AIRPLANE MAINTENANCE MANUAL

PA-28-181

ARCHER

(S/N'S 2843001–2843999, 2881001 AND UP)

AND

PIPER PILOT

(S/N'S 28020001 AND UP)

(See also the Piper Pilot Supplement, P/N 767-111)

PIPER AIRCRAFT, INC.

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Manufacturers Association

REVISION STATUS

1. Definitions

The following definitions apply to the columns in paragraph "2".

A. Revision

The data in the revision column is comprised of two elements:

(1) A Type of Revision Code: ORG = Original, CR = Complete Revision, and PR = Partial Revision.

NOTE: Partial Revisions (PR) are listed only until the next Complete Revision (CR) is published. Then they are removed.

(2) The Revision Date in YYYYMMDD format.

NOTE: The Revision Date is the date placed on each revised page. It exists to separate one version of a page from another. Revision Date does not indicate the calendar date when the revision was actually published and available to the public. However, ***this is the date that appears in the Current Revision Checklist*** in the Customer Service Information File.

B. Publication Date

Publication Date usage has varied over the years. 1997–2013 it generally was synchronized with the revision date regardless of when the revision was published. In mid 2013 and later, the Publication Date is the calendar date when the revision was actually published and available to the public.

NOTE: ***This date does not appear in the Current Revision Checklist*** in the Customer Service Information File.

2. Revisions

Revisions to this Maintenance Manual, (P/N 761-883), originally issued December 9, 1997 are as follows:

Revision	**Publication Date**
ORG19971209	December 9, 1997
CR19990909	September 9, 1999
CR20061130	November 30, 2006
CR20140306	April 4, 2014
PR20160601	August 4, 2016
PR20160901	October 25, 2016
PR20161101	August 29, 2017
PR20171115	December 19, 2017
PR20190515	March 17, 2020
PR20200315	March 17, 2020
PR20200531	July 13, 2020
PR20201015	November 16, 2020
PR20210730	August 3, 2021

3. Availability

This maintenance manual, related inspection reports and manuals, service publications (SB, SL, etc.) and other Piper publications are available as described in the Owner Publications Catalog (part of the Customer Service Information File, see below).

Consult the "Customer Service Information File" (a free download from the Piper Aircraft, Inc. website at http://www.piper.com/technical-publications-documents/) to verify that you have the latest revision.

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INTRODUCTION

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
Intro-List of Effective Pages	1	Jul 30/21			
	2	Mar 6/14			
Intro-Table of Contents	1	Jul 30/21			
	2	Mar 6/14			
Introduction	1	Oct 15/20			
	2	Jul 30/21			
	3	Oct 15/20			
	4	Oct 15/20			
	5	Oct 15/20			
	6	Oct 15/20			
	7	May 15/19			
	8	Mar 15/20			
	9	Mar 15/20			
	10	Jun 1/16			
	11	Jul 30/21			
	12	Jul 30/21			
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	14	Jul 30/21			
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	16	Jun 1/16			
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	6	May 31/20			
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	10	Oct 15/20			
	11	Jul 30/21			
	12	Jun 1/16			
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	15	Mar 15/20			
	16	Jun 1/16			

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INTRODUCTION

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INTRODUCTION

1. Instructions for Continued Airworthiness

WARNING: INSTRUCTIONS FOR CONTINUED AIRWORTHINESS (ICA) FOR ALL NON-PIPER APPROVED STC INSTALLATIONS ARE NOT INCLUDED IN THIS MANUAL. WHEN A NON-PIPER APPROVED STC INSTALLATION IS INCORPORATED ON THE AIRPLANE, THOSE PORTIONS OF THE AIRPLANE AFFECTED BY THE INSTALLATION MUST BE INSPECTED IN ACCORDANCE WITH THE ICA PUBLISHED BY THE OWNER OF THE STC. SINCE NON-PIPER APPROVED STC INSTALLATIONS MAY CHANGE SYSTEMS INTERFACE, OPERATING CHARACTERISTICS AND COMPONENT LOADS OR STRESSES ON ADJACENT STRUCTURES, PIPER PROVIDED ICA MAY NOT BE VALID FOR AIRPLANES SO MODIFIED.

The PIPER PA-28-181 Archer (i.e., Archer III, Archer, Archer LX, Archer TX, Archer DX(1)) Maintenance Manual constitutes the Instructions for Continued Airworthiness (ICA) in accordance with Federal Aviation Regulations (FAR) Part 23, Appendix G. Chapter 4 contains the Airworthiness Limitations Section (ALS) (4-00-00) and the Inspection Program is in Chapter 5 (5-20-00).

WARNING: BOTH THIS AIRPLANE MAINTENANCE MANUAL (AMM), P/N 761-883, AND THE PA-28-181 PIPER PILOT AIRPLANE MAINTENANCE MANUAL SUPPLEMENT (AMMS), P/N 767-111, ARE REQUIRED TO PROPERLY MAINTAIN PIPER PILOT AIRPLANES IN AN AIRWORTHY CONDITION.

This manual also constitutes part of the ICA for the PA-28-181 Piper Pilot series of airplanes. Information common to both the Archer and Pilot airplanes is addressed herein. Information unique to the Pilot airplanes is presented in the Piper Pilot Airplane Maintenance Manual Supplement (P/N 767-111).

2. General

This publication is prepared in accordance with the General Aviation Manufacturers Association (GAMA) Specification No. 2, with respect to the arrangement and content of the System/Chapters within the designated Chapter/Section-numbering system.

WARNING: USE ONLY GENUINE PIPER AIRCRAFT PARTS OR PIPER AIRCRAFT APPROVED PARTS OBTAINED FROM PIPER APPROVED SOURCES, IN CONNECTION WITH THE MAINTENANCE AND REPAIR OF PIPER AIRPLANES.

This manual generally does not contain hardware callouts for installation. Hardware callouts are only indicated where a special application is required. To confirm the correct hardware used, refer to the PA-28-181 Parts Catalog, P/N 761-898, and FAR 43 for proper use.

Genuine PIPER parts are produced and inspected under rigorous procedures to ensure airworthiness and suitability for use in PIPER airplane applications. Parts purchased from sources other than PIPER, even though identical in appearance, may not have had the required tests and inspections performed, may be different in fabrication techniques and materials, and may be dangerous when installed in an airplane.

Additionally, reworked or salvaged parts or those parts obtained from non-PIPER approved sources, may have service histories which are unknown or cannot be authenticated, may have been subjected to unacceptable stresses or temperatures or may have other hidden damage not discernible through routine visual or nondestructive testing. This may render the part, component or structural assembly, even though originally manufactured by PIPER, unsuitable and unsafe for airplane use.

(1) Information in this manual covers all aspects of the Archer DX airplane with the exception of the power plant and associated systems. The power plant and its installation is covered in the ICA provided by the STC holder.

PIPER AIRCRAFT, INC. expressly disclaims any responsibility for malfunctions, failures, damage or injury caused by use of non-PIPER approved parts.

NOTE: PIPER AIRCRAFT, INC. expressly reserves the right to supersede, cancel and/or declare obsolete any part, part numbers, kits or publication that may be referenced in this manual without prior notice.

Also, Piper Aircraft, Inc. may possess manufacturer's data which defines minimum type design beyond what may be assumed by an authorized repair entity. When a repair is proposed, it is the responsibility of the repairer per AC 43.13-1 to determine that the proposed repair is not contrary to manufacturer's data. The repairer or aircraft owner or his agent should contact Piper directly to determine that a proposed repair is not in conflict with minimum type design capability.

If you have any question concerning the care of your airplane, be sure to include the airplane serial number in any correspondence to Piper.

3. Effectivity

This maintenance manual is effective for PA-28-181 Archer (i.e., Archer III, Archer, Archer LX, Archer TX, Archer DX) airplane serial numbers 2843001 thru 2843999, 2881001 and up; and PA-28-181 Piper Pilot airplane serial numbers 28020001 and up. Model years for Acher airplanes are shown in Chart 1. Model years for Piper Pilot airplanes are in 767-111.

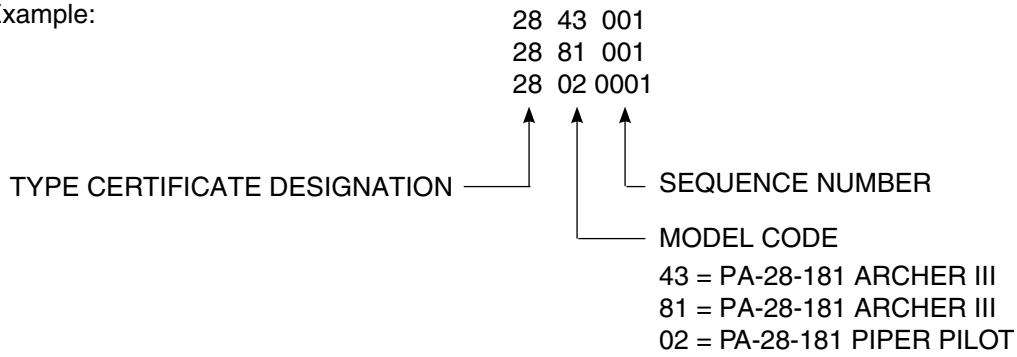
CHART 1
MODEL YEARS

Model Year	Serial Numbers	Model Year	Serial Numbers
1995	2843001 thru 2843020	2011	2843696 thru 2843699
1996	2843021 thru 2843066	2012	2843700 thru 2843702
1997	2843067 thru 2843111	2013	2843703 thru 2843750
1998	2843112 thru 2843202	2014	2843751 thru 2843797
1999	2843203 thru 2843414	2015	2843798 thru 2843822
2001	2843415 thru 2843500	2016	2843823 thru 2843862
2002	2843501 thru 2843541	2017	2843863 thru 2843937
2003	2843542 thru 2843585		2881001 thru 2881003
2004	2843586 thru 2843604	2018	2843938 thru 2843949
2005	2843606 thru 2843619		2881004 thru 2881093
2006	2843620 thru 2843647	2019	2881094 thru 2881282
2007	2843648 thru 2843663	2020	2881283 thru 2881430
2008	2843664 thru 2843673	2021	2881431 and up
2010	2843674 thru 2843695		

NOTE: The above is provided as a general reference only.

4. Serial Number Explanation

Example:



5. Assignment of Subject Material

This publication is divided into industry standard, three element, numeric subject groupings as follows:

A. System/Chapter - The various groups are broken down into major systems such as Environmental Systems, Electrical Power, Landing Gear, etc. They are assigned a number, which becomes the first element of the standardized numbering system. Thus, the element “28” of the number 28-40-01 refers to the chapter “Fuel”. Everything concerning the fuel system will be covered in this chapter.

B. Sub-System/Section - The major systems/chapters of an airplane are broken down into subsystems. These sub-systems are identified by the second element of the standard numbering system. The element “40” of the number 28-40-01 concerns itself with the indicating section of the fuel system.

C. Unit/Subject - The individual units within a sub-system/section may be identified by the third element of the standard numbering system. The element “01” of the number 28-40-01 is a subject designator. This element is assigned at the option of the manufacturer and is normally zeroed out by PIPER.

Refer to the Chapter/Section Index Guide, below, for a complete breakdown and list. The material is arranged in ascending numerical sequence.

6. Pagination

The Chapter - Section (i.e. - 28-40-00) numbering system (explained above) forms the primary page numbering system for this manual. Within each Section, pages are numbered consecutively beginning with Page 1 (i.e., 28-40-00, Page 1).

7. Identifying Revised Material

A vertical line (i. e., change bar) along the left-hand margin of the page (or text column) is used to identify revised text or illustrations.

Example.

A change bar in the left-hand margin opposite the footer (i.e., chapter/section/subject, page number and date), indicates that the text was unchanged but the material was relocated.

Change bars in the individual chapter Tables of Contents do not indicate a change to that page, but rather that the information in the actual paragraph has changed.

NOTE: Change bars are not used in the title pages, list of effective pages, the Index, or the Wiring Diagram (Schematics) Cross-Reference.

A revision to a page is defined as a change to the text or illustrations that existed previously. Revisions, additions and deletions are identified by a vertical line (i. e., change bar) along the left-hand margin of the page opposite only the text or illustration that was changed. Reformatted, but otherwise unchanged, text is not identified by a change bar.

8. Indexing

An alphabetically arranged subject Index follows this introduction to assist the user in locating desired information. In addition, each System/Chapter begins with an individual Table of Contents.

9. List of Effective Pages

Each System/Chapter has a List of Effective Pages preceding the Table of Contents to identify the effective revision date for each page in that chapter.

10. Warnings, Cautions and Notes

These adjuncts to the text are used to highlight or emphasize important points when necessary. WARNINGS call attention to use of materials, processes, methods, procedures or limits which must be followed precisely to avoid injury or death to persons. **CAUTIONS** call attention to methods and procedures which must be followed to avoid damage to equipment. NOTES call attention to methods which make the job easier. Warnings and Cautions precede and Notes follow the text to which they apply.

11. Accident/Incident Reporting

To improve our Service and Reliability system and aid in Piper's compliance with FAR 21.3, knowledge of all incidents and/or accidents must be reported to Piper immediately. To expedite and assist in reporting all incidents and accidents, Piper Form 420-01 has been created. See Service Letter 1041 for latest revision. This procedure is to be used by all Dealers, Service Centers and Repair Facilities.

12. Supplementary Publications

The following publications/sources provide servicing, overhaul and parts information for the PA-28-181 airplane and its various components. Use them to supplement this manual.

A. Piper Publications

		P/N
(1)	Parts Catalog	761-898
(2)	Periodic Inspection Report	767-008
(3)	Progressive Inspection Manual (50 hour)	767-023
(4)	Piper Pilot Airplane Maintenance Manual Supplement	767-111
(5)	Piper Pilot Periodic Inspection Report	767-113
(6)	Piper Pilot Progressive Inspection Manual (50 hour)	767-112

B. Vendor Publications

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY.

(1) AIR CONDITIONING COMPRESSOR

Vendor:	T/CCI Manufacturing LLC 2120 N. 22nd Street Decatur, Illinois 62526 http://tccimfg.com	PH: (217) 422-0055
Service Manual:	2-Cylinder Service Manual	
or		
	Sanden International (USA), Inc. 601 South Sanden Blvd. Wylie, Texas 75098 http://www.sanden.com/	PH: (972) 442-8400 FAX: (972) 442-8700

(2) ALTERNATOR

Vendor:	Hartzell Engine Technologies 2900 Selma Hwy Montgomery, Alabama 36108 http://hartzell.aero/	PH: FAX:	(877) 359-5355 (334) 386-5410
Overhaul Manual:	OE-A2 Starter and Alternator Overhaul Manual		
or,			
Vendor:	Hartzell Engine Technologies 2900 Selma Hwy Montgomery, Alabama 36108 http://www.plane-power.com/	PH: FAX:	(877) 359-5355 (334) 386-5410

(3) ANGLE OF ATTACK SYSTEM

Vendor:	Safe Flight Instrument Corporation 20 New King Street White Plains, NY 10604 http://www.safeflight.com/	PH:	(914) 946-9500
System Installation and User Manual:	Dwg. 56201-1		

(4) AUTOFLIGHT

Vendor:	Honeywell One Technology Center 23500 W. 105th St., M/D #45 Olathe, KS 66061-1950 http://www.bendixking.com/	(or)	S-TEC Corporation One S-TEC Way Mineral Wells, TX 76067 PH: (940) 325-9406 http://www.s-tec.com	
or,				
Vendor:	Garmin International See “Integrated Avionics System (IAS)” on page INTR7.			

(5) BATTERY

Vendor:	Concorde Battery Corporation 2009 San Bernardino Road West Corvina, California 91790 http://www.concordebattery.com	PH:	(626) 813-1234
RG® Series Emergency Aircraft Battery CMM:	Document No. 5-0167		
RG® Series Main Aircraft Battery CMM:	Document No. 5-0171		
or			
Vendor:	Gill/Teledyne 840 W. Brockton Ave. Redland, California 92374 http://www.gillbatteries.com	PH:	(800) 456-0070

(6) BRAKES AND WHEELS

Vendor:	Parker Hannifin Corp Aircraft Wheel and Brake Division 1160 Center Road Avon, Ohio 44011 http://www.parker.com/	PH:	(800) 272-5464

(7) EMERGENCY LOCATOR TRANSMITTER

Vendor:	Artex Aircraft Supplies 14405 Keil Road NE Aurora, Oregon 97002 http://www.artex.net/	PH:	(800) 547-8901

(8) ENGINE

Vendor:	Lycoming Engines 652 Oliver Street Williamsport, PA 17701 http://www.lycoming.com/	PH: FAX:	(717) 323-6181 (717) 327-7101
Overhaul Manual:	DIRECT DRIVE MODELS	- P/N 60294-7	
Parts Catalog:	O-360- ... -A4M, ... ENGINES	- P/N PC-306-1	
or	IO, -360-A, -B Series (B4A)	- P/N PC-406-1	
Operators Handbook:	O, HO, IO, AIO, HIO, TIO-360 Series	- P/N 60297-12	

(9) ELECTRIC FUEL PUMP

Vendor:	Weldon Pumps P.O. Box 46579 640 Golden Oak Parkway Oakwood Village, Ohio 44146 http://www.weldonpumps.com/index.html	PH: FAX:	(440) 232-2282 (440) 232-0606
or,	Parker Hannifin Corp. See “Vacuum Pump” on page INTR9.		

(10) FLIGHT DATA MONITORING SYSTEM

Vendor:	Appareo Systems 1810 NDSU Research Circle North Fargo, ND 58102 http://www.appareo.com	PH:	(701) 356-2200

(11) FIRE EXTINGUISHER (PORTABLE)

Vendor:	H3R Inc. 43 Magnolia Ave # 4 San Francisco, California 94123-2911 http://www.h3r.com/index.htm	PH:	(800) 249-4289

(12) HI-LOK FASTENERS AND TOOLS

Vendor:	Hi-Shear Corporation 2600 Skypark Drive Torrance, California 90509	PH:	(213) 326-8110

(13) INTEGRATED AVIONICS SYSTEM (IAS)

Vendor:	Garmin International 1200 East 151ST Street Olathe, KS 66062 http://www.garmin.com	PH:	(913) 397-8200

(a) S/N's 2843701, 2843703–2843822, 2843824–2843851, and 2843853

G1000 Pilots Guide:	P/N 190-01459-00
G1000 Line Maintenance and Configuration Manual:	P/N 190-00303-04

(b) S/N's 2843852, 28438544–2843999

G1000 Pilots Guide:	P/N 190-02130-00
G1000 Line Maintenance Manual:	P/N 190-02094-00

(c) S/N's 2843823, 2881001 and up

G1000 Pilots Guide:	P/N 190-02130-02
G1000 Line Maintenance Manual:	P/N 190-02413-00

(14) INTEGRATED FLIGHT DISPLAY SYSTEM (IFDS)

Vendor:	Avidyne Corporation 55 Old Bedford Road Lincoln, MA 01773 http://www.avidyne.com/index.htm	PH:	(800) 284-3963

Instructions for Continued Airworthiness (ICA)

Primary Flight Display and Magnetometer/OAT:	Document No. AVPFD-174
Multifunction Display:	Document No. AVMFD-167
Data Acquisition Unit:	Document No. AVSIU-011

(15) INTERCOM

Vendor:	Flightcom Corporation 7340 S.W. Durham Road Portland, OR 97224 http://www.flightcom.net	PH: Fax:	(800) 432-4342 (503) 620-2943
Installation/ Operation Manual:	Part No. 600-0036-00		

(16) LIGHTS - NAVIGATION, STROBE, AND STANDBY/MAP LIGHTS

Vendor:	Whelen Engineering Co. Inc. Route 145, Winthrop Rd. Chester, Connecticut 06412 http://www.whelen.com/	PH: FAX:	(860) 526-9504 (860) 526-2009

(17) MAGNETOS

Vendor:	Slick Aircraft Products Champion Aerospace P.O. Box 686 1230 Old Norris Road Liberty, SC 29657 http://www.championaerospace.com/	PH:	(904) 772-1909
Installation, Operation and Maintenance Instructions:	F1100 Master Service ManuaL, 4300/6300 Series Magneto Maintenance And Overhaul Manual - L-1363		
Installation, Operation, Maintenance, and Troubleshooting:	SlickSTART”” Installation, Operation, Maintenance, and Troubleshooting Manual L-1492		
or,			
Service Manual:	S-20/S-200 Magneto Service Support Manual P/N X42002-3		

(18) NAVIGATION, COMMUNICATIONS, AND GPS (NAV/COM/GPS)

Vendor:	Garmin International See “Integrated Avionics System (IAS)” on page INTR7.

(19) PROPELLER

Vendor:	Sensenich Corporation 14 Citation Lane Lititz, PA 17543 http://www.sensenich.com/	PH: FAX:	(717) 569-0435 (717) 560-3725
Repair Manual:	METAL PROPELLER REPAIR MANUAL - SPRM 590		

(20) STANDBY ATTITUDE INDICATOR

Vendor:	Mid-Continent Instruments Co., Inc. 9400 E. 34 TH Street N. Wichita, KS 67226 http://www.mcico.com/index.html	PH: FAX:	(316) 630-0101 (316) 630-0723
Installation Manual and Operating Instructions:	Manual No. 9015762		

(21) STANDBY ELECTRONIC FLIGHT DISPLAY

Vendor: Aspen Avionics, Inc. PH: (505) 856-5034
5100 Indian School Road NE
Albuquerque, NM 87110
http://www.aspenavionics.com

Tech Note for
Air Data Calibration: Tech Note 2009-09, Document No. 991-00023-001

or,

Vendor: Garmin
(See address under IAS on page “INTR7”)

G5 Pilots Guide: P/N 190-01112-12

(22) STARTER

Vendor: Sky-Tec PH: (800) 476-7896
350 Howard Clemmons Rd. FAX: (817) 573-2252
Granbury, Texas 76048
http://www.skytecair.com

or,

Hartzell Engine Technologies
See “Alternator” on page INTR5.

(23) VACUUM PUMP

Vendor: Aero Accessories, Inc. PH: (800) 822-3200
1240 Springwood Avenue
Gibsonville, NC 27249
http://www.aeroaccessories.com/index.html

or,

Parker Hannifin Corp. PH: (800) 382-8422
Airborne Division
711 Taylor Street
Elyria, Ohio 44035
http://www.parker.com

(24) VACUUM REGULATOR

Vendor: Parker Hannifin Corp.
See “Vacuum Pump”, above.

(25) VOLTAGE REGULATOR

Vendor: Electro Systems, Inc. PH: (888) 461-6077
Airport Complex
P. O. Box 273
Fort Deposit, Alabama 36032
http://www.kellyaerospace.com/index.htm/

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13. Chapter/Section Index Guide

NOTE: The following GAMA Specification No. 2 standard chapters are not included in this Maintenance Manual: 29, 35, 36, 38, 49, 54, 60, 72, 75, 81, and 83. These chapters are omitted because the subject system is either: not installed in these airplanes; adequately covered in vendor or other manuals; or, for ease of use, has been combined with another chapter.

CHAPTER	SECTION	TITLE
4		AIRWORTHINESS LIMITATIONS
	00	Airworthiness Limitations
5		TIME LIMITS/MAINTENANCE CHECKS
	00	General
	10	Time Limits
	20	Scheduled Maintenance
	30	Special Inspections
	50	Unscheduled Maintenance Checks
6		DIMENSIONS AND AREAS
	00	General
	10	Principal Dimensions
	20	Reference Lines
	40	Access Provisions
7		LIFTING AND SHORING
	10	Jacking
8		LEVELING AND WEIGHING
	10	Weighing
	20	Leveling
9		TOWING AND TAXIING
	10	Towing
	20	Taxiing
10		PARKING AND MOORING
	10	Parking
	20	Mooring
11		PLACARDS AND MARKINGS
	00	General
	20	Exterior Placards and Markings
	30	Interior Placards
12		SERVICING
	00	General
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	20	Scheduled Servicing

CHAPTER	SECTION	TITLE
20		STANDARD PRACTICES - AIRFRAME
	00	General
	20	Painting
21		ENVIRONMENTAL SYSTEMS
	00	General
	20	Distribution
	40	Heating
	50	Cooling
22		AUTO FLIGHT
	10	Autopilot
23		COMMUNICATIONS
	00	General
	60	Static Discharging
24		ELECTRICAL POWER
	00	General
	30	DC Generation
	40	External Power
	60	Electrical Load Distribution
25		EQUIPMENT/FURNISHINGS
	10	Flight Compartment
	60	Emergency
26		FIRE PROTECTION
	20	Extinguishing
27		FLIGHT CONTROLS
	00	General
	10	Aileron
	20	Rudder and Tab
	30	Stabilator and Tab
	50	Flaps
28		FUEL
	00	General
	10	Storage
	20	Distribution
	40	Indicating

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	00		General
	30		Pitot and Static
31	INDICATING / RECORDING SYSTEMS		
	20		Independent Instruments
	30		Recorders
	50		Central Warning Systems
32	LANDING GEAR		
	00		General
	10		Main Gear
	20		Nose Gear
	40		Wheels and Brakes
33	LIGHTS		
	10		Flight Compartment
	40		Exterior
34	NAVIGATION		
	00		General
	10		Flight Environment Data
	20		Attitude and Direction
	24		Strandby Display
		00	Aspen Avionics EBD Standby Display
		01	Garmin G5 Standby Display
	25		Integrated Avionics System
		01	Garmin G1000
		02	Garmin G1000 NXi in S/N's 2843852, 2843854–2843999
		03	Garmin G1000 NXi in S/N's 2843823, 2881001 and up
	50		Dependent Positioning Determining
37	VACUUM		
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	10		Distribution
	20		Indicating
39	ELECTRICAL/ELECTRONIC PANELS		
	10		Instrument and Control Panels
	20		Electrical and Electronic Equipment
51	STRUCTURES		
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	10		Investigation, Cleanup, and Aerodynamic Smoothness
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	80		Electrical Bonding

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	00	General
	10	Passenger/Crew
	30	Cargo
53		FUSELAGE
	20	Main
55		STABILIZERS
	00	General
	20	Stabilator
	30	Vertical Stabilizer
	40	Rudder
56		WINDOWS
	10	Flight Compartment
	20	Passenger Compartment
57		WINGS
	00	General
	10	Main Frame
	20	Auxiliary Structure
	40	Attach Fittings
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70		STANDARD PRACTICES - ENGINE
	00	General
71		POWER PLANT
	00	General
	10	Cowling
	20	Mounts
	60	Air Intakes
73		ENGINE FUEL AND CONTROL
	10	Distribution
	20	Controlling
	30	Indicating
74		IGNITION
	10	Electrical Power Supply
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AIRWORTHINESS LIMITATIONS

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AIRWORTHINESS LIMITATIONS

NOTE: The Airworthiness Limitations section is FAA approved and specifies maintenance required under §§ 43.16 and 91.403 of the Federal Aviation Regulations unless an alternative program has been FAA-approved.

1. Approved Mandatory Replacement Times for Type Certification

(PIR-TCDS 2A13, Rev. 60.)

No limitations, related to fatigue life of the airplane and its components, have been established for the PA-28-181 Archer III or PA-28-181 Piper Pilot airplanes.

2. Approved Mandatory Structural Inspection Intervals

None.

3. Inspection Procedures for those Approved Mandatory Structural Inspection Items

None.

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CHAPTER

5

TIME LIMITS / MAINTENANCE CHECKS

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GENERAL

Piper Aircraft, Inc. (Piper) takes a continuing interest in having the owner get the most efficient use from his airplane, and keeping the airplane in the best mechanical condition. To that end, Piper publishes a recurring maintenance schedule which is supplemented with Service Bulletins, Service Letters and Service Spares Letters as required.

A. The recurring maintenance schedule for the PA-28-181 Archer III is provided in 5-20-00. The recurring maintenance schedule for the PA-28-181 Piper Pilot airplanes is provided in 767-111.

B. Piper Service Bulletins are of special importance and Piper considers compliance mandatory.

C. Service Letters deal with product improvements and service hints pertaining to the affected aircraft. Owners should give careful attention to service letter information so they can ensure their airplane is properly serviced and kept up to date with the latest changes.

D. Service Spares Letters offer improved parts, kits and optional equipment which were not originally available. These may be of interest to the owner.

E. Service Bulletins, Service Letters and Service Spares Letters are emailed to Piper Dealers/Service Centers. Owners are encouraged to download these service publications from http://www.piper.com/.

NOTE: Piper mails flight manual (AFM / POH) revisions to the registered owner's name and address as shown on the Aircraft Registration Certificate. If the aircraft is based and/or operated at a different location (or locations) and/or by a person (or persons) other than those recorded on the aircraft registration, then the registered owner(s) is responsible for forwarding these to the operating location(s) or person(s).

Changes in aircraft registration may take a substantial amount of time to be recorded by the Federal Aviation Administration and received by Piper to change the mailing address. Owners and operators should make arrangements to keep abreast of flight manual revisions during this interim period through their Piper Dealer/Service Center.

The Federal Aviation Administration (FAA) publishes Airworthiness Directives (AD's) that apply to specific aircraft. They are mandatory changes and are to be complied within a time limit set by the FAA. When an AD is issued, it is sent to the latest registered owner of the affected aircraft and also to subscribers of their service. The owner is solely responsible for being aware of and complying with airworthiness directives.

NOTE: A searchable database of AD's is available on the FAA website. See the "Airworthiness Directives" link at "http://www.faa.gov".

Owners should periodically check with a Piper Dealer/Service Center to find out the latest information to keep his aircraft up to date.

Service Bulletins, Service Letters, and Service Spares Letters are also available by subscription. See the availability statement in Revision Status.

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TIME LIMITS

1. General

   A. Refer to 4-00-00 for the FAA-approved airworthiness limitations section. It sets forth each mandatory replacement time, structural inspection interval, and related structural inspection procedure required for type certification.

   B. Refer to 5-20-00 for Piper's recommended Inspection Programs. They include the frequency and extent of the inspections required for the continued airworthiness of these airplanes.

   C. Inspections required by Flight Hour or Calendar Year, if due, are included as part of the Annual / 100 Hour Inspection and/or the Progressive Inspection Event cycles, and are listed individually in 5-30-00.

2. Life Limited Parts Marking and Disposition

   14 CFR Part 43.10, Disposition of Life-Limited Aircraft Parts requires that proper procedures are followed when removing life limited parts with time and/or cycles remaining on them as well as the disposition of life limited parts with no time and/or cycles left. As stated in 4-00-00, there are no life limited parts defined by Type Certificate (TC). Other parts which are replaced or rebuilt after having accumulated cycles, hours, or other replacement limit are specified in 5-20-00 or 5-30-00.

   A. Parts that are removed prior to accumulating their life limit, are to be marked with indelible ink or marker with the part number, serial number and accumulated life status as defined in 14 CFR Part 43.10 in a manner that does not affect part structural integrity, i.e. - no surface deformation such as vibration/etching allowed.

   B. Parts that have accumulated the life limit shall be disposed of in accordance with the applicable FARs. Piper recommends life limited parts with no time and/or cycles remaining be completely destroyed.

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SCHEDULED MAINTENANCE

This section provides instructions for conducting inspections. Repair or replacement instructions for those components found to be unserviceable during inspections will be found in the applicable airplane system chapter. (See Introduction, Chapter/Section Index Guide.)

WARNING: GROUND THE MAGNETO PRIMARY CIRCUIT (P LEAD), BEFORE PERFORMING ANY MAINTENANCE OPERATION ON THE ENGINE.

1. Description

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The recurring maintenance schedule for the PA-28-181 Archer III is provided herein as an Annual / 100 Hour Inspection. A Progressive Inspection Program is available in a separate manual form. See the availability statement in Revision Status.

Piper inspection programs comply with the F.A.A. Federal Aviation Regulations Parts 43, 91 and 135. The owner/operator is primarily responsible for maintaining the airplane in an airworthy condition, including compliance with all applicable Airworthiness Directives and conformity with the requirements in FAR 91.409, 91.411 and 91.413.

The first overhaul or replacement of components should be performed at the given periods. The condition of various components can then be used as criteria for determining subsequent periods applicable to the individual airplane, depending on usage, providing the owner/operator has an established Part 91 Progressive Inspection Program (see 91.409(d)) or Part 135 Approved Aircraft Inspection Program (see 135.419).

The time periods given for inspections of various components are based on average usage and environmental conditions.

NOTE: The listed inspection, overhaul and replacement schedules do not guarantee that a particular item or component will reach the listed time without malfunction. Unique operating conditions encountered by individual airplanes cannot be controlled by the manufacturer.

2. Definitions

A. Inspections - Must be performed only by persons authorized by the FAA or appropriate National Aviation Authority who are qualified on these aircraft, using acceptable methods, techniques and practices to determine physical condition and detect defects.

(1) Routine Inspection - Consists of a visual examination or check of the aircraft and its components and systems without disassembly.

NOTE: This includes examination of all fasteners for broken torque stripe. If broken torque stripe is found, loosen fastener, retoque, apply torque stripe, and safety, as required. Please report broken torque stripe via the FAA Service Difficulty Reporting (SDR) system.

(2) Detailed Inspection - Consists of a thorough examination of the aircraft, appliance, component, or system; with disassembly as necessary to determine condition.

(3) Special Inspection - Involves those components, systems or structure which by their application or intended use require an inspection peculiar to, more extensive in scope or at a time period other than that which is normally accomplished during an event or annual inspection.

B. Checks - Can be performed by pilots and/or mechanics who are qualified on this aircraft and consist of examinations in the form of comparisons with stated standards for the purpose of verifying condition, accuracy and tolerances.

C. Approved Inspection - Means a continuing airworthiness inspection of an airplane and its various components and systems at scheduled intervals in accordance with procedures approved by the FAA under FAR Part 91.409(d) or Part 135.419.

D. Tests - Operation of aircraft components, appliances or systems to evaluate functional performance.

(1) Operational Test - A task to determine that an item, is fulfilling its intended purpose.The task does not require quantitative tolerances. This is a fault finding task.

(2) Functional Test - A quantitative check to determine, if one or more functions of an item performs within specified limits. This test may require the use of supplemental bench test equipment.

(3) In addition, each of the above tests must be performed by an FAA Certified Repair Station with appropriate ratings or by a Certified Mechanic who is qualified on this aircraft. The recording of the above function must be made in the permanent aircraft records by the authorized individual performing the test.

E. Bench Test - Means removal of component from the aircraft to inspect for cleanliness, impending failure, need for lubrication, repair or replacement of parts and calibration to at least the manufacturers specifications using the manufacturers recommended test equipment or standards or the equivalent.

Each bench test will be performed by a Piper Service Center, FAA Certified Repair Station with appropriate rating or by a certified mechanic. This test will be performed at the scheduled interval regardless of any bench test performed on a particular component while being repaired/ overhauled before scheduled interval bench test. After the component is installed into the aircraft, an operational test of the component and its related system should be performed to ensure proper function. Serviceable parts that were issued to the component will be filed in the aircraft permanent records. The person performing the test must make appropriate entries in the aircraft's permanent maintenance record.

F. Maintenance - The word maintenance as defined by FAR Part 1, means "inspection, overhaul, repair, preservation and the replacement of parts, but excludes preventive maintenance."

G. On Condition Maintenance - A primary maintenance process having repetitive inspections or tests to determine the condition of units, systems, or portions of structure with regard to continued serviceability (corrective action is taken when required by item condition.)

H. Time - as used in this manual.

(1) (Time-in-service for aircraft components, unless otherwise specified, is a cumulative total of flight hours or calendar time calculated from the time a new or overhauled component was first installed in any aircraft, and including:

(a) the aircraft time that elapses from the initial installation to the first removal, if any; and,

(b) the aircraft time that elapses from each subsequent installation to each subsequent removal, if any; or,

(c) the calendar time elapsed since the installation.

NOTE: Dates stamped on individual components at the time of manufacture are typically applied to determine shelf life - i.e. the maximum time allowed from manufacture/ assembly/cure until actually installed in an aircraft and are not relevant.

Do not, however; ignore markings applied to life-limited parts when removed with time and/or cycles remaining on them.

(2) Aircraft time, flight hours, or aircraft hours are the "Hobbs Time" shown on, or calculated from, the installed "Hour Meter."

3. Inspection Requirements

WARNING: INSTRUCTIONS FOR CONTINUED AIRWORTHINESS (ICA) FOR ALL NON-PIPER APPROVED STC INSTALLATIONS ARE NOT INCLUDED IN THIS MANUAL. WHEN A NON-PIPER APPROVED STC INSTALLATION IS INCORPORATED ON THE AIRPLANE, THOSE PORTIONS OF THE AIRPLANE AFFECTED BY THE INSTALLATION MUST BE INSPECTED IN ACCORDANCE WITH THE ICA PUBLISHED BY THE OWNER OF THE STC. SINCE NON-PIPER APPROVED STC INSTALLATIONS MAY CHANGE SYSTEMS INTERFACE, OPERATING CHARACTERISTICS AND COMPONENT LOADS OR STRESSES ON ADJACENT STRUCTURES, THE PIPER PROVIDED ICA MAY NOT BE VALID FOR AIRPLANES SO MODIFIED.

Inspections must be accomplished by persons authorized by the FAA or appropriate National Aviation Authority. Checks may be performed by a pilot or owner who is checked out on the airplane.

A. Annual / 100 Hour Inspection (See paragraph 4.)

Owners/operators may maintain the airplane solely under FAR 91.409 (a) and (b) inspection requirements. The 100 hour inspection cycle is a complete inspection of the airplane and is identical in scope to an annual inspection.

B. Progressive Inspection

The Progressive Inspection program is designed to permit the best utilization of the aircraft through the use of a planned inspection schedule. This schedule:

P/N 767-023, PA-28-181, Archer III (S/N's 2843001 & up)

is prepared in a manual form. See the availability statement in Revision Status.

Refer to Piper's Customer Service Information File P/N 1753-755 for a checklist to ensure obtaining latest issue.

NOTE: The 50 Hour Progressive Inspection Manual (P/N 767-023) referenced above is not a stand-alone document. It constitutes a snapshot of the Airworthiness Limitations and Inspection sections of the Instructions for Continued Airworthiness (ICA) and is current only at the time of printing. Use it as follows:

(1) Owners/operators desiring to establish a Part 91 Progressive Inspection Program (PIP) (see 91.409(d)) or a Part 135 Approved Aircraft Inspection Program (AAIP) (see 135.419) should use the appropriate Progressive Inspection Manual as a template for submission to their regional FAA office.

(2) Service centers conducting Event Cycle inspections under a FAA-approved PIP or AAIP can use the appropriate Progressive Inspection Manual as a working check-off list/form, provided they verify its currency against the FAA-approved PIP or AAIP.

C. Overlimits Inspection

If the airplane has been operated so that any of its components have exceeded their maximum operational limits, special inspections may be required by Piper and/or the component manufacturer. See 5-50-00 and applicable vendor publications.

4. Annual / 100 Hour Inspection Procedure

A. Scheduled Maintenance (i.e. - paragraph 5.)

(1) The required periodic inspection procedures are listed in paragraph 5. These inspection procedures are broken down into major groups which include Propeller, Engine, Cabin and Cockpit, Fuselage and Empennage, Wing, Landing Gear, Special Inspections, Operational Inspection, and General. The first column in each group lists the inspection or procedure to be performed. The second column is divided into two sub-columns indicating the required inspection interval of 50 hours or 100 hours. Each inspection or operation is required at each of the inspection intervals indicated by a circle (O). When a vendor publication specifies a time outside the 50 / 100 hour cycle, it will be listed as a special inspection in 5-30-00.

(2) Refer to the applicable chapter of this manual for instructions on how to gain access to remove any item that must be removed and is not completely accessible.

(3) Inspection Report Forms.

To help in the performance of periodic inspections, an Inspection Report form:

P/N 767-008, PA-28-181, Piper Archer III (S/N's 2843001 & up).

is available. See the availability statement in Revision Status.

NOTE: Service centers conducting Part 91 Annual / 100 Hour Inspections can use the Inspection Report Form (P/N 767-008), as a working check-off list, provided they verify its currency against an up-to-date copy of the ICA (i.e. - this Maintenance Manual, see 4-00-00 and 5-20-00).

(4) In addition to inspection intervals required in scheduled maintenance (i.e. - paragraph 5), preflight inspection must also be performed.

(5) References to maintenance manual applicable areas are per the "chapter - system/sub-system" assignment of subject material numbering system.

B. Special Inspections (See 5-30-00.)

C. Unscheduled Maintenance (See 5-50-00.)

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5. Scheduled Maintenance

Refer to Notes 1, 2, 3, and 4 before performing the following inspections.

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
A. PROPELLER GROUP		
WARNING: PROPELLER MAY KICK BACK WHEN ROTATED BY HAND; USE EXTREME CAUTION. SWITCH OFF (GROUND) BOTH MAGNETOS BEFORE ROTATING PROPELLER. IF MAGNETO(S) ARE NOT GROUNDED, TURNING PROPELLER MAY START ENGINE.		
1. Inspect spinner and back plate for cracks, dents, missing screws, & security	O	O
2. Inspect blades for nicks and cracks	O	O
3. Inspect spinner mounting brackets for cracks and security		O
4. Inspect propeller mounting bolts for condition and security. If safety is broken, re-torque and safety		O
5. Inspect hub for cracks and corrosion		O
6. Inspect complete propeller and spinner assembly for security, chafing, cracks, deterioration, wear, and proper installation		O
B. ENGINE GROUP		
WARNING: GROUND MAGNETO PRIMARY CIRCUIT PRIOR TO BEGINNING ANY ENGINE WORK		
NOTE: Read Note 5 prior to beginning this inspection group.		
1. Remove engine cowling. Inspect for damage	O	O
2. Clean and inspect cowling for cracks, distortion, and loose or missing fasteners		O
3. Conduct required engine manufacturer's periodic inspections per latest revision of Lycoming's Operator's Manual, P/N 60297-12, Section 4.	O	O
4. Drain oil sump	O	O
5. Clean suction oil strainer at oil change. Inspect strainer for foreign particles	O	O
6. Clean pressure oil strainer or change full flow (cartridge type) oil filter element. Inspect strainer or element for foreign particles. (See Note 6.)	O	O
7. Inspect oil temperature sender unit for leaks and security		O
8. Inspect oil lines and fittings for leaks, security, chafing, dents, and cracks	O	O
9. Clean and inspect oil cooler cooling fins		O
10. Fill engine with oil per Lubrication Chart, 12-20-00.	O	O
CAUTION: DO NOT CONTAMINATE VACUUM PUMP WITH CLEANING FLUID.		
11. Clean engine		O
12. Test cylinder compression. (See Lycoming Service Instruction No. 1191A and FAA AC 43.13-1, latest revisions.)		O
13. Inspect ignition harness and insulators for high tension leakage and continuity		O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
B. ENGINE GROUP (CONT.)		
14. Inspect ignition harness for routing, condition, and security. (See Ignition Harness - Inspection - Routing, 74-20-00.)		O
15. Inspect magneto for oil seal leakage		O
16. Inspect magnetos to engine timing. (See Note 7.)		O
17. Inspect intake seals for leaks and clamps for tightness	O	O
18. Remove air filter and tap gently to remove dirt particles. Replace as required	O	O
19. If equipped, clean fuel injector servo inlet line screen and fuel strainer	O	O
20. If equipped, inspect carburetor ice detector probe for condition and security		O
21. Inspect condition of carburetor heat or alternate air door and induction airbox	O	O
22. If equipped, inspect fuel injectors for condition, security, and leaks	O	O
23. Inspect vent lines for evidence of fuel or oil seepage	O	O
24. Inspect flexible fuel lines for condition		O
25. Inspect fuel system for leaks	O	O
26. If equipped, drain carburetor and clean inlet line fuel strainer	O	O
27. Remove and clean fuel strainer (gascolator) bowl and screen	O	O
28. Drain and completely flush the fuel strainer (i.e., gascolator) and, if equipped, carburetor bowl		O
29. In airplanes with FACET fuel pumps only, clean electric fuel pump screen and inspect operation of pump	O	O
30. Inspect engine-driven and electric fuel pumps for condition and operation. Replace as required		O
31. If installed, inspect and operationally test vacuum pump(s) and lines. (See Notes 17 and 18.)		O
32. Inspect throttle, carburetor heat/alternate air, and mixture controls for security, travel, and operating conditions		O
33. Inspect exhaust stacks, connections, and gaskets. Replace gaskets as required	O	O
34. Inspect exhaust system per 78-00-00, Exhaust System, 100 Hour Inspection. (If optional low noise muffler is installed, see also 78-00-00, Optional Low Noise Muffler, 100 Hour Inspection.)		O
35. Inspect breather tube for obstructions and security		O
36. Inspect crankcase for cracks, leaks, and security of seam bolts		O
37. Inspect engine mounts for cracks and loose mountings. (See Note 14.)		O
38. Inspect rubber engine mount bushings for deterioration. Replace as required		O
39. Inspect all engine baffles		O
40. Inspect all lines, air ducts, electrical leads (wiring) and engine attachments for general condition (chafing, cracks or cracked insulation, deterioration), security, proper routing, and correct installation	O	O
41. Inspect firewall seals		O
42. Remove drive belt and turn alternator rotor by hand to inspect for abnormal noise or roughness in the bearings		O
43. Inspect alternator drive belt condition and tension. (See 24-30-00.)		O
44. Lubricate alternator idler pulley (if installed) per lubrication chart. Do not lubricate if sealed bearings are installed		O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
45. Inspect security and condition of alternator, its mounting, and electrical connections. (For airplanes equipped with a HET (aka Kelly, Electrosystems) Alternator, see 100 Hour Inspections under HET Alternators in 24-30-00.)		O
46. Inspect security & condition of starter, its mounting, & electrical connections		O
47. If installed, inspect Standby Alternator per 100 Hour Standby Alternator Inspection, 24-30-00.		O
48. If installed, inspect air conditioning compressor oil level. (See Note 10.)		O
49. If installed, inspect compressor belt condition and tension. (See 21-50-00.)		O
50. If installed, inspect compressor clutch security and wiring. (See Note 11.)		O
51. If installed, inspect security of compressor mounting		O
52. Check fluid in brake reservoir. Fill as required.	O	O
53. Lubricate all controls per Lubrication Chart, 12-20-00		O
54. Install engine cowling	O	O
C. CABIN AND COCKPIT GROUP		
1. Inspect cabin entrance, door, and windows for damage and operation		O
2. Inspect window sealants for cracks and deterioration. Reseal if necessary		O
3. Inspect upholstery, headliner and carpet for tears		O
4. Inspect seats and attaching brackets and hardware for condition, security, and operation		O
5. Inspect seat belts and shoulder harnesses per 25-10-00, Restraint System		O
6. Check trim operation		O
7. Inspect rudder pedals for operation and adjustment. (See Note 19.)		O
8. Ensure no hose clamps or cable ties are attached to the torque tubes of the rudder pedal and bar assembly. Verify free and correct movement of the rudder pedal and bar assembly.		O
9. Inspect parking brake and brake handle for operation and cylinder leaks		O
10. Inspect control wheels, column, chain sprockets, turnbuckles, pulleys, and cables for condition, security, and full travel. Inspect cables per Control Cable Inspection, 27-00-00		O
11. Inspect condition of flap control cable and cable-to-lever attachment bolt. Inspect cable per Control Cable Inspection, 27-00-00		O
12. If installed, inspect disposable-type (non-gauged) fire extinguisher minimum weight as specified on nameplate		O
13. Inspect landing, navigation, cabin, and instrument lights	O	O
14. Inspect instruments, lines, and attachments		O
15. If installed, inspect gyro operated instruments and electric turn and bank. (Overhaul or replace as required)		O
16. If installed, replace vacuum regulator filter element		O
17. Inspect static system, altimeter (and Avidyne or Garmin PFD, or Aspen EBD, if installed) and transponder for installation/certification per latest revision of AC43.13-1 and current test/inspection per FAR's 91.411 and 91.413, respectively		O
18. Cycle each circuit breaker with airplane power off	O	O
19. Check fuel selector valve operation.		O
20. Inspect condition of heater controls and ducts		O
21. Inspect air vents condition and operation		O
22. If installed, inspect condition of air conditioning ducts		O
23. If installed, remove and clean air conditioning evaporator filter.		O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
D. FUSELAGE AND EMPENNAGE GROUP		
1. Remove inspection plates and panels. (See Figure 3, 6-00-00.)		O
2. Inspect baggage door, latch, and hinges		O
3. Inspect battery condition and security. If installed, clean and recharge acid recovery (vent) jar. (See Battery, 24-30-00.)	O	O
4. Conduct a general visual inspection of electrical and electronic installations (mounting, wiring, harnesses, shields, connectors, etc.) for condition and security. For Garmin G1000/ G1000 NXi equipped aircraft, see the 100 hour Inspection in 34-25-01, 34-25-02, or 34-25-03 as appropriate		O
5. Inspect skins, bulkheads, frames, and stringers for damage, irregularities, or structural defects (i.e. - skin cracks, distortion, dents, corrosion, and loose or missing rivets)		O
6. Inspect condition and security of antenna mounts and electric wiring		O
7. If installed, inspect air conditioning system for refrigerant leaks. (See Note 10.)		O
8. If installed, inspect refrigerant level in sight gauge of receiver-dehydrator. (Refer to 21-50-00 and Note 10.)	O	O
9. If installed, inspect air conditioning condenser air scoop rigging. (See Note 15.)	O	O
10. Inspect fuel lines, valves, and gauges for damage and operation		O
11. Inspect security of all lines		O
12. Inspect vertical fin for surface damage or irregularities (i.e. - skin cracks, distortion, dents, and corrosion); structural defects (i.e. - loose or missing rivets); misrigging; and attachment points for missing or worn hardware		O
13. Inspect vertical fin attachments		O
14. Inspect rudder for surface damage or irregularities (i.e. - skin cracks, distortion, dents, and corrosion); structural defects (i.e. - loose or missing rivets); misrigging; hinge damage, excessive wear, freedom of movement and proper lubrication; and attachment points for missing or worn hardware		O
15. Inspect rudder control stops. Verify stops are not loose and locknuts are tight		O
16. Inspect rudder hinge bolts for excess wear. Replace as required		O
17. Inspect stabilator and tab for surface damage or irregularities (i.e. - skin cracks, distortion, dents, and corrosion); structural defects (i.e. - loose or missing rivets); misrigging; hinge damage, excessive wear, freedom of movement and proper lubrication; and attachment points for missing or worn hardware. (See Stabilator Attach Fittings Corrosion Inspection, 55-20-00.)		O
18. Inspect stabilator control stops, verify stops are not loose and locknuts are tight		O
19. Inspect stabilator balance weight for condition and security		O
20. Inspect stabilator and tab hinge bolts and bearings for excess wear. Replace as required		O
21. Inspect stabilator trim mechanism		O
22. Clean and lubricate stabilator trim drum screw		O
23. Inspect aft wing attach fittings per Aft Wing Attach Fittings 100 Hour Inspection, 53-20-00		O
24. Inspect aileron, rudder, stabilator, and stabilator trim cables; and cable terminals, turnbuckles, guides, fittings, and pulleys for safety, condition, and operation per Control Cable Inspection, 27-00-00		O
25. Inspect rudder, stabilator, and stabilator trim cable tension per 27-00-00, Chart 2. Use a tensiometer.		O
26. Lubricate per Lubrication Chart, 12-20-00	O	O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
27. Inspect anti-collision light for security and operation.		O
28. If installed, inspect security of Autopilot bridle cables and clamps. Inspect condition of cables per Control Cable Inspection, 27-00-00. (See Note 12.)		O
29. Inspect all air ducts, electrical leads, lines, radio antenna leads, & attaching parts for security, routing, chafing, deterioration, wear, & proper installation		O
30. Inspect ELT battery mount for condition and security		O
31. Inspect ELT antenna for condition, security, and operation (test operation per Antenna Test in 25-60-00). Replace antenna if bent or damaged		O
32. Install inspection plates and panels		O
33. Inspect static wicks per Static Wicks, Inspection, 23-60-00. (Replace as required.)		O
E. WING GROUP		
1. Remove inspection plates and panels. (See Figure 3, 6-00-00.) (See Note 16.)		O
2. Inspect exterior surfaces, skins, and tips for damage and loose or missing fasteners (and condition of walk-way as applicable)	O	O
3. Inspect visible interior structural components (skins, spars, ribs, stringers, etc.) for condition, security, distortion, or failure		O
4. Inspect wing carry-through structure for condition, security, distortion, or failure		O
5. Inspect tip light shield for cracks, bonds, corrosion, or other damage.		O
6. Inspect ailerons for surface damage or irregularities (i.e. - skin cracks, distortion, dents, and corrosion); structural defects (i.e. - loose or missing rivets); misrigging; hinge damage, excessive wear, freedom of movement and proper lubrication; and attachment points for missing or worn hardware		O
7. Inspect aileron control stops, verify stops are not loose and locknuts are tight		O
8. Inspect aileron cables and cable terminals, turnbuckles, fittings, guides, pulleys, and bellcranks for safety, condition, and operation per Control Cable Inspection, 27-00-00		O
9. Inspect aileron cable tension per 27-00-00, Chart 2. Use a tensiometer		O
10. Inspect flaps for surface damage or irregularities (i.e. - skin cracks, distortion, dents, and corrosion); structural defects (i.e. - loose or missing rivets); misrigging; hinge damage, excessive wear, freedom of movement and proper lubrication; and attachment points for missing or worn hardware		O
11. Inspect condition of bolts used with hinges. Replace as required		O
12. Lubricate per Lubrication Chart, 12-20-00	O	O
13. Inspect wing fore and aft attach fittings, and bolts, for security, corrosion and condition		O
14. Retorque wing aft spar attach bolts per Wing Aft Spar-to-Fuselage Attachment Hardware 100 Hour Inspection, 57-40-00		O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
E. WING GROUP (cont.)		
15. Inspect fuel tanks and lines for leaks and water contamination		O
16. Inspect fuel caps, cap gaskets, fuel filler neck sealer, fuel gage transmitter gaskets, and gage transmitter access covers for condition, proper sealing, security, alignment, etc. Ensure to service and clean these areas, replacing parts as necessary		O
17. Inspect the interior of metal fuel tanks for signs of corrosion, which may indicate water contamination. If signs of contamination are found, alert the owner and fuel supplier of your findings for corrective action		O
18. Fuel tanks marked for minimum octane rating		O
19. Inspect all air ducts, electrical leads, lines, and attaching parts for security, routing, chafing, deterioration, wear, and proper installation		O
20. Install inspection plates and panels		O
21. Inspect static wicks per Static Wicks, Inspection, 23-60-00. (Replace as required.)		O
F. LANDING GEAR GROUP		
1. Inspect oleo struts for proper extension and evidence of fluid leakage. See Landing Gear, 12-10-00	O	O
2. Inspect nose gear steering control and travel		O
3. Inspect wheels for alignment		O
4. Put airplane on jacks		O
5. Inspect tires for cuts, uneven or excessive wear, and slippage		O
6. Remove wheels, clean, inspect, and repack bearings		O
7. Inspect wheels for cracks, corrosion, and broken bolts		O
8. Inspect tire pressure per Chart 1, 6-00-00	O	O
9. Inspect brake linings and discs for condition and wear		O
10. Inspect brake backing plates for condition and wear		O
11. Inspect hydraulic lines, electrical leads, and attaching parts for condition and security (i.e. - routing, chafing, damage, wear, etc.)		O
12. Inspect shimmy dampener		O
13. Inspect gear forks for damage		O
14. Inspect oleo struts for fluid leaks and scoring		O
15. Inspect gear struts and mounting bolts for condition and security		O
16. Inspect torque links for cracks, bolts for condition and security		O
17. Inspect torque link assembly for excessive side play		O
18. Lubricate per Lubrication Chart, 12-20-00	O	O
19. Remove airplane from jacks		O
G. SPECIAL INSPECTIONS		
Review inspections in 5-30-00. Perform all special inspections applicable to your aircraft and currently due per the given inspection interval.	O	O

5. Scheduled Maintenance (continued)

NATURE OF INSPECTION	Inspection Interval (Hrs) 50	100
H. OPERATIONAL INSPECTION		
1. Inspect fuel pump and fuel tank selector	O	O
2. Inspect fuel quantity, pressure and flow readings	O	O
3. Inspect oil pressure and temperature	O	O
4. Inspect alternator output	O	O
5. Inspect carburetor heat / alternate air	O	O
6. Inspect parking brake	O	O
7. If installed, Inspect operation of auxiliary vacuum pump system	O	O
8. If installed, Inspect vacuum gauge	O	O
9. If installed, Inspect gyros for noise and roughness	O	O
10. Inspect cabin heater operation	O	O
11. Inspect magneto switch operation	O	O
12. Inspect magneto rpm variation	O	O
13. Inspect throttle and mixture operation	O	O
14. Inspect propeller smoothness	O	O
15. Perform maximum power static rpm check per 71-00-00	O	O
16. Inspect engine idle	O	O
17. Inspect electronic equipment operation	O	O
18. If installed, Inspect air conditioner compressor clutch operation	O	O
19. If installed, Inspect air conditioner condenser scoop operation	O	O
20. If installed, Inspect operation of autopilot, including automatic pitch trim, and manual electric trim. (See Note 13.)	O	O
21. Inspect free and correct operation of flight controls	O	O
I. GENERAL		
1. Aircraft conforms to FAA Specifications	O	O
2. Latest revision of applicable FAA Airworthiness Directives complied with	O	O
3. Current and correct Pilot's Operating Handbook (POH) is in the airplane	O	O
4. Inspect airplane for required placards as specified in Section 2 of the POH		O
5. Appropriate entries made in the Aircraft and Engine Log books	O	O
6. Airworthiness & Registration Certificates in the aircraft and properly displayed	O	O
7. Aircraft Equipment List, Weight and Balance and FAA Form(s) 337 (if applicable) are in the aircraft and in proper order	O	O
8. Operational inspection and run-up completed	O	O
9. Aircraft cleaned and lubricated after wash (as required)	O	O

5. Scheduled Maintenance (continued)

J. NOTES

1. Refer to Piper's Customer Service Information File P/N 1753-755 (available online at http://www.piper.com/technical-publications-documents) for latest revision dates to Piper Inspection Reports/Manuals and this maintenance manual. References to Chapter/Section are to the appropriate Chapter/Section in this manual.

   WARNING: INSTRUCTIONS FOR CONTINUED AIRWORTHINESS (ICA) FOR ALL NON-PIPER APPROVED STC INSTALLATIONS ARE NOT INCLUDED IN THIS MANUAL. WHEN A NON-PIPER APPROVED STC INSTALLATION IS INCORPORATED ON THE AIRPLANE, THOSE PORTIONS OF THE AIRPLANE AFFECTED BY THE INSTALLATION MUST BE INSPECTED IN ACCORDANCE WITH THE ICA PUBLISHED BY THE OWNER OF THE STC. SINCE NON-PIPER APPROVED STC INSTALLATIONS MAY CHANGE SYSTEMS INTERFACE, OPERATING CHARACTERISTICS AND COMPONENT LOADS OR STRESSES ON ADJACENT STRUCTURES, THE PIPER PROVIDED ICA MAY NOT BE VALID FOR AIRPLANES SO MODIFIED.

2. Inspections or operations are to be performed as indicated by a "O" at the 50 or 100 hour inspection interval. Inspections or operations (i.e., component overhauls/replacements, etc.) required outside the 100 hour cycle are listed as special inspections in 5-30-00. Inspections must be accomplished by persons authorized by the FAA or appropriate National Aviation Authority. Checks may be performed by a pilot or owner who is checked out on the airplane.
   (a) The 50 hour inspection accomplishes preventive maintenance, lubrication and servicing as well as inspecting critical components.
   (b) The 100 hour inspection is a complete inspection of the airplane, identical to an annual inspection.

   NOTE: A log book entry should be made upon completion of any inspections.

3. Piper Service Bulletins are of special importance and Piper considers compliance mandatory. In all cases, see Service Bulletin/Service Letter Index P/N 762-332 (available online at http://www.piper.com/technical-publications-documents) to verify latest revision. See also "Chart 1" on page 052017.
4. Piper Service Letters are product improvements and service hints pertaining to servicing the airplane and should be given careful attention
5. Inspections given for the powerplant are based on the engine manufacturer's manuals (Lycoming's Operator's Manual, P/N 60297-12). Any changes issued to the engine manufacturer's manuals shall supersede or supplement the inspections outlined in this report.
6. See latest revision of Lycoming Service Bulletin No. 480.
7. Inspect magneto to engine timing per Slick's F-1100 Master Service Manual. If that publication is not available, use the procedures in 74-10-00. Overhaul or replace magnetos as required or at engine overhaul. For Slick 4300 Series magnetos, in no case may time-in-service exceed engine TBO.
8. Not used.
9. Not used.

5. Scheduled Maintenance (continued)

J. NOTES (cont.)

CAUTION: ENVIRONMENTAL REGULATIONS MAY REQUIRE SPECIAL EQUIPMENT AND PROCEDURES BE UTILIZED WHEN EVACUATING OR CHARGING AIR CONDITIONING SYSTEM.

10. Compressor oil level should not be checked unless a refrigerant leak has occurred, requiring an addition of refrigerant to the system.
11. Clean any traces of oil from clutch surface.
12. Check security and condition of autopilot servo bridle cables, clamps, and sheer pin.
13. Refer to Pilot's Operating Handbook for preflight and flight check for intended function in all modes.
14. For new airplanes, at the first 100 hour inspection, while the airplane is on jacks, retorque the engine mount-to-fuselage bolts.
15. When inspecting air conditioning condenser scoop rigging, pay particular attention to adjustment of the micro-switch. See 21-50-00, Condenser Assembly Rigging.
16. In S/N's 2843001 thru 2843546 only, for airplanes which have not installed Piper Kit No. 767-363, perform 100 Hour W.S. 36.22 R.H. Wing Rib Inspection, 57-10-00.
17. For airplanes equipped with Aero Accessories engine-driven vacuum pumps, for pumps which have accumulated 500 hours or more time-in-service, inspect vacuum pump vane wear per Vacuum Pump(s), Inspection, 37-10-00.
18. For airplanes equipped with Parker Hannifin / Airborne vacuum pump(s), verify compliance with Parker Hannifin / Airborne Service Letter No. 72.
19. In airplanes with a rudder pedal and bar assembly which has accumulated 1,200 hours or more time-in-service, each 100 hours time-in-service: conduct the Rudder and Steering Pedal Assembly Inspection in 27-20-00.

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6. Service Publications List

Chart 1 is a cumulative list of Piper service publications (i.e. - Service Bulletins, Service Letters, and Vendor Service Publications) applicable to the airplane model covered by this manual, with the following exceptions:

A. The following service publications have been incorporated into this manual and are not listed below: Service Bulletins 856, 978A, 990, 1006, 1048, 1139A, 1242, 1244C, 1245A; Service Letters 1041, 1052, 1066, 1069, 1074, 1081, 1087, 1106A, 1116A, 1124, 1135, 1141, 1153, 1161, 1229; and Vendor Service Publication 204.

B. Nor are service publications which have become obsolete.

Kits are listed when installation of that single kit indicates compliance with the associated service publication. Kits listed may be no longer available or may have been replaced.

Effectivity is listed by airplane model and year. See the individual service publication for specific serial number applicability.

CHART 1 (Sheet 1 of 3)
SERVICE PUBLICATIONS LIST

Model	Model Year	Pub No.	Kit No.	Subject
PA-28-181	1995	SB 992		Replacement of Spinner & Addition of New Forward Spinner Bulkhead
		SB 996		Missing Rivet in Upper Door Frame, Forward and Aft
	1995 - 1997	SB 1005		Addition of Drain Holes to Induction Air Inlet Scoops
		SL 1019A		Alternator Support Bracket Improvement
	1995 - 1998	SB 1019		Engine Oil Cooler Inspection
		SB 1020		ARTEX 110-4 ELT and ELT Attachment Bracket
		SB 1022		Inspection of Induction Air Filter, Distribution of Purolator SB090298.01
		SB 1026	753-910	Dry Air Pump Flexible Couplings, Inspection/Replacement, Parker Hannifin Airborne SL No. 48
	1995 - 1999	SB 1040		Flight Control Wheel Collar Retainer Mod
		SB 1041		Airborne Air Filter Elements Inspection and Cleaning, Airborne SL No. 56
		SB 1050		Engine Baffle Assemblies, Installation of Longer Screws
		SB 1079		Untested Fuel Pump
		SL 1024	766-666	Induction Airbox Air Valve Bushing Improvement
		SL 1028A		Induction Airbox Inspection/Repair, Inlet Air Duct Replacement
		SL 1033		Air Conditioning Hose Replacement
	1995 - 2001	SB 1057	766-681	Low Noise Muffler Replacement
		SB 1104		Air Box Interference
	1995 - 2002	VSP 142		Garmin SB 0204 - New GPS Software (Ver. 2.11)
		VSP 153		Lycoming SB 555 - Oil Pump Body Plug Inspection and Supplement No. 2

CHART 1 (Sheet 2 of 3)
SERVICE PUBLICATIONS LIST

Model	Model Year	Pub No.	Kit No.	Subject
PA-28-181	1995 - 2004	SB 1153		Overhead Switch Panel Assembly Modifications
(cont.)		SB 1167	767-519	Nose Gear Centering Springs
	1995 - 2007	VSP 184		Sky-Tec SB 07-01 PM Model Starter Replacement
	1995 - 2008	SB 1197D		Control Wheel Shaft Inspection
	1995 - 2010	SL 1125	88472 002	Stall Horn Bracket Replacement
	1995 - 2012	SB 1253		Firewall Inspection
	1995 - 2015	SB 1280A		H3R Fire Extinguisher Replacement
		VSP 249		Hartzell SB062 Alternator Bolt Replacement
	1995 - up	SB 1284B		Lycoming SB No. 622A Slick Magneto Replacement
		SB 1297		Altimeter Inspection / Replacement (United Instruments SB No. 13)
		SB 1309		Fuel Tank Selector Placards Inspection
		SL 1093B		Electrical Contactor Replacement Kit
		SL 1218A		Crew Seat Control Cable Modification
		VSP 116B		Lycoming Service Instr. 1492D - Piston Pin Plug Wear Inspection
		VSP 174		Lycoming Service Alert 569 - IO 360
		VSP 196		Unison Service Bulletin No. SB2-08
		VSP 197A		Unison Service Bulletin No. SB3-08A
		VSP 244A		Lycoming Service Letter SL L-264 Champion/Slick Magnetos
	1996 - 2015	SB 1279A		Wing Rib Inspection at Right Side WS 140.09
	1998 - 2002	SB 1135		GMA 340 Audio Panel Wiring
	1999	SB 1036		Garmin Service Bulletin No. 9905, “GNS430 Mod 1”
		SB 1037		Inspection for Gascolator Bowl Drain Security
	1999 - up	VSP 191		Avidyne Service Alert 08-001 - Avidyne PFD
	2001	SB 1083C	767-316	S-TEC System 55X Autopilot
	2001 - 2002	SB 1122		Wing Rib Inspection
	2002 - 2003	SB 1137		Stabilator Balance Weight Tube Assembly
	2003	SB 1141		Clip Nut Replacement
	2003 - 2004	SB 1278		Left Wing Station 131.60 Rib Inspection
	2004	SB 1151E		4130N Steel Hardness Discrepancy
	2004 - 2007	SL 1103	88414 002	Attitude Indicator Standby Battery Failure
	2004 - 2013	SL 1126A		Voltage Suppressor Advisory
	2004 - up	SB 1162B		European Requirements for Avidyne Entegra Equipped PA-28 Models
		SL 1123		Avidyne EXP5000 Software Upgrade
		SL 1089		Aircraft Painting Requirement
		VSP 172		Avidyne Service Alert 05-001- Avidyne PFD
		VSP 182		Avidyne Service Bulletin 601-00006-067 - Avidyne PFD (Supercedes VSP 172)
		VSP 183		Avidyne Service Bulletin 601-00006-075 - Avidyne PFD w/ Garmin 400W

CHART 1 (Sheet 3 of 3)
SERVICE PUBLICATIONS LIST

Model	Model Year	Pub No.	Kit No.	Subject
PA-28-181	2005 - up	VSP 175		S-TEC SB 06-001 - Pitch Servo Inspection
(cont.)	2006	SB 1179		Main Landing Gear Cylinder Assembly Inspection
	2006 - 2007	SB 1187		Voltage Suppressor Replacement
	2010	SB 1223		Wire Screen Replacement
		VSP 249		Fuel Calculator
	2010 - up	VSP 219		Garmin Service Alert 1047A - LNAV/NAV and LNAV+V Full Scale Deflection
	2012 - 2015	SL 1215		Garmin G1000 Software Update Version 1618.05
	2013 - 2017	SB 1302B		Rear Seat Inspection
	2014 - 2015	SB 1274		Control Wheel Bushing Inspection
		SB 1276A		Left Wing Root Rib Inspection
	2015 - 2017	SB 1307		Plane-Power Alternator Mounting Hardware Replacement
		SB 1316	88619-001	Plane-Power Alternator Idler Pulley Bracket Improvement
	2015 - 2018	SB 1323		Replacement of the Fuel Fitting Injection Servo Vapor Return Fitting
	2016 - 2017	SB 1324	85470-005	Winterization Kit Replacement for Fuel Injected PA-28-181 Archer III
		VSP 248		Aspen Avionics Operation Bulletin 2015-01, Aspen Avionics Software Update 2.6.3
	2016 - up	VSP 258		Garmin Service Alert 1711, Fuel Calculator
	2017	SB 1311		Baggage Compartment Closeout Inspection
	2018	SB 1313A		Stabilator Trim Pulley Bracket Inspection

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SPECIAL INSPECTIONS

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The following inspections are in addition to those listed in 5-20-00. These inspections are performed at intervals of:

- ✈ Flight hours;
- ✈ calendar year; or
- ✈ the specific operation being conducted or the environment being operated in.

Unless otherwise indicated, these inspections are to be repeated at each occurrence of the specified interval. Note that the items listed herein are guidelines based on past operating experience. Each operator should closely monitor his own unique operating conditions/environment and react accordingly to keep his aircraft airworthy.

NOTE: A log book entry should be made upon completion of any inspections.

1. Per Flight Hour

A. First 25 Hours

[ ] For airplanes equipped with a new, rebuilt, or overhauled Lycoming engine, conduct required engine manufacturer's inspection per latest revision of Lycoming's Operator's Manual, P/N 60297-12, Section 4.

B. Each 200 Hours

[ ] For airplanes with wing flap(s) which have accumulated ten (10) years time-in-service, conduct the following special inspection each 200 hours: Inspect the interior of the wing flap for evidence of dissimilar metal corrosion where aluminum sheet metal is in contact with steel flap brackets. Use a bore scope or other suitable tool. Installation of a new wing flap will relieve this inspection requirement until such time as the replacement wing flap reaches ten (10) years time-in-service.

C. Each 300 Hours

[ ] For airplanes equipped with the optional low noise muffler, each 300 hours time-in-service or annually, whichever comes first, conduct the Optional Low Noise Muffler 300 Hour Inspection in 78-00-00.

D. Each 400 Hours

[ ] For airplanes with a Lycoming engine, conduct required engine manufacturer's inspection per latest revision of Lycoming's Operator's Manual, P/N 60297-12, Section 4.

E. Each 500 Hours

[ ] (1) If installed, the auxiliary vacuum pump/motor assembly must be removed from service and replaced at 500 hours operating time as indicated on the elapsed time indicator, or at 10 years of installed time in the aircraft, whichever comes first.

[ ] (2) For airplanes equipped with Slick Magnetos: inspect and clean magneto(s) per the appropriate 500 Hour Inspection in the Slick F1100 Master Service Manual.

[ ] (3) In airplanes equipped with the SureSTART Magento Booster System, perform the 500 Hour Inspection found under SureSTART Magento Booster in 74-10-00.

[ ] (4) Replace induction air filter each 500 hours time-in-service, annually, or after 10 cleanings, whichever comes first.

[ ] (5) If installed, replace the vacuum system inlet air filter (i.e., central air filter, gyro filter, etc.) element each 500 hours time-in-service, annually, or at vacuum pump replacement, whichever comes first.

[ ] (6) Lubricate per 500 hour requirements in Lubrication Charts, 12-20-00.

[ ] (7) For airplanes equipped with Garmin G1000/G1000 NXi: each 500 hours time-in-service conduct the 500 Hour Inspection as specified in 34-25-01, 34-25-02, or 34-25-03, as appropriate.

[ ] (8) For airplanes equipped with a HET (formerly Kelly, Electrosystems) Alternator, each 500 hours time-in-service or every two (2) years, whichever comes first: perform the 500 hour inspection under HET Alternators in 24-30-00.

[ ] (9) In airplanes equipped with Concorde RG Series batteries and operating 1,000 hours per year or more and battery capacity was between 85% and 90% at the last check, conduct subsequent checks every 500 hours in service (± 100 hours). If battery capacity is less than 85%, remove battery from service.

F. First 1,000 Hours

[ ] In airplanes equipped with Concorde RG Series batteries and operating 1,000 hours per year or more, conduct the initial battery capacity check 1,000 hours after initial installation (± 100 hours).

G. Each 1,000 Hours

[ ] (1) Replace engine compartment flexible fuel and oil hoses as required; but not to exceed 1,000 hours time-in-service, eight (8) years, or engine overhaul, whichever comes first; except for TSO-C53a - Type D hoses which are replaced on-condition.

[ ] (2) If installed, replace A/C compressor belt each 1,000 hours time-in-service, or three (3) years, whichever comes first.

[ ] (3) Replacement of the standard muffler is recommended at or near 1,000 hours time-in-service. (See 78-00-00.)

[ ] (4) For airplanes equipped with Garmin G1000/G1000 NXi: each 1,000 hours time-in-service conduct the 1,000 Hour Inspection as specified in 34-25-01, 34-25-02, or 34-25-03 as appropriate.

[ ] (5) For airplanes equipped with a Plane-Power alternator, each 1,000 hours time-in-service or every five (5) years, whichever comes first: remove the field brush assembly and inspect brushes for excess wear. Replace brush assembly if brushes extend less than 0.250 inch from the edge of the brush holder case.

G. Each 1,000 Hours (cont.)

[ ] (6) In airplanes equipped with Concorde RG Series batteries and operating 1,000 hours per year or more and battery capacity was above 90% at the last check, conduct subsequent capacity checks every 1,000 hours in service (± 100 hours).

H. Each 1,600 Hours

[ ] (1) For airplanes in Usage Class "B" (see 57-10-00), which have accumulated 3,700 hours total time-in-service or 3,700 hours factored time-in-service; conduct Wing Spar Inspection, 57-10-00.

[ ] (2) For airplanes in Usage Class "C" (see 57-10-00); conduct Wing Spar Inspection, 57-10-00.

I. Each 2,000 Hours

[ ] (1) Each 2,000 hours or seven (7) years, whichever occurs first, remove interior cabinets, panels, and headliner and conduct detailed inspection of aircraft structure (skin, bulkheads, stringers, etc.) for condition and security. Inspection of structure concealed by headliner may be accomplished by alternate means (i.e. – through the use of a bore scope) without removing the headliner, providing access is obtained to all concealed areas and bore scope provides sufficient detail to adequately accomplish the inspection. At F.S. 73, examine the two steel Upper Forward Cockpit Fittings (one each side), P/N's 79553-000/-001 or 62522-000/ 68257-000, as applicable, for signs of corrosion.

[ ] (2) Each 2,000 hours, or as specified in the latest revision of Lycoming Service Instruction No. 1009, overhaul or replace engine.

[ ] (3) For airplanes equipped with Hartzell Engine Technologies (HET) Alternators (formerly Kelly Aerospace or Eelctrosystems), each 2,000 hours or twelve (12) years, or at engine overhaul, whichever comes first; overhaul or replace alternator(s).

[ ] (4) If installed, each 2,000 hours time-in-service or at engine overhaul, whichever comes first, replace the Standby Alternator with a new or overhauled unit.

[ ] (5) Recommended Time-Between-Overhaul (TBO) / Reconditioning of Sensenich fixed-pitch metal propellers is 2,000 hours, if propeller does not receive damage requiring immediate attention. Airplanes in flight school operations or operating from unpaved or poorly maintained runways may expose the propeller to increased foreign object damage which will require a shorter interval between overhauls. Reconditioning is removal of fatigued surface metal and accumulated small nicks too numerous to repair individually. Contact a Sensenich factory approved repair station. (Refer to latest revision of Sensenich Service Bulletin No. R17.)

[ ] (6) Each 2,000 operating hours or seven (7) calendar years time-in-service, whichever occurs first, inspect the aft wing attach fittings and adjacent structure for corrosion for signs of corrosion as specified in 57-40-00, Wing Aft Attach Fitting Inspection.

[ ] (7) In airplanes that have accumulated fifteen (15) calendar years time-in-service or more, each two thousand (2,000) hours or seven (7) calendar years time-in-service, whichever occurs first, perform the Stabilator Control System Special Inspection in 27-00-00.

J. Each 2,200 Hours

[ ] For airplanes equipped with Aspen Avionics EBD: replace the internal back-up battery in the EBD every three (3) years or 2,200 hours, or if it fails the operational test.

K. Each 2,700 Hours

[ ] For airplanes equipped with a Sky-Tec 149-NL/EC starter, overhaul or replace starter each 2700 hours time-in-service.

L. Each 6,000 Hours

[ ] For airplanes in Usage Class "A" (see 57-10-00), which have accumulated 62,900 hours total time-in-service; conduct Wing Spar Inspection as described in 57-10-00.

2. Per Calendar Year

A. Each Thirty (30) Days

NOTE: Pilots/owners may perform and sign-off the following checks.

[ ] (1) If installed, the pilot should check non-gauged disposable fire extinguisher for condition and charge. Verify nozzle is unobstructed and safety seal is intact. Determine charge by "hefting" extinguisher.

[ ] (2) If installed, the pilot should check gauged rechargeable fire extinguisher for condition and charge. Check to see that the extinguisher is undamaged; that the nozzle is unobstructed; that the gauge pressure is in the operable (green) range; that the lockpin and tamper seal is in place, and that the operating instructions are clearly visible.

B. Each Ninety (90) Days

[ ] (1) For airplanes equipped with Avidyne Entegra: if the Standby Attitude Indicator has not been operated in the previous 90 days, charge the battery of the Standby Attitude Indicator. See Standby Attitude Indicator, 34-22-00.

[ ] (2) Inspect battery, box and cables at least every 90 days. Flush box as required and fill battery per instructions on box. (Not required if maintenance-free battery installed.)

[ ] (3) Remove, drain, and clean fuel strainer bowl and screen.

[ ] (4) Operationally test ELT per the Test procedure in 25-60-00 for the appropriate installed ELT at least every 90 days.

C. Each Four (4) Months

[ ] Change the engine oil and full-flow cartridge oil filter each four (4) months or every 50 hours time-in-service, whichever comes first. Fill engine with oil per Lubrication Chart in 12-20-00.

D. Each Six (6) Months

[ ] In airplanes equipped with Concorde RG Series batteries and operating less than 1,000 hours per year and battery capacity was between 85 and 90% at the last capacity check, conduct subsequent checks every six months in service (± 1 month). If battery capacity is less than 85%, remove battery from service.

E. First Twelve (12) Months

[ ] In airplanes equipped with Concorde RG Series batteries and operating less than 1,000 hours per year, conduct the initial battery capacity check at 12 months after initial installation (± 1 month).

F. Each Twelve (12) Months

[ ] (1) For airplanes equipped with the optional low noise muffler, each 300 hours time-in-service or annually, whichever comes first, conduct the Optional Low Noise Muffler 300 Hour Inspection in 78-00-00.

[ ] (2) For airplanes equipped with the Avidyne Entegra IFDS: each twelve months, perform a full capacity test of the Standby Attitude Indicator battery. See Standby Attitude Indicator, 34-20-00.

D. Each Twelve (12) Months (cont.)

[ ] (3) Replace induction air filter each 500 hours time-in-service, annually, or after 10 cleanings, whichever comes first.

[ ] (4) If installed, replace the vacuum system inlet air filter (i.e., central air filter, gyro filter, etc.) element each 500 hours time-in-service, annually, or at vacuum pump replacement, whichever comes first.

[ ] (5) For vacuum system equipped airplanes, beginning at five years from date of check valve manifold manufacture and each twelve (12) months thereafter, inspect the Airborne 1H5 series check valve manifolds per the latest revision of Airborne SL 39A.

[ ] (6) For airplanes equipped with Garmin G1000/G1000 NXi: each twelve (12) months conduct the Annual Inspection as specified in 34-25-01, 34-25-02, or 34-25-03 as appropriate.

[ ] (7) For airplanes equipped with Aspen Avionics EBD: every 12 months inspect per the annual inspection in 34-24-00 under Aspen Avionics EBD Standby Display, Inspections.

[ ] (8) For airplanes equipped with Garmin G5 EFI: every 12 months inspect per the annual inspection in 34-24-01 under Garmin G5 EFI Standby Display, Inspections.

[ ] (9) If a rechargeable (gauged) fire extinguisher is installed, perform annual maintenance per “Gauged, Class 2B:C Fire Extinguisher Model No. A344T” in 26-20-00.

[ ] (10) In airplanes equipped with the Angle of Attack (AoA) System, perform the Ground Functional Test on page 273020.

[ ] (11) In airplanes equipped with Concorde RG Series batteries and operating less than 1,000 hours per year and battery capacity was above 90% at the last capacity check, conduct subsequent capacity checks every 12 months in service (± 1 month).

G. Each Two (2) Years

[ ] (1) Test and inspect the static pressure system and altimeters (including AHRS (in Avidyne PFD), or GDC 74A or GSU 75 (Garmin), and Aspen EBD or Garmin G5 EFI, if installed). Ensure compliance with the requirements of FAR 43, Appendix E. (See FAR 91.411.)

NOTE: For airplanes equipped with Aspen EBD, see also 34-24-00, Aspen Avionics EBD Standby Display, Inspections.

NOTE: For airplanes equipped with Garmin G1000/G1000 NXi Integrated Avionics System, see GDC 74A (or GSU 75) Tests in 34-25-01 (34-25-02 or 34-25-03), as appropriate, before beginning the above test and inspection.

[ ] (2) Test and inspect the transponder. Ensure compliance with the requirements of FAR 43, Appendix F. (See FAR 91.413.)

[ ] (3) Swing the magnetic compass and, if equipped, recalibrate the magnetometer/flux detector every two (2) years. See Magnetic Heading Systems, 34-20-00 and for airplanes equipped with Garmin G1000/G1000 NXi: see GRS/GMU Calibration Procedures in 34-25-01 or GSU/GMU Calibration Procedures in 34-25-02 or 34-25-03, as appropriate.

[ ] (4) For airplanes equipped with a HET (formerly Kelly, Electrosystems) Alternator, each 500 hours time-in-service or every two (2) years, whichever comes first: perform the 500 hour inspection under HET Alternators in 24-30-00.

H. Each Three (3) Years

[ ] (1) If installed, replace A/C compressor belt each 1,000 hours time-in-service, or three (3) years, whichever comes first.

[ ] (2) For airplanes equipped with Avidyne Entegra: replace the Standby Attitude Indicator battery as required, but at least every three (3) years. See Standby Attitude Indicator, 34-22-00.

[ ] (3) For airplanes equipped with Aspen Avionics EBD: replace the internal back-up battery in the EFD1000 every 3 years or 2,200 hours, or if it fails the operational test.

I. Each Five (5) Years

[ ] (1) For airplanes equipped with Garmin G1000/G1000 NXi Integrated Avionics System: update the GRS 77 AHRS (or GSU 75 ADAHRS) "Earth magnetic field model" each five (5) calendar years. The updated "Earth magnetic field model" is available from Garmin on July 1st of each fifth year - i.e., 2020, 2025, etc.

[ ] (2) For airplanes equipped with a Plane-Power alternator, each 1,000 hours time-in-service or every five (5) years, whichever comes first: remove the field brush assembly and inspect brushes for excess wear. Replace brush assembly if brushes extend less than 0.250 inch from the edge of the brush holder case.

[ ] (3) Replace the CO Detector with a new or recalibrated unit every five (5) years.

J. Each Six (6) Years

[ ] (1) For airplanes equipped with Aero Accessories, Inc. vacuum pump(s), replace the shear coupling each six (6) years time-in-service.

[ ] (2) If installed, completely discharge the rechargeable (gauged) fire extinguisher and have it professionally inspected and recharged each six years time-in-service.

K. Each Seven (7) Years

[ ] (1) Each seven (7) years time-in-service, drain and remove the metal fuel tank from each wing and inspect for corrosion as specified in Fuel Tank/Wing Spar Corrosion Inspection, 28-10-00.

[ ] (2) Each 2,000 hours or seven (7) years, whichever occurs first, remove interior cabinets, panels, and headliner and conduct detailed inspection of aircraft structure (skin, bulkheads, stringers, etc.) for condition and security. Inspection of structure concealed by headliner may be accomplished by alternate means (i.e. – through the use of a bore scope) without removing the headliner, providing access is obtained to all concealed areas and bore scope provides sufficient detail to adequately accomplish the inspection. At F.S. 73, examine the two steel Upper Forward Cockpit Fittings (one each side), P/N's 79553-000/-001 or 62522-000/68257-000, as applicable, for signs of corrosion.

[ ] (3) Each 2,000 hours or seven (7) calendar years time-in-service, whichever occurs first, inspect the aft wing attach fittings and adjacent structure for signs of corrosion as specified in 57-40-00, Wing Aft Attach Fitting Inspection.

[ ] (4) In airplanes that have accumulated fifteen (15) calendar years time-in-service or more, each two thousand (2,000) hours or seven (7) calendar years time-in-service, whichever occurs first, perform the Stabilator Control System Special Inspection in 27-00-00.

L. Each Eight (8) Years

[ ] Replace engine compartment flexible fuel and oil hoses as required; but not to exceed 1,000 hours time-in-service, eight (8) years, or engine overhaul, whichever comes first; except for TSO-C53a - Type D hoses which are replaced on-condition.

M. Each Ten (10) Years

[ ] (1) Each ten (10) years time-in-service, test fuselage and wing fluid hoses to system pressure. Visually inspect for leaks. Hoses that pass inspection may remain in service, but must be rechecked each five (5) years additional time-in-service. No fluid hose may exceed twenty (20) years total time-in-service.

[ ] (2) If installed, the auxiliary vacuum pump/motor assembly must be removed from service and replaced at 500 hours operating time as indicated on the elapsed time indicator, or at 10 years of installed time in the aircraft, whichever comes first.

[ ] (3) For airplanes equipped with Avidyne Entegra: replace the CMOS battery in the multifunction display (MFD) as required, but at least each 10 years. See Multifunction Display (MFD), 34-22-00.

[ ] (4) For airplanes equipped with an auxiliary vacuum system, replace Airborne 1H5 series check valve manifolds at ten (10) years from check valve manifold date of manufacture. See latest revision of Airborne SL 39A.

N. Each Twelve (12) Years

[ ] (1) If installed, replace disposable-type (non-gauged) fire extinguishers at twelve (12) years from date of manufacture.

[ ] (2) If installed, hydrostatically test rechargeable (gauged) fire extinguishers each twelve (12) years time-in-service.

[ ] (3) For airplanes equipped with Hartzell Engine Technologies (HET) Alternators (formerly Kelly Aerospace or Electrosystems), each 2,000 Hours or Twelve (12) Years, or at engine overhaul, whichever comes first; overhaul or replace alternator(s).

O. Each Twenty (20) Years

[ ] No fluid hose may exceed twenty (20) years total time-in-service.

3. Per Specific Operation / Operating Environment

A. Operation in High Dust or Industrial Pollution Environment

CAUTION: DISCONNECT LINES FROM PITOT/STATIC SYSTEM BEFORE CONDUCTING THIS INSPECTION.

	Item	Inspection	Inspection Interval
☐	Engine Air Filter.	Clean and inspect.	Daily.
☐	Cabin Environmental and Instrument Air Filters.	Inspect and replace if necessary.	100 Hours.
☐	Pitot/Static system.	Check for obstruction. Reverse flow to lines.	100 Hours or as required.
☐	Landing Gear Oleos	Clean.	Before each flight.
		Inspect.	100 Hours.
☐	Landing Gear Wheel Bearings.	Clean, inspect and repack.	50 Hours.
☐	Windows.	Inspect for cracks, erosion, crazing, visibility, and cleanliness.	Daily.
☐	Structure drain holes.	Clean with pipe cleaner.	Before each flight.

B. Operation in High Salt or High Humidity Environment

	Item	Inspection	Inspection Interval
☐	Fuselage, Empennage Wings, and Control Surfaces.	Remove floor panels and exterior access plates; inspect for corrosion using a borescope or other suitable tool.	200 Hours.
☐	Landing Gear.	Inspect for corrosion and lubrication.	200 Hours.
	WARNING: ENSURE BOTH MAGNETO SWITCHES ARE OFF (GROUNDED), BEFORE TURNING PROPELLER. ENGINE MAY START IF BOTH SWITCHES ARE NOT OFF. USE EXTREME CAUTION WHEN ROTATING PROPELLER BY HAND; PROPELLER MAY KICK BACK.		
☐	Engines with more than 50 hours total time.	Each five days, pull prop through five complete revolutions. Each 30 days, fly aircraft for 30 minutes or, ground run until oil temperature is in the green arc. Avoid excessive ground run.	Each 5 days and each 30 days.
☐	Engines with less than 50 hours total time.	Each day, pull prop through five complete revolutions. Each 30 days, fly aircraft for 30 minutes or, ground run until oil temperature is in the green arc. Avoid excessive ground run.	Daily and each 30 days.
☐	Instruments and Wiring.	Inspect for proper seal of cases and corrosion.	100 Hours.
☐	Interior.	Inspect upholstery, seat belts, seats and rugs for corrosion and integrity.	100 Hours.
	NOTE: Do not use metallic tie downs (i.e. - chains, cables, etc.) in high salt or high humidity environments.		

C. Operation in Extreme Cold

	Item	Inspection	Inspection Interval
☐	Hydraulic, Pneumatic and Environmental.	Check all fittings and attachments for security and leaks.	First 100 Hour, then as required.

D. Operation from Soft or Unusual Terrain

	Item	Inspection	Inspection Interval
☐	Landing Gear.	Inspect for cracks, attachment, damage, cleanliness and lubrication.	100 Hours.
☐	Wheels.	Inspect for cracks, damage, chipped rims; bearings for damage, corrosion and lubrication.	100 Hours.
☐	Tires.	Inspect for cuts, wear, inflation and deterioration.	Daily.
☐	Brakes.	Inspect for damage, foreign material, cracks and overheating.	Daily.
☐	Flaps, Lower Fuselage and Wing.	Inspect for damage, cracks and corrosion.	100 Hours.

E. At Engine Overhaul

	Item	Inspection	Inspection Interval
☐	Oil Cooler.	Replace or overhaul at engine overhaul.	Each Occurrence.

F. Before Each Flight

	Item	Inspection	Inspection Interval
☐	H3R Fire Extinguishers, Model RT-A600, with date codes 2012 and later, only. NOTE: See 26-20-00.	Weigh unit. Weight must be above the minimum specified on the nameplate.	Each Flight.

UNSCHEDULED MAINTENANCE CHECKS

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

The following inspections are required in response to specific anomalies encountered during aircraft operation. Note that the items listed herein are guidelines based on past operating experience. Each operator should closely monitor his own unique operating conditions/environment and react accordingly to keep his aircraft airworthy.

NOTE: A log book entry should be made upon completion of any inspections.

1. Lightning Strike

	Item	Inspection	Inspection Interval
☐	Propeller.	Refer to latest Sensenich Repair Manual. Recondition prior to return to service.	Each occurrence, before further flight.
☐	Engine.	See latest revisions of engine manufacturer's Service Bulletins and Overhaul Manual.	Each occurrence, before further flight.
☐	Electrical and Avionics Systems.	Inspect and check harness, connections, and equipment for high voltage damage, burns and insulation degradation. Replace or overhaul as required. Consult with appropriate avionics vendor(s) for inspections and operational checks. Bench test alternator and voltage regulator(s), see 24-30-00. Inspect all antenna connections for condition and security.	Each occurrence, before further flight.
☐	All exterior surfaces, skins, and structure.	Inspect for burns, evidence of arcing, and damage on surfaces and bearings. Check for correct material properties in the area of the strike path. Degauss engine mount. Replace or repair affected areas/parts.	Each occurrence, before further flight.
☐	System Components.	Inspect instrumentation, vacuum, pitot/static, and fuel systems, for damage and correct operation.	Each occurrence, before further flight.
☐	Static Wicks.	Replace.	Each occurrence, before further flight.
☐	Bearings.	Inspect all control surface hinges and bearings, and landing gear and wheel bearings for pitting and damage. Replace as required.	Each occurrence, before further flight.

2. Engine Overtemp, Loss of Oil, Sudden Stoppage, or Propeller Strike

	Item	Inspection	Inspection Interval
☐	Engine.	See latest revisions of engine manufacturer's Service Bulletins and Overhaul Manual.	Each occurrence, before further flight.
☐	Propeller. (Sudden stoppage only.)	Refer to latest Sensenich Repair Manual. Remove and recondition before return to service, if required.	Each occurrence, before further flight.
☐	Engine Mount and Attachments.	Inspect for distortion and damage. Replace or repair as required.	Each occurrence, before further flight.

3. Severe Turbulence, Hard or Overweight Landing

CAUTION: MINOR OR APPARENTLY SUPERFICIAL DAMAGE MAY INDICATE A MORE SEVERE CONDITION SOMEWHERE ELSE IN THE STRUCTURE.

A. Place aircraft in a normal level attitude.

B. Make a preliminary inspection of checking alignment and out-of-track condition of engine, wings, tail, landing gear and doors.

C. Follow Piper and Lycoming Maintenance Manual procedures. If there are any questions regarding repairs or procedures, contact your Piper Dealer's Service Advisor (DSA).

D. Inspect the following items closely to determine the extent of damage:

	Item	Inspection	Inspection Interval
☐	Landing Gear Struts. (Not required for severe turbulence.)	Cracks, signs of overstress deformation, loose or damaged strut housings. Axles for cracks, bending or flat spots. Damaged oleos and seals, hydraulic leaks and landing gear alignment.	Each occurrence, before further flight.
☐	Wheels, Tires, Brakes. (Not required for severe turbulence.)	Cracks, chips, loose or cracked mounting bolts, alignment of slippage marks, sidewall distress, hydraulic or air leaks. Inspect the wheels (dye penetrant method) and wheel bolts (magnetic particle method).	Each occurrence, before further flight.
☐	Wheel Wells and Landing Gear attach points. (Not required for severe turbulence.)	Buckling, cracks, overstress, wing skin buckling, and side brace for damage and condition. Inspect landing gear attachment bolts (magnetic particle method).	Each occurrence, before further flight.

3. Severe Turbulence, Hard or Overweight Landing (continued)

	Item	Inspection	Inspection Interval
☐	Wings.	Wing attach bolts for slippage, damage and overstress. Fasteners securing wing attachment fittings to fuselage frames and wing spars for condition and security. Upper and lower wing skins for wrinkles, cracks, popped or loose rivets. Remove access plates and inspect for internal damage to ribs, stringers and sparwebs; and fuel tanks for damage, attachment, and leaks.	Each occurrence before further flight.
☐	Engine.	Engine mounts for distortion and damage to elastomeric parts. Propeller for evidence of ground strike (i.e. - hard or overweight landing).	Each occurrence, before further flight.
☐	Fuselage.	Loose or missing rivets, door alignment, windows and attachments for overstress, cracks or damage. Wing carry through member for overstress damage. Stringers, bulkheads, keel beams for buckling, cracks, or damage. Avionics, instruments and accessories installation for security and operation.	Each occurrence, before further flight.
☐	Empennage.	Skins for buckling wrinkles, loose or missing rivets. Stabilator, rudder, and vertical fin for security of attachment and overstress of bolts. Ribs, stringers for buckling, cracks and damage.	Each occurrence, before further flight.

4. Flaps Extended Above Maximum Flap Extension Speed (V_{FE})

	Item	Inspection	Inspection Interval
☐	Flap torque tube/pushrod.	Inspect for distortion. Replace as required. (See Flap Torque Tube/Pushrod Distortion Inspection, 27-50-00.)	Each occurrence, before further flight.
☐	Flaps.	Inspect for damage to the skin and attach points. Replace as required.	Each occurrence, before further flight.

5. Flood Damage, Immersion in Water

A. These guidelines are general in nature and should be applied or varied to fit the individual aircraft according to water level, length of time of exposure and other variables. Only those areas that might not be obvious to the mechanic are addressed.

CAUTION: MAKE ALL REPAIRS AND/OR ADJUSTMENTS IN ACCORDANCE WITH THE APPROPRIATE PIPER MAINTENANCE MANUAL, THE COMPONENT MANUFACTURER'S MAINTENANCE MANUAL, AND FAR PART 43. PAY PARTICULAR ATTENTION TO SILT, CORROSION AND CONTAMINANTS.

B. Follow Piper and Lycoming Maintenance Manual procedures. If there are any questions regarding repairs or procedures, contact your Piper Dealer's Service Advisor (DSA).

C. Determine the water level on the aircraft. Determine which operating and/or electrical components have been exposed to the water.

D. If the following items were immersed, inspect them closely to determine the extent of damage:

	Item	Inspection	Inspection Interval
☐	Airframe.	Clean silt and contaminants from airframe.	If immersed, each event before further flight.
☐	Tubular Structures. (i.e. - Engine Mounts, etc.)	Check for internal corrosion. Clean and represerve as required. (See 71-20-00 - Engine Mount Corrosion Inspection, Immersion in Water.)	If immersed, each event before further flight.
☐	Wings.	Inspect to ensure that contaminants are cleaned from fuel cell areas.	If immersed, each event before further flight.
☐	Landing Gear and associated Bearings, Torque Links, Shimmy Dampeners, etc.	Jack airplane and cycle landing gear oleos and torque links to ensure proper operation.	If immersed, each event before further flight.
☐	Control Surfaces.	Remove surface, clean and check all bearings - relube or replace as necessary. Rebalance before installation.	If immersed, each event before further flight.
☐	Flight Control System.	Clean and inspect all cables, pulleys, and bearings for evidence of corrosion. Replace corroded cables. Re-preserve galvanized cable with MIL- C-11796 Class 2 (hot).	If immersed, each event before further flight.
☐	Trim Control System.	Clean and inspect all trim system cables, pulleys, drums, bearings, jack screws, etc. Do not apply preservation to trim cables.	If immersed, each event before further flight.
☐	Actuating Cables.	Inspect "push-pull" actuating cables for powerplant, heating and ventilating system, fuel system, etc. for proper operation.	If immersed, each event before further flight.

5. Flood Damage, Immersion in Water (continued)

	Item	Inspection	Inspection Interval
☐	Engine.	Remove, disassemble, and inspect. Examine all parts paying particular attention for evidence of corrosion, rust or contaminants imbedded on bearing surfaces, piston, mounting flanges or any aluminum, magnesium or bronze surface that may be porous. Remove evidence of rust, or corrosion. If pitting in stressed areas is found the part should not be reused. Silt imbedded in porous surfaces may be removed. Be certain oil passages, dowel holes and similar hidden openings and recesses are thoroughly free from contaminants. Test electrical components and fuel metering devices in accordance with manufacturer's instructions to determine fitness for future use. Reassemble engine using new seals, gaskets, stressed bolts nuts and crankshaft sludge tubes. All reused parts must conform with Lycoming Table of Limits No. SSP-1776 for fits and clearances. See latest revision of Lycoming Service Bulletin No. 357.	If immersed, each event, before further flight.
☐	Engine Accessories.	Inspect. Aircraft systems that supply either fuel or oil to the engine must be thoroughly cleaned, including oil cooler, lines, valves, etc. to prevent contamination of the engine after reinstallation.	If immersed, each event, before further flight.
☐	Propeller.	Inspect and repair as necessary in an authorized propeller shop.	If immersed, each event, before further flight.

5. Flood Damage, Immersion in Water (continued)

	Item	Inspection	Inspection Interval
☐	Electrical Systems.	Replace all circuit breakers and switches	If immersed, each event before further flight.
		Replace all solenoids, relays and master contactors.	
		Replace battery.	
		Disassemble all connectors; clean and inspect for corrosion. Replace all corroded or pitted connectors. Inspect for wire corrosion at connector.	
		Check all harness assemblies for entrapped contaminants. Clean and check for short circuits.	
		Remove electric motors and electric pumps.	
		Remove all potted solid state electrical equipment such as alternator inop. switches, low fuel warning switches, etc. Clean, dry and bench check per appropriate maintenance manual.	
		Clean and check voltage regulators and overvoltage relays. Replace as necessary	
		Clean and check all strobe light power supplies. Refer to appropriate maintenance manual.	
		Replace all fuel senders, etc.	
		Clean, inspect and check heated pitot systems.	
☐	Autopilot System. (If Installed.)	Bench check in accordance with appropriate maintenance manual. Pay particular attention to clutch settings.	If immersed, each event before further flight.

5. Flood Damage, Immersion in Water (continued)

	Item	Inspection	Inspection Interval
☐	Vacuum and Pitot-Static Systems.	Replace gyros. Replace filters. Clean and inspect all lines, and pitot and static vents. Clean and check all regulating valves. Remove and inspect engine driven and auxiliary vacuum pumps.	If immersed, each event before further flight.
☐	Induction System.	Clean and inspect for silt and corrosion. Check all ducts and gaskets. Replace as necessary. Clean and inspect all heat shrouds and ducting.	If immersed, each event before further flight.
☐	Fuel System.	Perform Fuel Tank/Wing Spar Corrosion Inspection, 28-10-00. Clean all associated lines and pumps. Clean and inspect all fuel tank vents, cap vents and vent lines.	If immersed, each event before further flight
☐	Instruments.	Clean and inspect instruments. Bench check per appropriate maintenance manual.	If immersed, each event before further flight.
☐	Heating and Ventilating Systems.	Replace blower. Clean and inspect all distribution boxes, ducting and valves. Inspect and check system control cables. Replace corroded or binding cables. If installed, clean and inspect air conditioning evaporator, condenser, and compressor.	If immersed, each event before further flight.

5. Flood Damage, Immersion in Water (continued)

	Item	Inspection	Inspection Interval
☐	Avionics Systems.	Replace avionics. Clean and inspect antennas and connectors.	If immersed, each event, before further flight.
☐	Insulation and Upholstery.	Remove all wet insulation and upholstery. Thoroughly clean and dry (or replace) to ensure corrosion is not promoted in adjacent structures.	If immersed, each event before further flight.

CHAPTER

6

DIMENSIONS AND AREAS

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CHAPTER 6

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CHAPTER 6 - DIMENSIONS AND AREAS

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GENERAL

Leading Particulars

The leading particulars are listed in “Chart 1” on page 06002. The airplane serial number is located on the Manufacturers Aircraft Association (MAA) plate located on the left side of the fuselage at approximately F.S. 255. The engine serial number plate is located on the right side of the engine oil sump just below cylinder number 4.

CHART 1 (Sheet 1 of 2)
LEADING PARTICULARS

MODEL	**PA-28-181 ARCHER III**
ENGINE	
Manufacturer	**Lycoming**
Model	O-360-A4M or IO-360-B4A (1)
Rated Power / Speed	180 HP / 2700 RPM
Oil Sump Capacity	8 U.S. qts
Fuel, Aviation Grade, Minimum	100 Low Lead
Magnetos	**Slick**
Left	4371 (2) 4373 (3)
Right	4370 (2) 4373 (3)
Timing	25° BTC (Verify on Engine Data Plate)
Points Setting	.010 ± .002 inch (Use “E” Gap method)
Spark Plug Gap Setting	0.016–0.022 inch
Firing Order	1-3-2-4
Starter	
Electrosystems (24 Volt)	MHB 4016
SkyTec	149-NL/EC
Alternator	
Electrosystems (70 amp)	ES 4032 (4)
Hartzell Engine Technologies (HET) (75 amp)	ALU-8532LS (5)
Plane-Power (70 amp)	AL24-P70-3 (6)
PROPELLER-FIXED PITCHED	
Manufacturer	Sensenich
Model	76EM8S14-0-62
Diameter Max/Min (Measurement taken at 30 inch station.)	76.0 in.

(1) Available in S/N’s 2843821, 2843852 and up.
(2) With O-360-A4M engine.
(3) With IO-360-B4A engine.
(4) S/N’s 2843001 thru 2843725.
(5) S/N’s 2843726 and up, with Standard or Garmin G500; and, in S/N’s 2843726–2843822, 2843824–2843851, and 2843853, with Garmin G1000.
(6) With IO-360-B4A engine; and, in S/N’s 2843823, 2843852, 2843854–2843999, with Garmin G1000/G1000 NXi.

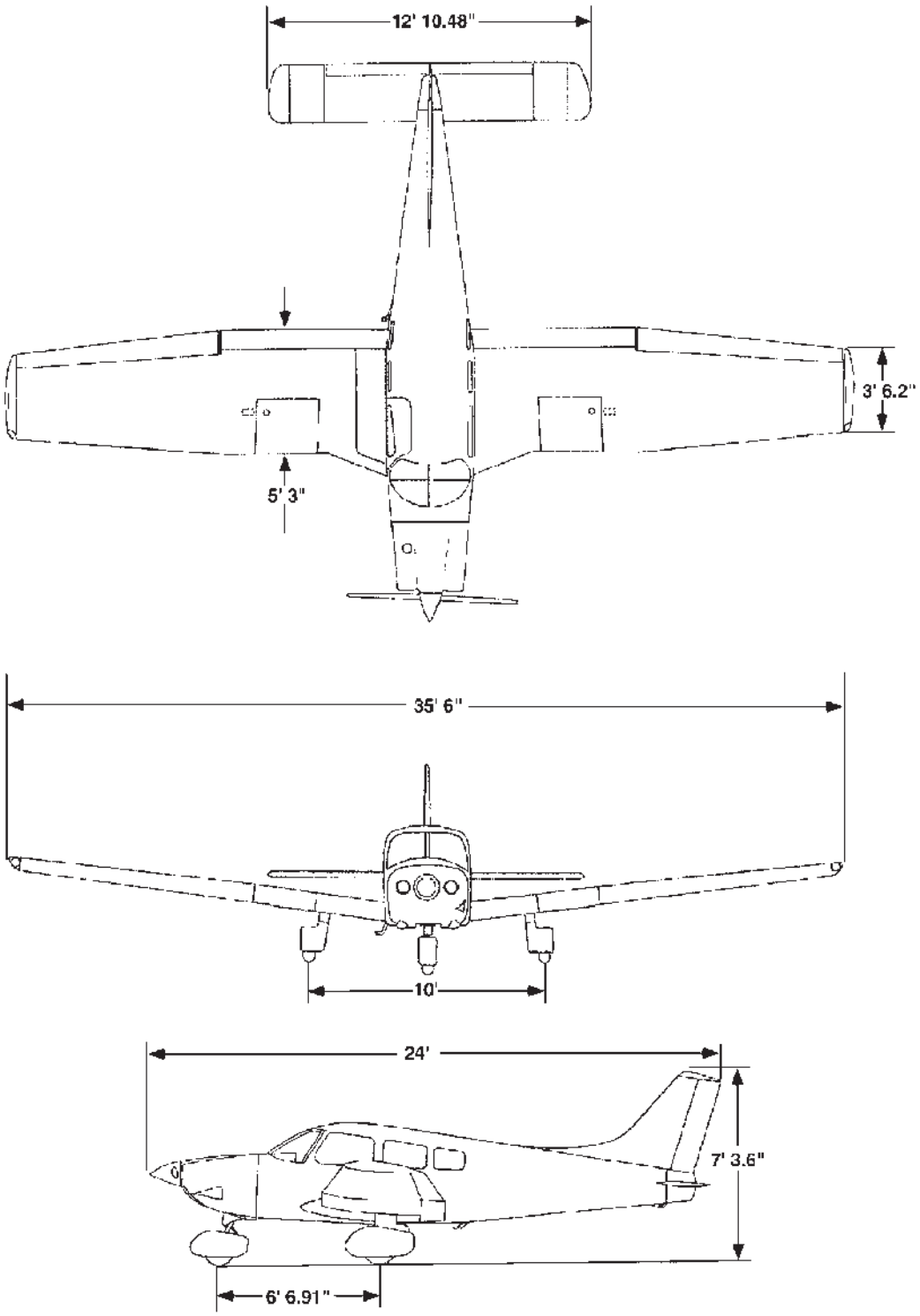
CHART 1 (Sheet 2 of 2)
LEADING PARTICULARS

FUEL SYSTEM	CAPACITY (US Gal.)
Left Wing Tank	25
Right Wing Tank	25
Total Fuel Onboard	50
Total Unusable	2
LANDING GEAR	
Type	Fixed
Shock Strut Type	Combination Air-Oil
Wheel Track	10.0 ft
Wheel Base	6 ft 6.9 in
Turning Radius, Nose Wheel 13 ft	
Nose Wheel Travel	20° ± 1° Left & Right
Wheel	
Nose	6:00 x 6 Cleveland 40-56B, or McCauley D-30665
Main	6:00 x 6 Cleveland 40-86B
Brake Type	Cleveland 30-55
Tires	
Nose	6:00 x 6, 4 Ply Rating
Main	6:00 x 6, 4 Ply Rating or 6:00 x 6, 6 Ply Rating
Tire Pressure	
Nose	18 psi
Main	24 psi

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PRINCIPAL DIMENSIONS

See Figure 1.

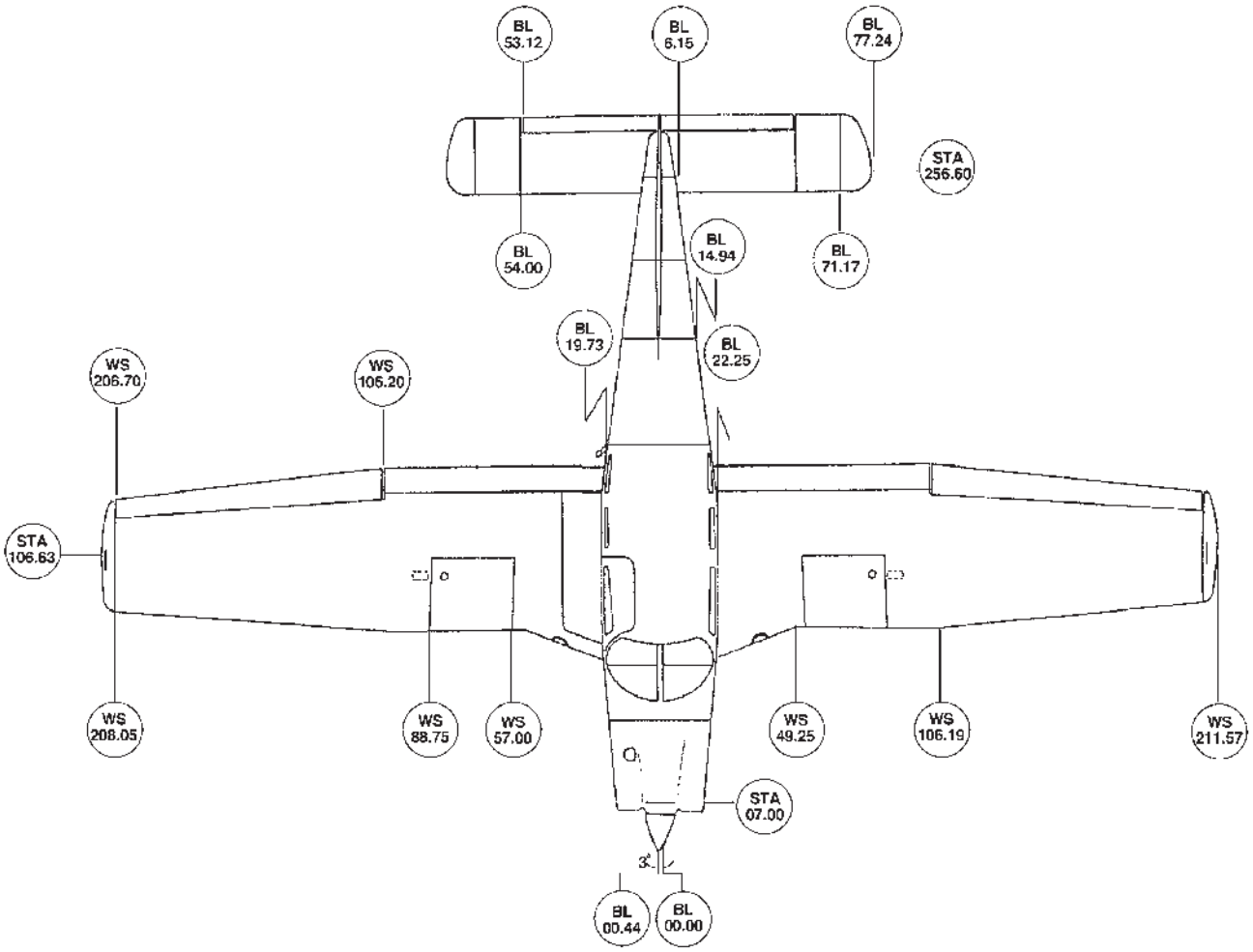


Three View
Figure 1

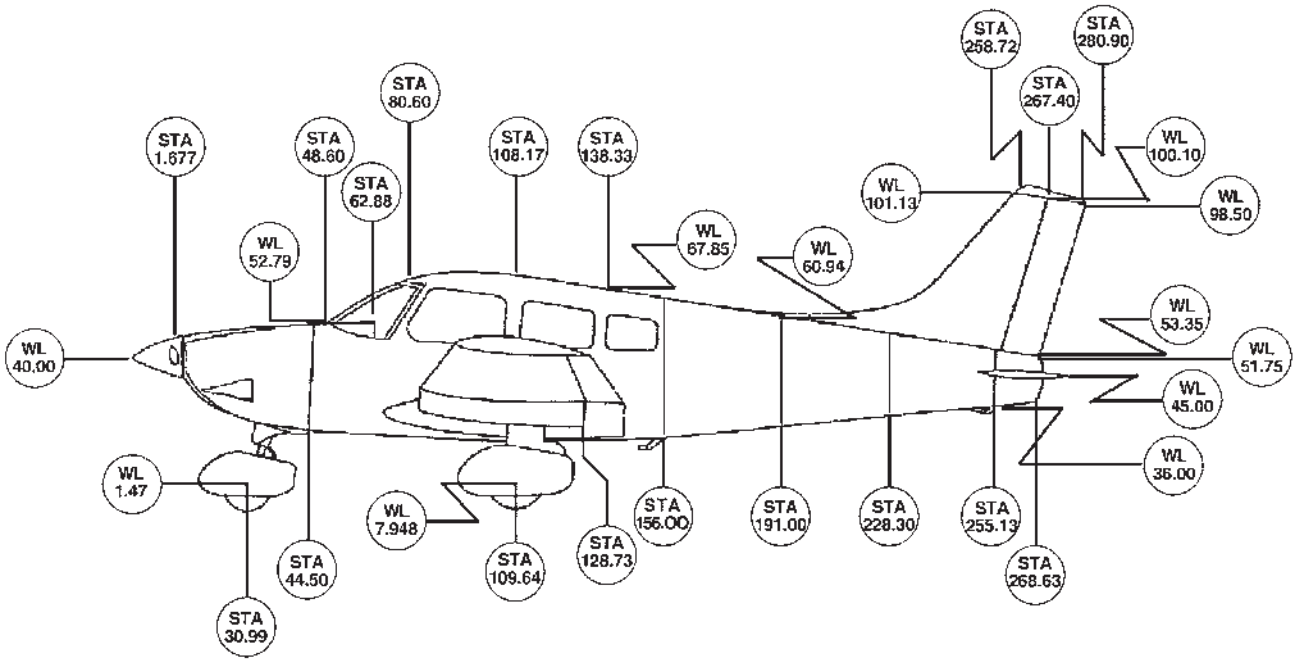
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REFERENCE LINES

A method using fuselage station, wing station, buttock line and waterline designations is frequently employed in this manual to locate various airplane components that require maintenance and servicing (see “Figure 1”). Fuselage stations (F.S.), wing stations (W.S.), buttock lines (B.L.), and water lines (W.L.) are reference points measured in inches vertically or horizontally from a given reference line. The reference datum line (F.S. 0) is located 78.4 inches ahead of the wing leading edge at the intersection of the straight and tapered section; B.L. 0 is the centerline of the airplane; and W.L. 0 is 20.5 inches below the cabin floor, measured at the rear wing spar with the airplane level.



Station Reference Lines
Figure 1 (Sheet 1 of 2)



Station Reference Lines
Figure 1 (Sheet 2 of 2)

ACCESS PROVISIONS

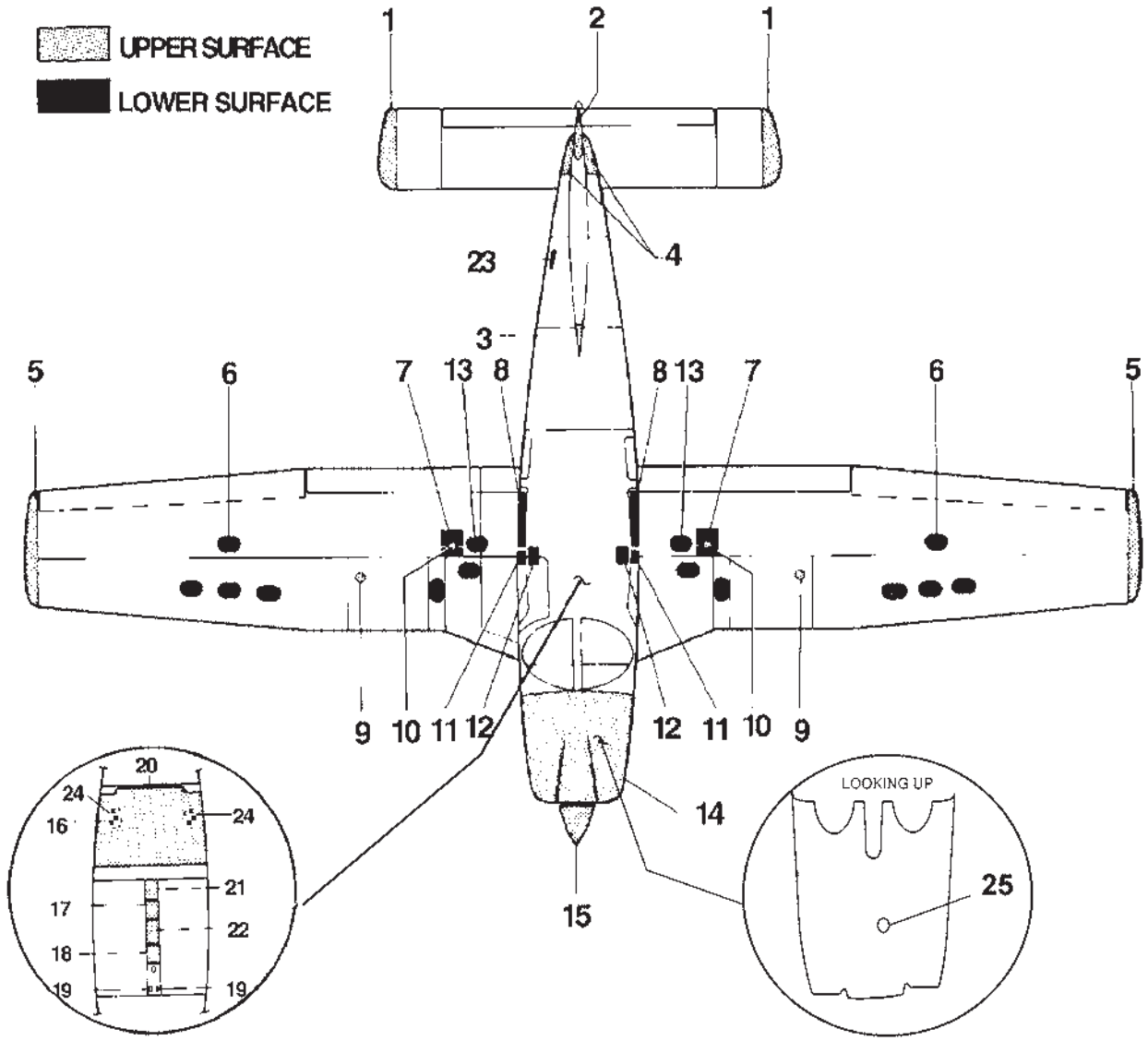
The access and inspection provisions for the PA-28-181 are shown in “Figure 1” on page 06402. The component to be serviced or inspected through each opening is identified in the illustration. All access plates and panels are secured by either metal fasteners or screws.

“Access plates and panels” includes all forms of covers, cowlings, fairings, plates, panels, tips, etc. identified in “Figure 1” which may be removed for access to any component or space.

All access plates and panels should be inspected for damage or excessive wear, including all mounting holes, each time they are removed. If damage or excessive wear is detected, installation of a new access plate or panel is recommended.

CAUTION: BEFORE ENTERING THE AFT SECTION OF THE FUSELAGE, BE SURE THE AIRPLANE IS SUPPORTED AT THE TAIL SKID.

To enter the aft section of the fuselage, open baggage compartment door and remove access panel.



1. STABILATOR TIP
2. RUDDER TIP
3. FIN FAIRING
4. TAIL CONE, CONTROL CABLE AND TRIM SCREW
5. WINGTIP
6. BELLCRANK AND CONTROL CABLE COVERS
7. MAIN GEAR STRUT ASSEMBLY FAIRING
8. FLAP TORQUE TUBE AND AILERON CABLES FAIRING
9. FUEL TANK FILLER CAP
10. STRUT FILLER CAP
11. FUEL LINES FAIRING
12. FUEL LINES AND SPAR ATIACHMENT BOLTS COVER.
13. CONTROL CABLES COVER
14. ENGINE COWL ASSEMBLY
15. PROPELLER SPINNER
16. SEAT, BATIERY, TORQUE TUBE AND CONTROL CABLES
17. MIDDLE CONTROL CABLE PLATE
18. FORWARD AILERON AND STABILATOR CONTROL CABLE PLATE
19. AILERON AND STABILATOR CONTROL CABLE COVERS
20. AFT FUSELAGE AREA AND STROBE LIGHT POWER SUPPLY COVER
21. REAR CONTROL CABLES AND PULLEYS COVER
22. FLAP CONTROL COVER
23. EMERGENCY LOCATOR TRANSMITIER
24. CONTROL CABLE INSPECTION PANELS
25. ACCESS HOLE FOR FUEL SERVO ADJUSTMENT (S/N's 2843891 & UP)

Access Plates and Panels
Figure 1

CHAPTER

7

LIFTING AND SHORING

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CHAPTER 7

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CHAPTER 7 - LIFTING AND SHORING

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JACKING

Jacking the airplane is necessary to service the landing gear and for other operations. The jacking operation is normally performed using tripod jacks. In other situations (i.e. - emergency, post-accident lifting, etc.), slings or air bags may be more appropriate.

If wing or fuselage shoring is required, make sure the support is contoured to conform to the surface it is supporting.

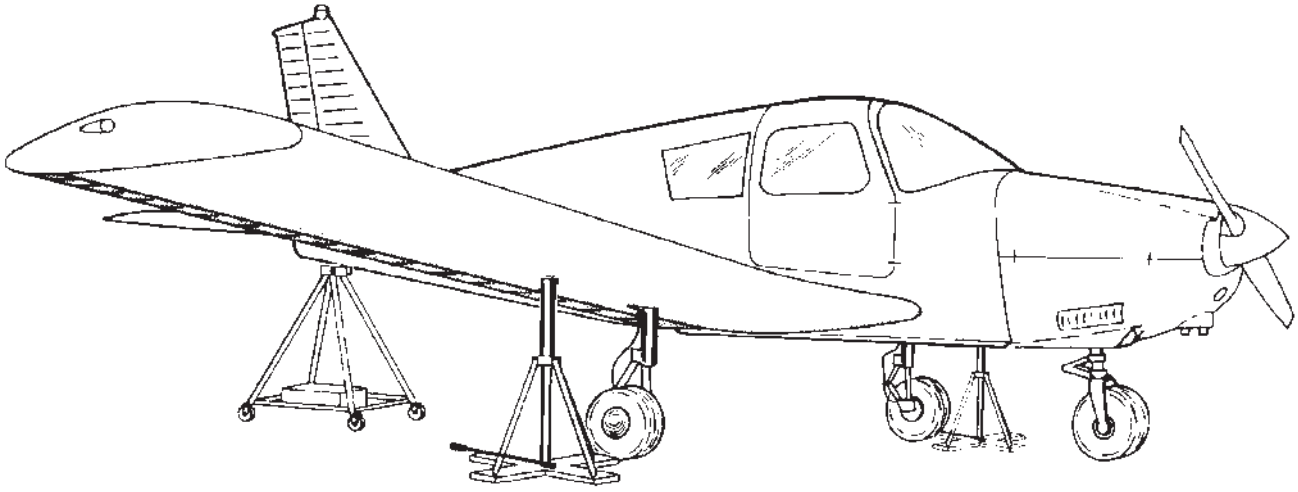
Jacking

A. Align jacks under the wing respective pads on the wing front spar. (Refer to Figure 1.)

CAUTION: BE SURE TO APPLY SUFFICIENT TAIL SUPPORT BALLAST. OTHERWISE, THE AIRPLANE WILL SLIP FORWARD AND FALL ON THE FUSELAGE NOSE SECTION.

B. Attach a tail stand with approximately 250 pounds ballast to tail skid.

C. Carefully raise jacks until all three wheels are clear of the surface.



Jacking
Figure 1

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CHAPTER

8

LEVELING AND WEIGHING

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CHAPTER 8 - LEVELING AND WEIGHING

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WEIGHING

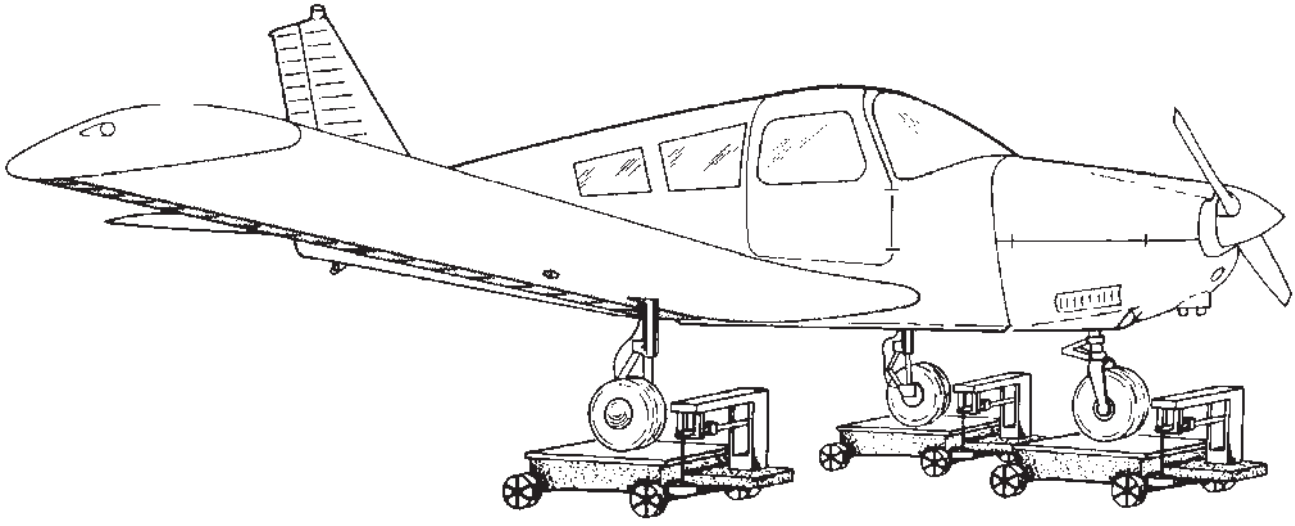
The airplane may be weighed by the following procedure (See Figure 1):

A. Position a scale and ramp in front of each of the three wheels.

B. Secure the scales from rolling forward and tow the airplane up onto the scales. (Refer to Towing, 9-10-00.)

C. Remove the ramp so as not to interfere with the scales.

D. If the airplane is to be weighed for weight and balance computations, level the airplane per 8-20-00.



Weighing Airplane
Figure 1

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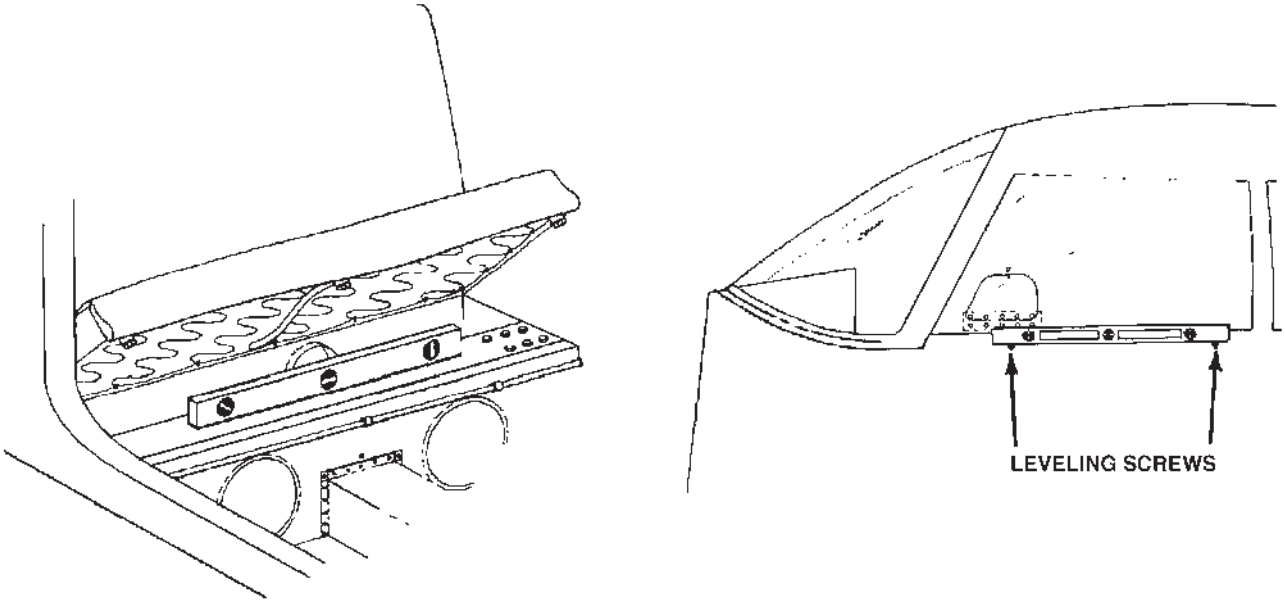
LEVELING

All configurations of the airplane are provided with a means for longitudinal and lateral leveling. The airplane may be leveled while on jacks, during the weighing procedure while the wheels are on scales, or while the wheels are on the ground. To level the airplane for purposes of weighing or rigging, the following procedures may be used:

NOTE: Always level the airplane laterally first, then level it longitudinally.

A. To laterally level the airplane, place a spirit level under rear seat.Raise or lower one wing tip by deflating the tire on the high side of the airplane or adjust either jack until the bubble of the level is centered. (Refer to Figure 1.)

B. To longitudinally level the airplane, partially withdraw the two leveling screws located immediately below the left front side window. (Refer to Figure 1.) Place a spirit level on these screw heads. If airplane is on scales or on floor, block main gear oleos to full extension and deflate the nose wheel tire or adjust the jacks until the bubble of the level is centered.



Leveling Airplane
Figure 1

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CHAPTER

9

TOWING AND TAXIING

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CHAPTER 9 - TOWING AND TAXIING

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TAXIING	9-20-00	1

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TOWING

Before towing airplane, ground personnel must be informed by a qualified pilot or other qualified personnel about tow turning limits of nose gear and any other system functions required to properly and safely move the airplane.

CAUTION: WHEN TOWING WITH POWER EQUIPMENT, TURNING THE NOSE GEAR IN EITHER DIRECTION BEYOND THE STEERING RADIUS LIMITS WILL RESULT IN DAMAGE TO THE NOSE GEAR AND STEERING MECHANISM.

CAUTION: PUSHING ON THE TRAILING EDGE OF THE AILERONS, WHEN MOVING THE AIRCRAFT FORWARD BY HAND, WILL CAUSE THE AILERON CONTOUR TO CHANGE RESULTING IN AN OUT-OF-TRIM CONDITION.

CAUTION: DO NOT TOW AIRPLANE WITH CONTROL LOCKS INSTALLED.

CAUTION: WHEN TOWING WITH POWER EQUIPMENT, USE ONLY EQUIPMENT THAT IS APPROPRIATE FOR THE AIRPLANE CLASS AND WEIGHT.

CAUTION: IF AIRPLANE IS STUCK (OR THE WHEELS ARE OTHERWISE NOT FREEWHEELING), DO NOT TOW THE AIRPLANE FROM THE NOSE. ATTACH TOW LINES TO THE MAIN GEAR STRUTS AS DESCRIBED BELOW.

The airplane may be moved by using the nose wheel steering bar that is stowed in the baggage compartment or by using power equipment that will not damage or cause excess strain to the nose gear steering assembly. Tow bar engages lugs at the top of the nose gear fork.

In the event towing lines are necessary, lines (rope) should be attached to both main gear struts as high up on the tubes as possible. Lines should be long enough to clear the nose and/or tail by not less than 15 feet, and a qualified person to ride in the pilot's seat to maintain control by use of the brakes.

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TAXIING

Before attempting to taxi the airplane, ground personnel should be checked out by a qualified pilot or other responsible person. Engine starting and shutdown procedures should be covered as well. Check that the propeller back blast and taxi areas are clear, apply power to start the taxi roll and perform the following checks:

CAUTION: DO NOT OPERATE THE ENGINE AT HIGH RPM WHEN RUNNING UP OR TAXIING OVER GROUND CONTAINING LOOSE STONES, GRAVEL OR ANY LOOSE MATERIAL THAT MAY CAUSE DAMAGE TO THE PROPELLER BLADES.

CAUTION: OBSERVE WING CLEARANCES WHEN TAXIING NEAR BUILDINGS OR OTHER STATIONARY OBJECTS. IF POSSIBLE, STATION A GUIDE OUTSIDE THE AIRPLANE TO OBSERVE.

CAUTION: WHEN TAXIING ON UNEVEN GROUND, AVOID HOLES AND RUTS.

A. Taxi forward a few feet and apply brakes to determine their effectiveness.

B. While taxiing, make slight turns to check steering effectiveness.

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CHAPTER

10

PARKING AND MOORING

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CHAPTER 10

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
10-List of Effective Pages	1	Mar 6/14			
	2	Mar 6/14			
10-Table of Contents	1	Mar 6/14			
	2	Mar 6/14			
10-10-00	1	Mar 6/14			
	2	Mar 6/14			
10-20-00	1	Mar 6/14			
	2	Mar 6/14			

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CHAPTER 10 - PARKING AND MOORING

TABLE OF CONTENTS

SUBJECT	SECTION	PAGE
PARKING	**10-10-00**	1
Parking		1
Locking Airplane		1
MOORING	**10-20-00**	1

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PARKING

CAUTION: ENSURE AIRCRAFT IS SUFFICIENTLY PROTECTED AGAINST ADVERSE WEATHER CONDITIONS AND PRESENTS NO DANGER TO OTHER AIRCRAFT.

CAUTION: MOOR AIRCRAFT WHEN PARKED ANY LENGTH OF TIME OR OVERNIGHT.

CAUTION: BE SURE TO PARK AIRCRAFT INTO THE WIND.

CAUTION: USE CARE WHEN SETTING OVERHEATED BRAKES, OR DURING COLD WEATHER WHEN ACCUMULATED MOISTURE COULD FREEZE BRAKES.

CAUTION: BE SURE TO LOCK AILERON AND STABILATOR BY USING FRONT SEAT BELT.

1. Parking

When parking the airplane, ensure that it is sufficiently protected against adverse weather conditions and presents no danger to other aircraft. When parking the airplane for any length of time or overnight, it is recommended that it be moored.

A. To park the airplane, head it into the wind, if possible.

B. Set the parking brake by pulling back the brake lever and depressing the knob attached to the left side of the handle.

C. To release the parking brakes, pull back on the brake lever to disengage the catch mechanism. Then allow the handle to swing forward.

D. The aileron and stabilator controls may be secured with the pilot's seat belt.

2. Locking Airplane

The cabin door and the baggage compartment door are provided with a key lock on the outside. A locking gas cap is optional. All use the same key, except the locking gas caps, original equipment and service replacements in 2006 and up, which have a separate key.

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MOORING

Moor airplane to ensure immovability, protection, and security under various weather conditions.

CAUTION: BE SURE AIRCRAFT IS PARKED INTO THE WIND WITH WHEELS BLOCKED TO ENSURE AIRCRAFT WILL NOT MOVE.

CAUTION: USE SQUARE OR BOWLINE KNOTS. DO NOT USE SLIP KNOTS.

CAUTION: SECURE TIE-DOWN ROPES TO WING TIE-DOWN RINGS AND TAIL SKID AT APPROXIMATELY 45 DEGREE ANGLES TO GROUND.

CAUTION: ROPE OF NON-SYNTHETIC MATERIAL MUST HAVE SUFFICIENT SLACK TO AVOID AIRCRAFT DAMAGE WHEN ROPES CONTRACT DUE TO MOISTURE.

CAUTION: LOCK AILERON AND STABILATOR CONTROLS USING FRONT SEAT BELT OR CONTROL SURFACE BLOCKS.

CAUTION: INSTALL PITOT TUBE COVER.

CAUTION: ADDITIONAL HIGH WIND PREPARATIONS INCLUDE USING TIE-DOWN ROPES FROM THE LANDING GEAR FORKS, AND SECURING RUDDER.

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CHAPTER

11

PLACARDS AND MARKINGS

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CHAPTER 11

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
11-List of Effective Pages	1	May 31/20			
	2	Mar 6/14			
11-Table of Contents	1	May 31/20			
	2	Mar 6/14			
11-00-00	1	Mar 6/14			
	2	Jun 1/16			
	3	Jun 1/16			
	4	Jun 1/16			
11-20-00	1	Mar 6/14			
	2	Jun 1/16			
	3	Nov 15/17			
	4	Mar 6/14			
11-30-00	1	May 31/20			
	2	May 31/20			
	3	May 15/19			
	4	May 15/19			
	5	May 15/19			
	6	May 15/19			
	7	May 15/19			
	8	May 15/19			
	9	May 15/19			
	10	Mar 15/20			

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CHAPTER 11 - PLACARDS AND MARKINGS

TABLE OF CONTENTS

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Placards, Nameplates, and Decals		2
Self-Adhesive Installation		2
Application of Edge Sealer (Decals only)		2
Without Adhesive - Installation with Tape		3
Meyercord / Ink Transfers (Decals)		4
EXTERIOR PLACARDS AND MARKINGS	11-20-00	1
INTERIOR PLACARDS	11-30-00	1

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GENERAL

1. Decals

NOTE: See also, Placards, Nameplates, and Decals, below.

For proper adhesion of decals, all surfaces must be clean and free of wax, oil, etc. Porous surfaces must be sealed. Sand and clean rough surfaces to remove any residue.

A. Paper Decals

(1) Installation

Soak paper decals in water 1 to 3 minutes. Place one decal edge on receiving surface and slide decal off of paper backing. Blot water around decal with a soft absorbent cloth. Remove bubbles trapped beneath decal by wiping carefully towards nearest edge with a cloth. Coat decal with clear varnish to protect from deterioration and peeling.

(2) Removal

Remove paper decals by rubbing with cloth dampened with lacquer thinner. Use lacquer thinner sparingly if decals are applied over painted or doped surfaces.

B. Vinyl Film Decals

(1) Installation

Separate paper backing from vinyl film. Remove paper adhering to film by rubbing with a clean water saturated cloth or a piece of masking tape. Apply cyclohexanone or equivalent, to adhesive side of film. Position and apply decal while adhesive is still tacky. Work a roller across decal until all air bubbles are removed.

(2) Removal

To remove a vinyl decal, place cloth saturated with cyclohexanone or methyl ethyl ketone on decal. Scrape with micarta scraper. Remove remaining adhesive with cloth dampened with dry cleaning solvent.

C. Metal Decals

(1) Installation

(a) Cellophane backed.

1) Immerse in water 1 to 3 minutes.
2) Remove and dry.
3) Remove cellophane backing.
4) Position on receiving surface. (For large foil decals, position center or receiving surface and work outward from center.)
5) Roll with rubber roller and press all edges firmly.

(b) Paper backed.

1) Peel backing from decal.
2) Apply light coat of cyclohexanone.
3) Position and smooth per steps 4 and 5 of cellophane backed decals.

(c) Metal decals with no adhesive.

1) Apply cement MIL-A-5092 to decal and receiving surface.
2) Allow cement to dry until tacky.
3) Apply and smooth down decal.
4) Remove excess adhesive with aliphatic naphtha.

(2) Removal

To remove metal decals, moisten decal edge with aliphatic naphtha and peel off decal.

2. Placards, Nameplates, and Decals

(PIR-PPS65103, Rev. H.)

A. Self-Adhesive Installation

NOTE: For proper adhesion of decals, apply when the temperature is between 55° and 90° F.

(1) Wipe the area where the item is to be installed, using a clean cloth soaked with isopropyl alcohol. Dry with a clean, dry cloth.

NOTE: Dry the area before the alcohol evaporates, to avoid discoloration and deposits at the edges of the wetted area.

(2) Lift a corner or an edge of backing from the item, without touching the adhesive.

(3) Position as specified by the applicable figure in 11-20-00 or 11-30-00 and secure a free corner or edge.

(4) Using a squeegee or equivalent item to hold the item to the panel, draw the rest of the backing from the item.

(5) Press the item in place with the squeegee by wiping across it from end to end, around edges and corners, and into recessess using firm, deliberate strokes and heavy pressure.

(6) Remove the premask from face of item by using an eraser or finger nail to lift a corner. Then peel back as parallel to the item as possible.

NOTE: Any attempt to squeegee the item after the premask is removed will cause shiny marks, blemishing the item's background.

(7) If any portion of the item tends to lift off, remove and discard it. Repeat this procedure to install a replacement item.

B. Application of Edge Sealer (Decals only)

Decals intended for use in harsh environments have an overlay film applied during manufacturing. This overlay film typically extends 0.25 inch around the perimeter of the decal. If any of the overlay film is cut or otherwise damaged during installation, then an additional sealer must be applied along all edges of the decal. 3M 3950 Edge Sealer (P/N 279-474) is an acceptable sealer to be used in this application. Apply Edge Sealer as follows:

(1) Apply edge sealer to the edges of the decal by straddling the decal and the aircraft structure with a ¼-inch brush or dauber.

(2) Wipe this interface in a smooth continuous motion to provide a fillet-type seal that extends onto the decal and aircraft structure by aprroximately 0.1 inch.

(3) Ensure that the entire edge is sealed without gaps or air bubbles.

(4) Touch up as required with additional brush coats of edge sealer within 2 hours.

(5) Wipe all excess sealer with a dry wiper.

(6) Air dry or apply heat (up to 250˚F) to achieve a tack-free condition of the edge sealer.

C. Without Adhesive - Installation with Tape

(PIR-PPS50085, Rev. B.)

To bond to the aircraft any placards, etc., that do not have an adhesive backing, use 3M™ VHB™ pressure sensitive acrylic foam tape, as follows.

(1) Confirm the correct location to adhere the double-sided tape. Place location marks (using masking tape) on the aircraft surface to precisely position the tape.

(2) Clean the surface using acetone or a 50:50 mixture of isopropyl alcohol and water. Wipe the area dry with a clean, lint-free, dry cloth.

(3) Wipe both faying surfaces with a brush or swab saturated with 3M Tape Primer 94 (P/N 279-127, one quart). Use the minimum amount of primer that will coat the desired surface areas. Allow 10 minutes for it to dry.

NOTE: Use a minimum coating of the tape primer. More is not better.

CAUTION: THE ADHESION OF THE TAPE IS ALMOST IMMEDIATE, SO USE EXTREME CARE AND APPLY VERY LITTLE CONTACT PRESSURE. AVOID THE NEED TO REPOSITION THE TAPE.

(4) Handle the double-sided acrylic foam tape by its edges only; carefully position the tape using the location marks. Adhesion is almost immediate, so use extreme care and as little contact pressure as possible. Avoid the need to reposition the tape, but if necessary, move the tape immediately and cautiously.

(5) When the placement of the tape is completed, use a roller, squeegee, or other smoothing tool to apply even pressure against the tape. Pay close attention to the tape edges.

CAUTION: DO NOT TRY TO REPOSITION THE TAPE AFTER USING SMOOTHING TOOL. DO NOT PULL ON THE TAPE OR ATTEMPT TO LIFT OR PEEL ITS EDGES.

(6) Keep the outward liner intact on the tape and position the item to be bonded with the tape over the location. Remove the tape liner using the pull tabs, folding them back at an approximately 45-degree angle from under the item, exposing the tape to the item.

(7) Once bonded, if there is some bowing or deformation of the adhered item, apply pressure not exceeding 5 psi with a shot bag (approximately 6 inches wide, 36 lbs per foot is only 1 psi) for 30 minutes.

3. Meyercord / Ink Transfers (Decals)

(PIR-PPS65104, Rev. J.)

Transfers (decals) installed on the instrument panel are Meyercord / Ink type. The following procedures should be followed in the event one or more of these transfers (decals) must be replaced.

A. Removal

CAUTION: DO NOT USE LACQUER THINNER ON ANY PANEL THAT HAS BEEN PAINTED WITH ENAMEL OR LACQUER. INSTRUMENT PANELS ARE PAINTED AT THE FACTORY WITH POLYURETHANE PAINTS.

Remove placard to be replaced with a clean cloth dampened with lacquer thinner.

CAUTION: BUTYL GLYCOL ETHER/ISOPROPYL ALCOHOL SOLVENT SOLUTION WILL REMOVE ENAMEL, LACQUER, AND POLYUTHERANE BASED PAINT PRODUCTS IF LIQUID IS DROPPED ONTO PAINTED SURFACE AND NOT REMOVED IMMEDIATELY.

If panel is painted with enamel or lacquer use a clean cloth dampened with Butyl Glycol Ether/Isopropyl Alcohol Solvent Solution to remove placard to be replaced.

B. Installation

(1) If required, clean surface to receive transfer (decal) using alcohol. Newly painted surfaces need no preparation.

(2) Mix a solution consisting of six (6) parts water and one (1) part Butyl Glycol Ether/Isopropyl Alcohol Solvent Solution (P/N 179-497).

(3) Submerge transfer (decal) in the mixed Butyl Glycol Ether/Isopropyl Alcohol Solvent Solution for approximately 3 to 5 seconds.

(4) Remove transfer (decal) from mixed solvent solution and lay in position.

(5) Using a plastic squeegee, squeegee out from center to edges to remove excess solution.

(6) Wait approximately 30 to 60 seconds, then slide the backing paper off and wipe up the excess solution with a damp cloth.

(7) Wait at least 30 minutes at room temperature before wiping the face of the transfer (decal) with a damp cloth to remove excess solvent residue.

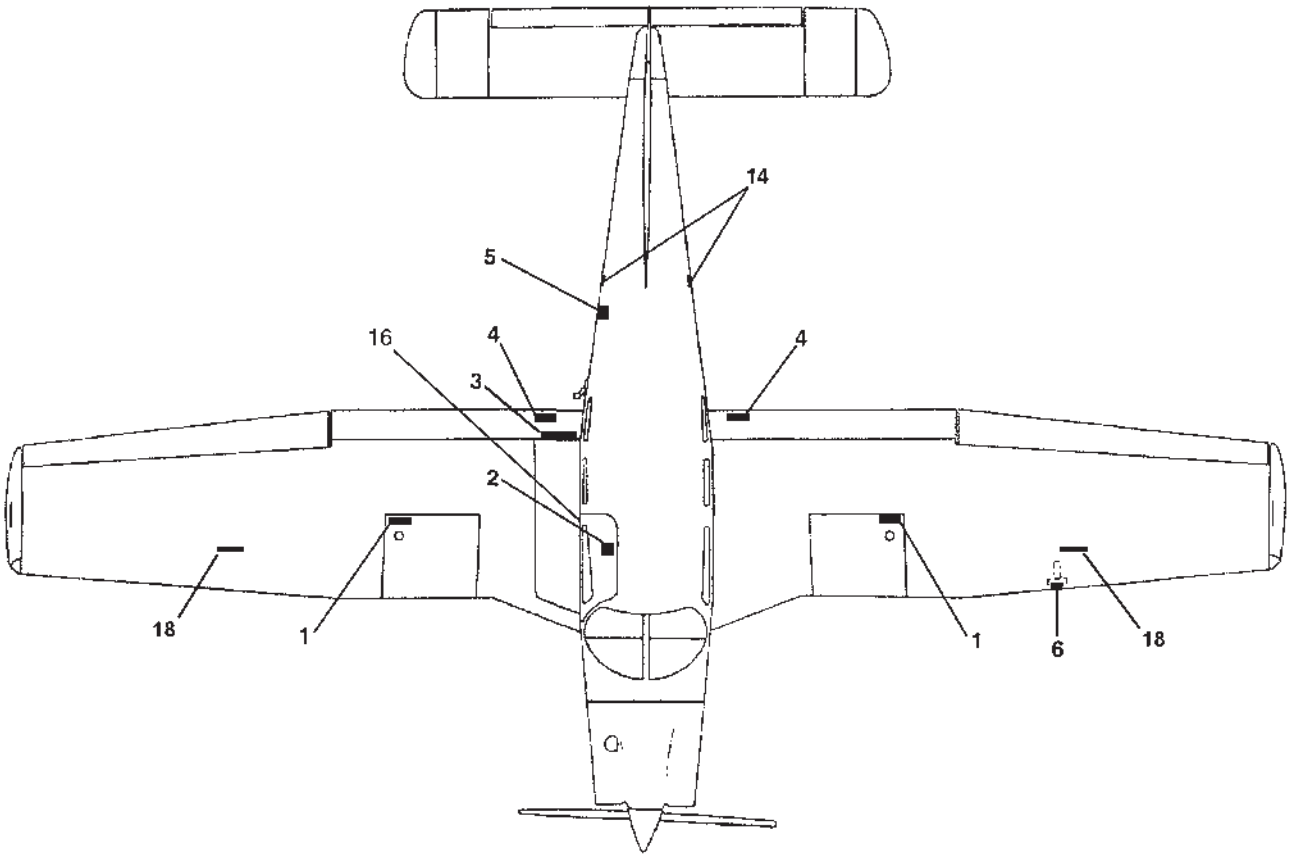
(8) Allow the transfer (decal) to dry thoroughly, (tack free in 2 hours at room temp) before handling.

EXTERIOR PLACARDS AND MARKINGS

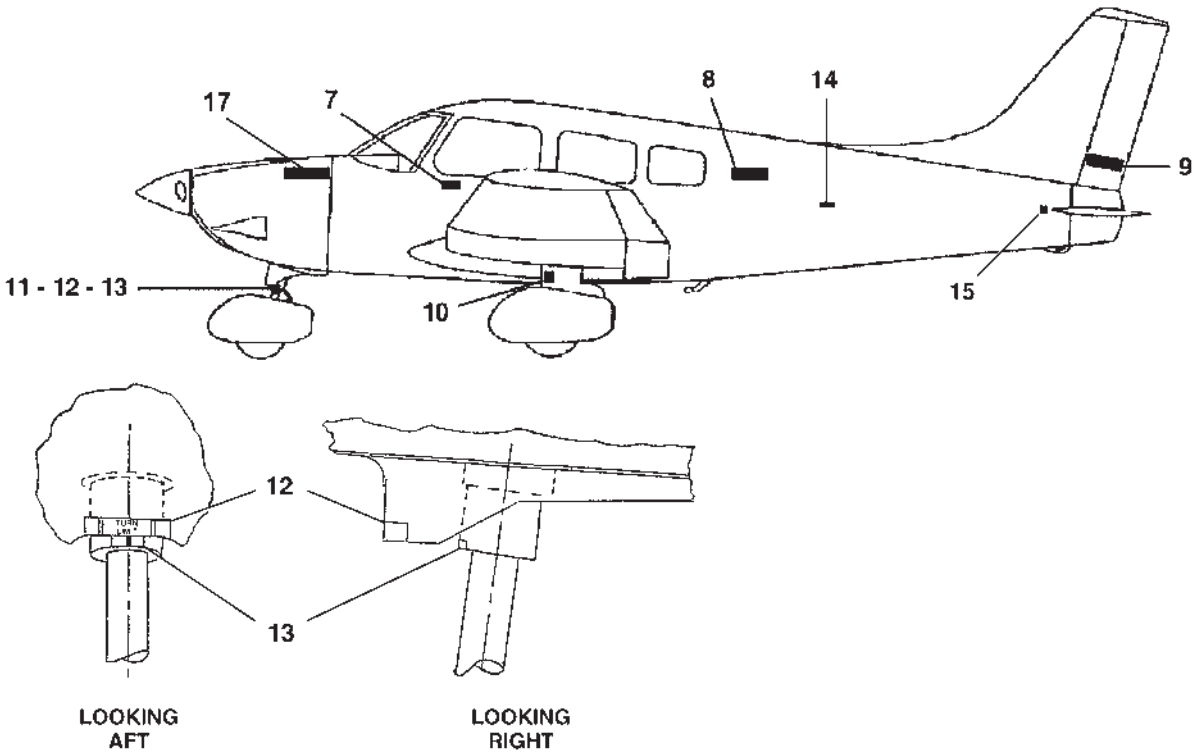
The airplane nameplate placard (“Figure 1” on page 11202, Item 15) is located on the left side of the fuselage near the stabilator leading edge at approximately F.S. 278.60. The placard identifies the airplane by its model number and serial number. Should a question arise concerning the care of the airplane, it is important to include the airplane serial number in any correspondence to your Piper Dealer’s Service Advisor (DSA).

NOTE: Any time an airplane is repainted or touched up, inspect all placards to ensure that they are not covered with paint, are legible, and securely attached.

36991 CV
107510 H



Exterior Placards and Markings
Figure 1 (Sheet 1 of 2)



S/N'S 2843547 & UP
(OPTIONAL IN S/N'S 2843415 THRU 2843546)

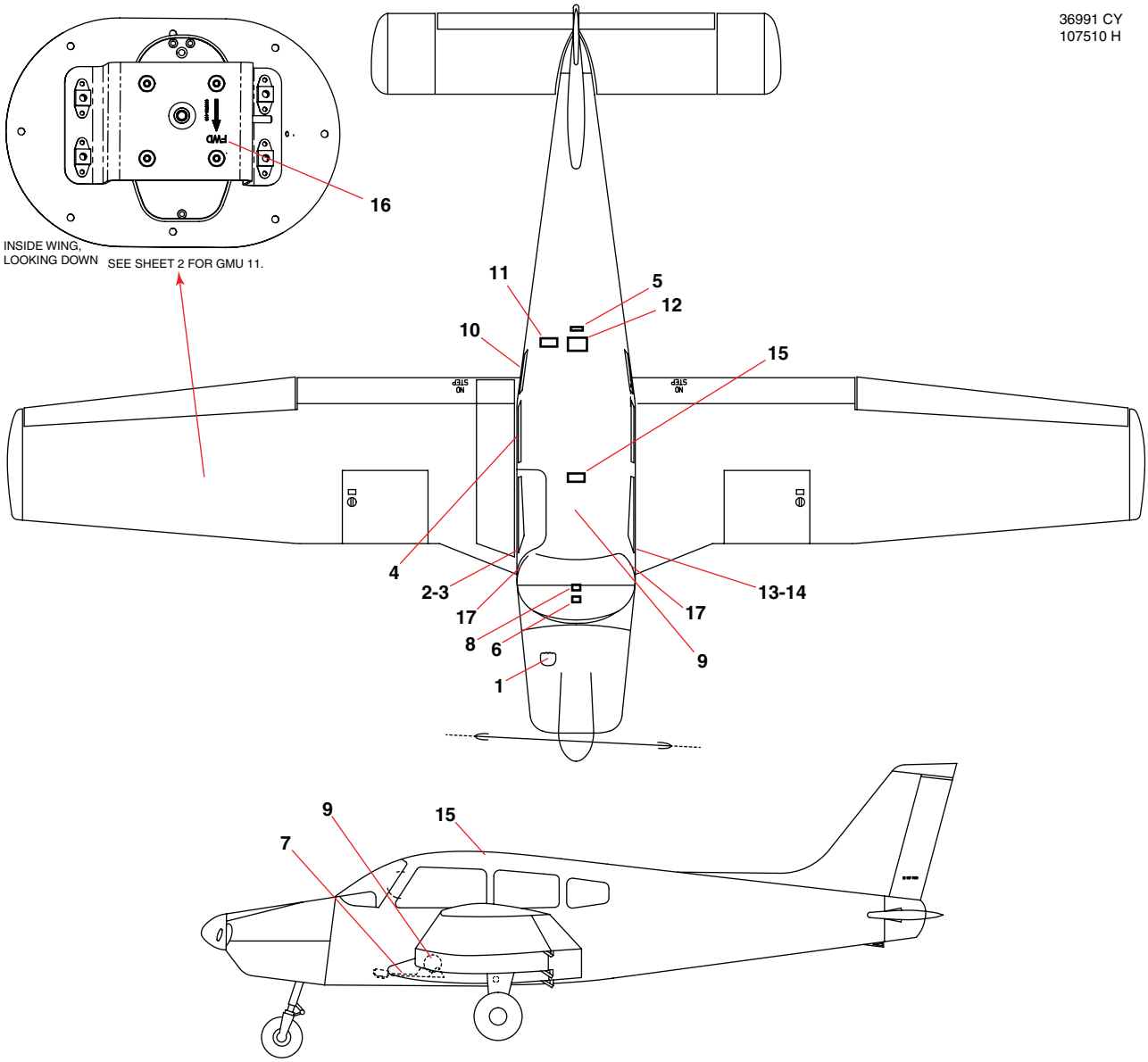
1. DECAL - AVGAS
2. PLACARD - DOOR RELEASE
3. PLACARD - FLAP WARNING
4. PLACARD - NO STEP
5. PLACARD-EXTERNALPOWER
6. DECAL - LIFT DETECTOR
7. PLACARD- LEVEL POINT
8. DECAL- PIPER LOGO
9. PLACARD - DO NOT PUSH
10. DECAL- OLEO SERVICE INSTRUCTION (MOUNTS INBOARD)
11. PLACARD - OLEO SERVICE INSTRUCTION
12. PLACARD - TURN LIMIT
13. PLACARD- TURN LIMIT POINTER
14. DECAL- STATIC VENT (WITH S-TEC AUTOPILOT ONLY)
15. PLATE - MAA (I.E., SERIAL NUMBER PLATE)
16. PLACARD - DOOR OPEN (S/N'S 2843203 & UP)
17. DECAL- MODEL NAME
18. DECAL- USE ONLY NON-MAGENETIC ..... .
    (UNDERSIDE, WITH GARMIN G1000, S/N'S 2843701, 2843703 AND UP)

Exterior Placards and Markings
Figure 1 (Sheet 2 of 2)

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INTERIOR PLACARDS

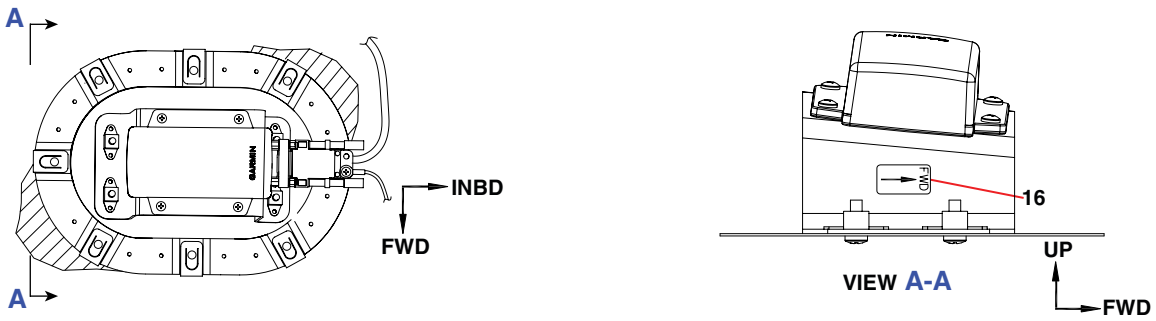
36991 CY
107510 H



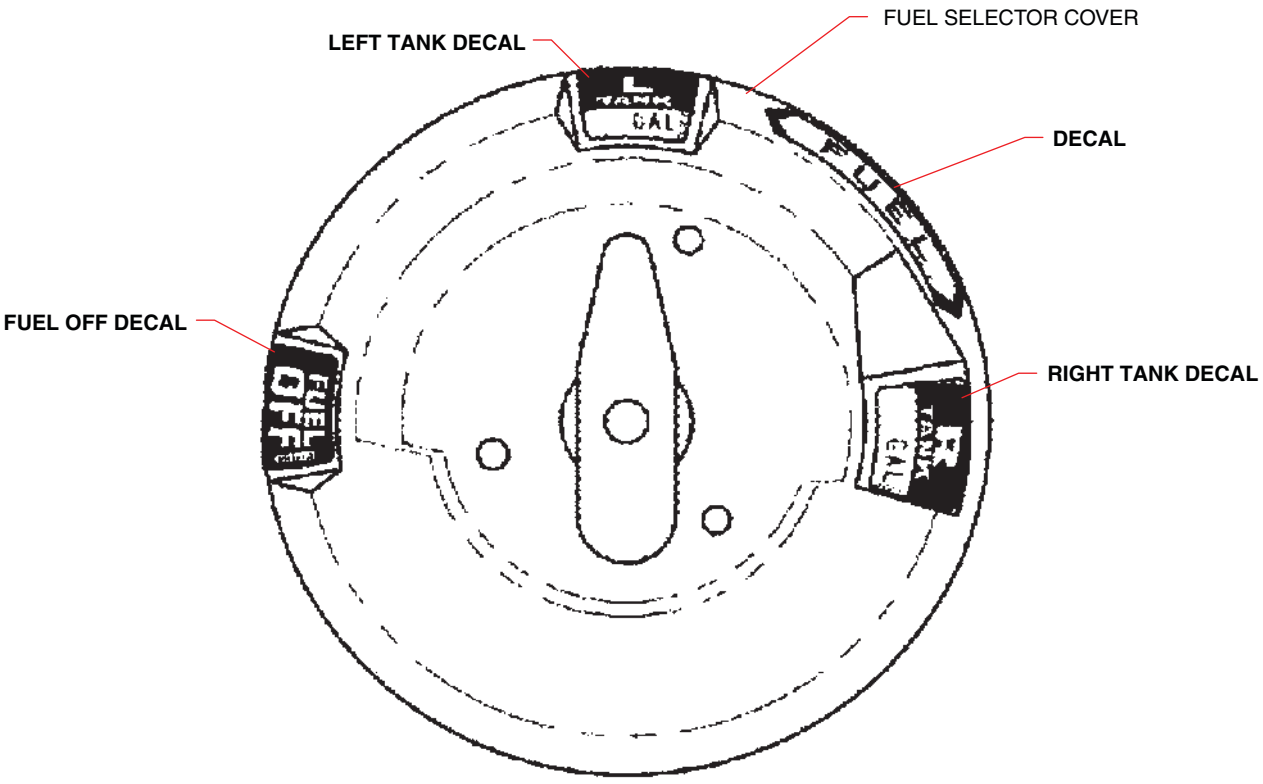
1. DECAL - OIL SPECIFICATIONS
2. PLACARD - OPEN
3. PLACARD - LATCH
4. PLACARD - MAXIMUM GROSS WEIGHT (1)
5. PLACARD - HAT SHELF LIIMITS
6. PLACARD - COMPASS DEVIATION (2)
7. PLACARD - FLAP LEVER
8. PLACARD - RUDDER TRIM
9. PLACARD - STABILATOR TRIM
10. PLACARD - BAGGAGE LIMITS (3)
11. PLACARD - BAGGAGE LIMITS (4)
12. DECAL - BAGGAGE / PASSENGER LOAD VS. OP. CATEGORY
13. PLACARD - LIMITATIONS INFORMATION
14. FUEL SELECTOR DECALS - OFF, FUEL, L TANK, AND R TANK (7)
15. DECAL - CABIN AIR PULL - CLOSED
16. DECAL - FWD (5) (8)
17. DECAL - HAND HOLD (6)

(1) With Garmin G1000 in 2843701, 2843703 and up; or with Garmin G500 in 2843736 and up.
(2) Not installed with Garmin G1000 in 2843701, 2843703 and up.
(3) With Baggage Door.
(4) Without Baggage Door.
(5) With Garmin G1000 in 2843701, 2843703 and up.
(6) 2843852 and up.
(7) See Sheet 2.
(8) See Sheet 2 FOR GMU 11.

Interior Placards
Figure 1 (Sheet 1 of 2)



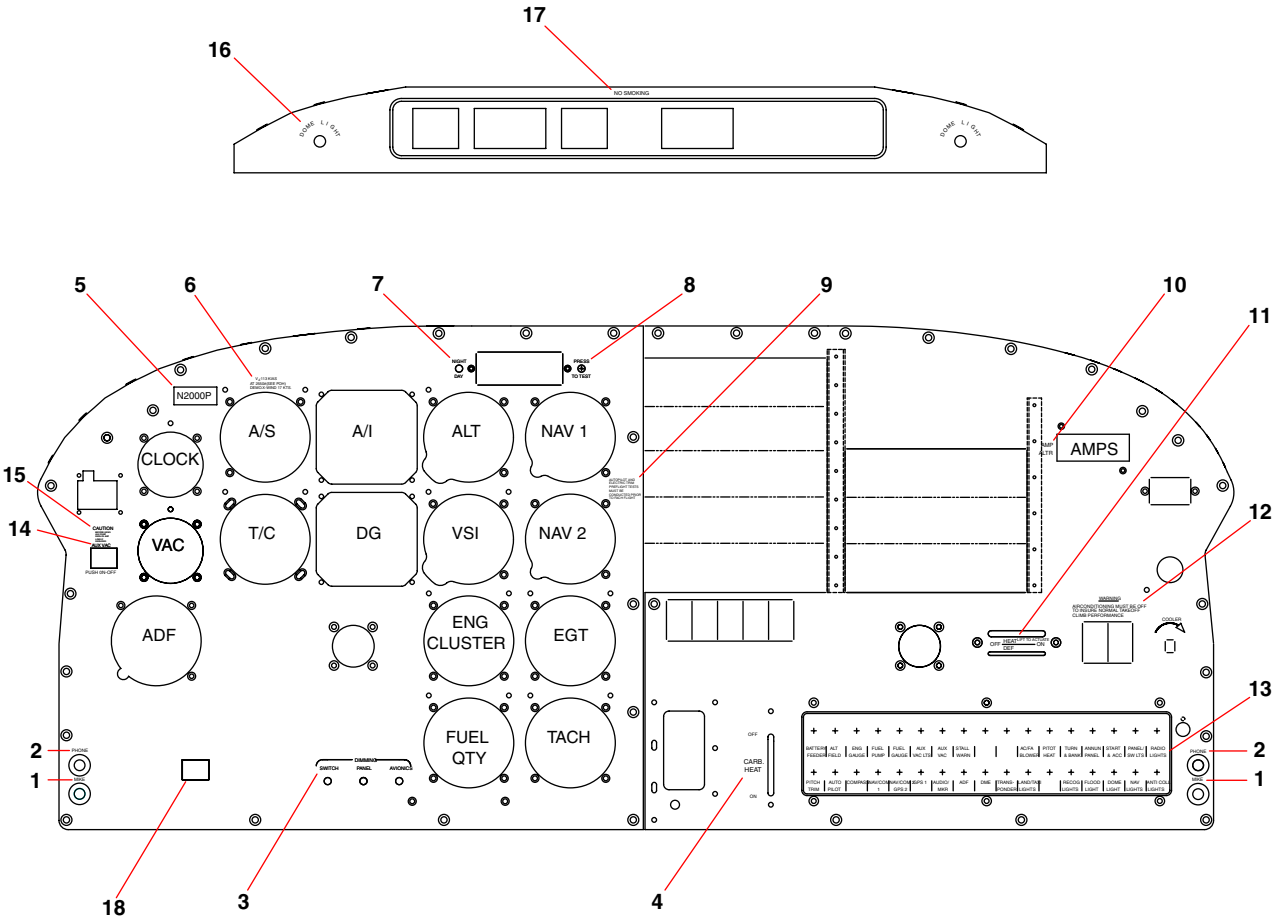
GMU 11



Interior Placards
Figure 1 (Sheet 2 of 2)

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06327 AC
36991 CD

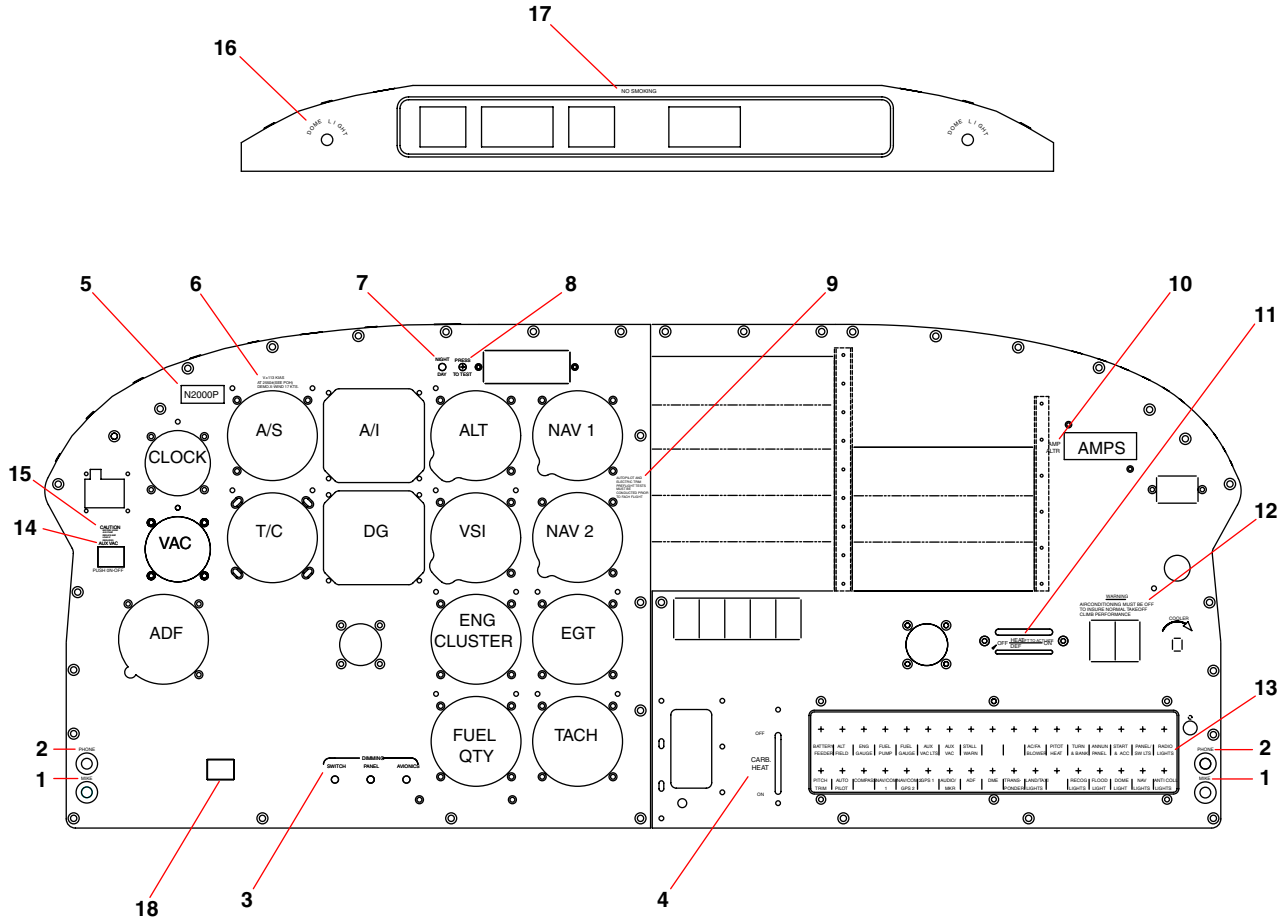


1. PLACARD - MIC
2. PLACARD - PHONE
3. PLACARD - DIMMING
4. PLACARD - CARB HEAT
5. PLATE - AIRCRAFT REGISTRATION NUMBER
6. PLACARD - AIRSPEED AND CROSSWIND
7. PLACARD - DAY/NIGHT
8. PLACARD - PRESS TO TEST
9. PLACARD - AUTOPILOT
10. PLACARD - AMPS ALTERNATOR
11. PLACARD - HEAT/DEFROST
12. PLACARD - AIR COND. (OPT)
13. PLACARD - CIRCUIT BREAKER
14. PLACARD - AUX. VAC
15. PLACARD - CAUTION
16. DECAL - DOME LIGHT
17. DECAL - NO SMOKING
18. PLACARD - ELT CAUTION

Effectivity
2843001 thru 2843202

Interior Placards
Figure 2

06327 AC
36991 CD

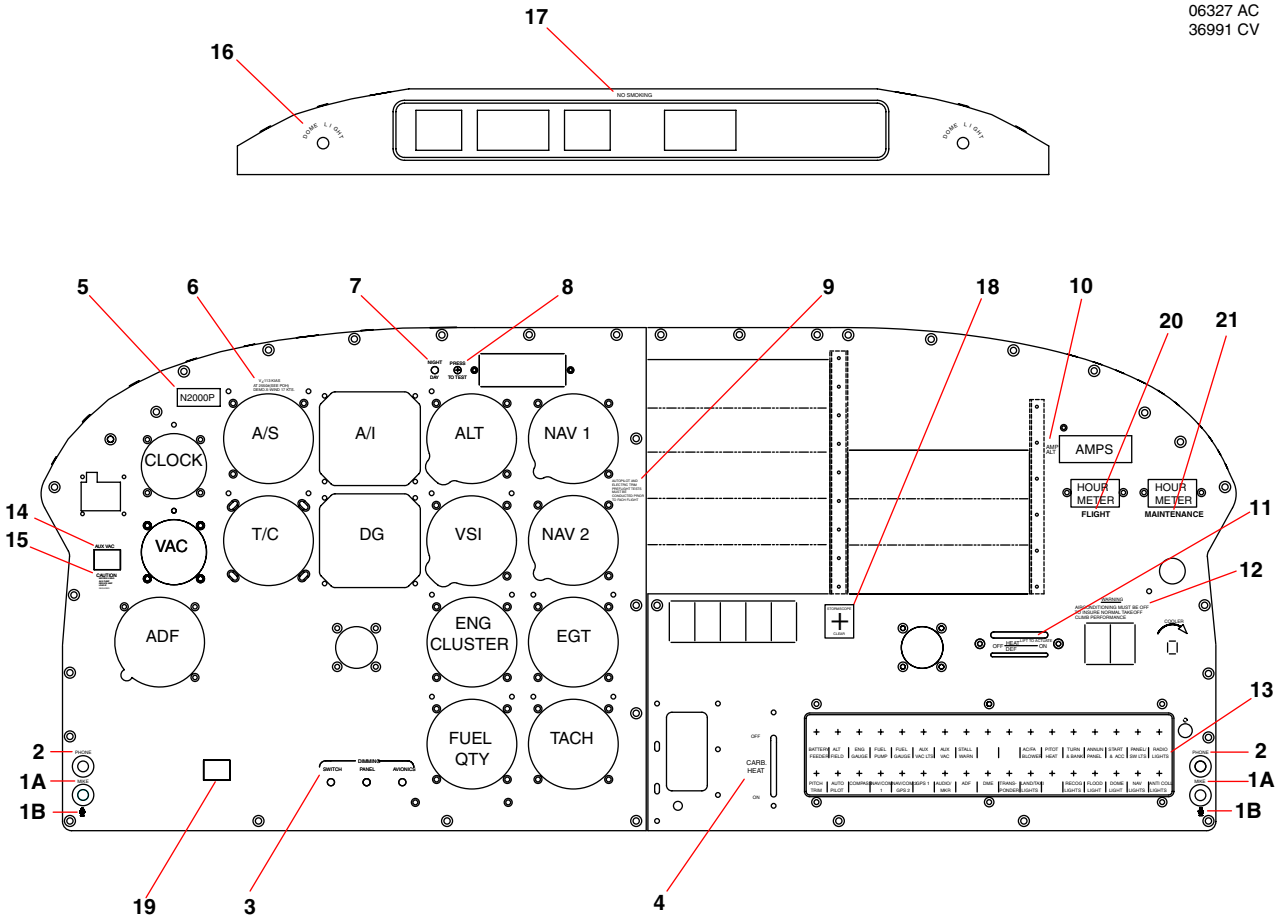


1. PLACARD - MIC
2. PLACARD - PHONE
3. PLACARD - DIMMING
4. PLACARD - CARB HEAT
5. PLATE - AIRCRAFT REGISTRATION NUMBER
6. PLACARD - AIRSPEED AND CROSSWIND
7. PLACARD - DAY/NIGHT
8. PLACARD - PRESS TO TEST
9. PLACARD - AUTOPILOT
10. PLACARD - AMPS ALTERNATOR
11. PLACARD - HEAT/DEFROST
12. PLACARD - AIR COND. (OPT)
13. PLACARD - CIRCUIT BREAKER
14. PLACARD - AUX. VAC
15. PLACARD - CAUTION
16. DECAL - DOME LIGHT
17. DECAL - NO SMOKING
18. PLACARD - ELT CAUTION

Interior Placards
Figure 3

Effectivity
2843203 thru 2843414

06327 AC
36991 CV



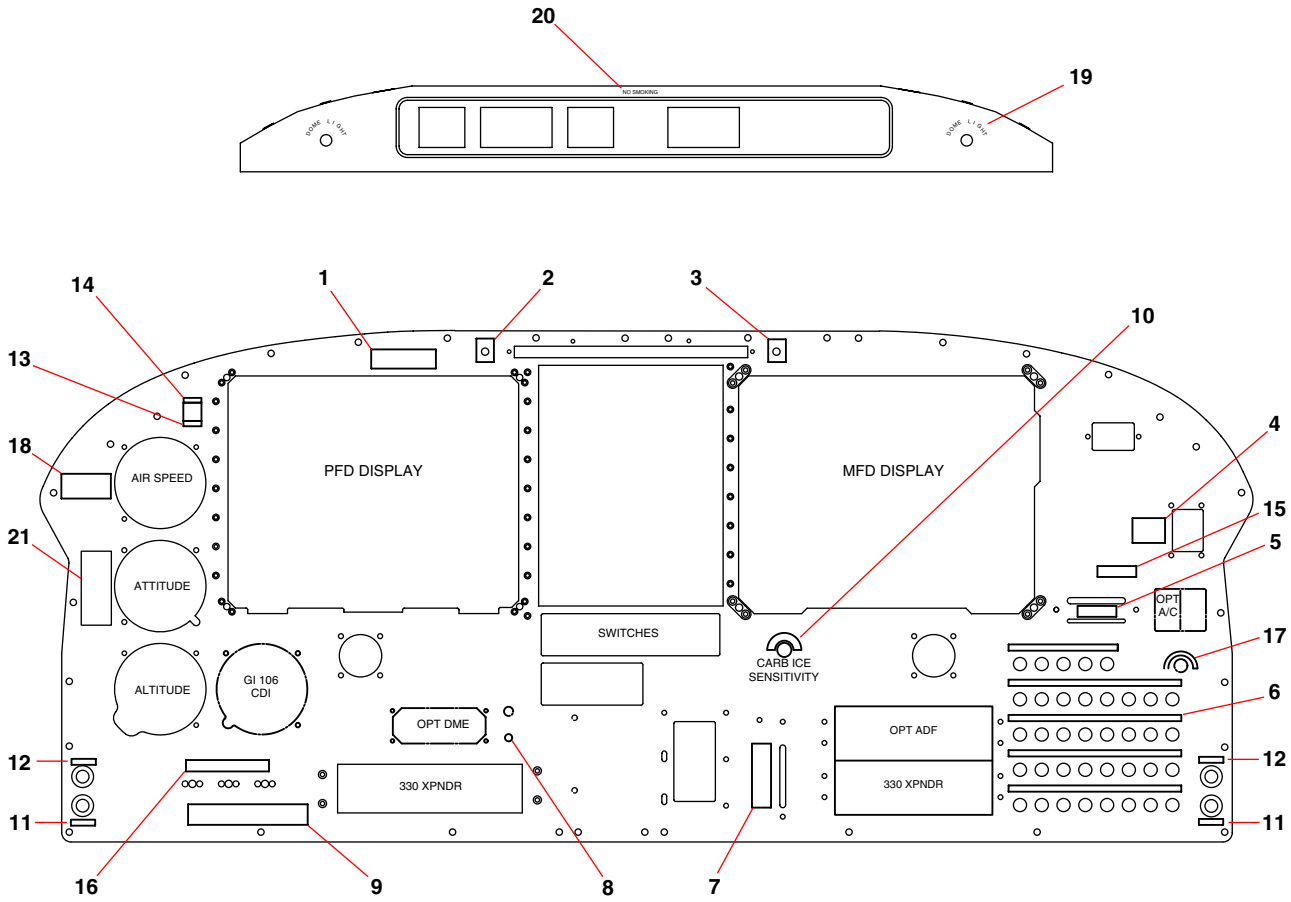
1A. PLACARD - MIC
1B. TRANSFER - MIC SYMBOL(1)
2. PLACARD - PHONE, or
TRANSFER - PHONE SYMBOL(1)
3. PLACARD - DIMMING
4. PLACARD - CARB HEAT
5. PLATE - AIRCRAFT REGISTRATION NUMBER
6. PLACARD - AIRSPEED AND CROSSWIND
7. PLACARD - DAY/NIGHT
8. PLACARD - PRESS TO TEST
9. PLACARD - AUTOPILOT
10. PLACARD - AMPS ALTERNATOR
11. PLACARD - HEAT/DEFROST
12. PLACARD - AIR COND. (OPT)
13. PLACARD - CIRCUIT BREAKER
14. PLACARD - AUX. VAC
15. PLACARD - CAUTION
16. DECAL - DOME LIGHT
17. DECAL - NO SMOKING
18. PLACARD - STORMSCOPE (OPT)
19. PLACARD - ELT CAUTION
20. TRANSFER - FLIGHT (OPT) (2)
21. TRANSFER - MAINTENANCE (2)

(1) In 2843852 and up.
(2) In 2843807 and up, installed only when optional Flight Time Meter is installed.

Effectivity
Standard or Garmin G500
2843415 thru 2843597
2843683 thru 2843700
2843702, 2843736 and up

Interior Placards
Figure 4

36991 CG
106645 B

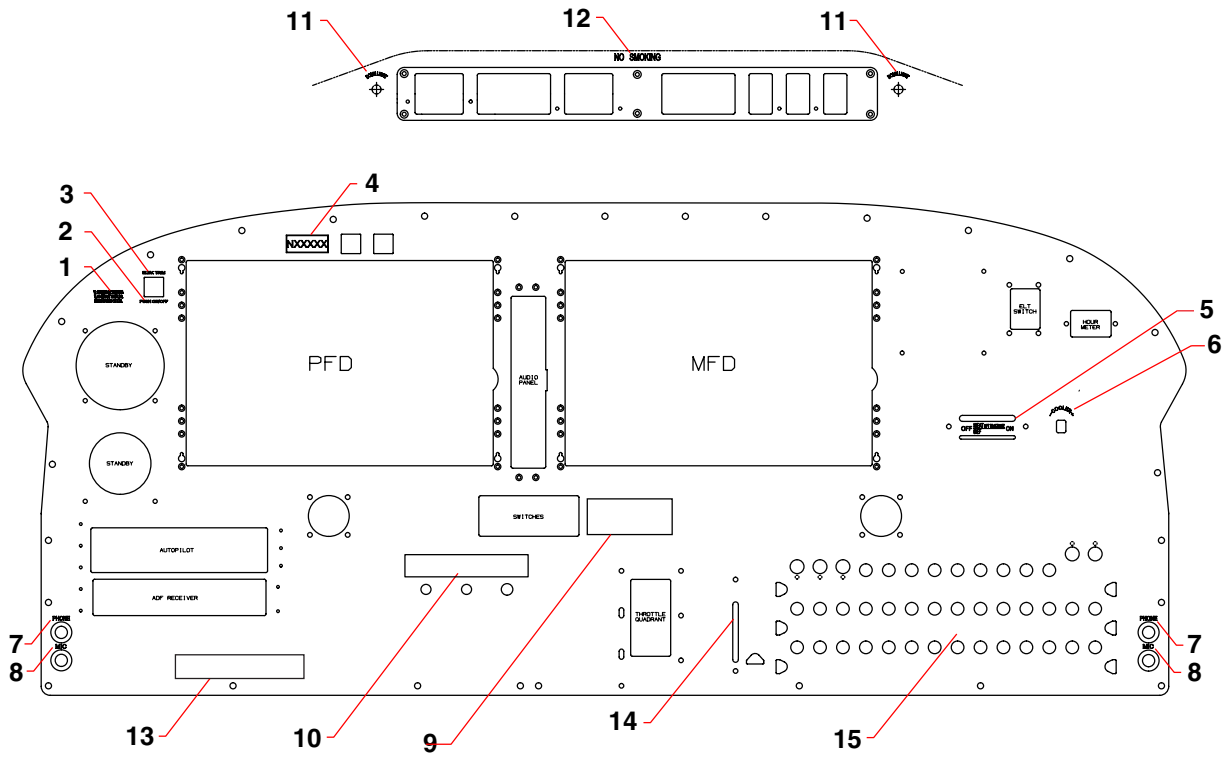


1. PLATE - AIRCRAFT REGISTRATION NUMBER
2. PLACARD - DAY/NIGHT
3. PLACARD - PRESS TO TEST
4. PLACARD - ELT CAUTION (OPTIONAL, S/N's 2843657 & UP)
5. PLACARD - HEAT/DEFROST
6. DECAL - CIRCUIT BREAKER
7. PLACARD - CARB HEAT
8. PLACARD - DME SELECT (OPT)
9. PLACARD - ALT STATIC
10. PLACARD - CARB ICE SENSITIVITY (OPT)
11. PLACARD - MIC
12. PLACARD - PHONE
13. PLACARD - PUSH ON/OFF (OPT)
14. PLACARD - ELV TRIM (OPT)
15. PLACARD - AIR COND. (OPT)
16. PLACARD - DIMMING
17. PLACARD - THERMOSTAT (OPT)
18. PLACARD - AIRSPEED
19. DECAL - DOME LIGHT
20. DECAL - NO SMOKING
21. DECAL - ATTITUDE INDICATOR

Interior Placards
Figure 5

Effectivity
with Avidyne Entegra

107510 H
107570 F
107571 A

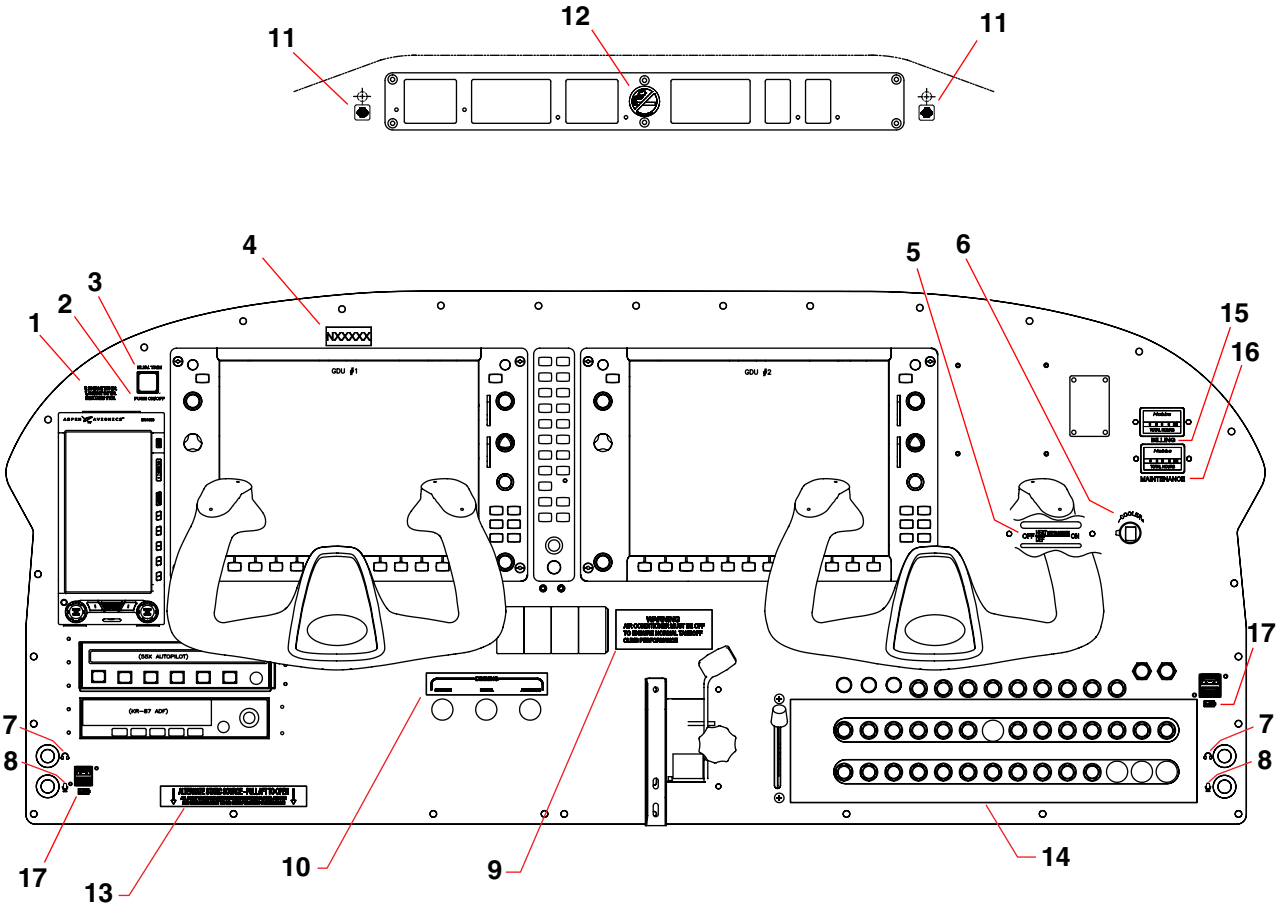


1. TRANSFER - AIRSPEED & CROSSWIND
2. TRANSFER - PUSH ON/OFF (OPT)
3. TRANSFER - ELV TRIM (OPT)
4. PLATE - AIRCRAFT REGISTRATION NUMBER
5. TRANSFER - HEAT/DEFROST
6. TRANSFER - THERMOSTAT (OPT)
7. TRANSFER - PHONE
8. TRANSFER - MIC
9. TRANSFER - AIR COND. (OPT)
10. TRANSFER - DIMMING
11. DECAL - DOME LIGHT
12. DECAL - NO SMOKING
13. E.L. PLACARD - ALTERNATE STATIC SOURCE
14. E.L. PLACARD - CARB HEAT, or
    E.L. PLACARD - ALT AIR (WITH IO-360-B4A ENGINE)
15. E.L. PLACARD - CIRCUIT BREAKERS

Effectivity
with Garmin G1000

Interior Placards
Figure 6

107510 H
108004 B

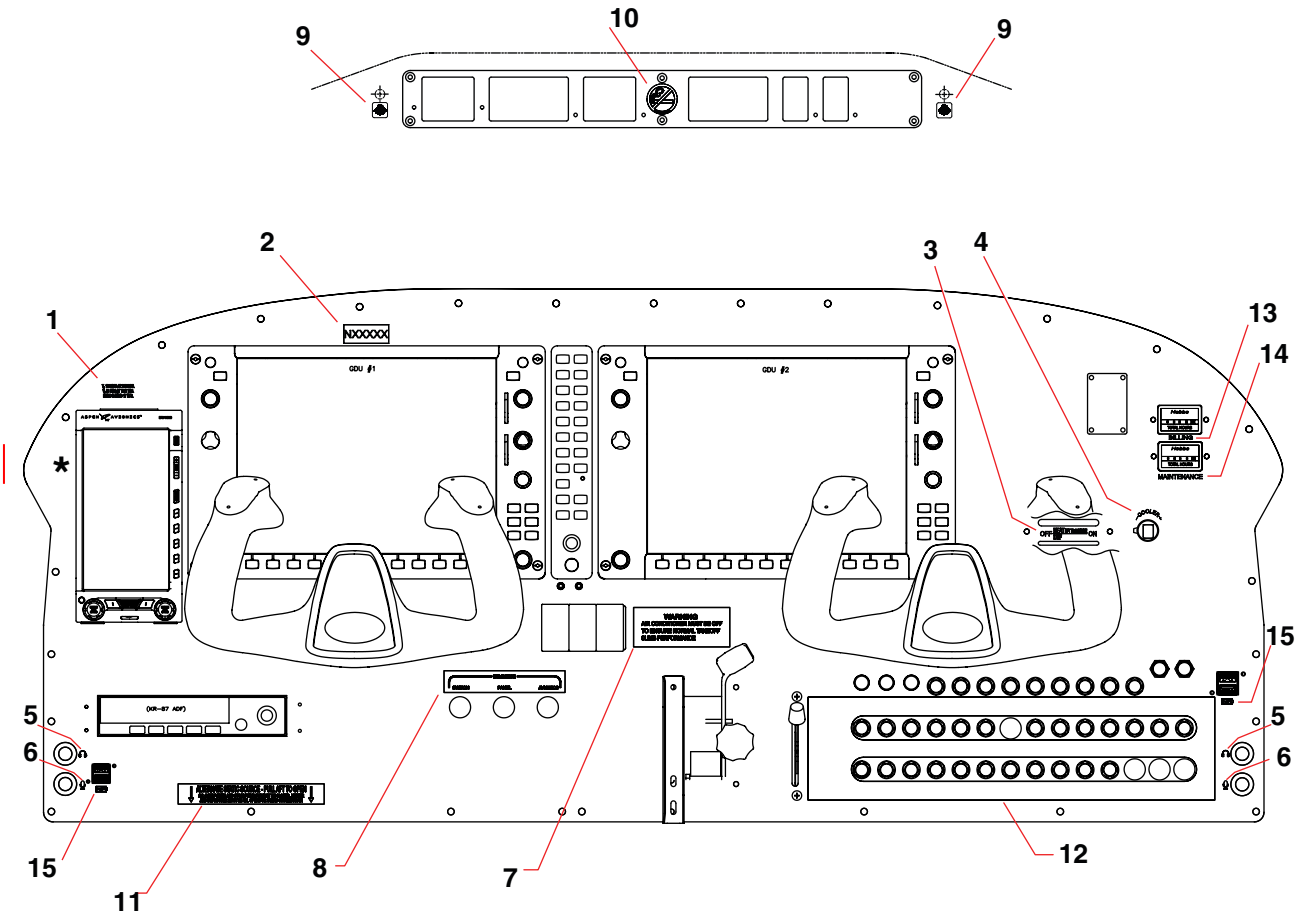


1. TRANSFER - AIRSPEED & CROSSWIND
2. TRANSFER - PUSH ON/OFF (OPT.)
3. TRANSFER - ELV TRIM (OPT.)
4. PLATE - AIRCRAFT REGISTRATION NUMBER
5. TRANSFER - HEAT/DEFROST
6. TRANSFER - THERMOSTAT (WITH A/C)
7. TRANSFER - PHONE SYMBOL
8. TRANSFER - MIC SYMBOL
9. TRANSFER - AIR CONDITIONING WARNING (WITH A/C)
10. TRANSFER - DIMMING
11. DECAL - DOME LIGHT SYMBOL
12. DECAL - NO SMOKING SYMBOL
13. E.L. PLACARD - ALTERNATE STATIC SOURCE
14. E.L. PLACARD - CIRCUIT BREAKERS & CARB HEAT, or
    E.L. PLACARD - CIRCUIT BREAKERS & ALT AIR (WITH IO-360-B4A ENGINE)
15. BILLING (OPT.)
16. MAINTENANCE (OPT.)
17. USB SYMBOL (OPT.)

Interior Placards
Figure 7

Effectivity
with Garmin G1000 NXi
2843852, 2843854–2843999

107510 K
108004 D



1. TRANSFER - AIRSPEED & CROSSWIND
2. PLATE - AIRCRAFT REGISTRATION NUMBER
3. TRANSFER - HEAT/DEFROST
4. TRANSFER - THERMOSTAT (WITH A/C)
5. TRANSFER - PHONE SYMBOL
6. TRANSFER - MIC SYMBOL
7. TRANSFER - AIR CONDITIONING WARNING (WITH A/C)
8. TRANSFER - DIMMING
9. DECAL - DOME LIGHT SYMBOL
10. DECAL - NO SMOKING SYMBOL
11. E.L. PLACARD - ALTERNATE STATIC SOURCE
12. E.L. PLACARD - CIRCUIT BREAKERS & CARB HEAT, or
    E.L. PLACARD - CIRCUIT BREAKERS & ALT AIR (WITH IO-360-B4A ENGINE)
13. BILLING (OPT.)
14. MAINTENANCE (OPT.)
15. USB SYMBOL (OPT.)

* ASPEN EBD SHOWN, GARMIN G5 IS HALF SIZE.

Effectivity
with Garmin G1000 NXi
2843823, 2881001 and up

Interior Placard
Figure 8

CHAPTER

12

SERVICING

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CHAPTER 12

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
12-List of	1	Jul 30/21	12-20-00	1	May 15/19
Effective Pages	2	Mar 6/14		2	Sep 1/16
				3	Sep 1/16
12-Table of Contents	1	Mar 15/20		4	May 31/20
	2	Jul 30/21		5	May 31/20
				6	May 31/20
12-00-00	1	Sep 1/16		7	May 31/20
	2	Sep 1/16		8	May 15/19
	3	Sep 1/16		9	Nov 15/17
	4	Mar 6/14		10	Mar 15/20
				11	Nov 15/17
12-10-00	1	Mar 6/14		12	Nov 15/17
	2	Mar 6/14		13	May 15/19
	3	May 15/19		14	Nov 15/17
	4	May 15/19		15	Nov 15/17
	5	Mar 15/20		16	Nov 15/17
	6	May 15/19		17	Nov 15/17
	7	Mar 6/14		18	Jul 30/21
	8	Mar 6/14		19	Nov 15/17
	9	Mar 6/14		20	Nov 15/17
	10	May 15/19			
	11	Mar 6/14			
	12	Mar 6/14			

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CHAPTER 12 - SERVICING

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GENERAL

This chapter covers all routine servicing of airplane, scheduled and non-scheduled, including replenishment of fuel, oil, brake fluid, oxygen (if installed), tire pressure, lubrication requirements, servicing of oleo struts with air and oil, etc. Pay special attention to all WARNINGS and CAUTIONS.

1. Aircraft Finish Care

WARNING: DO NOT USE FUELS (I.E., GASOLINE, KEROSENE, JET-A); SOLVENTS (I.E., ALCOHOL, BENZENE, CARBON TETRACHLORIDE, PAINT THINNER, ACETONE); OR WINDOW CLEANING SPRAYS TO CLEAN AIRPLANE.

The entire airplane is carefully finished inside and out to assure maximum service life. The external surfaces are coated with a durable polyurethane enamel finish.

A. Axalta Imron Paint System

CAUTION: FAILURE TO OBSERVE THE PROPER "FINISH CARE" GUIDELINES MAY RESULT IN DAMAGE OR LOSS OF SHINE OF THE AIRCRAFT PAINT. IMPROPER CARE MAY ALSO VOID THE WARRANTY REGARDING THE AIRCRAFT FINISH.

Piper aircraft use the Axalta Imron paint system. Imron polyurethane enamels are designed to provide superior protection against the elements the airplane will face throughout its long life. To maximize the benefits of Imron, use the following guidelines:

(1) For the first 30 days after painting, when the finish is still fresh:
   (a) Hand wash the aircraft often. Use fresh water only.
   (b) Avoid parking under trees or places where birds roost. If sap, bird droppings, or insect remains are discovered, rinse them off immediately. (Sap, bird droppings, or insect remains will damage the paint during this period.)

(2) For the first 60 days after painting:
   (a) To remove heavy soil, use mild liquid soap. Never use detergent.
   (b) DO NOT WAX THE AIRCRAFT WITHIN 60 DAYS OF PAINTING!

(3) For long term paint finish protection:
   (a) Park in a sheltered area (i.e., hangar) whenever possible.
   (b) Wash your aircraft often, especially when exposed to salty, dusty, acidic or alkaline environments.
   (c) Hand washing is preferred. If using a pressure washer, avoid spraying areas that have visible chips in the paint or are freshly painted.
   (d) When washing your aircraft, use non-abrasive, neutral pH (non acidic or alkaline) detergent. Do not use solvent-based solutions for washing large areas. Avoid abrasive cleaners, chemicals, abrasive wax, or brushes.
   (e) Avoid washing with stiff bristles. Soft cloth and felt brushes are recommended.
   (f) Never use a scraper to remove ice or snow from painted surfaces. Ice and snow should be brushed, not scraped off.
   (g) Do not allow spilled gasoline, oil, anti-freeze, hydraulic fluid, or windshield washer fluid to stand on the paint – remove immediately by rinsing with water. With some oils and hydraulic fluids, allowable exposure time before staining occurs is significantly shortened if fluid or paint surface is hot. (The finish may be permanently damaged.)
   (h) Do not wash the aircraft using extremely hot water or while the surface is hot. (Do not wash aircraft in the hot sun.)

(i) Never wipe the finish with a dry cloth, always use fresh water.

(j) Have paint nicks or scratches touched up as soon as possible to maintain the aircraft's corrosion protection.

(4) Should your aircraft's finish become damaged, have it repaired as soon as possible. Contact a reputable repair shop near you and specify the Axalta Imron quality used for the original finish. This will help assure the best possible color match and the same durability and appearance as when your aircraft was new.

To summarize, Piper aircraft using the Axalta Imron paint system need special attention in the early days of ownership.

B. Cleaning

CAUTION: IF PAINT IS LESS THAN SIX MONTHS, SEE "AXALTA IMRON PAINT SYSTEM" ON PAGE 12001.

CAUTION: DO NOT DIRECT ANY STREAM OF WATER OR CLEANING SOLUTION AT THE OPENINGS IN THE PITOT HEAD, STATIC PORTS, ALTERNATE STATIC PORTS OR FUSELAGE DRAINS.

The airplane should be washed with a mild soap and water solution. Harsh abrasives or alkaline soaps or detergents could scratch painted or plastic surfaces or corrode metal. Cover areas where a cleaning solution could cause damage. To wash the airplane use the following procedure:

(1) Flush away loose dirt with water.

(2) Apply cleaning solution with a soft cloth, a sponge or a soft brush.

(3) To remove exhaust stains, allow the solution to remain on the surface longer.

(4) To remove stubborn oil and grease stains, use a soft cloth dampened with naphtha - sparingly. Flush immediately with a mild soap and water solution.

(5) Rinse all surfaces thoroughly.

(6) Any good automotive wax may be used to protect and preserve painted surfaces. Soft cleaning cloths or a chamois should be used to prevent scratches when cleaning or polishing. A heavier coat of wax on leading surfaces will reduce the abrasion problems in these areas.

2. Cleaning

A. Exterior Surfaces

See "Aircraft Finish Care" on page 12001 and "Windshield and Windows" on page 12003.

B. Engine Compartment

Before cleaning the engine compartment, place strips of tape on the magneto vents to prevent any solvent from entering these units.

(1) Place a pan under the engine to catch waste.

CAUTION: DO NOT SPRAY SOLVENT INTO THE ALTERNATOR, STARTER, VACUUM PUMP(S), AIR INTAKE AND ALTERNATE AIR INLETS.

(2) With the engine cowling removed, spray or brush the engine with solvent or a mixture of solvent and degreaser, as desired. It may be necessary to brush areas that were sprayed where heavy grease and dirt deposits have collected in order to clean them.

(3) Allow the solvent to remain on the engine from five to ten minutes; then rinse the engine clean with additional solvent and allow to dry.

CAUTION: DO NOT OPERATE ENGINE UNTIL EXCESS SOLVENT HAS EVAPORATED OR OTHERWISE BEEN REMOVED.

(4) Remove the protective covers from the magnetos.

(5) Lubricate controls, bearing surfaces, etc., per Lubrication Charts,12-20-00.

C. Landing Gear

(1) Struts and Torque Links

Before cleaning the landing gear struts and torque links, remove wheel pants and place a plastic cover or similar material over the wheel and brake assembly.

(a) Place a pan under the gear to catch waste.

(b) Spray (low pressure only) or brush the gear area with solvent or a mixture of solvent and degreaser.

(c) Allow the solvent to remain on the gear for five to ten minutes. Rinse gear with additional solvent and allow to dry.

(d) Remove cover from wheel and remove the catch pan.

(e) Lubricate gear per Lubrication Chart, 12-20-00.

(f) Reinstall wheel pants, if not proceeding to wheels and brakes, below.

(2) Wheels and Brakes

CAUTION: DO NOT USE HIGH PRESSURE SPRAY WASH EQUIPMENT. ITS USE CAN INJECT SOAP SOLUTION AND WATER INTO THE WHEEL BEARINGS AND OTHER INTERNAL CAVITIES RESULTING IN CORROSION AND REDUCED SERVICE LIFE.

(a) Remove wheel pants, if not already removed, above.

(b) Hand wash wheels and brakes with a mild soap and water solution.

(c) Rinse with low-pressure spray.

(d) Lubricate gear per Lubrication Chart, 12-20-00, if not already done, above.

(e) Reinstall wheel pants.

D. Windshield and Windows

WARNING: DO NOT USE GASOLINE, ALCOHOL, BENZENE, CARBON TETRACHLORIDE, THINNER, ACETONE OR WINDOW CLEANING SPRAYS.

(1) Remove dirt, mud, etc., from exterior surfaces with clean water.

(2) Wash with mild soap and warm water, or an aircraft plastic cleaner using a soft cloth or sponge and a straight rubbing motion. Do not rub surfaces harshly.

(3) Remove oil and grease with a cloth moistened with kerosene.

(4) After cleaning plastic surfaces, apply a thin coat of hard polishing wax. Rub lightly with a soft cloth. Do not use a circular motion.

(5) A severe scratch or mar in plastic can be removed by using jeweler's rouge to rub out the scratch. Smooth both sides and apply wax.

(6) To improve visibility through windshield and windows during flight through rain, a rain repellent such as REPCON should be applied to windshield and windows. The surfaces of the windshield and windows treated becomes so smooth that water beads up and readily flows off the surface. Apply this product in accordance with the manufacturer's instructions. (See 91-10-00, Consumable Materials and Vendor Contact Information.)

E. Headliner, Side Panels and Seats

(1) Clean headliner, side panels, and seats with a stiff brush and vacuum where necessary.

WARNING: SOLVENT CLEANERS REQUIRE ADEQUATE VENTILATION.

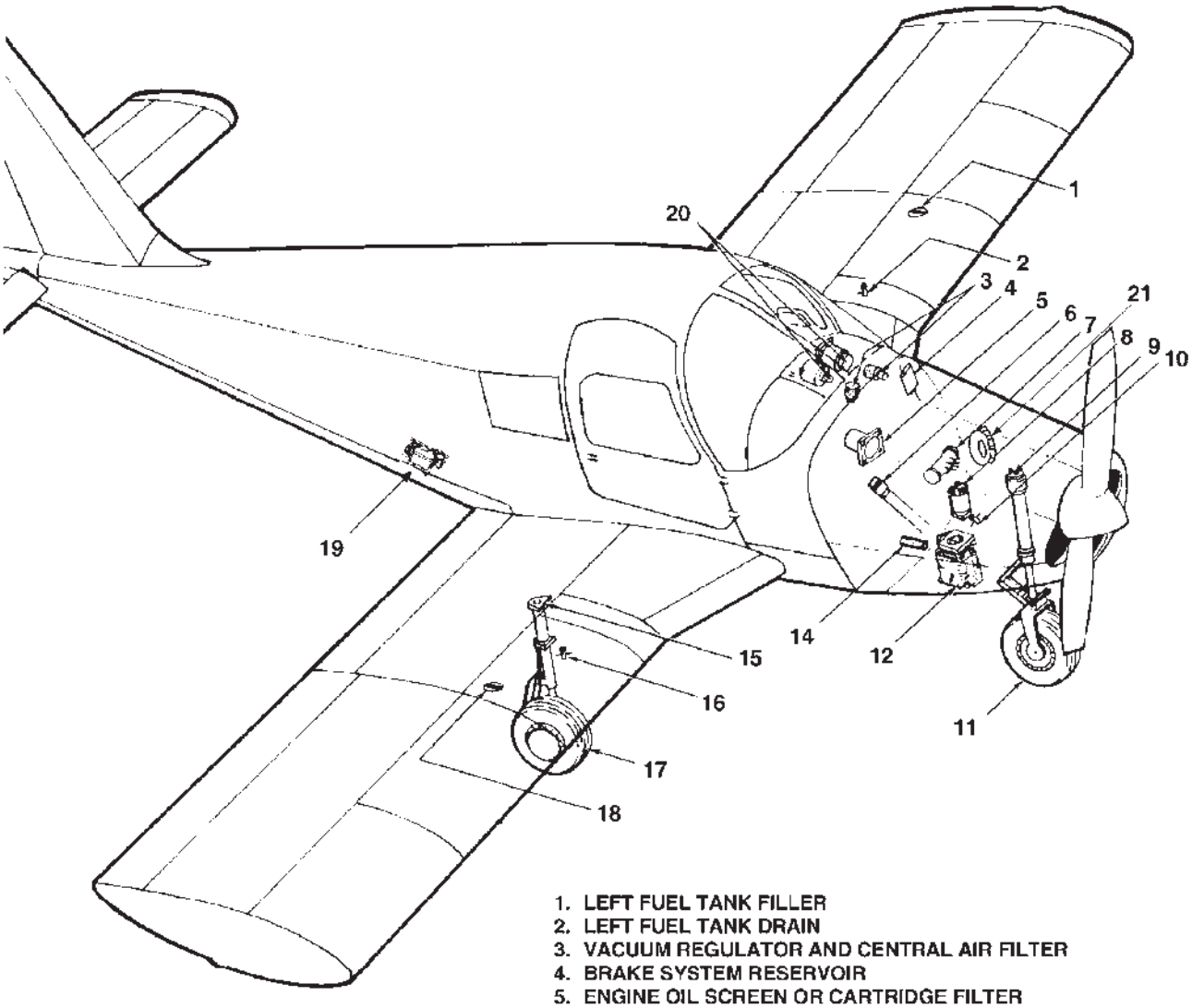
(2) Soiled upholstery, except leather, may be cleaned by using an approved air drying type cleaner or foam upholstery cleaner. Carefully follow the manufacturer's instructions. Avoid soaking or harsh rubbing.

(3) Leather material should be cleaned with saddle soap or mild soap and water.

F. Carpets

Use a small whisk broom or vacuum to remove dirt. For soiled spots, use a non-inflammable dry-cleaning fluid.

REPLENISHING



1. LEFT FUEL TANK FILLER
2. LEFT FUEL TANK DRAIN
3. VACUUM REGULATOR AND CENTRAL AIR FILTER
4. BRAKE SYSTEM RESERVOIR
5. ENGINE OIL SCREEN OR CARTRIDGE FILTER
6. ENGINE OIL FILLER / INDICATOR
7. ELECTRIC FUEL PUMP FILTER
8. FUEL DRAIN SCREEN
9. NOSE OLEO SHOCK STRUT FILLER
10. ENGINE FUEL DRAIN
11. NOSE TIRE
12. ENGINE CARBURETOR
13. NOT USED
14. ENGINE OIL SUCTION SCREEN
15. LEFT AND RIGHT OLEO SHOCK STRUT FILLER
16. RIGHT FUEL TANK DRAIN
17. LEFT AND RIGHT TIRE
18. RIGHT FUEL TANK FILLER
19. BATTERY
20. INSTRUMENTS (WITHOUT CENTRAL AIR FILTER) FILTER
21. ENGINE INDUCTION AIR FILTER CARTRIDGE

Service Points
Figure 1

1. Service Points

See Figure 1.

2. Fuel System

A. Fuel Tanks

(1) Filling

The fuel tanks of each wing are filled through filler necks located on the forward slope of the wings. Each wing tank holds a capacity of 25 U.S. gallons. Observe all required safety precautions for handling gasoline. Fill the tanks with fuel as specified on the placard adjacent to the filler neck.

(2) Water Contamination

WARNING: PILOTS, OWNERS, OPERATORS, MAINTENANCE, AND SERVICE PERSONNEL SHOULD ASSUME SOME WATER EXISTS IN THE FUEL SYSTEM.

Water may enter the fuel tank system via any penetration in the wing fuel tank and from moisture condensation inside the tank. Water in the fuel may come out of solution, settle and make its way to a drain location in the form of a blob, pea, or BB-shaped translucent mass found at the bottom of the sampler cup. Water suspended in the fuel may lead to a cloudy or hazy appearance in the sampler cup. Water may have dissolved in the fuel, but conditions have not yet occurred to cause the water to come out of solution and perhaps adhere to the dry tank upper surface or walls (similar to condensation).

See FAA Special Airworthiness Information Bulletin (SAIB) No. CE-12-06 for additional information.

(3) Draining Moisture

CAUTION: WHEN DRAINING ANY AMOUNT OF FUEL, INSURE THAT NO FIRE HAZARD EXISTS BEFORE STARTING ENGINE.

NOTE: There is no single point of drainage that can be used to check for all fuel system contaminants simultaneously.

The fuel system should be drained daily prior to first flight and after refueling to avoid the accumulation of water sediment. Each aluminum fuel tank is equipped with an individual quick drain located at the lower inboard rear corner of the tank. This allows each wing to be drained individually. The fuel strainer (gascolator) is provided with a quick drain valve (see Figure 2). Drain fuel tanks and strainer per the following:

With the airplane in the normal ground attitude and starting at the highest drain location, check all drain locations for contaminants before every flight, whether or not refueling has occurred. Have fuel sample disposal provisions and proper lighting at your disposal to properly check for fuel tank system contamination.

(a) Drain at least one cup of fuel (using a clear sampler cup) from each drain location:

1) drain each tank through its individual quick drain located at the lower inboard rear corner of the tank;

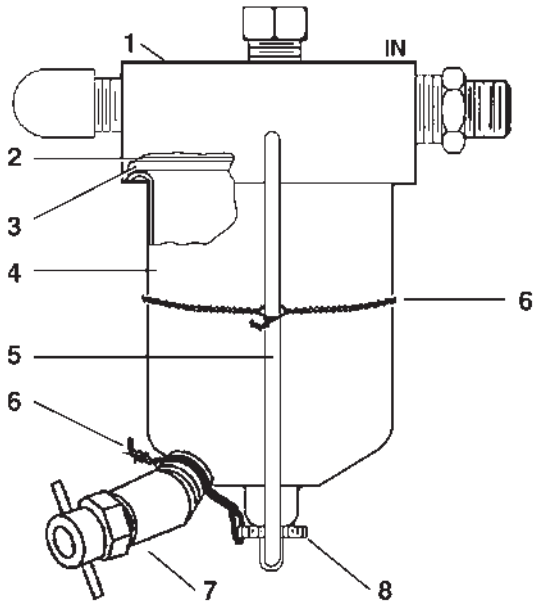
2) drain the fuel strainer as required to completely flush its contents in each of the fuel selector positions. Drain the fuel strainer by opening the quick drain on the strainer.

(b) Check for water, clarity, cloudiness, haze, proper fuel type/grade (i.e.; 100LL is light blue in tint, jet fuel is clear or yellowish), odor, or other contaminants.

(c) Allow time between fueling and draining. It takes time for any contaminates to settle to sump area prior to draining tanks.

(d) If any contamination is detected in the fuel tank system, thoroughly drain all drain locations again.

1. COVER
2. SCREEN
3. GASKET
4. STRAINER BOWL
5. BAILWIRE
6. SAFETY WIRE
7. DRAIN VALVE
8. BAIL ASSEMBLY



Gascolator
Figure 2

(e) If contamination is observed, take further samples until the fuel appears clear, and gently rock the airplane in both the roll and pitch axis to move any additional contaminants to the drain points.

(f) Take repeated samples from all drain locations until all contamination has been removed.

(g) If contaminants are still present, do not fly the airplane. Have qualified maintenance personnel drain and purge the fuel tank system. Remove all evidence of contamination prior to further flight.

(4) Draining Entirely

CAUTION: WHEN DRAINING ANY AMOUNT OF FUEL, ENSURE THAT NO FIRE HAZARD EXISTS BEFORE STARTING ENGINE.

Fuel may be drained from the system by opening the valve at the inboard end of each aluminum fuel tank. The flush type drain valve requires the drain cup pin to hold the valve open. The remaining fuel in the system may be drained through the filter bowl. Either wing may be drained by closing the selector valve and then draining as desired.

B. Anti-icing Fuel Additive

Use of an anti-icing fuel additive conforming to MIL-DTL-85470 is permitted. See Lycoming Service Letter No. L172C (or later).

CAUTION: ENSURE ADDITIVE IS DIRECTED INTO THE FLOWING FUEL STREAM. ADDITIVE FLOW SHOULD START AFTER AND STOP BEFORE FUEL FLOW.

CAUTION: DO NOT LET CONCENTRATED ADDITIVE DIRECTLY CONTACT PAINTED SURFACES OR INTERIOR SURFACES OF FUEL TANK.

CAUTION: SOME FUELS HAVE ANTI-ICING FUEL ADDITIVES PRE-BLENDED, SO NO FURTHER BLENDING IS NECESSARY.

CAUTION: ANTI-ICING FUEL ADDITIVES MUST NOT BE USED AS A SUBSTITUTE FOR PREFLIGHT DRAINING OF FUEL SYSTEM DRAINS.

The additive must:

(1) Be uniformly blended with fuel while refueling.

(2) Not exceed 0.15 percent by volume of the refueled quantity.

(3) Be blended not less than 0.10 percent by volume.

One and one-half liquid ounces per ten gallons of fuel falls in this range. Use blender supplied by additive manufacturer. Use the above information, and carefully follow manufacturer's mixing or blending instructions.

3. Oil System

CAUTION: DO NOT INTRODUCE ANY TRADE ADDITIVE TO THE BASIC LUBRICANT UNLESS RECOMMENDED BY THE ENGINE MANUFACTURER.

Engine oil level should be checked before each flight. The engine oil and full flow cartridge filter should be changed every 50 hours or four months, whichever occurs first. When changing oil, remove and clean oil screen(s). Use a quality brand Aviation Grade Oil of the proper season viscosity. See Lubrication "Chart 7" on page 122018.

A. Oil Sump

(1) Draining

CAUTION ENGINE MUST BE WARMED TO OPERATING TEMPERATURE TO ENSURE COMPLETE DRAINING OF OLD OIL.

To drain the oil sump, provide a suitable container with a minimum capacity of that required to fill the sump. Remove the engine cowling and open the oil drain located on the underside of the engine by pushing the arms of the drain up and turning counterclockwise. This will hold the drain in the open position.

(2) Filling

The oil sump should normally be filled with oil to the mark on the engine dipstick. The quantity of oil required for the engine may be found in 6-00-00, Chart 1. The specified grade of oil may be found in the Lubrication "Chart 7" on page 122018; on the inside surface of the engine oil filler access door; or in the appropriate vendor publication. To service the engine with oil, open the quick release oil filler access door on top of the cowl, and remove the oil filler cap with dipstick.

(3) Oil Screen (Suction)

CAUTION: EXAMINE FOR INTERNAL DAMAGE IF MATERIAL SUCH AS CHIPS OR PARTICLES FROM BEARINGS ARE TRAPPED IN FILTER. EVIDENCE OF INTERNAL ENGINE DAMAGE FOUND IN THE OIL FILTER JUSTIFIES FURTHER EXAMINATION TO DETERMINE CAUSE.

The oil suction screen, located on the bottom aft end of the engine sump is installed horizontally. To remove, cut the safety wire and remove the hex head plug. The screen should be cleaned at each oil change to remove any accumulation of sludge and to examine for metal filings or chips. If metal particles are found in the screen, the engine should be examined for internal damage. To avoid possible damage to the screen, after cleaning and inspection, place the screen inside the recess in the hex head plug, and insert the screen into the housing When certain that the screen is properly seated, tighten and safety the plug with MS-20995-C41 safety wire.

(4) Recommendations for Changing Oil

The engine manufacturer recommends that the oil and filter for engines equipped with full flow oil filters be changed each 50 hours of engine operation or every four months, whichever comes first. Always start and warm the engine to operating temperature before performing an oil change.

NOTE: See latest revision of Lycoming's Service Bulletin No. 480. And, the latest revision of Lycoming Service Instruction No. 1014 for lubricating oil recommendations.

(a) For the first 50 hours of engine operation when new and following the replacement of one or more cylinders, or until oil consumption has stabilized, use mineral oil that meets MIL-L-6082 or SAE J 1966 in accordance with the engine manufacturer's instructions.

(b) After the first 50 hours of engine operation when new and following the replacement of one or more cylinders, or until oil consumption has stabilized, use ashless dispersant oil that meets MIL-L-22851 or SAE J 1899 in accordance with the engine manufacturer's instructions.

(c) A change to additive oil should be made with a degree of caution in engines that have been operating on straight mineral oil for several hundred hours, since the cleaning action of some additive oils will tend to loosen sludge deposits and cause plugged oil passages. On any engine that has been operating on straight mineral oil, and is known to be in excessive dirty condition, do not switch to an additive or compounded oil until the engine has been overhauled.

(d) When changing from straight mineral oil to ashless dispersant oil, the following precautionary steps should be taken:

1) Do not add additive oil to straight mineral oil. Drain the straight mineral oil from the engine and fill with additive oil.

2) Do not operate the engine longer than five hours before the first oil change.

3) Check all oil screens for evidence of sludge or plugging and change oil every ten hours if sludge conditions are evident. Resume normal oil drain periods after sludge conditions improve.

B. Oil Filter

(1) The oil filter should be replaced after each 50 hours of engine operation. This is accomplished by removing the lockwire from the bolt head at the end of the filter housing, loosening the bolt, and removing the filter assembly from the adapter.

CAUTION: EVIDENCE OF INTERNAL ENGINE DAMAGE FOUND IN THE OIL FILTER JUSTIFIES FURTHER EXAMINATION TO DETERMINE CAUSE.

(2) Before discarding the throwaway filter, remove the element for inspection by using a Champion cutter tool, CT-470, available from Champion Aerospace (http://www.championaerospace.com), Liberty, South Carolina, 29657. It will cut open any spin on type oil filter for inspection. Examine the material trapped in the filter for evidence of internal engine damage, such as chips or particles from bearings. In new or newly overhauled engines, some small particles of metallic shavings might be found; these are generally of no consequence and should not be confused with particles produced by impacting, abrasion or pressure.

(3) After the filter has been replaced, tighten the cartridge to 15 to 18 foot-pounds of torque. Lockwire the bolt through the loops on the side of the housing to the drilled head of the thermostatic valve. Be sure the lockwire is replaced at both the attaching bolt head and the thermostatic oil cooler bypass valve. Use MS-20995-C41 safety wire.

4. Landing Gear

The landing gear consists of tires, brakes and oleo strut assemblies. These should be inspected for scored piston tubes, possible hydraulic fluid leakage and security and condition of all connection points. Check the brake linings for wear and frayed edges, and brake discs for scoring. Replace if necessary. Minor servicing is described in the following paragraphs. See Chapter 32 for additional information.

A. Oleo Struts

(1) Servicing Oleo Struts (PIR-PPS50045, Rev. L.)

WARNING: DO NOT RELEASE AIR BY REMOVING THE STRUT VALVE CORE OR FILLER PLUG. DEPRESS THE VALVE CORE PIN UNTIL STRUT CHAMBER PRESSURE HAS DIMINISHED.

CAUTION: DO NOT EXCEED SPECIFIED TUBE EXPOSURES.

CAUTION: DIRT AND FOREIGN PARTICLES ACCUMULATE AROUND THE FILLER PLUGS OF THE LANDING GEAR STRUTS. THEREFORE, BEFORE ATTEMPTING TO REMOVE THESE PLUGS, THE TOPS OF THE STRUTS SHOULD BE CLEANED WITH COMPRESSED AIR AND/OR WITH A DRY SOLVENT.

Air-oil struts are incorporated in each landing gear oleo to absorb the shock resulting from the impact of the wheels on the runway during landing. To obtain proper oleo action, the nose gear oleo strut must have approximately 3.25 ± .25 inches of piston tube exposed, while the main gear struts require approximately 4.50 ± .50 inches of tube exposure. These measurements are taken with the airplane on a level surface under normal static load (empty weight of airplane plus full fuel and oil).

If the strut has less tube exposed than that prescribed, determine whether it needs air or oil as follows:

(a) Raise the airplane on jacks.

(b) With the strut extended, remove the cap from the air valve at the top of the housing and depress the valve core to allow air to escape from the strut piston until it is fully compressed.

(c) Allow the foam from the air-oil mixture to settle and then determine if oil is visible up to the bottom of the filler plug hole.

(d) If the oil is visible at the bottom of the hole, then all that is required is the valve be checked for unsatisfactory conditions and air added as described in Inflating Oleo Struts, below.

(e) Should fluid be at any level below the bottom of the filler plug hole, the oleo should be checked for leaks, etc, and oil added as described in Filling Nose Gear Oleo Strut, below; or, Filling Main Gear Oleo Strut, below, respectively. Minor repair procedures are in Chapter 32.

(2) Filling Nose Gear Oleo Strut

To fill the nose gear oleo strut with either a large or small amount of hydraulic fluid (MIL-PRF-5606):

(a) Raise the airplane on jacks until the nose wheel is completely clear of the ground. (Refer to 7-10-00.)

(b) Place a pan under the gear to catch spillage.

(c) If not previously accomplished, remove the engine cowling and relieve air from the strut housing chamber by removing the cap from the air valve and depressing the valve core.

(d) Fill the strut chamber using one of the following methods:

1) Method 1:

a) Remove valve core from filler plug at the top of strut housing. Do not remove plug.

b) Attach one end of a clear plastic hose to the valve stem of the filler plug and submerge the other end in a container of hydraulic fluid. Ascertain that the end of the hose on the valve stem is tight and the fluid container is approximately equal in height to the top of the strut housing.

c) Fully compress and extend strut (at least 10 and no more than 12 inches of piston tube exposed) to draw fluid from the fluid container and expel air from strut chamber. Watch the fluid pass through the plastic hose. Stop when bubbles cease to flow through the hose - i.e. - when the strut is full and no air is present in the chamber.

d) When air bubbles cease to flow through hose, compress strut fully and remove hose from the valve stem.

e) With strut compressed, remove filler plug and check that the fluid level is visible up to the bottom of filler plug hole.

f) Install core in filler plug. Apply an appropriate thread lubricant to threads of filler plug and install plug in top of strut housing. Torque plug to 45 foot-pounds.

2) Method II:

a) Remove filler plug from top of strut housing.

b) Raise strut piston until fully compressed.

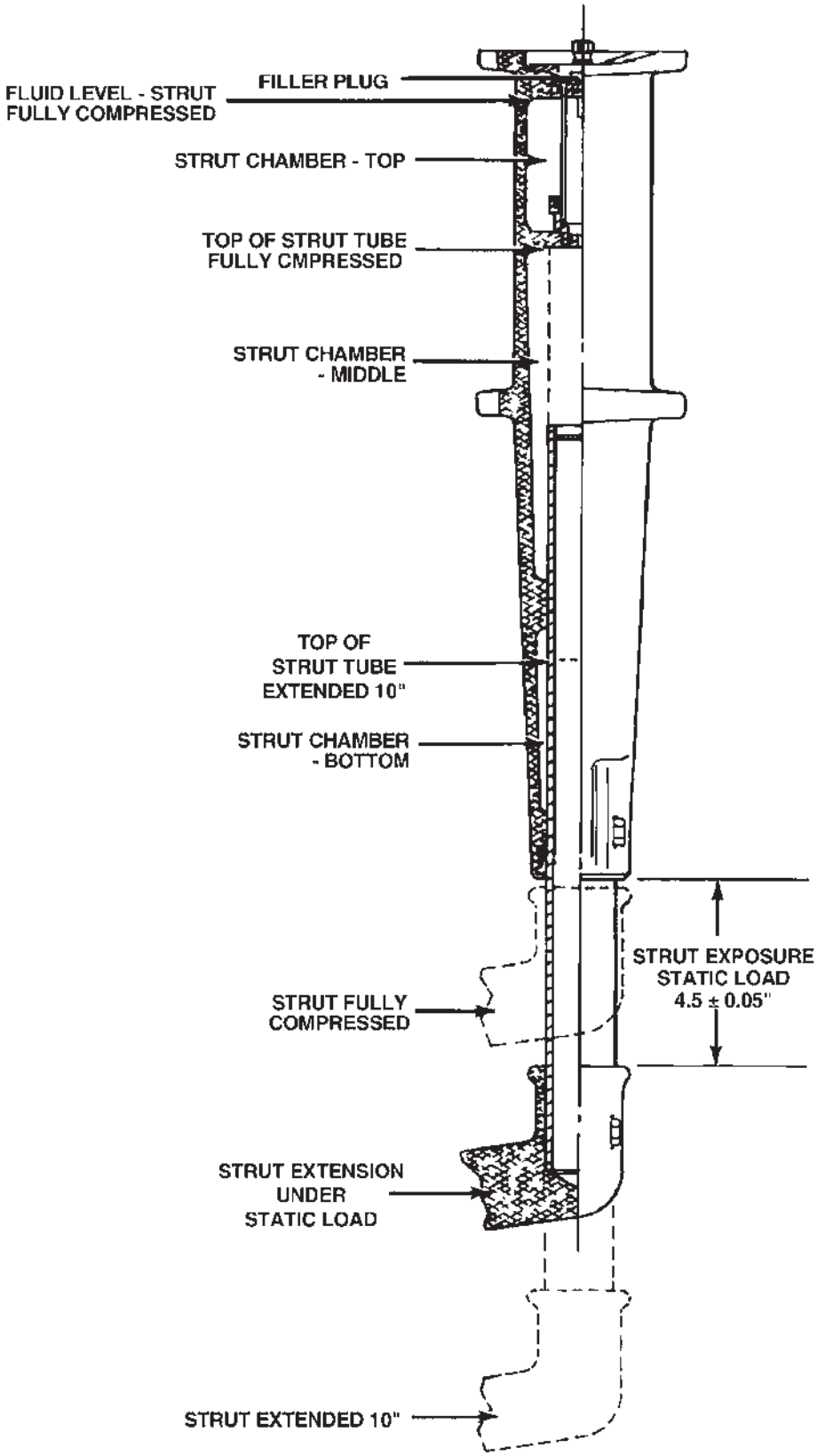
c) Pour fluid from a clean container through filler opening until the fluid level reaches bottom of filler plug hole. If housing was completely or nearly emptied, allow time for fluid to drain from the upper chamber into the middle chamber.

d) Lower strut until fully extended (at least 10 and no more than 12 inches of piston tube exposed). Fully compress and extend the strut two or three times to remove any air that may be trapped in housing.

e) Raise strut to full compression and fill with fluid to bottom of filler plug, if needed.

f) Apply an appropriate thread lubricant to the threads of filler plug and install filler plug in the top of strut housing. Torque plug to 45 foot-pounds.

(e) With airplane raised, compress and extend the gear strut several times. Ensure strut actuates freely. The weight of the gear fork and wheel should extend strut.



Main Gear Oleo Strut (cut-away view)
Figure 3

(f) Clean off overflow of fluid, and inflate strut as described in Inflating Oleo Struts, below.

(g) Check that fluid is not leaking from around strut piston at bottom of housing.

(3) Filling Main Gear Oleo Strut (See Figure 3.)

Fill partly full or completely emptied main gear oleo strut with MIL-PRF-5606 fluid as follows:

(a) Raise the airplane on jacks until the main gear torque link assembly approaches full travel.

(b) Place a pan under the gear to catch spillage.

(c) Remove cap on top wing to gain access to top of strut housing. Release air from strut housing chamber by removing cap from air valve and depressing valve core.

(d) Remove any one of the three torque link bolts and raise airplane until at least 10 and no more than 12 inches of piston tube is exposed.

NOTE: With torque link bolt removed, piston tube is free to slide from strut housing.

(e) Fill the main gear housing by one of the following methods:

1) Method I:

a) Remove valve core from filler plug at the top of strut housing. Do not remove plug.

b) Attach one end of a clear plastic hose to the valve stem of the filler plug and submerge the other end in a container of hydraulic fluid. Ascertain that the end of the hose on the valve stem is tight and the fluid container is approximately equal in height to the top of the strut housing.

c) Fully compress and extend strut (at least 10 and no more than 12 inches of piston tube exposed) to draw fluid from the fluid container and expel air from strut chamber. Watch the fluid pass through the plastic hose. Stop when bubbles cease to flow through the hose - i.e. - when the strut is full and no air is present in the chamber.

d) When air bubbles cease to flow through hose, compress strut fully and remove hose from the valve stem.

e) With strut compressed, remove filler plug and check that the fluid level is visible up to the bottom of filler plug hole.

f) Install core in filler plug. Apply an appropriate thread lubricant to threads of filler plug and install plug in top of strut housing. Torque plug to 45 foot-pounds.

2) Method II:

a) Remove filler plug from top of strut housing.

b) Raise strut piston until fully compressed.

c) Pour fluid from a clean container through filler opening until the fluid level reaches bottom of filler plug hole. If housing was completely or nearly emptied, allow time for fluid to drain from the upper chamber into the middle chamber.

d) Lower strut until fully extended (at least 10 and no more than 12 inches of piston tube exposed). Fully compress and extend the strut two or three times to remove any air that may be trapped in housing.

e) Raise strut to full compression and fill with fluid to bottom of filler plug, if needed.

f) Apply an appropriate thread lubricant to the threads of filler plug and install filler plug in the top of strut housing. Torque plug to 45 foot-pounds.

(f) Replace torque link bolt. Tighten bolt only enough to allow no side play in the connection.

(g) With airplane raised, compress and extend the strut several times to ascertain that the strut actuates freely. The weight of gear fork and wheel should extend strut.

(h) Clean off overflow of fluid and inflate strut as described in Inflating Oleo Struts, below.

(i) Check that fluid is not leaking around the strut piston at the bottom of the housing.

(4) Inflating Oleo Struts

Make certain that each oleo strut has sufficient fluid and the torque links are properly connected. Inflate oleo struts to:

(a) Proper visible piston extension (see above), or

(b) To pressures of 150 ± 15 psi.

When using the visible extension method, the aircraft should be fully serviced with fuel and engine oil and resting on its landing gear. Inflate strut until correct inches of piston is exposed. Rock aircraft several times to ascertain that gear settles back to the correct strut position. If a strut pump is not available, raise aircraft and use line pressure from a high pressure air system. Lower aircraft and, while rocking it, bring strut down to proper extension by releasing air from valve.

When using pressure method pistons must be fully extended by raising aircraft off ground. (Refer to 7-10-00.)

Check for valve core leakage before capping valve.

B. Brake System

The brake system incorporates a hydraulic fluid reservoir through which the brake system is periodically serviced. Fluid is drawn from the reservoir by the brake cylinders to maintain the volume of fluid required for maximum braking efficiency. Spongy brake pedal action is often an indication that the brake fluid reservoir is running low on fluid. Instructions for filling the reservoir are given in Filling Brake Cylinder Reservoir. When found necessary to accomplish repairs to any of the brake system components, or to bleed the system, these instructions may be found in 32-40-00.

(1) Filling Brake Cylinder Reservoir

The brake cylinder reservoir is located on the left side of the firewall in the engine compartment. It should be checked at every 50 hour inspection and replenished as necessary. Fill with MIL-PRF-5606 fluid to level marked on reservoir. No brake adjustment is necessary, though they should be checked periodically per instructions given in 32-40-00.

(2) Draining Brake System

(a) Connect a hose to bleeder fitting on the bottom of the cylinder.

(b) Place other end of hose in a suitable container.

(c) Open bleeder fitting and slowly pump hand brake lever and appropriate brake pedal until fluid ceases to flow.

(d) Clean brake system by flushing with denatured alcohol.

C. Tires

Discounting tire growth after initial inflation, once the tire has been inflated, the maximum permissible pressure drop is 5% in any 24 hour period. When the loss rate exceeds 5% in 24 hours, check tires for damage.

Some aircraft tires are designed to permit air or nitrogen that is trapped in the cords or that diffuses through the liner or tube to escape through special sidewall vents. This venting prevents pressure build-up within the cord body which might cause tread, sidewall or ply separation. Tires requiring lower sidewall vents will have either a green or white paint dot applied to the area of each vent hole. Tires not needing lower sidewall vents will have no color dot in this zone.

When water or a soap solution is brushed over the outside of a vented tire, bubbles form. Some vents may emit a continuous stream of bubbles. Others may produce intermittent bubbles. And some may not bubble at all. This variety is normal and does not mean that there is anything wrong with the tire. In fact, as long as a vented tire is inflated, there will be some diffusion from the vents. Vents should remain open, so check periodically to make sure they have not been covered over or closed by tire paint or spilled solvent. And since vents may be covered during retreading, check for evidence that your retreads have been revented.

However, as stated above, any drop in pressure exceeding 5% in any 24 hour period is not acceptable.

(1) Several basic characteristics of aircraft tires may be mistaken for problems:
   (a) Tire growth in the first 12 to 24 hours after inflation will result in a seemingly severe pressure drop. Simply inflate, wait for another 24 hours, then check pressure. It will probably be within specs.
   (b) Make sure that initial inflation is to recommended operating pressure to ensure full tire growth.
   (c) It is normal for tires to show a small amount of pressure leakage throughout the life of the tires.

(2) Maintain tires at pressure specified in 6-00-00, Chart 1. When checking tire pressure, examine tires for wear, cuts, bruises, and slippage. Apply Age-Master #1 to tires to protect against ozone attack and weathering as follows:
   (a) Clean oil and grease from all tire surfaces.
   (b) Apply single heavy coat using brush at 0.4 - 0.5 fluid ounces per square foot. Cover surface completely and evenly; allow to dry for 5 - 10 minutes.
   (c) Apply second coat per step (b); allow to dry for 20 - 30 minutes before handling.
   (d) Remove agent on wheel assembly with cleaning solvent.
   (e) Re-apply as conditions dictate.

5. Battery

CAUTION: DO NOT ALLOW SODA SOLUTION TO ENTER BATTERY.

The battery is located in the aft fuselage, aft of the rear baggage compartment. Check battery for proper fluid level. Do not fill battery above the baffle plates. Do not fill the battery with acid - use water only. A hydrometer check will determine the percent of charge in the battery.

A. Battery Corrosion Prevention

The battery must be checked for corrosion each 50 hour inspection or every 30 days, whichever comes first. Should corrosion be found on or around the battery, remove the battery and clean it and the surrounding area in accordance with the instructions in Chapter 24.

B. Check and service acid recovery (vent) jar and related hoses every 50 hours or following extensive battery use, such as a hard start requiring repeated use of starter. Refer to 24-30-00.

C. Check electrolyte level. Add distilled water as necessary.

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SCHEDULED SERVICING

Routine cleaning and lubrication of the airplane and its component parts will significantly extend its service life and reduce the frequency of repairs.

1. Fuel Strainer (i.e. - Gascolator)

   At intervals of 50 hours or 90 days, whichever comes first, clean the fuel strainer (i.e. - gascolator). The fuel strainer (Figure 2, 12-10-00) is mounted on the forward left side of firewall.

2. Induction Air Filter

   Check induction air filter each 50 hour maintenance inspection. Clean or replace if found to be dirty. Replace the filter after one year, ten cleanings or 500 flight hours, whichever comes first.

   A. Removal

      (1) Remove lower engine cowling.
      (2) Remove 6 screws that retain filter cover. Remove cover.
      (3) Remove the 2 wing nuts and washers securing the retaining plate located on lower left engine section.
      (4) Remove air filter.

   B. Installation

      After cleaning or replacing the filter, install the filter as follows:

      (1) Position air filter on engine.
      (2) Secure air filter using retaining and washers with wingnuts
      (3) Install cover and secure with 6 retaining screws.
      (4) Install lower engine cowling.

