AUTOPILOT FLIGHT CONTROL COMPONENTS - DESCRIPTION AND OPERATION.

1. Surface Servo

A. Three surface servos are used in the autopilot system. They position the rudder, elevator and aileron control tabs in response to control signals, thereby, providing correct positioning of the main control surfaces.

B. Each surface servo consists of a servo motor and rate generator, Autosyn synchro, gear train, accurate electromagnetic clutch, and a special pulley and servo mount. Provisions for adjustable torque limiting are included.

(1) The servo motor is a two-phase induction motor having a rate generator integrally mounted on its top. A torque-limiting resistor and a phase-shifing capacitor for the motor are included inside the servo housing.

(2) The Autosyn position follow-up synchro is geared to the motor output shaft through a series of split gears which connect to the Autosyn rotor shaft, giving a 334:1 speed reduction.

(3) The gear train consists of a differential and two stages of planetary gears. The cage of the differential is spring loaded to the servo housing. Two torque-limit microswitches, one mounted to either side of the cage, are actuated by an arm on the cage. The planetary gearing gives a 27:1 speed reduction. The low-side of the gearing drives the solenoid clutch directly.

(4) The solenoid clutch provides positive on-off action only. Action is obtained in a multi-disc clutch assembly placed between the rotating solenoid clutch and the output shaft. The solenoid clutch has no sliding parts, thereby, eliminating the possibility of binding or jamming which can occur when sliding parts move into contact. Instead, the translating parts are supported on a diaphragm and bellows and only flexing of the diaphragm results when the clutch is engaged or disengaged. The bellows and diaphragm place a spring load between the driving and driven clutch parts for positive, accurate on-off action. The slip clutch is a dry, sintered bronze type comprising five discs. It has an adjustable slip value. This clutch drives the splined output shaft. This shaft is hollow and contains a coaxial shaft that is connected to the input side of the solenoid clutch. The output end of the smaller shaft is hexagonal for attachment of a torque wrench, permitting check readings on the actual servo torque (not servo torque less rigging friction).

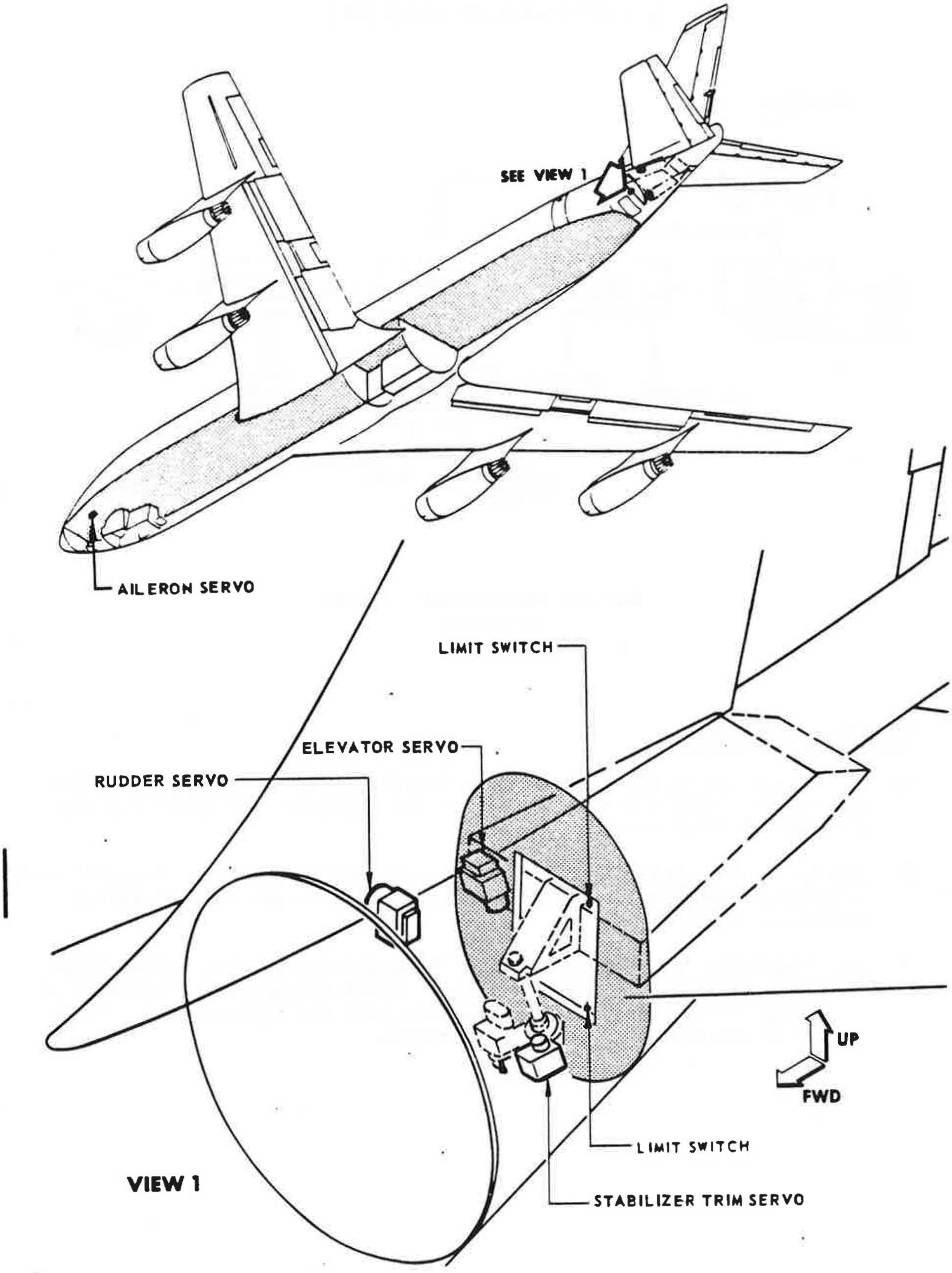
(5) The servo is mounted in the airplane on a special pulley and servo mount assembly by means of four screws. This mounting makes it possible to remove the servo without disturbing the rigging.

(6) Electrical connections are made at a terminal board inside the servo housing. It is readily accessible through an outside cover plate. Connection to the external wiring is made through an electrical connector mounted on the side of the servo housing.

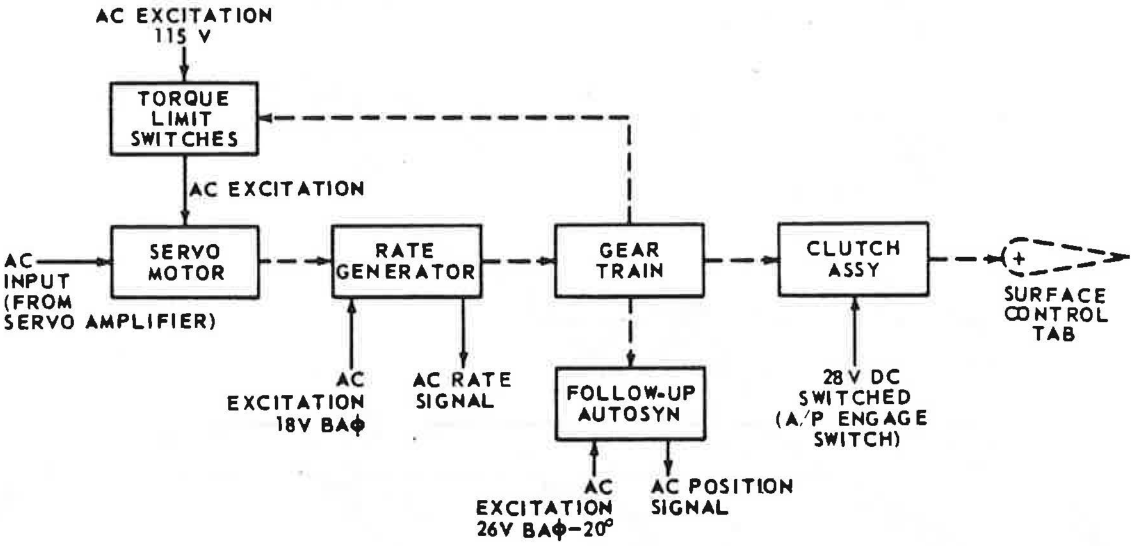
C. The aileron surface servo is located in the lower nose compartment, mounted to a bracket on the underside of the floor beam, approximately two feet to the left of the centerline and just aft of the pilot's control stand aileron cable drum. (See figure 1.) It is readily accessible through the floor hatch beneath the navigator's table in the control cabin, or from the rear of the lower nose compartment. The elevator servo is located in the tail section, approximately six feet aft of the rear cabin pressure bulkhead at floor beam level. It is centered below the stabilizer torque box. (See figure 1.) The elevator servo is accessible through the stabilizer jack-screw access panel on the bottom of the tail section. The rudder servo is located in the tail section two feet below the overhead, above the stabilizer torque box, and approximately six feet aft of the rear cabin pressure bulkhead. (See figure 1.) It also is accessible through the stabilizer jack-screw access panel.

D. In the operation of each surface servo, its motor fixed phase is excited by 115-volt, 400-cycle power from the power junction box. The variable phase is supplied by a type SA-40-A1 magnetic amplifier in the amplifier and computer case. The variable phase voltage is developed from the input control signal and varies from a nominal zero vaue to a maximum of approximately 240 volts. The torque-limiting resistor connected in series with the variable phase reduces the maximum voltage, thereby, lowering servo torque as required for safe airplane operation. A capacitor placed across the variable phase adjusts its phase angle. In response to a control signal the motor drives the rate generator and follow-up Autosyn to provide electrical rate feedback and servo shaft position signals respectively to the servo amplifier. Also, the motor pinion drives the solenoid clutch through the differential and planetary gear train. Torque applied to the differential results in a proportional movement of the cage against the springs. When a preset torque limit is reached, the cage arm will actuate one or the other torque limit switches to remove fixed-phase motor excitation. The switch actuated, depends upon the direction of applied torque. When the solenoid clutch is engaged it drives through the diaphragm and bellows into the multi-disc slip clutch. The disc clutch then drives the splined output shaft which mates with the internal spline of the pulley on the support bracket. The pulley is rigged with control system cables to activate the control surface tabs. Figure 2 gives the surface servo block diagram.

E. Separate protective fusing for servo power is not required, since it is provided in the autopilot power junction box.



Autopilot Flight Control Components Location
Figure 1



Surface Servo Block Diagram
Figure 2

2. Stabilizer Trim Servo

A. The trim servo supplies automatic pitch trim, through actuation of the horizontal stabilizer assembly, to remove sustained out-of-trim loads from the elevator servo.

B. The trim servo consists of a motor, two reversing clutches, a power relay, a direction sensor, a harmonic drive, a clutch brake, and two safety interlock relays.

(1) The motor is a 0.5 horsepower, three-phase, 400-cps, 4-wire motor with thermal overload protection. Speed-torque characteristics at rated voltage and frequency are adequate for both automatic pitch trim and mach trim control functions.

(2) The motor drives into two reversing clutches that are solenoid-actuated. The current to the clutches is controlled by the operation of two safety interlock relays which in turn are actuated by the demodulator amplifier (SA-76).

(3) The output of the two clutches is fed into the differential of the direction sensor. The cage of the differential is spring-loaded to the structure so that if torque is transmitted for either direction of rotation, microswitches are actuated by the movement of the cage. The output of the differential in turn is fed into the harmonic drive.

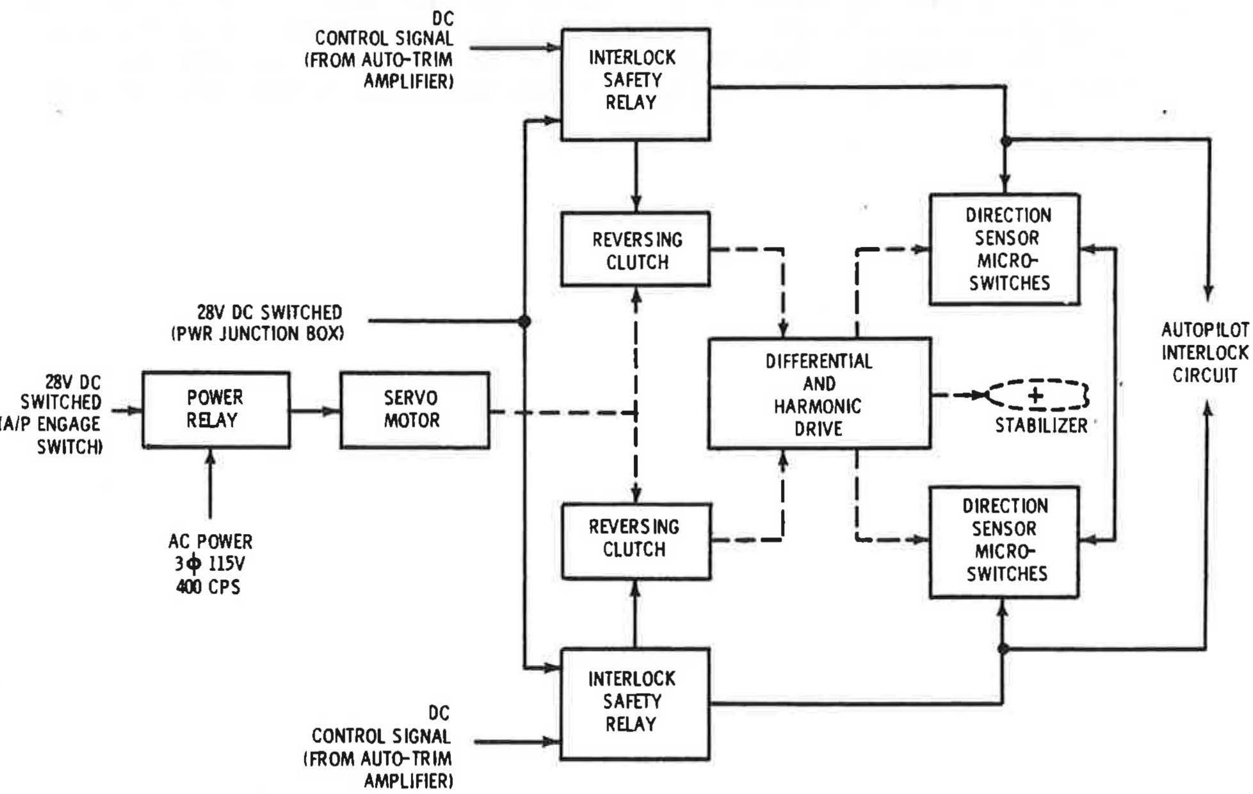
(4) The harmonic drive is a reduction gear train (76:1) consisting of a wave generator, a ball bearing, a flexible gear and an internal gear. The output of the harmonic drive is connected to a metal slip clutch that limits torque. The output of this clutch is the output of the servo.

(5) It is necessary to back-drive the servo output shaft at low torque values (for instance, when manual-electric trim is used). Since back-driving the reversing clutches through the harmonic drive would offer considerable reflected inertia, torque is cut down by clutching the internal gear of the harmonic drive to the structure when driving with the motor, and then unclutching and permitting it to freewheel (similar to that of a planetary gear) when back-driving. The clutch that does this is a solenoid-actuated dog clutch. This clutch has two dogs and works for both directions of rotation.

C. The stabilizer trim servo is located in the tail section aft of the rear cabin pressure bulkhead. It is mounted on the stabilizer ball nut and jackscrew assembly. (See figure 1.) It is accessible through the stabilizer jackscrew access panel on the underside of the tail section.

D. The stabilizer trim servo operates in parallel with the elevator servo. The servo motor, supplied with 115-volt, 400 cps, three-phase power through the power relay, is under constant drive while the autopilot is engaged. Part of the variable-phase power for the elevator servo is fed to a demodulator amplifier (SA-76) in the amplifier and computer package which supplies direct current to control one or the other clutch solenoid. Output of one polarity clutches the output gear for clockwise rotation; output of the opposite polarity clutches for counterclockwise drive. A third clutch, actuated when either of the previous two clutches is engaged, locks internal gear of the harmonic drive to the structure. The harmonic drive provides the necessary speed reduction and connects through the slip clutch to the output shaft. The output shaft drives the stabilizer jackscrew, providing movement of the stabilizer surface to establish the pitch trim required. The safety interlock relays and the microswitches are connected into the autopilot interlock circuit as a safety protection against operational malfunction. A closed circuit is maintained if both clutch solenoids are de-energized or if either is energized. If both solenoids are energized simultaneously, or if there is any type of relay or switch malfunction, an open circuit will occur disengaging the autopilot. A block diagram of this servo is given in figure 3.

E. Trim servo power is furnished from the radio and T-R circuit breaker panel (P5).



Stabilizer Trim Servo Block Diagram
Figure 3

PILOTS CONTROL AND INDICATING COMPONENTS - REMOVAL/INSTALLATION

5. General

A. The Control Panel is designed for installation and removal from the aircraft without the need of special tools. Installation of the unit does not require any special pre-installation checkout, alignment, or adjustment procedures. However, Control Panel, part number 5050351-9, and Digital Amplifier Unit, part number 5050365-9, must be installed as a pair in the aircraft.

6. Removal of Autopilot Control Panel

A. Open and tag AUTOPILOT and A/P & MACH TRIM (DISENG LT) circuit breaker located on P5 overhead circuit breaker panel.

B. Release quarter-turn fasteners at the edge of the panel by pushing in and turning one-quarter turn counterclockwise.

C. Pull panel from mounting space far enough to disconnect electrical connectors. Remove nearby control heads as necessary to gain access to the connectors.

D. Disconnect the electrical connectors.

E. Install conductive end caps on the three rear connectors of the Control Panel.

7. Installation of Autopilot Control Panel

A. To install the Control Panel refer to fig. 1 and proceed as follows:

B. Remove the conductive end caps from the three rear connectors of the Control Panel.

C. Locate Control Panel near panel opening.

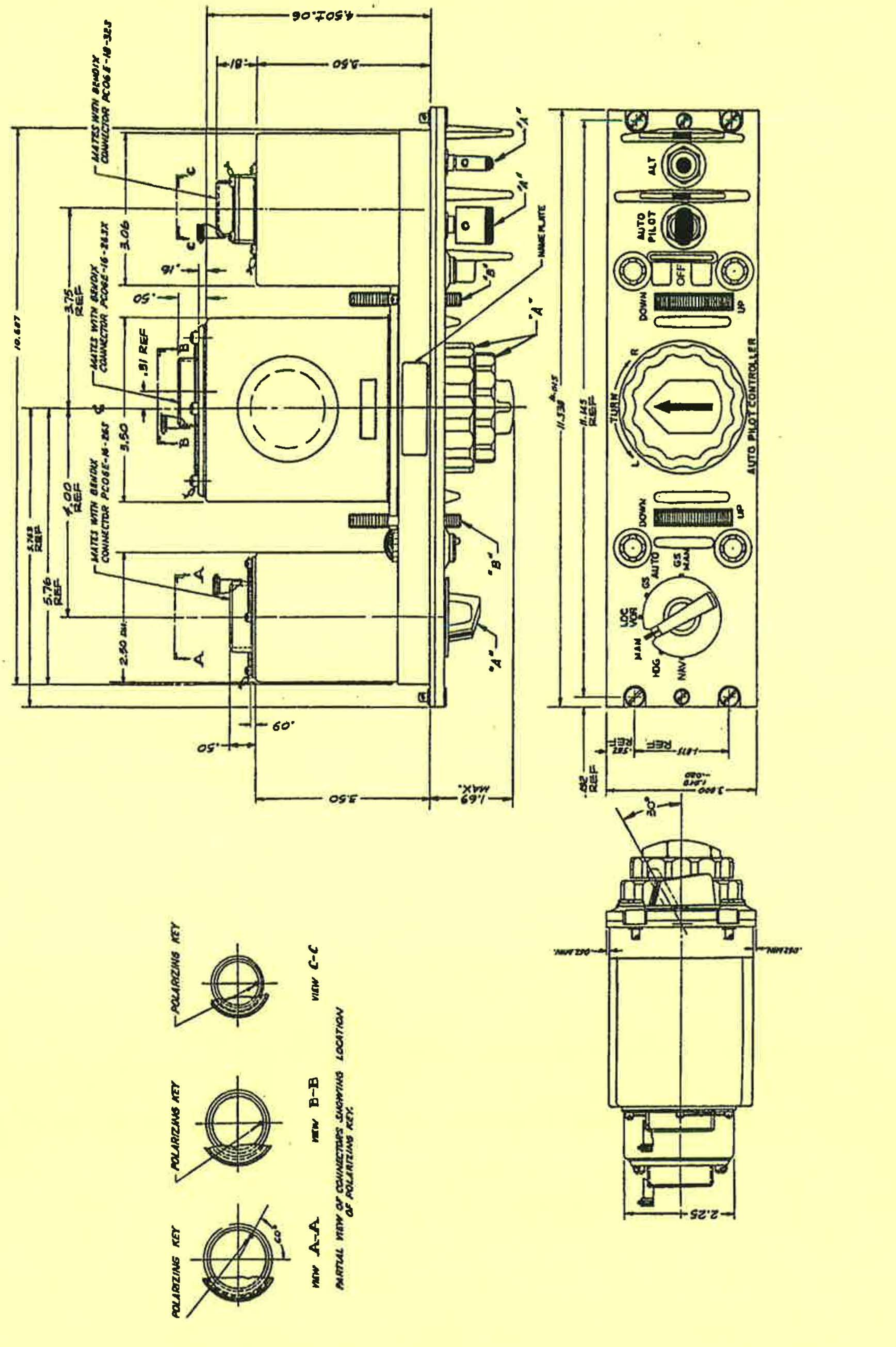
D. Attach mating connectors of aircraft cables to connectors at rear of Control Panel.

E. Place panel in mounting area.

F. Secure quarter-turn fasteners by pushing in and turning one-quarter turn clockwise. Install any nearby control heads removed for access to the control panel connectors.

G. Remove tags and close AUTOPILOT and A/P & MACH TRIM (DISENG LT) circuit breakers located on P5 overhead circuit breaker panel.

IICN: 1P-AMMX-22-0028-01



Control Panel
Fig. 1

8. Test

A. Perform test after installation as follows:

(1) See Chapter 22-42-0 "Autopilot System - Adjustment/Test".

- Step (6) Engage Interlocks and Warning

  (a) Autopilot Engagement

  (b) Autopilot Disengagement and Warning

- Step (9) Turn Knob Interlock

- Step (10) Mode Selector Switch Interlock

- Step (14) Mode Interlocks

  (a) Turn Controller Priority

  (b) HDG Interlock

  (c) Altitude Hold - Autopilot Pitch Wheel Disabling

  (f) NAV Mode - VOR/LOC Interlocks

  (g) GS Auto Interlocks

  (h) GS MAN Interlocks

- Step (16) MAN Mode

  (a) Turn Knob Tests

  (b) Coarse Turn Knob Sensitivity

- Step (18) Clutched Heading

  (c) Pitch Wheel Sensitivity

- Step C. Restore Aircraft to Normal.

END

INTENTIONALLY LEFT BLANK

AUTOPILOT SENSING COMPONENTS - DESCRIPTION AND OPERATION

1. Three-Axis Rate Transmitter

A. The Three-Axis Rate Transmitter senses the rate of change of airplane pitch, roll and yaw attitudes. It provides electrical signals which represent the angular rate in each of these airplane control axes. The roll rate transmitter is not used and is disconnected at the transmitter.

B. The Three-Axis Rate Transmitter is essentially three rate-gyro assemblies each consisting of a three-phase gyro motor with pivots extending from the gimbal housing, an autosyn synchro transmitter mounted above the gimbal pivot, a centering spring and a dashpot damping mechanism. The three gyro assemblies differ only in the way they are mounted and in the position of their spin axes. Each is mounted and in the position of their spin axes. Each is mounted on the base and the entire unit is enclosed by a gasket-sealed, evacuated case. Electrical connections are made through an connector mounted in the base.

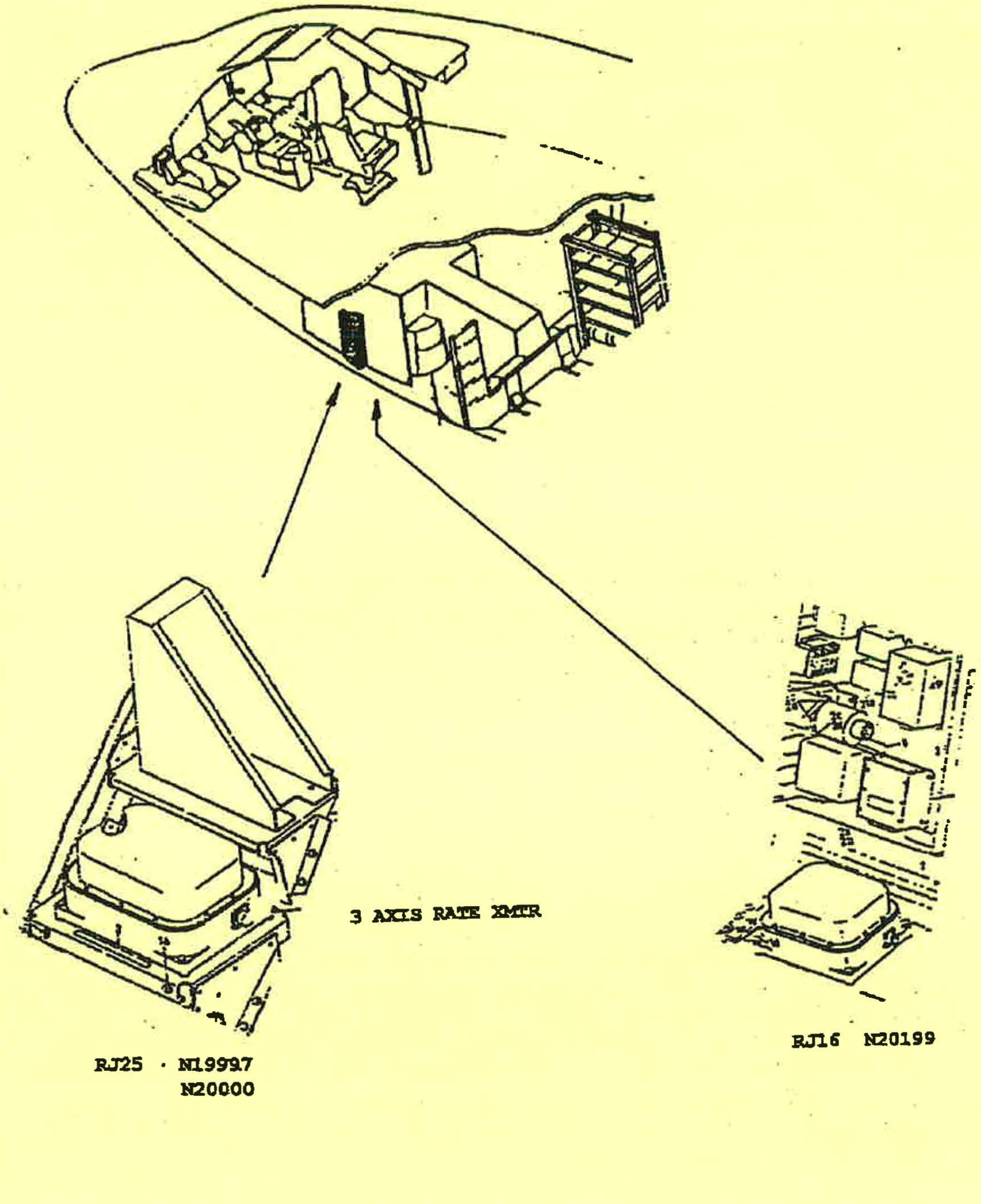
C. The Three-Axis Rate Transmitter is mounted behind a cover on an equipment shelf located at the lower left side of lower nose compartment forward of the left electronic equipment rack (see fig. 1). It is directly accessible through the lower nose compartment.

D. The Three-Axis Rate Transmitter utilizes gyroscopic precession principles for its operation.

(1) A separate gyro is provided for measuring rate of the rotation of the airplane in pitch, roll or yaw. All gyros function in a similar manner. For the roll gyro, as the airplane rolls, the spin axis is rotated in space about an axis perpendicular to the spin axis. This rotation generates forces tending to cause precession about an axis mutually perpendicular to the axis of spin and the roll axis. Forces tending to cause precession are proportional to the angular rate at which the spin axis is changed. Since this torque reacts against a linearly calibrated spring, the angle through which the gyro precesses is proportional to the rate of airplane roll (see fig. 2).

(2) The gyro is mounted in gimbal rings so precession can take place about an axis at right angles to the spin axis. An autosyn signal generator is coupled to the gimbal axis about which precession takes place. Output of the autosyn is then one phase of voltage, zero or a voltage reversed in phase 180 degrees (see fig. 3. for action of the three gyros).

ICN: 1P-AMMX-22-0010-01

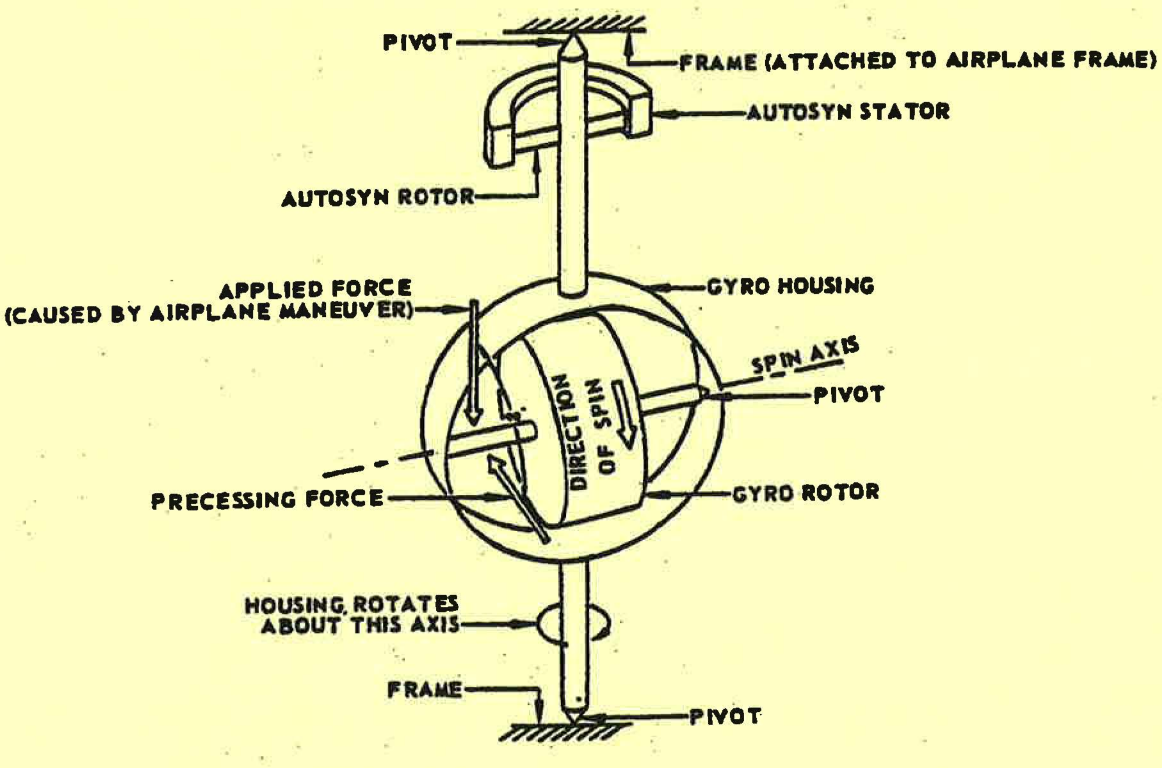


Autopilot Sensing Components Location
Fig. 1

(3) The centering spring is constructed and attached to the gyro housing pivot in such a manner that precession loads the spring which returns the gyro to center when the precession force is removed. This spring also contributes to giving a linear signal output. The dashpot mechanism provides damping action which allows the gyro to remain relatively unaffected by vibration and transient accelerations.

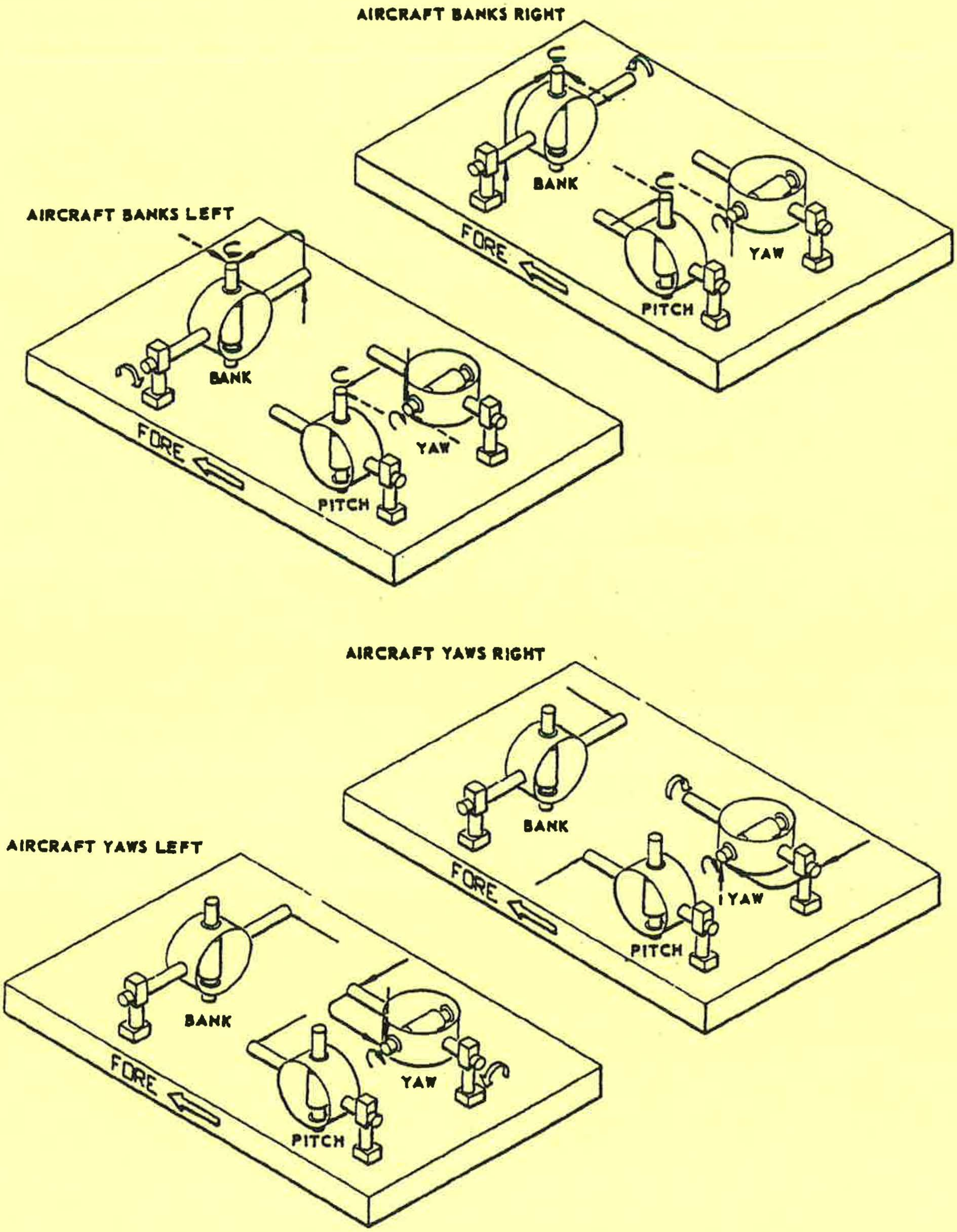
(4) A schematic diagram of the unit is given in fig. 4. Power is received from the autopilot power junction box. Protective fusing is provided only at the power junction box primary input.

ICN: 1P-AMMX-22-0011-01



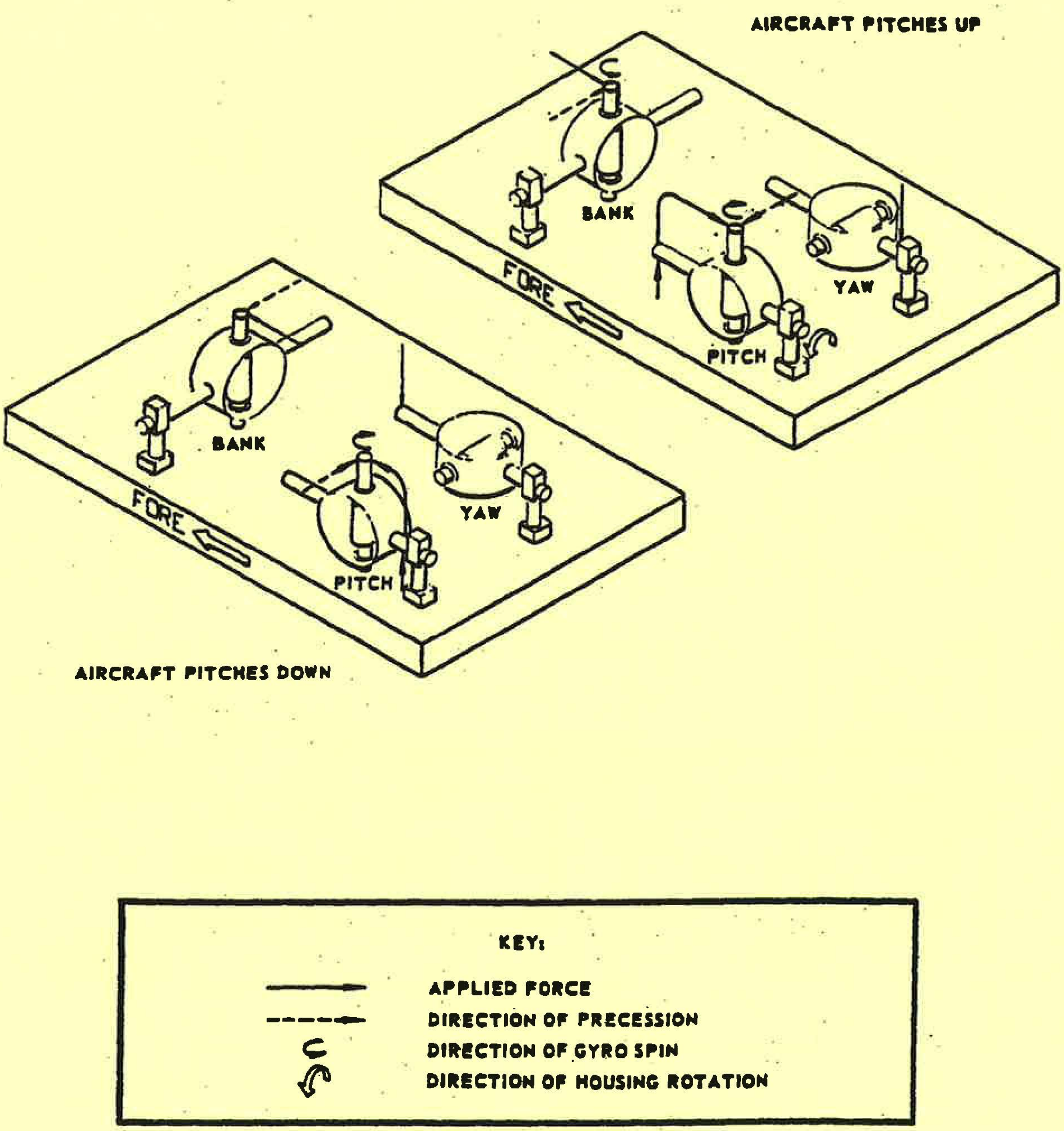
Rate Gyro Principles
Fig. 2

ICN: 1P-AMMX-22-0012-01



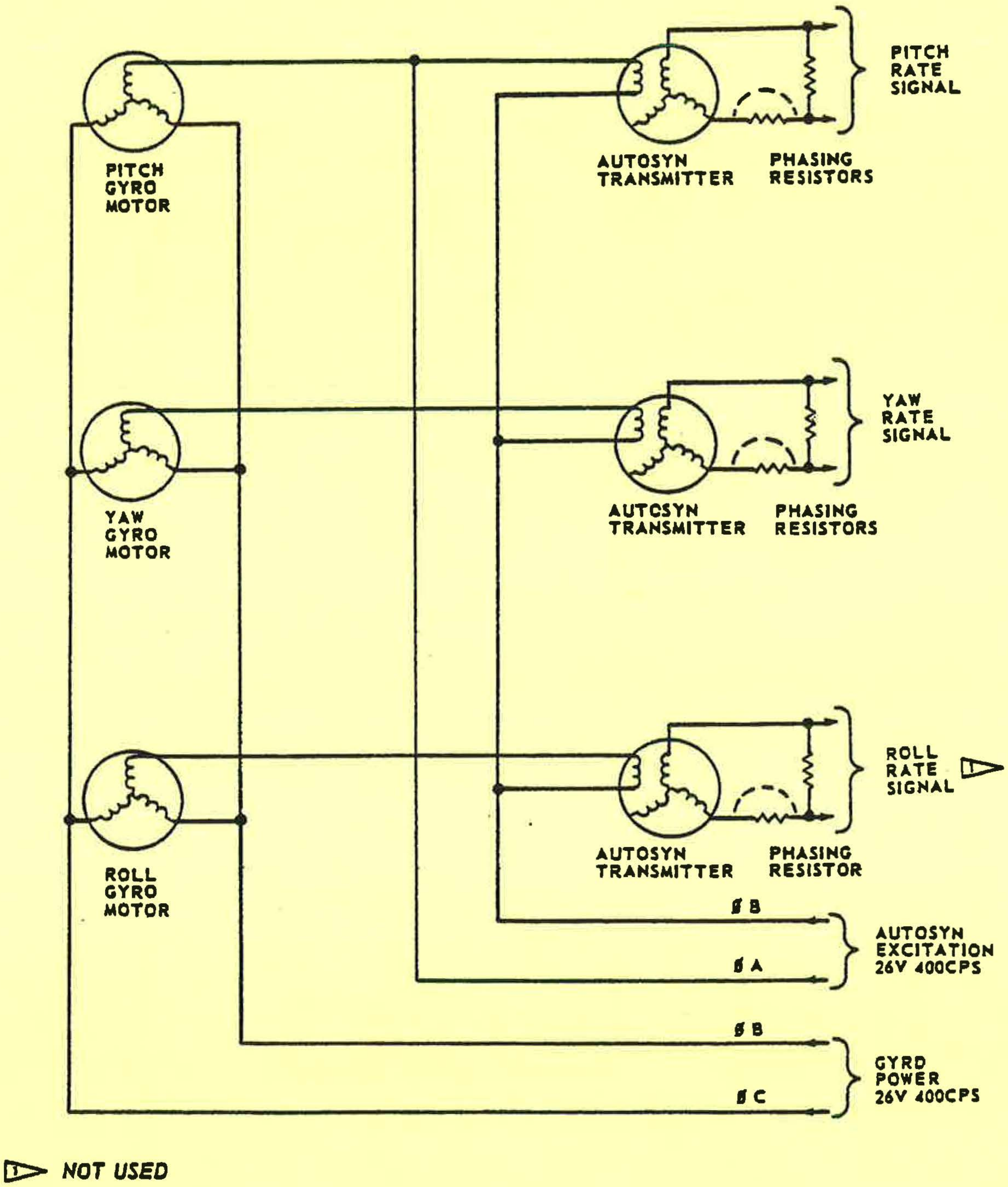
Rate Gyros Action
Fig. 3 (Sheet 1 of 2)

IICN: 1P-AMMX-22-0013-01



Rate Gyros Action
Fig. 3 (Sheet 2 of 2)

IICN: 1P-AMMX-22-0014-01



Three-Axis Rate Transmitter Schematic
Fig. 4

2. POSITION TRANSMITTER

A. The Position Transmitter measures the displacement from reference position of a control surface. It furnishes an electrical position follow-up signal which represents magnitude and direction of the displacement. Two Position Transmitters are used in the autopilot, one for left outboard spoiler position, and one for right outboard spoiler position.

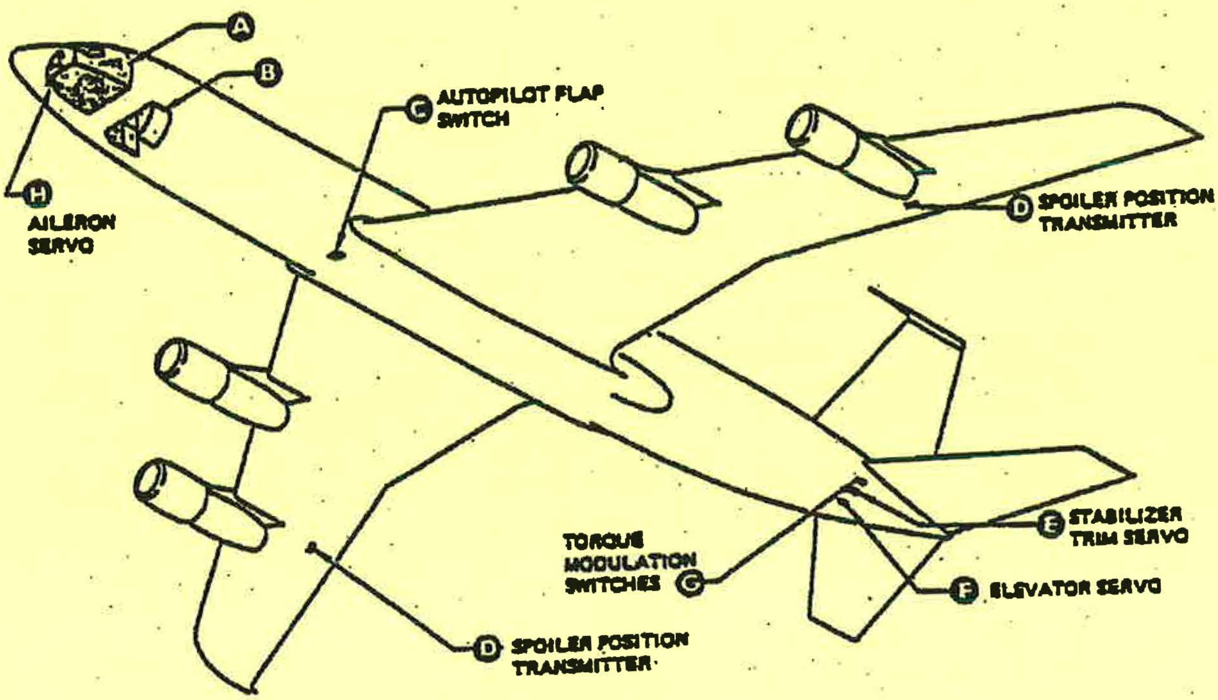
B. The Position Transmitter consists essentially of a high precision autosyn whose rotor is linked by an anti-backlash mechanism with one-to-one coupling to a splined input shaft. A selected phasing resistor is mounted across the back end of the autosyn. The autosyn assembly is enclosed in a gasket-sealed case, constructed to withstand weather enviroment. Electrical connections are made through a pigmy receptacle on the rear of the transmitter case.

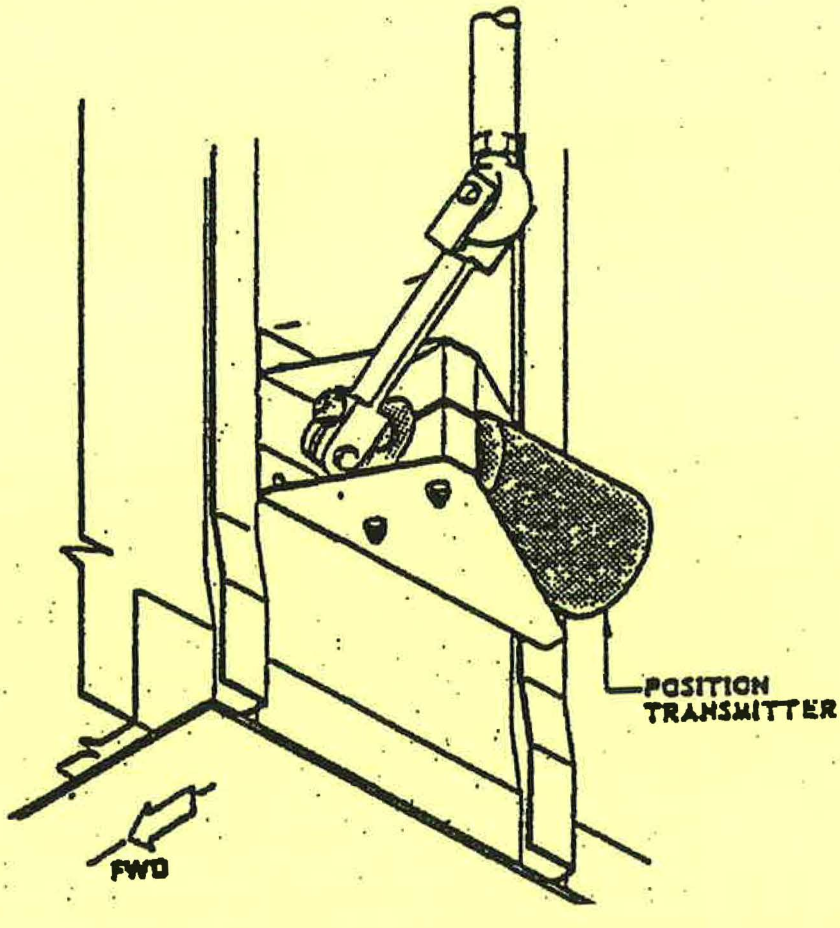
C. The spoiler Position Transmitters are mounted in clamping brackets, located on the aft side of the rear wing spars, one below spoiler number 1 and one below spoiler number 8 (see fig. 5). Access to either Position Transmitter is through cove-lip door number 1363 in the bottom wing panels. Refer to Chapter 12, "Access Doors and Panels."

D. The autosyn is utilized in the Position Transmitter as a direct sensor. Since this basic device is also used throughout the autopilot in conjunction with other elements and components as a mechanical to electrical transducer, its importance warrants a brief description of its operation.

(1) A mechanical displacement, suitably linked to the autosyn rotor produces a corresponding angular displacement of the rotor with respect to the stator. Excitation voltage in the rotor winding then magnetically induces an output voltage in the stator winding which is sinusoidally proportional to the magnitude and direction of the rotor displacement. A null, corresponding to a zero deviation of the movement from reference condition, occours when the rotor is in a position within the stator such that the induced voltage for all practical purposes is zero. Then displacement to one side of null position results in a output of one particular phase while displacement to the other side of null results in an output of opposite phase (see fig. 6). Output amplitude increases as the displacement for either side of null position increases and is maximum at 90 degrees displacement. Due to the internal configuration of the rotor and stator magnetic circuits, the autosyn output voltage is out-of-phase with the excitation voltage, leading it by 20 degrees. External phasing resistors are generally incorporated in the output and/or excitation circuits to properly adjust the output phase to signal reference phase for use in various autopilot circuits. This phasing resistor also sets the autosyn sensitivity (volts output per degree of rotor shaft rotation).

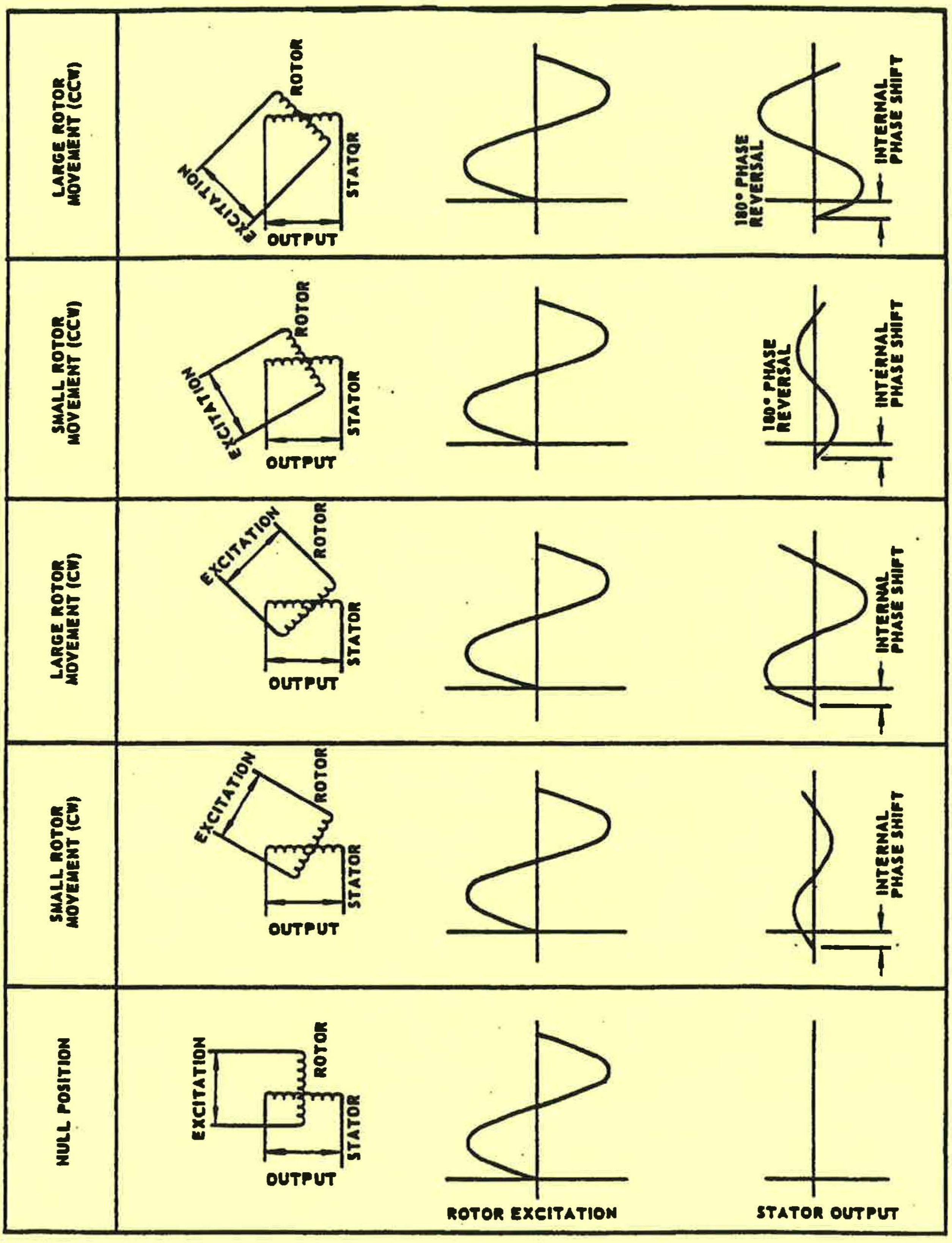
IICN: 1P-AMMX-22-0015-01





Spoiler Position Transmitter Location
Fig. 5

IICN: 1P-AMMX-22-0016-01



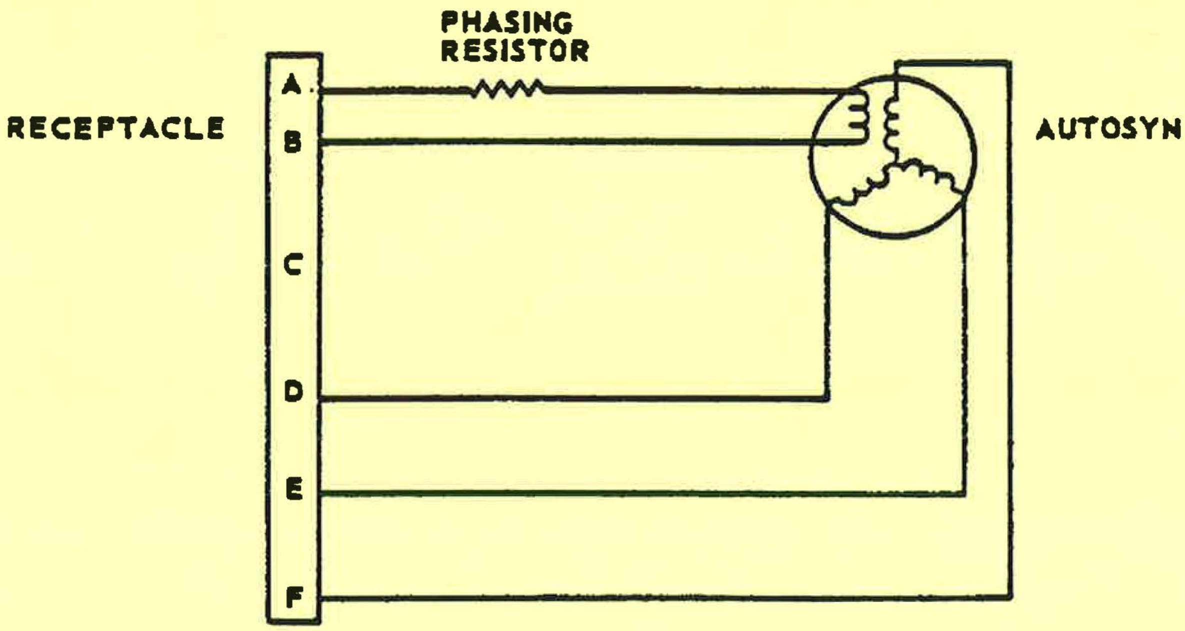
Schematic Representation of Autosyn Voltages
Fig. 6

(2) The Position Transmitter autosyn receive single-phase, 26 volt, 400 cycle excitation from the autopilot power junction box. Protective fusing is provided only at the power junction box primary input. An autosyn phasing resistor is connected in one rotor lead. It provides a sensitivity of 200 milivolts per degree.

(3) In performing its function, each spoiler Position Transmitter shaft is attached to the outboard spoiler just aft of its hinge line through an arm and control rod and is positioned simultaneously with the positioning of the spoiler.

(4) The circuit of this unit is given in fig. 7.

IICN: 1P-AMMX-22-0017-01



Position Transmitter Circuits
Fig. 7

END

SERIES YAW DAMPER SYSTEM - DESCRIPTION AND OPERATION

1. General

   A. The series yaw damper system provides automatic control of the yaw axis of the airplane and is electrically connected to the hydraulic power control unit of the rudder control system. (see chapter 27.) The series yaw damper allows the pilot to operate the rudder pedals in a normal manner without having an opposing force from the yaw damper. The limit of authority of the yaw damper system is $\pm$ 4° of rudder travel from rudder trim position.

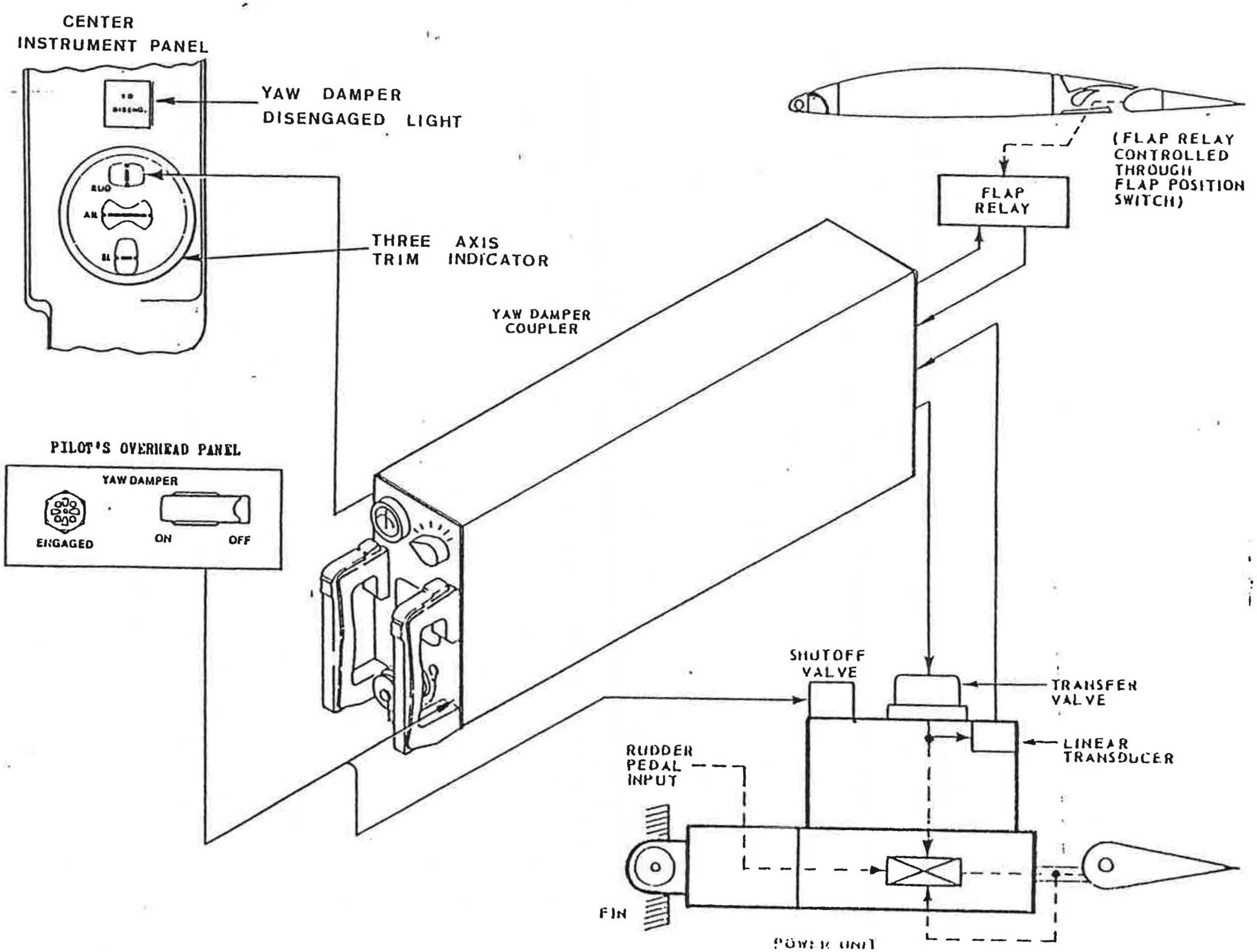
      NOTE: When the series yaw damper is installed in the airplane all reference to the rudder channel (parallel yaw damper) of the autopilot system may be disregarded.

   B. The series yaw damper system consists of a yaw damper coupler, yaw damper engage switch, yaw damper engage light (P13), yaw damper disengage light (P9) hydraulic pressure switch, rudder trim indicator, flap position relay controlled by the flap position (trim servo speed) switch, yaw damper-autopilot interlock relay and the yaw damper components of the rudder power control unit. The yaw damper components are the transfer valve, linear position transducer and the engage solenoid (shutoff) valve. The rudder trim indicator is part of the autopilot three axis trim indicator. A pictorial diagram of the yaw damper is shown in figure 1.

   C. The series yaw damper system is a full time yaw damper and. may be engaged during, taxiing takeoff or landing as well as during normal flight regime. When the yaw damper is engaged. the yaw rate gyro senses any change in the yaw axis and the yaw damper provides the necessary signals to the rudder power control unit and tie rudder to stabilize the airplane in the yaw axis. Normal operation of the yaw damper may be monitored by observing the opposing movement of the rudder trim meter, as the airplane is turned during taxiing maneuvers or in flight.

2. Yaw-Damper Coupler

   A. One yaw damper coupler is installed in the lower nose section of the airplane below the pilot's control column. The coupler consists of a rate gyro, power supply and several plug-in modules. The plug-in modules are the yaw rate filter, yaw synchronizer, yaw servo amplifier and the yaw gain calibrator. On the front panel of the coupler is the self test meter, self test switch, system test set connector and. the 115 volt a-c



Yaw Damper Pictorial Diagram
Figure 1

circuit breaker. The self test feature for the yaw damper system is provided by the self test switch and self test meter. If the self test switch is in any position but center OFF, the autopilot disengage warning lights will be on.

B. The yaw rate gyro is of conventional design and is provided with a torquer winding so that the gyro can be electrically rotated through a small angle. This feature is used in the self test checks to verify normal operation of the yaw damper coupler and the associated yaw damper components of the rudder power control unit.

C. The yaw damper coupler power supply furnishes all the a-c excitation and d-c control and power voltages required for the yaw damper coupler and the yaw damper components of the rudder power control unit. The coupler also provides the control voltage for the yaw damper-autopilot interlock relay used in the autopilot interlocks. (See figure 2.)

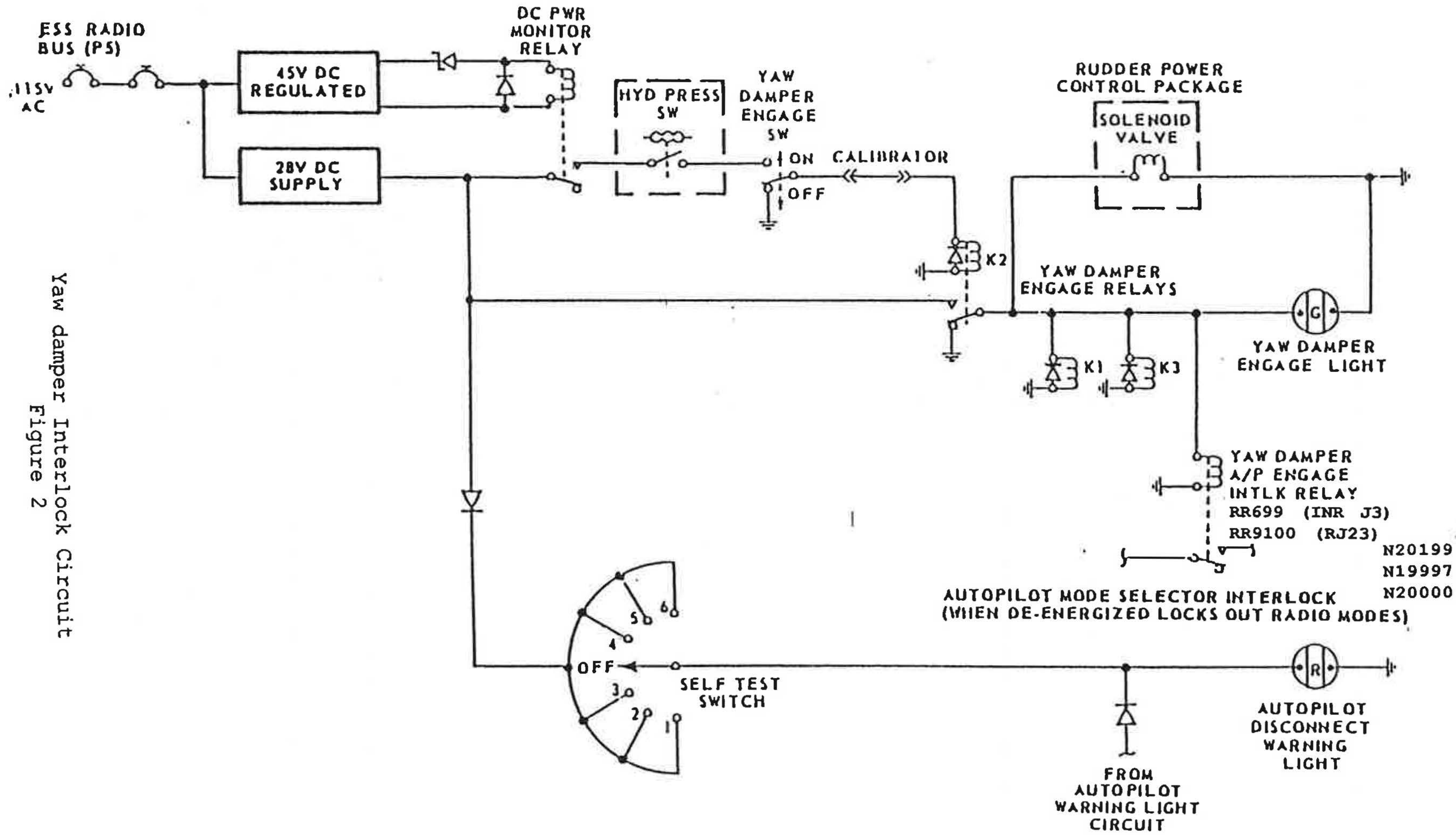
D. The plug-in modules contain the electronic components needed for amplification, synchronizing or band pass filtering of the yaw damper signals. The yaw gain calibrator contains various resistors, to adjust the gains of the yaw damper coupler to the requirements of the rudder power control unit, as well as a meter demodulator for the rudder trim meter and an amplifier for the aileron crossfeed signal from the autopilot system.

3. <u>Yaw Damper Hydraulic Pressure Switch</u>

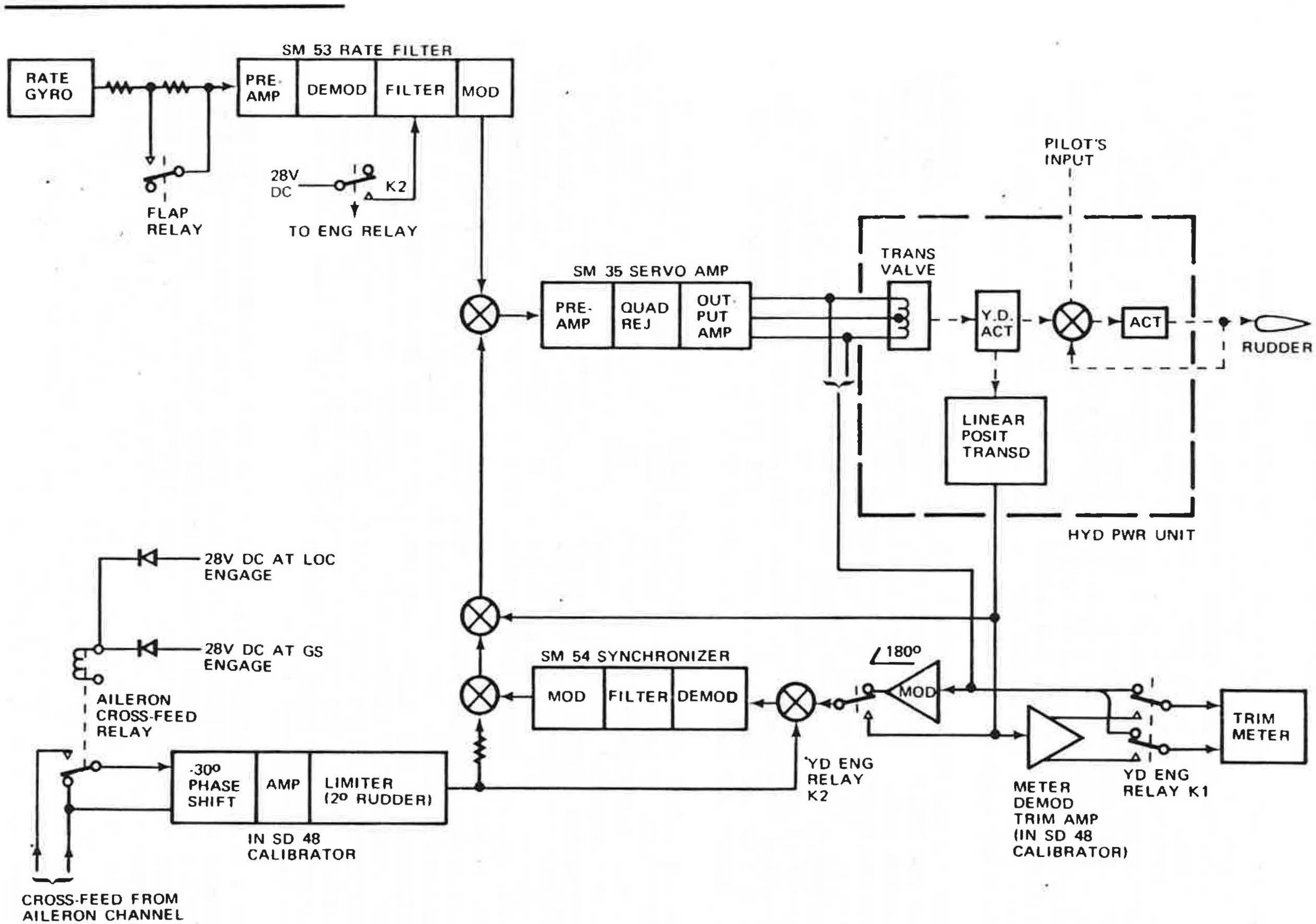
A. The yaw damper hydraulic pressure switch is similar to the rudder hydraulic low pressure warning switch, but of a lower setting. (See Chapter 27.) The two switches are mounted on the inner wall of the right hand wheel well at station 910. The yaw damper pressure switch contacts close when the hydraulic pressure exceeds 1000 psi and are used in the yaw damper interlock circuit. (See figure 2.)

4. <u>Yaw Damper Operation</u>

A. The series yaw damper system receives power from the yaw damper 115-volt a-c circuit breaker on the P5 panel. (See figure 2.) When the yaw damper coupler 45-volt d-c regulated supply reaches approximately 30 volts, the zener diode fires and energizes the d-c power monitor relay. This provides 28-volts dc for the interlock circuit through the hydraulic pressure switch, the yaw damper engage switch and the yaw damper calibrator wiring. When the yaw damper engage switch is OFF, both sides of the yaw damper engage relays and the engage solenoid valves are grounded. When the yaw damper engage switch is ON the yaw damper engage relays, the solenoid valve, the yaw damper engage light and the yaw damper-autopilot interlock relay are all energized. Failure of the 45-volt d-c regulated supply, 28-volt d-c supply, yaw damper engage relays, hydraulic pressure switch or hydraulic pressure will disengage the yaw damper system, extinguish the yaw damper engage light and return the autopilot to MAN mode if it was being used in a radio mode of operation.



Yaw damper Interlock Circuit
Figure 2



Yaw Damper Signal Circuit
Figure 3

B. When the yaw damper switch is OFF the yaw damper system is in a pre-engage or synchronizing mode. The output of the filter section of the rate filter (SM53) module is shorted to prevent the system from synchronizing to a yaw rate signal. (See figure 3.) The a-c output of the servo amplifier (SM55) is fed to the synchronizer module (SM54) through the modulator section prior to engagement to synchronize out all the signal chain unbalances. The rudder trim indicator is across the servo amplifier (SM55) output and displays whether the signal chain is nulled and ready for use at engagement. The rudder position transducer is also at a null, the yaw damper actuator is centered by the rudder power control unit caging springs and the de-energized solenoid valve shuts off hydraulic pressure to the yaw damper transfer valve.

B. When the yaw damper switch is ON, the yaw damper is engaged, energizing the yaw damper engage relays and the solenoid valve in the rudder power control unit. This ports hydraulic pressure to the transfer valve and the yaw damper coupler is now capable of operating the rudder control system. The output of the filter section of the rate filter module is unshorted and the input signal to the rate filter section of the synchronizer (SM54) is transferred to the rudder power control unit position transducer. The rudder trim indicator is transferred to the output of the rudder power control unit transducer through the trim amplifier section of the yaw calibrator (SD48).

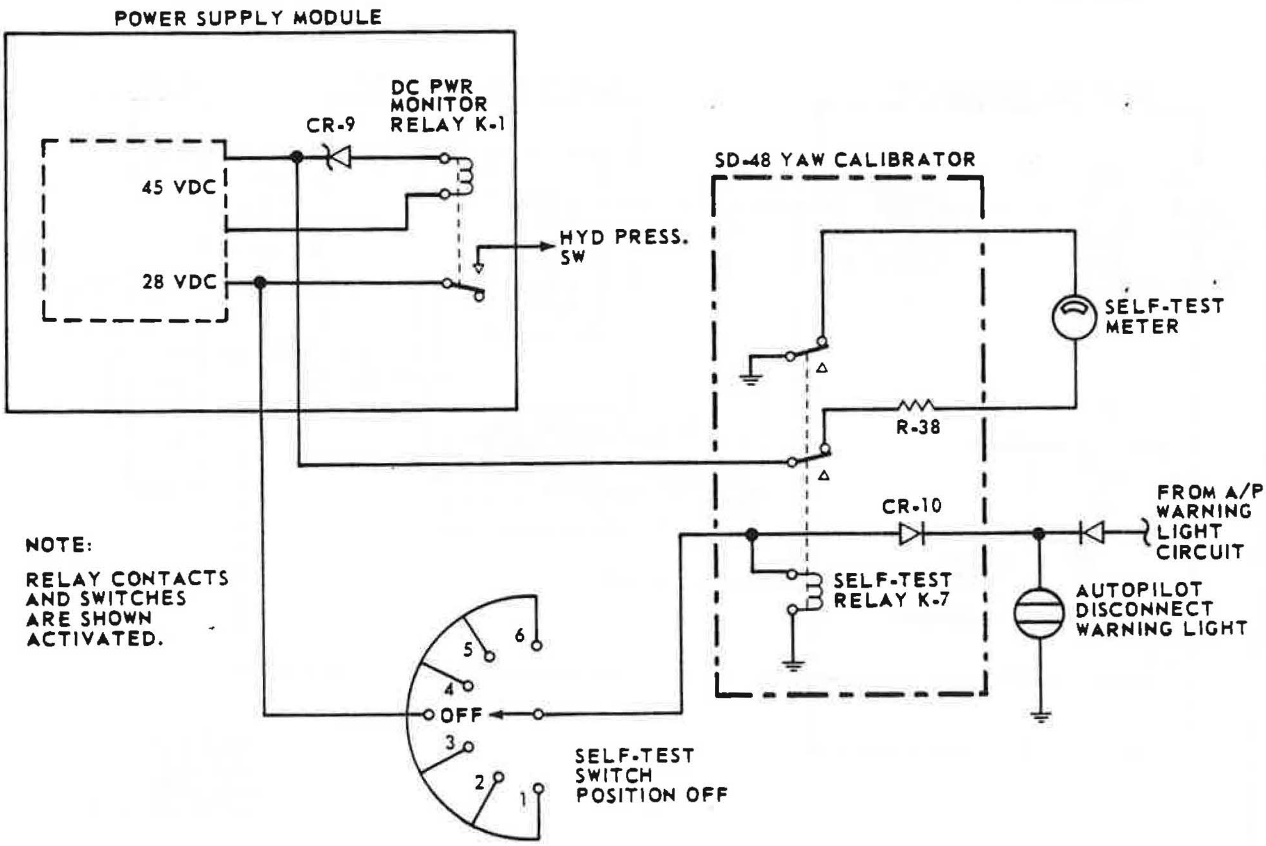
D. In the series yaw damper system the signal from the yaw rate gyro is introduced to the yaw rate filter module (SM53) through a gain change circuit controlled by a relay that is operated by a flap position switch. (See figure 3.) This gain change circuit compensates for differences in aerodynamic damping in the approach (flaps down) and cruise (flaps up) flight conditions. The band pass rate filter allows the signals in the dutch roll frequency range to pass, but eliminates both the steady yaw rate signals present during turns and the higher body bending frequencies. The output of the rate filter module is summed with the linear position transducer and synchronizer (SM54) outputs at the input to the servo amplifier (SM55). The summation of these signals produces a servo amplifier output signal that operates the yaw damper transfer valve in the rudder power control unit . The transfer valve ports hydraulic pressure to the yaw damper actuator which in turn displaces the main control valve. The control valve, when displaced, ports hydraulic pressure to the actuating piston which moves the rudder surface to satisfy the command signal.

E. An aileron crossfeed signal from the autopilot system is fed to the yaw damper system to provide for better stability and more accurate tracking of the localizer beam during an automatic approach. An amplitude limiter in the yaw damper coupler calibrator prevents saturation of the yaw damper signal chain due to the crossfeed signal. The aileron crossfeed signal is fed to the servo amplifier input and to the synchronizer input after passing through the amplifier-limiter (SD-48). The crossfeed signal is never synchronized since the amount of signal sent to the synchronizer is equal and opposite to the amount of transducer output caused by the servo amplifier input.

5. Yaw Damper System Self-Test Operation

A. The yaw damper coupler incorporates self-test circuitry to enable preflight operational check-out of the yaw damper system. Four functional tests are available by means of a spring-loaded self-test switch mounted on the front panel of the yaw damper coupler. The self-test switch connects a suitable input signal and output metering circuit to the system function undergoing test. Test results are indicated on the self-test meter which is also mounted on the front panel of the yaw damper coupler. Positions 2 and 4 of the self-test switch check the response of the complete yaw damper system to simulated yaw rate gyro signals and require that the yaw damper system be engaged. Positions 1 and 6 of the self-test switch check internal functions of the yaw damper coupler only and are performed with the yaw damper system disengaged. The center OFF position of the self-test switch connects the self-test meter across the output of the yaw damper coupler power supply module to provide continual monitoring of the 45 volt dc output. Positions 3 and 5 of the self-test switch are not used and are provided for possible future use. The individual tests are described and illustrated in the following paragraphs.

