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AUTOFLIGHT - DESCRIPTION AND OPERATION

1. General
   A. The autoflight systems consist of an Autopilot and Yaw Damper System and a Mach Trim System. Partial provisions for an Auto Throttle System are also installed on the airplane.
2. Autopilot and Yaw Damper Systems
   A. Autopilot System
      (1) The autopilot system controls the airplane about the pitch and roll axes and provides automatic airplane stabilization whenever the pitch and roll channels are engaged. Automatic stabilizer trim, which is a function of the pitch channel, compensates for changes in airplane trim due to fuel burnoff etc. Each autopilot channel may be engaged separately and, in some modes, are not dependent upon each other. Various mode selections enable the pilots to command the autopilot to fly the airplane onto a selected heading, maintain altitude, or fly to a selected VOR or localizer course as well as make automatic approaches to runways equipped with ILS facilities. The pilots may also manually control the airplane in a normal manner with the control wheel/column (control wheel steering) without disengaging the pitch or roll axes of the autopilot system. Pilots can then assist the autopilot system in flying to a selected heading or course. Use of control wheel steering does not disengage either channel of the autopilot system.
   B. Yaw Damper System
      (1) The yaw damper system provides airplane stabilization about the yaw axis. Automatic damping of dutch roll is sensed and corrected for by the yaw damper system whenever the system is engaged. Rudder deflections are not felt by the pilots since the rudder pedals are not moved by the yaw damper system. No interlocks are provided between the yaw damper system and the autopilot system.
      (2) The autopilot and yaw damper systems operate in conjunction with hydraulic power control units which drive the airplane ailerons, elevators and rudder (AMM Chapter 27, Flight Controls).
3. Mach Trim System
   A. The mach trim system provides automatic repositioning of the elevators as a function of mach number. As the airplane enters the mach tuck region, the elevator is repositioned to provide a new neutral in an upward direction which is proportional to the increase in mach. The mach trim system operates with or without the autopilot system engaged.
   B. The mach trim system operates in conjunction with the elevator hydraulic power control units and stabilizer/elevator neutral shift mechanism (AMM Chapter 27, Elevator and Tab Control System).

4. Auto Throttle System

A. The auto throttle system (if installed) automatically monitors airplane airspeed and adjusts throttle settings to maintain a selected airspeed. The auto throttle system is intended to assist the pilots in maintaining a constant airspeed when making landing approaches. The auto throttle system is independent of the autopilot system and operates with or without the autopilot system engaged.

AUTO FLIGHT - MAINTENANCE PRACTICES

1. General
   A. In order to maintain the integrity of the auto flight systems, it may become necessary to replace electro-hydraulic components on the elevator, aileron, and rudder power control units. This would involve handling BMS 3-11 hydraulic fluid. The following information is provided to aid in handling BMS 3-11.
2. BMS 3-11 Handling Procedure
   A. BMS 3-11 is the general specification for the fluid that is used in all hydraulically operated systems. BMS 3-11 is a fire-resistant base fluid to which suitable additives have been incorporated. All fluids meeting this specification can be intermixed in any amount with no separation, precipitation or cloudiness. All airplane equipment using BMS 3-11 are identified by nameplates.

      CAUTION: BMS 3-11 FLUID IS NOT COMPATIBLE WITH MIL-H-5606 FLUID. MIXING THESE FLUIDS, IN ANY AMOUNT, WILL COMPLETELY RUIN THE INTEGRITY OF THE HYDRAULIC SYSTEM.

   B. Areas where hydraulic fluid may leak are designated as possible BMS 3-11 contamination areas. Special paint and protective finishes are used to prevent damage in these areas. Refer to Chapter 51, Structures - Protective Finishes.
   C. BMS 3-11 has little or no effect on the metals used in aircraft construction up to 240°F. Only materials compatible with BMS 3-11 fluids, such as ethylene propylene, butyl rubber and teflon, are used for system seals, gaskets, O-rings, and hose linings.
   D. Skin irritation in the form of dry or cracked skin may result from prolonged or repeated contact with BMS 3-11. It also causes painful but temporary irritation to the eyes and may produce a burning sensation to other sensitive parts of the body. Inhalation of the fluid in the form of spray of fine mist may cause irritation of the upper respiratory tract. To minimize contact with BMS 3-11, wear gloves, goggles, face shields or safety glasses whenever there is danger of exposure. If exposure occurs, flush the eyes immediately with water and report to medical for observation.

      WARNING: BMS 3-11 HEATED TO TEMPERATURES ABOVE 450°F CAN BE DECOMPOSED INTO TOXIC FUMES AND GASES. AVOID INHALATION OF FUMES AND VAPORS FROM OVERHEATED BMS 3-11. IF IT IS NECESSARY TO WORK IN THESE FUMES AND VAPORS, SAFETY APPROVED EYE AND RESPIRATORY PROTECTION ARE MANDATORY.

AUTOPILOT AND YAW DAMPER SYSTEM - DESCRIPTION AND OPERATION

1. General
   A. The Sperry SP-77 Automatic Flight Control System (autopilot and yaw damper system) provides automatic operation of airplane flight controls to maintain altitude, attitude and heading references, yaw damping, automatic tracking and control of omnirange navigation throughout the various flight regimes, and automatic approach.
   B. The autopilot and yaw damper system is characterized by the following features:
      (1) Control wheel steering (CWS) for pitch and bank rate maneuvering.
      (2) Full time series yaw damping with control cabin self-test and full-time yaw damper position indication.
      (3) All angle radio beam capture with control wheel steering maneuvering.
      (4) Solid state (microelectronic) interlock, and operating circuits.
      (5) Plug-in modules on hinged cards for ease of maintenance.
   C. The autopilot and yaw damper attain 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 system integrity are attained by integrating the displacement errors with respect to time and by making control surface movement a function of indicated airspeed. The following is a brief explanation of these principles of operation:
      (1) 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 forces based on this rate of change 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 needed. Thus, considerable 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 damped by input 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.

(2) Integration in the autopilot provides an extremely accurate means of automatically retaining reference flight conditions when sustained or recurring displacement 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 corrective 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. The integrated signal remains at the value required to overcome the displacement error, compensating for required changes in the original flight references.

(3) The response of the airplane to control surface movement is aerodynamically a function of 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 control of the airplane at all airspeeds, continuous adjustment of the autopilot channel control surface gains is provided.

D. Individual components of the autopilot and yaw damper system are shown in Fig. 1. The following are the associated airplane electronic systems upon which successful autopilot operation depends (Ref Chapter 34).

(1) Vertical gyros
(2) Compass (directional gyros)
(3) Air data computer (ADC)
(4) Low range radio altimeter (when installed)
(5) VHF NAV and glide slope receivers
(6) Flight instruments

NOTE: Refer to Chapter 27 for airplane rudder, aileron and elevator power control units.

E. Yaw axis control is provided by the yaw damper. The yaw damper minimizes dutch roll during manually and automatically controlled flight by providing rudder displacement proportional to, and opposing, the yaw rate of the airplane. Since a series hydraulic actuator tie-in is used, there is no damper control feed back to the rudder pedals. This allows the pilot to maneuver the airplane in a normal manner without having an opposing force from the yaw damper.

F. Pitch axis control is provided by the pitch control channel and associated components. The operating modes of the pitch axis are the manual mode, including pitch attitude control and pitch commands through control wheel steering; altitude hold; glide slope control during ILS, with automatic or manual capture; automatic pitch (stabilizer) trim; and turbulence control. The pitch control channel operates in conjunction with the airplane vertical gyro, air data computer, glide slope receiver systems and the radio altimeter. The vertical gyro system supplies pitch angle signals, the air data computer supplies altitude error and altitude rate signals and the glide slope receiver supplies glide slope deviation signals, and the radio altimeter provides signals for glide slope gain programming. The output of the pitch control channel is applied to the transfer valve of the elevator power control unit to deflect the elevators. The pitch channel modes are selected by the mode selectors on the autopilot control panel.

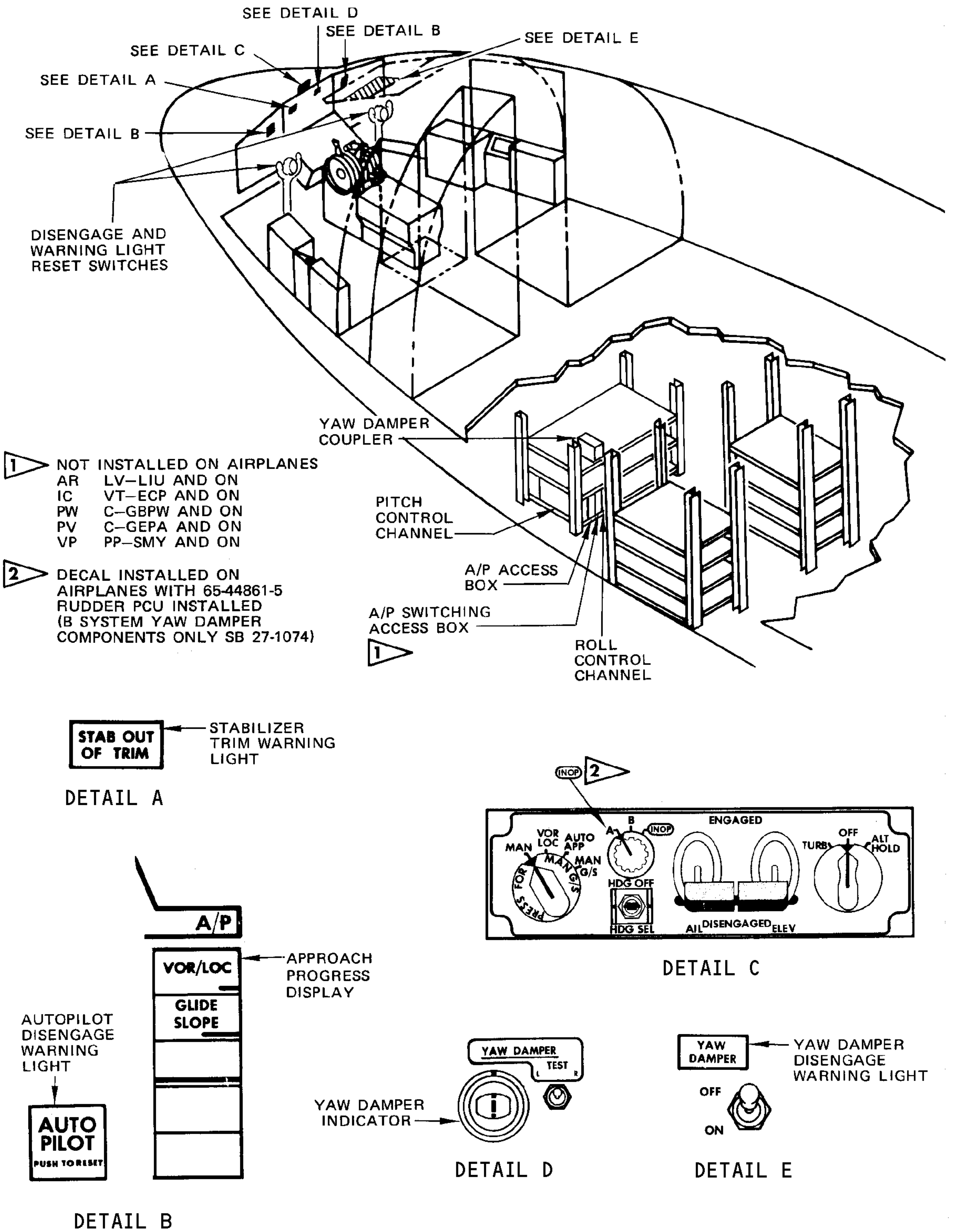
G. Roll axis control is provided by the roll control channel and associated components. The operating modes of the roll axis are the manual mode, including heading and roll attitude control and roll commands through control wheel steering; heading select; heading hold, and localizer and omnirange control. The roll control channel operates in conjunction with the airplane compass (directional gyro) flight instrument, vertical gyro, VOR navigation systems and the radio altimeter. The compass system supplies heading reference, the flight instrument system supplies preset heading and preset course signals, the vertical gyro supplies bank angle signals, the VOR navigation system supplies localizer and omnirange deviation signals and the radio altimeter supplies signals for localizer gain programming.

