NAVWEPS 01-40ALF-2

Handbook
Maintenance Instructions

NAVY MODELS

A-1H • A-1J
AIRCRAFT

SECTION VI
INSTRUMENTS

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SECTION VI

GENERAL INFORMATION

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6–1. INSTRUMENTS.

6–2. DESCRIPTION. Most of the instruments and instrument systems can be classified as either flight instruments or engine instruments, and are grouped accordingly. For convenience, the remaining few instruments are considered as belonging to the category of miscellaneous instruments.

6–2A. The Table of Contents on page 285 preceding this section should be consulted to determine where specific information can be found within the section. The Alphabetical Index at the end of the handbook should also be consulted for determining where specific information is contained within the handbook.

6–3. TROUBLE SHOOTING. To facilitate trouble shooting, the instruments can be classified into four groups according to operating principles: instruments operated from the pitot-static system, instruments operated by pressure, electrically operated instruments, and instruments which are self-contained.

6–4. Trouble occurring in instruments operated from the pitot-static system will generally be one of the following: an obstruction in the air hose or line, a leak around the glass or case of the instrument, a leak in the tubing or connections, moisture in the lines, or bent or broken lines. If any of the foregoing conditions are found and corrected, and the instrument is still defective, the instrument should be removed for overhaul.

6–5. Trouble occurring in instruments operated by pressure will generally be one of the following: an obstruction in the hose or piping, a leak in the lines or connections, foreign substances in the lines, or bent or broken lines. (For trouble occurring in instruments operated by hydraulic pressure, refer also to general principles of hydraulic trouble shooting in section III.) If any of the foregoing conditions are found and corrected, and the instrument is still defective, the instrument should be removed for overhaul.

6–6. Trouble occurring in electrically operated instruments will usually be one of the following: defective fusing, reversed connections to the power supply, loose or broken connections, low or high voltage supply, circuit shorted to ground, or circuit open between power unit and instrument. (Refer also to general principles of electrical trouble shooting in section VII.) If any of the foregoing conditions are found and corrected, and the instrument is still defective, the instrument should be removed for overhaul.

6–7. Troubles common to all instruments are the following: indicator out of calibration, foreign matter inside of dial cover, indicator hands or pointer rubbing on cover glass or dial, pointer stuck, or pointer loose on shaft. If the instrument remains defective after all remedial steps have been taken, the instrument should be removed for overhaul.

6–8. REMOVAL. Since all instrument mountings in the pilot's instrument panel are essentially the same, the following removal procedure is applicable to all instruments.

Note

The standby compass and the master direction indicator are exceptions to the following procedure.

a. Remove two lower screws of instrument mask and slide mask down and out of retaining clip.

b. Remove instrument mounting screws and pull instrument aft out of instrument panel.

CAUTION

Before proceeding with step c, make certain that systems related to instrument are at zero pressure and/or electrical circuit is open. If instrument is air operated, plug hose and cover instrument fitting with dust cap.

c. Disconnect instrument power connection (air or fluid lines and/or electrical connector plug).

6–9. INSTALLATION. The following installation procedure is applicable to all instruments.

Note

The standby compass and the master direction indicator are exceptions to the following procedure.

a. Connect instrument power connection (air or fluid lines and/or electrical connector plug). If instrument contains electrical receptacle, apply anti-seize compound (Specification JAN-A-669) to receptacle threads. If instrument is air operated, remove hose plug and dust cap before making connection.

b. Position instrument in instrument panel from cockpit side of panel. Align screw holes of instrument mask retaining clip with upper mounting screw holes of instrument and install instrument mounting screws.

c. Slide upper edge of instrument mask under mask retaining clip and install two lower screws in mask.

6–10. PILOT'S INSTRUMENT PANEL.

6–11. DESCRIPTION. (See figure 6–1.) The pilot's instrument panel is shock-mounted in the cockpit at fuselage station 100.375. The panel is stabilized in a vertical position by two tie rods which are mounted through grommets in the upper portion of the panel and shock-mounted to brackets on the gun sight support. A removable metal glare shield, extending forward from the top of the instrument panel to the bottom of the windshield, is provided to prevent reflection of the instrument lights from the windshield.

6–12. REMOVAL. (See figure 6–1.)

a. Remove arresting hook control handle from arresting hook control panel.

b. Remove Mark 3 Mod 5 control console from right-hand control panel.

c. Remove glare shield. Refer to section II.

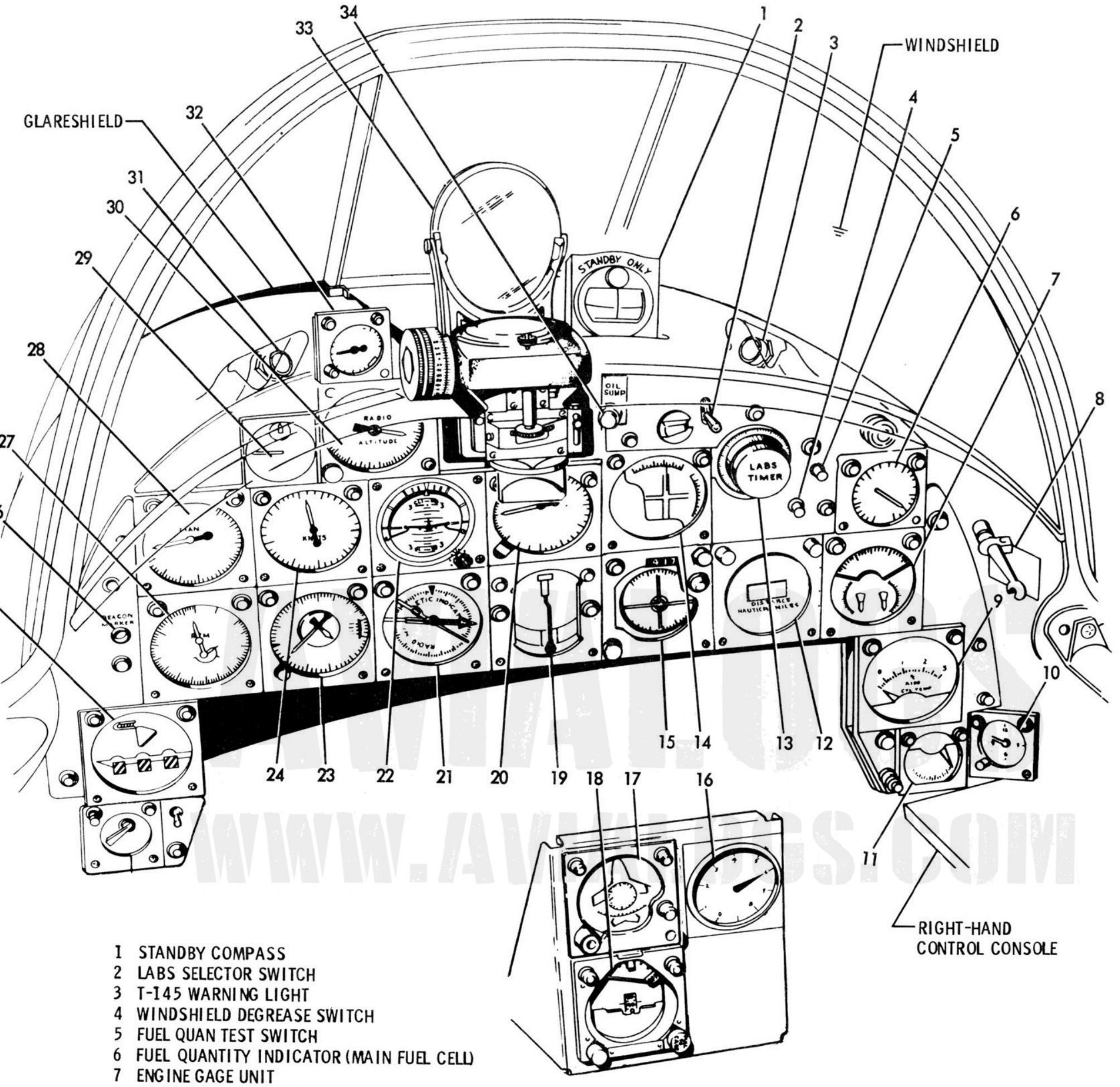
d. Remove nuts from tie rods that stabilize panel in vertical position.

e. Remove mounting bolts at lower portion of panel and swing right-hand side of panel aft.

f. Support panel and disconnect panel wiring at disconnect receptacle at lower right-hand corner of panel. Disconnect and cap fluid and/or air lines to instruments at disconnect receptacles on instruments and remove instrument panel.

CAUTION

Before disconnecting power connections, make certain that systems related to instrument are at zero pressure and/or electrical circuit is open. If instrument is air operated, plug hose and cover instrument fitting with dust cap.

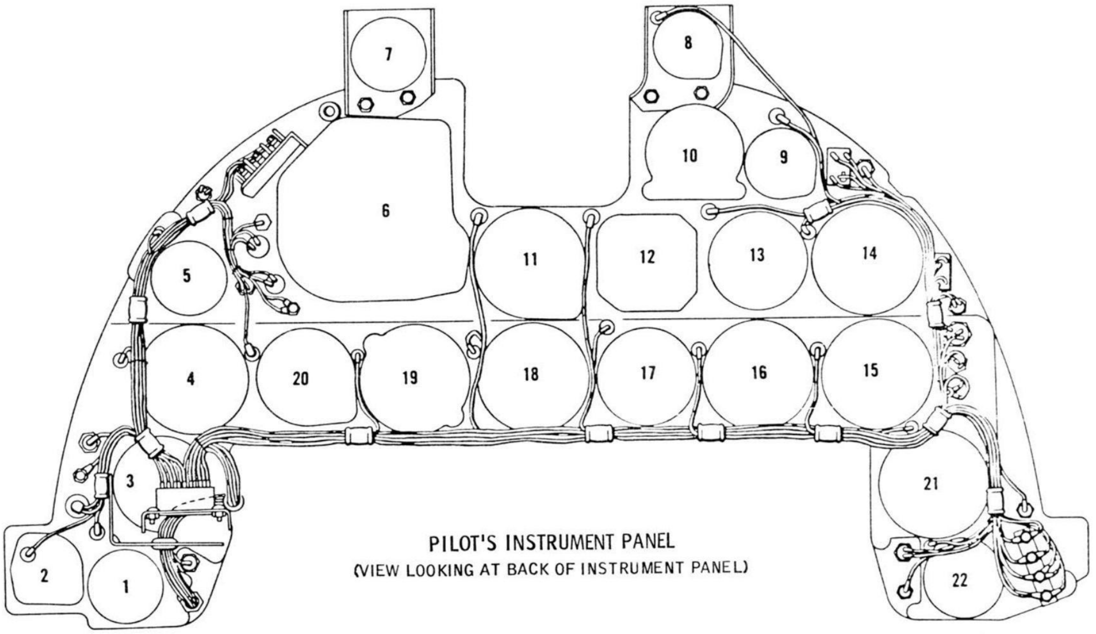


1 STANDBY COMPASS
2 LABS SELECTOR SWITCH
3 T-145 WARNING LIGHT
4 WINDSHIELD DEGREASE SWITCH
5 FUEL QUAN TEST SWITCH
6 FUEL QUANTITY INDICATOR (MAIN FUEL CELL)
7 ENGINE GAGE UNIT
8 ARRESTING GEAR HOOK CONTROL PANEL
9 ENGINE CYLINDER HEAD TEMPERATURE INDICATOR
10 8-DAY CLOCK
11 AIR TEMPERATURE INDICATOR
12 ID-310/ ARN RANGE INDICATOR
13 AERO 18A OR 18C ARMAMENT TIMER
14 AERO 18A OR 18C ARMAMENT INDICATOR
15 ID-249/ ARN COURSE INDICATOR
16 FUEL QUANTITY INDICATOR (EXTERNAL FUEL)
17 MASTER DIRECTION INDICATOR
18 GYRO HORIZON INDICATOR
19 TURN AND BANK INDICATOR
20 RATE OF CLIMB INDICATOR
21 ID-250/ ARN RADIO MAGNETIC INDICATOR
22 VERTICAL GYRO INDICATOR
23 PRESSURE ALTIMETER
24 AIRSPEED INDICATOR
25 LANDING GEAR AND FLAP POSITION INDICATOR
26 MARKER BEACON INDICATING LIGHT
27 TACHOMETER INDICATOR
28 MANIFOLD PRESSURE GAGE
29 TORQUE PRESSURE INDICATOR
30 AN/APN-22 RADIO HEIGHT INDICATOR
31 BOMB DIRECTOR INDICATING LIGHT
32 ACCELEROMETER
33 GUNSIGHT
34 OIL SUMP CONTAMINATION INDICATOR

EFFECTIVITY - BUNO.
FACTORY: NONE
SERV CHG: ALL AIRPLANES REWORKED TO A-1/ASC 635, 656, 695A AND 701

ALF-2-2S6 P-3592-5B

Figure 6-1. Pilot's Instrument Panel (Sheet 1)



1 AIR TEMPERATURE INDICATOR
2 8-DAY CLOCK
3 ENGINE CYLINDER HEAD TEMPERATURE INDICATOR
4 ENGINE GAGE UNIT
5 FUEL QUANTITY INDICATOR (MAIN FUEL CELL)
6 AERO 18A OR 18C ARMAMENT TIMER AND INDICATOR
7 STANDBY COMPASS
8 ACCELEROMETER
9 TORQUE PRESSURE INDICATOR
10 AN/ APN-22 RADIO HEIGHT INDICATOR
11 RATE OF CLIMB INDICATOR
12 VERTICAL GYRO INDICATOR
13 AIRSPEED INDICATOR
14 MANIFOLD PRESSURE GAGE
15 TACHOMETER INDICATOR
16 PRESSURE ALTIMETER
17 ID-250 / ARN RADIO MAGNETIC INDICATOR
18 TURN AND BANK INDICATOR
19 ID-249/ ARN COURSE INDICATOR
20 ID-310/ ARN RANGE INDICATOR
21 WHEEL AND FLAP POSITION INDICATOR
22 INGITION SWITCH

EFFECTIVITY - BUNO
FACTORY: NONE
SERV CHG: ALL AIRPLANES REWORKED TO A-1/ASC 635,656,672 AND 695A

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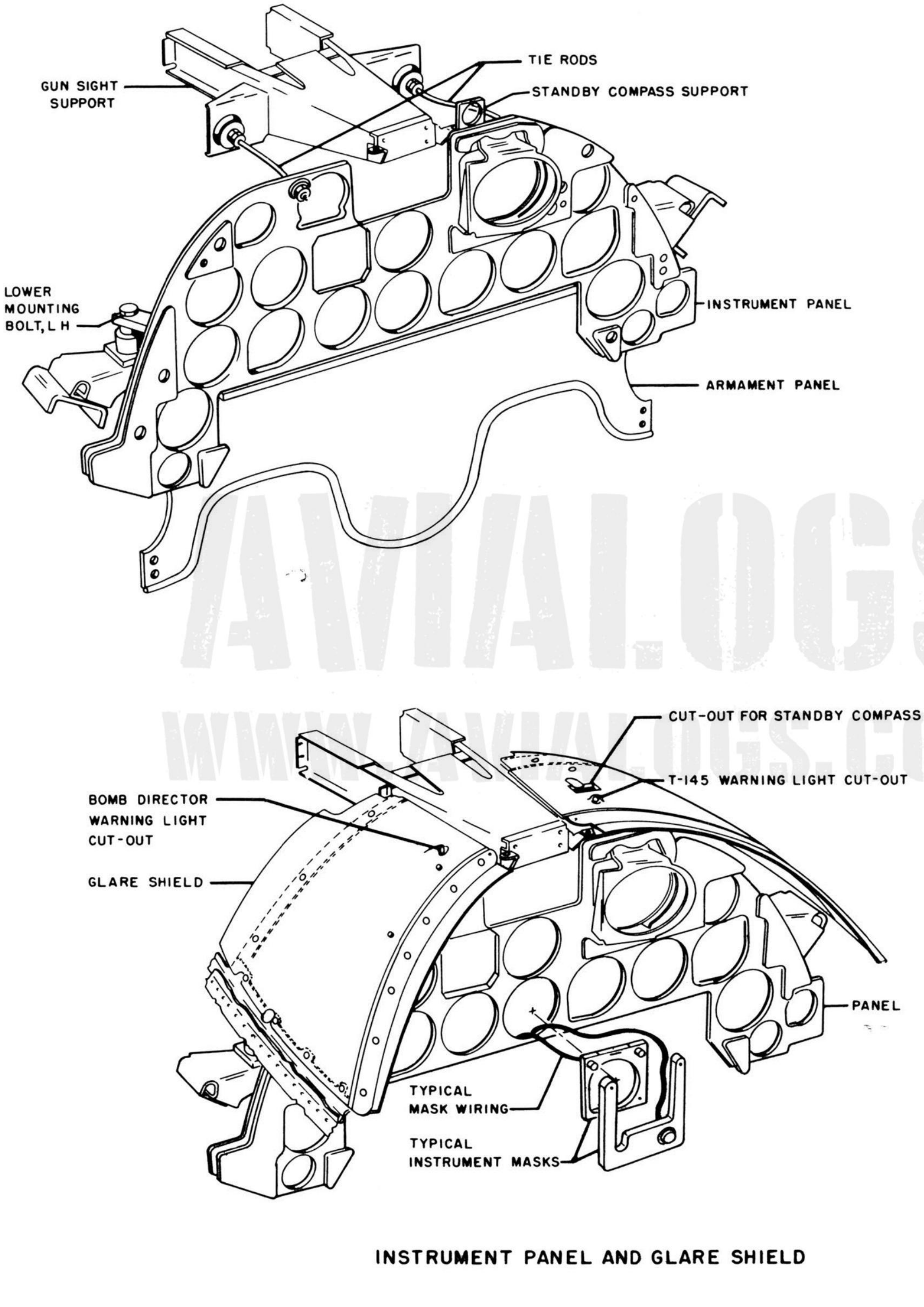
Figure 6-1. Pilot's Instrument Panel (Sheet 2)



INSTRUMENT PANEL AND GLARE SHIELD

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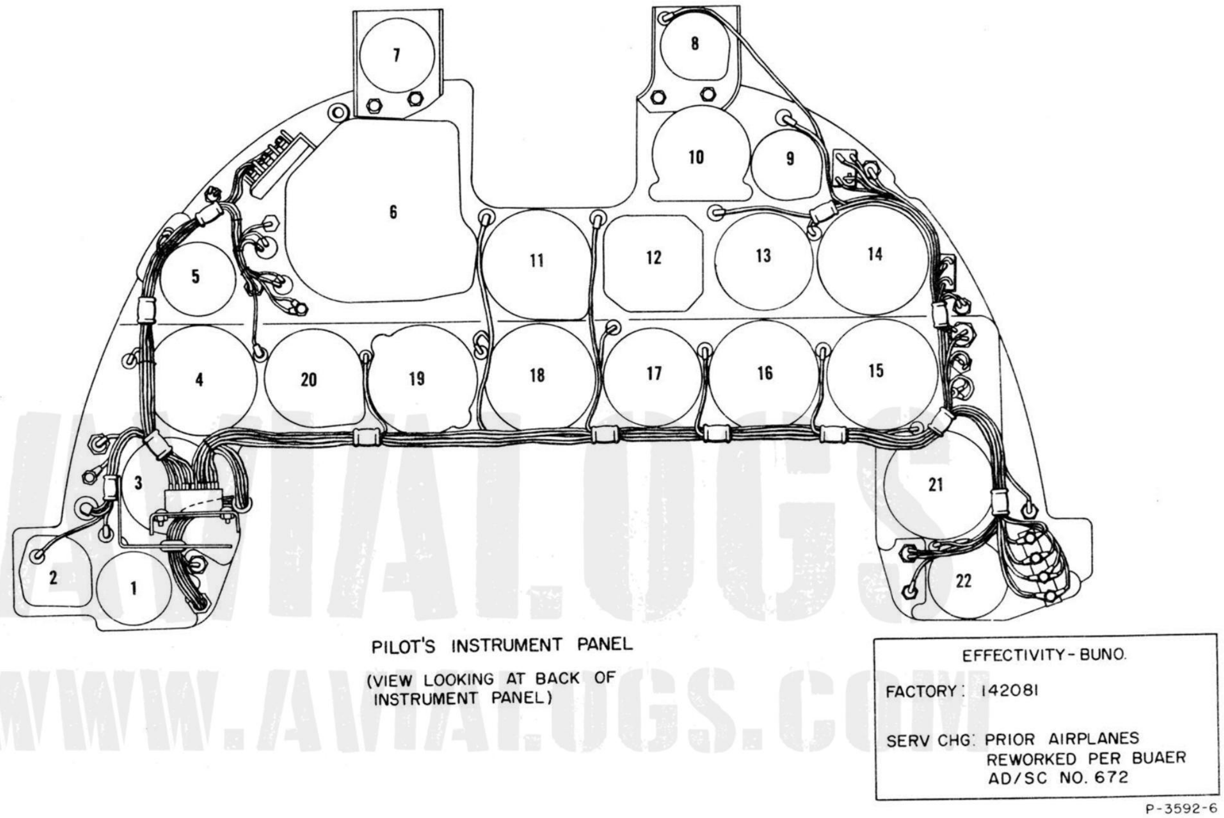
Figure 6-1. Pilot's Instrument Panel (Sheet 3)



1. Air temperature indicator
2. 8-day clock
3. Engine cylinder head temperature indicator
4. Engine gage unit
5. Main fuel-cell fuel quantity indicator
6. Aero 18A armament timer and indicator
7. Stand-by compass
8. Accelerometer
9. Torque pressure indicator
10. AN/APN-22 height indicator
11. Rate of climb indicator
12. Vertical gyro indicator
13. Airspeed indicator
14. Manifold pressure gage
15. Engine tachometer
16. Pressure altimeter
17. ID-250/ARN course indicator
18. Turn and bank indicator
19. ID-249/ARN course indicator
20. External fuel quantity indicator
21. Landing gear and flap position indicator
22. Ignition switch

Figure 6–1. Pilot's Instrument Panel (Sheet 5)

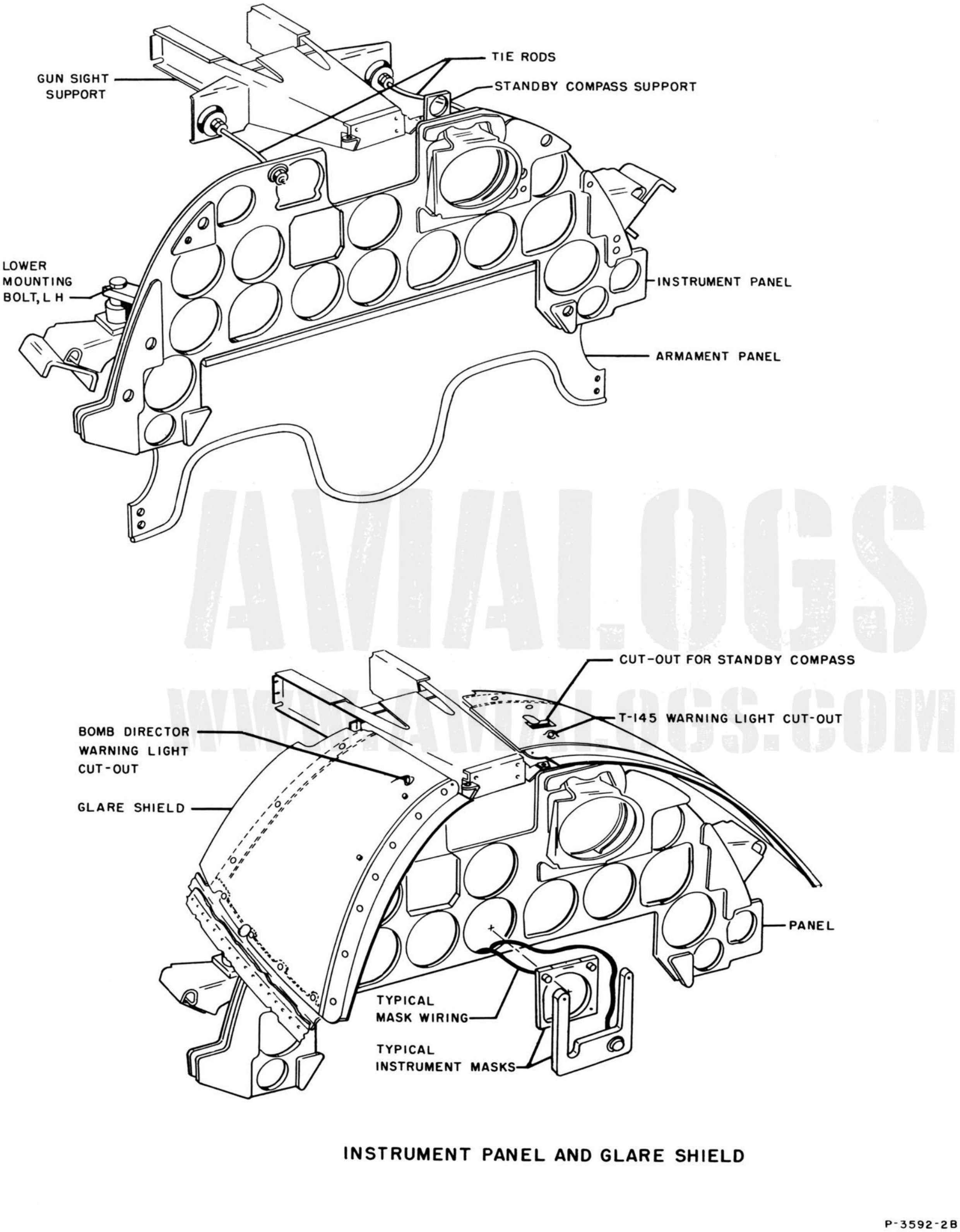


Figure 6–1. Pilot's Instrument Panel (Sheet 6)

6-13. MINOR REPAIR. The vibration insulators on the instrument panel upper and lower mounting brackets can be removed and replaced when the instrument panel is removed.

6-14. INSTALLATION. (See figure 6-1.)

NOTE

The arresting hook control handle must be removed from the arresting hook control panel and the Mark 3 Mod 5 console from the right-hand control panel.

a. With right-hand side of instrument panel swung aft, position left-hand side of panel to support.

b. Support instrument panel and connect panel wiring at disconnect receptacle at lower right-hand corner of panel. Apply antiseize compound (Fed. Spec. TT-A-580) sparingly to receptacle threads. Uncap and connect fluid and/or air lines to applicable instruments at receptacles on instruments.

c. Position right-hand side of instrument panel and install mounting bolts at lower portion of panel.

d. Guide panel to engage tie rods and install retaining nuts. Adjust so that panel face is within one degree of perpendicular to fuselage reference plane.

e. Install arresting hook control handle on arresting hook control panel.

f. Install Mark 3 Mod 5 control panel in right-hand control console.

g. Install glare shield. Refer to section II.

6-15. PILOT'S INSTRUMENT PANEL MASKS.

6-16. DESCRIPTION. (See figure 6-1.) The instruments on the pilot's instrument panel are provided with individual masks for indirect lighting of the instruments by the light assemblies mounted on the instrument panel.

6-17. REMOVAL. Refer to paragraph 6-8.

6-18. INSTALLATION. Refer to paragraph 6-9.

6-19. FLIGHT INSTRUMENTS AND INSTRUMENT SYSTEMS.

6-20. DESCRIPTION. The flight instruments and instrument systems include the following:

Vertical gyro indicating system
Pitot-static system
G-2 compass system
MA-1 compass system
P-1 automatic pilot system
Turn-and-bank indicator system
Turn-and-bank indicator
Landing gear and wing flap
  indicating system
Air temperature indicating system
Accelerometer
Standby compass
Compass correction card
8-day clock
Elapsed time clock
Engine torque pressure indicating system
Magnetic chip detector warning system.

6-20A. VERTICAL GYRO INDICATING SYSTEM.