B. Self-Test Switch: Position OFF. With the self-test switch in the center OFF position the 45 volt dc output of the power supply module is continually monitored by the self-test meter through the normally-closed contacts of self-test relay K7. (See figure 4.) Correct voltage is

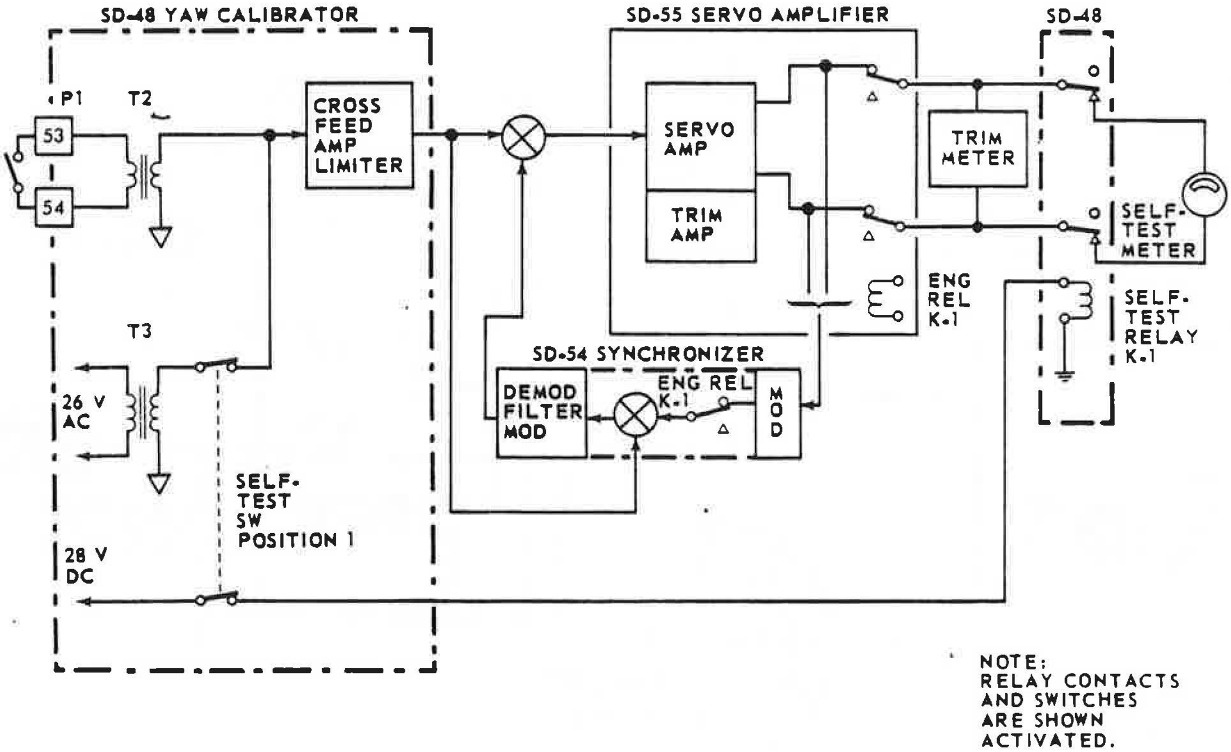


Self-Test Switch Position - OFF
Figure 4

indicated by the self-test meter reading in the mid-green area. Displacement of the self-test switch from the center OFF position applies 28 volts dc across self-test relay K7 and the autopilot disconnect warning light. Operation of relay K7 disconnects the self-test meter from the power supply output while steady illumination of the autopilot disconnect light provides warning that the self-test switch is not in the OFF position.

C. Self-Test Switch: Position 1. To perform this test the yaw damper system must be disengaged. Figure 5 shows the circuit configuration when the self-test switch is set to position 1 and the yaw damper engage switch is set to OFF. Under these conditions, the response of the limiter amplifier and the yaw synchronizer to a simulated crossfeed signal is checked at the output of the servo amplifier. The self-test meter must initially indicate in the mid-green area and slowly return to the null-green area in less than 18 seconds.

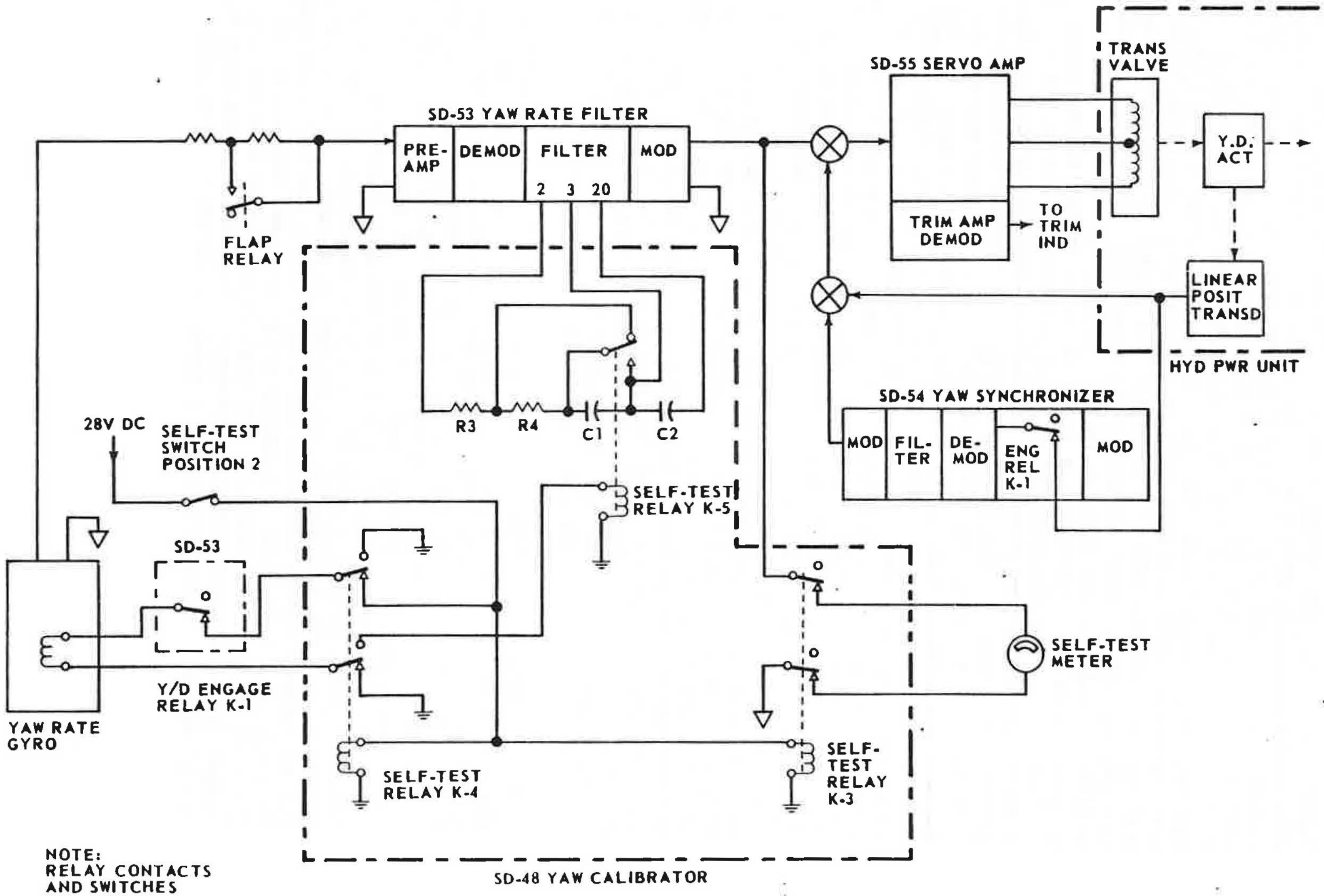
D. Self-Test Switch: Position 2. The yaw rate filter test verifies normal operation of the yaw damper system in response to a step output signal from the yaw rate gyro. This test requires engagement of the yaw damper system which in turn requires that the auxiliary hydraulic pump and the No. 1 rudder power switch on the overhead control panel be ON. The yaw damper hydraulic pressure switch closes when the hydraulic pressure exceeds 1000 psi and arms the yaw damper ENGAGE switch. (See figure 2.) Placing the yaw damper ENGAGE switch to the ON position



Self-Test Switch Position 1
Figure 5

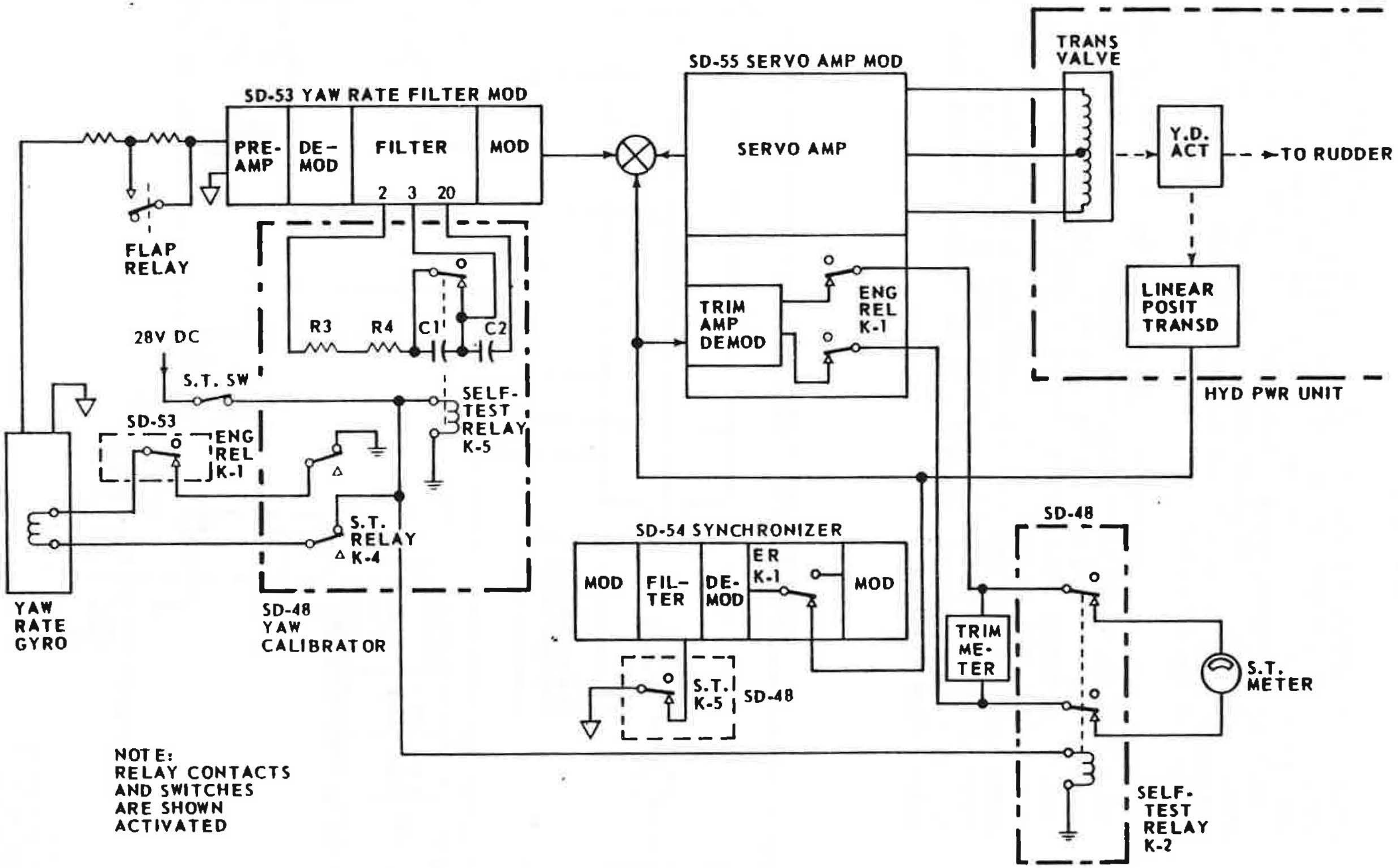
applies 28 volts dc to the yaw damper engage relays, the yaw damper engage light, and the solenoid valve in the rudder power control unit. This ports hydraulic pressure to the transfer valve and the yaw damper coupler is now capable of operating the rudder control system. The step input test signal is generated by holding the self-test switch in position 2. (See figure 6.) This energizes the torquer winding in the yaw rate gyro which causes the gyro to rotate through a small angle and produce a step output signal that is applied to the yaw rate filter module. In the yaw rate filter module the signal is demodulated and applied through a series capacitor-resistor (CR) filter circuit and a modulator stage to the input of the servo amplifier. This produces an output signal that operates the transfer valve in the rudder power control unit. The transfer valve ports hydraulic power to the yaw damper actuator which in turn displaces the main control valve and the linear position transducer. The control valve, when displaced, ports hydraulic pressure to the actuating piston which moves the rudder surface to satisfy the command signal. The linear position transducer produces a follow-up signal that is proportional to the amount of rudder deflection. This signal is fed back to the input of the servo amplifier and the yaw synchronizer in phase-opposition to the test input signal. As capacitor C-1 in the yaw rate filter circuit charges to the steady-state level of the test signal the output of the yaw rate filter module decreases. This causes the opposite phase follow-up signal to predominate and drive the rudder surface back to the streamlined position. This action is monitored by the self-test meter, which is connected across the output of the yaw rate filter module, through the normally-open contacts of self-test relay K-3. Correct operation of the system is indicated by the pointer of the self-test meter rising to the mid-green area approximately one second after the self-test switch is set to position 2 and then slowly returning to the null-green area approximately 14 seconds later. Concurrently, the rudder surface and the rudder trim meter will deflect to the left and then return to center.

E. Self-Test Switch: Position 4. The follow-up gain test is similar to the yaw rate filter test (paragraph D.) in that it provides an operational check of the complete yaw damper system. In this case however capacitor C-1 in the filter section of the yaw rate filter module is shorted out by the normally-open contacts of self-test relay K-5. Consequently, the steady-state test signal from the yaw rate gyro is not washed out in the yaw rate filter and thus functions as a constant level command signal to the servo amplifier. In addition, the output of the yaw synchronizer is shorted to ground. As a result, the rudder control system displaces the rudder until the follow-up signal from the linear position transducer nulls the test command signal at the input of the servo amplifier. The test is performed with the yaw damper system engaged and the self-test switch held in position 4. The resulting circuit configuration is shown in Figure 7. The self-test



NOTE:
RELAY CONTACTS
AND SWITCHES
ARE SHOWN
ACTIVATED.

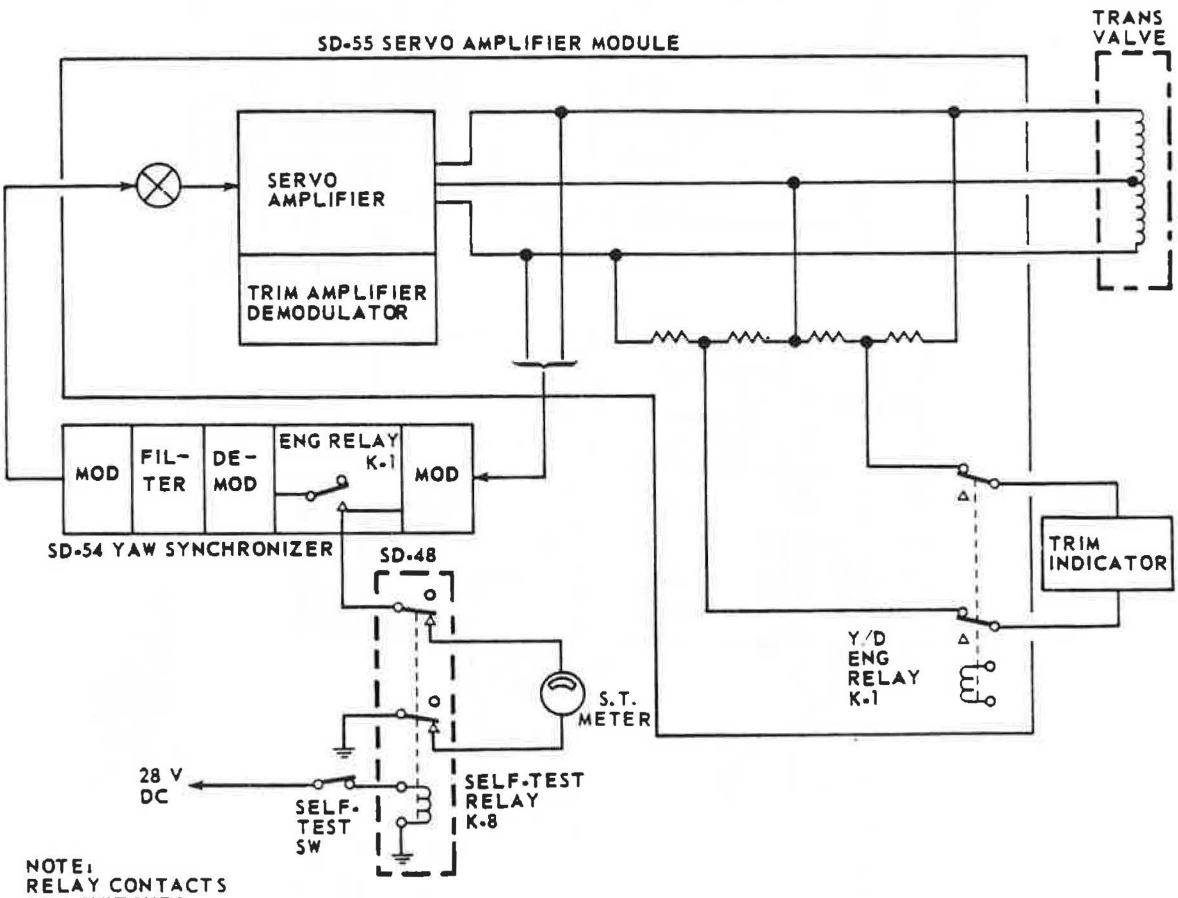
Self-Test Switch Position 2
Figure 6



Self-Test Switch Position 4
Figure 7

meter is connected across the output of the trim amplifier demodulator through normally-open contacts of yaw damper engage relay K-1 and self-test relay K-2 to monitor the output of the follow-up linear position transducer. Correct system operation is indicated by the self-test meter reading in the midgreen area. The rudder control surface and the rudder trim meter will be deflected to the right and remain there as long as the self-test switch is held in position 4.

F. Self-Test Switch: Position 6. The channel balance test checks the pre-engage operation of the yaw synchronizer to null any long term residual signal imbalance at the output of the servo amplifier. The test is performed with the yaw damper system disengaged and the self-test switch held in position 6. As shown in Figure 8 the self-test meter monitors the output of the servo amplifier through the modulator stage of the yaw synchronizer. Servo loop quiescent balance is shown by the self-test meter indicating in the null-green area and the trim meter being centered.



Self-Test Switch Position 6
Figure 8

YAW DAMPER SYSTEM - TROUBLE SHOOTING

1. General

A It is virtually impossible to trouble shoot the yaw damper system without having a thorough understanding of the system. The maintenance man should, from the malfunction report, be able to determine whether the fault is in the engage interlock circuit or the yav damper signal channel. The trouble shooting, charts are designed in a manner to facilitate this. It is obvious that adequate flight reports are essential to permit intelligent maintenance action. Flight reports such as Inoperative without additional description of operation involved, forces the maintenance man to perform a complete functional test of the system. Even with more complete flight reports, it might not always be possible to indicate the shortest route to the faulty component. It the flight report is of such a nature that it is not immediately apparent which trouble shooting chart to use, it is recommended that the applicable Adjusment/Test procedures be used.

B. Experience has shown, that in electronic systems, airplane wiring is responsible for less than two percent of all system malfunctions. This means that more than 98 percent of the failures occur in the system components. Emphasis is therefore placed on locating the faulty component and replacing it. A fault in the airplane wiring however, is always a possibility and wiring integrity should be checked when replacement of the suspected component does not clear the system fault.

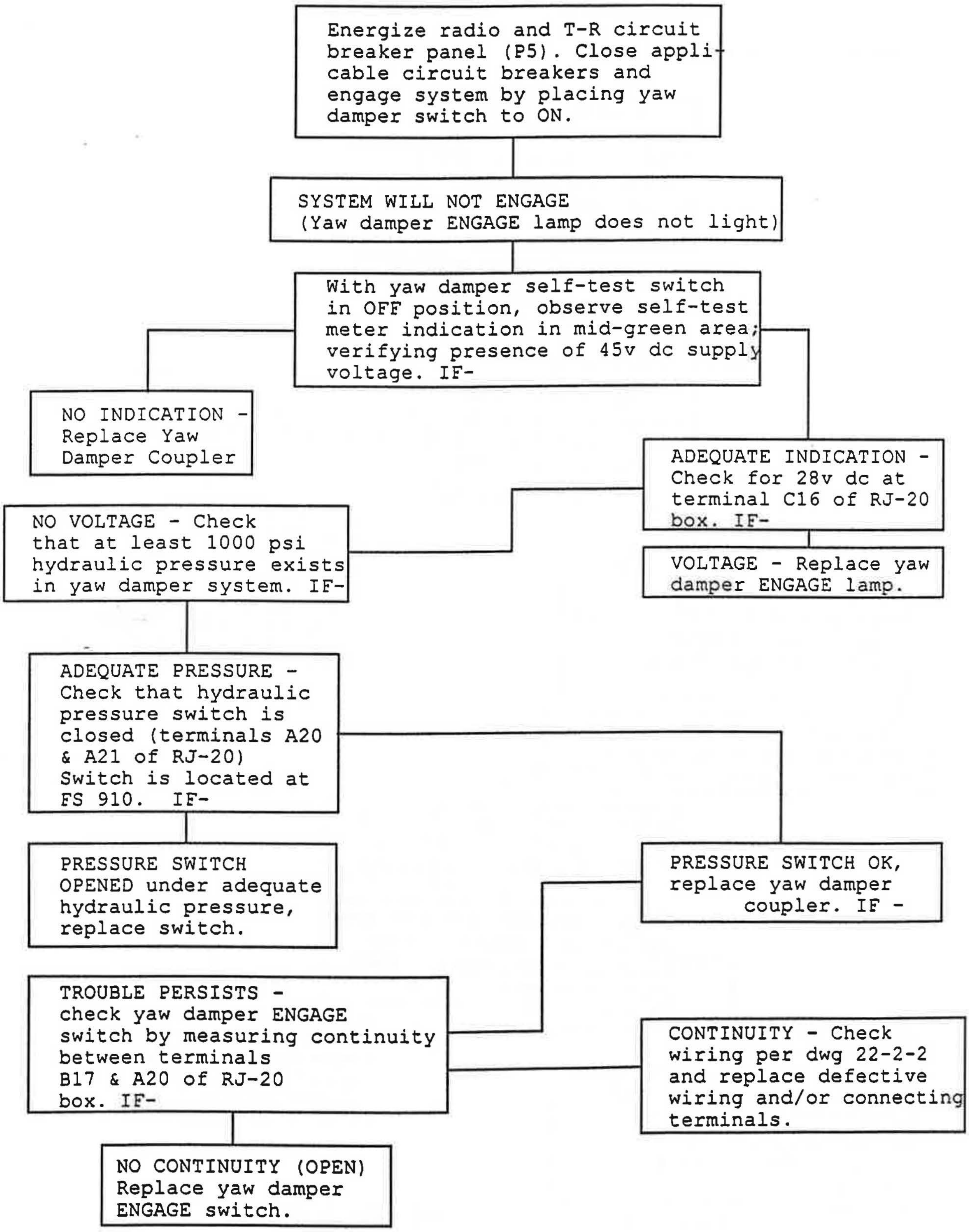
C. In every case in which the system fails to operate correctly, the first check should determine that all interconnecting wiring connectors are properly attached and that operating power is available to the component which receive it. Voltage measurements (using a standard multimeter) may be made at terminals in the autopilot junction box. Consult applicable wiring diagram when making power cheeks.

NOTE: All terminal numbers that are called out in this section are located in RJ20, if not otherwise specified.

D. An attempt has been made to list the more common types of malfunctions. The troubles are listed in the first colimn, expressed in a manner similar to the way they would be written up in flight reports. The second column lists probable faulty components causing the malfunction, and the third colimn lists the particular table charts to use to isolate the trouble. This listing should always be consulted first, so that the correct trouble shooting charts may be used.

Report of Malfunction	Probable Cause	Trouble Charts
	Engage Interlock Circuit	
System will not engage in YAW DAMPER	1. Circuit Breaker (Yaw Damper Coupler) 2. Yaw Damper Engage Switch 3. Yaw Damper Coupler 4. Hydraulic Pressure Switch	A
No control in yaw axis and no rudder trim indication with damper engaged	1. Three Axis Rate Transmitter 2. Yaw Damper Coupler	B
No control in yaw axis, but rudder trim indication normal for damper operation	1. Rudder Hydraulic Power Control Unit 2. Yaw Damper Coupler	B
No rudder trim indication, but control in yaw axis normal for damper operation	1. Three Axis Trim Indicator 2. Yaw Damper Coupler	B
Excessive undamped response in yaw axis	1. Rudder Hydraulic Power Control Unit 2. Yaw Damper Coupler	B
Control in yaw axis sluggish during damper operation	1. Yaw Damper Coupler 2. Rudder Hydraulic Power Control Unit	B
In glide slope mode, turns are not coordinated	1. Yaw Damper Coupler 2. Amplifier and Computer (Autopilot) 3. Autopilot Adapter (check correct pot setting) 4. Aileron/Rudder Crossfeed Relay	
Tendency to yaw increases after damper is engaged	1. Rudder Hydraulic Power Control Unit 2. Yaw Damper Coupler	
Airplane susceptible to Dutch Roll or "snaking" action	1. Yaw Damper Coupler 2. Rudder Hydraulic Power Control Unit 3. Three Axis Rate Transmitter 4. Amplifier and Computer (Autopilot) 5. Airplane Rigging, see chapter 27 6. Flap Switch	

A. Control Switching Interlock Circuit (Series Yaw Damper)



B. Series Yaw Damper Signal Chain

Energize radio and T-R circuit breaker panel (P5). Close applicable circuit breakers and engage system in yaw damper - ON

RUDDER CONTROL SURFACE DEFLECTS, BUT RUDDER TRIM INDEX DOES NOT DEFLECT IN THREE-AXIS TRIM INDICATOR.

Disengage yaw damper system and place yaw damper self-test switch to position 1. Read self-test meter. IF -

Self-test meter initially indicates in mid-green area and returns to mid-green area in less than 18 seconds; then yaw damper system limiter and sync response is OK. IF -

NO RESPONSE on self-test meter, relay K1 is not energizing, or 26v ac test voltage not available. Replace yaw damper coupler.

Rudder trim Index does not deflect during test, replace three-axis trim indicator. IF -

RUDDER CONTROL SURFACE DOES NOT AUTOMATICALLY FUNCTION DURING YAW DAMPER OPERATION -

Engage yaw damper system and place yaw damper self-test switch to position 2. Read self-test meter. IF -

Self-test meter reads in green area after one second and slowly returns to null-green area approximately 14 seconds later, yaw damper system response is OK. IF -

NO RESPONSE, or incorrect response on self-test meter, replace yaw damper coupler.

CONTINUED ON FOLLOWING PAGE

CONTINUED FROM PRECEDING PAGE

RUDDER control surface or rudder trim index does not deflect (left, then centered) during self-test, cheek for defective rudder hydraulic power control unit (per chapter 27). IF -

Previous self-tests OK, place self-test switch to position 4 and reed self-teSt meter. IF -

Self-test meter indicates in mid-green area as long as self-test switch is held in position 4, then yaw damper system response is OK. IF -

NO RESPONSE, or incorrect response on self-test meter, replare yaw damper coupler.

Rudder control surface nor rudder trim index does not deflect (right) during self-test, check for defective rudder hydraulic power control unit (per Chapter 27). IF -

Previous self-tests OK, disengage yaw damper and place self-test switch to position 6. Read self-test meter. IF -

Self-test meter indicates in null-green area, and rudder trim index reads centered, then yaw damper channel balance in OK. IF -

NO REPONSE, or incorrect response on self-test meter, replace yaw damper coupler.

PREVIOUS SELF-TESTS check out OK, tune in localizer station; engage autopilot in GS AUTO mode and cheek that aileron X-feed relay energizes properly. IF -

RELAY ENERGIZES OK check pot setting in A/P adapter (RJ-16 for N20199 RJ23 for N19997 & N20000). Readjust or replace as necessary if aileron x-feed signal has been found to be such that rudder control surface deflection does not limit to 2-1/2° left or right.

RELAY DOES NOT ENERGIZE Cheek for 28v dc at terminal G28 or L19 of RJ-3 junction box. IF -

VOLTAGE - replace defective x-feed relay.

No VOLTAGE - Trouble exists in autopilot amplifier computer.

SERIES YAW DAMPER SYSTEM - ADJUSTMENT/TEST

1. General

A. This system test insures that all components are properly installed and wired. It also checks the operation of the yaw damper on the ground for proper operation of the various components in the signal chain. It is assumed that all system furnishing signal data to the Yaw Damper are in proper working order, and the rudder control system is properly rigged (see Chapter 27 for rudder control system and Chapter 29 for rudder hydraulic system operation.). The dynamic response of the airplane cannot be checked on the ground.

B. Input signal data to the Yaw Damper is supplied by the yaw rate gyro in the Yaw Damper Coupler, a crossfeed signal from the aileron channel of autopilot and the linear position transducer of the rudder (hydraulic) power (control) unit. The Yaw Damper Coupler processes these signals and supplies the corrective signals to the transfer valve of the Rudder Power Unit to damp out unwanted turn motions of the airplane. The Yaw Damper Coupler also supplies 28 volts dc to the rudder power unit engage solenoid (shutoff) valve which then ports hydraulic fluid within the Rudder Power Unit, so that the yaw damper hydraulic spool valve may control the main hydraulic spool valve and move the rudder.

C. The series yaw damper system may be checked without any auxiliary equipment, but an optional test of the Rudder Power Control Unit and its yaw damper components may be performed by use of Boeing Elevator/Rudder Power Control Unit Test Set No. F72785. Boeing Document D6-6891 should also be obtained for full description of the tests.

2. Prepare to Test Series Yaw Damper System

A. Gain access to Yaw Damper Coupler in lower nose section and establish communications with another technican in the control cabin.

B. With external power supplied, establish that yaw damper and autopilot circuit breakers are pushed in, except where otherwise specified, and that yaw damper system and other associated systems are powered. Detemine that hydraulic pressure is available to operate rudder power unit.

WARNING: PRESSURIZING HYDRAULIC SYSTEM WILL SUPPLY POWER TO RUDDER AND ASSOCIATED HYDRAULIC SYSTEMS. CARE SHOULD BE TAKEN TO ISOLATE AND TAG HYDRAULIC SYSTEM NOT BEING TESTED TO PREVENT INJURY TO PERSONNEL OR DAMAGE TO AIRPLANE AND EQUIPMENT.

3. Test Yaw Damper Interlock and Engage Light

A. With rudder hydraulic power on and yaw damper and autopilot circuit breakers pushed in, engage Yaw Damper and autopilot. The yaw damper engage light shall illuminate.

B. With autopilot in MAN mode, turn rudder hydraulic power off. Yaw damper engage light shall extinguish and autopilot disengage. The autopilot warning light will flash and the audio alert will sound.

C. Turn rudder hyraulic power on, yaw damper engage light will illuminate. Pull yaw damper circuit breaker. Yaw damper engage light will extinguish. Push yaw damper circuit breaker in. Yaw damper engage light shall illuminate.

D. Engage autopilot and select LOC/VOR mode. Turn yaw damper engage switch OFF. Yaw damper engage light will extinguish and autopilot will disengage. Reselect LOC/VOR mode. Mode selector switch shall not latch-up in LOC/VOR mode. Turn Yaw Damper Switch ON, yaw damper engage light will illuminate.

E. Repeat step D. but use G/S AUTO and the G/S MAN positions of Autopilot Selector Switch.

F. Engage Yaw Damper. Place Yaw Damper Self Test Switch on Yaw Damper Coupler in any position but OFF. Autopilot disengage warning light-shall illuminate steady. Set Yaw Damper Self Test Switch to OFF position. Autopilot warning light shall extinguish.

4. Test Yaw Damper Power Mode (Self Test Procedure)

A. Place rudder in power mode configuration at zero trim knob and centered rudder pedals. Establish that flaps are in up (zero degrees) position unless otherwise specified. Engage yaw damper. Yaw damper engage light shall illuminate and self test meter on Yaw Damper Coupler shall read in the middle green area, showing proper output d-c voltage from Yaw Damper Coupler.

B. Perform yaw damper system self tests as outlined in fig. 1. The autopilot disengaged warning light will be extinguished for OFF position of self test switch, but illuminated for any other position of switch.

C. If no further requirement for external power, return all system to normal and remove external power.

Carry out a test according to the following list

Test procedure	Power mode YAW DAMPER	Position of SELF-TEST switch	A/P warning lamp	Trim indicator	Rudder excursion	SELF-TEST meter
+45V DC	Switched on or off	OFF	OFF	Middle position	Middle position	Middle green range
Synchro-nizer loop	Switched off	1	ON	Left, then middle-position	Remains in middle position	Middle green range, then decreasing until green zero range in approx. 18 sec.
Switch the SELF-TEST switch OFF and wait 10 sec.						
Rate time constant	Switched on	2	ON	Left, then middle position	Left, then middle position	Max. output in middle green range for approx. 1 sec., then drop to the green zero range
Switch the YAW DAMPER OFF and wait 30 sec.						
Follow-up	Switched on	4	ON	Right	Right	Middle green range
Follow-up	Switched on, land-ing flaps deployed	4	ON	Right	Right	Larger than with the landing flaps retracted
Channel balance	Switched off	6	ON	Middle position	Middle position	Decreasing indication until green zero range

Fig. 1

END

INTENTIONALLY LEFT BLANK

SERIES YAW DAMPER SYSTEM - ADJUSTMENT/TEST

1. General

A. This system test insures that all components are properly installed and wired. It also checks the operation of the yaw damper on the ground for proper operation of the various components in the signal chain. It is assumed that all systems furnishing signal data to the yaw damper are in proper working order, and the rudder control system is properly rigged. (See Chapter 27 for rudder control system and Chapter 29 for rudder hydraulic system operation.) The dynamic response of the airplane cannot be checked on the ground.

B. Input signal data to the yaw damper is supplied by the yaw rate gyro in the yaw damper coupler, a crossfeed signal from aileron channel of autopilot and the linear position transducer of the rudder (hydraulic) power (control) unit. The yaw damper coupler processes these signals and supplies the corrective signals to the transfer valve of the rudder power unit to damp out unwanted turn motions of the airplane. The yaw damper coupler also supplies 28 volts dc to the rudder power unit engage solenoid (shutoff) valve which then ports hydraulic fluid within the rudder power unit, so that the yaw damper hydraulic spool valve may control the main hydraulic spool valve and move the rudder.

C. The series yaw damper system may be checked without any auxiliary equipment, but an optional test of the rudder power control unit and its yaw damper components may be performed by use of Boeing Elevator/Rudder Power Control Unit Test Set No. F72785. Boeing Document D6-6891 should also be obtained for full description of the tests.

2. Prepare to Test Series Yaw Damper System

A. Gain access to yaw damper coupler in lower nose section and establish communications with another technician in the control cabin.

B. With external power supplied, establish that yaw damper and autopilot circuit breakers are pushed in, except where otherwise specified, and that yaw damper system and other associated systems are powered. Determine that hydraulic pressure is available to operate rudder power unit.

WARNING: PRESSURIZING HYDRAULIC SYSTEMS WILL SUPPLY POWER TO RUDDER AND ASSOCIATED HYDRAULIC SYSTEMS. CARE SHOULD BE TAKEN TO ISOLATE AND TAG HYDRAULIC SYSTEMS NOT BEING TESTED TO PREVENT INJURY TO PERSONNEL OR DAMAGE TO AIRPLANE AND EQUIPMENT.

3. Test Yaw Damper Interlock and Engage Light

A. With rudder hydraulic power on and yaw damper and autopilot circuit breakers pushed in, engage yaw damper and autopilot. The yaw damper engage light shall illuminate.

B. With autopilot in MAN mode, turn rudder hydraulic power off. Yaw damper engage light shall extinguish and autopilot will remain egaged, (N20199) disengage (N19997, N2000.)

C. Turn rudder hydraulic power on, yaw damper engage light will illuminate Select autopilot HDG mode. Pull yaw damper circuit breaker. Yaw damper engage light will extinguish and autopilot will remain engaged in HDG mode. Push yaw damper circuit breaker in. Yaw damper engage light shall illuminate.

D. Select LOC/VOR mode. Turn yaw damper engage switch off. Yaw damper engage light will extinguish and autopilot will remain engaged, but revert to MAN mode. Reselect LOC/VOR mode. Mode selector switch shall not latch-up in LOC/VOR mode. Turn yaw damper switch on, yaw damper engage light will illuminate.

E. Repeat stem D. but use GI/S AUTO and then G/S MAIN positions of autopilot mode selector switch.

F. Engage yaw damper. Place yaw damper self test switch on yaw damper coupler in any position but OFF. Autopilot disengage warning light shall illuminate steady. Set yaw damper self test switch to CFF position. Autopilot warning light shall extinguish.

NOTE: Para 3C/ D/ E/ for N20199 only

4. Test Yaw Damper Power Mode (Self Test Procedure)

A. Place rudder in power mode configuration at zero trim knob and centered rudder pedals. Establish that flaps are in up (zero degrees) position unless otherwise specified. Engage yaw damper. Yaw damper engage light shall illuminate and self test meter on yaw damper coupler shall read in the middle green area, showing proper output d-c voltage from yaw damper coupler.

B. Perform yaw damper system self tests as outlined in figure 501. The autopilot disengaged warning light will be extinguished for OFF position of self test switch. but illuminated for any other position of switch.

C. If no further requirement for external popwer, return all systems to normal and remove external power.

Carry out a test according to the following list

Test procedure	Power mode YAW DAMPER	Position of SELF-TEST switch	A/P warning lamp	Trim indicator	Rudder excursion	SELF-TEST meter
+45V DC	Switched on or off	OFF	OFF	Middle position	Middle position	Middle green range
Synchronizer loop	Switched off	1	ON	Left then middle position	Remains in middle position	Middle green range then decreasing until green zero range in approx. 18 sec.
Switch the SELF-TEST switch OFF and wait 10 sec.						
Rate time constant	Switched on	2	ON	Left then middle position	Left then middle position	Max. output in middle green range for approx. 1 sec. then drop to the green zero range
Switch the YAW DAMPER off and wait 30 sec.						
Follow-up	Switched on	4	ON	Right	Right	Middle green range
Follow-up	Switched on, landing flaps deployed	4	ON	Right	Right	Larger than with the landing flaps retracted
Channel balance	Switched off	6	ON	Middle position	Middle position	Decreasing indication until green zero range

Figure 501

SERIES YAW DAMPER RUDDER POWER UNIT COMPONENTS - MAINTENANCE PRACTICES

1. General

   A. The series yaw damper rudder power unit components are externally mounted on the rudder power control unit. Refer to 27-17-121, Rudder Power Control Unit. The yaw damper components consist of an engage solenoid (shutoff) valve, a transfer valve, and a position transducer. (See figure 201.) It is assumed that rudder, rudder control tab and rudder control system are properly rigged and in working order. See 27-17-0.

   WARNING: PRESSURIZING HYDRAULIC SYSTEMS WILL SUPPLY POWER TO RUDDER AND ASSOCIATED SYSTEMS. CARE SHOULD BE TAKEN TO ISOLATE OR TAG SYSTEMS NOT BEING TESTED TO PREVENT INJURY TO PERSONNEL OR DAMAGE TO AIRPLANE AND EQUIPMENT.

2. Removal/Installation Yaw Damper Components

   A. Remove Yaw Damper Component

      (1) Depressurize auxiliary hydraulic system. Refer to 27-17-0, Rudder, Rudder Control Tab and Rudder Control System.

      (2) Remove left and right rudder tab control panels (1510) and (1514) and rudder nose fairing access panel 1572. See Chapter 12, Access Doors and Panels.

      (3) Break lockwire and remove 4 screws using 3/16-inch allen wrench. Rotate rudder to gain access to components.

      (4) Lift valve from mounting pad or pull position transducer straight back approximately 3-1/2 inches to clear probe housing.

      (5) Cover all openings.

      CAUTION: CARE SHOULD BE TAKEN TO PREVENT CONTAMINATION OF HYDRAULIC FLUID.

   B. Install Yaw Damper Component

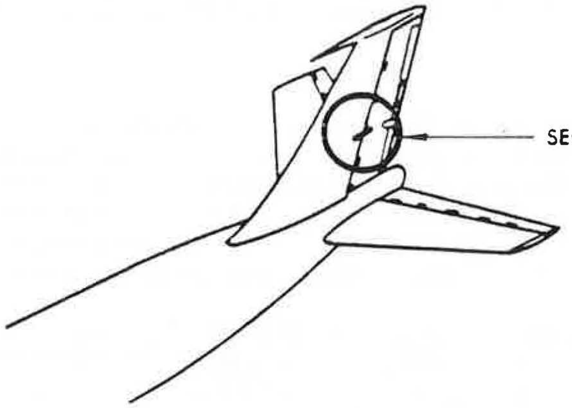
      (1) Verify that seal plate and/or o-ring are satisfactory or replace.

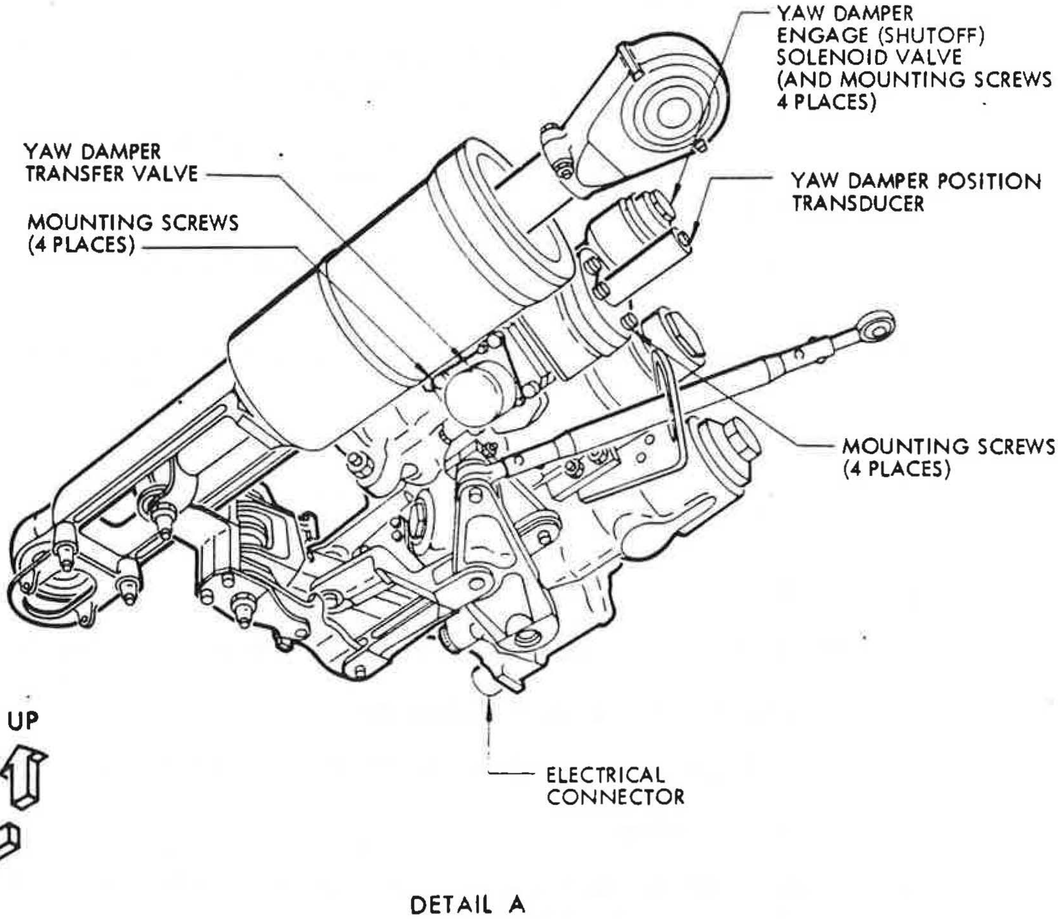
      (2) Install component on mounting pad.

      (3) Install screws and tighten to 25 to 35 pound-inches torque.

      (4) Lockwire screws.

      (5) Install access panels after checking for hydraulic fluid leakage and operation of component.





DETAIL A

Series Yaw Damper Rudder Power Control Unit Components
Figure 201

3. Inspection/Check Yaw Damper Components

A. General

(1) The yaw damper components are electrically interconnected by internal wiring between the electrical connector and individual internal connectors for each component. The continuity of the electrical wiring can be checked by using appropriate pin connectors and a testmeter, Simpson model 260 or equivalent.

(2) Check electrical continuity of wiring between pins of electrical connector. Pin to pin resistance shall agree with table below.

1 to 2	80	(± 8)	ohms	- Yaw damper solenoid valve
1 to 4	0	(short)	ohms	
5 to 6	1000	(± 100)	ohms	- Transfer valve solenoid
7 to 8	1000	(± 100)	ohms	- Transfer valve solenoid
9 to 10	90	(± 9)	ohms	- Position transducer primary
11 to 12	70	(± 7)	ohms	- Position transducer secondary

4. Adjustment/Test Yaw Damper Components

A. Refer to 22-6-0, Series Yaw Damper - Adjustment/Test.

NOTE: An optional test of the rudder power control unit and its yaw damper components may be performed by use of Boeing Elevator/Rudder Power Control Unit Test Set No. F72785. The tests are outlined in Boeing document D6-6891, Handbook of Operating Instructions - Elevator/Rudder Power Control Unit Test Set No. F72785.

SERIES YAW DAMPER HYDRAULIC PRESSURE INTERLOCK SWITCH REMOMAL/INSTALLATION

1. General

   A. A container will be necessary to catch fluid from disconnected hydraulic lines. Should any fluid spill on the airplane, decontaminate the area. See Chapter 12, Cleaning and Washing.

2. Equipment and Materials

   A. Main Gear Door Downlock, P71127 or equivalent

   B. O-ring, P/N NAS1612-4

3. Prepare to Remove Series Yaw Damper Hydraulic Pressure Interlock Switch

   A. Open SERIES YAW DAMPER circuit breakers on radio and T-R circuit breaker (P5).

   B. Depressurize rudder hydraulic system by turning rudder switch ON and de flecting the rudder with the power control unit until the rudder stops operating hydraulically. Turn rudder switch OFF.

   C. Open left main gear, heel well door and install main gear door downlock.

   WARNING: PERSONNEL SHOULD STAND CLEAR OF WHEEL WELL DOOR PATH.

   D. Depressurize utility hydraulic reservoir by slowly unscrewing filler cap three full turns. Tighten cap after compressed air has escaped.

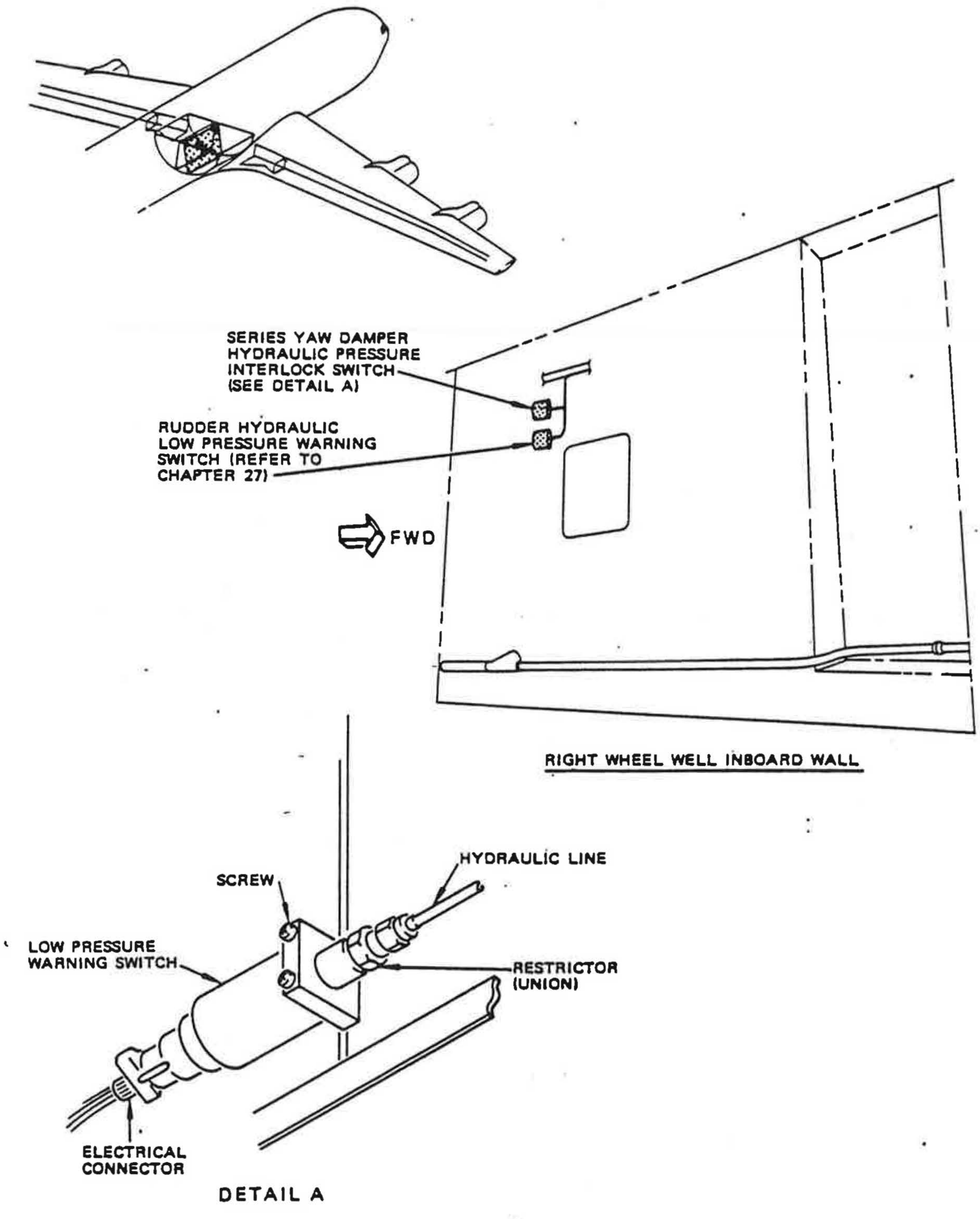
   CAUTION: DO NOT REMOVE UP UNTIL PESERVOIR IS DEPRESSURRIZED OTHERWISE FLUID WILL ESCAPE.

4. Remove Series Yaw Damper Pressure Interlock Switch

   A. Remove electrical connector plug from switah. (See figure 401.)

   B. Disconnect hydraulic line from restrictor and cap end of line.

   C. Remove switch mounting screw and remove pressure switch with restrictor. Discard the used O-ring.



Rudder Hydraulic Low Pressure Warning Switch Installation
Figure 401

5. Install Series Yaw Damper Hydraulic Pressure Interlock Switch

A. Lubricate 0-ring with fire resistant hydraulic fluid BMS 3-11 or grease MCS 352 (Monsanto). Install 0-ring, restrictor (union) on interlock switch. (see figure, 401.)

B. Position witch in mounting position and install mounting screws.

C. Connect hydraulic line to restrictor (union).

D. Install electrical connector to switch.

E. Pressurize interlock switch.

(1) Close SERIES YAW DAMPER circuit breakers on radio and T-R circuit breaker panel (P5).

(2) Connect external electrical power and position auxiliary pump switch ON.

F. Check switch and restrictor (union) for leakage.

G. Test series yaw damperr hydraulic pressure interlock switch. (See 22-6-21, Adjustment/test.)

H. Remove main gear door downlock and close left maingear wheel well door.

WARNING: PERSONNEL SHOULD STAND CLEAR OF WHEEL WELL DOOR PATH.

STABILIZER OPERATED ELEVATOR SERVO TORQUE SWITCHING SYSTEM - DESCRIPTION AND OPERATION

1. General

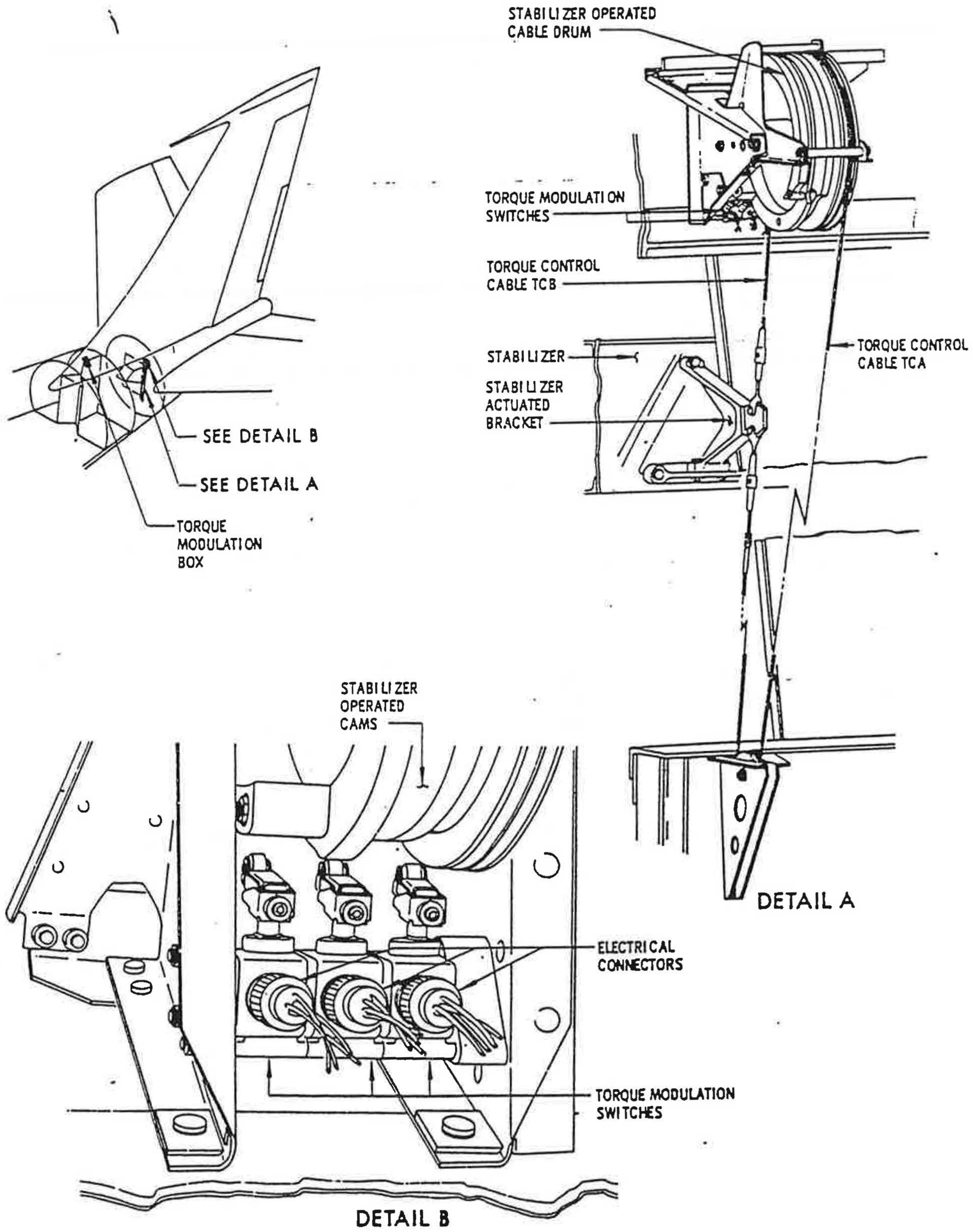
A. The stabilizer operated elevator servo torque switching system (torque modulation system) provides improved autopilot performance in the high stick-force/"g" flight regime while conforming to airplane load maneuver requirements. This improvement in autopilot pitch axis control is obtained from a fail safe stabilizer operated system. When the stabilizer is in the airplane nose-up trim region, the torque switching system decreases the value of the torque limiting resistor and thereby increases the available servo torque for the elevator.

B. The torque modulation system consists of three mechanically operated cams (driven by the stabilizer surface) which operate three torque modulation switches. The system also contains an electronic torque modulation box which is mounted in the pressurized area forward of the aft bulkhead, and is accessible from the right rear lavatory. Three torque limiting resistors are mounted in the autopilot component box RJ16 for N20199, RJ24 for N19997 - N20000.

C. The torque modulation system controls the elevator output channel of the autopilot. The control switching is mechanical and the torque modulation box selects the proper torque limiting resistors that apply the excitation voltage to the elevator servo for various positions of the stabilizer. No additional power or signal sources are necessary for the operation of the system. See fig. 1 for system component location.

2. Torque Modulation Box

A. The torque modulation box contains two capacitors, a diode and a relay and is mounted on the rear bulkhead of the airplane and is accessible through a wall panel in the right rear lavatory.



Stabilizer Operated Elevator Servo Torque Switching
System Component Location
Figure 1

3. Torque Limiting Resistors

A. The torque limiting resistors are mounted in the autopilot component box RJ16 or RJ24.

4. Torque Modulation Cams and Switches

A. The cams and switches are located in the unpressurized area adjacent to the stabilizer. (See figure 1.)

5. Operation

A. The stabilizer-operated switching is shown in figure 2.

(1) A set of three mechanically-operated cams, driven from the stabilizer control surface, operates switches S1(RS302), S2(RS303) and S3(RS304) in accordance with switching program shown in figure 2. As the stabilizer travels through the switching region between 0.5 units nose down and 1.5 units of nose up trim, these switches will operate to produce a self-monitoring function, and at 0.5 units of nose up trim (torque changeover point) will operate the double pole relay K1. The normally-open contacts of pole A are used to bypass the torque limiting resistors R1 and R2 for high torque and the remaining contacts are used in the autopilot engage interlock circuits.

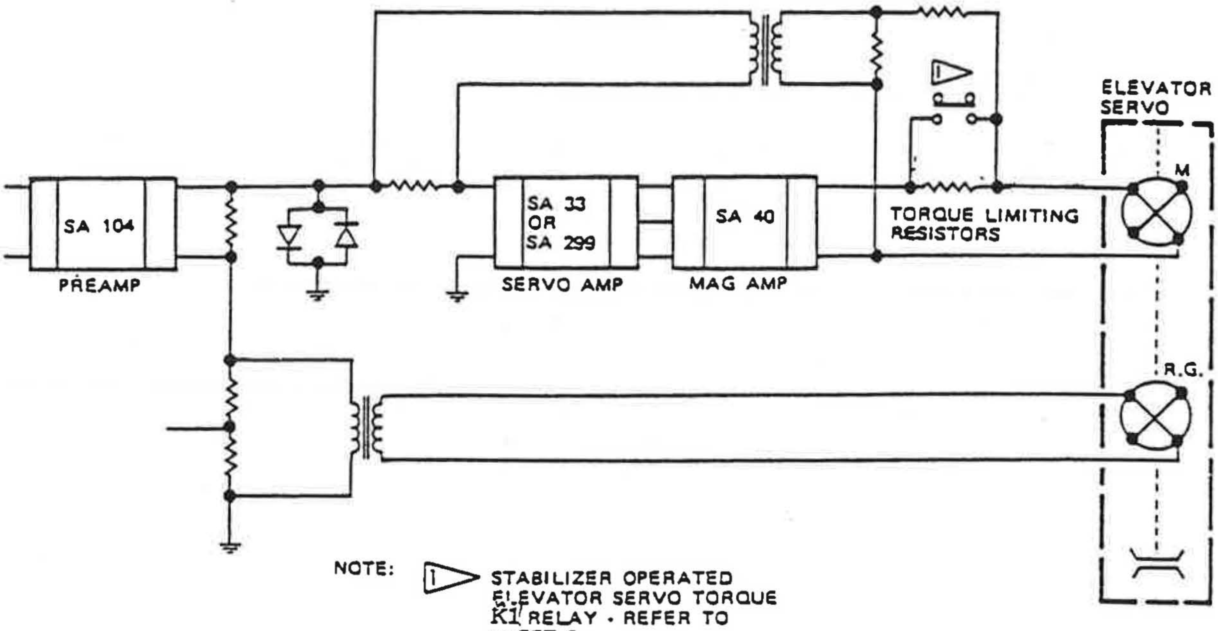
(2) Capacitor C1, which is in parallel with the autopilot engage holding solenoid and relay K1 solenoid, provides sufficient delay to prevent nuisance autopilot disengagements during contact transfer of relay K1 (events 3 to 4) and switch S1 (events 5 to 6 and 6 to 7).

(3) Diode D1 is used to provide suppression of transient voltages across the relay solenoid and switch S3.

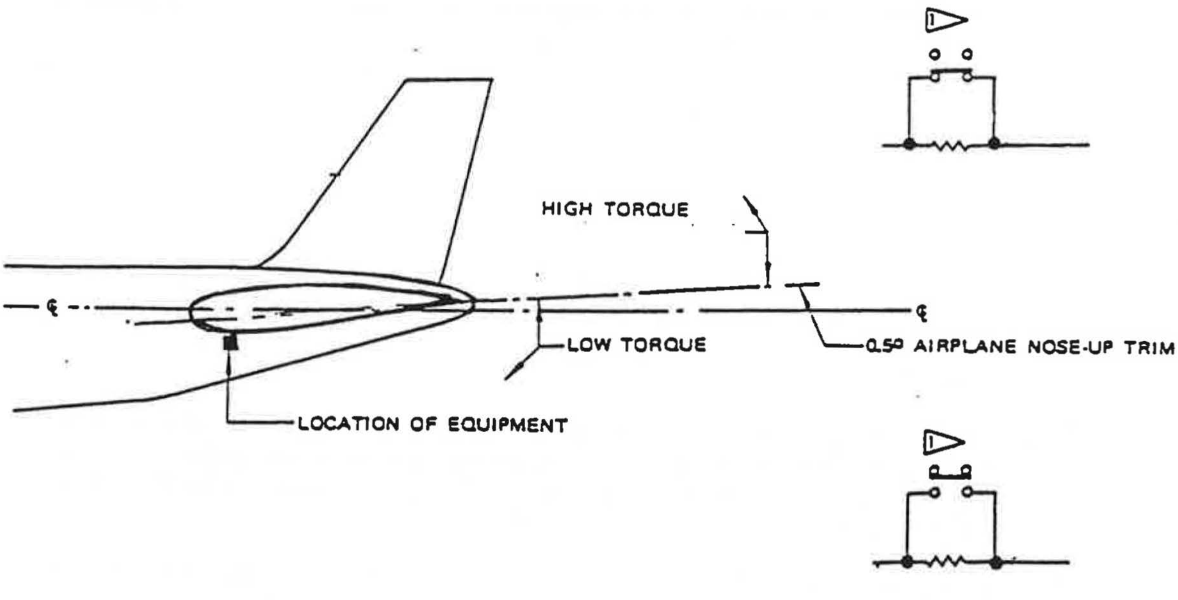
B. During normal auto-stab-trim operation the interlock network provides continuity to the autopilot engage holding solenoid through alternate current paths using the contacts of switches S1, S2 and relay K1. As the stabilizer moves in the airplane nose up direction (events 0 to 7) the following operation results:

(1) For event (0 to 1) switch S2 transfers to the Y position. This switching operation is to provide autopilot disengagement if switch S1 has failed in the Y position. During normal operation this switch event provides no function.

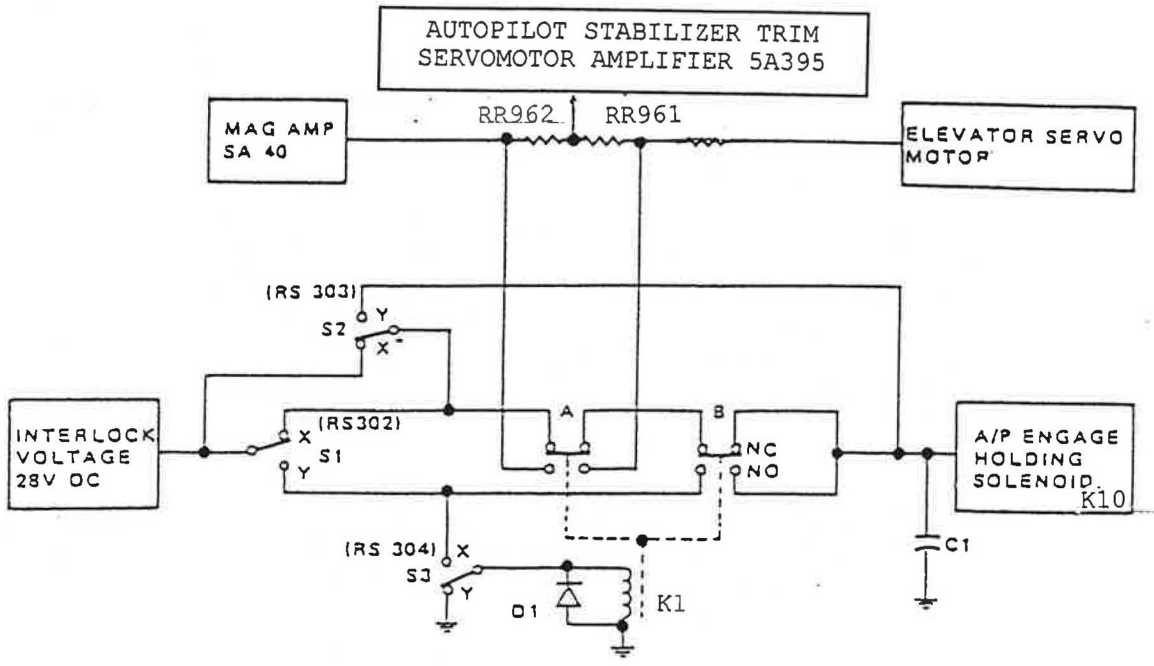
(2) For event (1 to 2) switch S2 returns to X position. The return of S2 to its original position establishes a current path necessary during the subsequent switching event.



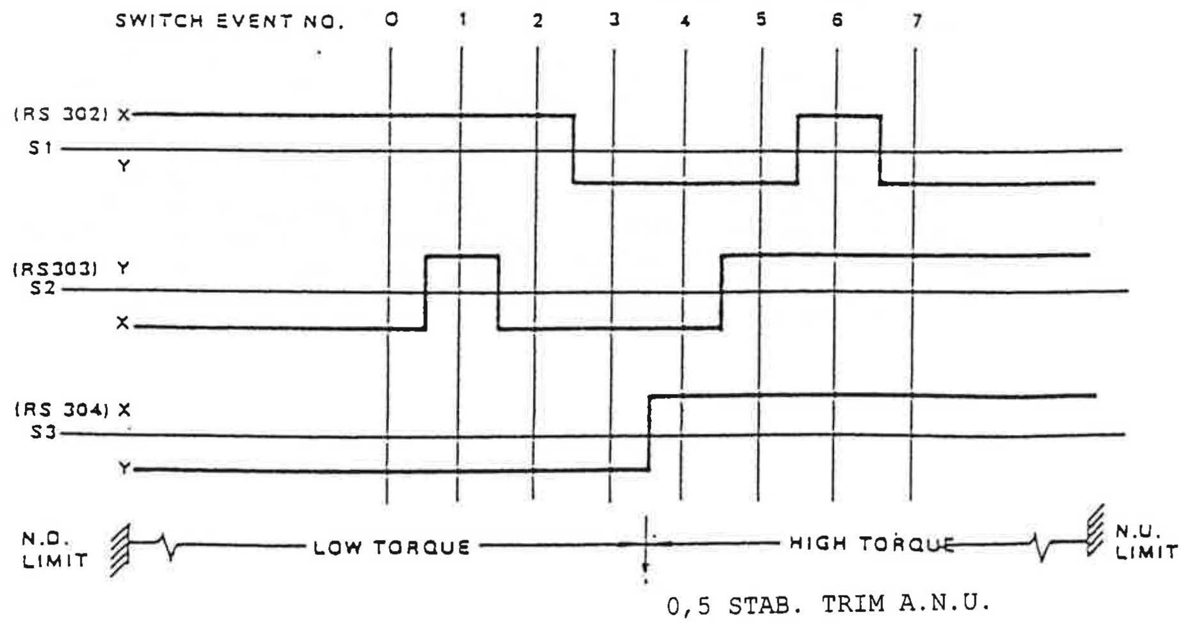
NOTE: 1 STABILIZER OPERATED ELEVATOR SERVO TORQUE K1 RELAY - REFER TO SHEET 2



Stabilizer Operated Variable Torque System
Figure 2 (Sheet 1)



INTERLOCK NETWORK - SHOWN IN LOW TORQUE MODE (EVENT NO. 0)



Stailizer Operated Variable Torque System
Figure 2 (Shet2)

(3) For event (2 to 3) switch S1 transfers to Y position. The purpose of this switching operation is to apply the interlock voltage (28 volts dc) to the X contact of S3 in preparation for switching the network to the high torque mode. Another purpose for this switching sequence is to provide interlock continuity via S2 (X) and relay K1 normally closed contacts.

(4) For event (3 to 4) switch S3 transfers to X position. Twenty-eight volts dc is applied to the K1 relay solenoid via S1(Y) and S3(X) contacts. After the relay contacts have transferred to the energized position, the contacts of pole A short-out torque limiting resistors R1 and R2 to obtain high servo torque. Interlock continuity is provided via contacts of S1(Y) and pole B of relay K1. Capacitor C1 holds autopilot engage holding solenoid energized during the switching operation.

(5) For event (4 to 5) switch S2 transfers to Y position. The purpose of this switching operation is to provide autopilot disengagement if relay contacts have failed to transfer to normally open position; and to provide a current path to keep relay K1 energized subsequent to the following switching event.

(6) For event (5 to 6) switch S1 returns to X position. During the contact transfer time of S1, the interlock voltage is removed from the relay solenoid, however, capacitor C1 provides sufficient holding current during this period to keep the relay energized. After the contacts have transferred the relay solenoid is energized via the contacts of S1(X), S2(Y), relay pole B, and S3(X).

(7) For event (6 to 7) switch S1 transfers to Y position. The purpose of this switching operation is to enable relay K1 to re-energize subsequent to an interruption in interlock voltage via switches S1(Y) and S3(X). This will allow high torque autopilot operation whenever the autopilot is engaged(or re-engaged) while the stabilizer position is in the high torque region.

C. A similar operation of the autopilot engage interlock circuit will result when the stabilizer is moving in the opposite direction (nose-down trim) or traversing any portion of the switching region. In the event of a switch or relay failure, where a contact fails to transfer electrically, as with a welded contact, the interlock network will disengage the autopilot when the stabilizer is in the switching region. A capacitor or diode failure may result in nuisance autopilot disengagements since these elements are used to provide nuisance free autopilot operation during switch and relay transients.

STABILIZER-OPERATED ELEVATOR SERVO TORQUE SWITCHES - ADJUSTMENT/TEST

1. General

   A. The stabilizer-operated servo torque switches provide an increase in available elevator servo torque at stabilizer settings between 0.5 units of airplane noseup trim and the airplane noseup stabilizer limit. (Refer to Stabilizer Trim System, Chapter 27.)

2. Torque Modulation Swithes Adjusment

   A. Equipment and Materials

      (1) Rigging Pin - 5" long x 5/16" diameter

      (2) Scale - 0 to 4 feet in inches and hundredths

      (3) Tensiometer - 0-100 lb. capacity

   B. Prepare for Adjustment

      (1) Remove stabilizer jackscrew access panel No. 1185.

      (2) Set stabilizer A dimension at 31.89 (± 0.03) inches.

         NOTE: This is 3 degrees, 30 minutes stabilizer down.

      (3) Insert rigging pin through cable drum and outer bracket. (See figure 501.

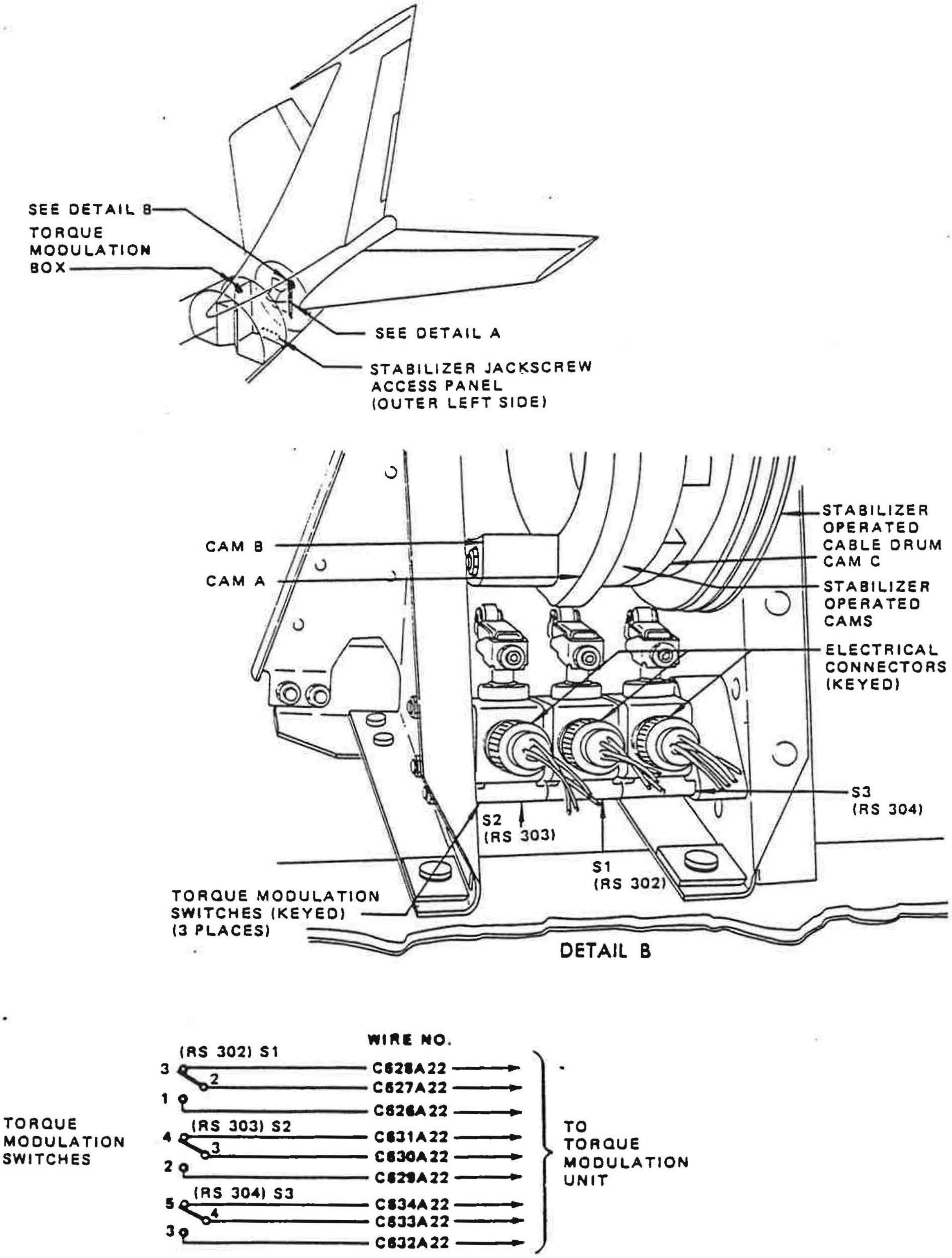
      (4) Rig cables to load indicated by temperature in Table 1.

      (5) Remove rigging pin.

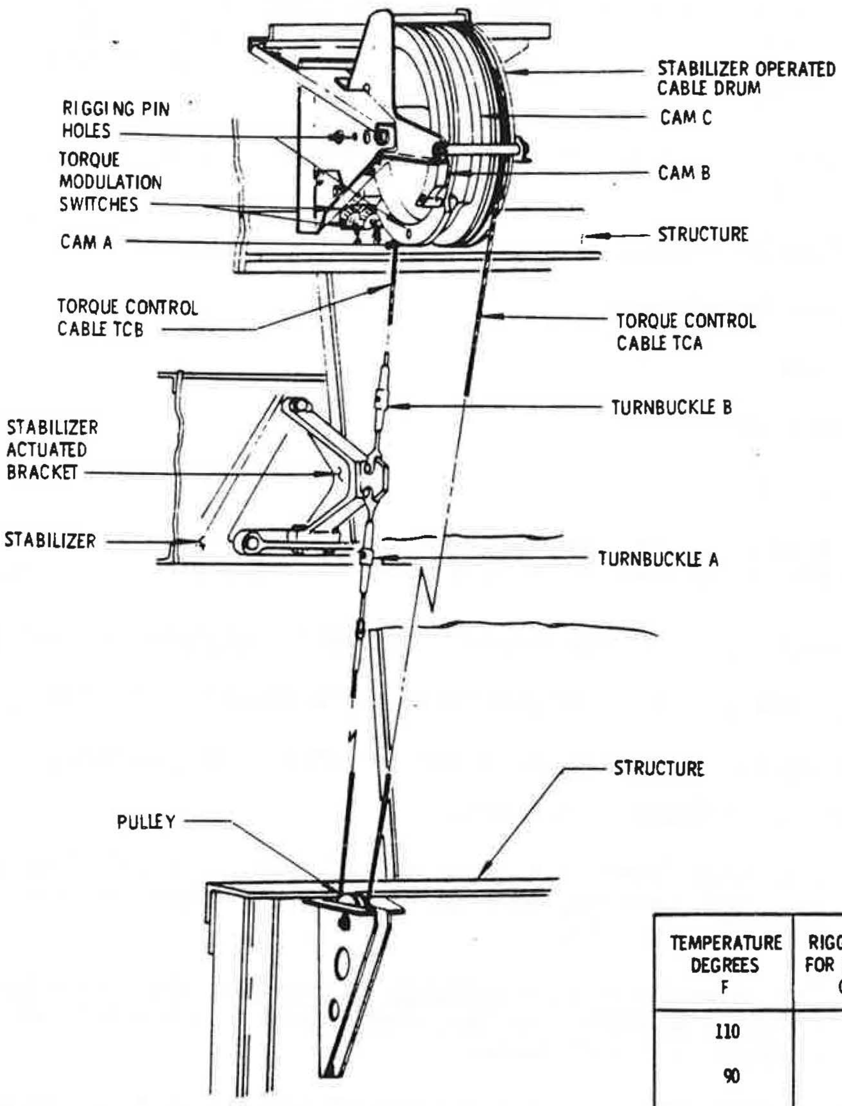
   C. Adjust Torque Modulation Switches

      (1) Position stabilizer to 2 degrees 50 minutes down by setting stabilizer A dimension to 33.01 (± 0.10) inches. set switch RS303 to actuate with cam B.

      (2) Position stabilizer to 3 degrees 10 minutes clown by setting stabilizer A dimension to 32.45 (± 0.10) inches. Set switch RS302 to actuate with cam A.



Stabilizer Operated Elevator Servo Torque Switching Installation
Figure 501 (Sheet 1)



DETAIL A

TEMPERATURE DEGREES F	RIGGING LOAD ± 10 LBS FOR STABILIZER TORQUE CONTROL CABLE
110	90
90	82
70	75
50	68
30	60
10	53
-10	45
-30	42
-40	34

TABLE I

Stabilizer Operated Elevator Servo Torque Switching Installation
Figure 501 (Sheet 2)

(3) With cables rigged as above, position stabilizer to 3 degrees 30 minutes down by setting stabilizer A dimension to 31.89 (± 0.03) inches. Set switch RS304 to actuate at this position with cam C. (See figure 501.)

(4) Test stabilizer-operated elevator torque switching system. Refer to paragraph 3.

3. Torque Modulation Switches Test

A. Equipment and Materials

(1) AC voltmeter

(2) Torque wrench

B. Prepare for Test

(1) Remove cover plate (covering hex shaft) from housing of elevator servo and mount torque wrench with adapter on hex shaft of elevator servo.

(2) Set stabilizer at approximately 0.5 units airplane nose-up position.

NOTE: Flaps must be up for these tests unless otherwise specified.

(3) Place Split Axis elevator ON-OFF switch in OFF position.

(4) Disable stabilizer trim drive.

(a) Tape stabilizer trim servo up and down limit switches in actuated position, or remove either wire from terminal AB16 in RJ23 (G19 in RJ3).

(b) Tape stabilizer trim actuator up and down limit switches in actuated position, or open stabilizer trim actuator circuit breaker on the P2 panel.

(5) Connect ac voltmeter across (output of SA40) terminals F16 and F14 in RJ23 (A26 and A28 in RJ3).

(6) Manually trim the stabilizer to zero units trim position.

C. Test Torque Modulation Switches

(1) Engage autopilot and slowly insert a pitchdown signal. Check that:

(a) Output of SA40 card is between 105 and 133 volts.

(b) Torque developed at maximum voltage is between 72 and 88 pound-inches (low torque region).

NOTE: Use calibration curve for torque wrench to correct readings. BØ excitation to SA40 card must be 135 volts.

(2) Repeat step (1) using pitchup signal.

(3) Connect ac voltmeter across terminals F13 and F14 in RJ23 (A29 and A28 in RJ3). With a pitchup signal manually trim stabilizer toward 1 unit nose-up trim. Check that voltage output increases at some position between 0-1 unit nose-up trim (high torque region). Disengage autopilot at this position.

(4) Measure stabilizer dimension "A" on stabilizer jackscrew. Check that the dimension is 31.9 (± 0.1) inches.

(5) With stabilizer positioned as in step (3), re-engage autopilot and slowly insert a pitchdown signal. Check that maximum torque developed is between 103 and 125 pound-inches. Disengage autopilot.

(6) Repeat step (5) using a pitchup signal.

NOTE: Airplanes with 10-3056-400 elevator torque adapter installed, proceed to step (7). If not, restore system to normal.

(7) With stabilizer in high torque region and with flaps down, engage autopilot, and slowly insert a pitchdown signal. Check that maximum torque developed is between 147 and 180 pound-inches.

(8) Disengage autopilot and repeat step (7) in opposite direction.

D. Restore Airplane to Normal

INTENTIONALLY LEFT BLANK

NOTE: The insertion of this TR has to be listed in the Record of Temporary Revisions at the beginning of Volume 1.

MACH TRIM SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Mach Trim System

A. General

(1) The mach trim system analyzer is designed to locate malfunctions in the mach trim system which occur outside the mach trim coupler. The mach trim coupler should be removed, the tester installed in its place and the test procedure outlined in Table I, should be followed.

(2) If the outlined procedure does not locate a malfunction in the system, the faulty component may safely be assumed to be the mach trim coupler. The mach trim coupler should be replaced with a coupler known to be in good working condition.

(3) If the test procedure does locate a fault, the faulty component should be replaced or wiring repaired as necessary. The mach trim coupler that was removed prior to the test may be installed if a single fault is assumed.

NOTE: The mach trim coupler has not been checked. To make sure that the complete system is operating properly, a mach trim coupler known to be in good working condition should be installed. This last step is necessary only when the system has a double fault.

B. Equipment and Materials

(1) Mach Trim System Analyzer, Boing F-71375 or equivalent.

(2) Pitot-Static System Tester, Boing EDS/PS-200 or equivalent.

