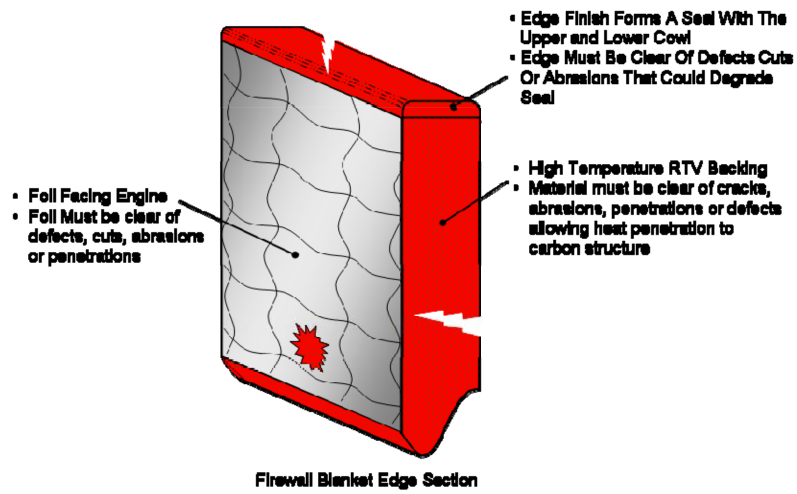


Figure 71-38 Firewall Blanket Edge Section

3. Inspect firewall blanket around cabin heat valve box for tight fit or any blanket settling, reference Chapter 21 – *Environmental Systems* for additional information.
4. Inspect all Firewall Blanket penetrations for damage or loss of high temperature caulk insulation.
5. Inspect Firewall Blanket cowl contacting edges for damage. Edges
6. Install lower and upper cowl in accordance with the Cowling Inspection procedure on page 7 of this chapter.

This completes the Firewall Blanket Inspection procedure.

Section 30-02 Firewall Blanket Troubleshooting Guide

Use this troubleshooting guide to resolve issues with the engine.

Complaint	Possible Cause	Remedy
Tears or discoloration	• In service wear • Excess heat exposure	• Replace blanket
Heat box fit loose	• Loss of sealant	• Reseal heat box
	• In service blanket wear	• Replace blanket
Worn edge sealing RTV	• Cowl chafe	• Redress RTV material with GE406, Red High Temp.

Table 71-4 Firewall Blanket Troubleshooting Guide

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Section 71-50 Engine Electrical Harness

Many of the sensor and control functions of the FADEC system are linked to airframe components by an electrical harness called the Low Voltage Harness. This harness is connected to airframe systems and components through two “Cannon plug” bulkhead connectors. These connectors are located on the firewall.

Additional circuits including starter and alternator wires, oil pressure and temperature sensors, etc., are connected through individual wires on the engine and connectors on the firewall.

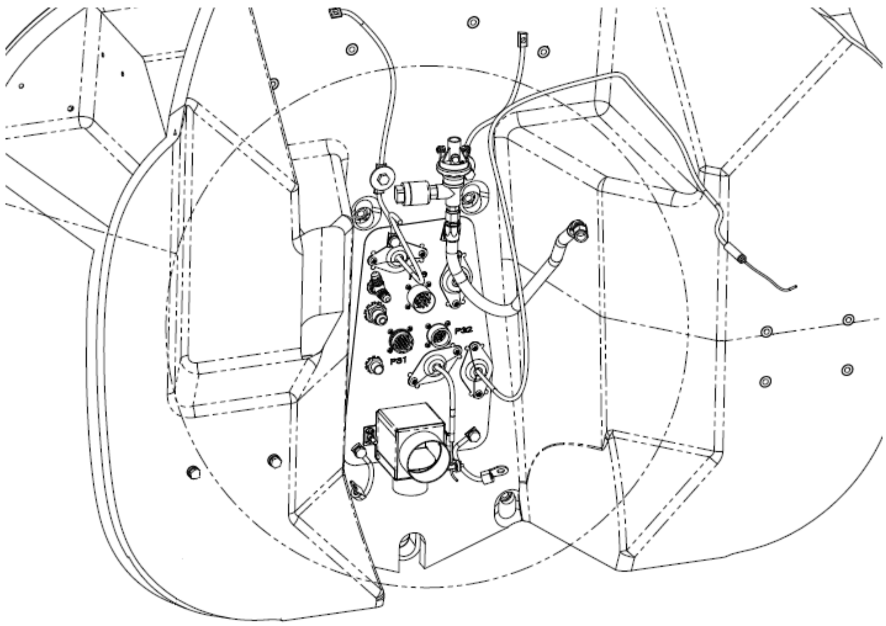


Figure 71-39 Firewall Interface Feed-through Panel

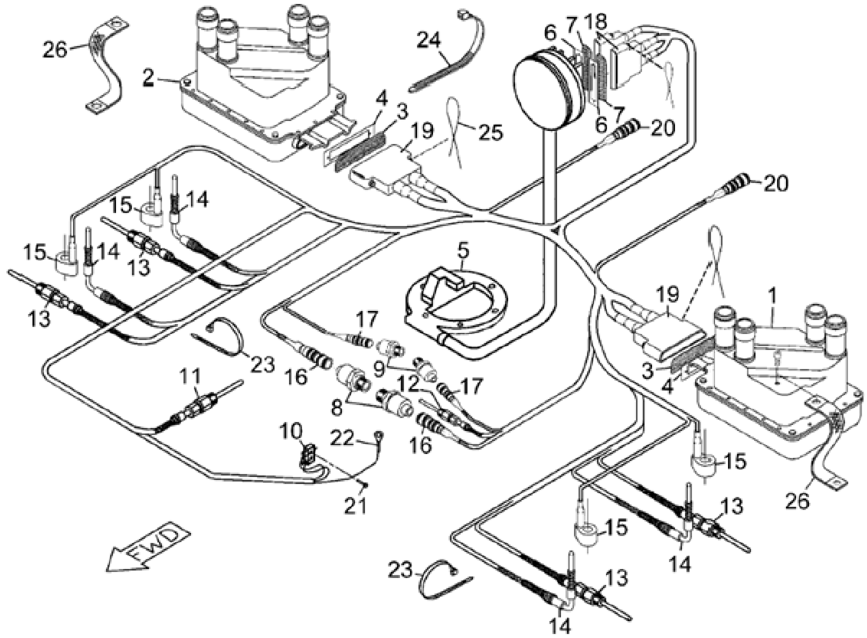
Section 50-01 FADEC Low Voltage Harness Description

The low voltage harness connects all essential components of the FADEC System. This harness acts as a signal transfer buss interconnecting the two Electronic Control Units (ECUs) with aircraft power sources, the Ignition Switch, Speed Sensor Assembly (SSA), Health Status Annunciator (HSA) or two-light indicator panel,

Temperature and pressure sensors, the fuel injector coils and all sensors, except the SSA and Fuel Pressure and Manifold Pressure Sensors, are hardwired to the low voltage harness.

This harness transmits sensor input to the ECUs via a 50-pin connector. The harness connects to the engine mounted pressure sensors via cannon connectors. The 25-pin connectors connect the harness to the speed sensor signal conditioning unit. The low voltage harness attaches to the cabin harness by firewall-mounted bulkhead fittings or connectors. Information from the ECUs is conveyed to the HSA and the cockpit-mounted data port through the same cabin harness/bulkhead connector assembly.

Figure 71-40 shows the low voltage harness installed on an IOF-240-B series engine.



1. Electronic Control Unit
2. Electronic Control Unit
3. Sealant Strip
4. EMI Gasket
5. Speed Sensor
6. EMI Gasket
7. Sealant Strip
8. MAP Sensor
9. Fuel Pressure Sensor
10. Throttle Position Switch (TPS)
11. MAT Sensor
12. MAT Sensor
13. EGT Sensor
14. CHT Sensor
15. Fuel Injector Solenoid
16. Six-pin connector
17. Six-pin connector
18. Twenty-five-pin Connector
19. Fifty-pin Connector
20. Nineteen-pin Connector
21. Screw
22. Ground Terminal
23. Cable Tie
24. Safety Tie
25. Lockwire
26. Ground Strap

Figure 71-40 Low Voltage Harness on IOF-240-B Series Engine

Section 50-02 FADEC Ignition System Harness

The Ignition System consists of the high voltage harness and spark plugs. Figure 71-41 shows the high voltage harness for a four-cylinder engine.

Since there are two spark plugs per cylinder on all engines, a four-cylinder engine has eight leads and eight spark plugs.

One end of each lead on the high voltage harness attaches to a spark plug and the other end of the lead wire attaches to the spark towers on each ECU. The spark tower pair is connected to opposite ends of one of the ECU's coil packs. Two coil packs are located in the upper portion of the ECU.

The high voltage harness carries energy from the ECU spark towers to the spark plugs on the engine.

Electronic Control Unit 1 fires the top and bottom spark plugs for Cylinders 1 & 2; ECU 2 fires the top & bottom spark plugs for Cylinders 3 & 4.

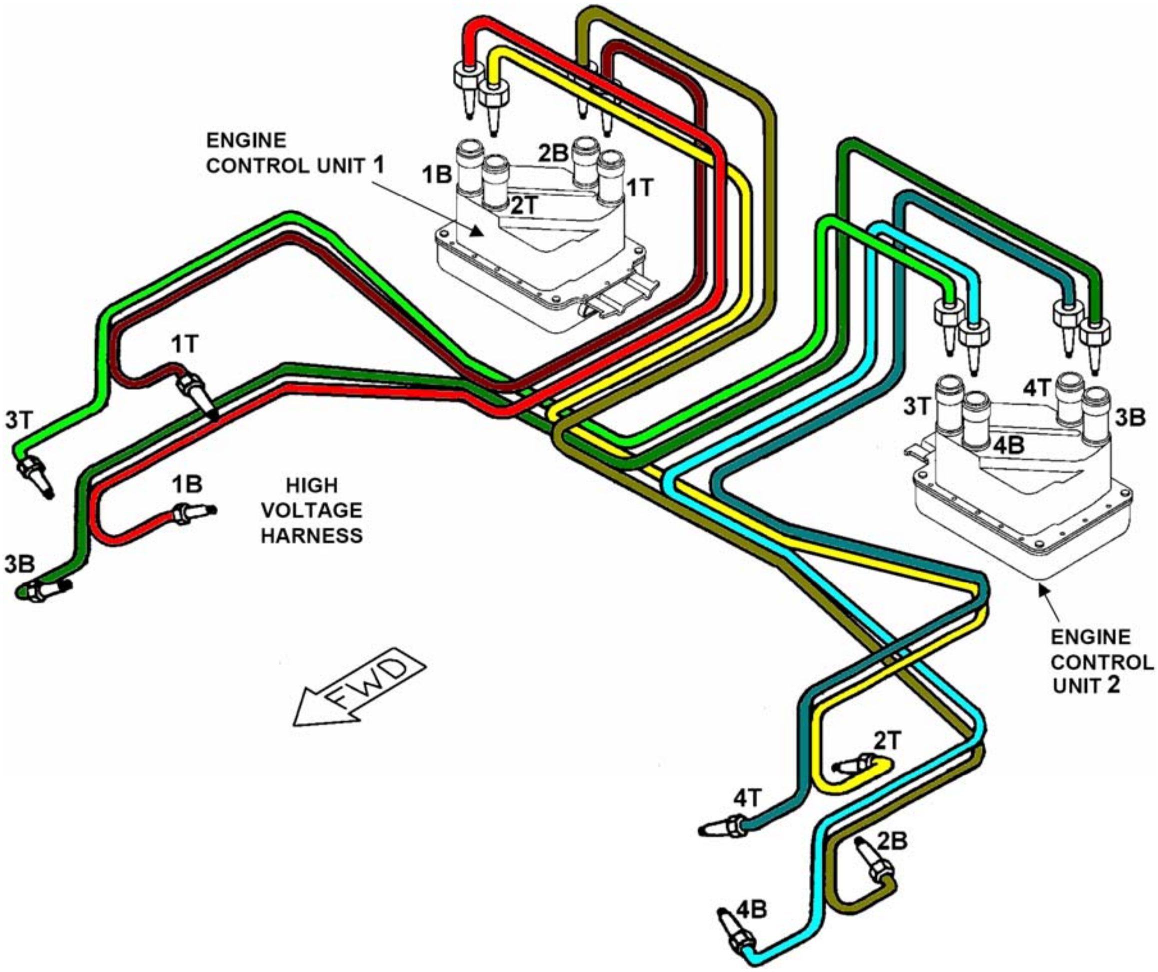


Figure 71-41 FADEC Ignition System Harness

Section 50-03 FADEC Harness Assembly Procedures

This section contains the procedures for the FADEC harness assemblies. As described previously, there are two principle FADEC harnesses, the ignition system harness and the low voltage harness. Procedures to follow provide remove, replace, test and inspection procedures for each harness system. Selected procedures may be performed in accordance with schedules described in Chapter 05 or on condition.

IGNITION SYSTEM HARNESS REMOVAL

Perform the following procedure to remove ignition harness leads from the engine. Procedure applies to any one or all eight leads.

1. Locate and remove all “P” clamps securing the ignition lead to engine baffle and structure. Inspect each clamp location for evidence of damage
2. Restrain the lead wire ferrule to prevent kinking of the lead.
3. Rotate the B-nut until is free of the barrel.
4. Extract the lead by pulling straight back from the barrel.

CAUTION

Do not side load or twist the lead while removing it from the barrel damage to the silicone insulator may result. A gentle rocking force will be sufficient to free the lead end.

5. Route the lead out of the engine installation taking care not to disturb adjacent leads and harnesses.
6. Repeat the above steps for all other ignition leads to be removed.

This completes the

Ignition System Harness Removal Procedure

IGNITION SYSTEM HARNESS INSTALLATION

Perform the following procedure to install ignition harness leads to the engine. Procedure applies to any one or all eight leads.

1. Route the ECU end of the ignition lead wires to the appropriate ECU. Each ignition lead wire has a metal identification (ID) tag that designates the cylinder and spark plug assignment. For example, 2T designates the lead is for Cylinder 2 top spark plug and 2B is for Cylinder 2 bottom spark plug.

Figure 71-42 High Voltage Harness Spark Plug Attachment

2. Spray a small amount of MS 122 DF Spray in each ECU spark tower.
3. Connect each high voltage harness lead wire to the appropriate ECU spark tower. Each spark tower is identified with a two-digit code of the same format as the ignition lead wire ID tags (Figure 71-43).

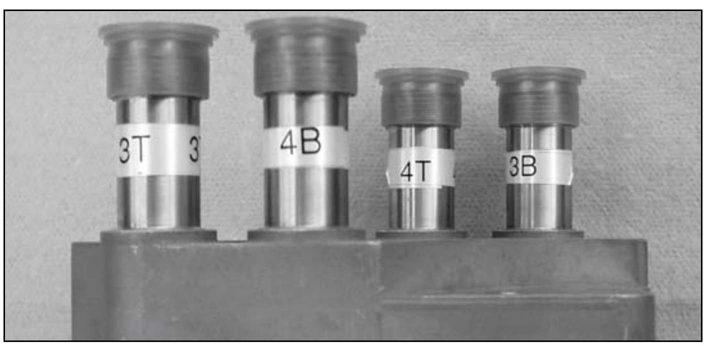


Figure 71-43 Ignition Lead Wire Connections to ECUs

4. To prevent the ignition lead wire sleeves from sticking and to minimize twisting of the ferrule, coat the insulating sleeves with MS 122 DF Spray (Figure 71-44).

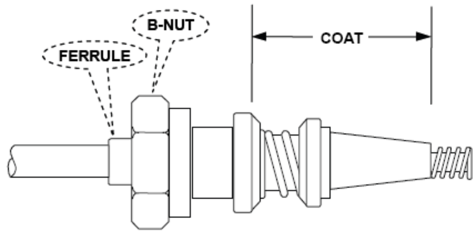


Figure 71-44 Coating Insulating Sleeve: MS 122 DF Spray on Ignition Lead Wire

5. Connect the lead wires as follows:

- Insert the spring-end of the lead into the spark tower.
- While holding the ferrule, firmly push the rubber insulator into the tower.
- When the B-nut thread makes contact with the spark tower threads, turn the B-nut clockwise.
- Restrain the lead wire ferrule to prevent kinking of the lead.
- Continue rotating the B-nut until it seats.
- Torque the B-nut per Teledyne Continental Motors M-22 Maintenance Manual, Appendix B.

Do not allow the ignition lead wire ferrule to twist while tightening the B-nut. If twisting is observed, hold the knurled portion of the B-nut shoulder with a wrench while tightening the B-nut.

6. Using “P” clamps removed previously, route leads as far away as possible from exhaust pipes to ensure they are not exposed to temperatures in excess of 400°F (204°C).

7. Route leads so as to prevent chafing and damage due to vibration. Refer to Appendix C of Teledyne Continental Motors M-22 Maintenance Manual for guide lines.

This completes the Ignition System Harness Installation procedure

IGNITION SYSTEM HARNESS INSPECTION PROCEDURE

Perform this procedure to inspect the ignition system.

1. Perform the FADEC Level I Diagnostic in accordance with Teledyne Continental Motors M-22 Maintenance Manual.
2. Visually inspect the ignition leads for chafing, deterioration, and insulation breakdown. Worn or frayed ignition wire must be replaced.
3. Verify high voltage leads are securely fastened to the ECU spark towers.

CAUTION

Failure to maintain the ignitions leads can cause engine damage or failure.

4. Verify high voltage leads are securely fastened to the spark plugs
5. Inspect engine ignition lead “P: clamps for damage or chafing of ignition leads.

This completes the Ignition System Harness Inspection Procedure

LOW VOLTAGE HARNESS REMOVAL

Perform this procedure to remove the low voltage harness.

1. Position split master switch – OFF
2. Position BAT1 circuit breaker – OPEN
3. Position FADEC A and B power switches - OFF

CAUTION

Power is to remain OFF while the low voltage harness is removed.

4. Remove upper and lower cowl in accordance with the Cowling Removal procedure on page 29 of this chapter.
5. Remove engine baffle system in accordance with Chapter 75 – *Air Inductions*.
6. Remove the low voltage wire harness in accordance with Teledyne Continental Motors IOF-240 Overhaul Manual OH-22, Chapter 6.

This completes the Low Voltage Harness Removal procedure

LOW VOLTAGE HARNESS INSTALLATION

Perform this procedure to install the low voltage harness.

NOTE

Additional detailed information on this subject may be found in Teledyne Continental Motors IOF-240 Overhaul Manual OH-22, Chapter 11.

The engine low voltage harness (Figure 71-40) connects essential components of the FADEC System to the ECUs. Sensors are hardwired to the harness, except for the speed sensor array (SSA) and pressure sensors. When all connectors are properly routed connected, refer to “Harness Routing” in Appendix C of the Teledyne Continental Motors Overhaul manual OH-22 to complete the harness installation.

CAUTION

The engine low voltage harness is shipped with protective covers installed on all connectors to prevent damage and contamination. Remove the covers at the time of installation.

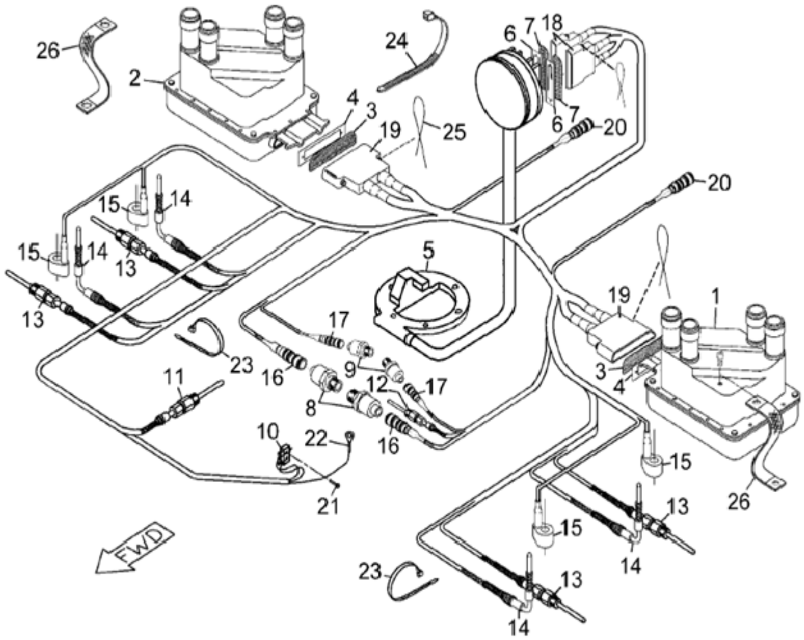
1. Position split master switch – OFF
2. Position BAT1 circuit breaker – OPEN
3. Position FADEC A and B power switches - OFF
4. if not already performed, remove engine baffle in accordance with Chapter 75 – *Air Induction* procedures.
5. Install low voltage wire harness in accordance with Teledyne Continental Motors IOF-240 Overhaul Manual OH-22, Chapter 11.
6. Install engine baffle system in accordance with Chapter 75 – *Air Induction* procedures.
7. Install upper and lower cowl in accordance with the Cowling Installation procedure on page 30 of this chapter.
8. Position BAT (CB001) circuit breaker – CLOSED.
9. Perform Inspection and engine functional check in accordance with the Engine Removal procedure on page 21 of this chapter.

This completes the Low Voltage Harness Installation procedure.

LOW VOLTAGE HARNESS INSPECTION

Perform this procedure to inspect the low voltage harness.

1. Inspect each low voltage path for indications of insulation abrasion or other physical damage. Refer to Figure 71-45.



1. Electronic Control Unit
2. Electronic Control Unit
3. Sealant Strip
4. EMI Gasket
5. Speed Sensor
6. EMI Gasket
7. Sealant Strip
8. MAP Sensor
9. Fuel Pressure Sensor
10. Throttle Position Switch (TPS)
11. MAT Sensor
12. MAT Sensor
13. EGT Sensor
14. CHT Sensor
15. Fuel Injector Solenoid
16. Six-pin connector
17. Six-pin connector
18. Twenty-five-pin Connector
19. Fifty-pin Connector
20. Nineteen-pin Connector
21. Screw
22. Ground Terminal
23. Cable Tie
24. Safety Tie
25. Lockwire
26. Ground Strap

Figure 71-45 Low Voltage Harness Inspection

2. Inspect speed sensor connectors for security and indication of contamination.
3. Inspect ECU connectors for security and indication of contamination.
4. Inspect CHT probe connections (4) for security, indications or wear or contamination.
5. Inspect EGT probes (4) for damage, excess bending or contamination.
6. Inspect MAP probes (2) for connector security or evidence of contamination.
7. Inspect fuel pressure sensor connectors for security or evidence of contamination.
8. Inspect injector solenoids for security, damage, or evidence of contamination.

9. Inspect WOT switch installation for follower wear, cam corrosion or connector damage.
10. Perform engine function test in accordance with the Engine Operational Check procedure on page 21 of this chapter.

This completes the Low Voltage Harness Inspection procedure

Section 50-04 Engine Electrical Harness Troubleshooting Guide

Use this troubleshooting guide to resolve issues with the engine.

Complaint	Possible Cause	Remedy
Abnormal RPM or Performance Drop	• Ignition Lead Fault	• Perform diagnostics in accordance with TCM Maintenance Manual M-22 Chapter 8
	• Spark plug fault	
	• ECU fault	
Abraded ignition lead	• Faulty “P” clamp	• Replace ignition lead • Replace faulty “P” Clamp
	• Abrading on adjacent structure	• Replace ignition lead • Correct routing fault
FADEC Power On Faults	• FADEC system failure	• Run Level I diagnostic in accordance with TCM Maintenance Manual M-22 Chapter 8 and follow fault tree
	• Primary battery failure	• Replace primary battery
	• Secondary battery failure	• Replace secondary battery
	• Alternator failure	• Replace alternator
	• Low Voltage harness fault as a result of routing fault	• Repair or replace low voltage harness • Correct harness routing

Table 71-5 Engine Electrical Harness Troubleshooting Guide

Section 71-60 Air Intakes

Two openings in the engine cowling, on either side of the propeller, admit air to the engine compartment. Baffles secured to the engine, and sealing against the interior of the cowling, direct the majority of this air past the cylinder cooling fins.

The normal intake for engine combustion air is through an air filter located on the top of the engine (on the high pressure / low temperature side of the engine baffles). Alternate air, when selected by the pilot, enters the engine from below the engine (on the high temperature / low pressure side of the engine baffles).

A duct on the left side of the firewall supplies high pressure / low temperature air to the oil cooler. Air leaving the oil cooler will exit through the aft opening in the lower cowling.

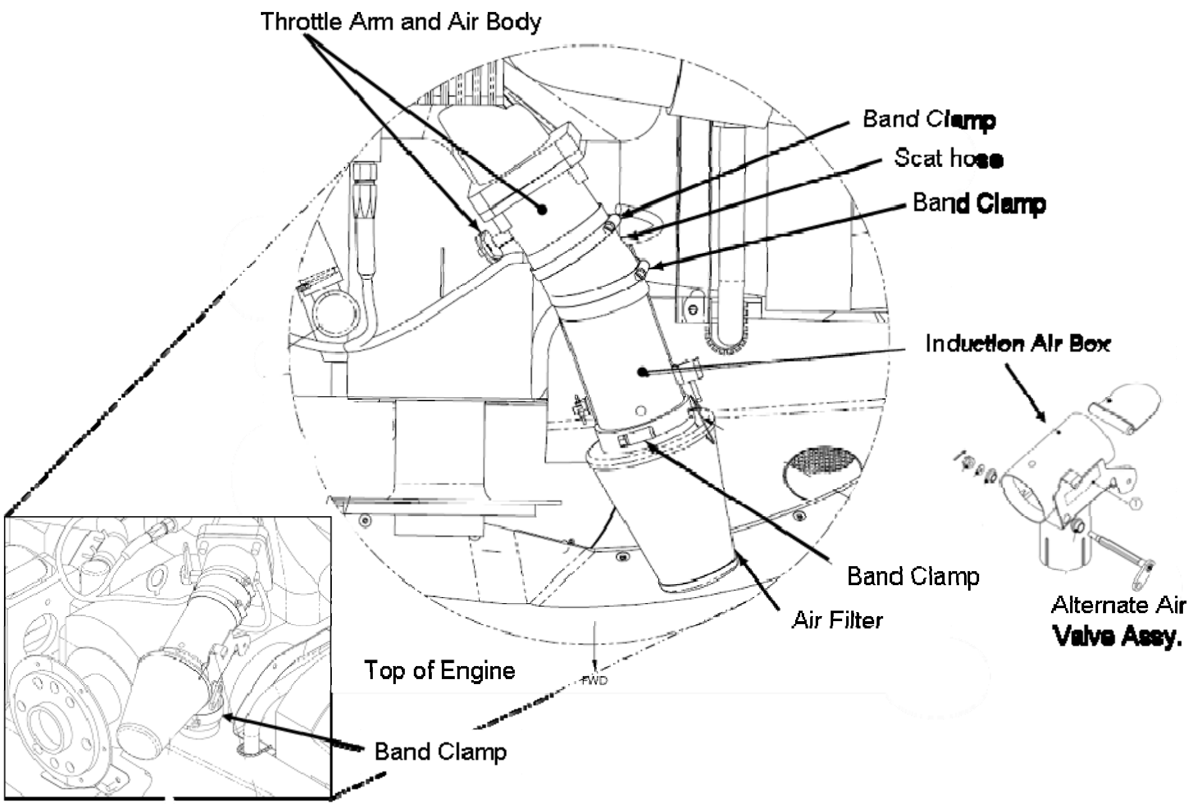


Figure 71-46 Air Induction System

Section 60-01 Air Intake Procedures

This section contains the procedures to remove and install the air intake system.

AIR INTAKE REMOVAL

Perform this procedure to remove the air intake.

11. Position the aircraft split master switch – OFF
12. Position BAT1 circuit breaker – OPEN
13. Position SYSTEM, START circuit breaker – OPEN
14. Remove upper cowl in accordance with the Cowling Removal procedures on page 29 of this chapter.
15. Remove air filter by removal of .010 safety wire and the loosening of securing band clamp as shown in Figure 71-46.
16. Disconnect alternate air cable by removal of alternate air arm bolt and nut hardware as shown in Figure 71-47.

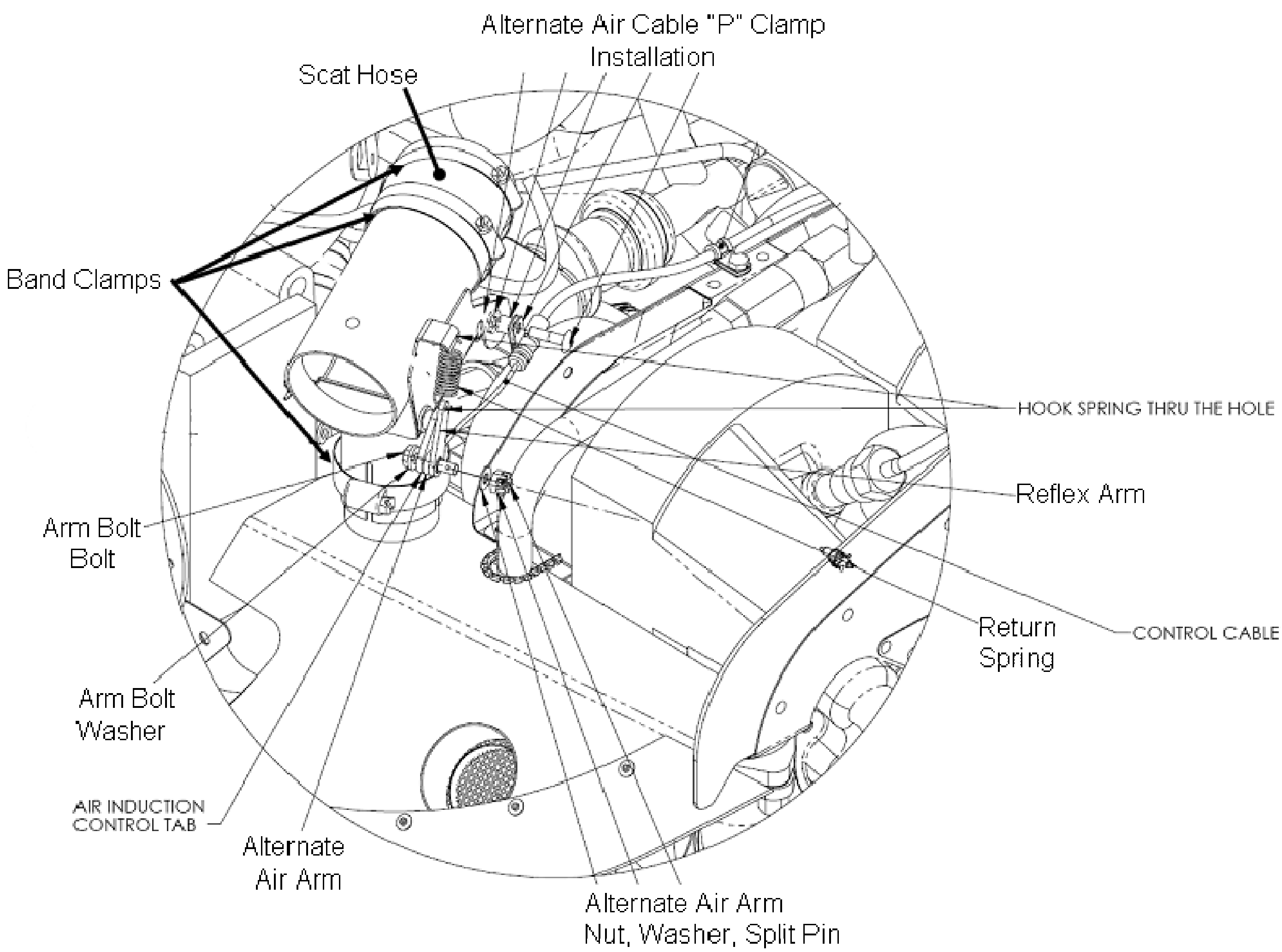


Figure 71-47 Alternate Air Control Installation

17. Remove alternate air cable “P” clamp bolt, nut, washer and spacer hardware as shown in Figure 71-47 and position cable clear of assembly.
18. Loosen three (3) band clamps securing the alternate air box assembly to the forward baffle and to the engine induction manifold.
19. Slide scat tube clear of induction manifold
20. With a gentle upward motion slide the air box assembly off of the forward baffle flange and clear of the engine installation.

This completes the Air Intake Removal procedure.

AIR INTAKE INSTALLATION

Perform this procedure to install the air intake.

1. Inspect the alternate air box for debris, flapper valve freedom of movement and return spring function.
2. Slide scat tube over alternate air box in the location indicated in Figure 71-47.
3. Position band clamp over the air box end of the scat tube and secure
4. Pre-position three (2) band clamps on the alternate air box assembly as shown in Figure 71-47. One to secure scat tube to the engine induction manifold and one to secure the alternate air box to the forward engine baffle.
5. Slide the alternate air box assembly over the forward baffle flange and position inlet tube to align with engine induction manifold.
6. Slide lower band clamp down over the alternate air box tube split to appoint half way down the split. Tighten the band clamp in this position.
7. Slide the scat tube over the manifold inlet and secure with band clamp.
8. Connect the alternate air control cable end to the flapper valve arm with bolt, nut and washer removed previously. Do not install a new split pin at this time.
9. Install alternate air cable "P" clamp as shown in Figure 71-47 using bolt, nut, washer, and spacer removed previously.
10. From the cabin control, exercise the alternate air flapper valve and verify full range of motion. Adjust the alternate air arm cable as required to achieve full range of motion.
11. Secure the connection for the alternate air arm cable with a new split pin.
12. Install the air filter by sliding the band clamp loosely over the filter opening. Slide the filter into the alternate air box assembly in let tube. Secure the filter band clamp by tightening.
13. Install .020 in safety wire between the filter base and alternate air arm-mounting bracket.
14. Install upper cowl in accordance with the Cowling Installation procedure on page 30 of this chapter.

This completes the Air Intake Removal procedure

ALTERNATE AIR FLAPPER VALVE ARM REMOVAL

Perform this procedure to remove the arm for the alternate air flapper valve.

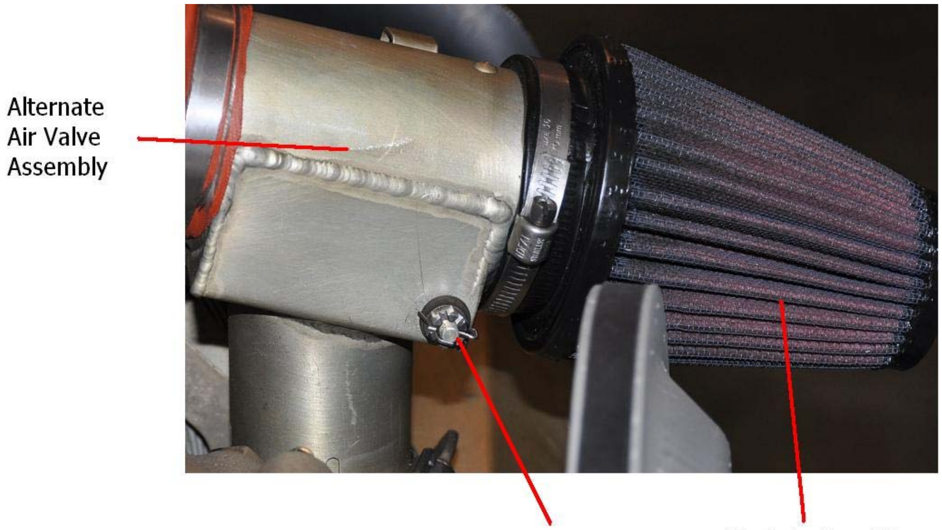


Figure 71-48 Alternate Air Valve Assembly

1. Position the aircraft split master switch – OFF
2. Position BAT1 circuit breaker – OPEN
3. Position SYSTEM, START circuit breaker – OPEN
4. Remove upper cowl in accordance with the Cowling Removal procedures on page 29 of this chapter.
5. Cut and discard the safety wire securing the air induction filter.
6. Loosen the clamp securing the filter to the alternate air valve assembly. Remove the filter.
7. Remove and discard the split pin securing the nut that holds the cable of the alternate air valve cable to the valve arm.
8. Loosen the nut sufficiently to allow the cable to be disconnected from the valve arm.
9. Remove and discard the split pin securing the valve arm to the alternate air assembly.
10. Loosen and remove the nut and washer from the valve arm.
11. Slide the arm out of the alternate air assembly.

This completes the Alternate Air Flapper Valve Arm Removal procedure.

ALTERNATE AIR FLAPPER VALVE ARM INSTALLATION

Perform this procedure to install the arm for the alternate air flapper valve.

1. Position the aircraft split master switch – OFF
2. Position BAT1 circuit breaker – OPEN
3. Position SYSTEM, START circuit breaker – OPEN
4. Remove upper cowl in accordance with the Cowling Removal procedures on page 29 of this chapter.
5. If the filter is mounted on the alternate air assembly, loosen the clamp securing the filter to the alternate air valve assembly. Remove the filter.
6. Before inserting the valve arm, inspect the threads for any damage.
7. Align the valve arm as shown in the Figure 71-49. Insert the valve arm in to the alternate air assembly. The arm must go through the flapper valve.

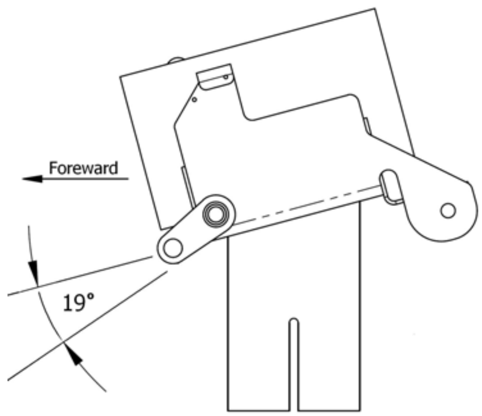


Figure 71-49 Install The Flapper Arm With The Control Tab Pointing Forward

8. Install the flat washer and castle nut.
9. Apply torque to the castle nut in accordance with Chapter 20 –*Standard Practices*.
10. Install a new split pin in accordance with Chapter 20 –*Standard Practices*.
11. Rotate the valve arm through the entire range. Movement should be smooth and without binding. The valve flapper should seat over both openings.
12. Connect the alternate air control cable end to the flapper valve arm with bolt, nut and washer removed previously. Do not install a new split pin at this time.
13. Install alternate air cable "P" clamp as shown in Figure 71-47 using bolt, nut, washer, and spacer removed previously.
14. From the cabin control, exercise the alternate air flapper valve and verify full range of motion. Adjust the alternate air arm cable as required to achieve full range of motion.

15. Secure the connection for the alternate air arm cable with a new split pin.
16. Install the air filter by sliding the band clamp loosely over the filter opening. Slide the filter into the alternate air box assembly in let tube. Secure the filter band clamp by tightening.
17. Install .020 in safety wire between the filter base and alternate air arm-mounting bracket.
18. Install upper cowl in accordance with the Cowling Installation procedure on page 30 of this chapter.

This completes the Air Intake Removal procedure

AIR INTAKE INSPECTION

Perform this procedure to inspect the air intake.

1. Inspect air filter for condition and time in service in accordance with Chapter 05 - Time Limits/Maintenance Checks/Inspection Intervals and Chapter 04 - Airworthiness Limitations or this manual.
2. Inspect alternate air assembly for corrosion or damage.
3. Remove the alternate air valve arm in accordance with the Alternate Air Flapper Valve Arm Removal procedure on page 69 of this chapter.
4. Inspect the threads on the shaft to insure there is no damage to the threads on the valve arm.
5. Install the valve are in accordance with the Alternate Air Flapper Valve Arm Installation on page 70 of this chapter.
6. Cycle alternate air flapper valve and verify full range of motion.
7. Inspect alternate air cable for binding
8. Inspect scat tube for damage and clamp security
9. Inspect baffle band clamp for security.
10. Inspect air filter safety wire for security.

This completes the Alternate Air Flapper Valve Arm Removal

Perform this procedure to remove the arm for the alternate air flapper valve.

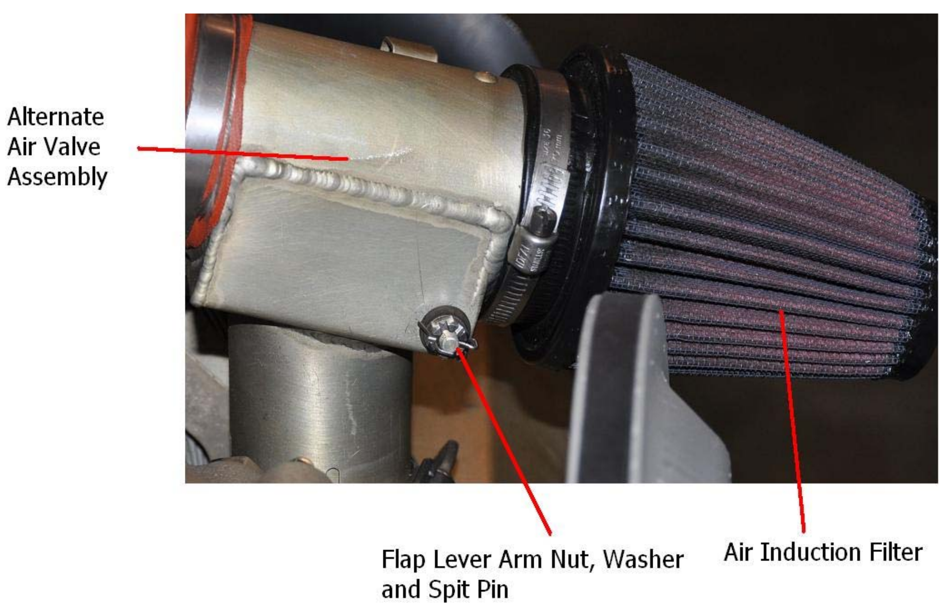


Figure 71-50 Alternate Air Valve Assembly

11. Position the aircraft split master switch – OFF
12. Position BAT1 circuit breaker – OPEN
13. Position SYSTEM, START circuit breaker – OPEN

14. Remove upper cowl in accordance with the Cowling Removal procedures on page 29 of this chapter.
15. Cut and discard the safety wire securing the air induction filter.
16. Loosen the clamp securing the filter to the alternate air valve assembly. Remove the filter.
17. Remove and discard the split pin securing the nut that holds the cable of the alternate air valve cable to the valve arm.
18. Loosen the nut sufficiently to allow the cable to be disconnected from the valve arm.
19. Remove and discard the split pin securing the valve arm to the alternate air assembly.
20. Loosen and remove the nut and washer from the valve arm.
21. Slide the arm out of the alternate air assembly.

This completes the Alternate Air Flapper Valve Arm Removal procedure.

ALTERNATE AIR FLAPPER VALVE ARM INSTALLATION

Perform this procedure to install the arm for the alternate air flapper valve.

22. Position the aircraft split master switch – OFF
23. Position BAT1 circuit breaker – OPEN
24. Position SYSTEM, START circuit breaker – OPEN
25. Remove upper cowl in accordance with the Cowling Removal procedures on page 29 of this chapter.
26. If the filter is mounted on the alternate air assembly, loosen the clamp securing the filter to the alternate air valve assembly. Remove the filter.
27. Before inserting the valve arm, inspect the threads for any damage.
28. Align the valve arm as shown in the Figure 71-49. Insert the valve arm in to the alternate air assembly. The arm must go through the flapper valve.

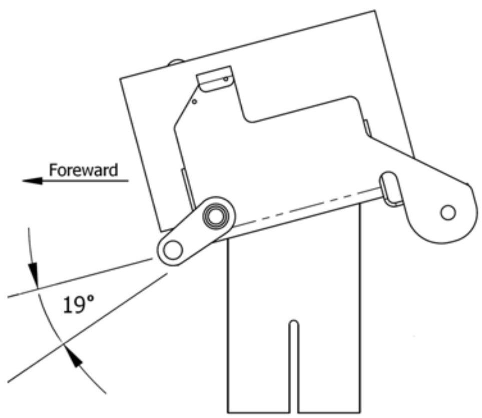


Figure 71-51 Install The Flapper Arm With The Control Tab Pointing Forward

29. Install the flat washer and castle nut.
30. Apply torque to the castle nut in accordance with Chapter 20 –*Standard Practices*.
31. Install a new split pin in accordance with Chapter 20 –*Standard Practices*.
32. Rotate the valve arm through the entire range. Movement should be smooth and without binding. The valve flapper should seat over both openings.
33. Connect the alternate air control cable end to the flapper valve arm with bolt, nut and washer removed previously. Do not install a new split pin at this time.
34. Install alternate air cable "P" clamp as shown in Figure 71-47 using bolt, nut, washer, and spacer removed previously.
35. From the cabin control, exercise the alternate air flapper valve and verify full range of motion. Adjust the alternate air arm cable as required to achieve full range of motion.

36. Secure the connection for the alternate air arm cable with a new split pin.
37. Install the air filter by sliding the band clamp loosely over the filter opening. Slide the filter into the alternate air box assembly in let tube. Secure the filter band clamp by tightening.
38. Install .020 in safety wire between the filter base and alternate air arm-mounting bracket.
39. Install upper cowl in accordance with the Cowling Installation procedure on page 30 of this chapter.

This completes the Air Intake Removal procedure

Air Intake Inspection procedure.

Section 60-02 Air Intake Troubleshooting Guide

Use this troubleshooting guide to resolve issues with the air intakes.

Complaint	Possible Cause	Remedy
Filter damaged	• Time in service • Impact damage	• Replace
Filter time in service expired	• End of service life in accordance with Chapter 04	• Replace
Alternate air flapper valve will not move full travel	• Control cable out of adjustment	• Adjust cable
	• Cable “P” clamp loose	• Tighten “P” clamp
Stiff alternate air control	• Cable corrosion	• Lubricate cable • Replace cable if damaged
	• Alternate air arm mechanism damage	• Repair or replace alternate air box mechanism
	• Cable pinched	• Replace cable if damage found
Leaking inlet scat tube	• Loose band clamp	• Tighten band clamp
	• Damaged scat tube	• Replace scat tube
Corrosion	• In service wear	• Repair superficial surface corrosion • Replace alternate air box assembly for penetrating corrosion.
Loose filter safety wire	• Faulty installation	• Replace safety wire
	• Damage to filter base	• Replace filter

Table 71-6 Air Intakes Troubleshooting Guide

Section 71-70 Engine Drains

A small-diameter drain tube exits the intake area of each cylinder head to allow excess (liquid) fuel to drain. The four cylinder drains are combined to form a single engine fuel drain.

Check valves are provided to prevent backflow of ambient air into the intake manifold through the drains. At each scheduled maintenance interval, check to insure that the drains function properly.

Section 70-01 Manifold Fuel Drain Procedures

The section contains the procedure to perform in accordance with Chapter 05 – *Time Limits/Maintenance Checks/Inspection Intervals* scheduled maintenance.

MANIFOLD FUEL DRAIN CHECK PROCEDURE

Perform the Manifold Fuel Drain Procedure in accordance with Chapter 05 – *Time Limits/Maintenance Checks/Inspection Intervals* and on condition.

1. Apply light vacuum source to airframe fuel drain outlet.
2. Monitor fuel drain to ensure free airflow.
3. Apply slight positive air pressure to airframe fuel drain outlet.
4. Monitor fuel drain to ensure cylinder drain check valve closes (no airflow).
5. An additional drain is installed on dry bay of engine-mounted fuel pump. If fuel is present at drain indicates current (or impending) failure of fuel pump drive seals or internal components.

This completes the Manifold Fuel Drain Procedure.

Section 70-02 Engine Drains Troubleshooting Guide

Use this troubleshooting guide to resolve issues with the engine.

Complaint	Possible Cause	Remedy
Drain blockage	• Residue build up	• Repair tube damage
	• Tube damage	• Perform drain check
Fuel present	• Engine driven fuel pump failure is pending or has occurred.	• Inspect engine pump and replace in accordance with TCM Maintenance Manual M-22

Table 71-7 Engine Drains Troubleshooting Guide

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CHAPTER 72

ENGINE

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Table of Contents

SECTION 72-00 GENERAL 5
SECTION 00-01 ENGINE MODEL DESCRIPTION 5
SECTION 72-10 FRONT SECTION DESCRIPTION 7
SECTION 72-20 POWER SECTION DESCRIPTION 9
SECTION 72-30 CYLINDER SECTION DESCRIPTION 11
SECTION 30-01 BAFFLES 11
SECTION 30-02 BAFFLE PROCEDURES 12
PORT BAFFLE REMOVAL 13
PORT BAFFLE INSTALLATION 14
STARBOARD BAFFLE REMOVAL 15
STARBOARD BAFFLE INSTALLATION 16
FORWARD BAFFLE REMOVAL 17
FORWARD BAFFLE INSTALLATION 20
AFT BAFFLE REMOVAL 21
AFT BAFFLE INSTALLATION 24
SECTION 30-03 BAFFLE TROUBLESHOOTING GUIDE 26
SECTION 72-40 ACCESSORY CASE DESCRIPTION 27
SECTION 72-50 LUBRICATION 29

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Section 72-00 General

This chapter provides a descriptive overview of the IOF-240-B engine installed on the airplane. Detailed information for routine line maintenance for each engine subsection or system is provided in the appropriate chapter. More detailed information for repairs and maintenance on systems and components specific to the IOF-240-B engine (in particular, the FADEC system) are provided in the current release of the Teledyne Continental Motors Maintenance Manual for IOF-240-B series engines, TCM p/n: M-22.

Section 00-01 Engine Model Description

The airplane is powered by a Teledyne Continental Motors IOF-240-B engine rated at 125 bhp maximum continuous power. Recommended cruise power is 90 bhp. The engine drives a two-blade wooden fixed pitch propeller.

The IOF-240-B-X is a four-cylinder air-cooled engine. Its type designation reflects the following:

I – the engine is fuel injected

O – Cylinder layout - horizontally opposed

F– The engine uses a Full Authority Digital Engine Control system (FADEC)

240 – Engine displacement is 240 cubic inches

B – Model designation indicating that the engine is designed for use with a fixed-pitch propeller, with a doweled six bolt hole propeller flange and no provision for a hydraulic propeller governor

X – Specific configuration of accessories as supplied to Liberty Aerospace, Inc., for use on the airplane.

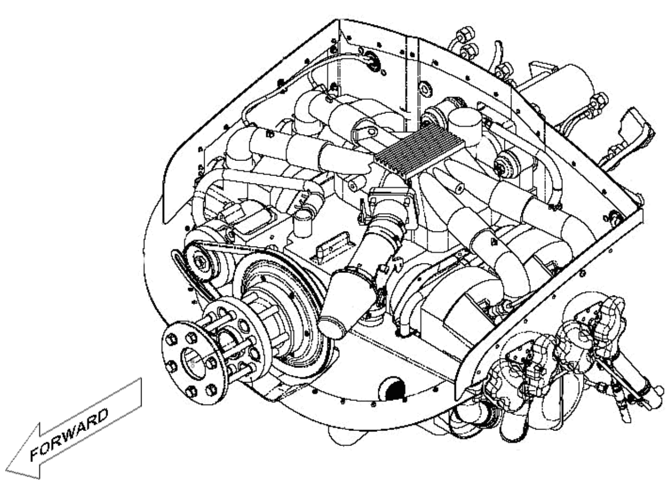

Figure 72-1 Engine Assembly

Because of the nature and complexity of the FADEC system, procedures covered in this maintenance manual include routine line servicing, replenishment of operating fluids, and replacement of "consumable" components such as spark plugs only. Detailed information and procedures covering maintenance and troubleshooting of the FADEC system and related components are provided in the Teledyne Continental Motors Maintenance Manual for IOF-240-B series engines, TCM P/N: M-22.

Section 72-10 Front Section Description

The IOF-240-B engine is direct drive; there is no reduction gear. A front crankshaft seal prevents leakage of engine oil.

In the airplane installation, the alternator is installed at the front of the engine and belt-driven via pulleys on the alternator and engine crankshaft. A solid propeller shaft extension is installed on the crankcase flange, with the propeller and spinner installed at the forward end of the shaft extension.

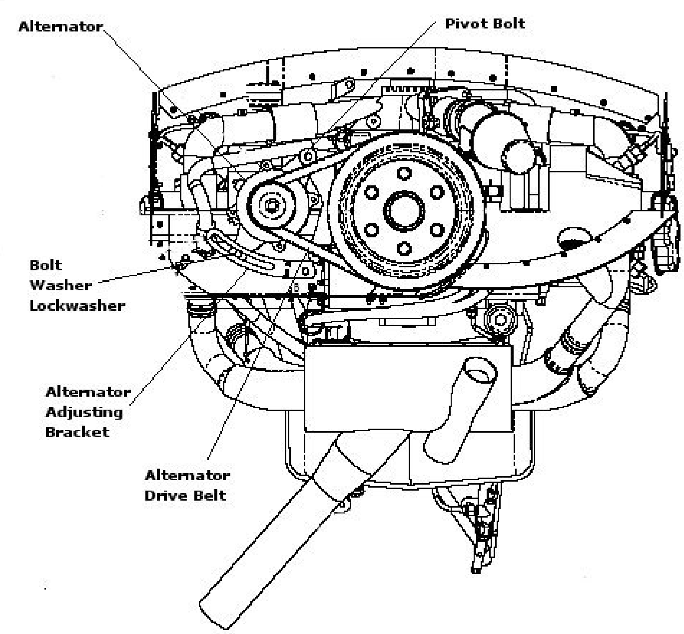

Figure 72-2 Engine Front View

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Section 72-20 Power Section Description

The engine power section includes the crankshaft, camshaft, connecting rods, valve lifters, and related bearings.