   C. Cleaning

      Air filter must be removed and cleaned daily when operating in dusty conditions. If any holes or tears are noticed, air filter must be replaced immediately (see PA-28-181 Parts Catalog, P/N 761-898). Clean filter element by shaking off loose dirt by rapping filter element on a hard flat surface.

      Clean air filter housing by wiping with a clean cloth soaked in a suitable quick drying type solvent. Install and seal filter element when housing dries.

3. Carburetor Heat / Alternate Air Door

   The carburetor heat / alternate door is located in the air induction box to provide a source of air to the engine should there be an air stoppage through the filter system. In airplanes equipped with the O-360-A4M engine, it additionally provides heated air to the carburetor. The following should be checked during inspection:

   A. Check that air door seals are tight and that the hinge is secure.

   B. Check that when the cockpit control is in the closed position the door is properly seated in the closed position.

      NOTE: In airplanes equipped with a Fuel-Injected engine only, ensure that the control lever has 0.160 inch clearance from the instrument panel when the door and the lever are both in the closed position and again when both are in the fully open position.

   C. Actuate the door by operating the control lever in the cockpit to determine that it is not sticking or binding.

   D. Check the cockpit control cable for free travel.

4. Propeller

Inspect spinner, back plate and propeller surfaces for nicks, scratches, corrosion and cracks. Remove minor nicks and scratches per instructions in 61-10-00. Paint face of each blade with a flat paint to retard glare. Wipe surfaces with a light oil or wax to prevent corrosion.

5. Electrical System

Servicing the electrical system starts with the battery, see Battery, 12-10-00. The security of all electrical connections should be checked as well as the operation of all lights, general condition of the generator or alternator and starter. All electrical wires should be inspected for chafing and bare wires. For detailed information on this system, refer to Chapter 24 of this manual.

6. Tire Balance

Proper balancing is critical for the life of aircraft tires. If a new tire is balanced upon installation it will usually remain balanced for the life of the tire without having any shimmy or flat spots.

A. Tire Balancer

An inexpensive balancing fixture that will balance almost any light aircraft tire can be made from the materials shown in Figure 1.

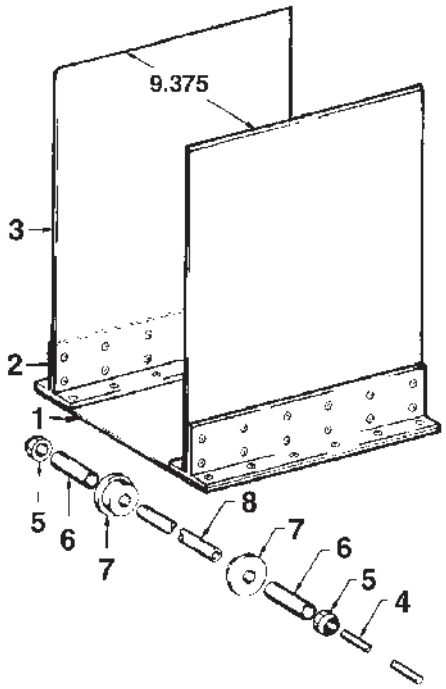
(1) Chamfer top edges of -3 sides, leaving 1/16 inch flat on top of the inboard edge. Rivet -2 tee's to -3 sides using AN 470-AD5 rivets, with 2 inch spacing, and using AN 426-AD5 rivets (2 inch center to center) to secure -2 tee's to -1 base. If tee extrusion is unavailable, heavy angle extrusion could be used. -3 sides must be vertical.

(2) The -4 axle must slide through the -8 pipe, the -5 nuts are made by reaming the existing threads in the AN 365-624 nuts with an R drill, then tapping them with a 1/8-27 pipe tap.

(3) The -6 spacers were made from 1/2 inch aluminum tubing, the two lengths of spacers are suitable for balancing most any aircraft wheel.

(4) The -7 bushings may be made from one inch phenolic or aluminum using a 1-1/2 inch hole saw to cut out the smaller bushing and a 1-3/4 hole saw to cut out the larger. By inserting a 1/4 inch long threaded bolt through the pilot hole and securing with a washer and nut, a drill press and file may be used to make the off-set on the bushing. The turned-down part should just slide inside the bearing race and then ream the pilot hole to slide over the -8 pipe threads.

(5) The -8 pipe was made from a piece of 1/8 inch black pipe and threaded with a 1/8-27 pipe die, this will be thread 3 inches in from each end of the pipe.



USE THE FOLLOWING LIST OF MATERIALS TO MAKE THE BALANCER

- 1. 1 EA BASE	12 X 11	0.190 2024 T3 CLAD ALUMINUM ALLOY
- 2. 2 EA TEE	2.5 X 2 X 11	0.190 2024 T4 EXTRUDED ALUMINUM ALLOY
- 3. 2 EA SIDES	14 X 11	0128 2024 T3 CLAD ALUMINUM ALLOY
- 4. 2 EA AXLE	0.125 X 10.25	4130 STEEL, NOMALIZED
- 5. 2 EA NUTS	AN 365-624	
- 6. 2 EA SPACER	0.50 X 2.25	5052-0 ALUMINUM TUBING
2 EA SPACER	0.50 X 1.25	5052-0 ALUMINUM TUBING
- 7. 2 EA BUSHING	1.480 X 1.625 X 1.00	PHENOLIC OR ALUMINUM
2 EA BUSHING	2.240 X 1.37 X 1.00	PHENOLIC OR ALUMINUM
- 8. 1 EA PIPE	1/8 X 9.3	BLACK STEEL PIPE
* 2 EA BEARINGS	SAVE TWO OF EACH SIZE WORN WHEEL BEARINGS FROM PREVIOUS INSPECTIONS.	

Tire Balancer
Figure 1

B. Procedure

Balance tires as follows:

(1) Mount the tire and tube (if one is used) on the wheels, but do not install the securing bolts. Install the wheel bearings in the wheels; then, using the -7 bushings, -6 spacers, and -5 nuts, install the wheel/tire assembly on the -8 pipe. Secure the -5 nuts finger tight so that the wheel halves touch each other. Be sure the bolt holes are aligned. Insert the -4 axle through the -8 pipe and place the wheel in the center of the balancer. Make sure the axle is only on the chamfered edges of the balancer and that it is at 90° to the sides of the balancer.

(2) Release the tire. If it is out of balance it will rotate, coming to rest with the heaviest point on the bottom. Tape a one-half (½) ounce patch across the top center of the tire. Rotate the tire 45° and release it again. If the tire returns to the same position, add a one (1) ounce patch and again rotate the tire and release it. Continue this procedure until the tire is balanced.

(3) When balance is attained, put a chalk mark on the sidewall directly below the patch. Use one mark for each half ounce of weight needed. Mark the valve stem location on the tire and the opposite wheel half to assure reassembly in the same position. Remove the wheel from the balance stand, break it down and clean the inside of the tire with toluol. Apply a coat of patch cement to both the patch and the inside center of the tire in line with the chalk marks. When the cement has dried, install the patches making certain they are on the centerline of the tire and aligned with the chalk marks on the sidewall. Burnish the patches to remove trapped air, etc.

(4) When reassembling the wheel, powder the inside of the tire. Mount the tire on the valve side of the wheel in the same position it was in when it was balanced. Install the other wheel half, aligning the chalk marks. Install the bolts and tighten to required torque, then inflate the tire to the pressure specified in 6-00-00, Chart 1, and recheck the balance. The wheel should not be more than one ounce out of balance.

7. Winterization

See “Figure 2” on page 12205.

A. Description

(1) In airplanes equipped with the O-360-A4M engine.

A winterization plate is standard and should be used to reduce air flow through the engine oil cooler whenever the ambient temperature is 50° F or less. When not in use the winterization plate should be stored in the aircraft cockpit pilot side panel.

(2) In airplanes equipped with the IO-360-B4A engine.

Winterization plate(s) is(are) standard and should be used to reduce air flow through the engine oil cooler(s) whenever the ambient temperature is at a specified value (see below) or less; or, whenever the oil temperature cannot be maintained at or above 140° F during continuous operation. When not in use the winterization plate(s) is(are) stored in the aircraft cockpit pilot side panel.

(a) S/N’s 2843821, 2843852–2843937 had a 15° F ambient temperature winterization plate as standard equipment. While this is acceptable for continued use, greater flexibility is availble by installation of the 50° F ambient temperature winterization plate per Piper Service Bulletin No. 1324.

(b) S/N’s 2843938–2843999, and 2881001 and up, have a 50° F ambient temperature winterization plate for the standard oil cooler as standard equipment. A winterization plate is included with the optional (lower) oil cooler.

B. Standard (Upper) Oil Cooler Winterization Plate

(1) Installation

(a) Remove the upper engine cowling per Removal in 71-10-00.

(b) Remove the nuts and washers from the upper inboard and the lower outboard oil cooler mounting bolts.

(c) Install the winterization plate on the forward side of the oil cooler with Slots A over the upper inboard and the lower outboard oil cooler mounting bolts.

(d) Secure the winterization plate with the washers and nuts.

(e) Install the upper engine cowling per Installation in 71-10-00.

(2) Removal

(a) Remove the upper engine cowling per Removal in 71-10-00.

(b) Remove the nuts and washers from the upper inboard and the lower outboard oil cooler mounting bolts – i.e., Slots A of the winterization plate.

(c) Remove the winterization plate.

(d) Secure the oil cooler with the washers and nuts.

(e) Install the upper engine cowling per Installation in 71-10-00.

C. Optional (Lower) Oil Cooler Winterization Plate

(1) Installation

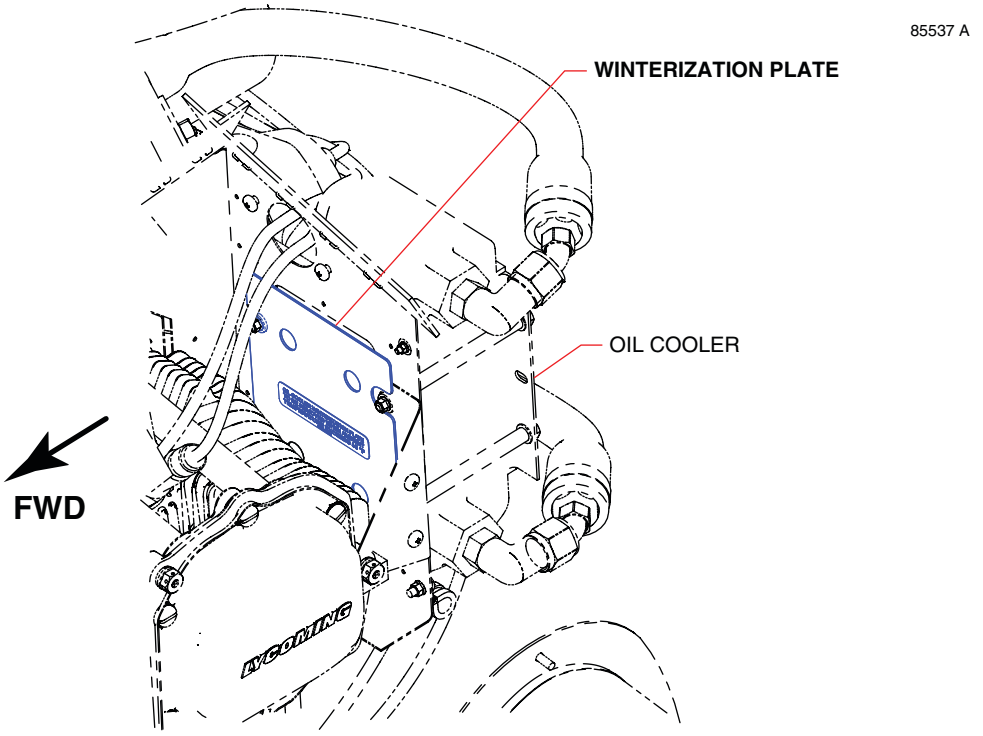
(a) Remove the upper engine cowling per Removal in 71-10-00.

(b) Loosen the four bolts securing the air duct to the top of the oil cooler.

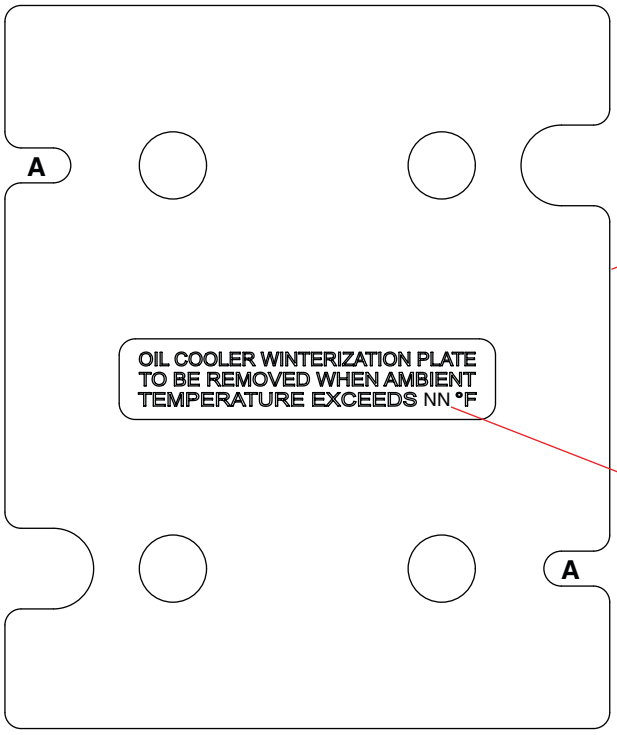
(c) From the forward side, slide the winterization plate between the oil cooler and the air duct.

(d) Secure the installation by tightening the four bolts securing the air duct to the top of the oil cooler.

(e) Install the upper engine cowling per Installation in 71-10-00.



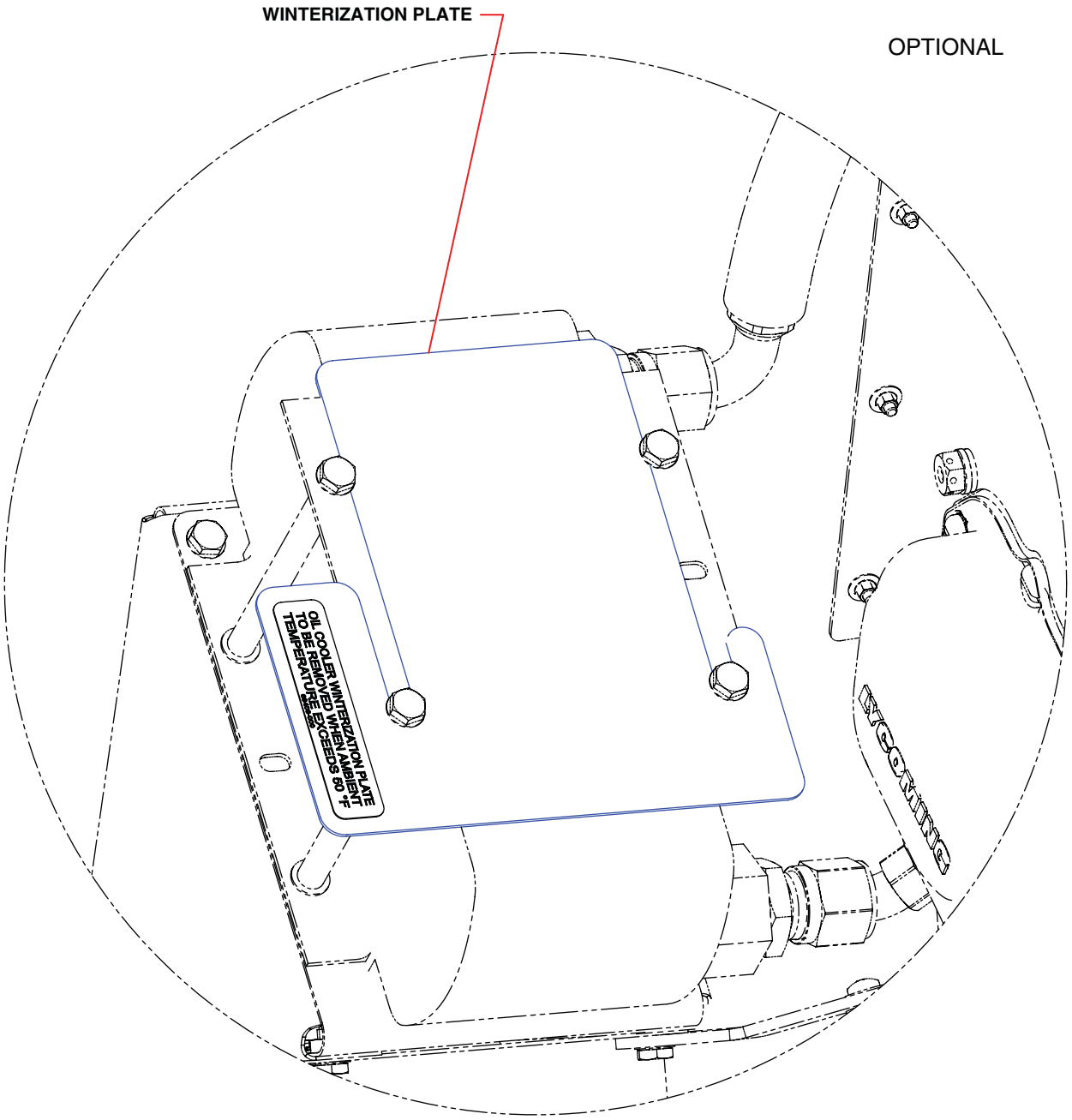
ENGINE BAFFLES REMOVED FOR CLARITY



STANDARD

INSTALL FORWARD OF ENGINE OIL COOLER USING EXISTING HARDWARE AT SLOTS “A”.

Winterization Plate Installations
Figure 2 (Sheet 1 of 2)



LOOKING DOWN AND AFT AT THE LOWER OIL COOLER
(AIR DUCT REMOVED FOR CLARITY)

Winterization Plate Installations
Figure 2 (Sheet 2 of 2)

(2) Removal

(a) Remove the upper engine cowling per Removal in 71-10-00.

(b) Loosen the four bolts securing the air duct to the top of the oil cooler.

(c) Slide the winterization plate forward and remove it.

(d) Secure the air duct to the oil cooler by tightening the four bolts .

(e) Install the upper engine cowling per Installation in 71-10-00.

8. Lubrication

Proper lubrication procedures are valuable both as a means of prolonging the service life of the airplane and reducing the frequency of repairs. The periodic application of recommended lubricants to their relevant bearing surfaces, together with cleanliness, ensures the maximum efficiency and utmost service life of all moving parts. Instructions regarding the locations, time intervals, and types of lubricant used are found in the Lubrication Charts. See also 91-10-00, Chart 10, Consumable Materials.

NOTE: If the airplane is inactive for long periods of time, lubricate in accordance with Lubrication Charts at least every 90 days.

A. Precautions

To ensure the best possible results from the application of lubricants, observe the following precautions:

CAUTION: MIL-PRF-23827 AND MIL-PRF-81322, CONTAIN CHEMICALS WHICH MAY BE HARMFUL TO PAINTED SURFACES.

CAUTION: DRY LUBRICANT (I.E. - PTFE BASED MS-122) WILL ATTACK ANY ACRYLIC BASED PLASTIC (LUCITE), POLYCARBONATES (LEXAN), POLYSTYRENE AND ITS COPOLYMERS (ABS), AND CELLULOSE ACETATE.

CAUTION: AFTER THOROUGHLY WASHING AIRPLANE, ENSURE LANDING GEAR, FLIGHT CONTROLS, FLAP TRACKS, STABILATOR TRIM SCREW, AND ENGINE COMPARTMENT ARE STILL PROPERLY LUBRICATED.

(1) Use recommended lubricants. Where general purpose lubricating oil is specified, but unavailable, clean engine oil may be used.

(2) Check components to be lubricated for evidence of excessive wear and replace as required.

(3) Remove excess lubricant from components to prevent collecting dirt and sand in quantities capable of causing excess wear or damage to bearing surfaces.

B. Application of Grease

Before using a grease gun, ensure that gun is filled with new, clean grease of the grade specified for the particular application.

(1) Where a reservoir is not provided around a bearing, apply the lubricant sparingly and wipe off any excess.

(2) Remove wheel bearings from the wheel hub and clean thoroughly with a suitable solvent. When repacking with grease, be sure the lubricant enters the space between the rollers in the retainer ring. Do not pack the grease into the wheel hub.

C. Application of Oil

Whenever specific lubrication instructions for individual components are not available, observe the following precautions:

(1) Apply oil sparingly, never more than enough to coat the bearing surfaces.

(2) Do not oil control cables.

D. Lubrication Charts

(PIR-66539. Rev. J, PIR-PPS65102-134, Rev. NEW.)

The lubrication charts consist of individual illustrations for the various aircraft systems. Each component to be lubricated is indicated by a number, the type of lubricant and the frequency of application. Special instructions are listed in Chart 2 before the lubrication charts.

NOTE: When the average ambient air temperature is approximately at the dividing line, use the lighter oil.

While the specified lubricant should be used if available, lubricants listed in the Lubrication Charts should all be considered “or equivalent.” Lubricant specifications become problematic over time. Where a specific product is called out, that manufacturer may go out of business, may be sold, or renamed. The named product may subsequently be no longer available, or renamed itself. Many lubricant military specifications have been superceded over the last several years. Accordingly, a cross-reference chart (“Chart 1”) is provided for lubricants where specification or product changes have been identified.

CHART 1 (Sheet 1 of 2)
LUBRICANT SPECIFICATION CROSS-REFERENCE

Old Spec / Product	superceded by	**New Spec / Product**	**Product Type**
Lubriplate 907		Lubriplate Aero	Grease, Lithium Base, Multipurpose.
MIL-A-907		MIL-PRF-907 (aka Kopr-Kote *)	Anti-Seize Thread Compound, High Temp. (up to 566 Degrees C) (up to 1050 Degrees F)
MIL-A-5092		MMM-A-1617	Adhesive, (Rubber Base) General Purpose.
MIL-C-16173		MIL-PRF-16173	Corrosion Preventative Compound, Solvent Cutback, Cold Application.
MIL-DTL-27686		See MIL-I-27686 below.	
MIL-G-3278		MIL-PRF-23827	Grease, Aircraft & Instrument, Gear and Actuator Screw.
MIL-G-3545		MIL-PRF-81322	Grease, Aircraft, General Purpose, Wide Temp. (-54 to 177 Degrees C) (-65 to 350 Degrees F).
MIL-G-6032		SAE-AMS-G-6032	Grease, Plug Valve, Gasoline & Oil Resistant.
MIL-G-7711		MIL-PRF-81322	See MIL-PRF-81322 above.
MIL-G-18709		DOD-G-24508	Grease, High Perfomance, Multipurpose.
MIL-G-21164		MIL-G-21164	Grease, Molybdenum Disulfide, for Low & High Temperatures.
MIL-G-23827		MIL-PRF-23827	See MIL-PRF-23827 above.
MIL-G-81322		MIL-PRF-81322	See MIL-PRF-81322 above.

CHART 1 (Sheet 2 of 2)
LUBRICANT SPECIFICATION CROSS-REFERENCE

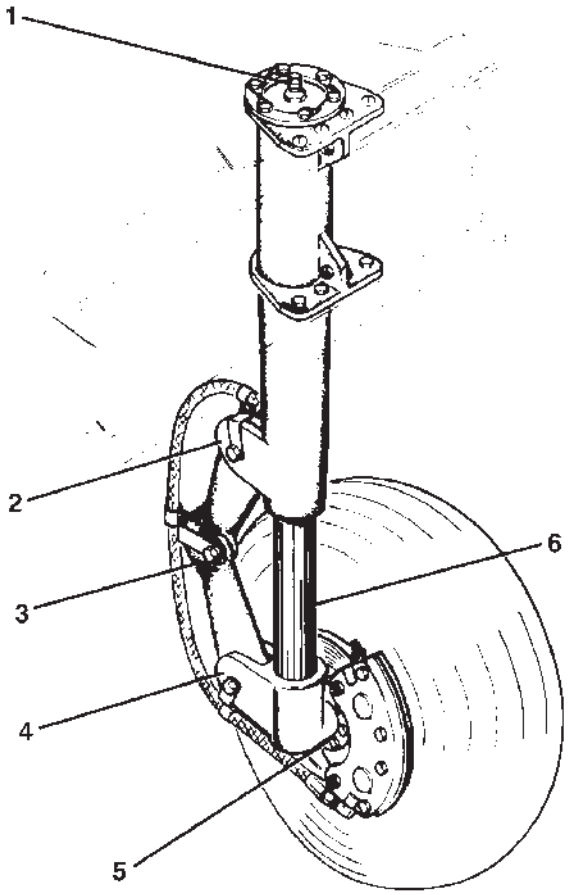
Old Spec / Product	superceded by	New Spec / Product	Product Type
MIL-H-5606		MIL-PRF-5606	Hydraulic Fluid, Petroleum Base, Aircraft, Missile, and Ordnance.
MIL-I-27686		MIL-DTL-85470	Inhibitor, Icing, Fuel System, High Flash, NATO Cold Number S-1745.
MIL-L-6082		SAE J 1966	Oil, Lubricating, Aircraft Piston Engine (Non-Dispersant Mineral Oil)
MIL-L-7870		MIL-PRF-7870	Oil, Lubricating, Low Temp.
MIL-L-22851		SAE J 1899	Oil, Lubricating, Aircraft Piston Engine (Ashless Dispersant).
MIL-L-25567		MIL-PRF-25567	Leak Detection Compound, Oxygen Systems.
MIL-L-60326		MS-122AD *	Dry-Lubricant.
MIL-M-7866		SAE-AMS-M-7866	Molybdenum Disulfide, Technical, Lubrication Grade.
MIL-S-11031B		A-A-59293	Adhesive (Curing), Sealing Compound (Polysulfide Base).
MIL-S-22473		ASTM-D-5363	Adhesive, Anaerobic Single-Component.
MIL-S-8660		SAE-AS-8660	Silicone Compound, Nato S-736, (-54 to 204 Degrees C) (-65 to 400 Degrees F)
MIL-T-5544		SAE-AMS-2518	Thread Compound, Anti-Seize, Graphite-Petrolatum
MIL-T-27730		A-A-58092	Tape, Anti-Seize, Polytetrafluoroethylene
MS-122 *		MS-122AD *	Dry-Lubricant
MS-122-6075 *		MS-122AD *	Dry-Lubricant
Parker O-Ring Lube *		Parker O-LUBE *	O-Ring Lubricant
Parker 6PB * or 6PB Parker *		MIL-PRF-907 (aka Kopr-Kote *)	See MIL-PRF-907 above.
TT-A-580 (JAN-A-669)		TT-A-580 (TT-S-1732)	Sealing Compound, Pipe Joint and Thread, Lead Free, General Purpose.
* Product Nomenclature			

CHART 2
SPECIAL INSTRUCTIONS

1. BEARINGS AND BUSHINGS - Clean exterior with a quick-drying solvent before lubricating.
2. LUBRICATION POINTS - Wipe all lubrication points clean of old grease, oil, dirt, etc., before lubricating.
3. WHEEL BEARINGS - For nose and main wheels, see bearing installation procedure in 32-40-00, under Main Wheel Assembly, Assembly and Installation, for grease requirements and packing instructions. Wheel bearings also require cleaning and repacking after exposure to any abnormal quantity of water.
4. OLEO STRUTS AND BRAKE RESERVOIR - Fill per instructions on unit or container, or refer to applicable chapter in this manual.
5. DOOR SEALS - Apply release agent/dry lubricant to door seals at least once a month to improve sealing characteristics and to prevent the seal from sticking.
6. CONTROL CABLES - Do not oil control cables. Grease control cables where they pass over a pulley or through a fairlead.
7. AIR FILTER - To clean filter, tap gently to remove dirt particles or wash in warm water and mild detergent and dry. Do not blow out with compressed air. Do not use oil. Replace filter if damaged.
8. OIL AND FILTER - Change the oil and filter every 50 hours or four months, whichever comes first. See latest revision of Lycoming's Service Bulletin No. 480, "Oil Change".
9. See the latest revision of Lycoming Service Instruction No. 1014 for lubricating oil recommendations.
10. O-RING, CONTROL WHEEL SHAFT BUSHING - Disassemble the retainer plates and lubricate the O-ring around the control wheel shaft bushing as required. Ensure thick retainer plate is reassembled with slot up.
11. NOT USED.
12. AIR CONDITIONING CONDENSER DOOR ACTUATING TRANSMISSION - Transmission to be 1/2 full of grease. Apply grease during assembly and lubricate transmission ball nut and screw with MIL-PRF-23827 grease.
13. AILERON HINGES - Do not lubricate aileron hinges with teflon sleeves. Aileron hinges without teflon sleeves should be first cleaned with a quick-drying solvent and then lubricated.

CHART 3
LUBRICATION - MAIN LANDING GEAR

	COMPONENT	LUBRICANT	FREQUENCY
1.	OLEO STRUT FILLER POINT (See Spec. Instr. 4 & 6)	MIL-PRF-5606	AS REQUIRED
2.	TORQUE LINK BEARING, UPPER (See Spec. Instr. 1)	MIL-PRF-23827	100 HRS
3.	TORQUE LINK CONNECTOR BUSHING (See Spec. Instr. 1)	MIL-PRF-23827	100 HRS
4.	TORQUE LINK BEARING, LOWER (Spec. Instr. 1)	MIL-PRF-23827	100 HRS
5.	MAIN GEAR WHEEL BEARINGS (See Spec. Instr. 3)	MOBIL GREASE 28 MIL-PRF-81322E	100 HRS
6.	EXPOSED MAIN OLEO STRUT (See Spec. Instr. 2)	RELEASE AGENT/ DRY LUBRICANT MS-122AD	100 HRS
7.	BRAKE RESERVOIR (See Spec. Instr. 4 & 6)	MIL-PRF-5606	AS REQUIRED



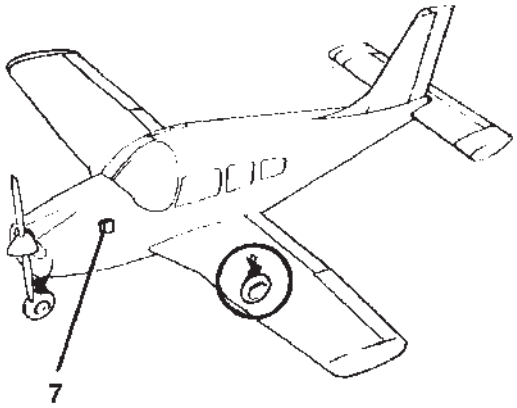


CHART 4
LUBRICATION - NOSE LANDING GEAR

	COMPONENT	LUBRICANT	FREQUENCY
1.	NOSE GEAR OLEO STRUT FILLER POINT (See Spec. Instr. 4)	MIL-PRF-5606	AS REQUIRED
2.	STEERING BELLCRANK PIVOT POINTS AND STEERING ROD ENDS (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS
3.	SHIMMY DAMPENER PIVOT POINT (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS
4.	TORQUE LINK ASSEMBLY (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS
5.	NOSE WHEEL BEARINGS (See Spec. Instr. 3)	MOBIL GREASE 28 MIL-PRF-81322E	100 HRS
6.	STEERING ROD (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS
7.	BUNGEE SEAL	PARKER O-RING LUBRICANT	100 HRS
8.	EXPOSED OLEO STRUT (See Spec. Instr. 2)	RELEASE AGENT/ DRY LUBRICANT MS-122AD	100 HRS

CAUTION: DO NOT USE HYDRAULIC FLUID WITH A CASTOR OIL OR ESTER BASE.

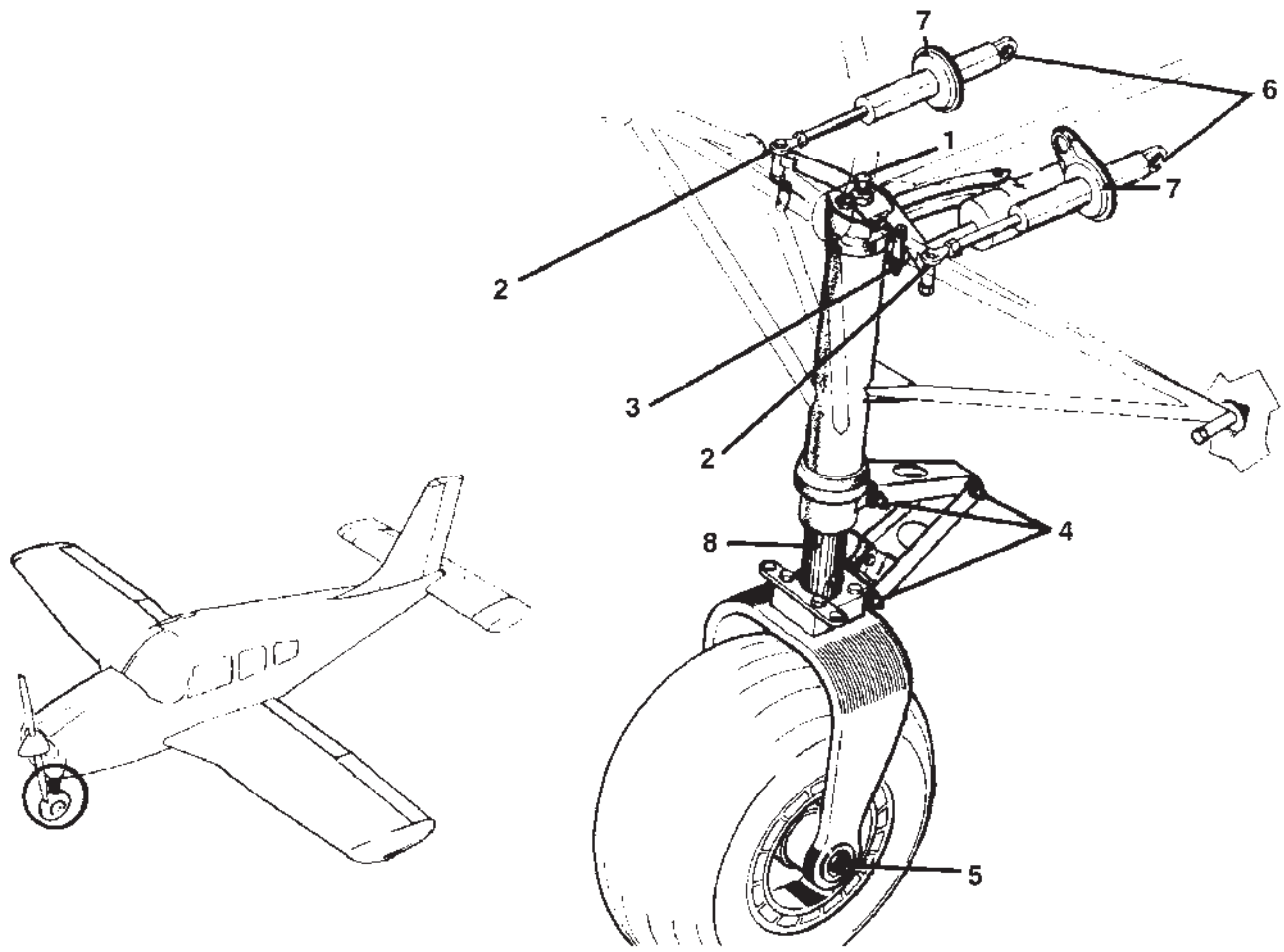
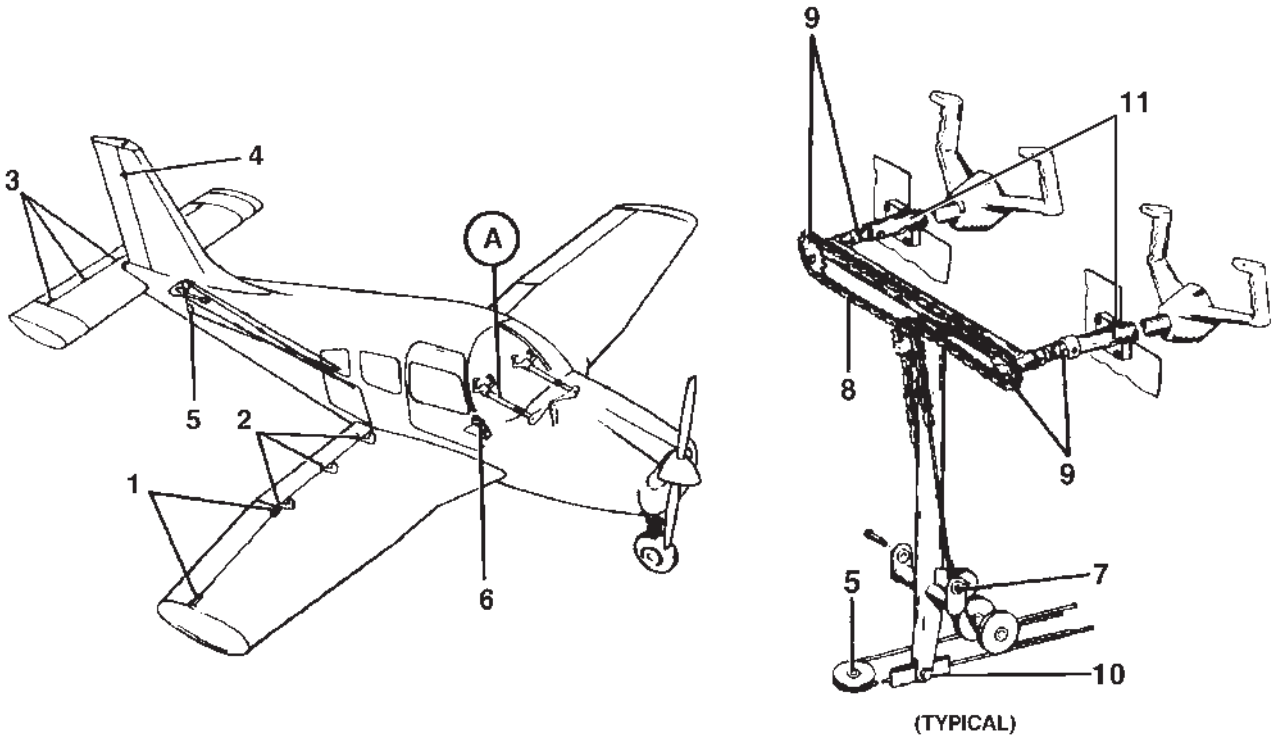


CHART 5 (Sheet 1 of 4)
LUBRICATION - CONTROL SYSTEM

	COMPONENT	LUBRICANT	FREQUENCY
1.	AILERON HINGE PINS (See Spec. Instr. 1 & 13)	MIL-PRF-7870C	100 HRS
2.	FLAP HINGE BEARINGS (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS
3.	STABILATOR HINGE PINs (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS
4.	RUDDER HINGE BEARINGS (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS
5.	CONTROL CABLE PULLEYS (See Spec. Instr. 6)	MIL-PRF-7870C	100 HRS
6.	STABILATOR TRIM CONTROL WHEEL (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS
7.	TEE BAR PIVOT POINT (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS
8.	CONTROL COLUMN CHAIN (See Spec. Instr. 2)	MIL-PRF-7870C	500 HRS
9.	CONTROL COLUMN FLEX JOINTS AND SPROCKET (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS
10.	STABILATOR CONTROL (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS
11.	O-RING, CONTROL WHEEL SHAFT BUSHING (See Spec. Instr. 10)	PARKER O-LUBE	AS REQUIRED
12.	AILERON, STABILATOR, AND RUDDER CONTROL CABLES, AND STABILATOR TRIM CABLES (See Spec. Instr. 2 & 6) (Not Shown)	AEROSHELL 33 / MIL-PRF-23827	100 HRS

CAUTION: DO NOT LUBRICATE CONTROL WHEEL SHAFT OR BUSHING. CLEAN ONLY, USING ALCOHOL OR OTHER SUITABLE SOLVENT.

CAUTION: DO NOT OVER-LUBRICATE COCKPIT CONTROLS.



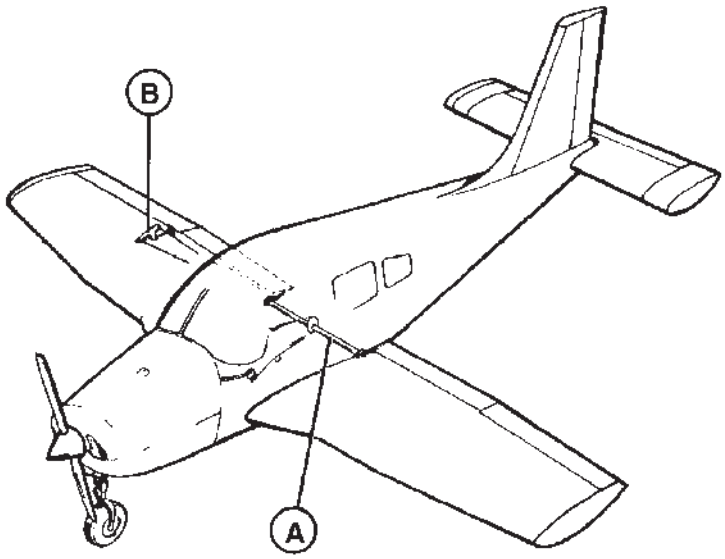
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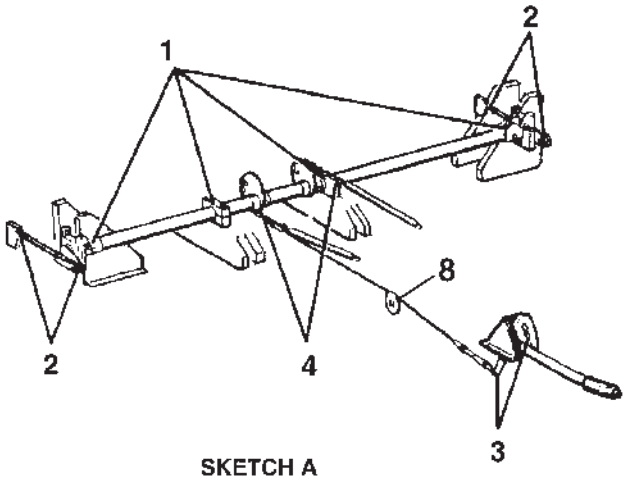
SKETCH A

CHART 5 (Sheet 2 of 4)
LUBRICATION - CONTROL SYSTEM

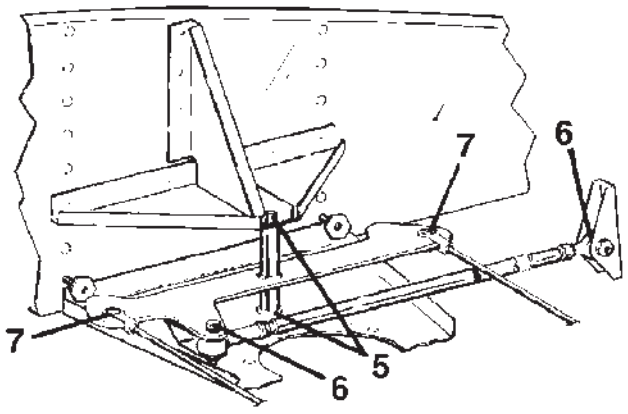
	COMPONENT	LUBRICANT	FREQUENCY
1.	FLAP TORQUE TUBE BEARING BLOCKS (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS
2.	FLAP CONTROL ROD END BEARINGS (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS
3.	FLAP HANDLE PIVOT POINT, LOCK MECHANISM, AND TURNBUCKLE END (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
4.	FLAP RETURN AND TENSION CHAINS (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS
5.	AILERON BELLCRANK PIVOT POINTS (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS
6.	AILERON CONTROL ROD END BEARINGS (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS
7.	AILERON BELLCRANK CABLE ENDS (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS
8.	FLAP CABLE (See Spec. Instr. 2 & 6)	AEROSHELL 33 MIL-PRF-23827	100 HRS

CAUTION: DO NOT OVER-LUBRICATE COCKPIT CONTROLS.





SKETCH A

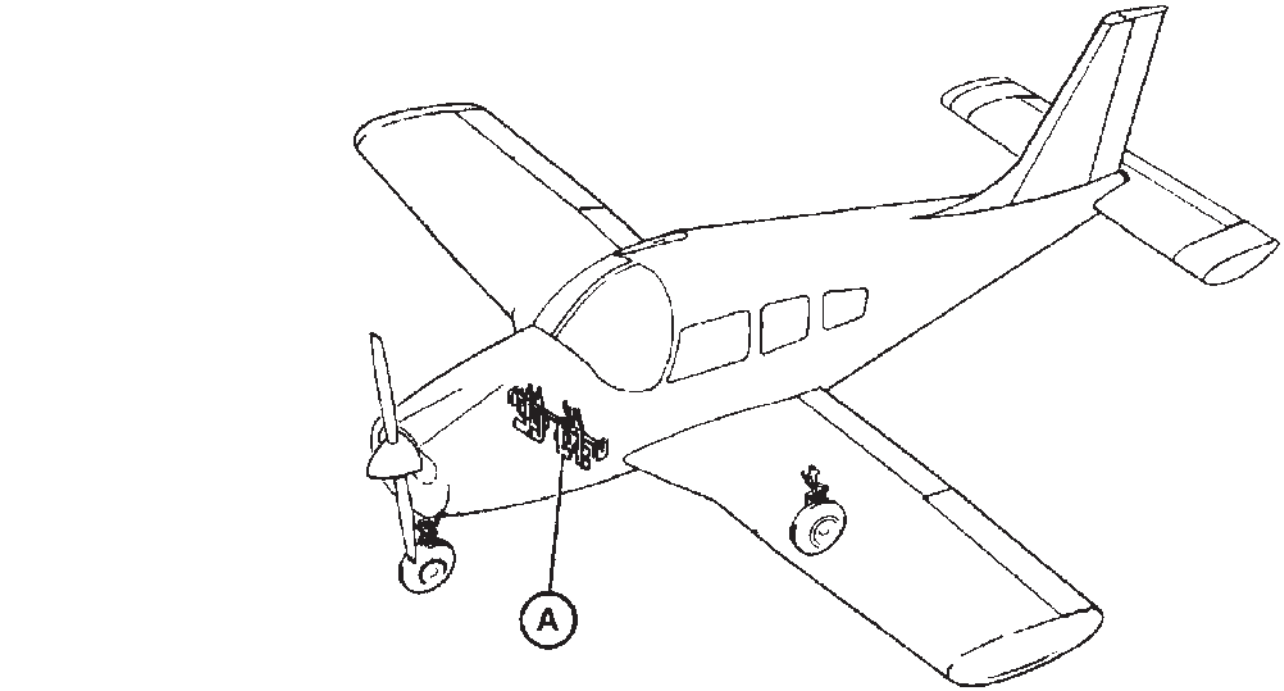


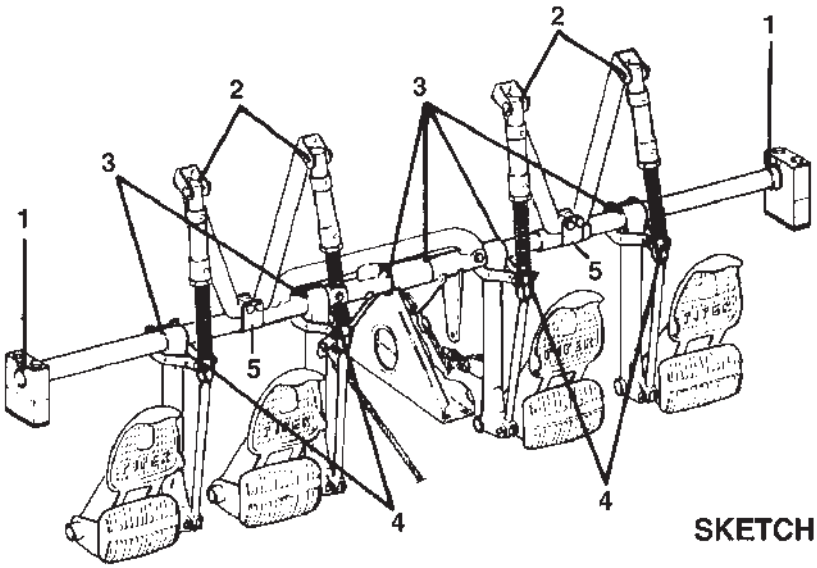
SKETCH B

CHART 5 (Sheet 3 of 4)
LUBRICATION - CONTROL SYSTEM

	COMPONENT	LUBRICANT	FREQUENCY
1.	RUDDER TUBE BEARING BLOCKS (See Spec. Instr. 2)	RELEASE AGENT/ DRY LUBRICANT MS-122AD	100 HRS
2.	TOE BRAKE CYLINDER ATTACHMENTS (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS
3.	RUDDER TUBE CONNECTIONS (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS
4.	BRAKE ROD ENDS (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS
5.	TOE BRAKE BRACE (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS

CAUTION: DO NOT OVER-LUBRICATE COCKPIT CONTROLS.



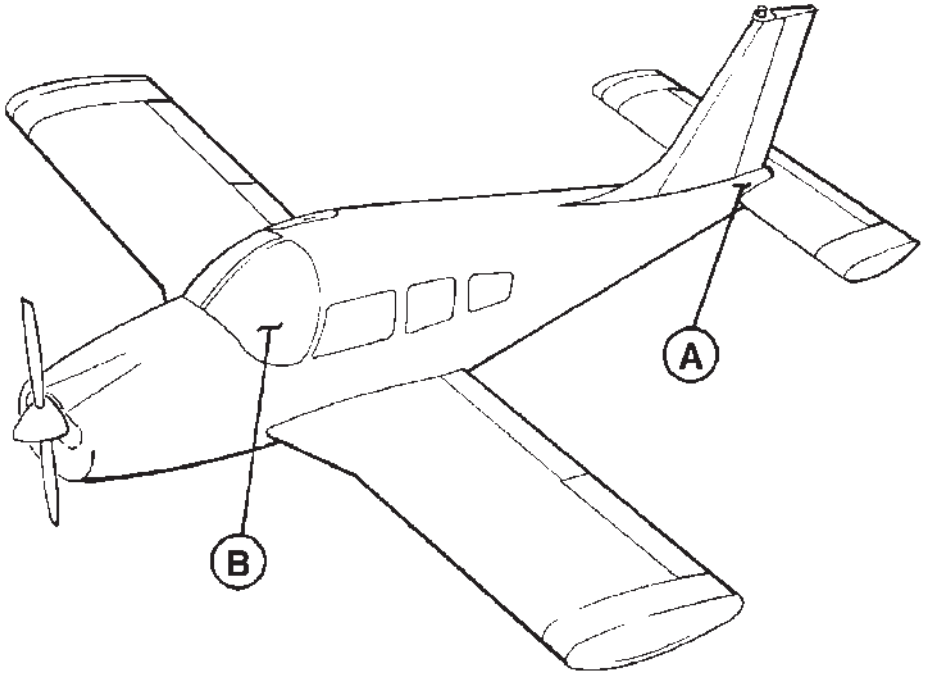


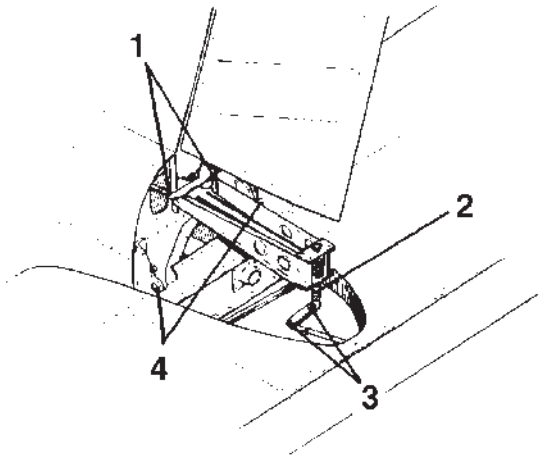
SKETCH A

CHART 5 (Sheet 4 of 4)
LUBRICATION - CONTROL SYSTEM

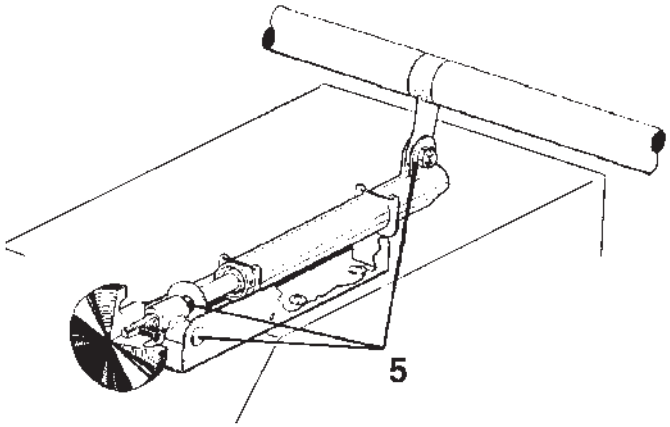
	COMPONENT	LUBRICANT	FREQUENCY
1.	RUDDER ARM CABLE ENDS (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS
2.	STABILATOR TRIM SCREW (See Spec. Instr. 2)	LUBRIPLATE AERO	500 HRS
3.	STABILATOR SCREW/TAB LINKS (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS
4.	STABILATOR HINGE POINTS (See Spec. Instr. 1 & 2)	MIL-PRF-7870C	100 HRS
5.	RUDDER TRIM ASSEMBLY (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS

CAUTION: DO NOT OVER-LUBRICATE COCKPIT CONTROLS





SKETCH A



SKETCH B

CHART 6
LUBRICATION - CABIN DOOR, BAGGAGE DOOR, AND SEATS

	COMPONENT	LUBRICANT	FREQUENCY
1.	DOOR HINGES (See Spec. Instr. 2)	MIL-PRF-7870C	100 HRS
2.	DOOR SEALS (See Spec. Instr. 5)	RELEASE AGENT/ DRY LUBRICANT MS-122AD	50 HRS
3.	DOOR LATCH MECHANISMS (See Spec. Instr. 2)	LUBRIPLATE AERO	500 HRS
4.	SEAT TRACK ROLLERS, STOP PINS AND REAR SEAT LEG RETAINER (CLIP AND CAM) (See Spec. Instr. 2)	LUBRIPLATE AERO	100 HRS

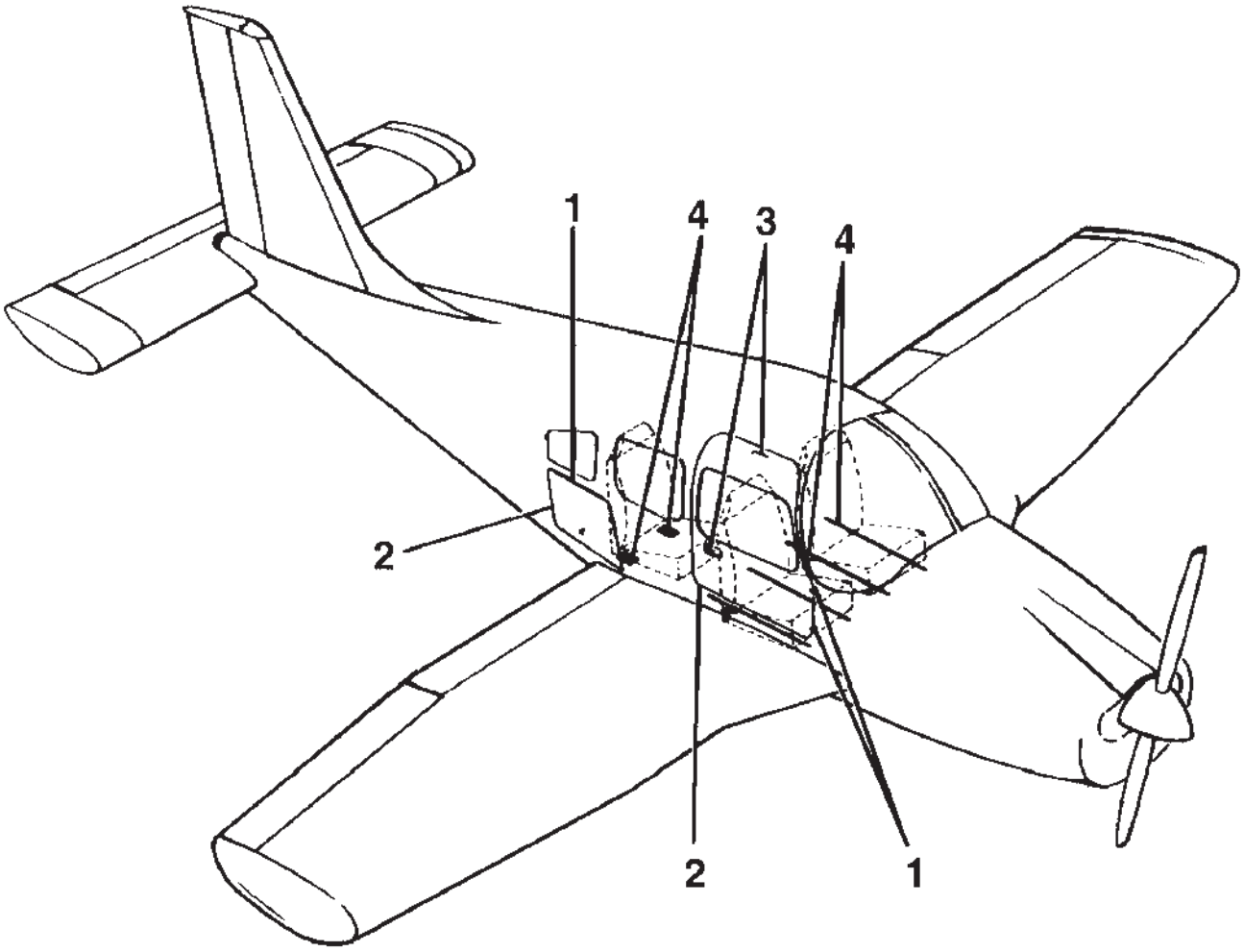


CHART 7
LUBRICATION - POWER PLANT AND PROPELLER

	COMPONENT			LUBRICANT	FREQUENCY
1.	ENGINE SUMP LUBRICATING OIL, RECIPROCATING ENGINE (PISTON)	AIR TEMPERATURE	SAE J1966 (Ashless Dispersant)	SAE J1899	
		ALL	SAE 20W-50	SAE 15W50, 20W50, 25W60	
	(See Spec. Instr. 8 and 9)	ABOVE 80°F (26.67°C)	SAE 60	SAE 60	50 HRS
		ABOVE 60°F (15.55°C)	SAE 50	SAE 40, 50	
		30° TO 90°F (-1.11° TO 32.22°C)	SAE 40	SAE 40	
		0° TO 70°F (-17.77° TO 21.11°C)	SAE 30	SAE 30, 40	
	(LYCOMING SI 1014N)	0° TO 90°F (-18° TO +32°C)	SAE 20W-50	SAE 20W-50, 15W-50	
		BELOW 10°F (-12.22°C)	SAE 20W-50	SAE 30	
2.	CARTRIDGE TYPE OiL FiLTERS (See Spec. Instr. 8 & 9)				50 HRS
3.	AIR FILTER (See Spec. Instr. 7)				50 hrs
4.	ENGINE CONTROL AND ENVIRONMENTAL CONTROL PIVOT POINTS (See Spec. Instr. 1 & 9)			MIL-PRF-7870C	100 HRS
5.	FRESH AIR VENT SHAFTS (See Spec. Instr. 2)			MIL-PRF-7711	500 HRS
6.	ALTERNATOR IDLER PULLEY BEARING (See Spec. Instr. 2)			MIL-PRF-81322E	100 HRS
7.	CARBURETOR AIRBOX SHAFT ASSEMBLY (See Spec. Instr. 2)			RELEASE AGENT/ DRY LUBRICANT MS-122AD	100 HRS

CAUTION: DO NOT OVER-LUBRICATE COCKPIT CONTROLS.

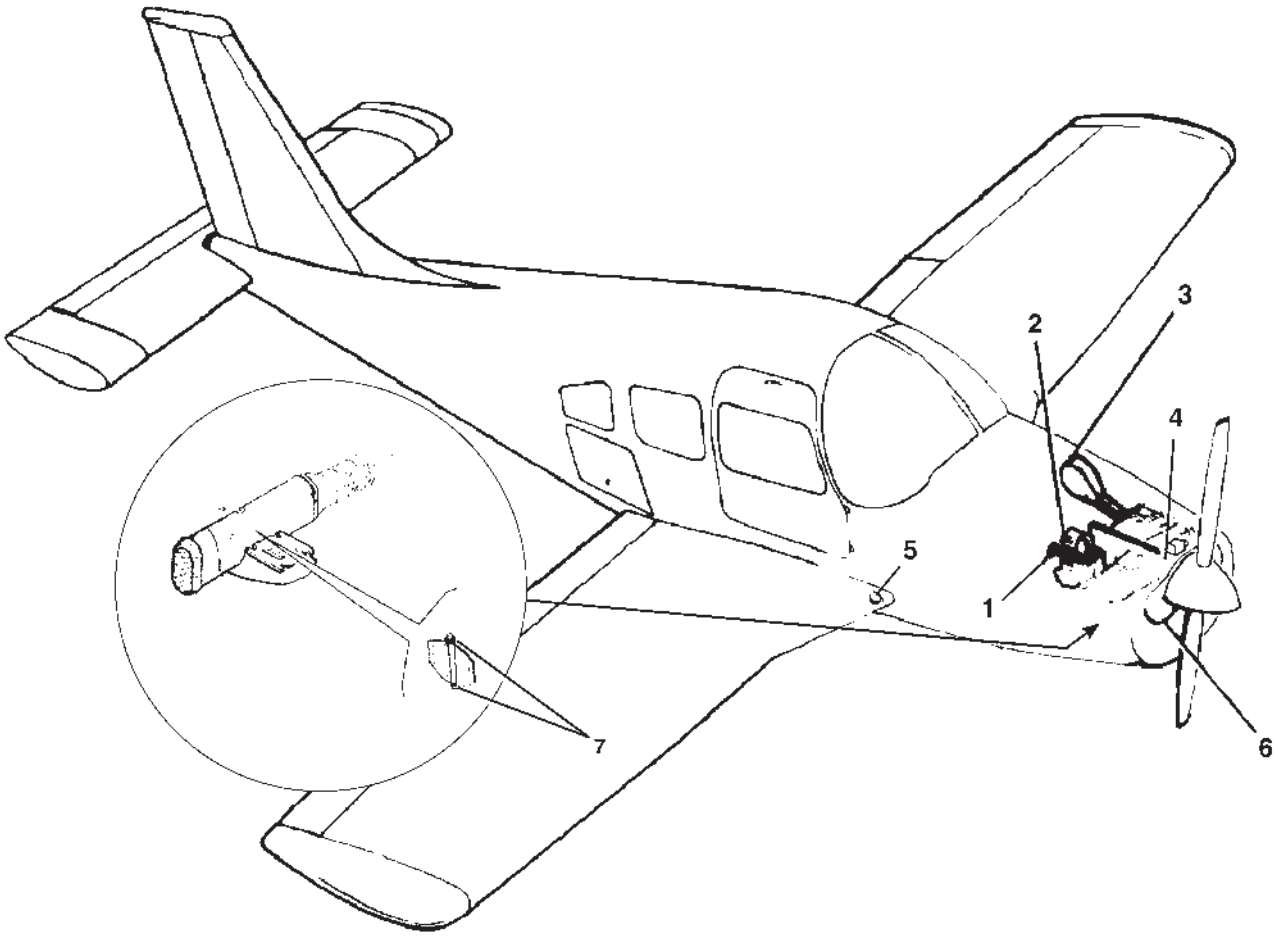


CHART 8
LUBRICATION - AIR CONDITIONING CONDENSER

	COMPONENT	LUBRICANT	FREQUENCY
1.	CONDENSER HINGE AND ACTUATORS (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS
2.	CONDENSER DOOR ACTUATING TRANSMISSION (See Spec. Instr. 12)	MIL-PRF-23827C	500 HRS

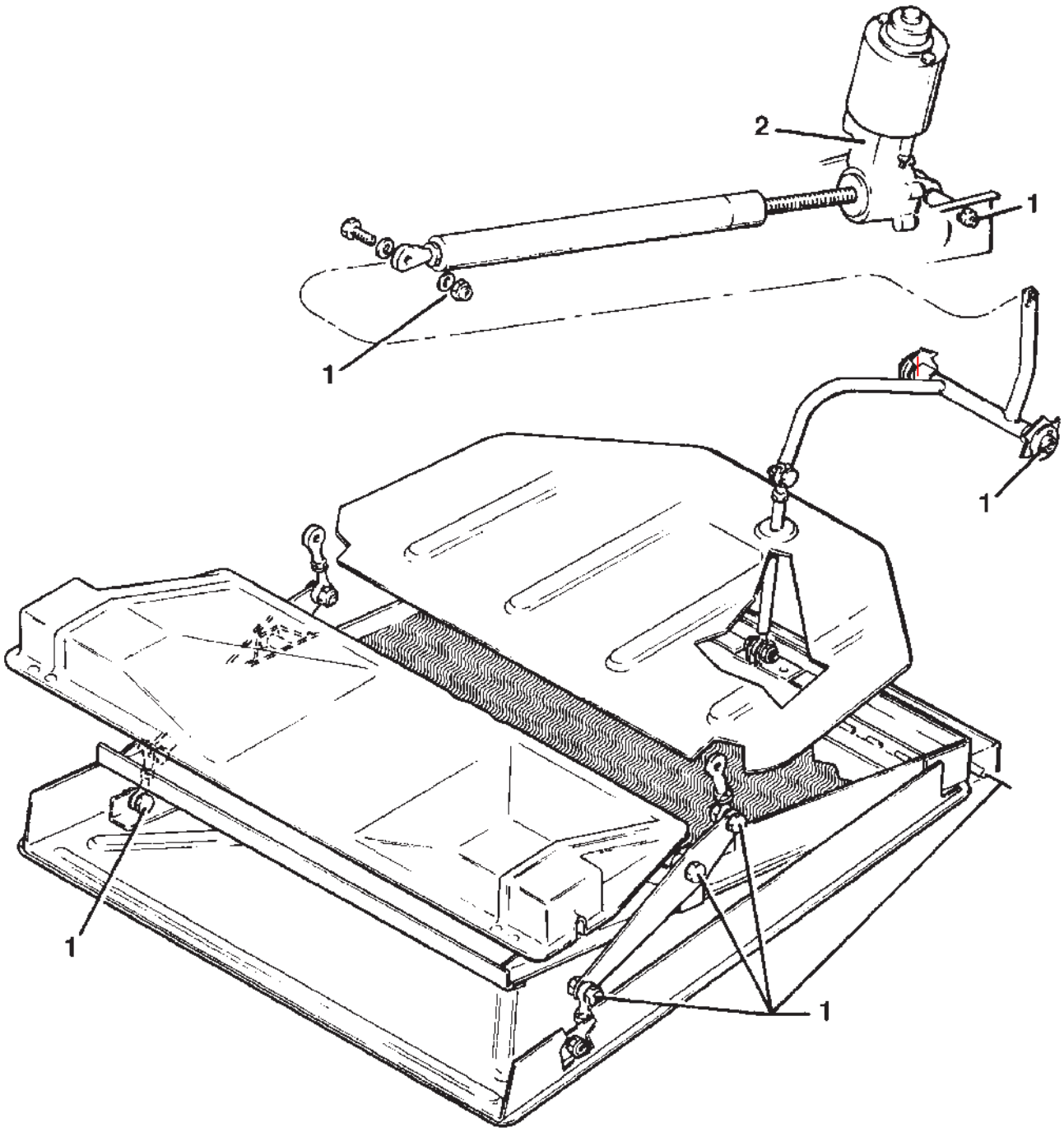
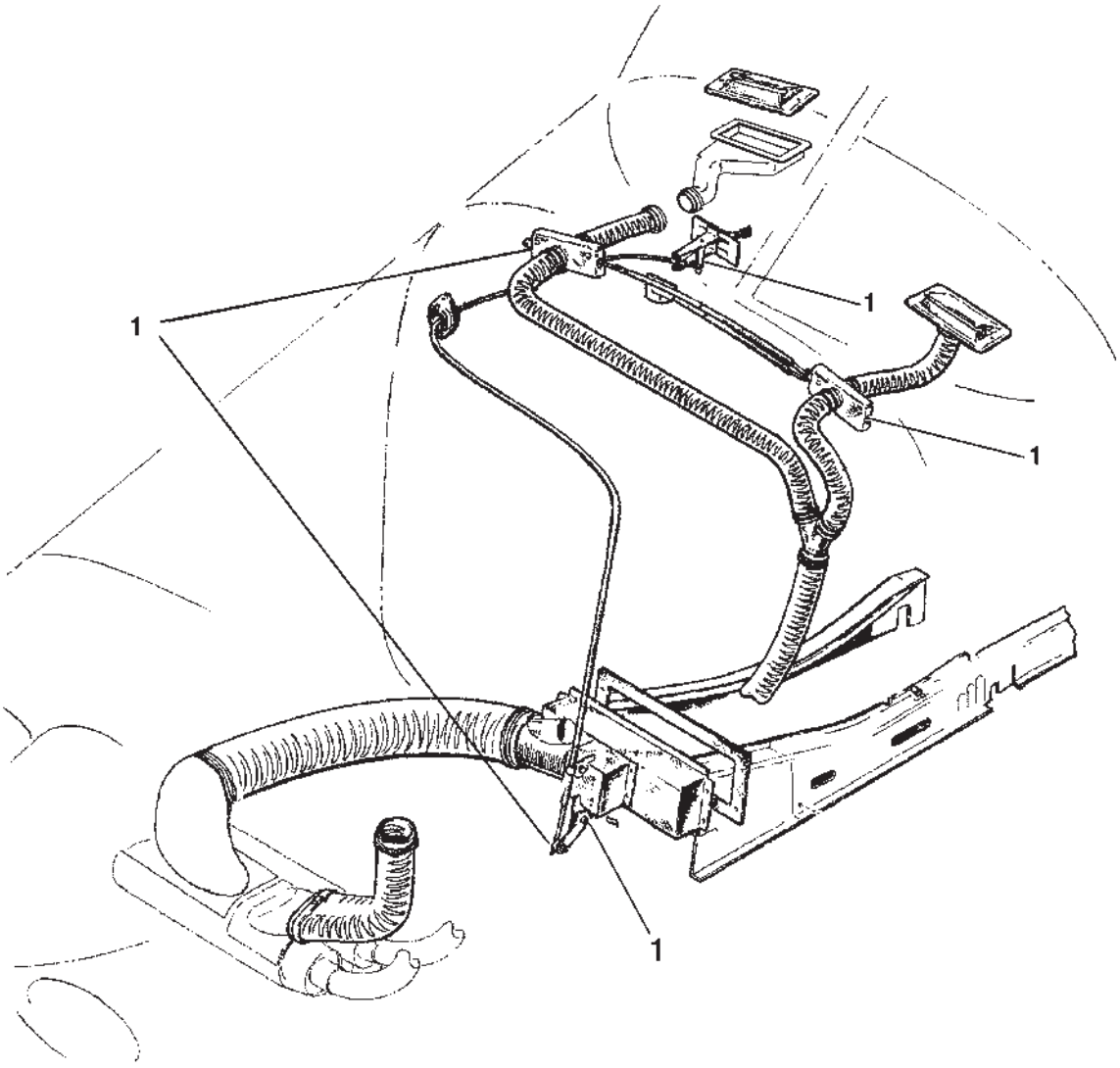


CHART 9
LUBRICATION - HEAT AND DEFROST SYSTEM

	COMPONENT	LUBRICANT	FREQUENCY
1.	Cabin Heat and Defroster Mechanisms (See Spec. Instr. 1)	MIL-PRF-7870C	100 HRS



CHAPTER

20

STANDARD PRACTICES - AIRFRAME

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CHAPTER 20

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
20-List of Effective Pages	1	Nov 15/17			
	2	Mar 6/14			
20-Table of Contents	1	Nov 15/17			
	2	Mar 6/14			
20-00-00	1	Mar 6/14			
	2	Mar 6/14			
	3	Mar 6/14			
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	6	Mar 6/14			
	7	Mar 6/14			
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	10	Mar 6/14			
20-20-00	1	Mar 6/14			
	2	Mar 6/14			
	3	Mar 6/14			
	4	Mar 6/14			
20-97-00	1	Nov 15/17			
	2	Nov 15/17			
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	6	Nov 15/17			

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CHAPTER 20 - STANDARD PRACTICES - AIRFRAME

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Removing Cherrylock Rivet		3
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Flareless Tube Assemblies		7
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Airframe Water Leak Check		10
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EMI Shielding		1
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Wrapped Drain Wire.		1
Installing The PRT-S Tubing.		2
Shrinking.		3
SPC5092 Wrap-A-Shield Tubing		6
Cut to size.		6
Attaching Drain Wire.		6
Install the Wrap-A-Shield Tubing.		6

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GENERAL

1. Description

This chapter contains general information pertaining to standard aircraft hardware installation and removal practices, as well as general information on corrosion control and painting issues.

For standard repair practices of a minor nature, see 51-70-00 and AC 43.13-1 (latest revision).

If non-destructive testing is needed after repair of 4130 steel, use a magnetic particle inspection method such as Magnaflux.

Testing and inspecting of aluminum castings and machined aluminum parts may be done by the dye penetrant method.

Usually, a good visual inspection with a 10X magnifying glass will show any damage or defect in a repair that is of a significant nature.

2. Torque Wrenches

Torque wrenches should be checked daily and calibrated by means of weights and a measured lever arm to ensure that inaccuracies are not present. Checking one torque wrench against another is not sufficient and is not recommended. Some wrenches are quite sensitive to the way they are supported during a tightening operation. Any instructions furnished by the manufacturer must be followed explicitly.

When it is necessary to use a special extension or adapter wrench together with a torque wrench, a simple mathematical equation must be worked out to arrive at the correct torque reading. Following is the formula to be used: (Refer to “Figure 1”.)

T = Torque desired at the part.

A = Basic lever length from center of wrench shank to center of handle or stamped on wrench or listed for that model wrench.

B = Length of adapter extension, center of bolt to center of shank.

C = Scale reading needed to obtain desired torque (T).

$$\text{The formula:} \quad C = \frac{A \times T}{A + B}$$

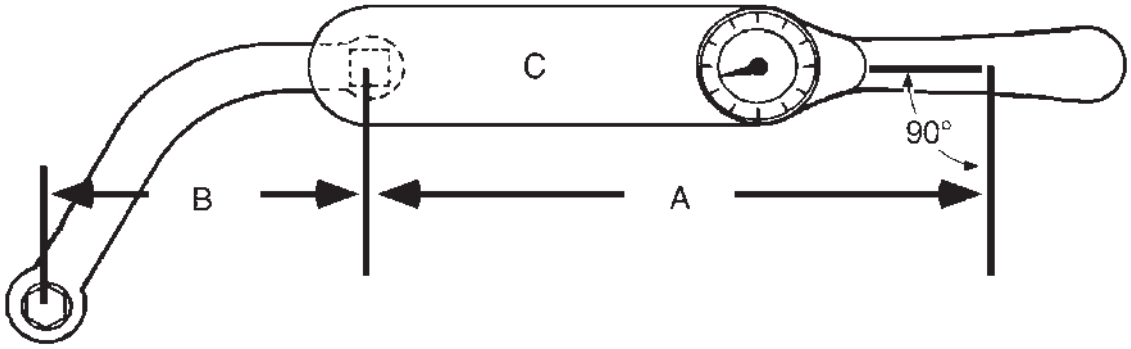
EXAMPLE: A bolt requires 30 foot pounds and a 3 inch adapter (one-quarter of a foot or 0.25’) is needed to get at it. You want to know what scale reading it will take on a one-foot lever arm wrench to obtain the 30 foot pounds at the bolt.

$$C = \frac{1 \times 30}{1 + 0.25} \quad \text{or} \quad C = \frac{30}{1.25}$$

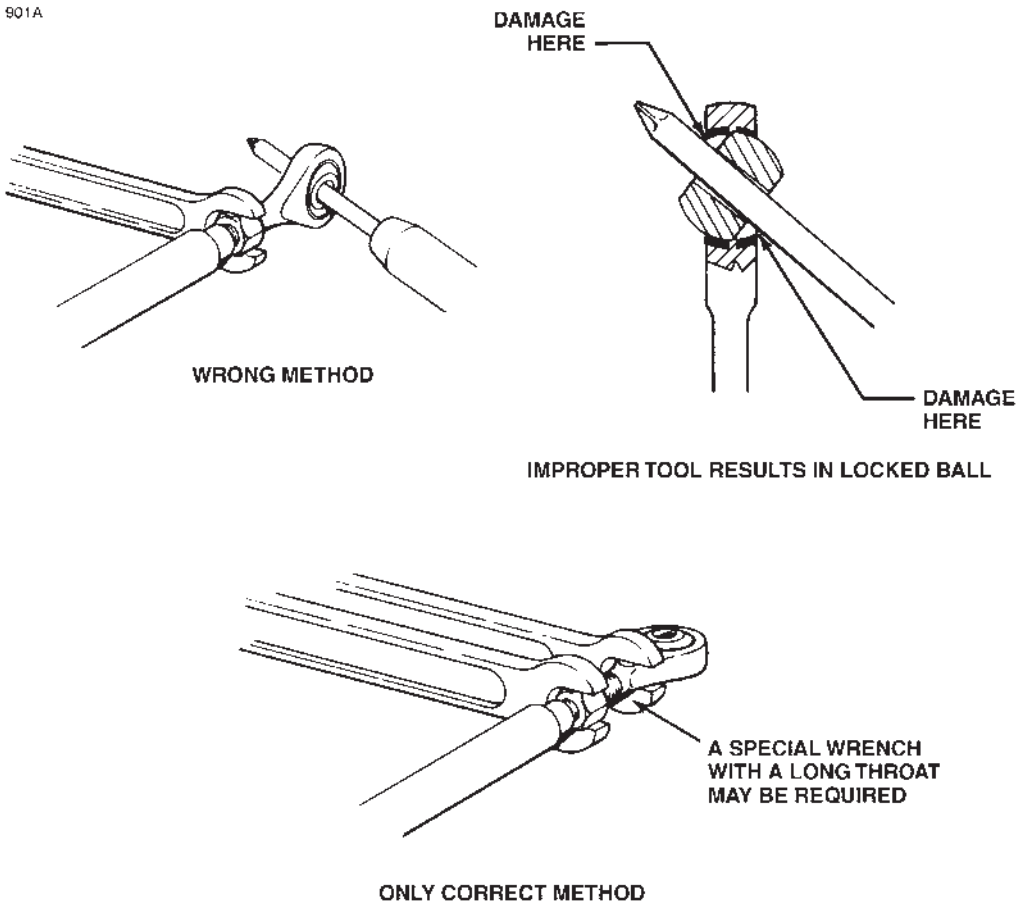
Remember, the 3 inch adapter must be projecting 3 inches straight along the wrench axis. In general, avoid all complex assemblages or adapters and extensions of flex joints.

3. Installing Rod End Bearings

Install rod end bearings as shown in “Figure 2”.



Torque Wrench Formula
Figure 1



Installing Rod End Bearings
Figure 2

4. Removing Cherrylock Rivets

Use following procedure to remove cherrylock rivets:

A. To remove from thick material, use a tapered steel drift pin to drive out rivet stem. (See “Figure 3”, View 1.)

CAUTION: DRIVING OUT THE LOCKED STEM OF RIVETS INSTALLED IN THIN MATERIAL MAY DAMAGE THE MATERIAL.

NOTE: Drilling completely through the rivet sleeve, when removing rivets, tends to enlarge hole.

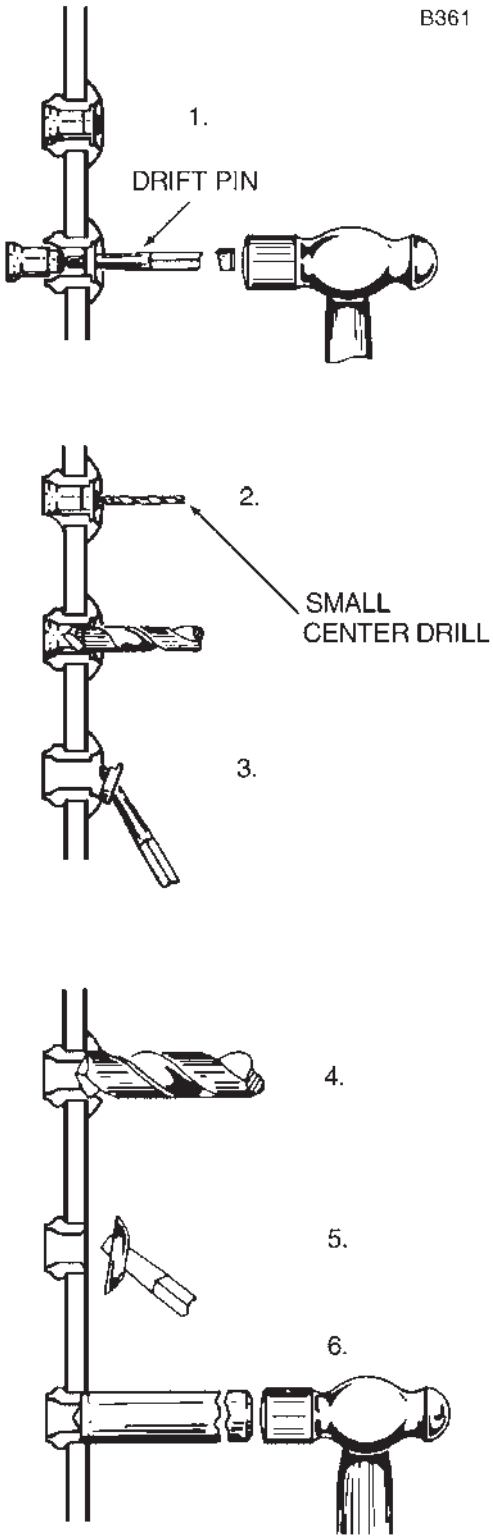
B. To remove from thin material, drill away tapered portion of stem to destroy the lock. Use a small center drill bit on top of the rivet stem to provide a guide for a larger bit. (See “Figure 3”, Views 2 and 3.)

C. Pry remainder of locking collar out of rivet head with a drift pin. (See “Figure 3”, View 3.)

D. Drill almost, but not completely, through head of rivet. Use a drill bit the same size as the rivet shank. (See “Figure 3”, View 4.)

E. Use a drift pin as a lever to break off rivet head. (See “Figure 3”, View 5.)

F. Drive out remaining rivet shank with a pin having same diameter as rivet shank. (See “Figure 3”, View 6.)



Removing Cherrylock Rivets
Figure 3

5. Identification of Fluid Lines

Refer to “Figure 4”.

Aircraft fluid lines are identified by color code markers, words and geometric symbols. The markers identify each line`s function, content, primary hazard, and the direction of fluid flow.

Most fluid lines are marked with 1 inch tape or decals. Paint is used on lines in the engine induction system.

Certain lines may also be identified as to the specific function within a system. For example: DRAIN, VENT, PRESSURE or RETURN.

Lines conveying fuel may be marked FLAM. Lines containing toxic materials are marked TOXIC. Line containing physically dangerous materials, such as oxygen, nitrogen, or freon, are marked PHDAN.

The aircraft and engine manufacturer is responsible for the original installation of identification markers, Aircraft maintenance personnel are responsible for their replacement when it becomes necessary.

Tapes, paint, tags and decals are placed on both ends of a line and at least once in each compartment through which the line runs. Identification markers are also placed immediately adjacent to each valve, regulator, filter or other accessory within a line.

6. Inspection of Flexible Hoses

NOTE: During the manufacturing process, a condition known as “rubber strike-through” occasionally occurs. This condition is such that rubber material protrudes through the wire braid cover. This condition has no effect on hose quality.

It is recommended that flexible hoses be inspected every 100 hours, especially those in the engine compartments. When inspecting hoses, look for the following conditions:

A. Check each installation to be sure the hose is not kinked, twisted, or distorted. Check for evidence of abrasion, cuts, and broken wires. Random broken wires are acceptable since wire breaks sometimes occur during manufacture. Discard hose if two or more broken wires are found per plait (braid) or more than six broken wires per lineal foot. Broken wires in an area where kinking is evident is also a cause for rejection.

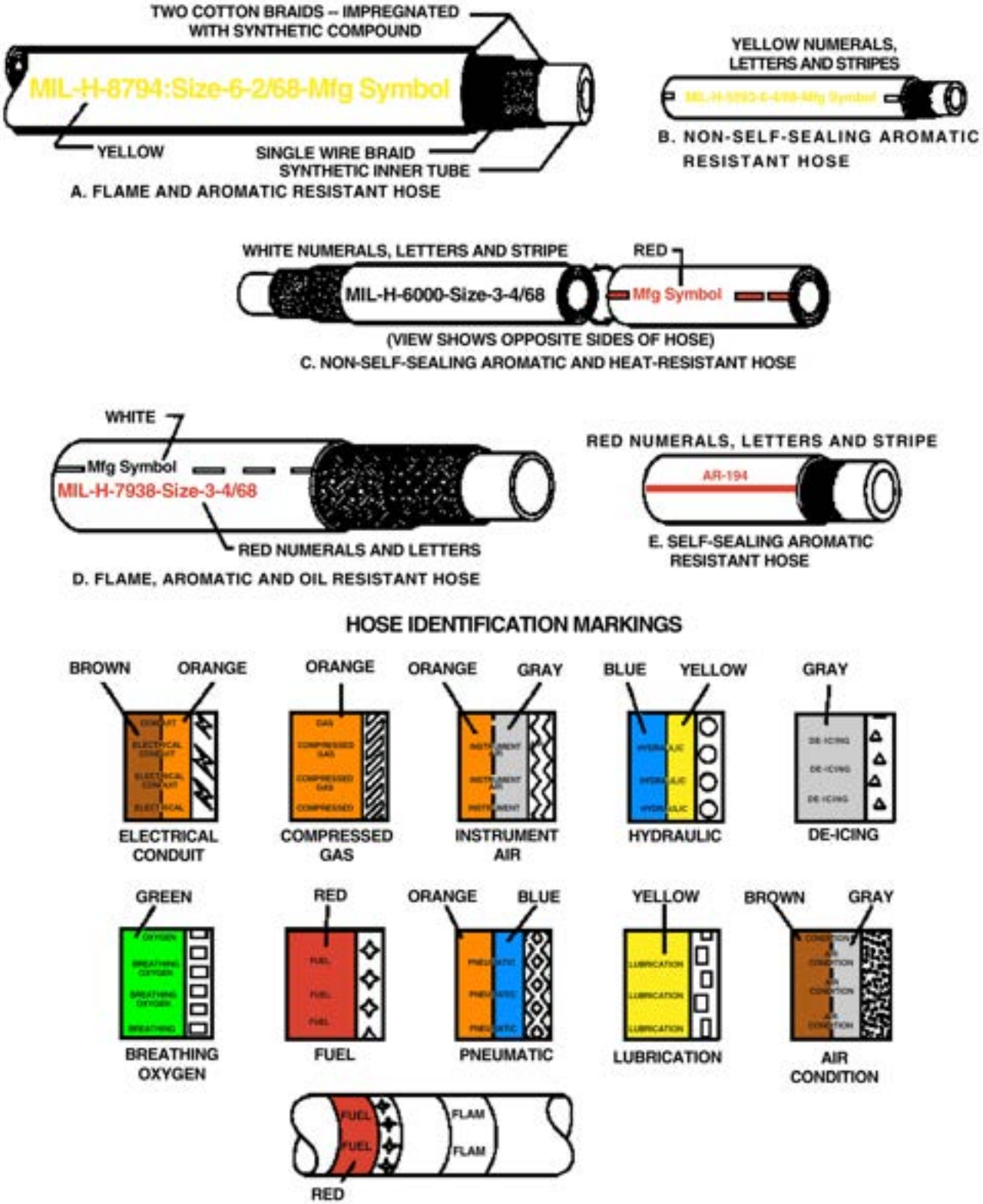
CAUTION: PUNCTURING THE OUTER COVER OF THE HOSE MAY CAUSE DAMAGE TO THE HOSE.

B. Check each assembly for deterioration, ply separation of cover or braid, cracks, weather checking, lack of flexibility, blisters or bulging, collapse, or sharp bending. Blisters on the outer synthetic cover do not necessarily indicate a faulty hose.

C. Remove hose from assembly if hose shows any visible wear. Inspect hose interior and check for signs of deterioration, tube collapse, cut rubber, wire braid puncture, or restriction. To inspect hoses with elbow fittings, use flexible inspection light and viewer, or inspection ball as described in “Chart 1” Replace hose if any deterioration exists.

CHART 1
BALL DIAMETERS FOR TESTING HOSE RESTRICTIONS

Hose Size	Ball Size
- 4	5/64
- 5	9/65
- 6	13/64
- 8	9/32
- 10	3/8
- 12	1/2
- 16	47/64
- 20	61/64

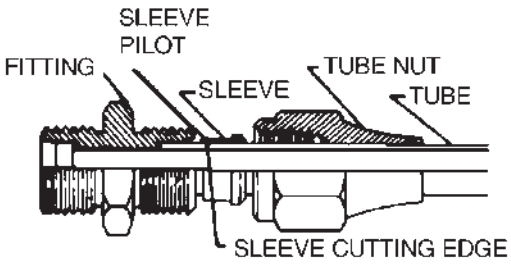


Hose, Tube, and Line Markings
Figure 4

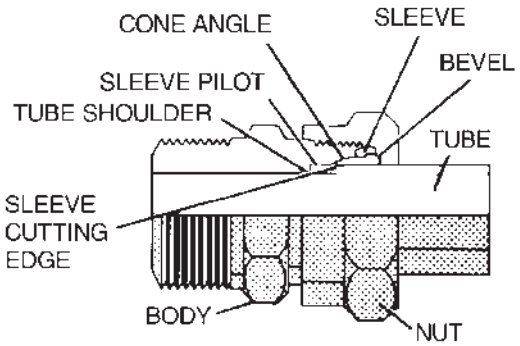
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TUBING AND HOSE LUBRICANTS

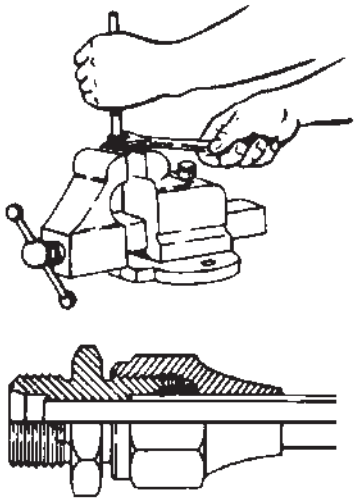
TUBING SYSTEM	LUBRICANT
HYDRAULIC	MIL-H-5606
FUEL	MIL-H-5656
OIL	SYSTEM OIL
PNEUMATIC	MIL-L-4343



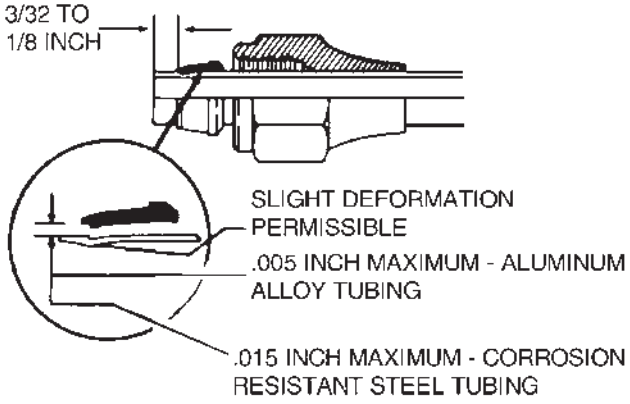
STEP 1



FLARELESS-TUBE FITTING



STEP 2



STEP 3

Flareless Tube Fittings
Figure 5

7. Flareless Tube Assemblies

The use of flareless tube fittings eliminates all tube flaring. An operation, referred to as presetting, is necessary prior to installing a new flareless tube assembly. Presetting is performed as follows:

A. Cut tube to correct length. Ensure ends are perfectly square. Deburr inside and outside of tube. Slip nut, then sleeve, over the tube. (Refer to “Figure 5”, Step 1.)

B. Lubricate fitting and nut threads as specified in table contained in “Figure 5”.

C. Place fitting in a vise (refer to “Figure 5”, Step 2). Hold tubing firmly and squarely on seat in fitting. (Tube must bottom firmly in the fitting.) Tighten nut until cutting edge of sleeve grips tube. This point is determined by slowly turning tube back and forth while tightening nut. When tube no longer turns, nut is ready for final tightening.

D. Final tightening depends upon type and size of tubing. On aluminum alloy tubing up to and including half inch outside diameter, tighten nut from 1 to 1-1/6 turns. On aluminum alloy tubing over half inch outside diameter, or steel tubing, tighten nut from 1-1/6 to 1-1/2 turns.

E. After presetting the sleeve, disconnect tubing from fitting and check the following points (refer to “Figure 5”, Step 3):

(1) Tube extends 3/32 to 1/8 inch beyond sleeve pilot to prevent blow off.

(2) Sleeve pilot contacts tube. A maximum clearance of 0.005 inch for aluminum alloy tubing, or 0.015 inch for steel tubing, is acceptable.

(3) A slight collapse of tube at sleeve cut is permissible. No movement of sleeve pilot, except rotation, is permissible.

8. Support Clamps

CAUTION: MAKE CERTAIN THAT CLAMPS ARE OF THE CORRECT SIZE. CLAMPS OR SUPPORTING CLIPS SMALLER THAN THE OUTSIDE DIAMETER OF THE HOSE MAY RESTRICT THE FLOW OF FLUID THROUGH THE HOSE.

Support clamps are used to secure the various lines to the airframe or power plant assemblies. Several type of support clamps are used for this purpose. The rubber cushioned and plain are the most commonly used clamps. The rubber cushioned clamp is used to secure lines subject to vibration; the cushioning prevents chafing of the tubing. The plain clamp is used to secure lines in areas not subject to vibration.

A teflon cushioned clamp is used in areas where the deteriorating effects of Skydrol 500,hydraulic fluid (MIL-H-5606),or fuel is expected, however, because it is less resilient, it does not provide as good a vibration damping effect as other cushion materials.

Use bonded clamps to secure metal hydraulic, fuel and oil lines in place. Unbonded clamps should be used only for securing wiring. Remove any paint or anodizing from the portion of the tube at the bonding clap location.

All plumbing lines must be secured at specified intervals. The maximum distance between supports for rigid fluid tubing is shown in “Chart 2”.

CHART 2
MAXIMUM DISTANCE BETWEEN SUPPORTS FOR FLUID TUBING

Tube O.D. (IN.)	Distance Between Supports (IN.) Aluminum Alloy	Steel
1/8	9-1/2	11-1/2
3/16	12	14
1/4	13-1/2	16
5/16	15	18
3/8	16-1/2	20
1/2	19	23
5/8	22	25-1/2
3/4	24	27-1/2
1	26-1/2	30

9. Dye Penetrant Inspections

NOTE: The following procedure is general in nature. See manufacturer's instructions, included with dye penetrant kit, for specifics.

A. Using a volatile cleaner, thoroughly remove dirt, loose scale, oil and grease from surface to be inspected.

B. Heat surface to at least 70°F (21°C), but not exceeding 130°F (54°C).

C. Apply penetrant by brushing, spraying, or dipping. Let stand 2 to 15 minutes, depending on temperature.

D. Remove surplus penetrant by applying special cleaner recommended by penetrant manufacturer, or by rinsing with water. Allow housing to dry.

E. Apply a light, even coat of developer by spraying, brushing, or dipping. Cracks or other opening in surface being inspected will appear as bright red. An indication of size of the defect may be obtained by watching the size and rate of growth of red indication.

10. Threaded Fastener Installation

(PIR-PPS20015-1, Rev. AD.)

A. Fastener Lengths

Fastener shank lengths must be long enough to prevent bearing loads on threads. In addition, the fastener or screw must extend through the nut.

The specified fastener grip length can be varied by one size (longer or shorter) to meet requirements stated above.

B. Washer Usage

Add a maximum of two NAS1149 washers (of the correct diameter, material and finish that matches the fastener being installed) under fastener heads or nuts to correct for variations in material thickness within the tolerances permitted. Where needed, use a maximum of two standard filler washers (spacers) under the nut to adjust for fastener length or alignment of cotter key hole. Where nutplates are used, adjust for protruding head fastener length by using up to a maximum of two standard filler washers under the fastener head.

C. Self-locking Fasteners

The use of self-locking nuts, fasteners and screws, including fasteners with non-metallic inserts is subject to the following limitations:

(1) Fasteners incorporating self-locking devices must not be reused if they can be run up using less than the required minimum torque values specified or as shown in Chart 2, 91-10-00. They may be reused, if hand tools are required to run them up, providing there is no obvious damage to the self-locking device prior to installation.

(2) Fasteners 5/16 inch diameter and over with cotter pin holes may be used with self-locking nuts.

(3) Self-locking nuts must not be used at joints which subject either the nut or the fastener to rotation.

(4) Self-locking fasteners shall never be tapped or rethreaded. Nuts, fasteners and screws with damaged threads or rough ends shall not be used, or rethreaded.

D. Torque

See Torque Requirements, 91-10-00.

11. Thread Lubrication

Lubricate all fittings on external lines, including attachment points at engine and other components, with proper lubricant as specified in “Chart 3”.

When applying thread lubricants, proceed as follows:

A. Thoroughly clean threads before applying lubricant.

B. Use thread lubricant sparingly.

C. Apply thread lubricant to male threads only.

D. Lubricate first three threads only on straight fittings.

E. Do not lubricate first two threads on tapered fittings; apply lubricant to next three threads only.

F. Ensure lubricant does not enter fittings or flared areas.

G. Lubricate any fittings going to engine with same fluid going through lines.

CHART 3
THREAD LUBRICANTS

Line	Lubricant
Brakes	MIL-PRF-5606
WARNING: DO NOT PERMIT MIL-T-5544 ANTI-SEIZE COMPOUND TO ENTER SYSTEM. APPLY TO FITTING THREADS ONLY.	
Air Conditioning Refrigerant	MIL-T-5544, Anti-Seize Compound, Graphite Petrolatum
Fuel	MIL-T-5544, Anti-Seize Compound, Graphite Petrolatum
Landing Gear Air Valve	MIL-PRF-907E, Anti-Seize Thread Compound, High Temperature
Oil	SAE-AMS-G-6032, Grease (Gasoline and Oil Resistant)
Pitot and Static	TT-A-580 (TT-S-1732) Anti-Seize Compound
CAUTION: LUBRICATE ENGINE FITTINGS ONLY WITH THE FLUID CONTAINED IN THE PARTICULAR LINE.	

12. Airframe Water Leak Check

(PIR-PPS60047, Rev. C.)

Use the following procedures to verify watertight integrity of the fuselage.

A. Required Equipment

(1) Choice of one of the following size/length combinations of standard garden hose:

3/4 inch inside diameter, maximum length 150 feet, or

5/8 inch inside diameter, maximum length 75 feet, or

1/2 inch inside diameter, maximum length 25 feet.

(2) A hose nozzle capable of adjustment. (Green Garden Nozzle 1220-C or equivalent is acceptable.)

B. Procedure

(1) Attach hose with nozzle to water tap at city water pressure. Check to insure that there are no leaks in the hose or fittings which would impair the check or cause other damage, with the tap full open.

(2) The check (except for baggage compartments located apart from the cabin) shall be conducted with an observer inside the aircraft.

(3) All cabin and baggage compartment doors and other controllable external closures shall be closed and latched during rainfall simulation.

(4) The spray shall be aimed directly at the openings around doors and windows, and at any other cockpit/cabin/separate baggage compartment closures, for a period of one-half to one and one-half minutes for each door, window, and other closure(s). The nozzle shall be held three to ten feet from point of water contact.

(5) If a leak allowing water to enter the cockpit/cabin/separate baggage compartment is detected, the sealant or caulking (see 91-10-00, Chart 10) shall be repaired, as required, and the check shall be repeated in the area where the leak was noted.

PAINTING

1. Painting Safety

WARNING: OVERSPRAY FROM CERTAIN ENAMELS, IF PUT IN WATER, IS FLAMMABLE. STORE ALL OVERSPRAY IN COVERED CONTAINERS AWAY FROM BUILDINGS WHERE SPRAYING OPERATIONS ARE CONDUCTED.

WARNING: WASH ALL RAGS AND SPONGES USED TO APPLY ANY PHOSPHORIC ACID CONVERSION COATINGS (ALODINE) BEFORE DISPOSAL. IF MATERIAL DRIES ON RAG, THERE IS DANGER OF SPONTANEOUS COMBUSTION.

WARNING: MIX DOPES AND LACQUERS WITH AIR DRILL. DO NOT USE ELECTRIC DRILL. ARCING ELECTRIC DRILL MOTOR WILL IGNITE FUMES.

WARNING: VERIFY SPRAY ROOM IS WELL VENTILATED. A CONCENTRATION OF FUMES WILL CAUSE A DANGEROUS FIRE HAZARD OR INSUFFICIENT OXYGEN FOR THE OPERATOR.

CAUTION: DO NOT ALLOW PAINT STRIPPER TO CONTACT FIBERGLASS REINFORCED PARTS SUCH AS RADOMES, RADIO ANTENNAS, WING PARTS, OR WING TIPS. FIBERGLASS STRUCTURES MAY BE FINISHED WITH ACRYLIC LACQUER OR POLYURETHANE ENAMEL AND ARE DAMAGED BY THE STRIPPER.

2. Polyurethane Paint Safety

WARNING: POLYURETHANE PAINT MAY BE DANGEROUS TO YOUR HEALTH. SERIOUS INJURY WILL RESULT IF SAFETY PRECAUTIONS ARE NOT FOLLOWED.

WARNING: DURING TRANSIT AND STORAGE CHECK FOR SIGNS OF A BULGING CAN, OTHER THAN NORMAL ODOR, OR A CHANGE IN RESIN FROM A CLEAR TO A CLOUDY STATE. A SLOW CARBON DIOXIDE BUILDUP WILL CAUSE CAN TO BURST. REMOVE AND PROPERLY DISPOSE ANY DEFECTIVE CANS.

WARNING: ENSURE ADEQUATE VENTILATION AND WEAR APPROPRIATE BREATHING PROTECTION FACE MASK WHEN PAINTING.

WARNING: POLYURETHANE PAINTS CAN PRODUCE IRRITATION OF THE SKIN, EYES, AND RESPIRATORY TRACT DURING MIXING AND APPLICATION. EXPOSURE TO SPRAY VAPORS AND MISTS DURING SPRAY APPLICATION MAY CAUSE BREATHING DIFFICULTY, SHORTNESS OF BREATH, AND DRY COUGH. INDIVIDUAL SUSCEPTIBILITY IS A CONTROLLING FACTOR. ONCE SENSITIZED, MANY PEOPLE CANNOT TOLERATE ANY EXPOSURE AND MUST THEREAFTER AVOID EXPOSED WORK AREAS.

WARNING: PRODUCTION TYPE MIXING AND SPRAY PAINTING OPERATIONS MUST BE IN SPECIALLY DESIGNED, EXHAUST-VENTILATED AREAS.

WARNING: PAINTERS MUST BE FULLY CLOTHED WITH COLLARS BUTTONED AND SLEEVES TAPED AT THE WRIST. PAINTERS MUST WEAR FITTED, DOUBLE CARTRIDGE ORGANIC VAPOR RESPIRATOR WITH FRESH CARTRIDGE INSERTED DAILY, SOLVENT-RESISTANT GAUNTLET STYLE GLOVES, AND SAFETY GOGGLES.

3. Paint Application

WARNING: GROUND AIRCRAFT BEFORE PAINTING SO NO STATIC ELECTRICITY CHARGES BUILD UP AND DISCHARGE.

CAUTION: PROTECT WINDSHIELD WHEN MASKING AIRCRAFT. PAINT STRIPPERS, METAL BRIGHTENERS, AND SOLVENTS WILL DAMAGE WINDSHIELD.

CAUTION: BALANCE MOVABLE CONTROL SURFACES AFTER PAINTING. REFER TO APPROPRIATE MAINTENANCE MANUAL SECTIONS.

CAUTION: BEFORE FORCE DRYING AT ELEVATED TEMPERATURES, VERIFY THAT ALL FUEL TANK VENTS ARE UNOBSTRUCTED AND WILL NOT RESULT IN EXPANDED FUEL SPILLING ON NEWLY PAINTED SURFACES OR PAINT BOOTH FLOOR.

CAUTION: DO NOT PAINT PITOT TUBES, GAS CAPS, OR ANTENNA COVERS THAT WERE NOT FACTORY PAINTED.

CAUTION: DO NOT USE METALLIC PAINTS ON RADAR CONES OR ANTENNA COVERS.

CAUTION: DO NOT ALLOW SILICONE LUBRICANTS TO CONTACT ANY SURFACES TO BE PAINTED. SILICONE LUBRICANT IS VERY DIFFICULT TO REMOVE COMPLETELY.

4. Painting Sequence

For primer, tack, finish coats, and lacquer application:

A. Position airplane so airflow is from tail toward nose and overspray ahead of you.

B. To minimize overspray problems, have two painters work simultaneously on opposite sides of airplane.

C. Paint difficult areas such as landing gear, and wheel wells before flat surfaces. Paint the ends and leading edges of ailerons and flaps. Paint flap and aileron wells, wing tips, and leading and trailing edges.

D. Paint the bottom of the airplane first including bottom of horizontal tail surfaces. Starting at the root and working outward, spray chordwise. Work up fuselage and allow spray to cover sides. Work up to engine. Spray wing bottom. Start each painter at the root and work toward tip, spraying chordwise.

E. Lower airplane tail enough to reach fin top. When spraying fuselage top, tilt spray gun so overspray is ahead of area being painted and new paint will wipe out overspray. Spray primer across fuselage, vertical and horizontal tail surfaces, and wing.

5. Color Matching

See aircraft logbooks for color codes.

6. Trim and Registration Numbers

Apply predominant color first over entire surface. Apply trim colors over base color after it dries. When top of fuselage is to be painted white with a dark color adjoining it, apply light color and feather into area to be painted with dark color. When light color dries, place masking tape and paper along separation line, and apply dark color.

Allow paint to dry several hours before removing masking tape. Remove tape by pulling slowly parallel to surface. This will reduce the possibility of peeling off finish with tape.

Apply registration numbers by painting or affixing self-adhering plastic figures. They must be solid color lines contrasting with background. Location and size of identification numbers vary, per aircraft size. Location and size is found in Federal Aviation Regulations.

7. Paint System Compatibility

Before applying new paint, find what type finish was used previously. Refer to the Piper parts catalog for correct paint number and color.

Identify paint finishes by applying engine oil to a small surface area. Old nitrocellulose finishes will soften in a few minutes. Acrylics, urethanes, and epoxy finishes show no effects.

If not identified, wipe down a small area with rag wet with methy ethyl ketone. MEK picks up pigments from acrylic finishes, but not from epoxy or cured urethane coatings. Wipe surface, do not rub. Heavy rubbing picks up epoxy and urethane pigments from coatings not fully cured.

The use of different types of paint, with several coatings, make repair of damaged and deteriorated areas difficult. Paint finishes are not always compatible. The following are general rules for compatibility and are not necessarily listed in order of importance.

A. Old type zinc chromate primer may be used directly for touchup of bare metal surfaces and on interior finishes. It may be overcoated with wash primers if in good condition. Acrylic lacquer finishes will not adhere to this material.

B. Modified zinc chromate primer will not adhere to bare metal. Never use it over a dried film of acrylic nitrocellulose lacquer.

C. Nitrocellulose coatings will adhere to acrylic finishes, but reverse is not true. Do not use acrylic nitrocellulose lacquers over old nitrocellulose finishes.

D. Acrylic nitrocellulose lacquers will not adhere to nitrocellulose and epoxy finishes and to bare metal. For best results, apply lacquers over fresh, successive coatings of wash primer and modified zinc chromate. They also adhere to freshly applied epoxy coatings (dried less than 6 hours).

E. Epoxy topcoats adhere to all paint systems in good condition. Use epoxy for general touch touchup, including touchup of defects in baked enamel coatings.

F. Old wash primer coats may be overcoated directly with epoxy finishes. Apply a new second coat of wash primer if an acrylic finish is to be applied.

G. Old acrylic finishes may be refinished with new acrylic provided old coating is thoroughly softened using acrylic nitrocellulose thinner before paint touchup.

H. Repair damage to epoxy finishes by using more epoxy. Neither lacquer finish will stick to epoxy surfaces. In some instances, air drying enamels may be used for touchup of epoxy coatings if edges of damaged areas are roughened with abrasive paper.

8. Common Paint Troubles

A. Poor Adhesion - Paint properly applied to correctly pretreated surfaces will adhere satisfactorily. When thoroughly dry, paint must not be easily removed. Poor adhesion can result from:

(1) Inadequate cleaning and pretreatment.
(2) Inadequate stirring of paint or primer.
(3) Coating at incorrect time intervals.
(4) Application under adverse conditions.
(5) Bad application.

B. Spray Dust - Spray dust caused by atomized particles drying before reaching surface being painted fail to flow as a continuous film. Usual causes are incorrect air pressure or distance gun is held from work.

C. Sags and Runs - Excess paint causes wet paint film to move by gravity and presents a sagging appearance. Incorrect viscosity, air pressure, and gun handling, or inadequate surface preparation are frequent causes.

D. Spray Mottle - Sometimes known as orange peel or pebble, is caused by incorrect paint viscosity, air pressure, spray gun setting, or the distance the gun is held from work.

E. Blushing is one of the most common troubles. It appears as clouding or blooming of paint film. It is more common with cellulose than synthetic materials. It may be caused by moisture in air supply line, adverse humidity, drafts, or sudden temperature changes.

9. Storage

A. Store paint, enamel, and other finishing material in dry storage away from direct sunlight and heat. Mark each container with a code for identification.

B. Storage facilities must comply to Occupational Safety and Health Act (OSHA) requirements regarding air circulation, lighting, and fire protection. Lock storage facilities to prevent children and unauthorized personnel entry.

C. Invert pigmented materials every inventory so pigments will not pack to can bottom. Properly dispose of empty containers.

D. Use older materials first. Useful life of some finishes is limited.

E. Storage area temperatures must be approximately 50-90°F. If finishes are stored in temperature extremes, allow them to return to room temperature before using.

10. Painting Facility

WARNING: DO NOT BREATHE PAINT FUMES. FUMES DEPLETE THE OXYGEN SUPPLY REQUIRED BY THE BODY.

A. Painting facilities must conform to local, state, and OSHA standards with respect to air circulation, exhaust emissions, lighting, and fire protection.

B. Provide sufficient air movement in painting area so there is only a slight finishing material odor. Exhaust fans must be belt-driven and located near floor level. Locate fan's motor away from fumes.

C. All spraying area personnel must wear approved respiration safety equipment.

11. Aircraft Finish Care

See 12-00-00.

WIRING

Airplanes equipped with the Garmin GFC 700 Automatic Flight Control System have additional EMI shielding on some harness assemblies. If repair is required, use the following procedures as appropriate.

EMI Shielding

A. PRT-S Wrap Around Heat Shrink Tubing

The following procedure is for installing “PRT-S” wrap-a-round, heat shrinkable, EMI shielded tubing.

(1) Cut to size.

(a) Select a tubing diameter appropriate for the intended repair per “Chart 1”.

NOTE: Selecting a PRT-S tubing size smaller than recommended may result in the material splitting during the heating process.

(b) Cut the tubing to length and also cut an appropriate length of drain wire.

NOTE: When PRT-S is removed from the original spool it may have a flat profile with creased edges. Do not be alarmed by this shape, it will disappear when the product shrinks.

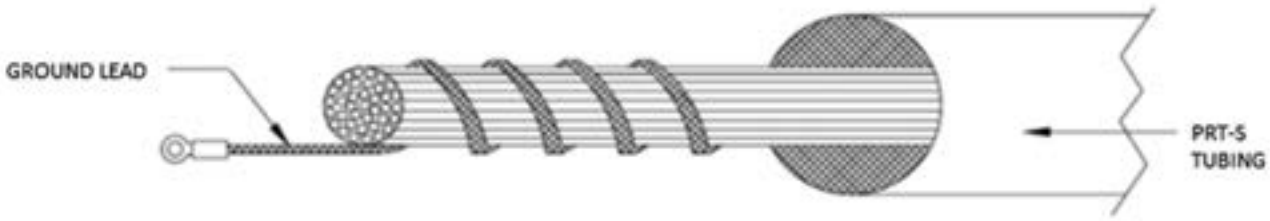
(2) Wrapped Drain Wire.

A ground lead can be added by simply wrapping a piece of woven braid around the primary wire bundle prior to installing the PRT-S tubing over the cable. The ground lead should be helically wound five revolutions around the outside of the primary wires. The helical winding ensures that the bare lead will always be in contact with the shield due to the compressive contact of the shrink tubing. This winding will also create enough friction to keep the lead from pulling out if tugged on. See “Figure 1” on page 20972.

CHART 1
EMI SHIELDING SPECIFICATIONS

Harness	EMI Shielding
A/P Servo (Pitch/Roll)	SPC5092 or ZT14-04-005-3
A/P Servo (Pitch Trim/Yaw)	SPC5092 or ZT14-04-005-3
Magnetometer	SPC5092 or ZT14-04-005-1
OAT Probe	SPC5092 or ZT14-04-005-1
Wire Harness Size (Diameter Inches)	**EMI Shielding Product Name / Manufacturer / P/N**
up to 0.5	Wrap-A-Shield / Pro Power (aka SPC Tech) / SPC5092
0.125–0.187	PRT-S / Zippertubing / ZT14-04-005-1
0.25–0.37	PRT-S / Zippertubing / ZT14-04-005-3

(PIR-107872 New.)



Drain Wire
Figure 1



Wrapping PRT-S Around Harness
Figure 2

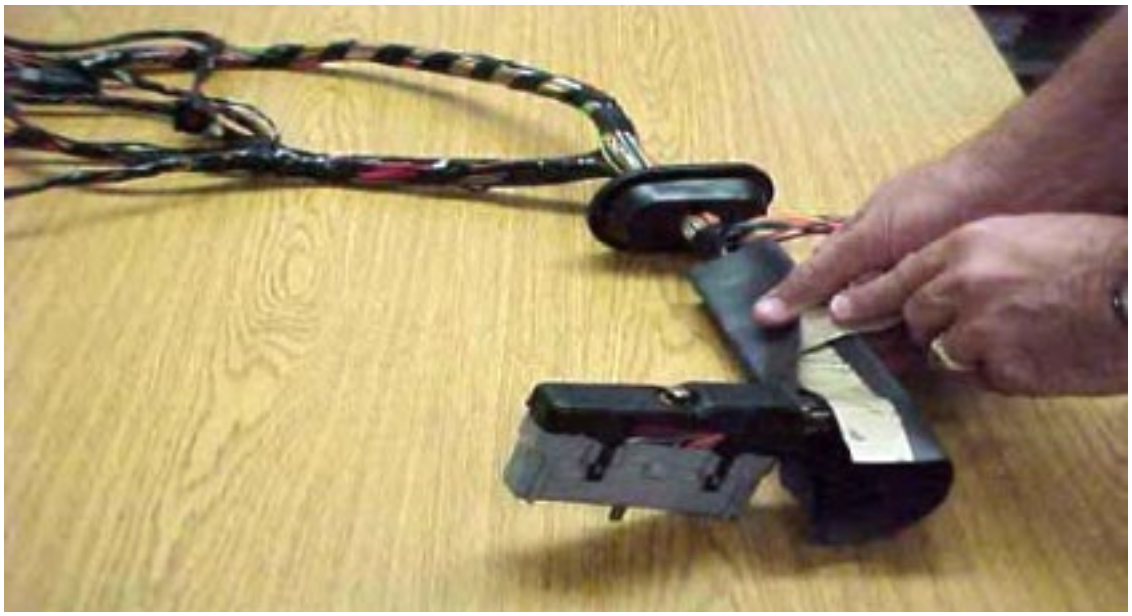
(3) Installing The PRT-S Tubing.

(a) Lay the tubing out so the metallic EMI shield material faces up, install the PRT-S tubing around the cable so the metallic shield cloth will be sandwiched between the wires and the heat shrinkable jacket material. Wrap the loose edge of the shield cloth around the wires and cover as much of the cable as possible. Peel off an inch or two of the adhesive's paper release liner on the shield material edge. Stick the adhesive edge to the wire bundle or cables. Wrap the entire assembly length in this manner. See "Figure 2".

NOTE: Wrap the shield around the cable as much as possible to ensure complete EMI shield coverage when the tubing is fully shrunk.

(b) Remove a short section of the paper release liner that covers the adhesive on the shrink tubing. Position the non-adhesive edge of the tubing over the exposed adhesive section and align so all the adhesive is covered but no more. Never over-wrap the tubing so much that you leave a loose flap of material at the overlap. Avoid contacting the adhesive with your fingers as much as possible. Remove short sections of the release liner until the entire length of tubing is sealed closed. Rub the overlap seam area firmly using your thumb to work out any wrinkles. Minor wrinkles will disappear when the tubing shrinks. See"Figure 3".

NOTE: Do not be concerned about the large size of the sealed tubing versus the wire size. It will always appear extremely large. You may shrink the tubing in place immediately after sealing; the closure adhesive system has no cure time.



PRT-S tubing
Figure 3

(4) Shrinking.

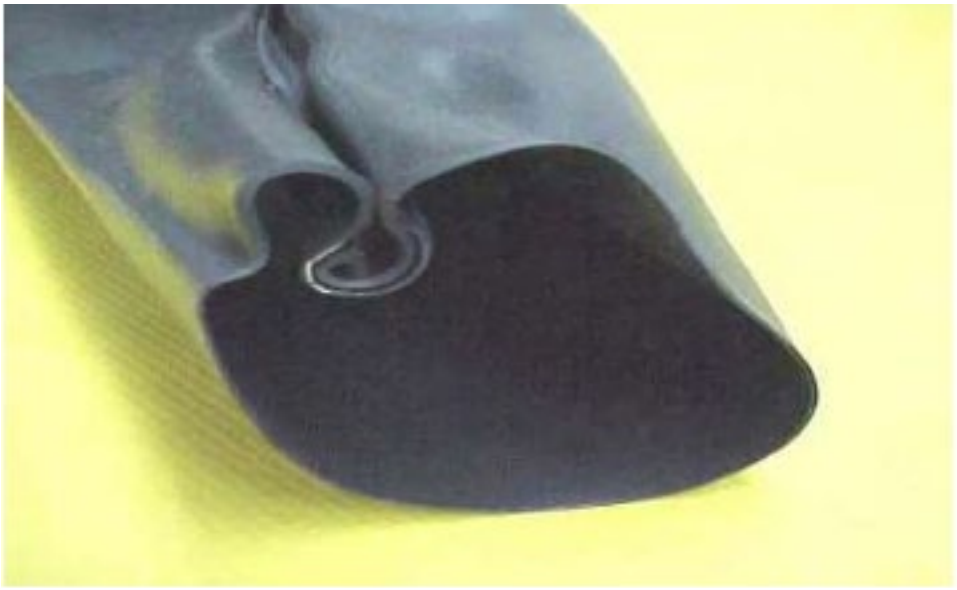
(a) PRT-S tubing begins to shrink at 90°C (194°F), however to achieve a timely and uniform installation it's desirable to use a heat gun temperature setting of 120-130°C (248-266°F). Always use a small reducer nozzle that will allow you to localize the stream. Never use a tubing reflector designed to spread the heat evenly or infrared type of heat gun.

(b) Begin shrinking the PRT-S tubing by directing the heat source at the overlap seam area near the center of the tubing. See "Figure 4" on page 20974. Apply heat to the overlap area only! The overlap area will tend to curl up into a "U" shape as it is heated. See "Figure 5" on page 20974. Continue applying heat in short bursts to this area until the surrounding tubing begins to pull the "U" shape back down flat. Do not be afraid to apply extra heat to the overlap area, as it will take extra dwell time for the inside layer of the overlap to shrink. Do not be alarmed if the outer layer of the overlap flap tends to lift off the adhesive and curl back slightly during initial heat exposure. With continued heating this lifted area will generally lay back down on its own, if not, lightly tap the lifted edge as necessary with a wooden stick. Shrink the overlap seam along its entire length before attempting to shrink the remaining tubing.

(c) After the overlap seam has fully shrunk begin applying heat to the remaining tubing. With small diameter tubing sizes you may find that most of the remaining tubing has already shrunk. Visually examine the entire circumference of the tubing and apply localized heat to any areas that show evidence of insufficient heating. While the tubing is hot you may work or form the PRT-S tubing and cable as necessary. Once properly positioned allow the finished assembly to cool to the point of being warm.



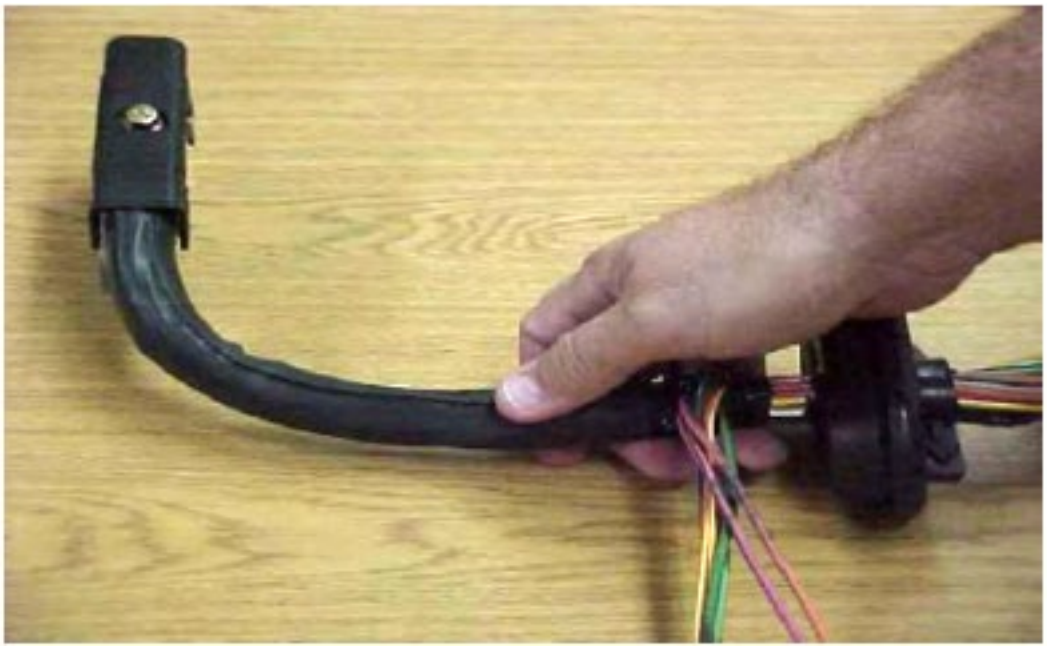
Heat Shrinking PRT-S
Figure 4



PRT-S Curling into "U" Shape
Figure 5

CAUTION: NEVER ATTEMPT TO REMOVE EXCESS ADHESIVE FROM THE OVERLAP AREA USING SOLVENT.

(d) The finished assembly may display a shiny, sticky area, parallel to the overlap seam. This is residual adhesive that was exposed as the overlap material pulled back during shrinking. The excess adhesive can be removed while the tubing is warm by rubbing your thumb along the overlap seam line. Rubbing the seam area will ensure good overlap contact and the excess adhesive will ball up ahead of your thumb. Remove excess adhesive as necessary and discard. See “Figure 6” and “Figure 7”.



Rubbing Adhesive Seam PRT-S
Figure 6



Residual Adhesive Ball PRT-S
Figure 7

B. SPC5092 Wrap-A-Shield Tubing

The following procedure is for installing “Wrap-A-Shield” tubing (SPC5092), a wrap around, EMI shielded tubing. See “Figure 8”.

(1) Cut to size.

Cut the tubing to length and also cut an appropriate length of drain wire.

(2) Attaching Drain Wire.

(a) Locate the braid strap, and then solder the drain wire to the braid strap.

(b) Once the solder has cooled, wrap the drain wire around the primary wire bundle prior to installing the shield tubing over the cable.

NOTE: The ground lead should be helically wound four revolutions around the outside of the primary wires. This will form a strain relief to keep the lead from pulling out if tugged on.

(3) Install the Wrap-A-Shield Tubing.

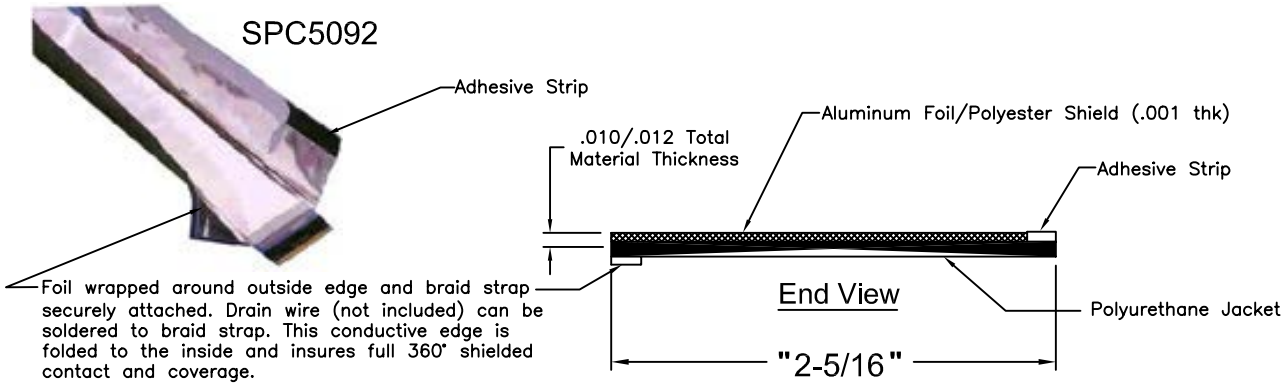
(a) Lay the tubing out so the metallic EMI shield material faces up, with the ground lead attached.

(b) Lay in your primary wire bundle, (don’t forget to wrap your ground lead as mentioned in Step (2)) starting from the braid strap edge begin to roll your shield and wire bundle towards the adhesive strip.

(c) Removing the backing paper from the adhesive strip, continue to roll your bundle, closing the tube as you roll over the exposed adhesive.

(d) Now secure the shield using MS3367-7-9 cable ties, starting one (1) inch from each end and at six (6) inch intervals.

(e) Finalize the assembly by rubbing your thumb along the overlapped seam to ensure good overlap contact with the adhesive.



SPC5092 Wrap-A-Shield
Figure 8

CHAPTER

21

ENVIRONMENTAL SYSTEMS

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CHAPTER 21

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CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
21-List of Effective Pages	1	May 15/19	21-50-00 (cont.)	19	Mar 6/14
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				21	Mar 6/14
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21-50-00	1	Sep 1/16		46	May 15/19
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	5	Mar 6/14		50	Sep 1/16
	6	Mar 6/14		51	Sep 1/16
	7	Mar 6/14		52	Mar 15/20
	8	Mar 6/14		53	Sep 1/16
	9	Sep 1/16		54	Sep 1/16
	10	Mar 6/14		55	Sep 1/16
	11	Mar 6/14		56	Sep 1/16
	12	Sep 1/16		57	Sep 1/16
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	14	Mar 6/14			
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GENERAL

Environmental control is available in a standard ram air configuration or one of two optional configurations.

1. Standard

   A. Ram Air Cabin Ventilation

      See 21-20-00.

      Fresh air from an inlet on the left rear fuselage is ducted into overhead cabin vents. Additionally, fresh air inlets in the leading edge of each wing feed adjustable outlets near the floor at each seat location.

   B. Ram Air Heat and Defrost

      See 21-40-00.

      Fresh air from a heater inlet duct on the lower cowling passes through a shroud around the standard muffler and is ducted to the windshield and through the cabin.

2. Optional

   A. Fresh Air Blower

      See 21-20-00.

      Fresh air from an inlet on the left rear fuselage is ducted into overhead cabin vents by a two-speed blower mounted aft of the rear closeout panel.

   B. Air Conditioning

      See 21-50-00.

      If installed, the optional air conditioning system uses a two-speed blower to recirculate cabin air through an evaporator and filter located in the aft fuselage. The cooled air is then ducted back into the cabin through the overhead vents.

      A compressor mounted on the front of the engine takes heat-laden, vaporized, refrigerant from the evaporator and pumps it to a condenser mounted in the aft fuselage where it is cooled to a liquid state and pumped back into the evaporator.

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DISTRIBUTION

1. Overhead Vent System

When the optional air conditioning system is not installed cabin ventilation and cooling is provided solely by fresh air through the overhead vent system (“Figure 1”).

A. Ram Air Installation (Standard)

Air enters an inlet located on the rear left side of the fuselage and is ducted through the overhead vents to the cabin. Fresh air flow is controlled by a flapper valve positioned in the duct just forward of F.S. 220. A CABIN AIR knob, located in the cockpit overhead, controls the flapper valve.

B. Overhead Vent Blower (Optional)

The blower is mounted in the aft section of the fuselage and is connected to the overhead vent system. The blower draws air in from the left side of the fuselage and forces it through the ducting. The blower switch is mounted on the instrument panel and has a three position switch providing two blower speeds.

(1) Removal

(a) Remove the access door from the aft wall of the baggage area.

(b) With the master switch off, disconnect the plug assemblies at the blower assembly.

(c) Remove the inlet and outlet hoses from the blower assembly by removing the clamps.

(d) Remove the screws, washers and nuts that secure the blower assembly to the hanger braces.

(e) Remove the screws and washers which secure the blower assembly to the retainer and hangers.

(f) Remove the blower assembly from the aircraft.

(2) Installation

(a) Position blower assembly in hangers and retainer. Install washers and screws.

(b) Secure blower assembly to hanger braces nuts, washers, and screws.

(c) Seal all hose joints with 3M 390 Duct Tape.

(d) Install inlet and outlet hoses. Secure with clamps.

(e) Ensure master switch is OFF. Connect plug and receptacles to blower (See “Chart 1”).

(f) Check blower for proper operation.

(g) Install access door to aft wall of baggage area. Secure with attaching hardware.

NOTE: Pin number 1 is at pointed side of plug and receptacle.

(3) Disassembly

(a) Remove the hose duct from the forward edge of the blower assembly by removing the nuts, washers and screws.

(b) Remove the cover from the blower assembly by removing the nuts, washers and screws.

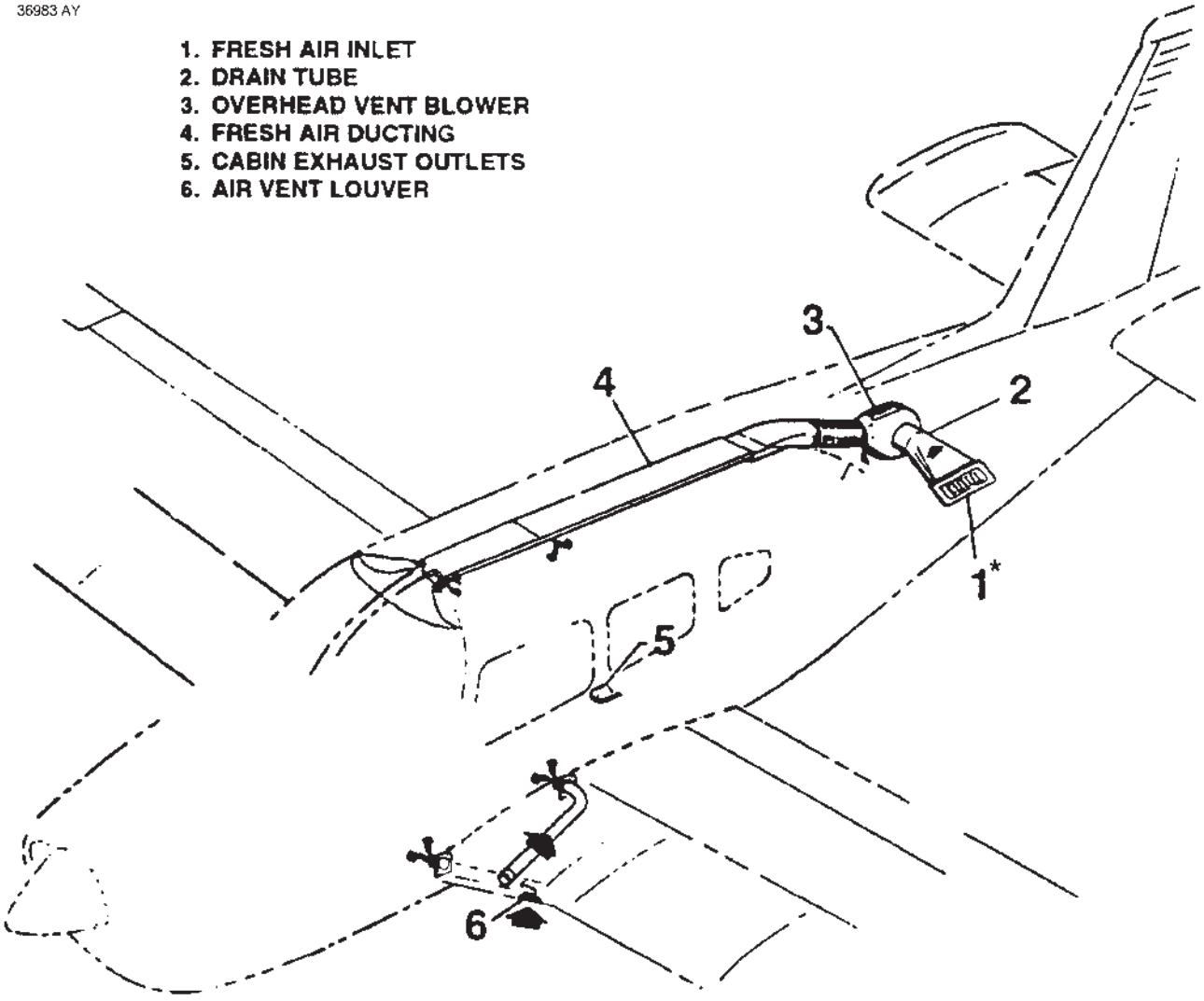
(c) Remove the blower fan from the motor shaft by removing the set screw.

(d) For removal of the motor, proceed as follows:

1) Separate the plate from the motor cover by carefully drilling out the connecting rivets.

2) Cut the motor wires at the edge of the receptacle and plug and remove the wire ends from the blocks.

3) Remove the motor from the mounting plate by removing the nuts, washers and bolts.



* In S/N's 2843501 and up, if removed, fay surface seal area of contact between duct assembly and fresh air inlet louver with 3M Strip Caulk 08578 (Piper P/N 279-110) upon reinstallation.

Overhead Vent System
Figure 1

CHART 1
BLOWER SYSTEM WIRE COLOR CODES

	Motor Wires				Aircraft Wires		
		Pin No.(1)	Motor - P/N 761-778 (Dukes 1482-22-1)		Aircraft Harness	Pin No.(1)
Ground	Plug	2	Green	Receptacle	AC26A	2
Low Speed		1	Red		Black	1
	Receptacle	2		Plug	White(2)	2
High Speed		1	Orange		Red	1

NOTES: 1. Pin No. 1 is at the pointed side of the plug or receptacle.
2. Not used (i.e. - folded back and taped off.

(4) Assembly

(a) Mount motor on plate and secure with bolts, washers, and nuts. Check that motor nuts are snug and shaft spins freely.

(b) Position cover over motor plate with motor wires protruding through cover grommet.

(c) With holes in cover matching holes in motor plate, secure the two parts together with rivets.

(d) Fill any opening left where wires are pass through cover grommet with Sealant (PRC-5000), Airframe and Window (see 91-10-00, Chart 8).

(e) Install wires in plug and receptacle.

(f) Position blower fan on motor shaft and secure with set screw.

(g) Position hose duct on blower assembly and secure with screws, washers, and nuts. Install screws with heads inside duct.

(h) Clean old sealant from all surfaces where duct attaches to blower assembly.

(i) Seal all surfaces where duct attaches to blower assembly with Sealant, Airframe and Window (see 91-10-00, Chart 8).

2. Avionics Cooling Blower

A. Standard and Avidyne Installations

An avionics cooling blower is mounted to a bracket attached to the fuselage side-panel just forward of (i.e. - behind) the instrument panel. In S/N's 2843001 thru 2843202, the blower is on the left side; in S/N's 2843203 thru 2843700 and 2843702, the blower is on the right side. See "Figure 2".

NOTE: Actual equipment served will vary depending on optional equipment installed.

(1) Removal

(a) Remove ducts from top of blower. (Ducts are held in place by Ty-Raps.)

(b) Disconnect blower motor wires at connector.

(c) Remove screws, washers and nuts that attach the blower assembly to its mounting bracket.

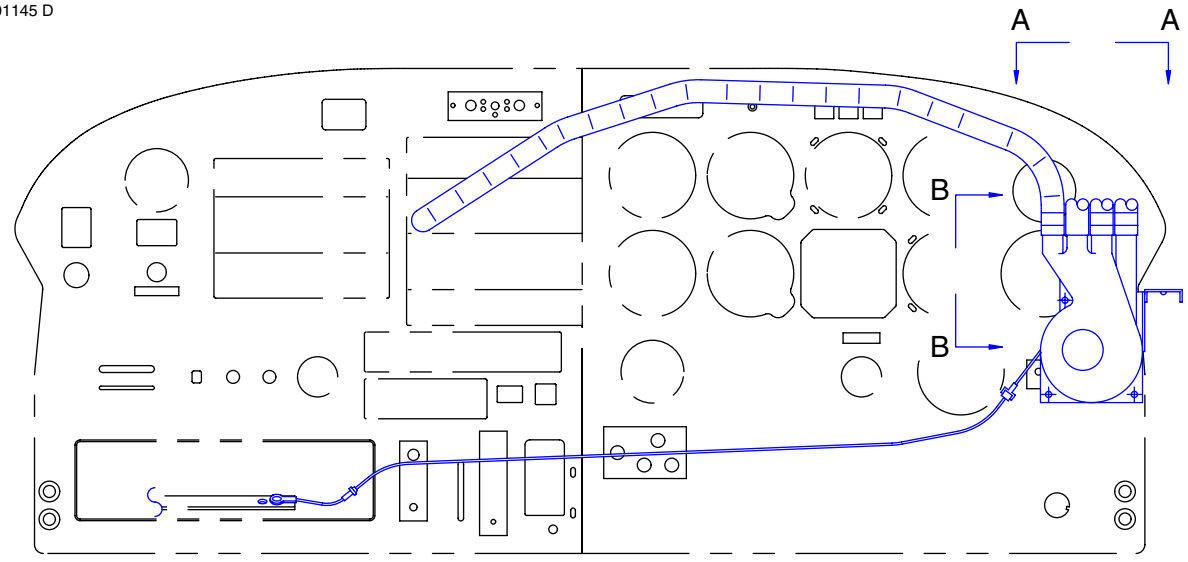
(2) Installation

(a) Attach the blower assembly to the mounting bracket with the screws, washers and nuts.

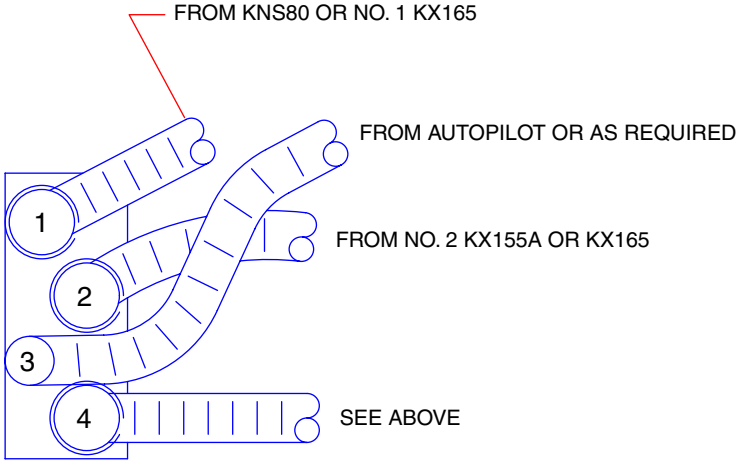
(b) Reconnect the blower motor wires.

(c) Attach the ducts to the tops of the blower assembly. Ensure that the duct ends overlap the ports on top of the blower one-half inch minimum. Secure with Ty-Raps.

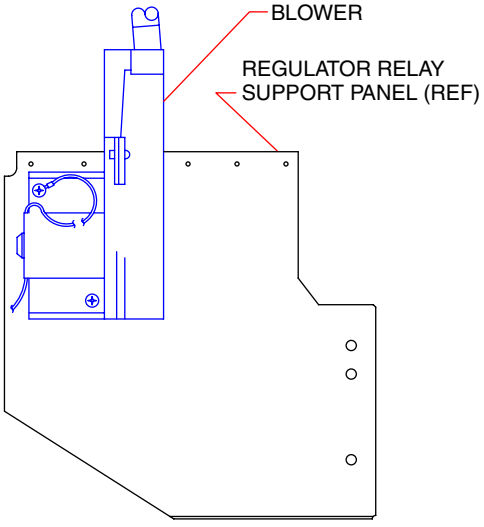
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INSTRUMENT PANEL-VIEW LOOKING AFT



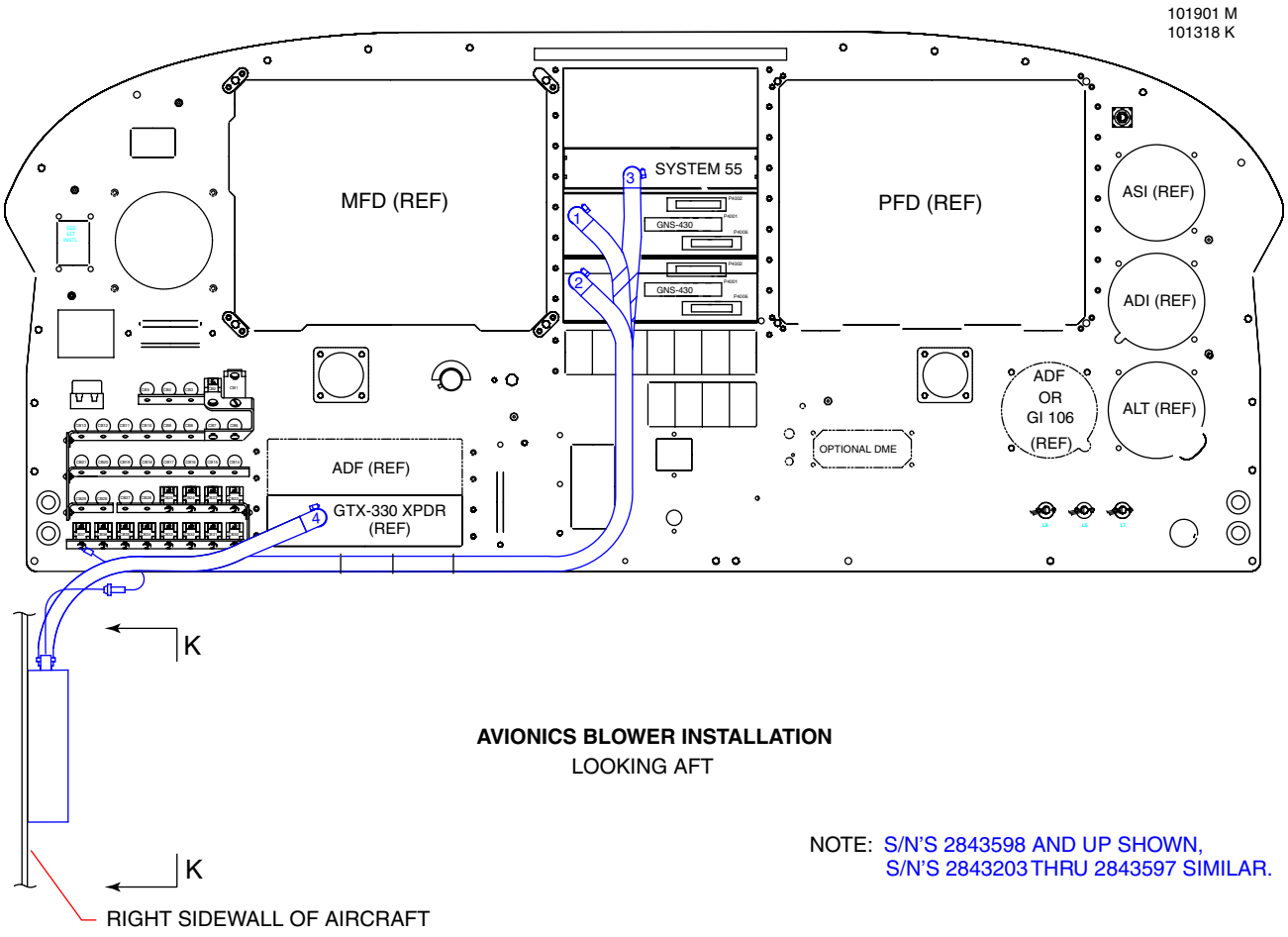
VIEW A - A



VIEW B - B

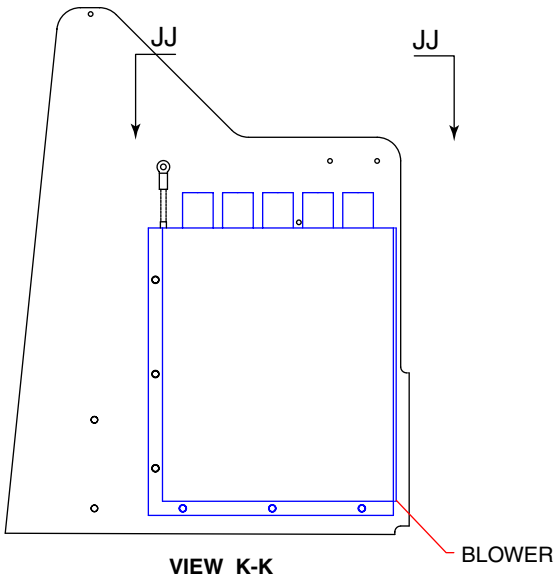
Effectivity
2843001 thru 2843202

Avionics Cooling Blower
Figure 2 (Sheet 1 of 2)

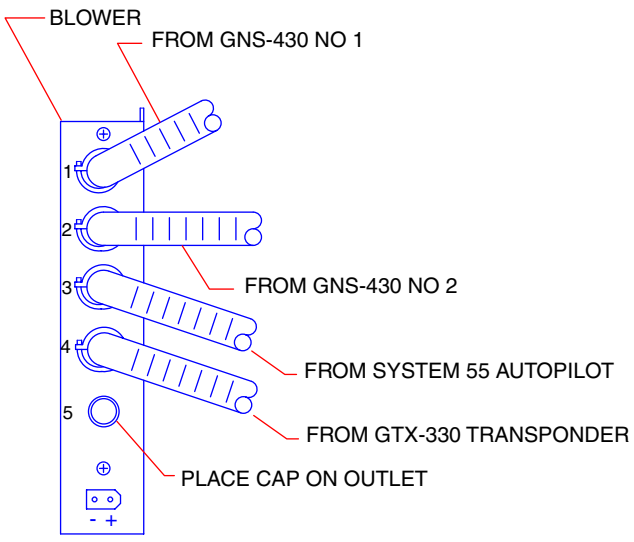


AVIONICS BLOWER INSTALLATION
LOOKING AFT

NOTE: S/N'S 2843598 AND UP SHOWN,
S/N'S 2843203 THRU 2843597 SIMILAR.



VIEW K-K
AVIONICS BLOWER MOUNTING



VIEW JJ-JJ
ROTATED 90° CCW

Avionics Cooling Blower
Figure 2 (Sheet 2 of 2)

Effectivity
2843203 thru 2843700
and 2843702

B. Garmin G1000 and G1000 NXi Installations

In S/N's 2843701, 2843703 and up, an avionics cooling blower is mounted to a bracket attached to the right fuselage side-panel just forward of (i.e., behind) the instrument panel. In addition, two cooling fans are mounted in the avionics equipment racks attached to the instrument panel. See "Figure 3" on page 21207, "Figure 4" on page 21208, and "Figure 5" on page 21209.

NOTE: Actual equipment served will vary depending on optional equipment installed.

(1) Avionics Cooling Blower

(a) Removal

1) Remove ducts from top of blower ports. (Ducts are secured to the blower by Ty-Raps and secured to the ports by hose clamps.)

2) Disconnect blower motor wires at connector.

3) Remove screws, washers and nuts that attach the blower assembly to its mounting bracket.

(b) Installation

1) Attach the blower assembly to the mounting bracket with the screws, washers and nuts.

2) Reconnect the blower motor wires.

3) Attach the ducts to the tops of the blower assembly. Ensure that the duct ends overlap the ports on top of the blower one-half inch minimum. Secure to blower ports by hose clamps and to the blower assembly by Ty-Raps.

(2) Rack-mounted Cooling Fans

(a) Removal

1) Disconnect fan wires at connector.

2) Remove screws (6 ea.) that a secure fan rack to the avionics equipment mounting rack.

3) Remove fan rack and fan.

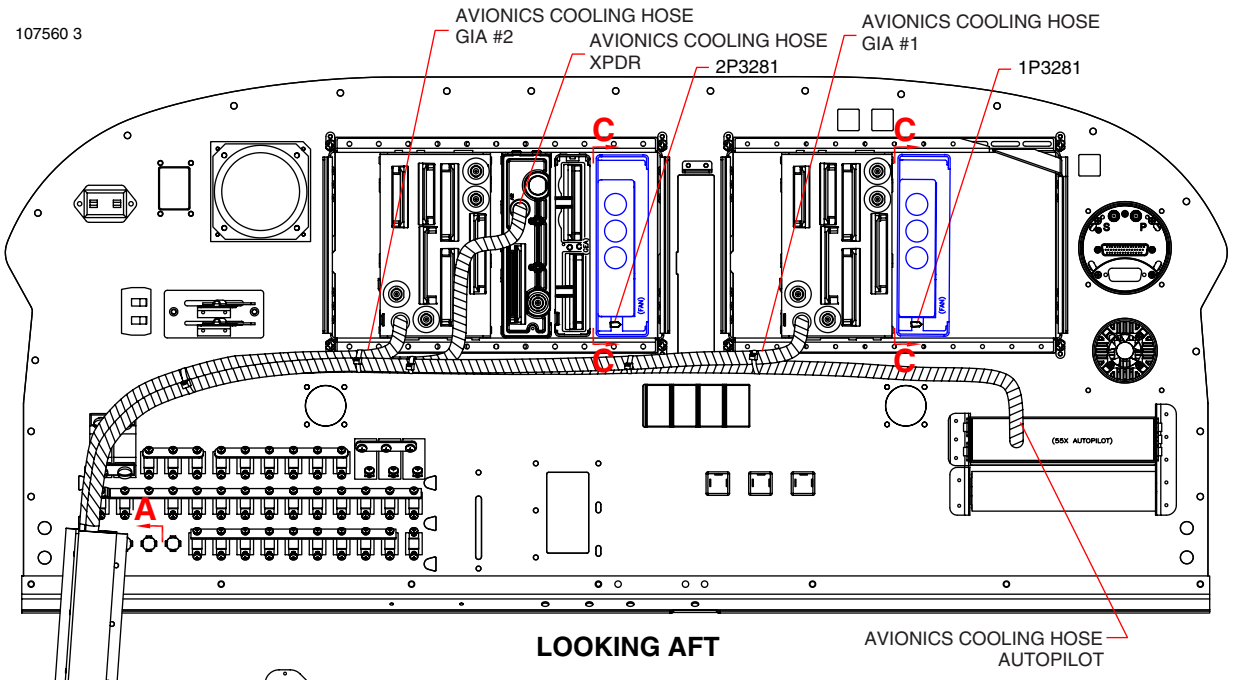
4) Remove screws (3 ea.) that secure fan to rack and remove fan.

(b) Installation

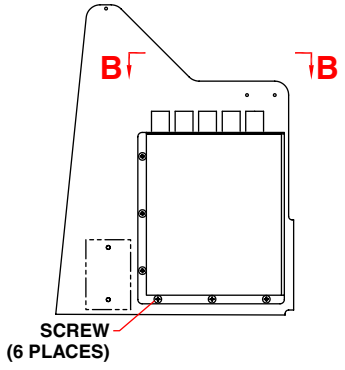
1) Insert fan into fan rack and secure with screws (3 ea.)

2) Insert fan rack into avionics equipment mounting rack and secure with screws (6 ea.).

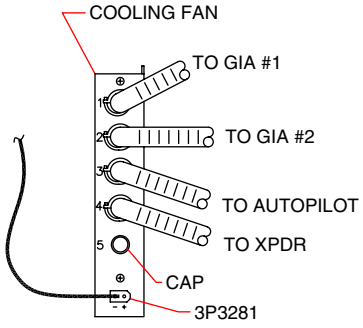
3) Connect fan wires to fan.



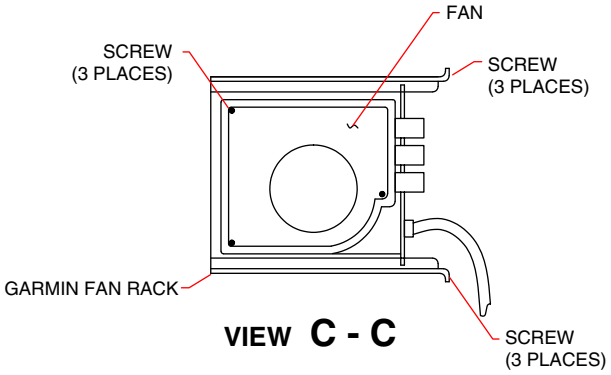
LOOKING AFT



VIEW A - A



VIEW B - B
ROTATED 90° CLOCKWISE

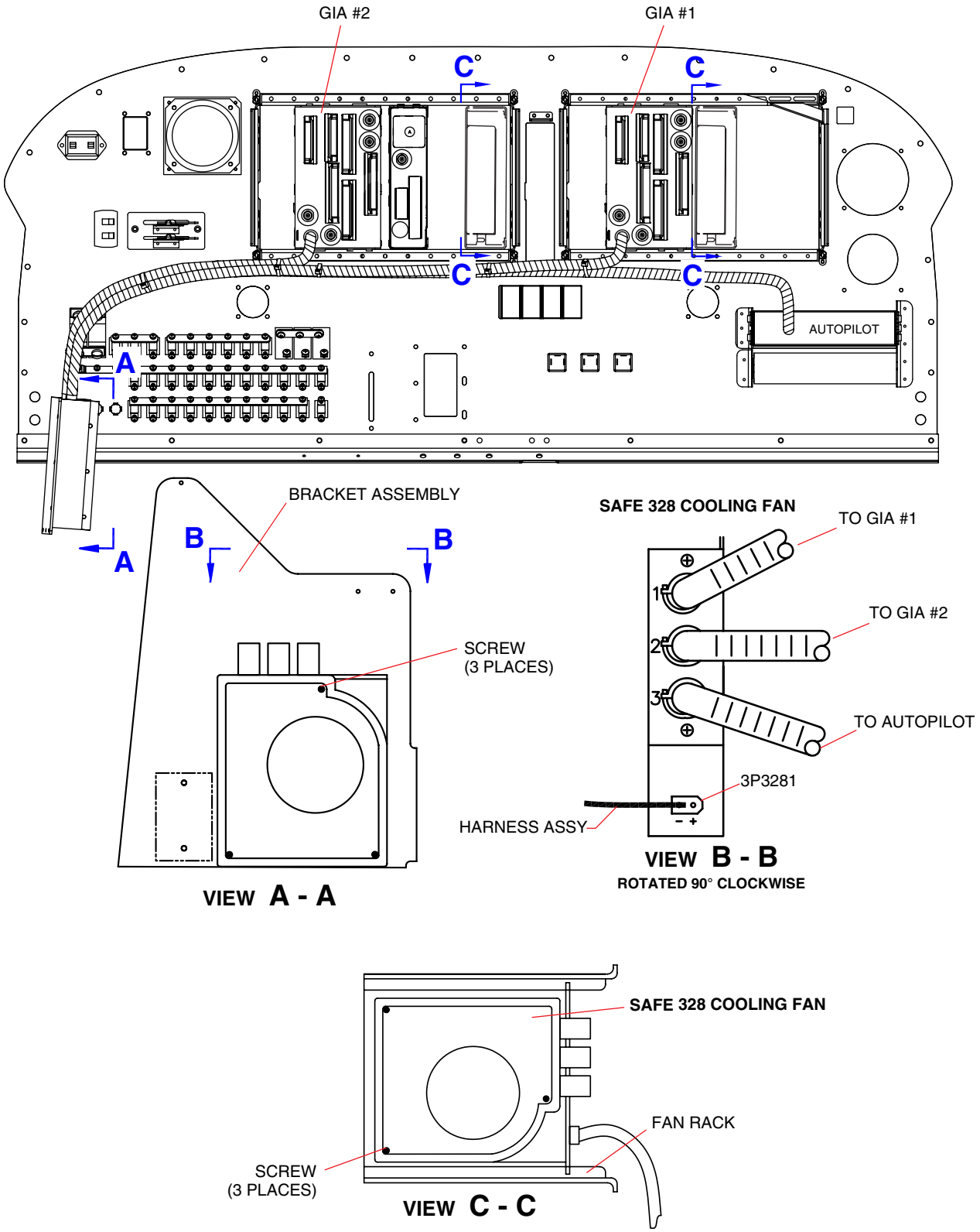


VIEW C - C

Avionics Cooling
Figure 3

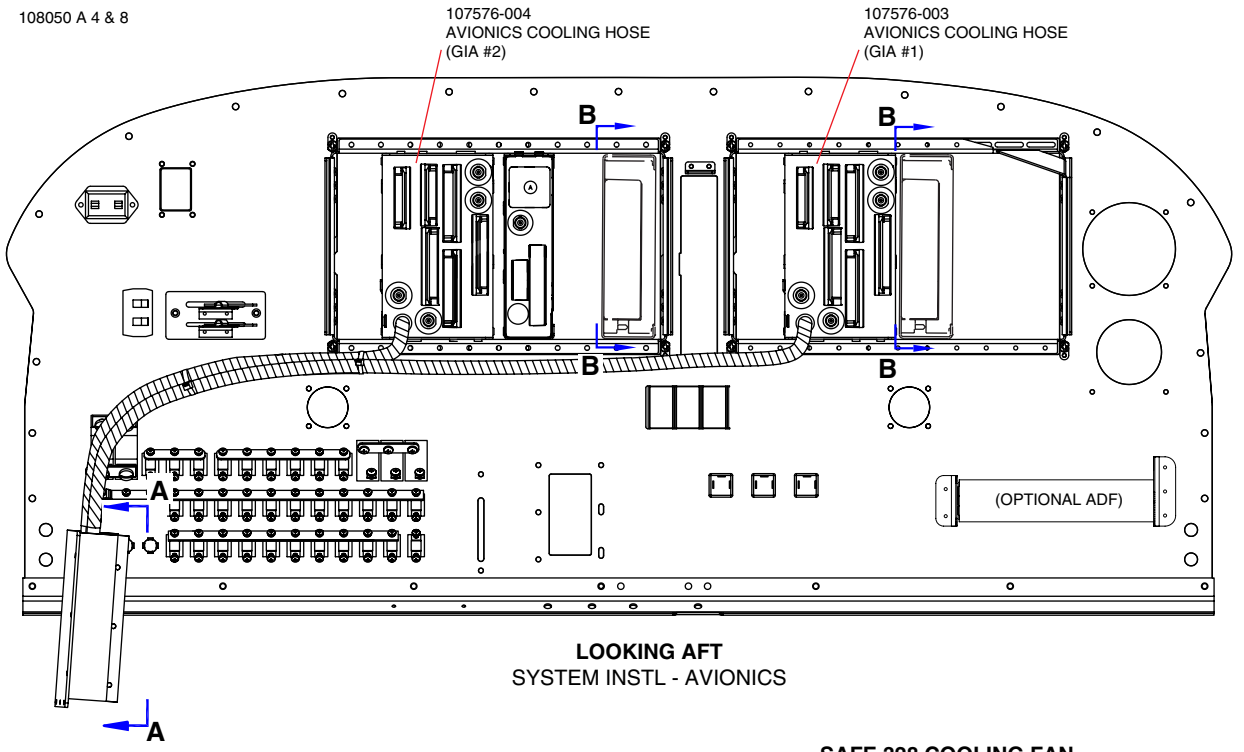
Effectivity
with Garmin G1000

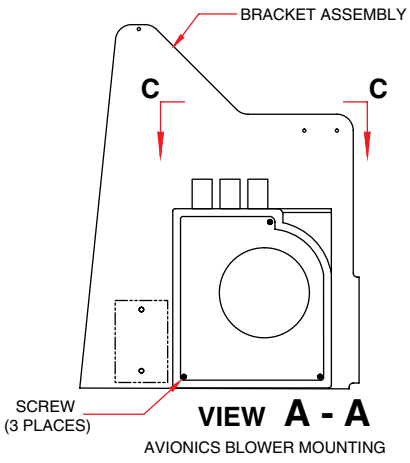
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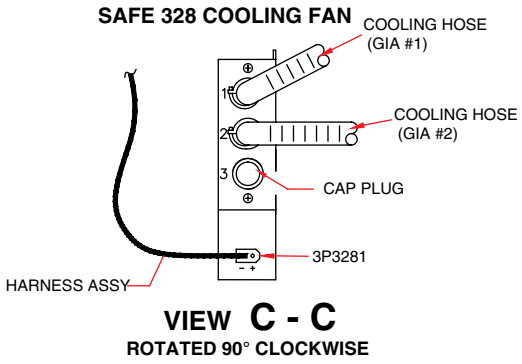


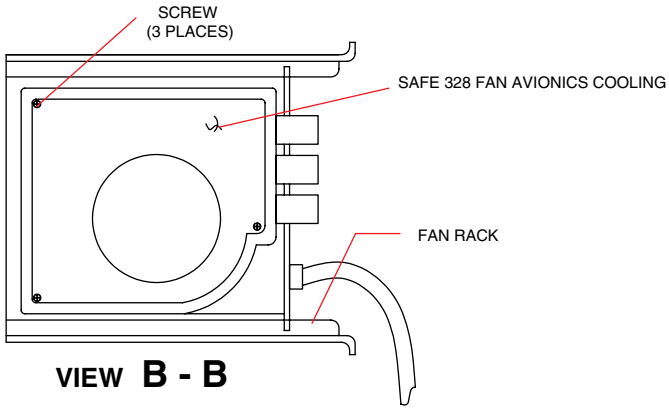
Effectivity
with Garmin G1000 NXi
2843852, 2843854–2843999

Avionics Cooling
Figure 4









Avionics Cooling
Figure 5

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HEATING

1. Description and Operation

See “Figure 1”.

CAUTION: HEAT DUCT SURFACE BECOMES HOT WHEN CABIN HEAT IS OPERATING. BURNS CAN RESULT IF ARMS OR LEGS ARE PLACED TOO CLOSE TO HEAT DUCT OUTLETS OR CONTACT SURFACE.

Fresh air is ducted from the heating intake vent located on the left front of the lower cowling to the heater shroud which is attached to the muffler. The heated air is then ducted to the valve box mounted on the firewall. When the valve is open, heated air enters the heat ducts located along each side of the center console. Outlets in the heat ducts are located at each seat location. Airflow to the rear seats can be regulated by controls in the heat ducts located between the front seats. The temperature of the cabin is regulated by the heater control located on the right side of the instrument panel.

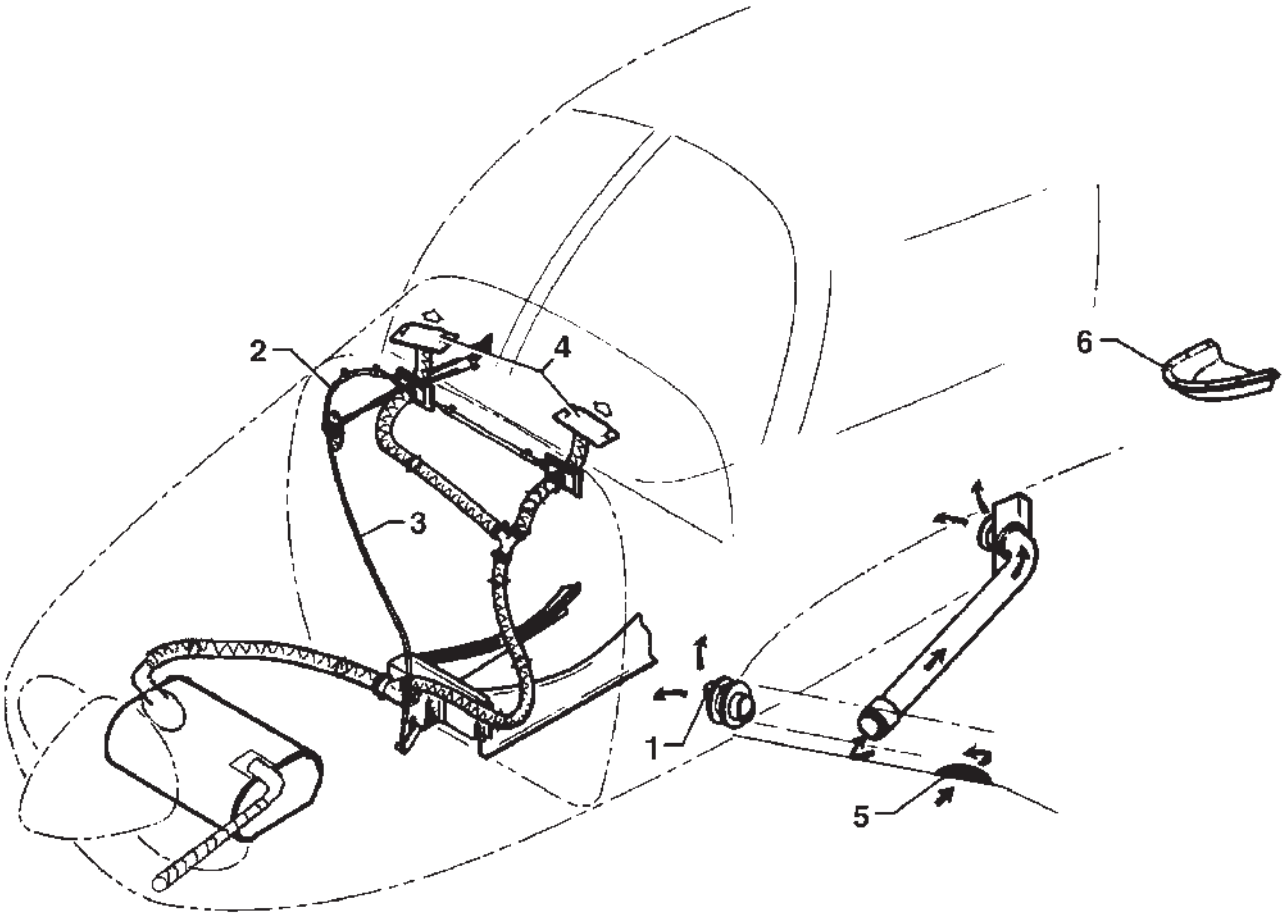
Defrosting is accomplished by heat outlets located on the right and left side of the cowl cover. Heated air is ducted directly from the heater valve box to the defroster shut-off valves at the firewall and then to the defroster outlets. The airflow is regulated by a defroster control located below the heat control.

To aid air distribution, the cabin air is exhausted overboard by an outlet located on the bottom of the fuselage. Cabin exhaust outlets are located below and outboard of the rear seats.

2. Heater Maintenance

WARNING: INSPECT EXHAUST MANIFOLD REGULARLY. DEFECTIVE EXHAUST MANIFOLD WILL CAUSE CARBON MONOXIDE FUMES IN CABIN AREA. (SEE 78-00-00.)

If the exhaust manifold should become defective, carbon monoxide fumes may be discharged into the cabin area. Therefore it is imperative that the exhaust manifold be inspected regularly. Refer to 78-00-00 for the inspection procedure. The heater shroud must be removed in order to inspect the manifold assembly. Check the operation of the push-pull controls to ensure the valve doors function properly. When the controls are pulled out, the door should be completely open to permit full airflow. When the controls are pushed in, the valves should close off all air passage and vent the air into the engine compartment.



FRESH AIR

CABIN HEAT

1. FRESH AIR CONTROL
2. DEFROSTER CONTROL
3. HEATER CONTROL
4. DEFROSTER AIR OUTLET
5. FRESH AIR INLET
6. CABIN EXHAUST OUTLET

Cabin Heater, Defrosters, and Fresh Air System
Figure 1

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COOLING

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Fresh Air (Standard)

   See 21-20-00.

   When the optional air conditioning system is not installed cabin ventilation and cooling is provided solely by fresh air through the overhead and cabin floor vents.

2. Air Conditioning (Optional)

   A. Description and Operation

      Refer to “Figure 1”.

      WARNING: WHEN SERVICING THE AIR CONDITIONING SYSTEM, BE SURE TO IDENTIFY THE SPECIFIC REFRIGERANT, LUBRICANT, AND COMPONENTS USED IN THE PARTICULAR INSTALLATION. THE AIR CONDITIONING SYSTEM INSTALLED IN THESE AIRPLANES USES HFC-134A REFRIGERANT.

      NOTE: Operate air conditioning system at least once a month to keep system lubricated and prevent sticking valves.

      NOTE: “HFC-134a” means air conditioning refrigerant which is generically identified and documented as 1,1,1,2-Tetrafluoroethane and/or CAS# 811-97-2. Brand names and commercial designations which meet these requirements include (but are not limited to):

      HFC134a, HFA134a, R134a, Dymel® 134a, Fluorocarbon 134a, Forane® 134a, Genetron® 134a, Halocarbon 134a, KLEA® 134a, Norfluane, Referon® 134a, and SUVA™ 134a.

      This installation consists of a compressor with its special bracketry, and an evaporator, condenser, receiver-dehydrator, circulating fan, thermal expansion valve, and related plumbing.

      The evaporator filters, dehumidifies, and cools air. The evaporator is mounted in a fabricated housing with the receiver-dehydrator, circulating fan, thermal expansion valve, and related plumbing. The housing is in the rear cabin, aft of baggage area closeout panel. There are two types of air conditioning compressors used. The T/CCI (York) two cylinder or the Sanden five cylinder compressor. Both are piston type supported by special bracketry at engine front. A V-belt connected to engine ring gear drives the compressor through a magnetic clutch. The condenser is on a hinge mounted door, on the bottom of the fuselage tail section. The condenser door is hinge mounted to allow extension into airstream during system operation. The condenser door is electrically activated to the system. When ON it is fully extended. When the system is OFF the door is fully retracted.

      A Ranco type pressure switch protects the system and automatically controls condenser maximum head pressure by temporarily declutching the compressor if pressure becomes excessively high. The air conditioning control switch, a fan control switch to govern cold air velocity, and a temperature control are on aircraft instrument panel adjacent to heater and defroster levers.

      System design does not increase aircraft drag during take-off. During maximum power demands the compressor is de-clutched and the condenser door automatically retracts.

The air conditioning system is a recirculating, independent unit. It filters, dehumidifies, and cools air as air cycles through evaporator. The unit operates from controls mounted on right side instrument panel. The air conditioning master switch has two positions, ON-OFF. If AIR COND position is selected, the compressor clutch engages, the condenser scoop opens, and the circulating fan is turned on. Temperature is controlled by temperature control selector thermostat. A three position fan switch (LOW-OFF-HIGH) operates the blower. The fan may be operated to circulate air without using air conditioning unit.

The air conditioning system installed in Archer III airplanes uses HFC-134a refrigerant. Refrigerant enters the compressor as a vapor. The compressor pressurizes the heat laden vapor until the vapor temperature becomes warmer than the outside air temperature. The compressor then pumps the vapor to the condenser where the refrigerant is cooled and changes to liquid. The liquid now passes to the receiver-dehydrator. The receiver-dehydrator filters, removes moisture, and ensures a steady flow of liquid refrigerant into the evaporator through the expansion valve. The expansion valve is a temperature controlled metering valve which regulates the flow of liquid refrigerant to evaporator. The evaporator enables the liquid refrigerant to absorb heat from the outside air passing over coils, converting it back to a vapor. From the evaporator, heat laden refrigerant in a vapor state returns to compressor, and the cycle repeats.

B. Troubleshooting

Troubles peculiar to air conditioning system components are listed in “Chart 1”, with probable causes, and suggested remedies. Correct trouble and check entire system for security and components operation. See definitions under Servicing Cooling System, below.

CAUTION: UNITED STATES ENVIRONMENTAL REGULATIONS REQUIRE USE OF A COLLECTION SYSTEM WHEN EVACUATING REFRIGERANT FROM AIR CONDITIONER.

NOTE: Check all environmental regulations for your local area before servicing air conditioning system.

CHART 1 (Sheet 1 of 4)
TROUBLESHOOTING AIR CONDITIONING SYSTEM

Trouble	Cause	Remedy
High discharge pressure.	Refrigerant overcharge.	Purge excess refrigerant.
	Air in system.	Check for leaks. Bleed charge from system. Evacuate and recharge system.
	Overheated condenser due to blocked air passage.	Clean bugs and dirt from condenser fins. Straighten bent fins.
	Flooded evaporator indicated by heavy frosting on suction line and compressor suction service valve.	Check capillary bulb is securely clamped to suction line. If capillary bulb is OK, replace expansion valve.
	Restriction in liquid line from condenser.	Check for kinked hoses and clogged filter.
Low discharge pressure.	Refrigerant undercharge.	Check system for leaks. Evacuate and recharge system.
	Damaged compressor valves or dirt under valves.	Replace compressor.
	Damaged compressor. Worn or broken piston or piston rings.	Replace compressor.
Low suction pressure accompanied by icing of evaporator.	Low air supply through evaporator.	Repair blower or blower motor. Clean stoppage in air ducts.
	Very dirty evaporator fins and coils.	Clean and flush with water.
Low suction pressure. (Evaporator not cold enough.) Suction gauge reads vacuum indicating evaporator lacks refrigerant.	Refrigerant undercharge. Moisture freezing in expansion valve. Valve shows frost.	Add refrigerant. Install new dryer. Evacuate and recharge.
clogged. Inoperative expansion	Expansion valve inlet screen	Remove screen. Clean with solvent and replace.
	valve. Valve stuck closed or capillary bulb has lost charge.	Warm capillary by holding in hand. If suction pressure does not change, replace expansion valve.
	Restriction in liquid line. Restriction will show frost.	Locate restriction and repair.

CHART 1 (Sheet 2 of 4)
TROUBLESHOOTING AIR CONDITIONING SYSTEM

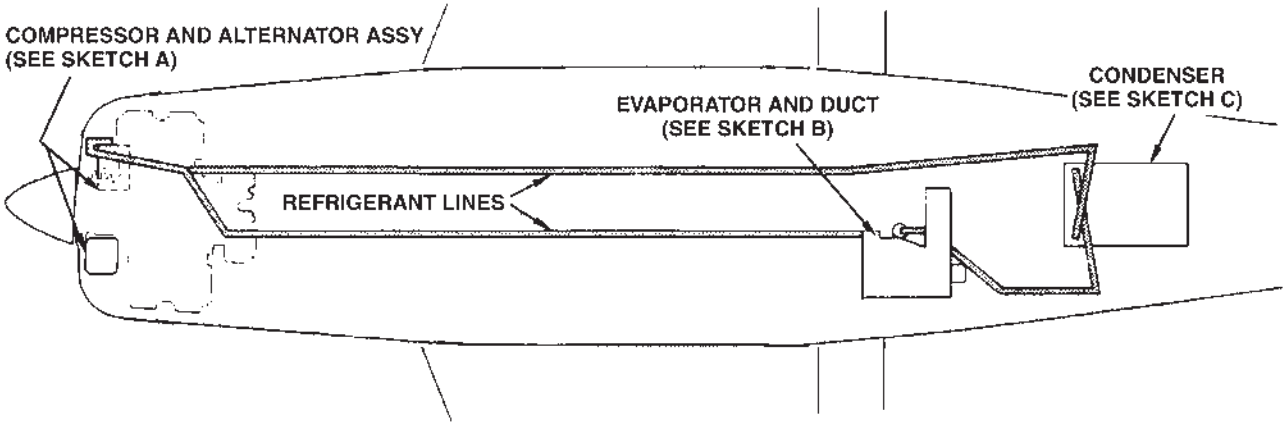
Trouble	Cause	Remedy
High suction pressure.	Capillary bulb clamp loose on suction line. Suction line shows frost.	Clean contact surfaces of suction line and cap bulb. Tighten clamp.
	Expansion valve not closing. Evaporator flooded. Suction line frosted to compressor.	Replace expansion valve.
	Compressor drive belt slipping.	Adjust belt tension.
	Magnetic clutch slipping.	Check electrical circuit for proper voltage to clutch coil. Clean oily clutch surfaces.
	Leaking or broken compressor valves.	Replace compressor.
Condenser door will not close when air conditioner switch is in OFF position.	Faulty K-2 relay.	Replace relay.
System does not cool.	**If electrical:**	
	Blown fuse in control head.	Replace fuse.
	Open circuit breaker.	Set circuit breaker.
	Broken or disconnected electrical wire.	Check all terminals for loose connections. Check wiring for hidden breaks.
	Broken or disconnected ground wire.	Check ground wire is not loose, broken, or disconnected.
	Clutch coil burned out or disconnected.	Verify voltage to clutch. Replace if inoperative.
	Thermostat sensing element defective.	Check thermostat and cabin comfort control panel.
	Blower motor disconnected or burned out.	Verify voltage to blower motor. Repair or replace if inoperative.
	If mechanical:	
	Loose or broken drive belt.	Replace drive belts and tighten to specifications.
	Compressor partially or completely frozen.	Remove compressor. Service or replace.
	Expansion valve stuck in open position.	Replace expansion valve.

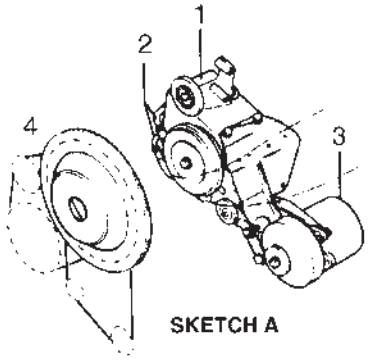
CHART 1 (Sheet 3 of 4)
TROUBLESHOOTING AIR CONDITIONING SYSTEM

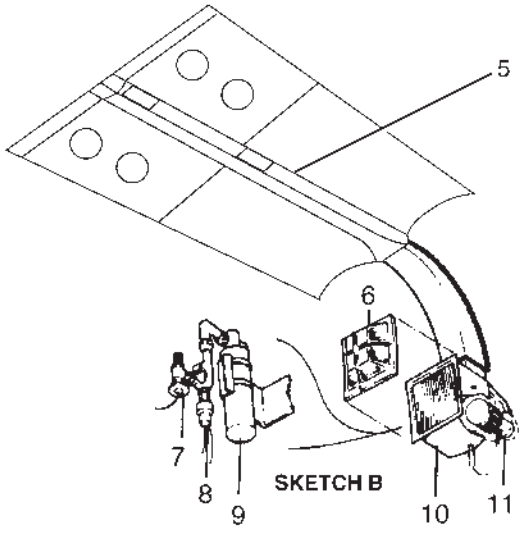
Trouble	Cause	Remedy
System does not cool. (cont.)	**If refrigeration:**	
	Broken refrigerant line.	Examine all lines for evidence of breakage by external stress or rubbing wear.
	Leak in system.	Evacuate system, apply static charge, leak test system, and repair leak as necessary.
	Compressor shaft seal leaking.	Replace compressor.
	Clogged screen or screens in receiver-dehydrator or expansion valve; plugged hose or coil.	Repair as necessary.
System cooling inadequate.	**If electrical:**	
	Blower motor operation sluggish.	Remove blower motor for service or replacement.
	If mechanical:	
	Compressor clutch slipping.	Remove clutch assembly for service or replacement. Check clutch airgap and coil.
	Obstructed blower passage.	Examine passage for obstruction. Correct as necessary.
	Insufficient air circulation over condenser coils; fins clogged with dirt or bugs.	Clean condenser coils.
	Clogged evaporator filter.	Clean with solvent.
	If refrigeration:	
	System refrigerant low.	Recharge system until a minimum 20°F temperature drop is achieved at the evaporator during the Post Charging Operational Check and gauge readings stabilize to specifications.
	Clogged screen in expansion valve.	Purge system, replace expansion valve.
	Expansion valve thermal bulb has no charge.	Purge system, replace expansion valve.

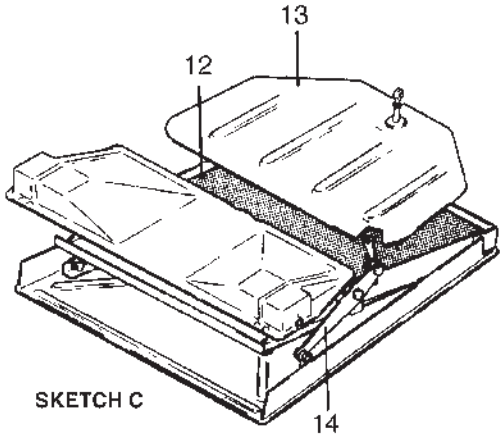
CHART 1 (Sheet 4 of 4)
TROUBLESHOOTING AIR CONDITIONING SYSTEM

Trouble	Cause	Remedy
System cooling inadequate. (cont.)	**If refrigeration:** (cont.)	
	Clogged receiver-dehydrator screen.	Purge system, replace receiver-dehydrator.
	Excessive moisture in system.	Purge system, replace receiver-dehydrator.
	Air in system.	Purge, evacuate, and charge system. (Replace receiver-dehydrator.)
Excessively noisy system.	**If electrical:**	
	Defective winding or connection in compressor clutch coil.	Replace or repair as necessary.
	If mechanical:	
	Loose or worn drive belts, crankshaft pulley, or idler pulley or bearing.	Tighten or replace as required.
	Engine components such as: alternator, water pump, valves, timing or mounts.	Check.
	Compressor mounting bolts or brackets - broken or loose.	Check, repair, replace.
	Compressor oil level low.	Fill with proper amount of specified oil.
	Compressor failure.	Check shaft turning smoothness. Remove compressor for service or replacement.
	Magnetic clutch failure.	Check airgap, clutch pulley, front plate, coil, and bearing. Adjust, repair, or replace, as required.
	If refrigeration:	
	Excessive system charge.	Remove excess refrigerant until high pressure gauge drops within specifications.
	Low system charge.	Check system for leaks. Recharge system.
	Excessive moisture in system.	Replace receiver-dehydrator; purge, evacuate, and recharge system.









1. COMPRESSOR
2. REFRIGERANT LINES
3. ALTERNATOR
4. PULLEY ARRANGEMENT
5. DUCT ASSEMBLY
6. EVAPORATOR FILTER AND COVER
7. EXPANSION VALVE
8. PRESSURE SWITCH
9. RECEIVER-DEHYDRATOR
10. EVAPORATOR
11. BLOWER MOTOR
12. CONDENSOR
13. COVER ASSEMBLY
14. BELLCRANK ASSEMBLY

Air Conditioning Installation
Figure 1

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3. Servicing Cooling System

(PIR-PPS50003-3, Rev. H / PPS50003-5 Rev. B.)

NOTE: The maximum refrigerant capacity is 2.5 pounds. The total refrigerant capacity required is determined separately for each system and is the amount that will result in a minimum 20° F temperature drop at the evaporator, as specified in the post charging operational check.

A. Definitions

(1) High Side. The “high side” consists of all lines and components between the compressor outlet and the expansion valves. It includes the condenser and receiver-dehydrator sight gauge.

(2) Low Side. The “low side” consists of all lines and components between the expansion valves and the compressor inlet. It includes the evaporator.

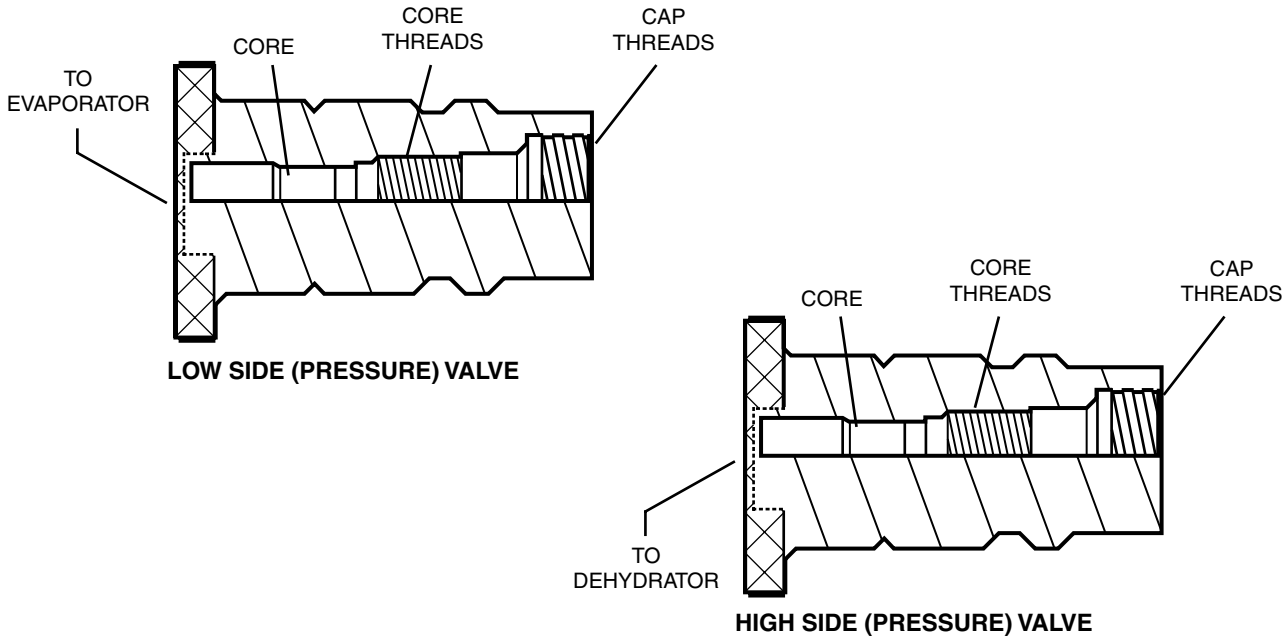
B. Service Valves

See “Figure 2”.

CAUTION: IF AIR CONDITIONING REFRIGERANT LINES OR SYSTEM IS OPENED, LINES AND FITTINGS MUST BE CAPPED AND SEALED IMMEDIATELY TO PREVENT DIRT AND OTHER CONTAMINANTS FROM ENTERING THE SYSTEM. (DO NOT PUT A PLUG INTO THE HOSES OR FITTINGS.)

Use inline service valves for all normal air conditioning service (i.e. - testing, bleeding, evacuating, and charging). The aircraft is equipped with inline service valves mounted in the suction and discharge lines of the evaporator assembly located behind the cabin rear closeout panel. These inline service valves are quick disconnect type Schrader valves.

NOTE: If a Schrader service valve is not serviceable, the core assembly must be replaced.



Service Valves
Figure 2

C. Malfunction Detection

NOTE: If the cooling system has leaked refrigerant or is discharged, the compressor oil level must be checked.

Detection of system malfunction largely depends on the mechanic's ability to interpret gauge pressure readings into system problems. A system operating normally will have low side gauge pressure reading that will correspond with the temperature of the refrigerant evaporating, allowing for a few degrees temperature rise due to loss in tube walls and fins. The high side will have a gauge pressure that will corresponds with the temperature of the refrigerant condensing in the condenser, allowing for a few degrees temperature drop due to loss in tube walls and fins.

Any deviation from that which is normal indicates a malfunction within the system due to faulty control device, obstruction, defective part, or improper installation.

Detection of system malfunction is made easier with knowledge of the relationship between temperature and pressure of refrigerant HFC-134a. Refer to "Chart 2" for specific values.

NOTE: Gauge readings are about one inch mercury or 1/2 psi higher than chart reads for each 1000 feet elevation above sea level.

Actual air temperature of air passing over the evaporator coils will be several degrees warmer allowing for a temperature rise caused by the loss in the fins and tubing of the evaporator.

The importance of a seasonal check up of the air conditioning system should be brought to the attention of the customer whenever possible. A thorough check of the system performed in a methodical manner will reveal trouble the customer is often not aware of. Locating and repairing the trouble early will usually result in savings to the customer both in time and additional troubles that too often result from neglect.

A performance test of the system is the only positive way in which the complete system can be checked for efficient operation. The air conditioning system should be given this test before work is begun on the system whenever possible. However, if the system is completely inoperative, repairs must be performed before the system can be properly tested. The test can uncover further work that must be performed before the system is brought to its full operating efficiency. The performance test should always be performed after repair work has been done and before the airplane is released to the customer. The serviceman performing this test carefully will ensure that the repairs have been properly performed and that the system will operate satisfactorily.

NOTE: The air conditioning system should be operated at least once a month to prevent sticking valves and to keep the system lubricated.

The performance test, when properly performed includes a thorough examination of the outside of the system as well as the inside. Many related parts are overlooked because it is felt they are of no bearing on the operating efficiency of the unit. For this reason, a thorough visual inspection of the complete system should be performed, followed by an operating inspection of the system.

CHART 2
TEMPERATURE VS. PRESSURE

Refrigerant HFC134a Evaporator Pressure Gauge Reading psi	Refrigerant HFC134a Evaporatore Temperature °F
-5	-27
0	-15
2	-9
4	-4
6	0
8	4
10	7
12	11
14	14
16	17
18	20
20	22
22	25
24	28
26	30
28	33
30	35
32	37
34	39
36	41
38	43
40	45
42	47
44	49
46	51
48	53
50	54
55	58
60	62
65	66
70	69

D. Special Servicing Procedures

The air conditioning system should be serviced by a qualified shop with trained personnel. The following procedures and precautions should be observed.

CAUTION: UNITED STATES ENVIRONMENTAL REGULATIONS REQUIRE THAT AIR CONDITIONING SYSTEM REPAIRS BE ACCOMPLISHED BY A QUALIFIED SHOP WITH APPROPRIATELY TRAINED PERSONNEL.

The efficiency of this system depends upon the pressure-temperature relationship of pure refrigerant. As long as the system contains only pure refrigerant plus a specified amount of compressor oil (which is mixed with the refrigerant), it is considered to be chemically stable. Foreign materials within the system will affect the chemical stability, contaminate the system, and decrease its efficiency.

(1) Refrigerant Safety Precautions:

WARNING: REFRIGERANT HFC-134A IS ODORLESS AND COLORLESS IN EITHER THE LIQUID OR GASEOUS STATE. REFRIGERANT FOR CHARGING REFRIGERATION SYSTEMS IS SUPPLIED IN PRESSURIZED CONTAINERS (APPROX. 70 PSI AT 70°F) IN LIQUID FORM. SINCE THIS MATERIAL IS ESSENTIALLY INERT AT ROOM TEMPERATURES THE DANGERS ARE PRIMARILY ASSOCIATED WITH THE PRESSURE AND THE REFRIGERATION EFFECTS OF THE RELEASE AND SUBSEQUENT EVAPORATION OF THIS PRESSURIZED LIQUID.

WARNING: WEAR SUITABLE EYE PROTECTION WHEN HANDLING REFRIGERANT DUE TO THE POSSIBILITY OF FREEZING OF THE EYE IF CONTACTED BY ESCAPING LIQUID REFRIGERANT. IF LIQUID REFRIGERANT DOES STRIKE THE EYE, THE FOLLOWING ACTIONS SHOULD BE TAKEN:

1 DO NOT RUB THE EYE.
2 Splash large quantities of cool water into the eye to raise the temperature.
3 Tape on an eye patch to avoid the possibility of dirt entering the eye.
4 Rush to a physician or hospital for immediate professional aid.
5 DO NOT ATTEMPT TO TREAT IT YOURSELF.

WARNING: WEAR SUITABLE CLOTHING AND GLOVES. IF LIQUID REFRIGERANT STRIKES THE SKIN, FROSTBITE CAN OCCUR. IF EXPOSED, TREAT WITH WARM (NOT HOT) WATER AND SEEK MEDICAL ATTENTION.

WARNING: DO NOT DISCHARGE LARGE QUANTITIES OF REFRIGERANT INTO CLOSED ROOMS. IT MAY DISPLACE MOST OF THE AIR IN THE ROOM AND THIS COULD CAUSE OXYGEN STARVATION. GASEOUS REFRIGERANT IS HEAVIER THAN AIR AND FLOWS TO THE BOTTOM OF A CONTAINER.

WARNING: DO NOT APPLY DIRECT FLAME OR OTHER HIGH HEAT SOURCE TO A REFRIGERANT CONTAINER DUE TO THE HIGH PRESSURES WHICH WILL RESULT. IF ANY HEATING IS DONE TO REFRIGERANT CONTAINERS THE CONTAINER PRESSURE SHOULD BE MONITORED AND KEPT BELOW 150 PSI.

(2) System Servicing Precautions:

WARNING: DISCHARGE SYSTEMS SLOWLY TO PREVENT THE ESCAPE OF LIQUID REFRIGERANT AND THE LOSS OF THE LUBRICATING OIL.

NOTE: The term “Discharge,” as used throughout this section, in no sense implies or suggests discharging refrigerant to atmosphere. In all cases when discharging, an environmentally approved refrigerant recovery station is to be used.

WARNING: DO NOT LEAVE SYSTEMS OPEN TO THE ATMOSPHERE WHEN DISCHARGED. MOISTURE AND OTHER CONTAMINATION MAY ENTER AND DAMAGE OPEN SYSTEMS.

NOTE: When HFC-134a comes in contact with moisture it absorbs it into the system, which will lead to system failure.

WARNING: USE ONLY APPROVED REFRIGERATION OIL IN THE COMPRESSOR:

-- POLYALKYLENE GLYCOL (PAG) OR POLYOL ESTER (POE) FOR HFC-134A SYSTEMS

IF ANY DOUBT EXISTS ABOUT THE CLEANLINESS OF THE COMPRESSOR OIL, REPLACE IT WITH NEW OIL.

WARNING: NEVER INTRODUCE ANYTHING BUT PURE REFRIGERANT AND THE APPROPRIATE REFRIGERANT OIL INTO A SYSTEM.

WARNING: KEEP REFRIGERANT OIL CONTAINERS TIGHTLY SEALED AND CLEAN TO PREVENT ABSORPTION OF MOISTURE OR OTHER CONTAMINATION.

WARNING: NEVER REUSE OIL REMOVED FROM THE SYSTEM -- DISCARD IT.

CAUTION: WHEN LOCTITE REFRIGERANT SEALANT HAS BEEN USED ON A JOINT IT MUST BE HEATED TO 400°F PRIOR TO DISASSEMBLY. LOCTITE MUST BE USED TO SEAL ANY PIPE THREADS IN THE SYSTEM LINES.

CAUTION: REPLACE THE RECEIVER-DEHYDRATOR ASSEMBLY ON ANY SYSTEM WHICH HAS BEEN OPERATING WITH A LEAK ALLOWING AIR TO ENTER THE SYSTEM. IF A RECEIVER-DEHYDRATOR IS LEFT OPEN TO THE ATMOSPHERE IT SHOULD BE REPLACED DUE TO THE LOSS OF EFFECTIVENESS OF THE DRYING COMPOUND IT CONTAINS.

CAUTION: A NEW RECEIVER-DEHYDRATOR SHOULD BE OPENED AND CONNECTED TO THE SYSTEM ONLY WHEN READY TO CHARGE THE SYSTEM WITH REFRIGERANT.

CAUTION: RECOMMENDED TORQUE VALUES MUST BE USED ON ALL FLARE FITTING AND O-RING JOINTS. SEE “Chart 3”.

CAUTION: IF AIR CONDITIONING REFRIGERANT LINES OR SYSTEM IS OPENED, LINES AND FITTINGS MUST BE CAPPED AND SEALED IMMEDIATELY TO PREVENT DIRT AND OTHER CONTAMINANTS FROM ENTERING THE SYSTEM. (DO NOT PUT A PLUG INTO THE HOSES OR FITTINGS.)

CAUTION: UNITED STATES ENVIRONMENTAL REGULATIONS PROHIBIT THE RELEASE OF REFRIGERANT INTO THE ATMOSPHERE. SPECIAL EQUIPMENT IS REQUIRED WHEN TESTING, DISCHARGING, OR CHARGING THE SYSTEM.

CAUTION: UNITED STATES ENVIRONMENTAL REGULATIONS REQUIRE THAT AIR CONDITIONING SYSTEM REPAIRS BE ACCOMPLISHED BY A QUALIFIED SHOP WITH APPROPRIATELY TRAINED PERSONNEL.

CHART 3
RECOMMENDED TORQUE SPECIFICATIONS

ALUMINUM TUBING		
Metal Tube O.D.	**Thread and Fitting Size**	**Ft./Lb.**
1/4	7/16	5-7
3/8	5/8	11-13
1/2	3/4	15-20
5/8	7/8	21-27
3/4	1-1/16	28-33

FLARE CONNECTIONS			O-RING CONNECTIONS		
Tube OD	**Thread size**	**Ft./Lb.**	**Tube OD**	**Thread size**	**Ft./Lb.**
3/8	5/8	18-20	3/8	5/8	11-13
1/2	3/4	36-39	1/2	3/4	15-20
5/8	7/8	52-57	5/8	7/8	21-27

E. Servicing the System with a Charging Stand

CAUTION: MINERAL OIL AND PAG ARE NOT COMPATIBLE. USE A SEPARATE MANIFOLD TEST SET AND / OR TEST/CHARGING STAND AND RECOVERY SYSTEM FOR EACH REFRIGERANT TYPE.

CAUTION: USE RECOVERY UNIT SPECIFICALLY DESIGNED FOR THE TYPE OF REFRIGERANT USED IN THE AIRCRAFT SYSTEM. UNINTENDED AFFECTS MAY OCCUR IF REFRIGERANTS ARE COMBINED.

(1) Discharging (Bleeding/Purging) the System

CAUTION: APPLIES TO ROBINAIR 34700 OR SIMILAR CHARGING/RECOVERY STATION. IF USING DIFFERENT EQUIPMENT, SEE OPERATOR'S MANUAL OF STATION BEING USED FOR DETAILED INSTRUCTIONS FOR DISCHARGING SYSTEM.

This procedure is for a Robinair 34700 or similar charging stand/recovery station (see "Figure 3" and "Figure 6"). Required only if the system contains refrigerant.

NOTE: The term "Discharge," as used throughout this section, in no sense implies or suggests discharging refrigerant to atmosphere. In all cases when discharging, an environmentally approved refrigerant recovery station is to be used.

(a) Gain access to service valves by removing rear closeout panel in cabin.

(b) Remove protective caps from service valves.

(c) Connect high side (red) hose to air conditioner high side service valve. On systems equipped with quick disconnect connections, open coupler valve.

(d) Connect low side (blue) hose to air conditioner low side service valve. On systems equipped with quick disconnect connections, open coupler valve.

(e) Check the low side gauge (GAUGE 1) and high side gauge (GAUGE 2) to determine that there is pressure in the system. If there is no pressure, there is no refrigerant in the system to recover.

(f) Check that the oil drain valve is closed.

(g) Open both the low side and high side valves on control panel.

(h) Open the red GAS (vapor) valve and the blue LIQUID valve on the charging station's refrigerant tank.

(i) Slowly open the oil drain valve to see if system oil separator contains oil. If it does, let oil drain into the oil drain bottle (located at the bottom of the rear side of the charging station) until separator is empty.

(j) Close the oil drain valve. Dispose of collected oil in an environmentally accepted manner. Return collection bottle to its place on the charging stand.

(k) Plug unit into a proper voltage outlet. Turn MAIN POWER switch ON.

(l) Press the RECOVER key on charging station keypad.

(m) To assure complete recovery of refrigerant:

1) Wait 5 minutes. Observe pressure gauges for a rise above zero.

2) If a rise occurs, press the HOLD/CONT key.

3) Repeat as necessary until system maintains pressure for two minutes.

(n) Slowly open oil drain valve. Drain oil into the oil catch bottle. When all recovered oil has been completely drained, close oil drain valve.

NOTE: Drain oil separator after each job. Display will indicate OIL (OUNCES) or OIL (GRAMS) as a reminder.

(o) Measure the amount of oil in the catch bottle. The same amount of new oil must be added to the system before charging the system.

(p) To enter diagnostic mode, simultaneously press the SHIFT/RESET and ENTER keys. To display amount of refrigerant recovered by the unit, press the 3 key. The panel display will read the amount of recovered refrigerant in pounds or kilograms.

(q) Simultaneously press the SHIFT/RESET and ENTER keys to clear internal counter. Press SHIFT/RESET to return to the main menu.

(2) Leak Detection

Refer to “Figure 3” and “Figure 4”.

When using a charging stand, a leak may be located as follows:

(a) Ensure that aircraft and/or ground power is OFF.

(b) Close all valves on the charging stand.

(c) Remove the protective caps from the high and low side service ports on the evaporator.

(d) Connect the blue and red hoses to the service ports as shown in “Figure 6”.

(e) Proceed following the instructions in either paragraph (f) or (g), below.

(f) Using gaseous dry nitrogen:

1) Remove the pressure switch located on the evaporator assembly.

2) Connect a regulated (0-300 psig) gaseous dry nitrogen source to the pressure switch port on the evaporator assembly.

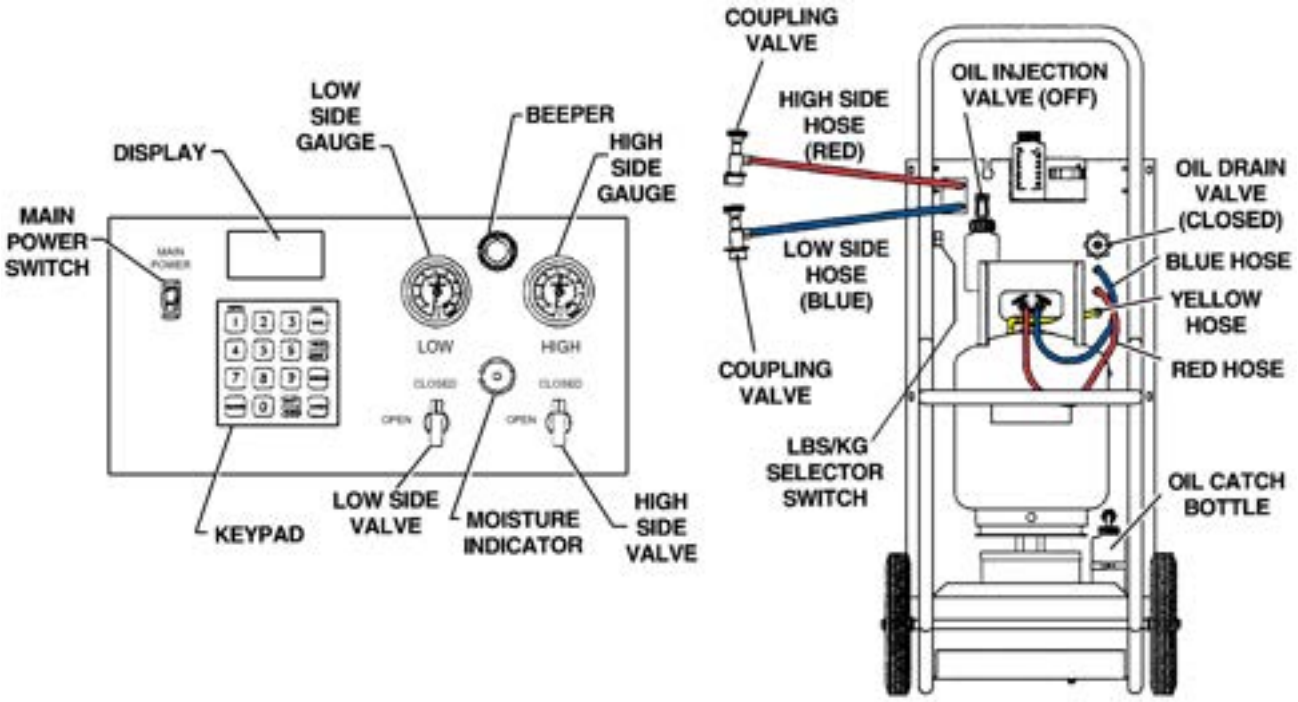
3) Slowly pressurize the system to 200 psig max. with nitrogen and turn off the nitrogen source. Ensure gauges used have a 0-400 psig range.

4) Monitor pressure on the charging station gauge for 30 minutes. A leak free system will maintain the 200 psig pressure for 30 minutes.

5) If there is no pressure drop for 30 minutes, slowly release nitrogen pressure and disconnect the nitrogen source from the evaporator assembly. Re-install the pressure switch, lubricating the threads with Ester-25065 oil (P/N 197-511). Proceed to step (i).

6) If there is a pressure drop, find leak(s) by applying a soap solution to all connections.

7) Tighten/re-tighten fittings as necessary to stop leak(s). If leaks are due to damaged or worn components, proceed with refrigerant recovery/system discharge, perform repairs or component replacement and repeat leak detection procedure.



Robinair 34700 Charging Stand
Figure 3

(g) Using HFC-134a refrigerant:

1) Ensure that there is at least five pounds of refrigerant in the charging cylinder.

2) Open the high pressure control valve and the refrigerant control valve on the charging stand. Allow one pound of refrigerant to enter the system.

3) Close the high pressure control valve and the refrigerant control valve.

CAUTION: IT IS RECOMMENDED THAT A THICK SOLUTION OF SOAP AND WATER BE USED TO CHECK FOR LEAKS INSTEAD OF THE PROPANE LEAK DETECTOR THAT IS PROVIDED WITH SOME BRANDS OF CHARGING STANDS.

4) Locate leak(s) using an electronic leak detector designed to detect HFC-134a refrigerant. Or, use soap and water in a thick solution.

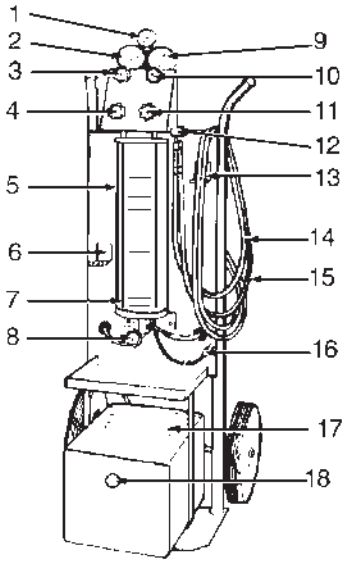
5) If no leaks are found, proceed to step (h).

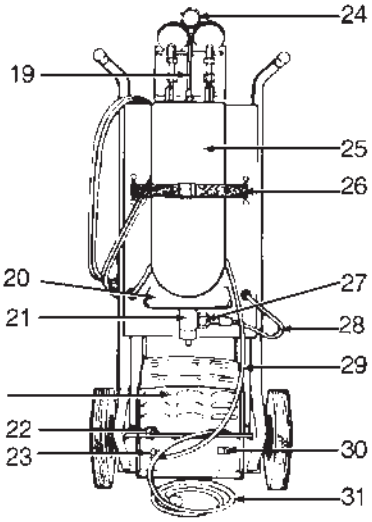
6) Tighten/re-tighten fittings as necessary to stop leak(s). If leaks are due to damaged or worn components, proceed with refrigerant recovery/system discharge, perform repairs or component replacement and repeat leak detection procedure.

(h) Recover remaining refrigerant from system using the Robinair 34700 (or other approved) charging station (see Discharging the System, above). Any quantity of oil recovered from aircraft must be measured and an equal amount of new oil (i.e. - PAG or POE with HFC-134a) must be added to system before recharging.

(i) Evacuate the system, and charge see below.

1. CYLINDER PRESSURE GAUGE
2. COMPOUND GAUGE
3. VALVE, LOW PRESSURE CONTROL
4. VALVE, VACUUM CONTROL
5. CHARGING CYLINDER
6. BRACKET
7. SIGHT GLASS
8. CYLINDER BASE VALVE
9. HIGH PRESSURE GAUGE
10. VALVE, HIGH PRESSURE CONTROL
11. VALVE, REFRIG. CONTROL
12. CHARGING LINE HOSE HOLDER
13. BRACKET
14. LOW PRESSURE CHARGING LINE
15. HIGH PRESSURE CHARGING LINE
16. HEATING ELEMENT PLUG
17. VACUUM PUMP
18. OIL FILL LOCATION
19. NECK ASSEMBLY
20. REFRIGERANT DRUM SUPPORT
21. REFRIGERANT DRUM VALVE
22. VACUUM PUMP VALVE
23. VACUUM PUMP EXHAUST PORT
24. TOP CYLINDER VALVE
25. REFRIGERANT DRUM





26. WEB STRAP
27. REFRIGERANT DRUM REDUCER
28. CHARGING CYLINDER HOSE
29. VACUUM PUMP INTAKE HOSE
30. VACUUM PUMP SWITCH
31. VACUUM PUMP POWER CORD

Kent Moore J23500 Charging Stand
Figure 4

(3) Evacuating the System

NOTE: Perform a Leak Detection check, above, before evacuating the system.

If the system has been operated in a discharged condition or anytime the system has been open to atmospheric pressure, the receiver-dehydrator must be replaced and the system evacuated to remove any trapped air and moisture which has entered it. A vacuum pump capable of pulling 29 inches of mercury or better should be used. As the pressure in the air conditioning system is lowered, the boiling temperature of the water (moisture) that may be present is also lowered. This then forces any moisture, in the form of water vapor, out of the system. “Chart 4” demonstrates the effectiveness of moisture removal under a given vacuum.

(a) Using a Kent Moore J23500 or similar charging stand: (See “Figure 4”)

1) Ensure that aircraft and/or ground power is OFF.
2) Close all valves on the charging stand.
3) Remove closeout panel at the rear of the cabin to gain access to the service valves.
4) Remove protective caps from the high and low side service ports on the evaporator.
5) Remove the protective cap from the vacuum pump outlet.
6) Connect the blue and red hoses to the service ports.
7) Start the vacuum pump.
8) Open the valve on the vacuum pump. Open the low pressure control valve and the vacuum control valve on the charging stand.
9) After five minutes of pump operation, the high pressure gauge should indicate slightly below zero.
   a) If it doesn’t, stop the pump and eliminate the blockage in the system replacing the faulty component, then repeat steps (1)-(9).
   b) If it does, open the high pressure control valve on the charging stand and continue to evacuate the system.
10) Operate the vacuum pump for fifteen minutes, or until the compound gauge indicates 24 to 26 in. Hg. whichever occurs first.

CHART 4
USING VACUUM TO EVACUATE MOISTURE

	System Vacuum	Boiling Point of Water (°F)
COMPOUND GAUGE READING IN INCHES OF MERCURY VACUUM	27.95	101
	28.74	84
	29.53	52
	29.76	29
	29.84	15
	29.88	1

NOTE: Compound gauge reading will be approximately one inch lower, numerically, for each 1000 feet elevation above sea level.

11) Close the low pressure control valve and the high pressure control valve on the charging stand. Stop the vacuum pump and observe the compound gauge. If the gauge rises at a rate faster than 1 in. Hg. in 5 minutes, there is a leak in the system. Locate and fix the leak. Repeat the evacuation steps above.

12) Open the low pressure control valve and the high pressure control valve on the charging stand. Continue pumping and hold the system pressure below 26 in. Hg. for a minimum of 30 minutes. All the pumping time specified above may be included in the 30 minutes provided that no leaks or blockages are noted, and provided that the system is not opened by removal or disconnection of components.

13) Close the low pressure control valve, the high pressure control valve and the vacuum control valve. Stop the vacuum pump and perform the charging procedure immediately.

(b) Using a Robinair 34700 or similar charging/recovery stand: (See “Figure 3”)

1) Ensure that aircraft and/or ground power is OFF.

2) Close all valves on charging stand.

3) Remove closeout panel at rear of cabin to gain access to service valves.

4) Remove protective caps from the high and low side service ports on the evaporator.

5) Connect the blue and red hoses to the service ports (see “Figure 6”), on systems equipped with quick disconnect connections, open coupler valves.

6) Open blue (low side) valve (1) on unit’s control panel.

7) Open both the red GAS (vapor) valve and the blue LIQUID valve on the tank (ref. “Figure 6”).

8) Program the length of evacuation time.

a) Press the VACUUM key on control panel key pad.

b) Display will show unit is in VACUUM mode.

c) Refer to operator’s manual for further detail.

9) Enter the required time in minutes and seconds (30:00 minutes minimum) by pressing appropriate keys and then ENTER on keypad. The display will show selected time in minutes and seconds. Example: one hour and fifteen minutes (1:15) would be entered as 7500. The display will show 75:00. Thirty minutes is entered as 3000. the display will show 30:00.

10) To start the vacuum pump press the VACUUM key on keypad again.

11) Vacuum sequence will continue for the programmed time. Digital display will then show CPL, indicating that the evacuation is completed.

12) If, after 5 minutes of pump operation, the RED gauge does not indicate a little below zero:

a) Stop the pump by pressing the “1” key or the SHFT/RESET key.

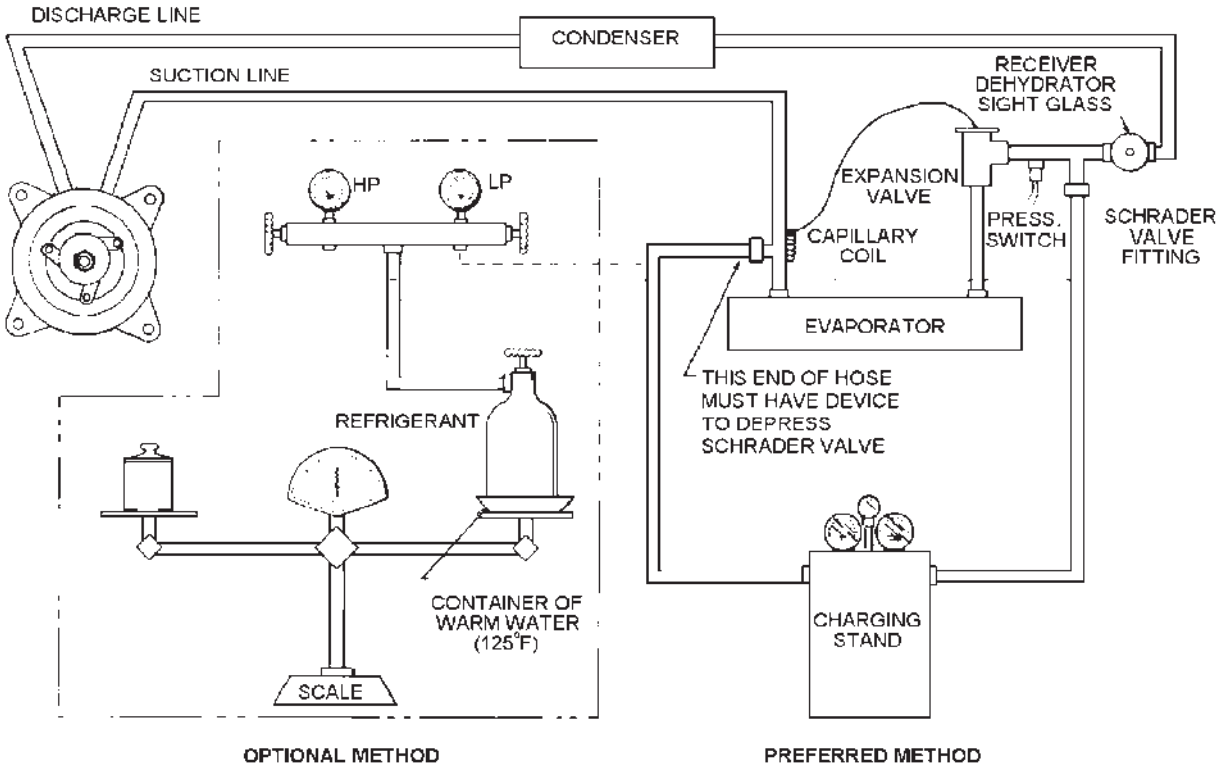
b) Eliminate blockage in the system by replacing faulty parts.

c) Repeat steps (1) through (12).

13) If, after 5 minutes of pump operation, the RED gauge indicates a little below zero, open red (high side) valve (2), and continue evacuation.
    a) System vacuum (i.e. - low side gauge (GAUGE 1)) should attain 24 to 26 inches of mercury (in. Hg.) in 10 to 15 minutes.
    b) Allow pump to hold a vacuum of 26 in. Hg. (or below) for a minimum of 15 minutes.
    c) Failure to achieve or hold a vacuum of 26 in. Hg. (or below) in either (a) or (b), above, indicates a leak in the system. Locate leak as described in Leak Detection, above.
    d) Repair leak. Repeat steps (1) through (13).
14) With the low side (1) and high side (2) valves OPEN, continue pumping, holding system below 26 in. Hg. for a minimum of 30 minutes.

    NOTE: All specified pumping times may be included in the 0:30 minutes, provided no blockage or leaks are noted, and provided the system is not opened by disconnecting or removing components.

15) When panel display reads CPL (complete), close both the low side valve (1) and the high side (2) valves.
16) Perform charging procedure immediately, see Charging the System, “Figure 5”.



Charging Hookup
Figure 5

(4) Charging the System

NOTE: Always evacuate the system (see above), before charging.

(a) Using a Kent Moore J23500 or similar charging stand (refer to “Figure 3” and “Figure 4”)

1) Open the valve at the base of the charging cylinder and fill the charging cylinder with sufficient refrigerant to charge the system. If refrigerant stops filling the cylinder, open the bleed valve at the top of the charging cylinder to relieve head pressure and allow refrigerant to continue filling the charging cylinder.
2) Close the bleed valve and the valve at the base of the charging cylinder.
3) Turn the charging cylinder sight glass to match the pressure reading on the charging cylinder pressure gauge. Keep the sight glass in this position during the remainder of the charging operation.
4) Connect the heating element plug to a 110 volt power outlet.
5) With the low pressure control valve (3) closed, open the refrigerant control valve (11) and the high pressure control valve (10).
6) Allow the correct amount of refrigerant to enter the high side of the system.
7) Close the high pressure control valve (10) and the refrigerant control valve (11).
8) Disconnect the hoses from the airplane’s system.

(b) Using a Robinair 34700 charging station or equivalent (Refer to “Figure 3” and “Figure 6”)

CAUTION: APPLIES TO ROBINAIR 34700 OR SIMILAR CHARGING/RECOVERY STATION. IF USING DIFFERENT EQUIPMENT, SEE OPERATOR’S MANUAL OF STATION BEING USED FOR DETAILED INSTRUCTIONS FOR DISCHARGING SYSTEM.

1) Check that main power switch and/or ground power is OFF.
2) Check that the LBS/KG selector switch on back of unit is in desired measurement mode. Be sure to turn OFF the main power switch before changing the measurement mode.

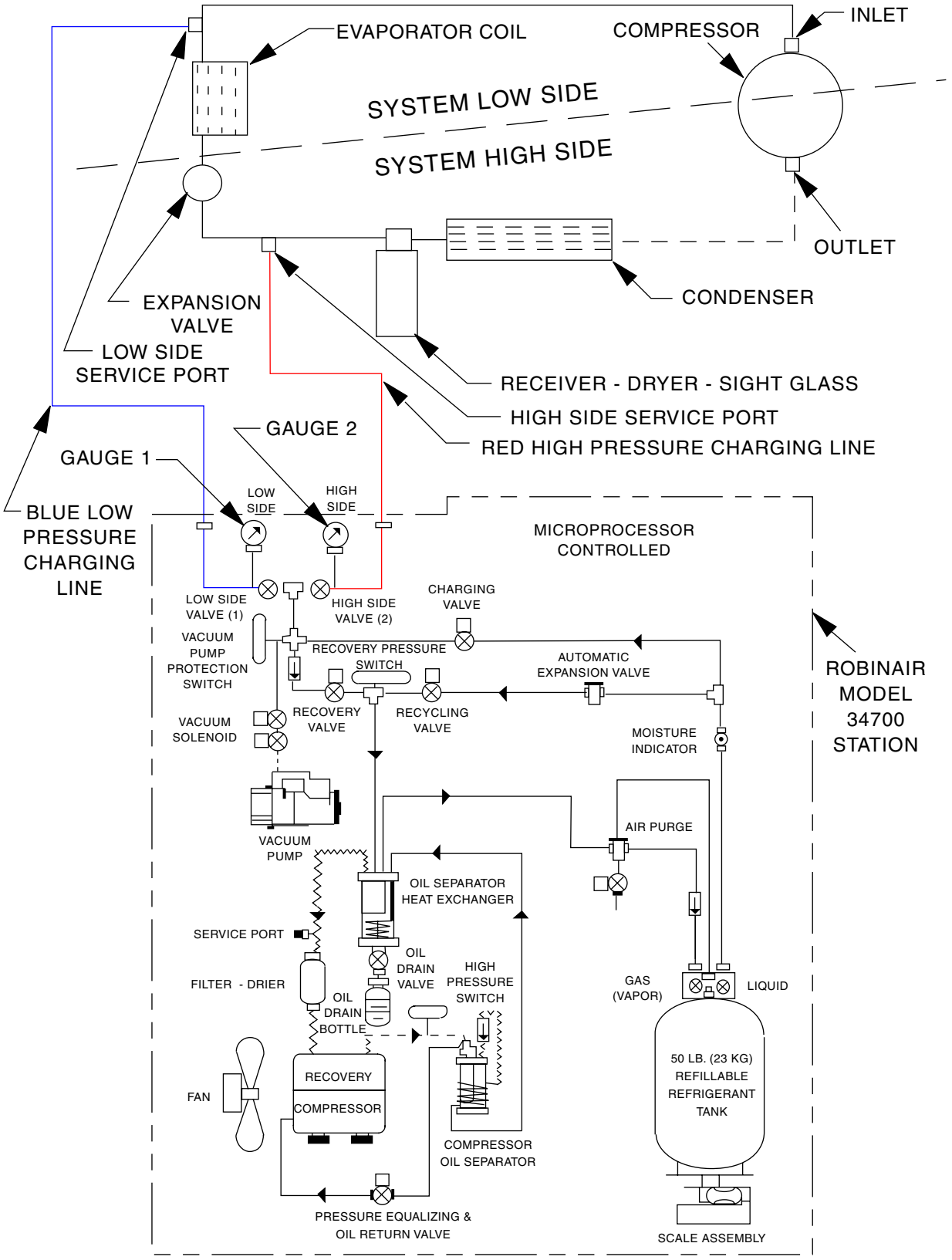
NOTE: You may enter the amount of refrigerant to be charged when the unit is turned ON. The unit will store the amount in memory until it is turned off.

3) Remove protective caps from the high and low side service ports on the evaporator.
4) Connect the blue and red hoses to the service ports (see “Figure 6”), on systems equipped with quick disconnect connections, open coupler valves.

CAUTION: DO NOT PLACE ANY WEIGHT, INCLUDING HANDS AND/OR FEET, ON REFRIGERANT TANK OR SCALE DURING CHARGING PROCESS. ANY WEIGHT DISTURBANCE WILL CAUSE AN INDIRECT TRANSFER OF REFRIGERANT.

CAUTION: ADD REFRIGERANT THROUGH THE LOW PRESSURE SIDE ONLY.

5) Open the low side (blue) valve on the unit’s control panel.



Robinair 34700 Hose Hookup
Figure 6

6) If the messages PROGRAM and CHARGE do not display, press the CHG key to enter the PROGRAM mode.

NOTE: The amount of refrigerant required must be determined for each airplane. It is the amount that will result in a minimum 20° F temperature drop at the evaporator. The PA-28-181 requires approximately 2.5 LBS.

7) Enter amount of refrigerant required to charge the system by pressing the appropriate number keys and ENTER on keypad.

NOTE: You may enter the amount of refrigerant to be charged when the unit is turned ON. The unit will store the amount in memory until it is turned off.

8) To begin charging process, press CHG key on keypad.
   a) The digital display will read AUTOMATIC and show the amount of refrigerant programmed for the charge.
   b) As the solenoid opens, it will make an audible sound.
   c) The display will count down to zero, and display message CPL, when charging is complete.
9) Close low side (blue) valve. Check that the high (red) valve is also closed. Also close coupler valves.
10) Perform Post Charging Operational Check, see below.

(5) Post Charging Operational Check

See “Figure 3”, “Figure 4”, and “Figure 6”. Numbers in parentheses refer to “Figure 4”, numbers in brackets refer to “Figure 6”.

(a) Ensure that the low pressure control valve (3) [1] and the high pressure control valve (10) [2] are closed. Hook up the charging stand to the system as shown in “Figure 5” and “Figure 6”.

(b) With Robinair 34700 style stands equipped with quick disconnect couplings only, ensure coupler valves are open.

CAUTION: ASCERTAIN THAT THE AREA AROUND THE AIRPLANE IS CLEAR AND THAT A QUALIFIED PERSON IS AT THE CONTROLS OF THE AIRPLANE.

CAUTION: ENSURE THE AIRPLANE IS HEADED INTO THE WIND.

(c) Start aircraft engine, bring to normal ramp idle rpms until oil temperature rises to normal operating range. Increase engine to 1000 rpms, activate the air conditioning system controls at maximum cooling high blower speed for 2 minutes. Then operate the engine, at 2000 rpm, and air conditioning system for an additional 2 minutes.

(d) Check the sight gauge on the receiver-dehydrator during the engine operation at 1000 rpm and 2000 rpm to ensure refrigerant flow. Normal HFC-134a operation may appear from clear to foamy. Any indication of bubbles passing the sight gauge does not indicate that additional refrigerant is required. Confirm that there is a minimum of a 20° F temperature drop across the evaporator. If doors are closed or sealed, cabin ambient will continue to drop and this delta will decrease.

(e) If additional refrigerant is required, add it slowly through the refrigerant control valve (11) (Kent Moore J23500 style stands only) and the low pressure control valve (3) [1] until the temperature drop across the evaporator is a minimum of a 20° F. Regulate the flow of refrigerant with the low pressure control valve. Do not allow the compound gauge (2) or the low side gauge [1] to exceed a reading of 40 psi.

(f) With the engine operating at 1,000 rpm, the low and high side gauges should indicate as shown in “Chart 5”.

(g) With the charge properly established, stop the engine and, with Robinair 34700 style stands equipped with quick disconnect couplings only, close the coupler valves.

(h) Close the low pressure control valve (3) [1] and, with Kent Moore J23500 style stands only, the refrigerant control valve (11).

(i) Remove the charging stand. Replace all protective caps and covers.

CHART 5
AMBIENT TEMPERATURE INDICATION

GAUGE	AMBIENT TEMPERATURE	GAUGE INDICATION
Low Pressure	All	10 to 35 psig
High Pressure	Up thru 75°F	125 psig min to 175 psig max
High Pressure	Over 75°F	150 psig min to 275 psig max

F. Servicing the System with a Manifold Set

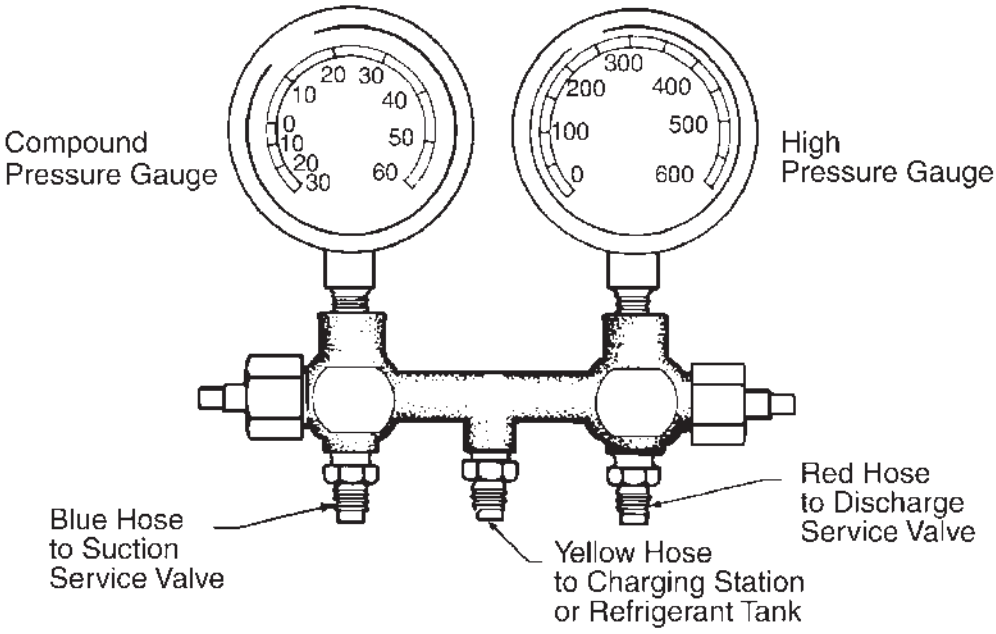
(1) Test Gauge and Manifold Set

The proper testing and diagnosis of the air conditioning system require that a manifold gauge set (or a charging stand) be attached to the system. This set consists of two gauges mounted to a manifold. One gauge is a high pressure gauge used in the discharge side of the system. The other is a low pressure gauge used in the suction side of the system. The manifold is a device having fittings for both gauges and connection hoses with provisions for controlling the flow of refrigerant through the manifold. (Refer to “Figure 7” and “Figure 8”.)

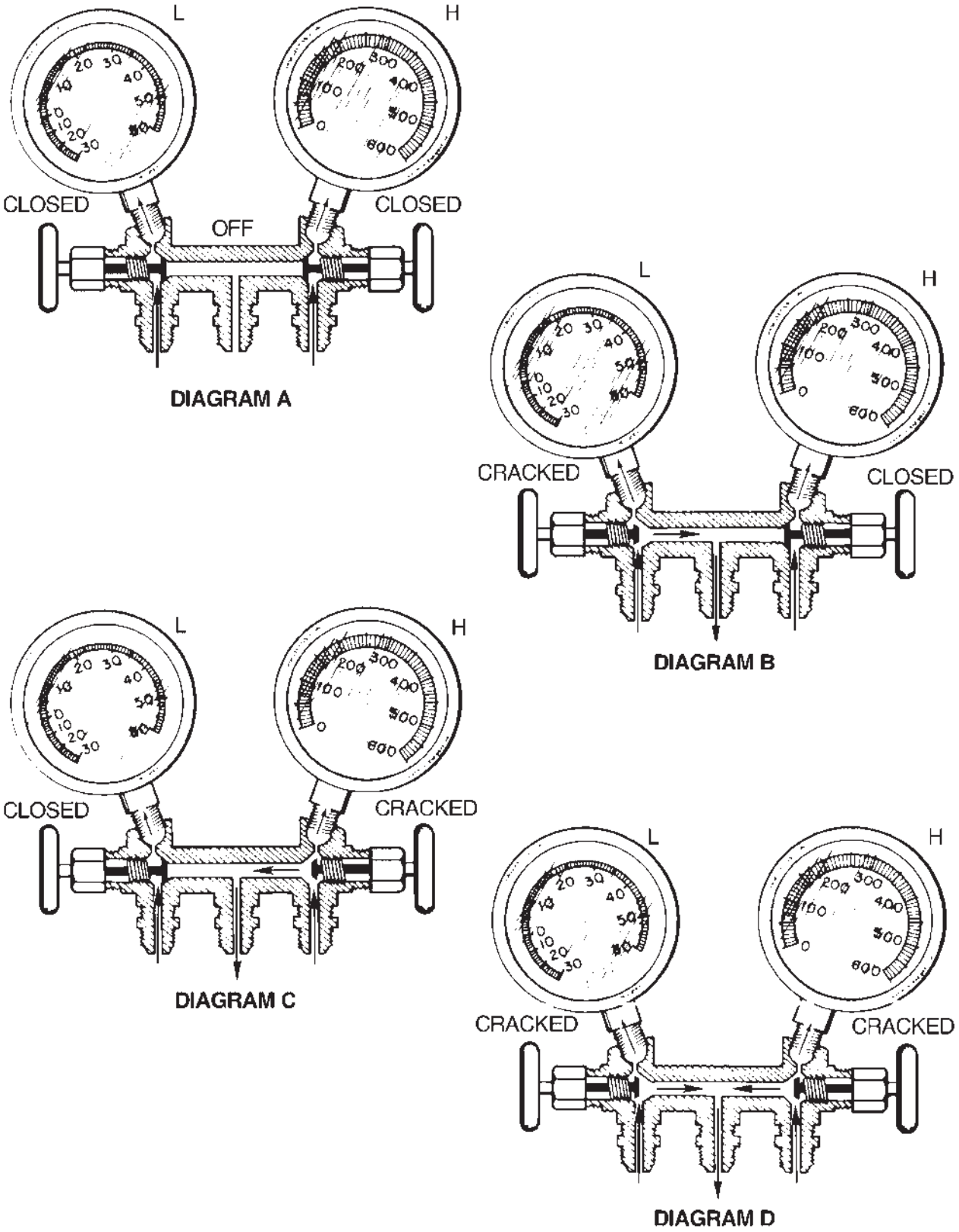
The center port of the manifold set is used for charging or evacuation procedures, or any other service that may be necessary.

Both the high or low side of the manifold have hand shut-off valves. When the hand valve is turned all the way in, in a clockwise direction, the manifold is closed. The pressures on that side of the system will, however, be recorded on the gauge above the hose.

Cracking the hand valve, in the counterclockwise direction, opens the systems to the middle service port of the manifold set. This is desirable only when it is necessary to let refrigerant out of or into the system. (Refer to “Figure 7” and “Figure 8”.)



Test Gauge and Manifold Set
Figure 7



Manifold Set Operation
Figure 8

(2) Leak Detection

Refer to “Figure 9”.

(a) Close both the low side and high side valves on manifold hand set.

(b) Connect manifold hand set middle port (yellow) hose to a regulated (0-300 psig) gaseous dry nitrogen source or a container of HFC-134a refrigerant.

(c) Open nitrogen source or refrigerant container service valve.

(d) Open the manifold hand set high side valve until a pressure of 50 psig is reached on low side gauge. Close high side valve.

(e) Locate leak(s) using soap and water in a thick solution; or, if using HFC-134a, an electronic leak detector designed to detect HFC-134a refrigerant can also be used.

(f) Tighten/re-tighten fittings as necessary to stop leak(s). If leaks are due to damaged or worn components, proceed with refrigerant recovery/system discharge, perform repairs or component replacement and repeat leak detection procedure.

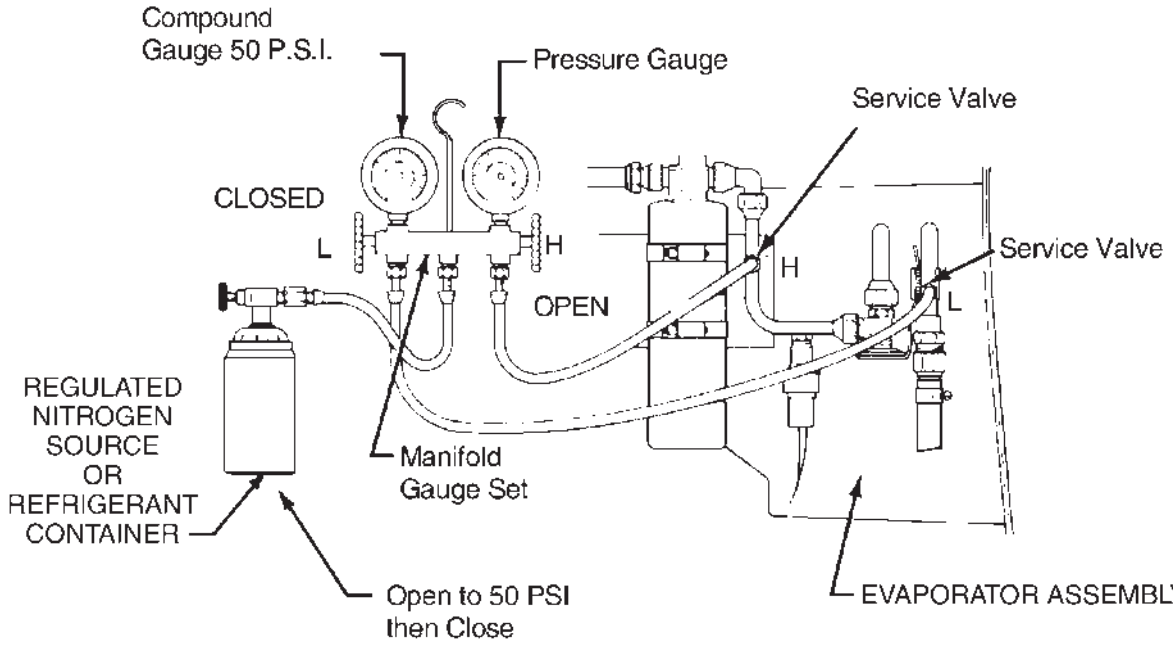
(g) Check that the both high side and low side valves on the manifold hand set are closed.

(h) Close service valve on nitrogen source or refrigerant container. Disconnect yellow manifold hand set center hose from nitrogen source or refrigerant container.

(i) On systems equipped with quick disconnect connections, close coupler valves. Disconnect manifold hand set red and blue hoses from airplane service ports. Remove manifold hand set.

(j) If HFC-134a refrigerant was used, recover remaining refrigerant from system using the Robinair 34700 (or other approved) charging station. Any quantity of oil recovered from aircraft must be measured and an equal amount of new oil (i.e. - PAG or POE with HFC-134a) must be added to system before recharging.

(k) When refrigerant recovery is complete, on systems equipped with quick disconnect connections, close coupler valves. Disconnect charging/test station from service ports.



Leak Test Hookup
Figure 9

(l) Evacuate the system, see below.

(m) Immediately charge the system, see below.

(3) Evacuating the System

See “Figure 10”.

NOTE: Perform a Leak Detection check, above, before evacuating the system.

If the system has been operated in a discharged condition or anytime the system has been open to atmospheric pressure, the receiver-dehydrator must be replaced and the system evacuated to remove any trapped air and moisture which has entered it. A vacuum pump capable of pulling 29 inches of mercury or better should be used. As the pressure in the air conditioning system is lowered, the boiling temperature of the water (moisture) that may be present is also lowered. This then forces any moisture, in the form of water vapor, out of the system. “Chart 4” demonstrates the effectiveness of moisture removal under a given vacuum.

(a) Ascertain that all system pressure is released.

(b) Connect the manifold set hoses to the service ports and vacuum pump as shown in “Figure 10”.

(c) Close the high side (pressure) and low side (suction) hand valves on the manifold set.

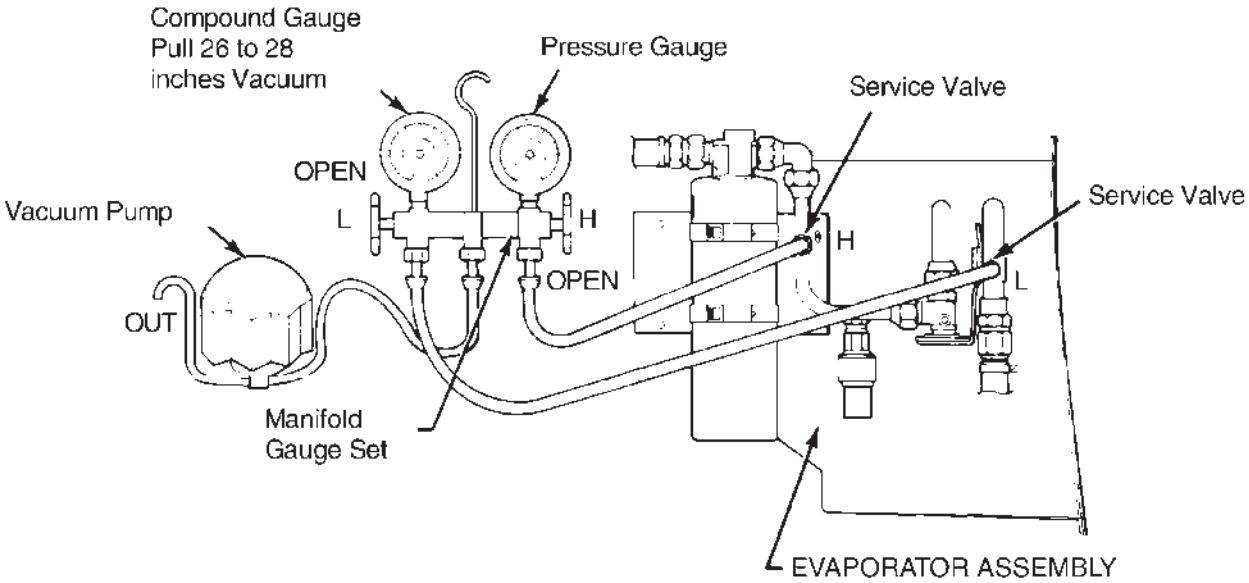
(d) Start the vacuum pump.