H. The autopilot and yaw damper are single channel systems. Provisions are made for a complete dual channel system with a dual flare coupler for all weather landing capability.

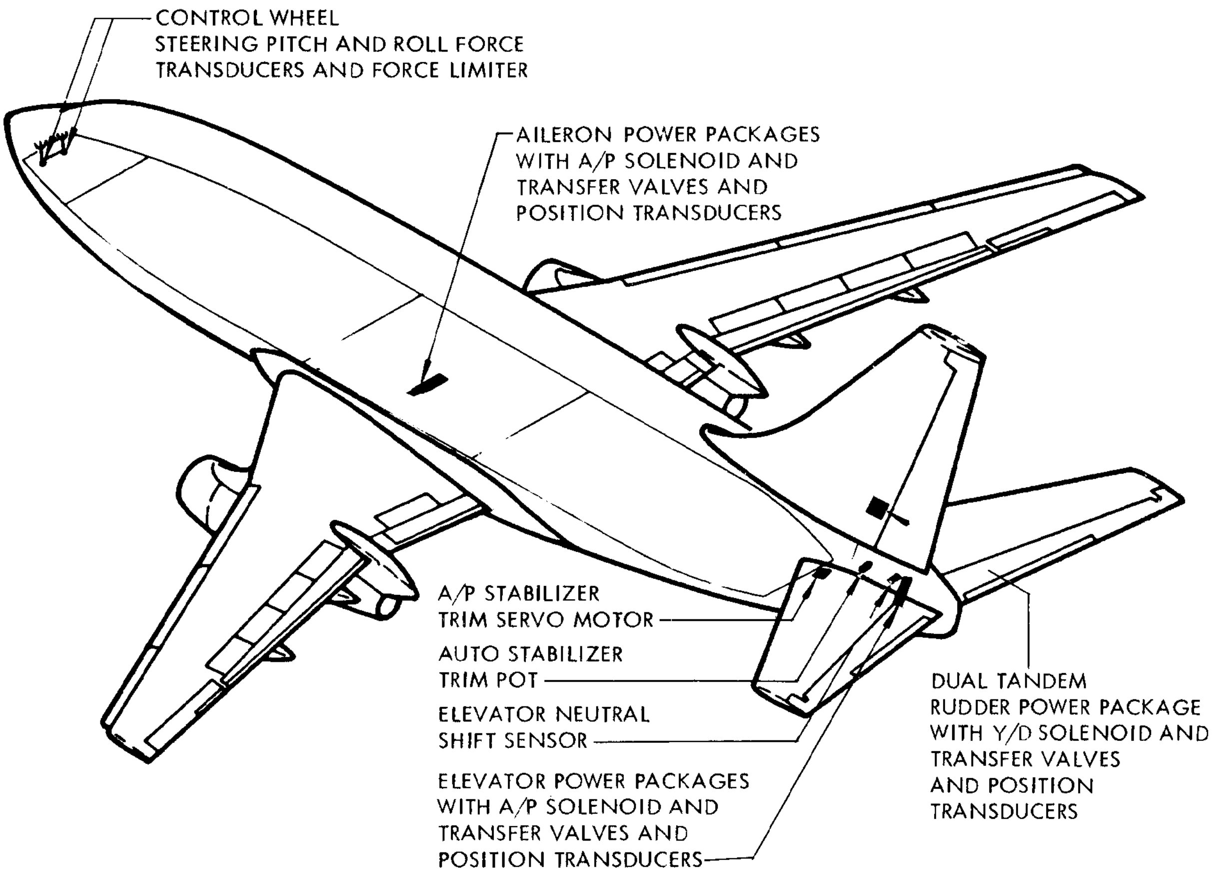
I. Since the SP-77 automatic flight control system makes maximum use of solid state microelectronics and high level digital logic circuits, circuit functions throughout the description and operation are assigned two values (0, 1) of the binary numbers system. An active circuit is considered a 1 state and an inactive circuit is considered a 0 state. The assignment of a 1 or 0 is based on the following example: ROLL ENGAGE is 1 provided the system is roll engaged, also ROLL ENGAGE is 1 provided the system is not engaged. If the system is not engaged, ROLL ENGAGE is 0. If the system is engaged, ROLL ENGAGE is 0. Circuit symbology is standard. Boolean algebra functions are used and described as follows:

(1) An AND circuit requires all inputs of a logic circuit to be high to obtain a high output. A * symbol is used to designate AND.

(2) An OR circuit requires any one input of a logic circuit to be high to obtain a high output. A + symbol is used to designate OR.



Autopilot and Yaw Damper System Component Location
Figure 1 (Sheet 1)



Autopilot and Yaw Damper System Component Location
Figure 1 (Sheet 2)

(3) A bar over a function designates a logical inversion (NOT function), e.g., if PITCH ENGAGE = 0, PITCH ENGAGE = 1; conversely, if PITCH ENGAGE = 1, PITCH ENGAGE = 0.

(4) A double NOT bar over a function negates the NOT function, e.g., PITCH ENGAGE = PITCH ENGAGE.

(5) By definition, all functions which actuate (close) a switch or complete a circuit are considered 1 when the affected circuit is in the same state as the noted function. For example, the function ALT HOLD+G/S ENG = SWITCH CLOSED (1) when the altitude hold is selected or when glide slope beam is captured. Also the function

(6) PITCH ENG*CWS OD = PITCH ENG + CWS OD = SWITCH CLOSED (1) when the pitch channel is not engaged or control wheel steering is not out of detent. Also the function G/S ENG*PLUS 10 SEC = G/S ENG+ PLUS 10 SEC = SWITCH CLOSED (1) when the pitch channel is in the glide slope intercept mode (glide slope not engaged) or the 10 second glide slope capture period has terminated (plus 10 seconds NOT NOT). The switch will open when the pitch channel is in the glide slope engage mode (the VBS has dropped out) and the 10-second timer has not clocked 10 seconds. This function may be shown as G/S ENG*PLUS 10 SEC = SWITCH CLOSED = 0.

J. To aid discussion of autopilot and yaw damper system operation, reference numbers are assigned to summing points and solid-state switches. Solid-state switches are presented by conventional switch symbols. The logic function opposite each switch indicates the condition(s) necessary to actuate (close) the switch. System gain adjustments, attenuation and parallel summing resistors are not shown on diagrams and schematics for clarity and simplification.

2. <u>Yaw Damper Components (All except coupler P/N 4084042)</u>

<u>NOTE</u>: For aircraft with coupler P/N 4084042 installed (incorporating SB 27A1206), see 22-12-01

A. Yaw Damper Coupler

(1) The yaw damper coupler is installed on the E1-1 shelf of the electronic equipment racks in the electrical and electronic compartment.

(2) The yaw damper coupler part No. 2588880 consists of a yaw rack assembly and several plug-in modules. The plug-in modules are the yaw rate gyro, yaw computer, yaw calibrator and provisions for an optional turn indicator amplifier that may be used with a remote turn indicator. The yaw rate gyro is identical to and interchangeable with the rate gyros of the pitch and roll channels. On the front of the chassis is a self-test switch and meter, and a gyro test switch. The yaw calibrator contains fixed resistors to adapt the gains of the yaw damper coupler to a specific airplane configuration.

(3) The yaw damper coupler part No. 4030952 contains solid-state components on modern plug-in cards with printed circuit boards. The cards and subassembly installed in the yaw coupler rack are: yaw computation, BITE, and power supply cards, and a rate sensor with yaw BITE panel subassembly. The front panel of the computer contains 2 lights (FAIL, PASS), a GO pushbutton for initiating each BITE test which is annunciated by a digital indicator, a LAMP TEST pushbutton for checking lamp continuity and a 2-position BITE toggle switch labeled OFF and ARM. Test points are on the lower portion of the panel.

(4) The rate gyro senses the rate of change of the airplane yaw attitude. It generates an ac output voltage with the magnitude and phase as a function of the angular yaw rate and direction of the motion. The gyro consists of a small, viscous-damped, single degree-of-freedom gyro composed of a motor and rotor, gimbal, signal pickoff, torsion bar and viscous damper. With a yaw rate, a precessional torque is developed by the gyro which displaces the gimbal. The torsion bar and viscous damper, attached to the gimbal, try to restrain it. The pickoff rotor, also attached to the gimbal, rotates within a stator which contains both primary (excitation) and secondary (output) windings. The angular position of the pickoff rotor due to the yaw rate determines the magnitude and sense of the output voltage.

(5) On airplanes with yaw damper coupler part No. 4030952-906 installed, yaw acceleration signal derived from the yaw rate gyro is used to provide improved ride quality characteristics. In addition, a roll signal from vertical gyro is used to provide turn coordination.

(6) The yaw computer provides for amplification and shaping of the rate signals for operation of the yaw damper hydraulic transfer valve and synchronizes the signal chain prior to engagement.

B. Yaw Damper Controls

(1) The yaw damper controls consist of circuit breakers, an engage switch, a disengage warning light, a test switch, a yaw damper indicator and a system select switch which is not used on later airplanes that have hydraulic system switching (A-B) removed and have (B) system only on autopilot control panel. Airplanes incorporating SB 27-1074 (rudder PCU with B system yaw damper components only) must position system select switch to (B) position for yaw damper operation (Fig. 1).

(2) Yaw damper power is supplied through two circuit breakers located on the main load control center (P6-2). One circuit breaker is for 115 volts ac and the other is for 28 volts dc.

(3) The YAW DAMPER ENGAGE switch is located on the overhead panel (P5) and is the solenoid held type. When the engage switch is placed to the ON position, and if the yaw damper engage interlocks are satisfied, the solenoid will hold the switch engaged. The yaw damper is then in the yaw damping mode.

(4) Whenever the yaw damper is, or becomes, disengaged, the amber light, labeled YAW DAMPER, on the overhead panel (P5-3) will illuminate steadily. The MASTER CAUTION lights and the FLT CONT master caution annunciator light, located on the light shield, will also illuminate.

(5) The YAW DAMPER TEST switch and indicator are located on the center panel (P2). When the yaw damper is operative; i.e., engaged and hydraulic power available, if the test switch is actuated, a test voltage is applied to the yaw damper coupler that torques the rate gyro. As the gyro is displaced, its output signal is sent through the normal signal chain and the rudder will deflect. The position transducer on the rudder power control unit senses the rudder displacement from neutral and sends a signal to the indicator which shows rudder deflection. If the test switch is placed left, L position, the bar on the indicator should deflect left and if the switch is placed right, R position, the bar should deflect right. In flight, the indicator shows rudder deflection from neutral resulting from yaw damper inputs.