The crankshaft rotates on three bearings (front, intermediate, and rear). The bearings are of the metal-to-metal type and are lubricated by oil under pressure fed through drillings in the crankshaft. Additional drilled oil passages provide lubrication to the connecting rods.

The camshaft is supported by three main bearings and is gear driven (at a 1:2 ratio) by the crankshaft. The crankshaft/camshaft gear train is located at the rear of the engine. Self-adjusting hydraulic valve lifters (tappets) are moved by the camshaft to operate cylinder intake and exhaust valves.

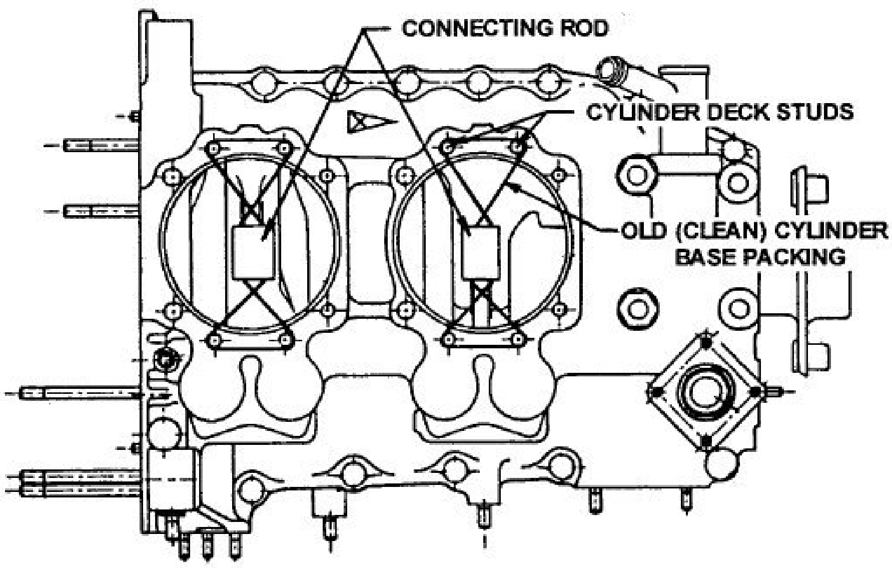

Figure 72-3 Engine Case

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Section 72-30 Cylinder Section Description

There are four (identical) cylinder assemblies. Each assembly consists of a steel cylinder barrel with an aluminum alloy cylinder head assembly screwed and shrunk to the barrel. Cooling fins are machined in both the barrel and the head. The interior of the barrel has a nitride coating for increased wear.

The intake and exhaust valves are installed in valve guides in the cylinder head and each have two concentric valve springs. Rocker arms, installed via brass bushings on a single rocker shaft in each cylinder head, transmit motion from the valve pushrods to the valve stems. The pushrods and rockers are drilled to provide an oil passage from the hydraulic valve lifters in the crankcase, through the pushrods and the rockers, to the rocker arm bearings and to the valve stem faces. Oil spray from the rocker arm bearings lubricates the valve stems. Excess oil returns to the crankcase via the pushrod tubes, which are spring-loaded to seal against the cylinder head and the crankcase.

A cylinder head temperature (CHT) sensor is installed in each cylinder head to provide required data to the FADEC system computers. An exhaust gas temperature (EGT) sensor is installed in each cylinder's exhaust pipe, approximately two inches from its attachment to the cylinder exhaust port, to provide required data to the FADEC system computers.

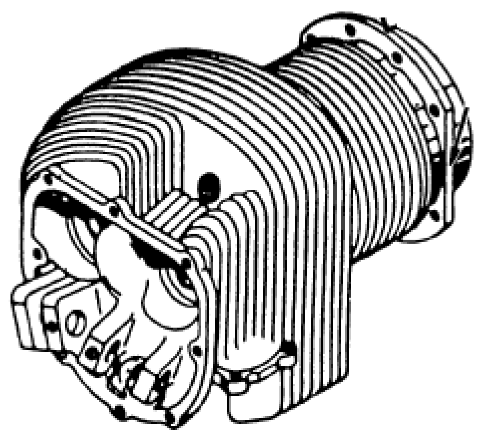


Figure 72-4 Cylinder Head Assembly

Section 30-01 Baffles

Baffles are installed in the Liberty XL2 airplane to direct airflow through the engine compartment to augment engine. Each baffle assembly consists of a sheet metal baffle and a silicon cowl seal riveted to the edge of the baffle. Cutouts in the baffle fit around engine parts so as to direct the airflow aft and down through the engine compartment.

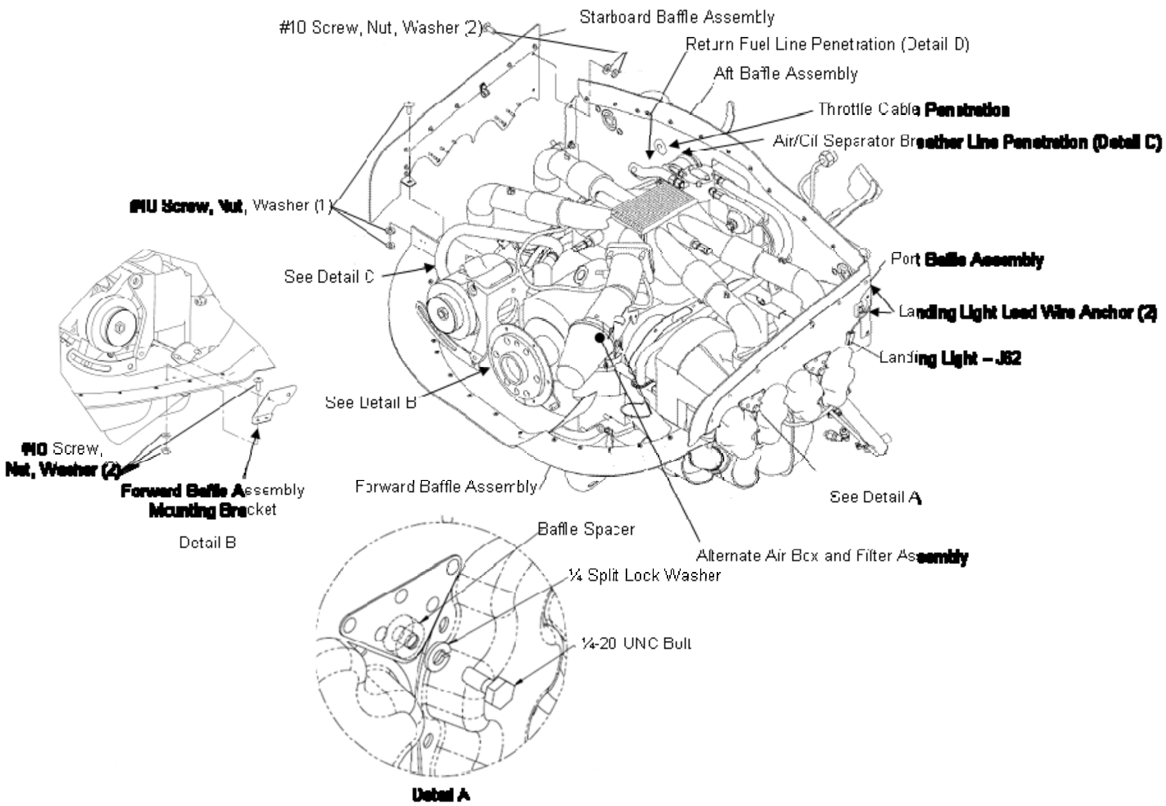


Figure 72-5 Engine Baffle System

The baffle system is made up of 4 assemblies as follows:

1. Port Baffle Assembly
2. Starboard Baffle Assembly
3. Aft Baffle Assembly
4. Forward Baffle Assembly

Figure 72-5 depicts placement of each baffle assembly along with relates attachment hardware. Port and starboard baffle assemblies can be removed without disturbing engine systems penetrate the forward and aft baffles assemblies. Forward and aft baffle assembly removal requires some engine system removal. Procedures to follow identify supporting manual chapters required to support baffle removal and installation.

Section 30-02 Baffle Procedures

The following Baffle Procedures perform removal and installation of the engine baffle system. Procedures for each baffle assembly may be run together or individually as required by the maintenance operation supported.

PORT BAFFLE REMOVAL

This performs Port Baffle Removal. Refer to Figure 72-5 for location of hardware.

1. Position the aircraft split master switch – OFF
2. Position BAT1 circuit breaker – OPEN
3. Position SYSTEM, START circuit breaker – OPEN
4. Remove upper and lower cowl in accordance with Chapter 71 – Power Plant of this manual
5. Locate and remove "P" clamp securing the ignition lead above cylinders
6. Remove wire tie securing landing light lead and connector J62 to the baffle mounted wire tie anchor.
7. Remove one (1) #10 screw, washer and nut set securing the baffle to the forward baffle assembly
8. Remove two (2) #10 screw, washer and nut sets securing the baffle to the aft baffle assembly.
9. Remove two (2) ¼-20 bolt, lock washer and spacers securing the lower baffle section to the cylinders
10. Remove the port baffle.

This completes the Port Baffle Removal procedure.

PORT BAFFLE INSTALLATION

This performs Port Baffle Installation. Refer to Figure 72-5 for location of hardware.

1. Position the aircraft split master switch – OFF
2. Position BAT1 circuit breaker – OPEN
3. Position SYSTEM, START circuit breaker – OPEN
4. Position the baffle aligning with the cylinder ¼-20 bolt hole locations.
5. Install two (2) ¼-20 bolt, lock washer and spacers securing the lower baffle section to the cylinders. Do not tighten at this time.
6. Install two (2) #10 screw, washer and nut sets securing the baffle to the aft baffle assembly. Do not tighten at this time
7. Install one (1) #10 screw, washer and nut set securing the baffle to the forward baffle assembly. Do not tighten at this time.
8. Locate and install "P" clamp securing the ignition lead above cylinders
9. Install wire tie to secure landing light lead and connector J62 to the baffle mounted wire tie anchor.
10. Align baffle with adjacent baffle structure and tighten #10 hardware.
11. Torque ¼-20 baffle bolts (2) in accordance with Teledyne Continental Motors manual M-22 Appendix B.
12. Install upper lower cowl in accordance with Chapter 71 – *Power Plant* of this manual
13. Position BAT1 circuit breaker – CLOSED
14. Position SYSTEM, START circuit breaker – CLOSED

This completes the Port Baffle Removal procedure.

STARBOARD BAFFLE REMOVAL

This performs Starboard Baffle Removal. Refer to Figure 72-5 for location of hardware.

1. Position the aircraft split master switch – OFF
2. Position BAT1 circuit breaker – OPEN
3. Position SYSTEM, START circuit breaker – OPEN
4. Remove upper and lower cowl in accordance with Chapter 71 – *Power Plant* of this manual.
5. Locate and remove "P" clamp securing the ignition lead above cylinders
6. Remove one (1) #10 screw, washer and nut set securing the baffle to the forward baffle assembly
7. Remove two (2) #10 screw, washer and nut sets securing the baffle to the aft baffle assembly.
8. Remove two (2) ¼-20 bolt, lock washer and spacers securing the lower baffle section to the cylinders.
9. Remove the port baffle.

This completes the Starboard Baffle Removal procedure.

STARBOARD BAFFLE INSTALLATION

This performs Starboard Baffle Installation. Refer to Figure 72-5 for location of hardware.

1. Position the aircraft split master switch – OFF
2. Position BAT1 circuit breaker – OPEN
3. Position SYSTEM, START circuit breaker – OPEN
4. Position the baffle aligning with the cylinder ¼-20 bolt hole locations.
5. Install two (2) ¼-20 bolt, lock washer and spacers securing the lower baffle section to the cylinders. Do not tighten at this time.
6. Install two (2) #10 screw, washer and nut sets securing the baffle to the aft baffle assembly. Do not tighten at this time
7. Install one (1) #10 screw, washer and nut set securing the baffle to the forward baffle assembly. Do not tighten at this time.
8. Locate and install "P" clamp securing the ignition lead above cylinders
9. Align baffle with adjacent baffle structure and tighten #10 hardware.
10. Torque the two ¼-20 baffle bolts in accordance with Teledyne Continental Motors manual M-22 Appendix B.
11. Install upper lower cowl in accordance with Chapter 71 – *Power Plant* of this manual
12. Position BAT1 circuit breaker – CLOSED
13. Position SYSTEM, START circuit breaker – CLOSED

This completes the Starboard Baffle Installation procedure.

FORWARD BAFFLE REMOVAL

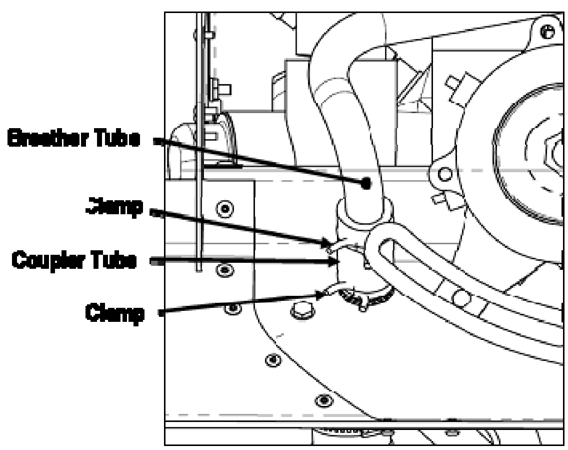
This performs Forward Baffle Removal. Refer to Figure 72-5 for location of hardware

1. Position the aircraft split master switch – OFF
2. Position BAT1 circuit breaker – OPEN
3. Position SYSTEM, START circuit breaker – OPEN
4. Remove upper and lower cowl in accordance with Chapter 71 – *Power Plant* of this manual.
5. Remove alternate air box assembly in accordance with Chapter 71 – *Power Plant* of this manual

NOTE

The XL-2 can be fitted with one of two different breather installations. The first is a direct overboard breather and the second is an air/oil separator. If an air/oil separator is installed proceed to step 9. If an overboard breather is installed proceed with the following step.

6. Loosen clamps securing the coupler tube to the breather line
7. Slide the breather tube up and out of the couple tube
8. Remove the couple tube by sliding it upward off of the lower breather tube



Detail C
Overboard Breather Penetration

Figure 72-6 Overboard Breather Penetration

NOTE

If an air/oil separator is installed proceed with the following step.

9. Remove #10 screw, nut and washer sets (2) from the forward baffle upper bracket assembly as shown in Figure 72-7.

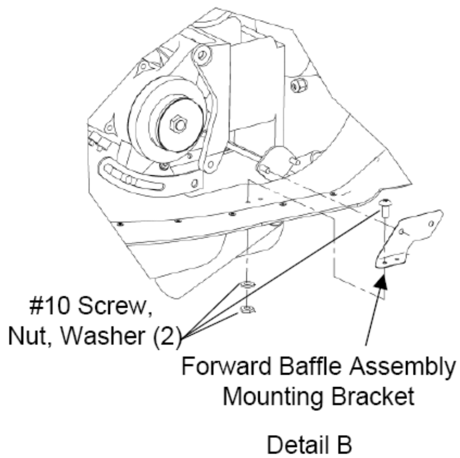


Figure 72-7 Forward Baffle Assembly Upper Bracket

10. Remove #10 screw, nut and washer sets (2) from the forward baffle lower bracket assembly as shown in Figure 72-8.

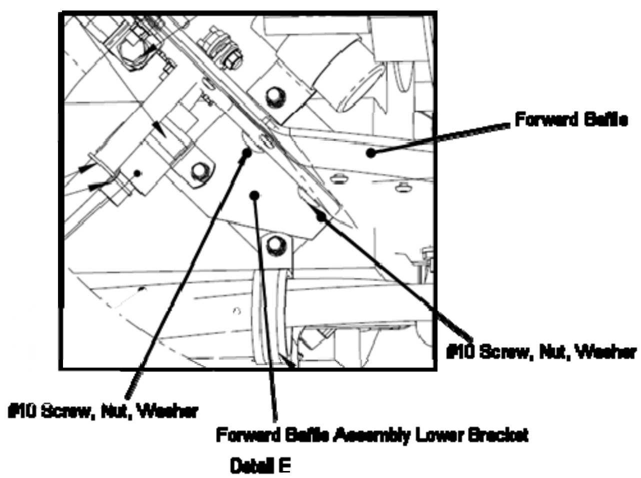


Figure 72-8 Forward baffle Assembly Lower Bracket

11. Remove the #10 screw, nut and washer set (1) connecting the forward baffle assembly to the port baffle assembly.
12. Remove the #10 screw, nut and washer set (1) connecting the forward baffle assembly to the starboard baffle assembly.
13. Remove the #10 screw and washer from the port and starboard cylinder inter-baffle assemblies as shown in Figure 72-9.

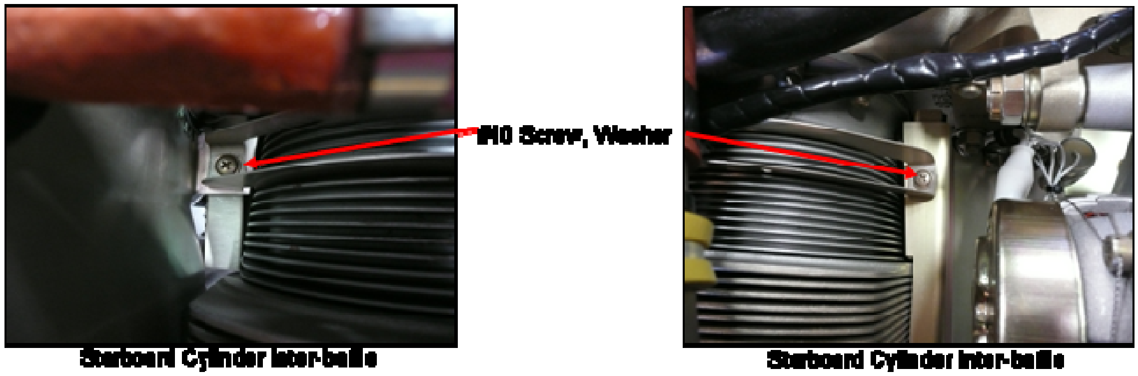


Figure 72-9 Cylinder Inter-Baffle Screws

14. Remove the forward baffle assembly from the engine.

This completes the Forward Baffle Removal procedure

FORWARD BAFFLE INSTALLATION

The following procedure performs Forward Baffle Installation

1. Position the aircraft split master switch – OFF
2. Position BAT1 circuit breaker – OPEN
3. Position SYSTEM, START circuit breaker – OPEN
4. Position the forward baffle assembly at the from of the engine
5. Install the #10 Screw and washer from the port and starboard cylinder inter-baffle assemblies as shown in Figure 72-9. Do not tighten hardware at this time.
6. Install the #10 screw, nut and washer set (1) connecting the forward baffle assembly to the starboard baffle assembly. Do not tighten hardware at this time.
7. Install #10 screw, nut and washer sets (2) from the forward baffle lower bracket assembly as shown in Figure 72-8. Do not tighten hardware at this time.
8. Install #10 screw, nut and washer sets (2) from the forward baffle upper bracket assembly as shown in Figure 72-7.

NOTE

The XL-2 can be fitted with one of two different breather installations. The first is a direct overboard breather and the second is an air/oil separator. If an air/oil separator is installed do not perform the next four (4) steps.

9. Install the couple tube by sliding it downward onto the lower breather tube
10. Secure the lower tube section with clamps removed previously
11. Slide the breather tube down and into the couple tube
12. Secure the breather tube and couple tube with clamp removed previously.
13. Align forward baffle with port and starboard baffle and torque all hardware.
14. Install alternate air box assembly in accordance with Chapter 71 – *Power Plant* of this manual
15. Install upper and lower cowl in accordance with Chapter 71 of this manual.
16. Position SYSTEM, START circuit breaker – CLOSED
17. Position BAT1 circuit breaker – CLOSED

This completes the Forward Baffle Installation procedure.

AFT BAFFLE REMOVAL

The following procedure performs Aft Baffle Removal

1. Position the aircraft split master switch – OFF
2. Position BAT1 circuit breaker – OPEN
3. Position SYSTEM, START circuit breaker – OPEN
4. Remove upper and lower cowl in accordance with Chapter 71 – *Power Plant* of this manual.
5. Remove the throttle arm linkage bolt, nut, washer and split key.
6. Remove the throttle cable from the throttle cable bracket sliding "B" nuts off the end of the throttle cable.

Figure 72-10 Throttle Cable and Bracket

NOTE

The XL-2 can be fitted with one of two crank case breather systems. One is a direct overboard breather and the other is an air/oil separator. For engines fitted with an air/oil separator perform the following step else go to step 8.

7. Remove the air/oil separator line by removing the separator band clamp as shown in Figure 72-11.
8. Slide the hose out of the aft engine baffle.

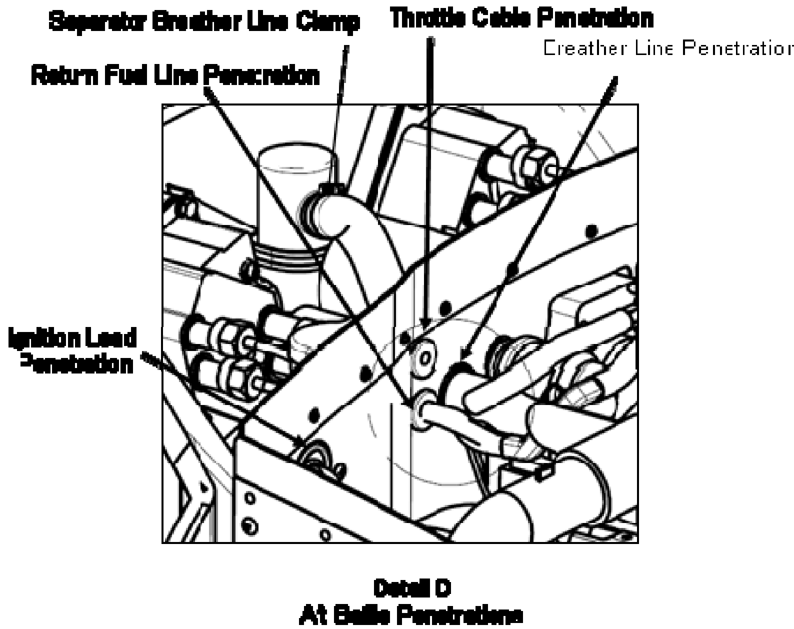


Figure 72-11 Aft Baffle Penetrations

9. Remove port ignition leads by removal of #10 screw, nut washer sets (2) and sliding the leads aft out of the baffle.
10. Remove starboard ignition leads by removal of #10 screw, nut washer sets (2) and sliding the leads aft out of the baffle.

CAUTION

In the following step a fuel line will be opened. Some residual fuel and vapors may be released as the line is opened. Capture residual fuel in a container. No heat sources can be exposed during this procedure.

11. Remove and cap fuel return line connected to the recirculation valve assembly and route the line back out of the aft baffle as shown in Figure 72-12.

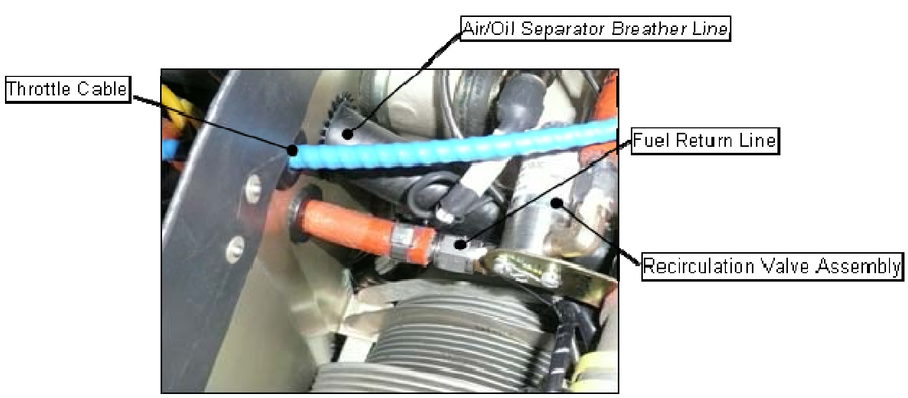


Figure 72-12 Fuel Return Line Penetration

12. Remove wire tie securing landing light lead and connector J62 to the baffle mounted wire tie anchor.
13. Remove the #10 screw and washer from the port and starboard cylinder inter-baffle assemblies securing the aft baffle.
14. Remove port #10 screw, nut, washer assemblies (2) connecting the aft baffle assembly to the port side baffle assembly as shown in Figure 72-5
15. Remove starboard #10 screw, nut, washer assemblies (2) connecting the aft baffle assembly to the starboard side baffle assembly as shown in Figure 72-5.
16. Disconnect the alternate air box control cable at the box.
17. Slide the aft baffle up and out of the engine guiding the throttle cable and alternate air control cable out of the baffle as it is removed.

This completes the Aft Baffle Removal procedure

AFT BAFFLE INSTALLATION

The following procedure performs Aft Baffle Installation

1. Position the aircraft split master switch – OFF
2. Position BAT1 circuit breaker – OPEN
3. Position SYSTEM, START circuit breaker – OPEN
4. Position the aft baffle above the engine accessory case and feed the throttle cable and alternate air control cable through the aft baffle.
5. Install port #10 screw, nut, washer assemblies (2) connecting the aft baffle assembly to the port side baffle assembly as shown in Figure 72-5. Do not tighten at this time.
6. Install starboard #10 screw, nut, washer assemblies (2) connecting the aft baffle assembly to the starboard side baffle assembly as shown in Figure 72-5. Do not tighten at this time.
7. Install the #10 screw and washer from the port and starboard cylinder inter-baffle assemblies securing the aft baffle. Do not tighten at this time.
8. Install wire tie securing landing light lead and connector J62 to the baffle mounted wire tie anchor.
9. Route the fuel return line through the aft baffle and connect to the recirculation valve assembly as shown in Figure 72-12.
10. Install port ignition leads by sliding leads through the aft baffle and securing the baffle seal with #10 screw, nut washer sets (2).
11. Install starboard ignition leads by sliding leads through the aft baffle and securing the baffle seal with #10 screw, nut washer sets (2).

NOTE

The XL-2 can be fitted with one of two crank case breather systems. One is a direct overboard breather and the other is an air/oil separator. For engines fitted with an air/oil separator perform the following step else go to step 14.

12. Slide the hose through the aft engine baffle.
13. Secure the air/oil separator line by installing the separator band clamp as shown in Figure 72-11.
14. Slide throttle cable “B” nuts over the cable assembly and secure throttle cable to the engine cable bracket.
15. Connect the throttle cable end to the air body arm and secure with split key.
16. Connect control cable to alternate air box control arm and secure with new split key.

17. Align aft baffle with port and starboard side baffles and tighten hardware.
18. Install upper and lower cowl in accordance with Chapter 71 – *Power Plant* of this manual
19. Position SYSTEM, START circuit breaker – CLOSED
20. Position BAT1 circuit breaker – CLOSED

This completes the Aft Baffle Installation procedure

Section 30-03 Baffle Troubleshooting Guide

Use this troubleshooting guide to resolve issues with the baffle system. This guide provides initial troubleshooting guidance.

Complaint	Possible Cause	Remedy
Baffle Cracked	• Vibration • Loose installation • Corrosion	• Standard repair in accordance with Chapter 51 – *Standard Practices Structures* • Replace
Baffle Corrosion	• In service Wear	• Repair in accordance with Chapter 51 – *Standard Practices Structures* if superficial • Replace if greater than superficial
Baffle Seal Damage	• In service wear • Faulty cowling installation	• Replace seal • Install cowl in accordance with Chapter 71 – *Power Plant*
Damaged baffle penetration grommets	• In service wear	• Replace

Table 72-1 Troubleshooting Table

Section 72-40 Accessory Case Description

The aluminum alloy casting of the accessory case is attached to the rear of the engine crankcase, aligned with crankcase dowels; the accessory case is secured to the crankcase-by-crankcase studs and various attaching hardware. Accessory mount pads on the rear surface are machined in one plane parallel to the machined parting flange, which surrounds the front side of the casting. Mounting pads for the magnetos, alternator cover, starter, tachometer drive, oil filter adapter, a relief valve for the oil pressure and an oil suction screen boss are provided. The accessory case casting has two holes above and three studs to attach the starter and starter adapter. A mounting pad is provided for the oil screen housing in lieu of the screw-on type oil filter.

The oil pump housing is machined into the internal portion of the accessory case. A machined, threaded boss is located on the lower right side of the accessory case for installation of a non-adjustable valve for oil pressure relief. The chamber for the oil pump gears are machined in the interior of the accessory case. The hole for the gear shaft of the oil pump drive is machined in-line with the camshaft and the driven gear shaft hole is directly above it.

A semicircular opening at the accessory case bottom is a machined threaded hole to accommodate installation of the oil suction tube. Passages cast into the accessory case allow oil to flow from the oil suction tube to the oil pump gears, pressure relief valve, and main oil gallery. The tachometer drive shaft is the slotted end of the oil pump driven gear shaft.

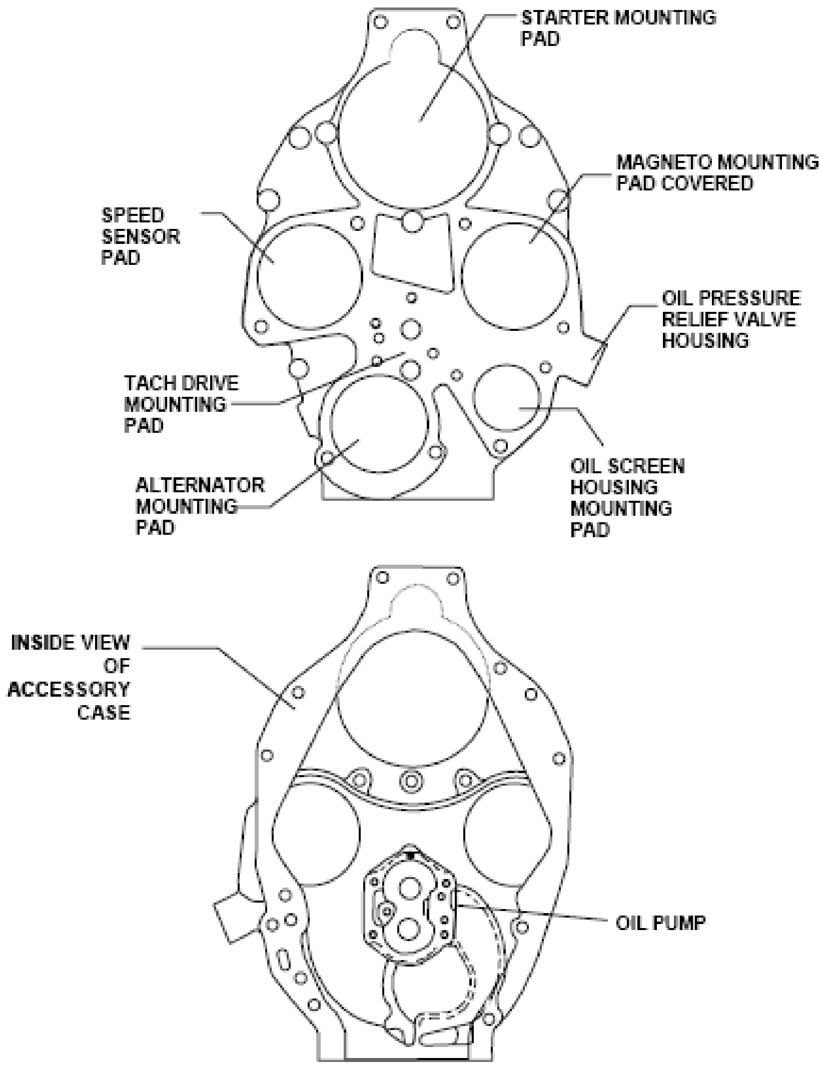


Figure 72-13 Accessory Case

Section 72-50 Lubrication

Oil is stored in an oil sump bolted to the bottom of the crankcase near the rear of the engine. A gear-type oil pump at the rear of the engine is driven by the camshaft drive gear. The oil sump incorporates an external filler tube to allow oil replenishment. A calibrated dipstick is installed in the oil filler tube.

A suction tube extends from the oil pump into the oil tank. Oil leaves the oil pump under pressure. A regulating valve allows some oil to return to the oil tank to maintain oil pressure within limits at varying engine speeds. Oil from the pump is routed to an oil cooler adapter on the left side of the engine accessory case.

The oil cooler adapter has hose connections to route oil to the firewall-mounted oil cooler, which incorporates a “Vernatherm” thermostatically controlled valve to control oil temperature, and to the oil filter element. An integral bypass valve will open in the event of blockage of the oil cooler. The oil temperature sensor for the engine instruments are is located on the oil cooler adapter. Oil pressure is located remotely and not installed on the engine.

A full-flow oil filter is mounted to an adapter on the lower accessory case. The oil filter is located just aft of the engine oil sump.

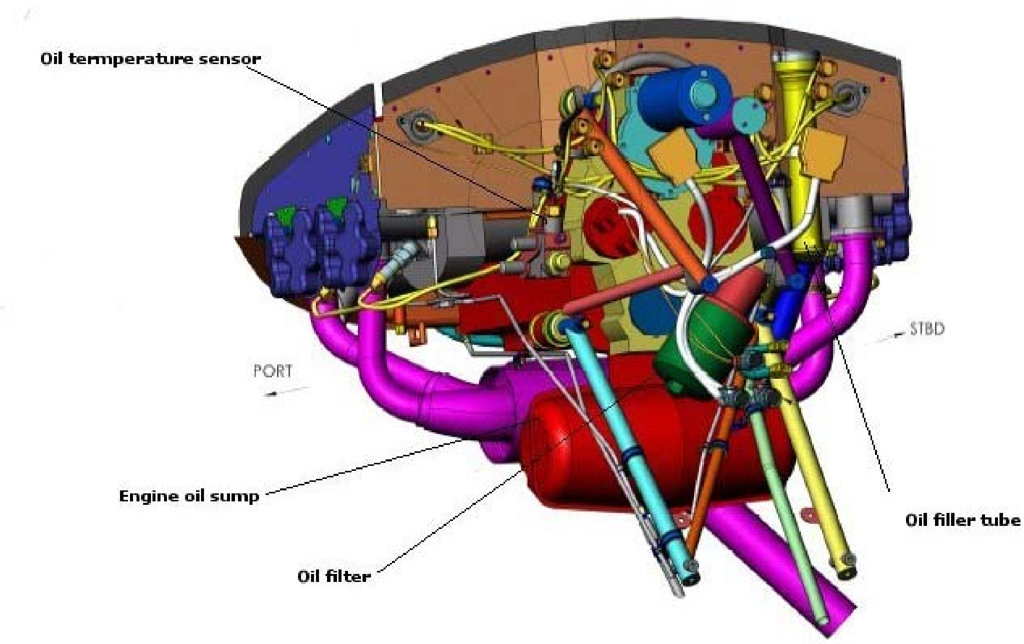


Figure 72-14 Cut-A-Way View of Engine and Oil Related Components

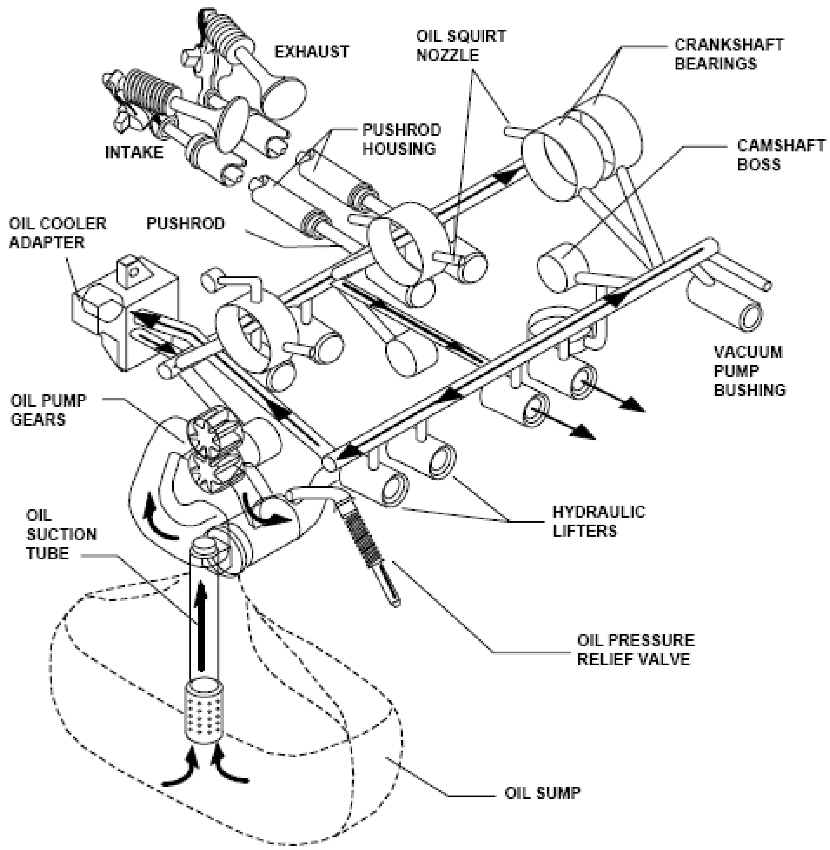


Figure 72-15 Oil Distribution

Figure 72-15 shows the flow of oil with in the engine. Oil galleries and drilled passages inside the engine route oil to the crankshaft and camshaft main bearings and to the hydraulic valve lifters. Drillings inside the crankshaft route oil to the connecting rod lower bearings. Oil escaping from the main and connecting rod bearings creates an oil mist inside the crankcase which lubricates the connecting rod upper bearings and cylinder walls, as well as the cam lobes and lower faces of the lifters. In addition, oil nozzles on the main bearings direct a jet of oil at the undersides of the pistons to cool them.

Oil in the hydraulic lifters is routed via the hollow valve pushrods to the cylinder rocker boxes, where it lubricates the rockers, valve stems, and valve guides before returning, via the pushrod housings, to the crankcase. Any excess oil in the crankcase returns, via the large-diameter opening between the crankcase and the sump, to the oil sump.

CHAPTER 73

ENGINE FUEL CONTROL

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Table of Contents

SECTION 73-00	GENERAL	5
SECTION 73-01	FUEL DISTRIBUTION	7
SECTION 01-01	ENGINE DRIVEN FUEL PUMP	7
SECTION 01-02	PERIODIC MAINTENANCE	7
SECTION 01-03	ENGINE DRIVEN FUEL PUMP PROCEDURES	7
ENGINE DRIVEN PUMP REMOVAL		8
ENGINE DRIVEN PUMP INSTALLATION		9
ENGINE DRIVEN PUMP OPERATION CHECK AND INSPECTION		10
SECTION 01-04	ENGINE DRIVEN FUEL PUMP TROUBLESHOOTING GUIDE	11
SECTION 01-05	BYPASS FUEL FILTER	11
SECTION 01-06	PERIODIC MAINTENANCE	12
SECTION 01-07	BYPASS FUEL FILTER PROCEDURES	12
BYPASS FUEL FILTER REMOVAL		13
BYPASS FUEL FILTER INSTALLATION		14
BYPASS FUEL FILTER OPERATION CHECK AND INSPECTION		16
SECTION 01-08	BYPASS FUEL FILTER TROUBLESHOOTING GUIDE	17
SECTION 01-09	FUEL DISTRIBUTION BLOCK FILTER	17
SECTION 01-10	PERIODIC MAINTENANCE	18
SECTION 01-11	FUEL DISTRIBUTION BLOCK FILTER PROCEDURES	18
FUEL DISTRIBUTION BLOCK FILTER OPERATION CHECK AND INSPECTION		19
SECTION 01-12	FUEL DISTRIBUTION BLOCK FILTER TROUBLESHOOTING GUIDE	20
SECTION 73-02	FUEL CONTROL AND GOVERNING	21
SECTION 02-01	FUEL INJECTORS	21
SECTION 02-02	PERIODIC MAINTENANCE	21
SECTION 02-03	FUEL INJECTOR PROCEDURES	21
FUEL INJECTOR SOLENOID REMOVAL		22
FUEL INJECTOR REMOVAL		23
FUEL INJECTOR INSTALLATION		24
FUEL INJECTOR SOLENOID INSTALLATION		26
FUEL INJECTOR OPERATIONAL CHECK AND INSPECTION		28
SECTION 02-04	FUEL INJECTOR TROUBLESHOOTING GUIDE	30
SECTION 73-03	FUEL INDICATING	31
SECTION 03-01	FUEL PRESSURE SENSORS	31
SECTION 03-02	PERIODIC MAINTENANCE	31
SECTION 03-03	FUEL PRESSURE SENSOR PROCEDURES	31
FUEL PRESSURE SENSOR REMOVAL		32
FUEL PRESSURE SENSOR INSTALLATION		33
FUEL PRESSURE SENSOR OPERATIONAL CHECK AND INSPECTION		35
SECTION 03-04	FUEL PRESSURE SENSOR TROUBLESHOOTING GUIDE	37

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Section 73-00 General

The IOF-240B engine uses an electronically controlled fuel injection system to provide optimally metered and timed amounts of fuel to each cylinder at the appropriate point prior to each combustion event. In addition to engine-mounted components including a mechanical fuel pump, fuel distribution plumbing and components, and electro magnetically operated fuel injection nozzles for each cylinder.

Fuel from the airframe fuel system is routed to the intake of the engine-driven fuel pump, which is mounted on the front left side of the engine and gear-driven from the camshaft.

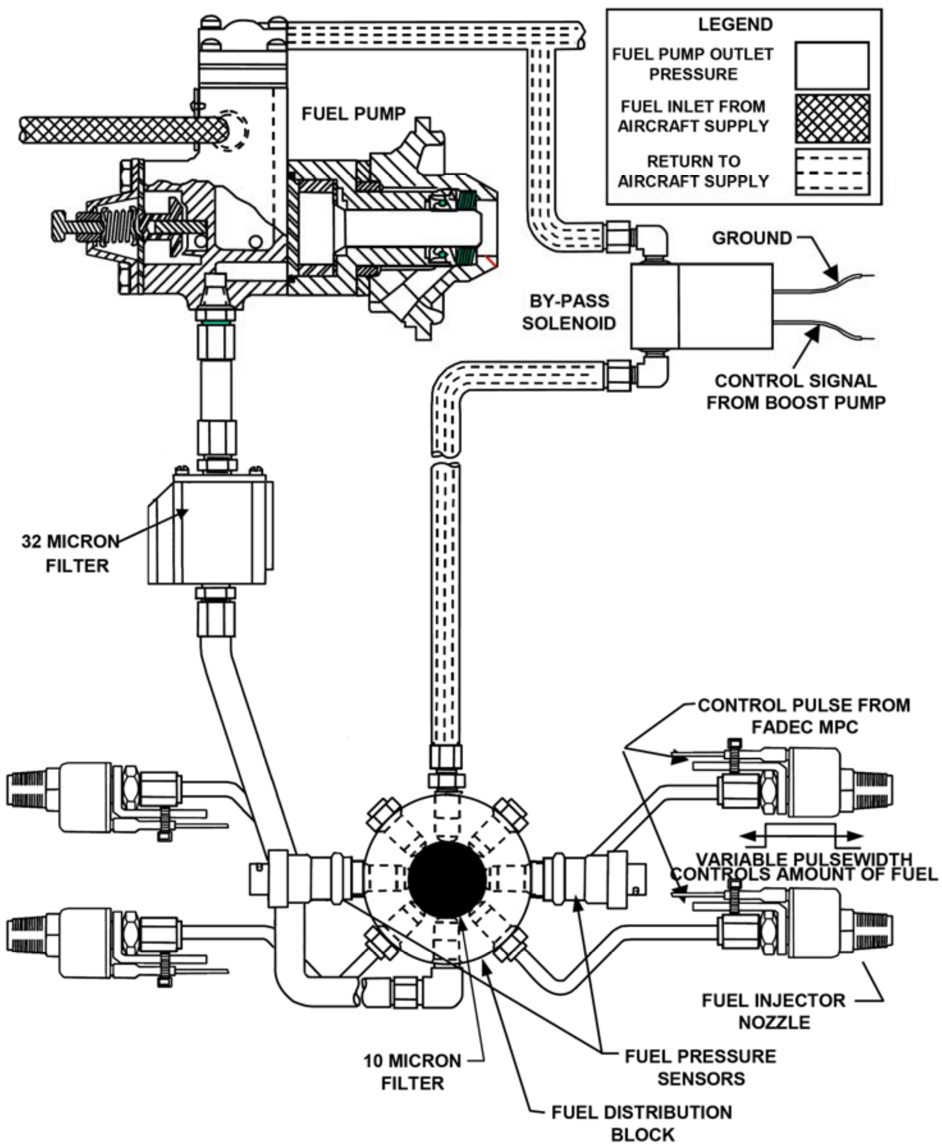

Figure 73-1 Engine Fuel System

In operation, fuel enters the engine driven pump passing through a centrifugal separator where fuel vapor is removed and returned to the airplane fuel tank. Next, it passes into the pump element, where its pressure is increased. Since pump effectiveness varies with engine speed, the pump is designed to produce more pressure and flow than is required by the engine fuel injectors.

An adjustable relief valve regulates pump output at lower RPM, while an adjustable internal orifice regulates pressure at high RPM. These adjustments are critical to the proper functioning of the FADEC system.

Fuel pressure is sensed at the fuel distribution block by two pressure transducers connected to the FADEC system. Fuel pressure sensor measurements are used by FADEC software to manage engine operation in real time. FADEC also provides fuel pressure measurements in a data stream read by the VM-1000FX Integrated Engine Instrument System for display to the pilot. This same data stream is routed to the Engine Data Interface (EDI) where it is recorded as a fuel pressure history that can be retrieved by a qualified A & P mechanic for diagnostic purposes. For an explanation of the EDI System, refer to Chapter 31 - Indicators and Recording Systems.

This chapter provides a descriptive overview of the fuel system installed on the IOF-240B engine. Information on the airplanes fuel system, which includes the fuel boost pump, gascolator, and fuel switch are in Chapter 28 - *Fuel System*. This chapter deals with the fuel system from the firewall forward. See Figure 73-1 for a diagram of the engine fuel system.

Detailed information for repairs and maintenance on systems and components specific to the IOF-240B engine (in particular, the FADEC system) are provided in the current release of the Teledyne Continental Motors Maintenance Manual for IOF-240-B series engines, Teledyne Continental Motors P/N M-22.

Section 73-01 Fuel Distribution

Section 01-01 Engine Driven Fuel Pump

An engine-driven, positive displacement vane fuel pump shown in Figure 73-2 supplies fuel as required to the fuel injectors. Fuel enters the fuel pump at the well of the swirl chamber where the fuel is centrifuged and the liquid is separated from fuel vapor. The liquid fuel is directed to the fuel pump blades. The fuel pump blades force the fuel to the fuel pump outlet through various fittings and fuel lines.

The fuel pump is directly driven at the same speed as the crankshaft. Therefore, fuel flow and fuel pressure vary directly with engine speed. An adjustable relief valve maintains pump flow and pressure for the lower engine speeds, while an adjustable orifice controls fuel pump pressure for the higher engine speeds. This combination of mechanical control circuits ensures proper pump pressure and delivery for all engine operating speeds.

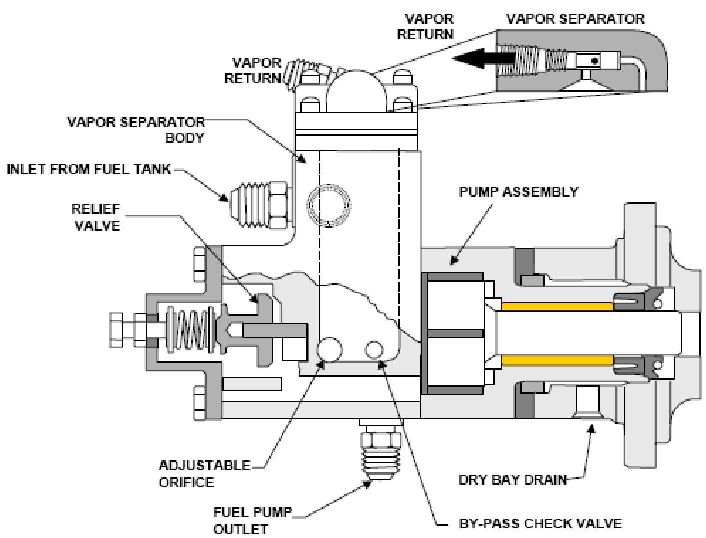

Figure 73-2 Engine Drive Fuel Pump

Section 01-02 Periodic Maintenance

Periodic maintenance of the engine driven pump entails an operational check performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 Engine Operational Checklist. Operation checks are performed in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 6. Adjustments to high speed and low speed fuel pressure settings are performed on condition in accordance with Teledyne Continental Motors Maintenance Manual M-22 procedures.

Section 01-03 Engine Driven Fuel Pump Procedures

This section has the procedures for removal installation and operational check of the engine driven fuel pump.

ENGINE DRIVEN PUMP REMOVAL

Perform this procedure to remove the engine driven fuel pump.

WARNING

The components mounted on the engine are very close to the propeller. Secure all electrical power as prescribed below prior to starting this procedure.

1. Position the split master switch OFF.
2. Position FADEC A and B power switches OFF.
3. Locate circuit breaker CB003, START and pull OPEN.
4. Locate circuit breaker CB029, BOOST and pull OPEN.
5. Position cockpit fuel selector OFF.
6. Remove upper and lower cowl in accordance Chapter 71 – *Power Plant.*

CAUTION

In the next several steps, fuel lines will be opened. Residual fuel may be present in these lines. Remove all sources of ignition from the work area and prepare a fuel rated container to capture fuel released as lines are opened. Cap all fuel lines after opening to prevent leakage and entry of contaminants. Pack the immediate area around the fuel lines with absorbent materials to absorb any fuel spillage.

7. Perform Fuel Pump Removal in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 10.

This completes the Engine Driven Pump Removal procedure.

ENGINE DRIVEN PUMP INSTALLATION

Perform this procedure to install the engine driven pump

WARNING

The components mounted on the engine are very close to the propeller. Secure all electrical power as prescribed below prior to starting this procedure.

1. Position the split master switch OFF.
2. Position FADEC A and B power switches OFF.
3. Locate circuit breaker CB003, START and pull OPEN.
4. Locate circuit breaker CB029, BOOST and pull OPEN.
5. Position cockpit fuel selector OFF.

CAUTION

In the next several steps, fuel lines will be opened. Residual fuel may be present in these lines. Remove all sources of ignition from the work area and prepare a fuel rated container to capture fuel released as lines are opened. Cap all fuel lines after opening to prevent leakage and entry of contaminants. Pack the immediate area around the fuel lines with absorbent materials to absorb any fuel spillage.

6. Perform Fuel Pump Installation in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 10. Do not perform operational checks until directed in the following steps.
7. Pack the area around the fuel pump with absorbent materials to absorb any fuel that may leak.
8. Position cockpit fuel selector ON.
9. Locate circuit breaker CB003, START and push to CLOSE.
10. Locate circuit breaker CB029, BOOST and push to CLOSE.
11. Perform Fuel Pump leak check and Engine Operational Check steps in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 10 Fuel Pump Installation.
12. Remove the packing around the fuel pump.
13. Install upper and lower cowl in accordance with Chapter 71 – *Power Plant*.

This completes the Engine Driven Pump Installation procedure.

ENGINE DRIVEN PUMP OPERATION CHECK AND INSPECTION

Perform this procedure to operational check and inspect the engine driven fuel pump.

WARNING

The components mounted on the engine are very close to the propeller. Secure all electrical power as prescribed below prior to starting this procedure.

1. Position the split master switch OFF.
2. Position FADEC A and B power switches OFF
3. Locate circuit breaker CB003, START and pull to OPEN.
4. Locate circuit breaker CB029, BOOST and pull OPEN.
5. Position cockpit fuel selector OFF.
6. Remove upper and lower cowl in accordance with Chapter 71 – *Power Plant*.
7. Perform a visual inspection of the engine driven fuel pump installation. Verify all fuel line fittings are secure with no indication of fuel staining. Fuel staining is an indication of a leak.

CAUTION

Do not proceed if there is evidence of leaks. Refer to Engine Driven Pump Troubleshooting Guide for corrective actions.

8. Position cockpit fuel selector ON.
9. Locate circuit breaker CB003, START and push to CLOSE.
10. Locate circuit breaker CB029, BOOST and push to CLOSE.
11. Perform Engine Operational Check in accordance with Teledyne Continental Motors Maintenance Manual Chapter 6.
12. Install upper and lower cowl in accordance with Chapter 71 – *Power Plant*.

This completes the Engine Driven Pump Operation Check and Inspection procedure.

Section 01-04 Engine Driven Fuel Pump Troubleshooting Guide

Table 73-1 Engine Driven Fuel Pump Troubleshooting Guide

Complaint	Possible Cause	Remedy
Fuel line pump connection leak	Loose fittings	Torque in accordance with Teledyne Continental Motors Maintenance Manual M-22 Appendix B
	Worn fuel lines	Replace
	Fitting contamination	Remove, clean and torque fitting in accordance with Teledyne Continental Motors Maintenance Manual M-22 Appendix B.
Level I diagnostic fails fuel pressure test	Fuel pump fault	Replace
	Fuel pump out of adjustment	Adjust in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 9.
	Contaminated 32 Micron filter element	Clean filter of contaminate
		Replace if damaged
	Contaminated 10 Micron filter element	Clean filter of contaminate
		Replace if damaged

Section 01-05 Bypass Fuel Filter

The engine driven pump forces fuel through a 32-Micron bypass fuel filter assembly located between the fuel pump and the fuel distribution block. The fuel filter assembly contains a removable filter screen element as shown in Figure 73-3.