(3) Two sets of interphones.

C. Prepare for Trouble Shooting of Mach Trim System

(1) Close ESSENTIAL RADIO BUS circuit breakers (3) and RADIO BUS NO.2 circuit breakers (3) on A-C BUS NO.2 circuit breaker panel (P2).

(2) Close ESSENTIAL RADIO BUS and RADIO BUS NO.2 Toggle Switches on radio and T-R circuit breaker panel (P5)

(3) Close applicable vertical gyro circuit breaker or insert fuse on radio and T-R circuit breaker panel (P5) as applicable.

(4) Open all autopilot and mach trim a-c and d-c ciruit breakers or remove fuses as applicable on radio and T-R circuit breaker panel (P5)

(5) Open KIFIS breaker or remove fuse as applicable on radio and T-R circuit breaker panel (P5).

(6) Remove mach trim coupler from electronic equipment rack.

(7) Install mach trim system analyzer in place of mach trim coupler.

(8) Install an interphone communication system between the pilot or copilot station and the electronic equipment rack station. This can be accomplished by using existing airplane wiring and interphone-jack facilities (See Chapter 23).

NOTE: Two operators are needed to perform the test and it is essential for the operators to be able to communicate with each other.

(9) Position one operator at the pilot's or copilot's station.

(10) Position one operator at the electronic equipment rack with the mach trim analyzer.

NOTE: Components called out in the following testing procedure are located on the mach trim system analyzer if not otherwise specified. Terminal numbers are located in RJ3 for LX-N20199, RJ23 for LX-N19997 & LX-N20000 if not otherwise specified.

(11) Follow the testing procedure outlined in Table I

Step	Indication OK	Malfunction-Replace Component or Repair Wiring as Necessary
1. Close A/P & MACH TRIM DISENG LT D-C circuit breaker on circuit breaker panel.	MACH TRIM DISENGAGED light (on engine instrument panel (P9).	NO light, check for 28V DC at * Check warning light and warning light on circuitry. *D30 for LX-N20199 *AD13 for LX-N19997, LX-N20000.
2. Close AUTOPILOT A-C circuit breaker (on radio and T-R circuit breaker panel).	AUTOPILOT ON light on (if POWER Switch is in AUTOPILOT position, lights will also come on as in step 3.).	No light, see trouble shooting Chart "A".
3. Close MACH TRIM CONTROL A-C ciruit breaker (on radio and T-R circuit breaker panel (P5). Power Switch must be Mach Trim position.	TRIM RELAY light and SAFETY SW TEST light on.	a. If TRIM RELAY light fails to come on, see trouble shoot ing Chart "B" b. If SAFETY SW TEST light fails to come on, see trouble shooting Chart "C" c. Note step 35.
4. On airplanes having mach potentiometer in copilot's computer unit, (KIFIS control unit), close copilot's KIFIS circuit breaker or insert fuse (on P5) as applicable.		
5. Set POWER Switch to AUTOPILOT Position.	∅ A, ∅ B and ∅ C A-C POWER lights on. PHASE ROTATION RIGHT light on; PHASE ROTATION WRONG light out.	If any or all lights are out, check autopilot circuit breaker (on P5).
6. Set POWER Switch to MACH TRIM position.	∅ A, ∅ B and ∅ C A-C POWER lights on. PHASE ROTATION RIGHT light on; PHASE ROTATION WRONG light out.	If any or all ∅ lights are out, check mach trim circuit breaker (on P5). If all ∅ lights are on, but PHASE ROTATION WRONG light is ON and PHASE ROTATION RIGHT light is OFF; check phase rotation.

Table I (Sheet 1 of 9)

Step	Indication OK	Malfunction-Replace Component or Repair Wiring as Necessary
7. Hold STAB TRIM MOTOR NOSE UP - NOSE DN Switch in NOSE UP position. For single speed trim servo hold for 6 seconds. For two speed trim servo hold for 8 seconds.	Stabilizer position indicator (on the control stand) shall move up 1 (± 1/4) unit.	If stabilizer position indicator does not move up as required, check trim servo (on lower right side of the actuator assembly) and limit Switches (on airplane structure adjacent to the jackscrew). SAFETY SW LIGHT may blink but should not be off longer than it takes the operator to say ONE quickly.
8. Hold STAB TRIM MOTOR SHORTS Switch in SHORTS position while holding STAB TRIM MOTOR NOSE UP-NOSE DN Switch in NOSE UP position.	Stabilizer trim shall stop.	If stabilizer trim does not stop, assure that wire to pin 17 of plug J1 (RD1317) is not grounded.
9. Relase STAB TRIM MOTOR SHORTS Switch; release STAB TRIM MOTOR NOSE UP-NOSE DN Switch.		
10. Hold STAB TRIM MOTOR NOSE UP-NOSE DN Switch in NOSE DN position. For single speed trim servo hold for 6 seconds. For two speed trim servo hold for 8 seconds.	Stabilizer position indicator shall move down 1 (± 1/4) unit.	If stabilizer position indicator does not move up as required, check trim servo (on lower right side of the actuator assembly) and limit switches (on airplane structure adjacent to the jackscrew). SAFETY SW LIGHT may blink but should not be off longer than it takes the operator to say ONE quickly.

Table I (Sheet 2 of 9)

Step	Indication OK	Malfunction-Replace Component or Repair Wiring as Necessary
11. Hold STAB TRIM MOTOR SHORTS Switch in SHORTS position while holding STAB TRIM MOTOR NOSE UP-NOSE DN Switch in NOSE DN position.	Stabilizer trim shall stop.	If stabilizer trim does not stop, assure that wire to pin 17 (of plug J1 RD1317) is not grounded.
12. Release STAB TRIM MOTOR SHORTS Switch; release STAB TRIM MOTOR NOSE UP-NOSE DN Switch. 13. Trim stabilizer (manually or with main electric stabilizer trim motor) so that trim indicator reads 0 units. <u>NOTE:</u> to employ main electric trim motor, observe step 24. Set "STAB TRIM" cutout Switch in the ON position, or the STAB TRIM MAIN ELECTRIC cutout Switch in NORMAL (on right hand side of control stand) as applicable. 14. Set voltmeter selector Switch to POS XMTR.	 Voltmeter indication should be as shown in Table II.	 a. If no, or incorrect voltage reading, see trouble shooting Chart D. b. Note step 18.
15. Hold POS XMTR SHORTS Switch in upward position.	No significant change in voltage noted previously.	If a change in voltage is noted, assure that wires to pins 24 and 32 of plug J2 (M7W RD1318) are not grounded.

Table I (Sheet 3 of 9)

Step	Indication OK	Malfunction-Replace Component or Repair Wiring as Necessary
16. Hold POS XMTR SHORTS Switch in downward position.	No significant change in voltage noted previously.	If a change in voltage is noted, assure that wires to pins 24 and 32 of plug J2 (M7W RD1318) are not grounded.
17. Release POS XMTR SHORTS Switch. 18. Hold POS XMTR POLARITY Switch in downward position.	Voltage shall decrease 0.60 (± 0.25) volts from reading obtained in step 14.	If voltage does not decrease as required, assure that wiring is not reversed in airplane or resistor box. Assure that stabilizer position transmitter linkage is not reversed NOTE: Looking at stabilizer position transmitter and stabilizer from left side of airplane when stabilizer leading edge moves down, position transmitter shaft must rotate clockwise.
19. Release POS XMTR POLARITY Switch. 20. Trim stabilizer 1 unit nose down manually or with main electric trim motor (see NOTE in step 13.).	Voltmeter reading shall decrease from reading attained in step 14 (Se Table II.).	If voltmeter indication is incorrect, check resistors in resistor box (See wiring diagram 22-1-6.). Assure that position transmitter linkage is not reversed.
21. Hold STAB TRIM MOTOR SHORTS Switch in SHORTS position.	Voltmeter reading shall go to zero volts.	If voltmeter reading does not go to zero volts, assure that wire to pin 25 of plug J2 (M/W RD1318) is not grounded.

Table I (Sheet 4 of 9)

Step	Indication OK	Malfunction-Replace Component or Repair Wiring as Necessary
22. Release STAB TRIM MOTOR SHORTS Switch. 23. Set MACH TRIM CUTOUT Switch to ON or AUTOMATIC STABILIZER trim cutout Switch to NORMAL (on control stand) as applicable. 24. Close STAB TRIM CONT DC circuit breaker (on P5) and STAB TRIM A-C circuit breaker (on P2 breaker panel). NOTE: This step may have been done with step 13. 25. Hold pilot trim Switch (on pilot's wheel) in NOSE UP position momentarily.	TRIM RELAY light out; STAB TRIM ON warning light (on control stand) on.	a. If TRIM RELAY light fails to come out, check STAB TRIM cut out Switch or STAB TRIM MAIN ELECTRIC cutout Switch (on control stand) whichever is used and manual STAB TRIM CONT relay (on J18 relay shield). b. If STAB, TRIM ON warning light fails to come on, check warning light and warning light wiring.
26. Hold pilot trim Switch in downward position momentarily.	TRIM RELAY light out; STAB TRIM ON warning light on.	If TRIM RELAY light fails to come on, check STAB TRIM cutout Switch or STAB TRIM MAIN ELECTRIC cutout Switch (on control stand) whichever is used.

Table I (Sheet 5 of 9)

Step	Indication OK	Malfunction-Replace Component or Repair Wiring as Necessary
27. Repeat step 25 and step 26 using copilot's trim Switch (on copilot's wheel).		
28. Set autopilot engage Switch (on autopilot control panel to the AUTOPILOT position). Airplanes with AUTOMATIC STAB TRIM cutout Switch require Switch in NORMAL position before MACH SW and AUTOPILOT ENGAGE lights will go on.	AUTOPILOT ENGAGED LIGHTS ON, AUTOPILOT DISENGAGED lights (on P9 engine instrument panel) OFF and autopilot audio alerter will not sound.	If lights fail to come on as specified, check autopilot, and wires to the following: pin 30, J1 (M/W-RD1317) pin 31, J1 (M/W-RD1317) pin 23, J1 (M/W-RD1317) pin 24, J1 (M/W-RD1317)
29. Set POWER Switch to AUTOPILOT position.	NOTE: COMPARISON UNIT INTERLOCK Switch must be in BYPASS if comparison unit is part of system.	
30. Hold STAB TRIM MOTOR NOSE UP-NOSE DN Switch in NOSE UP position. Push on control column. For single speed trim servo push for 6 seconds. For two speed trim servo push for 24 seconds.	Stabilizer position indicator (on control stand) shall move up 1 (± 1/4) unit.	If stabilizer position indicator does not move up as required, check autopilot.
31. Hold STAB TRIM MOTOR NOSE UP-NOSE DN Switch in NOSE DN position. Pull on control column. For single speed trim servo pull for 6 seconds. For two speed trim servo pull for 24 seconds.	Stabilizer position indicator shall move down 1 (± 1/4) unit.	If stabilizer position indicator does not move down as required, check autopilot.
NOTE: Operation of STAB TRIM MOTOR Switch is unnecessary in steps 30 and 31 for airplanes incorporating Bendix E/P two speed trim servo.		

Table I (Sheet 6 of 9)

Step	Indication OK	Malfunction-Replace Component or Repair Wiring as Necessary
32. Release STAB TRIM MOTOR NOSE UP-NOSE DN Switch. Push on control column.		If stabilizer position indicator moves, check trim servo. Check for trim servo wiring shorting to plug J1 (M/W RD1317).
NOTE: Delete step 32 for airplanes with two speed trim servo installed.		
33. WARNING: WAIT 30 SECONDS (FOR TRIM SERVO MOTOR ARMATURE TO STOP ROTATING) BEFORE PROCEEDING.		
34. Set POWER Switch to MACH TRIM position.		
35. Hold SAFETY SW TEST Switch in downward position.	SAFETY SW TEST light out.	If light remains on, check limit switches and trim servo and repair or replace as necessary.
36. Pull AUTOPILOT A-C circuit breaker (P5).		
37. Pull A/P & MACH TRIM DISENG LT CB (P5).		
38a.Set MACH TRIM cutout Switch to on or AUTOMATIC STAB TRIM cutout Switch to NORMAL as applicable.	MACH TRIM CUTOUT Switch ON light on; MACH TRIM DISENGAGE light (on P9 engine instrument panel off).	Perform step 38b.
38b.Set MACH TRIM cutout switch or AUTOMATIC STAB TRIM cutout switch as applicable to CUTOUT position (control stand).	MACH TRIM CUTOUT Switch ON light on; MACH TRIM CUTOUT Switch ON light off, and MACH TRIM DISENGAGED light will be dimmer than usual.	If lights do not perform as required, check MACH TRIM cutout Switch or AUTOMATIC STAB TRIM cutout Switch (on control stand) as applicable. Check diodes in resistor box (See wiring diagram 22-1-6).

Table I (Sheet 7 of 9)

Step	Indication OK	Malfunction-Replace Component or Repair Wiring as Necessary
39. Position voltmeter selector Switch to BIAS position.	Voltmeter indication should be as shown in Table II.	If voltmeter reading is not as specified, check resistors in resistor box. (See wireing diagram 22-1-6).
40. Position voltmeter selector Switch to MACH POT position.	Voltmeter shall indicate 0 volts.	In the event of an erroneous indication, check mach synchrotel transmitter as applicable; check KIFIS control unit or servo torque unit as applicable.
41. Hold MACH TRIM TEST Switch (on P9 engine instrument panel) in TEST position.	Voltmeter reading should be as shown in Table II.	In the event of erroneous indications: a. Check resistors in resistor box or mach trim component box. (See wiring diagram 22-1-6). b. Note step 44.

Table I (Sheet 8 of 9)

Step	Indication OK	Malfunction-Replace Component or Repair Wiring as Necessary
42. Release MACH TRIM TEST Switch		
43. On airplanes with mach potentiometer in KIFIS control unit only: Position AUTO STAB TRIM cutout Switch to NORMAL (on control stand) as applicable and hold copilot's KIFIS Switch (copilot's instrument panel) in TEST position.	MACH SW CLOSED light on. Light should be steady and of normal brilliance. Voltmeter readings should be as shown in Table II.	a. If MACH SW CLOSED light shows indications different from those specified, or if voltmeter readings are erroneous, check resistors in resistor box (aft of left electronic equipment rack). b. Note step 44.
44. NOTE: This step is not necessary if steps 41 through 43 are satisfactory and if malfunction of synchrotel transmitter is not suspected. With a pitot-static system tester, supply the airplane pitot-static system with pitot and static pressures sufficient to simulate mach numbers between 0.75 and 0.94.	 Voltmeter readings should be as shown in Table II.	 See steps 41 through 43.

Table I (Sheet 9 of 9)

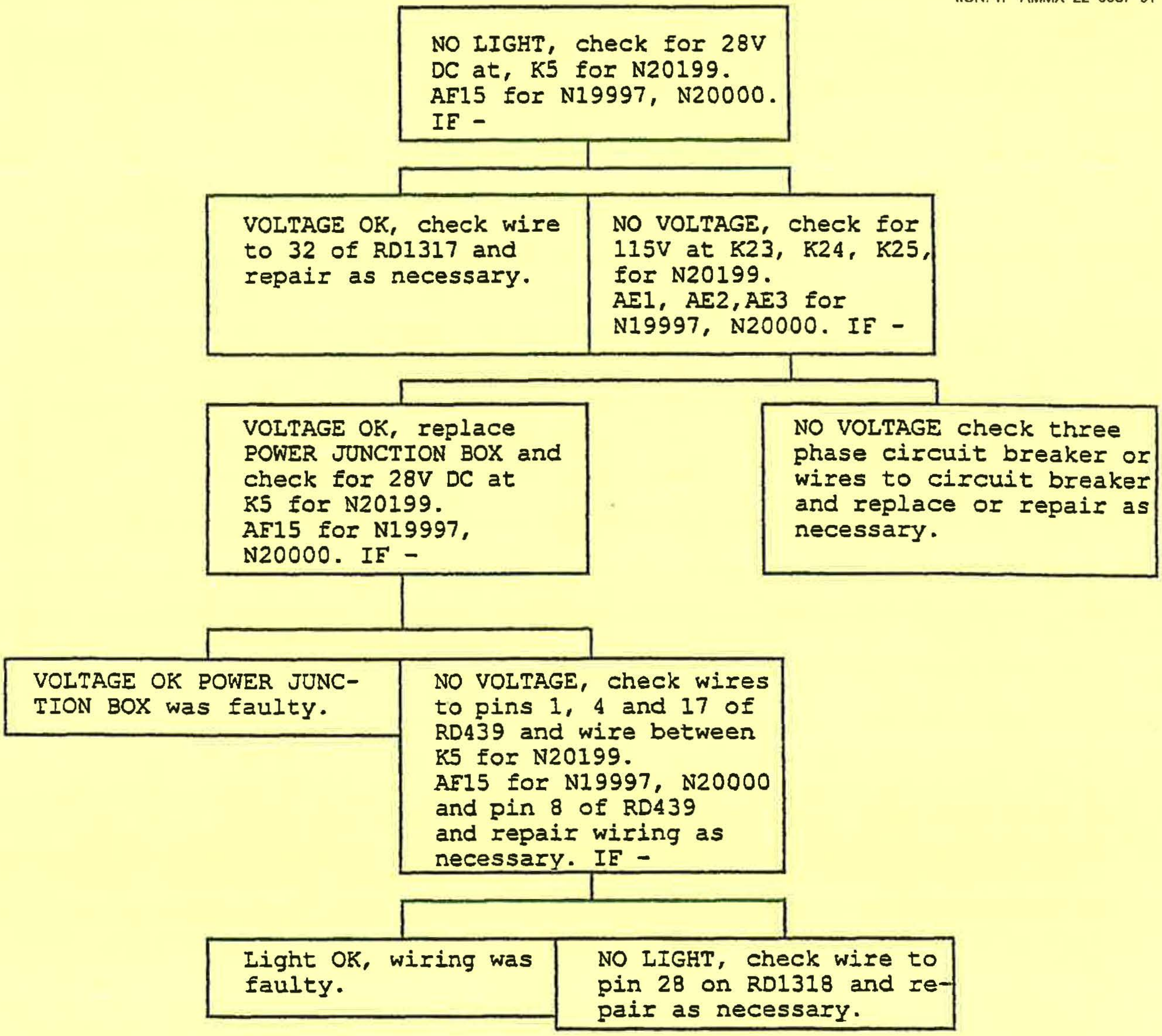
CONDITION	707-300C
POSITION TRANSMITTER- 0 Units of Trim on Trim Indicator	0.50 to 1.5 V
Trim Stabilizer Airplane Nose Down to	Voltage decreases 0.25 to 0.45 V/unit
BIAS	1.1 to 1.4 V
MACH TRIM TEST If from Resistor Box	
MACH TRIM TEST or F/O If from Mach Pot (.917 M)	0.70 to 1.10 V
OUTPUT OF MACH POTENTIOMETER ENGINES [1]	0.81 M
M = 0.87	
M = 0.90	
M = 0.86	0.10 to 0.20 V
M = 0.88 [2]	0.20 to 0.40 V [2]
M = 0.92	0.80 to 1.10 V

[1] When mach number is increasing, MACH SWITCH CLOSED light comes on 0.02 to 0.00 Mach below this point. Output of MACH POT begins 0.00 to 0.02 mach above this point and increases smoothly if mach number is increased slowly,then decreases smoothly if mach number is decreased slowly to mach number where output began. MACH SWITCH CLOSED light shall go off 0.00 to 0.02 mach below the point at wich it came on with increasing mach.

@@@ [2] Voltage at 0.88 M shall be 2.0 (± 0.5) times the voltage at 0.86 M but do not exceed table value.

Table II

ICN: 1P-AMMX-22-0037-01



NOTE: RD439 is the PWR J-BOX 22-1-3.

Trouble Shooting Chart A

IICN: 1P-AMMX-22-0038-01

TRIM RELAY LIGHT fails to come on, check for 28V DC at H23 for N20199 AC7 for N19997, N20000.
IF -

NO VOLTAGE, check wire to pin 28 of RD1317 and repair as necessary.

VOLTAGE OK, check for 28V DC at K10 for N20199, AC6 for N19997, N20000.
IF -

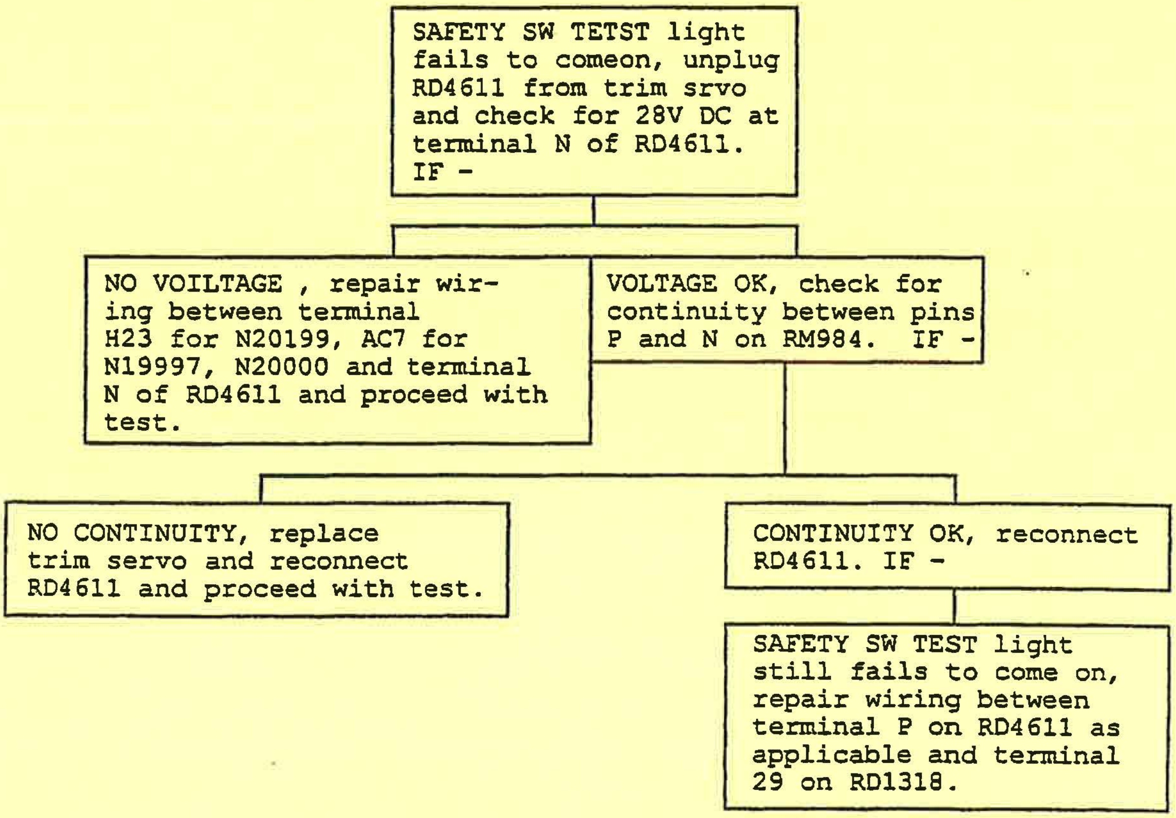
NO VOLTAGE , check stabilizer trim control relay (on STAB TR8IMI CONT SCHIELD J18) and replace or repair as necessary.
IF -

VOLTAGE OK, check wire from K10 for N20199, AC6 for N19997, N20000 to pin 29 of RD1317 and repair as necessary.

NO LIGHT, check wires from H23 for N20199, AC7 for N19997, N20000 and K10 for N20199, AC6 for N19997, N20000 to stabilizer trim control relay and repair as necessary.

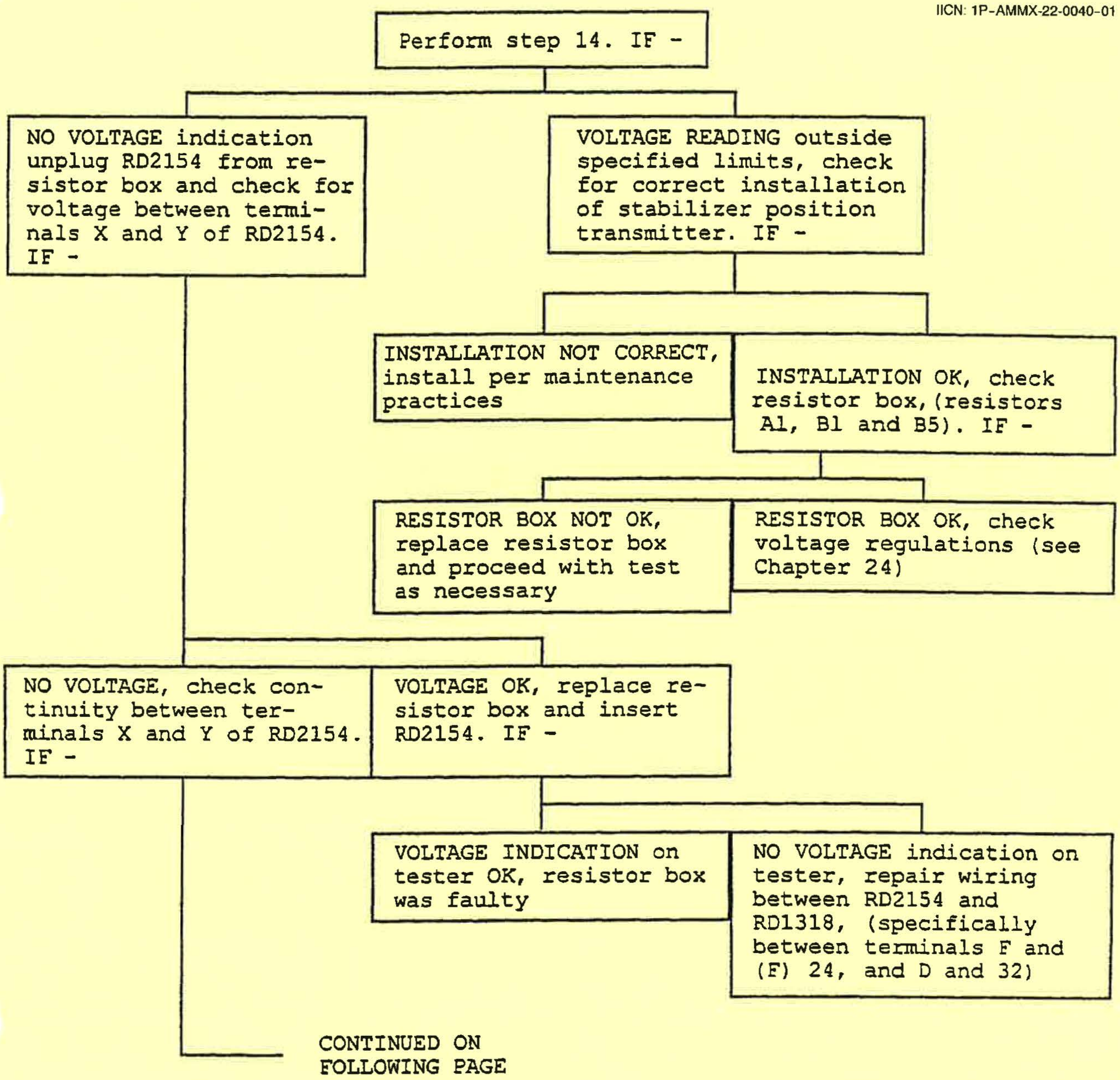
Trouble Shooting Chart B

ICN: 1P-AMMX-22-0039-01



Trouble Shooting Chart C

IICN: 1P-AMMX-22-0040-01



Trouble Shooting Chart D (Sheet 1)

IICN: 1P-AMMX-22-0041-01

CONTINUED FROM
PRECEDING PAGE

CONTINUITY OK, Plug in RD2154, remove RD1318 from mach trim coupler and check for continuity between pins 25 and 26 of RD1318. IF -

NO CONTINUITY, replace stabilizer position transmitter and recheck.

CONTINUITY OK, check for voltage at pins 25 and 26 of RD1318 receptacle on mach trim coupler. IF -

NO CONTINUITY, replace stabilizer position transmitter and recheck.

VOLTAGE OK, replace stabilizer position transmitter and recheck.

NO VOLTAGE, replace mach trim coupler.

Trouble Shooting Chart D (Sheet 2)

END

INTENTIONALLY LEFT BLANK

INTENTIONALLY LEFT BLANK

NOTE: The insertion of this TR has to be listed in the Record of Temporary Revisions at the beginning of Volume 1.

MACH TRIM SYSTEM - DESCRIPTION AND OPERATION

1. General

A. The purpose of the mach trim system is to position the horizontal stabilizer of the airplane as a function of mach number in order to provide a positive control column force gradient at high mach numbers when the autopilot is disengaged. The system consists of a Mach Trim Coupler, stabilizer position transmitter, stabilizer trim servo, automatic stabilizer trim cutout switch, Monitor Test Switch, resistor box and warning light (see fig. 1). The stabilizer trim unit contains a servo motor with two different speed-torque characteristics. The mach trim system drives the high speed motor and the autopilot, when engaged, drives the motor at both speeds. Refer to Chapter 22-24-0 Autopilot Flight Control Component.

B. It is not possible to use the automatic mach trim system when the autopilot is engaged, and when the stabilizer trim system is electrically driven. The system is designed to function at airplane speeds of mach 0.82 and above. When the above conditions are met, the stabilizer will be positioned automatically as a function of mach number (see fig. 2).

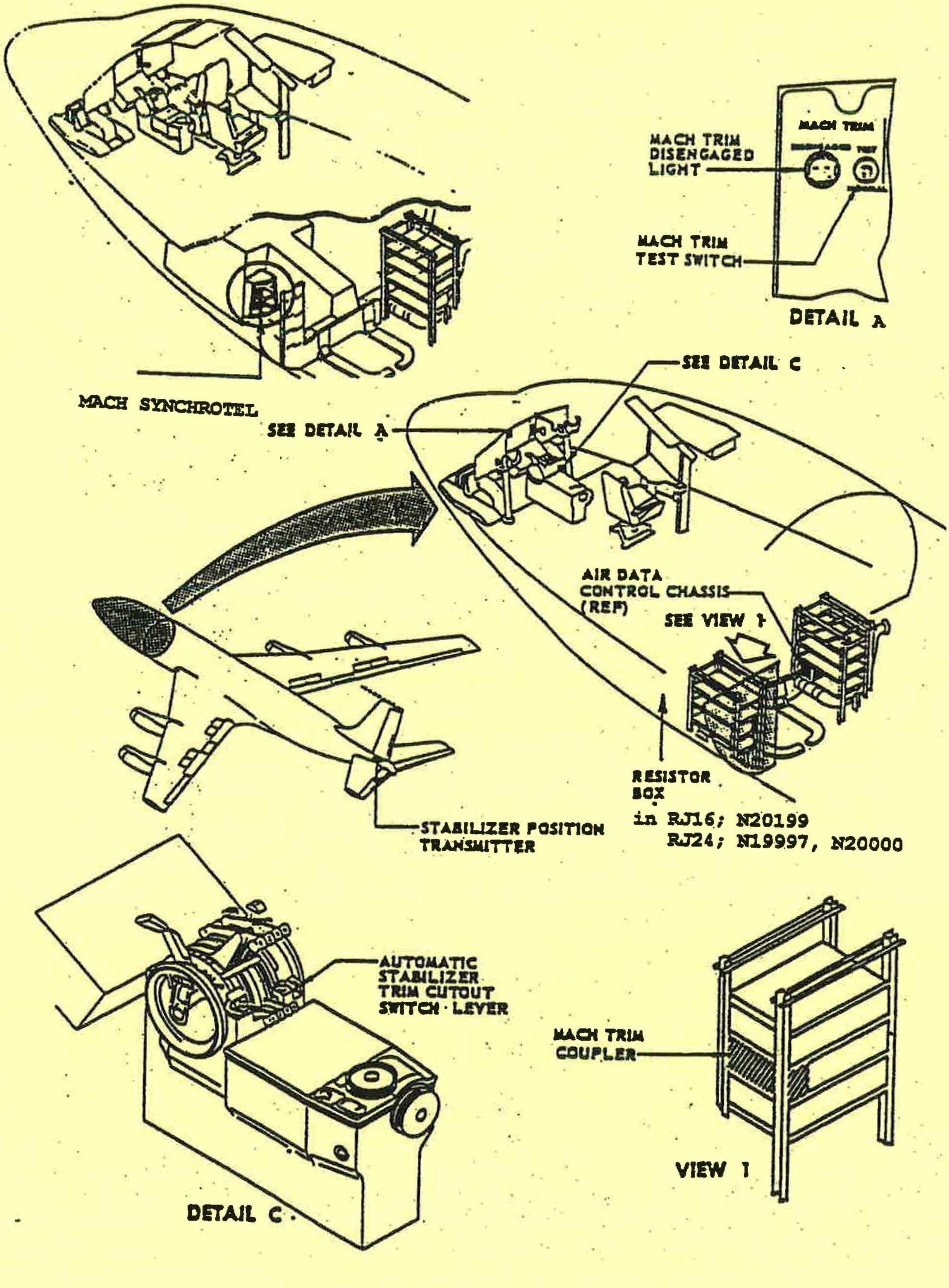
C. The mach trim system receives mach number information from a mach synchrotel transmitter (see fig. 1). This information then goes to the (copilot's) computer unit in the Integrated Air Data System, (KIFIS) which operates the mach switch and mach pot located in the computer unit, to control the mach trim system.

D. The automatic stabilizer trim cutout switch is located within the control stand. When the actuating lever on the top of the control stand is in the NORM position, the Switch is ON. In the aft CUTOUT position, the Switch is OFF.

2. Mach Trim System Controls

A. Direct manual control of the mach trim system is provided by the automatic stabilizer trim cutout switch. When the switch lever is in the NORMAL position, the circuits to the trim servo clutch relay coils are closed, and the system warning light is armed (see fig. 2). When one of the relays is energized, 28V d-c is furnished to the power relay and the clutches for either clockwise or counterclockwise rotation of the stabilizer. The direction of rotation depends on which relay is being energized (see fig. 3). Three-phase power is now being furnished to the fast speed stabilizer trim servo motor. When the automatic stabilizer trim cutout switch lever is in the CUTOUT position, the mach trim system is de-energized, the warning light is on to indicate the system is inoperative, and it also opens the autopilot interlock circuit.

IICN: 1P-AMMX-22-0031-01



Mach Trim System Component Location
Fig. 1

B. Simultaneous operation of the main electrical stabilizer trim system and mach trim system is prevented by a manual trim relay in the Mach Trim Coupler. This relay is energized when the stabilizer trim control switch is not operated. When the stabilizer trim control switch is in use, the manual trim is not energized, the signal paths to the trim servo clutch relays are opened and the clutch solenoids are shorted.

3. Mach Trim Coupler

A. General

(1) The Mach Trim Coupler contains a signal preamplifier, an electro-mechanical synchronizer, a control amplifier, an auto-trim actuator amplifier, a monitor amplifier and several relays. The coupler acts as the control center for the mach trim system and supplies the nesessary control signals to the trim servo. It also contains protective and warning circuits to guard against malfunctions and possible violent maneuvers during system operation.

B. Coupler Power Supply

(1) Operating power for the mach trim system is taken from A and C phases of the flight instrument bus No.2 in the radio and T-R circuit breaker panel (P5). This power is converted in the Mach Trim Coupler power supply to the voltages necessary for all components of the system as follows:

(a) 5-volts phase A is supplied to the mach signal potentiometer for furnishing control signals to the system.

(b) 30-volts phase C, and 15-volts phase A plus 180 degrees, are supplied to energize the electro-mechanical synchronizer card motor.

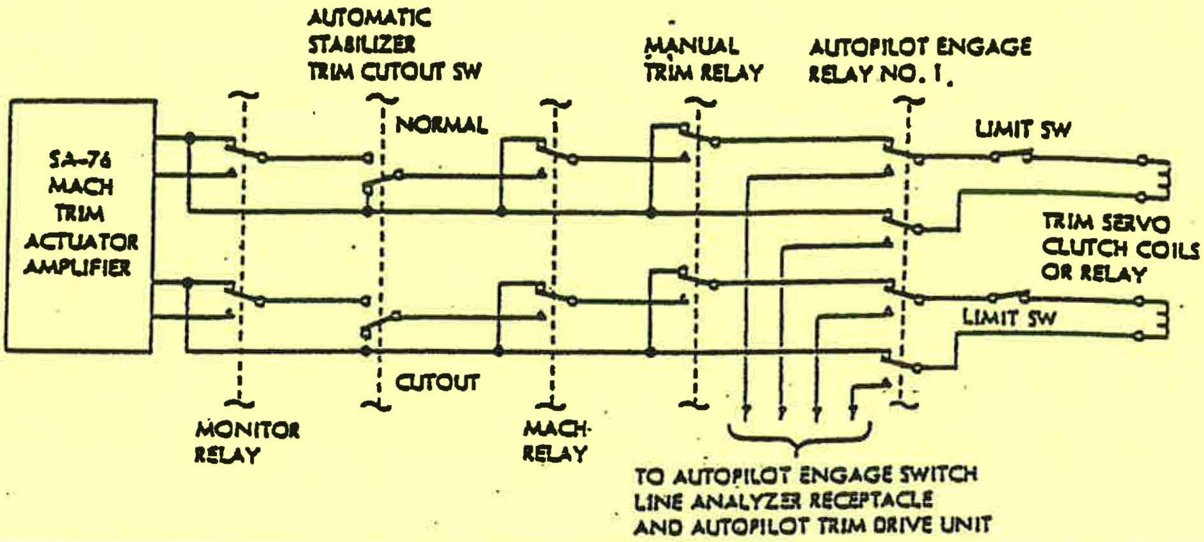
(c) 19.4-volts phase A and 10.3-volts phase C are combined to supply 26-volts at a phase angle lagging phase A by 20 degrees. This lagging voltage is used to excite the electro-mechanical synchronizer signal synchro and stabilizer trim position transmitter.

(d) 28-volts d-c power is supplied to all relay control circuits.

(e) 28-volt filtered d-c power is supplied to the amplifier and monitor cards.

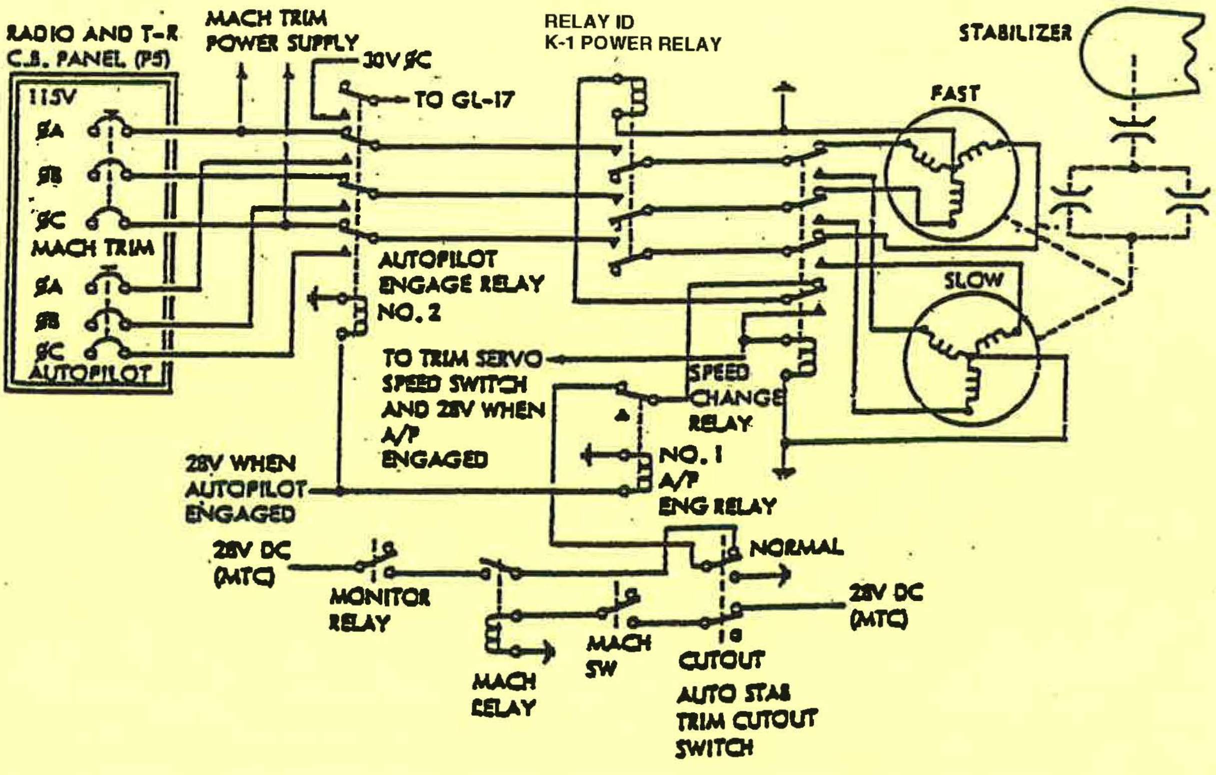
(f) 0-volts phase A is supplied as a reference.

IICN: 1P-AMMX-22-0002-01



Trim Servo Power Control Circuit
Fig. 2

IICN: 1P-AMMX-22-0026-01



Trim Servo Power Control Circuit
Fig. 3

(2) The Mach Trim Coupler power supply is energized whenever the flight instrument bus No.2 in the radio and T-R circuit breaker panel (P5) is energized.

C. Autopilot Engage Relays

(1) Two interlock relays prevent simultaneous operation of the mach trim system and autopilot system. These relays are autopilot engage relay No. 1 and autopilot engage relay No. 2. Both relays are energized when the autopilot is engaged (see fig. 3).

(2) When autopilot engage relay No. 1 is energized, control of the trim servo clutches and power relay is transferred from the mach trim actuator amplifier to the autopilot actuator amplifier.

(3) When autopilot engage relay No. 2 is energized, operating power for the stabilizer trim servo motor is transferred from the mach trim power source to the autopilot power source, and power is supplied to the electro-mechanical synchronizer.

D. Manual Trim relay (K2 is in Mach Trim Coupler)

(1) The manual trim relay may be energized from the autopilot power junction box or the Mach Trim Coupler power supply when the main stabilizer trim relay is not energized (see Chapter 27). Diodes prevent current flow between the power supply sources (see fig. 4). The relay is energized through a thermal delaying circuit which delays the relay closing by 0.75 second.

(2) When the manual trim relay is de-energized (main stabilizer trim system is electrically operated) the autopilot interlock circuit is opened, the control circuit to the trim servo clutches are opened and power is supplied to the synchronizer motor.

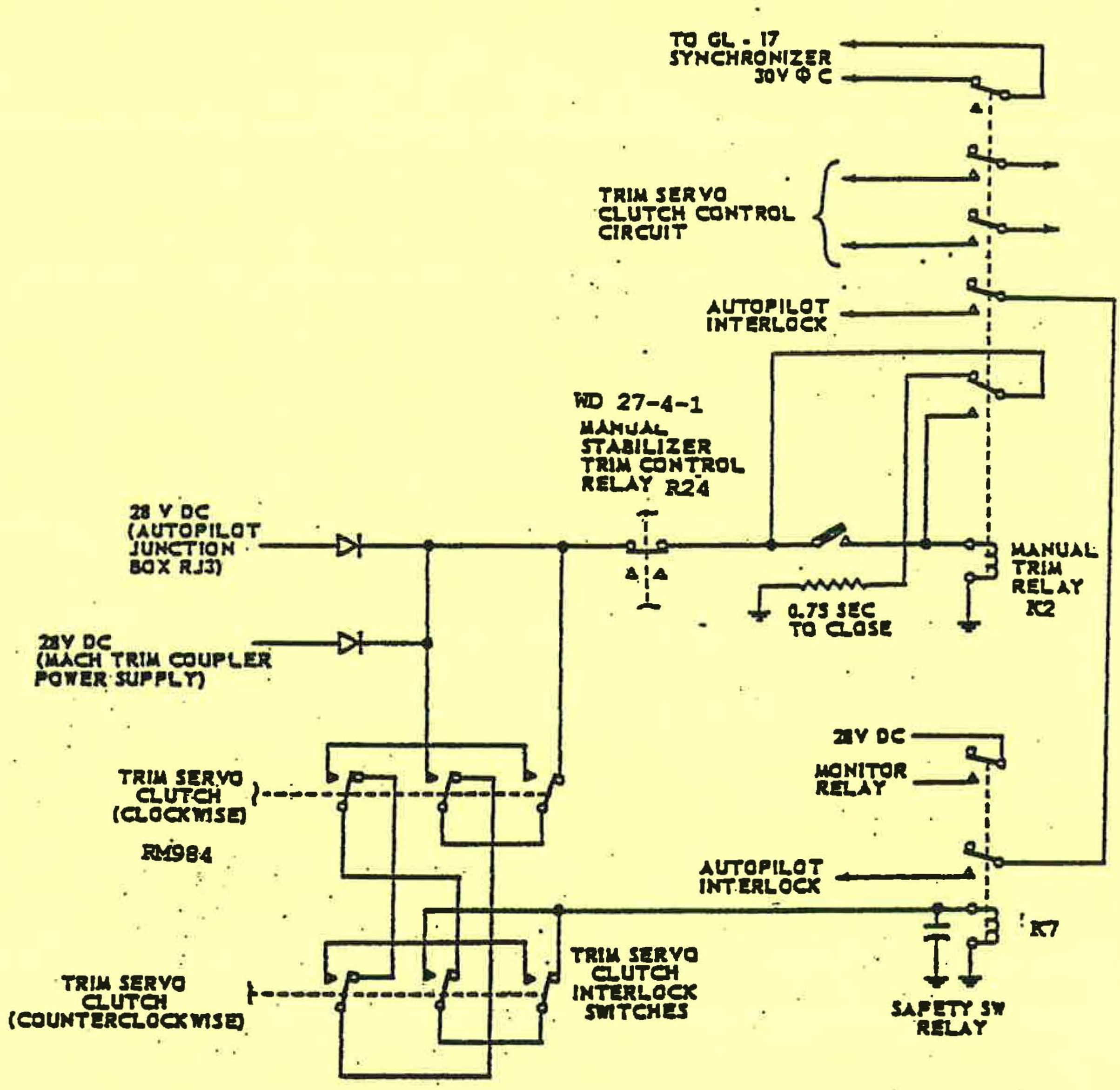
(3) When the manual trim relay is energized, a closed path is provided for the autopilot interlock circuit and for signals to the trim servo coil clutches.

E. Safety Switch Relay (K7 is in Mach Trim Coupler)

(1) The safety switch relay coil is connected in series with interlock switches in the trim servo so that the relay will be de-energized if both trim servo clutches should be engaged at once (see fig. 4).

(2) When the safety switch relay is energized, two sets of contacts are operated. One set of contacts is connected in series with contacts of the manual trim relay in the autopilot interlock circuit. The second set of contacts is connected in series with other relay contacts to complete a ciruit to the monitor relay.

IICN: 1P-AMMX-22-0032-01



For more details, see WD 22-1-0 SCH 7

Manual Trim and Safety Switch Relay Circuits
Fig. 4

F. Monitor Relay

(1) Autoprotection against malfunction is furnished by the monitor amplifier which controls the monitor relay. Contacts on the monitor relay control output of the auto-trim actuator amplifier and control application of power to the trim servo motor power relay and clutches. An additional set of contacts is used in an interlock for the monitor relay coil so that the relay cannot be energized after it has been tripped unless the automatic stabilizer trim switch is turned to CUTOUT and then to NORMAL. (see fig. 5).

(2) When the automatic stabilizer trim cutout switch lever is in the NORMAL position and power is applied to the Mach Trim Coupler, the relay will be pulled in by the initial pulse of current through the capacitors, and if the safety switch relay is energized and the monitor amplifier senses proper circuit operation, the monitor relay will stay energized by the power through the closed contact in the monitor amplifier. When the automatic stabilizer trim cutout switch lever is in CUTOUT position the monitor relay will stay energized in all conditions.

G. Mach Relay

(1) When the automatic stabilizer trim cutout switch lever is in the NORMAL position the mach relay will be energized if the mach switch is closed (see fig. 6).

(2) When the mach relay is energized, three sets of contacts complete the control circuit between the auto-trim actuator amplifier and trim servo clutches, and the circuit to the trim servo power relay. When the mach relay is not energized, these circuits are opened.

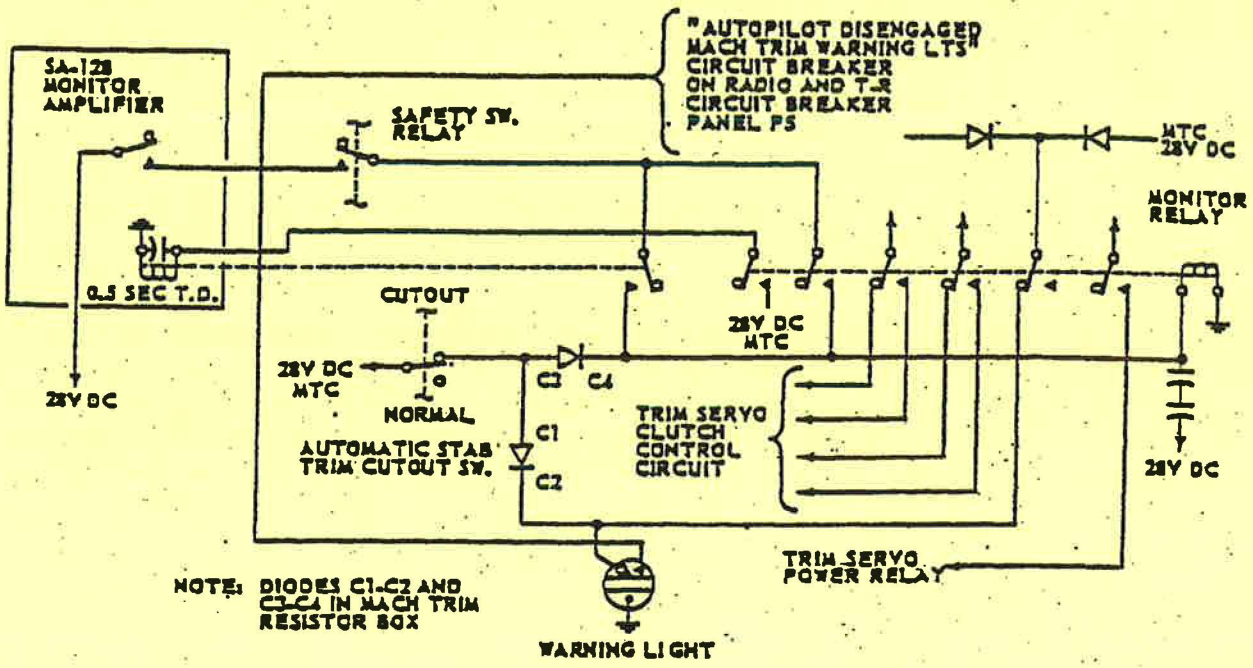
H. Preamplifier

(1) Mach signals from the mach potentiometer in the integrated air data instrument system, synchronizer signals from a synchronizing autosyn and position signals from the stabilizer position transmitter are combined and fed to a preamplifier. If the combined inputs from the three signal sources are not balanced, an error signal will exist. This error signal is amplified and sent to a control amplifier (see fig. 7).

I. Control Amplifier

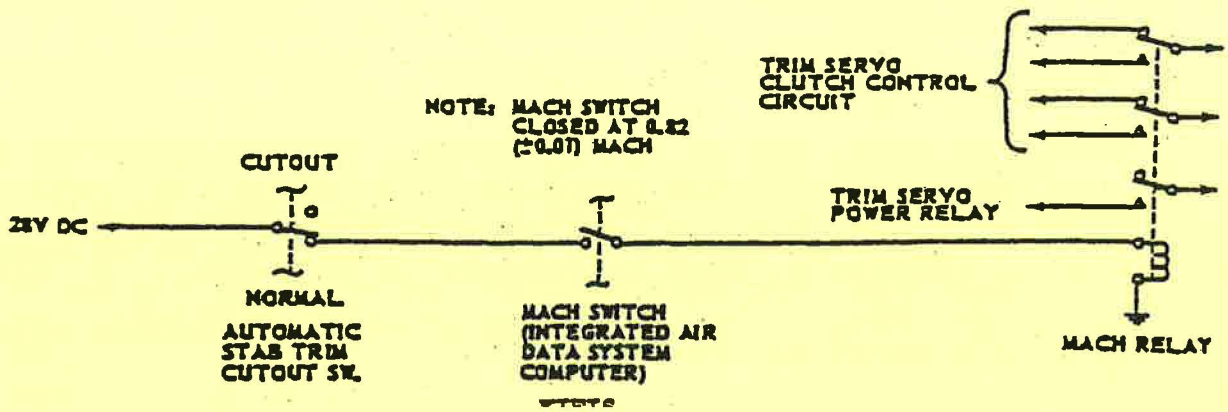
(1) The control amplifier amplifies the a-c error signal and converts the amplified signal into a d-c signal in a Ramey toroid phase discriminator. The d-c signal is fed to a magnetic amplifier to develop controlled a-c power for operation of the electro-mechanical synchronizer. The control amplifier also provides an a-c signal to an auto-trim actuator amplifier (see fig. 7).

IICN: 1P-AMMX-22-0033-01



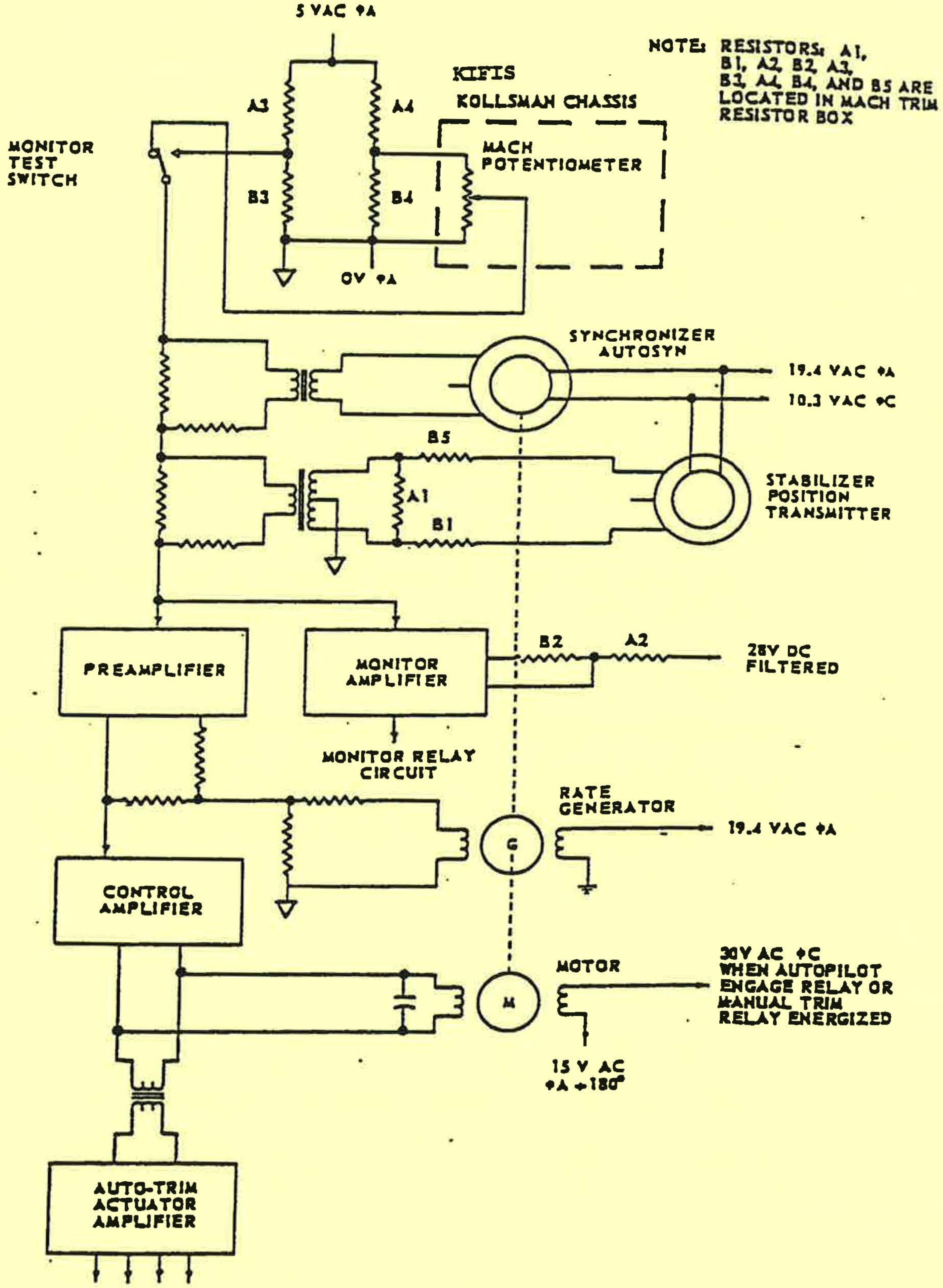
Monitor Relay and Warning Circuit
Fig. 5

IICN: 1P-AMMX-22-0034-01



Mach Relay Circuit
Fig. 6

IICN: 1P-AMMX-22-0035-01



Mach Trim Signal Circuit
Fig. 7

(2) Whenever the autopilot is engaged, or whenever the main electrical stabilizer trim system is operated, an electro-mechanical synchronizer motor receives excitation for the reference phase. Error signals developed at the input of the preamplifier are amplified and used by the variable phase of the synchronizer motor. The motor drives a rate generator and autosyn transmitter. Output of the autosyn is used to cancel any difference between the mach input signal and stabilizer position signal, thus keeping the mach trim system synchronized with manual flight control or autopilot. Output of the synchronizer rate generator is fed through the control amplifier and used to damp out any tendency of the synchronizer motor to overshoot when the error signal is cancelled.

J. Auto Trim Actuator Amplifier

(1) Output of the control amplifier is fed to an auto-trim actuator amplifier. This amplifier limits the signal by diode limiters and amplifiers it in a step-up transformer to increase sensitivity and reduce power loss. The a-c signal is then changed to dc in magnetic amplifier demodulators and used to operate magnetic amplifier switches. The amplifier switches control the clutches in the trim servo which actuates the stabilizer trim mechanism (see fig. 2 and 7).

K. Synchronizer

(1) In order to prevent rapid changes in attitude when the mach trim system is initially engaged, balancing or synchronizing signals must be supplied to make up the difference between signals from the mach potentiometer and stabilizer position transmitter. The synchronizing signal is supplied by a motor driven autosyn (see fig. 7).

(2) The two-phase synchronizer motor receives excitation on the reference phase only when the autopilot is engaged or when the main electrical stabilizer trim system is operated. Any error signals appearing at the input of the control amplifier is amplified and fed to the variable phase of the motor, which then operates and turns the synchronizer autosyn to a new position. When the autosyn reaches a position at which its output cancels the error signal, the system is synchronized and the motor stops.

(3) A rate generator is used to prevent synchronizer overshoot. Output of the rate generator is fed through the control amplifier to the variable phase of the synchronizer motor. When the error signal is balanced, rate generator output act in opposition to motor operation, and rapidly stops the motor to prevent overshoot.

L. Monitor Amplifier

(1) Protection against violent maneuvers or extreme attitudes caused by malfunction of the mach trim system is provided by a monitor amplifier. This amplifier senses the error signal and compares it to a reference signal. If the error signal reaches a predetermined magnitude the monitor amplifier causes the monitor relay to become de-energized and disconnects the mach trim system from the trim servo. A warning light on the engine instrument panel is then energized to inform the pilot that the system is disconnected. The monitor trips if signal voltage exceeds that equivalent to one degree of stabilizer movement. The warning light will also illuminate if the automatic stabilizer trim cutout switch lever is in the CUTOUT position (see fig. 5 and 7).

(2) A one-half second delayed opening relay is in the monitor amplifier to eliminate nuisance disconnect of the system when transient signal difference is presented. The warning light will blink momentarily, but the system is not disconnected. (see fig. 5)

4. <u>Monitor Test Switch</u>

A. The monitor amplifier can be checked during ground operation by a monitor test switch on the engine instrument panel. When the switch is actuated, a fixed error signal is substituted for the mach signal and applied to the Mach Trim Coupler input. The error signal should cause the monitor amplifier to disconnect the mach trim system. The monitor circuit cannot be armed again until the automatic stabilizer trim cutout switch lever is actuated to CUTOUT and back to NORMAL, and the fixed error signal is removed from the amplifier input (see fig. 5 and 7).

<u>WARNING</u>: DO NOT ACTUATE THE MONITOR TEST SWITCH DURING FLIGHT.

5. <u>Mach Potentiometer and Mach Switch</u>

A. Electrical signals corresponding to mach number are taken from a potentiometer in the KIFIS system (see fig. 6 and 7). The potentiometer is driven by a servo system controlled by output of a mach transmitter in the machmeter. Refer to Chapter 34-17-0, "Altitude Alert System (KIFIS)".

B. The KIFIS also operates the mach switch. The switch is closed at approximately mach 0.83 and is open at speeds less than mach 0.83. When the switch is closed, the mach relay in the Mach Trim Coupler is energized.

6. Stabilizer Position Transmitter

A. The stabilizer position transmitter is an autosyn which is connected to the stabilizer through a mechanical linkage. The autosyn motor is excited by 26-volt a-c, and induces a voltage in the stator windings. The phase and magnitude of the induced voltage is compared to reference signals to determine direction and magnitude of trim error.

7. Resistor Box

A. The resistor box contains two diodes used in the warning light circuit and fixed resistors used an voltage dividers and resistance bridges to modify error and reference voltages for use by the mach trim system. The resistor box is located in the lower nose compartment (see fig. 1).

8. Mach Trim Warning Lights

A. A warning light is provided to indicate that the mach trim system is disengaged. The system is disengaged when the automatic stabilizer trim cutout switch lever is in the CUTOUT position, or when the monitor relay is de-energized. The warning light is located on the pilot's engine instrument panel (see fig. 1).

9. Operation

A. During normal flight operation, the mach trim system is on stand-by when the mach trim circuit breaker and the autopilot and mach trim disengage light circuit breaker are closed on radio and T-R circuit breaker panel (P5), and the automatic stabilizer trim cutout switch lever is in NORMAL position.

B. The system will start operation when the main electrical stabilizer trim system is not used, the autopilot disengaged, and the mach number is 0.82 or above. When these conditions exist, any change in mach number will cause the horizontal stabilizer to be positioned to correct for the change. For example, if the airplane increases speed because of a nose down attitude the mach trim system will cause the stabilizer to move a nose up trim position. Conversely, if the mach number decreases, the mach trim system will cause the stabilizer to move to a nose down trim position. The system will therefore automatically adjust the horizontal stabilizer position in accordance with the proper function of mach number such that positive static stability is obtained throughout the airplane speed range. Positive static stability is defined as a requirement for an increasing positive stick force for trim with increasing speed.

C. When the main electrical stabilizer trim system or autopilot is used, the mach trim system is inoperative and the synchronizer is energized to wash out any signal feed back from the stabilizer position transmitter. However, operation of the emergency manual stabilizer trim system will not override the mach trim system and the stabilizer position transmitter feed back signal will not be washed out.

END

MACH TRIM SYSTEM - DESCRIPTION AND OPERATION

1. General

A. The purpose of the Mach Trim System is to position the horizontal stabilizer of the airplane as a function of mach number in order to provide a positive control column force gradient at high mach numbers when the autopilot is disengaged. The system consists of a mach trim coupler, stabilizer position transmitter, stabilizer trim servo, automatic stabilizer trim cutout switch, monitor test switch, resistor box and a warning light. (See figure 1.) The stabilizer trim unit contains a servo motor with two different speed-torque characteristics. The mach trim system drives the high speed motor and the autopilot, when engaged, drives the low speed motor. Refer to autopilot flight control components.

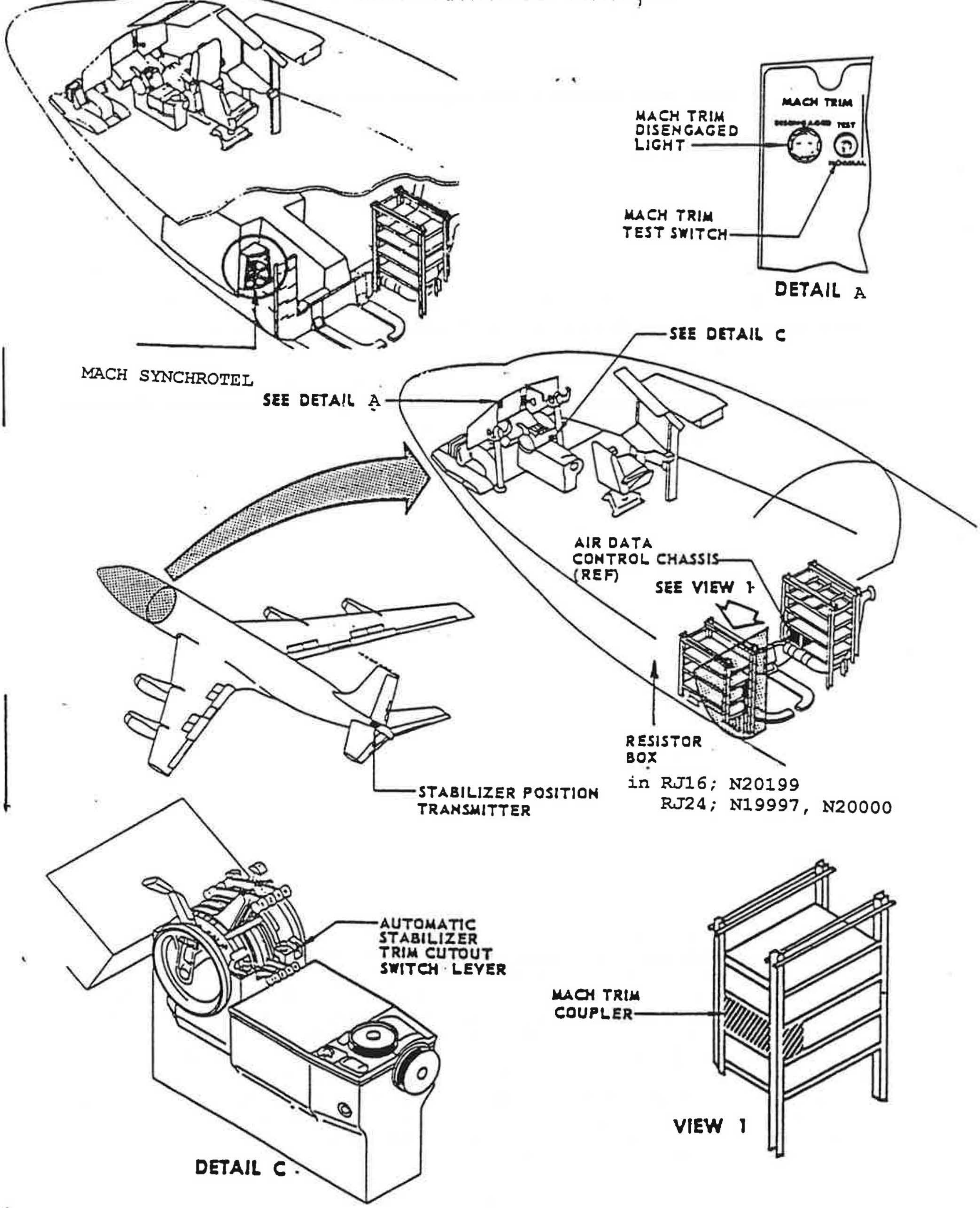
B. It is not possible to use the automatic mach trim system when the autopilot is engaged, and when the stabilizer trim system is electrically driven. The system is designed to function at airplane speeds of mach C.82 and above. When the above conditions are met, the stabilizer will be positioned automatically as a function of mach number. (See figure 2.)

C. The Mach Trim System receives, mach number information from a mach synchrotel transmitter. (See figure 1.) This information then goes to the (copilot's) computer unit in the integrated Air Data System, (KIFIS) which operates the mach switch and mach pot located in the computer unit,to control the mach trim system.

D. The automatic stabilizer trim cutout switch is located within the control stand. When the actuating lever on the top of the control stand is in the "NORM" position, the switch is on. In the aft "CUTOUT" position, the switch is off.

2. Mach Trim System Controls

A. Direct manual control of the mach trim system is provided by the automatic stabilizer trim cutout switch. When the switch lever is in the "NORMAL" position, the circuits to the trim servo clutch relay coils are closed, (figure 2) and the system warning light is armed. When one of the relays is energized, 28V d-c is furnished to the power relay and the clutches for either clockwise or counterclockwise rotation of the stabilizer. The direction of rotation depends on which relay is being energized. (See figure 3.) Three-phase power is now being furnished to the fast speed stabilizer trim servomotor. When the automatic stabilizer trim cutout switch lever is in the "CUTOUT" position, the mach trim system is de-energized, the warning light is on to indicate the system is inoperative, and it also opens the autopilot interlock circuit.



Mach Trim System Component Location
Figure 1

B. Simultaneous operation of the main electrical stabilizer trim system and mach trim system is prevented by a manual trim relay in the mach trim coupler. This relay is energized when the stabilizer trim control switch is not operated. When the stabilizer trim control switch is in use, the manual trim relay is not energized, the signal paths to the trim servo clutch relays are opened and the clutch solenoids are shorted.

3. Mach Trim Coupler

A. General

(1) The mach trim coupler contains a signal preamplifier, an electro-mechanical synchronizer, a control amplifier, an auto-trim actuator amplifier, a monitor amplifier and several relays. The coupler acts as the control center for the mach trim system and supplies the necessary control signals to the trim servo. It also contains protective and warning circuits to guard against malfunctions and possible violent maneuvers during system operation.

B. Coupler Power Supply

(1) Operating power for the mach trim system is taken from A and C phases of the flight instrument bus No. 2 in the radio and T-R circuit breaker panel (P5). This power is converted in the mach trim coupler power supply to the voltages necessary for all components of the system as follows:

(a) 5-volts phase A is supplied to the mach signal potentiometer for furnishing control signals to the system.

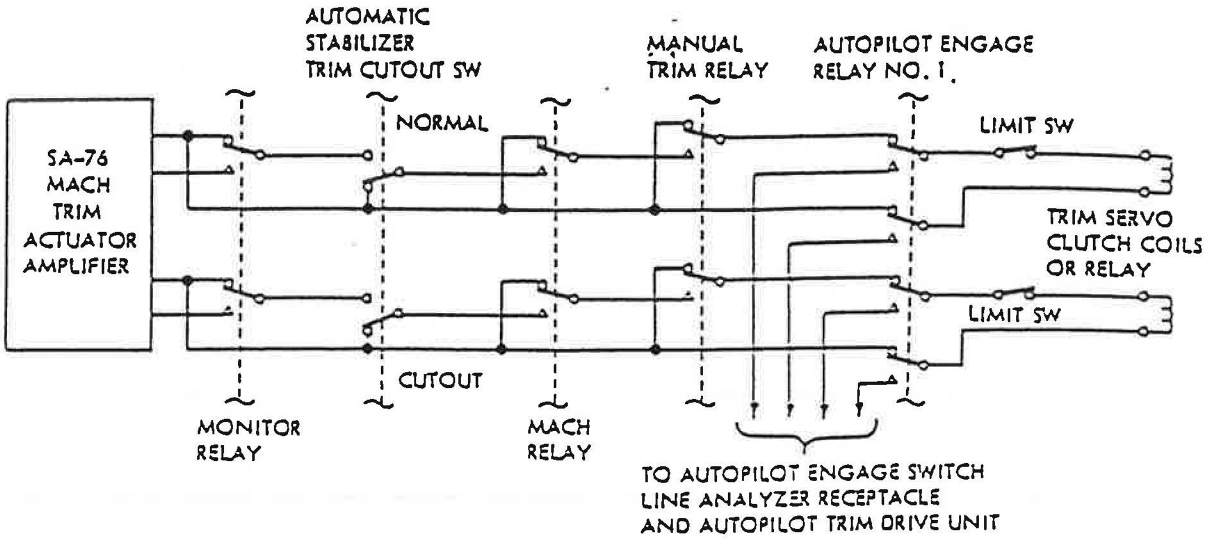
(b) 30 volts phase C, and 15 volts phase A plus 180 degrees, are supplied to energize the electro-mechanical synchronizer card motor.

(c) 19.4 volts phase A and 10.3 volts phase C are combined to supply 26-volts at a phase angle lagging phase A by 20 degrees. This lagging voltage is used to excite the electro-mechanical synchronizer signal synchro and stabilizer trim position transmitter.

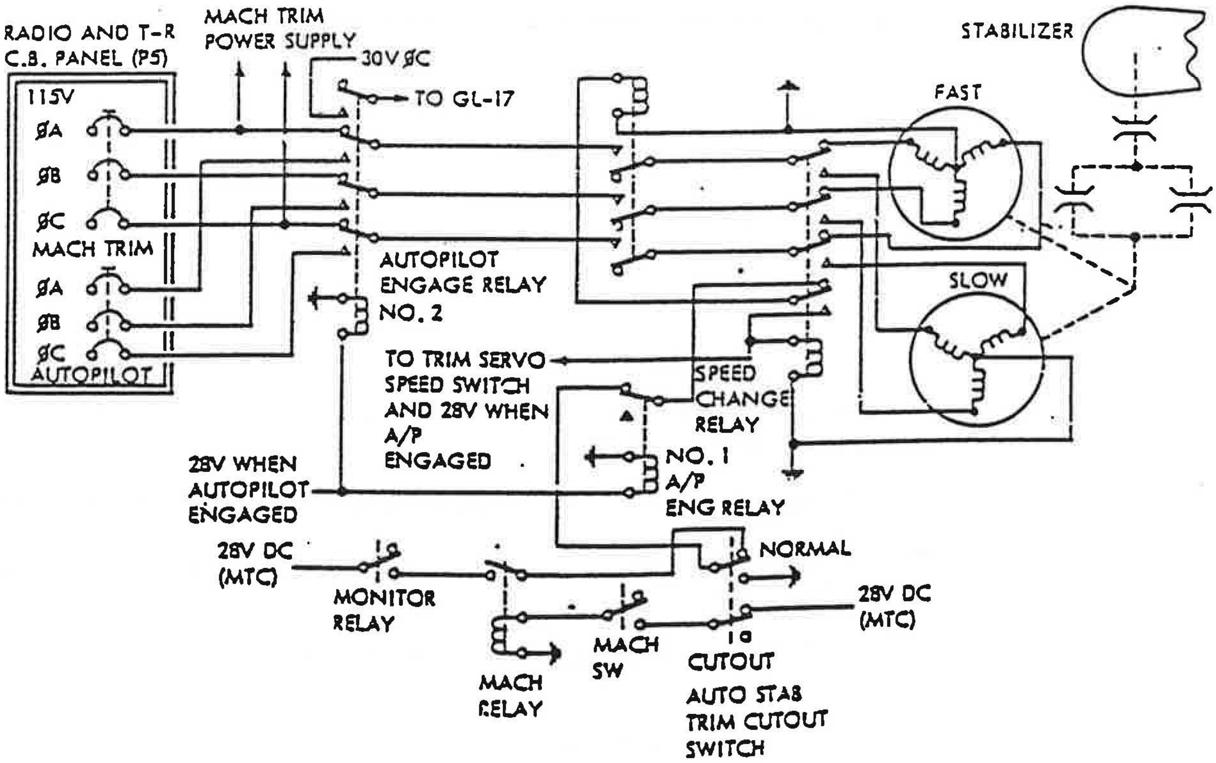
(d) 28-volt d-c power is supplied to all relay control circuits.

(e) 28-volt filtered d-c power is supplied to the amplifier and monitor cards.

(f) 0-volt phase A is supplied as a reference.



Trim Servo Clutch Control Circuit
Figure 2



Trim Servo Power Control Circuit
Figure 3

(2) The mach trim coupler power supply is energized whenever the flight instrument bus No. 2 in the radio and T-R circuit breaker panel (P5) is energized.

C. Autopilot Engage Relays

(1) Two interlock relays prevent simultaneous operation of the Mach trim system and autopilot system. These relays are autopilot engage relay No. 1 and autopilot engage relay No. 2. Both relays are energized when the autopilot is engaged. (See figure 3.)

(2) When autopilot engage relay No. 1 is energized control of the trim servo clutches and power relay is transferred from the mach trim actuator amplifier to the autopilot actuator amplifier.

(3) When autopilot engage relay No. 2 Is energized operating power for the stabilizer trim servo motor is transferred from the mach trim power source to the autopilot power source, and power is supplied to the electro-mechanical synchronizer.

D. Manual Trim Relay (K2 is in MACH TRIM COUPLER)

(1) The manual trim relay may be energized from the autopilot power junction box or the mach trim coupler power supply when the main stabilizer trim relay is not energized. (See Chapter 27.) Diodes prevent current flow between the power supply sources. (See figure 4.) The relay is energized through a thermal delaying circuit which delays the relay closing by 0.75 second.

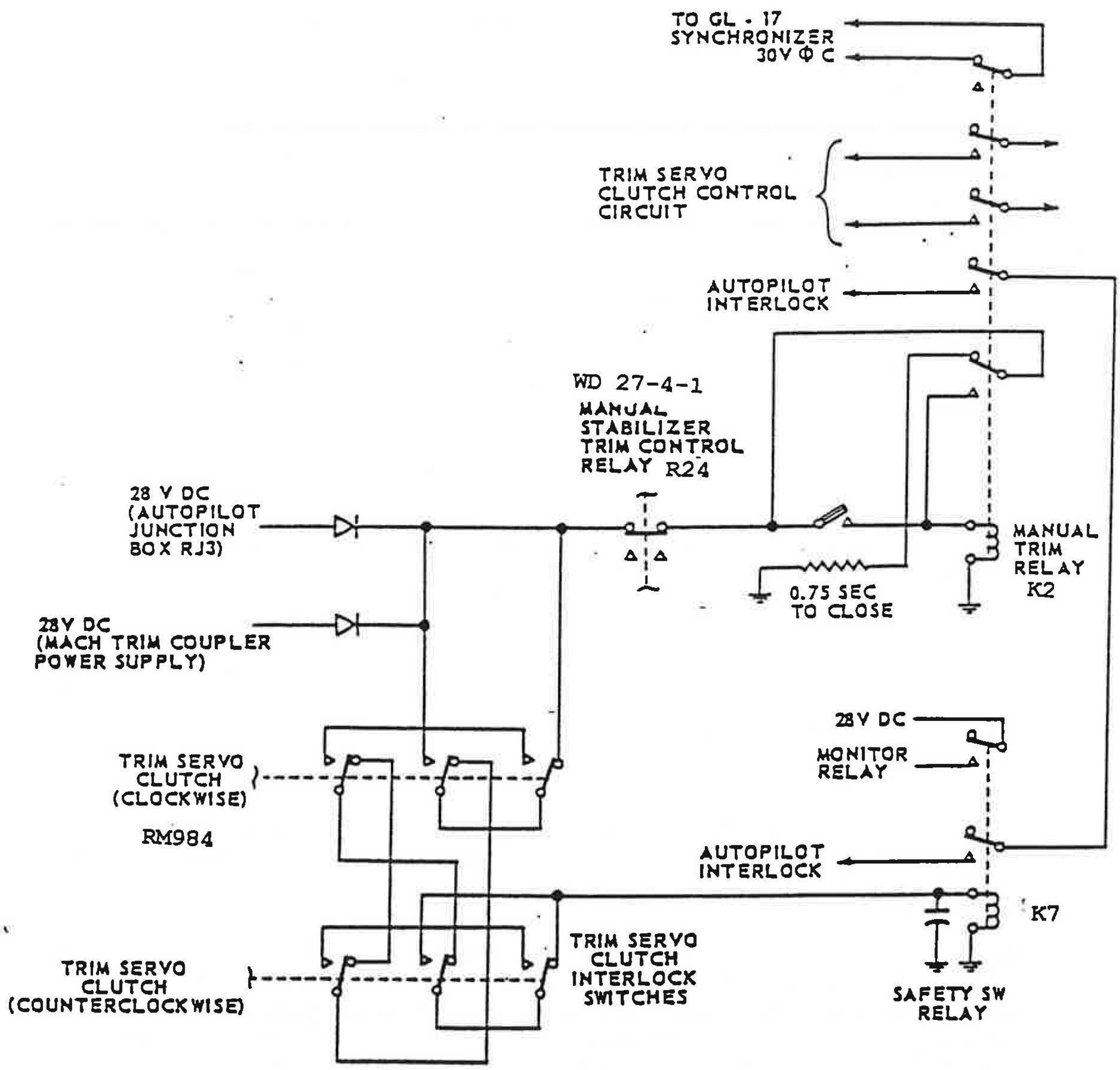
(2) When the manual trim relay is de-energized (main stabilizer trim system is electrically operated) the autopilot interlock circuit is opened, the control circuit to the trim servo clutches are opened and power is supplied to the synchronizer motor.

(3) When the manual trim relay is energized, a closed path in provided for the autopilot Interlock circuit and for signals to the trim servo coil clutches.

E. Safety Switch Relay(K7 is in MACH TRIM COUPLER)

(1) The safety switch relay coil is connected in series with interlock switches in the trim servo so that the relay will be de-energized if both trim servo, clutches should be engaged at once. (See figure 4.)

(2) When the safety witch relay is energized, two sets of contacts are operated. One set of contacts is connected in series with contacts of the manual trim relay in the autopilot interlock circuit. The second set of contacts is connected in series with other relay contacts to complete a circuit to the monitor relay.



For more details, see WD 22-1-0 SCH 7

Manual Trim and Safety Switch Relay Circuits
Figure 4

F. Monitor Relay

(1) Automatic protection against malfunction is furnished by the monitor amplifier which controls the monitor relay. Contacts on the monitor relay control output of the auto-trim actuator amplifier and control application of power to the trim servo motor power relay and clutches. An additional set of contacts is used in an interlock for the monitor relay coil so that the relay cannot be energized after it has been tripped unless the automatic stabilizer trim switch is turned to "CUTOUT" and then, to "MORMAL." (See figure 5.)