(6) On airplanes with system switching, less airplanes incorporating SB 27-1074, the system select switch is on the autopilot control panel, which is located on the glare shield above the center instrument panel (P2). The switch has three positions; A, B, and AB; but for a single channel autopilot and yaw damper system, the AB position is locked out. With the switch in A position, the yaw damper system A-B transfer relay in the autopilot switching accessory box directs the yaw damper signals to and from the A system solenoid valve, transfer valve, and position transducer. The yaw damper is then supplied power by the A hydraulic power system. If the switch is placed to B position, the system A-B transfer relay then switches control to the B hydraulic system components on the rudder power control unit. With the switch in A position and if system A hydraulic pressure fails, the select switch can be placed in B position, thereby disengaging the yaw damper system. The yaw damper system must then be manually re-engaged.

(7) On airplanes with no system select switching, plus airplanes incorporating SB 27-1074, B position only, the yaw damper signals are applied to a single solenoid valve and transfer valve with feedback from a single position transducer. The system is supplied power by the B hydraulic system. Airplanes incorporating SB 27-1074 retain the system switching capability; however, system select switch must remain in B position for yaw damper operation.

C. Position Transducers

(1) There are two linear position transducers mounted on the rudder power control unit; one for the A hydraulic system and one for the B system. The transducers supply rudder position feedback signals to the yaw damper coupler. The transducers are variable reluctance transformers which are excited by 26 volts ac. The magnitude of the output signal varies directly with the length of the input stroke and the phase of the signal changes with the direction of the stroke from the center null position.

D. Transfer and Solenoid Valves

(1) The transfer and solenoid valves control hydraulic flow for yaw damper control at the rudder power control unit.

(2) The yaw damper solenoid valves are mounted on the rudder power control unit. They are solenoid actuated valves that are energized when the yaw damper is engaged and they supply hydraulic power to the transfer valves.

(3) The transfer valves, also mounted on the rudder power control unit, converts the yaw damper electrical signals into hydraulic flow to move the rudder. The unit consists of a torque motor which moves either a flapper valve or a jet pipe assembly which regulates the hydraulic flow to the control valve.

3. Autopilot Components

A. Pitch Control Channel

(1) The pitch control channel is installed on the El-3 shelf of the electronic equipment racks in the electrical and electronic compartment. The control channel consists of a pitch rack assembly and several plug-in modules which are the pitch rate gyro or rate deriving network, the vertical path coupler, the pitch computer, the pitch servo-amplifier, the control wheel steering (CWS) coupler and the pitch calibrator.

(2) The pitch rate gyro on pitch channel part No. 2588810-901, -902 senses airplane pitch attitude rate of change. This unit is identical to the yaw rate gyro but is oriented in the pitch control channel to detect airplane angular pitch rate. The rate deriving network, installed on pitch channel part No. 2588810-903, -904, develops a rate signal from the vertical gyro signal feeding the channel. Throughout the text, references to rate gyro implies the rate signal whether generated by the rate gyro or developed by the rate deriving network. The vertical path coupler receives dc signals from the glide slope receiver, and amplifies and modulates the signals to ac. The unit also provides for signal gain programming. The pitch computer provides pitch control error signals to the servo-amplifier and consists essentially of a vertical path filter and an electromechanical computer. The pitch servo-amplifier supplies the stabilization signals for operating the elevator actuator transfer valve. The control wheel steering coupler provides quadrature rejection and control wheel steering sensor level detection. The pitch calibrator contains fixed resistors to adapt the gains of the pitch control channel to a specific airplane configuration.

(3) The pitch rack assembly is the base of the pitch control channel and supplies the necessary support and interconnection for the plug-in modules. The unit houses the pitch axis interlock logic and power supplies. The front of the rack contains the self-test switch and meter, and the gyro test switch.

(4) The type of construction used in the pitch control channel is identical to the construction of the yaw damper coupler and some of the circuits are interchangeable.

B. Roll Control Channel

(1) The roll control channel is installed on the E1-3 shelf of the electronic equipment racks in the electrical and electronics compartment. The roll control channel consists of a roll rack assembly and several plug-in modules which are the roll rate gyro or rate deriving network, the lateral path coupler, the roll computer, the heading synchronizer, the roll servo-amplifier and the roll calibrator.

(2) The roll rate gyro on roll channel part No. 2588812-901 senses, airplane roll attitude rate of change. This unit is identical to the yaw and pitch rate gyros but is oriented in the roll control channel to detect airplane angular roll rate. The rate deriving network, installed on roll channel part No. 2588812-902, develop a rate signal from the vertical gyro feeding the channel. Throughout the text, references to rate gyro implies the rate signal whether generated by the rate gyro or developed by the rate deriving network. The lateral path coupler receives dc signals from the omnirange and localizer receiver. The unit provides ac turn command signals proportional to radio beam deviation to control the airplane along the desired lateral beam flight path. The roll computer provides roll command error signals to the roll servo-amplifier. The roll computer consists essentially of amplifier limiters and an electromechanical computer. The heading synchronizer supplies heading error signals and synchronizes heading prior to engagement. The roll servo-amplifier supplies stabilization signals to operate the aileron actuator transfer valve. The roll calibrator module consists of fixed resistors to adapt the gains of the roll control channel to a specific airplane configuration. The roll rack assembly is the base of the roll control channel and supplies the necessary support and interconnection for the roll axis modules. The unit houses the roll axis interlock logic and power supplies. The front of the rack contains a self-test switch and meter, and a gyro test switch.

(3) The construction of the roll rack assembly and the plug-in modules is the same as the yaw damper coupler and some of the circuits are interchangeable with yaw damper and pitch channel circuits.

C. Autopilot Control Panel, Controls, and Indicators

(1) The autopilot controls consist of circuit breakers, the autopilot control panel, the disengage and warning light reset switches, the approach progress display, the disengage warning lights, and the stabilizer out of trim warning light.

(2) All electrical power is supplied from electronic bus-1 to the autopilot through circuit breakers located on the main load control center (P6-2). The circuit breakers are labeled for the control channel or component to which they supply power and whether the power is 115 volts ac or 28 volts dc.

(3) The control panel is located on the glare shield above the pilots' center instrument panel where it is accessible to both the captain and first officer. The control panel contains the controls to enable the captain and first officer to engage the aileron and elevator axes and to select various modes of operation. These controls are the aileron and elevator engage switches, the navigation mode select switch, the heading off-heading select switch, the system select switch, and the pitch mode select switch.

(a) The aileron and elevator engage switches, labeled AIL and ELEV are two position switches. The positions are DISENGAGED and ENGAGED. The switches have a latch that holds the switch to the DISENGAGED position. If certain engage interlocks are satisfied, a solenoid releases the latches so that the switch may be moved to the ENGAGED position. When in the ENGAGED position, if certain other interlocks are satisfied, the solenoid will hold the switch engaged. If the interlocks are broken, the solenoid will release the switch to the DISENGAGED position. The engage interlocks are so arranged that the roll (AIL) engage switch does not require engagement before the pitch (ELEV) engage switch will engage.

(b) The navigation (NAV) mode select switch is a four-position rotary switch that is spring-loaded to the first position and solenoid held in the other positions. The positions are manual (MAN), omnirange/localizer (VOR LOC), automatic approach (AUTO APP) and manual glide slope (MAN G/S). The switch must be depressed and rotated to select the MAN G/S position. Certain interlocks must be satisfied for the switch solenoid to remain energized in each position except manual.

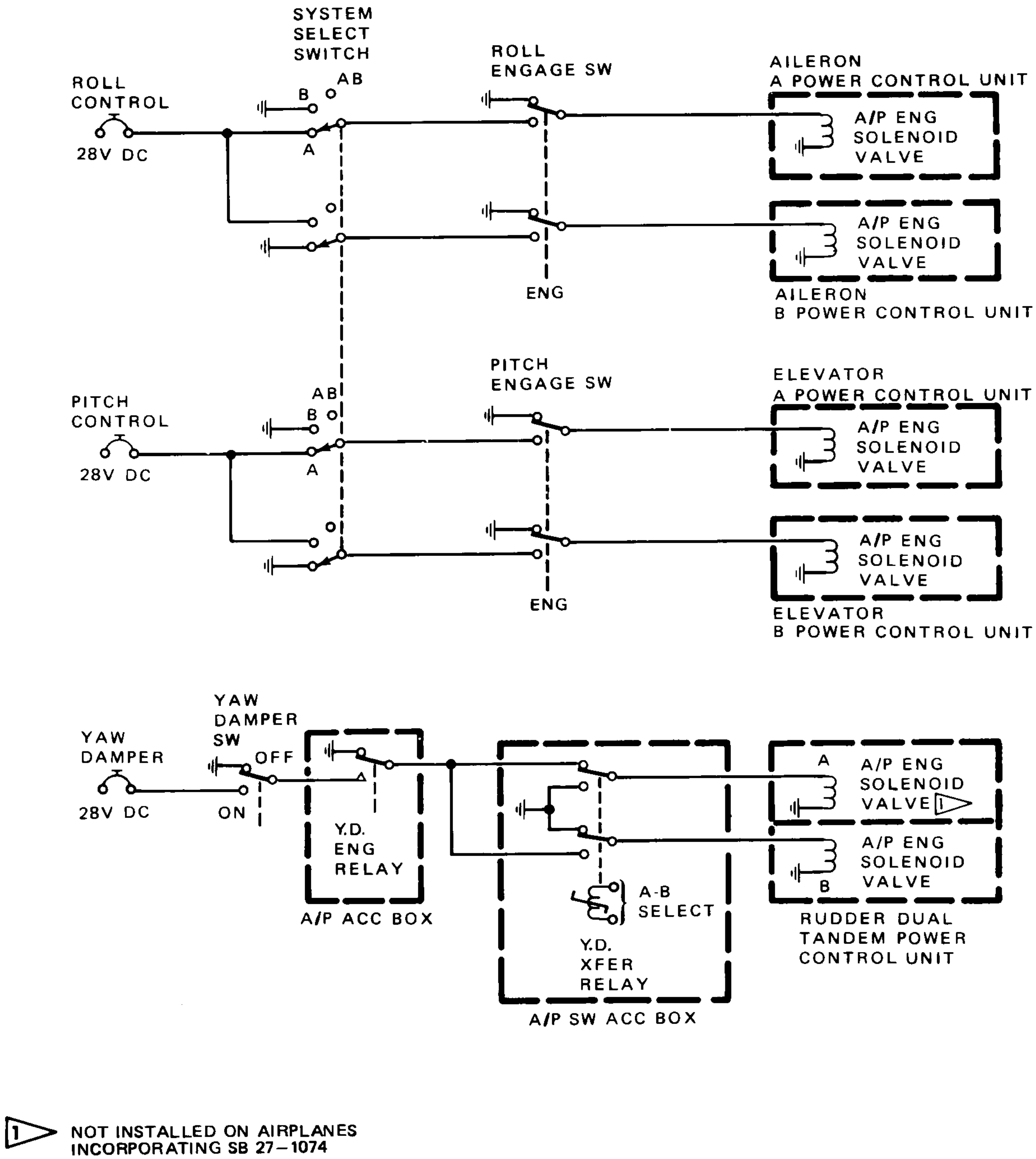
(c) The pitch mode select switch is a three-position rotary switch with a center OFF position. The switch is solenoid held in the altitude hold (ALT HOLD) position, so that it may be automatically released upon glide slope engagement. The turbulence (TURB) position is selected by rotating the switch counterclockwise.