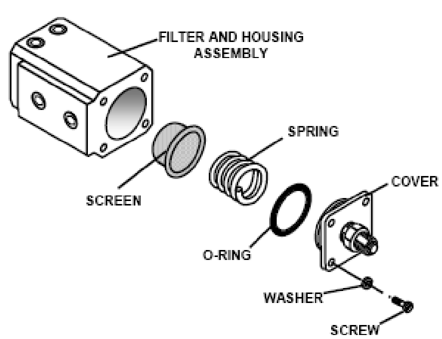

Figure 73-3 Bypass Fuel Filter Assembly

Section 01-06 Periodic Maintenance

Periodic maintenance of the 32-Mocron Bypass Fuel Filter entails an inspection and cleaning check performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 Engine Operational Checklist. Inspections are performed in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 7.

Section 01-07 Bypass Fuel Filter Procedures

The following procedures perform removal, installation, inspection and operational checks of the bypass fuel filter assembly. Periodic removal of the filter for inspection is required at prescribed maintenance intervals provided and on condition.

BYPASS FUEL FILTER REMOVAL

Perform this procedure to remove the bypass fuel filter.

WARNING

The components mounted on the engine are very close to the propeller. Secure all electrical power as prescribed below prior to starting this procedure.

1. Position the split master switch OFF.
2. Position FADEC A and B power switches OFF.
3. Locate circuit breaker CB003, START and pull OPEN.
4. Locate circuit breaker CB029, BOOST and pull OPEN.
5. Position cockpit fuel selector OFF.
6. Remove upper and lower cowl in accordance with Chapter 71 – *Power Plant*.

CAUTION

In the next several steps, fuel lines will be opened. Residual fuel may be present in these lines. Remove all sources of ignition from the work area and prepare a fuel rated container to capture fuel released as lines are opened. Cap all fuel lines after opening to prevent leakage and entry of contaminants. Pack the immediate area around the fuel lines with absorbent materials to absorb any fuel spillage.

7. Pack the area around the fuel pump with absorbent materials to absorb any fuel that may leak.
8. Disconnect the fuel lines from each end of the filter.
9. If replacing the filter later, place protective caps on the fuel lines.
10. Remove the filter from the mounting bracket.

This completes the Bypass Fuel Filter Removal procedure.

BYPASS FUEL FILTER INSTALLATION

Perform this procedure to install the bypass fuel filter.

WARNING

The components mounted on the engine are very close to the propeller. Secure all electrical power as prescribed below prior to starting this procedure.

1. If installing the replacement filter immediately after removal of the previous filter, go to step 8 of this procedure.
2. Position the split master switch OFF.
3. Position FADEC A and B power switches OFF.
4. Locate circuit breaker CB003, START and pull OPEN.
5. Locate circuit breaker CB029, BOOST and pull OPEN.
6. Position cockpit fuel selector OFF.
7. Remove upper and lower cowl in accordance with Chapter 71 – *Power Plant*.

CAUTION

In the next several steps, fuel lines will be opened. Residual fuel may be present in these lines. Remove all sources of ignition from the work area and prepare a fuel rated container to capture fuel released as lines are opened. Cap all fuel lines after opening to prevent leakage and entry of contaminants. Pack the immediate area around the fuel lines with absorbent materials to absorb any fuel spillage.

8. Install the filter assembly on the mounting bracket and secure hardware with .020 safety wire.
9. Connect the fuel lines to each end of the fuel filter assembly. Torque the "B" nut in accordance with Teledyne Continental Motors Maintenance Manual M-22 Appendix B.
10. Position the cockpit fuel selector ON.
11. Locate circuit breaker CB029, BOOST and push to CLOSE.
12. Position the split master switch ON.
13. Position the fuel boost pump switch ON.
14. Verify the fuel boost pump is running and normal fuel pressure is reported on the VM1000FX display.
15. Inspect the bypass filter and connections for indications of leaks.

CAUTION

If any of the fuel connections leak, immediately turn off the fuel boost pump and master switch. Refer to the troubleshooting guide at the end of this section for corrective action.

16. Position the fuel boost pump switch OFF
17. Position the split master switch OFF
18. Install the upper and lower cowl in accordance with Chapter 71 – *Power Plant*.
19. Locate circuit breaker CB003, START and push to CLOSE.

This completes the Bypass Fuel Filter Installation procedure.

BYPASS FUEL FILTER OPERATION CHECK AND INSPECTION

Perform this procedure to inspect and check the operation of the bypass filter. This procedure supports the scheduled maintenance in accordance with Liberty Maintenance Manual Chapter 05.

1. Position the split master switch OFF.
2. Position FADEC A and B power switches OFF.
3. Locate circuit breaker CB003, START and pull OPEN.
4. Locate circuit breaker CB029, BOOST and pull OPEN.
5. Position cockpit fuel selector OFF.
6. Remove upper and lower cowl in accordance with Chapter 71 – *Power Plant*.
7. Perform Bypass Fuel Filter Service in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 7.
8. Position the cockpit fuel selector ON.
9. Locate circuit breaker CB029, BOOST and push to CLOSE.
10. Position the split master switch ON.
11. Position the fuel boost pump switch ON.
12. Verify the fuel boost pump is running and normal fuel pressure is reported on the VM1000FX display.
13. Inspect the bypass filter and connections for indications of leaks.

CAUTION

If any of the fuel connections leak, immediately turn off the fuel boost pump and master switch. Refer to the troubleshooting guide at the end of this section for corrective action.

14. Position the fuel boost pump switch OFF.
15. Position the split master switch OFF.
16. Install the upper and lower cowl in accordance with Chapter 71 – *Power Plant*.
17. Locate circuit breaker CB003, START and push to CLOSE.

This completes the Bypass Fuel Filter Operation Check and Inspection procedure.

Section 01-08 Bypass Fuel Filter Troubleshooting Guide

Use Table 73-2 to aid in troubleshooting the bypass filter.

Complaint	Possible Cause	Remedy
Bypass filter connection leaks	Incorrect connection torque	• Torque in accordance with Teledyne Continental Motors Maintenance Manual M-22 Appendix B.
	Connection contamination	• Remove, clean and Torque in accordance with Teledyne Continental Motors Maintenance Manual M-22 Appendix B
	Deteriorated fuel lines	• Replace
Bypass filter housing leaks	Incorrect housing screw torque	• Perform Bypass Fuel Filter Service in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 7.
	Housing O ring failure	
Bypass filter reduced flow	Contaminated screen element	• Remove and clean in accordance with Teledyne Continental Motors Maintenance Manual Chapter 9. • Replace element if damaged

Table 73-2 Bypass Fuel Filter Troubleshooting Guide

Section 01-09 Fuel Distribution Block Filter

Fuel exits the 32-micron bypass fuel filter and enters the fuel distribution block where it encounters another filter as shown in Figure 73-4 that eliminates particulates larger than 10-microns. From this point, fuel travels to each fuel injector through rigid injector lines. No further filtering is performed once fuel leaves the fuel distribution block.

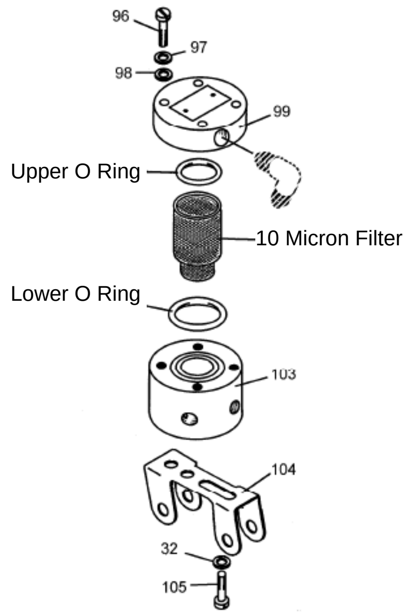

Figure 73-4 Fuel Distribution Block Filter Assembly

Section 01-10 Periodic Maintenance

Periodic maintenance of the Fuel Distribution Block Filter entails an inspection and cleaning check performed at intervals specified in Chapter 05 – *Time Limits/Maintenance Checks/Inspection Intervals* of this manual. Inspections are performed in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 7.

Section 01-11 Fuel Distribution Block Filter Procedures

This section contains the information about the removal, installation, inspection, and operational checks of the fuel distribution block filter assembly.

Periodic removal of the filter for inspection is required at prescribed maintenance intervals and on condition. Removal and replacement of the distribution block assembly is done on condition only. Refer to Teledyne Continental Motors Overhaul Manual OH-22 Chapter 6 for Fuel Distribution Block removal procedures and Chapter 11 for installation procedures.

Presented here is the procedure to check the operation and inspect the fuel distribution block

FUEL DISTRIBUTION BLOCK FILTER OPERATION CHECK AND INSPECTION

Perform this procedure to inspect and check the operation of the fuel distribution block filter.

WARNING

The components mounted on the engine are very close to the propeller. Secure all electrical power as prescribed below prior to starting this procedure.

Position the split master switch OFF.

1. Position FADEC A and B power switches OFF.
2. Locate circuit breaker CB003, START and pull OPEN.
3. Locate circuit breaker CB029, BOOST and pull OPEN.
4. Position cockpit fuel selector OFF.
5. Remove the engine cowlings in accordance with Chapter 71 – *Power Plant*.
6. Perform Fuel Distribution Block Service in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 7.
7. Position the cockpit fuel selector ON.
8. Locate circuit breaker CB029, BOOST and push to CLOSE.
9. Position the split master switch ON.
10. Position the fuel boost pump switch ON.
11. Verify the fuel boost pump is running and normal fuel pressure is reported on the VM1000FX display.
12. Inspect the Fuel Distribution Block and connections for indications of leaks.

CAUTION

If any of the fuel connections leak, immediately turn off the fuel boost pump and master switch. Refer to the troubleshooting guide at the end of this section for corrective action.

13. Position the fuel boost pump switch OFF.
14. Position the split master switch OFF.
15. Install the engine cowlings in accordance with Chapter 71 – *Power Plant*.
16. Locate circuit breaker CB003, START and push to CLOSE.

This completes the Fuel Distribution Block Filter Operation Check and Inspection procedure.

Section 01-12 Fuel Distribution Block Filter Troubleshooting Guide

Use Table 73-3 to aid in troubleshooting issues with the fuel distribution block.

Complaint	Possible Cause	Remedy
Fuel Distribution Block connection leaks	Incorrect connection torque	• Torque in accordance with Teledyne Continental Motors Maintenance Manual M-22 Appendix B.
	Connection contamination	• Remove, clean and Torque in accordance with Teledyne Continental Motors Maintenance Manual M-22 Appendix B
	Deteriorated fuel lines	• Replace
Fuel Distribution Block housing leaks	Incorrect housing screw torque	• Perform Bypass Fuel Filter Service in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 7.
	Housing Upper O ring failure	• Replace
	Housing Lower O ring failure	• Replace
Fuel Distribution Block filter reduced flow	Contaminated filter element	• Remove and clean in accordance with Teledyne Continental Motors Maintenance Manual Chapter 9. • Replace element if damaged

Table 73-3 Fuel Distribution Block Filter Troubleshooting Guide

Section 73-02 Fuel Control and Governing

This section contains information on fuel control and governing.

Section 02-01 Fuel Injectors

The fuel injector assembly is threaded on both ends and is made up of two parts: the control coil and the injector (Figure 73-5). The internal components of the fuel injector consist of a pintle valve and a spring. The solenoid creates an electromagnetic field to lift the pintle valve and open the path for fuel to flow. The solenoid coil fits over the pintle valve body. A support bracket is fitted on top of the solenoid to strengthen the electrical wiring harness. A single jam nut secures the injector, solenoid and bracket assembly. The valve design and injector end forms a self atomizing feed for the fuel, therefore, no air bleeds are needed to assist in fuel atomization.

The fuel injector assembly is designed to permit individual injector solenoid body replacement independent of the injector nozzle body. Likewise, individual injector nozzle bodies can be replaced independent of the injector coil body.

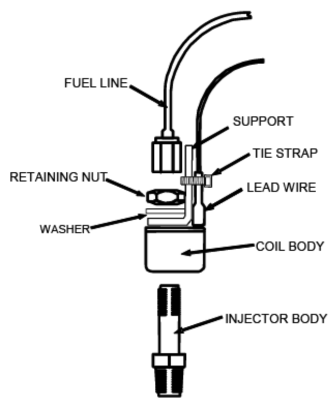

Figure 73-5 Fuel Injector Assembly

Section 02-02 Periodic Maintenance

Periodic maintenance of the fuel injector assemblies entails an operational check and inspection performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 Engine Operational Checklist. Inspections are performed in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 8.

Section 02-03 Fuel Injector Procedures

The following procedures perform fuel injector assembly removal, installation, operational checks and inspections. Injector assemblies are not routinely removed for periodic maintenance. Operational checks performed during periodic maintenance determine if a fault condition exists that requires injector assembly removal and installation procedures to be performed.

FUEL INJECTOR SOLENOID REMOVAL

The following procedure performs Fuel Injector Solenoid Removal. This procedure may be applied to each of four injector solenoids installed on the engine.

WARNING

The components mounted on the engine are very close to the propeller. Secure all electrical power as prescribed below prior to starting this procedure.

1. Position the split master switch OFF.
2. Position FADEC A and B power switches OFF.
3. Locate circuit breaker CB003, START and pull OPEN.
4. Locate circuit breaker CB029, BOOST and pull OPEN.
5. Position cockpit fuel selector OFF.
6. Remove upper and lower cowl in accordance with Chapter 71 – *Power Plant*.

CAUTION

In the next several steps, fuel lines will be opened. Residual fuel may be present in these lines. Remove all sources of ignition from the work area and prepare a fuel rated container to capture fuel released as lines are opened. Cap all fuel lines after opening to prevent leakage and entry of contaminants. Pack the immediate area around the fuel lines with absorbent materials to absorb any fuel spillage.

7. Perform Fuel Injector Solenoid Removal in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 10.

This completes the Fuel Injector Solenoid Removal procedure.

FUEL INJECTOR REMOVAL

The following procedure performs Fuel Injector removal. This procedure may be applied to each of four injectors installed on the engine.

CAUTION

Fuel injector nozzles are not field serviceable components. Maintenance of fuel injector nozzle bodies is limited to replacement in the event of a failure for any reason. Never attempt to correct an injector fault or obstruction by insertion of an object into either the supply or spray end of injector body.

1. Perform Fuel Injector Solenoid removal in accordance with Liberty Maintenance Manual Chapter 73.
2. Perform Fuel Injector Removal in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 10.

This completes the Fuel Injector removal procedure.

FUEL INJECTOR INSTALLATION

The following procedure performs Fuel Injector Installation. This procedure may be applied to each of four fuel injectors installed on the engine.

WARNING

The components mounted on the engine are very close to the propeller. Secure all electrical power as prescribed below prior to starting this procedure.

1. Position the split master switch OFF.
2. Position FADEC A and B power switches OFF.
3. Locate circuit breaker CB003, START and pull OPEN.
4. Locate circuit breaker CB029, BOOST and pull OPEN.
5. Position cockpit fuel selector OFF.
6. Remove upper and lower cowl in accordance with Chapter 71 – *Power Plant*.

CAUTION

In the next several steps, fuel lines will be opened. Residual fuel may be present in these lines. Remove all sources of ignition from the work area and prepare a fuel rated container to capture fuel released as lines are opened. Cap all fuel lines after opening to prevent leakage and entry of contaminants. Pack the immediate area around the fuel lines with absorbent materials to absorb any fuel spillage.

7. Perform Fuel Injector Installation in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 10.
8. Perform Fuel Injector Solenoid Installation in accordance with Liberty Maintenance Manual Chapter 73. Do not perform FADEC Level I Diagnostics or Engine Operational Check at this time.
9. Position the cockpit fuel selector ON.
10. Locate circuit breaker CB029, BOOST and push to CLOSE.
11. Position the split master switch ON.
12. Position the fuel boost pump switch ON.
13. Verify the fuel boost pump is running and normal fuel pressure is reported on the VM1000FX display.
14. Inspect the fuel injector and connections for indications of leaks.

CAUTION

If any of the fuel connections leak, immediately turn off the fuel boost pump and master switch. Refer to the troubleshooting guide at the end of this section for corrective action.

15. Position the fuel boost pump switch OFF.
16. Position the split master switch OFF.
17. Locate circuit breaker CB003, START and push to CLOSE.
18. Perform a FADEC Level I Diagnostic in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 8.
19. Perform an Engine Operational Check in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 6.
20. Install the upper and lower cowl in accordance with Chapter 71 – *Power Plant*.

This completes the Fuel Injector Installation procedure.

FUEL INJECTOR SOLENOID INSTALLATION

The following procedure performs Fuel Injector Solenoid Installation. This procedure may be applied to each of four fuel injector solenoids installed on the engine.

WARNING

The components mounted on the engine are very close to the propeller. Secure all electrical power as prescribed below prior to starting this procedure.

1. Position the split master switch OFF.
2. Position FADEC A and B power switches OFF.
3. Locate circuit breaker CB003, START and pull OPEN.
4. Locate circuit breaker CB029, BOOST and pull OPEN.
5. Position cockpit fuel selector OFF.
6. Remove upper and lower cowl in accordance with Chapter 71 – *Power Plant*.

CAUTION

In the next several steps, fuel lines will be opened. Residual fuel may be present in these lines. Remove all sources of ignition from the work area and prepare a fuel rated container to capture fuel released as lines are opened. Cap all fuel lines after opening to prevent leakage and entry of contaminants. Pack the immediate area around the fuel lines with absorbent materials to absorb any fuel spillage.

7. Perform Fuel Injector Solenoid Installation in accordance with Liberty Maintenance Manual Chapter 73. Do not perform FADEC Level I Diagnostics or Engine Operational Check at this time.
8. Position the cockpit fuel selector ON.
9. Locate circuit breaker CB029, BOOST and push to CLOSE.
10. Position the split master switch ON.
11. Position the fuel boost pump switch ON.
12. Verify the fuel boost pump is running and normal fuel pressure is reported on the VM1000FX display.
13. Inspect the fuel injector and connections for indications of leaks.

CAUTION

If any of the fuel connections leak, immediately turn off the fuel boost pump and master switch. Refer to the troubleshooting guide at the end of this section for corrective action.

14. Position the fuel boost pump switch OFF.
15. Position the split master switch OFF.
16. Locate circuit breaker CB003, START and push to CLOSE.
17. Perform a FADEC Level I Diagnostic in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 8.
18. Perform an Engine Operational Check in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 6.
19. Install the upper and lower cowl in accordance with Chapter 71 – *Power Plant*.

This completes the Fuel Injector Solenoid Installation procedure.

FUEL INJECTOR OPERATIONAL CHECK AND INSPECTION

Perform this procedure to inspect and check the operation of the fuel injectors.

WARNING

The components mounted on the engine are very close to the propeller. Secure all electrical power as prescribed below prior to starting this procedure.

1. Position the split master switch OFF.
2. Position FADEC A and B power switches OFF.
3. Locate circuit breaker CB003, START and pull OPEN.
4. Locate circuit breaker CB029, BOOST and pull OPEN.
5. Position cockpit fuel selector OFF.
6. Remove upper and lower cowl in accordance with Chapter 71 – *Power Plant*.
7. Inspect each of four injector assemblies installed on the engine. Verify no indication fuel leaks, mechanical, solenoid or solenoid wire damage. In the event faults are detected, do not continue. Refer to the Fuel Injector Troubleshooting Guide at the end of this section for corrective action.
8. Position the cockpit fuel selector ON.
9. Locate circuit breaker CB029, BOOST and push to CLOSE.
10. Position the split master switch ON.
11. Position the fuel boost pump switch ON.
12. Verify the fuel boost pump is running and normal fuel pressure is reported on the VM1000FX display.
13. Inspect the fuel injector assemblies and connections for indications of leaks.

CAUTION

If any of the fuel connections leak, immediately turn off the fuel boost pump and master switch. Refer to the troubleshooting guide at the end of this section for corrective action.

14. Position the fuel boost pump switch OFF.
15. Position the split master switch OFF.
16. Locate circuit breaker CB003, START and push to CLOSE.
17. Perform a FADEC Level I Diagnostic in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 8.

18. Perform an Engine Operational Check in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 6.
19. Install the upper and lower cowl in accordance with Chapter 71 – *Power Plant*.

This completes the Fuel Injector Operational Check and Inspection procedure.

Section 02-04 Fuel Injector Troubleshooting Guide

Table 73-4 FUEL INJECTION SYSTEM TROUBLESHOOTING TABLE:

Complaint	Possible Cause	Remedy
Rough idle	Injector restricted	• Replace
	Injector solenoid fault	• Replace
Poor acceleration	Worn throttle linkage	• Repair
	Injector restricted	• Replace
	Injector solenoid fault	• Replace
Engine runs rough	Injector restricted	• Replace
	Injector solenoid fault	• Replace
Leaking injector	Injector torque incorrect	• Torque in accordance with Teledyne Continental Motors Maintenance Manual M-22 Appendix B.
	“B” nut torque incorrect	
	Faulty injector	• Replace

Section 73-03 Fuel Indicating

Section 03-01 Fuel Pressure Sensors

Engine fuel pressure indication is measured electronically at the fuel distribution block. In the airplane, fuel pressure indication is derived electronically in the FADEC and sent to the VM1000FX integrated engine instrument display.

Engine fuel flow is calculated internally by the FADEC and provided to the VM1000FX.

Section 03-02 Periodic Maintenance

Fuel pressure sensors are not field serviceable units. Maintenance is limited to remove and replace on condition. Periodic maintenance of the fuel pressure sensors entails an operational check performed at intervals specified in the Liberty Maintenance Manual, Chapter 05 Engine Operational Checklist. Operation checks are performed in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 6.

Section 03-03 Fuel Pressure Sensor Procedures

The following procedures perform fuel pressure sensor removal, installation, operational checks and inspections. Pressure sensors are not routinely removed for periodic maintenance. Operational checks performed during periodic maintenance determine if a fault condition exists that requires pressure sensor removal and installation procedures to be performed.

FUEL PRESSURE SENSOR REMOVAL

Perform this procedure to remove the fuel pressure sensor.

NOTE

Fuel pressure sensors are not a field serviceable item; maintenance is limited to removal and replacement.

WARNING

The components mounted on the engine are very close to the propeller. Secure all electrical power as prescribed below prior to starting this procedure.

1. Position the split master switch OFF.
2. Position FADEC A and B power switches OFF.
3. Locate circuit breaker CB003, START and pull OPEN.
4. Locate circuit breaker CB029, BOOST and pull OPEN.
5. Position cockpit fuel selector OFF.
6. Remove upper and lower cowl in accordance Chapter 71 – *Power Plant*.

CAUTION

In the next several steps, fuel lines will be opened. Residual fuel may be present in these lines. Remove all sources of ignition from the work area and prepare a fuel rated container to capture fuel released as lines are opened. Cap all fuel lines after opening to prevent leakage and entry of contaminants. Pack the immediate area around the fuel lines with absorbent materials to absorb any fuel spillage.

7. Ensure there is no pressure in the fuel system by loosing a fuel line to the side of the block that the fuel pressure sensor is to be removed from.
8. Cut and remove wire ties as necessary to allow sufficient freedom in engine Low Voltage Harness to remove sensor cable.
9. Disconnect engine harness circular connector (P9 or P10) from affected sensor.
10. Unthread fuel pressure sensor from fuel distribution block.

This completes the Fuel Pressure Sensor Removal procedure.

FUEL PRESSURE SENSOR INSTALLATION

Perform this procedure to install the fuel pressure sensor

WARNING

The components mounted on the engine are very close to the propeller. Secure all electrical power as prescribed below prior to starting this procedure.

1. Position the split master switch OFF.
2. Position FADEC A and B power switches OFF.
3. Locate circuit breaker CB003, START and pull OPEN.
4. Locate circuit breaker CB029, BOOST and pull OPEN.
5. Position cockpit fuel selector OFF.
6. Remove upper and lower cowl in accordance with Chapter 71 – *Power Plant*.
7. Clean sensor mounting threads as necessary and apply thread sealant (Teledyne Continental Motors P/N 646940) to second and third threads of sensor.
8. Thread sensor into fuel distribution block and torque to 130-150 in/lbs.
9. Using a digital voltmeter, measure resistance between engine harness connector body and engine ground. Resistance must be less than 0.5 ohms.
10. Reconnect engine harness to sensor (P9 or P10). Orient alignment keys of connector and push connector together until connector body bottoms.
11. Twist locking ring clockwise while maintaining toward sensor until it “snaps” into position and allows no further rotation. When properly mated and locked, bayonet pins are visible in locking ring observation port.
12. Connect fuel lines at fuel distribution block; torque to 55-60 in/lbs.
13. Position the cockpit fuel selector ON.
14. Locate circuit breaker CB029, BOOST and push to CLOSE.
15. Position the split master switch ON.
16. Position the fuel boost pump switch ON.
17. Verify the fuel boost pump is running and normal fuel pressure is reported on the VM1000FX display.
18. Inspect the fuel distribution block, sensors and connections for indications of leaks.

CAUTION

If any of the fuel connections leak, immediately turn off the fuel boost pump and master switch. Refer to the troubleshooting guide at the end of this section for corrective action.

19. Position the fuel boost pump switch OFF.
20. Position the split master switch OFF.
21. Locate circuit breaker CB003, START and push to CLOSE.
22. Perform fuel injection system diagnostic in accordance with Teledyne Continental Motors Maintenance Manual M-22, Chapter 8.
23. Install the upper and lower cowl in accordance with Chapter 71 – *Power Plant.*

This completes the Fuel Pressure Sensor Installation procedure.

FUEL PRESSURE SENSOR OPERATIONAL CHECK AND INSPECTION

Perform this procedure to inspect and check the operation of the fuel pressure sensor.

WARNING

The components mounted on the engine are very close to the propeller. Secure all electrical power as prescribed below prior to starting this procedure.

1. Position the split master switch OFF.
2. Position FADEC A and B power switches OFF.
3. Locate circuit breaker CB003, START and pull OPEN.
4. Locate circuit breaker CB029, BOOST and pull OPEN.
5. Position cockpit fuel selector OFF.
6. Remove upper and lower cowl in accordance with Chapter 71 – *Power Plant*.
7. Inspect each of two fuel pressure sensor assemblies installed on the engine. Verify no indication fuel leaks, mechanical faults or sensor wire damage. In the event faults are detected, do not continue. Refer to the Fuel Pressure Sensor Troubleshooting Guide at the end of this section for corrective action.
8. Position the cockpit fuel selector ON.
9. Locate circuit breaker CB029, BOOST and push to CLOSE.
10. Position the split master switch ON.
11. Position the fuel boost pump switch ON.
12. Verify the fuel boost pump is running and normal fuel pressure is reported on the VM1000FX display.
13. Inspect the fuel pressure sensor assemblies for indication of leaks.

CAUTION

If any of the fuel connections leak, immediately turn off the fuel boost pump and master switch. Refer to the troubleshooting guide at the end of this section for corrective action.

14. Position the fuel boost pump switch OFF.
15. Position the split master switch OFF.
16. Locate circuit breaker CB003, START and push to CLOSE.

17. Perform a FADEC Level I Diagnostic in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 8.
18. Perform an Engine Operational Check in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 6.
19. Install the upper and lower cowl in accordance with Chapter 71 – *Power Plant*.
20. This completes the Fuel Pressure Sensor Operational Check and Inspection procedure.

Section 03-04 Fuel Pressure Sensor Troubleshooting Guide

Complaint	Possible Cause	Remedy
Low fuel pressure indication	Restricted flow to fuel manifold block and fuel pressure sensors	• Check for restriction between pump and manifold block
	Inadequate pressure from fuel pump	• Adjust pump in accordance with Teledyne Continental Motors Maintenance Manual M-22 section 23-2.
High fuel pressure indication	Restricted flow at injector	• Check for restriction; replace injector as required.
	Restricted recirculation passage in fuel pump	• Replace fuel pump
	Excessive pressure from fuel pump	• Adjust pump in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter 9.
	Fuel pressure sensor	• Replace sensor
Fluctuating or erroneous fuel pressure indication	Fuel vapor as a result of high ambient temperatures.	• Operate electric fuel pump for several minutes. If no change, check for clogged vent in fuel pump vapor separator cover in accordance with Teledyne Continental Motors Maintenance Manual M-22 Chapter Clean only with solvent, no wires.
	Defective fuel pressure sensor	• Replace sensor
Excessive pressure split reported in level I diagnostic	Fuel pressure sensor fault	• Replace sensor
	low voltage harness fault	• Repair

Table 73-5 Fuel Pressure Sensor Troubleshooting Guide

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CHAPTER 74

ENGINE IGNITION

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Table of Contents

SECTION 74-00	GENERAL	5
SECTION 00-01	TORQUE VALUES FOR SPARK PLUGS AND HIGH TENSION LEADS	5
SECTION 74-10	ELECTRICAL POWER SUPPLY	7
SECTION 74-20	DISTRIBUTION	9
SECTION 74-30	SWITCHING	11
SECTION 74-40	FADEC TROUBLESHOOTING	13
SECTION 40-01	FADEC PROCEDURES	14
FADEC ECU MODULE REMOVAL		15
FADEC ECU INSTALLATION		17

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Section 74-00 General

This chapter provides a descriptive overview of the Full Authority Digital Engine Controller (FADEC) engine ignition system installed on the airplane. More detailed information for repairs and maintenance on systems and components specific to the IOF-240B engine (in particular, the FADEC system) are provided in the current release of the Teledyne Continental Motors Maintenance Manual for IOF-240B series engines, TCM P/N: M-22.

The ignition system of the IOF-240B engine is part of the FADEC. Unlike conventional magneto ignition systems, its timing is electronically determined by the FADEC computers, and is not adjustable by maintenance technicians.

The ignition system uses TCM 630049 or Champion RHB38E spark plugs. Clean and gap plugs in accordance with plug manufacturer's specifications.

Section 00-01 Torque Values for Spark Plugs and High Tension Leads

Unless otherwise specified in the Teledyne Continental Motors Maintenance Manual for IOF-240B series engines, TCM P/N: M-22, the torque on the spark plugs and high tension connection are as follows:

- The torque on the spark plugs needs to be 300-360 in-lbs.
- The torque on the "B" nut on the high tension connection of the spark plug needs to be 120-150 in-lbs.

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Section 74-10 Electrical Power Supply

Unlike conventional magneto ignition systems, the ignition subsystem of the FADEC is not self-powered, but requires DC power from the airplane's electrical system.

To provide necessary system redundancy, FADEC has two power sources. In the airplane, the primary power source, labeled FADEC A, is the airplane's main power distribution bus, which is powered by the alternator and/or the airplane's primary battery.

The secondary power source, labeled FADEC B, is powered by a separate battery which is dedicated only to this application. When the airplane's main power distribution bus is energized, a charging circuit constantly recharges the FADEC B or secondary battery, thus indirectly powering the FADEC B bus.

In the event of failure of the airplane primary DC system, including discharge of the primary battery, the secondary battery will power FADEC, by way of the FADEC B connection, for a period sufficient to locate and land at a suitable airport.

FADEC is fully operational when powered by the FADEC A bus, the FADEC B bus, or both. Switches are provided to check operation on both systems before flight. The FADEC Health Status Annunciator contains five LED annunciations providing FADEC health and status information as follows:

- FADEC WARN - Illuminates red upon a potential total failure of the FADEC system
- FADEC CAUTION - Illuminates amber when there is a fault in the FADEC system
- PPWR FAIL - Illuminates red upon failure of the airplane primary power system
- EBAT FAIL – Illuminates amber in the event of a secondary battery failure or in the event primary power fails and FADEC is running exclusively from the secondary battery.
- FUEL PUMP - Illuminates green when FADEC calls for boost pump operation.

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Section 74-20 Distribution

FADEC incorporates two Electronic Control Unit (ECU) modules; each module in turn contains two high voltage coil packs and associated circuitry. The module installed on the right side of the firewall (as seen from the pilot's seat) connects to the top and bottom spark plugs on the rear pair of cylinders (cylinder 1 on the right side of the engine, cylinder 2 on the left side of the engine). The module installed on the left side of the firewall is connects to the top and bottom spark plugs on the forward pair of the cylinders (cylinder 3 on the right, cylinder 4 on the left). See Figure 74-1.

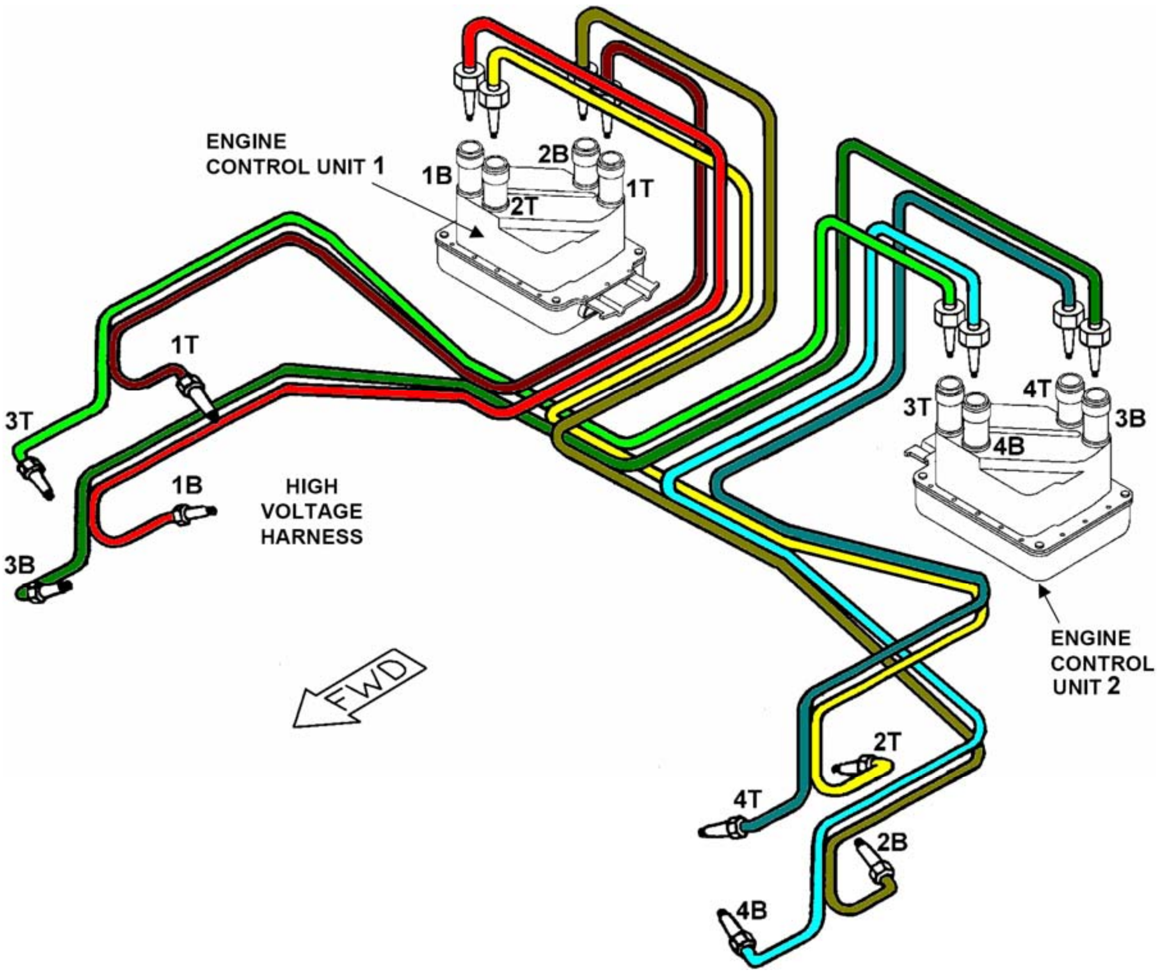


Figure 74-1 ECU Connections to the Spark Plugs

Each Electronic Control Unit (ECU) module contains two microprocessors; each microprocessor controls two opposing cylinders. A four-cylinder installation therefore requires two (2) modules. ECU module #1 controls cylinders 1 and 2. ECU module #2 controls cylinders 3 and 4.

Each microprocessor also provides secondary control for the opposing cylinder's fuel injector, thus providing redundancy. The "A" and "B" channel microprocessors monitor each other's performance. In the event one microprocessor stops controlling, the other takes over control of the cylinders fuel injector. Ignition and fuel injection redundancy is provided as follows:

Engine Cylinder	Primary Control	Secondary Control
1	ECU 1, Channel A	ECU 1, Channel B
2	ECU 1, Channel B	ECU 1, Channel A
3	ECU 2, Channel A	ECU 2, Channel B
4	ECU 2, Channel B	ECU 2, Channel A

Table 74-1 ECU Channels and Engine Cylinders

FADEC ignition is of the “waste spark” type. In this type of ignition, each spark plug fires once per engine revolution (twice per complete 4-stroke cycle), once during its cylinder’s compression stroke (the “working” spark) and once during its cylinder’s exhaust stroke (the “waste” spark). This second spark is used to burn any unspent fuel or other combustibles before they leave the engine.

Section 74-30 Switching

A conventional aircraft ignition switch controls the operation of the ignition portion of the FADEC. Note this switch is separate from, and independent of, the FADEC A and FADEC B power switches, which control operation of the FADEC system as a whole (both ignition and fuel control functions).

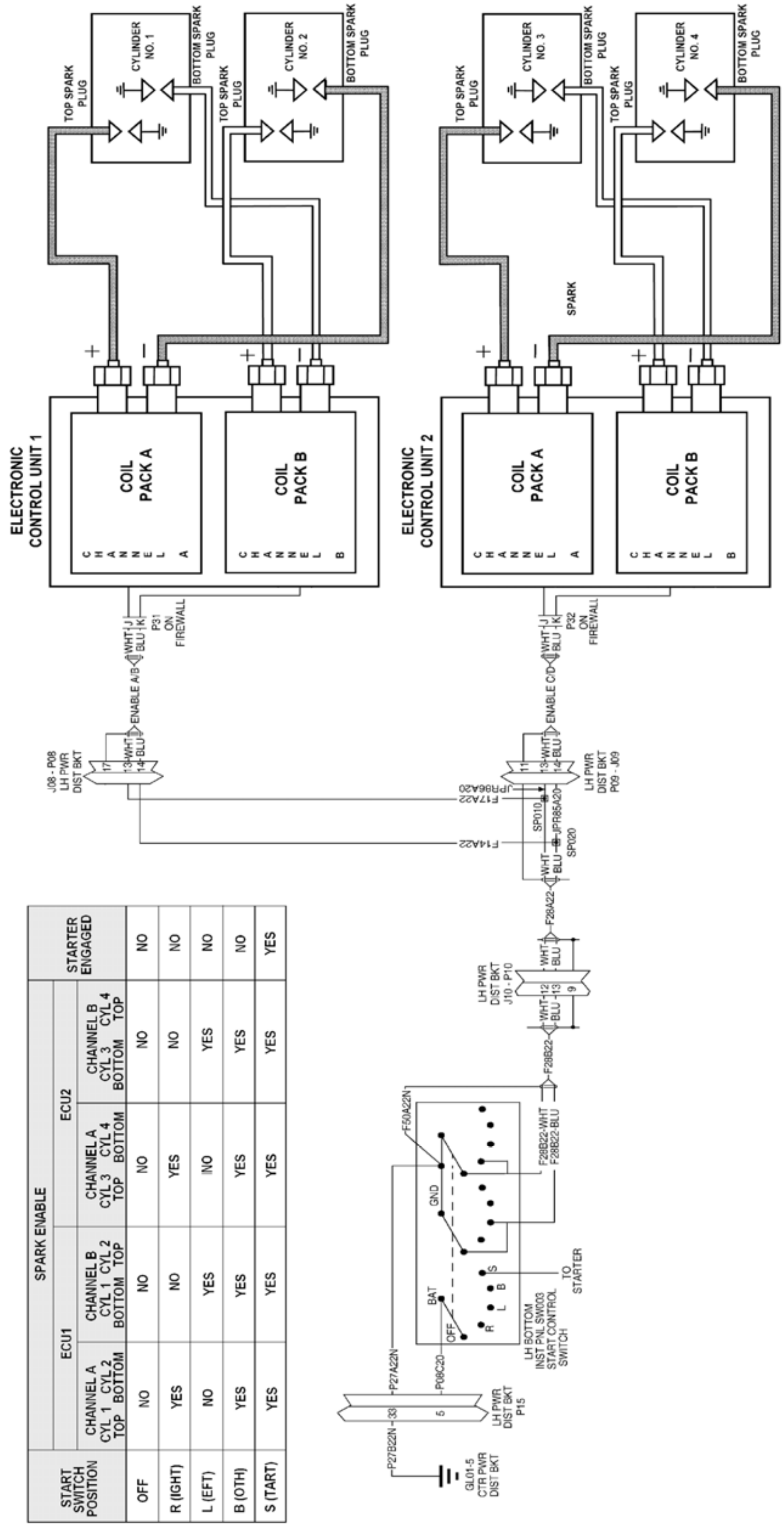
With the ignition switch in the OFF position, disables all spark plugs from fire.

Placing the ignition switch in the R position enables the spark plugs connected to channel A of ECU #1 (Cylinder 1 top, Cylinder 2 bottom) and to channel A of ECU #2 (Cylinder 3 top, and Cylinder 4 bottom) to fire as needed.

Placing the ignition switch in the L position enables the spark plugs connected to channel B of ECU #1 (Cylinder 1 bottom, Cylinder 2 top) and to channel B of ECU #2 (Cylinder 3 bottom, and Cylinder 4 top) to fire as needed.

Placing the ignition switch in the BOTH or START position enables the spark plugs connected to ECU #1 (Cylinder 1 top/bottom, Cylinder 2 top/bottom) and to ECU #2 (Cylinder 3 top/bottom, and Cylinder 4 top/bottom) to fire as needed.

Figure 74-2 shows the ignition switch and ECUs. The figure also shows how they work together to fire the spark plugs.



START SWITCH POSITION	SPARK ENABLE				STARTER ENGAGED
	ECU1		ECU2		
	CHANNEL A CYL 1 TOP CYL 2 BOTTOM	CHANNEL B CYL 1 BOTTOM CYL 2 TOP	CHANNEL A CYL 3 TOP CYL 4 BOTTOM	CHANNEL B CYL 3 BOTTOM CYL 4 TOP	
OFF	NO	NO	NO	NO	NO
R (IGHT)	YES	NO	YES	NO	NO
L (EFT)	NO	YES	INO	YES	NO
B (OTH)	YES	YES	YES	YES	NO
S (TART)	YES	YES	YES	YES	YES

Figure 74-2 Ignition System

Section 74-40 FADEC Troubleshooting

If a problem with the ignition system is suspected, perform a system diagnostic check as described in Chapter 6 - *Troubleshooting* of TCM IOF-240-B4B Maintenance Manual; P/N: M-22. If this check does not indicate a system fault, then refer to TCM Manual M-22, Chapter 4.

Table 74-2 gives troubleshooting and fault isolation information.

Complaint	Possible Cause	Remedy
Engine will not start	Ignition switch OFF or grounded wires	Turn switch ON, check for grounded wires
	Spark plugs fouled, improperly gapped, or loose	Remove and clean. Check the gap of each spark plug in accordance with spark plug manufacturer's specifications. Install and torque to 300-360 in/lbs.
	Open circuit in FADEC Low Voltage Harness	Inspect harness, ensure all connectors properly fastened to components
	FADEC fault	Perform FADEC fault Isolation per TCM manual section 12-7.
Rough idling	Spark plugs fouled or improperly gapped	Remove and clean. Check the gap of each spark plug in accordance with spark plug manufacturer's specifications. Install and torque to 300-360 in/lbs.
	High Voltage harness improperly installed	Verify high voltage lead connections in accordance with TCM manual section 12-8
	High energy leak in ignition harness	Inspect, replace lead as required
	High Voltage harness cable damaged	Inspect, replace lead as required
	FADEC fault	Perform FADEC fault Isolation per TCM manual section 12-7.
Rough engine at speeds above idle	Loose or improperly gapped spark plugs	Remove and clean. Check the gap of each spark plug in accordance with spark plug manufacturer's specifications. Install and torque to 300-360 in/lbs.
	High energy leak in ignition harness	Inspect, replace lead as required
	FADEC fault	Perform FADEC fault Isolation per TCM manual section 12-7.

Complaint	Possible Cause	Remedy
	Malfunctioning fuel injector assembly	See TCM manual sec. 12-3, FADEC fuel system troubleshooting
Sluggish operation and/or excessive RPM drop	Fouled or dead spark plugs	Remove and clean. Check the gap of each spark plug in accordance with spark plug manufacturer's specifications. Install and torque to 300-360 in/lbs.
	Improperly gapped spark plugs	Adjust to proper gap
	FADEC fault	Perform FADEC fault isolation per TCM manual section 12-7.

Table 74-2 Ignition Fault Isolation Table

Section 40-01 FADEC Procedures

The procedure to remove and install most of the components of the FADEC system is covered in the service manual TCM IOF-240-B4B Maintenance Manual; P/N: M-22. The procedures in this section are those that are specific to the Liberty Aerospace, Inc. XL2 airplane.

CAUTION

The FADEC ECU modules are not interchangeable. When replacing the ECU module, it is required to replace the existing module with an ECU module that has the same part number.

WARNING

THE OUTPUT OF THE ECUS TO THE SPARK PLUGS IS HIGH TENSION VOLTAGE. DO NOT HANDLE THE WIRING WHILE ENGINE IS RUNNING OR ATTEMPTING TO START.

FADEC ECU MODULE REMOVAL

Perform this procedure to remove the FADEC ECU module from the airplane.

CAUTION

This device contains electro-statically sensitive parts. Take ESD precautions before handling this or any other electronic device on the airplane.

1. Pull circuit breaker BAT 1 (CB001) to OPEN.
2. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
3. Remove the upper and lower engine cowling.
4. Use a 7/8-inch open-end wrench to loosen and remove the four spark plug connectors from the front of the ECU.
5. Use a 5/32-inch hex wrench to loosen and remove the ground wires from the ECU.
6. If the airplane has the air/oil separator mounted between the ECUs, use a 7/16-inch socket to remove the three bolts that secure the air/oil separator mounting bracket to the ECUs.
7. Position the air/oil separator away of the effected ECU.

NOTE

The next step removes the bolts that hold the ECU module to the firewall. Between the module and the firewall there is a stand-off spacers. There are two sizes of these stand-off spacers. Be sure to note the location of the space as they are removed.

8. Use a 7/16-inch open-end wrench or socket to remove the remaining bolts securing the module to the firewall. Retain the spacers for later use to install the replacement module.
9. Carefully lift the module to gain access to the ECU connector.
10. Cut and remove the safety wire from the connector.
11. Cut and remove the cable tie wrapped around the connector and bracket.
12. Use a flat blade screwdriver to loosen the two screws securing the connector to the module. Disconnect the ECU cable and connector.

CAUTION

There is a grey wire mess gasket and a GelTeK sealant strip between the connector and the module. This gasket and sealant strip must be discarded and a replacement used when reconnecting the connector to the module.

13. Remove the ECU from the airplane.

CAUTION

If it is required to protect the end of the connectors of the ECU cable and/or the connectors on the ECU module, do not use any type of tape that may come in contact with the electrical connections. The tape's adhesive can leave a residue that can cause damage to the connections.

14. If replacing the ECU module later, install caps on to all connectors. Reposition the air/oil separator and secure it with the hardware removed in step 6 above. Install the upper and lower engine cowlings.

This completes the FADEC ECU Module Removal procedure.

FADEC ECU INSTALLATION

Perform this procedure to install the FADEC ECU module from the airplane.

CAUTION

The FADEC ECU modules are not interchangeable. When replacing the ECU module, it is required to replace the existing module with an ECU module that has the same part number.

CAUTION

This device contains electro-statically sensitive parts. Take ESD precautions before handling this or any other electronic device on the airplane.

CAUTION

There is a grey wire mesh gasket and a GelTeK sealant strip that are installed between the connector and the ECU. Use only a new gasket and sealant strip when installing the replacement ECU. The ECU comes with these parts in sealed bags as part of the kit. Do not open the bag until the gaskets are needed and only handle the gaskets with clean gloved hands

1. If replacing the previous module with a replacement ECU module, go to step 7 below.
2. Pull circuit breaker BAT 1 (CB001) to OPEN.
3. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
4. Remove the upper and lower engine cowlings.
5. If the airplane has the air/oil separator mounted between the ECU modules, use a 7/16-inch socket to remove the three bolts that secure the air/oil separator mounting bracket to the modules.
6. Position the air/oil separator away of the effected module.
7. Remove connector caps, inspecting the connectors for any type of contamination.

CAUTION

There is a grey wire mesh gasket and a GelTeK sealant strip that are installed between the connector and the ECU module. Use only a new gasket and sealant strip when installing the replacement module. The ECU module comes with these parts in sealed bags as part of the kit. Do not open the bag until the gaskets are needed and only handle the gaskets with clean, gloved hands

CAUTION

Inspect the grey wire mesh gasket before installation. There are to be no protruding wires from the gasket.

8. Position a new GelTek sealant strip and wire mesh gasket on to the ECU connector. See Figure 74-3.
9. Connect the ECU cable connector to the bottom of the module.
10. Secure the connector with a large tie wrap.
11. Secure the connector screws with safety wire.

NOTE

The next step installs the bolts that hold the ECU to the firewall. Between the module and the firewall there is a stand-off spacers. There are two sizes of these stand-off spacers. Be sure to note the location of the space as they are removed. See Figure 74-3

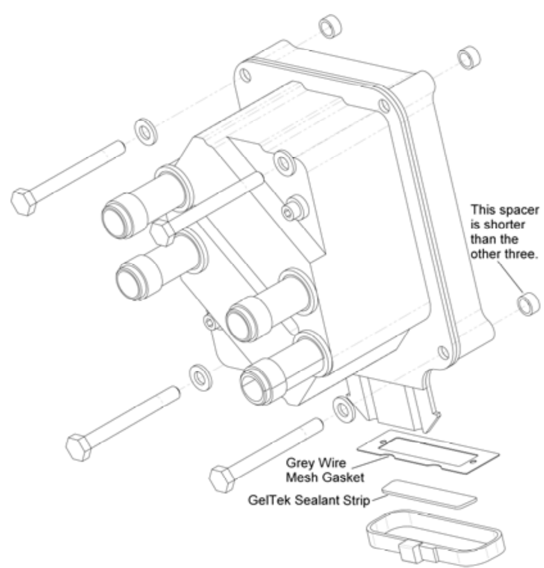


Figure 74-3 ECU Bolt and Spacer Locations

12. Position the ECU on the firewall and secure the ECU with the bolts and spacers removed in step 8 in the FADEC ECU Module Removal procedure on page 15.
13. If the airplane has the air/oil separator that mounts between the modules, position the air/oil separator between the modules and secure with hardware removed in step 6 in the FADEC ECU Module Removal procedure on page 15.
14. Connect the ground wires to the hex socket screws on the upper surface of the ECU module using a 5/32 hex wrench.

NOTE

The spark plug wires have metal identification tags that indicate where to connect the wire to the ECU. The ECU module has matching identification tags to match the tags on the wires.

15. Connect the spark plug wires to the ECUs securing the connectors with a 7/8-inch open-end wrench. Torque the nuts to 120-150 in-lbs.
16. Run an Engine Level 1 Diagnostic system check. To see the procedure to do an Engine Level 1 Diagnostic system check, see Chapter 71 – *Power Plant*.

This completes the FADEC ECU Installation procedure.

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CHAPTER 75

AIR INDUCTION

NOTE

The information that was in this chapter has been moved to Chapter 71 – Power Plant and Chapter 72 – Engine.

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CHAPTER 76

ENGINE CONTROLS

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Table of Contents

SECTION 76-00 GENERAL 5

SECTION 00-01 FADEC SYSTEM DESCRIPTION AND FUNCTIONAL OVERVIEW 6
SECTION 00-02 HEALTH STATUS ANNUNCIATOR AND POWER TRANSFER CHECK PROCEDURES 7
FADEC POWER TRANSFER CHECK 8

SECTION 76-10 POWER CONTROL 11

SECTION 10-01 POWER (THROTTLE) CABLE REMOVAL AND REPLACEMENT 12
THROTTLE CABLE REMOVAL 13
THROTTLE CABLE INSTALLATION 14
THROTTLE CABLE RIGGING PROCEDURE 15

SECTION 76-20 EMERGENCY SHUTDOWN 17

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Section 76-00 General

This chapter provides a descriptive overview of the control systems for the IOF-240-B engine installed on the airplane. Detailed information for routine line maintenance for each engine subsection or system is provided in the appropriate chapter. More detailed information for repairs and maintenance on systems and components specific to the IOF-240B engine FADEC system are provided in the current release of the Teledyne Continental Motors Maintenance Manual for IOF-240-B series engines, TCM p/n: M-22.