(2) When the automatic stabilizer trim cutout witch lever is in the "NORMAL" position and power is applied to the mach trim coupler, the relay will be pulled in by the initial pulse of current throughthe capacitors, and, if the safety switch relay is energized and themonitor amplifier senses proper circuit operation, the monitor relay will stay energized by the power through the closed contact in the monitor amplifier. When the automatic stabilizer trim cutout switch lever is in "CUTOUT" position the monitor relay will stay energized in all conditions.

G. Mach Relay

(1) When the automatic stabilizer trim cutout witch lever is in the "NORMAL" position the mach relay will be energized if the mach switch is closed. (See figure 6.)

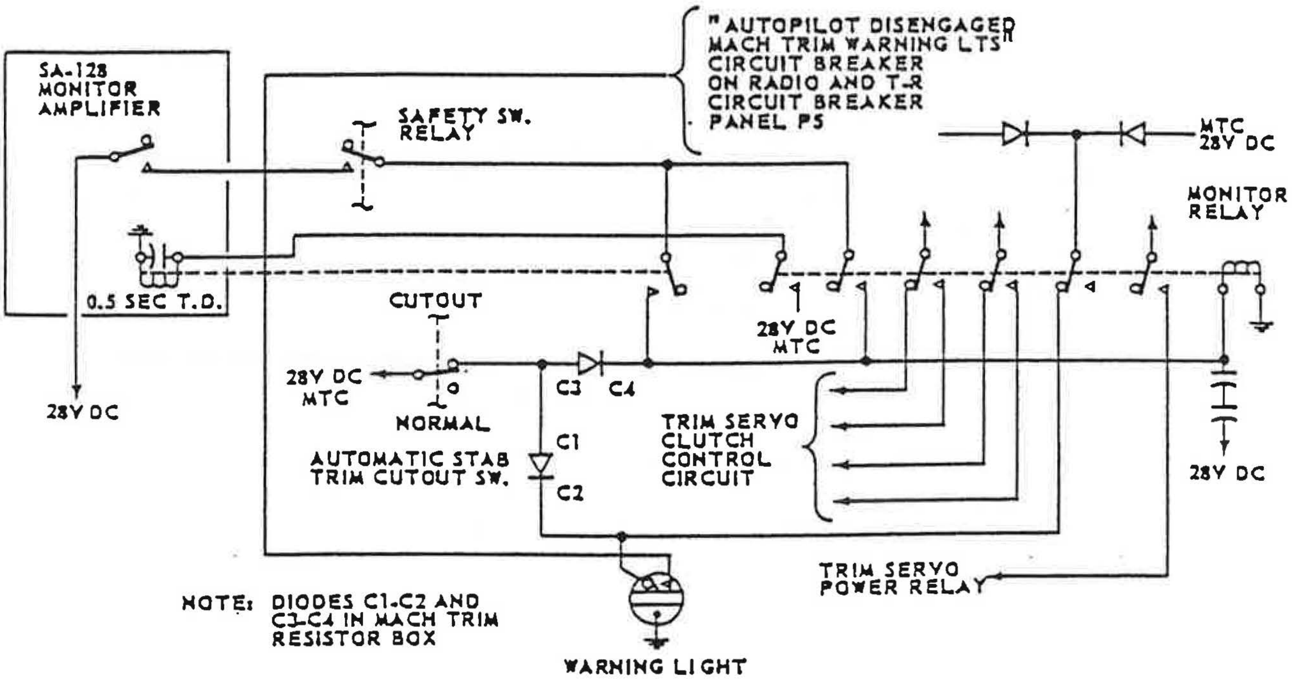
(2) When the mach relay is energized, three sets of contacts complete the control circuit between the auto-trim actuator amplifier and trim servo clutches, and the circuit to the trim servo power relay. When the mach relay is not energized, these circuits are opened.

H. Preamplifier

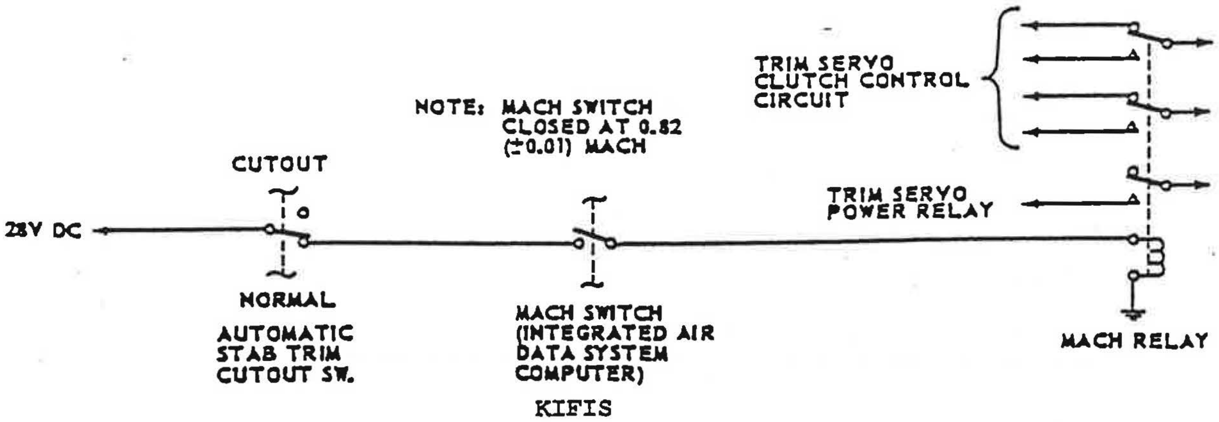
(1) Mach signals from the mach potentiometer in the integrated air data instrument system, synchronizer signals from a synchronizing autosyn and position signals from the stabilizer position transmitter are combined and fed to a preamplifier. If the combined inputs from the three signal sources are not balanced, an error signal will exist. This error signal is amplified and sent to a control amplifier. (See figure 7.)

I. Control Amplifier

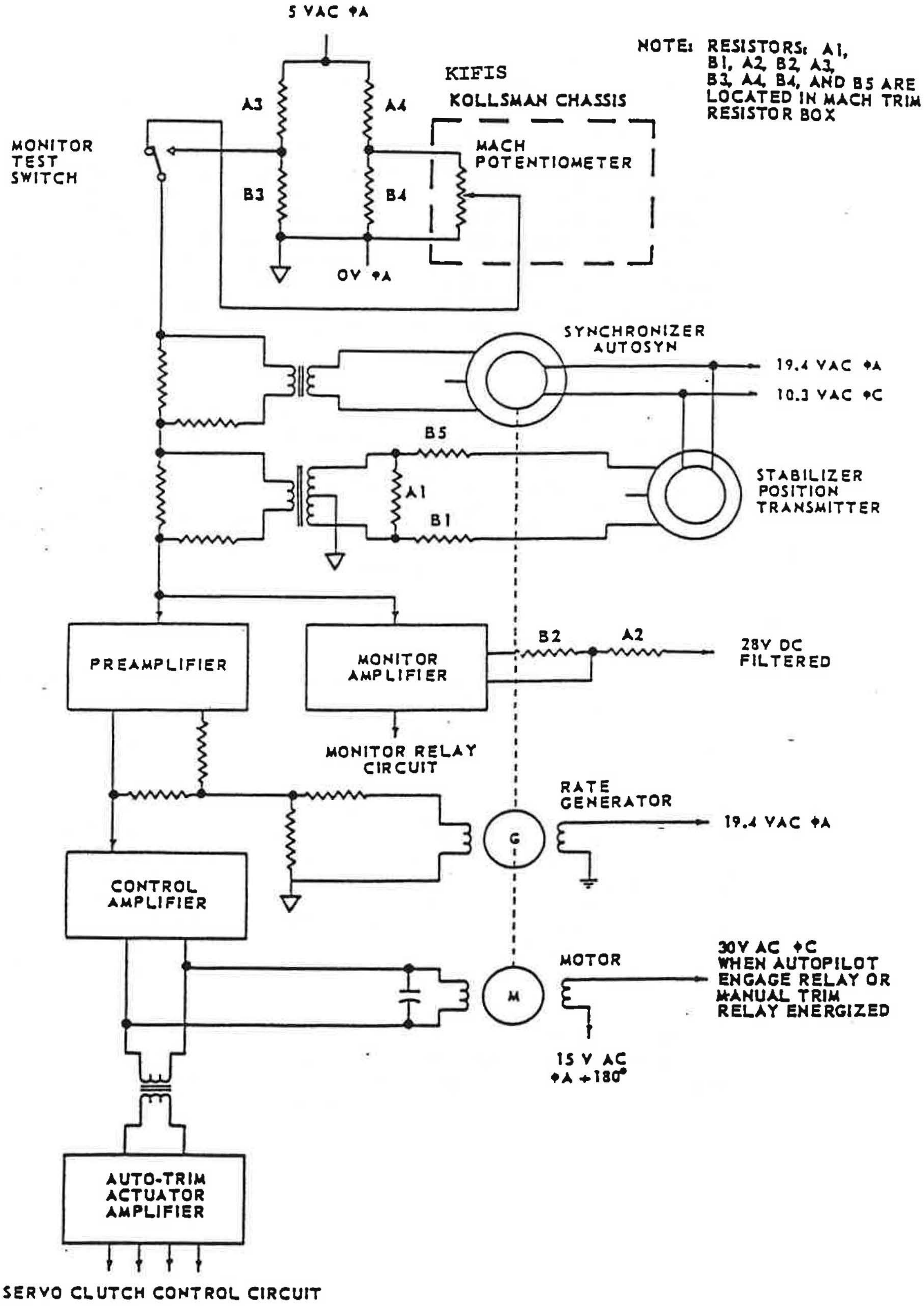
(1) The control amplifier amplifies the a-c error signal, and converts the amplified signal into a d-c signal in a Ramey toroid phase discriminator. The d-c signal is fed to a magnetic amplifier to develop controlled a-c power for operation of the electro-mechanical synchronizer. The control amplifier also provides an a-c signal to an auto-trim actuator amplifier. (See figure 7.)



Monitor Relay and Warning Light Circuit
Figure 5



Mach Relay Circuit
Figure 6



Mach Trim Signal Circuit
Figure 7

(2) Whenever the autopilot is engaged, or whenever the main electrical stabilizer trim system is operated, an electro-mechanical synchronizer motor receives excitation for the reference phase. Error signals developed at the of the preamplifier are amplified and used by the variable phase of the synchronizer motor. The motor drives a rate generator and autosyn transmitter. Output of the autosyn is used to cancel any difference between the mach input signal and stabilizer position signal, thus keeping the mach trim system synchronized with manual flight control or autopilot. Output of the synchronizer rate generator is fed through the control amplifier and used to damp out any tendency of the synchronizer motor to overshoot when the error signal is cancelled.

J. Auto-Trim Actuator Amplifier

(1) Output of the control amplifier is fed to an auto-trim actuator amplifier. This amplifier limits the signal by diode limiters and amplifies it in a step-up transformer to increase sensitivity and reduce power loss. The a-c signal is then changed to dc in magnetic amplifler demodulators and used to operate magnetic amplifier switches. The magnetic amplifier switches control the clutches in the trim servo which actuates the stabilizer trim mechanism. (See figures 2 and 7.)

K. Synchronizer

(1) In order to prevent rapid changes in attitude when the mach trim system is initially engaged, balancing or synchronizing signals must be supplied to make up the difference between signals from the mach potentiometer and stabilizer position transmitter. The chronizing signal is supplied by a motor driven autosyn. (See figure 7.)

(2) The two-phase synchronizer motor receives excitation on the reference phase only when the autopilot is engaged or when the main electrical stabilizer trim system is operated. Any error signal appearing at the input of the control amplifier is amplified and fed to the variable phase of the motor, which then operates and turns the synchronizer autosyn to a new position. When the autosyn reaches a position at which its output cancels the error signal, the system is synchronized and the motor stops.

(3) A rate generator is used to prevent synchronizer overshoot. Output of the rate generator is fed through the control amplifier to the variable phase of the synchronizer motor. When the error signal is balanced, rate generator output acts in opposition to motor operation, and rapidly stops the motor to prevent overshoot.

L. Monitor Amplifier

(1) Protection against violent maneuvers or extreme attitudes caused by malfunction of the mach trim system is provided by a monitor amplifier. This amplifier senses the error signal and compares it to a reference signal. If the error signal reaches a predetermined magnitude the monitor amplifier causes the monitor relay to become de-energized and disconnects the mach trim system from the trim servo. A warning light on the engine instrument panel is then energized to inform the pilot that the system is disconnected. The monitor trips if signal voltage exceeds that equivalent to one degree of stabilizer movement. The warning light will also illuminate if the automatic stabilizer trim cutout switch lever is in the "CUTOUT" position. (See figures 5 and 7.)

(2) A one-half second delayed opening relay is in the monitor amplifier to eliminate nuisance disconnect of the system when transient signal difference is presented. The warning light will blink momentarily, but the system is not disconnected. (See figure 5).

4. Monitor Test Switch

A. The monitor amplifier can be checked during ground operation by a monitor test switch on the engine instrument panel. When the switch is actuated, a fixed error signal is substituted for the mach signal and applied to the mach trim coupler input. The error signal should cause the monitor amplifier to disconnect the mach trim system. The monitor circuit cannot be armed again until the automatic stabilizer trim cutout switch lever is actuated to "CUTOUT." and back to "NORMAL", and the fixed error signal is removed from the amplifier input. (See figure 5 and 7.)

WARENING: DO NOT ACTUATE THE MONITOR TEST SWITCH DURING FLIGHT.

5. Mach Potentiometer and Mach Switch

A. Electrical signals corresponding to mach number are taken from a potentiometer in the KIFIS system. (See figures 6 and 7.) The potentiometer is driven by a servo system controlled by output of a mach transmitter in the machmeter. Refer to Chapter 34, "Integrated Air Data Instrument System."

B. The KIFIS also operates the mach switch. The switch is closed am approximately mach 0.83 and is open at speeds less than Mach 0.83. When the switch is closed, the mach relay in the mach trim coupler is energized.

6. Stabilizer Position Transmitter

A. The stabilizer position transmitter is an autosyn which is connected to the stabilizer through a mechanical linkage. The autosyn motor is excited by 26-volt a-c, and induces a voltage in the stator windings. The phase and magnitude of the induced voltage is compared to reference signals to determine direction and magnitude of trim error.

7. Resistor Box

A. The resistor box contains two diodes used In the warning light circuit and fixed resistors used an voltage dividers and resistance bridges to modify error and reference voltages for use by the mach trim system. The resistor box in the lower nose compartment.(See figure 1.)

8. Mach Trim Warning Light

A. A warning light is provided to indicate that the mach trim system is disengaged. The system is disengaged when the automatic stabilizer trim cutout switch lever is in the "CUTOUT" position, or when the monitor relay is de-energized. The warning light is located on the pilot's engine instrument panel. (See figure 1.)

9. Operation

A. During normal flight operation, the mach trim system is on stand-by when the mach trim circuit breaker and the autopilot and mach trim disengage light circuit breaker are closed on radio and T-R circuit breaker panel (P5), and the automatic stabilizer trim cutout switch lever is in "NORMAL" position.

B. The system will start operation when the main electrical stabilizer trim system is not used, the autopilot disengaged, and the mach number is 0.82 or above. When these conditions exist, any change in mach number will cause the horizontal stabilizer to be positioned to correct for the change. For example, it the airplane increases speed because of a nose dawn attitude the mach trim system will cause the stabilizer to move to a nose up tria.position. Conversely, if the mach number decreases the mach, trim system will cause the stabilizer to move to a nose down trim position. The system will therefore, automatically adjust the horizontal stabilizer position in accordance with the proper function of mach number such that positive static stability is obtained throughout the airplane speed range. Positive static stability is defined as a requirement for an increasing positive stick force for trim with increasing speed.

C. When the main electrical stabilizer trim system or autopilot is used, the mach trim system is inoperative and the synchronizer is energized to wash cut any signal fed back from, the stabilizer position transmitter. However, operation of the emergency manual stabilizer trim system will not override the mach trim system and the stabilizer position transmitter feed back signal will not be washed out.

MACH TRIM SYSTEM - TROUBLE SHOOTING

1. Trouble Shooting Mach Trim System

A. General

(1) The mach trim system analyzer is designed to locate malfunctions in the mach trim system which occur outside the mach trim coupler. The mach trim coupler should be removed, the tester installed in its place and the test procedure outlined in Table I should be followed.

(2) If the outlined procedure does not locate a malfunction in the system, the faulty component may safely be assumed to be the mach trim coupler. The mach trim coupler should be replaced with a coupler known to be in good working condition.

(3) If the test procedure does locate a fault, the faulty component should be replaced or wiring repaired as necessary. The mach trim coupler that was removed prior to the test may be installed if a single fault is assumed.

NOTE: The mach trim coupler has not been checked. To make sure that the complete system is operating properly, a mach trim coupler known to be in good working condition should be installed. This last step is necessary only when the system has a double fault.

B. Equipment and Materials

(1) Mach Trim System Analyzer, Boeing F-71375 or equivalent

(2) Pitot-Static System Tester, Boeing EDS/PS-200 or equivalent

(3) Two sets of interphones

C. Prepare for Trouble Shooting of Mach Trim System

(1) Close ESSENTIAL RADIO BUS circuit breakers (3) and RADIO BUS NO. 2 circuit breakers (3) on A-C BUS NO. 2 circuit breaker panel (P2).

(2) Close ESSENTIAL RADIO BUS and RADIO BUS NO. 2 toggle switches on radio and T-R circuit breaker panel (P5).

(3) Close applicable vertical gyro circuit breaker or insert fuse on radio and T-R circuit breaker panel (P5) as applicable.

(4) Oven all autopilot and mach trim a-c and d-c circuit breakers or remove fuses as applicable on radio and T-R circuit breaker panel (P5).

(5) Open KIFIS breaker or remove fuse as applicable on radio and T-R circuit breaker panel (P5).

(6) Remove mach trim coupler from electronic equipment rack.

(7) Install mach trim system analyzer in place of mach trim coupler.

(8) Install an interphone communication system between the pilot or copilot station and the electronic equipment rack station. This can be accomplished by using existing airplane wiring and interphone-jack facilities. See Chapter 23.

NOTE: Two operators are needed to perform the test and it is essential for the operators to be able to communicate with each other.

(9) Position one operator at the pilot's or copilot's station.

(10) Position one operator at the electronic equipment rack with the mach trim system analyzer.

NOTE: Components called out in the following testing procedure are located on the mach trim system analyzer if not otherwise specified. Terminal numbers are located in RJ3 for N20199, RJ23 for N19997 & N20000 if not otherwise specified.

(11) Follow the testing procedure outlined in Table I.

Step	Indication OK	Malfunction-Replace Component or Repair Wiring as Necessary
1. Close A/P & MACH TRIM DISENG LT D-C circuit breaker on circuit breaker panel	"MACEH TRIM DISENGAGED" light (on engine instrument panel (P9))	NO light, check for 28V DC at * check warning light and warning light on circuitry. * D30 for N20199. * AD13 for N19997, N20000.
2. Close AUTOPILOT A-C circuit breaker (on radio and T-R circuit breaker panel).	AUTOPILOT ON" light on. (If POWER switch is in AUTOPILOT position, lights will also come on as in step 3.)	No light, see trouble shooting Chart "A."
3. Close MACH TRIM CONTROL A-C circuit breaker (on radio and T-R circuit breaker panel (P5).) POWER switch must be MACH TRIM position.	"TRIM RELAY" light and "SAFETY SW TEST" light on.	a. If TRIM RELAY light falls to come on, see trouble shooting Chart "B." b. If SAFETY SW TEST light fails to come on, see trouble shooting Chart "C." c. Note step 35.
4. On airplanes having mach potentiometer in co pilot,s computer unit, (KIFIS control unit), close copilot's KIFIS circuit breaker or insert fuse (on P5) as applicable.		
5. Set POWER switch to AUTOPILOT Position.	Ø A, Ø B and Ø C A-C POWER lights on. PEASE ROTATION RIGHT light on; PHASE ROTATION WRONG light out.	If any or all lights are out, check autopilot circuit breaker (on P5).
6. Set POWER switch to MACH TRIM position.	ØA, Ø B and Ø C A-C POWER lights on. PHASE ROTATION RIGHT light on; PHASE ROTATION Wrong light out.	It any or all ø lights are out, cheek mach trim circuit breaker (on P5). If all 0 lights are on, but PHASE ROTATION WRONG light is on and PHASE ROTATION RIGHT light is out; cheek phase rotation.

Table 1 (Sheet 1 of 9)

Step	Indication OK	Malfunction - Replace Component or Repair Wiring as Necessary
7. Hold STAB TRIM MOTOR NOSE UP - NOSE DN switch in NOSE UP position. For single speed trim servo hold for 6 seconds. For two speed trim servo hold for 8 seconds.	Stabilizer position indicator (on the control stand) shall move up 1 (± 1/4) unit.	If stabilizer position indicator does not move up as required, check trim servo (on lower right side of the actuator assembly) and limit switches (on airplane structure adjacent to the jackscrew). "SAFETY SW LIGHT" may blink but should not be off longer than it takes the operator to say ONE quickly.
8. Hold "STAB TRIM MOTOR SHORTS" switch in "SHORTS" position while holding "STAB TRIM MOTOR NOSE UP - NOSE DN" switch in "NOSE UP" position.	Stabilizer trim shall stop.	If stabilizer trim does not stop, assure that wire to pin 17 of plug J1 (RD1317) is not grounded.
9. Release "STAB TRIM MOTOR SHORTS" switch; release "STAB TRIM MOTOR NOSE UP - NOSE DN" switch.		
10. Hold "STAB TRIM MOTOR NOSE UP - NOSE DN" switch in "NOSE DN" position. For single speed trim servo hold for 6 seconds. For two speed trim servo hold for 8 seconds.	Stabilizer position indicator shall move down 1 (± 1/4) unit.	If stabilizer position indicator does not move down as required, check trim servo (on lower right side of the actuator assembly) and limit switches (on airplane structure adjacent to the jackscrew). "SAFETY SW LIGHT" may blink, but should not be off longer than it takes the operator to say ONE quickly.

Table I (Sheet 2)

Step	Indication OK	Malfunction - Replace Component or Repair Wiring as Necessary
11. Hold "STAB TRIM MOTOR SHORTS" switch in "SHORTS" position while holding "STAB TRIM MOTOR NOSE UP - NOSE DN" switch in "NOSE DN" position.	Stabilizer trim shall stop.	If stabilizer trim does not stop, assure that wire to pin 17 (of plug J1 RD1317) is not grounded.
12. Release "STAB TRIM MOTOR SHORTS" switch; release "STAB TRIM MOTOR NOSE UP - NOSE DN" switch.		
13. Trim stabilizer (manually or with main electric stabilizer trim motor) so that trim indicator reads 0 units. NOTE: To employ main electric trim motor, observe step 24. Set "STAB TRIM" cutout switch in the on position, or the "STAB TRIM MAIN ELECTRIC" cutout switch in "NORMAL" (on right hand side of control stand) as applicable.		
14. Set voltmeter selector switch to POS XMTR.	Voltmeter indication should be as shown in Table II.	a. If no, or incorrect voltage reading, see trouble shooting chart "D." b. Note step 18.
15. Hold POS XMTR SHORTS switch in upward position.	No significant change in voltage noted previously.	If a change in voltage is noted, assure that wires to pins 24 and 32 of plug J2 (M/W RD1318) are not grounded.

Table I (Sheet 3)

Step	Indication OK	Malfunction - Replace Component or Repair Wiring as Necessary
16. Hold POS XMTR SHORTS switch in downward position.	No significant change in voltage noted previously.	If a change in voltage is noted, assure that wires to pins 24 and 32 of J2 (M/W RD1318) are not grounded.
17. Release POS XMTR SHORTS switch.		
18. Hold POS XMTR POLARITY switch in downward position.	Voltage shall decrease 0.60 (± 0.25) volts from reading obtained in step 14.	If voltage does not decrease as required, assure that wiring is not reversed in airplane or resistor box. Assure that stabilizer position transmitter linkage is not reversed. NOTE: Looking at stabilizer position transmitter and stabilizer from left side of airplane when stabilizer leading edge moves down, position transmitter shaft must rotate clockwise.
19. Release POS XMTR POLARITY switch.		
20. Trim stabilizer 1 unit nose down manually or with main electric trim motor. (See NOTE in step 13.)	Voltmeter reading shall decrease from reading attained in step 14. See Table II.	If voltmeter indication is incorrect, check resistors in resistor box. (See wiring diagram 22-1-6.) Assure that position transmitter linkage is not reversed.
21. Hold STAB TRIM MOTOR SHORTS switch in SHORTS position.	Voltmeter reading shall go to zero volts.	If voltmeter reading does not go to zero volts, assure that wire to pin 25 of plug J2 (M/W RD1318) is not grounded.

Table I (Sheet 4)

Step	Indication OK	Malfunction-Replace Component or Repair Wiring as Necessary
22. Release STAB TRIM MOTOR SHORTS switch 23. Set "MACE TRIM CUTOUT' switch to on or "AUTO-MATIC STABILIZER trim cutout switch to "NORMAL" (on control stand) as applicable 24. Close "STAB TRIM CONT DC circuit breaker (on P5) and "STAB TIM A-C circuit breaker (on P2 breaker panel). NOTE: This step may have been done with step 13. 25. Hold pilot trim switch (on pilot's wheel) in "NOSE UP" position momentarily.	TRIM RELAY light out; STAB TRIM ON warning light (on control stand) on.	a. If TRIM RELAY light fails to come out, check "STAB TRIM cut out switch or "STAB TRIM MAIN ELECTRIC" cutout switch (on con-trol stand) whichever is used and manual STAB TRIM CONT relay (on J18 relay shield) b. If STAB, TRIM ON warning light fails to come on, cheek warning light and warning light wiring.
26. Hold pilot trim switch in downward position momentarily.	TRIM RELAY light out; STAB TRIM ON warning light on.	If TRIM RELAY light fails to come on, check "STAB TRIM" cutout switch or "STAB TRIM MAIN ELECTRIC cutout switch (on control stand) whichever is used.

Table 1 (Sheet 5 of 9)

Step	Indication OK	Malfunction - Replace Component or Repair Wiring as Necessary
27. Repeat step 25 and step 26 using copilot's trim switch (on copilot's wheel).		
28. Set autopilot engage switch (on autopilot control panel to the "AUTOPILOT" position). Airplanes with AUTOMATIC STAB TRIM cutout switch require switch in NORMAL position before MACH SW and AUTOPILOT ENGAGE lights will go on.	"AUTOPILOT ENGAGED" light on; "AUTOPILOT DISENGAGED" light (on P9 engine instrument panel) off.	If lights fail to come on as specified, check autopilot, and wires to the following: pin 30, J1 (M/W-RD1317) pin 31, J1 (M/W-RD1317) pin 23, J1 (M/W-RD1317) pin 24, J1 (M/W-RD1317)
29. Set POWER switch to AUTOPILOT position.	NOTE: COMPARISON UNIT INTERLOCK SWITCH must be in BYPASS if comparison unit is part of system.	
30. Hold STAB TRIM MOTOR NOSE UP --NOSE DN switch in NOSE UP position. Push on control column. For single speed trim servo push for 6 seconds. For two speed trim servo push for 24 seconds.	Stabilizer position indicator (on control stand) shall move up 1 (± 1/4) unit.	If stabilizer position indicator does not move up as required, check autopilot.
31. Hold STAB TRIM MOTOR NOSE UP - NOSE DN switch in NOSE DN position. Pull on control column. For single speed trim servo pull for 6 seconds. For two speed trim servo pull for 24 seconds.	Stabilizer position indicator shall move down 1 (± 1/4) unit.	If stabilizer position indicator does not move down as required, check autopilot.
NOTE: Operation of STAB TRIM MOTOR switch is unnecessary in steps 30 and 31 for airplanes incorporating Bendix E/P two speed trim servo.		

Table I (Sheet 6)

Step	Indication OK	Malfunction - Replace Component or Repair Wiring as Necessary
32. Release STAB TRIM MOTOR NOSE UP - NOSE DN switch. Push on control column.	Stabilizer position indicator shall not move.	If stabilizer position indicator moves, check trim servo. Check for trim servo wiring shorting to plug J1 (M/W RD1317).
NOTE: Delete step 32 for airplane with two speed trim servo installed.		
33. WARNING: WAIT 30 SECONDS (FOR TRIM SERVO MOTOR ARMATURE TO STOP ROTATING) BEFORE PROCEEDING.		
34. Set POWER switch to MACH TRIM position.		
35. Hold SAFETY SW TEST switch in downward position.	SAFETY SW TEST light out.	If light remains on, check limit switches and trim servo and repair or replace as necessary.
36. Pull AUTOPILOT A-C circuit breaker (P5).		
37. Pull A/P & MACH TRIM DISENG LT CB (P5).		
38a. Set MACH TRIM cutout switch to on or AUTOMATIC STAB TRIM cutout switch to NORMAL as applicable.	MACH TRIM CUTOUT SWITCH ON light on; MACH TRIM DISENGAGED light (on P9 engine instrument panel off).	Perform step 38b.
38b. Set MACH TRIM cutout switch or AUTOMATIC STAB TRIM cutout switch as applicable to CUTOUT position (control stand).	MACH TRIM CUTOUT SWITCH OFF lights on; MACH TRIM CUTOUT SWITCH ON light off, and MACH TRIM DISENGAGED light (P9) on. NOTE: In this step, one of the OFF lights will be dimmer than the other; also MACH TRIM DISENGAGED light will be dimmer than usual.	If lights do not perform as required, check MACH TRIM cutout switch or AUTOMATIC STAB TRIM cutout switch (on control stand) as applicable. Check diodes in resistor box. (See wiring diagram 22-1-6.)

Table I (Sheet 7)

Step	Indication OK	Malfunction-Replace Component or Repair Wiring as Necessary
39. Position voltmeter selector switch to BIAS position.	Voltmeter indication should be as shown in Table II.	If voltmeter reading is not as specified, check resistors in resistor box. (See wiring diagram 22-1-6.)
40. Position voltmeter selector switch to MACH POT Position.	Voltmeter shall indicate 0 volts. mach synchrotel trans	In the event of an erroneous indication, check mitter as applicable; check KIFIS control unit or servo torque unit as applicable.
41. Hold MACH TRIM TEST switch (on P9 engine instrument panel) in TEST position.	Voltmeter reading should be as shown in Table II.	In the event of erroneous indications: a. Check resistors in resistor box or mach trim component box. (See wiring diagram 22-1-6.) b. Note step 44.

Table I (Sheet 8)

Step	Indication OK	Malfunction-Replace Component or Repair Wiring as Necessary
42. Release MACH TRIM TEST switch		
43. On airplanes with mach potentiometer in KIFIS control unit only: Position AUTO STAB TRIM cutout switch to NORMAL (on control stand) as applicable and hold co-pilot's KIFIS switch (copilot's instrument panel) in TEST position.	MACE SW CLOSED light on. Light should be steady and of normal brilliance. Voltmeter readings should be as shown in Table II.	a. If MACH SW CLOSED light shows indications different from those specified, or if voltmeter reading are erroneous,, check resistors in resistor box (aft of left electronic equipment rack) b. Note step 44.
44. NOTE: This step is not necessary if steps 41 through 43 are satisfactory and if malfunction of synchrotel transmitter is not suspected. With a pitot-static system tester, supply the airplane pitot-static system with pitot and static pressures sufficient to simulate mach numbers between 0.75 and 0.94.	Voltmeter readings should be as shown in Table II.	See steps 41 through 43

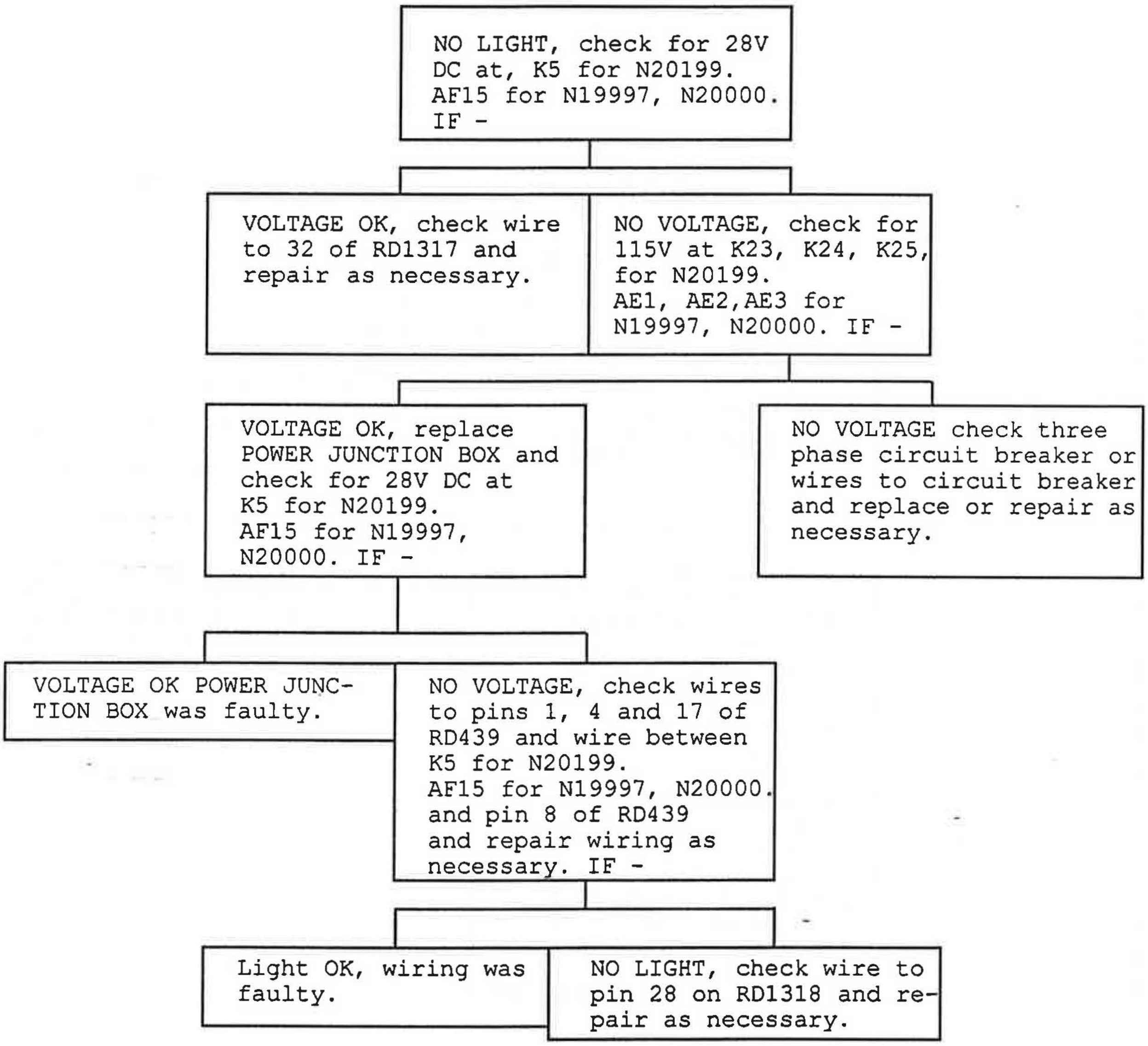
Table 1 (Sheet 9)

CONDITION	707-300C
POSITION TRANSMITTER - 0 Units of Trim on Trim Indicator	0.50 to 1.5 V
Trim Stabillzer Airplane Nose Down to	Voltage decreases 0.25 to 0.45 V/unit
BIAS	1.1 to 1.4 V
MACH TRIM TEST If From Resistor Box	
MACH TRIM TEST or F/O KIFIS TEST If From Mach Pot (.917 M)	0.70 to 1.10 V
OUTPUT OF MACH POTENTIOMETER ENGINS [1]	0.81 M
M = 0.87	
M = 0.90	
M = 0.86	0.10 to 0.20 V
M = 0.88 [2]	0.20 to 0.40 V [2]
M = 0.92	0.80 to 1.10 V

[1] When mach number is increasing, MACH SWITCH CLOSED light comes on 0.02 to 0.00 Mach below this point. Output of MACH POT begins 0.00 to 0.02 mach above this point and increases smoothly if mach number is increased slowly, then decreases smoothly if mach number is decreased slowly to mach number where output began. MACH SWITCH CLOSED light shall go off 0.00 to 0.02 mach below the point at which it cam on with increasing mach.

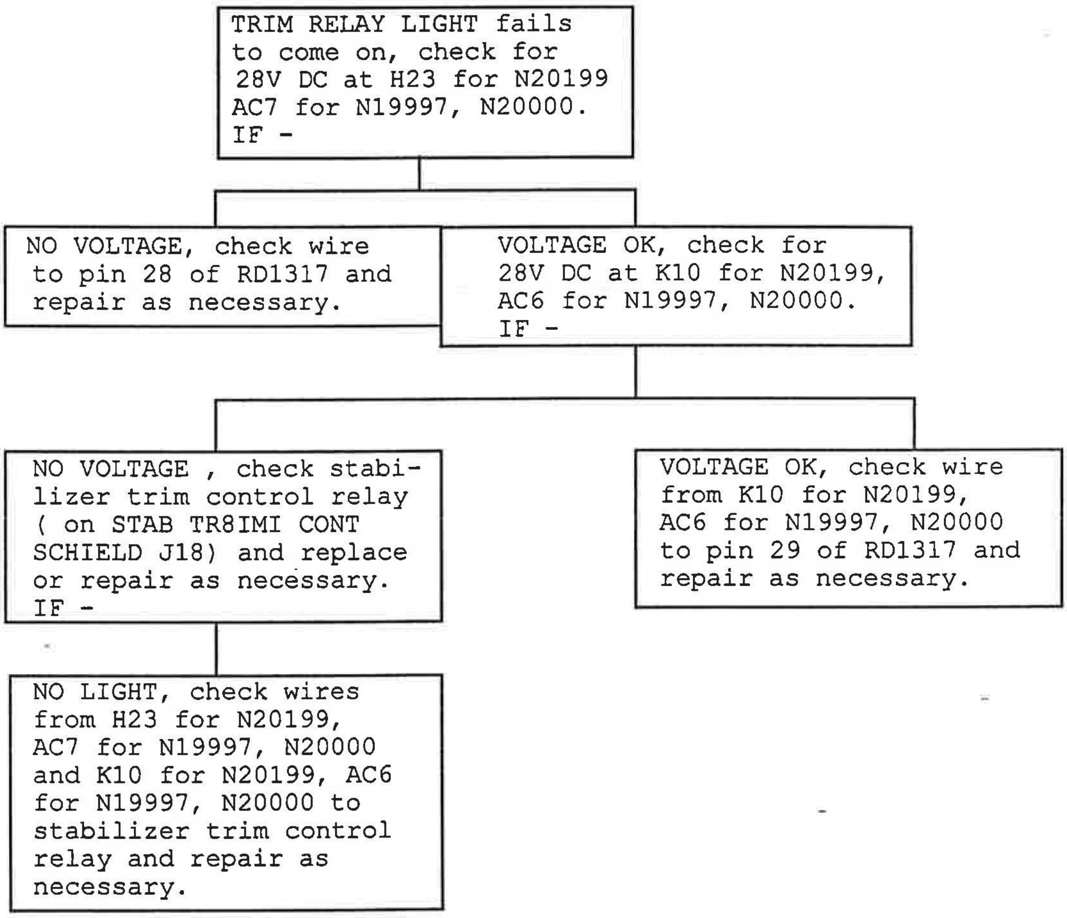
[2] Voltage at 0.88 M shall be 2.0 (± 0.5) times the voltage at 0.86 M but do not exceed table value.

Table II



NOTE : RD439 is the PWR J-BOX 22-1-3

Trouble Shooting Chart A



Trouble Shooting Chart B

SAFETY SW TETST light fails to comeon, unplug RD4611 from trim srvo and check for 28V DC at terminal N of RD4611. IF -

NO VOILTAGE , repair wir-ing between terminal H23 for N20199, AC7 for N19997, N20000 and terminal N of RD4611 and proceed with test.

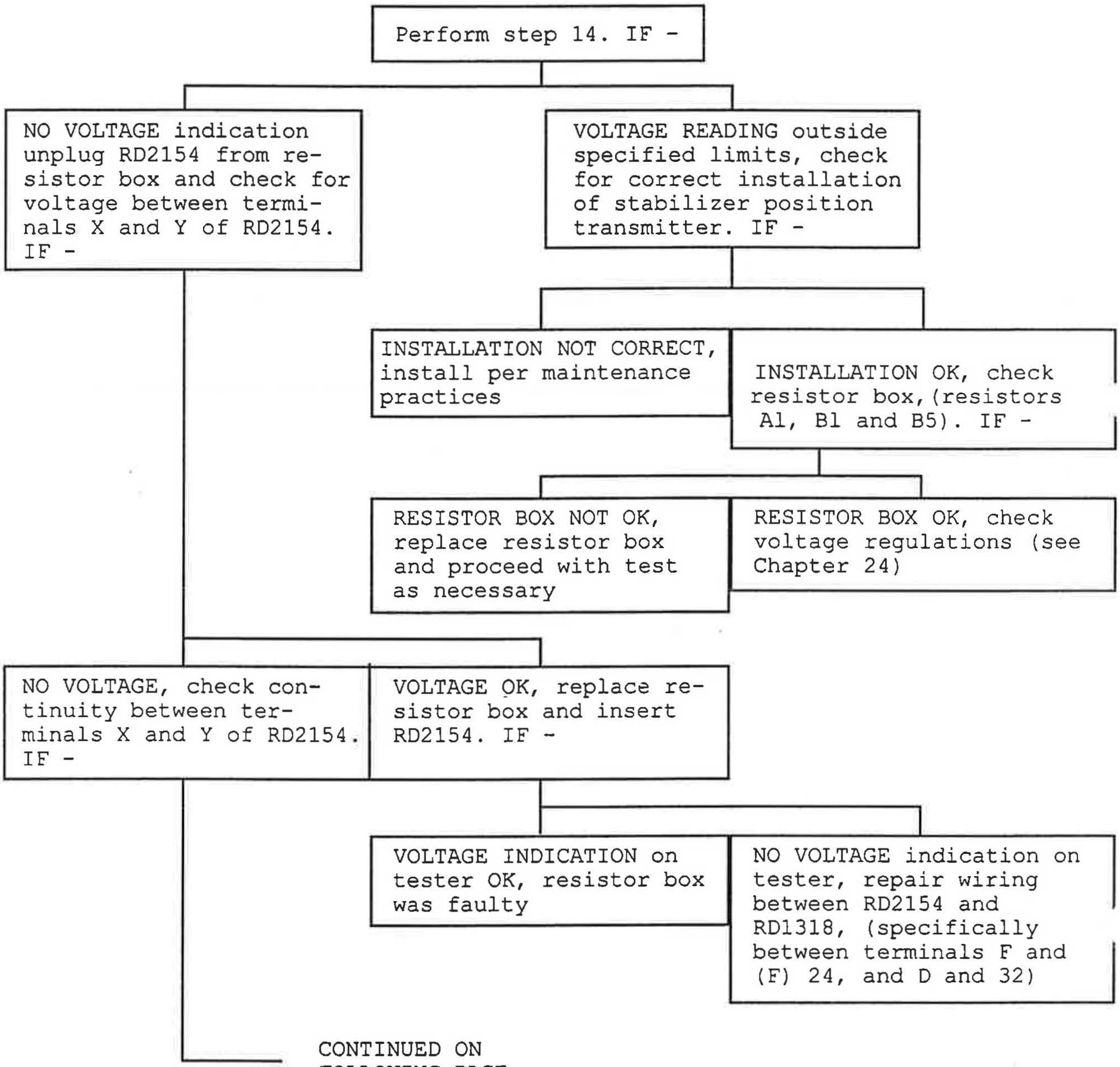
VOLTAGE OK, check for continuity between pins P and N on RM984. IF -

NO CONTINUITY, replace trim servo and reconnect RD4611 and proceed with test.

CONTINUITY OK, reconnect RD4611. IF -

SAFETY SW TEST light still fails to come on, repair wiring between terminal P on RD4611 as applicable and terminal 29 on RD1318.

Trouble Shooting Chart C



CONTINUED ON
FOLLOWING PAGE

Trouble Shooting Chart D (Sheet 1)

CONTINUED FROM
PRECEDING PAGE

CONTINUITY OK, Plug in RD2154, remove RD1318 from mach trim coupler and check for continuity between pins 25 and 26 of RD1318. IF -

NO CONTINUITY, replace stabilizer position transmitter and recheck.

CONTINUITY OK, check for voltage at pins 25 and 26 of RD1318 receptacle on mach trim coupler. IF -

NO CONTINUITY, replace stabilizer position transmitter and recheck.

VOLTAGE OK, replace stabilizer position transmitter and recheck.

NO VOLTAGE, replace mach trim coupler.

Trouble Shooting Chart D (Sheet 2)

MACH TRIM SYSTEM - MAINTENANCE PRACTICES

1. Adjustment/Test Mach Trim System

   A. Equipment

      (1) Pitot-static pressure test equipment.

         (a) Air pressure source, 0-15 inches a of mercury, with adapter hose for connection to pitot head and gauge for measuring pressure.

         (b) Vacuum, source, 0-19 inches of mercury, with fittings for connections to static line test fittings and gauge for measuring vacuum.

         (c) Controllable cross -fed assembly, enabling static source (vacuum) to be fed to pitot lines.

   B. Test Mach Trim System

      (1) Test mach trim interlock and warning light.

         (a) With essential radio bus and radio bus No. 2 energized, check mach trim circuit brezaker, autopilot circuit breaker and autopilot and mach trim disengaged light circuit breaker are closed on radio and T-R circuit breaker panel (P5).

         (b) Set AUTO TRIM cutout switch to "CUTOUT" Mach trim disengaged light should be illuminated.

         (c) Set AUTO TRIM cutout switch to on and open mach trim circuit breaker. Mach trim disengaged light should be illuminated.

         (d) Place "AUTOPILOT-OFF" switch on autopilot control panel in "AUTOPILOT" position and release. Autopilot should be engaged as evidenced by Increased loading on control systems

         (e) Disengage autopilot, close mach trim circuit breaker and again engage autopilot. Mach trim disengaged light should not be illuminted, and autopilot should remain engaged.

(f) Disengage autopilot and then momentarily operate mach trim monitor test witch to "TEST " (operate for 5 to 10 seconds). Mach trim disengaged light should illuminate and remain illuminated after test switch in returned to "NORMAL". (Stabilizer trim wheel may drive momentarily before the light illuminates.

(g) Momentarily move AUITO TRIM cutout witch to "CUTOUT" and back to on. Mach trim disengaged light should go out and remain unlighted.

(h) Repeat (f) and (g) using "KIFIS" test switch.

(i) Engage autopilot and temporarily lift wire C498C22 from terminal H23 for N20199, AC7 for N19997, N20000 in autopilot junction box. Autopilot should disengage and trim disengaged light should illuminate.

(j) Replace wire C498C22.

(k) Mach trim disengaged light should remain illuminated.

(2) Test stabilizer trim synchronization.

(a) With AUTO TRIM cutout switch in "CUTOUT" position, move stabilizer using stabilizer trim witch through 2 or 3 units of travel as indicated on the stabilizer trim indicator and then set to zero trim (within 1/4 unit). Note the exact position.

(b) Switch AUTO TRIM cutout switch to on. Mach trim disengaged light should not illuminate.

(c) Manually rotate stabilizer trim wheel in nose up trim direction until mach trim disengaged light illuminates. Trim position indicated should be no less than 1 unit and no more than 2 units from the position noted in (a).

(d) Manually rotate stabilizer trim wheel back to zero trim and momentarily move AUTO TRIM cutout switch to "CUTOUT." The mach trim disengaged light should go out and remain unlighted.

(e) Repeat (a) through (d) for a nose down trim condition.

(3) Test mach relay operation.

(a) With a pressure vacuum source connected to copilot's machmeter pitot and static lines (see Chapter 34, Pitot-static systems) and AUTO TRIM cutout switch in on, slowly adjust the applied pressure to increase the simulated mach indication until stabilizer trim servomotor starts to run. The mach value should be no less than 0.79 and no more than 0.81

(4) Test mach trim calibration.

(a) With a pressure vacuum source connected to MACH Synchrotel Transmitter pitot and static lines and AUTO TRIM cutout switch in CUTOUT position, move stabilizer using stabilizer trim switch through 2 or 3 units of travel an indicated on the stabilizer trim indicator and then set to zero trim (within 1/4 unit). Note the exact position.

(b) Move automatic stabilizer trim cutout switch to on (NORMAL).

(c) Adjust pitot and static pressures as necessary to produce the following mach readings as indicated on machmeter and measure the stabilizer trim angles. Verify that measured stabilizer angles check with the following tables and stabilizer deflects nose-up.

NOTE: Stabilizer trim wheel may be used to determine stabilizer travel. (12.4 revolutions of trim wheel equals 1 degree stabilizer movement.)

Mach Indicator	Rev. Trim Wheel	Stabilizer Angle
0.81	0.0 to 1.0	0.0 to 0.1
0.86	5.0 (± 1.5)	0.4 (± 0.12)
0.88	10.0 (± 3.0)	0.8 (± 0.25)
0.92	25.0 (± 3.0)	2.0 (± 0.25)

(d) Verify that motion of control column at any of the above settings should not cause stabilizer to move.

(e) Hold control column at neutral position and engage autopilot system.

(f) Push control column forward. Stabilizer trim wheel should move towards nose-up.

(g) Hold control column at neutral and turn stabilizer trim wheel about three revolutions by hand. Wheel should remain in final position showing no effects of trim servo operation.

(h) Disengage autopilot and disconnect pitot-static pressure equipment.

C. Directional Test of Mach Trim System

NOTE: The following test should be performed whenever the mach trim coupler or stabilizer position transmitter is replaced.

(1) Connect external power to the airplane.

(2) Place auto stab trim cutout switch to on (NORMAL).

(3) Position stabilizer with manual electric trim to zero units trim position.

(4) Position stabilizer to 2 units airplane nose-up position using manual stabilizer trim wheel crank.

NOTE: The mach trim monitor light will illuminate prior to reaching 1-1/2 units airplane nose-up stabilizer position.

(5) Hold the KIFIS test or mach trim test switch in TEST position.

(6) Position the auto stab trim cutout switch to CUTOUT position.

(7) Restore the auto stab trim cutout witch to on (N0RMAL) position.

(8) The mach trim monitor light shall extinguish and remain out. If this light illuminates, the mach trim system wiring may be incorrect and should be specifically checked.

NOTE: The amount and direction of trim observed will vary with airplane model and is not relevant to this check.

(9) Determine it there is further need for external power, if not, remove from airplane.

2. Mach trim system test

CAUTION : NO MECHANICS IN THE BACK.

1.	Stabilizer trim cutout switch "ON". Auto trim cutout switch in "NORMAL"	Mach trim warning light off.
2.	Pull the Mach Trim Control AC breaker on panel P5.	Mach trim warning light on.
	Close the Mach Trim Control AC breaker on panel P5.	Mach trim warning light goes off.
3.	Set mach trim test switch in "test". Set auto trim switch in "cutout". Briefly actuate the electric with the left Trim switch.	Mach trim warning light on.
	Set auto trim switch in "NORMAL".	Mach trim warning light goes off.
	Release mach trim test switch. (springs back to normal)	Mach trim warning light on.
	Set auto trim switch in "cutout" Briefly actuate the electric trim switch.	
	Set auto trim switch in "NORMAL".	Mach trim warning light goes off.
4.	Set KIFIS test switch to "test" and hold it in the position.	Mach trim warning light on. The trim may move briefly.
5.	Set auto trim cutout switch in "CUTOUT".	
	Try to engage the autopilot.	Should not engage on N20199.
	Briefly actuate the electric trim with the right switch.	Should engage on N19997 & N20000.
	Set auto trim cutout switch in "NORMAL"	Mach trim warning light goes off
6.	Try to turn the trimwheel in both directions.	Increased resistance must be felt in both directions.
	Release KIFIS test switch. Set auto trim cutout switch in "CUTOUT" and back to "NORMAL"	Mach trim warning light on. The trimwheel may move briefly.
7.	Engage the autopilot. Disconnect autopilot with wheel disconnect switch.	
8.	Manually turn the trimwheel in nose up direction.	The mach trim warning lamp illuminates after 17 $\pm$ 5 rotations of the trimwheel. From zero trim position.
9.	Set auto trim cutout switch in "CUTOUT" and back to "NORMAL"	Mach trim warning lamp goes off.
10.	Determine if there is further need for external power, if not, remove from the airplane.	

CAUTION : **THE STABILIZER TRIM MOTOR IN NOT DESIGNED FOR CONTINUOUS OPERATION. MAXIMUM SWITCHING PERIOD 2 MINUTES, THEN 15 MINUTES PAUSE.**

THE TRIMWHEEL MAY DO 0.5 ROTATION IN AN ANTI-CLOCKWISE DIRECTION AT TEST POINTS THAT RELATE TO THE TRIMMING.

MACH TRIM CUTOUT SWITCH - MAINTENANCE PRACTICES

1. Removal/Installation Mach Trim Cutout Switch

A. Remove Mach Trim Cutout Switch (See figure 201)

(1) Remove screws from access panel on lower forward right side of control stand.

(2) Remove panel.

(3) Remove pins fastening rod ends to both switches on mounting plate. The aft switch is the mach switch.

(4) Remove four allen head screws holding mounting plate for mach switch.

(5) Lift switches and mounting plate up enough so the electrical plug is accessable.

(6) Disconnect electrical plug, RD2431.

(7) Remove two mounting screws for mach trim cutout switch.

NOTE: Retain boot from switch shank for installation on new switch.

(8) Remove switch.

B. Install Mach Trim Cutout Switch

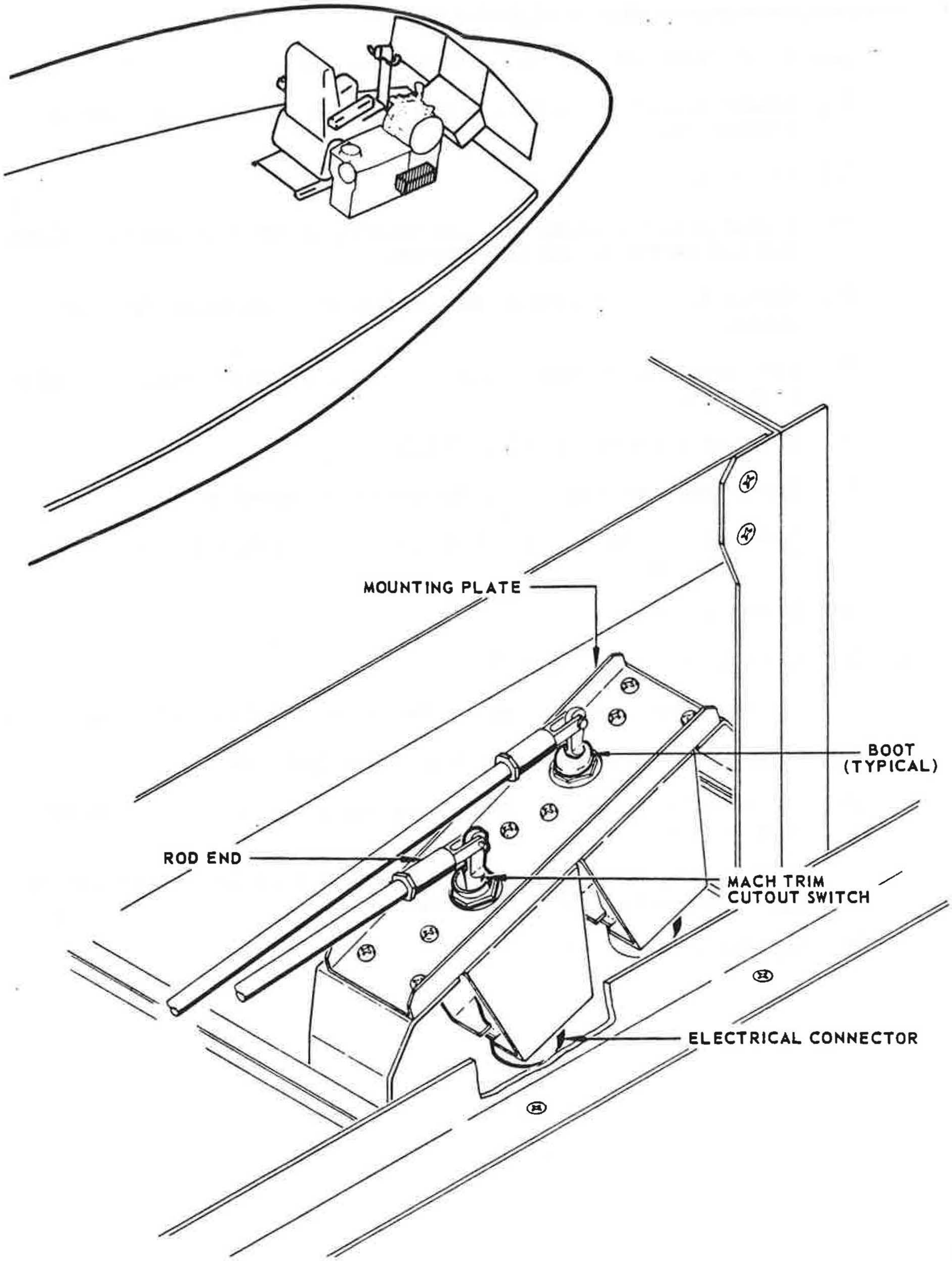
(1) Replace screws securing mach trim cutout switch to mounting plate.

(2) Connect electrical leads by connecting plug RD2431.

(3) Put mounting plate and switch assembly in place securing it with four screws.

(4) Install boot over shank of switch and replace rod ends on switch operating handles using new pins.

(5) Replace access panel.



Mach Trim Cutout Switch
Figure 201

MACH TRIM SYSTEM STABILIZER POSITION TRANSMITTER - MAINTENANCE PRACTICES

1. Removal/Installation Stabilizer Position Transmitter

A. Prepare for Removal

(1) Rotate manual stabilizer trim control wheels to the stops for an airplane nose down condition.

(2) Remove stabilizer jackscrew access panel. Refer to Chapter 12, "Access Doors and Panels."

(3) Deactivate autopilot and stabilizer trim nose up limit switches by depressing actuating plunger with 3/4" OD washer and securely tape in position to hold them down.

(4) With power on to autopilot system and stabilizer trim system actuate autopilot and stabilizer trim system to determine that the stabilizer can not be moved electrically.

NOTE: In lieu of steps (3) and (4), either a stabilizer support jack assembly F52238-501 or a stabilizer trim lock assembly F71336 or equivalents may be used to prevent stabilizer movement.

(5) Flag manual stabilizer control wheels so they can not be operated.

(6) Disconnect power from system by opening circuit breakers on radio and T-R circuit breaker panel (P5).

WARNING: WHILE A MAN IS IN THE REQUIRED POSITION TO WORK ON THE STABILIZER POSITION TRANSMITTER HE COULD BE CRUSHED IF STABILIZER IS ALLOWED TO MOVE.

B. Remove Position Transmitter (See figure 201.)

(1) Crawl through the opening under the stabilizer to reach station 1578 and LBL 15.

(2) Remove the electrical connector on the end of position transmitter.

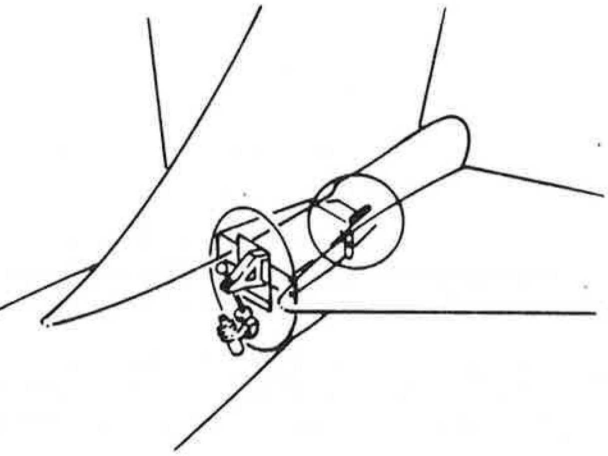
(3) Loosen locking bolt holding actuating arm on shaft of transmitter.

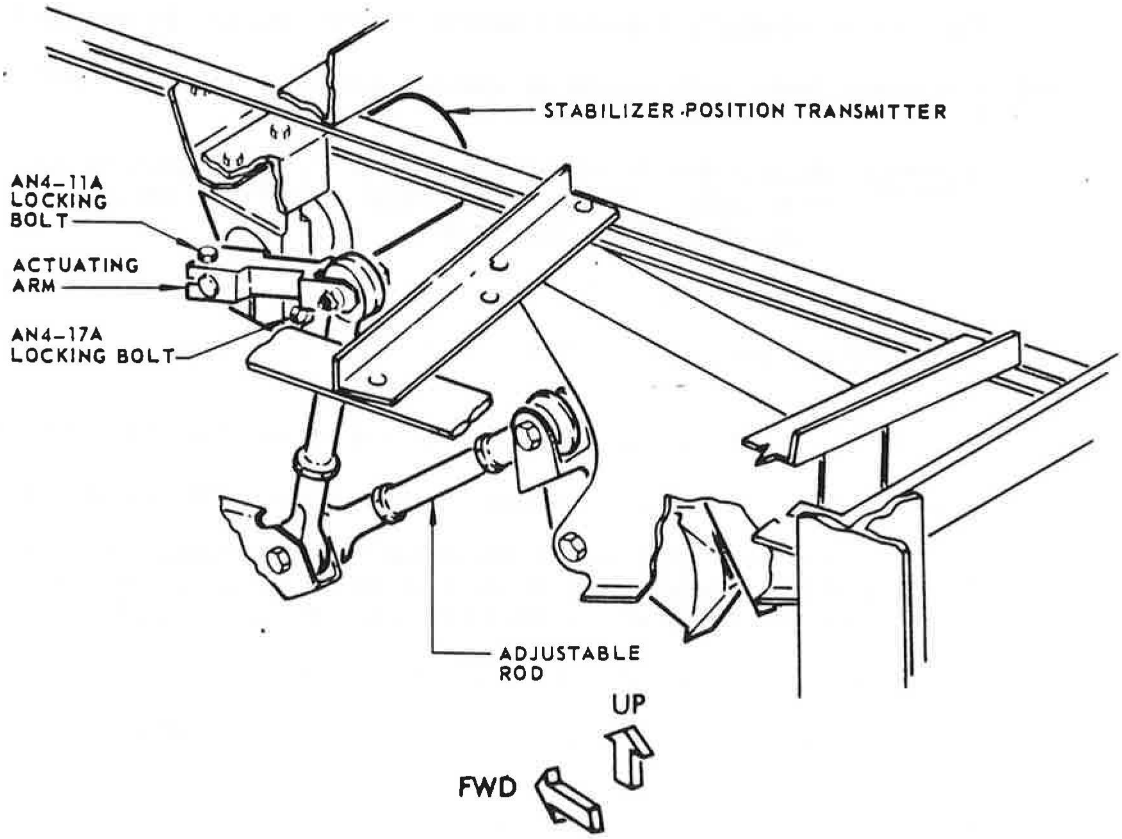
NOTE: Observe the position of the actuating arm relative to the position transmitter; it must be reinstalled in the same position pointing aft from the shaft of the transmitter.

(4) Remove actuating arm from position transmitter shaft.

(5) Loosen locking bolt clamping position transmitter in mounting bracket.

(6) Slide position transmitter out of mounting bracket.





Stabilizer Position Transmitter
Figure 201

B. Install Position Transmitter

(1) Verify that stabilizer is still positioned for an airplane full nose down position and safety precautions axe observed. (See figure 201.) See steps A.(1), (2), (3), (4), (5).

WARNING: WHILE A MAN IS IN THE REQUIRED POSITION TO WORK ON THE STABILIZER POSITION TRANSMITTER THE COULD BE CRUSHED IF STABILIZER IS ALLOWED TO MOVE.

(2) Insert position transmitter in mounting bracket and secure locking bolt in position.

(3) Place actuating arm on end of position transmitter shaft. Actuating arm and pushrod connection must point aft.

(4) Align index mark on transmitter shaft with index mark on transmitter case and tighten locking bolt clamping actuating arm to the position transmitter shaft.

(5) Add one AN960D416 washer under nuts of AN4-11A and AN4-17A locking bolts.

(6) Tighten all locking bolts securely (around 70 to 100 inch/lb torque). Esure that transmitter cannot turn in mounting bracket.

(7) Insert electrical connector.

(8) Activate limit switches by removing tape and washers.

(9) Install access panel. (185) or (1185).

(10) Check system. See 22-10-0, Mach Trim System Maintenance Practices.

(11) Remove flags and power from systems.

AUTOPILOT FLIGHT CONTROL COMPONENTS - DESCRIPTION AND OPERATION

EFFECTIVITY

AIRPLANES WITH TWO-SPEED AUTOPILOT STABILIZER TRIM SERVO

1. Surface Servo

A. TWO surface servos are used in the autopilot system. They Position the rudder, elevator and aileron control tabs in response to control signals, thereby providing correct positioning of the main control surfaces. The rudder servo is not installed in airplanes with series yaw damper system.

B. Each surface servo consists of a servomotor and rate generator, Autosyn synchro, gear train, accurate electromagnetic clutch, and a special pulley and servo mount. Provisions for adjustable torque limiting are included.

(1) The servomotor is a two-phase induction motor having a rate generator integrally mounted on its top. A torque-limiting resistor and a phaseshifting capacitor for the motor are included inside the servo housing.

(2) The Autosyn position follow-up synchro is geared to the motor output shaft through a series of split gears which connect to the Autosyn rotor shaft, giving a speed reduction.

(3) The gear train consists of a differential and two stages of planetary Sears. The cage of the differential is spring loaded to the servo housing. Two torque-limit microswitches, one mounted to either side of the cage, are actuated by an am on the cage. The planetary gearing gives a speed reduction. The low side of the gearing drives the solenoid clutch directly.

(4) The solenoid clutch provides positive on-off action-only. Action is obtained in a multi-disc clutch assembly placed between the rotating solenoid clutch and the output shaft. The solenoid clutch has no sliding parts, thereby eliminating the possibility of binding or jamming which can occur when sliding parts move into contact. Instead, the translating parts are supported on a diapharm and bellows and only flexing of the diaphragm results when the clutch is engaged or disengaged. The bellows and diapharm place a spring load between the driving and driven clutch parts for positive, accurate on-off action. The slip clutch is a dry, sintered bronze type comprising five discs. It has an adjustable slip value. This clutch drives the splined output shaft. This shaft is hollow and contains a coaxial shaft that is connected to the input side of the solenoid clutch. The output end of the smaller shaft is hexagonal for attachment of a torque wrench, permitting check readings on the actual servo torque (not servo torque less rigging friction.)

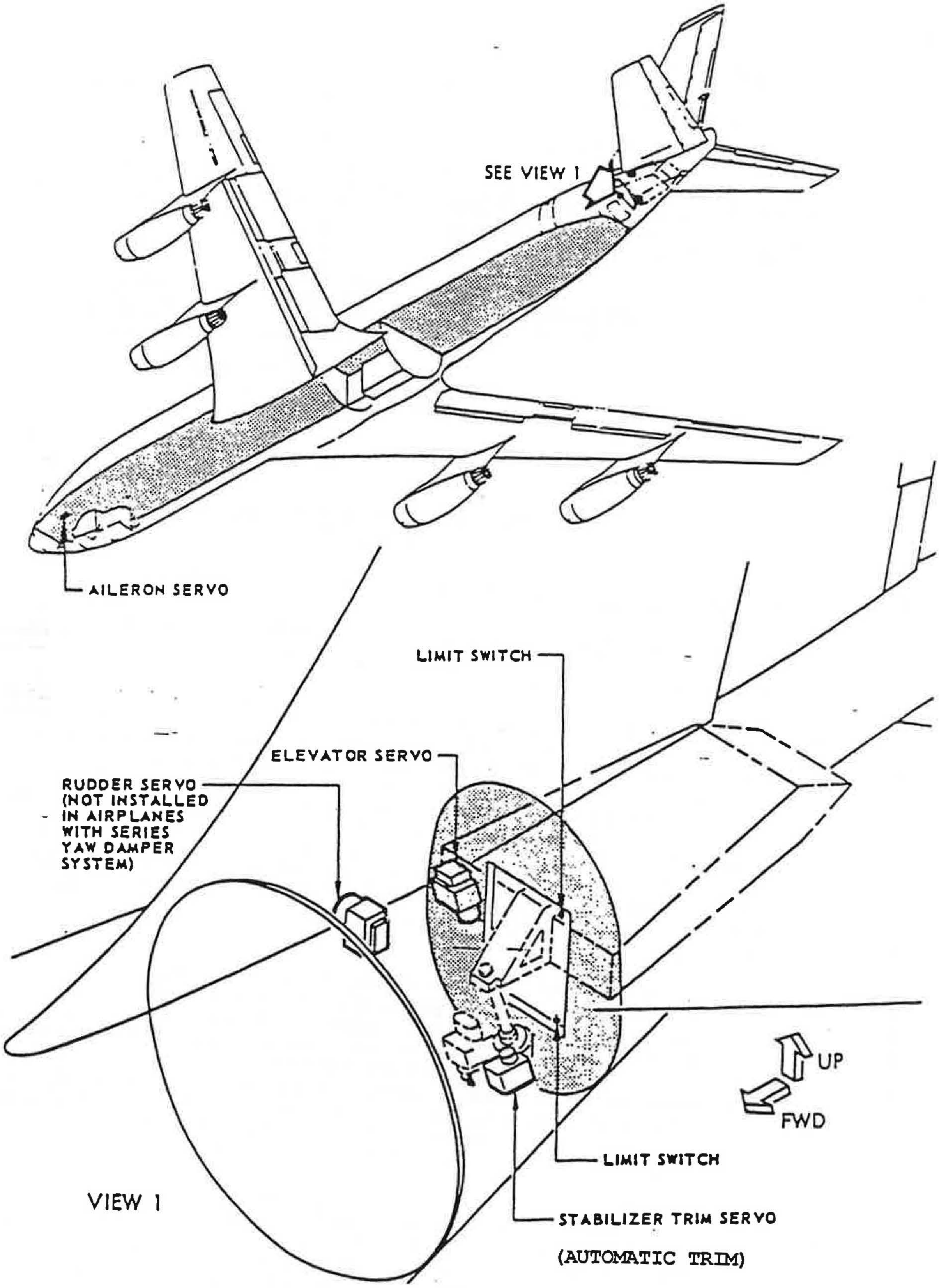
(5) The servo is mounted in the airplane on a special pulley and servo mount assembly by means of tour screws. This mounting makes it possible to remove the servo without disturbing the rigging.

(6) Electrical connections are made at a terminal board inside the servo housing. It is readily accessible through an outside cover plate. Connection to the external wiring is made through an electrical connector mounted on the side of the servo housing.

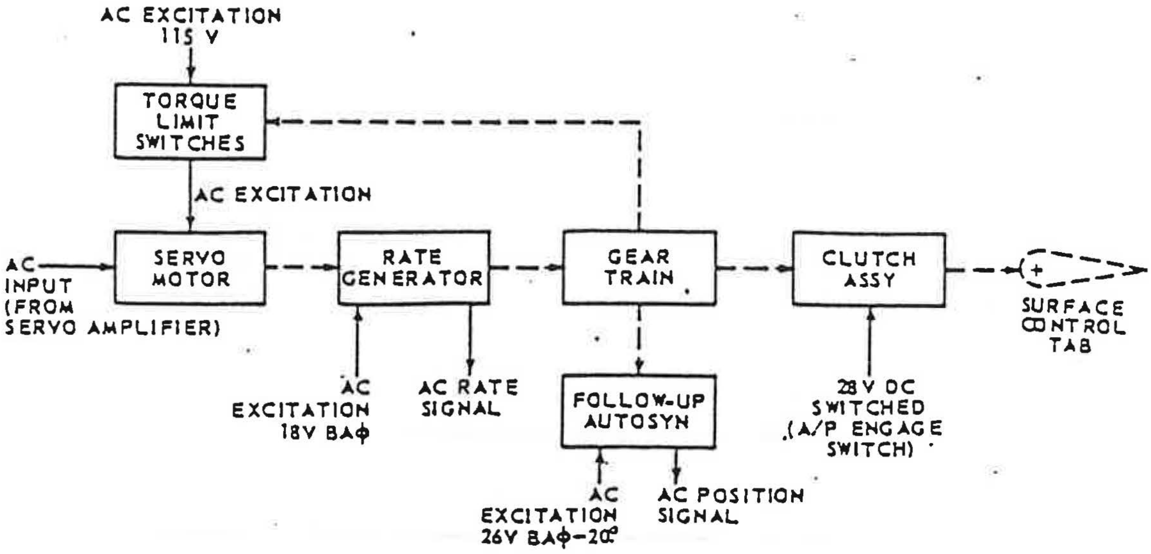
C. The aileron surface servo is located in the lover nose compartment, mounted to a bracket on the underside of the floor beam, approximately two feet to the left of the centerline and just aft of the pilot's control stand aileron cable drum. (See figure 1.) It in readily accessible through the floor hatch beneath the navigator's table in the control cabin, or from the rear of the lower nose compartment. The elevator servo is located in the tail section, approximately six feet aft of the rear cabin pressure bulkhead at floor beam level. It to cerntered below the stabilizer torque box. (See figure 1.) The elevator servo in accessible through the stabilizer jack-screy access panel on the bottom of the tail section.

D. In the operation of each surface servo, its motor fixed phase is excited by 115-volt, 400-cycle power from the power junction box. The variable phase is supplied by a type SA-40 magnetic amplifier in the amplifier and computer case. The variable phase voltage is developed from the input control signal and varies from a nominal zero vaue to a maximum of approximately 240 volts. The torque-limiting resistor connected in series with the variable phase reduces the maximum voltage, thereby, lowering servo torque as required for safe airplane operation. A capacitor placed across the variable phase adjusts its phase angle. In response to a control signal the motor drives the rate generator and follow-up Autosyn to provide electrical rate feedback and servo shaft position signals respectively to the servo amplifier. Also, the motor pinion drives the solenoid clutch through the differential and planetary gear train. Torque applied to the differential results in a proportional movement of the cage against the springs. When a preset torque limit is reached, the cage arm will actuate one or the other torque limit switches to remove fixed-phase motor excitation. The switch actuated, depends upon the direction of applied torque. When the solenoid clutch is engaged it drives through the diaphragm and bellows into the multi-disc slip clutch. The disc clutch then drives the splined output shaft which mates with the internal spline of the pulley on the support bracket. The pulley in rigged with control system cables to activate the control surface tabs. Figure 2 gives the surface servo block diagram.

E. Separate protective fusing for servo power is not required, since it is provided in the autopilot power junction box.



Autopilot Flight Control Components Location
Figure 1



Surface Servo Block Diagram
Figure 2

2. Stabilizer Trim Servo, (Bendix/Eclipse Pioneer 16729 series)

A. The trim servo supplies automatic pitch trim, through actuation of the horizontal stabilizer assembly, to remove sustained out-of-trim loads from the elevator servo.

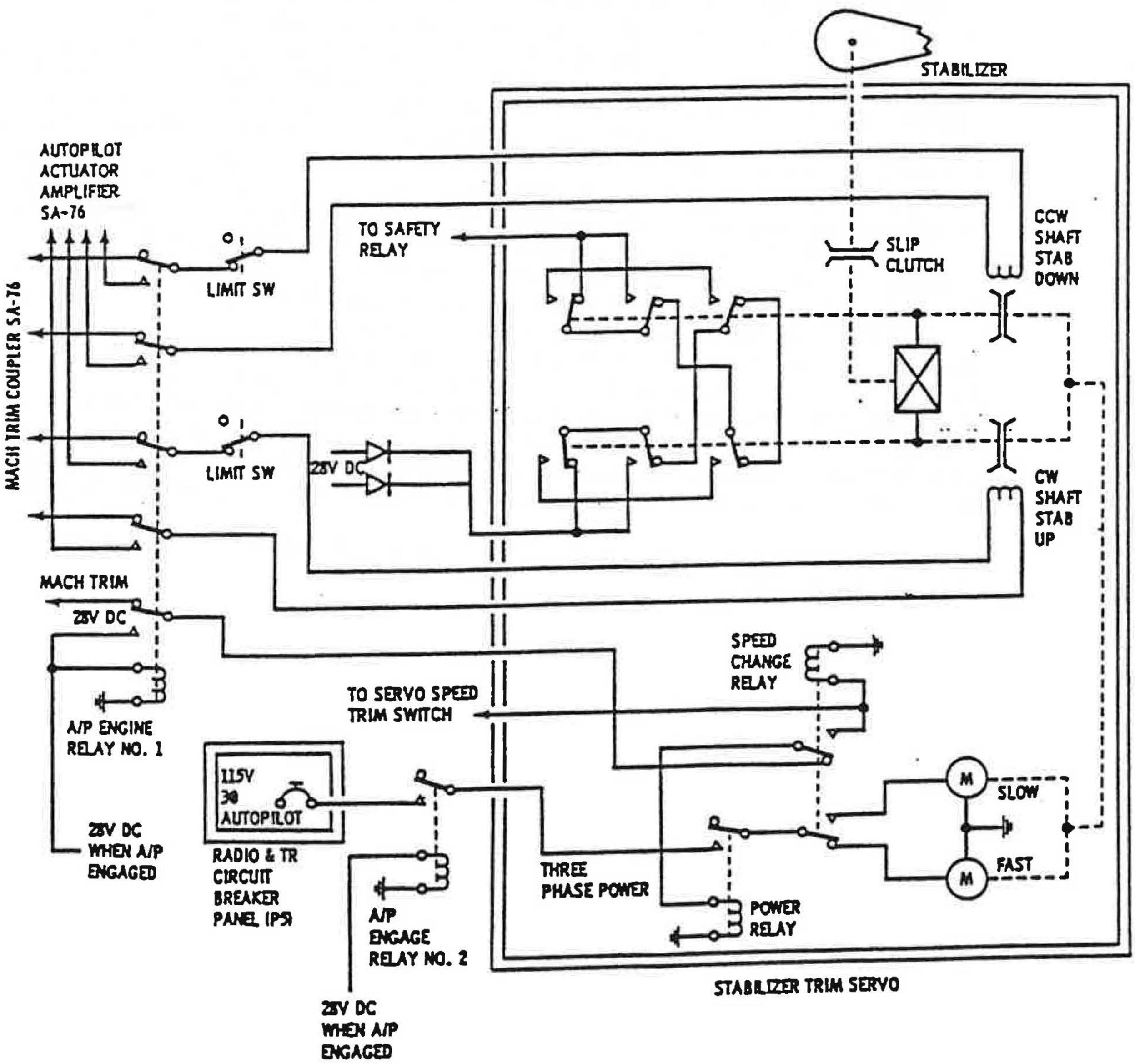
B. The trim servo consists of a motor with two separate field windings, two magnetic solenoid clutches, power relay, speed change relay, interlock switch assembly, slip clutch and output gear train enclosed in a cast housing. The servo motor has two different speed-torque characteristics. Three-phase power is supplied to the slow speed servo motor when both the power relay and the speed change relay are energized. When the speed change relay is deenergized (by moving the trim servo speed switch to the flaps down position, or the turn controller out of detent In manual mode), power is supplied to the high speed servo motor. (See figure 4.) When the mach trim system is operational the high speed motor is used. See Chapter 22-30-0.

(1) A pinion gear is keyed to the motor shaft and secured with a nut and lock washer. The motor pinion meshes with a gear train which turns a gear splined to a clutch assembly. This gear also meshes with a corresponding splined gear on a duplicate clutch assembly. The solenoid clutch in each assembly is a double-disc and diaphragm type in which sliding parts are eliminated and the spring action of the diaphragm gives sharp, clean engagement and disengagement. A solenoid shaft, screwed into the armature, then exerts a squeezing force on the shaft and diaphragm assembly, locking it to the clutch body.

(2) The two interlock switch assemblies each consists of three microswitches on a mounting bracket. One of these assemblies is mounted with each clutch assembly so that linear motion produced by clutch engagement and disengagement operates the microswitches.

(3) The driven shaft of each clutch assembly meshes with the primary gear of the output gear train. This gear train consists of a small and large shaft and associated speed reduction gears mounted in the servo housing. An extension of the large shaft through the housing forms a splined output shaft. Leads from the safety interlock switches are connected to a terminal board which is part of the output gear train assembly.

(4) An electrical connector for the external power leads is mounted on the side of the servo housing.