6-20B. DESCRIPTION. (See figure 6-1A.) The vertical gyro indicating system which is installed on all airplanes is the primary flight attitude indicating system of the airplane. It is comprised of a control assembly which is installed in the radio compartment, an indicator which is located on the instrument panel, and the necessary circuitry. The system utilizes both d-c power and constant frequency a-c power, and is activated by placing the d-c power control switch in BAT & GEN posi-

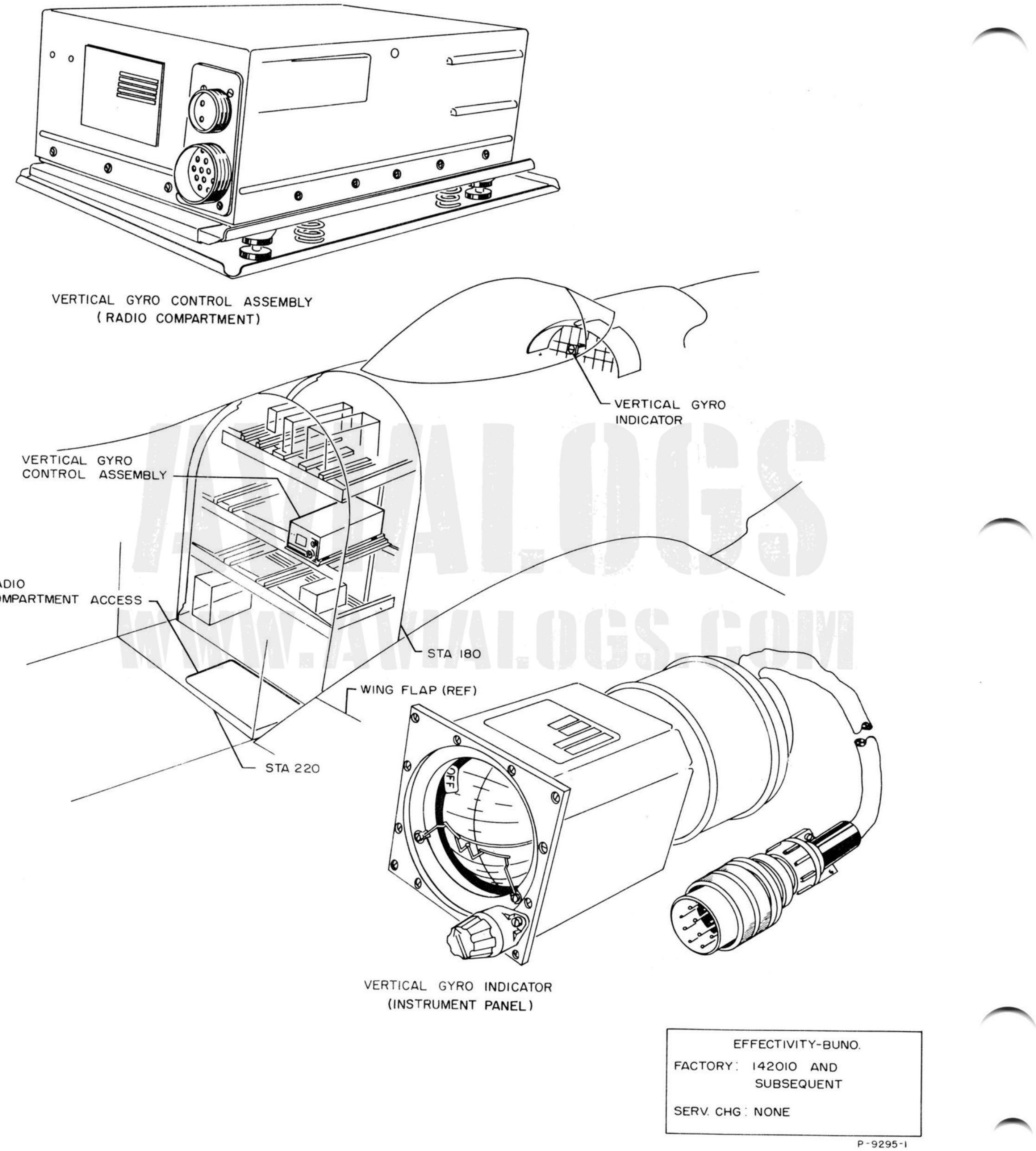


Figure 6–1A. Vertical-Gyro Indicating System

tion. After approximately a two-minute warm-up period, the vertical gyro system is automatically uncaged; the power off flag disappears from the face of the indicator, and the system is ready for use.

6-20C. VERTICAL GYRO CONTROL ASSEMBLY.

6-20D. DESCRIPTION. The vertical gyro control assembly, which is installed on all AD-7 airplanes, is the brain of the vertical gyro indicating system. It includes the vertical gyro, amplifier channels, a rate-of-turn gyro, a pitch-bank erection system, and associated components. A slight change in airplane flight attitude displaces the vertical gyro. This displacement creates electrical signals which are amplified and transmitted to the servos in the indicator. As the maneuver is initiated, the rate-of-turn gyro actuates the pitch-bank erection system, thus assuring precession of the vertical gyro back to its true vertical position. The control assembly is located in the radio compartment on the right-hand side, looking forward, in place of the MA-1 compass amplifier which has been moved aft and to the left side of the compartment.

6-20E. VERTICAL GYRO INDICATOR.

6-20F. DESCRIPTION. The vertical gyro indicator, which is installed in all AD-7 airplanes, is the primary flight attitude instrument of the airplane. It includes servos, an indicating sphere, a power off flag, and an electrical pitch trim knob. The servos receive gyro displacement signals from the control assembly and mechanically transmit these signals to the indicating sphere. The sphere visually depicts the attitude of the earth's horizon as compared to a fixed airplane reference. The power off flag, containing the word "OFF" gives visual indication as to whether or not the system is operating. The electrical pitch trim knob, which is located on the face of the indicator, provides a manual means for electrically trimming the indicator to aircraft attitudes of 20 degrees nose-up and 10 degrees nose-down. The indicator is located in the center of the instrument panel in place of the P-1 automatic pilot gyro horizon which has been moved below the instrument panel to a console installation.

6-20G. VERTICAL GYRO INDICATING CIRCUIT.

6-20H. DESCRIPTION. The vertical gyro indicating circuit includes the following components:

Name	*Location*
Fuse, 3-ampere a-c (2 required)	A-c fuse panel (fwd equip compt—LH side) looking fwd
*Interlock relay	Terminal panel 16 (fwd equip compt—RH)
Circuit breaker	Cockpit circuit breaker panel
Control assembly	Radio compt—RH fwd
Indicator	Instrument panel

*Applies only to airplanes BuNo. 142029 and subsequent.

6-20J. On airplanes BuNo. 142029 and subsequent an interlock relay is incorporated into the vertical gyro indicating circuit to prevent the application of a-c power without d-c power. D-c power is needed to actuate the quick erection timer to decrease the a-c power to the gyro erection motor from 58 volts to 28 volts after the required warm-up period. This prevents the voltage that is needed for the quick erection from exceeding the duty cycle of the a-c operated erection motor. (See figure 10-42B.)

CAUTION

On airplanes BuNo. 142010 through 142028, DO NOT disengage the vertical gyro indicating circuit breaker, "GYRO HORIZ," at any time.

6-21. PITOT-STATIC SYSTEM.

6-22. DESCRIPTION. (See figure 6-2.) The pitot-static system includes the following principal components:

Name	*Para Ref*
Pitot tube	6-25
Static boom	6-29
Static tube	6-30
Airspeed indicator	6-33
Airspeed correction card	6-34
Pressure altimeter	6-35
Rate-of-climb indicator	6-37
Pitot heat control switch	6-40

6-23. Pitot pressure is obtained from the pitot tube; static pressure is obtained from the static boom on the vertical stabilizer. Pitot pressure and static pressure are utilized to operate the airspeed indicator; static pressure only is utilized to operate the altimeter and rate-of-climb indicator.

6-24. DRAINING. Moisture which condenses and collects in the pitot-static system lines can be drained by uncapping the drain tees installed in the lines. After draining, anti-seize compound (Specification JAN-A-669) should be used to coat the threads of the tee before the cap is reinstalled.

6-25. PITOT TUBE.

6-26. DESCRIPTION. (See figure 6-2.) The pitot tube is installed on the underside of the right-hand wing outboard panel, just forward of the wing spar at approximately wing station 250. Ram air pressure enters the pitot tube and is routed to the airspeed indicator on the pilot's instrument panel. The pitot tube is equipped with an electrical heating element for anti-icing.

6-27. REMOVAL. (See figure 6-2.)

a. Through access panels on underside of right-hand wing outboard panel, forward and aft of front spar at approximately wing station 235, remove pitot pressure line clamp.

b. Remove pitot tube mounting screws.

c. Pull pitot tube down and disconnect pitot pressure line.

d. Disconnect pitot tube electrical plug and remove pitot tube.

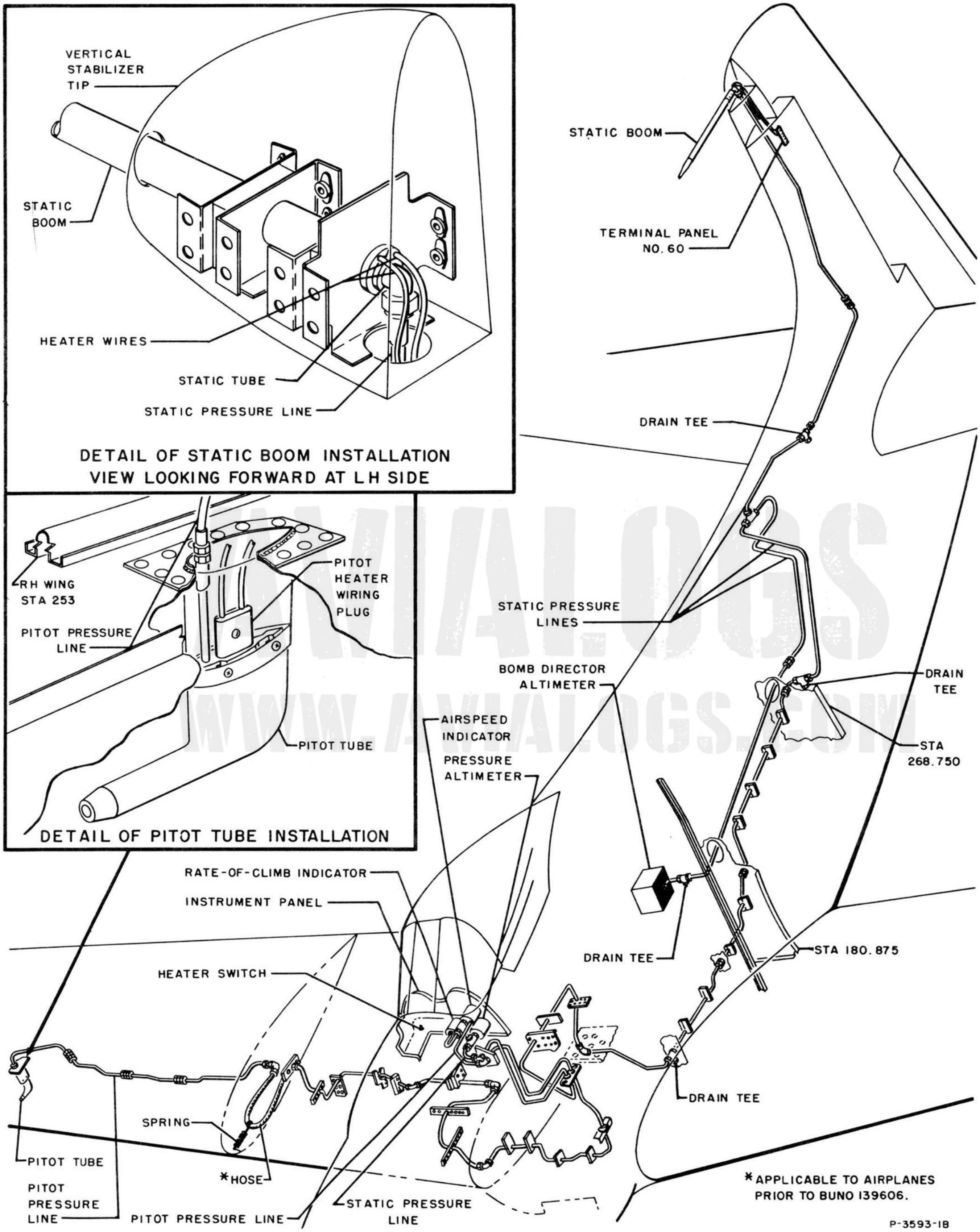


Figure 6–2. Pitot-Static System (Sheet 1)

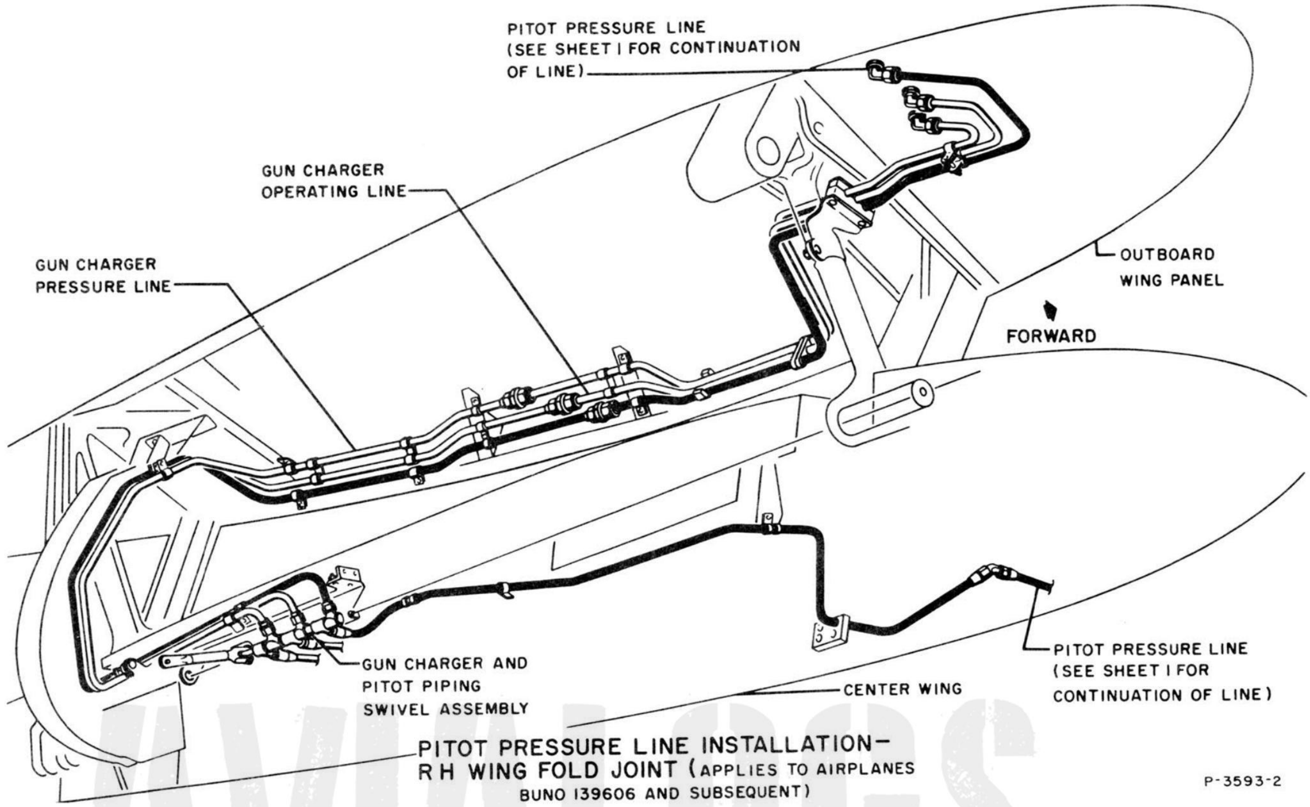


Figure 6–2. Pitot-Static System (Sheet 2)

6–28. INSTALLATION. (See figure 6–2.)

a. Apply anti-seize compound (Specification JAN-A-669) to threads of line fittings and electrical receptacle on pitot tube.

b. Connect electrical plug and pitot pressure line to pitot tube.

c. Place tube on support and install mounting screws.

d. Through access panels, install pitot pressure line clamp.

6–29. STATIC BOOM.

6–30. DESCRIPTION. (See figure 6–2.) The static boom, which projects forward from the leading edge of the vertical stabilizer just above vertical station 111.187, contains the static tube of the pitot-static system. The static tube furnishes static pressure to the airspeed indicator, the pressure altimeter, the rate-of-climb indicator, and to the bomb director altimeter. The static tube is equipped with an electrical heating element for anti-icing.

6–31. REMOVAL. (See figure 6–2.)

a. Remove vertical stabilizer tip by removing tip attaching screws, and carefully slide tip off static boom.

b. Disconnect static pressure line from static tube.

c. Disconnect static tube electrical connections.

d. Remove static boom attaching screw and bolt and remove boom from mounting support.

e. To remove static tube from static boom, first remove screws attaching static boom tip to boom, then remove screws placed along static boom and pull static tube and capped pitot tube from boom.

Note

The pitot tube in the static boom is not used in the system, and is therefore capped.

6–32. INSTALLATION. (See figure 6–2.)

a. To install static tube in static boom, first place static tube and capped pitot tube in static boom and install screws along static boom, then install static boom tip on boom.

b. Apply anti-seize compound (Specification JAN-A-669) to threads of static boom line fittings.

c. Connect static pressure line to static tube.

d. Connect static tube electrical wiring.

e. Position static boom to mounting supports and install attaching screw and bolt.

f. Carefully slide vertical stabilizer tip on static boom, align tip with stabilizer and install tip attaching screws.

g. Seal around boom at vertical stabilizer tip with MIL-S-7502 Class B Type II sealing compound.

6–33. AIRSPEED INDICATOR.

6–34. DESCRIPTION. (See figures 6–1 and 6–2.) The airspeed indicator is mounted in the left-hand portion of the pilot's instrument panel and is connected to the pitot pressure line and to the static pressure line of the pitot-static system. The indicator dial is calibrated in knots and indicates an airspeed range of 40 to 400 knots. An airspeed correction card is carried in a retainer just below the cockpit right-hand rail.

6–35. PRESSURE ALTIMETER.

6–36. DESCRIPTION. (See figures 6–1 and 6–2.) The pressure altimeter is mounted in the left-hand portion of the pilot's instrument panel and is connected to the static pressure line of the pitot-static system. The instrument indicates the altitude of the airplane in relation to sea level or in relation to a pre-selected reference point. There are three pointers on the indicator dial to indicate altitude in units of 100 feet, 1000 feet, and 10,000 feet, through a range of 0 to 50,000 feet.

6–37. RATE-OF-CLIMB INDICATOR.

6–38. DESCRIPTION. (See figures 6–1 and 6–2.) The rate-of-climb indicator is mounted near the center of the pilot's instrument panel and is connected to the static pressure line of the pitot-static system. The instrument indicates the rate of change in airplane altitude through a range of 0 to 6,000 feet per minute.

6–39. PITOT-STATIC HEATER CIRCUIT.

6–40. DESCRIPTION. The pitot-static heater circuit supplies power to the heating elements of the pitot tube in the right-hand wing outboard panel and to the static tube in the vertical stabilizer static boom. The circuit receives power from the secondary bus through a 15-ampere circuit breaker, designated OIL DILUTE and PITOT HEAT. The heater control switch is mounted in the cockpit right-hand control panel.

6–41. G-2 COMPASS SYSTEM.

6–42. DESCRIPTION. (See figure 6–3.) The G-2 compass system, installed on airplanes BuNo. 134466 through 134637, 135223 through 135406 and 137492 through 137632, includes the following principal components:

Name	*Para Ref*
Master direction-indicator	6–46
Remote compass transmitter	6–50
Compass amplifier	6–52
Compass adapter	6–55
Compass control switch	6–57
ID-250/ARN course indicator	6–58

6–43. The G-2 compass system operates on the combined principles of the gyroscope and the magnetic compass to eliminate the short-period oscillation errors of the compass and the relatively long-period drift errors of the gyro. The system thus provides a stable indication of azimuth heading of the airplane. The gyro element is controlled by the compass element, which acts as the primary direction detector of the system. Both stabilized (compensated) and unstabilized (uncompensated) compass readings are relayed to the master direction-indicator. A voltage equivalent of the stabilized average indication of the airplane heading is relayed from the master direction-indicator to the stator of a heading synchro in the ID-250/ARN course indicator. The synchro responds by developing a torque to turn its rotor to align with the heading and, in so doing, positions the circular scale of the indicator to denote the magnetic heading of the airplane. For further information regarding the ID-250/ARN course indicator, refer to paragraph 6–58V.

6–44. An automatic pilot system pick-off is provided in the master direction-indicator to furnish the automatic pilot system, when engaged, with a stabilized directional signal.

6–45. ADJUSTMENT (COMPENSATION). The compass system must be compensated after reinstallation or replacement of the remote compass transmitter. If the master direction-indicator is replaced, the airplane must be reswung and the compass system compensated, if necessary.

Note

Steps a, b, and c must be observed before step d is attempted.

a. Make certain that personnel engaged in compass compensation operations remove from their persons all ferrous items, tools, pocket knives, badges, mechanical pencils and other magnetic articles which might affect compass readings. Use only non-magnetic screw driver to adjust compensating magnets.

b. Remove from area all equipment constituting magnetic disturbances, such as automobiles, trucks, tractors, battery carts, fire extinguisher carts, towing bars, and other airplanes.

c. Secure all magnetic equipment in airplane in positions occupied in normal flight. Place all control levers in normal positions.

d. Place airplane in three-point attitude on compass rose, with vertical center line through hub of right-hand main landing gear as near center of turntable as possible.

Note

Care must be taken to keep the airplane from moving off the desired heading while readings are being taken.

e. Align remote compass transmitter so that mounting screws are in center of adjusting slots, and tighten screws.

f. Start engine and, after standard warm-up, operate engine at sufficient speed for generator to maintain normal operating voltage for electrical system.

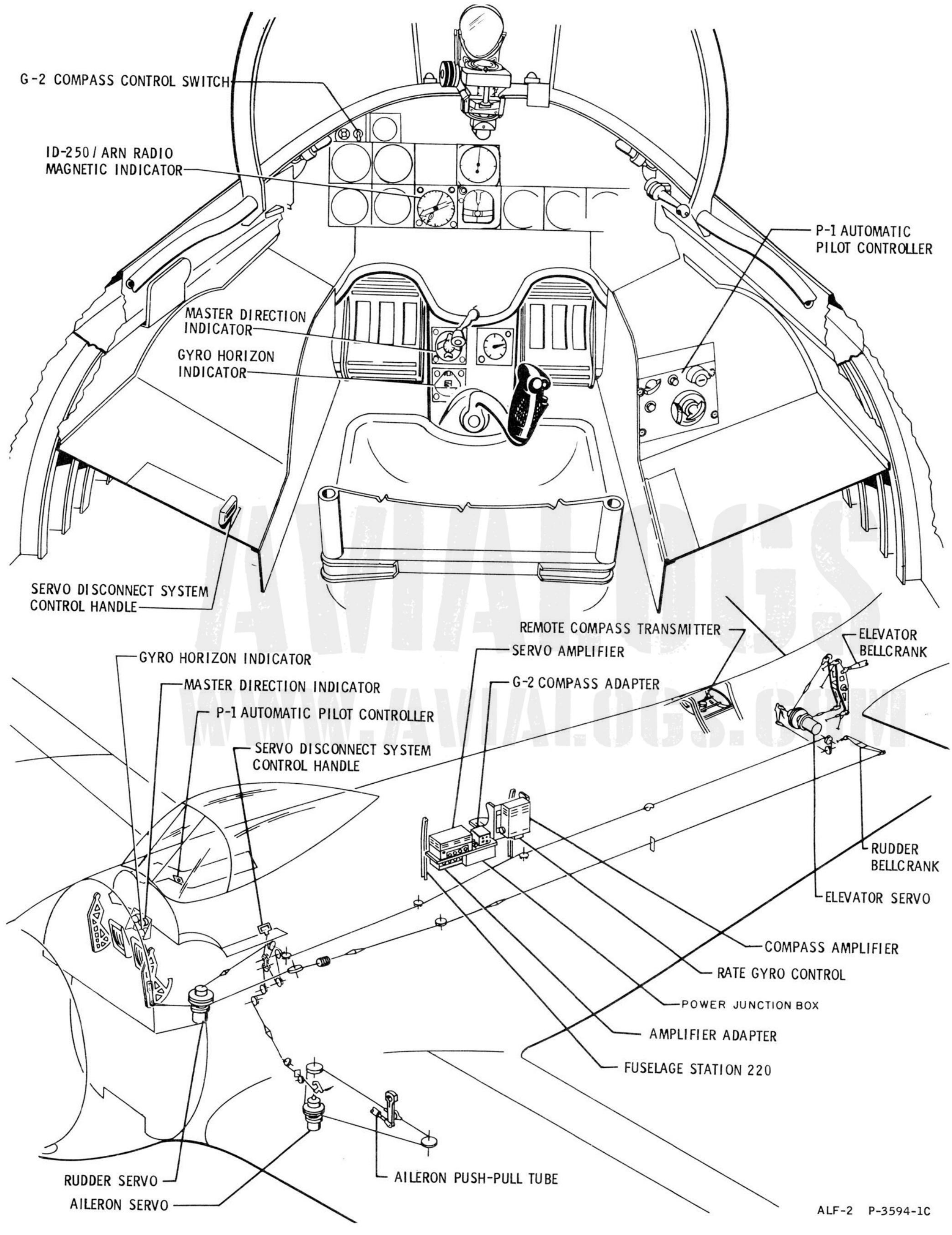


Figure 6-3. G-2 Compass and P-1 Automatic Pilot System

g. Operate following equipment during entire period of compass compensation:

Gunsight lamp	Fuel quantity indicator
Radar altimeter	Battery (switch ON)
Search radar (if installed)	

h. Turn airplane to south heading. Place compass control switch in COMPASS CONTROL and set N-S and E-W compensators on compass transmitter to null positions. Wait five minutes for gyro to attain speed.

i. Deleted.

j. Move master direction indicator outer dial 10 to 20 degrees out of agreement with indication of inner (magnetic) dial. (Caging knob should be held in at least five seconds to allow caging to occur.) Observe rate at which outer dial moves toward inner dial indication: rate should be between 2 and 6 degrees per minute.

k. Repeat step h with compass control switch in FREE D.G.: master direction indicator outer dial should not move toward synchronization with inner dial (except for negligible drift).

CAUTION

Failure of foregoing tests indicates faulty control switch, wiring, or other component of system. Compensation should not be continued until fault has been corrected.

l. Place compass control switch in COMPASS CONTROL. Synchronize inner and outer dials of master direction indicator. Wait for outer dial to stabilize before taking reading. (Stabilized headings change less than one degree per minute; however, inner dial and outer dial indications may differ as much as eight degrees.)

m. Record stabilized readings on compensation chart. Refer to table 6-1.

n. Turn airplane to west, north, and east headings successively, and in each case wait one minute after dials synchronize before recording readings. Determine difference in degrees between uncompensated readings and accurate magnetic heading, i.e., compass rose degree headings 180, 270, 000, and 090. Assign algebraic signs to readings as follows: if uncorrected dial reading is greater than accurate magnetic heading, deviation is minus; if dial reading is less than accurate magnetic heading, deviation is plus.

NOTE

Allowable spread between maximum positive and maximum negative deviation should be determined for compass rose being used. If spread falls within allowable tolerance limits, the system is acceptable to this point and the compensation procedure may be continued. If spread exceeds allowable limits, the compass is being influenced by local magnetic material which must be removed before compensation operation is continued.

TABLE 6-1. G-2 COMPASS COMPENSATION—SAMPLE WORK SHEET

True Airplane Headings (On Compass Rose)	Uncompensated Readings (Outer Dial)	Deviations
SOUTH 180	175-1/2	+4-1/2
WEST 270	276	-6
NORTH 000	0006-1/2	-6-1/2
EAST 090	090	0

$$\text{N-S adjustment} = \frac{\text{N-S}}{2} = \frac{(-6\text{-}1/2) - (+4\text{-}1/2)}{2} = \frac{-11}{2} = -5\text{-}1/2$$

Dial reading should be changed from 006-1/2 to 001 (5-1/2 degrees lower).

$$\text{E-W adjustment} = \frac{\text{E-W}}{2} = \frac{(0) - (-6)}{2} = \frac{+6}{2} = +3$$

Dial reading should be changed from 090 to 093 (3 degrees higher).

$$\text{Realignment adjustment} = \frac{N + E + S + W}{4} = \frac{(-6\text{-}1/2) + (0) + (+4\text{-}1/2) + (-6)}{4} = \frac{-8}{4} = -2$$

Remote compass transmitter housing must be rotated clockwise until reading on outer dial is 091 (2 degrees less than 093).

o. Determine adjustment of N-S compensator as follows:

$$\frac{(\text{North deviation} - \text{South deviation})}{2} = \text{N-S adjustment}$$

p. Place airplane on north heading and adjust N-S compensating screw at remote transmitter until outer dial has changed amount determined in step n. If N-S adjustment is minus, rotate screw clockwise; if adjustment is plus, rotate screw counterclockwise. Allow sufficient time for outer dial to stabilize, and wait one minute after stabilized heading has been obtained before taking reading.

q. Place airplane on east heading and make similar adjustment of E-W compensating screw at remote transmitter. Determine E-W adjustment as follows:

$$\frac{(\text{East deviation} - \text{West deviation})}{2} = \text{E-W adjustment}$$

r. After N-S and E-W adjustments have been completed, determine overall realignment adjustment by averaging all four deviations as follows:

$$\frac{(N + E + S + W)}{4} = \text{realignment adjustment}$$

s. Make no adjustment if answer is less than plus or minus one degree. If answer is not within tolerance, loosen transmitter mounting bolts and turn housing in adjustment slots clockwise if answer is plus, counterclockwise if answer is minus. Allow sufficient time for dial to stabilize on new heading before concluding adjustment. Tighten mounting screws.

t. Check adjustments by swinging airplane through complete circle and recording compass readings at intervals of 15 degrees: no deviation at any heading should exceed four degrees; deviation at master direction-indicator headings of 000, 090, 180, and 270 degrees must not exceed one degree.

u. Correct compensation for compass deviations current in locality in which compensation is made.