The IOF-240B engine installed in the airplane is equipped with a "PowerLink" ™ Full Authority Digital Engine Control (FADEC) manufactured by the Aerosance Corporation. This system controls both ignition timing and fuel delivery to the engine. The ignition section of the FADEC replaces the conventional magneto system, while the fuel section replaces the carburetor or hydro-mechanical fuel injection system found in conventional installations.

In a fixed-pitch propeller installation such as this airplane, primary power control is through a single throttle lever. Fuel mixture is controlled by FADEC; there is no cockpit mixture or pitch control. The following are additional engine controls:

- Include a manual knob for selection of alternate induction air (used only in case of blockage of the primary induction air filter),
- An ignition switch to select between the L and R channels of the FADEC system,
- FADEC A and FADEC B power switches to control electric power to the FADEC system
- A FADEC Health Status Annunciator (HSA) which indicates various FADEC malfunctions or operating modes
- A Boost Pump Mode Switch (BPMS)
- An emergency fuel shutoff

Section 00-01 FADEC System Description and Functional Overview

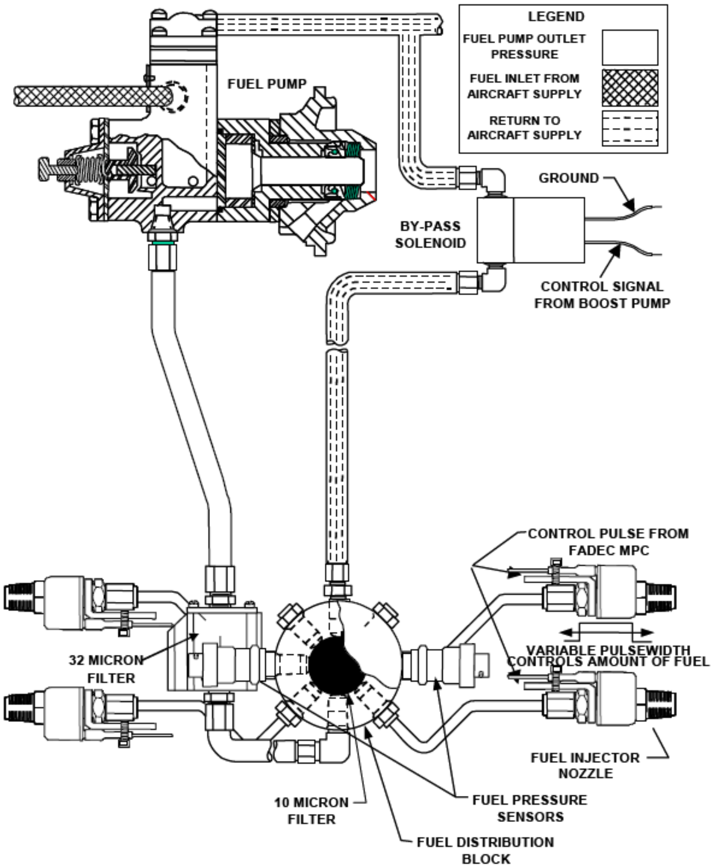

Figure 76-1 Fuel Supply System

Components of the FADEC system include dual Electronic Control Units (ECU) mounted on the engine compartment firewall. Each contains dual redundant microprocessors and dual high voltage coil sections. Each microprocessor controls ignition for two cylinders. Complete failure of a single microprocessor and/or its associated high voltage section will result only in the loss of spark to one of the two spark plugs in the two affected cylinders; fuel injection functions are shared by all four microprocessors in the system.

In the extremely unlikely event of complete failure of an entire ECU (both processors and both high voltage sections), the engine will continue to run at reduced power, as the ECU's are arranged to control cylinders paired on opposite sides of the engine. The remaining cylinders will respond normally to cockpit controls.

An engine-mounted sensor set supplies information to the two ECUs. Sensors include cylinder head temperature for each cylinder, exhaust gas temperature for each cylinder, temperature and absolute pressure of the air in the induction manifold, pressure of fuel supplied to the fuel injector nozzles, engine rotational speed, and crankshaft position. Engine speed and crankshaft position are sensed by dual magnetic pickups installed in the crankcase, which sense the passage of “targets” (holes) milled into the camshaft drive gear. Signals from the magnetic sensors are further conditioned by a unit installed on the engine accessory case in the left magneto opening. The right magneto opening is covered.

In addition, in fixed-pitch propeller installations, throttle position is sensed to the wide-open-throttle (WOT) position or RPM. When WOT is sensed, the FADEC fuel system adjusts fuel flow for the best-power configuration. At any other throttle systems, the FADEC fuel system adjusts flow as required for best-cruise operation.

To provide system redundancy, many sensors, including engine speed and crank position, intake manifold pressure and temperature, and fuel pressure, are installed in duplicate. In addition, should all information from any sensor(s) be lost, system software will revert to internally stored default values to allow continued engine operation.

The FADEC system manages each cylinder individually. As a combustion event for a given cylinder approaches, the FADEC calculates mass airflow based on engine speed and intake manifold air density (manifold pressure and air temperature). At the appropriate time, the coil for the cylinder’s fuel injector is energized for a specific length of time based on fuel pressure and calculated to provide the correct amount of fuel for best economy mixture (at intermediate throttle positions) or best power mixture (at wide-open throttle).

At the appropriate time before the cylinder reaches top dead center (TDC) on the compression stroke, the spark plugs are fired. Engine timing is established by the FADEC ECU’s and cannot be manually set or verified.

Calculation of the correct timing and duration of fuel injection (and of ignition timing when engine speed at or below 2000 RPM) is repeated for each subsequent combustion event.

The FADEC system has no direct control over throttle position, which is managed directly by the pilot. However, in the event of engine over-speed, the FADEC will attempt to limit the over-speed condition, initially by leaning the fuel mixture, then by sequentially disabling the ignition in one or more cylinders. Transient engine roughness may be experienced because of the FADEC’s attempts to limit engine over-speed.

Section 00-02 Health Status Annunciator and Power Transfer Check Procedures

A Health Status Annunciator (HSA) panel is provided to allow the pilot to monitor correct function of the FADEC and verify correct operation of its backup modes during a preflight check. This section contains the procedure to check the power transfer check.

FADEC POWER TRANSFER CHECK

Perform this procedure to check the FADEC power transfer.

1. Start engine per Aircraft Flight Manual / Pilot's Operating Handbook (AFM/POH).
2. Allow oil temperature to warm up to minimum 100 deg. F.
3. Point airplane into the wind, and free of all obstructions. Chock wheels or hold brakes (do not depend on parking brake).
4. Set throttle to 1700 RPM.
5. Check oil pressure 30-60 psig.
6. Set BPMS switch OFF>
7. Check fuel pressure within limits.
8. Set BPMS switch ON (pump will run continuously) or AUTO (pump will be commanded on or off by FADEC).
9. Check alternator load on VM1000FX.
10. Press TEST switch on HSA. Verify all bulbs illuminate.
11. Release test switch.
12. Turn FADEC PWR A switch OFF. Verify that there is no RPM drop or bump (RPM does not change). Verify AUX PWR lamp illuminates and BATT LO lamp may or may not illuminate.
13. Turn FADEC PWR A switch ON. Verify that there is no RPM drop or bump and that AUX PWR and BATT LO lamps extinguish.
14. Turn FADEC PWR B switch OFF. Verify that there is no RPM drop or bump and that BATT LO light illuminates.
15. Turn FADEC PWR B switch ON. Verify that there is no RPM drop or bump and that BATT LO light extinguishes.

WARNING

IF AN RPM DROP OCCURS DURING FADEC PWR A AND PWR B TRANSFER CHECK (ABOVE STEPS), DETERMINE FAULT AND REPAIR BEFORE FLIGHT.

16. Set ignition switch to L position, check RPM and HSA:
17. Verify RPM drop is no less than 10 RPMs and no more than 150 RPMs.
18. Verify HSA right side channel lamps ILLUMINATE.

NOTE

If ignition switch remains in L position for more than 30 seconds, the FADEC warning lamp will illuminate.

19. Return ignition switch to BOTH position, check RPM and HSA: There should be no drop in RPM.
20. Verify all HSA lamps extinguished.
21. Verify RPM drop is no less than 10 and no greater than 150.
22. Verify HSA left side channel lamps: ILLUMINATE

NOTE

If ignition switch remains in R position for more than 30 seconds, the FADEC warning will illuminate.

23. Return the ignition switch to the BOTH position. Check the RPM and the HSA. There should be no drop in RPM.
24. Verify all HSA lamps extinguished.

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Section 76-10 POWER CONTROL

The power cable, comprised of a fixed outer sheath and a moving inner element, transmits movement from the cockpit throttle lever to the air throttle unit at the top of the engine. Attachments at both ends secure the outer sheath to the airframe. A swaged spherical rod-end bearing at the rear end is attached to the throttle lever by means of an AN3-10 bolt and terminated using a self locking, castellated nut and a cotter pin. While a swaged spherical rod-end bearing at the front is attached to the air throttle bell-crank using the same means for termination as the rear attachment.

To prevent failure of the swage rod-end bearings at both ends of the cable, a washer of sufficient size is placed between the castle nut attached to the bolt and the bearing on the rod-end. The washer will interfere with any undetected separation of the bearing from the rod-end.

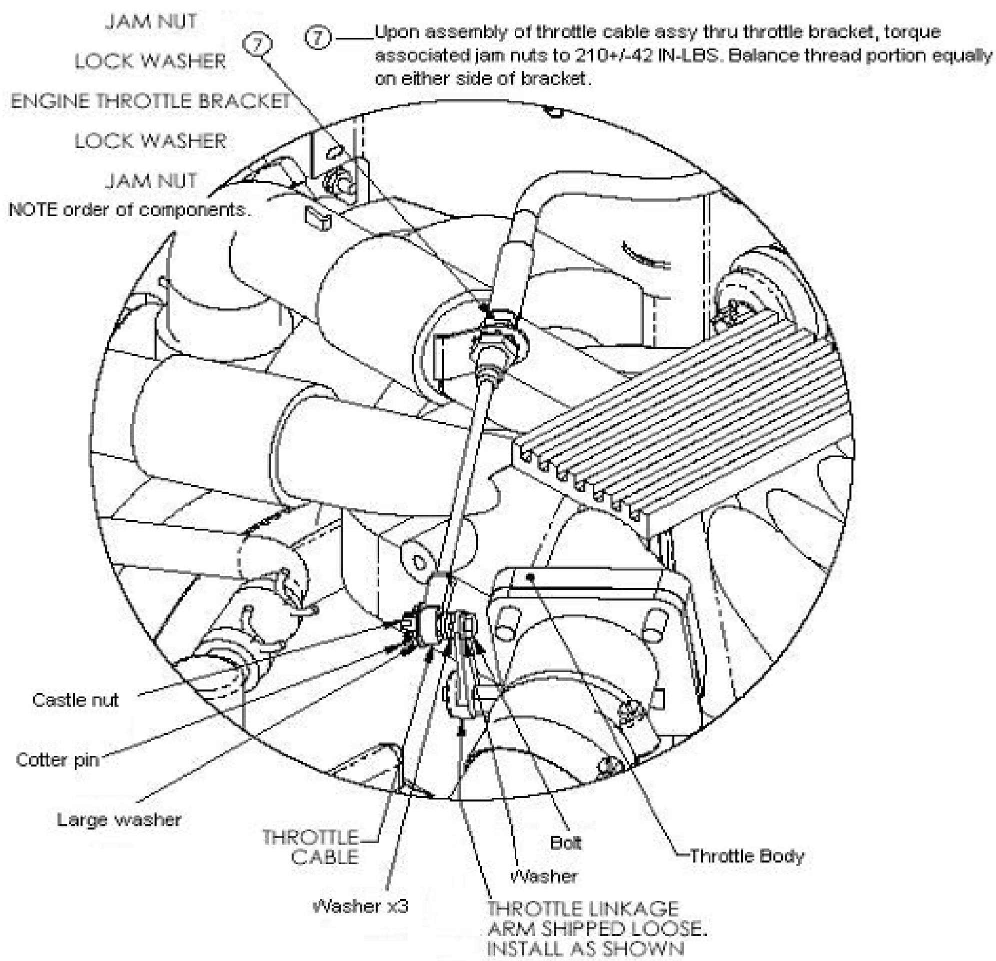

Figure 76-2 Throttle Cable Assembly

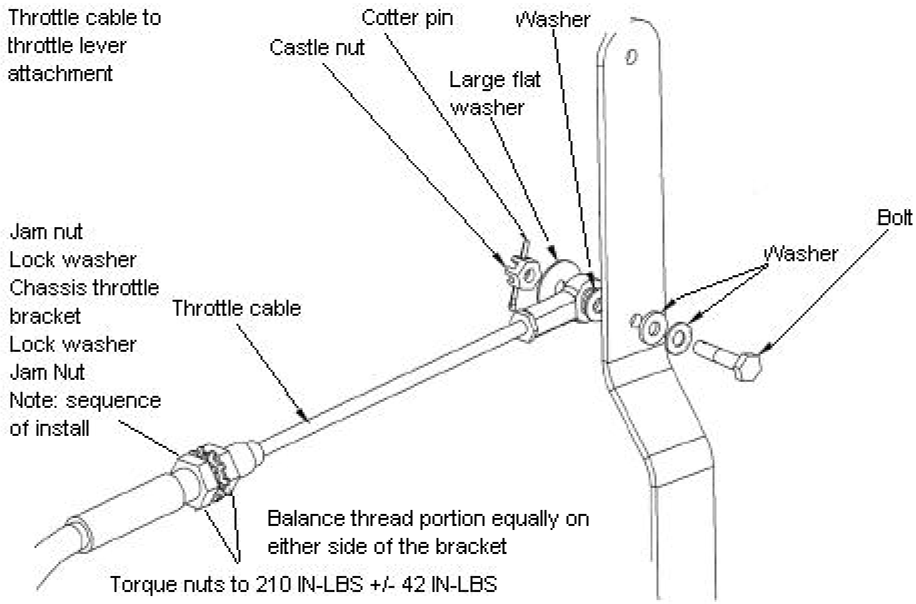

Figure 76-3 Throttle Cable and Lever Attachment

Section 10-01 Power (Throttle) Cable Removal and Replacement

This section contains the procedures to remove and install the power (throttle) cable.

THROTTLE CABLE REMOVAL

1. Remove belly panel.
2. Remove upper and lower engine cowls.
3. Remove and discard cotter pin on engine end of throttle cable.
4. Loosen castle nut and remove large washer. Note position of all washers.
5. Remove throttle cable rod end from bolt. Unthread rod end from throttle cable.
6. Remove washer stack from bolt, and remove bolt from throttle body lever.
7. Loosen rod end side jam nut and slide nut and lock washer off throttle cable.
8. Slide the throttle cable towards the cabin through engine bracket and rear engine baffle.
9. Remove cable side jam nut and lock washer by sliding off cable.
10. Lay throttle cable end on engine.
11. Loosen jam nut at firewall and slide jam nut and lock washer off cable.
12. Loosen and remove WDG clamp aft of firewall from cable.
13. Remove and discard cotter pin from throttle lever end of throttle cable.
14. Loosen castle nut and remove large washer. Note position of all washers.
15. Remove throttle cable rod end from bolt. Unthread rod end from throttle cable.
16. Remove washer stack from bolt, and remove bolt from throttle lever.
17. Loosen rod end side jam nut and slide nut and lock washer off throttle cable.
18. Remove cable side jam nut and lock washer by sliding off cable.
19. Slide the throttle cable towards the firewall through chassis bracket.
20. Remove firewall jam nut and lock washer by sliding off cable.
21. Pull throttle cable slowly through firewall pass-thru from under the aircraft.

THROTTLE CABLE INSTALLATION

1. Pass throttle cable through firewall pass-thru from under the aircraft moving towards the engine compartment taking care to rout cable up and thru rear engine baffle.
2. Reverse the above steps.
3. Torque all jam nuts to 210 IN-LBS +/- 42 IN-LBS.

THROTTLE CABLE RIGGING PROCEDURE

1. Ensure that rod ends at both ends of the cable are threaded approximately mid-way on the threaded ends.
2. Place the power lever in the cockpit at its rear most position.
3. Perform throttle rigging to current TCM Manual P/N IO-22 Chapter 5-2.6.

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Section 76-20 Emergency Shutdown

The normal means of shutting down the airplane engine is by moving the ignition switch to the OFF position. This grounds all four high voltage coils in the MPC units. This method is also the primary means of emergency engine shutdown.

Should the engine fail to stop, the following additional means of shutting down are available in order of preference:

1. Turn both FADEC PWR A and FADEC PWR B switches OFF.
2. Turn BPMS (Boost Pump Mode Switch) to OFF.

This disables both channels of the FADEC, interrupting both fuel injection and ignition. It also disables the airframe electric boost pump in case a FADEC fault or pilot action has commanded it ON.

3. Turn cockpit fuel selector OFF.
4. Turn BMPS OFF.

This physically halts the flow of fuel to the engine fuel pump. It also disables the airframe electric boost pump (which is upstream of the fuel selector valve) in case a FADEC fault or pilot action.

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CHAPTER 77

ENGINE INDICATING

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Table of Contents

SECTION 77-00 GENERAL 5
SECTION 77-40 SERVICING THE VM1000FX 7
SECTION 40-01 VM1000FX SYSTEM PROCEDURES 7
DPU REMOVAL 8
DPU INSTALLATION 10
DISPLAY PANEL REMOVAL 12
DISPLAY PANEL INSTALLATION 14
HEALTH STATUS ANNUNCIATOR REMOVAL 15
HEALTH STATUS ANNUNCIATOR INSTALLATION 17
PRIMARY BATTERY CURRENT SENSOR REMOVAL 19
PRIMARY BATTERY CURRENT SENSOR INSTALLATION 22
OIL PRESSURE SENSOR REMOVAL 25
OIL PRESSURE SENSOR INSTALLATION 27
OIL TEMPERATURE SENSOR REMOVAL 28
OIL TEMPERATURE SENSOR INSTALLATION 30
ENGINE HOURS RE-PROGRAMMING PROCEDURE FOR VM1000FX 32
SECTION 40-02 VM1000FX TROUBLESHOOTING 33

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Section 77-00 General

The airplane uses a Vision MicroSystems VM1000FX integrated engine instrument display system for all engine and systems monitoring information.

The system consists of a Display Processing Unit (DPU) mounted behind the CB (circuit breaker) panel, a display panel mounted in the instrument panel, and sensors for certain engine and systems related functions installed at several locations (see below). In addition, information about a number of engine functions is derived directly from the FADEC system. This information is transmitted electronically to the DPU, and displayed on the VM1000FX indicator.

The VM1000FX unit itself is powered from the Primary A power bus only through circuit breaker CB032 (VM1000). The data card model SBC100 mounted inside the VM1000FX DPU is powered from both Primary A and Secondary B buses through CB014 (B3/HSA) and CB027 (A3/HSA).

Backlighting of the display panel is provided via a connection from the airplane panel lights dimmer to the DPU (not to the panel itself). A small inverter in the DPU converts airplane lighting DC voltage to the AC voltage required by the panel backlight.

Table 77-1 shows what information the VM1000FX panel displays and its source:

Information displayed	Information source	Sensor location
Manifold pressure	FADEC	intake manifold (2)
RPM	FADEC	SSA (speed sensor assembly) on crankcase
Oil Pressure	oil pressure sensor	firewall
Oil Temperature	oil temperature sensor	oil cooler adaptor
Fuel Pressure	FADEC	fuel distribution block
(4) Exhaust gas temperatures	FADEC	(4) cylinder exhaust pipe temperature probes
(4) cylinder head temperatures	FADEC	(4) cylinder head temperature sensors
Electrical system voltage	“voltage monitor” circuit breaker	airplane power “A” distribution bus
Alternator output amperage	ammeter sensor	behind instrument panel
Engine percent power	FADEC	computed internally

Table 77-1 Information displayed on the VM1000FX Panel

NOTE

In the event that errors occur in the transmission of data from the FADEC to the VM1000FX, the digital RPM display will be replaced by the display “CE xx,” where “xx” is the number of errors encountered (from 00 to 99). If the display advances rapidly to “CE 99,” assume the interconnection between the FADEC and the VM1000FX is defective.

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Section 77-40 Servicing the VM1000FX

The components of the VM1000FX system (DPU, display panel, sensors, interconnecting cables) are not field-serviceable. Servicing any of these components is at the unit replacement level only.

Section 40-01 VM1000FX System Procedures

This section contains the removal and installation procedures for the Display Processing Unit, display panel, Health Status Annunciator, and the various sensors. There is also a procedure for reprogramming the VM1000FX to reflect the correct number of engine hours.

For procedures to remove and install the Health Status Annunciator indicator panel, mounted to the instrument panel, refer to Chapter 31 – *Indicating/Recording Systems*.

DPU REMOVAL

Perform this procedure to remove the DPU from the airplane.

CAUTION

This device contains electro-statically sensitive parts. Take ESD precautions before handling this or any other electronic device on the airplane.

1. Write down the current engine operating hours.
2. Pull circuit breaker BAT 1 (CB001) to OPEN.
3. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
4. Locate two post lamps installed near the upper panel edge and remove hoods. Set them aside for later installation.
5. Locate and remove seven (7) 8-32 machine screws securing the circuit breaker panel to the instrument console. Retain these screws for later installation.

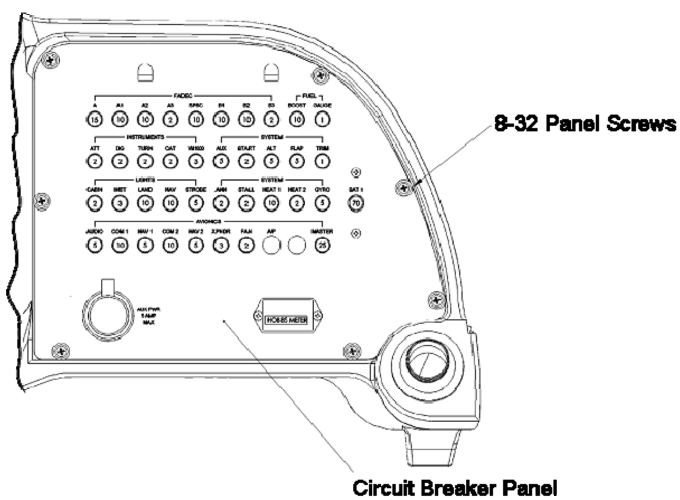

Figure 77-1 Circuit Breaker Panel

NOTE

The following step will remove the CB panel. There are a number of electrical wire harness connections to the instrument console and ground plane. Take care when removing the CB panel from the instrument console Guide wire harnesses during the removal process in order to prevent damage.

6. Prepare a protective cushion surface just below and in front of the circuit breaker panel console position.

7. Gently pull the circuit breaker panel out of the instrument console and pivot face down in front of the console on the protective cushion surface.
8. Connect an anti-static strap from you body to the metallic surface of the power distribution harness.
9. Disconnect engine data processing unit ribbon cable connector PVM03A from DPU connector PVM03.
10. Disconnect engine data processing unit cable connector PVM01 from DPU connector PVM01.
11. Disconnect engine data processing unit cable connector PVM04 from DPU connector PVM04.
12. Remove four (4) 6-32 screws holding the DPU mounting bracket to the circuit breaker panel assembly. See Figure 77-2 for the location of the DPU mounting screws.

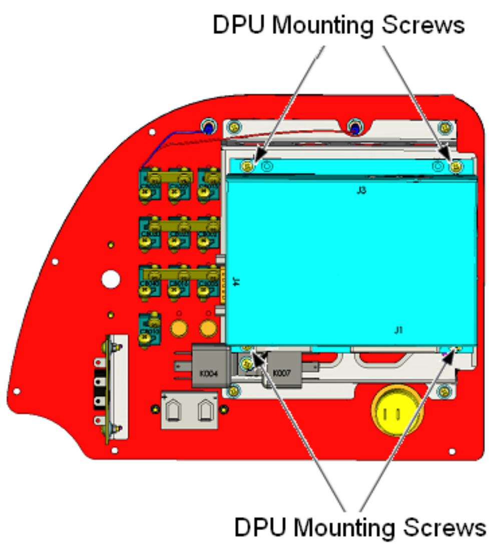

Figure 77-2 DPU Bracket Mounting Screws

13. Remove the DPU and mounting bracket as an assembly.
14. Verify software version.
15. Write down the DPU serial number.
16. If the installation of the DPU will take place later, install the CB panel in to the instrument console. Secure the instrument panel with the seven screws removed in step 5 above.

This completes the DPU Removal procedure.

DPU INSTALLATION

Perform this procedure to install the DPU into the airplane.

CAUTION

This device contains electro-statically sensitive parts. Take ESD precautions before handling this or any other electronic device on the airplane.

1. Ensure all electrical switches are off.
2. If installing a replacement DPU immediately after removing the previous DPU indicator, then proceed to step 8 below.
3. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
4. Locate two post lamps installed near the upper panel edge and remove hoods. Set them aside for later installation.
5. Locate and remove seven (7) 8-32 machine screws securing the circuit breaker panel to the instrument console. Retain these screws for later installation.

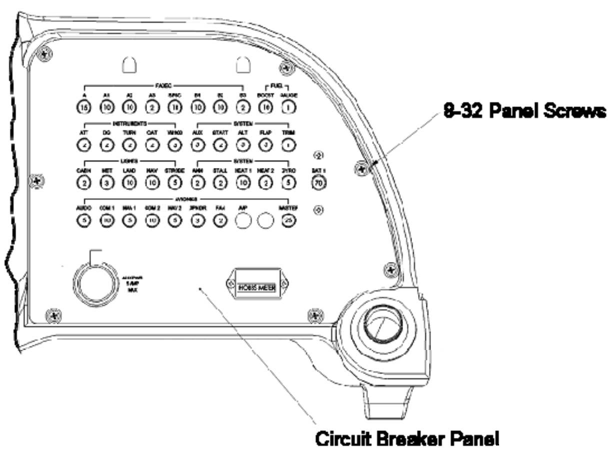

Figure 77-3 Circuit Breaker Panel

NOTE

The following step will remove the CB panel. There are a number of electrical wire harness connections to the instrument console and ground plane. Take care when removing the CB panel from the instrument console Guide wire harnesses during the removal process in order to prevent damage.

6. Prepare a protective cushion surface just below and in front of the circuit breaker panel console position.
7. Gently pull the circuit breaker panel out of the instrument console and pivot face down in front of the console on the protective cushion surface.

8. Connect an anti-static strap from you body to the metallic surface of the power distribution harness.
9. Install new DPU using the four screws removed in step 12 in the DPU Removal procedure.
10. Connect engine data processing unit's ribbon cable connector PVM03A from DPU connector PVM03.
11. Connect engine data processing unit's cable connector PVM01 from DPU connector PVM01.
12. Connect engine data processing unit's cable connector PVM04 from DPU connector PVM04.
13. Verify software version.
14. Install the CB panel in to the instrument console.
15. Secure the panel with the screws removed earlier.
16. Use the Engine Hours Re-Programming Procedure for VM1000FX on page 32, to set the value of the engine hours recorded in the DPU Removal procedure on page 8 of this chapter

This completes the DPU Installation procedure.

DISPLAY PANEL REMOVAL

Perform this procedure to remove the VM1000FX display panel.

This device contains electro-statically sensitive parts. Take ESD precautions before handling this or any other electronic device on the airplane.

1. Ensure all electrical switches are off.
2. Pull circuit breaker BAT 1 (CB001) to OPEN.
3. Place a cover over the pilot yoke control.
4. Have a large block of soft foam on your lap to place the instrument panel on.
5. Remove the top two screws and one of the bottom screws from the VM1000FX display panel.
6. Remove the ten screws securing the instrument panel to the instrument console. See Figure 77-4 for location of the screws.

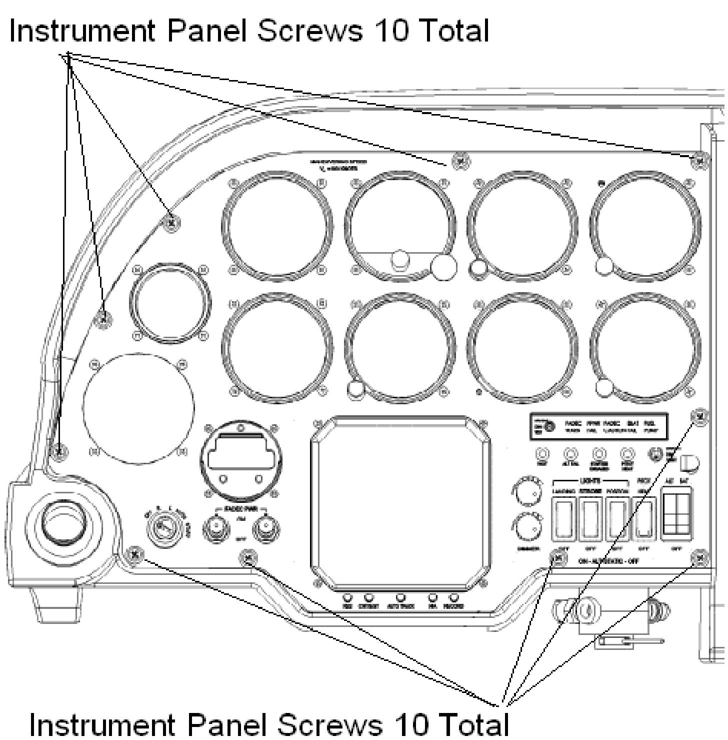

Figure 77-4 Location of the Screws Securing the Instrument Panel

7. Gentle pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
8. Do not to disturb electrical connections, pitot or static lines.

9. Connect an anti-static strap from you body to the metallic surface of the power distribution harness.
10. Disconnect ribbon connector, remove display panel.
11. Remove the remaining screw securing display panel to instrument panel.
12. Gently push the display into the instrument panel and let the display drop down through the bottom of the instrument console.
13. If the installation of the VM1000FX display panel will take place later, install the instrument panel in to the instrument console. Secure the instrument panel with the ten screws removed in step 6 above.

This completes the Display Panel Removal procedure.

DISPLAY PANEL INSTALLATION

Perform this procedure to install the VM1000FX display panel.

CAUTION

This device contains electro-statically sensitive parts. Take ESD precautions before handling this or any other electronic device on the airplane.

CAUTION

To clean face of display panel, only use a soft cloth dampened with clear water, then wipe dry. Do not use any solvents or other cleaning products to clean face of display panel.

1. Ensure all electrical switches are off.
2. If installing a replacement display panel immediately after removing an existing display panel, then proceed to step 7 below.
3. Place a cover over the pilot yoke control.
4. Have a large block of soft foam on your lap to place the instrument panel on.
5. Remove the ten screws securing the instrument panel to the instrument console. See Figure 77-4 for location of the screws.
6. Gentle pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
7. Connect an anti-static strap from you body to the metallic surface of the power distribution harness.
8. Connect ribbon cable to replacement display panel and secure.
9. Place display panel in instrument panel cutout and secure with the screws removed in the Display Panel Removal procedure.
10. Install instrument panel into the instrument console taking care not to disturb electrical connections, pitot or static lines.
11. Secure the instrument panel with the ten instrument panel screws.
12. Apply power to airplane electrical system and perform functional check of integrated engine instrument system.

This completes the Display Panel Installation procedure.

HEALTH STATUS ANNUNCIATOR REMOVAL

Perform this procedure to remove the health status annunciator, HSA, from the airplane.

CAUTION

This device contains electro-statically sensitive parts. Take ESD precautions before handling this or any other electronic device on the airplane.

1. Ensure all electrical switches are off.
2. Pull circuit breaker BAT 1 (CB001) to OPEN.
3. Place a cover over the pilot yoke control.
4. Have a large block of soft foam on your lap to place the instrument panel on.
5. Remove the ten screws securing the instrument panel to the instrument console. See Figure 77-4 for location of the screws.

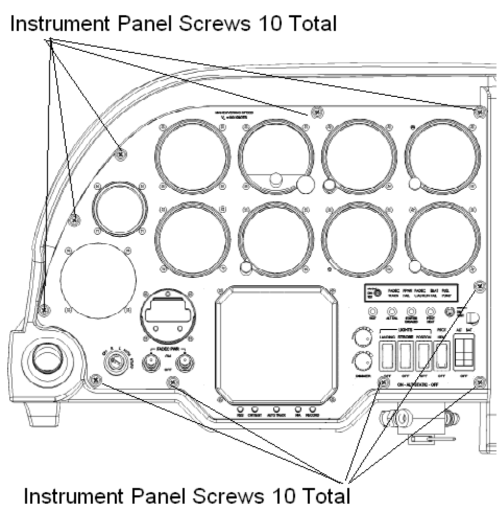

Figure 77-5 Location of the Screws Securing the Instrument Panel

6. Gentle pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
7. Do not to disturb electrical connections, pitot, or static lines.
8. Connect an anti-static strap from you body to the metallic surface of the power distribution harness.

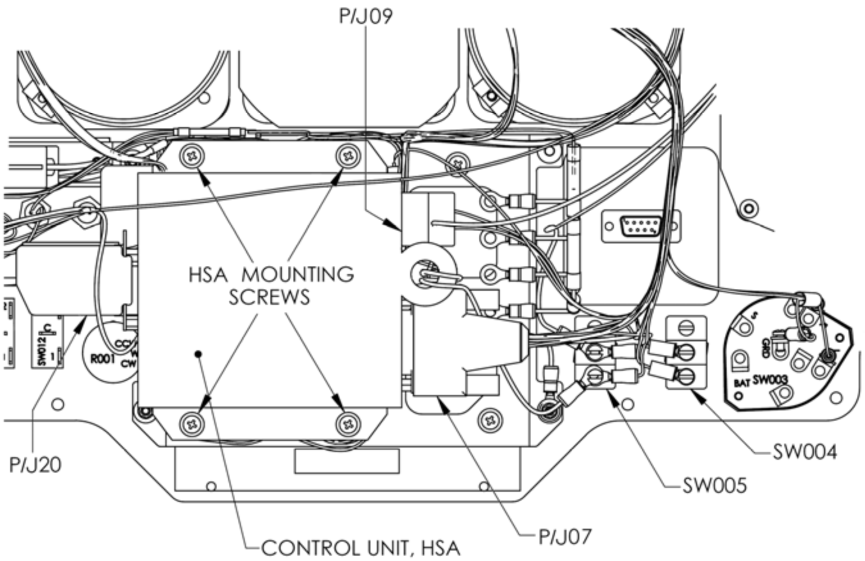

Figure 77-6 HSA Control Unit Showing the Mounting Screws, Connectors and Switch SW005

9. Disconnect P/J7, P/J9, and P/J20 from the HSA module. See Figure 77-6 for location of these connectors.
10. The cable for P/J03 goes from the instrument panel to the instrument console. Disconnect P/J03.

CAUTION

Failure to disconnect P/J03 can cause damage to the secondary battery system as the terminal connector on wire P24A10 is treaded through the current sensor on the HSA.

11. Remove the wire P24A10 from the terminal on switch SW005.
12. Carefully thread the wire back through the current sensor.
13. Remove the four screws securing the HSA to the instrument panel.
14. Secure the wire P24A10 to the same terminal on switch SW005.
15. If the installation of the HSA will take place later, install the instrument panel in to the instrument console. Secure the instrument panel with the ten screws removed in step 5 above.

This completes the Display Panel Removal procedure.

HEALTH STATUS ANNUNCIATOR INSTALLATION

Perform this procedure to install the health status annunciator, HSA.

CAUTION

This device contains electro-statically sensitive parts. Take ESD precautions before handling this or any other electronic device on the airplane.

1. Ensure all electrical switches are off.
2. If installing a replacement the HSA immediately after removing an existing HSA, then proceed to step 7 below.
3. Place a cover over the pilot yoke control.
4. Have a large block of soft foam on your lap to place the instrument panel on.
5. Remove the ten screws securing the instrument panel to the instrument console. See Figure 77-4 for location of the screws.
6. Gentle pull the instrument panel towards you, placing it face down on to the block of soft foam rubber.
7. Connect an anti-static strap from you body to the metallic surface of the power distribution harness.
8. Disconnect P/J03.
9. Disconnect wire P24A10 from switch SW005.
10. Loop the wire through the current sensor as shown in Figure 77-7. From P03 route the wire P24A10 upward through the current senor ring, making two loops through the sensor, wire enters at the bottom. The ring terminal end will exit at the top of the sensor.

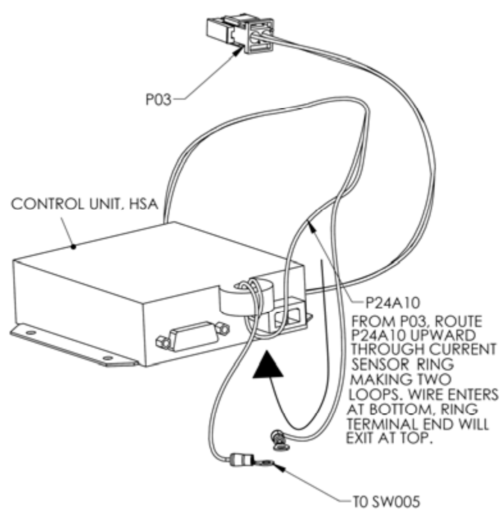

Figure 77-7 Showing the Looping of the Wire Through the Current Sensor

11. Secure the HSA to the bracket using the four screws from step 13 of the Health Status Annunciator Removal procedure.
12. Connect P/J7, P/J9, and P/J20 from the HSA module. See Figure 77-6 for location of these connectors.
13. Connect P/J03.
14. Install instrument panel into the instrument console taking care not to disturb electrical connections, pitot, or static lines.
15. Secure the instrument panel with the ten instrument panel screws.
16. Apply power to airplane electrical system and perform functional check of integrated engine instrument system.

This completes the Health Status Annunciator Installation procedure.

PRIMARY BATTERY CURRENT SENSOR REMOVAL

Perform this procedure to remove the primary battery current sensor.

CAUTION

This device contains electro-statically sensitive parts. Take ESD precautions before handling this or any other electronic device on the airplane.

1. Pull circuit breaker BAT 1 (CB001) to OPEN.
2. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
3. Locate two post lamps installed near the upper panel edge and remove hoods. Set them aside for later installation.
4. Locate and remove seven (7) 8-32 machine screws securing the circuit breaker panel to the instrument console. Retain these screws for later installation.

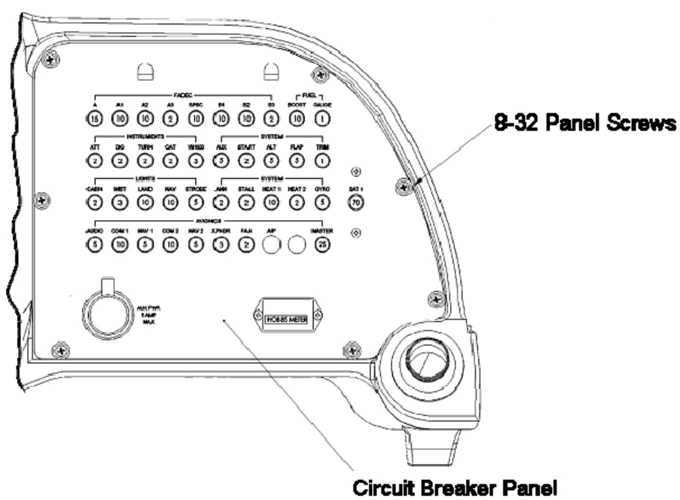

Figure 77-8 Circuit Breaker Panel

NOTE

The following step will remove the CB panel. There are a number of electrical wire harness connections to the instrument console and ground plane. Take care when removing the CB panel from the instrument console Guide wire harnesses during the removal process in order to prevent damage.

5. Prepare a protective cushion surface just below and in front of the circuit breaker panel console position.
6. Gently pull the circuit breaker panel out of the instrument console and pivot face down in front of the console on the protective cushion surface.

7. Connect an anti-static strap from you body to the metallic surface of the power distribution harness.
8. Disconnect wire P03B6 from terminal 2 of TB01 on the power distribution harness. See Figure 77-9 for location of the terminals on TB01.

CAUTION

Use caution when disconnecting wire P03A6 from terminal 2 of TB01. There can be primary battery voltage on the terminal.

9. Disconnect wire P10A6 from terminal 3 of TB01 on the power distribution harness.

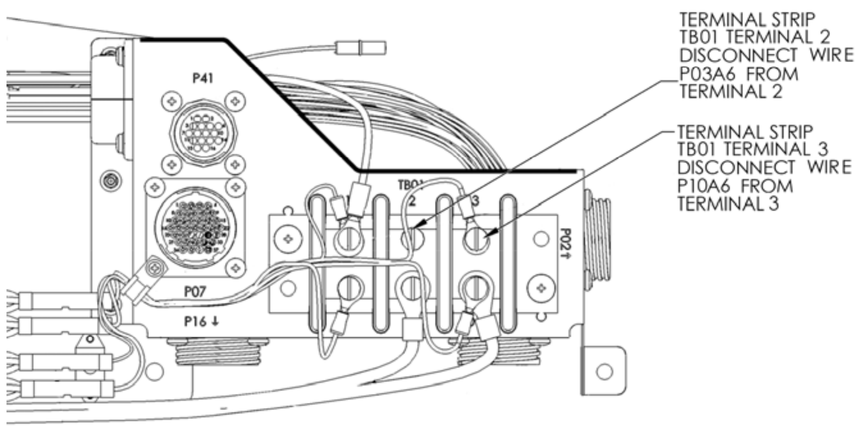

Figure 77-9 Right Side the Power Distribution Assembly Showing Location for the Connection for Wire P10A6

10. Cut the terminal end from wire P10A6.
11. Thread wire P10A6 back through the current sensor. See Figure 77-10 for location of the primary power current sensor.

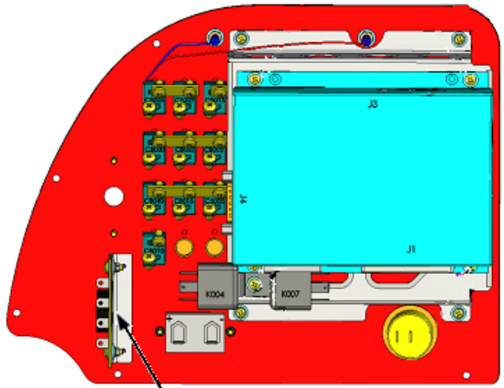

Figure 77-10 Location of the Primary Power Current Sensor

12. Disconnect the four wires from the primary power current sensor board.
13. Remove the two screws that secure the current sensor board to the mounting bracket.
14. Remove the board.
15. If the installation of the current sensor will take place later, secure wire P10A6 such that it will not come in contact with other circuits, save current sensors board screws, and install the instrument panel in to the instrument console. Secure the instrument panel with the ten screws removed in step 4 above.

This completes the Primary Battery Current Sensor Removal procedure.

PRIMARY BATTERY CURRENT SENSOR INSTALLATION

Perform this procedure to install the primary battery current sensor into the airplane.

CAUTION

This device contains electro-statically sensitive parts. Take ESD precautions before handling this or any other electronic device on the airplane.

1. Ensure all electrical switches are off.
2. If installing a replacement current sensor board immediately after removing the previous current sensor board indicator, then proceed to step 8 below.
3. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
4. Locate two post lamps installed near the upper panel edge and remove hoods. Set them aside for later installation.
5. Locate and remove seven (7) 8-32 machine screws securing the circuit breaker panel to the instrument console. Retain these screws for later installation.

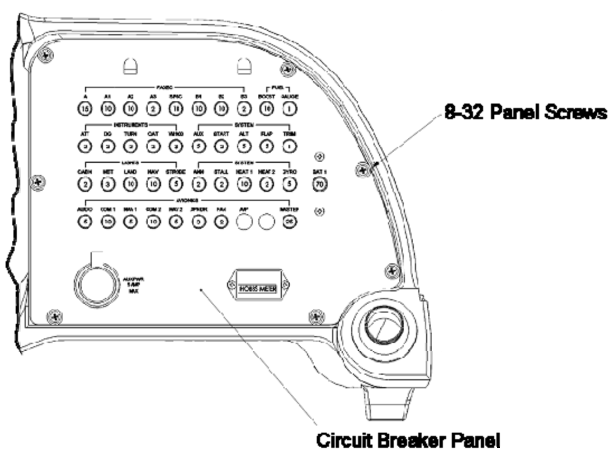

Figure 77-11 Circuit Breaker Panel

NOTE

The following step will remove the CB panel. There are a number of electrical wire harness connections to the instrument console and ground plane. Take care when removing the CB panel from the instrument console Guide wire harnesses during the removal process in order to prevent damage.

6. Prepare a protective cushion surface just below and in front of the circuit breaker panel console position.

7. Gently pull the circuit breaker panel out of the instrument console and pivot face down in front of the console on the protective cushion surface.
8. Connect an anti-static strap from you body to the metallic surface of the power distribution harness.
9. Mount the replacement current sensor board to the mounting bracket use the screws removed in step 13 in the Primary Battery Current Sensor Removal procedure on page 21.
10. Connect cable V13A20 for the current sensor board. See Table 77-2 for the correct wire connections.

Wire Color from Cable V13A20	Primary Battery Current Sensor Board Connector
White (no tracer)	WHT
White/Blue	BLK
White/Orange	RED
White/Green	GRN

Table 77-2 Cable Connections for the Primary Battery Current Sensor Board

11. Carefully thread wire P10A6 through the current sensor.

NOTE

Thread the wire through the current sensor board BEFORE mounting a terminal to the end of the wire.

12. Mount an appropriate sized wire terminal to the end of the wire. To complete the installation of the wire terminal apply heat shrink tubing to the end of the wire.
13. Connect wire P10A6 to terminal 3 of the TB01 on the power distribution harness.

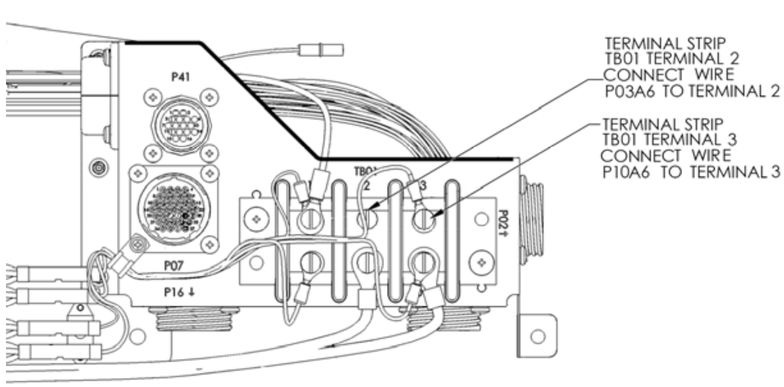

Figure 77-12 Location of TB01 Terminal 3

14. Connect wire P03A6 to terminal 2 of TB01 of the power distribution harness.

CAUTION

Use caution when connecting wire P03A6 from terminal 2 of TB01. There can be primary battery voltage on the terminal.

15. Install the CB panel into the instrument console.
16. Secure the CB panel to the instrument console using the screws removed in step 5 of this procedure.

This completes the Primary Battery Current Sensor Installation procedure.

OIL PRESSURE SENSOR REMOVAL

Perform this procedure to remove either of the oil pressure sensors. The oil pressure sensors mount to the firewall, below the level of the Electronic Control Units, ECU. There are two sensors at this location. See Figure 77-13. The sensor on top is for the Hobbs meter, the sensor on the side is for the Health Status Annunciator. As these sensors are co-located, the procedure is the same for both. Any differences are noted in the text of the procedure.

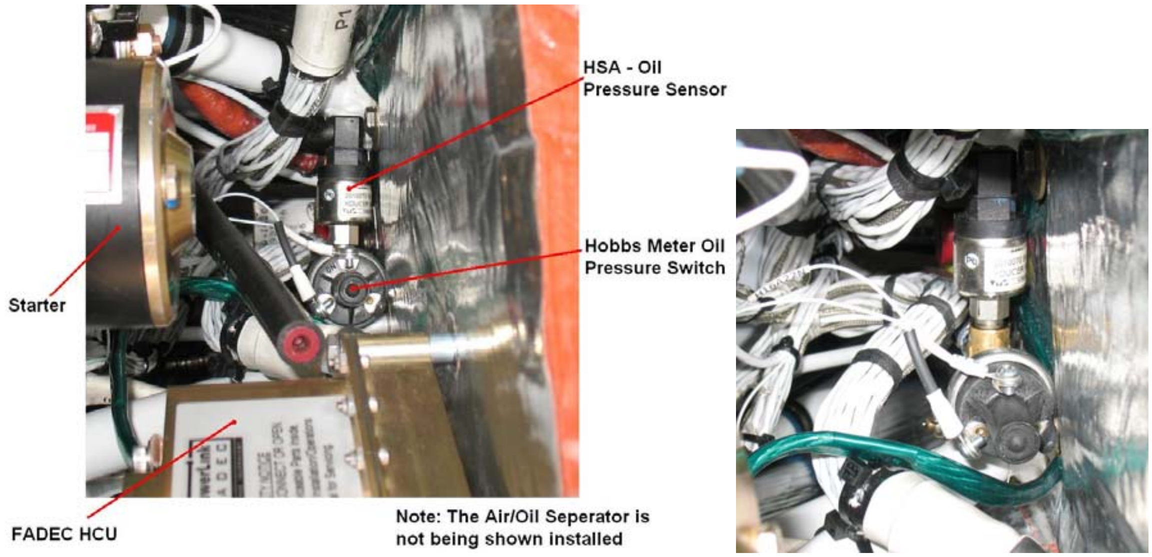

Figure 77-13 Location of the Oil Pressure Switch (Hobbs) and the Oil Pressure Sensor (HSA) (The Air/Oil Separator Not Shown)

1. Pull circuit breaker BAT 1 (CB001) to OPEN.
2. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
3. Remove the upper and lower engine cowling.
4. If the airplane is equipped with an air/oil separator mounted to the firewall between the Electronic Control Units (ECU), disconnect the air/oil separator and bracket from the Electronic Control Units.
5. Refer to the exploded view shown in Figure 77-14. Have someone hold a wrench on the nut from inside the cockpit.
6. Turn the bolt securing the mounting bracket to the firewall. Remove the bolt being careful to retain the flat washers and the sleeve spacer
7. Remove the bracket securing the oil pressure sensor T-fitting to the firewall.

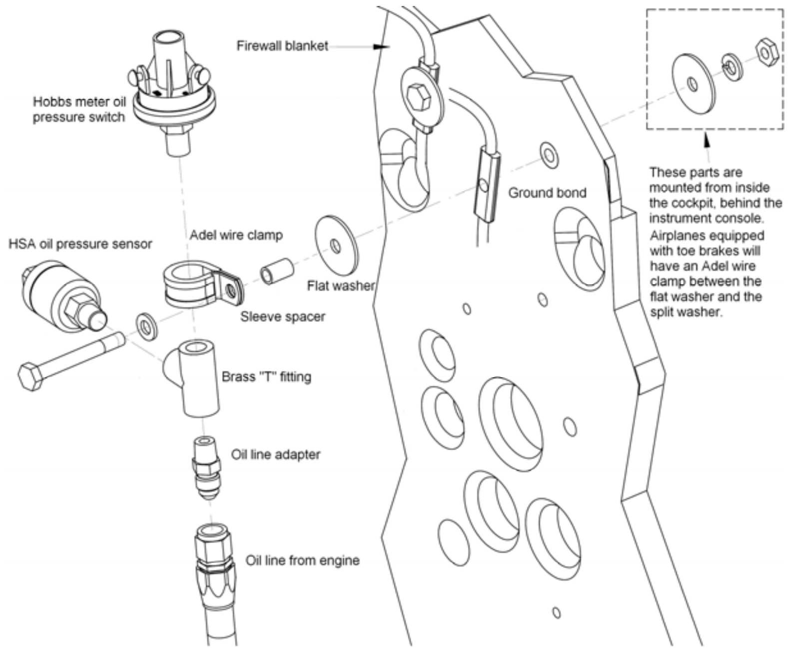

Figure 77-14 Exploded View of the Oil Pressure Mount

8. Bring the oil pressure sensor assembly up enough to allow access to the electrical connections on the sensors.
9. If removing Hobbs oil pressure switch, go to step 13.
10. Remove the screw in the center of the connector.
11. Disconnect the cable going to pressure sensor.
12. Go to Step 14.
13. Using a Phillips screwdriver, remove the two wires that connect to the top of the Hobbs oil pressure switch.
14. Wrap the T-fitting with a rag to catch any oil.
15. Use a 7/16-inch wrench to remove the oil pressure sensor.
16. If installation of a replacement sensor will happen later, cap the open end of the T-fitting, and secure.
17. If an air/oil separator was removed, temporarily install the air/oil separator and secure.

This completes the Oil Pressure Sensor Removal procedure.

OIL PRESSURE SENSOR INSTALLATION

Perform this procedure to install the oil temperature sensor.

1. If installing a replacement oil pressure sensor immediately after removing the previous oil pressure sensor, then proceed to step 9 below.
2. Pull circuit breaker BAT 1 (CB001) to OPEN.
3. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
4. Remove the upper and lower engine cowling.
5. If the airplane is equipped with an air/oil separator mounted to the firewall between the Electronic Control Units (ECU), disconnect the air/oil separator and bracket from the Electronic Control Units.
6. Refer to the exploded view shown in Figure 77-14. Have someone hold a wrench on the nut from inside the cockpit.
7. Turn the bolt securing the mounting bracket to the firewall. Remove the bolt being careful to retain the flat washers and the sleeve spacer
8. Remove the bracket securing the oil pressure sensor T-fitting to the firewall.
9. Install the oil pressure sensor to the T-fitting.
10. If installing Hobbs oil pressure switch, go to step 14.
11. Connect the cable going to pressure sensor.
12. Secure the connector with the screw in the center of the connector.
13. Go to Step 15.
14. Using a Phillips screwdriver, secure the two wires to the top of the Hobbs oil pressure switch.
15. Check the connector or connections on the other oil pressure sensor to make sure the connections are secure.
16. Using Figure 77-14 as a guide, assemble the mounting bracket and hardware.
17. Push the bolt through the firewall blanket to the inside of the cockpit.
18. Have some one hold the bolt from the engine compartment. Assemble the hardware from the cockpit side as shown in Figure 77-14.
19. Tighten the nut to secure the hardware.
20. Remove any rags that were absorbing oil.
21. Clean oil spilled during this procedure.