(e) Open the low side manifold set hand valve. The low side gauge should show a vacuum.

(f) After five minutes of pump operation the high side gauge should indicate slightly below zero. If it does not, stop the pump and eliminate the blockage in the system by replacing the faulty component, then repeat the previous evacuation steps.

(g) Operate the vacuum pump for fifteen minutes or until the low side gauge indicates 24 to 26 in. Hg. whichever occurs first.

(h) Close the low side hand valve, stop the vacuum pump and observe the low side gauge. If the gauge rises at a rate faster than 1 in. Hg. in 5 minutes, there is a leak in the system. Locate and repair the leak, then repeat the previous evacuation steps.



Evacuation Hookup
Figure 10

(i) With both the low and high side valves open, continue the pumping and hold the system below 26 in. Hg. for a minimum of 30 minutes. All the previous pumping time may be included in the 30 minutes provided that no leaks or blockages are noted, and provided that the system is not opened by removal or disconnection of components.

(j) Close the low and high side hand valves, stop the vacuum pump and perform the charging procedure immediately.

(4) Charging the System Using the Airplane Compressor

This method is the least desirable due to the requirement of operating the airplane's engine to run the compressor.

CAUTION: ASCERTAIN THAT THE AREA AROUND THE AIRPLANE IS CLEAR AND THAT A QUALIFIED PERSON IS AT THE CONTROLS OF THE AIRPLANE.

(a) Keep the system under the vacuum established during the evacuating procedure with both hand valves in the closed position.

(b) Attach a container of the refrigerant (HFC-134a) to the manifold set and open the container service valve.

(c) Loosen the center hose at the manifold set until a hiss can be heard. Allow the gas to escape for 2 to 3 seconds, then tighten the connection.

(d) Open the high side manifold set hand valve, observe the low side gauge, then close the high side hand valve. The low side gauge should immediately change from an indication of a vacuum to an indication of pressure. If it does not, the system is blocked, and the blockage must be corrected before proceeding.

(e) Start the engine and operate it at 1000 rpm.

(f) Adjust the airplane air conditioning controls for maximum cooling, high blower speed.

(g) Keep the refrigerant cylinder in an upright position. A slug of liquid refrigerant entering the system would damage the compressor.

(h) Open the low side manifold set hand valve and allow two pounds of refrigerant in the gas state to enter the system.

(i) Close the low side manifold set hand valve.

(j) Proceed with the Post Charging Operational Check, below.

(5) Post Charging Operational Check

CAUTION: ASCERTAIN THAT THE AREA AROUND THE AIRPLANE IS CLEAR AND THAT A QUALIFIED PERSON IS AT THE CONTROLS OF THE AIRPLANE.

NOTE: Head the airplane into the wind during these checks.

(a) With the manifold set installed, and both hand valves closed, start aircraft engine, bring to normal ramp idle rpms until oil temperature rises to normal operating range. Increase engine to 1000 rpms, activate the air conditioning system controls at maximum cooling high blower speed for 2 minutes. Then operate the engine, at 2000 rpm, and air conditioning system for an additional 2 minutes.

(b) Check the sight gauge on the receiver-dehydrator during the engine operation at 1000 rpm and 2000 rpm to ensure refrigerant flow. Normal HFC-134a operation may appear from clear to foamy. Any indication of bubbles passing the sight gauge does not indicate that additional refrigerant is required. Confirm that there is a minimum of a 20° F temperature drop across the evaporator. If doors are closed or sealed, cabin ambient will continue to drop and this delta will decrease.

(c) Add additional refrigerant slowly through the low side manifold set hand valve until the temperature drop across the evaporator is a minimum of a 20° F.

(d) Close the low side hand valve and refrigerant container valve.

(e) With the engine operating at 1,000 rpm, the gauges should indicate as shown in “Chart 5”.

(f) Once the charge is properly established, stop the engine, close the refrigerant container service valve. Remove the manifold set and replace all protective caps and covers.

(6) Adding Partial Charge to System

The system can be topped off with refrigerant by the following method:

(a) Remove the closeout panel at the rear of the cabin.

(b) Connect a charging hose to a refrigerant cylinder and also to the low pressure Schrader valve fitting on the manifold assembly.

(c) Purge the charging hose by allowing a small amount of refrigerant gas to escape at the Schrader valve fitting.

CAUTION: ASCERTAIN THAT THE AREA AROUND THE AIRPLANE IS CLEAR AND THAT A QUALIFIED PERSON IS AT THE CONTROLS OF THE AIRPLANE.

NOTE: Head the airplane into the wind during this procedure.

(d) Start the engine, operate at 1000 rpm and turn the air conditioner on maximum cool.

(e) Remove the plastic plug (if installed) from the sight glass in top of the receiver-dehydrator and check to ensure refrigerant flow

(f) With a low refrigerant charge in the system, the temperature drop across the evaporator will be less than 20° F when the system is operating.

(g) Open the valve on the refrigerant cylinder.

(h) Allow refrigerant to flow into the system until the temperature drop across the evaporator is a minimum of a 20° F.

(i) Close the refrigerant valve and check to see that the temperature drop across the evaporator remains a minimum of a 20° F during system operation.

(j) When the temperature drop across the evaporator remains a minimum of a 20° F, add an additional pound of refrigerant to the system. (Engine should be operating at 1,000 rpm.)

NOTE: This is done with OAT at 70° F, or higher, with the air conditioner operating.

(k) Shut off the air conditioner and engine. Remove the charging hose from the Schrader valve with care due to refrigerant remaining in the line.

(l) Reinstall closeout panel.

G. T/CCI (York) Compressor

Original equipment S/N's 2843001 thru 2843783.

See "Figure 11".

The engine driven compressor is mounted on the right front of the engine. A V-belt connected to the ring gear pulley drives the compressor through a magnetic clutch.

(1) Compressor Service

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Do not service the compressor in the field. Service must be done by a qualified shop which has the special equipment and trained personnel required to properly service the unit.

Maintenance of the unit is limited to removal and installation of compressor, checking the compressor oil level and replacement of worn drive belt or compressor clutch. Contact Climate Control Inc., for special tools and instructions for detailed compressor maintenance.

(2) Removal

CAUTION: PREVENT DIRT OR FOREIGN MATERIAL FROM ENTERING THE SYSTEM. ALL HOSE AND TUBING ENDS SHOULD BE CAPPED IMMEDIATELY. ANY LUBRICATION REQUIRED IN THE ASSEMBLY OF THE COMPONENTS SHOULD BE REFRIGERANT OIL OF THE TYPE USED IN THE COMPRESSOR.

The system must be discharged before removing compressor. Refer to Discharging, above. Remove the compressor as follows:

(a) Be sure the air conditioning circuit protector is in the OFF position.

(b) Remove engine cowling and right front baffles.

(c) Disconnect the electrical leads to the magnetic clutch on the compressor.

(d) Discharge the air conditioning system to an appropriate environmentally approved refrigerant recovery station. See Discharging, above.

(e) Remove the suction and discharge lines from the connections on the compressor.

NOTE: All open lines should be capped immediately to prevent dirt and moisture from entering the system.

(f) Loosen bolt securing compressor idler pulley to release belt tension and remove belt from pulley. (Do not force belt over pulleys.)

(g) Support compressor and remove six (6) bolts securing compressor to engine mounting brackets.

(h) Remove compressor.

(3) Installation

Install the compressor as follows:

(a) Align compressor mounting holes with mounting brackets. Install bolts and progressively tighten to a torque of 20 foot pounds. (Safety all bolts with 0.032 wire.) If required, adjust compressor drive sheave forward and aft alignment by adding one (1) NAS1149F0663P washer and/or one (1) NAS1149F0632P washer, alone or in combination, between the compressor mounting bracket and compressor and between mounting bracket and engine to achieve a gap of .000 to .030 inch.

(b) After mounting compressor attach wrench to center nut of compressor and slowly rotate in direction of engine rotation 15 times. This step blows out any residual oil in the compressor head that could cause damage when the system is operated under engine power for the first time.

(c) Check oil level in compressor as described in Checking Compressor Oil, below.

CAUTION: DO NOT FORCE BELT INTO PULLEY SHEAVE. IF NECESSARY, REMOVE IDLER ASSEMBLY.

(d) Place drive belt over clutch pulley and adjust alignment of pulleys and belt as described in Drive Belt Service, below.

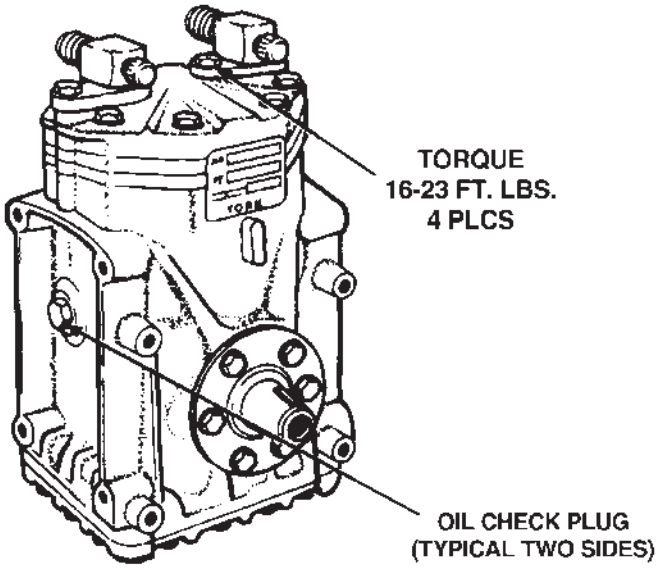
(e) Connect discharge and suction lines to compressor.

(f) Evacuate the system per Evacuating the System, above.

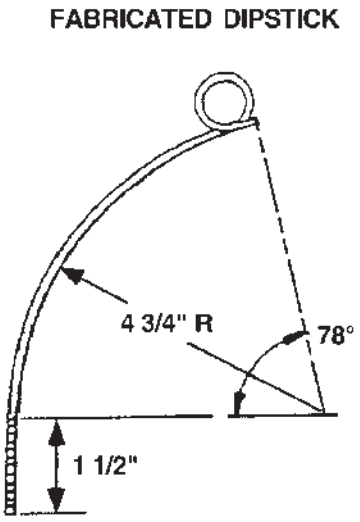
(g) Charge the system per Charging the System, above.

(h) Install right front baffle.

(i) Install engine cowling.



F209R-25019 COMPRESSOR USES PAG-21941
(POLYALKYLENE-GLYCOL) LUBRICANT



12 NOTCHES
1/8 INCH APART

(4) Checking Compressor Oil Level

Total system oil capacity is 14 fluid ounces; 10 in compressor and 4 in system.

Whenever a system component has been replaced or there is an obvious leak, use the following procedure to check the compressor oil level after making necessary repairs. I.E. - check oil level each time the system is discharged.

(a) Discharge system per Discharging the System, above.

(b) Recover all refrigerant from the system. Be careful not to lose oil.

(c) Remove 0.375 inch oil fill plug in top side of compressor crankcase.

(d) Verify (and rotate as necessary) the “K” stamped on the compressor clutch front face is in the UP position.

(e) Using a dipstick as shown in “Figure 11”, measure oil level to lowest point in compressor crankcase. Use “Chart 6” to translate measurement into ounces of oil.

(f) Do not operate compressor with less than six (6) ounces of oil. Do not add more than ten (10) fluid ounces. Use compressor oil, P/N 197-511.

(g) Install the oil filler plug.

(h) Evacuate and charge the system per Evacuating the System and Charging the System Procedures in this section.

CHART 6
COMPRESSOR OIL CHARGE (INSTALLED IN AIRPLANE)

Oil Charge	6	8	10
Dipstick Reading (inches)	13/16	1.00	1-3/16

(5) Magnetic Clutch

See “Figure 12”.

(a) Removal

CAUTION: DO NOT USE A WHEEL PULLER ON OUTER FLANGE OF PULLEY. THIS WILL DAMAGE PULLEY GROOVES OR CLUTCH BEARINGS.

1) Remove self-locking capscrew and washer from compressor shaft.
2) Insert 5/8 - 11 UNC-2B bolt in threaded part of hub and tighten. Pressure exerted by the bolt on compressor crankshaft end will force off rotor pulley assembly without clutch or compressor damage.
3) Remove four bolts securing field assembly against compressor bosses and remove bolts, washers, and field assembly.

(b) Installation

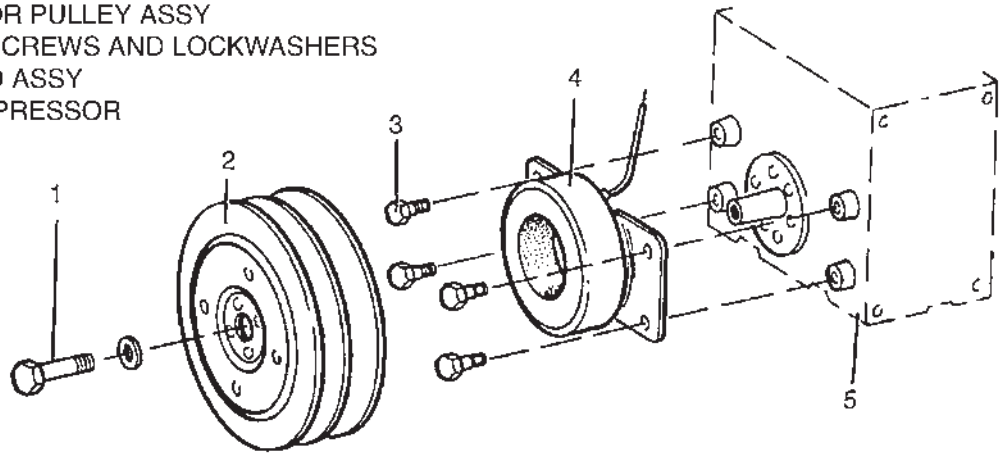
CAUTION: COMPRESSOR SHAFT MUST BE CLEAN AND FREE FROM BURRS.

1) Position field assembly against the compressor bosses, with electrical leads to cylinder side of compressor.
2) Secure field assembly with four capscrews and lockwashers, handtight only.
3) Connect electrical lead from the field assembly. The ground wire must also be connected.
4) Slide pulley assembly over field assembly and onto crankshaft, now tighten field assembly to a torque of 8 ft.-lbs. Secure pulley assembly with washer and new self-locking capscrew. Tighten capscrew to a torque of 20 ft.-lbs.

NOTE: If clutch is not engaged while tightening capscrew, insert a spanner into hole in armature face.

5) Spin pulley by hand to check interference between the field and rotor pulley assemblies. If there is interference, a rubbing noise can be heard as pulley rotates. Remove rotor pulley assembly and adjust field assembly mounting until the interference is eliminated.

1. CAPSCREW
2. ROTOR PULLEY ASSY
3. CAPSCREWS AND LOCKWASHERS
4. FIELD ASSY
5. COMPRESSOR



Magnetic Clutch
Figure 12

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H. Sanden Compressor

Original equipment in S/N's 2843784 and up. Authorized service replacement in S/N's 2843001 thru 2843783 per Piper Kit 88551-001.

See "Figure 13".

The engine driven compressor is mounted on the right front of the engine. A V-belt connected to the ring gear pulley drives the compressor through a magnetic clutch.

(1) Compressor Service

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Do not service the compressor in the field. Service must be done by a qualified shop which has the special equipment and trained personnel required to properly service the unit.

Maintenance of the unit is limited to removal and installation of compressor, checking the compressor oil level and replacement of worn drive belt or compressor clutch.

(2) Removal

(a) Be sure the air conditioning circuit protector is in the OFF position.

(b) Remove engine cowling and right front baffles.

(c) Disconnect the electrical leads to the magnetic clutch on the compressor.

(d) Discharge the air conditioning system to an appropriate environmentally approved refrigerant recovery station. Refer to Discharging.

CAUTION: PREVENT DIRT OR FOREIGN MATERIAL FROM ENTERING THE SYSTEM. ALL HOSE AND TUBING ENDS SHOULD BE CAPPED IMMEDIATELY.

(e) Remove the suction and discharge lines from the connections on the compressor.

(f) Loosen bolt securing compressor idler pulley to release belt tension and remove belt from pulley. (Do not force belt over pulleys.)

(g) Support compressor and remove bolts (4 ea.), washers (as required), and nuts (4 ea.) securing compressor to mounting brackets.

(h) Remove compressor.

(3) Installation

(a) Drain and charge the compressor with oil.

Ensure that compressor ports are capped air tight.

1) Drain compressor of all oil by removing oil plug and draining oil reservoir.

2) Turn compressor on end so that oil may drain out through suction and discharge ports. Rotate compressor pistons serveral times to make sure residual oils are drained.

3) Fill compressor with 3.4 fluid ounces of compressor oil (Piper PN 197-511).

a) Ensure that the seat and O-ring are clean and wet with oil.

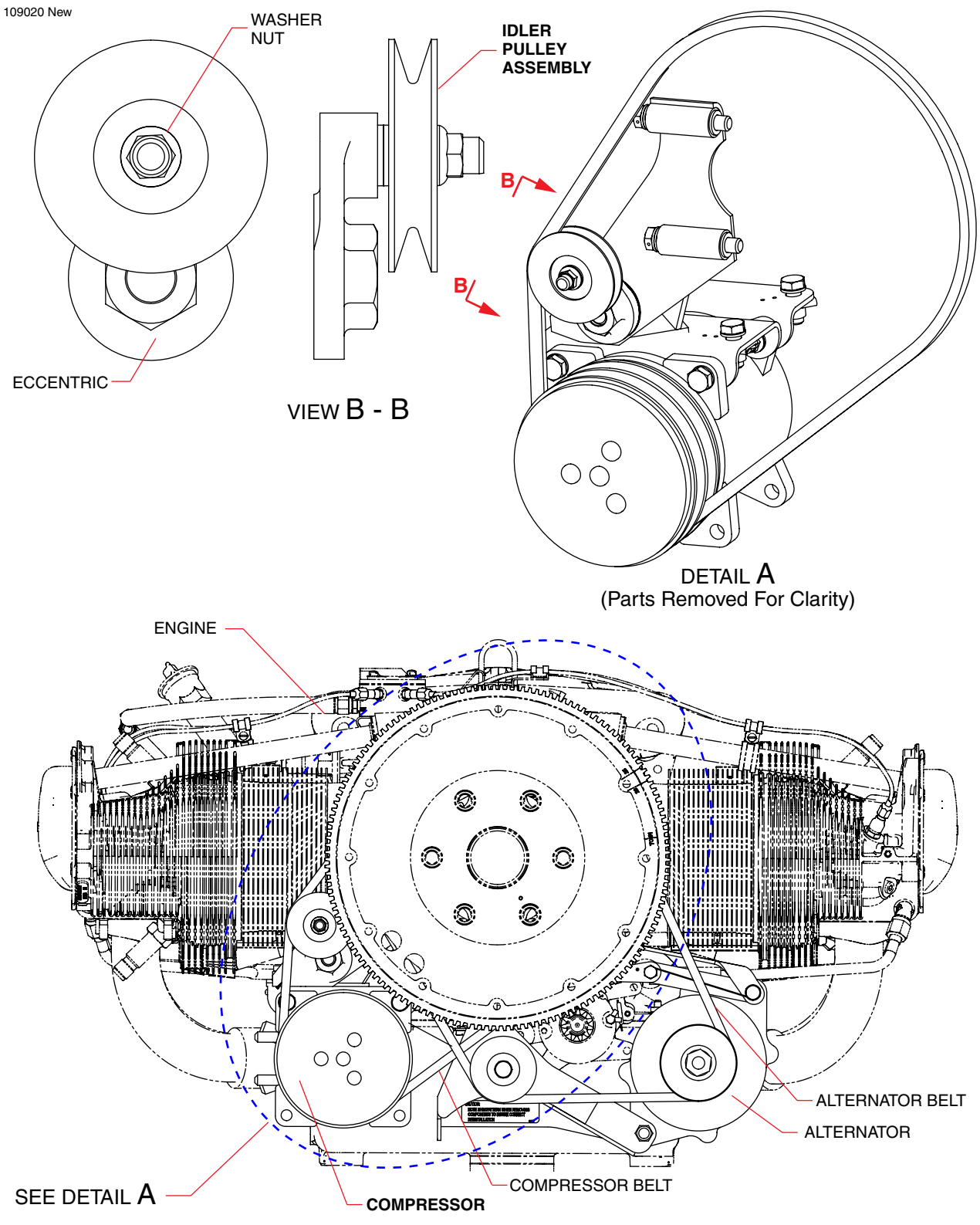
b) Reinstall the oil plug, first checking that the sealing O-ring is not twisted.

CAUTION: DO NOT OVER TIGHTEN THE PLUG TO STOP A LEAK. IF THE PLUG LEAKS, REMOVE IT AND INSTALL A NEW O-RING.

c) Torque the plug from 5 to 8 foot-pounds (6.8 to 10.8 N-m).

(b) Align compressor mounting holes with mounting brackets.

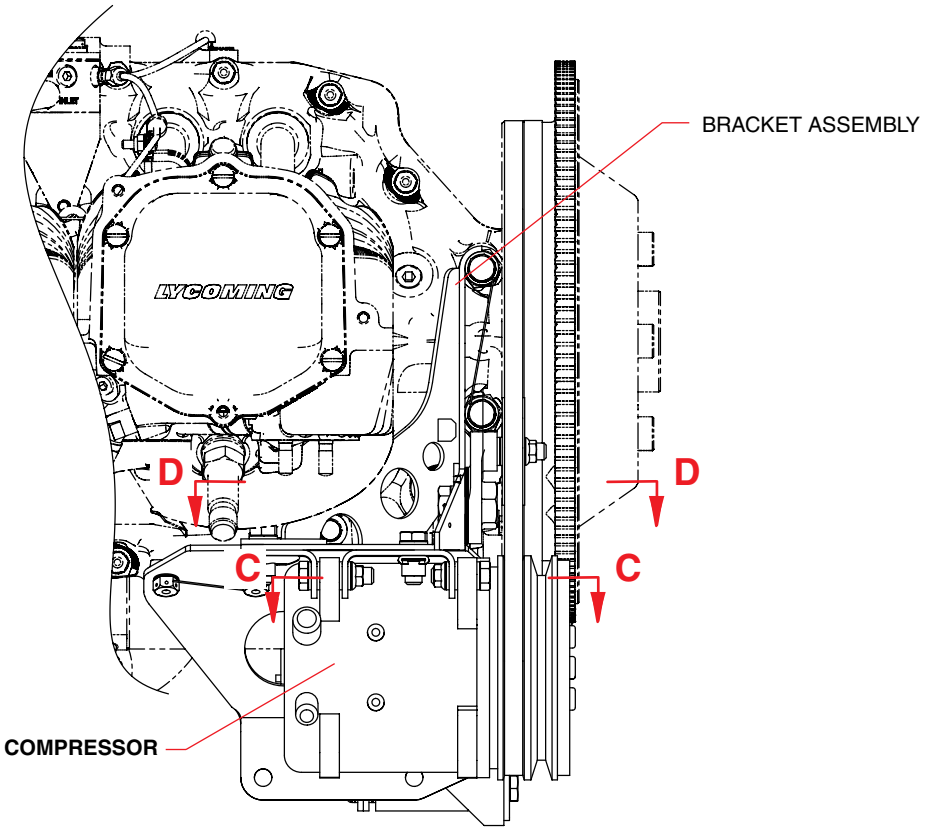
(c) Install bolts (4 ea.), washers (as required to align pulleys), and nuts (4 ea.).



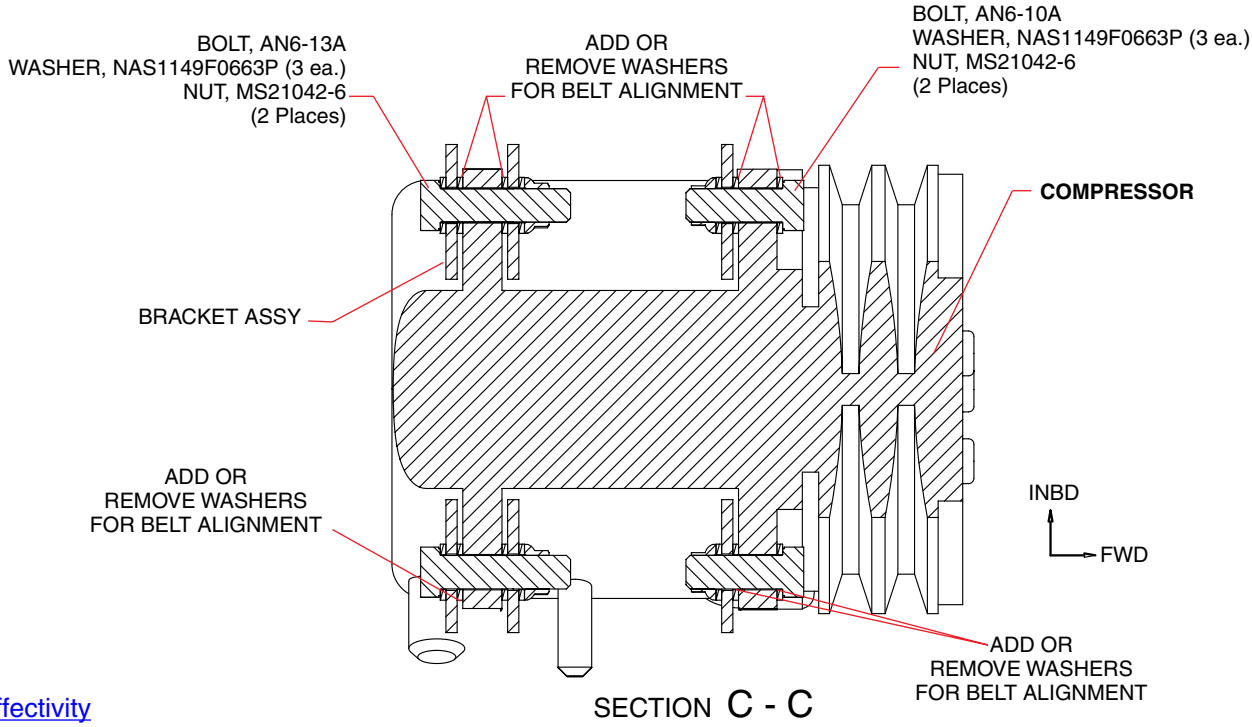
Sanden Compressor Installation
Figure 13 (Sheet 1 of 3)

Effectivity
2843784 and up,
or 2843001 thru 2843783
with Piper Kit 88551-001

109020 New



LOOKING LEFT

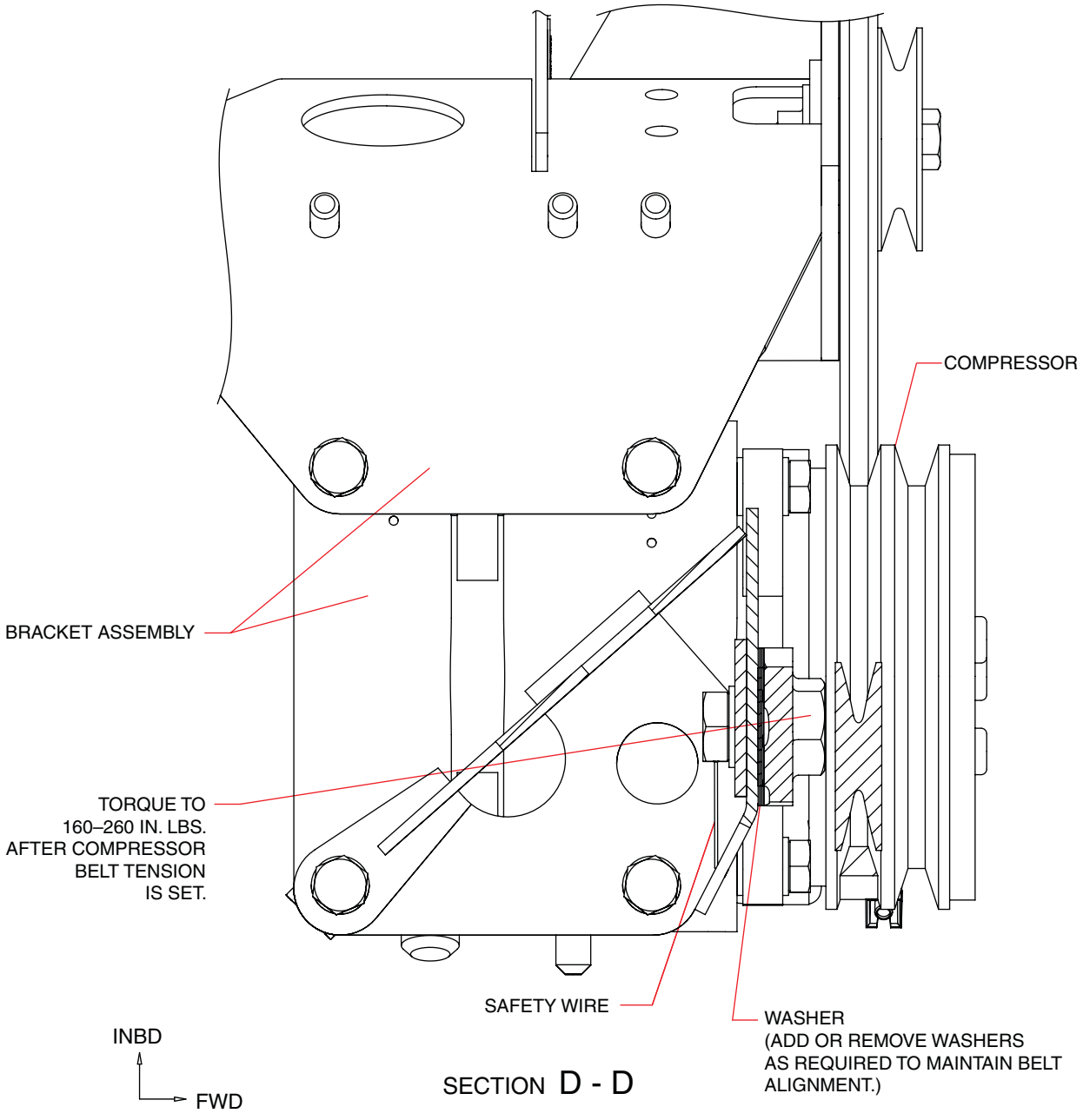


SECTION C - C

Effectivity
2843784 and up,
or 2843001 thru 2843783
with Piper Kit 88551-001

Sanden Compressor Installation
Figure 13 (Sheet 2 of 3)

109020 New



Sanden Compressor Installation
Figure 13 (Sheet 3 of 3)

Effectivity
2843784 and up,
or 2843001 thru 2843783
with Piper Kit 88551-001

(d) Safety all bolts with 0.032 wire.

CAUTION: DO NOT FORCE BELT INTO PULLEY SHEAVE. IF NECESSARY, REMOVE IDLER ASSEMBLY.

(e) Place drive belt over clutch pulley and adjust alignment of pulleys and belt as described in Drive Belt Service, below.

(f) Connect discharge and suction lines to compressor.

(g) Evacuate the system per Evacuating the System, above.

(h) Charge the system per Charging the System, above.

(i) Install right front baffle.

(j) Install engine cowling.

(4) Checking Compressor Oil Level

NOTE: If servicing a charged system recover refrigerant.

(a) Check using a dipstick of a configuration shown in “Figure 14”.

(b) Determine the mounting angle of the compressor by placing a propeller protractor across the flat surfaces of the two front mounting ears.

(c) Read the mounting angle to the closest degree.

(d) If the increments read on the dipstick do not agree with the table, add or subtract oil to obtain mid-range value.

Example: If mounting angle is 10º, add oil in one (1) fluid ounce increments until 6 is read on the dipstick. (Use Piper Code No. 197-511, oil, purch: when adding oil).

(e) Remove the oil filler plug.

(f) Rotate the counterweight located on the front of the clutch until it is at the angle shown in relation to oil filler hole.

(g) Insert the dipstick to its “stop” position which is the angle near the tip of the dipstick.

(h) Point of the angle must be to the left (as viewed from the front) (see “Figure 15”).

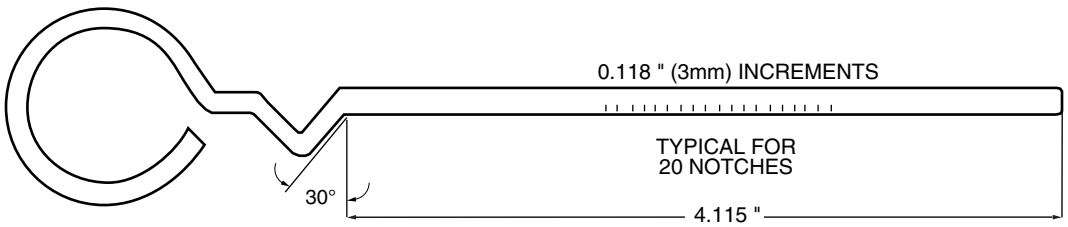
(i) Remove the dipstick. Count increments of oil.

(j) Use mounting angle “Chart 7” below to determine correct oil level for the compressor.

(k) When installing the oil filler plug, make sure that the sealing o-ring is not twisted and that no dirt or other contamination is on the o-ring or seat.

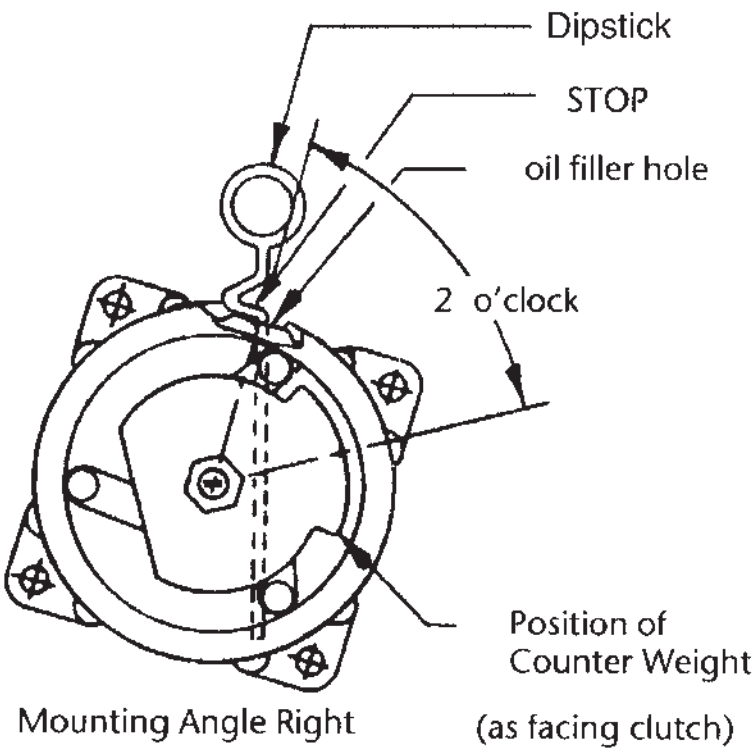
CAUTION: DO NOT OVERTIGHTEN.

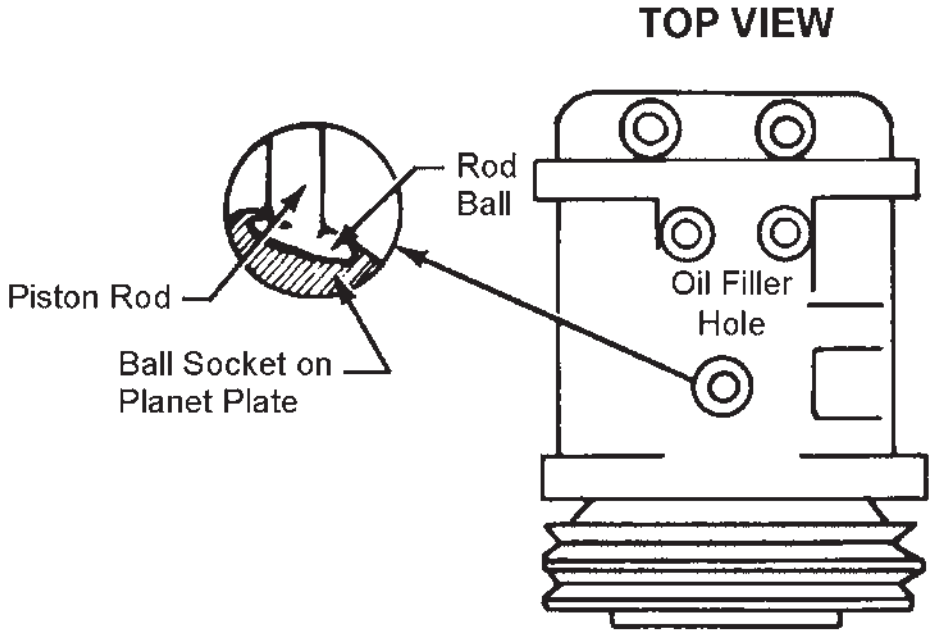
(l) Torque the plug from 5 to 8 foot-pounds (6.8 to 10.8 N-m).



Effectivity
2843784 and up,
or 2843001 thru 2843783
with Piper Kit 88551-001

Sanden Compressor Oil Dipstick
Figure 14





Compressor Mounting Angle and Oil Filler Hole
Figure 15

Effectivity
2843784 and up, or
2843001 thru 2843783
with Piper Kit 88551-001

CHART 7
SANDEN MOUNTING ANGLE OIL LEVELS

Mounting Angle (Degrees)	Acceptable Oil Level In Increments
0	3-5
10	5-7
20	6-8
30	7-9
40	8-10
50	8-10
60	9-11
90	9-11

(m) To stop a leak; replace the o-ring.

I. Drive Belt Service

See “Figure 16”.

(1) Replacement

(a) Remove old belt by removing spinner, propeller, nose cowl, engine baffles as required, starter ring gear assembly, and drive belt.

CAUTION: DO NOT FORCE BELT INTO PULLEY SHEAVE. REMOVE IDLER ASSEMBLIES, IF NECESSARY, AND ALTERNATOR LOWER MOUNTING BOLTS TO INSTALL BELT.

(b) Position new belt on starter ring gear sheaves.

(c) Install starter ring gear assembly, propeller, and spinner.

(d) Route belt to proper pulley sheave per “Figure 16”.

(e) Check compressor and alternator belt and pulley alignment as follows:

CAUTION: VERIFY THE STRAIGHTEDGE HAS SOLID CONTACT WITH THE SURFACE OF THE RING GEAR TO ENSURE TRUE READINGS.

1) Establish a datum line for checking belt and pulley alignment. Establish a nominal dimension between forward edge of compressor belt and forward machined surface of ring gear. This dimension must be taken at ring gear assembly where the belt is in its sheave. Check for offset at the other pulley sheaves by using a stiff straightedge long enough to extend from front of ring gear to component sheaves.

2) Obtain a basic measurement from top of ring gear by measuring the width of starter ring gear plus the dimension from the forward machined surface of ring gear to forward edge of compressor drive belt. (See “Figure 16”.)

3) Alignment and adjustment of compressor and alternator drive belts require different procedures. See below.

(2) Alignment

See “Figure 16”.

(a) Compressor Belt

CAUTION: VERIFY THE STRAIGHTEDGE HAS SOLID CONTACT WITH THE SURFACE OF THE RING GEAR TO ENSURE TRUE READINGS.

1) Place straightedge against right forward side of ring gear and measure belt alignment at compressor sheave (Point-B).

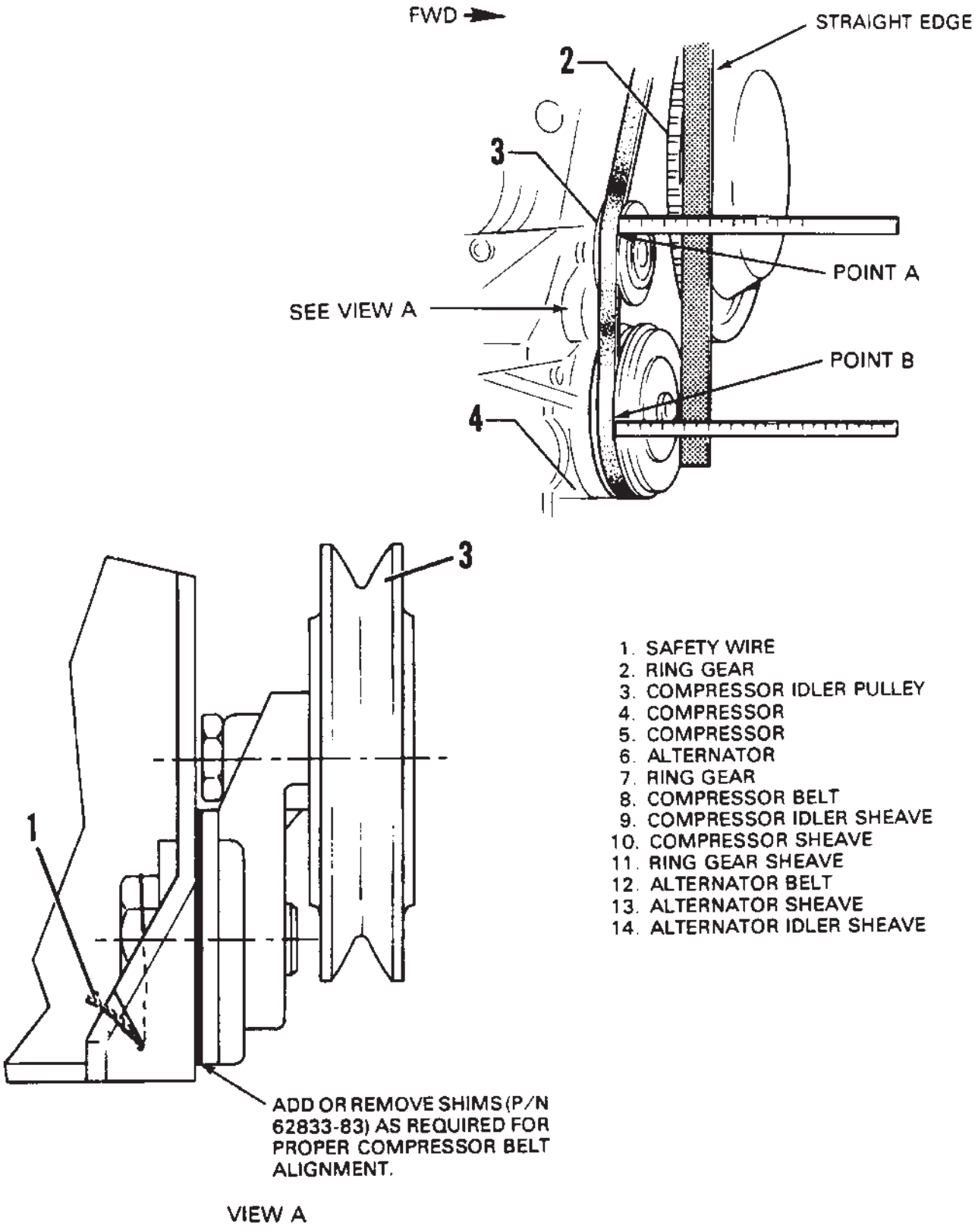
2) Measure belt alignment at compressor idler pulley (Point-A). Belt misalignment at Point-A must be half the offset of Point-B and the dimension at top of ring gear and same direction fore and aft. (See “Chart 8”.)

3) If Point-A nominal offset is not within 0.030 inch, per Step (b), add or remove shims, as shown in “Figure 16”, as required. Belt alignment must be made as close to nominal as shims will allow.

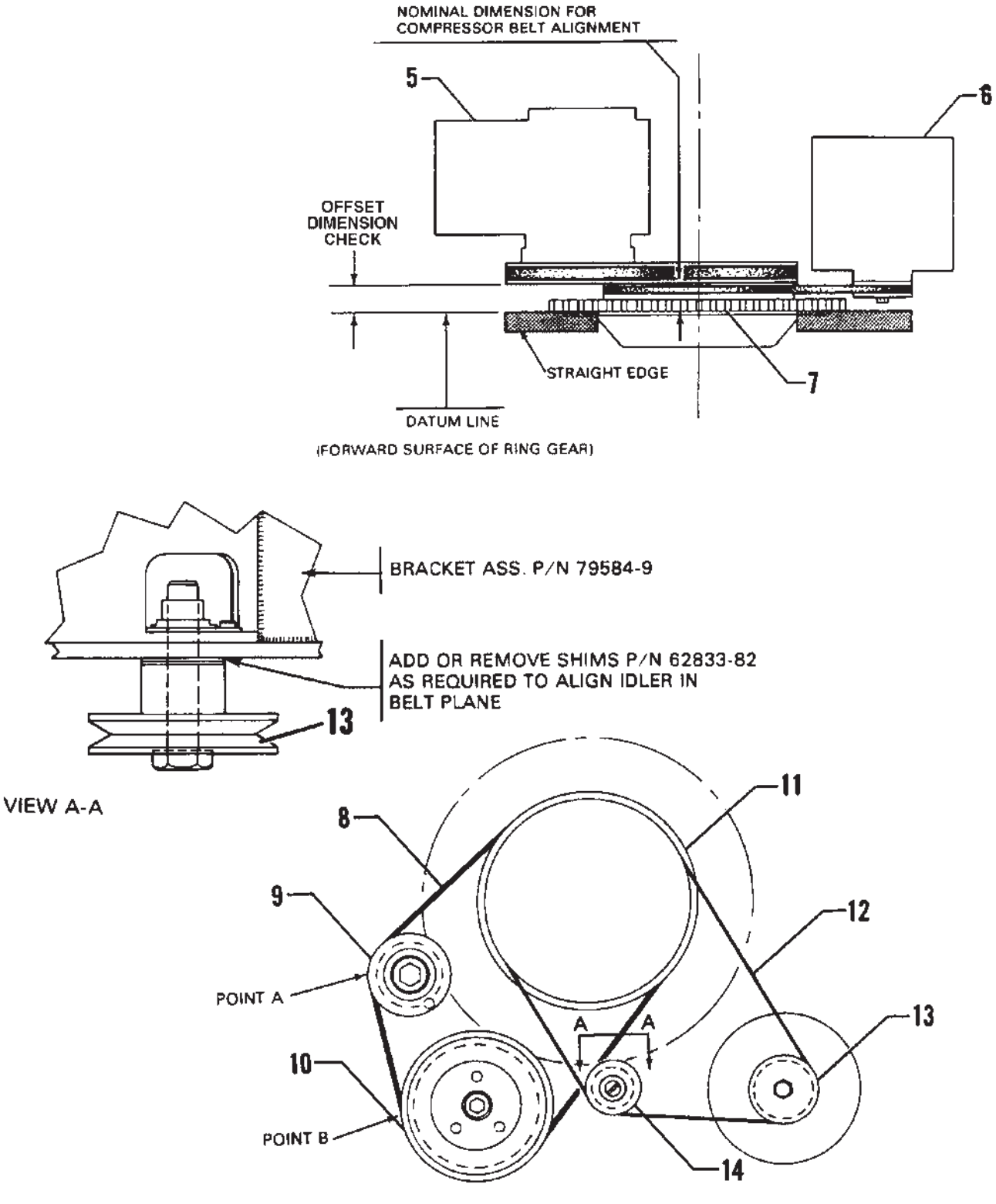
NOTE: If the compressor sheave offset is extreme, it may be adjusted by adding (or removing) washers between the compressor, the compressor mounting bracket, and or the engine, as described in Compressor Installation. If zero offset is not obtainable, note the measured offset.

(b) Alternator Belt

Align idler pulley in the belt plane by adding or removing shims, with alternator belt installed, as shown in “Figure 16”.



Compressor and Alternator Belt Installation
Figure 16 (Sheet 1 of 2)



Compressor and Alternator Belt Installation
Figure 16 (Sheet 2 of 2)

CHART 8
COMPRESSOR OFFSET AND IDLER PULLEY NOMINAL LOCATION

T/CCI (YORK)						
Compressor Pulley Offset From Ring Gear Pulley	0.010	0.020	0.030	0.040	0.050	0.060
Idler Pulley Nominal Location (in notches)	0.006	0.011	0.017	0.022	0.028	0.033
SANDEN						
Compressor Pulley Offset From Ring Gear Pulley		0.020	0.030	0.040	0.050	0.060
Idler Pulley Nominal Location (in inches)		0.010	0.015	0.020	0.025	0.030

(3) Adjustment of Drive Belt Tension

Adjustment of the compressor and/or alternator drive belts is very important to obtain long belt life and proper component operation. Using a belt tensiometer, adjust the belt tensions as follows:

NOTE: P/N's shown are for reference only. Always refer to the Parts Catalog, P/N 761-898, before ordering new belts.

(a) Compressor

1) In-service belts:

Part Number	Tension (Lbs.)
73965-010	73 to 87
73965-020	50 to 70

2) New belts:

NOTE: The high tensions specified for new belts compensates for the initial stretch that takes place as soon as they are operated. Do not apply these higher tension values to belts which have been used previously.

Part Number	Tension (Lbs.)
73965-010	73 to 87
73965-020	50 to 70

3) Torque compressor idler pivot bolt to as shown in "Figure 13" on page 215039 and safety after compressor belt tension is set.

(b) Alternator Belts

See Alternator Belt Tension Adjustment in 24-30-00.

(c) Install engine baffles if removed and install engine cowl.

CAUTION: IF THE AIR CONDITIONER IS TO BE OPERATED DURING GROUND SERVICING, THE TEST AREA SHOULD BE CLEAN AND FREE OF ANY LOOSE OBJECTS LYING ON THE RAMP.

(d) Run the engine for 15 minutes at 1200 rpm.

(e) Shut down the engine and recheck the belt tensions per above.

(f) Check compressor / alternator belt tension every 100 hours or annually, whichever occurs first.

(g) Check all idler and bracket bolts for safety and replace the engine cowling.

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J. Refrigerant Lines And Routing

CAUTION: DISCHARGE SYSTEM COMPLETELY BEFORE HOSE COUPLINGS ARE UNCOUPLED. (SEE DISCHARGING, ABOVE.)

NOTE: United States environmental regulations prohibit the release of refrigerant into the atmosphere. Special equipment is required when discharging or recharging system.

NOTE: If performing the initial charge after changing the lines that feed the condenser, add a minimum two (2) ounces of oil to the system in addition to any oil that was recovered during the evacuation procedure.

(1) Description

The refrigerant lines in this airplane are flexible, high-pressure hoses and should be handled carefully. The hoses in the power plant area are routed so as to provide maximum protection from heat and abrasion.

They couple at the firewall to hoses routed through the two inboard, external hat sections on bottom of fuselage, up through floor to condenser and evaporator in tail cone. Discharge is in the right hand hat section. The suction is in the left hand hat section.

(2) Repair / Replacement

The air conditioning compressor was changed in 2014. Hose material was changed in 2016. Accordingly, repair/replacement of hoses and hose fittings is accomplished as follows:

(a) S/N's 2843001 thru 2843822

Three kits are available.

1) Piper Kit 88555-001 is used to repair fuselage suction lines damaged or leaking at the evaporator hose fitting.

2) Piper Kit 88563-001 is used to repair fuselage suction lines damaged or leaking in the middle hose section at least eight inches from any end fitting.

3) Piper Kit 88571 in S/N's 2843001 thru 2843783 only,

88571-001 replaces fuselage discharge lines.

88571-002 replaces fuselage suction lines.

88571-003 replaces compressor lines, but requires that the 88551-001 A/C Compressor Replacement Kit and the 88571-001 and 88571-002 kits be installed first.

88571-004 replaces all system lines, but requires that the 88551-001 A/C Compressor Replacement Kit be installed first.

(b) S/N's 2843823 and up (PIR-PPS20029, Rev. B.)

Aeroquip Nylon Veneer Tube hoses are installed. Leaking hose fittings may be replaced as follows:

1) Required Equipment

Aeroquip Plier Connecting Tool FT1357. Widely available from various suppliers.

2) Assembly Procedure

a) Consult the appropriate Piper Parts Catalog, and order the appropriate replacement fitting, cage, and clips. (See "Figure 17")

b) Prior to installing the fittings on the hose, ensure the hose is cut such that the cut is made square to the hose length.

c) Install two proper sized clips onto the cut end of the hose ("Figure 18" Item A).

d) Lubricate the nipple with a generous amount of Oil, Air Conditioner, HFC-134a, POE (see 91-10-00, Chart 10.).

NOTE: This must be done to lower the force of nipple insertion.

e) Insert the nipple into the hose: (“Figure 18”, Item B)

1] There should be no gap between the end of the hose and the shoulder on the nipple.

2] Avoid kinking or damaging the hose while inserting nipple.

f) Snap the cage into the groove on the nipple. (“Figure 18”, Item C)

NOTE: When the cage has been correctly installed in the cage groove, the cage will be able to rotate in the groove.

g) Slide the clips over the cage arms and into the channels on each arm (“Figure 18”, Item D).

h) Using Aeroquip tool FT1357, close the clips (“Figure 18”, Item E).

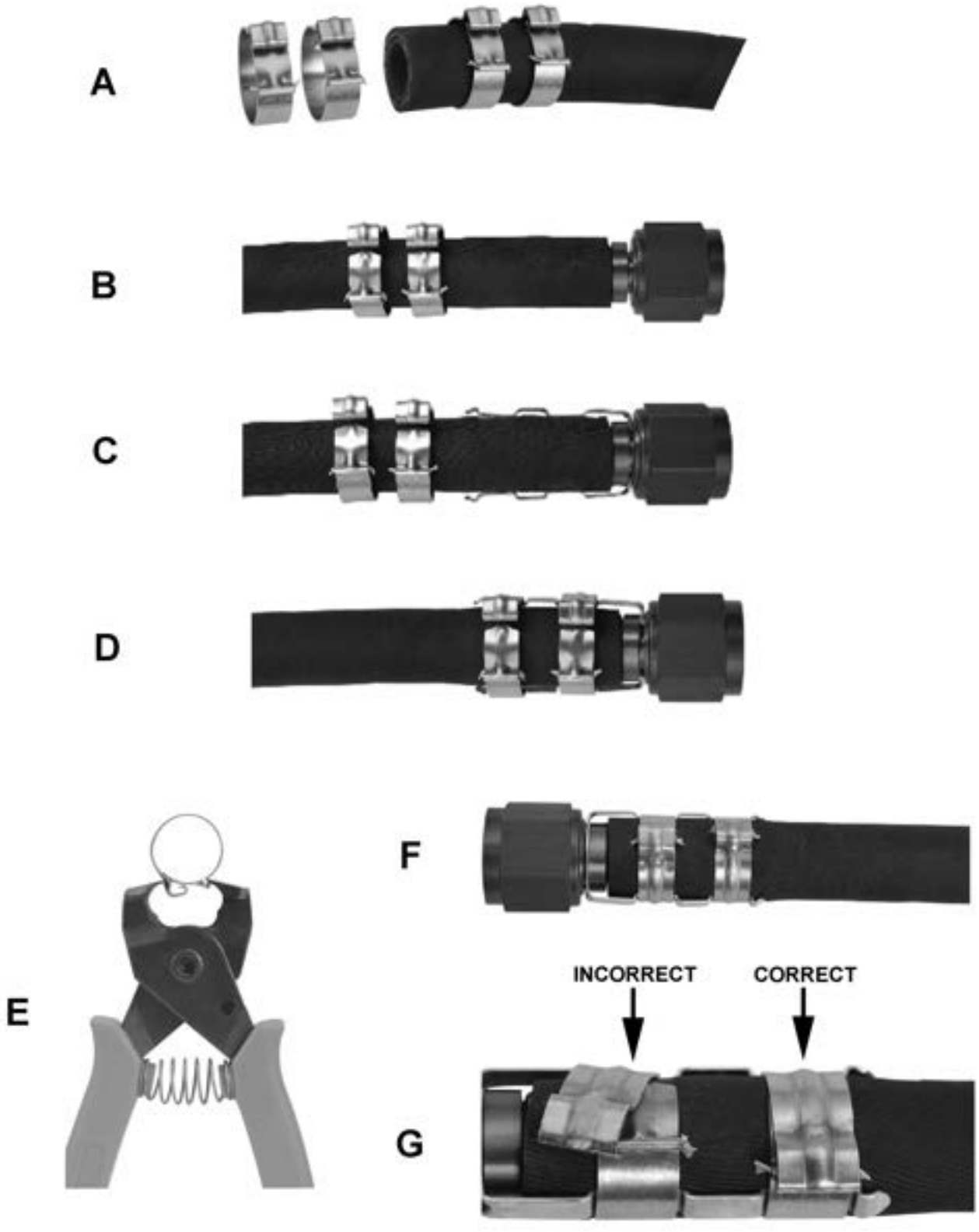
NOTE: The tool should be positioned squarely on the clip connection points and should remain square during the closing of the clip.

i) Inspect finished assembly for correct installation (“Figure 18”, Items F and G).



A/C Hose Fitting Installation
Figure 17

Effectivity
2843823 and up



Effectivity
2843823 and up

A/C Hose Fitting Assembly
Figure 18

K. Receiver-Dehydrator

See “Figure 19” on page 215052.

The unit is mounted on inboard side of evaporator assembly housing.

(1) Removal

CAUTION: IF RECEIVER-DEHYDRATOR IS NOT SERVICEABLE, IT MUST BE REPLACED. RECEIVER-DEHYDRATOR MUST BE REPLACED WHEN SYSTEM HAS OPERATED WITHOUT A CHARGE OR HAS BEEN LEFT OPEN.

(a) Discharge system of all refrigerant. (See Discharging above.)

(b) Uncouple refrigerant lines at receiver-dehydrator. (See Special Servicing Procedures above.)

(c) Remove clamp attaching unit to evaporator housing.

(2) Installation

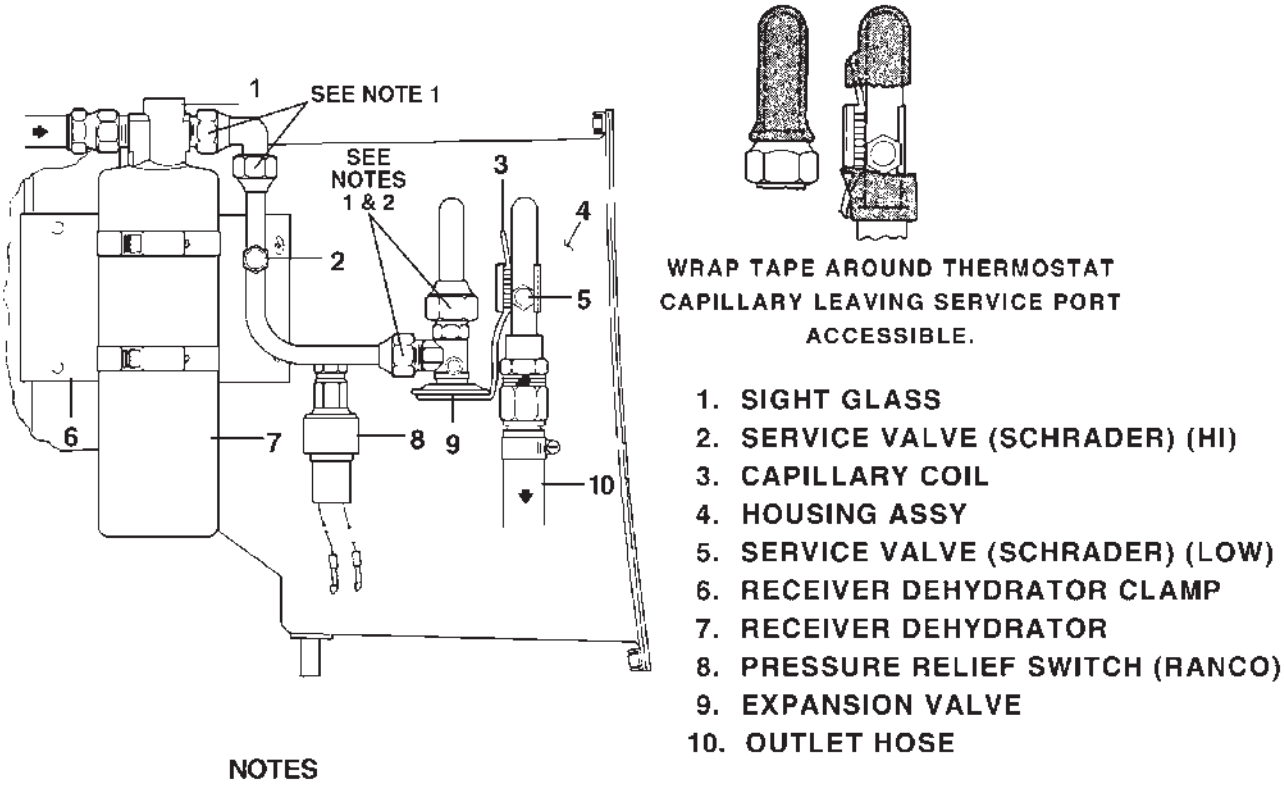
Use only receiver-dehydrators marked with a GREEN arrow.

(a) Slip mounting bracket around receiver and put it in place on evaporator housing with tube fitting on top. Align fittings to proper line before securing mounting bracket.

(b) Replace O-rings.

(c) Tighten fittings to torque listed in “Chart 3”.

(d) Evacuate and charge system per Evacuating the System and Charging the System.



Component Installation
Figure 19

L. Expansion Valve

See “Figure 19” and “Figure 20” on page 215053.

(1) Removal

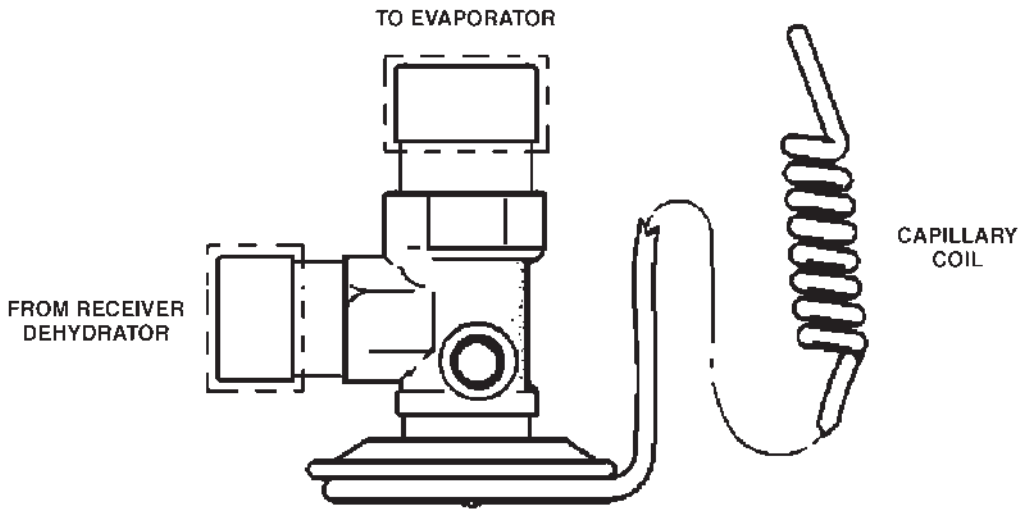
The expansion valve is in evaporator assembly between receiver dehydrator and evaporator inlet. The capillary coil is attached to evaporator outlet line.

NOTE: If expansion valve is not serviceable, it must be replaced with a new part.

(a) Remove access panels, and discharge system. (See Discharging, above.)
(b) Remove capillary coil from outlet line. (Do not kink capillary tube.)
(c) Uncouple all related tube fittings. (See Special Servicing Procedures above.)

(2) Installation

(a) Install expansion valve in inlet line of evaporator core. Apply P.A.G. lubricant on O-rings and replace O-rings on fittings, torque fittings per “Chart 3” on page 215014.
(b) Secure capillary coil to evaporator outlet line.
(c) Evacuate and charge system per above
(d) Check for leaks. (See leak detection.)
(e) Replace access panels.



Expansion Valve (Typical)
Figure 20

M. Evaporator

Evaporator assembly consists of evaporator core, receiver-dehydrator, expansion valve, circulating fan, pressure switch, necessary housing, and plumbing. The housing is made of thermoplastic material and the condensed moisture is dumped overboard through a hose clamped to fitting on bottom of evaporator housing.

Evaporator assembly is behind cabin rear closeout panel, attached to mounting panel with 12 screws, washers, and a bracket securing the back to mounting panel.

(1) Removal

CAUTION: DISCHARGE THE SYSTEM BEFORE DISASSEMBLING ANY COMPONENTS FOR SERVICE.

NOTE: United States environmental regulations prohibit the release of refrigerant into the atmosphere. Special equipment is required when discharging or recharging system.

(a) Remove air conditioning filter cover, filter, and rear access panels.

(b) Uncouple the liquid line from inlet side of receiver-dehydrator and suction line from evaporator core outlet (see Special Servicing Procedures, above).

(c) Disconnect related electrical wires.

(d) Remove flexible air duct from housing outlet and remove drain hose from housing.

(e) Remove temperature probe from evaporator housing.

(f) Remove screws attaching support bracket and evaporator housing to mounting panel.

(g) Remove assembly through access hole in bulkhead.

(2) Installation

(a) Cement gasket in place on flanges of evaporator housing and attach large end of mounting gasket to back of housing.

(b) Install housing through access hole with air duct outlet on top and mate mounting flanges to surface of mounting panel and insert screws (Do not tighten at this time).

(c) Line mounting bracket with mating holes in mounting panel, insert screws and tighten. Tighten screws in flange and check that the gasket is in place, flange seal must be air tight.

(d) Couple suction and discharge lines to their proper fittings (apply Loctite refrigerant sealant to tube flares only).

(e) Evacuate and charge system per above.

(f) Check for leaks (See Leak Detection,above). If no leaks are detected, seal, and install access panel on evaporator housing.

(g) Couple flexible air duct and drain tube.

(h) Make and check electrical connections. (See 91-21-50.)

(i) Check blower operation and refrigerant systems.

(j) Install rear closeout panel.

N. Pressure Relief Switch

CAUTION: BEFORE RELIEF SWITCH REMOVAL, AIR CONDITIONING SYSTEM MUST BE DISCHARGED. (REFER TO DISCHARGING.)

NOTE: United States environmental regulations prohibit the release of refrigerant into the atmosphere. Special equipment is required when discharging or recharging system.

(1) Removal

(a) Remove electrical connections from switch.

(b) Remove switch assembly from service port on steel line.

(2) Installation

(a) Apply sealant sparingly to flare. Lube O-ring with PAG or POE oil.

(b) Install new switch.

(c) Charge system per above.

O. Condenser

(1) Removal

Condenser is mounted in a frame assembly in fuselage bottom between stations 156.00 and 191.00.

(a) Discharge system per above. (See also Special Servicing Procedures, above.)

(b) Remove access panel from aft bulkhead of cabin.

(c) Remove condenser forward cover panel.

(d) Uncouple suction and discharge hoses at condenser fitting. (See Special Servicing Procedures, above.) Remove hose clamps holding hoses to condenser frame.

(e) Remove AN-3 bolts from upper ends of side hinges and rod ends.

(f) Support condenser assembly and remove bolt attaching actuating rod to condenser assembly.

(g) Lower aft end of assembly on the piano hinge at assembly forward end.

(h) Remove eight screws attaching piano hinge to condenser frame assembly and remove.

(i) To remove condenser core from assembly, remove screws in the side mounting frame.

(2) Installation

(a) Install condenser core to frame assembly with hose fittings forward and RT fitting pointed inboard.

(b) Place condenser and frame assembly to fuselage frame mounting bracket and insert the eight screws into piano hinge.

(c) Attach side hinges, actuating rod, and rig per condenser assembly rigging instructions.

(d) Seal and couple hose fittings (seal with Loctite refrigerant sealant applied to flares only).

(e) Adjust condenser per condenser assembly rigging instructions, below.

WARNING: EXHAUST GASES CAN SEEP INTO CABIN, IF CONDENSER COVER PANEL(S) ARE NOT SEALED PROPERLY. CONDENSER COVER PANEL(S) MUST BE REPLACED AND SEALED IN THE ORIGINAL MANNER.

WARNING: TEST FOR CARBON MONOXIDE (CO) ON GROUND AND IN FLIGHT WITH AND WITHOUT AIR CONDITIONER OPERATING. PRESENCE OF CO MUST NOT EXCEED ONE (1) PART IN 20,000.

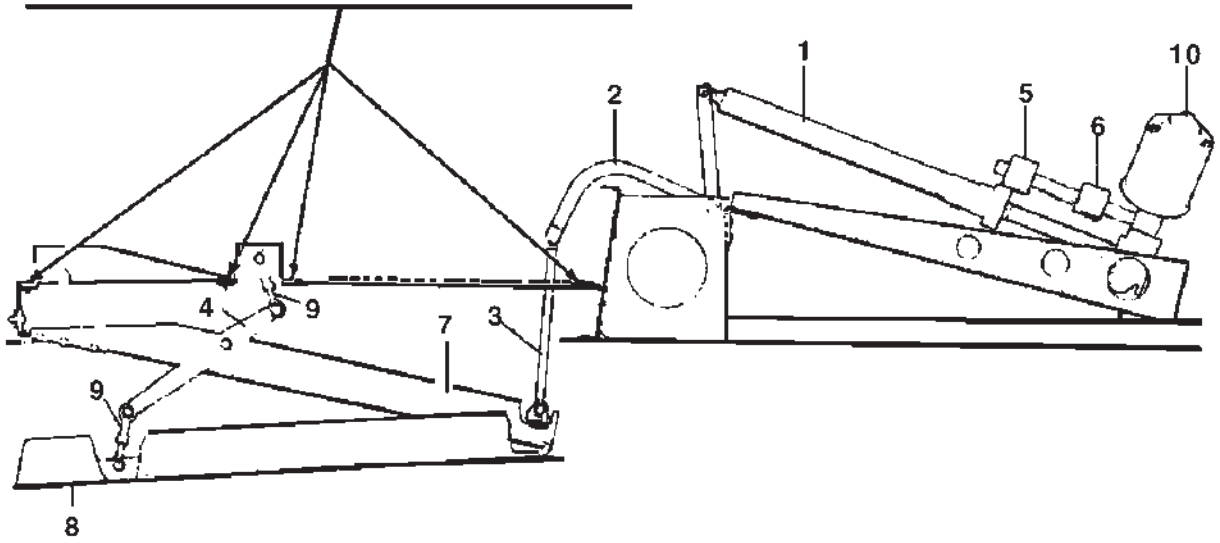
(f) Seal around condenser forward cover panel (and aft cover panel if removed) with Permagum Bead No. 576 purchased from Prestolite Engineering Company. See “Figure 21” on page 215056.

(3) Condenser Door Actuator

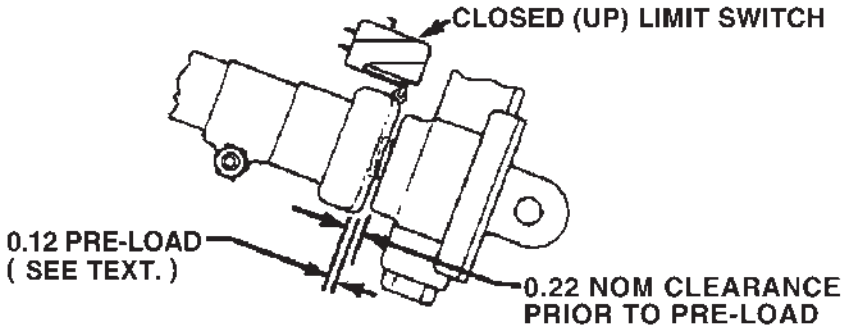
The actuator is on a bracket mounted between two bulkheads in tail cone. It is coupled to the condenser assembly through a bellcrank mounted to a bracket on bulkhead aft of condenser. Actuator travel is controlled by two limit switches. Both up and down switches are on the actuator. See “Figure 21” on page 215056 for switch locations.

99855 AB

SEAL ALL AROUND FORWARD AND AFT COVERS WITH PERMAGUM BEAD NO. 576 PURCHASED FROM PRESTOLITE ENGINEERING COMPANY.



1. ACTUATING TRANSMISSION ASSY.
2. BELLCRANK ASSY. (CONDENSER)
3. PUSH ROD ASSY.
4. BELLCRANK ASSY. (MECHANISM)
5. OPEN LIMIT SWITCH
6. CLOSED LIMIT SWITCH
7. CONDENSER
8. CONDENSER DOOR
9. PUSH ROD
10. TRANSMISSION MOTOR ASSY.



CLOSED LIMIT SWITCH ADJUSTMENT

Condenser Air Scoop Installation
Figure 21

(4) Condenser Assembly Rigging Instructions

See “Figure 21” on page 215056.

Condenser assembly is actuated by an electric motor through bellcranks, push rods, and limit switches. Condenser door must fit flush with fuselage skin, and with increased force along forward edge. Use the following steps:

(a) Adjust open limit switch to open condenser door 5.00 ± 0.50 inches measured from leading edge of door to fuselage skin.

(b) Adjust side push rods so a vertically measured gap of 0.16 inch exists along trailing edge of door the instant forward edge of door is flush with fuselage skin.

(c) Fully close door and adjust CLOSED limit switch so actuator travels an additional 0.12 inch with door fully closed - this is necessary to preload mechanism.

(d) Cycle assembly several times. Verify proper operation without binding.

P. Electrical Installation

The wiring harness is connected to switches in the climate control center on the right side of the instrument panel. The harnesses cross the instrument panel to the left side where two wires are taken off for the compressor clutch. The harness then passes aft along the left side of fuselage connecting to the blower motor, the pressure relief switch, and the condenser actuating motor. Two fuses behind the air conditioning system control panel and a 20 amp circuit breaker mounted in circuit breaker panel protect the complete air conditioning electrical system.

(1) Adjustment of Throttle Switch

The throttle switch is mounted forward and below the throttle arm. The switch must be adjusted to actuate at the last quarter inch of full open throttle travel. Position the switch so that the throttle arm contacts the center of the switch actuator button.

(2) Fuse Replacement

Locate the fuse to be replaced behind the air conditioning system control panel.

(a) Open the fuse holder by applying a slight pushing, counterclockwise twisting, pressure.

(b) Remove blown fuse and insert a new 5 amp fuse.

(c) Close the fuse holder by applying a slight pushing and clockwise twisting pressure.

(3) Electrical Schematic

See 91-21-50.

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CHAPTER

22

AUTOFLIGHT

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CHAPTER 22

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AUTOPILOT

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. General

An Autopilot/Flight Director (APFD) system is optional in the PA-28-181 Archer III airplane.

A. S/N's 2843001 thru 2843202 (1995–1998)

A King Autopilot/Flight Director (A.P.F.D.), manufactured by then Allied Signal (now Honeywell), was installed in these airplanes. Maintenance information for those systems is not included in this manual. Follow the service literature published by the A.P.F.D. equipment manufacturer. This includes mechanical service such as: adjusting bridle cable tension, servo removal and installation, servo clutch adjustments, etc.

Honeywell (King/Allied Signal) technical support, parts support, and service literature can be obtained as described under Supplementary Publications, Vendor Publications, in the Introduction.

B. S/N's 2843203 and up (1999 and up)

The S-TEC System 55 / 55X is installed in these airplanes. Maintenance information for this system is provided in "S-TEC System 55 / 55X", below.

C. S/N's 2843823, 2881001 and up (2017 and up)

The Garmin GFC 700 Automatic Flight Control System (AFCS) is available in Garmin G1000 NXi-equipped airplanes S/N's 2843823, 2881001 and up. When installed, it fully integrats into the G1000 NXi system which is installed in these airplanes. Maintenance information for this system is provided in "Garmin GFC 700 Automatic Flight Control System (AFCS) (Optional)" on page 221027.

2. S-TEC System 55 / 55X

A. S-TEC System 55

This system was adopted in 1999 and is installed as optional equipment in S/N's 2843203 thru 2843414 only.

B. S-TEC System 55X

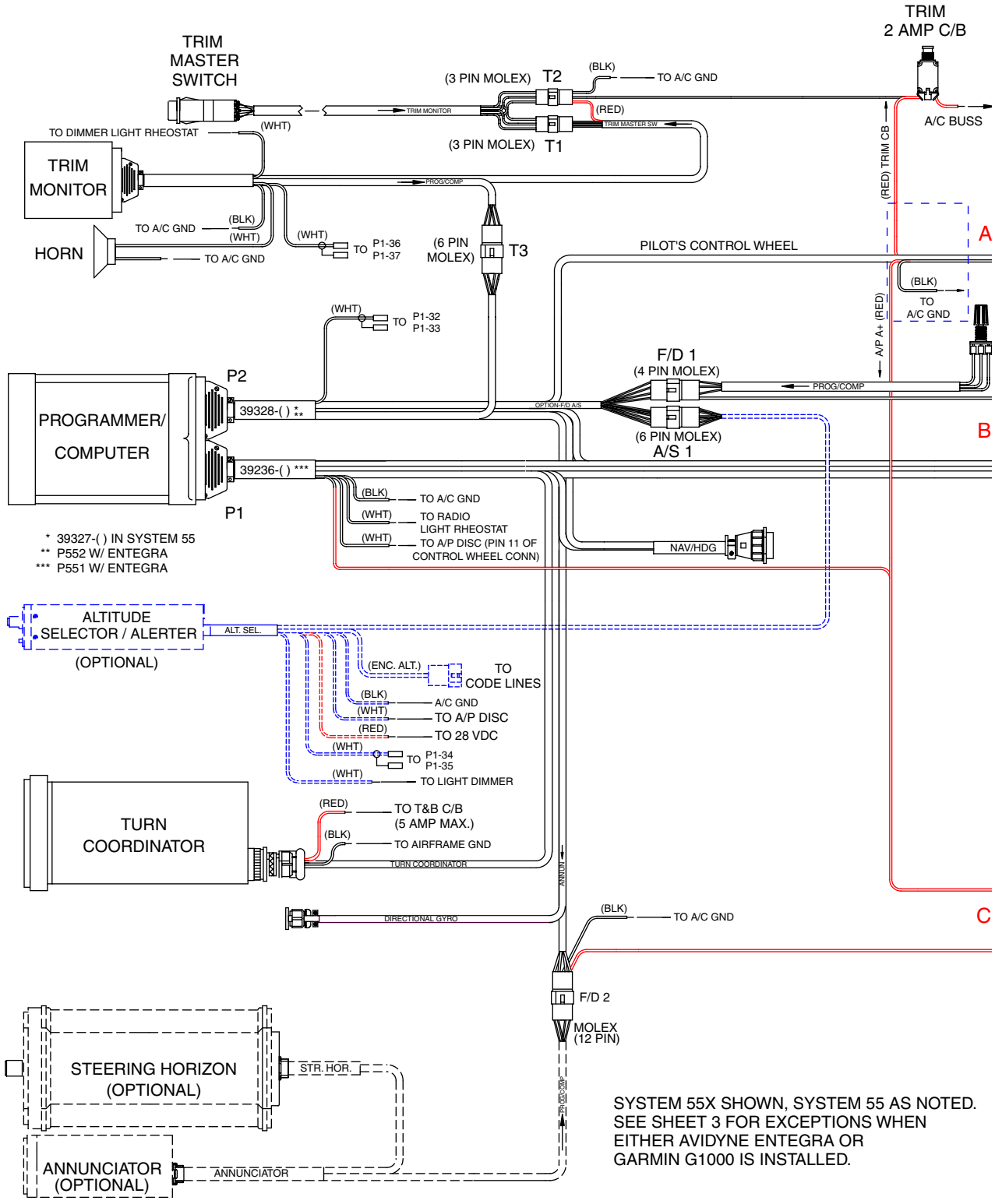
This system was adopted in 2001 and is installed as optional equipment in S/N's 2843415 and up.

C. Description

See "Figure 1" on page 22102.

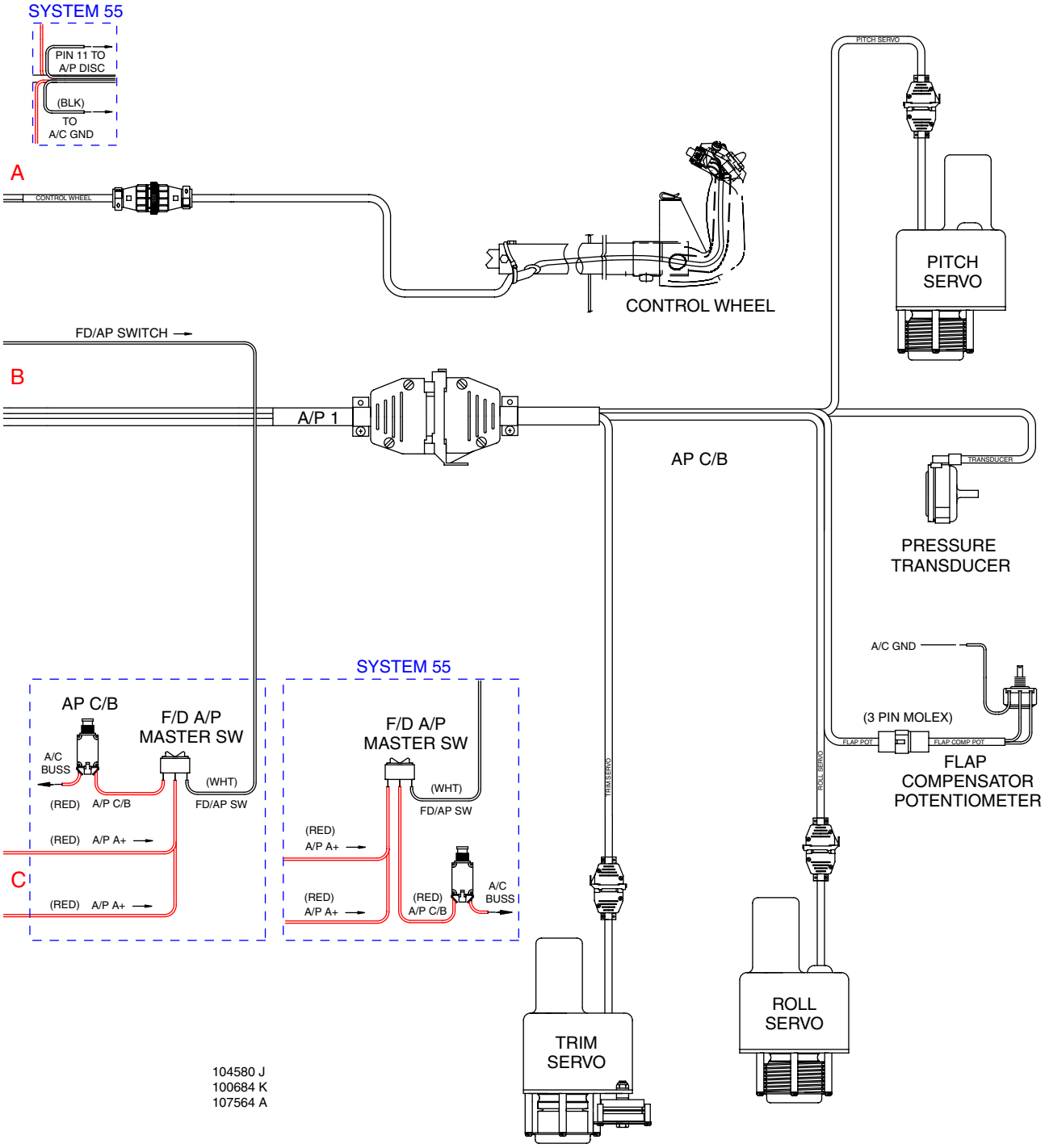
The S-TEC System 55/55X is a rate based autopilot that controls the roll and pitch axis of the aircraft. The autopilot's main function is to convert pilot commands to logic signals for the roll and pitch computers. As the pilot enters the desired mode by pressing the appropriate mode selector switch, the computer acknowledges the mode, causing the appropriate annunciator to illuminate.

The Roll Computer receives select input signals from the Directional Gyro (DG) or Horizontal Situation Indicator (HSI), VHF Omnidirectional Radio (VOR), Localizer (LOC) or Global Positioning System (GPS), Deviation Indicators, and the Turn Coordinator. It then computes roll servo commands for stabilization, turns, navigation intercepts, and tracking.



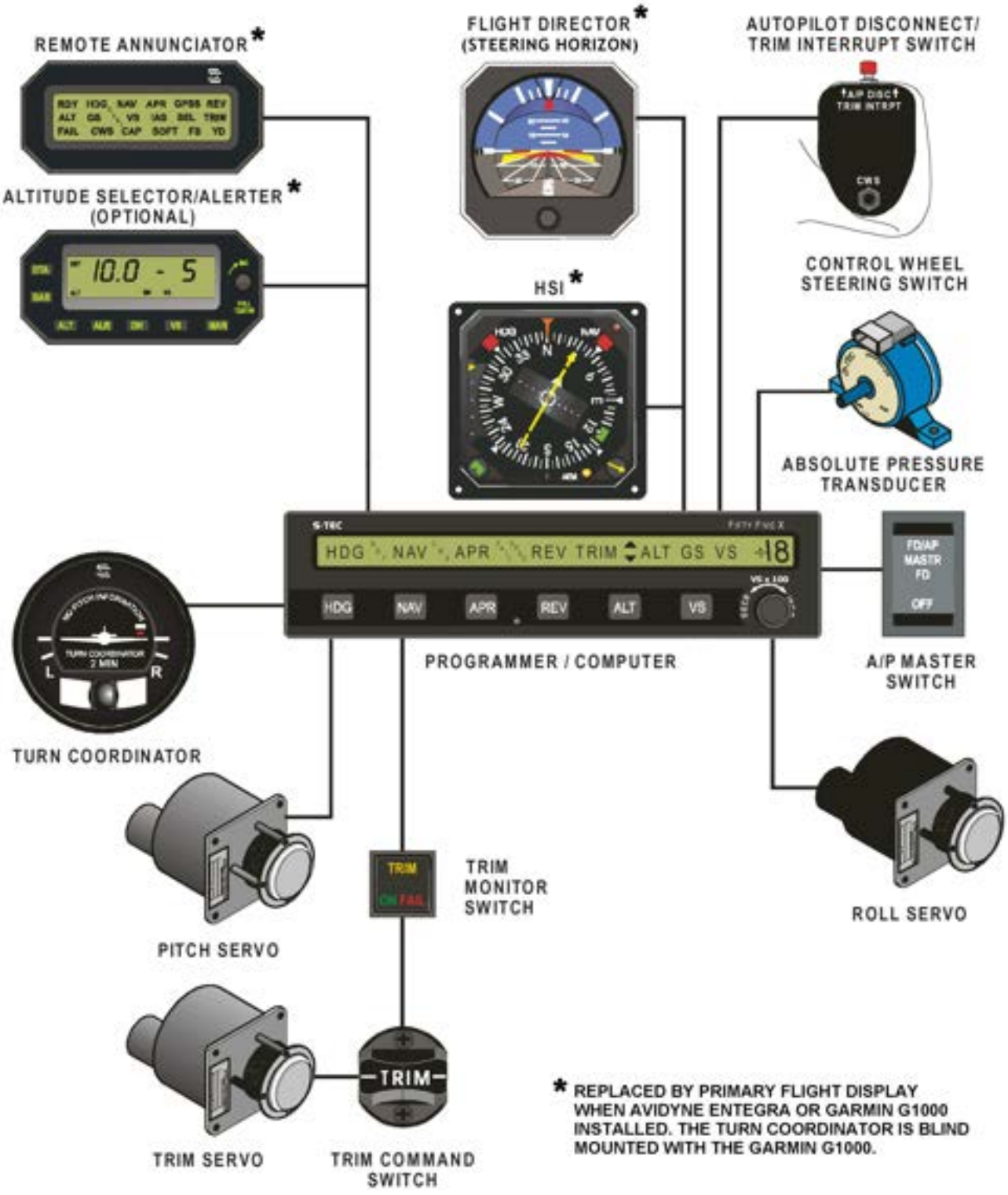
Effectivity
2843203 and up

System 55 / 55X Autopilot Installation
Figure 1 (Sheet 1 of 3)



System 55 / 55X Autopilot Installation
Figure 1 (Sheet 2 of 3)

Effectivity
2843203 and up



Effectivity
2843203 and up

System 55 / 55X Autopilot Installation
Figure 1 (Sheet 3 of 3)

The Pitch Computer receives select input signals from the Altitude Pressure Transducer, Accelerometer, Glideslope Deviation Indicator and Altitude Selector/Alerter (if installed). It then computes pitch servo commands for vertical speed, altitude hold and glideslope intercept and tracking. Sensing for trim annunciation or automatic stabilator trim is provided by the pitch servo. Drive for the stabilator trim servo is provided by the pitch computer.

A typical S-TEC System 55/55X Autopilot installation includes the following:

(1) Panel Mounted

Programmer/Computer, Turn Coordinator, Annunciator, D.G. or HSI, and Steering Horizon.

(2) Remote Mounted

Roll Servo, Pitch Servo, Trim Servo, Trim Monitor, A/P Disconnect switch, and Altitude (Pressure) Transducer.

Servo installations use aluminum brackets to secure the servos to the airframe. Attachment to the airplane's primary flight control and trim systems is accomplished with bridle cables and extension attachments.

D. Troubleshooting

System functionality can be determined using functional checks described in the AFM Supplement and autopilot Pilot's Operating Handbook. More detailed troubleshooting should be accomplished by authorized S-TEC Dealers, holding the appropriate FAA certification, with required test equipment and service data.

E. GPSS (System 55X only)

The Global Positioning System Steering (GPSS) is a function of the 55X autopilot only. In the GPSS mode, the converter receives ground speed and bank angle digital signals that are calculated and converted to a commanded turn rate. The turn rate is then scaled and converted to a DC heading error signal that is compatible with S-TEC autopilots. The end result is an autopilot that can be directly coupled to the roll steering commands produced by the GPS Navigator, eliminating the need for the pilot to make any further adjustments to the HSI course arrow or the DG's heading bug.

F. System Operation

Operation of the autopilot and other systems is described in the FAA-approved Airplane Flight Manual Supplement (AFMS) - see airplane Pilot's Operating Handbook (POH), Section 9. Specialized controls, annunciation, operation and interpretation are covered in this supplement and in the S-TEC Autopilot POH that supplements the approved AFMS.

G. Maintenance

Except as provided in 5-20-00, servicing and/or maintenance of the autopilot system is On-Condition.

NOTE: Servicing of S-TEC System 55/55X Autopilot installations is best accomplished by approved S-TEC dealers holding the appropriate FAA-certification. Locations of and access to the components installed are described and depicted individually below. Removal and replacement of components is generally indicated by functional checks provided in the AFM Supplement, S-TEC Autopilot POH and/or below.

H. Ground Checks

Complete the appropriate checkout procedure after any maintenance to the system is performed.

NOTE: The Systems 55/55X incorporate a SELF-TEST that requires a 100% pass rate before the autopilot can be engaged.

(1) Standard, Avidyne Entegra, and Garmin G500 Equipped Airplanes

NOTE: For airplanes equipped with Avidyne Entegra (see 34-22-00), references below to the remote annunciator, flight director (FD), and HSI are to those functions in the Primary Flight Display (PFD).

(a) Apply aircraft power.

(b) Avionics Master Switch — ON

(c) Autopilot Master Switch — Set to FD / AP

NOTE: Observe that all segments of the Programmer / Computer display and annunciators illuminate for five (5) seconds during test. Satisfactory completion of the SELF-TEST is indicated when the Ready (RDY) annunciator remains on at the end of the five (5) second self-test. Should a fault be detected, the FAIL annunciator will remain on at the conclusion of the self-test and the autopilot will not operate.

(d) Trim Master (ON / OFF) Switch — ON

(e) HDG and VS switches — PRESS / RELEASE

Ensure that HDG and VS illuminate on the Fifty Five X annunciator.

(f) VS Knob — ROTATE CW

Pitch control (i.e. - the control yoke) should move slowly out (pilot may have to assist a heavy yoke).

(g) VS Knob — ROTATE CCW

Pitch control should move slowly in.

(h) A/P DISC Trim Interrupt Switch (on control yoke) — PRESS

Verify the autopilot disconnects.

(i) HDG Mode — ENGAGE

(j) DG or HSI HDG bug — MOVE LT / RT

Roll control should follow the HDG bug.

NOTE: If HSI equipped, center the course arrow under the lubber line and push the NAV button. Move the course arrow on the HSI left then right. Roll control should follow the course arrow. Channel a valid VOR signal and move course arrow just enough to deflect the left / right needle one (1) or two (2) dots. Roll control should follow the Course Deviation Indicator (CDI) left / right needle during the test. (This test is only valid if the left / right needle is centered with the course arrow under the lubber line.)

NOTE: If DG equipped, center the HDG bug under the lubber line. Channel a valid VOR signal. Move the OBS to cause left / right CDI needle deflection. The roll control should follow the left / right needle movement.

(k) REV Mode button — PUSH

Roll control should respond opposite to the course arrow and CDI left / right needle inputs.

(l) Altitude Hold (ALT) button — PUSH

Slowly pull out (nose up) on the pitch control (i.e. - control yoke). Autotrim should run nose down with TRIM flashing on the remote annunciator and the autopilot computer / programmer after approximately 3 seconds. Slowly move control yoke forward (nose down). After 3 seconds, autotrim should move nose up with TRIM flashing on the remote annunciator and the autopilot computer / programmer after approximately 3 seconds.

(m) Trim Master (ON / OFF) Switch — OFF

(n) Manual Electric Trim Test

1) Trim Master (ON / OFF) Switch — ON

a) Move each segment of the Manual Electric Trim Command Switch FWD and AFT. Trim should not run.

b) Move both segments of the Trim Command switch FWD. Trim should run nose down.

c) Move both segments of the Trim Command switch AFT. Trim should run nose up.

2) Re-trim aircraft for takeoff and check controls for freedom of movement. Be sure the autopilot and trim servos are dis-engaged.

(o) Flight Director Test

3) Autopilot Master Switch — SELECT FD

Note the roll, pitch and trim servos are disengaged. The steering bar should be in view on the attitude indicator.

4) HDG Mode — ENGAGE

MOVE HDG bug 45 degrees left. The roll steering bar should slowly indicate a left steering command. Repeat the same test for the right side.

5) VS Mode — ENGAGE

SELECT 1500 FPM rate of climb. Note the pitch steering bar moves slowly up. Repeat the same test for the down direction.

6) Autopilot Master Switch — SELECT FD / AP

The servos should re-engage.

7) Trim Master ON / OFF Switch — ON

8) Manual Electric Trim Command Switch — MOVE FWD or AFT

The autopilot should disconnect.

NOTE: The Manual Electric Trim Command Switch will disconnect the autopilot only if there is a Pitch Mode engaged.

(2) Garmin G1000 Equipped Airplanes (S/N's 2843701, 2843703 and up)

(a) Autopilot Power-Up Test

1) Select the AVION MASTR switch to ON.

2) Select AP master switch to "FD/AP".

3) Verify RDY is displayed on AP controller and the PFD AFCS status box within 3 minutes.

4) Pull AUTO PILOT circuit breaker.

5) Verify AP controller display is blank.

6) Select AP master switch OFF. Reset all circuit breakers.

(b) Autopilot Engagement Test

1) Select AP master switch to "FD/AP".

2) Verify RDY is displayed on AP controller and PFD AFCS status box within 3 minutes.

3) Press the HDG and ALT buttons on the AP computer.
4) Verify HDG and ALT are displayed on AP computer and the PFD AFCS status box.
5) Verify the pitch and roll clutches engage and resist movement of the control wheels.
6) Press the AP DISC switch on the pilot's control wheel.
7) Verify the autopilot disengages, accompanied by an annunciation on the AP computer and an aural alert.
8) Verify that the control wheels have free motion in the roll and pitch axes.
9) Select AP master switch to "FD".
10) Select the heading and altitude modes by pressing HDG and ALT buttons on the AP computer.
11) Verify that HDG and ALT are displayed on the AP computer.
12) Verify that FD, HDG, and ALT are displayed on the PFD AFCS status box, and that the command bars are displayed on the PFD.
13) Verify that the AP remains disengaged (the control wheels have free motion)

(c) FD/AP Operational Test

NOTE: The command bars on the PFD should be aligned with the aircraft reference and the control wheels should be centered and stationary. There may be some roll motion of the yokes if the aircraft is not level.

1) Select AP master switch to "FD/AP".
2) Push the HDG knob on the PFD to synchronize the heading bug to the current aircraft heading.
3) Press the HDG and ALT buttons on the AP computer.
4) Verify that HDG and ALT are displayed on the AP computer and the PFD AFCS status box, and that the command bars are displayed on the PFD.
5) Turn the HDG knob on the PFD counterclockwise.
6) Verify the command bars on the PFD and the control wheels rotate counterclockwise.
7) Turn the HDG knob on the PFD clockwise.
8) Verify the command bars on the PFD and the control wheels rotate clockwise
9) Push the HDG knob to synchronize the heading bug to the current aircraft heading. Wait until the flight director command bars align with the aircraft symbol.
10) Press the VS button on the AP computer.
11) Turn the VS knob on the AP computer clockwise.
12) Verify the command bars on the PFD pitch up and the control wheels move aft. (In some aircraft, the down spring may require manual assistance to get control stick movement.)
13) Turn the VS knob on the AP computer counterclockwise.
14) Verify the command bars on the PFD pitch down and the control wheels move forward.
15) Disengage the AP.

(d) Autopilot Clutch Torque and Cable Tension Test

1) Select AP master switch to "FD/AP" and TRIM switch "ON".
2) Press the HDG and ALT buttons on the AP computer.
3) Verify that HDG and ALT are displayed on the AP computer and the PFD AFCS status box, and that the command bars are displayed on the PFD.

4) Verify the autopilot clutches can be manually overpowered in the pitch and roll axes.

NOTE: If servo clutches cannot be overpowered, check the servo clutch torque settings. Refer to "Servo Clutch Torque Adjustment" on page 221022 for torque settings.

5) Press the HDG and ALT buttons on the AP computer.
6) Verify that HDG and ALT are displayed on the AP computer and the PFD AFCS status box, and that the command bars are displayed on the PFD.
7) Actuate and hold PITCH TRIM switch in either the NOSE UP or NOSE DOWN direction.
8) Verify the autopilot disconnects, RDY is displayed on the autopilot computer flashing for 5 seconds and then remains, and TRIM is displayed on the PFD flashing while the trim is running.
9) While trim is running, restrain the aircraft pitch trim wheel and verify that the trim clutch can be overpowered.

NOTE: If trim servo clutch cannot be overpowered, check the trim servo clutch torque setting. Refer to the autopilot installation drawing for torque settings.

10) Actuate and hold PITCH TRIM switch in the NOSE UP direction.
11) Verify the trim wheel moves smoothly throughout the entire trim range during manual electric trim operation, and TRIM is displayed on PFD while the trim is running.
12) Actuate and hold PITCH TRIM switch in the NOSE DOWN direction.
13) Verify the trim wheel moves smoothly throughout the entire trim range during manual electric trim operation, and TRIM is displayed on PFD while the trim is running.

(e) Electric Trim and CWS

1) Select AP master switch to "FD/AP" and TRIM switch "ON".
2) Actuate both sections of the PITCH TRIM switch to activate Manual Electric Pitch Trim (MEPT). Verify the trim clutch engages the trim wheel and trim tab drive in the requested direction, and TRIM is displayed on the PFD flashing while the trim is running. Check operation in the up and down direction.
3) Activate each half of the MEPT switch. Verify the trim clutch does not engage, the trim wheel does not move.
4) Press the AP DISC TRIM INTER switch and hold while actuating the manual electric trim switch. Verify trim does not run and the trim wheel rotates freely when moved manually. Release the switches.

NOTE: If the trim is run for approximately two seconds with AP DISC TRIM INTER switch held, the trim monitor will enter a FAIL state. To reset, depress the trim switch twice to turn the system off and back on.

5) Activate the MEPT switch. Remove power from the Pitch Trim servo by pulling the Pitch Trim circuit breaker. Verify trim does not run and the trim wheel rotates freely when moved manually. Release the switches.
6) Restore power to the Pitch Trim Servo.
7) Press the HDG and ALT buttons on the AP computer.
8) Verify that HDG and ALT are displayed on the AP computer and the PFD AFCS status box, and that the command bars are displayed on the PFD.
9) Press and hold the CWS switch and verify the control wheel moves freely when moved manually and CWS is displayed on the AP computer.

10) Release the CWS switch and press the AP DISC TRIM INTER switch on the pilots control wheel. Verify the autopilot disengages and RDY is displayed on the autopilot computer flashing for 5 seconds, accompanied by an aural alert. Verify the control wheels move freely in pitch and roll axes.

(f) Nav Receiver Interface Test

NOTE: Perform the following test using ramp test equipment. Operate the equipment according to the test equipment manufacturer's instructions.

NOTE: The PFD HSI does not show a course deviation bar unless a valid VHF NAV frequency is tuned.

1) Select AP Master Switch to "FD/AP".
2) Simulate a VOR signal on radial 360° relative to aircraft. Tune the NAV 1 and NAV 2 receivers to the simulation frequency.
3) Set the HSI on the PFD to VOR1 by pressing the CDI soft key until VOR1 is selected. Press the CRS knob on the PFD.
4) Verify full scale deflection of VOR1 by varying the selected course at least 10° left and right. Reset course pointer to 360° relative to aircraft.
5) Repeat Steps 3 and 4 with VOR 2. When finished, set the PFD to VOR 1.
6) Engage the autopilot and press the NAV and ALT keys on the autopilot computer. Using the CRS1 knob alter course by 10° to the right. Verify the flight director and aircraft controls respond by flying to the VOR course. Repeat to the left.
7) Set CRS1 course pointer to 360° relative to aircraft.
8) Simulate a Localizer/Glideslope signal. Tune this signal on NAV 1 and NAV 2 receiver. Set the PFD HSI to LOC1. Use the test equipment to center the deviation bars (localizer and glideslope) on the PFD.
9) Press the ALT and APR key on the autopilot computer. Verify that NAV, ALT, and APR annunciations are displayed on the PFD. Apply right/left and up/down localizer/glideslope signals using the test equipment. Verify that the Flight Director and flight controls respond appropriately.

(g) Autopilot Flap Compensation Check

1) Ensure FD/AP Master switch is set to "FD/AP" and TRIM switch is ON.
2) Set the control yoke to a neutral position.
3) Press HDG and ALT buttons on the autopilot computer.
4) Pull the flap lever to set flaps to 10 degrees.
5) Verify that the control yoke moves forward.
6) Pull the flap lever to set flaps to 25 degrees.
7) Verify that the control yoke moves forward.
8) Pull the flap lever to set flaps to 40 degrees.
9) Verify that the control yoke moves forward.
10) Move flap lever forward to set flaps to 25 degrees.
11) Verify that the control yoke moves aft.
12) Move flap lever forward to set flaps to 10 degrees.
13) Verify that the control yoke moves aft.
14) Move flap lever forward to set flaps to 0 degrees.
15) Verify that the control yoke moves aft.
16) Set FD/AP Master switch to "FD".

I. Components

(1) Panel-Mounted Components

The flight director, HSI, autopilot programmer/computer, altitude selector/alerter (if installed), remote annunciator, and turn coordinator are either face-mounted or rack-mounted in the instrument panel. See 39-10-00 for removal and installation instructions.

(2) Component Locator

See “Figure 2” on page 221012.

(3) Trim Monitor

The trim monitor is mounted to a bracket under the instrument panel on the left side. See “Figure 2” on page 221012.

(a) Removal

1) Disconnect autopilot harness.
2) Remove screws (4) holding trim monitor to mounting bracket and remove trim monitor.

(b) Installation

1) Place trim monitor in position on mounting bracket and secure with screws (4).
2) Connect autopilot harness.
3) Perform “Ground Checks” on page 22106.

(4) Pressure Transducer

The pressure transducer is located on the aft upper left side of the bulkhead at F.S. 191.00. See “Figure 2” on page 221012.

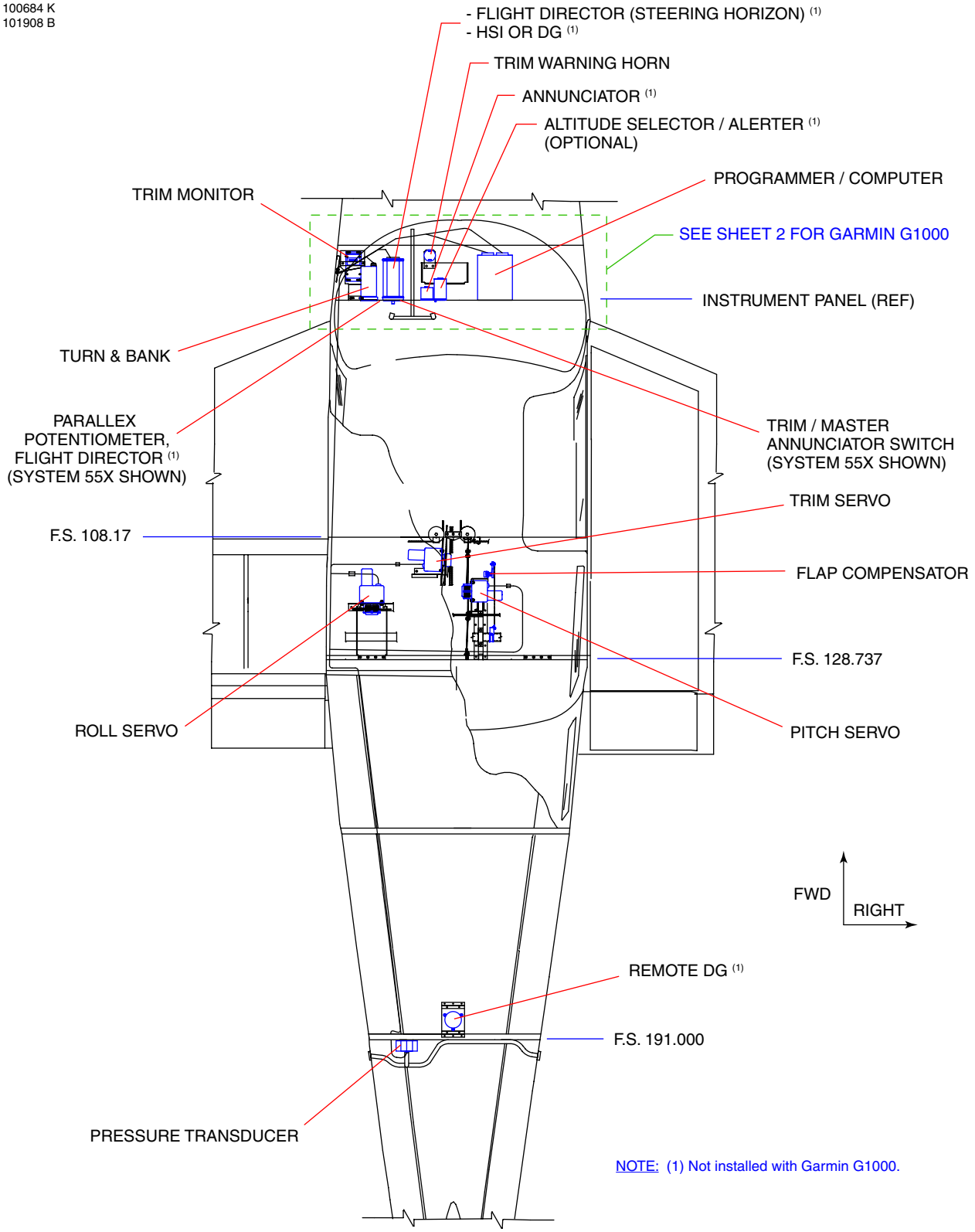
(a) Removal

1) Remove the Ty-Rap and disconnect the transducer from the static-system by removing the flexible hose.
2) Disconnect the autopilot harness.
3) Remove screws and washers (2 ea.) and remove transducer.

(b) Installation

1) Place transducer in position. Secure transducer to instrument panel with screws and washers (2 ea.)
2) Connect the transducer to the static system by sliding the flexible hose over the hose barb. Then position and tighten Ty-Rap.
3) Connect the autopilot harness.
4) Perform “Ground Checks” on page 22106.

104580 T
100684 K
101908 B

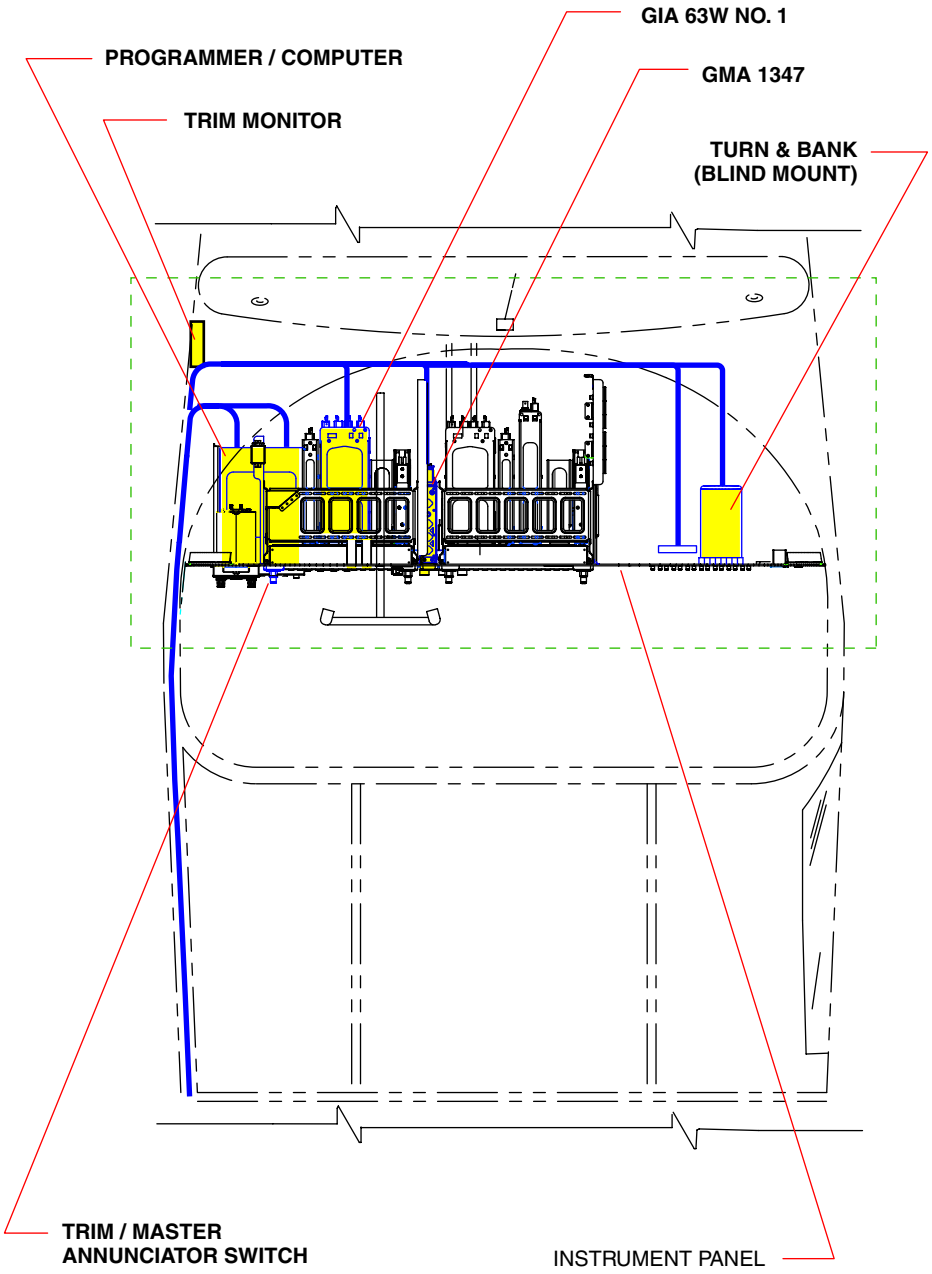


NOTE: (1) Not installed with Garmin G1000.

Effectivity
2843203 and up

Component Locator
Figure 2 (Sheet 1 of 2)

107564 A



Component Locator
Figure 2 (Sheet 2 of 2)

Effectivity
with Garmin G1000 or G1000 NXi
2843701, 2843703–2843999

Effectivity
with S-Tec System 55/55X

(5) Roll Servo

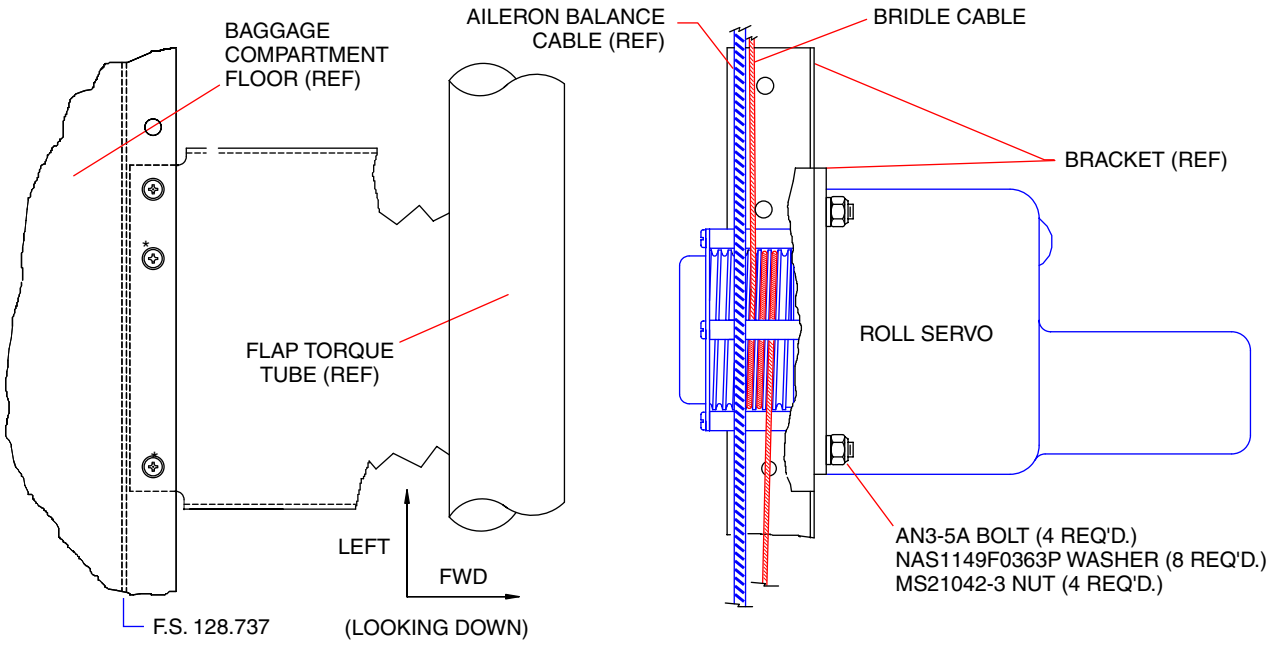
The roll servo is mounted underneath the left rear seat. A bridle cable and clamps attach the servo capstan to the aileron balance cable. See “Figure 2” on page 221012 and “Figure 3” on page 221015.

(a) Removal

1) Remove the rear seats.
2) Remove adjacent carpet.
3) Remove screws securing floor panel and remove panel.
4) Disconnect autopilot harness.
5) Remove nuts and bolts (2 ea.) securing each cable clamp (2) and remove cable clamps from aileron balance cable and autopilot bridle cable.
6) Remove nuts and bolts (4 ea.) and washers (8 ea.) securing roll servo to mounting bracket and remove roll servo with attached bridle cable.

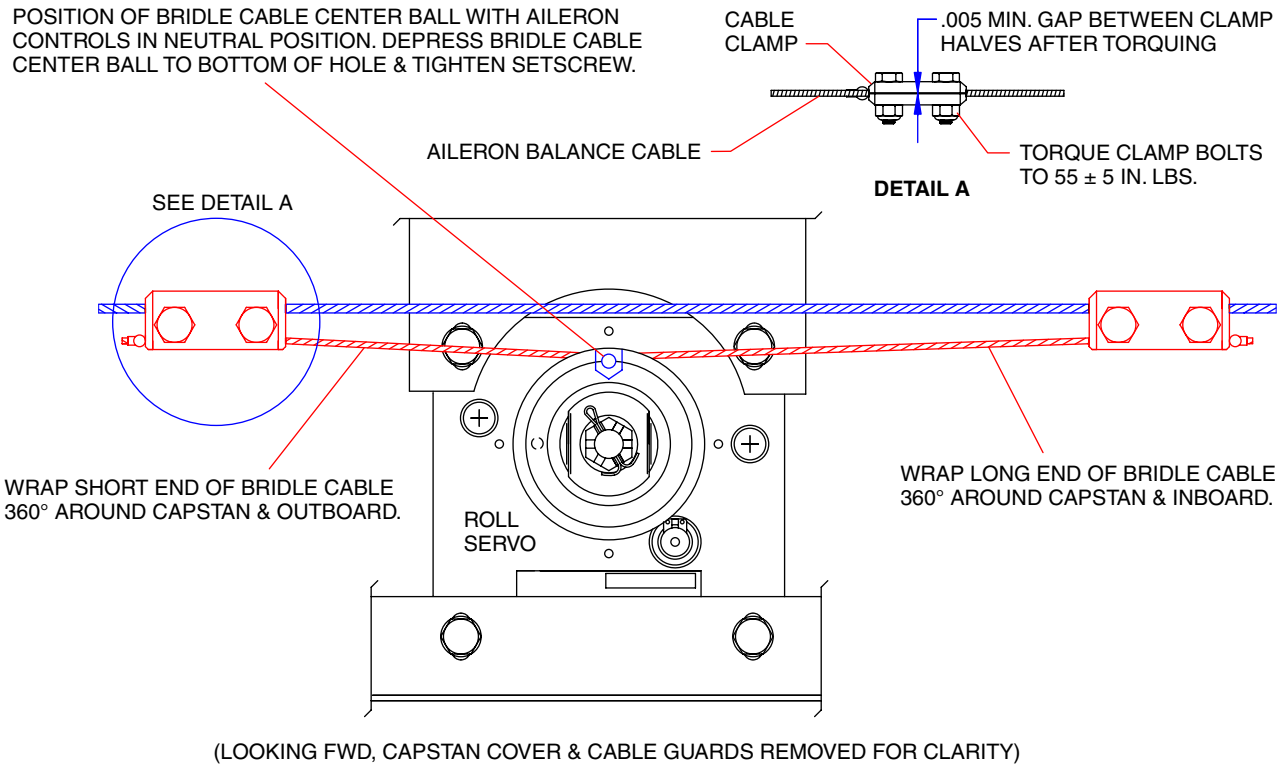
(b) Installation

1) Rig ailerons per Aileron Control Rigging and Adjustment, 27-10-00.
2) Place the control column tee bar in full forward position and secure by use of a suitable tool or by placing weights on the aft side of the stabilator, if stabilator cables have been previously tensioned.
3) Lock the ailerons in neutral (i.e. - aligned with flaps) position using a suitable contour fixture at the inboard ends of the ailerons and the outboard ends of the flaps. Verify control wheels are centered and secure in that position.
4) Remove screws (4) and remove capstan cover and cable guards from servo.
5) Adjust roll servo clutch torque per Servo Clutch Torque Adjustment, below.
6) Wrap autopilot bridle cable, align capstan, and and tighten center-ball setscrew as shown in “Figure 4” on page 221015.
7) Replace cable guards and capstan cover, secure with screws (4).
8) Position servo as shown in Figure 3 and install and secure nuts (4 ea.) , washers (8 ea.) , and bolts (4 ea.) holding servo to mounting bracket.
9) Position cable clamps (2) as shown in Figure 3 and tighten nuts and bolts (2 ea.). Adjust cable clamps in or out along the aileron balance cable to obtain a bridle cable tension of 15 + 10, -2 lbs (System 55X) or 15 ± 2 lbs (System 55). Torque cable clamp bolts to torque specified in “Figure 4” on page 221015.
10) Remove the locking fixtures at the inboard ends of the ailerons. Aileron neutral (i.e., aligned with flaps) position should be maintained with the control wheels in neutral. A droop of 1/8 inch is allowable.
11) Remove the control wheel/tee bar locks. Check to ensure that the left aileron up and right aileron down stops are contacted simultaneously and vice versa. Adjust stops as required.
12) Rotate the left (pilot’s) control wheel in each direction until the bellcranks contact the stops. The sprocket stops on the tee bar shall not be contacted until additional “override” movement (cushion) of the wheel occurs. A “cushion” on 0.030 to 0.040 inches is to be maintained as measured between the sprocket pin and adjustable control wheel stop bolts.



Roll Servo Installation
Figure 3

Effectivity
2843203 and up



Roll Servo Capstan Wrapping
Figure 4

Effectivity
2843203 and up

13) Place the ailerons in the neutral (aligned with the flaps) position. For each aileron, from the neutral position, check that the “up” travel and the “down” travel are within the limits shown in Figure 8, 27-10-00.

a) Center bubble of a protractor over surface of aileron at neutral position. Note reading.

b) Move aileron full up and down. Check degree of travel in each direction. Degree of travel on protractor is determined by taking the difference between protractor reading at neutral and up, and neutral and down. Bubble must be centered at each reading.

When measuring “down” travel from the neutral position, a light “up” pressure shall be maintained at the center of the aft edge of the aileron. When measuring “up” travel from the neutral position, a light “down” pressure shall be maintained at the center of the aft edge of the aileron (at the “up” position only), just sufficient to remove the slack between the bellcrank and the aileron.

14) If steps (10)–(13), above, reveal the aileron controls out of rig, repeat steps (1)–(9).

15) Connect autopilot harness.

16) Check aileron controls for free and correct movement.

17) Perform “Ground Checks” on page 22106.

18) Replace floor panel and secure with screws.

19) Replace carpeting.

20) Replace the rear seats.

(6) Trim Servo

The trim servo is located just left of the aircraft centerline, under the rear seats, just aft of the bulkhead at F.S. 108.17. The left stabilator trim cable wraps around the servo idler pulley and servo capstan. See “Figure 2” on page 221012 and “Figure 5” on page 221017.

(a) Removal

1) Remove rear seats. Remove carpet and floor panel.

2) Tie a pull rope to the left stabilator trim cable exposed beneath the floor panel and tie-off the pull rope to structure aft.

3) Tie a pull rope to the left stabilator trim cable aft of the turnbuckle in the rear fuselage aft of the trim servo and tie-off the pull rope to structure forward.

NOTE: The pull ropes apply tension to the trim cables to prevent the cables from unwrapping from the trim wheel drum or the trim barrel, and to prevent the cables from fouling at any of the pulleys.

4) Slack-off the turnbuckle in the left stabilator trim cable segment aft of the trim servo sufficient to relieve tension on the left stabilator trim cable as it wraps around the trim servo idler pulley and capstan.

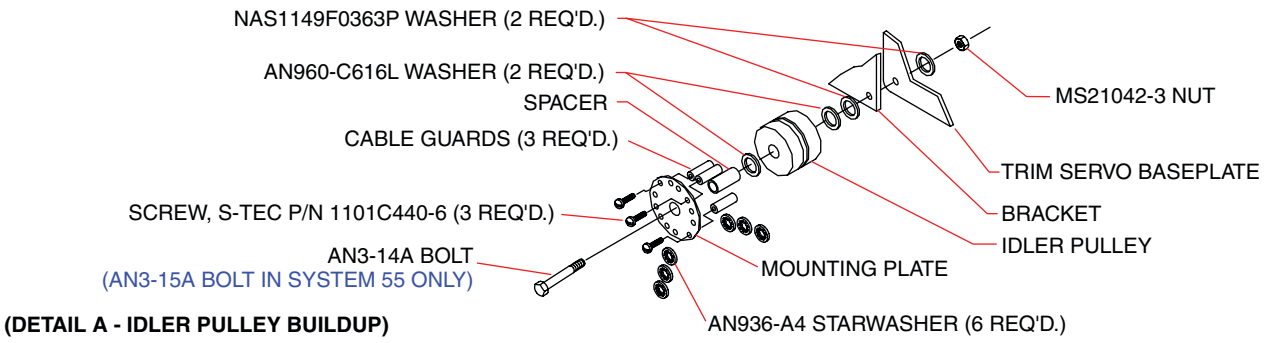
5) Disconnect the autopilot harness.

6) Remove the capstan cover and cable guards (4) by removing the retaining screws (4).

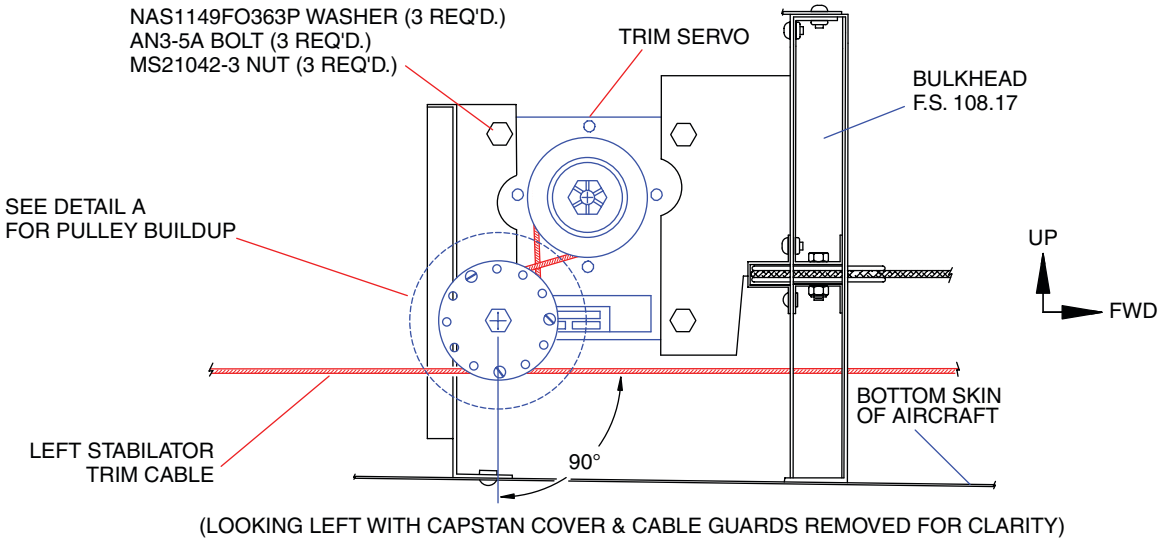
7) Remove the bolt, nut, and washer securing the idler pulley to the trim servo baseplate and mounting bracket and remove the idler pulley components.

NOTE: The idler pulley breaks down into the following components upon removal of the bolt, above: mounting plate/cable guard assembly, idler pulley, and two washers.

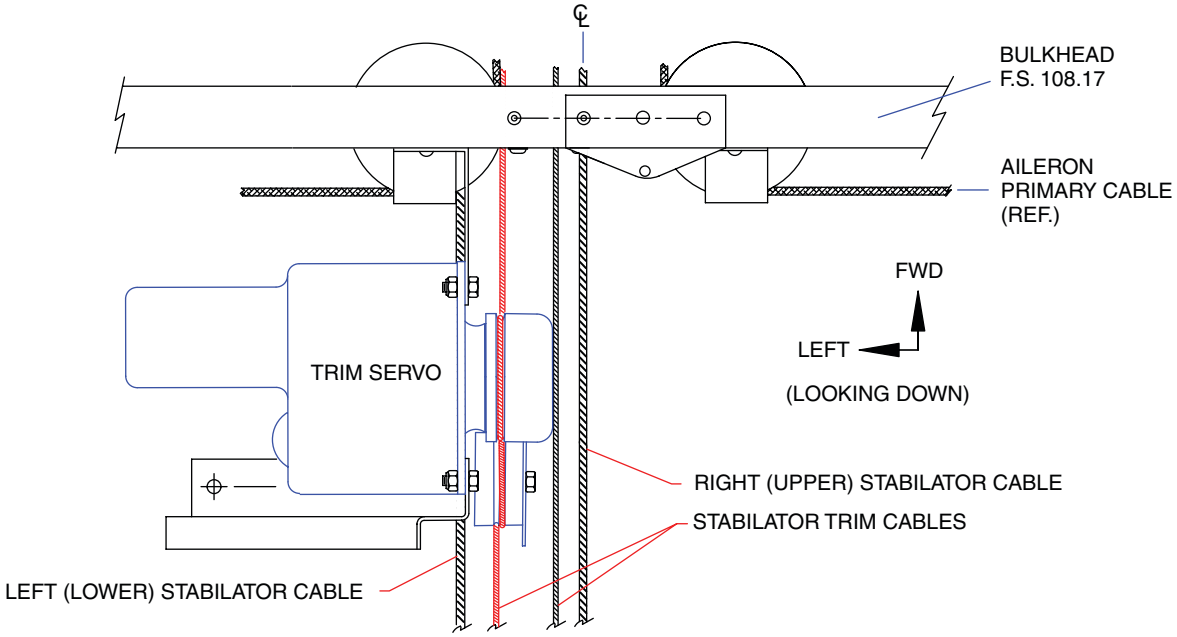
8) Remove the remaining bolts, nuts, and washers (3 ea.) securing the trim servo to its mounting bracket and remove the trim servo.



(DETAIL A - IDLER PULLEY BUILDUP)



(LOOKING LEFT WITH CAPSTAN COVER & CABLE GUARDS REMOVED FOR CLARITY)



Trim Servo Installation
Figure 5

Effectivity
2843203 and up

(b) Installation

1) Adjust trim servo clutch torque per “Servo Clutch Torque Adjustment” on page 221022.
2) With the capstan cover and cable guards removed, position the trim servo as shown in “Figure 2” on page 221012 and “Figure 5” on page 221017. Secure with bolts, nuts, and washers (3 ea. - i.e., aft two and forward left).
3) Assemble the idler pulley cable guards (3) to the mounting plate with screws (1 ea.) and star washers (2 ea.). Place the center bolt through the mounting plate/cable guard assembly and slide a washer over the threaded end and up against the mounting plate. Set the mounting plate/cable guard/bolt assembly aside.
4) Drape the slack left stabilator trim cable over the servo capstan.

CAUTION: IN STEPS 5 THRU 9, BELOW, USAGE OF LEFT AND RIGHT IS RELATIVE TO THE VIEW OF THE TECHNICIAN IN THE CABIN BAGGAGE AREA LOOKING AFT, EXCEPT WHERE AIRPLANE COMPONENT PARTS ARE SPECIFICALLY NAMED.

5) Place thumb and forefinger on top of the capstan over the trim cable in its groove. Pressing the trim cable into its groove, slide thumb and forefinger down around opposite sides of the servo capstan and pull the trim cable slack towards you and to your left.
6) Holding the trim cable in that position, install the capstan cover and cable guards as shown in “Figure 5” on page 221017.
7) Hold the idler pulley aft of the trim servo and to the right of the airplane’s left stabilator trim cable. Move the idler pulley left to the left stabilator trim cable and capture the trim cable in the bottom cable groove on the left of the idler pulley.
8) Keeping the trim cable in the bottom groove, slide the idler pulley forward along the trim cable, left of the servo capstan, and bring it approximately to its installed position (See “Figure 5” on page 221017.). At this point, the aft portion of the trim cable should be routed left, around the front, and to the right of the idler pulley and to the left, around the rear, and to the right of the capstan.
9) Holding the idler pulley in this position, reach down and pull the forward portion of the trim cable over the idler pulley and seat it in the top cable groove on the idler pulley. The left trim cable should now be routed as shown in “Figure 5” on page 221017.
10) Place the spacer inside the idler pulley.
11) Position the mounting plate/cable guard/bolt assembly as shown in “Figure 5” on page 221017 and slide the bolt through the spacer inside the idler pulley.
12) Place a washer over the bolt end and put the bolt through the trim servo mounting bracket and baseplate. Secure with a nut and washer, taking care to ensure that the cable guards are positioned, and the left stabilator trim cable is routed, as shown in “Figure 5” on page 221017.
13) Take up the slack in the left trim cable with the turnbuckle and remove the pull ropes.
14) Rig stabilator trim per Stabilator Trim Rigging and Adjustment, 27-30-00.
15) Connect the autopilot harness.
16) Perform “Ground Checks” on page 22106.
17) Check elevator trim controls for free and correct movement.
18) Reinstall floor panel and carpet.
19) Reinstall rear seats.

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(7) Pitch Servo

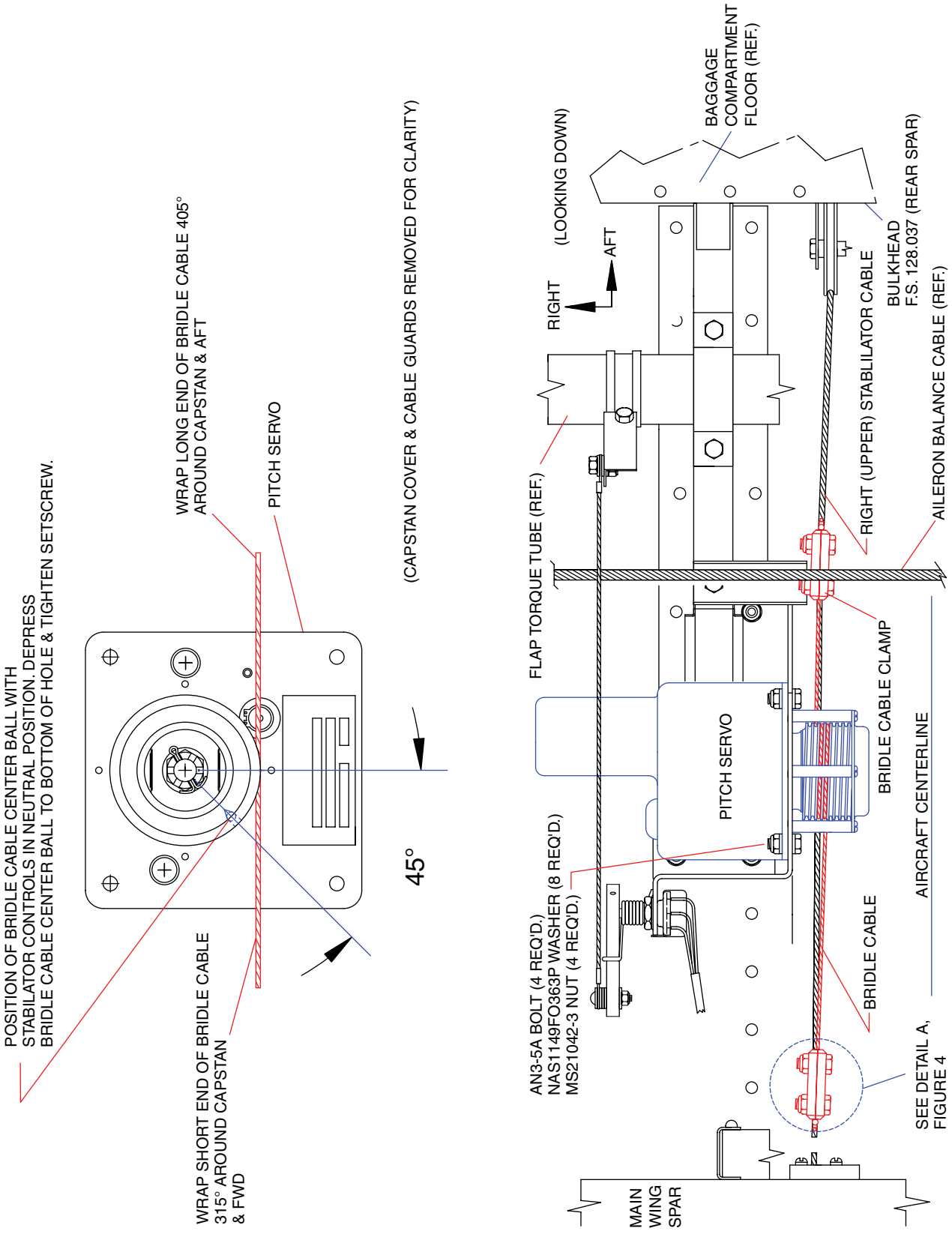
The pitch servo is located under the right rear seat. A bridle cable and clamps attach the servo capstan to the right (upper) stabilator cable. See “Figure 2” on page 221012 and “Figure 6” on page 221021.

(a) Removal

1) Remove rear seats.
2) Remove carpet and floor panel.
3) Disconnect autopilot harness.
4) Remove nuts and bolts (2 ea.) securing each cable clamp (2) and remove cable clamps from right (upper) stabilator cable and autopilot bridle cable.
5) Remove nuts and bolts (4 ea.) and washers (8 ea.), securing pitch servo to mounting bracket and remove pitch servo with attached bridle cable.

(b) Installation

1) Rig stabilator controls per Stabilator Controls Rigging and Adjustment, 27-30-00.
2) Remove screws (4) and remove capstan cover and cable guards from servo.
3) Adjust pitch servo clutch torque per Servo Clutch Torque Adjustment, below.
4) Wrap autopilot bridle cable, align capstan, and tighten center-ball setscrew as shown in “Figure 6” on page 221021.
5) Replace cable guards and capstan cover, secure with screws (4).
6) Position pitch servo as shown in “Figure 6” on page 221021 and secure with bolts, nuts, and washers (4 ea.).
7) Position cable clamps (2) as shown in “Figure 6” on page 221021 and tighten nuts and bolts (2 ea.). Adjust cable clamps in or out along the stabilator cables to obtain a bridle cable tension of 15 ± 2 lbs (System 55) or 15 +10, -2 lbs (System 55X). Torque cable clamp bolts as specified in “Figure 4” on page 221015.
8) Connect autopilot harness.
9) Perform “Ground Checks” on page 22106.
10) Check stabilator controls for free and correct movement.
11) Reinstall floor panel and carpet.
12) Reinstall rear seats.



Pitch Servo Installation
Figure 6

Effectivity
2843203 and up

Effectivity
with S-Tec System 55/55X

(8) Servo Clutch Torque Adjustment

See “Figure 7” on page 221023 and “Chart 1” on page 221023.

(a) Remove servo per instructions under specific servo, above.

(b) Place servo in a holding fixture (i.e., vice) with capstan up.

(c) Remove capstan cover, cable guards, and cable.

(d) Check capstan torque by attaching the capstan adjusting tool (special tool – see parts catalog) to the capstan and using a currently calibrated torque wrench as shown in “Figure 7” on page 221023.

1) Acceptable torque is specified in “Chart 1” on page 221023.

2) If adjustment is required, proceed as follows.

(e) Remove cotter pin from end of servo shaft and remove castle nut, shim washers, and tension washers.

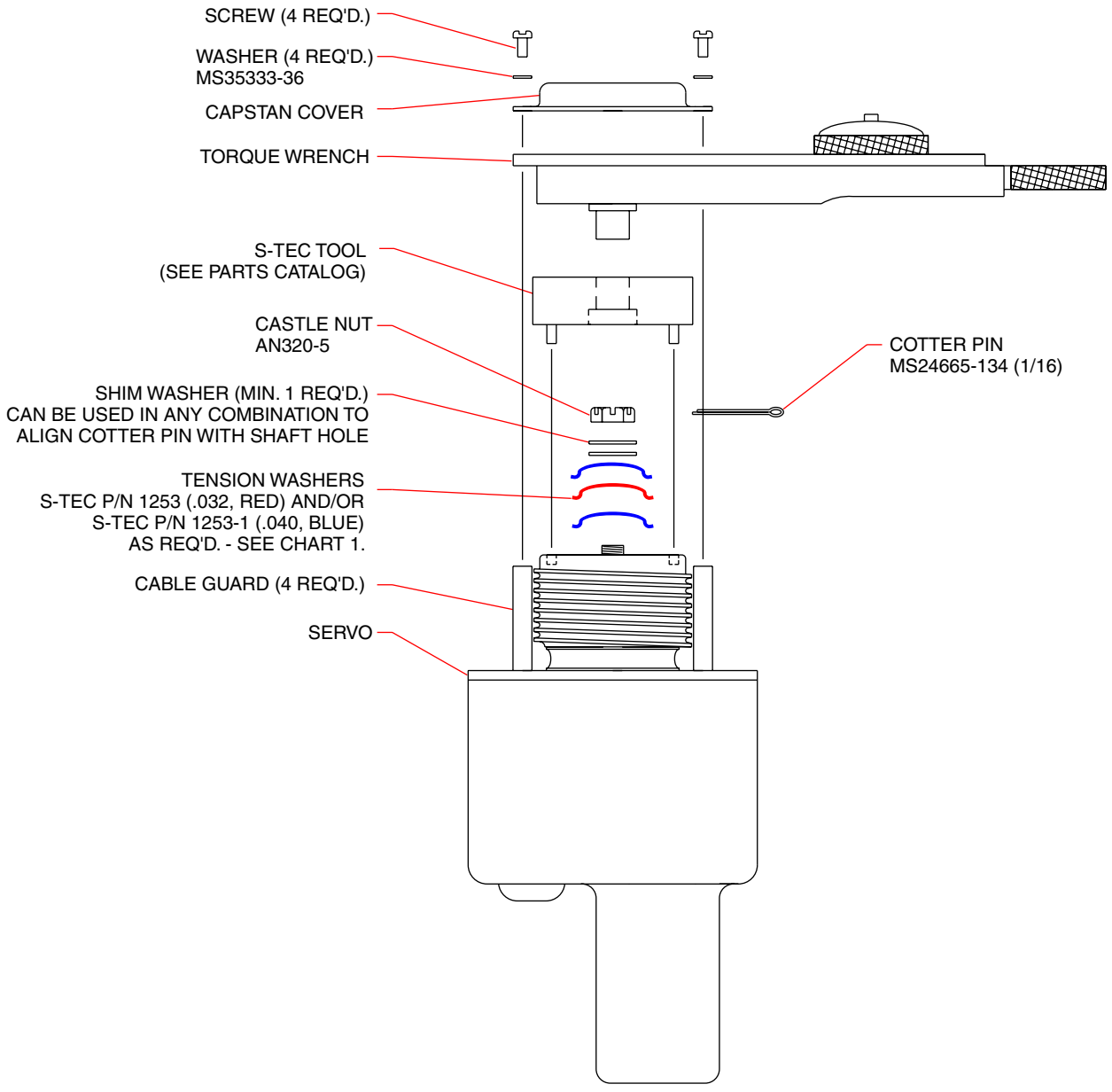
(f) Replace tension washers as required (see “Chart 1” on page 221023).

(g) Replace shim washers and castle nut.

(h) Tension castle nut so that capstan torque is as specified in “Chart 1” on page 221023.

CHART 1
AUTOPILOT SERVO CLUTCH TORQUE

SERVO	TORQUE (In. Lbs.) System 55/55X	WASHERS REQUIRED System 55	WASHERS REQUIRED System 55X
Roll	35 ± 2	One .032; Two.040	Three.032
Pitch	40 ± 2	One .032; Two .040	One .032; Two .040
Trim	29 ± 2	One .032; Two .040	Three.032



Servo Clutch Torque Adjustment
Figure 7

Effectivity
2843203 and up

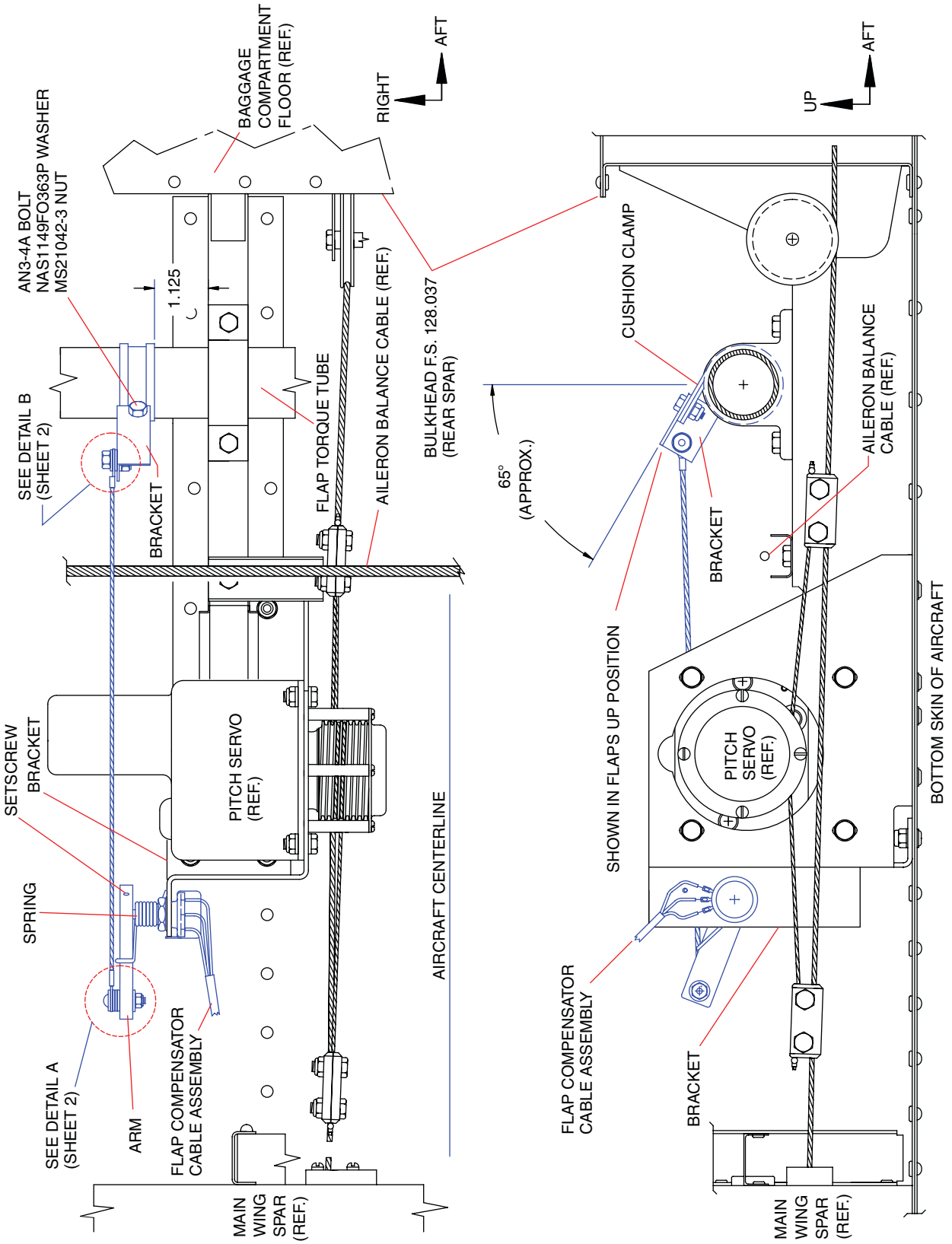
Effectivity
with S-Tec System 55/55X

(9) Flap Compensator

A flap compensator potentiometer (pot) is mounted underneath the right rear seat. The pot is mounted on a bracket attached to the forward side of the pitch servo mounting bracket. The arm of the pot is linked to the flap torque tube by a cable assembly. See “Figure 8” on page 221025.

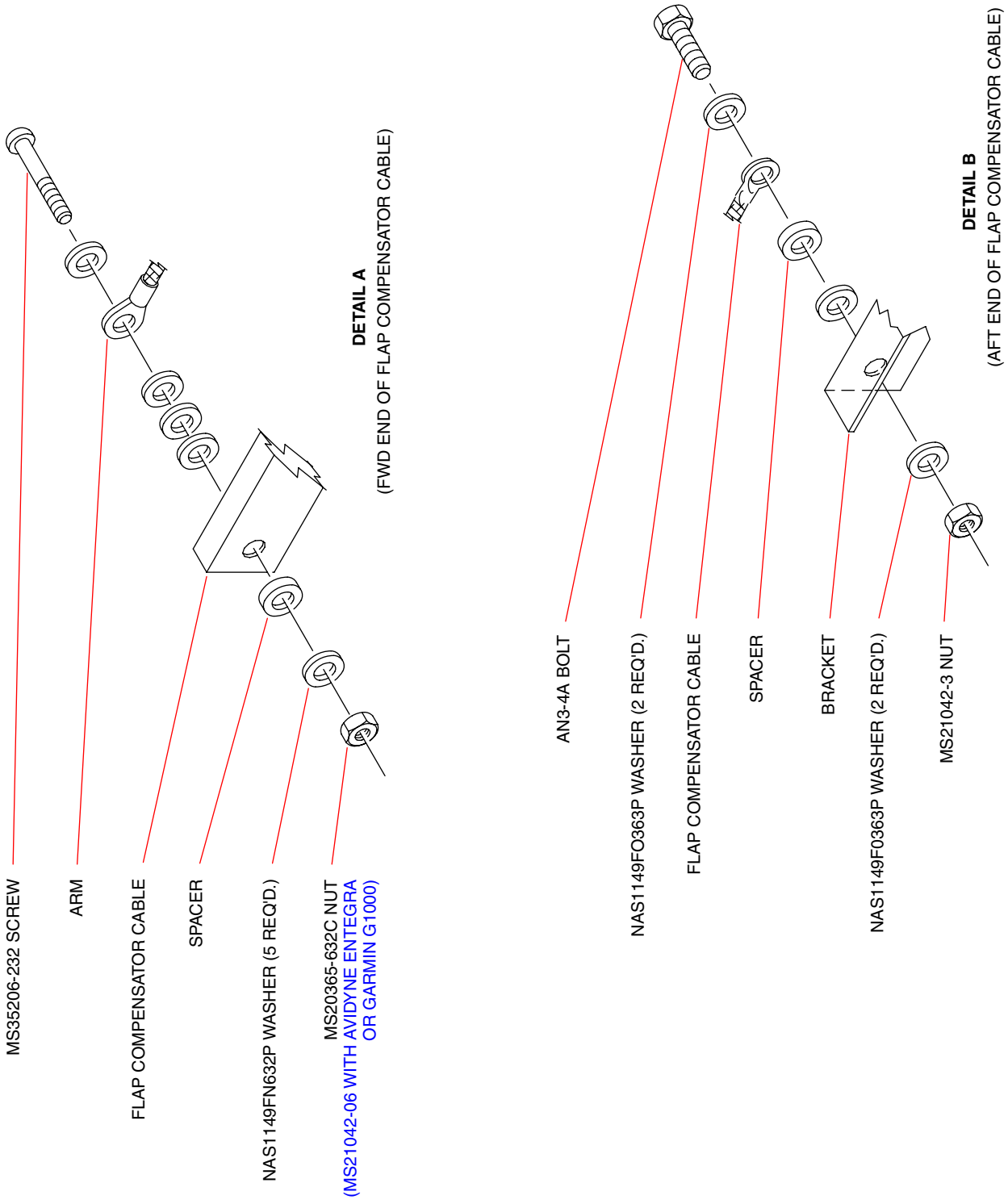
Adjustment

1) With flaps in the full up position and set screw loose, turn A/P master switch ON.
2) Connect a digital voltmeter (3 1/2 digit) between airframe ground and center terminal (wiper) of potentiometer (pot).
3) Turn pot shaft clockwise to stop. Voltmeter should read 5.00 vdc.
4) Turn pot shaft slowly counter-clockwise until voltage just starts to decrease from 5.00 volts.
5) Tighten set screw and recheck wiper voltage for 4.95 to 5.00 vdc.



Flap Compensator Installation
Figure 8 (Sheet 1 of 2)

Effectivity
2843203 and up



Effectivity
2843203 and up

Flap Compensator Installation
Figure 8 (Sheet 2 of 2)

Effectivity
with S-Tec System 55/55X

3. Garmin GFC 700 Automatic Flight Control System (AFCS) (Optional)

This system is available in Garmin G1000 NXi equipped airplanes S/N's 2843823, 2881001 and up.

A. Description

See "Figure 9" on page 221036 and "Figure 11" on page 221039. See also 34-25-03.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY.

CAUTION: THIS SECTION PROVIDES INFORMATION SUFFICIENT ONLY TO LOCATE, REMOVE, AND INSTALL COMPONENTS. DETAILED TROUBLESHOOTING OR REPAIR IS REQUIRED, IT SHOULD BE PERFORMED BY A QUALIFIED AVIONICS REPAIR SHOP USING THE APPROPRIATE MAINTENANCE MANUALS.

The GFC 700 Automatic Flight Control System (AFCS) provides a three-axis fail-safe digital flight control with Flight Director, Autopilot, and Pitch Auto-Trim functions. The GFC 700 functionality is distributed across several units of the G1000 NXi avionics system and is designed for precision approach to Category 1 minimums. The controls are located on the MFD bezel. The mode logic and Flight Director computations are performed within the GIA 63W Integrated Avionics units. The autopilot computations and monitoring are performed within the GSA 8X Servo Actuators with GSM 86 Servo Gear Boxes. Dual Control Display Units and Integrated Avionics units provide redundancy.

B. Troubleshooting

Perform "Ground Checks" below and see Troubleshooting in 34-25-03.

C. Ground Checks

(108059, Rev. New)

(1) Setup
  (a) Ensure G1000 NXi software is loaded, properely configured, and checked out per 34-25-03.
  (b) Level the airplane per 8-20-00.
  (c) Apply external power.

(2) References
  (a) Appropriate Garmin Cockpit Reference Guide for Piper PA-28-181 Archer.
  (b) Appropriate Pilot's Operating Handbook (POH) serialized to the airplane.
  (c) Avionics/Electrical System Schematics found in Chapter 91.

NOTE: In the sections that follow, an Autopilot disconnect should be accompanied by an aural alert (two-second tone) unless otherwise specified.

(3) Pre-Flight Test (PFT)

(a) Set the AVION MASTR switch to OFF. After 30 seconds select the AVION MASTR switch to ON. Verify the GFC 700 begins an automatic pre-flight test after AHRS and ADC parameters become valid.

(b) Verify that a white 'PFT' annunciation is displayed on the PFD. A momentarily red 'AFCS' annunciation displayed before PFT starts is acceptable.

(c) Upon successful completion of the test, an aural alert will sound and the annunciation will clear. If the 'PFT' annunciation turns red, the test has failed. See Troubleshooting, above.

(d) Repeat Steps (a)–(c) to test the PFT aural alert for the other GIA 63W. (For even-interval system power-up, the aural alert is generated by GIA 63 #1, whereas odd-interval system power-ups are generated by GIA 63 #2.)

(4) AFCS Switch Checks

Verify that the AFCS system buttons and switches are operating correctly by performing the following steps (for both the pilot and copilot control yokes where applicable).

(a) Actuate both sections of the PITCH TRIM (TRIM UP/TRIM DN) switch to activate Manual Electric Pitch Trim (MEPT). Verify the trim clutch engages and the trim wheel and trim tab drive in the requested direction. Check operation in both the up and down direction.

NOTE: "TRIM" is displayed flashing on the PFD while the trim is running.

(b) Activate Co-Pilot's MEPT switch in the UP/DN direction. Then activate the Pilot's MEPT switch in the opposite direction. Verify the Pilot's MEPT switch takes priority.

(c) Activate each half of the MEPT switch. Verify the trim clutch does not engage, the trim wheel does not move, and PTRM annunciation on the PFD after approximately 3 seconds.

(d) Press the AP DISC TRIM INTER switch and hold while actuating the manual electric trim switch. Verify trim does not run and the trim wheel rotates freely when moved manually. Release the switches.

(e) Activate the MEPT switch. Remove power from the Pitch Trim servo by pulling the Pitch Trim circuit breaker. Verify trim does not run and the trim wheel rotates freely when moved manually. Release the switches.

(f) Pull Autopilot circuit breaker. Reset Pitch Trim and Autopilot circuit breakers.

(g) Engage the autopilot by pressing the AP key on the MFD. Verify the pitch and roll clutches engage and resist movement of the control wheel. Press and hold the CWS switch and verify the control wheel moves freely when moved manually.

(h) Release the CWS switch and press the AP DISC TRIM INTER switch on the pilots control wheel. Verify the autopilot disengages with a flashing amber 'AP' annunciation on the PFD, accompanied by an aural alert. Verify that the control wheel is free in pitch and roll axes.

(i) Press the GO AROUND button on the power lever. Verify 'TO' is annunciated on the PFD for both PITCH and ROLL modes and the command bars should be at ten (10) degrees nose up and wings-level.

(j) Press the Flight Director (FD) key on the MFD to deactivate the GA mode. Press the AP key to engage the autopilot. Press the CWS button for a minimum of 5 seconds and release; verifying there is no residual force on the control stick for the pitch and roll axis while the CWS is pressed. Servos reengage after CWS is released.

(k) Disengage the autopilot by pressing the AP DISC TRIM INTER switch on the co-pilot's control wheel. Engage VS mode by pressing the VS key on the MFD. Verify the PFD displays 'VS' in green and indicates a pitch reference of '0 FPM'.

NOTE: The Alt. Sel must be activated for the FLC key to function properly.

(l) Press the FLC key on the MFD and verify that 'FLC' is annunciated on the PFD.

(m) Press the ALT key on the MFD and verify that the 'ALT' annunciation is displayed in green on the PFD with an altitude reference equal to the aircraft altitude (within the nearest 20 feet).

(n) Press the FD key and verify that the mode annunciations and command bars are removed from the display.

(5) Autopilot Disconnect Checks – Normal Mode

Engage the autopilot by pressing the AP key on MFD prior to each of the following checks and reset the circuit breaker (if required) prior to the subsequent check. For each check, verify a five-second flashing amber AP annunciation, a three-second disconnect aural alert, and FD bars remaining on display.

(a) Press the MEPT switch on the pilot's control yoke.

(b) Press and hold the left section of the MEPT switch on the pilot's control yoke. Verify the trim wheel rotates freely when moved manually.

(c) Press and hold the right section of the MEPT switch on the pilot's control yoke. Verify the trim wheel rotates freely when moved manually.

(d) Press and hold the left section of the MEPT switch on the co-pilot's control yoke. Verify the trim wheel rotates freely when moved manually.

(e) Press and hold the right section of the MEPT switch on the co-pilot's control yoke. Verify the trim wheel rotates freely when moved manually.

(f) Press GA button on the throttle handle. Press the FD key to remove the GA mode annunciation from the display.

(g) Press the A/P DISC TRIM INTER switch on the pilot's control yoke.

(h) Press the A/P DISC TRIM INTER switch on the co-pilot's control yoke.

(6) Autopilot Disconnect Checks – Abnormal Mode

Engage the autopilot by pressing the AP key on the MFD prior to each of the following checks and reset the circuit breaker (if required) prior to subsequent check. For each check, verify a continuous flashing red/white AP, a red boxed AFCS annunciation (unless specified otherwise), and a continuous disconnect aural alert (until silenced by the AP DISC TRIM INTER switch).

NOTE: After resetting each circuit breaker, allow time for the AHRS or GPS to initialize prior to conducting the next test.

(a) Pull the INTEG AV 1 circuit breaker. Verify th FD bars are removed from the display. Various Crew Alerting System (CAS) messages will appear due to GIA 1 powering down.

(b) Pull the INTEG AV 2 circuit breaker. Verify th FD bars remain on the display. Various CAS messages will appear due to GIA 2 powering down.

(c) Pull the AHRS 1 circuit breaker. Verify the red boxed ACFS annunciation does not appear. Verify the FD bars are removed from the display.

(d) Pull the AUTOPILOT circuit breaker. After verification, pull the PITCH TRIM circuit breaker and reset both breakers.

(7) Autopilot Clutch Overpower Check

The GFC 700 uses electronic torque limiting as well as mechanical slip clutches to limit the maximum servo effort. When the system is on the ground, the electronic torque limiting is removed, allowing manual checks of the slip-clutch settings.

(a) Engage the Autopilot by pressing the AP key on the MFD.

(b) Manually overpower the autopilot clutches in pitch and roll. If the Autopilot clutches cannot be overpowered, check the servo clutch torque settings. See “Chart 2”.

(c) Actuate and hold PITCH TRIM switch in either the NOSE UP or NOSE DOWN direction to disconnect the autopilot. While the trim is running, restrain the aircraft pitch trim wheel and verify that the trim clutch can be overpowered. If it cannot be overpowered, check the trim servo clutch torque setting. See “Chart 2”.

(d) Engage the autopilot by pressing the AP key on the MFD. Actuate and hold the manual electric trim switch in either the up or down direction to disconnect the autopilot. Verify that the trim wheel moves smoothly in both directions throughout the entire trim range during manual electric trim operation. If the trim wheel hesitates, this may indicate that the pitch trim clutch is slipping and proper clutch setting and cable tension should be verified. See Troubleshooting, above; and Trim Servo and “Chart 2”, below. If both clutch setting and cable tension are within tolerance, check the aircraft pitch trim system for excessive friction. See Rigging and Adjustment under Elevator Trim Controls in 27-30-00.

CHART 2
SERVO CLUTCH TORQUE SETTINGS

Servo	Clutch Part Number (Piper/Garmin)	Torque (Lbs.)
Pitch	PS50193-100/ 011-02147-16	81
Pitch Trim	PS50193-69/ 011-2147-05	24
Roll	PS50193-70/ 011-02147-12	72

(8) Manual Electric Pitch Trim Speed Check

(a) Run the MANUAL ELECTRIC PITCH TRIM in one direction until it runs against the mechanical stop.

(b) Run the trim in the opposite direction and, using a stop watch or equivalent device, time the trim speed to the opposite mechanical stop. Verify that the elapsed time for full travel measures 30 ± 5 seconds.

(9) Autopilot Operation Checks

(a) Engage the Autopilot by pressing the AP key on the MFD. Push the HDG knob to synchronize the heading bug to the current aircraft heading. Select HDG mode by pressing the HDG key on the MFD. Verify the command bars are level and the control wheel is stationary. (There may be some roll motion in the yoke if the aircraft not perfectly level.)

(b) Turn the HDG knob to the left and right and check that the command bars move in the correct direction and the control wheel follows the command bars.

(c) Push and hold the CWS button and pull the control wheel to its aft limit. Release the CWS button. Verify the autopilot clutches re-engage and hold the wheel stationary.

(d) Apply continuous forward pressure, slowly moving the control wheel. After a brief delay, verify the trim wheel begins moving in a trim up direction.

(e) Grip the control wheel and press the CWS button. Verify trim motion stops. Move the control wheel to the forward limit and release the CWS button. Slowly move the control wheel aft. After a brief delay, verify the trim wheel begins to trim down. Relieve pressure on the wheel and verify the trim motion stops. Check that the trim wheel is free to turn. Hold the control wheel and press the A/P DISC TRIM INTER switch to disconnect the autopilot.

(10) VOR/LOC/GS Test

Check the VOR, ILS, and Glideslope functions with ramp test equipment. Operate the equipment according to the test equipment manufacturer's instructions.

NOTE: The PFD HSI does not show a course deviation bar unless a valid VHF NAV frequency is tuned.

(a) Simulate a VOR signal on a radial 360°. Tune the NAV1 and NAV2 receivers to the simulation frequency.

(b) Verify full scale deflection of VOR1 CDI by varying the selected course at least 10° left and right. Reset course pointers to 360°.

(c) Engage the autopilot and press the NAV key on the MFD. Using the CRS knob, alter course by 10° to the right. Verify the Flight Director and aircraft controls respond by flying to the VOR course. Repeat to the left.

(d) Set CRS course pointer to 360°.

(e) Repeat steps (b) - (e) with VOR2.

(f) Simulate a Localizer/Glideslope signal. Tune this signal on NAV1 and NAV2 receiver.

(g) Set the PFD HSI to LOC1 by pressing CDI soft key until LOC1 is selected. Use the test equipment to center the deviation bars (localizer and glideslope) on the PFD.

(h) Press the AP key on the MFD. Press the APR key on the MFD. Verify that the LOC and GS annunciations are green on the PFD. Apply right/left and up/down localizer/glideslope signals using the test equipment. Verify that the Flight Director and flight controls respond appropriately.

(i) Repeat steps (h) - (i) with LOC2.

(11) ESP Functional Check (Optional)

Required only for airplanes with the optional Enhanced AFCS features.

(a) Ensure the airplane has valid GPS reception.

(b) On the MFD, navigate to the SYSTEM SETUP page of the AUX group using the FMS knob. Press SETUP 2 softkey.

(c) Verify that there is a window for Stability & Protection and the status is “ENABLED”.

(d) Press MSG softkey. Verify that there are no “ESP FAIL” or “ESP OFF” system messages.

(e) Cover both GPS1 and GPS2 antennas.

(f) Verify on PFD 1 the ESP Roll Indices are displayed at 45° on the roll indicator.

(g) On the MFD, activate the cursor then rotate the large FMS knob to select Stability & Protection.

(h) Rotate the small FMS knob to change the status to “DISABLED”.

(i) Verify on PFD 1 the ESP Roll Indices are not displayed at 45° on the roll indicator on the Attitude Display. Press MSG softkey. Verify that there is an “ESP OFF” system message.

(j) Remove the cover from both GPS1 and GPS2 antennas.

(k) Remove power to the aircraft.

(l) Wait ~ 15 seconds before re-applying aircraft power.

(m) After AFCS PFT is complete and prior to GPS acquisition, verify on PFD that the ESP Roll Indices are displayed at 45° on the roll indicator.

(n) Once GPS has acquired satellites, verify on PFD that the ESP Roll Indices are not displayed at 45° on the roll indicator on the Attitude Display.

(o) On the MFD, navigate to the SYSTEM SETUP page of the AUX group using the FMS knob.

(p) Press Setup 2 softkey. Verify there is a window for Stability & Protection and the status is “ENABLED”.

(q) Press MSG softkey on PFD. Verify that there are no “ESP FAIL” or “ESP OFF” system messages.

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D. Components

See “Figure 10”.

(1) Garmin Servos

The Garmin servos are a two part system the GSA 8X Servo Actuator and the GSM 86 Servo Gearbox. See “Figure 10”.

(a) GSA 8X Servo Actuator

The GSA 8X servo actuator is an electromechanical unit that provides automatic control of a single flight axis (pitch, roll, or yaw), or their associated trims.The GSA 8X receives serial RS-485 data packets from two GIA 63W Integrated Avionics units, which contain data from the Flight Director, the GRS 77 AHRS, and the GDC 74 Air Data Computer. The drive clutch solenoid engages the GSM 86 drive clutch.The GSA 8X is mounted to the GSM 86 and can be removed and replaced without removing or de-rigging the GSM 86.

1) Removal

a) Turn the master switch off. Pull the AUTOPILOT (and, if appropriate, PITCH TRIM) circuit breaker(s).

b) Disconnect the servo actuator electrical harness connector.

c) Remove the bolts and washers securing the servo actuator to the servo gearbox and remove the servo actuator.

2) Installation

a) Grease the servo actuator output gear with AeroShell 33MS.

NOTE: Two O-ring seals are pressed into grooves in the back plate of the servo gearbox units. If during installation, removal, or reinstallation of a servo gearbox either of these O-rings becomes detached from the rear plate, carefully inspect and ensure there are no cuts or other damage to the O-ring(s). If no damage is found, reinstall the O-ring(s) in the rear plate, being careful not to stretch or compress it along its length while installing into the groove in the rear plate. If either O-ring is missing or damaged, replace with Garmin P/N 251-20031-00.

b) Position the servo actuator to the servo gearbox, secure with bolts and washers (4 ea.) as follows:

1] Insert all bolts and thread in approximately two turns by hand.

2] Starting with two opposite corner bolts, tighten all bolts until 1/64” to 1/32” (½ to 1 turn) from fully seated.

3] Verify that the GSA 8x is still slightly loose relative to the servo gearbox, ensuring there is still relative movement between the two units.

4] Starting with two opposite corner bolts, tighten all bolts to proper torque per “Figure 10”.

c) Connect the servo actuator electrical harness connector.

d) Push-in the AUTOPILOT (and, if appropriate, PITCH TRIM) circuit breaker(s).

(b) GSM 86 Servo Gearbox

See “Figure 9”

The GSM 86 servo gearbox is mounted to the aircraft structure, via a custom mounting bracket, and is responsible for transferring the output torque of the GSA 8X servo actuator to the mechanical flight control surface linkage. The GSM 86 houses its internal components within an cast aluminum housing. There are multiple variations of the GSM 86, all designed to accommodate various flight-control system interfaces and torque requirements. The slip-clutch settings and type of output capstan are described elsewhere in this chapter.

Seals

No scheduled maintenance of the GSM 86 seals is required. However, if during installation or replacement of the GSM 86 it is determined that an O-ring seal in the unit back plate is missing or damaged, the seal must be replaced with the appropriate Garmin part number.

NOTE: Two O-ring seals are pressed into grooves in the back plate of GSM 86 units. If either of the O-rings becomes detached from the back plate, carefully inspect the O-ring and ensure there are no cuts or any other damage. If no damage is found, reinstall the O-ring(s) in the back plate, being careful not to stretch or compress it along its length while installing into the groove in the back plate. If either O-ring is missing or damaged, replace with the appropriate Garmin part number item.

(c) Inspection

1) First 1000 hours, three years, or any time the GSA 8X servo actuator is separated from servo mount: clean and grease the servo actuator output gear with AeroShell 33MS.

2) Each twelve months, perform the following inspections:

a) Visually inspect:

1] The servos, connectors, support structures, and bridle cables to ensure that no corrosion, chafing, cracks, or other defects exist.

2] The servo bridle cables to ensure no fraying, corrosion, or other damage exists. Replace the cable if the condition is questionable.

3] The GFC 700 system wiring to ensure no chafing, wear, or other damage exists.

b) Manually move the ailerons (for roll servo), elevators (for pitch servo), elevator trim wheel (for pitch trim servo), and rudder pedals (for yaw servo) from stop to stop and observe the servo, capstan, and control surface rigging. Ensure there is no binding in the bridle cabling, and that the capstan pulleys rotate freely.

c) Check the tension of each servo bridle cable per “Figure 12”, ”Figure 13”, or ”Figure 14”, as appropriate.

d) Manually override the slip clutch every 12 months per the Autopilot Clutch Overpower Check under “Ground Checks” on page 221027.

(d) Return to Service

1) Original Servo(s) Reinstalled

No software loading is required if the removed servo(s) is re-installed. Continue to “Ground Checks” on page 221027.

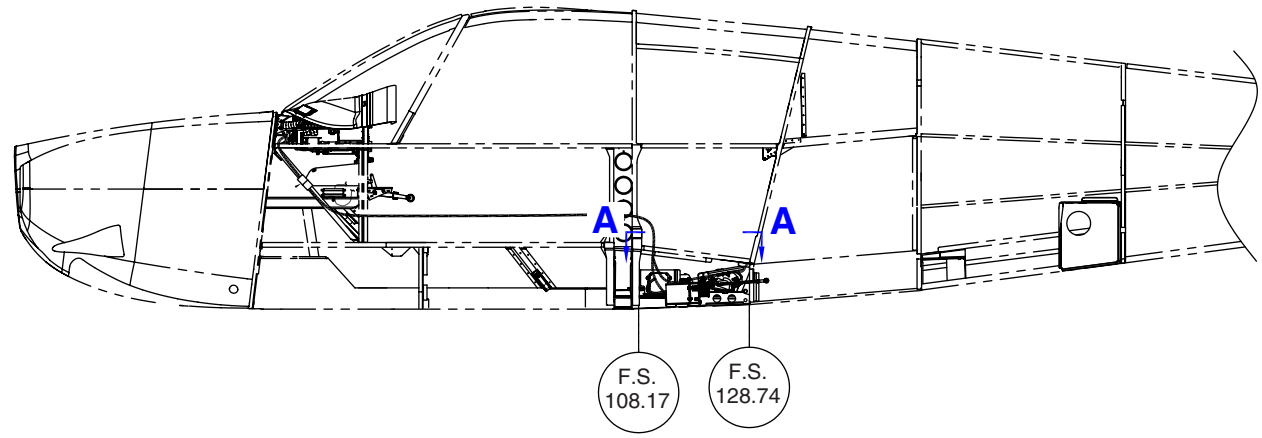
NOTE: This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process.

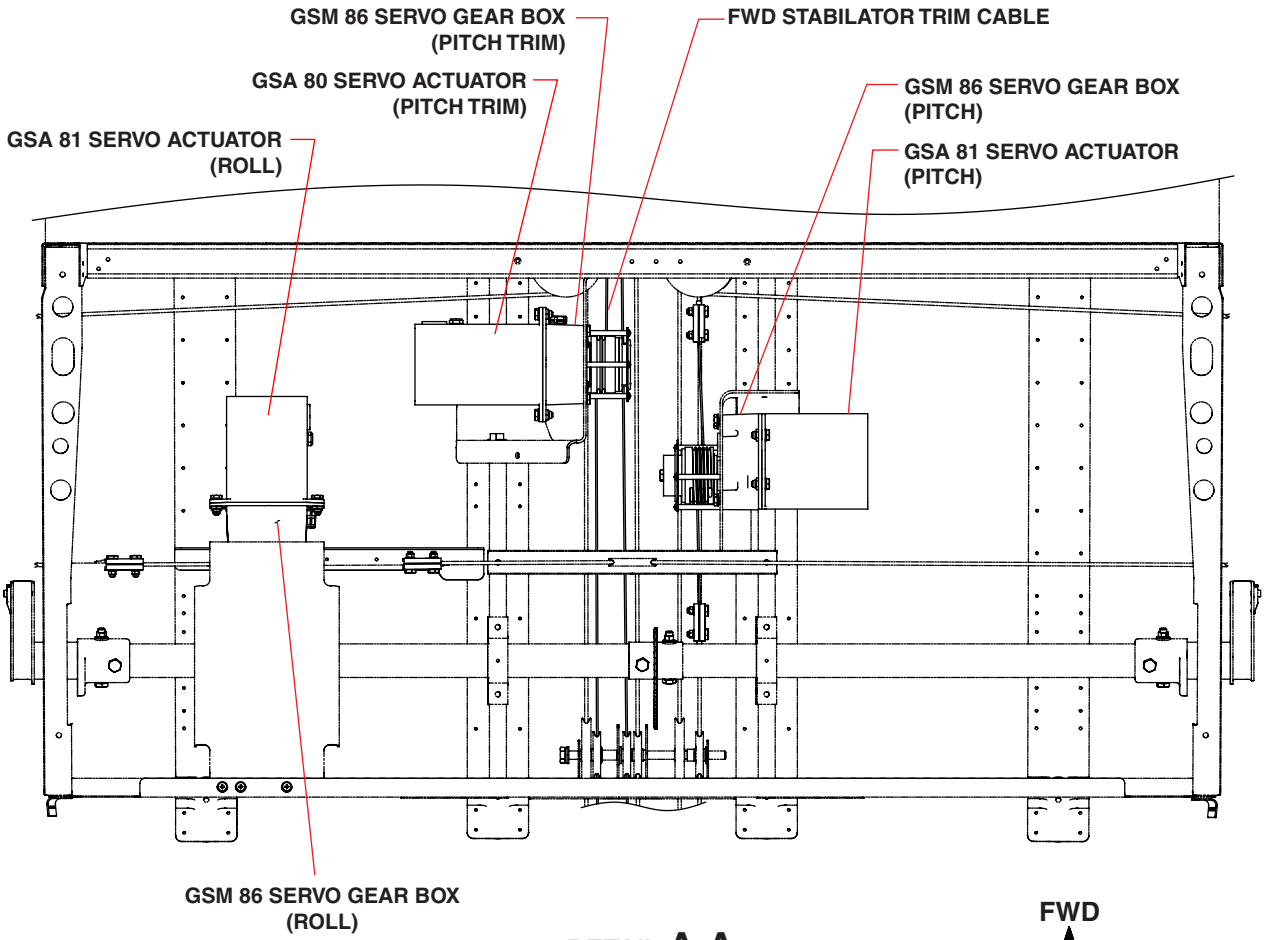
2) New, Repaired, or Exchanged Servo(s) Installed

If a new, repaired, or exchanged servo is installed, the correct software files and certification gains must be loaded to the unit. See G1000 NXi Software/Configuration Procedure in 4-25-03 and then continue to “Ground Checks” on page 221027.

3) If no other service is to be performed, continue to Ground Checks, above.

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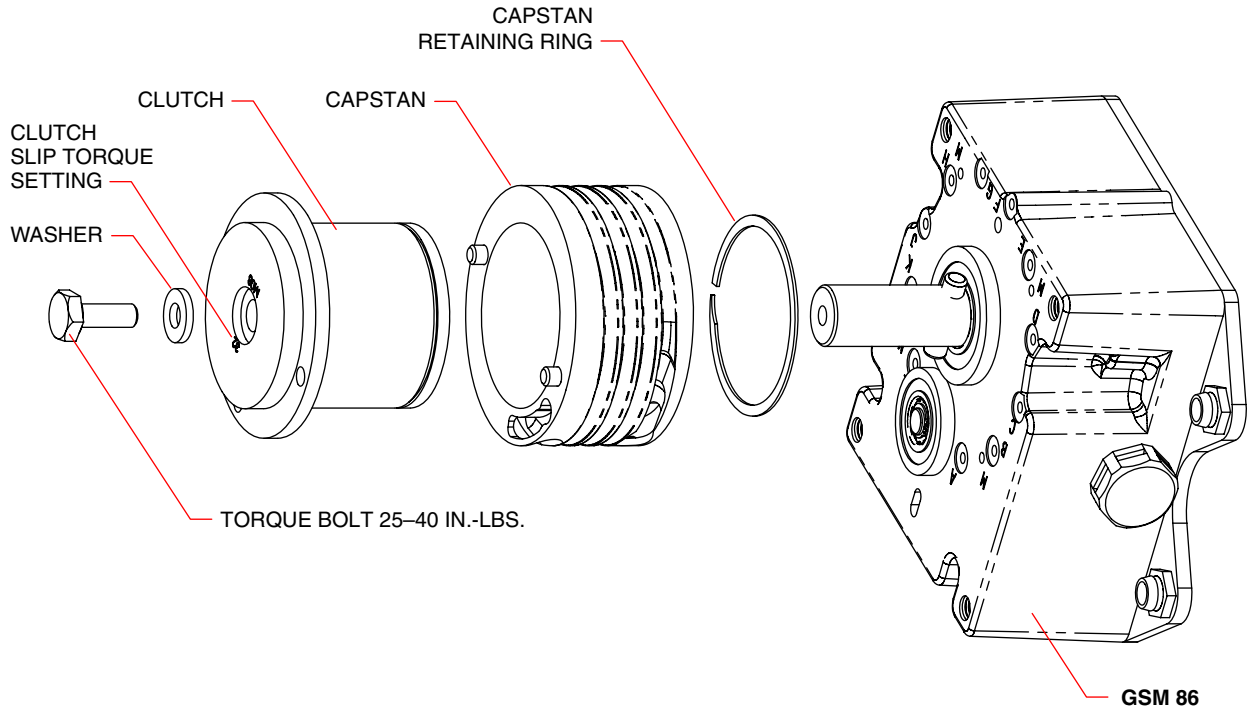




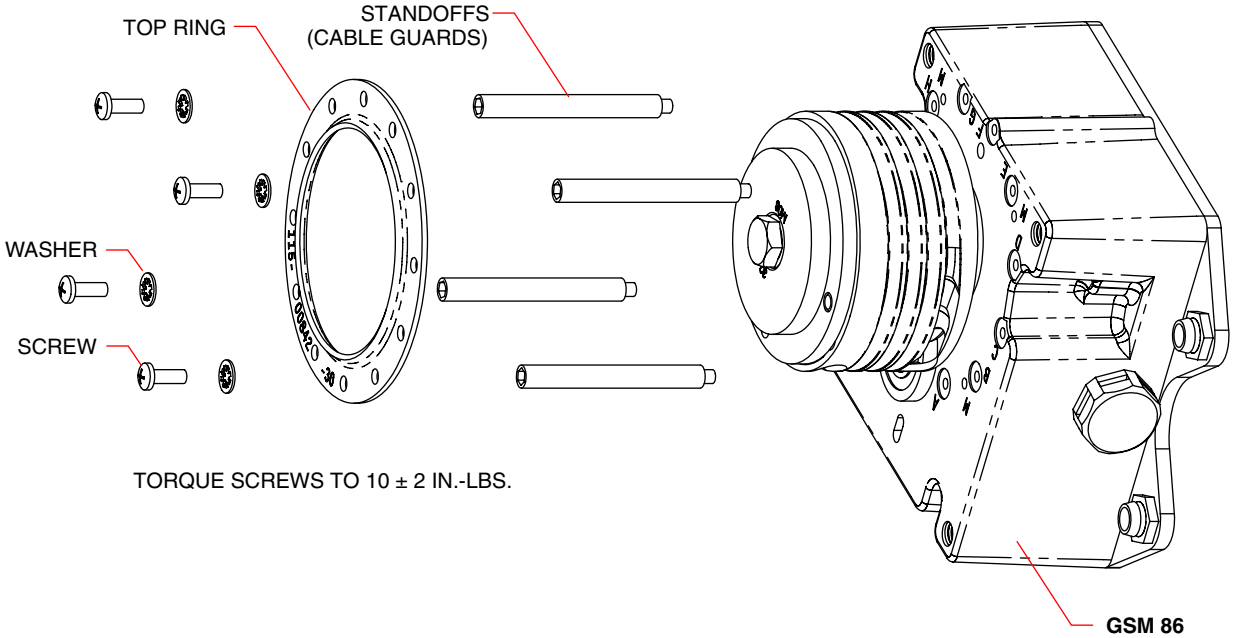
DETAIL A-A
ROTATED 90° CLOCKWISE

FWD
RHS

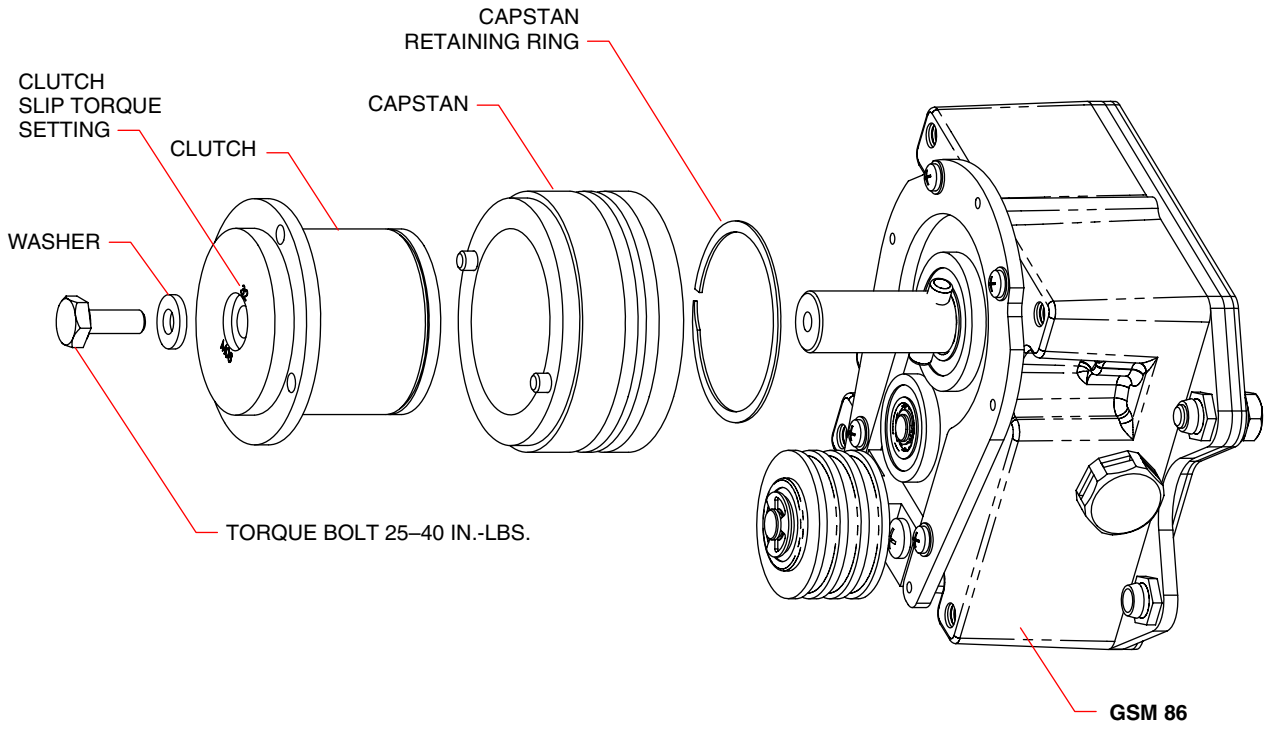
GFC 700 Autopilot Component Locator
Figure 9



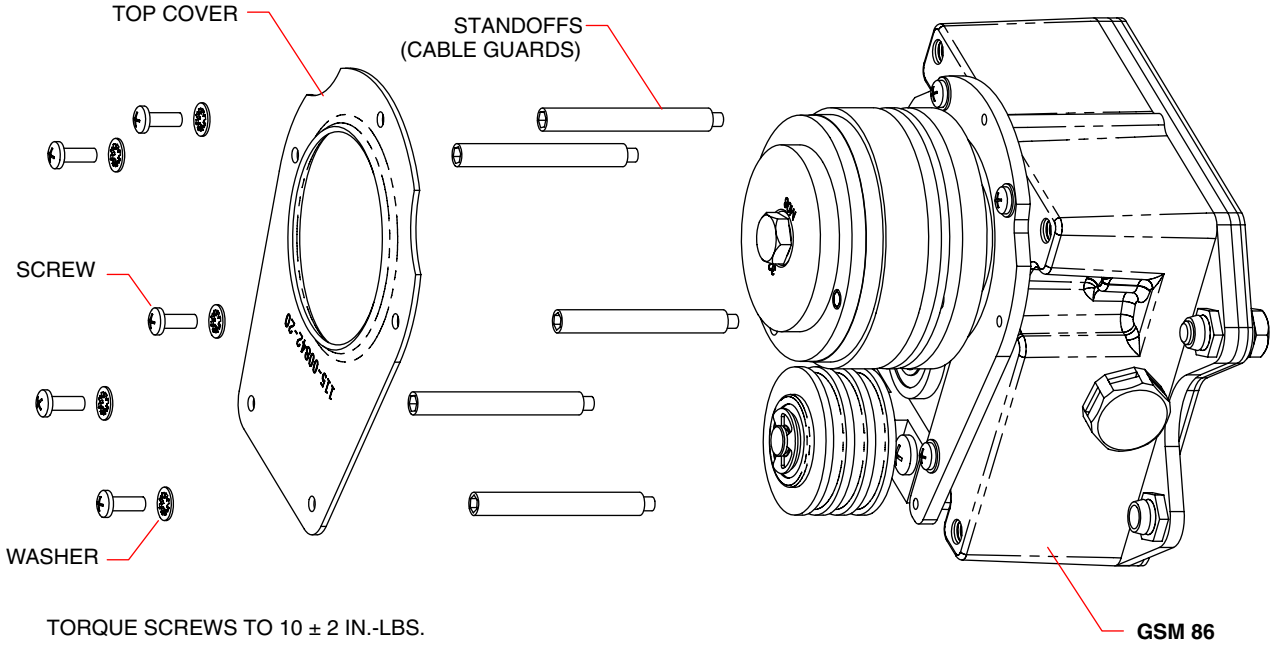
NOTE: RECORD NUMBER AND CLOCKING OF STANDOFFS PRIOR TO DISASSEMBLY.



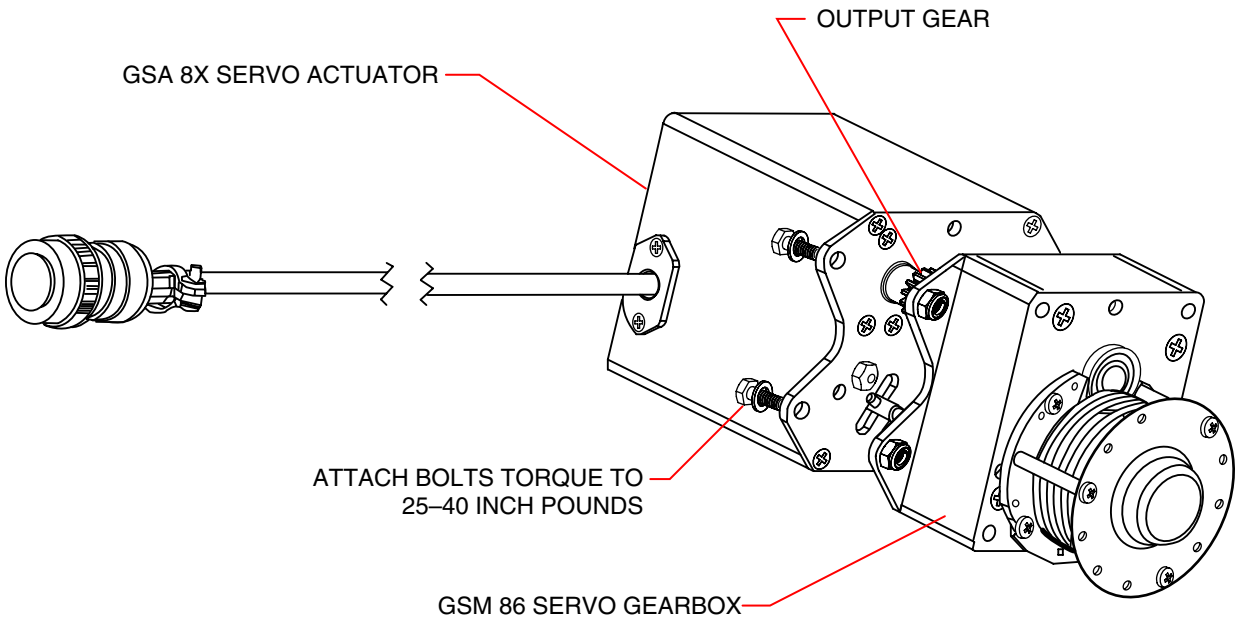
GSM 86 Servo Gearbox
Figure 10 (Sheet 1 of 2)



NOTE: RECORD NUMBER AND CLOCKING OF STANDOFFS PRIOR TO DISASSEMBLY.



GSM 86 Servo Gearbox
Figure 10 (Sheet 2 of 2)



GSA 8X/GSM 86 (Typical)
Figure 11

(2) Roll Servo

The roll servo is mounted at the far right side of the aircraft under the floor panel below the passenger seat. Access it from the baggage door. See “Figure 12”.

(a) Removal

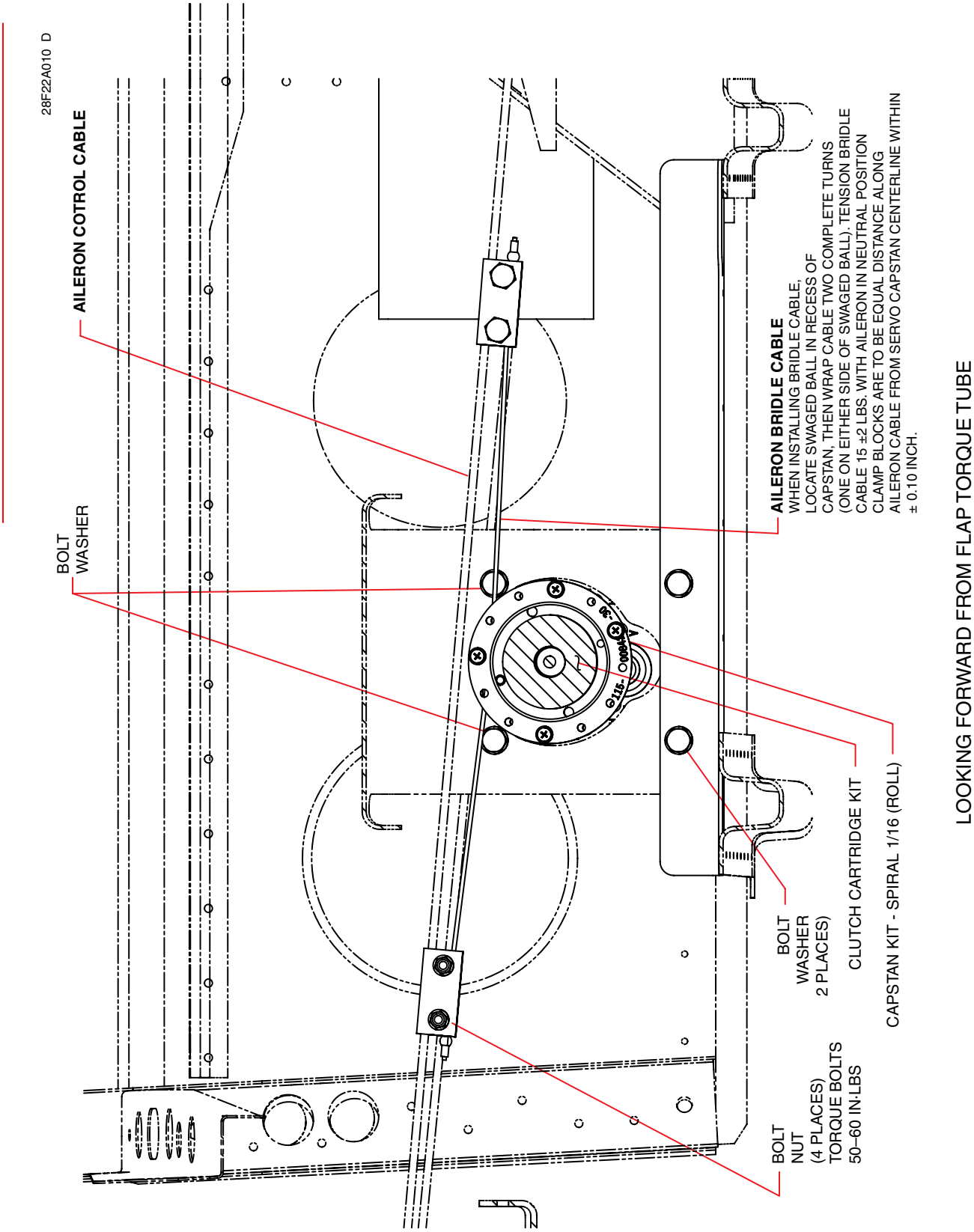
1) Remove the passenger seats.
2) Remove underlying carpet and floor panel.
3) Disconnect wiring harness and cables from the roll servo.
4) Remove nuts and bolts (2 ea.) securing each cable clamp (2) and remove cable clamps from aileron balance and autopilot bridle cables.
5) Remove bolts and washers (4 ea.) securing roll servo (GSA 81/GSM 86) to mounting brackets and remove roll servo with attached bridle cable.

(b) Installation

1) Rig ailerons per Aileron Control Rigging and Adjustment, 27-10-00.
2) Place the control column tee bar in full forward position and secure by use of a suitable tool or by placing weights on the aft side of the stabilator, if stabilator cables have been previously tensioned.
3) Lock the ailerons in neutral (i.e. - aligned with flaps) position using a suitable contour fixture at the inboard ends of the ailerons and the outboard ends of the flaps. Verify control wheels are centered and secure in that position.
4) Remove screws (4) and remove capstan cover and cable guards from servo.
5) Wrap autopilot bridle cable, as described in “Figure 12” and align capstan.
6) Replace cable guards and capstan cover, secure with screws (4).
7) Position roll servo (GSA 81/GSM 86) as shown in “Figure 12” and secure with bolts and washers (4 ea.) holding servo to mounting brackets. Torque bolts as specified in “Figure 12”.
8) Position cable clamps (2) as shown in “Figure 12” and tighten nuts and bolts (2 ea.). Adjust cable clamps in or out along the aileron balance cable to obtain the bridle cable tension specified in “Figure 12”.
9) Torque cable clamp bolts as specified in “Figure 12”.
10) Remove the locking fixtures at the inboard ends of the ailerons. Aileron neutral (i.e. - aligned with flaps) position should be maintained with the control wheels in neutral. A droop of 1/8 inch is allowable.
11) Remove the control wheel/tee bar locks. Check to ensure that the left aileron up and right aileron down stops are contacted simultaneously and vice versa. Adjust stops as required.
12) Rotate the left (pilot’s) control wheel in each direction until the bellcranks contact the stops. The sprocket stops on the tee bar shall not be contacted until additional “override” movement (cushion) of the wheel occurs. A “cushion” on 0.030 to 0.040 inches is to be maintained as measured between the sprocket pin and adjustable control wheel stop bolts.

13) Place the ailerons in the neutral (aligned with the flaps) position. For each aileron, from the neutral position, check that the “up” travel and the “down” travel are within the limits as specified in 27-00-00, Chart 2.
    a) Center bubble of a protractor over surface of aileron at neutral position. Note reading.
    b) Move aileron full up and down. Check degree of travel in each direction. Degree of travel on protractor is determined by taking the difference between protractor reading at neutral and up, and neutral and down. Bubble must be centered at each reading.

       When measuring “down” travel from the neutral position, a light “up” pressure shall be maintained at the center of the aft edge of the aileron. When measuring “up” travel from the neutral position, a light “down” pressure shall be maintained at the center of the aft edge of the aileron (at the “up” position only), just sufficient to remove the slack between the bellcrank and the aileron. Total free play measured at the aileron trailing edges shall not exceed 0.120 inches.
14) If either step “10)” thru “13)” reveal the aileron controls out of rig, repeat steps “1)” thru “9)”, above.
15) Connect autopilot harness.
16) Check autopilot system for binding and proper operation, see Ground Checks, above.
17) Check aileron controls for free and correct movement.
18) Perform Post-Maintenance Operational Checkout, above.
19) Replace floor panel and secure with screws.
20) Replace carpeting.
21) Replace the passenger seats.
22) Review “Return to Service” on page 221035, to determine if software needs to be reloaded.



Roll Servo Installation
Figure 12

(3) Trim Servo

The trim servo is located at the airplane centerline just aft of F.S. 127.175, behind the cabin rear closeout panel. The left stabilator trim cable wraps around the servo ilder pulley and the servo capstan. See “Figure 9” and “Figure 13”.

(a) Removal

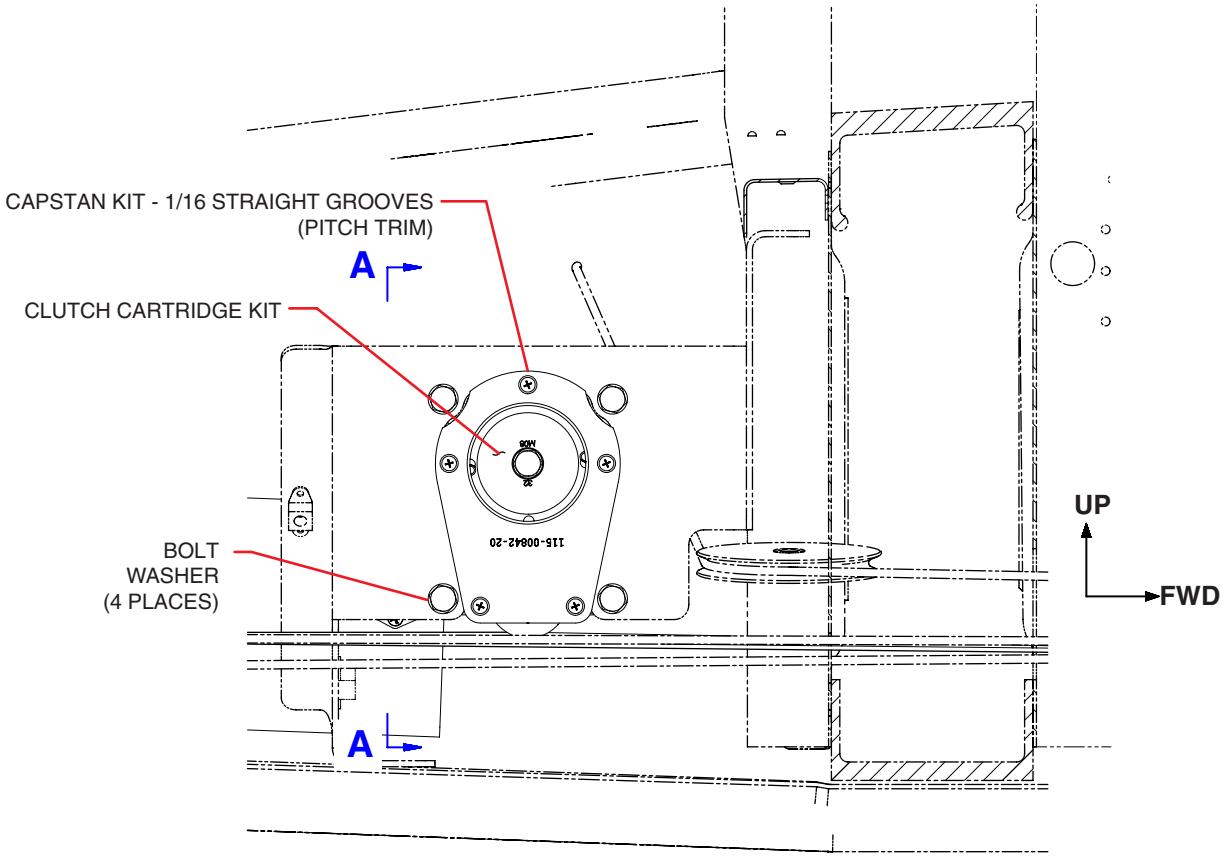
1) Attach a tail stand under the tail skid.
2) Remove the closeout panel at the back of the passenger cabin.
3) Remove the baggage compartment carpet and control cable inspection access covers.
4) Tie a pull rope to the left stabilator trim cable exposed beneath the floor panel and tie-off the pull rope to structure aft.
5) Tie a pull rope to the left stabilator trim cable aft of the turnbuckle in the rear fuselage aft of the trim servo and tie-off the pull rope to structure forward.

NOTE: The pull ropes apply tension to the trim cables to prevent the cables from unwrapping from the trim wheel drum or the trim barrel, and to prevent the cables from fouling at any of the pulleys.

6) Slack-off the turnbuckle in the left stabilator trim cable segment aft of the trim servo sufficient to relieve tension on the left stabilator trim cable as it wraps around the trim servo idler pulley and capstan.
7) Disconnect the autopilot harness.
8) Remove the capstan cover and cable guards (5) by removing the retaining screws (5).
9) Remove the cable from capstan and pulley
10) Remove the bolts and washers (4 ea.) securing the trim servo (GSA 80/GSM 86) to its mounting bracket and remove the trim servo.

(b) Installation

1) With the capstan cover and cable guards removed, position the trim servo (GSA 80/GSM 86) as shown in “Figure 13”.
2) Secure with bolts and washers (4 ea.) holding servo to mounting brackets. Torque bolts as specified in “Figure 13”.
3) Drape the slack left trim cable over the servo capstan.
4) Wrap the cable around the capstan and pulley two (2) complete turns while pressing the trim cable into its groove as shown in “Figure 13”.
5) Tighten turnbuckles and remove tie ropes.
6) Rig stabilator trim per Stabilator Trim Rigging and Adjustment, 27-30-00.
7) Connect the autopilot harness.
8) Perform Post-Maintenance Operational Checkout, above.
9) Check stabilator trim controls for free and correct movement.
10) Install the capstan cover and cable guards (5) as shown in “Figure 13”.
11) Replace cabin closeout panel.
12) Replace control cable inspection access covers in baggage compartment floor and replace baggage compartment carpet.
13) Reinstall rear seats.
14) Remove tail stand.
15) Review “Return to Service” on page 221035, to determine if software needs to be reloaded.



Trim Servo Installation
Figure 13

(4) Pitch Servo

The pitch servo is located at the centerline in the aft fuselage, between bulkheads F.S. 127.175 and F.S. 108.758. Access is through the cabin rear closeout panel. A bridle cable and clamps attach the servo capstan to the right (upper) stabilator cable. See “Figure 9”, and “Figure 14”.

(a) Removal

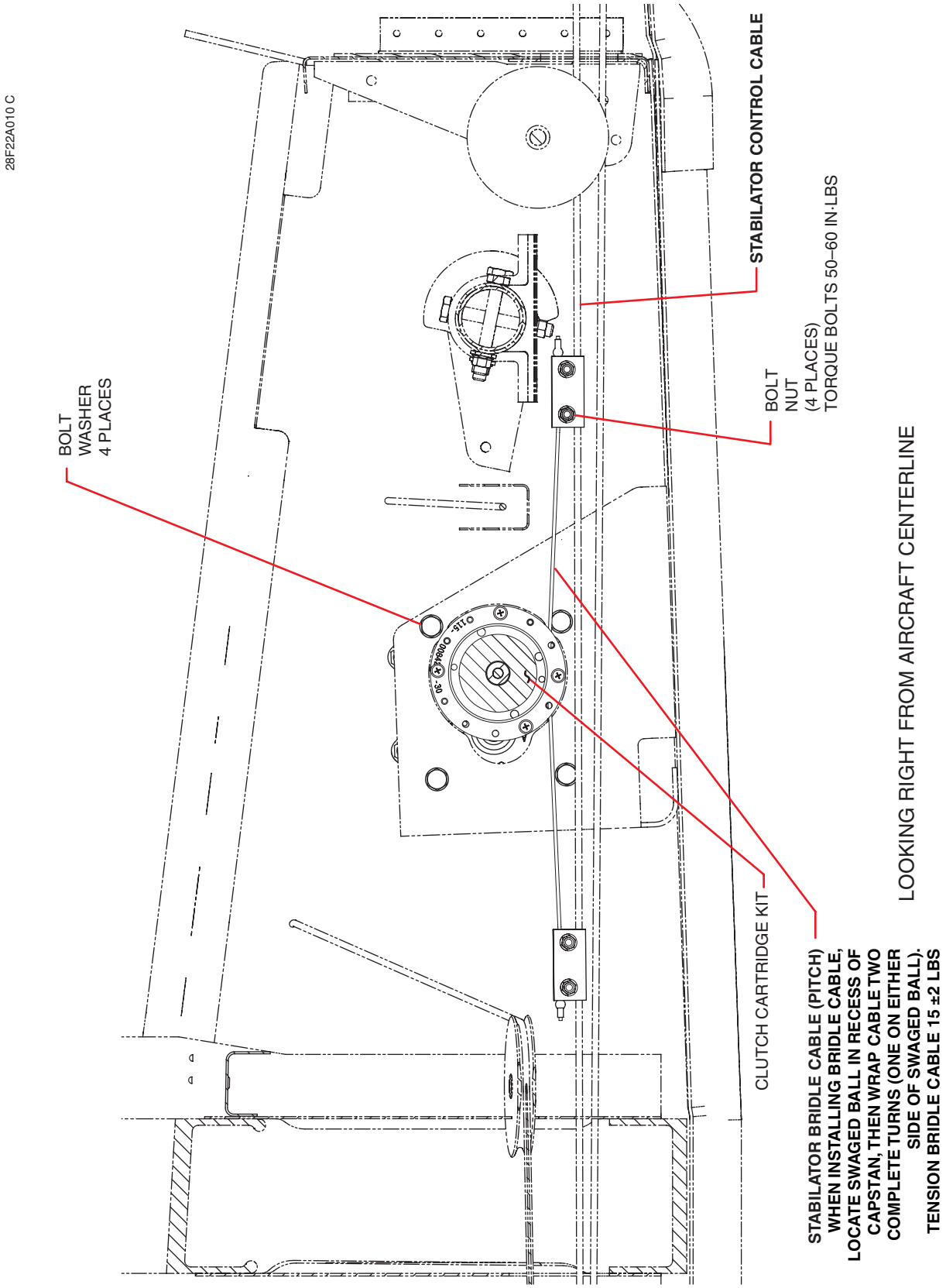
1) Attach a tail stand under the tail skid.
2) Remove the cabin rear closeout panel.
3) Crawl into the tailcone to the pitch servo.
4) Disconnect autopilot harness connector.
5) Remove nuts and bolts (2 ea.) securing each cable clamp (2) and remove cable clamps from upper and lower stabilator cables and autopilot bridle cable.
6) Remove cotter pin from bridle cable pulley and pull upper portion of bridle cable free of the pulley.
7) Remove bolts and washers (4 ea.) securing pitch servo (GSA 81/GSM 86) to mounting bracket and remove pitch servo with attached bridle cable.

(b) Installation

1) Rig stabilator controls per Rigging and Adjustment, Stabilator Controls, 27-30-00.
2) Remove screws (4) and remove capstan cover and cable guards from servo.
3) Wrap autopilot bridle cable, align capstan, and press center ball in place in “Figure 14”
4) Replace cable guards and capstan cover, secure with screws (4).
5) Position pitch servo (GSA 81/GSM 86) as shown in “Figure 14” and secure with bolts and washers (4 ea.) holding servo to mounting brackets. Torque bolts as specified in “Figure 14”.
6) Lead upper portion of bridle cable through pulley as shown in “Figure 14”. Reinstall cotter pin.
7) Position cable clamps (2) as shown in “Figure 14” and tighten nuts and bolts (2 ea.).
8) Adjust cable clamps in or out along the stabilator cable to obtain the bridle cable tension specified in “Figure 14”.
9) Torque cable clamp bolts as specified in “Figure 14”.
10) Connect autopilot harness.
11) Check autopilot system for binding and correct operation, see Ground Checks, above.
12) Check stabilator controls for free and correct movement.
13) Reinstall and secure the cabin rear closeout panel.
14) Remove tail stand.
15) Review “Return to Service” on page 221035, to determine if software needs to be reloaded.

(5) Flap Position Sensor (Optional)

See 27-50-00.



Pitch Servo Installation
Figure 14

WIRING

Airplanes equipped with Garmin GFC 700 Automatic Flight Control Systems have additional EMI shielding on some harness assemblies. If repair is required, see EMI Shielding in 20-97-00.

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CHAPTER

23

COMMUNICATIONS

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CHAPTER 23

LIST OF EFFECTIVE PAGES

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	2	Mar 6/14			
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	2	Mar 6/14			
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	3	Nov 15/17			
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CHAPTER 23 - COMMUNICATIONS

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GENERAL

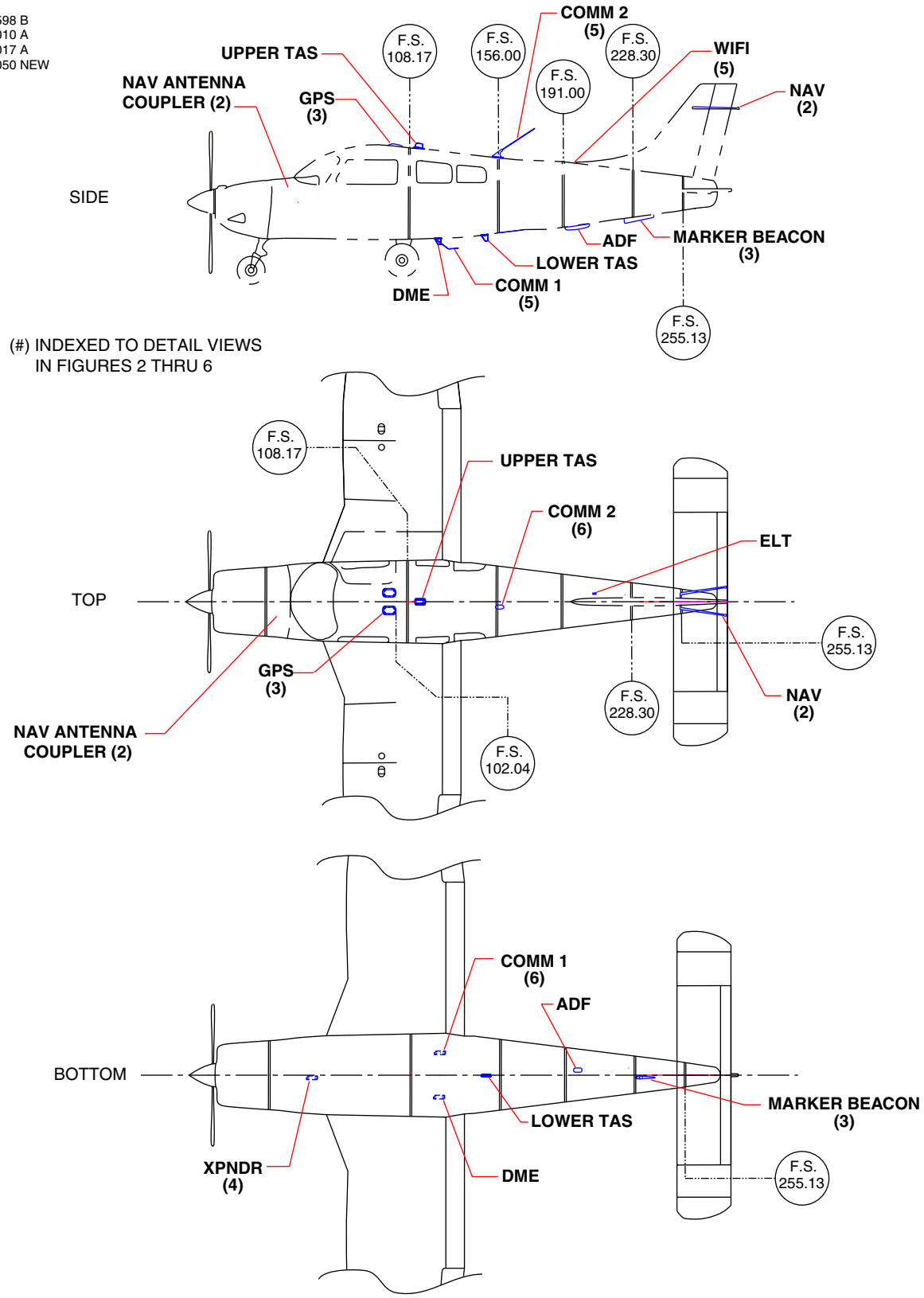
1. Radio Master Switch

   A separate master switch for the avionics is located in the switch panel in the center of the instrument panel. Both this switch and the battery master switch in the overhead panel must be ON for the avionics to work.

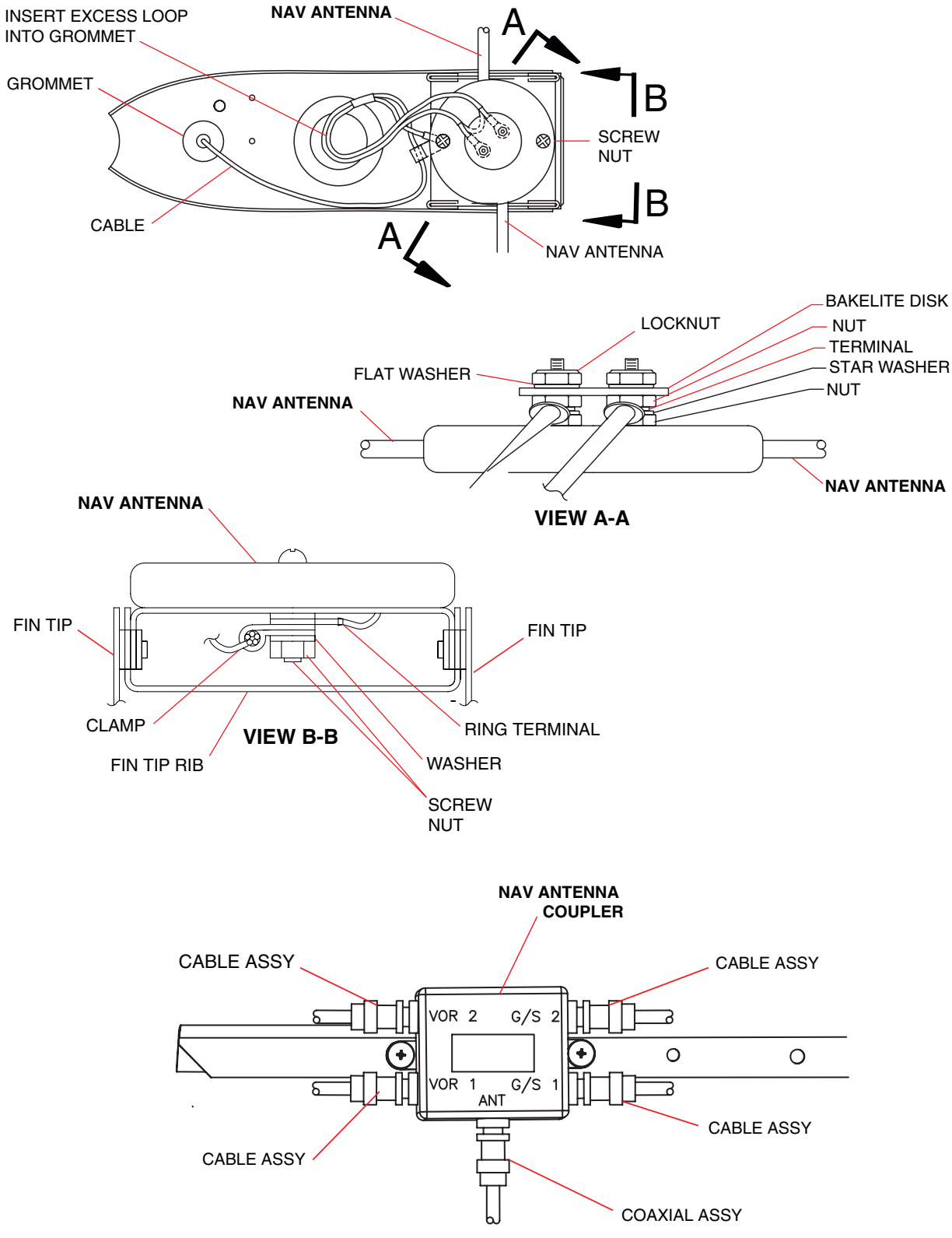
2. Antennas

   See “Figure 1” on page 23002.

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108010 A
108017 A
108050 NEW

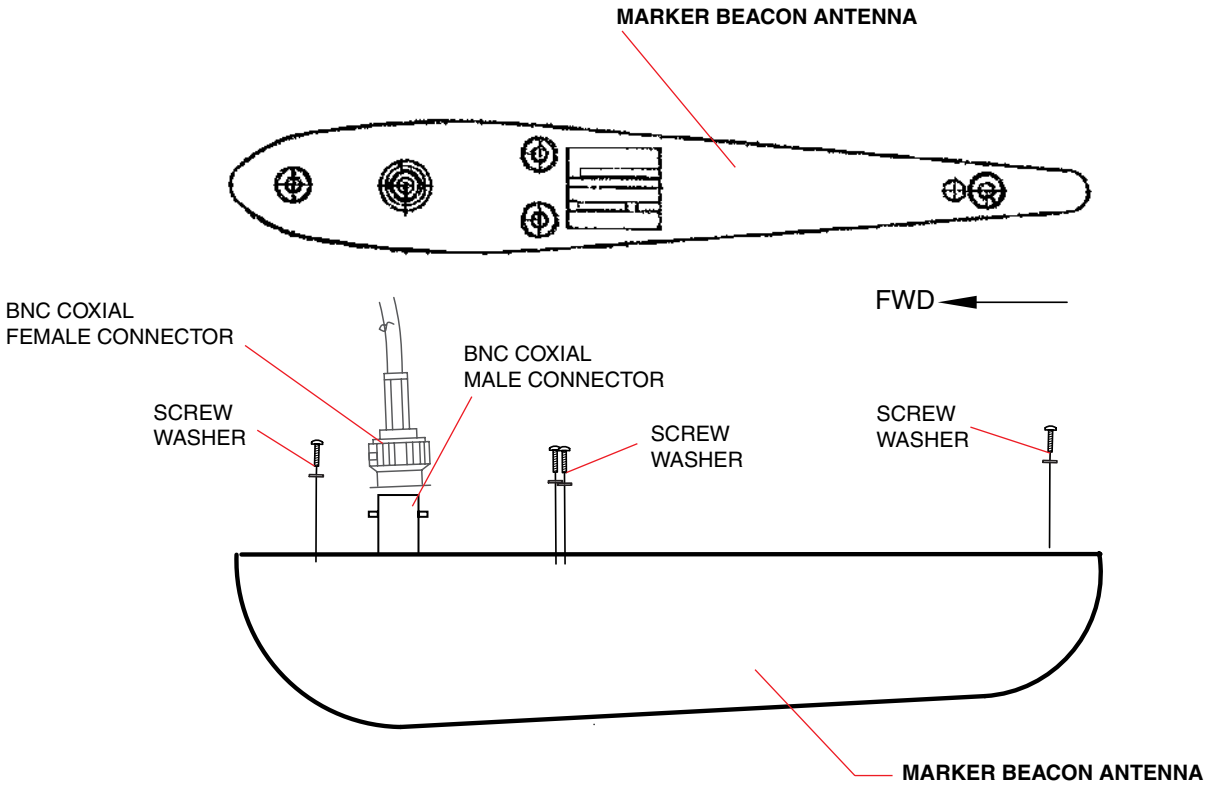


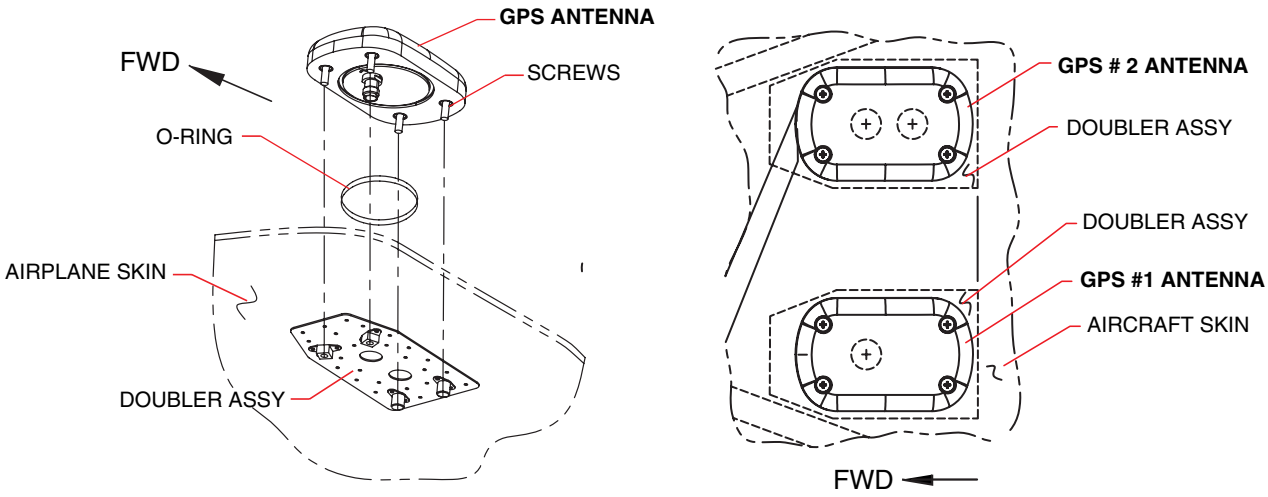
Antenna Installations
Figure 1



Antenna Installation
Figure 2

Effectivity
with Garmin G1000 or G1000 NXi

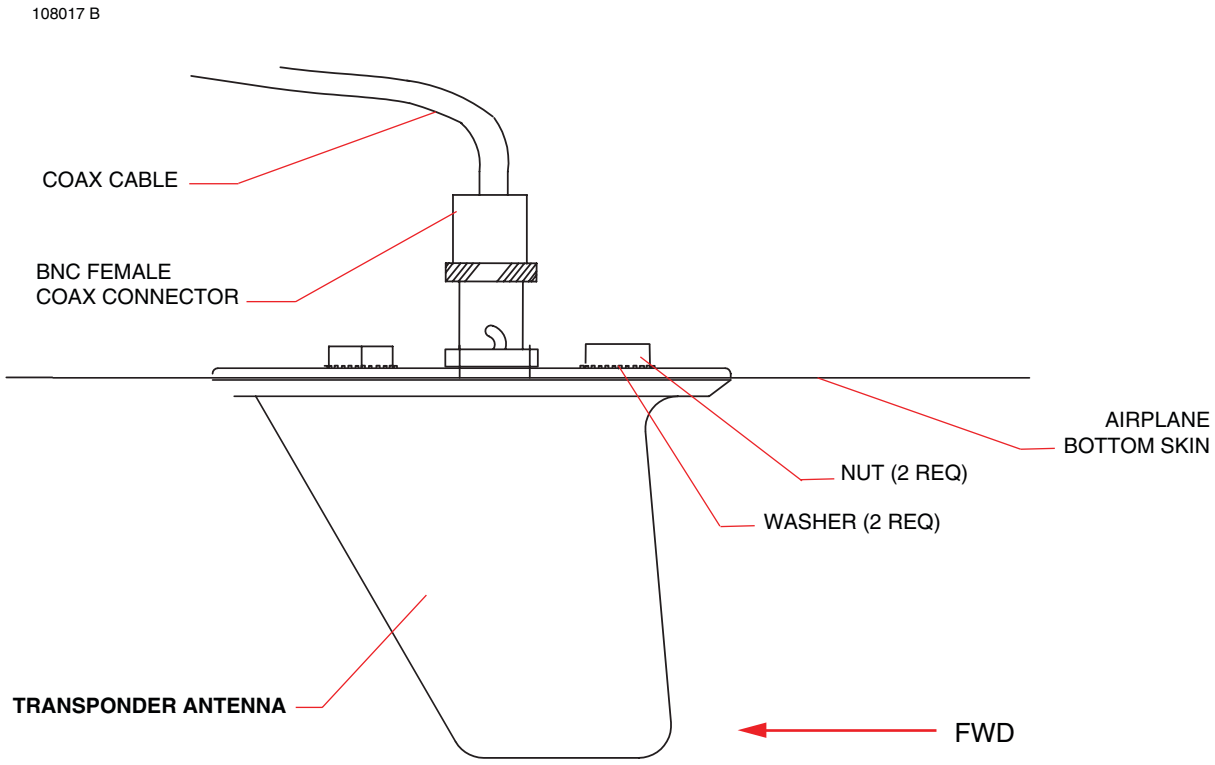




GPS ANTENNA INSTALLATION

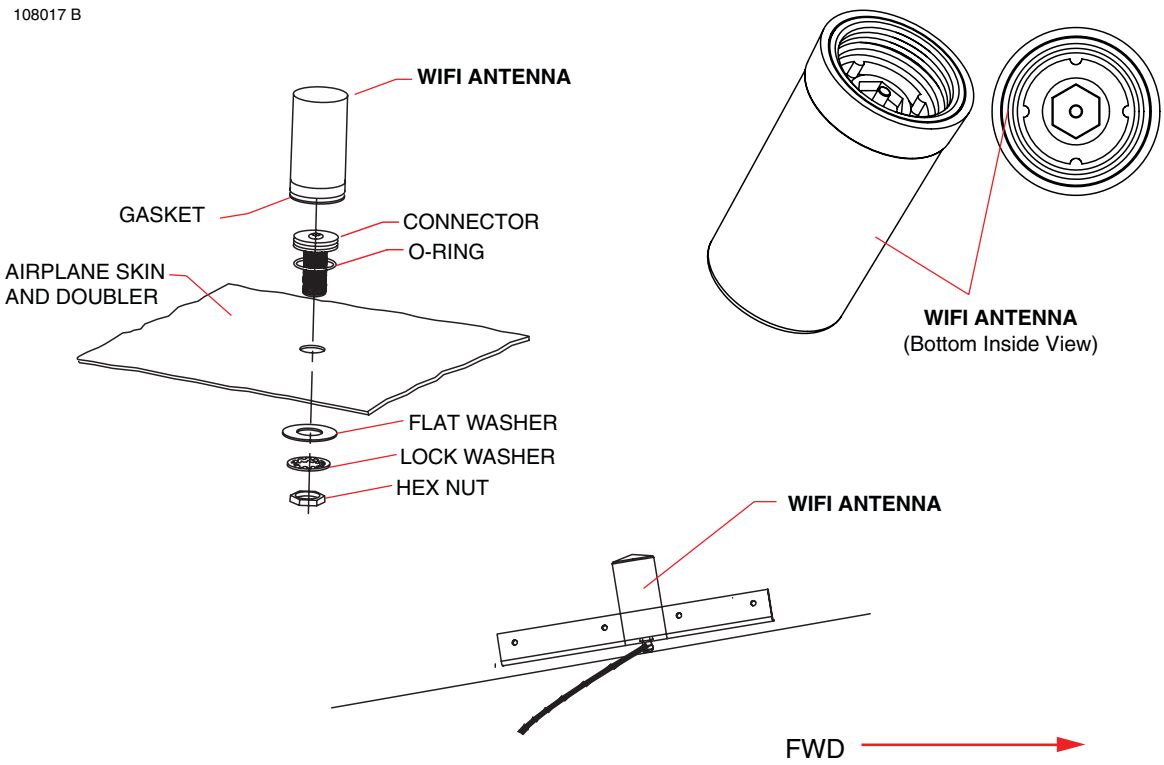
Effectivity
with Garmin G1000 or G1000 NXi

Antenna Installation
Figure 3



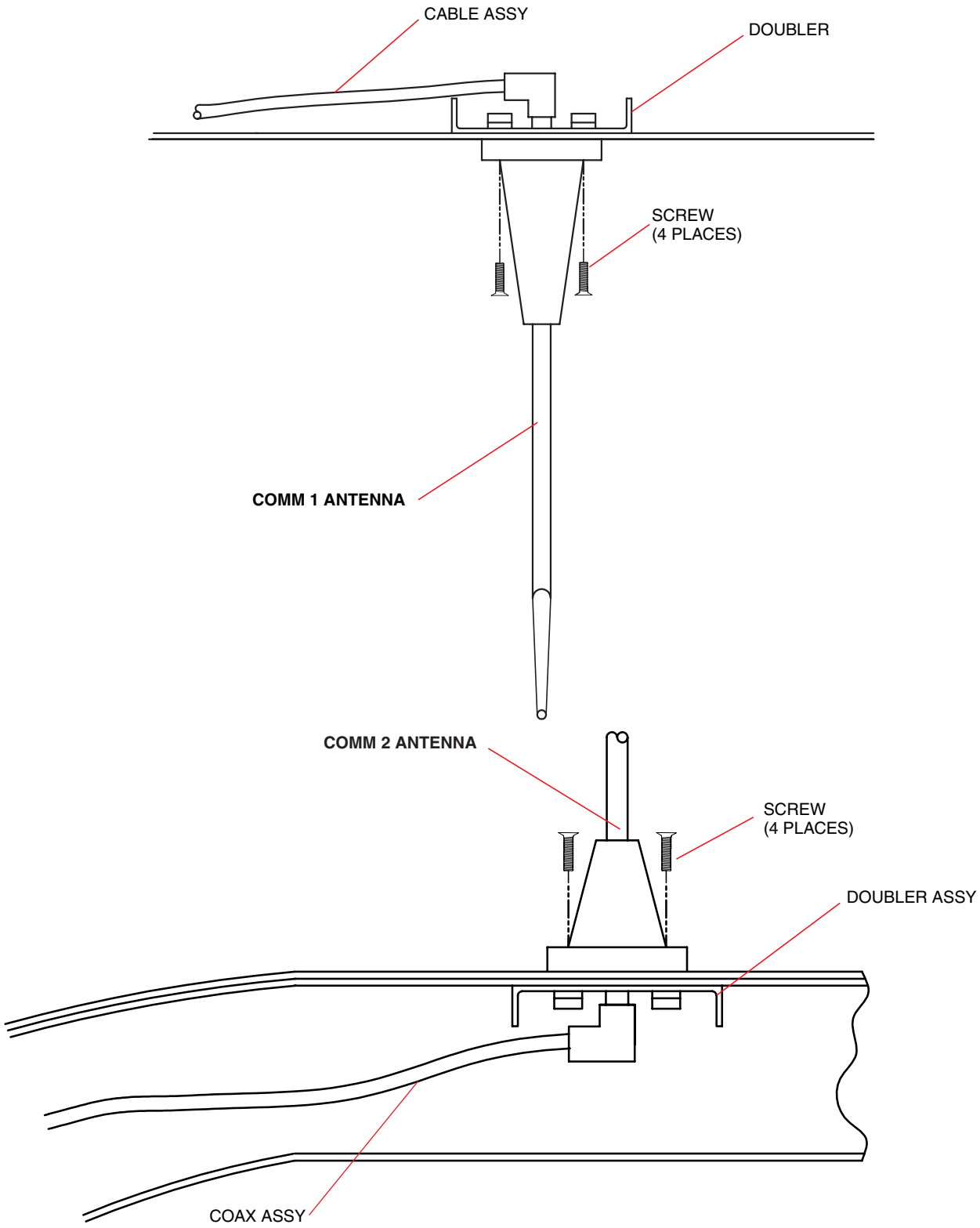
Antenna Installation
Figure 4

Effectivity
with Garmin G1000 or G1000 NXi



Antenna Installation
Figure 5

Effectivity
with Garmin G1000 NXi
2843852, 2843854–2843999,
2843823, 2881001 and up



Antenna Installation
Figure 6

Effectivity
with Garmin G1000 or G1000 NXi

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AUDIO INTEGRATING

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Garmin GMA 340 Audio Panel

See “Figure 1”.

The GMA 340 is factory-installed in Standard, Avidyne Entegra and Garmin G500 equipped airplanes, S/N’s 2843203 and up.

The Garmin GMA 340 Audio Panel integrates the audio switching, amplifier, and intercom system with a marker beacon receiver. Maintenance of the GMA 340 is “on condition” only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop in accordance with the GMA 340 Audio Panel Maintenance Manual (Garmin P/N 190-00149-02).

Information provided in this manual is intended to aid the removal and installation of the GMA 340 unit and its associated wiring and to permit basic system functional test and adjustment.

A. Removal and Installation

See Rack-Mounted Avionics, Removal and Installation, 39-10-00.

B. System Functional Test

See “Figure 2” on page 23503.

(1) On Ground

The following known good equipment is required prior to performing this test: microphone, headset, speaker and avionics receivers.

(a) Lamp Test

1) Apply power to the unit by rotating the pilot intercom knob clockwise.

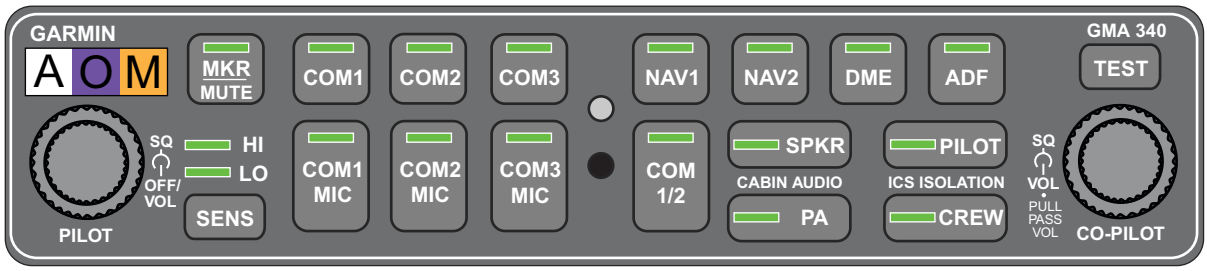
2) The test button checks the internal LED annunciator and marker beacon lamps. Press TEST to confirm operation of the LED’s. Cover the photocell with a finger and observe that the LED annunciators dim automatically. Check the front panel back-lighting and dimming function. Each annunciator contains a lamp for illumination.

(b) Fail-safe Operation Check

1) Turn the unit off by rotating the pilot intercom knob counterclockwise.

2) Check fail-safe operation by exercising the COM 1 microphone, microphone key and audio over the headphones.

3) Turn the unit back on to continue testing.



GMA 340 Audio Panel
Figure 1

Effectivity
Standard, Avidyne Entegra,
Garmin G500, 2843203 and up

(c) Transceiver Operational Check

1) Perform a ramp test radio check by exercising the installed transceivers, microphone, microphone key and audio over the headphones and speaker.

2) Verify that communications are loud and clear and push-to-talk (PTT) operation is correct.

(d) Intercom System (ICS) Check

1) Set the intercom to the ALL mode (Crew and Pilot LED's off).

2) Plug in headsets at each ICS position.

3) Adjust squelch and volume for each position and verify that the ICS is working properly.

4) Check Pilot and Copilot ICS positions for isolation and proper operation of volume and squelch controls.

5) Press the PA button. Verify that microphone audio is heard over the speaker.

(e) Aircraft Receivers Check

1) Select the audio source corresponding to each installed avionics unit and check for audio over the headsets.

2) Check for Pilot/Copilot audio isolation when pressing the COM 1/2 button.

3) Press the SPKR button and verify that any selected audio is heard over the speaker.

(f) Music System Check

Connect a stereo music source to MUSIC 2. Press the CREW button to set the ICS to the crew mode. Verify that stereo audio is heard in the passenger headsets only.

(2) In Flight

Verify proper operation of the marker lamps and marker audio, including the marker audio mute function. Check proper operation of the marker sensitivity selection (using the SENS button) by flying towards the outer marker position initially using HI sensitivity. When the OM audio is just barely audible in the headset, switching to LO sensitivity should reduce or eliminate the audio.

C. Adjustment

See "Figure 2" on page 23503.

CAUTION: USE ONLY A 2 MM (MAX BLADE WIDTH) FLAT-BLADE NONCONDUCTIVE SCREW DRIVER AS AN ADJUSTMENT TOOL. BE CAREFUL WHEN INSERTING ADJUSTMENT TOOL THROUGH THE TOP COVER. THE UNIT MAY BE DAMAGED IF AN ADJUSTMENT TOOL IS ACCIDENTALLY FORCED AGAINST UNINTENDED COMPONENTS OR CIRCUIT BOARD PATHS.

The following adjustments can be made through access holes in the top cover of the GMA 340:

(1) Marker beacon audio level.

Counter-clockwise adjustment increases the marker audio level.

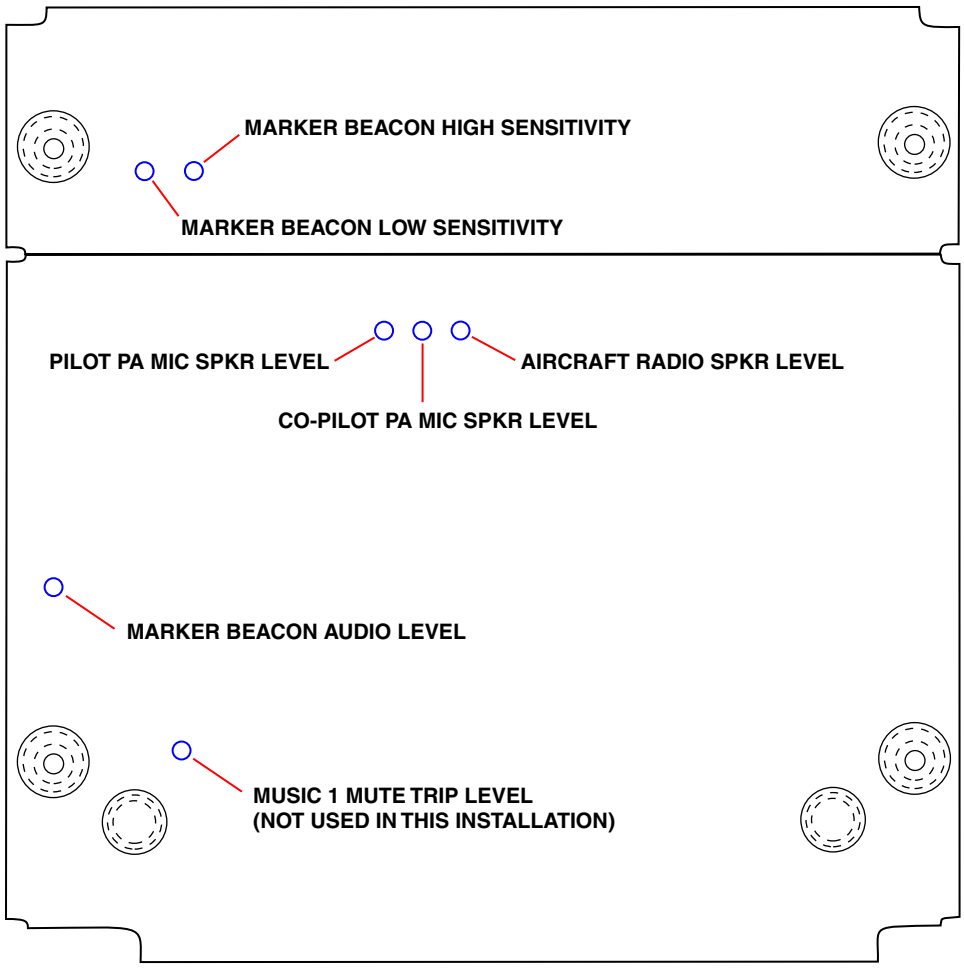
(2) Marker beacon sensitivity.

Clockwise (CW) adjustment increases the sensitivity. LOW sensitivity can be adjusted without affecting HIGH sensitivity setting. Adjusting HIGH sensitivity will, however, affect the LOW sensitivity. If the HIGH sensitivity setting is adjusted, then the LOW sensitivity setting should be checked and adjusted afterwards, as needed. If your GMA-340 top cover does not have the marker beacon sensitivity adjustment access holes as indicated in Figure 2, and you need to adjust the sensitivity, contact Garmin for instructions.