(d) The heading off-heading select switch is a three-position toggle switch, spring loaded to the center off (heading hold) position and solenoid held in the HDG SEL or HDG OFF position. The switch will automatically return to the center position from the HDG SEL position if a lateral radio beam (VOR or localizer) is engaged or if the CWS is out of detent. The switch will hold in the HDG OFF position only when the navigation mode select switch is in the MAN position. It will automatically return to off from either position when a selected VOR radial is captured.

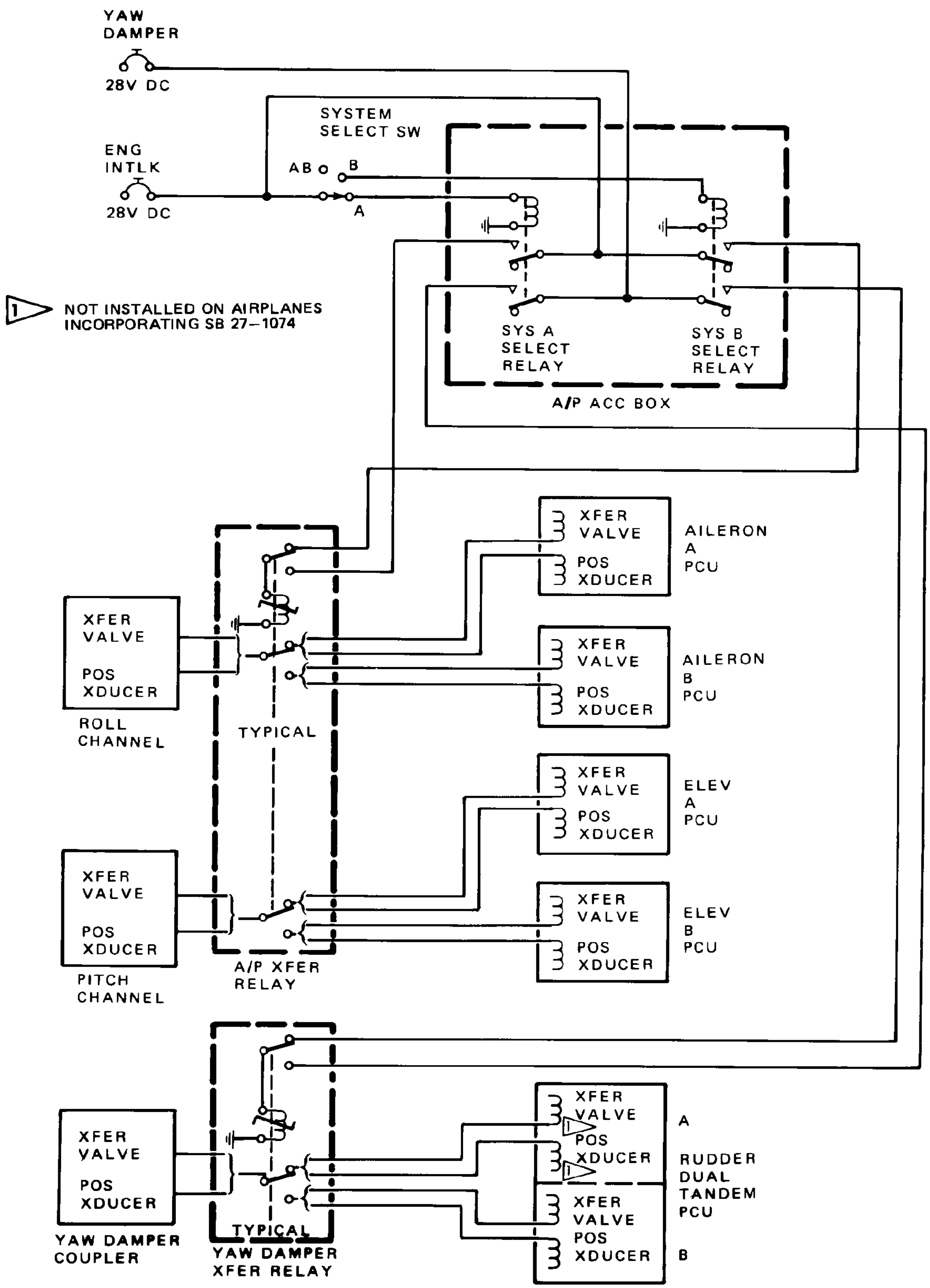
(e) The system select switch is a three-position rotary switch with A, B, and AB position; however, the AB position is inoperative and the A position cannot be used on airplanes with B system yaw damper only (Fig. 2, Fig. 3). The switch energizes a system A or system B select relay which in turn energize the yaw damper system A-B transfer relay and the autopilot A-B transfer relay. The transfer relays then connect the transfer and solenoid valves and position transducers on the rudder, aileron and elevator power control units, to be used by the autopilot and yaw damper. One set of valves is fed by the A hydraulic system and the other by the B hydraulic system. On airplanes with B position only, the autopilot and yaw damper signals are connected directly to rudder, aileron, and elevator control units powered by the B hydraulic system. The autopilot and yaw damper systems will disengage any time the system select switch is moved on airplanes with A, B switching and only the autopilot will disengage on airplanes with B position only.

(4) The approach progress displays are located on the captain's and first officer's instrument panels. The displays are annunciators utilizing colored lights to indicate when certain switching functions have taken place within the autopilot.

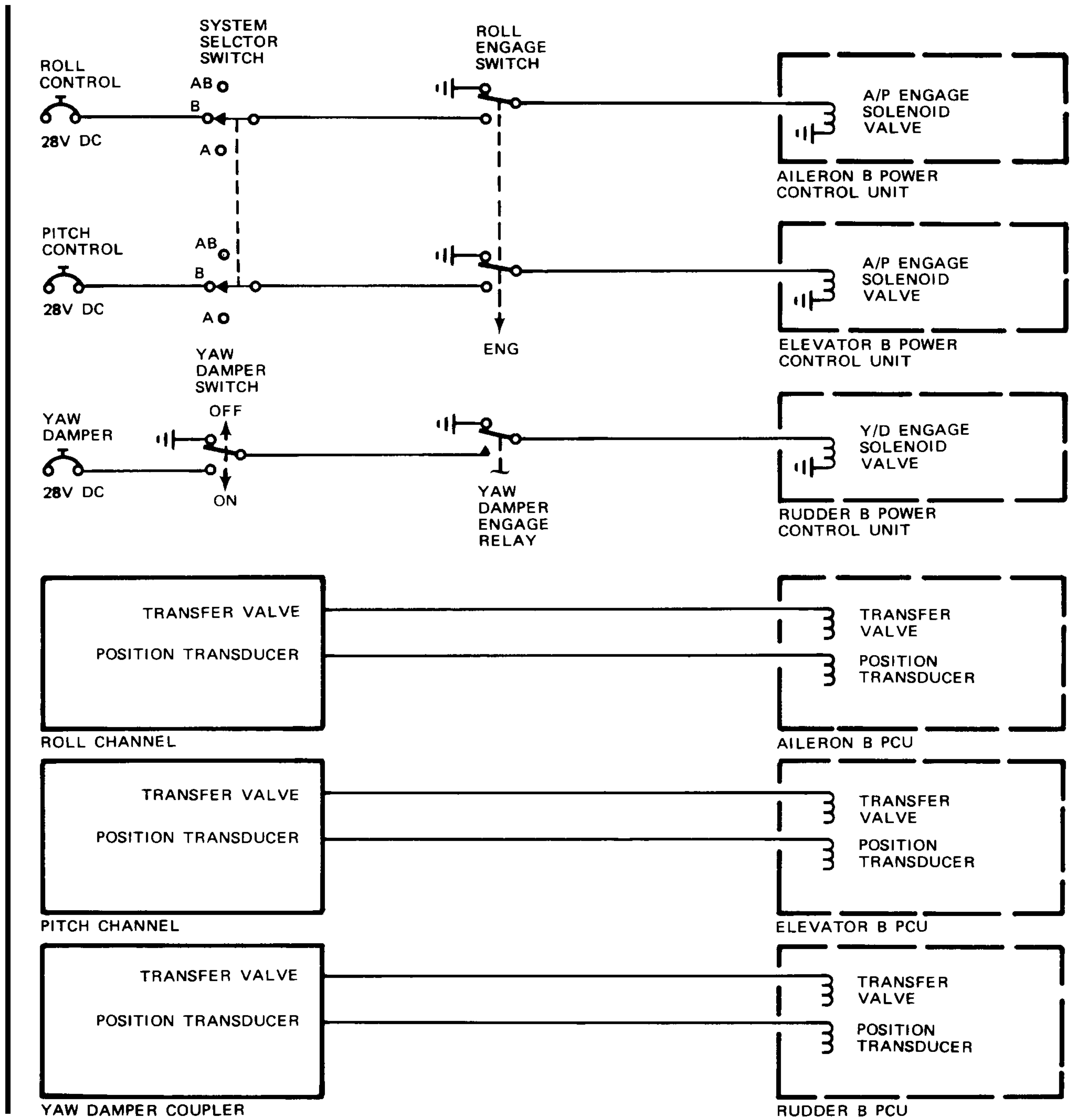
(5) The disengage and warning light reset switches are located on the outboard horn of each control wheel. The normally-closed contacts of the switches are in the engage interlock circuits, and if either switch is actuated, the autopilot will disengage. The normally-open contacts of the switches are used to reset the disengage warning light circuit. If the autopilot is disengaged using the control wheel switch, the warning lights will be reset at the same time, therefore; the lights will not flash.



Autopilot and Yaw Damper System A-B Select Circuits
Figure 2 (Sheet 1)



Autopilot and Yaw Damper System A-B Select Circuits
Figure 2 (Sheet 2)



Autopilot and Yaw Damper Circuit
Figure 3

(6) The disengage warning lights are located on the captain's and first officer's instrument panels. The lights are covered with a red cap. The light assembly incorporates a light reset push-to-test switch. When the light is pushed in, the warning light circuit is reset and the lights flash. The lights will extinguish when the light is released. If either light is held in, both lights should flash until the light is released. The lights will illuminate steadily if any one of the autopilot or yaw damper system self-test switches are in any position other than OFF.

(7) The stabilizer out of trim warning light is an amber light located on the center instrument panel (P2). The light is controlled by the stabilizer trim monitor in the pitch control channel and will illuminate whenever an out of trim condition exists between the stabilizer and elevator for more than approximately 12 seconds.

D. Stabilizer Trim Servo

(1) The stabilizer trim servo is located on the stabilizer jackscrew gearbox assembly in the tail cone. The trim servo consists of a nose-up relay, a nose-down relay, a speed change relay, a two-speed three-phase ac motor, a gearbox assembly with an electromagnetic clutch and an output shaft. Stabilizer trim control signals (trim up or trim down) are generated in the pitch control channel and are sent to the appropriate relay. Actuation of a relay supplies 115-volt ac, three-phase power to drive the servomotor in the desired direction. An additional set of contacts of the relays complete 28 volts dc from the pitch channel to the electromagnetic clutch solenoid.