This completes the Oil Pressure Sensor Installation procedure.

OIL TEMPERATURE SENSOR REMOVAL

Perform this procedure to remove the oil temperature sensor. The oil temperature sensor is located on the port side of the engine up next to the firewall. See Figure 77-15 for the location of the oil temperature sensor. The location makes it very difficult to remove and/or install. Before attempting to remove/install the sensor, check all of the wiring and connections. Make sure the fault is not in the wiring or connectors.

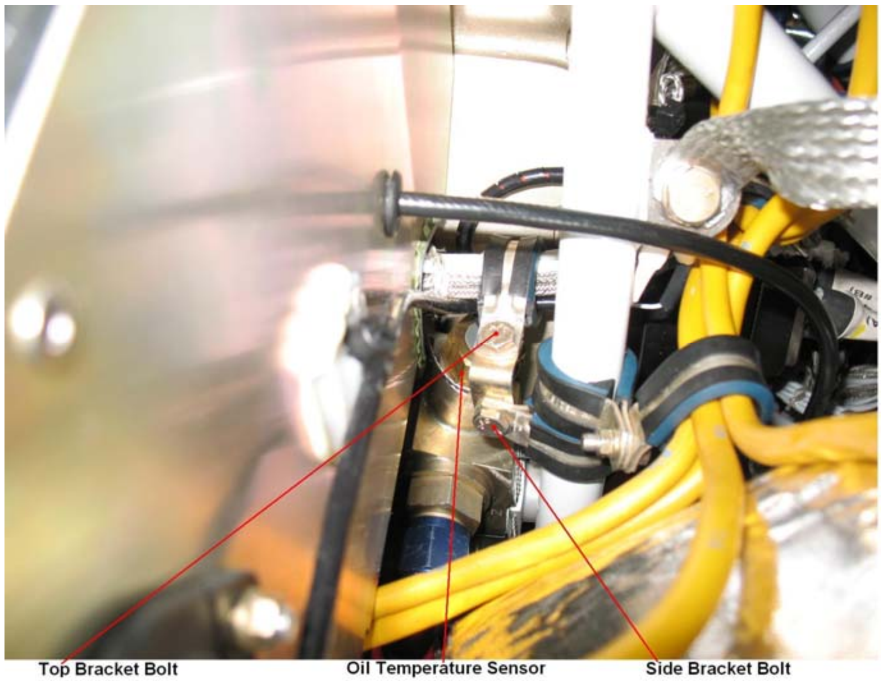

Figure 77-15 Location of the Oil Temperature Sensor

1. Pull circuit breaker BAT 1 (CB001) to OPEN.
2. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
3. Remove the upper and lower engine cowling.
4. Remove the top bracket 3/8-inch bolt and ¼-inch nut that secures the wire clamp to the L-bracket. Carefully remove the cable clamp from the wire bundle.
5. Loosen the side bracket bolt and nut that secures the wire clamp to the engine support tube.

NOTE

Spraying the cable clamp with a small amount of alcohol will allow the cable clamp to slide easier on the support tubing.

6. Slide the wire clamp down the tube and out of the way.

7. Cut and remove the safety wire from the sensor.
8. Carefully remove any cable ties securing the bundle of wires above the sensor.
9. Disconnect the P/J61.
10. Position the wire bundle away from the top of the oil temperature sensor.
11. Pack the area of the around the sensor to catch any oil that may come out of the fitting.
12. Use a 7/8-inch open-ended wrench to remove the oil temperature sensor.

NOTE

With the two Adel wire clamps out of the way, there is just enough room to get the wrench on to the sensor and turn it just enough to break the threads free. Then bring the wrench straight down from the top and complete the removal of the sensor. See Figure 77-16 to see the correct positioning to complete the removal of the oil temperature sensor.

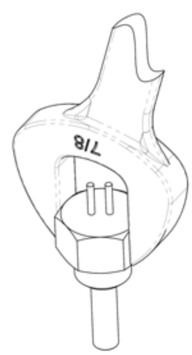

Figure 77-16 Positioning the Wrench to Remove the Oil Temperature Sensor

13. With the oil temperature sensor removed, remove the remainder of the safety wire from the sensor-mounting bracket. Install new safety wire in to the bracket to prepare for the installation of a replacement sensor.

NOTE

Liberty Aerospace, Inc. recommends to start the installation of the safety wire at this point, while the sensor is out of the engine.

14. If replacing the oil temperature sensor later, plug the open hole where the sensor goes, secure all wires and connectors, secure mounting hardware.

This completes the Oil Temperature Sensor Removal procedure.

OIL TEMPERATURE SENSOR INSTALLATION

Perform this procedure to install the oil temperature sensor.

1. Pull circuit breaker BAT 1 (CB001) to OPEN.
2. If installing replacement oil temperature sensor immediately after removing the previous oil temperature sensor, then proceeds to step 6 below.
3. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
4. If installed, remove the upper and lower engine cowling.
5. Reposition the cable mounting hardware associated with the cable above the oil temperature sensor.
6. Reposition the bundle of cables above the sensor.
7. Wrap the area with a rag to absorb and oil.
8. Remove any plug from the hole for the sensor.
9. If a new safety wire was not installed previously, insert a fresh safety wire and prepare it for the sensor.
10. Assemble the oil temperature sensor’s copper crush gasket on to the sensor.

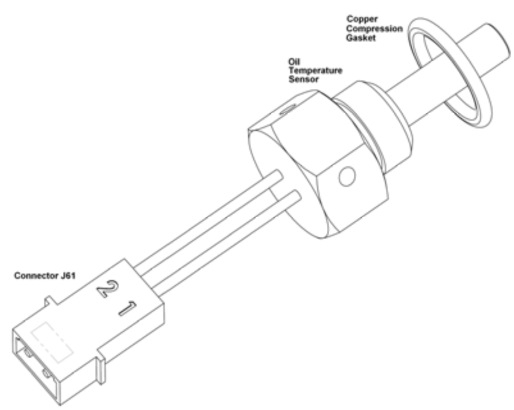

Figure 77-17 Oil Temperature Sensor Showing the Copper Compression Gasket

11. Insert oil temperature sensor assembly into the hole for the oil temperature sensor. Use a 7/8-inch wrench as shown in Figure 77-16 to turn-in the oil temperature sensor.
12. Torque the sensor to 220 in-lbs.
13. Complete the installation of the safety wire.
14. Connect P61 to J61.
15. Secure the cable bundle using wire ties.
16. Move the cable clamp back into position and tighten the side bracket bolt.

17. Assemble the wire bundle cable clamp to the L-bracket. Secure the clamp to the L-bracket using the bolt and nut removed in step 4 in the Oil Temperature Sensor Removal procedure on page 28 of this chapter.
18. Remove any rags that were absorbing oil.
19. Clean oil spilled during this procedure.

This completes the Oil Temperature Sensor Installation procedure.

ENGINE HOURS RE-PROGRAMMING PROCEDURE FOR VM1000FX

To re-load engine hours, follow the steps below:

1. "Accessing Set Mode": with power off, while pressing and holding buttons 1, 3, and 5, turn on power. Wait until displays start reading out values (approximately 15 seconds) and release buttons.
2. "Selecting Decade to Change: press button 4 once to move to next higher decade on digital engine hours display area. No visual indication is displayed until STEP 3. Pressing button moves the selection left increases decade. Upon reaching highest decade, pressing button again returns to lowest decade.
3. "Changing the Decade Value": press button 5 to increase decade value. Increasing display above 9 increases to next decade. To erase engine hours value, select highest decade press button 5 continually until display clears. Do this TWICE, then repeat steps 1 thru 3.
4. When the display indicates the correct number of engine hours, turn VM1000FX power OFF.
5. Turn power ON and confirm current Tachometer Time has been set. Turn power OFF.
6. Check DPU functionality by normal engine starting and shutdown procedures.
7. When no defects noted, confirm compliance in logbook entry.

This completes the Engine Hours Re-Programming Procedure for VM1000FX procedure.

Section 40-02 *VM1000FX Troubleshooting*

Table 77-3 Integrated Engine Instrument Troubleshooting

Complaint	Possible cause	Remedy
VM1000FX completely inoperative (panel remains blank)	Defective power circuit breakers (2)	Replace
	Defective power wiring	Replace
	Defective DPU	Replace DPU
	Defective display panel	Replace display panel
	Defective ribbon cable from DPU to display	Replace ribbon cable
VM1000FX backlighting inoperative	Defective wiring from panel lights dimmer to DPU	Repair
	Defective inverter in DPU	Replace display
	Defective EL backlight panel in display	Replace DPU
Erratic or absent oil pressure indication	Defective oil pressure sender	Replace sender
	Defective oil pressure sender wiring	Repair
Erratic or absent oil temperature indication	Defective oil temperature sender	Replace sender
	Defective oil temperature sender wiring	Repair
Erratic or absent amperage indication	Defective amp sensor	Replace sensor
	Defective amp sensor wiring	Repair
Erratic or absent voltage indication	Defective VOLT MON circuit breaker	Replace circuit breaker
	Defective wiring	Repair wiring

Complaint	Possible cause	Remedy
Display of momentary, erratic values, then automatically returning to normal operation, OR Blank display followed by automatic restart to normal operation mode or automatic restart to self-test diagnostic cycle first, and then the display automatically returns to normal operation, OR Display demo-mode, then returning to normal operation after CB cycle.	Defective software	Retrofit by removing DPU 4010083 with Software 3036003, Revision 1.010, or earlier, and replacing with Software 3036003, Revision 1.011, or later.
Any of following indications absent or erratic: RPM, Manifold Pressure, EGT, CHT, Fuel Flow (confirmed by indication of FADEC Health Status Annunciator or erratic engine operation)	FADEC or FADEC sensor fault	See Chapter 72 in this manual and/ or Chapter 12 in TCM Engine Maintenance Manual (TCM P/N M-22)
Any of following indications absent or erratic: RPM, Manifold Pressure, EGT, CHT, Fuel Flow (FADEC Health Status Annunciator normal)	Interconnection fault	Replace interconnection cable between FADEC and connector VM-04 on DPU.
	DPU fault	Replace DPU

CHAPTER 78

ENGINE EXHAUST

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Table of Contents

SECTION 78-00 GENERAL 5

SECTION 78-10 EXHAUST SYSTEM COMPONENTS 7

SECTION 10-01 EXHAUST SYSTEM PROCEDURES 8
COLLECTOR/MUFFLER REMOVAL 9
COLLECTOR/MUFFLER INSTALLATION 10
EXHAUST PIPE REMOVAL 11
EXHAUST PIPE INSTALLATION 12
HEAT EXCHANGER SHROUD REMOVAL 13
HEAT EXCHANGER SHROUD INSPECTION AND INSTALLATION 14
SECTION 10-02 EXHAUST SYSTEM TROUBLESHOOTING GUIDE 15

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Section 78-00 General

The exhaust system installed on the airplane's TCM IOF-240B engine includes individual exhaust pipes from each cylinder and a single muffler located below the engine. A single overhead discharge pipe extends through the right side of the lower cowling.

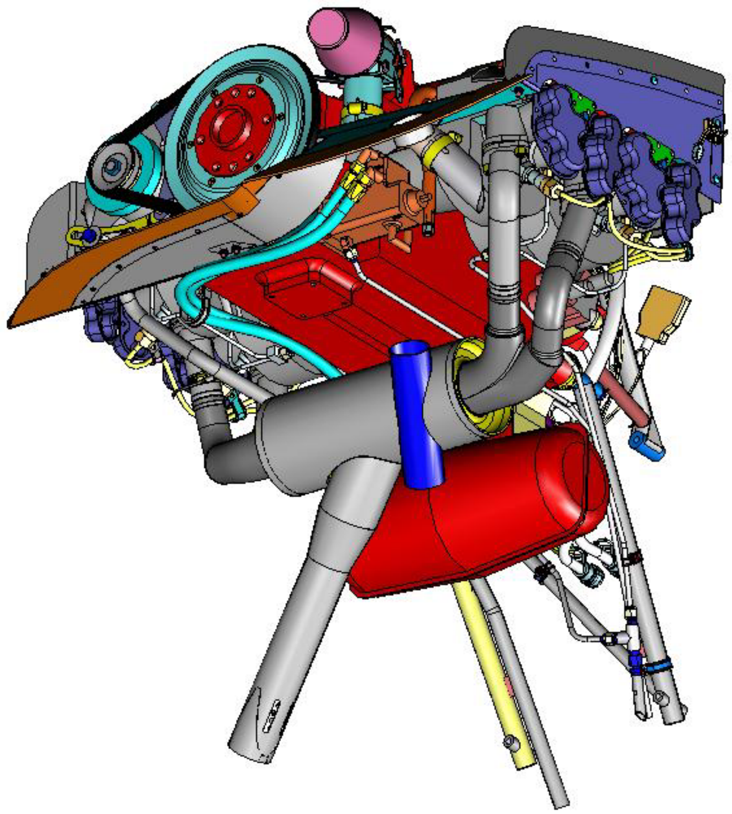

Figure 78-1 Engine Exhaust System

Two versions of the exhaust system exist: a normal exhaust system and a quiet exhaust system. All procedures are identical for both configurations. The Illustrated Parts Catalog shows the part numbers for each configuration.

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Section 78-10 Exhaust System Components

Individual exhaust pipes are secured to the exhaust port by brass exhaust nuts screwed onto the cylinder exhaust studs.

CAUTION

Exhaust nuts are intended for one-time use only. Re-use of exhaust nuts could result in inflight loosening of exhaust connections and possible inflight fire.

An EGT (Exhaust Gas Temperature) sensor for the FADEC system is installed on each exhaust pipe approximately two inches below the cylinder exhaust port. The sensors are secured by band clamps.

The two exhaust pipes from each side of the engine are connected to the left and right ends of the muffler, which is cylindrical in shape and installed below the engine with its axis running perpendicular to the engine. Slip joints in the exhaust pipe accommodate the dimensional changes that result from temperature changes in the exhaust system. Clamps secure the exhaust pipes to the muffler.

A single discharge pipe extends downward and to the (airplane's) right from the muffler for overboard discharge of exhaust gases.

The muffler is fitted with an internal heat exchanger that is used to heat outside air and provide heated air to the cabin heater box. The muffler heat exchanger is connected to the cabin heat box and the forward engine compartment baffle by heat resistant hoses and cable clamps.

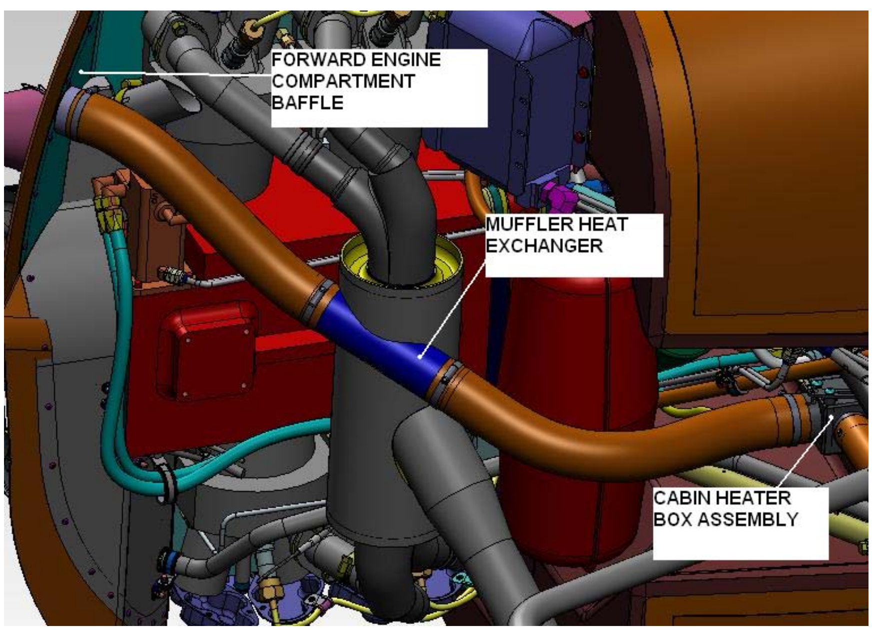

Figure 78-2 Muffler Heat Exchanger to Cabin Heater Box

Section 10-01 Exhaust System Procedures

This section contains the procedures for the exhaust system. These procedures include the collector (muffler) removal and installation, exhaust pipe removal and installation, and the heat exchanger shroud removal, inspection and installation.

COLLECTOR/MUFFLER REMOVAL

Perform this procedure to remove the collector or muffle.

1. Ensure all electrical switches are off.
2. Remove upper cowling.
3. At the port side of lower cowling, disconnect landing light; remove lower cowling.
4. Disconnect brackets securing muffler to airframe.
5. Disconnect clamp securing heat exchanger to cabin heater box hose.
6. Disconnect clamp securing heat exchanger to forward engine compartment baffle.
7. Disconnect (4) bead clamps securing cylinder exhaust pipes to muffler. Remove the muffler.

COLLECTOR/MUFFLER INSTALLATION

Perform this procedure to install the collector/muffler.

WARNING

BEFORE INSTALLING A PREVIOUOSLY USED COLLECTOR/MUFFLER, REMOVE THE HEAT EXCHANGER SHROUD AND INSPECT THE ENTIRE SURFACE OF THE COLLECTOR/MUFFLER FOR CRACKS. USE A MIRROR TO INSPECT ALL SURFACES ON THE COLLECTOR/MUFFLER. IF THERE ARE ANY CRACKS FOUND IN THE COLLECTOR/MUFFLER, EITHER REPLACE THE COLLECTOR/MUFFLER OR FIX THE CRACKS BY WELDING. IF REPAIRING THE CRACKS BY WELDING, THE WELDS MUST COMPLY WITH MIL-STD-1595A USING AMS 5680 FILLER.

1. Ensure all electrical switches are off.
2. Using new AN3-4A bolts and MS20365-1032C nuts, position muffler tubes in line with cylinder exhaust pipes and secure with (4) bead clamps. Torque hardware to 20-25 in-lbs.
3. Connect clamp securing heat exchange to forward engine compartment baffle.
4. Connect clamp securing heat exchanger to cabin heater box hose.
5. Connect brackets securing muffler to airframe.
6. Install lower and upper engine cowlings.

EXHAUST PIPE REMOVAL

1. Remove upper and lower cowling.

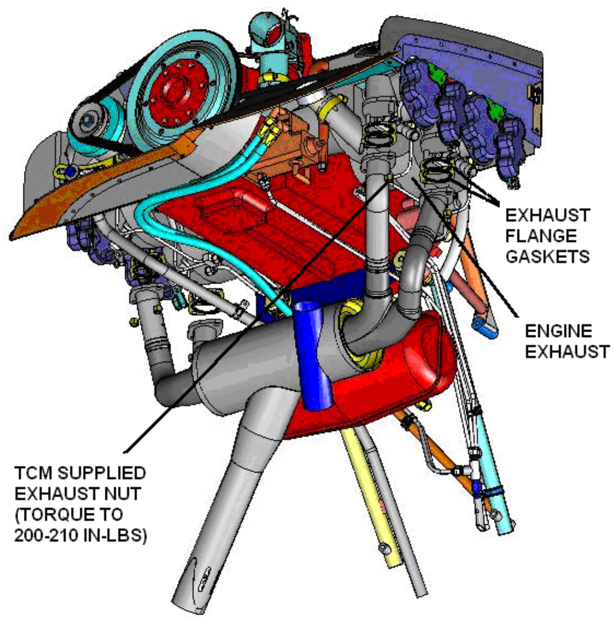

Figure 78-3 Engine exhaust system

2. Disconnect clamp at junction of exhaust pipe and muffler.
3. Cut wire ties as required to provide sufficient slack to remove EGT probe from exhaust pipe.
4. Loosen and remove B-nut securing EGT probe to compression fitting on exhaust pipe.
5. Carefully remove EGT probe from compression fitting, taking care not to kink or crimp EGT probe wire. Secure EGT probe to engine or structure to prevent damage to probe or harness.
6. Remove and discard brass exhaust nuts and lock washers from exhaust port stubs.
7. Remove exhaust pipe.
8. Remove and discard exhaust gasket.

NOTE

Do not remove EGT probe compression fitting from threaded boss on exhaust pipe unless necessary. If compression fitting must be removed, it should be discarded and replaced with a new one.

EXHAUST PIPE INSTALLATION

1. Install new exhaust gasket on cylinder exhaust port.
2. Place exhaust pipe in position. Align lower end with muffler inlet and loosely install clamp.
3. Apply high temperature anti-seize compound to threads of cylinder studs.
4. Install NEW brass exhaust nuts and lock washers. Torque to 200-210 +/- 5 in/lbs.
5. Using new AN3-4A bolts and MS20365-1032C nuts, position muffler tube in line with cylinder exhaust pipe and secure with bead clamp. Torque clamp hardware to 20-25 in-lbs.
6. Apply small amount of high temperature anti-seize compound to second and third threads of EGT probe compression fitting.
7. Carefully insert EGT probe into compression fitting. Thread on B-nut and torque to 70 +/- 5 in/lbs.
8. Use wire ties to secure EGT probe harness away from hot areas.
9. Adjust alternate air supply header to exhaust pipe (forward, port) to 3/8" clearance.

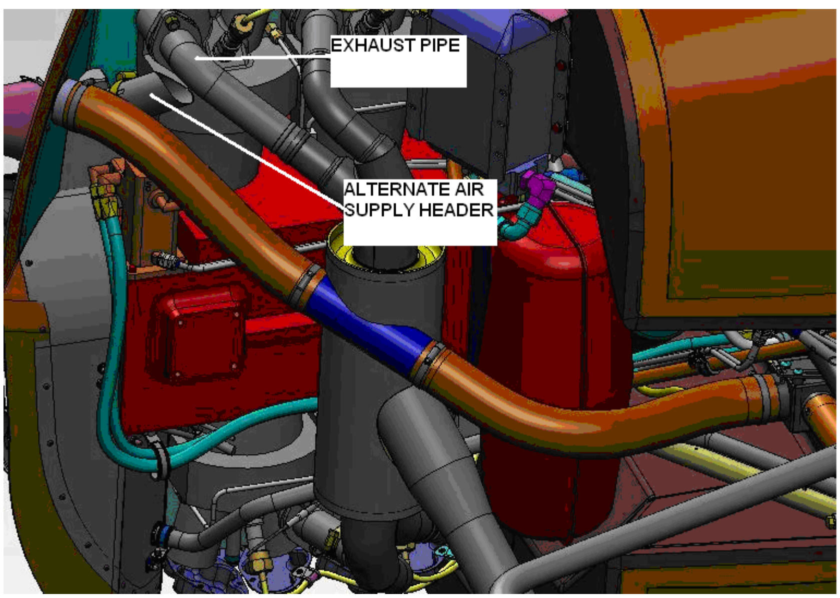

Figure 78-4 Exhaust Pipe to Alternate Air Supply Header

HEAT EXCHANGER SHROUD REMOVAL

Perform this procedure to remove the heat exchanger shroud. Refer to Figure 78-5 during this procedure.

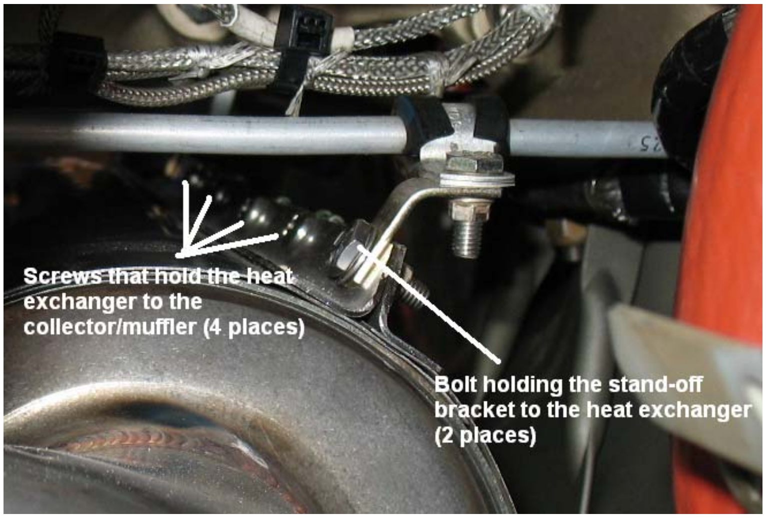

Figure 78-5 Heat Exchanger Hardware

1. Ensure all electrical switches are off.
2. Remove upper cowling.
3. At the port side of lower cowling, disconnect landing light; remove lower cowling.
4. Disconnect clamp that secures the heat exchanger to cabin heater box hose.
5. Disconnect clamp that secures the heat exchanger to forward engine compartment baffle.
6. Remove the two bolts that secure the stand-off brackets to the collector/muffler.
7. Remove the four screws that hold the heat exchanger on to the collector/muffler.
8. Slowly work the heat exchanger cover down the exhaust pipe to remove the cover from the collector/muffler.

HEAT EXCHANGER SHROUD INSPECTION AND INSTALLATION

Perform this procedure to install the heat exchanger shroud.

WARNING

BEFORE INSTALLING THE HEAT EXCHANGER SHROUD, INSPECT THE ENTIRE SURFACE OF THE COLLECTOR/MUFFLER FOR CRACKS. USE A MIRROR TO INSPECT ALL SURFACES ON THE COLLECTOR/MUFFLER. IF THERE ARE ANY CRACKS FOUND IN THE COLLECTOR/MUFFLER, EITHER REPLACE THE COLLECTOR/MUFFLER OR FIX THE CRACKS BY WELDING. IF REPAIRING THE CRACKS BY WELDING, THE WELDS MUST COMPLY WITH MIL-STD-1595A USING AMS 5680 FILLER.

1. Inspect the collector/muffler for cracks. Use a mirror to inspect all surfaces on the collector/muffler.
2. Slide the heat exchanger shroud up the exhaust pipe and attach it to the collector/muffler.
3. Install the four screws removed in step 7 of the Heat Exchanger Shroud Removal procedure on page 13 of this chapter.
4. Install the two bolts removed in step 6 of the Heat Exchanger Shroud Removal procedure on page 13 of this chapter.
5. Install the air hose and hose clamp removed in step 5 of the Heat Exchanger Shroud Removal procedure on page 13 of this chapter.
6. Install the air hose and hose clamp removed in step 4 of the Heat Exchanger Shroud Removal procedure on page 13 of this chapter.
7. Install the lower and upper cowling

Section 10-02 Exhaust System Troubleshooting Guide

This section details information on inspecting and troubleshooting the exhaust system. When inspecting the exhaust system, specifically the collector/muffler, remove the heat exchanger shroud to inspect for cracks in the exhaust system that is beneath the shroud. Table 78-1 has a troubleshooting chart to aid in troubleshooting issues with the exhaust system.

WARNING

IT IS REQUIRED TO REMOVE THE HEAT EXCHANGER SHROUD TO INSPECT THE COLLECTOR/MUFFLER FOR CRACKS. USE A MIRROR TO INSPAECT ALL SURFACES ON THE COLLECTOR/MUFFLER. IF THERE ARE ANY CRACKS FOUND IN THE COLLECTOR/MUFFLER, EITHER REPLACE THE COLLECTOR/MUFFLER OR FIX THE CRACKS BY WELDING. IF REPAIRING THE CRACKS BY WELDING, THE WELDS MUST COMPLY WITH MIL-STD-1595A USING AMS 5680 FILLER.

Complaint	Possible Cause	Remedy
Excessive engine noise	defective muffler	replace
Cracks in exhaust pipes or muffler[2] Use a mirror to inspect all surfaces on the collector/muffler. See Warning concerning the inspection of the exhaust system	thermal stress	Weld[1] or replace
Exhaust gas or odor in cockpit	cracks in muffler	Weld[1] or replace
	holes in SCAT ducts under cowling	replace
	leaks to muffler heat exchanger	replace muffler and heat exchanger
Traces of exhaust gas (gray stains) on cylinder	defective exhaust gasket	replace
	warped exhaust pipe flange	replace pipe

[1] All welds must comply with MIL-STD-1595A using AMS 5680 filler.
[2] It is required to remove the heat exchanger shroud to inspect the collector/muffler for cracks.

Table 78-1 Exhaust System Troubleshooting

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CHAPTER 79

ENGINE OIL

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Table of Contents

SECTION 79-00	GENERAL	5
SECTION 79-10	STORAGE	7
SECTION 79-20	OIL SYSTEM DISTRIBUTION	9
SECTION 20-01	OIL COOLER	9
SECTION 20-02	AIR/OIL SEPARATOR	9
SECTION 20-03	OIL SYSTEM PROCEDURES	9
	OIL COOLER OR OIL COOLER HOSE REMOVAL	10
	OIL COOLER OR OIL COOLER HOSE INSTALLATION	11
	AIR/OIL SEPARATOR REMOVAL	12
	AIR/OIL SEPARATOR INSTALLATION	13
	OIL FILTER ELEMENT REMOVAL	14
	OIL FILTER ELEMENT INSTALLATION	15
	ENGINE OIL DRAINING PROCEDURE	16
SECTION 20-04	TROUBLESHOOTING GUIDE	17
SECTION 79-30	INDICATING	19
SECTION 30-01	OIL INDICATING PROCEDURES	19
	OIL PRESSURE SENSOR REMOVAL	20
	OIL PRESSURE SENSOR INSTALLATION	22
	OIL TEMPERATURE SENSOR REMOVAL	23
	OIL TEMPERATURE SENSOR INSTALLATION	25
SECTION 30-02	TROUBLESHOOTING GUIDE	27

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Section 79-00 General

This chapter describes only those components of the engine oil system that are external to the engine. For internal components, refer to the “lubrication” subchapter of the engine chapter. For more detailed information about internal components consult the Teledyne Continental Motors IOF-240-B Maintenance Manual, TCM p/n: M-22.

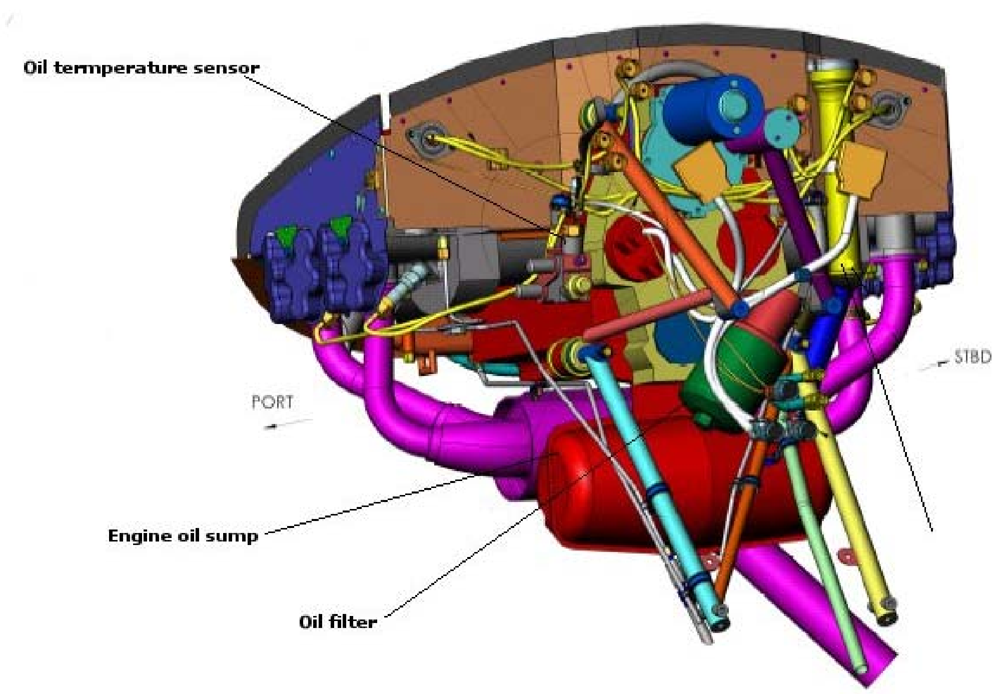

Figure 79-1 View Of Engine Oil Components, from Aft Looking Forward

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Section 79-10 Storage

The IOF-240-B's oil supply is stored in a welded steel oil sump secured to the bottom of the crankcase. Capacity of the sump is 6.0 US quarts.

An oil filler tube is secured to the (airplane's) right side of the sump and extends upward. Its top end is accessible through a small door inset into the engine upper cowling. The removable oil filler tube cap incorporates a dipstick for preflight determination of oil quantity (See Figure 79-1).

A fitting at the rear of the oil sump accepts a hose connection allowing oil from the engine crankcase breather hose to return to the sump by gravity. No other external oil lines are connected to the sump: the engine oil pump suction tube extends through the center of the opening where the sump is bolted to the crankcase, while return oil drains from the engine into the sump around the periphery of the same opening.

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Section 79-20 Oil System Distribution

Components of the oil system external to the engine include a full flow oil filter and oil cooler.

The full flow oil filter element is attached directly to the oil filter adapter at the lower right rear of the engine accessory gear-case. Oil flows to and from the filter element via internal passages in the adapter.

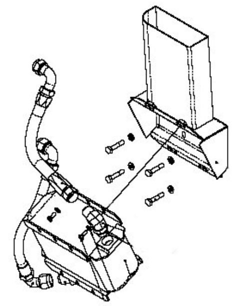

Figure 79-2 Exploded View of Engine Oil Cooler Installation

Section 20-01 Oil Cooler

Mounted on the left side of the firewall is the oil cooler. The oil cooler connects to the oil cooler adapter by supply and return hoses. An air duct directs air from the low-temperature/high-pressure portion of the cowling (above the engine cooling baffles) downward to the oil cooler. After passing through the oil cooler, the air is discharged overboard through the opening at the rear of the lower cowling. An internal “Vernatherm” unit in the oil cooler allows part or all of the oil flow to bypass the cooling passages in the oil cooler, thus regulating oil temperature.

Section 20-02 Air/Oil Separator

Some Liberty Aerospace, Inc. XL-2 airplanes come with an air/oil separator mounted on a bracket secured to the inboard electronic control unit (ECU) mounting bolts, see Figure 79-3. Crankcase oil vapors are routed from the crankcase breather to the air/oil separator, where the oil is separated from the vapors. The oil is then routed back to the oil tank, and the vapors are routed overboard, at the bottom of the engine cowling.

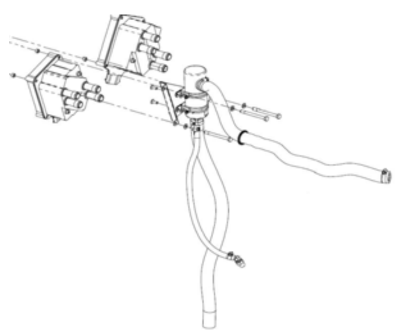

Figure 79-3 Exploded View of Air/Oil Separator

Section 20-03 Oil System Procedures

This section contains the procedures to remove and install the oil cooler and oil cooler hose, the air oil separator, the oil filter, and to change the engine oil.

OIL COOLER OR OIL COOLER HOSE REMOVAL

Perform this procedure to remove the oil cooler or oil cooler hoses.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Pull the BAT1 (CB001) circuit breaker to OPEN.
3. Pull the SYSTEM, START circuit breaker to OPEN
4. Remove upper and lower cowling in accordance with Chapter 71 – *Power Plant* of this manual
5. Position container and/or absorbent material below connections to be removed to accommodate unavoidable minor spillage.

NOTE

It is not necessary to drain engine oil for maintenance on oil cooler, oil cooler connections, or oil filter.

6. Drain engine oil in accordance with TCM Maintenance Manual M-22 Chapter 9.
7. Perform Oil Cooler removal procedures in accordance with TCM Maintenance Manual M-22 Chapter 10.
8. If hose(s) have been disconnected from oil cooler but remain attached to engine, secure free end of hose above level of top of oil sump to prevent possible (unlikely) siphoning or spillage.
9. Install caps on hoses and/or engine and oil cooler connections as required.
10. If necessary, remove bolts securing oil cooler to air duct; remove oil cooler.

This completes the Oil Cooler or Oil Cooler Hose Removal procedure

OIL COOLER OR OIL COOLER HOSE INSTALLATION

Perform this procedure to install the oil cooler or oil cooler hose.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Pull the BAT1 (CB001) circuit breaker to OPEN.
3. Pull the SYSTEM, START circuit breaker to OPEN
4. Perform oil cooler and hose installation in accordance with TCM Maintenance Manual M-22 Chapter 10.
5. Torque the nuts in accordance with table B-4, TCM IOF-240 Series Engine Maintenance Manual. M-22
6. Install engine oil in accordance with TCM Maintenance Manual M-22 Chapter 9
7. Push the BAT1 (CB001) circuit breaker to CLOSE.
8. Push the SYSTEM, START circuit breaker to CLOSE
9. Perform engine operation test in accordance with Chapter 71 – *Power Plant* of this manual.
10. Inspect oil cooler and hoses for leaks
11. Install upper and lower cowl in accordance with Chapter 71 – *Power* Plant of this manual.

This completes the Oil Cooler or Oil Cooler Hose Installation procedure.

AIR/OIL SEPARATOR REMOVAL

Perform this procedure to remove the air/oil separator.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Pull the BAT1 (CB001) circuit breaker to OPEN.
3. Pull the SYSTEM, START circuit breaker to OPEN
4. Remove upper and lower cowling in accordance with Chapter 71 – *Power* Plant of this manual.

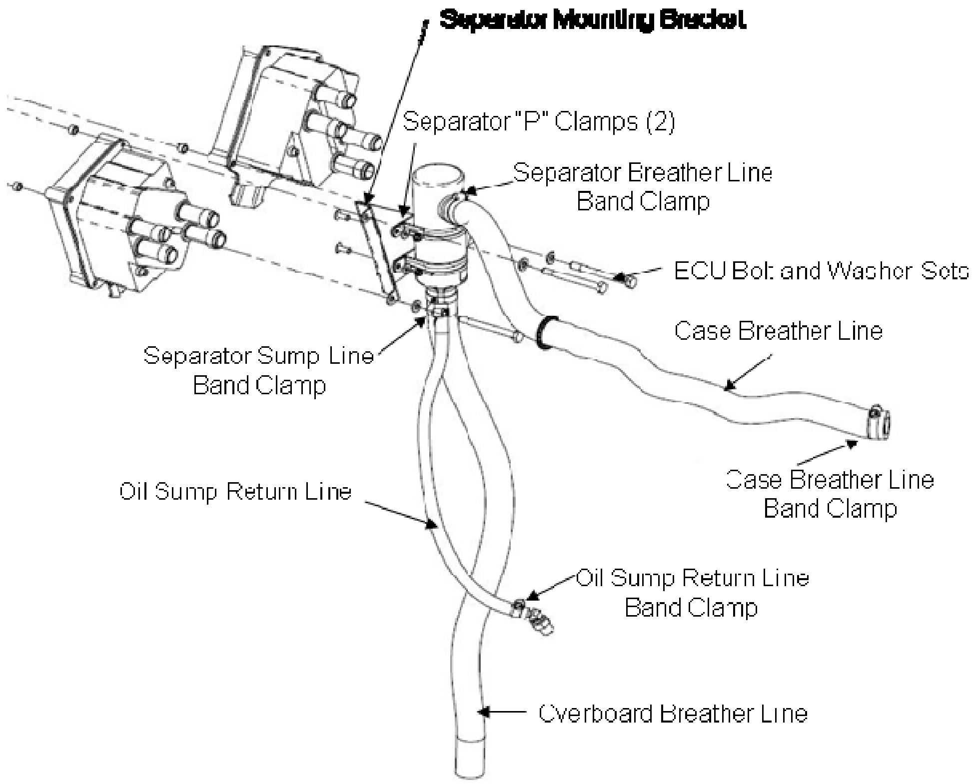

Figure 79-4 Air/Oil Separator Removal

5. Remove the clamps and hoses from the separator as shown in Figure 79-4
6. Remove the inboard mounting bolts and hardware from #1 and #2 ECU's.
7. Remove the "P" clamps holding the separator to the mounting bracket.

This completes the

Air/Oil Separator Removal procedure

AIR/OIL SEPARATOR INSTALLATION

Perform this procedure to install the air/oil separator.

1. Install the “P” clamps holding the separator to the mounting bracket.
2. Install the inboard mounting bolts and hardware from #1 and #2 ECU’s.

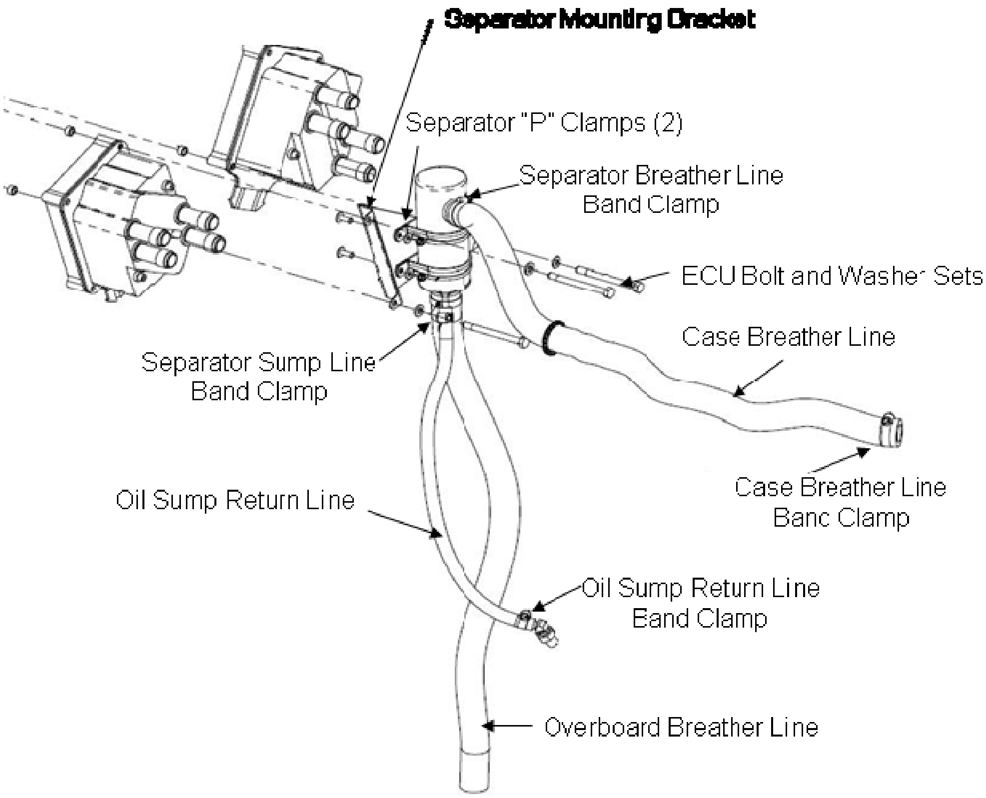

Figure 79-5 Air/Oil Separator Removal

3. Install the clamps and hoses from the separator as shown in Figure 79-5
4. Inspect line routing and adjust lines clear of abrading structures.
5. Position the BAT1 (CB001) circuit breaker - CLOSED.
6. Push the SYSTEM, START circuit breaker to CLOSE
7. Perform engine operation test in accordance with Chapter 71 – *Power Plan*t of this manual.
8. Inspect oil cooler and hoses for leaks
9. Install upper and lower cowl in accordance with Chapter 71 – *Power Plant* of this manual.

This completes the Air/Oil Separator Installation procedure.

OIL FILTER ELEMENT REMOVAL

Perform this procedure to remove the oil filter.

1. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
2. Pull the BAT1 (CB001) circuit breaker to OPEN.
3. Pull the SYSTEM, START circuit breaker to OPEN
4. Remove upper and lower cowling in accordance with Chapter 71 – *Power* Plant of this manual.
5. Position container and/or absorbent material below oil filter element to accommodate unavoidable minor spillage.
6. Using filter wrench, remove oil filter.

NOTE

It is recommended that filter element be opened (use oil filter cutter tool) and entire element be examined for metal or other foreign material.

This completes the

Oil Filter Element Removal procedure

OIL FILTER ELEMENT INSTALLATION

Perform this procedure to install the oil filter.

1. Place a thin film of clean engine oil on gasket of replacement filter element.
2. Thread filter element onto adapter and torque to 192-216 in-lbs.
3. Secure filter with safety wire.
4. Push the BAT1 (CB001) circuit breaker - CLOSED
5. Push the SYSTEM, START circuit breaker to CLOSE
6. Perform engine operation test in accordance with Chapter 71 – *Power* Plant of this manual.
7. Inspect oil filter installation for leaks.
8. Install upper and lower cowling in accordance with Chapter 71 – *Power* Plant of this manual

This completes the Oil Filter Element Installation procedure.

ENGINE OIL DRAINING PROCEDURE

Perform this procedure to drain the engine oil.

1. Start engine and allow oil to reach normal operating temperature.
2. After shutdown, ensure all electrical switches are off.
3. Remove upper and lower engine cowling.
4. Position appropriate container under engine oil sump drain plug.
5. Cut and remove safety wire.
6. Remove drain plug and gasket, drain oil.
7. Perform engine oil sampling in accordance with TCM IOF-240 Series Engine Maintenance Manual, Chapter 9.
8. When draining is complete, reinstall oil drain plug with NEW gasket.
9. Torque drain plug to 190-210 in-lbs.
10. Secure with safety wire.

Section 20-04 Troubleshooting Guide

Use Table 79-1 as an aid in troubleshooting issues with the oil system.

Complaint	Possible Cause	Remedy
Oil separator flow obstructed	Oil residue	Clean with engine solvent
Oil leaking from filter seal	Filter seal damaged Insufficient filter torque	Replace filter
Oil cooler lines leaking	Loose fittings	Tighten to spec IAW TCM Maintenance Manual M-22
	Damages hose	Replace hoses
Oil Cooler Element leaking	Remove and inspect if no damage reinstall at correct torque.	Tighten fittings Replace oil cooler

Table 79-1 Oil System Troubleshooting Guide

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Section 79-30 Indicating

An electronic oil temperature sensor is threaded into the oil cooler adapter. Information from this sensor is displayed on the VM1000FX Integrated Engine Instrument Display System

An electronic oil pressure sensor is mounted on the firewall and plumbed to the oil pressure port on the oil cooler adapter. Information from this sensor is displayed on the VM1000FX Integrated Engine Instrument Display System.

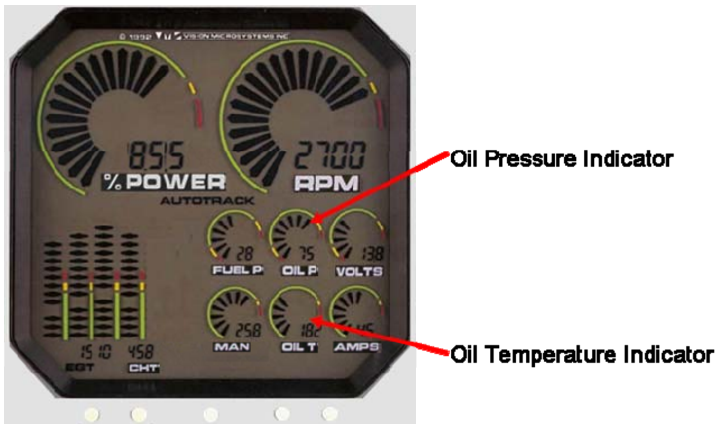
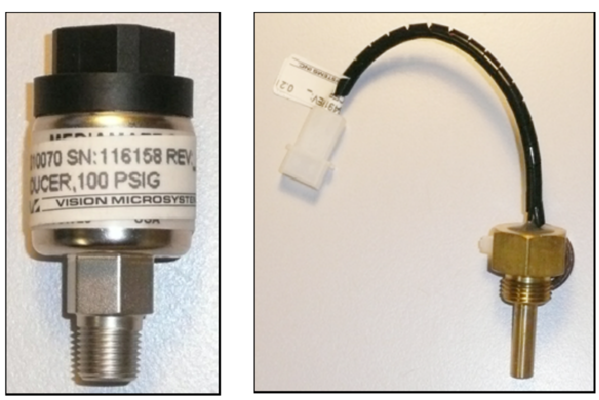


Figure 79-6 VM1000FX Oil Temperature and Pressure Indicators

Section 30-01 Oil Indicating Procedures

This section contains the removal and installation procedures for the oil temperature and pressure sensors as shown in Figure 79-7. The sensors are electronic components. Care must be taken to avoid application of excess forces during procedures and to protect electrical connections from contamination or damage.



Oil Pressure Sensor

Oil Temperature Sensor

Figure 79-7 Oil Temperature and Pressure Sensors

OIL PRESSURE SENSOR REMOVAL

Perform this procedure to remove either of the oil pressure sensors. The oil pressure sensors mount to the firewall, below the level of the Electronic Control Units, ECU. There are two sensors at this location. See Figure 79-8. The sensor on top is for the Hobbs meter, the sensor on the side is for the Health Status Annunciator. As these sensors are co-located, the procedure is the same for both. Any differences are noted in the text of the procedure.

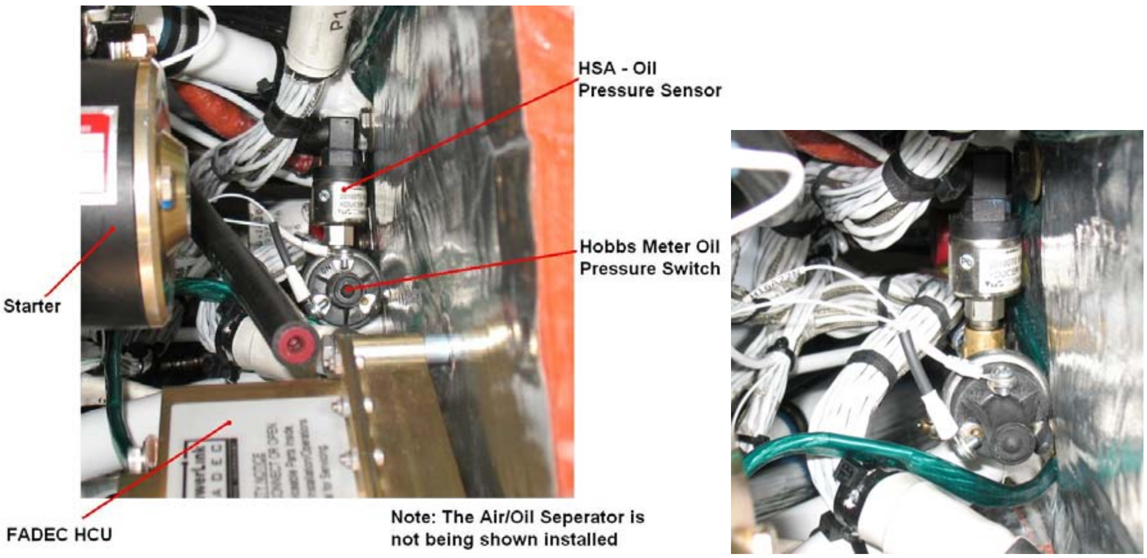


Figure 79-8 Location of the Oil Pressure Switch (Hobbs) and the Oil Pressure Sensor (HSA) (The Air/Oil Separator Not Shown)

1. Pull circuit breaker BAT 1 (CB001) to OPEN.
2. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
3. Remove the upper and lower engine cowling.
4. If the airplane is equipped with an air/oil separator mounted to the firewall between the Electronic Control Units (ECU), disconnect the air/oil separator and bracket from the Electronic Control Units.
5. Refer to the exploded view shown in Figure 79-9. Have someone hold a wrench on the nut from inside the cockpit.
6. Turn the bolt securing the mounting bracket to the firewall. Remove the bolt being careful to retain the flat washers and the sleeve spacer
7. Remove the bracket securing the oil pressure sensor T-fitting to the firewall.