6-46. MASTER DIRECTION INDICATOR.

6-47. DESCRIPTION. (See figures 6-1 and 6-3.) The master direction indicator is mounted on an extension to the pilot's instrument panel located on the centerline of the cockpit just below the armament panel. All components of the G-2 compass system are encased in the master direction indicator except the remote compass transmitter, the compass amplifier and adapter, and the ID-250/ARN radio magnetic indicator. The master direction indicator consists essentially of a gyroscopic mechanism mounted in bearings and geared to the main dial of the indicator and to various pick-off coils, plus control coils and a resetting mechanism. The magnetic heading of the airplane is detected by the remote compass transmitter, which is electrically connected through the master direction indicator detection circuit to the center dial or correspondence indicator of the master direction indicator. The correspondence indicator is therefore always in agreement with the remote compass transmitter. The detector signal from the remote compass transmitter is also transmitted to the compass amplifier where the signal is modified and sent to the torque motor of the master direction indicator. By means of the amplified detector signal from the compass amplifier, the torque motor precesses the gyro element of the master direction indicator until agreement occurs between the master direction indicator and the remote compass transmitter. The precession rate of the gyro element is approximately four degrees per minute, with the result that the gyro does not instantly follow every variation in the remote compass transmitter detector signal, supplying instead a stabilized average indication of the airplane heading on the master direction indicator outer dial. Indication from the master direction indicator outer dial is relayed to the compass section of the ID-250/ARN radio magnetic indicator.

6-48. The master direction indicator is constructed in two sections. The front section contains the various coils (detector and automatic pilot pick-off), dials, vanes and resetting mechanism. The rear section contains the universally mounted gyro element and the precessing motor. Mechanical stops restrict the relative motion of the motor gimbal to approximately 88 degrees in either direction from the normal position, so that when the limit of freedom in the gimbal is reached, the stop causes a tumbling action of the gyro element producing the same indication but rotating the motor gimbal around the stop in the opposite direction. The tumbling action of the gyro allows the gyro to be used during any extreme maneuvers of the airplane.

6-49. The resetting mechanism of the master direction indicator is controlled by a knob identified as PUSH TO RESET COMPASS. Pressing the resetting knob cages the gyro and, at the same time, actuates a clutch over-ride switch which energizes the coil of the clutch relay to disengage the clutch from the automatic-pilot servo units. Upon release of the resetting knob, the gyro is uncaged at its new setting and the clutch over-ride switch is opened, de-energizing the coil of the relay to re-engage the automatic pilot.

CAUTION

Internal components of the master direction indicator are subject to internal damages and misalignment during transportation or handling due to uncontrolled movement of their mass components.

6-49A. To prevent damage from occurring to the master direction indicator the following detailed instructions as outlined in Avionics Bulletin 50 dated 19 June 1964 must be adhered to:

a. The master direction indicator must be locked when prepared for shipment or removed from the airplane. Care to be exercised when unlocking during installation.

b. Personnel performing repair, inspection or assembly on master direction indicator must lock indicator prior to release for shipment. Operational personnel must return any indicator received in an unlocked position to the cognizant repair activity from which it was received, with necessary FUR information citing Electronic Bulletin 50 as authority.

6-49B. REMOVAL. Refer to paragraph 6-8.

CAUTION

Handle indicator with extreme care. Keep instrument away from equipment and materials which might magnetically influence instrument. Always place instrument, when removed from mountings, on rubber or felt pad.

6-49C. INSTALLATION. Refer to paragraph 6-9.

6-50. REMOTE COMPASS TRANSMITTER.

6-51. DESCRIPTION. (See figure 6-3.) The remote compass transmitter is located on the center line of the airplane between fuselage stations 291.250 and 308. The transmitter detects the magnetic heading of the airplane and electrically transmits the heading to the master direction-indicator. The transmitter magnetic element, which consists essentially of a pair of bar magnets fastened to a liquid-tight float submerged in compass liquid

in a liquid-tight chamber, aligns itself with the horizontal component of the earth's magnetic field. The local field produced by the magnetic element influences the flux pattern in the core of the transmitter coil, generating unbalanced voltages in the coil. These voltages are transmitted to the master direction-indicator detector coil and correspondence-indicator coil to furnish visual indications of magnetic headings of the airplane.

6–52. COMPASS AMPLIFIER.

6–53. DESCRIPTION. (See figure 6–3.) The compass amplifier is mounted on a support installation just aft of fuselage station 240, on the right-hand side of the airplane. The compass amplifier functions as a junction box and power source for other various circuit components, and as an amplifier and rectifier for the detector signal from the master direction-indicator. The detector signal varies proportionately with the degree of misalignment of the gyro element with the remote compass transmitter and is transformed to a differential d-c signal which is capable of operating the torque motor of the gyro. This action causes the gyro to align accurately with the average compass transmitter heading.

6–54. The amplifier supplies 26-volt, 400-cycle power to the remote compass transmitter, and to the detector coil and correspondence-indicator coil of the master direction-indicator.

6–55. G-2 COMPASS ADAPTER.

6–56. DESCRIPTION. (See figure 6–3.) The compass adapter is mounted on a support installation between fuselage stations 233.312 and 240 on the right-hand side of the airplane. The compass adapter is employed to connect the G-2 compass system to the P-1 automatic pilot system to provide the automatic pilot system with a stable directional reference.

6–57. G-2 COMPASS CONTROL SWITCH.

6–58. DESCRIPTION. (See figure 6–3.) The compass control switch is mounted on the pilot's instrument panel. The switch is designated G-2 COMPASS, and has two positions, "COMPASS CONTROL" and "FREE D.G." When the switch is in "COMPASS CONTROL," the outer dial of the master direction-indicator is enslaved to the inner correspondence of the indicator, indicating an average heading on the outer dial. When the switch is in "FREE D.G.," the gyro in the master direction-indicator is released from all influence of the remote compass transmitter and, therefore, acts as a free directional gyro. The free gyro directional indication from the master direction-indicator outer dial is relayed to the compass section of the ID-250/ARN course indicator.

6–58A. MA-1 COMPASS SYSTEM.

6–58B. DESCRIPTION. (See figure 6–3A.) The MA-1 compass system, installed on airplanes BuNo. 139606 and subsequent, includes the following principal components:

Name	*Para Ref*
Transformer	6–58E
Compass transmitter	6–58G
Directional gyro	6–58L
Amplifier	6–58N
Compass controller	6–58R
Compass automatic pilot adapter	6–58T
ID-250/ARN course indicator	6–58V

6–58C. The MA-1 compass system combines the functions of both a directional gyro and a magnetic compass to provide an accurate, stabilized indication of the aircraft heading through 360 degrees in azimuth. The compass transmitter is the direction-sensing component of the system. It detects the direction of the horizontal lines of force of the earth's magnetic field and transmits this information electrically to the amplifier. The amplifier, in turn, determines the degree of misalignment existing between the transmitter and gyro heading and transmits a voltage proportional to the degree of misalignment to a slaving torque motor in the directional gyro. The torque motor then precesses the gyro horizontally until it is aligned with the heading determined by the transmitter. Thus the gyro is slaved to the magnetic meridian of the earth and, as the airplane turns, the case rotates about the gyro. This angular rotation displaces the stator of the gyro heading synchro in relation to its rotor and causes a voltage to be induced which is transmitted through the amplifier to the stator of a synchro in the ID-250/ARN course indicator. The synchro responds by developing a torque which turns its rotor to align electrically with the rotor of the gyro heading synchro and, in so doing, positions the circular scale of the indicator to denote the magnetic heading of the airplane. When the system is operated as a free-gyro without the magnetic sensing characteristics, the indicator denotes the heading of the gyro only. For further information regarding the ID-250/ARN indicator, refer to paragraph 6-58V.

6–58D. The MA-1 compass system is also utilized to provide stabilized heading information for the automatic pilot system. This information is supplied electrically from the amplifier through the compass automatic pilot adapter.

6–58E. The MA-1 compass system receives power from the 28-volt primary d-c bus and from the 115-volt, constant-frequency, 400-cycle a-c power supply through a transformer which increases the voltage output to 200 volts.

6–58F. For additional information pertaining to the MA-1 compass system, refer to Handbook of Operation and Service Instructions for MA-1 Compass-Controlled Directional-Gyro System, NAVAER 05-15C-501.

6–58G. COMPASS TRANSMITTER.

6–58H. DESCRIPTION. The compass transmitter is the magnetic direction-sensing component of the MA-1 compass system. It is installed in a supporting bracket which is located on the center line of the airplane between

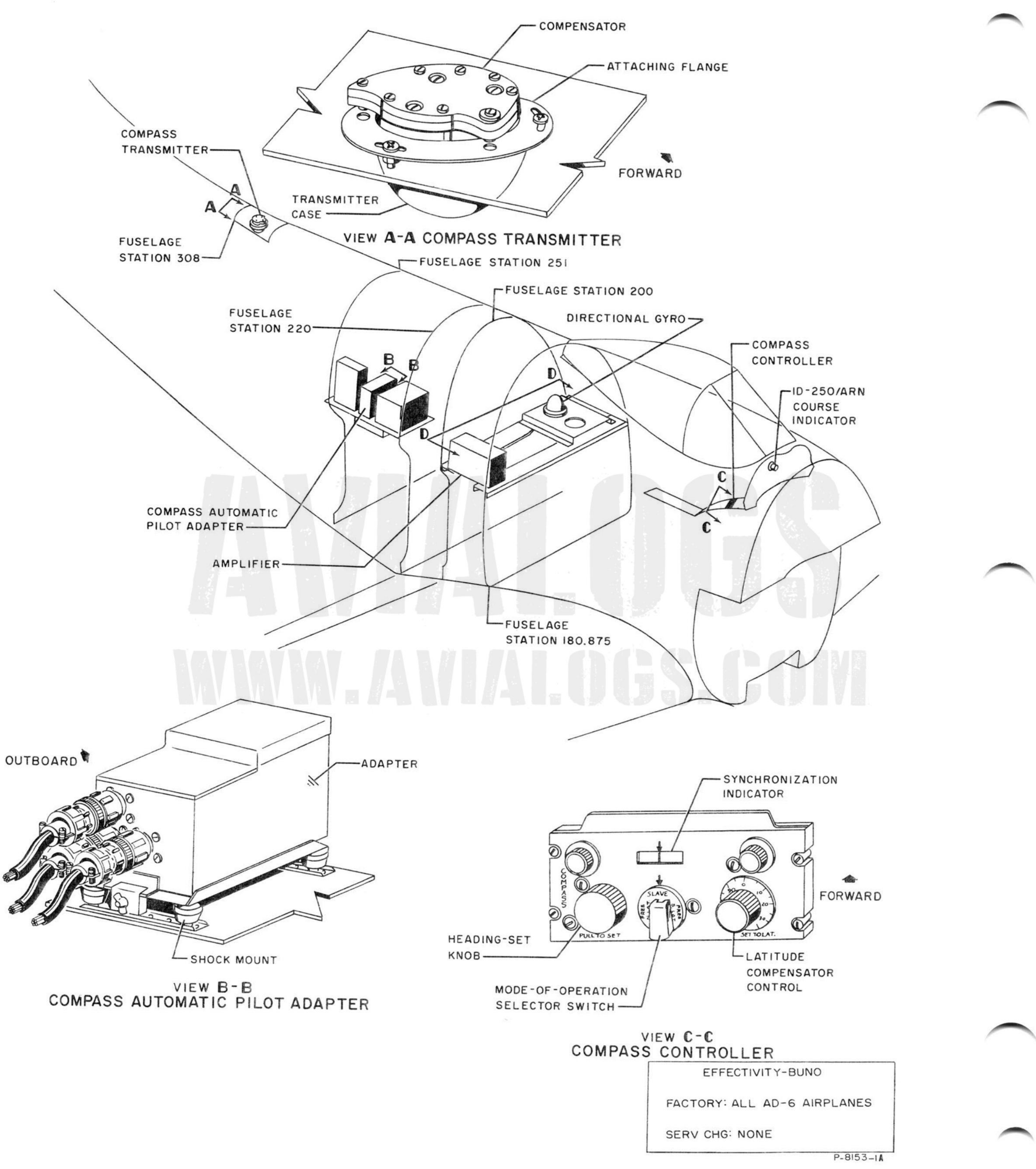
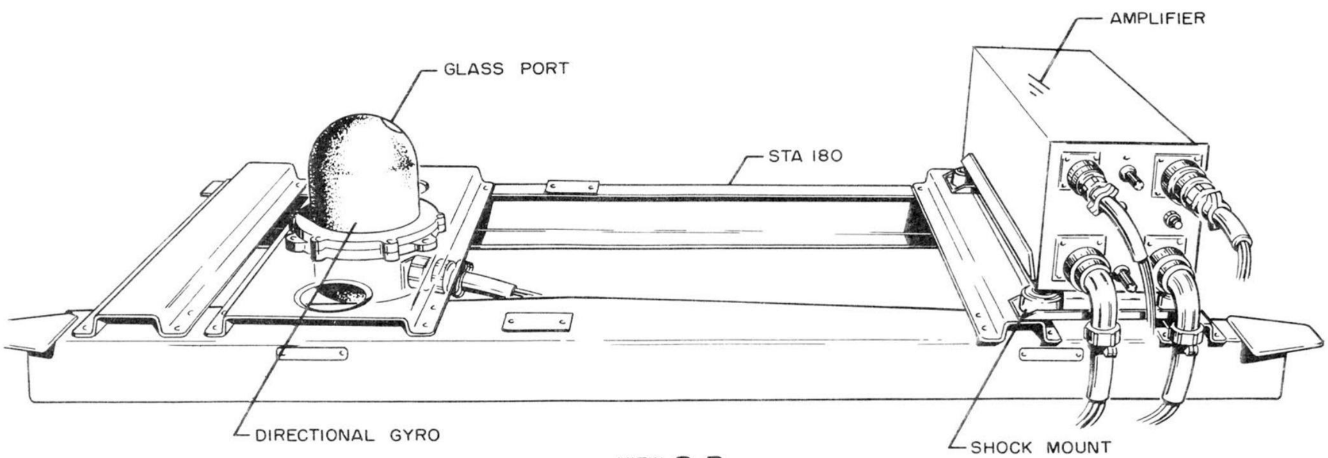
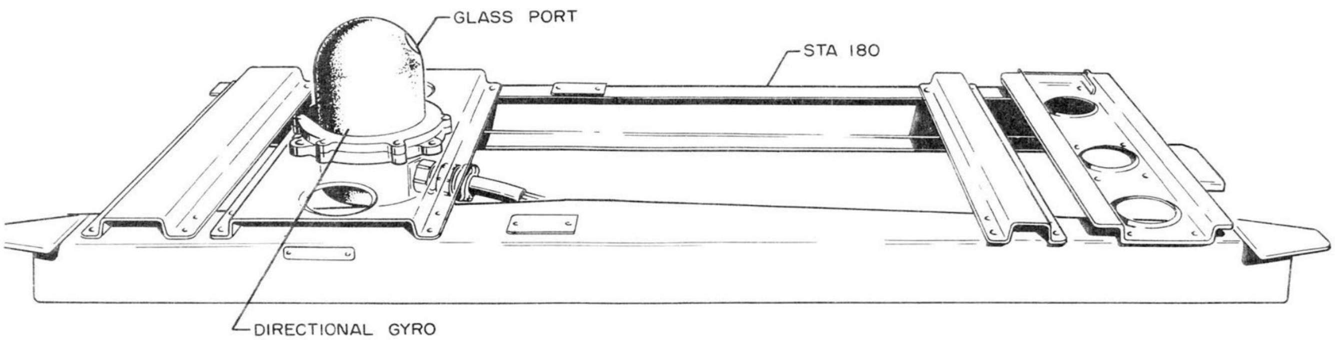


Figure 6–3A. MA-1 Compass System (Sheet 1)



VIEW D-D
DIRECTIONAL GYRO AND AMPLIFIER
(VIEW LOOKING FORWARD)

EFFECTIVITY-BUNO.
FACTORY: ALL AD 6 AIRPLANES
SERV. CHG: NONE

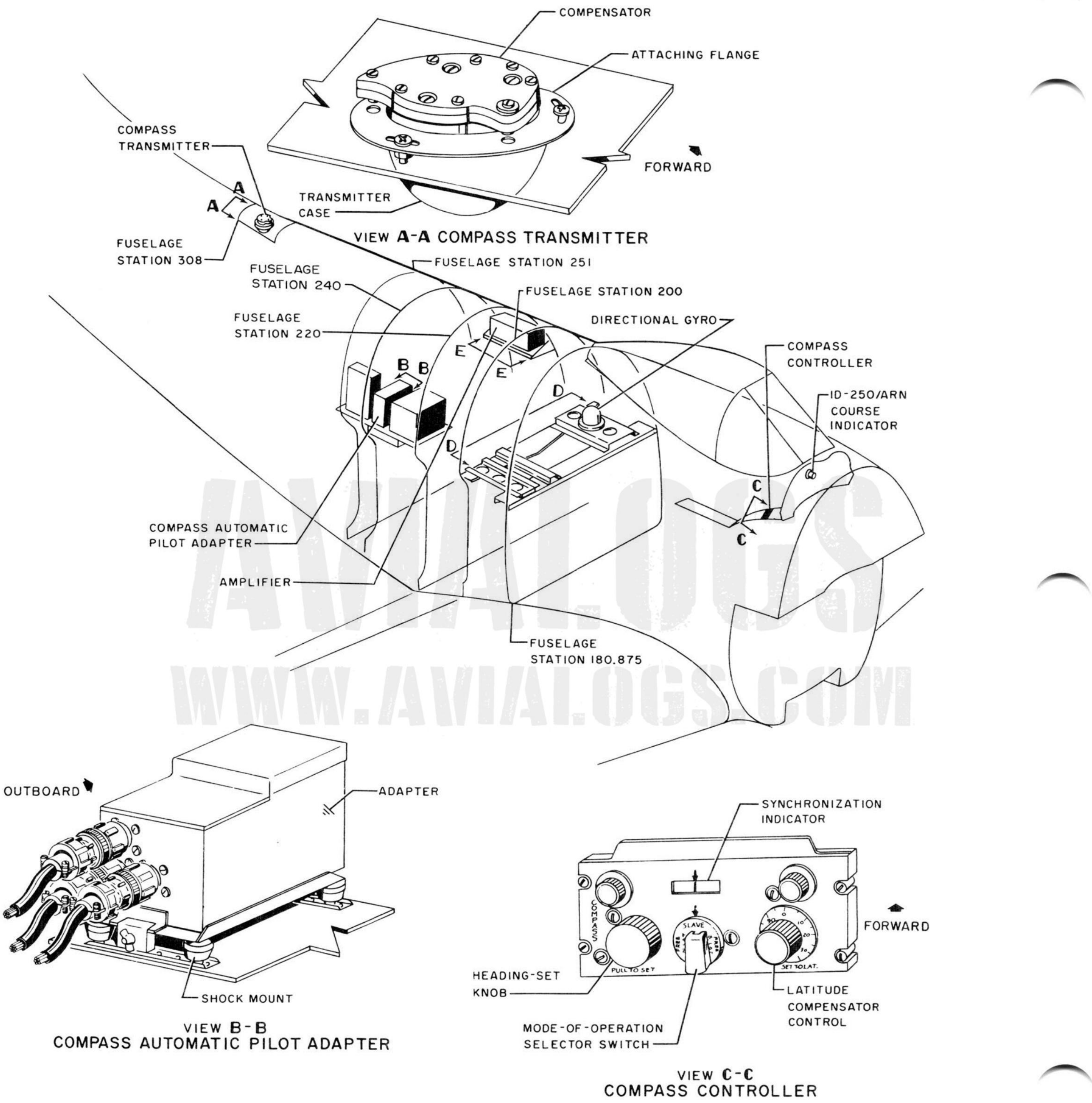


VIEW D-D
DIRECTIONAL GYRO
(VIEW LOOKING FORWARD)

EFFECTIVITY-BUNO.
FACTORY: ALL AD 7 AIRPLANES
SERV. CHG: NONE

P-8153-2A

Figure 6–3A. MA-1 Compass System (Sheet 2)



EFFECTIVITY-BUNO.
FACTORY: ALL AD 7 AIRPLANES
SERV. CHG: NONE

P-8153-3

Figure 6–3A. MA-1 Compass System (Sheet 3)

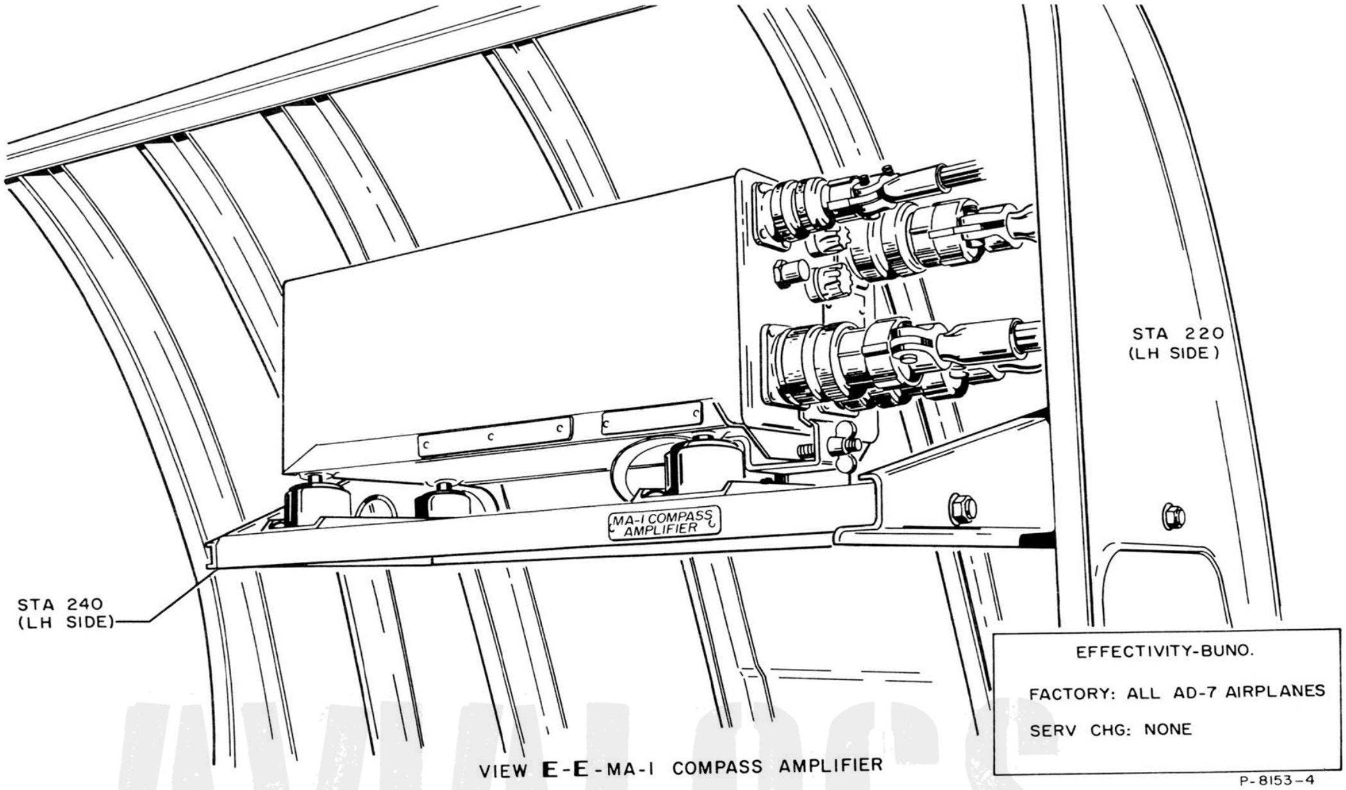


Figure 6–3A. MA-1 Compass System (Sheet 4)

fuselage stations 291.750 and 308. Access for maintenance is provided by the radio compartment access door. The functioning assemblies of the transmitter are enclosed within a hemispherical case that contains an attaching flange with an engraved index on the forward edge; for installation reference, the index is marked in 2-degree increments to encompass an angle of 10 degrees to the right or to the left of the fore-and-aft axis of the transmitter. The direction-sensing element consists of a spider, fabricated of highly permeable metal and resembling a three-spoke wheel. A slit is made through the rim between the spokes and the hub is widened to receive a core about which a primary or exciting coil is wound. Each leg of the spider is encircled with a secondary or pick-up coil. The element is suspended on a universal joint, thus allowing it to remain pendant through an arc of 30 degrees in pitch and roll; dampening fluid in the case prevents excessive oscillations during flight. Terminals on top of the case connect through lead wires to the spider coils. By energizing the primary coil with an alternating current, a voltage is induced in each pick-up coil. The magnitude of the induced voltage varies according to the angular position of its related spider leg with magnetic north. The voltage induced in each coil is transmitted to the amplifier which in turn supplies a voltage to energize the slaving torque motor to precess the directional gyro until its axis is aligned with the magnetic heading detected by the compass transmitter. A single-cycle compensator is installed on top of the transmitter case to eliminate transmitter errors caused by constant magnetic distortions from airplane parts and electrical equipment. The compensator contains four permanent magnets that are rotated by two slotted shafts to oppose the deflecting forces.

6–58J. REMOVAL.

a. Remove compensator from top of compass transmitter.

b. Disconnect electrical leads from terminals on transmitter case.

c. Remove screws attaching transmitter case flange to supporting bracket.

6–58K. INSTALLATION.

a. Place compass transmitter in position on supporting bracket with index on flange forward, and temporarily install attaching screws.

b. Remove compensator, if installed.

c. Connect electrical leads to applicable terminals on transmitter.

d. Install compensator.

e. Align center mark of index on transmitter case flange with mark on supporting bracket; then tighten attaching screws.

6–58L. DIRECTIONAL GYRO.

6–58M. DESCRIPTION. The directional gyro is the stabilizing component of the MA-1 compass system that responds to the direction sensed by the compass transmitter, thereby providing a means for obtaining an accurate indication of the airplane magnetic heading. It may also be operated as a free gyro without the magnetic sensing characteristics. The principal components of the mechanism are a gyro, a leveling torque motor, a slaving torque motor and a heading synchro. All components are hermetically sealed at a pressure of one atmosphere within a dome-shaped, helium-filled container and electrical connections to the components are made through a single receptacle near the base of the container. The mechanism is installed on the left-hand side of the center shelf located in the forward equipment compartment between fuselage stations 180.875 and 200. The gyro consists of a three-phase, high-speed motor with two separate stators, each surrounded by a rotor with a common horizontal axis. A supporting or inner gimbal is attached to the stator bearing and shaft assembly between the two rotors and is pivoted horizontally within an outer gimbal. The outer gimbal is pivoted vertically within the container and can rotate in azimuth without limit. The leveling torque motor consists of a two-phase stator which is fastened to the container and a hysteresis rotor attached to the top of the outer gimbal. When the gyro tilts in flight the motor exerts a torque about the vertical axis of the outer gimbal which precesses the gyro about its horizontal axis until the spin axis is level with the horizontal plane of the earth. The slaving torque motor consists of two sets of coils which are attached to the outer gimbal and two disk-shaped permanent magnets fastened to the inner gimbal. When the coils are energized, a torque is produced about the horizontal axis of the gyro and it precesses about the vertical axis until the spin axis is aligned with the direction detected by the compass transmitter. The stator of the heading synchro is attached to the bottom of the gyro container and the rotor is connected through gearing to the outer gimbal. Thus, any angular rotation in azimuth by the gyro will change the heading synchro rotor in relation to its stator. This causes a voltage to be induced which is transmitted to the indicator in the cockpit for interpretation. A compass card, used for reference purposes, is attached to the top of the outer gimbal and can be viewed through a glass port near the top of the container.

6–58N. AMPLIFIER.