(3) Aircraft radio speaker output level.

(4) Pilot PA microphone speaker output level.

(5) Copilot PA microphone speaker output level.



GMA 340 Audio Panel Adjustments
Figure 2

Effectivity
Standard, Avidyne Entegra,
Garmin G500, 2843203 and up

2. Garmin GMA 1347 Audio Control Panel

The GMA 1347 is factory installed in Garmin G1000/G1000 NXi equipped airplanes S/N's 2843701, 2843703–2843999. The GMA 1347 integrates NAV/COM digital audio, intercom system and marker beacon controls. The Audio Panel communicates with both GIA 63Ws using RS-232 digital interface. The GIA 63W contains the GPS receiver, VHF COM/NAV receivers.

Maintenance of the GMA 1347 is "on condition" only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop. (See 34-25-01 or 34-25-02, as appropriate.)

Information provided in this manual is intended to aid the removal and installation of the GMA 1347 unit and its associated wiring and to permit basic system functional test. In addition, a detailed electrical schematic is provided in 91-23-50.

A. Removal and Installation

See Components, GMA 1347, Removal and Installation, in 34-25-01 or 34-25-02, as appropriate.

B. System Functional Test

See LRU Test Procedures, GMA 1347, in 34-25-01 or 34-25-02, as appropriate.

3. Garmin GMA 1360 Audio Control Panel

The GMA 1360 is factory installed in Garmin G1000 NXi equipped airplanes S/N's 2843823, 2881001 and up. The GMA 1360 integrates NAV/COM digital audio, intercom system and marker beacon controls. The Audio Panel communicates with both GIA 64 Integrated Avionics Unit using RS-232 digital interface. The GIA 64 Integrated Avionics Unit contains the GPS receiver, VHF COM/NAV receivers.

Maintenance of the GMA 1360 is "on condition" only and, with the exception of swapping complete units, should be performed only by a qualified avionics shop. (See 34-25-03.)

Information provided in this manual is intended to aid the removal and installation of the GMA 1360 unit and its associated wiring and to permit basic system functional test. In addition, a detailed electrical schematic is provided in 91-23-50.

A. Removal and Installation

See Components, GMA 1360, Removal and Installation, in 34-25-03.

B. System Functional Test

See LRU Test Procedures, GMA 1360 in 34-25-03.

4. Aft Passenger Intercom (Optional)

A. Description

S/N's 2843711 and up may be equipped with an aft passenger intercom with two way radio communications capability. This system consists of an instrument panel-mounted intercom (Flightcom Model 403) and intercom jack assemblies with integral push to transmit switches located above the armrest in each interior side panel adjacent to the aft seats. The system is designed and wired such that the pilot has transmit priority over the passengers/copilot.

B. Troubleshooting

See "Chart 1".

CHART 1 (Sheet 1 of 3)
TROUBLESHOOTING - AFT PASSENGER INTERCOM SYSTEM

Trouble	Cause	Remedy
Difficulty transmitting on the radio.	Microphone jack wiring may be loose or connected incorrectly.	Verify integrity of wiring. Repair as required. If wiring connections are good, compare wiring to electrical schematic and correct if necessary.
Sidetone is distorted while transmitting on radio, even though tower receives you loud and clear.	Distorted sidetone while transmitting may be due to improper shielding of aircraft antenna cable.	Check the shielding of antenna cable.
No sidetone while transmitting on radio.	Radio sidetone level may be adjusted too low or turned off.	Increase radio sidetone output.

CHART 1 (Sheet 2 of 3)
TROUBLESHOOTING - AFT PASSENGER INTERCOM SYSTEM

Trouble	Cause	Remedy
The unit appears completely dead.	Voltage may not be available through the power supply lead (pin 20).	Restore voltage by reconnecting pin 20.
	Fuse or circuit breaker may be blown or pulled.	Replace fuse or reset circuit breaker.
	One or more wires may be loose or connected incorrectly.	Check integrity of wiring. Repair as required. If wiring connections are good, compare wiring to electrical schematic and correct if necessary.
	Headphones may not be plugged in completely.	Reseat headphone jack(s).
	Headset volume may be turned down.	Turn up headset volume.
	Headphone plugs may be corroded.	Clean headset plugs with steel wool or crocus paper.
	Aircraft master switch may be turned OFF.	Turn ON aircraft master switch.
	Squelch may be set incorrectly. (Model 403 intercom is silent when muted.)	Reset squelch control.
High pitch whine that varies with engine RPM.	Microphone jacks may be shorting to aircraft frame.	Verify microphone jacks are properly insulated from aircraft frame.
	The harness connection to Pin 13 on the intercom is loose or disconnected.	Verify integrity of Pin 13 and the harness connector. Correct as required.
	The ground lead from Pin 1 on one of the microphone jacks may be loose or disconnected.	Reconnect ground lead to Pin 1 of affected microphone jack.
	Aircraft battery may need to be replaced.	Replace the aircraft battery.
High pitch whine or squeal is always present.	If the noise is coming from the turn and bank indicator or other electric gyro, the intercom may be shorting to the airframe.	1. Verify microphone jacks are properly insulated from aircraft frame. 2. Check integrity of intercom ground (Pin 1).

CHART 1 (Sheet 3 of 3)
TROUBLESHOOTING - AFT PASSENGER INTERCOM SYSTEM

Trouble	Cause	Remedy
Intercom is not loud enough, or you lose parts of your sentences.	Microphone may not be close enough to your lips.	Place microphone 1/8 inch from your lips or the side of your mouth.
	Squelch may be adjusted too high.	While no one is talking, slowly rotate the Squelch knob counterclockwise until you hear background noise, and then rotate the knob clockwise until the noise just disappears.
	Headset may be putting out weak signal.	Adjust the microphone pre-amp located on the headset microphone or in the earcup to equalize the sensitivity of headset.
Not able to squelch out all background noise.	If using a David Clark H10-30 headset, the microphone pre-amp may not be turned down.	Rotate small nylon screw at front of the microphone completely counterclockwise to decrease the microphone pre-amp output.
	If using single earsets or other tubular microphone-type headsets, they may not have the noise canceling capability necessary for proper squelch control.	Select a headset with a full earmuff and a high quality noise canceling electret microphone or amplified microphone.
Background noise level is too high while speaking.	Microphone you are using may have poor noise-canceling capability (only one poor microphone will make the entire system noisy).	Switch to a premium headset with a small electret element, such as the Flightcom 5DX. As an alternative, turn down the volume control on headset until the noise is at an acceptable level, and speak louder with the microphone touching your lips to increase the signal-to-noise ratio by increasing the strength of your voice and decreasing the apparent strength of the noise.
Intercom still does not work.	The problem may be within the radios and/or headsets.	Check the radio and headsets in another aircraft.

C. Flightcom Model 403 Intercom

The Flightcom intercom is mounted in the instrument panel above and slightly to the right of the copilot's control wheel. See "Figure 3".

(1) Removal

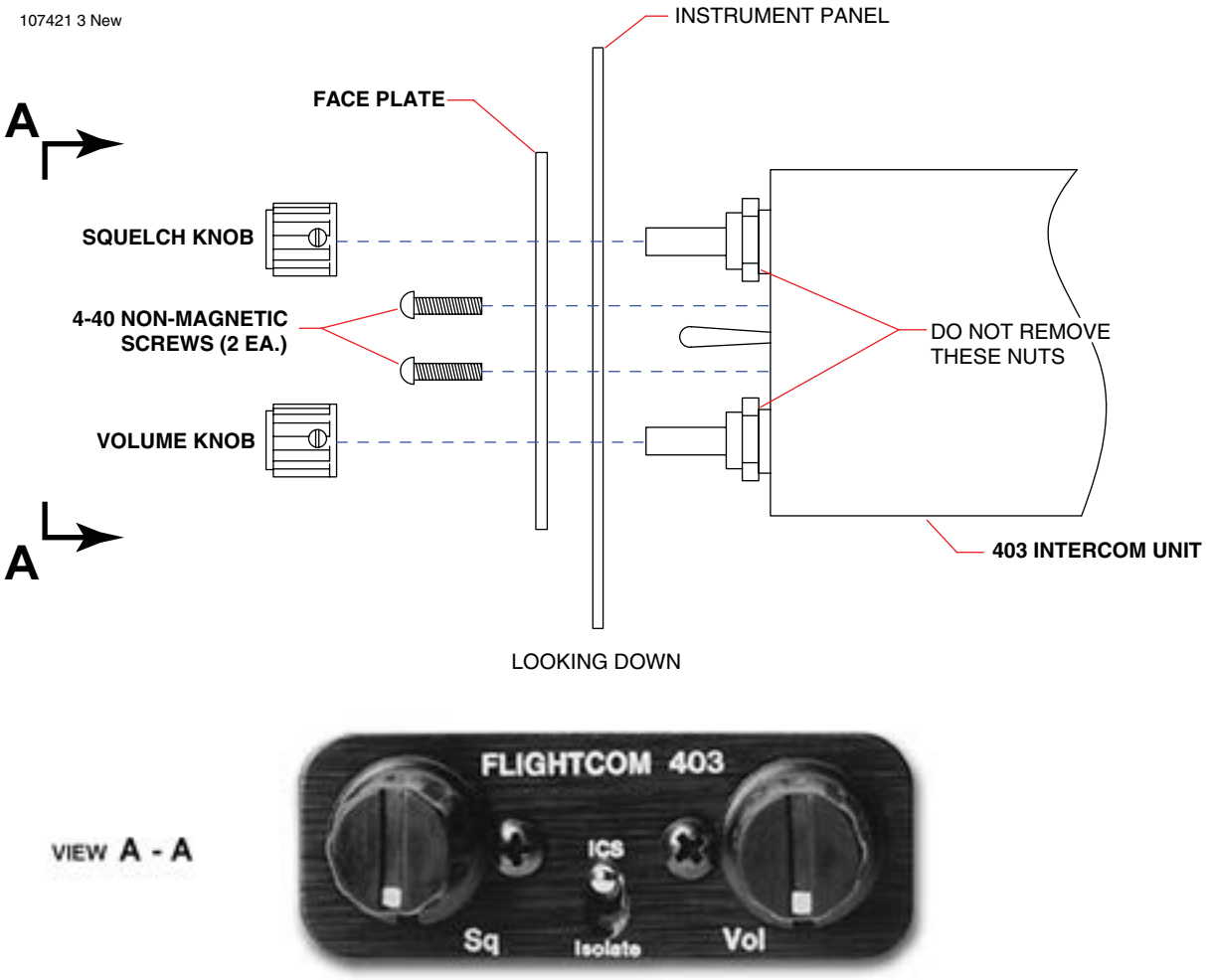
(a) Remove electrical power from the airplane.

(b) Disconnect the 25-pin D-sub connector from the rear of the intercom.

(c) Remove the squelch and volume controls knobs by loosening the set screws and pulling them off of the control shafts.

(d) Reaching behind the instrument panel, support the intercom unit. Remove the two phillips head screws and the intercom face plate.

(e) Remove the intercom unit.



Effectivity
2843711 and up

Flightcom Model 403 Intercom
Figure 3

(2) Installation

(a) Verify electrical power is removed from the airplane.

(b) Reach behind the instrument panel and place the intercom in position with the squelch and volume control shafts and the isolate switch protruding through the appropriate holes in the instrument panel.

(c) Continue to support the unit while installing the intercom face plate and securing with two phillips head screws.

(d) Install the squelch and volume controls knobs as follows:

1) Rotate both the squelch and volume control shafts counter-clockwise until they stop;

2) Position each control knob so that it points to seven o'clock and push onto the control shaft;

3) Secure each control knob with its set screws.

(e) Connect the 25-pin D-sub connector to the rear of the intercom unit.

(f) Conduct an operational test of the intercom system.

D. Push-to-Transmit (PTT) Switch and Phone/Mike Jacks

See "Figure 4" on page 235010.

The Push-to-Transmit (PTT) Switch and Phone/Mike Jacks are mounted in the interphone cover plate assembly in the left and right lower interior side panels adjacent to the aft seats.

(1) Interphone Cover Plate Assembly

The cover plate assembly is secured in the interior panel by friction at the bottom and spring clips at the top.

(a) Removal

1) Remove electrical power from the airplane.

2) Grasping the top of the cover plate assembly with your fingertips, press down and pull out to free the spring clips.

3) If screws are present apply light pressure to the cover plate and use a screwdriver and the four screws.

4) Lift the cover plate assembly out of the interior panel sufficient to disconnect the harness connectors.

5) Remove the cover plate assembly.

(b) Installation

1) Place the cover plate assembly adjacent to the opening in the interior panel and connect the two harness connectors.

2) Install the cover plate assembly in the interior panel bottom first and secure it by pressing the spring clips home at the top.

3) If screws are present install the cover plate assembly in the interior panel and secure it with four screws.

4) Conduct an operational test of the intercom system.

(2) Push-to-Transmit (PTT) Switch

The PTT switch is press fit into the Interphone Cover Plate Assembly. Its wiring is soldered at the switch and to Pins 1 and 2 of the adjacent Mike Jack.

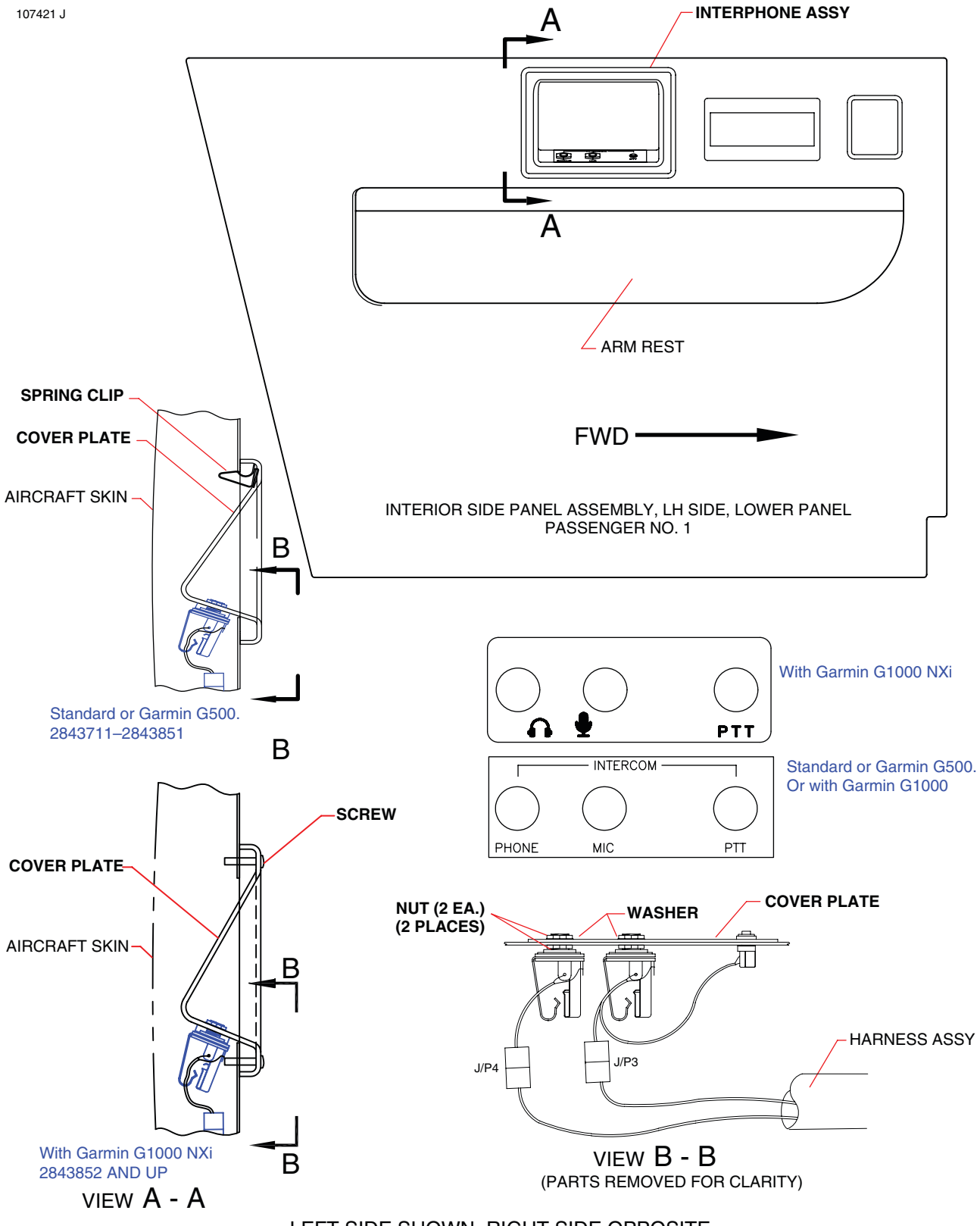
(a) Removal

1) Remove the Interphone Cover Plate Assembly per Removal, above.

2) Cut the wires leading from the PTT switch to the Mike Jack.

3) Press the PTT switch up and out of its hole in the cover plate assembly.

107421 J



LEFT SIDE SHOWN, RIGHT SIDE OPPOSITE

Effectivity
2843711 and up

Push-to-Transmit Switch and Phone/Mike Jacks
Figure 4

(b) Installation

NOTE: If installing a new PTT switch, solder wire pigtails onto Pins 1 and 2 of the new switch. Use 22 AWG wire unless otherwise specified in 91-23-50 Figure 4, Sheet 3.

1) Feed wire pigtails through hole in cover plate assembly and press switch into hole.
2) Splice wire pigtails to wires from the Mike Jack, Pin 1 to Pin 1 and Pin 2 to Pin 2.
3) Reinstall Interphone Cover Plate Assembly per Installation, above.

(3) Mike and Phone Jacks

The Mike and Phone jacks are mounted in the Interphone Cover Plate Assembly and secured by an external jamnut.

(a) Removal

1) Remove the Interphone Cover Plate Assembly per Removal, above.
2) Remove the upper jamnut and remove the jack.

NOTE: Note position of brown plastic washer and fiber washer for reinstallation.

NOTE: If a mike jack, cut the wires leading from the PTT switch to the jack.

(b) Installation

NOTE: If installing a new jack, solder wire pigtails from the old harness connector as shown in 91-23-50 Figure 4, Sheet 3 for the appropriate jack. If a mike jack, solder additional wire pigtails onto Pins 1 and 2 of the new jack to accomodate the PTT switch wiring. Use 22 AWG wire unless otherwise specified in 91-23-50 Figure 4, Sheet 3.

1) Postion the jack in the appropriate hole as shown in “Figure 4”.
2) Ensure brown plastic and fiber washers are positioned as noted during removal.
3) Secure the jack with the upper jamnut.
4) If a mike jack, splice the wire pigtails to wires to the PTT switch, Pin 1 to Pin 1 and Pin 2 to Pin 2.
5) Reinstall Interphone Cover Plate Assembly per Installation, above.

E. Post Installation Configuration and Functional Test

(PIR-107823 New.)

Use the following procedures to setup, adjust volume and squelch control, checkout radio transmission and checkout passenger isolate. See “Figure 3” to reference Flightcom Control Panel configuration.

(1) Adjust the Intercom and Headset Configuration and Volume.

(a) Plug headsets into the copilot and passenger jacks in the aircraft.

(b) Turn on the aircraft BATT MASTR switch to turn on the intercom.

(c) Set the intercom Volume control knob to the 11 o’clock position.

(d) Set the intercom Squelch control knob to the 3 o’clock position.

NOTE: Using stereo headphones without a mono setting will cause only one earphone to be active.

(e) Turn each headset volume to ½ the available volume control.

1) Position the headset boom microphone 1/8” from your lips to the side of your mouth.

2) While speaking loudly, adjust the volume controls on the Garmin audio panel to set the copilot volume to a comfortable level. Set the Flightcom 403 switch to “ICS,” the Flightcom volume should then be adjusted to set passenger ICS volume to personal preference.

(2) Adjust the squelch control:

(a) While no one is talking, turn the intercom Squelch control knob as far clockwise as possible while still blocking background noise.

(b) Re-adjust the setting in flight to compensate for different noise levels.

NOTE: If you set the squelch too high by turning the Squelch control knob counterclockwise, voice will be cut out unless you talk very loudly; if you set the squelch too low by turning the Squelch control knob clockwise, the background noise will be heard occasionally. The intercom will not interfere with the normal use of the radio and will allow passengers to hear the aircraft radio and sidetone.

(3) Checkout Radio Transmission

(a) Transmit on the copilot’s position and each passenger position using the following checkout procedure:

(b) Verify the volume and squelch control setup was performed.

(c) Plug a headset into the copilot’s phone and microphone jack and speak into the microphone without pressing the push-to-talk (PTT) button and get a response from someone connected first to the pilot’s phone and microphone jack and then to each of the passenger phone and microphone jacks. This will confirm that the copilot’s and each passenger intercom and sidetone are performing clearly and correctly.

(d) From the copilot’s position use the yoke PTT to transmit to a ground station or another aircraft to verify a clear copilot’s transmission.

(e) From each passenger position use that passenger’s PTT to transmit to a ground station or another aircraft to verify a clear passenger transmission.

(f) From one passenger position use that passenger’s PTT to transmit to a ground station or another aircraft a long count and override with a transmission from the copilot position to verify the passenger’s transmission can be overridden by the copilot.

NOTE: Only the person whose PTT switch is depressed will be heard over the radio. No other intercom conversations will be transmitted over the radio at that time.

(4) Checkout Copilot Isolate Switch

Perform the following Isolation switch checkout procedure:

Move the Flightcom 403 switch from “ICS” to “Isolate” and verify using the passenger’s PTT that each passenger position cannot transmit. The passengers should still have sidetone and intercom between each other and the copilot.

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STATIC DISCHARGING

1. Static Wicks

   A. Description

      This airplane is equipped with nine (9) static wicks: one (1) each on the inboard and outboard ends of each aileron; one (1) on each outboard end of the stabilator; one (1) on the left-hand side of the rudder at the bottom; and one (1) in the center of each flap.

   B. Inspection

      Each 100 hours or annually, whichever comes first, inspect static wicks as follows:

      (1) General appearance and physical condition.
      (2) Security of attachment to airframe.
      (3) Discharge points visible.
      (4) Resistance = 0.5 ±20% megohms (500 to 1,000 volt megohmmeter.
      (5) Base resistance to airframe (one (1) ohm maximum).

2. Bonding Straps

   NOTE: Perform Electrical Bonding - On Condition Inspection, 51-80-00, whenever a bonding strap is installed/reinstalled.

   To aid in dissipating static electricity buildup the ailerons, stabilator, stabilator tab, and rudder are bonded to either the control's hinge or spar.

   When replacing the jumper assemblies (bonding straps), secure the end of the jumper that mounts to the control's hinge or spar as follows:

   A. Clean an area of 1 1/2 times the diameter of the jumper's washer down to bare metal.

   B. Attach the jumper and washer to the control's hinge or spar.

   C. Seal the cleaned area with waterborne, chromated, fluid resistant, epoxy primer (i.e. - PRC Desoto) and acrylic lacquer.

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CHAPTER

24

ELECTRICAL POWER

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CHAPTER 24

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GENERAL

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

This chapter contains instructions for correcting difficulties which may arise in electrical system operation. Included is a general description and function of each part of the system along with test and adjustments of various components.

1. Description and Operation

The electrical system is a 28-volt, direct current, single wire, negative ground system. All electrical equipment is grounded to the metal structure of airplane. The structure takes the place of a second wire. A 24-volt battery is incorporated into the system to furnish power for starting and as a reserve power source in case of alternator failure. The master battery and 70 (or 75) amp alternator are connected to the bus bar that powers all electrical equipment, except the starter, which receives its power direct from battery. The battery relay and alternator field circuit are controlled by independent rocker type master (BATT MASTR) and alternator (ALTR) switches located in the overhead switch panel. The alternator switch (ALTR) may be used to turn OFF the alternator field in the event of alternator failure.

The annunciator or crew alerting system (depending on how the airplane is equipped) will alert the pilot if the alternator fails to produce current, accompanied by a zero indication on the ammeter. A low voltage alert will occur when alternator output is lost and the system is drawing power from the battery alone.

The master (BATT MASTR) switch must be ON before any electrical equipment will operate. The airplane is equipped with standard navigation lights and wing tip strobe type anti-collision lights, which are controlled by rocker type switches (NAV LIGHT and STROB LIGHT) located in the overhead switch panel. The airplane is also equipped with two landing light; one located in each wing. The right wing landing light may also be used as a taxi light. These lights are controlled by a rocker type switch (OFF-LDG LIGHT) located in the overhead switch panel.

In S/N's 2843622 thru 2843682, with Avidyne Entegra IFDS installed, a Standby Alternator is also installed as an emergency backup to the primary alternator system.

In airplanes equipped with Garmin G1000 an emergency battery is installed in addition to the main battery. The emergency battery has a rated capacity of 6.0 ampere hours and is mounted just forward of the main battery.

2. Troubleshooting

WARNING: ALL CHECKS AND ADJUSTMENTS OF ALTERNATOR AND COMPONENTS MUST BE MADE WITH ENGINE STOPPED. TO COMPLETE SOME CHECKS OR ADJUSTMENTS, REMOVE THESE UNITS FROM AIRPLANE AND PLACE ON A TEST STAND.

Troubles peculiar to the alternator system are listed in 24-30-00, "Chart 1", with probable causes and suggested remedies. Troubles peculiar to the battery system are listed in 24-30-00, "Chart 7", with probable causes and suggested remedies.

The wiring diagrams included in Chapter 91 give a physical breakdown of the different electrical circuits used in the airplane.

After trouble is corrected, check entire electrical system for security and operation of its components.

3. Contactor Maintenance

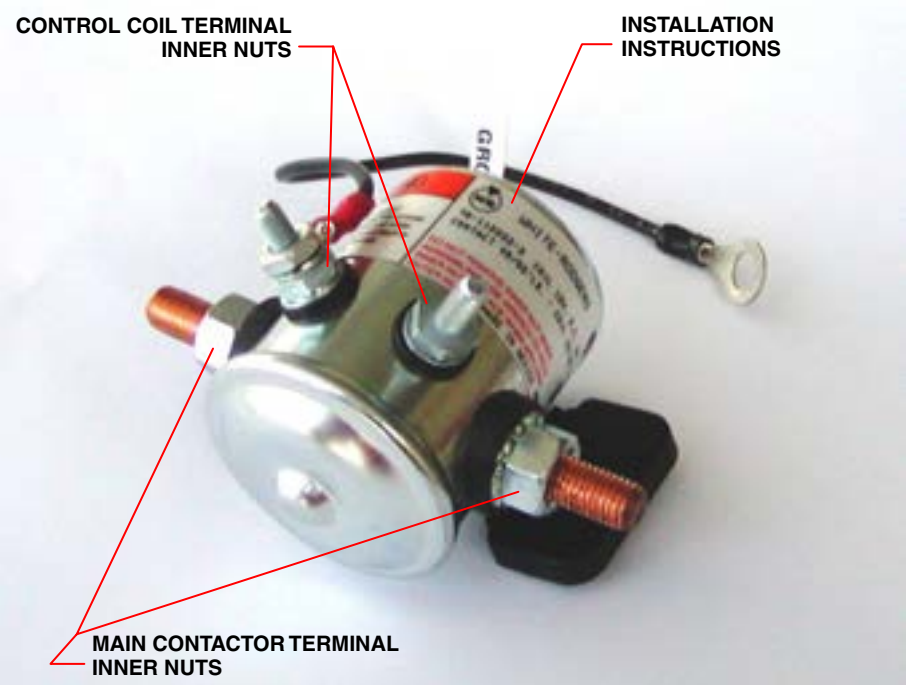
See “Figure 1”.

WARNING: A STARTER CONTACTOR THAT IS DAMAGED AS DESCRIBED BELOW CAN FAIL UNEXPECTEDLY IN THE CLOSED POSITION, CAUSING AN UNCOMMANDED ROTATION OF THE PROPELLER THE NEXT TIME THE MAIN POWER BUS IS ENERGIZED.

Contactor terminals may be damaged if correct procedure is not followed when loosening or tightening connections. Specifically, wrenching torque may be transmitted to components inside the contactor, causing damage that may go undetected.

Service spares replacement contactors are configured out-of-the-box as shown in “Figure 1”. The contactor is labeled with a note in red text that reads: “DO NOT REMOVE INNER NUT WHEN ASSEMBLING. USE WRENCH TO HOLD INNER NUT IN PLACE WHEN APPLYING OUTER NUT.”

The “INNER NUT” referred to in this note is the nut(s) that are supplied with the contactor. When installing wire terminals onto the threaded posts, two wrenches must be used: One wrench (the back-up wrench) is placed on the inner nut to hold it stationary, while the other wrench turns and tightens the outer nut.



Contactor (Typical)
Figure 1

The proper installation torque for nuts installed on the threaded terminal posts is labeled on the contactor. If the label is missing or illegible, torque as follows:

White-Rodgers:

- On the main contactor terminal nuts (5/16 – 24 thread), use 45 to 55 inch-pounds (in-lb).
- On the control coil terminals nuts (#10 – 32 thread), use 12 to 18 in-lb.

Lamar:

- On the main contactor terminal nuts (5/16 – 24 thread), use 35 to 45 in-lb.
- On the control coil terminals nuts (#10 – 32 thread), use 10 to 15 in-lb.

NOTE: Fastener torques shown above reflect the specifications of the contactor manufacturer at the time of publication of this maintenance alert service letter, and are subject to change. Adhere to vendor installation instructions provided with any Piper approved replacement contactor.

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D.C. GENERATION

1. General

Electrical power is generated by a 70 or 75 amp alternator. Three different alternators may be installed. The Electrosystems 70 Amp ES 4032 and the Hartzell Engine Technologies (HET) 75 Amp ALU-8532LS are similar and are addressed together. The Plane-Power 70 Amp AL24-P70-1 is addressed separately.

2. Precautions

The following precautions must be observed when testing or servicing the electrical system. Failure to observe these precautions will result in serious damage to the electrical equipment.

CAUTION: REFER TO WIRING DIAGRAM ("FIGURE 4" ON PAGE 243014) WHEN INSTALLING OR TESTING ELECTROSYSTEMS OR HARTZELL ALTERNATOR.

CAUTION: DISCONNECT BATTERY BEFORE CONNECTING OR DISCONNECTING TEST INSTRUMENTS, EXCEPT VOLTMETER, OR BEFORE REMOVING OR REPLACING ANY UNIT OR WIRING. ACCIDENTAL GROUNDING OR SHORTING AT REGULATOR, ALTERNATOR, AMMETER, OR ACCESSORIES, WILL CAUSE SEVERE DAMAGE TO UNITS AND/OR WIRING.

CAUTION: THE OUTPUT LEAD MUST NOT BE REMOVED FROM ALTERNATOR WHILE ROTOR WINDING IS ENERGIZED AND ALTERNATOR IS OPERATING.

CAUTION: DO NOT ATTEMPT TO POLARIZE ALTERNATOR. NO POLARIZATION IS REQUIRED. ANY ATTEMPT TO DO SO COULD DAMAGE ALTERNATOR, REGULATOR, OR CIRCUITS.

CAUTION: GROUNDING ALTERNATOR OUTPUT TERMINAL WILL DAMAGE ALTERNATOR AND/OR CIRCUIT AND COMPONENTS.

CAUTION: REVERSED BATTERY CONNECTIONS WILL DAMAGE RECTIFIERS, WIRING, AND OTHER CHARGING SYSTEM COMPONENTS. BATTERY POLARITY MUST BE CHECKED WITH A VOLTMETER BEFORE CONNECTING BATTERY. THIS AIRCRAFT IS NEGATIVE GROUND.

CAUTION: IF BOOSTER BATTERY OR FAST CHARGER IS USED, ITS POLARITY MUST BE CONNECTED CORRECTLY TO PREVENT DAMAGE TO ELECTRICAL SYSTEM COMPONENTS.

3. Troubleshooting

WARNING: ALL CHECKS AND ADJUSTMENTS OF ALTERNATOR AND COMPONENTS MUST BE MADE WITH ENGINE STOPPED. TO COMPLETE SOME CHECKS OR ADJUSTMENTS, REMOVE THESE UNITS FROM AIRPLANE AND PLACE ON A TEST STAND.

Troubles peculiar to the alternator system are listed in 24-30-00, "Chart 1", with probable causes and suggested remedies.

The wiring diagrams included in Chapter 91 give a physical breakdown of the different electrical circuits used in the airplane.

After trouble is corrected, check entire electrical system for security and operation of its components.

CHART 1 (Sheet 1 of 4)
TROUBLESHOOTING ALTERNATOR SYSTEM

Trouble	Cause	Remedy
Zero output indicated on ammeter regardless of rpm. (See alternator system test procedure.)	Open field circuit.	With the battery switch turned on, check for battery voltage from the airplane's main bus through the entire field circuit to the alternator field terminal. Measure the voltage from the ground (-) to the following points (+) in sequence: bus bar, field circuit breaker (5A), field terminals of master switch voltage regulator and alternator field terminal. Interruption of voltage through any of these points isolates the faulty component or wire which must be replaced. (See schematic, 91-24-30.)
	Open output circuit.	With the battery switch turned on, check for battery voltage from the airplane's main bus through the entire output circuit to the alternator battery post. Measure voltage from ground (-) to the following points (+) in sequence: bus bar, output current limiter, ammeter, and alternator battery post. Interruption of voltage through any of these points isolates the faulty component or wire which must be replaced. (See schematic, 91-24-30.) Open circuit in alternator output will usually burn out ALT annunciator and the 50 ohm resistor. Check 5.0 amp in line fuse.

CHART 1 (Sheet 2 of 4)
TROUBLESHOOTING ALTERNATOR SYSTEM

Trouble	Cause	Remedy
Zero output indicated on ammeter regardless of rpm. (See alternator system test procedure.) (continued)	Open field winding in alternator.	Disconnect field terminal of alternator from field wiring and check for continuity from field terminal to ground with ohmmeter (20-100 ohms) depending on brush contact resistance.
		WARNING: TURN MAGNETO SWITCH TO OFF BEFORE TURNING PROP.
		Pull propeller slowly by hand turning alternator rotor through 360 of travel.
		If resistance is high, check brushes for spring tension and excessive wear and replace if necessary. If brushes are good and field reads open, replace alternator.
Output indicated on ammeter does not meet minimum values specified in alternator system test procedure.	Faulty voltage regulator.	Start engine, turn on load (see alternator test procedure), set throttle at 2300 rpm. Check voltage at buss bar (remove cigar lighter/power point and check from center contact (+) to ground (-)). Voltage should be 27.5 volts minimum. If voltage is below this value, replace regulator.
	High resistance connections in field or output circuit.	Check visually for loose binding posts at the various junction points in system, alternator battery post, lugs on ammeter, connections at voltage regulator, circuit breaker, etc. (See schematic, 91-24-30.) Examine crimped terminal ends for signs of deterioration at crimp or strands of broken wire at crimp. Tighten any loose binding posts or replace bad wire terminals.

CHART 1 (Sheet 3 of 4)
TROUBLESHOOTING ALTERNATOR SYSTEM

Trouble	Cause	Remedy
Output indicated on ammeter does not meet minimum values specified in alternator system test procedure. (continued)	Open rectifier.	If any of the six rectifiers pressed into the rear bell housing of the alternator open up internally, it will result in a definite limitation on the current that can be drawn from the alternator. After having checked the previous causes of low output it can be assumed that a faulty rectifier exists. See appropriate vendor publication as specified in Introduction, Supplementary Publications, Vendor Publications, Alternator.
Field circuit breaker trips.	Short circuit in field circuit.	Disconnect field wiring at terminal of alternator. Turn on master switch. If breaker continues to trip, disconnect each leg of field circuit, working from alternator towards the circuit breaker until breaker can be reset and will hold. Replace component or wire which was isolated as defective. (See schematic, 91-24-30.)
	Short circuit in field winding of alternator.	Disconnect field wiring at terminal of alternator. Turn on master switch. Set breaker, and if breaker fails to trip, this isolates short circuit to field of alternator itself. Check brush holders for shorting against frame. If there are no obvious signs of a physical short circuit at field terminal or brush holder, replace alternator.
	NOTE: Intermittent short circuit. Internal short circuit of the field can occur at various positions of the rotor, therefore, reconnect field, reset breaker.	

CHART 1 (Sheet 4 of 4)
TROUBLESHOOTING ALTERNATOR SYSTEM

Trouble	Cause	Remedy
WARNING: TURN MAGNETO SWITCH TO OFF BEFORE TURNING PROP.		
Field circuit breaker trips. (continued)	Short circuit in field winding of alternator.	Pull propeller slowly by hand turning alternator rotor through 360° of travel. Observe circuit breaker for signs of tripping.
Ammeter indicates 70 amps at 1400 rpm and above, ALT annunciator light ON.	Short to ground in alternator output wiring.	Check output wiring for short to ground. Check other wiring for chafing, etc.
Excessive ammeter fluctuation.	Excessive resistance in field circuit.	Check all connections and wire terminals in field circuit for deterioration such as loose binding posts, broken wire strands at terminals, etc. Tighten all connections and replace faulty terminals.
	High field circuit resistance.	If problem persists, jump across the terminals of the following components one at a time until the faulty unit is isolated. a. Field 5-amp (alternator circuit protector). b. Alternator switch. c. Overvoltage relay.
	Defective voltage regulator.	Replace voltage regulator.
	Faulty grounds, from grounding points.	Completely clean all corrosion. Ensure solid ground.

4. Checking Alternator System

With electrical equipment off (except master switch) the ammeter will read amount of charging current demanded by battery. The amount will vary, depending on percentage of battery charge. As battery is charged, current displayed on ammeter will reduce to approximately two amperes. Current reading on ammeter will show if alternator system is operating normally.

NOTE: Current reading on ammeter is the load in amperes demanded by the electrical system from alternator. Check, for example, a condition where battery is demanding 10 amperes charging current, then switch on landing light. Note value in amperes placarded on circuit breaker panel for landing light circuit breaker (10 amps) and multiply by 80 percent. You will arrive at a current of 8 amperes. This is approximate current drawn by the light. When light is switched on, there will be an increase of current from 10 to 18 amperes on ammeter. As each unit of electrical equipment is switched on, current will add up and the total, including battery, will read on ammeter.

For example, the airplane's maximum continuous load with all equipment on is approximately 60 amperes for 70 ampere alternator. This approximate 60 ampere value, plus approximately two amperes for a fully charged battery, will appear continuously under these flight conditions. If the ammeter reading were to go much below this value, under the aforementioned conditions, trouble with the alternator system would be indicated and corrective action should be taken by switching off the least essential equipment. Locate faulty components as follows:

CAUTION: ON AIR-CONDITIONED AIRCRAFT, FULL ALTERNATOR OUTPUT ON GROUND MUST BE LIMITED TO NOT MORE THAN 10 MINUTES. REFER TO PILOT'S OPERATING HANDBOOK.

A. Check airplane is positioned so prop blast will not interfere with other operations going on nearby. Start engine and set throttle for 1000 to 1200 rpm.

B. Switch on the following loads and observe ammeter output increase as follows:

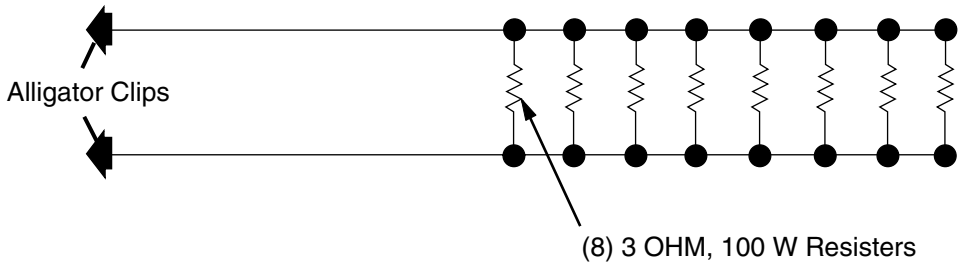
(1) Rotating beacon - 3 to 6 amps.

(2) Navigation and instrument lights - 4 to 6 amps.

(3) Landing lights - 7 to 9 amps each.

If the alternator does not meet the above readings refer to "Chart 1". Follow troubleshooting procedure outlined on the chart. Check each cause and isolation procedure under a given trouble.

If desired, the load required by the test can be simulated by connecting 8, 3-ohm, 100-watt resistors wired in parallel from the main bus (+) to airframe ground (-). See "Figure 1".



Eight Resistor-Bank Load
Figure 1

5. Service Procedures/Service Tips

The alternator and regulator are designed for use on only one polarity system. Observe the following when testing or servicing the electrical system: (Failure to observe cautions will result in serious damage to electrical equipment.)

CAUTION: DISCONNECT BATTERY BEFORE CONNECTING OR DISCONNECTING TEST INSTRUMENTS (EXCEPT VOLTMETER) OR BEFORE REMOVING OR REPLACING ANY UNIT OR WIRING. ACCIDENTAL GROUNDING OR SHORTING AT REGULATOR, ALTERNATOR, AMMETER, OR ACCESSORIES, WILL CAUSE SEVERE DAMAGE TO UNITS AND/OR WIRING.

CAUTION: ALTERNATOR MUST NOT BE OPERATED ON OPEN CIRCUIT WITH ROTOR WINDING ENERGIZED.

CAUTION: DO NOT ATTEMPT TO POLARIZE ALTERNATOR. NO POLARIZATION IS REQUIRED. POLARIZATION WILL RESULT IN DAMAGE TO ALTERNATOR, REGULATOR, OR CIRCUITS.

CAUTION: GROUNDING ALTERNATOR OUTPUT TERMINAL WILL DAMAGE ALTERNATOR AND/OR CIRCUIT AND COMPONENTS.

CAUTION: REVERSED BATTERY CONNECTIONS WILL DAMAGE RECTIFIERS, WIRING, OR OTHER CHARGING SYSTEM COMPONENTS. BATTERY POLARITY MUST BE CHECKED WITH VOLTMETER BEFORE CONNECTING BATTERY. MOST AIRCRAFT ARE NEGATIVE GROUND.

CAUTION: IF A BOOSTER BATTERY OR FAST CHARGER IS USED, POLARITY MUST BE CONNECTED CORRECTLY TO PREVENT DAMAGE TO ELECTRICAL SYSTEM COMPONENTS.

6. Alternator Belt Tension Adjustment

CAUTION: AN IMPROPERLY TENSIONED BELT WILL WEAR RAPIDLY, SLIP, AND REDUCE ALTERNATOR OUTPUT. BELT MUST BE CHECKED FOR PROPER TENSION AT INSTALLATION, AFTER FIRST 25 HOURS OPERATION, AND EACH 100 HOURS THEREAFTER.

NOTE: For airplanes with air conditioning installed, See 21-50-00, for replacement and adjustment of compressor drive belt. See also latest revision of Lycoming Service Instruction No. 1129.

There are three acceptable methods of checking alternator belt tension. The first method described is preferred by most maintenance personnel. It is technically simple and requires less time.

A. Torque Method

This method of checking belt tension consists of measuring torque required to slip the belt at the small pulley as follows:

(1) Apply a torque indicating wrench to the nut attaching pulley to alternator and turn in a clockwise direction. Observe torque shown on wrench at the instant the pulley slips.

(2) Adjust belt to proper tension as specified in “Chart 2” on page 24308.

B. Deflection Method

Check belt tension by measuring the amount of deflection caused by a predetermined amount of tension as follows:

(1) Attach the hook of a small spring-scale to the belt at the mid-point between rear gear support and alternator.

(2) Pull scale until it reads 14 pounds. (10 pounds for used belts.)

(3) Measure the distance the belt moved with the 10 or 14 pound load. The distance (deflection) must be 5/16 inch. If less than 5/16 inch, belt is too tight. For aircraft equipped with Plane Power alternator, distance (delfection) must be 13/32 inch. If less than 13/32 inch, belt is too tight.

C. Belt Tension Gauge

Use a belt tension gauge such as the Burroughs deluxe model or equivalent.

(1) Attach gauge over the belt per gauge manufacturer's instructions.

(2) Quickly release handle and read tension.

(3) Repeat steps (1) and (2) several times to eliminate the possibility of an inaccurate reading.

NOTE: Slight variations in readings taken at different locations on the belt are normal.

(4) If a new belt is installed, set tension 25 percent above operating range to allow for stretch that will occur as soon as belt is operated.

D. Complete alternator belt tension adjustment, and tighten alternator pivot bolts to a torque of 225 to 255 inch-pounds.

E. See "Chart 3" on page 24309 to troubleshoot alternator belt issues.

CHART 2
ALTERNATOR BELT TENSION

Width of Belt	Condition	Torque (Ft.-Lbs.)*
3/8 inch	New	11 to 13
	Used	7 to 9
13/32 inch	New	8 to 9
	Used	7 to 9
1/2 inch	New	13 to 15
	Used	9 to 11
		* indicated at alternator pulley

NOTE: Higher tension specified for a new belt will compensate for the initial stretch at the first operation. Higher tension values must not be applied to previously used belts.

CHART 3 (Sheet 1 of 2)
TROUBLESHOOTING ALTERNATOR BELTS

Trouble	Cause	Remedy
Belt slippage.	Lack of tension.	Increase tension.
	Overloaded drive.	Reduce load or check pulley size.
	Worn pulley or V grooves.	Replace pulley.
	Oily drive conditions	Clean drive; resolve oil leak.
Belt turns over.	Misaligned drive.	Realign shafts and pulleys
	Worn pulley or V grooves.	Replace pulley
	Heavy impluse loads.	Check idler pulley (spring or not) Check tension
	Excessive vibration.	Tension v-belt. If tension is good, inspect belt condition or replace.
	Broken cords caused by prying belt over pulley.	Replace belts. Do not pry belt over pulley.
	Wrong pulley or belt.	
Rapid belt wear.	Worn pulley grooves.	Replace pulley.
	Pulleys misaligned.	Align the pulleys & brackets.
	Mismatched belts. (Multi belt installations only.)	Replace with matched belts.
	Belt slippage.	Increase tension.
	Pulley diameter too small.	Check P/N and/or replace pulley.
Belt separated.	Foreign materials in drive belt.	Check that pulleys are protected.
	Belt slippage. (over much time.)	Increase tension.
	Heavy start up loads.	Reduce loads before starting.
	Belts damaged during installation.	Install new belts properly.
Belt Stretch.	Excessive drive tension.	Use proper tension.
	Broken cords using multiple belts.	Replace belts with matched set.
	Large misalignment.	Realign brackets and pulleys.
Belt squeal.	Belt slippage.	Increase tension.
	Insufficient arc of contact.	Increase center distance.
	Over loaded drive.	Check pulley size.

CHART 3 (Sheet 2 of 2)
TROUBLESHOOTING ALTERNATOR BELTS

Trouble	Cause	Remedy
Belt chirp.	Misaligned drive.	Align the pulleys & brackets.
	Belt tilted in pulley groove.	Correct angle of belt.
	Belt riding on side of pulley groove.	Adjust bracket in or out to seat belt.
Belt bottom cracks.	High surrounding temperature.	Provide ventilation.
	Pulley diameter too small.	Redesign drive.
	Belt slippage.	Increase tension.
Overheated bearings.	Continous belt slippage.	Increase tension.
	Excessive drive tension.	Tension drive properly.
	Bearings not lubed or damaged.	Replace bearing.

7. Hartzell Engine Technologies (HET) Alternators (formerly Kelly Aerospace, Electrosystems)

These alternators are similar and serviced the same.

A. Electrosystems 28 vdc, 70 amp Alternator ES 4032

Installed in S/N's 2843001 thru 2843725.

NOTE: In S/N's 2843622 thru 2843682, with Avidyne Entegra installed, see also Standby Alternator, below.

B. HET 28 vdc, 75 amp Alternator ALU-8532LS

Installed in S/N's 2843726–2843820, 2843822; and in S/N's 2843823 and up with the O-360-A4M engine and/or Garmin G1000.

C. Inspections

(PIR_Hartzell OE-A2, Rev. 4.)

Times given below are alternator times-in-service.

(1) First Ten (10) Hours

Perform an initial check of the alternator assembly.

(a) Inspect through bolt security and proper safety wire application. Re-torque if bolts are found loose.
(b) Inspect pulley for security.
(c) Inspect for signs of overheating or electrical arcing.
(d) Inspect the cooling duct connection and associated air inlet for security and cleanliness.
(e) Ensure the alternator assembly is clear of interference with any engine or airframe structure.
(f) Inspect alternator to engine mounting bolts for condition and security.
(g) Inspect drive belt tension per "Alternator Belt Tension Adjustment" on page 24307.
(h) Inspect drive belt for unusual wear.
(i) Inspect all terminal hardware for tightness and insulators for condition.

(2) First Fifty (50) Hours

Perform a check of the alternator assembly.

(a) Inspect through bolt security and proper safety wire application. Re-torque if bolts are found loose.
(b) Inspect alternator to engine mounting bolts for condition and security.
(c) Inspect drive belt tension per "Alternator Belt Tension Adjustment" on page 24307.
(d) Inspect drive belt for unusual wear.
(e) Inspect pulley for security.
(f) Inspect for signs of overheating or electrical arcing.
(g) Inspect all terminal hardware for tightness and insulators for condition.

(3) Each 100 Hours or Annually, whichever comes first

Perform a check of the alternator assembly.

(a) Inspect through bolt security and proper safety wire application. Re-torque if bolts are found loose.
(b) Inspect alternator to engine mounting bolts for condition and security.
(c) Inspect drive belt tension per "Alternator Belt Tension Adjustment" on page 24307.
(d) Inspect drive belt for unusual wear.
(e) Inspect pulley for security.

(f) Inspect area around the brush holder for soot. If a large amount of soot appears, remove each brush and check for wear or damaged brushes. If severe wear has occurred, check slip rings for gouges or scratches.

(g) Inspect for oil leaks at cooling air outlet spout.

(h) Inspect all terminal hardware for tightness and insulators for condition.

(4) Each 500 Hours or each Two (2) Years, whichever comes first

(a) Remove each brush and inspect for wear or damaged brushes. If brush shows more than 50% wear, has chips or damage, or broken brush lead strands, replace brushes (brushes must be replaced as a set with the brush holder). If brushes are to be reused, return them in the position and orientation as removed, as a set.

(b) Inspect alternator to engine mounting bolts for condition and security.

(c) Inspect drive belt tension per “Alternator Belt Tension Adjustment” on page 24307.

(d) Inspect drive belt for unusual wear. Replace if required.

(e) Inspect pulley for security. Replace if required.

(f) Inspect insulators for cracks or burns.

(g) Inspect the aluminum housings for condition and for surface corrosion, clean and treat if required.

(h) If conditions are found that require replacement of parts other than the brush holder assembly, drive gear, or hardware; overhaul alternator.

(5) Each 2000 Hours, every Twelve (12) Years, or at any Engine Overhaul, whichever comes first

Recommend overhaul or replacement of alternator assembly. This a maximum time in service, actual overhaul time may vary based on cooling, electrical load, and conditions of use. The engine manufacturer recommended TBO, if less shall supercede this TBO.

D. Components

The principal alternator components are: brush holder assembly, slip ring end head, rectifiers, stator, rotor, and drive end head. (The following paragraph numbering are keyed to “Figure 3”.)

(1) The brush and holder assembly contains two brushes, two brush springs, a brush holder, and insulator. One brush is connected to a terminal stud and insulated from ground. The other brush is connected to ground through brush holder. Brush and holder assembly are easily removed for inspection or brush replacement.

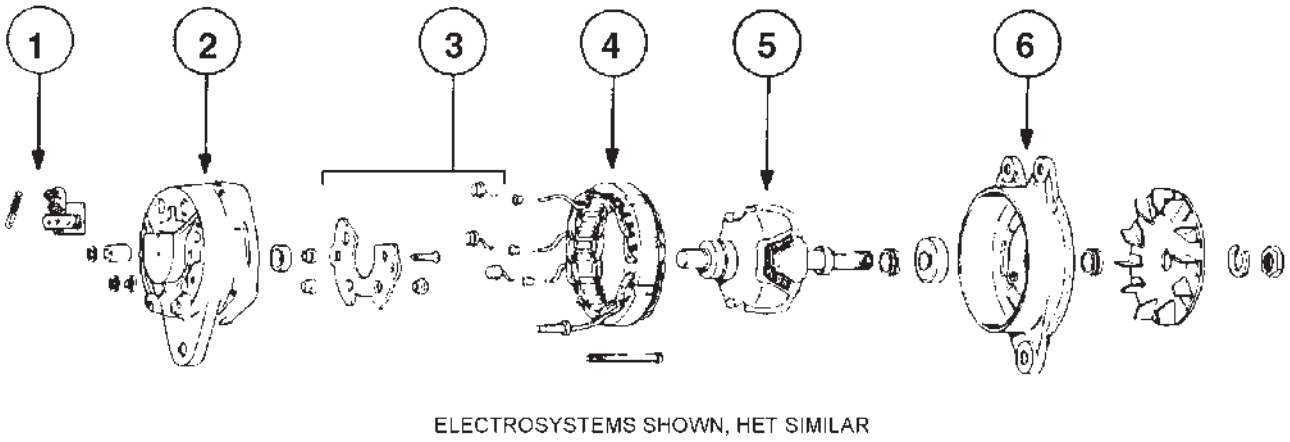
(2) The slip ring end head provides mounting for rectifiers and rectifier mounting plate, output and auxiliary terminal studs, and brush and holder assembly. Slip ring end head contains a roller bearing, outer race assembly, and a grease seal.

(3) The rectifiers used are rated at minimum 150 peak inverse voltage for transient voltage protection. Three positive rectifiers are mounted in rectifier mounting plate while three negative rectifiers are mounted in slip ring end head. Each rectifier pair is connected to stator lead with high temperature solder. Stator leads are anchored to rectifier mounting plate with epoxy cement for vibration protection.

(4) The stator contains a special lead connected to center of the three phase windings and is used to activate low voltage warning systems or relays. The stator is treated with a special epoxy varnish for high temperature resistance.

(5) The rotor contains slip ring end bearing inner race, and spacer on slip ring end of shaft. Rotor winding and winding leads are specially treated with high temperature epoxy cement to provide vibration and temperature resistance characteristics. High temperature solder is used to secure winding leads to slip rings.

(6) The drive end head supports a sealed, pre-lubricated ball bearing in which drive end of rotor shaft rotates.



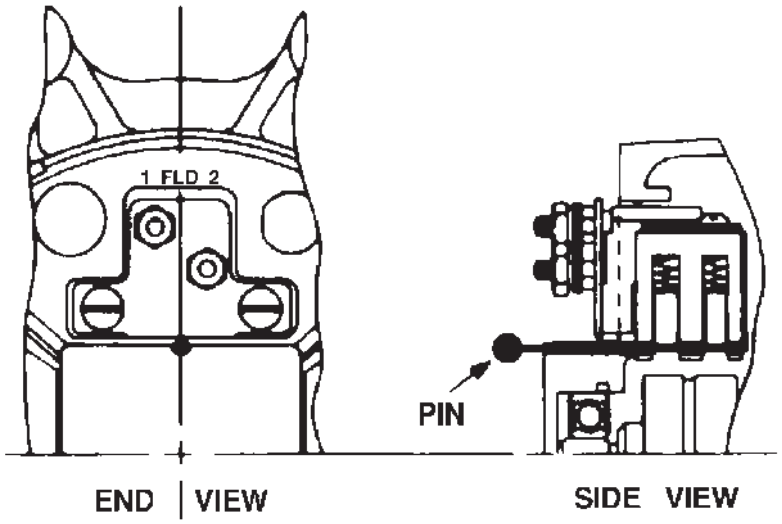
Alternator Exploded View
Figure 2

E. Brushes

These units have a separate brush holder assembly installed after alternator assembly. Brush holder has a small hole that intersects the brush cavities. Use a pin or wire, as in “Figure 2” to hold brushes in holder during assembly. Remove pin after brush holder retaining screws are tightened. Check brushes are seated against slip rings.

F. Drive Pulley

Tighten drive pulley retaining nut to a torque of 35-45 foot-pounds.



Alternator Brush Installation
Figure 3

G. Tests

(PIR_Hartzell OE-A2.)

(1) Bearing Test

Test for a faulty bearing by holding the alternator in one hand and snap-spin the pulley/shaft with the other. A defective bearing will be heard or felt. In noisy locations, press the alternator adjusting ear to your ear lobe and snap-spin the pulley/shaft a few times.

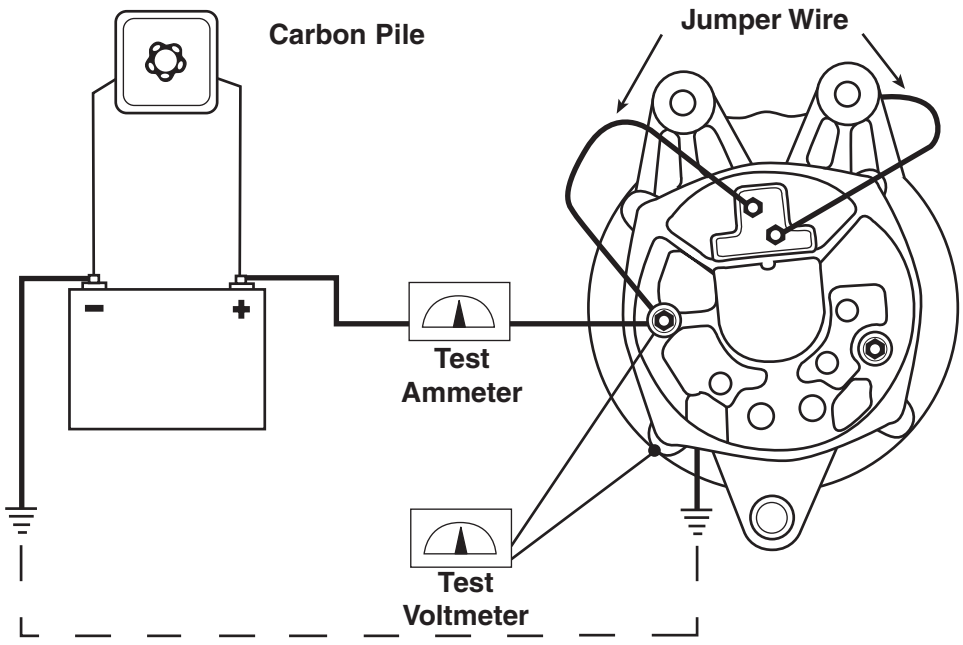
(2) Output Test

(a) Setup

1) The alternator is to be at room temperature 70°-80° F (21°to 26°C) before beginning the test.
2) Connect the alternator so that it is supplying its own field current. Refer to “Figure 4”.
3) The output voltage is to be controlled by an adjustable load.
4) The alternator is not to be run for longer than 2 minutes for each test point.
5) The 8500 Series alternators use a slip ring end cover that has a hose type connection for air pressure ventilation. Remove this cover when bench testing the alternators.

(b) Test

1) Adjust carbon pile to obtain specified voltage.
2) Run alternator to RPMs specified in “Chart 4”. Verify outputs within specified ranges.
3) If specified output cannot be obtained replace or overhaul alternator.



Testing Alternator
Figure 4

CHART 4
ELECTROSYSTEMS / HET ALTERNATOR TEST SPECIFICATIONS

Alternator Model	ES4032		ALU-8532LS	
Voltage	28-volts		28-volts	
Rated Output	70 amperes		75 amperes	
Ground Polarity	Negative		Negative	
Rotation	Bi-Directional		Bi-Directional	
Rotor (70–80° F):				
Current Draw	2.4 to 4.0 amps @ 28.0-volts		1.9 to 2.3 amps @ 24.0-volts	
Resistance	7.0–11.0 ohms		10.1–11.7 ohms	
Output Test (70–80° F):				
Volts	28.0	28.0	28.0	28.0
Min. Amperes	25.0	70.0	23.0	61.0
Alternator rpm	4000 min	6000 min	4000 min	8000 min

H. Overhaul

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

When required, overhaul the Electrosystems/HET alternator using the appropriate vendor publication as specified in Introduction, Supplementary Publications, Vendor Publications, Alternator.

8. Plane-Power 28 vdc, 70 amp Alternator AL24-P70-1 or AL24-P70-3

Installed with the IO-360-B4A engine and/or Garmin G1000 NXi in S/N 2843821, 2843823, 2843852 and up; and 2881001 and up. See "Figure 5" on page 243018 and "Figure 6" on page 243022.

NOTE: See latest revision of Piper Service Spares Letter 488 to install the AL24-P70-3 alternator as a service replacement in S/N's 2843001–2843822, 2843824–2843862.

In airplanes equipped with the IO-360-B4A engine and/or Garmin G1000 NXi:

- AL24-P70-1 alternators were factory installed in S/N's 2843821, 2843823, 2843852–2843880; and 2881001–2881004.
- AL24-P70-3 alternators were factory installed in S/N's 2843881 and up; and 2881005 and up.

The following instructions address the AL24-P70-3 alternator installation. For airplanes with the AL24-P70-1 still installed, refer to "Figure 5 (Sheet 3 of 4)" on page 243020.

NOTE: Airplanes equipped with the AL24-P70-1 alternator should install Piper Kit No. 88617-001 per Piper Service Bulletin No. 1307. This reworks the alternator into the AL24-P70-3 configuratrion.

A. Inspections

(1) 100 Hour / Annual

Each 100 hours time-in-service or annually, whichever comes first, remove drive belts and turn alternator rotors to inspect condition of bearings for abnormal noise or roughness.

(2) Five (5) year / 1,000 Hour

Each five (5) years or 1,000 hours time-in-service, whichever comes first:

(a) Remove alternator per "Removal", below.

(b) Remove field brush assemblies per "Figure 7" on page 243026 and inspect brushes for excess wear. Replace brush assemblies if brushes extend less than 0.250 inches from the edge of holder case.

(c) Install alternator per "Installation", below.

B. Removal

The alternator is mounted in a fixed position to support brackets and a clevis and arm assembly. Alternator belt tension is controlled solely by the idler pulley mounted on the lower bracket.

(1) Disconnect the negative (ground) battery lead.

CAUTION: NOTE NUMBER AND LOCATION OF SHIM(S) WHEN REMOVING IDLER PULLEY TO INSURE CORRECT REINSTALLATION.

(2) Remove the alternator idler pulley to reduce belt tension.

(3) Slip the alternator belt off of the alternator pulley when tension is released.

(4) Disconnect the electrical connections from the rear of alternator.

(5) Remove the lower alternator mount bolt, spacer, sleeve, washers and nut.

(6) Remove safety wire from the upper upper alternator mounting bolt.

(7) Supporting the alternator, remove the upper alternator mounting bolt and washers, and remove alternator.

CHART 5
ALTERNATOR PULLEY OFFSET VS. IDLER PULLEY NOMINAL LOCATION

Alternator Pulley Offset From Ring Gear Pulley	0.020	0.030	0.040	0.050	0.060
Idler Pulley Nominal Location	0.010	0.015	0.020	0.025	0.030

C. Installation

(1) Position the alternator upper mounting ear inline with the arm and support bracket. Loosely secure with bolt and washer(s).

NOTE: As shown in "Figure 5" on page 243018, or "Figure 6 (Sheet 1 of 4)" on page 243022 add or remove washers as required to maintain belt alignment. The alternator pulley is allowed to be a maximum of 0.060 misaligned from the ring gear pulley. The idler pulley should be shimmed per "Chart 5".

(2) Rotate the alternator lower mounting ear inline with the bottom support bracket and secure with bolt, spacer, washers (5 ea.), bushings (2 ea.), and nut. See Section A–A in "Figure 5".

(3) Returning to the upper mounting ear, tighten bolt and secure with safety wire.

(4) Connect the electrical connections to the rear of alternator. Torque per "Figure 5".

CAUTION: DURING INSTALLATION, NEVER FORCE BELT OVER PULLEY. REINSTALL WITH THE BELT ALREADY SEATED IN THE IDLER GROOVE.

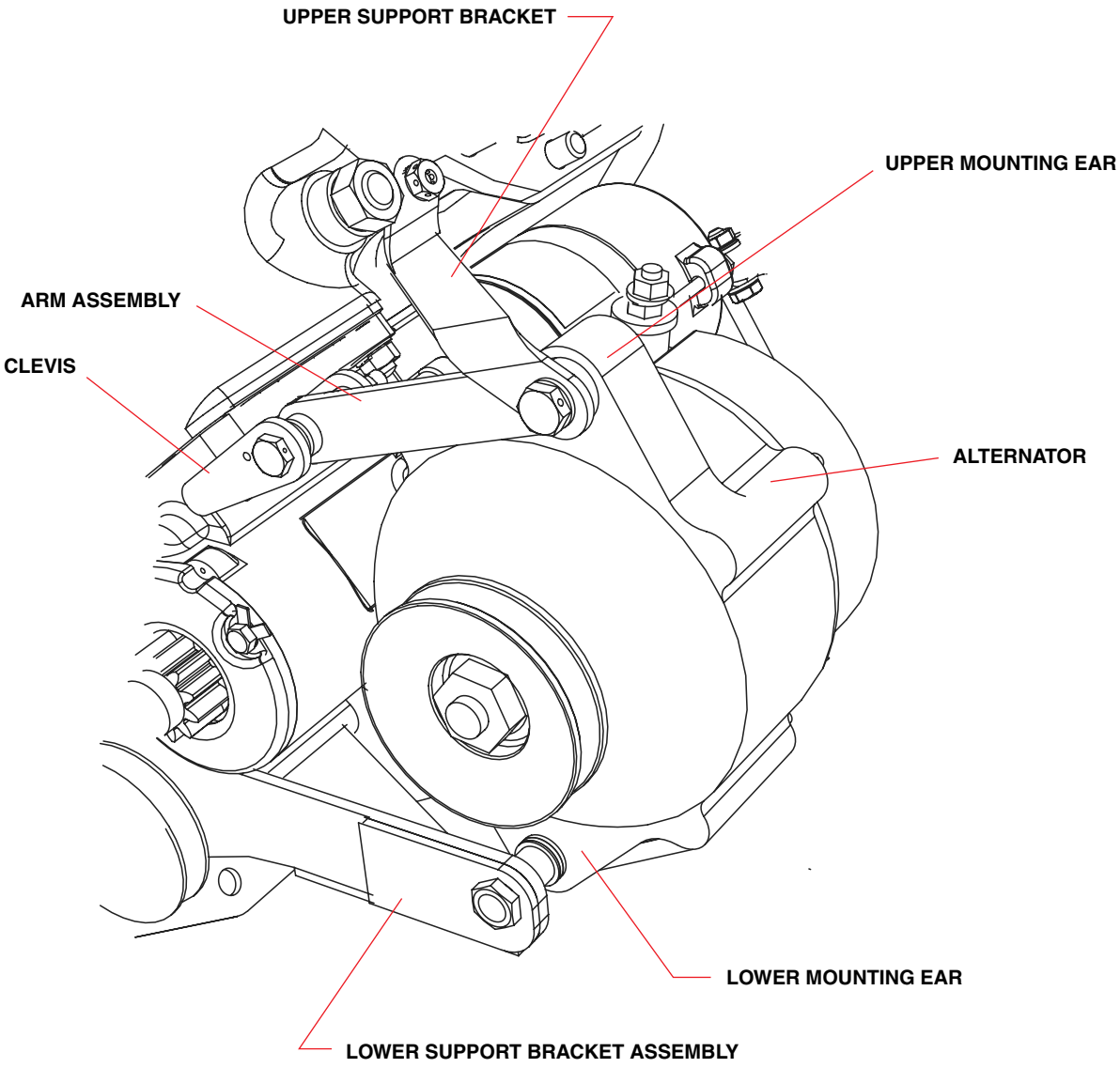
(5) Reinstall the alternator belt over the ring gear pulley and alternator pulley.

(6) Reinstall idler pulley with the belt already seated in the idler pulley groove.

(7) Tighten all alternator mounting bolts.

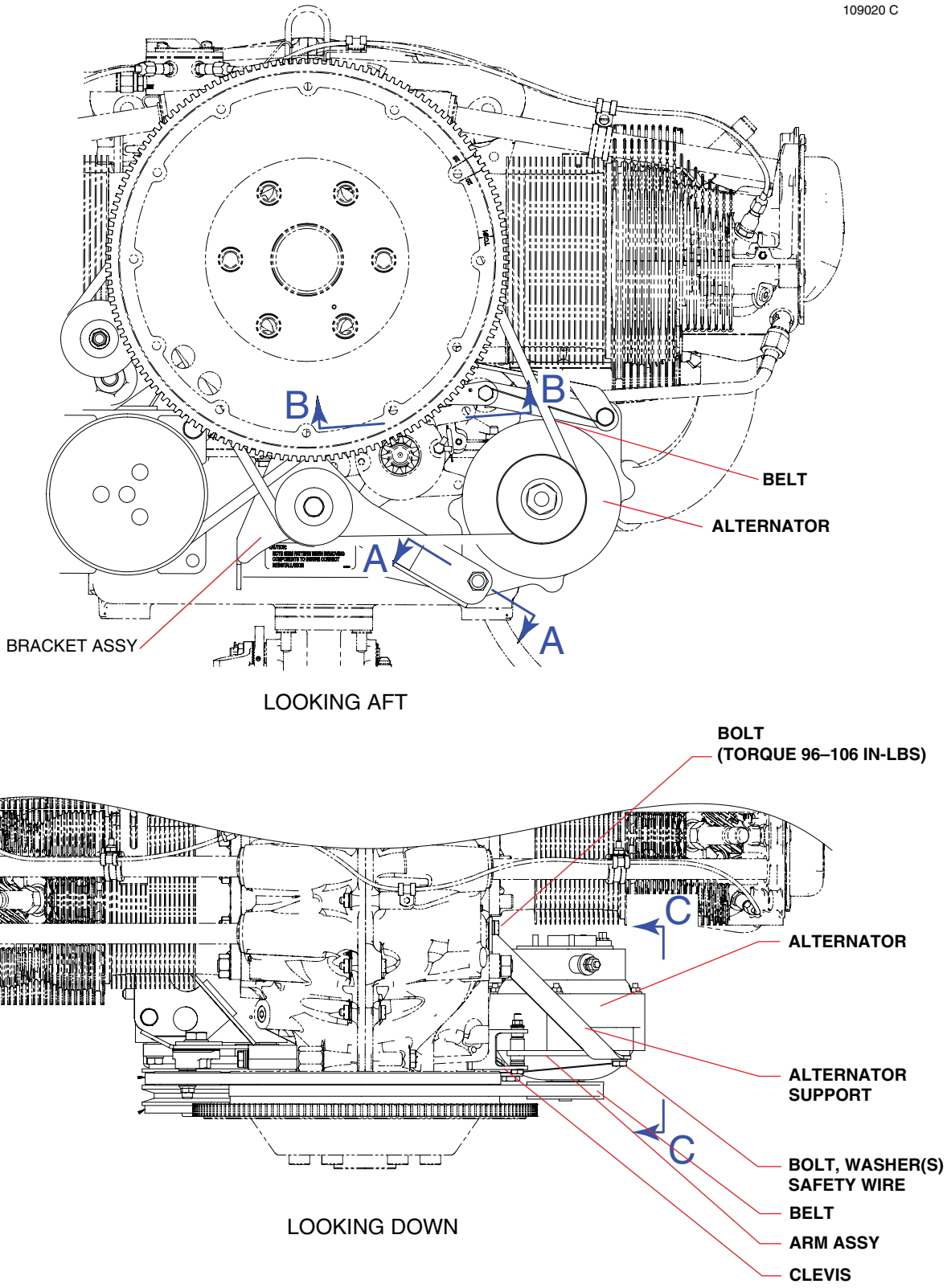
(8) Adjust belt tension per "Alternator Belt Tension Adjustment" on page 24307.

(9) Connect negative (ground) battery cable.

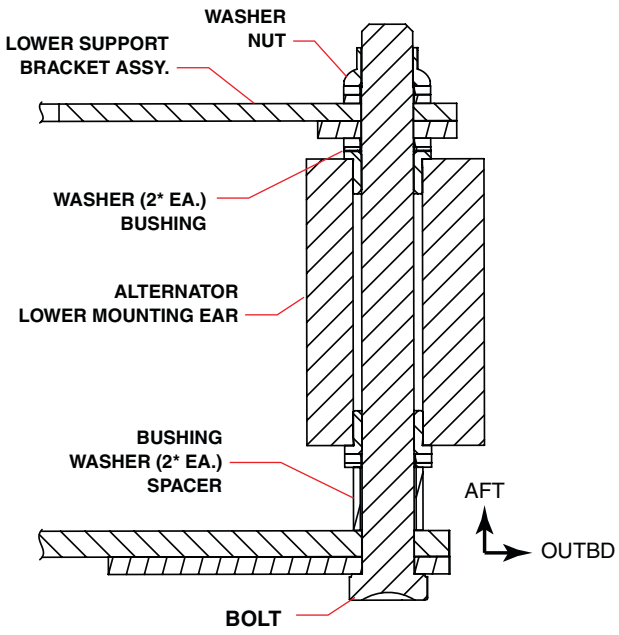


Alternator Installation
Figure 5 (Sheet 1 of 4)

Effectivity
with Plane-Power Alternator in 2843001–2843949;
and 2881001–2881230 pre-Piper SB 1340



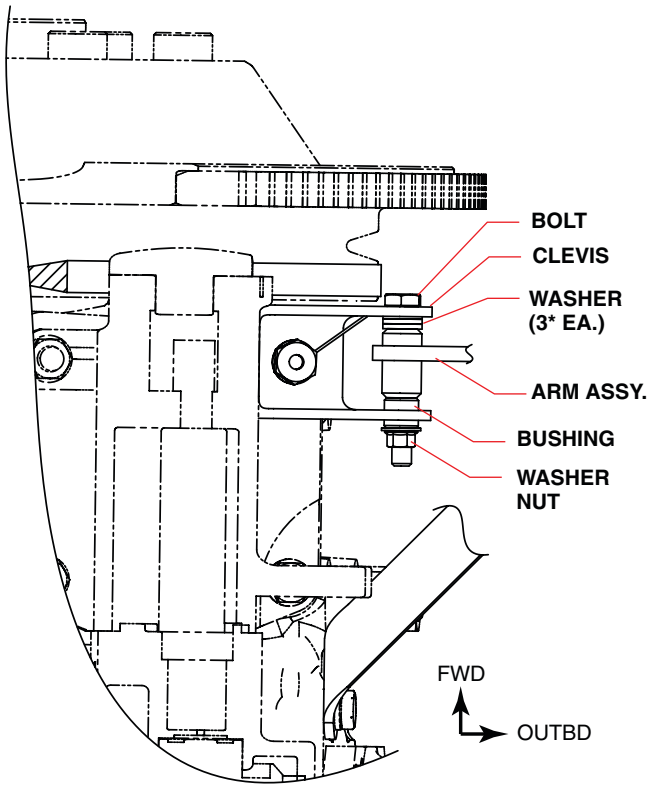
Alternator Installation
Figure 5 (Sheet 2 of 4)



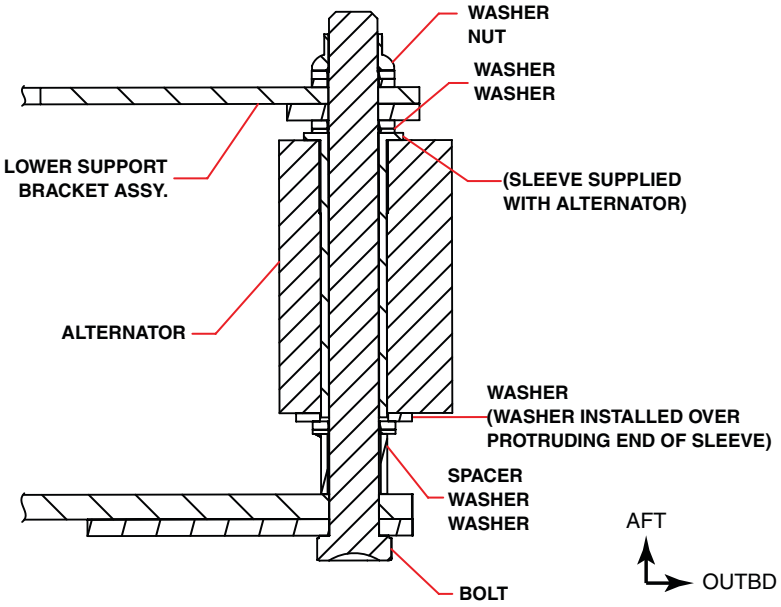
SECTION A - A

AL24-P70-1 CONFIGURATION

* NUMBER OF WASHERS AS REQUIRED, SEE NOTE IN TEXT.



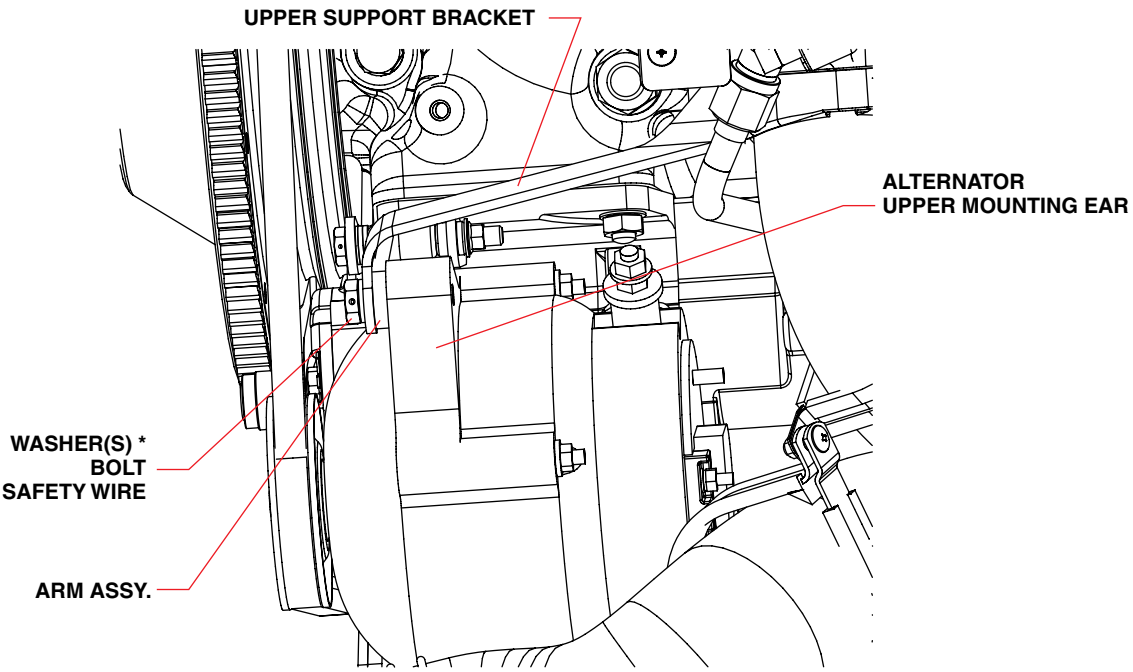
SECTION B - B



SECTION A - A

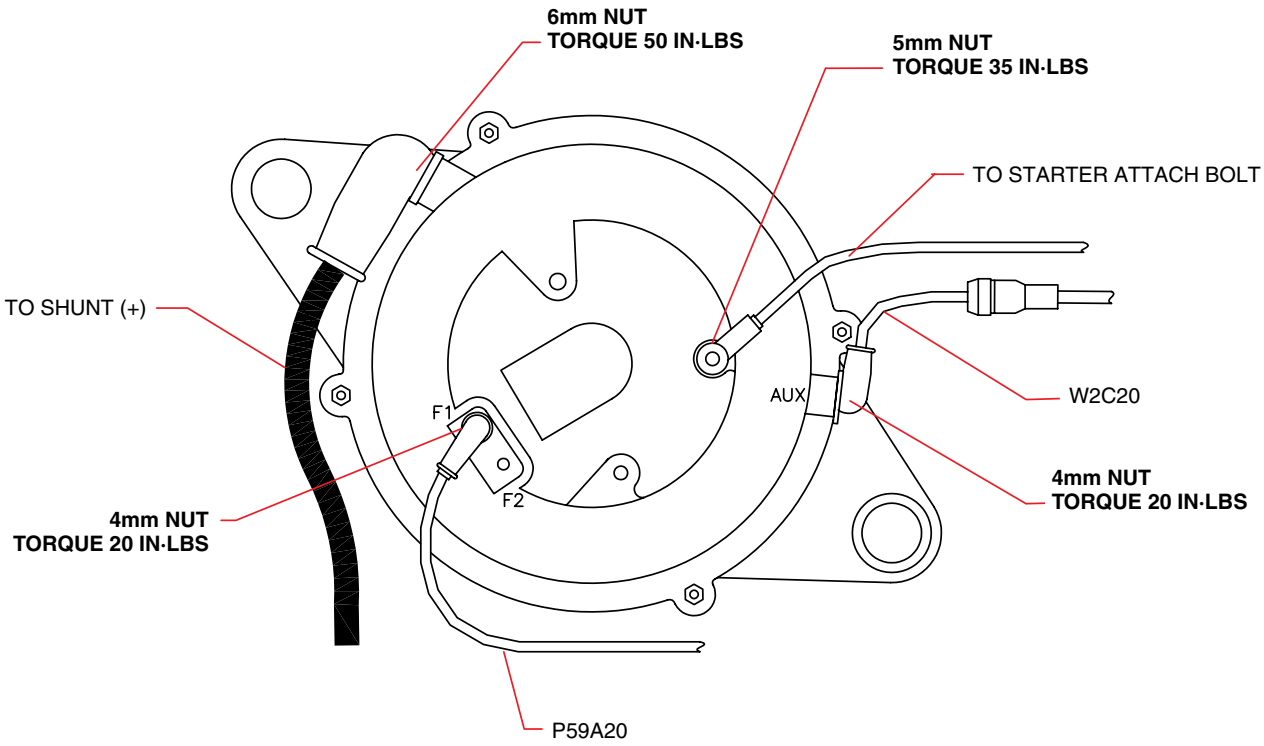
AL24-P70-3 CONFIGURATION

Alternator Installation
Figure 5 (Sheet 3 of 4)



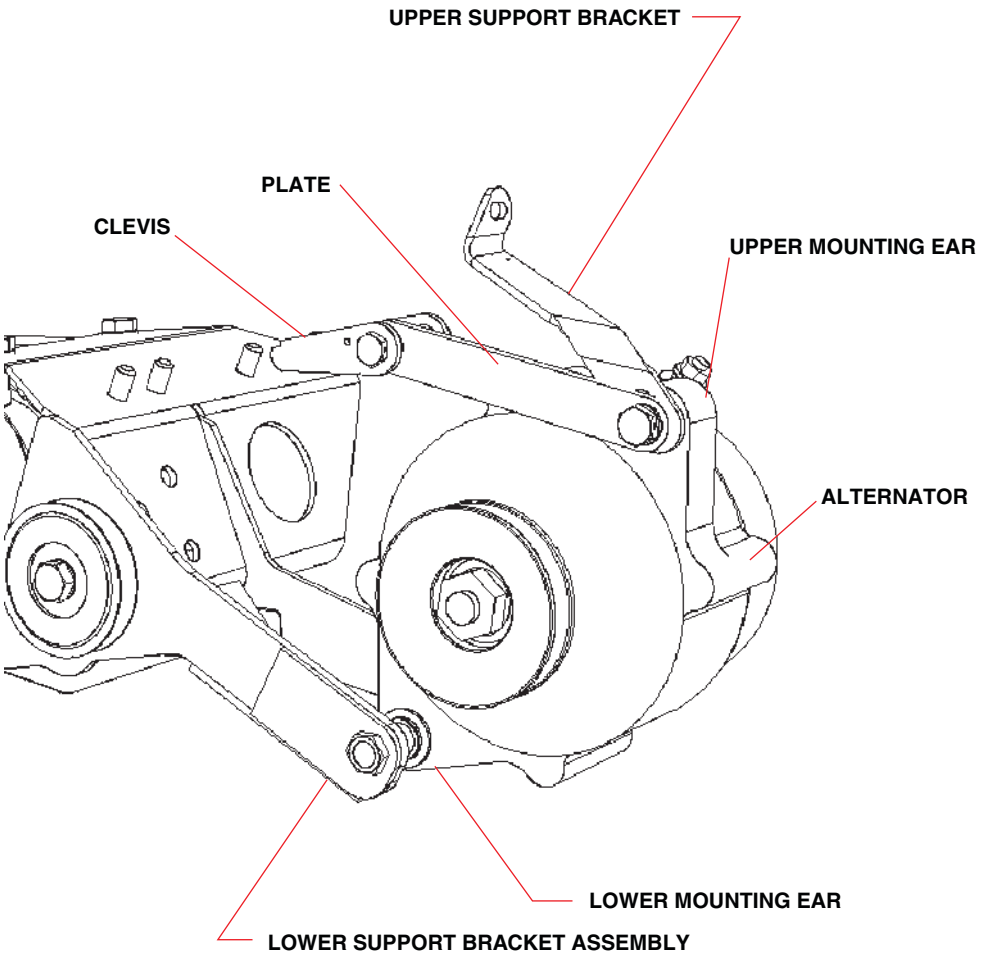
VIEW C - C

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LOOKING FORWARD AT REAR OF ALTERNATOR

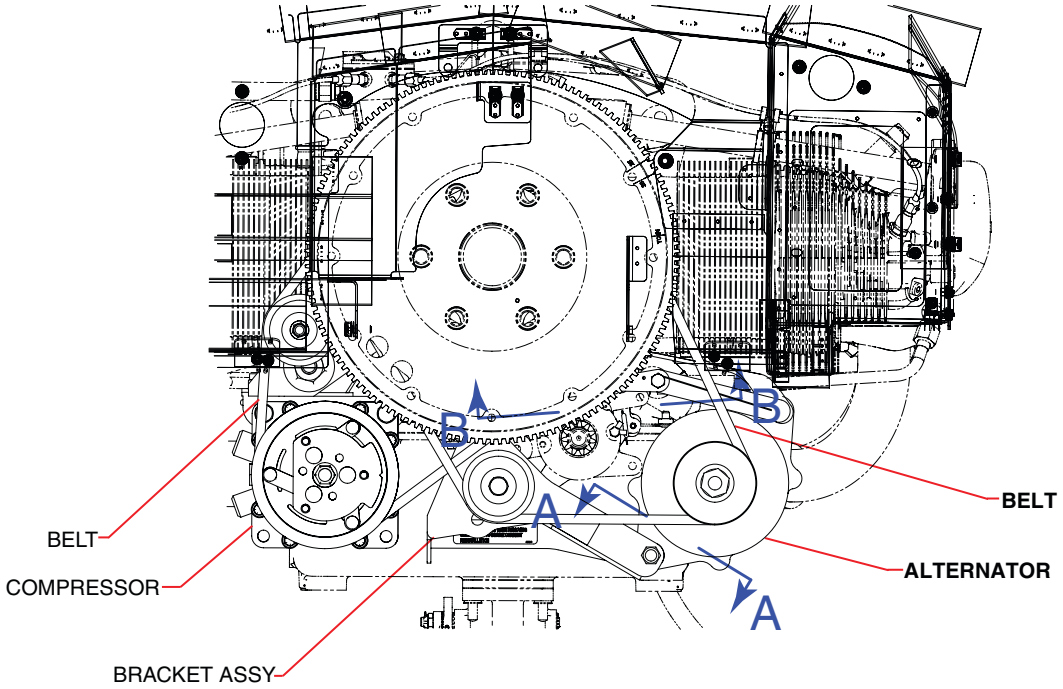
Alternator Installation
Figure 5 (Sheet 4 of 4)



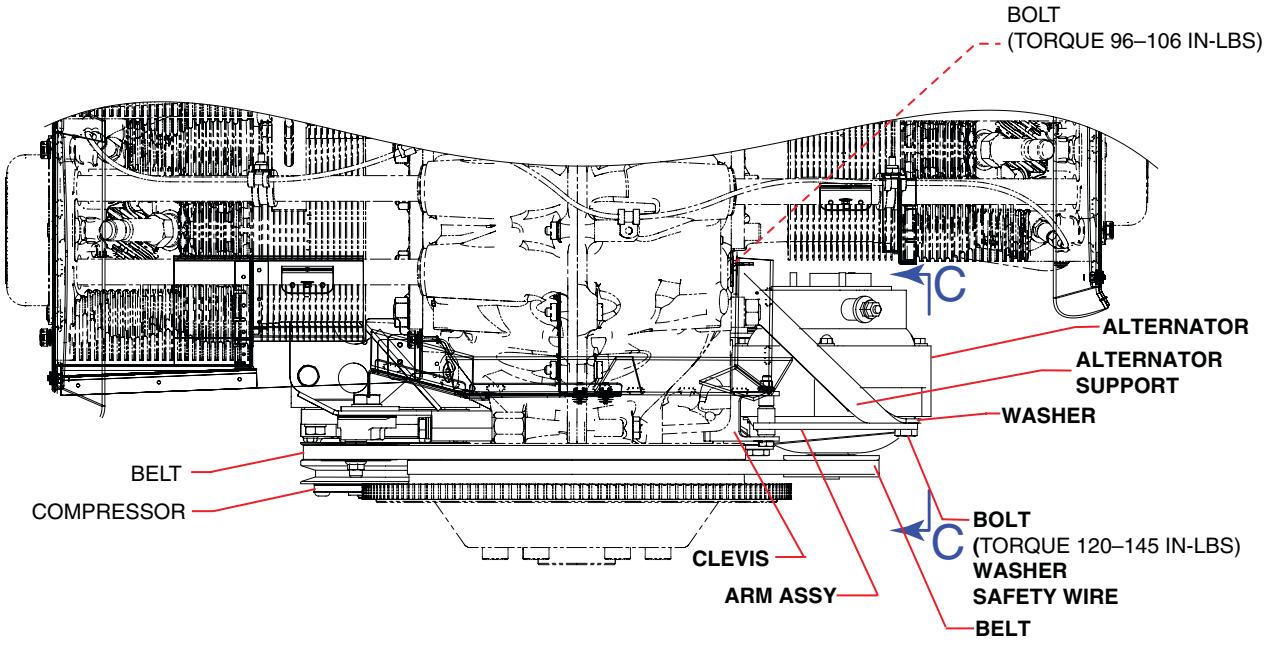
Alternator Installation
Figure 6 (Sheet 1 of 4)

Effectivity
with Plane-Power Alternator in 2881231 and up;
and in earlier airplanes post-Piper SB 1340

109020 H

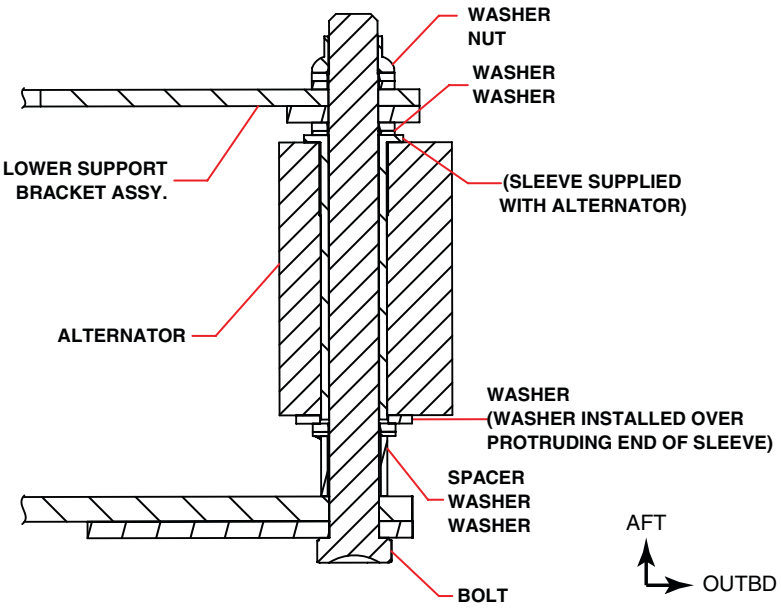


LOOKING AFT



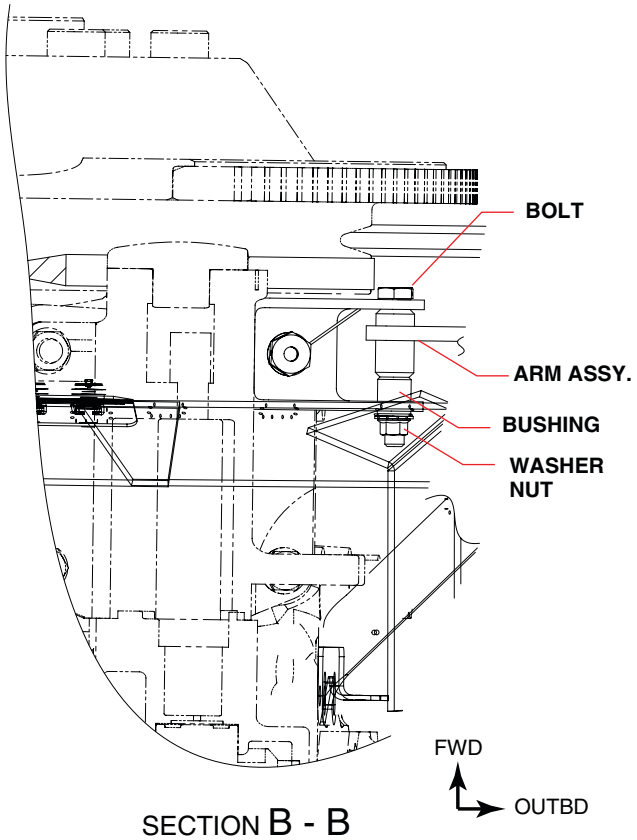
LOOKING DOWN

Alternator Installation
Figure 6 (Sheet 2 of 4)



SECTION A - A

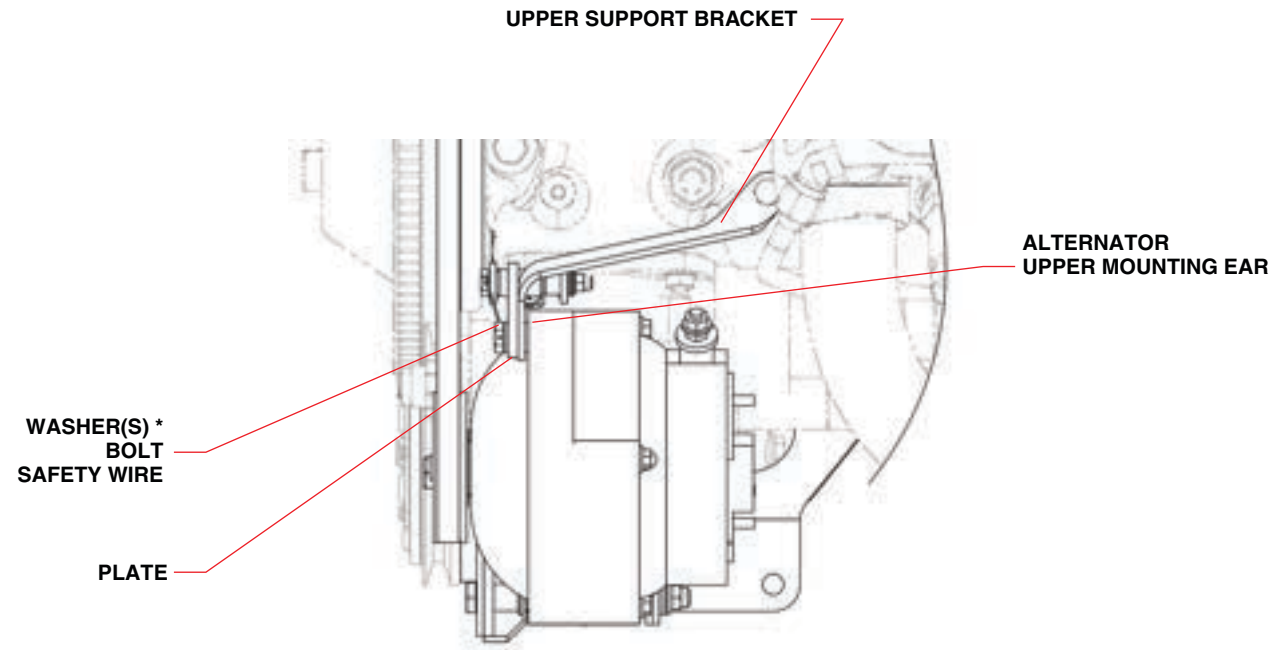
AL24-P70-3 CONFIGURATION



SECTION B - B

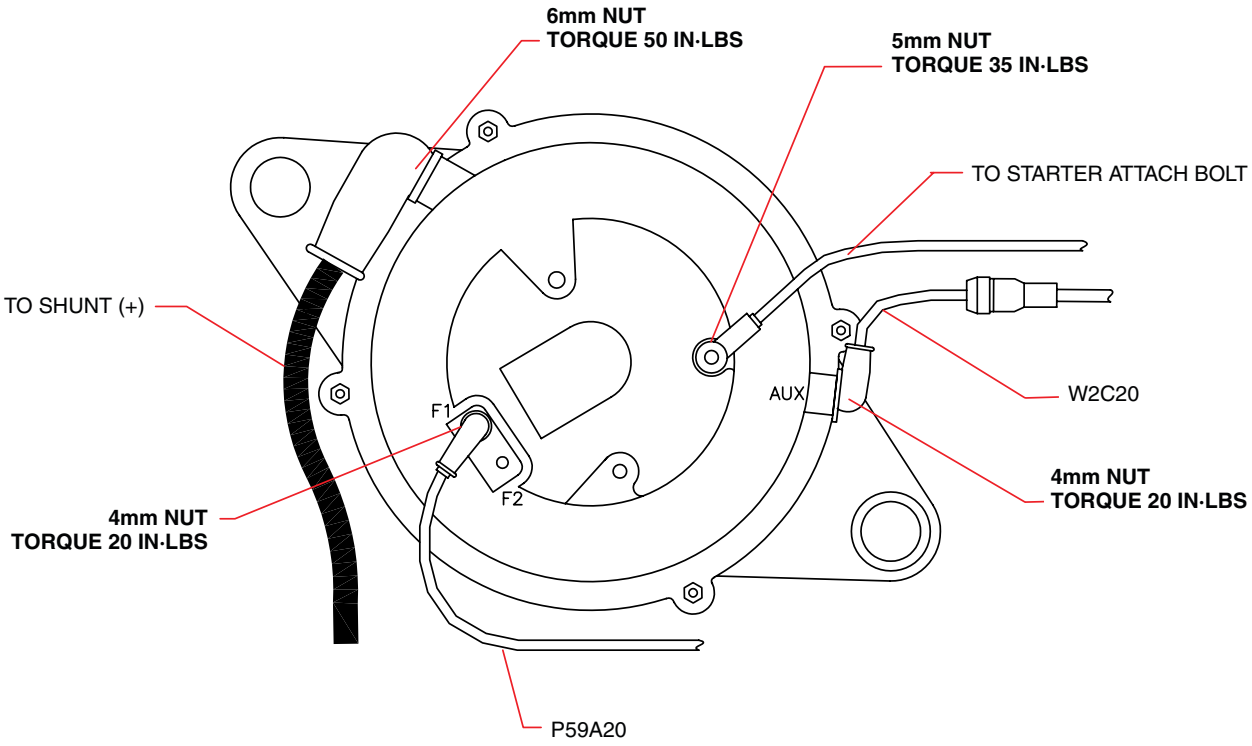
Alternator Installation
Figure 6 (Sheet 3 of 4)

Effectivity
with Plane-Power Alternator in 2881231 and up;
and in earlier airplanes post-Piper SB 1340



VIEW C - C

107570 F

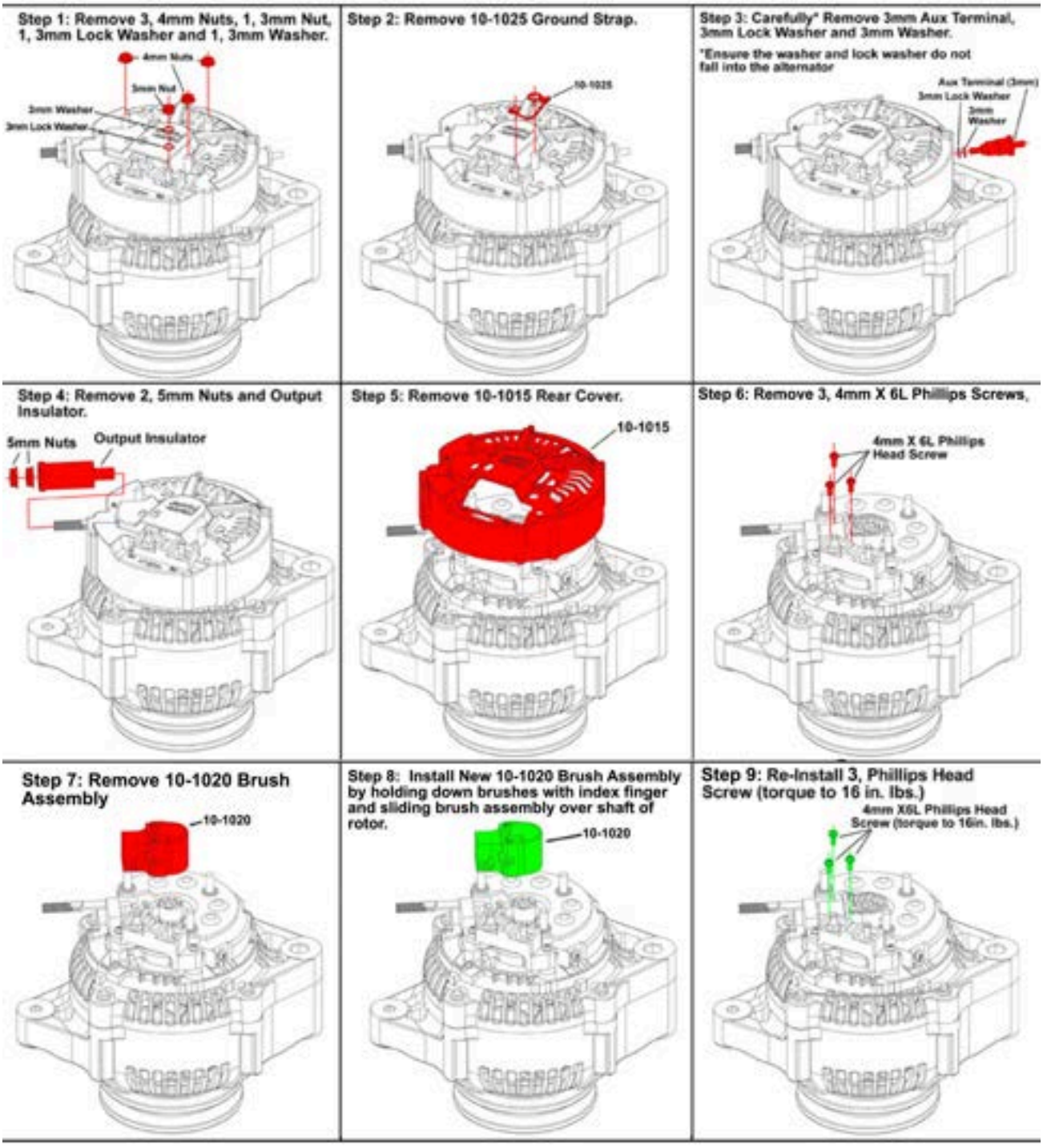


LOOKING FORWARD AT REAR OF ALTERNATOR

Alternator Installation
Figure 6 (Sheet 4 of 4)

Replacing Brush Assembly

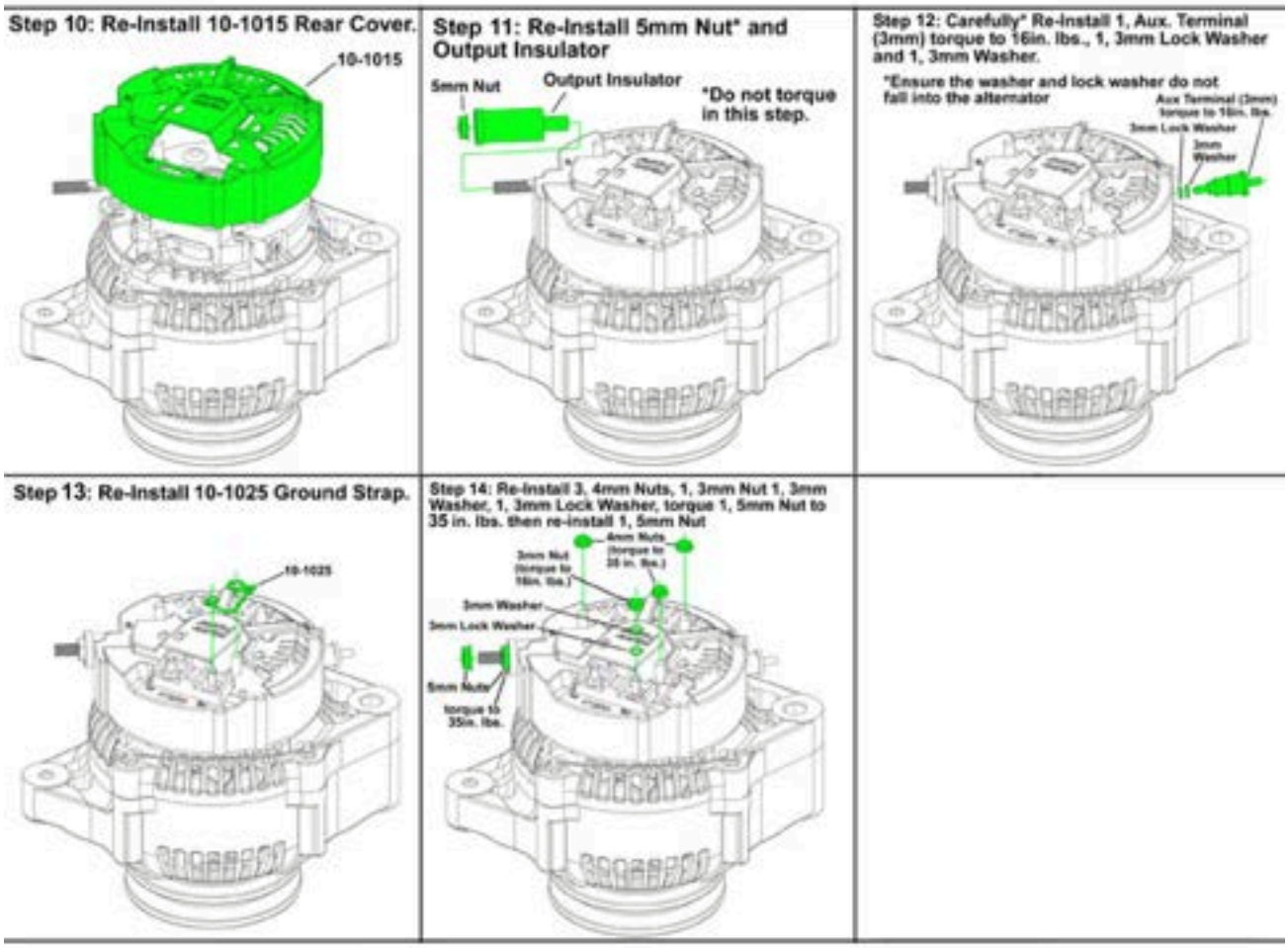
Drawing: 10-9999-3



Replacing Brush Assembly
Figure 7 (Sheet 1 of 2)

Replacing Brush Assembly

Drawing: 10-9999-3



Replacing Brush Assembly
Figure 6 (Sheet 2 of 2)

9. Standby Alternator

Installed with Avidyne Entegra in S/N's 2843622 thru 2843682.

A. Description

The standby system provides 20 amps of power to support continued flight in the event of primary alternator failure. It turns on automatically, annunciating its operation to the pilot through the panel mounted annunciator which doubles as a standby alternator load monitor.

If the primary alternator fails in flight, the standby regulator will sense the drop in system voltage and automatically activate the standby alternator. If the current requirement is over 20 amps when the standby alternator is activated, the STBY ALT ON annunciator light will flash. Reducing the current usage to 20 amps or less will cause the annunciator light to cease flashing and light steadily. The pilot may choose which equipment he needs for the given flight conditions by simply keeping the total load below the flashing point of the annunciator. This will reserve battery energy for transient loads (gear, flaps, landing lights, etc.) during approach. Loads may be beyond the flashing point of the annunciator light for up to five (5) minutes without damaging the standby alternator.

The standby alternator is mounted on the vacuum pump drive of the engine accessory case. Panel mounted equipment includes the addition of a STBY ALT ON light to the annunciator, a STBY ALTR rocker switch and three standard pull type circuit breakers (1 amp, 5 amp and 40 amp). The regulator is mounted under the floor in the forward baggage compartment, right side.

See 39-20-00 for illustrated component locations and Chapter 91 for electrical schematics.

B. Troubleshooting

(PIR-BC203-2D-trouble.wpd, Rev. C.)

See "Chart 6".

C. Inspection

100 Hour Standby Alternator Inspection

Each 100 Hours time-in-service, or at each annual inspection, whichever comes first, inspect the standby alternator installation as follows:

(1) Standby Alternator (BC410)
  (a) Inspect security and condition of standby alternator mounting and wiring.
  (b) Perform a normal preflight inspection
  (c) Move the aircraft to an area safe for engine start.
  (d) Perform a normal engine start and allow the engine to reach proper temperature for runup RPM
  (e) Assure that the standby alternator "Field" and "Sense" circuit breakers and the "Standby Alt" master are in the on position.
  (f) Reduce system electrical loads to approx. 10-15 amps.
  (g) Set engine to 2000 RPM minimum.
  (h) Switch primary alternator field switch to OFF.
  (i) Check that the "Stby Alt On" annunciator lights.
  (j) Increase the electrical load to over 20 amps. The "Stby Alt On" annunciator should be blinking. Reduce the electrical load to less than 20 Amps. The "Stby Alt On" annunciator should be ON steadily.
  (k) Switch primary alternator field switch to ON and verify primary alternator operation. The "Stby Alt On" annunciator should be OFF.
  (l) Return the engine to idle RPM.

CHART 6 (Sheet 1 of 3)
TROUBLESHOOTING STANDBY ALTERNATOR REGULATOR (BC203-2D)

Following this procedure will locate the most common system problems encountered. When calling for additional factory support, the information gained from these checks should be readily available to aid your Piper Dealer's Service Advisor in narrowing the field of possibilities as quickly as possible. Record the various measurements in the spaces indicated.

Refer to the electrical schematics in 91-24-30 and a high impedance (preferably digital) volt/ohmmeter (DVM) to make the following checks.

1. Setup

   A. Ensure the engine is not running,

   B. The magnetos are OFF, and

   C. That no external power is applied to the aircraft electrical system.

2. Turn all switches off. Use the lowest resistance scale on the DVM. Check resistance between the battery negative (-) terminal and both Terminal 7 of the regulator and the engine case. Measurements over 0.5 Ohm to either would be cause for investigation. In this case, check the engine ground strap, battery ground strap, and regulator ground wire for loose or contaminated connections, broken conductors or bad crimp joints. If these measurements are less than 0.5 Ohm, any of these three points may be used as reference (-) for the following measurements.

   A. Resistance from battery to Terminal 7: _____Ohms.

   B. Resistance from battery to engine case: _____Ohms.

3. Turn on the battery master and alternator field switches. Measure the voltage on the battery bus and on Terminal 1 of the regulator. The voltages should be equal within 0.2 volts. A difference of greater than 0.2 volts may be caused by using a breaker as the source for Terminal 1 that supplies another device of considerable load. Change to a lightly loaded breaker or a breaker dedicated to Terminal 1 and the low voltage lamp. It is recommended that Terminal 1 not be jumped to Terminal 6. If Terminal 1 has no voltage, the regulator will not operate.

   A. Bus voltage: _____ volts.

   B. Terminal 1 voltage: _____volts.

4. Measure the voltage on Terminal 6 of the regulator. It should be within 0.5 volts of the bus voltage. A difference of greater than 0.5 volts may be caused by poor contacts in the field breaker or field switch, poor crimp joints, or loose screw terminals in the wiring between the bus and Terminal 6. Absence of voltage on Terminal 6 will prevent the regulator from operating.

   Terminal 6 voltage:_____volts.

5. Check the voltage on Terminal 5 of the regulator. The voltage should be between 13 and 15 volts. A voltage outside this range may indicate a bad regulator.

   Terminal 5 voltage:_____volts.

CHART 6 (Sheet 2 of 3)
TROUBLESHOOTING STANDBY ALTERNATOR REGULATOR (BC203-2D)

6. Move to the engine compartment. Without disconnecting the field connector, measure the field voltage on the alternator. Use a thin probe or small gage wire wrapped around the probe to reach through the connector body and measure the voltage on the male blade coming out of the alternator. It should measure within 0.5 volts of the measurement on Terminal 5 of the regulator. A lack of voltage may indicate an open circuit between Terminal 5 of the regulator and the field terminal. If an open field circuit is suspected, the switches may be turned off, the alternator field connector removed, and a resistance measurement made between the connector and Terminal 5 of the regulator. Look for near 0 Ohms. Typically the field resistance of the alternator will be between 3 and 10 Ohms from the male field terminal blade to alternator case.

   A. Field terminal voltage:_____volts.

   B. Alternator field resistance:_____Ohms.

7. With the switches on, check the voltage between the alternator output post (or B lead) and ground. It should be battery voltage. If not, check the wiring between the alternator B lead and the battery positive (+) terminal. Look for loose or contaminated connections, broken wires, or an open breaker or fuse.

   Alternator B lead voltage:_____volts.

8. If all of the voltages in the first 6 steps are close to the value specified, the charging system should be operative. If not, check for a broken or loose alternator belt or, if the alternator is spline driven, check that the spline drive shear coupling is not sheared. Note that on spline drive alternators the engine speed must be at or above run-up RPM before checking for useable output.

   NOTE: During the following tests, if the annunciator is always ON or always OFF, check the annunciator circuit by removing the wire from terminal 2 of the regulator. The lamp should be OFF with the wire disconnected and should illuminate if the wire is connected to ground.

   If the lamp circuit works properly but the lamp still remains fixed either ON or OFF all the time with the lamp wire connected to Terminal 2, assume that the lamp driver is bad in the regulator. Have the regulator repaired or replaced.

9. If the charging system seems to be generating power, but the STBY ALT ON annunciator will not flash above 20 amps output, check the following:

   A. With the Battery and Standby Alternator Master switches ON but the engine not running, check the voltage between terminals 4 and 7 of the regulator. The voltage should be 10.0, " 0.1 volts. If it is not within tolerance, remove the wire from Terminal 4 and re-check the voltage. If it is still not within tolerance the regulator may be bad. If the voltage is now in tolerance, suspect a bad current sensor or shorted sensor wire.

   B. If the voltage originally measured in step A was in tolerance, divide the measurement by 2. This value should be the voltage measured between terminals 3 and 7 of the regulator. If the voltage measured is not close to 5.0 Volts, suspect the current sensor in the standby alternator output lead.

CHART 6 (Sheet 3 of 3)
TROUBLESHOOTING STANDBY ALTERNATOR REGULATOR (BC203-2D)

C. If the measurements in Steps A and B are satisfactory, prepare to monitor the voltage between terminals 3 and 7 of the regulator with the engine running. Move the aircraft to a safe location for a 2000 RPM runup. Start the engine. Minimize the bus loads and bring the engine RPM up to approximately 2000. Switch OFF the primary alternator. Wait for the STBY ALT ON annunciator to illuminate. At light bus loads this could take a minute or two. Note the voltage between terminals 3 and 7 of the regulator and begin switching ON additional bus load. The monitored voltage should increase at the rate of 0.056 Volts for each Amp of standby alternator load.

(1) If the voltage does not change, verify standby alternator output with the ship's ammeter or a clamp-on ammeter probe. If the standby output is verified and there is still no change in the monitored voltage, suspect a bad current sensor.

(2) If the voltage decreases instead of increasing, the 10 Ga. current sensor wire is connected backward. Reverse the connection and re-run the test.

(3) If the voltage increases at approximately the correct rate, continue adding load until the voltage is greater than 6.2 volts. At or above this voltage, the lamp should be flashing. If not suspect a bad regulator.

10. Intermittent problems are the hardest to find. Temporarily bring small test wires into the cockpit from 2 or 3 points in question to allow monitoring them with the DVM during periods of system failure. Double check all screw terminals for security. Try a 5 pound pull test on all crimp joints and make sure that the terminal is crimped on the wire, not the insulation.

11. Noise problems are also difficult to find. A few tips to help with curing noise problems follow:

A. A unitized grounding system helps prevent noise problems by preventing voltage differences between different ground points.

B. The battery acts as a noise filter in the system. Poor connections to the battery or a battery that is going bad can add to or even cause noise problems.

C. Shielding of low level audio leads (especially microphone leads or headset leads) is required. Sometimes the shields in the cables can separate from repeated flexing. Try checking shield continuity with an ohmmeter or substituting another headset, microphone, etc.

D. Wire routing may be important in some installations. Separation of noise carrying conductors such as P leads from other wiring may help. Running noisy wiring parallel to other wiring in the same bundle is asking for trouble. Wires at 90 degrees to one another, however, do not couple noise.

E. Running transmitter feed lines close to and in parallel with other wiring can cause a problem. Normally, problems will only be encountered if there is a mismatch and therefore a high SWR in the antenna system. If noise or charging system breaker tripping occurs during Comm transmit only or when the transponder is ON only, check the corresponding antenna system carefully or separate the transmission line from other wiring.

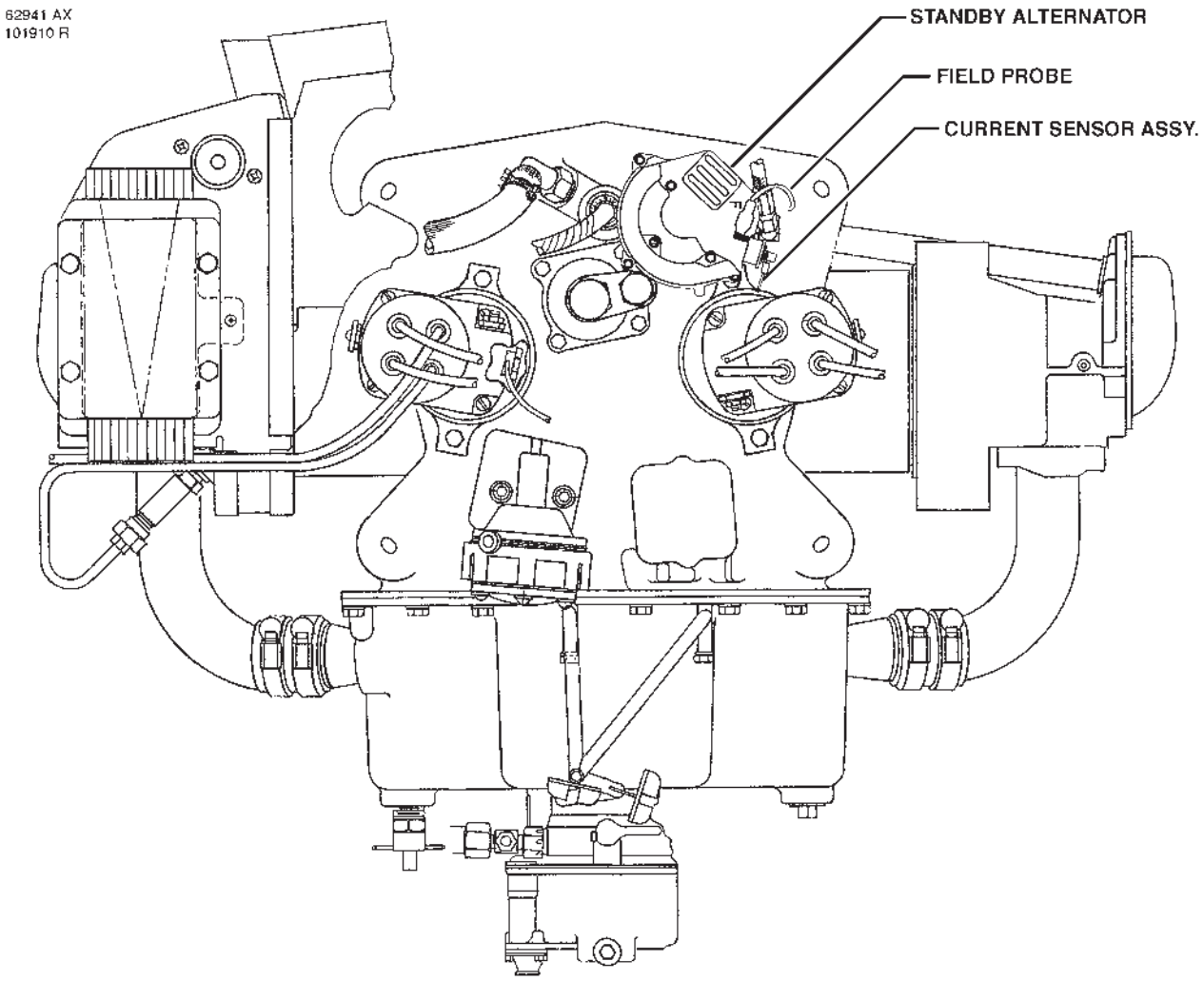
F. The best plan is to stop the noise at its source. Once the noise is loose, it can be difficult to filter it out of all affected systems. Try to locate the offending item and correct the problem at that point. Switching off the alternator, the mags (first one then the other), or any other electrical equipment that generates noise should help to find the offender.

(2) Standby Alternator Regulator (BC203-3D)
  (a) Inspect security and condition of the case mounting bolts and wiring.
  (b) Inspect overvoltage protection by performing Overvoltage Test, below.

(3) Standby Alternator Regulator (BC203-3D)
  (a) Inspect security and condition of the case mounting bolts and wiring.
  (b) Inspect overvoltage protection by performing Overvoltage Test.

D. Removal

(1) Disconnect wiring harness.
(2) Remove washers and nuts (4).
(3) Pull standby alternator and gasket from drive pad.



Standby Alternator Installation
Figure 8

E. Installation

(1) Orientate standby alternator as shown in “Figure 8”.
(2) Locate new gasket and standby alternator on drive pad.
(3) Secure with washers and nuts (4 each).
(4) Torque nuts 60 to 70 in.-lbs.
(5) Connect wiring harness.

F. Set Point Voltage Check

Voltage adjustment of the BC203-2D is not normally required. Deviation from the factory set point of 26.0 volts by more than 0.2 volts may indicate the need for repair or replacement. This set point voltage can be checked as follows:

(1) Connect a calibrated digital voltmeter directly between terminals 1 and 7 of the regulator.
(2) Ensure the airplane is in a suitable area and start the engine.
(3) Running the engine at over 2000 RPM and with a total bus load of under 3 amps, the voltmeter should indicate 26.0 ± 0.2 volts.

G. Overvoltage Test

The BC203-2D contains internal over-voltage protection. Grounding for both regulation and overvoltage protection is achieved through terminal 7 of the regulator, through the case mounting bolts and through the grounding stud provided under the terminal strip. Overvoltage protection may be tested for correct operation using either of the following two tests:

(1) Test 1
  (a) Disconnect the aircraft wiring harness from terminal 6 of the regulator.
  (b) Connect a current limited power supply with an output voltage adjustable between zero and 35 volts to terminals 6 and 7 with the positive lead on terminal 6.
  (c) Connect a 10 ohm, 10 watt resistor from terminal 6 to terminal 5.

  CAUTION: THE POWER SHOULD BE REMOVED FROM TERMINAL 6 WITHIN FIVE (5) SECONDS OF ACHIEVING THE SHORTED CONDITION.

  (d) Limiting the output current to 5 amps or less, gradually raise the power supply voltage until the regulator shorts the output of the power supply. Remove power from terminal 6 as soon as the short occurs.
  (e) The short should occur between 32.0 and 33.0 volts. No short indicates the failure of the overvoltage protection circuitry of the regulator and necessity for repair or replacement of the regulator.
  (f) If the test is satisfactory, switch power OFF, disconnect the power supply, remove the 10 ohm resister, and reconnect terminal 6 to the aircraft wiring harness.
(2) Test 2
  (a) Disconnect the aircraft wiring harness from terminal 6 of the regulator.
  (b) Connect a 5 amp in-line fuse from the aircraft bus to the negative terminal of a 12 volt lantern battery.
  (c) Connect a 10 ohm, 10 watt resistor from terminal 6 to terminal 5.
  (d) Energize the aircraft Bus and momentarily connect the positive terminal of the lantern battery to terminal 6 of the regulator.

(e) The fuse should blow immediately. If the fuse does not blow, the over-voltage protection circuit has failed and the regulator must be replaced or repaired.

(f) If the test is satisfactory, switch power off, reconnect terminal 6 to the aircraft wiring harness, and remove the 10 ohm resistor.

H. Repair

Failure due to broken wires or damaged connectors may be corrected in the field using repair procedures complying with the latest revision of AC43.13-1. All other repairs are by replacement only.

10. Battery (Gill G-243)

Installed in S/N's 2843001 thru 2843597. And in S/N's 2843598–2843865 when Standard, Avidyne Entegra, or Garmin G500 equipped. See "Figure 9" on page 243038.

CAUTION: BATTERY FLUID MUST NOT BE FILLED ABOVE BOTTOM OF THE SPLIT RING.

Access battery through the aft side of the baggage compartment. It is in a protective box with vent system that incorporates a lead acid recovery jar. To draw off fumes produced during the battery charging process, the jar is vented overboard through a vent tube in the floor of the airplane. Any electrolyte overflow will be captured by the jar to be disposed of in an approved manner during servicing.

A. Troubleshooting

See "Chart 7" on page 243036.

B. Removal

CAUTION: ALWAYS REMOVE THE GROUND CABLE FIRST AND INSTALL IT LAST TO PREVENT ACCIDENTAL SHORT CIRCUITING OR ARCING.

(1) Remove the access panel to aft section of fuselage.
(2) Cut ty-wrap and remove vent jar tube from battery elbow.
(3) Remove wing nut and disconnect negative (ground) cable.
(4) Pull back rubber battery terminal boot on positive cable, loosen wing nut and remove the positive cable.
(5) Remove the two each AN3-73A bolts, AN970-3 washers, and 84316-2 insulators securing battery cover. Remove cover.
(6) Remove battery.

C. Installation

(1) Check battery and battery tray are clean and free of acid.
(2) Position battery in battery tray.
(3) Install vent jar tube to battery elbow. Secure with an MS3367-4-9 ty-wrap.
(4) Install battery cover and secure with two each 84316-2 insulators, AN970-3 washers, and AN3-73A bolts.

CAUTION: DO NOT TIGHTEN BATTERY COVER HOLD DOWN BOLTS MORE THAN 10 INCH-POUNDS

(5) Connect positive lead to positive battery terminal and secure.
(6) Connect ground cable to negative battery terminal and secure.
(7) Install the access panel to aft section of fuselage.

D. Cleaning

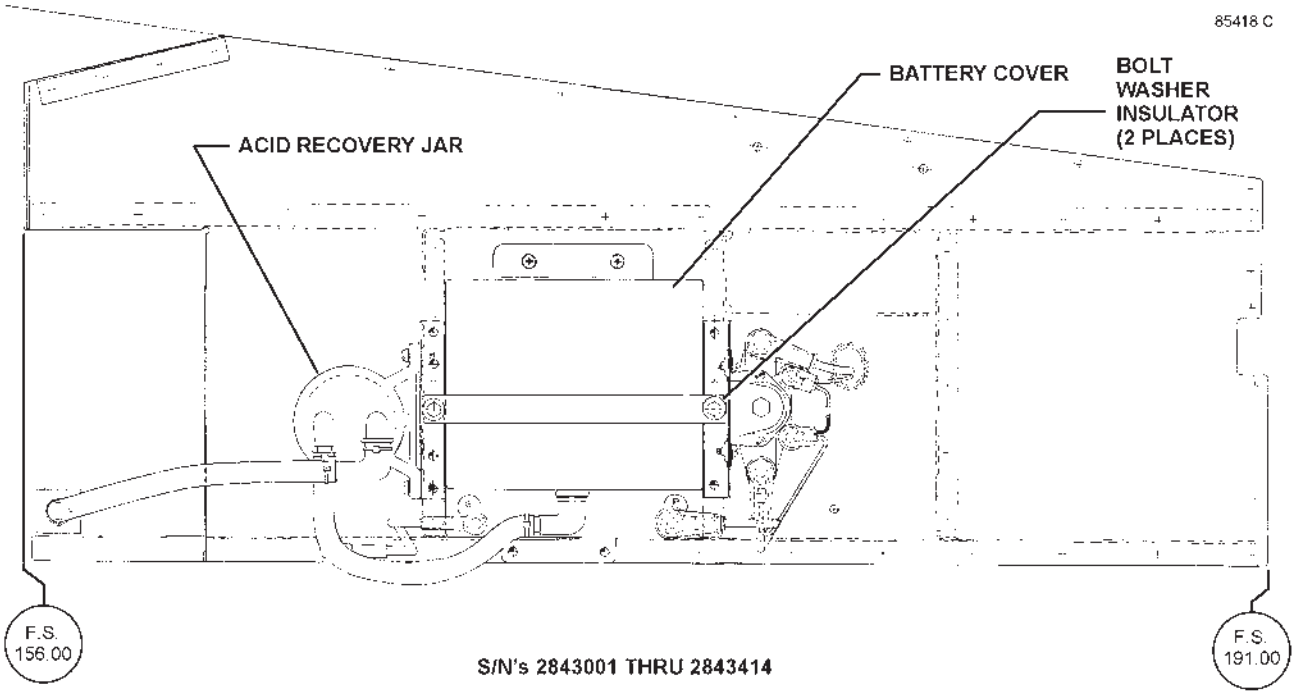
(1) Remove all accumulated contamination from the battery exterior with a stiff bristle brush. (Do not use a metal brush or abrasive materials.) Wipe exterior of battery and interior of manifold, including manifold top cover, with a cloth saturated with a solution of bicarbonate of soda mixed - one part soda to twenty parts of water. (Check that cell plugs are tight - do not allow soda solution to enter any cells.)
(2) Wash entire battery with clear water and dry thoroughly.
(3) Wash down the battery support and floor area, hold down supports, connectors and cable ends with a soda solution followed by clear water. Dry entire area and component parts thoroughly. Apply fresh acid resistant paint if required.

CHART 7 (Sheet 1 of 2)
TROUBLESHOOTING BATTERY

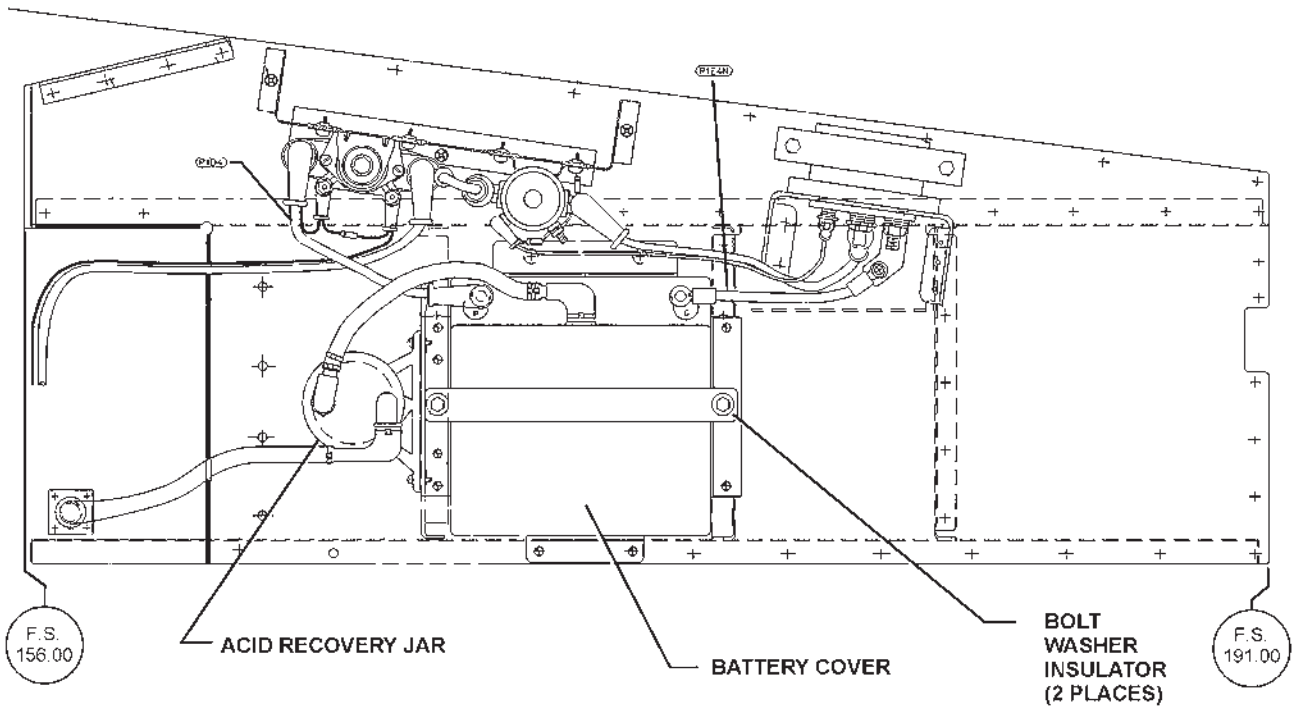
Trouble	Cause	Remedy
Battery installed with reversed polarity.	Battery charged backwards.	Remove battery and install with correct polarity.
		Remove battery. Connect load such as landing light, lamp, or similar load, and discharge battery. Charge with correct polarity and test each cell for signs of damage due to reversed charging.
NOTE: This condition can only occur where a discharged battery has been removed from airplane and put on a charger with polarity reversed. Such a reversal in polarity cannot occur in airplane due to a faulty alternator system.		
Discharged battery.	Battery worn out.	Replace battery.
	Improper charging rate.	Adjust voltage regulator.
	Standing too long.	Remove and charge battery if left in unused airplane three weeks or more.
	Equipment left on accidentally.	Remove and charge battery.
	Impurities in electrolyte.	Replace battery.
	Short circuit (ground) in wiring.	Check wiring.
	Broken cell partitions.	Replace battery.
Battery life is short.	Overcharge due to level of electrolyte being below top of plates.	Maintain proper electrolyte level.
	Sulfation due to disuse.	Replace battery.
	Impurities in electrolyte.	Replace battery.
	Low charging rate.	Adjust voltage regulator.
Cracked cell partition(s).	Hold-down bracket loose.	Replace battery and properly tighten hold-down bracket.
	Frozen battery.	Replace battery.
Compound on top of battery melts.	Charging rate too high.	Reduce charging rate by adjusting voltage regulator or replace transistorized regulator.

CHART 7 (Sheet 2 of 2)
TROUBLESHOOTING BATTERY

Trouble	Cause	Remedy
Electrolyte runs out of vent plugs.	Too much water added to battery and charging rate too high.	Drain and keep at proper level. Adjust voltage regulator.
Excessive corrosion inside container.	Spillage from over filling.	Use care in adding water.
	Vent lines leaking or clogged.	Repair or clean.
	Charging rate too high.	Adjust voltage regulator.
Battery freezes.	Discharged battery.	Replace.
	Water added and battery not charged immediately.	Always recharge battery for 1/2 hour following addition of water in freezing weather.
Battery polarity reversed.	Connected backwards on charger.	Battery must be slowly discharged completely and then charged correctly and tested.
Battery consumes excessive water.	Charging rate too high (if in all cells).	Correct charging rate.
	Cracked partition (one cell only).	Replace battery.



LOOKING DOWN
(AFT OF BAGGAGE COMPARTMENT REAR CLOSEOUT PANEL)



Battery Installation
Figure 9

E. Acid Recovery (Vent) Jar

(1) Removal

CAUTION: DO NOT REMOVE EITHER VENT HOSE FROM VENT JAR CAP.

(a) Remove two MS35207-264 screws and AN960-10 washers securing jar to battery bracket assembly.

(b) Carefully position jar so that it can be unscrewed from the jar top.

(c) Remove jar from airplane.

(2) Installation

CAUTION: CHECK BATTERY VENT TUBES LEADING TO AND FROM VENT JAR FOR OBSTRUCTIONS. REMOVE OBSTRUCTIONS TO ENSURE POSITIVE BATTERY BOX VENTILATION. REPLACE ONLY WITH SPECIAL ACID PROOF HOSES SPECIFIED IN PARTS CATALOG.

(a) Carefully position jar top so that jar can be screwed into it.

(b) Attach jar to lid.

(c) Attach jar assembly to battery bracket using two each MS35207-264 screws and AN960-10 washers.

(3) Cleaning

(a) Visually inspect all vent lines for kinks, cracks, flexibility, and loose connections. Replace only with special hoses from parts manual. (DO NOT REPLACE WITH ORDINARY RUBBER HOSE.)

(b) Slowly pour the soda solution into the vent hose, still attached to fuselage floor vent, using a small funnel. The solution will flow out the fuselage vent.

(c) Follow with a final purge of clear water to flush the vent line and then blow dry with low pressure air. This ensures that the vent line is not kinked or restricted and that it is neutralized.

(d) Wipe down the lower fuselage area surrounding the vent with soda solution and clear water. Apply a fresh coat of high quality aircraft wax to entire area.

(e) Unscrew the bottom of the recovery jar and separate from the top. (See Removal, above.) Remove jar pad and empty jar contents into suitable container for safe disposal.

(f) Thoroughly wash and neutralize the jar, pad, top (including bracket), and the short length of vent hose still attached to the jar top with a soda solution and a clear water rinse.

(g) Thoroughly dry all components and recharge the jar with 0.75 of an inch of bicarbonate of soda. Place dry jar pad in the jar on top of the soda charge.

(h) Screw jar back together and keep it in a vertical position.

(i) Install in aircraft per installation instructions, above.

F. Battery Charging - (Gill Model G-243)

CAUTION: IN THE OPERATION OF THE BATTERY, GASES ARE FORMED WHICH MAY BE EXPLOSIVE IF IGNITED. NEVER CREATE SPARKS OF ANY KIND OR BRING AN OPEN FLAME NEAR THE BATTERY. VENTILATE THE BATTERY WHEN CHARGING TO DISPOSE OF THE GAS GENERATED BY THE BATTERY.

The National Electric Code forbids charging batteries installed in aircraft or within 10 feet of fuel tank areas. The battery must be removed from the aircraft for charging. Further, an aircraft battery should not be allowed to deteriorate to the point where safety of flight is jeopardized. The batteries emergency capacity should be sufficient to power the essential bus for a minimum of thirty minutes.

(1) Remove cell plugs and ensure that vents in plugs are open and that vent valves operate freely.

(2) Check that the electrolyte level in each cell is at the bottom of the split ring.

(3) A hydrometer check of each cell should be accomplished. (Refer to Hydrometer Reading and Battery Charge. See “Chart 8” on page 243041.)

CAUTION: WEAR EYE PROTECTION - DON’T TAKE CHANCES WHEN CHARGING. MAKE SURE THE CHARGING AREA IS WELL VENTILATED. IF CENTRAL AIR CONDITIONING IS USED, THE BATTERY CHARGING AREA SHOULD BE VENTED TO THE OUTSIDE AIR TO PREVENT HYDROGEN GASSES FROM BEING CIRCULATED THROUGHOUT THE BUILDING.

(4) It is recommended that vent caps be left on the battery while charging. In addition a wet cloth should be placed over the vent caps within the manifold.

(5) The battery may be charged at any rate, in amperes, not to exceed that point which would produce bubbling and gassing of the electrolyte or a cell temperature of 115° F in any case.

NOTE: If a cell temperature reaches the 115° F. limit, the charging rate shall be reduced and the charge completed at 3 amperes or less. DO NOT CHARGE AT A HIGHER RATE WHEN CELLS ARE GASSING. Refer to Gill Service Manual G.S.M.-682 for alternate charging methods and service procedures.

(6) If a constant current (recommended) charge is available, the charge should be started at 3 amperes and reduced in half if and when cells start gassing until fully charged.

(7) As charging occurs, if any cells sputter or flood, the electrolyte level is too high and the excess must be removed. In any case the electrolyte level shall be adjusted at the end of the charge. The level will rise due to acid returning to the electrolyte mix, normal gassing, and expansion due to temperature rise.

(8) Thoroughly clean battery after charging to prevent remaining acid bridges which can form during charging.

NOTE: Never allow lead acid batteries or tools used on them to be near Ni-Cad batteries and Ni-Cad battery tools.

G. Hydrometer Reading and Battery Charge

See “Chart 8”.

CAUTION: IN THE OPERATION OF THE BATTERY, GASES ARE FORMED WHICH MAY BE EXPLOSIVE IF IGNITED. NEVER CREATE SPARKS OF ANY KIND OR BRING AN OPEN FLAME NEAR THE BATTERY. VENTILATE THE BATTERY WHEN CHARGING TO DISPOSE OF THE GAS GENERATED BY THE BATTERY.

Whenever checking the battery, ascertain that all connections are clean and tight and that the fluid level is above the baffle plates. If necessary to add fluid, fill cell with distilled water to the bottom of the split ring. After adding water, charge the battery until gassing before taking a hydrometer reading. Otherwise, the water and electrolyte will not be mixed, giving a false reading. Temperatures different from the established norm will effect the hydrometer readings. Refer to “Chart 9” for temperature corrections. Specific gravity values for a fully charged battery are as follows:

(1) To adjust low specific gravity, charge the battery (see Charging Battery) until it is gassing and until the specific gravity rises no higher over a 3-hour period. Then remove some electrolyte and replace with 1.300 specific gravity electrolyte. Repeat this step if, after one hour of charging, the specific gravity is still too low. DO NOT ADJUST A CELL THAT DOES NOT GAS.

(2) To adjust high specific gravity, charge the battery (see Charging Battery) until it is gassing and until the specific gravity rises no higher over a 3-hour period. Remove some electrolyte and replace with distilled water. Repeat this step if, after one hour of charging, the specific gravity is still too high.

CHART 8
PERCENT OF CHARGE

Hydrometer Reading	Percent of Charge
1280	100
1250	75
1220	50
1190	25
1160	Very little useful capacity
1130 or below	Discharged

CHART 9
SPECIFIC GRAVITY TEMP. CORRECTION

Electrolyte Temperature		Correction	
°C	°F		
60	140	1.024	
55	130	1.020	
49	120	1.016	Add
43	110	.012	to
38	100	.008	Reading
33	90	.004	
27	80	0.000	--------
23	70	- .004	
15	60	- .008	Subtract
10	50	- .012	from
5	40	- .016	Reading
-2	30	- .020	
-7	20	- .024	
-13	10	- .028	
-18	0	- .032	
-23	-10	- .036	
-28	-20	- .040	
-35	-30	- .044	

H. Battery Discharge

See “Chart 10”.

The capacity of a storage battery is measured in units of ampere hours, which is the product of the electrical current in amperes multiplied by the time in hours. Although current may be obtained after the end of the time, the voltage of the battery has dropped to a point beyond which it is not very useful. The ampere hours which may be obtained from a battery are greater for a long low-rate or intermittent rate discharge than for a short high-rate discharge because the voltage will drop faster at the higher discharge rate. The maximum permissible rate of discharge is limited only by the current-carrying ability of the wiring, motor, or other apparatus to which the battery is connected or by the current-carrying ability of the cell terminals and connectors and not by the plates themselves. Listed below are recommended discharge rates:

I. Battery Temperature Considerations

Operation of storage batteries beyond their ambient temperature or charging voltage limits will result in excessive cell temperatures leading to electrolyte boiling, rapid deterioration of the cell, and finally battery failure. The relationship between the maximum charging voltage and the number of cells in the battery is also significant, since this will determine (for a given ambient temperature and state of charge) the rate at which energy is absorbed as heat within the battery. The maximum voltage per cell should not exceed 2.35 - volts, and the maximum temperature should not exceed 115° F.

Low electrolyte temperatures temporarily reduce the battery capacity and the freezing point depends on the specific gravity. To prevent freeze damage, maintain the specific gravity at a reasonably high level as indicated by “Chart 11”.

NOTE: Lead-acid batteries are subject to a constant discharge due to the internal chemical action.

CHART 10
DISCHARGE RATE

Battery Type	Volts	Ampere Hours (5 Hours)
Gill G-243	24	15.5

CHART 11
ELECTROLYTE FREEZING POINT

Specific Gravity	Freezing Point	
	°C	°F
1.300	-70	-95
1.275	-62	-80
1.250	-52	-62
1.225	-37	-35
1.200	-26	-16
1.175	-20	- 4
1.150	-15	5
1.125	-10	13
1.100	-8	19

J. Battery Repairs, Storage, and Service Tips

The internal parts of the battery have been designed to wear at approximately the same rate, making it uneconomical to replace any of the parts with new ones. Replacing the entire battery is simpler and cheaper.

Before storing the battery, it should be properly charged, the vent plugs put tightly in place, and the leads disconnected to prevent use during idle periods. The battery should be charged at intervals during the idle period. Before returning the battery to service, it should be thoroughly charged. The battery will be sufficiently charged when, after a 3-hour period, the specific gravity does not rise any higher with the electrolyte gassing and a charging rate of 1-1/2 amperes.

Long battery life and trouble-free service is obtained from the battery if the following simple tips are observed:

(1) Keep it clean.
(2) Keep it charged.
(3) Maintain proper electrolyte levels.
(4) Keep specific gravity equal among all cells.

K. Preparing New Dry Charged Battery for Installation

CAUTION: PRIOR TO INSTALLING A NEW DRY-CHARGED BATTERY, FOLLOW THE PREPARATION INSTALLATION INSTRUCTIONS FURNISHED WITH THE BATTERY BY TELEDYNE.

The Teledyne, dry-charged, Model GILL/G-243 Battery shall be stored as received from Teledyne. Do not remove vent seals, add acid, nor attempt to charge a dry-charged battery until the time arrives to install the battery into an airplane.

11. Batteries - Concorde Valve-Regulated Sealed Lead-Acid

See “Figure 10” on page 243046.

Garmin G1000/G1000 NXi equipped airplanes (S/N's 2843701, 2843703 and up) have two Concorde 24 volt batteries; one main battery and one emergency battery.

Standard or Garmin G500 equipped airplanes S/N's 2843866 and up have a single Concorde 24 volt main battery.

Service and maintain these batteries per the appropriate Concorde Component Maintenance Manuals (CMM). See Vendor Publications under Supplementary Publications in the Introduction.

A. Precautions

See precautions at the beginning of this section and in the appropriate Concorde CMM.

B. Troubleshooting

See Fault Isolation Summary in the appropriate Concorde CMM.

C. Inspections

(1) Each 100 Hours or Annually, whichever comes first.

Visually check the battery and battery compartment for general condition, electrolyte venting, and corrosion. As required, clean per Cleaning, below.

(2) Capacity Check (Airplanes operating less than 1,000 hours per year.)

See appropriate Concorde CMM for capacity check procedures.

(a) Initial check at 12 months after initial installation (± 1 month).

(b) As long as the capacity is above 90% at the last check, subsequent capacity checks every 12 months in service (± 1 month).

(c) If the capacity is between 85 and 90% at the last check, subsequent checks every 6 months in service (± 1 month).

(d) If the capacity is less than 85%, remove battery from service.

(3) Capacity Check (Airplanes operating 1,000 hours per year or more.)

See appropriate Concorde CMM for capacity check procedures.

(a) Initial battery capacity check 1,000 hours after initial installation (± 100 hours).

(b) As long as the capacity is above 90% at the last check, subsequent checks every 1,000 hours in service (± 100 hours).

(c) If the capacity is between 85 and 90% at the last check, subsequent checks every 500 hours in service (± 100 hours).

(d) If battery capacity is less than 85%, remove battery from service.

D. Cleaning

CAUTION: DO NOT USE ANY TYPE OF SOLVENT TO CLEAN THE BATTERY. SOLVENTS MAY DAMAGE THE BATTERY.

(1) Clean the outside surfaces of the battery with a lint-free shop cloth that is clean, dry, and free of oil.

(2) If the battery has caked-on dirt or grime, use a cloth dampened with tap water, then wipe dry.

(3) Wipe down the battery support and floor area, hold-down supports, connectors and cable ends.

(4) If there is any evidence of electrolyte venting / corrosion clean with a cloth saturated with a solution of baking soda and water. Rinse with a clean cloth wet with plain water. Dry thoroughly. Touchup paint as required.

E. Main - Model RG24-11M

See “Figure 10” on page 243046.

The main battery has a rated capacity of 11.0 ampere hours, 160 cold cranking amps and is mounted aft of the baggage compartment rear closeout panel on a shelf along the right side of the airplane. When installed with an emergency battery, the main battery is the aft battery.

(1) Removal

(a) Remove the aft baggage compartment rear closeout panel.

CAUTION: REMOVE NEGATIVE CABLE FIRST, BEFORE EXPOSING AND LOOSENING POSITIVE TERMINAL NUT, TO AVOID SHORT CIRCUITS AND ELECTRICAL BURNS.

(b) Remove nut and disconnect negative (ground) battery cable.

(c) Pull back rubber battery terminal boot, remove nut and disconnect positive battery cable.

WARNING: BATTERIES ARE HEAVY. USE APPROPRIATE LIFTING DEVICES OR EQUIPMENT. USE BATTERY HANDLES WHERE PROVIDED.

(d) Remove battery hold down bolts. Lift battery tie-down off battery and place away from carpeting, etc.

(e) Lift battery out of airplane.

(2) Installation

WARNING: BATTERIES ARE HEAVY. USE APPROPRIATE LIFTING DEVICES OR EQUIPMENT. USE BATTERY HANDLES WHERE PROVIDED.

(a) Place the battery into position on its shelf.

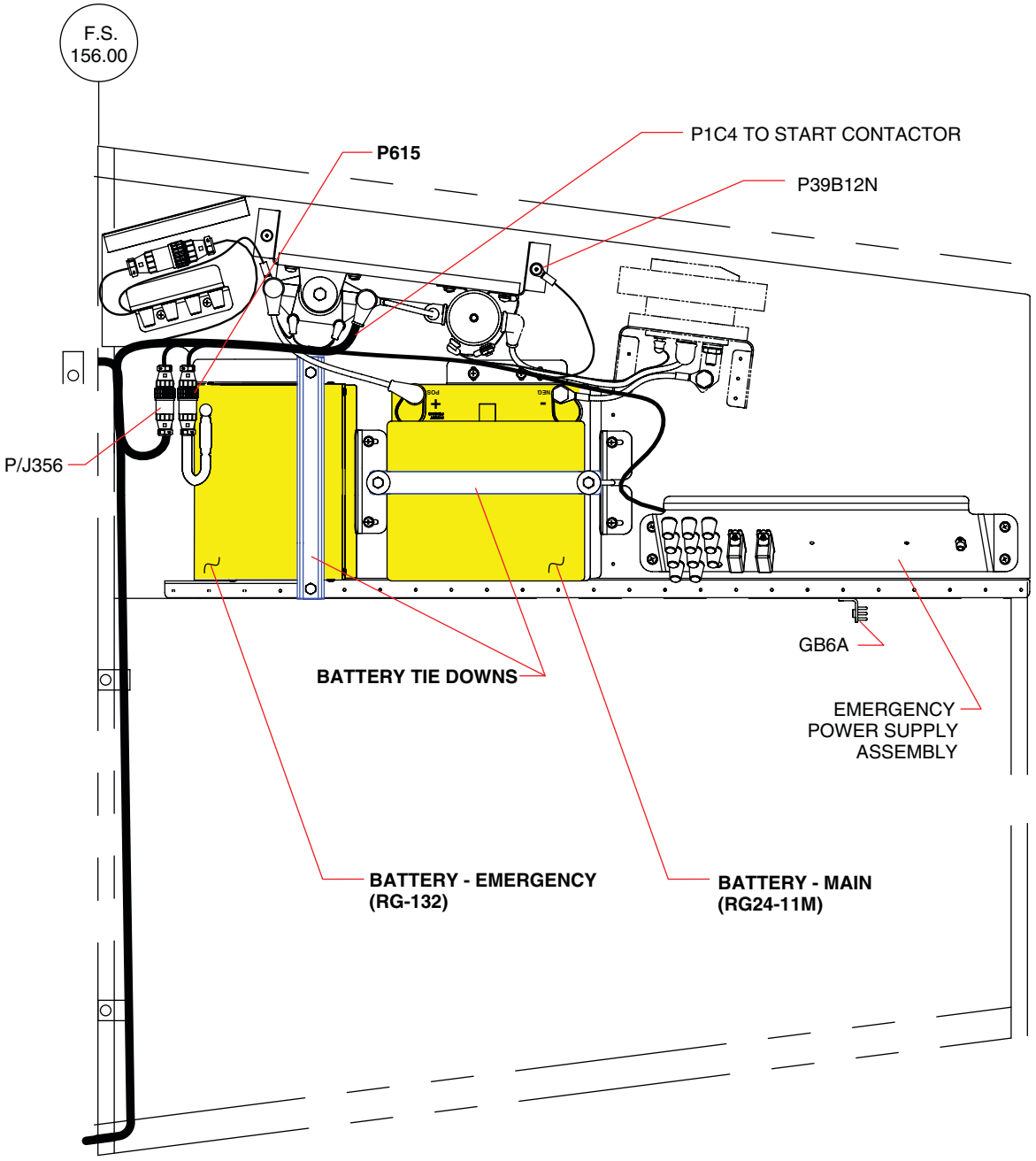
(b) Place battery tie-down on battery and install battery hold-down bolts.

(c) Connect positive battery cable, tighten nut and push rubber battery terminal boot back on.

(d) Connect negative (ground) cable and tighten nut.

(e) Install the aft baggage compartment rear closeout panel.

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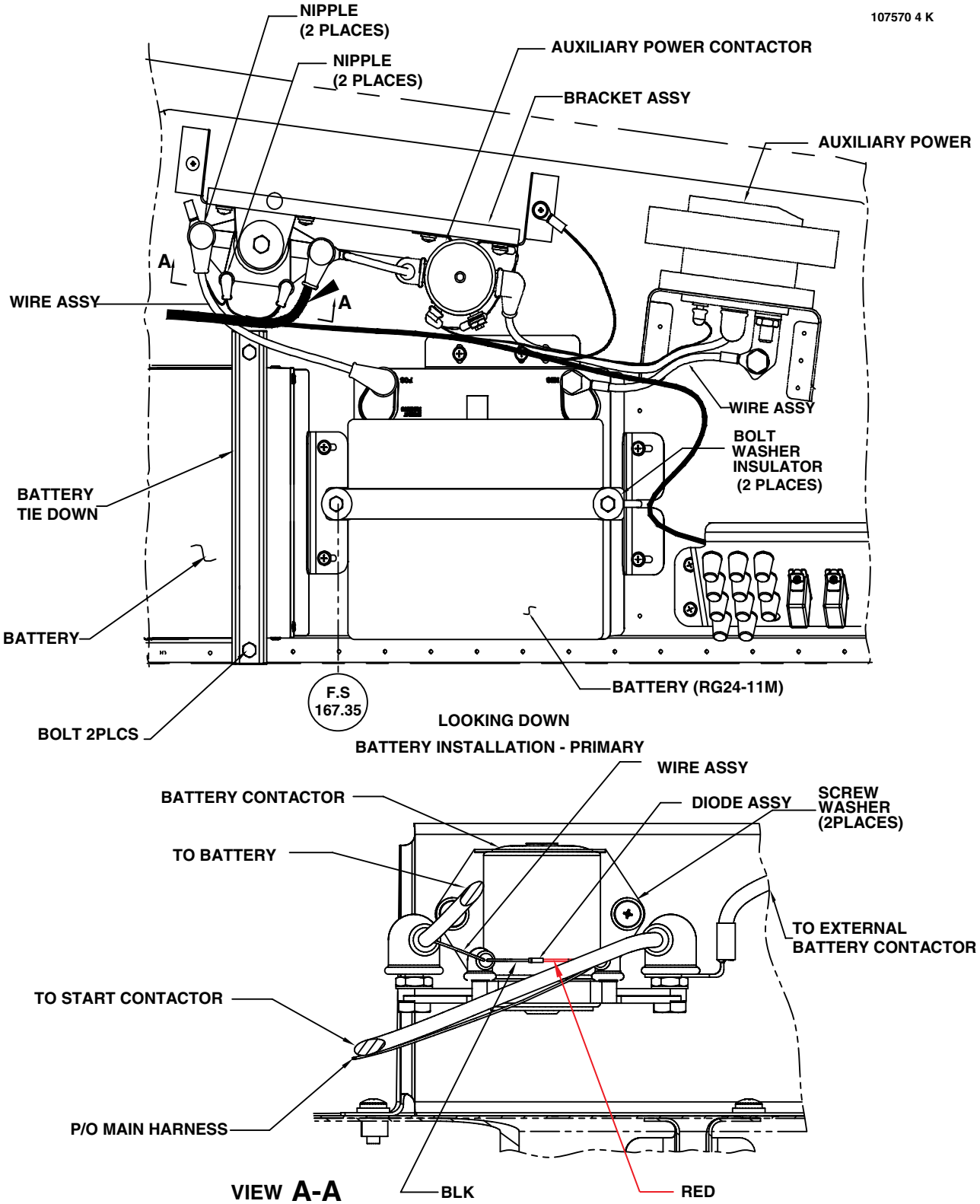


LOOKING DOWN
(AFT OF BAGGAGE COMPARTMENT REAR CLOSEOUT PANEL)

Effectivity
with Garmin G1000/G1000 NXi

Battery Installation - Concorde
Figure 10 (Sheet 1 of 2)

Effectivity
with Concorde Battery(ies)



Battery Installation - Concorde
Figure 10 (Sheet 2 of 2)

Effectivity
Standard or Garmin G500
2843866 and up

F. Emergency - Model RG-132

NOTE: Charging these batteries requires a locally-made charging harness. See the appropriate Concorde Component Maintenance Manual (CMM) per Vendor Publications under Supplementary Publications in the Introduction. The charging harness requires a special connector, AMP 206153-1, four terminals AMP 66098-6, and a clamp AMP 206062-1 which can be procured locally or ordered from Piper as P/N's 555-806, 588-792 (4 ea.), and 554-612, respectively.

When installed, the emergency battery has a rated capacity of 6.0 ampere hours and is mounted with the main battery aft of the baggage compartment rear closeout panel on the right side of the airplane. The emergency battery is the forward battery.

(1) Removal

See "Figure 10" on page 243046.

(a) Remove the aft baggage compartment rear closeout panel.

(b) Disconnect the connector to the emergency power supply assembly.

WARNING: BATTERIES ARE HEAVY. USE APPROPRIATE LIFTING DEVICES OR EQUIPMENT. USE BATTERY HANDLES WHERE PROVIDED.

(c) Remove battery hold down bolts. Lift battery tie-down off battery and place away from carpeting, etc.

(d) Lift battery out of airplane.

(2) Installation

WARNING: BATTERIES ARE HEAVY. USE APPROPRIATE LIFTING DEVICES OR EQUIPMENT. USE BATTERY HANDLES WHERE PROVIDED.

(a) Place the battery into position on its shelf.

(b) Place battery tie-down on battery and install battery hold-down bolts.

(c) Connect the connector to the emergency power supply assembly.

(d) Install the aft baggage compartment rear closeout panel.

12. Checking Voltage Regulator

The regulator is a fully transistorized unit. The components are encapsulated in epoxy, making field repair of the unit impractical. Before performing any of the following checks on regulator, verify proper functioning of the alternator per manufacturer's instructions. If it does not meet the specifications, the alternator must be replaced.

A. Use only a good quality, adjustable dc power source.

CAUTION: ALL TESTS MUST BE ACCOMPLISHED WITH THE REGULATOR OUT OF THE CIRCUIT.

B. Use a quality, accurate voltmeters with at least a 35-volt scale.

CAUTION: AMBIENT TEMPERATURES SURROUNDING VOLTAGE REGULATOR MUST BE BETWEEN 50°F AND 100°F.

(1) Voltmeter connections.

Refer to "Figure 11" on page 243050.

(a) Connect positive voltmeter lead to red wire (supply) at regulator harness connector or wire .

(b) Connect negative voltmeter lead to regulator ground wire (black).

(2) Light Bulb Connections.

Refer to "Figure 11" on page 243050.

(a) Connect one bulb lead to blue wire (field) at regulator harness connector or wire.

(b) Connect other bulb lead to regulator ground wire (black).

(3) In the regulation check procedure, increase voltage to regulator and monitor both the voltmeter and bulb.

(4) As the regulation point of a properly functioning control unit is approached, the bulb will blink off and on. At regulation, the bulb will be ON continuously.

(5) If regulator does not regulate to 28 Volts, it is out of calibration, and must be replaced.

C. If regulator checks good, check airplane for:

(1) Poor or loose connections.

(2) Poor ground on regulator housing.

(3) Shorted alternator windings.

(4) A grounded wire.

D. After completing test, connect regulator into circuit.

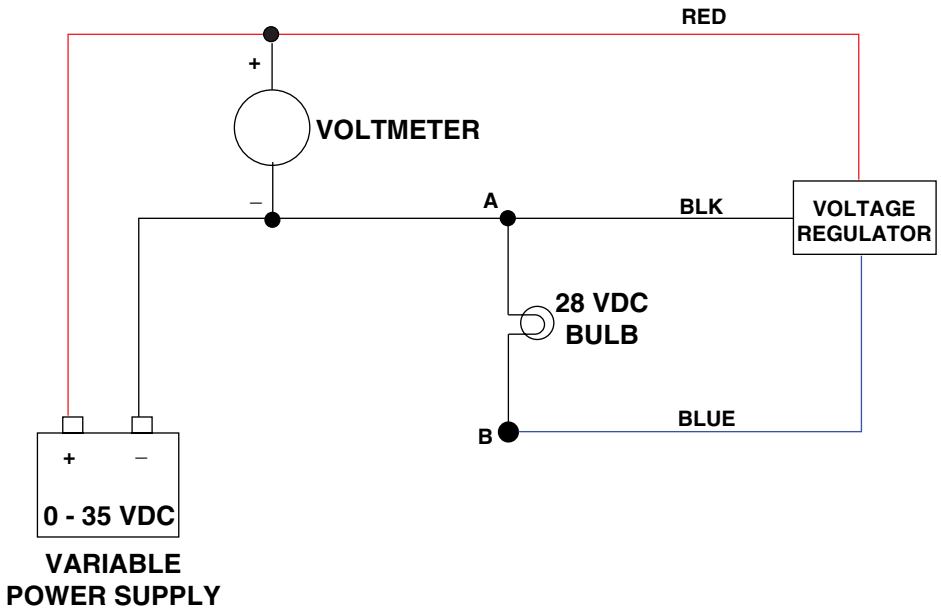
E. 28 Vdc System Regulator Overvoltage Check

The Lamar regulator has an incorporated overvoltage relay. To check relay operation use the same test connections as testing the regulator, except connect the bulb across the RED and BLUE wires. (Refer to "Figure 12" on page 243050.) Test as follows:

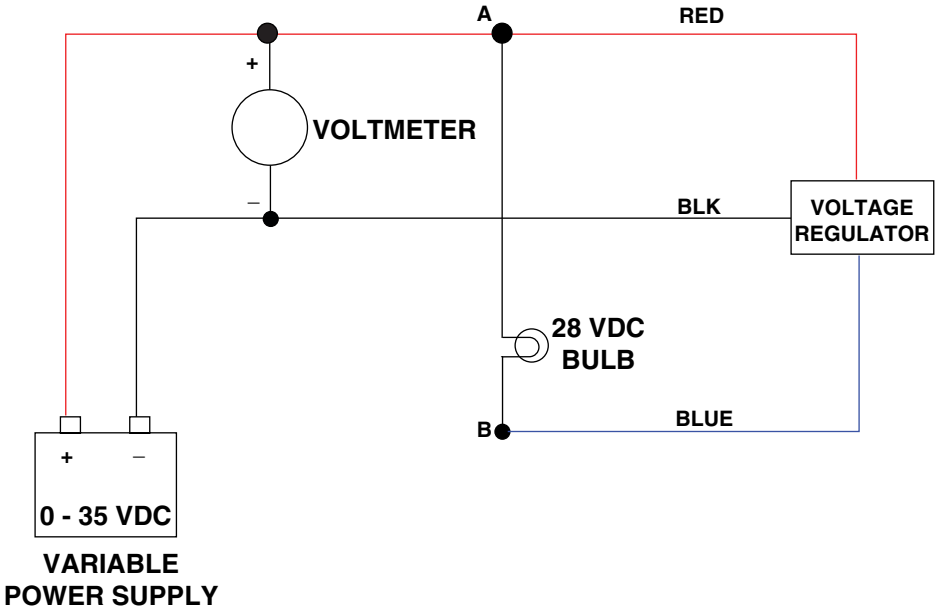
(1) While monitoring both the voltmeter and the light bulb, increase voltage to circuit slowly to 32 volts.

(2) After a slight delay, the bulb will light.

(3) If overvoltage control fails to operate at 32 Vdc, it must be replaced.



Lamar 28 Vdc Regulator Check
Figure 11



28 Vdc System Lamar Overvoltage Check
Figure 12

EXTERNAL POWER

1. Description

The external power receptacle is located near the battery on the right side aft fuselage, aft of the baggage compartment door.

2. Operation

A. Ground Power

CAUTION: WHEN USING AUXILIARY POWER UNIT, ENSURE THE VOLTAGE AND POLARITY ARE SET TO CORRESPOND WITH THE AIRCRAFT SYSTEM VOLTAGE AND POLARITY.

(1) Set the ground power unit to the proper voltage and polarity or, if using a single voltage ground power unit (or battery) ensure it is the proper voltage and polarity.

NOTE: If using a battery, connect the RED lead of jumper cable to the POSITIVE (+) terminal of external 24 volt battery, as appropriate, and the BLACK lead goes to the NEGATIVE (-) terminal.

(2) Turn aircraft Master Switch and all electrical equipment OFF.

(3) Insert the plug of the jumper cable into the socket located per Description, above.

(4) When using external power for operation of any of the airplane's equipment, the Master Switch must be ON.

B. Starting

Proceed as follows when using external power for engine start or operation of any of the airplane's equipment:

(1) Turn the master switch OFF.

(2) Turn all electrical equipment OFF.

(3) Insert plug of a standard 28Vdc power source into the external power socket in the fuselage. Note that, after inserting plug, the electrical system is ON.

(4) Proceed with the normal engine starting technique.

NOTE: Starter manufacturers recommend limiting cranking to thirty second periods with a two minute rest between periods. Lengthy cranking will shorten the starter's life.

C. After engine start:

(1) Reduce power to the lowest possible rpm to reduce sparking when disconnecting jumper cable.

(2) Disconnect the jumper cable from the aircraft.

(3) Turn the master and alternator switches ON and check the alternator ammeter. DO NOT ATTEMPT FLIGHT IF THERE IS NO INDICATION OF ALTERNATOR OUTPUT. If aircraft battery is weak, charging current will be high. DO NOT TAKE OFF until charging current falls below 20 amps.

NOTE: For all normal operations using jumper cables, the master switch should be OFF. The aircraft battery can be used in parallel with the external battery by turning the master switch ON. This will give longer cranking capabilities, but will not increase the amperage. Exercise care, because if the aircraft battery has been depleted, the external power supply can be drawn down to the level of the aircraft battery. This can be tested by turning the master switch ON momentarily while the starter is engaged. If cranking speed increases, the aircraft battery is at a higher level than the external power supply.

(4) When the engine is firing evenly, advance throttle to 800 RPM. If oil pressure is not indicated within thirty seconds, stop engine and determine trouble. It will take a few seconds longer in cold weather to get an oil pressure indication. If the engine fails to start, refer to the Lycoming Operating Handbook: Engine Troubles and Their Remedies.

ELECTRICAL LOAD DISTRIBUTION

1. Electrical Load Analysis (ELA)

The following charts are provided to aid in determining available electrical capacity when adding or removing equipment over the life of the airplane.

A. Standard Electro-Mechanical Instruments

(PIR-VB-712, Rev. 5.)

S/N's 2843001 and up.

See "Chart 1" on page 24603.

B. Avidyne Entegra equipped

((PIR-VB-712, Rev. 8.)

S/N's 2843598 thru 2843682.

See "Chart 2" on page 24604.

C. Garmin G1000 equipped

(PIR-VB-2280, Original.)

S/N's 2843701, 2843703–2843822, 2843824–2843851 & 2843853.

See "Chart 3" on page 24606.

D. Garmin G1000 NXi equipped

(1) S/N's 2843852, 2843854–2843999.

(PIR-VB-2591, Original.)

See "Chart 4" on page 24608.

(2) S/N's 2843823, 2881001 and up.

Information pending.

2. USB Charging Ports (Optional)

Available in S/N's 2843823, 2843852, 2843854 and up.

When installed, two dual USB charging ports are provided. One is installed in each side the instrument panel.

A. Description

The Dual USB Charging Ports convert 28 VDC ship's current to standard 5V power for any electronic product that charges using a USB connector. Two Universal Serial Bus-A (USB-A) ports are at each location and are designed as DCPs (Dedicated Charging Ports) to industry-standard protocol per the USB Battery Charging 1.2 Compliance Plan. Both ports can provide up to 2.1 amps of power simultaneously to charge any USB device.

B. Operation

(1) Electrical Performance

The Dual USB Charging Port converts an aircraft (DC) input voltage within the range specified to a 5V (DC) output. This output power is applied to a dual USB-A connector in accordance with the USB Implementers Forum.

The USB D+ and D- data lines communicate with the USB portable device to tell the device it is a dedicated charging port (DCP), capable of a higher current than a standard USB port. This allows the USB portable device to draw up to 2.1 Amps.

The unit is designed as a DC-to-DC converter with a series switch on each output to regulate current applied to that output. Each series switch independently reduces the output current to a safe level if the USB portable device draws excess current, is shorted or has a fault.

If the temperature of the unit becomes elevated due to a fault or excessive load, the device will seamlessly communicate with the USB portable device to lower the charge current. This allows the device to continue charging while the unit returns to a temperature within designed limits. When the temperature returns to a safe level the unit will automatically reestablish the higher charge current level with the device and continue charging.

(2) Protective Features

(a) Short Circuit Protection

The unit is capable of surviving a short circuit event without permanent damage. The unit goes into an over-current condition so that the average current is significantly reduced and the device is protected.

(b) Over-Current Protection

The unit monitors the current draw individually on each port. During an over-current condition the voltage is reduced. If the voltage falls below 3.8 VDC the output is turned off for a period of 12 seconds. The output is then checked for continued over-current conditions every 16 milliseconds. This condition is referred to as a hiccup mode. The device stays in this mode until the over-current condition is removed, then returns to normal operation.

(c) Low Input Voltage Shutdown

If the input voltage applied to the unit drops below 10 VDC the unit will shut down until the applied voltage returns to a level within range.

(d) Over-Temperature

When the temperature of the unit becomes elevated, the unit communicates with the USB portable device to reduce the charge current output (1 amp limit). When the temperature returns to an acceptable level the unit automatically returns to a higher charge current as required (up to 2.1 amps).

C. Troubleshooting

Troubleshooting the Dual USB Charging Ports is limited to verifying correct harness wiring: 10–28 VDC at Pin A and a good ground at Pin B; and that the harness is then connected to the unit. If the harness has been verified and is properly connected and energized and the unit fails to function properly - remove and replace the unit.

D. Removal

Each unit is face-mounted to the instrument panel. Both units are removed and installed by the same procedure.

(1) Disconnect the negative (ground) battery cable.

(2) From behind the instrument panel, disconnect the electrical harness from the USB unit.

(3) At the front of the instrument panel, remove the two screws securing the USB unit to the panel and remove the unit.

E. Installation

(1) Position charging port unit behind the instrument panel and secure with two screws.

(2) Reconnect wiring harness to the USB unit.

(3) Reconnect the negative (ground) battery cable.

(4) Turn ON the master switch.

(5) Connect a USB device to each charging port and verify charging.

(a) If charging occurs at each port, install is complete.

(b) If charging does not occur at one or both ports, troubleshoot per”C. Troubleshooting”, above.

CHART 1
ELECTRICAL LOAD ANALYSIS

Component / System	Amps/ Unit	Start 0.5 Min	Taxi (Night) 30 Min	Take Off & Climb 20 Min	Cruise (Night) 60 Min	Land (Night) 10 Min
Starter	175	87.5				
Starter Contactor	0.74	0.37				
Battery Master	0.50	0.25	15.0	10.0	30.0	5.0
Alternator Field	2.30	1.15	69.0	46.0	138.0	23.0
Electric Turn & Bank	0.30	0.15	9.0	6.0	18.0	3.0
Pitot Heat	6.0		180.0	120.0	360.0	60.0
Engine Gauge	0.20	0.10	6.0	4.0	12.0	2.0
Auxilliary Fuel Pump	5.0	2.5		100.0		50.0
Primer Valve	0.40	0.20				
Radio Master	0.70	0.35				
Instrument Panel Lights	0.80	0.40	24.0	16.0	48.0	8.0
Switch Lights	0.80	0.40	24.0	16.0	48.0	8.0
Position Lights	4.0		120.0	80.0	240.0	60.0
Strobe Lights	3.5			70.0	210.0	35.0
Cockpit Flood Light	0.60		18.0	12.0	36.0	6.0
Cabin Lights	0.90				50.0	
Landing/Taxi Lights	3.6 (7.1)		108.0			7.1
Electric Clock	Negligible					
Radio Lights	0.50		15.0	10.0	30.0	5.0
Avionics Blower	0.50	0.25	15.0	10.0	30.0	5.0
Cigar Lighter	4.0					
Auxiliary Vacuum Pump	11.0					
Electrical Total		118.4	603.0	500.0	1250.0	341.0
KMA-24 Audio Panel	1.6		48.0	32.0	96.0	16.0
KX-165 COM/NAV 1	0.4 / 6.0		12.0	8.0	24.0	4.0
KX-165 COM/NAV 2	0.4 / 6.0		12.0	8.0	24.0	4.0
KT-76A Transponder	1.1 / 1.8		33.0	22.0	66.0	11.0
KR-87 ADF	1.0		30.0	20.0	60.0	10.0
KI-206 VOR/GS Indicator	0.10		3.0	2.0	6.0	1.0
KI-206 VOR/LOC Indicator	0.10		3.0	2.0	6.0	1.0
AR-850 Altitude Reporter	0.4 / 1.0		12.0	8.0	24.0	4.0
KN62A DME	0.53		16.0	10.6	32.0	5.3
KAP-100 Autopilot	3.7		111.0	74.0	222.0	
Avionics Total			280.0	186.6	560.0	56.3
System Total (without A/C)			883.0	686.6	1810.0	397.3
Air Conditioning	9.5		285.0		570.0	
Electrical			603.0	500.0	1250.0	341.0
Avionics			280.0	186.6	560.0	56.3
System Total (with A/C)			1168.0	686.6	2380.0	397.3

CHART 2 (Sheet 1 of 2)
ELECTRICAL LOAD ANALYSIS

Equipment	Current Draw	Start 0.5 Min	Taxi 10 Min	T/O Climb 15 Min	Cruise 60 Min	Landing 10 Min	Notes
Starter	175.00	87.50					
Starter Contactor	0.74	0.37					
Battery Master Contactor	0.50	0.25	5.00	7.50	30.00	5.00	
Alternator Field	2.30		23.00	34.50	138.00	23.00	
Turn Rate Indicator	0.30	0.15	3.00	4.50	18.00	3.00	
Aux Fuel Pump	5.00	2.50		25.00		25.00	5 min during T/O & Landing
Primer Valve	0.40	0.20					
Annunciator Light Assembly	0.05	0.03	0.50	0.75	3.00	0.50	0.96A @ 5% duty
Pilot Heat	6.00						
Instrument Panel Lights	0.56	0.28	5.60	8.40	33.60	5.60	7 lights @ 0.08A
Switch Lights	1.56	0.78	15.60	23.40	93.60	15.60	26 lights @ 0.6A (50% dim)
Avionics Lights	0.50		5.00	7.50	30.00	5.00	
Cockpit Flood Lights	0.60	0.30					
Strobe (Anti-Collision) Lights	3.50		35.00	52.50	210.00	35.00	
Cabin (Dome) Lights	0.90					9.00	
Landing/Taxi Light	7.10		71.00			71.00	
Position Lights	4.00	2.00		60.00	240.00	40.00	
Avionics Master Switch	0.70	0.35					
Avionics Blower	0.50	0.25	5.00	7.50	30.00	5.00	
Air Conditioning	10.00		100.00	150.00	600.00		
Audio Panel (GMA 340)	2.20		22.00	33.00	132.00	22.00	
COM/NAV/GPS GNS 430 #1	1.51		15.10	22.65	90.60	15.10	
COM/NAV/GPS GNS 430 #1 Tx (7.51A @ 5% duty)	0.38		3.76	5.63	22.53	3.76	5% Tx duty
COM/NAV/GPS GNS 430 #2	1.67		16.70	25.05	100.20	16.70	includes DC50
GTX 330 Transponder	0.79		7.90	11.85	47.40	7.90	
DME-450	1.20		12.00	18.00	72.00	12.00	
Altitude Encoder (AK 350)	0.18		1.80	2.70	10.80	1.80	
System 55X Autopilot	2.50		25.00	37.50	150.00	25.00	
Trim Servo (0.5A @ 5% duty)	0.03		0.25	0.38	1.50	0.25	5% duty
Stormscope (WX-500)	0.38		3.80	5.70	22.80	3.80	
Skywatch (SKY-497)	2.50		25.00	37.50	150.00	25.00	

CHART 2 (Sheet 2 of 2)
ELECTRICAL LOAD ANALYSIS

Equipment	Current Draw	Start 0.5 Min	Taxi 10 Min	T/O Climb 15 Min	Cruise 60 Min	Landing 10 Min	Notes
MFD	1.70	0.85	17.00	25.50	102.00	17.00	includes dalalink Rx
MFD Datalink Tx (2A @ 5% duty)	0.10	0.05	1.00	1.50	6.00	1.00	5% Tx duty
PFD	3.00	1.50	30.00	45.00	180.00	30.00	
PFD Autopilot Relay	0.32	0.16	3.20	4.80	19.20	3.20	
DAU	0.13	0.07	1.30	1.95	7.80	1.30	
Energy Demand (amp*min)		97.6	454.5	660.3	2541.0	428.5	
Amp Demand (amps)		195.2	45.5	44.0	42.4	42.9	43.8 amps (avg.)
Alternator Output (amps)		0	70	70	70	70	
Alternator Energy (amp*min)		0	700	1050	4200	700	
Total Energy (amp*min)		-97.6	245.5	389.7	1659.0	271.5	Complete flight fully charged

CHART 3 (Sheet 1 of 2)
ELECTRICAL LOAD ANALYSIS

ITEM	# UNITS	AMPS/ UNIT	DURATION	TIME START	TAXI	T/O	CRUISE	LAND	REMARKS	AMPS * MINUTES START	TAXI	T/O	CRUISE	LAND
Starter	1	175	INT	0.25	0	0	0	0	15 second crank time	43.75	0	0	0	0
Starter Contactor	1	0.74	INT	0.25	0	0	0	0	Same time as starter engaged	0.185	0	0	0	0
Primary Battery Contactor	1	0.5	CONT	1	15	15	60	10		0.5	7.5	7.5	30	5
Emergency Power Contactor	1	0.25	CONT	0	0	0	0	0		0	0	0	0	0
ALT Field	1	3.2	CONT	0	15	15	60	10		0	48	48	192	32
Avionics Contactor	1	0.7	INT	1	0	0	0	0	Denergized when avionics “ON”	0.7	0	0	0	0
Flight Hour Meter (Hobbs)	1	0.05	CONT	1	15	15	60	10		0.05	0.75	0.75	3	0.5
Auxiliary Fuel Pump	1	5	INT	1	0	3	0	3	T/O, Landing and Emergency Use	5	0	15	0	15
Primer Valve	1	0.4	INT	1	0	3	0	3	T/O, Landing and Emergency Use	0.4	0	1.2	0	1.2
Strobe Lights	1	3.5	CONT	0	0	15	60	10		0	0	52.5	210	35
Position Lights	2	0.25	CONT	0	15	15	60	10		0	7.5	7.5	30	5
Landing Lights	2	1.8	INT	0	0	5	0	5		0	0	18	0	18
Instrument Panel Lights	1	0.05	CONT	0	15	15	60	10		0	0.75	0.75	3	0.5
Switch Lights	1	0.8	CONT	1	15	15	60	10		0.8	12	12	48	8
Avionics Lights	1	0.25	CONT	1	15	15	60	10		0.25	3.75	3.75	15	2.5
Cabin Dome (Map) Light	1	0.3	INT	0	0	0	60	0	50% Utilization	0	0	0	18	0
Pitot Heat	1	6	CONT	0	15	15	60	10		0	90	90	360	60
Vent Fan (A/C on Hi speed)	1	3.7	INT	0	15	7.5	0	5	Not typically required in cruise flight	0	55.5	27.75	0	18.5
Avionics Cooling Fans	3	0.4	INT	0.5	8	8	30	5	worst case duty cycle 50%	0.6	9.6	9.6	36	6
STBY Instrument (ASPEN)	1	1.5	CONT	1	15	15	60	10	worst case temp with heaters ON	1.5	22.5	22.5	90	15
STBY Instrument (ASPEN)	1	0.9	INT	1	5	5	20	3	worst case temp with heaters ON, 30% duty cycle	0.9	4.5	4.5	18	2.7
GDU 1040	2	1.3	CONT	1	15	15	60	10		2.6	39	39	156	26
GDU 1040	2	1.2	INT	1	15	2	5	1	worst case temp with heaters on	2.4	36	4.8	12	2.4
GDC 74A	1	0.235	CONT	1	15	15	60	10		0.235	3.525	3.525	14.1	2.35
GRS 77	1	0.3	CONT	0	15	15	60	10		0	4.5	4.5	18	3
GRS 77 (Initial)	1	1	INT	1	0	0	0	0	Peak occurs momentarily at start-up	1	0	0	0	0
GMA 1347	1	1.75	CONT	0	15	15	60	10		0	26.25	26.25	105	17.5
GIA 63W (INTEG AV)	2	1.25	CONT	0	15	15	60	10	includes superflag output draw from WOW relay	0	37.5	37.5	150	25
GIA 63W (COM)	2	0.3	CONT	1	15	15	60	10		0.6	9	9	36	6
GIA 63W (COM)	2	4	INT	0	2	2	6	1	10% duty cycle transmit	0	16	16	48	8
GEA	1	0.15	CONT	1	15	15	60	10		0.15	2.25	2.25	9	1.5
GEA	1	0.17	INT	1	5	5	20	3	estimated to be 1/3 of full potential tranducer load	0.17	0.85	0.85	3.4	0.51
S-TEC Autopilot	1	0.9	INT	0	15	15	60	10		0	13.5	13.5	54	9
Turn Coordinator	1	0.3	CONT	1	15	15	60	10	Autopilot sensor only	0.3	4.5	4.5	18	3
GTS 800 TAS	1	1.3	CONT	0	15	15	60	10	1.5 Max op, 1.3 normal op	0	19.5	19.5	78	13
GTX 33 Transponder	1	0.85	CONT	0	15	15	60	10	maximum quiescent	0	12.75	12.75	51	8.5
GTX 33 Transponder	1	0.81	INT	0	2	2	6	1	maximum full TSO reply rate	0	1.62	1.62	4.86	0.81
KR-87 ADF	1	0.625	CONT	0	15	15	60	10		0	9.375	9.375	37.5	6.25
KN-63 DME	1	0.61	CONT	0	15	15	60	10		0	9.15	9.15	36.6	6.1

CHART 3 (Sheet 2 of 2)
ELECTRICAL LOAD ANALYSIS

ITEM	# UNITS	AMPS/ UNIT	DURATION	TIME START	TIME TAXI	TIME T/O	TIME CRUISE	TIME LAND	REMARKS	AMPS * MINUTES START	AMPS * MINUTES TAXI	AMPS * MINUTES T/O	AMPS * MINUTES CRUISE	AMPS * MINUTES LAND
GDL 69A XM Receiver	1	0.35	CONT	0	15	15	60	10		0	5.25	5.25	21	3.5
GRT 10 XM Remote	1	0.044	CONT	0	15	15	60	10		0	0.66	0.66	2.64	0.44
									TOTAL AMP * MINUTES	62.09	512.87	540.62	1905.46	367.32
									UTILIZATION LOAD (AMPS)	62.06	34.2	36.0	31.76	36.73

CHART 4 (Sheet 1 of 2)
ELECTRICAL LOAD ANALYSIS

ITEM	# UNITS	AMPS/ UNIT	DURATION	TIME START	TAXI	T/O	CRUISE	LAND	REMARKS	AMPS * MINUTES START	TAXI	T/O	CRUISE	LAND
Starter	1	175	INT	0.25	0	0	0	0	15 second crank time	43.75	0	0	0	0
Starter Contactor	1	0.74	INT	0.25	0	0	0	0	Same time as starter engaged	0.185	0	0	0	0
Primary Battery Contactor	1	0.5	CONT	1	15	15	60	10		0.5	7.5	7.5	30	5
Emergency Power Contactor	1	0.25	CONT	0	0	0	0	0		0	0	0	0	0
ALT Field	1	2.43	CONT	0	15	15	60	10		0	36.45	36.45	145.8	24.3
Avionics Contactor	1	0.7	INT	1	0	0	0	0	Denergized when avionics “ON”	0.7	0	0	0	0
Flight Hour Meter (Hobbs)	1	0.05	CONT	1	15	15	60	10		0.05	0.75	0.75	3	0.5
Auxiliary Fuel Pump	1	5	INT	1	0	3	0	3	T/O, Landing and Emergency Use	5	0	15	0	15
Primer Valve	1	0.4	INT	1	0	3	0	3	T/O, Landing and Emergency Use	0.4	0	1.2	0	1.2
Tail Strobe Light	1	0.11	CONT	0	0	15	60	10		0	0	1.65	6.6	1.1
Strobe Lights	2	0.5	CONT	0	0	15	60	10		0	0	15	60	10
Position Lights	2	0.17	CONT	0	15	15	60	10		0	5.1	5.1	20.4	3.4
Landing Lights	2	0.6	INT	0	0	5	0	5		0	0	6	0	6
Instrument Panel Lights	1	0.05	CONT	0	15	15	60	10		0	0.75	0.75	3	0.5
Switch Lights	1	0.8	CONT	1	15	15	60	10		0.8	12	12	48	8
Avionics Lights	1	0.25	CONT	1	15	15	60	10		0.25	3.75	3.75	15	2.5
Cabin Dome (Map) Light	1	0.3	INT	0	0	0	60	0	50% Utilization	0	0	0	18	0
Pitot Heat	1	6	CONT	0	15	15	60	10		0	90	90	360	60
Vent Fan (A/C on Hi speed)	1	3.7	INT	0	15	7.5	0	5	Not typically required in cruise flight	0	55.5	27.75	0	18.5
Avionics Cooling Fans	3	0.4	INT	0.5	8	8	30	5	worst case duty cycle 50%	0.6	9.6	9.6	36	6
STBY Instrument (ASPEN)	1	1.5	CONT	1	15	15	60	10	worst case temp with heaters ON	1.5	22.5	22.5	90	15
STBY Instrument (ASPEN)	1	0.9	INT	1	5	5	20	3	worst case temp with heaters ON, 30% duty cycle	0.9	4.5	4.5	18	2.7
GDU 1050	2	0.9	CONT	1	15	15	60	10		0	27	27	108	18
GDU 1050	2	1.0	INT	1	15	2	5	1	worst case temp with heaters on	2	30	4	10	2
GSU 75 ADAHRS	1	0.38	CONT	1	15	15	60	10		0	5.7	5.7	22.8	3.8
GMA 1347	1	1.75	CONT	0	15	15	60	10		0	26.25	26.25	105	17.5
GIA 63W (INTEG AV)	2	0.3	CONT	0	15	15	60	10	includes superflag output draw from WOW relay	0	9	9	36	6
GIA 63W (COM)	2	0.3	CONT	1	15	15	60	10		0.6	9	9	36	6
GIA 63W (COM)	2	4	INT	0	2	2	6	1	10% duty cycle transmit	0	16	16	48	8
GEA	1	0.5	CONT	1	15	15	60	10		0.5	7.5	7.5	30	5
GEA	1	0.17	INT	1	5	5	20	3	estimated to be 1/3 of full potential tranducer load	0.17	0.85	0.85	3.4	0.51
S-TEC Autopilot	1	0.9	INT	0	15	15	60	10		0	13.5	13.5	54	9
Turn Coordinator	1	0.3	CONT	1	15	15	60	10	Autopilot sensor only	0.3	4.5	4.5	18	3
GTS 800 TAS	1	1.5	CONT	0	15	15	60	10	1.5 Max op, 1.3 normal op	0	22.5	22.5	90	15
GTX 345R Transponder	1	0.36	CONT	0	15	15	60	10	maximum quiescent	0	5.4	5.4	21.6	3.6
GTX 345R Transponder	1	0.29	INT	0	2	2	6	1	maximum full TSO reply rate	0	0.58	0.58	1.74	0.29
KR-87 ADF	1	0.45	CONT	0	15	15	60	10		0	6.75	6.75	27	4.5
KN-63 DME	1	0.61	CONT	0	15	15	60	10		0	9.15	9.15	36.6	6.1
GDL 59	1	0.572	CONT	0	15	15	60	10		0	8.58	8.58	34.22	5.72

CHART 4 (Sheet 2 of 2)
ELECTRICAL LOAD ANALYSIS

ITEM	# UNITS	AMPS/ UNIT	DURATION	TIME START	TAXI	T/O	CRUISE	LAND	REMARKS	AMPS * MINUTES START	TAXI	T/O	CRUISE	LAND
GRT 10 XM Remote	1	0.044	CONT	0	15	15	60	10		0	0.66	0.66	2.64	0.44
GDL 69eA XM Receiver	1	0.36	CONT	0	15	15	60	10		0	5.4	5.4	21.6	3.6
FS-510	1	0	CONT	0	15	15	60	10	included in GDU 1050	0	0	0	0	0
ELT 1000	1	0.17	CONT	1	15	15	60	10		0.17	2.55	2.55	10.2	1.7
Appareo Vision 1000	1	0.15	CONT	1	15	15	60	10		0.15	2.25	2.25	9	1.5
FlightCom	1	0.16	CONT	1	15	15	60	10		0.16	2.4	2.4	9.6	1.6
Maintenance Hour Meter	1	0.05	CONT	1	15	15	60	10		0.05	0.75	0.75	3	0.5
									TOTAL AMP * MINUTES	58.2	456.7	441.8	1560.5	297.8
									UTILIZATION LOAD (AMPS)	58.2	30.45	29.45	26.01	29.78

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CHAPTER

25

EQUIPMENT / FURNISHINGS

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CHAPTER 25

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CHAPTER 25 - EQUIPMENT / FURNISHINGS

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FLIGHT COMPARTMENT

1. Pilot's Seat Lock and Release Rigging

Refer to "Figure 1".

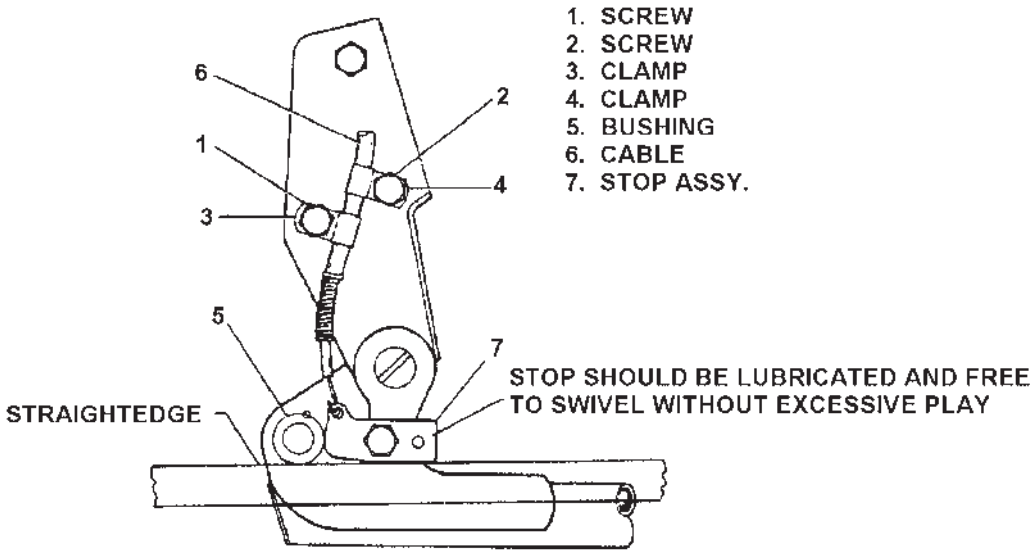
A. Loosen screws and loosen clamps to allow push-pull cable to move within the clamps.

B. Place a straightedge along lower surface of seat back release bushing.

C. Adjust push-pull cable by raising or lowering until lower surface of the stop assembly is parallel to straightedge.

D. Secure push-pull cable in position by tightening screws on clamps. The stop must be lubricated and free to swivel without excessive play.

E. Push on seat back to check stop assembly engagement. Rotate seat back release handle and check for seat back disengagement.



Seat Back Lock
Figure 1

2. Lumbar Support

Refer to “Figure 2” on page 25103.

A. Description

The pilot and co-pilot seats incorporate a Lumbar support as standard equipment, except in the Archer TX 2013 and up. The installation consists of an inflatable bladder attached to the seat back filler and an inflation bulb located under and on the inboard side of each pilot and co-pilot seat.

B. Removal

(1) Remove seat from airplane.
(2) Loosen velcro securing seat back filler cover.
(3) Remove only enough of seat back filer cover to expose lumbar bladder.

NOTE: Inflation tube may be removed before or after bladder is removed from seat back filler. Tube is not glued to nipple attachment; it can be removed by carefully pulling on tube.

(4) Remove inflation tube from bladder.

CAUTION: DO NOT USE A CHEMICAL SOLVENT TO REMOVE BLADDER. SOLVENT MAY DAMAGE SEAT BACK FILLER

CAUTION: TO AVOID OR MINIMIZE DAMAGE TO SEAT BACK FILLER DURING REMOVAL, USE ONE HAND TO RETAIN SEAT BACK FILLER IN PLACE, WHILE GENTLY REMOVING BLADDER WITH OTHER HAND.

(5) Starting at either right or left edge of bladder, carefully and slowly pull bladder and pad assembly from seat back filler.

C. Assembly

(1) Apply a layer of 3M 847 cement to smooth side of bladder pad.
(2) Apply a layer of 3M 847 cement to back side of bladder (side away from inflation tube nipple).
(3) Attach bladder pad to bladder.

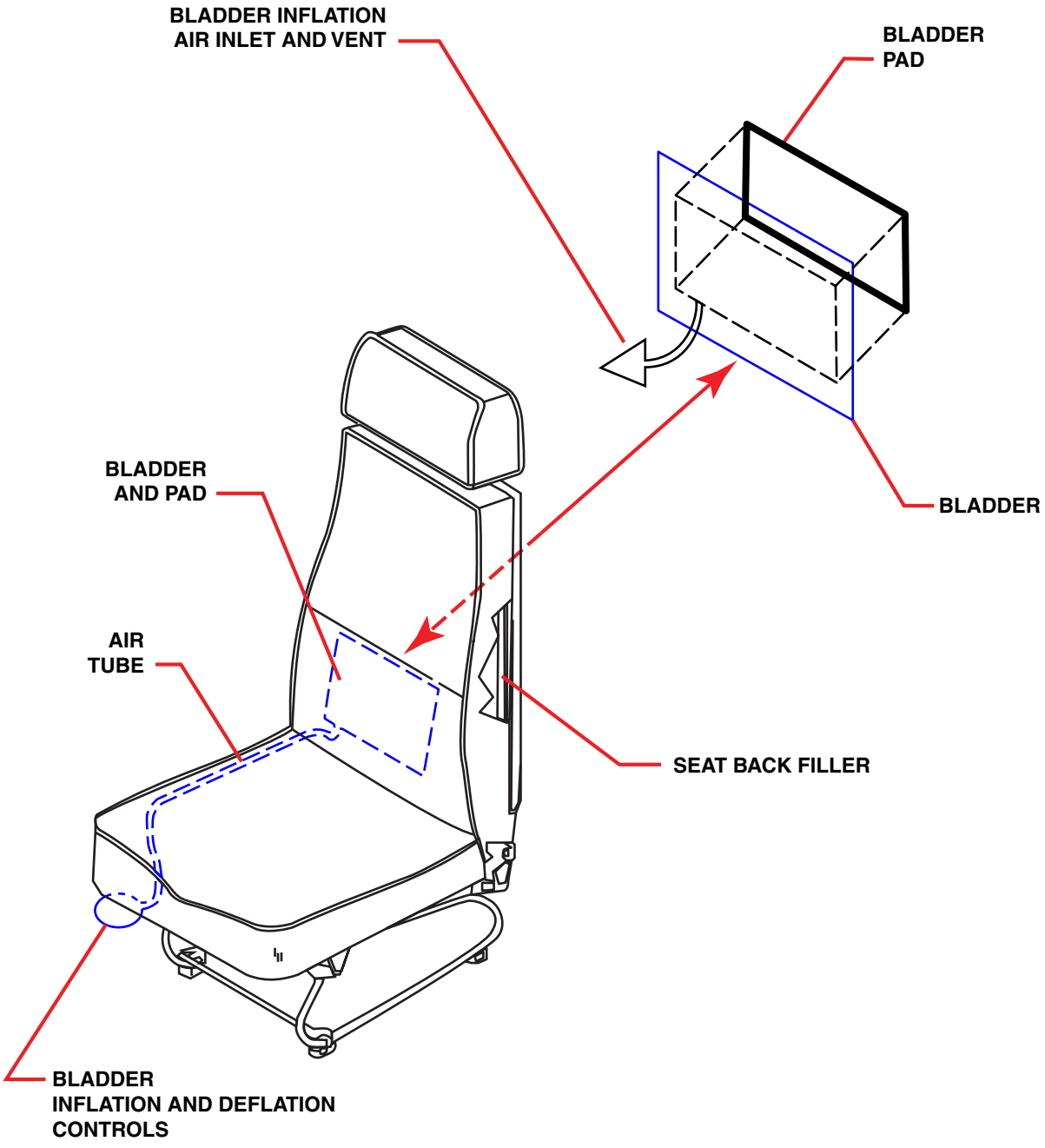
NOTE: While cement does not set immediately, there is no need to wait before attaching bladder and pad to seat back filler.

D. Installation

(1) Apply a layer of 3M 847 cement to rough side of bladder pad.
(2) Apply a layer of 3M 847 cement to seat back filler where bladder is to be located.
(3) Attach bladder and pad assembly to seat back filler. Depending on temperature and humidity, allow 0:30 minutes to 1:00 hour for cement to set.
(4) Install seat back filler cover and secure velcro fastenings.
(5) Install seat in airplane.

3. Carpets

The carpets are individually fastened to the floor with Velcro fasteners and adhesive (Hysol EA9309 NA).



Lumbar Seat Bladder Installation
Figure 2

4. Restraint System

An integrated shoulder harness / lap belt restraint system is installed in S/N's 2843337 and up. Earlier airplanes employ a traditional lap belt with a separate, detachable, shoulder harness.

NOTE: AmSafe Seatbelt Airbags are available in S/N's 2843711 and up. When ordered, they are installed per STC and their Instructions for Continued Airworthiness (ICA) are the sole responsibility of the STC holder (i.e., AmSafe). Further mention in this manual (if any) typically addresses interface with the Piper airframe or other Piper installed components.

A. Inspection

(1) In S/N's 2843001 thru 2843336 only:

(a) Shoulder Harness

1) Inspect ends and attachment points for condition and security.

2) Inspect harness web material for condition and wear over its entire length. Particularly look for wear and fraying where harness web passes in and out of inertial reel. If excessively worn, replace.

3) Check inertia reel mechanism by pulling sharply on strap. Verify reel will lock in place under sudden stress.

(b) Lap Belt

1) Inspect ends and attachment points for condition and security.

2) Inspect harness web material for condition and wear over its entire length. Particularly look for wear and fraying where harness web passes in and out of adjustable buckle end. If excessively worn, replace.

3) Inspect shoulder harness keeper nylon bushing. If excessively worn or missing, replacement of that half of the lap belt is required.

(c) Additionally, in S/N's 2843001 thru 2843013 only:

1) If Piper Service Bulletin No. 990 has been complied with, no further action is required. If not, then inspect seat belts at each position.

2) Replace all belts manufactured by Pacific Scientific and the seat belt has a date stamp of 6/94 thru 8/95, if the seat belt buckle end has an old Piper logo (large stylized "P") or no logo, and if the seat belt fitting end I.D. tag has a date stamp of 6/94 thru 8/95.

3) Replacement belts, fitting end and buckle end, should be ordered under the same part number(s) found on the suspect seat belts as no new part numbers were issued. Order the appropriate quantity of: Seat Belt, Fitting End - P/N 564-867; Buckle End (short) - P/N 564-868; Buckle End (medium) - P/N 564-889; and/or, Buckle End (long) - P/N 564- 862.

4) Check the new seat belts for operation and security, after installation.

5) Make an appropriate logbook entry of compliance with Service Bulletin No. 990.

6) Pending replacement of the suspect seat belts, the aircraft may be flown, provided the Pilot-In-Command inspects each seat belt being used for secure positive latching and that the seat belts are free of any anomalies prior to each flight and during each use.

(2) In S/N's 2843337 and up:

Integrated Shoulder Harness / Lap Belt (Schroth)

(a) Inspect ends and attachment points for condition and security.

(b) Inspect harness web material for condition and wear over its entire length. Particularly look for wear and fraying where harness web passes in and out of inertial reel and in and out of the adjusting buckle. If excessively worn, replace.

(c) Check inertia reel mechanism by pulling sharply on strap. Verify reel will lock in place under sudden stress.

B. Inertial Reel Adjustment

The inertial reel locking feature prevents the shoulder strap from extending and holds occupant in place. For normal movement strap will extend and retract as required. If required, adjust inertial reel as follows:

(1) Allow harness to wind up on reel as much as possible.

(2) On end of reel, pry off plastic cover over spring. Make sure spring does not come out of plastic cover. Set aside plastic cover.

(3) Unwind the harness completely. Measure and mark the harness 24 inches from the reel center.

(4) Wind harness onto reel until the 24 inch mark is reached. Hold reel and place cap with spring over reel shaft end.

(5) Align slot in shaft with spring tang. Wind spring 6 1/2 turns and snap plastic cover into holes in reel end shaft.

(6) Release harness and allow harness to wind up. Extend harness several times to check reel for smooth operation.

(7) Hold inertia reel with reel completely wound and inertia mechanism end up. Pry off plastic cover over mechanism and set reel aside.

(8) Install nut in plastic cover so that stud in cover is flush with nut surface. Position cover over reel and snap cover into place. Extend harness several times to ensure reel operates smoothly.

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EMERGENCY

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S) WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

This section contains information necessary to perform operational checks of the Emergency Locator Transmitter (ELT), with pilot's remote switch. Included are the appropriate removal and installation instructions to facilitate battery replacement.

1. Artex ELT 110-4 Emergency Locator Transmitter (ELT)

Installed as standard equipment in S/N's 2843001 through 2843656.

A. Description

The Artex ELT 110-4 transmits on 121.5 mHz and 243.0 mHz, and is designed to meet or exceed the requirements of TSO C91a and FAR Part 91. Electrical power for the ELT is totally supplied by its own self-contained battery. The battery must be replaced on or before the replacement date marked on battery pack label. Also, the battery must also be replaced any time it has been used in an emergency situation or after more than one hour of accumulated test time.

The Artex ELT 110-4 cannot be accidently activated by dropping the unit, handling it roughly, or during shipping. However, when properly mounted, and locked into its mounting tray, the ELT will activate in a crash, regardless of the cockpit remote switch or ELT-unit switch position. The normal position of the ELT-unit switch is in the down or OFF position. The normal position of the remote cockpit switch is in the down or ARM position.

Whenever the ELT is activated, a red light located just above the remote cockpit switch will blink to alert the pilot or maintenance personnel. Should the ELT be activated accidently, it must be reset. To reset:

(1) Position the remote cockpit switch to ON, then immediately reposition it to ARM, or;
(2) Position the ELT-unit switch to ON, then immediately reposition it to OFF.

B. Inspections

(1) 100 Hour

Inspect per the 100 hour inspection (including the Antenna Test) under Artex ME406 ELT, below.

(2) Annual

Every twelve (12) months, inspect the ELT installation as follows:

NOTE: The following inspection satisfies the requirements of FAR 91.207.

(a) Remove ELT from its mounting tray per Removal, below.

Inspect the mounting hardware. Ensure the hardware is free of cracks or other obvious damage.

(b) Battery Inspection

1) Remove the four (4) securing screws from the bottom of the ELT.
2) Position the ELT product label (arrow) side down, and carefully lift the battery pack away from the ELT and lay it alongside the ELT.
3) Carefully disconnect the harness from the connnector in the black plastic housing. Use a flat-bladed screwdriver to pry the connector out of its mating plug.
4) Inspect the battery pack and ELT chassis. The battery cells, components and connectors should be free of corrosion.

5) Inspect for broken wires, connections, or damage.
6) Ensure the battery housing is free of cracks or other visible damage.
7) Verify the battery expiration date. If the battery pack has not expired it may be reinstalled. The battery pack must be replaced with a new one:
   a) After use in an emergency;
   b) After an inadvertent activation of unknown duration;
   c) When the total of all known transmissions exceeds one (1) hour; or
   d) On or before the battery replacement (expiration) date.

(c) Perform "G-switch Check" on page 25605 (i.e., under Testing, below).
(d) Reinstall ELT into airplane per Installation, below.
(e) Perform the "Antenna Test" on page 256013 (i.e., under Artex ME406 ELT).
(f) Reset ELT by turning ELT switch to "ON" then to 'OFF/ARM" position.
(g) Make an appropriate logbook entry documenting completion of this inspection and whether or not the ELT passed or failed.

C. Removal

The ELT is located in the aft fuselage/tail cone, just forward of the stabilator leading edge. Access is through a removable panel on the right side.

(1) In the cabin, remove the rear closeout panel.
(2) Disconnect the ground (negative) cable from battery.
(3) At the right side of the empennage, remove four (4) screws and remove ELT access panel.
(4) Remove ELT from the airplane by:

CAUTION: DO NOT USE CONTACT CLEANER ON ELT COMPONENTS. SUCH CHEMICAL AGENTS CAN BE HIGHLY DESTRUCTIVE TO THE MOUNTING HARDWARE AND ELT HOUSING.

(a) Loosening the two screws on the front of the mounting tray and pull mounting tray cap off.
(b) Disconnecting coax (antenna) cable.
(c) Disconnecting the Molex cable from the ELT unit.
(d) Visually inspect and confirm proper seating of all connector pins. Special attention should be given to coaxial center conductor pins which are prone to retracting into the connector housing.
(e) Remove unit from airplane.

D. Installation

(1) Install unit into mounting tray:
   (a) Connect molex and coax cables to ELT unit.
   (b) Install mounting tray cap and secure to front of mounting tray with the two screws.
(2) Replace ELT access panel and secure with four (4) screws.
(3) Reconnect the ground (negative) cable to the airplane battery.
(4) In the cabin, replace the rear closeout panel.
(5) Test transmitter per "Testing" on page 25604.

E. Battery Replacement

See “Figure 1”, below.

(1) Removal

(a) If required, remove the ELT from the airplane per Removal, above.

(b) Remove the four screws on the bottom of the ELT securing the battery pack.

(c) Disconnect battery pack connector from main unit. Use a flat-bladed screwdriver to pry the connector out of its mating plug.

(d) Remove battery pack from unit.

(2) Installation

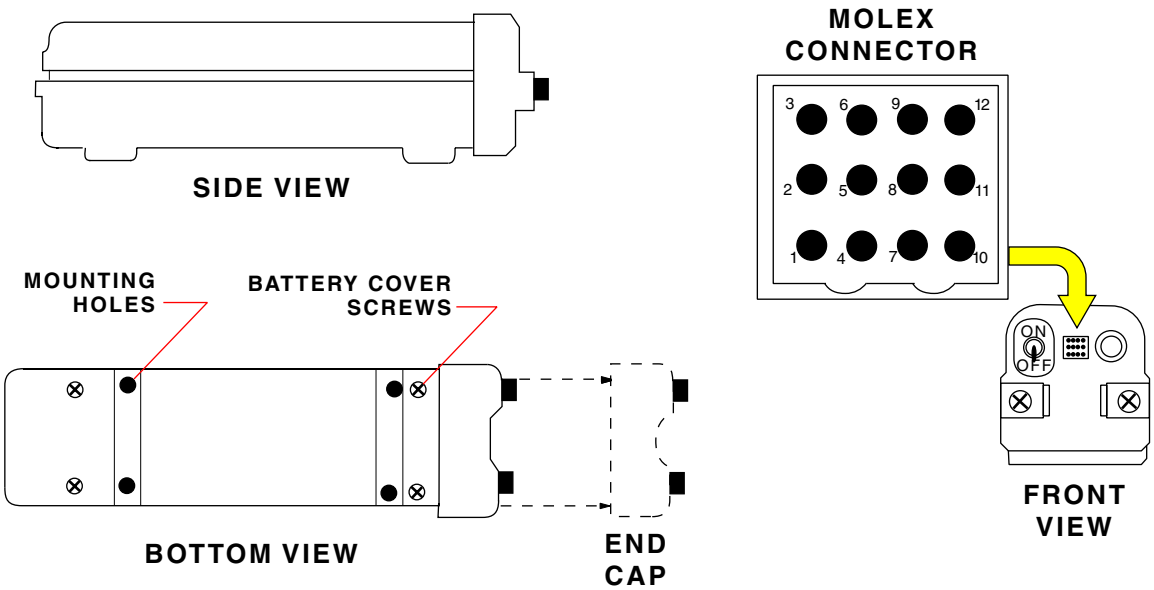
(a) Securely plug in new battery pack connector to main unit.

(b) Immediately reset unit by positioning unit switch to ON, then to OFF.

(c) Fit new battery pack into place. Ensure all gaskets are properly aligned.

(d) Replace the four screws. Dress wires away from standoffs to avoid pinching wires between standoffs and the battery pack.

(e) Install ELT into airplane per Installation, above.



Artex ELT 110-4
Figure 1

F. Testing

CAUTION: ALL ELT "ON" TESTS SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES AFTER THE HOUR UTC OR AS REQUIRED BY LOCAL OR NATIONAL AUTHORITIES. NOTIFY ANY NEARBY CONTROL TOWER OF YOUR INTENTIONS.

CAUTION: DO NOT ALLOW TEST DURATION TO EXCEED FIVE (5) SECONDS.

CAUTION: CONSULT FAA ADVISORY CIRCULAR AC 91-44A, LATEST REVISION, FOR DETAILED INFORMATION CONCERNING UNSHIELDED TESTING.

Always perform the tests within the first five (5) minutes of the hour. Notify any nearby control tower of your intentions. If outside of the US, always follow all local or national regulations for testing ELTs. Do not allow test duration to exceed five (5) seconds. Any time the ELT is activated it is transmitting a 121.5 MHz distress signal.

The transmitter operates on the emergency frequencies of 121.5 and 243.0 MHz; both of these frequencies are monitored by the various FAA installations. Before performing any operational test of the ELT, the following precautions should be observed:

- Test should be no longer than three audio sweeps.
- Test should be conducted only within the time period made up of the first five minutes after any hour.
- If the operational tests must be made at a time not included within the first five minutes after the hour, the test should be coordinated with the closest FAA Tower or Flight Service Station.

(1) Installed Transmitter Test (Self-Test)

Every 90 Days.

(a) Turn both the airplane master switch and the radio master switch ON.

(b) Tune airplane communications receiver to 121.5 mHz and select SPKR on the audio panel.

(c) Turn the ELT aircraft panel switch to "ON", wait for three (3) sweeps on the receiver, which takes about one (1) second, and then turn the switch back to the "ARM" position.

(d) To pass the test, you must hear the three (3) sweeps and see the front panel light immediately begin to flash continuously until the unit is switched to "ARM".

1) During the "ON" to "ARM" transition, the microprocessor in the ELT checks the "G-Switch" (automatic activation switch) latching circuit, pins 5 and 8 on the tray connector.

2) If there is a problem, the processor will not immediately turn on the cockpit light during those first few sweeps. For example, if the jumper between pins 5 and 8 was not installed or was open, the cockpit light would flash momentarily upon ELT activation and the stay off for approximately three (3) seconds before beginning to flash continuously.

3) Repairs should be done only by a licensed aviation radio repair shop.

(2) Antenna Test

Each 100 Hours perform the "Antenna Test" on page 256013 (i.e., under Artex ME406 ELT).

(3) G-switch Check

CAUTION: AS WITH ALL BEACON-TESTING, THIS TEST SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES OF THE HOUR, AND ANY LOCAL CONTROL TOWER SHOULD BE ADVISED OF THE TEST.

Test operation of the G-switch annually as follows:

(a) Setup

1) Remove the ELT from the airplane per Removal, above.

2) Fabricate a “shorting plug” by obtaining an appropriate connector and installing a jumper (i.e. - short) between Pins 5 and 8. (Artex sells a test plug for this purpose under part number 151-2012.)

NOTE: Be sure the correct pins are shorted. Some connections will force the ELT to activate. Others will keep the unit from activating in any circumstance. No combination of shorts will cause permanent damage to the ELT, however all wrong pin combinations erroneously indicate a faulty ELT.

If the ELT activates without any pins shorted it is defective and should be returned for repair.

3) Install the “shorting plug” on the ELT.

4) For a more thorough test, monitor the transmission with an AM receiver tuned to 121.5 MHz, as described in Transmitter Test, below.

(b) Procedure

The ELT should remain OFF until an acceleration of three (3) Gs or more is applied axially, in the rearward direction. This action should activate the unit, transmitting immediately on 121.5 MHz.

CAUTION: MAINTAIN A FIRM GRIP ON THE ELT.

1) Apply acceleration greater than three (3) Gs to the ELT, in the rearward direction. This can be achieved by using a rapid forward (throwing) motion in the direction of the label arrow, then rapidly reversing the direction.

2) Monitor ELT activation by observing the ELT LED (and AM receiver, if desired).

3) Allow ELT to transmit only long enough to verify operation.

4) Reset ELT by turning ELT switch to “ON” then to ‘OFF” position.

G. Electrical Schematic

See 91-25-60.

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2. Artex ME406 ELT

Installed as standard equipment in S/N's 2843657 thru 2843822; 2843824 thru 2843851 and 2843853; and in Standard or Garmin G500-equipped airplanes, S/N's 2843854–2843999.

See "Figure 2" on page 256011.

A. Description and Operation

The Artex ME406 ELT is a fully TSO-C126 / ETSO-2C126 certified, type AF (automatic fixed) beacon 406 MHz ELT. All functions of the ME406 are under microprocessor control. A self-test routine checks ELT operation and installation, then presents the results as visual and auditory "error codes" to aid in troubleshooting and to indicate status. The ME406 is pre-programmed at the factory using a short message format. The following User Protocols are supported: Serial Number; Tail Number; 24-Bit Aircraft Address; and Aircraft Operator Designator/Serial Number.

The battery pack consists of two D-size lithium cells mounted in a cover assembly, and is field replaceable. Rated life is one hour of use or the expiration date on the battery pack, whichever comes first, as specified by FAR 91.207(c).

In the event of a crash, the ME406 activates automatically, and transmits the standard swept tone on 121.5 MHz lasting until battery power is gone. In addition, for the first 24 hours of operation, a 406 MHz signal is transmitting at 50 second intervals. This transmission lasts 440 ms and contains identification data programmed into the beacon and is received by Cospas-Sarsat satellites. Position accuracy of the 121.5 MHz signal is approximately 15-20 km radius about the transmitter. Accuracy of the 406 MHz signal is within about a 3 km radius.

In a crash, an acceleration activated crash sensor (G-switch) turns the ELT "ON" automatically when the ELT experiences a change in velocity (or deceleration) of 4.5 fps ± 0.5 fps. Activation is also accomplished by means of the cockpit mounted remote switch or the local switch on the ELT. To deactivate the ELT, set either switch to the "ON" position, then back to 'ARM'.

The ELT does not have an "OFF" position. Instead, a jumper between two pins on the front D-sub connector must be in place for the G-switch to activate the unit. The jumper is installed on the mating half of the connector so that when the connector is installed, the beacon is armed. This allows the beacon to be handled or shipped without "nuisance" activation (i.e. - front connector removed).

NOTE: The ELT can still be manually activated using the local switch on the front of the ELT. Care should be taken when transporting or shipping the ELT not to move the switch or allow packing material to become lodged such as to toggle the switch.

If the ELT is inadvertently activated, deactivate it by setting either switch to the "ON" position and then back to "ARM".

B. Troubleshooting

See "Chart 1" on page 25608.

C. Inspections

(1) 100 Hour

Each 100 hours time-in-service, inspect the ELT installation as follows:

(a) Inspect the ELT unit and mount for proper installation and secure mounting.

(b) Inspect wiring and conduits for proper routing, secure mounting, and obvious defects.

(c) Inspect bonding and shielding for proper installation and condition.

(d) Inspect antenna for condition, secure mounting, and proper operation (per Antenna Test under Testing, below).

CHART 1 (Sheet 1 of 2)
TROUBLESHOOTING ME406 ELT

Trouble	Cause	Remedy
3 Flash Error after performing Self Test.	Bad load detect. Detects open or short condition on the antenna output or cable.	Check that the RF cable is connected and in good condition.
		Perform continuity check of center conductor and shield. Check for a shorted cable.
		Check for intermittent connection in the RF cable.
		If this error code persists there may be a problem with the antenna installation. Check this with a VSWR meter. Check the antenna for opens, shorts, resistive ground plane connection.
4 Flash Error after performing Self Test.	Low power detected. Occurs if output power is below about 33 dBm (2 watts) for the 406 signal or 17 dBm (50 mW) for the 121.5 MHz output.	Verify battery voltage. Replace battery if low voltage (~5.6 VDC) or if 7 Flash error is also present.
	Also may indicate that 406 signal is off frequency.	Verify 406 MHz frequency. If bad, return for repair / replacement.
5 Flash Error after performing Self Test.	Indicates that the ELT has not been programmed.	Read ELT 406 MHz signal to verify programming.
6 Flash Error after performing Self Test.	Indicates that G-switch loop between pins 5 and 12 at the D- sub connector is not installed. ELT will not activate during a crash.	Check that the harness D-sub jumper is installed by verifying less thant one (1) ohm of resistance between pins 5 and 12. If missing, install jumper wire.
7 Flash Error after performing Self Test.	Indicates that the ELT battery has too much accumulated operation time (more than one hour) per regulation.	Replace battery.
	May also indicate damage to the battery circuit.	If error does not clear after battery replacement, check continuity of battery circuit and correct function of circuit components.

CHART 1 (Sheet 2 of 2)
TROUBLESHOOTING ME406 ELT

Trouble	Cause	Remedy
Remote Switch LED always ON (steady).	Wiring error or frayed wires shorting out pins on back of Remote Switch.	Verify wiring. Verify integrity of all crimp or solder connections on harness.

(2) Annual

Every twelve (12) months, inspect the ELT installation as follows:

NOTE: The following inspection satisfies the requirements of FAR 91.207.

(a) Remove ELT from its mounting tray per Removal, below.

Inspect the mounting hardware. Ensure the hardware is free of cracks or other obvious damage.

(b) Battery Inspection

NOTE: The battery pack contains static sensitive parts, take ESD precautions before handling.

1) Remove the eight (8) securing screws from the battery-side cover.

NOTE: Battery pack is identified by the embossed text: “BATTERY ACCESS ON THIS SIDE”.

CAUTION: DO NOT PULL ON THE FLEXIBLE PORTION OF THE CABLE - USE THE RIGID SECTION OF THE FLEX CIRCUIT AT THE CONNECTOR AS A HANDLE.

2) Carefully lift the battery cover (battery pack) away from the ELT and unplug the flex-cable connected to the pack.

3) Inspect the battery pack and ELT chassis. The battery cells, components and connectors should be free of corrosion.

4) Inspect flex-circuit for broken connections or damage.

5) Ensure the battery housing is free of cracks or other visible damage.

6) Verify the battery expiration date. If the battery pack has not expired it may be reinstalled. The battery pack must be replaced with a new one:

a) After use in an emergency;

b) When the transmitter has been in use for more than one (1) cumulative hour (i.e., 7 flash error);

c) After an inadvertent activation of unknown duration; or

d) On or before the battery replacement (expiration) date.

(c) Perform “G-switch Check” on page 256013.

(d) Reinstall ELT into airplane per ”Installation” on page 256010.

(e) Perform “Antenna Test” on page 256013.

NOTE: For a fee, http://www.406Test.com allows testing of the ELT through-the-satellite.

(f) Perform “Digital Message Verification” on page 256014.

(g) Reset ELT by turning ELT switch to “ON” then to ‘ARM” position.

(h) Check ELT for signs of registration. In the US, NOAA supplies a beacon registration label that is applied to the ELT when it is registered.

(i) Make an appropriate logbook entry documenting completion of this inspection and whether or not the ELT passed or failed

D. Removal

See “Figure 2” on page 256011.

(1) In the cabin, remove the rear closeout panel.

(2) Disconnect the ground (negative) cable from battery.

(3) At right side of empennage, remove four (4) screws and remove ELT access panel.

(4) Remove ELT Connections:

(a) Loosen the thumbscrews and remove the D-sub and RF connectors.

(b) Visually inspect and confirm proper seating of all connector pins. Special attention should be given to coaxial center conductor pins which are prone to retracting into the connector housing.

(5) Loosen the retention strap and remove the ELT from its mounting tray.

E. Installation

See “Figure 2” on page 256011.

(1) Insert the ELT into the mounting tray at an angle so that the locking ears at the end fit into the mounting tray locking slots.

(2) Fasten the retention strap around the ELT so that it is firmly held in place.

(3) Insert the D-sub and RF connectors ensuring that they are seated properly.

(4) Tighten the thumbscrews.

(5) At right side of empennage, place ELT access panel in position and secure with four (4) screws.

(6) In the rear cabin, reconnect the ground (negative) cable to battery.

(7) Reinstall the rear closeout panel.

F. Battery Replacement

NOTE: When replacing the battery pack, a replacement kit containing the battery pack, replacement gasket, hardware and labels is available.

(1) Remove ELT per Removal, above.

(2) Remove the battery pack as follows:

NOTE: The battery pack contains static sensitive parts, take ESD precautions before handling.

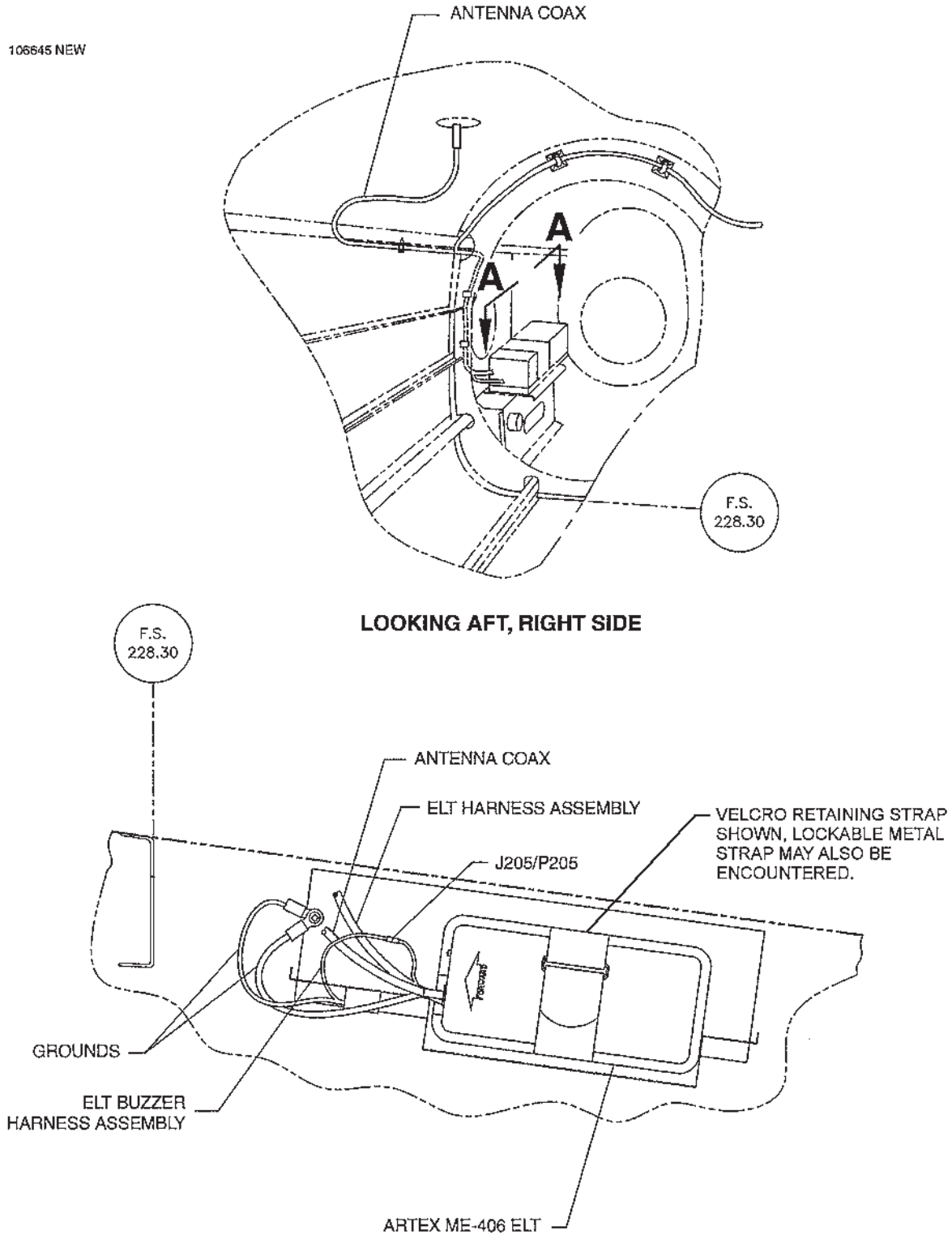
(a) Remove the eight (8) securing screws from the battery-side cover.

NOTE: Battery pack is identified by the embossed text:
“BATTERY ACCESS ON THIS SIDE”.

CAUTION: DO NOT PULL ON THE FLEXIBLE PORTION OF THE CABLE - USE THE RIGID SECTION OF THE FLEX CIRCUIT AT THE CONNECTOR AS A HANDLE.

(b) Carefully lift the battery cover (battery pack) away from the ELT and unplug the flex-cable connected to the pack.

(c) Discard/recycle the old battery pack.



Artex ME406 ELT Installation
Figure 2

(3) Lay the new battery pack on the work surface with the batteries facing up.

(4) Install a replacement seal in the slot along the perimeter of the housing.

(5) Leaving the battery 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. The cable should not be twisted and the connector should 'click' into place.

NOTE: The battery connector is keyed to prevent incorrect installation.

(6) Mate the ELT to the battery, making sure that the seal is positioned correctly during the process.

(7) Replace the eight (8) securing screws and torque to 10–12 inch-lbs.

(8) Install ELT per "Installation" on page 256010.

(9) Perform Installed Transmitter Test (Self Test) under Testing, below.

(10) Enter pertinent battery replacement information in the aircraft log book and fill out any other documentation required by local authority.

G. Testing

CAUTION: ALL ELT "ON" TESTS SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES AFTER THE HOUR UTC OR AS REQUIRED BY LOCAL OR NATIONAL AUTHORITIES. NOTIFY ANY NEARBY CONTROL TOWER OF YOUR INTENTIONS.

CAUTION: DO NOT ALLOW TEST DURATION TO EXCEED FIVE (5) SECONDS.

CAUTION: CONSULT FAA ADVISORY CIRCULAR AC 91-44A, LATEST REVISION, FOR DETAILED INFORMATION CONCERNING UNSHIELDED TESTING.

Always perform the tests within the first five (5) minutes of the hour. Notify any nearby control tower of your intentions. If outside of the US, always follow all local or national regulations for testing ELTs. Do not allow test duration to exceed five (5) seconds. Any time the ELT is activated it is transmitting a 121.5 MHz distress signal. If the unit operates for approximately 50 seconds, a 406 MHz distress signal is transmitted and is considered valid by the satellite system.

(1) Installed Transmitter Test (Self Test)

Every 90 Days.

(a) Turn both the airplane master switch and the radio master switch ON.

(b) Tune a receiver (usually the aircraft radio) to 121.5 MHz.

(c) Turn the ELT aircraft panel switch to "ON", wait for three (3) sweeps on the receiver, which takes about one (1) second, and then turn the switch back to the "ARM" position while paying special attention of the LED activity upon entering the 'ARM" condition.

(d) To pass the test, you must hear the three (3) sweeps and see the front panel light immediately begin to flash continuously until the unit is switched to "ARM".

1) During the "ON" to "ARM" transition, the microprocessor in the ELT checks the "G-Switch" (automatic activation switch) latching circuit, pins 5 & 12 on the D-sub connector at the ELT; the 406 MHz transmitter for proper RF output and a battery check.

2) If the ELT is working properly, the sequence following entry to the "ARM" condition will result in the panel LED staying illuminated for approximately one (1) second (one pulse), and then extinguishing. The buzzer should also sound once.

(e) If the panel LED and buzzer present more than one (1) pulse when switched to "ARM", determine the problem from the list in "Chart 1" on page 25608.

(2) Antenna Test

Each 100 Hours use a low quality AM broadcast receiver to determine if energy is being transmitted from the antenna.

(a) Hold the antenna of the radio (tuning dial on any setting) about six (6) inches from the activated ELT antenna. The ELT aural tone should be heard on the AM broadcast receiver.

(b) This is not a measured check, but it does provide confidence that the antenna is radiating sufficient power to aid search and rescue.

NOTE: Use of the aircraft's VHF receiver, tuned to 121.5 MHz, is not recommended. This receiver is more sensitive and could pick up a weak signal even if the radiating ELT antenna is disconnected. While it will confirm the ELT is active, it does not check the integrity of the ELT system

(3) G-switch Check

Test operation of the G-switch annually as follows:

CAUTION: EVEN WITHOUT AN ANTENNA CONNECTED, THE SIGNAL CAN BE RECEIVED BY A SATELLITE. AS WITH ALL BEACON-TESTING, THIS TEST SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES OF THE HOUR, AND ANY LOCAL CONTROL TOWER SHOULD BE ADVISED OF THE TEST.

(a) Setup

1) The ELT must be removed from the airplane. See "Removal" on page 256010.

2) Fabricate a "shorting plug" by obtaining a standard 15-pin D-sub connector and installing a jumper (i.e. - short) between Pins 5 and 12. (Artex sells a test plug for this purpose under part number 150-1130.)

NOTE: Be sure the correct pins are shorted. Some connections will force the ELT to activate. Others will keep the unit from activating in any circumstance. No combination of shorts will cause permanent damage to the ELT, however all wrong pin combinations erroneously indicate a faulty ELT.

If the ELT activates without any pins shorted it is defective and should be returned for repair.

3) Install the "shorting plug" on the ELT.

4) For a more thorough test, monitor the transmission with an AM receiver tuned to 121.5 MHz, as described in "Installed Transmitter Test (Self Test)" on page 256012.

(b) Procedure

The ELT should remain OFF until an acceleration of three (3) Gs or more is applied axially, in the rearward direction. This action should activate the unit, transmitting immediately on 121.5 MHz.

CAUTION: MAINTAIN A FIRM GRIP ON THE ELT.

1) Apply acceleration greater than three (3) Gs to the ELT, in the rearward direction. This can be achieved by using a rapid forward (throwing) motion in the direction of the label arrow, then rapidly reversing the direction.

2) Monitor ELT activation by observing the ELT LED (and AM receiver, if desired).

3) Allow ELT to transmit only long enough to verify operation.

CAUTION: ENSURE ELT IS RESET BEFORE 50 SECONDS HAS ELAPSED FROM SUCCESSFUL ACTIVATION. FAILURE TO DO SO WILL PERMIT A "LIVE" SIGNAL TO BE SENT TO THE SEARCH AND RESCUE SATELLITES.

4) Reset ELT by turning ELT switch to "ON" then to 'ARM" position.

(4) Digital Message Verification

The ARTEX 406 MHz ELT transmits a 406 MHz message upon reset, which is encoded such that it will be ignored by the SAR satellite system. The 15-digit number contained in this transmission is used to register the ELT with the appropriate 406 MHz ELT registration authority. In the US, the National Oceanic and Atmospheric Administration (NOAA) maintains the database of registered ELT's. The information in this database provides the Search and Rescue system with aircraft identification data in the event an actual distress signal is transmitted.

Verify the 406 MHz digital message as follows:

(a) Required Equipment

A test set capable of receiving and decoding the message - i.e., Artex ELT Test Set P/N 453-1000, or equivalent.

(b) Procedure

1) At right side of empennage, remove four (4) screws and remove ELT access panel.
2) Disconnect the antenna coax cable at the ELT, connect test set or terminate as applicable.
3) Perform all necessary steps to prepare Test Set to receive 406 MHz signal including (but not limited to) turning on power, activating program or any other steps required for the particular Test Set being used.
4) Perform the Installed Transmitter Test (Self Test) as described above.
5) Watch the screen on the Test Set to ensure that a message has been received. Repeat "Self Test" if necessary.
6) View message, ensure that all applicable information is correct (country code, aircraft ID, etc.).
7) The 15 digit ID hex ID (for example "ADC6492640D3411F1') should match what is shown on the ELT product label. This is the 15 digit hex ID (Unique Identification Number or 'UIN") that is used to register the ELT.
8) At right side of empennage, place ELT access panel in position and secure with four (4) screws.

H. Electrical Schematic

See 91-25-60.

3. Artex ELT 1000

Installed as standard equipment in Garmin G1000 NXi-equipped airplanes 2843823, 2843852, 2843854–2843999; and 2881001 and up.

See “Figure 3”, below, and “Figure 4” on page 256017.

A. Description and Operation

The Artex ELT 1000 is a TSO-C126 / ETSO-2C126 certified and COSPAS-SARSAT TAC 251 approved, type AF (Automatic Fixed) beacon, which transmits on 121.5 and 406 MHz. When the 406 MHz signal is detected by the Cospas–Sarsat satellite LEO (Low Earth Orbit system), a position is calculated and the 121.5 MHz signal is used to home in on the crash site. The ELT 1000 is pre-programmed at the factory to use a transmitted short message format. The information contained in the message includes: (a) beacon serial number, (b) an aircraft identification or registration number, (c) a country of registration and country code and (d) position coordinates, if beacon is programmed to receive position data from the aircraft navigation system. ELT operation checks are performed by a series of self-tests with results as visual and auditory “error codes” to aid in troubleshooting and to indicate status.

The battery pack consists of two D-size lithium cells mounted in a covered assembly, and is field replaceable. Rated life is six (6) years unused or with total of transmissions exceeding one hour of use, whichever comes first, as specified by FAR 91.207(c). The lithium battery packs used on the Artex ELT 1000 are certified under TSO C142a.

In the event of a crash, the ELT 1000 activates automatically, and transmits the standard swept tone (homing signal) on 121.5 MHz lasting until battery power is gone, which with lithium cells is at least 50 hours. In addition, for the first 24 hours of operation, a 406 MHz signal is transmitting at 50 second intervals. This transmission lasts 440 ms and contains identification data programmed into the beacon and is received by Cospas-Sarsat satellites. Position accuracy of the 121.5 MHz signal is approximately 15–20 km radius about the transmitter. Accuracy of the 406 MHz signal is within about a 3 km radius. If position information is extracted from the aircraft navigation system, the accuracy improves to approximately 100 meters.



Artex ELT 1000 Emergency Locator Transmitter
Figure 3

In a crash, an acceleration activated crash sensor (G-switch) turns the ELT “ON” automatically when the ELT experiences a change in velocity (or deceleration) of 4.5 fps ± 0.5 fps. Activation is also accomplished by means of the cockpit mounted remote switch or the local switch on the ELT. To deactivate the ELT, set either switch to the “ON” position, then back to ‘ARM’.

The ELT does not have an “OFF” position. Instead, a jumper between two pins on the front D-sub connector must be in place for the G-switch to activate the unit. The jumper is installed on the mating half of the connector so that when the connector is installed, the beacon is armed. When this connector is removed the ELT can not be activated. This allows the beacon to be handled or shipped without “nuisance” activation (i.e., front jumper connector removed).

NOTE: The ELT can still be manually activated using the local switch on the front of the ELT. Care should be taken when transporting or shipping the ELT not to move the switch or allow packing material to become lodged such as to toggle the switch.

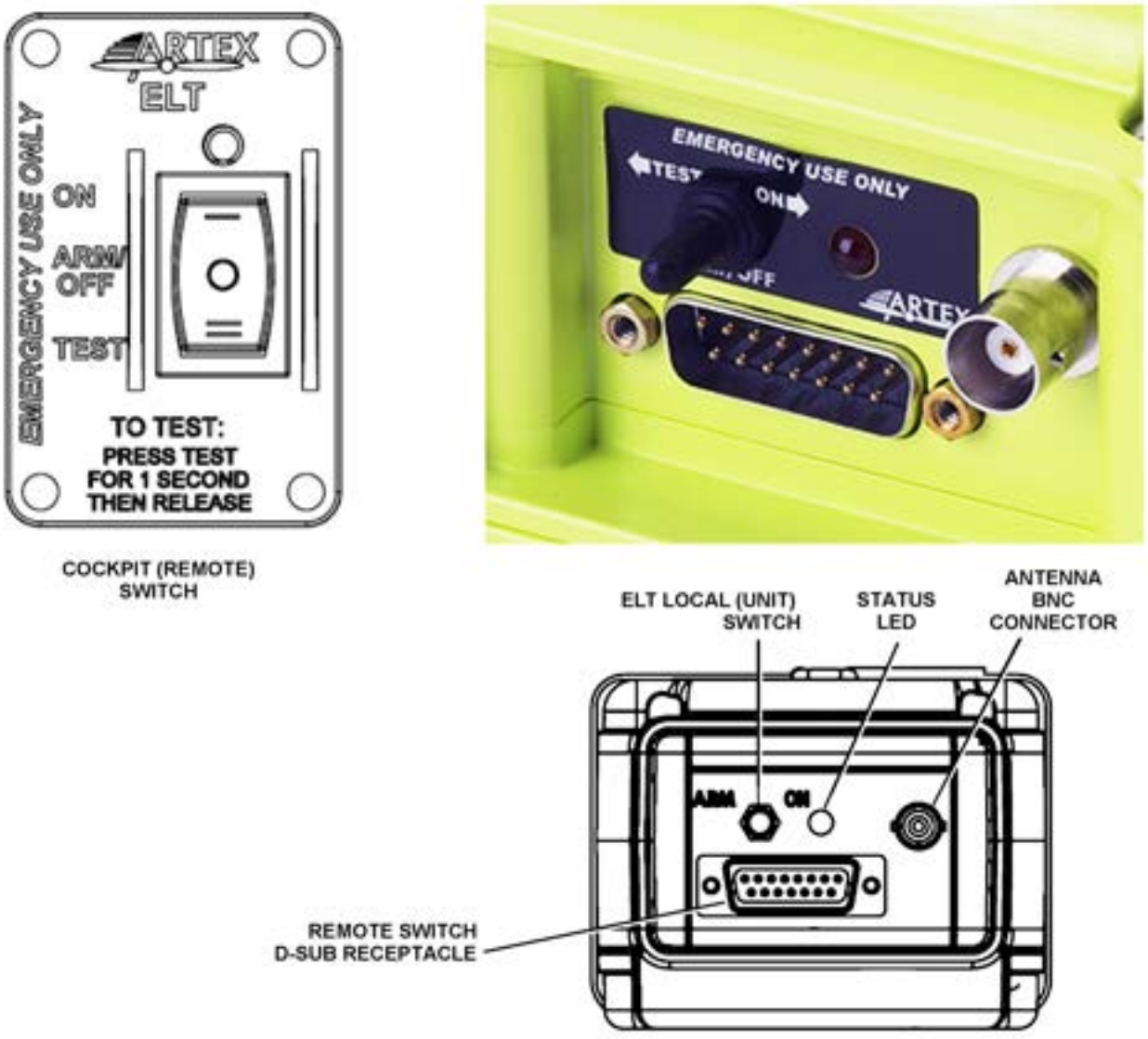
If the ELT is inadvertently activated, deactivate it by setting either switch to the “ON” position and then back to “ARM”. If the switch is already in the “ON” position, move it to the “ARM” position.