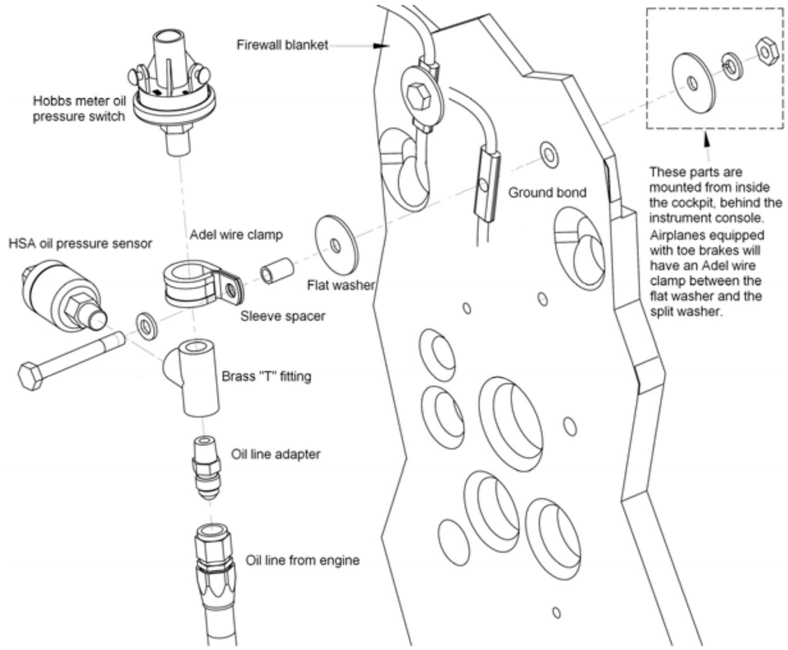


Figure 79-9 Exploded View of the Oil Pressure Mount

8. Bring the oil pressure sensor assembly up enough to allow access to the electrical connections on the sensors.
9. If removing Hobbs oil pressure switch, go to step 13.
10. Remove the screw in the center of the connector.
11. Disconnect the cable going to pressure sensor.
12. Go to Step 14.
13. Using a Phillips screwdriver, remove the two wires that connect to the top of the Hobbs oil pressure switch.
14. Wrap the T-fitting with a rag to catch any oil.
15. Use a 7/16-inch wrench to remove the oil pressure sensor.
16. If installation of a replacement sensor will happen at a later time, cap the open end of the T-fitting, and secure.
17. If an air/oil separator was removed, temporarily install the air/oil separator and secure.

This completes the Oil Pressure Sensor Removal procedure.

OIL PRESSURE SENSOR INSTALLATION

Perform this procedure to install the oil temperature sensor.

1. If installing a replacement sensor for the oil pressure immediately after removing the previous oil pressure sensor, then proceed to step 9 below.
2. Pull circuit breaker BAT 1 (CB001) to OPEN.
3. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
4. Remove the upper and lower engine cowling.
5. If the airplane is equipped with an air/oil separator mounted to the firewall between the Electronic Control Units (ECU), disconnect the air/oil separator and bracket from the Electronic Control Units.
6. Refer to the exploded view shown in Figure 79-9. Have someone hold a wrench on the nut from inside the cockpit.
7. Turn the bolt securing the mounting bracket to the firewall. Remove the bolt being careful to retain the flat washers and the sleeve spacer
8. Remove the bracket securing the oil pressure sensor T-fitting to the firewall.
9. Install the oil pressure sensor to the T-fitting.
10. If installing Hobbs oil pressure switch, go to step 14.
11. Connect the cable going to pressure sensor.
12. Secure the connector with the screw in the center of the connector.
13. Go to Step 15.
14. Using a Phillips screwdriver, secure the two wires to the top of the Hobbs oil pressure switch.
15. Check the connector or connections on the other oil pressure sensor to make sure the connections are secure.
16. Using Figure 79-9 as a guide, assemble the mounting bracket and hardware.
17. Push the bolt through the firewall blanket to the inside of the cockpit.
18. Have some one hold the bolt from the engine compartment. Assemble the hardware from the cockpit side as shown in Figure 79-9.
19. Tighten the nut to secure the hardware.
20. Remove any rags that were absorbing oil.
21. Clean oil spilled during this procedure.

This completes the Oil Pressure Sensor Installation procedure.

OIL TEMPERATURE SENSOR REMOVAL

Perform this procedure to remove the oil temperature sensor. The oil temperature sensor is located on the port side of the engine up next to the firewall. See Figure 79-10 for the location of the oil temperature sensor. The location makes it very difficult to remove and/or install. Before attempting to remove/install the sensor, check all of the wiring and connections. Make sure the fault is not in the wiring or connectors.

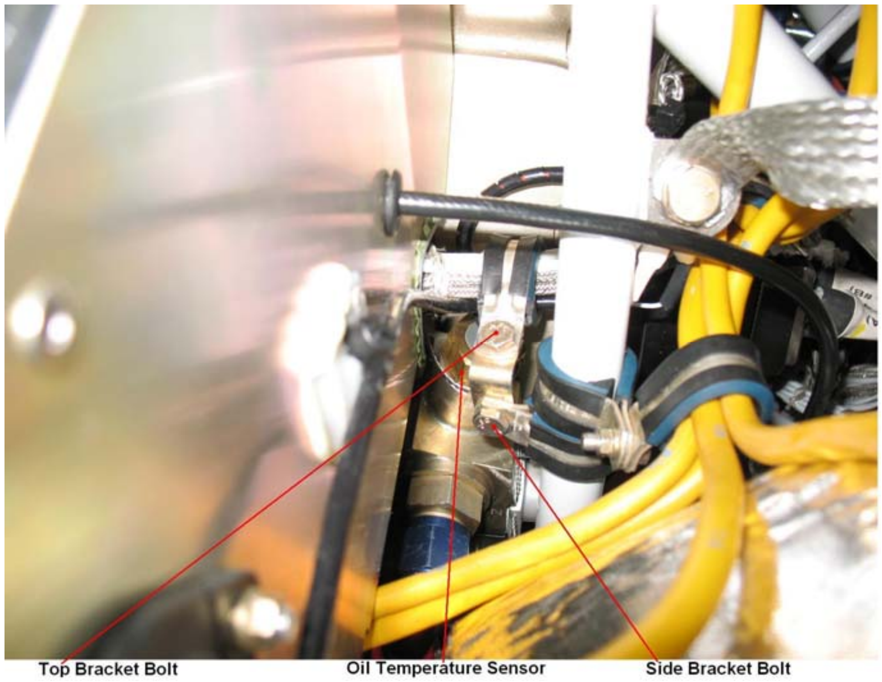


Figure 79-10 Location of the Oil Temperature Sensor

1. Pull circuit breaker BAT 1 (CB001) to OPEN.
2. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
3. Remove the upper and lower engine cowling.
4. Remove the top bracket 3/8-inch bolt and ¼-inch nut that secures the wire clamp to the L-bracket. Carefully remove the cable clamp from the wire bundle.
5. Loosen the side bracket bolt and nut that secures the wire clamp to the engine support tube.

NOTE

Spraying the cable clamp with a small amount of alcohol will allow the cable clamp to slide easier on the support tubing.

6. Slide the wire clamp down the tube and out of the way.

7. Cut and remove the safety wire from the sensor.
8. Carefully remove any cable ties securing the bundle of wires above the sensor.
9. Disconnect the P/J61.
10. Position the wire bundle away from the top of the oil temperature sensor.
11. Pack the area of the around the sensor to catch any oil that may come out of the fitting.
12. Use a 7/8-inch open-ended wrench to remove the oil temperature sensor.

NOTE

With the two Adel wire clamps out of the way, there is just enough room to get the wrench on to the sensor and turn it just enough to break the threads free. Then bring the wrench straight down from the top and complete the removal of the sensor. See Figure 79-11 to see the correct positioning to complete the removal of the oil temperature sensor.

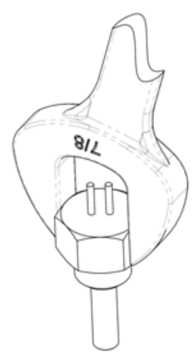


Figure 79-11 Positioning the Wrench to Remove the Oil Temperature Sensor

13. With the oil temperature sensor removed, remove the remainder of the safety wire from the sensor-mounting bracket. Install new safety wire in to the bracket to prepare for the installation of a replacement sensor.

NOTE

Liberty Aerospace, Inc. recommends to start the installation of the safety wire at this point, while the sensor is out of the engine.

14. If replacing the oil temperature sensor later, plug the open hole where the sensor goes, secure all wires and connectors, secure mounting hardware.

This completes the Oil Temperature Sensor Removal procedure.

OIL TEMPERATURE SENSOR INSTALLATION

Perform this procedure to install the oil temperature sensor.

1. Pull circuit breaker BAT 1 (CB001) to OPEN.
2. If installing replacement oil temperature sensor immediately after removing the previous oil temperature sensor, then proceeds to step 6 below.
3. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
4. If installed, remove the upper and lower engine cowling.
5. Reposition the cable mounting hardware associated with the cable above the oil temperature sensor.
6. Reposition the bundle of cables above the sensor.
7. Wrap the area with a rag to absorb and oil.
8. Remove any plug from the hole for the sensor.
9. If a new safety wire was not installed previously, insert a fresh safety wire and prepare it for the sensor.
10. Assemble the oil temperature sensor’s copper crush gasket on to the sensor.

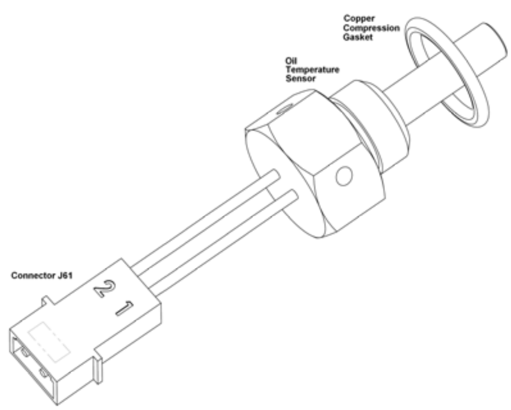


Figure 79-12 Oil Temperature Sensor Showing the Copper Compression Gasket

11. Insert oil temperature sensor assembly into the hole for the oil temperature sensor. Use a 7/8-inch wrench as shown in Figure 79-11 to turn-in the oil temperature sensor.
12. Torque the sensor to 220 in-lbs.
13. Complete the installation of the safety wire.
14. Connect P61 to J61.
15. Secure the cable bundle using wire ties.
16. Move the cable clamp back into position and tighten the side bracket bolt.

17. Assemble the wire bundle cable clamp to the L-bracket. Secure the clamp to the L-bracket using the bolt and nut removed in step 4 in the Oil Temperature Sensor Removal procedure on page 23 of this chapter.
18. Remove any rags that were absorbing oil.
19. Clean oil spilled during this procedure.

This completes the Oil Temperature Sensor Installation procedure.

Section 30-02 Troubleshooting Guide

Use Table 79-2 to aid in troubleshooting the oil sensors.

Complaint	Possible Cause	Remedy
Oil Temperature Sensor:		
Temperature Intermittent	loose wiring or connector Sensor bad	Repair Replace
No indication	loose wiring or connector Sensor bad	Repair Replace
Oil Pressure Sensor:		
Pressure Intermittent	loose wiring or connector Sensor bad	Repair Replace
No indication	loose wiring or connector Sensor bad	Repair Replace

Table 79-2 Troubleshooting Oil Sensors

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CHAPTER 80

STARTING

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Table of Contents

SECTION 80-00 GENERAL	5
SECTION 00-01 STARTER REMOVAL AND INSTALLATION	5
STARTER REMOVAL	6
STARTER INSTALLATION	7
SECTION 00-02 STARTER TROUBLESHOOTING	8
SECTION 80-10 STARTING	9
SECTION 10-01 STARTING PROCEDURES AT LOW TEMPERATURES	9
ENGINE PREHEATING WITH COMBUSTION HEATER	10
LOW TEMPERATURE ENGINE STARTING	12
SECTION 10-02 TROUBLESHOOTING ENGINE STARTING ISSUES	14

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Section 80-00 General

The IOF-240-B engine installed on the airplane is equipped with a lightweight electric starter.

The starter is installed on the accessory case at the top center of the rear of the engine and drives the crankshaft directly through internal gears. See Figure 80-1 for the location of the starter. When the starter is energized, a solenoid on the starter engages its pinion with the crankshaft drive gear. As the solenoid reaches its fully extended position, it closes the primary power circuit to the starter and the starter rotates the crankshaft to start the engine.

When the engine starts and exceeds starter cranking speed, spiral splines on the starter pinion drive shaft cause initial disengagement of the pinion. Removing power from the starter solenoid removes power from the starter and completes the disengagement of the starter drive train.

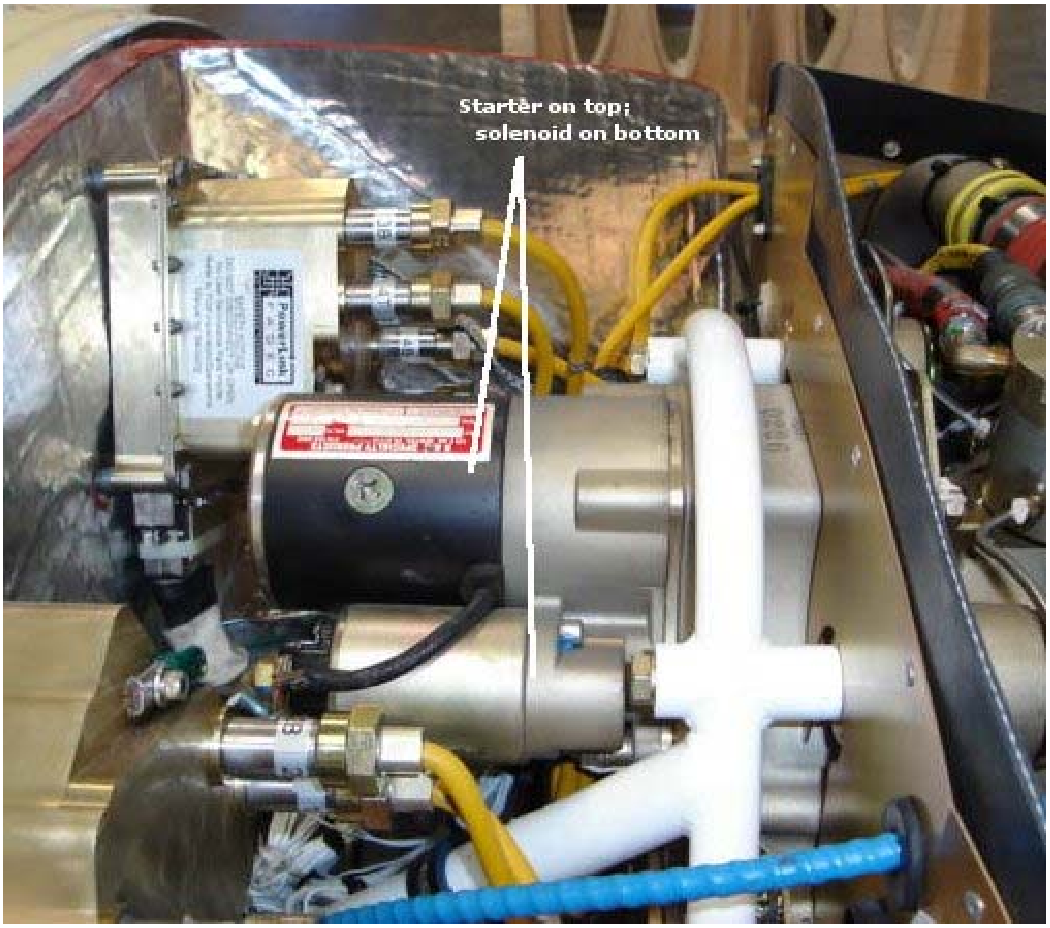

Figure 80-1 Location of the Starter for the Engine

The starter is not a field serviceable item. Maintenance is limited to starter removal and replacement.

Section 00-01 Starter Removal and Installation

This section contains the procedures to remove and install the starter.

STARTER REMOVAL

Perform this procedure to remove the starter from the engine.

1. Pull circuit breaker BAT 1 (CB001) to OPEN.
2. Position the ALT and BAT master switches, the FADEC PWR A and B switches, and the ignition switch to OFF.
3. Remove the upper engine cowling.
4. Disconnect cable P04A4 from power stud on starter.
5. Remove (3) nuts, lock washers, and seal washers securing starter to studs on rear of accessory case.
6. Remove through- bolt, lock washer, and plain washers securing top of starter to accessory case.

NOTE

Engine rear lifting eye, if installed, is attached to through-bolt.

7. Gently “tap” starter to break gasket adhesion and remove by moving straight aft, away from engine.

STARTER INSTALLATION

Perform this procedure to install the starter.

1. Ensure mating surfaces on starter face and engine accessory gear-case are clean and free of oil.
2. Coat surfaces lightly with TCM approved gasket sealant.
3. Place NEW starter gasket over accessory case studs.
4. Place starter over accessory case studs. Install (3) seal washers, (3) lock washers, and (3) nuts.
5. Install lock washer, plain washers, and engine lifting eye (if present) on through-bolt. Install bolt through top of engine accessory case and into starter; torque to 240 +/- 20 in/lbs.
6. Torque (3) nuts on starter studs to 240 +/- 20 in/lbs.
7. Reconnect starter cable; torque to 100 +/- 5 in/lbs.
8. Reconnect negative battery cable; reinstall aft baggage compartment closeout.
9. Perform functional test.

Section 00-02 Starter Troubleshooting

This section contains a table that aids in troubleshooting issues with the starter.

Complaint	Possible Cause	Remedy
Starter inoperative (“STARTER ENGAGED” does not illuminate)	defective “START” circuit breaker	replace
	defective ignition/start switch	replace
	defective start contactor	replace
	defective wiring	repair
Starter inoperative (“STARTER ENGAGED”) illuminates	defective starter	replace
	defective cable from start contactor to starter	repair or replace
Starter operates but does not rotate engine	defective solenoid	replace starter
“START ENGAGED” annunciator remains on after start switch is released	defective starter contactor	replace
	defective ignition/start switch	replace
	defective (shorted) wiring	repair
Starter operation sluggish	battery discharged	charge battery
	defective starter	replace
	defective wiring / loose contacts at starter contactor and/or between contactor and starter	repair or replace cables as necessary
	defective ground strap or loose contact between engine case and airframe	repair or replace ground strap

Table 80-1 Starter System Troubleshooting

Section 80-10 Starting

The starter solenoid is energized by DC power from the spring-loaded "start" position of the ignition/starter key switch. This circuit is powered by a 2-amp circuit breaker on the main distribution bus. Primary power is provided by a heavy-gauge cable from the battery contactor in the aft fuselage. This cable is energized any time the master battery contactor is closed (BAT/ALT Master Switch in ON position).

An instrument panel annunciator illuminates to indicate that the start contactor has closed which supplies power to the starter. This circuit is protected by an inline 1-amp fuse. A press-to-test feature checks the annunciator lamps only.

Section 10-01 Starting Procedures at Low Temperatures

This section contains the procedures to pre-heat and start the engine after long-term (≥ 2-hours) exposure to very low temperature (≤ 20°F/ -7°C). Engine preheating is required to facilitate engine starting during cold weather and when the engine has been exposed to temperatures below 20ºF/-7°C for more than 2 hours. The preferred method of preheating is to place the aircraft in a heated hangar for a minimum of 4 hours prior to flight.

WARNING

FAILURE TO PROPERLY PREHEAT A COLD-SOAKED ENGINE MAY RESULT IN OIL CONGEALING WITHIN THE ENGINE, OIL HOSES, AND OIL COOLER WITH SUBSEQUENT LOSS OF OIL FLOW, POSSIBLE INTERNAL DAMAGE TO THE ENGINE, AND SUBSEQUENT ENGINE FAILURE.

WARNING

SUPERFICIAL APPLICATION OF PREHEAT TO A COLD SOAKED ENGINE CAN CAUSE DAMAGE TO THE ENGINE. AN INADEQUATE APPLICATION OF PREHEAT MAY WARM THE ENGINE ENOUGH TO PERMIT STARTING BUT WILL NOT DE-CONGEAL OIL IN THE SUMP, LINES, COOLER, FILTER, ETC. CONGEALED OIL IN THESE AREAS REQUIRES CONSIDERABLE PREHEAT. THE ENGINE MAY START AND APPEAR TO RUN SATISFACTORILY, BUT CAN BE DAMAGED FROM LACK OF LUBRICATION DUE TO THE CONGEALED OIL BLOCKING PROPER OIL FLOW THROUGH THE ENGINE. THE AMOUNT OF DAMAGE WILL VARY AND MAY NOT BECOME EVIDENT FOR MANY HOURS. HOWEVER, THE ENGINE MAY BE SEVERELY DAMAGED AND MAY FAIL SHORTLY AFTER APPLICATION OF HIGH POWER.

ENGINE PREHEATING WITH COMBUSTION HEATER

Perform this procedure to preheat the engine using a combustion heater high volume combustion heater with ducts directed to the engine oil sump, cylinders, and oil cooler.

CAUTION

Proper engine preheating procedures require thorough application of preheat to all parts of the engine. Apply hot air directly to the oil sump and external oil lines, the cylinders, air intake, and oil cooler. Because excessively hot air can damage non-metallic components such as seals, hoses, and drive belts, do not attempt to hasten the preheat process.

1. Select a high volume hot air heater.

NOTE

Small electric heaters inserted in the cowling opening do not appreciably warm the oil and may result in superficial preheating.

2. Preheat the following engine parts. Apply preheated air directly to these listed parts for at least 30 minutes:

- Oil sump
- Oil filter
- External oil lines
- Oil cooler
- Cylinder assemblies
- Air intake

3. Periodically feel the top of the engine for warmth. Apply heat directly to the induction tubes and cylinders will promote vaporization and ease starting. Alternately, heat the sump and engine cylinders until engine start.
4. After preheating, remove heater and all associate air ducts from the area around the airplane.
5. Start the engine immediately after completion of the preheating process.

CAUTION

If there is no indication of oil pressure within 30 seconds, shut down the engine and determine the cause. Operating the engine without oil pressure may result in engine malfunction or stoppage.

6. Operate the engine at 1000 RPM until there is some indication of oil temperature.

7. Monitor the oil pressure closely. If necessary, retard the throttle to maintain oil pressure below 100 psi. If oil pressure is less than 30 psi, or cannot be maintained below 100 psi, shut the engine down and repeat the preheat process.

8. Monitor the oil temperature until it reaches at least 75°F (24°C).

CAUTION

Do not operate the engine at speeds above 1700 RPM unless the oil temperature is at least 75°F (24°C) and the oil pressure is between 30 to 60 psi.

9. Run the engine up to 1700 RPM; approach this RPM in increments to prevent oil pressure from exceeding 100 psi.

WARNING

OPERATING THE ENGINE ABOVE 1700 RPM BEFORE REACHING THE MINIMUM OIL TEMPERATURE MAY RESULT IN ENGINE MALFUNCTION, ENGINE FAILURE, INJURY, OR DEATH.

CAUTION

Continually monitor oil pressure during run up.

10. When oil temperature has reached 100°F (38°C) and oil pressure does not exceed 60 psi at 2500 RPM, the engine has been warmed sufficiently to accept full rated power.

LOW TEMPERATURE ENGINE STARTING

Perform this procedure to start the engine at temperatures between 20°F/-7°C and 40°F/4°C.

1. Check the charge on the airplane's main and secondary battery. If not fully charged, charge the batteries before attempting to start the engine.

CAUTION

Attempting to start your engine with a partially discharged aircraft primary battery may result in damage to the starter relay and possible engine kickback, resulting in a broken starter gear.

WARNING

OVER-PRIMING CAN CAUSE A FLOODED INTAKE RESULTING IN A "HYDRAULIC LOCK" EVENT AND SUBSEQUENT ENGINE MALFUNCTION OR FAILURE. ENSURE ALL FUEL HAS DRAINED FROM THE INTAKE MANIFOLD AND/OR CYLINDER PRIOR TO ATTEMPTING TO START AN OVER-PRIMED OR FLOODED ENGINE.

CAUTION

If there is no indication of oil pressure within 30 seconds, shut down the engine and determine the cause. Operating the engine without oil pressure may result in engine malfunction or stoppage.

2. Operate the engine at 1000 RPM until there is some indication of oil temperature.
3. Monitor the oil pressure closely. If necessary, retard the throttle to maintain oil pressure below 100 psi. If oil pressure is less than 30 psi, or cannot be maintained below 100 psi, shut the engine down, and follow preheat instructions to prevent engine damage.
4. Check the oil temperature; it should be at least 75°F (24°C).

CAUTION

In the next step, do not operate the engine at speeds above 1700 RPM unless the oil temperature is at least 75°F (24°C) and the oil pressure is between 30 to 60 psig.

5. Run the engine up to 1700 RPM; approach this RPM in increments to prevent oil pressure from exceeding 100 psi.

WARNING

OPERATING THE ENGINE ABOVE 1700 RPM BEFORE REACHING THE MINIMUM OIL TEMPERATURE MAY RESULT IN ENGINE MALFUNCTION, ENGINE FAILURE, INJURY OR DEATH.

CAUTION

Continually monitor oil pressure during run up.

6. When oil temperature has reached 75°F (24°C) and oil pressure does not exceed 60 psi at 2500 RPM, the engine is warm enough to accept full rated power.

Section 10-02 Troubleshooting Engine Starting Issues

Refer to the table below to troubleshoot issues starting the engine after exposure to low temperature.

Complaint	Possible Cause	Remedy
Insufficient oil pressure	Congealed oil in lines	Shut down engine and repeat preheat procedure
Oil pressure remains above 100 psi at low RPM	Congealed oil in lines	Shut down engine and repeat preheat procedure
Engine not turning	Insufficient charge on the primary battery	Charge primary battery
Engine is turning but does not start	Insufficient charge on the secondary battery	Charge secondary battery

Table 80-2 Troubleshooting Table for Engine Starting Issues

CHAPTER 91

WIRING DIAGRAMS

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Table of Contents

SECTION 91-00	GENERAL	5
SECTION 00-01	WIRING STANDARDS	5
SECTION 00-02	WIRE NUMBERING	6
SECTION 00-03	WIRE COLOR CODE DESIGNATION	7
SECTION 91-01	AIRPLANE BLOCK DIAGRAM	9
SECTION 91-02	SCHEMATIC DIAGRAMS	13

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Section 91-00 General

This chapter contains the wiring diagrams, schematics, and wire lists for the Liberty Aerospace, Inc. XL-2 airplane. These drawings are up-to-date as of the release of this chapter and are for general information only. Table 91-1 has the list of Liberty Aerospace, Inc. drawings and their current versions and forms a part of this chapter by reference. However, the most current version levels available from Liberty Aerospace, Inc. of these drawings should be used as the definitive source for information.

Part Number	Description	Revision
135A-81-001	Schematic, Electric System General Layout	B
135A-81-010	Schematic, Power Generation and Distribution	P
135A-81-011	Schematic, Starting System	H
135A-81-012	Schematic, Fuel System	J
135A-81-013	Schematic, FADEC System Cable Harness	P
135A-81-014	Schematic, VM1000FX	K
135A-81-015	Schematic, Annunciation	J
135A-81-016	Schematic, Flap System	L
135A-81-017	Schematic, Pitch Trim System	L
135A-81-018	Schematic, Interior Lighting	M
135A-81-019	Schematic, Exterior Lighting	M
135A-81-020	Schematic, Electric Gyro Instruments	K
135A-81-023	Schematic, Facility Socket	E
135A-81-024	Schematic, Emergency Locator Transmitter (121.5/243MHz)	A
135A-81-026	Schematic, M803 OAT/ Clock	C
135A-81-027	Schematic, Avionics System Garmin Layout	A
135A-81-028	Schematic, Stall Warning	F
135A-81-029	Schematic, Hour Meter	C
135A-81-031	Schematic, EDI-200 Interface Harness	A
135A-81-034	Schematic, Garmain Avionics Full Stack Interconnect	H
135A-81-037	Schematic, Pitot Heat	B
135A-81-044	Schematic, Artex Emergency Locator Transmitter (121.5/406MHz)	A

Table 91-1 Liberty Aerospace, Inc. XL-2 Airplane Schematic Drawings

Section 00-01 Wiring Standards

Wiring used in the Wiring Diagrams for non-shielded wires are per MIL-W-22759. All non-shielded wires are 22 AWG unless otherwise noted. Wiring used in the Wiring Diagrams for shielded wires are per MIL-W-27500.

Section 00-02 Wire Numbering

Each wire in the XL-2 airplane has a unique number. The number defines what circuit or system the wire belongs to, the route for the wire, the route subsection, the gauge of the wire, and if the wire is a travelling ground.

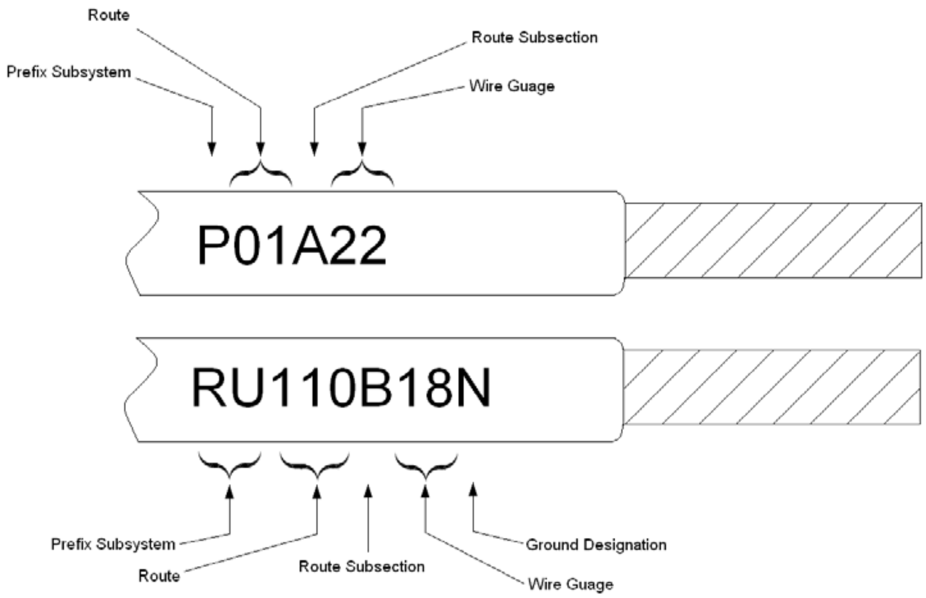

Figure 91-1 Wire Number Coding Standard

Table 91-2 gives the system description for the subsystem designation on the wire number. The route subsection designation for the route from source to termination.

Sub System Designation	System Description
A	Antenna
AP	Autopilot
C	Clock and Facility socket
E	Fuel System
EGND	Earth Ground – ground to aircraft frame
F	FADEC
G	Gyro Instrumentation
H	Pitot Heat, Hour Meter, Outside Air Temperature
IGND	Instrumentation Ground – ground path isolated from Earth Ground
JPR	Jumper Wire
L	Lighting
P	Power Distribution
R	Avionics General

Sub System Designation	System Description
RD	Avionics Display Systems (glass cockpit)
RG	Avionics Garmin
RU	Avionics UPS Aviation Technologies
SP	Splice
ST	Solder Sleave
T	Trim Flaps
V	VM1000FX
W	Annunciation, Stall Warning

Table 91-2 Subsystem Prefix Designation for Wire Numbers

Section 00-03 Wire Color Code Designation

Table 91-3 defines the abbreviations used in the schematics for various wire colors. If a wire has multiple colors, the different colors are separated by a forward slash (/) between each color. The base color is first, followed by the second significant color then the third significant color and so on.

For example, a wire that has a blue base color, then red then green, would have the designation BLU/RED/GRN.

Color Code	Color
BLK	Black
BRN	Brown or Tan
RED	Red
ORG	Orange
YEL	Yellow
GRN	Green
BLU	Blue or Azure
VIO	Violet or Purple
SLT	Grey or Slate
WHT	White
SLD	Shield
TIP	Center conductor of a Coax Cable

Table 91-3 Wire Color Code Abbreviations

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Section 91-01 Airplane Block Diagram

This section has the block diagram of the airplane's electric system. Figure 91-2 shows the block diagram.

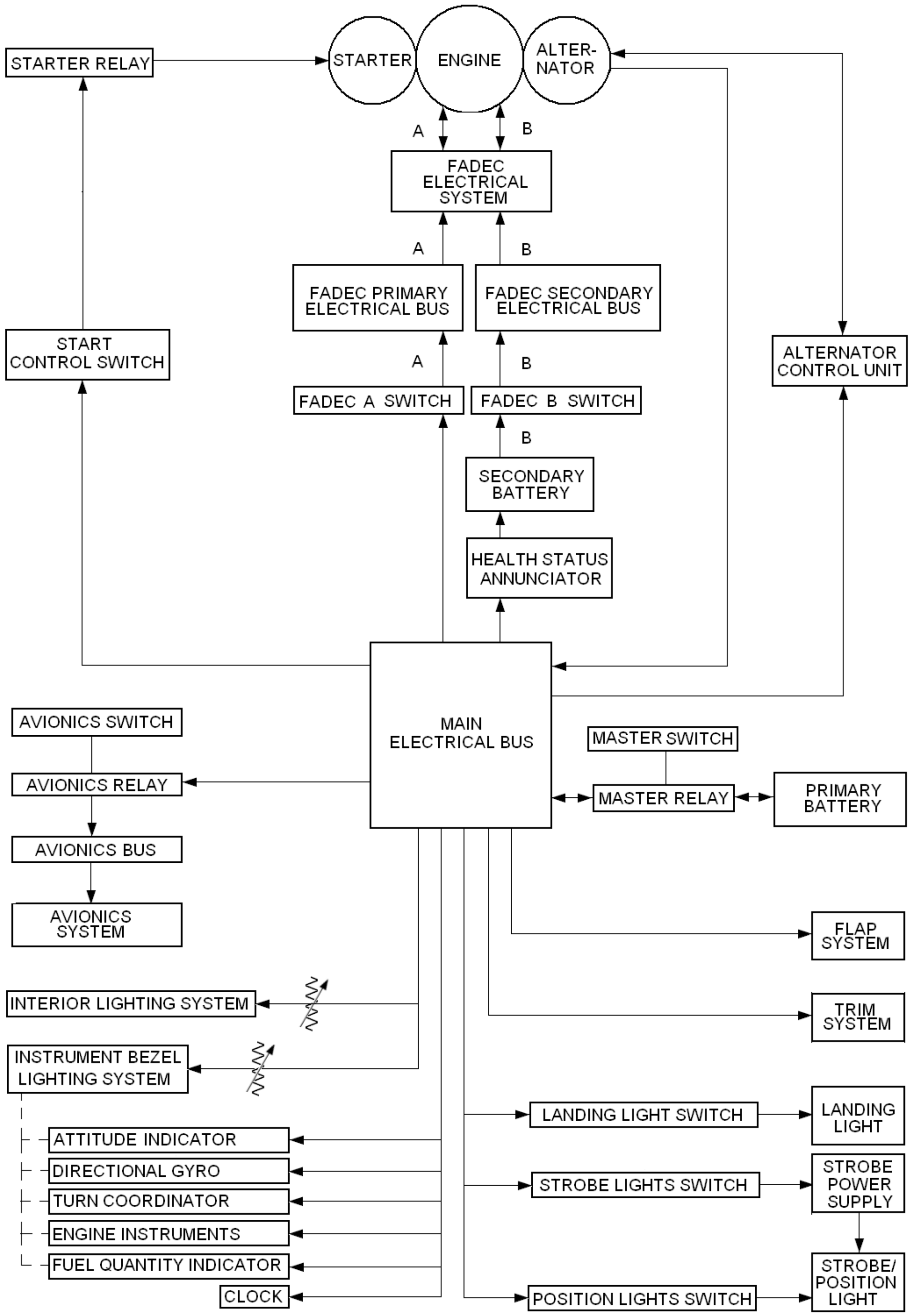

Figure 91-2 Block Diagram of the Liberty XL-2 Airplane

Figure 91-3 through Figure 91-5 shows the hierarchy of the schematic drawings. The numbers shown within the dashed boxes are Liberty Aerospace, Inc. drawing numbers. Use this diagram as an aid to locate the correct drawings for a given system.

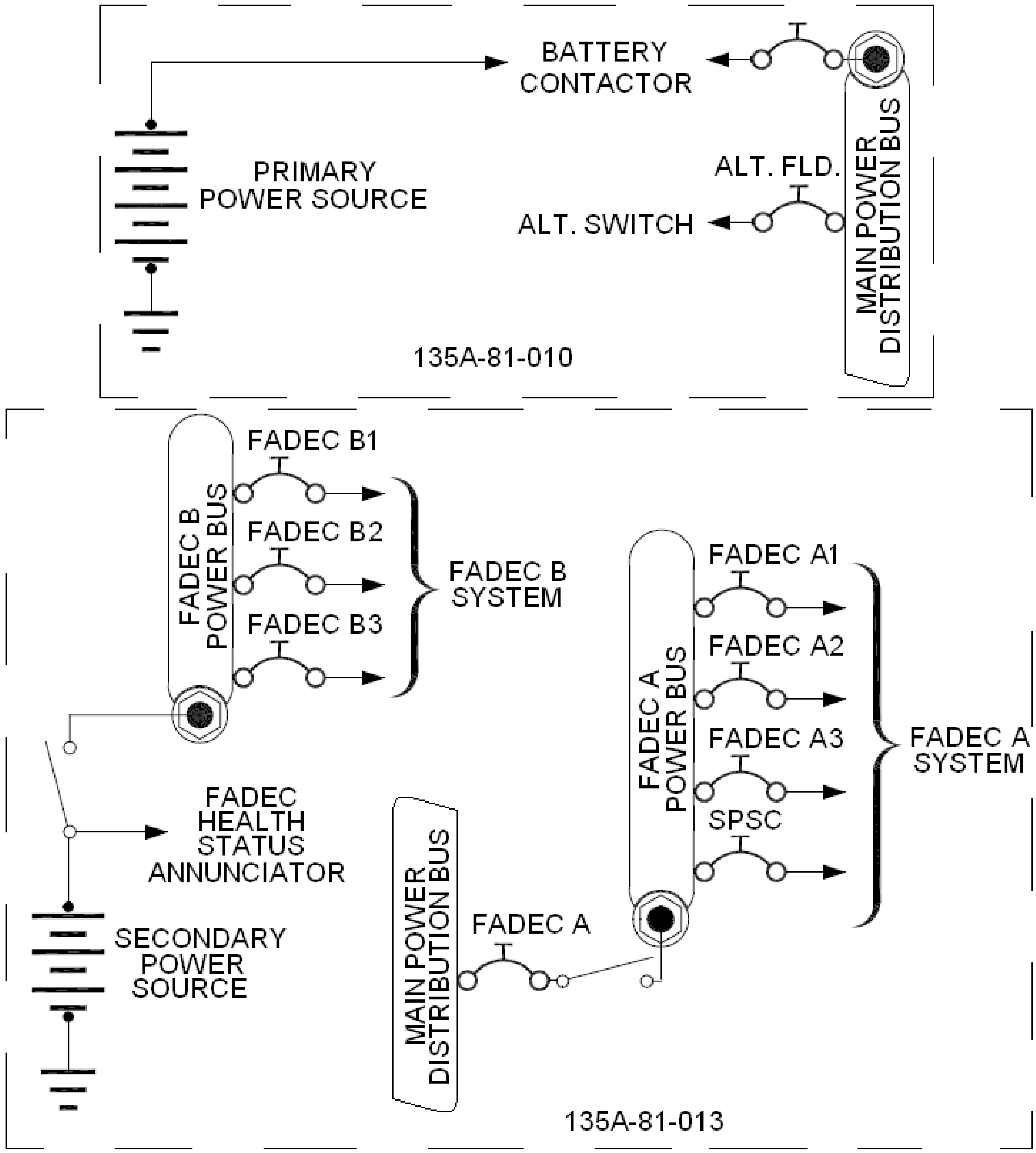

Figure 91-3 Drawings For the Two Battery Systems

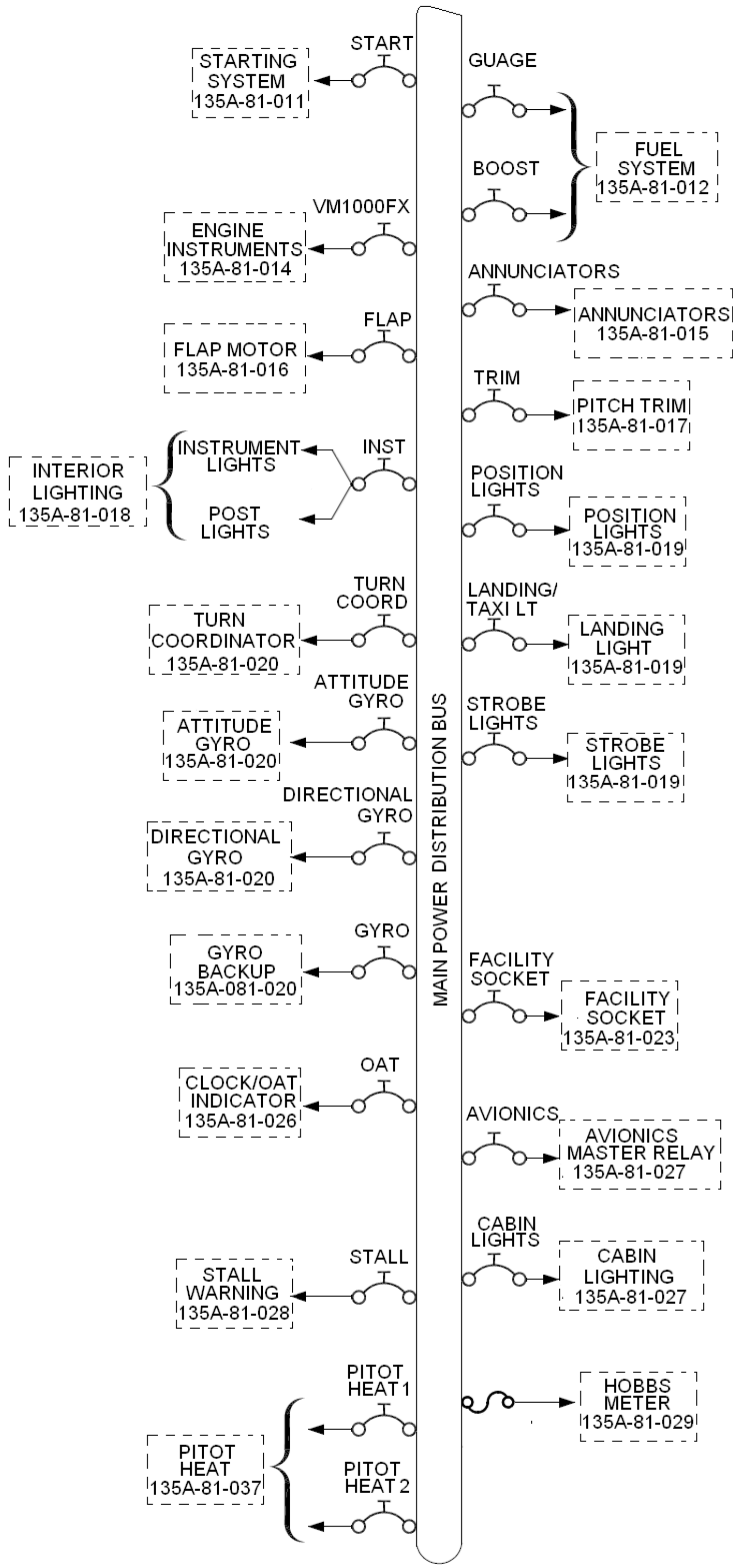

Figure 91-4 Drawing Hierarchy of the Main Power Distribution Bus

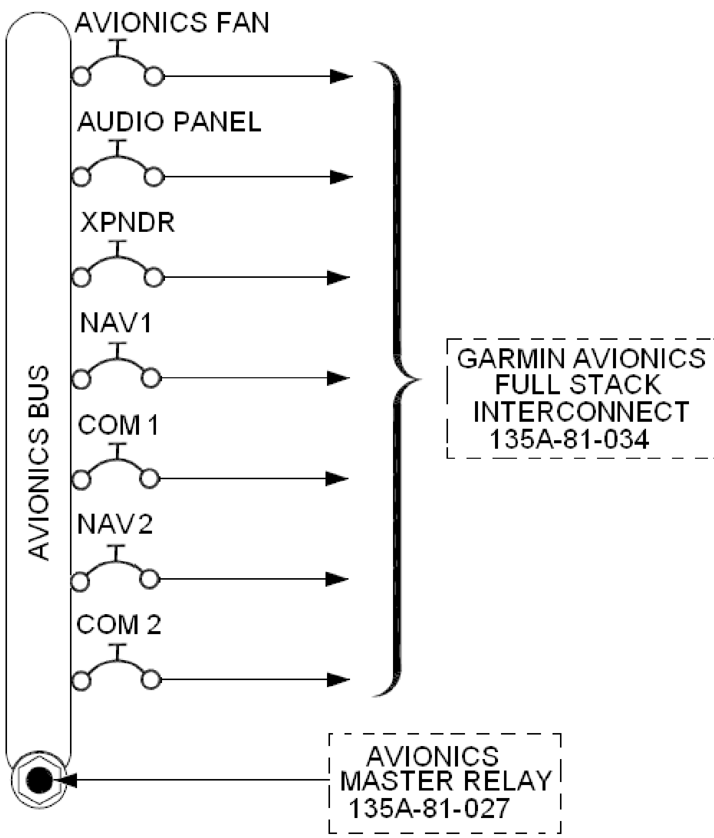

Figure 91-5 Drawing Hierarchy

Section 91-02 Schematic Diagrams

This section contains schematic diagrams for the Liberty Aerospace, Inc. XL-2 airplane. These diagrams are for reference only and can be used for basic troubleshooting. The Liberty Aerospace, Inc. drawings called out in Figure 91-3 through Figure 91-5 and Table 91-1 are the most up-to-date and should be used as the most accurate. Differences between these drawings and a specific airplane are due to updates and changes that occurred after production of the airplane.

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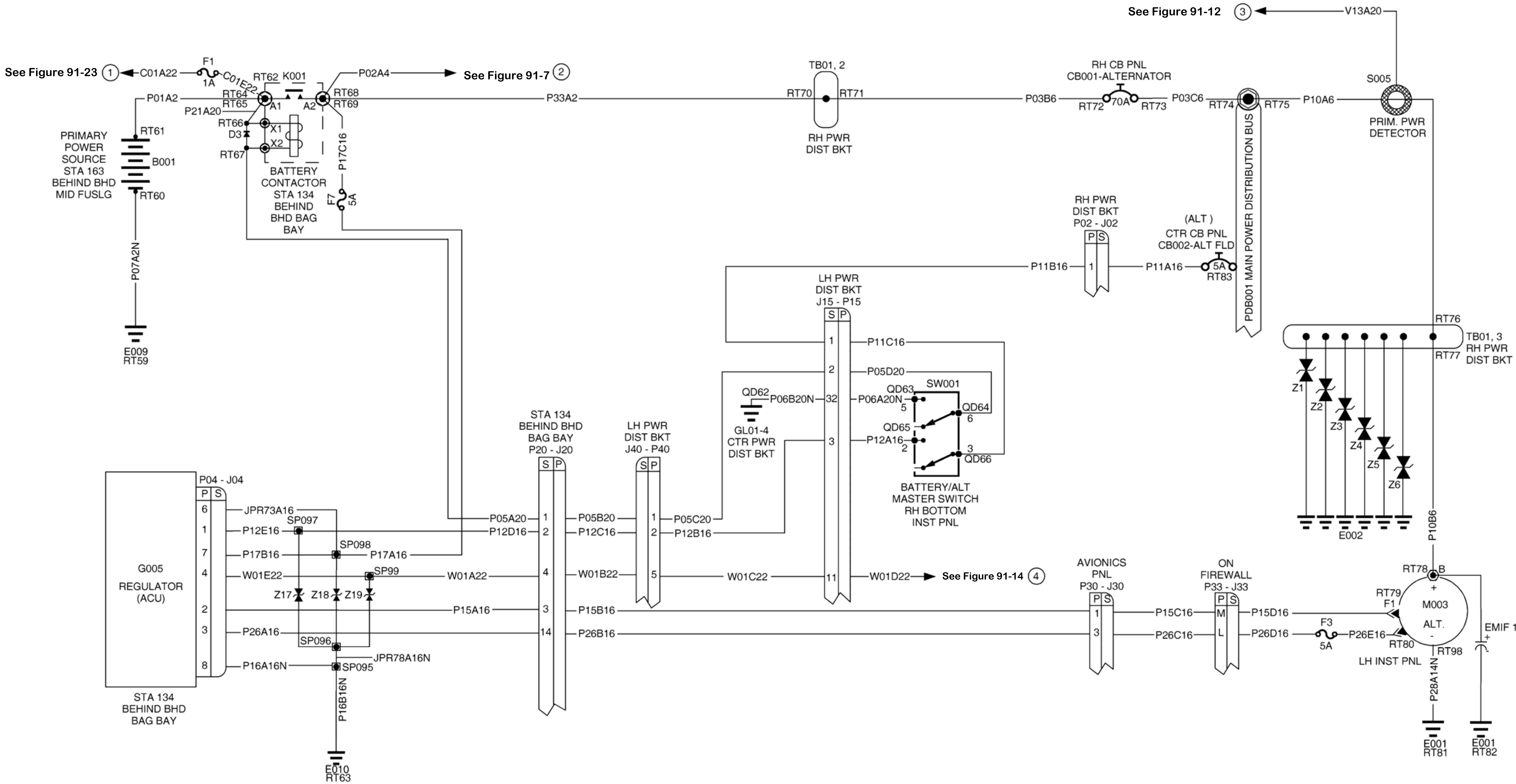

Figure 91-6 Power Generation and Distribution

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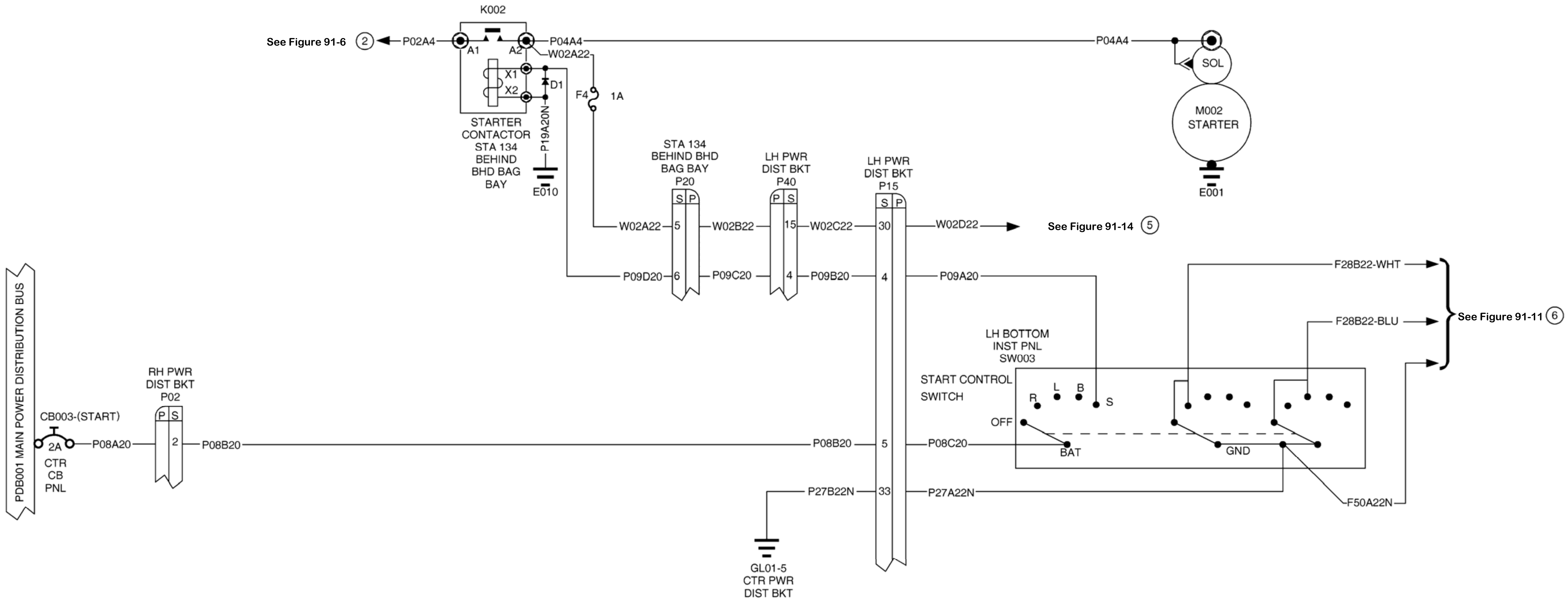