6–58P. DESCRIPTION. The amplifier functions as a junction box and power supply for the components of the MA-1 compass system. On all model AD-6 airplanes it is installed in the radio compartment on the right-hand side at approximate station 200, above the center line of the airplane. On all model AD-7 airplanes it is moved aft in the radio compartment, and installed on the left-hand side at approximate station 263. Functioning components of the amplifier include a servo unit, a servo amplifier, a compass amplifier and a leveling amplifier. For detailed information regarding operation of the mechanism, refer to Handbook of Operation and Service Instructions for MA-1 Compass-Controlled Directional-Gyro System, NAVAER 05-15C-501.

6–58R. COMPASS CONTROLLER.

6–58S. DESCRIPTION. The compass controller provides a means for operating the MA-1 compass system. It also contains indicators which show system synchronization and power failure. Principal components of the controller are a heading-set knob, a mode-of-operation selector switch, a latitude compensator control and a synchronization indicator. All components are contained within a rectangular case equipped with a plastic panel which attaches by fasteners to the right-hand control panel. Electrical connections to the components are made through a single connector on the bottom of the case. The heading-set knob, identified as PULL TO SET, permits rotation of a synchro differential which is electrically connected between the output synchro of the gyro and a synchro control transformer in the amplifier. This arrangement makes it possible to change the system heading without actually moving the gyro. Pulling the knob out also actuates a switch that breaks the circuit to the automatic pilot system when the heading is being changed. The mode-of-operation selector switch, identified as FREE N. LAT—SLAVED—FREE S. LAT, is utilized to operate the gyro either slaved to the compass transmitter or as a free directional-gyro. For free-gyro operation there are two switch positions: one for northern latitudes, and one for southern latitudes. The latitude compensator, identified as SET TO LAT, is effective only when the system is operated as a free-gyro. When the control is set, direct current is allowed to flow in the coils of the gyro slaving torque motor, and the gyro is precessed at a rate equal and opposite to the apparent drift due to the earth's rotation. The synchronization indicator incorporates a white pointer that is visible through a window in the controller panel. When the pointer is in line with the white arrow on the panel, the system is in synchronization and the angular position of the amplifier servo shaft corresponds to the direction sensed by the compass transmitter. In the event a power failure occurs in the system, a red warning flag is displayed in the synchronization indicator window. The controller may be removed by loosening the fasteners, lifting the unit from the control panel and disconnecting the electrical connector below the case.

6–58T. COMPASS AUTOMATIC PILOT ADAPTER.

6–65U. DESCRIPTION. The compass automatic pilot adapter is installed on a supporting bracket on the right-hand side of the radio compartment between fuselage stations 220 and 251.500. The adapter is utilized to electrically connect the MA-1 compass system to the P-1 automatic pilot system. The MA-1 compass system provides the P-1 automatic pilot system with a stable directional reference.

6–58V. ID-250/ARN COURSE INDICATOR.

6–58W. DESCRIPTION. The ID-250/ARN course indicator is an electrically operated mechanism which is capable of displaying directional information from three independent systems. All functioning components of the indicator are enclosed within a cylindrical case and installed on the instrument panel. The indicator dial consists of a movable circular scale which is marked in 2-degree increments through 360 degrees. On airplanes prior to BuNo. 139606, voltage derived from the G-2 compass system energizes a synchro which positions the scale below a fixed index to denote the magnetic heading of the airplane. This voltage is supplied by the MA-1 compass system on airplanes BuNo. 139606 and subsequent. Two pointers, which revolve about an axis in the center of the circular scale, operate in conjunction with independent radio navigational systems. The single-barred pointer, identified as "1," indicates bearing to a station relative to the longitudinal axis of the airplane and is actuated by a mechanism which is energized by the AN/ARA-25 automatic direction finding system. The actuating mechanism for the double-barred pointer, identified as "2," is energized by the AN/ARN-6 radio compass system (or an AN/ARN-14E navigation system) on airplanes prior to BuNo. 139606, and positions the pointer to denote magnetic bearing to a station. On airplanes BuNo. 139606 and subsequent, this actuating mechanism is energized by the AN/ARN-21 navigational radio homing system. The single-barred pointer is slaved to the double-barred pointer when the AN/ARA-25 system is inoperative and either the AN/ARN-6, AN/ARN-14E or the AN/ARN-21 system is in use. However, when the AN/ARA-25 system is operated simultaneously with one of the other systems, both pointers function independently to display the heading information from their respective systems. When the AN/ARA-25 system only is utilized, the double-barred pointer is inoperative. For further information regarding the actuating systems for the indicator pointers, refer to section VIII.

6-59. P-1 AUTOMATIC PILOT SYSTEM.

6-60. DESCRIPTION. (See figure 6-3.) The P-1 automatic pilot is a system of automatic controls which is electrically powered and gyroscopically controlled. The system includes the following principal components:

Name	*Para Ref*
Controller	6-68
Gyro horizon control	6-73
Rate gyro control	6-76
Power junction box	6-78
Servo amplifier	6-80
Amplifier adapter	6-82
Servos	6-84
Servo disconnect system	6-88
Servo disconnects	6-91
Clutch switch	6-93

6-61. The automatic pilot system is controlled from the automatic pilot control console mounted in the cockpit right-hand control panel. The system automatically operates airplane flight controls to maintain any selected magnetic heading of the airplane and, simultaneously, retain stabilized pitch and bank attitudes of the airplane. Operation of the system is dependent upon electrical signals emanating from three sources: the direction signal originating in the remote compass, the rate-of-turn signal originating in the rate gyro control, and the pitch and bank signal originating in the gyro horizon control. The signals control the direction and degree of rotation of servo units which are cable-connected to the airplane flight controls, thereby providing conversion of electrical impulses into mechanical action.

6-62. To prevent over-control of the airplane by the automatic pilot control signals, a follow-up signal, originating in each of the servo units, is provided to oppose the control signal. By proper combination of control signals and opposing follow-up signals, the airplane is returned to the selected attitude or heading, when momentary displacements occur, without oscillating overswing, or "hunting."

6-63. When engaged, the automatic pilot system receives a stabilized directional signal from the master direction-indicator of the G-2 compass system.

6-63A. On airplanes BuNo. 139643 and subsequent, and prior airplanes reworked per BuAer AD/SC No. 525, a single-pole, single-throw switch is installed in the automatic pilot control console and provides a means for disengaging the elevator control servo when the automatic pilot system is engaged. The switch has two indicated positions, "OFF" and "ON," and is identified on the control console as ELEV CONT.

6-64. REMOVAL. Before attempting to remove any of the automatic pilot equipment, make certain that the following conditions are observed:

a. Battery switch: "OFF."

b. External power source disconnected.

6-65. The following general steps should be taken to remove any component of the automatic pilot system:

a. Remove any cover, shield, or other protection from the unit.

b. Disconnect all plugs, wiring, or bonding braid attached to the unit to be removed. Secure plugs, receptacles, or open wiring to prevent damage thereto.

c. Remove attaching parts and remove unit.

6-66. INSTALLATION. The following general steps should be taken to install any component of the automatic pilot system:

a. Place unit on bracket or support and install attaching parts.

b. Apply anti-seize compound (Specification JAN-A-669) to threaded connections of unit, and connect all wires, plugs, or receptacles to unit. (For correct wiring information, refer to applicable wiring diagram in section X.)

c. Make certain that all required bonding between unit and airplane structure is free of non-conducting materials.

d. Install protective covering on unit.

6-67. TESTING. Table 6-2 provides an outline of procedures for testing the entire automatic pilot system. Control settings and/or operations are listed in recommended performance sequence wherever performance can be visually verified. If the results noted are obtained, it may be assumed that the equipment is in a condition satisfactory for flight. However, a flight operational test is required to ascertain that the performance of the equipment is properly adjusted to the particular characteristics of the individual airplane on which it is installed. If the noted results are not obtained, more extensive tests and adjustments should be made, in accordance with applicable service publications, by qualified personnel.

6-68. P-1 AUTOMATIC PILOT CONTROLLER.

6-69. DESCRIPTION. (See figure 6-3.) The automatic pilot controller provides a means of altering the heading or course of the airplane without disengaging the automatic pilot system. The controller is installed in the right-hand control panel and includes a TURN control knob, a BANK TRIM control wheel and a PITCH control wheel. The TURN control knob enables the pilot to make co-ordinated, constant-rate turns. Operation of this knob governs the signals from the bank, rate, and pitch servos in the automatic pilot servos. In straight flight, the TURN control knob is retained in a center position by a lever which engages the turn control shaft.

TABLE 6-2. TESTING PROCEDURE — P-1 AUTOMATIC PILOT SYSTEM

The following conditions are mandatory during all ground tests and should be verified before proceeding with the test:
a. Battery switch: "ON."
b. External power source connected.
c. No extraneous material or equipment close to control surfaces.

Note

Allow sufficient time between steps in the procedure for instruments and associated equipment to reach proper operating conditions.

Control Settings and/or Operation	*Desired Results*
Energize a-c power circuit as follows: Flight instrument power selector switch: "INVERTER 1." Appropriate No. 1 and No. 2 inverter circuit breakers and d-c instrument circuit breaker: depress.	Both inverters start operating.
Flight instrument power selector switch: "INVERTER 2 & AUTO PILOT."	Both inverters start operating.
No. 2 inverter circuit breaker: open.	No. 2 inverter stops; inverter warning light becomes illuminated.
Energize flight control circuits as follows: AUTO PILOT circuit breaker: depress.	No change.
Gyro horizon control cage switch: "UNCAGE."	Gyro horizon-indicator energized; master direction-indicator completes uncaging cycle.
Control stick: neutral.	Elevators and ailerons in neutral position.
Rudder pedals: neutral.	Rudder in neutral position.
Clutch switch: depress.	Automatic pilot servos energized; controls held in neutral position.
Gyro horizon control cage switch: "CAGE."	Clutch switch releases; automatic pilot servos de-energized; controls not held in neutral.
Gyro horizon control cage switch: "UNCAGE."	Gyro horizon-indicator energized; master direction-indicator completes caging cycle.
Clutch switch: depress.	Automatic pilot servos energized; controls become fixed.
Automatic pilot controller turn knob: rotate clockwise.	Rudder: right. Ailerons: right wing down. Elevators: up.
Automatic pilot controller turn knob: rotate counterclockwise.	Rudder: left. Ailerons: left wing down. Elevators: up.
Automatic pilot controller pitch trim wheel: "DOWN."	Elevators: down.
Automatic pilot controller pitch trim wheel: "UP."	Elevators: up.

The BANK control wheel when operated changes the relative positions of the bank autosyn rotor and stator, thereby generating a bank signal. The PITCH control wheel when operated changes the relative positions of the pitch autosyn rotor and stator, thereby generating a pitch signal.

6-70. BENCH ADJUSTMENT OF AUTOMATIC PILOT CONTROLLER.

6-71. DESCRIPTION. (See figure 6-3B.) Before installing a new or overhauled controller in the model A-1H airplane, make certain the controller will meet the requirements of the bench adjustment procedure outlined in figure 6-3B. Controllers not meeting the bench adjustment requirements must not be used in the model A-1H airplane.

6-72. Deleted.

6-73. P-1 GYRO HORIZON INDICATOR

6-74. DESCRIPTION. (See figures 6-1 and 6-3.) The gyro horizon indicator is mounted on an extension to the pilot's instrument panel on the centerline of the cockpit just below the armament panel. The gyro horizon indicator contains an electrically driven, vertical-seeking, non-tumbling gyro which provides a stable vertical reference from which can be measured any departure in airplane pitch and bank. The instrument also contains the pitch and bank autosyns. The bank autosyn, with rotor mounted in the gyro yoke gimbal and stator secured to a stationary bracket, provides, by induced voltages in the stator, a signal which is in proportion to the degree of bank with the automatic pilot aileron servo unit. The pitch autosyn, with rotor mounted on the gyro motor housing and stator secured to the gyro yoke gimbal, similarly provides a signal which is in proportion to the degree of pitch with the automatic pilot elevator servo unit. The gyro horizon indicator furnishes visual indications of the degree of pitch and bank whether the automatic pilot is engaged or disengaged, and a window in the center of the instrument face furnishes a numerical indication of the degree of pitch of the airplane.

6-75. The gyro horizon indicator caging mechanism is manually controlled and is entirely mechanical. The caging mechanism is operated from the caging knob at the front of the instrument.

CAUTION

Internal components of the gyro horizon indicator are subject to internal damages and misalignment during transportation or handling due to uncontrolled movement of their mass components.

6-75A. To prevent damage from ocurring to the gyro horizon indicator the following detailed instructions as outlined in Avionics Bulletin 50 dated 19 June 1964 must be adhered to:

a. The gyro horizon indicator must be locked when prepared for shipment or removed from the airplane. Care to be exercised when unlocking during installation.

b. Personnel performing repair, inspection, or assembly of gyro horizon indicator must lock indicator prior to release for shipment.

c. Operational personnel must return any indicator received in an unlocked position to the cognizant repair activity from which it was received, with necessary FUR information citing Electronic Bulletin 50 as authority.

6-75B. REMOVAL. Refer to paragraph 6-8.

CAUTION

Cage gyro before removing gyro horizon control. Handle instrument with extreme care to protect delicate gyro element inside.

6-75C. INSTALLATION. Refer to paragraph 6-9.

6-76. P-1 AUTOMATIC PILOT RATE GYRO CONTROL (BANK AND TURN CONTROL).

6-77. DESCRIPTION. (See figure 6-3.) The rate gyro control is mounted on a support installation just forward of fuselage station 251.500 on the right-hand side of the airplane. The instrument contains the rate autosyn and an electrically driven gyro, mounted with axis athwartship. The gyro is supported in a housing pivoted in a line with the fore-and-aft axis of the airplane. Whenever the airplane turns, a force in proportion to the amount of turn is applied to the gyro axis, attempting to make the gyro turn also; because of the pivot

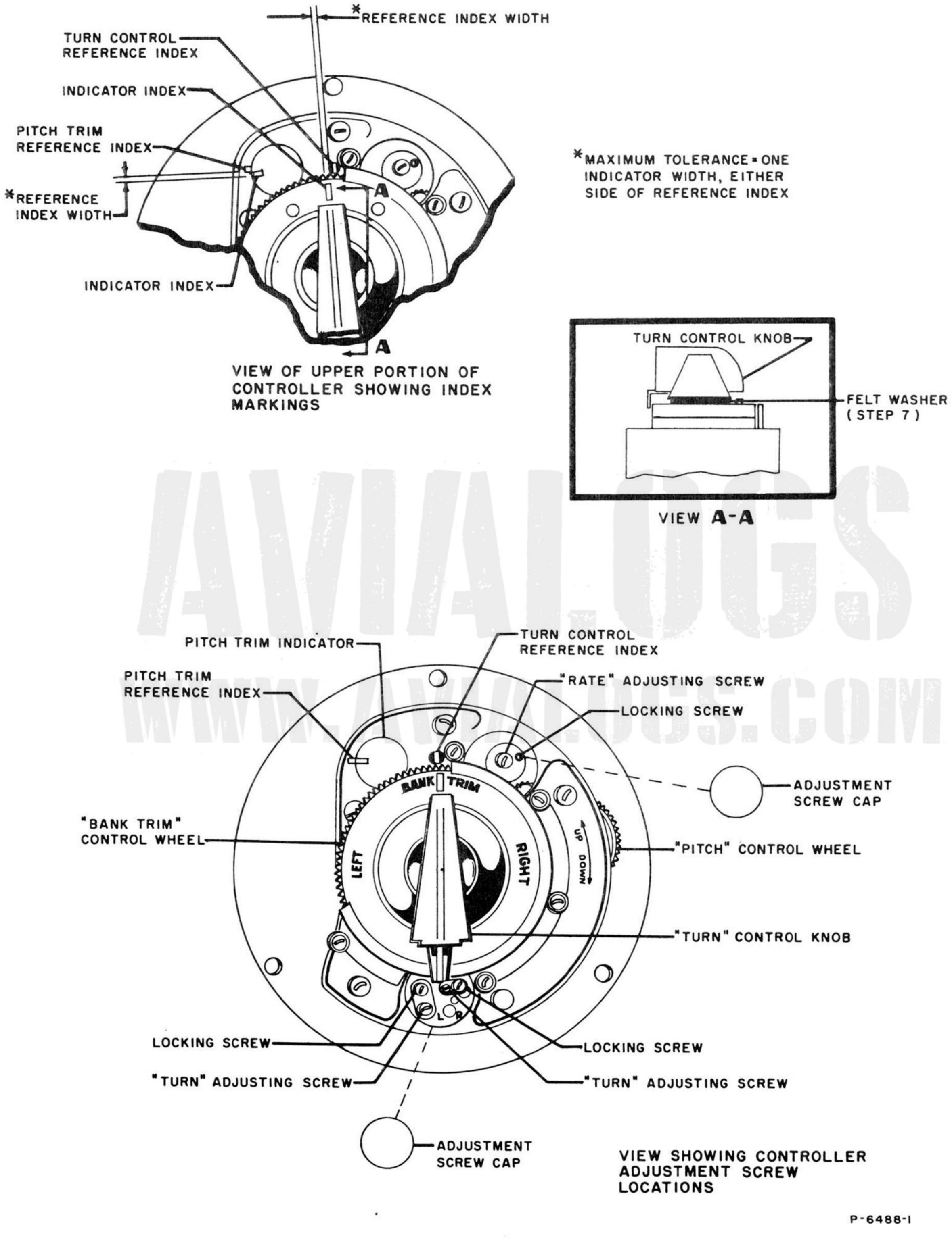
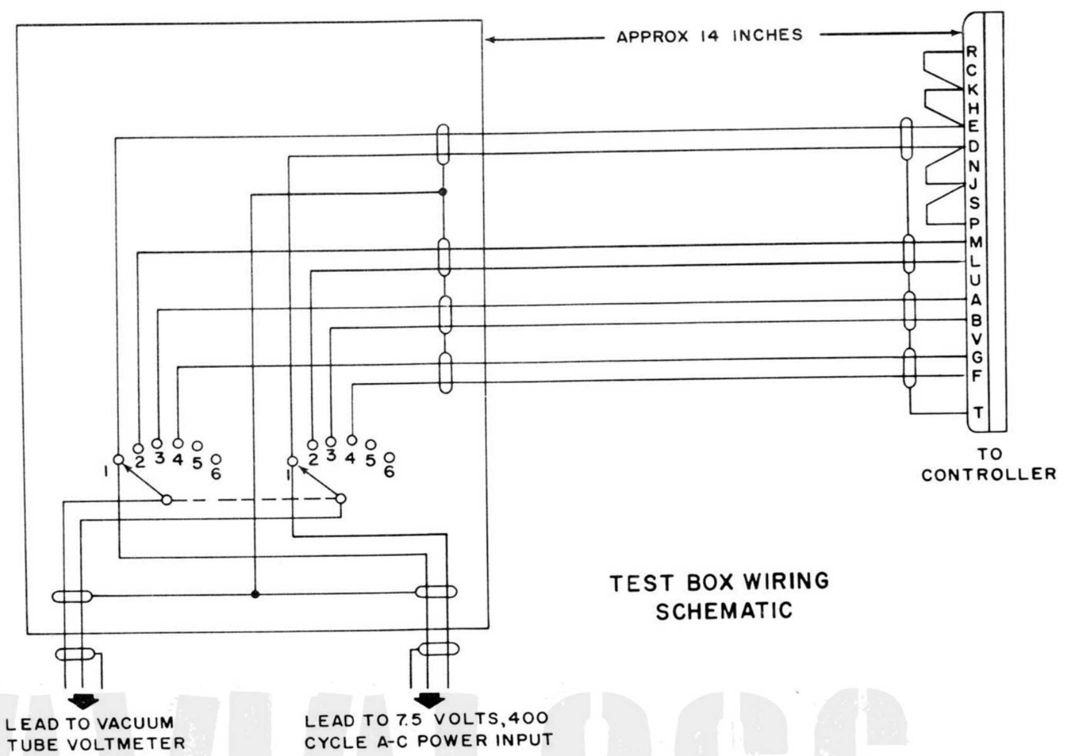


Figure 6–3B. P-1 Automatic Pilot Controller—Bench Check Out and Adjustment (Sheet 1)



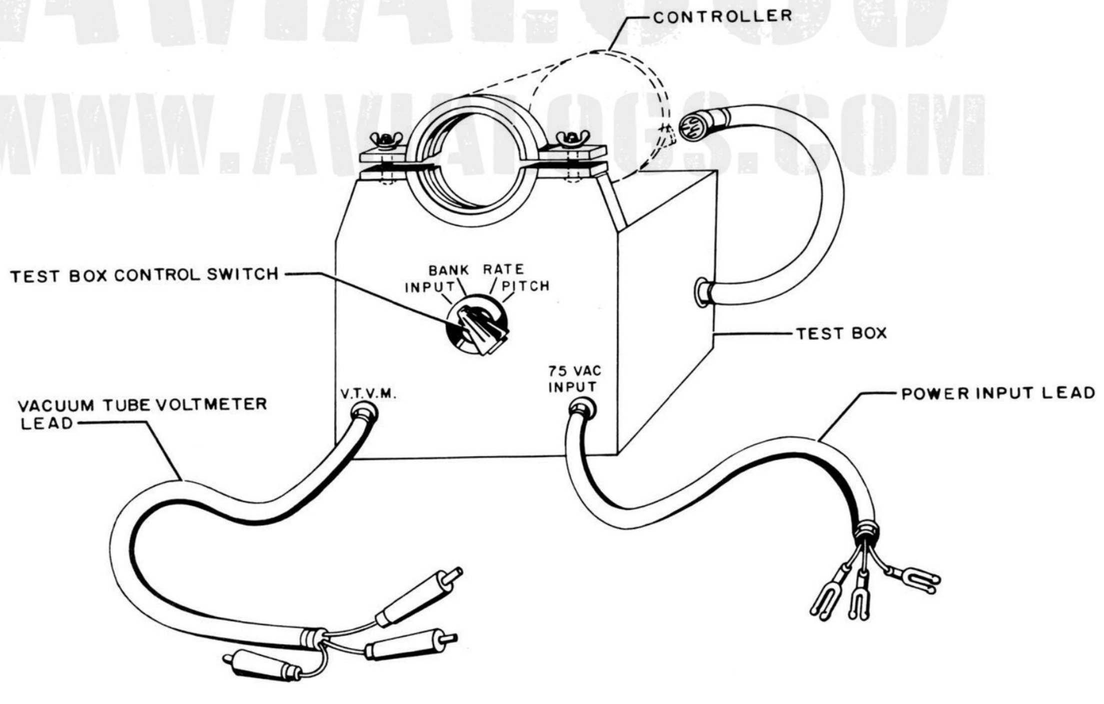


Figure 6–3B. P-1 Automatic Pilot Controller—Bench Check Out and Adjustment (Sheet 2)

Equipment required:

a. Vacuum tube voltmeter.

b. 7.5-volt a-c 400-cycle power source.

c. Test box. (See sheet 2.)

CAUTION

Input voltage must be adjusted to 7.5 volts for each step of the adjustment procedure.

TEST BOX SWITCH POS	CONTROL SETTING AND/OR OPERATION	VOLTMETER READING	REMARKS
1.	Set voltmeter on 10-volt scale. Voltages will be read on a-c scale.		Place meter in best position for accurate reading of scale.
2.	Connect controller to test box.		
3. INPUT	Connect 7.5-volt a-c 400-cycle power source.	7.5 volts a-c	An input of 7.5-volt a-c must be maintained throughout the entire test procedure.
4. PITCH	Center all control knobs. Rotate PITCH trim wheel "UP" until voltmeter reads 20 millivolts on 0.03-volt scale. Return wheel to center position.	20 millivolts normal. 27 millivolts maximum.	If the voltmeter reading exceeds 27 millivolts while the wheel is being returned to center position, reject controller.
5. PITCH	Rotate PITCH trim wheel "DOWN" until voltmeter reads 20 millivolts on 0.03-volt scale. Return wheel to center position.	20 millivolts normal. 27 millivolts maximum.	If voltmeter reading exceeds 27 millivolts while wheel is being returned to center, reject controller.
6. PITCH	Rotate PITCH trim wheel "UP" and "DOWN" through center until voltmeter reads minimum voltage on 0.03-volt scale.	10 millivolts maximum allowable voltage.	If minimum voltage reading exceeds 10 millivolts when the reference and indicator index markers coincide within the width of one marker width, reject controller. If controller is satisfactory, leave PITCH trim wheel in minimum voltage position and proceed with step 7.
7. BANK	Rotate BANK trim control until voltmeter reads minimum on 0.03-volt scale.	8 millivolts maximum allowable voltage.	If voltmeter reading exceeds 8 millivolts when the reference and index markers coincide within one marker width, reject controller. With BANK trim knob in minimum voltage position, rotate TURN control through detent several times. If the BANK trim control is slipping, voltage readings may exceed 8 millivolts. If this occurs, add felt washer under TURN control knob or reject controller.
8. RATE	Rotate turn control through detent several times. Read minimum voltage on 0.03-volt scale.	15 millivolts maximum allowable voltage.	If voltmeter reading exceeds 15 millivolts when TURN control is returned to detent from either direction, reject controller.
9. BANK	With all controls in the positions achieved by the previous steps, rotate TURN control to LEFT, out of detent, until voltmeter reads one volt. Leave TURN control in this position and remove screw cap located in the lower center controller face.	One volt.	
PITCH	Loosen locking screws. Turn "L" PITCH adjustment screw until voltmeter reads 170 (plus or minus 10) millivolts. Tighten locking screw.	170, plus or minus 10 millivolts.	
	Remove screw cap located in the upper right quarter of the controller face. (See sheet 1.)		

Figure 6–3B. P-1 Automatic Pilot Controller—Bench Check Out and Adjustment (Sheet 3)

TEST BOX SWITCH POS	CONTROL SETTING AND/OR OPERATION	VOLTMETER READING	REMARKS
RATE	Loosen RATE adjustment locking screw. While holding TURN control knob, turn RATE adjustment screw until voltmeter reads 280 (plus or minus 30) millivolts on the 0.03-volt scale. Tighten locking screw. Return TURN control to center position.	280, plus or minus 30 millivolts.	
10. BANK	Rotate TURN control to RIGHT, out of detent until voltmeter reads one volt on 0.3-volt scale. Leave the TURN control in this position.	One volt.	
PITCH	Loosen locking screw for "R" TURN, adjusting screw until voltmeter reads 155 (plus or minus 10) millivolts on the 0.3-volt scale. Tighten locking screw.	155, plus or minus 10 millivolts.	
RATE	Upon switching test box to RATE, the voltmeter should read 280 (plus or minus 30) millivolts on the 0.3-volt scale. If it does not, split the difference between the reading obtained in step 9 and step 10 and readjust the RATE adjusting screw so that both tolerances can be met.	280, plus or minus 30 millivolts.	If tolerance in RATE voltage cannot be met, reject controller.
11. PITCH	While observing voltmeter, rotate TURN control SLOWLY, 90 degrees to right and left of center several times and stop in detent. If the voltmeter reads more than 10 millivolts on the 0.03 scale, readjust the PITCH trim wheel to minimum voltage reading. Repeat the above several times.	10 millivolts on 0.03-volt scale.	If PITCH trim wheel cannot be adjusted so that the voltmeter reading is 10 millivolts or less when the TURN control snaps into detent from either direction, reject controller. THE PITCH TRIM WHEEL MUST NOT BE MOVED FROM THE POSITION AS DETERMINED ABOVE DURING THE REMAINDER OF THE TEST.
12. BANK	Recheck minimum voltage reading given in step 7. Leave the BANK TRIM wheel in the minimum voltage position for the remainder of the test.		
PITCH and RATE	Recheck voltage and tolerance achieved in steps 9 and 10.		After several voltage checks, if the controller will not maintain voltage tolerances as specified, reject controller.
PITCH	Rotate TURN control RIGHT until voltmeter reading exceeds 50 millivolts on the 0.1-volt scale.	50 millivolts on the 0.1-volt scale.	
		65 volts maximum allowable on return to detent.	If the voltmeter reading exceeds 65 millivolts while the TURN control is being returned to center detent, reject the controller.
PITCH	Rotate TURN control LEFT until voltmeter reading exceeds 50 millivolts on the 0.1-volt scale.	50 millivolts on the 0.1-volt scale.	
		65 volts maximum allowable on return to detent.	If the voltmeter reading exceeds 65 millivolts while the TURN control is being returned to center detent, reject the controller.

Figure 6—3B. P-1 Automatic Pilot Controller—Bench Check Out and Adjustment (Sheet 4)

alignment of the gyro housing, a turn of the airplane causes the gyro axis and gyro housing to precess. The rotor of the rate autosyn is mounted on the gyro housing, and the stator of the rate autosyn is secured to the case. Therefore, precession of the gyro, due to turning of the airplane, causes relative movement between the autosyn rotor and stator, resulting in changes in relationship of induced voltages in the three autosyn stator leads. The relationship of voltages will always be proportional to the rate of turn of the airplane. When the airplane ceases to turn, centralizing springs return the gyro to a central position.