Stabilizer Trim Servo Circuit (Bendix)
Figure 4

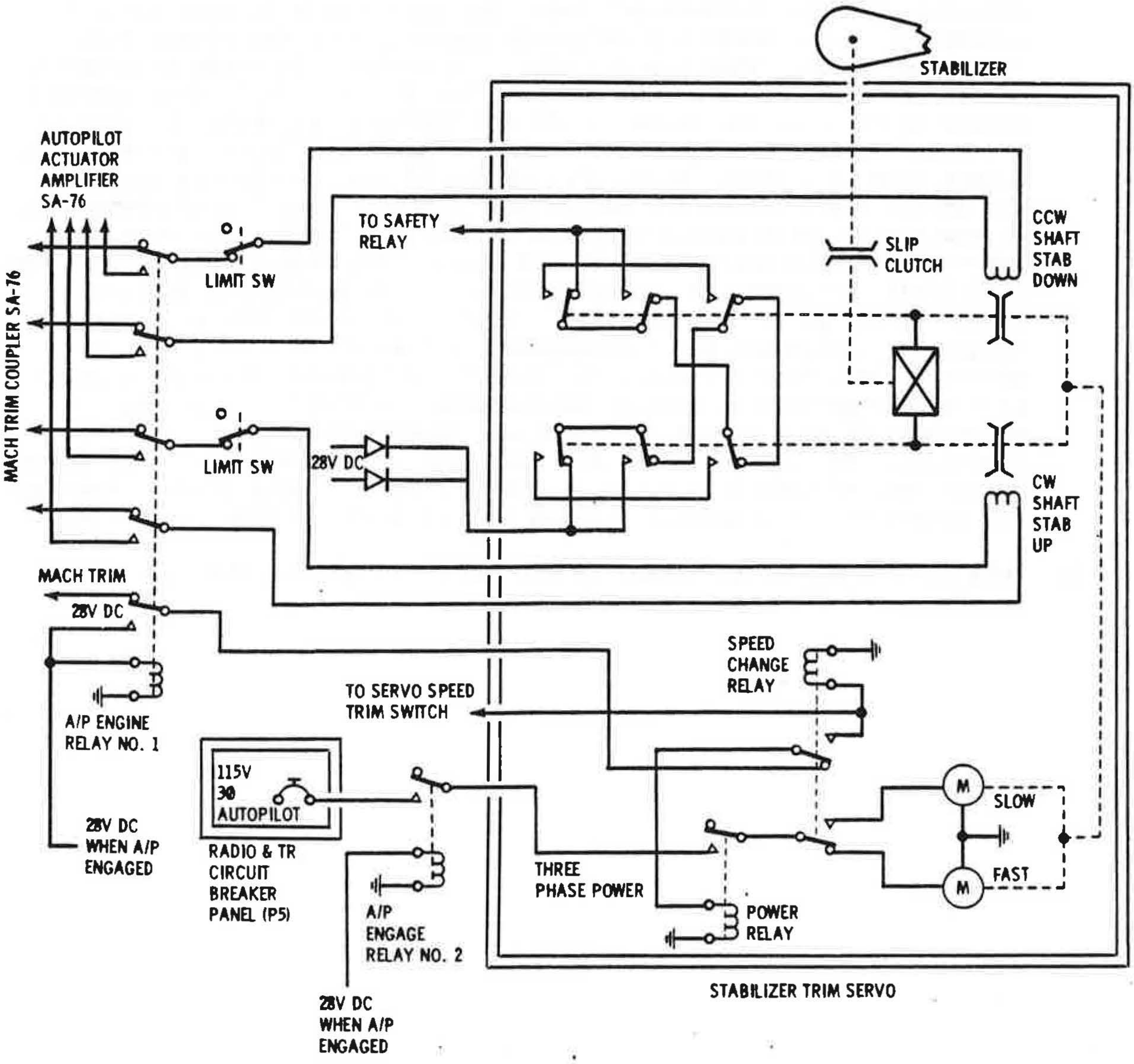
C. The stabilizer trim servo is located in the tail section aft of the rear cabin pressure bulkhead. It is mounted on the stabilizer ball nut and jackscrew assembly. (See figure 1.) It is accessible through the stabilizer jackscrew access panel on the underside of the tail section.

D. The stabilizer trim servo operates in parallel with the elevator servo. Part of the variable-phase power for the latter is fed to a demodulator amplifier (SA-76) in the amplifier and computer package which supplies direct current to one or the other clutch solenoids in accordance with the phase of the variable voltage delivered to the elevator servo. The servo motor, supplied with 115-volt, 400-cycle, three-phase power through the power relay and speed change relay is under constant drive while the autopilot is engaged. It turns the two clutch assembly gears, one clockwise and one counterclockwise. The clutch body is thus rotated around the center shaft and diaphragm assembly when the clutch solenoid is not energized. When the solenoid is energized, it draws an armature or keeper to the poles of the magnet. The solenoid shaft then exerts a squeezing force on the center shaft and diaphragm assembly, locking to the clutch body. The shaft and diaphragm assembly then rotate with the clutch body as a unit. Thus, when either of the clutches is energized, the driven shaft drives the output gear train. Direction of rotation is determined by which clutch supplies the driving force. The gear train drives the output shaft at a reduced speed. The output shaft drives the stabilizer jackscrew, providing movement of the stabilizer surface to establish the pitch trim required. Since the linear motion produced by the clutch engagement and disengagement actuates the interlock microswitches, they are connected into the autopilot interlock circuit as a safety protection against malfunction. A closed circuit is maintained if both clutch solenoids are de-energized or if either is energized. If both solenoids are energized simultaneously, or if there is any type of switch malfunction, an open circuit will occur disengaging the autopilot. A schematic diagram of this servo is given in figure 4.

E. Trim servo power is furnished by the radio and 28V d-c T-R circuit breaker panel (P5).

(3) The driven shaft of each clutch assembly meshes with the primary gear of the output gear train. This gear train consists of a small and large shaft and associated speed reduction gears mounted in the servo housing. An extension of the large shaft through the housing forms a splined output shaft. Leads from the safety interlock switches are connected to a terminal board which is part of the output gear train assembly.

(4) An electrical connector for the external power leads is mounted on the side of the servo housing.



Stabilizer Trim Servo Circuit (Bendix)
Figure 4

C. The stabilizer trim servo is located in the tail section aft of the rear cabin pressure bulkhead. It is mounted on the stabilizer ball nut and jackscrew assembly. (See figure 1.) It is accessible through the stabilizer jackscrew access panel on the underside of the tail section.

D. The stabilizer trim servo operates in parallel with the elevator servo. Part of the variable-phase power for the latter is fed to a demodulator amplifier (SA-76) in the amplifier and computer package which supplies direct current to one or the other clutch solenoids in accordance with the phase of the variable voltage delivered to the elevator servo. The servo motor, supplied with 115-volt, 400-cycle, three-phase power through the power relay and speed change relay is under constant drive while the autopilot is engaged. It turns the two clutch assembly gears, one clockwise and one counterclockwise. The clutch body is thus rotated around the center shaft and diaphragm assembly when the clutch solenoid is not energized. When the solenoid is energized, it draws an armature or keeper to the poles of the magnet. The solenoid shaft then exerts a squeezing force on the center shaft and diaphragm assembly, locking to the clutch body. The shaft and diaphragm assembly then rotate with the clutch body as a unit. Thus, when either of the clutches is energized, the driven shaft drives the output gear train. Direction of rotation is determined by which clutch supplies the driving force. The gear train drives the output shaft at a reduced speed. The output shaft drives the stabilizer jackscrew, providing movement of the stabilizer surface to establish the pitch trim required. Since the linear motion produced by the clutch engagement and disengagement actuates the interlock microswitches, they are connected into the autopilot interlock circuit as a safety protection against malfunction. A closed circuit is maintained if both clutch solenoids are de-energized or if either is energized. If both solenoids are energized simultaneously, or if there is any type of switch malfunction, an open circuit will occur disengaging the autopilot. A schematic diagram of this servo is given in figure 4.

E. Trim servo power is furnished by the radio and 28V d-c T-R circuit breaker panel (P5).

RUDDER SERVO CABLE DRUM AND BRACKET - MAINTENANCE PRACTICES

EFFECTIVITY

AIRPLANES INCORPORATING
FLIGHT CHARACTERISTIC IMPROVEMENT PROGRAM

1. Removal/Installation Rudder Servo Cable Drum and Bracket

A. Remove Rudder Servo Cable Drum and Bracket

(1) Remove rudder tab control quadrant access panel. See Chapter 12, "Access Doors and Panels."

(2) Remove stabilizer jackscrew access panel. See Chapter 12, "Access Doors and Panels."

(3) Depressurize auxiliary hydraulic system. See Chapter 27, "Rudder, Rudder Control Tab and Rudder Control System - Maintenance Practices."

WARNING: THE AUXILIARY HYDRAULIC SYSTEM MUST BE DEPRESSURIZED TO PREVENT INJURY TO PERSONNEL WORKING ON RUDDER AND VERTICAL FIN.

(4) Place rudder and rudder control tab in neutral.

(5) Install rigging pin through rudder control system aft quadrant. See Chapter 27, "Rudder, Rudder Control Tab and Rudder Control System - Maintenance Practices."

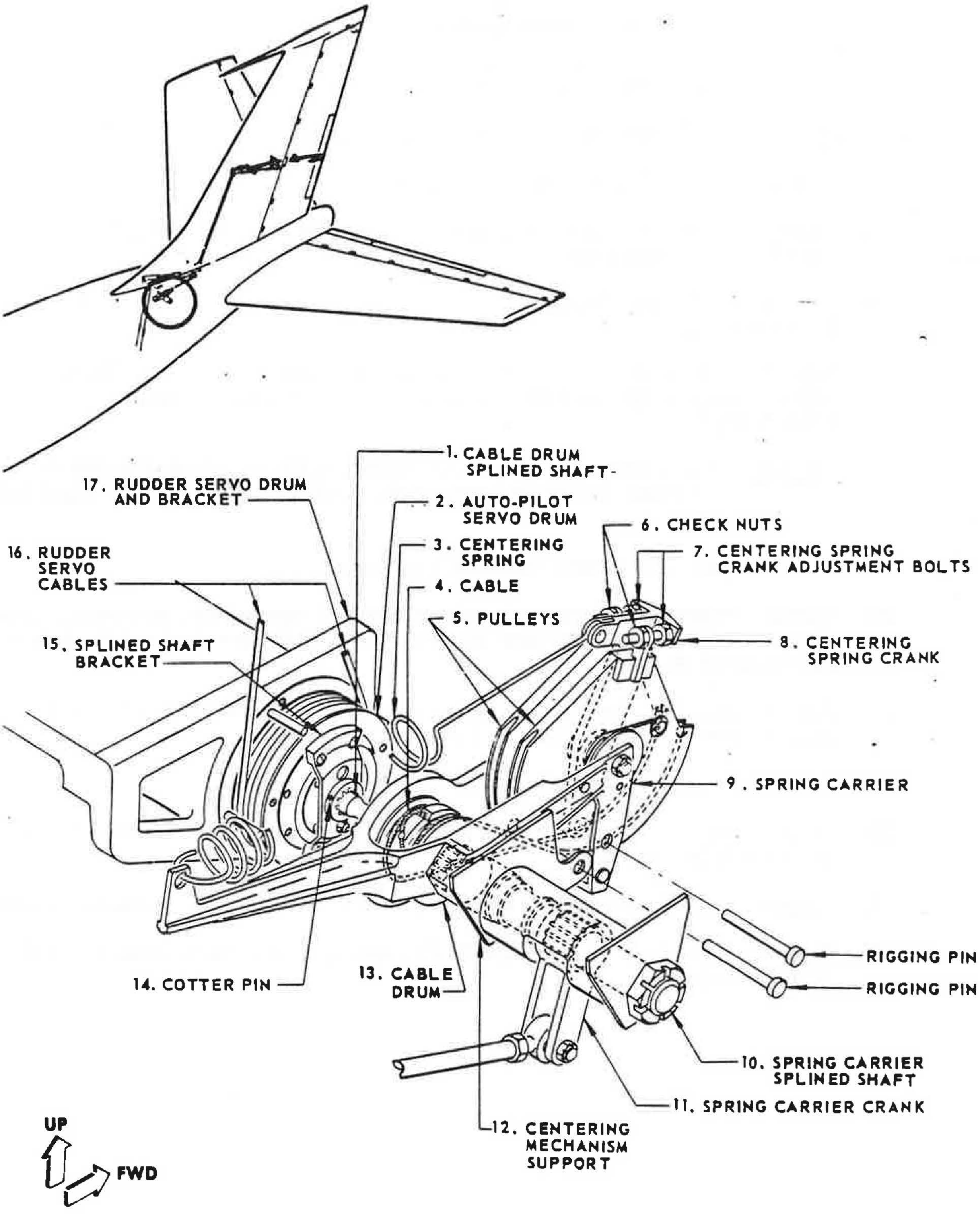
(6) Install rigging pin through centering mechanism support (12) and spring carrier (9). (See figure 201.)

(7) Disengage turnbuckles in rudder servo cables (16).

(8) Support autopilot rudder servo and remove electrical connector and four mounting bolts.

(9) Support rudder servo drum and bracket and remove four mounting bolts.

(10) Remove cable drum splined shaft (1) and splined shaft bracket (15) if necessary.



Rudder Servo Drum and Bracket and Centering Spring Installation
Figure 201

B. Install Rudder Servo Cable Drum and Bracket

(1) Install rigging pin through centering mechanism support (12) and spring carrier (9).

(2) Install splined shaft bracket (15) on servo drum when necessary and lockwire mounting bolts.

(3) Position servo drum and bracket and install four mounting bolts.

(4) Install rudder servo cables and adjust turnbuckles.

(5) Engage cable drum splined shaft (1) with splined shaft bracket (15), rotating spline if necessary.

(6) Secure cable drum splined shaft with cotter pin (14).

(7) Adjust centering spring crank adjustment bolts (7) until rigging pin can be installed without binding through spring carrier (9) and pulleys (5). Tighten check nuts (6) on adjustment bolts.

(8) Remove two rigging pins from centering mechanism and one rigging pin from rudder control system aft quadrant.

(9) Position autopilot rudder servo and install four mounting bolts and electrical connector.

(10) Install access panels.

(11) Test rudder trim control system.

SURFACE SERVO - MAINTENANCE PRACTICES

1. Removal/Installation Surface Servo

A. General

(1) The removal and Installation procedure of the two surface servos is the same. The aileron servo is located in the lower nose section at body station 240 and left buttock line 20. The aileron servo can be reached through the lower nose compartment interior access door.

(2) The rudder servo is located aft of the aft pressure bulkhead at body station 1484. Access to the servo is gained through the stabilizer jackscrew access panel. The rudder servo is not installed in airplanes with the series yaw damper system.

(3) The elevator servo is located aft of the aft pressure bulkhead and

under the stabilizer at body station 1519. Access to the servo is gained through the stabilizer jackscrew access panel and by crawling under the stabilizer.

WARNING: WHILE A MAN IS IN THE REQUIRED POSITION TO WORK ON THE ELEVATOR SERVO, HE COULD BE CRUSHED IF THE STABILIZER IS ALLOWED TO MOVE.

(4) To prevent injury, the following precautions should be taken.

(a) Rotate manual stabilizer trim control wheels to the stops for an airplane nose down condition.

(b) Remove stabilizer jackscrew access panel. Refer to Chapter 12, Access Doors and Panels.

(c) Deactivate autopilot and stabilizer trim nose up limit switches by depressing the actuating plunger with 3/4" OD washer and taping securely in position to hold them down.

(d) With power on to autopilot system and stabilizer trim system, actuate autopilot and stabilizer trim system to determine that the stabilizer cannot be moved electrically.

NOTE: In lieu of steps (c) and (d), either a stabilizer support Jack assembly F70057 or a stabilizer trim lock assembly F71336-500 or equivalents may be used to prevent stabilizer movement.

(e) Flag manual stabilizer control wheels so they will not be operated.

(f) Disconnect power from the system by opening circuit breakers on radio and T-R circuit breaker panel (P5).

B. Remove Surface Servo

(1) Disconnect the electrical connector on the servo.

(2) Supporting the servo, remove the four mounting bolts and remove the servo.

C. Prepare to Install Surface Servo

(1) Coat new "O" ring with Dow Corning Compound No. 4 and slip on servo.

D. Install Surface Servo

(1) Slide servo into mounting bracket and install four mounting bolts and lockwire.

(2) Connect electrical connector to servo.

(3) Activate stabilizer servo limit switches and remove flags from systems, where necessary.

(4) Test surface servo and stabilizer trim servo (where necessary) to ensure proper operation. See Autopilot System, Adjustment/Test.

2. Adjustment/Test Stabilizer Autopilot Trim Servo

A. Test Autopilot Trim Servo

(1) Check that horizontal stabilizer is free to travel.

(2) Position control column in full aft position and engage autopilot. See Autopilot System - Adjustment/Test.

(3) Move pitch wheel to pitch "UP" and check that stabilizer trim indicator on control stand indicates a change in airplane "NOSE UP TRIM."

(4) Disengage autopilot.

(5) Position control column full forward and hold. Engage autopilot.

(6) Position pitch wheel to pitch "DOWN" and check that stabilizer trim indicator indicates a change in airplane "NOSE DOWN TRIM."

(7) Disengage autopilot.

SURFACE SERVO - MAINTENANCE PRACTICES

1. Removal/Installation Surface Servo

A. General

(1) The removal and installation procedure of the three surface servos is the same. The aileron servo is located in the lower nose section at body station 240 and left buttock line 20. The aileron servo can be reached through the lower nose compartment interior access door.

(2) The rudder servo is located aft of the aft pressure bulkhead at body station 1484. Access to the servo is gained through the stabilizer jackscrew access panel. The rudder servo is not installed in airplanes with the series yaw damper system.

(3) The elevator servo is located aft of the aft pressure bulkhead and under the stabilizer at body station 1519. Access to the servo is gained through the stabilizer jackscrew access panel and by crawling under the stabilizer.

WARNING: WHILE A MAN IS IN THE REQUIRED POSITION TO WORK ON THE ELEVATOR SERVO, HE COULD BE CRUSHED IF THE STABILIZER IS ALLOWED TO MOVE.

(4) To prevent injury, the following precautions should be taken.

(a) Rotate manual stabilizer trim control wheels to the stops for an airplane nose down condition.

(b) Remove stabilizer jackscrew access panel. Refer to Chapter 12, Access Doors and Panels.

(c) Deactivate autopilot and stabilizer trim nose up limit switches by depressing the actuating plunger with 3/4" OD washer and taping securely in position to hold them down.

(d) With power on to autopilot system and stabilizer trim system, actuate autopilot and stabilizer trim system to determine that the stabilizer cannot be moved electrically.

NOTE: In lieu of steps (c) and (d), either a stabilizer support jack assembly F70057 or a stabilizer trim lock assembly F71336-500 or equivalents may be used to prevent stabilizer movement.

(e) Flag manual stabilizer control wheels so they will not be operated.

(f) Disconnect power from the system by opening circuit breakers on radio and T-R circuit breaker panel (P5).

B. Remove Surface Servo

(1) Disconnect the electrical connector on the servo.

(2) Supporting the servo, remove the four mounting bolts and remove the servo.

C. Prepare to Install Surface Servo

(1) Coat new "O" ring with Dow Corning Compound No. 4 and slip on servo.

D. Install Surface Servo

(1) Slide servo into mounting bracket and install four mounting bolts and lockwire.

(2) Connect electrical connector to servo.

(3) Activate stabilizer servo limit switches and remove flags from systems, where necessary.

(4) Test surface servo and stabilizer trim servo (where necessary) to ensure proper operation. See Autopilot System, Adjustment/Test.

AUTOPILOT FLAP SWITCH - REMOVAL/INSTALLATION

1. General

   A. The autopilot flap switch may be installed with splices or have a connector. Remove and install as applicable. (See figure 401.)

2. Remove Flap Switch

   A. Open main gear wheel well doors.

   B. Remove wire bundle retaining straps.

   C. Disconnect splices on switch leads or remove electrical connector from switch.

   D. If necessary, pull leads from wire bundle.

   E. Break lockwire, remove lower checknut and remove switch.

3. Install Flap Switch

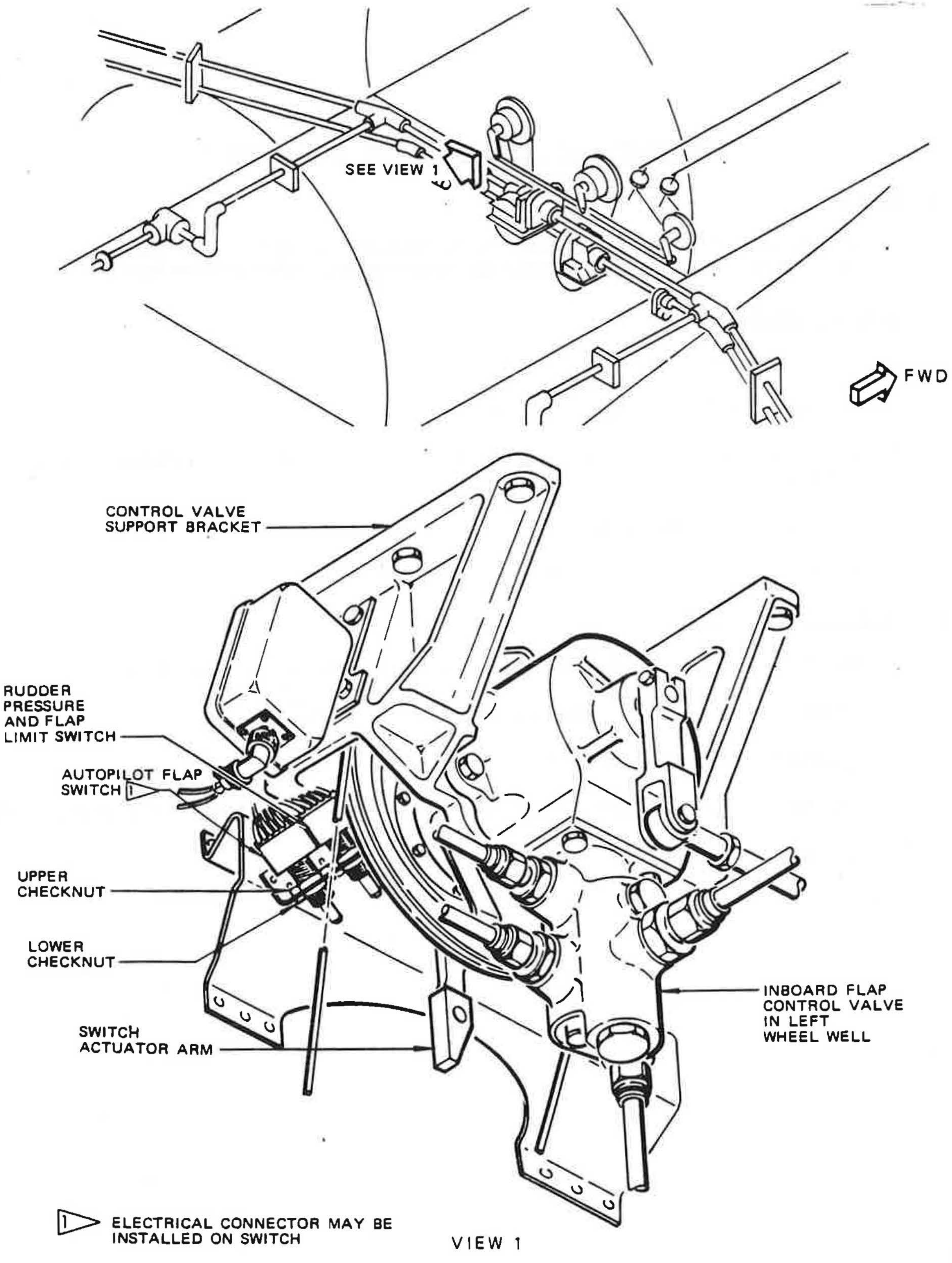
   A. Position switch in support bracket and install lower checknut.

   B. If necessary, thread leads through wire bundles.

   C. Splice switch leads to wiring bundle or attach connector.

   D. Adjust and test switch. Refer to Autopilot Flap Switch - Adjustment/Test.

   E. Lockwire switch.



Autopilot Flap Switch Installation
Figure 401

AUTOPILOT FLAP SWITCH - ADJUSTMENT/TEST

1. General

   A. The autopilot flap switch (RS224 or RS394) is operated when the flaps are lowered 0.5 degree or more. The following circuits are operated:

      (1) Stabilizer trim servo speed change relay is energized.

      (2) Yaw damper rate gyro output gain is increased.

      (3) On airplanes with elevator torque adapter, the autopilot flap switch RS394 is used to operate a relay in this adapter, increasing the maximum elevator channel output torque when the flaps are down. The flaps also operate a rudder pressure and flap limit switch (refer to Emergency Flap Drive Circuit Component, Chapter 27); one pair of contacts of this switch are used to monitor the operation of the autopilot flap switch. If either fails, dc power is completed to light B and the light comes on steady.

   B. The autopilot flap switch is located in the airplane left wheel well.

2. Autopilot Flap Switch Adjustment

   A. Equipment and Materials

      (1) Control Surface Protractor Assembly - F52485-500 or equivalent

      (2) Protractor Clamp Assembly - F71292-16 or equivalent

   B. Adjust Switch

      (1) Provide utility hydraulic power. Refer to Hydraulics, Chapter 29.

      (2) Install protractor assembly on inboard flaps and zero protractor.

      (3) Position flaps to 1 (+0.0/-0.5) degree.

(4) Adjust flap switch down toward switch actuator arm until switch actuates.

(5) Tighten checknuts and lockwire.

(6) Remove protractor.

(7) Turn off utility hydraulic power.

(8) Test autopilot flap switch.

3. Autopilot Flap Switch Test

A. Test Switch

(1) Provide electrical power.

(2) Provide utility hydraulic power. Refer to Hydraulics, Chapter 29.

(3) Test rudder gain change circuit.

(a) Open autopilot circuit breakers.

(b) Extend flaps 5 (± 2) degrees.

(c) On airplanes with yaw damper gain relay, apply 28 volts dc to terminal C18 in RJ20. On airplanes without yaw damper gain relay, delete step (c).

(d) Measure continuity between terminals D15 and E16 in RJ20. Check that resistance is less than 5 ohms.

(e) Retract flaps. Check that resistance increases.

(f) Close autopilot circuit breakers.

(4) Test speed change relay circuit.

(a) Engage autopilot and pull control column to full aft position.

(b) Move the pitch trim wheel in the nose up direction. Check that the time required for the stabilizer to move through 4 units of trim, as indicated on the control stand indicator, is 90 (± 18) seconds.

(c) Extend flaps greater than 3 degrees. Check that the time required for the stabilizer to move through 4 units of trim is 30 (± 6) seconds.

(d) Disengage the autopilot and retract flaps.

(5) Test elevator torque adapter circuit (if applicable). (Refer to Wiring Diagram Manual, 22-1-0, schematic 6.)

(a) Engage autopilot.

(b) Position flaps down 1 degree. Check that autopilot remains engaged.

NOTE: Autopilot disengage warning light B should not come on for more than 2 seconds.

(c) Raise the flaps. Check that the autopilot remains engaged.

(d) Turn off utility hydraulic power.

(e) If no longer required, remove electrical power from airplane.

MACH TRIM SYSTEM - DESCRIPTION AND OPERATION

EFFECTIVITY

AF F-BHSR AND ON
SAB OO-SJF AND ON

1. General

A. The purpose of the Mach Trim System is to position the horizontal stabilizer of the airplane as a function of mach number in order to provide a positive control column force gradient at high mach numbers when the autopilot is disengaged. The system consists of a mach trim coupler, stabilizer position transmitter, stabilizer trim servo, automatic stabilizer trim cutout switch, monitor test switch, resistor box and a warning light. (See figure 1.) The stabilizer trim unit contains a servo motor with two different speed-torque characteristics. The mach trim system drives the high speed motor and the autopilot, when engaged, drives the low speed motor. Refer to autopilot flight control components.

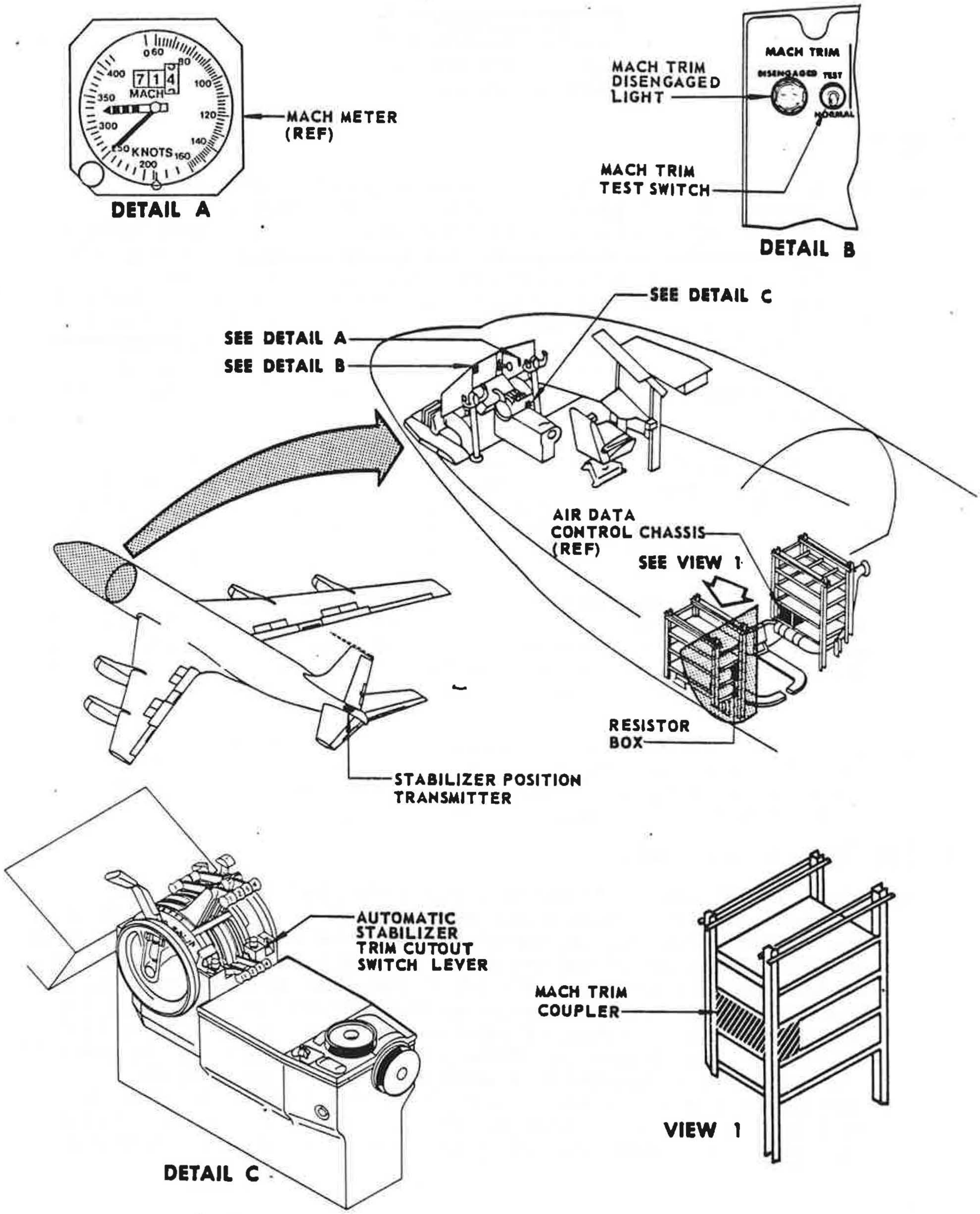
B. It is not possible to use the automatic mach trim system when the autopilot is engaged, and when the stabilizer trim system is electrically driven. The system is designed to function at airplane speeds of mach 0.82 and above. When the above conditions are met, the stabilizer will be positioned automatically as a function of mach number. (See figure 2.)

C. The Mach Trim System receives mach number information from a mach synchrotel transmitter in the copilot's machmeter. This information then goes to the copilot's computer unit in the Integrated Air Data System, which operates the mach switch and mach pot located in the computer unit, to control the mach trim system.

D. The automatic stabilizer trim cutout switch is located within the control stand. When the actuating lever on the top of the control stand is in the "NORMAL" position, the switch is on. In the aft "CUTOUT" position, the switch is off.

2. Mach Trim System Controls

A. Direct manual control of the mach trim system is provided by the automatic stabilizer trim cutout switch. When the switch lever is in the "NORMAL" position, the circuits to the trim servo clutch relay coils are closed, (figure 2) and the system warning light is armed. When one of the relays is energized, 28V d-c is furnished to the power relay and the clutches for either clockwise or counterclockwise rotation of the stabilizer. The direction of rotation depends on which relay is being energized. (See figure 3.) Three-phase power is now being furnished to the fast speed stabilizer trim servo motor. When the automatic stabilizer trim cutout switch lever is in the "CUTOUT" position, the mach trim system is de-energized, the warning light is on to indicate the system is inoperative, and it also opens the autopilot interlock circuit.



Mach Trim System Component Location
Figure 1

B. Simultaneous operation of the main electrical stabilizer trim system and mach trim system is prevented by a manual trim relay in the mach trim coupler. This relay is energized when the stabilizer trim control switch is not operated. When the stabilizer trim control switch is in use, the manual trim relay is not energized, the signal paths to the trim servo clutch relays are opened and the clutch solenoids are shorted.

3. Mach Trim Coupler

A. General

(1) The mach trim coupler contains a signal preamplifier, an electro-mechanical synchronizer, a control amplifier, an auto-trim actuator amplifier, a monitor amplifier and several relays. The coupler acts as the control center for the mach trim system and supplies the necessary control signals to the trim servo. It also contains protective and warning circuits to guard against malfunctions and possible violent maneuvers during system operation.

B. Coupler Power Supply

(1) Operating power for the mach trim system is taken from A and C phases of the flight instrument bus No. 2 in the radio and T-R circuit breaker panel (P5). This power is converted in the mach trim coupler power supply to the voltages necessary for all components of the system as follows:

(a) 5-volts phase A is supplied to the mach signal potentiometer for furnishing control signals to the system.

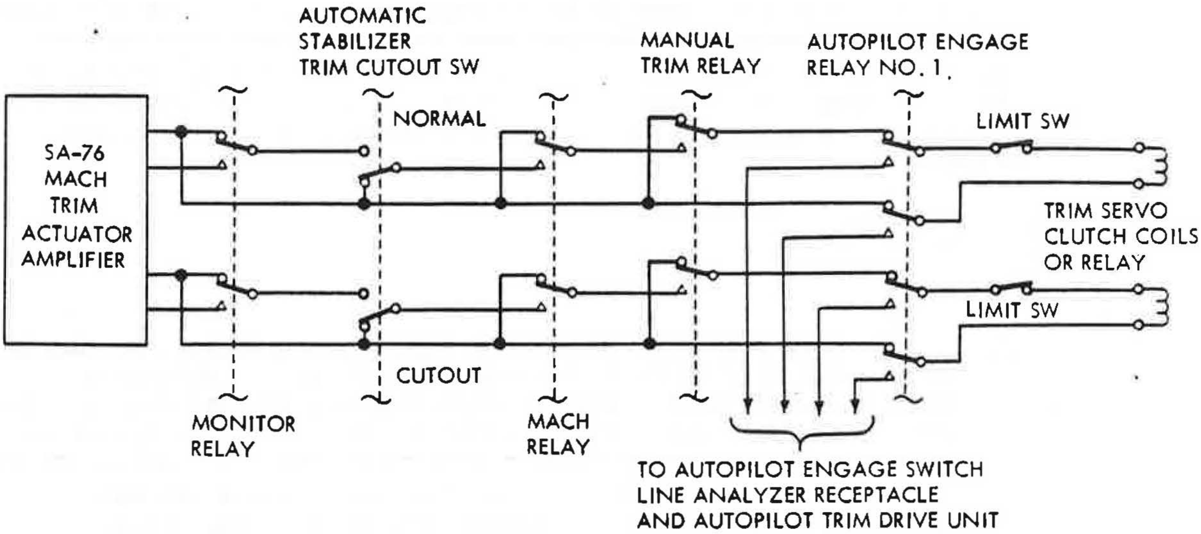
(b) 30 volts phase C, and 15 volts phase A plus 180 degrees, are supplied to energize the electro-mechanical synchronizer card motor.

(c) 19.4 volts phase A and 10.3 volts phase C are combined to supply 26-volts at a phase angle lagging phase A by 20 degrees. This lagging voltage is used to excite the electro-mechanical synchronizer signal synchro and stabilizer trim position transmitter.

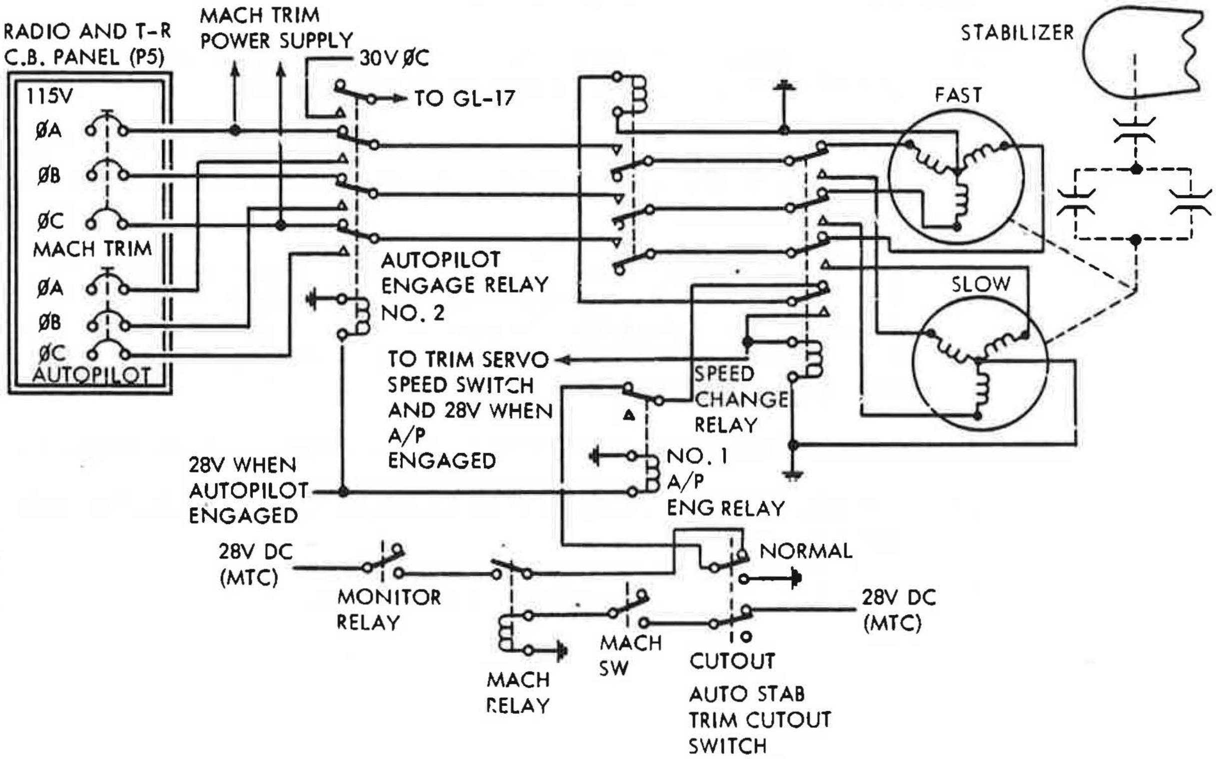
(d) 28-volt d-c power is supplied to all relay control circuits.

(e) 28-volt filtered d-c power is supplied to the amplifier and monitor cards.

(f) 0-volt phase A is supplied as a reference.



Trim Servo Clutch Control Circuit
Figure 2



Trim Servo Power Control Circuit
Figure 3

(2) The mach trim coupler power supply is energized whenever the flight instrument bus No. 2 in the radio and T-R circuit breaker panel (P5) is energized.

C. Autopilot Engage Relays

(1) Two interlock relays prevent simultaneous operation of the mach trim system and autopilot system. These relays are autopilot engage relay No. 1 and autopilot engage relay No. 2. Both relays are energized when the autopilot is engaged. (See figure 3.)

(2) When autopilot engage relay No. 1 is energized control of the trim servo clutches and power relay is transferred from the mach trim actuator amplifier to the autopilot actuator amplifier.

(3) When autopilot engage relay No. 2 is energized operating power for the stabilizer trim servo motor is transferred from the mach trim power source to the autopilot power source, and power is supplied to the electro-mechanical synchronizer.

D. Manual Trim Relay

(1) The manual trim relay may be energized from the autopilot power junction box or the mach trim coupler power supply when the main stabilizer trim relay is not energized. (See Chapter 27.) Diodes prevent current flow between the power supply sources. (See figure 4.) The relay is energized through a thermal delaying circuit which delays the relay closing by 0.75 second.

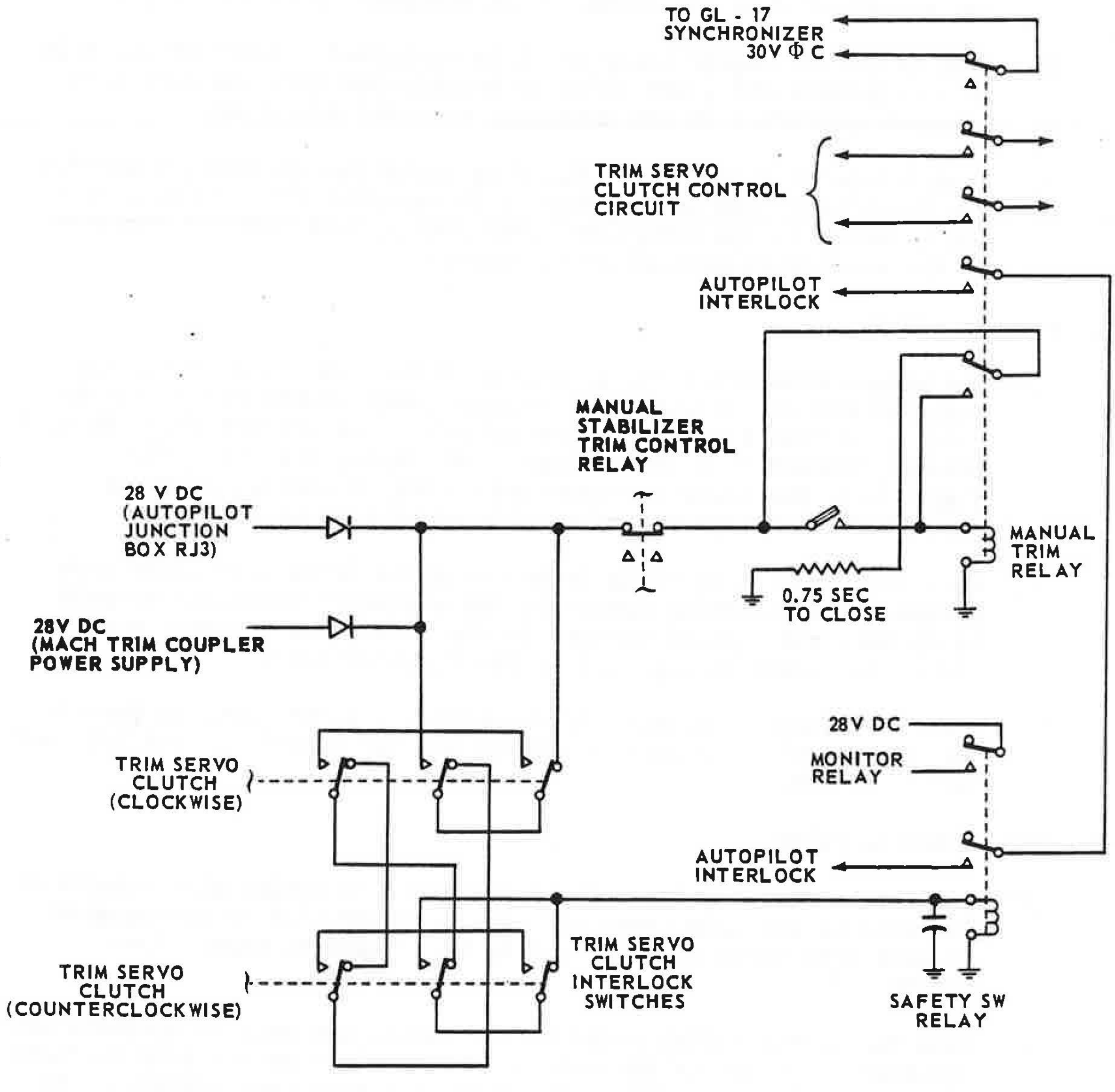
(2) When the manual trim relay is de-energized (main stabilizer trim system is electrically operated) the autopilot interlock circuit is opened, the control circuit to the trim servo clutches are opened and power is supplied to the synchronizer motor.

(3) When the manual trim relay is energized, a closed path is provided for the autopilot interlock circuit and for signals to the trim servo coil clutches.

E. Safety Switch Relay

(1) The safety switch relay coil is connected in series with interlock switches in the trim servo so that the relay will be de-energized if both trim servo clutches should be engaged at once. (See figure 4.)

(2) When the safety switch relay is energized, two sets of contacts are operated. One set of contacts is connected in series with contacts of the manual trim relay in the autopilot interlock circuit. The second set of contacts is connected in series with other relay contacts to complete a circuit to the monitor relay.



Manual Trim and Safety Switch Relay Circuits
Figure 4

F. Monitor Relay

(1) Automatic protection against malfunction is furnished by the monitor amplifier which controls the monitor relay. Contacts on the monitor relay control output of the auto-trim actuator amplifier and control application of power to the trim servo motor power relay and clutches. An additional set of contacts is used in an interlock for the monitor relay coil so that the relay cannot be energized after it has been tripped unless the automatic stabilizer trim switch is turned to "CUTOUT" and then to "NORMAL." (See figure 5.)

(2) When the automatic stabilizer trim cutout switch lever is in the "NORMAL" position and power is applied to the mach trim coupler, the relay will be pulled in by the initial pulse of current through the capacitors, and if the safety switch relay is energized and the monitor amplifier senses proper circuit operation, the monitor relay will stay energized by the power through the closed contact in the monitor amplifier. When the automatic stabilizer trim cutout switch lever is in "CUTOUT" position the monitor relay will stay energized in all conditions.

G. Mach Relay

(1) When the automatic stabilizer trim cutout switch lever is in the "NORMAL" position the mach relay will be energized if the mach switch is closed. (See figure 6.)

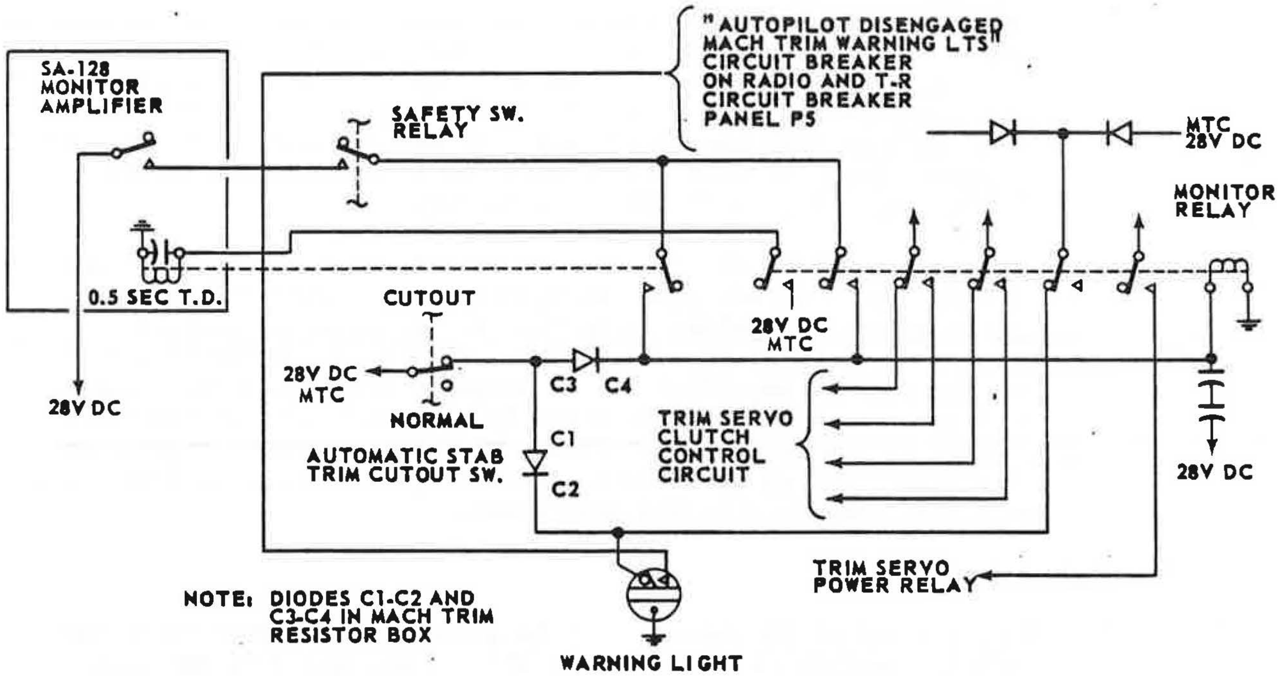
(2) When the mach relay is energized, three sets of contacts complete the control circuit between the auto-trim actuator amplifier and trim servo clutches, and the circuit to the trim servo power relay. When the mach relay is not energized, these circuits are opened.

H. Preamplifier

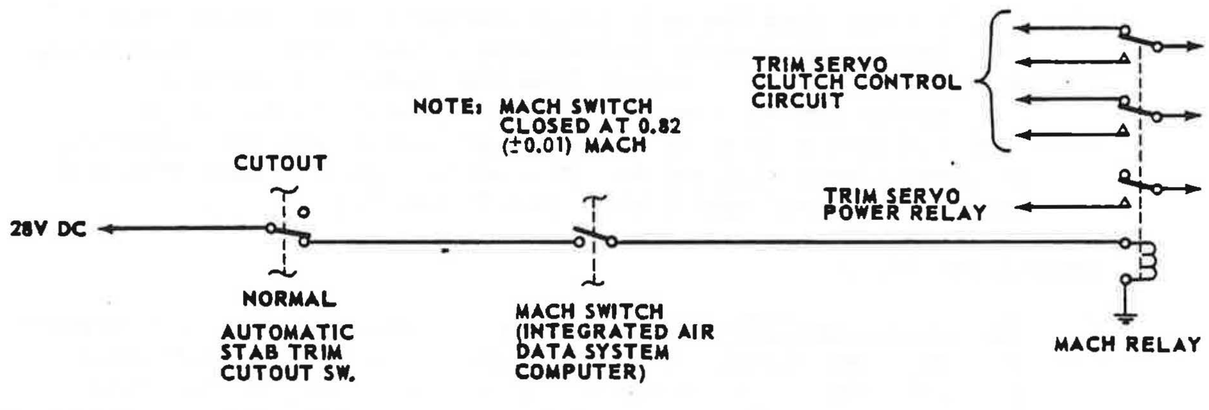
(1) Mach signals from the mach potentiometer in the integrated air data instrument system, synchronizer signals from a synchronizing autosyn and position signals from the stabilizer position transmitter are combined and fed to a preamplifier. If the combined inputs from the three signal sources are not balanced, an error signal will exist. This error signal is amplified and sent to a control amplifier. (See figure 7.)

I. Control Amplifier

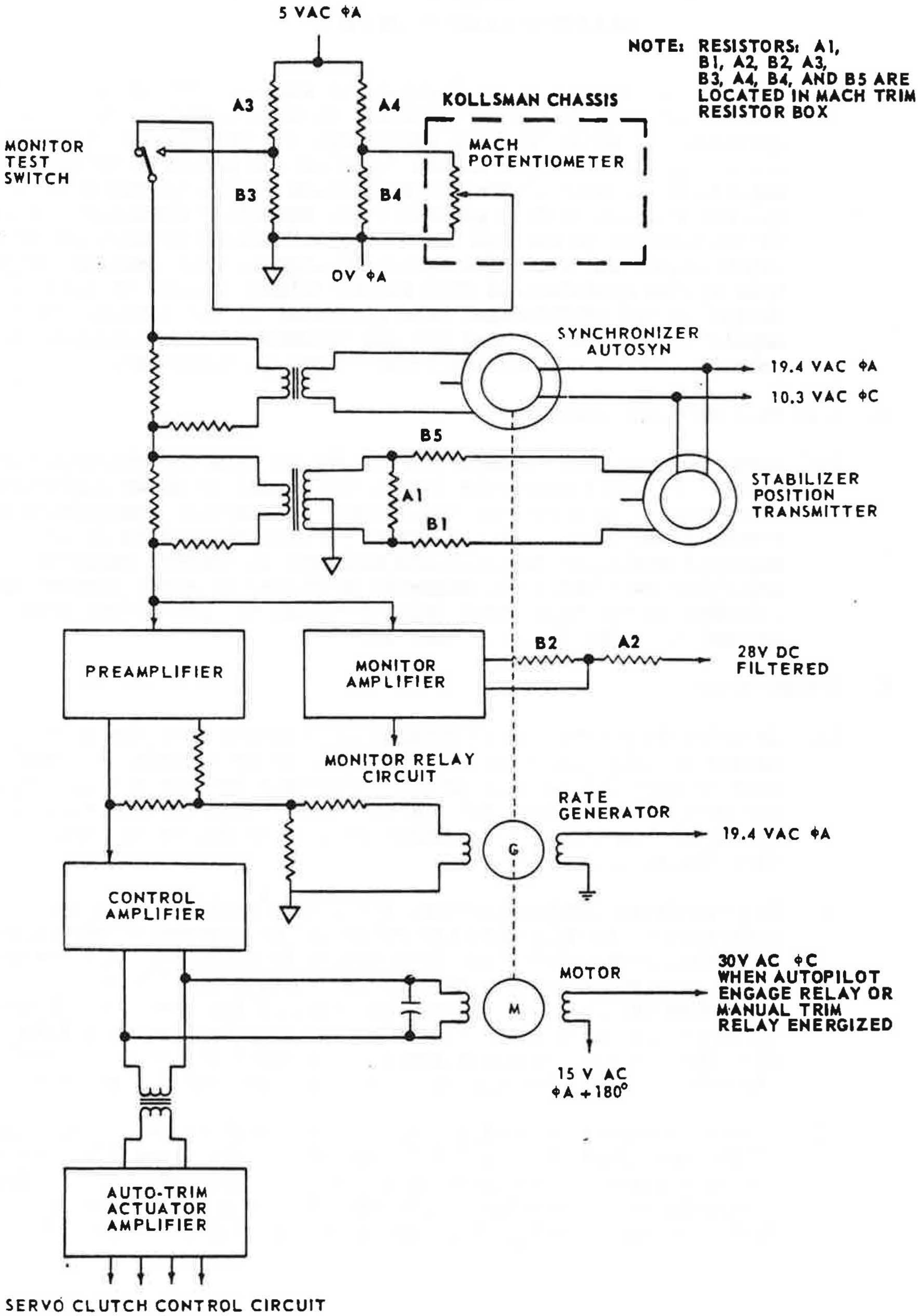
(1) The control amplifier amplifies the a-c error signal, and converts the amplified signal into a d-c signal in a Ramey toroid phase discriminator. The d-c signal is fed to a magnetic amplifier to develop controlled a-c power for operation of the electro-mechanical synchronizer. The control amplifier also provides an a-c signal to an auto-trim actuator amplifier. (See figure 7.)



Monitor Relay and Warning Light Circuit
Figure 5



Mach Relay Circuit
Figure 6



Mach Trim Signal Circuit
Figure 7

(2) Whenever the autopilot is engaged, or whenever the main electrical stabilizer trim system is operated, an electro-mechanical synchronizer motor receives excitation for the reference phase. Error signals developed at the input of the preamplifier are amplified and used by the variable phase of the synchronizer motor. The motor drives a rate generator and autosyn transmitter. Output of the autosyn is used to cancel any difference between the mach input signal and stabilizer position signal, thus keeping the mach trim system synchronized with manual flight control or autopilot. Output of the synchronizer rate generator is fed through the control amplifier and used to damp out any tendency of the synchronizer motor to overshoot when the error signal is cancelled.

J. Auto-Trim Actuator Amplifier

(1) Output of the control amplifier is fed to an auto-trim actuator amplifier. This amplifier limits the signal by diode limiters and amplifies it in a step-up transformer to increase sensitivity and reduce power loss. The a-c signal is then changed to dc in magnetic amplifier demodulators and used to operate magnetic amplifier switches. The magnetic amplifier switches control the clutches in the trim servo which actuates the stabilizer trim mechanism. (See figures 2 and 7.)

K. Synchronizer

(1) In order to prevent rapid changes in attitude when the mach trim system is initially engaged, balancing or synchronizing signals must be supplied to make up the difference between signals from the mach potentiometer and stabilizer position transmitter. The synchronizing signal is supplied by a motor driven autosyn. (See figure 7.)

(2) The two-phase synchronizer motor receives excitation on the reference phase only when the autopilot is engaged or when the main electrical stabilizer trim system is operated. Any error signal appearing at the input of the control amplifier is amplified and fed to the variable phase of the motor, which then operates and turns the synchronizer autosyn to a new position. When the autosyn reaches a position at which its output cancels the error signal, the system is synchronized and the motor stops.

(3) A rate generator is used to prevent synchronizer overshoot. Output of the rate generator is fed through the control amplifier to the variable phase of the synchronizer motor. When the error signal is balanced, rate generator output acts in opposition to motor operation, and rapidly stops the motor to prevent overshoot.

L. Monitor Amplifier

(1) Protection against violent maneuvers or extreme attitudes caused by malfunction of the mach trim system is provided by a monitor amplifier. This amplifier senses the error signal and compares it to a reference signal. If the error signal reaches a predetermined magnitude the monitor amplifier causes the monitor relay to become de-energized and disconnects the mach trim system from the trim servo. A warning light on the engine instrument panel is then energized to inform the pilot that the system is disconnected. The monitor trips if signal voltage exceeds that equivalent to one degree of stabilizer movement. The warning light will also illuminate if the automatic stabilizer trim cutout switch lever is in the "CUTOUT" position. (See figures 5 and 7.)

(2) A one-half second delayed opening relay is in the monitor amplifier to eliminate nuisance disconnect of the system when transient signal difference is presented. The warning light will blink momentarily, but the system is not disconnected. (See figure 5.)

4. Monitor Test Switch

A. The monitor amplifier can be checked during ground operation by a monitor test switch on the engine instrument panel. When the switch is actuated, a fixed error signal is substituted for the mach signal and applied to the mach trim coupler input. The error signal should cause the monitor amplifier to disconnect the mach trim system. The monitor circuit cannot be armed again until the automatic stabilizer trim cutout switch lever is actuated to "CUTOUT" and back to "NORMAL", and the fixed error signal is removed from the amplifier input. (See figure 5 and 7.)

WARNING: DO NOT ACTUATE THE MONITOR TEST SWITCH DURING FLIGHT.

5. Mach Potentiometer and Mach Switch

A. Electrical signals corresponding to mach number are taken from a potentiometer in the air data system. (See figures 6 and 7.) The potentiometer is driven by a servo system controlled by output of a mach transmitter in the machmeter. Refer to Chapter 34, "Integrated Air Data Instrument System."

B. The integrated air data instrument also operates the mach switch. The switch is closed at approximately mach 0.83 and is open at speeds less than mach 0.83. When the switch is closed, the mach relay in the mach trim coupler is energized.

6. Stabilizer Position Transmitter

A. The stabilizer position transmitter is an autosyn which is connected to the stabilizer through a mechanical linkage. The autosyn motor is excited by 26-volt a-c, and induces a voltage in the stator windings. The phase and magnitude of the induced voltage is compared to reference signals to determine direction and magnitude of trim error.

7. Resistor Box

A. The resistor box contains two diodes used in the warning light circuit and fixed resistors used as voltage dividers and resistance bridges to modify error and reference voltages for use by the mach trim system. The resistor box in the lower nose compartment, on the left side at body station 400. (See figure 1.)

8. Mach Trim Warning Light

A. A warning light is provided to indicate that the mach trim system is disengaged. The system is disengaged when the automatic stabilizer trim cutout switch lever is in the "CUTOUT" position or when the monitor relay is de-energized. The warning light is located on the pilot's engine instrument panel. (See figure 1.)

9. Operation

A. During normal flight operation, the mach trim system is on stand-by when the mach trim circuit breaker and the autopilot and mach trim disengage light circuit breaker are closed on radio and T-R circuit breaker panel (P5), and the automatic stabilizer trim cutout switch lever is in "NORMAL" position.

B. The system will start operation when the main electrical stabilizer trim system is not used, the autopilot disengaged, and the mach number is 0.82 or above. When these conditions exist, any change in mach number will cause the horizontal stabilizer to be positioned to correct for the change. For example, if the airplane increases speed because of a nose down attitude the mach trim system will cause the stabilizer to move to a nose up trim position. Conversely, if the mach number decreases the mach trim system will cause the stabilizer to move to a nose down trim position. The system will therefore, automatically adjust the horizontal stabilizer position in accordance with the proper function of mach number such that positive static stability is obtained throughout the airplane speed range. Positive static stability is defined as a requirement for an increasing positive stick force for trim with increasing speed.

C. When the main electrical stabilizer trim system or autopilot is used, the mach trim system is inoperative and the synchronizer is energized to wash out any signal fed back from the stabilizer position transmitter. However, operation of the emergency manual stabilizer trim system will not override the mach trim system and the stabilizer position transmitter feed back signal will not be washed out.

AUTOPILOT SYSTEM - DESCRIPTION AND OPERATION

1. General

The Honeywell autopilot consists of a Digital Amplifier Unit (DAU), Control Panel, Aileron Servo, Elevator Servo, Stabilizer Trim Servo, Power Junction Box, Three Axis Rate Gyro, Three Axis Trim Indicator, Autopilot Adapter, Spoiler Position Transmitters, and the Elevator Torque Adapter. The Digital Amplifier Unit is the control center of the Automatic Flight Control System (AFCS). It provides the computational and logic data required for maintaining the attitude, altitude and heading reference of the airplane. To fly to a selected heading or navigational reference (FMS, TACAN or VOR) or landing reference (ILS). The Control Panel provides the flight crew with controls for the AFCS and adjusting the attitude and heading, and for selecting altitude hold, navigation or landing modes.

The autopilot is characterized by the following salient features:

- It is a high-gain, completely digital system incorporating advanced electronic techniques and modular design. The digital Amplifier Unit and the Control Panel are Line Replaceable Units (LRUs) used on the airplane as a set. They are designed for installation and removal from the airplane without the need of special tools in case of malfunction. All computing is accomplished within the DAU. Airplane power for autopilot use is converted within the system.

WARNING: VOLTAGES EXISTING IN THE AUTOPILOT CIRCUITS ARE HAZARDOUS AND CAN BE FATAL.

- The principle performance characteristics of the autopilot system (DAU) are listed in fig. 1. The system operation provides this performance with one set of parameter adjustments for airplane. At any trimmed attitude of the airplane within its maneuverability limits, the autopilot produces no adverse reactions upon engagement. It causes the airplane to maintain the pitch attitude and heading existing at the time of engagement, except when engagement occurs during a maneuver. Engagement during a maneuver employing bank angles less than 35 degrees, results in a smooth adjustment of airplane roll attitude to give straight, wings-level-flight.

- Interlock monitoring of all essential functions required for safe and reliable system operation provides the needed safety. All interlocks are fail-safe and must be satisfied before the system will engage. The opening of any interlock while the autopilot is engaged will cause system disconnect with pilot warning.

- The autopilot operates in conjunction with the airplane compass, flight instrument, vertical gyro, VOR, TACAN, INS, and ILS navigation systems and one of the two Air Data Display Units selectable with the ADC 1/2 switch on the center instrument panel. The flight instrument system supplies preset heading signal and preset course signal; the navigation receiver system furnishes LOC/VOR deviation signals, localizer frequency switching signal, and the glide slope deviation signal; and the compass system supplies the autopilot with heading reference signal from the directional gyro. TACAN furnishes a lateral deviation signal. INS furnishes a STEER and Valid signal.

STEADY STATE ATTITUDE LIMITS:

(Smooth Air)	Roll	±1	degree
	Pitch	±1/2	degree
	Heading	±1/2	degree
(Moderate Turbulence)	Roll	±2	degrees
	Pitch	±1	degree
	Heading	±2	degrees
(Cyclic Wander)	Roll	±1/2	degree
	Pitch	0	
	Heading	0	

MANEUVERING CONTROL LIMITS:

Bank Angle	(manual & for level flight)	35 (±3) degrees
Bank Angle	(VOR, localizer, glide slope, preset heading)	30 (±3) degrees
Bank Angle	(glide slope after capture)	(1) Changes linearly from 30 (±3) to 10 (±2) degrees from 1500 feet to cut off point when autopilot (DAU) used. (2) Changes linearly from 30 (±3) to 5 (±2) degrees from 1000 feet to 60 feet when radar altimeter used.
Roll Rate	(preset heading, VOR, localizer, glideslope)	4 degrees per second maximum

AUXILIARY CONTROL LIMITS:

Altitude divergence	±50 feet

Principal Performance Characteristics
Fig. 1

- The autopilot attains optimum smoothness and stability of control by the use of error information based not only on the magnitude of the airplane and its control surface displacements from reference positions and angles, but also on the rate of change of these displacements. Optimum sensitivity and accuracy in maintaining reference parameters are attained by integrating the displacement errors with respect to time and by making control surface movement a function of indicated air speed. The following is a brief explanation of these principles of operation:

- Physically, when a sudden change in airplane attitude occurs, a large rate of change of position exists before there is time for a significant displacement actually to take place. By applying control based on this rate of change of position instead of waiting for a significant displacement to build up, the initial control forces are greatest at the inception of the error and consequently are most effective when most needed. Thus, considerable anticipatory corrective action is accomplished before the displacement becomes large and the possibilities of large deviations from reference conditions are greatly reduced. The rate at which these displacement corrections take place is naturally damped by the rate information to assure return to reference conditions with practically no overshoot. When the displacement error is no longer changing, the rate signals fall to zero and the error existing at the instance is acted upon by control forces that are proportional only to the error magnitude. The combination of displacement control and rate control provides the desired smoothness and stability of operation.

- Integration in the autopilot provides an extremely accurate means of automatically retaining reference flight conditions when sustained or recurring displacements from the references caused by wind or loading changes exist. A persistent displacement error, from a light reference, may exist at such a low level that it will not actuate the associated servo channel to cause correction action. Errors of this nature will produce increasing errors in airplane displacement as they are permitted to remain. Small signal errors are integrated against time to build up small displacement errors to usable values so they will correct the error through the associated servo channel. By integrating (summing) a portion of the displacement error signal with respect to the length of time that the displacement persists, another error signal of increasing magnitude is built up. This signal eventually reaches sufficient magnitude to cancel completely the displacement error signal and the airplane is returned to the reference position to be maintained. The integrated signal remains at the value required to overcome the displacement error, compensating for required changes in the original flight references.

- The response of the airplane to control surface movement is aerodynamically a function of the dynamic air pressure. As the dynamic air pressure increases, there is a decrease in the amount of control surface movement required to produce a given change or rate of change in airplane attitude. Therefore, to maintain accurate, automatic control of the airplane at all air speeds, continuous adjustment of the gains of the autopilot surface control channels as a function of dynamic air pressure is provided. Practically, these adjustments are made proportional to indicated air speed which is a function of dynamic air pressure and is easily measured (it is total air pressure, or pitot pressure, minus the ambient or static pressure).

- The locations of individual components of the autopilot system are shown in fig. 2. The principal autopilot operational components and those of associated systems upon which autopilot operation depends, are depicted in fig. 3.

- In its operation the autopilot system senses the following actions of the airplane and its control surfaces:

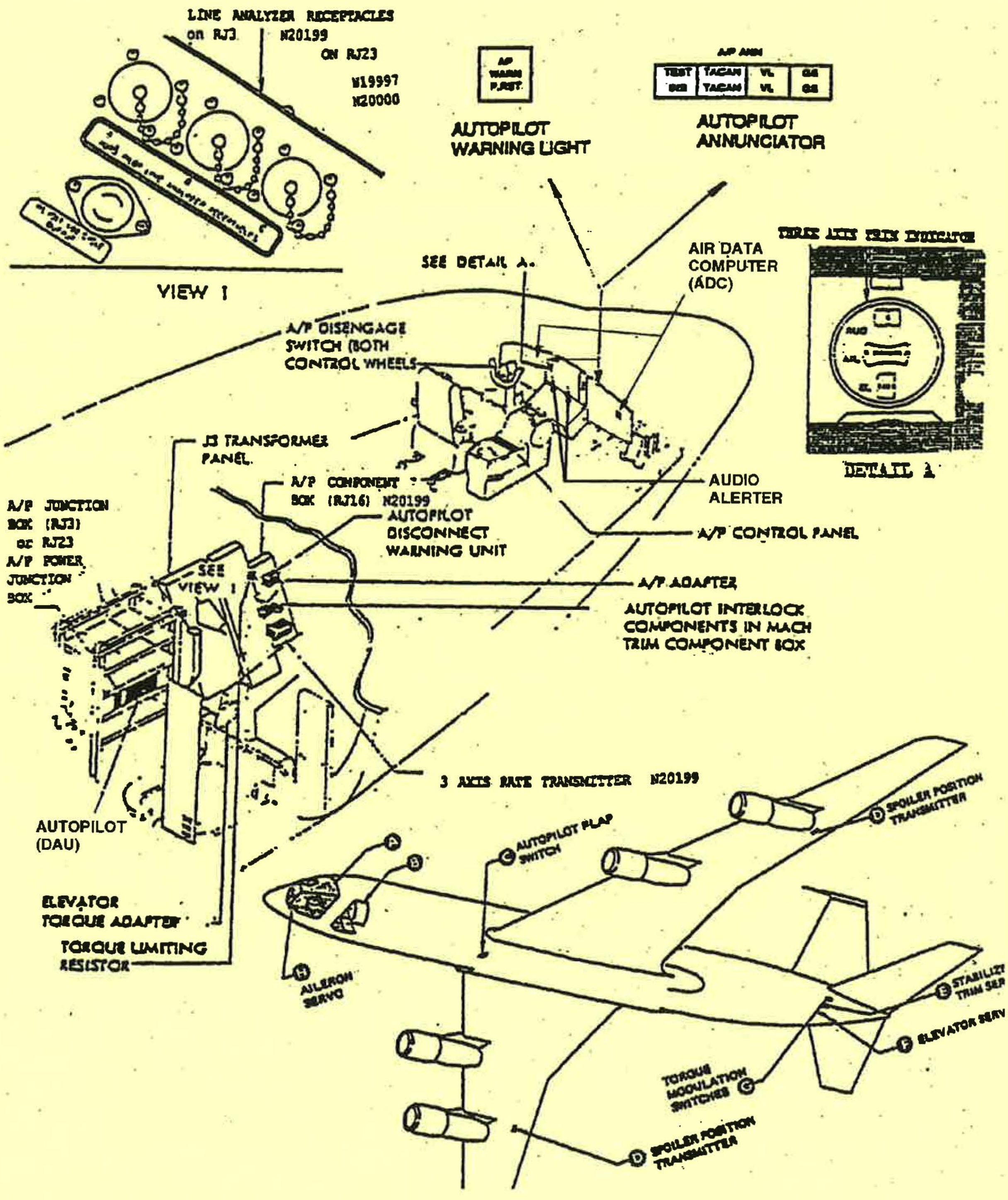
  - The rate of change of airplane pitch and roll attitudes.

  - The displacement of the airplane from the reference bank and pitch attitudes, reference heading and reference altitude.

  - The amount and rate of control surface displacement.

  - The indicated air speed (dynamic air pressure).

IICN: 1P-AMMX-22-0004-01



Autopilot System Component Location
Fig. 2

A/P Component Box (RJ24) for N19997, N20000

- The autopilot is basically a digital computer. The input signals are coming from different systems.

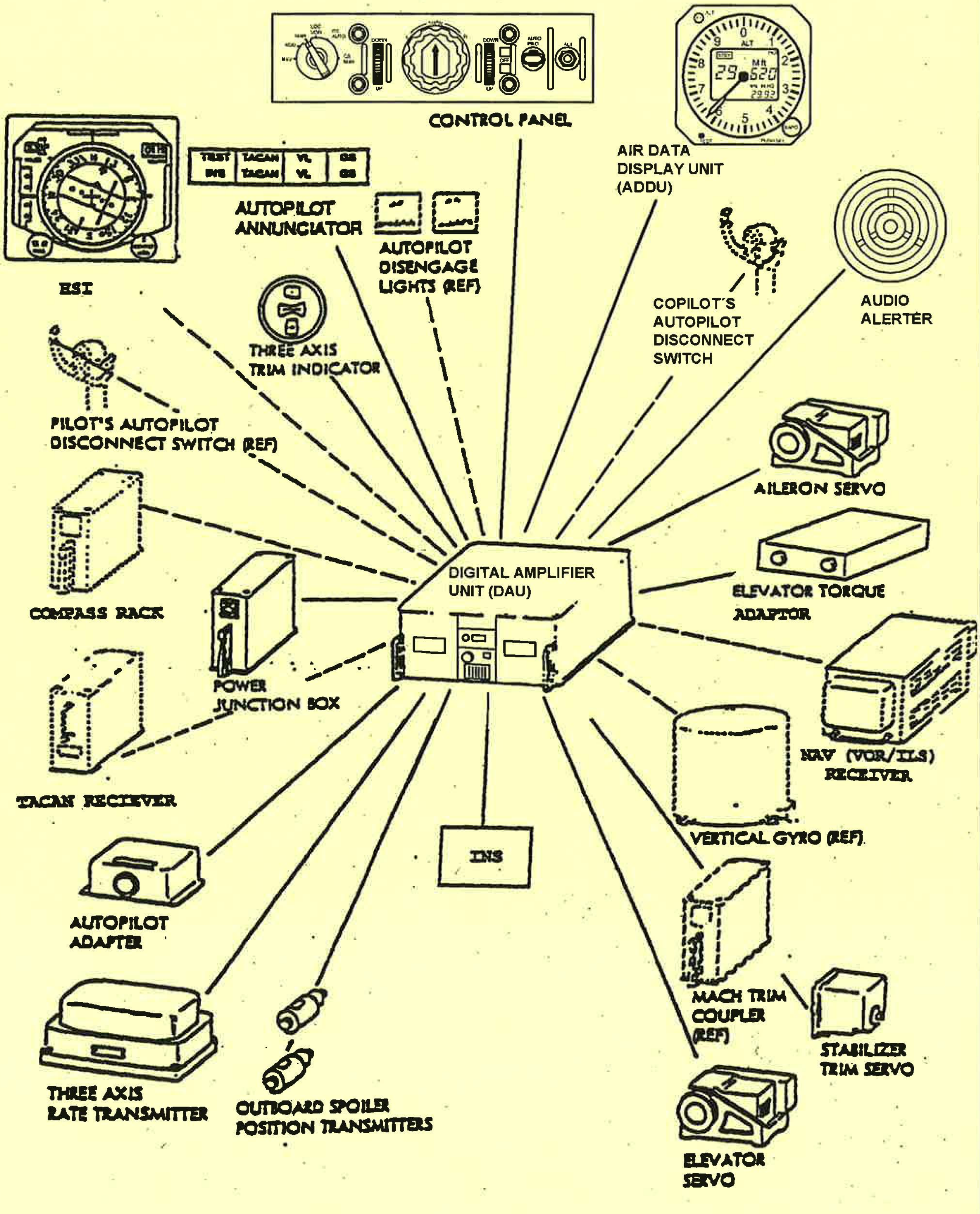
  - Vertical gyro and steering signals are a-c single phase. These voltages are fed directly to the multiplexers in the DAU and then through an A/D Converter and onto the INPUT/ OUTPUT (I/O) BUS. the compass heading signal is a 3-wire synchro signal.

  - Signals from the LOC/VOR/TACAN and G/S radio receivers are low level direct current and may be of either polarity. Direct current signals first pass through a multiplexer and then through the A/D converter and onto the I/O BUS.

  - Altitude error signals will be computed internally based on the ARINC 429 signal from the RVSM system.

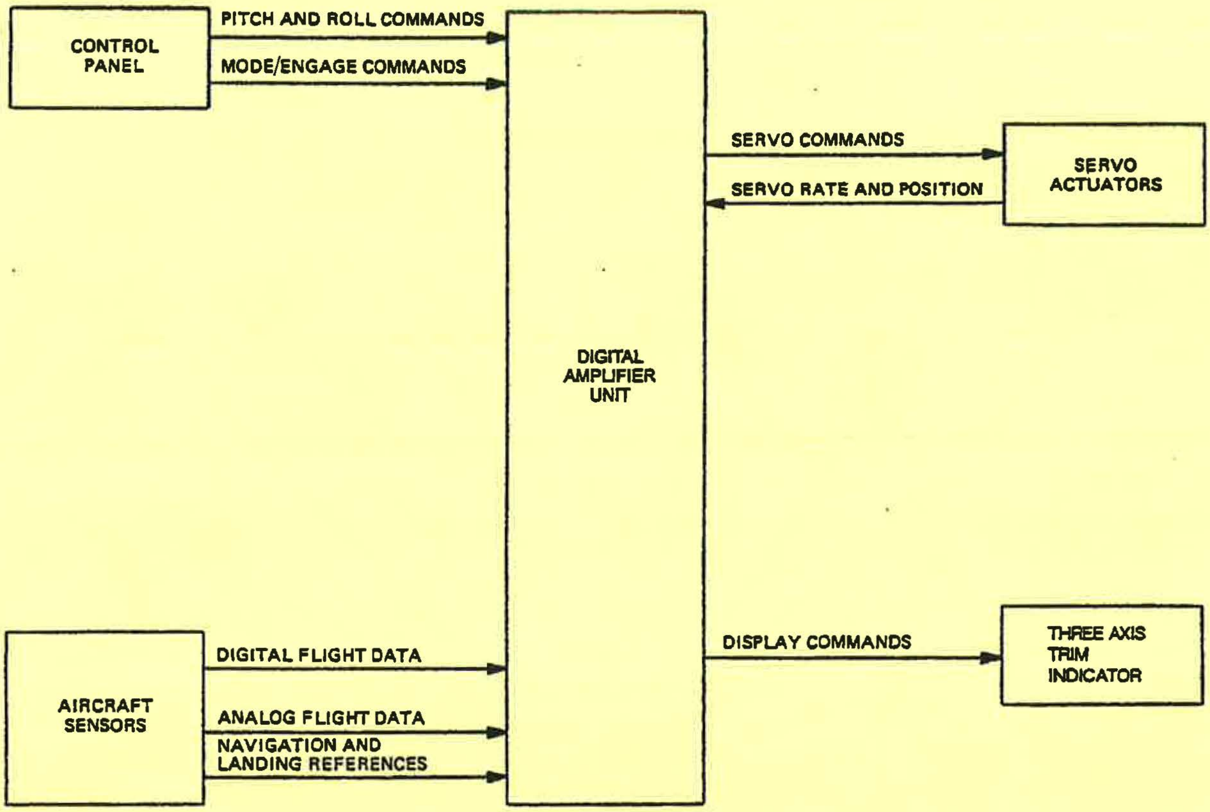
  - The autopilot uses these input signals to produce output voltages which are used by the autopilot servos to actuate the control surfaces (aileron, elevator and stabilizer), thereby reducing the control errors to zero. Basically, each servo is an electric motor which is linked by a gear train, a pulley (or equivalent mechanism) to its associated control system. The gear train increases the motor output torque sufficiently to move the control surface. A functional block diagram of basic autopilot system elements is given in fig. 4. For further information about the operation of the associated control systems refer to respective chapter in the Maintenance Manual.

IICN: 1P-AMMX-22-0003-01



Autopilot System Pictorial Diagram
Fig. 3

IICN: 1P-AMMX-22-0036-01



Autopilot Functional Block Diagram
Fig. 4

2. Operation of Autopilot System

A. System Functions

(1) The individual autopilot components are combined into separate aileron and elevator control channels to provide a system which will completely control the airplane in its roll and pitch axis.

(2) System control functions which may be selected are:

(a) AUTOPILOT

(b) ALT (Altitude hold)

(3) In the AUTOPILOT function for each mode selected, the system control switching circuitry sets up the components required in each control channel to provide proper operation in that mode. The following maneuvering control modes are selectable:

(a) Manual (MAN)- Basic Mode on Engagement

(b) Preset Heading (HDG)

(c) Localizer-omnirange (LOC-VOR / TACAN / INS STEER)

(d) Glide slope-automatic (GS AUTO)

(e) Glide slope-manual (GS MAN)

(f) Navigation (NAV)

NOTE: Navigation (NAV) Mode is disabled.

(4) The altitude hold function provides control in the elevator channel. The system control switching circuitry sets up the components required for proper operation.

B. System Engagement

(1) The system must be energized before it can be engaged in any of its control modes. This is accomplished by closing the radio bus switches and autopilot circuit breakers located on radio and TR circuit breaker panel (P5). Primary a-c, three-phase, 400-cycle voltage is then supplied to the autopilot power junction box. From this, various a-c excitation voltages are applied directly to all autopilot sensors, amplifiers, servos, and position transmitters. D-C operating and reference voltages are also supplied to signal amplifiers and to control circuit switches and

relays. Lack of dc gives fail-safe "no operation." When energized, the entire system is operating, except for various relays, servo and autosyns, which are controlled and become energized only upon engagement of a chosen control function or mode. The system must be energized for at least one minute before it can be engaged. This time is required to erect the vertical gyro and close its autopilot interlock. The vertical gyro must be energized simultaneously with the autopilot system.

(2) When the autopilot system is energized, the following conditions must also be established before the system may be engaged in any of its control modes.

(a) In flight, the airplane must be in trim.

(b) On the ground for service checking, flight controls must be set to neutral.

(c) The Autopilot Engage Switch must be in its off position.

(d) The Altitude Hold Switch must be in its OFF position.

(e) The Mode Selector Switch will automatically be in the MAN position.

(f) The controller TURN Knob must be manually positioned to its center detent.

(g) The Pitch Controller Wheels may be in any position prior to autopilot engagement.

(h) The vertical gyro in use must be energized and erected.

(i) The electrical Manual Trim Switches (one on each control wheel) must be in their center (OFF) position.

(j) The automatic stabilizer trim cutout switch lever must be in the NORMAL position (when installed in interlock chain).

(k) The series yaw damper system must be engaged (ON).

(l) The stabilizer operated elevator servo torque switching system interlock (where installed) must be closed.

(3) If these conditions, plus those necessary for reliable autopilot system operation are fulfilled, all interlocks will be satisfied and the system will remain engaged. The autopilot can then be engaged and any desired mode can be selected.

C. Autopilot Modes

(1) Manual Mode (MAN).