B. Troubleshooting

See “Chart 1” on page 256016. See also “Installed Transmitter Test (Self Test)” on page 256024 and “Chart 2” on page 256026.

CHART 1
TROUBLESHOOTING ELT 1000

Trouble	Cause	Remedy
Remote switch LED always on (Steady).	Improper wiring.	Verify wiring.
	Short circuit.	Verify integrity of all crimp and solder connections.
ELT Will Not Turn Off.	**Turn ELT “OFF” Using Local Switch. If ELT turns OFF, then:**	
	Defective remote switch harness reset circuit wiring.	Check wiring continuity and repair as necessary.
	Defective remote switch.	Replace remote switch.
	If ELT Does Not Turn “OFF” Using Local Switch. then:	
	ELT defective.	Remove battery pack to disable ELT and return ELT to manufacturer for servicing.
Flash Errors.	Self-Test.	See “Chart 2” on page 256026.



ELT 1000 Manual Activation Switches
Figure 4

C. Inspections

(1) 100 Hour

Each 100 hours time-in-service, inspect the ELT installation as follows:

(a) Inspect the ELT unit and mount for proper installation and secure mounting.

(b) Inspect wiring and conduits for proper routing, secure mounting, and obvious defects.

(c) Inspect bonding and shielding for proper installation and condition.

(d) Inspect antenna for condition, secure mounting, and proper operation (per “Antenna Test” on page 256024).

(e) Check ELT for signs of registration. In the US, NOAA supplies a beacon registration label that is applied to the ELT when it is registered.

(f) Make an appropriate logbook entry documenting completion of this inspection and whether or not the ELT passed or failed.

(2) Annual

Every twelve (12) months, inspect the ELT installation as follows:

NOTE: The following inspection satisfies the requirements of FAR 91.207.

(a) Remove ELT from its mounting tray per Removal, below.

Inspect the mounting hardware. Ensure the hardware is free of cracks or other obvious damage.

(b) Battery Inspection

NOTE: The battery pack contains static sensitive parts, take ESD precautions before handling.

1) Remove the eight (8) securing screws from the battery-side cover.

NOTE: Battery pack is identified by the embossed text:
“BATTERY ACCESS ON THIS SIDE”.

CAUTION: DO NOT PULL ON THE FLEXIBLE PORTION OF THE CABLE - USE THE RIGID SECTION OF THE FLEX CIRCUIT AT THE CONNECTOR AS A HANDLE.

2) Carefully lift the battery cover (battery pack) away from the ELT and unplug the flex-cable connected to the pack.

3) Inspect the battery pack and ELT chassis. The battery cells, components and connectors should be free of corrosion.

4) Inspect flex-circuit for broken connections or damage.

5) Ensure the battery housing is free of cracks or other visible damage.

6) Verify the battery expiration date. If the battery pack has not expired it may be reinstalled. The battery pack must be replaced with a new one:

a) After use in an emergency;

b) When the transmitter has been in use for more than one (1) cumulative hour (i.e., 7 flash error);

c) After an inadvertent activation of unknown duration; or

d) On or before the battery replacement (expiration) date.

(c) Perform “(3) G-switch Check” on page 256025.

(d) Reinstall ELT into airplane per “Installation” on page 256020.

(e) Perform “Antenna Test” on page 256024.

NOTE: For a fee, http://www.406Test.com allows testing of the ELT through-the-satellite.

(f) Perform “Digital Message Verification” on page 256028.

(g) Reset ELT by turning ELT switch to “ON” then to ‘ARM” position.

(h) Check ELT for signs of registration. In the US, NOAA supplies a beacon registration label that is applied to the ELT when it is registered.

(i) Make an appropriate logbook entry documenting completion of this inspection and whether or not the ELT passed or failed.

D. Removal

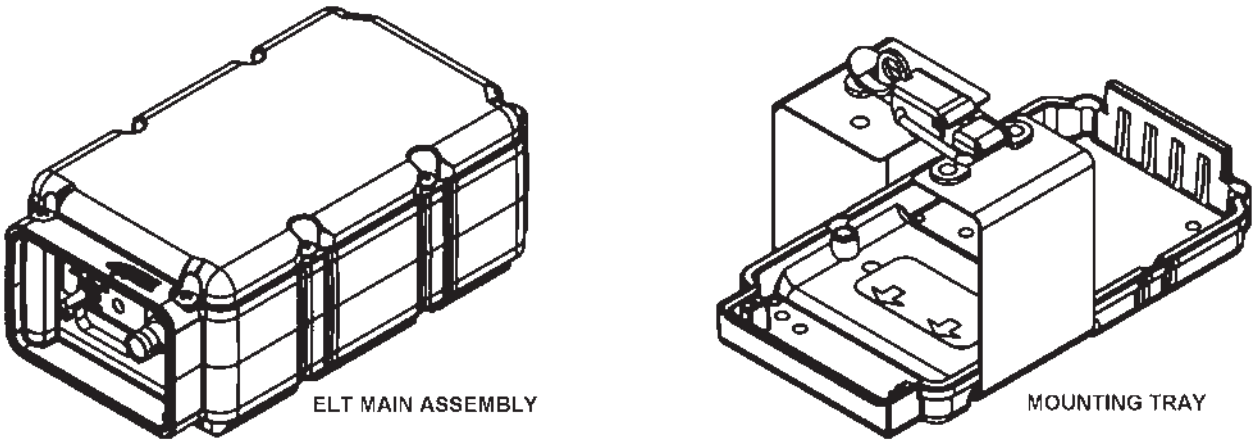
See ”Figure 5”, below, and “Figure 6” on page 256021.

(1) In the cabin, remove the rear closeout panel.
(2) Disconnect the ground (negative) cable from battery.
(3) At right side of empennage, remove four (4) screws and remove ELT access panel.
(4) Remove ELT Connections:
  (a) Loosen the thumbscrews and remove the D-sub and RF connectors.
  (b) Visually inspect and confirm proper seating of all connector pins. Special attention should be given to coaxial center conductor pins which are prone to retracting into the connector housing.
(5) Loosen the retention strap and remove the ELT from its mounting tray.

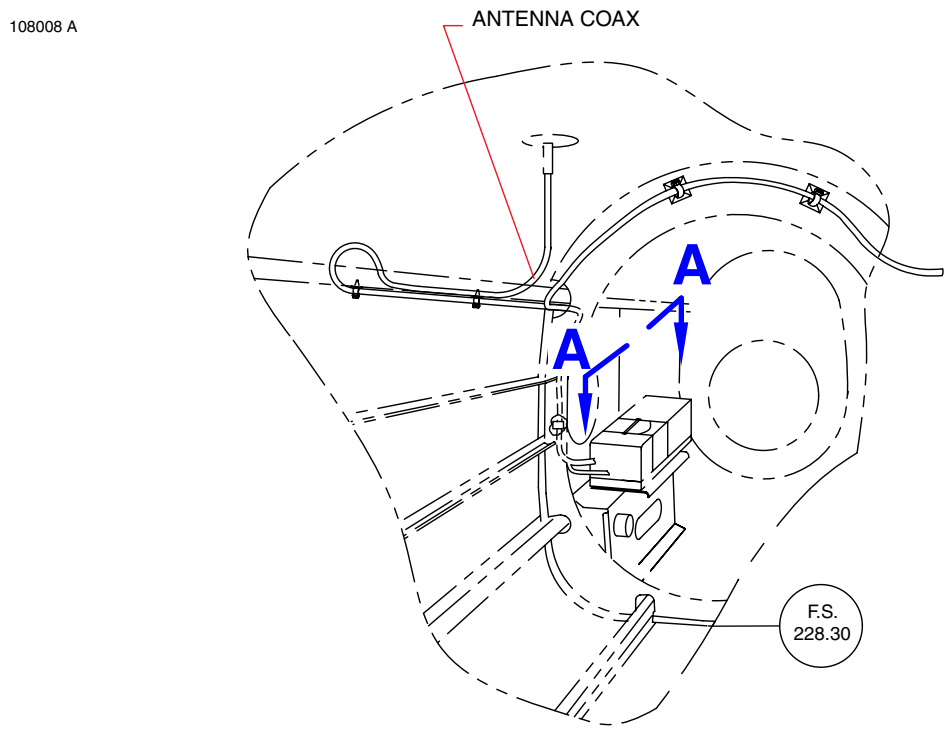
E. Installation

See ”Figure 5”, below, and “Figure 6” on page 256021.

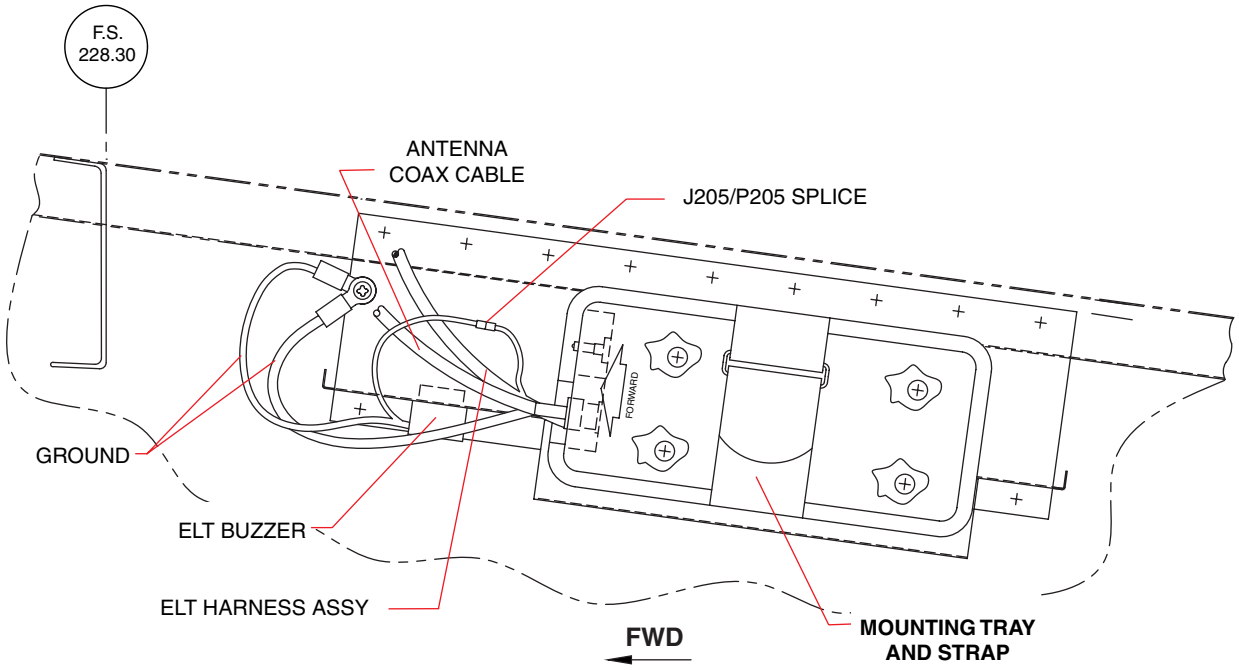
(1) Insert the ELT into the mounting tray at an angle so that the locking ears at the end fit into the mounting tray locking slots.
(2) Fasten the retention strap around the ELT so that it is firmly held in place.
(3) Insert the D-sub and RF connectors ensuring that they are seated properly.
(4) Tighten the thumbscrews.
(5) At right side of empennage, place ELT access panel in position and secure with four (4) screws.
(6) In the rear cabin, reconnect the ground (negative) cable to battery.
(7) Reinstall the rear closeout panel.



ELT 1000 Assembly
Figure 5



LOOKING AFT AT RIGHT SIDE



VIEW A - A

Artex ELT 1000
Figure 6

F. Battery Replacement

See “Figure 7” on page 256023.

NOTE: When replacing the battery pack, a replacement kit containing the battery pack, replacement gasket, hardware and labels is available.

(1) Remove ELT per “Removal” on page 256020.

(2) Remove the battery pack as follows:

NOTE: The battery pack contains static sensitive parts, take ESD precautions before handling.

(a) Remove the eight (8) securing screws from the battery-side cover.

NOTE: Battery pack is identified by the embossed text:
“BATTERY ACCESS ON THIS SIDE”.

CAUTION: DO NOT PULL ON THE FLEXIBLE PORTION OF THE CABLE - USE THE RIGID SECTION OF THE FLEX CIRCUIT AT THE CONNECTOR AS A HANDLE.

(b) Carefully lift the battery cover (battery pack) away from the ELT and unplug the flex-cable connected to the pack.

(c) Discard/recycle the old battery pack.

(3) Lay the new battery pack on the work surface with the batteries facing up.

(4) Install a replacement seal in the slot along the perimeter of the housing.

(5) Leaving the battery 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. The cable should not be twisted and the connector should ‘click’ into place.

NOTE: The battery connector is keyed to prevent incorrect installation.

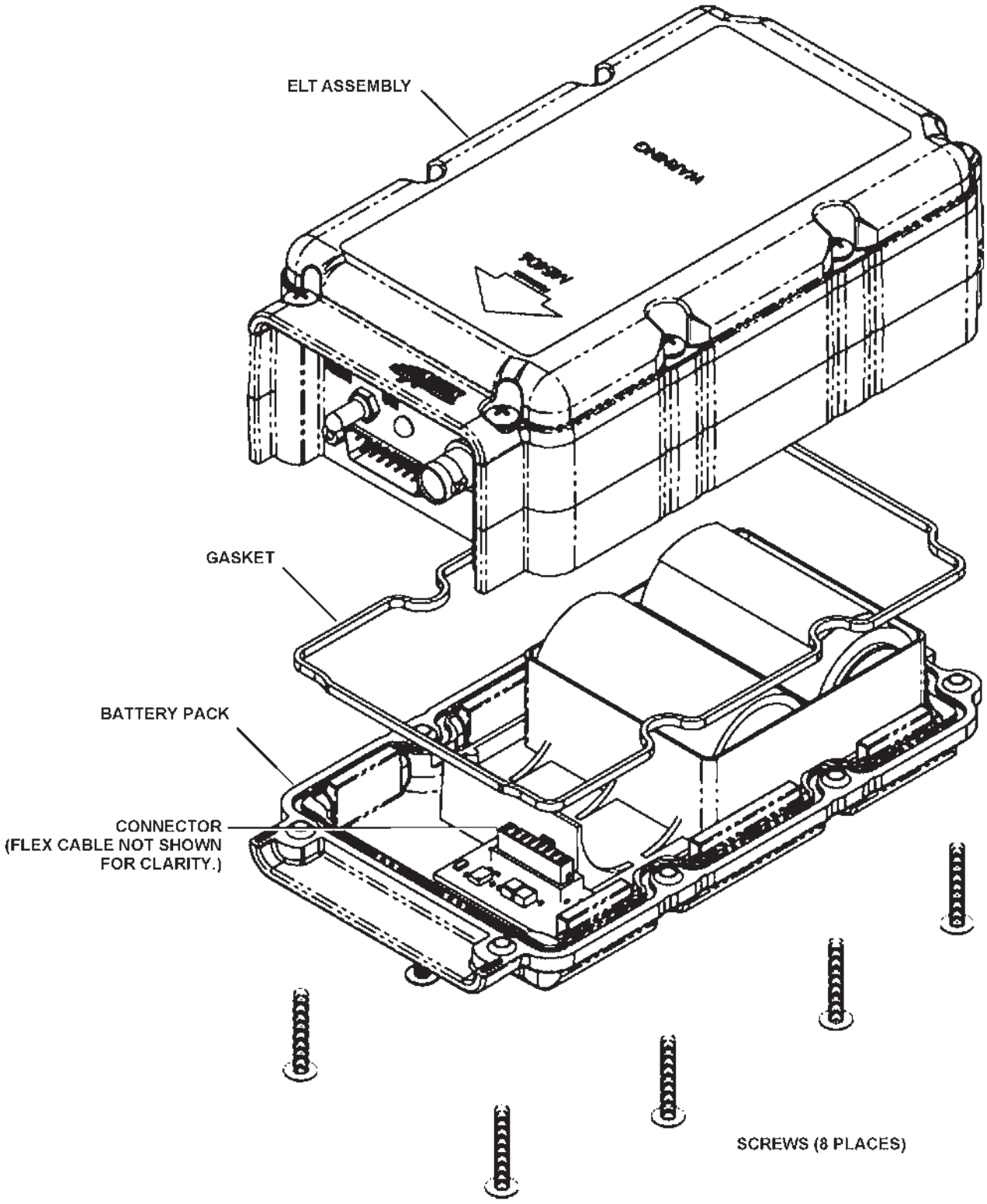
(6) Mate the ELT to the battery, making sure that the seal is positioned correctly during the process.

(7) Replace the eight (8) securing screws and torque to 10 - 12 inch-lbs.

(8) Install ELT per “Installation” on page 256020.

(9) Perform “Installed Transmitter Test (Self Test)” on page 256024.

(10) Enter pertinent battery replacement information in the aircraft log book and fill out any other documentation required by local authority.



Artex ELT 1000 Battery Assembly
Figure 7

G. Testing

CAUTION: ALL ELT "ON" TESTS SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES AFTER THE HOUR UTC OR AS REQUIRED BY LOCAL OR NATIONAL AUTHORITIES. NOTIFY ANY NEARBY CONTROL TOWER OF YOUR INTENTIONS.

CAUTION: DO NOT ALLOW TEST DURATION TO EXCEED FIVE (5) SECONDS.

CAUTION: CONSULT FAA ADVISORY CIRCULAR AC 91-44A, LATEST REVISION, FOR DETAILED INFORMATION CONCERNING UNSHIELDED TESTING.

Always perform the tests within the first five (5) minutes of the hour. Notify any nearby control tower of your intentions. If outside of the US, always follow all local or national regulations for testing ELTs. Do not allow test duration to exceed five (5) seconds. Any time the ELT is activated it is transmitting a 121.5 MHz distress signal. If the unit operates for approximately 50 seconds, a 406 MHz distress signal is transmitted and is considered valid by the satellite system.

(1) Installed Transmitter Test (Self Test)

Every 90 Days.

(a) Turn both the airplane master switch and the radio master switch ON.

(b) Tune a receiver (usually the aircraft radio) to 121.5 MHz.

(c) Turn the ELT aircraft panel switch to "ON", wait for three (3) sweeps on the receiver, which takes about one (1) second, and then turn the switch back to the "ARM" position while paying special attention of the LED activity upon entering the 'ARM" condition.

(d) To pass the test, you must hear the three (3) sweeps and see the front panel light immediately begin to flash continuously until the unit is switched to "ARM".

1) During the "ON" to "ARM" transition, the microprocessor in the ELT checks the "G-Switch" (automatic activation switch) latching circuit, pins 5 & 12 on the D-sub connector at the ELT; the 406 MHz transmitter for proper RF output and a battery check.

2) If the ELT is working properly, the sequence following entry to the "ARM" condition will result in the panel LED staying illuminated for approximately one (1) second (one pulse), and then extinguishing. The buzzer should also sound once.

(e) If the panel LED and buzzer present more than one (1) pulse when switched to "ARM", determine the problem from the list in "Chart 2" on page 256026.

(2) Antenna Test

Each 100 Hours use a low quality AM broadcast receiver to determine if energy is being transmitted from the antenna.

(a) Hold the antenna of the radio (tuning dial on any setting) about six (6) inches from the activated ELT antenna. The ELT aural tone should be heard on the AM broadcast receiver.

(b) This is not a measured check, but it does provide confidence that the antenna is radiating sufficient power to aid search and rescue.

NOTE: Use of the aircraft's VHF receiver, tuned to 121.5 MHz, is not recommended. This receiver is more sensitive and could pick up a weak signal even if the radiating ELT antenna is disconnected. While it will confirm the ELT is active, it does not check the integrity of the ELT system

(3) G-switch Check

Test operation of the G-switch annually as follows:

CAUTION: EVEN WITHOUT AN ANTENNA CONNECTED, THE SIGNAL CAN BE RECEIVED BY A SATELLITE. AS WITH ALL BEACON-TESTING, THIS TEST SHOULD BE PERFORMED WITHIN THE FIRST FIVE (5) MINUTES OF THE HOUR, AND ANY LOCAL CONTROL TOWER SHOULD BE ADVISED OF THE TEST.

(a) Setup

1) The ELT must be removed from the airplane. See “Removal” on page 256020.

2) Fabricate a “shorting plug” by obtaining a standard 15-pin D-sub connector and installing a jumper (i.e. - short) between Pins 5 and 12. (Artex sells a test plug for this purpose under part number 150-1130.)

NOTE: Be sure the correct pins are shorted. Some connections will force the ELT to activate. Others will keep the unit from activating in any circumstance. No combination of shorts will cause permanent damage to the ELT, however all wrong pin combinations erroneously indicate a faulty ELT.

If the ELT activates without any pins shorted it is defective and should be returned for repair.

3) Install the “front connector” on the ELT.

4) For a more thorough test, monitor the transmission with an AM receiver tuned to 121.5 MHz, as described in “Installed Transmitter Test (Self Test)” on page 256024.

(b) Procedure

The ELT should remain OFF until an acceleration of three (3) Gs or more is applied axially, in the rearward direction. This action should activate the unit, transmitting immediately on 121.5 MHz.

CAUTION: MAINTAIN A FIRM GRIP ON THE ELT.

1) Apply acceleration greater than three (3) Gs to the ELT, in the rearward direction. This can be achieved by using a rapid forward (throwing) motion in the direction of the label arrow, then rapidly reversing the direction.

2) Monitor ELT activation by observing the ELT LED (and AM receiver on 121.5 MHz, if desired). Verify activation by listening for the aural sweep tone on the receiver. The LED on the ELT housing will also begin to flash, indicating activation. For proof, the AM radio will verify that the RF output is operational.

3) Allow ELT to transmit only long enough to verify operation.

CAUTION: ENSURE ELT IS RESET BEFORE 50 SECONDS HAS ELAPSED FROM SUCCESSFUL ACTIVATION. FAILURE TO DO SO WILL PERMIT A “LIVE” SIGNAL TO BE SENT TO THE SEARCH AND RESCUE SATELLITES.

4) Reset ELT by turning ELT switch to “ON” then to ‘ARM” position.

CHART 2 (Sheet 1 of 2)
ELT 1000 SELF-TEST

Beeps/Flashes	Result	Indication/Action
1	System OK.	One pulse two seconds long of the LED, remote light and buzzer.
2	Code not used in ELT 1000.	
3	Code not used in ELT 1000.	
4		Low output power.
	Bad load or faulty ELT unit	1) Terminate antenna cable at antenna connection end w/50Ω load. Repeat self-test. If error clears proceed to step 3.
		2) Terminate antenna cable at antenna connection end w/50Ω load. Repeat self-test. If error clears proceed to step 3.
		3) Check coax cable and connectors for shorts and opens, bad center pin, etc. Disconnect, then reconnect cable. Repeat self-test.
		4) Check antenna ground for low resistance (should be <100 m Ω).
		5) Antenna should be 3 feet away from any vertical metal.
		6) Have antenna checked by factory.
5		No position data present (long message protocol only. Error is suppresed for short messages).
	Aircraft navigation system “OFF”.	Turn ON navigation system.
	Faulty system interface wiring or connections.	Check integrity of wiring and connections.
6		G-Switch loop between pins 5 and 12 is not present.
	Jumper open.	Verify D-Sub connector jumper is installed by checking for less than 1Ω between pins 5 and 12. Repair as necessary.
	Jumper missing.	Install G-SW enable jumper between pins 5 and 12 in D-sub harness connector.
7		Battery Check.
	Battery operating time is >1 hr.	Replace battery pack.
	Battery voltage low.	Replace battery pack.
	Battery memory read-write error.	Repeat self-test. Replace battery pack if 7-flash persists.

CHART 2 (SHEET 2 OF 2)
ELT 1000 SELF-TEST

Beeps/Flashes	Result	Indication/Action
8		Protocol Programming Data.
	Missing data such as ELT S/N or aircraft tail number.	ELT requires reprogramming w/appropriate data.

(4) Digital Message Verification

The ARTEX ELT 1000 transmits a 406 MHz message upon reset, which is encoded such that it will be ignored by the SAR satellite system. The 15-digit number contained in this transmission is used to register the ELT with the appropriate 406 MHz ELT registration authority. In the US, the National Oceanic and Atmospheric Administration (NOAA) maintains the database of registered ELT's. The information in this database provides the Search and Rescue system with aircraft identification data in the event an actual distress signal is transmitted.

Verify the 406 MHz digital message as follows:

(a) Required Equipment

A test set capable of receiving and decoding the message - i.e., Artex ELT Test Set P/N 453-1000, or equivalent.

(b) Procedure

1) At right side of empennage, remove four (4) screws and remove ELT access panel.
2) Disconnect the antenna coax cable at the ELT, connect test set or terminate as applicable.
3) Perform all necessary steps to prepare Test Set to receive 406 MHz signal including (but not limited to) turning on power, activating program or any other steps required for the particular Test Set being used.
4) Perform the "Installed Transmitter Test (Self Test)" on page 256024.
5) Watch the screen on the Test Set to ensure that a message has been received. Repeat "Self Test" if necessary.
6) View message, ensure that all applicable information is correct (country code, aircraft ID, etc.).
7) The 15 digit ID hex ID (for example "ADC6492640D3411F1') should match what is shown on the ELT product label. This is the 15 digit hex ID (Unique Identification Number or 'UIN") that is used to register the ELT.
8) At right side of empennage, place ELT access panel in position and secure with four (4) screws.

H. ELT Programming

(PIR-107994 C.)

(1) Required Equipment

Programming the Artex ELT 1000 requires a Artex TPS 8715 Handheld Tester/Programmer or equivalent.

(2) Procedure

(a) Plug the programming cable into the ELT and connect to the Handheld Programmer.

(b) Power on the Handheld. Press "Home". The Main Screen View should appear.

NOTE: Press "Home" until the Artex Icon appears on the Main Screen View.

(c) From the Handheld Programmer Main Screen View, Select "ARTEX".

(d) On the "ARTEX" ELT Programmer page, select "Program".

(e) On the "Program" page, begin programming the transmitter.

NOTE: Artex Programmer may default to the previously entered Model, Protocol and Country Codes.

1) On "Program" dropdown menu, select ELT Model "ELT1000/345".

2) Program the GPS Baud Rate to 4800.
   a) Select “Options”.
   b) Select the “Test” position on the ELT 1000 and release within 2 seconds AFTER selecting “Get Options” on the Artex Programmer.
   c) Select the Dropdown Menu.
   d) Select the GPS Baud “4800”.
   e) Select the “Test” Position on the ELT 1000 and release within 2 seconds AFTER selecting “Set” on the Artex Programmer.
   f) Verify “Successfully Programmed Speed” is displayed on the Artex Programmer.
   g) Select “Return”.
3) For U.S. aircraft and aircraft bound for export country that require Tail number registration on the ELT:
   a) Select “Aviation User/Tail No./Loc (L)”.
   b) Dropdown menu, select desired Country Code.
   c) Enter the tail number of the aircraft (N number), which may by up to seven alphanumeric characters. Use stylus to select “abc” (for alphabet) and select “123” (for numbers). (If keyboard was opened, press “Done”)
   d) When finished, select “Enter”.
4) For aircraft bound for export country that require a 24 bit address registration:
   a) Select “Std Loc/Aircraft 24-bit Addr (L)”.
   b) Select the desired number system (Octal: 8 digits from 0-7 or Hex: 6 digits from 0-9 or A-F) to match the format of the 24 bit address provided.
   c) Enter the 24-bit address. (If keyboard was opened, press “Done”)
   d) When finished, select “Enter”.

      NOTE: The 24 digit hex code is shown at the bottom of the screen with the abbreviated 15 digit hex code below it.

5) Select the test position on the ELT 1000 switch and release. Then within 2 seconds, press the “Program” icon on the programmer.
6) Verify that the screen shows “Programming Successful Verify ELT Using Sarcalc”.

(f) Label Printing:
1) Connect the Handheld Programmer to the applicable printer suing the specified cable and adapter.
2) After the ELT has been successfully programmed, select “Print”, from Printing ELT Data, select “Combined Label”. After printing is complete, select “Return”.

   NOTE: Place new label over current ELT’s label that reads, “This ELT is Programmed For and Must be Registered in Country, Three Digit Country Code, 15-Digit Hex ID”.

(g) Programming Verification:

NOTE: For a Direct Verification, connect the Handheld coax cable to the ELT and for a Remote Verification, remove coax cable and connect Handheld Whip Antenna. For a Remote Verification, a distance of 0-30 feet from the ELT antenna is recommended.

1) Select “Sarcalc”.
2) Select “Measure”.
3) The screen will display “Waiting for Data”.
4) Turn the ELT switch to “TEST”, then release.
5) Verify the message contains the correct information for the aircraft.
6) When completed, turn the Handheld OFF.

(3) Installation Verification

NOTE: Perform the following after ELT installation is completed in the aircraft.

(a) Perform the following functional check within the first 5 minutes after the hour (UTC), as required by AC 43.13-1, Chapter 12, § 12-22, Note 3.
(b) Tune an AM receiver to 121.50 MHz and locate it approximately 20’ to 30’ from the ELT antenna.
(c) Initiate the self-test by momentarily pressing the cockpit remote switch into the TEST position for approximately 1 second.
(d) On the AM receiver, listen for 3 audible sweeps, which takes about 1 second.
(e) Note the LED activity on the cockpit remote switch or the audio beeps from the ELT. If the ELT is working properly, the LED will stay on for approximately 1 second and then turn OFF accompanied by a short beep and a long beep.

I. Electrical Schematic

See 91-25-60.

CHAPTER

26

FIRE PROTECTION

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CHAPTER 26

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EXTINGUISHING

WARNING: AFTER DISCHARGE OF EXTINGUISHER AVOID EXPOSURE TO SMOKE, VAPORS AND OTHER BY-PRODUCTS OF FIRE. DO NOT INCINERATE EXTINGUISHER.

CAUTION: EXTINGUISHER IS A PRESSURE VESSEL. PROTECT FROM CORROSIVE CONDITIONS. IF THERE IS ANY CORROSION OR DAMAGE, EXTINGUISHER SHOULD BE CAREFULLY EMPTIED AND DISCARDED. USE ONLY AS DIRECTED.

Portable Fire Extinguisher

NOTE: No onboard fire extinguisher is required in these airplanes. Piper does, however; recommend one be carried.

A. Disposable, Class 2B:C Fire Extinguisher Model No. RT-A600

This extinguisher was installed as standard equipment in S/N's 2843001–2843818 and was available as a service replacement well into 2015. See "Figure 1".



Figure 1
Model RT-A600
(Piper P/N 459-887)

(1) Description

Containing 1.2 lbs. (550 grams) of Halon 1211/1301 blend, the extinguisher is located atop the tunnel cover just forward of the spar box and is secured by a quick release bracket.

To operate the extinguisher, remove it from the quick-release bracket, hold it upright in either hand by the handgrip, with the spray nozzle pointing forward. Remove the safety pin, direct the nozzle towards the base of the fire source, depress the lever. Maximum extinguishing effect is obtained if the fire fighter uses side to side motion and keeps moving in towards the base of the fire source as it is extinguished. Releasing the lever closes a secondary seal inside the operating head. This interrupts the flow of extinguishant, thus retaining part of the charge, for dealing with a flash back or re-ignition should they occur, without waste or leakage. A partly or totally discharged extinguisher should be replaced immediately after use.

(2) Inspection

Disposable type fire extinguishers should be maintained and inspected in accordance with the nameplate instructions.

(a) Before Each Flight

RT-A600 extinguishers with date codes 2012 and later are subject to H3R Service Bulletin No. 2015-01 (Piper Service Bulletin No. 1280A). These extinguishers should be labeled "Weigh this unit before each flight" and must be weighed before each flight. Weight must be above the minimum specified on the nameplate.

(b) Monthly

Inspect monthly or more frequently. Ensure nozzle is not obstructed and safety seal is intact. Inspection is a "quick check" that an extinguisher is available and will operate. It is intended to give reasonable assurance that the extinguisher is fully charged and operable. This is done by seeing that it is in its designated place, that it has not been actuated (discharged) or tampered with, and that there is no obvious physical damage or condition to prevent operation. Determine fullness by weighing or "hefting."

(c) 100 Hour / Annual

Each 100 hours or annually, whichever comes first, weigh the extinguisher. Replace and return to manufacturer if gross weight is below the minimum specified on the nameplate.

B. Gauged, Class 2B:C Fire Extinguisher Model No. A344T

This extinguisher is installed as standard equipment in S/N's 2843819 and up; and is available as service replacement in S/N's 2843001–2843818. See "Figure 2".

(1) Description

Containing 1.25 lbs. (567 grams) of Halon 1211, the extinguisher is located atop the tunnel cover just forward of the spar box and is secured by a quick release bracket.

To operate the extinguisher, remove it from the quick-release bracket, hold it upright in either hand by the handgrip, with the spray nozzle pointing forward. Remove the safety pin, direct the nozzle towards the base of the fire source, depress the lever. Maximum extinguishing effect is obtained if the fire fighter uses side to side motion and keeps moving in towards the base of the fire source as it is extinguished. Releasing the lever closes a secondary seal inside the operating head. This interrupts the flow of extinguishant, thus retaining part of the charge, for dealing with a flash back or re-ignition should they occur, without waste or leakage. A partly or totally discharged extinguisher should be replaced immediately after use.

Gauged type fire extinguishers should be installed, inspected, and maintained in accordance with the owner's manual supplied with the extinguisher and NFPA No. 10, "Portable Fire Extinguishers." The following summarizes portions of those two documents.

(2) Inspection

Fire extinguisher shall be inspected when initially placed in service and at minimum 30-day intervals or more frequently if circumstances dictate. Persons performing 30-day inspections are not be required to be certified.

Check to see that the extinguisher is undamaged; that the nozzle is unobstructed; that the gauge pressure is in the operable (green) range; that the lockpin and tamper seal is in place, and that the operating instructions are clearly visible. If the inspection reveals a deficiency in any of these conditions, corrective action must be taken.



Figure 2
Model A344T
(Piper P/N 459-750)

(3) Maintenance

(a) Annual

Extinguishers shall be subjected to maintenance not more than one year apart or when specifically indicated by an inspection. Maintenance is a thorough examination of the extinguisher, covering mechanical parts, extinguishing agent and expelling means. It is intended to give maximum assurance that an extinguisher will operate effectively and safely, and should be done professionally. Most authorities require special tags be attached to the extinguisher to verify this service.

(b) Six-Year Maintenance

Every six years, stored pressure fire extinguishers that require a 12-year hydrostatic test shall be emptied and subjected to applicable maintenance procedures. This should be done professionally, and involves a thorough inspection and the replacement of certain parts. When these maintenance procedures are performed during periodic recharging or hydrostatic testing, the six year requirement shall begin from that date.

(c) Twelve-Year Hydrostatic Test

All gauged H3R Aviation fire extinguishers require a 12 year hydrostatic test performed by certified technicians as per NFPA 10. Simply stated, the Hydrostatic test confirms the integrity of the cylinder, and at the 12 year mark, is performed in conjunction with the 6 year maintenance.

C. Replacement Intervals

Replace fire extinguishers as specified in 5-30-00.

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CHAPTER

27

FLIGHT CONTROLS

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GENERAL

1. Description and Operation

The PA-28-181 is controlled in flight by three standard primary control surfaces: ailerons, stabilator, and rudder. Controls are operated through movement of the dual control columns and dual rudder pedals. Individual surfaces are connected to their control components by cables and push-pull tubes. Directional and longitudinal trim control is provided by adjustable trim mechanisms for the rudder and stabilator. Flaps are mechanically operated and can be positioned at 0, 10, 25, and 40 degrees.

Aileron controls consist of two control wheels connected by torque tubes to sprockets on each end of the "T" of the T-bar control column. A chain is wrapped around sprockets and around a double sprocket on the vertical post of the control column. The chain is connected to the primary aileron control cable which is routed through the center of fuselage, to main spar, and out through wings to a bellcrank in each wing. A balance cable is also connected to the bellcrank. As control wheels are moved, control cables move bellcranks, actuating push-pull rods to move ailerons.

Stabilator controls are also connected to the control column. From connecting point, cables are routed around pulleys under the floor, and aft to tail section. Aft end of cables connect to stabilator balance arm which is connected to the stabilator. When control wheels are moved forward or aft, cables move balance arm up and down rotating the stabilator on hinge points.

The rudder is controlled by pilot and copilot rudder pedals. Cables are connected to both sides of rudder pedal assembly and routed aft through the bottom of fuselage to rudder horn. When one rudder pedal is pushed, cables move in opposite directions turning rudder horn and rudder. The wing flap system is operated by a lever between the front seats.

2. Standard Practices and Procedures

The following tips may be helpful in the removal, installation, and rigging of individual control system assemblies.

A. It is recommended, though not always necessary, to level and place the airplane on jacks during rigging and adjustment.

B. Remove turnbuckle barrels from cable ends before withdrawing the cables through the structures.

C. Tie a cord to the cable end before withdrawing the cable through the structures to facilitate installation of cable.

D. Mark cable ends, etc., before disconnecting, use a felt tip marking pen.

E. Assemble and adjust the turnbuckles so that each terminal is screwed an approximately equal distance into the barrel. Do not turn the terminals in such a manner that will put a permanent "twist" into the cables.

F. Cable tensions should be taken with the appropriate control surface in its neutral position.

NOTE: Cable rigging tensions specified must be corrected to ambient temperature in the area where the tension is being checked, using Chart 1. When installing new cables, initially tension them 20 to 30% over nominal tension, then loosen them to the "high side" of the tolerance. This "pre-stretching" will aid in maintaining specified tension after flight testing.

G. Check all cable ball ends for proper seating in retainers after setting cable tension.

H. After completion of each adjustment, check the turnbuckles to be sure not more than three terminal threads are visible outside the barrel. Install the locking clips, and check for proper installation by trying to remove the clips using fingers only. Both locking clips may be installed in opposite holes. Locking clips which have been installed and removed must be scrapped and not reused. Safety turnbuckles in accordance with AC 43.13-1, latest revision.

I. When pushrods or rod ends are provided with an inspection hole, the screw must be screwed in far enough to pass the inspection hole. This can be determined visually or by feel, inserting a piece of wire into the inspection hole. If no hole is provided, there must be a minimum of 0.375 (3/8) of an inch thread engagement.

J. When installing/adjusting rod end jamnuts, refer to Figure 2, 20-00-00, for proper method.

K. After completion of adjustments, each jamnut must be tightened securely.

NOTE: Torque all nuts in the flight control system (including nose wheel steering). Refer to 91-10-00, Chart 2.

L. Ensure all pulley guard pins are properly installed and secured.

3. Flight Control Surface Travel

See “Chart 2” for specifications, see appropriate section for rigging instructions.

4. Flight Control Cable Tension

CAUTION: CABLE TENSIONS GIVEN IN CHART 2 APPLY ONLY TO AIRPLANES WITHOUT AUTOPILOT BRIDLE CABLE INSTALLATIONS, UNLESS THE AUTOPILOT SYSTEM WAS FACTORY INSTALLED BY PIPER. IF AN AUTOPILOT USING BRIDLE CABLES HAS BEEN INSTALLED AFTER LEAVING THE FACTORY, CONSULT THE APPROPRIATE AUTOPILOT VENDOR DOCUMENTATION FOR CORRECT CABLE TENSIONS WITH AUTOPILOT BRIDLE CABLES ATTACHED.

A. See “Chart 2” for specifications, see appropriate section for rigging instructions.

B. When a new cable is installed, cable tension must be rechecked after flight test.

CHART 1
CABLE TENSION VS. AMBIENT TEMPERATURE

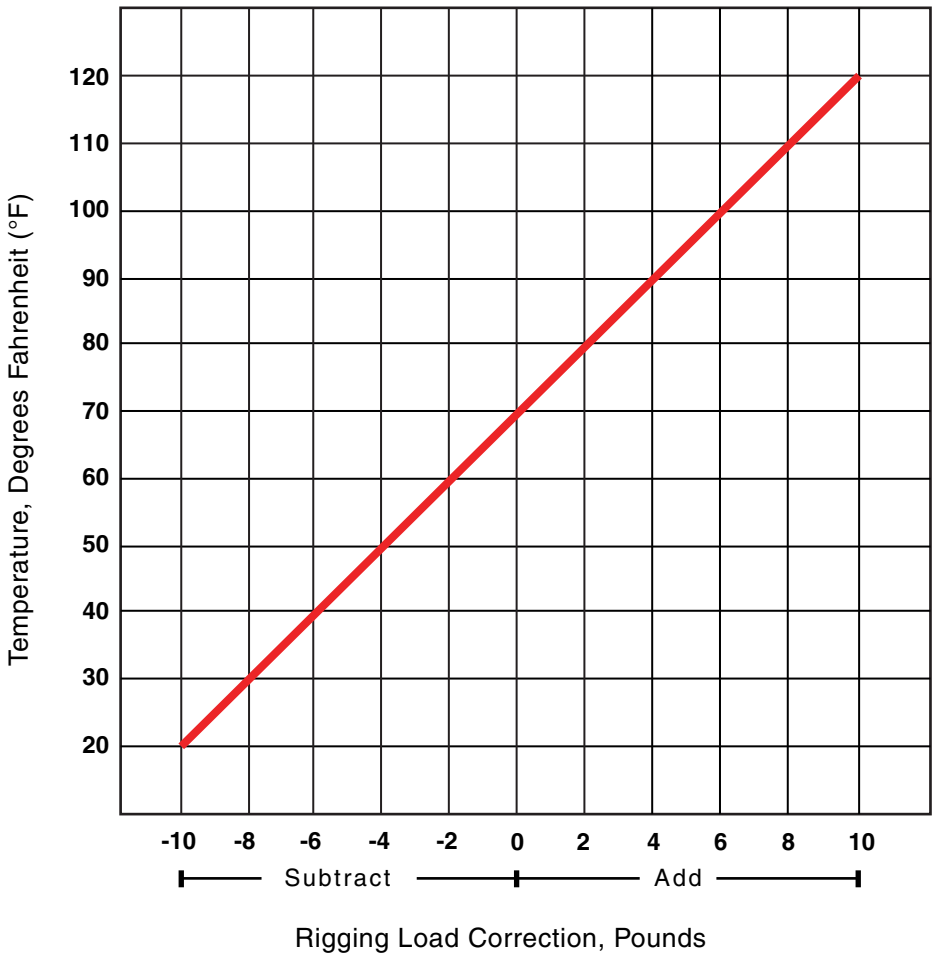
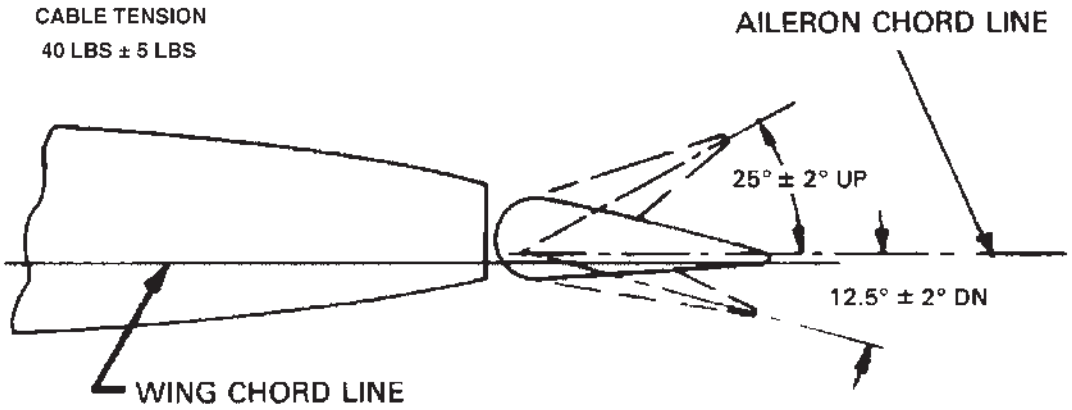


CHART 2 (Sheet 1 of 2)
FLIGHT CONTROL SURFACES RIGGING LIMITS

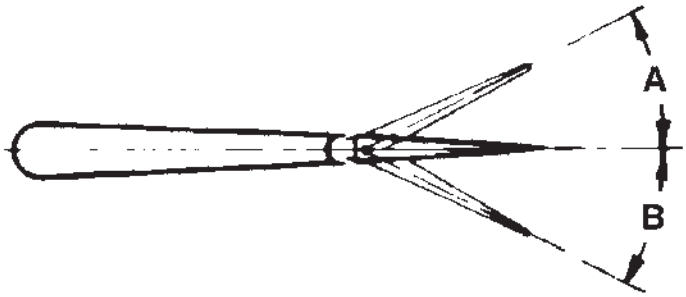
(PIR-PPS50021-2, Rev. K.)

CABLE TENSION
40 LBS ± 5 LBS



1. MAXIMUM FREE PLAY AT TRAILING EDGE (UP/DOWN MOVEMENT) IS 0.24 INCH.
2. MAXIMUM FREE PLAY SPANWISE (INBOARD/OUTBOARD MOVEMENT) IS 0.035 INCH.
3. IN S/N'S 2843001 THRU 2843202 ONLY, CABLE TENSION APPLIES ONLY TO AIRPLANES WITHOUT AUTOPILOT BRIDLE CABLES ATTACHED. SEE APPROPRIATE AUTOPILOT SERVICE MANUAL FOR CABLE TENSION WITH AUTOPILOT BRIDLE CABLES ATTACHED.

RUDDER TRAVEL	CABLE TENSION *
A - 28° ± 1° RIGHT	40 LB ± 5 LB
B - 28° ± 1° LEFT	



* IN S/N'S 2843001 THRU 2843202 ONLY, CABLE TENSION MAY NOT APPLY TO AIRPLANES WITH AUTOPILOT BRIDLE CABLES ATTACHED. SEE APPROPRIATE AUTOPILOT SERVICE MANUAL FOR CABLE TENSION WITH AUTOPILOT BRIDLE CABLES ATTACHED.

CHART 2 (Sheet 2 of 2)
FLIGHT CONTROL SURFACES RIGGING LIMITS

STABILATOR TRAVEL

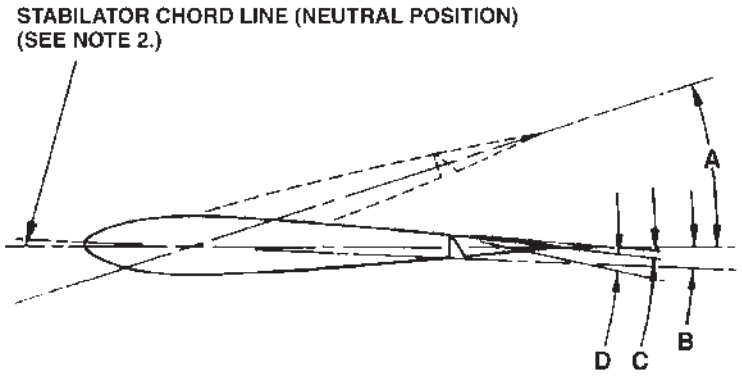
A - 14° ± 1° UP
B - 2° ± 1° DOWN

STABILATOR TAB TRAVEL

C - 3° ± 1° UP
D - 12° ± 1° DOWN

CABLE TENSION (SEE NOTE 3.)

STABILATOR - 40 LB ± 5 LB
STABILATOR TRIM - 14 LB ± 1 LB



1. MAXIMUM FREE PLAY OF STABILATOR TAB, MEASURED AT TAB TRAILING EDGE, IS 0.06 INCH.
2. STABILATOR NEUTRAL POSITION IS WHEN THE STABILATOR CHORD LINE IS PARALLEL TO THE TOP SURFACE OF THE FRONT SEAT SEAT TRACKS.
3. IN S/N'S 2843001 THRU 2843202 ONLY, CABLE TENSION MAY NOT APPLY TO AIRPLANES WITH AUTOPILOT BRIDLE CABLES ATTACHED. SEE APPROPRIATE AUTOPILOT SERVICE MANUAL FOR CABLE TENSION WITH AUTOPILOT BRIDLE CABLES ATTACHED.

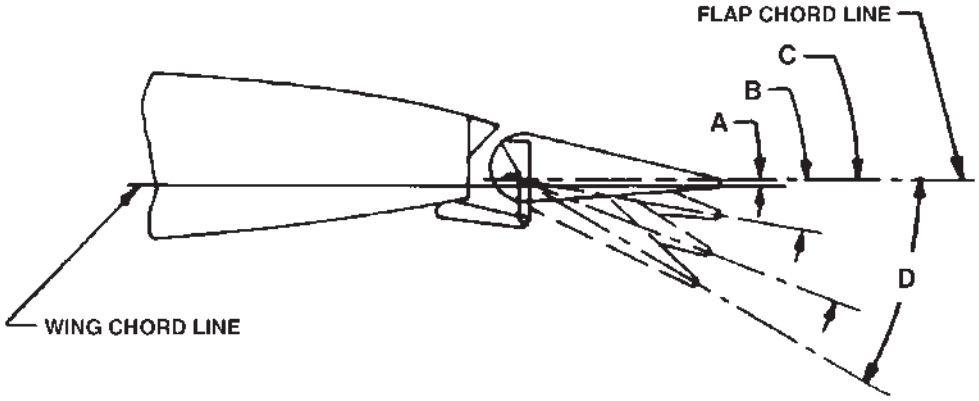
FLAP TRAVEL

A - 0° ± 1° FULL UP

B - 10° ± 2°

C - 25° ± 2°

D - 40° ± 2° FULL DOWN



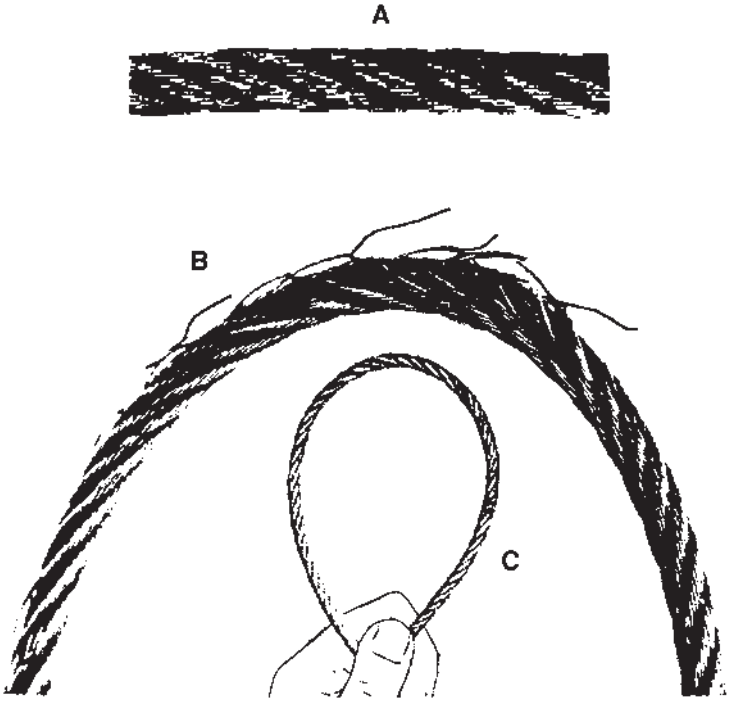
5. Control Cable Inspection

Aircraft control cable systems are subject to a variety of environmental conditions and forms of deterioration that, with time, may be easy to recognize as wire/strand breakage or the not-so-readily visible types of wear, corrosion, and/or distortion. The following data may help in detecting the presence of these conditions:

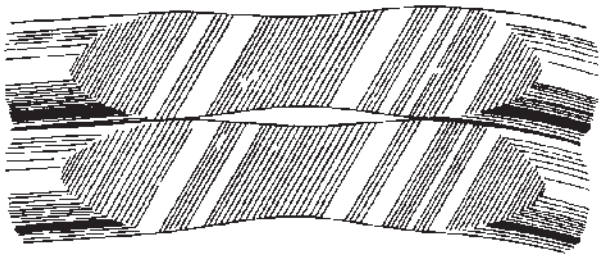
A. Cable Damage

Critical areas for wire breakage are sections of the cable which pass through fairleads and around pulleys. To inspect each section which passes over a pulley or through a fairlead, remove cable from aircraft to the extent necessary to expose that particular section. Examine cables for broken wires by passing a cloth along length of cable. This will clean the cable for a visual inspection, and detect broken wires, if the cloth snags on cable. When snags are found, closely examine cable to determine full extent of damage.

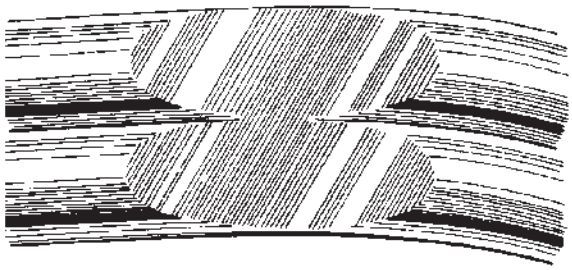
The absence of snags is not positive evidence that broken wires do not exist. “Figure 1”, View A, shows a cable with broken wires that were not detected by wiping, but were found during a visual inspection. The damage became readily apparent (View B) when the cable was removed and bent using the techniques depicted in View C.



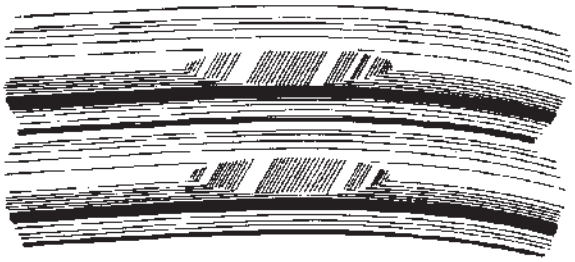
Control Cable Inspection Techniques
Figure 1



INDIVIDUAL OUTER WIRES
WORN MORE THAN 50%



INDIVIDUAL OUTER WIRES WORN
MORE THAN 40 - 50 %
(NOTE BLENDING OF WORN AREAS)

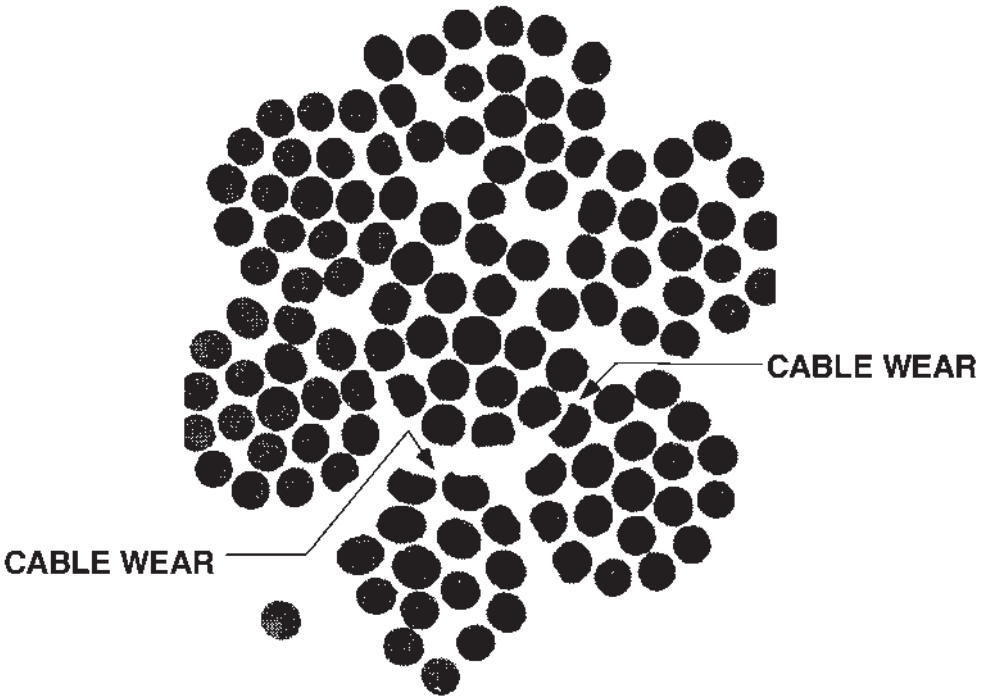


INDIVIDUAL OUTER WIRES WORN LESS THAN 40%
(WORN AREAS INDIVIDUALLY DISTINGUISHABLE)

Cable Wear Patterns
Figure 2

B. External Wear Patterns

Wear will normally extend along cable equal to the distance cable moves at that location. Wear may occur on one side of the cable only or on its entire circumference. Replace flexible and non-flexible cables when individual wires in each strand appear to blend together (outer wires worn 40-50 percent) as depicted in “Figure 2”.



Internal Cable Wear
Figure 3

C. Internal Cable Wear

As wear is taking place on the exterior surface of a cable, the same condition is taking place internally, particularly in the sections of the cable which pass over pulleys and quadrants. This condition, shown in “Figure 3”, is not easily detected unless the strands of the cable are separated. Wear of this type is a result of the relative motion between inner wire surfaces. Under certain conditions the rate of this type wear can be greater than that occurring on the surface.

D. Corrosion

Carefully examine any cable for corrosion that has a broken wire in a section not in contact with wear producing airframe components such as pulleys, fairleads, etc. It may be necessary to remove and bend the cable to properly inspect it for internal strand corrosion as this condition is usually not evident on the outer surface of the cable. Replace cable segments if internal strand rust or corrosion is found.

Areas especially conducive to cable corrosion are battery compartments, lavatories, wheel wells, etc., where concentrations of corrosive fumes, vapors, and liquids can accumulate.

NOTE: Check all exposed sections of cable for corrosion after a cleaning and/or metal-brightening operation has been accomplished in that area.

E. Cable Maintenance

CAUTION: TO AVOID REMOVAL OF CORROSION-PREVENTATIVE COMPOUNDS AND CABLE INTERNAL LUBRICANT, DO NOT USE VAPOR DEGREASING, STEAM CLEANING, METHYLETHYLKETONE (MEK) OR OTHER SOLVENTS.

CAUTION: DO NOT OIL CONTROL CABLES.

Frequent inspections and preservation measures such as rust prevention treatments for bare cable areas will help to extend cable service life. Where cables pass through fairleads, pressure seals, or over pulleys, remove accumulated heavy coatings of corrosion prevention compound. Provide corrosion protection for these cable sections by lubricating as specified in the Lubrication Chart, 12-20-00.

F. Cable Fittings (Terminals, Turnbuckles, etc.)

Clean the fittings thoroughly with a suitable solvent prior to inspection.

NOTE: Most control cables feature strands made of galvanized steel. By design, the galvanized coating corrodes sacrificially, protecting the steel strands underneath. This normal corrosion process creates zinc oxide powder, which can migrate along the length of the control cable, covering the surface of the fittings with an inert white speckled coating. This coating, along with any other contaminants such as dirt, oil or grease, must be removed in order to properly examine the fittings.

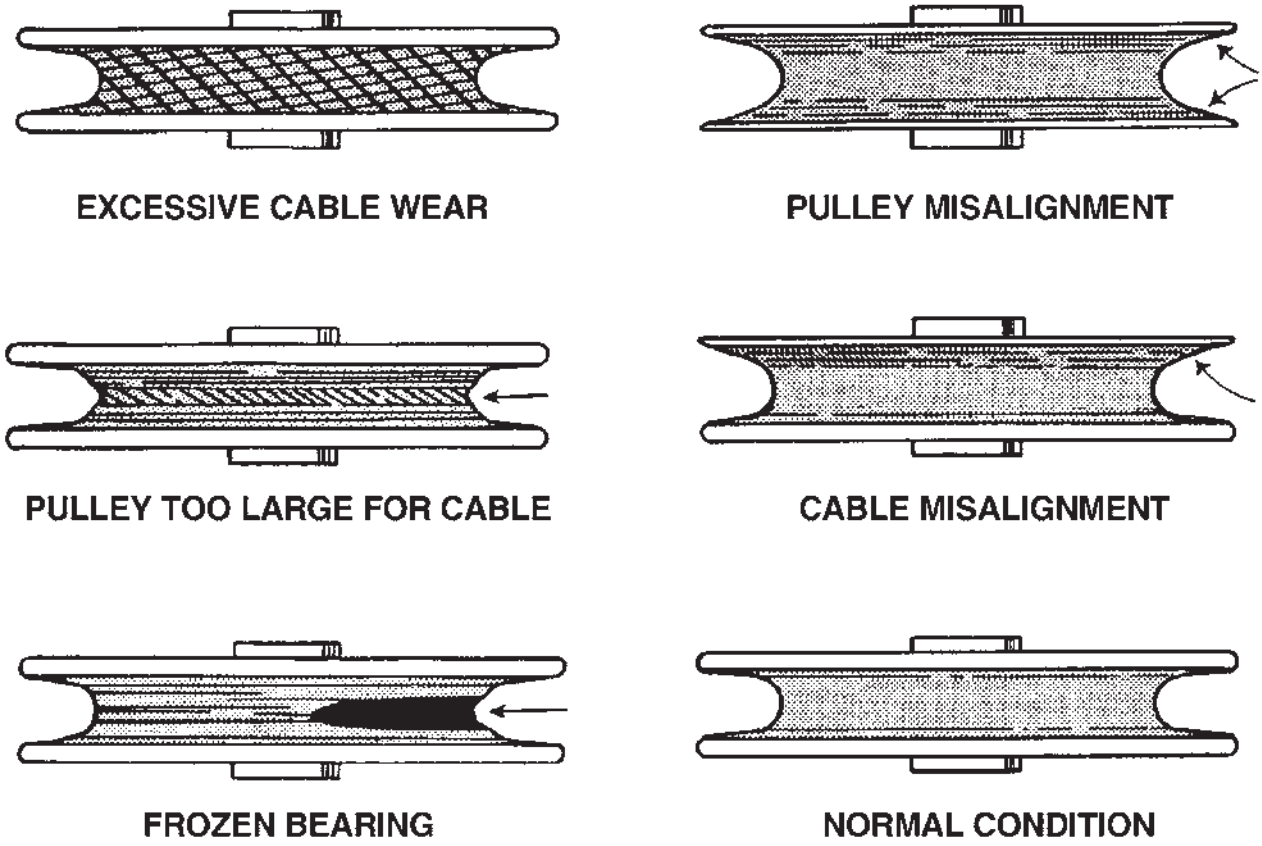
(1) 100 Hour Standard Inspection

Check swaged terminal reference marks for any indication of cable slippage within fitting. Inspect fitting assembly for distortion and/or broken strands at the terminal. Check that all bearings and swivel fittings (bolted or pinned) pivot freely to prevent binding and subsequent failure. Check turnbuckles for proper thread exposure and broken or missing safety wires/clips.

Pay particular attention to corrosion and “pitting” on cable terminals, turnbuckles and cable fittings. Any corrosion or pitting found requires replacement of the corroded fitting and/or cable.

(2) 100 Hour Special Inspection

For airplanes 15 years old or older, using a 10X magnifier, visually inspect the entire surface of each cable terminal, turnbuckle, or other cable fitting for corrosion or cracking. Inspect under safety wire or clips wrapped around the cable or fitting. Any evidence of corrosion or cracking, however minute, is cause for replacement. A logbook entry documenting the replacement of a cable terminal, turnbuckle, or other cable fitting relieves the inspection requirement for that fitting only, until such time as that fitting has been in service for 15 years.



Pulley Wear Patterns
Figure 4

G. Pulleys

Inspect pulleys for roughness, sharp edges, and presence of foreign material embedded in the grooves. Examine pulley bearings to assure proper lubrication, smooth rotation, freedom from flat spots, dirt, and paint spray. Periodically rotate pulleys, which turn through a small arc, to provide a new bearing surface for the cable. Maintain pulley alignment to prevent the cable from riding on flanges and chafing against guards, covers, or adjacent structure. Check all pulley brackets and guards for damage, alignment, and security.

H. Pulley Wear Patterns

Various cable system malfunctions may be detected by analyzing pulley conditions. These include such discrepancies as too much tension, misalignment, pulley bearing problems, and size mismatches between cables and pulleys. Examples of these conditions are shown in “Figure 4”.

I. Stabilator Control System Special Inspection

NOTE: This inspection incorporates the requirements of Piper Service Bulletin No. 1245A.

Beginning at the first regularly scheduled maintenance event after reaching fifteen (15) calendar years time-in-service and each two thousand (2,000) hours or seven (7) calendar years time-in-service thereafter, whichever occurs first, perform the following inspection.

NOTE: Documented replacement of a control cable or cable fitting (i.e., turnbuckle, terminal, etc.) relieves this inspection requirement for that component only until it reaches fifteen (15) calendar years time-in-service.

(1) Secure a jack stand to the tail skid (tie down).

(2) Locate the rear cabin bulkhead, which is behind the aft seats. The rear cabin bulkhead is accessible from inside the aircraft.

NOTE: Remove seats and/or other interior components as desired to ease accomplishment of this inspection.

(3) Remove and retain the access cover from the rear cabin bulkhead, in order to gain access to the tailcone area.

(4) Visually inspect the entire length of the stabilator flight control cable system (see Figure 1, 27-30-00), with special emphasis on the turnbuckle area, as described in the steps that follow.

(5) Identify the two (2) stabilator flight control cable turnbuckles (Item 16 in Figure 1, 27-30-00), which are located inside the tailcone, directly aft of the rear cabin bulkhead near F.S. 176.

(6) Disassemble the turnbuckle bodies to facilitate a complete inspection of the turnbuckle bodies and associated cable terminals.

NOTE: See Standard Procedures, above; and, Removal under Stabilator Controls in 27-30-00.

(7) Clean the turnbuckle bodies and associated cable terminals thoroughly per the instructions provided in Cable Fittings, above.

(8) Using a 10X magnifier, a mirror and a suitable light source, carefully examine the entire surface of each turnbuckle, cable terminal, and adjacent portion of the flight control cable, inspecting for cracks, corrosion, or broken cable strands. Any evidence of cracks or cable fraying, however minute, is cause for replacement. Any evidence of corrosion that remains after accomplishing the cleaning instructions above is cause for replacement.

(9) Reassemble the turnbuckle bodies and associated cable terminals. Rig and adjust stabilator travel and stabilator control cable tension.

NOTE: See Standard Procedures; and, Installation and Rigging and Ajustment under Stabilator Controls in 27-30-00.

(10) Reinstall the rear cabin bulkhead access cover.

NOTE: Reinstall seats and/or other interior components that were removed in Step (2).

(11) Remove the jack stand from the aircraft.

(12) Make a logbook entry indicating compliance with this inspection and documenting any components that were replaced.

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AILERON

1. Troubleshooting

   See “Chart 1”.

2. Control Column

   Refer to “Figure 1”.

   A. Removal

      (1) Remove either control wheel with tube as follows:

         (a) Separate control wheel tube from flexible joint located on either side of tee bar assembly, by removing nut, washer, and bolt. Pull tube from flexible joint.

         (b) To remove left control tube, slide the stop from tube.

         (c) Disconnect autopilot systems wiring in control tube at the quick disconnect terminals behind instrument panel. Draw wires back into the tube and out the tube forward end.

         (d) Remove control wheel assembly from instrument panel.

      (2) Remove tee bar assembly as follows:

         (a) Remove access panel or door to aft section of fuselage.

         (b) Relieve cable tension from stabilator control cables at one of the stabilator cable turnbuckles in the aft section of fuselage.

         (c) Relieve tension from aileron control cables and chains, and at the turnbuckle connecting chains to tee bar top.

         (d) Disconnect control chains from control cables, where the chains and cables join, by removing the cotter pins, nuts, bolts, and bushings.

         (e) Remove tunnel cover as follows:

            1) Remove rudder trim control knob and trim cover attaching screws.

            2) Roll back carpet from tunnel and remove tunnel plate located aft of tee bar assembly, by removing plate attaching screws.

         (f) Remove the two aileron control cable pulleys attached to lower section of tee bar by removing pulley attaching bolt.

         (g) Disconnect stabilator controls from lower end of tee bar assembly.

         (h) Disconnect the mixture control, throttle control, etc., to allow tee bar assembly removal.

         (i) Remove tee bar assembly by removing attaching bolts with washers and nuts (through each side of floor tunnel) and lift it up and out through right side of cabin.

   B. Installation

      (1) Install tee bar assembly as follows:

         (a) Swing tee bar assembly into place from right side of cabin and secure with attaching bolts, washers, and nuts inserted through each side of floor tunnel.

         (b) Connect stabilator controls to lower end of tee bar with bolt, washer, nut, and cotter pin. Cable ends must be free to rotate.

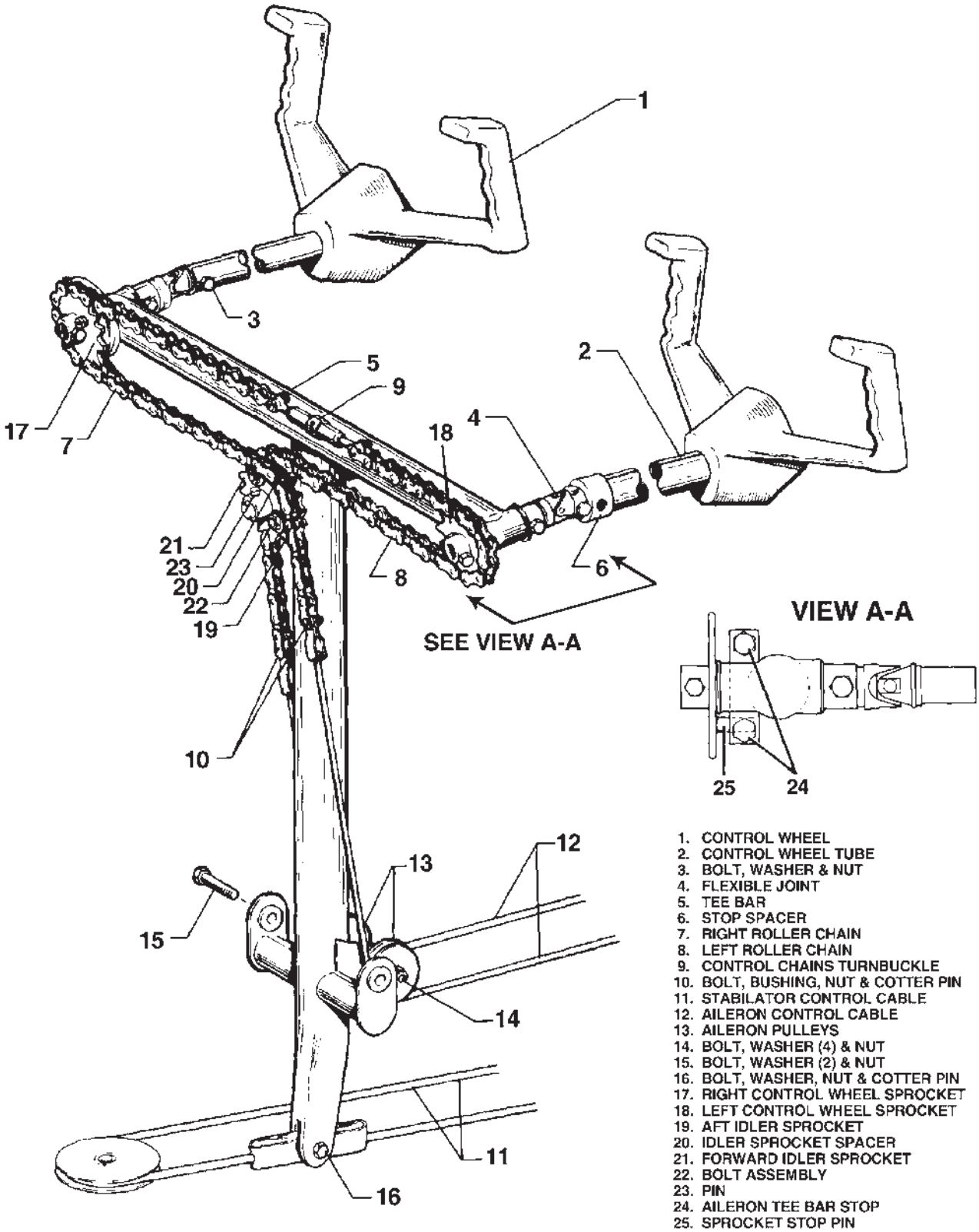
         (c) Place aileron control cables around pulleys that attach to the lower section of tee bar. Position pulleys and secure with bolt, washers, and nut.

         (d) Install control wheel tubes as described below.

         (e) Place control wheels in neutral (centered) position and install aileron control chains on control wheel sprockets and idler crossover sprockets. This turnbuckle must be centered between the two control wheel sprockets.

CHART 1
TROUBLESHOOTING AILERON CONTROL SYSTEM

Trouble	Cause	Remedy
Lost motion between control wheel and aileron.	Cable tension too low.	Adjust cable tension.
	Linkage loose or worn.	Check linkage and tighten or replace.
	Broken pulley.	Replace pulley.
	Cables not in place on pulleys.	Install cables correctly. Check cable guards.
Resistance to control wheel rotation.	System not lubricated properly.	Lubricate system.
	Cable tension too high.	Adjust cable tension.
	Control column horizontal chain improperly adjusted.	Adjust chain tension.
	Pulleys binding or rubbing.	Replace binding pulleys and/or provide clearance between pulleys and brackets.
	Cables not in place on pulleys.	Install cables correctly. Check cable guards.
	Bent aileron and/or hinge.	Repair or replace aileron and/or hinge.
	Cables crossed or routed incorrectly.	Check routing of control cables.
Control wheels not synchronized.	Incorrect control column rigging.	Re-rig control column.
Control wheels not horizontal when ailerons are neutral.	Incorrect rigging of aileron system.	Re-rig aileron system.
Incorrect aileron travel.	Aileron control rods not adjusted properly.	Adjust control rods.
	Aileron bellcrank stops not adjusted properly.	Re-rig controls.
Correct aileron travel cannot be obtained by adjusting bellcrank stops.	Incorrect rigging of aileron cables, control wheel and control rod.	Re-rig controls.
Control wheel stops before control surfaces reach full travel.	Incorrect rigging between control wheel and control cables.	Re-rig controls.



1. CONTROL WHEEL
2. CONTROL WHEEL TUBE
3. BOLT, WASHER & NUT
4. FLEXIBLE JOINT
5. TEE BAR
6. STOP SPACER
7. RIGHT ROLLER CHAIN
8. LEFT ROLLER CHAIN
9. CONTROL CHAINS TURNBUCKLE
10. BOLT, BUSHING, NUT & COTTER PIN
11. STABILATOR CONTROL CABLE
12. AILERON CONTROL CABLE
13. AILERON PULLEYS
14. BOLT, WASHER (4) & NUT
15. BOLT, WASHER (2) & NUT
16. BOLT, WASHER, NUT & COTTER PIN
17. RIGHT CONTROL WHEEL SPROCKET
18. LEFT CONTROL WHEEL SPROCKET
19. AFT IDLER SPROCKET
20. IDLER SPROCKET SPACER
21. FORWARD IDLER SPROCKET
22. BOLT ASSEMBLY
23. PIN
24. AILERON TEE BAR STOP
25. SPROCKET STOP PIN

Control Column Assembly
Figure 1

(f) Loosen connecting bolts of idler sprockets to allow snug fit of chain around control wheel sprockets and over idler sprockets.

(g) Connect aileron control cables to chain ends with bolts, bushings, nuts, and cotter pins.

(h) Adjust chain turnbuckle between the two control wheel sprockets to allow control wheels to be neutral and obtain proper cable tension. To have both control wheels neutral, set chain turnbuckle to neutralize wheels and set cable tension with turnbuckles (under floor panel aft of main spar) per rigging and adjustment of aileron controls. Before safetying turnbuckle, verify when ailerons are neutral, the control wheels are neutral, and chain turnbuckle centered. Aileron bellcranks must contact their stops before control wheel hits its stop. Maintain 0.030 to 0.040 inch clearance between sprocket pin and adjustable stop bolts on models with adjustable aileron tee bar stops.

(i) Set stabilator cable tension with turnbuckle in the aft section of fuselage per instructions in rigging and adjustment of stabilator controls. Complete adjustments and check safety of all turnbuckles.

(j) Tighten the connecting bolts of the idler sprockets. Torque 45 ± 5 in.-lbs.

(k) Install floor tunnel plate trim covers as follows:

1) Position tunnel plate and install and secure with proper screws.

2) Roll carpet into place and install rudder trim cover and knob with a clevis pin and cotter pin.

3) Replace fuel selector and rudder trim knob.

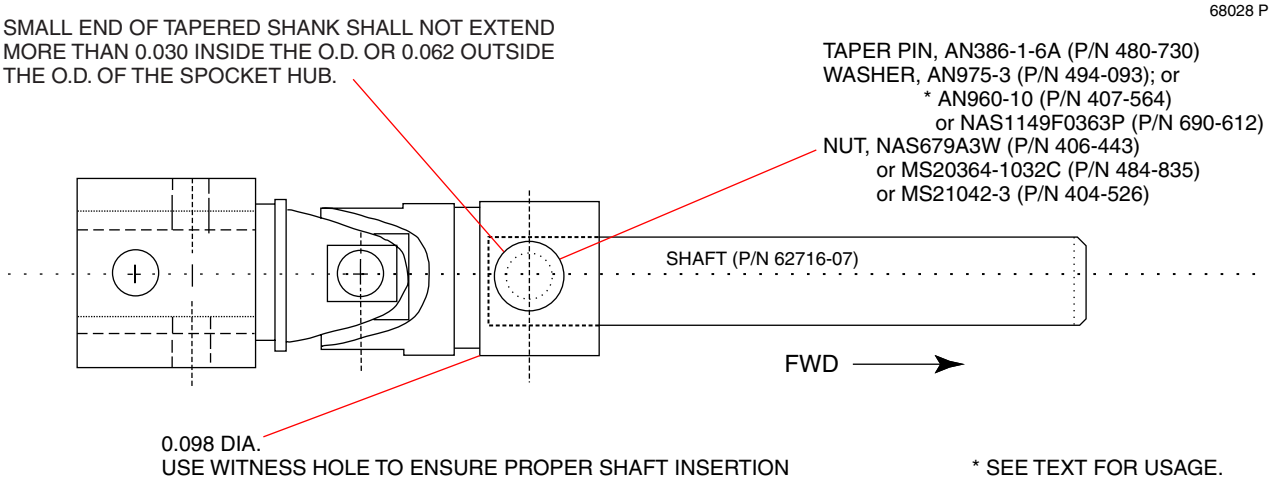
(2) Install either control wheel assembly as follows:

(a) Insert control wheel tube through instrument panel.

(b) Install autopilot systems wiring in the control tube and route the wires through hole in the forward side of tube and out the small hole in forward side. Position rudder grommet in the hole in the side of tube.

(c) Install stop on left control tube.

(d) Connect control wheel tube to the flexible joint of tee bar assembly. If the control cables or chains have not been removed or loosened, place ailerons in neutral and install control tube on the flexible joint to allow control wheel to be neutral. Install and tighten bolt, washer, and nut.



Flex (Universal) Joint Assembly
Figure 2

(3) Flex Joint Replacement (Refer to “Figure 1” and “Figure 2”.)

(a) Carefully lay out location for hole to be drilled in flex joint tube to match hole in control column shaft.

(b) Using a #5 (0.2055) drill bit, drill hole through flex joint tube at location determined, above.

(c) Ream drilled hole, in steps, with a #1 reamer, checking to ensure proper depth for taper pin and sufficient pin thread protrusion for proper installation.

NOTE: Reamer may be purchased from your Piper Dealer as P/N 906-713.

(d) Install pin through tube and shaft.

(e) If pin shoulder does not protrude past sprocket hub, install an AN960-10 washer.

(f) If pin shoulder protrudes past sprocket hub, install an AN975-3 washer.

(g) Install nut. Torque 35 to 40 inch-pounds.

3. Control Wheel (S/N's 2843112 and up.)

See "Figure 3".

The retainer clip (P/N 104687-002) and MS24693-S40 screws (2 ea.) are factory installed in S/N's 2844589 and up. S/N's 2843112 thru 2843588 must individually procure and retrofit the retainer clip and screws.

A. Removal

(1) Remove two (2) screws and retainer clip.
(2) Remove control wheel attachment screw and spacer.
(3) Slide control wheel off of tube.

B. Installation

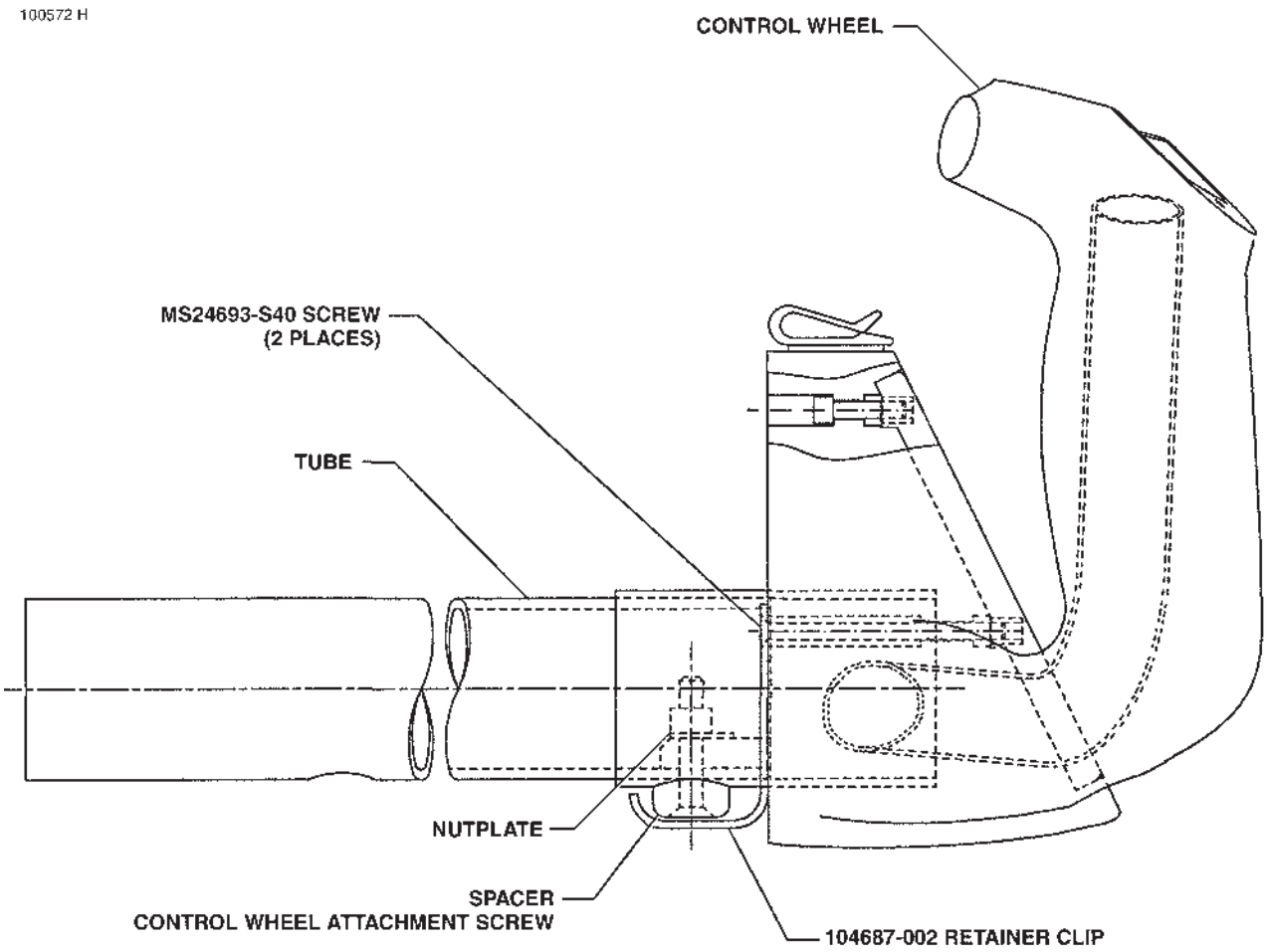
(1) Degrease aft end of tube and inside of control wheel using acetone or naptha. Allow to dry.
(2) Degrease control wheel attachment screw, the spacer, and the nutplate (inside the tube) using acetone or naptha. Allow to dry.
(3) Prime inside of control wheel with Loctite 7649 (Piper P/N 279-073). Allow to dry.
(4) Install control wheel onto tube using Loctite 271 (Piper P/N 279-128). Take care to ensure screw hole in control wheel aligns with screw hole in tube.

NOTE: This step must be accomplished promptly due to short cure time.

(5) Prime threads of control wheel attachment screw with Loctite 7649 (Piper P/N 279-073). Allow to dry.
(6) Install control wheel attachment screw (with spacer) into nutplate using Loctite 271 (Piper P/N 279-128).

NOTE: This step must be accomplished promptly due to short cure time.

(7) Position retainer clip to capture spacer and control wheel attachment screw. Secure with two (2) screws, being careful not to over tighten the screws and damage the logo medallion.





Control Wheel Installation
Figure 3

Effectivity
2843112 and up

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4. Aileron Control Cables

A. Removal

Refer to “Figure 4”.

(1) Remove rear seat and floor panel to access control cables in fuselage or wings.

(2) Remove right or left primary control cables in the fuselage as follows:

(a) Remove the two front seats.

(b) Remove tunnel cover located aft of tee bar assembly as follows:

1) Remove rudder trim control knob and trim cover attaching screws.

2) Roll back carpet from tunnel and remove tunnel plate, located aft of tee bar assembly, by removing the plate attaching screws.

NOTE: To aid in control cable installation mark the cable ends and attach a line before drawing them through fuselage or wing.

(c) Separate primary control cable at the turnbuckle under rear seat or floor panel aft of main spar.

(d) Remove cable pulleys attached to lower section of control column tee bar assembly by removing pulley attaching bolt.

(e) Move cable guard under pulley cluster located just aft of lower portion of tee bar by removing cotter pin from exposed end of guard. Slide guard left or right as required.

(f) Remove cotter pins (used as cable guards) at the pulley in forward area of floor opening aft of main spar.

(g) Disconnect cable from control chain at control column tee bar assembly by removing cotter pin, nut, bolt, and bushing that connect the two. Secure chains to prevent unwrapping from around sprockets.

(h) Pull cable back through floor tunnel.

(3) Remove primary control cable from either wing as follows:

(a) Remove access plate to aileron bellcrank on the underside of wing forward of the inboard end of aileron.

(b) Separate cable at the turnbuckle located aft of the main spar.

(c) Remove cotter pin, nut, washer, and bolt to disconnect cable from forward end of aileron bellcrank.

(d) Pull cable from wing.

(4) Remove either balance cable as follows:

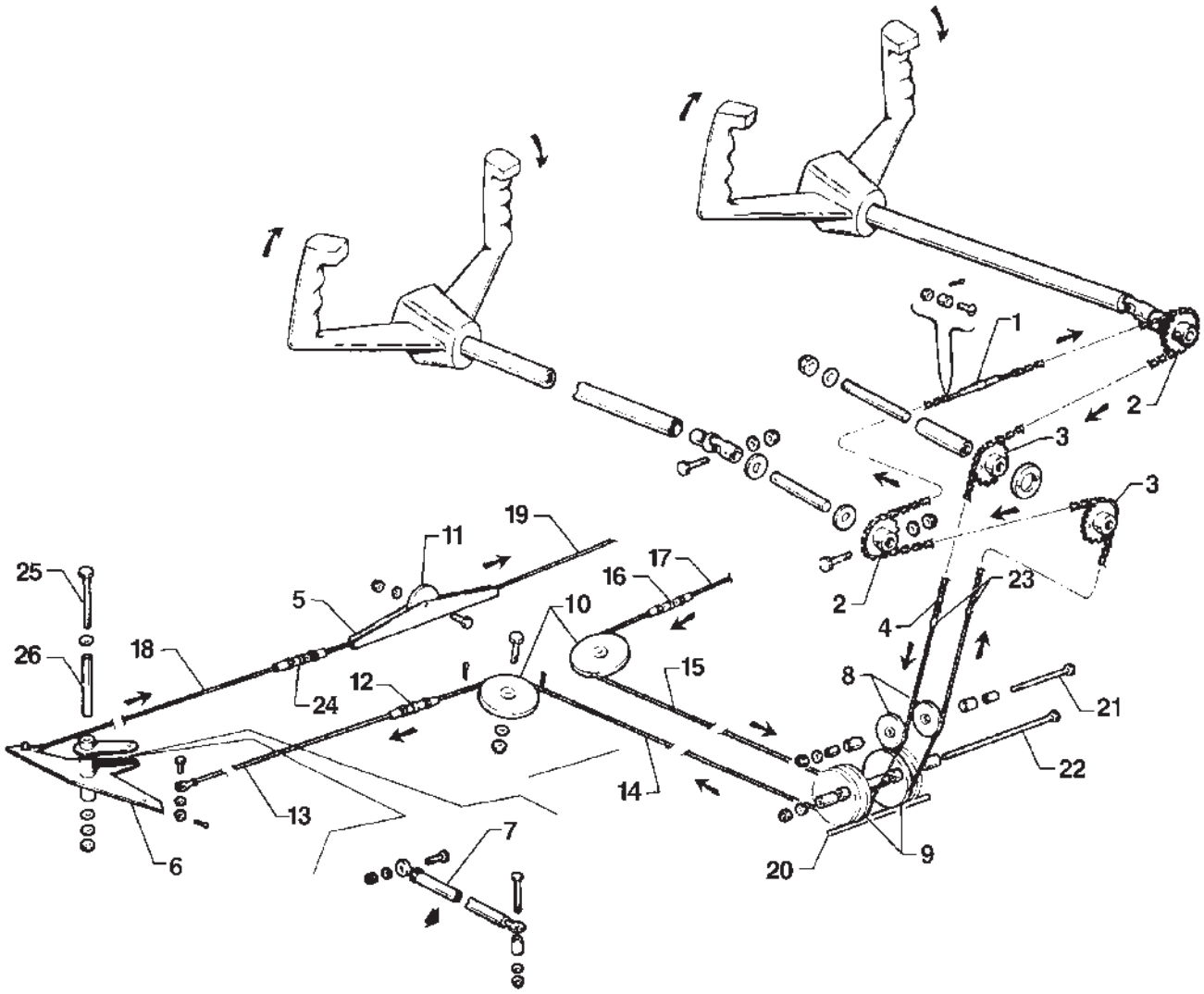
(a) Separate balance cable at turnbuckle in right side of the opening aft of main spar.

(b) If the left balance cable is to be removed, remove cotter pin used as a cable guard at the pulley in the center of the opening.

(c) Remove access plate to aileron bellcrank on the underside of wing forward of the inboard end of aileron.

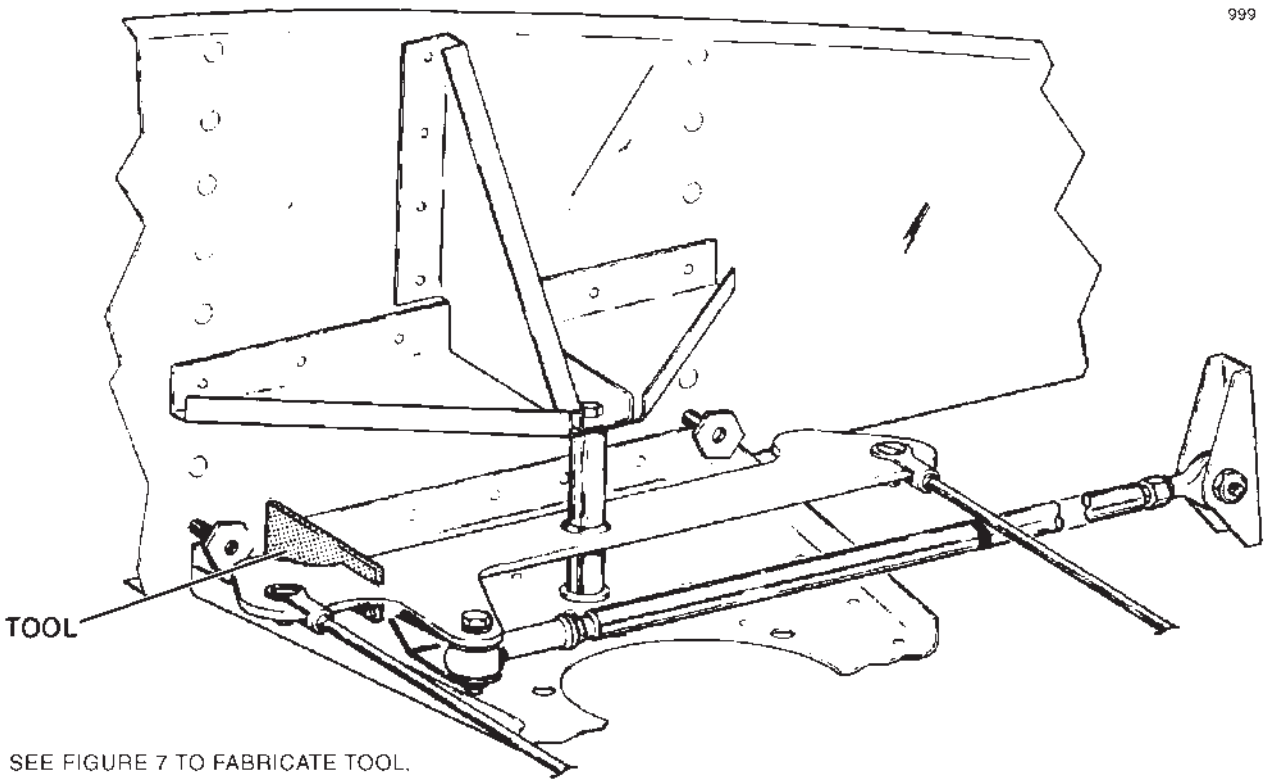
(d) Remove cotter pin, nut, washer, and bolt to disconnect cable from aft end of aileron bellcrank.

(e) Pull cable from wing.



1. CONTROL CHAINS TURNBUCKLE
2. CONTROL WHEEL SPROCKET
3. IDLER SPROCKET
4. AILERON CONTROL CHAIN
5. PULLEY BRACKET
6. AILERON BELLCRANK
7. AILERON CONTROL ROD
8. TEE BAR PULLEY
9. FORWARD CLUSTER PULLEY
10. PRIMARY FORWARD CONTROL CABLE
11. BALANCE CABLE PULLEY
12. RIGHT PRIMARY TURNBUCKLE
13. RIGHT WING PRIMARY CABLE
14. RIGHT FUSELAGE PRIMARY CABLE
15. LEFT FUSELAGE PRIMARY CABLE
16. LEFT PRIMARY TURNBUCKLE
17. LEFT WING PRIMARY CABLE
18. RIGHT BALANCE CABLE
19. LEFT BALANCE CABLE
20. CABLE GUARD ROD
21. BOLT, WASHER & NUT
22. BOLT, WASHER & NUT
23. BOLT, NUT, BUSHING & COTTER PIN
24. BALANCE CABLE TURNBUCKLE
25. BELLCRANK PIVOT BOLT
26. BELLCRANK BUSHING

Aileron Controls (Typical)
Figure 4

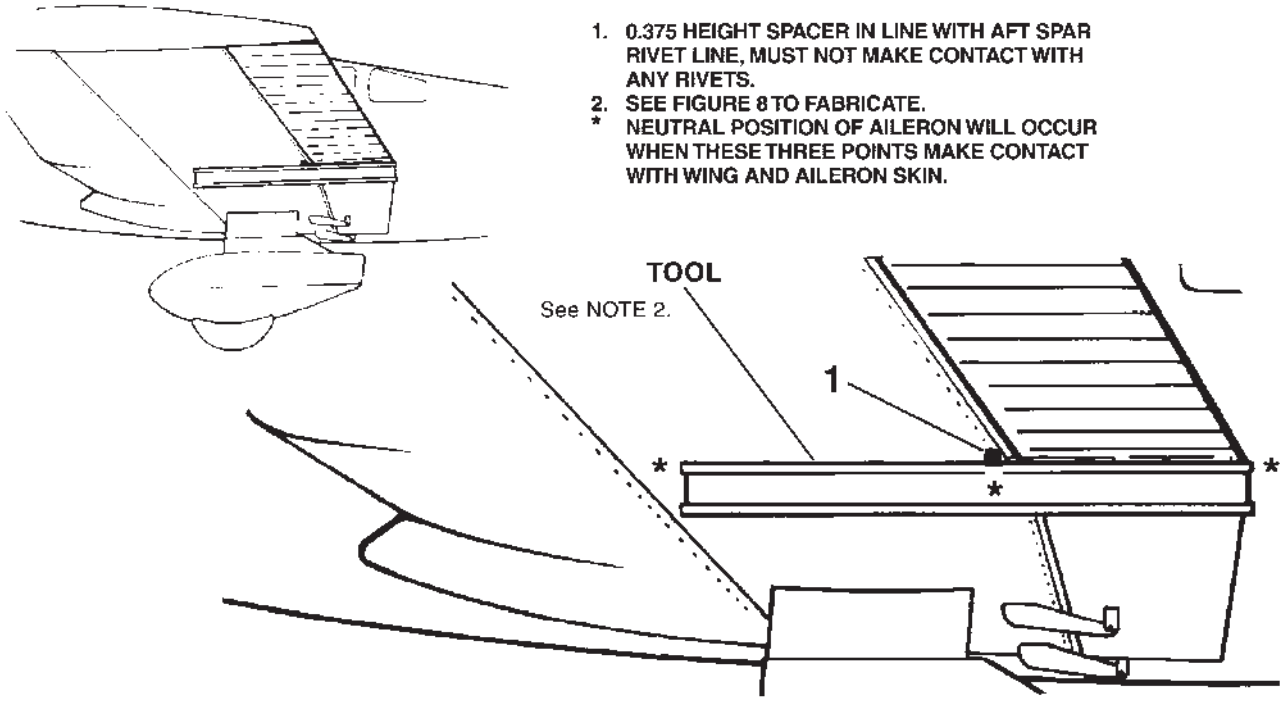


Aileron Bellcrank Rigging (Typical)
Figure 5

B. Installation

Refer to “Figure 4”.

(1) Install right or left primary control cable in fuselage as follows:
  (a) Pull cable through fuselage floor tunnel.
  (b) Connect cable to end of control chain and secure, using bushing, bolt, nut, and cotter pin.
  (c) Place cable around pulley located in the tunnel aft of tee bar. Install cable guard and secure with cotter pin.
  (d) Position cables and install cable pulleys to lower section of tee bar assembly. Secure with bolt, washer, and nut.
  (e) Place cable around pulley in access opening just aft of main spar and install cotter pin cable guards.
  (f) If primary control cable in wing is installed, connect control cable ends at the turnbuckle in the access opening just aft of main spar.
  (g) Check rigging and adjustment per Aileron Controls-Rigging and Adjustment, below.
  (h) Install floor tunnel plate trim covers as follows:
    1) Place tunnel plate into position and secure with attachment screws.
    2) Roll carpet into place and install rudder trim cover and knob.



Rigging Aileron Neutral
Figure 6

(2) Install primary control cable (either wing) as follows:
  (a) Insert control cable into wing.
  (b) Connect cable to forward end of aileron bellcrank, using bolt, washer, nut, and cotter pin. Allow cable end to rotate freely on bellcrank.
  (c) If primary control cable in fuselage is installed, connect the ends at turnbuckle under rear seat aft of main spar.
  (d) Check rigging and adjustment per Aileron Controls-Rigging and Adjustment, below.
  (e) Install access plate on wing underside.

(3) Install either balance cable as follows:
  (a) Insert cable into wing.
  (b) Connect cable to aft end of aileron bellcrank using a bolt, washer, nut, and cotter pin. Allow cable end to rotate freely on bellcrank.
  (c) Connect balance cable ends at the turnbuckle under rear seat aft of main spar.
  (d) If left cable was removed, install cotter pin cable guard at the pulley in fuselage aft of main spar.
  (e) Check rigging and adjustment per Aileron Controls - Rigging and Adjustment, below.
  (f) Install access plate on underside of wing.
  (g) Install floor panel, seat belt attachments, and seats.

5. Aileron Bellcrank Assembly

Refer to “Figure 4”.

A. Removal

(1) Remove rear seats and floor panel.

(2) Remove access plate to aileron bellcrank on underside of wing, forward of aileron inboard end.

(3) Relieve tension from aileron control cables by loosening balance cable turnbuckle in the opening aft of main spar.

(4) Disconnect primary and balance control cables from bellcrank assembly by removing cotter pins, nuts, washers, and bolts.

(5) Disconnect aileron control rod by removing cotter pin, nut, washer, and bolt.

(6) Remove nut (visible from wing underside), pivot bolt, and washers holding bellcrank.

(7) Remove bellcrank from wing.

B. Installation

(1) Check bellcrank pivot bushing is lubricated and install in the torque tube portion of bellcrank.

(2) Place bellcrank in position with a washer located between each end of the torque tube and mounting brackets.

NOTE: In S/N’s 2843591 and up, an additional washer (P/N 494-103) may be installed at the upper end of the torque tube to maintain wing contour.

(3) Install bellcrank pivot bolt with head up. Install washer and nut on bolt, and tighten nut to torque of 20 to 25 inch-pounds. Check bellcrank rotates freely with little up-down play.

(4) Install and adjust control rod and check aileron travel per rigging and adjustment of aileron controls.

(5) Connect ends of primary and balance control cables to bellcrank using bolts, washers, nuts, and cotter pins. Allow cable ends to rotate freely on bellcrank.

(6) Tighten control cables at balance cable turnbuckle in the floor opening aft of main spar. Check cable tension per Aileron Controls - Rigging and Adjustment, below.

(7) Install access plate on wing underside and replace floor panel, seat belt attachments, and seats.

6. Aileron Controls - Rigging and Adjustment

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF AILERONS. WHILE IT WOULD SEEM SELF-EVIDENT, FIELD EXPERIENCE HAS SHOWN THAT THIS CHECK IS FREQUENTLY MISINTERPRETED OR NOT PERFORMED AT ALL. ACCORDINGLY, UPON COMPLETION OF AILERON RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE RIGHT AILERON TRAILING EDGE MOVES UP AND THE LEFT AILERON TRAILING EDGE MOVES DOWN WHEN THE CONTROL WHEEL IS TURNED RIGHT; AND THAT THE LEFT AILERON TRAILING EDGE MOVES UP AND THE RIGHT AILERON TRAILING EDGE MOVES DOWN WHEN THE CONTROL WHEEL IS TURNED LEFT.

NOTE: Complete all rigging corrections. If an out of trim condition persists, it is possible trailing edge of aileron was used to move aircraft forward. This will result in slight bulging of aileron contour at the trailing edge and will cause an out of rig condition.

NOTE: Flap rigging and adjustment must be completed before starting aileron rigging and adjustment. See 27-50-00.

Check and adjust the rigging of the aileron controls as follows:

A. Ascertain that the control chains have been rigged per Control Column, Installation, above.

B. Remove the access plate to each aileron bellcrank located on the underside of the wing, forward of the center aileron hinge by removing the plate attaching screws.

C. Remove the two rear seats, and the front seats if desired.

D. Affix a bellcrank rigging tool (fabricated per dimensions given in “Figure 7”) between the forward arm of each bellcrank and the adjacent rib, as shown in “Figure 5”. The slotted end of the tool fits on the arm forward of and adjacent to the primary control cable end. The other end of the tool is positioned so that the side of the tool contacts the aft side of the bellcrank stop. The bellcrank must be moved to allow a snug fit of the tool between the bellcrank arm and rib. To do so, it may be necessary to loosen a primary control cable or the balance cable. The neutral position may also be found by locating the position at which the forward and aft cable connection holes are an equal distance from the adjacent outboard wing rib.

E. With each bellcrank set at neutral and secured by a suitable tool:

(1) Ascertain that the flaps are rigged in accordance with 27-50-00, and in their up locked position.

(2) Place the control column tee bar in the full forward position and maintain in this position by use of a suitable tool, or by placing weights on the aft side of the stabilator, if the stabilator cables have been previously tensioned.

(3) With the aileron control rod connected between the bellcrank and aileron, check that the trailing inboard edge of the aileron is even with the trailing outboard edge of the flaps. This is the aileron neutral position. (See “Figure 6” for an alternative method.) The ailerons may be allowed to droop by approximately 1/8 (0.125) inch.

(4) Should the two trailing edges (aileron and flap) not align, disconnect the aileron control push rod from the bellcrank and aileron, and loosen the jam nuts at each rod end and adjust the rod ends to obtain trailing edge alignment. Apply a slight up pressure against the trailing edge of the aileron while making this adjustment. Ascertain that there is sufficient thread engagement at both rod ends. Any adjustment should be distributed between both rod ends.

(5) With the control wheels locked in their centered position, adjust the turnbuckles (located in the access opening just aft of the main spar) of the primary and balance cables to their proper cable tension (see 27-00-00, “Chart 2”.). Primary cable tension will be slightly less than balance cable tension, but within the specifications in 27-00-00, “Chart 2”. Adjust cables in such a manner that when the specified tension is obtained, the inboard ends of the ailerons are aligned with the outboard ends of the flaps. Apply a slight up pressure against the trailing edge of the aileron (to take slack out of the hinge and linkage) while making this observation.

F. Remove bellcrank rigging tools and control wheel locks. Check to ensure that the left aileron up and right aileron down (i.e. - bellcrank) stops are contacted simultaneously and vice versa. Adjust stops as required.

G. Rotate the left (pilot’s) control wheel in each direction until the bellcranks contact the stops. The sprocket stops on the tee bar shall not be contacted until additional “override” movement (cushion) of the wheel occurs. A “cushion” of 0.030 to 0.040 inch is to be maintained as measured between the sprocket pin and adjustable stop bolts. (Refer to “Figure 1”.)

H. Place the ailerons in the neutral (aligned with the flaps) position. Adjust “up” and “down” aileron stops to obtain degrees “up” travel and degrees “down” travel of each aileron from the neutral position as specified in 27-00-00, “Chart 2”.

(1) Center bubble of a protractor over surface of aileron at neutral position. Note reading.

(2) Move aileron full up and down. Check degree of travel in each direction. Degree of travel on protractor is determined by taking the difference between protractor reading at neutral and up, and neutral and down. Bubble must be centered at each reading.

When measuring “down” travel from the neutral position a light “up” pressure shall be maintained at the center of the aft edge of the aileron. When measuring “up” travel from the neutral position a light “down” pressure shall be maintained at the center of the aft edge of the aileron (at the “up” position only), just sufficient to remove the slack between the bellcrank and the aileron.

I. Check complete system for operation and safety of turnbuckles, bolts, etc.

J. Install the access plates and panels, and any seats which were removed.

7. Aileron Post-Rigging Inspection

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

Upon completion of any aileron control cable replacement, disconnect, or rigging perform the following postrigging inspection:

A. Rotate the pilot’s control wheel CLOCKWISE, as if to bank the airplane to the RIGHT, until reaching the primary stop. Visually confirm the following:

(1) The RIGHT hand wing aileron is deflected UP. The trailing edge of the aileron will be noticeably and significantly ABOVE the trailing edge of the RIGHT wing at the far RIGHT end of the aileron, near the wing tip.

(2) The LEFT hand wing aileron is deflected DOWN. The trailing edge of the aileron has moved noticeably and significantly BELOW the trailing edge of the LEFT wing at the far LEFT end of the aileron, near the wing tip.

B. Rotate the pilot’s control wheel COUNTERCLOCKWISE, as if to bank the airplane to the LEFT, until reaching the primary stop. Visually confirm the following:

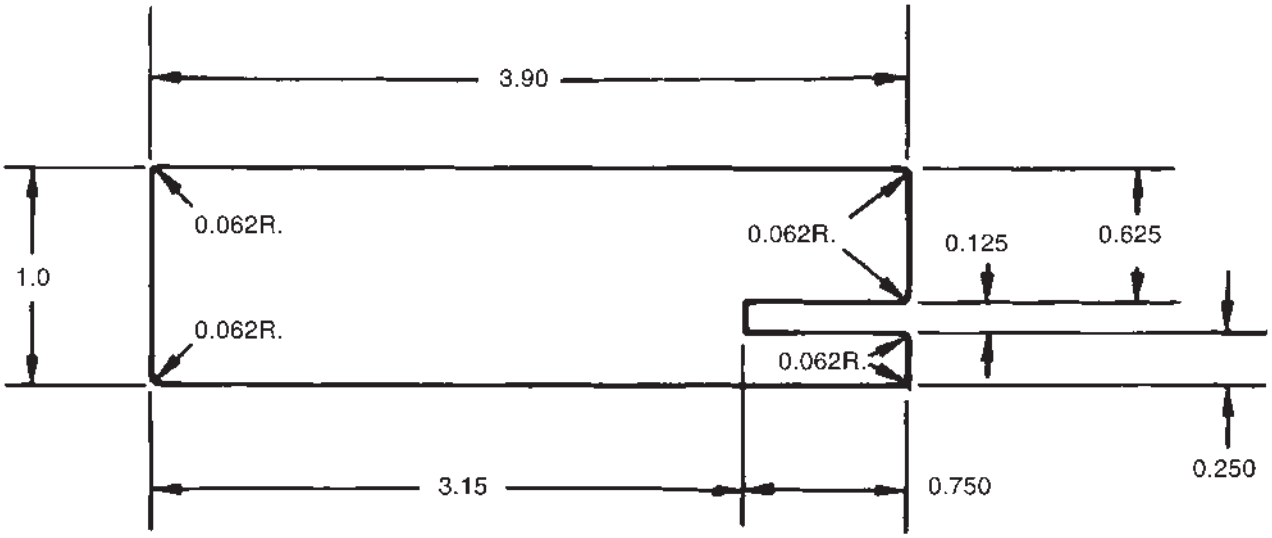
(1) The LEFT hand wing aileron is deflected UP. The trailing edge of the aileron has moved noticeably and significantly ABOVE the trailing edge of the LEFT wing at the far LEFT end of the aileron, near the wing tip.

(2) The RIGHT hand wing aileron is deflected DOWN. The trailing edge of the aileron has moved noticeably and significantly BELOW the trailing edge of the RIGHT wing at the far RIGHT end of the aileron, near the wing tip.

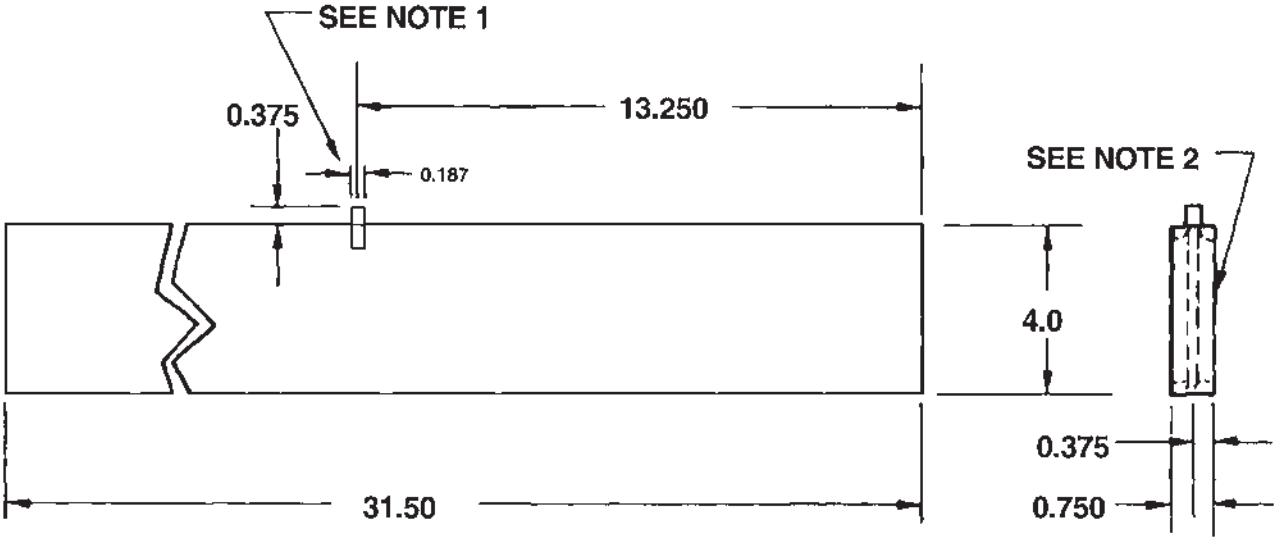
C. Repeat the above steps for the copilot controls.

D. Make a logbook entry documenting completion of this inspection.

MATERIAL:
0.125 X 1.0 ALUMINUM PLATE



Fabricated Aileron Bellcrank Rigging Tool
Figure 7



MATERIAL:
0.750 X 31.50 X 4.00 ALUM BAR OR
0.750 X 31.50 X 0.750 ALUM BAR STOCK (MIN)

NOTES:

1. DRILL AND TAP TO 10-32 NF AN-3 BOLT. JAM NUT AND INTERNAL STAR WASHER MAY BE USED FOR SPACER OR AN-3 BOLT WITH HEAD FILED TO REQUIRED LENGTH.
2. SPAR STOCK MAY BE USED IN PLACE OF ALUM BAR STOCK.

Fabricated Aileron and Flap Rigging Tool
Figure 8

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RUDDER AND TAB

1. Troubleshooting

See “Chart 1” and “Chart 2”.

CHART 1
TROUBLESHOOTING RUDDER CONTROL SYSTEM

Trouble	Cause	Remedy
Lost motion between rudder pedals and rudder.	Cable tension too low.	Adjust cable tension.
	Linkage loose or worn.	Check linkage and tighten or replace.
	Broken pulley.	Replace pulley.
	Bolts attaching rudder to bellcrank are loose.	Tighten bellcrank bolts.
Excessive resistance to rudder pedal movement.	System not lubricated properly.	Lubricate system.
	Rudder pedal torque tube bearing in need of lubrication.	Lubricate torque tube bearings.
	Cable tension too high.	Adjust cable tension.
	Pulleys binding or rubbing.	Replace binding pulleys and/or provide clearance between pulleys and brackets.
	Cables not in place on pulleys.	Install cables correctly. Check cable guards.
	Cables crossed or routed incorrectly.	Check routing of control cables.
Rudder pedals not neutral when rudder is streamlined.	Rudder cables incorrectly rigged.	Re-rig rudder cables.
Incorrect rudder travel.	Rudder bellcrank stop incorrectly adjusted.	Re-rig bellcrank stops.
	Nose wheel contacts stops before rudder.	Re-rig nose wheel stops.

CHART 2
TROUBLESHOOTING RUDDER TRIM CONTROL SYSTEM

Trouble	Cause	Remedy
Trim control knob moves with excessive resistance.	System not lubricated properly.	Lubricate system.

2. Rudder and Steering Pedal Assembly

Refer to “Figure 2”.

A. Inspection

WARNING: ANY CRACKED COMPONENTS DISCOVERED DURING INSPECTION MUST BE REPLACED PRIOR TO NEXT FLIGHT.

In airplanes whose rudder pedal and bar assembly has accumulated 1,200 hours or more time-in-service, conduct the following inspections each 100 hours time-in-service.

NOTE: Installation of a new rudder pedal and bar assembly and a new rudder bar support assembly relieves this recurring inspection requirement, until the new components accumulate 1,200 hours time-in-service.

(1) Gain access to the areas identified in “Figure 1”. Remove interior components as required.

(2) Use a 10X magnifier, a mirror, and a suitable light source or other equipment capable of providing equal or better resolution:

(3) Pedal and Bar Assembly Inspection

Visually inspect all welds, with specific emphasis on the locations identified in “Figure 1”.

(4) Rudder Bar Support Assembly Inspection

See “Figure 1”.

(a) Remove and retain the four (4) bolts that secure the rudder bar support assembly to the cockpit floor.

NOTE: Removing the rudder bar support assembly is not required to accomplish this inspection.

(b) Visually inspect the four (4) bolt holes for cracks and elongation. The diameter of the four bolt holes should measure 0.250 +0.005/-0.001 inches.

(c) Visually inspect the two (2) sheet metal bends common to the base for cracking, the entire length of each bend, both inner and outer surfaces.

(d) Reinstall the four (4) bolts that secure the rudder bar support assembly to the cockpit floor.

(5) Make a logbook entry documenting compliance with this inspection.

B. Removal

(1) Remove access panel to the aft section of fuselage.

(2) Relieve rudder and stabilator cable tension by loosening one of the rudder and stabilator cable turnbuckles in aft section of fuselage.

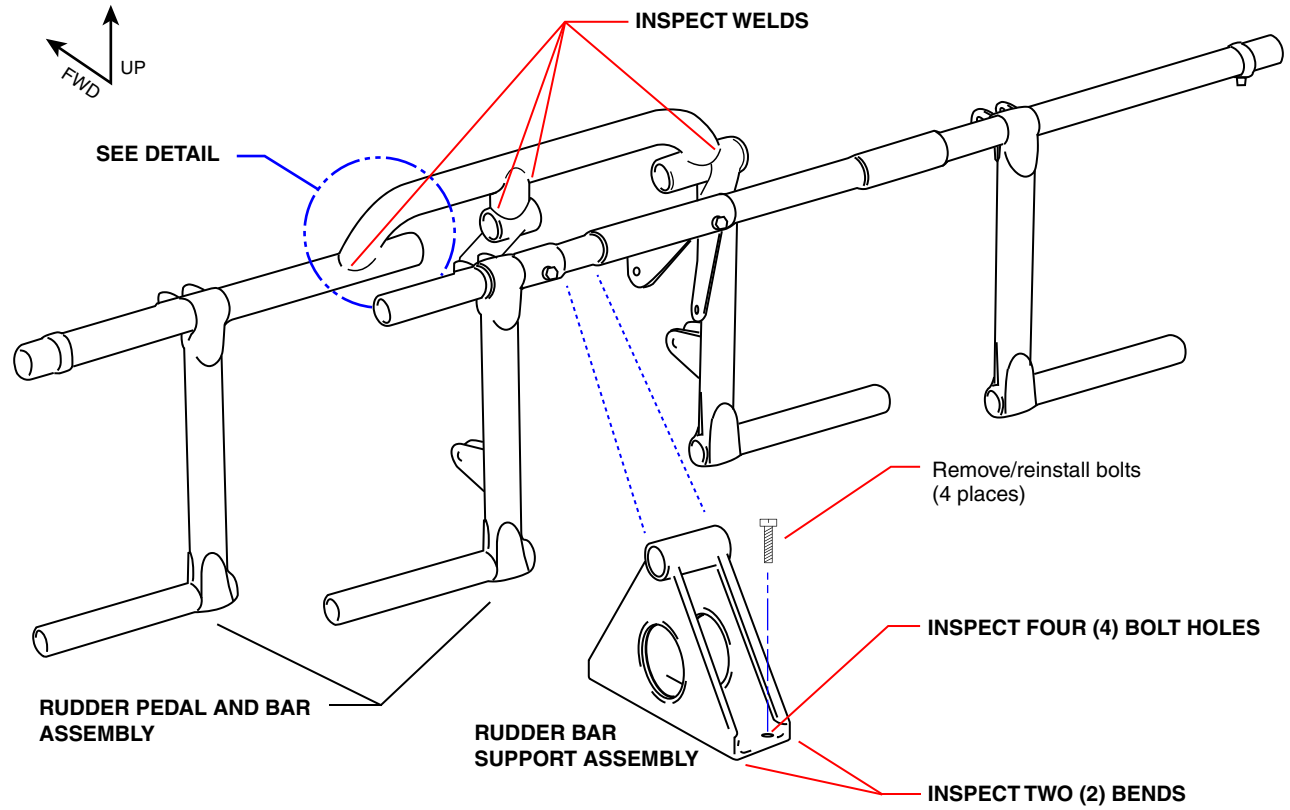
(3) Remove tunnel plate aft of tee bar assembly by removing rudder trim control knob, trim cover attaching screws, and trim cover. Roll back carpet from tunnel and remove tunnel plate aft of tee bar assembly by removing plate attaching screws.

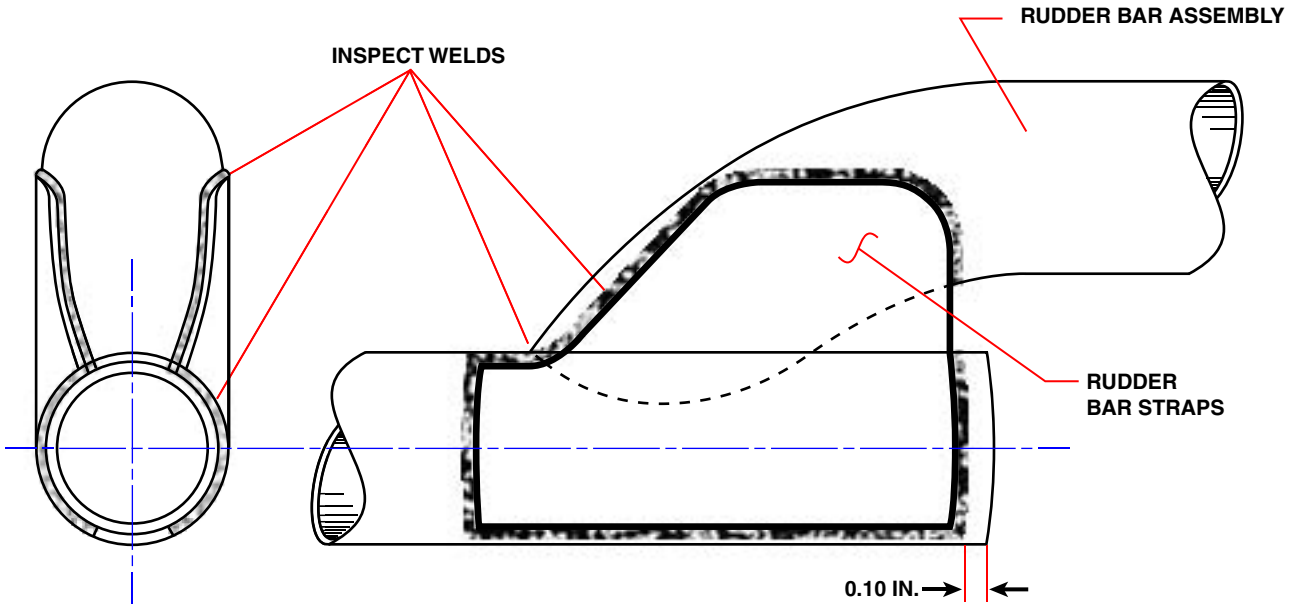
(4) Disconnect stabilator control cable from low end of tee bar assembly.

(5) Remove tee bar attaching bolts (with their washers and nuts) through each side of floor tunnel. Pull lower end of tee bar aft.

(6) Disconnect control cable ends from torque tube arms by removing cotter pins, washers, nuts and bolts.

(7) Disconnect rudder trim from torque tube assembly by removing cotter pin, washers, and the bolt connecting the arm to the trim. Remove cotter pin and clevis pin from rudder trim mechanism and remove mechanism from mounting channel. Remove engine control bracket assembly screw and swing it out of the way. Disconnect alternate air cable and move aside.





DETAIL

Rudder Pedal and Bar Assembly Inspection
Figure 1

(8) Disconnect steering rods at the rudder by removing nuts and bolts.

(9) Disconnect brake cylinders (low end of each cylinder rod) by removing cotter pins, washers, nut, and bolts.

(10) Disconnect vee brace(s) (two braces are used with right hand brakes) from torque tube by removing nuts, washers, and bolts holding strap bracket to vee brace.

(11) Disconnect torque tube support bracket by removing two bolts attached to the box under bracket, and the four bolts attached to forward bulkhead.

(12) Remove two bolts that extend through torque tube at the center of tube assembly over floor tunnel. Compress tubes. Remove left and right toe brake pedal assembly.

(13) Disconnect torque tube support blocks from their support brackets on each side of fuselage by removing attaching nuts, washers, and bolts.

(14) Remove trim side panels.

(15) Rotate rudder pedal bar assembly toward cabin door far enough to pull right pedal bar out. Rotate remaining assembly left and remove assembly. (Note the spacers and washers on each end and between support blocks.)

C. Installation

(1) Assemble torque tube assembly per “Figure 2”. Do not at this time install the two bolts through the center of tube assembly.

(2) Place upper support blocks on the ends of torque tube assembly. A washer is required on each tube end.

(3) Position support blocks on their mounting brackets each side of fuselage and secure with bolts, washers and nuts. A bushing is required in the upper support block bolt holes. A plate is required on top of upper block, between upper and lower blocks, and under block mounting bracket.

(4) Align bolt holes in center area of torque tube assembly. Install and tighten bolts, washers, and nuts.

(5) Position torque tube support bracket on floor tunnel and secure with bolts.

(6) Position vee brace(s) on torque tube. Install strap bracket around torque tube and brace, and secure with bolts, washers and nuts.

(7) Connect the ends of the brake cylinder rods and clevis rods to the idler arms. Secure with clevis and cotter pins.

(8) Connect steering rods to rudder pedals and secure with bolts and nuts. Check steering rod adjustment per Alignment, 32-20-00.

(9) Connect rudder trim to torque tube arm. Secure with bolt, washer, nut, and cotter pin. A thin washer is under the nut which is tightened only finger tight.

(10) Connect rudder control cables ends to torque tube arms. Secure with bolts, washers, nuts, and cotter pins. Allow ends to rotate freely.

(11) Swing tee bar into place, secure with attachment bolts, washers, and nuts. Insert bolts through each side of floor tunnel. (See “Figure 2”.)

(12) Connect stabilator control cables to lower end of tee bar with bolt, washer, and nut. Secure with cotter pin. (See “Figure 2”.) Allow cable ends to rotate freely.

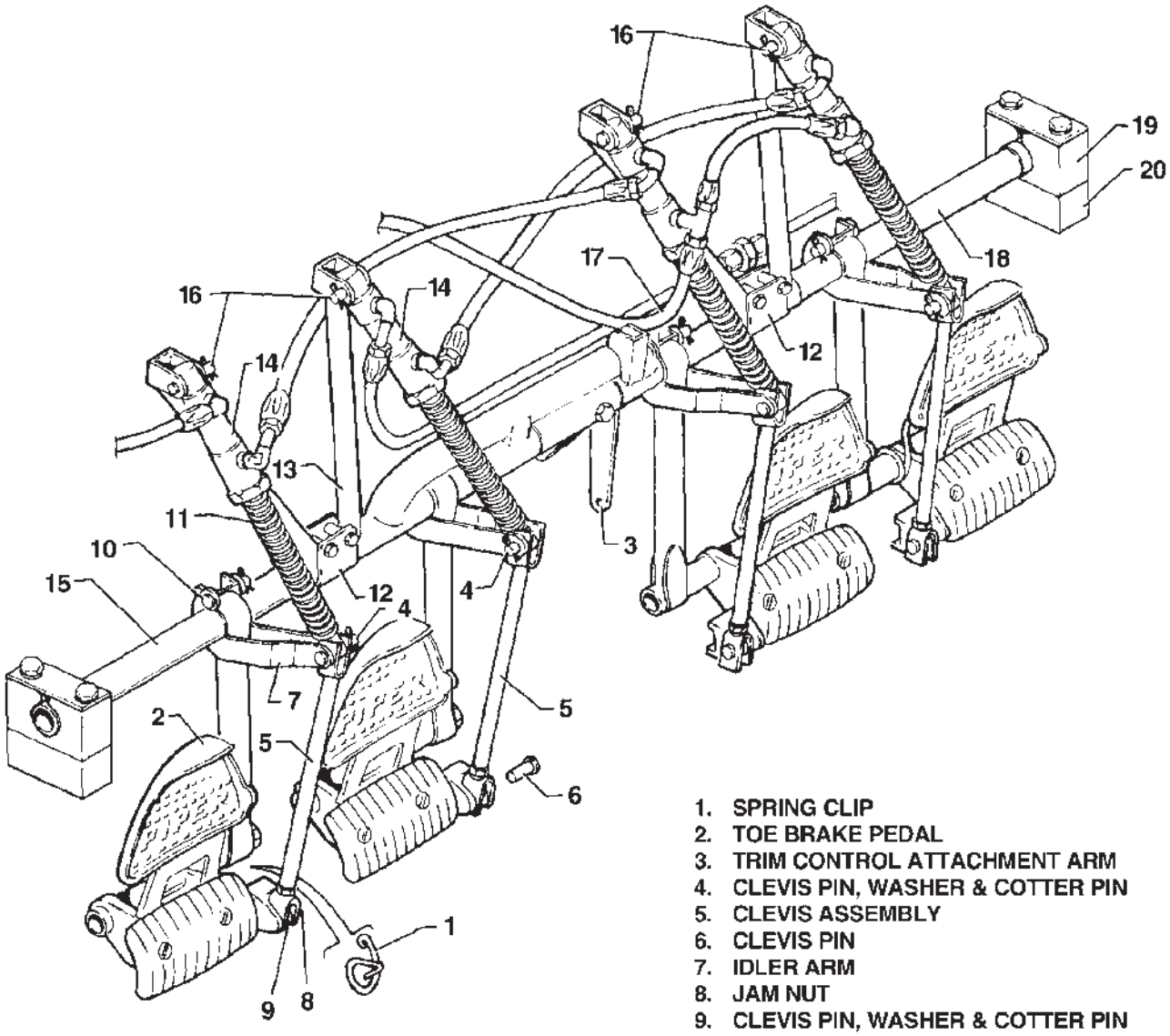
(13) Set rudder cable tension and check “Rudder Controls - Rigging and Adjustment” on page 27208.

(14) Set stabilator cable tension and check rigging and adjustment per “Stabilator Controls - Rigging and Adjustment” on page 27305.

(15) Check aileron cable tension per “Chart 2” on page 27004.

(16) Check safety of bolt and turnbuckles.

(17) Install floor tunnel plate and secure with screws. Fasten tunnel carpet in place.



1. SPRING CLIP
2. TOE BRAKE PEDAL
3. TRIM CONTROL ATTACHMENT ARM
4. CLEVIS PIN, WASHER & COTTER PIN
5. CLEVIS ASSEMBLY
6. CLEVIS PIN
7. IDLER ARM
8. JAM NUT
9. CLEVIS PIN, WASHER & COTTER PIN
10. CLEVIS PIN, WASHER & COTTER PIN
11. RETURN SPRING
12. BRACKET
13. BRACE ASSEMBLY
14. HYDRAULIC CYLINDER ASSEMBLY
15. LEFT TUBE ASSEMBLY
16. CLEVIS PIN & COTTER PIN
17. FLEXIBLE HOSE ASSEMBLY
18. RIGHT TUBE ASSEMBLY
19. UPPER SUPPORT BLOCK
20. LOWER SUPPORT BLOCK

Rudder Controls and Steering Pedal Assembly
Figure 2

(18) Install rudder trim cover and control knob.

(19) Install access to aft section of fuselage.

3. Rudder Control Cables

Refer to “Figure 3”.

A. Removal

(1) Remove access panel to the aft section of fuselage to remove the forward or aft rudder cables.

(2) Disconnect cable at the turnbuckle in aft section of fuselage.

(3) Remove either forward rudder cable as follows

(a) Remove rear seat, floor panel, and front seats.

(b) Remove cable guard pin from underside of pulley cluster in the aft area of flap torque tube.

(c) From within the area aft of the main spar, remove block attaching screws to remove cable rub blocks attached to spar housing.

(d) Remove rudder trim knob and cover attaching screws.

(e) Remove tunnel plate aft of tee bar by removing enough carpet from tunnel to allow plate attaching screws and plate to be removed.

(f) Remove cotter pin from the exposed end and slide it left or right to move cable guard pin, under pulley cluster aft of tee bar.

(g) Remove cotter pin, nut, washer, and bolt to disconnect the cable end from the arm on rudder pedal torque tube.

(h) Pull cable from floor tunnel.

(4) Remove aft rudder control cables as follows:

(a) Remove tail cone fairing by removing attaching screws.

(b) Remove cotter pin, nut, washer, and bolt to disconnect cable from rudder horn.

(c) Pull cable through fuselage.

B. Installation

(1) Install forward rudder control cables as follows:

(a) Pull control cable through floor tunnel.

(b) Install bolt, washer, nut, and cotter pin to connect cable end to the arm on rudder pedal torque tube. (Allow cable end to rotate freely on the arm.)

(c) Connect cable to aft control cable at the turnbuckle in aft fuselage. If the aft control cables are not installed, install at this time per instructions below Check each cable is in pulley groove.

(d) Move cable guard into position, in the forward area of tunnel, under pulley cluster and secure with cotter pin.

(e) Within the area aft of the main spar, install cable guard blocks onto spar housing and secure with screws.

(f) Install cable guard under pulley cluster aft of flap torque tube.

(g) Set cable tension and check rigging and adjustment per Rudder Controls-Rigging and Adjustment, below.

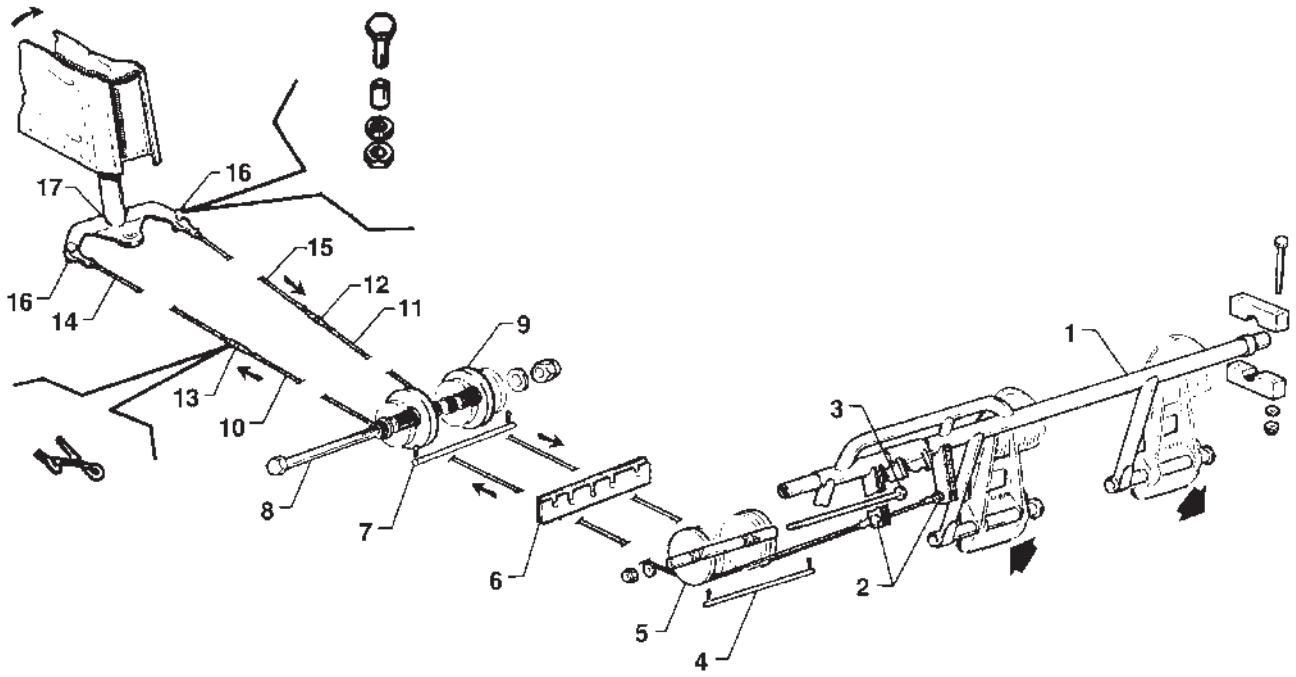
(h) Install forward tunnel plate aft of tee bar and secure with screws.

(i) Put floor carpet in place and secure.

(j) Install lower and upper selector covers and secure with screws.

(k) Install floor panel, rear seat, and front seats.

(2) Install the aft rudder control cables as follows:
  (a) Position control cable in fuselage.
  (b) Connect cable end to rudder horn with bolt, washer, nut, and cotter pin. Allow cable end to rotate freely.
  (c) Connect the other cable end to forward control cable at the turnbuckle in aft fuselage.
  (d) Set cable tension and check rigging and adjustment per rigging and adjustment of rudder controls.
  (e) Install tail cone fairing and secure with screws.
(3) Install cabin rear closeout panel to the aft section of fuselage.



1. RUDDER & STEERING PEDAL ASSEMBLY
2. BOLT, WASHER, NUT & COTTER PIN
3. BOLT, BUSHINGS, WASHER, & NUT
4. CABLE GUARD PIN
5. PULLEY CLUSTER
6. RUB BLOCKS
7. CABLE GUARD PIN
8. BOLT, BUSHINGS, WASHER, & NUT
9. PULLEY CLUSTER
10. RIGHT FORWARD CABLE
11. LEFT FORWARD CABLE
12. LEFT TURNBUCKLE
13. RIGHT TURNBUCKLE
14. RIGHT AFT CABLE
15. LEFT AFT CABLE
16. BOLT, BUSHING, WASHER, & NUT
17. RUDDER HORN

Rudder Controls
Figure 3

4. Rudder Controls - Rigging and Adjustment

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF RUDDER. WHILE IT WOULD SEEM SELF-EVIDENT, FIELD EXPERIENCE HAS SHOWN THAT THIS CHECK IS FREQUENTLY MISINTERPRETED OR NOT PERFORMED AT ALL. ACCORDINGLY, UPON COMPLETION OF RUDDER RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE RUDDER TRAILING EDGE MOVES RIGHT WHEN THE RIGHT PEDAL IS DEPRESSED; AND, THAT THE RUDDER TRAILING EDGE MOVES LEFT WHEN THE LEFT PEDAL IS DEPRESSED.

Inspect, rig and adjust rudder controls as follows:

NOTE: Raise nose wheel off of the ground when rigging the rudder. (See 7-10-00.)

A. Remove tail cone fairing and cabin rear closeout panel (to access the aft section of fuselage).

B. Check rudder travel by swinging rudder until it contacts its stop. If control cables are connected, use rudder pedals to swing rudder.

C. With rudder against its stop, place rigging tool against the side of the rudder and vertical stabilizer as shown in “Figure 4”. (Make tool from dimensions in “Figure 9”.) Verify tool is not contacting rivets. If no gap exists between rigging tool and rudder surface, and vertical stabilizer, the rudder stop for one direction of travel is correct per “Chart 2” on page 27004.

D. Swing rudder in the other direction and check travel as directed above.

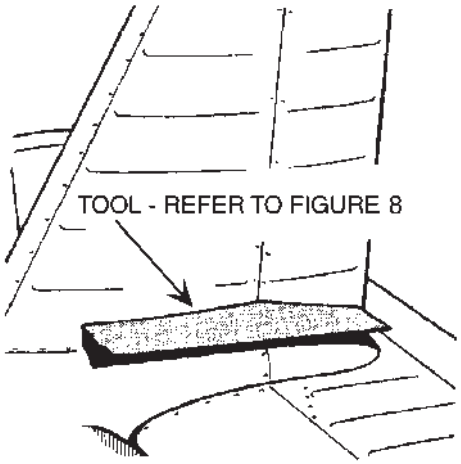
E. If rudder travel is incorrect (showing a gap between the tool and any part of the control surfaces), adjust rudder stops to correct rudder travel. (See per “Chart 2” on page 27004 and “Figure 5”.)

F. Align rudder in its neutral position (i.e. streamlined with vertical fin). Using a suitable tool, lock it in that position.

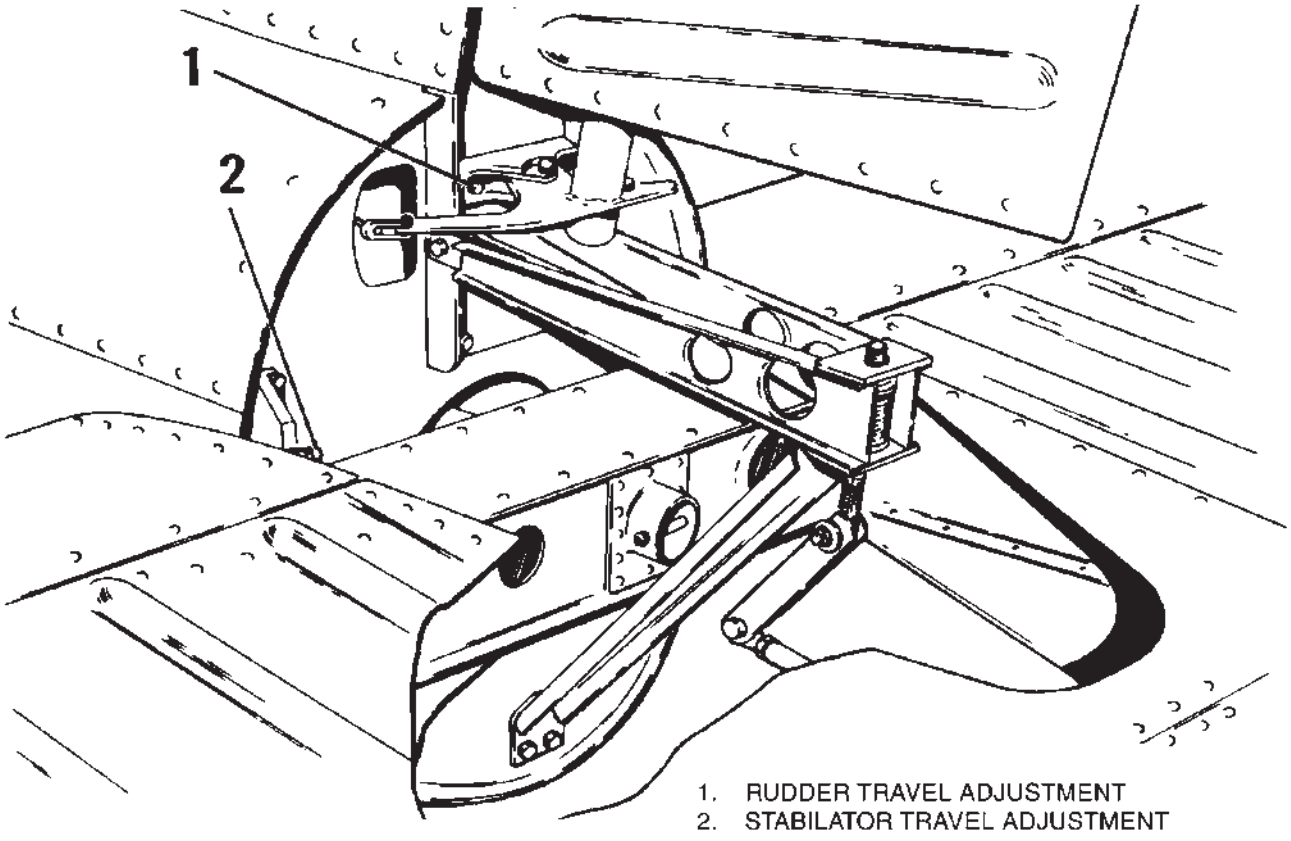
G. Clamp rudder pedals to align them with each other and in their neutral position as shown in “Figure 6” and “Figure 7”.

H. Align the nose wheel with the longitudinal axis of the airplane. Using a suitable tool, lock it in that position.

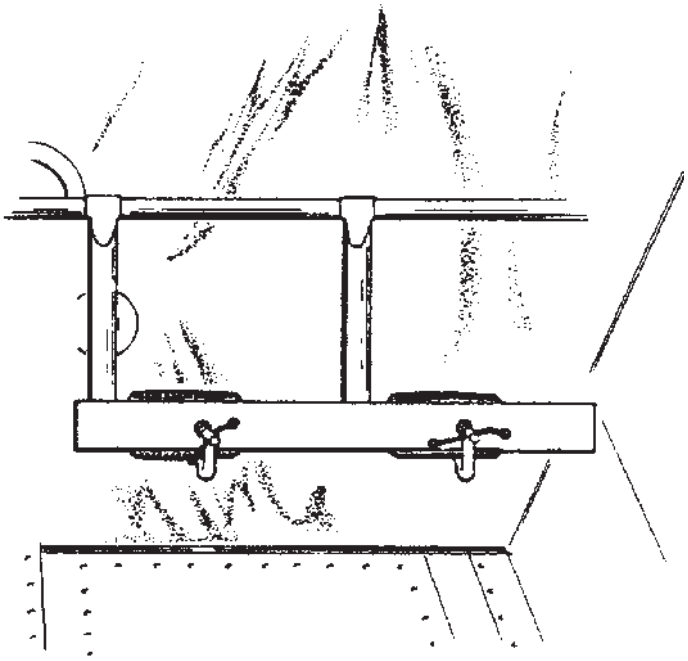
NOTE: This may require disconnecting and adjusting the nose wheel steering rod rod ends. See 32-20-00, Alignment of Nose Gear.



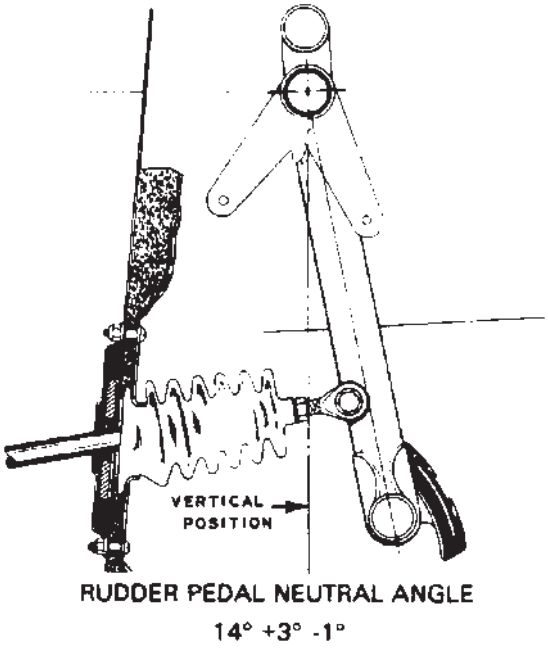
Rudder Rigging Tool
Figure 4



Rudder Travel Adjustment
Figure 5



Clamping Rudder Pedals
Figure 6



Rudder Pedals at Neutral
Figure 7

I. Adjust rudder cable turnbuckles in aft section of fuselage for proper cable tension as specified in “Chart 2” on page 27004. Adjust cables evenly to avoid uneven strain on aircraft components.

J. Check safety of turnbuckles and install all pulley guard pins.

CAUTION: DO NOT PUSH RUDDER HARDER THAN NECESSARY, TO AVOID CABLE STRETCH.

K. Remove centering tools.

L. Turn the nose wheel to the left until the rudder stop makes contact. Adjust nose wheel stop to provide .06 to .12 inch clearance. Repeat with nose wheel to the right. Nose wheel travel is 20° left, 20° right (± 1°).

M. Check complete system for free and correct operation.

N. Install tail cone fairing and cabin rear closeout panel.

5. Rudder Post-Rigging Inspection

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

NOTE: This inspection may need to be performed with the airplane on jacks.

Upon completion of any rudder control cable replacement, disconnect, or rigging perform the following postrigging inspection:

A. Depress the pilot’s RIGHT rudder pedal as if to yaw the airplane to the RIGHT, until reaching the primary stop. Visually confirm the following:

The trailing edge of the rudder will be noticeably and significantly RIGHT of the trailing edge of the vertical fin.

B. Depress the pilot's LEFT rudder pedal as if to yaw the airplane to the LEFT, until reaching the primary stop. Visually confirm the following:

The trailing edge of the rudder will be noticeably and significantly LEFT of the trailing edge of the vertical fin.

C. Repeat the above steps for the copilot controls.

D. Make a logbook entry documenting completion of this inspection.

6. Rudder Trim Controls

A. Troubleshooting

See "Chart 2".

B. Removal

Refer to "Figure 8".

(1) Remove attaching screws to remove cover from trim control assembly.

(2) Remove the rudder trim knob and cover attaching screws.

(3) Rotate trim knob to the extreme left (counterclockwise) trim position.

(4) Remove cotter pin, nut, washer, and bolt to disconnect housing lug from arm on rudder pedal torque tube.

(5) Remove cotter pin and clevis pin to remove threaded bushing from the aft end of mounting channel. (Some mounting channels have two holes in the aft end; note from which hole the clevis pin was removed.)

(6) Remove channel attaching screws at the inside of the channel to remove mounting channel.

C. Installation

Refer to "Figure 8" on page 272012.

Install rudder trim mechanism and set it at neutral (no load on spring) position. Perform the procedure only after all rudder and nose wheel rigging is complete.

(1) Install trim control mounting channel on upper side of floor tunnel. A spacer plate (some models) is between channel and tunnel. Install attaching screws secured with anchor nuts.

(2) Check clips are installed so safety wire is on top before attaching assembly to mounting channel. Check threaded bushing is on assembly shaft with the welded attachment bushing forward or toward housing.

(3) Attach housing lug to rudder pedal torque tube arm and secure with bolt, washer, and nut. Tighten nut finger tight and safety with cotter pin.

(4) Clamp rudder pedals in neutral and position threaded bushing in mounting channel. Turn control shaft until holes in bushing and channel align, and install clevis pin and cotter pin. If two through holes are in the aft end of the mounting channel, the pin must be installed through the hole that will give equal travel and hit rudder stops before bottoming out of the trim assembly.

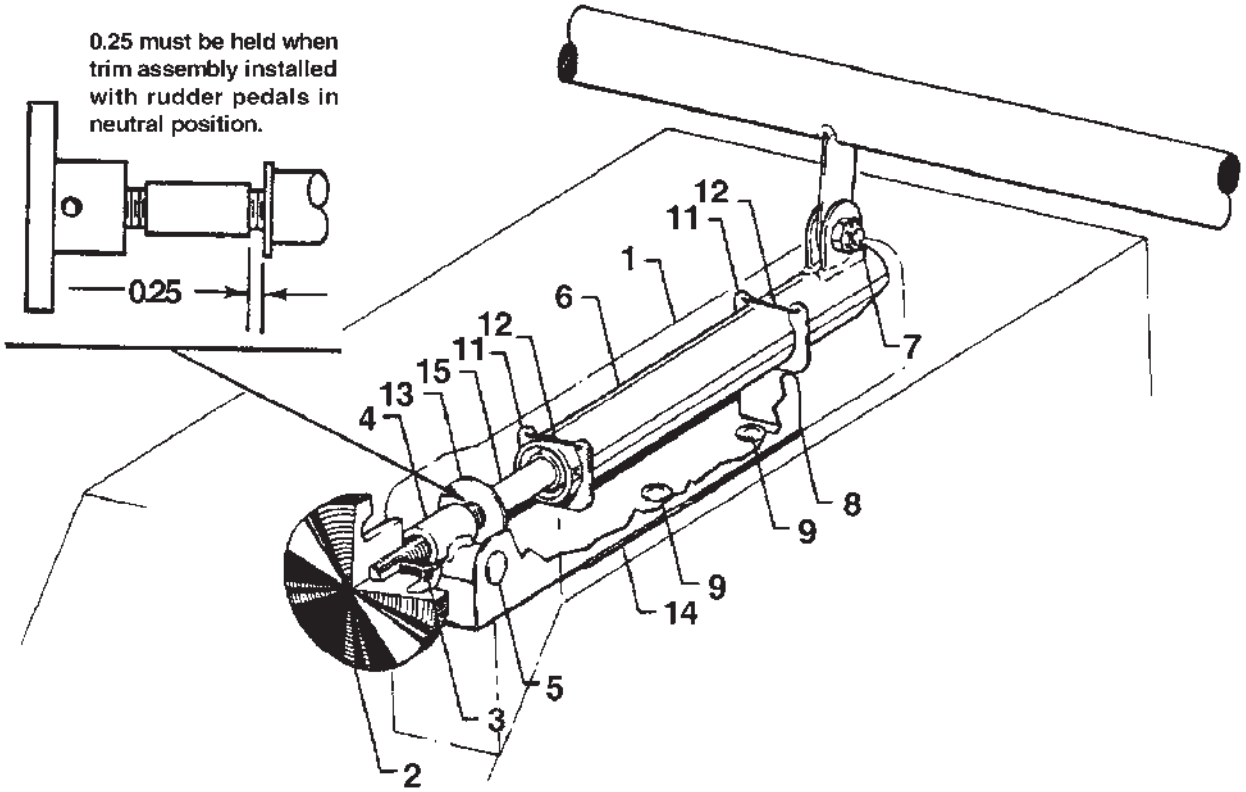
(5) With the rudder pedals neutral and no pressure fore or aft on the clevis pin, install the assembly cover so that the indicator washer and the neutral mark on the cover align.

(6) Install the trim cover, secure with screws, and install the trim control knob.

D. Rigging and Adjustment

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF RUDDER. WHILE IT WOULD SEEM SELF-EVIDENT, FIELD EXPERIENCE HAS SHOWN THAT THIS CHECK IS FREQUENTLY MISINTERPRETED OR NOT PERFORMED AT ALL. ACCORDINGLY, UPON COMPLETION OF RUDDER TRIM RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE RUDDER TRAILING EDGE MOVES LEFT WHEN THE RUDDER TRIM KNOB IS TRIMMED NOSE LEFT; AND, THAT THE RUDDER TRAILING EDGE MOVES RIGHT WHEN THE RUDDER TRAILING EDGE TRIM KNOB IS TRIMMED NOSE RIGHT.

Perform these procedures only after all rudder and nose wheel rigging is complete. No adjustments are necessary other than those required during installation of the assembly per installation of rudder trim controls.



1. COVER
2. RUDDER TRIM CONTROL KNOB
3. LOCK SCREW
4. BUSHING
5. CLEVIS PIN
6. CONTROL SHAFT HOUSING
7. BOLT ASSEMBLY
8. MOUNTING CHANNEL
9. SCREW
11. CLIPS
12. SAFETY WIRE
13. INDICATOR WASHER
14. SPACER PLATE
15. CONTROL SHAFT

Rudder Trim Control
Figure 8

E. Rudder Trim Control Post-Rigging Inspection

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

NOTE: This inspection may need to be performed with the airplane on jacks.

Upon completion of any rudder control cable replacement, disconnect, or rigging perform the following postrigging inspection:

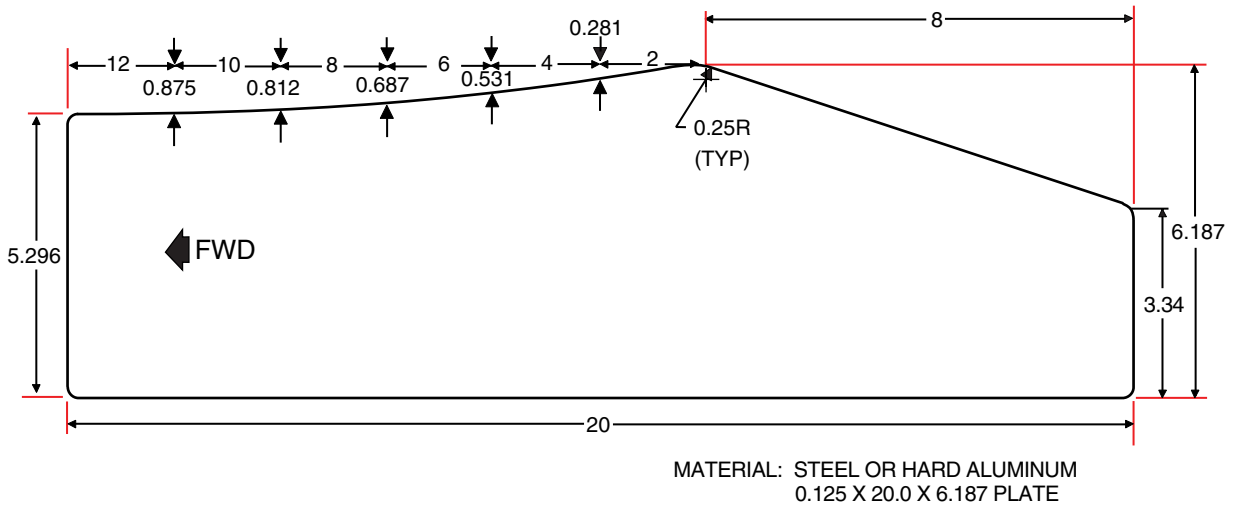
(1) When rudder trim knob is trimmed left. Visually confirm the following:

The rudder trailing edge moves left.

(2) When rudder trim knob is trimmed right. Visually confirm the following:

The rudder trailing edge moves right.

(3) Make a logbook entry documenting completion of this inspection.



Fabricated Rudder Rigging Tool
Figure 9

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STABILATOR AND TAB

1. Troubleshooting

See “Chart 1” and “Chart 2”.

CHART 1
TROUBLESHOOTING STABILATOR CONTROL SYSTEM

Trouble	Cause	Remedy
Lost motion between control wheel and stabilator.	Cable tension too low.	Adjust cable tension.
	Linkage loose or worn.	Check linkage and tighten or replace.
	Broken pulley.	Replace pulley.
	Cables not in place on pulleys.	Install cables correctly.
Resistance to stabilator control movement.	System not lubricated properly.	Lubricate system.
	Cable tension too high.	Adjust cable tension.
	Binding control column.	Adjust and lubricate.
	Pulleys binding or rubbing.	Replace binding pulleys and/or provide clearance between pulleys and brackets.
	Cables not in place on pulleys.	Install cables correctly.
	Cables crossed or routed incorrectly.	Check routing of control cables.
	Bent stabilator hinge.	Repair or replace stabilator hinge.
Incorrect stabilator travel.	Stabilator stops incorrectly adjusted.	Adjust stop screws.
Correct stabilator travel cannot be obtained by adjusting stops position.	Stabilator cables incorrectly rigged.	Re-rig stabilator cables.

CHART 2
TROUBLESHOOTING STABILATOR TRIM CONTROL SYSTEM

Trouble	Cause	Remedy
Lost motion between trim control wheel and trim tab.	Cable tension too low.	Adjust cable tension.
	Cables not in place on pulleys.	Install cables properly.
	Broken pulley.	Replace pulley.
	Linkage loose or worn.	Check linkage and tighten or replace.
Trim control wheel moves with excessive resistance.	System not lubricated properly.	Lubricate system.
	Cable tension too high.	Adjust cable tension.
	Pulleys binding or rubbing.	Replace binding pulleys. Provide clearance between pulleys and brackets.
	Cables not in place on pulleys.	Install cables properly.
	Trim tab hinge binding.	Lubricate hinge. If necessary, replace.
	Cables crossed or routed incorrectly.	Check routing of control cables.
Trim tab fails to reach full travel.	System incorrectly rigged.	Check and/or adjust rigging.
	Trim drum incorrectly wrapped.	Check and/or adjust rigging.
Trim indicator fails to indicate correct trim position.	Trim indicator unit not adjusted properly.	Adjust trim indicator.

2. Stabilator Control Cables

Refer to “Figure 1”.

A. Removal

(1) Remove access panel to aft section of fuselage in baggage compartment, two front seats, rear seats, and floor panel.

(2) Disconnect control cable at the turnbuckle in the aft section of fuselage.

(3) Remove either forward stabilator cable as follows:

(a) Remove tunnel carpet and cover plate as follows:

1) Remove rudder trim control knob and trim cover attaching screws.

2) Roll back carpet from tunnel and remove tunnel plate aft of tee bar assembly by removing the plate attaching screws.

(b) If the right (upper) stabilator control cable is to be removed, remove cotter pin guards at the pulley in the forward area of tunnel.

(c) To disconnect cables from low end of tee bar remove cotter pin, nut, washer, and bolt.

(d) In the access opening aft of main spar, remove cable rub blocks attached to spar housing by removing the block attaching screws.

(e) Remove cotter pin cable guard at pulley cluster in the access opening aft of main spar.

NOTE: Attach a line to cable end before removal to aid in control cable installation.

(f) Pull cable aft through floor tunnel.

(4) Remove either aft stabilator control cable as follows:

(a) Remove cotter pin, nut, washer, and bolt and disconnect cable end at the stabilator balance arm.

(b) Remove cotter pin cable guard at the pulley either above or below balance arm.

(c) Remove cable from airplane.

B. Installation

(1) Install forward stabilator cables as follows:

(a) Pull control cable through floor tunnel. Check right (upper) cable is routed around pulley(s) in the forward area of floor tunnel.

(b) Connect cables to low end of control column tee bar or idler arm with bolt, washer, nut, and cotter pin. Allow cable ends free to rotate.

(c) Install aft control cable per instructions below.

(d) Connect control cable to aft cable at the turnbuckle in aft section of fuselage.

(e) Set cable tension and check rigging and adjustment per Stabilator Controls - Rigging and Adjustment.

(f) For the right control cable, install the cotter pin cable guards at the pulley(s) in the forward area of the tunnel.

(g) Within the access opening at main spar, install cable rub blocks to spar housing and secure with screws.

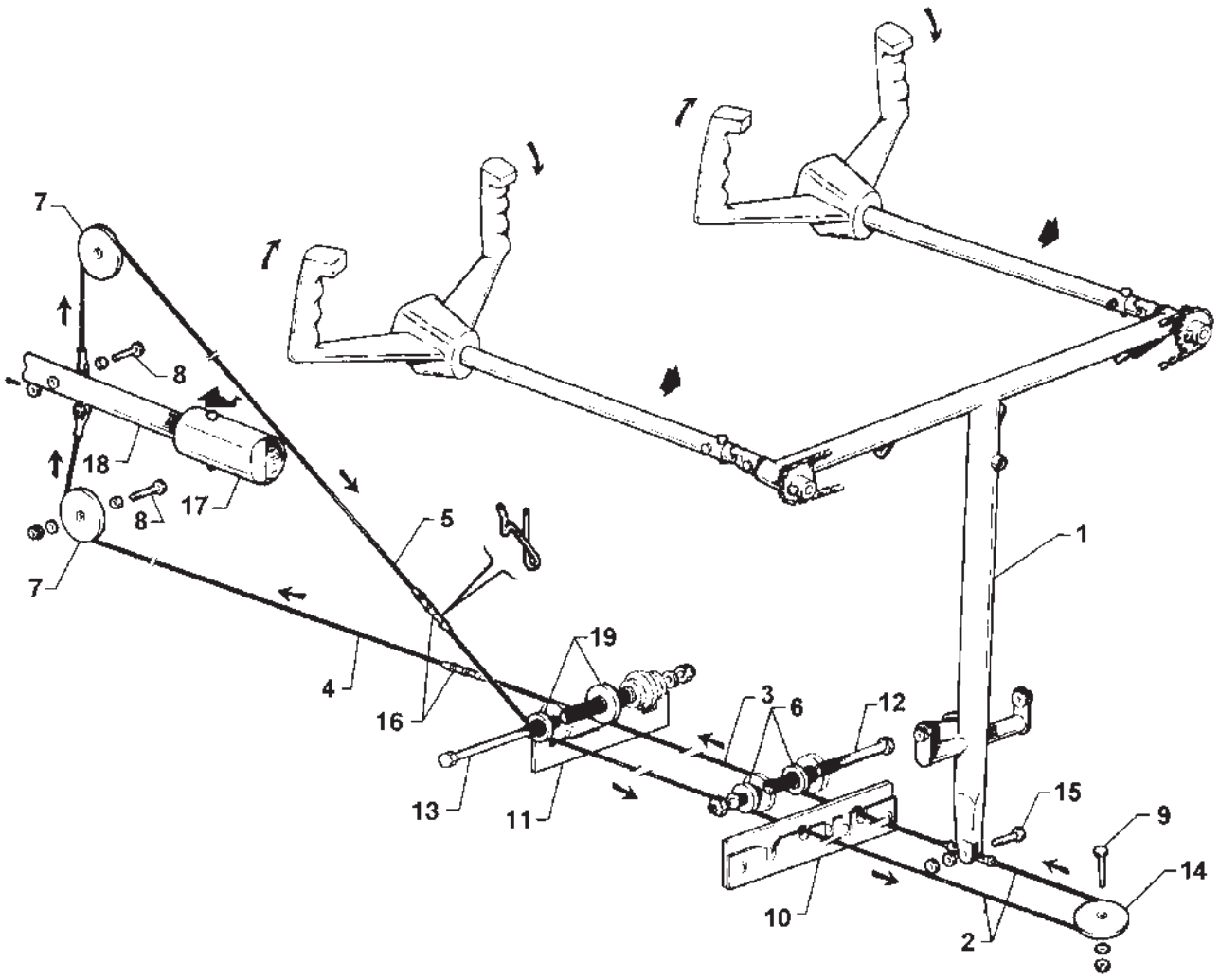
(h) Install cotter pin cable guard at the pulley cluster in the access opening.

(i) Install floor tunnel plate and trim covers as follows:

1) Place tunnel plate into position and secure with attaching screws.

2) Roll carpet into place and install rudder trim cover and knob.

(j) Install front seats, rear seat, and floor panel.



1. CONTROL COLUMN TEE BAR
2. FORWARD RIGHT CABLE
3. FORWARD LEFT CABLE
4. AFT LOWER LEFT CABLE
5. AFT UPPER RIGHT CABLE
6. FORWARD CLUSTER PULLEY
7. AFT PULLEYS
8. BOLT, WASHER, NUT & COTTER PIN
9. BOLT, WASHER & NUT
10. CABLE RUB BLOCK
11. CABLE GUARD
12. BOLT. WASHER (7) & NUT
13. BOLT, WASHER (11) & NUT
14. FORWARD PULLEY
15. BOLT, WASHER, NUT & COTTER PIN
16. TURNBUCKLE
17. BALANCE ARM WEIGHT
18. STABILATOR BALANCE ARM
19. PULLEYS

Stabilator Controls
Figure 1

(2) Install either aft stabilator control cable as follows:

(a) Route cable around its pulley either over or under stabilator balance arm.

(b) Connect cable to stabilator balance arm and secure with bolt, washer, nut, and cotter pin. (Check bushing is installed with bolt.)

(c) Connect cable to the forward cable at the turnbuckle in aft fuselage. (Upper aft cable connects to right forward cable and the lower cable to left cable.)

(d) Install cotter pin cable guard at the pulley, where required.

(e) Set cable tension and check rigging and adjustment per below.

(f) Install seats and access panels.

3. Stabilator Controls - Rigging and Adjustment

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF STABILATOR. WHILE IT WOULD SEEM SELF-EVIDENT, FIELD EXPERIENCE HAS SHOWN THAT THIS CHECK IS FREQUENTLY MISINTERPRETED OR NOT PERFORMED AT ALL. ACCORDINGLY, UPON COMPLETION OF STABILATOR RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE STABILATOR TRAILING EDGE MOVES UP WHEN THE WHEEL IS PULLED BACK; AND, THAT THE STABILATOR TRAILING EDGE MOVES DOWN WHEN THE WHEEL IS PUSHED FORWARD.

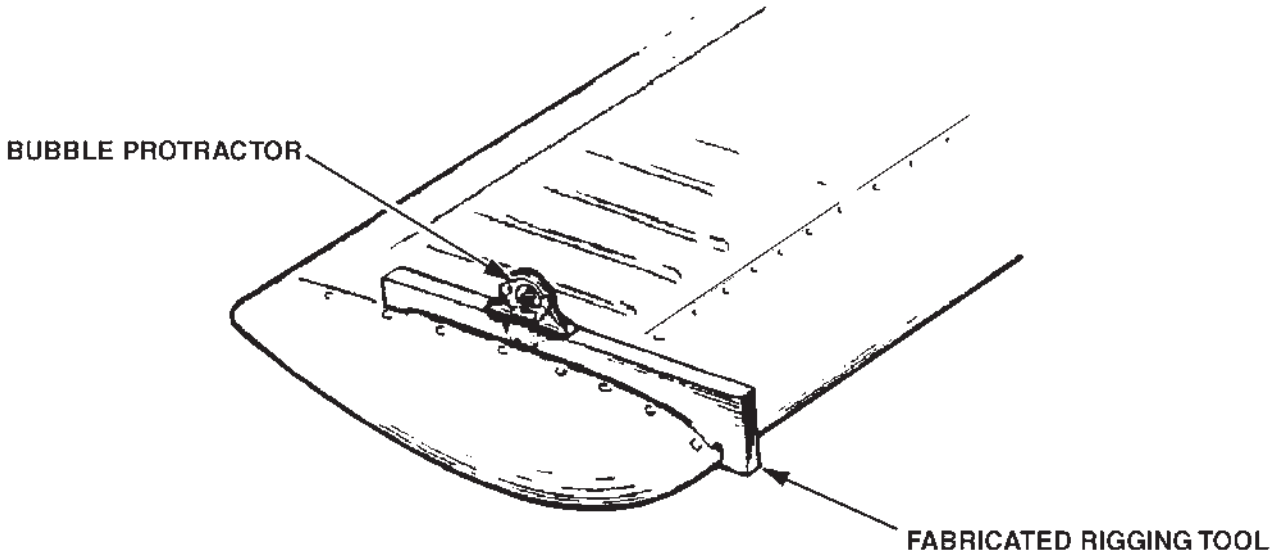
A. Level airplane per 8-20-00.

CAUTION: STABILATOR MUST CONTACT BOTH STOPS BEFORE CONTROL WHEEL CONTACTS ITS STOPS.

B. Check and set the proper degree of stabilator travel, as follows:

(1) To check the stabilator travel, place rigging tool (see “Figure 6” on page 273013 to fabricate tool) on upper surface of stabilator per “Figure 2”.

(2) Set, on a bubble protractor, the number of degrees up travel as given in “Chart 2” on page 27004, and place it on the rigging tool. Raise stabilator trailing edge and check protractor bubble is centered when the stabilator contacts its stops.



Stabilator Rigging
Figure 2

(3) Set, on a bubble protractor, the number of degrees down travel as given in “Chart 2” on page 27004. Place it on the rigging tool. Lower stabilator trailing edge. Check protractor bubble is centered when the stabilator contacts its stops.

(4) If stabilator travel is incorrect in up or down position, remove tail cone fairing by removing attaching screws. Use rigging tool and bubble protractor to turn the stops at each stabilator hinge stop in or out to obtain correct degree of travel (see “Chart 2” on page 27004 and “Figure 5” on page 27209.).

(5) Check stop screw lock nuts are secure and install tail cone fairing.

C. Check and set stabilator control cable tension, as follows:

(1) Check stabilator travel is correct.

(2) Remove cabin rear closeout panel to access the aft section of the fuselage.

(3) Secure control column in the near forward position, with tee bar positioned 0.25 + 0.12, -0.03 inch off of the forward tee bar stop.

(4) Check each control cable for correct tension per 27-00-00, “Chart 2”.

(5) If tension is incorrect, loosen lower cable turnbuckle in aft fuselage and adjust the upper cable turnbuckle for proper tension. Obtain cable tension with control wheel tee bar positioned 0.25 +0.12, -0.03 inch off of the forward tee bar stop and with stabilator contacting its stop.

(6) Check safety of all turnbuckles and bolts.

(7) With upper cable tension correct, and control wheel forward, adjust lower cable turnbuckle for proper tension.

(8) Check full travel of control wheel with relation to full travel of stabilator to verify stabilator contacts its stops before control wheel contacts its stops. With control wheel in the fore and aft positions, travel distance from the point where stabilator contacts its stops and control wheel contacts its stops should be approximately equal. Adjust turnbuckles if incorrect. Install all pulley guard pins.

(9) Install cabin rear closeout panel.

D. Remove airplane from jacks.

4. Stabilator Post-Rigging Inspection

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

Upon completion of any stabilator control cable replacement, disconnect, or rigging perform the following postrigging inspection:

A. Pull the pilot’s control wheel FULL AFT, as if to nose the airplane UP, until reaching the primary stop. Visually confirm the following:

The stabilator is deflected leading edge DOWN and trailing edge UP. The trailing edge of the stabilator will be noticeably and significantly ABOVE the leading edge.

B. Push the pilot’s control wheel FULL FORWARD, as if to nose the airplane DOWN, until reaching the primary stop. Visually confirm the following:

The stabilator is deflected leading edge UP and trailing edge DOWN The trailing edge of the stabilator will be noticeably and significantly BELOW the leading edge.

C. Repeat the above steps for the copilot controls.

D. Make a logbook entry documenting completion of this inspection.

5. Stabilator Trim Controls

A. Forward Stabilator Trim Assembly

Refer to “Figure 3”.

(1) Removal

CAUTION: DO NOT DAMAGE TRIM INDICATOR WIRE.

(a) To remove trim control wheel assembly and trim control cables, remove the panel to the aft section of airplane.

(b) If aft trim cable is not to be removed, block cables at the pulleys in upper aft section of fuselage to prevent unwrapping from trim drum. (Refer to “Figure 4”.)

(c) Loosen the cables if the trim control wheel is to be removed or disconnect if the cables are also to be removed. (Do this at the trim cable turnbuckles in the aft section of the fuselage.)

(d) Remove control wheel and drum as follows:

1) Remove control wheel cover by removing cover attaching screws.

2) Remove wheel assembly from mounting brackets by removing nut, washer, and bolt holding wheel between brackets. (Draw wheel from brackets. Do not damage trim indicator wire.)

3) Unwrap left cable from drum.

4) Wheel and drum are joined by a push fit. Separate these two items with their center bushing and unwrap right cable.

5) Tie cables forward to prevent slipping into floor tunnel.

(e) Remove trim control cables as follows:

1) Remove rear seats, floor panel, and front seats.

2) Unfasten carpet from aft floor tunnel and lay it forward.

3) Remove tunnel cover attaching screws. Remove tunnel cover located between trim control wheel and spar cover.

4) Remove cable pulleys located in the tunnel by removing cotter pin, washer, and clevis pin.

5) Remove cable rub blocks on aft side of main spar by removing cable rub block attaching screws.

6) Remove cable guard pin at pulley cluster aft of wing flap torque tube at station 127.175.

7) Remove nut, washer, bushing, and bolt and remove cable pulleys in aft section of fuselage at station 156.485.

8) With cables disconnected from trim control wheel, pull cable(s) through floor tunnel.

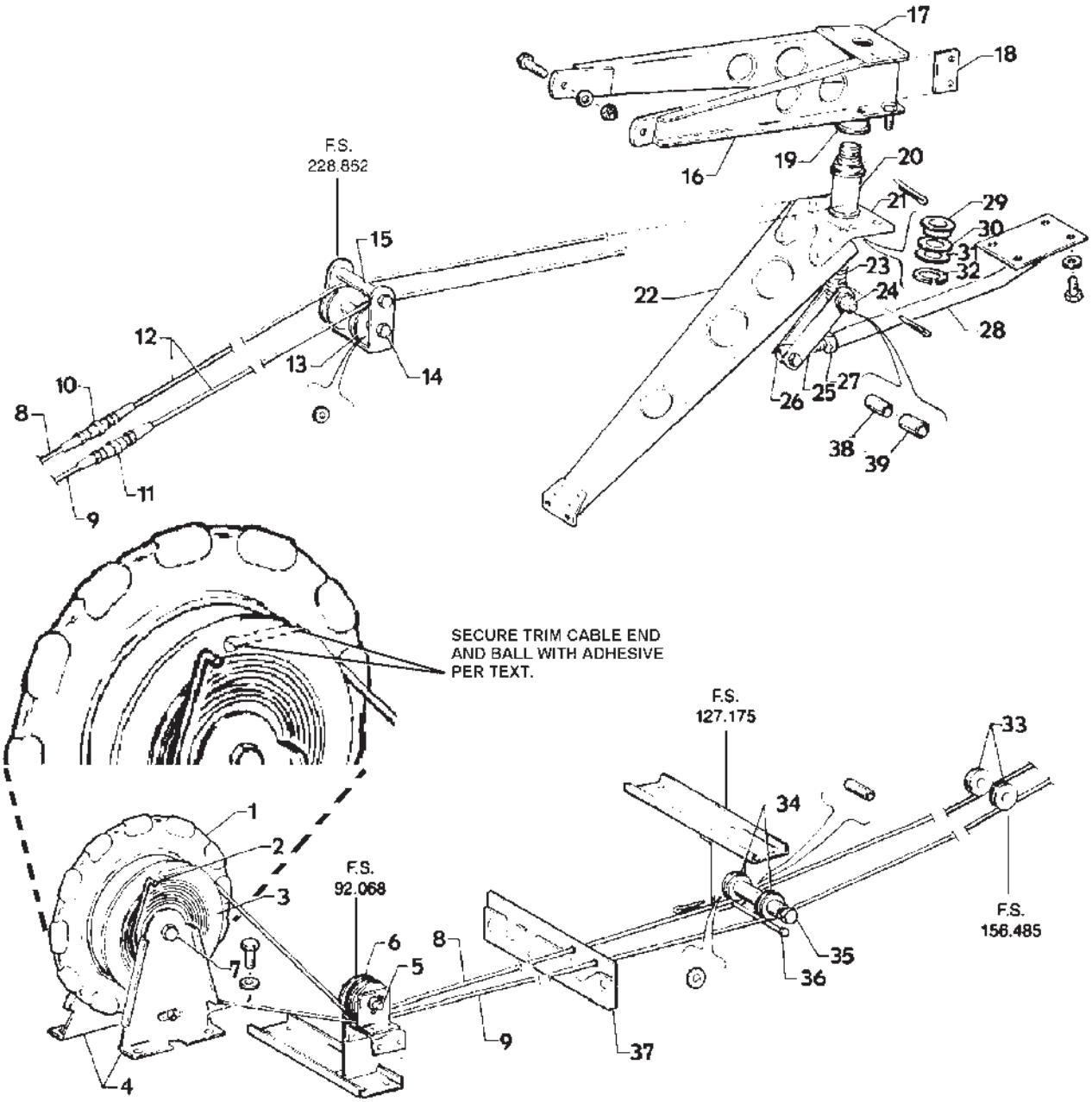
(2) Installation

(a) Install trim control wheel with drum as follows:

1) Wrap right trim cable on trim drum by inserting the swaged ball of cable in slot provided in the side (right side) of the drum that mates with control wheel. Wrap drum three wraps of cable in a clockwise direction. Secure trim cable and cable ball to drum with Loctite EA 9396 AERO A/B adhesive, P/N 279-473. See detail in “Figure 3” on page 27308.

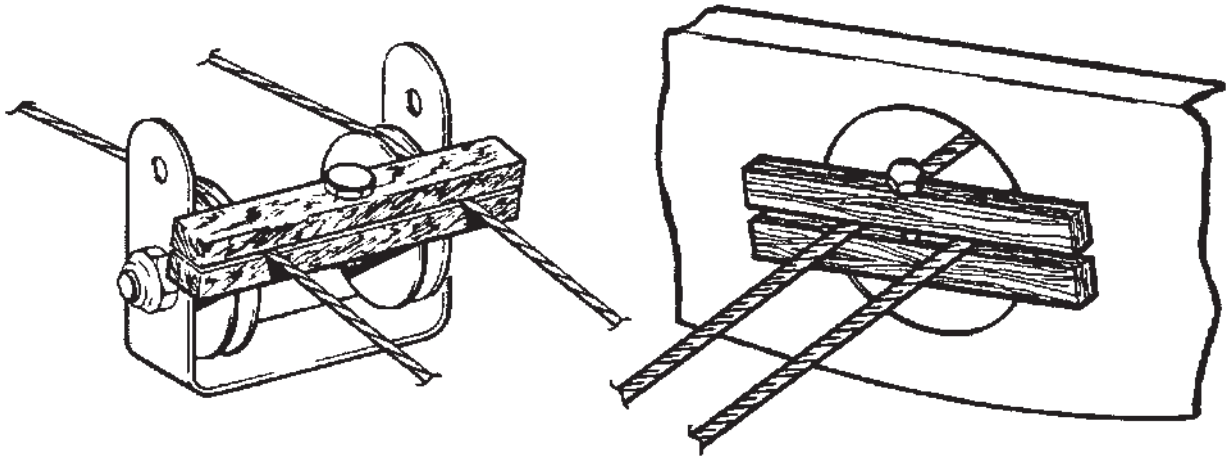
2) Align the long lug of drum with long slot of wheel and push together to attach control wheel to cable drum.

3) Wrap left trim cable on drum by inserting the swaged ball of cable in slot provided in the flanged side (left side) of drum and looking at this side, wrap drum with three wraps of cable in a clockwise direction. Secure trim cable and cable ball to drum with Loctite EA 9396 AERO A/B adhesive, P/N 279-473. See detail in “Figure 3” on page 27308.



1. CONTROL WHEEL, TRIM
2. INDICATOR, TRIM POSITION
3. DRUM, TRIM CABLE
4. MOUNTING BRACKET
5. CLEVIS PIN, WASHER & COTTER PIN
6. PULLEY CLUSTER
7. BOLT, BUSHING, WASHER & NUT
8. CABLE, RIGHT FORWARD
9. CABLE, LEFT FORWARD
10. TURNBUCKLE, RIGHT
11. TURNBUCKLE, LEFT
12. CABLE, AFT
13. PULLEY CLUSTER
14. BOLT, BUSHING, WASHER (3) & NUT
15. GUARD-BOLT, BUSHING, WASHER & NUT
16. RIB, HORIZONTAL SUPPORT
17. PLATE, RIB
18. GUARD
19. BUSHING, BARREL
20. BARREL, TRIM
21. CLIP, DIAGONAL RIB
22. RIB, DIAGONAL SUPPORT
23. SCREW, TRIM
24. BOLT, WASHER (2) & NUT
25. LINK
26. ROD END
27. JAM NUT
28. ARM, TRIM CONTROL
29. BUSHING BARREL
30. WASHER, THRUST
31. WASHER, SPACER
32. SNAP RING
33. PULLEY CLUSTER
34. PULLEY CLUSTER
35. BOLT, WASHER & NUT
36. GUARD, CABLE
37. RUB BLOCK
38. TRIM LINK BUSHING
39. TRIM LINK SLEEVE

Stabilator Trim Controls
Figure 3



Methods of Securing Trim Cables
Figure 4

4) Lubricate and install bushing in control wheel and drum.

5) Align control cables and position control wheel assembly between mounting brackets. Check end of trim indicator wire is positioned in the spiraled slot of the drum with no bind on the end. Install retainer bolt from left side and install washer and nut.

6) Install cover over control wheel and secure with screws, unless control cables have yet to be installed.

(b) Install trim control cables as follows:

1) Pull cable(s) through floor tunnel.

2) Wrap cable drum and install trim control wheel per, above.

3) Position cable pulleys on mounting bracket in floor tunnel and install clevis pin, washer, and cotter pin.

4) Connect cable to the aft cable at turnbuckle in the aft section of fuselage. Install aft cable if not already installed.

5) Install pulleys in aft lower section of fuselage at station 156.485 forward of cable turnbuckles.

6) Remove blocks holding aft trim cable and check cables are seated on their pulleys.

(c) Set cable tension and check rigging and adjustment per Rigging and Adjustment. Check safety of all turnbuckles.

(d) Install cable guard at underside of the pulleys aft of flap torque tube at station 127.175, and secure.

(e) Install cable rub blocks on aft side of main spar housing and secure with screws.

(f) Install tunnel cover on tunnel and secure with screws.

(g) Install carpet over floor tunnel.

(h) Install cover over trim control wheel and secure with screws and special washers.

(i) Install floor panel and seat belt attachments aft of main spar and secure panel with screws.

(j) Install panel to aft section of airplane and install seats.

(3) S-TEC Manual Electric Trim Clutch Check & Adjustment

Refer to “Figure 5”.

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

(a) S-TEC Manual Electric Trim is an optional system in Archer III S/N’s 2843058 and up. When installed, a servo motor, capstan, and idler pulley assembly are attached to the left stabilator trim cable just aft of Sta. 156.00.

(b) Adjustment

Check and adjust the servo clutch as follows:

1) Remove the trim cable guards and trim cable from capstan. Remove the capstan cover. Check capstan torque by attaching the capstan adjusting tool (special tool) to the capstan and using a currently calibrated torque wrench as shown in “Figure 5”. Acceptable torque is 29 ± 1 in.-lbs. If adjustment is required, proceed as follows.

2) Remove servo motor and capstan. Place servo motor and capstan in a holding fixture (vice) with capstan up. Remove capstan cover. Remove cotter pin from end of servo shaft and remove castle nut, shim washers, and tension washers.

3) Replace tension washers as required [1 ea. 0.032 in. (Red) and 2 ea. 0.040 in. (Blue)]. Replace shim washers and castle nut. Tension castle nut so that capstan torque is as specified above.

4) Reinstall servo motor and capstan. Adjust stabilator trim cable tension as specified in 27-00-00, “Chart 2”.

B. Aft Stabilator Trim Assembly

Refer to “Figure 3”.

(1) Removal

(a) Remove access panel to aft section of fuselage.

(b) Block trim cables at the first set of pulleys forward of cable turnbuckles in aft fuselage per “Figure 4”.

(c) Disconnect cable at the turnbuckles in aft section of fuselage.

(d) Remove tail cone attaching screws and remove tail cone.

(e) Remove nut, washer, bolt, bushing, and sleeve connecting link to screw, and disconnect link between trim screw and trim control arm.

(f) Remove cotter pin from top of screw, and turn screw down and out of barrel.

(g) Remove snap ring, washer, and thrust washer from bottom of barrel.

(h) Remove the four attaching nuts, washers, and bolts and disconnect the diagonal rib from horizontal rib that supports the trim assembly.

(i) Pull trim cable from fuselage.

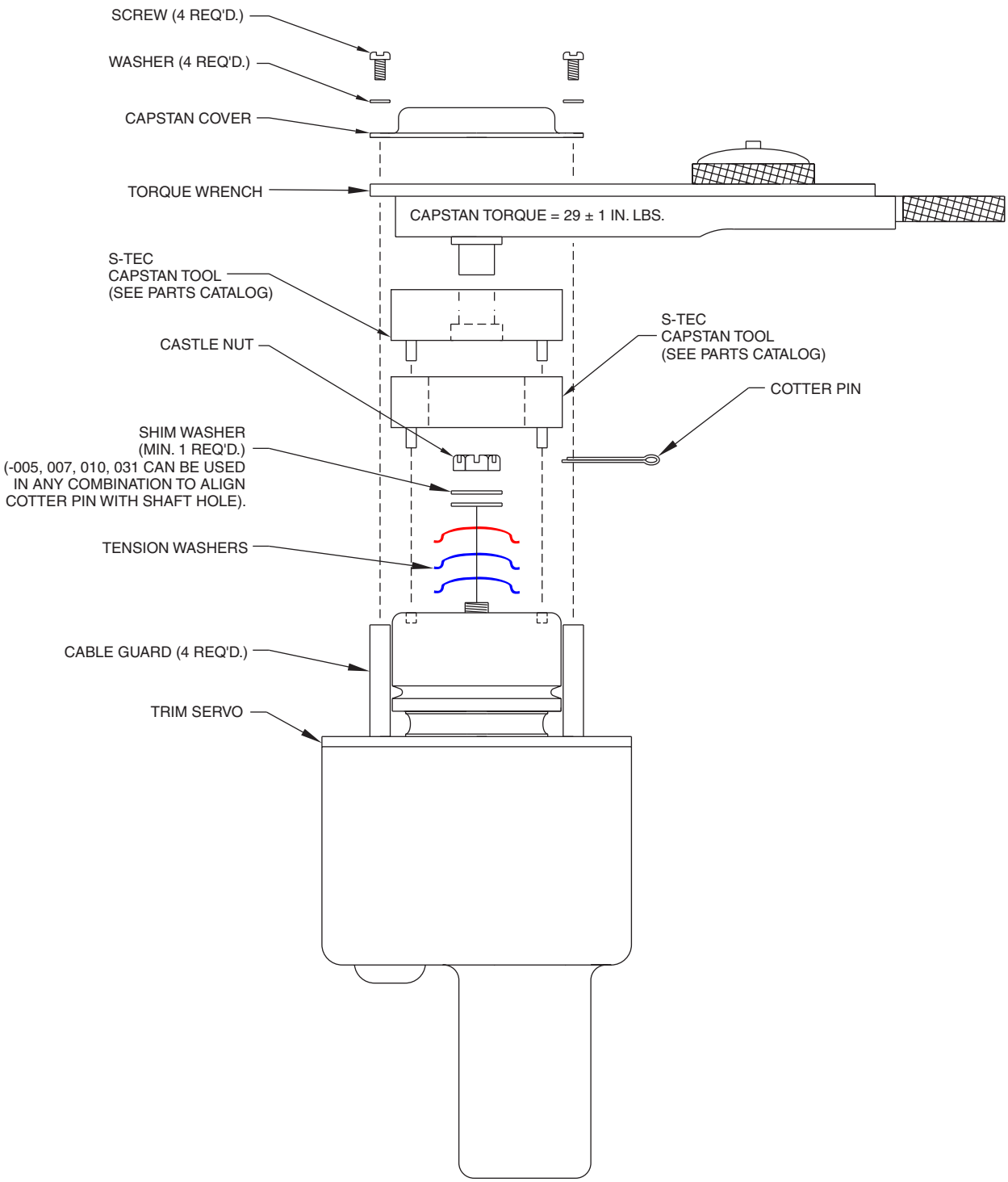
(2) Installation

(a) Wrap trim barrel as follows:

1) Lay center (as measured equally from each end to center of cable) of trim cable in barrel slot.

NOTE: Wrap cable evenly to obtain 23 wraps on barrel, as viewed from side opposite slot, and with cables extending out from slotted side.

100831 G



Manual Electric Trim Servo Clutch Torque Adjustment
Figure 5

2) Bring upper cable through diagonal slot in flange at upper end of barrel, and wrap down in counterclockwise direction.

3) Bring lower cable through diagonal slot in lower end of barrel, and wrap up in clockwise direction.

(b) Clamp cables between two pieces of wood to prevent unwrapping.

(c) Check barrel bushings are installed in rib plate and clip.

(d) Lubricate bushings and install trim barrel in the bushings between the two support ribs. Attach bottom diagonal rib to horizontal rib and secure with bolt, washers, and nuts.

(e) Install thrust washer, washer, and snap ring on low end of barrel.

(f) Install trim screw in barrel and secure each end with cotter pin through the screw.

(g) Attach link between trim screw and trim control arm. Check bushing and sleeve are in place before installing bolt, washer and nut.

(h) Route cables into fuselage and attach ends to forward trim cables.

(i) Remove blocks holding forward cables tight and holding aft cables at barrel.

(j) Set cable tension and check rigging and adjustment per Rigging and Adjustment, below. Check safety of all turnbuckles and install pulley guard pins.

(k) Install tail cone and secure with screws.

(l) Install access panel to aft section of fuselage.

C. Rigging and Adjustment

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF STABILATOR TAB. WHILE IT WOULD SEEM SELF-EVIDENT, FIELD EXPERIENCE HAS SHOWN THAT THIS CHECK IS FREQUENTLY MISINTERPRETED OR NOT PERFORMED AT ALL. ACCORDINGLY, UPON COMPLETION OF STABILATOR TRIM RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE STABILATOR TAB TRAILING EDGE MOVES UP WHEN THE TRIM WHEEL IS TRIMMED NOSE DOWN; AND, THAT THE STABILATOR TAB TRAILING EDGE MOVES DOWN WHEN THE TRIM WHEEL IS TRIMMED NOSE UP.

Refer to “Figure 3”.

(1) Level the airplane per 8-20-00.

(2) Remove cabin rear closeout panel to access the aft section of the fuselage.

(3) Secure control column in the near forward position, with tee bar positioned 0.25 +0.12, -0.03 inch off of the forward tee bar stop.

(4) Check and adjust stabilator trim control cables for correct tension per 27-00-00, “Chart 2”.

(5) Check safety of all turnbuckles and bolts.

(6) Set stabilator in neutral and lock in position. (To find neutral, place rigging tool on upper surface of stabilator per “Figure 2”. Zero a bubble protractor, place it on rigging tool and tilt stabilator to center bubble.)

(7) With stabilator in neutral, turn trim wheel to neutral position.

(8) Check stabilator tab is streamlined with the stabilator. If tab is not streamlined, adjust tab push rod to streamline tab with stabilator. This is the neutral position of the tab.

(9) Set protractor on neutral tab, center bubble, and turn trim control wheel both full up and full down to check tab travel per 27-00-00, “Chart 2”. (Travel on the protractor is determined by taking difference between protractor reading at neutral and up, and neutral and down.) Bubble must be centered each reading with the airplane level.

(10) To obtain correct travels, if incorrect, adjust by disconnecting the links at actuating arm rod end and turning the rod end in or out to correct travel. Reconnect links to rod end.

(11) Secure jam nut on actuating arm rod end.

(12) Use control wheel and trim wheel to put stabilator and tab in all extremes of travel.

(a) Check for turnbuckle and pulley clearance and tab indicator location.

(b) Verify stabilator up and down stops are contacted before control wheel tee bar stops.

(13) With stabilator held securely against either stop, determine freeplay of the stabilator tab does not exceed the limits in 27-00-00, “Chart 2”.

D. Stabilator Trim Post-Rigging Inspection

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

Upon completion of any stabilator control cable replacement, disconnect, or rigging perform the following postrigging inspection:

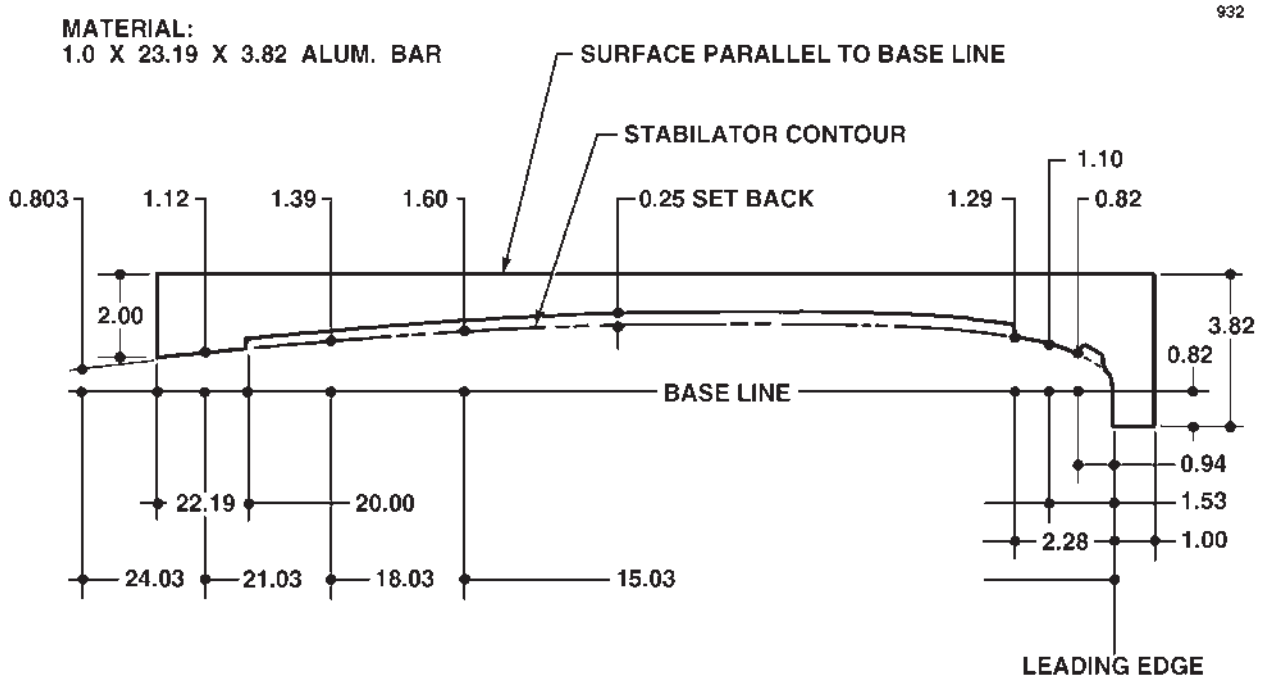
(1) When trim wheel is trimmed Down Visually confirm the following:

The Stabilator Tab trailing edge moves Up.

(2) When trim wheel is trimmed Up. Visually confirm the following:

The Stabilator Tab trailing edge moves Down.

(3) Make a logbook entry documenting completion of this inspection.



Stabilator Rigging Tool
Figure 6

6. Stall Warning

The stall warning system consists of a lift detector electrically connected to a stall warning horn. As stalling conditions are approached, the lift detector activates the stall warning horn.

The lift detector is on the leading edge of the left wing. A tab will extend beyond the leading edge at the point where lift detector is mounted. With the master switch in the ON position, gently lift the tab; the stall warning horn should sound.

The electrical circuit is protected by a 5 amp STALL WARNING circuit breaker mounted in the circuit breaker panel on the lower right side of the instrument panel.

A. Lift Detector

(1) Removal

CAUTION: MASTER SWITCH MUST BE OFF BEFORE WORKING ON LIFT DETECTOR OR WARNING HORN. WHEN INSTALLING WING, PLACE REFERENCE MARKS ON HOLDING PLATE AND WING SKIN.

(a) Remove four screws holding plate around tab. Pull lift detector and plate out as a unit.

(b) Mark electrical wires and terminals to aid installation. Remove electrical wires from lift detector; remove lift detector from airplane.

(2) Installation

NOTE: Refer to 91-27-30 for electrical schematic of stall warning system

(a) Attach electrical wires to their correct terminals on lift detector.

(b) Position lift detector with its mounting plate on wing, so sensor blade drops down freely; secure in position with the four screws previously removed.

(c) Perform “Flight Test Procedure”, below.

(3) Adjustment

The lift detector switch is adjusted at the factory when the airplane is test flown, and should not require further adjustment during normal service life of the airplane. If service on the wing requires removing the switch, the following instructions will aid in proper switch positioning.

CAUTION: NEVER ADJUST SWITCH BY BENDING VANE.

(a) Loosen the two Phillips head screws; one on either side of vane. If stall warning sounds too late, move switch up. If stall warning sounds too early, move switch down. Tighten screws after adjustments.

(b) The only way to test accuracy of the setting is to fly the airplane into a full stall condition and note speed at which the stall warning sounds. Stalls must be made with flaps up and power off. It may be necessary to make several test flights and alternate adjustments before the desired setting is obtained. The stall warning must sound not less than five mph, or more than ten mph before the stall occurs.

(c) Perform “Flight Test Procedure”, below.

B. Stall Warning Flight Test Procedure

(PIR-FTP2001-5, Rev. G / FTP2001-5-1, Rev. F.)

NOTE: The following procedure is typically performed after installation of the lift detector or installation of a replacement wing.

Fly airplane and at a rate of 1 KT/SEC maximum record the following warning and stall speeds:

	TRIM KIAS	RECORD:	WARNING KIAS	STALL KIAS
0° FLAPS =	71		_____	_____
FULL FLAPS =	62		_____	_____

(1) Verify that the stall warning system performs within the following limits:
   (a) Maximum: stall warning begins at 5 to 10 KIAS prior to stall and continues until stall occurs.
   (b) Desired: stall warning begins at 7 to 9 KIAS prior to stall and continues until stall occurs.

(2) If the stall warning system fails to perform within the limits above, adjust the system and refly the test.

(3) If not able to adjust the stall warning system sufficient to achieve the limits above, recheck the “replacement” wing installation to ensure it is installed and rigged correctly.

7. Angle of Attack (AoA) System

CAUTION: THE AIRCRAFT'S STALL WARNING SYSTEM IS THE PRIMARY STALL WARNING. THE AOA SYSTEM IS NON-REQUIRED EQUIPMENT AND IS TO BE USED ONLY AS SUPPLEMENTAL INFORMATION TO THE PILOT.

NOTE: For proper procedures in flight, consult the Pilot's Operating Handbook (POH).

The AoA Indication system consists of two parts: an Indexer Computer (Display) mounted on the glaresheild and a dedicated Lift Detector mounted in the leading edge of the right wing. The lift detector senses the wing's leading edge airflow field providing the AoA Indexer Computer an accurate and repeatable indication of the AoA.

A. Inspection

Annually or each time the Indexer Computer or Lift Transducer has been disconnected, perform the functional ground check per "Ground Functional Test" on page 273020.

B. Calibration

Each time the Lift Transducer is replaced, or anytime system calibration is in doubt, perform the calibration procedure per "Calibration" on page 273020.

C. Troubleshooting

The Indexer Computer and Lift Transducer are not field repairable. Check the system by applying power, if the system successfully completes the Power-On Self Test, the system is operational.

If the Power-On Self Test is unsuccessful, hold the top and bottom buttons on the Indexer for two (2) seconds while power is still supplied. The display will illuminate all reference marker segments and then will illuminate a particular segment of the Indexer Computer display. Make note of the illuminated segments, as these will aid in diagnosing the problem.

When contacting support, have the following information on hand: Unit Part Number, Unit Serial Number, Unit Software Revision (these are found on the nameplate on the lower surface of the Indexer) and the Fault Code (which LEDs are illuminated on the Indexer Computer).

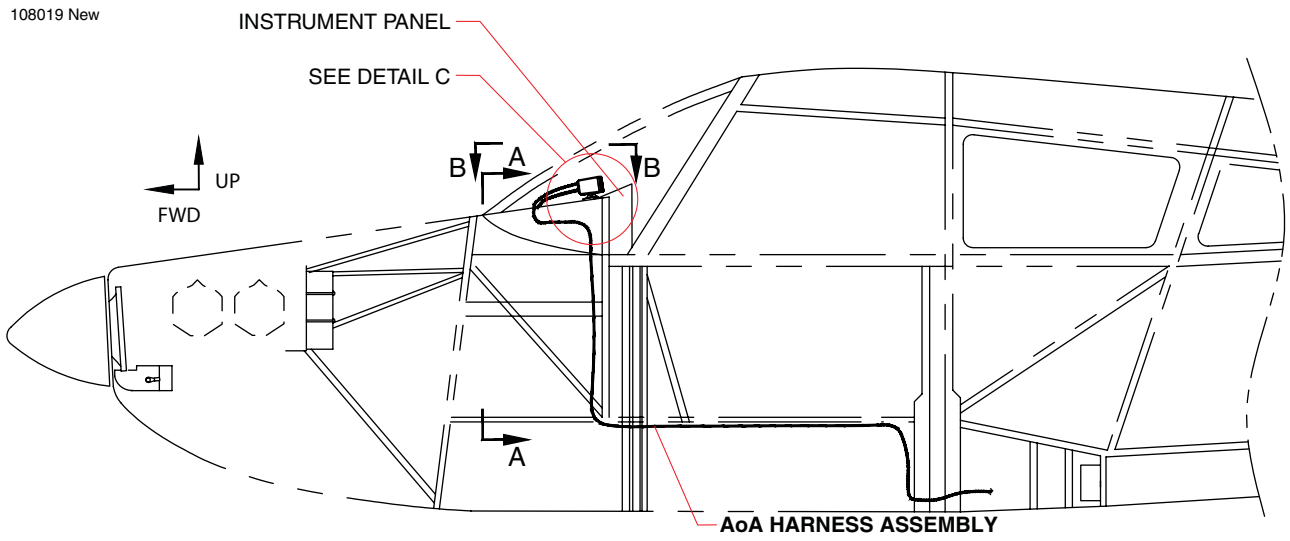
See http://www.safeflight.com for additional troubleshooting actions.

D. Indexer Computer

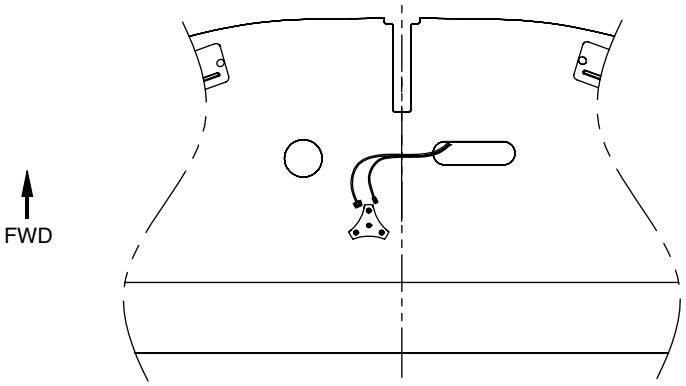
The Indexer Computer is installed on top of the instrument panel glareshield. See "Figure 7" on page 273017.

(1) Removal
  (a) Insert a flat-head screw driver into the slot at the end of each leg.
  (b) Lift up to release.
  (c) Rotate the cover to access the screw heads.
  (d) Pull gently on the Indexer to remove the system after the cover has been lifted.
  (e) Disconnect cable assembly from rear of unit.

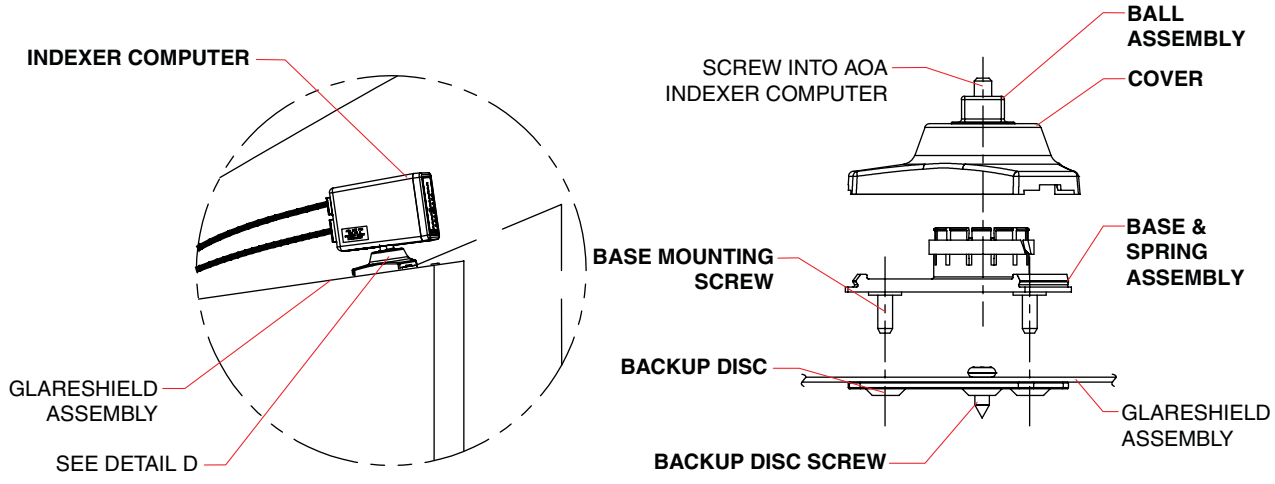
(2) Installation
  (a) Insert Ball Assembly into Base Assembly.
  (b) Rotate Indexer until view of front face is acceptable.
  (c) Lock cover onto Base Assembly.
  (d) Connect Cable Assembly to Indexer.



LOOKING OUTBD FROM LBL 3.0



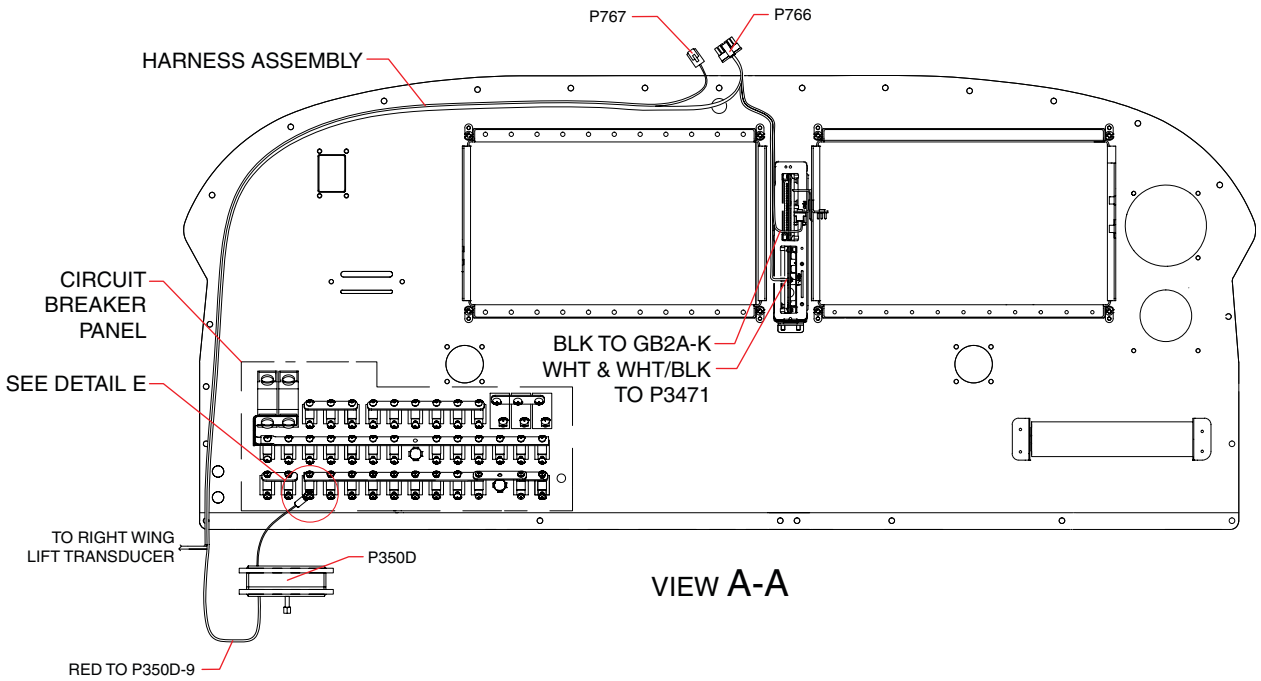
VIEW B-B



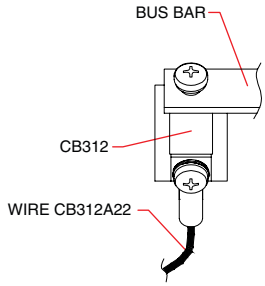
DETAIL C
LHS LOOKING INBOARD

DETAIL D

AoA System Installation
Figure 7 (Sheet 1 of 3)

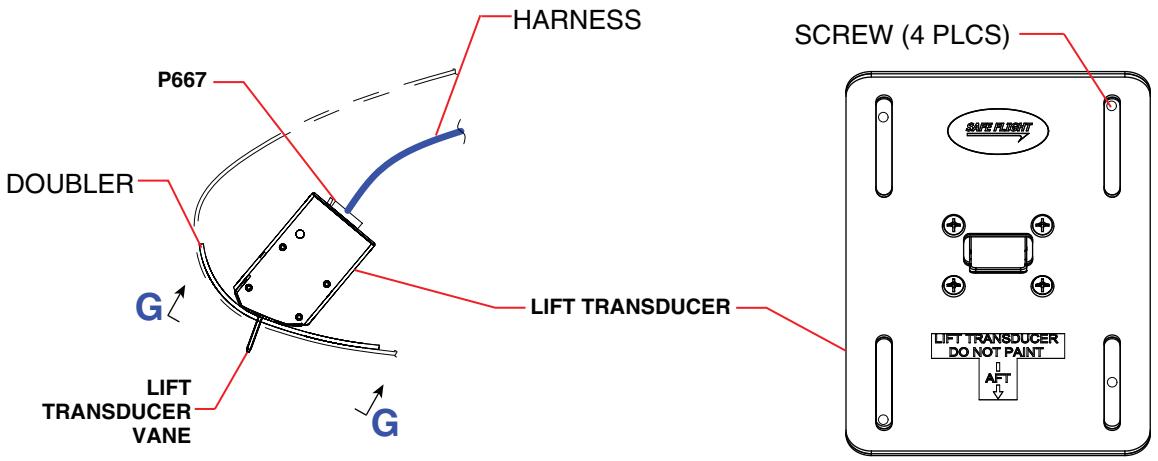


VIEW A-A



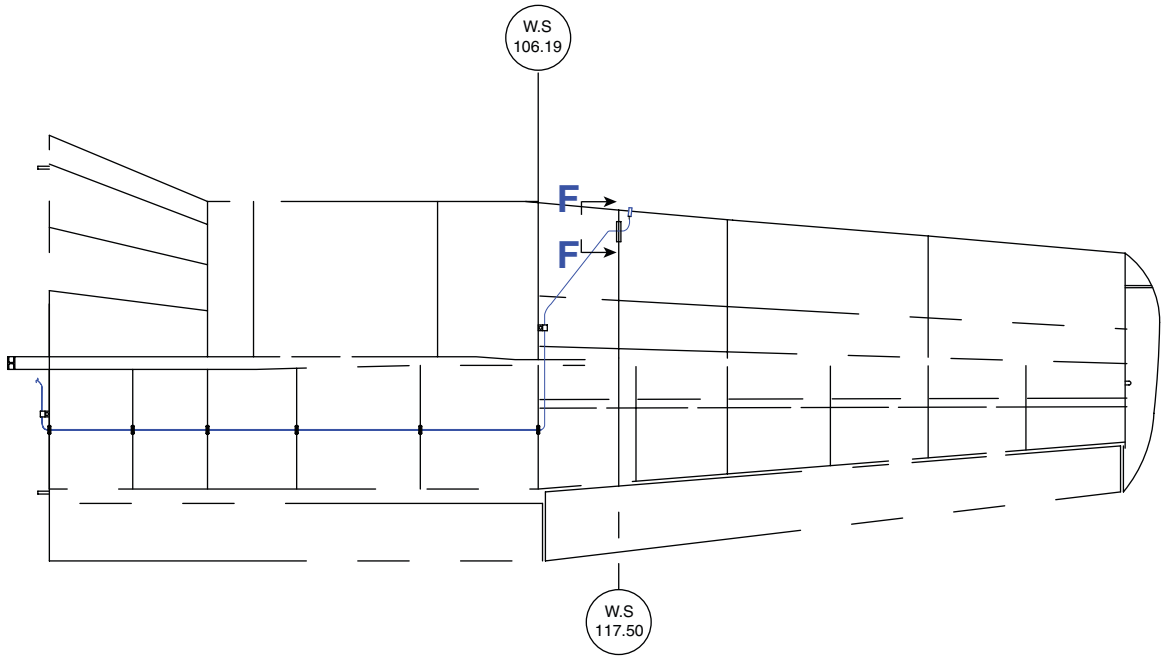
DETAIL E

AoA System Installation
Figure 7 (Sheet 2 of 3)



VIEW F - F
LOOKING OUTBOARD

VIEW G - G



LOOKING DOWN AT RIGHT WING

AoA System Installation
Figure 7 (Sheet 3 of 3)

E. Lift Transducer

The Lift Transducer is mounted in the leading edge of the right wing just outboard of W.S. 117.50.

(1) Removal

(a) Set master switch to the OFF position.

(b) Remove four (4) screws from lift transducer mounting plate slots.

(c) Carefully pull the lift transducer forward to gain access to the P667 connector.

(d) Take care to capture the lift transducer wiring harness so it cannot retract into the wing.

(e) Disconnect the P667 connector and remove the lift transducer.

(2) Installation

(a) Postion the lift transducer before the opening in the wing.

(b) Connect the P667 connector and release the wiring harness to allow it to retract into the wing.

(c) Slide the lift transducer into position and secure with four (4) screws.

F. Ground Functional Test

(PIR-SafeFlight Dwg. 56201-1, Rev. E.)

The Lift Transducer vane should move freely in both directions with no resistance.

With the electrical power OFF the Indexer Computer should not be lit.

Turn the main power ON. The Indexer Computer will perform a Power-on Self-Test, illuminating all of the LEDs for approximately five seconds. The audio alert will also be active during the self-test. After the self-test, the audio will cancel and the LEDs will indicate the current position of the Lift Transducer.

NOTE: The single red LED will continue to blink at 1 Hz until the system has been calibrated in flight.

Observe the Indexer Computer for a transition to the low-AoA side of the display, with a single green LED illuminated when the Lift Transducer vane is gently pushed down (AFT).

Gently push the vane up (FORWARD). The Indexer Computer should transition to the high-AoA side of the display, followed by the flashing of the two red LEDs, indicating Low Airspeed (high AoA) Awareness (LAA). Turn the navigation light switch ON and observe Indexer Computer display dims.

G. Calibration

At a safe altitude for possible stall recovery, using the POH recommended landing approach speed, power setting, and flap extension fly a simulated approach. Simultaneously, press and hold both the top and bottom buttons for two seconds to enter the calibration mode. All of the red, yellow, and green indicators will begin flashing as shown in “Figure 8” on page 273021.

Press the bottom button to select the Landing Approach calibration mode. Continue flying the aircraft at the normal approach airspeed and descent rate. The Indexer Computer will begin flashing slowly the green center ON-SPEED indication for five seconds.

Continue flying the aircraft at the normal approach airspeed and descent rate. The Indexer Computer will begin flashing quickly the green ON-SPEED indication while the system is recording data for five seconds.

If the calibration is successful, the green ON-SPEED indication will change to solid (non-flashing) as shown in “Figure 9” on page 273021.

If the calibration is unsuccessful, the Indexer Computer will return to flashing slowly the ON-SPEED indications for 5 seconds, and then will transition again to flashing quickly for another 5 seconds. This sequence will repeat until the calibration is successful.

When the approach setting has been successfully accomplished, without adjusting power, pitch the aircraft nose up, slowing the aircraft to the speed where aircraft stall warning just begins to sound. Hold this speed. Press the top button to transition to the LAA calibration mode. The two red lights will begin slowly flashing for five seconds as shown in “Figure 10”.

The two red lights will begin quickly flashing for another five seconds. During this time, maintain the stall warning target speed (at a constant power setting).

If the calibration is successful, the red LAA indication will change to solid (non-flashing).

If the calibration is unsuccessful, the system will return to flashing slowly for 5 seconds; then will transition again to flashing quickly for another 5 seconds. This sequence will repeat until the calibration is successful.

Press and hold both the top and bottom button for two seconds to exit the calibration mode.

NOTE: After a successful calibration, at a safe altitude, fly a simulated approach using the POH recommended landing approach speed, power, and flap extension. Verify that the given Onspeed (five dots) is lit within +/-5 knots of the POH referenced speed. Without adjusting power, pitch aircraft nose up, slowing the aircraft so that the Indexer Computer displays two amber LEDs. Verify LAA audio is active and stall warning is not. If verification fails, repeat calibration.

NOTE: Pressing and holding both top and bottom buttons for two seconds at any time exits the calibration procedure.

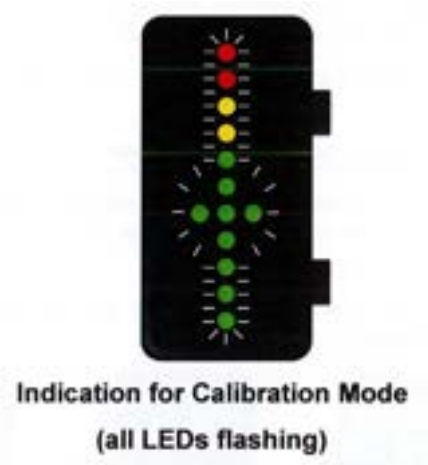


Figure 8



Figure 9



Figure 10

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FLAPS

1. Troubleshooting

   See “Chart 1”.

2. Inspection

   Torque Tube/Push Rod Distortion Inspection

   If flaps have been extended at or above V_{FE}, inspect the flap torque tube arms and pushrods for evidence of distortion.

   (1) If the paint is cracked or peeling anywhere along the torque tube arm or pushrod, torsional movement has occurred.
   (2) Remove the paint and inspect for cracks:
      (a) In the welds at the arm on the torque tube end.
      (b) In the rod ends and pushrod tube.
      (c) Use a dye penetrant method of inspection.
   (3) If cracks are not found, repaint the part(s) and reinstall.
   (4) If cracked, replace the affected part(s).

3. Wing Flap Controls

   A. Removal

   Refer to “Figure 1” on page 27503.

   (1) Remove flap torque tube assembly as follows:
      (a) Remove access plate between the underside of aft section of each wing and fuselage by removing attaching screws.
      (b) Remove two front seats, rear seat, and floor panel.
      (c) Disconnect left and right flap control tubes (rods) at the flaps by removing nuts, washers, and bolts, or at the torque tube cranks (arms) by removing the bolts and washers from inner side of each crank. Remove bolt through a hole in fuselage side skin located over the torque tube with flap handle moved to 40 degree position.

         CAUTION: FORWARD PRESSURE WILL BE ON THE HANDLE WITH THE TENSION SPRING DISCONNECTED.

      (d) With the flap handle, fully extend flaps and disconnect flap tension spring at the spar or aft end of control cable.

CHART 1
TROUBLESHOOTING FLAP CONTROL SYSTEM

Trouble	Cause	Remedy
Flaps fail to extend or retract.	Control cable broken or disconnected.	Replace or reconnect control cable.
Flaps not synchronized or fail to move evenly when retracted.	Incorrect rigging of system.	Adjust flaps.

(e) Grasp flap handle, release plunger, and allow flap return to retracted position.

(f) Disconnect flap return spring at spar or return chain.

(g) Remove cotter pin, nut, and clevis bolt and disconnect control cable from chain.

(h) Remove block attaching bolts and remove tube support blocks.

(i) Remove nuts, washers, and bolts holding right and left cranks and stop fittings on torque tube.

(j) From between each wing and fuselage, remove cranks from torque tube.

(k) Remove nuts, washers, and bolts and disconnect one bearing block from its mounting brackets.

(l) Slide tube from bearing block still attached to its brackets, raise the end, and lift it from floor opening.

(2) Remove flap control cable as follows:

(a) Remove the seats floor panel. Remove the front and rear seats.

CAUTION: FORWARD PRESSURE WILL BE ON THE HANDLE WITH THE TENSION SPRING DISCONNECTED.

(b) Extend flaps to relieve spring tension and disconnect flap tension spring from cable.

(c) Retract flap. Use care as forward pressure will be on the handle.

(d) Remove cotter pin, nut, clevis pin, and bushing to disconnect cable from chain.

(e) Remove flap handle bracket and cover.

(f) Lift aft section of tunnel carpet enough to remove screws holding tunnel cover between flap handle and spar cover. Remove cover.

(g) Remove cotter pin cable guard from flap cable pulley inside floor tunnel just ahead of spar housing.

(h) Remove the attaching screws and remove cable rub blocks located in floor opening on aft side of spar housing.

(i) Remove cotter pin, nut, and bolt and disconnect cable turnbuckle at flap handle. Check clevis bolt for wear. Replace bolt if worn.

(3) Disconnect cable turnbuckle from handle and remove bolts holding bracket to floor tunnel. Remove flap handle and bracket.

B. Installation

Refer to “Figure 1” on page 27503.

(1) Install flap torque tube assembly as follows:

(a) Install chain sprocket with chain on torque tube and secure with bolts, washers, and nuts.

(b) Slide tube stop fittings on respective ends of torque tube.

(c) Check one bearing block fitting is installed between its attachment brackets.

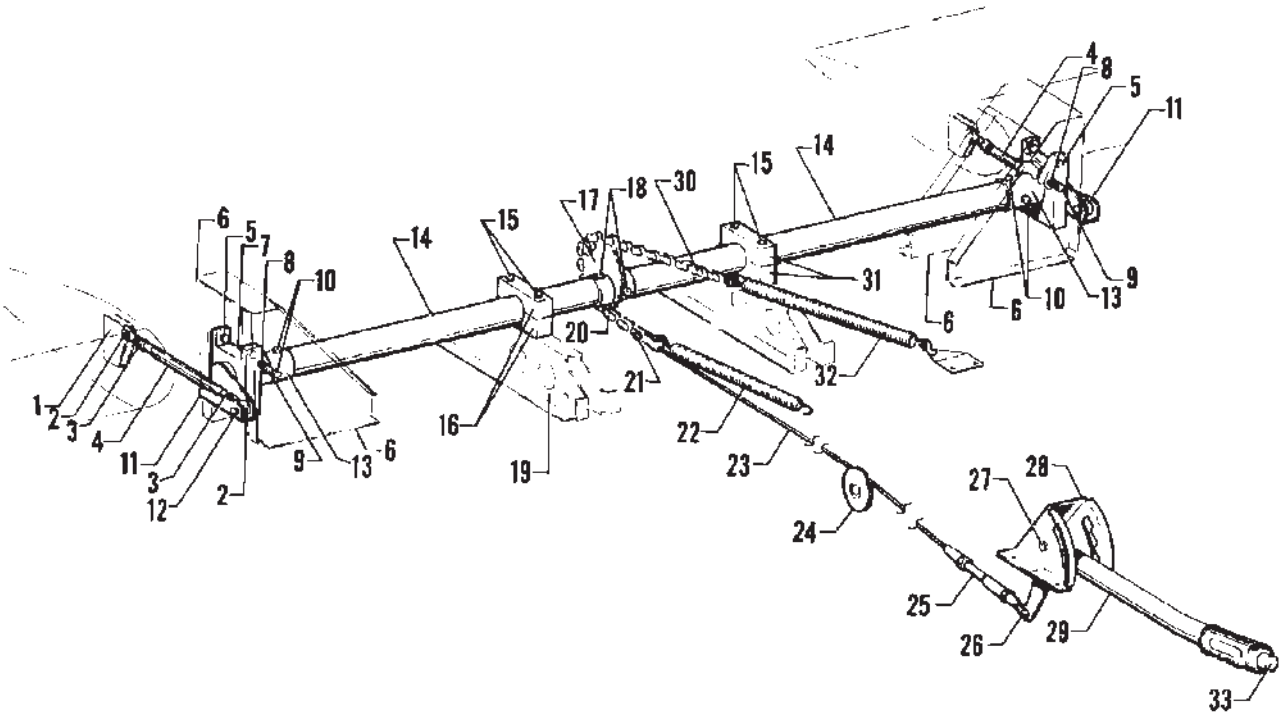
(d) Slide the other bearing block over its respective torque tube end.

(e) Position torque tube end with the bearing block on it between mounting bracket and slide the other end into previously attached bearing block.

(f) Position remaining bearing block and secure with bolts, washers, and nuts.

(g) Push torque tube cranks (arms) on each end of torque tube and slide stop fitting in place. Align crank bolt hole and stop fitting with torque tube holes and install bolts. Holes in stop fitting are elongated to allow stop fitting to be pushed against bearing blocks for no side play of assembly. Tighten bolt assemblies on stop fittings.

(h) Install tube support blocks on support brackets and secure with bolts.



1. BRACKET, ROD ATTACHMENT
2. BOLT, WASHER & NUT
3. JAM NUT
4. ROD, FLAP CONTROL
5. BOLT, BEARING BLOCK ATTACHMENT
6. BRACKET, BEARING BLOCK
7. BLOCK, BEARING
8. NUT, LOCK
9. SCREW, FLAP ADJUSTMENT
10. BOLT, WASHER & NUT
11. CRANK (ARM), TORQUE TUBE
12. BOLT, WASHER & BUSHING
13. FITTING, TORQUE TUBE STOP
14. TUBE, TORQUE
15. BOLT, WASHER & NUT
16. BLOCK, BEARING
17. SPROCKET, TENSION SPRING
18. BOLT, WASHER & NUT
19. BRACKET, BEARING BLOCK
20. CHAIN, TENSION SPRING
21. CLEVIS BOLT, BUSHING NUT & COTTER PIN
22. SPRING, TENSION
23. CABLE, FLAP CONTROL
24. PULLEY
25. TURNBUCKLE
26. CLEVIS BOLT, NUT & COTTER PIN
27. BOLT, BUSHING, WASHER & NUT
28. BRACKET, FLAP HANDLE
29. HANDLE, FLAP
30. CHAIN, RETURN SPRING
31. BLOCK, BEARING
32. SPRING, RETURN
33. BUTTON, FLAP RELEASE

Flap Control System
Figure 1

(i) Connect flap return spring to return chain and/or at spar housing.

(j) Connect control cable end to tension chain and secure with bushing, clevis bolt, nut, and cotter pin.

(k) Pull flap handle full back and connect tension spring. Release flap handle to forward position.

(l) Connect flap control tube to flap and/or torque tube crank, and secure. Install bolt and bushing (connecting control tube to crank) through a hole in fuselage side over torque tube.

(2) To install flap handle with bracket, place the assembly on floor tunnel and secure with bolts.

(3) Install flap control cable as follows:

(a) Attach cable and turnbuckle to flap handle arm, secure with clevis bolt, nut, and cotter pin. Check turnbuckle end is free to rotate on arm.

(b) Route cable through tunnel and spar housing.

(c) Install cable rub blocks on aft side of spar housing and secure with screws.

(d) Attach cable end to tension chain and secure with bushings, clevis bolt, nut, and cotter pin. (If the chain is not installed due to torque tube assembly being removed, install the assembly per instructions above.

(e) Pull flap handle full back and connect tension spring to cable end.

(f) Install cotter pin cable guard over pulley ahead of spar housing in floor tunnel.

(4) Install tunnel cover and secure with screws. Install tunnel carpet and bracket cover.

(5) Install and secure seats.

C. Rigging and Adjustment

WARNING: VERIFY FREE AND CORRECT MOVEMENT OF FLAPS. WHILE IT WOULD SEEM SELF-EVIDENT, FIELD EXPERIENCE HAS SHOWN THAT THIS CHECK IS FREQUENTLY MISINTERPRETED OR NOT PERFORMED AT ALL. ACCORDINGLY, UPON COMPLETION OF FLAP RIGGING AND ADJUSTMENT, VISUALLY CONFIRM THAT THE FLAP TRAILING EDGE MOVE UP WHEN THE FLAP HANDLE IS PUSHED DOWN; AND, THAT THE FLAP TRAILING EDGE MOVE DOWN WHEN THE FLAP HANDLE IS PULLED UP.

NOTE: Flap rigging and adjustment must be completed before starting aileron rigging and adjustment.

(1) Place flap handle in full forward position (i.e., flaps full up).

(2) Remove rear seat and floor panel. Remove flap handle cover and enough tunnel carpet to remove tunnel cover aft of handle.

(3) Adjust flap up stop as follows:

See “Figure 2” on page 27505.

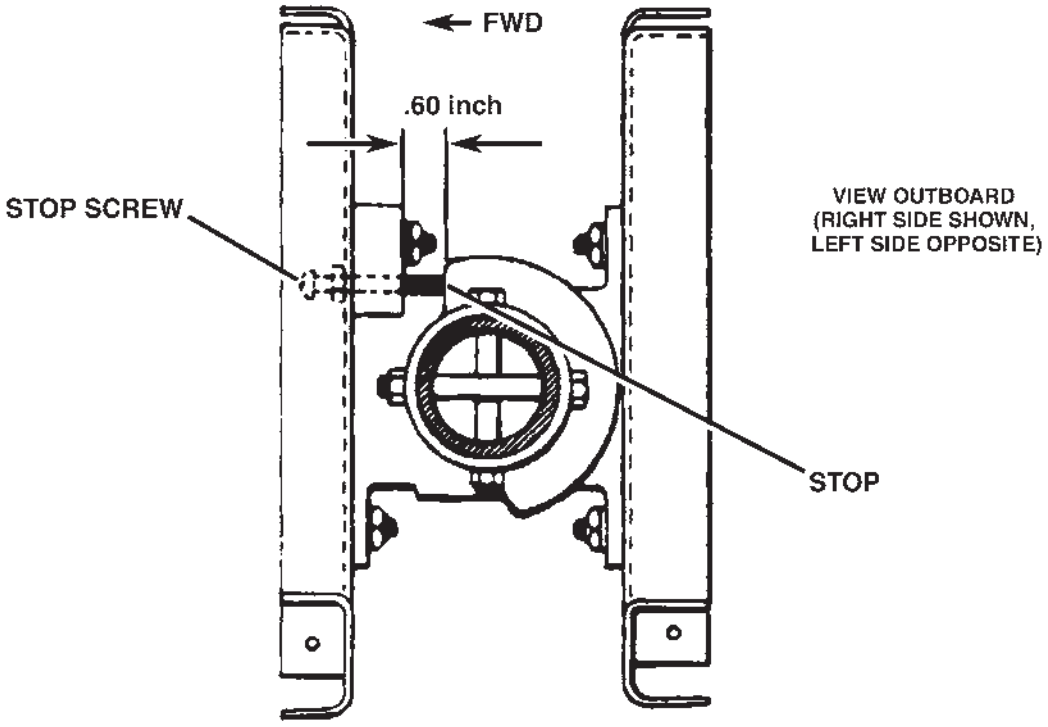
(a) Loosen left stop adjustment screw located in floor opening along outer end of flap torque tube.

(b) Loosen jam nut of right stop screw at opposite end of torque tube.

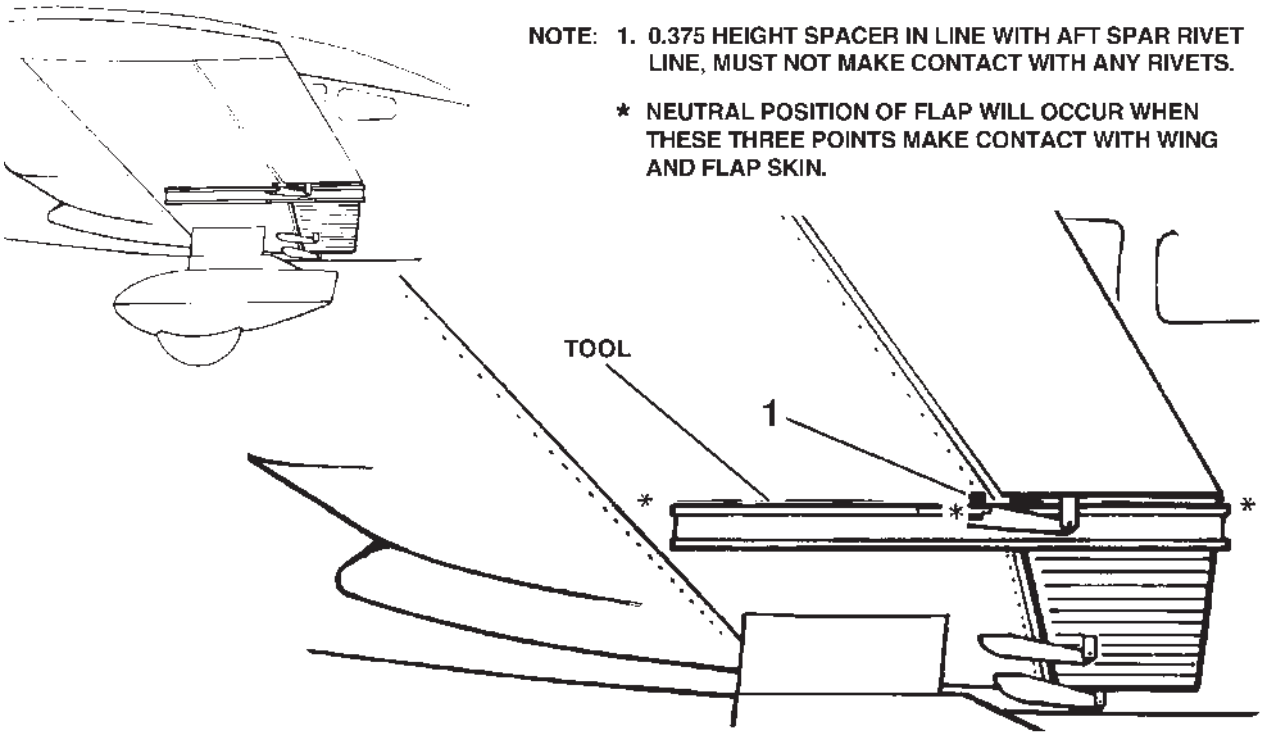
(c) Turn right stop screw to obtain approximately 0.60 inch between stop fitting and bearing block, as measured along top side of screw. Tighten jam nut.

(d) With right stop screw against its stop, adjust left stop screw until it contacts its stop fitting. Tighten jam nut.

CAUTION: DO NOT ROTATE TORQUE TUBE WHILE RETENSIONING CABLE, OR TIGHTEN ENOUGH TO ALLOW TUBE TO BE PULLED AWAY FROM STOPS.



Flap Stop Adjustment
Figure 2



Flap Rigging Tool
Figure 3

(e) Set flap control cable tension (handle next to floor, 0 degrees). Adjust turnbuckle (attached to low end of flap handle in floor tunnel) to remove all slack. Do not tighten cable to the point that the stop screw comes off its stop.

(f) Adjust and safety turnbuckle.

(g) Place a 0.125 inch spacer between the right stop screw and its stop fitting. Check that, when down pressure is applied on right flap, it stays in uplock position. If it extends, turn adjustment screw out (a few threads at a time) until flap remains in uplock position with spacer inserted. Adust left stop screw to match. Remove spacer.

(4) Verify up-neutral position of flaps as follows:

(a) With the flap handle in the "flaps up" position, adjust each flap push rod so that the chord line of the flap forms a zero (0) degree ± 1º angle with the wing chord at the outboard end of the flap. Maintain a light "up" pressure on the underside of the flap (sufficient to take the slack out of the linkage) while making this measurement. Tighten pushrod jam nuts.

NOTE: An alternative method is shown in "Figure 3" on page 27505. Make tool from dimensions given in "Figure 8" on page 271016.

NOTE: To remedy a wing heavy condition during flight, adjust flap down from neutral on heavy wing, by lengthening control rod. Check each rod inspection hole to make sure there are sufficient threads remaining and that a wire cannot be inserted through these holes. Maintain a minimum of 0.375 inch thread engagement on rod ends without check holes. Do not raise the flap of the other wing above neutral.

(5) Measuring from the "flaps up" position attained in the previous paragraph, and maintaining light "up" pressure on the underside of the flap as described therein, check flap travel. Travel shall be as specified in 27-00-00, "Chart 2". If flap travel is not within specifications in 27-00-00, "Chart 2", inspect for damaged components and repeat steps (1) thru (5).

(6) Check operation of the flap and flap handle ratchet mechanism.

(7) Install flap handle cover and tunnel cover and carpet. Install rear seat and floor panel and any other access plates and panels which may have been removed.

D. Flaps Post-Rigging Inspection

NOTE: Visual confirmation may require the aid of a second qualified mechanic or pilot.

Upon completion of any flap handle, cable, pushrod, or torque tube replacement, disconnect, or rigging perform the following post-rigging inspection:

(1) Position the flap control handle to the flaps DOWN position. Visually confirm the following:

Both flaps are fully DOWN and ALIGNED. That is the trailing edge of each flap is significantly LOWER than when streamlined with the wing.

(2) Position the flap control handle to the flaps UP position. Visually confirm the following:

Both flaps are fully UP and ALIGNED. That is the trailing edge of each flap is streamlined with the wing.

(3) Make a logbook entry documenting completion of this inspection.

4. Flap Position Sensor (Optional)

A flap position sensor installation is available in airplanes equipped with Garmin G1000 NXi.

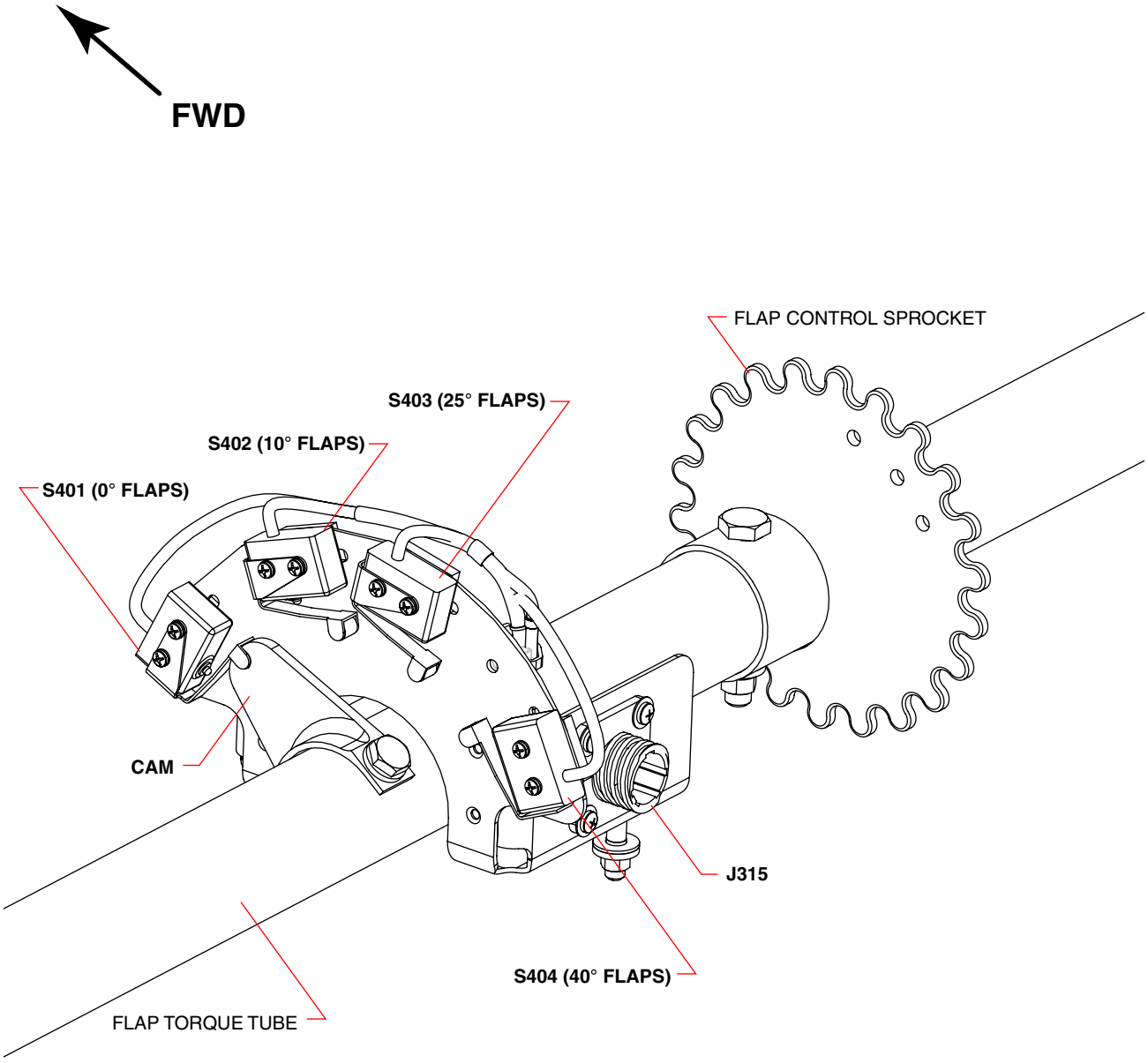
It consists of four (4) switches mounted on a bracket over the flap torque tube. The switches are activated as the flaps are raised and lowered by a cam mounted to the flap torque tube. When activated, each switch provides a signal to the GEA 71.