CAUTION

Handle rate gyro control with extreme care to protect delicate gyro element inside.

6–78. P-1 AUTOMATIC PILOT POWER JUNCTION BOX.

6–79. DESCRIPTION. (See figure 6–3.) The power junction box is mounted on a support installation on the right-hand side of the airplane at fuselage station 233.312. The power junction box interconnects the a-c and d-c power supplies, and contains a three-phase transformer assembly to step the voltage down from 115 volts to 26 volts a-c, a three-pole power relay, three condensers, and four standard electrical connector receptacles.

6–80. P-1 AUTOMATIC PILOT SERVO AMPLIFIER.

6–81. DESCRIPTION. (See figure 6–3.) The servo amplifier is shock-mounted on a support installation just aft of fuselage station 220, on the right-hand side of the airplane. The amplifier embodies three separate but identical channels, one each for rudder, aileron and elevator signals. The output of each channel furnishes power to the variable phase of the low inertia motor in the related servo unit.

6–82. P-1 AUTOMATIC PILOT AMPLIFIER ADAPTER.

6–83. DESCRIPTION. (See figure 6–3.) The amplifier adapter is mounted on a support installation just aft of fuselage station 220, on the right-hand side of the airplane. The adapter is the medium by which the automatic pilot is adjusted to the characteristics of the individual airplane. When installing a new or overhauled amplifier adapter in the model AD-6 airplane, the adapter must be adjusted to meet the flight characteristics of the airplane. Adjustment is made by means of four potentiometers and three double throw switches, accessible when the top cover plate is removed from the adapter.

6–83A. ADJUSTMENT.

a. Remove top cover plate from amplifier adapter.

b. Remove cotter pins securing three switches.

Note

Make adjustments in the following alphabetical sequence.

c. Set RUDDER switch to "B."

d. Set AILERON switch to "B."

e. Set ELEVATOR switch to "A."

f. Install cotter pins to secure three switches.

g. Set RATE control to 100. This varies the amount of rate-of-turn signal in proportion to the directional signal.

h. Set RUDDER control to 65. This varies the amount of follow-up signal in proportion to the combined rate-of-turn and directional signal.

i. Set AILERON control to 85. This varies the amount of follow-up signal in proportion to the bank signal.

j. Set ELEVATOR control to 100. This varies the amount of follow-up signal in proportion to the pitch signal.

WARNING

When it becomes necessary to change above settings due to unsatisfactory performance during flight check, *do not exceed following limits:*

Control	*High*	*Low*
RATE	100	90
RUDDER	75	55
AILERON	95	75
ELEVATOR	100	90

Note

If adjustments between these limits fail to give correction desired, the error is in some other section of the automatic pilot system. Return adjustments to original settings and check balance of system.

6–84. P-1 AUTOMATIC PILOT SERVOS.

6–85. DESCRIPTION. (See figure 6–4.) The automatic pilot system includes three servo units, one servo incorporated into each of the cable control systems related to the control surfaces involved in the automatic pilot system: the ailerons, the elevators, and the rudder. The aileron servo is located forward of the spar web in the left-hand wing center section just outboard of wing station 30.165. The elevator servo is located between fuselage stations 330.125 and 335.750 on the center line of the airplane. The rudder servo is located in the forward equipment compartment, just below the cockpit floor, approximately 17 inches to the left of the center line of the airplane. Each servo unit includes an induction-type low inertia motor, a damper, a clutch, and a follow-up autosyn. The output torque of the low inertia motor is transferred to the servo power shaft connected to the controls through the medium of a solenoid-operated clutch. When the solenoid is energized, through the automatic pilot clutch switch, the clutch is held in the engaged position. The damper is provided to eliminate oscillation or "hunting" of the servo. The follow-up autosyn provides the signals to check control surface movement as soon as any corrective trim adjustment of the surface has been made through the automatic pilot system.

6–86. REMOVAL. (See figure 6–4.) The following removal procedure is generally applicable to any of the three automatic pilot servo units:

a. For removal of the aileron servo, first remove left-hand wing access door.

b. Disconnect plug coupling nut at end of servo and remove electrical plug from receptacle.

c. For removal of aileron and elevator servos, loosen turnbuckles governing control cable tension at servo pulleys. For removal of rudder servo, loosen terminal nuts on rudder control drum only enough to relieve control cable tension.

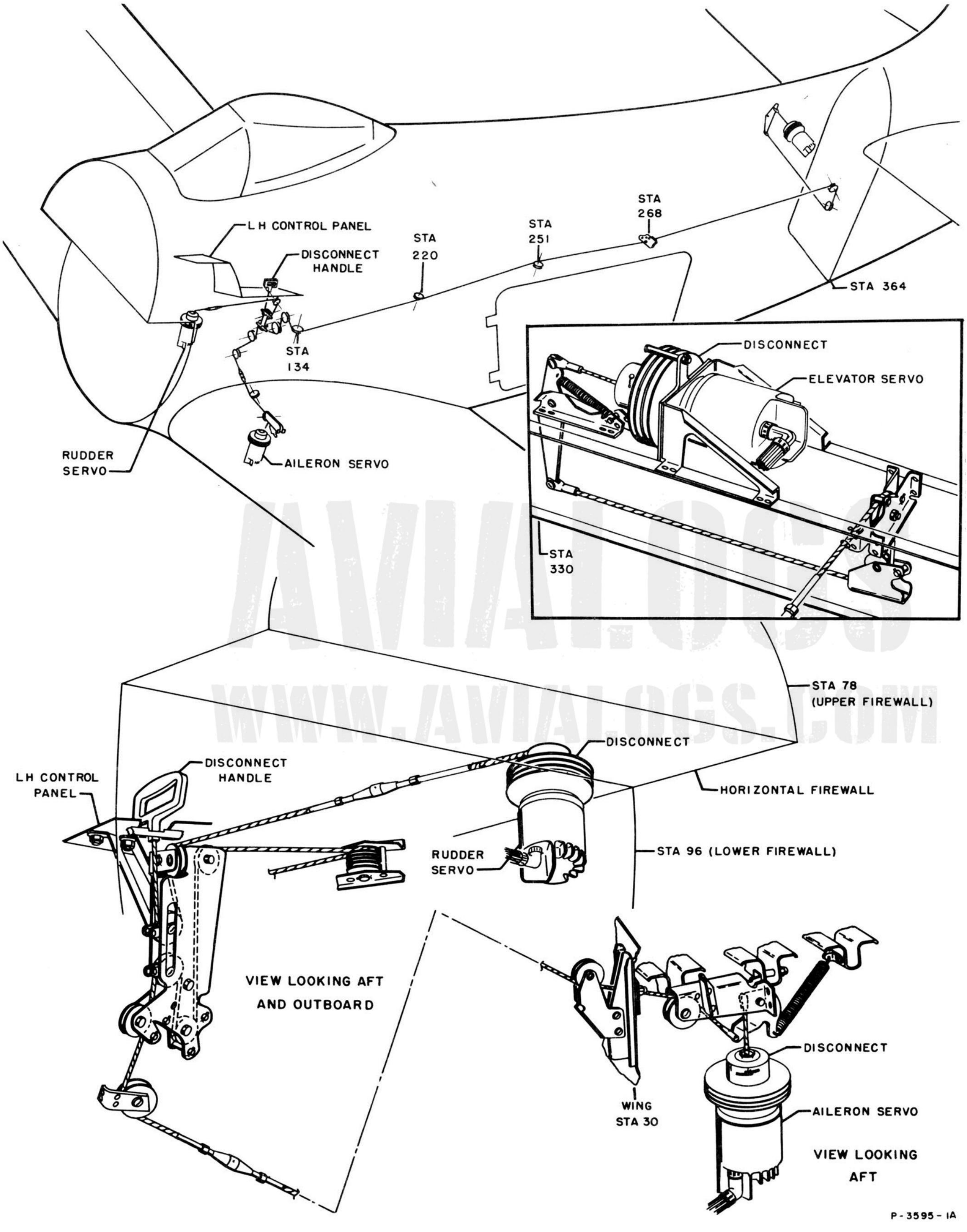
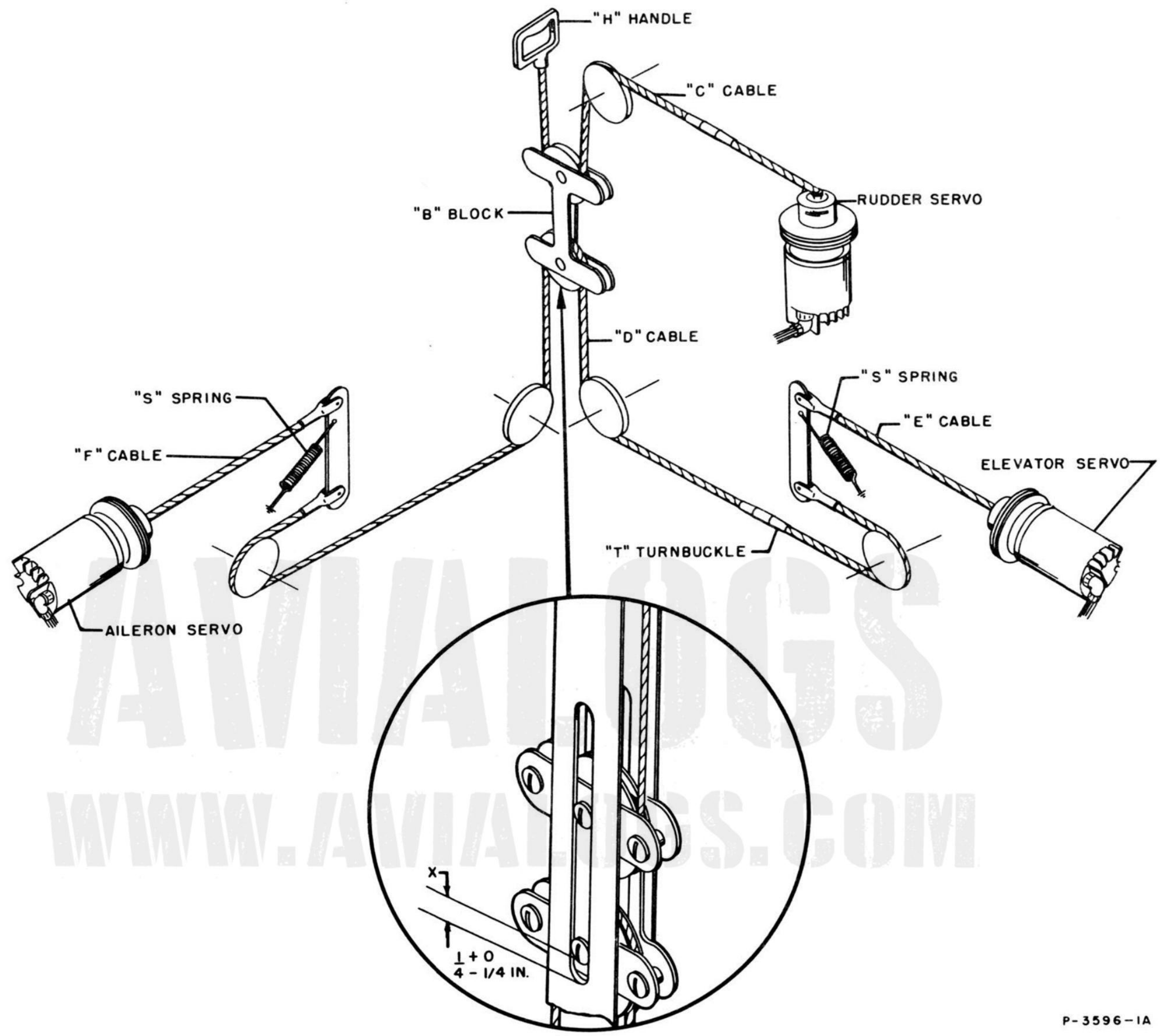


Figure 6–4. P-1 Automatic Pilot Servo Disconnect System



RIGGING PROCEDURE

STEP 1

Rig cable "D" by adjusting turnbuckle "T" in fuselage so that there is no slack in cables "D," "E," or "F" when block "B" is resting on bottom of slot in supporting structure or within ¼ inch of bottom of slot, as shown by dimension "X."

STEP 2

When step 1 is completed:

a. Cables "E" and "F" will have no slack, but will not be loaded.

b. Cable "D" will have no slack, and will be loaded only by force induced by spring "S."

STEP 3

Adjust cable "C" not to be loaded, but to have no slack with handle "H" against panel.

STEP 4

Check system:

Pull handle "H" and ascertain that all three servos are disconnected before handle reaches full travel.

STEP 5

Reset servos.

Figure 6–5. P-1 Automatic Pilot Servo Disconnect System Adjustment

d. At servo disconnect, break lockwire which safeties two clamp levers, squeeze clamp levers together (or loosen screw as applicable).

e. Break lockwire and remove bolts attaching servo to bracket and remove servo.

6-87. INSTALLATION. (See figure 6-4.) The following installation procedure is generally applicable to any of the three automatic pilot servo units:

a. Install and lockwire bolts attaching servo to bracket.

b. Position servo on bracket and engage servo power shaft with servo disconnect by squeezing together two clamp levers on servo disconnect.

NOTE

The servo disconnect clamp levers must be in the locked position after lockwiring or, if applicable, tightening of screw 30 to 40 inch-pounds. To check locking of clamp levers, grasp unit with both hands and pull unit hard to ascertain engagement of servo power shaft.

c. Rig control cables to correct tension. (Refer to section II.)

6-88. P-1 AUTOMATIC PILOT SERVO DISCONNECT SYSTEM.

6-89. DESCRIPTION. (See figure 6-4.) The automatic pilot servo disconnect system consists of the three servo disconnect units, a control handle, and the necessary pulleys and control cables. The disconnect handle, AUTOPILOT REL., is mounted in the cockpit left-hand control panel. When the handle is pulled, the cable system is operated to actuate the automatic pilot servo disconnect units simultaneously to disengage each from its related servo. The cable system is spring-loaded, causing the release handle to return to its original position after being pulled. When the automatic pilot has been disengaged by the servo disconnect system, the automatic pilot cannot be re-engaged in flight.

6-90. ADJUSTMENT. See figure 6-5.

6-91. P-1 AUTOMATIC PILOT SERVO DISCONNECTS.

6-92. DESCRIPTION. (See figure 6-4.) A servo disconnect is installed at each of the three automatic pilot servo units. The disconnect acts as a mechanical coupling between the servo power shaft and the cable drum of the related flight cable control. The servo disconnects are controlled from the handle, AUTO-PILOT REL., mounted in the left-hand control panel in the cockpit. When the handle is pulled, the servo disconnects are operated simultaneously by a cable control system to disengage the power shaft of each related servo unit. Once disengaged, the servo units cannot be re-engaged in flight. Re-engagement is accomplished on the ground individually at each disconnect by pressing outward on the two levers at the end of the disconnect and rotating the disconnect unit until it engages the servo power shaft.

6-93. P-1 AUTOMATIC-PILOT CLUTCH SWITCH.

6-94. DESCRIPTION. (See figure 6-3.) The automatic-pilot clutch switch is mounted on the automatic-pilot controller in the cockpit right-hand control panel. Actuation of the switch de-energizes the double-pole, double-throw clutch relay which energizes the solenoid-operated servo clutches and engages the servos to the automatic pilot. The clutch switch can be actuated to engage the clutches only when the gyros in the automatic pilot controls are uncaged. A clutch over-ride switch in the master direction indicator is actuated by the operation of the resetting knob of the indicator to energize the clutch relay and disengage the clutch whenever the master direction indicator gyro is caged. The clutch relay is located in the right-hand forward equipment compartment.

6-95. TURN-AND-BANK INDICATOR SYSTEM.

6-96. DESCRIPTION. The turn-and-bank indicator system includes a turn and bank indicator, a filter, a vacuum relief valve, and a vacuum regulator valve. Vacuum to drive the indicator is maintained by a vacuum (pressure) pump which is installed on the engine supercharger rear cover. The pump serves in both the turn-and-bank indicator system, and the anti-G system. When the pump is in operation, vacuum created by the pump is directed through a line to the turn-and-bank indicator to drive the indicator gyro unit. The vacuum regulator valve and vacuum relief valve are installed in the line between the pump and indicator to maintain a vacuum of 1.8 to 2.5 inches of mercury upon the indicator for all varying conditions of atmospheric pressure.

6-96A. TURN-AND-BANK INDICATOR.

6-96B. DESCRIPTION. (See figure 6-1.) The turn-and-bank indicator, located on the center portion of the pilot's instrument panel, consists of a vacuum driven gyro rate of turn indicator, and a ball-type inclinometer. The turn indicator gyro unit is driven by the suction of air from the indicator case, by the vacuum pump. The operating vacuum pressure in the turn-and-bank indicator is 1.8 to 2.5 inches of mercury. An air filter assembly is installed in the indicator air intake port. The bank indicator consists of a curved glass tube filled with liquid and containing a ball.

6-96C. VACUUM REGULATOR VALVE.

6-96D. DESCRIPTION. The vacuum regulator valve controls the amount of vacuum in the line between the vacuum relief valve and the turn-and-bank indicator. It maintains a relatively constant vacuum upon the turn-and-bank indicator under all conditions of air density, atmospheric pressure and air flow. When the vacuum system is inoperative, the pressure within the valve body is equal to atmospheric pressure and the valve is kept closed by the compression spring. When the vacuum system is functioning, the pressure in the valve body is reduced below atmospheric pressure, the valve disc is forced only slightly off the valve seat and air is admitted directly into the valve body.

6–96E. REMOVAL.

a. Open access door to forward equipment compartment for access to vacuum regulator valve.

b. Disconnect lines from regulator valve.

c. Remove regulator valve attaching screws, nuts, washers, and spacers and remove regulator valve.

6–96F. INSTALLATION.

a. Place regulator valve in position against support and secure valve to support with attaching screws, nuts, washers and spacers.

b. Connect lines to regulator valve.

6–96G. ADJUSTMENT. To adjust the vacuum regulator valve the check nut is unlocked and the adjusting screw is turned clockwise for increased vacuum and counterclockwise for decreased vacuum.

Note

It is important that the check nut be locked after this operation to avoid the possibility of an accidental change in the adjustment.

6–96H. VACUUM RELIEF VALVE.

6–96J. DESCRIPTION. The vacuum relief valve is located in the forward equipment compartment, and is installed in the vacuum line between the air pump and vacuum regulator valve. The vacuum relief valve controls the negative pressure exerted by the pump on the vacuum regulator valve.

6–96K. REMOVAL.

a. Open access door to forward equipment compartment for access to vacuum relief valve.

b. Disconnect lines from vacuum relief valve.

c. Remove vacuum relief valve attaching screws, nuts, washers, and remove vacuum relief valve.

6–96L. INSTALLATION.

a. Place vacuum relief valve in position against support and secure with attaching screws, nuts, and washers.

b. Connect lines to vacuum relief valve.

6–97. LANDING GEAR AND WING FLAP POSITION INDICATING SYSTEM.

6–98. DESCRIPTION. The landing gear and wing flap position indicating system includes the following components:

Name	*Para Ref*
Indicator	6–100
Landing gear locked position switch	6–102
Landing gear position switch	6–107
Tail gear position and warning light switch	6–112
Landing gear position warning light circuit	6–116
Landing gear warning light switch	6–120
Landing gear warning light transfer switch	6–125
Landing gear position warning light	6–127
Wing flap position transmitter	6–129

6–99. The landing gear and wing flap position indicating system receives power from the primary bus through a five-ampere circuit breaker designated LDG GEAR & FLAP IND.

6–100. LANDING GEAR AND FLAP POSITION INDICATOR.

6–101. DESCRIPTION. (See figure 6–1.) The landing gear and flap position indicator is mounted in the lower left-hand portion of the pilot's instrument panel. The position of the landing gear is indicated on the instrument dial by place cards which are "tumbled" into position by coils energized through the landing gear position switches to indicate the locked up and locked down positions of the gear, and the "OFF" condition of the instrument. The position of the wing flaps is shown on the indicator by a selsyn-type receiving mechanism.

6–102. LANDING GEAR LOCKED POSITION SWITCH.

6–103. DESCRIPTION. A main landing gear locked position switch is mounted on the upper lock linkage of each main landing gear assembly. When the main land-

ing gear is locked in the down position, the plunger of each lock switch is depressed, completing the indicator circuits from the landing gear position switch to ground.

6–104. REMOVAL.

a. Remove switch mounting screws and remove switch from bracket.

b. Disconnect switch wiring at terminal panel.

6–105. INSTALLATION.

a. Connect switch wires to terminal panel. (Refer to section X for circuit diagram.)

b. Fasten switch to mounting bracket with mounting screws.

6–106. ADJUSTMENT.

a. Safety dive brake control handle in cockpit in "CLOSE."

b. Jack airplane until main wheels are clear of ground.

c. Measure length of locked position switch plungers when landing gear is in down position.

d. Place landing gear approximately 30 degrees from full-down position and measure length of plungers.

e. Adjust switches so that plungers are depressed 3/16 inch minimum when landing gear is in full-down position.

6–107. LANDING GEAR POSITION SWITCH.

6–108. DESCRIPTION. A landing gear position switch is mounted on a bracket with, and just forward of, a landing gear warning light switch on the truss above each main gear. The plunger of each switch is actuated by a clip mounted on the related main gear outboard drag link. When the gear is retracted, the position switch plunger is depressed which completes the circuit from the "up" section of the landing gear and flap position indicator through the position switch to "ground." When the landing gear is extended, the transfer switch plunger is extended which completes the circuit from the "down" section of the landing gear and flap position indicator through the position switch to the locked position switch. When the landing gear is locked down, the circuit is completed through the locked position switch to ground.

6–109. REMOVAL. Refer to paragraph 6–104.

6–110. INSTALLATION. Refer to paragraph 6–105.

6–111. ADJUSTMENT.

a. Safety dive brake control handle in "CLOSE."

b. Remove main landing gear fairings.

c. Jack airplane until main wheels are clear of ground.

d. At each main landing gear, measure length of landing gear position switch plungers.

e. Fully retract landing gear and measure length of plungers in depressed position.

f. Adjust switches so that plungers are depressed 7/16 inch minimum when landing gear is retracted.

6–112. TAIL GEAR POSITION AND WARNING LIGHT SWITCH.

6–113. DESCRIPTION. The tail gear position and warning light switch is mounted on a bracket at fuselage station 364 bulkhead to the left of the tail gear pivot point. The switch plunger is depressed when the tail gear is extended, completing the tail gear "down" indicator circuit. When the tail gear is retracted, the switch plunger is extended, completing the tail gear "up" indicator circuit.

6–114. REMOVAL. Refer to paragraph 6–104.

6–115. INSTALLATION. Refer to paragraph 6–105.

6–116. LANDING GEAR WARNING LIGHT CIRCUIT.

6–117. DESCRIPTION. The landing gear warning light circuit is provided to give visual warning that the landing gear is not locked in the position indicated by the landing gear control handle. The circuit is completed to illuminate the landing gear warning light when any unit of the landing gear is in motion to extend or retract, when the main landing gear is not locked either in the extended position or the retracted position, or when the tail landing gear is not fully retracted or fully extended.

6–118. The landing gear warning light circuit is controlled by the combined function of the following switches: the landing gear warning light transfer switch, actuated by the landing gear control handle linkage; the landing gear warning light switches and the landing gear locked position switches, actuated by the main gear; and the tail gear position and warning light switch, actuated by the tail landing gear. When a unit of the main landing gear is extended but not locked down and the landing gear control handle is in "WHEELS DOWN," the warning light circuit is completed through the landing gear locked position switch; when a unit of the main landing gear is not fully retracted and the landing gear control handle is in "WHEELS UP," the warning light circuit is completed through the landing gear warning light switch. When the landing gear control handle is either in "WHEELS UP" or "WHEELS DOWN" and the tail gear position does not correspond to the position of the landing gear control handle, the warning light circuit is completed through the tail gear position and warning light switch to illuminate the warning light.

6–119. On airplanes BuNo. 134466 through 134637, 135223 through 135406, and 137492 through 137632, the landing gear position warning light circuit receives power from the primary bus through a five-ampere circuit breaker. On airplanes BuNo. 139606 through 139821, the circuit receives power from both the primary and secondary bus through five-ampere circuit breakers.

6–120. LANDING GEAR WARNING LIGHT SWITCH.

6–121. DESCRIPTION. The landing gear warning light switch is mounted on a bracket with, and just aft of, the landing gear position switch on the truss above each main gear. When the landing gear is in the fully retracted position, the switch plunger is depressed to break the circuit to the landing gear warning light in the landing gear control handle. When the landing gear control handle is in "WHEELS UP," and the landing gear is moving from the down-and-locked position, the switch plunger is extended to complete the circuit to the landing gear warning light in the landing gear control handle. (During downward movement of the main landing gear, with the landing gear control handle in "WHEELS DOWN," the warning light circuit is completed through one side of the landing gear locked position switch.)

6–122. REMOVAL. Refer to paragraph 6–104.

6–123. INSTALLATION. Refer to paragraph 6–105.

6–124. ADJUSTMENT. Refer to paragraph 6–111.

6–125. LANDING GEAR WARNING LIGHT TRANSFER SWITCH.

6–126. DESCRIPTION. The landing gear warning light transfer switch is mounted in the forward equipment compartment and is actuated by the landing gear control valve linkage. The switch is a two-position (landing gear "up" or "down") switch.

Note

After installation of landing gear position warning light transfer switch, make certain that switch is actuated when landing gear control handle is in "WHEELS UP."

6–127. LANDING GEAR WARNING LIGHT.

6–128. DESCRIPTION. The landing gear warning light is installed in the landing gear control handle. The complete warning light assembly consists of a lamp, a lamp socket and a bayonet-type cap.

6–128A. LANDING GEAR FLASHING WARNING SYSTEM.

6–128B. DESCRIPTION. (See figure 10–35A.) The landing gear flashing warning light system is installed on airplane BuNo. 142081, and prior airplanes reworked per BuAer AD/SC No. 652. It provides an additional cockpit warning light that flashes on and off if the pilot neglects to extend the landing gear after lowering the flaps and reducing engine power. Principle components of the system are as follows:

Name	*Para Ref*
Flasher Unit	6–128D
Warning Light	6–128F
Throttle Switch	6–128H
Flap Switch	6–128K

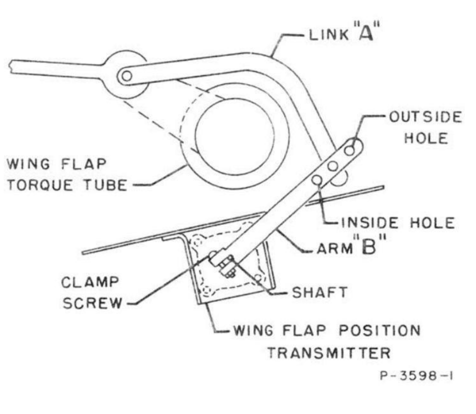
6–128C. Two switches provide the means for energizing the flashing warning light circuit: one switch is actuated by lowering the flaps, and the other by retarding the throttle approximately 90 percent of its normal quadrant travel. When both switches are actuated, power from the primary bus is supplied to the warning light through a flasher unit which causes the light to flash at 100 cpm.

6–128D. FLASHER UNIT.

6–128E. DESCRIPTION. The flasher unit is located in the forward equipment compartment. It is attached at station 110 to a channel on the lower surface of the cockpit floor, approximately six inches to the right of the airplane centerline. The unit is a hermetically sealed, RC oscillator type electronic device which has a contact rating of .5 amperes at 27 volts d-c lampload. A safety feature is incorporated in the unit which causes the warning light to burn steadily in the event of malfunction.

6–128F. WARNING LIGHT.

6–128G. DESCRIPTION. The landing gear flashing warning light assembly is located on the extreme left-hand side of the instrument glare shield. It consists of a push-to-test light assembly which is bracketed and shielded.



ADJUSTMENT

1. Energize electrical circuit.
2. Set flaps in full "up" position.
3. Attach link "A" to center hole of arm "B."
4. Loosen clamp screw on shaft.
5. Adjust shaft until indicator in cockpit shows full "up." Tighten clamp screw on shaft.
6. Operate flaps to full "down" position and check indicator in cockpit.
7. If necessary:

   For increased indication, attach link "A" to inside hole and adjust according to steps 4 through 6 above.

   For decreased indication, attach link "A" to outside hole and adjust according to steps 4 through 6 above.