(a) In the manual mode the autopilot provides a means by which the airplane control surfaces are automatically operated to maintain straight and level flight on a predetermined heading or direction. In addition, the system provides for low effort control of the airplane, without use of the control wheel, control column, or rudder pedals. By means of the turn controller knob, located on the Control Panel, the pilot has single knob control of coordinated, blanked turns associated with changes in roll attitude. Maximum bank angle limit using the TURN Knob is 35 degrees. The Pitch-Trim Wheels, also located on the Control Panel, permit manually controlled descents or ascents associated with changes in pitch attitude.

(2) Preset Heading Mode (HDG)

(a) Engagement of the Preset Heading Mode, following initial autopilot engagement, enables the Automatic Flight Control System (AFCS) to accurately track the heading selected with the heading bug on the pilot's HSI. The maximum bank angle limit in the HDG mode is 30 degrees. Maximum roll rate is 4.5 derees per second. Use of the TURN Knob in the HDG mode will result in the system reverting to the MAN mode.

(3) Localizer-omnirange

(a) LOC/VOR

1 With the autopilot Mode Selector in LOC/VOR position, the autopilot automatically guides and stabilizes the airplane to intercept and track a preselected VOR or radio beam. Beam displacement signal from the VOR radio receivers are summed algebraically with course signals from the HSI. The resulting signal provides the autopilot with a lateral command that bank the airplane into a turn towards the preselected VOR radial or localizer radio beam and rolls the airplane out of the bank as the beam center is approached. The autopilot then maintains the airplane centered on the beam with the crab angle required to counteract the effect of prevailing crosswinds.

(b) TACAN

1 In TACAN mode the autopilot automatically guides and stabilizes the airplane to intercept and track a preselected TACAN course. Beam displacement signal from TACAN receivers is summed algebraically with course signals from the HSI. The resulting signal provides the autopilot with a lateral command that bank the airplane into a turn towards the preselected TACAN ra-

dial and rolls the airplane out of the bank as the beam center is approached. The autopilot then maintains the airplane centered on the beam with the crab angle required to counteract the effect of prevailing crosswinds.

(c) INS STEER

1 Engagement of the NAV (INS) function following internal autopilot engagement enables the Automatic Flight Control System (AFCS) to automatically capture and track a specific course previously logged and active on the INS system being selected to drive the autopilot. Engagement of NAV function that the NAV mode selector switch be in the INS position and requires the control panel selector switch positioned to the VOR/LOC position. The maximum bank angle in the NAV (INS) mode is 30 degrees. Maximum roll rate in the NAV mode is 2.3 degrees per second. The autopilot will maneuver the aircraft on the course defined by INS waypoints that have been programmed into the inertial navigation system by the pilot.

(4) Glide Slope Automatic Mode (GS AUTO)

(a) The glide slope (GS) auto mode is selected by tuning the navigation receiver to a localizer frequency, engaging the autopilot and turning the mode select knob to the GS auto position. While the aircraft is outside of the GS capture threshold; the Automatic Flight Control System (AFCS) is in the GS arm mode. The GS auto mode will only engage after localizer capture has occured and the glideslope threshold has been attained. Upon GS capture, A/H mode will be automatically disengaged and pitch wheel commands will be inhibited. The AFCS will track the GS beam. GS gain is reduced as the aircraft descends from 1000 ft to 100 ft as measured by the radar altimeter.

(5) Glide Slope Manual Mode (GS MAN)

(a) The glide slope manual mode provides ILS tracking operation similar to the GS AUTO mode with the difference that transition to glide slope engage is performed manually by operation of the mode selector switch instead of automatically as a function of glide slope beam signal strength through the auto engage amplifier.

D. Altitude Hold (ALT HOLD) Control

(1) With the autopilot engaged in manual, preset heading, or localizer VOR modes, altitude control may be utilized to automatically hold the altitude at which the function was switched on. The ALT HOLD function can also be utilized in the GS AUTO mode, but only until the glide slope beam is engaged. To provide a smooth transition at the time of glide slope intercept, disengagement

of the altitude hold control is delayed for approximately ten seconds during which period the altitude control signal is washed out. This produces a damping function which counteracts the transient attitude changes produced by the sudden introduction of a nose-up glide slope signal (just prior to beam intercept) followed by a nose-down signal (as the beam is intercepted). This damping is especially desirable when the airplane is in the approach configuration, that is, with the flaps down. Under these conditions the elevator servo is switched for maximum torque output so that transient effects are more pronounced.

E. A/P Warning System Operation

(1) The A/P warning system consists of two press-to-test, press-to-reset warning lights and the audio alerter arranged in a circuit such that disengagement of the autopilot due to an open interlock, complete loss of autopilot dc power, or placing the autopilot engage switch to the OFF position causes both lights to flash and the audio alerter will sound; decreased dc power does not impair operation of the autopilot but is indicated by steady illumination of lights and audio alerter which, should an automatic disconnect subsequently occur, then extinguishes. Failure of either flap switch does not impair operation of the autopilot but is indicated by steady illumination of lights and audio alerter. Setting the YAW DAMPER SELF TEST switch out of its OFF position, causes the steady illumination at both A/P WARN LT and the Audio Alerter. The warning lights and audio alert may be extinguished by pressing either the warning light assembly or the autopilot disconnect switch on any control wheel.

END

INTENTIONALLY LEFT BLANK

MACH TRIM SYSTEM - TROUBLE SHOOTING

1. Refer to 22-10-0, Mach Trim System - Trouble Shooting

MACH TRIM SYSTEM - MAINTENANCE PRACTICES

1. Refer to 22-10-0, "Mach Trim System - Adjustment/Test."

AUTOPILOT SYYSTEM - DESCRIPTION AND OPERATION

1. General

A. The autopilot is a Bendix/Eclipse-Pioneer PB-20 Automatic Flight Control System. This autopilot system provides automatic maneuvering of the airplane in a coordinated manner to maintain attitude, altitude and heading references, and provides automatic tracking of VOR, localizer or glide slope beam references when employed with appropriate radio equipment. Yaw axis stability is provided by the series yaw damper system. In the earlier 707 airplanes yaw axis stability is provided by the parallel yaw damper of the autopilot. Some of the earlier installations are being modified to install the series yaw damper.

B. The autopilot is characterized by the following salient features:

(1) It Is a high-gain, completely transistorized system incorporating advanced electronic techniques and modular design. Plug-in units are mounted for easy removal and replacement in case of malfunction. Computing is accomplished by card amplifier and card motor-signal generator assemblies combined in servo, control loops. Data smoothing and synchronizing are performed by similar electromechanical and electronic computing devices. Airplane power is converted for, autopilot use within the system.

WARNING: VOLTAGES EXISTING IN THE AUTOPILOT CIRCUITS ARE HAZARDOUS AND CAN BE FATAL.

(2) The principle performance characteristics afforded by the PB-20 autopilot system are listed in figure 1. The system operation provides this performance with one set of paramter adjustments for all flight conditions throughout the operation range of the airplane. At any trimmed attitude of the airplane within its maneuverability limits, the autopilot produces no adverse reactions upon engagement. It causes the airplane to maintain the pitch attitude and heading existing at the time of engagement, except when engagement occurs during a meneuver. Engagement, during a maneuver employing bank angles less than 35 degrees results in a smooth adjustment of airplane roll attitude to give straight, wings-level flight.

(3) Interlock monitoring of all essential functions required for safe and reliable system operation provides the needed safety. All interlocks are fail-safe and must be satisfied before the system will engage. The opening of any interlock during engagement causes system disconnect with pilot warning.

(4) The autopilot operates in conjunction with the airplane compass, flight instrument, vertical gyro, VOR, TACAN, INS, and ILS naviga tion systems. The flight instrument system supplies preset heading signal and preset course signal the vertical gyro supplies bank, pitch and up-elevator signals; the VOR navigation system furnishes LOC/VOR deviation signal and localizer power switching signal, the ILS navigation system furnishes glide slope deviation signal; and the compass system supplies the autopilot with heading reference signal from the directional gyro on airlines using Sperry compass systems, from the RMI on airlines using the Bendix Continental compass system, and from the compass coupler on airlines using the Bendix Polar Path compass. TACAN furnishes deviation. INS furnishes STEER Signal only.

STEADY STATE ATTITUDE LIMITS :

(Smooth Air)	Roll	± 1 degree
	Pitch	± 1/2 degree
	Heading	± 1/2 degree
(Moderate Turbulence)	Roll	± 2 degrees
	Pitch	± 1 degree
	Heading	± 2 degrees
(Cyclic wander)	Roll	± 1/2 degree
	Pitch	0
	Heading	0

MANEUVERING CONTROL LIMITS :

Bank Angle (manual & for level flight)	35 (± 3) degrees
Bank Angle (VOR, localizer, glide slope, preset heading)	30 (± 3) degrees
Bank Angle (glide slope after capture)	(1) Changes linearly from 30 (± 3) to 10 (± 2) degrees from 1500 feet to cutoff point when air data sensor used. (2) Changes linearly from 30 (± 3) to 5 (± 2) degrees from 1000 feet to 60 feet when radar altimeter used.
Roll Rate (preset heading, VOR, localizer, glide slope)	4 degrees per second maximum

AUXILIARY CONTROL LIMITS:

Altitude divergence	± 50 feet

Principal Performance Characteristics
Figure 1

AMPLIFIER COMPUTERS COVERED IN SECTION 22-42-0	WIRED FOR PALS SYSTEM CONVERSION [1]	RUDDERS CARDS INSTALLED	COMPASS SYNC CARDS INSTALLED	REVISED GLIDE SLOPE DESENSITIZATION REQ	INTERCHANGEABLE WITH
10-3056-65	NO	NO	NO	NO	
10-3056-66	NO	NO	NO	YES	-200 -201
10-3056-67	NO	NO	YES	NO	-150
10-3056-68	NO	NO	YES	YES	-250 -251
10-3056-100 [2]	YES	NO	NO	NO	-101
10-3056-101 [2]	YES	YES	NO	NO	-100
10-3056-150 [3]	YES	NO	YES	NO	-151
10-3056-151 [3]	YES	YES	YES	NO	-150
10-3056-200	YES	NO	NO	NO	- 66 -201
10-3056-201	YES	YES	NO	YES	- 66 -200
10-3056-250	YES	NO	YES	YES	- 68 -251
10-3056-251	YES	YES	YES	YES	- 68 -250

[1] PALS : Precision Approach and Landing System

[2] N20199. Type 18756-1C/-2A/-3B.

[3] N19997, N20000. Type 18762-1C + LH Mod. 06-22-07

Amplifier - Computer Configuration Chart
Figure 1A

INTENTIONALLY LEFT BLANK

(5) The autopilot attains optimum smoothness and stability of control by the use of error information based not only on the magnitude of the airplane and its control surface displacements from reference positions and angles, but also on the rate of change of these displacements. Optimum sensitivity and accuracy in maintaining reference parameters are attained by integrating the displacement errors with respect to time and by making control surface movement a function of indicated air speed. The following is a brief explanation of these principles of operation:

(a) Physically, when a sudden change in airplane or control surface position or angle occurs, a large rate of change of position exists before there is time for a significant displacement actually to take place. By applying control based on this rate of change of position instead of waiting for a significant displacement to build up, the initial control forces are greatest at the inception of the error and consequently are most effective when most needed. Thus, considerable anticipatory corrective action is accomplished before the displacement becomes large and the possibilities of large deviations from reference conditions are greatly reduced. The rate at which these displacement corrections take place is naturally damped by the rate information to assure return to reference conditions with practically no overshoot. When the displacement error is no longer changing, the rate signals fall to zero and the error existing at that instance is acted upon by control forces that are proportional only to the error magnitude. The combination of displacement control and rate control provides the desired smoothness and stability of operation.

(b) Integration in the autopilot provides an extremely accurate means of automatically retaining reference flight conditions when sustained or recurring displacements from the references caused by wind or loading changes exist. A persistent displacement error, from a flight reference, may exist at such a low level that it will not actuate the associated servo channel to cause correction action. Errors of this nature will produce increasing errors in airplane displacement as they are permitted to remain. Small signal errors are integrated against time to build up small displacement errors to usable values so they will correct the error through the associated servo channel. By integrating (summing) a portion of the displacement error signal with respect to the length of time that the displacement persists, another error signal of increasing magnitude is built up. This signal eventually reaches sufficient magnitude to cancel completely the displacement error signal and the airplane is returned to the reference position to be maintained. The integrated signal remains at the value required to overcome the displacement error, compensating for required changes in the original flight references.

(c) The response of the airplane to control surface movement is aerodynamically a function of the dynamic air pressure. As the dynamic air pressure increases there is a decrease in the amount of control surface movement required to produce a given change or rate of change in airplane attitude. Therefore, to maintain accurate, automatic control of the airplane at all air speeds, continuous adjustment of the gains of the autopilot surface control channels as a function of dynamic air pressure is provided. Practically, these adjustments are made proportional to indicated air speed which is a function of dynamic air pressure and is easily measured (it is total air pressure, or pitot pressure, minus the ambient or static pressure).

C. The locations of individual components of the autopilot system are shown in figure 2. The principal autopilot operational components and those of associated systems upon which autopilot operation depends are depicted in figure 3.

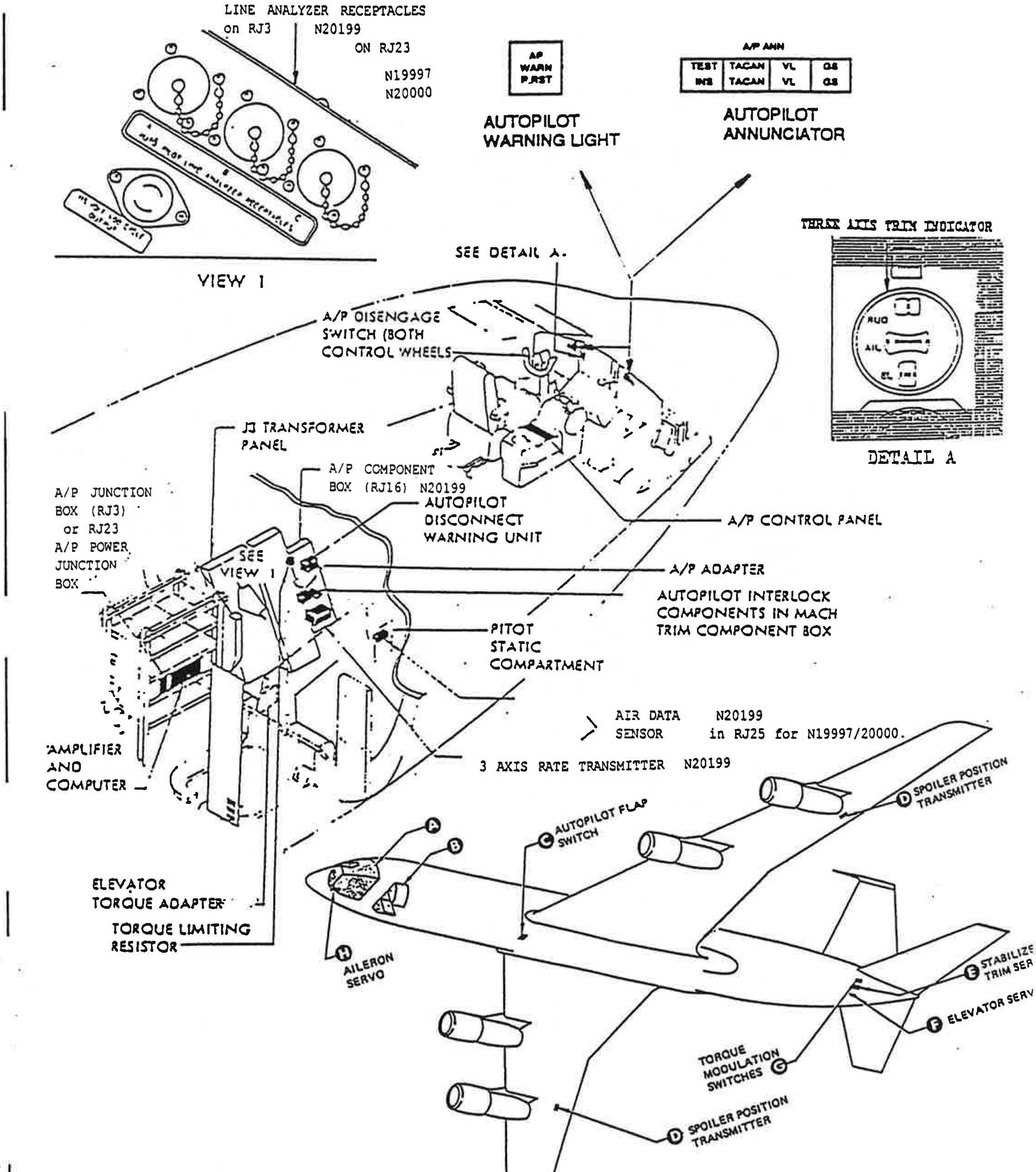
D. In its operation the autopilot system senses the following actions of the airplane and its control surfaces:

(1) The rate of change of airplane pitch and roll attitudes.

(2) The displacement of the airplane from the reference bank and pitch attitudes, reference heading and reference altitude.

(3) The amount and rate of control surface displacement.

(4) The indicated air speed (dynamic air pressure).



Autopilot System Component Location
Figure 2

A/P COMPONENT BOX (RJ24) for N19997, N20000.

E. The sensed data is in the form of autosyn synchro voltages which are compared with command voltages to develop control information. Control information is derived by circuitry consisting of mechanical linkages, electro-mechanical synchronizers and integrators (or electronic synchronizers), transistorized amplifiers and magnetic power amplifiers. The control information is generated in the form of autosyn displacement-error voltages and angular error voltages and time-integrated displacement and angular error voltages. These error voltages are appropriately combined and amplified to produce output voltages which are used by the autopilot servos to actuate the control surfaces (ailerons, elevators and stabilizer), thereby reducing the control errors to zero. Basically, each servo is an electric motor which is linked by a gear train and pulley (or equivalent mechanism) to its associated control system. The gear train increases the motor output torque sufficiently to move the control surface. A functional block diagram of basic autopilot system elements is given in figure 4. Refer to Chapter 27, "Flight Controls," for a discussion of the operation of the associated control systems.

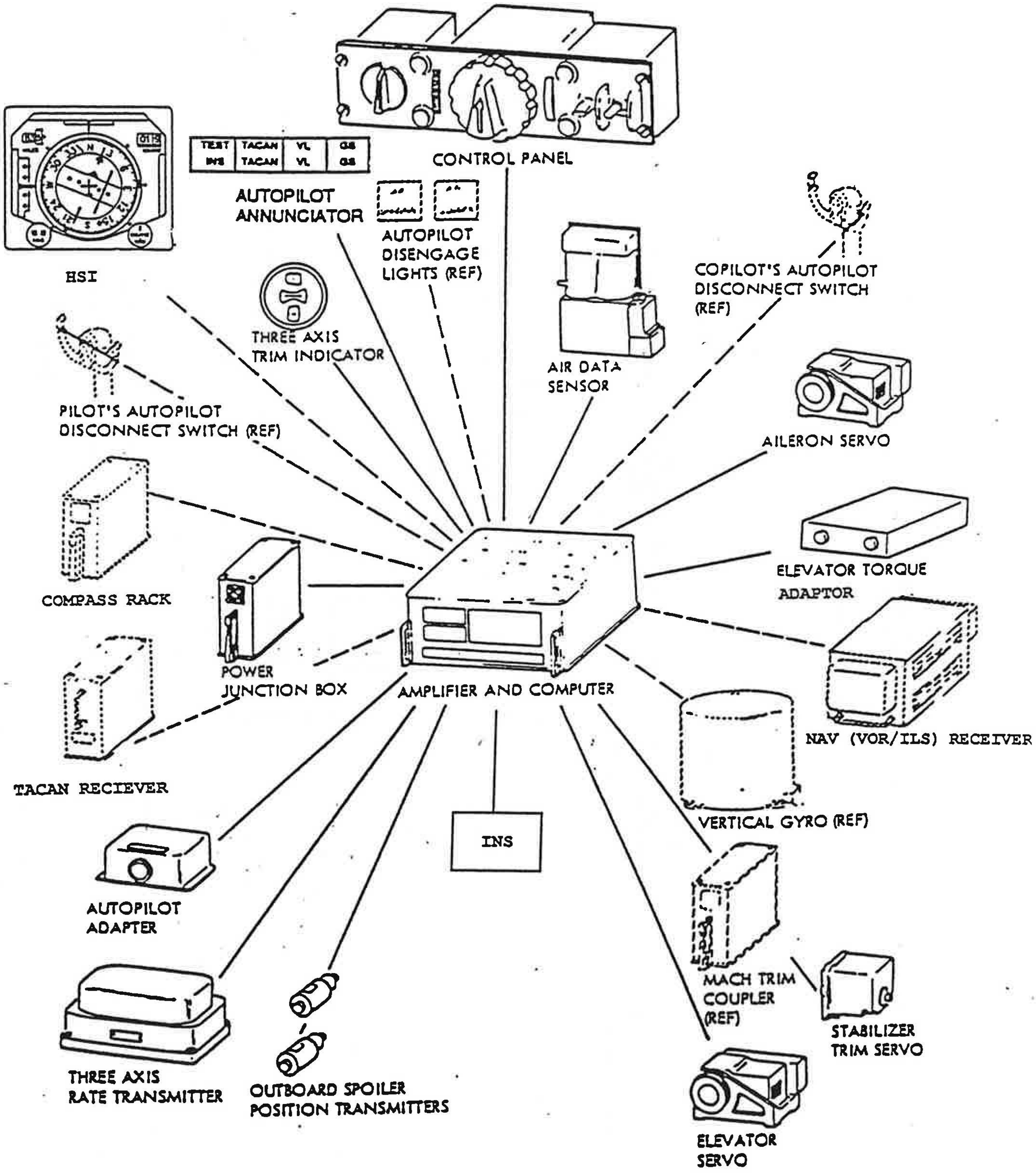
(1) The autopilot is basically a form of analogue computer. The signal sources may be either a-c or d-c voltages. In order that a form of intelligence may be used by the autopilot in the amplifier and computer it must be alternating current single phase and may be in phase or 180° out of phase.

(a) Vertical gyro and compass heading signals are a-c single phase and require no conversion. These voltages are fed directly to the mixing chain in the amplifier and computer after passing through gain controls.

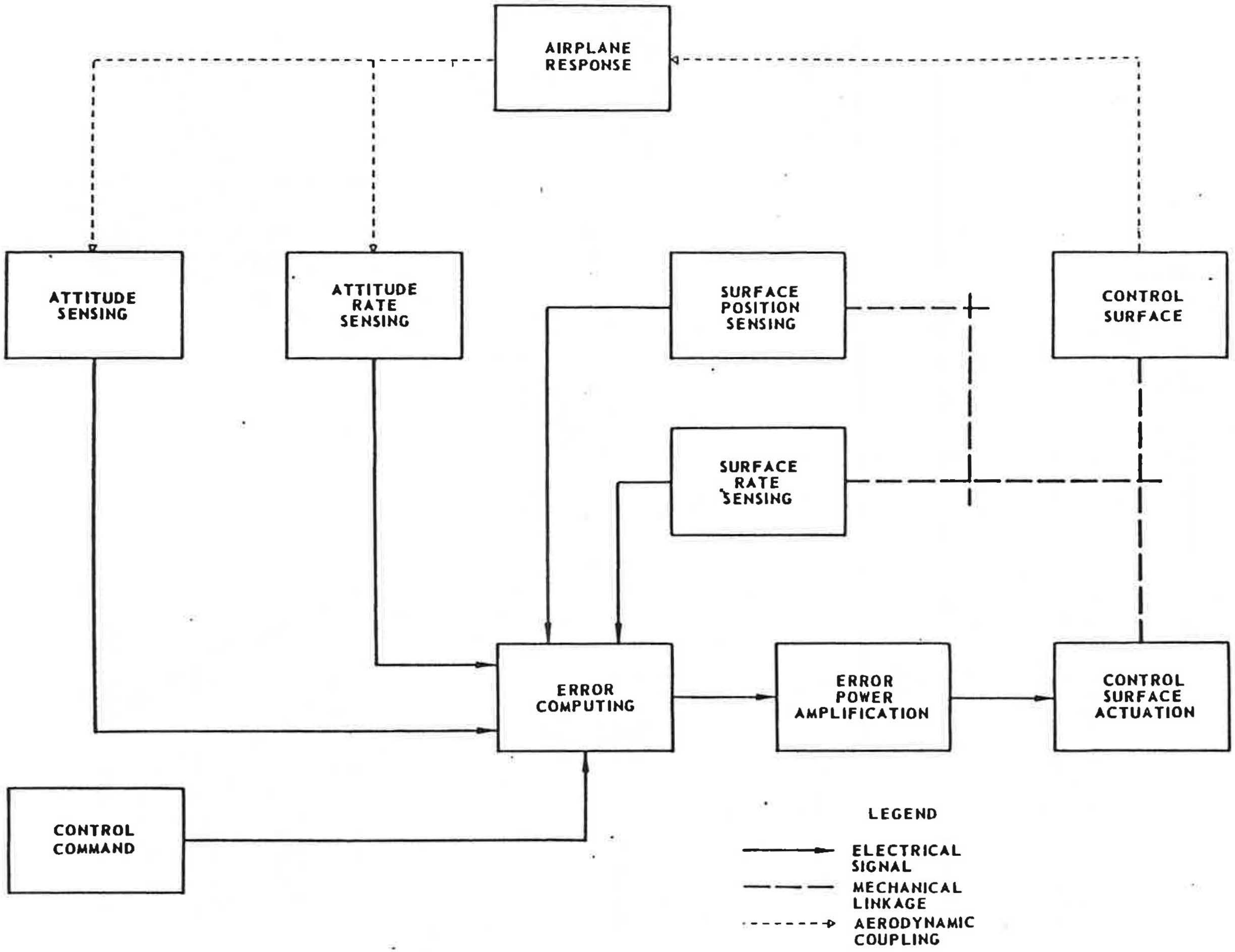
(b) Signals from the LOC/VOR and G/S radio receivers are direct current and may be of either polarity. Direct current signals first pass through a modulator section where they control the phase of the output of a phase discriminator. After suitable gain control and limiting, this resultant a-c signal is injected into the computing chain of the amplifier and computer.

(c) Altitude error signals from the air data sensor are directly usable in the computer after suitable gain control in the same manner as the other a-c signals.

(d) The amplifier and computer incorporates an external plug-in adapter. The adapter, which contains impedance matching networks to couple and match response signals of the airplane to the amplifier and computer, is a rectangular box which fits in the lower front section of the amplifier and computer. The use of the proper adapter makes it possible to interchange the amplifier and computer between different types of airplanes using the PB-20 autopilot system.



Autopilot System Pictorial Diagram
Figure 3



Autopilot Functional Block Diagram
Figure 4

2. Operation of Autopilot System

A. System Functions

(1) The individual autopilot components are combined into separate aileron and elevator control channels to provide a system which will completely control the airplane in its roll and pitch axis.

(2) The specific functions and control modes made available in the system operation are determined by manually setting the proper control switches on the autopilot control panel. System control functions which may be selected are:

(a) AUTOPILOT

(b) ALT (Altitude hold)

(3) In the AUTOPILOT function for each mode selected, the system control switching circuitry sets up the components required in each control channel to provide proper operation in that mode. The following maneuvering control modes are selectable:

(a) Manual (MAN)

(b) Preset Heading (HDG)

(c) Localizer-omnirange (LOC-VOR)

(d) Glide slope-automatic (GS AUTO)

(e) Glide slope-manual (GS MAN)

(4) The altitude hold function provides control in the elevator channel. The system control switching circuitry sets up the components required for proper operation.

B. System Engagement

(1) The system must be energized before it can be engaged in any of its control modes. This is accomplished by closing the radio bus switches and autopilot circuit breakers located on radio and TR circuit breaker panel (P5). Primary a-c, three-phase, 400-cycle voltage in this way is supplied to the autopilot power junction box. From this, various a-c excitation voltages are applied directly to

all autopilot sensors, amplifiers, servos, and position transmitters. Also, d-c operating and reference voltages are supplied to signal amplifiers and to control circuit switches and relays. Lack of dc gives fail-safe "no operation." When energized the entire system is operating, except for various relays and the servo, autosyn and gain potentiometer clutches which they control and which become energized only upon engagement of a chosen control function or mode. The system must be energized for at least one minute before it can be engaged. (This time is required to erect the airplane vertical gyro and close its autopilot interlock. The vertical gyro must be energized simultaneously with the autopilot system.)

(2) When the autopilot system is energized, the following conditions must also be established before the system may be engaged in any of its control modes.

(a) In flight, the airplane must be in trim.

(b) On the ground for service checking, flight controls must be set at neutral.

(c) The autopilot engage switch must be in AUTOPILOT.

(d) The altitude hold switch must be in its off position.

(e) The mode selector switch will automatically be in the MAN position.

(f) The controller turn knob must be manually positioned to its center detent.

(g) The pitch controller wheels may be in any position prior to autopilot engagement.

(h) The vertical gyro in use must be energized and erected.

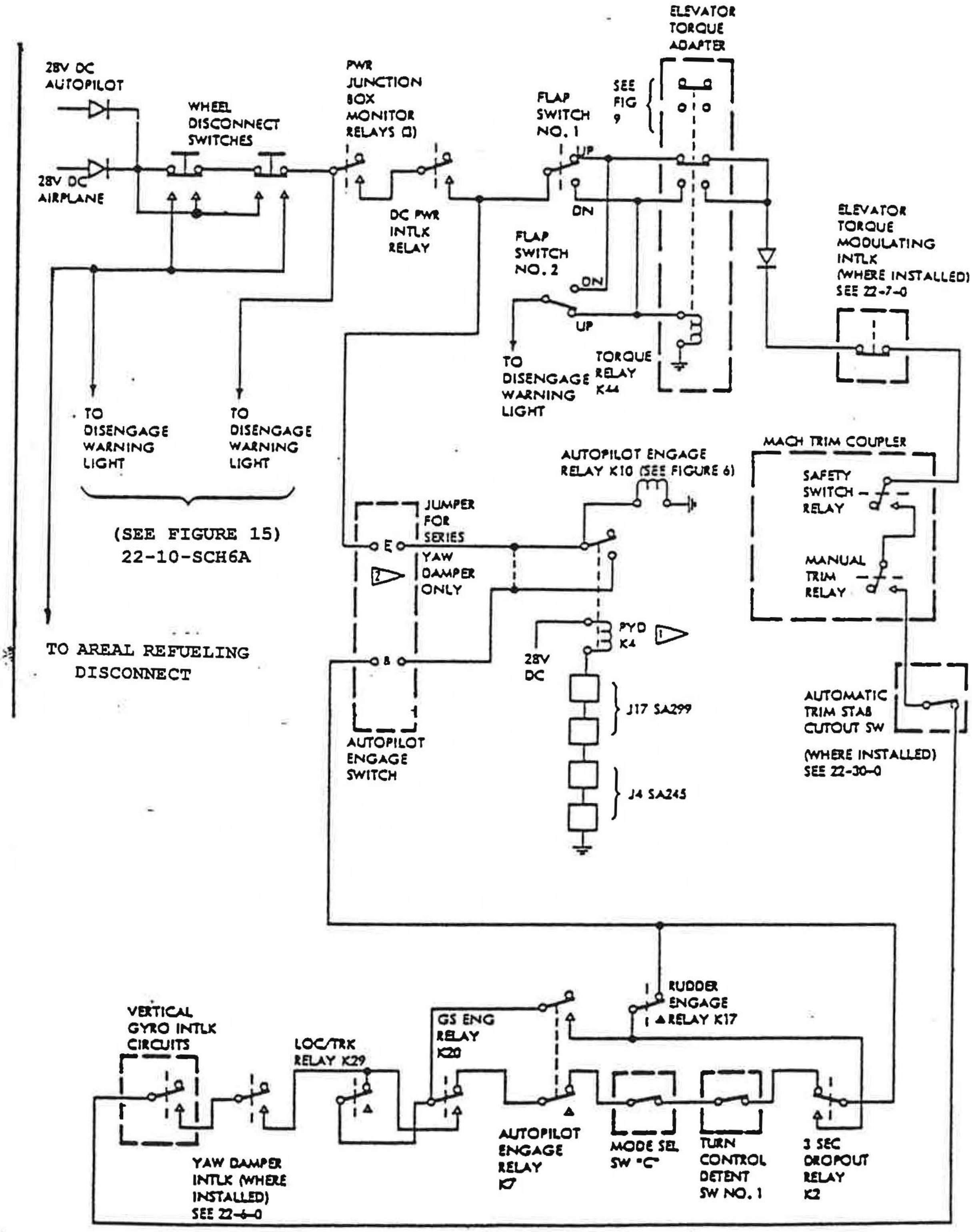
(i) The electrical manual trim switches (one on each control wheel) must be in their center (off) position.

(j) The automatic stabilizer trim cutout switch lever must be in the NORMAL position (when installed in interlock chain).

(k) The series yaw damper system (when installed in interlock chain) must be engaged (ON).

(l) The stabilizer operated elevator servo torque switching system interlock (where installed) must be closed.

(3) If these conditions, plus those necessary for reliable autopilot system operation are fulfilled, all interlocks will be satisfied and the system will remain engaged. The autopilot can then be engaged in any desired mode. (See figure 5.)



[1] PROVISIONS FOR PARALLEL YAW DAMPER WHEN RUDDER SERVO CARDS INSTALLED

[2] CONTACT E CLOSED FOR PARALLEL YAW DAMPER OPERATION ONLY

Autopilot Engage Interlock Circuit
Figure 5

C. Autopilot Modes

(1) Manual Control.

(a) With all interlock requirements satisfied, placing the autopilot engage switch to the AUTOPILOT position engages the autopilot in the MANUAL mode. Contact C of the spring-loaded mode selector switch is closed only while the switch is in the manual (MAN) position. (See figure 5.) Thus initial autopilot engagement can only be effected with the mode selector switch set to manual (MAN).

(b) In the manual mode the autopilot provides a means by which the aircraft control surfaces axe automatically operated to maintain straight and level flight on a predetermined heading or direction. In addition, the system provides for low effort control of the aircraft without use of the control wheel, control column, or rudder pedals. By means of the turn controller knob, located on the control panel, the pilot has single knob control of coordinated, banked turns and changes in roll attitude. The pitch-trim, wheels, also located on the control panel, permit manually controlled dives or climbs and changes in pitch attitude.

(c) Prior to engagement the autopilot continually monitors the aircraft attitude in pitch and yaw. This results in the autopilot, after engagement, maintaining the pitch attitude and heading which existed immediately prior to autopilot engagement.

(d) With the autopilot engage switch set to AUTOPILOT control switching circuits establish the circuit configuration for manual mode operation. D-C power is applied to the coil of autopilot engage relay K10 through closed contact B of the autopilot engage switch. Normally-open contacts of K10, in turn, apply power through closed contact A of the engage switch to the aileron and elevator servo clutch solenoids, and to the coils of autopilot engage relay K7 and autopilot engage relays K5 and K6 in the mach trim coupler. Operation of K10 also connects d-c power through closed contact F of the engage switch to rudder engage relay K3, rudder engage thermal relay K16, and warning light relay K1 In the autopilot disengage warning unit.

(e) The aileron and elevator clutch solenoids engage the aileron and elevator control tab surfaces to their respective servomotors. Rudder engage relay K3 closes to complete the d-c circuit to the autopilot engage switch holding solenoid.

Rudder engage thermal relay K16 closes one-half second later to energize rudder engage relay K17. Paralleled normally-open contacts of rudder engage relays K3 and K17 arm the Bendix autopilot disengage warning light circuit.

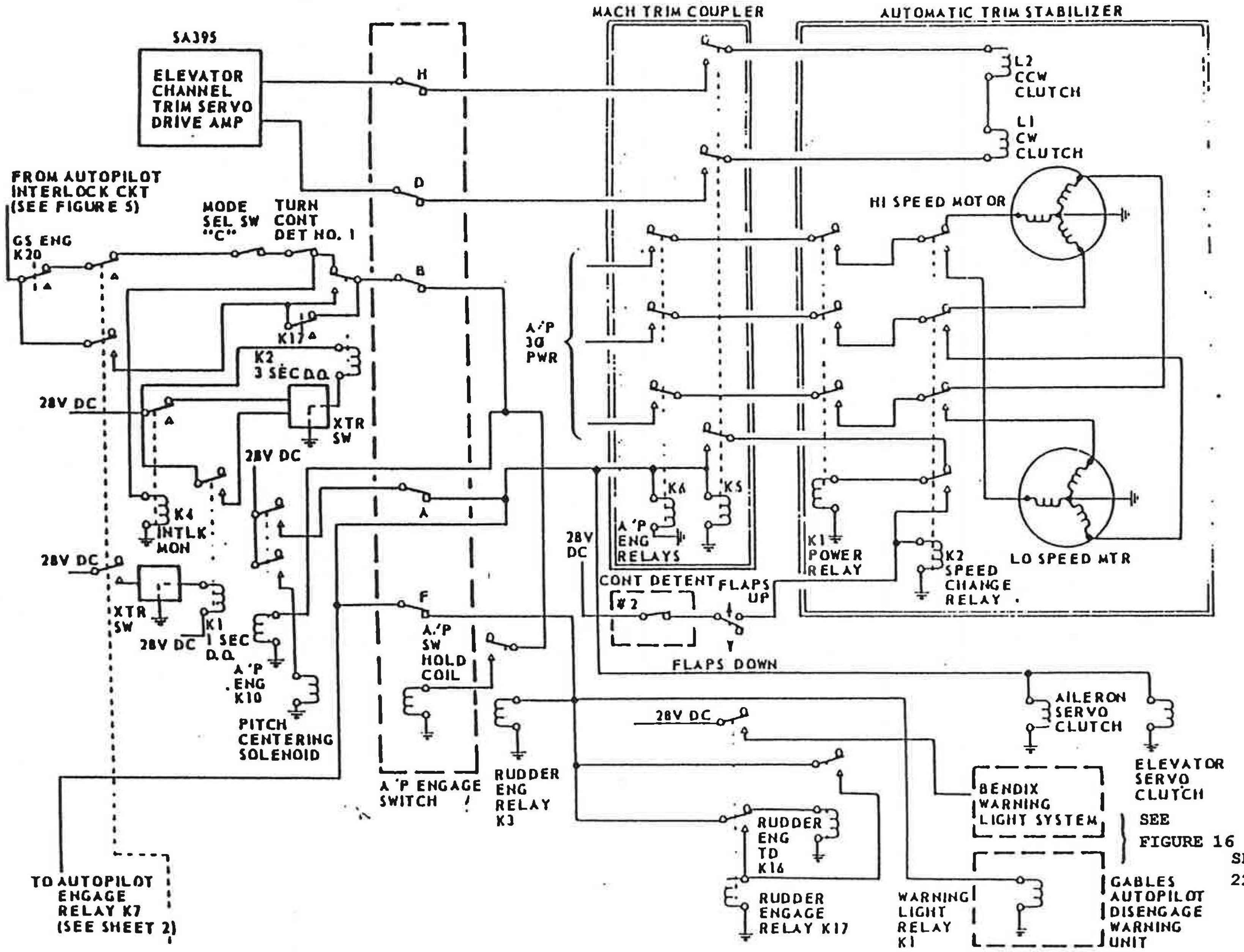
(f) Autopilot engage relay K7, in the energized condition performs the following functions:

1) Rearranges the autopilot interlock circuit (in conjunction with 3-second drop-out relay K2) to bypass contact C of the mode selector switch and detent switch No. 1 of the turn controller. (See figure 5) The autopilot can now be engaged in any other desired mode by mean of the mode selector switch and the turn controller can be used to manunlly execute coordinated turns.

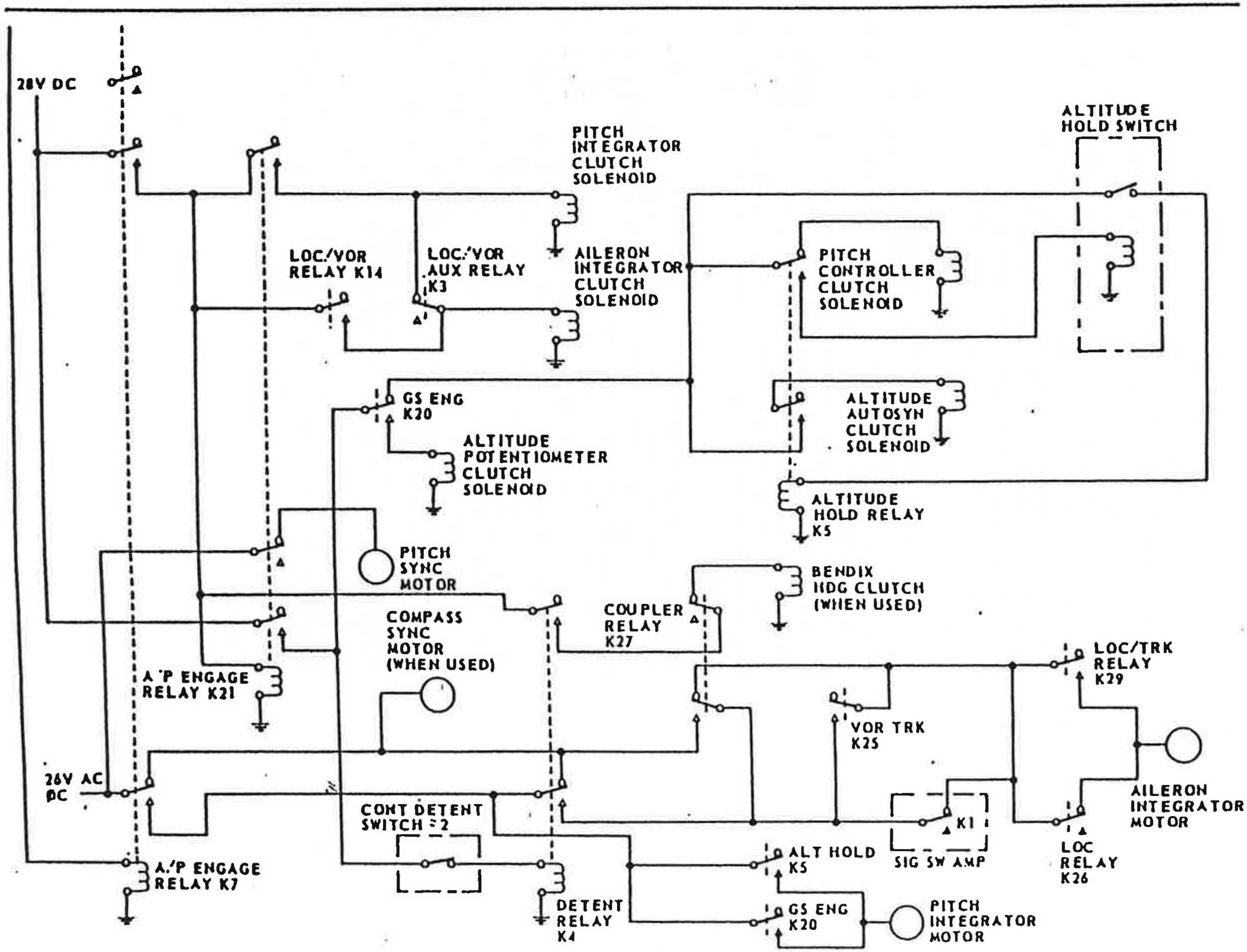
2) Removes power from interlock monitor relay K4. (See figure 6.) This action causes time-delay drop-out relay K2 to energize. The autopilot interlock circuit is now completed through the pulled-in contacts of K2. In the event of autopilot disengagement, subsequent re-engagement must be delayed for approximately four seconds until K2 releases. This allows time for all channels to synchronize before system re-engagement.

3) Removes 28 V a-c excitation power from the compass synchronizer motor on airplanes using Sperry compass system, which prior to autopilot engagement and in modes, other than manual, continually positions its autosyn in synchronism with the compass heading to give zero autosyn output. With the motor de-energized the synchronizer autosyn produces heading error signals that are proportional to heading deviations of the aircraft.

4) Completes the 26-volt a-c excitation circuit to the aileron integrator motor by energizing autopilot engeage relay K21 and in turn, through controller detent switch No. 2, detent relay K4. The 26-volt a-c power is now routed to the aileron integrator motor through relaxed contacts of signal switch amplifier relay K1 and LOC engage relay K26. Another set of K21 contacts completes the circuit to the aileron integrator clutch solenoid. The alleron integrator is now activated and provides correction signals to the aileron signal chain in order to oppose long-term steady-state compass heading errors.



Manual Mode Control Switching Circuits
Figure 6 (Sheet 1)



Manual Mode Control Switching Circuits
Figure 6 (Sheet 2)

(g) Normally-closed contacts of autopilot engage relay K21 open to remove the fixed-phase excitation voltage from the pitch synchronizer motor which operates prior to autopilot engagement to enable the aircraft to maintain the pitch angle existing at the time of engagement. Other normally-open contacts of K21 close to energize the pitch intergrator clutch solenoid and arm the altitude hold switch. Pitch integration, however, is employed only in the glide slope and altitude hold operating modes. The pitch controller clutch links the controller pitch trim wheels to the pitch potentiemeters so that manaul adjustment of pitch attitude is my available to the pilot.

(h) On airplanes using the Bendix compass system instead of the Sperry compass synchronizer a set of normally-open contacts an detent relay K4 closes and connects d-c power (through normally-closed contacts of coupler relay K27) to the Bendix heading clutch solenoid. This clutch engages an autosyn which provides a compass heading signal directly to the autopilot, making a compass synchronizer unnecessary. Prior to engagement or during control in modes other then manual, the heading clutch is de-energized and the autosyn is nulled by centering levers tc give zero output.

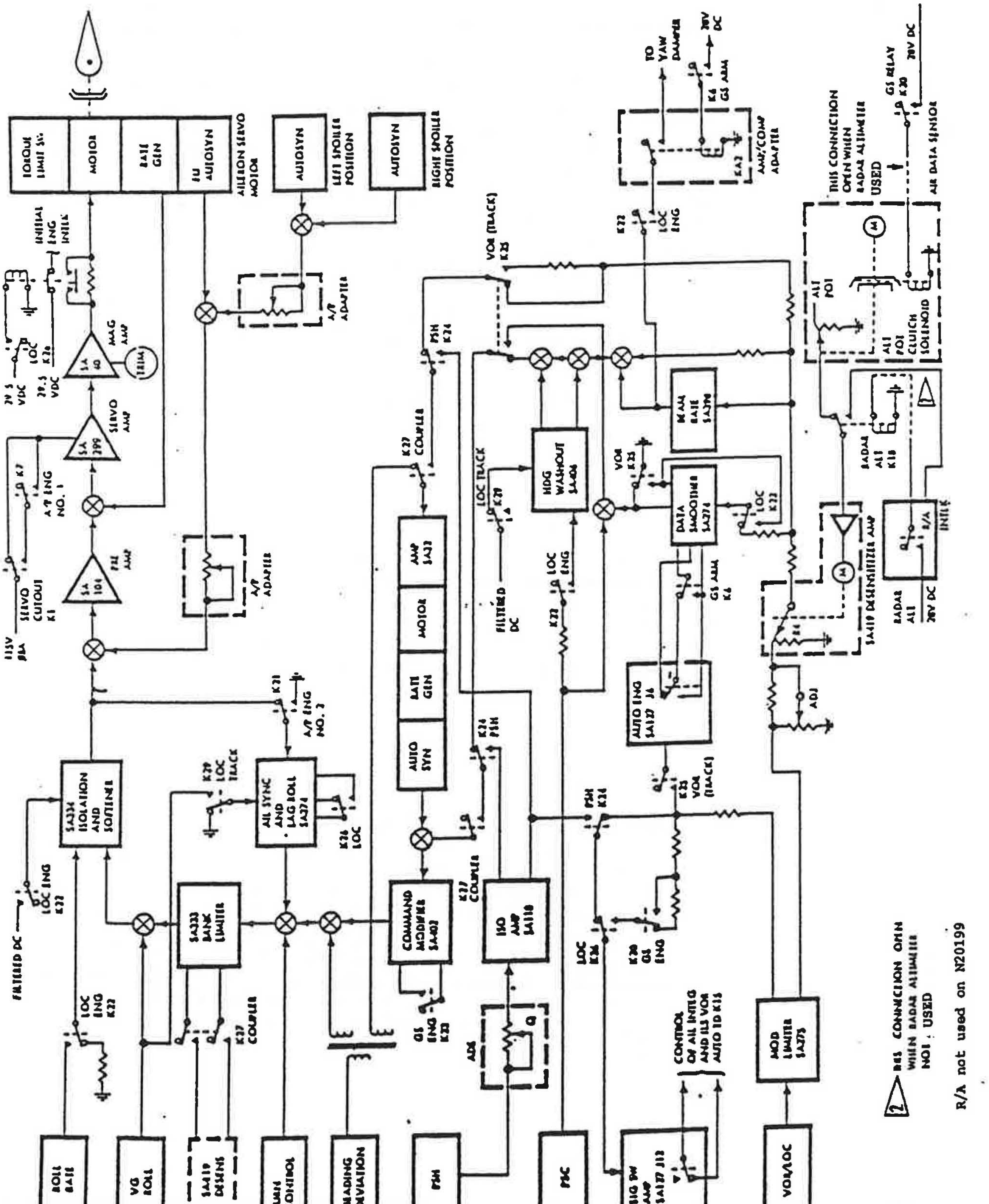
(i) Autopilot engage relays K5 and K6 located in the mach trim coupler, perform switching functions that transfer control of the automatic trim stabilizer from the mach coupler system to the autopilot system. (See 22-10-0.) Relay K5 pulls-in to transfer control of the auto trim clutches from mach trim servo drive amplifier SA76 to elevator channel trim servo drive amplifier SA395 and at the same time energizes mach trim power relay K1 through normally-closed contacts off speed change relay K2. Relay K6 pulls-in to transfer the three-phase excitation windings of the hi-speed or lo-speed trim stabilizer motors (depending on the condition of speed change relay K2) from the mach trim 3-phase power source to the autopilot 3-phase power source. Speed change relay K2 is energized through the autopilot interlock circuit and the trim servo speed switch which in turn is controlled by the position of the wing flaps. In the flaps up position K2 is energized and the 3-phase power is transferred from the hi-speed stabilizer motor to the lo-speed stabilizer motor. One degree extension of the flaps opens the trim servo speed switch to de-energize relay K2 and return power to the fast stabilizer trim motor.

(j) In summary, manual mode autopilot operation controls the aircraft through the aileron and elevator channels to maintain straight and level flight on the heading and with the pitch attitude existing at the time of engagement. In addition, pilot controlled turns, climbs, dives and changes in roll and pitch attitude can be executed through the controller turn knob and pitch trim wheels. Only vertical gyro roll attitude inputs and compass heading inputs constitute the roll command signal in the aileron channel. Compass heading signals are also fed to the aileron integrator which is activated by energizing the aileron integrator clutch and the application of fixed phase voltage to the integrator motor. The integrated compass heading signals null out long-term steady-state compass heading errors.

(k) If the controller turn knob is turned out of center detent to command a certain bank angle, the integrator servomotor is disabled and the compass heading inputs are removed. However, the aileron integrator clutch remains energized so that any integration signal existing at this point remains in the signal channel. The bank command signal generated by movement of the controller turn knob is now present in the signal chain and is nulled out by the roll attitude signal from the vertical gyro as the bank angle is attained. The airplane remains at this bank angle until the controller turn knob is returned to the center detent position. When this occurs, the bank command signal is reduced to zero and the vertical gyro roll attitude signal causes the aircraft to level off and begin tracking the new heading which is reintroduced into the aileron channel by operation of detent relay K4. Relay K4 also reactivates the aileron integrator which responds to any steady state residual type displacement from the new heading. A detailed description of aileron and elevator channel manual mode operation follows.

(l) Aileron Channel

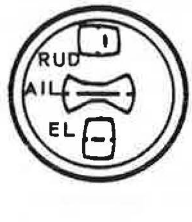
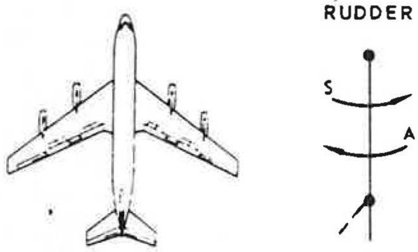
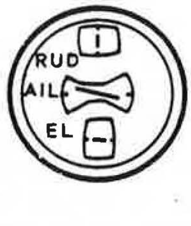
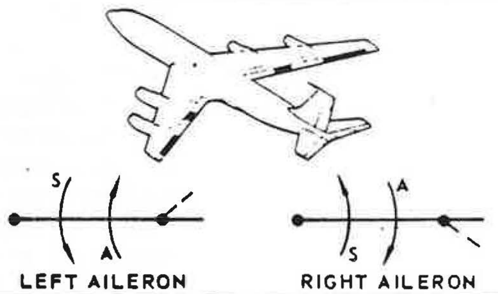
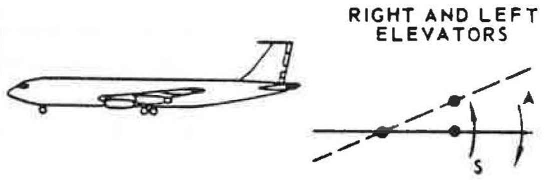
1) The autopilot aileron channel operation in manual mode is illustrated in the block diagram of figure 7. The various input signals are combined with feedback signals at summing points. The resulting signal fed to the servo-amplifier is the differential between input and feedback signals. Before system engagement continuous synchronization is maintained by the aileron servo and the SA274 electronic synchronizer. Synchronization establishes zero bank angle as the engagement reference even though the bank angle at engagement is other than zero. This is accomplished by feeding to the synchronizer any pre-engagement bank angle signal (from the vertical gyro) which develops during a banking maneuver.

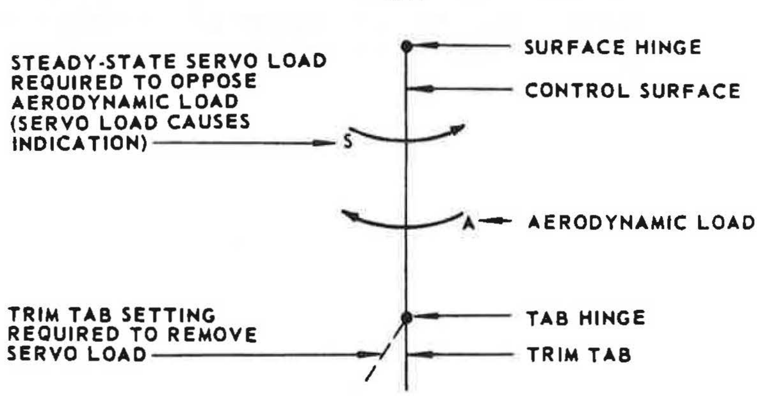


Aileron Channel Block Diagram
Figure 7

2) Prior to autopilot engagement the aileron servo clutch is disengaged. Bank attitude signals from the vertical gyro are fed into the signal chain at the input side of the isolation softener. Normally-closed contacts of autopilot engage relay K21 feed the output of this isolation softener to aileron synchronizer amplifier SA274. The synchronizer produces output signals that are delayed reproductions of the corresponding input signals. The output is connected 180 degrees out of phase with the input signal and thus cancels any steady-state bank attitude signals prior to autopilot engagement. When the autopilot is engaged, relay K21 energizes and grounds the input to the synchronizer. If the aircraft is in a bank at this time, a roll correction signal begins to build up as the synchronizer output decreases to zero. This produces a smooth transition into autopilot engage and causes the aircraft to level off in the roll axis.

3) When engaged in manual mode the aileron channel controls the airplane to compass heading existing at engagement. Any tendency of the airplane to wander from this heading creates a compass heading error signal which is fed to the aileron integrator circuit and to the bank angle limiter. The integrator builds up an output only when the heading error is repeated continually in the same direction because of a persistent wing-heavy condition. Under this condition, the integrator output combines with heading error signal and these are fed to the bank angle limiter. The limiter prevents autopilot response to heading error signals of amplitude sufficient to cause bank angles in excess of 35 degrees. The limiter output is then combined with any bank attitude signal being produced by the vertical gyro as a result of bank angle changes. The resultant error signal is applied first to the servo-preamplifier to increase its amplitude and then to the servo demodulator and magnetic amplifiers where it is further amplified to power the servomotor. The motor responds to this ordered bank signal and positions the ailerons, through movement of their control tabs, to correct for the bank and heading deviations. The servomotor also drives a servo rate generator and a servo follow-up autosyn.

INDICATION	TRIM REQUIRED	CONDITIONS PRIOR TO TRIM
	NOSE RIGHT	
	RIGHT WING DOWN	
	NOSE UP	

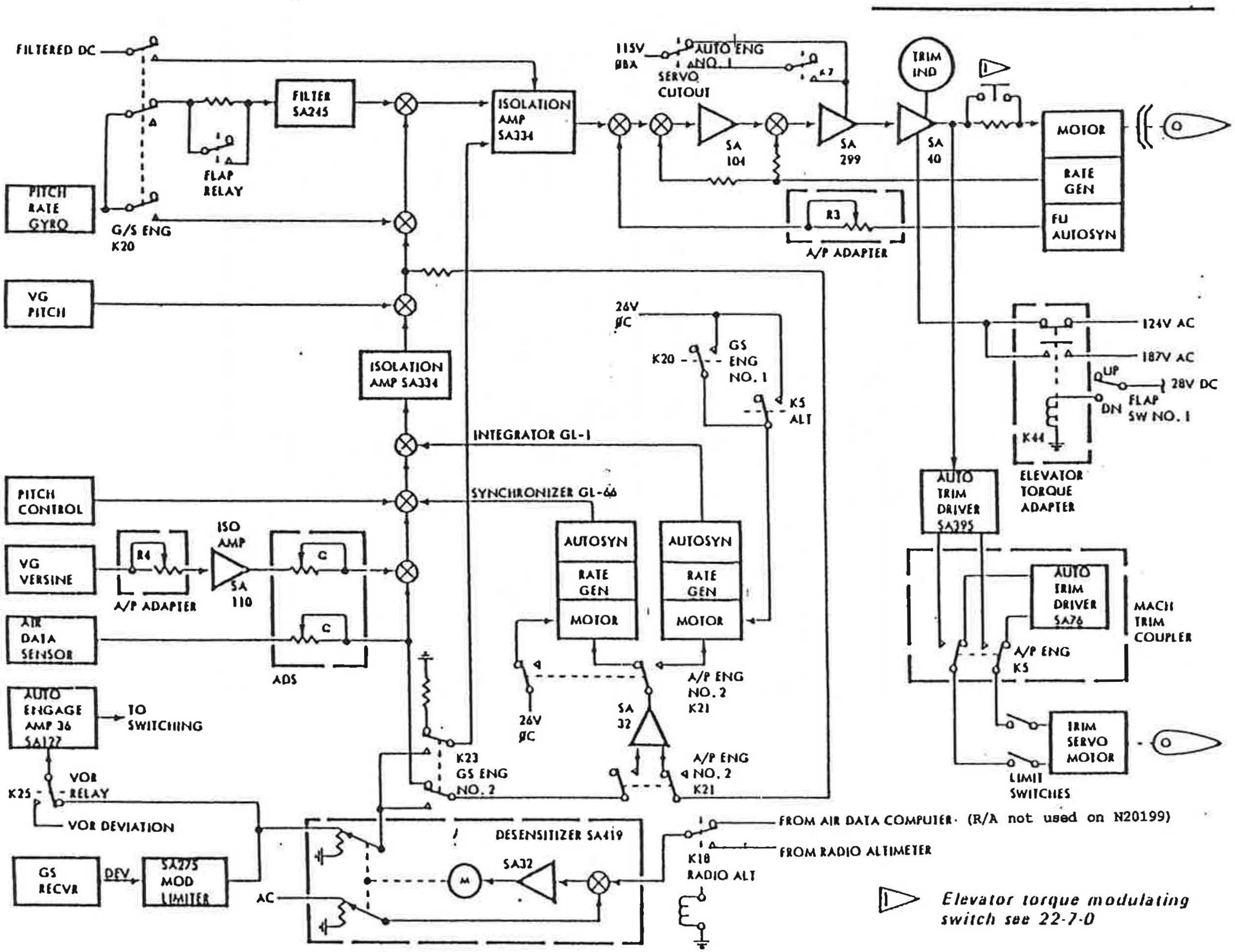


Typical Surface Loading and Resultant Trim Indication
Figure 8

4) The follow-up autosyn signal, representing the position of the ailerons, is combined with a signal from the two outboard spoiler position transmitter autosyns and fed back through the aileron gain adjust potentiometer (R2) to the preamplifier input in opposition to the command signal driving the motor. The spoiler position transmitter signals, are included to provide satisfactory lateral stability of the airplane while under autopilot control during rapid let dawn conditions with speed brakes (spoilers) extended. They are differentially connected to give zero resultant signal when both spoilers are extended equally regardless of the amount extended, and a reversible phase signal when one is extended more than the other. When the combined position follow-up signal cancels the command signal, the motor stops, leaving the aileron surfaces streamlined. The servo rate generator signal is fed back to the input of the servo demodulator amplifier to oppose changes in motor speed. Thus, it provides damping of the motor as it stops. Also, a negative feedback signal proportional to the motor drive voltage is produced in a feed back network and applied to the amplifier input to provide additional stabilization.

5) Overall aileron channel gain is established by adjustment of the combined position follow-up signal in the alleron gain potentiometer. Optimum gain, a function of airplane flight characteristics in roll, is predetermined and preset at installation. The aileron indicator bar in the three-axis trim indicator will deflect whenever the demodulator amplifier is giving an output voltage. If this out-of-trim indication persists in one direction, manual trim adjustment is required to relieve the sustained out-of-trim condition causing the indication. Figure 8 gives an illustration of the indication and the trim required. It is advisable to disengage the autopilot by pressing the wheel disconnect witch before manual trim correction is made. Then the autopilot may be re-engaged with the airplane in proper trim.

6) Should it be desirable or necessary to change airplane heading, a bank command signal my be inserted in the signal channel by operating the turn controller an the control panel. As the knob is moved off detent position, the controller detent switch No. 2 opens, removing the compass heading error signal from the signal chain. (See figure 6 sheet 2.) Power is also removed from the speed change relay, which relaxes to give high speed stabilizer trim servomotor operation for better stability in maximum turns. The



Elevator Channel Block Diagram
Figure 9

bank command signal then rolls the airplane at an angle depending on the amount the controller knob is manually turned. Since this signal is fed through the bank angle limiter, commands for roll angles in excess of 35 degrees will cause a maximum bank of only 35 degrees. As the bank command signal rolls the airplane, the bank attitude signal from the vertical gyro builds up to a value which exactly cancels the command signal. Thus, the airplane rolls to the desired bank angle and is maintained in that attitude until the command signal is modified. As the new heading is approached the turn knob must be returned manually to center detent position, rolling the airplane back to wings-level attitude on the new heading. The detent switch No. 2 will then close to re-engage the compass heading error signal circuit and apply heading error referenced to the new heading. During a manually controlled turn the compass synchronizer (where used) is energized to zero the synchronizer autosyn at the new heading, as is done prior to autopilot engagement. In systems using clutched heading autosyn in the RMI (or compass coupler) the autosyn is declutched and centered to zero output during manually controlled turns.

(m) Elevator Channel

1) The autopilot elevator channel operation in manual mode is illustrated in the block diagram of figure 9. Again, as in aileron channel operation, various input signals are combined with feedback signals at summing points and the resultant is fed to the servo-amplifiers. Prior to engagement of the elevator servo clutch any signal impressed on the signal chain as a result of pitch rate and pitch attitude sensing, or versine generator output is cancelled by the pitch synchronizer. (No output is available from the controller pitch potentiometers at this time since they are not clutched to the pitch wheels until system engagement and they are zeroed by a centering device Any unbalance signals at the servomotor drives the servo rate generator and follow-up autosyn until their outputs cancel the driving signal and the motor stops. Thus, the servomotor is continuously synchronized to the pitch attitude. Then when the elevator servo clutch is engaged, the autopilot will control to the pitch attitude reference existing at time of engagement. After engagement, while operating in manual control mode, pitch attitude may be changed by manually turning the controller pitch wheels. Controller adjusted pitch attitude then becomes autopilot pitch reference.

2) The elevator channel servo components function in much the same manner as those in the aileron channel except that the feedback signals are in some instances fed back to different points in the signal chain. For example, to assure high elevator stability, the servo rate generator output signal is fed both to the input of the high-gain preamplifier and the input of the servo demodulator amplifier. In this instance also, the servo position follow-up autosyn output is fed to the signal chain ahead of the preamplifier. In manual mode without altitude hold, the altitude error autosyn in the air data sensor is not clutched in and is zeroed, so does not contribute to this action. Also the elevator integrator receives no signal in manual mode, its motor receives no excitation, and the computer autosyn is zeroed, so it likewise does not contribute a signal. The net difference signal is fed to the isolation amplifier and softener which limits its magnitude. The isolation amplifier output is compared to the combined pitch attitude and pitch rate (if any) signals and the difference is the error signal fed to the servo-amplifiers.

3) An up-elevator signal is inserted in the signal chain to prevent loss of altitude in turns. A cosine function (cos θ) of bank angle, generated in the vertical gyro, is fed to an external network in which a versed sine value (1-cos θ) is derived. This signal is placed on the versine gain potentiometer (R4) in the autopilot adapter for gain adjustment in accordance with the amount of up-elevator required for a given bank angle. The gain adjusted signal is amplified in the isolation amplifier and is attenuated by the K/Q potentiometer in the air data sensor. The output signal is zero for zero bank angle and increases in the same direction for both right and left bank angles. The magnitude of the output signal is inversely proportional to the airspeed.

4) The pitch rate signal from the three-axis rate transmitter pitch autosyn is fed to a band-pass amplifier. There it is demodulated to give a d-c signal proportional to pitch rate. This signal is filtered to give zero output when the pitch rate signal is changing at either fast rates or slow rates and full output when changing at intermediate rates. Thus, steady-state pitch rate signals are blocked and output is obtained only when the pitch rate is changing within specified frequency limits. The filter band-pass is adjusted to peak at 0.85 cps for pitch rate filtering. The filtered d-c signal is then modulated back into an equivalent a-c signal which, after attenuation, is fed to the servo control signal chain.

5) Any error signal whether from a pitch command input or a sensed pitch change resulting from an out-of-trim condition, that is applied to the servo-preamplifier is amplified to increase the torque gradient of the servo-amplifier. The output of the servo-amplifier drives the servomotor to displace the elevator control surfaces in the proper direction. This changes the pitch attitude which is measured by the vertical gyro. As previously stated, the servo position follow-up signal is fed to the preamplifier input in opposition to the command signals. When this input and consequent output is decreased below the value of the pitch attitude signal from the vertical gyro, direction of the servomotor is reversed and the follow-up signal input to the preamplifier is reduced. Follow-up signal reduction continues until the output signal is equal and opposite to the vertical gyro signal. This signal balance occurs when the control surface is returned to streamline.

6) The variable phase output from the servo-amplifier not only drives the servomotor, but is simultaneously fed to the stabilizer auto-trim actuator (SA76) which changes it into d-c clutch-solenoid control current. Depending upon the phase of the amplifier output voltage, one or the other of the auto-trim clutches is engaged for clockwise or counter-clockwise rotation of the trim motor shaft. To obtain coordination between elevator and trim stabilizer, actuator sensitivity is set so that the trim motor is automatically clutched to the stabilizer only when the elevator servomotor is delivering large amounts of torque. Thus, the horizontal stabilizer is moved a small amount as the elevators are deflected to change airplane attitude. Because the trim actuator threshold sensitivity is lower than that of the elevator servomotor, the stabilizer remains somewhat deflected as the elevator servomotor reverses to drive the system to a signal balance. Even though the elevator control tabs and consequently the elevator control surfaces return to streamline at balance, the horizontal stabilizer will provide the airplane response necessary to fulfill the pitch attitude commanded. This process gives automatic trim in the pitch axis and there is no need for manual trim in pitch when autopilot is engaged. In fact, manual pitch trim, supplied either manually or by the pitch trim motor while the autopilot is engaged, will cause the autopilot to disconnect automatically. If the autopilot is purposely disconnected in flight by pressing the wheel disengage switches, the airplane will be in a "hands-off" trim condition in the pitch axis. In the event the stabilizer is driven beyond the travel limit in either direction, one of the limit switches will open the driving clutch solenoid circuit to disengage the stabilizer from the trim motor.

7) Elevator servo-amplifier output will also deflect the pitch trim movement in the three-axis trim indicator. However, no sustained indication will prevail since pitch trim is continuously provided. Figure 8 illustrates a typical momentary indication obtained during trim adjustment. During manually-controlled flight when autopilot is disengaged, but energized, a sustained out-of-trim condition in the pitch axis may develop but will not give a sustained indication on the three-axis trim indicator. This is because continuous synchronization of the elevator servomotor to existing pitch attitude nulls the servo-amplifier output.

8) Torque relay K44, located in the elevator torque adapter box, changes the elevator servo torque as a function of flap position to improve system performance in the approach configuration. When the flaps are down, flap switch No. 1 activates torque relay K44. A set of pulled-in contacts on K44 increases the excitation voltage to servo magamp SA40 from 124 volts ac to 187 volts ac. A second set of contacts on K44 functions in the main autopilot interlock circuit to cause a system disconnect in the event K44 fails to operate in response to actuation of the flap switches. Failure of either of the flap switches is indicated by a steady illumination of the Gables autopilot disconnect warning light although the autopilot does not disconnect.

9) The overall elevator channel gain is established by adjustment of the elevator gain potentiometer (R3) in the autopilot adapter which sets the level of the servo position follow-up signal. Optimum gain, a function of airplane flight characteristics in pitch, is predetermined and preset at installation.

(n) Rudder Channel

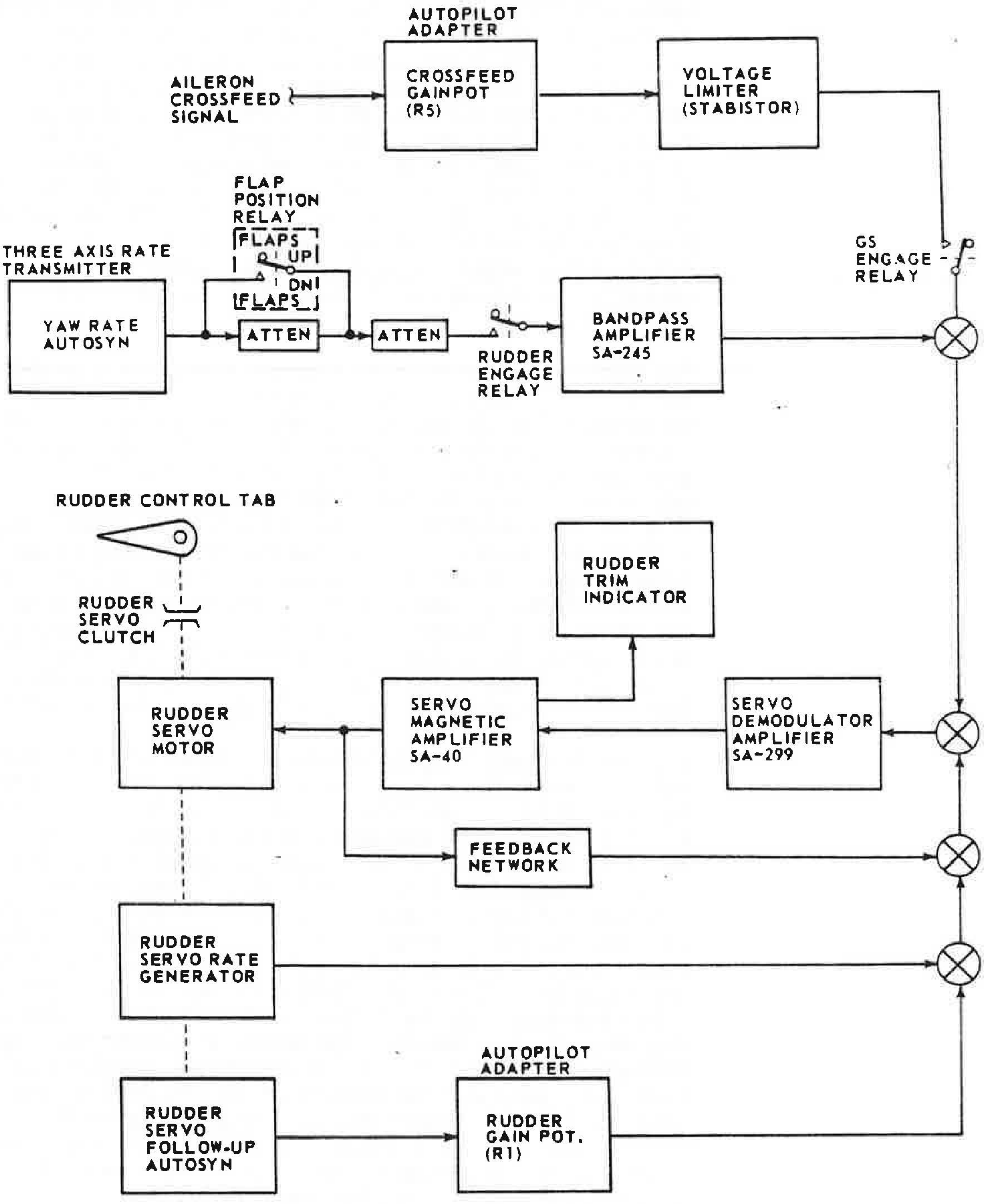
<u>NOTE</u>: The following paragraphs, apply to installations using the parallel yaw damper. The series yaw damper is covered in section 22-6-0.

1) The autopilot rudder channel operation in manual mode is illustrated in the block diagram of figure 10. Upon system engagement the rudder servo clutch engages the rudder control tab rigging to the rudder servomotor. A delay of 0.5 seconds then occurs before a yaw rate signal may be applied to the summing junction by the rudder engage relay. This delay assures solid engagement of the rudder servo clutch before a yaw rate signal is applied to the servo-amplifier. Since the airplane attitude at time of autopilot engagement is the reference attitude to which the autopilot controls, any yaw deviation from reference produces a yaw rate signal from the yaw rate gyro autosyn in the three-axis rate transmitter.

2) The yaw rate signal is fed to a band-pass amplifier SA245 where it is converted into a d-c signal proportional to the yaw rate. The d-c signal is passed through the band-pass filter which gives zero output when the signal is changing at either fast rates (high frequencies) or slow rates (low frequencies), and full output when it is changing at intermediate rates. Thus, any steady-state output from the yaw rate gyro autosyn is blocked by the filter and filter output is obtained only when the turn rate is changing within specified frequency limits. The filter pass-band is adjusted to peak at 0.4 cps for yaw rate filtering. The filtered signal is then modulated back into an a-c signal and applied to the servo demodulation amplifier SA299 and the servo magnetic amplifier SA40 where it is amplified to power the servomotor.

3) The motor responds to this command signal in a manner that positions the rudder, through movement of its control tab, to correct for the yaw deviation. The servomotor also drives a follow-up autosyn which measures the angular position of the driven shaft, and whose output is representative of surface position. This signal is fed back to the amplifier in opposition to the command signal driving the motor. When it cancels the driving signals the motor stops. In addition to the position follow-up signal a rate follow-up signal from the rate generator coupled to the motor is provided. Since the rate signal represents motor speed, it is fed back to the amplifier input phased to oppose change in speed. Thus, it performs a damping function to minimize shaft oscillation as the motor stops.

4) A third feedback signal inversely proportional to the motor drive voltage is produced in a feedback network and is applied to the amplifier input in opposition to the driving voltage to provide additional stabilization. The ratio of amplifier output signal to input signal (overall rudder channel gain) is estabilished by adjustment of the position follow-up signal in the rudder gain potentiometer. The optimum gain ratio, a function of airplane flight characteristics in yaw, is predetermined and preset at installation. The rudder indicator bar in the three-axis trim indicator will be displaced from center position whenever the demodulator amplifiers is giving an output voltage. If this out-of-trim indication persists in one direction, manual trim adjustment is required to put the airplane back in proper yaw axis trim. Figure 8 illustrates a typical indication, interprets the trim required, and diagrams the control surface and forces acting on it for the indication shown.



Rudder Channel Block Diagram
Figure 10

5) Any error signal whether from a pitch command input or a sensed pitch change resulting from an out-of-trim condition, that is applied to the servo-preamplifier is amplified to increase the torque gradient of the servo-amplifier. The output of the servo-amplifier drives the servomotor to displace the elevator control surfaces in the proper direction. This changes the pitch attitude which is measured by the vertical gyro. As previously stated, the servo position follow-up signal is fed to the preamplifier input in opposition to the command signals. When this input and consequent output is decreased below the value of the pitch attitude signal from the vertical gyro, direction of the servomotor is reversed and the follow-up signal input to the preamplifier is reduced. Follow-up signal reduction continues until the output signal is equal and opposite to the vertical gyro signal. This signal balance occurs when the control surface is returned to streamline.

6) The variable phase output from the servo-amplifier not only drives the servomotor, but is simultaneously fed to the stabilizer auto-trim actuator (SA76) which changes it into d-c clutch-solenoid control current. Depending upon the phase of the amplifier output voltage, one or the other of the auto-trim clutches is engaged for clockwise or counter-clockwise rotation of the trim motor shaft. To obtain coordination between elevator and trim stabilizer, actuator sensitivity is set so that the trim motor is automatically clutched to the stabilizer only when the elevator servomotor is delivering large amounts of torque. Thus, the horizontal stabilizer is moved a small amount as the elevators are deflected to change airplane attitude. Because the trim actuator threshold sensitivity is lower than that of the elevator servomotor, the stabilizer remains somewhat deflected as the elevator servomotor reverses to drive the system to a signal balance. Even though the elevator control tabs and consequently the elevator control surfaces return to streamline at balance, the horizontal stabilizer will provide the airplane response necessary to fulfill the pitch attitude commanded. This process gives automatic trim in the pitch axis and there is no need for manual trim in pitch when autopilot is engaged. In fact, manual pitch trim, supplied either manually or by the pitch trim motor while the autopilot is engaged, will cause the autopilot to disconnect automatically. If the autopilot is purposely disconnected in flight by pressing the wheel disengage switches, the airplane will be in a "hands-off" trim condition in the pitch axis. In the event the stabilizer is driven beyond the travel limit in either direction, one of the limit switches will open the driving clutch solenoid circuit to disengage the stabilizer from the trim motor.

7) Elevator servo-amplifier output will also cleflect the pitch trim movement in the three-axis trim indicator. However, no sustained indication will prevail since pitch trim to continuously provided. Figure 8 illustrates a typical momentary indication obtained during trim adjustment. During manually-controlled flight when autopilot is disengaged, but energized, a sustained out-of-trim condition in the pitch axis may develop but will not give a sustained indication on, the three-axis trim indicator. This is because continuous synchronization of the elevator servomotor to existing pitch attitude nulls the servo-amplifler output.

8) Torque relay K44, located in the elevator torque adapter box, changes the elevator servo torque as a function of flap position to improve system performance in the approach configuration. When the flaps are down, flap switch No. 1 activates torque relay K44. A set of pulled-in contacts on K44 increases the excitation voltage to servo magamp SA40 from 124 volts ac to 187 volts ac. A second set of contacts on K44 functions in the main autopilot interlock circuit to cause a system disconnect in the event K44 fails to operate in response to actuation of the flap switches. Failure of either of the flap switches is indicated by a steady illumination of the autopilot disconnect warning light although the autopilot does not disconnect.

9) The overall elevator channel gain is established by adjustment of the elevator gain potentiometer (R3) in the autopilot adapter which sets the level of the servo position follow-up signal. Optimum gain, a function of airplane flight characteristics in pitch, is predetermined and preset at installation.

(n) Rudder Channel N/A for N20199, N19997, N20000

NOTE: The following paragraphs, apply to installations using the parallel yaw damper. The series yaw damper is covered in section 22-6-0.

1) The autopilot rudder channel operation in manual mode is illustrated in the block diagram of figure 10. Upon system engagement the rudder servo clutch engages the rudder control tab rigging to the rudder servomotor. A delay of 0.5 seconds then occurs before a yaw rate signal may be applied to the summing junction by the rudder engage relay. This delay assures solid engagement of the rudder servo clutch before a yaw rate signal is applied to the servo-amplifier. Since the airplane. attitude at time of autopilot engagement is the reference attitude to which the autopilot controls, any yaw deviation from reference produces a yaw rate signal from the yaw rate gyro autosyn in the three-axis rate transmitter.