Figure 91-7 Starting System

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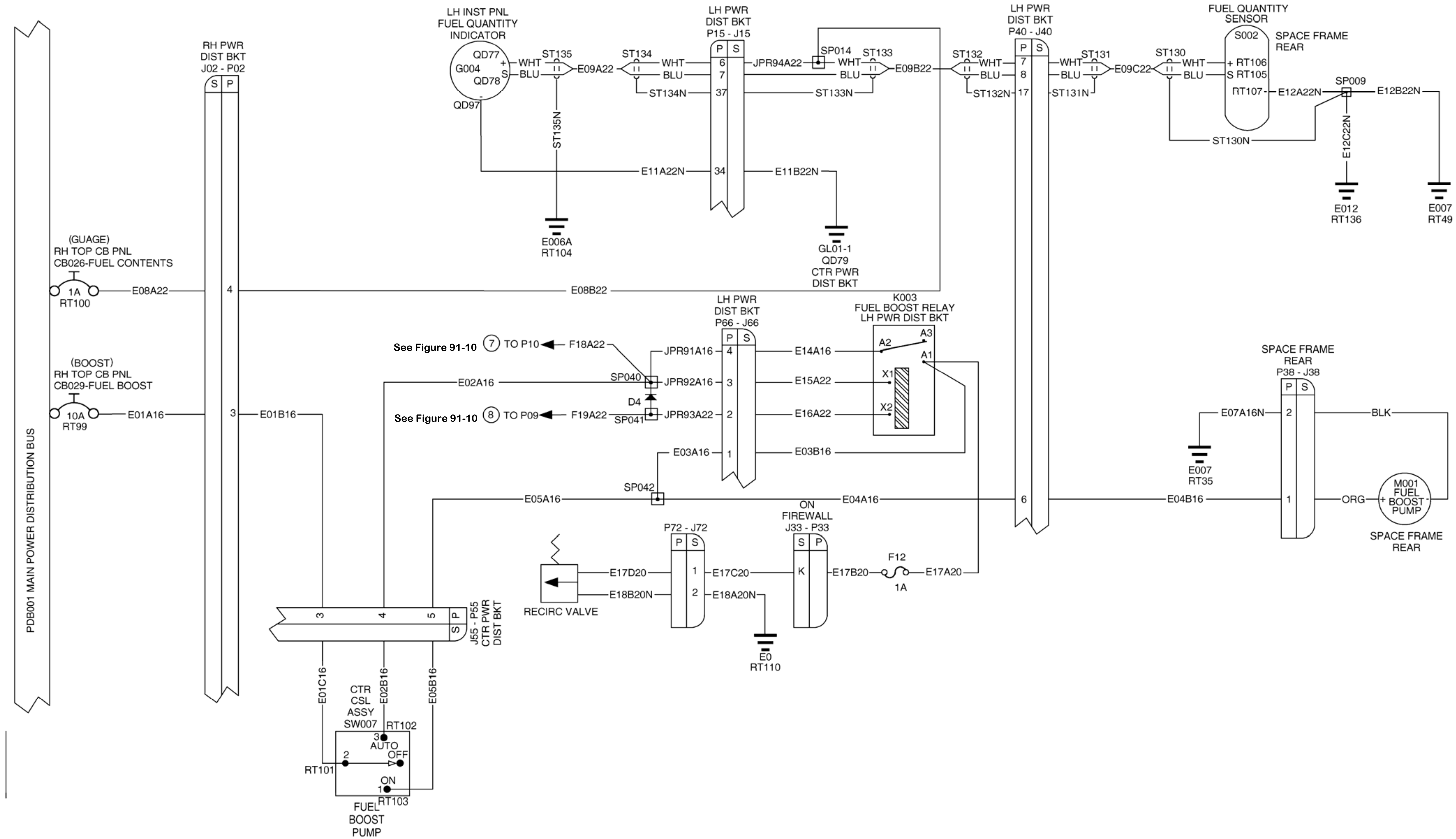

Figure 91-8 Fuel System

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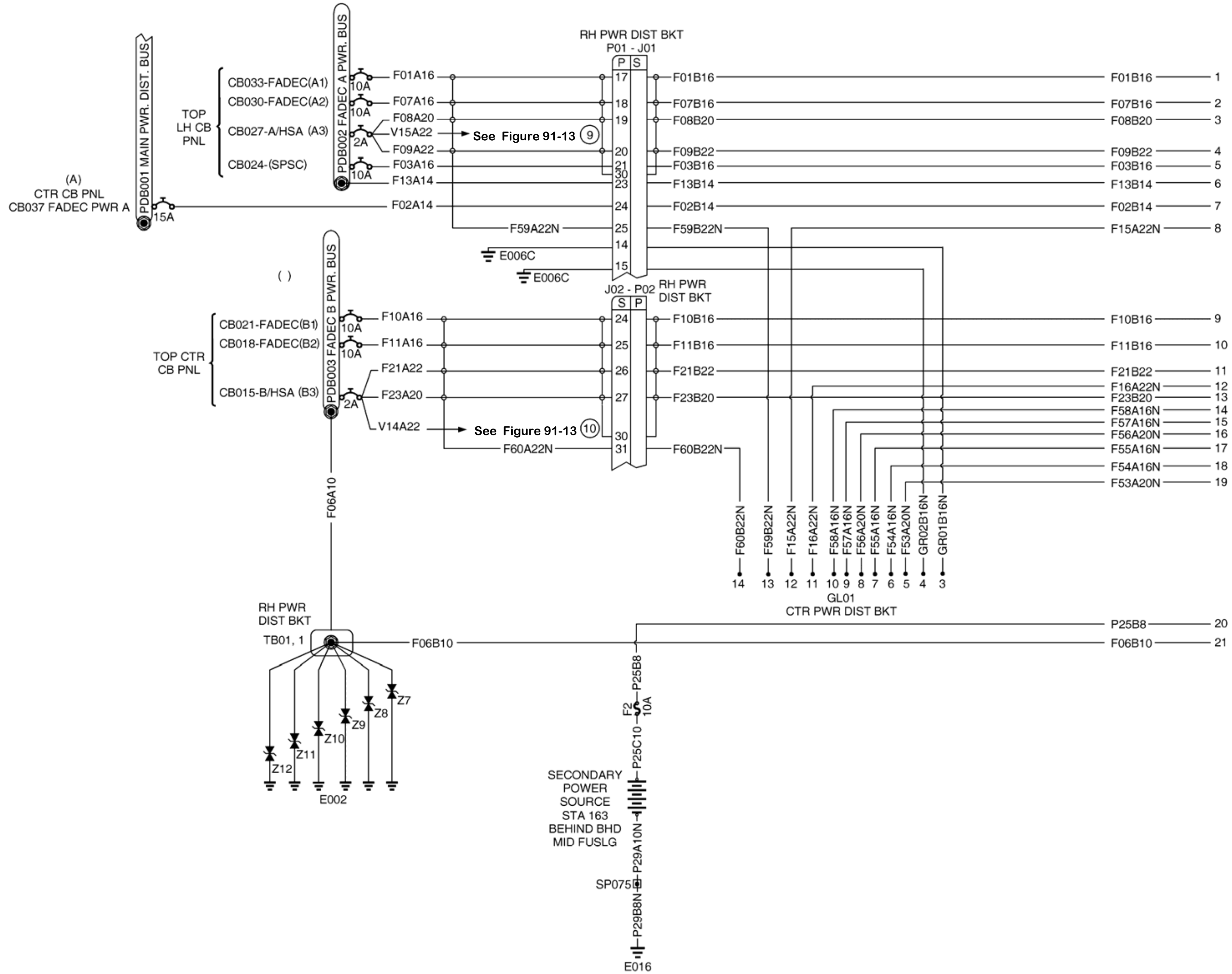

Figure 91-9 FADEC System Cabin Harness (1 OF 3)

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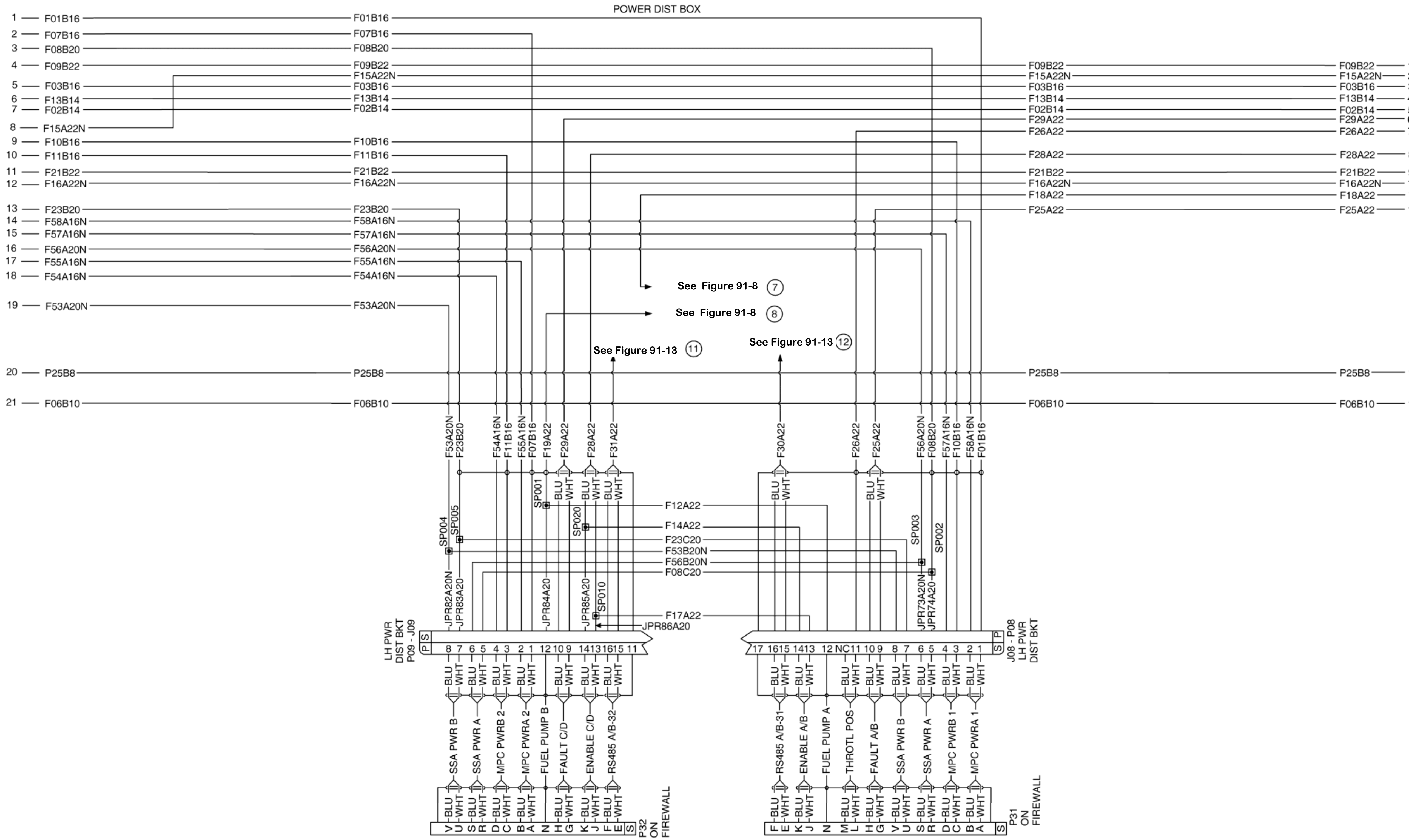

Figure 91-10 FADEC System Cabin Harness (2 OF 3)

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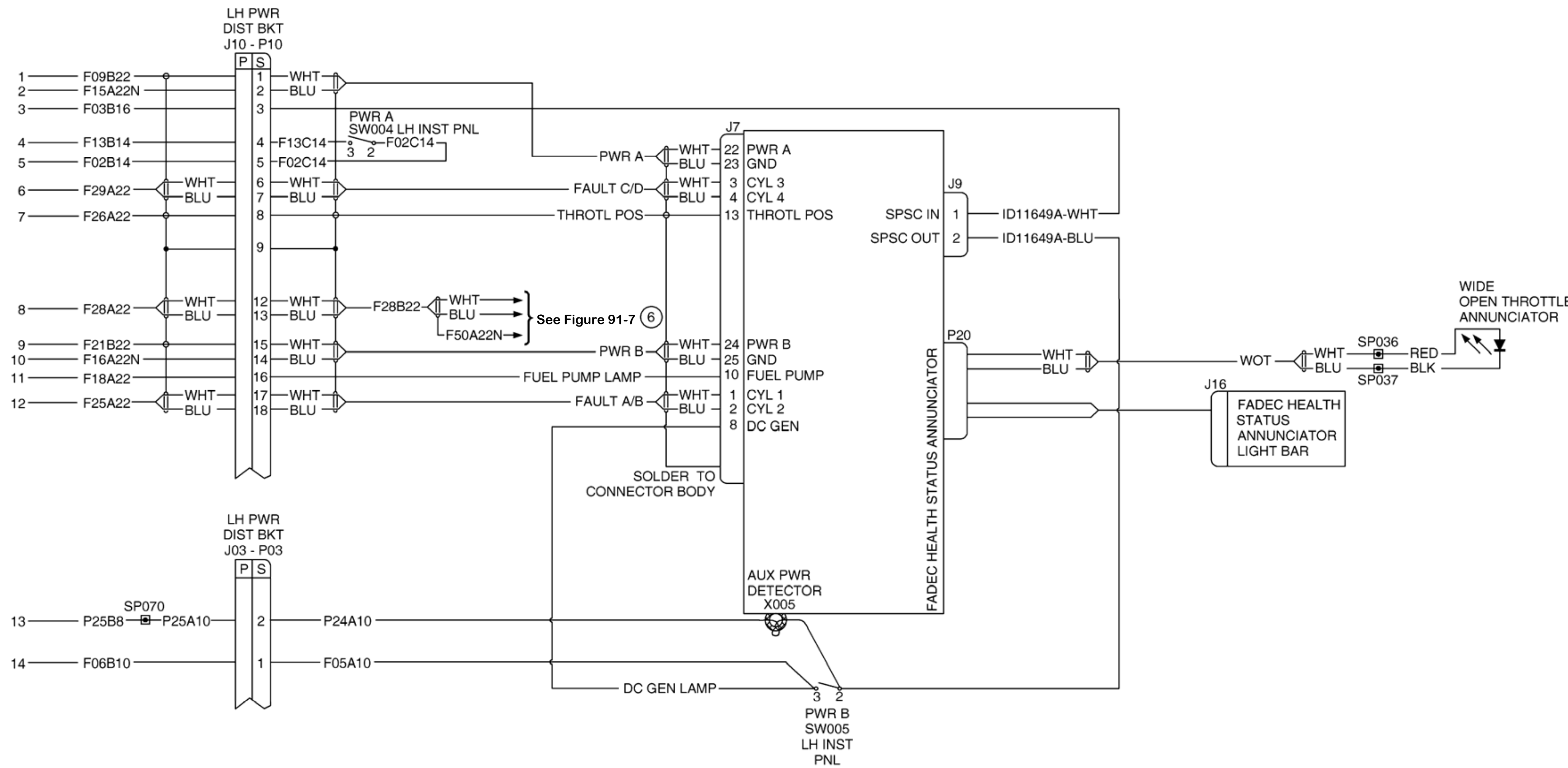

Figure 91-11 **FADEC System Cabin Harness (3 OF 3)**

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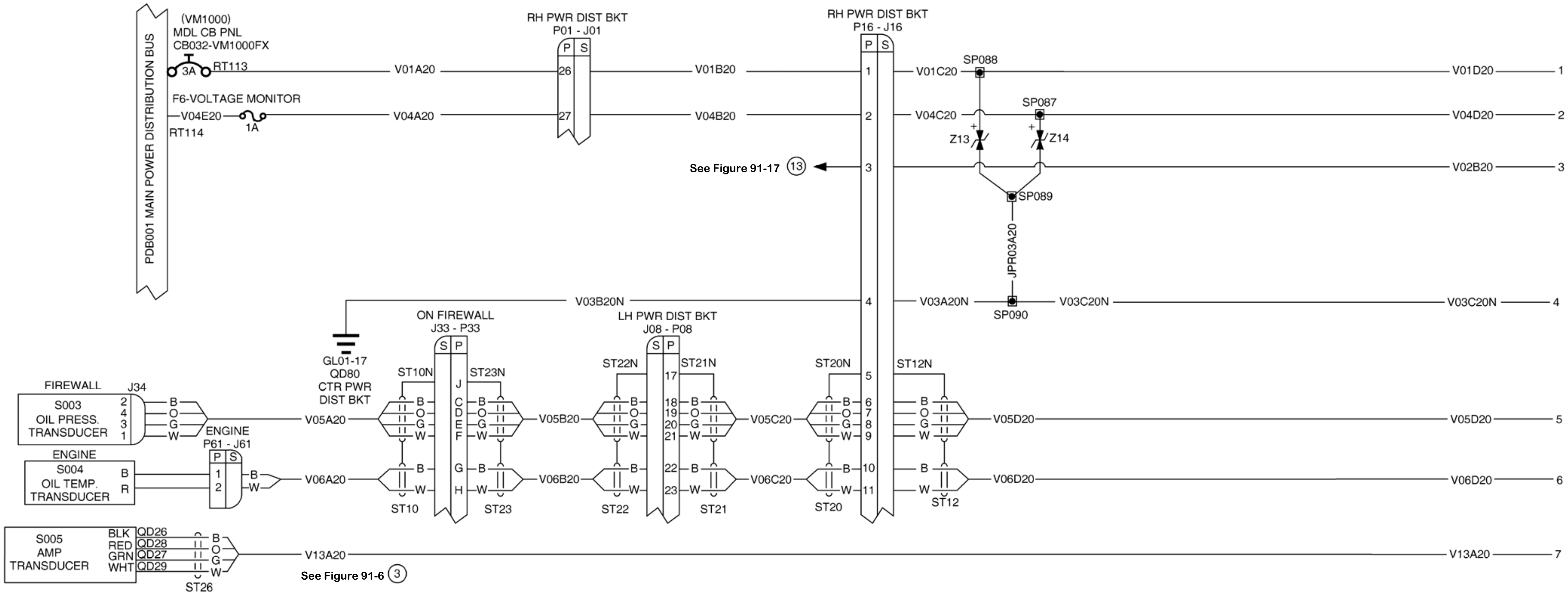

Figure 91-12 VM1000FX Engine Instrument System (1 OF 2)

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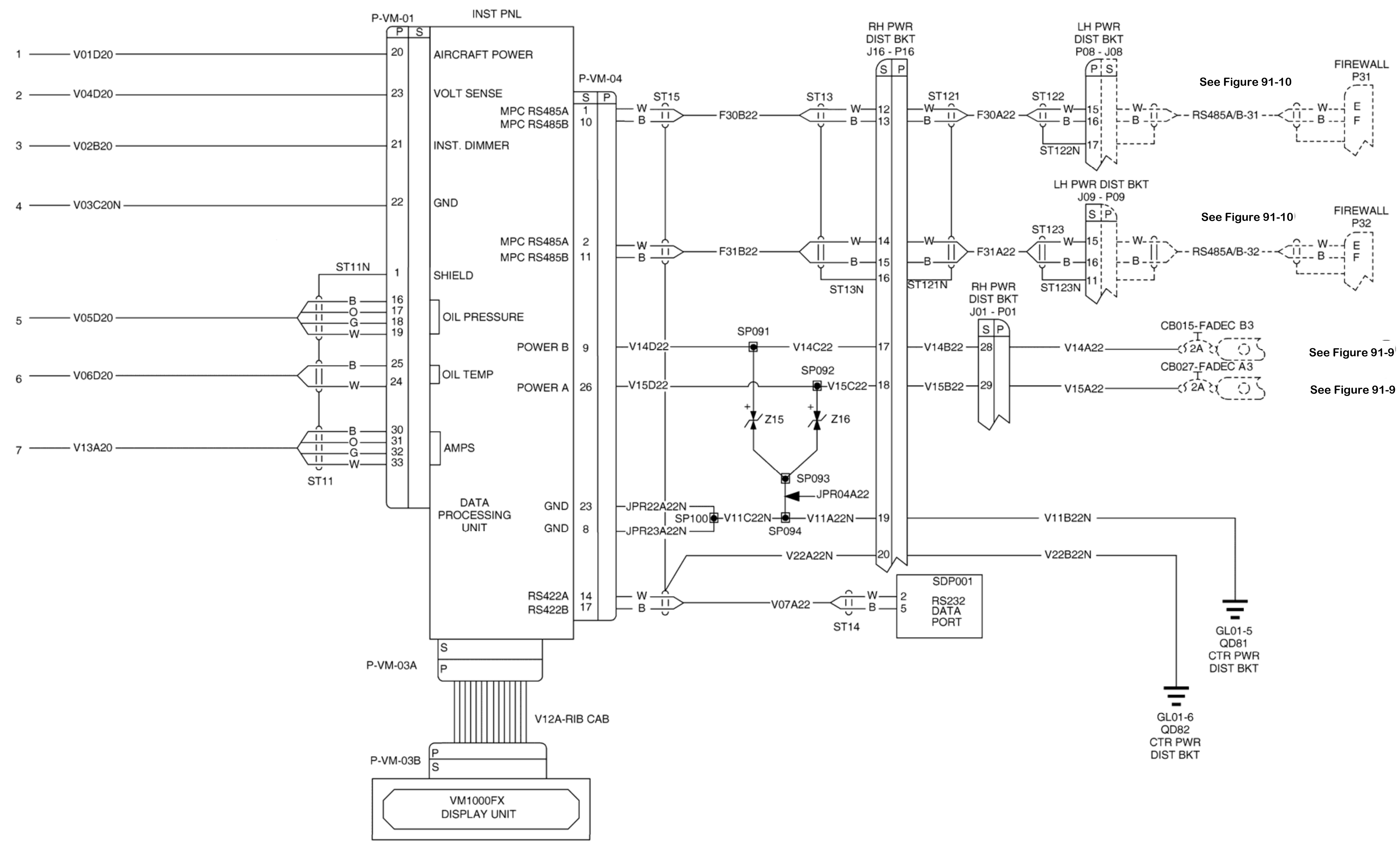

Figure 91-13 VM1000FX Engine Instrument System (2 of 2)

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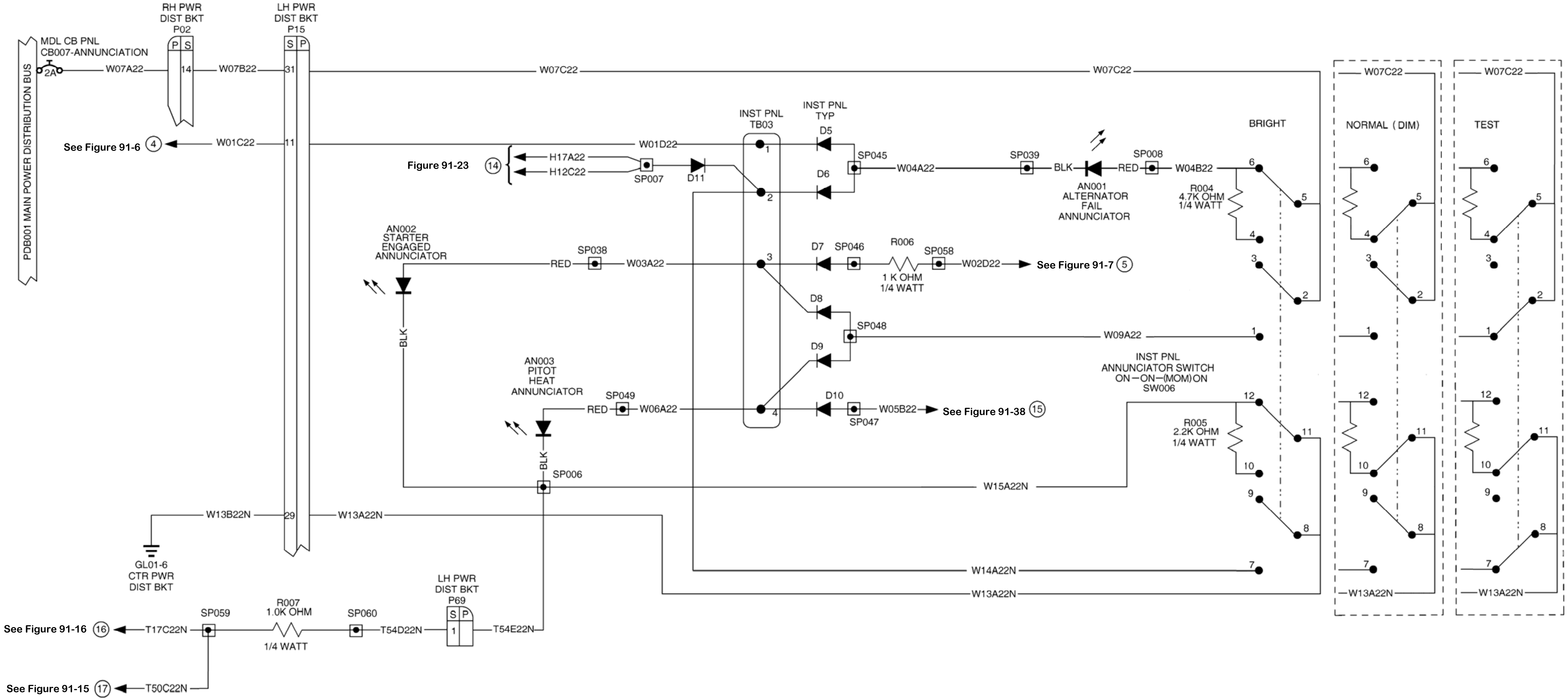

Figure 91-14 Annunciator System

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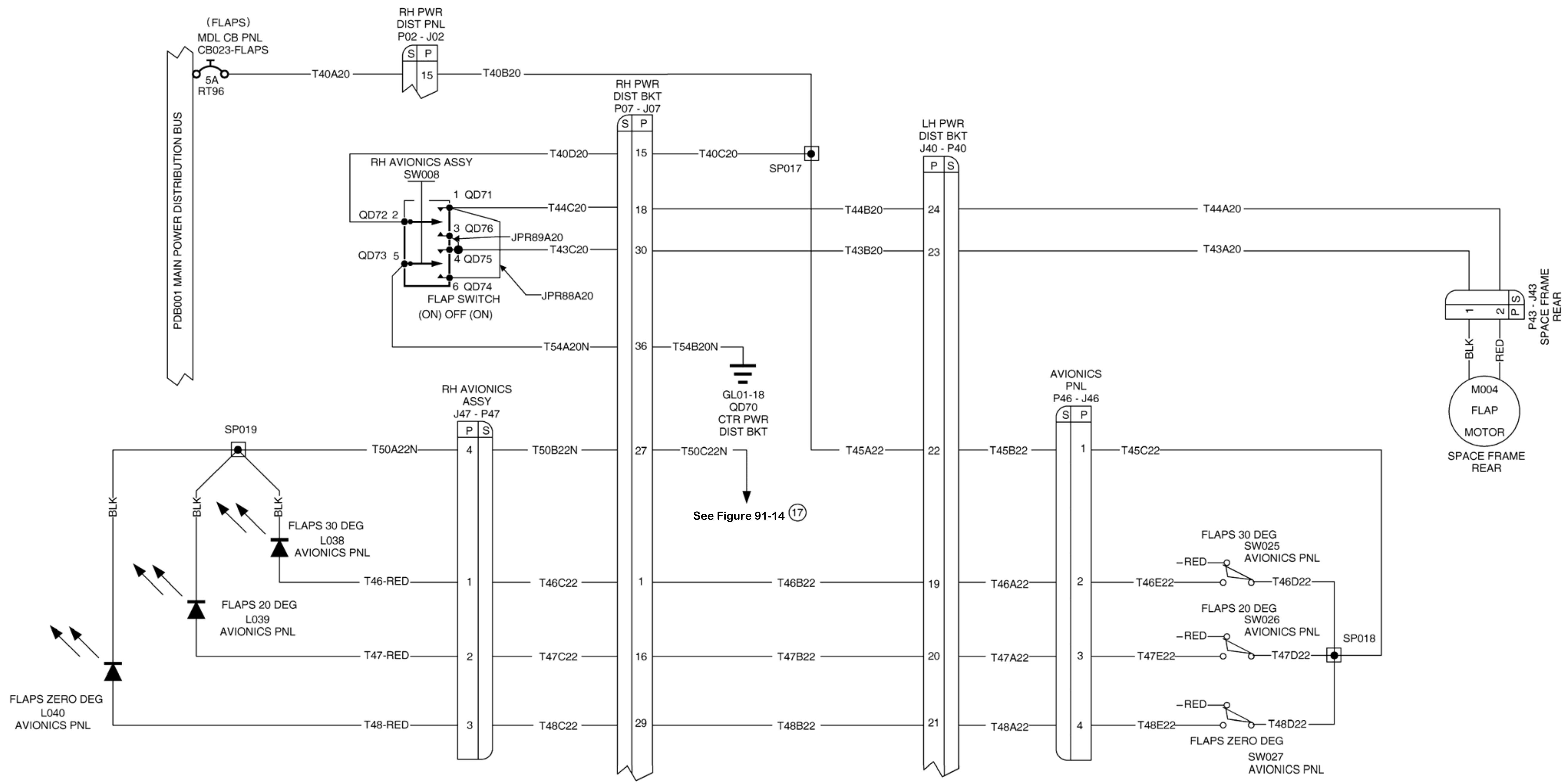

Figure 91-15 Flap System

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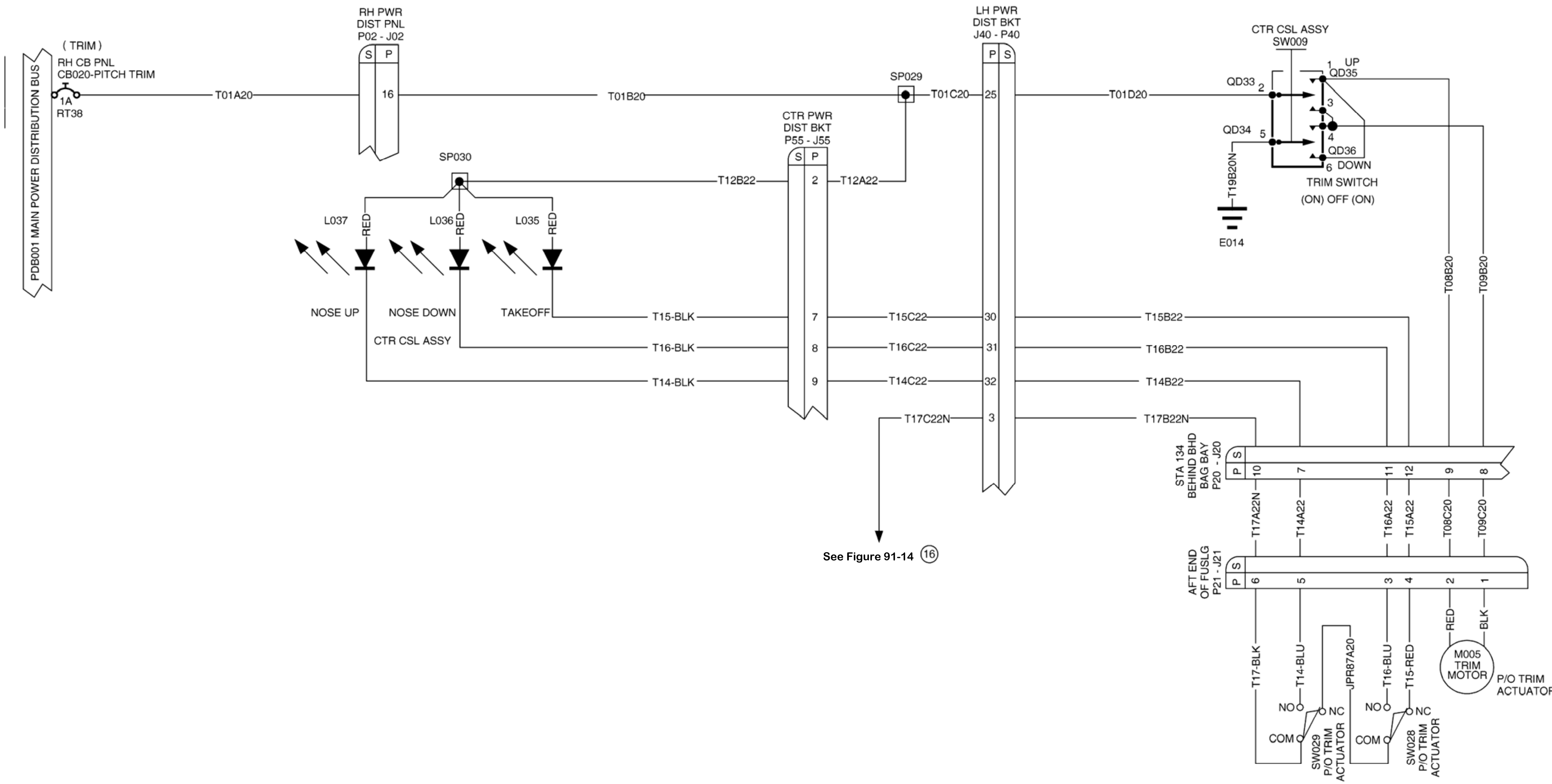

Figure 91-16 Trim System

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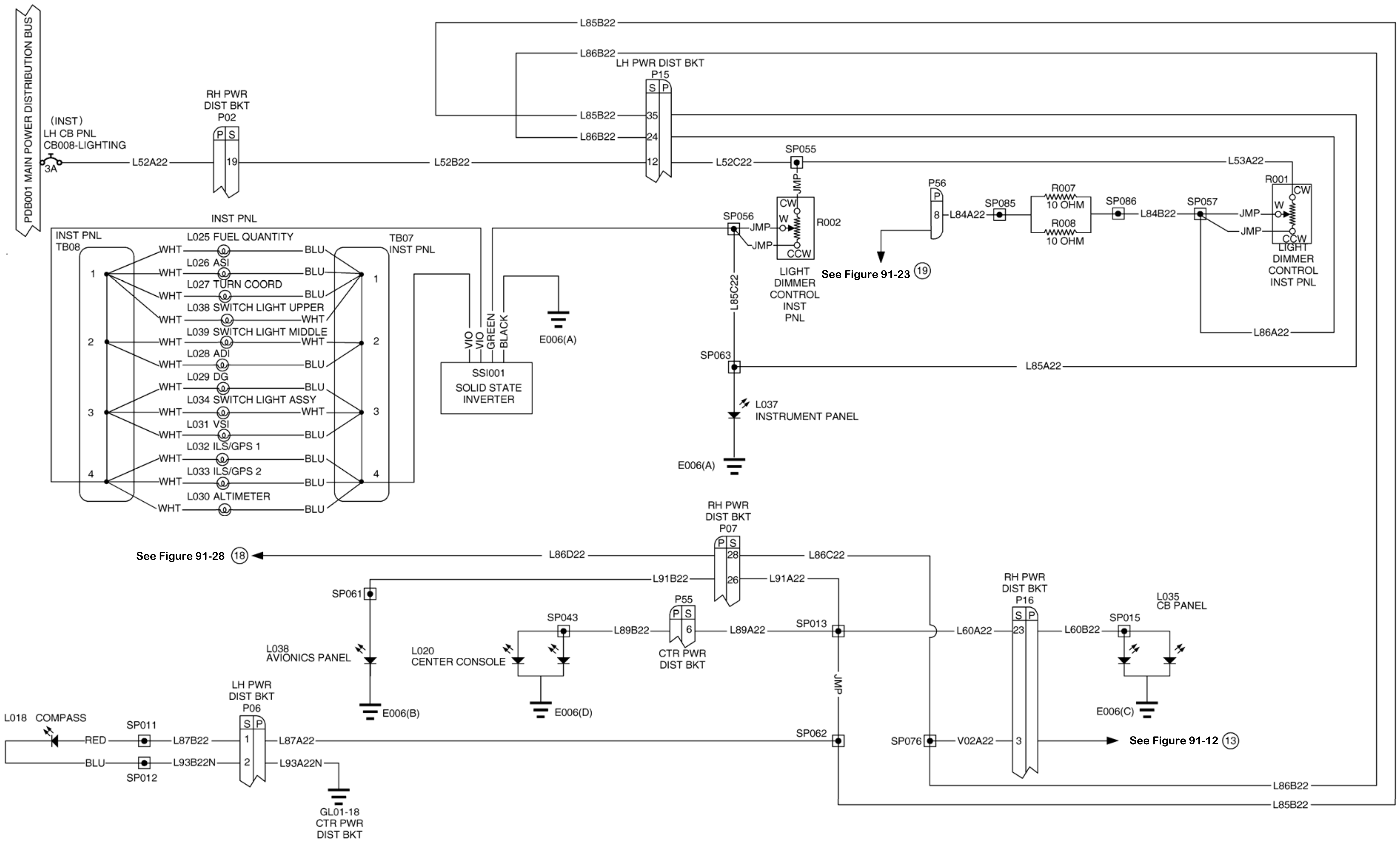

Figure 91-17 Interior Lighting

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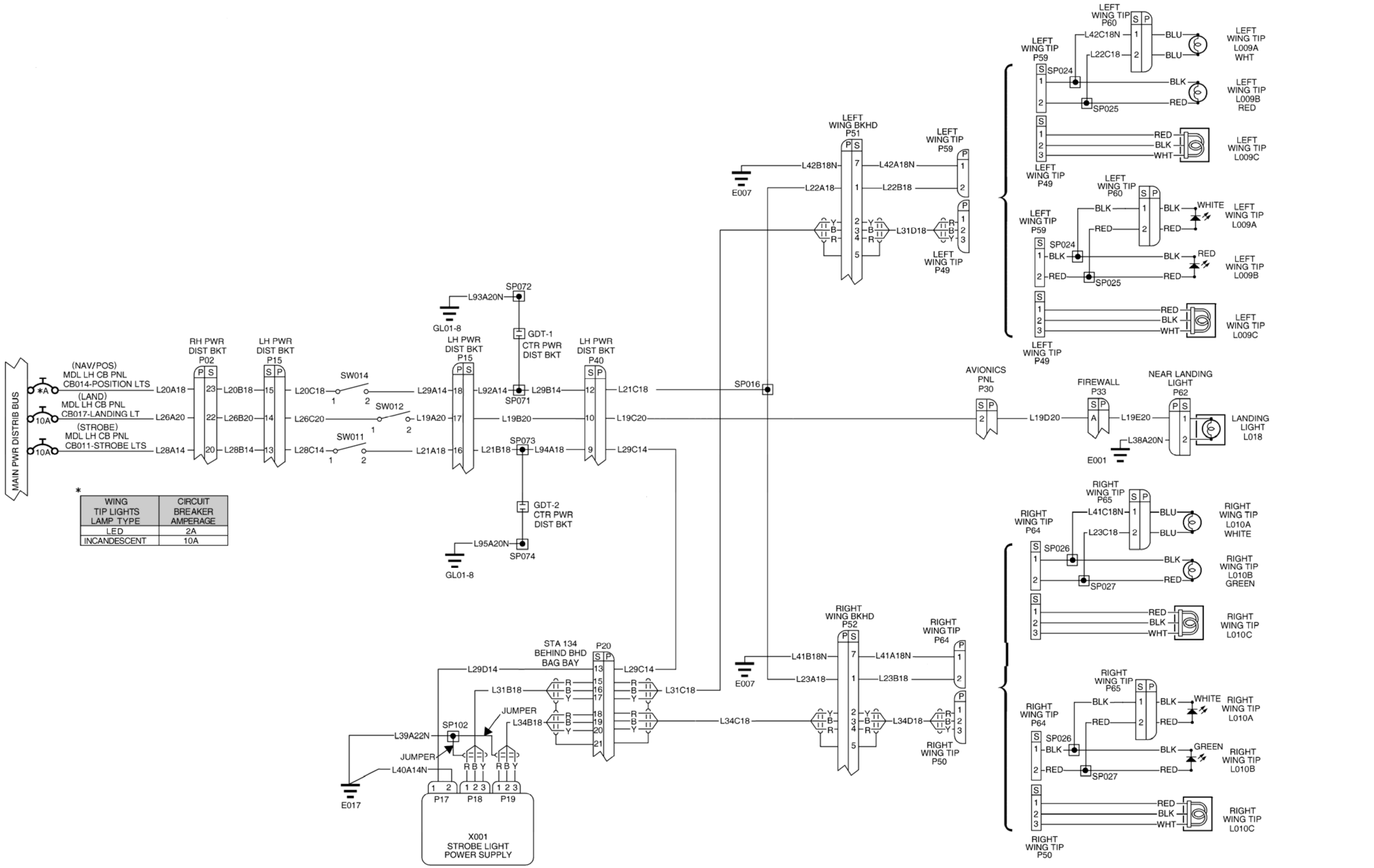

*

WING TIP LIGHTS LAMP TYPE	CIRCUIT BREAKER AMPERAGE
LED	2A
INCANDESCENT	10A

Figure 91-18 Exterior Lighting

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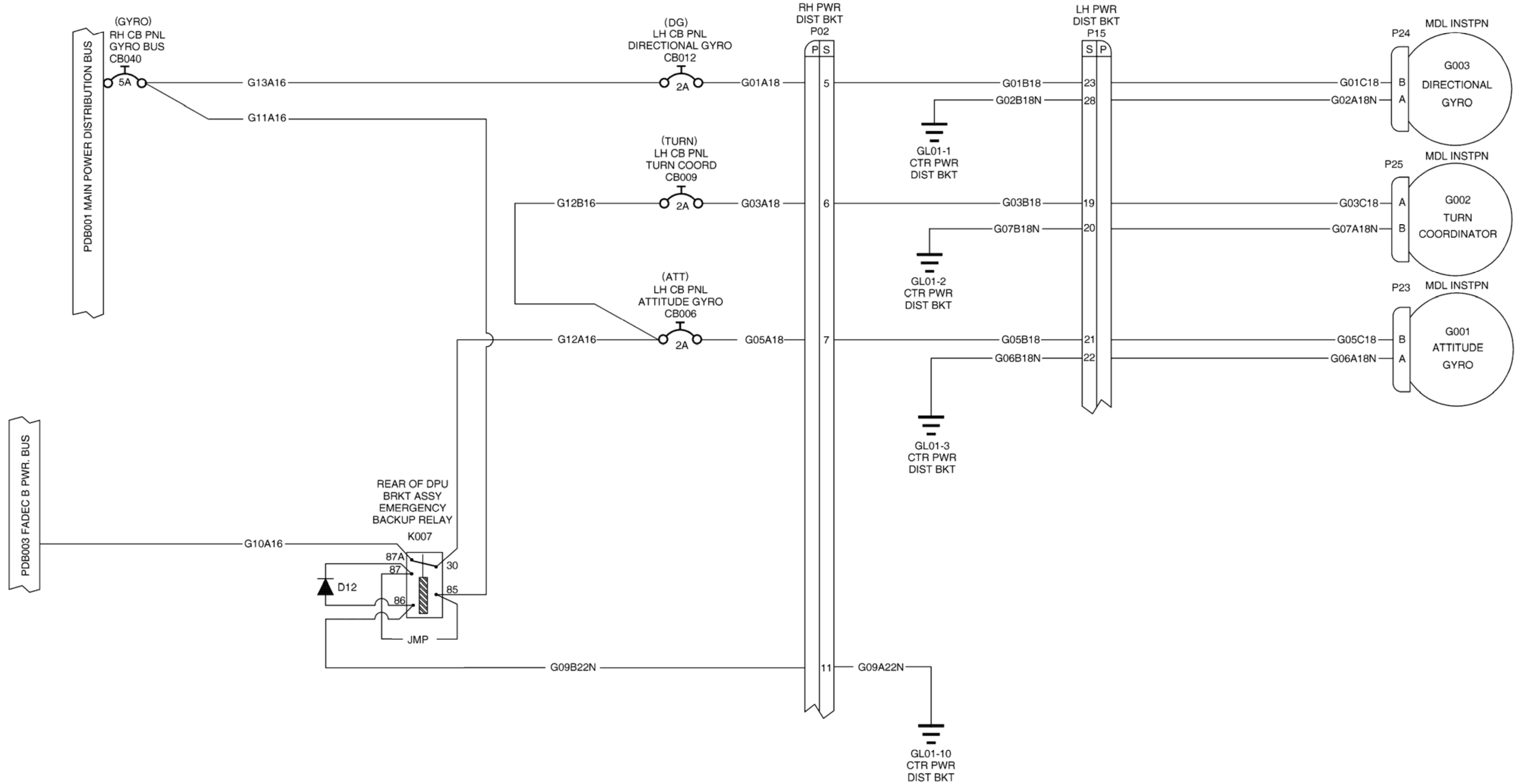

Figure 91-19 Electric Gyro Instrument System

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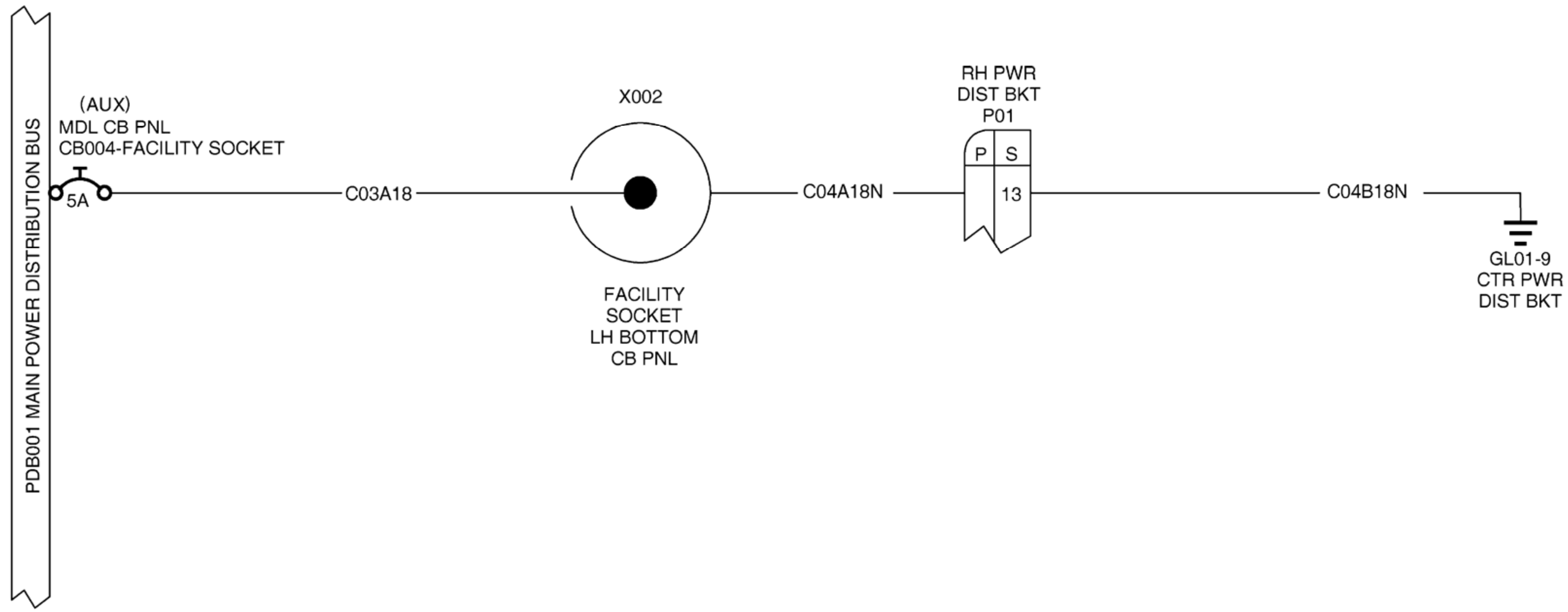

Figure 91-20 Facility Socket

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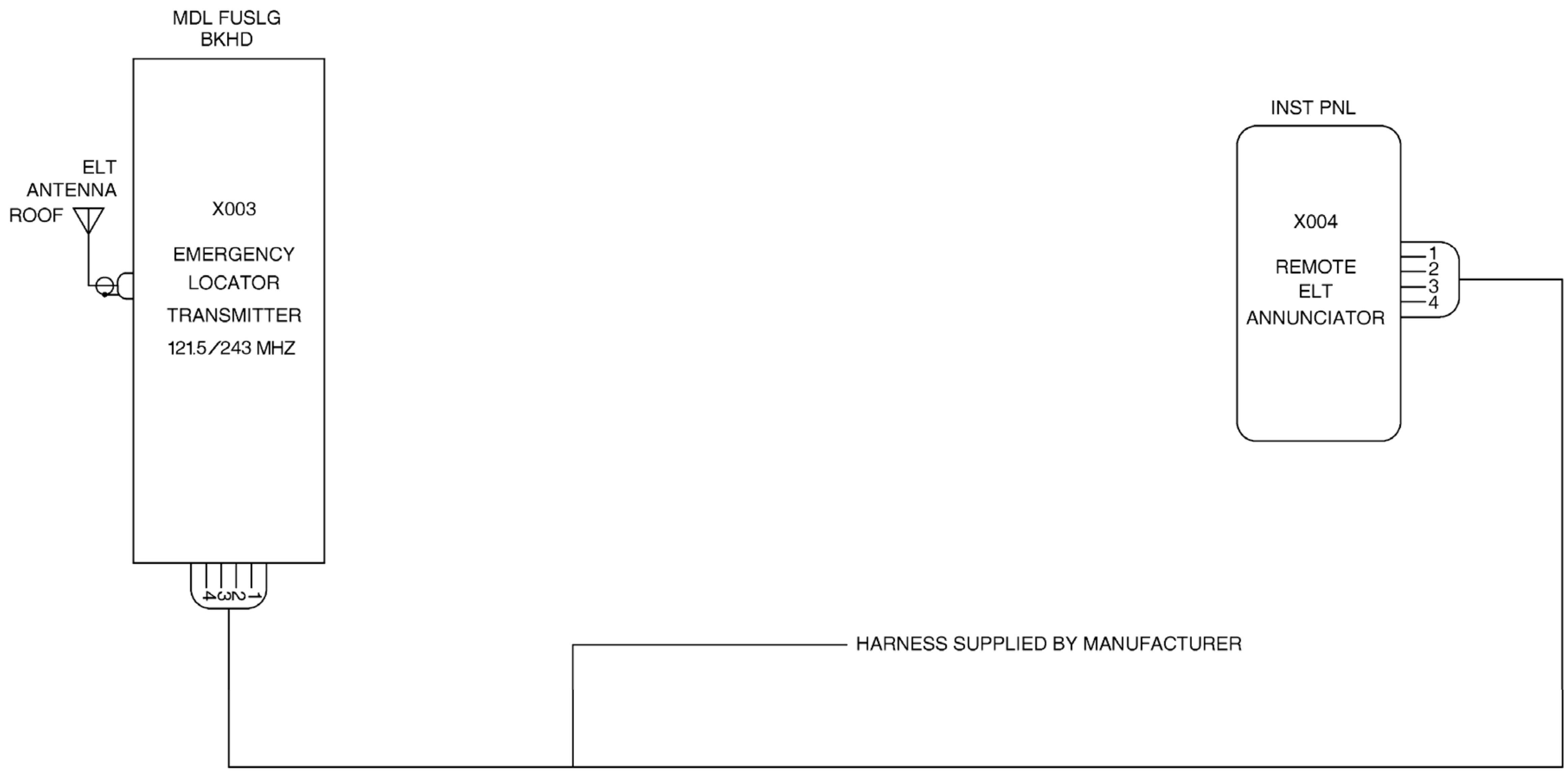

Figure 91-21 121.5/243 Mhz ELT Interconnect

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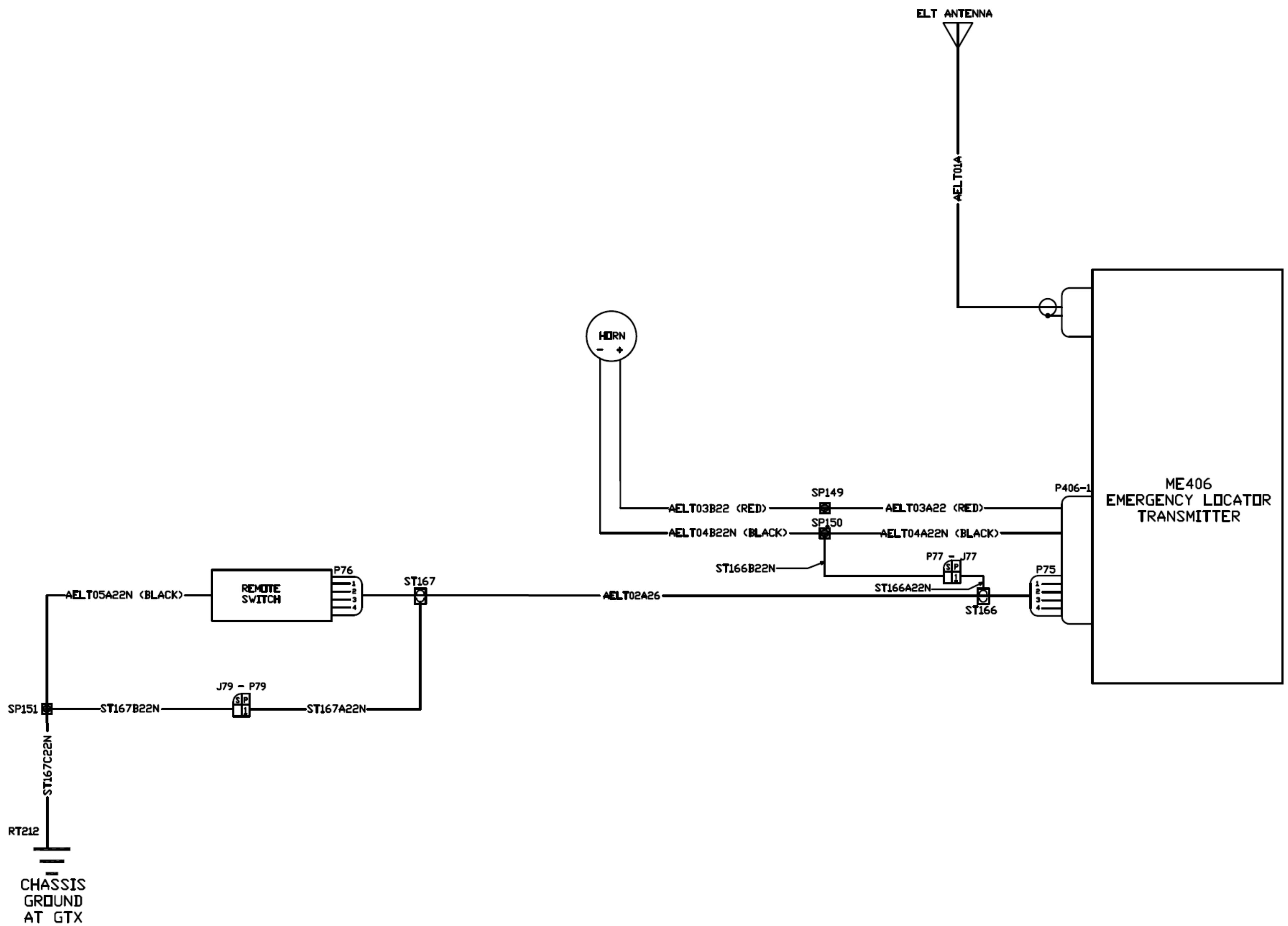