(2) Trim servomotor power is controlled by the autopilot stabilizer trim circuit in the pitch channel. Power to the clutch solenoid is supplied from the pitch control channel through the autopilot trim cutout switch on the control stand. When the autopilot trim cutout switch is placed in the CUTOUT position, power is removed from the motor and clutch solenoid. The stabilizer trim limit switches also remove power from the clutch solenoid. Protection against a hot short (simultaneous trim-up and trim-down command) is provided in the pitch channel.

E. Position and Trim Sensors

(1) The position and trim sensors consist of the linear position transducers, the automatic stabilizer trim potentiometer and the elevator neutral shift sensor. The linear position transducers furnish elevator and aileron position feedback to the pitch and roll control channels. The automatic stabilizer trim potentiometer and the elevator neutral shift sensor furnish signals to the stabilizer trim circuit in the pitch control channel.

(2) There are four identical linear position transducers, one integrally contained in each elevator and aileron power control unit. The transducers are variable reluctance transformers which are excited by 26 volts ac. The magnitude of the output signal varies directly with the length of the input stroke and the phase of the signal changes with the direction of the stroke from the center null position.

(3) The automatic stabilizer trim potentiometer provides variable control of the threshold of the automatic pitch trim. The potentiometer is mounted on the elevator and rudder feel computer unit and is driven by the stabilizer. The trim potentiometer consists of a potentiometer, an input isolation transformer, two fixed resistors, and a fixed calibration resistor. The input is fed from the pitch control channel with signals from the elevator position transducer and the elevator neutral shift sensor.

(4) The elevator neutral shift sensor is a three wire synchro using 26-volt ac excitation. The sensor is mounted next to and is driven by the elevator feel and centering unit to determine the neutral shift of the elevator as a function of stabilizer position or mach trim.

F. Autopilot Transfer and Solenoid Valves

(1) One autopilot solenoid valve is mounted on each elevator and aileron control unit. They are solenoid-actuated valves which are energized when the autopilot axes are engaged; they complete hydraulic power to the transfer valves in the power control units.

(2) A transfer valve is also mounted on each elevator and aileron power control unit. They convert the autopilot electrical signals into hydraulic flow. The unit consists of a torque motor which moves either a flapper valve or a jet pipe assembly which regulates the hydraulic flow to the power control unit output piston.

G. Autopilot Accessory Boxes

(1) On airplanes with A and B system selection on the autopilot control panel, plus airplanes incorporating SB 27-1074, the accessory boxes consist of the autopilot accessory box and the switching accessory box. The two boxes are mounted side by side on the E1-3 shelf of the equipment racks in the electrical and electronic compartment. On airplanes with B mode only on the control panel, less airplanes incorporating SB 27-1074, the switching accessory box is not installed.

(2) The autopilot accessory box contains the necessary components to interconnect the autopilot and yaw damper system. It contains the yaw damper engage relay, the system A and system B select relays, the glide slope engage relay, (if installed), the autopilot warning lights flashers, time delay gate module and the circuit interrupters for the yaw damper, auto pilot engage and mode switching circuits. The circuit interrupters ensure the engage and NAV mode holding coils drop out whenever the system select switch or NAV mode select switch is operated, thereby allowing the interlock circuits to be satisfied before the coils can be energized again. The system A and system B select relays control the system A-B transfer relays.

(3) The switching accessory box contains a yaw damper system A-B transfer relay, an autopilot system A-B transfer relay and loading resistors for the navigation and ILS receivers. The relays switch control circuits between the A and B elevator, aileron and rudder power control units whenever A or B is selected on the autopilot control panel. On airplanes that do not have the switching accessory box installed the loading resistors for the navigation and ILS receivers are located in the autopilot accessory box.

H. Force Transducers

(1) The force transducers supply control wheel steering signals to the pitch and roll control channels which are proportional to the amount of force applied to the control wheels by the pilots when manually flying the airplane with the autopilot channels engaged.

(2) There are two force transducers in the pitch axis and one in the roll axis. The two pitch force transducers are mounted in the forward control quadrants below the captain's and first officer's control columns. The roll force transducer is mounted between the control shaft and aileron drum below the captain's control column. Each force transducer assembly consists of two independent electrically isolated E pickoff type transducers.

I. Force Limiter

(1) The force limiter is splined to the aileron drum at the bottom of the captain's control column. The force limiter consists of an electromechanical clutch, switch and a spring-loaded arm and cam assembly. The electromechanical clutch and switch are activated when the roll axis is engaged. The force limiter mechanically limits the control wheel angle to 25 degrees by progressive increase of force within the system to cam out the autopilot control at the power control unit. This limits the rate of roll which the autopilot can command; thereby, ensuring against autopilot hardover maneuvers. The switch ensures the force limiter clutch is engaged. If the clutch disengages, the switch opens the aileron engage switch holding coil circuit.

(2) The captain or first officer may override the force limiter by applying sufficient force on the control wheel to drive through the limiting mechanism. The autopilot continues to drive the power unit throughout the control wheel steering regime after the control wheel is returned to less than 25 degrees. Manual control is used if the control wheel is turned greater than 25 degrees.

(3) The force limiter is deactivated when the autopilot is disengaged, regardless of control wheel position.

4. Yaw Damper Operation (All except coupler P/N 4084042)

NOTE: For aircraft with coupler P/N 4084042 installed (incorporating SB 27A1206), see 22-12-01.

A. General

(1) The yaw damper provides damping of the airplane's yaw axis movement by shaping, amplifying, and coupling rate gyro and yaw damper actuator position signals to control the rudder through the hydraulic actuator. Yaw damper operation is confined to synchronization mode and engaged mode.

(2) The description of yaw damper operation is based upon the schematic (Fig. 3, Fig. 4). Summing points on this schematic are coded with reference numbers to aid in the discussion.

B. Operating Modes

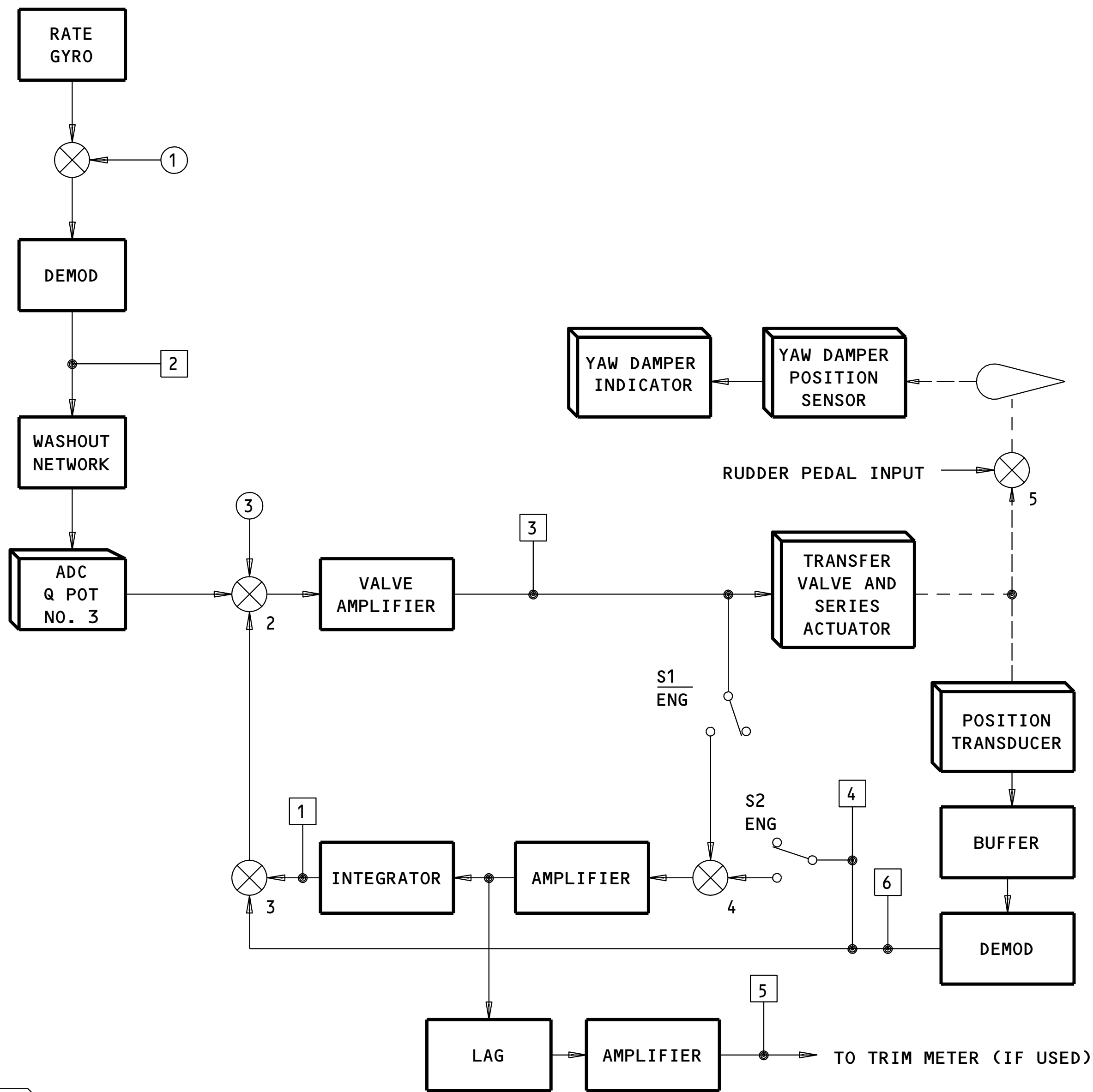
(1) Synchronization Mode

(a) Prior to yaw axis engagement, the yaw damper coupler is in the synchronization mode. The purpose of synchronization is to provide a null output to the electrohydraulic transfer valve to prevent yaw damper coupler engage transients.

(b) On yaw damper coupler part No. 2588880, synchronization is achieved by routing the valve amplifier output, past disengaged switch YS-1, to its input through the electronic integrator and summing point 2. By this action, the valve amplifier output is reduced to a null as the integrator output increases and cancels the yaw rate gyro signal at summing point 2.

(c) On yaw damper coupler part No. 4030952, synchronization is accomplished by routing the demodulated, parameter-controlled signal from summing point 2 through switch S1, summing point 4 and integrator, and back to summing points 3 and 2. Output from summing point 2 results in temporary output from the valve amplifier. The valve amplifier output is reduced to a null as the integrator output increases and cancels the signal at summing point 2.

(d) Prior to yaw damper engagement, the yaw damper actuator remains centered which nulls the position feedback to the yaw damper coupler.