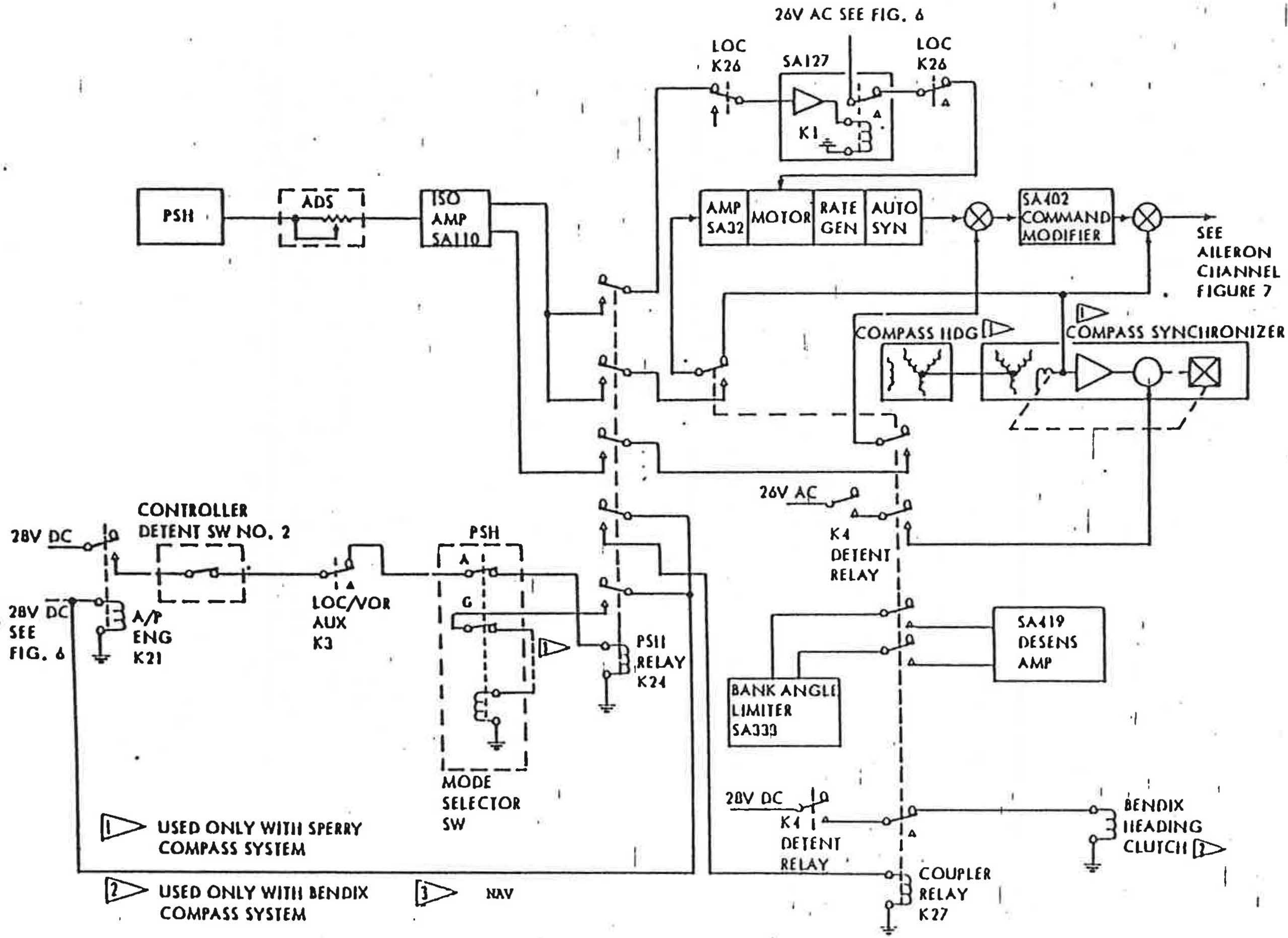
(2) Preset Heading Mode

(a) In the preset heading mode the autopilot automatically guides the aircraft to attain and hold a preselected heading. The new heading is set into the horizontal situation Indicator (HSI) by means of the HDG knob, after which the mode selector switch is set to the HDG position.
The autopilot then automatically banks the aircraft into a turn toward the selected heading and automatically rolls the aircraft out of the turn as the selected heading is approached.

(b) The autopilot must be engaged in manual mode prior to engaging the HDG mode. (See figure 6.) Thus the circuit configuration established by manual mode control switching remains in effect with the exception that normal compass heading signals from the compass synchronizer (or Bendix clutched compass heading) are removed from the aileron channel and replaced with the preselected heading signals. This switching is accomplished by energizing preset heading relay K24 and coupler relay K27.

(c) Preset heading relay K24 is energized through closed contact A of the mode selector switch. (See figure 11.) Normally-open contacts of K24 in turn, complete the circuit to the coil of coupler relay K27. With both relays energized the preselected heading signal is coupled directly into the aileron channel at the input of command modifier SA402 and to the aileron integrator at the input of integrator amplifier SA32. The preselected heading is also coupled to the input of aileron channel signal switch amplifier SA127 which operates signal switch relay K1 as a function of heading error signal strength. Large error signals energize signal switch relay K1 and disable the aileron integrator by removing the fixed-phase excitation voltage from the integrator motor. When the heading signal drops to a value corresponding to a heading error of 1 degree signal switch relay K1 releases and heading signal integration commences. Integration stops if the heading error exceeds 1 degree since signal switch relay K1 opens the circuit to the integrator motor.

(d) The normal (manual mode) compass heading input to the aileron channel is nulled by connecting the fixed-phase excitation voltage to the synchronizer motor through pulled-in contacts of coupler relay K27. The synchronizer servo loop then operates to continually null the output of the synchronizer autosyn to the aircrafts existing heading. When the HDG mode is released K27 de-energizes and power is removed from the synchronizer motor. The synchronizer autosyn output then functions as the control signal to provide heading hold on the newly acquired heading. In installations using the Bendix compass system instead of the compass synchronizer, relay K27 functions to de-energize the Bendix heading clutth solenoid. The declutched



Preset Heading Mode
Figure 11

autosyn is then centered to zero output. Releasing the HDG mode re-energizes the heading clutch solenoid and restores the compass heading signal to the aileron channel to provide heading hold on the newly acquired heading. Other separate contacts of coupler relay K27 connect a bias voltage from desensitizer amplifier SA419 to the limiter circuit in bank angle limiter SA333. This reduces the bank angle limits from 35 degrees to 30 degrees during heading mode operation.

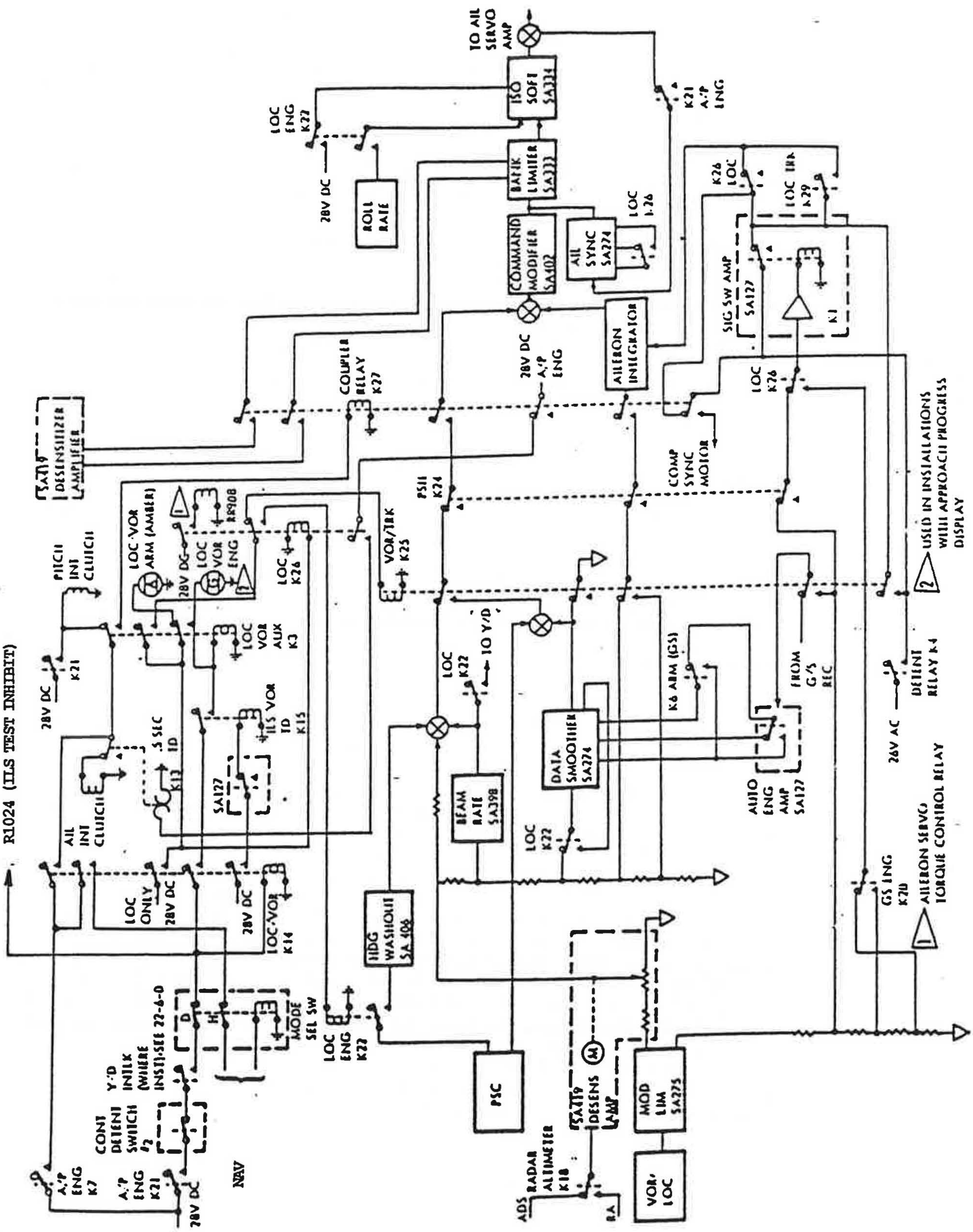
(e) The summed heading error and integrated heading error signals are applied to the input of command modifier SA402. The command modifier produces a linear output for command roll rates up to 4 degrees of roll per second but maintains this limit for input signals that exceed this rate. From this point on, operation of the aileron channel is identical to that for manual mode control.

(3) Localizer - VOR Mode:

(a) In localizer - VOR mode the autopilot automatically guides and stabilizes the aircraft to intercept and track a preselected VOR or localizer radio beam. Beam displacement signal from the localizer or VOR radio receivers are summed algebraically with course deviation signals from the HSI.
The resultant signal provides the autopilot with a lateral command that bank the aircraft into a turn towards the preselected VOR radial or locelizer radio beam and rolls the aircraft out of the bank as the beam center is approached. The autopilot then maintenance the aircraft centered on the beam with the crab angel required to counteract the effect of prevailing crosswinds.

(4) Localizer Mode Operation:

(a) The MANUAL or PSH, modes may be used, to fly the aircraft on a proper heading to intercept the desired localizer beam.
The course defined by the beam is then set on the HSI.
The localizer receiver is tuned to the proper frequency and the mode selector witch is set to LOC/VOR. These actions energize LOC/VOR relay K14 and LOC engage relay K26. (See figure 12.) Relay K14 is energized through contact D of the mode selector witch, normally-open contacts of the yaw damper interlock relay (where installed in this position), contact number 2 of the controller detent switch, and normally-open contacts of autopilot engage relay K21. Relay K26 then becomes energized through normally-open contacts of K14, providing the localizer receiver is tuned to a localizer frequency. Other contacts of relay K14 provide an alternate de path to the aileron integrator autosyn, clutch solenoid, energize the holding coil of the mode selector switch and in addition arm the contacts of signal witch amplifier relay K1 and ILS/VOR AUTO relay K15 for subsequent localizer engagement as a function of beam signal strength.



LOC/VOR Mode Control Switching
Figure 12

(b) Relay K26, in the energized state, connects the input of the signal switch amplifier to the proper output of modulator-limiter SA275 for automatic localizer engage operation (through relaxed contacts of GS ENG relay K20) and in addition removes the fixed-phase voltage from the aileron integrator servomotor. Thus, there is no integration in the aileron channel during this phase of the localizer approach. Other switching functions of relay K26 include:

1) Changing the time-constant of roll lag and synchronizer amplifier SA274 (from 10 to 30 seconds) for subsequent introduction of a lag roll damping signal into the aileron channel. (See figure 12.)

2) Connecting 28 volts dc to contacts E and F of the mode selector switch for subsequent glideslope engagement as the aircraft tracks the localizer inbound towards glideslope intercept. (See figure 13.)

3) Completing the circuit to the coil of LOC ENG relay K22 so that localizer rather than VOR operation ensues when signal switch amplifier relay K1 releases (and K3 energizes) as a function of decreasing localizer beam strength.

(c) The dc output of the localizer receiver is fed to the input of modulator-limiter SA275 which converts the dc signal to a 400 cps signal that corresponds in phase and amplitude to the polarity and magnitude of the dc input. After conversion the signal is amplified and separated into two branches. One branch is applied to a limiter circuit which sets an upper limit to the output signal amplitude. The limited signal is then applied to a voltage divider which includes a potentiometer for signal level adjustment, and then to localizer desensitizer potentiometer R4 which reduces the signal level as a function of altitude during glide slope descent to the runway. The signal attenuation compensates for the increased signal levels due to geometric convergence of the localizer beam. From altitude desensitizer potentiometer R4 the beam signal is applied to a second voltage divider which establishes the design signal levels required for coupling the beam and beam rate signals into the aileron signal channel. The second output of the modulation limiter is not limited. This signal, in localizer operation, is fed to the input of aileron channel signal switch amplifier SA127 which performs automatic switching as a function of localizer beam signal strength.

(d) While the aircraft is in the saturated portion of the localizer beam, coupler relay K27 is de-energized and the localizer beam and preset course signals to the aileron channel are interrupted. During this period the aileron channel, receives the normal heading input and flies the aircraft on its existing heading toward the selected localizer beam. Since the beam displacement signal is large while the aircraft is in the saturated portion of the beam signal switch amplifier relay Kl is energized and the circuit to the coil of ILS/VOR AUTO relay K15 is open. When the aircraft enters the unsaturated portion of the beam and the displacement signal decreases to approximately 150 microamperes, signal switch (SA127) amplifier relay Kl releases and relay K15 becomes energized one-half second later. The half-second time delay ensures stable signal conditions before localizer engagement.

(e) Relay K15, when pulled in, energizes LOC/VOR AUX relay K3, which in turn through separate sets of its own normally-open contacts, energizes coupler relay K27 and LOC ENG relay K22 while at the same time interrupting the dc circuit to the coil of PSH relay K24. With LOC/VOR AUX K3 in the pulled-in condition the aileron integrator clutch solenoid then receives its energizing power through closed contacts of K14. If the mode selector switch is now moved back to the MAMUAL or PSH modes relay K14 becomes de-energized. Its contacts then remove energizing power from the aileron integrator clutch and the integrator clutch solenoid is declutched. At the same time, power is removed from the coil of LOC/VOR AUX relay K3, which in the de-energized condition provides an alternate de circuit to the integrator clutch solenoid. However, the CR time delay circuit across the coil of K3 delays opening of the relay for one-half second. This period allows the aileron integrator autosyn to be nulled by centering levers before integration commences in the new mode.

(f) LOC ENG relay K22, in the energized condition, performs switching functions and circuit modifications as follows:

1) Connects the preset course signal to the input of heading washout amplifier SA406 which passes normal heading deviation signals but eliminates or "washes out" any long term steady state deviation signal such as that caused by a sustained crosswind. This feature enables the aircraft to track the localizer beam with the crab angle necessary to counteract a prevailing croswind.

2) Introduces roll rate signals into the aileron signal channel through isolation and softener amplifier SA334 while at the same time increasing the gain of amplifier SA334 for improved response. The roll rate signal damps out large sudden aircraft displacements due to beam noise, etc.

angle limlter is summed with the vertical gyro attitude signal at the input to isolation and softener amplifier SA334. Roll rate is also introduced for improved damping,, through normally open contacts of LOC ENG relay K22. The composite command signal is then fed to servo preamplifier SA104. The servo portion of the aileron channel responds to the command signal in the same manner as in manual mode operation. Elevator channel operation during LOC/VOR mode is the same as fcr manual mode.

(5) VOR Mode Operation

(a) The MANUAL or PSH modes may be used to fly the aircraft on a proper heading to intercept the desired VOR radial. The course defined by the VOR radial is then set on the HSI. The VOR receiver is tuned to the proper frequency and the mode selector switch set to VOR/LOC. This energizes LOC/VOR relay K14. The system is new armed for automatic VOR engagement as a function of beam signal strength by means of signal switch amplifier SA127. (See figure 12.)

(b) When the mode selector switch is set to the LOC/VOR position the condition of relay K26 determines whether the system is operating in the VOR or localizer modes. Relay K26 is energized through normally-open contacts of LOC/VOR relay K14 only when the localizer receiver is tuned to a localizer frequency. The system is then armed for subsequent localizer engage as described in the previous paragraph. With relay K26 de-energized the system is armed for subsequent VOR engagement.

(c) Initially, coupler relay K27 is de-energized and therefore the composite VOR beam and preset course signal is disconnected from the aileron signal channel. During this period the aileron channel receives the normal compass heading signal and the autopilot flies the aircraft on its existing heading toward the selected VOR radial. The VOR beam displacement signal however, is applied to signal switch amplifier SA127 through normally-closed contacts of relays K24 and K26. Consequently, since the beam displacement signal is large while the aircraft is in the saturated portion of the VOR beam the signal switch amplifier relay is energized and the circuit to the coil of ILS/VOR AUTO relay K15 is open. When the aircraft enters the unsaturated portion of the beam and the beam displacement signal decreases to approximately 90 microamperes the signal switch amplifier relay releases causing K15 to energize after a half-second delay. This completes the d-c circuit to the coil of LOC/VOR AUX relay K3, which in turn energizes VOR/TRK relay K25 through normally-closed contacts of LOC ENG relay K26 and coupler relay K27. The system is now engaged to track the selected VOR radial.

3) Arms aileron crossfeed relay KA1 for subsequent roll-to-yaw beam rate crossfeed, which occurs during glide slope operation when GS ARM relay K6 energizes. The roll-to-yaw crossfeed signal, which is limited to 2 1/2 degrees of rudder command, provides increased tracking stability during final approach.

4) Completes the interlock to the coils of glide slope engage relays K20 and K23 in preparation for subsequent glideslope engagement.

5) Disconnects data smoother SA274 from the output of modulator-limiter SA275 and at the same time connects the output of the data smoother back to its own input. This allows any accumulated charge on the data smoother filter capacitors to discharge through the input and return the amplifier to a unit gain condition for subsequent operation.

(g) Coupler relay K27, in the energized state, completes localizer engagement by replacing the normal compass heading signal in the aileron channel with the computed localizer lateral control signal and at the same time reduces the bank angle limits from 35 degrees to 30 degrees as in heading mode operation. These modifications are accomplished as follows:

1) A set of normally-open contacts couples the summed beam displacement, beam rate, and course deviation signals to the input of command modifier SA402.

2) Another set of contacts connects an attenuated beam displacement signal to the input of aileron integrator amplifier SA32. However, beam error integration does not commence until later in the approach when LOC/TRK relay K29 energizes and connects the fixed-phase excitation voltage to the aileron integrator motor.

3) Reduction of the bank angle limits from 35 degrees to 30 degrees is effected through two separate contacts of K27 which connect a limiter reference voltage from desensitizer amplifier SA400 to bank angle limiter SA333.

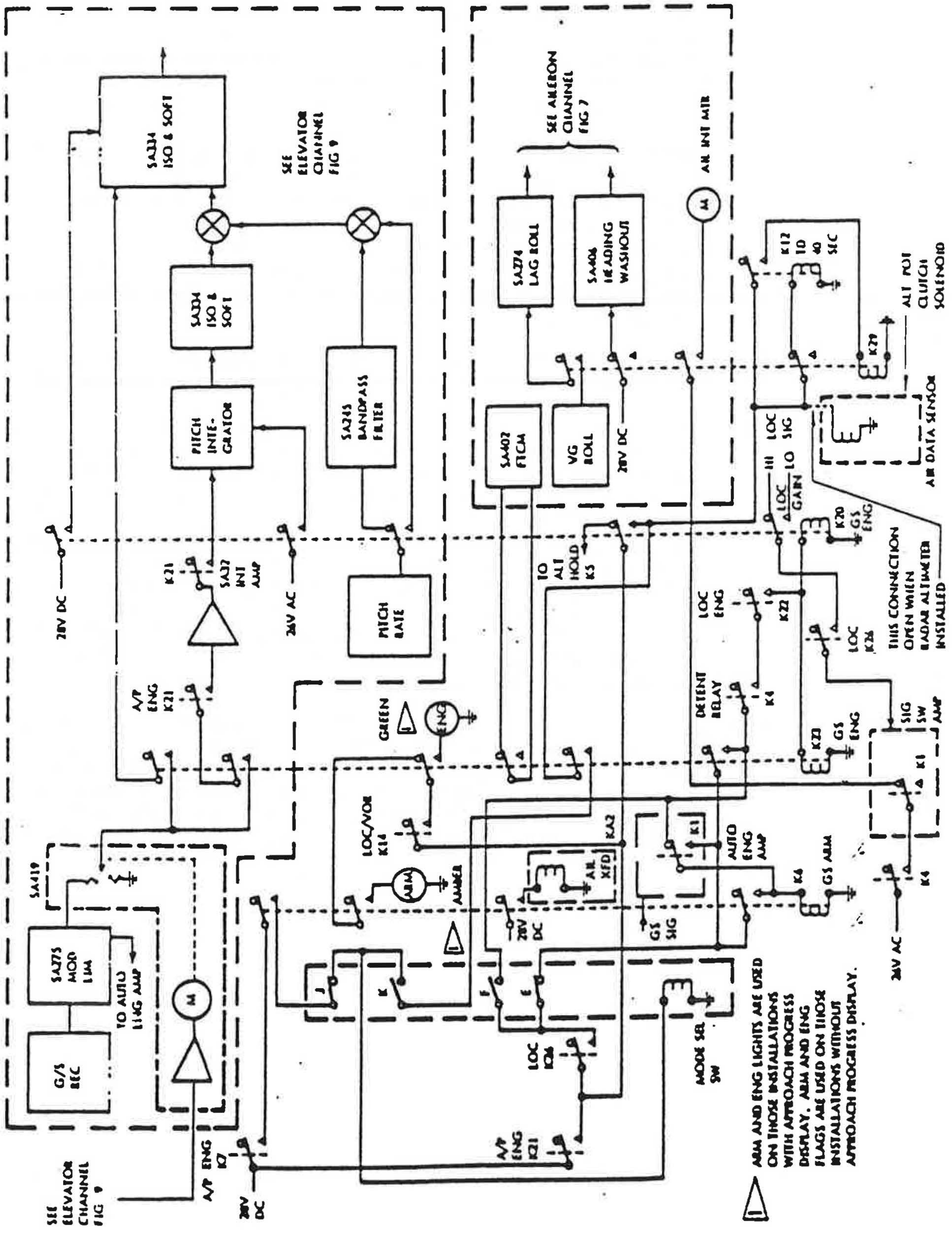
4) Other normally-open contacts of K27 remove the normal compass heading signals from the aileron channel either by exciting the compass synchronizer motor or de-energizing the Bendix heading clutch, whichever is installed. This action is fully described under preset heading mode operation.

(h) The composite localizer beam, beam rate and course deviation signal is fed to command modifier SA402 which produces a linear output for small input signals but softens and limits large inputs to an output that corresponds to a roll rate of 4 degrees per second. The output of the command modifier is then fed to bank angle limiter SA333. The output of the bank

(d) Preset course signals are summed with the VOR beam displacement signal from data smoother SA274 and coupled to the aileron signal channel through normally-open contacts of VOR TRK relay K25, normally-open contacts of PSH relay K24 and normally-open contacts of coupler relay K27. The VOR beam displacement signal is also coupled to the input of the aileron integrator through other contacts of the same relay configuration (K25, K24, K27). Other contacts of VOR TRK relay K25 perform the following functions: remove the beam rate signal from the aileron signal chain; provide an alternate path for the 26-volt fixed phase voltage to the aileron integrator servomotor independent of the signal switch amplifier relay; connect the nonlimited VOR signal to signal switch amplifier SA127 for data smoother control.

(e) The data smoother is a low pass filter amplifier (integrator) that bypasses rapid beam signal fluctuations to produce an averaged beam error output signal. The time constant of the low pass filter, either 10 seconds or 30 seconds, is determined by the instantaneous amplitude of the nonlimited VOR beam signal. This signal is applied to elevator signal switch amplifier SA127. When the amplitude of the signal exceeds 65 microamperes the signal switch amplifier closes and switches the data smoother time constant from 10 to 30 seconds. When the signal drops below 65 microamperes the signal switch amplifier opens and returns the time constant to the 10 second value. During actual over-the-station passage through the "cone of confusion" the two time constants may be switched back and forth several times in order to smooth out the rapid fluctuation of the signal.

(f) The composite lateral command signal consisting of the summed VOR displacement signal from the data smoother, the preset course deviation signal and the long-term correction signal from the aileron integrator is coupled into the aileron servo channel through command modifier SA402. From this point on operation of the aileron channel is the same as in manual mode.



Glide Slope Mode Switching
Figure 13

(6) Glide Slope Automatic Mode

(a) Prior to an automatic landing approach, the pilot engages the glide slope mode of operation by placing the mode selection switch to GS AUTO. In this mode the autopilot receives beam displacement signals from both localizer and glide slope receiving equipment and controls the aircraft in roll and pitch for an automatic ILS (Instrument Landing System) approach. Since the aircraft is also controlled by localizer during GS AUTO mode operation the circuit configuration set in the LOC/VOR mode is also set up in the GS AUTO mode. (See figure 12.) In addition, the system is now armed through operation of GS ARM relay K6 for automatic glide slope engagement as a function of glide slope beam signal strength. (See figure 13.)

(b) While the aircraft is in the saturated portion of the glide slope beam auto engage amplifier relay K1 is energized. This completes the d-c circuit to the coil of GS ARM relay K6 through normally-open contacts of autopilot engage relay K21, normally-open contacts of localizer relay K26 (which is energized when the localizer receiver is tuned to a localizer frequency), and closed contact E of the mode selector switch. Relay K6, once energized, is self-locked through a set of its own normally-open contacts and therefore is now independent of the operation of auto engage amplifier relay K1 as a function of glide slope beam signal strength.

(c) With relay K6 energized, power is applied to the GS ARM annunciator solenoid through normally-open contacts of LOC/VOR relay K14, normally-closed contacts of GS ENGAGE relay K23 and normally-open contacts of GS ARM relay K6. Other normally-open contacts of K6 complete d-c circuits to the holding coil of the mode selector switch and the coil of aileron crossfeed relay KA2. Relay KA2 operates to introduce beam rate crossfeed signals from the aileron channel to the yaw channel for improved stability and tracking accuracy during automatic approach.

(d) Since the aircraft must intersect the glide slope beam at a predetermined altitude, the pilot sets the altitude hold mode switch to the ALT position. The aircraft now tracks the localizer beam at an appropriate altitude for engaging the glide slope beam. As the aircraft intercepts the beam and the glide slope signal decreases to approximately 50 microamperes auto engage amplifier relay K1 releases and completes the d-c circuit to the coils of GS ENG relays K20 and K23 through normally-open contacts of detent relay K4 and LOC ENG relay K22.

(e) GS ENG relays K20 and K23 are self-locked through normally-open contacts of K23 and are therefore independent of any subsequent operation of auto engage amplifier relay K1 due to fluctuations in the strength of the glide slope signal. Other contacts of relay K23 perform the following switching functions:

1) Transfer dc power from the GS ARM annunciator solenoid to the GS ENGAGE annunciator solenoid, which now indicates that the glide slope system is engaged.

2) Couple the glide slope signal at the output of modulator-limiter SA275 from desensitizing potentiometer R4 to isolationsoftener amplifier SA334 and to the elevator integrator servo loop at the input of integrator amplifier SA32. At the isolation-softener amplifier the glide slope signal is added to the elevator signal chain. The application of the 400 cps glide slope signal to the elevator integrator servo loop is done through normally-open contacts of autopilot engage relay K21. The elevator integrator output nullifies sustained or long-term beam error signals caused by wind or loading which tend to keep the airplane from the center of the glide slope beam. In addition the integrator produces a signal that cancels out the steadystate vertical gyro pitch attitude signal caused by the nosedown attitude of the aircraft on the glide slope beam. This action establishes the glide path as the pitch attitude reference during glide slope operation.

3) Increases the time-constant of aileron full-time command modifier SA402 so that the rate output is reduced from 4 degrees of roll per second to 2.5 degrees per second.

NOTE : Only N20199
Used on N19997 and N20000 if R/A INOP.

(f) When a descent to touchdown is commenced beam gain desensitization (both glide slope and localizer) and bank angle limiting begin as a function of altitude sensed by either the radar altimeter or air data sensor in conjunction with desensitizer amplifier SA419. Beam desensitization as a function of altitude minimizes the effects of glide slope beam noise and localizer beam convergence and provides for a smooth transition to the flare out maneuver. In installations that do not include a radar altimeter, beam signal attenuation and bank angle limiting are performed through the air data sensor. This is accomplished by connecting the output of the air data sensor altitude potentiometer to the input of the desensitizer amplifier through the relaxed contacts of radar altimeter relay K18, which in this system configuration is permanently de-energized. (See figures 7 and 9.) The altitude potentiometer is clutched to the air data sensor servomotor by the altitude potentiometer clutch solenoid which is energized through pulled-in contacts of glide slope relay K20 during glide slope mode operation. The air data sensor input to the desensitization amplifier drives a servo loop that positions the wipers of the localizer and glide slope desensitization potentiometers to provide signal attenuation of 80% over a 1500 feet change in altitude for glide slope (4.0% for localizer) from the time of glide slope engagement. Desensitizer amplifier SA419 also generates a bias voltage output that varies as a function of altitude. The bias voltage is coupled feet change in altitude for glide slope (4.0% for localizer) from the time of glide slope engagement. Desensitizer amplifier SA419 also generates a bias voltage output that varies as a function of altitude. The bias voltage is coupled to the limiter circuit in bank angle limiter SA33 through separate normally-open contacts of coupler relay K27 and serves to reduce the bank angle limits from 30 degrees to 10 degrees over the 1500 feet change in altitude. (See figure11). Consequently, when using the air data sensor for beam desensitization and bank angle limiting, it is Important to know at what altitude the glide slope is engaged. For example, if glide slope is engaged at 2500 feet, attenuation will cease after the aircraft has descended to 1000 feet. This will cause a very low system gain resulting in a "sloppy approach. If this occurs the system should be recycled at an altitude of 1500 feet by turning the mode selector witch to LOC/VOR then back to GS AUTO. If engagement is at an altitude lower than 1500 feet the system gain will be higher than desired resulting in an "overactive" approach.

NOTE : N19997 and N20000 only.

(g) With installatlonz that include a radar altimeter in addition, to the air data sensor the desensitization, functions are performed by the radar altimeter. This is accomplished by connecting the output of the radar altimeter to the input of desensitization amplifier SA419 through the pulled-in contacts of radar altimeter relay K18. In this system configuration relay K18 is energized through an interlock in the radar altimeter which closes automatically at an altitude of 2500 feet. In this case the radar altimeter output drives the desensitizer servo, loop to position the wipers of the beam desensitizer potentiometers and produce the bank angle limiting bias. However,jthe gain programming produced by the radar altimeter is different from that produced by the air data sensor. With the radar altimeter, beam desensitization and bank angle limiting commence at an absolute altitude of 1000 feet. As the altitude decreases below 1000 feet attenuation conmences. Localizer beam gain is reduced from optimum gain at 1000 feet to 50% of this value at 50 feet. It then remains constant at this value to touchdown. Glide slope gain is reduced from optirmum gain at 1000 feet to 32% of this value at 200 feet and then to zero gain at zero feet. The bank angle limits are reduced from 30 degrees at 1000 feet to 5 degrees at 60 feet. The 5 degree bank limit is then maintained to touchdown. An adjustable trip located In the low range radio altimeter, functions to illuminate the MDA annunciator

(h) In the event of a radar altimeter malfunction the altimeter interlock opens and releases radar altimeter relay K18. This switches the input of.the desensitizer amplifier back to the output of the air data sensor. However, the air data sensor altitude potentiometer clutch is no longer energized by operation, of glide slope engage relay K20. Therefore it is inportant to realize that In the event of an altimeter malftion beam desensitization and bank angle limiting as a function of altitude are no longer available.

In fact, because the air data sensor altitude potentiometer is spring-returned to its maximum output position when the altitude potentiometer clutch is de-energized, a radar altimeter malfunction returns the beam gains and bank angle limits to their optimum or initial values. In this event, therefore, it would probably be advisable to disengage the GS AUTO mode.

(i) GS ENGAGE relay K20, energized in conjunction with GS ENGAGE relay K23, performs the following switching functions:

1) Introduces a 28 volt dc softener bias to elevator channel isolation and softener amplifier SA334 which modifies the response of the amplifier so that rapidly fluctuating glide slope input signals are eliminated from the signal chain.

2) Transfers the pitch rate signal directly into the elevator signal chain instead of through bandpass filter SA245. This action tightens the response of the elevator channel to sudden variations in pitch attitude.

3) Connects the 26 volt ac fixed phase voltage to the elevator integrator motor so that integration of sustained or long-term stand-off errors now commences.

4) Removes dc power from ALT HOLD relay K5, which in turn releases the ALT HOLD switch and de-energizes the altitude hold solenoid, and at the same time energizes the altitude potentiometer clutch solenoid in the air data sensor. (See figure 14.) (The above only applies to installations without a radar altimeter.)

5) Connects dc power through normally-open contacts of LOC TRK relay K29 to the coil of time delay relay K12. Relay K12 closes approximately 40 seconds later to complete the circuit to the coil of LOC/TRK relay K29. Relay K29, once energized latches in position and completes the circuit modifications required for final approach to touchdown.

(j) The primary function of relay K29 is to modify the aileron channel signal structure to compensate for the effect of shearwinds during the final phase of the approach. This is accomplished by replacing the preset course heading signal with a lag roll "pseudo heading" signal. Thus, from this point on localizer damping is accomplished with lagged roll attitude signals and beam rate signals instead of the preset course heading signal. Normally-open contacts of K29 close to remove the 28 volt dc bias voltage from heading washout amplifier SA406. This reduces the output level so drastically that preset course heading deviation signals are effectively removed from the aileron signal chain. Simultaneously, another set of K29 contacts connects the roll attitude signal from the vertical gyro to the input of lag roll amplifier SA274 which has a 30-second time delay circuit. Thus the output of the lag roll amplifier provides a correction for any long-term tendency of the aircraft to bank away from the beam but does not produce any instantaneous correction signals as a result of rapid heading changes caused by varying wind forces. Therefore there is no opposition to the natural "weather-cocking" action of the airplane under varying wind conditions as would be the case with heading signal damping. However, in the event of a sustained bank angle due to a long-term beam displacement a correction lag roll signal will be generated and maintained for the remainder of the descent.

(k) A secondary function of LOC/TRK relay K29 is to initiate integration in the aileron channel during localizer tracking. This is accomplished by connecting the 26-volt a-c fixed-phase voltage to the aileron integrator motor through normally-open contacts of detent relay K4, released contacts of aileron signal switch amplifier K1, and normally-open contacts of K29. The integrator produces a signal that compensates for sustained crosswind forces. If the airplane is blown off course, the beam error signal developed causes the autopilot to change the heading of the aircraft and return it to course. If the crosswind persists, successive corrections build up enough output from the integrator to give a heading which will maintain the desired course.

(l) Signal mixing in the elevator channel during final approach therefore, consists of glide slope beam error signals attenuated as a function of altitude, integrated beam error signals, pitch attitude signals from the vertical gyro and pitch rate signals from the three-axis rate gyro. These signals form the error signal for servo control. Operation of the elevator channel beyond the isolation amplifier is in accordance with the description given for manual mode.

(m) Signal mixing in the aileron channel during final approach consists of localizer beam error signals, beam rate signals integrated beam error signals, roll attitude signals from the vertical gyro, roll rate signals from the roll rate gyro, and a lagged roll attitude signal for localizer damping. These signals form the error signal for aileron servo control. Operation of the aileron channel beyond the isolation amplifier is in accordance with the description given for manual mode.

(7) Glide Slope Manual Mode

(a) The glide slope manual mode provides ILS tracking operation similar to the GS AUTO mode with the difference that transition to glide slope engage is performed manually by operation of the mode selector switch instead of automatically as a function of glideslope beam signal strength through the auto engage amplifier.

(b) Setting the mode selector switch to GS MAN closes switch contacts P end K. (Switch contacts E and J are now open.) This bypasses auto engage amplifier relay Kl and applies power to GS ENG relays K20 and K23 throuh normally-open contacts of detent relay K4 and LOG ENG relay K22. The mode selector switch holding coil is now energized through closed contact J of the selector switch and pulled-in contacts of K20 and K23. The energizing of GS ENG relay K20 and K23 then sets up glide slope operation as in GS AUTO mode operation.

D. Altitude Hold (ALT HOLD) Control

(1) With the autopilot engaged in manual, preset heading, or localizer VOR modes, altitude control may be utilized to automatically hold the airplane at a desired altitude. In a dive or climb, the airplane will level off and return smoothly to the altitude at which the function was switched In. The ALT HOLD function can also be utilized in the GS AUTO mode, but only until the glide slope beam is engaged. At this point (activation of GS ENG relays K20 and K23) the altitude hold function is automatically disabled. To provide a smooth transition at the time of glide slope intercept, disengagement of the altitude hold control is delayed for approximately ten seconds during which period the altitude control signal is washed out. This produces a damping function which counteracts the transient attitude changes produced by the sudden introduction of a nose-up glide slope signal (just prior to beam intercept) followed by a nose-down signal (as the beam is intercepted). This damping is especially desirable when the airplane is in the approach configuration, that is, with the flaps down. Under these conditions the elevator servo is switched for maximum torque output so that transient-effects are more pronounced.

(2) Altitude hold control switching and operation is shown in figure 14. When the altitude hold switch-on the autopilot control panel is set to the ALT position, dc power is applied to the coil of ALT HOLD relay K5. The circuit is through pulled-in contacts (19-20) of A/P EM relay K21, relaxed contacts 16-15 of GS ENG relay K20, relaxed Contacts (13-12) of pitch axis disengage relay Kpa (when installed), and the contacts of the ALT hold witch. Pulled-in contact of ALT HOLD relay K5 perform the following functions:

a) Contacts (7-8) energize ALT relay K41, the altitude autosyn clutch solenoid in the air date sensor through pulled-in contacts (7-2) of K41, and ALT relay K42 through relaxed contacts (7-1) of ALT T1ME DELAY relay K43. Pulled-in contacts (3-6) of ALT relay K42 provide an alternate holding path for autosyn clutch solenoid that is independent of ALT relay K41.

(b) Contacts (4-5) energize the altitude switch holding coil, and simultaneously disconnect the pitch trim clutch solenoid in the autopilot controller.

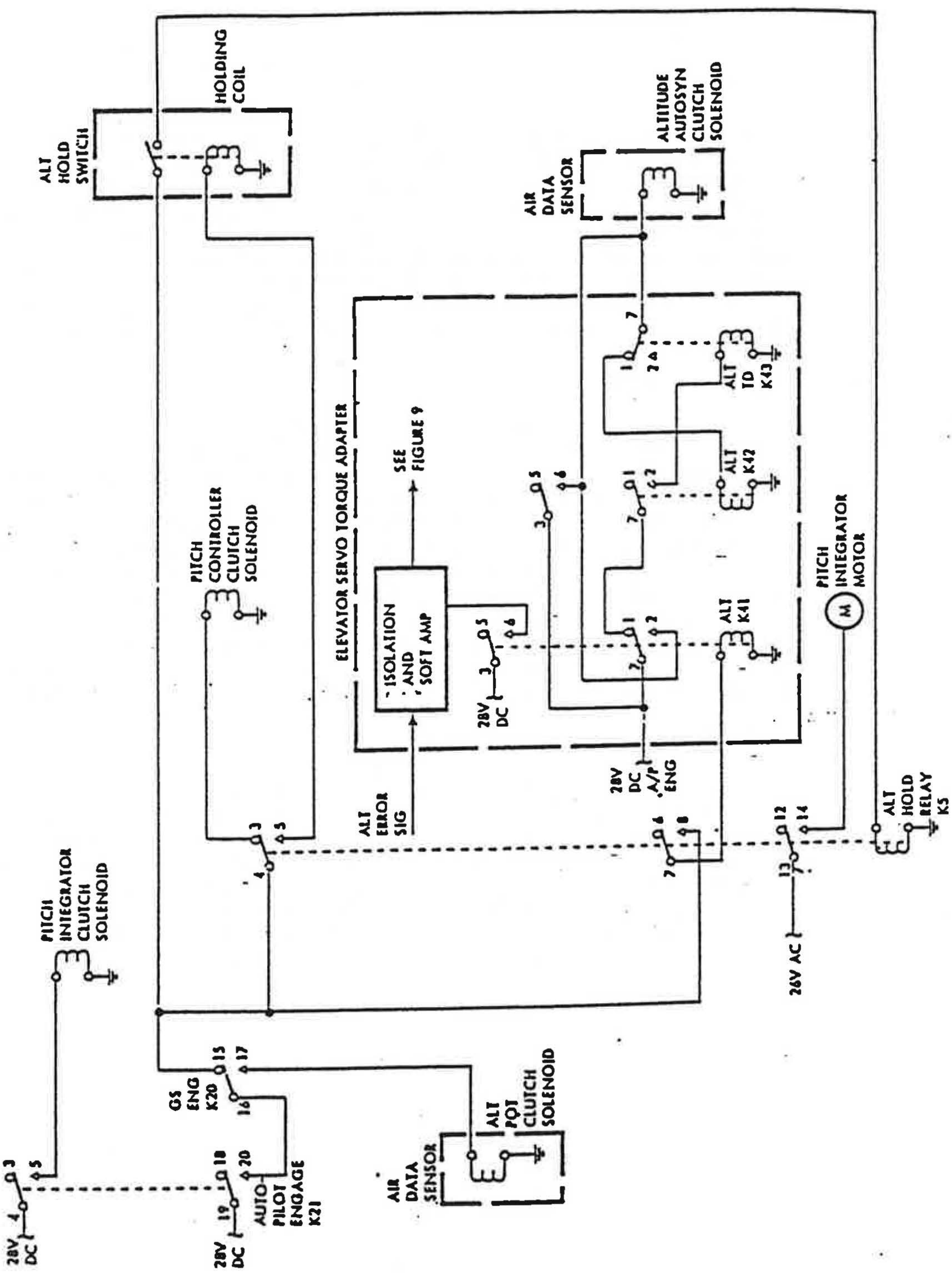
(c) Pulled-in contacts 13-14 apply fixed-phase ac excitation voltage to the pitch integrator motor.

(3) Energizing GS ENG relay K20 or de-energizing A/P ENG relays K21 or K7 removes power to the altitude switch holding coil and the switch returns to its off position. The altitude switch may also be manually returned to its off position. This removes excitation to ALT HOLD relay K5 which in turn de-energizes ALT relay K41. Relaxed contacts (7-1) of K41 connect dc power to the coil of ALT TIME DELAY relay K43 through held-in connects (7-2) of ALT relay K42. Relay K43 becomes energized approximately 10 seconds later and interrupts the circuit to the coil of K42. Pulled-in contacts (3-6) of relay K42 open and remove excitation to the autosyn clutch solenoid which disengages altitude hold control.

(4) The altitude hold function controls elevator channel operation only. The altitude error signal from the clutched-in altitude autosyn in the air data sensor is placed on an altitude-gain adjust potentiometer, also in the air data sensor, where its level is adjusted as a function of dynamic air pressure "Q," or indicated airspeed. The reason for this adjustment is to provide greater signal sensitivity at lower airspeeds, thus giving optimum response over the full range of airspeeds. The gain-adjusted signal is coupled through the isolation and softener amplifier in the torque adapter box to the input of isolation amplifier SA334; and through relaxed contacts (7-6) of GS ENG relay K23, to the input of integrator amplifier SA32. The output of the integrator is summed with the altitude error signal and the versine up-elevator signal at the input of isolation amplifier SA334.

(5) The torque adapter isolation amplifier provides the washout function referred to in paragraph (1). During ALT hold operation pulled-in contacts (3-6) of ALT relay K41 modify the time constant of the isolation amplifier so that altitude control signals are passed without delay. When the ALT mode is disengaged, ALT relay K41 is de-energized and contacts (3-6) open. This increases the time-constant of the amplifier to approximately 10 seconds. As the autosyn remains clutched-in for approximately 10 seconds after disengagement due to the operation of time-delay relay K43, the altitude error signal is washed out over this period.

THIS PAGE INTENTIONALLY LEFT BLANK



Altitude Hold Control Switching Circuit
Figure 14

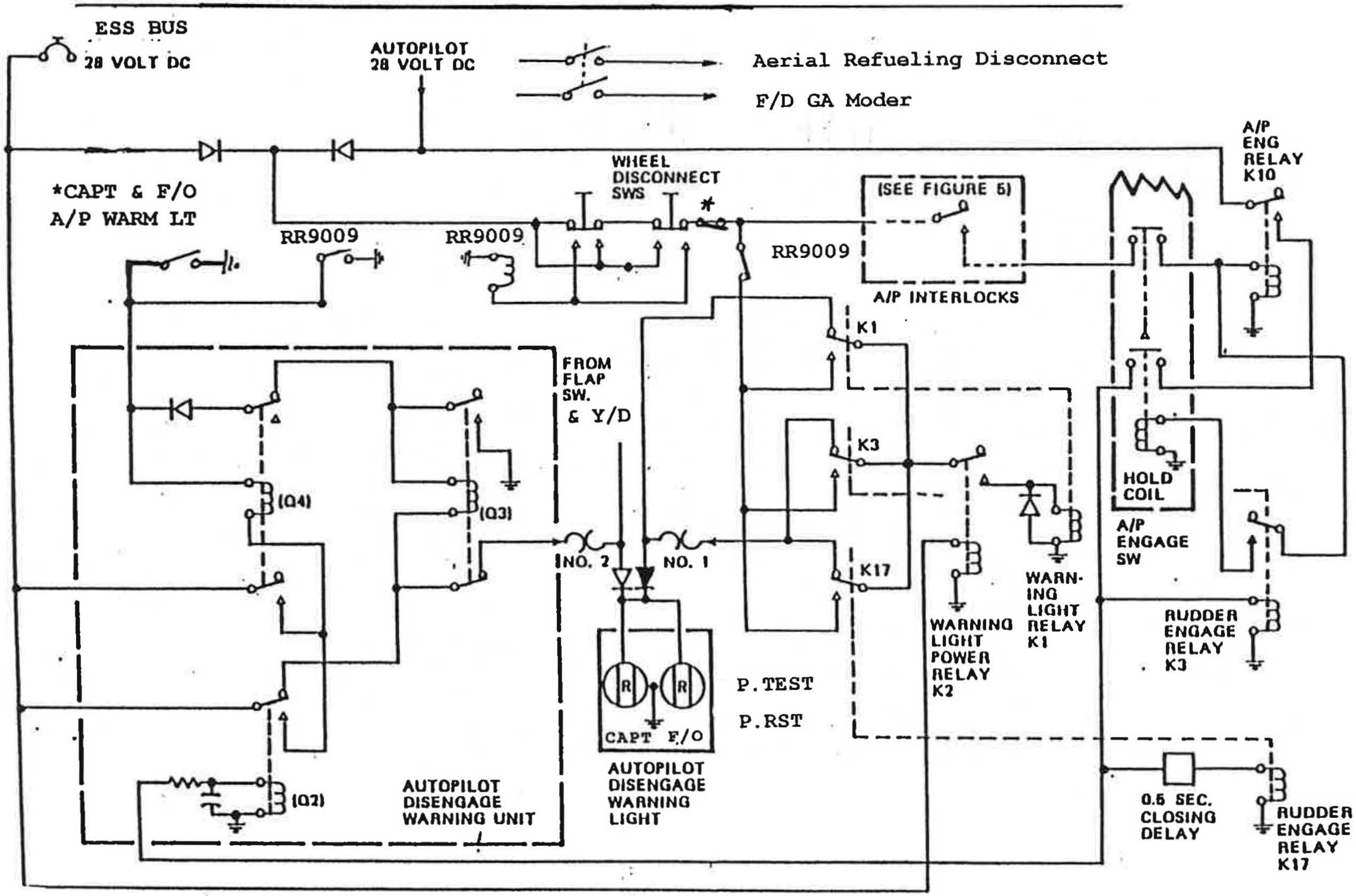
(6) The altitude error signal effectively replaced the pitch command signal used in manual mode since the latter is declutched from the controller pitch wheels and zeroed for altitude hold control. Thus, if the airplane should nose-down slightly, the air data sensor will send a nose-up command as it loses altitude. However, the vertical gyro will not sense a small nose-down attitude and will not send a nose-up command. Should the nose-down tendency persist, the integrator will in time, build up a steady signal to correct this condition. If in level flight the airplane should drop in altitude, the altitude error developed in the air data sensor will command nose-up. Should this nose-up attitude be large enough for the vertical gyro to sense, it will send a nose-down command to mix with the isolation amplifier output. When these two signal balance each, other the airplane will again be holding the desired altitude.

E. Warning Light Operation

(1) The warning light system consists of two press-to-test and press-to-reset warning lights arranged in a circuit such that disengagement of the autopilot due to an open interlock, loss of autopilot dc power, disengagement by means of the wheel disconnect switches or placing the autopilot engage switch to the OFF position causes both lights to flash; loss of Ess dc power, does not impair operation of the autopilot but is indicated by steady illumination of lights which, should an automatic disconnect subsequently occur, then extinguishes. Failure of either flap switch does not impair operation of the autopilot but is indicated by steady illumination of lights. (See figure 15.) Setting the YAW DAMPES SELF TEST switch out of this OFF position, causes the steady illuminationat both A/P WARN LT.

(2) WARNING light operates as follows:

(a) Airplane 28 volt Ess dc power energizes warning light powerrelay K2 when the airplane dc bus is turned on. With the autopilot engage switch set to ENGAGE, rudder engage relays K3 and K17 are energized, completing the circuit from the autopilot and Ess 28 volts dc to the coil of warning light relay K1. K1 is then self-locked through its own pulled-in contacts and cannot be relaxed until either the wheel disconnect switches are actuated or Ess dc power is lost or removed.



Warning Light Circuit
Figure 15

(b) When an interlock opens or the autopilot is disengaged by means of the autopilot engage switch or loss of autopilot d-c power occurs, rudder engage relays K3 and K17 open and the airplane and/or autopilot dc are connected to lights. through flasher #1, the relaxed contacts of K3 and K17, the pulled-in contacts of K1 and the wheel disconnect switches - causing lights to flash until either of the wheel disconnect switches or WARN LT is actuated to remove power from warning light relay K1.

(c) If Ess d-c power to the autopilot is lost, warning light power relay K2 opens and releases warning light relay K1. Autopilot d-c power is then applied continuously to the warning light through the wheel disconnect switches and WARN LT the pulled-in contacts of rudder engage relays K3 and K17, and the relaxed contacts of warning light relay K1. Thus, continuous illumination of lights indicates the loss of Ess d-c power although operation of the autopilot is not affected by this condition.

(d) When the Ess dc bus is connected warning lights flashes since voltage is available through the relaxed contacts of relays (Q2) and (Q3) and flasher #2. Depressing either WARN LT or wheel switch extinguishes lights since relay (Q3) closes and holds, removing voltage from the flasher. (Relay (Q3) is provided with power through the relaxed contacts of (Q2) and a ground through, the pulled-in contact of (Q3). Engaging the autopilot applies 28 volts de to relay(Q2) through the autopilot interlock circuit and engage witch. (Q2) closes and removes power from (Q3) allowing(Q3) to open. The circuit is now armed such that an open autopilot interlock, loss of autopilot d-c power, or disengagement of the autopilot through the autopilot engage witch causes lights to flash. Lights flashes because these three disconnect conditions all result in the release of relay (Q2) and the application of voltage to lights through the relaxed contacts of (Q2), (Q3) and flasher #2. Depressing either WARN LT or flasher #1 wheel disconnect switch extinguishes both lights. In the case of flasher #1 wheel disconnect switches place a ground on the coil of relay (Q3) causing it to close and hold through its own pulled in contacts and thus disconnect power from the lights. In the case offlasher #2 the wheel disconnect witches remove voltage from lights by de-energizing warning light relay KI.

(e) When the autopilot is disengaged by means of the wheel disconnect switches, either intentionally or inadvertently, relay (Q4) is provided with a ground. Simultaneously, excitation is removed from, the coil of relay (Q2). However, due to the time delay circult across the coil of (Q2) the relay does not release imedlately thus relay (Q4), which is still receiving 28 volts dc through the pulled-in contacts of (Q2), closes. This prevents relay (Q3) from energizing when (Q2) times out i.e. the ground return of (Q3) is interrupted by the pulled-in contacts of (Q4). Consequently when relay (Q2) does relax the circuit to lights is comlete through its own relaxed contacts and those of(Q3), causing lights to operate.

AUTOPILOT SYSTEM - REMOVAL/INSTALLATION

1. General

A. The Digital Amplifier Unit is designed for installation and removal from the aircraft without the need of special tools. Installation of the unit does not require any special pre-installation checkout, alignment or adjustment procedures. However, the Digital Amplifier Unit and the Control Panel must be installed as a pair in the aircraft.
Prior to installation and removal of the Digital Amplifier Unit, a visual inspection of the aircraft mounting rack, mating electrical connectors, and associated hardware shall be conducted.

2. Removal of Digital Amplifier Unit

A. Open and tag AUTOPILOT and A/P & MACH TRIM (DISENG LT) circuit breakers located on P5 overhead circuit breaker panel.

B. Press release triggers on handles and pull lever latches to open (horizontal) position.

WARNING: THE DIGITAL AMPLIFIER UNIT WEIGHS APPROXIMATELY 38 POUNDS. TWO PERSONNEL ARE REQUIRED TO REMOVE THE UNIT FROM THE RACK.

C. Grasp handles and pull unit out of rack until engagement cam clears the mounting surface.

D. Close and lock the lever latches and remove the Digital Amplifier Unit from the rack.

E. Install conductive end caps on the three rear connectors of the Digital Amplifier Unit.

3. Empty Rack Inspection

A. The visual inspection shall verify the following:

(1) No aircraft rack deformation.

(2) Aircraft electrical connectors in good condition and properly aligned in rack.

(3) No missing or loose parts.

(4) Forced air cooling gaskets in good condition, no cuts, not buckled, properly secured to rack.

4. <u>Installation of Digital Amplifier Unit</u>

A. To install the Digital Amplifier Unit, refer to Figure 1 (Sheet 1 and 2.) and proceed as follows:

B. Remove the conductive end caps from the three rear connectors of the Digital Amplifier Unit.

C. Press release triggers on handles and pull lever latches to open (horizontal) position.

<u>WARNING</u>: THE DIGITAL AMPLIFIER UNIT WEIGHS APPROXIMATELY 38 POUNDS. TWO PERSONNEL ARE REQUIRED TO INSTALL THE UNIT IN THE RACK.

D. Slide the Digital Amplifier Unit back on the rack until the rear connectors are partially engaged.

<u>CAUTION</u>: IMPROPER ADJUSTMENT OF LATCH ASSEMBLIES MAY CAUSE CIRCUIT MAL-FUNCTION OR EQUIPMENT DAMAGE.

E. Engage lever latch hook with fork assembly pin.

F. Start closing lever latch handle.

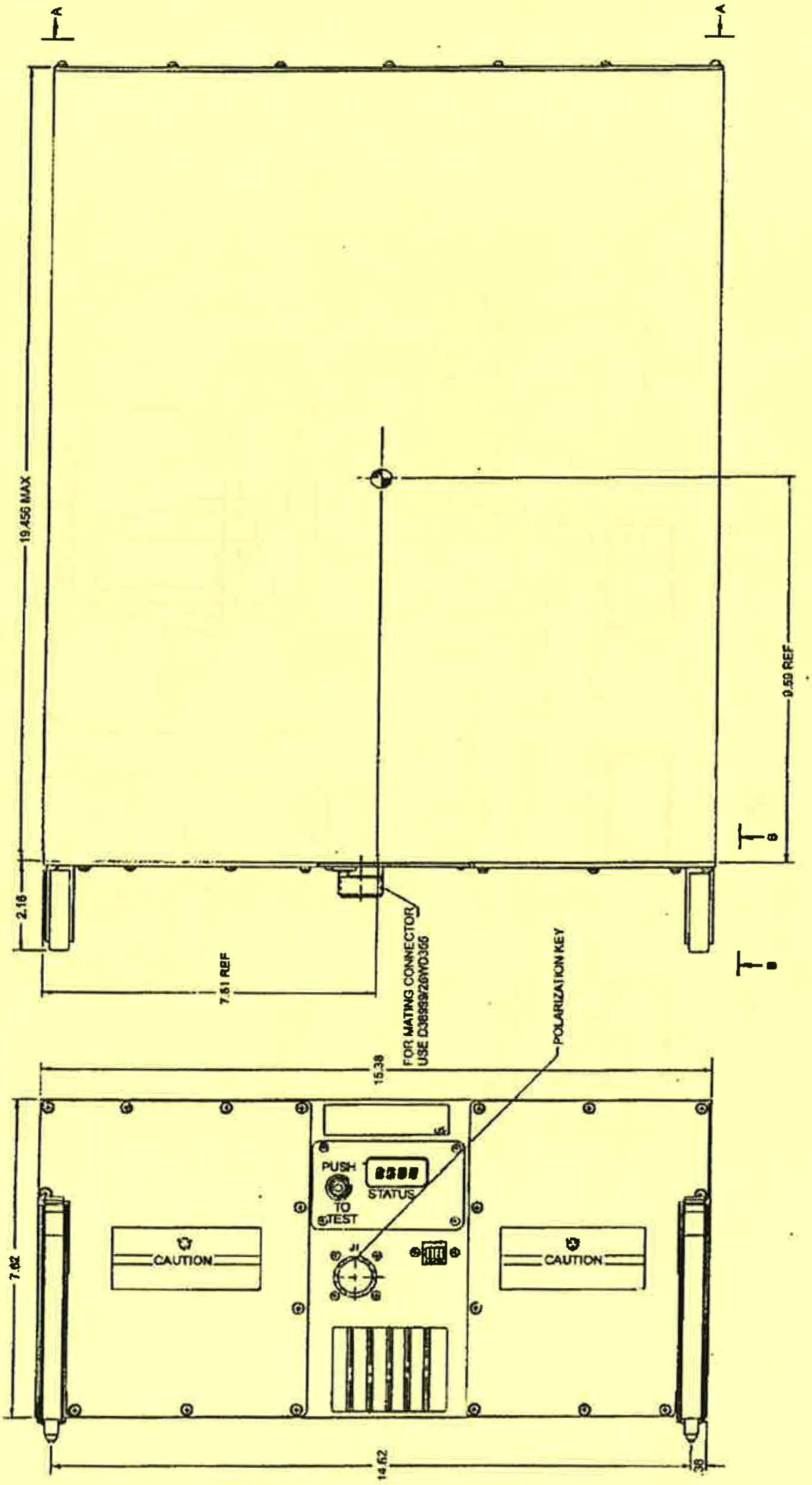
<u>NOTE</u>: It is possible to determine full connector engagement by feel. A sudden increase in handle pressure or resistance to handle movement indicates that the connectors are engaged.

G. Adjust fork assembly until resistance to handle movement occurs at the gap tolerance shown in Fig. 1 (Sheet 2 of 2).

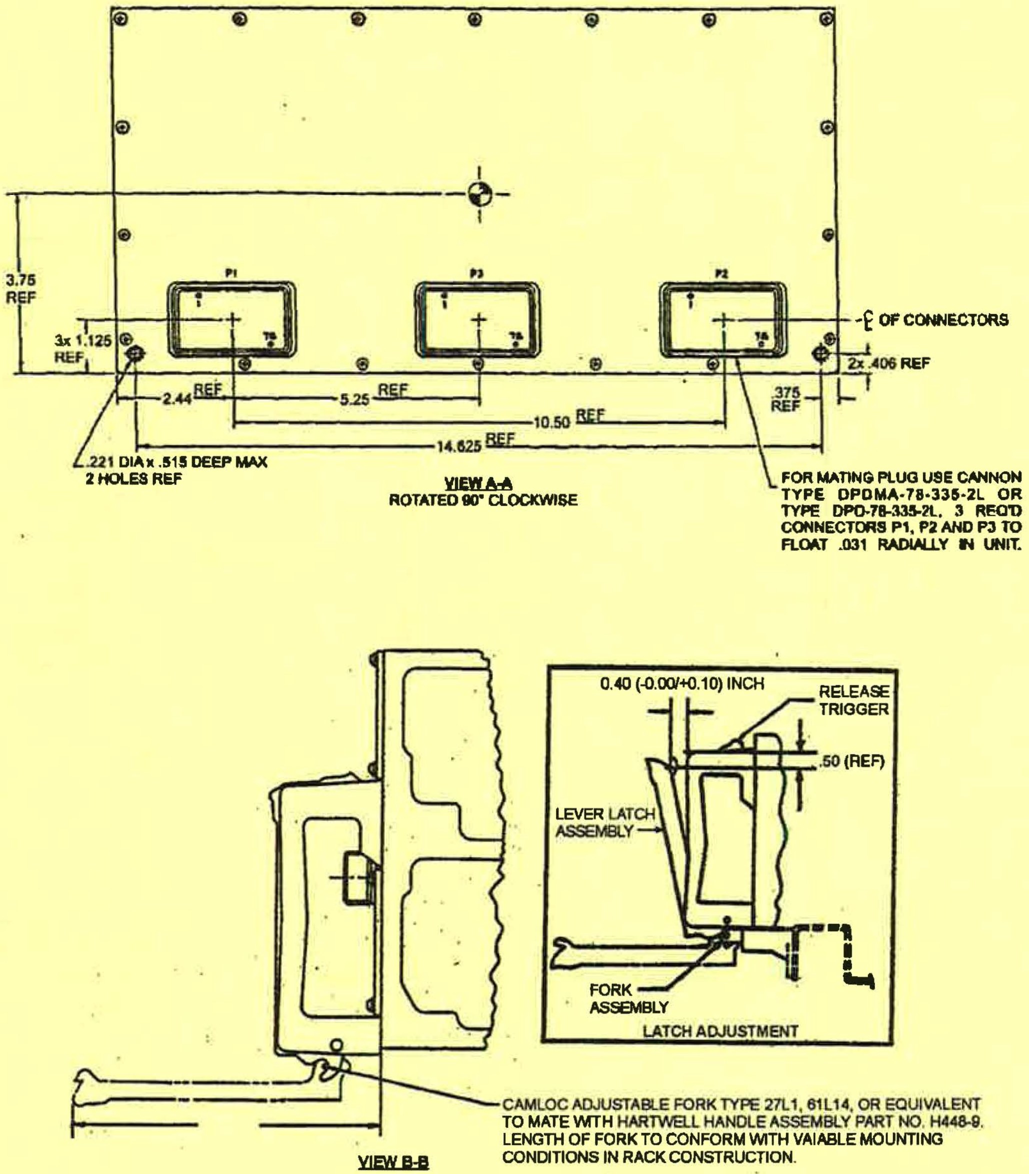
H. Close the handle until it is latched.

I. Remove tag and close AUTOPILOT and A/P & MACH TRIM (DISENG LT) circuit breaker located on P5 overhead circuit breaker panel.

IICN: 1P-AMMX-22-0030-01



Digital Amplifier Unit
Fig. 1 (Sheet 1 of 2)



IICN: 1P-AMMX-22-0029-01

Digital Amplifier Unit
Fig. 1 (Sheet 2 of 2)

5. Digital Amplifier Unit Operating Instructions

A. The operating controls and indicators for the Digital Amplifier Unit are listed and described in table 1 and identified in fig. 2. The BIT Display Module is the only Digital Amplifier Unit assembly that contains controls and indications.

(1) On-Board Initiated or In-Flight. This Built-In-Test (BIT) is performed in the aircraft either automatically at power on (or following a power outage of 75 milliseconds or more) or on the ground as initiated by the PUSH TO TEST switch on the Digital Amplifier Unit.
Annunciation is limited to a PASS/FAIL indication and refusal of the Autopilot to engage or stay engaged.

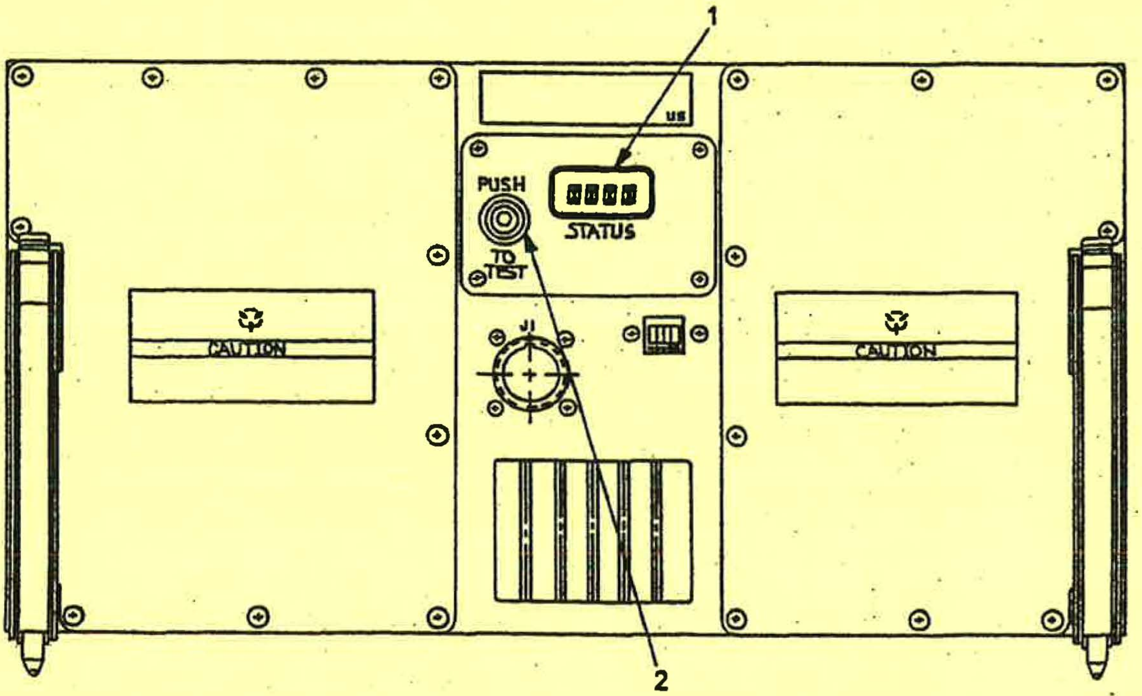
B. The BIT Display Module PUSH TO TEST switch (see fig. 2) is used to step through BIT functional test and to interrogate results stored in the Digital Amplifier Unit Non-Volatile Memory (NVM) for display on the STATUS Display.

(1) ON-BOARD INITIATED BIT results in a PASS/FAIL display for the digital Amplifier Unit, preceded by a readout of up to ten logged in-flight failures of associated aircraft units. Table 2 shows the sequence of tests and status displays for On-Board Initiated BIT. Note that the first depression of the PUSH-TO-TEST switch will result in one of four possible indications, either an instruction to disengage the Autopilot, a blank display indicating the need to press the PUSH-TO-TEST switch again, "END" if no sensor failures stored, or readout of the first stored sensor failure. The Table 3 is a cross reference of STATUS Display readouts associated with the various failed external units. Following readout of the last failure (up to ten failures may be stored), the STATUS Display indicates "END". The Non-Volatile-Memory (NVM) is erased upon readout. The PUSH-TO-TEST switch must then be depressed within ten seconds to perform the Digital Amplifier Unit On-Board Initiated BIT. Once BIT has started, the STATUS Display will step through eleven status indications to indicate the progress of the BIT. The final result will be a "PASS" or "FAIL" indication.

Index No.	Control or Indicator	Description
1	STATUS Display	Provides an alphanumeric display of each failed CCA, the failed components of the CCA and the BIT test that failed.
2	PUSH TO TEST switch	Provides the means to step through a series of BIT test in sequence.

Digital Amplifier Unit, Bit Display Module, Controls and Indications
Table 1

IICN: 1P-AMMX-22-0027-01



Digital Amplifier Unit, Bit Display Module, Controls and Indications
Fig. 2

PUSH TO TEST Switch	STATUS Display	FUNCTION / ACTION
1st press	APON	Disengage A/P
	Blank, 1st failure, or END if no failures in NVM (See Table 3 for codes)	Repeat 1st press
2nd through 10th press	2nd through 10th failure or END if fewer than 10 failures	Display will blank 10 seconds after END if PUSH TO TEST Switch not pressed.
11th press	END	Indicates end of NVM read. Display will blank in 10 seconds if PUSH tO TEST switch is not pressed.
1st press after end *	STR 10 20 30 40 50 60 70 75 80 90 100	Start of BIT BIT test time is divided into eleven parts. Time segments are displayed as tests are performed.
	PASS/FAIL	Result of test.
Last press	Blank or blanks in 10 seconds if PUSH TO TEST Switch not pressed	Indicates Digital Amplifier Unit is out of BIT mode.

* If FNVM is displayed instead of STR, the NVM has not been cleared; check unit.

NVM Read and ON-Board Initiated BIT test
Table 2

STATUS Display Fault Indication	Failed Unit
RDAC	DAC Roll Attitude
RXTM	Three Axis Rate Transmitter
DDAC	DAC Heading
PDAC	DAC Pitch Attitude
DAD1	Digital Air Data Computer (Pri)
DAD2	Digital Air Data Computer (Alt)
PCTL	Control Panel Pitch Controller
DAU	Digital Amplifier Computer Unit
ESRV	Elevator Servo
ASRV	Aileron Servo
Trim	Auto Stabilizer Trim Servo
Ralt	Radio Altimeter
IRS1	Internal Reference System (Pri)
IRS3	Internal Reference System (Alt)
FMC1	Flight Management System (Pri)
FMC2	Flight Management System (Alt)
ATSC	Autothrottle System Computer
CNTR	Control Panel (Arinc 429)
SPR1	Spare (Pri)
SPR2	Spare (Alt)

NVM Display Data Cross-Reference for On-Board Initiated BIT
Table 3

END

AUTOPILOT SYSTEM - ADJUSTMENT/TEST

1. Tools and Equipment

A. Table 501 lists tools and test equipment required for flightline maintenance of the autopilot system.

PART NO. OR TYPE DESIGNATOR	USE AND APPLICATION	NOMENCLATURE
TOOLS		
F52485-500 Boeing	Protractor	Measure control wheel angles.
F72790 Boeing	Protractor Mount	Mounts protractor to control wheel.
L-30M Amtec/Hunter Spring	30-pound Spring Scale	Measure control column forces.
F70183-500 Boeing	Fixture — Servo Torque Measuring	Measure aileron and elevator torque motor torque.
F70183-501 Boeing	Stabilizer Trim Lock Assembly	Disables stabilizer trim manual trim wheel.
TEST EQUIPMENT		
2TSJ60B00002 Boeing	Tilt Table — INS and Gyro Units	Used for tilting INS unit.
204-91950-1 Boeing	Adapter — Tilt Table	Provides interface between INS and tilt table and tilt table and aircraft.
GGS-764	Stop Watch	Time control responses.
AN/USM 341	Digital Multimeter	Measure servo ac voltages.
Model 260 Simpson	Analog Volt Ohmmeter	Measure ac and dc and continuity voltages.
AN/ASN 186Corad 28845	Ramp Test Set	Provides test signals for VHF NAV system.
T-30 B/B Tel-Instrument Corp	Ramp Test Set	Provides test signals for VHF NAV system.
980N-1 Collins Radio	Altimeter Test Set	Provide radio test altitude signal.

Autopilot System Tools and Test Equipment
Table 501 (Sheet 1 of 2)

PART NO. OR TYPE DESIGNATOR	USE AND APPLICATION	NOMENCLATURE
AN/ARM 25C	TACAN Test Set	Provide TACAN test signal.
TTU-205 C/E	Pressure Temperature Test Set	Provide altitude and airspeed inputs.
204-91920-1	Adapter-Pitot System	To connect test set to pitot probes.
33410-125-375 T	Adapter-Static System	To provide closure for aircraft static ports and connect test set.
Equivalent tools or test equipment are authorized for use.		

Autopilot System Tools and Test Equipment
Table 501 (Sheet 2 of 2)

2. Checkout

CAUTION: OBSERVE ALL REQUIREMENTS FOR COOLING AIR AS SPECIFIED IN THE CHECKOUT PROCEDURES. OTHERWISE, EQUIPMENT MAY BECOME DAMAGED BECAUSE OF OVER HEATING.

A. General

(1) Unless otherwise specified, autopilot is engaged in MAN mode. Establish communications between crew members. The checkout procedure is divided into individual tests that may be performed in sequence or on an individual basis as required. If complete test is to be performed, paragraphs B. (1) ,B. (2), B. (3) and B. (4) must be performed before starting test. In this case, step of the individual tests need not be performed. In any checks requiring the control wheel to be centered, the allowable tolerance is ±6 degrees, unless otherwise specified.

(2) The Restore Aircraft to Normal Procedure, paragraph C, must be performed after every test sequence if, no other tests are to be performed.

B. Preparation

(1) Autopilot Activation

(a) Apply electrical power to aircraft and turn on draw through cooling air.

(b) Establish communication between test personnel.

(c) If tilt table is required, perform paragraph B. (3)

WARNING: ENSURE THAT ALL PERSONNEL ARE CLEAR OF CONTROL SURFACES, CONTROL CABLES, SERVO MECHANISMS AND OTHER MOVABLE PARTS WHEN APPLYING OR REMOVING HYDRAULIC POWER, OR WHEN CONDUCTING TESTS THAT RESULT IN SURFACE MOVEMENT. UNEXPECTED MOVEMENT OF CONTROL SURFACES MAY CAUSE INJURY.

(d) Apply hydraulic power to aircraft.

(e) Close following circuit breakers on overhead circuit breaker panel P5:

NOTE: LX-N19997 and LX-N20000 only.

ESSENTIAL FLIGHT INSTR BUS

AUTOPILOT
A/P & MACH TRIM DISENG LT DC
F/O COMPTR NO 1 (AC)

ESSENTIAL RADIO BUS

SERIES YAW DAMPER (AC)

NOTE: LX-N20199 only

FLIGHT INSTR BUS NO 2
A/P & MACH TRIM DISENG LT DC

ESSENTIAL FLIGHT INSTR BUS
AUTOPILOT
FLT INST NO 1 FD NO 1

ESSENTIAL RADIO BUS
SERIES YAW DAMPER (AC)

(f) Set YAW Switch to ON on overhead panel Yaw Damper Control Panel.

(g) If the aircraft is equipped with Autopilot Attitude Source Select switch, set ADI switch on pilot instrument panel to VG.

(h) Set FLAP lever to FLAP UP position.

(i) Set IND LIGHT switch on center instrument panel to BRT.

(j) When necessary to engage the autopilot during checkout, perform the following substeps as necessary:

1 Check that Autopilot Mode Select Switch on AUTOPILOT CONTROLLER is in MAN position.

2 Set TURN Knob to center detent on AUTOPILOT CONTROLLER.

3 Set STAB TRIM AUTOMATIC switch on control stand to NORMAL.

4 Set Autopilot Engage Switch on AUTOPILOT CONTROLLER to ON.

NOTE: The autopilot is disengaged when the AUTOPILOT switch on the AUTOPILOT CONTROLLER returns to OFF. The AP WARN light will flash until reset by pressing the autopilot disconnect switch on either control wheel or by pressing the AP WARN light pushbutton lens assembly (on aircraft equipped with this type pushbutton light assembly). The autopilot warning horn will sound when the AP WARN light is flashing.

5 Set two SPOILERS position switches on overhead panel to OFF.

(2) VOR Navigation System Activation

(a) Activate No. 1 VOR navigation system as follows:

(b) Check that following circuit breakers on overhead circuit breaker panel are closed:

NOTE: LX-N19997 and LX-N20000 only.

ESSENTIAL FLIGHT INSTRUMENT BUS	FLIGHT INSTRUMENT BUS NO 2
BRG EXC (AC)	XMFR NO 2 (AC)
HDG EXC (AC)	COMPASS NO 2 (AC)
F/O COMPTR NO 1 (AC)	BRG EXC (AC)
VERT GYRO NO 1 (AC)	HDG EXC (AC)
NAV XFER RLY NO 1 (DC)	F/O COMPTR NO 2 (AC)
XMFR NO 1 (AC)	NAV XFER RLY NO 2 (DC)
COMPASS NO 1 (AC)	
ADC 1 (DC)	
ADC 2 (DC)	

ESSENTIAL RADIO BUS	RADIO BUS NO 2
VOR NO 1 (AC)	VOR NO 2 (AC)
VOR NO 1 (DC)	VOR NO 2 (DC)
GS NO 1 (DC)	GS NO 2 (DC)
IU NO 1 (DC)	IU NO 2
	ADC XFER

NOTE: LX-N20199 only.

ESSENTIAL FLIGHT INSTRUMENT BUS	FLIGHT INSTRUMENT BUS NO 2
BRG EXC (AC)	XMFR NO 2 (AC)
HDG EXC (AC)	COMPASS NO 2 (AC)
COMPTR NO 1 (AC)	BRG EXC (AC)
VERT GYRO NO 1 (AC)	HDG EXC (AC)
NAV XFER RLY NO 1 (DC)	COMPTR NO 2 (AC)
XMFR NO 1 (AC)	NAV XFER RLY NO 2 (DC)
COMPASS NO 1 (AC)	
ADC 1 (DC)	
ADC 2 (DC)	

ESSENTIAL RADIO BUS	RADIO BUS NO 2
VOR NO 1 (AC)	VOR NO 2 (AC)

VOR NO 1 (DC)	VOR NO 2 (DC)
GS NO 1 (DC)	GS NO 2 (DC)
IU NO 1 (DC)	ADC XFER (DC)
	IU NO.2 (DC)

(c) Verify that NAV receivers are OFF, Pilot NAV MODE switch in VOR/LOC and that NAV and GS flags on HSI, GS flag on ADI, are all in view.

NOTE: Whenever VHF navigation receiver is not required and no localizer or VOR signal is needed, receiver shall be detuned from any local station.

(d) Set pilot RMI selector on main instrument panel to TACAN/VOR.

(e) Set pilot NAV MODE selector to VOR/LOC.

(f) Check that BRG and HDG circuit breakers on flight instrument accessory units are closed.

(g) Make the following settings on both compass controllers located on overhead panel:

1 Select aircraft hemisphere.

2 Set MODE selector to DG.

3 Slew pilot HDG knob on pilot HSI compass card to 100 degrees heading under lubber line.

(h) Set control switch on No. 1 VHF NAV control panel to PWR. Check that dial lights come on.

(3) Vertical Gyro Installation on Tilt Table

CAUTION: VERTICAL GYRO MUST BE OFF FOR A MINIMUM OF 20 MINUTES BEFOR REMOVING FROM RACK TO PREVENT DAMAGE TO THE GYROS.

Remove Vertical Gyro 1 navigation unit from rack and install on tilt table as follows:

NOTE: Align Vertical Gyro to aircraft heading

(a) Set Mode Selector Switch on INs MODE SELECT to OFF if applicable.

(b) Open following circuit breakers on P5 circuit breaker panel:

ESSENTIAL FLIGHT INSTRUMENT BUS
VERT GYRO NO 1 (AC)

WARNING: NAVIGATION UNIT WEIGHS APPROXIMATELY 53 POUNDS. USE TWO PERSONNEL WHEN REMOVING NAVIGATION UNIT.

CAUTION: ALLOW 20 MINUTES FOR GYROS TO RUN DOWN BEFORE REMOVING UNIT OR GYRO AND ACCELEROMETER DAMAGE MAY RESULT. USE CARE DURING REMOVAL TO PREVENT DAMAGE TO AIRFLOW SEAL.

(c) Connect tilt table to Vertical Gyro rack connector using extension cables.

(d) Install tilt table adapter on tilt table.

(e) Install VERTICAL GYRO on tilt table.

(f) Level tilt table in pitch and roll 0 (±1) degree.

(4) Vertical Gyro Activation

(a) Activate both (if required) Vertical Gyro systems as follows:

1 Check that ESSENTIAL FLIGHT INSTRUMENT BUS circuit breaker is closed.

(b) Check that following circuit breakers on P5 are closed:

ESSENTIAL FLIGHT INSTRUMENT BUS	FLIGHT INSTRUMENT BUS NO 2
XMFR NO 1 (AC)	VERT GYRO NO2 (AC)
BRG EXC (AC)	COMPASS NO 2 (AC)
ATT EXC (AC)	XMFR NO 2 (AC)
HDG EXC (AC)	ATT EXC (AC)
COMPASS NO 1 (AC)	BRG EXC (AC)
VERT GYRO NO1 (AC)	
ADC 1 (DC)	
ADC 2 (DC)	

ESSENTIAL RADIO BUS	RADIO BUS NO 2
INS NO 1 PWR (AC)	INS NO 2 PWR (AC)
INS NO 1 HTR (AC)	INS NO 2 HTR (AC)
IU NO.1 (DC)	IU NO 2

(5) BIT Test

BIT Test shall be performed to verify the operational status of the Digital Amplifier Unit. Before DAU BIT test can be initiated, any autopilot system faults stored in DAU non-volatile memory must be read out and cleared from memory. The following tests shall be performed in the aircraft environment utilizing ground power.

(a) Cockpit Setup for BIT Test

Perform the following cockpit setup procedure prior to executing DAU BIT test:

1 NAV REC No. 1 should be turned OFF.

2 NAV SELECT MODE should be VOR/LOC.

3 DADC and Vertical Gyro power should be applied.

4 Radio Altimeter should be turned ON.

5 PSH bug on pilot HSI should be positioned under the lubber.

6 Course Pointer on pilot HSI should be positioned under the lubber.

7 TURN Knob should be positioned in center detent.

8 Heading and Course knobs and controls should not be moved during BIT execution.

(b) Failure Memory Read/Clear

1 Activate autopilot and vertical gyro system. Refer to paragraphs B. (1) and B. (4).

2 With autopilot disengaged, momentarily depress and release the TEST button on the DAU front panel until the BIT display reads END.

3 Wait approximately 10 sec. after END is displayed and verify that the BIT display becomes blank.

(c) BIT Inhibit

1 Activate autopilot and vertical gyro system. Refer to paragraphs B. (1) and B. (4).

2 Engage autopilot and verify that the DAU BIT display is blank.

3 With the autopilot engaged, momentarily depress the TEST button on the DAU front panel.

4 Verify that the DAU BIT display reads APON, indicating autopilot ON, which inhibits BIT execution.

5 Disengage autopilot and verify that the DAU BIT display is blank.

(d) BIT Execution

Momentarily depress and release the TEST button on the DAU front panel. While END is displayed, momentarily depress the TEST button one time. The BIT display will sequence through the following readouts.

STR
10
20
30
40
50
60
70
75
80
90
100
PASS

NOTE: The DAU might store old failures. If FAIL is annunciated on the DAU display, repeat the BIT test. It will clear memory and at the end of BIT test the display should read PASS.