Rigging

See “Figure 4” on page 27508.

(1) Prior to rigging the Flap Position Sensor, the flaps must be rigged per “Rigging and Adjustment” on page 27504.
(2) Disconnect the negative (i.e., ground) battery cable.
(3) Remove rear seat and floor panel.
(4) Raise flaps to the 0 deg position. Adjust the position of the Flaps 0 deg switch (S401) so that the cam makes full contact with the switch arm roller and the switch is actuated.
(5) Lower flaps to the 10 deg position and adjust the position of the Flaps 10 deg switch (S402) so that the cam makes full contact with the switch arm roller and the switch is actuated.
(6) Lower flaps to the 25 deg position and adjust the position of the Flaps 25 deg switch (S403) so that the cam makes full contact with the switch arm roller and the switch is actuated.
(7) Lower flaps to the 40 deg position and adjust the position of the Flaps 40 deg (S404) switch so that the cam makes full contact with the switch arm roller and the switch is actuated.
(8) Connect the negative (i.e., ground) battery cable.
(9) Verify proper rigging:
    (a) Start the PFD and MFD in configuration mode by applying power to each while holding the ENT key.
    (b) On PFD, use the FMS knob to navigate to the GEA71X group, DISCRETE IN page.
    (c) Activate the cursor by pressing in the FMS knob, and use the large FMS knob to scroll down until Discrete In* 11A through Discrete In* 14A are visible on screen.
    (d) Verify Discrete In* 11A is active with the flaps in the full up position, and Discrete In* 12A through 14A are inactive.
    (e) Extend the flaps to the first notch (10°). Verify Discrete In* 12A is active and the other three discretes are inactive.
    (f) Extend the flaps to the second notch (25°). Verify Discrete In* 13A is active and the other three discretes are inactive.
    (g) Extend the flaps to the final notch (40°). Verify Discrete In* 14A is active and the other three discretes are inactive.
    (h) Retract flaps one notch at a time and verify the proper discrete is active at each position.
        1) If successful, turn off the Garmin G1000 NXi system and proceed to step “(10)”.
        2) If not, repeat the rigging procedure.
(10) Install rear seat and floor panel and any other access plates and panels which may have been removed.

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Effectivity
with Garmin G1000 NXi

Flap Position Sensor Installation (Optional)
Figure 4

CHAPTER

28

FUEL

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CHAPTER 28

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GENERAL

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

This chapter contains information on fuel tanks, selector valves, filter screens, and fuel pumps. Instructions are given for difficulties which may arise in the normal operation of the fuel system. Carburetor or fuel injector maintenance is found in Chapter 73.

1. Description

See “Figure 1” on page 28003.

NOTE: See 12-10-00 for anti-icing fuel additives information.

The Archer III is equipped with two aluminum fuel tanks; one in the inboard leading edge section of each wing. Each tank has a 25-gallon capacity. A strainer is installed in the fuel outlet of each tank. A fuel line from the tank outlet is routed through the wings to the fuel selector valve which is located on the left side of cabin in front of pilot’s seat. A line from the fuel selector valve leads to the gascolator located on left forward face of firewall. A fuel line is routed from the gascolator to the electric fuel pump, the engine driven fuel pump, and then to the carburetor (or fuel injector) inlet port.

Beginning in 2020, a fuel return system is optional in airplanes with fuel injected engines. See “Figure 7” on page 282015. Fuel is plumbed from the engine fuel injection servo port using a TSO approved hose and is routed to a fitting in the firewall. Behind the firewall, the fuel system return lines are routed to a duplex fuel selector valve that has provisions for both the main fuel feed system lines as well as the fuel return system lines. The duplex valve allows the system to return fuel back to the tank that is feeding the engine.

A. In airplanes equipped with an O-360-A4M engine (i.e., with a carburetor):

The electric fuel pump is installed on the left forward side of the firewall. The outlet port on the pump is equipped with (or routed to) a tee. One side of the tee is routed through the engine driven pump to the carburetor. The other side of the tee is routed through the primer solenoid valve (clamped to the left side engine mount) to cylinders 1, 2, and 4. In the event of engine-driven fuel pump failure, the electric fuel pump can be activated continuously by an ON-OFF rocker type FUEL PUMP switch located in the overhead switch panel.

The electric fuel primer system may be used for cold engine start. A push ON, spring-loaded OFF primer (PRIME) switch, located adjacent to the starter (START) switch on the overhead switch panel, is used to activate the system. When pushed and held in, the switch activates a relay, which bypasses the FUEL PUMP switch, and operates the electric fuel pump. At the same time, a solenoid valve is opened electrically, permitting fuel to be pumped directly to cylinders 1, 2, and 4.

Fuel pressure (fuel flow with Avidyne Entegra or Garmin G1000) is provided to the triple gauge by a transducer fed from a tee in the fuel line at the inlet port of the carburetor. See 73-30-00.

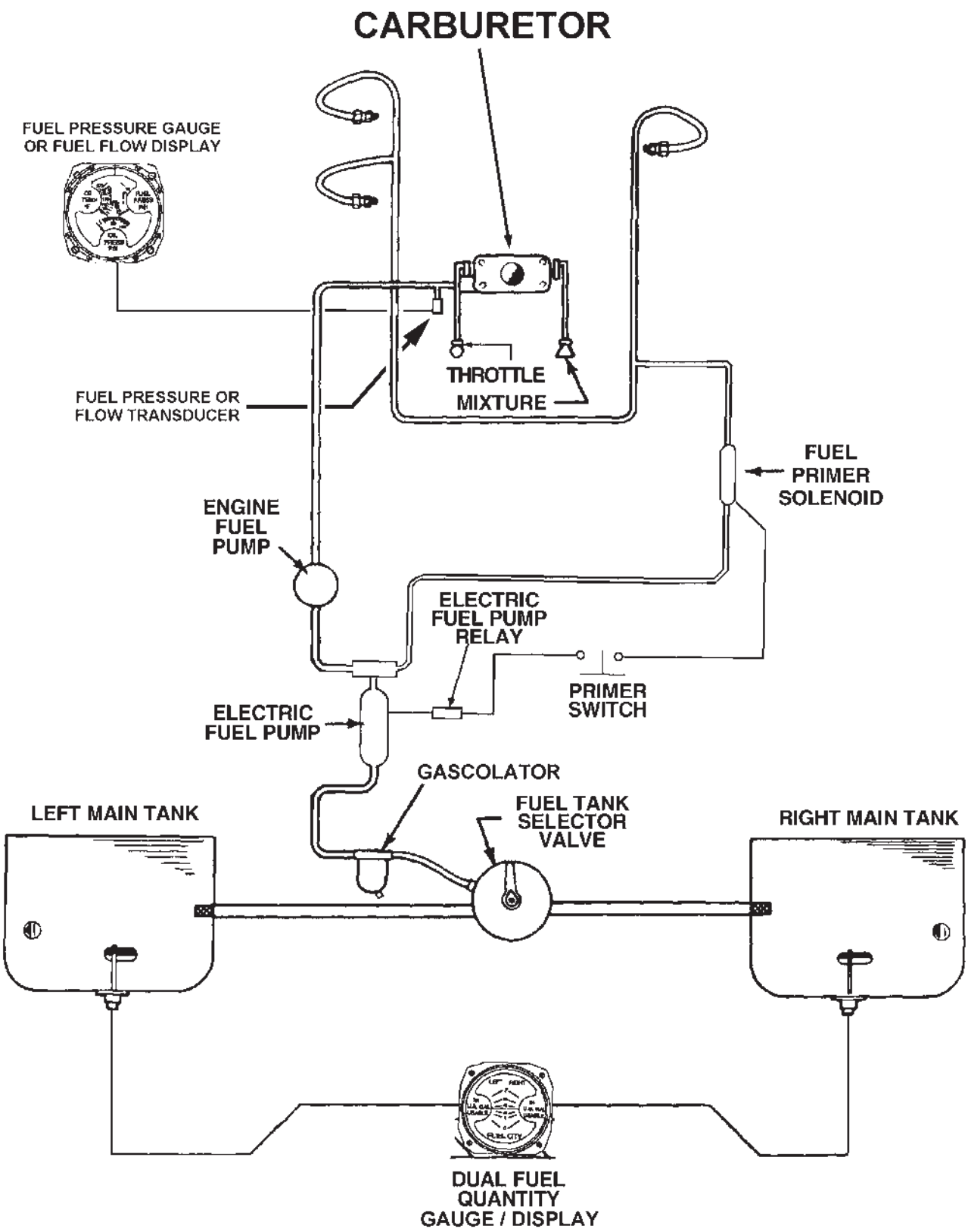
B. In airplanes equipped with a fuel injected engine:

The electric fuel pump is installed on the left forward side of the firewall. The outlet port on the pump is routed through the engine-driven pump to the fuel injector. In the event of engine-driven fuel pump failure, the electric fuel pump can be activated continuously by an ON-OFF rocker type FUEL PUMP switch located in the overhead switch panel.

Fuel flow data is provided by a transducer in the fuel line between the outlet port of the fuel injector and the fuel flow divider. See 73-30-00.

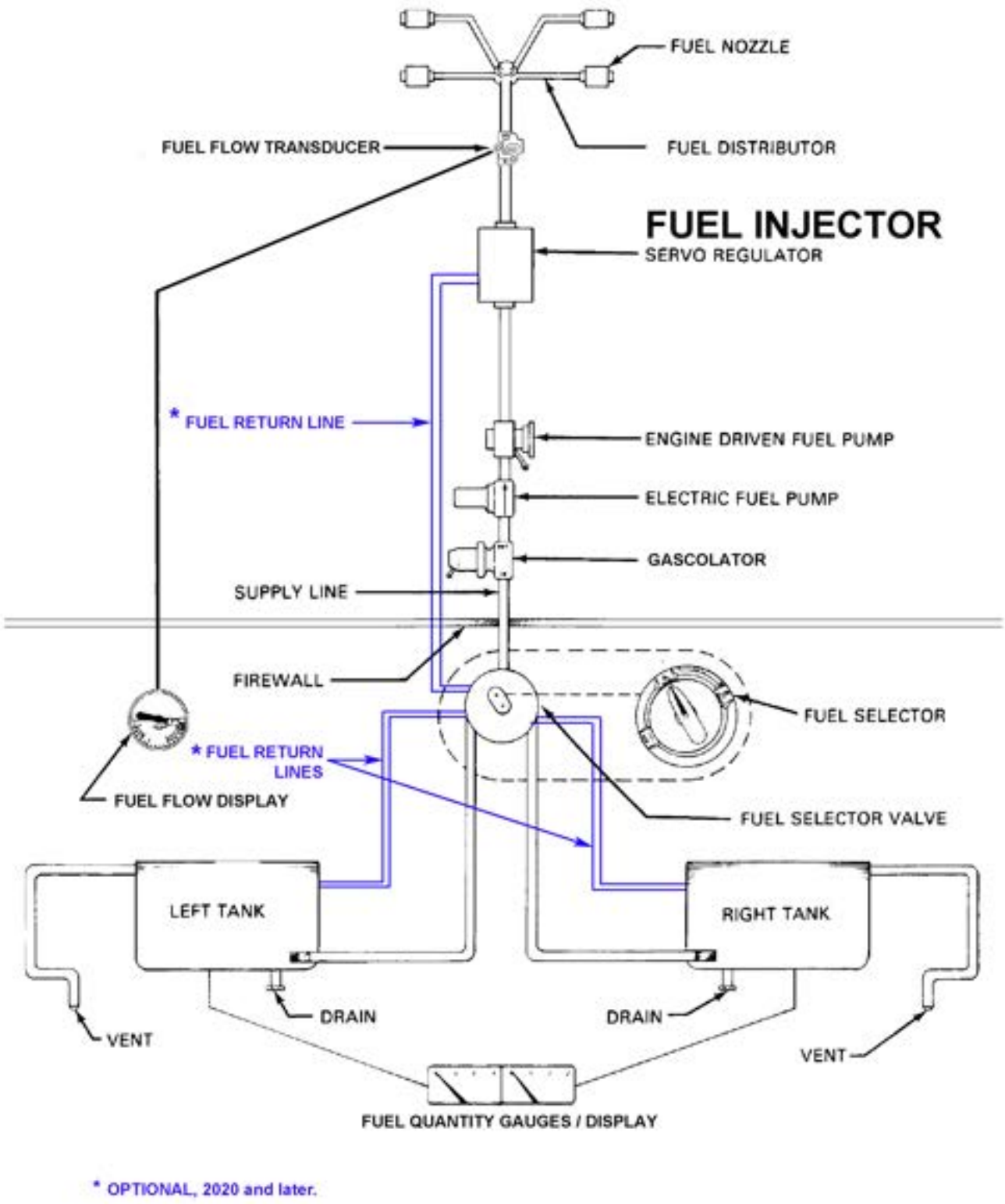
2. Troubleshooting

Troubles unique to the fuel system are listed in “Chart 1” on page 28006 along with their probable causes and likely remedies. When troubleshooting, check from the power supply to the item affected. If no trouble is found with this method, the trouble may be in the individual piece(s) of equipment. Remove and replace faulty equipment.



Fuel System
Figure 1 (Sheet 1 of 2)

Effectivity
with O-360-A4M Engine



Effectivity
with IO-360-B4A Engine

Fuel System
Figure 1 (Sheet 2 of 2)

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CHART 1 (Sheet 1 of 2)
TROUBLESHOOTING FUEL SYSTEM

Trouble	Cause	Remedy
Fuel will not flow.	Blocked fuel line.	Flush fuel system.
	Blocked cap vent.	Check and clean cap vent hole.
	Mechanical or electric fuel pump failure.	Check and replace if necessary.
	Fuel selector not in proper position.	Check selector position and adjust if required.
	Fuel selector valve damage.	Replace fuel valve.
Fuel gauge/display failure.	Broken wire.	Check and repair.
	Gauge/display inoperative.	Replace.
	Fuel inside sender float.	Replace sender float.
	Circuit breaker open or fuse blown.	Check and set or replace.
	Incomplete ground.	Check ground connections at fuel sender in wings.
	Float and arm assembly of fuel sender in wing sticking.	Check fuel sender in wing(s) and repair or replace.
Fuel gauge/display reads full when tanks are not full.	Complete ground on transmitter wire.	Check ground connections at fuel transmitter in wing(s).
No fuel pressure (or flow) reading.	Fuel valve stuck.	Check valve.
	No fuel in tanks.	Check fuel level and fill.
	Defective fuel pump.	Check pump for pressure build up. Check diaphragm and relief valves in engine pump. Check for obstructions in electric pump. Check bypass valve. Check for air leak in intake lines.
.	Defective gauge/display.	Replace gauge/display.
	Fuel selector not in proper position.	Check selector position and adjust if required.
Pressure low or pressure surges.	Obstruction in pump inlet side.	Trace lines and locate obstruction.
	Faulty bypass valve.	Replace.
	Faulty diaphragm.	Replace or rebuild pump.

CHART 1 (Sheet 2 of 2)
TROUBLESHOOTING FUEL SYSTEM

Trouble	Cause	Remedy
Unidentified leak.	Fuel line damaged or improperly installed.	Locate and repair or tighten.
Fuel valve leaks.	Worn O-rings	Replace O-rings or valve.

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STORAGE

1. Fuel Tanks

A. Removal

(1) Drain fuel from fuel tank. (Refer to 12-10-00.)
(2) Remove tank assembly screws. If installed remove door stop.
(3) Remove access panel at WS 54.00 and disconnect fuel line attached to tank.
(4) Pull tank from wing assembly enough to remove sender wire.
(5) Remove tank.

B. Installation

(1) Slide tank into position and connect sender wire.
(2) Slide tank completely into position and attach fuel line.
(3) If door stop was previously removed, be sure to reinstall it while installing and tightening screws around fuel tank.
(4) Fill fuel tank and check for leaks, unrestricted fuel flow, and proper quantity gauge readings. Refer to fuel quantity sender/gauge check (installed).
(5) Position access plate and secure with screws.

C. Inspection

See also"Fuel Tank/Wing Spar Corrosion Inspection" on page 28106.

(1) General
(a) Each 100 hours inspect each tank for signs of leaks (indicated by telltale stains).
(b) If leaks are detected, repair per Repair, below.

(2) Leak Test (PIR-PPS60049-4, Rev. C.)

CAUTION: THE FUEL TANKS MAY BE DAMAGED IF MORE THAN 1.5 PSI PRESSURE IS INTRODUCED INTO THE SYSTEM.

If a fuel tank leak is suspected, the following test can be used to confirm or deny and aid in locating a leak. (Refer to "Figure 1" on page 28103.)

(a) Setup
1) Ensure the fuel lines and components of the fuel system are readily accessible.
2) Defuel the airplane (or just the wing being tested) per Draining Fuel System, 12-10-00, prior to performing the following test.
3) A test fixture containing the following features is required to perform this test and the Fuel System Leak Test in 28-20-00: (See "Figure 1" to fabricate.)

- Two air filters
- Pressure Regulator
- Two fuel filler test caps each incorporating a 5 psi gauge and a shut-off valve
- Main pressure gauge (0 - 5 psi; marked red above 1.5 psi)
- Two main air supply shut-off valves (Valves #1 and #2)
- One main air supply needle valve (Valve #4)
- Manometer (with AIRCRAFT and REFERENCE sides labeled)
- Manometer needle valve (Valve #3)
- Flow meter (0 - .5 SCFH)
- Two flow meter shut-off valves

4) Ensure appropriate caps are available for closing open lines and vents.
5) Apply bubble testing liquid (P/N 279-246) to surfaces being checked for leaks.

(b) Procedure

NOTE: The flow meter and associated shut-off valves are not used in this procedure.

1) Cap all fuel vent outlets on the wing(s) being tested.
2) Verify fuel sump drain valve(s) (located underneath the LH and RH wings) are in the closed position.
3) Cap all fuel lines exiting the fuel tank(s) being tested.
4) On the test fixture, ensure that the following valves are selected to the OFF position:
   a) #1 and #2 main air supply shut-off valves
   b) Main air supply needle valve, valve #4
   c) Shut-off valve on each fuel test cap
5) Verify that the Manometer needle valve, valve #3 is OPEN.

   CAUTION: DO NOT OVERTIGHTEN.

6) Install special test unit fuel cap(s) at the fuel filler port in the wing(s) being tested. Ensure test unit fuel cap(s) is snugg, but do not overtighten.

   NOTE: If one wing is not being tested, attach that wing's special test unit fuel cap to the test fixture air supply line and leave the test unit fuel cap valve closed throughout the procedure.

7) Connect shop air to test fixture.

   CAUTION: DO NOT EXCEED THE 1.5 PSI LIMIT OR FUEL TANK DAMAGE MAY RESULT.

8) Select the test fixture "Valve 2" to the "ON" position, and using "Valve 1," slowly pressurize the line to 1.0 psi minimum; 1.5 psi maximum.

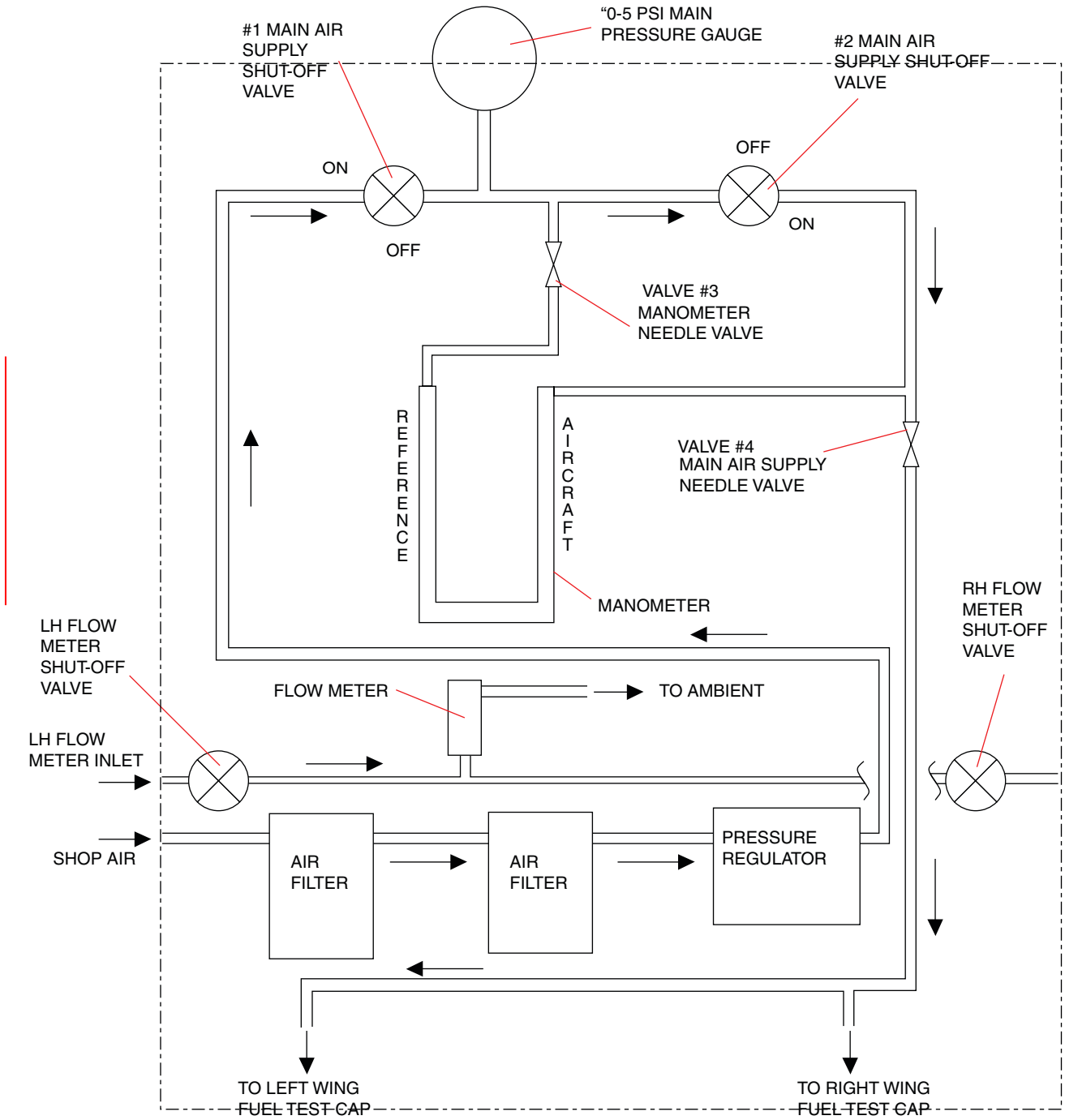
   NOTE: Monitor pressure on the main pressure gauge. Do not let air into aircraft fuel tanks if pressure reads above 1.5 psi.

   CAUTION: DO NOT EXCEED THE 1.5 PSI LIMIT OR FUEL TANK DAMAGE MAY RESULT.

9) Open the test cap shut-off valve(s) for the tank(s) being tested. Slowly open "Valve 4" and monitor pressure on the main pressure gauge. Adjust pressure to read between 1.0 psi minimum to 1.5 psi maximum.
10) Select the test fixture "Valve 1" to the "OFF" position, then "Valve 2" to the "OFF" position. Observe the position of test fluid in manometer. Both sides should be equal in height.
11) Disconnect shop air from the test fixture and monitor the leakage rate.

    NOTE: No leakage should occur over a five (5) minute period.

    a) If "AIRCRAFT" side of manometer moves upwards during 5 minute hold, then a leak exists at the connections. If leak exists, bubble testing liquid shall be applied to all fuel system connections during the hold period, and each connection shall be inspected for bubble formation.
    b) If "REFERENCE" side of manometer moves upwards, then there exists a temperature/pressure differential. Releasing half of the air in the system and then refilling the system may solve this issue.
12) If leaks are found in fuel tank(s) being tested, repair per "Repairs" on page 28104.



Fuel Tank/System Leakage Test Fixture
Figure 1

13) Remove all fuel vent caps (LH and RH wings) and verify that the remaining fuel tank pressure bleeds off through each vent.

14) Remove all test equipment and reconfigure aircraft fuel system to flight configuration.

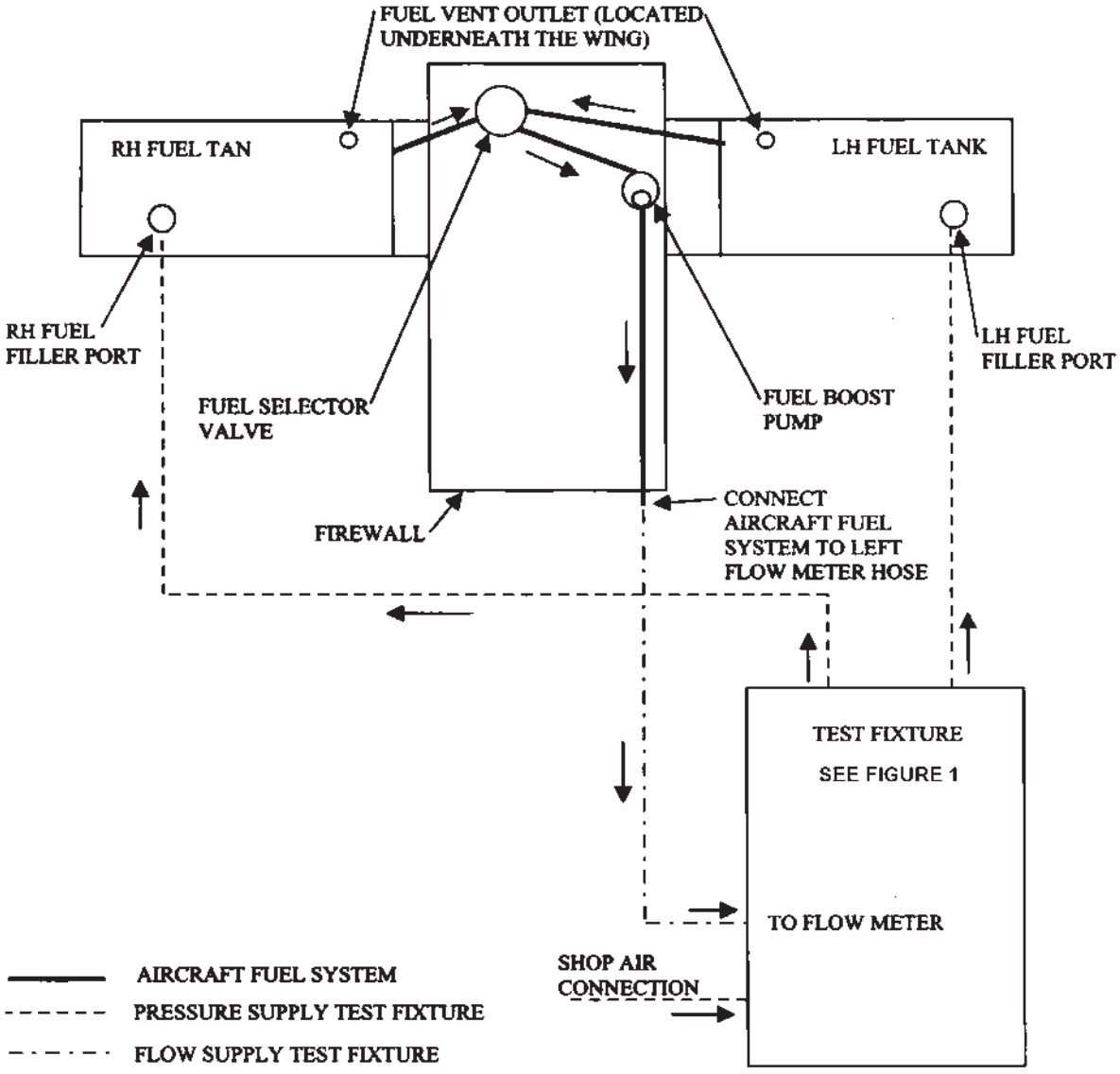
D. Repairs

WARNING: SLOSHING OF FUEL TANKS IS PROHIBITED.

(1) If leaks are detected in fuel tank skin joints, seal with Products Research Corporation (PRC) PR1422A series or PR1433G series sealant.

(2) For leaks in other areas, if upon inspection and tightening leak cannot be stopped, component replacement is required.

NOTE: The fuel tank should be replaced if the tank is damaged to the extent it cannot be repaired by the above method.



Fuel System Leakage Test Schematic
Figure 2

2. Fuel Tank/Wing Spar Corrosion Inspection

NOTE: The following inspection incorporates the requirements of Piper Service Bulletin No. 1006.

Each seven (7) calendar years time-in-service, remove the fuel tanks per Removal under paragraph 1 in 28-10-00 and inspect as specified below.

NOTE: Disassemble as necessary to determine the condition. This includes gaining access as necessary by removing panels or skin, removing rivets, removing paint and/or the use of NDT as required.

NOTE: Sub-paragraphs A and C are keyed to “Figure 3”.

A. Visually inspect for corrosion (intergranular, exfoliation, etc.). Minor surface oxidation is acceptable. Any corrosion exceeding that requires repair or replacement. Additionally, while all areas and parts are exposed check for other anomalies such as damage, cracking, or wear. Any part or area determined to be defective must be repaired or replaced using standard FAA-approved parts and methods. See “Figure 4” for examples of corroded and clean corrosion-free surfaces.

A1 Inspect the fuel tank attach hardware and gang channels (nut plate strips).

WARNING: SLOSHING OF FUEL TANKS IS PROHIBITED. ANY EVIDENCE OF SLOSHING COMPOUND REQUIRES REPLACEMENT OF THE FUEL TANK.

A2 Inspect the exterior of the fuel tanks for leaks. Inspect the interior for corrosion or sloshing compound. If either condition exists, clean, repair, or replace the fuel tank as required.

A3 Remove fuel quantity senders. Inspect for condition, operation, and security and freedom of movement of the float arm. Inspect condition of wires and terminals. Replace components as required. Replace gaskets.

A4 Inspect hard fuel vent lines for interior and exterior corrosion, wear, or deposits. Flush and clean with mineral spirits under pressure through a clean cloth. If excessive debris, deposits, or corrosion observed, replace the line.

A5 Replace flexible fuel vent hoses / couplings.

A6 Inspect the entire spar, all spar angles (cap), and ribs behind and adjacent to the fuel tank. If corrosion is detected, conduct a thorough inspection of the entire wing.

A7 Remove the lacing at the wing root and inspect the spar and forward wing attach fittings. In addition to visually inspecting for corrosion, also visually check for loose or damaged nuts or bolts.

B. If replacement of parts is required, refer to the current revision of the PA-28-181 Archer III Parts Catalog, P/N 761-898.

C. Treat the following areas with Dinotrol AV 8 (Piper P/N 89500-800) before reassembly.

NOTE: Ensure all areas to be treated with corrosion preventative are clean and any paint or primer is dry and fully cured.

B1 Wing spar at root and forward wing attach fittings.

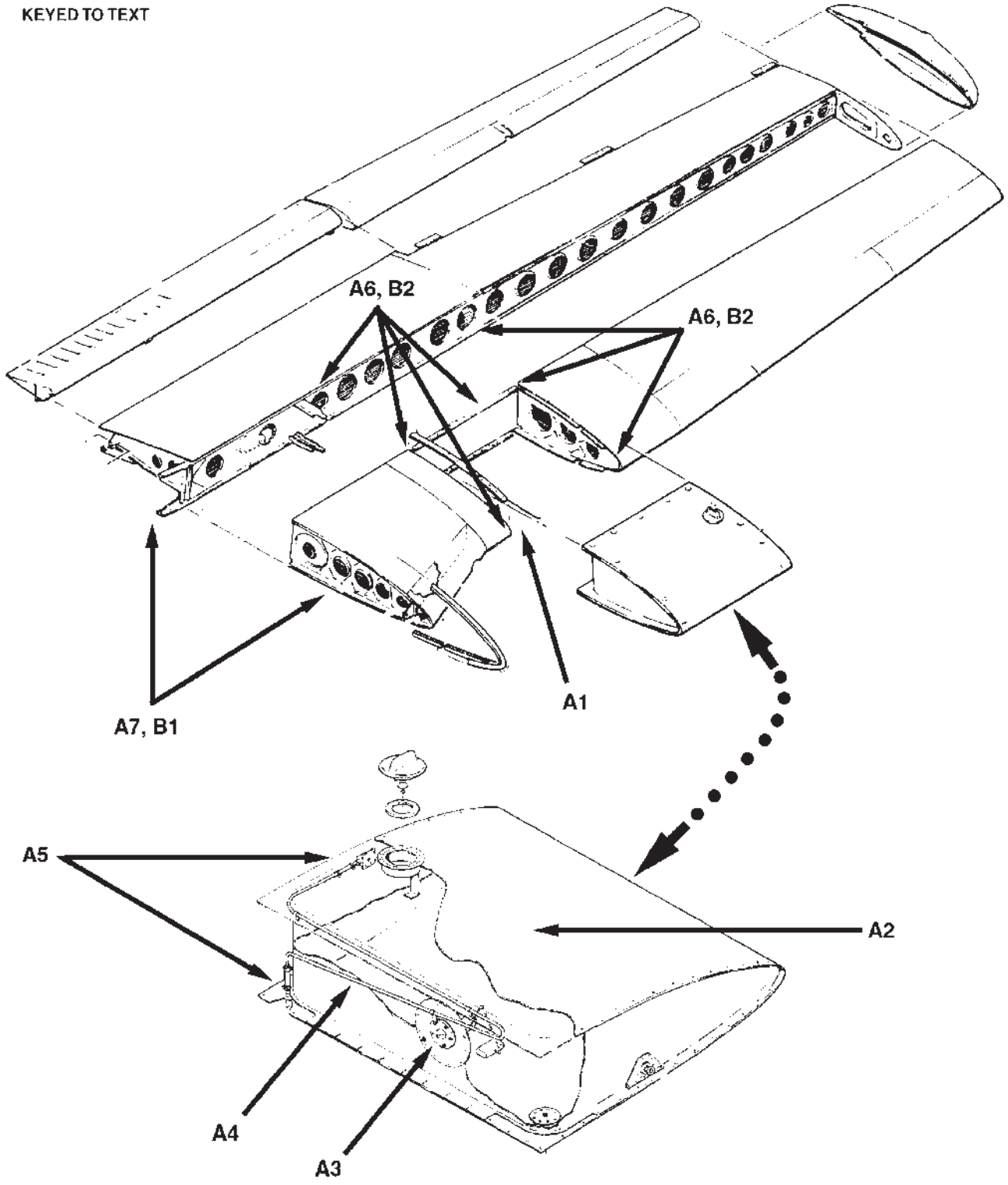
B2 Entire wing spar, spar angles (cap), and ribs behind and adjacent to the fuel tank.

D. Reinstall the fuel tanks per Installation under paragraph 1 in 28-10-00.

E. Prior to fueling the airplane, perform Fuel System Leak Test, 28-20-00.

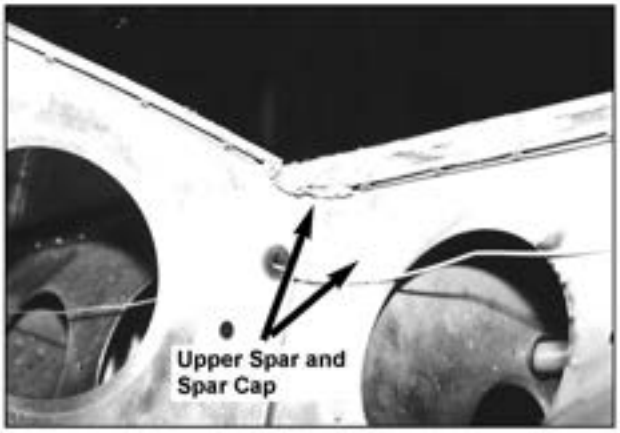
F. Prior to returning the aircraft to service, perform operational checks on any area or system disturbed by inspection, repair, or part replacement. Refer to the appropriate chapter for the applicable system for detailed information (i.e., Chapter 28 for fuel quantity, fuel feed, etc.).

G. Upon successful completion of this inspection, make an appropriate logbook entry documenting compliance.



Fuel Tank / Wing Spar Corrosion Inspection
Figure 3

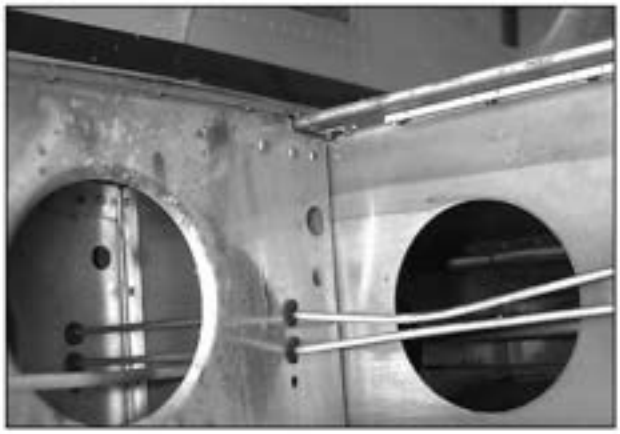
CORROSION EXAMPLES
(Spar Cap and Tank Bay)
below

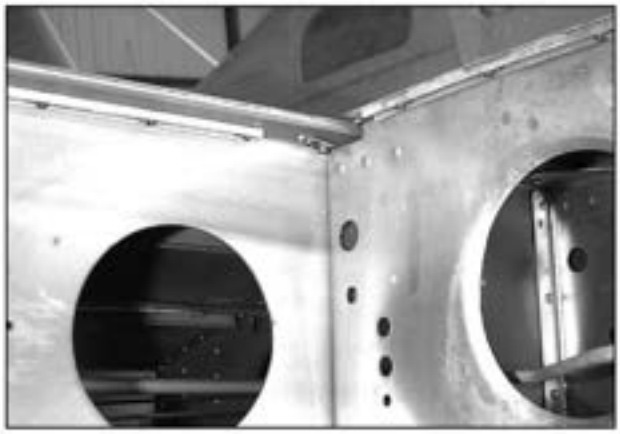












TYPICAL CLEAN INSPECTION
(Spar Cap and Tank Bay)
above

Fuel Tank / Wing Spar Corrosion Examples
Figure 4

3. Locking Fuel Cap

Refer to “Figure 5”.

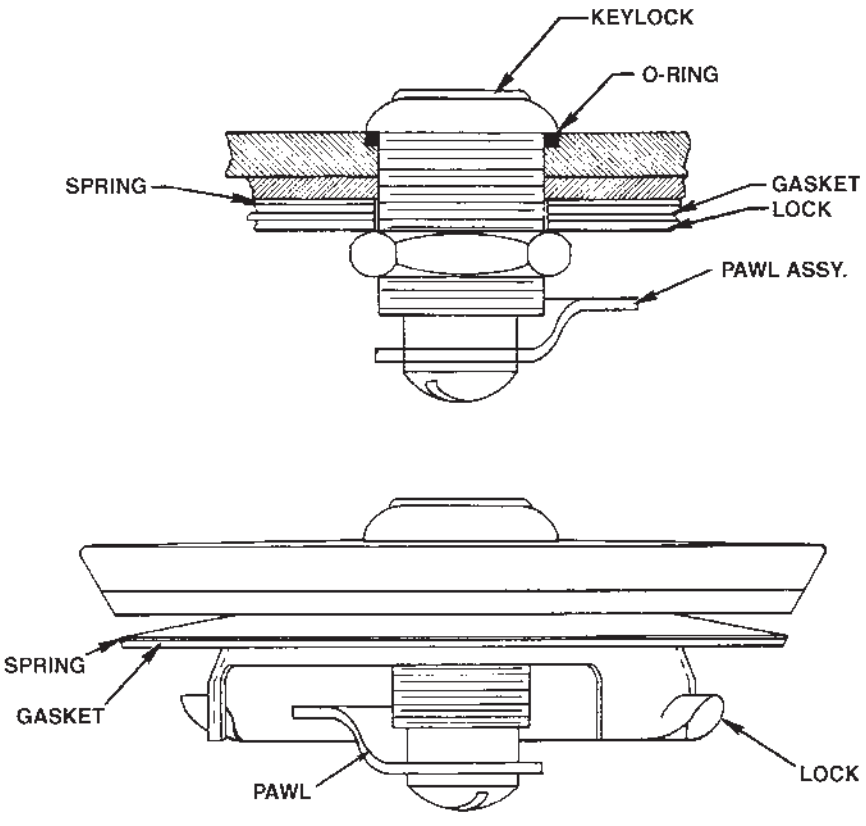
NOTE: The following instructions are not applicable to the new style (circa 2007) locking fuel cap, P/N 654-556 or 654-572.

A. Disassembly

(1) Remove the two screws from top of fuel cap.
(2) Remove screw and lock washer that secures pawl to bottom of key lock. Remove pawl.
(3) Remove nut that secures key lock to cover.
(4) Slide lock, gaskets, and spring over back of key lock.
(5) Remove key lock by pushing keylock through cover. Ensure that the O–ring is not lost.

B. Assembly

(1) Insert key lock through cover. Make sure that O–ring gasket is installed under head of key lock.
(2) Slide spring, gaskets, and lock over back of key lock.
(3) Install nut that secures key lock to cover.
(4) Apply loctite #211 or #680 to screw threads primed with Loctite 7649 N or 7471 T. Attach pawl to back of lock assembly with screw and lock washer.
(5) Apply a thin coating of PR-1422 sealant to shank and thread of the screws removed from top of cap. Install screws and lockwashers on top of fuel cap.



Locking Fuel Cap (Old Style)
Figure 5

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DISTRIBUTION

CAUTION: DRAIN THE FUEL SYSTEM PER THE PROCEDURES IN SECTION 12-10-00 BEFORE DISCONNECTING ANY COMPONENTS OF THE FUEL DISTRIBUTION SYSTEM.

1. Fuel Selector Valve

CAUTION: NO FIELD DISASSEMBLY OR REPAIR OF FUEL SELECTOR VALVES IS AUTHORIZED. MAINTENANCE IS LIMITED TO REMOVAL AND REPLACEMENT OF THE WHOLE UNIT.

Standard equipment.

A. Removal

(1) Remove three screws holding selector cover and screw holding handle. Remove side panel to access selector valve.
(2) Remove selector handle and cover.
(3) Disconnect fuel lines from selector valve.
(4) Remove fuel valve assembly attaching screws to remove fuel valve assembly.

B. Installation

(1) Secure valve to bulkhead attachment location with attaching screws.

CAUTION: SEAL ALL PIPE THREADS WITH LOCTITE NO. 567 PST (P/N 179-771)

(2) Connect fuel lines to valve.
(3) Install side panel.
(4) Install selector cover with attaching screws.
(5) Install valve control handle with attaching screws.
(6) Perform leak test per steps (7) and (8) in “Inspection and Tightening” on page 282013.

2. Duplex Fuel Selector

Installed with Optional Fuel Return System for Fuel-Injected Engines (2020 and later).
See “Figure 7 (Sheet 2 of 4)” on page 282016.

A. Removal

(1) Remove screw and selector lever.
(2) Remove four (4) screws and washers securing the faceplate, cover/side panel, and fuel valve assembly.
(3) Remove faceplate and cover/side panel to access fuel valve assembly.
(4) Disconnect and cap six (6) fuel lines from fuel valve assembly.
(5) Loosen the fastener connecting the LH bracket assembly to the longeron and swing the LH bracket assembly out of the way.
(6) Remove fuel valve assembly from bracket assemblies.

B. Installation

(1) With the LH bracket assembly swung out of the way to provide clearance, position the fuel valve assembly to the RH bracket assembly and then swing the LH bracket assembly into position and tighten screw, washer, and nut to secure LH bracket assembly to the longeron.
(2) Remove caps and connect six (6) fuel lines to fuel valve assembly. Torque fittings as specified in “Figure 7 (Sheet 2 of 4)” on page 282016.

(3) Temporarily secure faceplate and fuel valve assembly to left and right bracket assemblies with four (4) washers and screws.

(4) Install selector lever and secure with attaching screw.

(5) Perform leak test per steps (7) and (8) in “Inspection and Tightening” on page 282013.

(6) Upon successful completion of leak test, remove selector lever and screw, four (4) washers and screws, and faceplate.

(7) Install cover/side panel and faceplate and secure cover/side panel, faceplate, and fuel valve assembly with four (4) washers and screws.

(8) Install selector lever and secure with attaching screw.

3. Gascolator (aka Fuel Strainer)

Refer to “Figure 1” on page 28203.

A. Removal

CAUTION: REMOVE POWER FROM AIRPLANE BEFORE BEGINNING PROCEDURE.

(1) Pull the main power circuit breaker.

(2) Check fuel selector valve is in the OFF position.

(3) Remove engine cowling per Chapter 71.

(4) Place a container under gascolator to catch fuel.

(5) Open drain valve and drain gascolator bowl.

(6) Disconnect fuel lines from gascolator body.

(7) Cut safety wires, loosen bail nut, move bail wire to the side, and remove bowl.

(8) Spread the ends of the bail wire to remove gascolator body from mounting bracket.

B. Installation

(1) Position the gascolator body to mounting bracket, ensuring the word “IN” (on top of the inlet port) is positioned aft toward the firewall.

(2) Connect fuel lines.

NOTE: Before connecting outlet line, ensure that the stainless steel sleeve on the outlet tube assembly has been sprayed with MS122DF lubricant. Take care to ensure that no lubricant is sprayed on the threads of the nut.

(3) Spread bail wire ends and insert through holes in the side of mounting bracket and gascolator body.

(4) Position gascolator bowl and bail wire. Hand-tighten bail nut, then tighten one flat (i.e., 60 degrees) with a wrench. Do not over tighten.

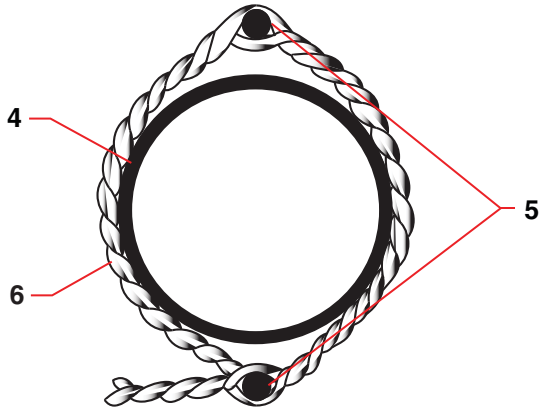
(5) Safety bail nut and bail wire with MS20995-C32 safety wire as shown in Figure 1. Use the “double twist” method as described in FAA Advisory Circular (AC) 43.13-1, latest revision, Chapter 7 (Aircraft Hardware, Control Cables, and Turnbuckles), Section 7 (Safetying). Download from the FAA web site: http://rgl.faa.gov/

CAUTION: TAKE CARE TO ENSURE NO FIRE HAZARD EXISTS BEFORE RESTORING POWER TO AIRPLANE.

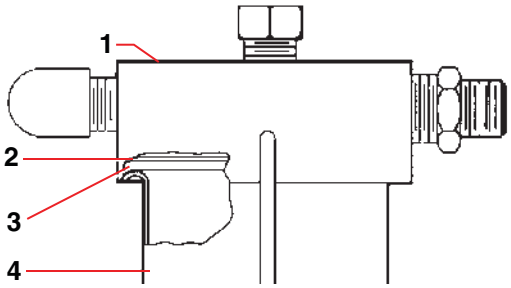
(6) Reset main power circuit breaker.

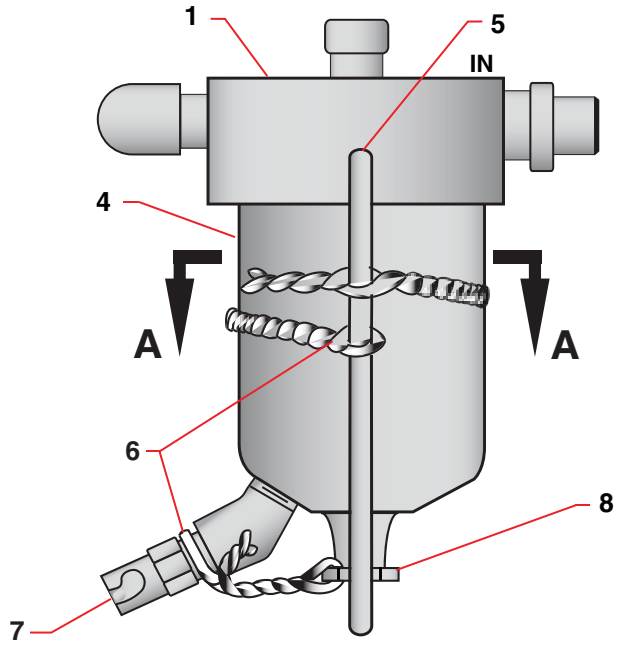
(7) Turn ON fuel pump and pressurize system. Check for leaks. Correct as required.

(8) Install engine cowling per Chapter 71.



Section A-A





1. COVER
2. SCREEN
3. GASKET
4. STRAINER BOWL
5. BAIL
6. SAFETY WIRE MS20995-C32
7. DRAIN VALVE
8. BAIL NUT

Gascolator (aka Fuel Strainer)
Figure 1

C. Cleaning and Inspection

(1) Prepare airplane and drain gascolator per steps (1)–(5), in Removal, above.

(2) Cut safety wires, loosen bail nut, move bail wire to the side, and remove bowl.

(3) Remove gasket and screen from gascolator body.

(4) Clean screen and bowl with acetone or a suitable dry type solvent.

(5) Inspect screen, gasket, bowl, gascolator body, bail wire and bail nut. Replace any worn or damaged components.

(6) Position gasket and screen in body.

(7) Install strainer bowl per steps (4)–(8), in Installation, above.

4. Electric Fuel Pump

The Archer III is equipped with a 28 Vdc electric fuel pump which is mounted on the left forward side of the firewall. Two different types of pumps are installed.

A. Facet Pump

Installed in S/N's 2843001–2843820, 2843822–2843851, 2843853–2943949, and 2881013–2881082 with O-360-A4M engines only. See "Figure 2" on page 28206 and "Figure 3" on page 28207.

(1) Removal

(a) Turn the master switch OFF.

(b) Remove the cowling by releasing cowling fasteners and attaching screws.

(c) Disconnect the negative (ground) battery cable (and, if Garmin G1000 equipped, the emergency battery P615 connector).

(d) Check fuel selector valve is in the OFF position.

CAUTION: CHECK ALL ELECTRICAL LEADS ARE DISCONNECTED BEFORE FUEL PUMP REMOVAL.

(e) Disconnect fuel pump electrical leads.

(f) Open drain valve on gascolator to drain fuel lines. Close drain valve when finished.

(g) Disconnect the fuel lines from each side of the tee fitting at pump outlet port.

(h) Disconnect fuel supply line from pump inlet port.

(i) Remove bolts (2 ea.) holding pump to the firewall and remove fuel pump and spacer.

(2) Installation

NOTE: If installing a replacement pump, be sure to transfer the tee fitting from the old pump to the new, if the new did not come with a tee fitting.

(a) Position fuel pump and spacer to firewall and secure with bolts (2 ea.).

(b) Connect fuel supply line to pump inlet port.

NOTE: Before connecting fuel supply line, ensure that the stainless steel sleeve on the tube assembly has been sprayed with MS122DF lubricant. Take care to ensure that no lubricant is sprayed on the threads of the nut.

(c) Connect the fuel lines to each side of the tee fitting at pump outlet port.

(d) Connect electrical leads to pump.

(e) Connect the negative (ground) battery cable (and, if Garmin G1000 equipped, the emergency battery P615 connector).

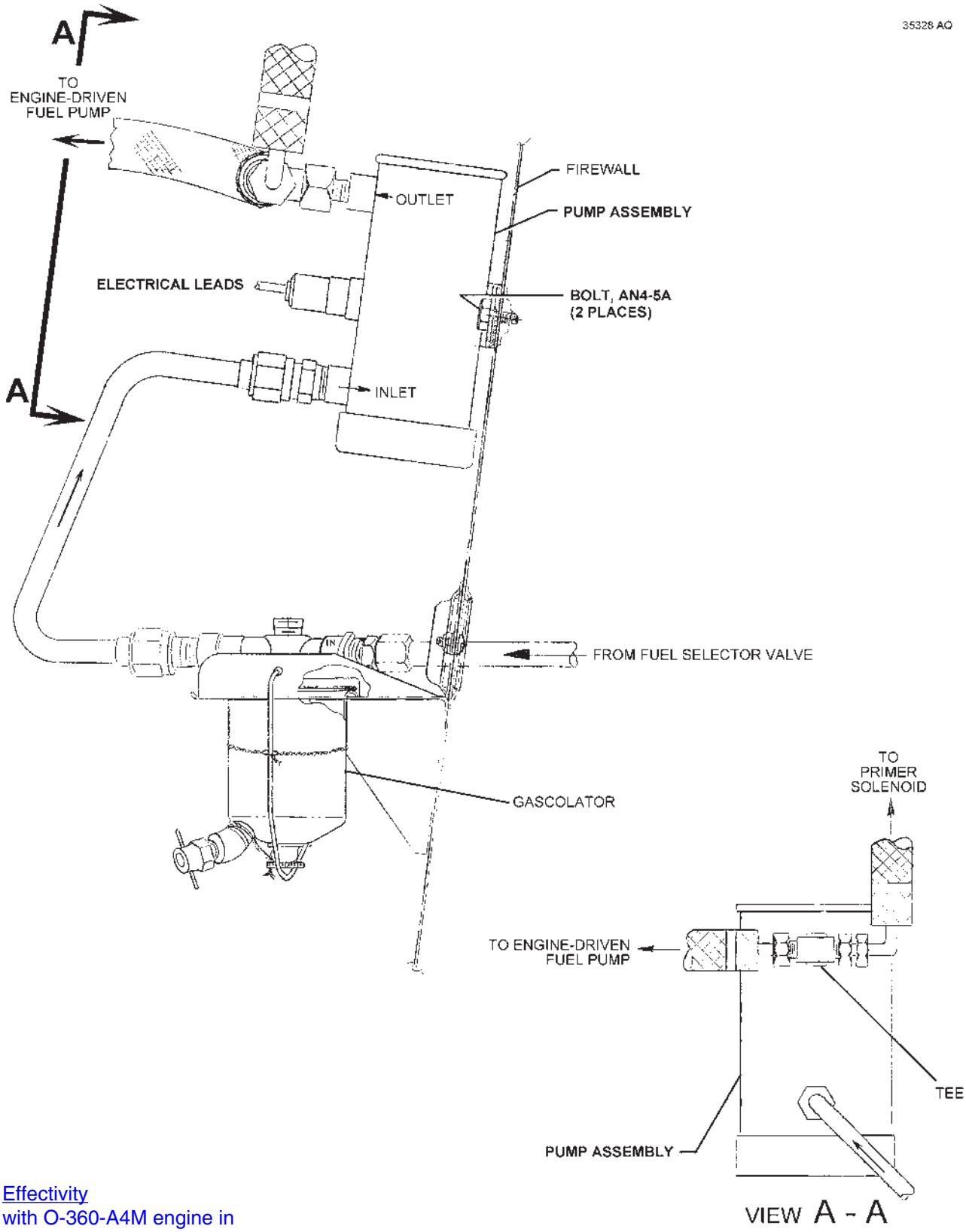
(f) Turn master switch ON.

(g) Turn fuel selector to the desired tank and operate fuel pump. Check all fuel lines and fittings for leakage.

1) If leaks are evident, tighten fittings, repair components as required.

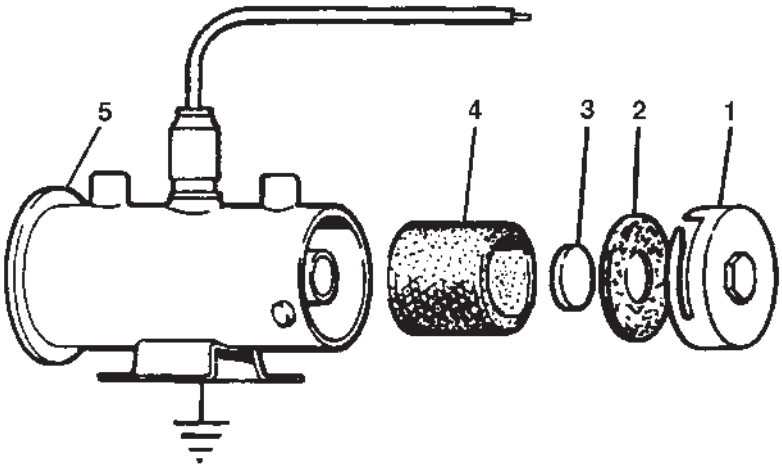
2) When no leaks are detected, proceed to the next step.

(h) Install cowling.



Effectivity
with O-360-A4M engine in 2843001–2843820, 2843822–2843851, 2843853–2943949, and 2881013–2881082 only

Electric Fuel Pump Installation
Figure 2



Electric Fuel Pump
Figure 3

Effectivity
Facet Pump

(3) Disassembly

CAUTION: DO NOT TAMPER WITH SEAL (CENTER OF MOUNTING BRACKET) AT THE SIDE OF THE PUMP. IT RETAINS THE DRY GAS SURROUNDING THE ELECTRICAL SYSTEM IN THE UPPER PORTION OF PUMP.

Disassemble fuel pump as follows: (Refer to “Figure 3”.)

CAUTION: DO NOT DAMAGE THE FILTER SCREEN (4) WHEN REMOVING IT FROM THE PUMP HOUSING.

(a) Cut safety wire and unscrew cover screw.

(b) Remove bottom cover (1), gasket (2), magnet (3), and filter screen (4) from pump.

(4) Cleaning, Inspection, And Repair

NOTE: Pump repair is limited to replacement of the filter screen (4) and cover gasket (2) only.

(a) Clean all parts with acetone or a suitable dry type solvent.

(b) Inspect pump for the following:

1) Check filter screen for damage or distortion.

2) Check cover gasket condition.

(c) Replace defective parts as required.

(5) Assembly

(a) Place cover gasket and magnet in the bottom cover with filter screw.

(b) Carefully guide screen around plunger spring cup. Screen must fit snug at both ends. Do not pinch or distort screw.

(c) Draw bottom cover tight with a wrench, and safety.

(6) Testing

(a) Bench Test

1) Check pump is sufficiently lubricated to prevent damage if run dry for more than five minutes.

2) Connect electrical lead of one pump to 28-volt dc power source.

3) Using a suitable container with the proper octane fuel, connect a fuel line from a container to inlet side of pump.

4) Connect another line from outlet side of pump to a pressure gauge and bypass valve, and back to the container.

5) Run pump with bypass valve open until there is a steady fuel flow. Close bypass valve and check pressure gauge for 4.00 to 4.75 psi maximum, no flow.

6) Replace the complete pump assembly if the proper pressure is not obtained.

(b) In Airplane

1) With cowling removed and fuel selector in the OFF position, remove fuel line from outlet end of pump.

2) Connect a test line with a bypass valve and pressure gauge to pump outlet end.

3) Place a container below pump to catch any fuel from test line during testing.

4) Turn fuel selector ON, open bypass valve on test line, and start pump.

5) When there is a steady fuel flow close bypass valve and check pressure gauge reading. Reading must be 4.00 to 4.75 psi maximum, no flow. (Do not keep bypass valve closed for more than one minute during pump operation and adjustment.)

6) Replace pump assembly if the proper pressure is not obtained.

7) Connect the original fuel line to pump. Open fuel selector and run pump. Check for fuel leaks.

8) Shut off pump, close fuel selector, and replace and secure cowling.

B. Weldon Pump

Installed in S/N's 2843821, 2843852, 2843854–2843935, and 2881001–2881090 with IO-360-B4A engine; and in 2881091 and up with any engine option. See "Figure 4" on page 28209.

NOTE: Authorized service replacement in S/N's 2843001–-2843949 and 2881001–2881090 with carbureted engines. Install Kit 88653-701.

NOTE: This pump is not serviceable. If it fails to perform as designed, it must be replaced.

(1) Removal

(a) Turn the master switch OFF.

(b) Remove the cowling by releasing cowling fasteners and attaching screws.

(c) Disconnect the negative (ground) battery cable (and, if Garmin G1000 equipped, the emergency battery P615 connector).

(d) Check fuel selector valve is in the OFF position.

CAUTION: CHECK ALL ELECTRICAL LEADS ARE DISCONNECTED BEFORE FUEL PUMP REMOVAL.

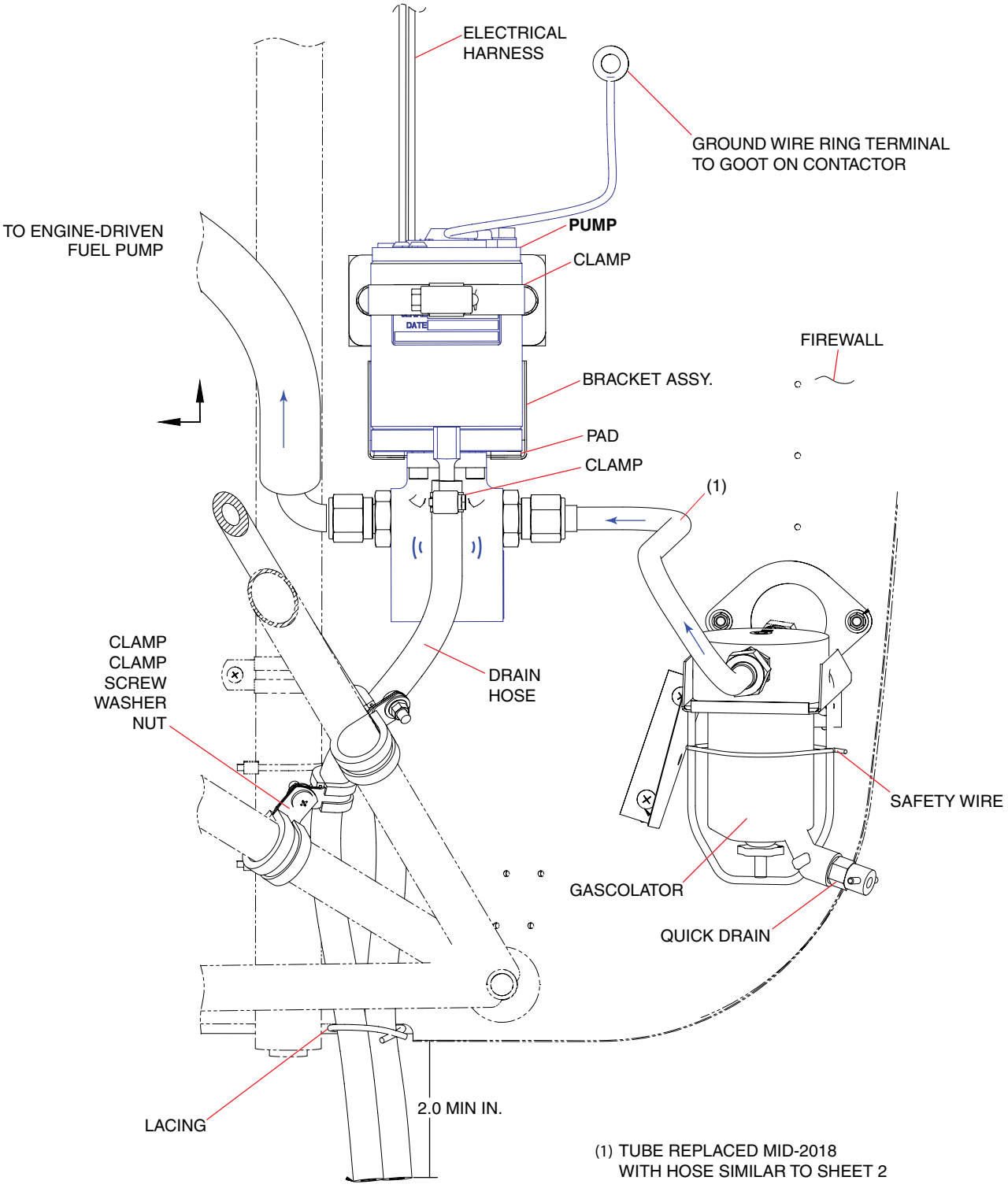
(e) Disconnect fuel pump electrical leads and ground strap.

(f) Open drain valve on gascolator to drain fuel lines. Close drain valve when finished.

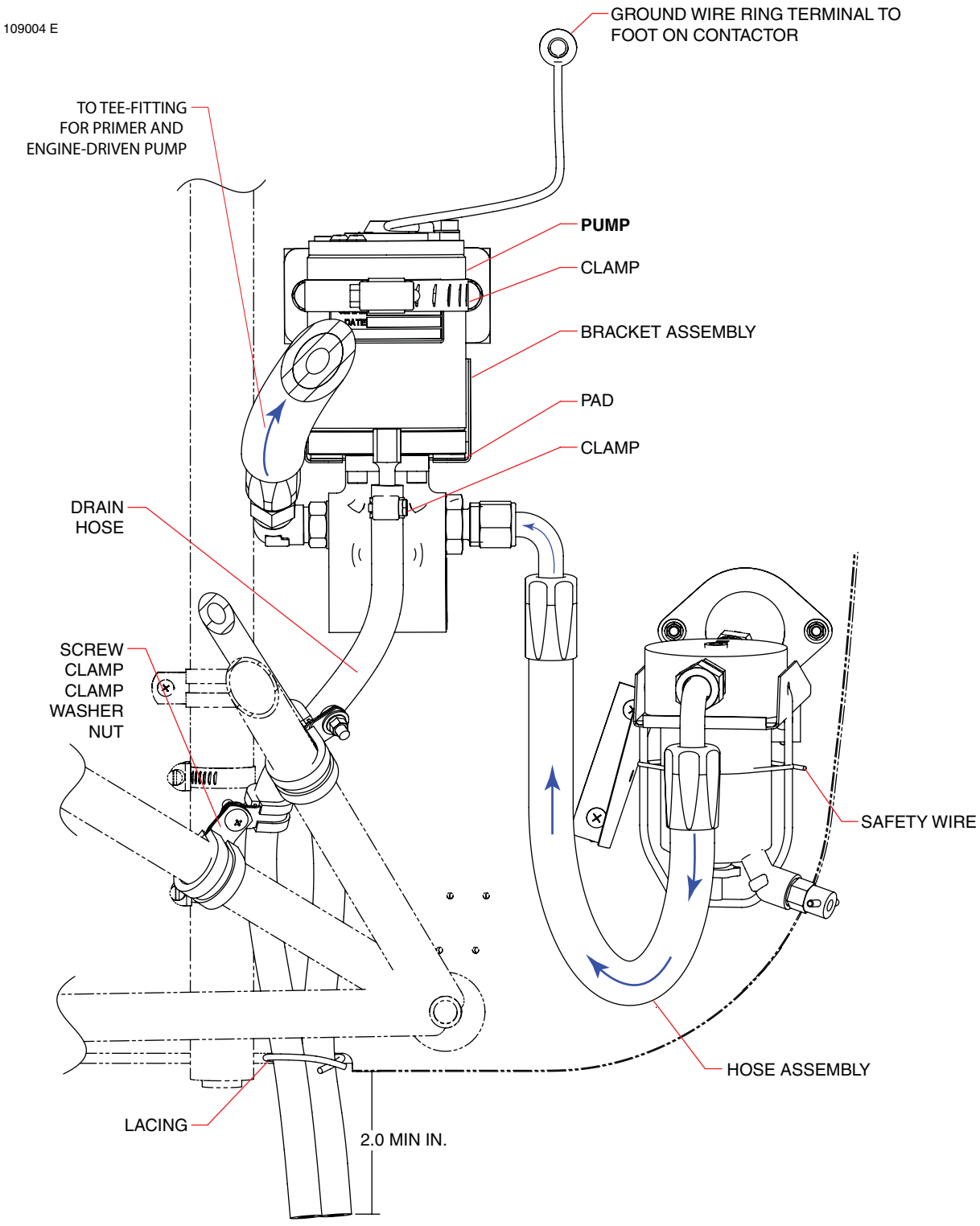
(g) Disconnect the fuel lines from each side of the pump.

(h) Disconnect the fuel pump drain hose.

109004 D



Electric Fuel Pump Installation
Figure 4 (Sheet 1 of 2)



Effectivity
with O-360-A4M engine option in 2881091 and up

Electric Fuel Pump Installation
Figure 4 (Sheet 2 of 2)

(i) Loosen the worm gear clamp holding pump to its mounting bracket sufficient to remove fuel pump.

(2) Installation

(a) Position fuel pump in its mounting bracket and secure with worm gear clamp.

(b) Connect fuel drain hose to fuel pump drain fitting.

(c) Connect the fuel lines to each side of the pump.

(d) Connect fuel pump electrical leads and grounding strap.

(e) Connect the negative (ground) battery cable and the emergency battery P615 connector.

(f) Turn master switch ON.

(g) Turn fuel selector to the desired tank and operate fuel pump. Check all fuel lines and fittings for leakage.

1) If leaks are evident, tighten fittings, repair components as required.

2) When no leaks are detected, proceed to the next step.

(h) Install cowling.

(3) Testing

This is a bench test to verify pump operation to minimum specifications. Remove / install the pump per “Removal” on page 28208 and “Installation” on page 282011.

(a) Setup

All tests shall be conducted using Aviation Gasoline, or Stoddard Solvent as the fluid. All of the following shall apply unless otherwise specified:

1) Voltage supplied shall be 28+ VDC

2) Fuel Temperature shall be 80+ 20 degrees F

3) Ambient temperature shall be 80+ 20 degrees F

4) All fuel lines shall be ½” O.D. tubing.

(b) Procedure

1) Install the pump in a test setup equivalent to that illustrated in “Figure 5” on page 282012.

2) Apply 28 VDC to motor. The motor should run smoothly with no unusual noise or vibration.

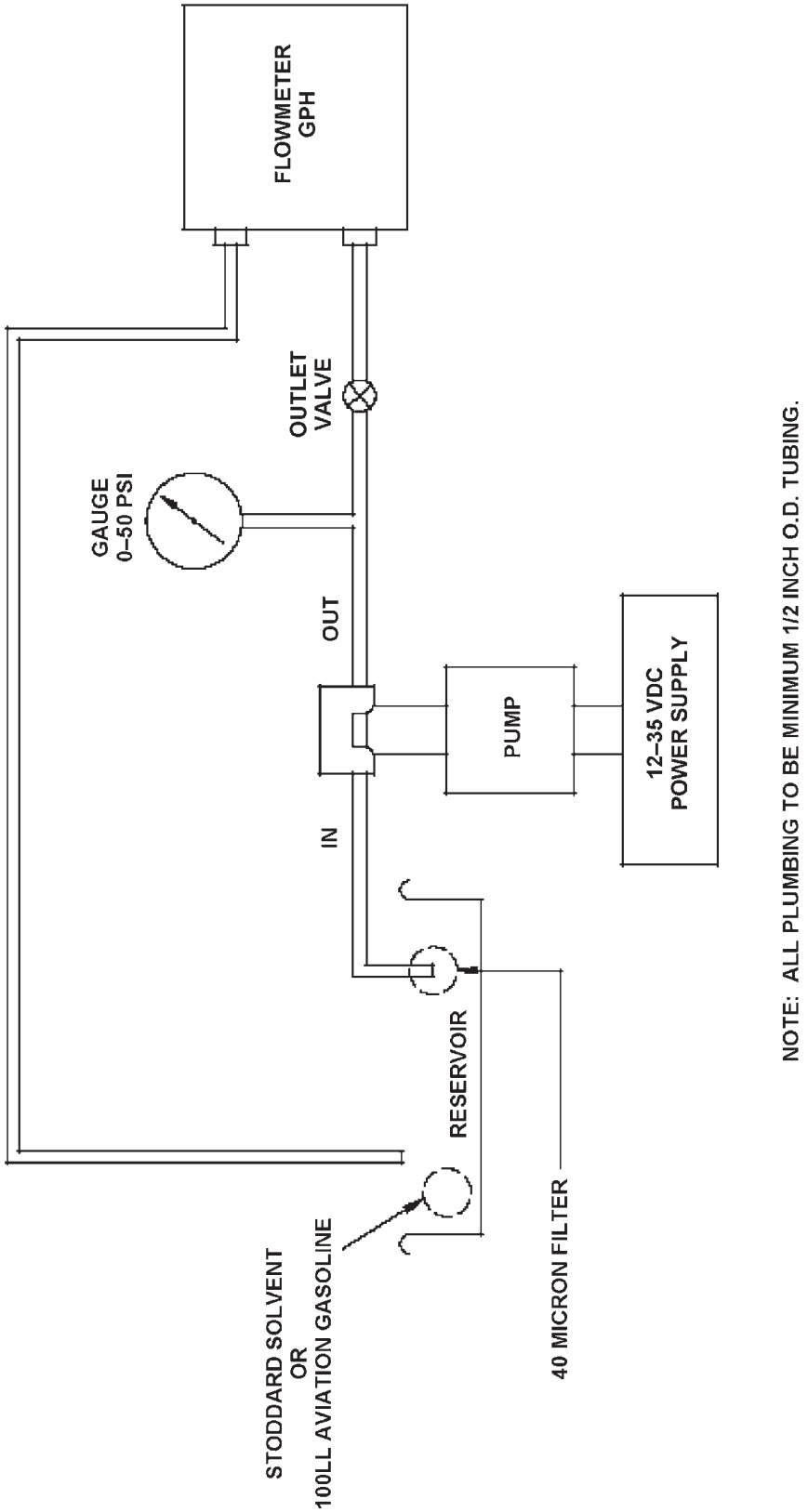
NOTE: Proper polarity of the motor is required. Connect the positive lead of the power supply to the white wire with orange stripe from the motor and the negative lead from the power supply to the white wire with black stripe from the motor.

3) Close the outlet valve and check pressure on the gauge. The gauge should read no greater than 30 PSIG.

4) Open the outlet valve until the pressure gauge reads 23 PSIG.

5) The flow meter must read at least 35 GPH or the pump is defective.

6) Operate pump for two (2) minutes and verify no leakage from the fuel drain fitting while the pump is operating.



Effectivity
Weldon Pump

Electric Fuel Pump Test Setup
Figure 5

5. Lines and Fittings

A. Inspection and Tightening

(1) Remove aft inboard inspection panel from the lower surface of right and left wing.

(2) Remove pilot's seat and left cabin side panel. Fold back carpeting and remove the cover from fuel line(s).

(3) Inspect all union fittings in fuel system for signs of leakage. (Note any leaking fittings for repair or replacement.)

CAUTION: A CROW'S FOOT ADAPTER OTHER THAN A TUBING TYPE WILL DEFORM OR DAMAGE UNION NUT AND WILL CAUSE A LEAK REQUIRING UNION AND TUBING REPLACEMENT.

CAUTION: IF DURING TORQUE CHECK A GALLED NUT AND UNION ARE SUSPECTED, BACK OFF THE NUT AND INSPECT THREADS. IF THE UNION IS SERVICEABLE, LUBRICATE THREADS WITH MIL-PRF-5606 HYDRAULIC FLUID. TORQUE THE NUT TO VALUES PER STEP 4. IF UNION IS UNSERVICEABLE REPLACE PER "REPLACEMENT OF FITTINGS", BELOW.

(4) Use a torque wrench and tubing crow's foot to carefully tighten each union fitting to the torques shown in "Chart 1". Apply thread lubricant to male connector threads only. Do not let lubricant enter the connector seat throat or contact ferrule seat face.

(5) After torquing each fitting measure distance between face of union nut and face of tubing nut. (Refer to "Figure 6" for tolerances.)

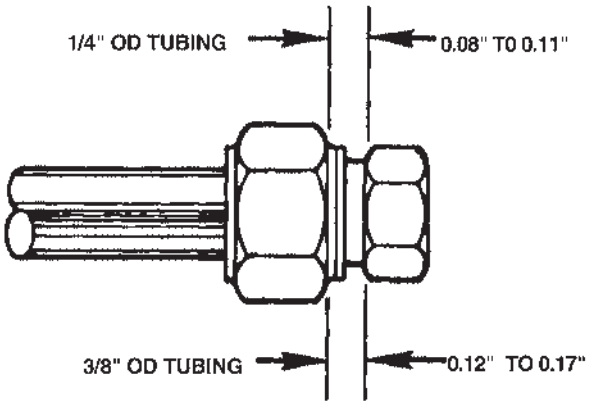
(6) Replace any out of tolerance fitting per Replacement of Fittings, below.

(7) Check all unions for proper tightness. Check all repairs are made. Verify airplane is full of fuel and run engine three to five minutes on each tank. (Perform engine operation in a safe manner and location.)

(8) Shut down engine and wiggle all unions. If any fittings are leaking, repair per Replacement of Fittings, below.

(9) When system is leak free, replace side panel, carpet, access plates, and seat.

(10) Make an appropriate logbook entry.



Tolerance, Union Nut and Tube Nut
Figure 6

CHART 1
FUEL LINE ALUMINUM FITTING TORQUE

Tube Size (OD)	Applied Torque (In·Lbs) (1) Compression Fitting (Swaged Ferrule and Nut)	Flared Fitting Nut (Standard AN nut)
1/4 inch (2)	75–95	50–65
3/8 inch (2)	175–195	110–130

(1) Using a tubing crow's foot.

B. Replacement of Fittings

NOTE: Airplane defueling may be required for union or tubing replacement.

(1) Repair fittings that show evidence of galling, do not meet "Figure 6" on page 282013 dimensional requirements, or leak after tightening.

(2) The recommended repair is to remove leaking union and replace with standard AN fitting as per latest revision of AC43.13-1. This will require cutting off swaged ferrule and adding a short length of tubing.

(3) A replacement tube and union purchased from Piper will have the ferrule pre-swaged onto tube. Install tube as follows:

(a) Lubricate male threads of union with MIL-PRF-5606 hydraulic fluid.

(b) Carefully align tube into union and snug up the nut with a wrench.

(c) Using the wrench, tighten nut one to two flats (1/6 to 1/3 turn).

(4) Repairs made using Parker Hannifin unions and tubes without pre-swaged ferrules must be installed as follows:

(a) Cut off tubing back from the fitting.

(b) De-burr tube end and prepare a short length of tube, to splice into line.

(c) Lubricate the ferrule and threads on the body and nut with MIL-PRF-5606 hydraulic fluid.

(d) Screw nut and ferrule onto union until solidly finger tight.

(e) Insert tubes into unions and check proper straight alignment of tubing and union.

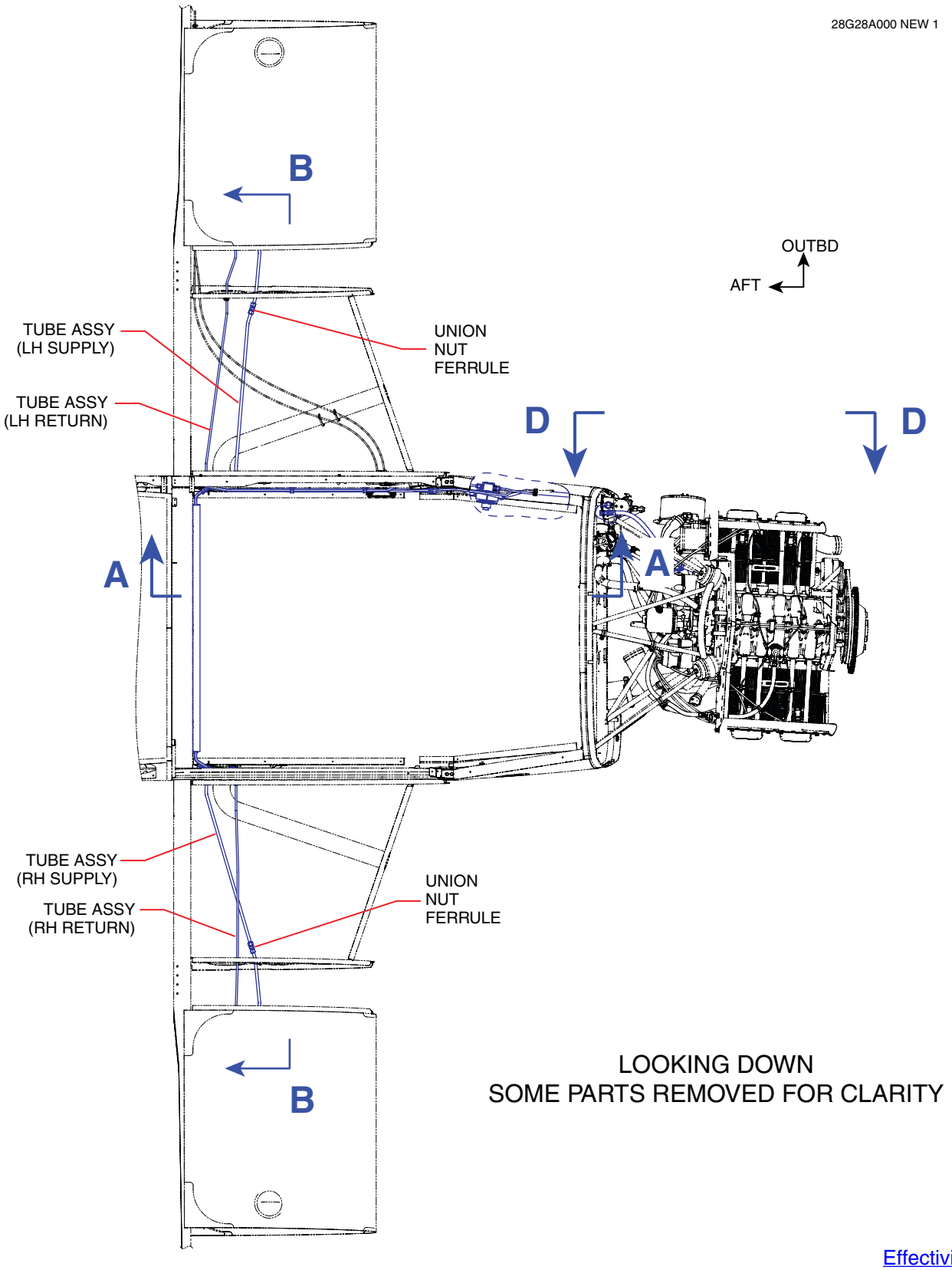
(f) Tighten nut one and one-quarter turns using a tubing wrench.

(5) Perform leak test per steps (7) and (8) in "Inspection and Tightening" on page 282013.

6. Fuel Return System

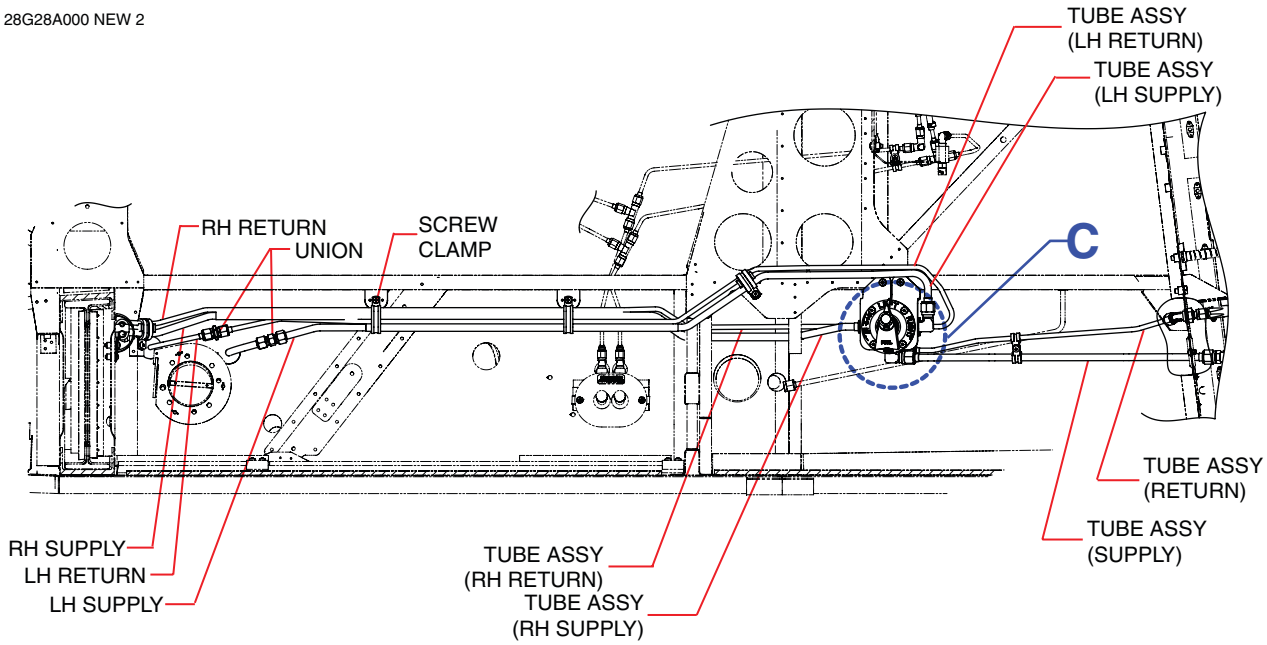
See "Figure 7" on page 282015.

Beginning in 2020, a fuel return system is optional in airplanes with fuel injected engines. Fuel is plumbed from the engine fuel injection servo port using a TSO approved hose and is routed to a fitting on the firewall. Behind the firewall, the fuel system return lines are routed to a duplex fuel selector valve that has provisions for both the main fuel feed system lines as well as the fuel return system lines. The duplex valve allows the system to return fuel back to the tank that is feeding the engine.

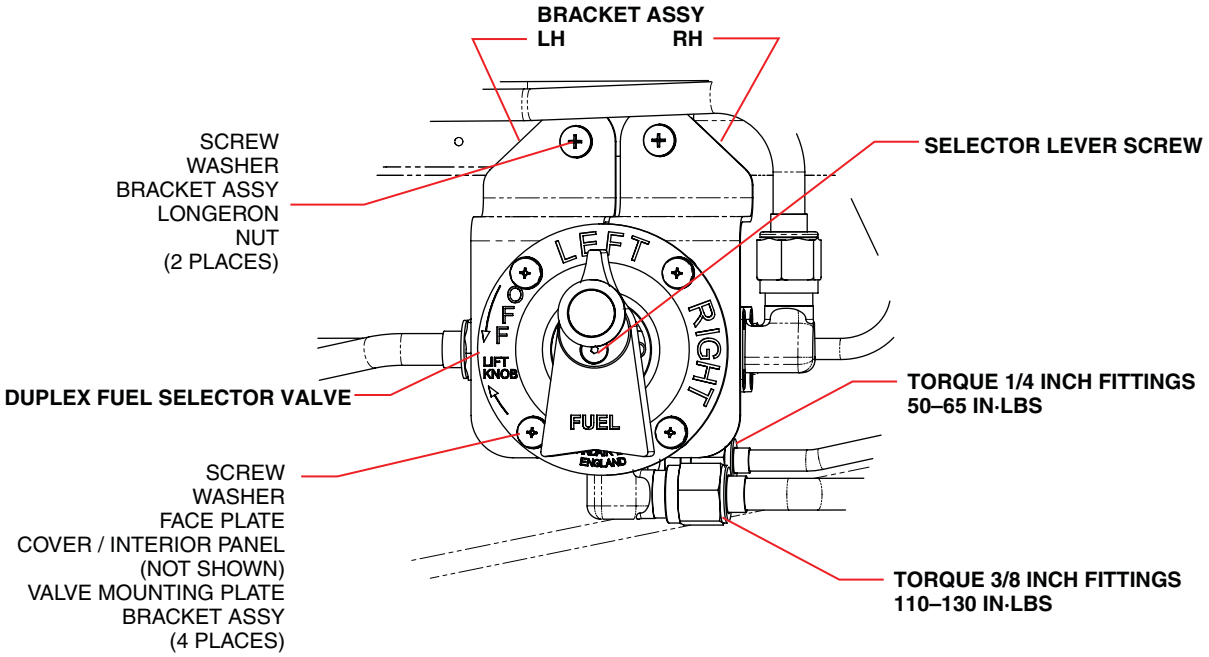


Fuel Return System
Figure 7 (Sheet 1 of 4)

Effectivity
Optional with
IO-360-B4A Engine



SECTION A – A
LOOKING OUTBOARD

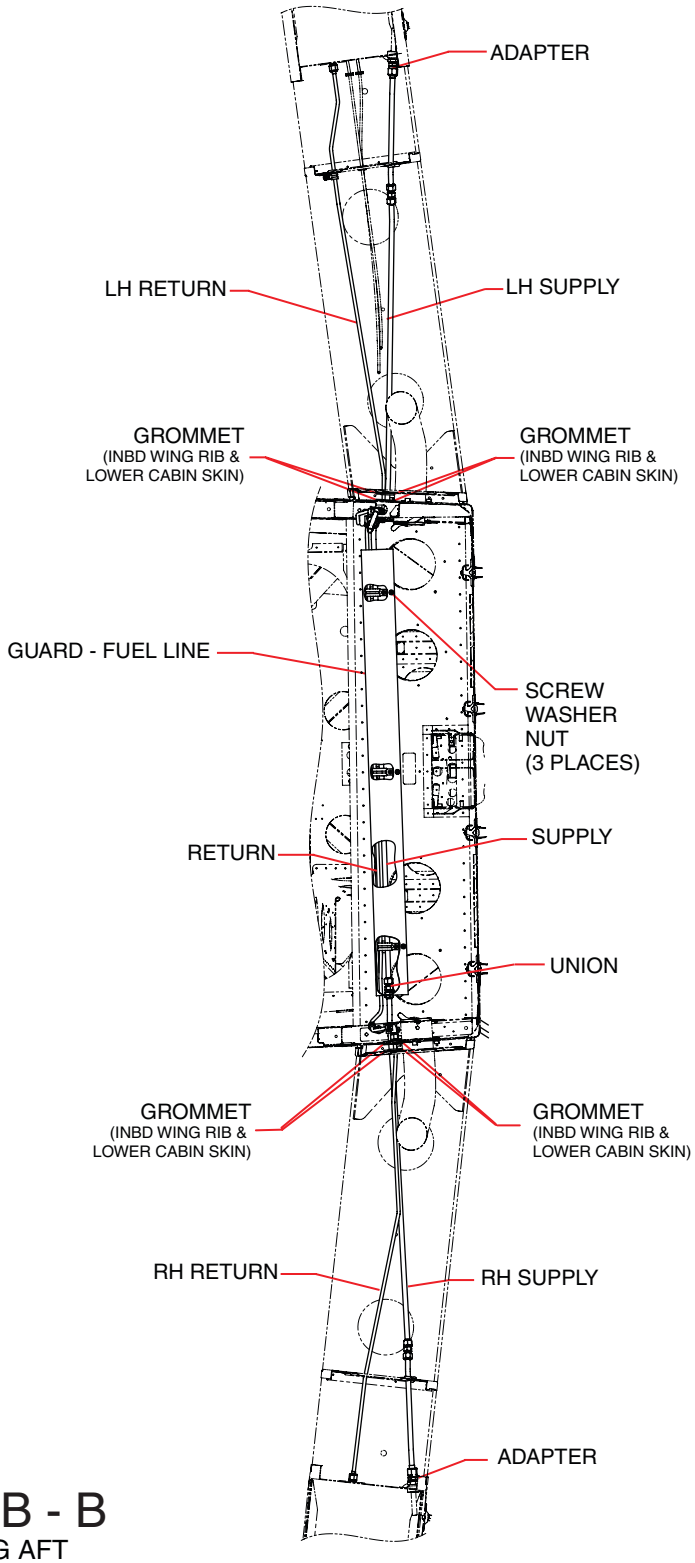


DETAIL C
LOOKING OUTBOARD

Effectivity
Optional with
IO-360-B4A Engine

Fuel Return System
Figure 7 (Sheet 2 of 4)

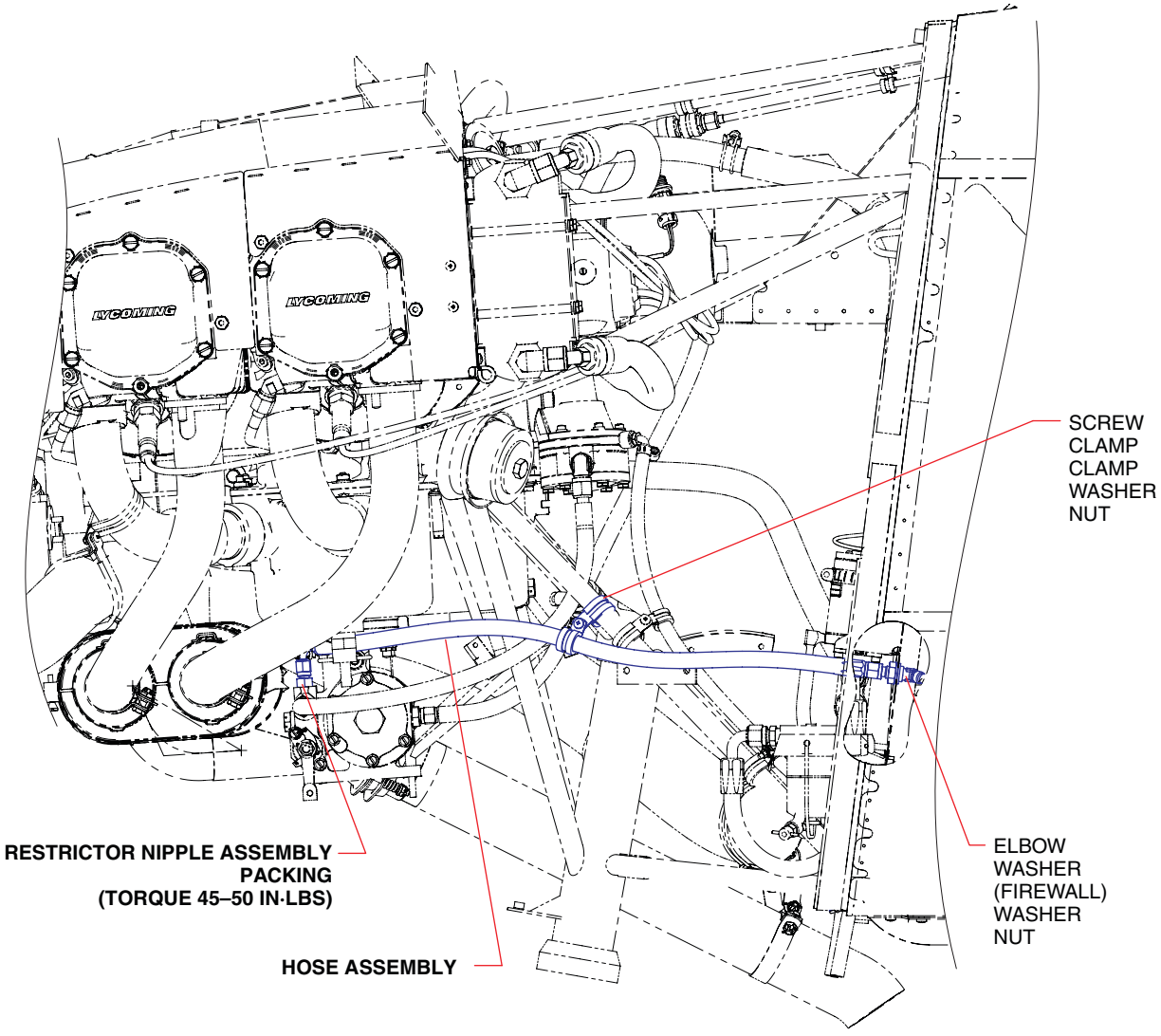
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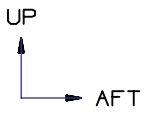
SECTION B - B
LOOKING AFT

Fuel Return System
Figure 7 (Sheet 3 of 4)

Effectivity
Optional with
IO-360-B4A Engine



VIEW D – D
LOOKING INBOARD



Effectivity
Optional with
IO-360-B4A Engine

Fuel Return System
Figure 7 (Sheet 4 of 4)

7. Fuel System Leakage Test

(PIR-PPS60049-4, Rev. C.)

A. Setup

See “Figure 1” on page 28103 and “Figure 2” on page 28105.

(1) Cap all fuel vent outlets on left and right wings.
(2) Verify fuel sump drain valves are in the closed position.
(3) On the test fixture, ensure that the following valves are selected to the OFF position.
  (a) #1 and #2 main air supply shut-off valves.
  (b) Main air supply needle valve (Valve #4).
  (c) Shut-off valve on each fuel test cap.
  (d) Both flow meter shut-off valves.
(4) Verify that the manometer needle valve (Valve #3) is open.
(5) Disconnect the main fuel line(s) forward of the firewall.
(6) Install special test unit fuel caps at the fuel filler port on each wing.

B. Procedure

(1) Leakage Test
  (a) Connect the test fixture left flow meter hose to the main fuel supply line at the firewall (see “Figure 2” on page 28105).

    NOTE: Aircraft equipped with the optional fuel return system have two fuel lines (supply line and return line) at the firewall. Correctly identify the fuel supply line and connect to the test fixture, the return line should be closed with an appropriate cap.

  (b) Select the Fuel Selector Valve (inside the fuselage) to “RIGHT TANK” (full forward position).
  (c) Connect shop air to test fixture.
  (d) Select the Test Fixture “Valve 2” to the “ON” position, and using “Valve 1”, pressurize the line to a minimum of 1.0 PSIG but no greater than 1.5 PSIG.

    NOTE: Monitor pressure on the main pressure gauge. Do not let air into aircraft fuel tanks if pressure reads above 1.5 PSIG.

    CAUTION: CARE MUST BE TAKEN NOT TO EXCEED THE 1.5 PSIG LIMIT TO PREVENT FUEL CELL DAMAGE.

  (e) Open both test cap shut-off valves, slowly open “Valve 4” and monitor pressure on the main pressure gauge. Close “Valve 1” IMMEDIATELY if pressure exceeds 1.5 PSIG, and open flow meter valve to bleed excess pressure.
  (f) Monitor pressure levels on fuel test cap gauges and allow aircraft fuel system to pressurize to a minimum of 1.0 PSIG but no greater than 1.5 PSIG.

    NOTE: The system should take approximately 10 minutes to fill.

  (g) Select the Test Fixture “Valve 1” to the “OFF” position, then “Valve 2” to the “OFF” position. Observe the position of test fluid inside manometer. The “AIRCRAFT” side and “REFERENCE” side of the manometer should be equal in height.

(h) Disconnect shop air from Test Fixture and monitor leakage rate.

CAUTION: THE SYSTEM SHALL NOT LEAK OVER A PERIOD OF 5 MINUTES.

1) If “AIRCRAFT” side of manometer moves upwards during 5 minute hold, then a leak exists at the connections. If leak exists, bubble testing liquid shall be applied to all fuel system connections during the hold period, and each connection shall be inspected for bubble formation.

2) If “REFERENCE” side of manometer moves upwards, then there exists a temperature/ pressure differential. Releasing half of the air in the system and then refilling the system may solve this issue.

(i) If leaks are found, repair per “Repairs” on page 28104.

(2) Flow Test

(a) Close fuel test cap valves. With the Fuel Selector Valve to the “RIGHT TANK”, open the flow meter valve that is connected to the fuel line forward of the firewall. Monitor that the flow meter gauge reads at least “0.5” SCFH.

(b) Select the Fuel Selector Valve to “OFF” (full aft position), and verify the flow meter gauge drops to “0” SCFH.

(c) Select the Fuel Selector Valve to “LEFT TANK” (center position), and verify the flow meter gauge reads at least “0.5” SCFH.

(d) Open the gascolator drain at the firewall, and verify that air exits through the drain. Close the drain.

(e) Select the Fuel Selector Valve to “OFF”, and verify flow meter gauge drops to “0” SCFH.

(f) Select the left flow meter valve to the “OFF” position.

(g) Push open each fuel sump drain valves (located underneath the LH and RH wings), and verify that air flows outside fuel tanks. Close the fuel sump drain valves.

(h) Remove all fuel vent caps (LH and RH wing), and verify that the remaining pressure bleeds off through each vent.

(i) In airplanes equipped with the optional fuel return system, repeat “Flow Check” steps (a) through (d) with fuel return line connected to the left flow meter hose of the test fixture and the supply line closed with an appropriate cap.

(j) Remove all test equipment and reconfigure aircraft fuel system to flight configuration.

INDICATING

Fuel pressure / fuel flow indication is addressed in 73-30-00.

1. Standard Installation

S/N's 2843001 thru 2843597; 2843683 thru 2843700, and 2843702 and up in airplanes equipped with traditional electro-mechanical instruments or Garmin G500.

The Archer III fuel quantity gauge is a dual instrument (two indicators in one case), calibrated in fractional divisions of 0, 5, 10, 15, 20 (gallons), and F (FULL).

A transmitter unit (i.e., sender), comprising a resistance strip and a movable control arm, is installed in each fuel tank. The arm position is controlled by a float in the fuel tank. The float position is transmitted electrically to the indicator display, which then show the amount of fuel in each tank.

A. Troubleshooting.

Troubles unique to the fuel indicating system are listed in "Chart 1" along with their probable causes and likely remedies.

B. Fuel Quantity Transmitter Unit

(1) Removal

(a) Remove fuel tank. (Refer to "Removal" on page 28101.)

(b) Disconnect transmitter wire from the connection post.

(c) Remove five bolts and remove unit.

(2) Installation

(a) Position transmitter and gasket to fuel tank. Apply Lubon 404 (Piper P/N 197-531) to bolt threads. Secure with five bolts, lock washers, and flat washers. (Tighten to torque of 25 inch-pounds.)

(b) Connect transmitter wire to the connection post.

(c) Install fuel tank. (Refer to "Installation" on page 28101.)

(3) Check and Adjustment (Piper P/N 548-671)

(a) Remove transmitter unit to be checked from the fuel tank.

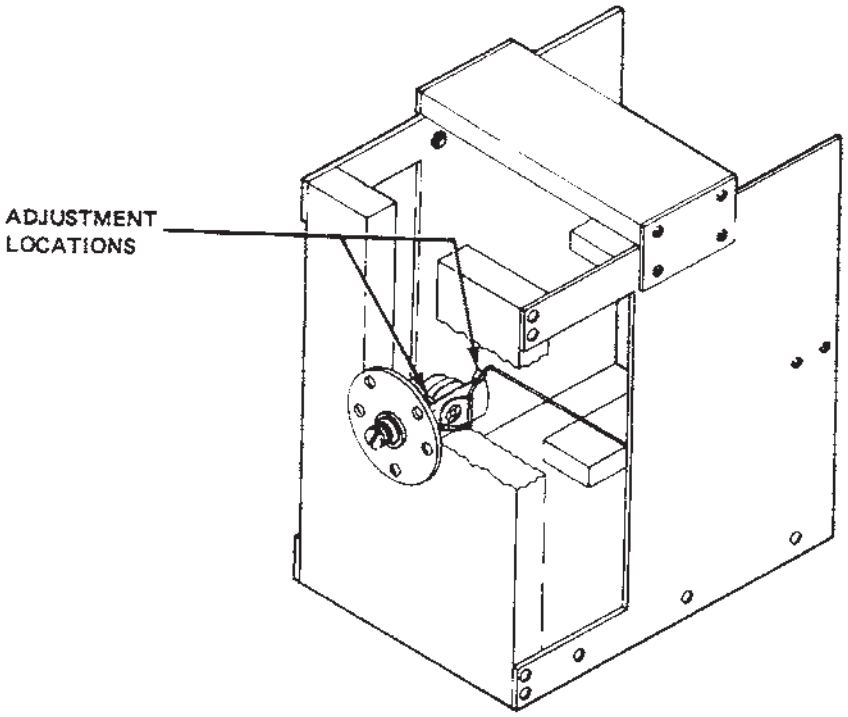
(b) Check and adjust transmitter unit float position as follows:

1) Fasten unit to a fabricated checking jig with washer and nuts as shown in "Figure 1". (Fabricate jig per "Figure 4" dimensions.)

2) Check with the float just touching the top, back of a 0.27 inch spacer. Resistance of the sending unit is 3 ohms ± 0.5 ohm. If the resistance of the sending unit is not in this range, adjust arm of the float assembly by gently bending it as shown in "Figure 1". The entire float must be kept horizontal to the base of the fixture.

CHART 1
TROUBLESHOOTING - FUEL QUANTITY INDICATORS

Trouble	Cause	Remedy
Fuel gauge fails to indicate.	Broken wiring.	Check and repair.
	Defective gauge.	Replace gauge.
	Tripped circuit breaker.	Reset circuit breaker. If breaker does not remain set, check wiring and components for a short.
Fuel gauge reads empty when tanks are full.	Incomplete ground.	Check ground connections at fuel transmitter in wings.
	Float completely filled or saturated.	Replace fuel transmitter.
	Gauge not properly adjusted.	Check gauge settings per appropriate calibration procedure.
	Defective gauge.	Replace gauge.
Fuel gauge reads full with tanks empty.	Incomplete ground.	Check ground at instrument.
	Float arm stuck.	Replace fuel transmitter.
	Gauge not properly adjusted.	Check gauge settings per appropriate calibration procedure.
	Defective gauge.	Replace gauge.
Fuel gauge reads incorrectly.	Defective gauge.	Replace gauge.
	Intermittent ground.	Check ground at transmitter and instrument.
	Float arm sticky.	Replace transmitter.
	Faulty transmitter.	Replace transmitter.
	Gauge not properly adjusted.	Check gauge settings per appropriate calibration procedure.



Fuel Quantity Transmitter Checking Jig
Figure 1

(c) Check transmitter unit for correct resistance and dead spots as follows:

1) Remove sending unit from the checking fixture and connect it to an ohmmeter. Allow float arm to rest on the bottom mechanical stop. Ohmmeter must read 0 ± 0.5 ohms resistance.

2) Move float arm to its upper mechanical stop. A resistance of 45 ± 2 ohms must be indicated.

3) Check for dead spots by slowly moving float arm from bottom stop, to upper stop, and back. Ohmmeter indicator must move steadily up and down scale without fluctuation as float arm is moved.

4) If there is incorrect resistance or dead spots, replace sending unit.

C. Fuel Quantity Gauge Accuracy Test

(PIR-PPS60032-12, Rev. D.)

The Archer III fuel gauge is a dual instrument with individual indicators for the left and right fuel tanks in a single case.

NOTE: This gauge was factory installed in S/N's 2843001 thru 2843597; and 2843683 thru 2843699. If a replacement gauge has been installed per Piper Kit No. 88536-001, see Electronic Fuel Quantity Gauge, below.

(1) Bench test.

The bench test requires the use of a resistance decade box containing a six position switch. Refer to "Figure 2".

(a) Set up:

1) Connect a 5 ohm resistor between the ground terminal (GND) and left or right sender terminal, appropriate to side not being tested.

2) Connect resistance decade to the left or right sender terminal, appropriate to side being tested.

3) Apply 28 ± 1 Vdc, through the resistance decade, to the ground (GND) and positive (+) terminals.

(b) Low end adjustment:

1) Select position F on resistor decade switch.

2) Verify the appropriate instrument needle points to zero (0).

3) If needle does not point to zero (0), adjust NULL potentiometer, located on bottom of instrument below side being tested, to center needle on zero (0).

(c) High end adjustment:

1) Select position A on resistor decade switch.

2) Verify the appropriate instrument needle points to F.

3) If needle does not point to F, adjust GAIN potentiometer, located on bottom of instrument below side being tested, to center needle on F.

(d) Full range check.

After low and high end adjustments have been completed check that, for each resistive value, the gauge indicates as specified in "Chart 2".

(2) Post installation functional check.

(a) Set up:

1) Level airplane, ± 1°, longitudinally and laterally. See 8-20-00.

2) Place battery switch in OFF position.

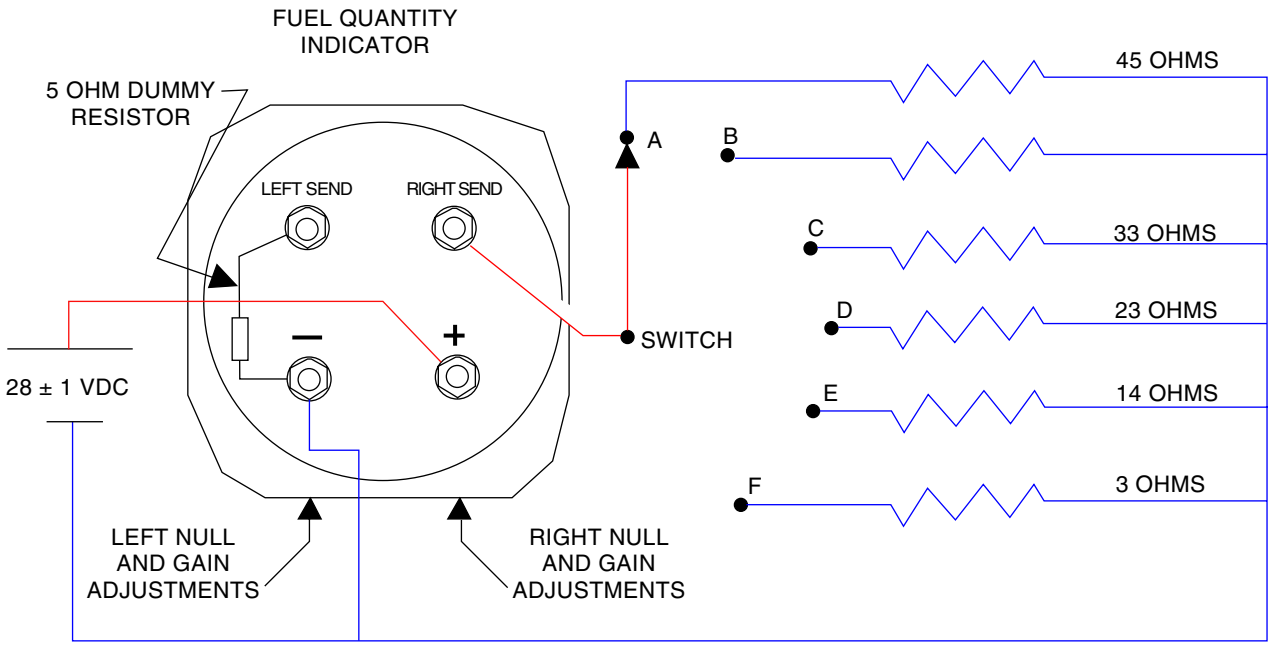
3) Completely drain fuel tank that relates to the gauge to be checked. (Refer to draining fuel system, 12-10-00.)

4) Connect external power supply to airplane's electrical system using APU connector.

5) Adjust power supply unit to provide 28 ± 1 Vdc. Allow gauges to warm-up for a minimum of three minutes.

(b) Low End Gauge Check:

Turn fuel selector to RIGHT or LEFT. With the tank(s) completely dry, place one (1) U. S. gallon of fuel in tank(s) of gauge(s). Check that the needle of each gauge points to 0. If not, adjust the respective "NULL" trim potentiometer until the needle centers on 0, + 0, -1. Record reading.



Resistance Decade Test Box Set-Up
Figure 2

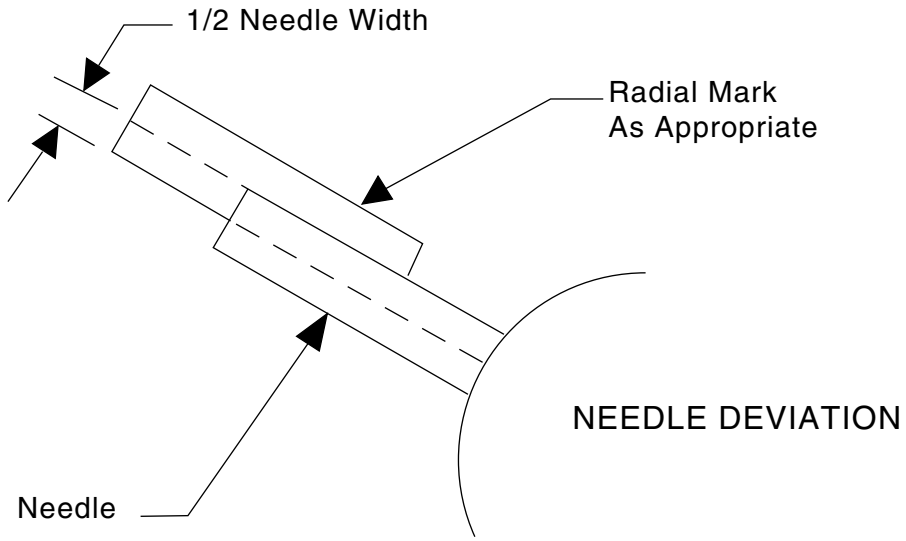
(c) Full Range Gauge Check:

Add fuel in increments specified in “Chart 3”. After the measured amount has been put in the tank, vibrate the tank by bumping its lower surface, and vibrate the gauge by tapping the gauge glass with fingers. Observe and record quantity indications.

NOTE: A Full Range Gauge Check can also be accomplished by starting with the tanks full and defueling each tank 5 gallons at a time.

(d) Full End Gauge Check:

With the tanks full, check that the needle of each gauge points to F. If not, adjust the respective “GAIN” trim potentiometer until the needle centers on F.



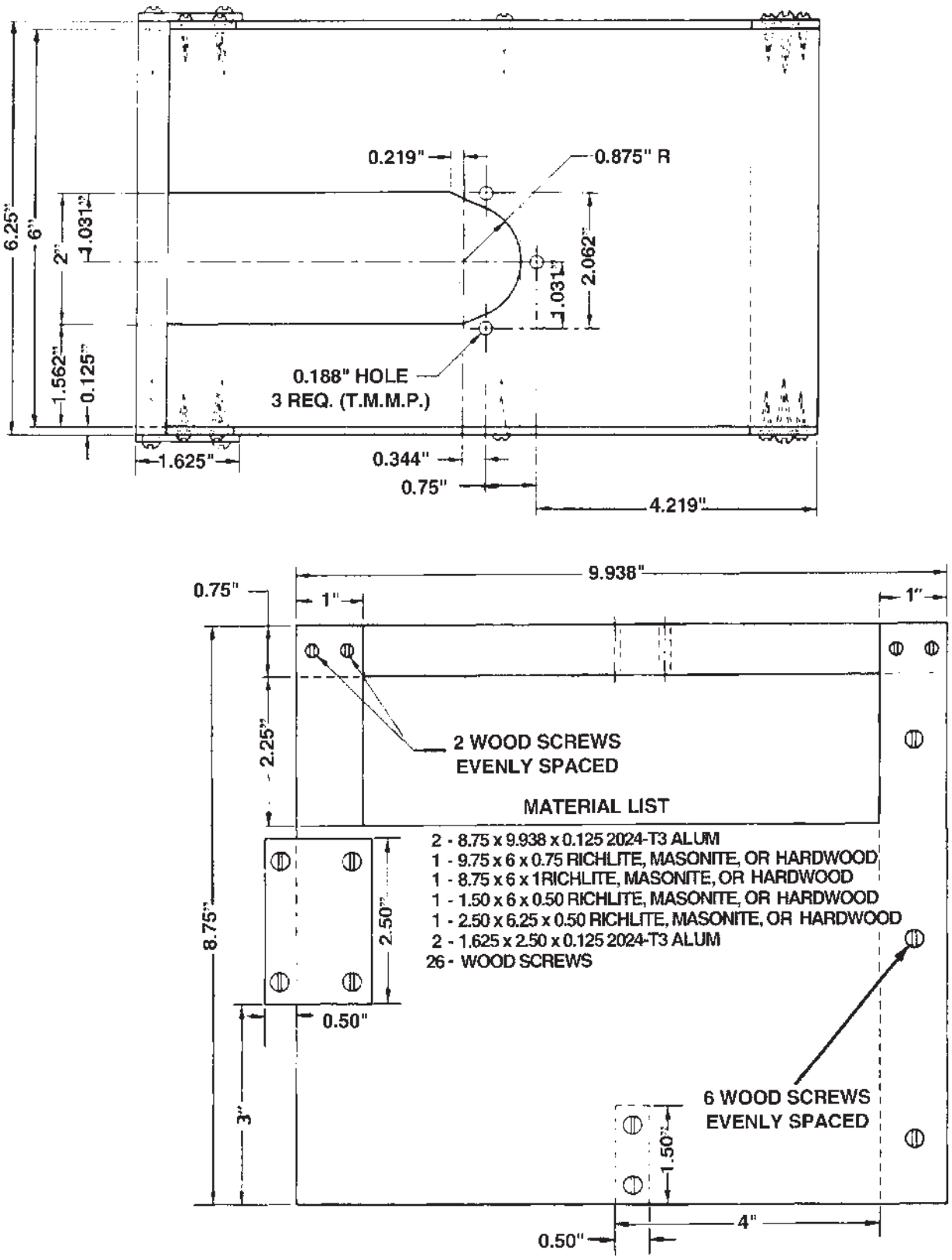
Fuel Quantity Needle Deviation
Figure 3

CHART 2
FUEL QUANTITY GAUGE BENCH TEST TOLERANCES

(1) Switch Position	Resistance in Ohms	Gauge Indication	(2) Tolerance (Needle Width)
F	3	0	+0, -1/2
E	14	5	± 1/2
D	23	10	± 1/2
C	33	15	± 1/2
A	45	F	± 1/2
	(1) See Figure 2	(2) See Figure 3	

CHART 3
FUEL QUANTITY GAUGE FULL RANGE CHECK TOLERANCE VALUES

Fuel Added (U.S. Gal.)	Actual Fuel in Tank (U.S. Gal.)	Gauge Indication	Tolerance (Needle Width)	Resistance Ref. (Ohms - both senders)
1	1	0	+0, -1	3
5	6	5	±1	14
5	11	10	±1	23
5	16	15	±1 1/2	33
9	25	F	±1 1/2	45



Fabricated Fuel Quantity Transmitter Checking Jig
Figure 4

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D. Electronic Fuel Quantity Gauge

(PIR-PPS60032-18, Rev. A.)

Factory installed in S/N's 2843700 and 2843702 and up in airplanes equipped with Garmin G500. Available as service replacement in S/N's 2843001 thru 2843597 and 2843683 thru 2843699, per Piper Kit No. 88536-001.

(1) Fuel Quantity Gauge Accuracy Check

(a) Ensure airplane is level ±1° laterally and longitudinally in accordance with 8-20-00. Drain fuel (including unusable) from left and right wing tanks per 12-10-00.

(b) Add one USG fuel to left and right wing tanks. Wait one minute to let fuel stabilize. Vibrate the tank by bumping its lower surface. Ensure gauge reads zero.

(c) Fill tanks to the increments indicated in "Chart 4".

(d) Verify fuel gauge meets the tolerance requirements in "Chart 4".

1) If gauge meets the specified requirements, proceed to step (e).

2) If gauge fails to meet the accuracy requirements of "Chart 4", calibrate per "(2) Fuel Quantity Gauge Calibration" and then repeat steps (a) thru (d).

(e) Document successful completion of this procedure in the airplane logbook.

CHART 4
FUEL INCREMENTS AND TOLERANCE

Total Fuel Quantity (USG)	Gauge Indication (USG)	Tolerance (USG)
1	0	0
7	6	±1
13	12	±1
19	18	±1
25	FULL	0

(2) Fuel Quantity Gauge Calibration

NOTE: This step only required if gauge fails the Fuel Quantity Gauge Accuracy Check.

NOTE: If the gauge is replaced, calibrate both sides. If senders are replaced on one side only, calibrate only that side.

(a) Setup

1) Drain fuel (including unusable) from left and/or right wing tanks.

2) With the master power (switch) on, disengage the fuel gauge circuit breaker.

3) Attach the Fuel Gauge Calibration Tool to the connector on the back of the fuel gauge and then reconnect airplane harness.

NOTE: If required, fabricate the Fuel Gauge Calibration Tool per "Fuel Gauge Calibration Tool" on page 284010.

4) Press and hold both TOP and BOT buttons on the calibration tool. Then apply power by resetting the fuel gauge circuit breaker.

5) When instructed on the fuel gauge display, release both buttons.

6) For general operation of calibration tool, BOT button is used to navigate various menu options, while TOP button is used to select menu options. Press TOP button when you see the screen shown in "Figure 5" to continue.

(b) Procedure

1) Select the tank to calibrate (“Figure 6”).

2) Navigate to the CONTINUE option, then proceed with the calibration (“Figure 7”).

NOTE: If desired, selecting ABORT will exit the calibration process and the gauge will return to previous calibration.

3) Navigate to START FROM ZERO option, then proceed to “4) Zero Fuel Calibration”.

4) Zero Fuel Calibration

a) Ensure that only the unusable fuel (1 USG) is in the tank to be calibrated (“Figure 8”).

b) Vibrate tank by bumping its lower surface and then wait approximately 20 seconds for the fuel to stabilize in the tank.

c) Once completed, continue to the next screen.

5) Full Range Calibration

a) When instructed as shown in “Figure 9”, add two USG of fuel to tank.

b) Vibrate tank by bumping its lower surface and then wait approximately 20 seconds for the fuel to stabilize in the tank.

c) Once completed, continue to the next screen.

d) Verify a green check “√” mark is in lower left of screen as shown in “Figure 10”.

e) Continue to follow the on-screen instructions adding fuel in 2 USG increments until reaching the 24 USG point.

f) At the 24 USG point, gauge will ask to exit the calibration process.

g) Select YES as shown in “Figure 11”.

h) The calibration table is being saved (“Figure 12”). Do not perform any further functions until prompted on the instrument screen.

i) Select either the next tank to calibrate or EXIT to terminate this function (“Figure 13”).

j) Return to Step 1) to calibrate the second tank.

k) Select EXIT to reboot the instrument and return to normal operating mode.

l) Upon completing calibration, perform accuracy check per “(1) Fuel Quantity Gauge Accuracy Check”.

(3) Fuel Gauge Calibration Tool

Using the parts in “Chart 5”, fabricate the Calibration Tool per “Figure 14”.



Figure 5



Figure 6



Figure 7



Figure 8



Figure 9



Figure 10



Figure 11



Figure 12

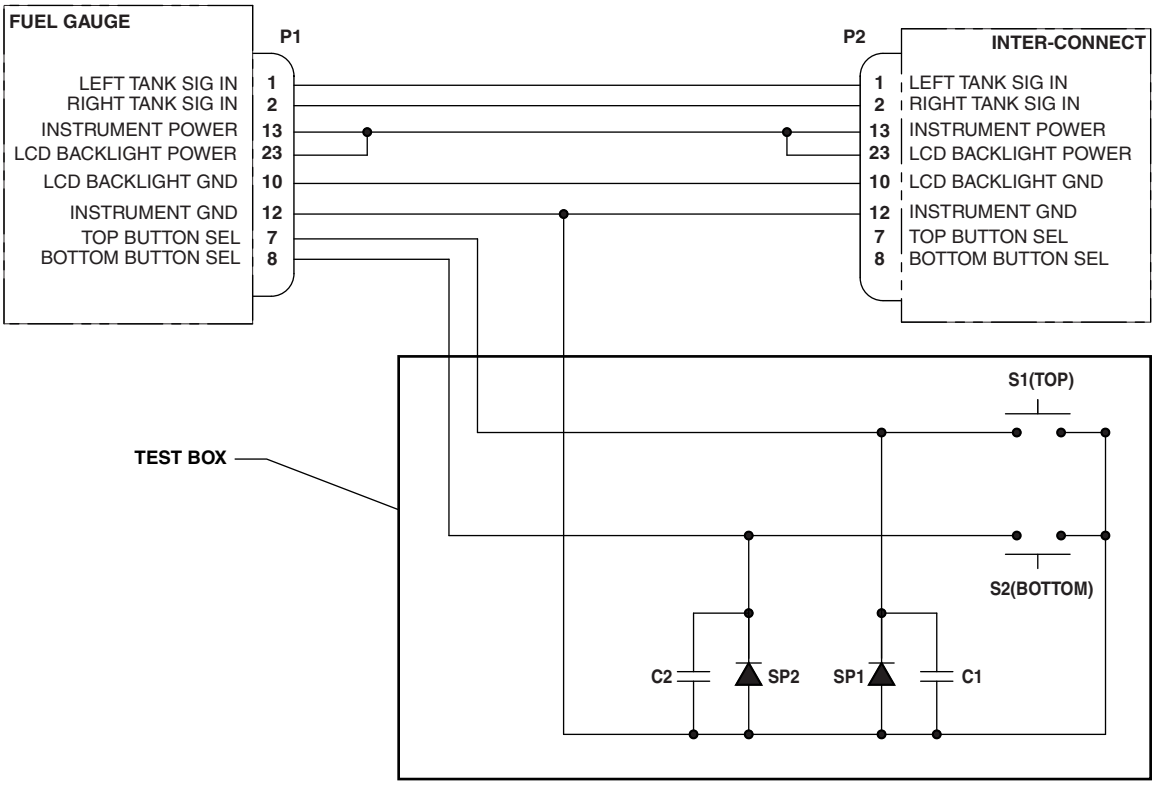


Figure 13

CHART 5
FUEL GAUGE CALIBRATION TOOL / HARNESS ASSEMBLY PARTS LIST

Designator	Description	P/N (Procure Locally *)	Qty
P1	Connector	DBMA-25S	1
P2	Connector	DBC25P	1
S1, S2	Switch	KSC421J70SHLFS	2
C1, C2	Capacitor	C0603C104K5RACTU	2
SP1, SP2	TVS Diode	SMAJ5.0A	2
Test Box	Enclosure (3”x2”x1”)	270-1801	1
Interconnect Wire	White	22 AWG	~ 18 Feet
ILB	Interleaved Bus PCB (1x.5)	8002	2
	Shrink tube	As required	

* Item procurement resources include Digi-Key, Mouser Electronics, Allied Electronics, ITT/Cannon and Radioshack.



Fuel Gauge Calibration Tool Wiring Diagram
Figure 14

2. Avidyne Entegra Installation

(PIR-PPS60212, Rev. A; PPS-60212-1, Rev. A.)

In S/N's 2843598 thru 2843682, with Avidyne Entegra installed, fuel quantity information is collected by the DAU and displayed on the MFD. Whenever the MFD, DAU or any fuel sender is replaced, calibrate the Fuel Quantity Indicator as follows:

NOTE: Fuel senders are the same as described in Fuel Quantity Transmitter Unit under Standard Installation, above.

NOTE: See 34-22-00 for additional information on the Avidyne Entegra system.

NOTE: The following procedure is written specifically for MFD Software No. 530-00180-002 (Latest Revision) and later. For airplanes using earlier software, differences will appear as "(** difference)".

A. Level the airplane per 8-20-00.

B. Drain fuel entirely per 12-10-00.

C. From the MFD maintenance page, depress line select key (L4) to access the Engine setup Page

D. Depress the Fuel Cal button (L1) to enter the fuel calibration page.

E. Depress "Begin Cal" button (R3) and follow the on-screen calibration procedure for left & right tank zero fuel and full fuel calibration points.

NOTE: At each calibration point, wait approximately three (3) minutes for the fuel level to stabilize before proceeding to the next step.

F. Use the left knob to select the current calibration point. The selected calibration point is highlighted and the value displayed is the current reported fuel quantity from the DAU. A message at the bottom of the screen prompts the operator to add the appropriate amount of fuel and then to select "Accept Value" button (R3) once the value reported from the DAU has stabilized. (wait approximately 3 minutes).

NOTE: If the DAU reported value is not within two (2) gallons of the test point value, a message "DAU reported Fuel Quantity Out of Tolerance" will be presented and the value will not be accepted.

G. Once all points have been calibrated, press "Calibration Complete" to cause the calibration factors to be computed and applied to the DAU reported fuel quantity.

H. Other options from the Calibration Underway state are to "Restore Last Cal" and "Clear Cal".

(1) Pressing "Restore Last Cal" causes the calibration values from the last completed calibration to be restored and the state to change to Calibrated.

(2) Pressing "Clear Cal" causes all calibration values to be cleared and the state to change to "Not Calibrated". An "Are You Sure?" prompt will give the operator a chance to reconsider the decision to either "Restore Last Cal" or "Clear Cal".

I. When the calibration procedure has been completed, press the Save button. If you decide not to save the changes, pressing the Cancel button from the Underway state causes the current calibration session to be aborted with any unsaved interim calibration values being discarded. Changes will not take effect until the MFD has restarted.

NOTE: The "empty" and "full" fuel readings on this set-up page will NOT display zero(0) or 24 gallons usable fuel for respective empty and full calibration points. However, the appropriate zero and full (24 U.S. gallons) will be displayed on the MFD in normal operation mode (not maintenance set-up).

J. Verify DAU Status Box reads: DAU(s) configured for PA-28-181 (Archer III). (** N/A)

K. Verify Calibration Status box reads: "Fuel Tank Calibration Status: Calibrated." (** N/A)

3. Garmin G1000 Installation

(PIR-PPS55034, Rev. B.)

Available in S/N's 2843701, 2843703–2843822, 2843824–2843851 and 2843853. The Garmin G1000 Integrated Avionics System is optional in these airplanes. Raw fuel quantity data from the fuel senders is sent to the #1 GIA 63W. The data then is fed through the PFD and the Fuel Quantity Indication is displayed on the Multi-Function Display (MFD).

CAUTION: ENTER "MAINTENANCE MODE" WHEN ACTIVATING THE GARMIN G1000 SYSTEM ON THE GROUND TO CONDUCT TESTS, INSPECTIONS, ETC. SEE MAINTENANCE MODE IN 34-25-01.

NOTE: Fuel senders are the same as described in Fuel Quantity Transmitter Unit under Standard Installation, above.

A. Fuel Tank Calibration

(1) Level the aircraft to within ± one (1) degree laterally and longitudinally per 8-20-00.

(2) Ensure the wing tanks are completely empty (12-10-00).

NOTE: The large FMS Knob changes Groups and the small FMS knob changes Pages within a Group.

(3) Apply ground power to the aircraft per 24-40-00.

(4) Start the PFD and MFD in the configuration mode by holding the ENT key on the PFD, and the far right softkey on the MFD while applying power.

NOTE: The G1000 System must be powered for three minutes before proceeding.

(5) Add unusable fuel to the wings (one (1) US gal. per side).

(6) After 5 minutes, select the 'CAL' group, then select 'FUEL CALIBRATION' page on the 'PFD'.

(7) Unlock the page by pressing softkeys 12 (far right softkey), 11, 10, 9 in sequence.

(8) Verify the 'CALIBRATION VALUE' indication for the Left Tank is stable before proceeding.

(9) Press the 'RAW DATA' softkey to de-select the scale function.

(10) Press the 'EMPTY' softkey. Press the 'ENT' key to acknowledge the prompt after the 'EMPTY' softkey is pressed.

(11) Verify the "ACTUAL QUANTITY" shown in the "CALIBRATION TABLE" in the lower right corner of the screen shows 0 gallons.

(12) Press the 'TNK SEL' softkey.

(13) Rotate the small FMS knob to generate a pick list and select the Right Tank.

(14) Press the 'ENT' key.

(15) Repeat steps (7) thru (10) for the right tank.

NOTE: For actual fuel quantity entries, the large FMS knob moves the cursor. The small FMS knob changes the digits.

(16) Add six (6) gal fuel to each wing.

(17) After at least 5 minutes, Verify the 'CALIBRATION VALUE' indication for the Right Tank is stable before proceeding.

(18) Press the 'FUEL QTY' softkey.

(19) Enter 6 gal for 'ACTUAL FUEL QUANTITY'. Press the 'ENT' key to accept.

(20) Press 'ENT' Key again to calibrate.

(21) Verify the "ACTUAL QUANTITY" shown in the "CALIBRATION TABLE" in the lower right corner of the screen shows 6 gallons.

(22) Press the 'TNK SEL' softkey.

(23) Rotate the small FMS knob to generate a pick list and select the Left Tank.

(24) Press the 'ENT' key.

(25) Repeat steps (16) thru (20) for the Left Tank.

(26) Add six (6) gal fuel to each wing.

(27) After at least 5 minutes, Verify the 'CALIBRATION VALUE' indication for the Left Tank is stable before proceeding.

(28) Press the 'FUEL QTY' softkey.

(29) Enter 12 gal for 'ACTUAL FUEL QUANTITY'. Press the 'ENT' key to accept.

(30) Press 'ENT' Key again to calibrate.

(31) Verify the "ACTUAL QUANTITY" shown in the "CALIBRATION TABLE" in the lower right corner of the screen shows 12 gallons.

(32) Press the 'TNK SEL' softkey.

(33) Rotate the small FMS knob to generate a pick list and select the Right Tank.

(34) Press the 'ENT' key.

(35) Repeat steps (26) thru (30) for the Right Tank.

(36) Add six (6) gal fuel to each wing.

(37) After at least 5 minutes, Verify the 'CALIBRATION VALUE' indication for the Right Tank is stable before proceeding.

(38) Press the 'FUEL QTY' softkey.

(39) Enter 18 gal for 'ACTUAL FUEL QUANTITY'. Press the 'ENT' key to accept.

(40) Press 'ENT' Key again to calibrate.

(41) Verify the "ACTUAL QUANTITY" shown in the "CALIBRATION TABLE" in the lower right corner of the screen shows 18 gallons.

(42) Press the 'TNK SEL' softkey.

(43) Rotate the small FMS knob to generate a pick list and select the Left Tank.

(44) Press the 'ENT' key.

(45) Repeat steps (36) thru (40) for the Left Tank.

(46) Add six (6) gal fuel to each wing.

(47) After at least 5 minutes, Verify the 'CALIBRATION VALUE' indication for the Left Tank is stable before proceeding.

(48) Press the 'FULL' softkey. Press the 'ENT' key to acknowledge the prompt after the 'FULL' softkey is pressed

(49) Verify the "ACTUAL QUANTITY" shown in the "CALIBRATION TABLE" in the lower right corner of the screen shows 24 gallons.

(50) Press the 'TNK SEL' softkey.

(51) Rotate the small FMS knob to generate a pick list and select the Right Tank.

(52) Press the 'ENT' key.

(53) Repeat steps (46) thru (48) for the Right Tank.

(54) Remove ground power from the aircraft.

B. Fuel Quantity Accuracy Check

NOTE: If an aircraft fails this check, troubleshoot to determine cause and corrective action and repeat the fuel calibration and accuracy check, if required.

(1) Level aircraft to within ± one (1) degree laterally and longitudinally per 8-20-00.

(2) Ensure the wing tanks are completely empty (12-10-00).98jh

(3) Apply ground power to aircraft per 24-40-00.

(4) At MFD splash screen press softkeys in the following sequence: 1, 2, 3, 1. Then press the far right softkey.

(5) Verify Maintenance Mode advisory message on PFD.

(6) Add unusable fuel to the wings (one (1) US gal per side) and allow to stabilize (5 min.).

(7) Verify the Garmin fuel indicator shows left and right fuel quantities at 0 gal.

(8) Verify fuel quantity gauge digits are RED and “L FUEL QTY” and “R FUEL QTY” warning CAS messages are displayed.

(9) Add three (3) gal of fuel to each wing and verify the corresponding reading on the Garmin system is within ± 1 gal per side (± 2 gal total).

(10) Add one (1) gal of fuel to each wing. Verify fuel quantity gauge digits turn from RED to AMBER and “L FUEL QTY” and “R FUEL QTY” warning CAS messages are extinguished when fuel quantity is > 3 gal per side. “L FUEL QTY” and “R FUEL QTY” caution CAS messages should appear after 10 seconds.

(11) Add two (2) gal of fuel to each wing. Verify fuel quantity gauge digits turn GREEN and “L FUEL QTY” and “R FUEL QTY” caution CAS messages are extinguished uponreaching > 5 gal of fuel per side. Verify reading is within ± 4 gal per side.

(12) Continue adding fuel in 3 gal increments (total of 6 increments) and verify the corresponding reading on the Garmin system is within ± 4 gal per side (± 8 gal total).

(13) If quantity readings at all intervals are within limits, then fuel accuracy check is complete.

(14) Remove ground power from the aircraft.

C. Calibration Table Delete/Edit

This procedure can be used to edit or delete an existing calibration point within the calibration table.

(1) Press the CAL TBL softkey.

(2) Scroll down to the entry to edited or deleted.

(3) To edit the entry press the EDIT softkey. Fill the selected tank to the level of the selected entry. Allow the CALIBRATION VALUE to stabilize.

(4) Press enter and confirm you want to overwrite the existing calibration point.

(5) To delete the entry press the DELETE softkey.

(6) Press and confirm you want to delete the calibration point.

4. Garmin G1000 NXi Installation

(PIR-107992 D, 108059 D.)

Available in S/N's 2843823, 2843852, 2843854–2843999; and 2881001 and up. The Garmin G1000 NXi Integrated Avionics System is optional in these airplanes. Raw fuel quantity data from the fuel senders is sent to the GEA which in turn outputs fuel quantity and diagnostic information to the GIA. The data then is fed through the PFD and the Fuel Quantity Indication is displayed on the Multi-Function Display (MFD).

CAUTION: ENTER "MAINTENANCE MODE" WHEN ACTIVATING THE GARMIN G1000 NXI SYSTEM ON THE GROUND TO CONDUCT TESTS, INSPECTIONS, ETC. SEE MAINTENANCE MODE IN 34-25-02 OR 34-25-03, AS APPROPRIATE.

NOTE: Fuel senders are the same as described in Fuel Quantity Transmitter Unit under Standard Installation, above.

A. Fuel Tank Calibration

(1) Level the aircraft to within ± 0.25 degree laterally and longitudinally per 8-20-00.

(2) Ensure the wing tanks are completely empty (12-10-00).

NOTE: The large FMS Knob changes Groups and the small FMS knob changes Pages within a Group.

(3) Apply ground power to the aircraft per 24-40-00.

(4) Start the PFD and MFD in configuration mode by applying power to each display while holding the ENT key.

NOTE: The G1000 NXi System must be powered for three minutes before proceeding.

(5) Add unusable fuel to the wings (one (1) US gal. per side).

(6) After 5 minutes, select the 'CAL' group, then select 'FUEL CALIBRATION' page on the 'PFD'.

(7) Unlock the page by pressing softkeys 12 (far right softkey), 11, 10, 9 in sequence.

(8) Verify the 'CALIBRATION VALUE' indication for the Left Tank is stable before proceeding.

(9) Press the 'RAW DATA' softkey to de-select the scale function.

(10) Press the 'EMPTY' softkey. Press the 'ENT' key to acknowledge the prompt after the 'EMPTY' softkey is pressed.

(11) Verify the "ACTUAL QUANTITY" shown in the "CALIBRATION TABLE" in the lower right corner of the screen shows 0 gallons.

(12) Press the 'TNK SEL' softkey.

(13) Rotate the small FMS knob to generate a pick list and select the Right Tank.

(14) Press the 'ENT' key.

(15) Repeat steps (7) thru (10) for the right tank.

NOTE: For actual fuel quantity entries, the large FMS knob moves the cursor. The small FMS knob changes the digits.

(16) Add six (6) gal fuel to each wing.

(17) After at least 5 minutes, Verify the 'CALIBRATION VALUE' indication for the Right Tank is stable before proceeding.

(18) Press the 'FUEL QTY' softkey.

(19) Enter 6 gal for 'ACTUAL FUEL QUANTITY'. Press the 'ENT' key to accept.

(20) Press 'ENT' Key again to calibrate.

(21) Verify the "ACTUAL QUANTITY" shown in the "CALIBRATION TABLE" in the lower right corner of the screen shows 6 gallons.

(22) Press the 'TNK SEL' softkey.

(23) Rotate the small FMS knob to generate a pick list and select the Left Tank.

(24) Press the 'ENT' key.

(25) Repeat steps (16) thru (20) for the Left Tank.

(26) Add six (6) gal fuel to each wing.

(27) After at least 5 minutes, Verify the 'CALIBRATION VALUE' indication for the Left Tank is stable before proceeding.

(28) Press the 'FUEL QTY' softkey.

(29) Enter 12 gal for 'ACTUAL FUEL QUANTITY'. Press the 'ENT' key to accept.

(30) Press 'ENT' Key again to calibrate.

(31) Verify the "ACTUAL QUANTITY" shown in the "CALIBRATION TABLE" in the lower right corner of the screen shows 12 gallons.

(32) Press the 'TNK SEL' softkey.

(33) Rotate the small FMS knob to generate a pick list and select the Right Tank.

(34) Press the 'ENT' key.

(35) Repeat steps (26) thru (30) for the Right Tank.

(36) Add six (6) gal fuel to each wing.

(37) After at least 5 minutes, Verify the 'CALIBRATION VALUE' indication for the Right Tank is stable before proceeding.

(38) Press the 'FUEL QTY' softkey.

(39) Enter 18 gal for 'ACTUAL FUEL QUANTITY'. Press the 'ENT' key to accept.

(40) Press 'ENT' Key again to calibrate.

(41) Verify the "ACTUAL QUANTITY" shown in the "CALIBRATION TABLE" in the lower right corner of the screen shows 18 gallons.

(42) Press the 'TNK SEL' softkey.

(43) Rotate the small FMS knob to generate a pick list and select the Left Tank.

(44) Press the 'ENT' key.

(45) Repeat steps (36) thru (40) for the Left Tank.

(46) Add six (6) gal fuel to each wing.

(47) After at least 5 minutes, Verify the 'CALIBRATION VALUE' indication for the Left Tank is stable before proceeding.

(48) Press the 'FULL' softkey. Press the 'ENT' key to acknowledge the prompt after the 'FULL' softkey is pressed

(49) Verify the "ACTUAL QUANTITY" shown in the "CALIBRATION TABLE" in the lower right corner of the screen shows 24 gallons.

(50) Press the 'TNK SEL' softkey.

(51) Rotate the small FMS knob to generate a pick list and select the Right Tank.

(52) Press the 'ENT' key.

(53) Repeat steps (46) thru (48) for the Right Tank.

(54) Remove ground power from the aircraft.

B. Fuel Quantity Accuracy Check

NOTE: If an aircraft fails this check, troubleshoot to determine cause and corrective action and repeat the fuel calibration and accuracy check, if required.

(1) Level aircraft to within ± 0.25 degrees of pitch and roll and ensure the wing tanks are completely empty.

(2) Add unusable fuel to the wings (1 US gal per side) and allow to stabilize (5 min.).

(3) Verify the fuel quantity indicator shows left and right fuel quantities at 0 gal.

(4) Verify fuel quantity gauge digits are RED .

(5) Add 3 gal of fuel to the wings and verify the corresponding reading on the Garmin system is within ± 1 gal per side (± 2 gal total).

(6) Add 1 gal of fuel to each wing. Verify fuel quantity gauge digits turn from RED to AMBER.

(7) Add 2 gal of fuel to each wing. Verify fuel quantity gauge digits turn GREEN upon reaching > 5 gal of fuel per side. Verify reading is within ± 4 gal per side.

(8) Continue adding fuel in 3 gal increments (6 additional increments) and verify the corresponding reading on the Garmin system is within ± 4 gal per side (± 8 gal total).

(9) If quantity readings at all intervals are within limits, then fuel accuracy check is complete. Turn aircraft power OFF.

C. Calibration Table Delete/Edit

This procedure can be used to edit or delete an existing calibration point within the calibration table.

(1) Press the CAL TBL softkey.

(2) Scroll down to the entry to edited or deleted.

(3) To edit the entry press the EDIT softkey. Fill the selected tank to the level of the selected entry. Allow the CALIBRATION VALUE to stabilize.

(4) Press enter and confirm you want to overwrite the existing calibration point.

(5) To delete the entry press the DELETE softkey.

(6) Press and confirm you want to delete the calibration point.

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CHAPTER

30

ICE AND RAIN PROTECTION

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CHAPTER 30

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
30-List of Effective Pages	1	Mar 6/14			
	2	Mar 6/14			
30-Table of Contents	1	Mar 6/14			
	2	Mar 6/14			
30-00-00	1	Mar 6/14			
	2	Mar 6/14			
30-30-00	1	Mar 6/14			
	2	Mar 6/14			

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CHAPTER 30 - ICE AND RAIN PROTECTION

TABLE OF CONTENTS

SUBJECT	SECTION	PAGE
GENERAL	**30-00-00**	1
PITOT AND STATIC	**30-30-00**	1

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GENERAL

The Archer III is not approved for operation in icing conditions.

Accordingly, limited ice and rain protection is provided in the form of an optional heated pitot-static mast (See 30-30-00), carburetor heat, and an optional carburetor detection system (See 71-60-00).

Carburetor heat is activated by a lever on the instrument panel which controls the flow of heated air from the heater muff to the carburetor airbox.

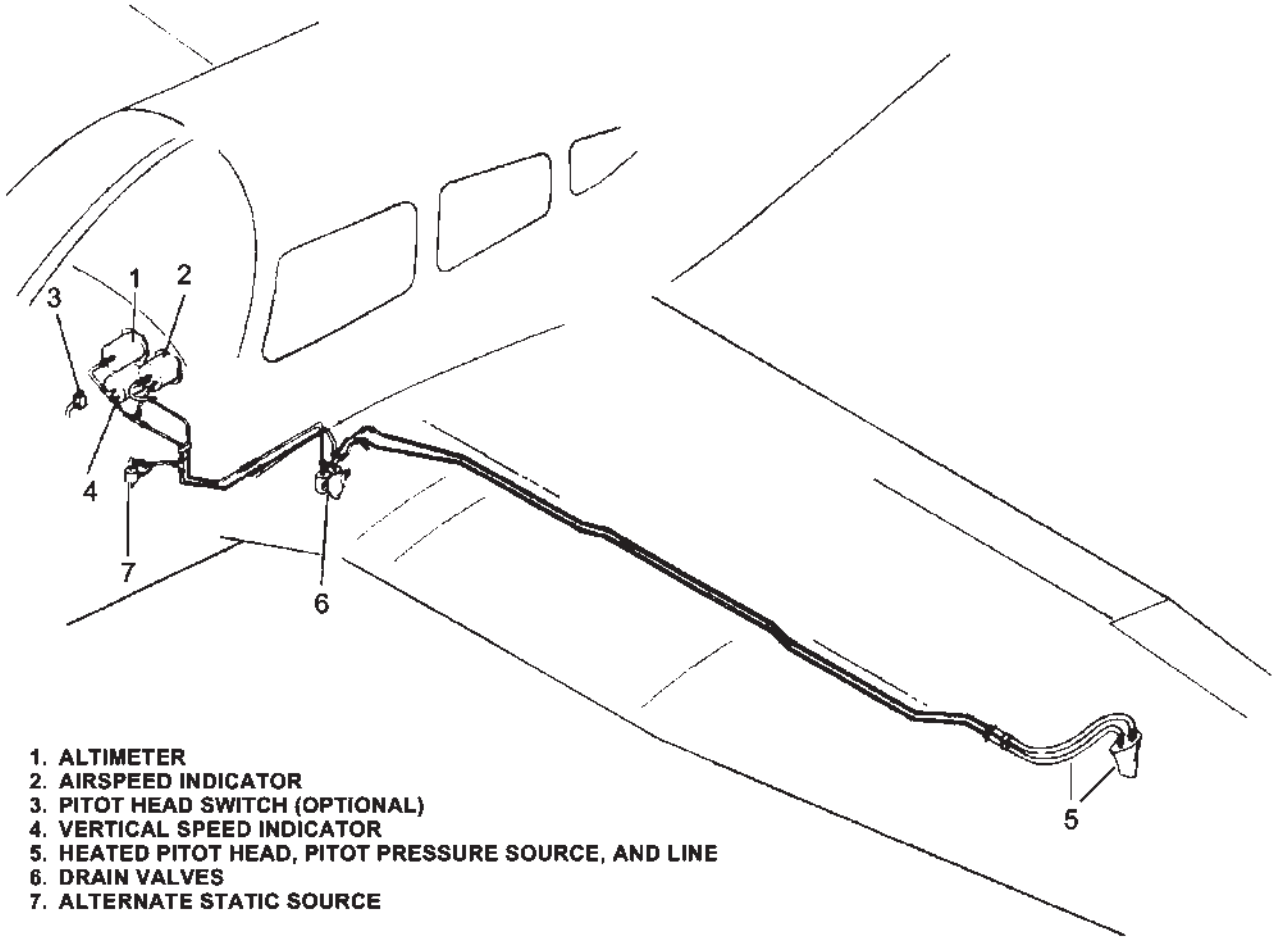
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PITOT AND STATIC

The pitot heat system (optional) is activated by the PITOT HEAT switch located on the center instrument panel just above the engine power quadrant. In addition to the switch, the system incorporates a heated pitot-static head and the 10 amp PITOT HEAT circuit breaker.

The pitot-static head is installed on the left wing.

NOTE: Refer to 91-30-30 for pitot heat wiring schematic.



Heated Pitot-Static System Installation (Optional)
Figure 1

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CHAPTER

31

INDICATING / RECORDING SYSTEMS

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CHAPTER 31

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE	CHAPTER SECTION	PAGE	DATE
31-List of Effective Pages	1	Jul 30/21			
	2	Mar 6/14			
31-Table of Contents	1	Jul 30/21			
	2	Nov 15/17			
31-20-00	1	Jul 30/21			
	2	Sep 1/16			
	3	Nov 15/17			
	4	Sep 1/16			
31-30-00	1	Sep 1/16			
	2	Mar 6/14			
	3	Mar 6/14			
	4	Sep 1/16			
	5	Mar 6/14			
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	8	Sep 1/16			
	9	Mar 15/20			
	10	Sep 1/16			
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	12	Nov 15/17			
	13	Nov 15/17			
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31-50-00	1	Nov 15/17			
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	3	Nov 15/17			
	4	Nov 15/17			
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	6	Nov 15/17			
	7	Nov 15/17			
	8	Nov 15/17			

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CHAPTER 31 - INDICATING / RECORDING SYSTEMS

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Removal		2
Installation		2
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Installation		1
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Description and Operation		1
Troubleshooting		4
Annunciation in Garmin G1000 / G1000 NXi-Equipped Airplanes		7
Description		7
Aural Alerts		7

INDEPENDENT INSTRUMENTS

WARNING: IF THE PFD DISPLAYS 'CO DET SRVC' IN THE MESSAGES WINDOW, THE CO DETECTOR HAS REACHED THE END OF ITS SERVICE LIFE AND MUST BE REPLACED.

WARNING: IF THE PFD DISPLAYS 'CO DET FAIL' IN THE MESSAGES WINDOW, THE CO DETECTOR MAYBE DEFECTIVE. SEE "UNIT FAILURE", BELOW.

Carbon Monoxide (CO) Dectector (Optional)

Available in Garmin G1000 NXi-equipped airplanes, S/N's 2843823, 2843852, 2843854–2843999; and 2881001 and up.

See "Figure 1" on page 31203.

The Carbon Monoxide (CO) Detector is designed to detect, measure, and provide a visual alert to the crew of reciprocating engine type aircraft before the cockpit level of carbon monoxide (CO) reaches a critical level.

When installed, the detector is mounted under the instrument panel to the left sidewall bracket just aft of the firewall. RS-232 data from the detector is transmitted to the GIA 63W No. 1 where it is processed and forwarded to the MFD which displays CO Level in the EIS strip. There is a 'CO RST' softkey on the engine page to reset the CO detector.

A. Operation

(1) Service Life

The CO Detector has a usable life of five (5) to seven (7) years. To ensure crew safety, and per the manufacturer's recommendation, replace the unit with a new or recalibrated unit every five (5) years.

(2) Crew Alerts

CO Level is displayed in the MFD EIS strip as 'CO PPM XX'. There is a three-minute delay at startup to stabilize the sensor before the unit will accurately sense CO levels.

(a) If CO PPM ≥ 200, then title and digits are RED. And the RED CAS Alert Message 'CO LVL HIGH' appears on the MFD.

(b) If CO PPM ≥ 50 AND < 200, then title and digits are AMBER. And the AMBER CAS Alert Message 'CO LVL HIGH' appears on the MFD.

(c) Otherwise, title is white and digits are GREEN.

NOTE: The carbon monoxide alarm level is calibrated to provide a visual alert within 5 minutes or less whenever the carbon monoxide level reaches 50 parts per million (PPM) by volume or greater per TSO C48A. The warning time is shortened at higher levels of CO concentrations and becomes approximately instant should the carbon monoxide level reach 400 parts per million by volume (PPM) or greater.

(3) Unit Failure

If the unit fails, the PFD will display the following system message in the messages window: 'CO DET FAIL'.

(a) In case of a failure indication, attempt to clear the failure condition by resetting the CO Detector. Press the 'CO RST' softkey on the engine page.

(b) If the failure condition continues after reset, cycle the CO Detector power by pulling and resetting the CO Detector circuit breaker. Wait three minutes for the sensor to stabilize.

(c) If the failure condition continues after softkey reset and cycling the power to the unit, replace the CO Detector.

B. Inspection

The carbon monoxide detector requires no periodic servicing or maintenance. At every power up the system performs a self-diagnostic check. If the unit fails, the PFD will display the following system message in the messages window: 'CO DET FAIL'. See "Unit Failure", above.

C. Removal

(1) Turn the master switch OFF.

(2) Disconnect the negative (ground) cable from the main battery. Disconnect the P615 connector from the emergency battery.

(3) Disconnect the electrical harness from the CO detector unit.

(4) Remove three scews and remove unit from its mounting angle.

D. Installation

(1) Position the CO detector unit on its mounting angle and secure with three screws.

(2) Connect the electrical harness to the CO detector unit.

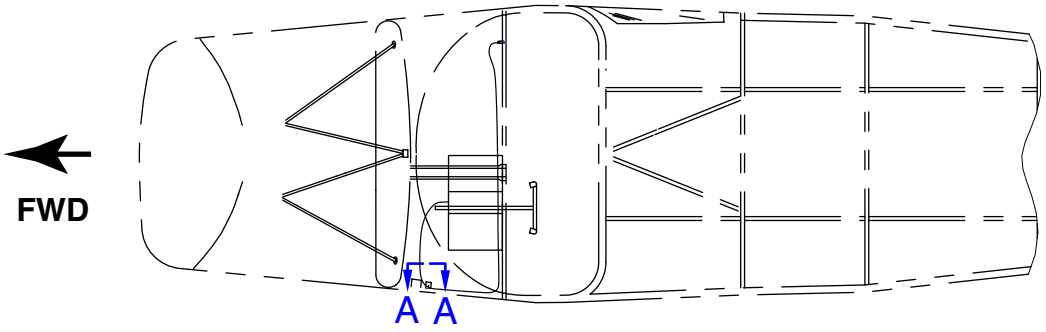
(3) Connect the P615 connector to the emergency battery.

(4) Connect the negative (ground) cable to the main battery.

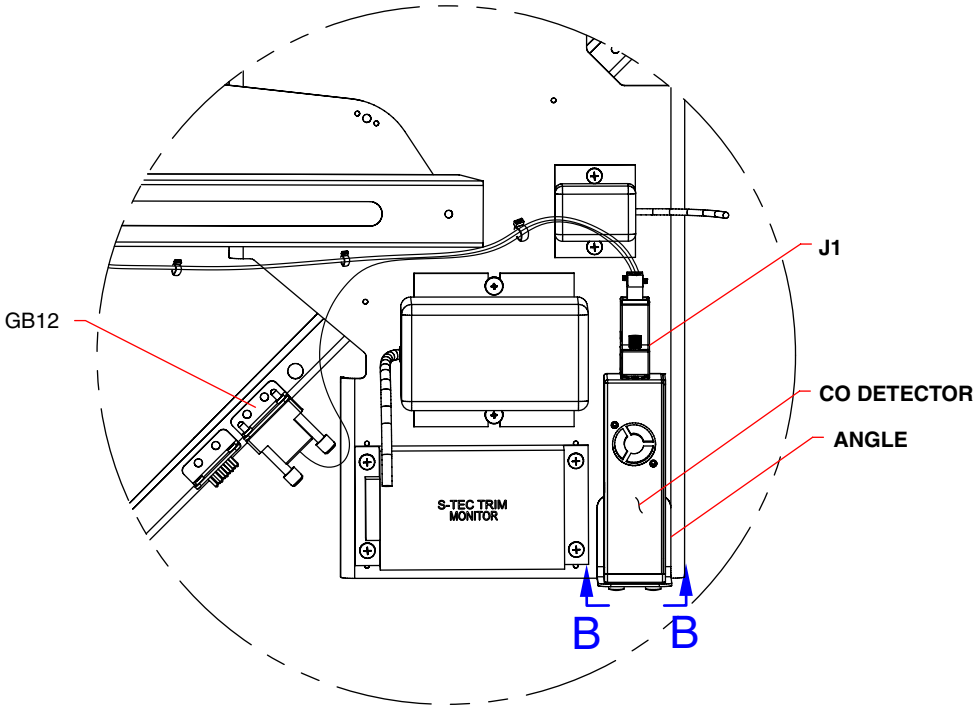
(5) Turn the master switch ON.

(6) Start the MFD and PFD normally. Verify 'CO PPM XX' appears in the EIS strip on the MFD and no CO detector system messages appear in the PFD messages window.

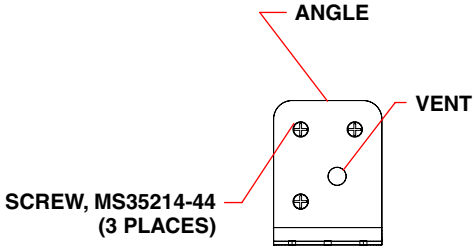
108040 NEW



LOOKING DOWN



VIEW A - A
(LHS LOOKING OUTBOARD)



VIEW B - B
(BOTTOM OF
CO-DETECTOR BRACKET)

CO Detector Installation (Optional)
Figure 1

Effectivity
with Garmin G1000 NXi

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RECORDERS

1. Vision 1000 Flight Data Monitoring System (Optional)

Available in S/N's 2843711 and up.

The Appareo Systems' Vision 1000 Flight Data Monitoring System gathers data from global positioning, image capturing, flight attitude acquisition and ambient audio recording to provide high fidelity information regarding aircraft movement through space. Data collected is written in two locations in real time. One location is in a protected back up internal memory and the other a removable SD card.

The installation consists of a Vision 1000 camera/recorder unit and a GPS antenna. An instrument panel mounted intercom system connected to passenger/copilot mic and phone jacks with an integral push to transmit button is also installed (see 23-50-00). That system is wired to give the pilot transmit priority over the passengers/copilot.



Vision 1000 Unit
Figure 1

A. Vision 1000 Unit

The Vision 1000 unit is mounted just aft of the co-pilot's head, 3.9 inches below the cockpit overhead and 4.7 inches right of the airplane centerline at F.S. 108.18. See "Figure 2" on page 31302.

(1) Removal

(a) Remove electrical power from the airplane.

(b) Disconnect the GPS antenna coax cable from the coax connector and the harness from the P1 Port.

(c) Remove nut, bolt, and two spacers and remove Vision 1000 unit.

(2) Installation

(a) Position Vision 1000 unit with clamping plate properly engaged and secure unit with bolt, nut, and two spacers.

(b) Connect the GPS antenna coax cable to the coax connector and the harness to the P1 Port.

(c) If the Vision 1000 unit installed is new or a replacement unit, configure it per Configuration, below.

NOTE: Also document installation of a new or replacement unit with an appropriate logbook entry.

(d) Conduct "Function Test" on page 31304.

B. GPS Antenna

The Vision 1000 GPS antenna is mounted to a bracket just above the cowl deck cover and to the right of the windshield center post. See "Figure 2" on page 31302.

(1) Removal

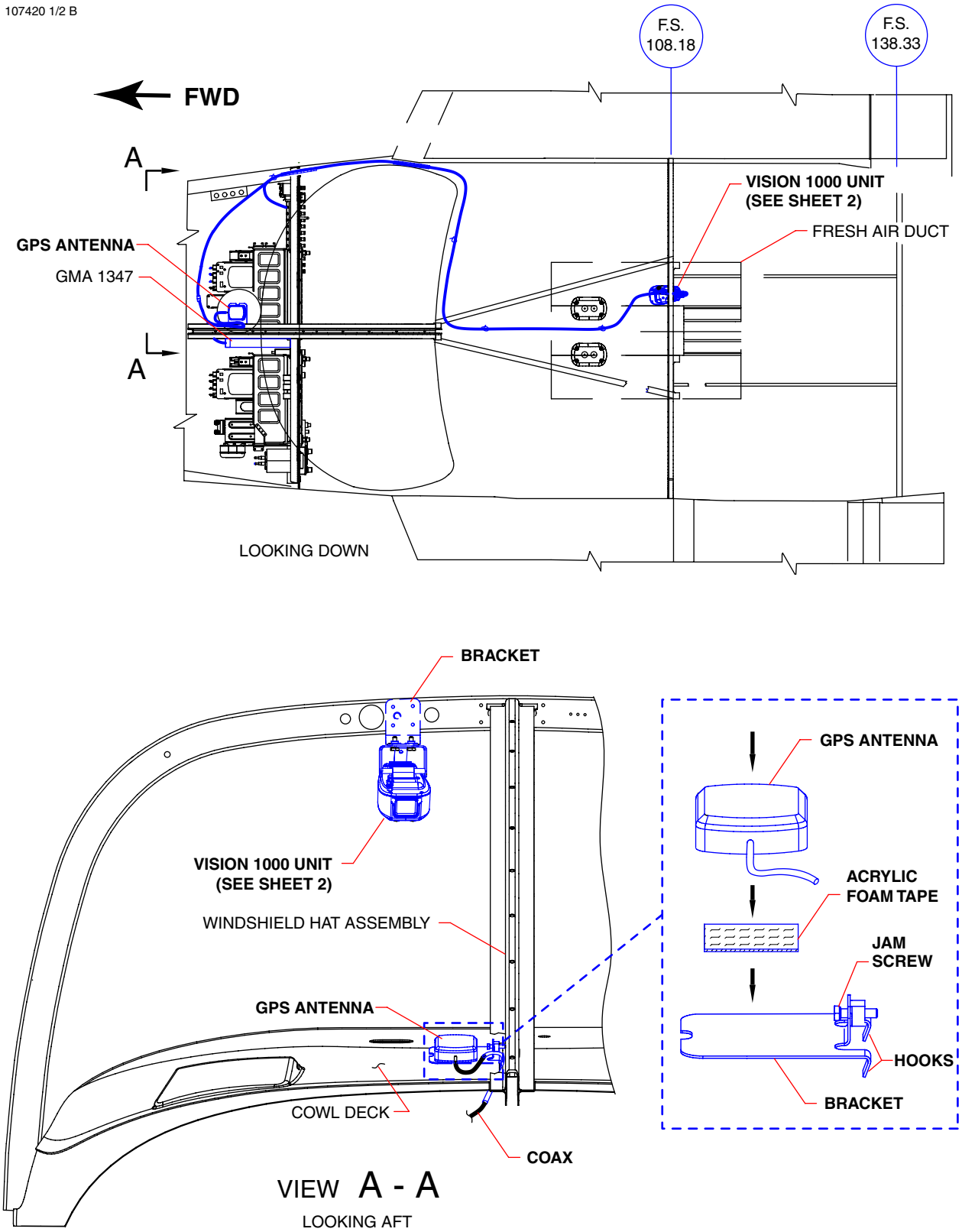
(a) Remove electrical power from the airplane.

(b) Behind the instrument panel and under the cowl deck, disconnect the SMB Connector in the GPS antenna coax cable.

(c) Loosen the jam screw on the antenna mounting bracket sufficient to disengage the mounting bracket hooks from the cowl deck.

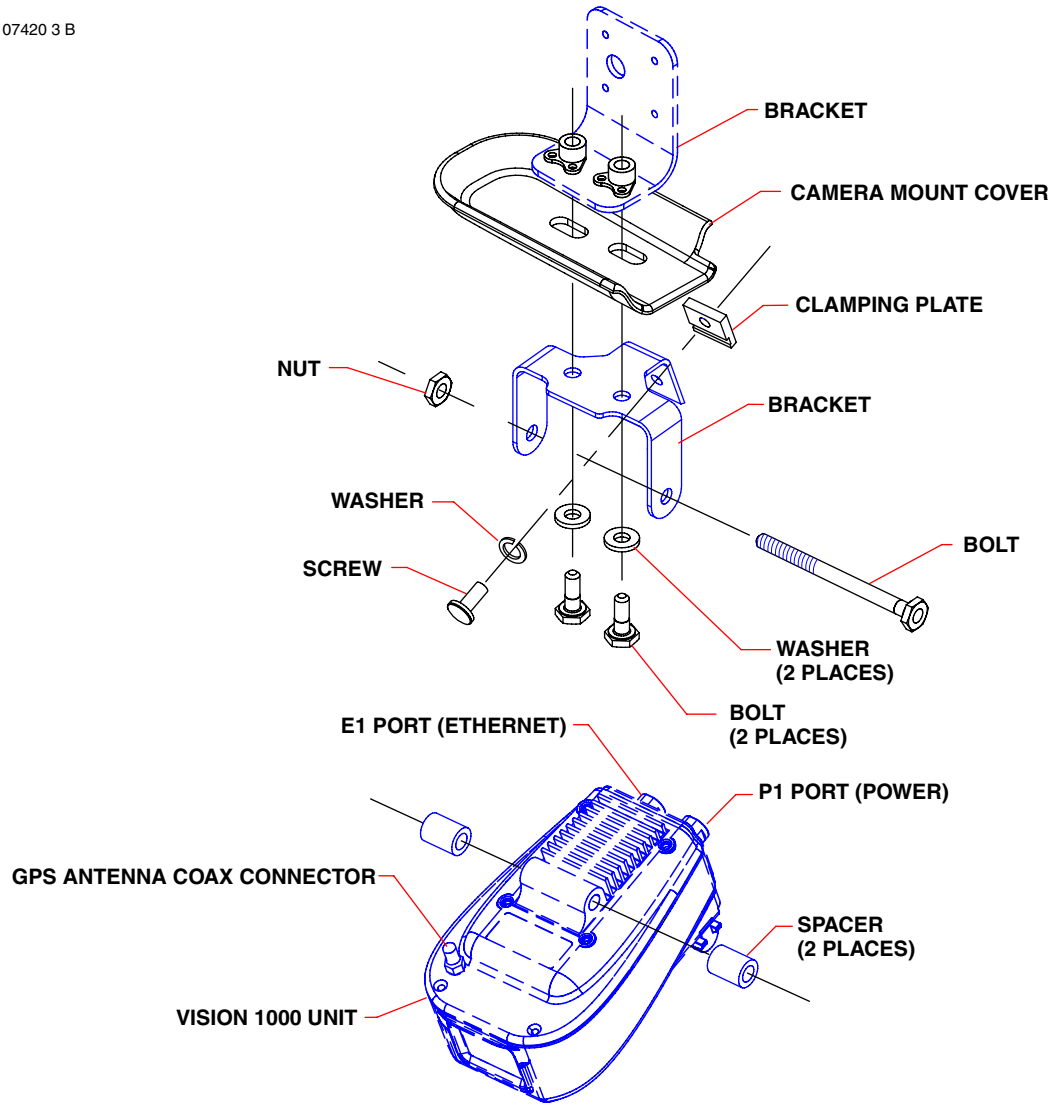
(d) Remove the antenna and mounting bracket, taking care to feed the coax cable pigtail out through the gap between the cowl deck and the windshield hat assembly.

107420 1/2 B



Effectivity
2843711 and up

Vision 1000 System Installation (Optional)
Figure 2 (Sheet 1 of 2)



Vision 1000 System Installation (Optional)
Figure 2 (Sheet 2 of 2)

Effectivity
2843711 and up

(2) Installation

(a) If replacing the antenna:

1) Separate the old antenna from the bracket with a knife.

2) Clean the double-stick foam tape residue from the bracket with an appropriate solvent such as Goo Gone.

3) Install the new antenna to the bracket with 3M VHB Foam Tape No. 5962 sufficient to cover the full footprint of the antenna.

(b) Feed the coax cable pigtail down through the gap between the cowl deck and the windshield hat assembly.

(c) Engage the mounting bracket hooks over the inboard end of the cowl deck adjacent to the windshield hat assembly and secure with the jam screw.

(d) Behind the instrument panel and under the cowl deck, connect the GPS antenna coax cable at the SMB Connector.

(e) Perform Function Test, below.

C. Function Test

Conduct the functional test in an area where the airplane has an unimpeded view of the sky, so that a proper GPS fix can be established.

(1) Connect power to the aircraft.

(2) Ensure the Avionics Master Switch is ON.

(3) Open the SD access door.

Push SD door latch to the side and swing card door open.

(4) Insert SD card.

(a) Place SD card vertically into the SD slot with the contacts facing forward.

(b) Push SD card into the Vision 1000 unit until it is recessed into the enclosure and release. The card should return a flush position.

(5) Observe LED, blue LED should be on continuously until unit has booted.

(a) Solid Green LED indicates that the system has booted and is functioning.

(b) If a solid Yellow LED is seen, refer to “Chart 1” on page 31305.

(6) Close the card door and latch shut.

(7) Remove power from aircraft.

D. Configuration

When installing a new or replacement Vision 1000 Unit, it must be configured per the following procedure.

(1) Required Equipment

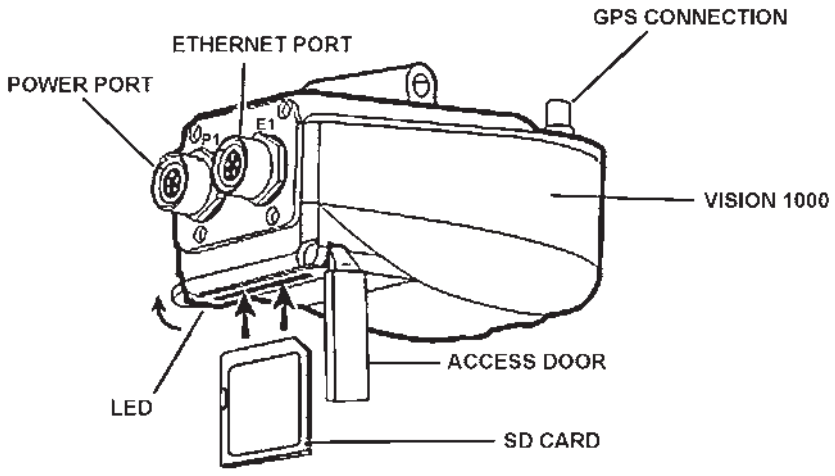
(a) System Configuration Tool Software (see Downloadable Software, below).

(b) Windows Laptop Computer

(c) Appaero System’s Ethernet Cord, P/N 153560-000004.

CHART 1
VISION 1000 LED STATUS

Item	LED Status	Configuration
1	Red	Contact Appareo Systems.
2	Blue	Booting.
3	Green	Operating.
4	Yellow	- SD card not inserted: insert SD card and verify Green LED. - SD card not formatted correctly: format SD to NTSF, verify Green LED. - GPS lock not received: Allow 15 minutes to clear, if problem persists contact Appareo Systems.
5	NO LED	- Make sure fuse is not blown: replace fuse. - Make sure avionics master switch is on. - Make sure avionics bus circuit breaker is closed. - Check power harness for continuity. - If all checks pass and LED is still not functioning, contact Appareo Systems.



Vision 1000 Unit Details
Figure 3

Effectivity
2843711 and up

(2) Procedure

The Vision 1000 must be powered ON during configuration.

(a) Connect a laptop to the Vision 1000 Unit:

1) Turn ON aircraft power and avionics master.

2) While the Vision 1000 is starting up, a blue LED located behind the SD card door will illuminate. After it has finished its start up, the LED will switch to green.

3) Once the LED has been green for approximately two (2) minutes, you may proceed.

4) Insert the appropriate end of the Ethernet cord into the computer's Ethernet port.

CAUTION: PINS ON THE ETHERNET CORD AND PORT MAY BEND IF THE ETHERNET CORD IS NOT HANDLED PROPERLY. ENSURE THE WHITE DOT OF THE ETHERNET CORD LINES UP WITH THE WHITE DOT ON THE E1 JACK ON THE BACK OF THE VISION 1000 UNIT.

5) Connect the other end of the Ethernet cord to the Vision 1000 E1 port, as shown in "Figure 3" on page 31305.

(b) Log in to the Vision 1000 configuration tool:

NOTE: The Configuration Tool can be obtained as specified under Downloadable Software, below.

1) Open Vision 1000 configuration tool on laptop.

2) Ensure the configuration tool is connected to a Vision 1000 and the Vision 1000 is powered ON (the status LED will be solid green if the Vision 1000 is powered on).

3) Wait until the Vision 1000 status orb is green and has the text "Device Connected" (top left corner).

4) Ensure that "Operator" is selected in the role field.

5) Type the password in the Password field. (Password = h2StAste .) (Password is case sensitive and cannot be altered.)

NOTE: Once logged in, you will have a live streaming image.

(c) Built-in-Test (BIT)

1) Select BIT tab on application screen.

2) Select "Start Built In Test" and ensure all tests pass.

a) SD card must be inserted.

b) Aircraft must be located where GPS signal can be received.

3) Overall status (top right of BIT screen) indicates:

a) all the BITs have passed (displays a green orb with the text "pass"),

b) a field-serviceable BIT has failed (displays a yellow orb with the text "warning")

c) one of the non-serviceable BITs have failed (displays a red orb with the text "fail").

NOTE: If either b) or c), contact Appareo Systems.

(d) Enter the tail number:

1) Under the IMU Zero Point configure you will see an attitude indicator, below that you will see an aircraft tail number input.

2) Enter the airplane registration number in the Tail # field and click enter.

NOTE: This information will be used in the creation of file names recorded by the Vision 1000.

(e) Configure the IMU Zero Point:

1) Level the airplane per 8-20-00.

2) Select configure tab; with aircraft stationary, enter zero for both roll and pitch and -6 for yaw in the appropriate fields.

3) Once you have entered the roll, pitch, and yaw click “configure”.

NOTE: If the configure button is not available (appears grayed out) an IMU Zero Point value is missing.

(f) You may now log out.

CAUTION: IN ORDER FOR THE CHANGES YOU MAKE WITH THE CONFIGURATION TOOL TO TAKE EFFECT, YOU MUST “POWER CYCLE” THE VISION 1000.

(g) Power cycle the Vision 1000:

1) For airplanes equipped with Garmin G1000 in S/N’s 2843701, 2843703 and up: Turn OFF avionics master switch (on the aircraft’s main circuit breaker panel) to power OFF the Vision 1000.

2) For airplanes equipped with the Garmin 500: Turn OFF radio master switch (on the aircraft’s instrument panel) to power OFF the Vision 1000.

3) Turn ON avionics or radio master switch to power ON Vision 1000.

4) Observe the Vision 1000’s status LED light, located behind the SD card access door. (See “Figure 3” on page 31305.)

a) While the Vision 1000 is starting up, the LED status light will be solid blue for several minutes.

b) Once the Vision 1000 has properly booted up, the LED status light will be solid green. This signifies that the Vision 1000 is ready to record flight data.

E. Downloadable Software

Download the Vision 1000 Configuration Tool and the Vision 1000 Playback Utility software programs.

(1) Navigate to http://www.appareo.com/account in an Internet Web browser.

(2) Sign in using an existing Appareo account, or create a new account by clicking the Click Here to create a new account link.

(3) Click Add Product.

(4) Select the appropriate software program from the product drop-down list (the software programs must be added to your account separately). Enter the appropriate product key from the fields below and click Submit.

(a) Software Product Key:

20420413 - 7EF14C60 - A6675 - 5E1020 - AEE1F

(b) Vision 1000 Configuration Tool Password:

h2StAste (Case Sensitive)

(c) Vision 1000 Playback Utility Software Product Key:

032517E8 - 42C84EAC - 98067 - 920010 - E4581

(5) Click Home (in the Manage Your Account column) and click the red download link for the product. Follow the on-screen instructions to finish the installation process. You may download the installer at any time using these links.

(a) If you do not currently have an Appareo Web account, you must create one before downloading these software program installers from the Appareo website. Creating an Appareo Web account is free and only takes a few minutes

(b) If you are unable to download these software programs from the Appareo website and a CD was not already included in your installation kit, contact Appareo to request an installer CD Vision 1000 Configuration Tool.

(6) Applying future Vision 1000 embedded software versions.

Appareo may release updated versions of the Vision 1000's embedded software in the future. Each embedded software release will be accompanied by a Service Bulletin that will include step-by-step instructions for how to apply the embedded software release to your Vision 1000; however, each company must use a unique key in order to download the release from the Appareo website. This embedded software key is included in the field below. Ensure that you retain this document so that you may apply any embedded software versions of your Vision 1000 in the future.

Vision 1000 Embedded Software Key:

2. Hobbs Meter(s)

A. Standard, Avidyne Entegra, or Garmin G500 Equipped Airplanes

S/N's 2843001 and up. See "Figure 4" on page 313010.

(1) Engine Hour Meter

An engine hour Hobbs meter is standard. It indicates and records actual engine operating time. The engine hour meter is installed in the right side of the instrument panel. It is supplied 28vdc power from the battery contactor through a 5 amp fuse and is grounded through an oil pressure switch mounted to the rear of the firewall. Oil pressure is sensed at engine start, the switch closes, the engine hour meter runs.

in S/N's 2843807 and up only, when the optional Flight Time Meter is installed, the engine hour meter will be labelled "MAINTENANCE".

(a) Removal

1) Turn the master switch OFF.
2) Disconnect the negative (-) (i.e., ground) battery cable.
3) Locate the meter in the rear instrument panel.
4) Disconnect the electrical leads from the rear of the meter.
5) Remove the screws, washers, and nuts (2 ea.) securing the meter to the panel.
6) Remove the meter from the cut-out in the panel.

NOTE: Early airplanes have the unit mounted through the instrument panel with the meter bezel exposed on the front (aft) side of the instrument panel. Later airplanes are mounted to the rear of the instrument panel with the bezel hidden from view.

(b) Installation

1) Ensure the master switch OFF and the negative (-) (i.e., ground) battery cable is disconnected.
2) Position the meter in the cut-out in the instrument panel.
3) Secure with screws, washers, and nuts (2 ea.).
4) Connect the electrical leads to the rear of the meter.
5) Connect the negative (-) (i.e., ground) battery cable.
6) At the earliest opportunity, start engine and verify engine hour meter operates properly.

(2) Flight Time Meter

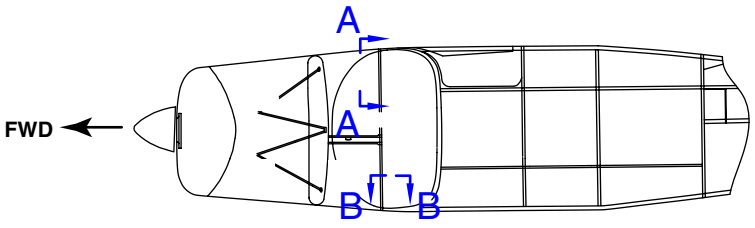
Available in S/N's 2843807 and up only.

An additional Hobbs meter is optional to indicate and record actual aircraft flight time. The flight time meter is installed to the left of the standard engine hour (i.e., maintenance) meter in the right side of the instrument panel. It shares a 28vdc common power source with the standard engine hour meter and is grounded through a pitot pressure switch mounted to left side of the airplane under the instrument panel. The flight time meter activates at pitot pressure greater than 49 KIAS.

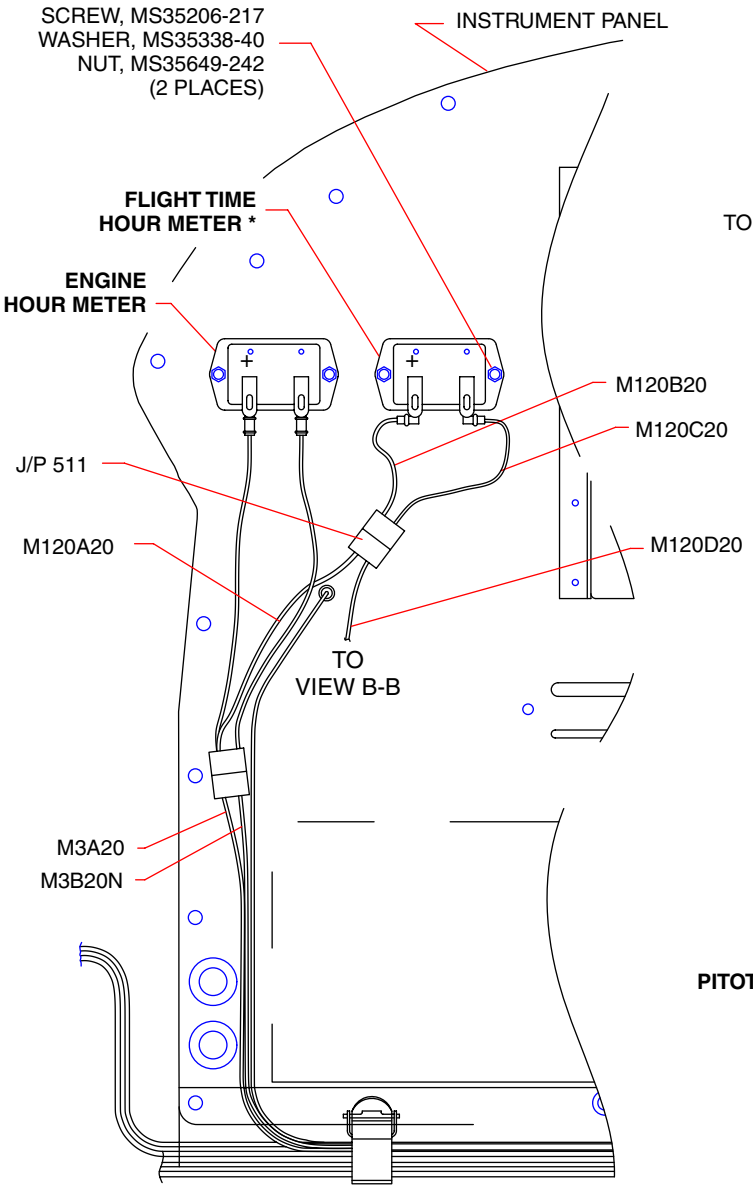
When installed, this meter will be labelled "FLIGHT".

Removal and Installation are the same as given under "Engine Hour Meter", above.

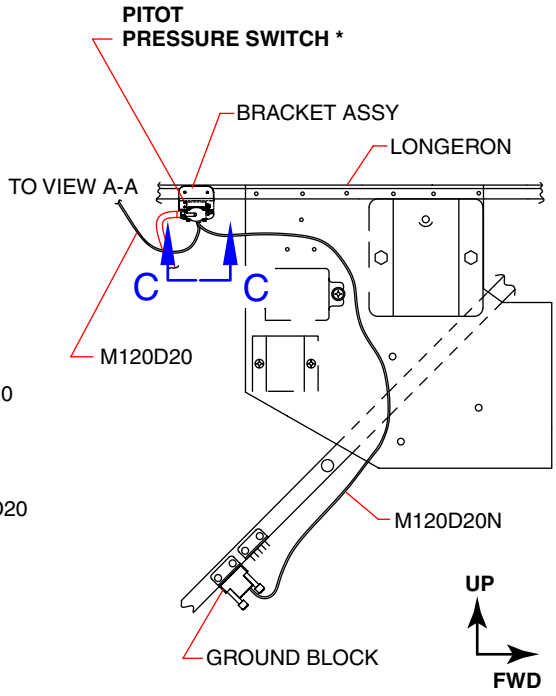
107844 C 2



LOOKING DOWN

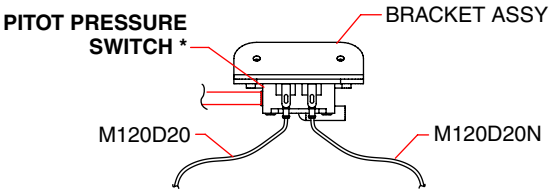


VIEW A - A
(LOOKING AFT AT INSTRUMENT PANEL)



VIEW B - B
(LOOKING OUTBOARD AT LEFT SIDE)

(* Optional in Standard or Garmin G500, S/N's 2843807 & up only.)



VIEW C - C
(LOOKING UP)

Effectivity
Standard, Avidyne Entegra, or Garmin G500

Hobbs Meter Installations
Figure 4

B. Garmin G1000/G1000 NXi Equipped Airplanes

(1) S/N's 2843701, 2843703–2843822, 2843824–2843851 and 2843853 (i.e., G1000)

(a) Hour Meter (Electro-Mechanical)

A Hobbs meter is standard. It is installed in the right side of the instrument panel. 28vdc power is supplied from the hot battery bus through a 5 amp fuse. The meter is then connected to both GIA 63Ws. Logic in the Garmin G1000 software activates the meter based on which "Hobbs Installation Configuration" has been selected in the G1000 software:

A. Runs when engine is running (determined by Garmin engine run logic)

B. Runs when aircraft is in the air (squat switches)

C. Runs whenever aircraft is electrically powered (GIA1 is operational)

Hobbs meter activation is suppressed when the G1000 system is in Maintenance mode.

Removal and Installation are the same as given under "Engine Hour Meter" on page 31309.

(b) Hour Meter (MFD "Tach Time" Display)

The Garmin G1000 system features an hour meter display in the MFD. Displayed as "Tach Time" this hour meter display is also configurable in the G1000 software. "Tach Configuration" can be set to one of the following options:

A. Run time for engine (tach time increases whenever engine is running)

B. Time with engine running AND aircraft in Air (basically a flight timer)

C. Scaled engine run time. Engine time increases at a scaled factor based on RPM. Low RPM makes the timer increase slowly, while high RPM runs at normal speed.

(2) S/N's 2843823, 2843852, 2843854–2843999; 2881001 and up (i.e., G1000 NXi)

See "Figure 5" on page 313013.

(a) Hour Meter (Electro-Mechanical)

A Hobbs meter is standard. It is installed in the right side of the instrument panel. 28vdc power is supplied from the hot battery bus through a 5 amp fuse. The meter is then connected to both GIAs. Logic in the Garmin G1000 NXi software activates the meter based on which "Hobbs Installation Configuration" has been selected in the G1000 NXi software:

A. Runs when engine is running (determined by Garmin engine run logic)

B. Runs when aircraft is in the air (squat switches)

C. Runs whenever aircraft is electrically powered (GIA1 is operational)

Hobbs meter activation is suppressed when the G1000 NXi system is in Maintenance mode.

When the optional "Maintenance Hour Meter" (see page 313012) is installed, this meter is configured to record flight time and is labelled "BILLING".

Removal and Installation are the same as given under "Engine Hour Meter" on page 31309.

(b) Hour Meter (MFD "Tach Time" Display)

The Garmin G1000 NXi system features an hour meter on the MFD. Displayed as "Tach Time" this hour meter display is also configurable in the G1000 NXi software. "Tach Configuration" can be set to one of the following options:

A. Run time for engine (tach time increases whenever engine is running)

B. Time with engine running AND aircraft in Air (basically a flight timer)

C. Scaled engine run time. Engine time increases at a scaled factor based on RPM. Low RPM makes the timer increase slowly, while high RPM runs at normal speed.

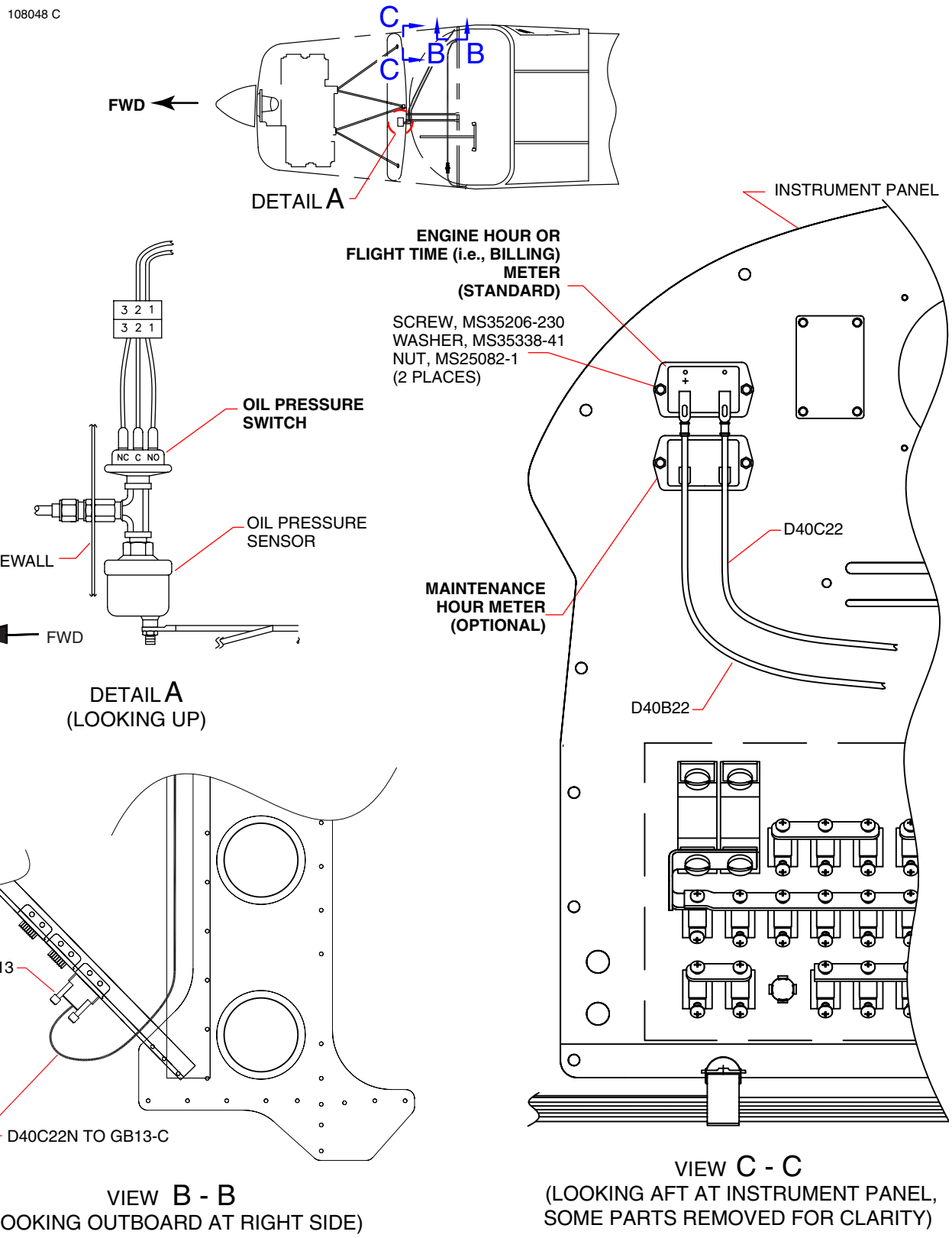
(c) Maintenance Hour Meter (Electro-Mechanical) (Optional)

See “Figure 5” on page 313013.

An additional Hobbs meter is available to indicate and record actual engine operating time. This meter is installed directly below the standard Garmin G1000 NXi-driven electro-mechanical meter, on the right side of the instrument panel. It shares the 28vdc power supplied from the hot battery bus through a 5 amp fuse to the standard meter and is grounded thru an oil pressure switch mounted to the rear of the firewall. Oil pressure is sensed at engine start, the switch closes, the engine hour meter runs.

When installed, this optional meter is labelled “MAINTENANCE”.

Removal and Installation are the same as given under “Engine Hour Meter” on page 31309.



Hobbs Meter Installations
Figure 5

Effectivity
with Garmin G1000 NXi

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CENTRAL WARNING SYSTEMS

1. Annunciator

Standard, Avidyne Entegra, or Garmin G500 equipped airplanes have a standalone annunciator panel located in the instrument panel. See “Figure 1” on page 31502.

A. Description and Operation

(1) The annunciator panel is a small cluster of lights which warn of malfunctions in the various circuits or systems. A malfunction is identified by the illumination of an individual warning light. Power is supplied from the bus bar through a 5-amp fuse located behind the switch panel.

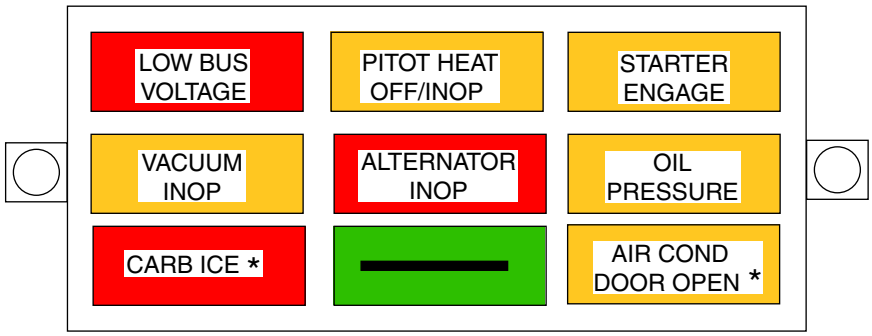
(2) An annunciator PRESS-TO-TEST switch is located to the right of the annunciator panel. With the BATT MAST switch in the ON position, depressing this switch will illuminate all installed annunciator lights.

(3) Functional Description

Annunciator	Color	Operation
ALTERNATOR INOP	RED	Illuminates when the alternator fails or is selected OFF.
CARB ICE	RED	Optional. Illuminates when carburetor ice detector senses ice in the carburetor throat.
LOW BUS VOLTAGE	RED	Illuminates when the electrical system drops from bus voltage (28 Vdc) to battery voltage (24.5 Vdc).
AIR COND DOOR OPEN	AMBER	Optional. Used if air conditioner is installed. Indicates proper A/C condenser door activation. Illuminates when A/C switch is ON, the condenser door is extended to an open position, and the fan switch is on (LOW or HIGH position).
OIL PRESSURE	AMBER	Illuminates when the in-line oil pressure sensor switch, located at the firewall, senses oil pressure has fallen below 35 psi.
PITOT HEAT OFF/INOP	AMBER	Illuminates when the pitot heat fails or is selected OFF.
STANDBY ALTNTR ON	AMBER	Illuminates when the primary alternator fails in flight and the standby alternator is automatically activated.
STARTER ENGAGE	AMBER	Illuminates when engine starter is engaged.
VACUUM INOP	AMBER	Illuminates when the vacuum sensor switch on the firewall is activated. The vacuum sensor switch is attached to a regulator, set to close when pressure falls below 3.5 inches of Hg.
	GREEN	Unassigned lights (2 or 3)

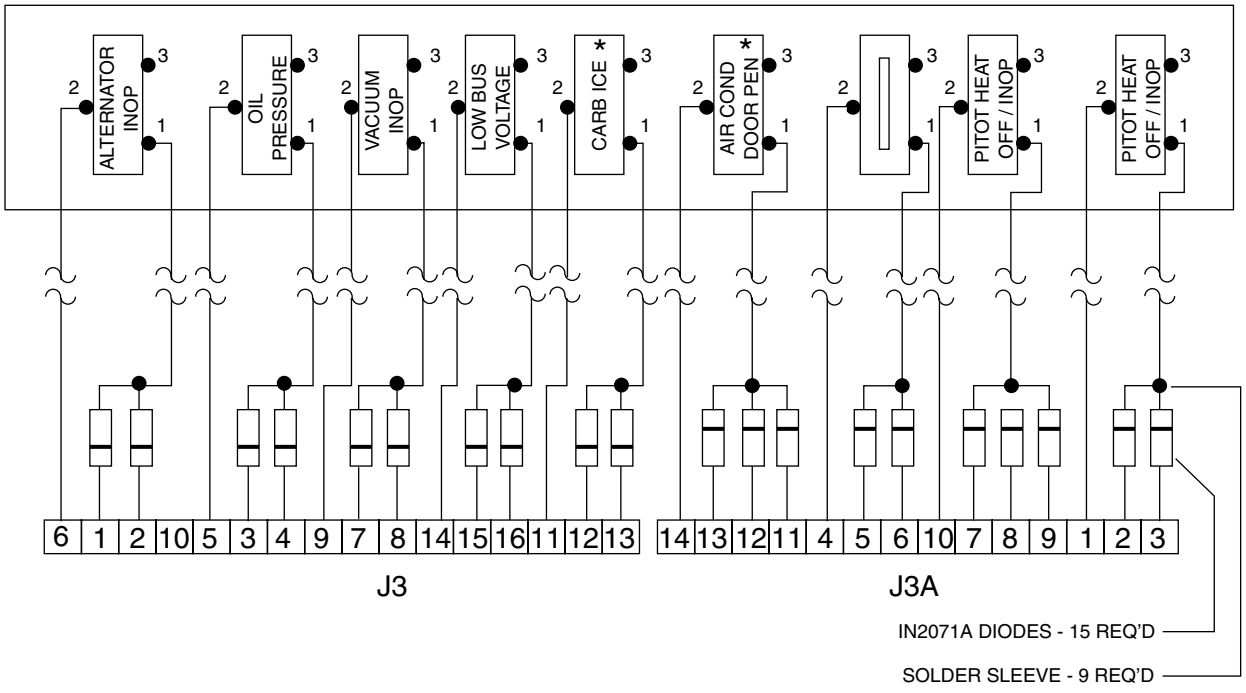
85372 M

LIGHT POSITIONS

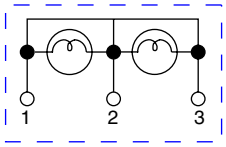


* OPTIONAL

ANNUNCIATOR LIGHT ASSEMBLY



* OPTIONAL



LIGHT SCHEMATIC

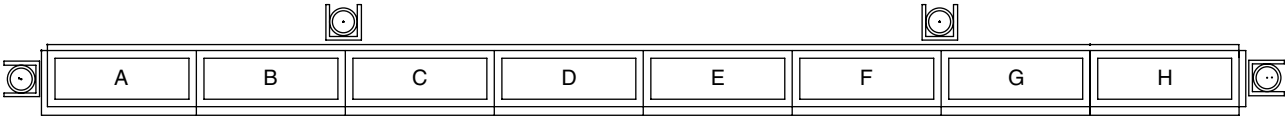
Effectivity
Standard or Garmin G500

Annunciator Panel
Figure 1 (Sheet 1 of 2)

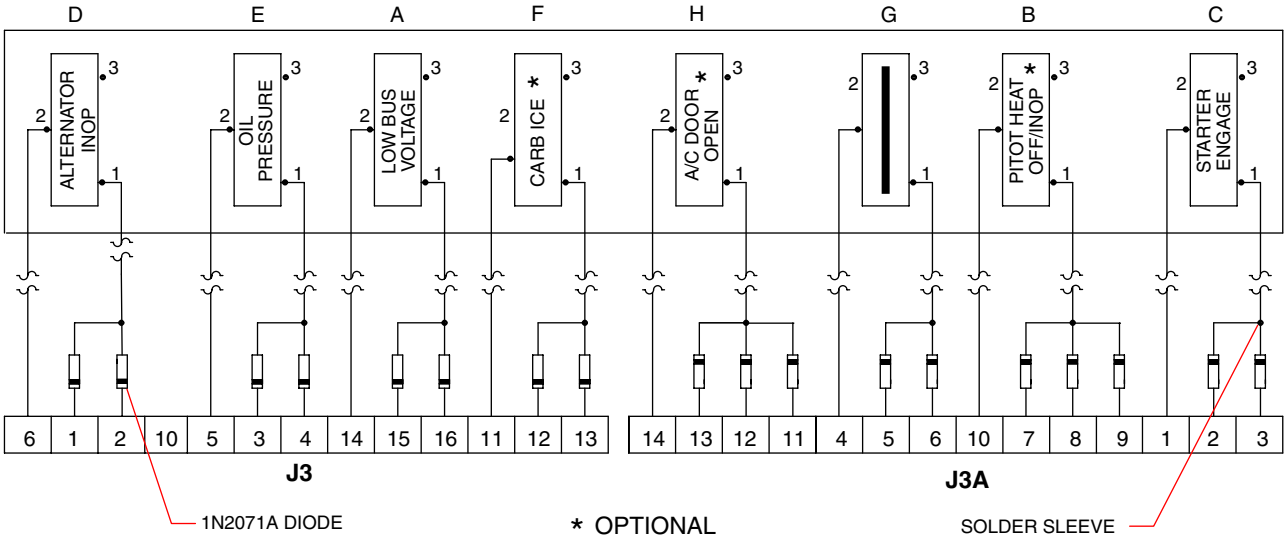
Effectivity
Standard, Avidyne Entegra, or
Garmin G500 equipped

85372 M

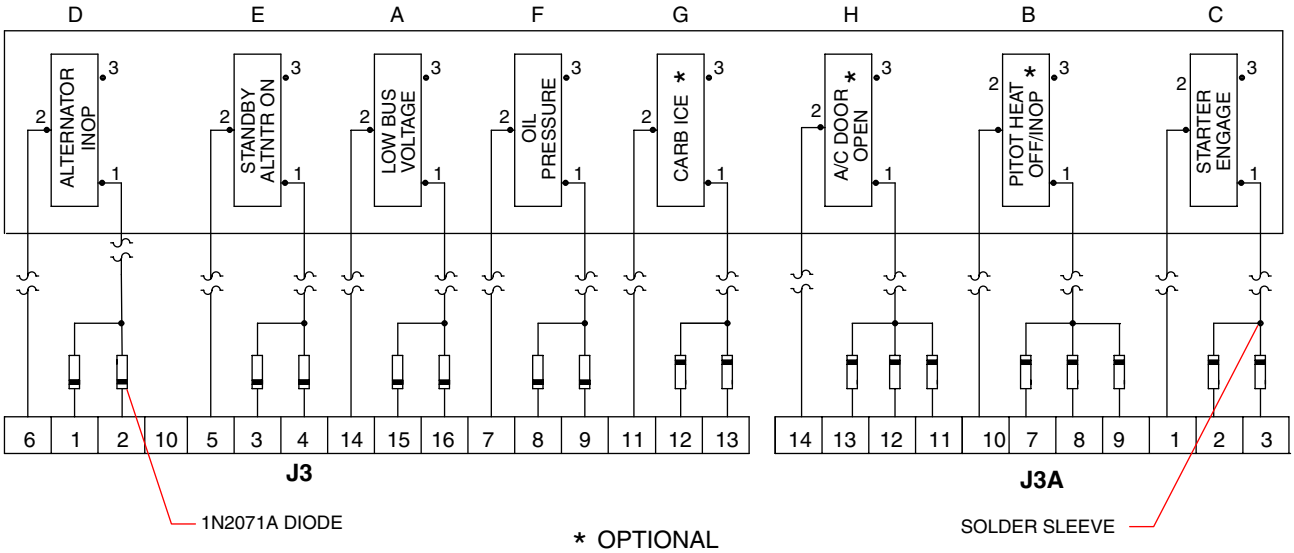
**LIGHT POSITIONS
(FRONT VIEW)**



ANNUNCIATOR LIGHT ASSEMBLY



ANNUNCIATOR LIGHT ASSEMBLY (W/ STANDBY ALTERNATOR)



Annunciator Panel
Figure 1 (Sheet 2 of 2)

Effectivity
with Avidyne Entegra

Effectivity
Standard, Avidyne Entegra, or
Garmin G500 equipped

B. Troubleshooting

See “Chart 1”.

CHART 1 (Sheet 1 of 3)
TROUBLESHOOTING ANNUNCIATOR

Trouble	Cause	Remedy
All warning lights fail to illuminate.	5 amp ANNUNCIATOR PANEL circuit breaker open.	Reset (push IN) circuit breaker.
	No current from bus.	Check all wire segments, connections, and receptacle at the left side of annunciator panel.
Test switch fails to activate switch, warning lights.	Bad switch or connections.	Check wires. Replace if necessary.
All warning lights do not go out after engine is running.	Test switch shorted or bad connections.	Check wire connections and replace switch if necessary.
Lights fail to dim when DAY-NIGHT switch placed in NIGHT position.	Defective K-3 relay in annunciator box assembly.	Replace relay.
	Open wires from day/night switch to dimmer box.	Check wire connections.
LOW BUS VOLTAGE fails to illuminate.	Blown bulb(s).	Replace bulb(s)
	Blown in-line fuse between bus bar and Low Voltage Monitor.	Replace fuse.
	Defective low voltage monitor in ammeter dimming control assembly.	Replace low voltage monitor.
LOW BUS VOLTAGE fails to dim with switch in NIGHT position.	Open wire between between bus bar and low voltage monitor; between low voltage monitor and annunciator box assembly; between annunciator box assembly and annunciator light.	Locate and repair open wire.
	Defective K-3 relay.	Replace relay.
PITOT HEAT OFF/INOP fails to illuminate.	Blown bulb(s).	Replace bulb(s).
	If pitot heat switch is OFF, faulty switch.	Replace switch.
	Faulty pitot sensor assembly.	Replace sensor assembly.
PITOT HEAT OFF/INOP fails to dim with switch in NIGHT position.	Open R-4 resistor in annunciator box assembly.	Replace resistor.
	Defective K-2 relay in annunciator box assembly.	Replace relay.

CHART 1 (Sheet 2 of 3)
TROUBLESHOOTING ANNUNCIATOR

Trouble	Cause	Remedy
STARTER ENGAGE fails to illuminate.	Blown bulb(s).	Replace bulb(s).
	Blown 5 amp in line fuse between starter contactor and annunciator panel.	Replace fuse.
STARTER ENGAGE fails to dim with switch in NIGHT position.	Open R-8 resistor in annunciator box assembly.	Replace resistor.
	Defective K-4 relay in annunciator box assembly.	Replace relay.
VACUUM INOP fails to illuminate.	Blown bulb(s).	Replace bulb(s).
	No voltage to sensor.	Check all wire segments and connections.
	Sensor activates at too low of a setting.	Replace. See Chapter 37.
	Defective sensor.	Replace. See Chapter 37.
VACUUM INOP light does not extinguish.	Sensor activates at too high of a setting.	Replace. See Chapter 37.
	Sensor terminals bridged.	Remove material between terminals.
	Defective sensor.	Replace. See Chapter 37.
VACUUM INOP fails to dim with switch in NIGHT position.	Open R-6 resistor in annunciator box assembly.	Replace resistor.
	Defective K-3 relay in annunciator box assembly.	Replace relay.
ALTERNATOR INOP fails to illuminate.	Blown bulb(s).	Replace bulb(s).
ALTERNATOR INOP fails to extinguish.	Blown 1/4-amp fuse between alternator and alternator out switch.	Replace fuse.
	Defective ALTERNATOR OUT switch.	Replace switch.
	Open wiring or connector between ALTERNATOR OUT switch an annunciator panel.	Check wires W2A & W2B and connectors R2-6 & P2-6 for open or bad connection. Repair as necessary.

CHART 1 (Sheet 3 of 3)
TROUBLESHOOTING ANNUNCIATOR

Trouble	Cause	Remedy
ALTERNATOR INOP fails to dim with switch in NIGHT position.	Open R-10 resistor in annunciator box assembly.	Replace resistor.
	Defective K-5 relay in annunciator box assembly.	Replace relay.
OIL PRESSURE fails to illuminate.	Blown bulb(s).	Replace bulb(s).
	No current to sensor.	Check all wire segments and connections.
	Sensor activates at too low of a setting.	Replace sensor. See Chapter 79.
OIL PRESSURE fails to extinguish.	Sensor activates at too high of a setting.	Replace. See Chapter 79.
	Sensor terminals bridged.	Remove material between terminals.
	Defective sensor.	Replace. See Chapter 79.
OIL PRESSURE fails to dim with switch in NIGHT position.	Open R-5 resistor in annunciator box assembly.	Replace resistor.
	Defective K-3 relay in annunciator box assembly.	Replace relay.

2. Annunciation in Garmin G1000 / G1000 NXi-equipped Airplanes

NOTE: Information provided herein is general in nature to provide an introduction to the Crew Alerting System. See the appropriate Pilot's Guide for your airplane for detailed descriptions of the annunciator system (CAS and Non-CAS) and all warnings, cautions and advisories.

A. Descriptiion

Annunciation is integrated into the Garmin G1000 / G1000 NXi systems. The Crew Alerting System (CAS) consists of Master Warning and Master Caution Indicators operating in conjunction with CAS text messages. CAS text messages appear in the lower left area of the MFD during normal operations and in the right side area of the PFD during reversionary mode operation. The Master Warning and Master Caution Indicators are illuminated push-button switches centered above the pilot's and copilot's PFDs in G1000 installations. In G1000 NXi installations, the Master Warning / Master Caution Indicators are softkeys in the CAS display. The severity of CAS messages are categorized as Warning, Caution and Advisory as follows: (See "Chart 2" on page 31508 for examples.)

(1) Warning (**red on black**) – Immediate crew awareness and action required; Master Warning triggered Warnings are accompanied by a continuous aural chime.

(2) Caution (**amber on black**) – Immediate crew awareness and possible future corrective action required; Master Caution triggered Cautions are usually accompanied by a single aural chime of approximately 1.5 seconds in duration.

(3) Advisory (**white**) – Advisory information; and are not accompanied by aural chimes. no immediate action required

B. Aural Alerts

Aural alerts are provided to alert the crew and call for their attention:

- Master Advisory (single chime) / Master Caution (double chime) / Master Warning (triple chime).
- Stall Warning (steady tone).
- When in a V_{ne} overspeed condition ("Airspeed, Airspeed").
- Terrain cautions/warnings.
- Traffic Alerts.

CHART 2
CREW ALERTING SYSTEM

— WARNINGS - RED - TRIPLE AURAL CHIME —	
Message	**Cause**
ALTR FAIL	Left ALTR switch selected ON and the alternator has failed.
L FUEL QTY	Left tank fuel quantity is less than or equal to 3.0 gallons; and/or total fuel quantity is less than or equal to 6.0 gallons.
R FUEL QTY	Right tank fuel quantity is less than or equal to 3.0 gallons; and/or total fuel quantity is less than or equal to 6.0 gallons.
START ENGD	The L starter contactor is engaged for greater than 30 seconds during engine start or anytime while the engine is running.
— CAUTIONS - AMBER - DOUBLE AURAL CHIME —	
Message	**Cause**
AC DOOR OPEN (Opt.)	Air conditioning condenser door is open.
L FUEL QTY	Left tank fuel quantity is less than or equal to 5.0 gallons; and/or total fuel quantity is less than or equal to 10.0 gallons.
R FUEL QTY	Right tank fuel quantity is less than or equal to 5.0 gallons; and/or total fuel quantity is less than or equal to 10.0 gallons.
PITOT HT FAIL	Pitot heat has failed.
PITOT HT OFF	Pitot heat is turned OFF. No aural chime accompanies this action.
— ADVISORIES - WHITE - SINGLE AURAL CHIME —	
Message	**Cause**
AC DOOR OPEN (Opt.)	Air conditioning condenser door is open.
EMERG BATT ON	Emergency power in use.
FUEL IMBAL	Fuel quantity imbalance greater than 10 gallons while the engine is OFF.
MAINT MODE ON	System is in maintenance mode for ground tests.
PFD FAN FAIL	PFD cooling fan has failed.

CHAPTER

32

LANDING GEAR

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CHAPTER 32

LIST OF EFFECTIVE PAGES

CHAPTER SECTION	PAGE	DATE
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	2	Mar 6/14
32-Table of Contents	1	Mar 15/20
	2	Jul 30/21
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32-10-00	1	Mar 6/14
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	5	Mar 6/14
	6	Mar 6/14
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	8	Mar 6/14
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	4	Mar 15/20
	5	Sep 1/16
	6	Sep 1/16
	7	Mar 6/14
	8	Mar 6/14
	9	Mar 6/14
	10	Mar 6/14
32-40-00	1	May 15/19
	2	May 15/19
	3	Jul 30/21
	4	May 15/19
	5	Jul 30/21
	6	May 31/20
	7	May 31/20
	8	May 31/20
	9	May 31/20
	10	Mar 6/14
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	12	Mar 6/14
	13	Mar 6/14
	14	Mar 6/14
	15	Mar 6/14
	16	Mar 6/14

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Disassembly		1
Assembly		3
Main Landing Gear		3
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Installation		4
Main Gear Strut and Wheel Fairings		6
Removal		6
Installation		6
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GENERAL

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

In this chapter are instructions for the removal, disassembly, inspection, overhaul, and installation of the landing gear, nose gear alignment, and the repair and service of the brake system and its components.

1. Description

The landing gear is a fixed tricycle type, fitted with three 6.00 x 6 wheels. The landing gear struts are the air-oil type. The nose gear, steerable through a wide arc, allows a short turning radius in each direction. A spring device is attached to the rudder pedal torque tube assembly to provide rudder trim and to aid in nose wheel and rudder centering. Bungee springs on the push rods make lighter and smoother ground steering possible. A shimmy damper is incorporated in the nose wheel steering mechanism.

The two main wheels are equipped with a single disc hydraulic brake assembly connected to a cylinder actuated by a hand lever, or by individual cylinders attached to each rudder pedal. A brake fluid reservoir is installed on the left forward face of the engine firewall.

Wheel pants were standard equipment 1995-2012. In airplanes 2013 and up, they are optional.

2. Troubleshooting

See “Chart 1”.

NOTE: When troubleshooting the landing gear system, it may be necessary to place the airplane on jacks (See 7-10-00).

CHART 1 (Sheet 1 of 2)
TROUBLESHOOTING LANDING GEAR

Trouble	Cause	Remedy
Nose landing gear shimmies during fast taxi, takeoff, or landing.	Internal wear in shimmy dampener.	Replace shimmy dampener.
	Worn steering horn bolt holes.	Replace horn assembly.
	Excessive free play in steering bungees.	Rework or replace.
	Shimmy dampener or bracket loose at mounting.	Replace necessary part and bolts.
	Tire out of balance.	Check balance and replace tire if necessary.
	Worn or loose wheel bearings.	Replace or adjust wheel bearings.
	Worn torque link bolts or bushings.	Replace bolts or bushings.
	Improper nose wheel fairing.	Replace with proper fairing.

CHART 1 (Sheet 2 of 2)
TROUBLESHOOTING LANDING GEAR

Trouble	Cause	Remedy
Excessive or uneven wear on nose tire.	Incorrect operating pressure.	Inflate tire to correct
	Wear resulting from shimmy.	Refer to preceding Trouble column for correction.
Nose gear does not steer properly.	Oleo cylinder binding in strut housing.	Lubricate strut housing. (See Lubrication Chart.)
	Cylinder or strut housing bushings damaged.	Replace.
	One brake dragging.	Determine cause and correct.
	Steering bellcrank loose on attachment plate.	Adjust and tighten.
	Steering bellcrank bearing or bolt worn.	Replace bearing or bolt.
	Shimmy dampener galling or binding.	Replace.
	Steering bungee broken internal spring.	Replace.
	Binding in either rudder trim or steering bungee.	Grease.
Nose or main landing gear shimmies during fast taxi, takeoff, or landing.	Tire out of balance.	Check balance and replace tire if necessary.
	Loose torque links.	Adjust side play.
	Worn or loose wheel bearings.	Replace/adjust wheel bearings.
	Worn torque link bolts or bushings.	Replace bolts or bushings.
Excessive or uneven wear on main tires.	Incorrect operating pressure.	Inflate tire to correct pressure.
Strut bottoms on normal landing or taxing on rough ground.	Insufficient air or fluid in strut.	Service strut with air or fluid.
	Defective internal parts in strut.	Replace defective parts.

MAIN GEAR

1. Main Gear Oleo

NOTE: Main gear axle (17) and piston tube assembly (16) are removed from cylinder housing (7) with the gear either removed from or installed on the airplane. The gear must be removed from the airplane in order to remove metering component in the top of housing. (See removal of gear.)

A. Cleaning, Inspection and Repair

NOTE: Oleo repair is limited to smoothing out minor scratches, nicks and dents, and replacement of parts.

(1) Clean all parts with a suitable quick drying type cleaning solvent.

(2) Inspect landing gear oleo components for the following:

(a) Bearing surfaces of housing for excess wear, corrosion, scratches, and overall damage.

(b) Retaining ring for cracks, burrs, etc.

(c) Cylinder tube for corrosion, scratches, nicks, excessive wear, and misalignment.

(d) Air valve for operation and general condition.

(e) Orifice plate for hole restriction.

B. Disassembly

Refer to “Figure 1”.

(1) Place airplane on jacks. (See 7-10-00.)

(2) Place drip pan under main gear to catch spillage.

(3) Remove the gear axle and piston tube assembly as follows:

(a) Remove air from oleo chamber by depressing air valve core pin found in the inspection hole on top of wing. After pressure in oleo chamber has diminished, remove valve core pin and attach a small hose to air valve. Drain fluid by slowly compressing piston tube. To extract more fluid, remove filler plug, insert a siphon hose, and drain fluid from upper area of housing.

(b) Disconnect flexible brake line at elbow on brake assembly.

(c) Disconnect torque link assembly by removing any one of three cotter pins, nuts, washers, and bolts. (Note arrangement of components for re-installation.) Carefully slide piston tube from cylinder housing.

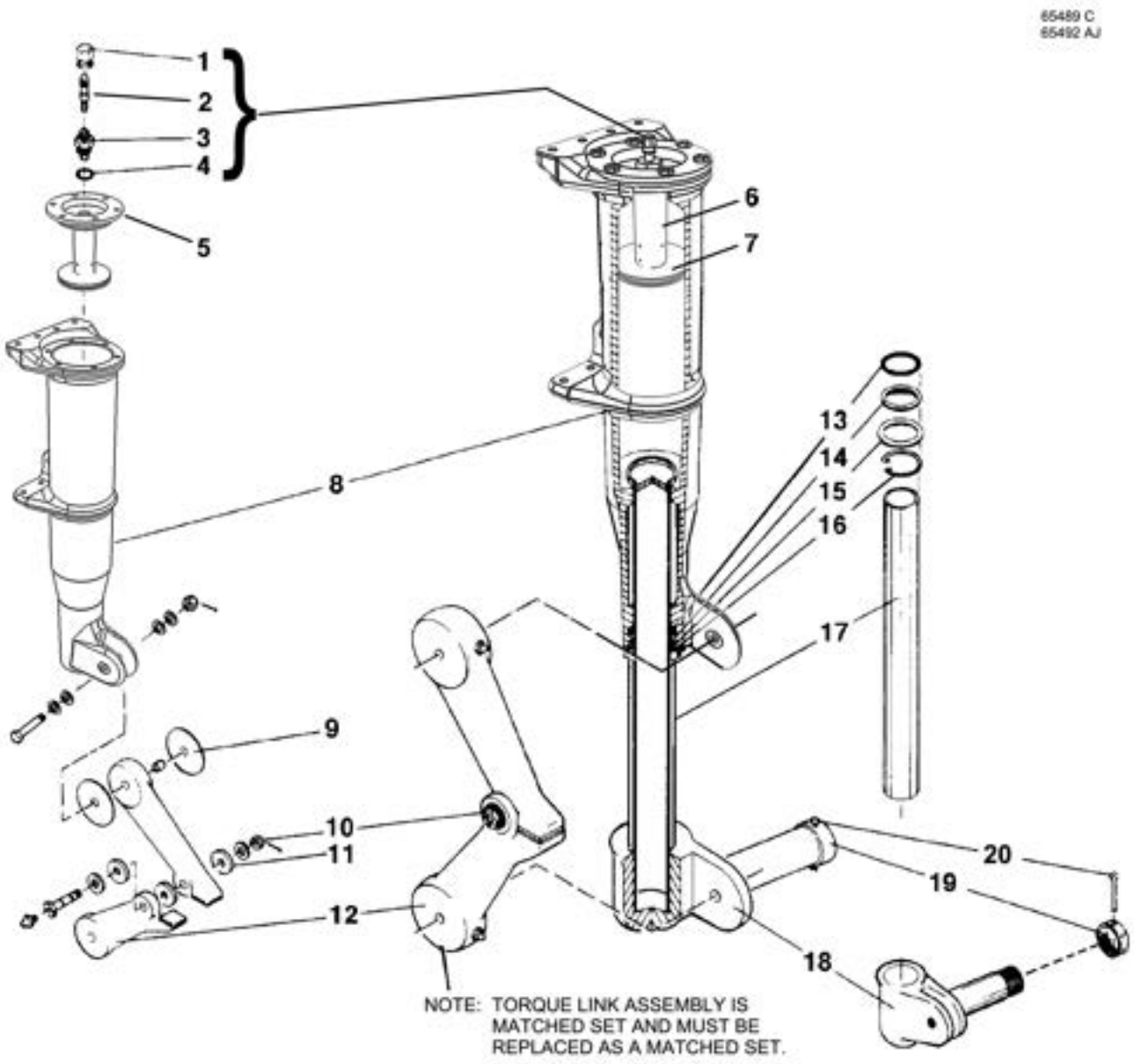
(d) Remove scraper ring, located inside lower end of cylinder housing, by first removing retainer ring (15), spacer ring (14), and then scraper ring (13).

(e) Remove O-ring seal (12), located just above the scraper ring, by using a curved wire or spoon shaped tool and inserting it under the ring.

(4) Remove cylinder head as follows:

(a) Cut safety wire and remove bolts (6 ea.) that secure cylinder head (4) in the top of housing. Remove cylinder head assembly from housing.

(b) Remove and discard O-ring. Remove all traces of sealant from around cylinder head and top of housing.



1. VALVE CAP
2. VALVE CORE
3. FILLER PLUG (i.e., Air Valve Body)
4. GASKET
5. HEAD ASSEMBLY
6. METERING TUBE
7. ORIFICE PLATE
8. STRUT HOUSING
9. WASHER
10. NUT
11. WASHER
12. TORQUE LINKS
13. QUAD RING
14. SCRAPER RING
15. SPACER RING
16. RETAINER RING
17. PISTON TUBE
18. AXLE STUB
19. WHEEL RETAINER NUT
20. COTTER PIN

Main Gear Oleo Strut
Figure 1

C. Assembly

Refer to “Figure 1”.

(1) Install cylinder head assembly as follows:

(a) Install new O-rings (3, 5) on cylinder head assembly. Lubricate O-rings with hydraulic fluid.

(b) Apply a thin layer of Loctite Gasket Eliminator No. 515 Sealant at the mating surface directly under flange of cylinder head (4).

(c) Press/push the cylinder head assembly into the cylinder.

(d) Position and secure cylinder head with bolts (6 ea.). Torque to 70 in.-lbs. and safety wire.

(2) Assemble piston tube components on the tube (16) by placing, in order, retainer ring (15), spacer ring (14), and scraper ring (13). Insert O-ring (12) into annular slot in bottom of housing.

CAUTION: DO NOT DAMAGE OR DISLOCATE O-RING IN HOUSING WHEN INSERTING PISTON INTO HOUSING.

(3) Lubricate piston wall and carefully insert piston into housing.

CAUTION: LINKS MUST BE LOOSE ENOUGH TO ALLOW FREE ACTION OF THE GEAR, BUT ALSO RESIST SIDE PLAY. OVER TIGHTENING WILL CAUSE BINDING OR DAMAGE TO LINKS.

(4) Verify bushings are installed in upper and lower torque links. Install links. At cable end of each link, install (with brake line hose bracket) bearing washers, bolt, washer, and nut. At connection point of upper and lower links, attach (with brake line hose brackets) spacer washers, grease bolt, washers, and nut. Install washers (NAS1149F0832P) under the head of bolt for firm sliding fit between the two links. Secure nuts with cotter pins.

(5) To eliminate side play of torque links, use 0.005 shim washers (Piper P/N 63311-03) as required. Maximum side play is 0.005/0.007. (Refer to “Figure 2”.)

(6) Slide scraper and spacer rings into place. Secure with retainer ring in the annular slot in bottom of housing.

(7) Install hydraulic brake line.

(8) If removed, install landing gear as described in Main Landing Gear - Installation, below.

(9) Service oleo strut as per Oleo Struts, 12-10-00.

(10) Actuate the gear several times by hand to be certain it operates freely.

(11) Remove drip pan and slowly lower airplane from jacks.

(12) Bleed brakes, if necessary. (See Bleeding Brakes, 32-40-00.)

2. Main Landing Gear

A. Cleaning, Inspection and Repair

NOTE: Landing gear repair is limited to reconditioning parts, parts replacement, smoothing out minor nicks and scratches, and repainting where chipped or peeled.

(1) Clean all parts with suitable dry type cleaning solvent.

(2) Inspect gear components for excessive wear, corrosion, and damage. Check cylinder housing and torque links for cracks, nicks, and misalignment.

(3) Use a 10-power glass to visually check cast gear cylinder top and bottom radii at drag link attach lugs for cracks. If no cracks found by visual inspection, use dye penetrant procedure to check for cracks. Refer to 20-10-00.

B. Removal

Refer to “Figure 2”.

(1) Place airplane on jacks. (See 7-10-00.)

(2) Place drip pan under main gear to catch spillage.

(3) If necessary, remove air from oleo chamber by depressing air valve core pin found in the inspection hole on top of wing. After pressure in the oleo chamber has diminished, remove valve core pin and attach a small hose to air valve. Drain fluid by slowly compressing piston tube. To extract more fluid from the chamber, remove filler plug, insert siphon hose, and drain fluid from upper area of housing.

(4) Remove fairing from around cylinder housing.

(5) Remove access plate located on bottom of wing, rear of housing, by removing attaching screws. Unhook hydraulic brake line inside wing assembly (accessible through the access plate). Cap line by using a threaded cap or wrapping with plastic.

(6) Remove top four bolts (hold bolts with slotted screwdriver and turn nut with wrench). Remove remaining six bolts with a wrench. Carefully remove gear assembly from wing.

C. Installation

Refer to “Figure 2”.

(1) Install main landing gear as follows:

(a) Position gear up in wing through access opening, secure with bolts, washers, and nuts.

(b) Connect brake line at point of disconnection.

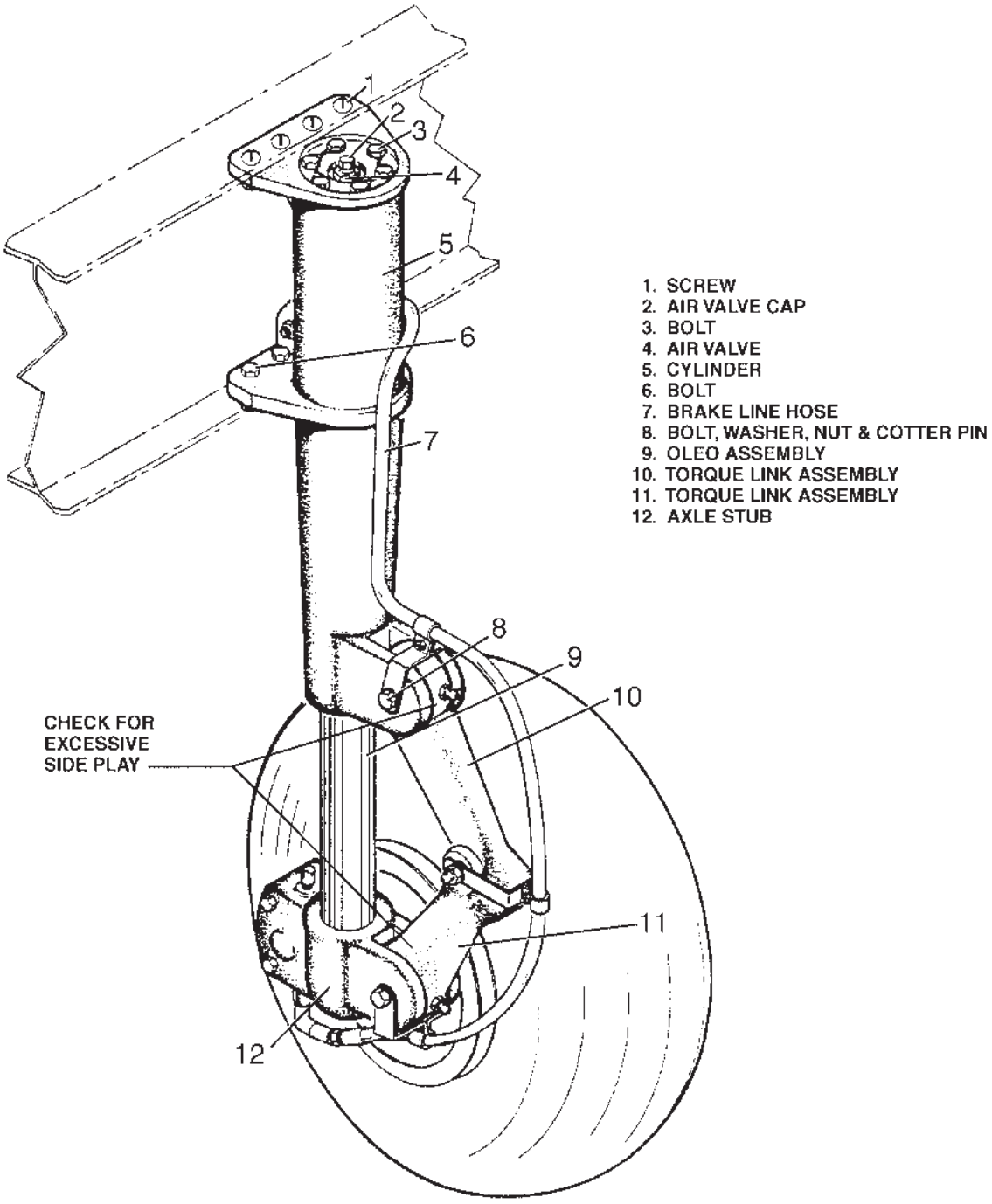
(2) Service oleo strut. (Refer to 12-10-00.)

(3) Service brake system. (Refer to 12-10-00.)

(4) Install access plate to bottom of wing.

(5) Install oleo housing fairing to gear.

(6) Remove drip pan from under gear and remove airplane from jacks.



Main Gear Installation
Figure 2

3. Main Gear Strut and Wheel Fairings

Refer to “Figure 3”.

A. Removal

(1) Wheel Fairing

(a) Remove the bolt and washer securing the wheel fairing to the wheel fairing support assembly.

(b) Remove the two screws on the top of the wheel fairing, outboard of the landing gear fairing.

(c) Loosen the eleven one-quarter turn fasteners (9) securing the wheel fairing shell to the wheel fairing panel and remove the shell.

(d) If removal of the wheel fairing panel is desired, remove the wheel fairing panel attachment bolt. Remove the panel.

(2) Strut Fairing

(a) Remove wheel fairing as described above.

(b) Loosen the eight (four inboard, four outboard) 1/4 turn fasteners (9) which secure the forward fairing half to the aft fairing half.

(c) Remove the six screws and washers securing the forward fairing half to the wing. Remove the forward fairing half.

(d) Remove the seven screws and washers securing the aft fairing half to the wing. Remove the aft fairing half.

B. Installation

(1) Strut Fairing

(a) Position the aft fairing half and secure to the wing using the seven screws and washers.

(b) Position the forward fairing half and secure to the wing using the six screws and washers.

(c) Secure the forward fairing half to the aft fairing half using the eight (four inboard, four outboard) 1/4 turn fasteners (9).

(d) Install wheel fairing as described below.

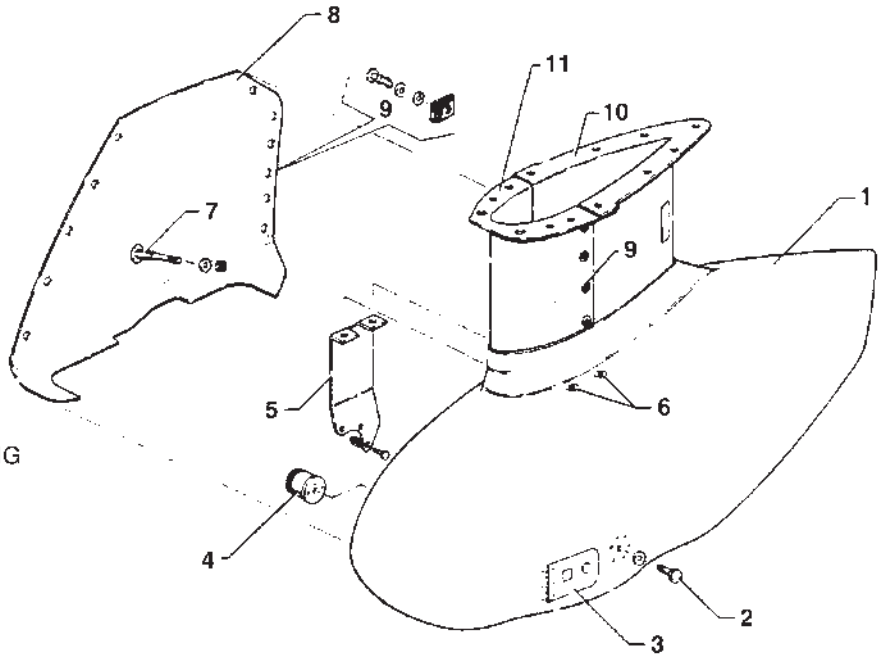
(2) Wheel Fairing

(a) Position the wheel fairing access panel and secure with the wheel fairing panel attachment bolt.

(b) Position the wheel fairing shell to the wheel fairing access panel and secure by tightening the eleven one-quarter turn fasteners (9).

(c) Install the two screws on the top of the wheel fairing, outboard of the landing gear fairing.

(d) Install the bolt and washer securing the wheel fairing to the wheel fairing support assembly.



1. SHELL, MAIN WHEEL FAIRING
2. BOLT AND WASHER
3. ACCESS DOOR
4. SUPPORT ASSEMBLY
5. BRACKET
6. SCREW AND WASHER
7. BOLT, WASHER AND NUT
8. ACCESS PANEL
9. 1/4 TURN FASTENERS
10. FAIRING, MAIN STRUT, AFT
11. FAIRING, MAIN STRUT, FORWARD

Main Gear Strut and Wheel Fairings
Figure 3

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NOSE GEAR

1. Nose Gear Oleo

Refer to “Figure 1”.

NOTE: Remove and disassemble the nose gear oleo strut assembly from strut housing with gear removed from or installed on airplane.

A. Disassembly

(1) Remove lower engine cowling. (Refer to 71-10-00.)

(2) Place airplane on jacks. (Refer to 7-10-00.)

(3) Place drip pan under nose gear to catch spillage.

(4) To remove air from strut, depress air valve core pin at top of strut assembly. After pressure in strut chamber has diminished, remove valve core pin, and attach a small hose to air valve. Drain fluid by slowly compressing piston tube. To extract more fluid from strut chamber, remove filler plug, insert siphon hose, and drain fluid from upper area of housing.

(5) To remove strut assembly from strut housing, cut safety wire at top of housing that secures steering horn attaching bolt to tube retainer nut. Remove steering horn attaching bolts to release steering horn from top of strut housing.

NOTE: It may be necessary to tap top of fork with a plastic mallet to get strut assembly out of housing.

(6) Loosen strut assembly retainer nut that secures strut assembly in strut housing. (At the same time, slide strut assembly out through bottom of strut housing.) Remove nut and washer from top of strut housing after assembly is removed.

(7) If necessary, remove top and bottom bearing from strut housing. (The bearings are compressed into place, and light tapping may be needed to free them.)

(8) Remove piston tube and fork assembly from cylinder as follows:

(a) Separate upper and lower torque links by removing cotter pin, connecting nut, washer, and bolt.

(b) Compress piston tube and fork assembly slightly and remove retainer ring from annular slot in bottom of cylinder tube.

(c) Remove piston tube and fork assembly by sliding out from the bottom of the cylinder tube.

(9) Remove bearing assembly from piston tube as follows:

(a) Release snap ring from top of piston tube.

(b) Slide bearing assembly off the end.

(c) If desired, carefully remove the wiper strip, back-up washer and quad ring or seal from inside of the bearing sleeve, and also the O-ring gasket from the outside of the bearing sleeve.

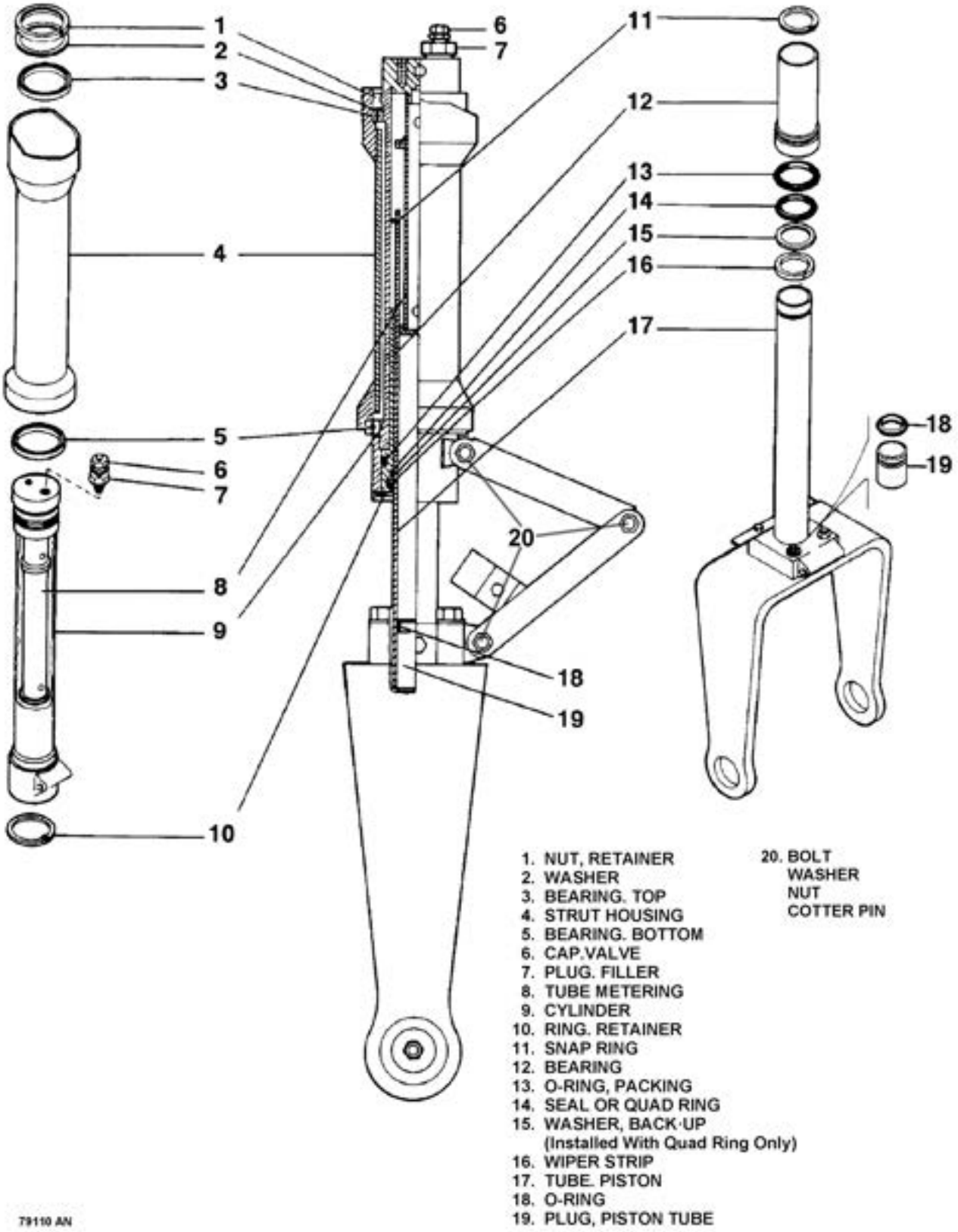
NOTE: Item (14) - Seal (P/N 485-315) replaced quad ring and backup washer in production airplanes cira 2003 and is authorized as service replacement in S/N’s 2843001 and up.

(10) Remove piston tube plug with O-ring in the lower end of the tube as follows:

(a) Remove nose wheel from fork. (Refer to removal and disassembly of nose wheel.)