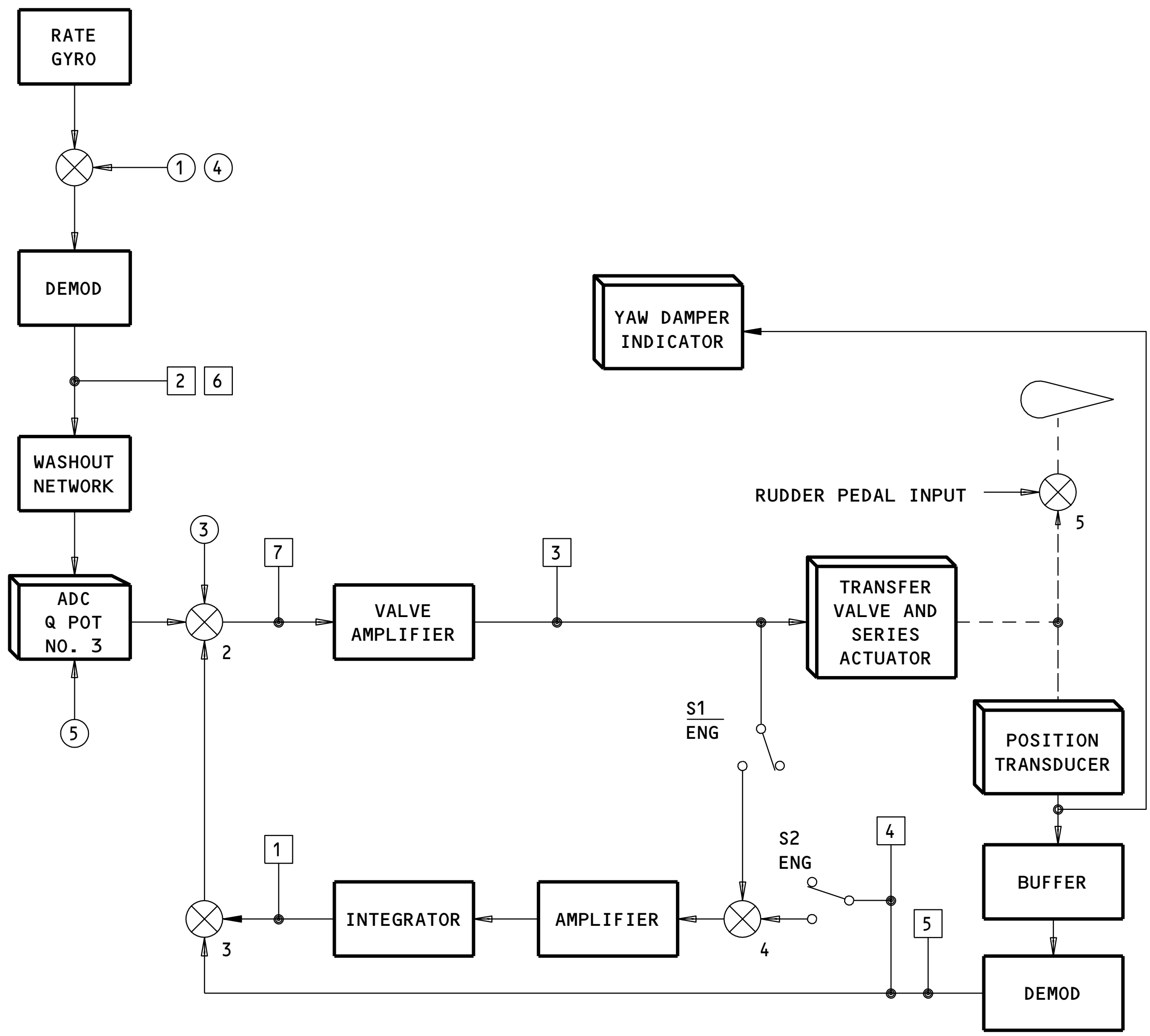
NOTES:

1 DENOTES EQUIPMENT EXTERNAL TO THE YAW DAMPER COUPLER

2 DENOTES BITE INPUT STIMULI POINTS

3 DENOTES BITE OUTPUT READ POINTS

Yaw Damper Block Diagram
Figure 4 (Sheet 1)



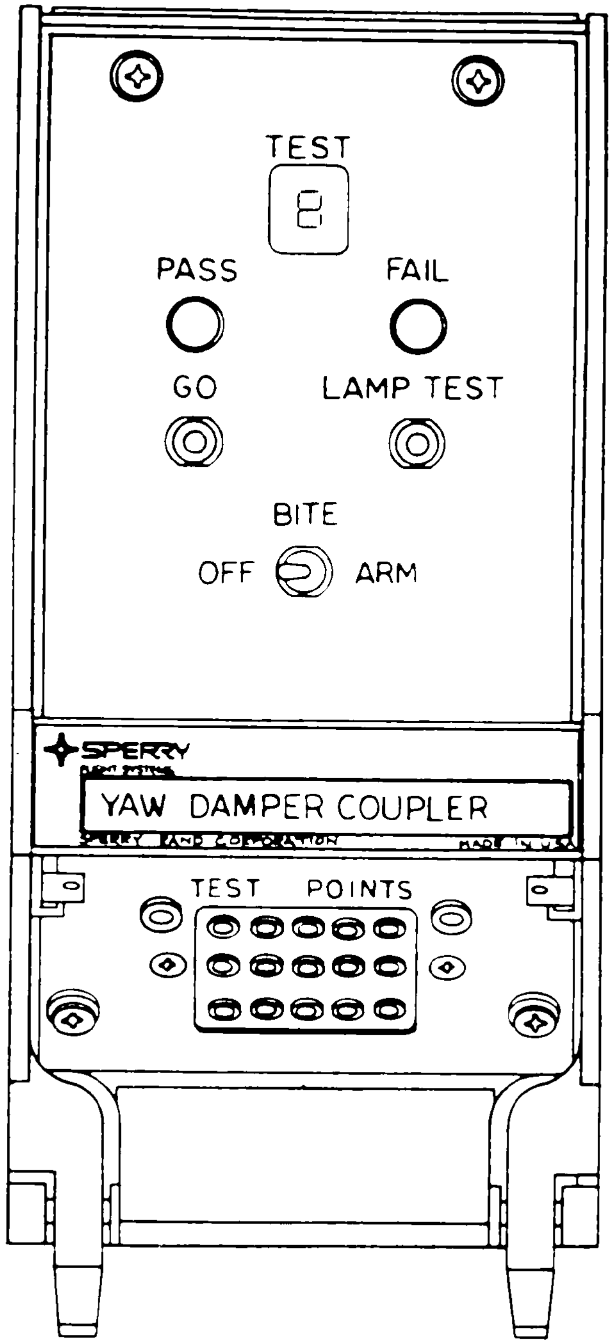
NOTES:

1 DENOTES EQUIPMENT EXTERNAL TO THE YAW DAMPER COUPLER

2 DENOTES BITE INPUT STIMULI POINTS

3 DENOTES BITE OUTPUT READ POINTS

Yaw Damper Block Diagram
Figure 4 (Sheet 2)



Yaw Damper Pictorial Diagram
Figure 5

(e) The yaw damper engage switch will remain in the ON position by holding the switch to ON for at least 0.1 second. This allows voltage through the holding coil and the logic circuit timer to complete a path to ground as provided by the logic OR gate circuit.

(f) The yaw damper engage relay is energized by a ground through the logic AND gate after going through the 2-second timer. Two sets of contacts apply power to the interlock circuit and to the respective rudder power system. An additional set of contacts open which extinguishes the warning light.

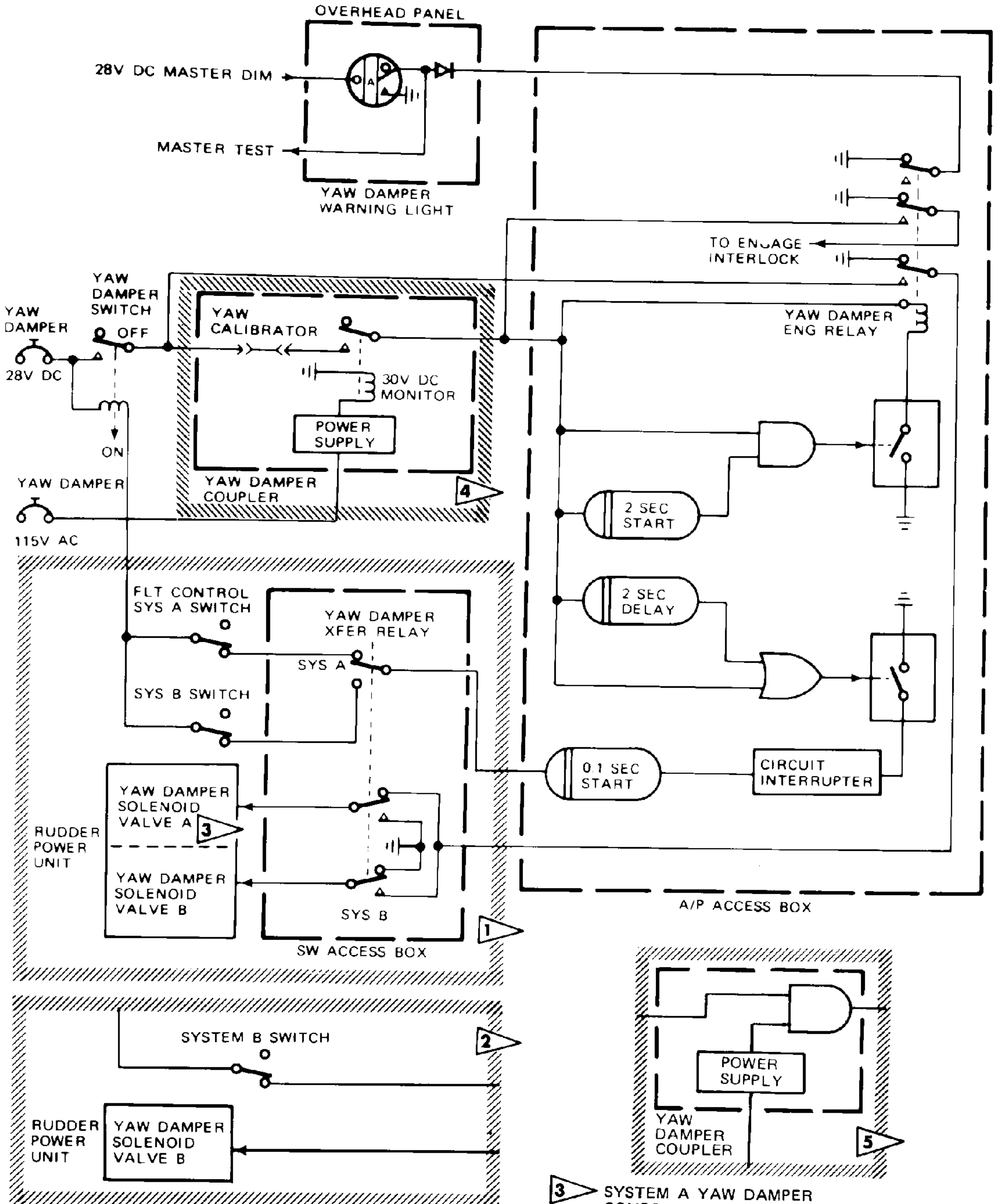
(g) Switching between hydraulic power systems causes the yaw damper system to disengage, however on later airplanes this switching is removed. Transferring between 115-volt ac generators will cause the monitor to be momentarily invalid. However, with a 2-second decay timer in the engage switch circuit, the yaw damper engage relay remains energized.

(2) Engaged Mode

(a) When the yaw damper coupler is engaged, processed rate gyro signals at summing point 2 produce a valve amplifier output. This output is applied to the series yaw damper actuator transfer valve to command movement. Movement of the yaw damper series actuator results in rudder movement and an output from the position transducer.

(b) Position transducer output is demodulated and applied to summing point 2 to cancel the processed rate gyro signal when the corresponding change in rudder position is appropriate for the rate of yaw attitude change. Position transducer output is also applied to control the integrator. The integrator lag is large enough so that the integrator output, which is applied to summing point 2, does not attenuate yaw damper action in the dutch roll frequency spectrum.