Figure 91-22 **406/121.5 Mhz ELT Interconnect**

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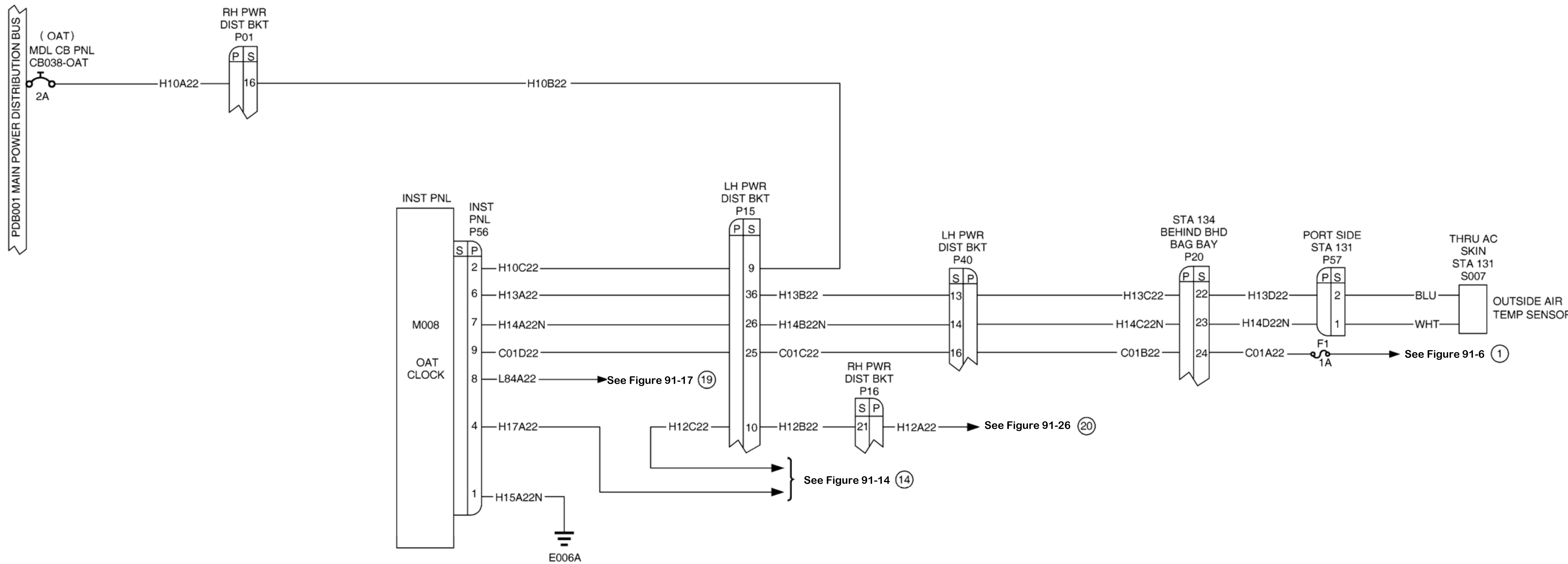

Figure 91-23 Outside Air Temperature Sensor and Clock System

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RADIO OPTION CB RATINGS			
MODEL	COM CB	NAV CB	ELIGIBLE STACK POSITIONS
GARMIN GNS 430	10	5	1, 2
GARMIN GNS 530	10	5	1
GARMIN SL30	5	2	1,2
GARMIN SL40	5	2	1,2

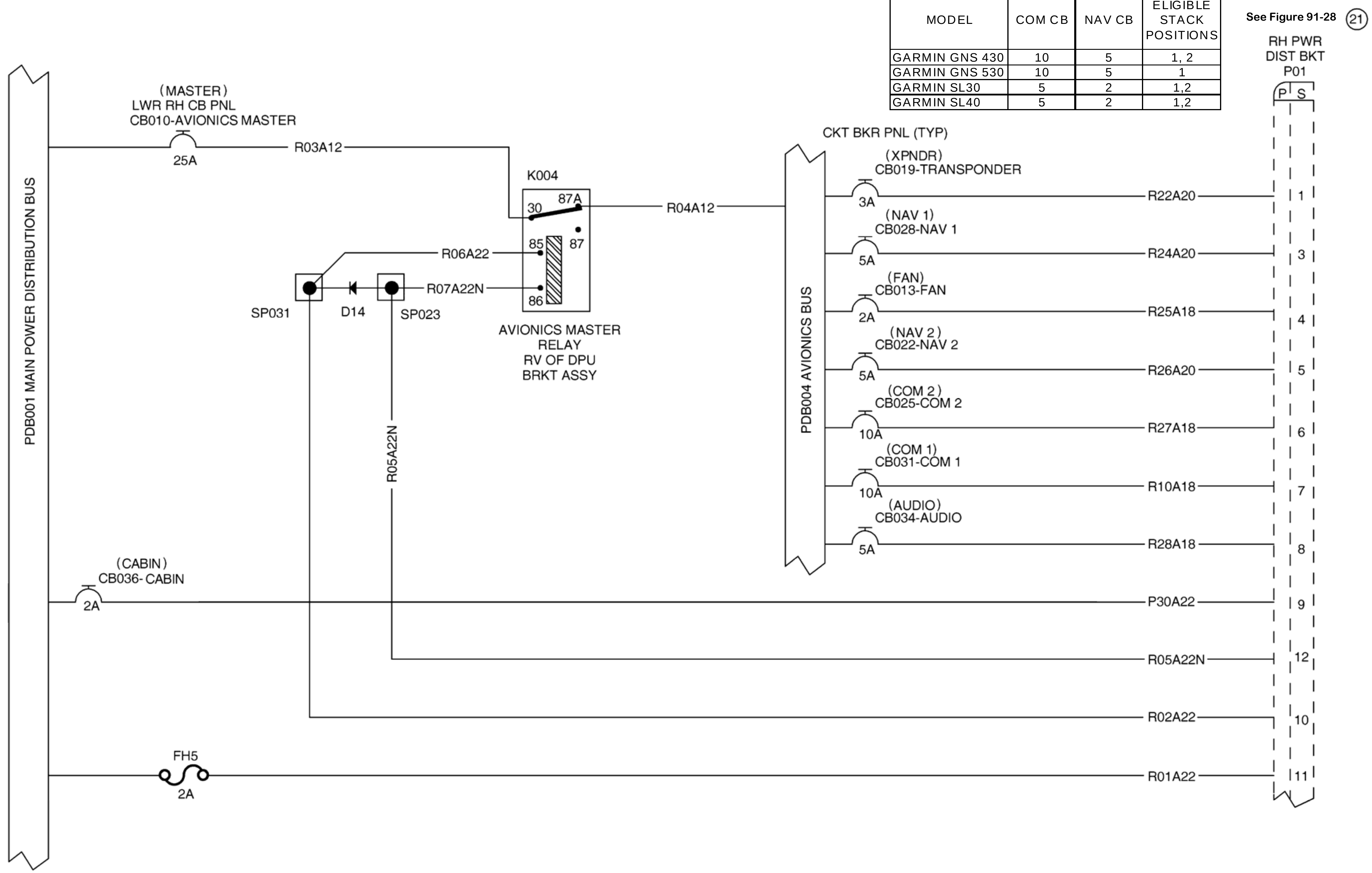

Figure 91-24 Avionics Power Distribution System

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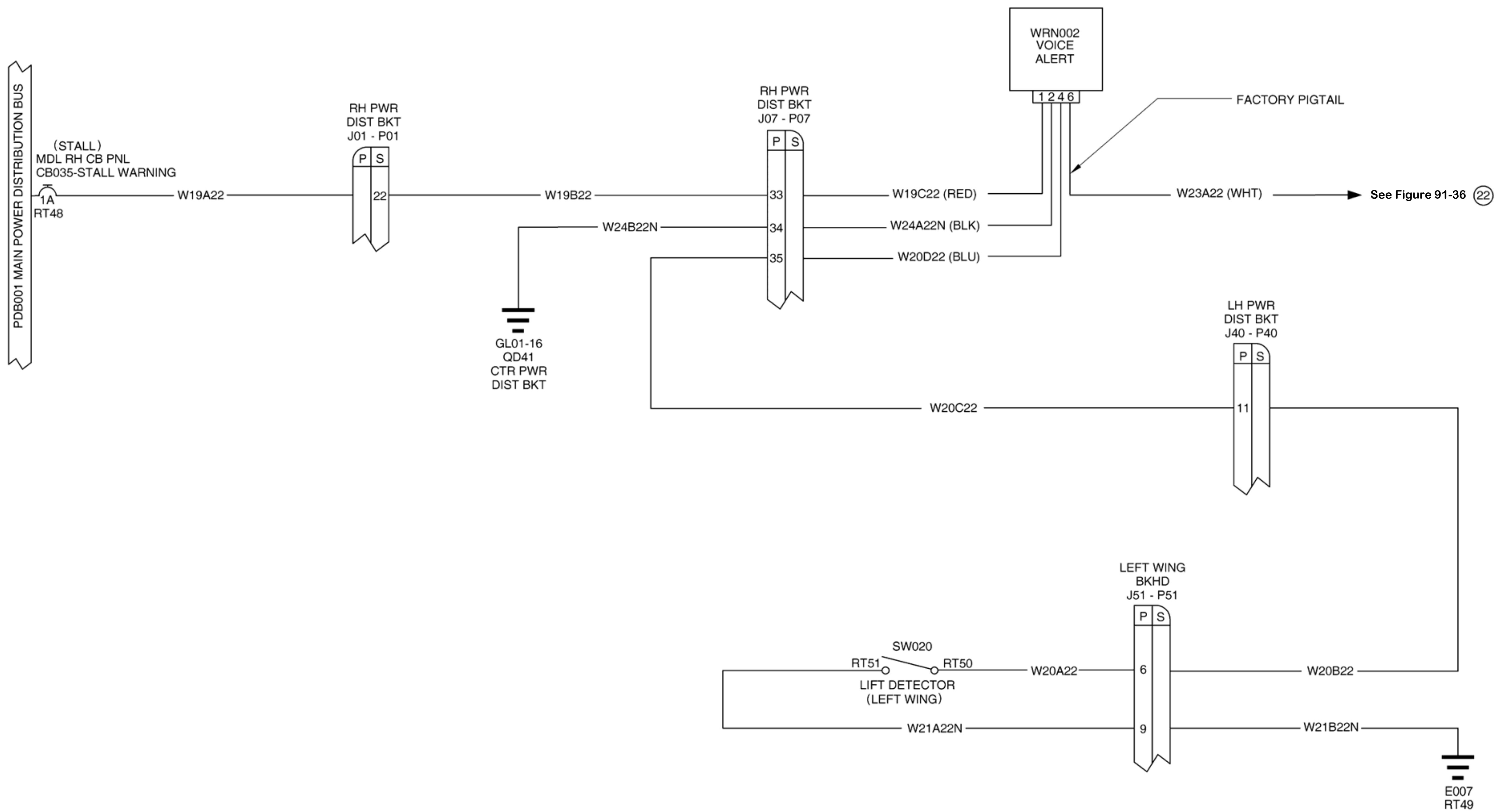

Figure 91-25 Stall Warning System

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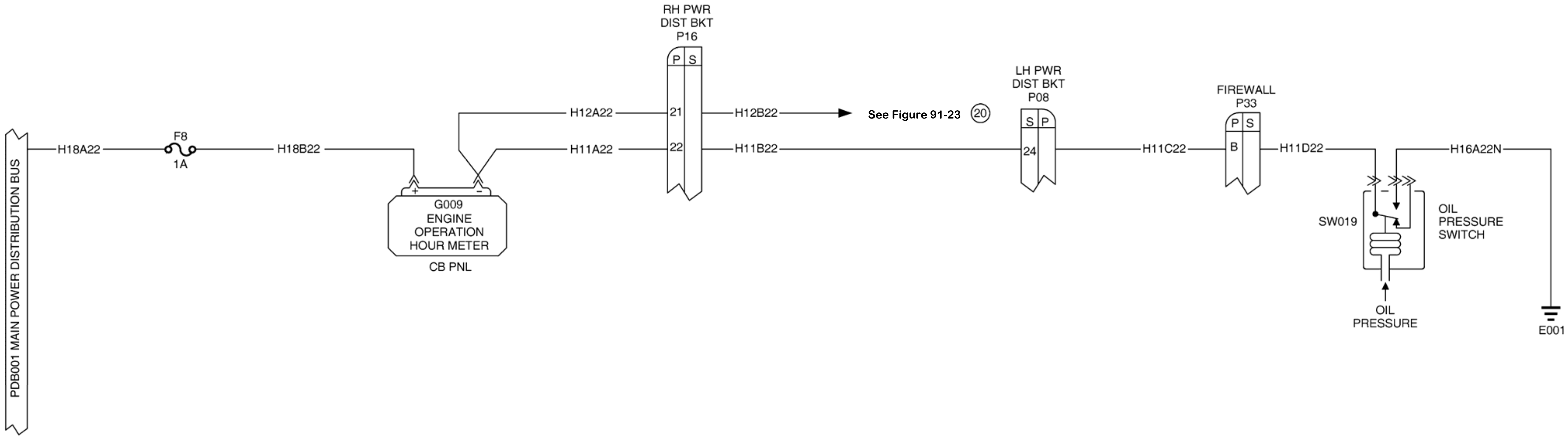

Figure 91-26 Hobbs Hour Meter

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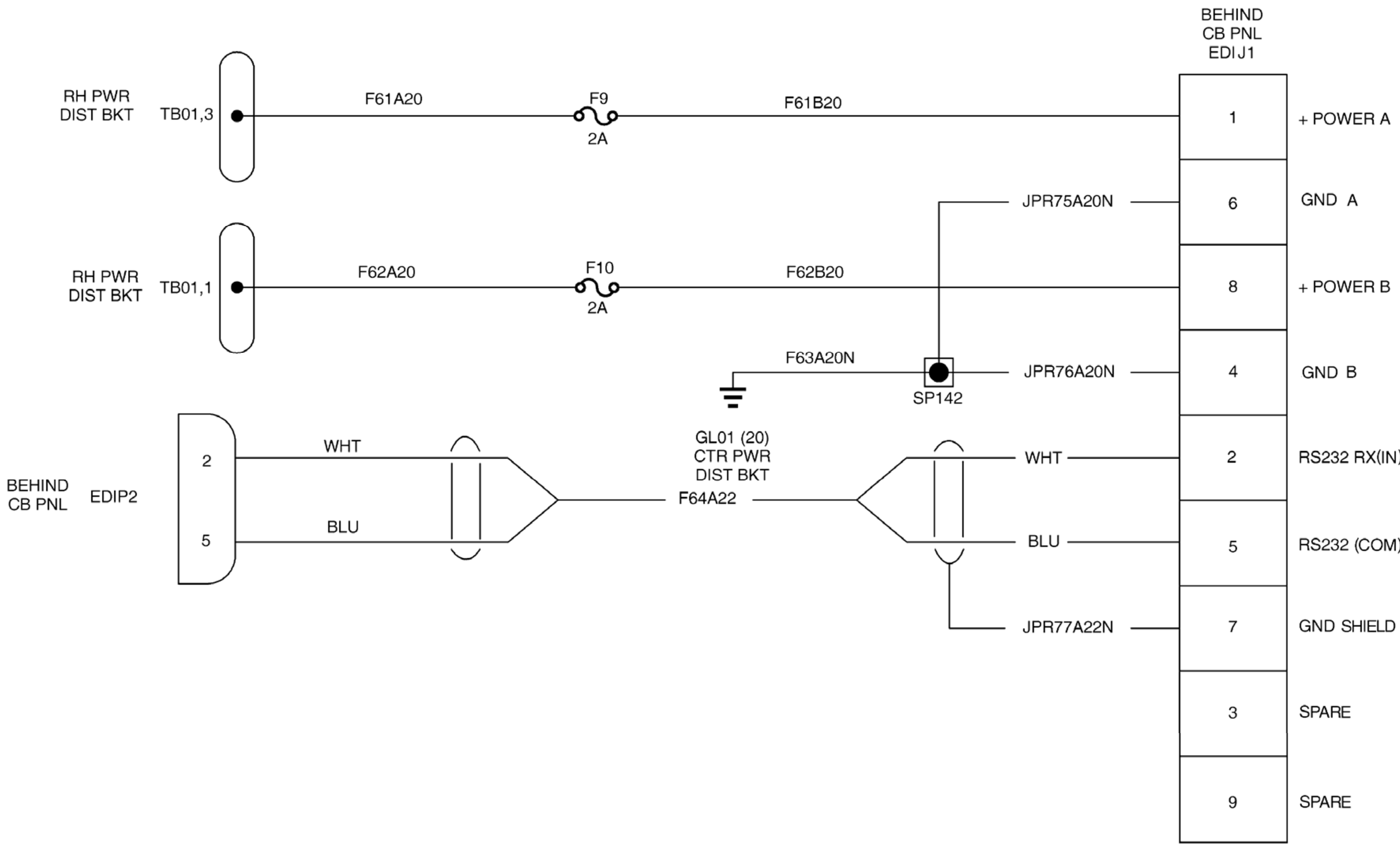

Figure 91-27 EDI-200 Interface Harness

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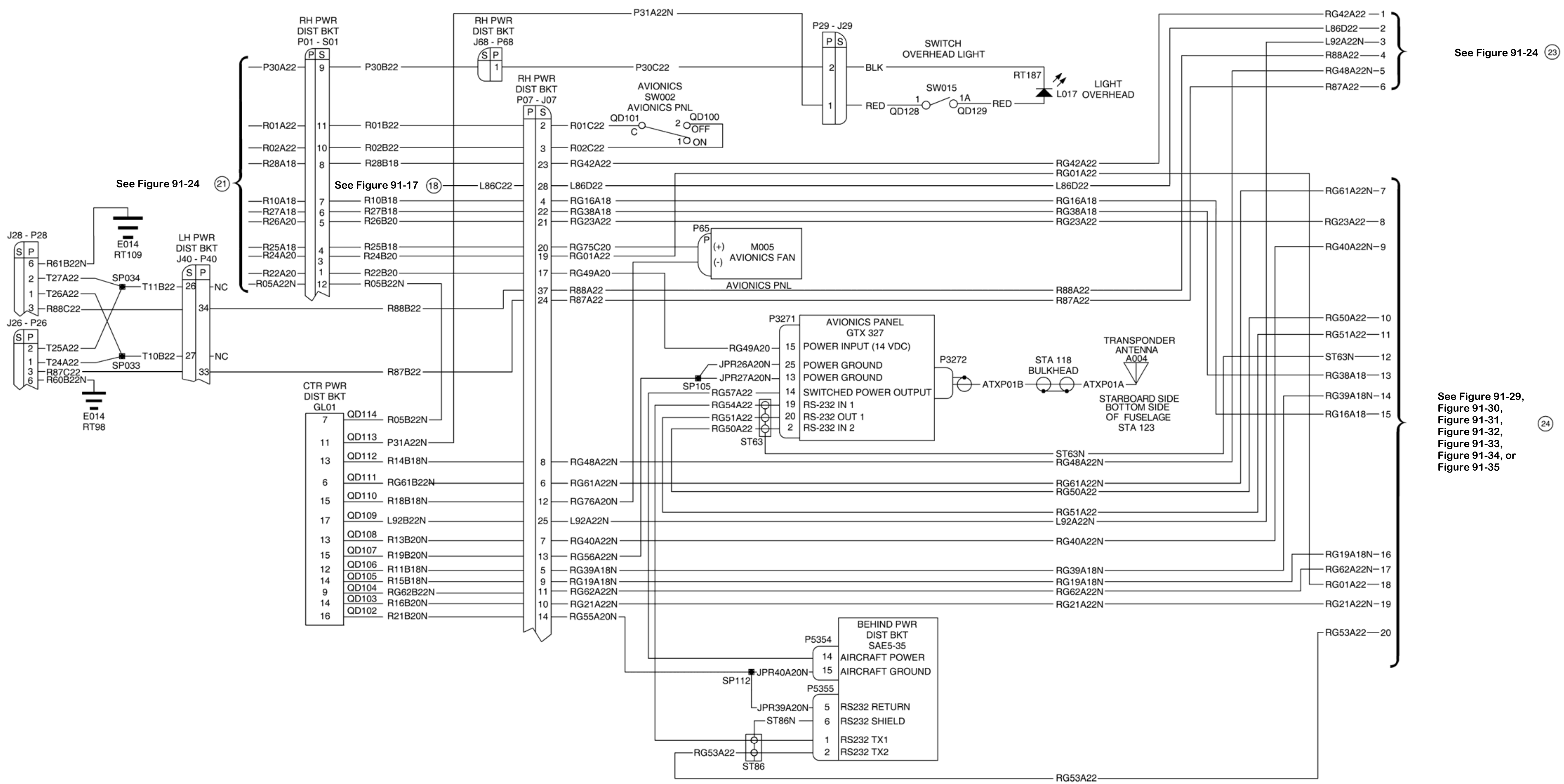

Figure 91-28 Avionics Stack

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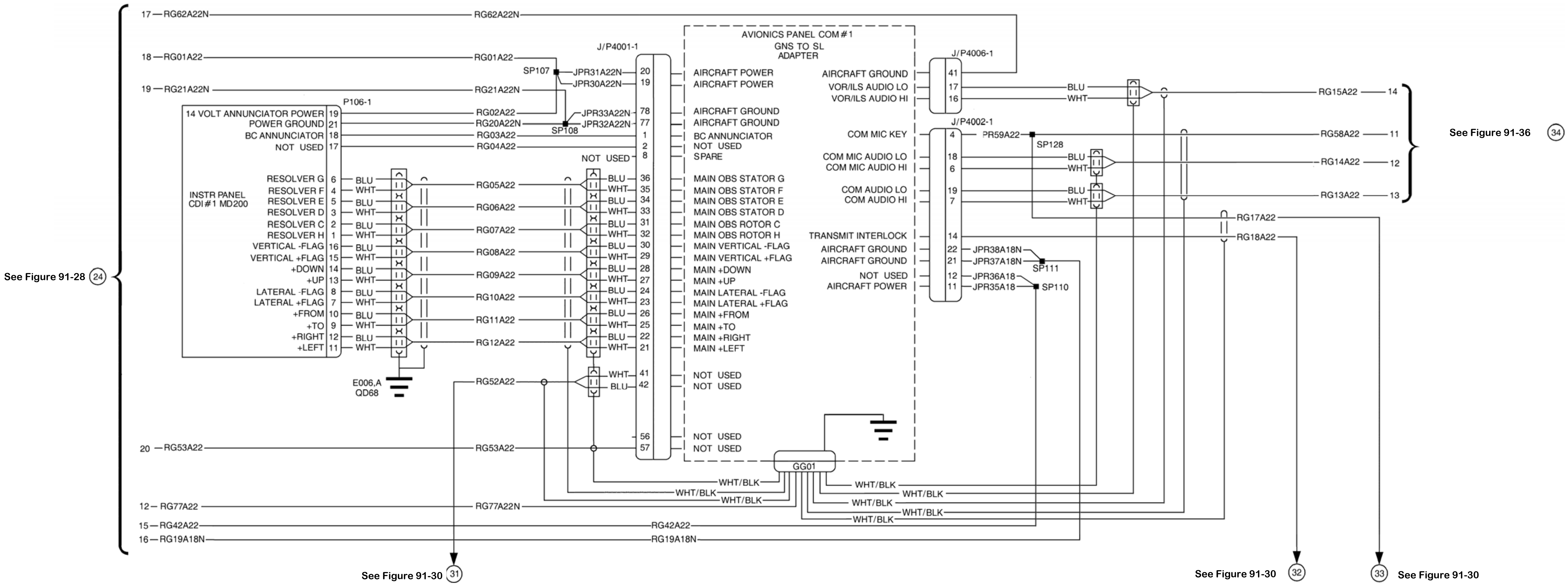

Figure 91-29 Avionics (Dual GNS System) 1 of 2

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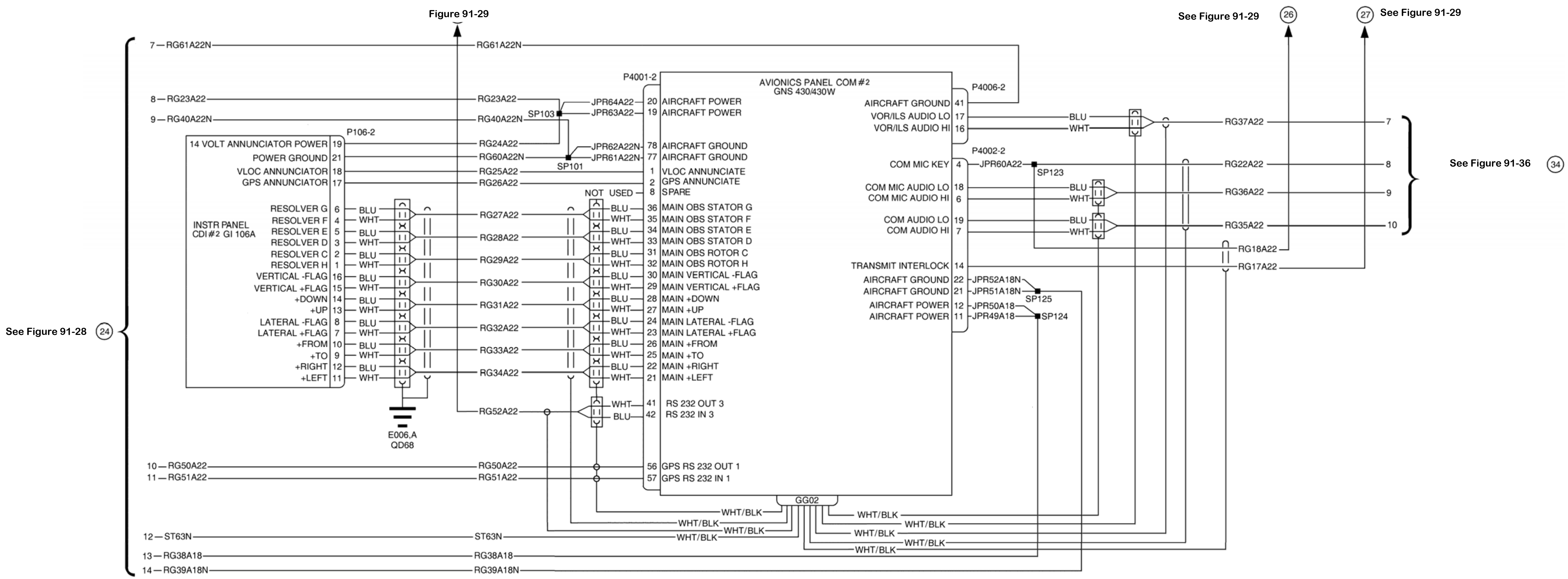

Figure 91-30 Avionics (Dual GNS System) 2 of 2

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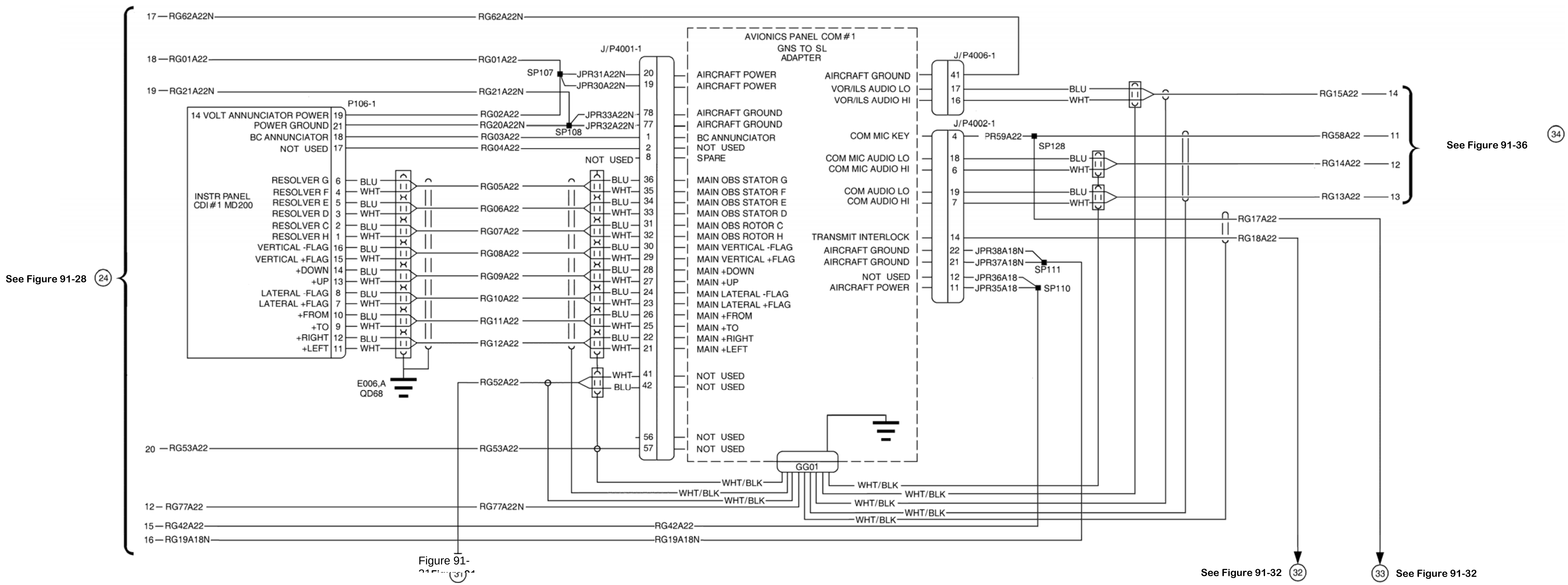

Figure 91-31 Avionics (Single GNS/Single SL System)1 of 2

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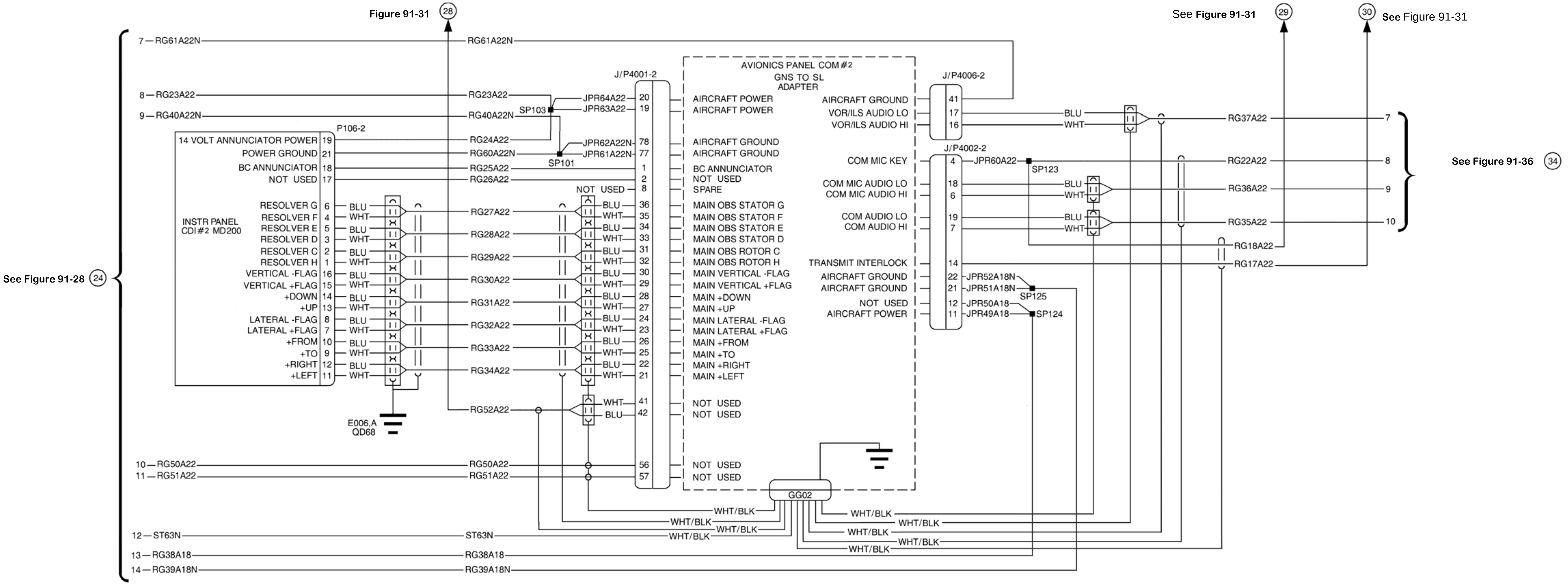

Figure 91-32 Avionics (Single GNS/Single SL System) 2 of 2

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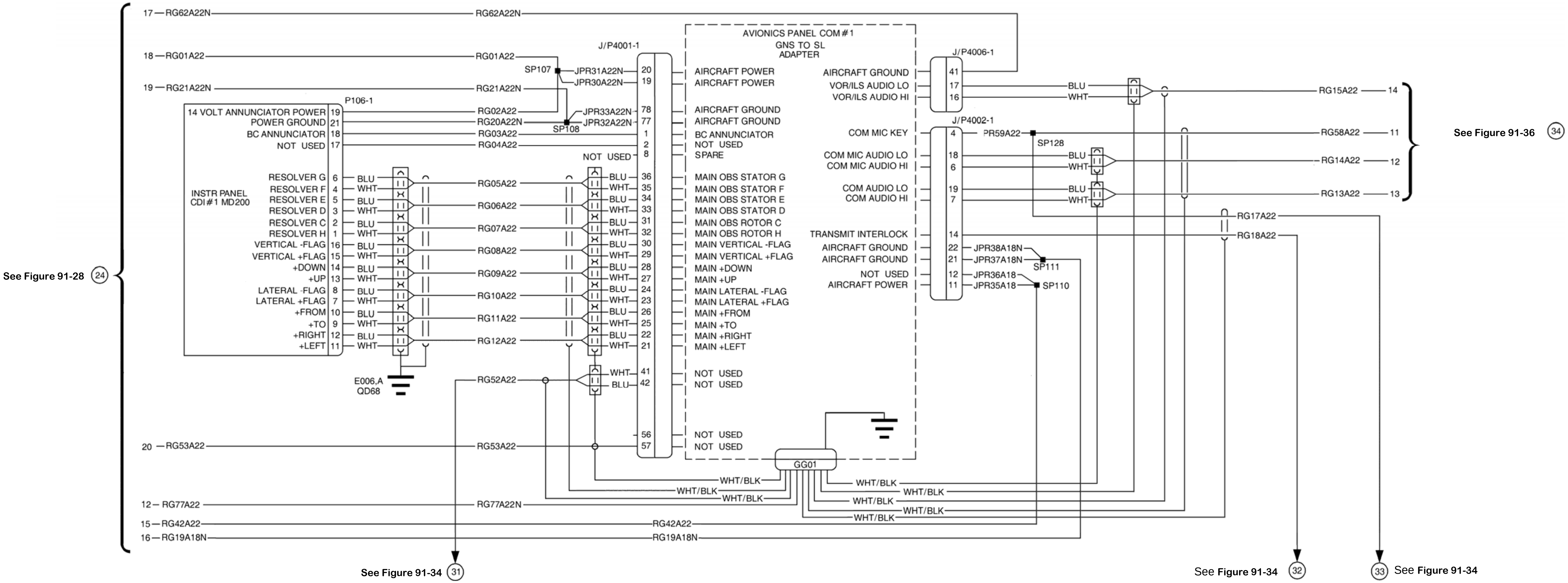

Figure 91-33 Avionics (Dual SL System) 1 of 2

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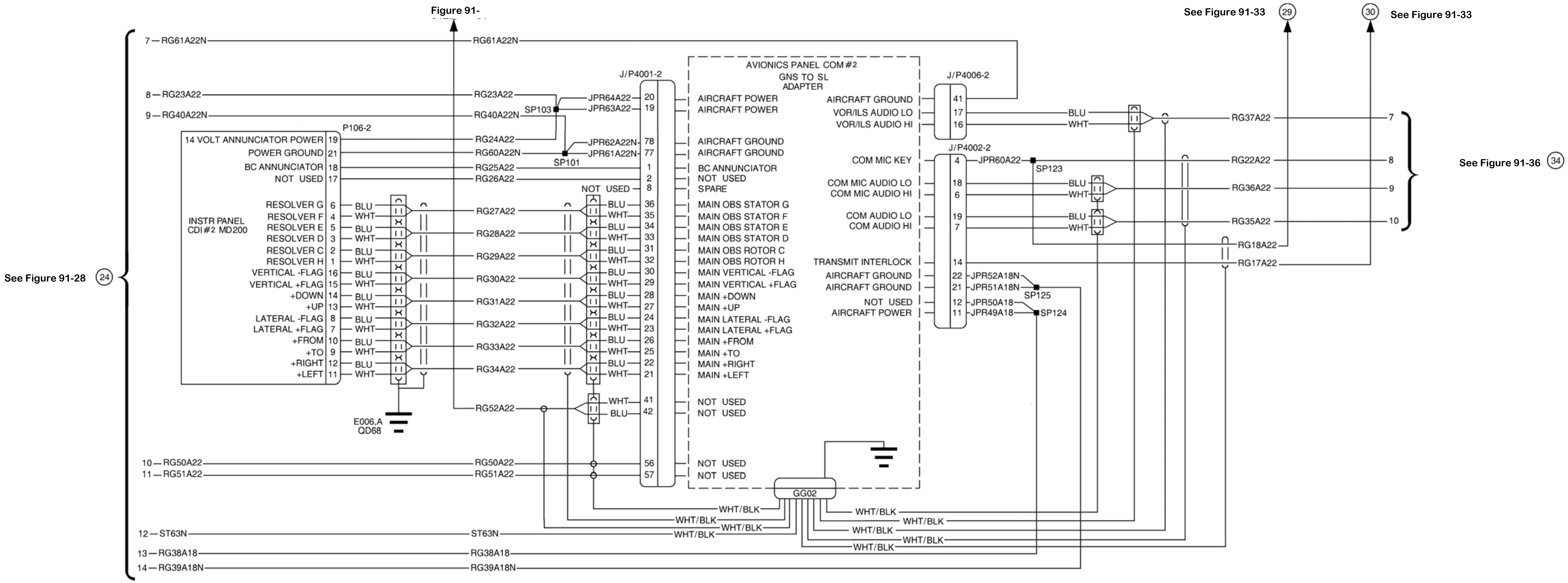

Figure 91-34 Avionics (Dual SL System) 2 of 2

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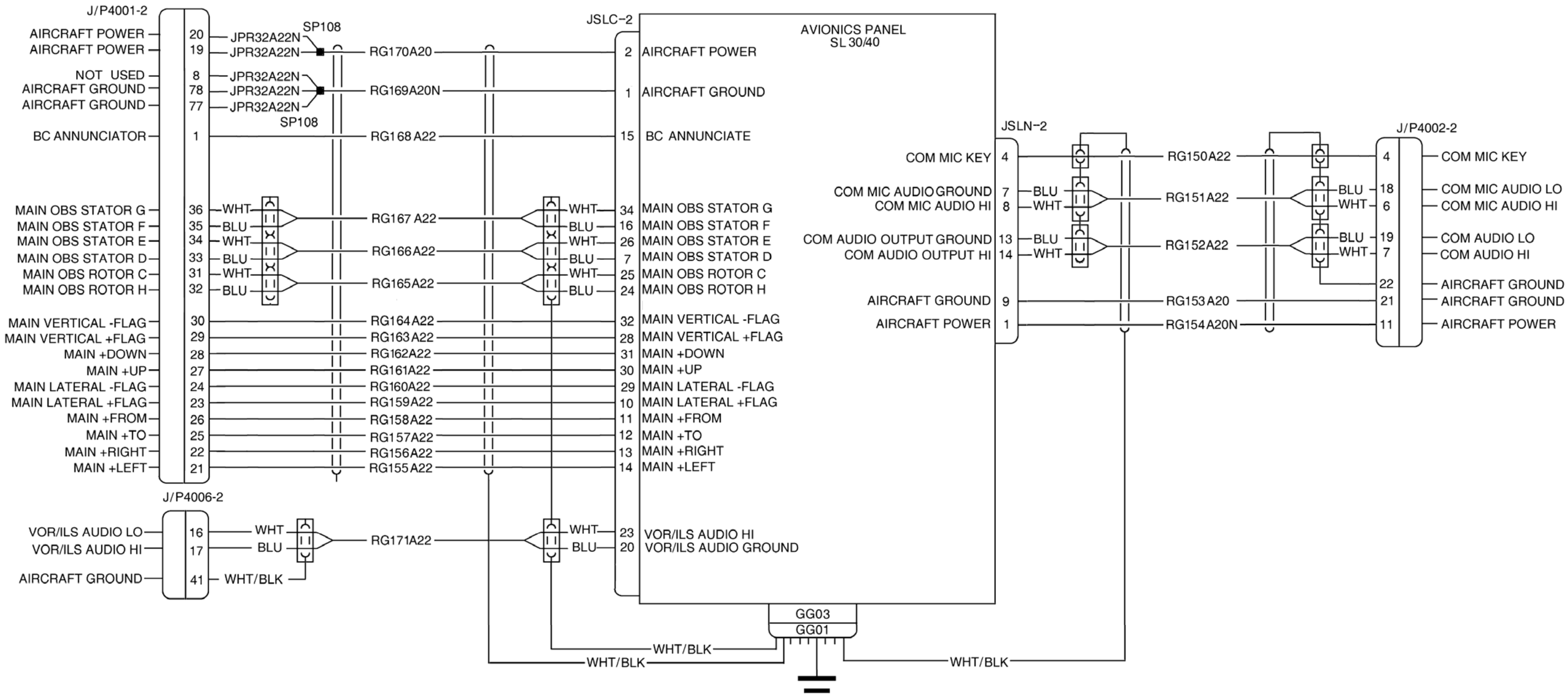

Figure 91-35 Avionics (GNS to SL Adapter)

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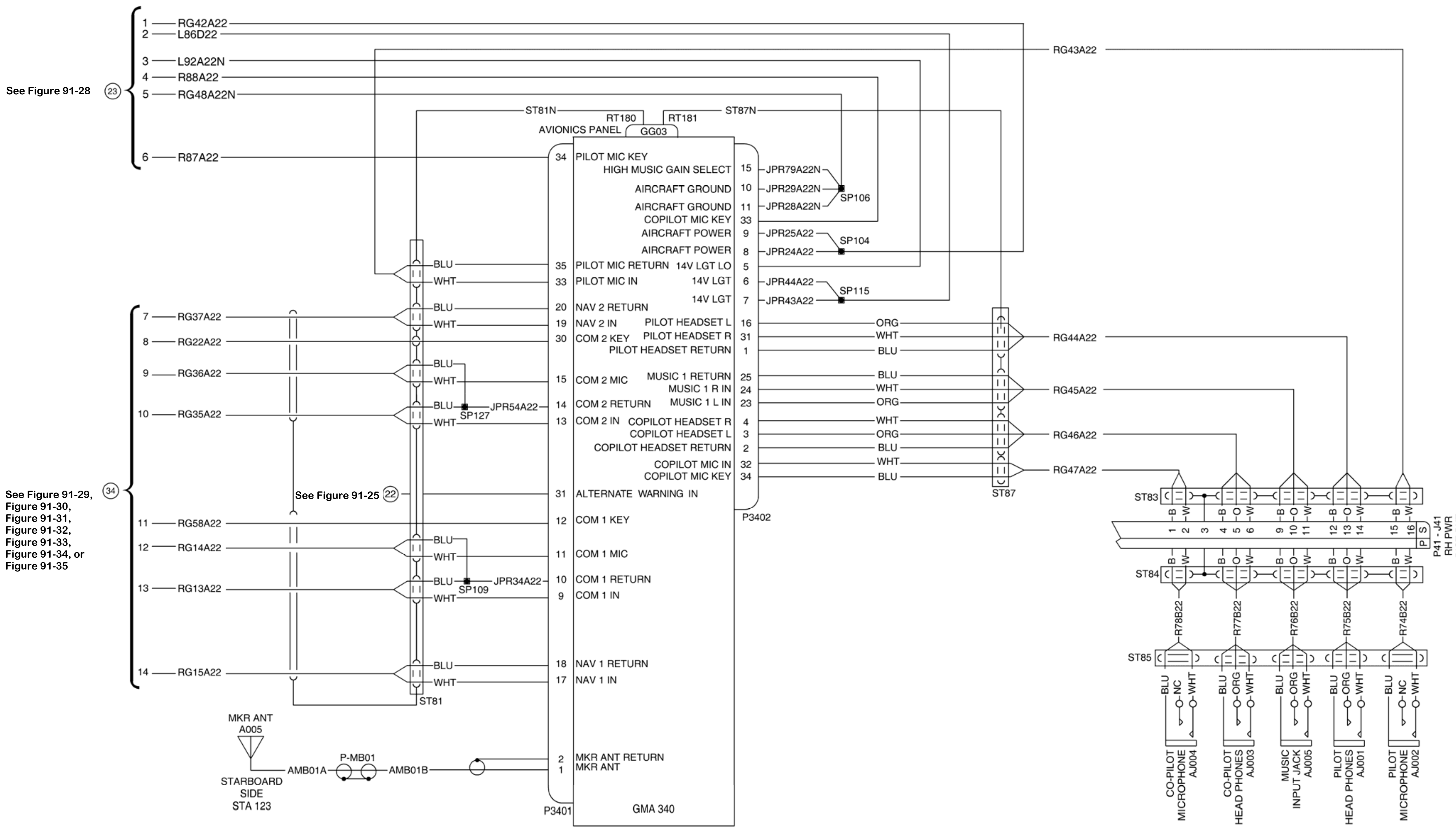

Figure 91-36 Avionics (Audio Panel)

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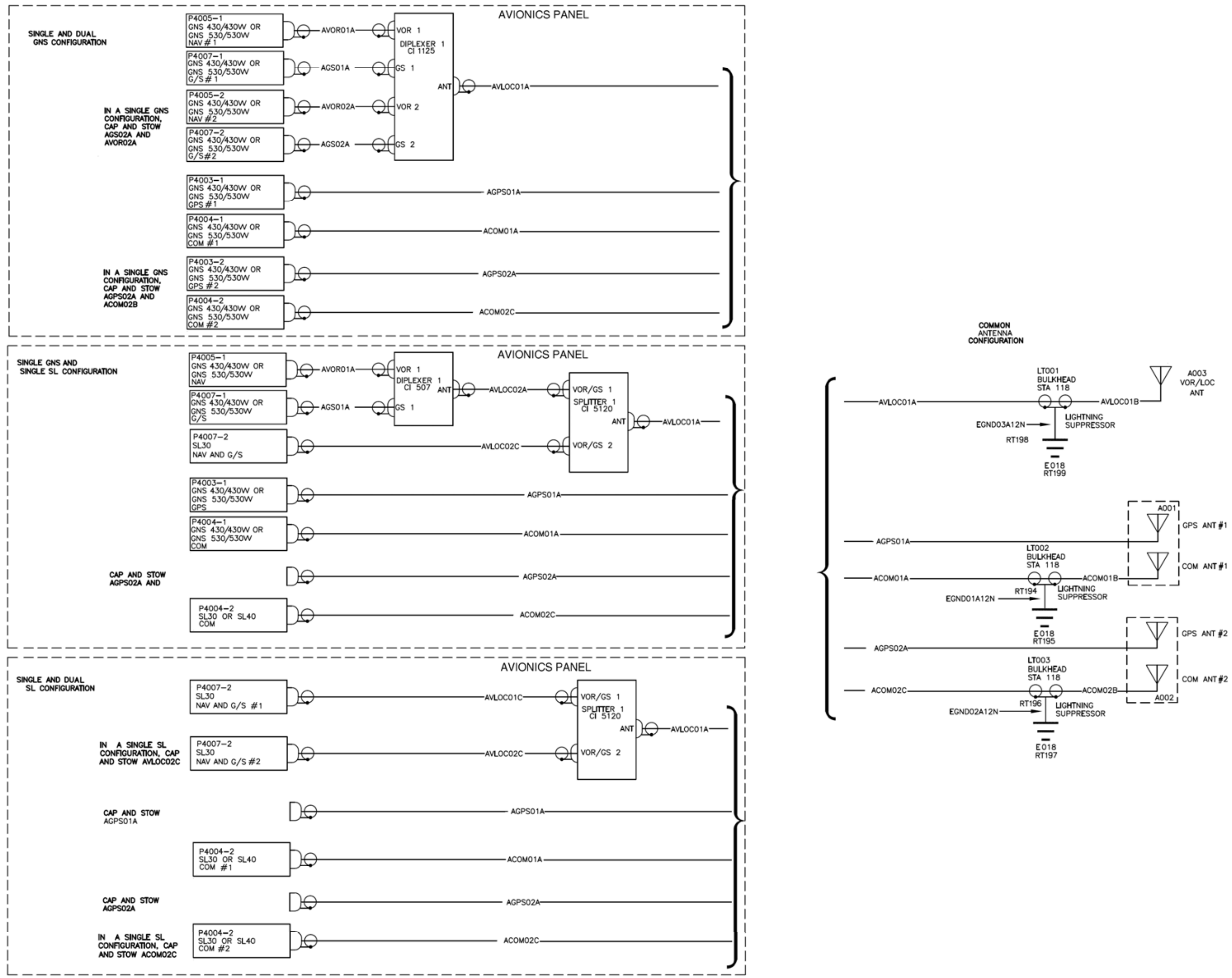

Figure 91-37 Avionics (Antenna Connections)

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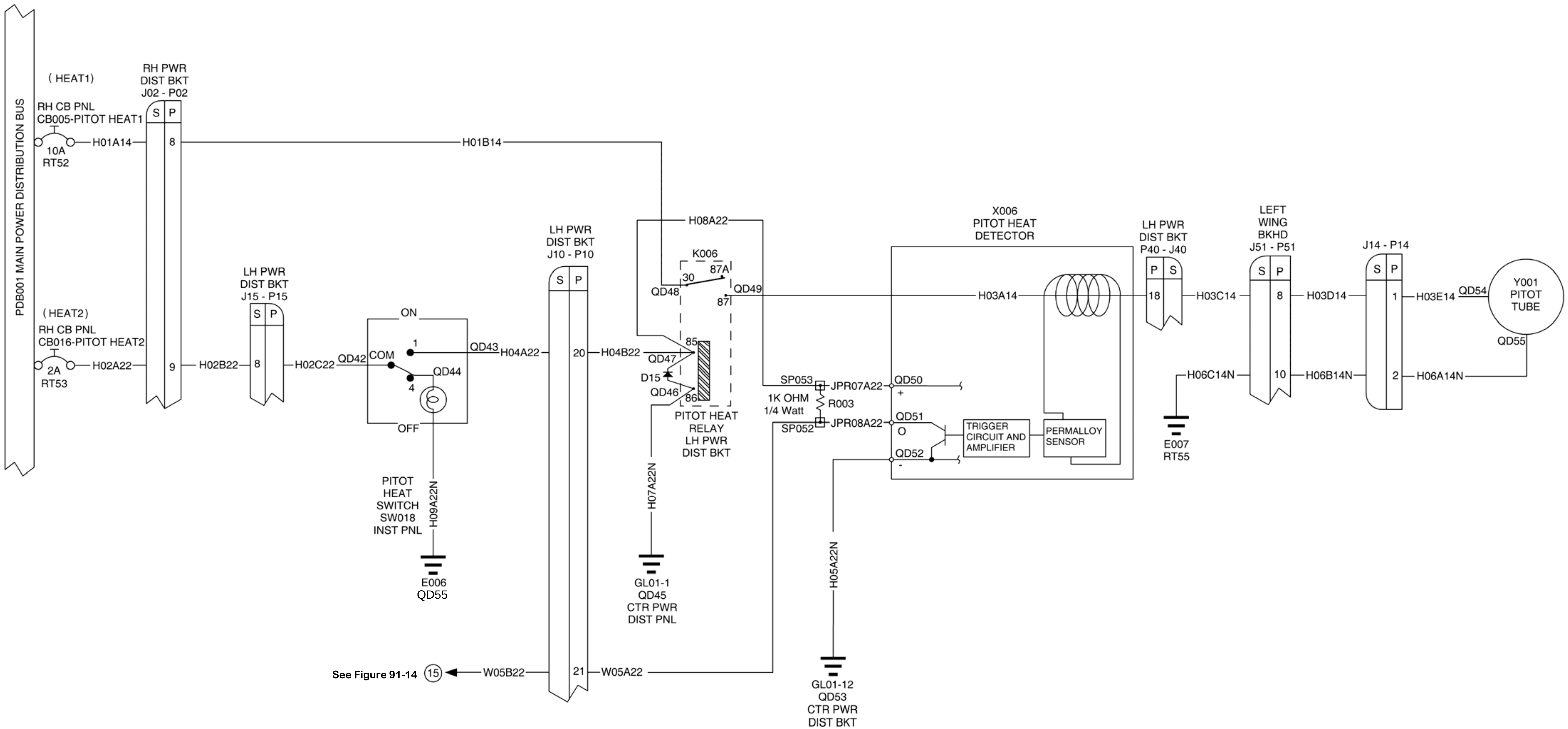

Figure 91-38 **Pitot Heating System**

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