(b) Loosen and remove bolt, washer and nut that extends through piston tube and block assembly.

(c) Push plug through top of piston tube. Use a rod inserted through tube bottom.



Nose Gear Oleo Strut
Figure 1

B. Assembly

Refer to “Figure 1”.

(1) Verify all parts are cleaned and inspected.

(2) Install piston tube plug as follows:

(a) Lubricate tube plug and O-ring with hydraulic fluid (MIL-H-5606) and install O-ring on plug.

(b) Lubricate inside wall of piston tube, and insert plug into top of tube, pushing it to the fork end.

(c) Align bolt holes of fork, tube, and plug; install the bolt, washer, and nut.

(3) Carefully install in the bearing assembly, the seal and the wiper strip as follows:

NOTE: Quad Ring (P/N 484-770) is no longer available (circa 2003). It and the Backup Washer (P/N 494-025) have both been replaced by a single Seal (P/N 485-315). The Seal is installed with the flared side (i.e., the O-ring and wiper) to the top of the strut.

(a) If not previously accomplished, in airplanes circa 2003 and earlier, remove the sharp edge on the top of the inner tube that will pass throught the seal during installation.

(b) Completely fill the snap ring groove on the inner tube with vaseline.

(c) The seal should be installed with the O-ring and wiper (the flared side) facing up towards the top of the strut (i.e., in the direction of the fuselage if strut is installed in airplane).

(d) Apply enough oil on the seal and liberally coat the upper end of the inner tube with vaseline to ensure slippery surfaces.

(e) Push inner tube into the bearing assembly quickly to prevent vaseline from being forced out of the snap ring groove.

(4) Slide the O-ring in place on the outside of the bearing assembly.

(5) Lubricate bearing assembly and carefully install on piston tube.

(6) Position snap ring on upper end of piston tube.

(7) Insert piston tube, with bearing assembly, in cylinder tube. Secure with retainer ring in annular slot at bottom of tube.

(8) Connect torque links on tube and fork, securing with bolt, washer, and nut. Tighten nuts only tight enough to retard side play, allowing links to rotate freely, align hole and insert cotter pin.

(9) Verify that upper and lower bearings are installed in strut housing. (Bearings are a press fit with grooves in the inner and outer races in the up position.)

(10) Position washer and strut assembly retainer nut on top of strut housing. Insert strut assembly up through washer until it contacts nut. Tighten nut to fit snug.

(11) Install steering horn assembly as follows:

(a) Insert hex bolt through side of horn and top of strut assembly. When it protrudes through the other side of steering horn, install washer and nut.

(b) Install steering horn attaching bolt through top of horn into strut assembly. Do not tighten bolt at this time. (If space appears between steering horn plate and top of strut assembly, install up to three spacer washer(s), (NAS1149F0416P), between horn and strut.)

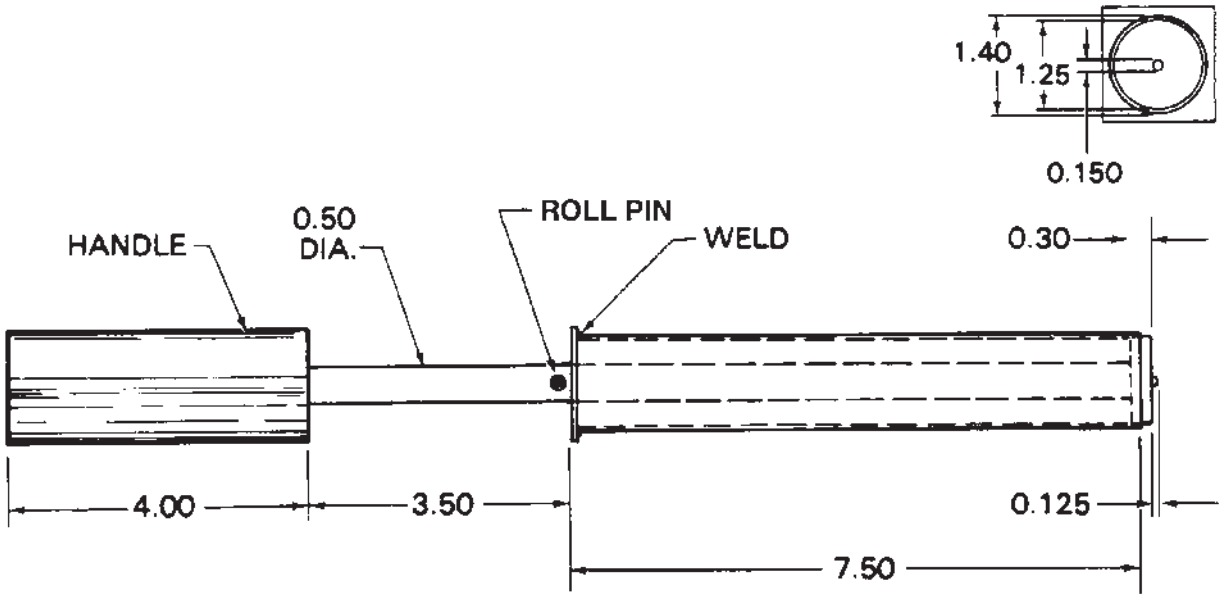
(c) Tighten bolt, and safety bolt to strut assembly retainer nut with MS20995C40 wire.

(12) Compress and extend strut several times to verify that strut operates freely. The weight of the gear wheel and fork should allow strut to extend.

(13) Service oleo strut with fluid and air. (Refer to 12-10-00.)

(14) Remove the drip pan from under nose wheel.

(15) Check gear for alignment. (Refer to alignment of nose landing gear.)



Retainer Ring Tool
Figure 2

C. Cleaning, Inspection and Repair

NOTE: Repair of oleo is limited to smoothing out minor scratches, nicks and dents, and parts replacement.

(1) Clean all parts with a suitable dry type cleaning solvent.
(2) Inspect landing gear oleo assembly component for the following:
  (a) Cylinder tube assembly for corrosion, scratches, nicks, and excessive wear.
  (b) Lock rings for cracks, burrs, and wear.
(3) Fork assembly for corrosion, scratches, nicks, and misalignment.
(4) Link assembly for elongated holes, cracks, corrosion, scratches, nicks, and straightness.
(5) General condition of air valve.

D. Nose Gear Oil Orifice Retainer Ring Installation

Refer to “Figure 1” on page 32102 through “Figure 3” on page 32206.

NOTE: A special tool can be made to simplify installation of the new retainer ring. (Refer to “Figure 2”.)

(1) With piston tube and fork removed from cylinder, verify all traces of the old retainer ring are removed from metering tube.
(2) Using the tool, position new retainer ring on end of tool with locating stud.
(3) Insert tool into cylinder with centering stud positioned into hole in base of metering tube.
(4) Hold tool tightly against metering tube and slide sleeve of tool towards metering tube. (This will move new retainer ring over end of metering tube and position itself into groove of metering tube.)

2. Nose Landing Gear

Refer to “Figure 3” on page 32206.

A. Removal

(1) Remove engine cowling. (Refer to 71-10-00.)
(2) Remove propeller. (Refer to 61-10-00.)
(3) Place airplane on jacks. (Refer to 7-10-00.)
(4) Remove engine. (Refer to 71-00-00.)
(5) Disconnect the two steering rod assemblies and centering springs at nose gear horn assembly by removing cotter pins, nuts, washers, and bolts.
(6) Disconnect oil lines, vacuum lines, fuel lines, hoses and wires secured to the mount with clamps and Koroseal lacing. Mark all wires and lines to aid installation.
(7) Remove nose gear and engine mount by removing the five bolts holding mount to firewall.

B. Installation

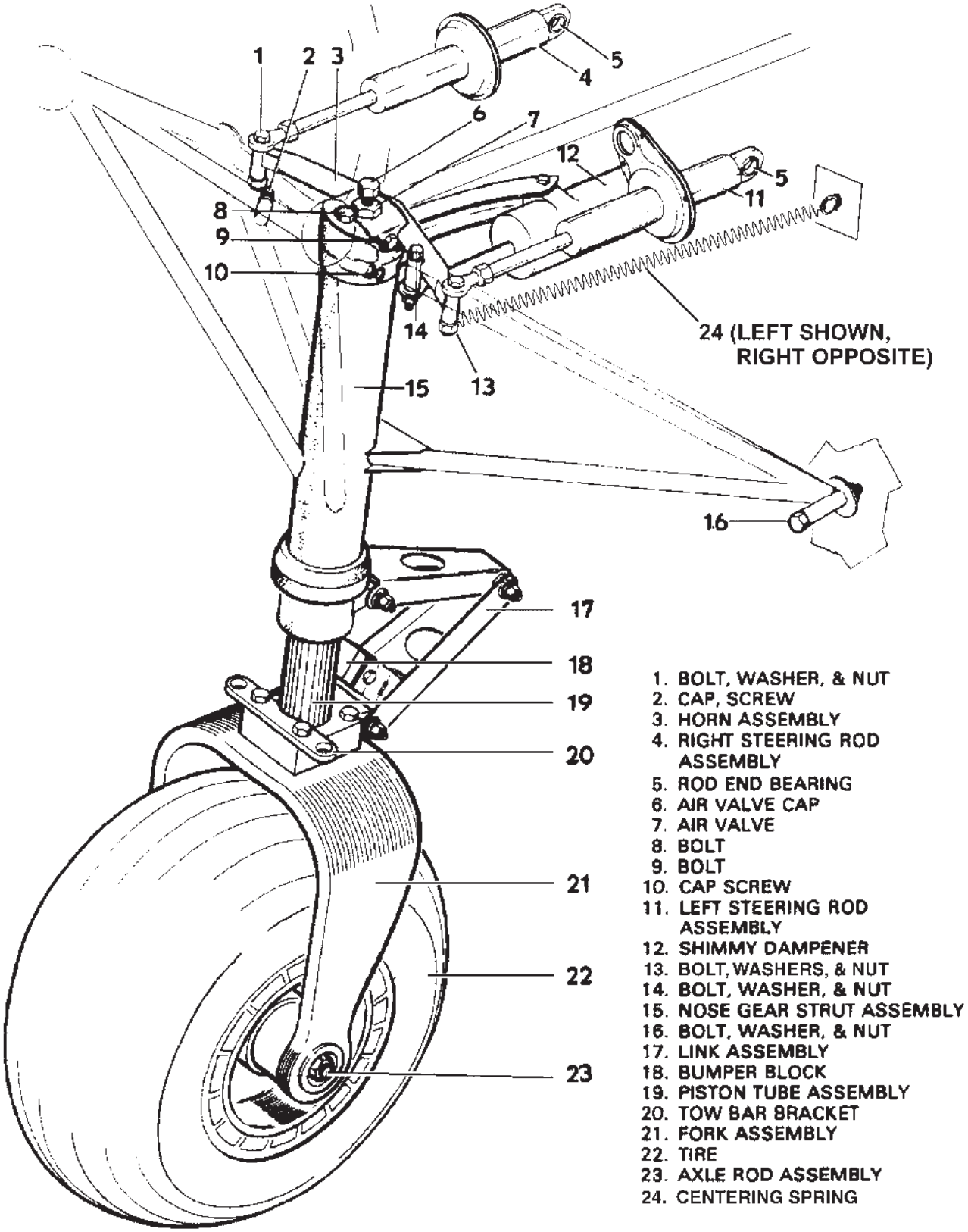
(1) Install nose gear and engine mount assembly to firewall with bolts, washers, and nuts. Tighten nuts to a torque of 240 to 270 inch-pounds.
(2) Attach two steering rod assemblies and centering springs to nose gear steering horn with bolts, washers, and nuts.
(3) Connect shimmy dampener to steering horn with bolts, washers, and nuts. (A spacer bushing and cotter pin are required at the body attachment point.)
(4) Install engine and connect controls. (Refer to 71-00-00.)
(5) Attach hoses, wires, and cables to engine mount tubing, secure with clamps and Koroseal lacing where required.
(6) Check nose gear alignment. (Refer to alignment of nose gear.)
(7) Remove airplane from jacks.
(8) Install propeller (refer to 61-10-00) and engine cowling (refer to 71-10-00).

C. Cleaning, Inspection and Repair

CAUTION: THE SHIMMY DAMPENER REQUIRES NO SERVICE OTHER THAN ROUTINE INSPECTION. IN CASE OF DAMAGE OR MALFUNCTION, DAMPENER MUST BE REPLACED. DO NOT REPAIR DAMPENER.

CAUTION: REPAIR TO LANDING GEAR IS LIMITED TO RECONDITIONING OF PARTS, SUCH AS REPLACING BEARINGS AND BUSHINGS, SMOOTHING OUT MINOR NICKS AND SCRATCHES, REPAINTING WHERE PAINT HAS CHIPPED OR PEELED, AND REPLACEMENT OF PARTS.

(1) Clean all parts with a suitable dry type cleaning solvent.
(2) Inspect nose gear assembly for the following:
  (a) Bolts, bearings, and bushings for excess wear, corrosion, and damage.
  (b) Strut housing and torque links for cracks, bends, or misalignment.
  (c) The shimmy damper (12) for condition and security. It requires no other service in normal use. Replace if damaged or malfunctioning.



Nose Gear Installation
Figure 3

D. Alignment

(1) Place airplane on a smooth, level floor.

(2) Place airplane on jacks. (Refer to 7-10-00.)

(3) Level airplane laterally and longitudinally. (Refer to 8-20-00.)

(4) From center of tail skid, extend a plumb bob and mark the contact point on the floor.

(5) Extend a chalk line from floor mark below tail skid, to a point approximately three feet forward of the nose wheel. Allow the line to pass under the wheel at the centerline of the tire. Snap the chalk line.

(6) Verify rudder is properly rigged and rudder cable tension is correct. (Refer to 27-20-00.)

(7) Clamp rudder pedals to align in a lateral position. (Refer to 27-20-00.)

(8) Verify nose wheel is in alignment with longitudinal axis of airplane or chalk line.

(9) Install steering rod assemblies between steering horn and rudder pedals without any load on rods. Adjust rod ends to obtain a no load condition. Connect rods to steering horn.

NOTE: Check rod ends have sufficient thread engagement. Use check hooks in the rods or verify a minimum 3/8 inch thread engagement.

(10) Verify rudder pedal stops are adjusted per instructions in 27-20-00.

(11) Check nose gear steering for maximum right and left travel as follows:

(a) Mark (on each side of the nose wheel) an angle line from the center line and wheel pivot point. (Refer to “Chart 1” for nose wheel turning angle.)

(b) Turn wheel to its maximum travel in both directions to check for allowable travel. If travel is too great in one direction and not enough in the other direction, check for possible damage to gear fork or torque links.

(12) Adjust horn stops to contact horn when nose wheel is turned as noted in “Chart 1”.

(13) Adjust shimmy dampener by turning nose wheel against stops and adjusting the rod end of dampener for adequate travel to both directions.

(14) Remove airplane from jacks.

CHART 1
NOSE GEAR ALIGNMENT TOLERANCES

Nose Wheel Travel	20 degrees left	± 1
	20 degrees right	± 1

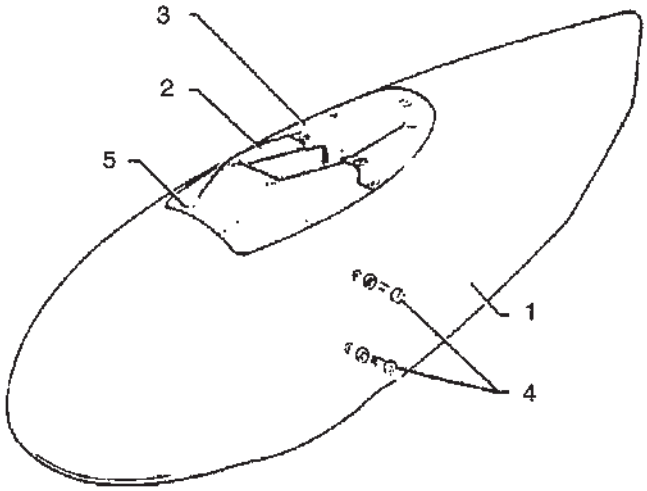
3. Nose Gear Wheel Fairing

A. Removal

(1) Remove the four bolts (two on each side) securing the nose gear fairing to the strut assembly.
(2) Remove the screws securing the forward and aft fairing cover assemblies to the nose gear fairing.
(3) Slide the fairing up on the strut assembly to allow access to the nose wheel assembly.
(4) Remove nose wheel as described in 32-40-00.
(5) Turn the fairing sideways and slide down and off of fork assembly.

B. Installation

(1) Turn the fairing sideways and slide up and on the fork assembly.
(2) Install nose wheel as described in 32-40-00.
(3) Slide the fairing down around the nose wheel assembly and position on the forward and aft fairing cover assemblies.
(4) Secure the nose gear fairing to forward and aft fairing cover assemblies with screws.
(5) Secure the nose gear fairing to the strut assembly using the four bolts (two on each side).



NOSE GEAR
FAIRING INSTALLATION

1. NOSE GEAR FAIRING
2. FAIRING COVER - FORWARD
3. FAIRING COVER - AFT
4. BOLT AND WASHER
5. SCREW AND WASHER

Nose Gear Strut and Wheel Fairings
Figure 4

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WHEELS AND BRAKES

1. Wheels

(PIR-PPS50025, Rev. AH.)

A. Main Wheel Assembly

See “Figure 1”.

(1) Removal and Disassembly

(a) Place airplane on jacks. (Refer to Jacking, 7-10-00.)

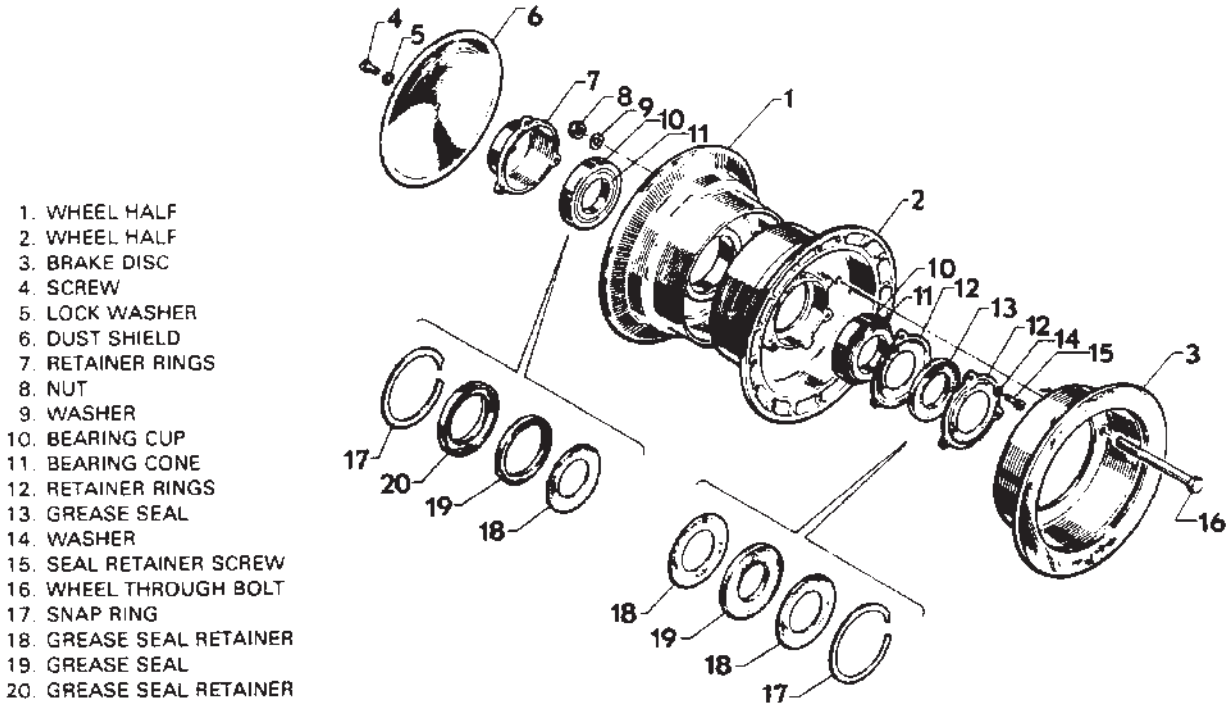
(b) If the wheel fairing is installed, remove the two fairing attachment bolts on each side of fairing and remove the small plate on top of fairing. Slide wheel fairing up gear fairing.

(c) To remove main wheel, remove the two cap bolts that join the brake cylinder housing and lining back plate assemblies. Remove back plate from between brake disc and wheel.

(d) Remove dust cover, cotter pin, and flat head pin that safeties the wheel nut, and remove the wheel nut. Slide wheel from axle.

(e) Separate the wheel halves by first deflating the tire. With tire sufficiently deflated, remove wheel through bolts. Pull wheel halves from tire by removing the inner half from tire first, then remove outer half.

(f) Remove wheel bearing assemblies from each wheel half by first removing retainer rings or snap rings holding the grease seal retainers, then the retainers, grease seals, and bearing cone. Bearing cups must be removed for replacement only. Remove by tapping them out evenly from the inside.



Main Wheel Assembly
Figure 1

(2) Assembly and Installation

Refer to “Figure 1” on page 32401.

CAUTION: DO NOT USE POWER TOOLS TO INSTALL NUTS AND BOLTS.

NOTE: Piper recommends that all tires on an airplane be from the same manufacturer and that the two main tires be the same part number.

NOTE: Tubes used must be from the same manufacturer as the tire. Do not mix tube and tire brands.

(a) Wheel Assembly

1) Check that bearing cup for each wheel half is properly installed.
2) Install tire with tube on wheel half with valve stem hole, inserting the stem through the valve hole. Align wheel bolt holes and then join two wheel halves.
3) Position brake disc in the inner wheel half and install wheel through-bolts, washers and nuts with the nuts on valve stem side, then tighten (draw up) the bolts in a criss-cross pattern to about 20 inch-pounds.

NOTE: Both plain and countersunk washers may be used. Some designs use bolts that have a radius between the head and shank, which requires the countersunk washers. Inspect bolts and washers prior to assembly. Properly oriented washers with countersunk surfaces to sit flush against the bolt head. Install washers and nuts on bolts (bolt heads are to be on brake disc side of wheel).

4) Remove all pressure in the tire.
5) Torque each nut again in a criss-cross pattern to about 45 inch-pounds before setting the final torque specified on the wheel placard. Do not use power tools to torque nuts.
6) Inflate the tire to the specified pressure per Chart 1, 6-00-00.

(b) Bearing Installation

CAUTION: DO NOT MIX AVIATION WHEEL BEARING GREASES WITH EACH OTHER.

NOTE: Mobil Aviation Grease SHC 100 has been used by the wheel manufacturer since March 2007 and is the preferred grease. If not available, greases meeting MIL-PRF-81322 or DOD-G-24508A are suitable alternatives. If old grease is unknown, thoroughly clean bearing components before packing with new grease.

CAUTION: HANDLE BEARING CONES WITH EXTREME CARE TO PREVENT CONTAMINATION OR DAMAGE.

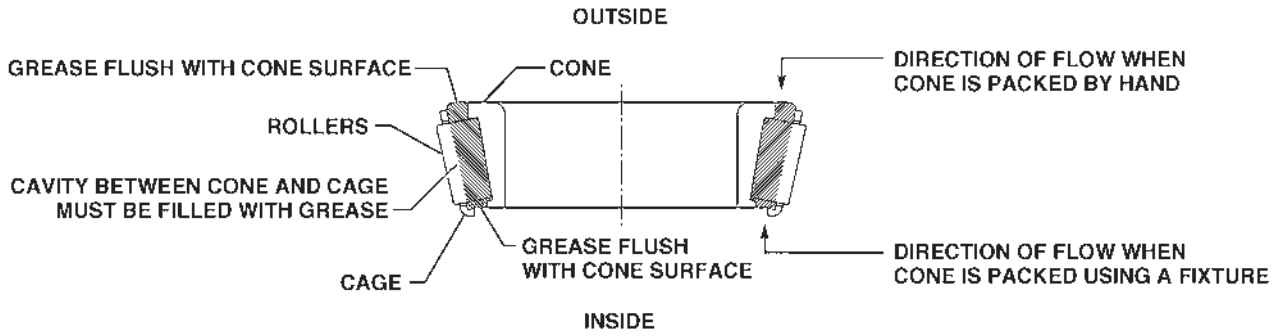
1) Pack the bearing cones as follows:

NOTE: Pack the bearing cones just before installation to prevent contamination.

NOTE: Bearing cones can be packed by hand or by using a mechanical bearing greaser. The mechanical bearing greaser will do a more thorough job of packing the grease.

a) Clean the bearing cones.
b) Push and force the grease up and out between the rollers, cone and cage.
c) The bearing is properly greased when no voids or daylight can be observed between the rollers and inner and outer races.
d) Disperse excess grease around each end and the tapered sides of each cone.

NOTE: Shaded area in “Figure 2” shows the recommended quantity of grease.



Wheel Bearing Packing
Figure 2

2) Liberally swab the bearing cup, bearing bore hub, and grease seal and snap ring areas with bearing grease.

3) If felt seals are used, lightly coat all surfaces of the felt with the wheel bearing grease. If rubber lip seals are used, lightly coat the rubber surfaces with bearing grease.

4) Install the bearing cones, grease seals (felts and rings or rubber lip seals) and snap rings.

   Excess grease will squeeze out. Remove the excess grease with an inward rotating movement against the bearing cone ID. Disperse any small amounts of grease on the exterior surface of the grease seal and snap ring and remove any grease from the hub outside surface.

5) Use care that bearing grease does not become contaminated.

(c) Wheel Installation

1) Install wheel assembly onto axle assembly.

2) Rotate the wheel/tire while torquing wheel nut to 150 to 200 inch-pounds to seat the bearing.

3) Back off wheel nut to zero torque.

4) Torque wheel nut to 30–40 inch·pounds while rotating wheel/tire.

5) Rotate wheel nut (clockwise or counterclockwise) to nearest slot and cotter pin hole.

   NOTE: Wheel must rotate freely without perceptible play.

6) Install cotter pin.

7) Recheck inflation pressure.

(d) Position brake lining back plates between wheel and brake disc and the brake cylinder on torque plate. Insert spacer blocks between back plates and cylinder, and install four bolts to secure assembly. If brake line was disconnected, connect the line and bleed brakes. (Refer to bleeding brakes.)

(3) Inspection

(a) Visually check all parts for cracks, distortion, defects, and excess wear.

(b) Check tie bolts for looseness or failure.

(c) Check internal diameter of felt grease seals. Replace felt grease seal if surface is hard or gritty.

(d) Check tire for cuts, internal bruises, and deterioration.

(e) Check bearing cones and cups for wear and pitting, and relubricate.

(f) Replace any wheel casting with visible cracks.

B. Nose Wheel Assembly

Refer to “Figure 3”.

(1) Removal and Disassembly

(a) Jack airplane so nose wheel is clear of the ground. (Refer to Jacking, 7-10-00.)

(b) If wheel fairing is installed, remove four bolts, two on each side, and the small plate on top held by metal screws. Slide fairing up gear until wheel is removed.

(c) To remove wheel:

1) Remove nut and washer from one end of axle rod and slide out rod and axle plugs.

CAUTION: DO NOT DAMAGE AXLE TUBE END IN ANY WAY. THIS WILL MAKE REMOVAL AND INSTALLATION EXTREMELY DIFFICULT.

2) Lightly tap axle tube out from the center of wheel assembly with an object of near equal diameter.

3) Remove spacer tubes and wheel assembly.

4) Slide down wheel fairing and remove by turning it sideways.

(d) Separate wheel halves by first deflating tire. With tire sufficiently deflated, remove wheel through bolts. Pull wheel halves from tire by removing wheel half opposite valve stem first and then the other half.

(e) Remove wheel bearing assemblies from each wheel half by removing three screws or snap ring holding the grease seal and seal retainers, and then the bearing cones. Remove bearing cup by tapping out evenly from the inside.

(2) Assembly and Installation

CAUTION: DO NOT USE POWER TOOLS TO INSTALL NUTS AND BOLTS.

NOTE: Piper recommends that all tires on an airplane be from the same manufacturer and that the two main tires be the same part number.

NOTE: Tubes used must be from the same manufacturer as the tire. Do not mix tube and tire brands.

(a) Check that bearing cup for each wheel half is properly installed.

(b) Install tire with tube on wheel half with valve stem hole, inserting the stem through the valve hole. Align wheel bolt holes and then join two wheel halves.

(c) Install wheel through-bolts, washers and nuts with the nuts on valve stem side, then tighten (draw up) the bolts in a criss-cross pattern to about 20 inch-pounds.

NOTE: Both plain and countersunk washers may be used. Some designs use bolts that have a radius between the head and shank, which requires the countersunk washers. Inspect bolts and washers prior to assembly. Properly oriented washers with countersunk surfaces to sit flush against the bolt head. Install washers and nuts on bolts.

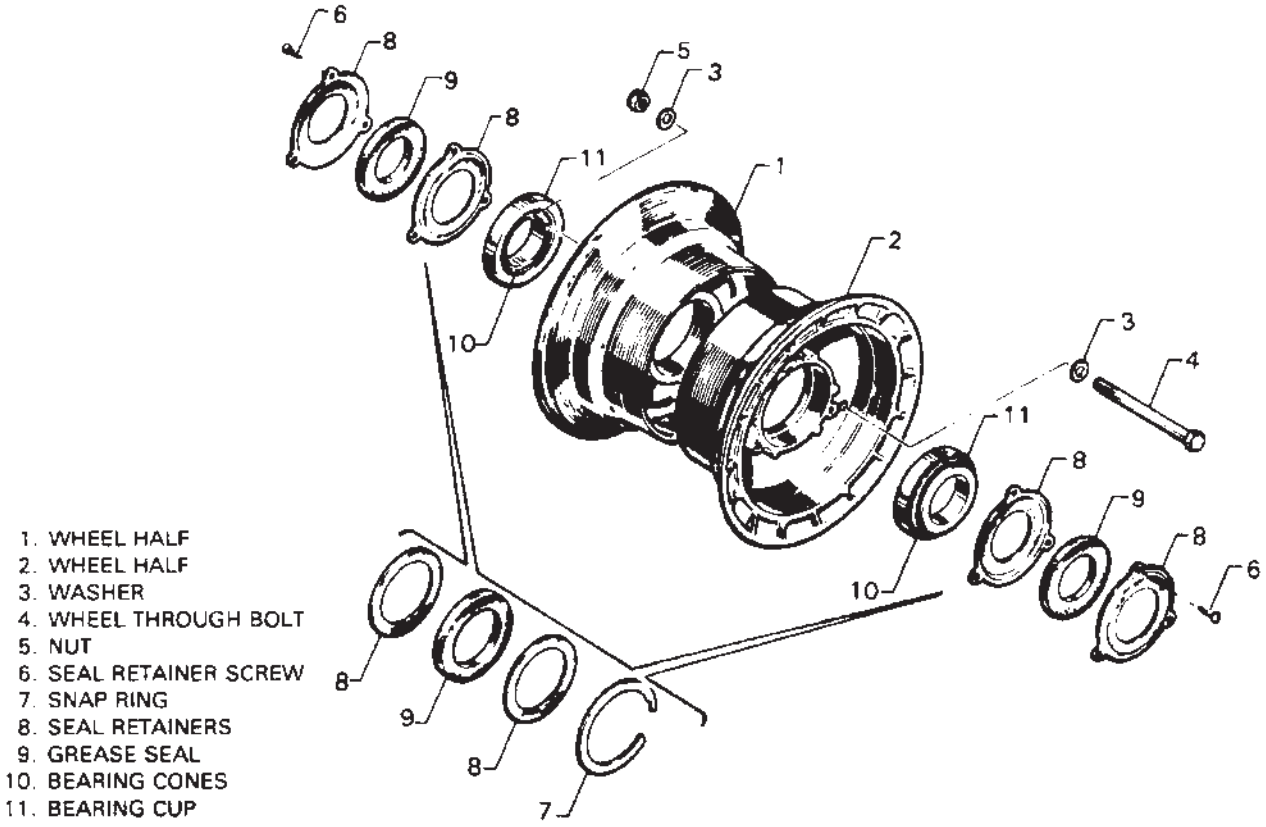
(d) Remove all pressure in the tire.

(e) Torque each nut again in a criss-cross pattern to about 45 inch-pounds before setting the final torque specified on the wheel placard. Do not use power tools to torque nuts.

On Cleveland wheel assemblies, final torque is per the placard on the wheel.

On McCauley wheel assemblies, final torque to 145 inch-pounds dry.

(f) Inflate the tire to the specified pressure per Chart 1, 6-00-00.



Nose Wheel Assembly
Figure 3

(g) Bearing Installation

Install bearings per “Bearing Installation” on page 32402.

(h) Wheel Installation

Install wheel per “Wheel Installation” on page 32403.

(i) Replace wheel fairing by turning sideways and slipping it up over fork assembly.

(j) Place spacer tubes one on each side of wheel and install unit in fork. Align and slide axle tube through spacer tubes and wheel assembly. Install axle plugs and rod with washer and nut. Tighten nuts until no side play is felt. (Allow wheel to rotate freely.)

(k) Turn fairing so it will fall into place. Install it with the four bolts and screws in the small plate.

(3) Inspection

(a) Visually check all parts for cracks, distortion, defects, and excess wear.

(b) Check tie bolts for looseness or failure.

(c) Check internal diameter of felt grease seals. Replace felt grease seal if surface is hard or gritty.

(d) Check tire for cuts, internal bruises, and deterioration.

(e) Check bearing cones and cups for wear and pitting, and relubricate.

(f) Replace any wheel casting having visible cracks.

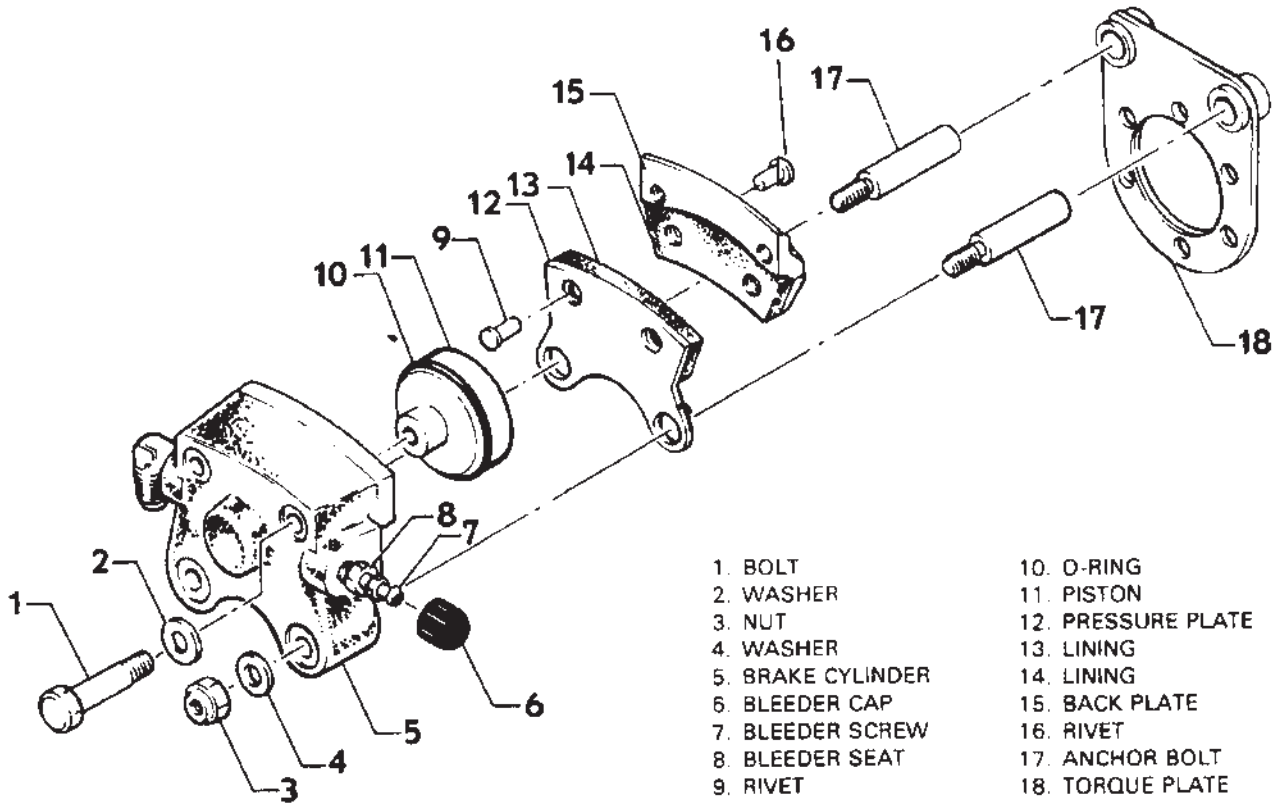
2. Brakes

A. Brake Adjustment and Lining Tolerance

The brake lining clearance is self-adjusting. Visually inspect the lining (it may be inspected visually while installed on the airplane). Linings are the riveted type and must be replaced if the thickness of any segment becomes worn below 0.099 inch or unevenly worn.

B. Wheel Brake Assembly

Refer to “Figure 4”.



Wheel Brake Assembly (Typical)
Figure 4

(1) Removal and Disassembly

(a) Disconnect brake line from brake cylinder at the tube fitting.

(b) Remove two cap bolts that join brake cylinder housing and lining back plate assembly. Remove back plate from between brake disc and wheel.

(c) Slide brake cylinder housing from torque plate.

(d) Remove pressure plate by sliding it off the anchor bolts of the housing.

(e) Remove piston by injecting low air pressure in cylinder fluid inlet and forcing piston from the housing.

(f) Check anchor bolt for wear.

(g) Remove anchor bolt as follows:

1) Position cylinder assembly on a holding fixture. (Refer to “Figure 5”.)

2) Use a suitable arbor press to remove anchor bolt from cylinder body.

(h) Install anchor bolt as follows:

1) Support anchor bolt in a holding fixture. (Refer to “Figure 5”, Step A.)

2) Align cylinder body over anchor bolt. (Refer to “Figure 5”, Step B.)

3) Use a suitable arbor press and apply pressure on the spot face directly over anchor bolt hole. (Refer to “Figure 5”, step C.)

(2) Assembly and Installation

(a) Lubricate piston O-ring with fluid MIL-PRF-5606 and install on piston. Slide piston in cylinder housing until flush with housing surface.

(b) Slide lining pressure plate onto anchor bolts of housing.

(c) Install cylinder assy in torque plate by sliding anchor bolts into bushings.

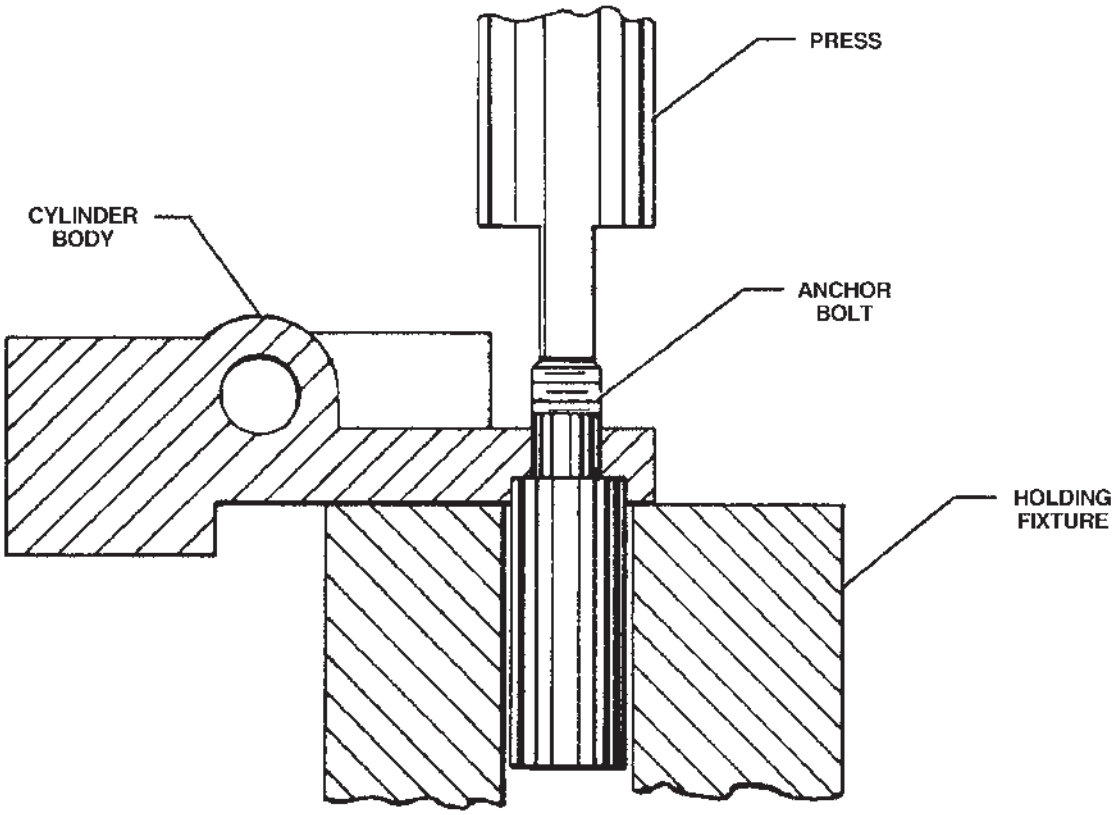
(d) Apply Sandstrom 28A Dry Film Lubricant (P/N 197-528) to anchor bolts and torque plate bushings. Ensure no other surfaces become contaminated with lubricant.

(e) Position lining back plate between wheel and brake disc. Install the two bolts to secure assembly. Dry torque 75–80 inch-pounds.

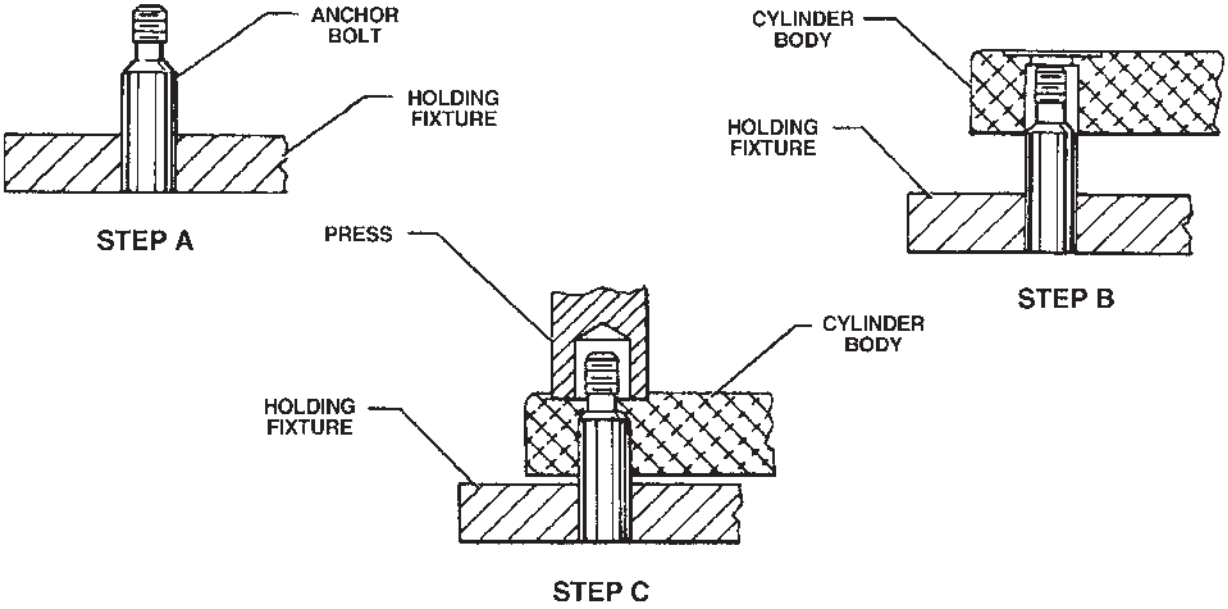
(f) Connect brake line to brake cylinder housing.

(g) Bleed brake system. (Refer to bleeding brakes.)

REMOVAL



INSTALLATION



Anchor Bolt
Figure 5

(3) Cleaning, Inspection and Repair

(a) Clean assembly with a suitable solvent and dry thoroughly.

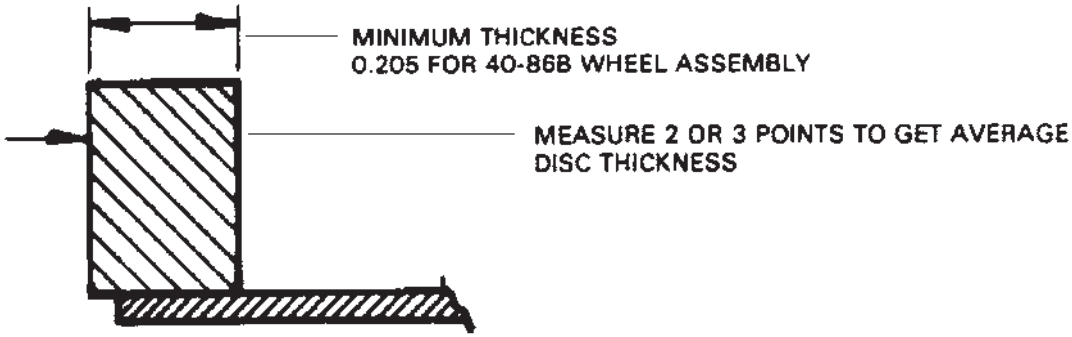
(b) Check cylinder housing wall and piston for scratches, burrs, corrosion, etc. that will damage O-rings.

(c) Check general condition of the brake bleeder screw and lines.

(d) Check brake disc for grooves, scratches, pits, or coning. Coning beyond 0.015 inch in either direction will require replacement. Minimum thickness of disc 164-20 used on wheel assembly 40-86B is 0.205 inch. (Refer to “Figure 6”.) A single groove or isolated grooves up to 0.031 inch deep will not require replacement. Grooving of the entire surface will reduce lining life and must be replaced. If a powdery rust appears on brake disc, one or two taxi-braking applications should clear it up. If rust is heavier, remove the disc for wire brushing, and sand with 220 grit sandpaper. To remove wheel disc, refer to removal and disassembly of main wheel.

(e) Remove lining from backing plates by drilling or punching out the old rivets, and installing a new set. Use the proper rivets and a rivet set that will properly stake the lining and form a correct flare of the rivet. (A rivet setting kit is available through Piper dealers as part number 754-165.)

(f) After new linings are installed, condition by performing a minimum of six light pedal effort braking applications from 25 to 40 mph, allowing brake discs to partially cool between stops.



Brake Disc Minimum Thickness
Figure 6

C. Hand/Parking Brake Master Cylinder

(1) Removal (Refer to “Figure 7”.)

(a) Disconnect inlet supply line from the fitting at the top of cylinder and drain fluid from reservoir and line into a suitable container.

(b) Disconnect pressure line from the fitting on cylinder and drain fluid from cylinder line.

(c) Disconnect the end of cylinder rod from brake handle by removing cotter pin that safeties the connecting clevis pin. Remove clevis pin and spacer washers.

(d) Disconnect base of cylinder from mounting bracket by removing attaching bolt assembly.

(e) Remove handle assembly by removing attaching bolt assembly holding handle to mounting bracket.

(2) Disassembly (See “Figure 8”.)

(a) Remove cylinder from mounting bracket.

(b) To disassemble cylinder, remove piston rod assembly by removing snap ring from the annular slot at the rod end of cylinder. Draw piston rod assembly from cylinder.

(c) Disassemble piston rod assembly by removing small snap ring securing the retainer bushing, spring, piston, seal, gland, and the large retainer spring.

(d) Remove O-rings from piston and gland.

(3) Assembly

NOTE: Use a small amount of hydraulic fluid (MIL-PRF-5606) on O-ring and component parts to prevent damage and for ease of handling during assembly.

(a) Install new O-ring on inside and outside of packing gland and on outside of piston. (Use a cone placed against the piston when installing Teflon O-ring on piston. The cone may be plastic or metal with dimensions per “Figure 8”.)

(b) To assemble piston rod assembly, install on rod, in order: roll pins, return spring retainer washer, retaining spring, packing gland with O-rings, seal, piston with O-ring, spring, and retainer bushing. Secure with snap ring on rod end.

(c) Insert piston rod assembly in housing and secure packing gland with snap ring.

(d) Install cylinder. (Refer to installation of brake master cylinder.)

(4) Installation (See “Figure 7”.)

(a) Install brake handle assembly between its mounting bracket and secure with bolt, washers, nut, and cotter pin. Washers must be placed on each side of handle, between the bracket, and under the nut.

(b) Place cylinder between mounting bracket and secure base end with bolt, washers, nut, and cotter pin. Place washers on each side of cylinder and under the nut.

(c) Connect rod end of cylinder to brake handle with a clevis pin and thin washers. Safety the clevis with a cotter pin.

(d) Connect pressure line to fitting at bottom of cylinder.

(e) Connect inlet supply line to the fitting at the top of cylinder and secure with spring clamp.

(f) Bleed brake system. (Refer to bleeding brakes.)

(5) Cleaning, Inspection and Repair

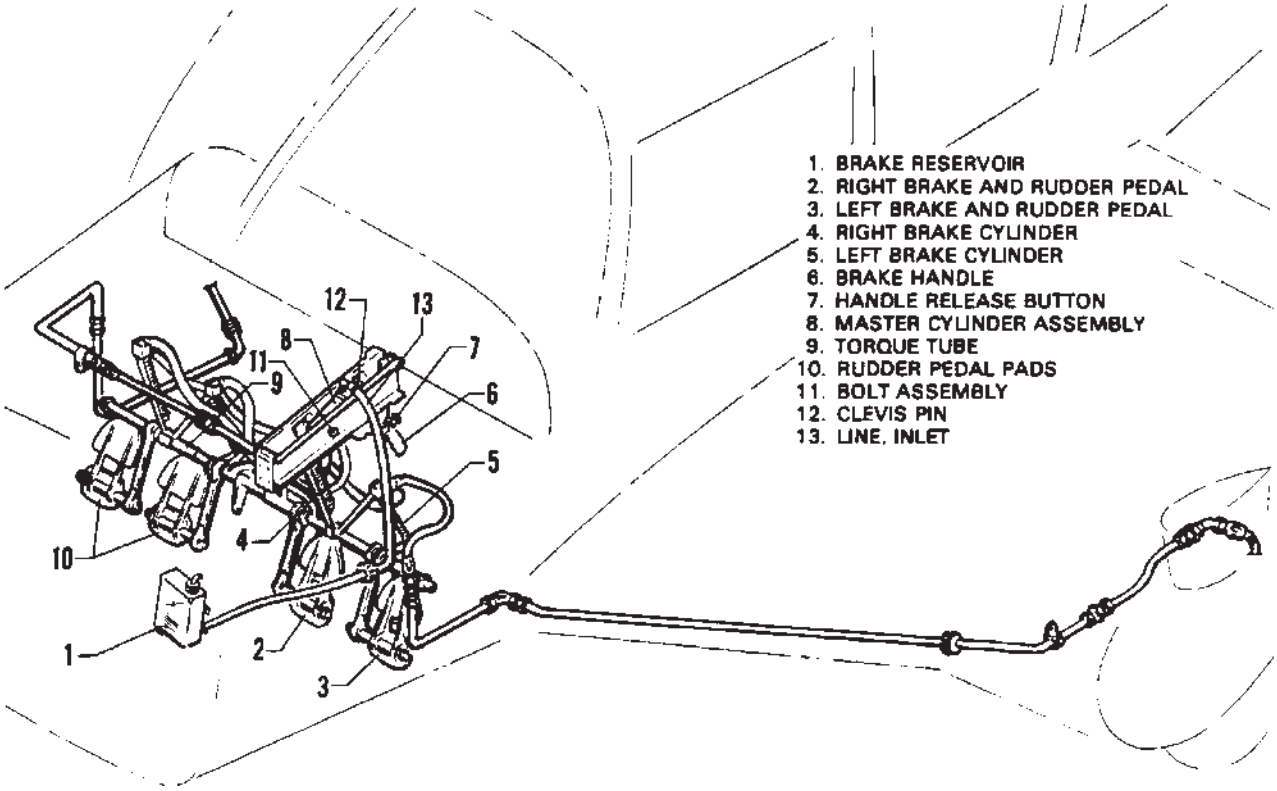
(a) Clean cylinder parts with a suitable solvent and dry thoroughly.

(b) Inspect interior walls of cylinder for scratches, burrs, corrosion, etc.

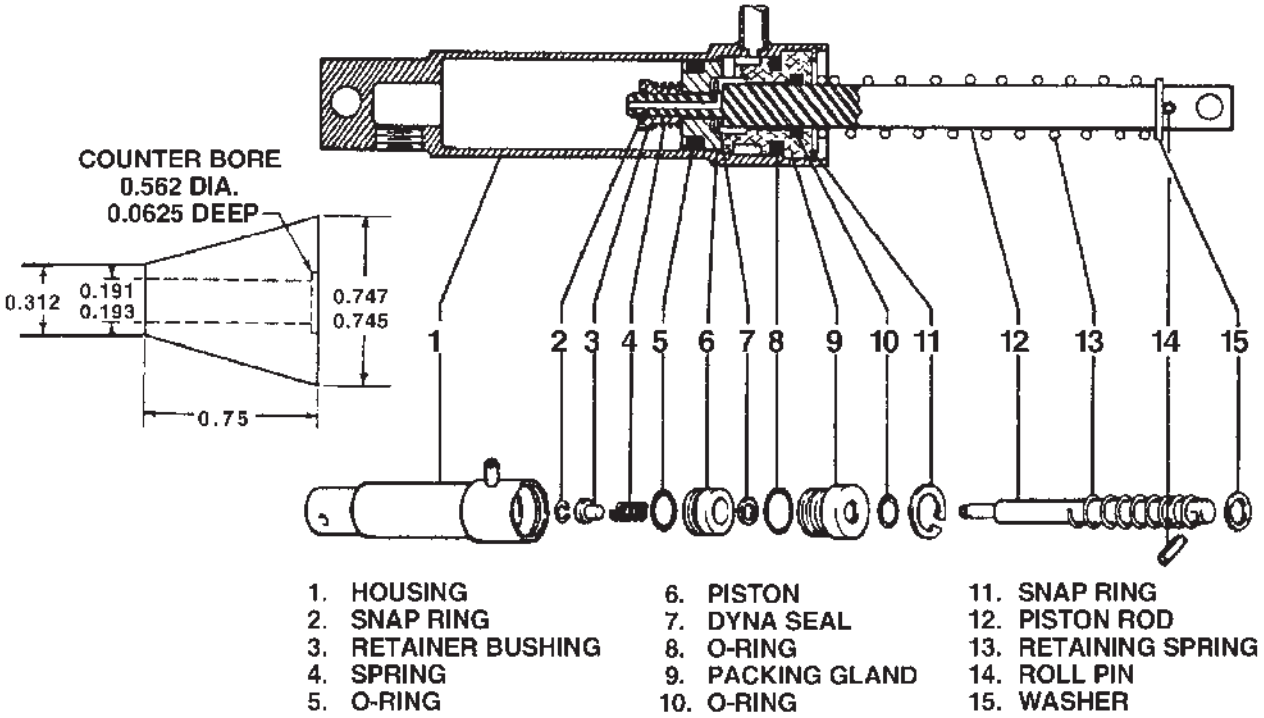
(c) Inspect the general condition of the fitting threads of cylinder.

(d) Check piston for scratches, burrs, corrosion, etc.

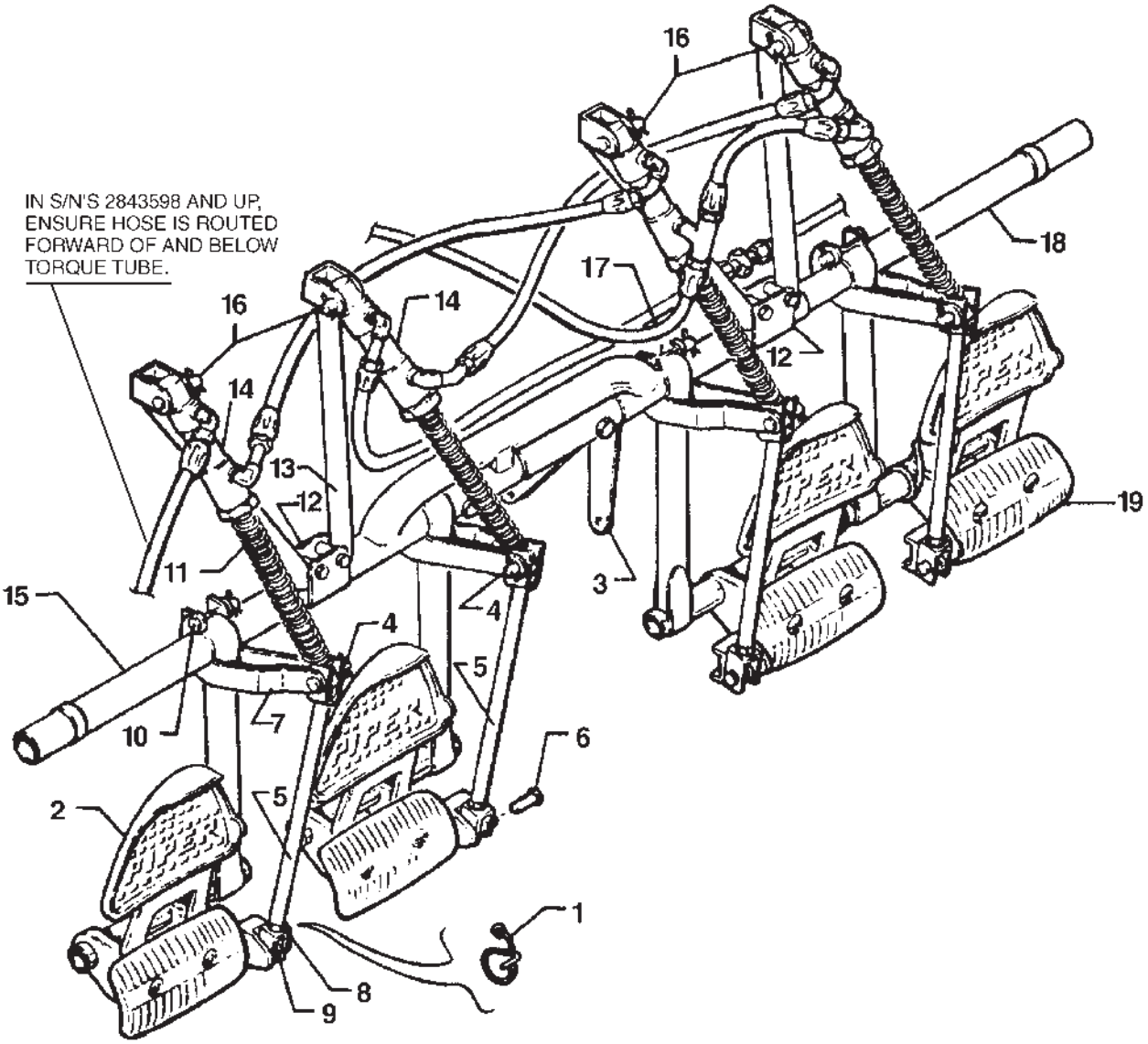
(e) Cylinder repairs are limited to polishing out small scratches, burrs, etc., and O-rings.



Brake System Installation
Figure 7



Hand / Parking Brake Master Cylinder
Figure 8



1. SPRING CLIP
2. TOE BRAKE PEDAL
3. TRIM CONTROL ATTACHMENT ARM
4. CLEVIS PIN, WASHER & COTTER PIN
5. CLEVIS ASSEMBLY
6. CLEVIS PIN
7. IDLER ARM
8. JAM NUT
9. CLEVIS PIN, WASHER & COTTER PIN
10. CLEVIS PIN, WASHER & COTTER PIN
11. RETURN SPRING
12. BRACKET
13. BRACE ASSEMBLY
14. HYDRAULIC CYLINDER ASSEMBLY
15. LEFT TUBE ASSEMBLY
16. CLEVIS PIN & COTTER PIN
17. FLEXIBLE HOSE ASSEMBLY
18. RIGHT TUBE ASSEMBLY
19. PEDAL PADS

Toe Brake Installation
Figure 9

D. Toe Brake Brake Cylinder(s)

(1) Removal (Refer to “Figure 9”.)

(a) Disconnect upper and lower lines from cylinder and cap lines to prevent fluid leakage, or drain fluid from brake reservoir and master cylinder.

(b) Remove cylinder from its attachment fittings by removing cotter pins that safety the cylinder attaching pins, and removing pins.

(2) Disassembly

Cleveland Cylinder Number 10-30. (Refer to “Figure 10”.)

(a) Remove cylinder from its mounting bracket. (Refer to removal of brake cylinder.)

(b) To disassemble cylinder, remove piston rod assembly by removing retaining ring from the annular slot in cylinder housing. Draw piston rod assembly from cylinder.

(c) Disassemble piston rod assembly by removing retaining ring, sleeve, spring, piston assembly, O-ring, gland, and return spring.

(d) Remove O-rings from piston and packing gland.

(3) Assembly

NOTE: Use a small amount of hydraulic fluid (MIL-PRF-5606) on O-ring and component parts to prevent damage, and for ease of handling during reassembly.

Cleveland Cylinder Number 10-30. (Refer to “Figure 10””.)

(a) Install new O-rings on inside and outside of packing gland on outside of piston.

(b) To assemble piston rod assembly, install on the rod, in order: roll pin, washer, spring, washer, packing gland with O-rings, seal, piston assembly with O-ring, spring, sleeve, and retaining ring.

(c) Insert piston rod assembly in cylinder and secure with retaining ring.

(d) Install cylinder. (Refer to installation of brake cylinder.)

(4) Installation (Refer to “Figure 9”.)

(a) Position cylinder at its mounting points and attach with clevis pins. Safety the pins with cotter pins.

(b) Connect brake lines to cylinder fittings.

(c) Bleed brakes.

(5) Cleaning, Inspection and Repair

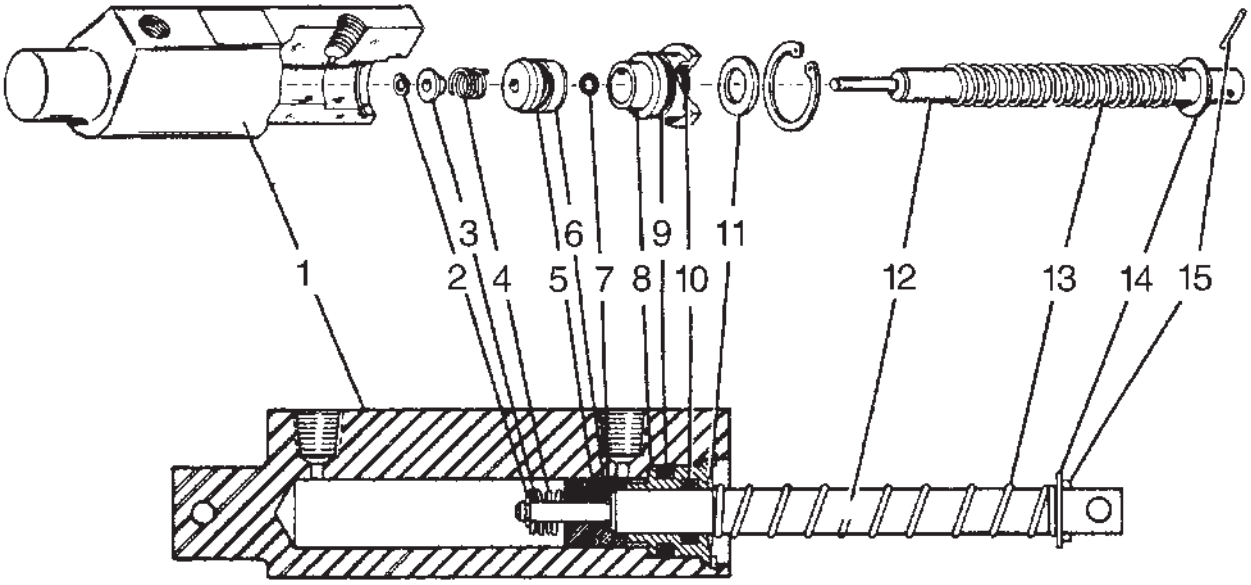
(a) Clean cylinder parts with a suitable solvent and dry thoroughly.

(b) Inspect interior walls of cylinder for scratches, burrs. corrosion, etc.

(c) Inspect general condition of the cylinder fitting threads.

(d) Check piston and valve for scratches, burrs, corrosion, etc.

(e) Repairs to cylinder are limited to polishing out small scratches, burrs, etc., and replacing valve, washer, seal, and O-rings.



1. HOUSING
2. RETAINING RING
3. SLEEVE
4. SPRING
5. O-RING
6. PISTON
7. O-RING
8. GLAND
9. O-RING
10. O-RING
11. WASHER WIPER
12. ROD
13. SPRING
14. WASHER
15. ROLL PIN

Toe Brake Cylinder
Figure 10

E. Bleeding Brakes

(1) Gravity Procedure

(a) Attach a clean, clear plastic tube to right landing gear brake bleeder. Extend the free end of the tube to a container partially filled with hydraulic fluid (MIL-PRF-5606). Check the end of tube is submerged in fluid. Open bleeder 1/2 to one (1) turn.

(b) Fill brake fluid reservoir, located on the firewall, with hydraulic fluid.

(c) Check right hand toe brake pedal(s) in cockpit have been pulled full aft.

(d) Pull hand brake handle and slowly pump master cylinder approximately 50 times or until hydraulic fluid is observed passing through plastic tube at brake bleeder.

NOTE: Fluid level in reservoir must be maintained to prevent air from entering system.

(e) Pump right brake cylinder very slowly approximately 12 times. This will purge air from toe brake cylinder system. Watch for any air forced through clear plastic tube during this operation. Verify air has been forced from toe brake system.

(f) Pump hand brake an additional 25 times or until no air is observed through clear plastic tube.

(g) Tighten brake bleeder and remove plastic tube.

(h) Repeat steps (a) through (f) for left main landing gear.

(2) Pressure Procedure

(a) Place a clean, clear plastic tube on vent fitting on top of brake fluid reservoir. Extend the free end of tube to a container partially filled with hydraulic fluid (MIL-PRF-5606). Verify tube end is submerged in fluid.

(b) Attach another clear plastic tube to right landing gear brake bleeder. Connect the free end of tube to pressure source. Open bleeder one to two (1 to 2) turns and pressure fill system with fluid.

(c) With fluid continually flowing through system, SLOWLY and simultaneously actuate hand brake and toe brake pedal, of side being bled, several times to purge air from cylinders. On dual brake installations, actuate both pedals for brake being bled.

NOTE: By watching the fluid pass through the plastic hose fluid reservoir and the bleeder fitting on gear being bled, check if any air is left in system. If air bubbles are evident, filling of the system must continue until all air is out of the system and a steady flow of fluid is obtained If brake handle remains spongy, disconnect the bottom of toe brake cylinders (Mar 6/14 to pedal) and rotate cylinder horizontally or above horizontal and use hand brake alone to purge air from system.

(d) Close the open bleeder fitting to which the pressure hose is attached. Do not remove tube from fluid reservoir until both brakes have been bled. Check brakes on the side being bled for proper pedal pressure. Place cap on bleeder fitting.

NOTE: Remove any trapped air in the top of wheel brake unit by applying pressure to the system with the brake hand lever and slowly opening bleeder and releasing hand lever.

(e) Repeat Steps (b) through (d) for left main landing gear.

(f) Drain excess fluid from reservoir to fluid level with syringe.

F. Brake System Leak Check

Pull for a good, firm hand brake and lock parking brake mechanism. Allow system to stand for approximately 10 minutes. Grip park brake handle; it must not be able to be pulled aft further than the original set. If handle can be pulled toward the panel and feels spongy, a leak is in the system. A leak may appear at any one of the connections throughout system or internally in master brake cylinder or wheel brake assemblies.

G. Bleeding Brake(s) After A Unit Has Been Changed

CAUTION: DO NOT ALLOW PRESSURE TO BLEED OFF BEFORE CLOSING BLEEDERS, AS THIS WILL ALLOW AIR IN SYSTEM. REPEAT PUMPING AND BLEEDING APPROXIMATELY 10 OR MORE TIMES OR UNTIL ALL AIR IS RELEASED FROM SYSTEM. DURING ALL BLEEDING, FLUID LEVEL OF RESERVOIR MUST BE MAINTAINED.

(1) Actuate hand brake handle until some pressure builds in system. Crack the attaching B nuts at any hose connection of replaced unit. Most of the spongy handle feeling will be displaced by this action.

(2) Actuate master cylinder and toe brake cylinder of the side unit that was changed, and bleed fluid through brake assembly on the wheel by pumping pressure and cracking bleeder until pressure drops.

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CHAPTER

33

LIGHTS

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CHAPTER 33

LIST OF EFFECTIVE PAGES

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CHAPTER 33 - LIGHTS

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CHAPTER 33 - LIGHTS

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FLIGHT COMPARTMENT

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Description and Operation

Electrical switches are located in an overhead switch panel. The pitot heat and radio master switches, along with other avionics switches, are located in the center instrument panel just below the radio stack. Air conditioning and fan switches are located on the lower right instrument panel. All switches are back lit rocker type.

Flight instruments are lit by post lights. All radios and navigation instruments are lit internally.

The switch lights, instrument post lights (or with Garmin G1000, LED panel light) and radio lighting are controlled by individual electronic dimmer control assemblies operated by adjustable potentiometers. The dimmer control assemblies are mounted on the aft side of the firewall to the right. The adjustable potentiometers are located on the lower left instrument panel below the pilot's control wheel.

The annunciator panel, located on the top right of the pilot's instrument panel, includes pitot heat off/inoperative, starter engaged, alternator inoperative, oil pressure, vacuum inoperative, low bus voltage and an optional air conditioning door open. A press-to-test switch is located to the right and a day-night toggle switch is located to the left of the annunciator panel.

A combined instrument panel flood/map light is located above the left and right pilot seats and are independently controlled by an adjustable potentiometer type control. Reading lights, each controlled by its own ON-OFF switch, are located above each rear seat. A courtesy light is installed above the main cabin entrance door. The courtesy light is wired directly to the battery so that it may be turned ON with the BATT MASTR switch OFF.

2. Instrument Panel Lights

Instrument post lights (or with Garmin G1000, LED panel light), switch lights, and avionics lights are controlled by individual electronic dimmer control assemblies, located to the right on the aft side of the firewall. They are operated by adjustable potentiometers located on the instrument panel below the pilot's control wheel. The instrument post and switch lights power supplies are protected by a 7.5 amp PANEL/SWITCH LIGHTS circuit breaker. The avionics power supply is protected by a 5 amp RADIO LIGHTS circuit breaker.

3. Dimmer Control Assembly

A. Removal

(1) Place BATT MASTR switch in OFF position and disengage (pull) BATTERY FEEDER circuit breaker.

(2) Disengage (pull) PANEL/SWITCH LIGHTS or RADIO LIGHTS circuit breaker, as appropriate.

(3) Gain access to appropriate dimmer control, mounted on the rear of the firewall, from under the right side instrument panel.

(4) Disconnect electrical plug from dimmer control.

(5) Remove machine screws holding dimmer control to the firewall.

(6) Remove dimmer control assembly.

B. Installation

(1) Position dimmer control in proper location on back of firewall.
(2) Install machine screws holding dimmer control to the firewall.
(3) Connect electrical plug to power supply.
(4) Engage (push in) PANEL/SWITCH LIGHTS or RADIO LIGHTS circuit breaker, as appropriate.
(5) Engage (push in) BATTERY FEEDER circuit breaker and place BATT MASTR switch in ON position.
(6) Check that lights associated with replaced dimmer control operate properly.
(7) Place BATT MASTR switch in OFF position.

4. Adjustable Potentiometer

A. Removal

(1) Place BATT MASTR switch in OFF position and disengage (pull) BATTERY FEEDER circuit breaker.
(2) Disengage (pull) PANEL/SWITCH LIGHTS or RADIO LIGHTS circuit breaker, as appropriate.
(3) Remove adjustable potentiometer control knob by loosening set screw with appropriate size allen wrench.
(4) Remove nut securing dimmer control potentiometer to instrument panel. Potentiometer should now hang down below bottom of panel.
(5) Mark wires attached to back of dimmer control potentiometer for proper location. Unsolder wires from back of dimmer control.
(6) Remove dimmer control assembly.

B. Installation

(1) Solder wires to proper contacts on back of new dimmer potentiometer.

NOTE: Ensure that the adjustable potentiometer is turned fully counterclockwise.

(2) Position potentiometer in proper location and insert from back of instrument panel. Check that key lugs on switch align with key holes in back of panel.
(3) Install nut securing potentiometer to face of instrument panel.
(4) Install control knob. Check that knob pointer is in appropriate position. Tighten allen screw.
(5) Engage (push in) PANEL/SWITCH LIGHTS or RADIO LIGHTS circuit breaker, as appropriate.
(6) Engage (push in) BATTERY FEEDER circuit breaker and place BATT MASTR switch in ON position.
(7) Check that lights associated with replaced potentiometer operate properly.
(8) Place BATT MASTR switch in OFF position.

5. Annunciator Panel

NOTE: Not installed with Garmin G1000/G1000 NXi. In these airplanes, the G1000/G1000 NXi system replaces the separate annunciator panel.

The annunciator panel is a small cluster of lights which warn of malfunctions in the various circuits or systems. See 31-50-00 for a functional description.

The DAY-NIGHT toggle switch, located to the left of the annunciator panel, is placed in the NIGHT position to dim the annunciator lights for night flight. This switch also activates the ammeter dimmer relay.

A. Troubleshooting

See 31-50-00.

B. Bulb Replacement

It is not necessary to remove the annunciator panel assembly to replace lamp bulbs. Simply “PUSH-IN” on the function light until it “clicks” and releases. The cover assembly will be partially ejected from the lamp base assembly. Pull the cover from the base and rotate to expose the lamp bulbs. Replace defective bulb(s) and reverse removal procedure. Select BATT MASTR switch ON; then depress “Test” switch to verify lamp is functioning.

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EXTERIOR

1. Description

Two landing lights are installed, one in each wingtip. These lights are also used for taxi operations. Both lights are controlled by a two position rocker switch (LDG LIGHT - OFF) located in the overhead switch panel. The circuit is protected by a 10 amp LANDING LIGHT circuit breaker located on the main circuit breaker panel. Incandescent landing lights are installed in S/N's 2843001–2843700 and 2843702. LED landing lights are installed in S/N's 2843701, 2843703 and up.

Navigation and position lights are installed in a single light assembly which also contains an anti-collision (i.e., strobe) light in each wingtip. The strobe lights are addressed separately below. The navigation / position lights consist of: a red light forward and a white light aft in the left wingtip; and, a green light forward and a white light aft in the right wingtip. The lights are controlled by a rocker type switch (NAV LIGHT) located in the overhead switch panel. The circuit is protected by a 5 amp NAV LIGHTS circuit breaker. An optional white tail light is also available which is mounted in the rudder tip fairing assembly. When installed, the tail light is powered from and controlled by the same NAV LIGHT switch as the navigation/position lights.

The exterior wingtip lighting assembly includes a pulsating anti-collision (i.e., strobe) white light. The strobe light is a Xenon flashtube with a separate power supply; or, an LED strobe light with no separate power supply. The strobe units are rated to flash approximately 50 times per minute. The anti-collision lights are controlled by an independent rocker switch (STROB LIGHT) in the overhead switch panel. The circuit is protected by a 5(1) amp ANTI-COLL LIGHTS circuit breaker.

An optional red fin tip ground reconigition/anti-collision (i.e., strobe) light is available in S/N's 2843115 and later. Again, the strobe light is a Xenon flashtube with a separate power supply; or, an LED strobe light with no separate power supply. When installed, the unit is controlled by the three-position STROB LIGHT - OFF - FIN STROB rocker switch in the overhead switch panel. This allows the fin tip strobe to be activated with, or independently of, the wing tip strobes.

When installed the separate power supply unit is located in the aft section of the fuselage.

(1) 10 amp with Garmin G1000 only.

2. Landing (Taxi) Lights

Two landing lights are installed, one in each wingtip. These lights are also used for taxi operations. Both lights are controlled by a two position rocker switch (LDG LIGHT - OFF) located in the overhead switch panel. The circuit is protected by a 10 amp LANDING LIGHT circuit breaker located on the main circuit breaker panel.

Incandescent landing lights are installed in S/N's 2843001–2843700 and 2843702. LED landing lights are installed in S/N's 2843823, 2843852 and up, with Garmin G1000/G1000 NXi.

See "Figure 3", "Figure 1" and "Figure 2".

NOTE: LED landing lights are authorized service replacements in S/N's 2843701, 2843703 and up. However, first time replacement requires replacing both right and left lamps.

A. Troubleshooting

(1) If equipped with LED lights, check the polarity of the wire connections.
(2) Verify that the appropriate lamp is installed.
(3) Verify correct voltage at the light. Should be 28 volts.
(4) Verify ground at the (GND) terminal.
(5) Replace light / lamp if confirmed faulty.

NOTE: If any LEDs are not illuminated, an LED light should be replaced.

B. Removal

(1) Remove nine (9) screws and washers and remove the Plexiglas lens from the wing tip.
(2) Remove the three (3) lamp retaining screws. Remove retaining ring and bushings (3 ea).

NOTE: Do not disturb adjustment bolts when removing landing light lamp(s). Should adjustment bolt settings be disturbed, check light alignment after installing new lamp.

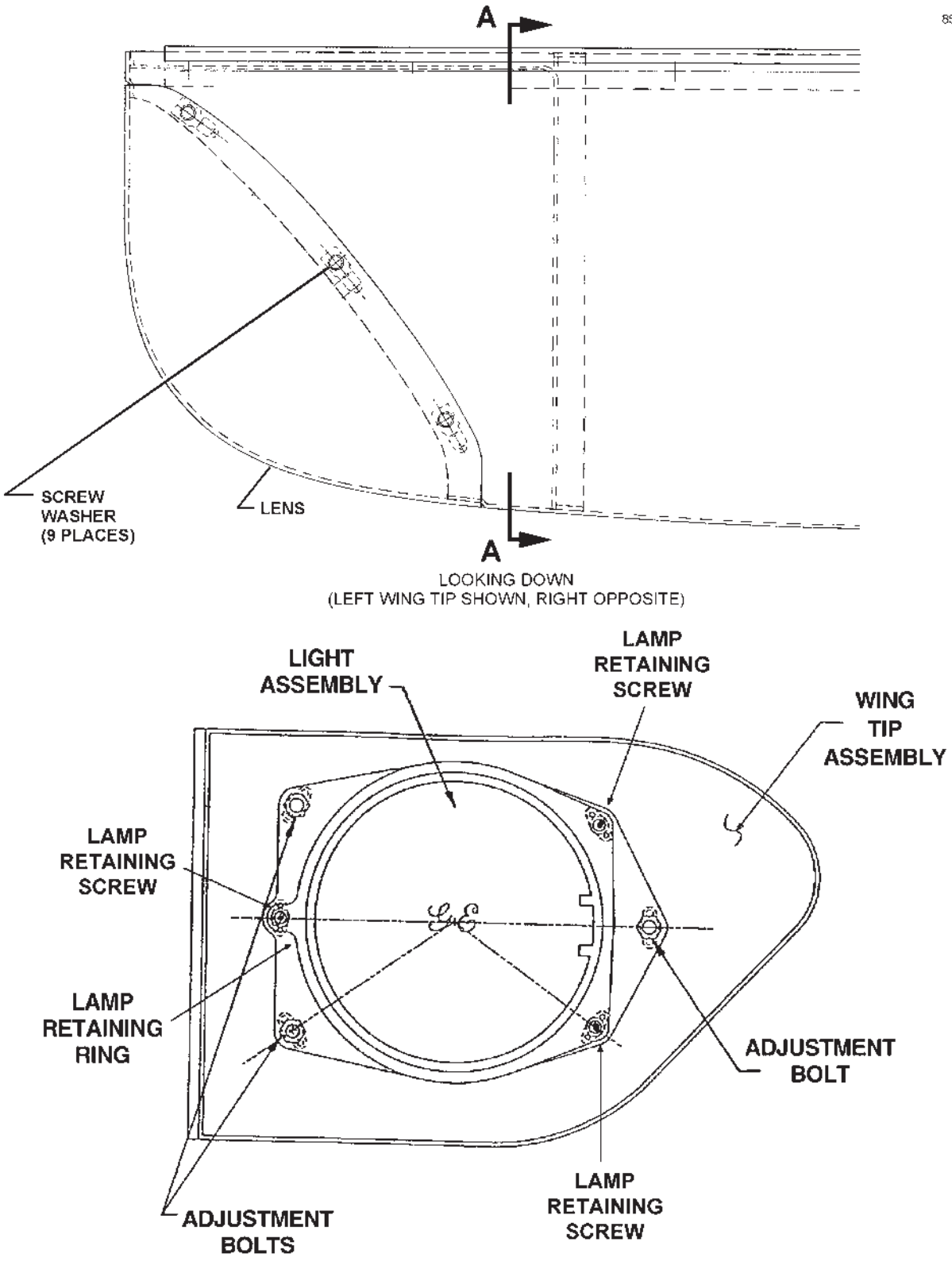
(3) Pull lamp forward far enough to gain access to the electrical leads on back of the lamp.
(4) Disconnect the electrical leads.
(5) Remove lamp.

C. Installation

(1) Connect electrical leads to lamp.

NOTE: If equipped with an LED lamp and the new lamp does not illuminate, remove power and reverse the electrical leads to the lamp. The LED light assembly is polarity sensitive; however, the unit should not be damaged if polarity is inadvertantly reversed.

(2) Position lamp in place and install bushings (3 ea.), retaining ring, and screws (3 ea.).
(3) Verify lamp function by turning BATT MASTR switch ON and selecting LDG LIGHT switch ON.
(4) Place BATT MASTR and landing light switch in OFF position.
(5) Install Plexiglas lens onto wing tip and secure with nine (9) screws and washers.



Landing LIght Installation
Figure 1

D. Adjustment (Alignment)

.(PIR-PPS50047-1, Rev B.)

See “Figure 1” and “Figure 2”.

(1) Locate airplane 20 feet from a target wall as shown in “Figure 2”.

(2) Level airplane both vertically and horizontally. (Refer to 8-20-00.)

(3) Remove nine (9) screws and washers and remove the Plexiglas lens from the wing tip.

(4) Connect airplane to a 28 Vdc external power supply.

(5) Position the BATT MASTR and LDG LIGHT switches ON.

(6) Adjust the three landing light adjustment bolts as necessary to move the landing light housing so that the center of the light beam illuminates the target wall at the following locations. (Refer to “Figure 2”.):

(a) Left light beam is centered at a point 32 ± 1 inches up from the floor and 15 feet ± 3 inches outboard to the left from the airplane’s centerline.

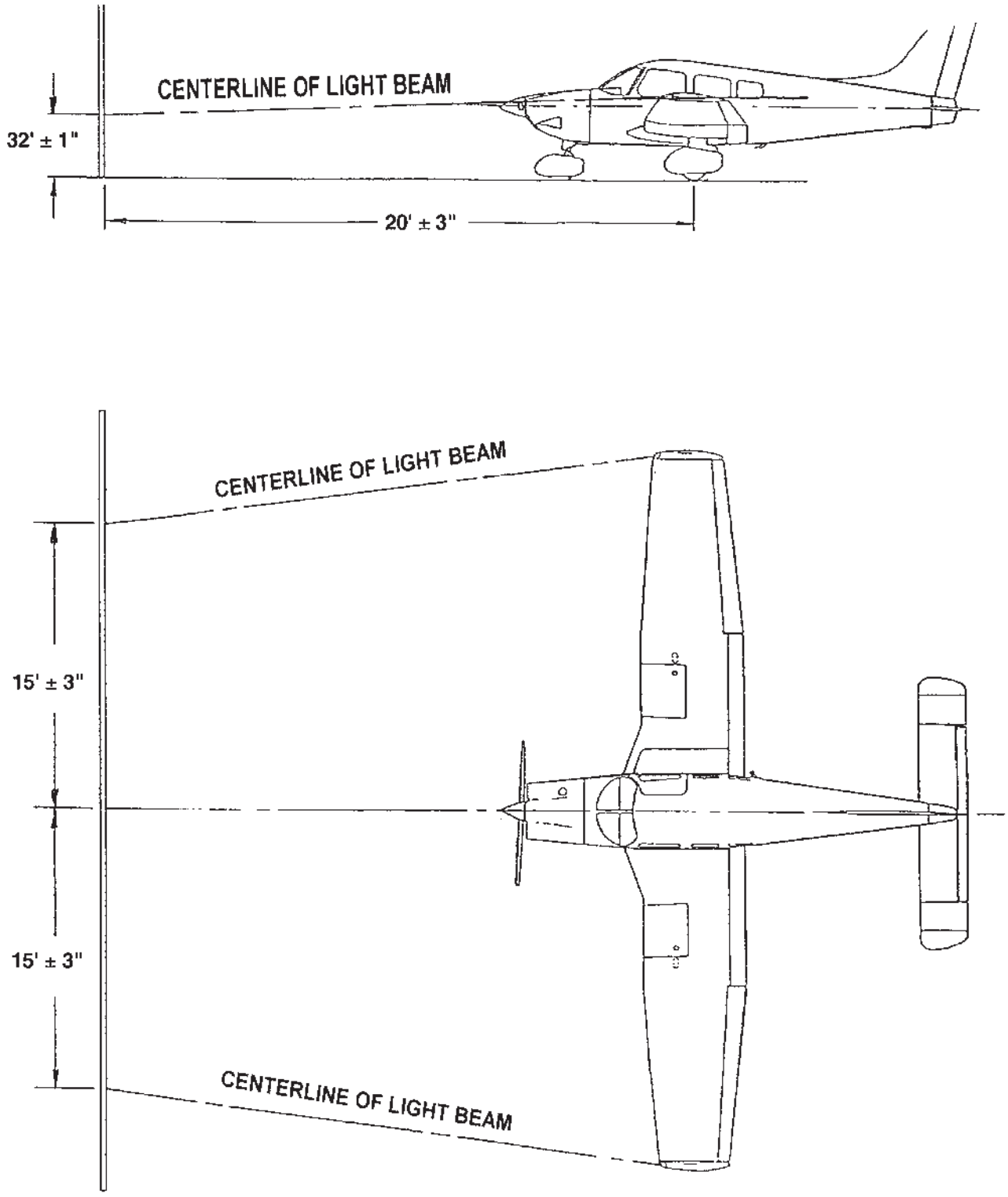
(b) Right light beam is centered at a point 32 ± 1 inches up from the floor and 15 feet ± 3 inches outboard to the right from the airplane’s centerline.

(7) Position the BATT MASTR and LDG LIGHT switches OFF.

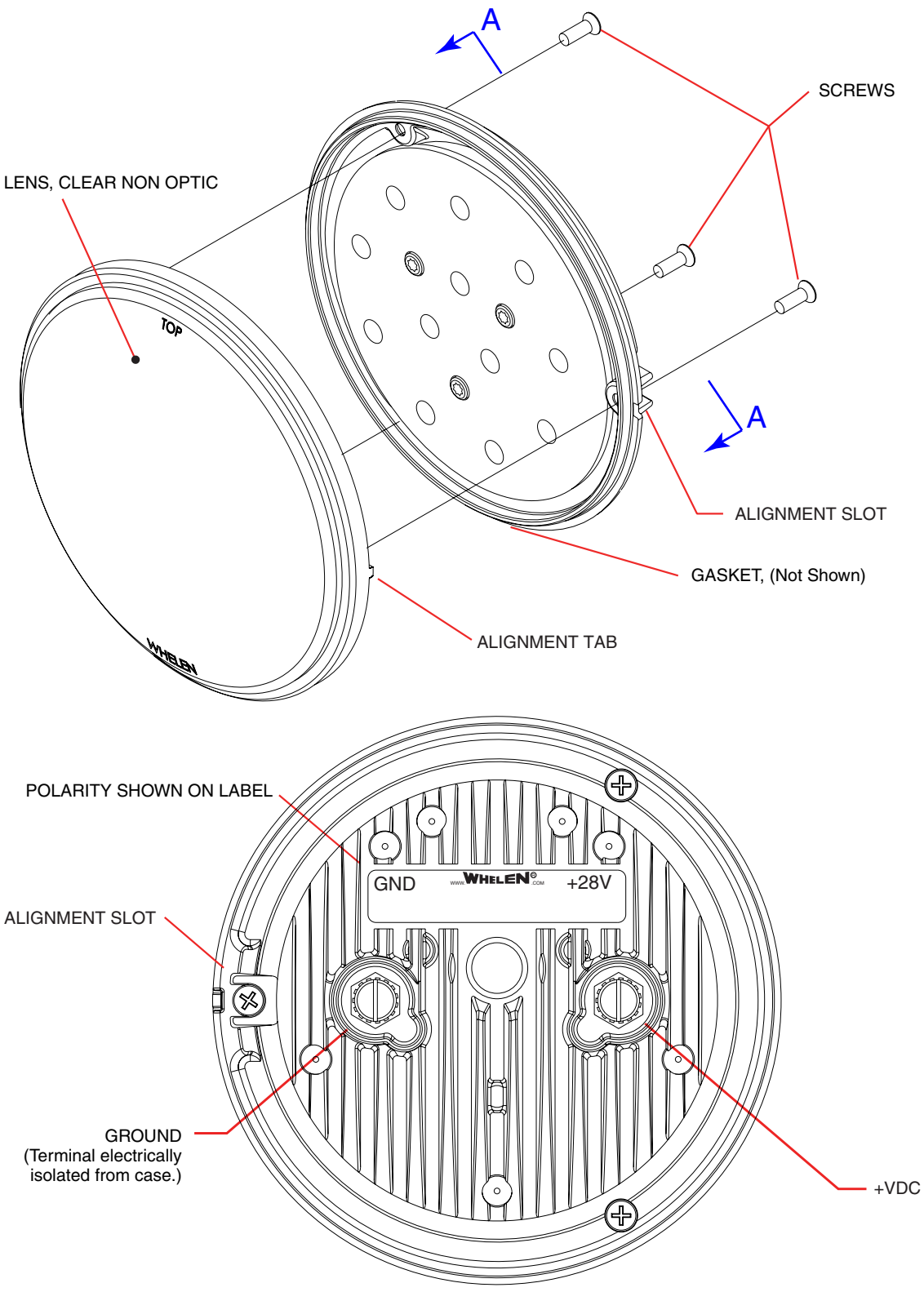
(8) Disconnect 28 Vdc external power supply.

(9) Install Plexiglas cover lens

(10) Remove any device used to level airplane. (Refer to 8-20-00.)



Landing Light Adjustment
Figure 2



VIEW A - A

LED Landing Light Assembly
Figure 3

Effectivity
2843701, 2843703 & up

3. Navigation / Position Lights

Navigation and position lights are installed in a single light assembly which also contains an anti-collision (i.e., strobe) light in each wingtip. The strobe lights are addressed separately in “Anti-Collision (i.e., Strobe) Lights” on page 334020.

The navigation / position lights consist of: a red light forward and a white light aft in the left wingtip; and, a green light forward and a white light aft in the right wingtip. The lights are controlled by a rocker type switch (NAV LIGHT) located in the overhead switch panel. The circuit is protected by a 5 amp NAV LIGHTS circuit breaker.

An optional white tail light is also available which is mounted in the rudder tip fairing assembly. When installed, the tail light is powered from and controlled by the same NAV LIGHT switch as the navigation/ position lights.

A. Wingtip

Three different installations exist.

(1) Installation A

Incandescent navigation / position lamps combined with a Xenon flashtube.

S/N’s 2843001–2843700, 2843702, 2843736 and up with Standard, Avidyne Entegra, or Garmin G500. See “Figure 4” on page 33408 and “Figure 5” on page 33409.

(a) Removal

1) Ensure that the BATT MASTR, STROB LIGHT, and NAV LIGHT switches are in OFF position.
2) Disengage (pull out) NAV LIGHTS and ANTI-COLL circuit breakers.
3) Disconnect the negative (i.e., ground) battery cable.
4) Remove the two screws securing the lens retainer.
5) Remove the lens retainer and one or both of the following: the red (left wing) or green (right wing) lens; or the aft clear lens.

NOTE: I.E., if only replacing one lamp, only remove the lens necessary to access that lamp.

a) To remove the left or right navigation incandescent lamp (i.e., behind the green or red lens), twist lamp counterclockwise to release lamp and pull out.
b) To remove the white aft lamp, pull straight out of its socket.

6) If removing the entire light assembly, continue as follows:
a) Remove the three screws securing the light assembly to the mounting plate.
b) Break sealant and move the light assembly out of the way. Draw electrical leads out to expose connectors. Capture wing harness to ensure it does not retract into wing.
c) Disconnect the light assembly electrical leads from the wing harness and remove the light assembly from the wing tip.
d) Remove old sealant from mating surfaces of light assembly and wingtip.

(b) Installation

1) If the entire light assembly has been removed, proceed as follows:

NOTE: If only one or both navigation/position lamps where removed, skip to step "2)".

a) Connect the light assembly electrical leads to the wing harness. Replace any Ty-Wraps or sleeves previously removed.

b) Release the wing harness and carefully feed the connector and strobe lamp electrical leads through the hole in the base.

c) Put the light assembly in place and secure to mounting plate with three screws.

CAUTION: MATING SURFACES MUST BE SEALED TO PREVENT WATER INTRUSION.

d) Fillet seal edges of mating surfaces of light assembly and wingtip. Use "Sealant, Airframe and Component" from Consumable Materials chart in 91-10-00.

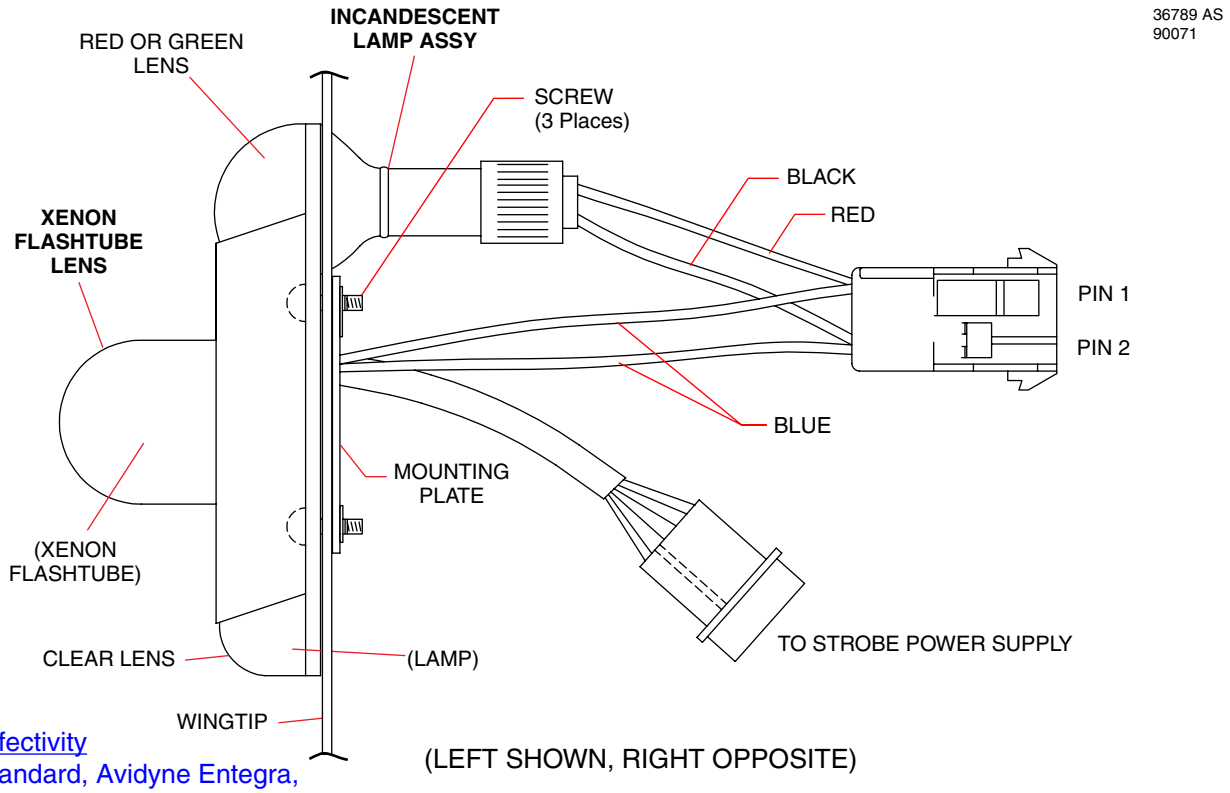
2) If only one or both navigation/position lamps were removed, proceed as follows:

a) Install the left or right forward incandescent lamp (i.e., behind the green or red lens):

1] Align lamp in socket.

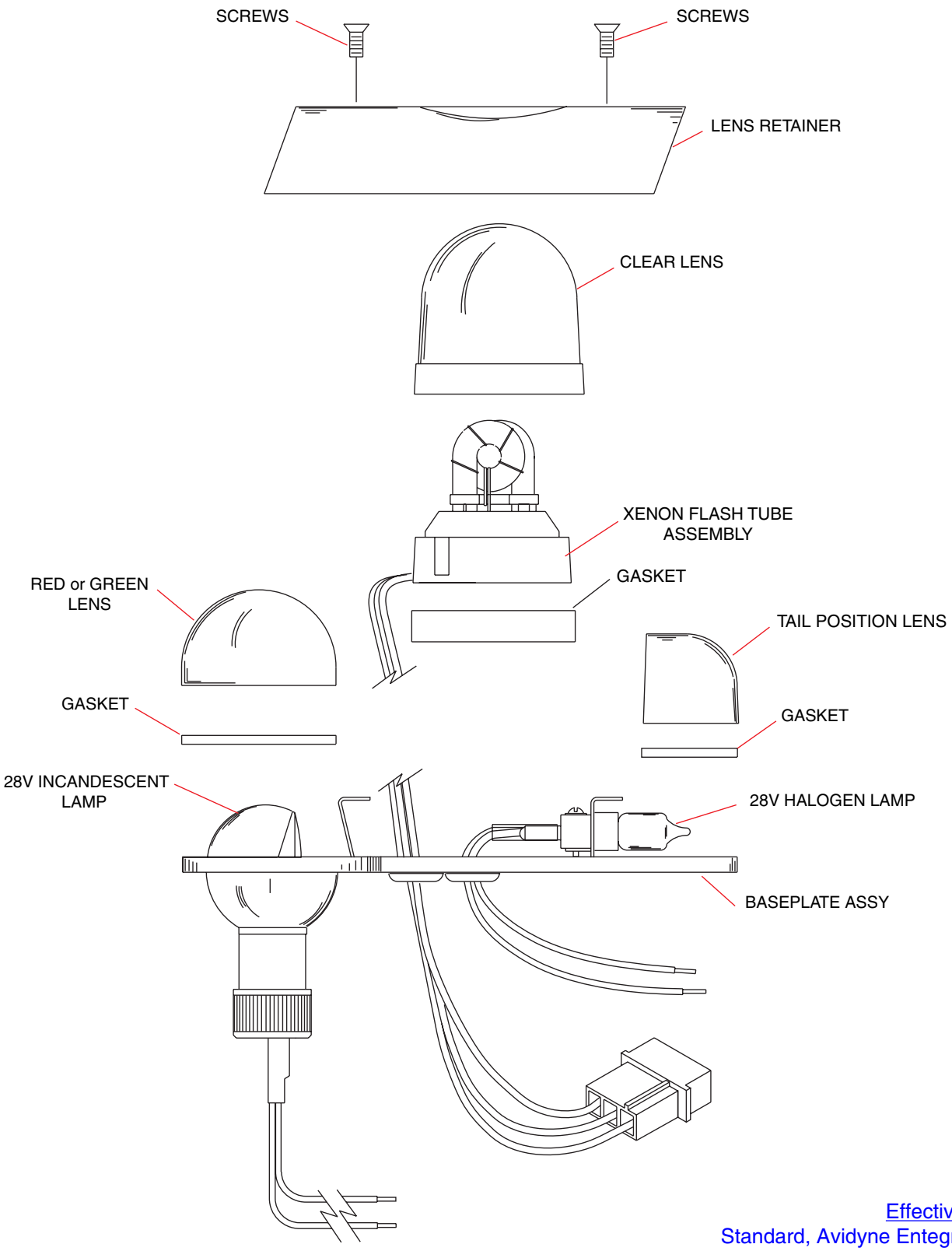
2] Push lamp to bottom of socket and twist clockwise to lock in place.

b) Install the white (aft) lamp by aligning the lamp with its socket and pushing straight in until lamp is seated.



Effectivity
Standard, Avidyne Entegra, or Garmin G500
2843001–2843700, 2843702, 2843736 and up

Wingtip Navigation/Position/Strobe Light Installation
Figure 4



Wingtip Navigation/Position/Strobe Light Assembly
Figure 5

Effectivity
Standard, Avidyne Entegra, or Garmin G500
2843001–2843700, 2843702, 2843736 and up

3) Connect the negative (i.e., ground) battery cable.
4) Set (push in) NAV LIGHTS circuit breaker.
5) Verify that lamp(s) is/are working by selecting BATT MASTR and NAV LIGHT switches ON and checking that all navigation light lamps are lit. Select BATT MASTR and NAV LIGHT switches OFF. Set (push in) ANTI-COLL circuit breaker.
6) Install lenses over each lamp. Install lens retainer and secure with two screws.
7) If entire light assembly was removed and reinstalled, verify function of strobe light before securing airplane.

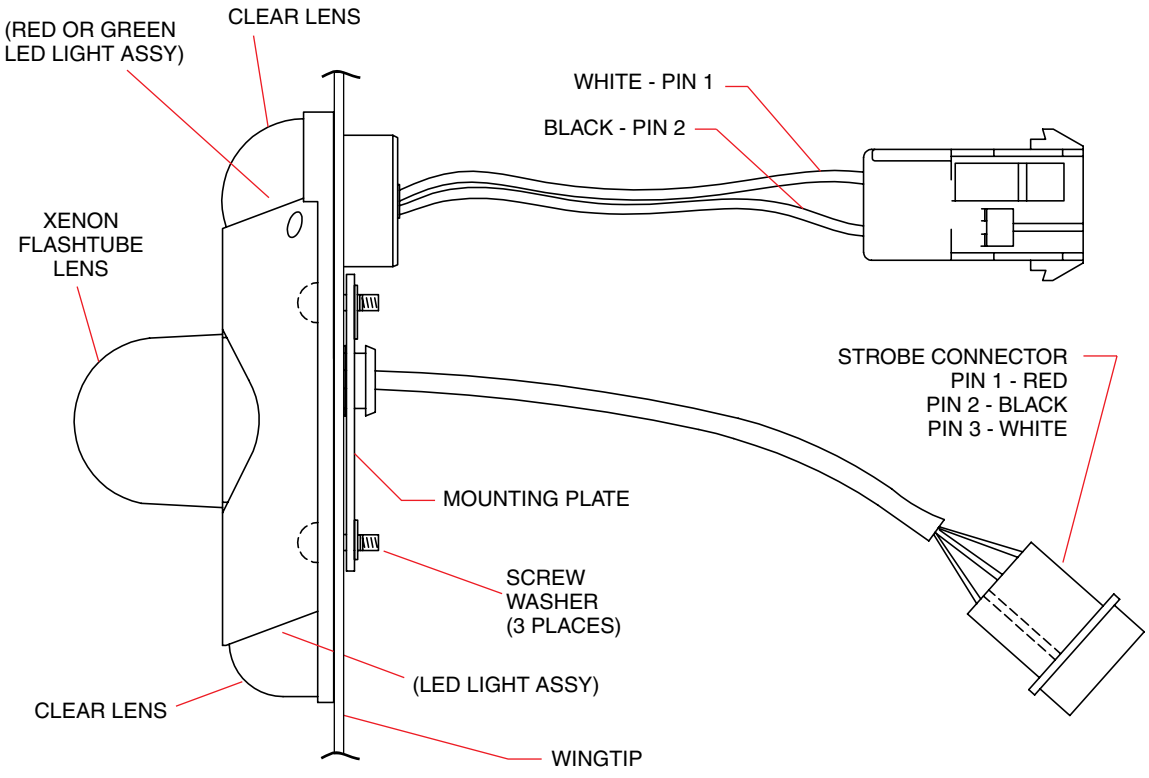
(2) Installation B

LED navigation / position lamps combined with a Xenon flashtube.

With Garmin G1000. See “Figure 6” on page 334011 and “Figure 7” on page 334012.

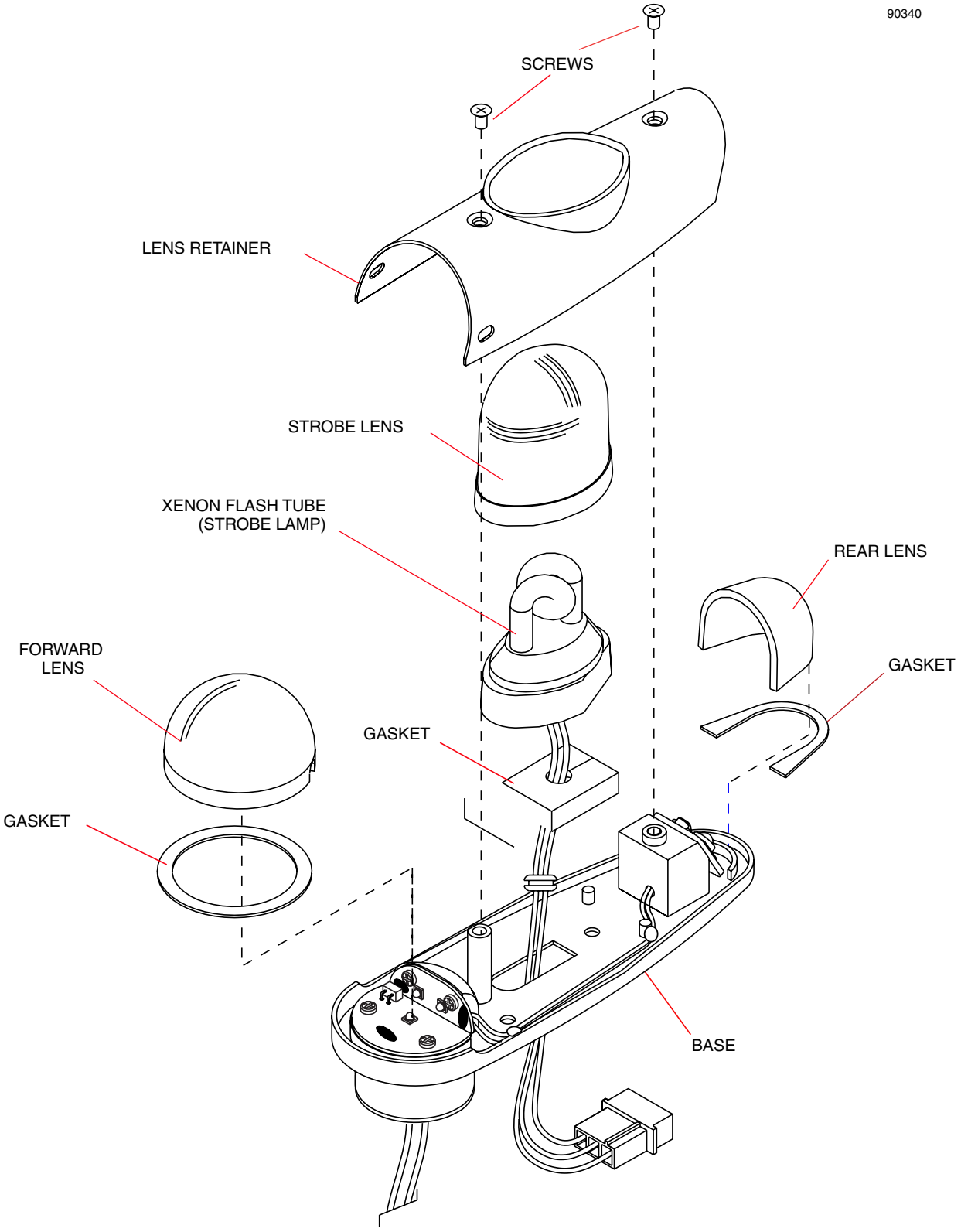
Removal and Installation is essentially the same as that shown under “Installation A” on page 33407 with the exception that replacement of either or both of the forward and aft navigation / position LED lamp assemblies requires removal and replacement of the entire LED lamp assembly base.





Wingtip Navigation/Position/Strobe Light Installation
Figure 6

Effectivity
with Garmin G1000



Effectivity
with Garmin G1000

Wingtip Navigation/Position/Strobe Light Assembly
Figure 7

(3) Installation C

LED navigation / position lamps combined with a LED strobe lamp.

With Garmin G1000 NXi. See “Figure 8” on page 334014 and “Figure 9” on page 334015.

If one LED fails the entire light assembly must be replaced.

(a) Removal

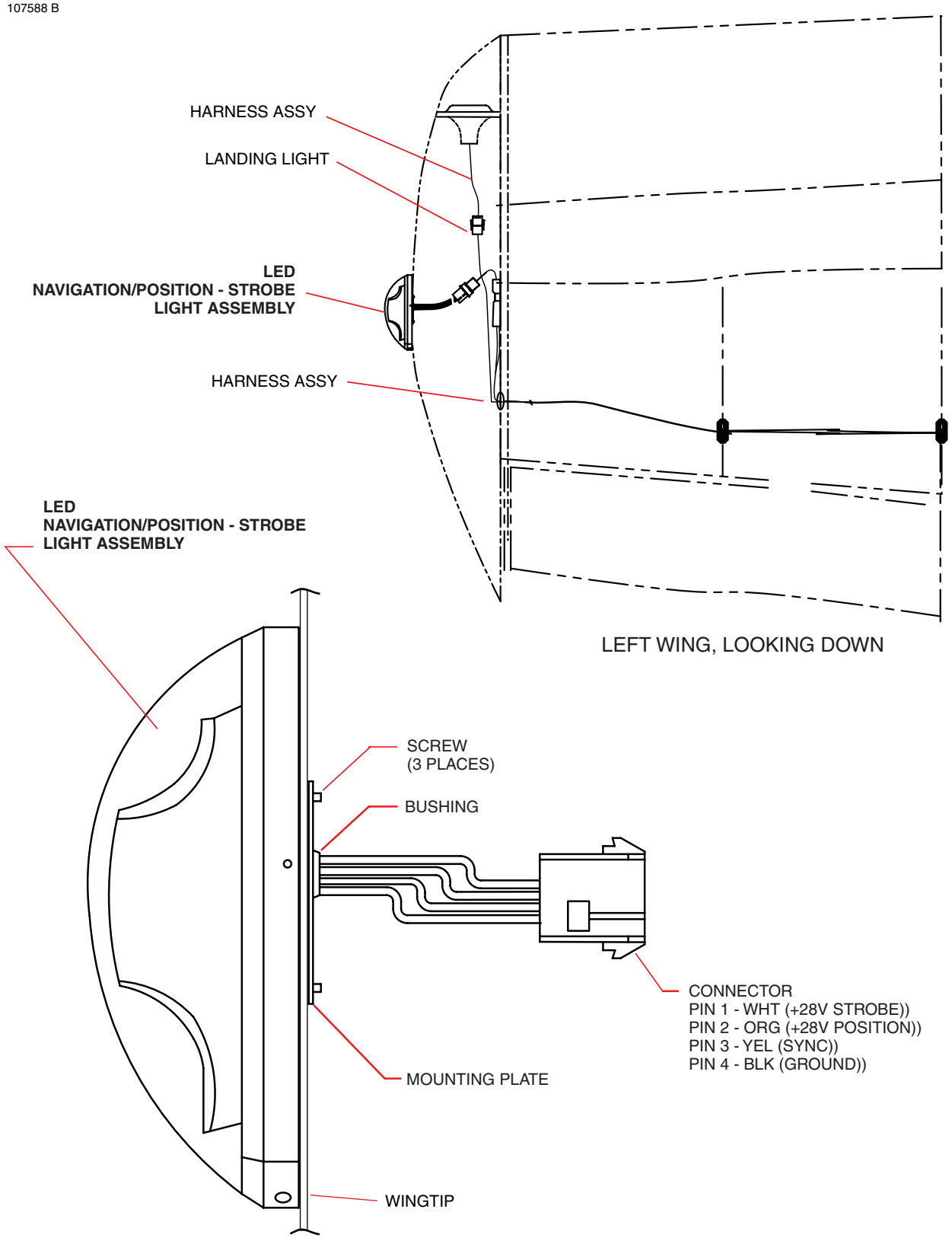
1) Ensure that the BATT MASTR and NAV LIGHT switches are in OFF position.
2) Disengage (pull out) NAV LIGHTS circuit breaker.
3) Disconnect the negative (i.e., ground) battery cable.
4) Lens
   a) Remove two (2) socket head screws from aft end of light assembly and remove lens retainer.
   b) Slide lens aft approximately 1/2 inch and lift away from light assembly.
5) Light Assembly
   a) Remove lens as described above.
   b) Remove three (3) phillips head screws and separate light assembly from baseplate.
   c) Pull light assembly away from wing tip to the attached wiring harness out to expose the connector. Capture the exposed wiring harness at the baseplate with tape to ensure it does not retract into the wing.
   d) Disconnect the connector and remove light assembly.
6) Baseplate

   Not required unless the existing baseplate is damaged or deteriorated.
   a) Remove Lens and Light Assembly as describe above.
   b) Remove three screws securing baseplate to wingtip and remove baseplate.
   c) Remove any remaining sealant from wingtip.

(b) Installation

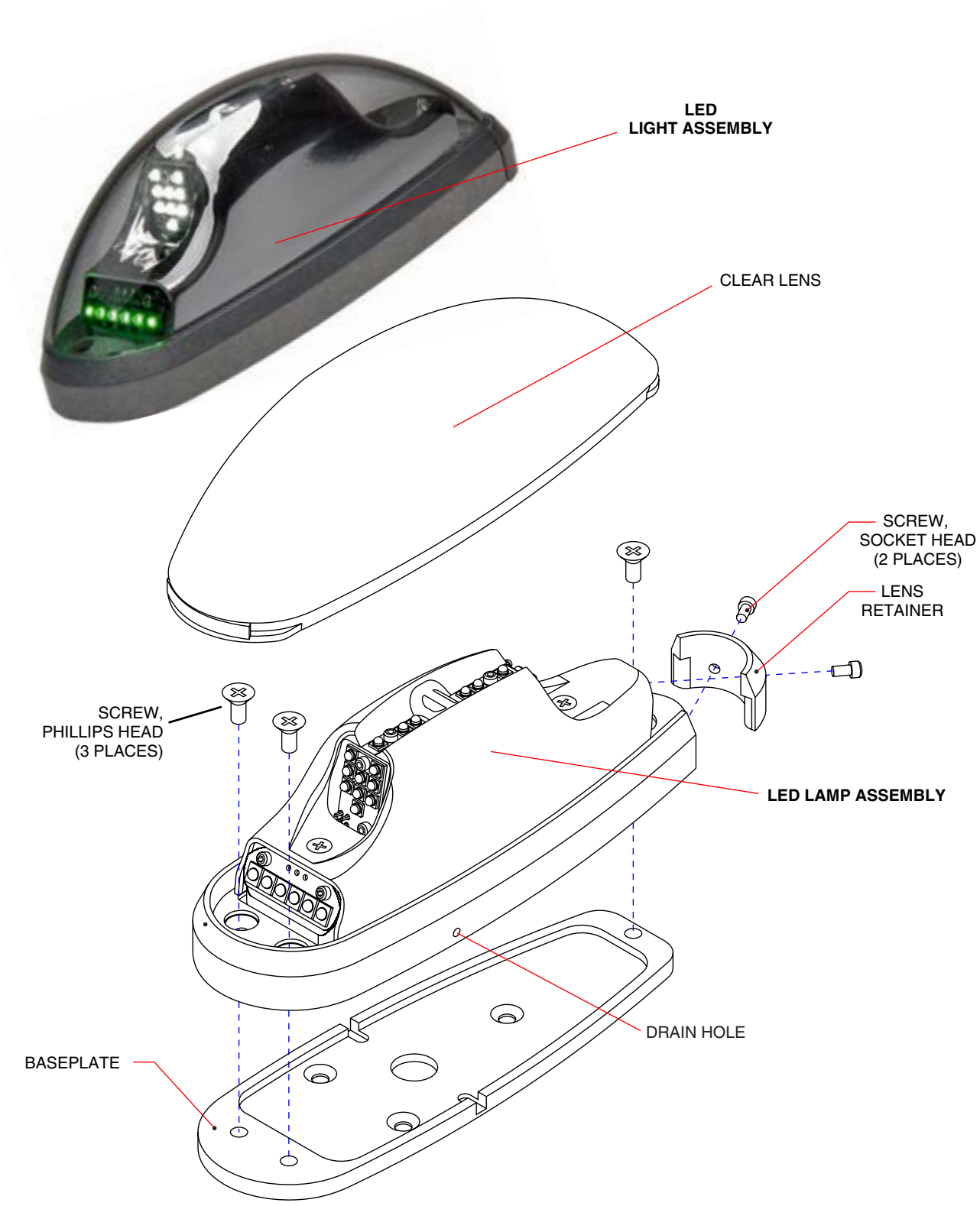
1) Baseplate (If required.)
   a) Ensure wingtip is clean and dry.
   b) Feed exposed wing wiring harness through hole in baseplate and position baseplate on wingtip.
   c) Secure with three screws.
   d) Fillet seal around the mating surfaces of the wingtip and baseplate to prevent water intrusion. Seal with “Sealant, Airframe and Component” per the Consumable Materials chart in 91-10-00.
2) Light Assembly
   a) Connect navigation/postion/strobe light assembly electrical connector to the exposed harness wiring connector at the baseplate.
   b) Feed the wiring harness through the hole in the baseplate and place the new light assembly in position.
   c) Secure with three phillips head screws.
3) Lens
   a) Place lens approximately 1/2 inch aft of light assembly, slide forward into position.
   b) Place lens retainer in position over aft end of lens.
   c) Secure lens retainer and lens with two (2) socket head screws.

107588 B



Effectivity
with Garmin G1000 NXi

Wingtip Navigation/Position/Strobe Light Installation
Figure 8



Wingtip Navigation/Position/Strobe Light Assembly
Figure 9

Effectivity
with Garmin G1000 NXi

4) Connect the negative (i.e., ground) battery cable.
5) Engage (push IN) the NAV LIGHTS circuit breaker.
6) Switch the BATT MASTR and NAV LIGHT switches ON and ensure the navigation/ position LEDs illuminate.
7) Switch NAV LiGHT and BATT MASTR OFF and secure airplane.

B. Tail (Optional)

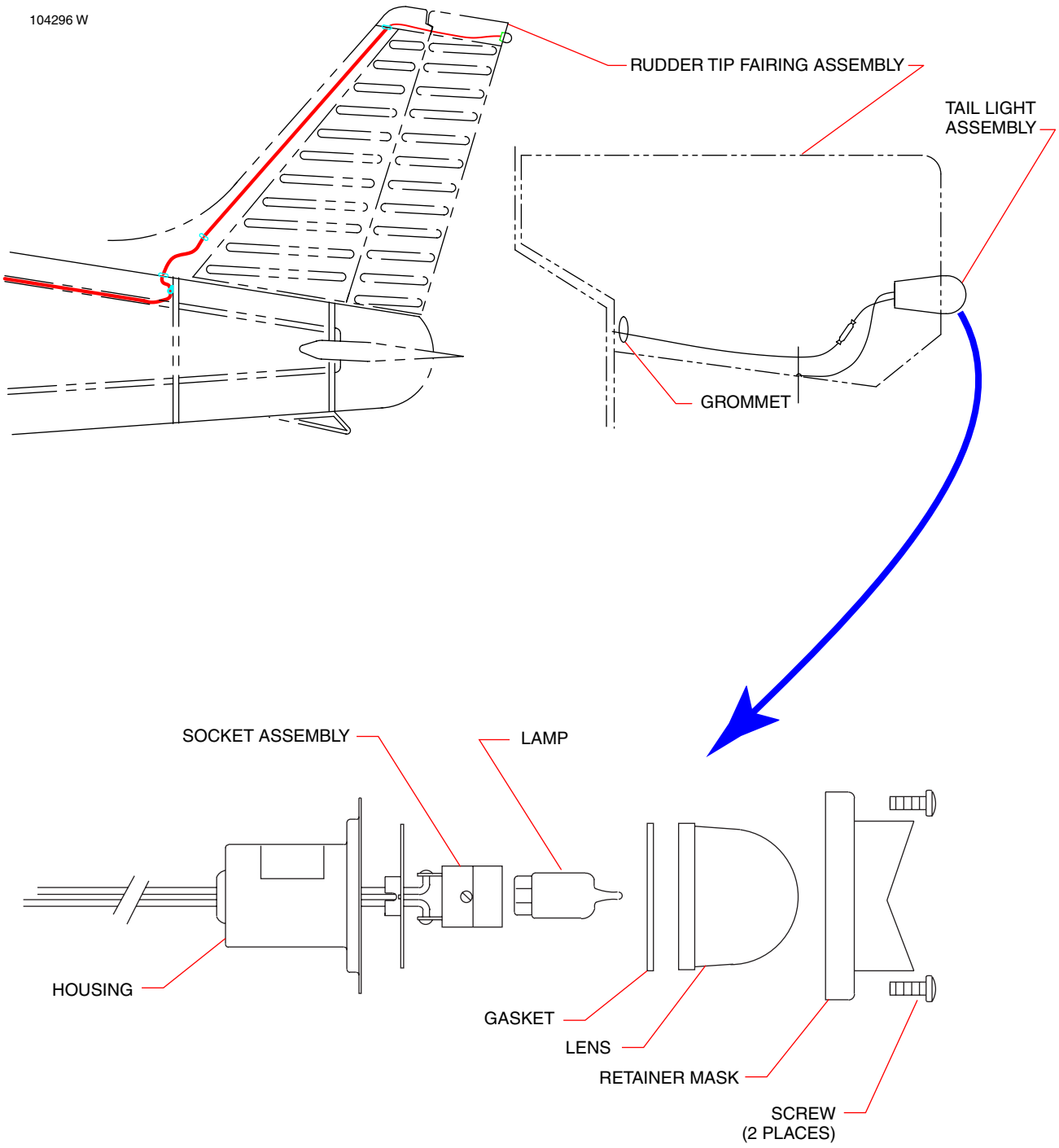
A white tail light is available. When installed it is mounted in the aft edge of the rudder tip fairing. Two different installations exist:

(1) Incandescent Tail Light

Installed with Standard, Avidyne Entegra, Garmin G500, or with Garmin G1000. See “Figure 10” on page 334017.

(a) Removal
1) Select BATT MASTR and NAV LIGHT switches OFF.
2) Disconnect the negative (i.e., ground) battery cable.
3) To replace the lamp:
a) Remove the two screws securing the retainer mask.
b) Remove retainer mask, lens, and lens gasket from light assembly.
c) Pull the lamp straight out of the socket assembly.
4) To remove the light assembly:
a) At the rudder:
1] Remove the screws securing the fairing assembly to the rudder.
2] Lift the fairing assembly off the rudder and disconnect the tail light ground lead by removing the screw, washer, and nut.
3] Disconnect the power lead at the splice.
b) At the bench:
1] Remove the two screws securing the retainer mask.
2] Remove retainer mask, lens, and lens gasket.
3] Remove the light assembly housing from the fairing assembly.

(b) Installation
1) If only replacing the lamp:
a) Align the new lamp with the socket assembly and press straight back until fully seated.
b) Install lens gasket, lens, and retainer assembly and secure with screws (2).
2) If reinstalling the entire light assembly:
a) At the bench:
1] Feed the electrical leads through the mounting hole in the rudder tip fairing assembly.
2] Position light assembly housing in place in the fairing assembly.
3] Position lens gasket, lens, and retainer assembly.
4] Apply Loctite 222MS to screws (2).
5] Secure light assembly to fairing assembly with screws (2).



Tail Light Installation (Optional)
Figure 10

Effectivity
Standard, Avidyne Entegra, or Garmin G500, or Garmin G1000

b) At the rudder:

1] Connect the light assembly ground lead to the rudder with screw, washer, and nut.

2] Connect the power lead at the splice.

3] Position fairing assembly in place atop the rudder and secure with screws.

3) Connect the negative (i.e., ground) battery cable.

4) Select BATT MASTR and NAV LIGHT switches ON, verify tail light illuminates.

5) Select BATT MASTR and NAV LIGHT switches OFF, secure airplane.

(2) LED Tail Light

With Garmin G1000 NXi. See “Figure 11” on page 334019.

If one LED fails the entire light assembly must be replaced.

(a) Removal

1) Select BATT MASTR and NAV LIGHT switches OFF.

2) Disconnect the negative (i.e., ground) battery cable.

3) At the rudder:

a) Remove the screws securing the fairing assembly to the rudder.

b) Lift the fairing assembly off the rudder and disconnect the tail light electrical leads at the connector.

4) At the bench:

a) Remove the two screws securing the retainer mask.

b) Remove retainer mask, lens, and lens gasket.

c) Remove the LED assembly from the fairing assembly.

(b) Installation

1) At the bench:

a) Feed the electrical leads through the mounting hole in the rudder tip fairing assembly.

b) Position LED assembly in place in the fairing assembly.

c) Position lens gasket, lens, and retainer assembly.

CAUTION: APPLY LOCTITE 609 TO SCREWS TO SEAL AGAINST WATER INTRUSION.

d) Apply Loctite 222MS to screws (2) and install screws to secure LED assembly, lens gasket, lens, and retainer assembly to fairing assembly.

2) At the rudder:

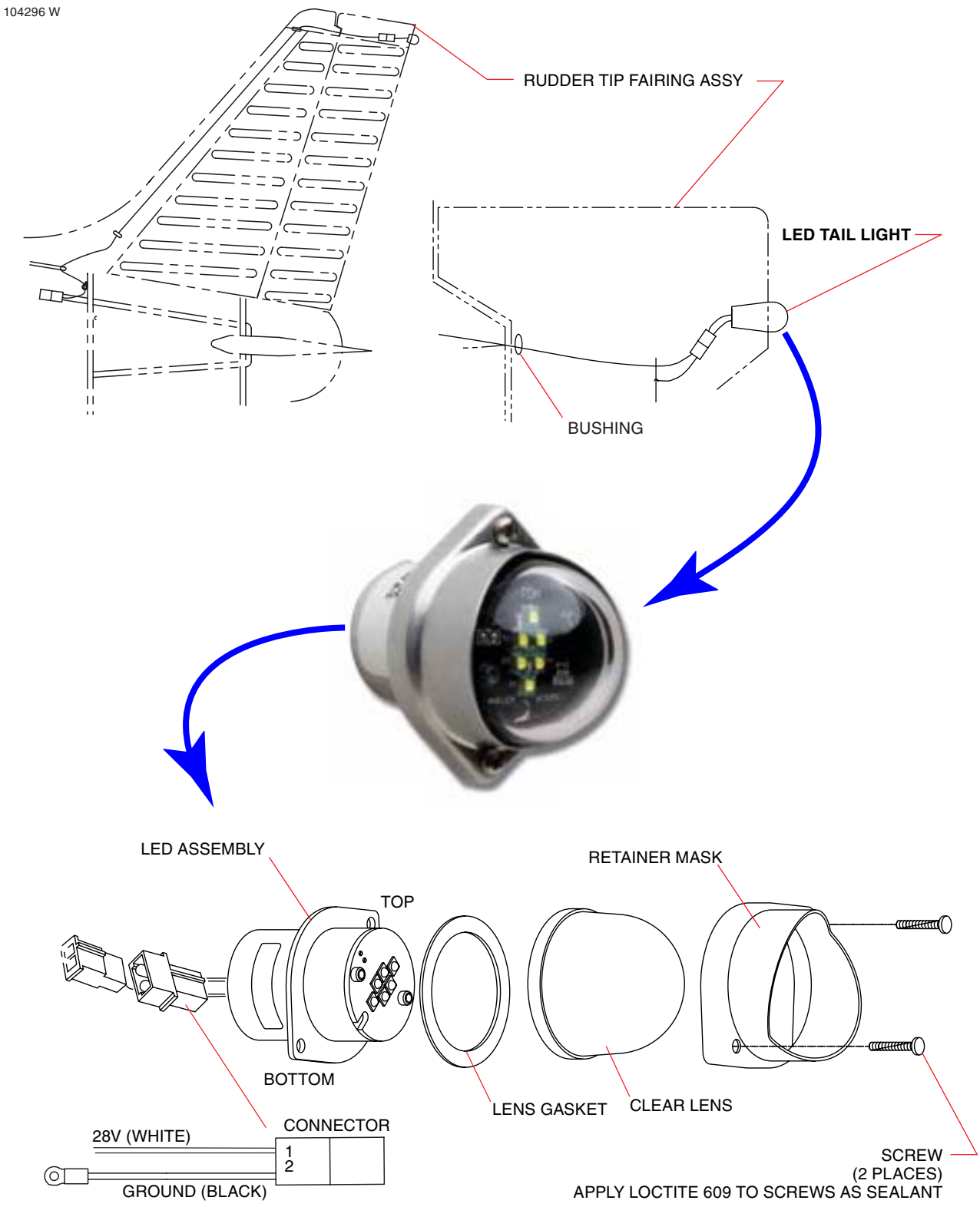
a) Connect the tail light electrical leads to the rudder wiring harness.

b) Position fairing assembly in place atop the rudder and secure with screws.

3) Connect the negative (i.e., ground) battery cable.

4) Select BATT MASTR and NAV LIGHT switches ON, verify tail light illuminates.

5) Select BATT MASTR and NAV LIGHT switches OFF, secure airplane.



Tail Navigation LED Light Assembly
Figure 11

Effectivity
with Garmin G1000 NXi

4. Anti-Collision (i.e., Strobe) Lights

A. Description

The exterior wingtip lighting assembly includes a pulsating anti-collision (i.e., strobe) white light. The strobe light is a Xenon flashtube with a separate power supply; or, an LED strobe light with no separate power supply. The strobe units are rated to flash approximately 50 times per minute. The anti-collision lights are controlled by an independent rocker switch (STROB LIGHT) in the overhead switch panel. The circuit is protected by a 5(1) amp ANTI-COLL LIGHTS circuit breaker.

An optional red fin tip ground reconigition/anti-collision (i.e., strobe) light is available in S/N's 2843115 and later. Again, the strobe light is a Xenon flashtube with a separate power supply; or, an LED strobe light with no separate power supply. When installed, the unit is controlled by the three-position STROB LIGHT - OFF - FIN STROB rocker switch in the overhead switch panel. This allows the fin tip strobe to be activated with or independently of the wing tip strobes.

When installed the separate power supply unit is located in the aft section of the fuselage.

B. Troubleshooting

When troubleshooting the strobe light system, check if the trouble is in flashtube or power supply. This can be accomplished by replacing the flash tube assembly with a good operating flash tube, or with the use of a strobe check unit. Normal operating power supply emits an audible tone of 1 to 1.5 kHz. If no sound is emitted, check system as follows: (When troubleshooting the system, use the appropriate schematic. Refer to 91-33-40.)

CAUTION: WHEN DISCONNECTING POWER SUPPLY, ALLOW TEN MINUTES OF BLEED DOWN TIME BEFORE HANDLING UNIT.

(1) Check input voltage at power supply is 28-volts.

CAUTION: WHEN DISCONNECTING AND CONNECTING POWER SUPPLY INPUT CONNECTIONS, DO NOT REVERSE CONNECTIONS. REVERSED POLARITY OF INPUT VOLTAGE FOR JUST AN INSTANT WILL PERMANENTLY DAMAGE POWER SUPPLY. REVERSED POLARITY DESTROYS A PROTECTIVE DIODE IN THE POWER SUPPLY, CAUSING SELF-DESTRUCTION FROM OVERHEATED POWER SUPPLY. THE DAMAGE IS SOMETIMES NOT IMMEDIATELY APPARENT, BUT WILL CAUSE SYSTEM FAILURE IN TIME.

(2) Check for malfunction in interconnecting cables.

CAUTION: A SHORT OF THE TYPE IN STEPS (A). AND (B). WILL NOT CAUSE PERMANENT DAMAGE TO POWER SUPPLY, BUT THE SYSTEM WILL BE INOPERATIVE. ANY CONNECTION BETWEEN PINS 1 AND 3 OF THE INTERCONNECTING CABLE DISCHARGE THE CONDENSER IN THE POWER SUPPLY AND DESTROY TRIGGER CIRCUITS.

(a) Check Pins 1 and 3 of interconnecting cable are not reversed.

(b) Use an ohmmeter to check continuity between Pin 1 and 3 of interconnecting cable. If a reading is obtained, cable is shorted and must be replaced.

(1) 10 amp with Garmin G1000 in S/N's 2843701, 2843703–2843822, 2843824–2843851, and 2843853, only.

(3) Check interconnecting cables for shorts.
   (a) Disconnect output cables from power supply outlets.
   (b) Use an ohmmeter for the following continuity checks.
      1) Check from Pin 1 to Pin 1, Pin 2 to Pin 2, and Pin 3 to Pin 3. If no continuity exists, cable is broken and must be replaced.
      2) Check continuity between Pins 1 and 2, 1 and 3, and 2 and 3 of the interconnecting cable. If continuity exists cable is shorted and must be replaced.

(4) Check tube socket assembly for shorts.
   (a) Disconnect tube socket assembly of the anti-collision light from the interconnecting cable.
   (b) Use an ohmmeter for the following continuity checks.
      1) Check continuity between Pin 1 of AMP connector to Pin 1 of tube socket. If no continuity exists, tube socket assembly is broken and must be replaced.
      2) Check Pin 2 of AMP connector to Pins 6 and 7 of tube socket.
      3) Check Pin 3 of AMP connector to Pin 4 of tube socket.
      4) When making these tests, if no continuity exists, tube socket assembly is broken and must be replaced.

(5) Clear all possible shorts at the power supply, by disconnecting the output cables from the power supply outlets, and connect an operating strobe lighthead assembly or a strobe check unit directly to the power supply outlet, then apply the required voltage to the power supply input. If this application proves positive the power supply is in working condition, and the problem may be with the interconnecting cables

(6) Most power supplies are protected against a short or open circuit on the output. In either case, the power supplies will effectively turn themselves off when subjected to a shorted output of a Xenon flash tube that refuses to flash.

(7) Strobe light power supplies are meant to be used, not to remain in an inactive state. Use them at all times, this will improve their proper functioning. Any strobe light power supply that has been out of service for a long period of time is subject to failure because the electrolytic condenser loses the polarity formation.

   If this is the case, it is recommended to start operating the system on a voltage that is reduced by 25 percent for 10 to 15 minutes before putting the power supply into normal service. This will prevent overheating of the condenser while they reform. If the power supply, after a long period of non use, is operated at full voltage immediately, there is an excellent possibility that the condenser will become overheated

C. Wing Tip

Three different installations exist.

(1) Installation A

Incandescent navigation / position lamps combined with a Xenon flashtube.

S/N's 2843001–2843700, 2843702, 2843736 and up with Standard, Avidyne Entegra, or Garmin G500. See "Figure 4" on page 33408 and "Figure 5" on page 33409.

(a) Removal

Removing the Xenon flashtube requires removing the entire light assembly.

1) Ensure that the BATT MASTR, STROB LIGHT, and NAV LIGHT switches are in OFF position.
2) Disengage (pull out) NAV LIGHTS and ANTI-COLL circuit breakers.
3) Disconnect the negative (i.e., ground) battery cable.
4) At the wing tip:
   a) Remove the two screws securing the lens retainer.
   b) Remove the lens retainer and the clear lens covering the strobe, as well as the lenses covering the forward and aft navigation/position lamps.
   c) Remove the three screws securing the light assembly to the mounting plate.
   d) Break sealant and move the light assembly out of the way. Draw electrical leads out to expose connectors. Capture wing harness to ensure it does not retract into wing.
   e) Disconnect the light assembly electrical leads from the wing harness and remove the light assembly from the wing tip.
   f) Remove old sealant from mating surfaces of light assembly and wingtip.
5) At the bench:

   Pull the strobe lamp (i.e., Xenon flashtube) and gasket away from the light assembly while carefully feeding the electrical leads through the hole in the base.

(b) Installation

1) At the bench:
   a) Ensure the gasket (and grommet, if any) are positioned properly.
   b) Carefully feed the strobe lamp (i.e., Xenon flashtube) electrical leads through the hole in the base.
2) At the wing tip:
   a) Connect the light assembly electrical leads to the wing harness. Replace any Ty-Wraps or sleeves previously removed.
   b) Release the wing harness and carefully feed the connector and strobe lamp electrical leads through the hole in the base.
   c) Put the light assembly in place and secure to mounting plate with three screws.

      CAUTION: MATING SURFACES MUST BE SEALED TO PREVENT WATER INTRUSION.

   d) Fillet seal edges of mating surfaces of light assembly and wingtip. Use "Sealant, Airframe and Component" from Consumable Materials chart in 91-10-00.
   e) Replace the clear lens over the strobe and the forward and aft navigation / position lens over their respective lamps. Capture lenses with the lens retainer.
   f) Secure the lens retainer with two screws.

3) Connect the negative (i.e., ground) battery cable.
4) Set (push in) ANTI-COLL circuit breaker.
5) Verify that the strobe is working by selecting BATT MASTR and STROB LIGHT switches ON and checking that strobe is flashing.
6) Select STROB LIGHT switch OFF. Set (push in) NAV LIGHTS circuit breaker.
7) Verify that navigaion/position lamp(s) are working by selecting NAV LIGHT switch ON and checking that all navigation light lamps are lit.
8) Select BATT MASTR and NAV LIGHT switches OFF.
9) Secure airplane.

(2) Installation B

LED navigation / position lamps combined with a Xenon flashtube.

With Garmin G1000. See “Figure 6” on page 334011 and “Figure 7” on page 334012.

Removal and Installation of the strobe lamp (i.e., Xenon flashtube) is the same as that shown under “Installation A” on page 334022.

(3) Installation C

LED navigation / position lamps combined with a LED strobe lamp.

With Garmin G1000 NXi. See “Figure 8” on page 334014 and “Figure 9” on page 334015.

If one LED fails the entire light assembly must be replaced.

See “Removal” and “Installation” under “Installation C” on page 334013.

D. Fin Tip (Optional)

Three different installations exist.

(1) Installation A

S/N's 2843115–2843129, 2843145–2843597, 2843683–2843700, 2843702, 2843736 and up with Standard or Garmin G500.

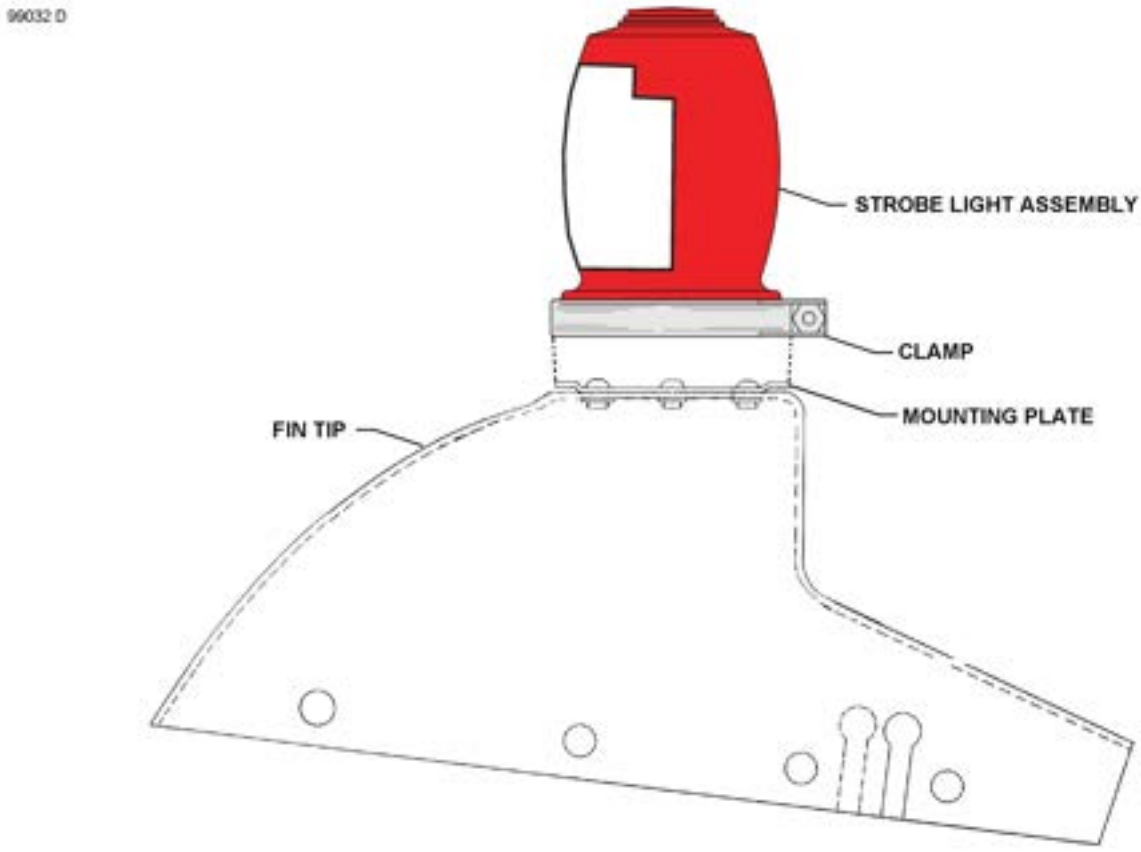
See "Figure 12" and "Figure 13" on page 334026.

(a) Removal

1) Ensure that the BATT MASTR, STROB LIGHT, and NAV LIGHT switches are in OFF position.
2) Disengage (pull out) NAV LIGHTS and ANTI-COLL circuit breakers.
3) Disconnect the negative (i.e., ground) battery cable.
4) At the fin tip:
   a) Remove screws and washers securing fin tip to vertical fin.
   b) Draw the fin tip away from the vertical fin until the electrical harness connector and ground wire are exposed.
   c) Capture the fin side of the harness to prevent its retracting into the fin.
   d) Disconnect the electrical harness connector and the remove the screw and washer securing the ground wire to the upper fin rib and remove the fin tip.
5) At the bench:
   a) Remove the screw and nut securing clamp around the base of the strobe light assembly and the mounting plate.
   b) Remove clamp and remove the strobe light assembly from the fin tip.
   c) Pull the flashtube assembly and gasket out of the the lens.

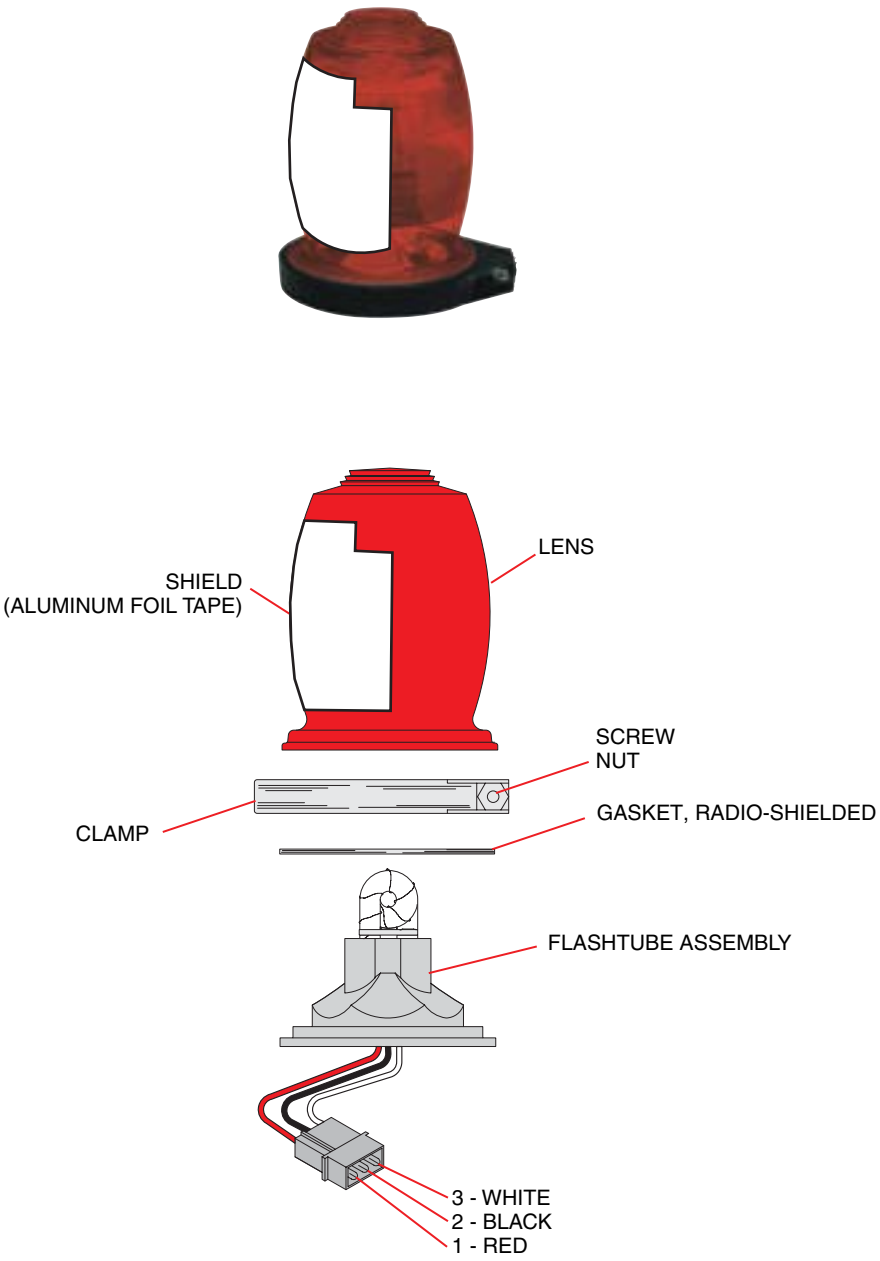
(b) Installation

1) At the bench:
   a) Position the gasket properly.
   b) Push flashtube assembly into the lens.
   c) Put the light assembly in place atop the fin tip and secure to mounting plate with clamp, screw, and nut.
2) At the fin tip:
   a) Connect the electrical harness connector and secure the ground wire to the upper fin rib with screw and washer.
   b) Release the wing harness and place the fin tip in position atop the fin.
   c) Secure with screws and washers.
3) Connect the negative (i.e., ground) battery cable.
4) Engage (push in) the ANTI-COLL circuit breaker.
5) Switch ON the BATT MASTR and STROB LIGHT switches. Verify the wing tip and fin tip strobes flash.
6) Switch the STROB LIGHT switch OFF and then to FIN. Verify the only fin tip strobe flashes.
7) Select STROB LIGHT switch OFF. Set (push in) NAV LIGHTS circuit breaker.
8) Select BATT MASTR switch OFF.
9) Secure airplane.



Fin Tip Strobe Light Installation
Figure 12

Effectivity
2843115–2843129,
2843145–2843597,
2843683–2843700,
2843702, 2843736 and up,
with Standard or
Garmin G500



Effectivity
2843115–2843129,
2843145–2843597,
2843683–2843700,
2843702, 2843736 and up,
with Standard or
Garmin G500

Fin Tip Strobe Light Assembly
Figure 13

(2) Installation B

With Avidyne Entegra; or with Garmin G1000.

See “Figure 14” on page 334028 and “Figure 15” on page 334029.

(a) Removal

1) Ensure that the BATT MASTR, STROB LIGHT, and NAV LIGHT switches are in OFF position.
2) Disengage (pull out) NAV LIGHTS and ANTI-COLL circuit breakers.
3) Disconnect the negative (i.e., ground) battery cable.
4) At the fin tip:
   a) Remove screws and washers securing fin tip to vertical fin.
   b) Draw the fin tip away from the vertical fin until the electrical harness connector and ground wire are exposed.
   c) Capture the fin side of the harness to prevent its retracting into the fin.
   d) Disconnect the electrical harness connector and the remove the screw and washer securing the ground wire to the upper fin rib and remove the fin tip.
5) At the bench:
   a) Remove the screws (4 ea.) securing the mounting bracket assembly to the fin tip.
   b) Break sealant around strobe light and fin tip and remove srobe light assembly from fin tip.
   c) Remove screws (4 ea.) securing lens/cover assembly to strobe light base and remove lens/cover assembly.
   d) Remove flashtube assembly from base/mounting bracket assembly by pulling the wiring harness through the base/mounting bracket assembly.
   e) Clean old sealant off fin tip, and, if it is to be reused, the lens/cover assembly.

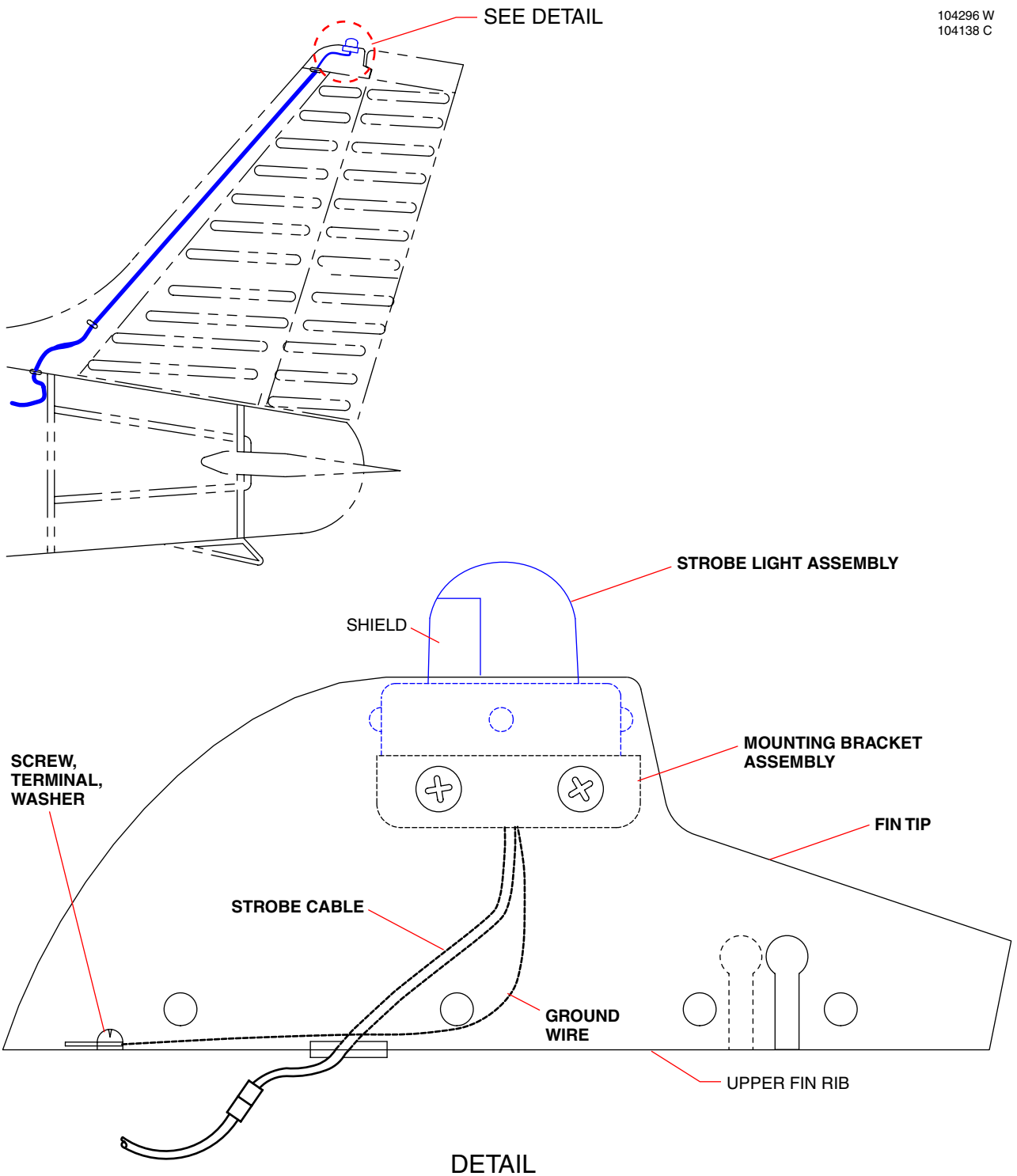
(b) Installation

1) At the bench:
   a) Feed the flashtube wiring harness through the base/mounting bracket assembly and position flashtube assembly on base/mounting bracket assembly.
   b) Position lens/cover assembly over flashtube assembly and strobe light base and secure with screws (4 ea.).
   a) Position strobe light assembly inside fin tip with lens protruding through top of fin tip.

      NOTE: Ensure shield is facing forward.

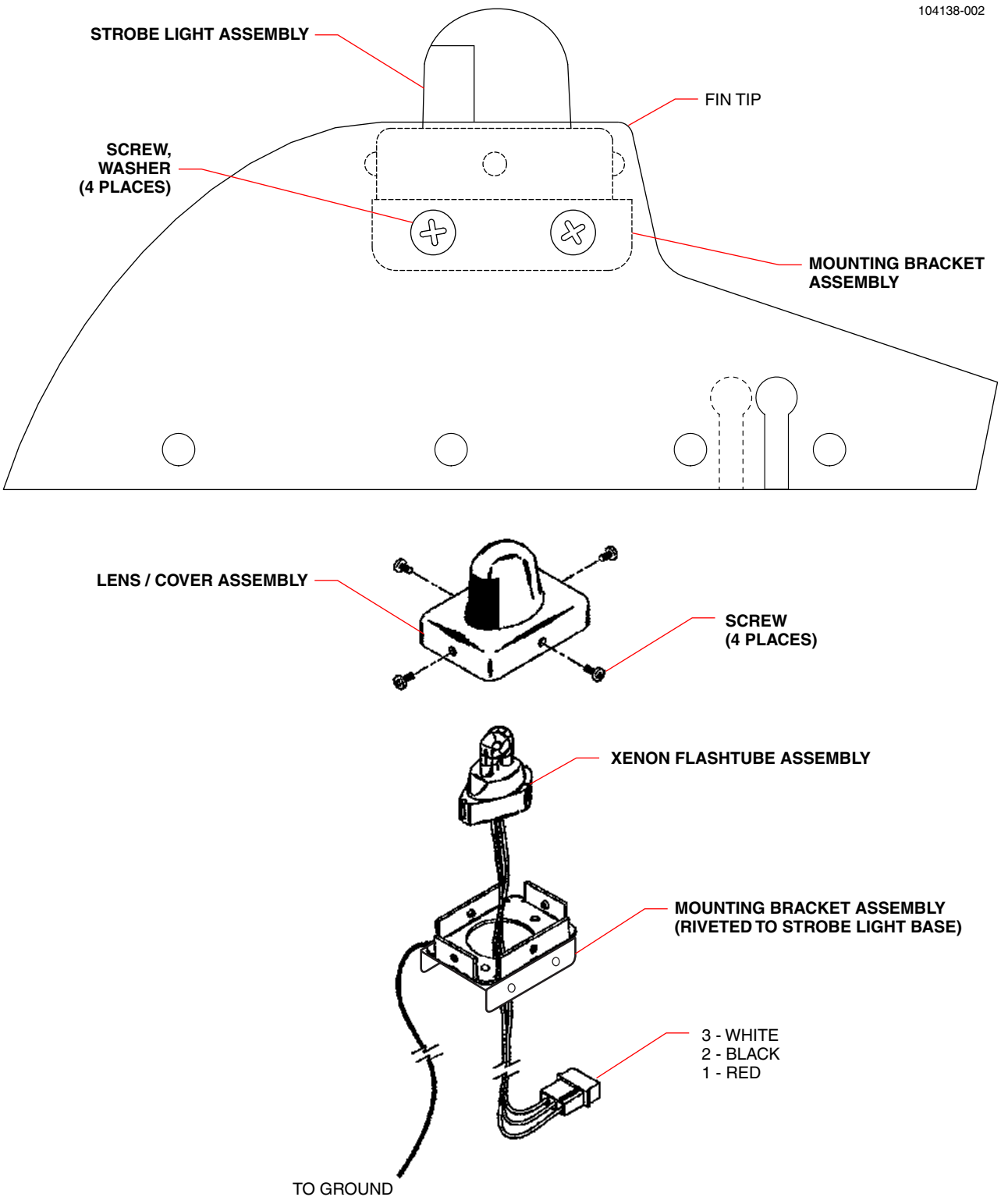
   b) Secure mounting bracket assembly to fin tip with screws (4 ea.).
   c) Fillet seal exterior joint between strobe light and fin tip. Use “Sealant, Airframe and Component” from Consumable Materials chart in 91-10-00.
2) At the fin tip:
   a) Connect the electrical harness connector and secure the ground wire to the upper fin rib with screw and washer.
   b) Release the wing harness and place the fin tip in position atop the fin.
   c) Secure with screws and washers.
3) Connect the negative (i.e., ground) battery cable.
4) Engage (push in) the ANTI-COLL circuit breaker.

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Effectivity
with Avidyne Entegra;
or, with Garmin G1000

Fin Tip Strobe Light Installation
Figure 14



Fin Tip Strobe Light Assembly
Figure 15

Effectivity
with Avidyne Entegra;
or, with Garmin G1000

5) Switch ON the BATT MASTR and STROB LIGHT switches. Verify the wing tip and fin tip strobes flash.
6) Switch the STROB LIGHT switch OFF and then to FIN. Verify the only fin tip strobe flashes.
7) Select STROB LIGHT switch OFF. Set (push in) NAV LIGHTS circuit breaker.
8) Select BATT MASTR switch OFF.
9) Secure airplane.

(3) Installation C

With Garmin G1000 NXi.

See “Figure 16” on page 334031 and “Figure 17” on page 334032.

(a) Removal

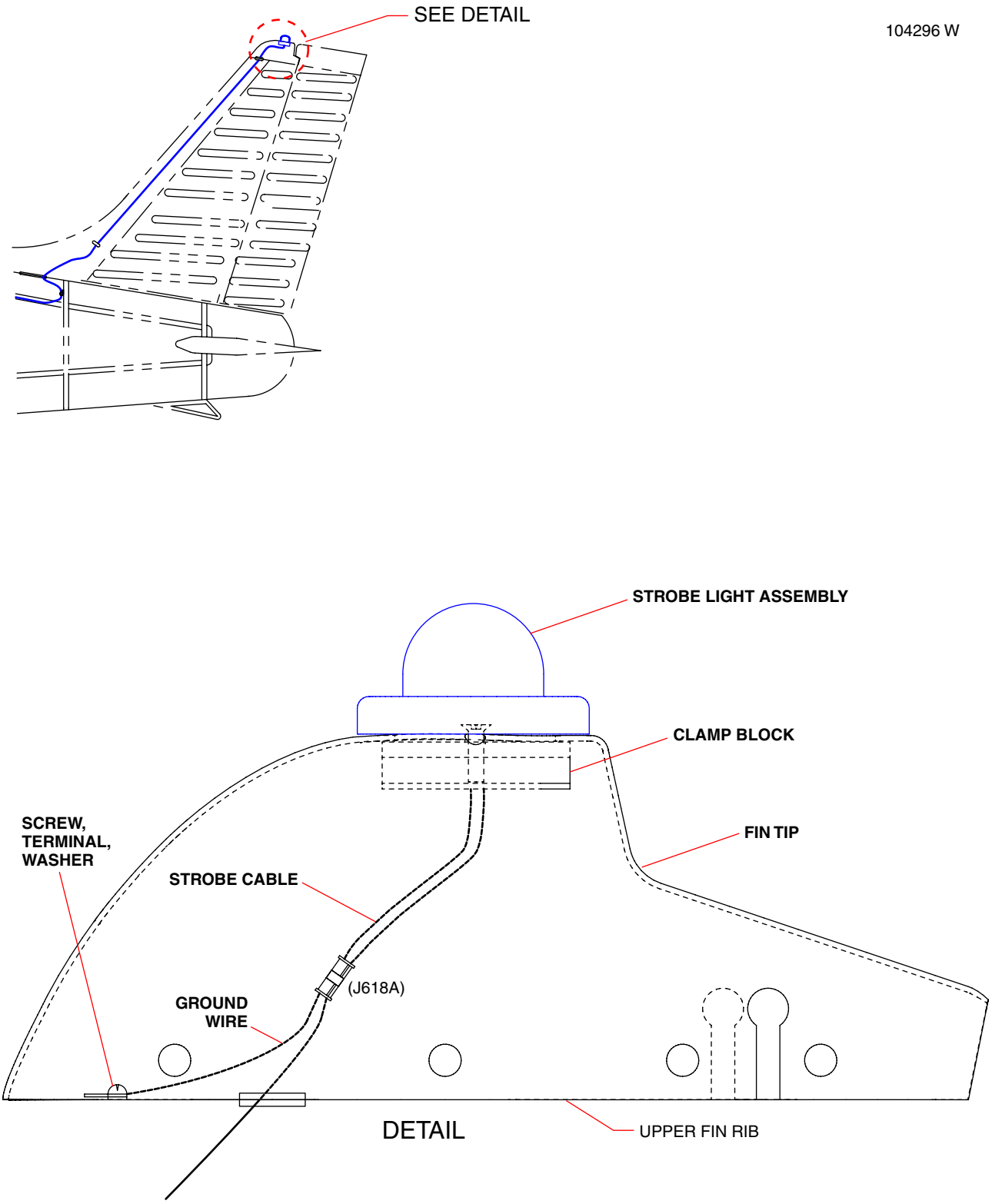
1) Ensure that the BATT MASTR, STROB LIGHT, and NAV LIGHT switches are OFF.
2) Disengage (pull out) NAV LIGHTS and ANTI-COLL circuit breakers.
3) Disconnect the negative (i.e., ground) battery cable.
4) At the fin tip:
   a) Remove screws and washers securing fin tip to vertical fin.
   b) Draw the fin tip away from the vertical fin until the electrical harness connector and ground wire are exposed.
   c) Capture the fin side of the harness to prevent its retracting into the fin.
   d) Disconnect the electrical harness connector and the remove the screw and washer securing the ground wire to the upper fin rib and remove the fin tip.
5) At the bench:
   a) Remove the screws (2 ea.) securing the strobe light assembly to the fin tip.
   b) Break sealant around strobe light base and fin tip and remove strobe light assembly from fin tip, pulling strobe light harness through hole in top of fin tip.
   c) Clean old sealant off fin tip, and, if it is to be reused, the strobe light base.

(b) Installation

1) At the bench:
   a) Feed the strobe light harness through the hole in fin tip.
   b) Position strobe light assembly atop fin tip and secure with screws (2 ea.).

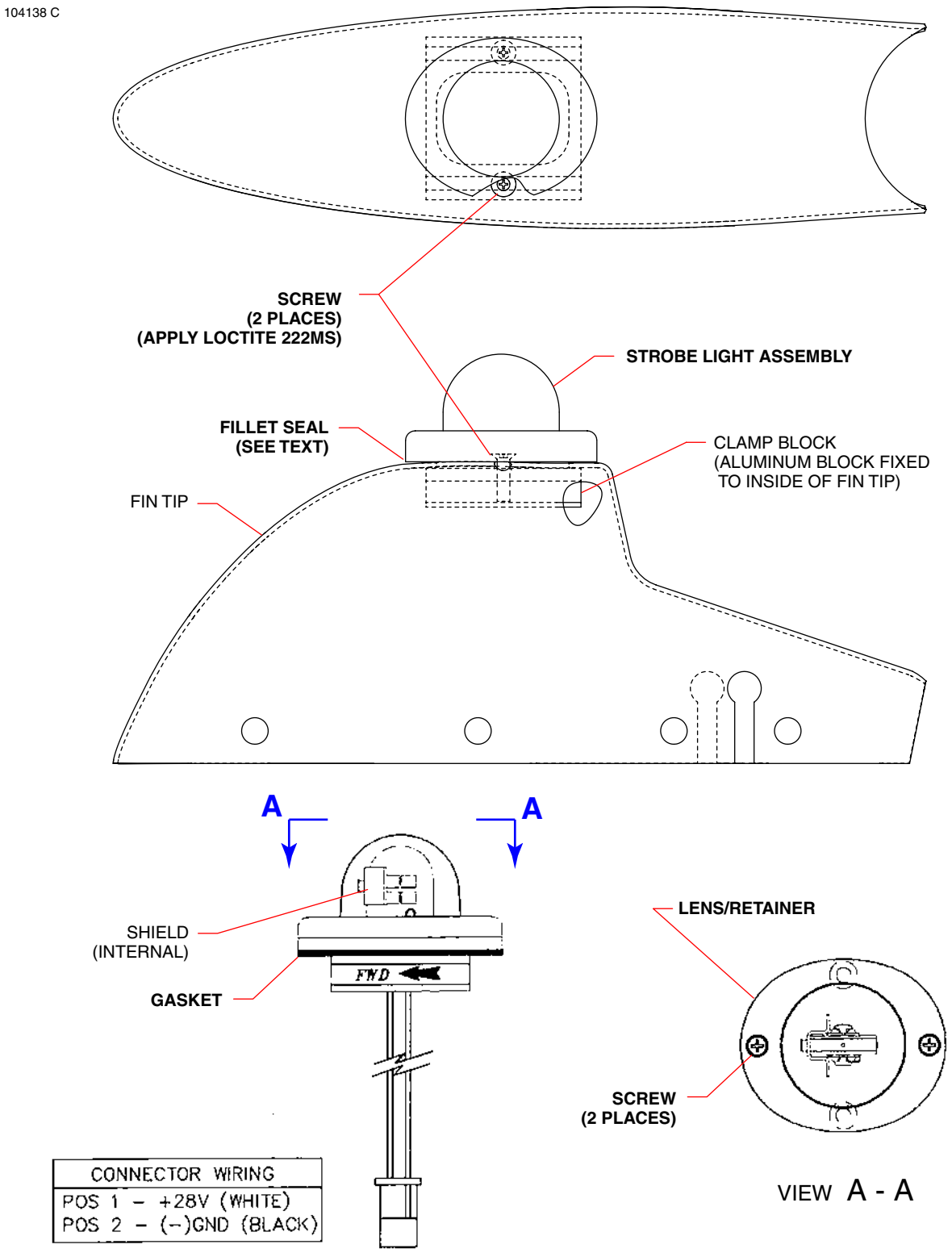
      NOTE: Ensure internal shield is facing forward.

   c) Fillet seal exterior joint between strobe light base and fin tip. Use “Sealant, Airframe and Component” from Consumable Materials chart in 91-10-00.
2) At the fin tip:
   a) Connect the electrical harness connector and secure the ground wire to the upper fin rib with screw and washer.
   b) Release the wing harness and place the fin tip in position atop the fin.
   c) Secure with screws and washers.
3) Connect the negative (i.e., ground) battery cable.
4) Engage (push in) the ANTI-COLL circuit breaker.
5) Switch ON the BATT MASTR and STROB LIGHT switches. Verify the wing tip and fin tip strobes flash.
6) Switch the STROB LIGHT switch OFF and then to FIN. Verify the only fin tip strobe flashes.



Fin Tip Strobe Light Installation
Figure 16

Effectivity
with Garmin G1000 NXi



Effectivity
with Garmin G1000 NXi

Fin Tip Strobe Light Assembly
Figure 17

7) Select STROB LIGHT switch OFF. Set (push in) NAV LIGHTS circuit breaker.
8) Select BATT MASTR switch OFF.
9) Secure airplane.

E. Strobe Power Supply

The strobe power supply is in the aft section of fuselage on the left side of the airplane mounted to a bracket fixed to the side and bottom of the fuselage.

S/N's 2843001–2843700, 2843702, 2843736 and up with Standard, Avidyne Entegra, or Garmin G500, or with Garmin G1000.

Refer to "Figure 18" on page 334034 and "Figure 19" on page 334035.

(1) Removal

(a) Ensure that the BATT MASTR and STROB LIGHT switches are in OFF position.

(b) Disengage (pull out) ANTI-COLL LIGHTS circuit breaker.

(c) Disconnect negative (i.e., ground) battery cable.

(d) Remove access panel to aft section of fuselage in the baggage compartment to gain access to the power supply.

(e) Note and mark location; then disconnect the strobe harness plugs.

(f) Remove screws and washers (4 ea.) holding power supply to bracket.

NOTE: Ground wires are connected to the screws adjacent to the strobe harness plugs. Note and mark their location as well.

(g) Remove power supply.

(2) Installation

(a) Position power supply in place and secure with screws and washers (4 ea.) being sure to capture the appropriate ground wires.

(b) Connect strobe harness plugs in proper place.

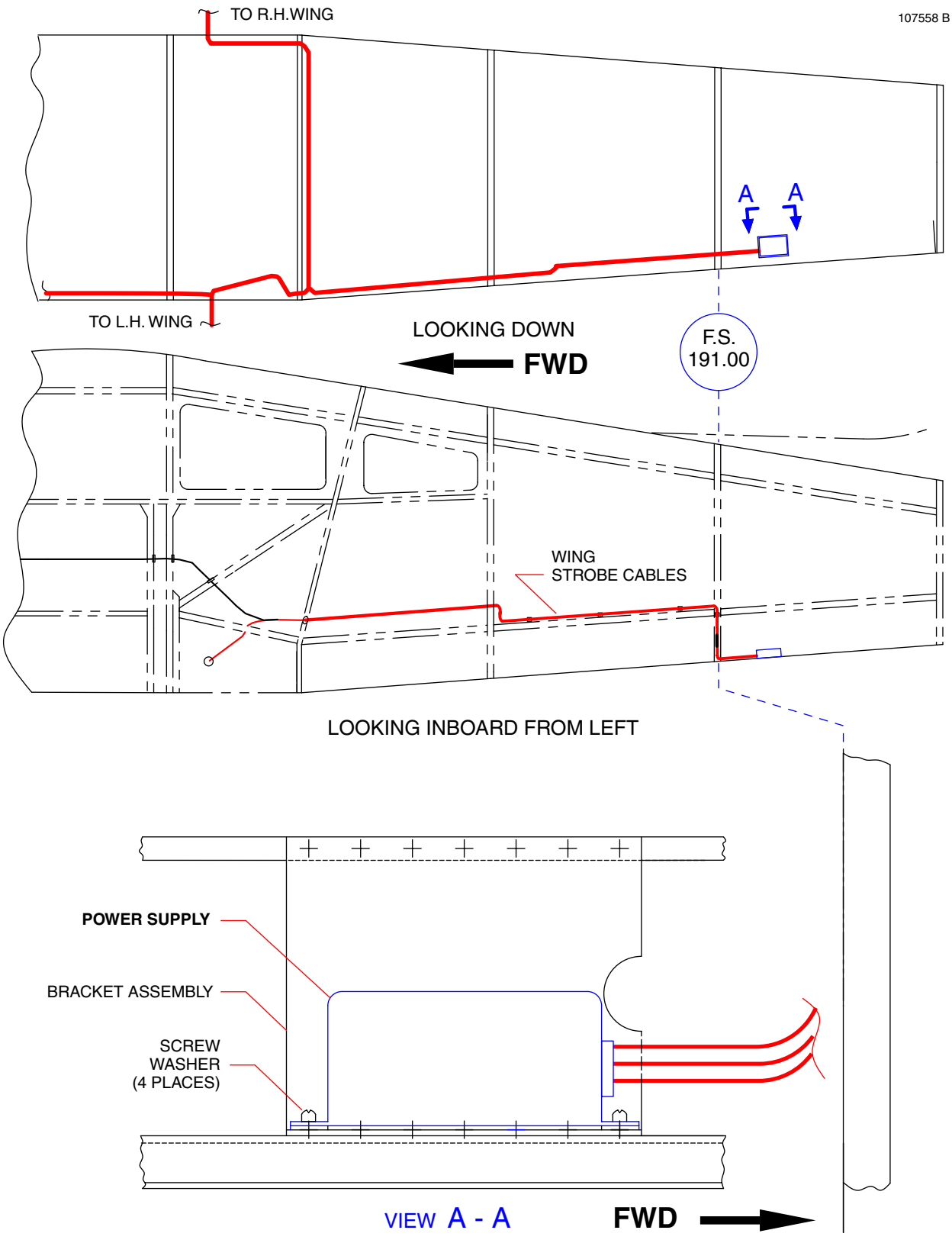
(c) Connect negative (i.e., ground) battery cable.

(d) Set (push in) ANTI-COLL LIGHTS circuit breaker.



Strobe Power Supply
Figure 18

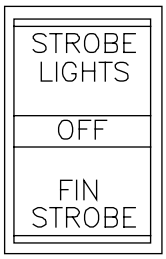
Effectivity
see page 334034



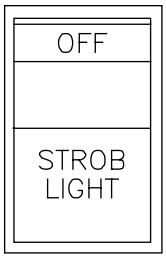
Strobe Power Supply Installation
Figure 19

Effectivity
see page 334034

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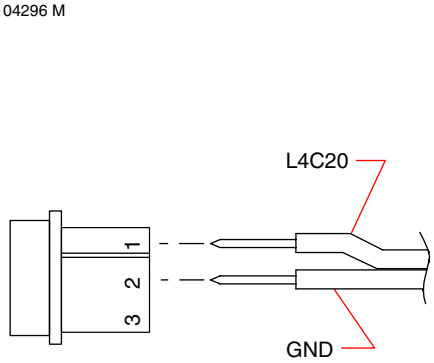


SWITCH FACE
IN AIRPLANES WITH
FIN STROBE OPTION INSTALLED

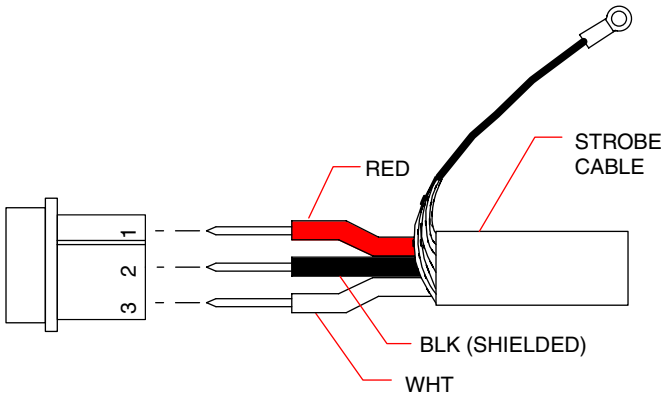


SWITCH FACE
IN AIRPLANES WITHOUT
FIN STROBE OPTION INSTALLED

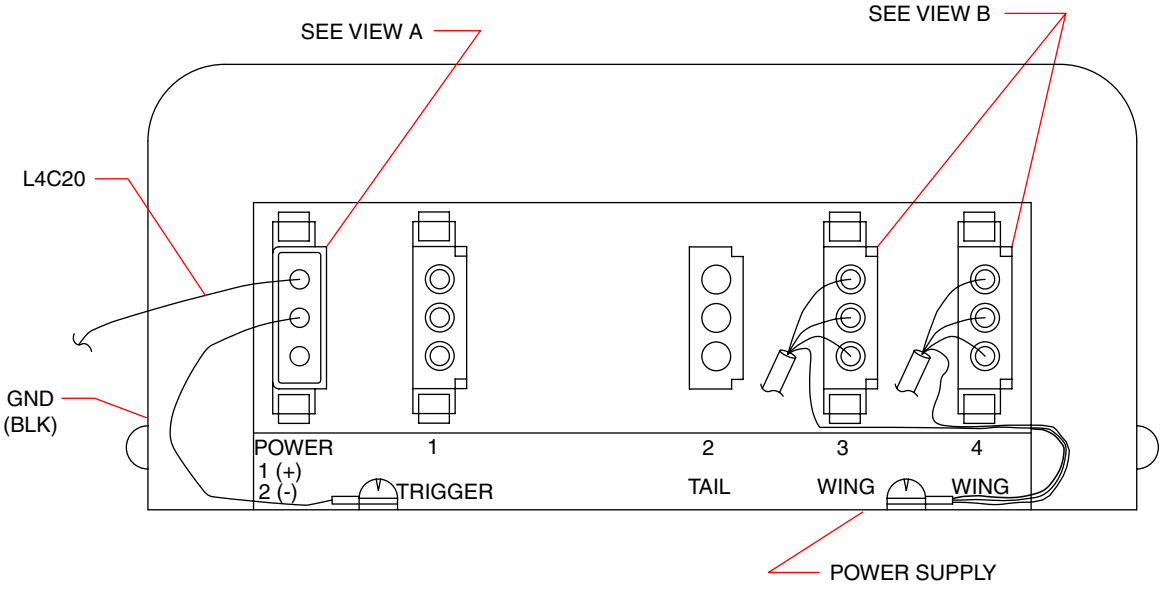
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VIEW A



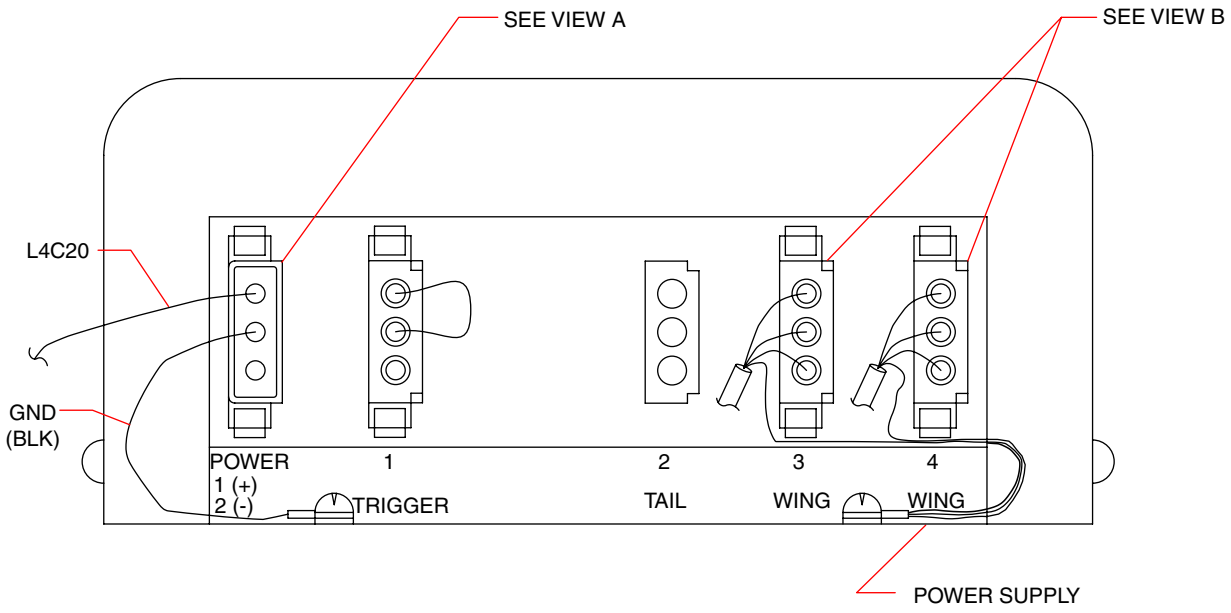
VIEW B



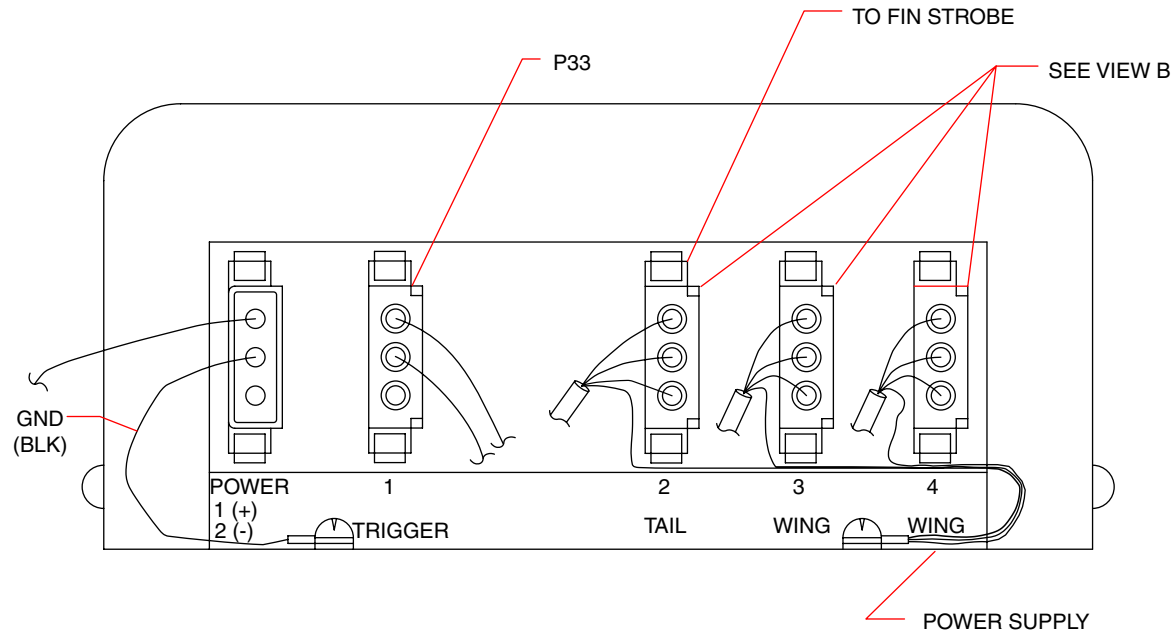
Effectivity
see page 334034

Strobe Power Supply Connections
Figure 20 (Sheet 1 of 2)

104296 M



STANDARD



WITH OPTIONAL FIN TIP STROBE

Strobe Power Supply Connections
Figure 21 (Sheet 2 of 2)

Effectivity
see page 334034

(e) Verify lights are working by selecting BATT MASTR and STROB LIGHT switches ON and checking that both left and right strobe lights flash. If fin tip strobe is installed verify its operation as well.

(f) Select BATT MASTR and STROB LIGHT switches OFF.

(g) Replace access panel in baggage compartment and secure airplane.

CHAPTER

34

NAVIGATION

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CHAPTER 34

LIST OF EFFECTIVE PAGES

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	4	Jul 30/21		4	Mar 6/14
	5	Jul 30/21		5	Nov 15/17
	6	Jul 30/21		6	Nov 15/17
	7	Jul 30/21		7	Nov 15/17
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	10	Jul 30/21	34-22-00	1	Nov 15/17
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	13	Jul 30/21		4	Nov 15/17
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	8	Mar 15/20		7	Nov 15/17
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GENERAL

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Standard

   S/N's 2843001 and up.

   These airplanes are equipped with conventional air, vacuum, and mechanically driven flight instruments as standard equipment. Most instruments are face mounted.

   Removal and installation instructions for the standard instruments are given in 39-10-00.

2. Avidyne Entegra IFDS

   Optional in S/N's 2843598 thru 2843682. See "Integrated Flight Display System" on page 34221.

   An Integrated Flight Display System (IFDS) (i.e., Avidyne Entegra) was available as an option in S/N's 2843598 thru 2843682. The IFDS installation consists of these components: Primary Flight Display (PFD), Multifunction Display (MFD), Data Acquisition Unit (DAU) and associated sensors, and Magnetometer/OAT Sensor Assembly.

   When installed, the Avidyne IFDS is the primary flight and navigation instrumentation. IFDS uses two 10.4-inch diagonal, high-resolution, sunlight-readable, full-color displays (PFD and MFD) and provides primary flight and engine information. Standard primary flight instruments are installed for redundancy. (See "Figure 1" on page 34221.)

   Line maintenance of this system is limited to basic troubleshooting which is addressed in 34-22-00.

   Standby altitude and airspeed indicators and a magnetic compass are also installed as secondary, independent sources of altitude, airspeed, and (non-stabilized) heading information. An electric attitude indicator is installed to provide an independent source for attitude information.

   Line maintenance of this system is limited to basic troubleshooting and that is addressed in 34-20-00.

3. Garmin G500 Flight Display System

   Optional in S/N's 2843676 and up.

   The Garmin G500 flight display system is available as an option. When ordered, it is installed by STC and Instructions for Continued Airworthiness (ICA) are the sole responibility of the STC holder (i.e., Garmin). References to the G500 system in this manual typically address how it interfaces with the Piper airframe and Piper provided components.

4. Garmin G1000 IAS

Optional in S/N's 2843701, 2843703–2843822, 2843824–2843851, and 2843853. See "Integrated Avionics System - Garmin G1000" on page 3425011.

The Garmin G1000 Integrated Avionics System (IAS) has a two-panel configuration with two (2) GDU 104X 10.4-inch LCD panels as the Primary Flight Display (PFD) and Multi-Function Display (MFD) which are functionally interchangeable. Functions by the system include flight instrumentation, position, navigation, communication, and identification information. Cobham (S-TEC) 55X Autopilot system is optional. The system consists of the following Line Replaceable Units (LRUs):

- GDU 104X Display Units (1 ea. PFD and MFD)
- GMA 1347 Audio Panel with Integrated Marker Beacon Receiver
- GIA 63W Integrated Avionics Unit (2 ea.)
- GEA 71 Engine/Airframe Unit
- GTX 33 (ES) Mode S Transponder
- GDC 74A Air Data Computer
- GTP 59 OAT Probe
- GRS 77 Attitude and Heading Reference System
- GMU 44 Magnetometer
- GDL 69A Data Link Unit (optional)
- GRT 10/GRC 10 Wireless Remote System (optional)
- GTS 800 Traffic Advisory System (optional)

An Aspen Avionics EBD is installed as a Standy Flight Display system.

5. Garmin G1000 NXi IAS

Optional in S/N's 2843852, 2843854–2843999. See "Integrated Avionics System - Garmin G1000 NXi" on page 3425021.

The Garmin G1000 Integrated Avionics System has a two-panel configuration with two (2) GDU 105X 10.4-inch LCD panels as the Primary Flight Display (PFD) and Multi-Function Display (MFD) which are functionally interchangeable. Functions by the system include flight instrumentation, position, navigation, communication, and identification information. Cobham (S-TEC) 55X Autopilot system is optional. The system consists of the following Line Replaceable Units (LRUs):

- GDU 105X Display Units (1 ea. PFD and MFD)
- GMA 1347 Audio Panel with Integrated Marker Beacon Receiver
- GIA 63W Integrated Avionics Unit (2 ea.)
- GEA 71 Engine/Airframe Unit
- GTX 335R Transponder (standard)
- GTX 345R Transponder (optional)
- GSU 75 Air Data Attitude and Heading Reference System (ADAHRS)
- GTP 59 OAT Probe
- GMU 44 Magnetometer
- GDL 59 (and antenna) Flight Parameter Recorder with Wi-Fi (optional)
- GDL 69eA Data Link Unit (optional)
- GRT 10/GRC 10 Wireless Remote System (optional)
- FS 510 Bluetooth and Wi-Fi Connectivity (optional)
- GTS 800 Traffic Advisory System (optional)

An Aspen Avionics Evolution Backup Display (EBD) is installed as a Standy Flight Display system.

6. Garmin G1000 NXi IAS

Optional in S/N's 2843823, 2881001 and up. See "Integrated Avionics System - Garmin G1000 NXi" on page 3425031.

The Garmin G1000 NXi Integrated Avionics System is installed in a two-panel configuration with two (2) GDU 105X 10.4-inch LCD panels as the Primary Flight Display (PFD) and Multi-Function Display (MFD) which are functionally interchangeable. Functions by the system include flight instrumentation, position, navigation, communication, and identification information. Garmin GFC 700 AFCS is optional. The system consists of the following Line Replaceable Units (LRUs):

- GDU 105X Display Units (1 ea. PFD and MFD)
- GMA 1360 Audio Selector panel
- GIA 64/64W Integrated Avionics Unit (2 ea.)
- GEA 71B Engine/Airframe Unit
- GTX 335R Transponder (standard)
- GTX 345R Transponder (optional)
- GSU 75 Air Data Attitude and Heading Reference System (ADAHRS)
- GTP 59 OAT Probe
- GMU 44/44B Magnetometer
- GDL 59 (and antenna) Flight Parameter Recorder with Wi-Fi (optional)
- GDL 69eA Data Link Unit (optional)
- GRT 10/GRC 10 Wireless Remote System (optional)
- FS 510 Bluetooth and Wi-Fi Connectivity (optional)
- GTS 800 Traffic Advisory System (optional)

One of two Standby Instrument Displays is installed as standard equipment:

- Aspen Avionics Evolution Backup Display (EBD); or,
- Garmin G5 Electronic Flight Instrument (EFI).

FLIGHT ENVIRONMENT DATA

1. Pitot and Static System

Refer to "Figure 1" on page 34102.

A. Description

The pitot air system consists of a pitot mast on the underside of the left wing with its related plumbing. Impact air pressure entering the pitot is transmitted from the pitot inlet through hose and tubing routed through the wing to the airspeed indicator (and, if installed, to the Primary Flight Display (PFD) of the Avidyne Entegra or Garmin G1000/G1000 NXi system) on the instrument panel. A partially or completely blocked pitot head will give an erratic or a zero instrument reading.

The static air system consists of interconnected static ports on the underside and aft side of the (pitot) static mast. The ports are connected to the airspeed indicator, altimeter, and vertical speed indicator (and to the PFD of the Avidyne Entegra or the Air Data Computer in the Garmin G1000 system, if installed) by hose and tubing routed through the wing. An alternate static air source is below the instrument panel in front of pilot. The alternate static source is part of the standard system and has a shutoff valve to close the port when not needed. Pitot and static lines are drained through separate drain valves on left lower side of fuselage interior.

NOTE: Aircraft equipped with either Avidyne Entregra or Garmin G1000/G1000 NXi have standby instruments/displays connected to the pitot/static system.

B. Test

(PIR-PPS60035, Rev AD.)

CAUTION: FOR AIRPLANES EQUIPPED WITH GARMIN G1000/G1000 NXI, ENTER “MAINTENANCE MODE” WHEN ACTIVATING THE SYSTEM ON THE GROUND TO CONDUCT TESTS, INSPECTIONS, ETC. SEE MAINTENANCE MODE UNDER 34-25-01, 34-25-02, OR 34-25-03, AS APPROPRIATE.

This test requires a pitot/static test fixture (i.e., Aerosonic Air Data Test Set - Model 90000-0168 or equivalent) and calibrated air source (i.e., airspeed simulator) and should be performed at any time an instrument, fitting, line, pitot head, or static button is disconnected. The test should be performed prior to the next flight.s

NOTE: For airplanes equipped with Garmin G1000/G1000 NXi, provide external power.

NOTE: For airplanes equipped with Garmin G1000/G1000 NXi, see GDC 74A (or GSU 75) Tests, under LRU Test Procedures, in 34-25-01, 34-25-02, or 34-25-03 as appropriate, before beginning this test.

NOTE: Ensure the lines and fittings are free of any entrapped moisture or restrictions.

(1) Attach the test fixture to the pitot head. Align the holes in the fixture with the holes in the head.

(2) Attach the airspeed simulator hose to the pitot (pressure) port of the fixture.

(3) When equipped with Avidyne Entegra, Garmin G500, Garmin G1000, or G1000 NXi, turn ON the PFD.

(4) Operate the simulator to obtain a reading of 75 knots on the airplane airspeed indicator(s).

(5) Check that the airspeed indicator needle(s) – (PFD airspeed for Garmin displays) – follow(s) in the same direction as the simulator airspeed indicator needle.

(6) Raise airspeed to 154 knots and wait 15 seconds to allow the airplane airspeed indicator(s) to stabilize.

(7) Observe the simulator and airplane airspeed indicators for 15 seconds. If a leak is present, the indicator needles/display will move toward zero. Maximum acceptable leak rate is one knot in 15 seconds.

(8) If a leak exceeds one knot in 15 seconds, check the fixture installation, hose connections, and pitot system lines and fittings. Repair the leak when found, then repeat steps (1) - (7), above.

(9) Steps for aircraft equipped with Avidyne Entegra or Garmin G1000/G1000 NXi system only:

(a) Operate the simulator, to indicate on the PFD, until the simulator indicates 140 knots.

(b) Verify that the PFD airspeed indicates within ± 3 knots of the simulator indication.

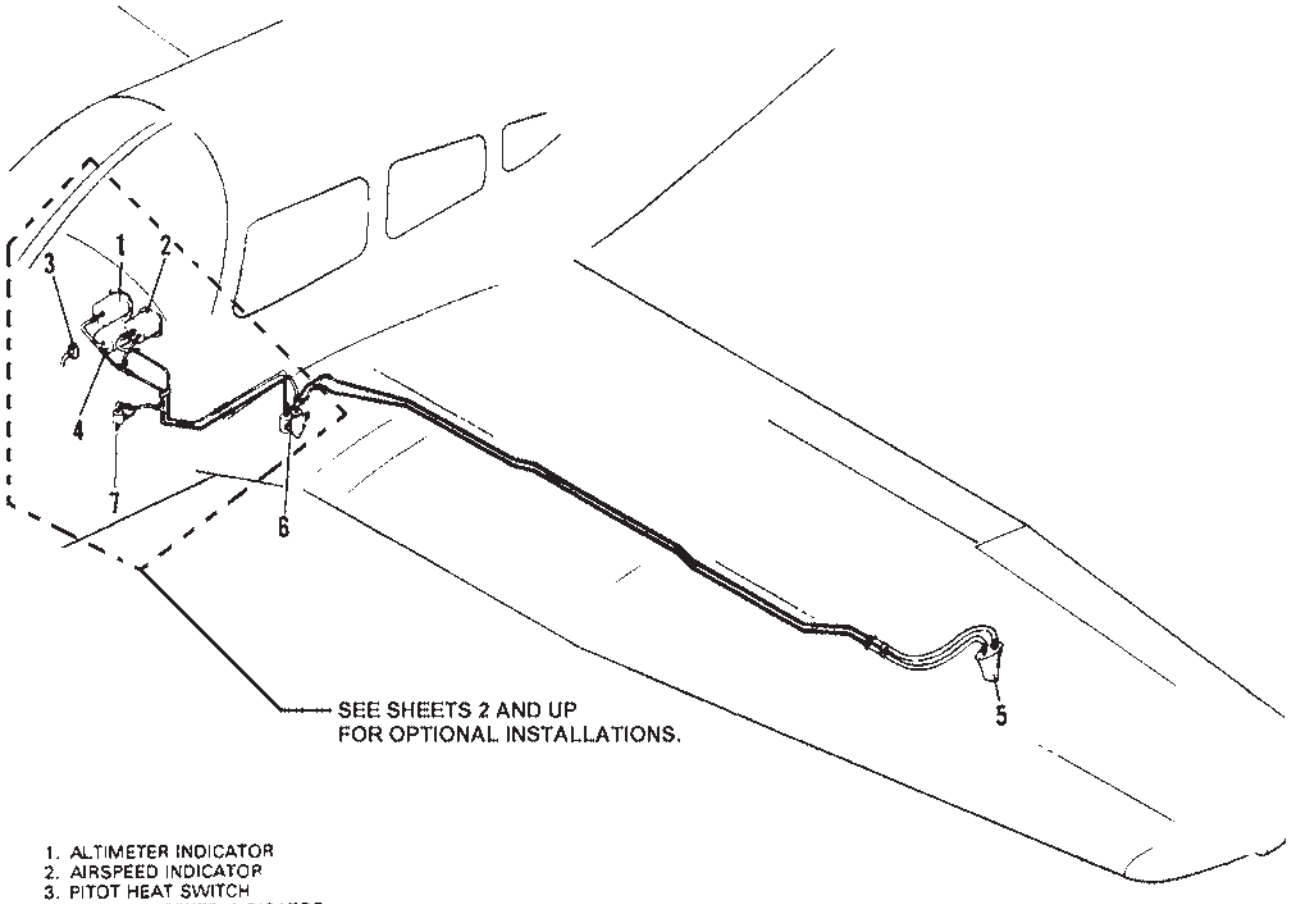
(c) On G1000/G1000 NXi equipped airplanes only: Operate the simulator, to indicate on the pilot PFD, until the overspeed alarm(s) (PFD data and audible) first activate. Verify that the simulator indicates within 154–157 knots.

(d) Increase the simulator airspeed indication to 167 knots.

(e) Decrease the simulator airspeed indication until the alarm(s) first stop. Verify that the simulator indicates an airspeed less than 154 knots.

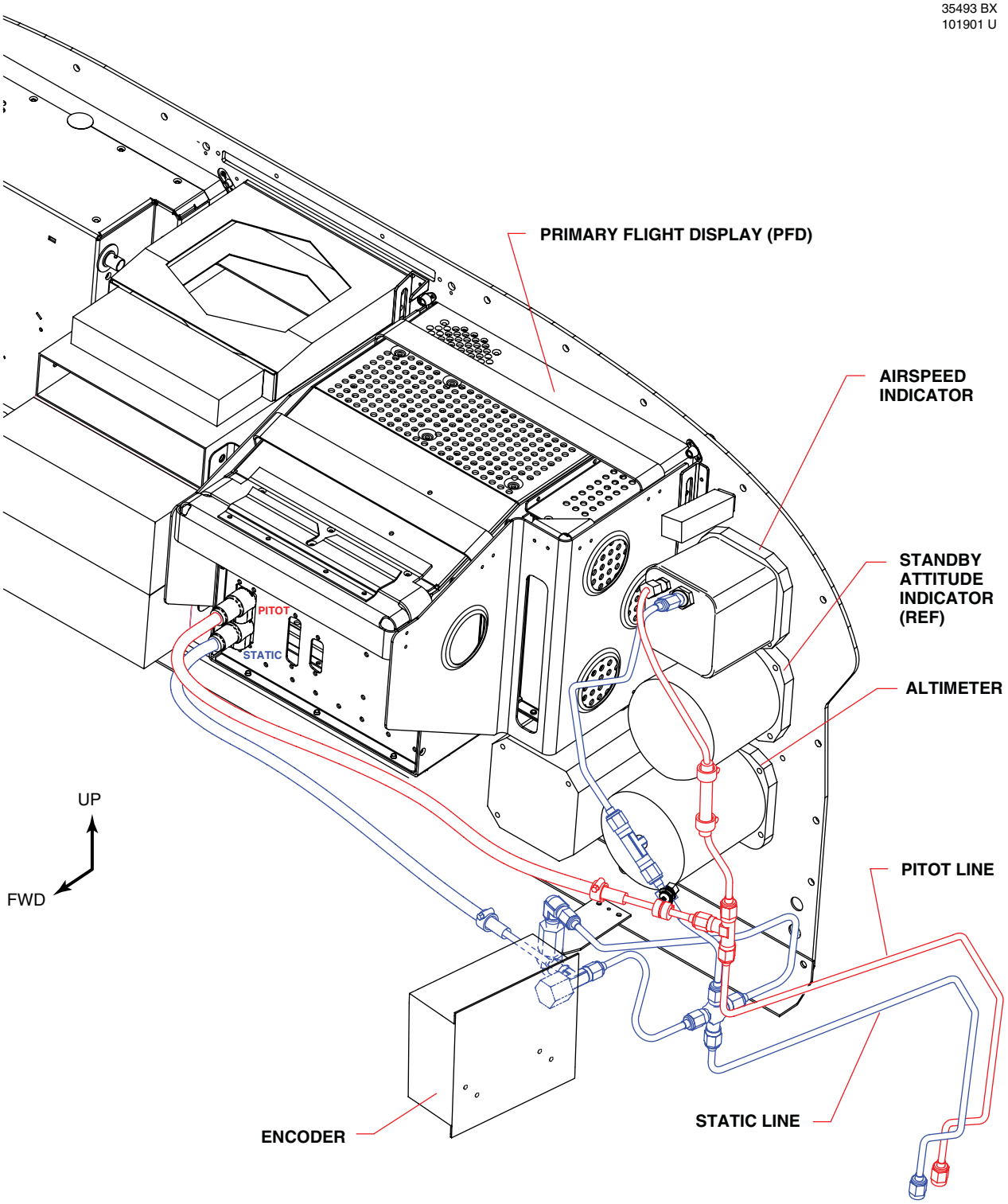
(f) On G1000 equipped airplanes only: Pull the PFD circuit breaker.

(g) On G1000/G1000 NXi equipped airplanes only: Reset (push-in) the PFD circuit breaker.



Pitot/Static System Installation
Figure 1 (Sheet 1 of 6)

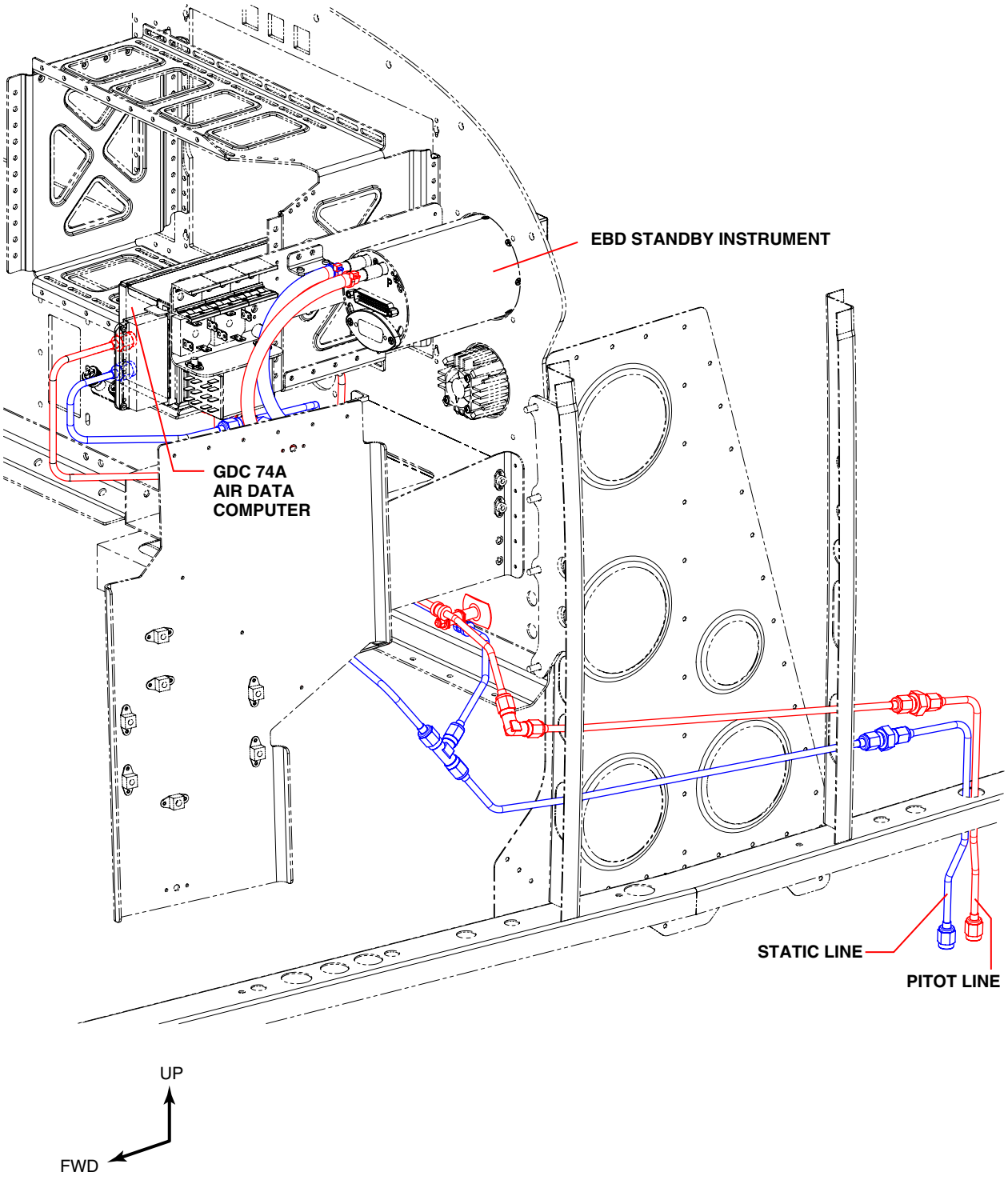
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Pitot/Static System Installation
Figure 1 (Sheet 2 of 6)

Effectivity
with Avidyne Entegra

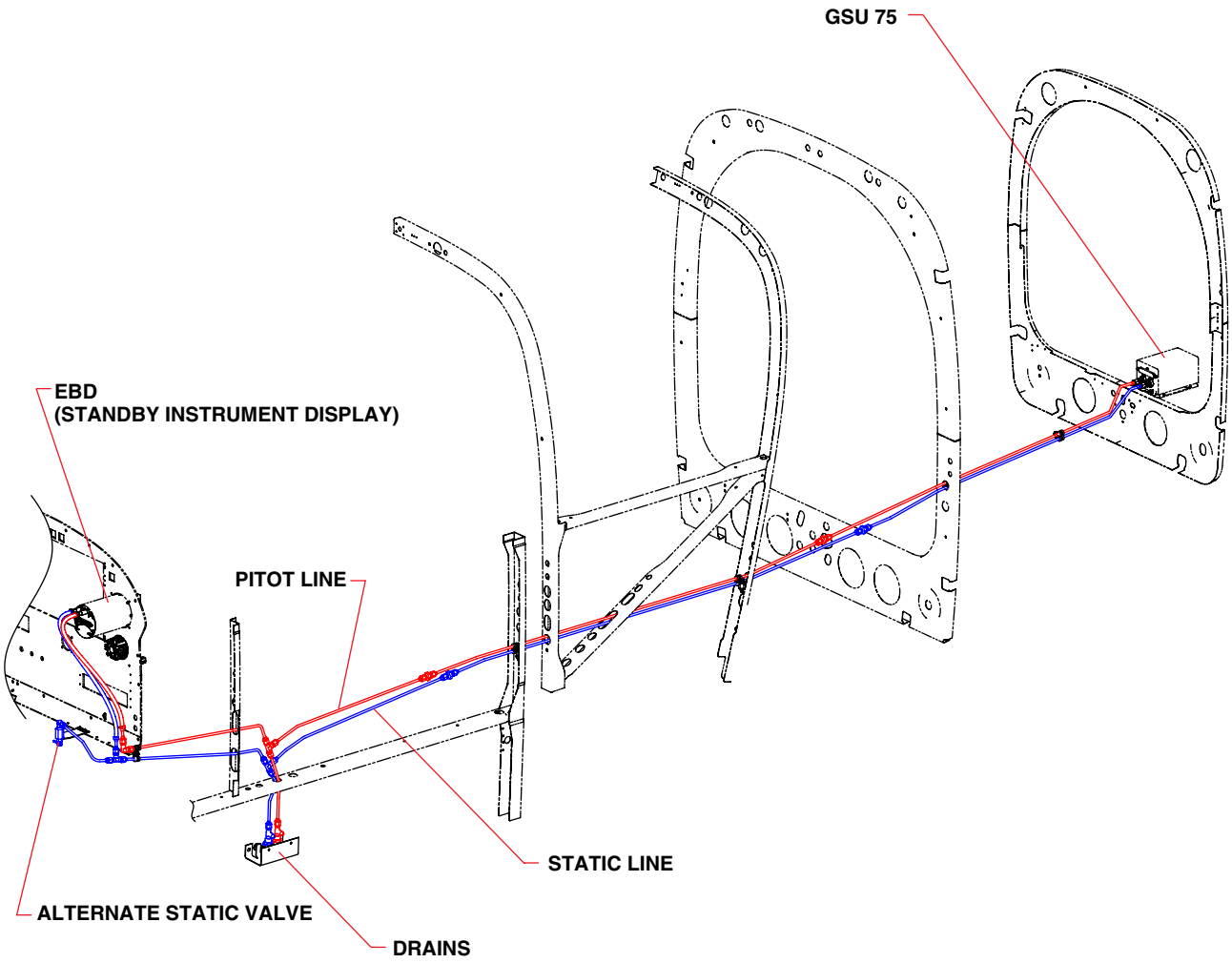
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Effectivity
with Garmin G1000

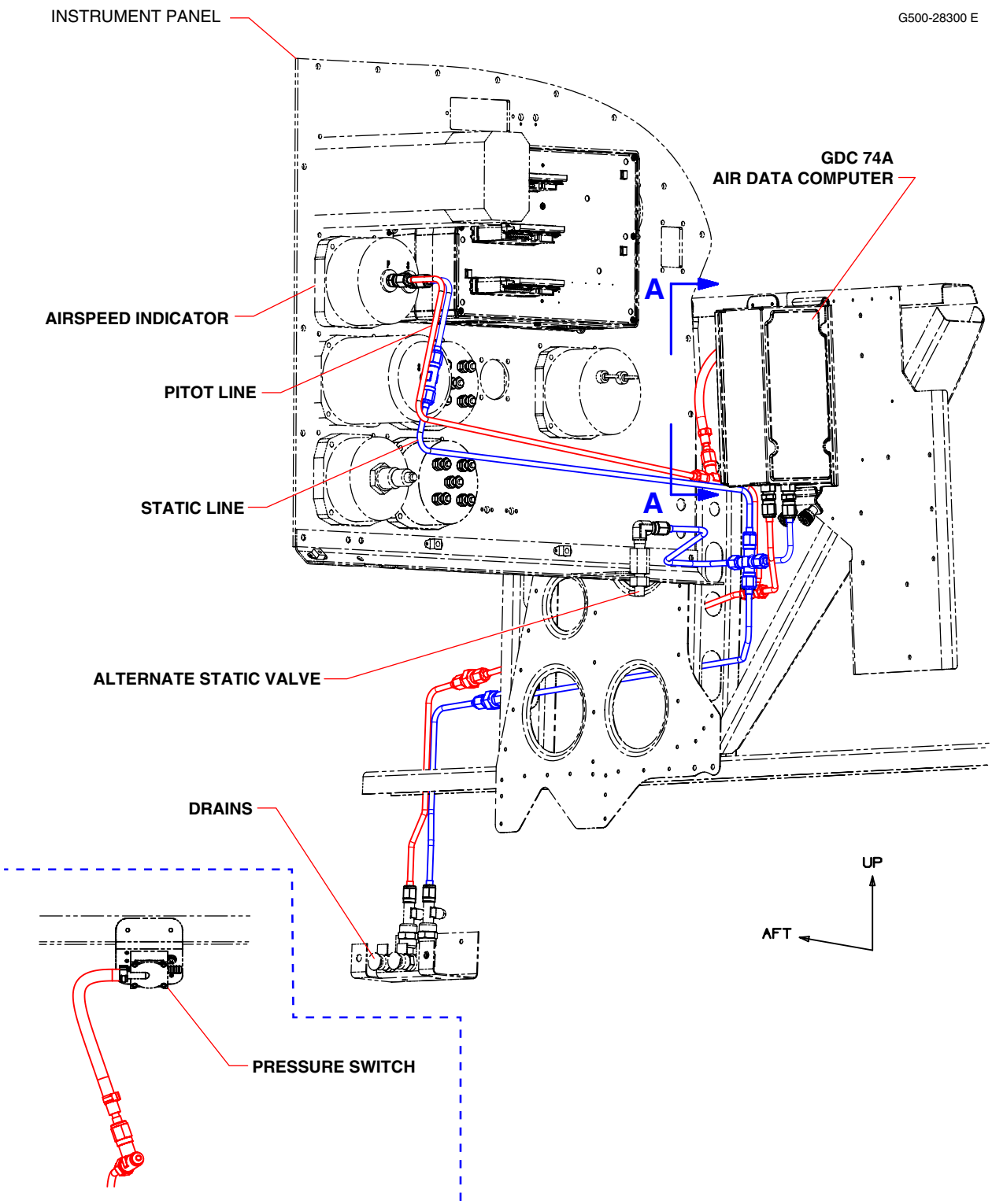
Pitot/Static System Installation
Figure 1 (Sheet 3 of 6)

106980 B



Pitot/Static System Installation
Figure 1 (Sheet 4 of 6)

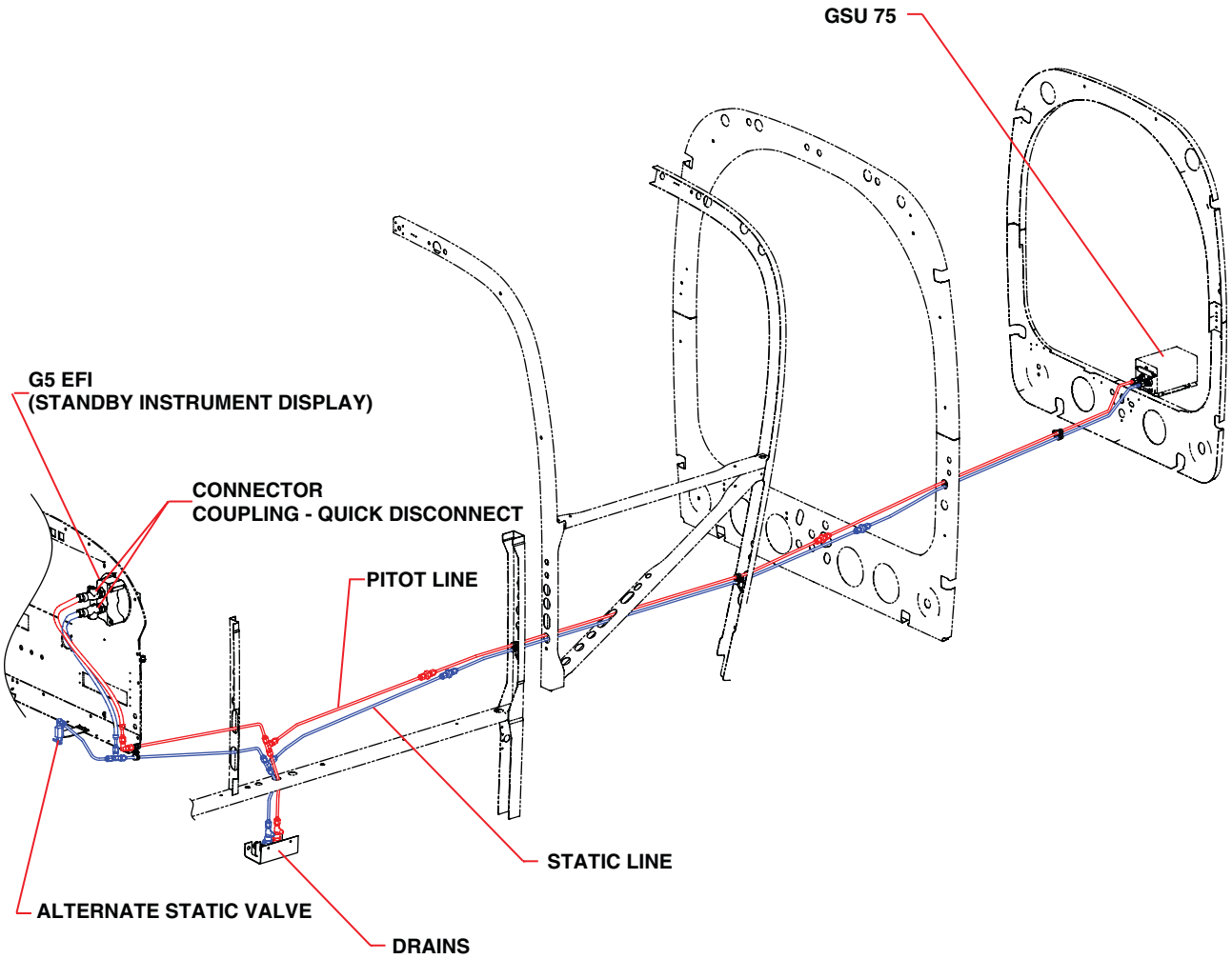
Effectivity
with Garmin
G1000 NXi and Aspen EBD



Effectivity
with Garmin G500

Pitot/Static System Installation
Figure 1 (Sheet 5 of 6)

106980 2



Pitot/Static System Installation
Figure 1 (Sheet 6 of 6)

Effectivity
with Garmin
G1000 NXi and G5 EFI

(10) Additional testing for aircraft equipped with optional airtime hour meter.

(a) Note the exact value on the airtime hour meter, including tenths of hours. Operate the simulator to indicate 60 knots on the airspeed indicator.

(b) Maintain indicated 60 knots for a minimum of 6 minutes.

(c) Verify that the time on the airtime hour meter has increased approximately 0.1 hours.

(11) Remove the test fixture from the pitot head.

(12) Attach static test fixture to the static port and pitot test port of the aircraft. Tape over the other static button.

(13) When equipped with Avidyne IFDS, Garmin G500, or Garmin G1000/G1000 NXi, verify the PFD is ON.

(14) Set the aircraft altimeter needles to read zero altitude. Operate the static simulator to cause the aircraft altimeter needles/display to read 1,000 feet altitude. Momentarily open the alternate static port. There should be a decrease in altimeter indication. If no change occurs the system is blocked and must be repaired prior to further testing.

(15) Increase altitude to 1,050 feet.

(16) Check that the aircraft altimeter shows an increase.

(17) Observe the aircraft altimeter. Loss of indicated altitude shall not exceed 100 feet in one minute.

(18) If a leak exceeds the above tolerance, check the fixture installation, plumbing and fittings. Repair the leak when found and repeat the static system checks above.

(19) When equipped with the optional Avidyne IFDS, Garmin G500, or Garmin G1000/G1000 NXi, turn OFF the PFD.

(20) Remove the test fixture. Remove tape from static button. When equipped with Garmin G1000/G1000 NXi remove the external power source.

2. Vertical Speed Indicator

NOTE: In airplanes equipped with Avidyne Entegra or Garmin G1000/G1000 NXi, this function is performed solely by the Primary Flight Display (PFD).

A. Description

The vertical speed indicator measures rate of change in static pressure when the airplane is climbing or descending. A pointer and dial will show the airplane's rate of climb or descent in feet per minute. Due to instrument lag, the airplane will be climbing or descending before the instrument starts to indicate. The instrument will continue to indicate a climb or descent after the airplane is level. Thus, the instrument is unreliable in rough air. This lag is not considered a malfunction.

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot / Static System, "Test" on page 34101.

B. Troubleshooting

See "Chart 1" on page 34109.

C. Removal and Installation

See 39-10-00.

CHART 1
TROUBLESHOOTING VERTICAL SPEED INDICATOR

Trouble	Cause	Remedy
Pointer does not set on zero when airplane is in accelerated level flight.	Aging of diaphragm.	Use set screw to reset pointer on zero. Tap instrument gently while resetting.
Pointer fails to respond.	Obstruction in static line.	Disconnect all instruments connected to static line. Clear line.
	Static vents frozen over.	Check individual instrument for obstruction in lines.
	Water in static line.	
	Obstruction in pitot head.	Clean lines and head.
Pointer oscillates.	Leaks in static lines.	Disconnect all instruments connected to static line.
		Check individual instruments for leaks. Connect instruments to static line and test installation for leaks.
	Defective mechanism.	Replace instrument.
Vertical Speed reads when airplane is banked.	Water in static line.	Disconnect static lines and blow out lines from cockpit out to pitot head.
Pointer has to be set before each flight.	Temperature compensator inoperative.	Replace instrument.
Pointer cannot be reset to zero.	Diaphragm distorted.	Replace instrument.
Instrument reads much less than actual rate during climb or descent.	Instrument case broken or leaking.	Replace instrument.

3. Altimeter

NOTE: For airplanes equipped with Avidyne Entegra or Garmin G1000/G1000 NXi, the primary altitude indicator is the Primary Flight Display (PFD).

A. Description

The altimeter indicates pressure altitude in feet above sea level. The indicator has three pointers and a dial scale; the long pointer is read in hundreds of feet, the middle pointer in thousands of feet, and the short pointer in ten thousands of feet. A barometric pressure window is on the right side of indicator dial and is set by a knob on lower left corner of instrument. The altimeter is a sealed diaphragm connected to the pointers through mechanical linkage. The diaphragm mounting is made of bi-metallic temperature sensitive strips that compensates for variations from standard temperatures. The instrument case is vented to static air system and, as static air pressure decreases, the diaphragm expands, causing the pointers to move through mechanical linkage.

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot / Static System, "Test" on page 34101.

B. Troubleshooting

See "Chart 2" on page 341011.

C. Removal and Installation

See 39-10-00.

CHART 2
TROUBLESHOOTING ALTIMETER

Trouble	Cause	Remedy
Excessive scale error.	Improper calibration adjustment.	Replace instrument.
Excessive pointer oscillation.	Defective mechanism.	Replace instrument.
High or low reading.	Improper venting.	Eliminate leak in static pressure system and check alignment of pitot/static mast.
Setting knob is hard to turn.	Wrong lubrication or lack of lubrication.	Replace instrument.
Inner reference marker fails to move when setting knob is rotated.	Out of engagement.	Replace instrument.
Setting knob set screw loose or missing.	Not tightened when altimeter was reset.	Tighten instrument screw, if loose.Replace instrument, if screw is missing.
Cracked or loose cover glass.	Case gasket hardened.	Replace instrument.
Dull or discolored markings.	Age.	If condition is undesirable, replace instrument.
Barometric scale and reference markers out of synchronization.	Slippage of mating parts.	Replace instrument.
Barometric scale and reference markers out of synchronization with pointers.	Drift in mechanism.	Refer to the latest revision of AC43.13-1.
Altimeter sticks at altitude or does not change with altitude change.	Water or restriction in static line.	Remove static lines from all instruments, blow line clear from cockpit to pitot/static mast.
Altimeter charges reading as aircraft is banked.	Water in static line.	Remove static lines from all instruments, blow line clear from cockpit to pitot/static mast.
Altimeter requires resetting frequently.	Temperature compensator inoperative.	Change instrument.

4. Airspeed Indicator

NOTE: For airplanes equipped with Avidyne Entegra or Garmin G1000/G1000 NXi, the primary altitude indicator is the Primary Flight Display (PFD).

A. Description

The airspeed indicator reads the speed of the airplane passing through the air. Airspeed indication is the differential pressure reading between pitot air pressure and static air pressure. The instrument has the diaphragm vented to the pitot air source. The case is vented to static air system. As airplane increases speed, pitot air pressure increases, causing diaphragm to expand. A mechanical linkage moves the pointer to show speed . The instrument dial is calibrated in knots and miles per hour, and has operating range markings for safe operation of the airplane.

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot / Static System, "Test" on page 34101.

B. Troubleshooting

See "Chart 3".

C. Removal and Installation

See 39-10-00.

CHART 3
TROUBLESHOOTING PITOT/STATIC TUBES AND AIRSPEED INDICATOR

Trouble	Cause	Remedy
Pointers of all instruments using static pressure do not indicate properly.	Leak in instrument case or in static lines.	Check for leak(s) and seal.
Pointer of instrument oscillates.	Defective mechanism.	Replace instrument.
Instrument reads high.	Pointer not on zero.	Replace instrument.
	Leaking static system.	Find leak and correct.
Instrument reads low.	Pointer not on zero.	Replace instrument.
	Leaking static system.	Find leak and correct.
	Pitot head not aligned correctly.	Realign pitot head.
Airspeed changes as aircraft is banked.	Water in pitot line.	Remove static lines from all instruments and blow line clear from cockpit to pitot/static mast.

5. Outside Air Temperature Gauge (OAT)

A. Standard

The Archer III has an internally lit OAT installed in the side panel just forward of the pilot's storm window. The gauge must be removed to replace the bulb.

(1) Removal

(a) Remove interior panel assembly around pilot's window.

(b) Disconnect power wire.

(c) Remove external sunshield by turning it counterclockwise. Turn by hand only.

(d) Unscrew instrument from support bushing. Note position of any washers or spacers.

(2) Installation

CAUTION: DURING INSTALLATION OF ALL OAT'S, ONLY FINGER TIGHTEN GAUGE AND EXTERNAL THREADED HEX TUBE (SUNSHIELD). OVERTORQUING WILL DAMAGE INSTRUMENT.

(a) Screw instrument into support bushing, placing any washers or spacers in the positions noted during removal.

(b) Install external sunshield by turning it clockwise. Hand tighten only.

NOTE: Check gauge position before hand tightening sunshield.

(c) Connect power wire.

(d) Install interior panel assembly around pilot's window.

B. Optional Avionics Systems

In S/N's 2843598 thru 2843682, with Avidyne Entegra installed, OAT is provided by the Magnetometer/ OAT Sensor Assembly (Mag/OAT) mounted in the underside of the outboard left wing, which provides air temperature data to the Primary Flight Display (PFD) and Multifunction Display (MFD). See 34-22-00, for additional information.

In airplanes with the Garmin G1000/G1000 NXi installed, the GTP 59 OAT Probe, mounted to the underside of the left wing, provides air temperature data to the PFD and MFD. For additional information see 34-25-01, 34-25-02, or 34-25-03, as appropriate.

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ATTITUDE AND DIRECTION

1. Attitude Deviation Indicator (ADI)

NOTE: In airplanes equipped with Avidyne Entegra or Garmin G1000/G1000 NXi, no air-driven attitude deviation indicator is installed. In those installations, this function is provided by the Primary Flight Display (PFD).

A. Description

The ADI is essentially an air driven gyroscope rotating in a horizontal plane and is operated by the same principle as the directional gyro. The gyro is driven primarily by the engine driven dry pneumatic vacuum pump. An electrically operated vacuum pump is also installed as a backup in the event of primary pump failure. Air volume, not air pressure, spins the gyro rotor. A pressure regulator will automatically adjust air pressure within proper limits. If the air filter becomes contaminated, restricting air flow, gyro rotor speed will be reduced.

Air flow directed at gyro buckets, causes the rotor to spin approximately 17,000 to 22,000 rpm, allowing gyroscopic ability to remain rigid in space. The instrument case moves freely about the spinning gyro rotor in three dimensions due to a gimbal assembly. A bar across the face of the indicator represents the horizon and aligning the miniature airplane to the horizon bar simulates the alignment of the airplane to the actual horizon. Any deviation simulates the deviation of the airplane from the true horizon. The ADI is marked for different degrees of bank. The display is stable and shows minute attitude changes of 1 degree. Unlike the directional gyro, the erection mechanism activity is seen by a rapidly wobbling and leveling pitch bar, when power is first applied. The instrument can be adjusted for parallax by a knob on the lower center instrument face.

When the optional autopilot system is installed in the Archer III, the pilot and (if installed) copilot's ADI's combine air driven gyro(s) with electrical pitch and roll inputs and outputs to the autopilot/flight director.

B. Troubleshooting

See "Chart 1".

C. Removal and Installation

See 39-10-00.

2. Standby Attitude Indicator

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

A. Description

In S/N's 2843598 thru 2843682, with Avidyne Entegra installed, an electric standby attitude deviation indicator is installed to the left of the Primary Flight Display (PFD). (See "Figure 1".)

Other than removing and replacing the unit itself (see 39-10-00), the only line replaceable part is the emergency power battery which is located in the rear of the instrument's case.

Required periodic maintenance is listed in 5-20-00 and 5-30-00. Checkout and test procedures and Instructions for Continued Airworthiness are provided in Mid-Continent Instruments Manual No. 9015762.

B. Removal and Installation

See 39-10-00.

CHART 1
TROUBLESHOOTING ATTITUDE INDICATOR

Trouble	Cause	Remedy
Bar fails to respond.	Observe vacuum gauge for insufficient vacuum.	If insufficient vacuum exist, check pump and tubing.
	Filter dirty.	Clean or replace filter.
	Defective instrument.	Replace gyro instrument.
Bar does not settle.	Insufficient vacuum.	Check line and pump. Adjust valve.
	Incorrect instrument.	Check part number.
	Defective instrument.	Replace.
Bar oscillates or shimmies continuously.	Instrument loose in panel.	Tighten mounting screws.
	Vacuum too high.	Adjust vacuum regulators.
	Defective mechanism.	Replace instrument.
Instrument does not indicate level flight.	Instrument not level in panel.	Loosen screws and level instrument.
	Aircraft out of trim.	Trim aircraft.
Bar high after 180° turn.	Normal, if it does not exceed 1/16 inch.	
Instrument tumbles in flight.	Observe vacuum gauge for low vacuum.	If vacuum is low, reset regulator.
	Dirty filter.	Clean or replace filter.
	Line to filter restricted.	Replace line.
	Plug missing or loose in instrument.	Replace or tighten plug.

3. Directional Gyro

NOTE: In airplanes equipped with Avidyne Entegra or Garmin G1000/G1000 NXi, no air-driven directional gyro is installed. In those installations, this function is provided by the Primary Flight Display (PFD).

A. Description

Air driven heading indicators are standard equipment on the Archer III, and are offered as an option for installation on the copilot's instrument panel.

The gyro stabilized heading indicator is a flight instrument incorporating an air driven gyro stabilized in the vertical plane. The gyro is rotated at high speed by lowering the pressure in the airtight case and simultaneously allowing atmospheric air pressure to enter the instrument against the gyro buckets. Due to gyroscopic inertia, the spin axis tends to continue pointing in the same direction, even though the aircraft yaws, turns, pitches or rolls. This relative motion between the gyro and the instrument case is shown on the instrument dial which, in earlier models, is similar to the compass card. The compass card has a 360° face.

Unless slaved to the Earth's magnetic field, the gyroscopic heading indicator has no sense of direction and must be set to the magnetic compass. This is also true of normally slaved heading indicator when it becomes necessary to deslave it. Once deslaved, it acts as free spinning gyroscope. The card, when set to agree with the airplane's magnetic compass, provides a positive heading indication free from acceleration/deceleration and turning errors. However, precession forces applied to the gyro during turns may cause the gyro to "drift" and, upon completion of the turn, result in a difference in readings between the directional gyro and the magnetic compass, necessitating resetting the gyro. Even while maintaining a given heading, the gyro compass tends to precess (drift) due to internal friction, spin axis error, air turbulence and airflow. Therefore, the gyro should be checked against the magnetic compass, and reset as necessary, at least every 15 minutes. Maximum acceptable precession is 5° in 15 minutes.

Some heading indicators are limited to 55° of roll and pitch. Should these limits be exceeded, the gyro will "tumble". This is evidenced by a rapid spinning of the compass card. The gyro in a properly operating instrument can be erected, after returning to straight and level flight, by caging the gyro and resetting it.

B. Troubleshooting

See "Chart 2".

Service is restricted to instrument installation and power source. Obvious malfunctions, such as failure to erect or compass card spinning that cannot be corrected by manually caging the instrument, require repair or replacement. Typical installation examples of heading indicator malfunctions are due to: restricted air flow from air line kinks or leaks, contaminated air filters, deteriorating electrical grounds, sagging instrument panel shock mounts, systems regulators, faulty vacuum/pressure gauges. (Air pressure must be 5.5 ±0.5 psig). Replace or repair the instrument only after the operating system has proven to be good.

C. Removal and Installation

See 39-10-00.

CHART 2
TROUBLESHOOTING DIRECTIONAL GYRO

Trouble	Cause	Remedy
Excess precession (drift) in either direction.	Setting error.	Review “Directional Gyro - Description,” above.
	Defective instrument.	Replace instrument.
	High or low vacuum. If vacuum is not correct, check for the following:	
	a. Relief valve improperly adjusted.	a. Adjust.
	b. Incorrect gauge reading.	b. Replace gauge.
	c. Pump failure.	c. Repair or replace.
	d. Vacuum line kinked or leaking.	d. Check and repair. Check for collapsed inner wall of hose.
Dial spins during turn.	Gimbal limits of (55° bank or pitch) exceeded.	Recage gyro in level flight.
Dial spins continuously.	Defective mechanism.	Replace.

4. Magnetic Heading Systems

A. Magnetic Compass

A magnetic compass is installed as standard equipment.

(1) Description

The magnetic compass is a self-contained instrument. The compass card is mounted on two magnets, which tend to align themselves with the Earth's magnetic field, permitting the compass to use Earth's magnetic north as a reference. Due to magnetic attractions created by nearby metal surfaces and the airplanes's electrical equipment, the compass magnets are frequently "pulled" away from magnetic north. Much of this error (not all) can be removed by a procedure called swinging the compass (see Adjustment of Magnetic Compass). This involves placing the airplane on a magnetic compass rose, and, using a brass screwdriver to adjust compensators incorporated in the instrument, removing as much of the error as possible while on headings of north, south, east and west. The remaining error is then noted on a compass correction (deviation) card for each 30° of heading. The completed correction card is then placed in receptacle mounted on the instrument.

For night operations, the instrument is internally lit. The light is powered by the airplane's instrument lighting circuit.

Swing the compass as described under Adjustment, below, as follows:

(a) When the accuracy of the compass is suspected.

(b) After any cockpit modification or major replacement involving ferrous metal.

(c) Whenever a compass has been subjected to a shock; e.g., after a hard landing or turbulence.

(d) After aircraft has passed through a severe electrical storm.

(e) After lighting strike.

(f) Whenever a change is made to the electrical system.

(g) Whenever a change of cargo is likely to affect the compass.

(h) When aircraft's area of operation is changed to a different geographic location with a major change in magnetic deviation. (e.g., from Miami, Florida to Fairbanks, Alaska.)

(i) After aircraft has been parked on one heading for over a year.

(j) When flux valves / magnetometers are replaced.

(2) Troubleshooting

See "Chart 3".

(3) Adjustment

Before attempting to compensate compass place the aircraft in simulated flight conditions. Check to see that the doors are closed, flaps in retracted position, engine running, throttle set at 1000 rpm or low idle, and aircraft in level flight attitude. Aircraft master switch, alternator switch and all radio switches should be in the ON position. All other cockpit controlled electrical switches should be in the OFF position. Use a brass or other non-magnetic screwdriver to make adjustments to compensator screws.

(a) Place airplane on a magnetic compass rose designed and intended for adjusting magnetic compasses.

(b) Set adjustment screws of compensator on zero. Zero position of adjusting screws is when the dot of the screw is lined up with the dot of the frame.

(c) Align airplane with magnetic North heading on the compass rose. Adjust N-S adjustment screw until compass reads exactly North.

CHART 3
TROUBLESHOOTING MAGNETIC COMPASS

Trouble	Cause	Remedy
Excessive card error.	Compass not properly compensated.	Compensate instrument.
	External magnetic interference.	Locate magnetic interference and eliminate, if possible.
Excessive card oscillation.	Insufficient liquid.	Replace instrument.
Card sluggish.	Weak card magnet.	Replace instrument.
	Excessive pivot friction or broken jewel.	Replace instrument.
	Excessive pivot friction or broken jewel.	Replace instrument.
Liquid leakage.	Loose bezel screws.	Replace instrument.
	Broken cover glass.	Replace instrument.
	Defective sealing gaskets.	Replace instrument.
Discolored markings.	Age.	Replace instrument.
Defective light.	Burned out lamp or broken circuit.	Check lamp or continuity of wiring.
Card sticks.	Altitude compensating diaphragm collapsed.	Replace instrument.
Card does not move, compensating screws are turned.	The gears that turn the compensating magnets are stripped.	Replace instrument.
Compass swings erratically when radio transmitter is keyed.	Normal.	

(d) Align airplane with magnetic East heading on the compass rose. Adjust E-W adjusting screw until compass reads exactly East.

(e) Align airplane with magnetic South heading and note error. Adjust N-S adjusting screw until one-half of the error has been removed.

(f) Align airplane with magnetic West heading and note error. Adjust E-W adjusting screw until one-half of the error has been removed.

(g) Align airplane with magnetic north. Note heading being indicated on magnetic compass (may not be North) and enter on compass deviation (correction) card.

(h) Align airplane with successive magnetic 30° headings, i. e., 030°, 060°, 090°, 120°, etc. Enter actual compass reading on each heading on compass deviation (correction) card.

(i) If deviations (difference between actual magnetic heading and what compass indicates on that particular heading) exceeds ±10° on any heading:

1) Check to be sure no magnetic metals are near compass (tools, flashlights, pocket knives, wristwatches etc.)

2) Check to be sure screwdriver being used to make adjustments is either fiber or non-magnetic metal, such as brass.

(j) When satisfied that errors in excess of 10° is fault of the instrument, replace instrument.

(k) After installing new instrument, repeat steps (a) through (h).

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION SUPPLEMENTARY PUBLICATIONS.)

B. King/Allied Signal Slaving Compass System (1995 thru 1998)

In S/N's 2843001 thru 2843202, slaving compasses may be installed as part of the optional King/Allied Signal (now Honeywell) Autopilot System. King/Allied Signal technical support, parts support, and service literature can be obtained from:

Honeywell
One Technology Center
23500 W.105th St., M/D #45
Olathe, Kansas 66061-1950
http://www.bendixking.com/

C. S-TEC ST-180 Slaving Compass System (1999-2004)

(PIR-PPS60191, Rev. New.)

In S/N's 2843203 thru 2843597, a S-TEC ST-180 slaving compass system may be installed as optional equipment.

(1) Flux Detector

With the optional S-TEC ST-180 slaving compass (HSI/autopilot) installation, a flux detector is installed in the wing tip of the left wing to provide heading data. (See "Figure 1".)

CAUTION: PERFORM MAGNETIC HEADING COMPENSATION / CALIBRATION, BELOW, WHENEVER THE FLUX DETECTOR IS CHANGED.

(a) Removal

1) Remove the left wing tip fairing to expose the flux detector.

2) Disconnect the wiring harness(es) from the top of the flux detector to be removed.

3) Unscrew and remove the three brass screws and washers and remove the flux detector.

(b) Installation

CAUTION: THE FLUX DETECTOR IS SECURED TO THE MOUNTING BRACKET WITH BRASS SCREWS. ENSURE ONLY BRASS SCREWS ARE USED WHEN REINSTALLING.

1) Place the flux detector into position on its mounting bracket and secure with brass screws and washers (3 ea.)

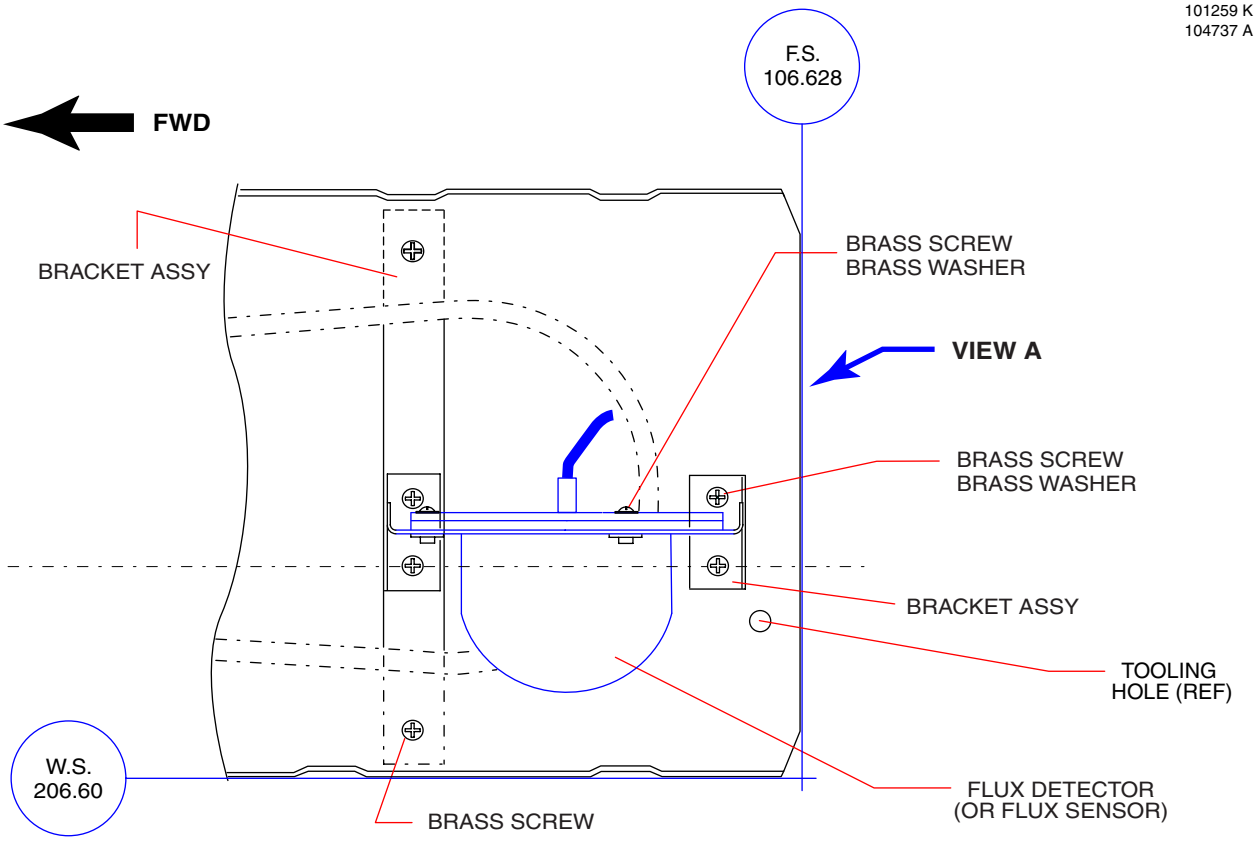
2) Connect wiring harness to the connector on top of the flux detector.