(c) Rate gyro signals are processed in the rate gyro filter. The rate gyro filter consists of a demodulator, amplifier, and washout network. Filtering accomplished in these circuits eliminates steady-state signals and unwanted high frequency signals. Consequently, the yaw damper coupler does not oppose normal turn maneuvers and does not respond to airplane vibration and bending. Frequency response is tailored to minimize the airplane dutch roll. The rate gyro filtered output is routed to summing point 2 through Q-potentiometer in the ADC. Rate gyro signal gain is varied by this potentiometer to compensate for changes in airplane dynamics as a function of airspeed.



1. AIRPLANES WITH SWITCHING ACCESSORY BOX INSTALLED (SYSTEM A AND B)
2. AIRPLANES WITHOUT SWITCHING ACCESSORY BOX INSTALLED. (SYSTEM B ONLY)
3. SYSTEM A YAW DAMPER COMPONENTS NOT INSTALLED ON RUDDER POWER UNIT ON AIRPLANES INCORPORATING SB 27-1074. (SYSTEM B ONLY)
4. YAW DAMPER COUPLER PART NO. 2588880
5. YAW DAMPER COUPLER PART NO. 4030952

Yaw Damper Engage Interlocks
Figure 6

C. Yaw Damper Test

(1) The yaw damper indicator, located on the center instrument panel, is a full time operating meter. Whenever the yaw damper system is engaged, the yaw damper indicator will reflect the movements of the rudder. Actuation of the yaw damper test switch to either the left or right applies a voltage to the yaw damper coupler (Fig. 7).

(2) Voltage to the coupler is applied to the yaw damper rate gyro torqueing coil which torques the rate gyro and simulates airplane movement. The coupler provides the properly related voltage to rudder power unit for rudder deflection consistent with direction selected. The yaw damper indicator responds to this action as a result of an output from the position transducer.

5. <u>Autopilot Roll Channel Operation</u>

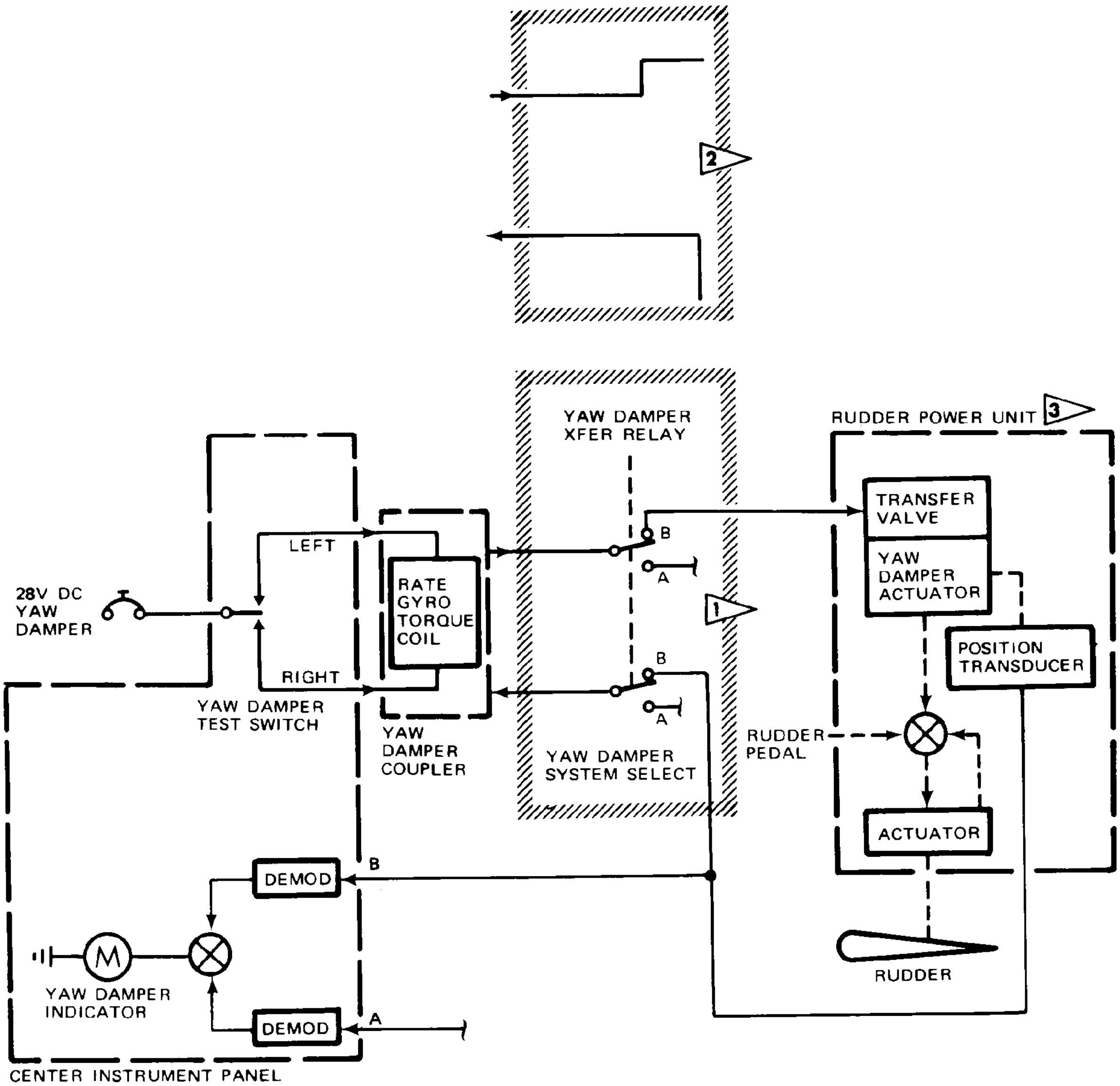
A. General

(1) The roll channel simplified block diagram (Fig. 6, Fig. 7, Fig. 8) presents the roll axis signal circuits. The various command input signals are summed at summing point 2, routed to the roll computer and combined with feedback signals at summing point 9. This resultant error signal is fed to the aileron power unit to command surface deflections. The system contains three main sections; the input section, the roll computer, and the output and feedback section. Except for control wheel steering (CWS), signal inputs from the input section are summed before going to the bank limiter. Individual input selection includes heading hold, attitude hold, heading select, VOR/LOC course selection. The output from the CWS force transducer supplies a signal to the rate limiter proportional to pilot force on the control wheel. The vertical gyro senses the change of roll attitude for an input to the computer. The roll computer section provides a composite roll error signal as an output for summing with the feedback section signal. The composite signal at summing point 9 is fed to the aileron power unit.

B. Operating Modes

(1) Synchronization Mode

(a) Synchronization of the roll channel is achieved prior to roll axis engagement whenever power is applied to the autopilot and auxiliary systems. Roll synchronization consists of the following: heading synchronization, roll attitude synchronization and valve-amplifier synchronization (Fig. 9).



1 EFFECTIVE ON AIRPLANES WITH A AND B SYSTEM SELECTION ON AUTOPILOT CONTROL PANEL PLUS AIRPLANES INCORPORATING SB 27-1074

2 EFFECTIVE ON AIRPLANES WITH B MODE ONLY ON AUTOPILOT CONTROL PANEL LESS AIRPLANES INCORPORATING SB 27-1074

3 SYSTEM A RUDDER POWER UNIT YAW DAMPER COMPONENTS NOT INSTALLED ON AIRPLANES INCORPORATING SB 27-1074

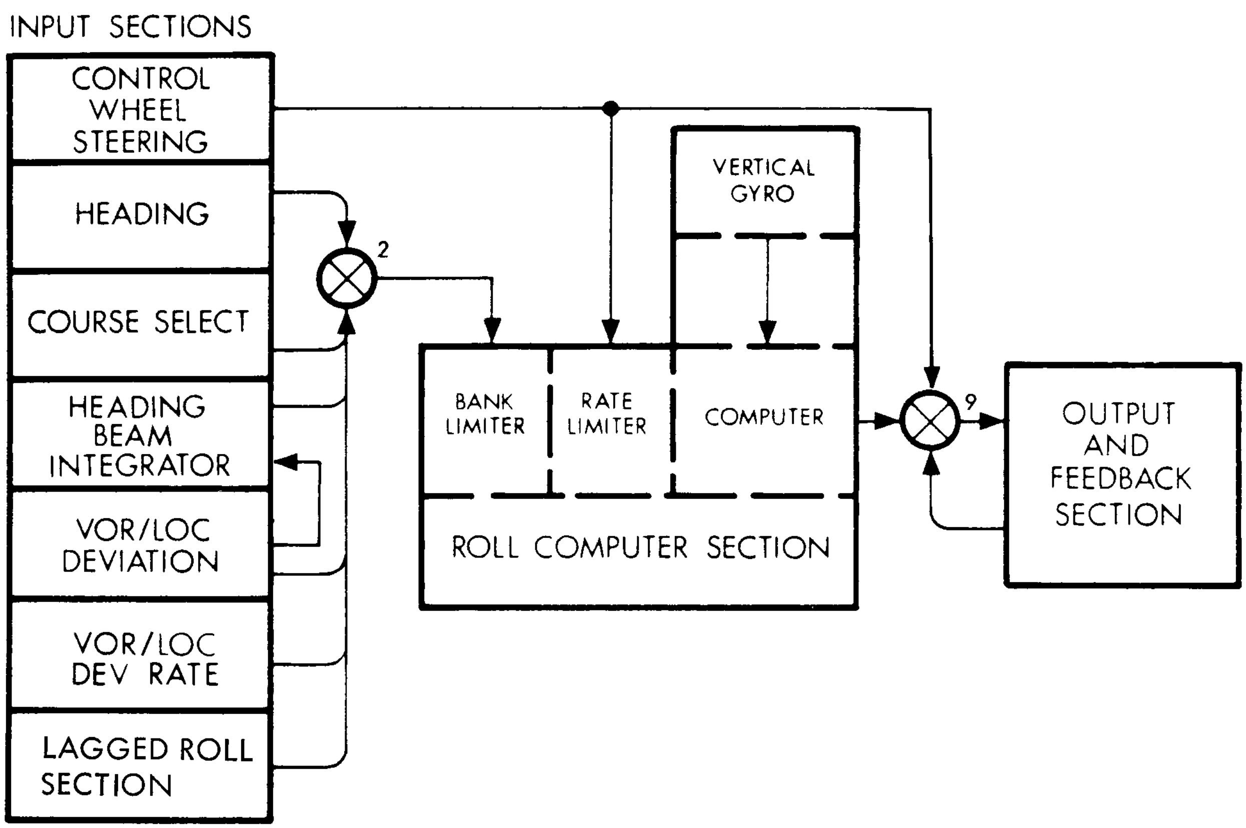
Yaw Damper Test Circuit
Figure 7

(b) During heading synchronization, an interlock signal is applied to the heading computer motor clamp circuit, which allows a power amplifier circuit to drive the heading computer. The heading computer will be driven in the direction which causes it to be synchronized with the airplane heading as sensed by the heading compass (directional gyro) system. The output of the heading computer control transformer (heading CT) is applied to summing point 4, amplified by the motor amplifier, and drives the heading servo in a direction to reduce the output of the heading CT to a null. This action produces heading synchronization because the heading CT stator is coupled to a heading synchro in the directional gyro. At this time energized switch RS-6 prevents heading CT output from being applied elsewhere in the roll control channel.