(e) BIT Exit

Verify that the BIT display automatically goes blank approximately 10 sec. after PASS is displayed. Record faults as required.

(6) Engage Interlocks and Warning

(a) Autopilot Engagement

NOTE: Whenever the control wheel is to be displaced, care should be exercised to assure travel is not restricted by the control system being up against its stops at either end of travel. Prior to each test requiring aileron control tab displacement, the control wheel shall be turned (CW & CCW) to its maximum possible position in each direction. The three axis trim indicator shall show a trimmed condition. If there is an out-of-trim indication when the wheel stops, the wheel should be manually moved until in-trim is indicated.

1 If required, activate autopilot and vertical gyro system. Refer to paragraphs B. (1) and B. (4).

2 Set STAB TRIM switches on control stand to NORMAL.

3 Set autopilot MODE Select Switch to MAN on AUTOPILOT CONTROLLER.

4 Set TURN Knob on AUTOPILOT CONTROLLER to detent.

5 Set Autopilot Engage Switch to AUTOPILOT on AUTOPILOT CONTROLLER.

6 Check autopilot engagement by noting increased effort required to move control column and control wheel. AP WARN lights are off.

7 Rotate TURN Knob on AUTOPILOT CONTROLLER clockwise and counterclockwise; control wheel rotates in direction that TURN Knob moves.

8 Rotate pitch wheel on AUTOPILOT CONTROLLER in nose UP and nose DOWN direction; control column moves aft and forward.

(b) Autopilot Disengagement and Warning

1 If required, activate autopilot and vertical gyro system. Refer to paragraph B. (1) and B. (4).

2 With autopilot engaged, momentarily press autopilot Disconnect Switch on Pilot Control Wheel; autopilot disengages and control column and control wheel move normally. The A/P WARN lights shall not begin flashing.

3 Engage autopilot and repeat step 2 using disengage switch on copilot control wheel.

4 With autopilot engaged, set Autopilot Engage Switch to OFF on AUTOPILOT CONTROLLER. AP WARN lights flash red and the AP WARN horn is active. Turn lights and horn OFF by momentarily pressing on the light cap.

5 With autopilot engaged, open AUTOPILOT circuit breaker on overhead circuit breaker panel. Autopilot Engage Switch returns to OFF and AP WARN lights flash red. The AP WARN horn shall be active.

6 Close AUTOPILOT circuit breaker and engage autopilot; AP WARN lights continue to flash. The AP WARN horn continues to be active.

7 Open and close A/P AND MACH TRIM DISENGAGE LT (DC) circuit breaker; AP WARN lights go off. The AP WARN horn shall be inactive.

(7) Vertical Gyro Attitude Valid

1 Activate autopilot and vertical gyro system. Refer to paragraphs B. (1) and B. (4).

2 Vertical Gyro may be default ON with flight instruments bus. OFF may have to be achieved by opening the circuit breaker.

3 Set Vertical Gyro 1 circuit breaker to OFF; autopilot disengages and AP WARN lights flash red. The AP WARN horn shall be active.

4 Momentarily press pilot control wheel autopilot disengage switch; AP WARN lights go out. The AP WARN horn shall be inactive. Set Vertical Gyro 1 circuit breaker to ON (Breaker closed). Repeat as necessary for alternate attitude systems.

(8) YAW Damper Interlocks

1 If required, activate autopilot and vertical gyro system. Refer to paragraphs B. (1) and B. (4). Activate the aircraft hydraulic system.

2 With autopilot disengaged, set Y/D Engage switch on overhead Yaw Damper Control Panel to OFF.

3 Set Autopilot Engage Switch to AUTOPILOT on AUTOPILOT CONTROLLER; AUTOPILOT switch will not hold in AUTOPILOT position.

4 Set Y/D Engage switch to ON on overhead Yaw Damper Control Panel.

5 Set Autopilot Engage Switch to ON on AUTOPILOT CONTROLLER. AUTOPILOT switch holds in AUTOPILOT position.

6 Set Y/D Engage switch to OFF on overhead panel Yaw Damper Control Panel; The AUTOPILOT switch shall drop to the center off position.

7 Set Y/D Engage switch to ON on overhead panel Yaw Damper Control Panel.

8 Set Autopilot Engage Switch to AUTOPILOT on AUTOPILOT CONTROLLER; The switch shall hold in the engaged position.

(9) Turn Knob Interlock

1 Verify AUTOPILOT disengage.

2 Set TURN Knob on AUTOPILOT CONTROLLER out of detent clockwise.

3 Set Autopilot Engage Switch on AUTOPILOT CONTROLLER to AUTOPILOT; switch will not hold but returns to OFF.

4 Repeat steps 2 and 3 for counterclockwise rotation of TURN Knob.

5 Return TURN Knob to center detent position.

NOTE: In the following interlock check procedures, the autopilot Mode Selector Switch should be set to the MAN position initially.

(10) Mode Selector Switch Interlock

1 With Autopilot Engage Switch set to OFF, manually hold the Autopilot MODE Select Switch in HDG Mode on AUTO-PILOT CONTROLLER.

2 Set Autopilot Engage Switch to AUTOPILOT on AUTOPILOT CONTROLLER; switch will not hold but returns to OFF.

3 Repeat steps 2 and 3 with Autopilot Mode Select Switch on AUTOPILOT CONTROLLER held in NAV, LOC/VOR, GS AUTO and GS MAN.

(11) Automatic Stabilizer Trim Cutout Switch Interlock

1 If required, activate autopilot and vertical gyro system. Refer to paragraph B. (1) and B. (4).

2 With autopilot engaged, set AUTOMATIC STAB TRIM switch to CUTOUT on control stand; the autopilot shall disengage.

3 Return the AUTOMATIC STAB TRIM switch to NORMAL.

(12) Manual/Electric Stabilizer Trim Interlock

1 If required, activate autopilot and vertical gyro system. Refer to paragraph B. (1) and B. (4).

2 With autopilot engaged, set STAB TRIM MAIN ELEC switch to NORMAL on control stand.

3 Press stabilizer trim switch on pilot control wheel for aircraft nose up; autopilot disengages.

4 Repeat steps 2 and 3 for nose down trim.

5 Repeat steps 1 to 4 on copilot side.

(13) Elevator Torque Control Interlocks

1 Ensure AUTOPILOT is on.

2 Set STAB TRIM to 0 position on control stand and engage autopilot.

3 Set FLAP level to 14 degrees; autopilot does not disengage.

4 Set FLAP level to 0; autopilot does not disengage.

5 Manually crank stabilizer trim wheel on control stand to obtain one unit nose up trim; autopilot does not disengage.

6 Manually crank stabilizer trim wheel to zero trim; autopilot does not disengage.

(14) Mode Interlocks

(a) Turn Controller Priority

1 If required, activate autopilot, VHF navigation, and vertical gyro systems. Refer to paragraphs B. (1), B. (2), and B. (4).

2 With autopilot engaged, and heading bug on HSI centered (under lubber line) set autopilot Mode Selector Switch to HDG on AUTOPILOT CONTROLLER.

3 Rotate TURN Knob on AUTOPILOT CONTROLLER clockwise out of detent; autopilot Mode Selector Switch returns to MAN.

a Set NO 1 NAV radio selector switch to PWR.

4 Set VHF navigation system No. 1 for a VOR test frequency. Set Autopilot Mode Select Switch to VOR LOC and repeat step 3.

5 Set VHF navigation system No.1 for glide slope test localizer frequency and a simulated one dot fly-up glide slope signal.

6 Set Autopilot Mode Select Switch to GS AUTO and repeat step 3.

7 Set Autopilot Mode Select Switch to GS MAN and repeat step 3.

(b) HDG Interlock

1 If required, activate autopilot and vertical gyro system. Refer to paragraphs B. (1) and B. (4).

2 With autopilot engaged, set Autopilot Mode Select Switch to HDG on AUTOPILOT CONTROLLER.

a Set ADI switch to VG.

3 Set pilot NAV MODE switch to INS, to VOR/LOC and then to TACAN; Autopilot Mode Select Switch remains in HDG.

(c) Altitude Hold - Autopilot Pitch Wheel Disabling

1 If required, activate autopilot and vertical gyro system. Refer to paragraphs B. (1) and B. (4).

2 With autopilot engaged, move pitch wheel on the AUTOPILOT CONTROLLER in down and up directions; control column moves forward and aft.

NOTE: Stabilizer trim wheel may rotate in nose down or up direction.

3 Set ALT switch to ALT on AUTOPILOT CONTROLLER.

4 Move pitch wheel in down and up direction on AUTOPILOT CONTROLLER; control column (or stabilizer trim wheel) does not move.

(d) Altitude Hold - Glide Slope Interlock

1 If required, activate autopilot, VHF navigation, and vertical gyro systems. Refer to paragraphs B. (1), B. (2), and B. (4).

2 With autopilot engaged, set VHF navigation receiver to a localizer frequency.

a Verify NAV MODE selector is in VOR/LOC.

3 Set ALT switch to ALT on AUTOPILOT CONTROLLER.

4 Set Autopilot Mode Select Switch to GS MAN; ALT switch returns to OFF.

(e) NAV Mode - TACAN VOR/LOC Interlocks

NOTE: The NAV mode interlocks, including VOR, TACAN etc. are not prerequisites to pass the systems test. The B707 is usually configured with these interlocks, however aircraft configuration changes over its life may have obviated these interlocks. In addition, for the interlock checks, the term NAV and INS may be used interchangeably.

1 If required, activate vertical gyro system. Refer to paragraphs B. (1) and B. (4).

2 With autopilot engaged, select VOR/LOC on the pilot NAV MODE.

3 Set Autopilot Mode Select Switch to LOC/VOR on AUTOPILOT CONTROLLER.

4 Set pilot NAV MODE switch to TACAN; Autopilot Mode Select Switch returns to MAN.

(f) NAV Mode - Radio Altimeter VOR/LOC Interlocks

1 If required, activate autopilot, VHF navigation, and vertical gyro systems.Refer to paragraphs B. (1), B. (2), and B. (4).

2 Select VOR/LOC using the Pilot NAV MODE switch and engage the autopilot.

3 Set No.1 VHF navigation receiver to a localizer frequency as follows:

a Tune VHF navigation receiver to 108.10 MHZ.

b Set T-30B/B ramp test set LOC course control to ON COURSE.

c Hold ramp test set LOC POWER switch down for duration of test.

4 Set autopilot mode selector to LOC/VOR on AUTOPILOT CONTROLLER.

5 Ensure Radio Altimeter is ON.

6 Press radio altimeter TEST switch on pilot instrument panel; radio altimeter indicator shows no change.

7 Set pilot NAV MODE switch to TACAN; Autopilot Mode Select Switch returns to MAN.

(g) GS Auto Interlocks

1 If required, activate autopilot, VHF navigation, and vertical gyro system. Refer to paragraphs B. (1), B. (2), and B. (4).

2 Ensure RADIO ALTIMETER is ON.

3 Set pilot NAV MODE switch to VOR/LOC.

4 Set VHF navigation receiver No. 1 to a localizer frequency and a simulated one dot fly-up glide slope signal as follows:

a Tune VHF navigation receiver to 108.10 MHZ.

b Set T-30B/B ramp test set LOC course control to on course.

c Set ramp test set GS course control to VARIABLE.

d Hold down ramp test set GS POWER and rotate GS VARIABLE control for a one dot fly-up indication of pilot HSI.

NOTE: Hold down GS POWER switch for duration of test.

5 With autopilot engaged, set Autopilot Mode Select Switch on AUTOPILOT CONTROLLER to GS AUTO.

6 SET No. 1 VHF navigation system to a VOR frequency (108.00 MHZ); Autopilot Mode Select Switch returns to MAN.

(h) GS MAN Interlocks

1 If required, activate autopilot, VHF navigation, and vertical gyro systems. Refer to paragraphs B. (1), B. (2), and B. (4).

2 Set pilot NAV MODE switch to VOR/LOC.

3 Set No. 1 VHF navigation receiver set to a localizer frequency and a simulated one dot fly-up glide slope signal as follows:

a Tune VHF navigation receiver to 108.10 MHZ.

b Set T-30B/B ramp test set LOC course control to ON COURSE.

c Set ramp test set GS course control to VARIABLE.

d Hold down ramp test set GS POWER and rotate GS VARIABLE control for a one dot fly up indication of pilot HSI.

NOTE: Hold down GS POWER switch for duration of test.

4 With autopilot engaged, set autopilot Mode Selector Switch on AUTOPILOT CONTROLLER to GS MAN.

5 Set no. 1 VHF navigation system to a VOR frequency; autopilot Mode Selector Switch returns to MAN.

(i) Radio Altimeter Interlocks

1 If required, activate autopilot, VHF navigation, and vertical gyro systems. Refer to paragraphs B. (1), B. (2), and B. (4).

2 Ensure RADIO ALTIMETER is ON.

3 With autopilot engaged, set pilot NAV MODE switch to VOR/LOC.

4 Set No. 1 VHF navigation receiver to localizer frequency.

5 Set T-30B/B LOC course switch to VARIABLE and GS course switch to on course.

6 Set autopilot mode selector to LOC/VOR on AUTOPILOT CONTROLLER; VL flight mode annunciators on pilot and copilot's panels come on green.

7 Rotate LOC VARIABLE control for one dot fly right indication on pilot HSI. Control wheel rotates clockwise; note angle position.

8 Immediately set RADIO ALTIMETER circuit breaker to OFF on overhead panel.

9 Verify that autopilot stays engaged and AP mode selector returns to MAN mode, control wheel rotates back to center.

10 Reselect LOC/VOR mode and verify control wheel rotates clockwise to a momentarily a greater angle than noted in step 7.

11 Set Autopilot Engage Switch on AUTOPILOT CONTROLLER to OFF and remove localizer signal.

(15) Stick Shaker Interlock

1 Engage autopilot.

2 At P13 overhead panel, perform Altitude Warning Test on Altitude Warning Panel.

3 Press "L Wing" switch and hold. Press also and hold "OLEO BYPASS" switch until stick shaker is activated.

4 Verify autopilot disengage.

5 Repeat step 3. This time press and hold "R Wing" switch in combination with "OLEO BYPASS" switch.

6 Verify autopilot disengage.

(16) Flight Mode Annunciator Lights

1 If required, activate autopilot per paragraph B. (1)

2 With autopilot disengaged, set TEST switch to TEST on the pilots instrument panel; A/P annunciator GS, VL, TACAN, TEST and INS come on.

3 With TEST indicator pressed, set IND LIGHTS switch to DIM; pilot annunciator indicator light intensity decreases. Return switch to BRT.

4 Repeat step (3) using copilot annunciator.

(17) MAN Mode

(a) Turn Knob Tests

NOTE: The following TURN Knob tests require autopilot inputs from the AUTOPILOT CONTROLLER Turn Knob, using the Vertical Gyro roll attitude source to center the control wheel. To prevent DAU integrator build-up, which could affect the test results, the Turn Knob and Vertical Gyro inputs should be moved together, to the greatest practical extent, to keep the control wheel centered. This will reduce integrator build-up and result in greater test results accuracy.

(b) Coarse Turn Knob Sensitivity

1 If required, activate autopilot, place Vertical Gyro 1 on tilt table, and activate vertical gyro system. Refer to paragraphs B. (1), B. (3), and B. (4)

2 With autopilot engaged and control wheel centered, set TURN Knob on AUTOPILOT CONTROLLER clockwise to the third detent (30 degrees).

3 Tilt Vertical Gyro 1 to simulate right roll until control wheel is centered; tilt required is between 28 and 32 degrees.

4 Level Vertical Gyro and place TURN Knob in center detent position. Disengage the autopilot using the wheel disconnect switch.

5 With autopilot engaged and control wheel centered, set TURN Knob on AUTOPILOT CONTROLLER counterclockwise to the third detent (30 degrees).

6 Tilt Vertical Gyro 1 to simulate left roll until control wheel is centered; tilt required is between 28 and 32 degrees

7 Level Vertical Gyro 1 and place TURN Knob in center detent position. Disengage the autopilot using the wheel disconnect switch.

(18) Bank Angle Limits

(a) Coarse Turn Knob Bank Angle Limit

1 If required, activate autopilot, place Vertical Gyro 1 on tilt table, and activate vertical gyro system. Refer to paragraphs B. (1), B. (3), and B. (4).

2 With autopilot engaged and control wheel centered, rotate TURN Knob to maximum clockwise direction.

3 Tilt Vertical Gyro 1 to simulate right roll until control wheel is centered; tilt required is between 30 and 40 degrees.

4 Level Vertical Gyro 1 and place TURN Knob in center detent position. Disengage the autopilot using the wheel disconnect switch.

5 With autopilot engaged and control wheel centered, rotate TURN Knob to maximum counterclockwise direction.

6 Tilt Vertical Gyro 1 to simulate left roll until control wheel is centered; tilt required is between 30 and 40 degrees.

7 Level Vertical Gyro 1and place TURN Knob in center detent position. Disengage the autopilot using the wheel disconnect switch.

(b) Wing Leveling

1 If required, activate autopilot, place Vertical Gyro 1 on tilt table, and activate vertical gyro system. Refer to paragraphs B. (1), B. (3), and B. (4).

2 With autopilot engaged and control wheel centered, tilt Vertical Gyro 1 to simulate right roll until control wheel moves counterclockwise approximately 30 degrees.

3 Set Autopilot Engage Switch to OFF on AUTOPILOT CONTROLLER and re-center the control wheel.

4 Set Autopilot Engage Switch to AUTOPILOT; control wheel remains centered momentarily, then rotates counterclockwise.

5 Level Vertical Gyro 1 and disengage the autopilot using the wheel disconnect switch

6 Repeat steps 2 through 5 for left roll and clockwise wheel rotation.

7 Level Vertical Gyro 1 and disengage the autopilot using the wheel disconnect switch.

(c) Aileron Gain

1 If required, activate autopilot, place Vertical Gyro 1 on tilt table, and activate vertical gyro system. Refer to paragraphs B. (1), B. (3), and B. (4).

2 With autopilot engaged, the two Spoiler Hydraulic Power Switches on the overhead panel in the OFF position and control wheel centered, tilt Vertical Gyro 1 to simulate 10 degrees right roll; control wheel rotates 32 to 44 degrees counterclockwise.

3 Set two Spoiler Hydraulic Power switches to the ON position; control wheel moves 10 to 20 degrees clockwise.

4 Level Vertical Gyro 1 and disengage the autopilot using the wheel disconnect switch.

5 Set two SPOILER switches to OFF.

6 Repeat steps 1 through 5 with a left roll.

7 Level Vertical Gyro 1 and disengage the autopilot using the wheel disconnect switch.

(19) Clutched Heading

(a) Heading Performance

1 Set pilot's compass to DG mode.

2 Remove mounting bolts of the Directional Gyro (DG). If tilt table is available remove DG and place it on tilt table.

3 Set Rotary Knob to MAN on AUTOPILOT CONTROLLER.

4 Engage autopilot.

5 Rotate DG clockwise (CW). Verify that the control wheel rotates counterclockwise (CCW).

6 Rotate DG CCW. Verify that Control wheel rotates CW.

7 Disengage autopilot.

8 Remove DG from tilt table when installed. Install Directional Gyro (Refer to MM 34-12-1).

(b) Spoiler/Speed Brake Operation

1 If required, activate autopilot and vertical gyro system. Refer to paragraphs B. (1), B. (3), and B. (4).

a Resync. the copilot's compass

2 Set two SPOILER switches on overhead panel to ON.

3 With autopilot engaged and control wheel centered, set speed brake lever to 10; control wheel remains centered within ±5 degrees.

4 Set speed brake to 30 and to 50; control wheel remains centered within ±5 degrees.

5 Return speed brake to zero.

6 Set two SPOILER switches to OFF.

7 Disengage the autopilot using the wheel disconnect switch.

(c) Pitch Wheel Sensitivity

1 If required, activate autopilot, place Vertical Gyro 1 on tilt table, and activate vertical gyro system. Refer to paragraphs B. (1), B. (3), and B. (4).

NOTE: Measurement can be taken from center of top of control column to the aft face of the main instrument panel. Be sure that all measurements are made using the same reference points. The control column neutral position is established by spring action when autopilot is disengaged.

2 Measure control column neutral position.

3 Set STAB TRIM switches on control stand to NORMAL.

4 Engage autopilot.

5 Rotate autopilot PITCH wheel one revolution in pitch up direction; control column moves aft.

6 Tilt Vertical Gyro 1 to simulate pitch up until control column returns to neutral; tilt required is between 5 and 9 degrees.

7 Level Vertical Gyro 1 and disengage the autopilot using the wheel disconnect switch.

(d) Elevator Gain

1 If required, activate autopilot, place Vertical Gyro 1 on tilt table, and activate vertical gyro system. Refer to paragraphs B. (1), B. (3), and B. (4).

2 Set STAB TRIM AUTOMATIC switch on control stand to NORMAL.

3 With autopilot disengaged, move control column aft to just off aft stops.

4 Engage autopilot and measure control column position.

5 Tilt Vertical Gyro 1 to simulate pitch up until control column moves forward 4 inches; tilt required is between 5 and 7.5 degrees.

6 Level Vertical Gyro 1 and disengage the autopilot using the wheel disconnect switch.

(e) Elevator Versine

1 If required, activate autopilot, place Vertical Gyro 1 on tilt table, and activate vertical gyro system. Refer to paragraphs B. (1), B. (3), and B. (4).

2 With STAB TRIM AUTOMATIC switch on control stand set to NORMAL and autopilot disengaged, move control column forward to just off forward stops.

3 Engage autopilot and measure control column position.

4 Tilt Vertical Gyro 1 30 degrees to simulate right roll; control column moves aft 1.5 to 4 inches.

5 Apply pitot pressure to simulate an increase in airspeed and verify that the control column moves forward.

6 Level Vertical Gyro 1 and disengage the autopilot using the wheel disconnect switch.

(f) Altitude Hold Integration

1 With pressures to simulate an altitude of 200 feet above airfield and an airspeed of 120 knots IAS applied to the pitot and static source and the control column held just off the forward stops, engage the autopilot and operate the Altitude Hold switch to ALT.

2 Slowly decrease the simulated altitude to ambient, maintaining the airspeed at 120 knots. The control column will move aft.

3 Tilt Vertical Gyro 1 to simulate pitch up until control column moves to approximately neutral. The control column will move forward, following the Vertical Gyro and then move slowly aft.

4 Level Vertical Gyro 1 and disengage the autopilot using the wheel disconnect switch. Decrease simulated altitude to ambient and airspeed to zero.

(g) Rate Transmitter - Roll Rate

1 If required, activate autopilot and vertical gyro system. Refer to paragraphs B. (1), B. (3) and B. (4).

2 Remove mounting bolts from three-axis rate transmitter.

3 Set pilot NAV MODE selector to VOR/LOC.

a Set course cursor under lubber line.

4 With autopilot engaged, select a LOC frequency (zero LOC and GS deviation) as follows:

a Tune VHF navigation receiver to 108.10 MHZ.

b Set T30B/B ramp test set LOC course control to ON COURSE.

c Set ramp test set GS COURSE control to ON COURSE.

d Hold down GS and LOC POWER switches for duration of test.

5 Set Autopilot Mode Select Switch to GS MAN on AUTOPILOT CONTROLLER; VL and GS indicators on flight mode annunciators come on green (engaged) on pilot instrument panels.

6 Rotate three-axis rate transmitter about roll axis at a constant rate to simulate right roll; control wheel rotates counterclockwise and remains as long as transmitter is rotated at a constant rate.

NOTE: If the three-axis rate transmitter rotation speed exceeds specified limits, the autopilot will automatically disengage and a roll rate fault will be stored in DAU non-volatile memory. If this condition should occur, reengage the autopilot and repeat the test, rotating the three-axis rate transmitter more slowly.

7 Stop transmitter rotation; control wheel returns to center.

8 Set Autopilot Mode Select Switch to MAN.

9 Repeat steps 6 and 7; verify that the control wheel responds to the rate transmitter rotation, as described in step 6, except simulate a left roll.

10 Disengage the autopilot using the wheel disconnect switch.

(h) Rate Transmitter - Pitch Rate

1 If required, activate autopilot and vertical gyro system. Refer to paragraphs B. (1), B. (3) and B. (4).

2 Ensure stabilizer is set to zero.

3 Remove mounting bolts from three-axis rate transmitter.

4 Engage the autopilot in the MAN mode.

5 Rotate the three-axis rate transmitter about pitch axis at a constant rate to simulate pitch up; control column momentarily moves forward, then returns to neutral.

NOTE: If the three-axis rate transmitter rotation speed exceeds specified limits, the autopilot will automatically disengage and a pitch rate fault will be stored in DAU non-volatile memory. If this condition should occur, reengage the autopilot and repeat the test, rotating the three-axis rate transmitter more slowly.

6 Stop transmitter rotation; control column returns to neutral.

7 Repeat steps 5 and 6 for simulated pitch down.

8 Disengage the autopilot using the wheel disconnect switch.

9 Reinstall three-axis rate transmitter on equipment shelf.

(k) Stabilizer Trim Rate

1 If required, activate autopilot and vertical gyro system. Refer to paragraphs B. (1) and B. (4).

2 Set control column just off aft stops.

a Set stabilizer trim to 0.

3 Engage autopilot.

4 With flaps up and TURN Knob in detent on AUTOPILOT CONTROLLER, rotate autopilot PITCH wheel in pitch up direction.

5 Measure time required for stabilizer to move through 4 units of trim as indicated on control stand trim indicator; time is 90 (±18) seconds.

6 Disengage autopilot, return stabilizer to 0 trim and reengage autopilot.

7 Set FLAP lever to 14 degrees then rotate PITCH wheel in full pitch up direction.

8 Measure time required for stabilizer to move through 4 units of trim: time is 30 (±6) seconds.

9 Disengage autopilot, return stabilizer to 0 trim and reengage autopilot.

10 Set FLAP lever to 0, TURN Knob out of detent and rotate PITCH wheel in full pitch up direction.

11 Measure time required for stabilizer to move through 4 units of trim: time is 30 (±6) seconds.

12 Set TURN Knob to detent.

13 Repeat the above testing, except with the control column held just off the forward stops and rotation of the pitch controller in the down direction from A units to 0 units.

14 Disengage the autopilot using the wheel disconnect switch.

(1) Stabilizer Trim Threshold

1 If required, activate autopilot and vertical gyro system. Refer to paragraphs B. (1) and B. (4).

2 Set stabilizer in zero trim position.

3 With control column in neutral position, engage autopilot.

4 Connect spring scale to center of control wheel.

5 Rotate PITCH wheel on AUTOPILOT CONTROLLER in pitch up direction while measuring force at center of control wheel; stabilizer trim motor starts to drive (STAB TRIM indicator on control stand moves) when measured force reaches 6 (±2) pounds.

6 Return PITCH wheel to neutral.

7 Repeat the above tests except rotate the PITCH wheel in the pitch down direction.

8 Disengage the autopilot using the wheel disconnect switch.

(m) Stabilizer Trim Limits

1 If required, activate autopilot and vertical gyro system. Refer to paragraphs B. (1) and B. (4).

2 Set stabilizer for -8 units of trim.

3 With autopilot engaged, rotate PITCH wheel in pitch up direction until stabilizer is driven to nose up electrical limits (STAB TRIM indicator stops); control stand trim indicator indicates approximately -10.4 units of trim and ELV trim on 3 axis trim indicator is displaced above in-trim indices.

4 Manually crank stabilizer trim wheel to nose up mechanical limit; control stand trim indicator indicates approximately -11 units of trim.

5 Set stabilizer for zero trim.

6 Rotate PITCH wheel in pitch down direction until stabilizer is driven to nose down electrical limits; control stand trim indicator indicates 2.4 units of trim and ELV trim indicator is displaced below in-trim indices.

7 Manually crank stabilizer trim wheel to nose down mechanical limits; control stand trim indicator indicates approximately 3.5 units of trim.

8 Disengage the autopilot if engaged using the wheel disconnect switch and return stabilizer to zero trim.

(20) Preset Heading Gain

1 If required, activate autopilot, place Vertical Gyro 1 on tilt table, and activate vertical gyro system. Refer to paragraphs B. (1), B. (3), and B. (4).

2 Apply pressure to pitot system to simulate 250 knots IAS.

3 Using HDG knob on pilot HSI, set preset heading bug to 5 degrees right of aircraft heading.

4 Engage autopilot and set autopilot mode selector to HDG; control wheel rotates clockwise and stops.

5 Tilt Vertical Gyro No 1 to simulate right roll until control wheel is centered; tilt required is between 6 and 9 degrees.

6 Decrease pitot pressure to simulate decrease in airspeed; control wheel rotates counterclockwise.

7 Level Vertical Gyro 1.

8 Set pitot pressure to ambient. Disengage the autopilot using the wheel disconnect switch.

(a) Bank Limits - HDG

1 If required, activate autopilot, place Vertical Gyro No 1 on tilt table, and activate vertical gyro system. Refer to paragraphs B. (1), B. (3), and B. (4).

2 With control wheel centered, apply pitot pressure to simulate 250 knots IAS.

3 Using HDG knob and pilot HSI, set preset heading cursor 90 degrees right of aircraft heading.

4 Engage the autopilot and set Autopilot Mode Select Switch on AUTOPILOT CONTROLLER to HDG; control wheel rotates clockwise.

5 Tilt Vertical Gyro No 1 to simulate right roll until control wheel is centered; tilt required is between 25 and 35 degrees.

6 Disengage the autopilot using the wheel disconnect switch. Level Vertical Gyro No 1.

7 Repeat steps 4 through 5, except set preset heading cursor for 90 degrees left of aircraft heading and simulate a left roll using Vertical Gyro No 1.

8 Reduce pitot pressure to ambient.

(b) INS / AUTOPILOT - Steering Command

1 Ensure that autopilot is activated and activate INS system.

2 Set pilots Autopilot INS Source Selector to INS 1.

3 On Mode Selector Control Panel set INS mode selector switch to ALIGN position.

4 Set INS data selector on control-display unit (CDU) to DSRTK/STS and wait for completion of INS vertical gyro alignment sequence until STS display indicates "55" or less (PI 5).

5 Set pilot NAV MODE selector to INS on glare shield.

6 Push in HSI- Course Knobs (both pilots).

7 On CDU set TEST/AUTO/MAN switch to MAN for a right hand INS steering command.

8 Set INS data selector on control display unit (CDU) to any position except DSRTK/STS.

9 Open LOW RANGE ALTM NO 1 circuit breaker on P5 overhead panel.

10 With autopilot engaged, set autopilot mode select switch to LOC/VOR on AUTOPILOT CONTROLLER; LOC/VOR mode does not hold.

11 Press and hold TEST switch (INS provides a valid Signal and a 15 ±2 degree right steering command to the autopilot).

12 With control wheel centered and autopilot engaged, set autopilot mode selector to LOC/VOR on AUTOPILOT CONTROLLER:

- LOC/VOR mode does hold and is activated
- Pilot / Copilot flight mode annunciator INS lights come on green (engaged)
- Control wheel rotates clockwise approx. 50 degrees.

13 Release TEST switch on CDU; Autopilot mode select switch goes back to MAN; control wheel comes back to center.

14 Set TEST/AUTO/MAN switch to AUTO for a left hand INS steering command.

15 With autopilot engaged, set autopilot mode select switch to LOC/VOR on AUTOPILOT CONTROLLER; LOC/VOR mode does not hold.

16 Press and hold TEST switch (INS provides a valid signal and a 15 ±2 degree left steering command to the autopilot).

17 With control wheel centered and autopilot engaged, set autopilot mode selector to LOC/VOR on AUTOPILOT CONTROLLER:

- LOC/VOR mode does hold and is activated
- Pilot / Copilot flight mode annunciator INS lights come on green (engaged)
- Control wheel rotates counterclockwise approx. 50 degrees.

18 Release TEST switch on CDU; Autopilot mode select switch goes back to MAN; control wheel comes back to center.

If aircraft is equipped with DUAL INS:

19 Select INS 2 Alternate at pilots Autopilot INS Source Selector.

20 Repeat steps 3 thru 18 for INS 2.

21 Disengage the autopilot using the wheel disconnect switch, deactivate INS system.

22 Close LOW RANGE ALTM NO 1 circuit breaker on P5 overhead panel.

(c) VOR Capture

1 If required, activate autopilot, VHF navigation, and vertical gyro systems. Refer to paragraphs B. (1), B. (2) and B. (4).Set pilot's HSI to North using copilot RMI.

2 Set pilot NAV MODE selector to VOR/LOC.

3 Set VHF navigation receiver to VOR test frequency (108.00 MHZ) with a simulated 0 degree radial signal.

a Slew HSI compass card to 90 degrees using the SET HDG control.

b Set T-30B/B ramp test set VOR course control to TO 270 degrees.

NOTE: Hold down VOR POWER switch on ramp test set for duration of test.

4 Set pilot HSI COURSE selector to provide 1-1/2 dot fly right indication.

5 With autopilot engaged, set Autopilot Mode Select Switch to VOR LOC on AUTOPILOT CONTROLLER; pilot and copilot flight mode VL annunciator lights come on amber (armed).

6 Slowly rotate HSI COURSE selector to reduce fly right indication until control wheel rotates clockwise; this occurs between 1/2 and one dot fly right indication. VL amber lights go off and VL green (engaged) lights come on.

7 Rapidly rotate HSI COURSE selector to provide on course indication and then to one dot fly left indication; control wheel rotates rapidly counterclockwise then clockwise following COURSE selector then continues rotating slowly clockwise.

8 Disengage the autopilot using the wheel disconnect switch.

(c) VOR Course Gain

1 If required, activate autopilot and VHF navigation system, place Vertical Gyro No 1 on tilt table, and activate vertical gyro system. Refer to paragraphs B. (1), B. (2), B. (3) and B. (4). Ensure that VOR receiver is OFF.

2 With pilot NAV MODE selector set to VOR/LOC and VHF navigation receiver set to VOR test frequency (of 108.00 MHZ), open FLT INSTR NO. 1 BRG EXC circuit breaker.

3 Set pilot HSI (deviation bar centered) course selector to 5 degrees right of aircraft heading.

4 Engage autopilot and set Autopilot Mode Select Switch to VOR LOC on AUTOPILOT CONTROLLER; control wheel rotates clockwise.

5 Tilt Vertical Gyro No 1 quickly to simulate right roll until control wheel is centered; tilt required is between 6 and 14 degrees.

6 Level Vertical Gyro No 1 and close FLT INSTR NO. 1 BRG EXC circuit breaker.

7 Disengage the autopilot using the wheel disconnect switch and level Vertical Gyro No 1.

(d) VOR Station Passage

1 If required, activate autopilot, VHF navigation, and vertical gyro systems. Refer to paragraphs B. (1), B. (3), and B. (4).

2 With pilot NAV MODE selector set to VOR/LOC and VHF navigation receiver set to VOR test frequency (of 108.00 MHZ), set pilot HSI COURSE selector to aircraft heading.

3 Set protractor for 20 degrees left.

4 With autopilot engaged, provide 1/2 dot fly right deviation signal to the navigation receiver as follows:

a Set the ramp test set VOR course control to VARIABLE.

b Hold down ramp test set VOR POWER switch and rotate VOR VARIABLE control for a 1/2 dot fly right deviation signal.

NOTE: Hold down VOR POWER switch for duration of test.

5 Set Autopilot Mode Select Switch to LOC/VOR on AUTOPILOT CONTROLLER; control wheel rotates slowly clockwise. Use a stopwatch to note duration and rate of wheel rotation.

6 Reduce deviation signal to zero by moving course knob; control wheel slowly returns to center.

7 Increase deviation signal for two dots fly right indication; control wheel rotates clockwise at approximately three times faster than noted in step 5. (Timing from 0 to 20 degrees).

8 Disengage the autopilot using the wheel disconnect switch.

(21) LOC Mode

(a) Localizer Capture

1 If required, activate autopilot, VHF navigation, and vertical gyro systems. Refer to paragraphs B. (1), B. (2), B. (3) and B. (4).

2 Set pilot NAV MODE selector to VOR/LOC.

3 Select LOC frequency and provide simulated localizer signal equivalent to three dots fly right.

4 Set pilot HSI COURSE selector to aircraft heading.

5 Engage autopilot and after 5 seconds set Autopilot Mode Select Switch to LOC/VOR on AUTOPILOT CONTROLLER; control wheel remains centered and pilot and copilot flight mode VL annunciator lights come on amber (armed).

6 Slowly decrease simulated localizer signal until control wheel rotates rapidly clockwise; this occurs prior to HSI deviation needle reaching one and one half dot indication. Flight mode VL annunciator amber lights go off and VL green (engaged) lights come on.

7 Disengage the autopilot using the wheel disconnect switch.

(b) Localizer Aileron Gain

1 If required, activate autopilot and VHF navigation system, place Vertical Gyro No 1 on tilt table, and activate vertical gyro system. Refer to paragraphs B. (1), B. (2), B. (3), and B. (4).

2 With pilot NAV MODE selector set to VOR/LOC, provide a localizer frequency (zero localizer deviation). Set

the two Spoiler Hydraulic Power Switches in the OFF position.

a Tune VHF navigation receiver to 108.10 MHZ.

b Set T-30B/B LOC course control to variable and set VARIABLE control to center HSI deviation bar.

NOTE: Hold down LOC POWER switch for duration of test.

3 Set pilot HSI COURSE selector to aircraft heading and engage autopilot.

4 Set Autopilot Mode Select Switch to VOR LOC.

5 Tilt Vertical Gyro 1 to simulate 5 degrees right roll; control wheel rotates counterclockwise between 35 and 65 degrees and AIL trim indicator is centered on three-axis trim indicator indicating that control wheel is not against stops.

6 Level Vertical Gyro No 1.

7 Disengage the autopilot using the wheel disconnect switch.

(c) Localizer Course Gain

1 If required, activate autopilot, VHF navigation, and vertical gyro systems. Refer to paragraphs B. (1), B. (2), B. (3) and B. (4).

2 With pilot NAV MODE selector set to VOR/LOC, provide a localizer frequency (zero localizer deviation) as follows:

a Tune VHF navigation receiver to 108.10 MHZ.

b Set LOC course control to variable and set VARIABLE control to center HSI deviation bar.

NOTE: Hold down LOC POWER switch for duration of test.

3 Set pilot HSI COURSE selector 5 degrees right and engage autopilot.

4 After 5 seconds, set Autopilot Mode Select Switch to LOC/VOR on AUTOPILOT CONTROLLER; control wheel rotates rapidly clockwise to between 45 and 75 degrees, then slowly counterclockwise towards center.

5 Disengage the autopilot using the wheel disconnect switch.

(d) Localizer Displacement Gain

1 If required, activate autopilot and VHF navigation system, place Vertical Gyro No 1 on tilt table, and activate vertical gyro system. Refer to paragraphs B. (1), B. (2), B. (3), and B. (4).

2 Pull the circuit breaker for the Radio Altimeters.

3 With pilot NAV MODE selector in VOR/LOC and localizer frequency selected, provide simulated 1/2 dot fly right indication as follows:

a Tune VHF navigation receiver to 108.10 MHZ.

b Set T-30B/B ramp test set LOC course control to VARIABLE.

c Hold down ramp test set LOC POWER switch and rotate LOC VARIABLE switch for a 1/2 dot fly right indication on pilot HSI.

NOTE: Hold down LOC POWER switch for duration of test.

4 With pilot HSI COURSE selector set to aircraft heading, engage autopilot.

5 After 5 seconds, set Autopilot Mode Select Switch to LOC/VOR on AUTOPILOT CONTROLLER; control wheel rotates clockwise.

6 Tilt Vertical Gyro No 1 to simulate right roll as required to center control wheel; tilt required is between 9 and 15 degrees.

7 Level Vertical Gyro No 1.

8 Disengage the autopilot using the wheel disconnect switch.

(e) Beam Rate Crossfeed (Rudder)

NOTE: Some aircraft do not use the localizer beam rate crossfeed function. Do not perform this test if this function is not used in the aircraft being tested.

1 If required, activate autopilot and VHF navigation, and vertical gyro systems. Refer to paragraphs B. (1), B. (2), B. (3) and B. (4).

2 With series yaw damper engaged, pilot NAV MODE selector set to VOR/LOC and localizer test frequency selected, provide simulated localizer signal for on-course indication as follows:

a Tune VHF navigation receiver to 108.10 MHZ.

b Set T-30B/B ramp test set LOC course control to VARIABLE.

c Hold down LOC POWER switch for duration of test.

d Set test set LOC VARIABLE control for 0 deviation on pilot HSI.

3 Set pilot HSI COURSE selector to aircraft heading and engage autopilot.

4 Set Autopilot Mode Select Switch to GS MAN on AUTOPILOT CONTROLLER.

5 After 20 seconds, increase simulated localizer signal (smoothly and at rapid rate) to provide two dot fly right indication; RUD trim indicator deflects to right on three-axis trim indicator during transition from on-course to two dots fly right and then returns to neutral.

6 Disengage the autopilot using the wheel disconnect switch.

(22) Glide Slope Modes

(a) GS Auto Engage

1 If required, activate autopilot, VHF navigation, and vertical gyro systems. Refer to paragraphs B. (1), B. (2), B. (3) and B. (4).

2 Set STAB TRIM AUTOMATIC switch to NORMAL on control stand.

3 With pilot NAV MODE selector set to VOR/LOC and localizer test frequency selected (zero deviation), provide simulated glide slope displacement signal to produce one dot fly up indication as follows:

a Tune VHF navigation receiver to 108.10 MHZ.

b Set T-30B/B ramp test set LOC course control to variable and set VARIABLE control to center HSI deviation bar.

c Set ramp test set GS course control to VARIABLE.

d Hold down ramp test set GS POWER and rotate GS VARIABLE control for a one dot fly up indication of pilot HSI.

NOTE: Hold down LOC and GS POWER switch for duration of test.

4 Engage autopilot and set Autopilot Mode Select Switch to GS AUTO; flight mode GS annunciator lights come on amber (ARM), control column does not move.

5 Slowly reduce glide slope displacement signal to produce 1/4 dot fly up indication; GS light remains amber.

6 Reduce glide slope displacement signal to zero (GS pointer centered); GS amber lights go off and GS green (engage) lights come on.

7 Increase glide slope displacement signal to 1/2 dot fly up and then decrease to 1/2 dot fly down deflection; control column moves aft and forward following displacement signal.

8 Disengage the autopilot using the wheel disconnect switch.

C. Restore Aircraft to Normal.

(1) Open following circuit breakers on P5 overhead circuit breaker panel:

NOTE: LX-N19997 and LX-N20000 only

AUTOPILOT
A/P & MACH TRIM (DISENG LT AC)
F/O COMPTER NO 1 (AC)

ESSENTIAL RADIO BUS

SERIES YAW DAMPER (AC)

NOTE: LX-N20199 only

FLIGHT INSTRUMENT BUS NO 2
A/P & MACH TRIM (DISENG LT AC)

ESSENTIAL FLIGHT INSTR BUS
AUTOPILOT
FLT INST NO 1 FD NO 1

ESSENTIAL RADIO BUS
SERIES YAW DAMPER (AC)

(2) Turn off INS as follows:

(a) Set mode selector to OFF on system mode control.

(b) Set mode selector to OFF on each INS mode select unit.

(c) Open following circuit breakers on P5 overhead panel:

ESS FLIGHT INSTR BUS

- ADC NO 1 (DC)
- NAV XFER RLY NO 1 (DC)
- F/O COMPTR NO 1 (AC)
- COMPASS NO 1 (AC)
- BRG EXC (AC)
- HDG EXC (AC)
- ATT EXC (AC)
- XMFR NO 1 (AC)
- VERT GYRO NO 1 (AC)

FLIGHT INSTRUMENT BUS NO 2

- NAV XFER RLY NO 2 (DC)
- ADC NO 2 (DC)
- COMPASS NO 2 (AC)
- BRG EXC (AC)
- HDG EXC (AC)
- VERT GYRO NO 2 (AC)
- ATT EXC (AC)
- XMFR NO 2 (AC)

ESS RADIO BUS

- INS NO 1 PWR (AC)
- INS NO 1 HTR (AC)
- VOR NO 1 (AC)
- GS NO 1 (DC)
- VOR NO 1 (DC)
- IU NO 1 (DC)

RADIO BUS NO 2

- INS NO 2 PWR (AC)
- INS NO 2 HTR (AC)
- VOR NO 2 (AC)
- GS NO 2 (DC)
- VOR NO 2 (DC)
- ADC XFER
- IU NO 2

(3) Remove navigation system test equipment.

(4) Remove hydraulic power if no longer required for other maintenance.

(5) Remove electrical power and draw through cooling air from aircraft if no longer required for other maintenance.

(6) Remove forced air cooling and disconnect Air Pressure warning Alarm System if no longer required for other maintenance.

END

(b) When the autopilot is disengaged by means of the wheel disconnect switches, either intentionally or inadvertently, relay K3(Q4) is provided with a ground. Simultaneously, excitation is removed from the coil of relay K1(Q2). However, due to the time delay circuit across the coil of K1(Q2) the relay does not release immediately thus relay K3(Q4), which is still receiving 28 volts dc through the pulled-in contacts of K1(Q2), closes. This prevents relay K2(Q3) from energizing when K1(Q2) times out, i.e. the ground return of K2(Q3) is interrupted by the pulled-in contacts of K3(Q4). Consequently when relay K1(Q2) does relax the circuit to light B is complete through its own relaxed contacts and those of K2(Q3), causing light B to operate.

(c) and (d) Deleted

F. Radio and Compass Transfer Interlock Circuit

(1) The schematic of radio and compass transfer interlock circuit is shown in figure 16. Through this circuit, 28 volts dc is applied to the mode selector holding coil when the mode selector is on HDG, LOC-VOR, GS AUTO or GS MAN position.

(2) The interlock circuit will open 28 volts dc path to the mode selector holding coil and return the mode selector to MAN mode if:

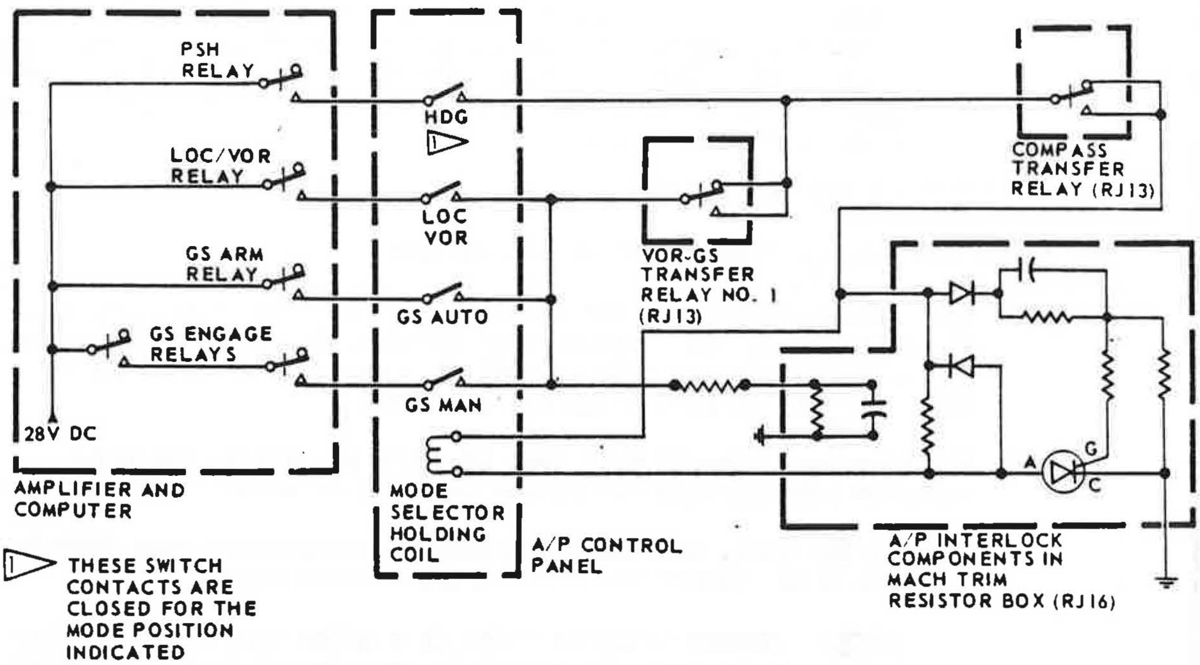
(a) In HDG mode, the compass switch is transferred from COMPASS N No. 1 to COMPASS No. 2 or in the reverse order.

NOTE: Compass transfer relay is a pulse type relay. Relay trips when the switch is transferred from COMPASS NO. 1 to COMPASS NO. 2 or in the reverse order.

(b) In LOC-VOR, GS AUTO or GS MAN mode, the pilot's deviation switch or the compass switch is transferred from one position to the other.

NOTE: VOR/GS relay trips when pilot's deviation switch changes position.

(3) The initial surge of current through the capacitor of the autopilot interlock components triggers the gate of the silicon controlled rectifier. The rectifier conducts and provides a ground for the mode selector holding coil. When the compass or navigation switch is operated the charge on capacitor prevents the rectifier from being retriggered, the rectifier stops conducting and the mode selector returns to MAN position.



Radio and Compass Transfer Interlock Circuit
Figure 16

AUTOPILOT ALTITUDE HOLD FUNCTION - ADJUSTMENT/TEST

1. RVSM Altitude Hold In-flight Performance Check

A. General

(1) The aircraft fulfills the requirements to operate in the Reduced Vertical Separation Minimum (RVSM) airspace. To comply with RVSM regulations, the autopilot system must be capable of maintaining a selected altitude within a specified range for a certain time, the following test has to be performed.

B. Test Procedure

NOTE: This test requires a verification flight to be performed by approved flight personell.

(1) Equipment required:

Data Tabulation Sheet (Refer to Fig. 1)

(2) Checkout

(a) Select a convenient altitude.

NOTE: The selected altitude must be within RVSM airspace (FL 290 thru FL 410).

(b) Allow aircraft to stabilize.

(c) Select ALT on autopilot control panel.

(d) Start recording data with aircraft flying in altitude hold mode according to Fig. 1.

NOTE:
- On longer legs at the same altitude, altitudes may be recorded only every 10 minutes.
- The maximum deviation from the desired altitude may not exceed ±65 ft.

(e) After performance check has been completed, select autopilot mode as required.

(f) Hand over the Data Tabulation Sheet to maintenance crew.

RVSM Altitude Hold In-Flight Performance Check Aircraft Registration:						
Departure Airfield			**Date/Time:**			
Destination Airfield			**Pilot:**			
Position	**Latitude:**		**Longitude:**			
	Instruments					
	Pilot			**Copilot**		
Time Min	Altimeter	Machmeter	Airspeed Indicator	Altimeter	Machmeter	Airspeed Indicator
0:00						
0:05						
0:10						
0:15						
0:20						
0:25						
0:30						
0:35						
0:40						
0:45						
0:50						
0:55						
0:60						

Fig. 1 Altitude Hold In-Flight Performance Check

1. Tools and Test Equipment.

   A. Table 501 lists tools and test equipment required for flightline maintenance of the autopilot system.

PART NO. OR TYPE DESIGNATOR	USE AND APPLICATION	NOMENCLATURE
TOOLS		
F52485-500 Boeing	Protractor	Measure control wheel angles.
F72790 Boeing	Protractor Mount	Mounts protractor to control wheel.
L-30M Amtek/Hunter Spring	Spring Scale - 30#	Measure control column forces.
F70183-500 Boeing	Fixture - Servo Torque Measuring	Measure aileron and elevator torque motor torque.
TEST EQUIPMENT		
2TSJ60B00002 Boeing	Tilt Table - INS and Gyro Units	Used for tilting INS unit.
204-91950-1 Boeing	Adapter - Tilt Table	Provides interface between INS and tilt table and tilt table and aircraft.
GGS-764	Stop Watch	Time control responses.
AN/USM 341	Digital Multimeter	Measure servo ac voltages.
Model 260 Simpson	DC Voltohmeter	Measure dc voltages.
AN/ASN 186 Corad 28845	Ramp Test Set	Provides test signals for VHF NAV system.
T-30 B/B Tel-Instrument Corp.	Ramp Test Set	Provides test signals for VHF NAV system.
980N-1 Collins Radio	Altimeter Test Set	Provide radio test altitude signal.

Autopilot System Tools and Test Equipment
Table 501 (Sheet 1 of 2)

PART NO. OR TYPE DESIGNATOR	USE AND APPLICATION	NOMENCLATURE
AN/ARM 25C	TACAN Test Set	Provide TACAN test signal.
TTU-205 C/E	Pressure Temperature Test Set	Provide altitude and air-speed inputs.
204-91920-1	Adapter-Pitot System	To connect test set to pitot probes.
33410-125-375T	Adapter-Static System	To provide closure for aircraft static ports and connect test set.
Equivalent tools or test equipment are authorized for use.		

Autopilot System Tools and Test Equipment
Table 501 (Sheet 2 of 2)

Ref.: DRW CDRL 43/44 Issue D dated May 19/89
Paragraph 7.36

2. Checkout.

CAUTION: OBSERVE ALL REQUIREMENTS FOR COOLING AIR AS SPECIFIED IN THE CHECKOUT PROCEDURES. OTHERWISE, EQUIPMENT MAY BECOME DAMAGED BECAUSE OF OVER HEATING.

A. General

(1) Unless otherwise specified, autopilot is engaged in MAN mode.

(2) Establish communications between crew members.

(3) The checkout procedure is divided into individual tests that may be per formed in sequence or on an individual basis as required. It complete test is to be performed, paragraphs B. (1) , 3 - (2), B - (3) and B. (4) must be performed before starting test. In this case step a of the indi vidual tests need not be performed.

(4) The restore aircraft to normal procedure, paragraph C, must be performed after every test sequence if no other tests are to be performed.

(5) In any checks. requiring the control wheel to be centered, the allowable tolerance is ± 6 degrees unless otherwise specified.

B. Preparation.

(1) Autopilot Activation.

(a) Apply electrical power to aircraft and turn on draw through cooling.

(b) Establish interphone communication between personnel.

(c) If tilt table is required, perform paragraph B. (3).

WARNING: ENSURE THAT ALL PERSONNEL ARE CLEAR OF CONTROL SURFACES, CON TROL CABLES, SERVO MECHANISMS AND OTHER MOVABLE PARTS WHEN APPLYING OR REMOVING HYDRAULIC POWER OR WHEN CONDUCTING TESTS THAT RESULT IN SURFACE MOVEMENT. UNEXPECTED MOVEMENT OF CONTROL SURFACES MAY CAUSE INJURY.

Ref.: DRW CDRL 43/44 Issue dated May 19/89
Paragraph 7.36

(d) Apply hydraulic power to aircraft.

(e) Close following circuit breakers on overhead circuit breaker panel P5:

AUTOPILOT 30

FLIGHT CONTROLS MACH TRIM

YAW DAMPERS SERIES OA

ESS FLIGHT INSTRUMENT BUS

FLT DIR NO. 1 COMPTR

(f) Set ON-OFF switch to ON on overhead panel YAW DAMPER control panel.

(g) Set AUTOPILOT INS SOURCE SELECT switch on pilot instrument panel to INS 1 NORMAL.

(h) Set FLAP lever to FLAP UP position.

(i) Set IND LIGHT switch on center instrument panel to BRT

(j) When necessary to engage the autopilot during checkout, perform the following Substeps as necessary:

1) Check that autopilot mode select switch is in MAN position on AUTO PILOT CONTROLLER.

2) Set TURN knob in detent on AUTOPILOT CONTROLLER.

3) Set STAB TRIM AUTOMATIC switch to NORMAL on control stand.

4) Set AUTOPILOT engage switch to ON on AUTOPILOT CONTROLLER.

NOTE: The autopilot is disengaged when the AUTOPILOT switch on the AUTOPILOT CONTROLLER returns to OFF. The AP WARN lights will flash until reset by pressing the light or the reset switch on either control wheel.

5) Set two SPOILERS position switches on overhead panel to OFF.

(2) VOR Navigation System Activation.

(a) Activate No. 1 VOR navigation system as follows:

Ref.: DRW CDRL 43/44 Issue D dated May 19/89
Paragraph 7.36

1) Check that following circult breakers on overhead circuit breaker panel are closed.

ESS FLIGHT INSTRUMENT BUS	FLIGHT INSTRUMENT BUS NO.2
BRG EXC	XMFR NO. 2
HDG EXC	COMPASS NO. 2
COMPTR (FIT DIR) NO. 1	BRG EXC
VG	HDG EXC
NAVXFER RLY NO. 1	COMPTR(FLT DIR)NO. 2
XMFR NO. 1 COMPASS NO. 1	NAV XFER RLY NO. 2

ESS RADIO BUS	RADIO BUS NO. 2
VOR NO. 1 (AC)	VOR No. 2 (AC)
VOR NO. 1 (DC)	VOR No. 2 (DC)
GS NO. 1	GS NO. 2

2) Set pilot NAV MODE selector on glare shield to VOR/LOC.

3) Set control switch on No. 1 VOR control panel to PWR. Check that dial lights come on.

4) Tune VOR navigation receiver to local station.

5) Check that NAV and GS flags on HSI, GS flag on ADI, and warning flags on RMI are all out of view.

6) Set pilot RMI selection on main instrument panel to TACAN/VOR.

7) Check that BRG and HDG circuit breakers on flight instrument accessory units (rack) are closed.

8) Make the following settings on both compass controllers located on overhead panel P13.

a) Select aircraft hemisphere.

b) Select aircraft latitude.

c) Set Mode selector to DG.

d) Push pilot HDG knob in and slew pilot HSI compass card to 100 degree heading under the lubber line. Check that copilot P141 agree with pilot HSI ± 2 degrees.

Ref.: DRW CDRL 43/44 Issue D dated May 19/89
Paragraph 7.36

(3) Install INS on Tilt Table. E-3A ONLY

Note: If A/P ATT Signals needs to be tested (ex.B.(9)), install VG1 on tilt table.

(a) Remove INS 1 navigation unit from rack and install on tilt table as follows:

CAUTION: INS MUST BE OFF FOR A MINIMUM OF 2 MINUTES BEFORE REMOVING FROM RACK TO PREVENT DAMAGE TO THE GYROS.

NOTE: Orient INS to aircraft heading.

1) Set mode selector switch on MSU to OFF.

2) Open following circuit breakers on P5 circuit breaker panel.

NAVIGATION
INS NO. 1
INS NO. 1 HTR

NOTE: Fasteners should pop out approximately 0.25 inch when released

(b) Open circuit breaker on front panel of battery unit No. 1.

WARNING: NAVIGATION UNIT WEIGHTS APPROXIMATELY 53 POUNDS. USE TWO MEN WHEN REMOVING NAVIGATION UNIT.

CAUTION: ALLOW 2 MINUTES FOR GYROS TO RUN DOWN BEFORE REMOVING UNIT OR GYRO AND ACCELEROMETER DAMAGE MAY RESULT. USE CARE DURING REMOVAL TO PREVENT DAMAGE TO AIRFLOW SEAL.

(c) Release cam-lock handle locking lever and remove navigation unit.

(e) Install plenum cover plate If cooling air is applied to navigation equipment rack.

(f) Connect tilt table to NU rack connector using extension cables (test cable W1 to J1A test cable W2 to J2A).

(g) Install tilt table adapter on tilt table.

(h) Install NU on tilt table.

(i) Close circuit breaker on front panel of batter unit No. 1.

(j) Secure door in closed position taking care not to crush extension cable.

(k) Level tilt table in pitch and roll 0 (±1) degree.

Ref.: DRW CDRL 43/44 Issue D dated May 19189
Paragraph 7.36

(4) INS Activation.

(a) Activate both (if required) INS system as follows:

1) Turn on Air Pressure Warning Alarm System and provide forced air cooling.

CAUTION: IF THE AIR PRESSURE WARNING ALARM SOUNDS, REMOVE POWER FROM ALL EQUIPMENT COOLED BY THE FORCED AIR SYSTEM.

DURING OPEN CABINET MAINTENANCE, THE ABSENCE OF AN AIR PRESSURE WARNING ALARM SOUND DOES NOT ASSURE THAT OPEN CABINETS APE RECEIVING ADEQUATE FORCED AIR COOLING. ALL EXISTING OPEN CABINET COOLING CONSTRAINTS MUST BE STRICTLY OBSERVED.

Note : Air Press, Warn Alarm syst. is an E-3A syst.

2) On circuit breaker panel PS check that following circuit breakers are closed:

NAVIGATION
INS NO. 1
NO. 1 HTR
NO. 2
NO. 2 HTR

ESS FLIGHT INSTRUMENT BUS
XFMR NO. 1
BPG NO. 1
ATT EXC
HDG EXC
COMPASS NO. 1
VG NO. 1

FLIGHT INSTRUMENT BUS
VG NO.2
COMPASS NO. 2
XFMR NO. 2
ATT EXC
BRG EXC
HDG EXC

AIR DATA
ALTM VIB NO. 1
AIR DATA
ALTM VIB NO. 2

CAUTION: IF WARN LIGHT ON CONTROL DISPLAY UNIT COMES ON WHILE

Ref.: DRW CDRL 43/44 Issue D dated May 19/89
Paragraph 7.36

3) Set mode selector to ALIGN on mode select unit.

4) Set data selector to DSRTK/STS on CDU.

5) Wait unit second digit from right in right data display on CDU displays 8 or below.

NOTE: Two to 60 minutes may be required for the 9 to change to 8 on CDU, depending on ambient temperature. Leave alternate INS in ALIGN.

6) Place mode selector to ATT on mode select unit or INS to be operated.

(5) Engage Interlocks and Warning.

(a) Autopilot engagement

NOTE: Whenever the control wheel is to be displaced, care should be exercised to assure travel is not restricted by the control system being up against its stops at either end of travel. Prior to each test requiring aileron control tab displacement, the control wheel shall be turned (CW & CCW) to its maximum possible position in each direction. The three axis trim indicator shall show a trimmed condition. If there is an out-of trim indication when the wheel stops, the wheel should be manually moved until in-trim is indicated.

1) If required, activate autopilot and INS system. Refer to paragraphs B. (1) and B. (4).

2) Set STAB TRIM switches on control stand to NORMAL.

3) Set autopilot mode select switch to MAN on AUTOPILOT CONTROLLER.

4) Set TURN knob on AUTOPILOT CONTROLLER in detent.

4a) For N19997 and N20000, ENGAGE YAW DAMPER Syst.

5) Set AUTOPILOT engage switch to AUTOPILOT on AUTOPILOT CONTROLLER.

6) Check autopilot engagement by noting increased effort required to move control column and control wheel. AP WARN lights are off.

7) Rotate TURN knob on AUTOPILOT CONTROLLER clockwise and counterclockwise; control wheel rotates in direction that TURN knob moves.

8) Rotate pitch wheel on AUTOPILOT CONTROLLER in nose UP and nose DOWN direction; control column moves aft and forward.

Ref.: DRW CDRL 43/44 Issue D dated May 19/89
Paragraph 7.36

b) Autopilot Disengagement and Warning.

1) If required, activate autopilot and INS system. Refer to paragraphs B.(1) and B.(4).

2) With autopilot engaged, momentarily press autopilot disconnect switch on pilot control wheel; autopilot disengages and control column and control wheel move normally and AP WARN lights on main instrument panels flash red. Turn lights off by pressing the switch again.

3) Engage autopilot and repeat step 2) using disengage switch on copilot control wheel.

4) With autopilot engaged, set AUTOPILOT engage switch to OFF on AUTOPILOT CONTROLLER. AP WARN lights flash red. Turn lights off by pressing light cap.

5) Momentarily press light cap on pilot or copilot AP WARN light. Lights come on when caps are pressed and go off when caps are re leased.

6) With autopilot engaged, open AUTOPILOT 3Ø circuit breaker on overhead circuit breaker panel. Autopilot engage switch returns to OFF and AP WARN lights flash red.

7) Set IND LIGHTS switch to DIM; intensity of AP WARN lights decreases. Return switch to BRT.

8) Close AUTOPILOT 30 circuit breaker and engage autopilot; AP WARN lights go OFF.

(6) VG1 Attitude Valid.

(a) If required, activate autopilot and VG1 system. Refer to paragraph B.(1)

(b) With autopilot engaged, pull VG NO 1 circuit breaker.

(c) Check for GYRO Flag in CAPT. ADI, with ADI source selector in VG.

(d) Check that in the same time, the A/P Engage switch retours to OFF.

Ref.: DRW CDRL 43/44 Issue D dated May 19/89
Paragraph 7.36

(e) Set Yaw Damper switch to ON on overhead panel Yaw Damper control panel. For N20199 ONLY

(f) Set AUTOPILOT engage switch to AUTOPILOT on AUTOPILOT CONTROLLER; autopilot engages.

(7) Turn Knob Interlock.

(a) If required, activate autopilot and INS system. Refer to paragraphs B.(1) and B.(4).

(b) Set TURN knob on AUTOPILOT CONTROLLER out of detent clockwise.

(c) Set AUTOPILOT engage switch on AUTOPILOT CONTROLLER to AUTOPILOT; switch will not hold but returns to OFF.

(d) Repeat steps (b) and (c) for counterclockwise rotation of TURN knob.

(e) Return TURN knob to detent.

(8) Mode Interlocks

(a) NAV Mode - INS Interlocks.

1) If required, activate autopilot and INS system. Refer. to para graphs B.(1) and B.(4)

2) With autopilot engaged, set pilot NAV MODE switch to INS on pilot glare shield.

3) Set autopilot mode select switch to NAV/LOC on AUTOPILOT. CONTROLLER.

4) Set pilot NAV MODE switch to TACAN; autopilot mode select switch returns to MAN.

(9) Flight Mode Annunciator Lights.

(a) If required, activate autopilot per paragraph B. (1)

(b) Press A/P ANNTEST indicator on pilot instrument panel; GS, VL, TACAN INS and TEST indicators come on.

Ref.: DRW CDRL 43/44 Issue D dated May 19/89
Paragraph 7.36

CAUTION: DO NOT SET IND LIGHTS SWITCH TO TEST; EQUIPMENT MAY BE DAMAGED. E-3A ONLY

(c) With TEST indicator pressed, set IND LIGHTS switch to DIM; pilot annunciator indicator light intensity decreases. Return switch to BRT.

(d) Repeat step (c) using copilot annunciator.

(10) INS Steering Mode.

(a) NAV Steering Gain.

1) If required, activate autopilot, place INS. 1 on tilt table and activate INS system. Refer to paragraphs B.(1), B. (3) and B.(4)

2) Set pilot NAV MODE selector to INS on glare shield.

3) Set AUTOPILOT INS SOURCE selector to INS 1 NORMAL.

4) Set INS 1 mode selector in ALIGN. Set INS 1 data selector on control display unit (CDU) to any position except DSRTK/STS and set AUTO MAN switch to MAN.

5) With control wheel centered and autopilot engaged, set autopilot mode select to NAV LOC. on AUTOPILOT CONTROLLER; pilot and copilot flight. mode annunciator INS lights come on green (engaged).

6) Press and hold TEST switch on INS 1 CDU; control wheel rotates.

7) Tilt VG1 to simulate right roll until control wheel is centered; tilt required is 15 (+2.1) degrees.

8) Release TEST switch on CDU and level INS 1.

9) Set INS 1 CDU AUTO MAN switch to AUTO.

10) Press and hold TEST switch on CDU; control wheel rotates.

11) Tilt VG1 to simulate left roll until control wheel is centered; tilt required is 15 (+2.1) degrees.

12) Release TEST switch on CDU and level INS 1.

Note: Test 7) and 11) are optinal.

(11) VOR Capture

(a) If required, activate autopilot, VOR. navigation, and VG1 systems.

Ref.: DRW CDRL 43/44 Issue D dated May 19/89
Paragraph 7.36

Refer to paragraphs B. (1), B. (2),

(b) Set pilot NAV MODE selector to VOR/LOC on glare shield.

(c) Set VOR navigation receiver to VOR test frequency (108.00 mHz) with a simulate 270 degree radial signal and a simulated heading of 90 degrees as follows:

1) Set DG/MAG switch to MAG on overhead DG control panel No 1.

2) Slew HSI compass card to 90° using the EDG control on HSI.

3) Set T-30B/B ramp test set VOR course control to TO 270°.

NOTE: Hold down VOR POWER switch on ramp test set for duration of test.

(d) Set pilot HSI COURSE selector to provide 1-1/2 dot fly right indication.

(e) With autopilot engaged, set autopilot mode select switch to NAV/LOC on AUTOPILOT CONTROLLER; pilot and copilot flight mode VL annunciator lights come on amber (armed).

(f) Slowly rotate HSI COURSE selector to reduce fly right indication until control wheel rotates clockwise; this occurs between 1/2 and one dot fly right indication. VL amber lights go off and VL green (engaged) lights come on.

(g) Rapidly rotate HSI COURSE selector to provide on course indication and then to one dot fly right indication; control wheel rotates rapidly counterclockwise then clockwise following COURSE selector then continues rotating slowing clockwise.

(12) VOR Course Gain.

(a) If required, activate autopilot and VOR navigation system, place VG 1 on tilt table, and activate INS system. Refer to paragraphs B. (1), B. (2) , B. (3).

(b) With pilot NAV MODE selector set to VOR/LOC on glare shield, and VOR navigation receiver set to VOR test frequency (of 108.00 mHz), open FLIGHT INSTRUMENTS NO. 1 BRG circuit breaker.

(c) Set pilot HSI (deviation bar centered) heading bug to 5 degrees right of aircraft heading.

Ref.: DRW CDRL 43/44 Issue D dated May 19/89
Paragraph 7.36

(d) Engage autopilot. and set autopilot mode select switch to NAV/LOC on AUTOPILOT CONTROLLER; control wheel rotates clockwise.

(e) Tilt VG 1 to simulate right roll until control wheel is centered; tilt required is between 6 and 14 degrees.

(f) Level VG 1 and close FLIGHT INSTRUMENTS NO. 1 BRG circuit breaker.

(13) VOR Station Passage.

(a) If required, activate autopilot, VOR navigation, and VG 1 System. (Refer to paragraphs B. (1), B. (3).

(b) With pilot NAV MODE selector set to VOR/LOC on glare shield and VOR navigation receiver set to VOR test frequency (or 108.00 mHz), set pilot HSI COURSE selector to aircraft heading.

(c) Set protractor for 20° left.

(d) With autopilot engaged, provide 112 dot fly right deviation signal the navigation receiver as follows:

1) Set T-30B/B ramp test set VOR course control to VARIABLE

2) Hold down ramp test set VOR POWER switch and rotate VOR VARIABLE control for a 1/2 dot fly right deviation signal.

NOTE: Hold down VOR POWER switch for duration of test.

(e) Set autopilot mode select switch to NAV/LOC on AUTOPILOT CONTROLLER; control wheel rotates slowly clockwise. Using stop watch, time and note rate of wheel rotation.

(f) Reduce deviation signal to below zero; control wheel slowly returns to past center.

(g) Increase deviation signal for two dots fly right indication; control wheel rotates clockwise at approximately three times the time inter val noted in step (e). (Timing from 0 to 20 degrees.)

(14) TACAN Mode.

(a) TACAN Capture.

1) If required, activate autopilot and, VG1 system. Refer to pargraphs B.(1).

2) Set pilot NAV MODE selector to TACAN on glare shield.

Ref.: DRW CDRL 43/44 Issue D dated May 19/89
Paragraph 7.36

3) Simulate TACAN bearing of 90 degrees using TACAN test set and an aircraft heading of 270 degrees as follows:

a) Check that following circuit brelkers are closed on P5 overhead panel:

FLIGHT NAVIGATION NO. 1 (or No. 2)
TACAN

FLIGHT INSTRUMENTS NO. 1 (or No. 2)

VG No. 1 (or No. 2)
COMPASS NO. 1 (or No. 2)
ESS FLIGHT INSTRUMENT BUS
XMFR No. 1 (or No. 2)

FLIGHT INSTRUMENTS NO. 1 (or No. 2)
XFMR-BRG

FLIGHT INSTRUMENTS NO. 1 (or No. 2)
XFMR-HDG

b) Set TACAN control panel mode switch to REC and X/Y switch to X.

c) Select low band channel marked on AN/ARM-25C test set on TACAN control panel.

d) Set COMP system heading to 270 degrees.

e) Disconnect wiring cables from receiver-transmitter upper (J1) and lower (J2) ANTENNA receptacles.

f) Connect AN/ARM-25C test set RF output cable to receiver-transmitter upper ANTENNA receptacle J1.

g) Connect test set power cable to equipment rack test power outlet.

h) Connect 50-ohm termination to ANTENNA or 50 TERM receptacle on test set.

i) Place test set ON-OFF switch to OFF. Allow a 30 minute warmup period. Test set STANDBY light should be on indicating chassis heater is energized.

j) Position test set controls as follows:

Ref.: DRW CDRL 43/44 Issue D dated May 19/89
Paragraph 7.36

CONTROL	POSITION
POWER/TUNE/OPERATE	OPERATE
IDENTITY	RANDOM
AZIMUTH	OFF
RANGE MILES	150
OUTPUT ATTENUATION DB	10
ON-OFF	ON
CHANNEL	LOW BAND (LEFT)
A/A MODE PRF	OFF

k) Check that teat set OPERATE light. is on and that blower is operating.

l) Adjust test set LEVEL SET control for a zero dBm indication on power level meter.

m) Set test set AZIMUTH switch to 90°. Power level meter indication will normally increase about one dBm when AZIMUTH switch is switched from OFF to 90 or 180° positions.

n) Set TACAN control panel mode selector switch to T/R.

4) With autopilot engaged, set pilot HSI COURSE selector to provide 1-1/2 dot fly right indication.

5) Set autopilot mode select switch to NAV/LOC on AUTOPILOT CONTROLLER; pilot and copilot flight mode TACAN annunciator lights come on amber (armed).

6) Slowly rotate HSI COURSE selector to reduce fly right indication until control wheel rotates clockwise; this occurs between 1/2 and one dot fly right indication. Flight mode-TACAN annunciator amber lights go off and TACAN green (engaged) lights come on.

7) Turn off TACAN and remove test set.

(b) TACAN Displacement Gain.

1) If required, activate autopilot, place VG 1 on tilt table, and activate INS systems. Refer to paragraphs B. (1), B. (3).

2) With pilot NAV MODE selector set to TACAN on glare shield, provide simulated TACAN bearing of 180 degrees and aircraft heading of zero degrees as follows:

Ref.: DRW CDRL 43/44 Issue D dataed May 19/89
Paragraph 7.36

a) Check that following circuit breakers are closed on P5 overhead panel:

FLIGHT NAVIGATION No. 1 (or No. 2)
TACAN

FLIGHT INSTRUMENTS No. 1 (or No. 2)
VG NO. 1 (or No. 2)
COMPASS No. 1 (or No. 2)
ESS FLIGHT INSTRUMENT BUS
XMFR No. 1 (or No. 2)

FLIGHT INSTRUMENTS No. 1 (or No. 2)
XFMR-BRG

FLIGHT INSTRUMENTS No. 1 (or No. 2)
XFMR-HDG

b) Set TACAN control panel mode switch to REC and X/Y switch to X.

c) Select low band channel marked on AN/ARM-25C test set on TACAN control panel.

d) Set COMP system heading to 0.

e) Disconnect wiring cables from receiver-transmitter upper (J1) and lower (J2) ANTENNA receptacles.

f) Connect AN/ARM-25C test set RF output cable to receiver-transmitter upper ANTENNA receptacle J1.

g) Connect test set power cable to equipment rack test power outlet.

h) Connect 50-ohm termination to ANTENNA or 50 TERM receptacle on test set.

i) Place test set ON-OFF switch to ON . Allow to 30 minute warmup period. Test set STANDBY light should be on indicating chassis heater is energized.

j) Position test set controls as follows:

CONTROL	POSITION
POWER/TUNE/OPERATE	OPERATE
IDENTITY	RANDOM
AZIMUTH	OFF

Ref.: DRW CDRL Issue D dated May 19/89
Paragraph 7.36

RANGE MILES	150
OUTPUT ATTENUATION DB	10
ON-OFF	ON
CHANNEL	LOW BAND (LEFT)
A/A MODE PRF	OFF

k) Check that test set OPERATE light is on and that blower is operating.

l) Adjust test set LEVEL SET control for a zero dBm indication on power level meter.

m) Set test set AZIMUTH switch to 180°. Power level meter indication will normally increase about one dBm when AZIMUTH switch is switched from OFF to 90 or 180°positions.

n) Set TACAM control panel mode selector switch to T/R.

3) With autopilot engaged, set pilot HSI COURSE selector to zero.

4) Provide TACAN test signal and tune TACAN receiver to test signal.

5) Tilt INS 1 to simulate right roll of 12 degrees; control wheel rotates counterclockwise.

6) Set autopilot mode select switch to NAV LOC.

7) Set HSI COURSE selector to provide 1/4 dot fly right indication while simultaneously changing aircraft (slew HSI compass card) heading to maintain zero course error; control wheel rotates clockwise towards center.

8) After 30 seconds, adjust VG 1 tilt to center control wheel; tilt required is between 8 and 16 degrees.

NOTE: Control wheel may 90 beyond center and require a roll beyond 12 degrees to center control wheel.

9) Disengage autopilot and level INS 1.

10) Turn off TACAN

(15) LOC Mode.

(a) Localizer Capture.

1) If required, activate autopilot, VOR navigation, and VG systems. Refer to paragraphs B.(2), B.(3).

Ref: DRW CDRL 43/44 Issue D dated May 19/89
Paragraph 7.36

2) Set pilot NAV MODE selector to VOR/LOC on glare shield.

3) Select LOC frequency and provide simulated localizer signal equivalent to three dots fly right as follows:

a) Tune VOR navigation receiver No. 1 to 108.10 mHz.

b) Set T-30B/B ramp test set LOC course control to VARIABLE.

c) Hold down test set LOC POWER switch and rotate LOC VARIABLE control for an indication of three dots fly right indication on pilot HSI.

NOTE: Hold down ramp test set LOC POWER switch for duration of test.

4) Set pilot HSI COURSE selector to aircraft heading.

5) Engage autopilot and after 5 seconds set autopilot mode select switch to NAV LOC on AUTOPILOT CONTROLLER; control wheel remains centered and pilot and copilot flight mode VL annunciator light come on amber (armed).

6) Slowly decrease simulated localizer signal until control wheel rotates rapidly clockwise; this occurs prior to HSI deviation needle reaching 1-1/2 dot indication. Flight mode VL annunciator amber lights go off and VL green (engaged) lights come on.

C. Restore Aircraft to Normal.

(1) Open following circuit breakers on P5 overhead circuit breaker panel:

AUTOPILOT 3Ø
YAW DAMPERS SERIES ØA
ESS FLIGHT AVIONICS
FLT DIR NO. 1 COMPT

(2) Turn off INS as follows:

(a) Set mode selector to OFF on system mode control.

(b) Set mode selector to OFF on each INS mode select unit

(c) Open following circuit breakers on circuit hreaker panel P5:

Ref.: DRW CDRL 43/44 Issue D dated May 19/89
Paragraph 7.36

NAVIGATION

INS
NO 1
NO 1 HTR
NO 2
NO 2 HTR
OMEGA NAV
RCVR CMPTR

ESS FLIGHT INSTRUMENT BUS
XFMR ATT
XFMR NO. 1
BRG EXC
HDG EXC
VG NO 1
AUTOPILOT 3Ø
NAV XFER RYL NO. 1
FLT DIR NO. 1

ESS RADIO BUS
VOR NO. 1 (AC & DC)
GS NO. 1

RADIO BUS NO. 2
VCR NO. 2 (AC & DC)
GS NO. 2

FLIGHT INSTRUMENT BUS NO. 2
VG NO. 2
COMPASS NO. 2
XFMR ATT
XFMR NO. 2
HDG EXC
BRG EXC
NAV XFER RLY NO. 2
FLT DIR NO. 2
AIR DATA
ALTM VIB NO. 1
AIR DATA
ALTM VIB NO. 2
COMPASS NO 2

(3) Remove navigation system test equipment.

(4) Open circuit breaker on front panel of INS battery unit No. 1.

(5) Remove INS test cables from rack.

(6) Remove and store plenum cover.

<u>CAUTION</u>: INS MUST BE OFF AT LEAST 2 MINUTES BEFORE REMOVING FROM TILT TABLE TO PREVENT DAMAGE TO GYROS.

Ref.: DRW CDRL, Issue D dated May 19/89
Paragraph 7.36

(7) Remove INS from tilt table and install in rack as follows:

WARNING: NAVIGATION UNIT WEIGHS APPROXIMATELY 53 POUNDS. USE TWO MEN WHEN LIFTING NAVIGATION UNIT.

CAUTION: USE CARE WHEN INSTALLING NAVIGATION UNITS TO PREVENT TEARING AIRFLOW SEAL.

(a) Slide navigation unit onto shelf. Engage cam-lock locking levers in shelf fork assemblies and actuate to fully seat connectors at rear of unit.

CAUTION: IMPROPER ADJUSTMENT OF CAM-LOCK LEVER LATCH FORK ASSEMBLIES ON EQUIPMENT SHELF MAY CAUSE SYSTEM MALFUNCTION OR EQUIPMENT DAMAGE.

(b) When connectors are fully seated, a noticeable increase in pressure is felt in locking levers. At this point, check that locking levers are 1/4-inch from locked position. if not, pull lever down and thread fork assembly in or out to obtain dimension within tolerance.

(c) Seat locking levers into handles.

(d) Close circuit breaker on front panel of applicable INS battery unit.

(8) Remove hydraulic power if no longer required for other maintenance.

(9) Remove electrical power and draw through cooling air from aircraft if no longer required for other maintenance.

(10) Remove forced air cooling and disconnect Air Pressure warning Alarm System if no longer required for other maintenance.. E-3A ONLY

Note: Item (4) to (7) for E-3A ONLY
If VG1 was installed on the TILT TABLE, reinstal VG1.

Ref.: DRW CDRL 43/44 Issue D dated May 19/89
Paragraph 7.36