Figure 6–6. Wing Flap Position Transmitter Adjustment

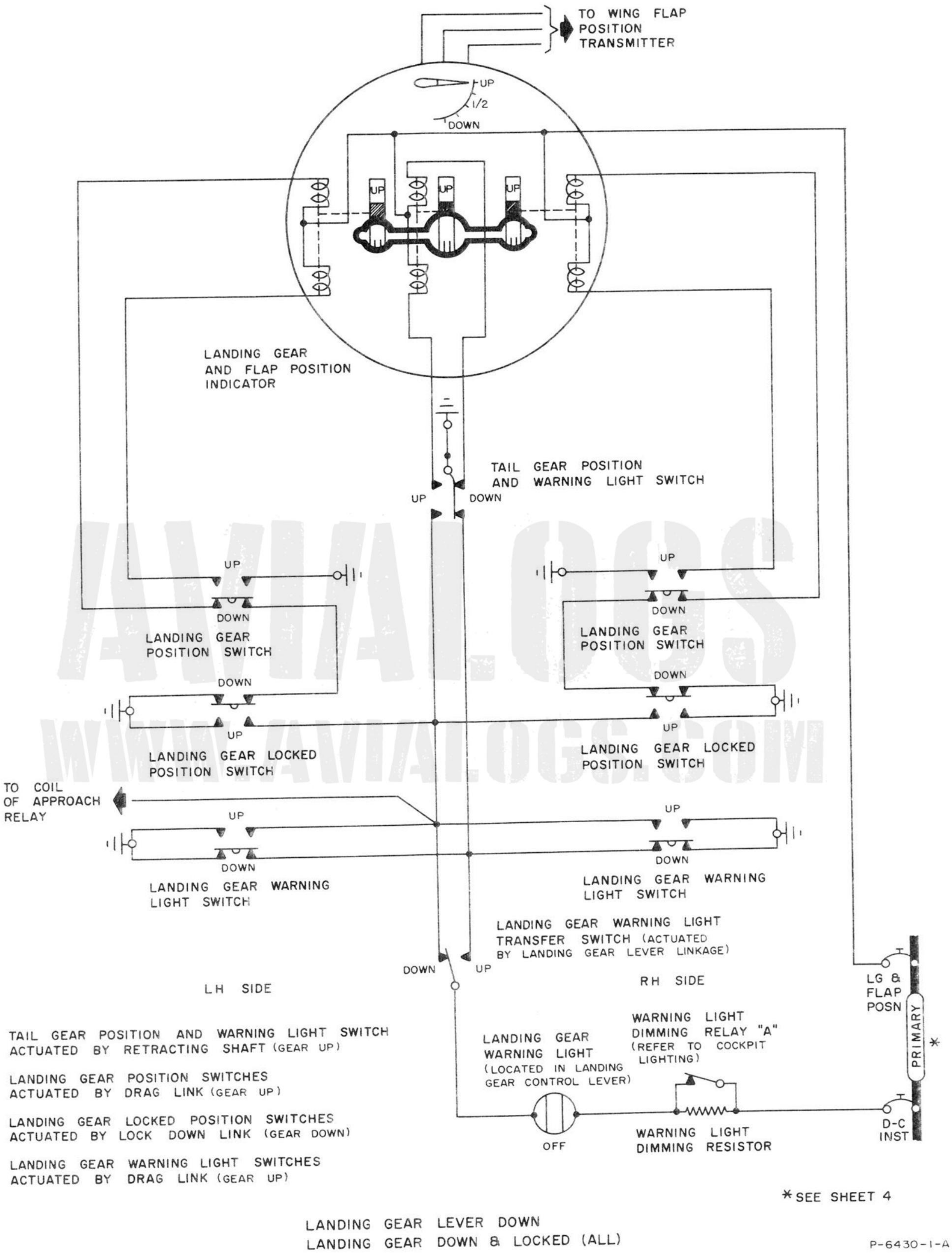


Figure 6–6A. Landing Gear Position and Warning Light Circuit (Sheet 1)

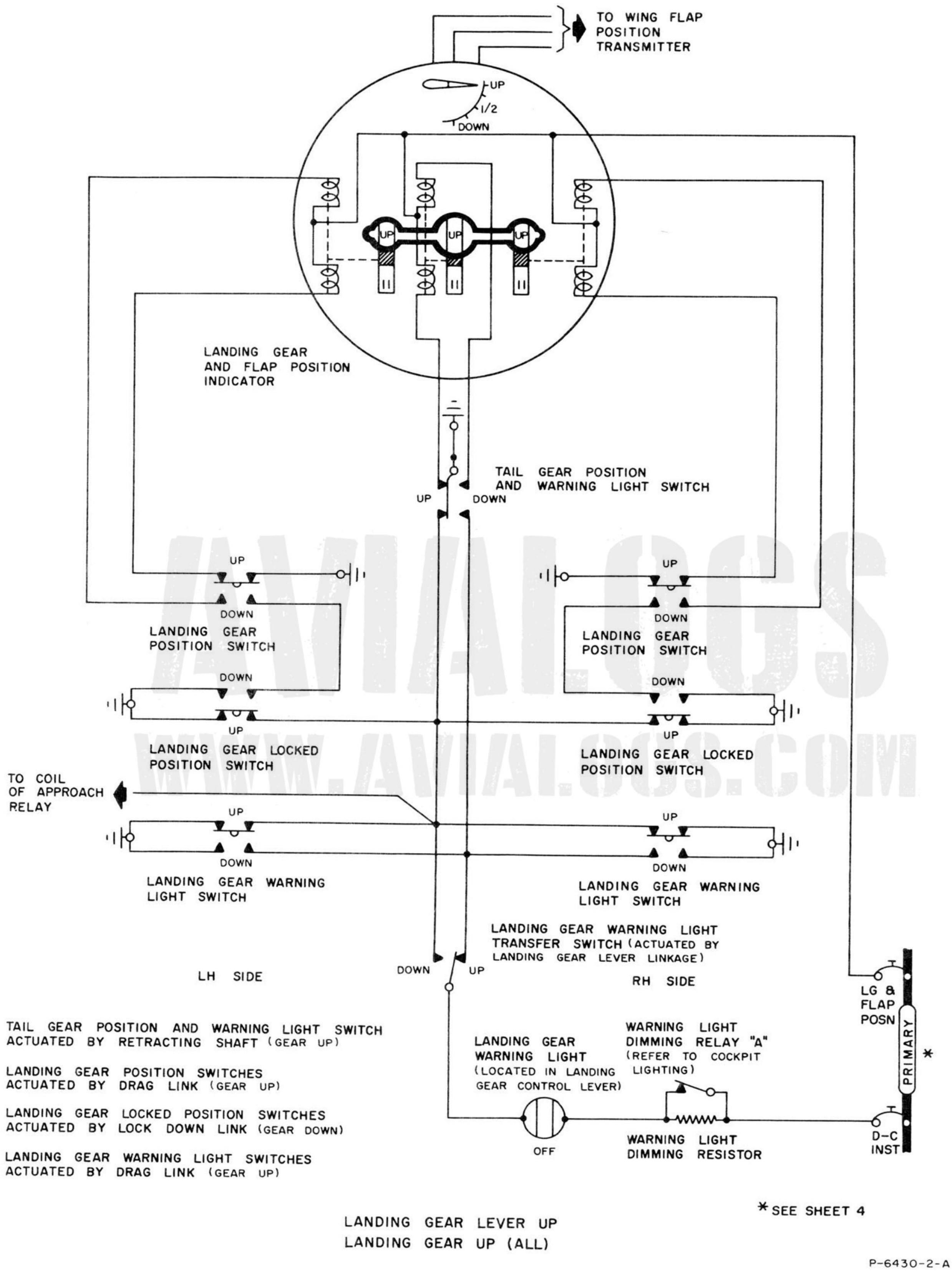
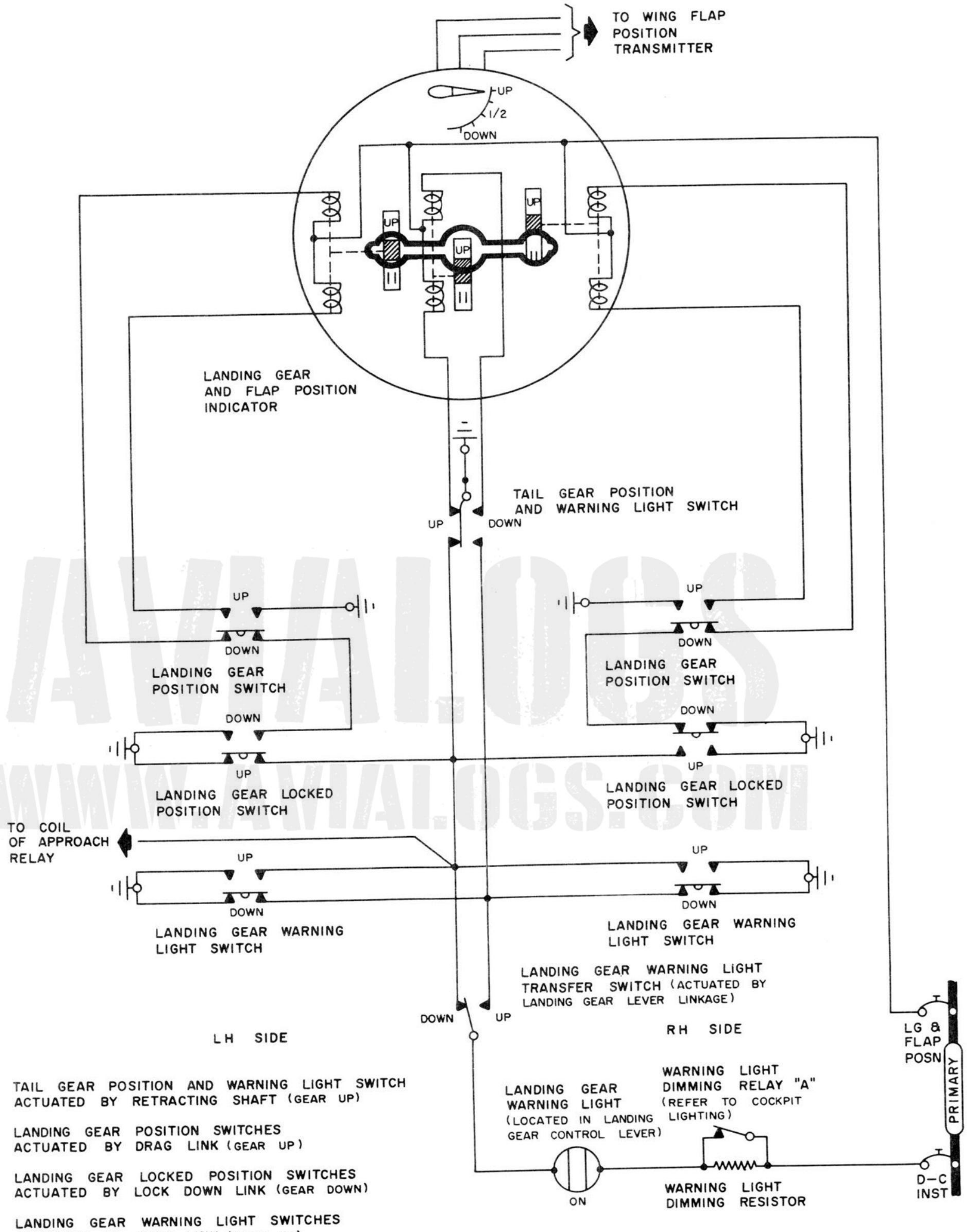


Figure 6–6A. Landing Gear Position and Warning Light Circuit (Sheet 2)



TAIL GEAR POSITION AND WARNING LIGHT SWITCH ACTUATED BY RETRACTING SHAFT (GEAR UP)

LANDING GEAR POSITION SWITCHES ACTUATED BY DRAG LINK (GEAR UP)

LANDING GEAR LOCKED POSITION SWITCHES ACTUATED BY LOCK DOWN LINK (GEAR DOWN)

LANDING GEAR WARNING LIGHT SWITCHES ACTUATED BY DRAG LINK (GEAR UP)

LANDING GEAR LEVER DOWN
RH GEAR DOWN AND LOCKED
LH GEAR BETWEEN UP AND DOWN
TAIL GEAR UP

*SEE SHEET 4

P-6430-3-A

Figure 6–6A. Landing Gear Position and Warning Light Circuit (Sheet 3)

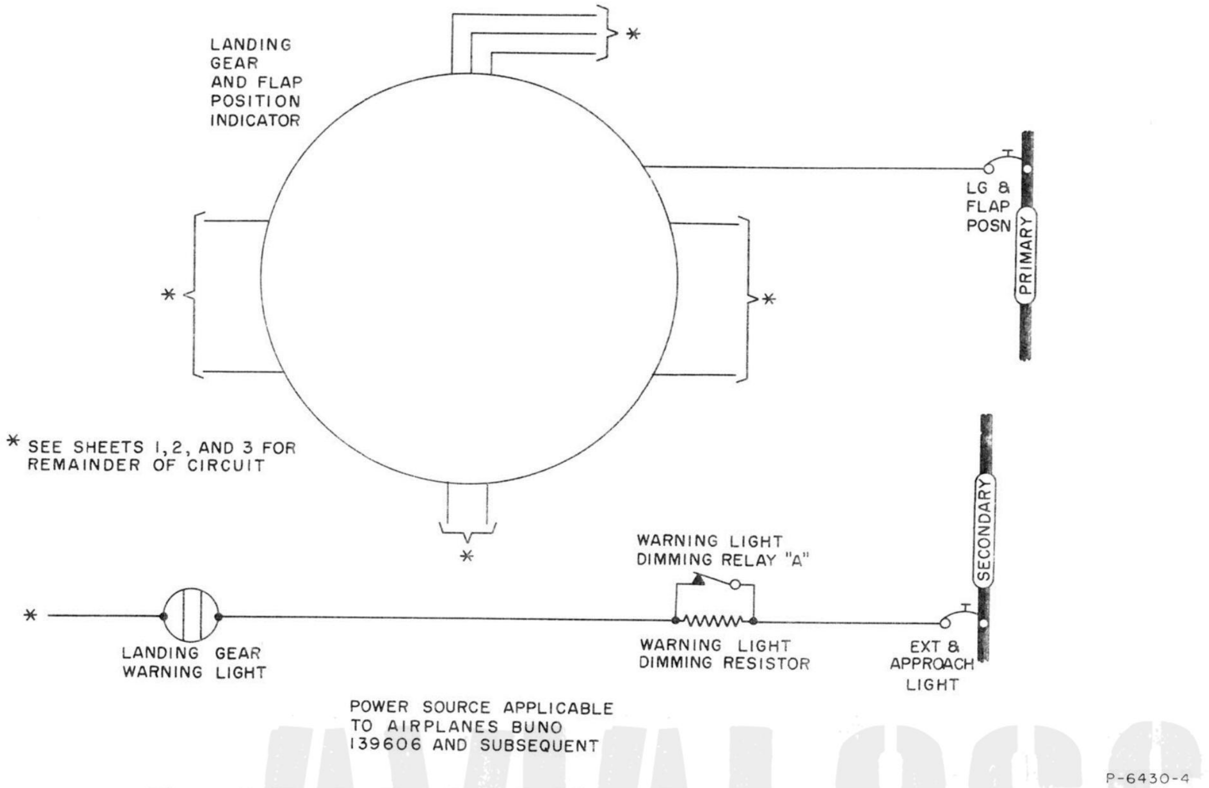


Figure 6–6A. Landing Gear Position and Warning Light Circuit (Sheet 4)

6–128H. THROTTLE SWITCH.

6–128J. DESCRIPTION. The landing gear flashing warning light throttle switch (AN3234-1) is attached to a slotted bracket at the base of the throttle control lever. The switch actuator rides forward on the control quadrant shaft, and if adjusted properly, it will engage the control quadrant shaft when the throttle lever is retarded 90 percent of its normal quadrant travel. Actuation of the switch completes only part of the circuit; the flap switch has to be closed in order to fully complete the circuit.

6–128K. FLAP SWITCH

6–128L. DESCRIPTION. The landing gear flashing warning light flap switch (MS25011-1) is located on the centerline of the airplane underneath the lower radio shelf at fuselage station 195. It is hinged to the flap-actuating-cylinder right-hand channel support, and is adjusted up or down by the aft attaching screw. When the flaps are extended, the flap actuating cylinder lowers. If the flap switch is adjusted properly, the actuating cylinder will engage the switch actuator, and complete the warning light circuit.

6–129. WING FLAP POSITION TRANSMITTER.

6–130. DESCRIPTION. The wing flap position transmitter is mounted on a support at fuselage station 187.625 on the center line of the airplane immediately below the wing flap torque tube. The instrument is a selsyn-type transmitter and is connected to the wing flap torque tube by a link and arm assembly. When the torque tube is rotated to operate the wing flaps, the shaft on the transmitter follows torque tube rotation with a similar rotation, and the transmitter transmits the degree of angular motion of the torque tube to the flaps section of the landing gear and flap position indicator.

6–131. REMOVAL.

a. Disconnect electrical plug from transmitter receptacle.

b. Loosen transmitter control arm clamp screw and slide arm off transmitter shaft.

c. Remove transmitter mounting screws and remove transmitter.

6–132. INSTALLATION.

a. Position transmitter on support and install mounting screws.

b. Apply anti-seize compound (Specification JAN-A-669) to threads of transmitter receptacle. Insert plug into receptacle and tighten knurled nut finger tight.

c. Slip transmitter control arm over transmitter shaft.

d. Adjust transmitter linkage.

6–133. ADJUSTMENT. See figure 6–6.

6–134. AIR TEMPERATURE INDICATING SYSTEM.

6–135. DESCRIPTION. The air temperature indicating system includes the following principal components:

Name	*Para Ref*
Indicator	6–138
Bulbs	6–140 and 6–145
Selector switch	6–150

6–136. The air temperature bulbs and the air temperature indicator constitute a resistance bridge circuit, with the bulbs supplying the variable resistance. Air temperature change varies the resistance of the bulbs, causing a corresponding change in current flow in the indicating circuit. Circuit current-flow changes are reflected on the indicator. One temperature bulb is installed in the trailing edge of the left-hand wing center section to provide free air temperature indications; the other air temperature bulb is installed in the carburetor air scoop to provide carburetor air temperature indications. The circuit is controlled by a double-pole selector switch. The switch has two positions, one position normally closed and one position momentarily closed. The carburetor air temperature bulb is connected through the normally closed position of the selector switch.

6–137. The air temperature indicating circuit receives power from the primary bus through a five-ampere circuit breaker.

6–138. AIR TEMPERATURE INDICATOR.

6–139. DESCRIPTION. (See figure 6–1.) The air temperature indicator is mounted in the extreme lower right-hand corner of the pilot's instrument panel. The indicator houses a meter, to indicate the direction of current flowing in the bridge circuit, and three fixed resistances of the system bridge circuit. The indicator is calibrated in degrees centigrade through a range of —70° to + 150°C (—90° to + 302°F).

6–140. CARBURETOR AIR TEMPERATURE BULB.

6–141. DESCRIPTION. The carburetor air temperature bulb is installed in the carburetor air scoop. The bulb serves as a variable-resistance arm for the system bridge circuit. The resistance of the bulb varies proportionately with a change in temperature, thereby causing deflection of the pointer on the air temperature indicator dial.

6–142. TESTING.

a. Remove carburetor air scoop.

b. Disconnect electrical plug from bulb.

c. Connect leads of accurate ohmmeter to bulb pins. Use low resistance range on ohmmeter: resistance should measure close to 100 ohms.

6–143. REMOVAL.

a. Remove carburetor air scoop.

b. Disconnect electrical plug from bulb.

c. Remove bulb from threaded hose in carburetor air scoop.

6–144. INSTALLATION.

a. Install bulb in threaded boss in carburetor air scoop.

b. Apply anti-seize compound (Specification JAN-A-669) to bulb receptacle threads and install electrical plug.

c. Install carburetor air scoop.

6–145. FREE AIR TEMPERATURE BULB.

6–146. DESCRIPTION. The free air temperature bulb is mounted in the trailing edge of the left-hand wing center section, approximately at wing station 109, between the rear shear web and the wing flap. The bulb serves as a variable-resistance arm for the system bridge circuit. The resistance of the bulb varies proportionately with a change in temperature, thereby causing deflection of the pointer on the air temperature indicator dial. The free air temperature bulb is connected into the indicating circuit only when the air temperature selector switch is pressed to its momentarily closed position.

6–147. TESTING.

a. Raise wing flaps.

b. Remove bulb adapter plate mounting screws and pull bulb away from wing structure.

c. Disconnect electrical plug from bulb.

d. Connect leads of accurate ohmmeter to bulb pins. Use low resistance range on ohmmeter: resistance should measure close to 100 ohms.

6–148. REMOVAL.

a. Raise wing flaps.

b. Remove bulb adapter plate mounting screws and pull bulb away from wing structure.

c. Disconnect electrical plug from bulb.

d. Remove nut holding bulb to adapter plate.

6–149. INSTALLATION.

a. Install adapter plate on bulb and tighten retaining nut.

b. Apply anti-seize compound (Specification JAN-A-669) to bulb receptacle threads and install electrical plug.

c. Push bulb assembly up through hole in plating and install adapter plate mounting screws.

6–150. AIR TEMPERATURE SELECTOR SWITCH.

6–151. DESCRIPTION. The air temperature selector switch is mounted on the pilot's instrument panel adjacent to the air temperature indicator. The switch is a two-position switch with one position normally closed. In the normally closed position, the switch completes the circuit from the carburetor air temperature bulb to the air temperature indicator. Outside air temperature readings can be obtained by pressing the selector switch to the momentarily closed position to complete the circuit from the free air temperature bulb to the air temperature indicator.

6–152. ACCELEROMETER.

6–153. DESCRIPTION. (See figure 6–1.) The accelerometer, mounted on the upper left-hand portion of the pilot's instrument panel, is a self-contained instrument with a dial calibrated in G units so that the reading can be interpreted as units of acceleration and deceleration.

CAUTION

Internal components of the accelerometer are subject to internal damages and misalignment during transporting or handling due to uncontrolled movement of their mass components.

6-153A. To prevent damage from occurring to the accelerometer the following detailed instructions as outlined in Avionics Bulletin 50 dated 19 June 1964 must be adhered to:

a. The accelerometer must be locked when prepared for shipment or removed from airplane. Care should be exercised when unlocking during installation.

b. Personnel performing repair, inspection or assembly of accelerometers must lock accelerometers prior to release for shipment.

c. Operational personnel must return any accelerometer received in an unlocked position to the cognizant repair activity from which it was received, with necessary FUR information citing Electronic Bulletin 50 as authority.

6-154. STANDBY COMPASS.

6-155. DESCRIPTION. (See figure 6-1.) The standby compass is installed on a bracket mounted on the right-hand side of the cockpit glare shield, immediately above the pilot's instrument panel. The standby compass is a short-period, direct-reading, magnetic compass which indicates heading of the airplane with reference to the earth's magnetic field. The standby compass is intended for use only when the G-2 compass system is inoperative. A standby compass deviation correction card is located in a retainer just below the cockpit right-hand rail. A light provided for the compass is controlled from a single-pole, single-throw switch in the right-hand control panel. Removal and installation are obvious.

NOTE

Compensation of the standby compass must be accomplished whenever the MK 20 MOD 4 gunsight has been removed and re-installed.

6-156. ADJUSTMENT (COMPENSATION). The standby compass should be compensated whenever the airplane is swung for compensation of the G-2 compass system. Refer to paragraph 6-45 for general procedures. Standby compass deviations should be corrected for compass deviations current in the locality in which compensation is made, and final compensations should be recorded on the standby compass deviation correction card.

6-157. EIGHT-DAY CLOCK.

6-158. DESCRIPTION. (See figure 6-1.) The eight-day clock is mounted on the right-hand side of the pilot's instrument panel. The instrument is a standard eight-day clock with a twelve-hour dial and a sweep second hand. A winding and setting knob is located on the lower left-hand corner of the clock.

6-159. ELAPSED TIME CLOCK.

6-160. DESCRIPTION. The elapsed-time clock is held in bond with the airplane's loose item equipment and is used as an interchangeable installation with the eight-day clock. The elapsed-time clock is provided with minute, hour, and sweep second hands. Successive depressions of the knob mounted on the face of the clock will cause the hands to (1) start, (2) stop, (3) return to zero simultaneously. The clock dial is graduated into 60 units. Minute figures are marked on the outer circumference of the graduations and hour figures are designated on the inner circumference of the graduations. The clock is spring wound by turning the knob of the face of the frame clockwise. Duration of operation under one winding is 36 hours.

6-161. ENGINE INSTRUMENTS AND INSTRUMENT SYSTEMS.

6-162. DESCRIPTION. The engine instruments and instrument systems include the following:

Tachometer system
Engine gage unit
Oil temperature bulb
Cylinder head temperature indicating system
Manifold pressure gage
Main fuel quantity indicating system
External fuel quantity indicating system
Engine torque pressure indicating system
Magnetic chip detector warning system.

6-163. TACHOMETER SYSTEM.

6-164. DESCRIPTION. The principal components of the tachometer system are an indicator and a generat generator.

6-165. TACHOMETER INDICATOR.

6-166. DESCRIPTION. (See figure 6-1.) The tachometer indicator is mounted on the left-hand side of the pilot's instrument panel. The instrument indicates revolutions per minute of the engine crankshaft. There are two pointers on the indicator dial: the long pointer indicates engine speed in increments of 10 rpm and completes one revolution for every 1000 rpm engine speed; the short pointer indicates engine speed in increments of 1000 rpm. The tachometer indicator has a range of 0 to 3500 rpm.

6-167. TACHOMETER GENERATOR.

6-168. DESCRIPTION. The tachometer generator is installed in the engine accessory compartment on the right-hand side of the supercharger rear housing. The frequency of the three-phase output of the tachometer generator is directly proportional to engine rpm. As engine speed changes, the output frequency of the generator follows, causing a corresponding change in the tachometer indicator.

6-169. ENGINE GAGE UNIT.

6-170. DESCRIPTION. (See figure 6-1.) The engine gage unit is mounted on the right-hand side of the pilot's instrument panel. The unit incorporates three instruments: an oil pressure gage (engine rear bank), an oil temperature indicator, and a fuel pressure gage. The oil pressure gage section of the engine gage unit indicates the pressure of the oil delivered from the oil pressure control valve to the rear cylinder bank of the engine. The dial is calibrated in pounds per square inch, from 0 to 200. The oil temperature indicator section of the unit indicates the temperature of the oil entering the engine. The dial is calibrated in degrees, from -70° to $+150^{\circ}$ C (-94° to $+ 302^{\circ}$F). The indicator operates in conjunction with an oil temperature bulb in the oil inlet line. The fuel pressure indicator section of the unit indicates the pressure of the unmetered fuel section of the carburetor air intake. The dial is calibrated in pounds per square inch, from 0 to 25. The engine gage unit receives power from the d-c primary bus through a five-ampere circuit breaker.

6-171. OIL TEMPERATURE BULB.

6-172. DESCRIPTION. The oil (inlet) temperature bulb is located in the engine rear oil sump. The bulb screws into a boss on the left-hand side of the oil sump casting. The resistance of the bulb varies proportionately with the change in temperature of the inlet oil, thereby causing a calibrated deflection of the engine gage unit oil temperature indicator.

6-173. TESTING.

a. Remove lower left-hand panel of engine accessory cowling.

b. Disconnect flexible conduit plug from receptacle on oil temperature bulb.

c. Connect leads of a special calibrated low-range accurate ohmmeter across oil temperature bulb contacts. Low resistance range of ohmmeter should be used, and resistance of bulb should measure close to 100 ohms.

6-174. REMOVAL.

a. Remove lower left-hand panel of engine accessory cowling.

b. Disconnect flexible conduit plug from receptacle on oil temperature bulb.

c. Remove lockwire securing oil temperature bulb to oil sump mounting.

d. Apply wrench to bulb hex fitting and unscrew bulb from oil sump.

6-175. INSTALLATION.

a. Screw oil temperature bulb into mounting on oil sump; tighten and secure oil temperature bulb with lockwire (MS20995N25) to oil sump mounting.

b. Apply antiseize compound (Fed. Spec. TT-A-580) to oil temperature bulb receptacle threads; insert conduit plug into receptacle and tighten knurled nut.

6-176. ENGINE CYLINDER HEAD TEMPERATURE INDICATING SYSTEM.

6-177. DESCRIPTION. The engine cylinder head temperature indicating system includes the following parts:

Temperature indicator
Indicator thermocouple leads
Thermocouple lead connector
Indicator thermocouple resistor
Temperature Indicator thermocouple.

6-178. TEMPERATURE INDICATOR.

6-179. DESCRIPTION. The temperature indicator is mounted on the right-hand side of the pilot's instrument panel. The instrument indicates in degrees centigrade the temperature of the No. 10 cylinder, on Model A-1H and No. 2 cylinder on Model A-1J Airplanes. The indicator is calibrated from -50° to $+300^{\circ}$C (-58° to $+572^{\circ}$F).