3) Seal the connector backshell and base as shown in "Figure 1" with DOW Corning 4 Electrical Insulating Compound.

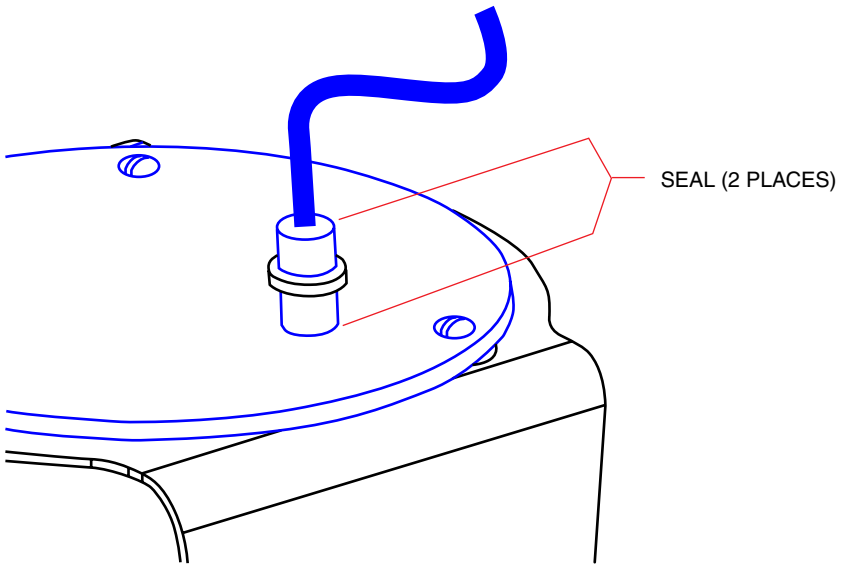
4) Reinstall the left wing tip.

NOTE: Ensure correct hardware is used when reinstalling wing tip over flux detector.

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LOOKING INBOARD
LEFTHAND WING



VIEW A

Flux Detector Installation
Figure 1

(2) Magnetic Heading Compensation / Calibration

Accuracy of the entire heading system is dependent on the location of the flux detector and proper calibration. Accuracies of plus or minus one degree are possible when care is taken during installation and calibration. To obtain such results:

The flux detector must be positioned so that it points in the direction of aircraft flight; and, The north-south and east-west correctors must be adjusted to compensate for extraneous magnetic fields near the location of the flux detector.

(a) Required Equipment

Instrument Flight Research Corp. RF signal generator NAV-40IL or equivalent. Characteristics of the signal generator include: Frequency ranges 108 to 118 MHz, 117 to 136 Mhz, 328 to 336 MHz; +1/-0.01% accuracy, output level continuously adjustable from 1.0uV to 0.1V into a 50 ohm load; 50 ohm output impedance; and internal adjustable or stepped VOR, LOC, and GS modulation. The RF signal generator should be portable and convenient for use while sitting in the aircraft cockpit.

(b) Procedure

1) Apply power to the Model ST-180 HSI System. Allow at least three (3) minutes for the gyro to erect and synchronize.
2) Prior to actual alignment of the flux sensor, turn the aircraft to both north and east headings. Apply power to electrical equipment such as navigation and beacon lights and verify that the compass system is not affected.
3) Align the aircraft to an approximate magnetic north heading. On "Chart 4", record the actual magnetic heading and the HSI heading card reading.
4) Determine and record on "Chart 4" the deviation between the actual magnetic heading and the heading card heading. If the heading card reads high, the deviation is a plus.
5) Repeat steps 3 and 4 for east, south, and west headings. Record actual magnetic headings, heading card readings, and deviations on "Chart 4".
6) Plot deviations on the initial deviation graph in "Figure 2".
7) Realign the aircraft to north. Adjust the north-south corrector on the Slaving Panel, for one half of the difference between the north and south deviations. Record the new deviation for north and south on the initial deviation graph in "Figure 2".
8) Realign the aircraft to east. Adjust the east-west corrector on the Slaving Panel, for one half the difference between the east and west deviations. Record the new deviation for east and west on the initial deviation graph in "Figure 2".
9) If the pattern is not centered around zero, rotate the flux sensor clockwise to correct for minus deviations or counterclockwise for plus deviations. Plot final deviations on the final deviation graph in "Figure 2".
10) The deviations should now center around the zero reference line of the graph. If the error exceeds the specified system error limits (±3 degrees), repeat the complete procedure.

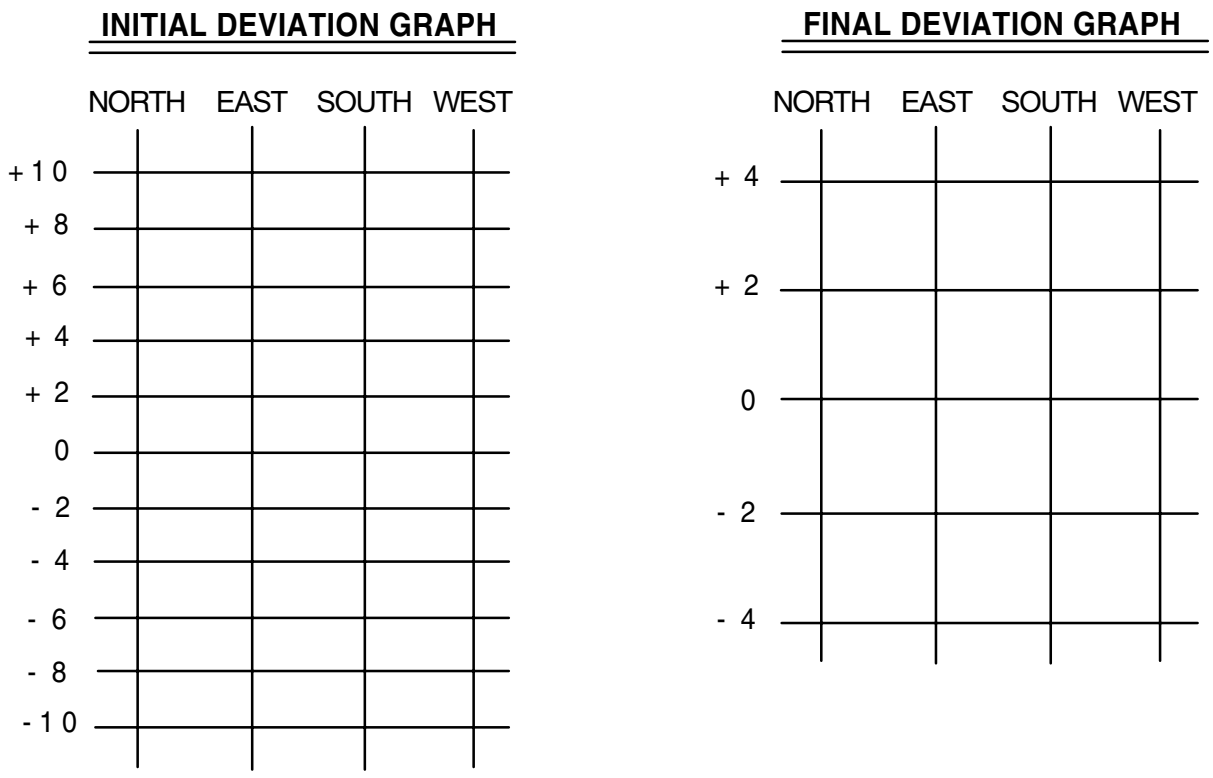
D. Avidyne Entegra IFDS

See 34-22-00.

CHART 4
DEVIATION CHART

Approx. Magnetic Heading	Actual Magnetic Heading	Heading Card Reading	Deviation
North			
East			
South			
West			

PPS60191



ST-180 Deviation Graphs
Figure 2

E. King KCS-55A Slaving Compass (with HSI)

(PIR-PPS60081, Rev. C)

The King KCS-55A may be installed as an option in non-Avidyne equipped airplanes, S/N's 2843673 and up.

(1) Flux Detector

See "Figure 1".

A flux detector installed in the left wing tip, at W.S. 207.40, provides heading data to the HSI.

CAUTION: PERFORM MAGNETIC HEADING COMPENSATION/CALIBRATION, BELOW, WHENEVER THE FLUX DETECTOR IS CHANGED.

(a) Removal

1) Remove the left wing tip fairing to expose the flux detector.
2) Disconnect the wiring harness from the top of the flux detector.
3) To remove the flux detector, remove the three (3) brass screws and washers attaching it to mounting bracket.

(b) Installation

CAUTION: THE FLUX DETECTOR IS SECURED TO THE MOUNTING BRACKET WITH BRASS SCREWS. ENSURE ONLY BRASS SCREWS ARE USED WHEN REINSTALLING.

1) Place the flux detector into position on its mounting bracket and secure with brass screws and washers (3 ea.).
2) Connect wiring harness to the connector on top of the flux detector.
3) Reinstall the left wing tip.

NOTE: Ensure correct hardware is used when reinstalling wing tip over flux detector.

(2) Flux Detector Index Error

(a) Place the airplane in simulated flight conditions:

1) Check to see that the doors are closed and flaps are in retracted position
2) Have the engine running at idle (1000 rpm), with the airplane at a level flight attitude.
3) Put the battery master, alternator and all radio switches in the ON position.
4) Power ON the KCS-55A (allow to synchronize).
5) Turn lights – recognition, instrument panel, navigation, position, and strobe – ON.

(b) Place the aircraft on a compass rose in each of the four cardinal headings. At each cardinal heading, with the slaving control in the "Free Gyro" position, center the slave meter with the manual slave buttons on the slaving control, and record the difference (plus or minus) between the cardinal heading and the indicator heading.

(c) Add the four differences, and divide the sum by four. The result is the index error.

(d) Loosen the mounting screws on the flux detector, and rotate the unit as required to cancel the index error (ex.: if the index error is +2°, set the indexing device at -2°).

(e) Retighten the mounting screws.

(3) Slaving Control Compensator Check and Adjustment

As in (a) above, place the aircraft in simulated flight conditions: engine running, battery master ON, alternator ON, all radios ON, and lights – recognition, instrument panel, navigation, position, and strobe – ON. Power ON the KCS-55A (allow to synchronize). Place the slaving control in the “Free Gyro” position. Use a brass or other non-magnetic screwdriver to make adjustments to compensator screws.

(a) Place the aircraft on the North compass rose heading. Rotate the indicator card to North using the manual slave buttons on the slaving control. With a non-magnetic screwdriver in the N/S opening on the slaving control, center the slave meter.

(b) Place the aircraft on the East compass rose heading. Rotate the indicator card to East using the manual slave buttons on the slaving meter. With a non-magnetic screwdriver in the E/W opening on the slaving control, center the slave meter.

(c) Place the aircraft on the South compass rose heading. Rotate the indicator card to South using the manual slave buttons on the slaving control. With a non-magnetic screwdriver in the N/S opening on the slaving control, adjust to remove one-half (1/2) of the existing deviation shown by the slave meter.

(d) Place the aircraft on the West compass rose heading. Rotate the indicator card to West using the manual slave buttons on the slaving control. With a non-magnetic screwdriver in the E/W opening on the slaving control, adjust to remove one-half (1/2) of the existing deviation shown by the slave meter.

(e) If deviation (difference between actual magnetic heading and what compass indicates on that particular heading) exceeds ± 10° on any heading, instrument replacement may be necessary. To verify, do the following:

1) Check whether any magnetic metals are near compass (tools, flashlights, pocket knives, wristwatches, etc.)

2) Check to be sure screwdriver being used to make adjustments is either fiber or a non-magnetic metal, such as brass.

3) When satisfied that errors in excess of 10° are the fault of the instrument, replace instrument. After installing a new instrument, repeat procedures (1) and (2).

(f) After all adjustments are completed, the indicator shall read within ± 2° of any compass rose heading on which the aircraft is placed.

F. Garmin G1000 or G1000 NXi IAS

See one of the following, as appropriate:

(1) "GRS/GMU Calibration Procedures" on page 342501118.

(2) "GSU/GMU Calibration Procedures" on page 342502122.

(3) "GSU 75/GMU Magnetometer Calibration" on page 342503124.

5. Turn Coordinator

NOTE: In airplanes equipped with Avidyne Entegra or Garmin G1000/G1000 NXi IAS, this function is performed solely by the Primary Flight Display (PFD).

A. Description

Archer III airplanes are equipped with turn coordinator(s). Conventual turn and slip indicator(s) are available as an option. The gyro in either one is electrically operated. The turn portion of both indicators is a gyroscope, while a ball sealed in a curved glass tube filled with dampening fluid is used to maintain coordinated flight. Both types work on the precession principle.

The gyro in the turn coordinator is installed so that the indicator, mechanically linked to the gyro, responds to both the rate of yaw or turn, and rate (not degree) of roll. With this indicator, if the aircraft is rolled right and left rapidly, the indicator will move while the airplane is rolling, indicating the rate (degrees per second) of roll. If the aircraft is then held in at a constant bank angle, and rudder is applied to maintain heading (such as when “slipping”), the indicator will come back to zero rate, indicating no roll or turn. Should the airplane be permitted to turn once a desired bank angle is established, the indicator will remain deflected in the direction of the roll/turn, now indicating rate of turn. Thus, using the flight controls to keep the indicator on the appropriate turn index, will result in the airplane becoming established in a coordinated, standard rate (3° per second) turn.

The gyro in the turn and slip indicator, which is the older style, is installed so that the needle responds only to rate of yaw or rate turn. It has a vertical needle in the center of the dial mechanically linked to the gyro. Unless the aircraft is turning or yawing, the needle will not move regardless of roll rate. The greater the rate of turn (degree of heading change per second), the more the gyro is precessed, resulting in greater needle deflection in the direction of turn.

B. Troubleshooting

See "Chart 5".

C. Removal and Installation

See 39-10-00.

CHART 5
TROUBLESHOOTING TURN COORDINATOR

Trouble	Cause	Remedy
Pointer fails to respond.	Foreign matter lodged in instrument.	Replace instrument.
Incorrect sensitivity.	Out of calibration.	Replace instrument.
Incorrect turn rate	Out of calibration.	Replace instrument.
	Aircraft not in coordinated turn. (Turn and slip indictor)	Center ball in turn.
Ball sticky.	Flap spot on ball.	Replace instrument.
Ball not in center when aircraft is correctly trimmed.	Instrument not level in panel.	Level instrument.
Instrument will not indicate a turn.	No power to instrument.	Check circuit and repair.
	Instrument malfunction.	Replace instrument.

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INTEGRATED FLIGHT DISPLAY SYSTEM (IFDS) - AVIDYNE ENTEGRA

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Integrated Flight Display System (IFDS) - Avidyne Entegra

S/N's 2843598 thru 2843682 may be equipped with the Avidyne Entegra IFDS installation.

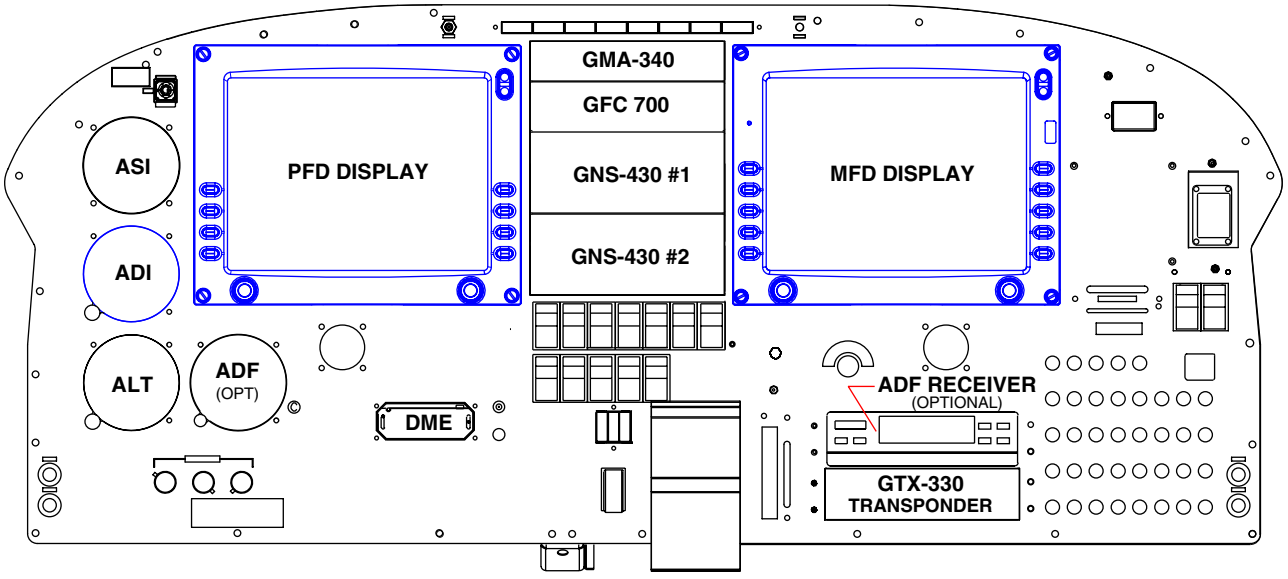
A. Description

See "Figure 1" and "Figure 2".

This system uses two large 10.4-inch diagonal, high-resolution, sunlight-readable full color displays (PFD and MFD), to provide primary flight and engine information as well as a wide variety of other data. Standard primary flight instruments (i.e. - airspeed, electric attitude indicator, and altimeter) provide redundancy.

The IFDS installation consists of the following components: Primary Flight Display (PFD), Multifunction Display (MFD), Data Acquisition Unit (DAU) and associated sensors, and Magnetometer/OAT Sensor Assembly.

101901 L



Instrument Panel - Avidyne Entegra
Figure 1

B. Maintenance

NOTE: An upgraded version of Avidyne software is installed in later airplanes. Accordingly, some items appear different in earlier airplanes, as cited below.

The Instructions for Continued Airworthiness (ICA) published by Avidyne provide the necessary information for maintaining this system as installed in Piper airplanes, except as noted below.

(1) Primary Flight Display (PFD) (PIR-PPS60213, Rev. New / PPS60213-1, Rev. New.)

Use 700-00006-0XX PFD & 700-00011-0XX Mag/OAT ICA, Avidyne Document No. AVPFD-174, Revision 02 or later, with the following exceptions:

NOTE: If any connection in the pitot / static system is opened for maintenance, the entire system must be rechecked per Pitot / Static System, Test, in 34-10-00.

NOTE: Before attempting to set-up the PFD, ensure the GNS-430(s) have been configured per the Post Installation Set-up Procedure under COM/NAV/GPS in 34-50-00.

NOTE: In S/N's 2843415 and up, before attempting to set-up the PFD, ensure the GTX-330 is configured per the Post Installation Set-up Procedure under "Transponder, S/N's 2843415 and up," in 34-50-00.

NOTE: Whenever the PFD is replaced perform the setup / calibration procedures specified in Avidyne Document No. AVPFD-174 as modified below.

(a) In para 6, Troubleshooting Information, in the chart where it says "OAT (Optional)," cross out "optional." The OAT is standard in the Piper installation.

(b) In para 7.2 and Figure 7, where the standard Avidyne installation describes alignment pins and retaining clips on the sides of the PFD, the Piper installation uses a single alignment pin on the top of the PFD engaging a slot in the upper rear cross bracket (see "Figure 2", Sheet 1.)

(c) In para 7.5.4.2, in the "Main RS232 Configuration Page" chart, for CHNL 3 under GNS-430 No 2, both Input and Output should read "Crossfill" instead of "Off."

(d) In para 7.5.5.2, below the heading "S-TEC System 55x Autopilot Unit," insert the following:

"If the PFD is replaced, the IRU calibration and Magnetometer calibration must be completed before continuing with the following steps."

(2) Multifunction Display (MFD) (PIR-PPS60212, Rev. A / PPS60212-1, Rev. A / PPS60212-2, Rev. New.)

Use 700-00004-0XX-() Multifunction Display ICA, Avidyne Document No. AVMFD-167, Revision 07 or later, with the following exceptions:

NOTE: The following is written for airplanes with MFD Software 530-00177-000 (latest revision); exceptions or differences for airplanes equipped with earlier software versions are noted (**) as required.

NOTE: Whenever the MFD is replaced, recalibrate the Fuel Quantity Indicator per Avidyne Entegra Installation, 28-40-00.

(a) In para 2, items 7 and 8 are standard in the Piper installation.

(b) In para 7.3.1, Maintenance Mode Access, replace 3, 4, and 5 with the following:

1) At the prompt "Press any bezel key to continue", press any Line Select Key (LSK). This places the screen at the Initial Usable Fuel Page. Depress the "Fuel Done" LSK R4.

2) Rotate the left knob until the AUX page is displayed.

3) Simultaneously press and hold the top left line select key (L1) and the third from top left line select key (L3) for at least five seconds or until line select key labels "blink" or are removed from the screen.

(c) Replace 7.3.2.1, GAMA 429 Graphics Setup (EX5000 only) with the following:

1) From the maintenance page, depress the LSK L1 (GPS) to access the GPS/FMS Setup page.

2) Using the right knob to select the configuration field and the left knob to select the desired configuration; configure the GPS/FMS Setup page for the following conditions:

Receiver 1:	GAMA 429 FORMAT
Port:	ARINC 429 RX 1
Speed:	LOW
Receiver 2:	GAMA 429 FORMAT
Port:	ARINC 429 RX 2
Speed:	Low

3) Press SAVE button.

(d) Replace para 7.3.5.1, TAS (SkyWatch, Bendix/King, or GTX-330) Setup, with the following:

1) From the maintenance page, depress the LSK L2 (Traffic) to access the Traffic Setup page.

2) Using the right knob to select the configuration field and the left knob to select the desired configuration; configure the Traffic Setup page for the following conditions:

Sensor:	TAS (if equipped with SKY-497, otherwise TIS-G)
Port:	ARINC 429 3
TAS Type:	SkyWatch (if equipped with SKY-497, otherwise Not Installed)

** With Software prior to #530-00177-000 (latest Rev.)

Sensor:	TAS (if equipped with SKY-497, otherwise Not Installed)
Port:	ARINC 429 3
TAS Type:	SkyWatch
External Controller:	Box NOT Checked

3) Press SAVE button.

(e) Replace 7.3.4, Lightning Sensor Interface and Setup, with the following:

1) From the maintenance page, depress LSK L3 (Lightning) to access the Lightning Setup page.

2) Using the right knob to select the configuration field and the left knob to select the desired configuration. Configure the Lightning Setup page for the following conditions:

Sensor:	WX-500 (if equipped, otherwise Not Installed)
Operating Mode:	Weather
Port:	RS232 3 (lightning default)
Stabilization type:	Use Map Heading/Track
Enable lightning ahead warning?:	Box CHECKED
Antenna on top:	Box NOT CHECKED

3) Press SAVE button.

(f) Replace para 7.3.10.1, Engine Sensor Setup, with the following:

1) From the maintenance page, depress LSK L4 to access the Engine Instrument Setup page.

2) Using the right knob to select the configuration field and the left knob to change the desired configuration; configure the Engine Instrument Setup page for the following conditions:

Aircraft Model:	PA28-181 (Archer)
Port:	RS232 4

3) Depress Save button (R1).

(g) Replace 7.3.8, Map Heading, with the following:

1) From the maintenance page, depress LSK R1 to access the Map Heading Setup page.

2) Using the right knob to select the configuration field and the left knob to change the desired configuration; configure the Map Heading Setup page for the following conditions:

Map Heading:	FMS/GPS

3) Press SAVE button.

(h) Aircraft Setup

1) From the maintenance page, depress LSK R2 to access the Aircraft Setup page.

2) Using the right knob to select the configuration field and the left knob to change the desired configuration; configure the Aircraft Setup page for the following conditions:

Narrowcast:	Not Installed
Port:	RS232 6
Broadcast:	XM Radio (if XM option installed, otherwise Not Installed)
Port:	RS232 1
Aux Data:	Not Installed
Port:	ARINC 429 4
Dimming Bus Voltage settings:	Shows Current Voltage - No setting required. (0 volts when annunciator day/night switch in DAY position. Dimming bus voltage shown when day/ night switch in NIGHT position).

** With Software prior to #530-00177-000 (latest Rev.):

Dimming Bus
Voltage settings:

Brightest Dimming Voltage:	24 Volts
Darkest Dimming Voltage:	5 Volts

3) When the Aircraft Setup has been configured correctly, press Save button. If you decide not to save the changes, press the Cancel button. Changes will not take effect until the MFD has been restarted.

(i) Fuel Quantity Indicator Calibration.

See 28-40-00.

(3) Data Acquisition Unit (DAU)

(a) Use 200-00041-000 DAU ICA, Avidyne Document No. AVSIU-011, with the following exceptions:

1) In para 6, in “Table 2 - DAU Pinout,” pins J1-2 and J1-21 have “No Connection” in the Piper installation.

2) In para 6, in “Table 4 - DAU Sensor Compatibility,” parameter “VAC” is not used in the Piper installation.

(b) Removal and Installation

NOTE: Whenever the DAU is replaced, recalibrate the Fuel Quantity Indicator per Avidyne Entegra Installation, 28-40-00.

The DAU is mounted on the left forward side of the frame assembly at F.S. 128.737 (see "Figure 2", Sheet 2, View B-B). Remove the rear seats and floor panel for access.

(4) Magnetometer/OAT Sensor Assembly

See "Figure 2", Sheet 3, View A-A.

The Magnetometer/OAT Sensor Assembly (Mag/OAT) is mounted on a wing access cover plate in the underside of the outboard left wing and supplies magnetic heading information to the Primary Flight Display (PFD).

NOTE: The cover plate and Mag/OAT sensor assembly is removed and installed as a unit.

(a) Removal

1) Remove eight (8) screws and support cover plate with your hand.

2) Drop cover plate down sufficient to reach inside and disconnect the wiring harness.

3) Remove cover plate and Mag/OAT sensor assembly, as a unit.

(b) Installation

NOTE: Whenever the Mag/OAT is replaced perform the PFD setup / calibration procedures specified in Avidyne Document No. AVPFD-174 as modified by Maintenance, Primary Flight Display (PFD), above.

1) Connect the Mag/OAT sensor assembly wiring harness.

2) Position the cover plate and Mag/OAT sensor assembly in the access hole with the arrow on the magnetometer pointing forward.

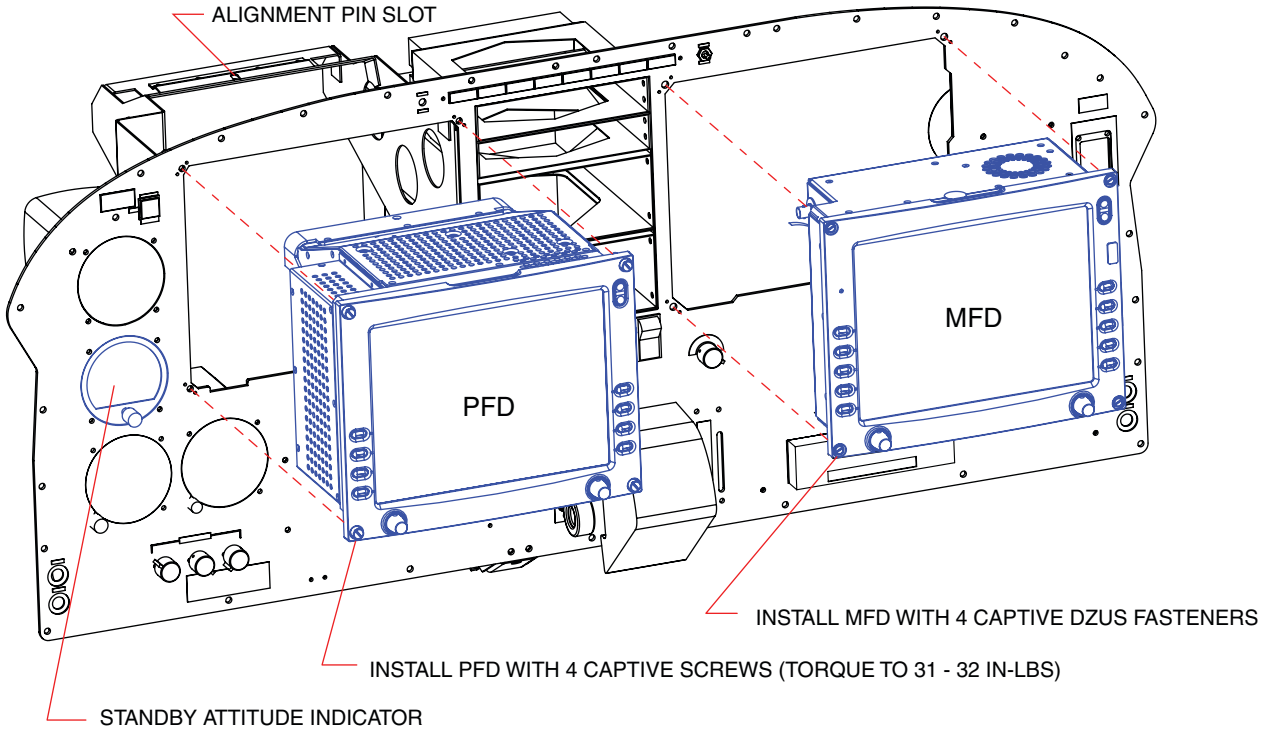
3) Secure with screws (8).

C. Component Locator

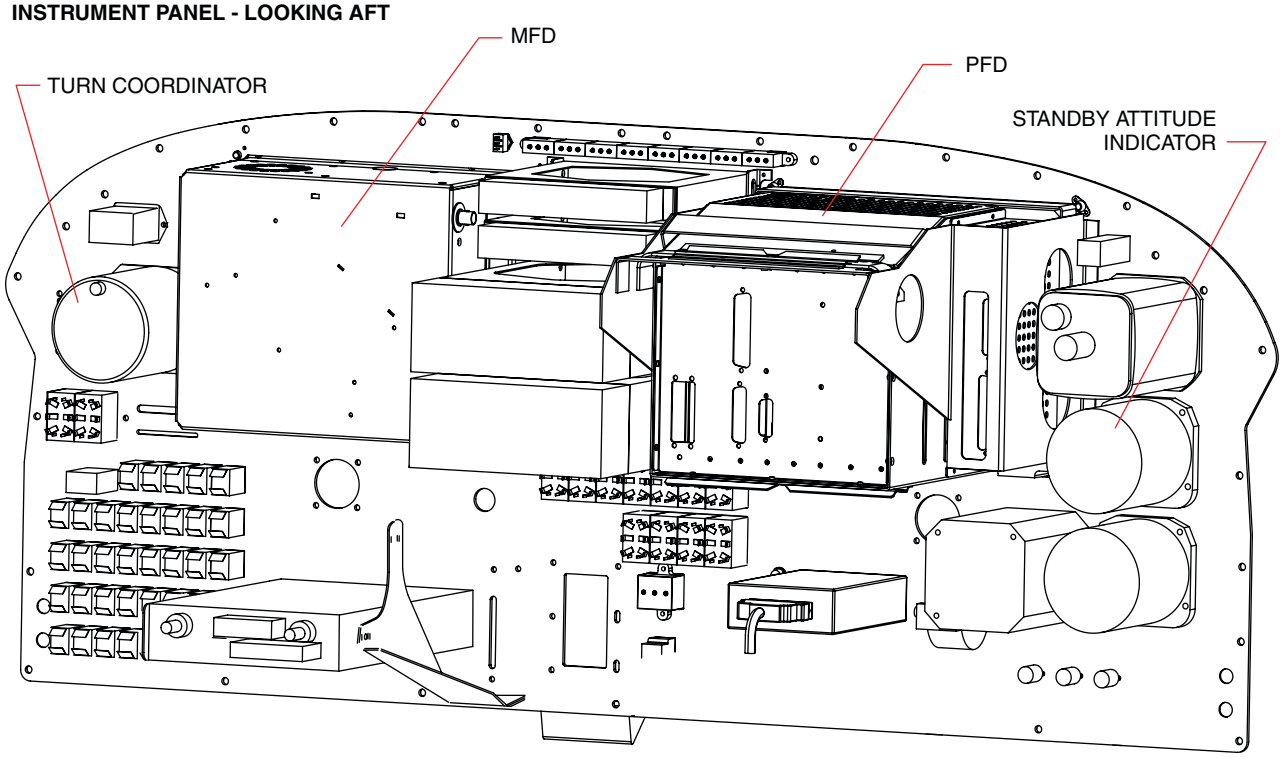
See "Figure 2".

INSTRUMENT PANEL - LOOKING FWD

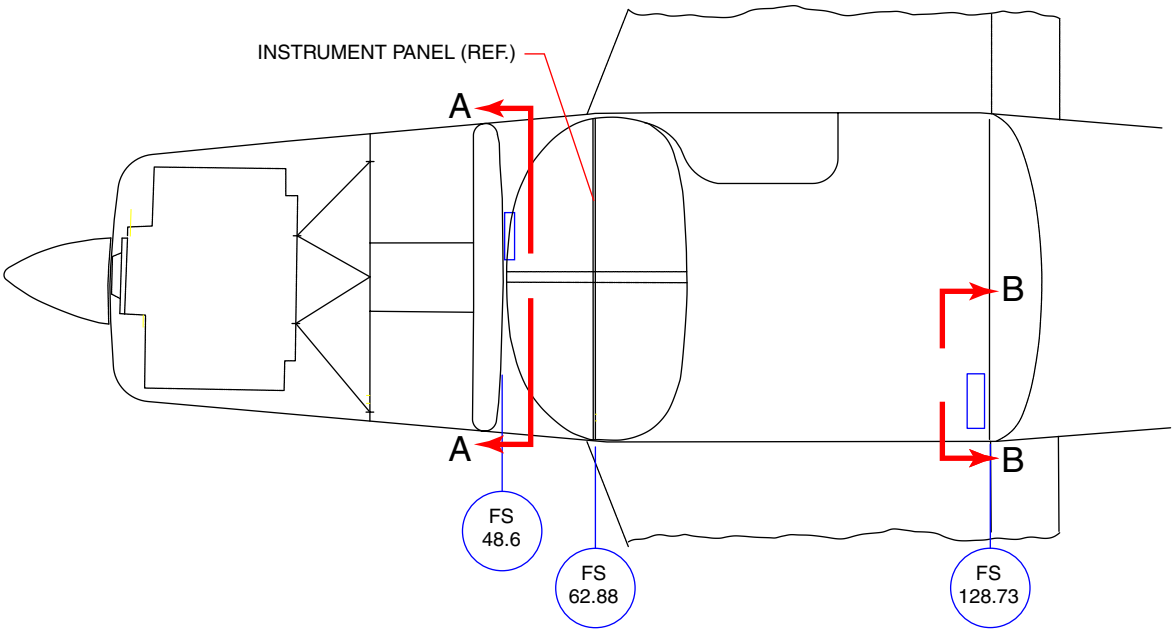
101901 L



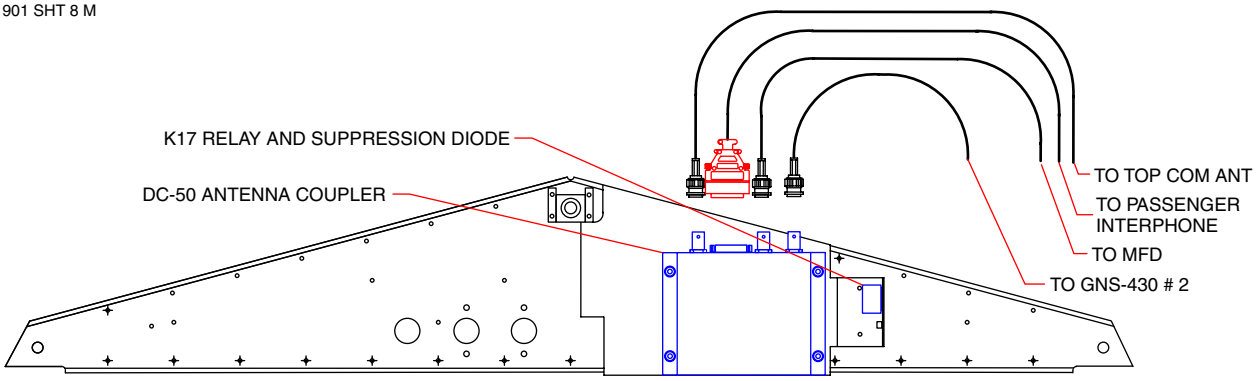
INSTRUMENT PANEL - LOOKING AFT



Component Locator
Figure 2 (Sheet 1 of 3)

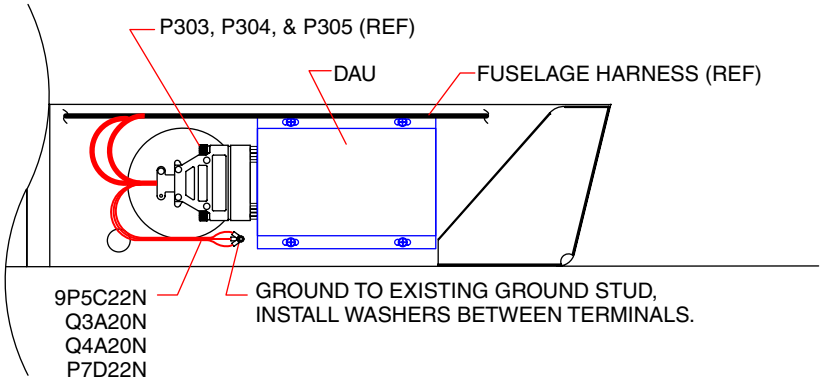


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VIEW A-A
ANTENNA COUPLER INSTALLATION - LOOKING FORWARD (STA 48.6)

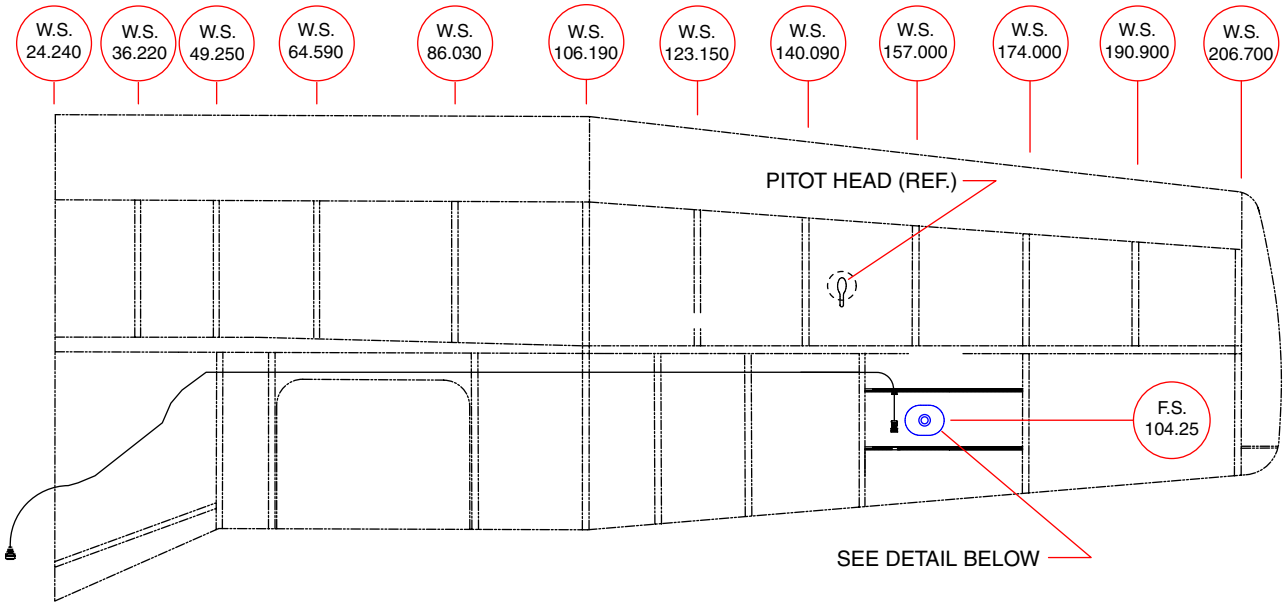
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VIEW B-B
DAU INSTALLATION - LOOKING AFT (STA 128.737)

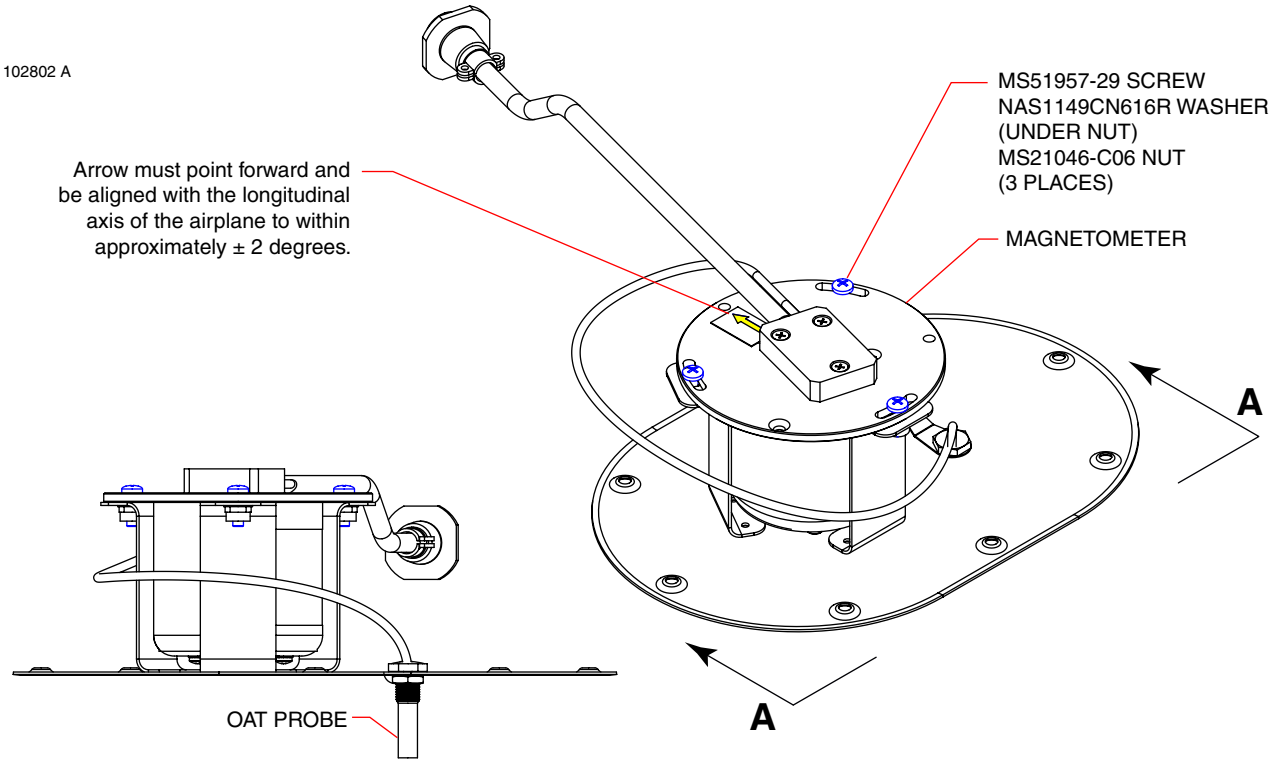
Component Locator
Figure 2 (Sheet 2 of 3)

101901 L



LEFT WING - LOOKING DOWN

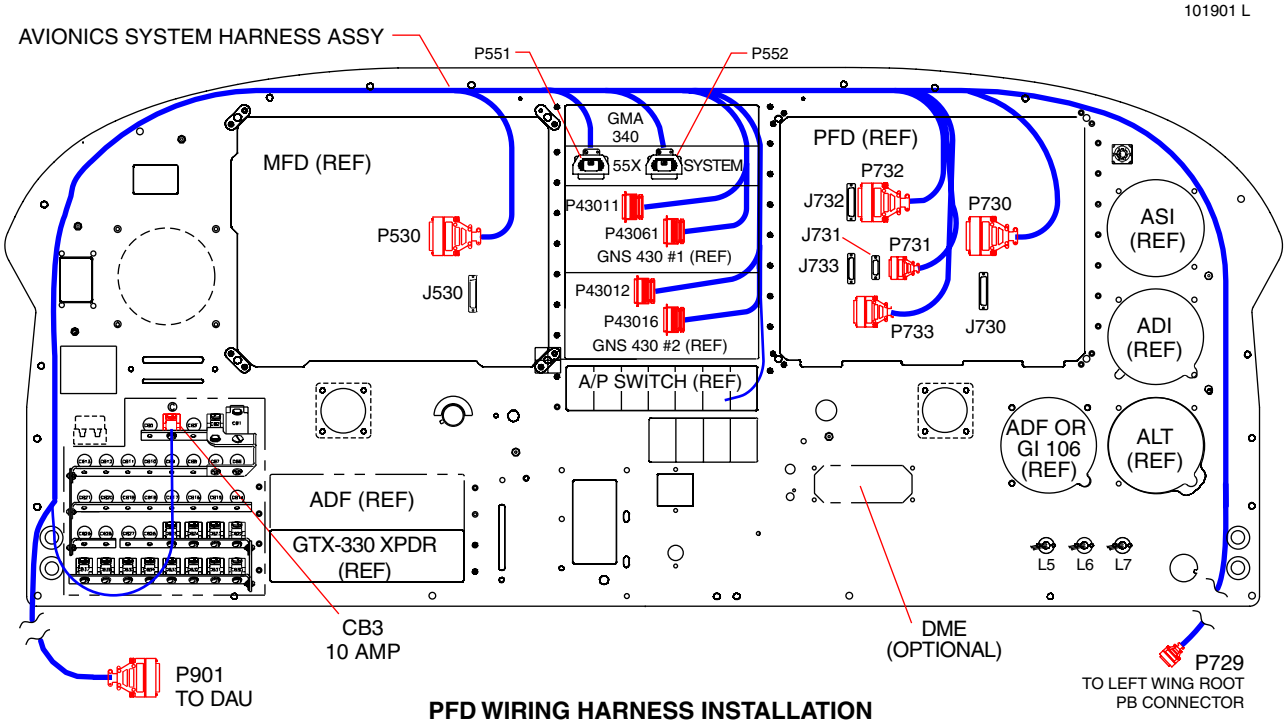
102802 A



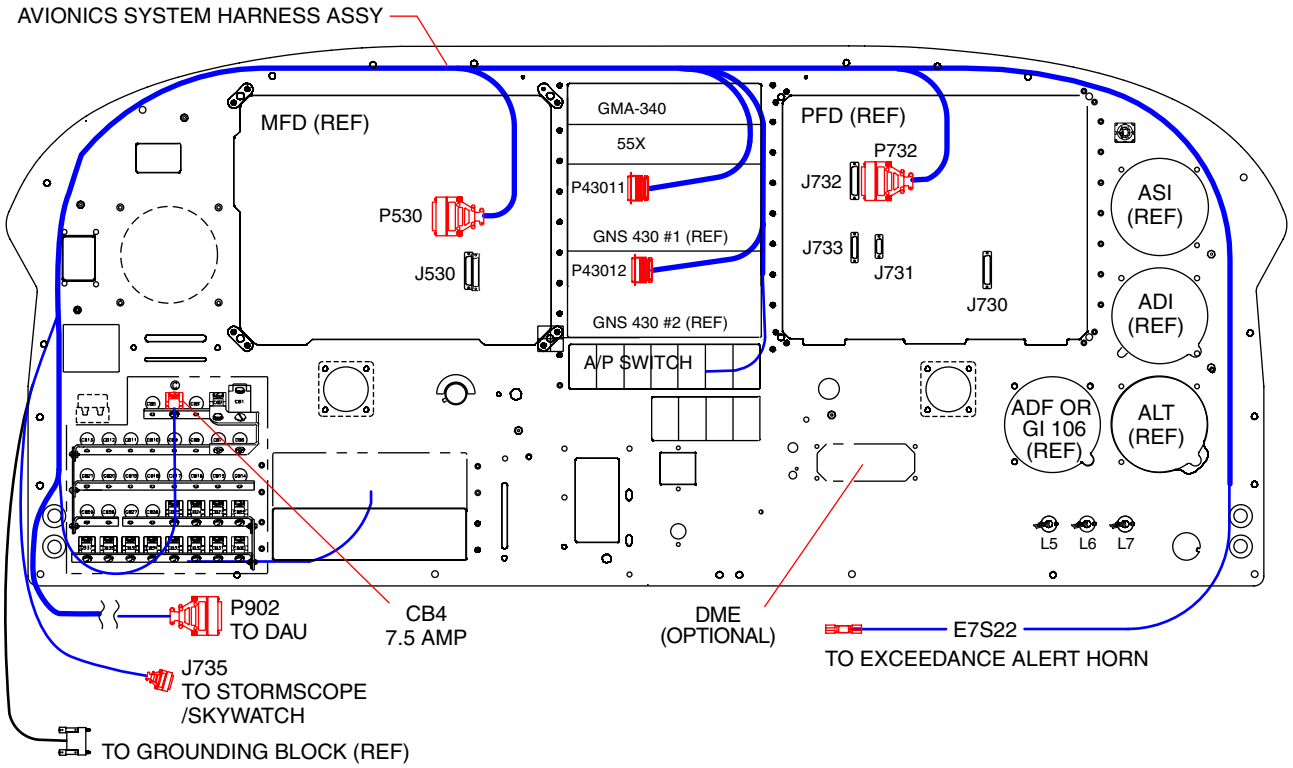
VIEW A - A

Component Locator
Figure 2 (Sheet 3 of 3)

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PFD WIRING HARNESS INSTALLATION
VIEW LOOKING AFT



MFD WIRING HARNESS INSTALLATION
VIEW LOOKING AFT

Wiring Harness Installation
Figure 3

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ASPEN AVIONICS EBD STANDBY DISPLAY

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Aspen Avionics EBD Standby Display

Installed as standard equipment in Garmin G1000 and some G1000 NXi equipped aircraft.

A. Description

The Aspen Avionics Evolution Backup Display (EBD) system provides a standby display for the primary IFDS system. The EBD contains an internal ADAHRS that is used to provide attitude and heading data for the display. The EBD as installed by Piper is connected to the emergency power buss and is powered by the emergencey battery to provide a minimum of 30 minute operation in the event of power loss under all foreseeable operating conditions. In addition, the EBD contains an internal battery which will provide an additional nominal 30 minute operation in the event emergency battery exhaustion.

Additional equipment is normally installed in support of the displays, including the Remote Sensor Module (RSM) and Configuration Module (CM).

The EBD system is configured as a basic system in which the EBD provides display of attitude, airspeed, altitude, direction of flight, vertical speed, turn rate, and turn quality. The system provides display of pilot-selectable indices (“bugs”), and annunciations to increase situational awareness and enhance flight safety.

NOTE: The terms EFD1000, EFD and EBD are used interchangeably.

B. Internal Battery Operation

When IAS is greater than 30 KIAS and the input voltage drops below 25.6 volts the EBD will automatically switch to internal battery (e.g., it assumes aircraft charging system failure).

Whenever indicated airspeed is invalid or below 30 KIAS (i.e., on the ground) the EBD will power up and power down with the application or removal of external power. A message is presented during the normal power down sequence to enable the pilot to abort the shutdown and switch to internal battery.

C. Troubleshooting

See "Chart 1" on page 3424002.

D. Inspections

(1) Annual

All units, brackets, installation hardware and wiring of the EBD system should be checked as defined below during annual inspection. Items found to be defective should be repaired or replaced prior to returning the aircraft to service.

(a) EBD Inspection

Inspect the EBD for damage and operation. Visually inspect the EBD wiring for damage, chafing, or excessive wear. The EBD braided bonding strap should be checked for proper termination at the EBD and aircraft grounding point to maintain HIRF and Lightning compliance.

Verify ≤ 3 milliohms from EBD ground stud to airframe ground. The installation of the EBD should be inspected for corrosion on the EBD and the structure it is mounted on. The fasteners should be inspected for tightness and general condition.

CHART 1 (Sheet 1 of 5)
TROUBLESHOOTING - EBD

Initialization Page Tests		
Test	**Cause**	**Remedy**
IOP.	a) Fail.	a) Replace EBD.
	b) System reboots after IOP test.	b) Replace EBD.
Config Module (CM).	a) Fail.	a) Check CM wiring. Replace CM.
	b) Wrong CM version.	b) Install correct SW version CM.
	c) System reboots after CM Test.	c) v2.0 or v2.1 display installed with a v2.2 CM. Install correct CM or EBD.
	d) Displays “Initializing” for more than 20 seconds.	d) CM unplugged or miswired.
RSM.	Fail (x).	Check RSM to EBD wiring for shorts or opens. Repair or replace RSM. Repair or replace EBD.
IMU.	Fail.	Replace EBD.
ADC.	Fail.	Replace EBD.
ADAHRS.	a) Fail.	a) Replace EBD.
	b) “Initializing” for more than 3 minutes.	b) Remove Pitot and Static line from back of EBD and reboot. If problem still exists then replace the EBD. If problem clears then repair Pitot or Static obstruction/kink.
	c) “Initializing” for more than 3 minutes with a RSM Fail above.	c) Repair RSM wiring or replace RSM.
NOTE: With software version 2.3 and later it is possible to advance the system beyond a failed Initialization Screen by holding the MENU key and line select keys #1 and #2 simultaneously for 3–4 seconds. The Diagnostic screen may then be accessed in the installation menu for further troubleshooting.		

CHART 1 (Sheet 2 of 5)
TROUBLESHOOTING - EBD

System Troubleshooting

Trouble	Cause	Remedy
Display does not power on. (NOTE: 20 second delay from the application of power to a visible display.)	a) EBD missing A/C power. there can be up to a	a) Check EBD circuit breaker, EBD on/off switch on panel, wiring, and A/C battery voltage > 11.5 volts.
	b) EBD may have been improperly shut down.	b) Switch unit off using “REV” button or “SHUT DOWN” command from Main Menu page 6.
	c) EBD missing A/C ground.	c) Check wiring to EBD.
	d) EBD is defective.	d) Repair or replace EBD.
Display does not power off. (NOTE: internal battery if airspeed is greater than 30kts.)	a) Airspeed is above 30kts. EBD will switch to	a) Normal operation.
	b) EBD may have been switched to internal battery.	b) Switch unit off using “REV” button or “SHUT DOWN” command from Main Menu page 6.
	c) EBD may have been improperly shut down.	c) Hold “REV” button for 20 seconds or unplug EBD internal battery for 3 seconds.
	d) EBD is defective	d) Repair or replace EBD.
“CONFIG MODULE LINK FAIL” message.	a) Configuration Module unplugged or miss wired.	a) Check CM plug and wiring from EBD to CM.
	b) Configuration module defective.	b) Repair or replace CM.
	c) EBD defective.	c) Repair or replace EBD.
“RSM LINK FAIL” message.	a) RSM to EBD communication lost.	a) Check RSM to EBD wiring for shorts or opens.
	b) RSM failed.	b) Repair or replace RSM.
	c) EBD failed.	c) Repair or replace EBD.
“WRONG CONFIG MODULE” message.	EBD is at one software level and config module is at a different software level.	Convert config module per appropriate service bulletin.

CHART 1 (Sheet 3 of 5)
TROUBLESHOOTING - EBD

System Troubleshooting - continued		
Trouble	**Cause**	**Remedy**
ALTIMETER, AIRSPEED, VSI FAIL (RED-X).	a) Air data sensor has not had sufficient warm-up time.	a) Allow up to 20 minutes at temps below -20ºC for flags to clear.
	b) Pitot/static lines reversed.	b) Connect pitot line to “P” port and static line to “S” port on EBD.
	c) Air data sensor failed.	c) Repair or replace EBD.
Vmo Fail.	Vmo configured incorrectly.	Re-check MOL programming.
ATTITUDE FAIL or DIRECTION FAIL (RED-X). (NOTE: up to 3 minutes to clear at temps below -20 ºC).	a) AHRS sensor has not completed initialization. Attitude flags could take	a) Allow up to 3 minutes for AHRS to initialize.
	b) RSM failed/data missing.	b) Check RSM to EBD wiring. Repair or replace RSM.
	c) Pitot and/or Static lines crossed, unplugged, or blocked.	c) Correct pitot/static plumbing. issue.
	d) EBD is defective.	d) Repair or replace EBD.
ATTITUDE FAIL and DIRECTION FAIL associated with “CHECK PITOT HEAT” message.	In Flight, Normal if pitot blockage due to ice or other.	Use pitot heat or check pitot system for blockage.
CROSS CHECK ATTITUDE message (yellow) (also see sluggish or poor AHRS (ADI) performance below)	a) If it occurred on system start.	a) RESET AHRS.
	b) Normal after abrupt maneuvers on ground or in air.	b) RESET AHRS.

CHART 11 (Sheet 4 of 5)
TROUBLESHOOTING - EBD

System Troubleshooting - continued		
Trouble	**Cause**	**Remedy**
"ERRONEOUS CALIBRATION VALUES" message during RSM Cal or Excessive Heading errors in one quadrant, or errors that are higher than actual in some quadrants and lower than actual in other quadrants.	a) RSM is tilted more than: ±4º to the aircraft longitudinal axis; ±10º to the lateral (wings level) axis; ±10º to the zero pitch axis.	a) Shim RSM to within limits stated.
	b) Poor RSM calibration.	b) Re-run RSM calibration at constant rate turns on flat ground.
	c) RSM calibrated too close to buildings or ferrous objects.	c) Re-run RSM calibration away from buildings and other ferrous objects.
	d) Ferrous hardware used to mount RSM.	d) Only non-ferrous screws, nuts, washers may be used on RSM.
	e) Airframe or external magnetic interference.	e) Check for magnetized areas on airframe close to RSM. Verify no ferrous hardware is near RSM. Degauss magnetized area(s).

CHART 11 (Sheet 5 of 5)
TROUBLESHOOTING - EBD

System Troubleshooting - continued

Trouble	Cause	Remedy
Sluggish or Poor AHRS (ADI) performance. Poor AHRS performance in steep bank turns. Sluggish compass card. (NOTE: movement associated with "Cross Check Attitude" message.	a) RSM magnetic interference. May or may not be	a) Survey RSM location using handheld compass. Move the compass around the location looking for needle deflection. There should be no more than 2º of compass needle movement within an area 18" x 18" around the location. If excessive needle is found, degauss the affected area.
	b) RSM has become magnetized.	b) With power removed from EBD system degauss RSM and general area.
	c) "Pitch Attitude Trim" or "Panel Tilt Pitch Compensation" adjustment made without performing a subsequent RSM Calibration.	c) Perform an RSM Calibration.
	d) Pitot and/or Static line connections at EBD blocked, kinked, or unplugged.	d) Check pitot/static connections and plumbing for blockage. Check IAS and ALT sensor per Ground Test Procedure.
	e) Normal after abrupt maneuvers.	e) Perform AHRS Reset.
Excessive Heading Lead / Lag during or after turns (>7°).	Magnetic Interference.	Check for RSM magnetic interference as in "Sluggish or Poor AHRS (ADI) performance", above. If that does not correct the issue, contact Piper Customer Service.

(b) EBD Internal Battery Test

Perform the EBD Internal Battery Test under Ground Test Procedure.

(c) RSM Inspection

Visually inspect the RSM for damage and wear on the bonding strap. RSM wiring should be checked for damage, chafing, or excessive wear. Verify RSM bonding from the ground stud to access plate ground is ≤ 3 milliohms to maintain HIRF and Lightning compliance. The RSM installation should be inspected for corrosion on the RSM and the wing access plate. The installation should be inspected for cracks and loose or damaged fasteners.

(d) Configuration Module Inspection

Visually inspect the Configuration Module(s) for damage. Visually inspect the Configuration Module wiring for damage, chafing, or excessive wear.

(2) Each Twenty-Four (24) Calendar Months

Use procedures in Aspen Avionics Tech Note 2009-09, Rev. B or later, Document Number 991-00023-001, Tech Note for Air Data Calibration to test and inspect the static pressure system and altimeters. Ensure compliance with the requirements of FAR 43, Appendix E. (See FAR 91.411.)

(3) Each Three Years or 2,200 Hours, Whichever Comes First

Replace the EBD internal battery per EBD Battery Replacement under Components, EBD Display, below.

E. Components

(1) EBD Display

The EBD is a digital system that consists of a high resolution 6” diagonal color LCD display, user controls, photocell and Micro SD data card slot. The rear portion of the EBD includes a non-removable electronics module which contains a full air data computer, attitude heading reference system, power supplies, backup battery, and dual processor electronics. Also on the rear of the unit, a fan is provided to cool the backlight and electronics.

The EBD mounts to the front surface of the instrument panel. The electronics module and cooling fins on the back are sized to fit into instrument panel holes. A recess-mount bracket mounts the displays nearly flush with the instrument panel.

The instrument is installed in a vertically oriented pair of instrument openings, without interfering with the surrounding instruments. The EBD contains a microSD card port and reader at the bottom of the display bezel. When authorized, software updates and system upgrades can be installed using the card port. The port is also used for database information in the EBD MFD.

The EBD is a digital system and supports the ARINC 429 digital interface. The PFD connects directly to the interfaced equipment.

(a) Removal

Verify power is off. Carefully insert a flat blade screw driver into the locking mechanism on the top center of the EBD. While gently prying pull back the top of the EBD and extract from bracket. Remove nut securing braided ground strap to EBD. Remove pitot and static quick connectors (EBD only) by pulling back outer spring loaded locking sleeve while unplugging connectors. To remove 44 pin D-sub connector unscrew both jackscrews fully and pull connector straight back.

(b) Installation

Verify power is off. Install 44 pin D-sub connector and tighten jackscrews until connector is fully seated. Install pitot and static lines (EBD only) to back of EBD by firmly pressing the fitting until fully seated (pitot and static quick connectors are keyed and cannot be crossed). Gently pull on connector to ensure proper connection. Connect braided bonding strap to EBD with nut. Insert bottom of EBD into bracket and pivot top forward until it locks into place on bracket.



EBD Standby Display
Figure 1

(c) Return to Service

1) Verify all system interfaces are functional.

2) Verify proper bonding per Bonding Check under Post Installation Setup Procedure, below.

3) Perform a System Leak Test per Ground Test Procedure, below.

(d) EBD Battery Replacement

EBD battery replacement must only be performed by a properly certified individual or facility. Remove EBD from panel as above. Remove two screws on each end of the football shaped cover plate on rear of the EBD. Unplug electrical connector and slide battery out of EBD. Install new battery in EBD then connect battery plug. Replace cover plate and tighten the cover screws. Tighten to 12 in-lbs. Reinstall and test EBD as above.

(e) EBD Knob Replacement Procedure

The knob(s) are removed by gripping the knob with the fingers of your hand and then pulling straight off the shaft using moderate force. Install the new knob P/N 520-00020-001 by pushing the knob on to the shaft with the fingers of your hand until it bottoms out. After knob replacement power up the system and verify the knob can turn freely and that the “push to enter” function operates properly by accessing the HDG bug or CRS pointer function.

(2) Remote Sensor Module (RSM)

The RSM is required for each EBD and connects directly to the display. It physically resembles a traditional GPS antenna.

The RSM contains all of the sensors that must be remotely located from the EBD display unit. It is mounted in a magnetically quiet environment. The RSM is installed internally on an access cover on the underside of the aircraft right wing.

The RSM is powered by the EBD through a shielded wire harness and contains the Magnetic “flux” sensors.

(a) Removal

CAUTION: DO NOT USE A MAGNETIC TIPPED SCREW DRIVER WHEN REMOVING AND REPLACING THE RSM.

1) Verify power is off.
2) Remove screws (8 ea.) securing access cover to wing skin. Take care to support access cover as it becomes detached from wing.
3) Disconnect RSM electrical connector from wing wiring harness and remove RSM and access cover from airplane.
4) To remove RSM from mounting bracket, first disconnect ground wire from mounting bracket stud. Then remove four (4) non-ferrous screws, washers, and nut from RSM and separate RSM from mounting bracket taking care to feed wiring pigtail through hole in bracket.

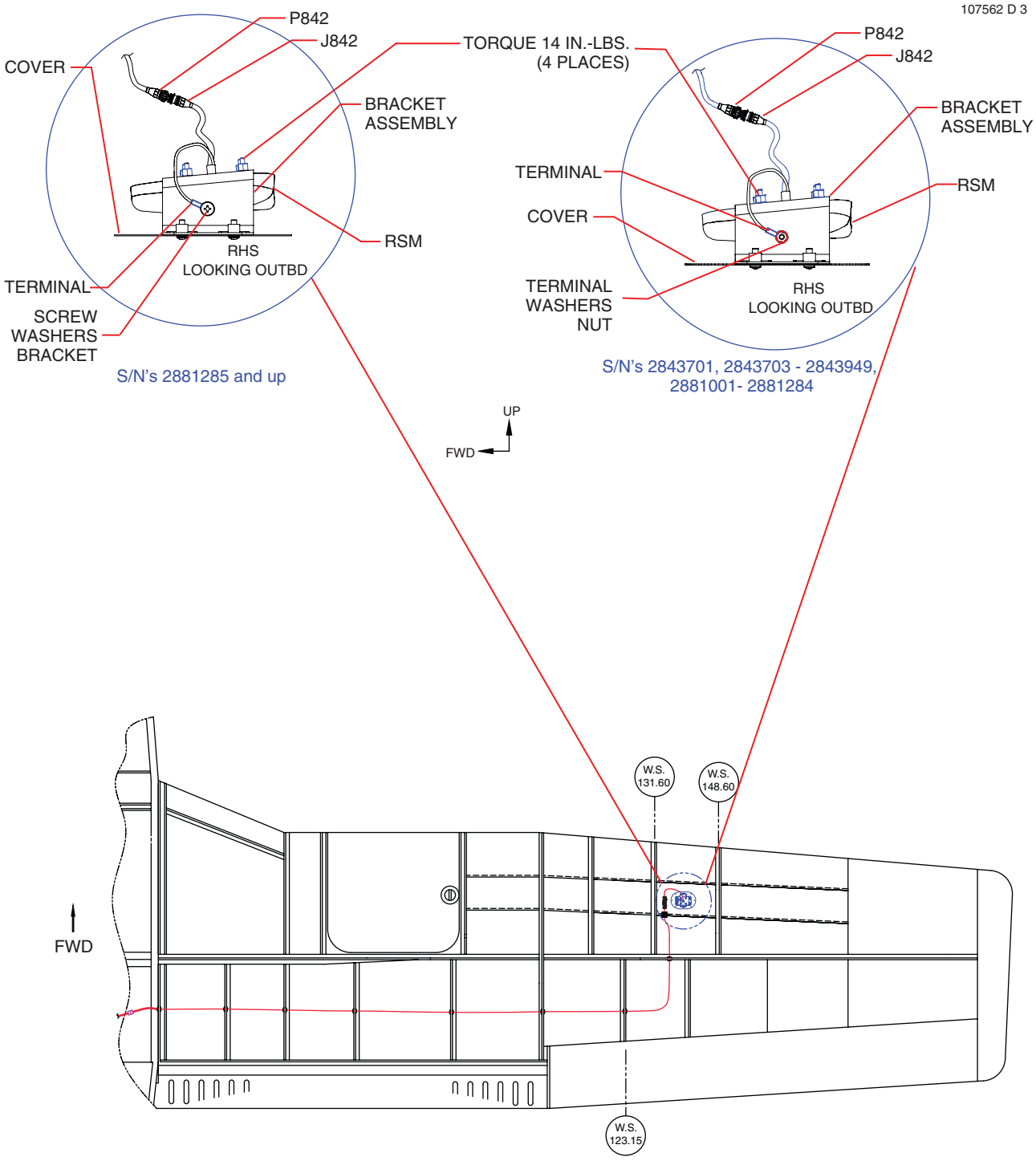
(b) Installation

CAUTION: DO NOT USE A MAGNETIC TIPPED SCREW DRIVER WHEN REMOVING AND REPLACING THE RSM.

1) Verify power is off.
2) If RSM has been separated from its mounting bracket, feed wiring pigtail through hole in bracket and place RSM in bracket with flat end forward (see arrow on mounting bracket). Secure with four (4) non-ferrous screws, washers, and nut. Torque as specified in "Figure 2".
3) Secure the RSM ground wire to stud on mounting bracket.
4) Connect RSM electrical connector to wing wiring harness.
5) Verify proper bonding per Bonding Check under Post Installation Setup Procedure, below.
6) Place RSM and access cover in position at wing skin opening, ensuring the arrow is pointing forward.
7) Secure access cover to wing skin with non-ferrous screws (8 ea.)

(c) Return to Service

Perform RSM Calibration.



RHS LOOKING DOWN

RSM Installation
Figure 2

(3) Configuration Module (CM)

The Configuration Module retains system configuration settings and calibration data. The Configuration Module connects to the EBD through a short fabricated harness and is fastened to the main wiring bundle of the display.

Each display has an associated Configuration Module that retains that display's aircraft specific configuration, calibration data and user settings. This permits the display hardware to be replaced without re-entering the installation settings or re-calibrating the EBD.

(a) Removal

Verify power is off. Cut the two (2) cable ties affixing the CM to the PFD wiring harness. Unplug the Molex connector by pressing down on the locking tab and gently pulling the connector from the module.

(b) Installation

Verify power is off. Plug the Molex connector into the module until it clicks. Cable tie the module to the PFD wiring harness.

(c) Return to Service

1) Configure unit per Installation Menu – Unit Configuration under Post Installation Setup Procedure, below.

2) Perform RSM Calibration under Post Installation Setup Procedure, below.

F. Post Installation Setup Procedure

(PIR-107821, Rev. B.)

(1) Configuration and Equipment Checkout

The following Test Equipment will be required to complete the remaining steps in the ground test procedure:

- Pitot Static Test Set
- Digital Multimeter
- Electrical Bonding Tester

(2) Bonding Check

(a) Verify EBD mounting bracket is bonded to instrument panel with no greater than 3 milliohms resistance.

(b) Verify RSM base plate is bonded to access cover with no greater than 3 milliohms resistance.

(3) Initial System Turn On

(a) Remove power by pulling applicable circuit breakers.

(b) Install the EBD, RSM, and Configuration Module, if required.

(c) Push in all applicable circuit breakers and apply power.

(d) Verify the EBD displays the initialization page after 5 seconds.

NOTE: AHRS Flags may take up to 3 minutes to clear. Airspeed and Altitude flags may take up to 20 minutes to clear at temperatures below -20ºC.

(4) System Configuration

Configure the EBD system prior to running the ground test procedure. The configuration pages are accessed through the EBD display using the MENU button and the lower Right Control Knob labeled MODE/SYNC.

(a) Main Menu Access

The Main Menu operation is accessed by pushing the “MENU” button. See EBD Pilot Guide for Main Menu operation.

(b) Menu Navigation

When no fields are enabled for editing, rotating the right control knob clockwise advances to the next menu page and counterclockwise advances to a previous menu page.

Editable menu items are displayed in white text on a blue background, non-editable menus items are green text on a blue background while grey text on a blue background is disabled from editing.

(c) Edit Mode

Pushing the line select key adjacent to an editable field enables the associated field for editing. The field turns magenta when enabled and the right control knob reads “Edit Value”.

When the field is enabled for editing rotating the right control knob will adjust the value. Pushing the right control knob or the adjacent line select key will exit from the editable field.

(d) Main Menu Configuration

The Main Menu consists of pages that are pilot selectable. Refer to the EBD Pilot Guide for Main Menu configuration if necessary.

(e) Approved Software

Installation of approved software can be verified as follows:

1) Push the “Menu” key.
2) Rotate the right control knob clockwise until the system status page (see "Figure 3") appears.
3) Compare displayed software versions with "Chart 2".

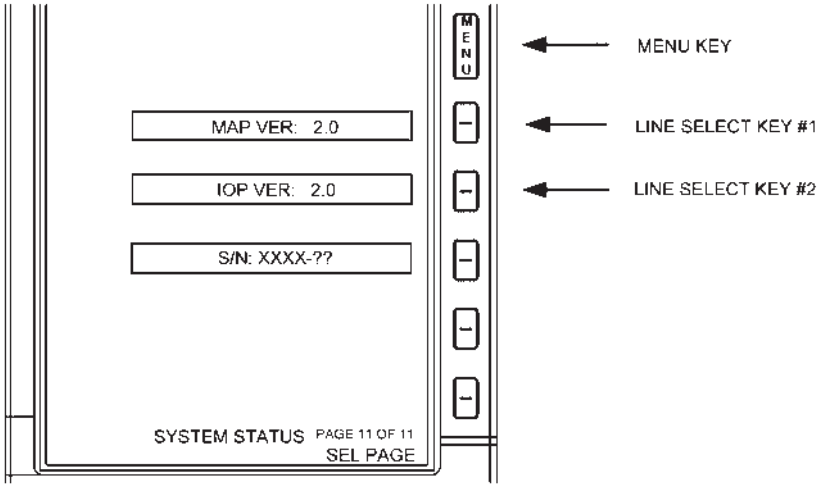
(f) Installation Menu – Unit Configuration

The Installation Menu is entered from the Main Menu’s “SYSTEM STATUS PAGE” (last page). Simultaneously push and hold the MENU key, Line Select Key #1 and Line Select Key #2 for 3 seconds while the airspeed is below 30 units.

Whenever the warning message in "Figure 4" is displayed, pressing either control knob shall advance the Installation menu.

To exit the Installation Menu at any time press the MENU button. All data will be saved as displayed. The system will reboot and “INITIALIZING” will appear on the display for approximately 40 seconds.

Configure the EBD system per the following instructions.



Example Installation Menu Access
Figure 3

CHART 2
APPROVED SOFTWARE

Identifier	Version	Part Number
MAP	2.6.3	302-00007-015
MAP	2.9	302-00007-026(1)
IOP	2.0.5	302-00013-005
IOP	2.1.1	302-00013-007(1)

PIR-107757-002, Rev. D.

(1) Factory installed in late 2017 and up.



Installation Menu Warning
Figure 4

(a) EBD Installation Menu Configuration

Installation Date: _____

Aircraft Model: _____ EBD S/N: _____

Aircraft Type: _____ RSM S/N: _____

Aircraft S/N: _____ ACU S/N: _____

CM S/N: _____

1) INSTALLATION MENU PAGE - IAS CONFIG A

Feature	Options	Actual Setting
MOL TYPE	1,2,3,4,5,6	1
MOL PARAM SEL	See instructions in table below	V_{ne}
MOL PARAM VALUE	See instructions in table below	154
MOL MARKER/BAND	BARBER POLE, REDLINE	REDLINE
MACH DSPL TYPE	ALTITUDE, SPEED	SPEED

2) INSTALLATION MENU PAGE – IAS CONFIG B - Set Speed Bands

Feature	Options	Actual Setting
OVERSPEED ALERT	DISABLE, ENABLE	DISABLE
Vno	0 to 450	125
Vfe	0 to 450	102
Vs	0 to 450	50
Vso	0 to 450	45

3) INSTALLATION MENU PAGE – IAS CONFIG C - Set Speed Markers

Feature	Options	Actual Setting
Vyse	0 to 450	0
Vmc	0 to 450	0
Triangle	0 to 450	0
Not Used		
Not Used		

4) INSTALLATION MENU PAGE - IAS CONFIG D - Rotorcraft Only

5) INSTALLATION MENU PAGE - IAS CONFIG E - Color of Airspeed Tape Speed Bands

Feature	Options	Actual Setting
SPD Band 2	YELLOW, CLEAR	YELLOW
SPD Band 3	GREEN, WHITE, CLEAR	GREEN
SPD Band 4	WHITE, CLEAR	WHITE
Not Used		
Not Used		

6) INSTALLATION MENU PAGE – IAS CONFIG F

Feature	Options	Actual Setting
IAS UNITS	KTS, MPH	KTS
TAPES	UNLOCKED, LOCK OFF, LOCK ON	LOCK ON
VSPD EDIT	UNLOCKED, LOCKED	LOCKED
Not Used		
Not Used		

7) INSTALLATION MENU PAGE - IAS CONFIG G

Feature	Options	Actual Setting
IAS DISPLAY	DISABLE, ENABLE	DISABLE
ALT DISPLAY	DISABLE, ENABLE	DISABLE
BARO DISPLAY	DISABLE, ENABLE	ENABLE
VSI DISPLAY	ALWAYS ON, ALWAYS OFF, AUTO	ALWAYS ON
Not Used		

8) INSTALLATION MENU PAGE – NAV SETUP A

Feature	Options	Actual Setting
GPS / NAV #1	NONE,A,B,C,D,E,F,G,H,I,J,K,L,M,P,Q,R,S	NONE
GPS / NAV #2	NONE,A,B,C,D,E,F,G,H,I,J,K,L,M	NONE

9) INSTALLATION MENU PAGE – NAV SETUP B

Feature	Options	Actual Setting
429 IN PORT 1	NONE, GPS1, VLOC1, VLOC1+ACU, GPS1+ACU, GPS1+ACU+VLOC1, GPS2, VLOC2, VLOC2+ACU, GPS2+ACU, GPS2+ACU+VLOC2	NONE
429 IN PORT 2	Same as IN PORT 1 options	NONE
429 IN PORT 3	Same as IN PORT 1 options	NONE
429 IN PORT 4	Same as IN PORT 1 options	NONE
429 IN PORT 5	NONE, TRAFFIC	NONE

10) INSTALLATION MENU PAGE – NAV SETUP C

Feature	Options	Actual Setting
429 OUT PORT SPEED	HIGH, LOW	LOW
429 IN PORTS 1 & 2 SPEED	HIGH, LOW	LOW
429 IN PORTS 3 & 4 SPEED	HIGH, LOW	LOW
ADF CONFIG	NONE, 1, 2	NONE
NAV #2 POSITION SOURCE	GPS1, GPS2	GPS1

11) INSTALLATION MENU PAGE – RS232 PORT CONFIG A

Feature	Options	Actual Setting
232 IN PORT 0	NONE, RSM, GPS TYPE 4, GPS TYPE 5, DFC A/P	NONE
232 IN PORT 1	NONE, GPS TYPE 1, GPS TYPE 2, GPS TYPE 3, GPS TYPE 4, GPS TYPE 5, XM, DFC A/P	NONE
232 IN PORT 2	NONE, GPS TYPE 1, GPS TYPE 2, GPS TYPE 3, GPS TYPE 4, GPS TYPE 5, DFC A/P	NONE
232 IN PORT 3	NONE, WX500, GPS TYPE 4, GPS TYPE 5, DFC A/P	NONE
232 IN PORT 4	NONE, EFD1000 MFD, EFD1000 PFD, EFD500 MFD	NONE

12) INSTALLATION MENU PAGE – RS232 PORT CONFIG B

Feature	Options	Actual Setting
232 IN PORT 5	NONE, EFD1000 MFD, EFD1000 PFD, EFD500 MFD	NONE
232 OUT PORT 0	NONE, RSM, GPS TYPE 4, GPS TYPE 5, DFC A/P 1, DFC A/P 2	NONE
232 OUT PORT 1	NONE, XM, ADC TYPE 1, ADC TYPE 2, GPS TYPE 4, GPS TYPE 5	NONE
232 OUT PORT 2	NONE, EFD, ADC TYPE 1, ADC TYPE 2, GPS TYPE 4, GPS TYPE 5, DFC A/P 1, DFC A/P 2	NONE
232 OUT PORT 3	NONE, WX500, ADC TYPE 1, ADC TYPE 2, GPS TYPE 4, GPS TYPE 5, DFC A/P 1, DFC A/P 2	NONE

13) INSTALLATION MENU PAGE – RS232 CONFIG C

This page is grayed out and not configurable.

14) INSTALLATION MENU PAGE – ACU CONFIG A

Feature	Options	Actual Setting
ACU HSI TYPE	0,1,2,3	0
ACU FD TYPE	0,1,2,3,4,5,6,7,8,9,10,11,12	0
ACU DATUM	NORMAL,REVERSED	NORMAL
FD PITCH OFFSET ADJ	-10.0 to +10.0 (degrees)	0.0
FD ROLL OFFSET ADJ	-10.0 to +10.0 (degrees)	0.0

15) INSTALLATION MENU PAGE – ACU CONFIG B

Feature	Options	Actual Setting
Digital Nav Tx Speed	HIGH. LOW	LOW
Digital Nav Rx Speed	HIGH, LOW	LOW

16) INSTALLATION MENU PAGE – MISC CONFIG A

Feature	Options	Actual Setting
RSM Orientation	TOP, BOTTOM (Inverted orientation)	BOTTOM
RSM GPS Enable	DISABLE, ENABLE, INTERCOM	DISABLE
RSM GPS USAGE	EMER ONLY/ MODE 2	EMER ONLY
RAD ALT CONFIG	NONE, TYPE 1, TYPE 2, TYPE 3, TYPE 4, TYPE 5, TYPE 6, TYPE 7	NONE
WIND DISPLAY	DISABLE, ENABLE>=30KIAS, ENABLE>=40KIAS, ENABLE>=50KIAS, ENABLE>=60KIAS, ENABLE>=70KIAS, ENABLE>=80KIAS, ENABLE>=90KIAS	DISABLE

17) INSTALLATION MENU PAGE – MISC CONFIG B

Feature	Options	Actual Setting
ELEC SYSTEM	14 VOLT, 28 VOLT	14 VOLT
EFD BATTERY CONFG:	INTERNAL, REMOTE	REMOTE*
ATTITUDE REF SYMBOL ADJ:	-5.0 to +5.0 degrees	0.0
PANEL TILT PITCH ADJ	-10.0 to +20.0 degrees	0.0
PANEL ROLL ADJ	-2.0 to +2.0 degrees	0.0

18) INSTALLATION MENU PAGE – MISC CONFIG C

Feature	Options	Actual Setting
COMPOSITE PHASE (VOR1,2)	(0,0) (180,0) (0,180) (180,180)	0,0
GPSS GAIN:	0.5 to 2.0 (0.1 increments)	1.0
CRS SDI	COMMON, NAV 1/2	COMMON
OBS DISPLAY	DISABLE, ENABLE	DISABLE

19) INSTALLATION MENU PAGE – MISC CONFIG D

Feature	Options	Actual Setting
OAT SOURCE	NONE, RSM, PROBE, INTERCOM	NONE
OAT BIAS	0 to -8 degrees	0.0

20) INSTALLATION MENU PAGE – RSM CALIBRATION

The following menu will be used in the next section to calibrate and validate the magnetometer in the RSM. Heading errors of up to +/- 6.0º can be calibrated at 30º increments beginning with North. Follow RSM Calibration under Post Installation Setup Procedure, below.

Feature	Options
START CALIBRATION	Press to initiate
ACCEPT CALIBRATION?	Press to ACCEPT Cal
REJECT CALIBRATION?	Press to REJECT Cal
HDG SEL	030° to 360° (in 30° increments)
HDG ADJ	-6.0 to +6.0 (degrees)

Use the following table to record the HDG ADJ values used to bring the compass heading values into specification:

Rsm Calibration	Page 12	Options	Actual Setting
HDG SEL: 030	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 060	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 090	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 120	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 150	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 180	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 210	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 240	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 270	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 300	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 330	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____
HDG SEL: 360	KEY 4		
HDG ADJ:	KEY 5	Editable: -6 to +6	_____

21) INSTALLATION MENU PAGE – ACCEL BIAS CAL
This page is for Factory Calibration only.

22) INSTALLATION MENU PAGE – WX-500

Feature	Options	Actual Setting
HEAD TYPE	NONE, DISPLAY, CONTROL	NONE
SYSTEM DATA	PAGE1, PAGE2, PAGE3, PAGE4	
STRIKE TEST	Press to Initiate	
NOISE MONITOR	Press to Initiate	
ANTENNA MOUNT	TOP, BOTTOM	

23) INSTALLATION MENU PAGE – NETWORK PAGE
This menu is for diagnostic purposes only.

Feature	Options
IP ADDR	0-255
SUBNET MASK	0-255
GATEWAY	0-255
PORT	0-9999
Not Used	

NOTE: Repeated presses of top 3 line select keys will select one of 4 selectable fields.

24) INSTALLATION MENU PAGE - DIAGNOSTICS

This menu is for diagnostic purposes only. See "Figure 5".

NOTE: Operational Time and Flight Time are stored in the configuration module of the aircraft and these times are cumulative for the installation not the EBD unit itself.

25) EXITING / SAVING DATA

To exit the Installation Menu press the MENU button. All data will be saved as it was displayed on each page. The system will reboot and “INITIALIZING” will appear on the display for approximately 40 seconds. Normal operation continues.



Diagnostics
Figure 5

(b) IAS Configuration

1) Definitions

V_{ne}: Never Exceed speed (beginning of red band)

V_{no}: Maximum Structural Cruise speed (beginning of yellow band).

V_{fe}: Maximum Flap Extend speed (top of white band)

V_s: No Flap Stall speed (bottom of green band)

V_{so}: Full Flap Stall speed (bottom of white band)

V_{yse}: Not Applicable to Single Engine Airplanes

V_{mc}: Not Applicable to Single Engine Airplanes

◁ Initial Flap Extension Speed

2) Use the following table to fill in Configuration Menu Page 1:

INDICATED AIRSPEEDS (IAS) CONFIGURATION

IAS Setting (KIAS)	Band Color	Band Range	Description	Pass
V_{ne} = 154	Red	>V_{ne}	Red arc displayed at all speeds above V_{ne}	_____
V_{no} = 125	Yellow	V_{no} - V_{ne}	Yellow arc extending from V_{no} to V_{ne}	_____
V_s = 50	Green	V_s - V_{no}	Green arc extending from V_s to V_{no}	_____
V_{fe} = 102	White	V_{so} - V_{fe}	Top of White arc	_____
V_{so} = 45	White		Bottom of White Arc	_____
V_{yse} = 0	Blue Marker	= V_{yse}	Blue Marker at V_{yse}	_____
V_{mc} = 0	Red Marker	= V_{mc}	Red Marker at V_{mc}	_____
◁ = 102	Triangle (White)	= ◁	White triangle at initial flap extension airspeed	_____

3) ATTITUDE REF SYMBOL ADJ

This is used to make small changes to the pitch attitude reference mark on the EBD should the aircraft fly consistently slightly nose high or nose low in cruise trim. It is not necessary to perform the RSM Calibration after making changes to this setting.

4) EXITING / SAVING DATA

To exit the Installation Menu press the MENU button. All data will be saved as it was displayed on each page. The system will reboot and “INITIALIZING” will appear on the display for approximately 40 seconds. Normal operation continues.

(5) RSM Calibration

(a) Calibration Overview

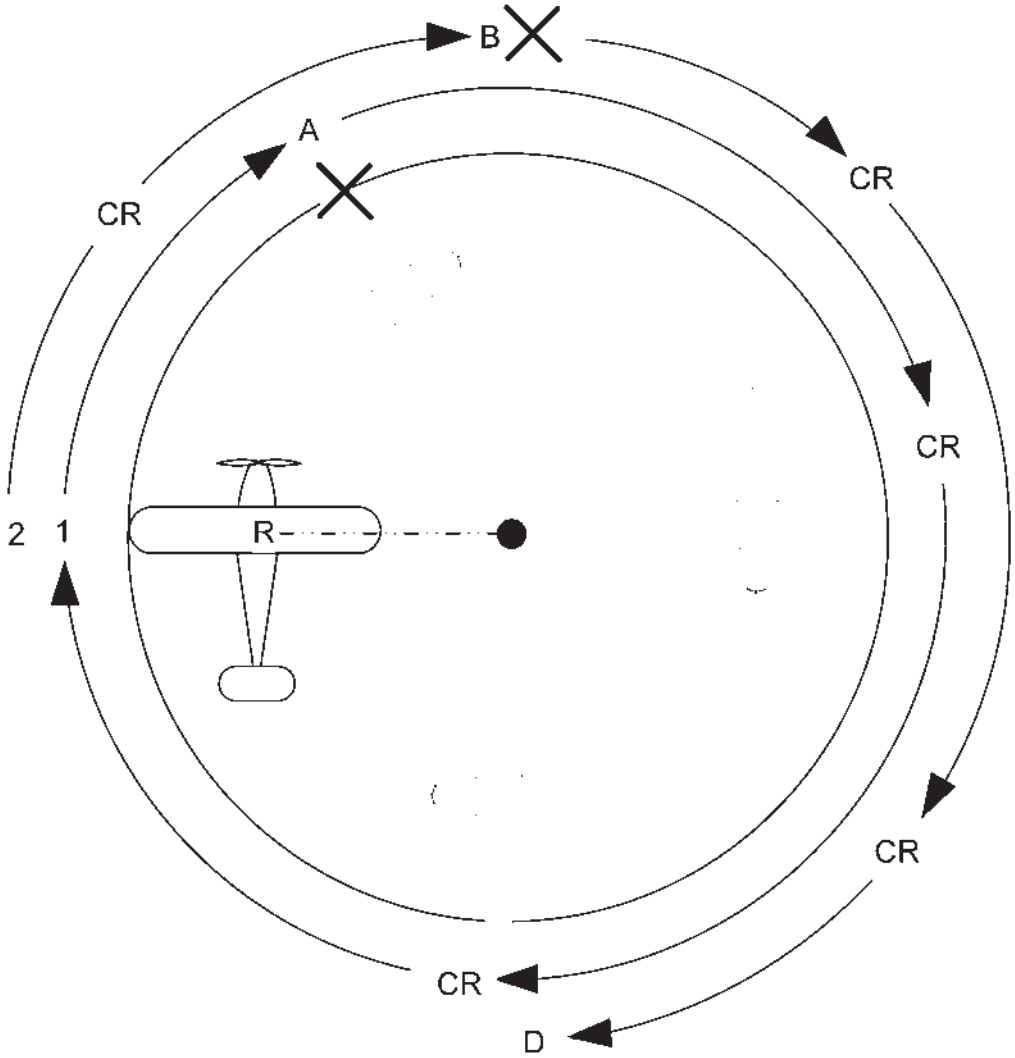
The Remote Sensor Module must be calibrated by performing a compass swing in the aircraft for any maintenance activities that could affect RSM accuracy.

Such activities might include but are not limited to the replacement of the RSM, replacement of the Configuration Module, installation of a mechanical or electrical device in the vicinity to the RSM, installation of an appliance that might generate a magnetic interference.

An overview of the RSM Calibration procedure is as follows (see "Figure 6"):

CAUTION: WHEN A CALIBRATION IS INITIATED, THE AIRCRAFT MUST BE TURNED AS DESCRIBED BELOW. IF THE CALIBRATION IS INITIATED AND THEN ACCEPTED WITHOUT MOVING THE AIRCRAFT, AN ERRONEOUS CALIBRATION WILL BE WRITTEN TO THE CONFIGURATION MODULE, POTENTIALLY RESULTING IN FAILURE TO INITIALIZE.

- A heading will be checked to verify the reasonableness of EBD heading prior to calibration.
- The aircraft will be taxied to a magnetically quiet and level area at least 200ft from metal buildings and clear of metal grates, manhole covers and rebar within the concrete. A Compass Rose is ideal for this procedure.
- The aircraft can start from any heading.
- With engines running, all electrical equipment operating, and the aircraft stationary the RSM CAL page will be entered and Start Calibration will be initiated (see "Figure 7").
- After a 15 second countdown timer the pilot/operator will begin to taxi the aircraft in a circle (cw or ccw) with the radius of approximately twice the length of the aircraft's wing as viewed from the cockpit (≈ 30ft).
- The aircraft will be taxied under its own power at a constant rate around a circle until a 60 second timer elapses. The aircraft must not stop until the timer has exhausted.
- At the completion of the 60 seconds the aircraft will have made at least a 450° circle (360° + 90°) to approximately two complete circles (720°).
- At the end of the 60 second timer four headings about 90° apart will be checked against a calibrated heading source (i.e., site compass, compass rose).
- If EBD heading is acceptable then the calibration is Accepted.
- If the EBD heading is not within tolerance then it is Rejected and the calibration procedure is re-run.
- After the calibration is accepted headings are checked using a calibrated reference (i.e., a sight compass) every 30° (starting from North) to verify that the heading accuracy is within ±4°.
- Heading can be calibrated every 30° if necessary.



● = Imaginary pivot point
R = Radius to pivot point – approximately equal to wingspan (≈30ft)
1 = Starting Point
A = Established Constant Rate turn ≈ 10 seconds after 1
CR = Constant Rate Turn must continue until end of 60sec timer
2 = 360° turn complete – continue at Constant Rate – Do not stop
B = 60sec timer must not reach Zero before this point
D = Accept/Reject CAL message occurs between B & D

RSM Calibration Graphic
Figure 6

(b) Calibration Procedure

The RSM calibration routine is accomplished using the Installation Menu "RSM CALIBRATION" menu page. See Installation Menu - Unit Configuration, above for instructions on entering the INSTALLATION MENU.

On the "RSM CALIBRATION" menu page the current calibrated heading (to the nearest 0.1 degrees) will be continuously displayed adjacent to the "CAL HDG:" menu field.

NOTE: Verify CAL HDG is within 25 degrees of a calibrated compass source prior to starting the RSM Calibration. This is a rough check of the hard and soft iron effects (magnetic interference) in the vicinity of the RSM. Should the CAL HDG be greater than 25 degrees then the RSM location should be re-surveyed.

With aircraft stationary at (POSITION 1) of "Figure 6" – RSM Calibration Graphic press the "START CALIBRATION" line select key. The annunciation shown in "Figure 8" will be displayed with a countdown timer that begins with 15 secs and counts down to 0 secs.

CAUTION: DO NOT PRESS ACCEPT CALIBRATION WITHOUT MOVING THE AIRCRAFT IN THE PROCEDURE BELOW AS CORRUPTION TO THE CONFIGURATION MODULE COULD OCCUR.



RSM Calibration Page
Figure 7

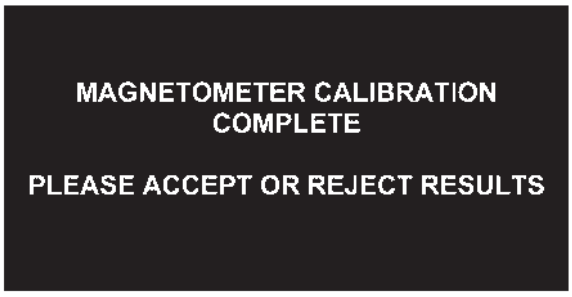


Calibration in Process
Figure 8



Aircraft Turning
Figure 9

When the menu of "Figure 9" is displayed immediately begin taxiing the aircraft clockwise or counter-clockwise at a constant rate of no faster than 1 turn every 30 seconds. About half normal taxi speed or a brisk walking speed is about right. Approximately 10 seconds after initial movement (see POSITION A) the aircraft should be taxiing at a constant rate (CR) throughout the rest of the procedure. When the countdown timer is reached between one and a quarter turns (450º) (POSITION B) and two turns (720º) (POSITION D) should have been completed. While turning the aircraft do not stop the aircraft until the end of the 60 second timer and "Figure 10" appears.



Accept/Reject Results
Figure 10

If you find that the timing of the turns was not right such that “Magnetometer Calibration Complete” message occurs between B & D of "Figure 6", then REJECT the results and re-run the procedure.

At the end of the calibration routine the “ACCEPT CALIBRATION?” and “REJECT CALIBRATION?” menu options will be enabled.

NOTE: If the message “Erroneous Calibration Values” is displayed then magnetic interference exists in the vicinity of the calibration area or the RSM is mounted in a magnetically noisy area. Try the calibration process again in a flat magnetically quiet area. If the message is displayed again the RSM location must be surveyed for magnetic interference.

To determine whether to ACCEPT or REJECT the results check four headings approximately 90° apart against a known good heading source (i.e., aircraft compass, sight compass, compass rose).

If the headings are within ±10° then press ACCEPT and use the Heading Offset Adjustment, below, to align each heading value to actual.

NOTE: Typically with a clean RSM location the heading values will be within 3 degrees of actual heading. If they are outside this value the RSM mounting area should be rechecked for magnetic contamination (internal interference) or the calibration can be rerun in a new location (external interference) to see if the values are improved. RSMs tilted to the upper end of the limit may not be able to achieve the 3 degree or less of error without using the Heading Offset Adjustment).

Pressing the “ACCEPT CALIBRATION” selection shall accept the calibration results, display the annunciation shown in "Figure 11" for 5 seconds, and return the “RSM CALIBRATION” menu page to its initial state.

Pressing the “REJECT CALIBRATION” selection shall reject the calibration results. Reject the results if the calibration was poor or a previously stored calibration has better heading accuracies.



Results Accepted
Figure 11



Results Rejected
Figure 12

(c) Heading Offset Adjustment

When the calibration routine is complete and the results have been accepted the CAL HDG value will be displayed.

Position the aircraft so that it is at a heading of 30° as verified by a sight compass or other calibrated means. Press the line select key next to “HDG SEL” and then turn the right knob until HDG SEL: 030° is displayed. Now press the line select key next to “HDG ADJ” and turn the right knob until the CAL HDG of "Figure 13" displays exactly 030.0° as shown in "Figure 14" or as close to 030° as possible within a ±4° tolerance. The HDG ADJ field will display the amount of correction (+/- 6.0 degrees) that was required. Press the HDG ADJ line select key to accept the setting.



Calibration Heading before adjustment
Figure 13

Now repeat the process above for all other headings in 30° increments from 60° to 360°. Press the MENU key to exit the Installation Menu. Continue with heading accuracy tests below.

NOTE: In some aircraft, prop wash and wind during ground operations can create inconsistent pressures in the pitot-static system. The pressures can affect the ADAHRS, resulting in small pitch and heading perturbations. Before reading the aircraft headings for the purposes of calibration, ensure the attitude solution has stabilized and is not influenced by external winds and pressures. The disturbances normally settle out over a period of 15 to 60 seconds. Idle power or temporarily selecting the alternate static source can sometimes eliminate the effect.



Calibration Heading after adjustment
Figure 14

(d) Heading Accuracy Test

As a final check, position the aircraft on the headings shown in the **HEADING ACCURACY TEST** table below and verify heading (viewed on HSI display) is within +/- 4 degrees using a calibrated heading source (i.e., sight compass, compass rose).

If any heading is outside ±4° then rerun the Calibration Procedure and or Heading Offset adjustment.

HEADING ACCURACY TEST

Calibrated Heading Source	Tolerance	Actual EBD Heading	Calibrated Heading Source	Tolerance	Actual EBD Heading
30	+/- 4	_____	210	+/- 4	_____
60	+/- 4	_____	240	+/- 4	_____
90	+/- 4	_____	270	+/- 4	_____
120	+/- 4	_____	300	+/- 4	_____
150	+/- 4	_____	330	+/- 4	_____
180	+/- 4	_____	360	+/- 4	_____

(e) Heading Interference Test

With aircraft engine(s) running monitor current aircraft heading on EBD and exercise flight controls stop to stop including flaps and any electric trim tabs. Verify the heading (viewed on HSI display) does not change on the EBD by more than 2°. If movement of flight controls causes more than a 2° heading change then it may be necessary to degauss the flight controls including the cables. A handheld degausser can be found at most audio and video stores.

- Operate all electrical and environmental equipment including:
- Blowers, fans, heaters, air conditioner
- Deice boots, fuel pump(s), backup vacuum pumps
- Landing, logo, NAV lighting
- Operate pulse equipment – transponder, WX radar, DME
- Key all VHF communication radios.
- Operate autopilot so that all servos run (roll, pitch, yaw, trim)

If the operation of any electrical system causes the heading to change by more than 2° the RSM wiring may need to be relocated away from the offending system. The offending system may also have a bonding issue to the airframe that needs to be corrected.

Run engine from idle to take off power and verify that the heading does not change by more than 2°. Prop wash at higher engine RPMs may cause a heading shift, try an alternate static source if this is a issue.

This completes all RSM calibration and heading tests.

(6) Ground Test Procedure

CAUTION: DO NOT EXCEED THE AIRCRAFT'S MAXIMUM AIRSPEED, ALTITUDE, OR VERTICAL SPEED AT ANYTIME DURING THE TESTING. DAMAGE COULD RESULT TO THE PRE-EXISTING AIRCRAFT INSTRUMENTS.

NOTE: When changing indicated airspeed or altitude on the ground using pitot/static test equipment, changes in the AHRS display of pitch, roll and heading will result, possibly accompanied by a CROSS CHECK ATTITUDE annunciation. This behavior is the result of the Kalman Filter algorithms employed in the EBD attitude solution. These changes in pitch, roll or heading are normal and do NOT indicate a system failure. The integrated nature of the EBD AHRS algorithms is such that AHRS performance can only be properly evaluated during flight or ground maneuvers.

(a) Check the following items for proper operation:

1) Airspeed Tape, Altitude Tape
2) AHRS Sensor
3) EMI Test
4) Flight Control Interference Check

(b) Indicated Airspeed Display

Use the Indicated Airspeeds (IAS) Configuration table under IAS Configuration to record the aircraft speed settings from the Aircraft Flight Manual in the IAS Setting column. Set the Pitot/Static test set for 5,000 ft above field elevation. Increase airspeed to Vne and check all Speed Bands and Speed Markers listed in table.

(c) Altitude Display

With the Pitot/Static tester still set for 5000 ft above field elevation and with BARO Set to 29.92 in. Hg. on the EBD, verify altitude tape displays altitude within ±40 ft of the calibrated test set altitude.

(d) System Leak Test

Perform a pitot-static system leak test per 34-10-00 or set the Pitot Static Test Set to 1,000 ft above field elevation and without additional pumping for a period of 1 minute the aircraft static system should not lose more than 100 ft of altitude.

(e) AHRS Sensor Test

Verify that correct aircraft attitude information is presented on the Attitude Indicator portion of the EBD (i.e., pitch, roll, and slip/skid indicators are all present and reflect current airplane attitude).

NOTE: The attitude flag must be cleared prior to beginning the AHRS sensor test. The attitude flag may take up to 5 minutes to clear when the ambient temperature is below -20° C. Typically the attitude solution will be available in less than 3 minutes. Flag will clear automatically when AHRS alignment is achieved.

(f) EBD Internal Battery Test

Test the internal back-up battery in the EBD every 12 months. Replace the battery if it fails the following operational test.

1) Setup

This test must be run at room temperature approximately 25º C.

2) Procedure

a) Turn on the EBD

b) Press MENU Key

c) Select POWER SETTINGS page from the Main Menu

d) Press the BATTERY line select key

e) BAT LEVEL IN --.-- will be displayed for a short period of time as battery capacity is being measured. This could take up to 10 minutes if the ambient temperature is below 0º C.



f) Once the capacity is measured ON BAT XX% REM will be displayed.



g) The “ON BAT” indication must read a minimum of 80% to continue. If the battery capacity is below 80% then the battery should be charged by returning the EBD to external power. The battery will charge as long as the EBD is turned on and external power is supplied.

h) With the battery displaying greater than 80% charge set a timer for 30 minutes. After the 30 minute time has elapsed the EBD must still be operating on battery. If the internal battery will not supply the minimum 30 minutes operating time or fails to charge above 80%, replace the battery and return the failed battery to Aspen Avionics.

i) Instructions for battery replacement are under Components, EBD Display, above.

j) Switch the EBD back to external power and recharge the internal battery to 80% or greater prior to release to service.

(g) EMI Test

Monitor the EBD for Flags, Red-X's, Red Slashes, heading changes, altitude changes, airspeed changes, attitude changes or any error messages while performing the following Test:

1) Transmit on all Comm radios for 20 seconds each at 118.000MHz, 126.900MHz, and 136.950MHz.
2) Turn on all transponders, DMEs, Wx Radar, and all other pulse type equipment for 20 seconds each.
3) Operate all aircraft lighting including position lights, strobe lights, navigation lights, and all other forms of lighting for 20 seconds each.
4) Operate all environmental equipment including fans, air conditioning, heaters, and all other forms of environmental control equipment for 20 seconds each.
5) Operate Fuel pump(s), deice boots, windshield heat, prop heat, flaps.
6) Operate autopilot, servos, yaw damper, and all electric trim servos.
7) Operate engine(s) and verify no interference.

(h) Flight Control Interference Check

CAUTION: VERIFY CONTROL SURFACES ARE FREE AND CLEAR.

With the EBD mounted in the instrument panel push the control column (yoke or stick) all the way forward (nose down) and verify there is sufficient clearance between the EBD, and their knobs, and the control column. With the control column fully forward move it from lock to lock (full right to full left) and verify there is sufficient clearance between all EBD, and their knobs, and the control column.

(7) Post Installation Flight Check

CAUTION: ONLY PERFORM FLIGHT TEST IN DAY VFR CONDITIONS WITH AN APPROPRIATELY RATED PILOT FOR THE AIRCRAFT TYPE TO BE FLOWN.

(a) Basic ADI Flight Checks

Fly the aircraft in straight and level flight and verify that the ADI roll indication is level with reference to the horizon. Observe the Slip Indicator is centered under the Roll pointer and adjust rudder trim if available to center.

1) Make a coordinated 30 degree banked turn to the right and verify that the ADI roll indication is correct with reference to the horizon.
2) Make a coordinated 30 degree banked turn to the left and verify the ADI roll indication is correct with reference to the horizon.
3) Pitch the aircraft up 10 degrees and verify the ADI pitch indication is correct with reference to the horizon.
4) Pitch the aircraft down 10 degrees and verify the ADI pitch indication is correct with reference to the horizon.

(b) Basic HSI/DG Flight Checks

1) Make a 180 degree coordinated turn to the right and verify that the compass scale and numerical heading indication correctly track the aircraft heading during the turn.
2) Make a 180 degree coordinated turn to the left and verify that the compass scale and numerical heading indication correctly track the aircraft heading during the turn.
3) Then turn from West to North (30° Angle of Bank) and, using an outside reference, roll out to a northerly heading. (In the Southern Hemisphere, also turn West to South)
4) Immediately after the aircraft rolls out, record the heading indication.
5) Maintain the same heading by outside visual reference. There may be some movement of the heading indicator as the heading system stabilizes.
6) When the heading stabilizes, record the heading again. Then perform the test from East to North (In the southern Hemisphere, also turn east to South).
7) If the difference between the heading on rollout and the heading after stabilizing is more than 7°, refer to the Troubleshooting Chart above.

(c) Document Test Flight

This completes the flight test. If everything was satisfactory then document the completion of the Test Flight in aircraft log book in accordance with FAR 91.407(b).

GARMIN G5 EFI STANDBY DISPLAY

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

1. Garmin G5 EFI Standby Display

   A. Description

   The Garmin G5 EFI configured as an attitude indicator is shown in "Figure 1". The G5 is an electronic flight instrument display operating as a standalone flight display. It features a bright, sunlight readable, 3.5-inch color display which is sized to fit in a standard 3-1/8-inch instrument cutout. The G5 contains integrated attitude/air data sensors that provides display of attitude and secondary display of air data information.

   The G5 calculates aircraft attitude using information from its built-in inertial sensors. If the G5 senses that the attitude solution is valid, but not yet within the internal accuracy limits, “ALIGNING” is displayed. The G5 can align itself both while taxiing and during level flight. The G5 will also use GPS and airspeed data to provide the most accurate attitude information. If none of these additional sources of information are available, attitude calculations will still be valid but accuracy may be slightly affected.

   The G5 has a Turn Rate Indicator located at the bottom of the display. Tick marks to the left and right of the displayed heading denote standard turn rates (3 deg/sec). A magenta Turn Rate Trend Vector shows the current turn rate. A standard-rate turn is shown on the indicator by the trend vector stopping at the standard turn rate tick mark.

   The G5 displays slip/skid information as indicated by the location of the ball at the lower center portion of the indicator.



Garmin G5 EFI Standby Display
Figure 1

(1) The G5 receives a GPS input for attitude aiding via the glareshield mounted GPS antenna only.

(2) The following functions are provided by the GPS input:

  (a) Ground Track

  (b) Ground Speed (GS)

(3) The G5 installation is connected to the aircraft Pitot / Static system which provides the following functions:

  (a) Secondary Airspeed indicator

  (b) Secondary Barometric altimeter

  (c) Secondary Vertical speed

  (d) Secondary Altimeter Barometric Setting

  (e) Secondary Altimeter

  (f) Secondary Altitude Display

  (g) Selected altitude setting, bug and visual altitude alerting

  (h) Secondary Vertical Speed Indicator

  (i) Vspeed references

(4) The G5 unit will turn on automatically when power is applied to the essential bus. The unit will turn off automatically 45 seconds after power is removed. Alternatively, holding the button on the left lower corner of the unit for three seconds will turn the unit off.

B. Troubleshooting

Garmin G5 EFI is a self contained unit that performs a self test during power up. If the unit fails to power up, check connections, if all connections are correct and power is going to the unit, replace unit.

C. Inspections

(1) Annually

Each twelve (12) calendar months conduct a battery capacity check of the Garmin G5 EFI battery per "(e) Battery Capacity Check" on page 3424015.

(2) Each Twenty-Four (24) Calendar Months

Each twenty-four (24) calendar months conduct an Altimeter System Test of the Garmin G5 EFI Unit in accordance with 14 CFR Part 43 Appendix E.

D. Basic Control & Operation

The Garmin G5 EFI unit will turn on automatically when power is applied to the essential bus. The unit will turn off automatically 45 seconds after power is removed. Alternatively, holding the button on the left lower corner of the unit for three seconds will turn the unit off.

Pressing the power button brings up the backlight controls as well as the current battery status. The backlight level adjusts automatically via a photocell. Temporary adjustments to the backlight level can be made by turning the knob on the lower right corner of the unit to adjust. When power is cycled to the unit, the backlight level is reset to use the photocell.

E. Components

(1) Garmin G5 EFI Unit

See "Figure 3" on page 3424016.

(a) Removal

1) Insert a hex drive tool through the access hole in the front cover of the G5 and loosen the unit from the mounting ring.
2) Disconnect the pitot and static lines from the G5 unit.
3) Disconnect the electrical connector and GPS coaxial cable connector from the G5 unit.
4) Remove the G5 unit.

(b) Installation

1) Inspect the electrical and coaxial connectors and pins for signs of damage. Repair any damage.
2) Reconnect the electrical connector and GPS coaxial connector to the G5 unit.
3) Reconnect the pitot and static lines to the G5 unit. The ports are labeled on the unit using the abbreviations “P” and “S” respectively.
4) Insert the alignment pin located at the top of the unit into the mating hole in the mounting ring, pushing the unit flush with the instrument panel.
5) Insert a hex drive tool through the access hole in the front cover of the G5 and tighten the unit to the instrument panel using 10-12 in·lbs torque.
6) See "Post Installation Setup and Checkout Procedures" on page 34240110.

(c) Return to Service

If the G5 unit is replaced, perform an altimeter system test of the G5 in accordance with 14 CFR Part 43 Appendix E.

See "Return to Service - System" on page 34240119.

(2) Battery

See "Figure 2" on page 3424015.

The G5 has an external lithium-ion battery that provides up to 4 hours of emergency power. The G5 charges the battery whenever the aircraft is powered. The G5 is connected to the main bus. When the main bus is not active, and the emergency bus is active it also charges the standby battery. The only time the G5 uses the internal battery is when all electrical power is lost to the aircraft.

NOTE: An new battery pack was introduced into production in S/N's 2881497 and up. In S/N's 2881106, 2881285 thru 2881293, 2881304 thru 2881310, 2881313 thru 2881318, 2881336 thru 2881471, when replacing the original factory installed battery pack (P/N 683-805) with the new battery pack (P/N 601-939), the G5 Software 006-B2304-23, Version 6.40, must be upgraded to G5 Software 006-B2304-34, Version 6.82. See "Garmin G5 EFI Software Loader Card Creation Procedure" on page 34240120.

Removal

1) Remove the G5 unit per Garmin G5 EFI Unit "Removal" on page 3424014.
2) Use a screwdriver to loosen the battery jackscrews on the back of the unit.
3) Remove the battery from the G5 unit.

(d) Installation

1) Position the battery on the back of the G5 unit.
2) Use a screwdriver to tighten the battery jackscrews to back of the G5 unit.
3) Install the G5 unit per Garmin G5 EFI Unit "Installation" on page 3424014.

(e) Battery Capacity Check

Each twelve (12) calendar months:

1) Without power applied to the aircraft, turn on the G5 by pressing the power button in the lower left corner of the unit.
2) Note the remaining battery capacity (%) at the top left corner of the display.
3) After about a minute, the remaining capacity will change from (%) to time (hour:min).
4) If the remaining capacity is less than one hour (1:00), allow the battery to charge until the capacity shows greater than 95% and repeat the check.

   NOTE: Apply power to the airplane to charge the G5 external battery.

5) If the remaining capacity is less than one hour (1:00) after charging, the battery must be replaced.



G5 Battery
Figure 2

(3) Mounting Ring

(a) Removal

1) Remove the G5 unit per Garmin G5 EFI Unit "Removal" on page 3424014.

2) Remove the three #6-32 screws that secure the mounting ring to the instrument panel.

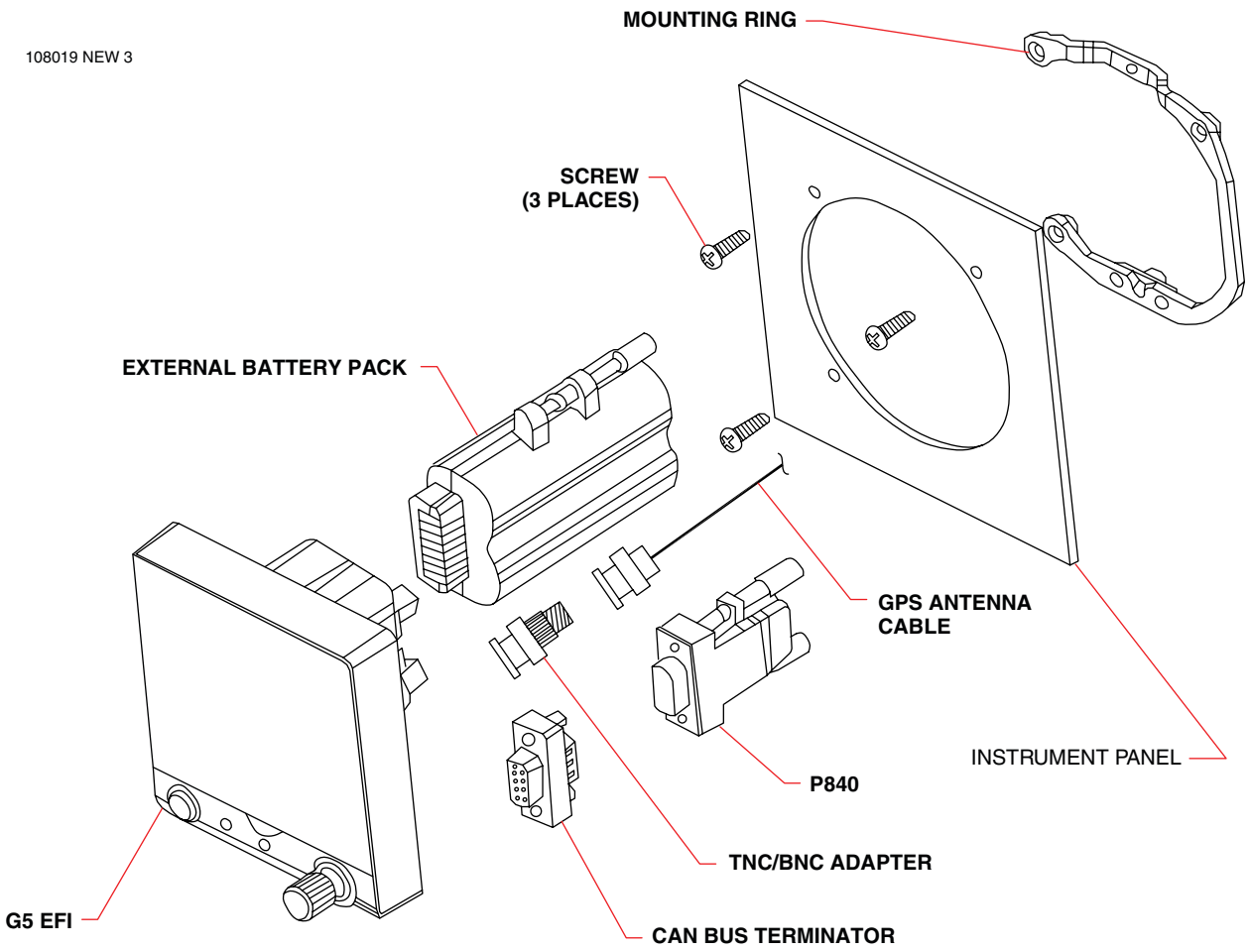
3) Remove the G5 mounting ring.

(b) Installation

1) Position the new mounting ring, and secure to the instrument panel using the three #6-32 screws.

2) Evenly torque the screws to 10-12 in-lbs.

3) Install the G5 unit per Garmin G5 EFI Unit "Installation" on page 3424014.

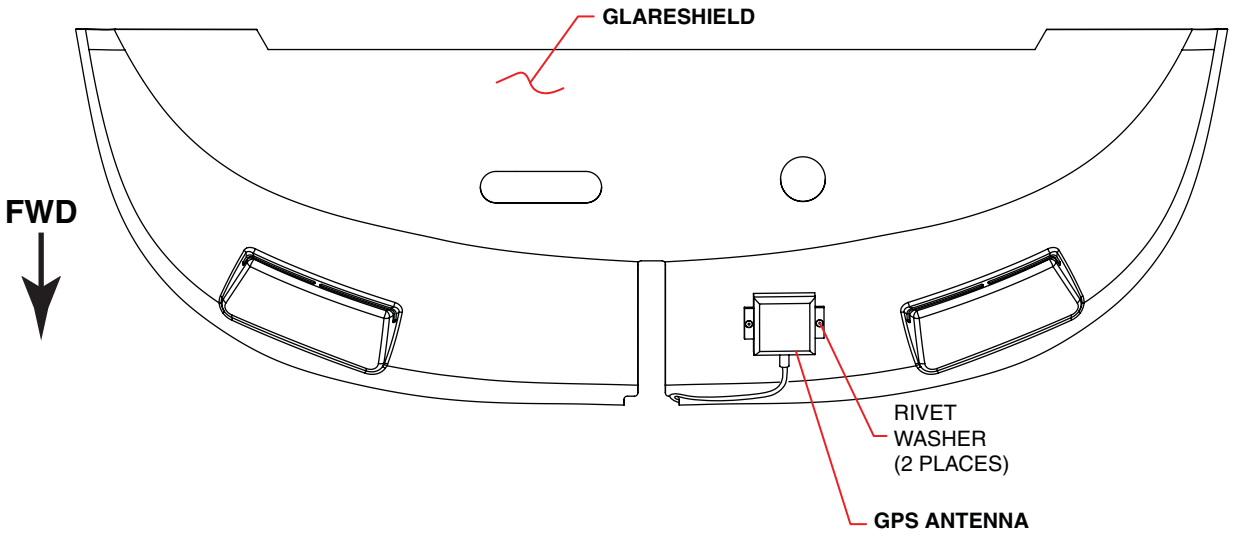


Garmin G5 EFI Installation
Figure 3

(4) GPS Antenna

The glareshield-mounted Garmin GPS antenna feeds a GPS receiver inside the G5 which provides position, velocity, and time data to support various functions. This antenna requires a TNC/BNC adaptor, for G5 Interface to GPS antenna.

108062 NEW 2



LOOKING DOWN



Garmin G5 Glareshield Mounted GPS Antenna
Figure 4

(5) GMU 11

The Garmin GMU 11 Magnetometer is a remote mounted device that interfaces with the Garmin G5 EFI to provide heading data. It is installed in the right wing, mounted to a bracket (cover) assembly attached to the bottom of the wing at approximately W.S. 137.00. The GMU 11 Magnetometer uses magnetic field measurements to create electronically stabilized heading data.

CAUTION: WHEN REMOVING AND/OR INSTALLING THE GMU 11, ALWAYS ENSURE THAT AIRCRAFT POWER IS OFF. UNPLUG ANY AUXILIARY POWER SUPPLIES.

(a) Troubleshooting

See "5) Magnetometer Interference Check" on page 34240118.

(b) Removal

1) Remove the eight screws securing the bracket (cover) assembly to the bottom of the wing. Remove the bracket (cover) assembly and attached GMU 11 being careful not to stress the wiring harness.
2) Disconnect the P842 connector.
3) Unscrew the four screws, washers, and nuts that hold the GMU 11 to the mounting bracket.

NOTE: Make sure the hardware removed from the GMU 11 mounting is retained for reinstallation. If hardware needs to be replaced, ensure the proper non-ferrous hardware is used per the parts catalog.

NOTE: Be sure to use non-magnetic tools to avoid harming the GMU.

(c) Installation

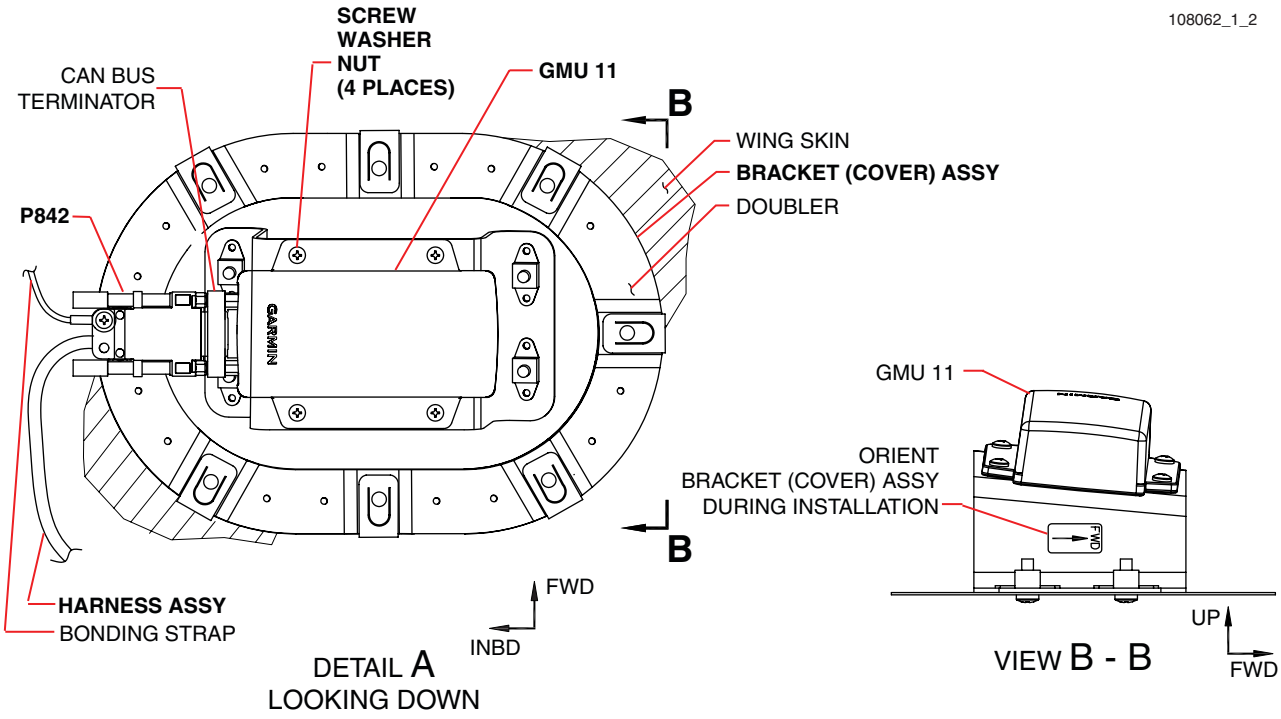
1) Place the GMU 11 in its mounting bracket and align its holes with the mounting holes.
2) Insert the four mounting screws and secure the GMU 11 in place with washers and nuts. Tighten screws until snug, plus 1/4 turn.
3) Inspect the P842 connector and pins for signs of damage. Repair any damage.
4) Connect the P842 connector to the GMU 11.
5) Taking note of the FWD arrow on the GMU 11 mounting bracket, place the GMU 11 access cover into the bottom of the wing and secure with eight screws.

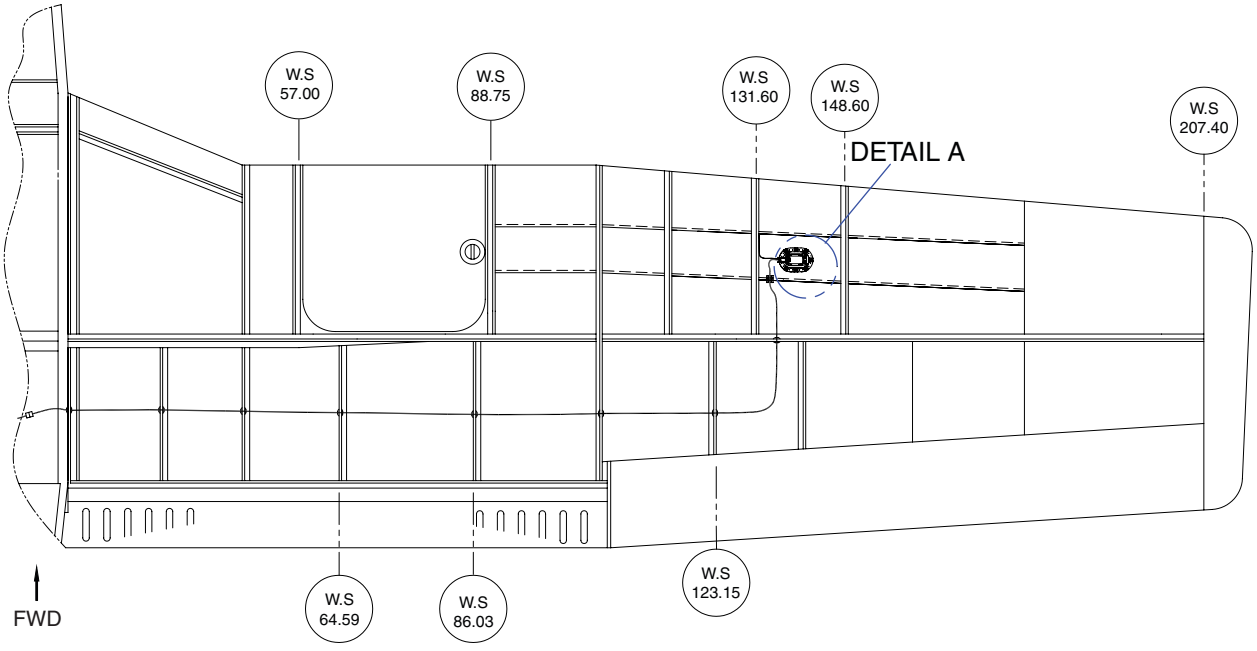
(d) Return to Service

Each time the GMU 11 is removed and reinstalled, or if it is replaced, conduct the "GMU 11 Magnetometer Calibration" on page 34240115.



GMU 11
Figure 5





G5 STANDBY
(RIGHT WING ONLY)
LOOKING DOWN

GMU 11 Mounting
Figure 6

F. Post Installation Setup and Checkout Procedures

(PIR-108066, Rev. A)

(1) Required Test Equipment

The following Test Equipment will be required to complete the remaining steps in the ground test procedure:

(e) Ground power equipment
(f) Pitot-Static Test equipment (Refer to "Pitot and Static System" on page 34101)
(g) Electrical Bonding Tester
(h) Digital Multi-Meter
(i) Aircraft jacks
(j) Digital Level (6 inch size)
(k) Digital Level (24-36 inch size)

(2) Post Installation Wiring Checks

NOTE: Use the seat rails as aircraft ground reference.

(a) Ground Checks
1) Ensure the connector is disconnected from the G5.
2) Using a multi-meter, verify continuity to aircraft ground plane for each ground pin defined in "Chart 1".

(b) Power Checks
1) Primary Power
a) Ensure the connector is disconnected from the G5.
b) Ensure applicable circuit breakers are closed.
c) Apply external power to aircraft.
d) Switch Battery Master to ON.
e) Using a voltmeter, verify 24V (+/- 4) power for each power pin defined in "Chart 1".
2) Emergency Power
a) Ensure the connector is disconnected from the G5.
b) Ensure Battery Master is OFF.
c) Ensure external power is removed from aircraft.
d) Ensure applicable circuit breakers are closed.
e) Switch Emergency Battery to ARM.
f) Using a voltmeter, verify 24V (+/- 4) power for each power pin defined in "Chart 1".

CHART 1
POST INSTALLATION WIRING GROUND CHECKS

Connector Pin	Unit	Note
P840-7	Garmin G5	Aircraft Power
P840-9	Garmin G5	Aircraft Ground

(3) Initial System Turn On

Close all applicable circuit breakers and apply external power to aircraft. On the G5, verify attitude is displayed and there are no invalid parameters.

Configure the system per "(4) System Configuration", prior to system checkout.

NOTE: On power-up, “Magnetometer not calibrated.” may be displayed on the G5 if a GMU 11 is installed. This will cause the compass card on the HSI/DG to display GPS track if installed as an HSI and nothing if installed as a DG. Magnetometer calibration is completed later in "GMU 11 Magnetometer Calibration".

(4) System Configuration

(a) Software Loading Procedure

NOTE: If the G5 unit reports the same software P/N and version as indicated in"Chart 5" on page 34240120 completing this section is not required.

If required, create a software loader card per "Garmin G5 EFI Software Loader Card Creation Procedure" on page 34240120.

1) Power on the G5, then insert a properly formatted microSD™ card into the microSD™ card slot.
2) A software update pop-up will appear on the screen. Select OK to begin the update.
3) The unit will reboot to the software update screen and the software update will begin automatically. This screen will show the progress of the software update.
4) Ensure power is not removed while the update is being performed.
5) The unit will reboot after the update is complete. The new software version will be displayed while the unit powers on or in the configuration mode menu.

(b) Device Configuration

1) Power off the G5 by holding the power button until it turns off. Remove the microSD™ card from the microSD™ card slot.
2) Hold down the knob while powering on the G5 by pressing the power button.
3) The unit will reboot to configuration mode.
4) Use the knob by rotating and pressing to set the configurations per "Chart 2" on page 34240112.

CHART 2 (Sheet 1 of 2)
IAS SETTINGS

Config Page	Config Option	Configuration Setting
Device Information	Installation Type	Standalone Instrument
	Diagnostic/Data Log	Enabled
Attitude	Indicator Type	Normal
	Pitch Display	Normal
	Roll Display	Ground Pointer
	Standard Turn Roll Reference	Show
	User Pitch Offset	Disabled
	Automatic Declutter	Enabled
Air Data	Air Data Sensors	Enabled
	Outside Air Temp Sensor	Disabled
	Vertical Speed Indicator	± 2000 FPM
Airspeed NOTE: Configure Units prior to setting airspeeds.	VNE	154
	VNO	125
	VSO	45
	VS1	50
	VFE	102
	VA	---
	VX	---
	VY	---
	VG	---
	VR	---
	VMC	---
	VYSE	---
Magnetometer	Magnetometer	Enabled
	Orientation	Connector Port
Flight Controls	Garmin Autopilot Servos	None
	Analog Autopilot Interface	None

CHART 2 (Sheet 2 of 2)
IAS SETTINGS

Config Page	Config Option	Configuration Setting
Backlight	Curent Mode	Automatic
	Default Mode	Automatic
	Minimum Photocell Input	10
	Minimum Display Brightness	20
	Maximum Photocell Input	70
	Maximum Display Brightness	100
	Filter Time Constant	6
Display	HSI Page	Disabled
	Power Up Page	PFD
GPS	Internal GPS Receiver	Enabled
	GPS Data Fields	Show
Navigation	Navigation Data	Disabled+
Units	Altitude	Feet
	Distance	Nautical
	Airspeed	Nautical
	Ground Speed	Nautical
	Ground Track	Magnetic
	Vertical Speed	Feet/Minute
	Pressure	Inches
	Temperature	Fahrenheit or Celsius (either can be selected as there is no OAT probe)
	Data Field	Normal
RS-232	Input Format	None
	Output Format	None
ARINC 429	Output 1	None
	Output 2	None
	Input 1	None
	Input 2	None
	Input 3	None
	Input 4	None

(c) Pitch/Roll Calibration

1) Level the aircraft in pitch and roll (0 ± 0.25°).
2) From the Attitude page (in config mode), select “Calibrate Pitch/Roll…”
3) Select Start, then wait 30 seconds for calibration to complete.
4) Press the knob to proceed after calibration completes.G5 Software

(d) Engine Run Procedures

1) Vibration test

a) Start the aircraft engine in accordance with the aircraft POH.
b) Start the G5 in config mode by holding down the knob while powering it on.
c) From the Attitude page, select “Vibration Test”
d) Ensure the aircraft complies with all on-screen instructions, then select Start.
e) Gradually increase engine power from idle to full power and back to idle in 1-2 minutes. A screen will be displayed showing the test in progress.
f) After the gradual increase and decrease in aircraft power verify for passing or failing results.

1] If the green check mark remains during the engine power sequence, this indicates Passing results, select Done.
2] Failing results will be indicated by a red “X” as shown in "Figure 7". Select Done when the test has been completed and follow the instructions in Step g).

g) If failures are indicated, repeat the test. If failures persist, the installation is unreliable until the source of the vibration problem is identified and remedied. Record the out of range limit values for future reference.

NOTE: In some aircraft, attempting the engine run-up test on a day with very strong and/or gusty winds may cause the test to occasionally fail. However, a failure during windy conditions cannot not be taken as evidence that the test would pass in calm conditions; an actual pass is required before the installation can be considered adequate.



Engine Vibration Test Engine Failure
Figure 7

2) GMU 11 Magnetometer Calibration

a) At the compass rose, align the aircraft to a heading of magnetic north (+/- 10°).

b) Restart the G5 in configuration mode.

c) Go to the “Magnetometer” page Group on the G5 unit.

d) Select the “Calibrate Magnetometer…” page.

e) The G5 will display instructions. Follow the on-screen instructions and select “Next” to cycle through each page of instructions.

f) On the final page, select “Start” to begin the calibration.

g) The G5 display advises the operator when to turn the aircraft, when to stop, and when to turn again.

h) Upon instruction to turn, taxi the aircraft in a right turn. After approximately 25° to 30° of turn from the last heading, the G5 display advises the operator when to stop the aircraft.

NOTE: If smooth accurate turns are not made, the G5 may incorrectly interpret a station and instruct to “Hold current position…” prior to or after completion of the heading change. If this scenario is encountered, it is best for the operator to temporarily ignore the “Hold current position…” command and instead use compass rose references to complete the heading change. Dwelling at these 30° increments for the time recommended by the G5 should result in successful calibration.

i) The G5 instructs the operator to dwell at multiple headings around a complete circle.

j) Repeat the turn-and-stop process until the G5 advises that a successful calibration is complete and “Calibration: Success” is displayed on the G5.

k) Select “Done” on the G5 to conclude this procedure

l) Exit configuration mode.

3) Compass Swing

a) With the G5 in normal mode, all of the aircraft and avionics systems powered and operating normally, position the aircraft on a known compass rose at a heading 360° (North).

b) Record the heading displayed on the G5 in "Chart 3" on page 34240116.

c) Repeat step 2 for each of the headings listed in "Chart 3" on page 34240116.

d) Calculate the heading errors by subtracting the displayed (B) value from the actual (A) value for each of the headings in "Chart 3" on page 34240116.

e) If all calculated heading errors for the G5 are +/-10° from the surveyed heading, the installation is acceptable and no further work is required to correct the GMU 11 installation. Proceed to "G5 Post Installation Checks".

f) If a Heading Error (A-B) is greater than +/- 10°, calculate the average error by adding all errors and dividing by 12. This is the angle by which the GMU 11 must be physically rotated to correct the installation.

g) Modify the installation to rotate the GMU 11 by the amount calculated in the previous step. When looking down at the GMU 11, rotate clockwise for positive values, and counterclockwise for negative values.

h) After physically correcting the GMU 11 installation, repeat the procedures in "GMU 11 Magnetometer Calibration" and this section.

CHART 3
HEADING VERIFICATION

Heading A	Displayed G5 Heading (B)	Heading Error (A-B)
360°		
30°		
60°		
90°		
120°		
150°		
180°		
210°		
240°		
270°		
300°		
330°		

(e) G5 Post Installation Checks

1) Initial Power On Checks

a) Apply power to the aircraft and aircraft systems and allow the aircraft systems to initialize.

b) On the G5 Attitude display, verify the following;

- No red-X's displayed
- Attitude display
- Heading display
- Airspeed Tape (Digital read out will be dashes)
- Altitude Tape (Digital readout should be reasonable to field elevation)
- Slip skid display
- Current track bug
- Barometric Pressure setting field

2) Indicated Airspeed Display

WARNING: THIS TEST MUST BE ACCOMPLISHED BY AN APPROPRIATELY RATED MECHANIC OR REPAIRMAN (14 CFR 43.4).

Refer to "Chart 2" on page 34240112 for IAS Setting values. Set the Pitot/Static Test Set for 5000ft above field elevation. Increase airspeed to Vne and check all Speed Bands and Speed Markers listed in "Chart 4".

CHART 4
SPEED BANDS AND SPEED MARKERS

IAS Setting	Band Color	Band Range	Description
Vne	Red	>Vne	Red arc displayed at all speeds above Vne
Vno	Yellow	Vno – Vne	Yellow arc extending from Vno to Vne
Vs1	Green	Vs – Vno	Green arc extending from Vs1 to Vno
Vfe	White		Top of White Arc
Vs0	White		Bottom of White Arc
Vyse	Blue marker		
Vmc	Red marker		

3) Altitude Display

Perform an altimeter system test of the G5 in accordance with 14 CFR Part 43 Appendix E up to an altitude of 20,000 feet.

Air Data Static Pressure Calibration

NOTE: Procedure is only required if Section 6.5.3 results for the G5 fall outside of tolerances of the Part 43 Appendix E tests.

NOTE: The static pressure calibration requires the use of a pressure control system (test set) with an altitude accuracy of at least ±5 ft. at sea level and ±20 ft. at 30,000 ft. It is necessary to recalibrate to sea level (0 ft.), 10,000 ft., 20,000 ft., and optionally to 30,000 ft. The operator is allowed to finish the calibration at the end of the 20,000 ft. calibration if the aircraft operational ceiling is below 20,000 ft.

a) Start the G5 instrument in config mode.
b) Select the Air Data configuration page.
c) Select Calibrate Static Pressure.
d) Ensure all on-screen instructions have been complied with, then press Start.
e) At each calibration point the display will present a screen indicating the pressure altitude to set. Once the altitude is set, select Ready to calibrate this pressure.
f) During the calibration at each pressure, the pressure must be held constant for 30 seconds for the calibration to be successful. The calibration may be cancelled at any point should the test startup require adjusting before repeating. A progress screen will be displayed showing the status of the test.
g) Select Done when the static pressure calibration is successfully completed.
h) Exit configuration mode.

4) AHRS Sensor Test

Verify that correct aircraft attitude information is present on the Attitude Indicator portion of the EBD (i.e. pitch, roll & slip/skid indicators are all present and reflect current aircraft attitude).

5) Magnetometer Interference Check

With the G5 in configuration mode, initiate the magnetometer interference test procedure by performing the following steps:

a) Select the “Magnetometer” page group on the G5

b) Select the “Interference Test…” page.

c) Follow the instructions on each page displayed on the G5. Select “Next” to cycle through the pages and “Back” to go to the previous page.

d) Select “Start” to begin the test procedure.

e) The operator should operate all electrical and environmental equipment including:

1] Blowers, fans, heaters, air conditioner.

2] Deice boots, fuel pump(s), backup vacuum pumps.

3] Landing, recognition, strobe, NAV lighting.

4] Operate pulse equipment – transponder, WX radar, DME.

5] Key all VHF communication radios.

6] Operate autopilot so that all servos run (roll, pitch, yaw, trim).

f) When the test sequence is completed, wait approximately five seconds and then select “Done” to complete the test.

g) The G5 informs the operator if the installation has passed or failed the magnetometer interference test. The results displayed indicate the worst case percentage of limit and the time at which it occurred. If the test passes, no further action is required.

h) If the test fails, the installation should be considered unreliable until the source of magnetic interference is identified and remedied. The magnetometer interference test must be repeated until passed. When the magnetometer interference test fails, record the magnetometer maximum deviation value and its corresponding timestamp. Compare the corresponding timestamp with the test sequence to identify which action produced the problem.

G. Return to Service - System

After all G5 configuration and testing is completed per "Post Installation Setup and Checkout Procedures" on page 34240110, complete the following steps:

(1) Position the aircraft to allow a GPS signal to be received.

(2) Apply power to the aircraft and aircraft systems and allow the aircraft systems to initialize.

(3) If new software was updated, verify the correct software part number is displayed.

(4) On the G5 Attitude display, verify the following;

- No Red-X's displayed
- Attitude display
- Heading display (if interfaced to a GMU 11) Ground Track display (if not interfaced to a GMU 11)
- Airspeed Tape (Digital read out will be dashes)
- Altitude Tape (Digital readout should be reasonable to field elevation)
- Ground speed field
- Slip skid display
- Current track bug
- Barometric Pressure setting field

(5) On the G5 HSI display, verify the following;

- No Red-X's displayed
- Heading/Ground Track display
  - ¤ Heading Display - if interfaced to a GMU 11, indicated by white numbers indicating magnetic heading.
  - ¤ Ground Track display (if not interfaced to a GMU 11, indicated by magenta numbers indicating GPS ground track.
- Ground speed field (configurations without a GPS input will not display Ground Speed)
- Current track bug (configurations without a GPS input will not display Current Track bug)
- Rotating compass rose.

2. Garmin G5 EFI Software Loader Card Creation Procedure

A. To obtain the latest software version (reference "Chart 5" Required Equipment List for latest approved version), visit www.Garmin.com and navigate to the Aviation Product / General Aviation / Indicators / G5 page.

B. Download the self-extracting update file onto your hard drive.

C. Insert a FAT32-formatted micro SD card with at least 20 MB of free space into your card reader.

D. Open the self-extracting update file and follow the instructions provided by the installer application. You will be prompted to specify the drive letter of the SD card you wish to use for the update. When the transfer is complete, safely eject the SD card.

CHART 5
REQUIRED EQUIPMENT LIST (REL)

GARMIN G5 SOFTWARE CONFIGURATION
REQUIRED EQUIPMENT LIST (REL)

LRU Identifier	Part Number	Version	Remarks
G5	006-B2304-34	6.82(1–2)	
G5	006-B2304-23	6.40(3–4)	
GMU 11	006-B2770-01	2.00	

(1) 108065, Rev. A.
(2) Factory installed in S/N's 2881472 and up.
(3) Factory installed in S/N's 2881106, 2881285–2881293, 2881304–2881310, 2881313–2881318, 2881336–2881471.
(4) In S/N's 2881106, 2881285–2881293, 2881304–2881310, 2881313–2881318, 2881336–2881471, when replacing the original factory installed battery pack (P/N 683-805) with the new battery pack (P/N 601-939), the G5 Software 006-B2304-23, Version 6.40, must be upgraded to G5 Software 006-B2304-34, Version 6.82.

INTEGRATED AVIONICS SYSTEM - GARMIN G1000

WARNING: FAILURE TO CONSULT APPLICABLE VENDOR PUBLICATION(S), WHEN SERVICING OR INSPECTING VENDOR EQUIPMENT INSTALLED IN PIPER AIRCRAFT, MAY RENDER THE AIRCRAFT UNAIRWORTHY. (SEE INTRODUCTION - SUPPLEMENTARY PUBLICATIONS.)

Integrated Avionics System - Garmin G1000

(PIR-190-00907-00, Rev B.)

Optional in S/N's 2843701, 2843703 thru 2843822, 2843824 thru 2843851, and 2843853.

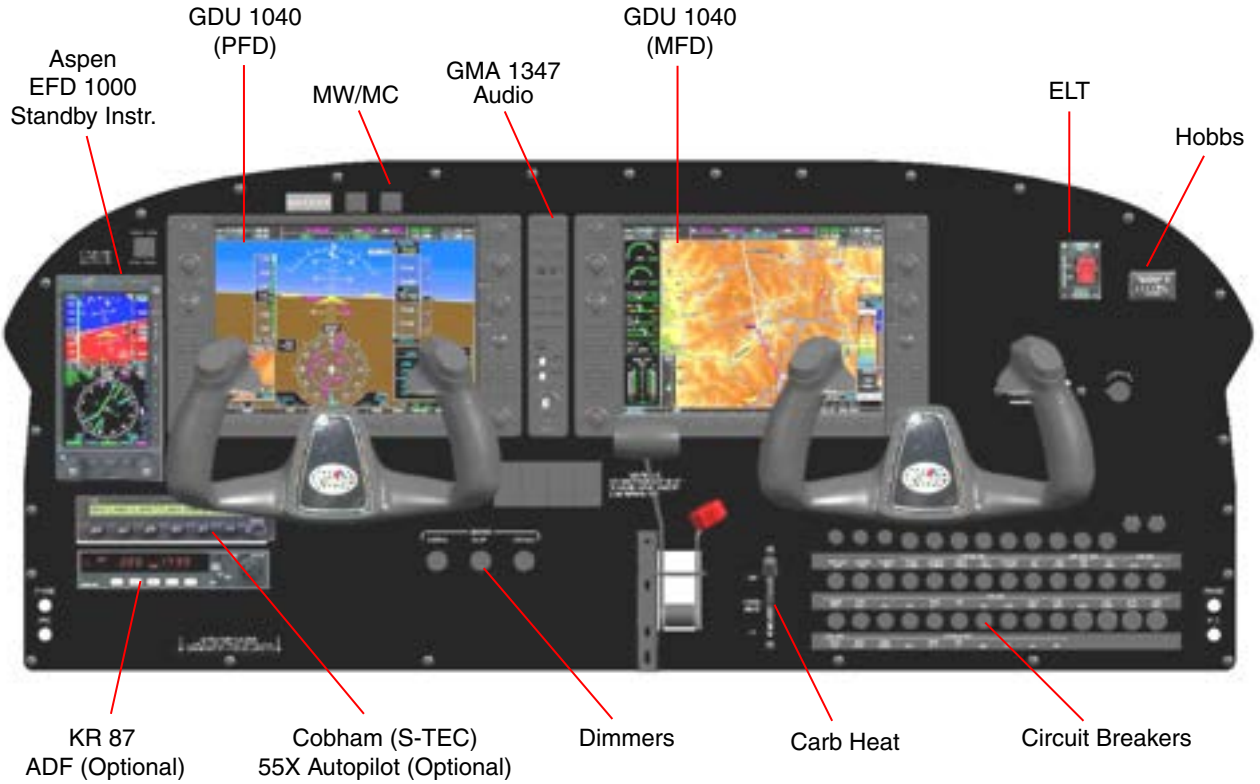
A. Description

See "Figure 1".

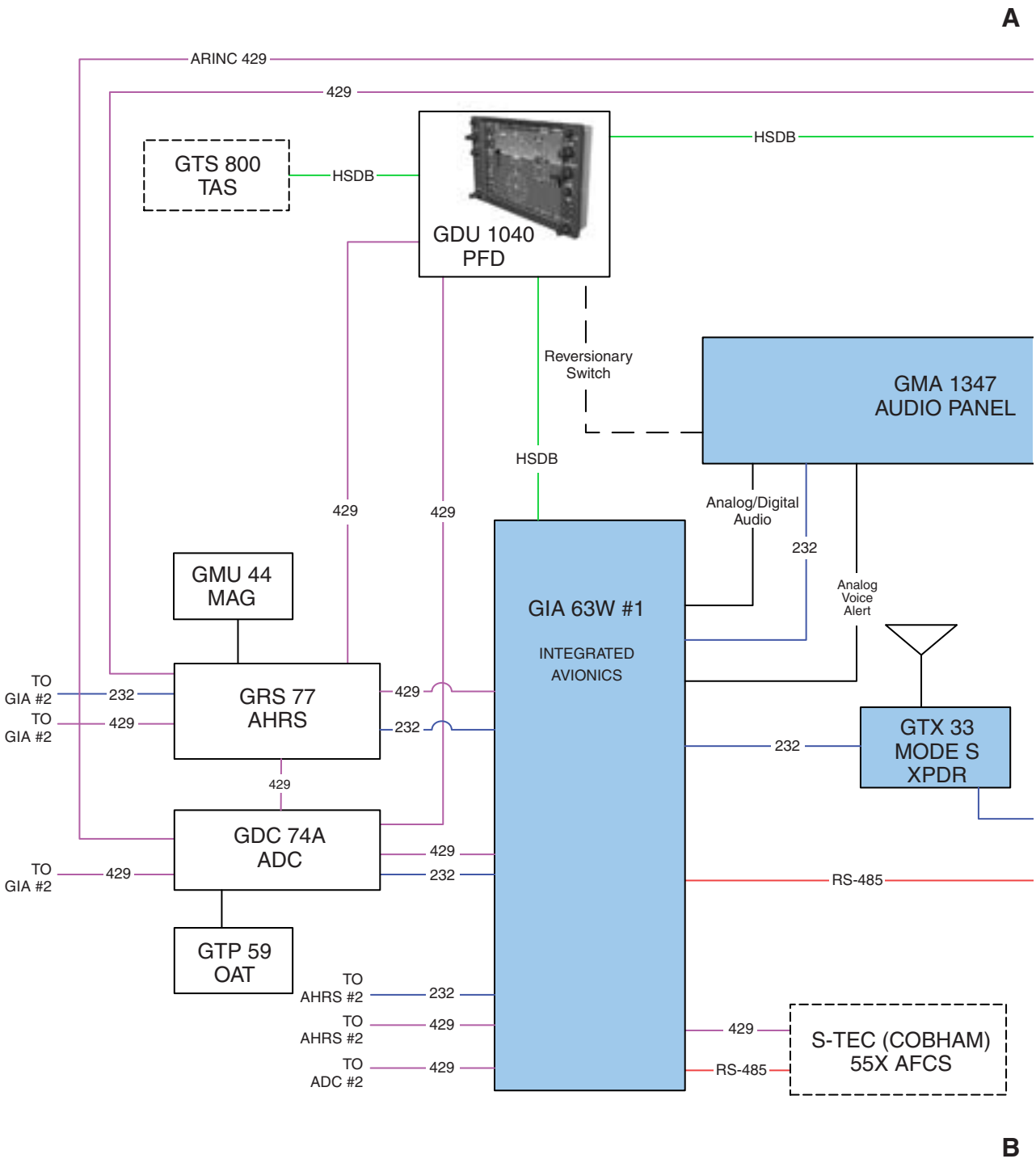
The Garmin G1000 Integrated Avionics System consists of two (2) GDU 1040 displays that serve as the primary flight display (PFD) and multi-function display (MFD). Functions provided by the system include display of attitude, heading, navigation, engine and airframe status, and a moving map display with position derived by GPS. In addition to display functions, GPS navigation, VHF Com, and transponder functions are provided by the G1000 system (via knobs and buttons on the display bezel).

A single AHRS and Air Data computer are standard. Several options that may be incorporated into the G1000 system include S-TEC 55X Autopilot, GTS 800 traffic advisory system (TAS), and TAWS-B capabilities.

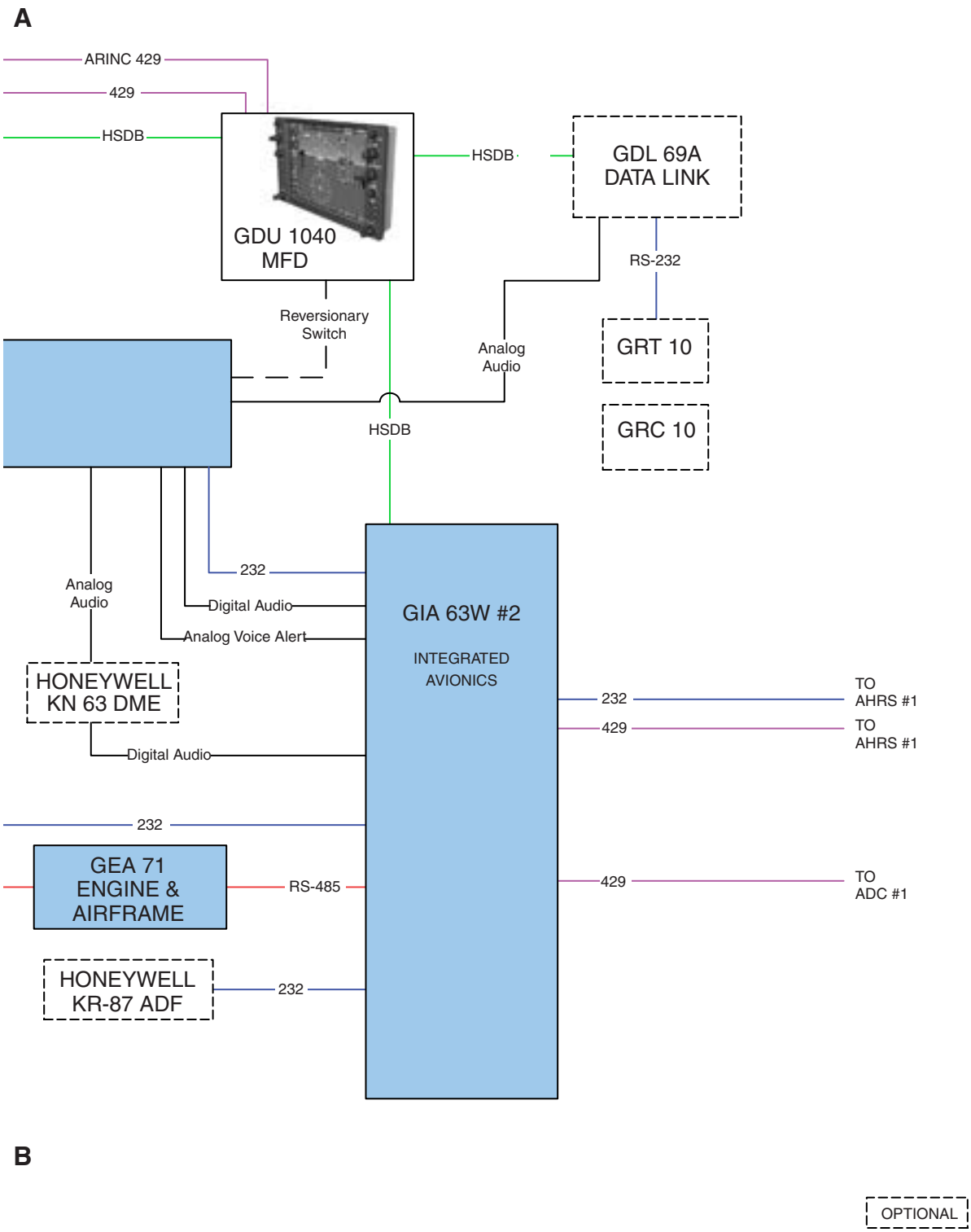
See the latest revision of the Garmin G1000 Pilots Guide for control and operation information. See Vendor Publications under Supplementary Publications in the Introduction.



Garmin G1000 Instrument Panel
Figure 1



Electrical/Avionics Component Block Diagram
Figure 2 (Sheet 1 of 4)



Electrical/Avionics Component Block Diagram
Figure 2 (Sheet 2 of 4)

(1) Flight Instrumentation

GRS 77 AHRS, GDC 74A Air Data Computer and GMU 44 Magnetometer units are responsible for providing the G1000 system with aircraft attitude, heading, altitude, airspeed, vertical speed, and outside air temperature information, all displayed on the PFD.

Primary data outputs from the GRS and GDC are sent directly to the PFD via ARINC 429. Secondary data paths connect the GRS and GDC to the MFD in the standard installation. Additional cables connect the GRS and GDC to both GIA 63W units.

The GRS 77 receives GPS data from the GIAs, airspeed data from the GDC, and magnetic heading from the GMU. Using these three external sources, combined with internal sensor data, the GRS accurately calculates aircraft attitude and heading.

(2) Engine Indicator System

The GEA 71 provides engine and airframe data to the G1000 system. The unit interfaces to transducers. Analog data is received from the transducers and is converted to a digital signal by the GEA 71. Digital information is then sent through the primary RS-485 serial path to the #1 GIA 63W. From the GIA, data is sent through the HSDB connection to the PFD, then on to the MFD for display. A backup data path exists, from the GEA to the #2 GIA 63W and on to the MFD for display, in the event the primary path fails.

(3) Communications/Navigation Systems

The GIA 63W IAUs contain VHF COM, VHF NAV, and GPS receivers. COM and NAV audio is sent via analog or digital audio to the GMA 1347 Audio Panel.

GPS information is sent to the GRS 77 AHRS and to both displays for processing.

A single GTX 33ES Mode S Transponder communicates with both GIAs. Transponder data is sent from the GIAs to the PFD where control and operation occurs.

(4) Components Details

Detailed descriptions of the individual G1000 system components/Line Replaceable Units (LRU's) are found below.

B. Power Distribution

See "Figure 3".

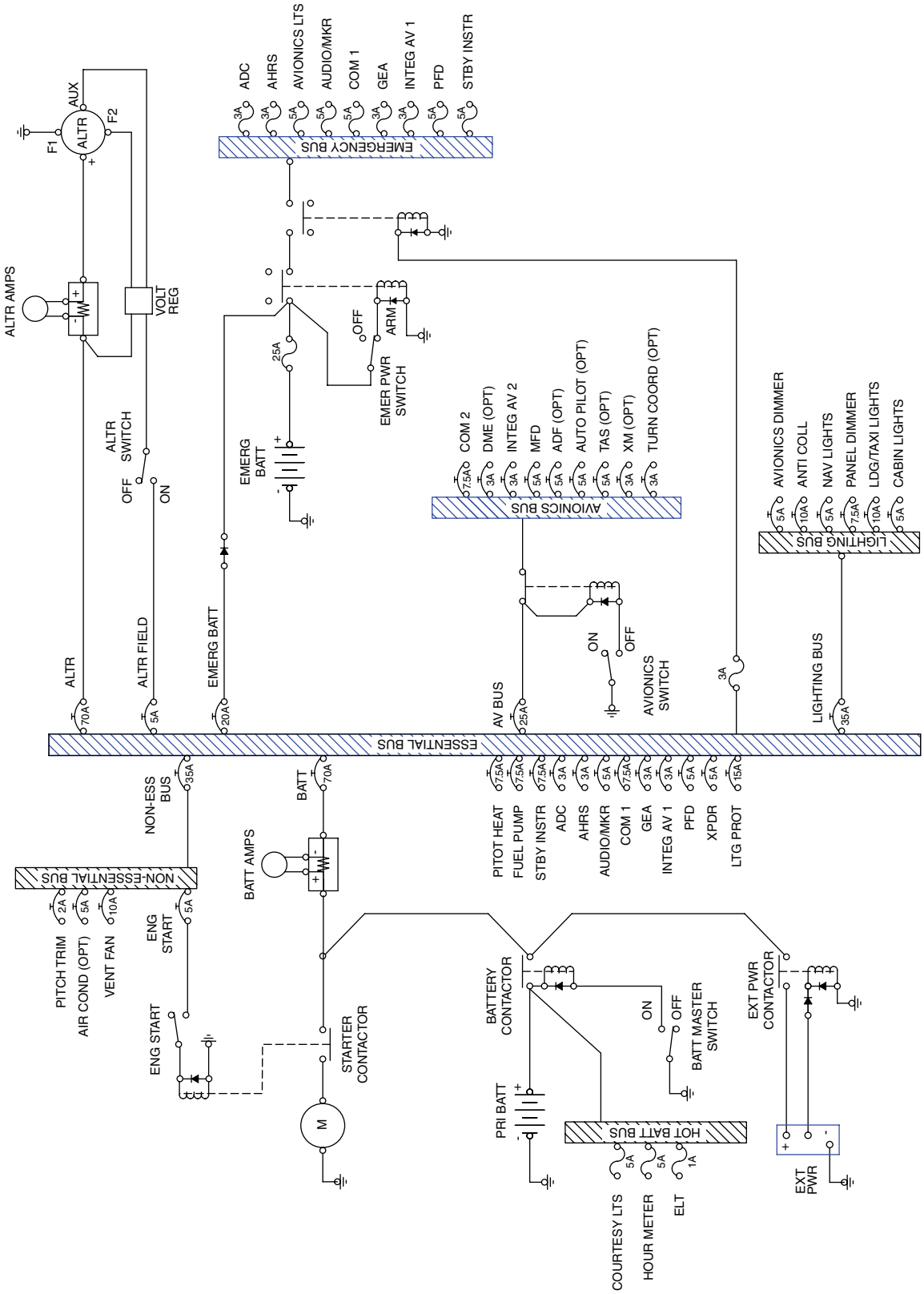
The electrical system has a tie point to connect the external power source, primary battery, starter motor, alternator and essential bus feed. Distribution of power for the G1000 system occurs on three aircraft electrical busses.

Essential Bus: Electrical current is fed to the essential bus from the alternator, isolated by one 70-amp circuit breaker. It is also 70-amp circuit breaker isolated from the battery. Essential bus loads are isolated from non-essential loads.

Emergency Bus: The emergency bus serves as a backup for the G1000 system and is isolated from all other busses. When the EMER switch is turned on, the battery master switch must be turned "OFF" to remove power from the essential bus. This closes the relay connecting the emergency bus to the emergency battery, providing emergency power to the connected equipment.

Avionics Bus: There is a single avionics bus that is isolated from its essential bus feed by a relay. It is controlled by the avionics switch in the cockpit. The avionics bus is fed through independent contactor that is protected by a 25-amp circuit breaker.

The electrical system also includes one non-essential bus and one lighting bus which are isolated by circuit breakers from the essential bus feed.



Bus Feeds and Component Power Sources
Figure 3

C. G1000 System Controls and Operation

(1) System Controls

The G1000 system controls are located on the PFD and MFD bezels. Indicators above the PFD, MFD, NAV, and COM keys are illuminated when their respective control mode(s) are selected. See the Garmin G1000 Pilots Guide for detailed controls descriptions. The unit is in MFD control mode by default on system power-up. NAV/COM radio tuning can be accomplished in either PFD or MFD control mode. The appropriate frequency box on the selected display is outlined by a light blue selection box, which flashes for a few seconds to indicate Control Unit activity. Selection of a different display control or radio tuning mode results in cancelation of the previous radio tuning mode.

(2) System Operation

The displays are connected together via multiple data busses, thus allowing for high-speed communication. The PFD and MFD are directly connected to the GIA63W unit on the respective side, as shown in "Figure 2" on page 3425012. This section discusses the normal and reversionary modes of operation as well as the various AHRS modes of the G1000 system.

(a) Normal Operation

1) PFD

In normal mode, the PFD presents graphical flight instrumentation (attitude, heading, airspeed, altitude and vertical speed), thereby replacing the traditional flight instrument cluster. The PFD also offers control for COM and NAV frequency selection.

2) MFD

In normal mode, the right portion of the MFD displays a full-color moving map with navigation information, while the left portion of the MFD is dedicated to the Engine Indication System (EIS). "Figure 4" gives an example of the G1000 displays in normal mode.

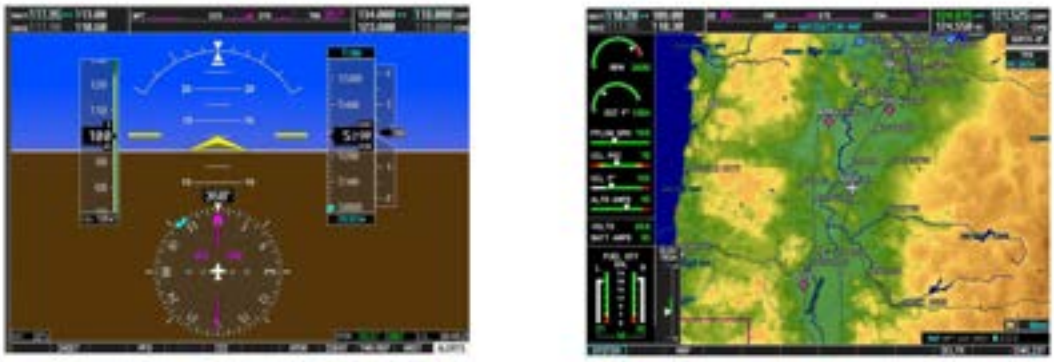
(b) Reversionary Mode

NOTE: The G1000 system alerts the pilot when backup paths are utilized by the LRUs. Refer to the Appendices in the appropriate Garmin Pilot's Guide for further information regarding system-specific alerts.

Reversionary mode is a mode of operation in which all important flight information is presented on at least one functioning displays (see "Figure 5"). Transition to reversionary mode should be straight-forward for the pilot, for flight parameters are presented in the same format as in normal mode. In the event of a PFD failure, the G1000 system automatically switches to reversionary (backup) mode. In reversionary mode, all important flight information is presented on the remaining display, whether the MFD or PFD.

- PFD failure – Press the DISPLAY BACKUP button on the GMU 1347 (see "Figure 6" on page 3425018) to put MFD into reversionary mode.
- MFD failure – The PFD automatically goes into reversionary mode.

Both displays may be set to reversionary mode at any time by pressing DISPLAY BACKUP on the GMU 1347 (see "Figure 6").



Normal Mode
Figure 4



Reversionary Mode (PFD or MFD Failure)
Figure 5



Audio Control and Display Backup (GMA 1347)
Figure 6

D. Maintenance Mode

CAUTION: ENTER “MAINTENANCE MODE” WHEN ACTIVATING THE GARMIN G1000 SYSTEM ON THE GROUND TO CONDUCT TESTS, INSPECTIONS, ETC.

The “Hobbs” Meter is activated by the GIA 63W Integrated Avionics Units and will run when one of the following is true, depending on how it was configured:

(1) the engine is running (Option A); or

(2) the airplane is inflight (Option B); or

(3) the GIA 63W No. 1 is ON (Option C).

To prevent needless accumulation of “hours” (i.e., to disable the “Hobbs” Meter) on the ground a Maintenance Mode has been implemented in the Garmin G1000 software.

Enter “Maintenance Mode” before exiting the splash screen at system startup by pressing SoftKeys 1, 2, 3, 1 in that order. The “MAINT MODE ON” advisory will then be displayed while the system remains in maintenance mode.

E. Troubleshooting

Troubleshoot the Garmin G1000 system by first identifying, then isolating the specific failure to the responsible LRU. There are several indications that the G1000 presents to the pilot or technician, showing overall system condition. A course of action should be determined based on the information presented on the display.

The fifth AUX group page on the MFD (see "Figure 7") provides LRU health status by means of a green check or a red ‘X’. Also, LRU software versions are shown along with other database versions and dates. If a red ‘X’ is shown for an LRU, see the appropriate LRU troubleshooting chart for guidance.

Typically, the G1000 Alerting System provides alerts/annunciations in conjunction with the information presented at the fifth AUX page.

CAUTION: POST INSTALLATION SET-UP,” BELOW, PROVIDES DETAILED INSTRUCTIONS ON EQUIPMENT CONFIGURATION AND RETURN-TO-SERVICE TESTING. ANYTIME A G1000 COMPONENT OR LRU IS REMOVED, SWAPPED, OR REPLACED, THE TECHNICIAN MUST FOLLOW THE PROCEDURES GIVEN IN “POST INSTALLATION SET-UP” TO ENSURE PROPER OPERATION OF THE SYSTEM.

Troubleshooting information is provided as follows:

(1) Line Replaceable Unit (LRU) Failures

See "Chart 1" on page 34250113 as the first step in troubleshooting any LRU.

(2) Engine / Airframe Sensor Failures - see "Chart 2" on page 34250118.

(3) TAWS (Optional) - "Chart 3" on page 34250120.

(4) FAILED PATH - see "Failed Path Messages." on page 34250121.

(5) GDU 1040 - see "Chart 9" on page 34250158 and "Chart 10" on page 34250160.

(6) GMA 1347 - see "Chart 11" on page 34250166 and "Chart 12" on page 34250167.

(7) GIA 63W - see "Chart 13" on page 34250168 and "Chart 14" on page 34250170.

(8) GEA 71 - see "Chart 15" on page 34250176.

(9) GTX 33 - see "Chart 16" on page 34250177.

(10) GRS 77 - see "Chart 17" on page 34250178 and "Chart 18" on page 34250179.

(11) GMU 44 - see "Chart 17" on page 34250178 and "Chart 18" on page 34250179.

(12) GDC 74A - see "Chart 19" on page 34250181.

(13) GDL 69A (Optional) - see "Chart 20" on page 34250182 and "Chart 21" on page 34250183.

(14) GTS 800 - see "Chart 22" on page 34250184.



System Status Page (AUX Group) - Example
Figure 7

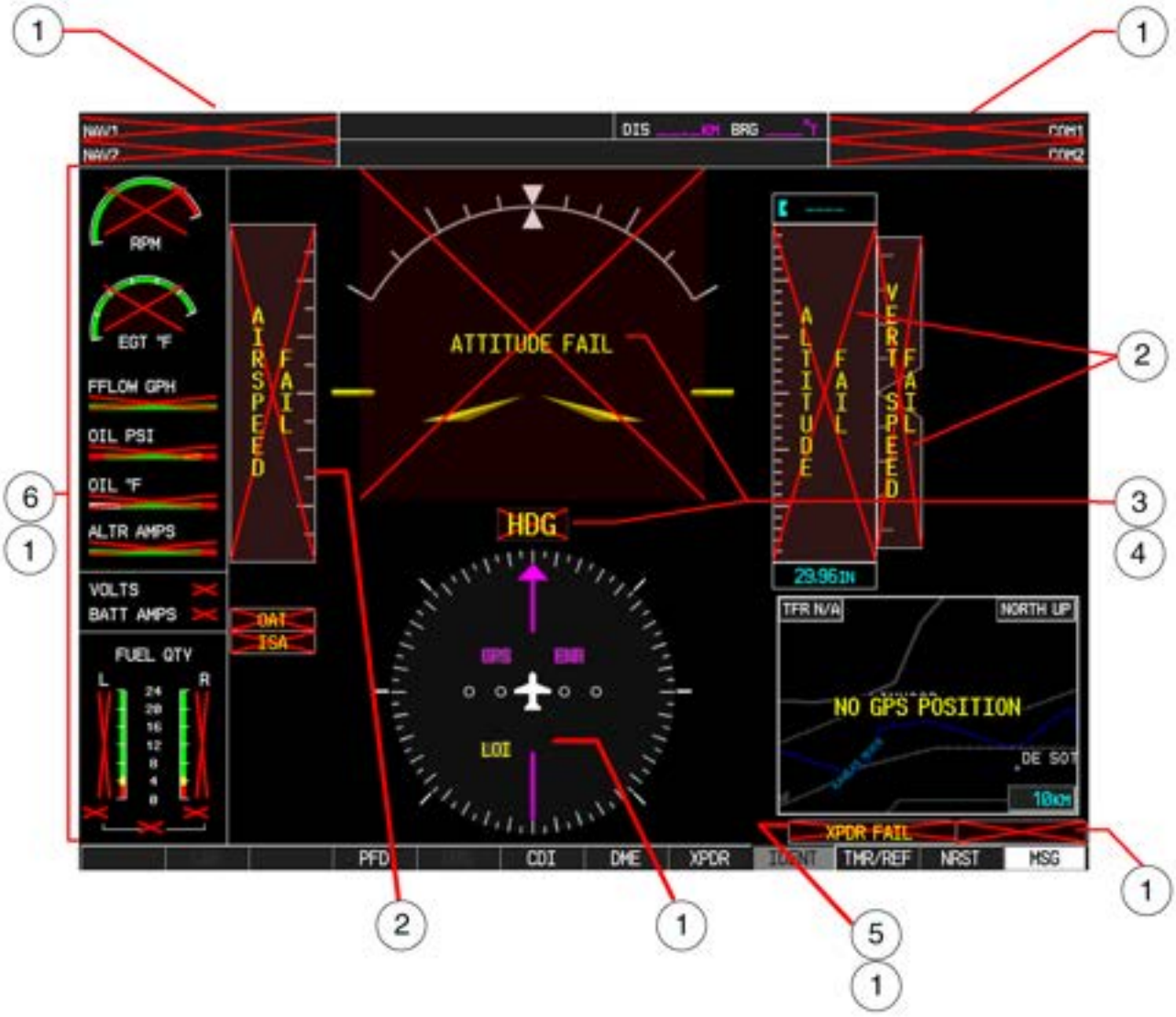
(15) Software Configuration and Status Verification (PIR-PPS55034, Rev. D.)

(a) Enter the configuration mode on the PFD and the MFD by applying power while holding ENT on each display.

(b) Verify on the SYSTEM CONFIGURATION PAGE that the following are valid. (Indicated by green annunciations): MFD1, PFD1, GIA1, GIA2, and GDL 69A (optional).

(c) Verify on the SYSTEM STATUS PAGE that the following are valid. (Green annunciations indicate that the system is healthy): MFD1, PFD1, GIA1, GIA2, and GDL 69A (optional).

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1. GIA 63W Integrated Avionics Units
2. GDC 74A Air Data Computer
3. GRS 77 AHRS
4. GMU 44 Magnetometer
5. GTX 33 Transponder
6. GEA 71 Engine Airframe Units

System Fail
Figure 8

CHART 1 (Sheet 1 of 5)
LRU TROUBLESHOOTING

System Failure Troubleshooting

The following table provides basic troubleshooting guidance for LRU failures.

Invalid Data Field	Associated LRU(s)	Solution
Low Speed Awareness Band permanently displayed 	GDU	• With aircraft weight on wheels and the G1000 in Configuration Mode, check that the **RH GEAR ON GROUND** discrete in indication on GIA I/O CONFIGURATION page is illuminated. • If indication is illuminated Green, switch GIA1 and GIA2, to verify location of problem: ✓ If problem is resolved, replace defective unit. ✓ If indication is not illuminated, check for wiring faults between the GIAs and aircraft squat switch system.
Low Speed Awareness Band permanently inhibited 	GDU	• With aircraft weight off wheels and the G1000 in Configuration Mode, check that the **RH GEAR ON GROUND** discrete in indication on GIA I/O CONFIGURATION page is not illuminated. • If indication is not illuminated Green, switch GIA1 and GIA2, to verify location of problem: ✓ If problem is resolved, replace unit. ✓ If indication is illuminated, check for wiring faults between the GIAs and aircraft squat switch system.
NAV1 & COM1 	GIA1	• Check configuration settings for GIA1 and PFD. • Swap GIA1 and GIA2, to verify location of problem: ✓ If problem follows GIA1, replace GIA1. • Check Ethernet interconnect from GIA1 to PFD. ✓ If problem persists, replace PFD.
NAV2 & COM2 	GIA2	• Check configuration settings for GIA2 and MFD. • Swap GIA1 and GIA2, to verify location of problem: ✓ If problem follows GIA2, replace GIA2. • Check Ethernet interconnect from GIA2 to MFD. ✓ If problem persists, replace MFD.

CHART 1 (Sheet 2 of 5)
LRU TROUBLESHOOTING

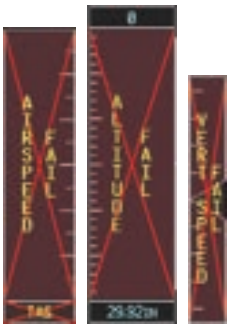
Invalid Data Field	Associated LRU(s)	Solution
GPS DR 	GIA1 or GIA2	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Check GPS1 and GPS2 signal strength on AUX page 4. Refer to GIA GPS section if GPS can not acquire a position lock for troubleshooting. • Check corresponding GPS antenna and cable for faults. • Check Ethernet interconnect between the PFD to GIA1 or MFD to GIA2 for faults. • Swap GIA1 and GIA2, to verify location of problem: ✓ If problem follows the GIA, replace the GIA. ✓ If problem persists, replace the PFD or MFD.
XPDR FAIL 	GTX 33	• Check GTX 33 configuration settings for GIA1 and GIA2. • Check GTX 33/GIA1 interconnect. • Check GTX 33/GIA2 interconnect. • Perform a SET>ACTV configuration reset on the GTX Config page for each GTX, and verify the aircraft registration is present. • Replace GTX 33. ✓ If problem persists, replace GIA.
TAS FAIL AIRSPEED FAIL ALTITUDE FAIL VERT SPEED FAIL 	GDC 74A	• Inspect GDC 74A pitot/static plumbing integrity. • Inspect pitot/static ports and associated equipment for faults. • For TAS failure, also check GTP 59 OAT probe. • Check GDC 74A configuration settings for the PFD, MFD, GIA1, and GIA2. ✓ If PFDs, MFD, and GIA configuration settings are correct, replace the GDC configuration module. ✓ If problem persists, replace the GDC 74B.
OAT 	GTP 59	• Check OAT probe wiring and connectors for faults. • Check GDC config module wiring for damage, replace if any is found. • Replace GDC config module • Replace GDC 74A with a known good unit: ✓ If problem persists, replace the GTP 59

CHART 1 (Sheet 3 of 5)
LRU TROUBLESHOOTING

Invalid Data Field	Associated LRU(s)	Solution
HDG FAIL 	GRS 77 & GMU 44	• Ensure metal objects (tool boxes, power carts, etc.) are not interfering with the magnetometer and aircraft is not in hangar, near other buildings, parked over metal drainage culverts or on hard surfaces that may contain steel reinforcements. • Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Cycle power after moving aircraft away from metal objects to determine if metal objects were the source of the interference. Allow up to five minutes for the heading to reinitialize. • Perform a Magnetometer Interference Test to check for interference from onboard electrical system components (e.g. NAV lights). Pay particular attention to any new electrical devices that have been installed since the aircraft was new. Correct any discrepancies that do not allow this test to pass before continuing. • Ensure GRS 77 and GMU 44 connectors are secure. • Check the wiring and any inline connectors between the GRS and GMU for faults. • Recalibrate the GMU 44. • Load configuration files to the PFD, MFD, GIA1, and GIA2. ✓ If problem persists replace the GMU 44. ✓ If problem persists, replace the GRS 77.
	GRS 77 & GMU 44	• If this message persists longer than five minutes, perform AHRS calibration procedures as described later in this chapter.

CHART 1 (Sheet 4 of 5)
LRU TROUBLESHOOTING

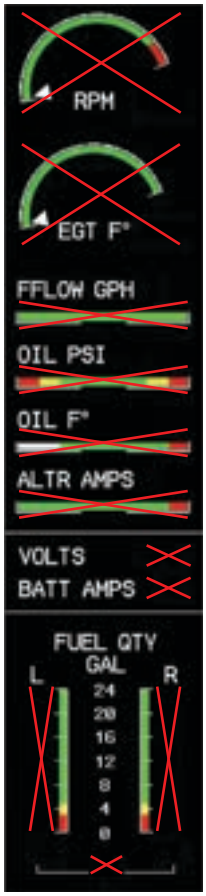
Invalid Data Field	Associated LRU(s)	Solution
ATTITUDE FAIL 	GRS 77	• Ensure that a cell phone or a device using cell phone technology is not turned on (even in a monitoring state) in the cabin. • Cycle GRS 77 power to restart initialization. • Ensure GRS 77 connecter is secure and proper wire harness strain relief is provided. • Ensure the GRS 77 is fastened down tightly in it's mounting rack and that the mounting rack is not loose (CAUTION - do not loosen the mounting rack hardware to the airframe shelf or the aircraft will need to be re-leveled and the PITCH/ROLL OFFSET procedure performed). • Ensure GPS has acquired at least four satellites, has a 3D navigation solution, and a DOP of less than 5.0. This is particularly important for an ATTITUDE FAIL that appears during ground operation only. • Perform an Engine Run-Up Test to check if engine vibration is causing the GRS 77 to go offline. • Load configuration files to the PFD, MFD, GIA1, and GIA2. • Calibrate the GRS 77 (Pitch/Roll Offset and Magnetometer Calibration). • Replace GRS 77. ✓ If problem persists replace GRS 77 configuration module. ✓ Contact Garmin Aviation Product Support if condition continues after replacing the GRS 77 and config module for additional assistance.

CHART 1 (Sheet 5 of 5)
LRU TROUBLESHOOTING

A DME or ADF failure is represented by the following red X's. The PFD displays function as a control head for the remote-mounted Honeywell KR 87 ADF and KN 63 DME.

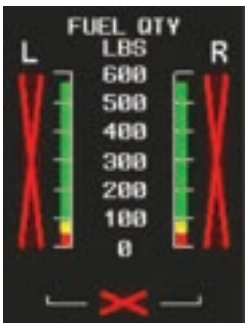
Invalid Field	Sensor	Solutions
 	Honeywell ADF (Optional)	• Ensure that GIA 63W #2 is properly functioning. • Reload the ADF option configurations. • Check for proper operation of the ADF receiver. Ensure that the receiver is receiving power. • Check ADF – GIA2 interconnect.
 	Honeywell DME (Optional)	• Ensure that GIA 63W #2 is properly functioning. • Reload the DME option configurations. • Check for proper operation of the DME receiver. Ensure that the receiver is receiving power. • Check DME – GIA2 interconnect.

CHART 2 (Sheet 1 of 2)
TROUBLESHOOTING ENGINE / AIRFRAME SENSOR FAILURES

Invalid Data Field	Associated LRUs	Solution
Engine/Airframe Sensors (All Invalid) 	GEA 71 & GIA 63W	• If software was loaded to a new GDU display, be sure that the user settings for the replaced display were cleared. Clear user settings by pressing the CLR key on the replaced display while applying power to it. Acknowledge the on-screen prompt by pressing the ENT key or the right-most softkey. • Check for GEA related Alert messages on the PFD. Correct any Alerts concerning software or configuration errors by reloading software or configuration as noted. • Verify GEA internal power supply, configuration, and calibration status in configuration mode. ✓ The internal power supply, configuration and calibration boxes should be green. If they are red, replace the GEA 71. • Verify internal, external, and reference voltages listed in the Main Analog and I/O A Analog boxes are not dashed out (does not include Aircraft Power 1 and 2). ✓ If any voltages are dashed out, replace the GEA. • Ensure the GEA is online (green checkmark on the AUX – SYSTEM STATUS page). ✓ If GEA is not online verify unit is receiving power at the rack connector. ✓ Check the GIA/GEA interconnects for faults. ✓ Reload configuration files to both GIA's and the GEA. • If problem persists, replace the GEA 71.

Invalid Field	Sensor	Possible Solutions (for applicable engine/system)
	Propeller RPM	• Check engine tachometer sensor wiring. • Replace engine tachometer sensor. • Replace GEA 71.
	Exhaust Gas Temp (EGT)	• Check EGT sensor wiring. • Replace EGT sensor . • Replace GEA 71.

CHART 2 (Sheet 2 of 2)
TROUBLESHOOTING ENGINE / AIRFRAME SENSOR FAILURES

Invalid Field	Sensor	Possible Solutions (for applicable engine/system)
	Fuel Flow	• Check fuel flow transmitter wiring. • Check fuel flow transmitter - GEA wiring. • Check power input to fuel flow transmitter. • Replace fuel flow transmitter. • Replace GEA 71.
	Oil Pressure	• Check oil pressure sensor – GEA wiring. • Replace oil pressure sensor. • Replace GEA 71.
	Oil Temperature	• Check oil temperature sensor – GEA wiring. • Replace oil temperature sensor. • Replace GEA 71.
	Fuel Quantity	• Check Fuel Senders - #1 GIA 63W wiring. • Check #1 GIA 63W - #1PFD wiring. • Check MFD - #1 PFD connections. • Replace Fuel Senders left or right • Replace #1 GIA 63W • Replace #1 PFD • For more details see 28-00-00.
	Amp/Volts	• Check Alt amp sensor - GEA 71 wiring. • Replace Alt amp sensor. • Replace GEA 71. • Check main bus - GEA 71 wiring. • Check emergency bus - GEA 71 wiring. • Replace GEA 71

Engine/Airframe Instrument Failures

The above table provides guidance for troubleshooting individual engine/airframe sensor failures. Be sure to also follow previous guidance given for the GEA 71. The technician should troubleshoot to isolate the fault by checking sensor-to-GEA wiring, replacing the suspect sensor, and finally by replacing the GEA 71. Replace one part at a time. Check for correct operation of the sensors and GEA 71 after any part has been replaced. See also Wiring Diagrams in Chapter 91.

CHART 3
TAWS (OPTIONAL) TROUBLESHOOTING

Annunciation	Cause	Solution
	A TAWS system failure has occurred.	• If message occurred on the first power up after unlocking TAWS, cycle power to initialize TAWS. • Ensure each GDU contains the correct database data card. • Ensure the G1000 is configured for TAWS: ✓ If the system is not configured for TAWS, reconfigure per TAWS Unlock Procedure, herein. • Verify GIAs are online. • Ensure a database or GDU SW mismatch has not occurred. ✓ If a mismatch has occurred, load correct database/software files or replace the terrain card.
	TAWS system is currently being tested.	N/A – Test will be conducted up to two minutes.
	TAWS system alerting is disabled	Enable TAWS system alerting by pressing the INHIBIT softkey on the MAP – TAWS page.
	GPS system integrity not high enough to enable TAWS	• Ensure valid GPS position is received from the AUX – GPS STATUS page. • Check GPS antenna & associated coaxial cabling. • Troubleshoot GIAs.

(16) Backup Communications Path Checks.

The G1000 system architecture is designed with redundant communication ports for several LRUs so that critical information can continue to be displayed in the event of an equipment or wiring failure. Of most importance is flight attitude, heading, and air data information. The GRS 77 and GDC 74A each have four separate ARINC 429 data lines which are all capable of sending data to the displays. The GEA 71, GTX 33, and GMA 1347 each have two redundant serial communication paths for the same purpose. See ""Figure 2" on page 3425012" for a basic G1000 block diagram depicting this architecture. When troubleshooting, refer to the G1000 Wiring Diagrams listed in Chapter 91.

(17) Failed Path Messages.

The following message indicates there is a data path connected to the GDU (PFD or MFD) or the GIA 63W (1 or 2) that has failed.

FAILED PATH – A data path has failed.

The FAILED PATH message is triggered by a timeout of any one digital channel. The channels that are checked are listed on these pages in Configuration Mode:

(a) GDU RS-232 / ARINC 429 CONFIG (PFD and MFD).

(b) GIA RS-232 / ARINC 429 CONFIG (GIA1 and GIA2).

(c) GIA CAN / RS-485 CONFIGURATION (GIA1 and GIA2).

NOTE: Once the FAILED PATH message has been triggered, it will remain on the list of messages until the next power cycle. This latching was implemented so that for intermittent failures, the message would remain at the end of the flight (to alert maintenance crew). Also, this keeps the crew from having to acknowledge message repeatedly in the case of intermittent failures.

The box next to each channel indicates the current status of the channel per the below:

Red = data path is known to be failed.

Black = data path status is unknown.

Green = data path is known to be good.

See "Chart 4" thru "Chart 8" on page 34250129 for suggested remedies for specific channels / LRUs.

NOTE: Check the PFD Alert Window for configuration or software error messages regarding the displays or LRUs. Correct any errors before proceeding.

CHART 4
G1000 - MESSAGE ADVISORIES / TROUBLESHOOTING

Data Path Failures		
Failure Message	**Cause**	**Solutions**
FAILED PATH – A data path has failed.	A communications data path status is not being received by the G1000.	Determine which data path has failed: See G1000 backup path test below. Check wiring continuity for the failed path. Swap or replace the affected LRU.

CHART 5
FAILED PATH MESSAGES, CONFIG PAGE (PFD, MFD, AND GIA) - PROCESS

SELECT UNITS = PFD / MFD / GIA1 / GIA2 – RS-232 / ARINC-429 / RS-485 Serial Config Pages

Indicator	Status
GREEN	SELECT UNIT/LRU data path is functioning correctly.
RED	SELECT UNIT/LRU data path is not functioning correctly. • Check avionics status page for SELECT UNIT or LRU configuration or software error messages. Correct any errors before proceeding. • If applicable, check the SELECT UNIT/LRU interconnect wiring and unit connector pins for faults. • Replace LRU. • Reload SELECT UNIT config file. • If problem remains, replace SELECT UNIT. • Remove SELECT UNIT and verify power and ground are present at the appropriate connector. ✈ If power or ground is not present, troubleshoot aircraft wiring for faults. ✈ If power and ground are present, check SELECT UNIT unit and LRU connector for damaged or pushback pins. • Swap PFD to confirm if the problem is in the original PFD. • Replace original PFD if box displays green check after swapping displays. • Ensure LRU connecter is secure and proper wire harness strain relief is provided. • Swap LRU #1 and LRU #2 to confirm if the problem is in the original LRU #1. • Replace original LRU #1 if box displays green check after swapping units.
BLACK	SELECT UNIT/LRU data path functionally is unknown. • Reload SELECT UNIT configuration file.

CHART 6 (Sheet 1 of 2)
FAILED PATH MESSAGES, CONFIG PAGE (PFD AND MFD)

PFD ARINC 429

Channel	LRU	Indicator	Status
IN 1	GRS 77	GREEN	PFD/GRS 77 data path is functioning correctly.
		RED	PFD/GRS 77 data path is not functioning correctly. • Verify GRS 77 status is “OK” using the System Status page on the MFD. If it is not, correct condition before proceeding, referencing GDC troubleshooting sections. o Swap GRS with a known working GRS unit to confirm if the problem is in the original GRS. o Replace original GRS if box turns green after swapping units. • Load PFD configuration file. • Swap PFD with MFD to confirm if the problem is in the PFD. o Replace original PFD if box turns green after swapping displays. • Swap GRS with a known working GRS unit to confirm if the problem is in the original GRS. o Replace original GRS if box turns green after swapping units. Check the PFD/GRS 77 interconnect wiring for faults.
		BLACK	PFD/GRS 77 data path functionally is unknown. Reload PFD configuration file.
IN 2	GDC 74A	GREEN	PFD/GDC 74 data path is functioning correctly.
		RED	PFD/GDC 74 data path is not functioning correctly. • Verify GDC 74 status is “OK” using the System Status page on the MFD. If it is not, correct condition before proceeding, referencing GDC troubleshooting sections. • Load PFD and GDC 74 configuration files. • Swap PFD with MFD to confirm if the problem is in the PFD. o Replace original PFD if box turns green after swapping displays. • Check the PFD/GDC 74 interconnect wiring for faults. Replace GDC 74 if problem remains.
		BLACK	PFD/GDC 74 data path functionality is unknown. Reload PFD configuration file.

CHART 6 (Sheet 2 of 2)
FAILED PATH MESSAGES, CONFIG PAGE (PFD AND MFD)

MFD ARINC 429

Channel	LRU	Indicator	Status
IN 1	GRS 77	GREEN	MFD/GRS 77 data path is functioning correctly.
		RED	MFD/GRS 77 data path is not functioning correctly. • Verify GRS 77 status is “OK” using the System Status page on the MFD. If it is not, correct condition before proceeding, referencing GDC troubleshooting sections. o Swap GRS with a known working GRS unit to confirm if the problem is in the original GRS. o Replace original GRS if box turns green after swapping units. • Load MFD configuration file. • Swap MFD with PFD to confirm if the problem is in the MFD . o Replace original MFD if box turns green after swapping displays. • Swap GRS with a known working GRS unit to confirm if the problem is in the original GRS. o Replace original GRS if box turns green after swapping units. Check the MFD/GRS 77 interconnect wiring for faults.
		BLACK	MFD/GRS 77 data path functionally is unknown. Reload MFD configuration file.
IN 2	GDC 74A	GREEN	MFD/GDC 74A data path is functioning correctly.
		RED	MFD/GDC 74A data path is not functioning correctly. • Verify GDC 74 status is “OK” using the System Status page on the MFD. If it is not, correct condition before proceeding, referencing GDC troubleshooting sections. • Load MFD and GDC 74 configuration files. • Swap MFD with PFD to confirm if the problem is in the MFD. o Replace original MFD if box turns green after swapping displays. • Check the MFD/GDC 74 interconnect wiring for faults. Replace GDC 74 if problem remains.
		BLACK	MFD/GDC 74A data path functionality is unknown. Reload MFD configuration file.

CHART 7 (Sheet 1 of 4)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

GIA1 RS-232

Channel	LRU	Indicator	Status
CHNL 1	GDC 74A	GREEN	GIA1/GDC 74A data path is functioning correctly.
		RED	GIA1/GDC 74A data path is not functioning correctly. • Verify GDC 74A status is OK on the System Status page on the MFD. If it has a red "X", troubleshoot why the GDC is offline before proceeding. • Swap GIA1 and GIA2, reconfigure both GIAs to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA1 if box turns green after swapping units. • Check the GIA1/GDC 74A interconnect wiring and connector pins for faults. Replace GDC 74A if problem remains.
		BLACK	GIA1/GDC 74A data path functionality is unknown. Reload GIA1 configuration files.
CHNL 5	GTX 33 w/TIS	GREEN	GIA1/GTX 33 data path is functioning correctly.
		RED	GIA1/GTX 33 data path is not functioning correctly. • Load GIA1 and GTX 33 configuration files. • Swap GIA1 and GIA2, reconfigure both GIAs to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA1 if box turns green after swapping units. • Check the GIA1/GTX 33 interconnect wiring for faults. Replace GTX 33 if problem remains.
		BLACK	GIA1/GTX 33 data path functionality is unknown Reload GIA1 configuration files.

CHART 7 (Sheet 2 of 4)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

GIA2 RS-232

Channel	LRU	Indicator	Status
CHNL 5	GTX 33 w/TIS	GREEN	GIA2/GTX 33 data path is functioning correctly.
		RED	GIA2/GTX 33 data path is not functioning correctly. • Load GIA2 and GTX 33 configuration files. • Swap GIA2 and GIA1, reconfigure both GIAs to their new locations to confirm if the problem is in the original GIA2. o Replace original GIA2 if box turns green after swapping units. • Check the GIA2/GTX 33 interconnect wiring for faults. Replace GTX 33 if problem remains.
		BLACK	GIA2/GTX 33 data path functionality is unknown Reload GIA2 configuration files.

CHART 7 (Sheet 3 of 4)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

GIA1 RS-232

Channel	LRU	Indicator	Status
CHNL 6	GRS 77	GREEN	GIA1/GRS 77 data path is functioning correctly.
		RED	GIA1/GRS 77 data path is not functioning correctly. • Verify GRS 77 status is OK on the System Status page on the MFD. If it has a red "X", troubleshoot why the GRS is offline before proceeding. • Load GIA1 configuration files. • Swap GIA1 and GIA2, reconfigure both GIAs to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA1 if box turns green after swapping units. • Swap GRS1 out with a known good spare (reconfigure as required) to confirm if the problem is in original GRS1. o Replace original GRS1 if box turns green after swapping units. Check the GIA1/GRS 77 interconnect wiring for faults.
		BLACK	GIA1/GRS 77 data path functionality is unknown. Reload GIA1 configuration files.
CHNL 7	GMA 1347	GREEN	GIA1/GMA 1347 data path is functioning correctly.
		RED	GIA1/GMA 1347 data path is not functioning correctly. • Verify GMA 1347 status is OK on the System Status page on the MFD. If it has a red "X", troubleshoot why the GMA is offline before proceeding. • Load GIA1 and GMA 1347 configuration files. • Swap GIA1 and GIA2, reconfigure both GIAs to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA1 if box turns green after swapping units. Check the GIA1/GMA 1347 interconnect wiring for faults.
		BLACK	GIA1/GMA 1347 data path functionality is unknown. Reload GIA1 configuration files.

CHART 7 (Sheet 4 of 4)
FAILED PATH MESSAGES, GIA RS-232 / ARINC 429 CONFIG PAGE (GIA1 AND GIA2)

GIA1 ARINC 429

Channel	LRU	Indicator	Status
IN 5	GDC 74A	GREEN	GIA1/GDC 74A data path is functioning correctly.
		RED	GIA1/GDC 74A data path is not functioning correctly. • Verify GDC 74A status is OK on the System Status page on the MFD. If it has a red "X", troubleshoot why GDC is offline before proceeding. • Swap GIA1 and GIA2, reconfigure both GIAs to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA1 if box turns green after swapping units. • Check the GIA1/GDC 74A interconnect wiring and unit connector pins for faults. • Replace GDC 74A .
		BLACK	GIA1/GDC 74A data path functionality is unknown. Reload GIA1 configuration files.
IN 6	GRS 77	GREEN	GIA1/GRS 77 data path is functioning correctly.
		RED	GIA1/GRS 77 data path is not functioning correctly. • Verify GRS 77 status is OK on the System Status page on the MFD. If it has a red "X", troubleshoot why GRS is offline before proceeding. • Ensure GRS 77 connecter is secure and proper wire harness strain relief is provided. • Swap GIA1 and GIA2, reconfigure both GIAs to their new locations to confirm if the problem is in the original GIA1. o Replace original GIA1 if box turns green after swapping units. • Check the GIA1/GRS 77 interconnect wiring and unit connector pins for faults. • Replace GRS 77.
		BLACK	GIA1/GRS 77 data path functionality is unknown. Reload GIA1 configuration files.

CHART 8
FAILED PATH MESSAGES, GIA RS-485 CONFIG PAGE (GIA1 AND GIA2)

GIA2 RS-485

Channel	LRU	Indicator	Status
CHNL 1	GEA 71	GREEN	GIA2/GEA 71 data path is functioning correctly.
		RED	GIA2/GEA 71 data path is not functioning correctly. • Verify GEA is powered on using the GEA Status page. • Load GIA2 and GEA configuration files. • Swap GIA2 and GIA1, reconfigure GIAs to their new locations to confirm if the problem is in the original GIA2. o Replace original GIA2 if box turns green after swapping units. o Replace original GEA if box turns green after swapping units. Check the GIA2/GEA 71 interconnect wiring for faults.
		BLACK	GIA2/GEA 71 data path functionality is unknown. Reload GIA2 configuration files.

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F. Recommended Tools

The following tools (or equivalent) are recommended to perform various maintenance tasks on the G1000 system.

(1) Calibrated voltmeter capable of measuring 0-32 Volts DC.
(2) #2 Phillips Screwdriver.
(3) 3/32-inch Hex Tool.
(4) Calibrated digital Level with 0.25 degrees of accuracy capability.
(5) Calibrated VHF NAV/COM/ILS ramp tester.
(6) Calibrated transponder ramp tester including Mode S capability for Mode S transponder equipped aircraft.
(7) Air Data Test Set (ADTS) capable of simulating altitude up to the aircraft's service ceiling.
(8) Headset/Microphone.
(9) Outdoor line-of-site to GPS satellite signals or GPS indoor repeater.
(10) Ground Power Unit (Capable of supplying 28 Vdc).
(11) Calibrated Flight Control Cable Tension Meter or equivalent.
(12) Calibrated digital thermometer suitable for measuring ambient temperature.
(13) Laptop with RS232 emulation software.
(14) M22885/108T8234 extraction tool.
(15) An 0.060" 6-Spline wrench.
(16) WX-PA portable analyzer kit (If Stormscope is installed).
(17) Calibrated torque wrench capable of measuring 0–70 in-lbs.
(18) Standard sockets & wrenches (3/8", 9/16", and 13/16").

G. Inspections

(1) Annual / 100 Hour Inspection

Each 12 months or 100 hours time-in-service, whichever comes first:

(a) Under Post-Installation Procedures, below:

Perform the "G1000 Connection Check".

(b) Under LRU Test Procedures, Initial Display Testing, below, perform the "PFD Fan Fail CAS Test".

(2) 500 Hour Inspection

Perform the following function tests of the Avionics, PFD, and MFD cooling fans:

(a) Avionics Cooling Fan

1) Turn OFF all electrical components and avionics systems.
2) Pull the INTEG AV1 circuit breaker.
3) Turn the master switch ON.
4) Push IN the INTEG AV1 circuit breaker. Verify airflow from the Avionics Cooling Blower by motor sound and, if required, by sight (i.e., use a flashlight to observe the fan turning in the intake opening).
5) After checking blower, turn OFF master switch.

(b) PFD and MFD Cooling Fans

The PFD and MFD cooling fans are wired into the PFD and MFD circuit breakers, respectively. Accordingly, function test them as follows:

1) Turn OFF all electrical components and avionics systems.
2) Pull the PFD and MFD circuit breakers.
3) Turn the master switch ON.
4) Push IN the PFD circuit breaker. Verify operation of the PFD cooling fan by motor sound. Pull the PFD circuit breaker.
5) Push IN the MFD circuit breaker. Verify operation of the MFD cooling fan by motor sound. Pull the MFD circuit breaker.
6) After checking cooling fans, turn OFF master switch and push IN PFD and MFD circuit breakers.

(3) 1000 Hour Inspection

Each 1000 hours time-in-service visually inspect each unit listed below for corrosion, damage, or other defects. Replace any damaged parts as required. Inspection may require the temporary removal of a unit or units to gain access to connectors.

(a) GDU 1040 PFD and MFD

Remove the PFD and MFD. Inspect the mounting surface and connector for corrosion, heavy oxidation or other damage.

(b) GMA 1347 Audio Panel

Inspect the GMA 1347 unit(s), rack and connectors for corrosion or other defects.

(c) GIA 63W Integrated Avionics Units

Inspect the GIA 63W #1 and #2 unit, rack and connectors for corrosion or other defects.

(d) GEA 71 Engine/Airframe Unit

Inspect the GEA 71 unit, rack, and connectors for corrosion or other defects.

(e) GTX 33ES Mode S Transponder.

Inspect the GTX 33ES unit, rack and connectors for corrosion or other defects.

(f) GDL 69A - Datalink (if installed)

Inspect the GDL 69A unit, rack and connectors for corrosion or other defects.

(g) PFD and MFD Cooling Fans

Inspect the PFD and MFD cooling fans for accumulation of dust or other damage. Remove excess dust as required.

(h) Avionics Cooling Fan

Inspect the avionics cooling fan for accumulation of dust or other damage. Remove excess dust as required.

(i) GRS 77 AHRS

Inspect the GRS 77 unit, rack and connector for corrosion or other defects.

(j) GMU 44 Magnetometer

Remove the GMU 44 and inspect the mounting hardware and GMU 44 for corrosion or other damage.

(k) Perform electrical bond tests on all G1000 equipment. See 51-80-00.

(l) Perform visual checks on the shield terminations for any degradation.

H. Components

CAUTION: ENTER “MAINTENANCE MODE” WHEN ACTIVATING THE GARMIN G1000 SYSTEM ON THE GROUND TO CONDUCT TESTS, INSPECTIONS, ETC. SEE MAINTENANCE MODE UNDER TROUBLESHOOTING, ABOVE.

(1) General

Maintenance of the G1000 system components is “On Condition” only. The following provides basic information regarding unit removal, installation, and troubleshooting. Basic guidance on loading and configuring software is provided under Post Installation Set-up, below. When removing and/or replacing any G1000 component, always ensure that aircraft power is off. Unplug any auxiliary power supplies.

Before removing any G1000 LRU, it is required that the technician verify the LRU software part numbers.

To check an LRU software part number and/or version:

(a) Start the G1000 system in Configuration Mode as described in Configuration Mode Overview under Post Installation Set-up, below.

(b) The System Status page shows a list of LRUs in the LRU window. Activate the cursor and highlight the LRU window.

(c) Use the FMS knob to scroll through the list in the window and select the desired LRU.

(d) The software part number and version is displayed in the DATA window. Compare this to the latest configuration shown in the Required Equipment List (REL), "Chart 29".

NOTE: If a faulty LRU is not reporting its software version and part number, check aircraft maintenance logs for last software version loaded and verify against the latest configuration.

(2) GDU 1040 PFD and MFD

See "Figure 9".

CAUTION: THE GDU 1040S USE A LENS COATED WITH A SPECIAL ANTI-REFLECTIVE COATING THAT IS VERY SENSITIVE TO SKIN OILS, WAXES AND ABRASIVE CLEANERS. CLEANERS CONTAINING AMMONIA WILL HARM THE ANTI-REFLECTIVE COATING. IT IS VERY IMPORTANT TO CLEAN THE LENS USING A CLEAN, LINT-FREE CLOTH AND AN EYEGLASS LENS CLEANER THAT IS SPECIFIED AS SAFE FOR ANTI-REFLECTIVE COATINGS.

(a) Description

Two Garmin GDU 1040 displays are installed in the instrument panel. The GDU 1040 units, 10.4-inch LCD displays with 1024x768 resolution, and either may serve as a PFD or MFD. The stand-by instruments are located to the left of the pilot's display. A GMA 1347 Audio Panel is located between the two GDU 1040 displays.

The GDU 1040 units communicate with each other and the GIA 63W units through high-speed data bus (HSDB) Ethernet connections.



GDU 1040 - PFD and MFD Displays
Figure 9

(b) Troubleshooting

See "Chart 1" on page 34250113 ,"Chart 9" on page 34250158, and "Chart 10" on page 34250160.

(c) Removal

1) Using a 3/32-inch hex tool, rotate all four quarter-turn fasteners counterclockwise until they reach their stops.
2) Carefully remove the display from the panel.
3) While supporting the display, disconnect the connector.

NOTE: Retain the database SD card(s) from the original display to install in the new display after loading software.

(d) Installation

NOTE: If replacing the PFD with a new, repaired, or exchanged GDU, the new GDU must be placed in the MFD location to load the GDU system to prevent the deletion of configuration files residing in the PFD configuration module. It is not necessary to install a display in the PFD position to load GDU software into a display installed in the MFD position. After loading GDU software into the display in the MFD position, turn off the system and move the GDU to the PFD position. Then, continue the GDU installation by loading the PFD configuration files.

1) Visually inspect the connector and pins for signs of damage. Repair any damage. While supporting the display, connect the connector to the rear of the unit.
2) Carefully insert the display into the panel cutout, ensuring that all 4 quarter-turn fasteners align with the corresponding holes.
3) Seat the display in the panel cutout. Do not use excessive force while inserting the display.
4) Once seated, rotate all four quarter-turn fasteners clockwise to lock the display to the panel.

(e) Return to Service

1) Original Displays Reinstalled.

If the removed display(s) are re-installed in their original positions, no software or configuration loading is required. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process. Continue to the PFD/MFD Test procedure under LRU Test Procedures.

2) New, Repaired or Exchanged Display(s) Installed.

If a new, repaired or exchanged GDU 1040 is installed, the correct software and configuration files must be loaded to the unit. See Software Files, G1000 Software/ Configuration Procedure, above. If ChartView or TAWS were previously installed, these must be reactivated. See Chartview and TAWS Unlock Procedure respectively. If any other options were previously installed, these must also be reactivated per Optional Equipment Configuration. Then continue to the PFD/MFD Test Procedure under LRU Test Procedures.

(3) GMA 1347 Audio Panel

See "Figure 10".



GMA 1347 Audio Panel
Figure 10

(a) Description

The Garmin GMA 1347 Audio Panel integrates NAV/ COM digital audio, intercom system and marker beacon controls. The GMA 1347 panel provides control of all cockpit intercom/mic systems as well as NAV/COM/ILS audio. The unit also provides display reversion mode control through a large red Display button. Power to the audio panel is provided by the battery switch or by applying external power. The GMA 1347 unit interfaces with the existing marker beacon antenna as well as existing mic and phone jacks.

(b) Troubleshooting

See "Chart 1" on page 34250113, "Chart 11" on page 34250166, and "Chart 12" on page 34250167.

(c) Removal

1) Using a 3/32-in hex tool, turn the hex nut counter-clockwise until the GMA 1347 is unlocked from its location.

2) Carefully remove the GMA 1347 from its rack.

(d) Installation

1) Visually inspect the connectors using a flashlight to ensure there are no bent or damaged pins. Repair any damage.

2) Gently insert the GMA 1347 into the rack until the locking tab engages the rack.

3) Begin to turn the hex nut clockwise. This draws the unit into the rack until seated. Do not overtighten the nut.

(e) Return to Service

1) Original GMA 1347 Reinstalled.

No software or configuration loading or testing is required if the original panel is reinstalled. This does not include units that were returned for repair: their software and configuration files are deleted during the repair testing process. Continue to the Return to Service Procedure, below.

2) New, Repaired or Exchanged GMA 1347 Installed.

If a new, repaired, or exchange GMA 1347 is installed, load the following files after removing the Supplemental Database cards from the bottom slots of each display.

Identify which card is removed from the MFD so it is reinserted into the MFD when software and configuration loading is complete. It contains a FliteChart or ChartView database which is used by only the MFD. Reinstall all the cards after the software and configuration file loads are complete.

a) GMA Software

b) GMA Configuration

c) GMA Files for optional equipment installed on the aircraft (i.e. ADF, HF radio, etc.).

Then, continue to GMA 1347 Test Procedure under LRU Test Procedures.

(4) GIA 63W Integrated Avionics Unit

See "Figure 11".

(a) Description

Two Garmin GIA 63W Integrated Avionics Units (IAUs) contain the VHF COM/NAV receivers, WAAS GPS receiver, Flight Director, and system integration microprocessors. The GIAs also serve as a communication interface to all other G1000 LRUs in the system. Each GIA 63W communicates directly with the on-side GDU 1040 display using a HSDB Ethernet connection. Both GIAs are located directly behind the PFD and MFD, installed into their respective LRU racks. Power is provided to GIA1 by the essential bus or the emergency bus. GIA2 is powered by the avionics bus.



GIA 63W Integrated Avionics Unit
Figure 11

1) Both GIA 63s interface to the following equipment:

- VOR/LOC/Glideslope Antenna System
- VHF COM #1 & #2 Antennas
- GA 36 and GA 37 GPS/WAAS Antennas
- GMA 1347
- GEA 71
- GDU 1040, #1 & #2
- GTX 33ES
- GDC 74A #1
- GDC 74A #2 (if installed)
- GRS 77 #1
- GRS 77 #2 (if installed)
- Fuel Quantity Sensors (left and right)

2) The GIA 63 #2 interfaces to the following additional equipment:
   - S-TEC 55X AFCS (if installed)
   - KR 87 ADF (if installed)
   - KN 63 DME (if Installed)
   - WX 500 Stormscope (if installed)

NOTE: The GIA 63W is compatible only with the antennas listed in the GIA 63W Installation Manual, Garmin P/N 190-00303-05.

(b) Troubleshooting

See "Chart 1" on page 34250113, "Chart 13" on page 34250168, and "Chart 14" on page 34250170.

(c) Removal

1) Remove the MFD as described above, under “GDU 1040 PFD and MFD.”
2) Unlock the GIA 63W handle by loosening the Phillips screw on the handle.
3) Pull the handle upward to unlock the GIA 63W. Gently remove the unit from the rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GIA 63W into its rack. The handle should engage the dogleg track.
3) Press down on the GIA 63W hand to lock the unit into the rack.
4) Lock the handle to the GIA 63W body using the Phillips screw.
5) Reinstall the MFD as described above.

(e) Return to Service

1) Original GIA 63W(s) Reinstalled

   No software or configuration loading is required if the removed GIA is reinstalled in its original position (GIA1 and GIA2 in their original racks). This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process. Continue to the Return to Service Procedure, below.

2) Original GIA 63W(s) Swapped for Troubleshooting

   No software loading is required if the original GIA units are reinstalled in opposite positions (GIA1 and GIA2 in opposite unit racks). However, configuration loading is required. See G1000 Software/Configuration Procedure, and then continue to the GIA 63W Test Procedure under LRU Test Procedures.

3) New, Repaired or Exchanged GIA 63W(s) Installed

   If a new, repaired or exchanged GIA 63 is installed, the correct software and configuration files must be loaded to the unit. See G1000 Software/Configuration Procedure, and then continue to the GIA 63W Test Procedure under LRU Test Procedures.

(5) GEA 71 Engine/Airframe Unit

See "Figure 12".

(a) Description

The Garmin GEA 71 Engine/Airframe Unit provides engine/airframe data to the G1000 system. Data received from transducers/sensors is processed and sent to GIA 63Ws (via RS-485 digital interface)interface), and subsequently to the GDU 1040 MFD. Engine parameters are normally displayed on the MFD. In the event of a MFD failure, all engine instruments can be displayed on the PFD. The GEA is located behind the MFD and is mounted in a vertical orientation. Power is received from the essential bus or the emergency bus.



GEA 71 Engine/Airframe Unit
Figure 12

The GEA interfaces to the following:

- Exhaust Gas Temperature Sensors
- Amp/Volts Sensor
- Oil Pressure Sensor
- Oil Temperature Sensor
- Fuel Flow Sensor
- Fuel Quantity Sensors
- Propeller Speed Sensor (Tachometer

(b) Troubleshooting

See "Chart 1" on page 34250113 and "Chart 15" on page 34250176.

(c) Removal

1) Remove the MFD as described above.
2) Unlock the GEA 71 handle by unscrewing the Phillips screw.
3) Pull the handle upward to unlock the GEA 71.
4) Gently remove the GEA 71 from its rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GEA 71 into the rack. The handle should engage the dogleg track.
3) Press down on the handle to lock the unit into place.
4) Lock the handle to the GEA 71 body using the Phillips screw.
5) Reinstall the MFD as described above.

(e) Return to Service

1) Original GEA 71 Reinstalled

No software or configuration loading is required if the removed GEA 71 is reinstalled. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process. Continue to the Return to Service Procedure, below.

2) New, Repaired or Exchanged GEA 71 Installed

If a new, repaired or exchanged GEA 71 is installed, the correct software and configuration files must be loaded to the unit. See G1000 Software/Configuration Procedure, and then continue to the GEA 71 Test Procedure under LRU Test Procedures.

(6) GEA 71 Backshell Thermocouple

See "Figure 13".

(a) Removal

1) Remove GEA 71 as described above.

2) Remove GEA connector backplate.

3) Remove connector J701 (5) from the backplate.

4) Remove cover (6) from the backshell.

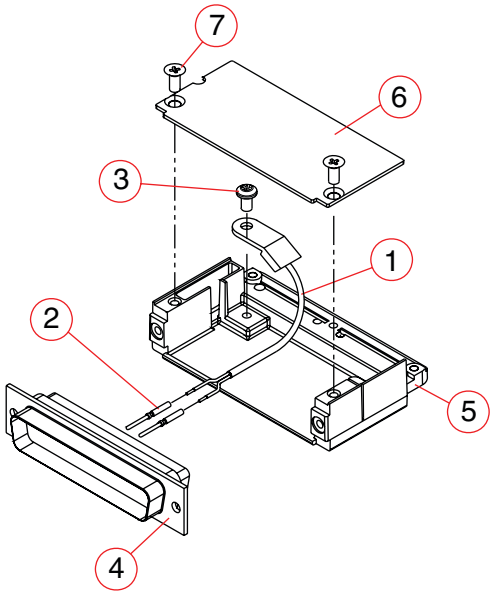
5) Unscrew thermocouple from boss on backshell. Extract the thermocouple pins from the connector.

(b) Installation

1) Crimp pins (2) onto each of the thermocouple wires (1). Ensure that pre-stripped wire length is 1/8" prior to crimping.

2) Insert newly crimped pins and wires into the appropriate connector housing location (4) as specified in the electrical schematic.

3) Place thermocouple body (1) onto the backshell boss (5). Place the thermocouple as shown in "Figure 13" so that the wires exit towards the bottom of the backshell.



GEA 71 Backshell Thermocouple
Figure 13

4) Fasten thermocouple tightly to backshell using the provided screw (3).

5) Fasten cover (6) to backshell using the provided screws (7).

(7) GTX 33(ES) Mode S Transponder

See "Figure 14".



GTX 33 Mode S Transponder
Figure 14

(a) Description

The Garmin GTX 33(ES) provides Mode A, C, and S altitude and position reporting information from the G1000 system. The unit is mounted directly behind the MFD on the right side avionics shelf. Power is received from the essential bus. Similarly to the GEA 71, the GTX 33(ES) unit sends data via RS-232 directly to the on-side GIA 63W. Information is then sent to the PFD, where the pilot can control the transponder. The GTX 33(ES) unit interfaces with the transponder antenna.

In the ES, or Extended Squitter, version the GTX 33 provides ADS-B compliance. Using precise GPS-referenced positioning information, the extended squitter technology enables transponders to automatically transmit traffic surveillance data – including aircraft flight ID, position, altitude, velocity, climb/descent, and heading information.

(b) Troubleshooting

See "Chart 1" on page 34250113 and "Chart 16" on page 34250177.

(c) Removal

1) Remove the MFD as described above.
2) Unlock the GTX 33 handle by loosening the Phillips screw on the handle.
3) Pull the handle upward to unlock the GTX 33. Gently remove the unit from the rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GTX 33 into its rack. The handle should engage the dogleg track.
3) Press down on the GTX 33 handle to lock the unit into the rack.
4) Lock the handle to the GTX 33 body using the Phillips screw.
5) Reinstall the MFD as described above.

(e) Return to Service

1) Original GTX 33(ES) is Reinstalled

No software or configuration loading is required if the removed GTX 33 is reinstalled. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process. Continue to GTX 33 Test Procedure under LRU Test Procedures.

2) New, Repaired or Exchanged GTX 33(ES) is Installed

If a new, repaired or exchanged GTX 33 is installed, the correct software and configuration files must be loaded to the unit. See G1000 Software/Configuration Procedure. Also enter the aircraft registration number in the GTX Config page after loading software and configuration files. Then, continue to Transponder Configuration Procedure under LRU Test Procedures.

(8) GDC 74A Air Data Computer

See "Figure 15".

(a) Description

The Garmin GDC 74A ADC compiles information from the pitot/static system and the GTP 59 outside air temperature (OAT) probe to digital air data computations to the G1000 system. The GDC 74A communicates with the GIA 63W, GDU 1040, and GRS 77 using ARINC 429 digital interface. The unit is mounted behind the PFD. Power is provided to the GDC 74A from the essential bus or the emergency bus.



GDC 74A Air Data Computer
Figure 15

NOTE: If any connections in the pitot / static system are opened for maintenance, the entire system must be rechecked per Pitot and Static Systems, Test, in 34-10-00.

(b) Troubleshooting

See "Chart 1" on page 34250113 and "Chart 19" on page 34250181.

(c) Removal

1) Remove the PFD as described above.
2) Disconnect the pitot/static plumbing from the rear of the unit. Disconnect the single connector.
3) Loosen each thumbscrew on the hold-down clamp and remove the clamp.
4) Carefully remove the unit from its mount.

(d) Installation

1) Place the unit in the mounting tray.
2) Position the locking clamp and fasten using the thumbscrews.
3) Connect the pitot/static plumbing.
4) Inspect the connector and pins for damage. Repair any damage. Connect the connector to the unit.
5) Reinstall the PFD as described above.

(e) Return to Service

NOTE: If any connections in the pitot / static system are opened for maintenance, the entire system must be rechecked per Pitot and Static Systems, Test, in 34-10-00.

1) Original GDC 74A is Reinstalled

No software or configuration loading is required if the removed GDC 74A is reinstalled. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process. Continue to GDC 74A Test Procedure under LRU Test Procedures.

2) New, Repaired or Exchanged GDC 74A is Installed

If a new, repaired or exchanged GDC 74A is installed, the correct software and configuration files must be loaded to the unit. See G1000 Software/Configuration Procedure, and then continue to GDC 74A Test Procedure under LRU Test Procedures.

3) New GDC 74A Configuration Module is Installed

The correct configuration files must be loaded if the GDC 74A configuration module has been replaced.

See G1000 Software/Configuration Procedure, then continue to GDC 74A Test Procedure under LRU Test Procedures.

(9) GTP 59 OAT Probe

See "Figure 16".

(a) Description

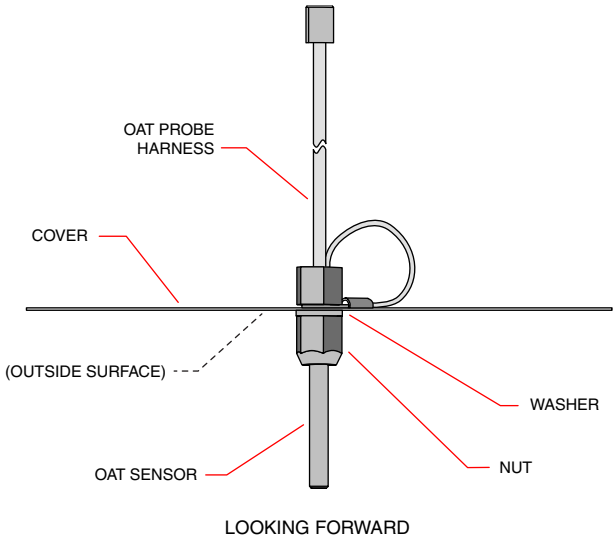
The Garmin GTP 59 OAT Probe provides the GDC 74A with air temperature data. The OAT probe is mounted on an access cover in the underside of the left wing.

(b) Troubleshooting

See "Chart 1" on page 34250113.

(c) Removal

1) Remove the access cover.
2) Disconnect the connector.
3) Use an open-end wrench to hold the probe in place on the inside of the access cover. On the outside of the access cover, loosen the GTP 59 nut and remove the GTP 59, nut and washer.



GTP 59 OAT Probe
Figure 16

(d) Installation

1) Place the GTP-59 grounding strap O-ring connector over the OAT probe end and insert the probe through the access cover from the inside side. Secure with nut and washer and washer on the outside of the access cover. Use an open-end wrench to hold the GTP-59 on the inside and torque the nut on the outside to 100 ± 20 in. lbs.
2) Reconnect the connector to the airplanes wiring harness.
3) Install the access cover.

(10) GRS 77 Attitude and Heading Reference System

See "Figure 17".



GRS 77 Attitude and Heading Reference System
Figure 17

(a) Description

The Garmin GRS 77 AHRS unit provides attitude and heading information to the G1000 system. The unit is mounted on the aft equipment shelf just forward of F.S. 191.00. It contains advanced tilt sensors, accelerometers, and rate sensors. The unit interfaces with the GDC 74A and GMU44 Magnetometer and utilizes GPS signals from the GIA 63Ws. Actual attitude and heading information is sent using ARINC 429 digital interface to both GDU 1040s and GIA 63Ws. Power is provided to the GRS 77 unit from the essential bus or the emergency bus. The GRS 77 interfaces with and provides power to the GMU 44 Magnetometer. The GRS 77 supplies attitude and heading information directly to the PFD, MFD and GIAs.

NOTE: GRS 77 Earth Magnetic Field Updates

The GRS 77 utilizes an Earth magnetic field model which is updated once every five years. The update is expected to be available from Garmin in 2020 and every five years thereafter, as long as the GRS 77 remains a Garmin-supported product. The G1000 system alerts the operator that the magnetic field database is out of date by issuing the message “AHRS SERVICE – AHRS Magnetic-field model needs update”. Garmin will distribute update instructions when updates are available.

(b) Troubleshooting

See "Chart 1" on page 34250113, "Chart 17" on page 34250178, and "Chart 18" on page 34250179.

(c) Removal

1) Remove the baggage compartment rear closeout panel.
2) Disconnect the AHRS connector.
3) Remove the four Phillips thumbscrews with a screwdriver and set them aside.
4) Gently lift the GRS 77 from the mounting plate. (If the mounting plate is removed, the GRS 77 must be re-calibrated. See Post-Installation Setup, below.

(d) Installation

1) Place the GRS 77 on the mounting plate, ensuring the orientation is correct.
2) Fasten the unit to the plate using the Phillips thumbscrews.
3) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage. Connect the connector to the GRS 77.
4) Calibrate and test the GRS 77 according to Post-Installation Setup, below.
5) Reinstall the baggage compartment rear closeout panel.

(e) Return to Service

1) Original GRS 77 is Reinstalled

If the original GRS 77 was reinstalled, then no software loading is required. This does not include a unit that were returned for repair as their software is deleted during the repair testing process.

a) If the GRS rack was not removed or loosened, continue to the GRS/GMU Test Procedure under LRU Test Procedures.

b) If the GRS rack was removed or loosened, continue to GRS/GMU Calibration Procedures under LRU Test Procedures. .

2) New, Repaired or Exchanged GRS 77 is Installed

If a new, repaired or exchanged GRS 77 is installed, then software must be loaded and the unit must be calibrated. See G1000 Software/Configuration Procedure for software loading. Then, proceed as follows:

a) If the GRS rack was not removed or loosened, continue to the GRS/GMU Test Procedure under LRU Test Procedures.

b) If the GRS rack was removed or loosened, continue to the GRS/GMU Calibration Procedures under LRU Test Procedures.

3) New GRS 77 Configuration Module is Installed

If the GRS 77 configuration module was replaced, no software loading is required. Continue to the GRS/GMU Calibration Procedures under LRU Test Procedures.

(11) GMU 44 Magnetometer

See "Figure 18".



GMU 44 Magnetometer
Figure 18

(a) Description

The GMU 44 provides horizontal and vertical magnetic field information to the GRS 77 ARHS. This LRU is powered via the GRS 77. It communicates with the GRS 77 using a RS-485 digital interface. The GMU 44 bracket is mounted on a wing access cover assembly at the bottom of the left wing between WS 148.60 and 131.60.

(b) Troubleshooting

See "Chart 1" on page 34250113 and "Chart 19" on page 34250181.

(c) Removal

1) Remove the eight non-ferrous screws securing the magnetometer cover assembly to the underside of the left wing.

2) Disconnect the wiring harness connection and remove the magnetometer and cover assembly.

3) Remove the three non-ferrous screws securing the magnetometer to its mounting bracket. Disconnect the ground strap. Remove the magnetometer.

(d) Installation

CAUTION: THE MAGNETOMETERS ARE SECURED TO THE MOUNTING BRACKET WITH BRASS SCREWS. ENSURE ONLY BRASS SCREWS ARE USED WHEN REINSTALLING.

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.

2) Place the GMU 44 into its mounting bracket, make sure the arrow on the GMU44 aligns with the arrow on the mounting bracket, and secure three non-ferrous screws. Torque 12-15 in-lbs. Connect the grounding strap.

3) Reconnect the GMU 44 pigtail to the wing wiring harness.

4) Reinstall the magnetometer cover assembly, make sure the arrow on the GMU44 is pointing forward, and secure the assembly with eight (8) non-ferrous screws.

(e) Return to Service

1) Original GMU 44 is Reinstalled

If the original GMU 44 was reinstalled, then no software loading is required. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process. Continue to the GRS/GMU Test Procedure under LRU Test Procedures.

2) New Repaired or Exchange GMU 44 is Installed

If the repaired or exchange GMU 44 was replaced with a new unit then software must be loaded. See G1000 Software/Configuration Procedure. Then continue to the GRS/GMU Calibration Procedures under LRU Test Procedures.

(12) GDL 69A Data Link Unit (Optional.)

See "Figure 19".



GDL 69A Data Link Unit
Figure 19

(a) Description

The GDL 69A provides XM Radio weather and music entertainment through means of a dedicated satellite data link. The GDL 69A is mounted to a rack on the upper left side of the fuselage forward of F.S. 191.00. Power to the GDL 69A is received from the avionics bus. The GDL 69A sends weather data via HSDB connection to the MFD, where the data link interface is controlled. It is linked directly to the GMA 1347 Audio Panel for distribution of the audio signal.

(b) Troubleshooting

See "Chart 20" on page 34250182 and "Chart 21" on page 34250183.

(c) Removal

1) Remove the baggage compartment rear closeout panel.
2) Unlock the GDL 69A handle by unscrewing the Phillips screw.
3) Pull the handle upward to unlock the GDL 69A.
4) Gently remove the GDL 69A from its rack.

(d) Installation

1) Visually inspect the connectors to ensure there are no bent or damaged pins. Repair any damage.
2) Gently insert the GDL 69A into the rack. The handle should engage the dogleg track.
3) Press down on the handle to lock the unit into place.
4) Lock the handle to the GDL 69A body using the Phillips screw.
5) Reinstall the baggage compartment rear closeout panel.

(e) Return to Service

1) Original GDL 69A is Reinstalled

No software or configuration loading is required if the original GDL 69A is reinstalled. This does not include units that were returned for repair as their software and configuration files are deleted during the repair testing process. Continue to the GDL 69A Test Procedures under LRU Test Procedure.

2) New, Repaired or Exchanged GDL 69A is Installed

If a new, repaired or exchanged GDL 69A is installed, the correct software and configuration files must be loaded to the unit and the XM Satellite Radio subscription must be reactivated. Then, continue to the GDL 69A Test Procedure under LRU Test Procedures.

(13) GRT 10/GRC 10 Wireless Remote System (Optional.)

See"Figure 20".

(a) Description

The optional GRT 10 wireless transceiver and GRC 10 wireless remote control are for passenger use to control the audio functions (volume adjustment and channel changing) of the Garmin GDL 69A Datalink Receiver. The system consists of two components:

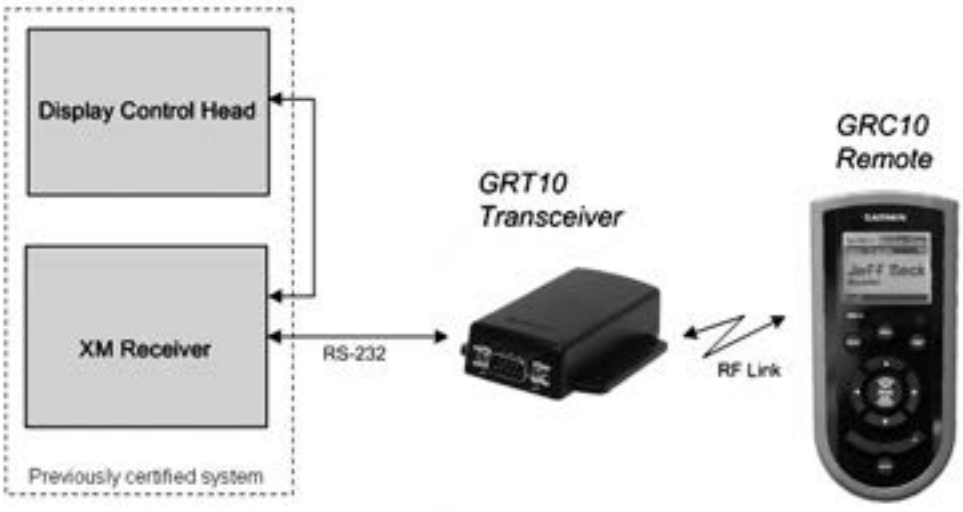
1) The GRT 10 Wireless Transceiver installed in the aircraft adjacent to the GDL 69A and connected to the GDL 69A serial port, and
2) the GRC 10 Wireless Remote with an LCD display.

(b) Removal

1) Disconnect backshell assembly from unit.
2) Use a 3/16” hex drive tool to remove each of the four mounting screws.

(c) Installation

1) Inspect connector for damaged pins.
2) Hold unit flush with the mounting plate.
3) Use a 3/16” hex drive tool to tighten each of the four mounting screws.
4) Connect backshell assembly to unit.



GRT 10 (Optional)
Figure 20

(14) GTS 800 Traffic Advisory System

See "Figure 21".

(a) Description

The optional GTS 800 Traffic Advisory System provides real-time traffic information to the PFD (and, indirectly, to the MFD). The Garmin GTS 800 connects with the #1 GDU 1040 and communicates with the MFD through an HSDB connection. Power is received from the Avionics Bus.

(b) Troubleshooting

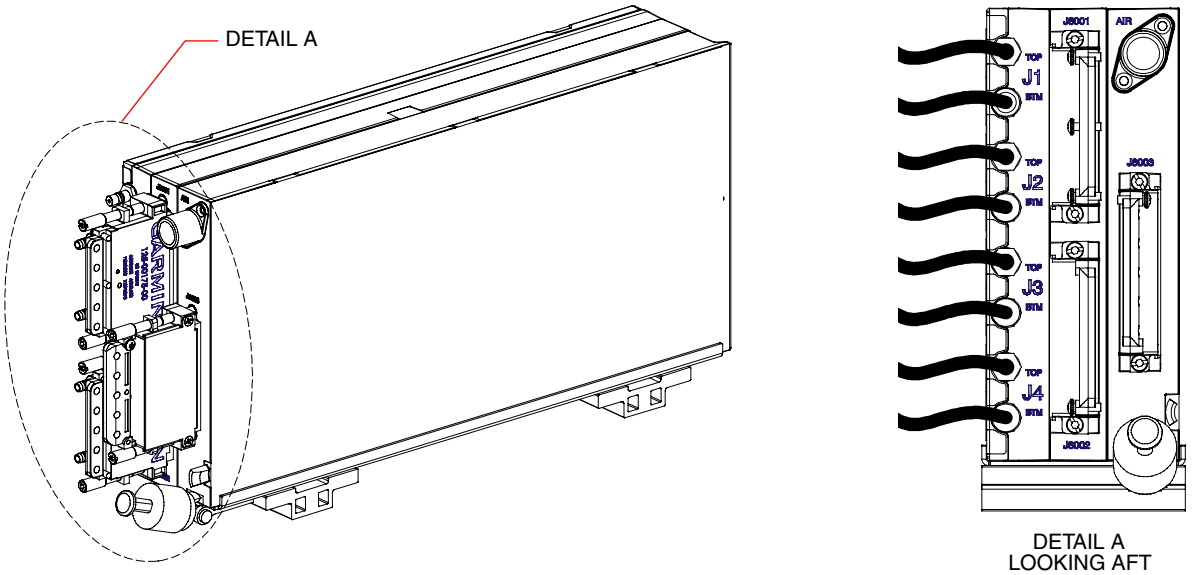
See "Chart 22" on page 34250184.

(c) Removal

1) Remove the aft cabin hatshelf assembly to gain access to the aft avionics shelf.
2) Disconnect the GTS 800 connectors.
3) Loosen the GTS 800 unit by unscrewing the knurled retainer.
4) Move the retainer arms downward. Gently remove the GTS 800 from its rack.

(d) Installation

1) Insert the GTS 800 into the rack and lift each retainer arm to engage lip.
2) Gently screw retainer to secure the GTS 800 unit into place.
3) Visually inspect the connectors and pins for damage. Repair any damage. Connect the connectors to the unit.
4) Reinstall the aft cabin hatshelf assembly.



GTS 800 TAS
Figure 21

(15) Configuration Module

See "Figure 22".

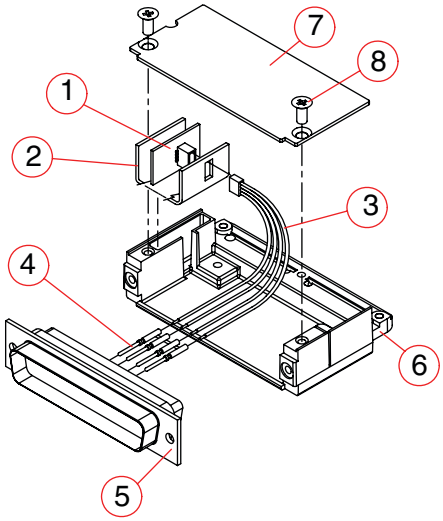
The configuration module is located in the backshell of the connector it is terminated to.

(a) Removal

1) Disconnect connector from LRU.
2) Remove 2 screws (8) from cover (7) and remove cover.
3) Unplug connector from configuration module (1).
4) Remove configuration module.

(b) Installation

1) Inspect connector for damaged pins (4).
2) Place configuration module (1) in position.
3) Insert connector into configuration module (1).
4) Assembly of the connector is the reverse of disassembly.



Configuration Module
Figure 22

NOTE: If the GRS 77 AHRS Configuration Module is replaced:

All three GRS 77/GMU 44 calibration procedures must be performed. Proceed to Post-Installation Setup later in this chapter.

NOTE: If the GDC 74A Configuration Module is replaced:

Configuration settings must be reloaded to the GDC 74A. Proceed to Post-Installation Setup later in this chapter.

NOTE: If the GEA 71 Configuration Module is replaced:

Perform the GEA 71 Test described later in this chapter.

NOTE: If the Master Configuration Module is replaced:

1. Start the G1000 system in Configuration Mode.
2. Go to the Configuration Upload Page on the PFD.
3. Press the UPDT CFG softkey.

NOTE: If both the PFD and Master Configuration Module are replaced:

The G1000 system (except GRS 77/GMU 44 and GDC 74A) must be reconfigured. Proceed to Post-Installation Setup later in this chapter.

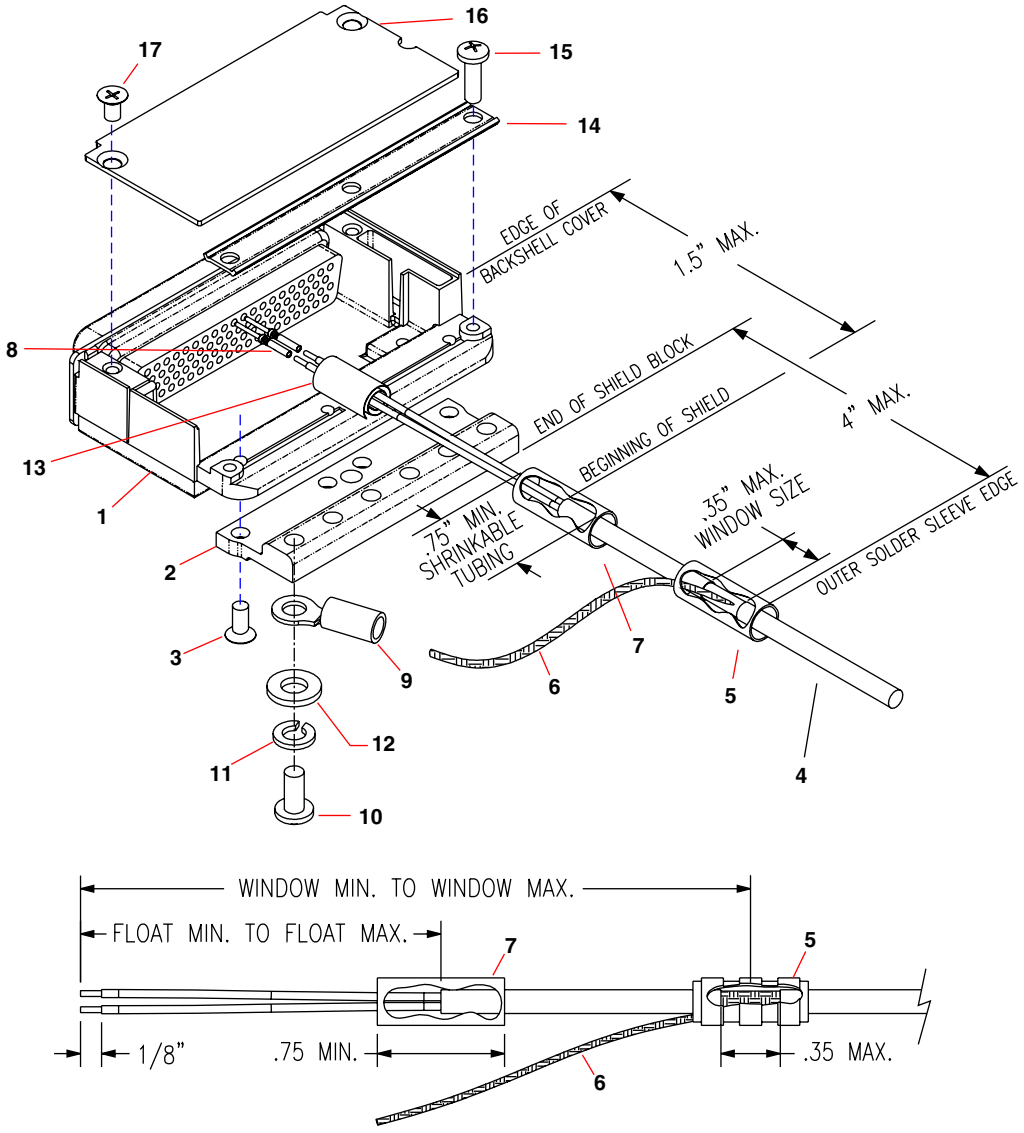
(16) Shield Block Installation

See "Figure 23".

Most G1000 connectors employ a shield block grounding system to provide necessary ground reference to shielding and/or transducers.

(17) Backshell Connectors

See "Figure 24".



1. CAST HOUSING
2. SHIELD BLOCK(S)
3. SCREW
4. MULTIPLE CONDUCTOR SHIELDED CABLE (2–CONDUCTOR DEMONSTRATED HERE)
5. DRAIN WIRE SHIELD TERMINATION (METHOD OPTIONAL)
6. BRAID, FLAT
7. FLOATING SHIELD TERMINATION (METHOD OPTIONAL)
8. PINS
9. RING TERMINAL, #8, INSULATED
10. SCREW
11. SPLIT WASHER
12. FLAT WASHER
13. SILICON FUSION TAPE
14. STRAIN RELIEF
15. SCREW
16. LID
17. SCREW

Shield Block Installation to Backshell
Figure 23

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