6-180. INDICATOR THERMOCOUPLE LEADS.

6-181. DESCRIPTION. Two indicator thermocouple leads are connected between the engine cylinder temperature thermocouple and the temperature indicator on the pilot's instrument panel. The yellow lead contains the indicator thermocouple resistor.

6-182. THERMOCOUPLE LEAD CONNECTOR.

6-183. DESCRIPTION. The thermocouple lead connector is mounted on the upper left-hand side of the fire-wall at fuselage station 78. The large pin of the plug is connected to the iron (black wire) lead to the indicator thermocouple resistor; the small pin is connected to the constantan (yellow wire) lead.

NOTE

A rubber gasket should be inserted between the thermocouple lead connector and the fire-wall to prevent gas leakage.

6-184. INDICATOR THERMOCOUPLE RESISTOR.

6-185. DESCRIPTION. The indicator thermocouple resistor is of the spool-type, and is installed on the upper right-hand engine mount. The resistor is adjusted so that a total resistance of 8 ± 0.05 ohms exists in the complete thermocouple circuit.

6-186. ADJUSTMENT. (See figure 6-7.) The resistance of a new resistor is more than the amount required by the circuit. When installed, a new resistor should be adjusted as follows:

a. Place cowl flap control switch in **OFF** position.

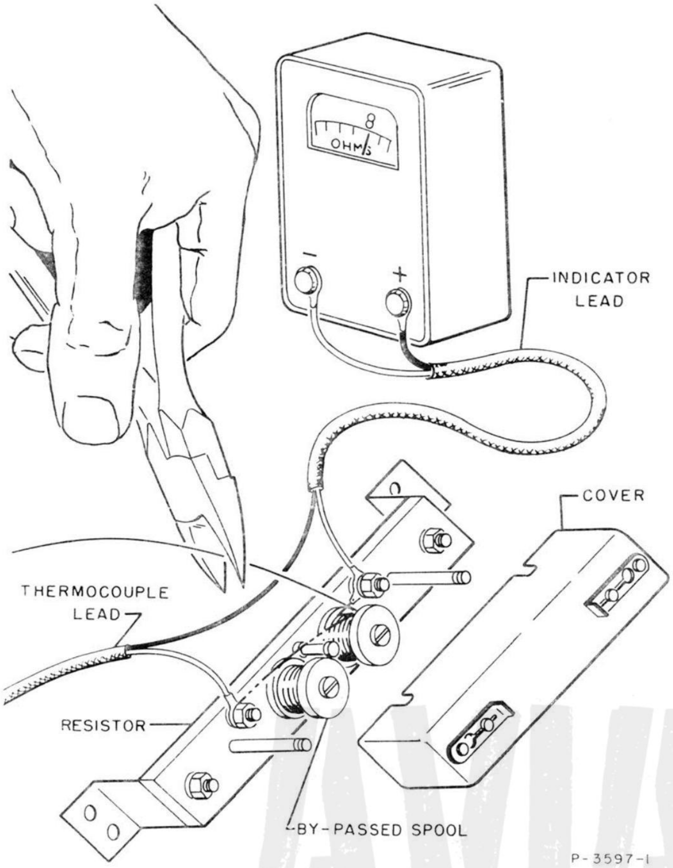


Figure 6-7. Engine Cylinder-Head Temperature Indicator Thermocouple Resistor Adjustment

b. Through right-hand instrument access panel, disconnect thermocouple lead connectors from engine cylinder head temperature indicator. Connect leads to accurate ohmmeter.

c. Open snap slides on resistor case and remove cover.

NOTE

Only one of two spools is connected with circuit. Make certain that connected wire does not make contact with bypassed spool.

d. Disconnect spool wire from resistor terminal and cut off and discard short length of wire. Reconnect to terminal and note ohmmeter (low range) reading. Repeat as necessary until reading of 8 ± 0.05 ohms is obtained.

e. Replace resistor cover.

f. Disconnect thermocouple leads from ohmmeter and reconnect leads to engine cylinder head temperature indicator: black wire to positive terminal and yellow wire to negative terminal.

6-187. TEMPERATURE INDICATOR THERMOCOUPLE.

6-188. DESCRIPTION. The temperature indicator thermocouple, with thermocouple lead, is connected to cylinder No. 10, on Model A-1H, and cylinder No. 2, on Model A-1J Airplanes.

CAUTION

When installing thermocouple in engine, care must be taken not to bend thermocouple case. If case is bent, thermocouple will be shorted and circuit will become inoperative.

6-189. MANIFOLD PRESSURE GAGE.

6-190. DESCRIPTION. (See figure 6-1.) The manifold pressure gage is mounted in the left-hand side of the pilot's instrument panel. The instrument indicates manifold pressure of the engine and is calibrated in inches of mercury from 10 to 75 inches absolute. Manifold pressure is directed from the manifold section of the engine to the gage through a line which attaches between the supercharger rear housing and the manifold pressure gage.

6-191. MAIN FUEL QUANTITY INDICATING SYSTEM.

6-192. DESCRIPTION. The main fuel quantity indicating system is an electronic capacitance system provided to indicate the quantity of fuel in the airplane fuel tanks. For detailed information concerning the fuel quantity indicating system and system components, refer to section V of this manual.

6-192A. EXTERNAL FUEL QUANTITY INDICATING SYSTEM.

6-192B. DESCRIPTION. The external fuel quantity indicating system is an uncompensated capacitor-type indicating system in which three tank probes wired in parallel form one leg of a self-balanced capacitive bridge circuit. For detailed information concerning the external fuel quantity indicating system and system components, refer to Section V of this manual.

6-192C. ENGINE TORQUE PRESSURE INDICATING SYSTEM.

6-192D. DESCRIPTION. The engine torque pressure indicating system provides a direct indication of the power delivered to the propeller shaft in terms of torque pressure. For detailed information concerning the engine torque pressure indicating system and system components, refer to Section V of this manual.

6-192E. MAGNETIC CHIP DETECTOR WARNING SYSTEM.

6-192F. DESCRIPTION. The magnetic chip detector warning system is an electrical warning system provided to indicate the presence of metal chips in the oil system. For detailed information concerning the magnetic chip detector system and system components, refer to Section V of this manual.

6-192G. TORQUE PRESSURE INDICATOR.

6-192H. DESCRIPTION. (See figure 6-1.) The torque pressure indicator denotes the amount of

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power delivered to the propeller shaft in terms of torque oil pressure. Essentially it is a single, servoed, autosyn-type indicator, hermetically sealed within a cylindrical case and installed on the lower right-hand side of the instrument panel. Electrical connections to the components are made through a single receptacle at the end of the case opposite from the dial face. A large pointer revolves about an axis in the center of the circular dial which is calibrated in increments of ten from 50 to 350 psi torque pressure. A small pointer near the top of the dial rotates through an arc of 360 degrees to provide a vernier-type indication on a sub-scale. The sub-scale is divided into single increments from 0 to 10 psi pressure and the pointer makes one complete revolution for each increment covered by the large pointer. Both pointers are actuated through a gear train by a servo motor in the indicator.

6-192J. The stator of the indicator autosyn is connected in parallel to the torque pressure transmitter synchro stator. Thus, when an electrical misalignment occurs between the respective rotors, voltage is induced in the transmitter synchro stator and transmitted to the indicator autosyn stator to align its rotor. Concurrently voltage is supplied from the indicator autosyn rotor through an amplifier to energize the servo motor and as the indicator rotor moves to align electrically with the transmitter rotor, the servo motor positions the indicator pointers on the dial. When the rotors are aligned electrically, voltage is no longer induced in the stators. This de-energizes the servo motor and the pointers are retained in position to denote the torque oil pressure. For complete maintenance instructions on this item refer to Handbook Overhaul Instructions, Autosyn Sensitive Pressure Indicators.

6-193. MISCELLANEOUS PRESSURE GAGES.

6-194. DESCRIPTION. The several pressure gages which are components of various operating systems are listed in table 6-4.

6-195. IN-FLIGHT FUELING SYSTEM CONTROL CONSOLE. For information concerning the control console, refer to the in-flight fueling system paragraphs in Section IV.

TABLE 6-4. MISCELLANEOUS PRESSURE GAGES

System	Manual Section Ref	Gage Name	Type	Pressure Range (Psi)	Location in Airplane
Arresting gear	II	Hold-down cylinder air pressure	Air	0-600	Aft of fuselage station 365, LH side
Cockpit sliding enclosure control	II	Air pressure	Air	0-2000	Cockpit, station 134.5, LH side
Hydraulic power supply	III	Hydraulic system	Hydraulic	0-5000	Cockpit, LH control panel
Hydraulic power supply	III	Hydraulic accumulator	Air	0-5000	Station 96 firewall, forward face, LH side

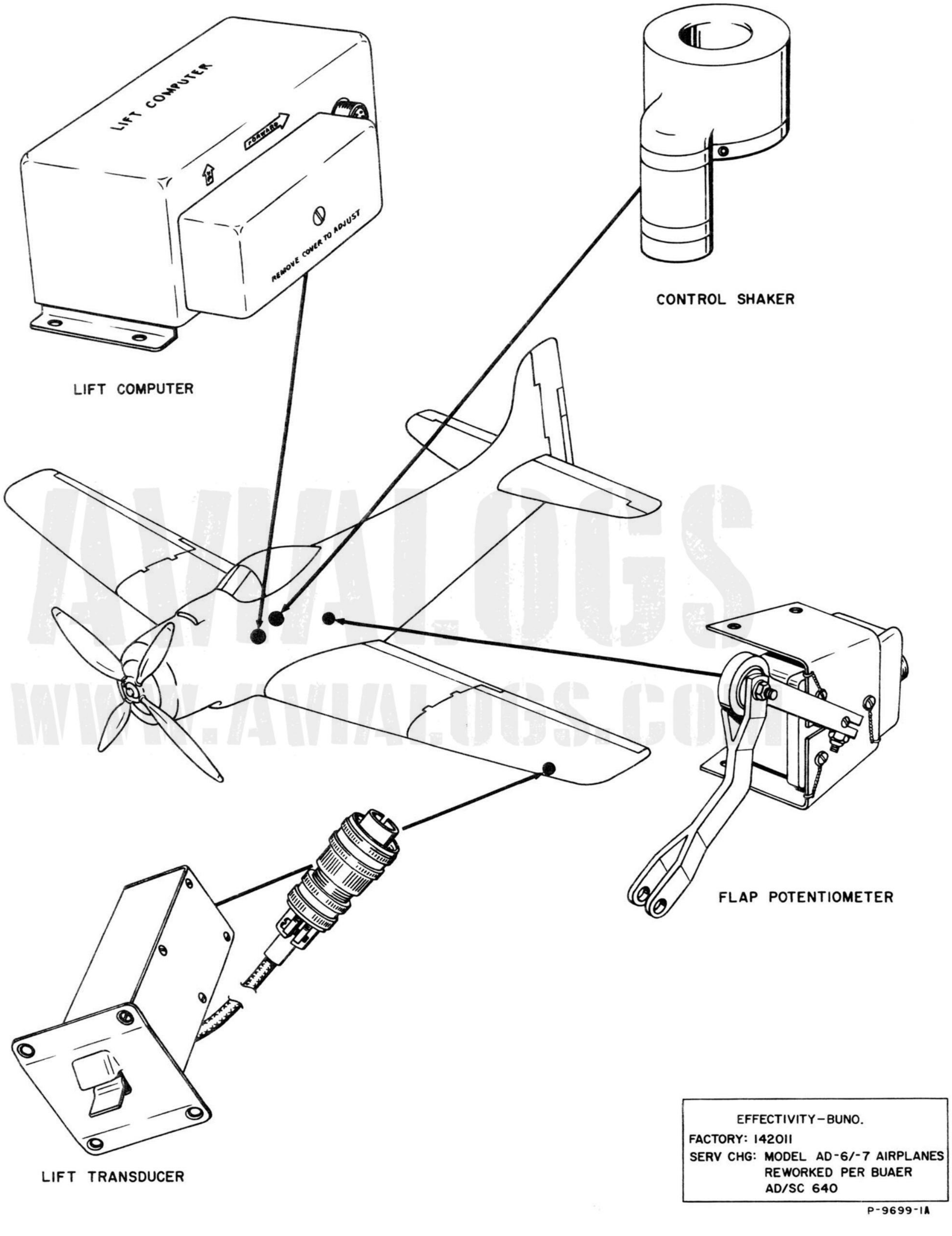


Figure 6–9. Stall Warning System Components

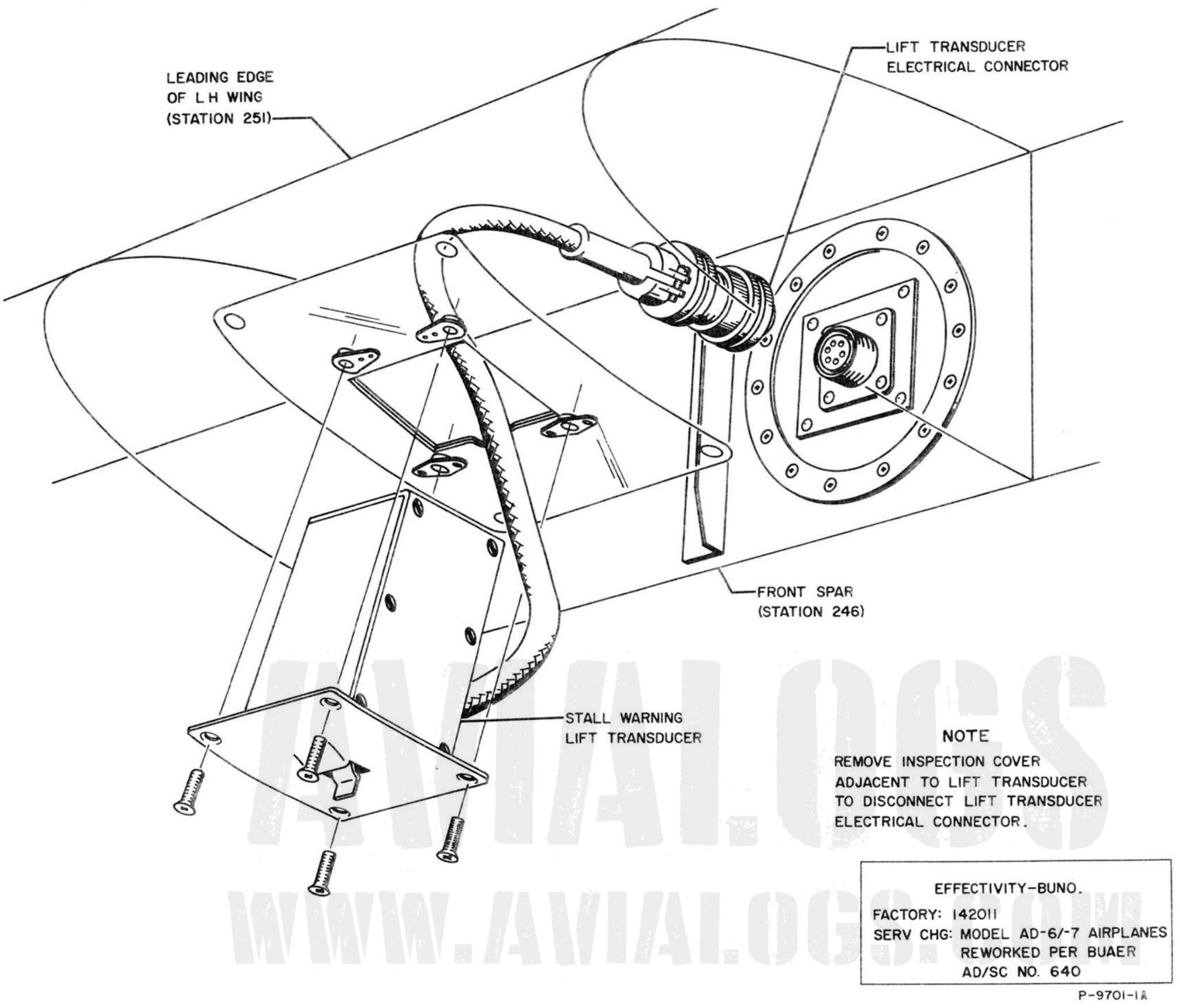


Figure 6–10. Stall Warning Lift Transducer, Removal and Installation

6–196. STALL WARNING SYSTEM.

6–197. GENERAL. (See figure 6–9.) The stall warning system is installed on airplane BuNo. 142011 and model AD-6 and AD-7 airplanes reworked per BuAer AD/SC No. 640. When the airplane approaches a stall, the stall warning system, by means of electrical circuitry, transforms air pressure at the leading edge of the left-hand wing into a physical, vibrating signal at the pilot's control stick. The stall warning system consists of the following major components:

Name	*Location*
Lift Transducer	Wing (LH—Sta 251)
Flap Potentiometer	Flap Torque Tube (fuselage centerline—Sta 187)
Lift Computer	Cockpit (adjacent to LH rudder pedal)
Control Shaker	Pilot's Control Stick

6–198. TROUBLE SHOOTING. Refer to table 6–5.

6–199. ADJUSTMENT. Refer to table 6–6.

6–200. TESTING. Refer to table 6–7.

6–201. LIFT TRANSDUCER.

6–202. DESCRIPTION. (See figure 6–9.) The lift transducer of the stall warning system is an air pressure sensing unit which is installed under the leading edge of the left-hand wing at station 251. The sensing vane of the lift transducer is deflected to a degree proportional to the air pressure hitting the vane. The vane is attached to a spring which operates the armature of a variable reluctance transformer, thus changing the transformer's output proportional to the vane's position. When the airplane is approaching a stall, the transformer's output is strong enough to cause the lift computer circuitry to close the control shaker's circuit. The control shaker, when energized, shakes the pilot's control stick as a warning of the approaching stall.

6–203. REMOVAL. (See figure 6–10.)

a. Remove inspection access adjacent to lift transducer.

b. Remove lift-transducer electrical connector from front spar receptacle.

c. Remove screws and free edges of lift transducer case.

CAUTION

Do NOT pull on sensing vane of lift transducer.

d. Remove transducer from wing.

6–204. INSTALLATION. (See figure 6–10.)

a. Connect lift-transducer connector to receptacle which is located on left-hand wing front spar station 248.

b. Insert transducer into access in leading edge of wing.

c. Secure transducer case to wing with screws.

6–205. FLAP POTENTIOMETER.

6–206. DESCRIPTION. (See figure 6–9.) The Flap Potentiometer is used in the stall warning circuitry to provide an electrical signal which is proportional to the effect the extended flaps have on the stalling speed of the airplane. When the flaps are extended, the flap potentiometer controls the actuation of the control shaker; consequently the control shaker is energized eight to nine knots above the stalling speed of the airplane. In order for the potentiometer to vary according to flap position, the arm of the potentiometer is connected by linkage to the flap torque tube. When the flaps are extended, the arm moves across the resistor changing the output voltage. The output voltage is transmitted to the lift-computer bridge circuitry. The flap potentiometer is attached to the left-hand, flap-actuating-cylinder channel support which is located left of the centerline, between stations 181 and 205. Access to this installation is through the radio compartment access.

6–207. REMOVAL. (See figure 6–11.)

a. Remove electrical connector from receptacle of flap potentiometer.

b. Remove bolt which connects flap-potentiometer link to flap-potentiometer arm.

c. Remove bolts that secure flap-potentiometer support bracket and remove complete assembly from airplane.

6–208. INSTALLATION. (See figure 6–11.)

a. Install flap potentiometer assembly to left-hand side of flap-actuating-cylinder channel support, 6½-

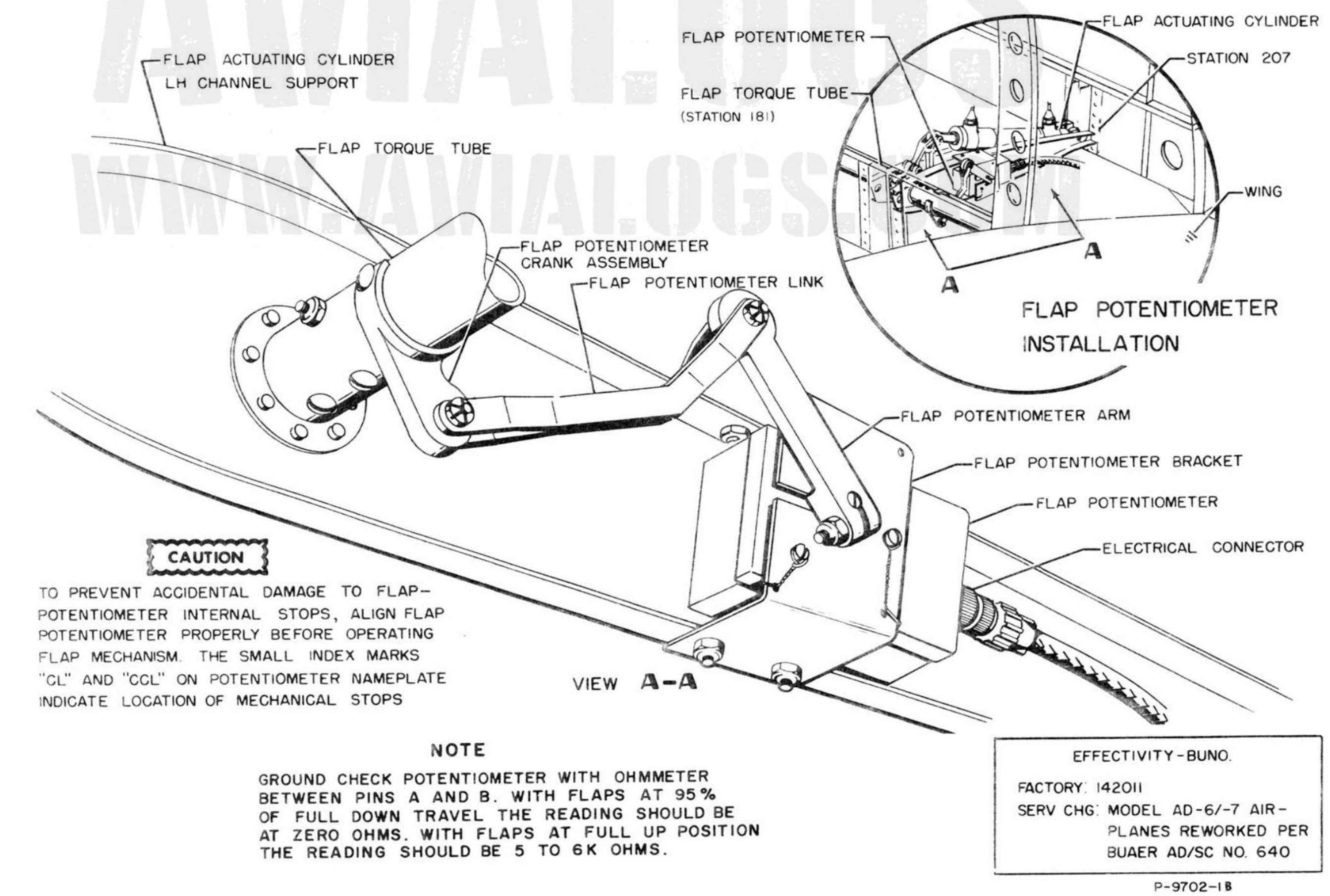


Figure 6-11. Stall Warning Flap Potentiometer, Removal and Installation

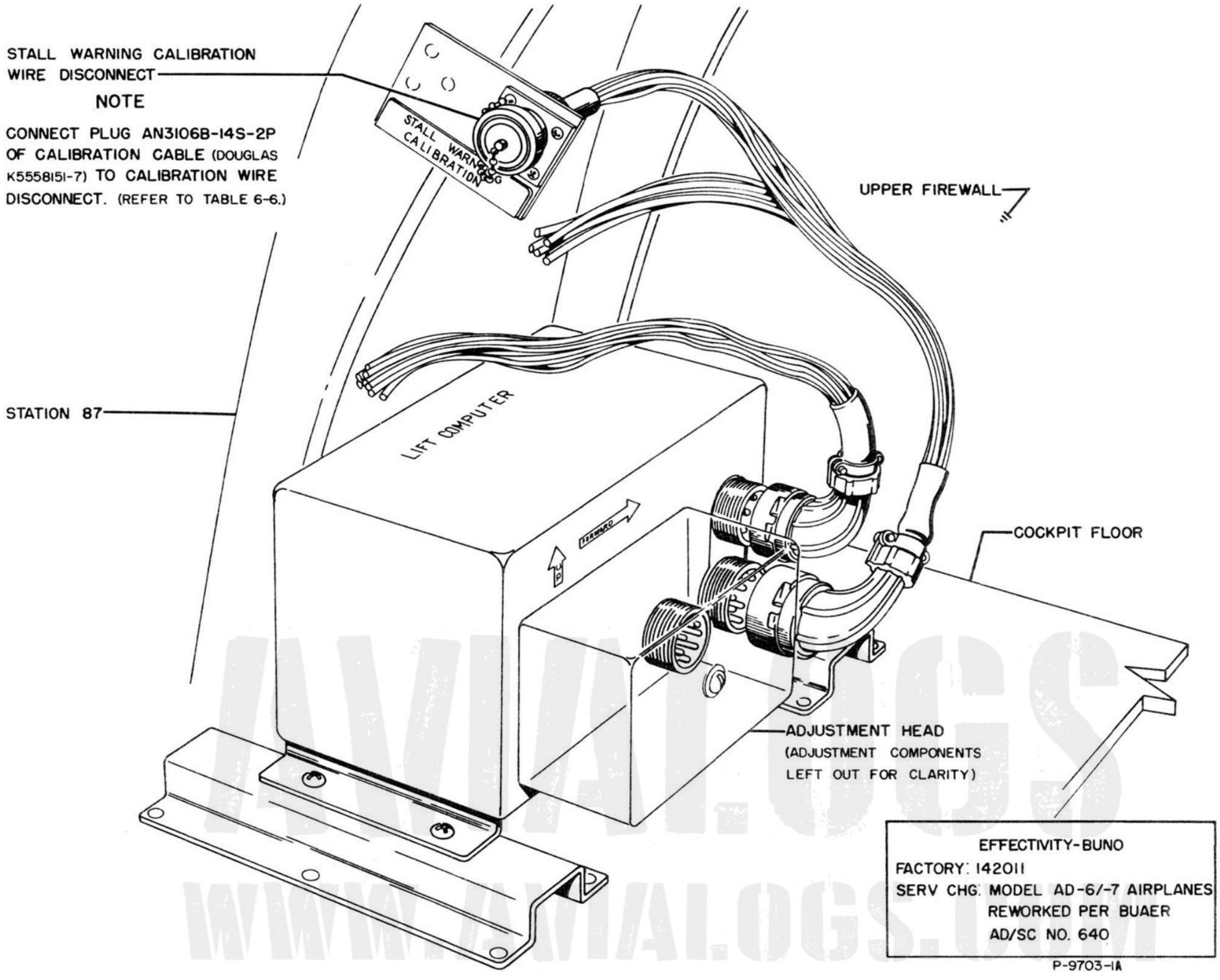


Figure 6–12. Stall Warning Lift Computer, Removal and Installation

inches aft of flap-torque-tube center with linkage facing forward.

b. Connect flap potentiometer link to flap-potentiometer arm.

c. Connect electrical connector to electrical receptacle on flap potentiometer.

6–209. LIFT COMPUTER.

6–210. DESCRIPTION. (See figure 6–9.) The lift computer of the stall warning system measures the signal obtained from the lift-transducer, and by means of bridge type circuits, compensates for effects of aircraft power and flap position. When the signal gains enough potential, the control shaker circuit is completed and transforms the electrical signal into a centrifugal vibration which shakes the pilot's control stick. The lift computer is attached to the left-hand side of the cockpit floor adjacent to the left-hand rudder pedal.

6–211. REMOVAL. (See figure 6–12.)

a. Remove electrical connectors from lift computer.

b. Remove screws securing lift computer to cockpit floor and remove unit from airplane.

c. Install safety screw into underside of lift computer.

6–212. INSTALLATION. (See figure 6–12.)

a. Remove safety screw from underside of lift computer.

CAUTION

Avoid rough handling after removal of safety screw from lift computer.

b. Install lift computer to brackets located on the right side of the cockpit floor at station 85.

c. Install electrical connectors to lift computer receptacles.

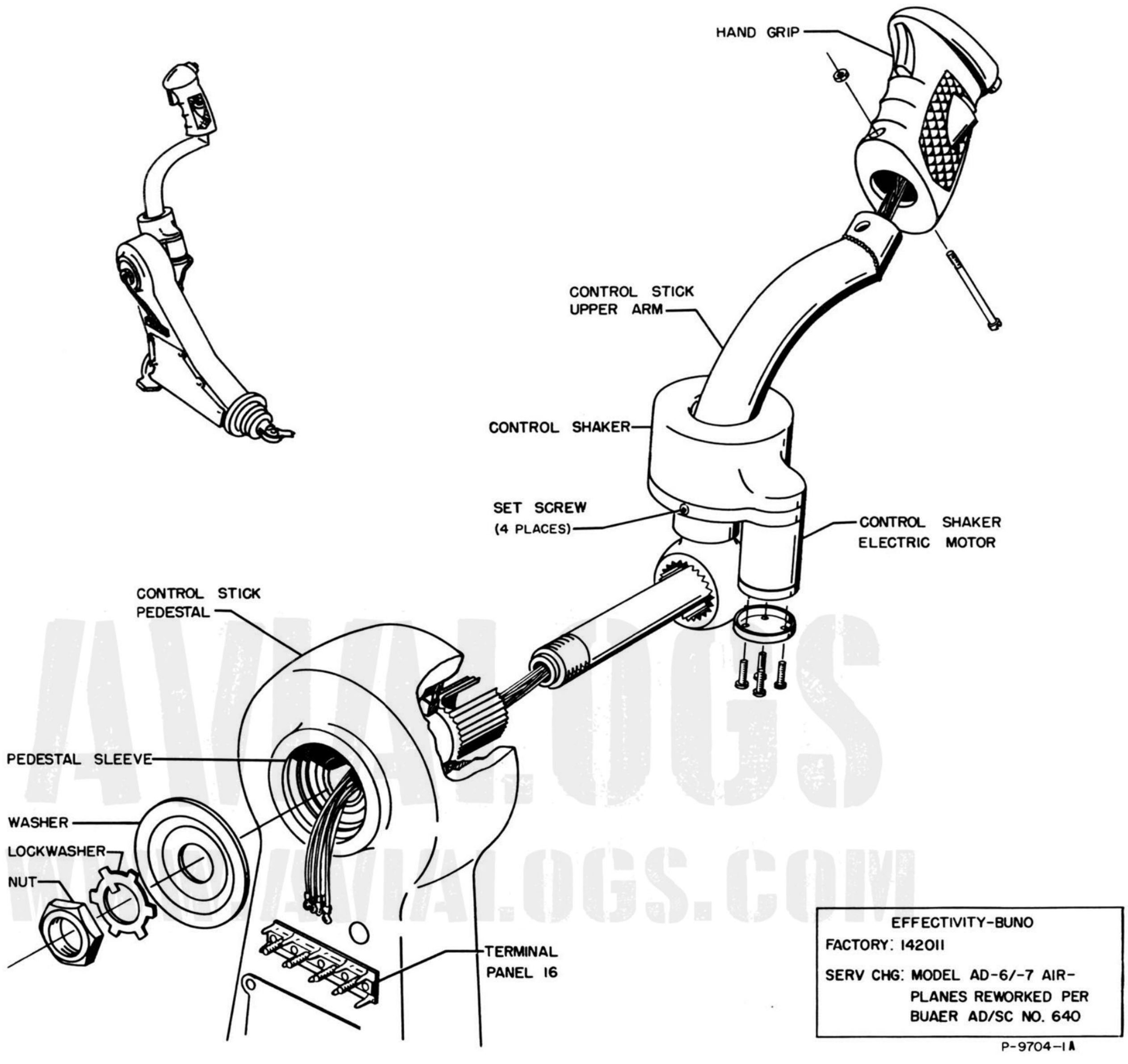


Figure 6–13. Stall Warning Control Shaker, Removal and Installation

6–213. CONTROL SHAKER.

6–214. DESCRIPTION. (See figure 6–9.) The control shaker is a small centrifugal unit which is attached to the base of the upper arm of the pilot's control stick. When the airplane approaches a stall, the motor of the control shaker is energized by the lift computer. The control shaker motor spins a large weight around the control stick and simulates a low frequency, high amplitude buffet signal by trying to pull the stick away from its axis. Access to the control shaker is clear, but removal is accomplished by removing the control-stick hand grip and the control-stick wiring. (Refer to paragraph 6–13.)

6–215. REMOVAL. (See figure 6–13.)

a. Remove terminal guard from terminal panel 16.

b. Disconnect hand grip wiring from terminal panel 16.

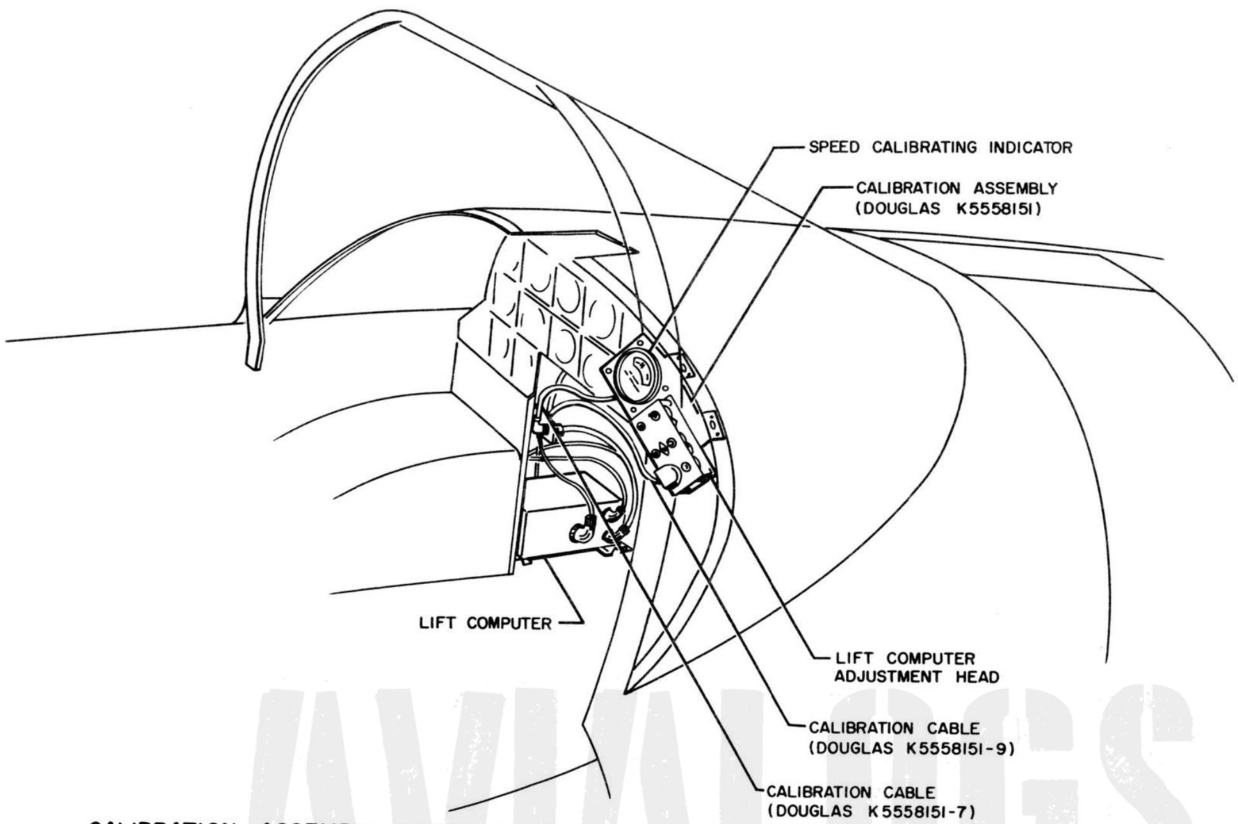
c. Disconnect control shaker wiring by removing shaker motor cover and disconnecting wires from motor terminals.

Note

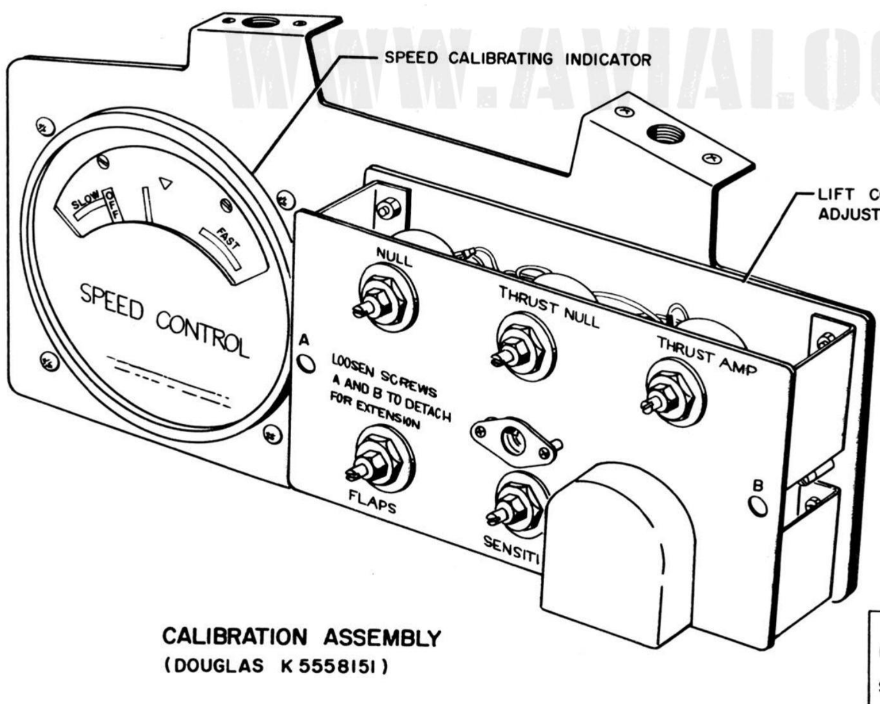
For removal and installation of hand grip on airplanes prior to BuNo. 135344, not reworked per BuAer AD/SC No. 569, refer to section II, paragraph 206.

d. Remove nut, lockwasher, and washer that connect arm assembly to control-stick pedestal.

e. Partially remove arm assembly and wiring from pedestal sleeve.



CALIBRATION ASSEMBLY INSTALLATION



NOTE

REMOVE ADJUSTMENT HEAD FROM LIFT COMPUTER AND INSTALL TO CALIBRATION ASSEMBLY

CALIBRATION ASSEMBLY
(DOUGLAS K5558151)

EFFECTIVITY-BUNO.
FACTORY: 142011
SERV CHG: MODEL AD-6/-7 AIRPLANES REWORKED PER BUAER AD/SC NO. 640

P-9700-1A

Figure 6–14. Stall Warning Adjustment Installation

TABLE 6–5. TROUBLE SHOOTING STALL WARNING SYSTEM

Trouble or Symptom	*Probable Cause*	*Isolation Procedure*	*Remedy*
1. Power failure.	a. 3 ampere fuse.	Turn d-c power control switch on. Check for input of 115 V, 400 cycles at pin L of lift computer.	Replace fuse. (Fuse panel fwd equip compt.) If this does not correct trouble, check number 2 inverter.
	b. 5 ampere circuit breaker.	Check for input of 26 V d-c at pin D of lift computer.	Replace circuit breaker. (Cockpit circuit breaker panel.) If this does not correct trouble, check battery.
2. Control shaker does NOT shake during flight adjustment.	a. Control Shaker	Disconnect control shaker wiring from shaker motor. Apply 27.5 V d-c across its terminal.	If control shaker does NOT shake, replace shaker. (Refer to paragraph 6–213.) If shaker shakes reconnect shaker wiring and proceed with trouble shooting.
	b. Stall Warning Relay. **Note** Turn d-c power control switch off.	Turn on d-c power control switch. Check continuity between pins A1 and A2 of relay.	If there is no continuity between pins A1 and A2, replace relay.
	c. Aircraft wiring between control shaker and lift computer.	See figure 10–34A and perform continuity check.	Replace defective aircraft wiring.
	d. Lift Transducer.	Through inspection access adjacent to lift transducer, disconnect transducer wiring and perform continuity between pins A and B, C and D, and D and E. Attach a gram gage to sensing vane. Record force it requires to lift vane to upper stop. Record force it requires to pull vane against lower stop. The difference between forces recorded should not be more than 2 grams.	If transducer fails either one of these tests, replace transducer. If transducer passes, proceed with trouble shooting.
	e. Aircraft wiring between lift transducer and lift computer.	See figure 10–34A and perform continuity check.	Replace defective wiring.
	f. Lift Computer.	Test lift computer. (Refer to table 6–7.)	Replace defective lift computer.
3. Control shaker continues to shake when flaps are extended.	a. Flap potentiometer linkage.	Visual operation of flap-potentiometer linkage operation.	Replace defective linkage. (See figure 6–11.)
	b. Flap Potentiometer.	Extend flaps, then disconnect plug from flap potentiometer and record resistance across pins A and B.	With flaps at 95% of full down travel the ohmmeter should register zero ohms; at full up position the reading should be 5 to 6K ohms.
	c. Aircraft wiring between flap potentiometer and lift computer.	See figure 10–34A and perform continuity check.	Replace defective wiring.
	d. Lift Computer.	Test lift computer. (Refer to table 6–7.)	Replace defective lift computer.

TABLE 6–6. STALL WARNING SYSTEM ADJUSTMENT

Note

Final adjustments to Stall Warning System are made during flight; therefore the following adjustment procedure should be considered preliminary preparation for flight adjustment of the Stall Warning System.

1. Preliminary Procedure:

a. Level airplane (refer to section I).

b. Remove cover from lift-computer adjustment head by removing screw located in center of cover.

c. Remove adjustment head from lift computer by removing two captive screws A and B.

d. Install adjustment head to calibration assembly (Douglas K5558151).

e. Connect socket AN3101A-18-1S of calibration cable (Douglas K5558151-9) to adjustment head.

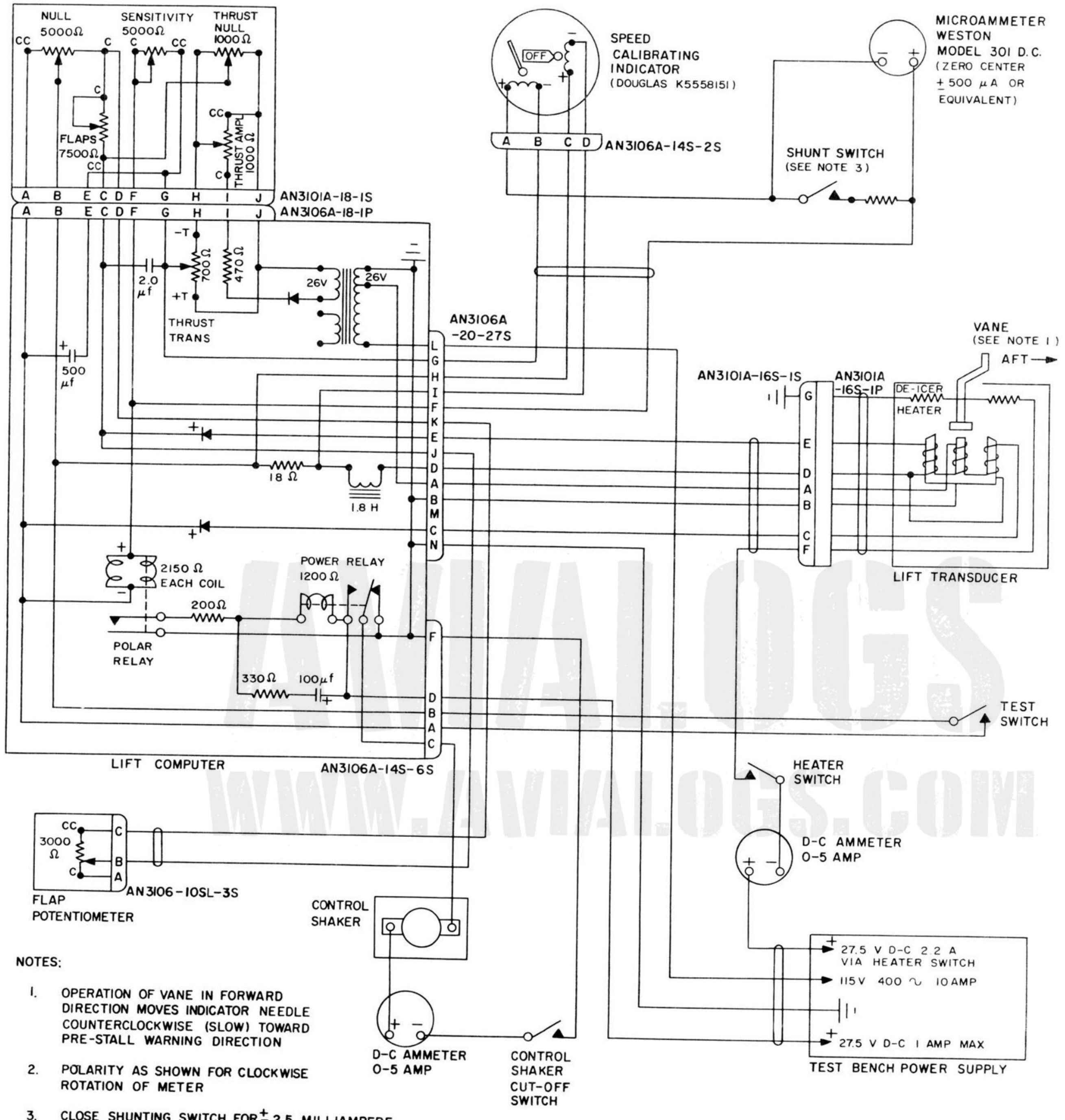
f. Connect plug AN3108B-18-1P of calibration cable (Douglas K5558151-9) to lift computer.

g. Connect plug AN3106B-14S-2P of calibration cable (Douglas K5558151-7) to calibration wire disconnect plug AN3100A-14S-2S which is located left and above lift computer, and attached to former station 87.

h. Connect plug AN3108B-14S-2P of calibration cable (Douglas K5558151-7) to calibration indicator.

i. Install calibration adjustment assembly to right-hand edge of instrument glare shield. (See figure 6–14.)

j. Partially loosen control-shaft lock nuts on adjustment head potentiometers and rotate lock nuts counterclockwise to permit adjustment of potentiometers.



NOTES:

1. OPERATION OF VANE IN FORWARD DIRECTION MOVES INDICATOR NEEDLE COUNTERCLOCKWISE (SLOW) TOWARD PRE-STALL WARNING DIRECTION
2. POLARITY AS SHOWN FOR CLOCKWISE ROTATION OF METER
3. CLOSE SHUNTING SWITCH FOR ± 2.5 MILLIAMPERE RANGE. SELECT RESISTOR FOR REQUIRED SHUNT.
4. ALL INTER-UNIT WIRES TO BE 20 GAGE WIRE
5. FLAP CONDITION POTENTIOMETER
   ELECTRICAL ROTATION – 40 DEGREES
   MECHANICAL ROTATION – 30 DEGREES

EFFECTIVITY-BUNO
FACTORY : 142011

SERV CHG: MODEL AD-6/-7 AIRPLANES REWORKED PER BUAER AD/SC NO. 640

P-9707-1B (7436742-6780-C-SC640)

Figure 6–15. Stall Warning Test Wiring — Schematic Diagram

TABLE 6–6. STALL WARNING SYSTEM ADJUSTMENT (Continued)

k. Set potentiometer control shafts as follows:

Name	*Position*
NULL	Full clockwise
THRUST NULL	Full clockwise
THRUST AMPLITUDE	Full counterclockwise
FLAPS	Full counterclockwise
SENSITIVITY	Full counterclockwise

l. Turn airplane d-c power control switch on. Calibration indicator "OFF" flag should disappear.

Note

Flaps should be fully retracted.

2. Preflight Adjustment Procedure:

Procedure	*Desired Results*
a. Rotate SENSITIVITY control clockwise.	Calibration indicator should deflect clockwise into "fast" zone.
b. Rotate NULL control counterclockwise.	Indicator pointer should deflect counterclockwise into "slow" zone. Control shaker should shake.
c. Adjust NULL control until pointer is at right-hand edge of "slow" amber zone on indicator dial.	Control shaker should stop shaking.
d. Rotate THRUST AMPLITUDE control clockwise.	Indicator pointer should deflect clockwise into "fast" zone.
e. Rotate THRUST NULL control counterclockwise.	Indicator pointer should deflect counterclockwise into "slow" zone. Control shaker should shake.
f. Adjust THRUST NULL control until pointer is at the right-hand edge of "slow" zone.	Control shaker should stop shaking.
g. Rotate THRUST AMPLITUDE control full counterclockwise.	Indicator pointer should remain at right-hand edge of "slow" zone.
h. Rotate FLAPS control full clockwise.	Indicator pointer should deflect counterclockwise into "slow" zone. Control shaker should shake.
i. Extend flaps of airplane to full extended position.	Indicator pointer should move clockwise to approximately right-hand edge of "slow" zone. Control shaker should stop shaking.
j. Fully retract flaps of airplane.	Control shaker should shake.
k. Adjust NULL control to bring indicator pointer under center mark of dial.	Control shaker should stop shaking.
l. Move sensing vane of lift transducer forward.	Control shaker should shake as indicator pointer enters "slow" zone.
m. Move sensing vane of lift transducer aft.	Control shaker should stop shaking as indicator pointer leaves "slow" zone.
n. Turn pitot-heat switch on.	Lift transducer mounting plate should begin to warm-up.
o. Turn airplane pitot-heat switch off.	Lift transducer mounting plate should cool-off.
p. Turn airplane d-c power control switch off.	
q. Return airplane to normal position.	
r. Flight Test and adjust Stall Warning System.	

Note

Upon completion of Stall Warning Adjustment Flight, remove calibration equipment from airplane and restore adjustment head to lift computer.

TABLE 6–7. TESTING PROCEDURE—STALL WARNING SYSTEM

1. Equipment Needed:

a. Microammeter, Weston model 30, (500 microamperes each side of center), or equivalent.

b. A shunt for meter in step a consisting of a calibrated resistor and a single-pole, single-throw switch. Shunt should decrease 500 ua to 2.5 ma.

c. Speed calibrating indicator (Douglas K-5558151).

d. Two d-c ammeters (0–5 amperes).

e. Three single-pole, single-throw switches.

f. Twenty gage wire necessary to connect Stall Warning units and test equipment together. (See figure 6–15.)

2. Preliminary Procedure:

a. Remove Stall Warning System to level bench.

b. Connect stall warning units in accordance with test wiring schematic (see figure 6–15).

c. Remove safety screw from bottom of lift computer.

d. Remove adjustment cover from lift-computer adjustment head exposing adjustment units.

e. Make initial control adjustments as follows:

Name	*Adjustment*
NULL control	Approximately centered
THRUST NULL control	Approximately centered
THRUST AMPLITUDE control	Zero (full counterclockwise)
FLAPS control	Zero (full counterclockwise)
SENSITIVITY control	Maximum (full clockwise)

f. Adjust flap potentiometer to full counterclockwise position.

g. Mount lift transducer in temporary manner with nameplate and vane free.

h. Leave heater cut-off switch open and control shaker cut-off switch open.

i. Apply 115 V, 400 cps a-c and 27.5 V d-c to system (see figure 6–15). (.1 ampere maximum.)

Note

"OFF" flag should disappear from speed calibrating indicator.

(Continued on page 308P)

TABLE 6–7. TESTING PROCEDURE—STALL WARNING SYSTEM (Continued)

3. Test Procedure:

Procedure	*Desired Results*
a. With shunt switch open, adjust NULL control for zero microamperes.	
b. Close shunt switch.	
c. Move vane of lift transducer to upper limit.	Indicator pointers should deflect to left.
d. Record microammeter reading for step c.	Should be minus value in milliamperes.
e. Move vane of lift transducer to lower limit.	Indicator pointers should deflect to right.
f. Record microammeter reading for step e.	Value should be positive and in milliamperes.
g. Find amount of milliamperes between upper limit value in step d and lower limit value in step f. Record this value.	Amount should not exceed 3.0 ± 0.6 milliamperes.
h. Adjust THRUST AMPLITUDE control to maximum, full clockwise.	
i. Open shunt switch and adjust THRUST NULL control until microammeter zeros.	
j. With electrical receptacles of lift computer facing operator, lift left end of lift computer box so that horizontal arrow points down at a minimum of 45 degrees.	Indicator pointers should move to left of zero. Thrust should be 11 +6 —3 percent of value found in step g.
k. Lift right end of lift computer box so that horizontal arrow points upward at a minimum of 45 degrees.	Indicator pointers should deflect to right of zero. Thrust should be 11 +6 —3 percent of value in step g.

Note

Provided readings are made on a level bench, the two readings in steps j and k should be within 50 microamperes of each other.

Procedure	*Desired Results*
l. Turn flap potentiometer arm full clockwise.	
m. Repeat steps j and k.	There should be no change in values found in steps j and k.
n. Adjust THRUST AMPLITUDE control to zero, full counterclockwise.	
o. Close shunt switch (2.5 ma range)	
p. Adjust flap potentiometer arm to maximum, full clockwise.	Indicator pointers should move left of zero. Microammeter should read 17 +10 —5 percent of milliampere value in step g.
q. Adjust flap potentiometer arm to minimum, full counterclockwise.	Indicator pointers should return to zero +0 —5 microamperes.
r. Return flap potentiometer arm to full clockwise position.	There should be no change in value found in step p.

Note

Entire change due to rotation of flap potentiometer takes place over a 40 degree segment of its rotation.

Procedure	*Desired Results*
s. Open test switch.	Indicator pointers should deflect to left. Microammeter should read minus 0.2 milliampere minimum or minus .5 milliampere maximum.
t. Close control-shaker, cut-off switch.	
u. Close test switch.	Control shaker should shake. Maximum current for control shaker is 1.0 amperes.

Note

After test switch is opened, control shaker will continue to operate for a short time while control circuit relay coils are being discharged. Indicator pointers will also show this condition, then slowly return from left to zero.

Procedure	*Desired Results*
v. Reduce d-c voltage input (pin D of six pin connector) to 20 volts d-c, and check that control shaker operates when test switch is closed.	
w. Restore control shaker voltage to 27.5 V d-c.	
x. Open microammeter shunt switch.	
y. Adjust NULL control *very slowly* counterclockwise until control shaker shakes. (Polar relay closed value.)	Microammeter should read between zero microamperes and minus 35 microamperes.
z. Adjust NULL control *very slowly* clockwise until control shaker stops shaking. (Polar relay open value.)	Microammeter should read within 30 microamperes of value found in step y.
aa. Open control-shaker, cut-off switch.	

Note

If desired, an indicator light may be used in place of control shaker.

Procedure	*Desired Results*
ab. Adjust NULL control to bring speed calibrating indicator pointer under center mark.	Microammeter should read 50 ± 10 microamperes.
ac. Adjust NULL to bring speed calibrating indicator pointer to left end of "slow" zone.	Microammeter should read minus 50 ± 10 microamperes.
ad. Adjust NULL control to bring speed calibrating indicator pointer to right end of "fast" zone.	Microammeter should read +150 ± 15 microamperes.
ae. Close heater cut-off switch.	Current in heater should be 2.0 ± 0.2 amperes. Mounting plate of lift transducer should warm to touch in 30 seconds.
af. Open heater cut-off switch.	
ag. Replace cover to adjustment head.	

CAUTION

Replace safety screw to bottom of lift computer. *Avoid* tightening safety screw beyond minimum necessary to take up slack between screw head and computer box. Remove safety screw when installing lift computer into airplane.

Procedure	*Desired Results*
ah. Re-install Stall Warning units to airplane.	
ai. Perform flight test and flight adjust Stall Warning System.	

CAUTION

Before fully removing arm assembly from the pedestal sleeve, mark the spline on the arm assembly and on the pedestal sleeve with a lacquer paint to insure correct re-positioning of arm assembly.

f. Remove terminals from wires and attach a length of cord to end of each wire.

g. Pull hand grip and wires through arm assembly until one-half foot of cord appears.

h. Disconnect wires from cords.

i. Loosen set screws that hold control shaker to arm assembly and lift shaker off of arm assembly.

6–216. INSTALLATION. (See figure 6–13.)

a. Remove hand grip from arm assembly. (Refer to paragraph 6–215.)

b. Remove control-shaker motor cover.

c. Connect control shaker wiring to control-shaker motor terminals.

d. Re-install control-shaker motor cover.

e. Slip control shaker, with motor facing down, over arm assembly, and anchor control shaker to base of arm assembly by aligning one control-shaker set screw with hole in arm assembly and tightening set screw.

f. Tighten remaining control-shaker set screws.

g. Tie ends of hand grip wires to cords protruding from arm assembly and begin to work cords and wires through arm assembly.

h. Install arm assembly to pedestal sleeve, using index marks to align assembly with pedestal sleeve.

i. Install washer, lockwasher, and nut to arm assembly attaching bolt.

j. Install terminals to hand grip wires.

k. Connect hand grip wiring to terminal panel 16.

l. Replace terminal panel guard.