(c) During roll attitude synchronization, airplane bank angle information from the vertical gyro is stored in the roll computer. The stored information is used to provide a reference for the roll attitude hold function in the synchronization mode. An interlock signal applied to the motor clamp circuit allows the power amplifier circuit to drive the roll computer. When the roll computer is driven to the point where the roll computer control transformer (roll CT) output is at a null, then the existing bank angle is stored in the roll CT and the roll computer resolver. The roll CT output is maintained at a null during synchronization as follows:

1) The roll CT will produce an output signal if the rotor position fails to correspond to the airplane bank angle sensed by the vertical gyro. This output signal is developed as a result of the roll CT being coupled back-to-back with the roll synchro in the vertical gyro.

2) The roll CT output signal is fed to the motor amplifier past de-energized switch RS-1 and to summing point 8. The motor amplifier and the power amplifier amplify the signal so that it drives the roll computer motor-tachometer generator.



Roll Channel Simplified Block Diagram
Figure 8

3) The motor-tachometer generator drives the roll CT in the direction which reduces the CT output. The roll computer will continue to drive until the roll CT output reaches a null.
4) Because the resolver (RS) is ganged to a common shaft with the roll CT, it stores bank angle information simultaneously with the roll CT. The resolver has two windings, the outputs from which are proportional to the sine and cosine of bank angle. The output from the cosine winding is applied to a lift compensation circuit which derives a signal proportional to the versine (1-cosine) of bank angle. This versine signal is routed to the pitch axis where it is used to compensate for decreased vertical lift in banked turns. During synchronization, the resolver (RS) signal is inhibited
5) from reaching summing point 7 by solid-state switch RS-2 which is energized (closed). All other signals are removed from the input of the bank rate limiter during synchronization by action of other solid-state switches.
6) The output from the tachometer portion of the motor-tachometer generator is fed to summing point 8 to provide damping for the roll computer servo loop. Part of this tachometer signal is shunted to ground during synchronization by switches RS-3 and RS-4. The resulting decrease in tachometer feedback allows the roll computer servo loop to follow up error signals more rapidly during synchronization.

(d) Valve amplifier synchronization consists of establishing a null at the output of the valve amplifier to eliminate roll axis engagement transients. Prior to roll axis engagement, a modulated output of the valve amplifier is fed to the electronic integrator. This amplifier output is applied to the electronic integrator through de-energized switch RS-9. During synchronization, the other integrator input is grounded by switch RS-10. Therefore, any output produced by the integrator during synchronization is a result of input signals from the valve amplifier. The integrator output will be of the phase required to cancel the inputs to summing points 9 and 10, which are causing valve amplifier output. The net result of the closed synchronization servo loop is to reduce the valve amplifier output to a null.

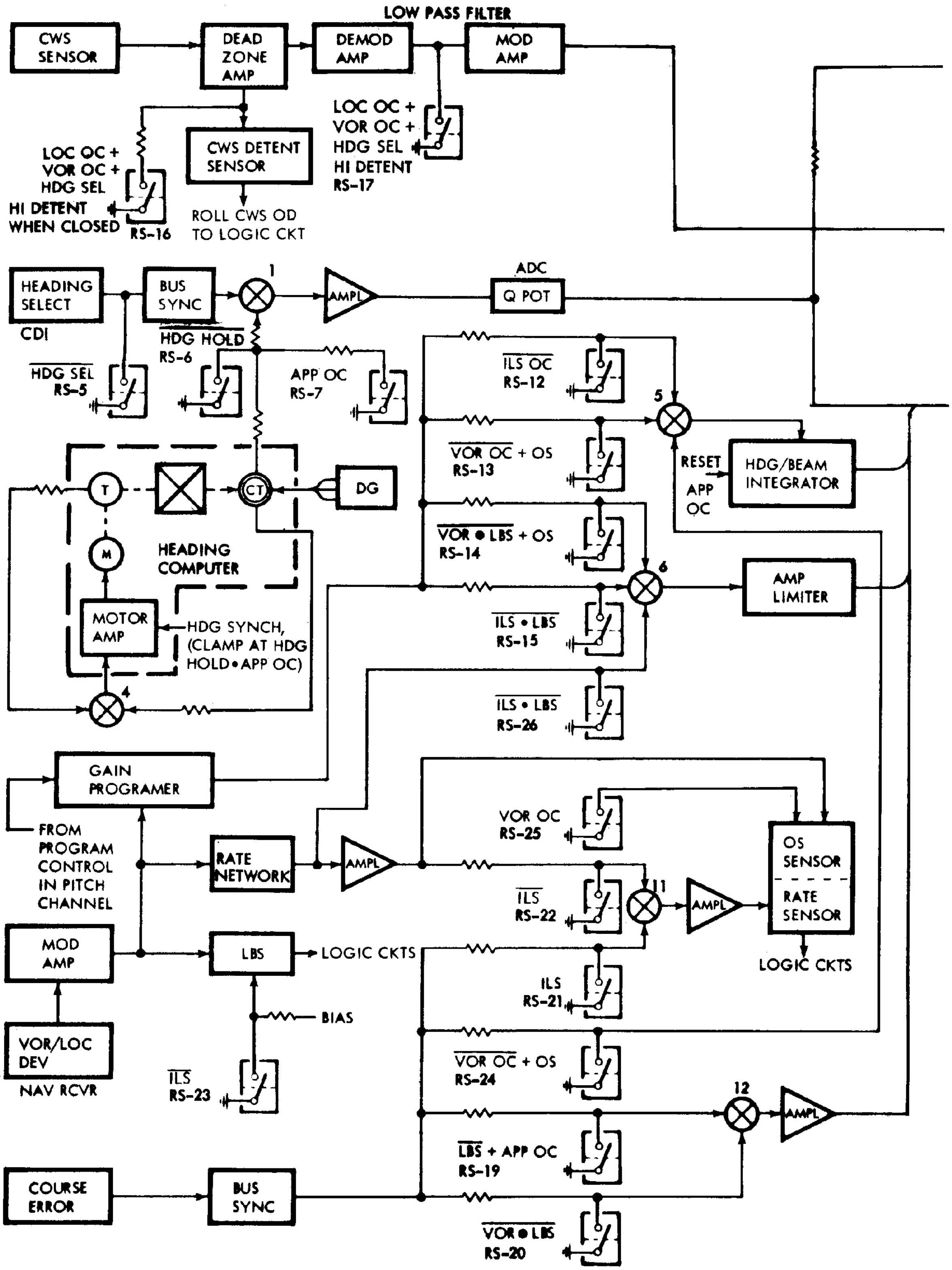
(2) Manual Mode

(a) Roll Engagement

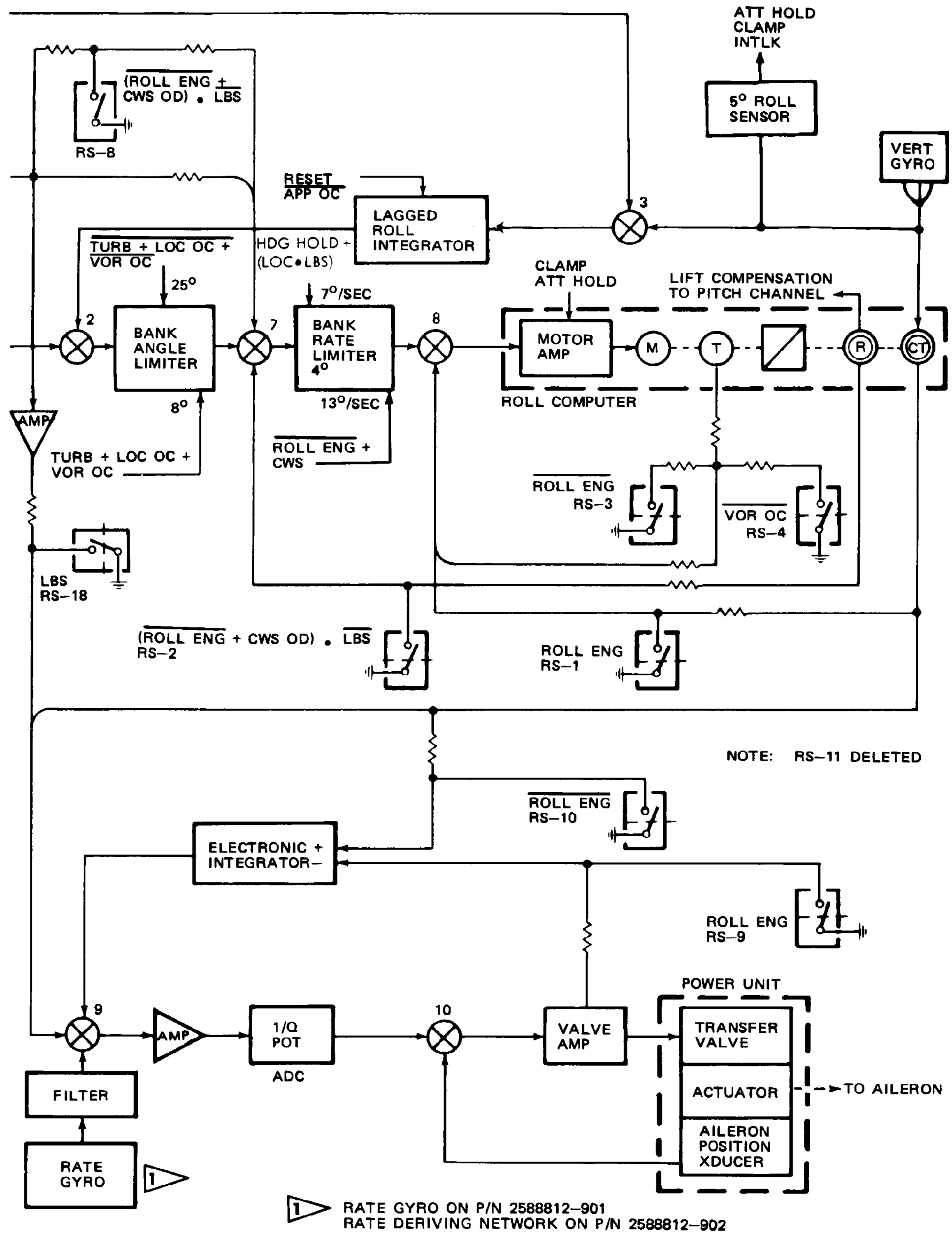
1) The roll axis will engage into a roll attitude hold or heading hold condition when the roll engage interlocks are satisfied (Fig. 10). These interlocks are as follows: flight control A or B switches on, for airplanes with A and B system selection on autopilot control panel, flight control B switch on, for airplanes with B system selection only on autopilot control panel, derived rate 115 vac for roll channel part No. 2588812-902 on later airplanes plus airplanes incorporating SB 22-1018, standby power switch is positioned to AUTO and dc is good from battery bus, 28-volt flag voltage from the air data computer (ADC) is good, the 30-volt dc roll channel power supply is good, the directional gyro and vertical gyro are good, and roll control wheel steering is not out of detent, or the AIL engage switch is engaged; and voltage through the roll calibrator is present.

2) The 28-volt dc engage interlock power is routed through the autopilot disengage switches to the pitch engage interlock circuit and to the system select switch (when installed) in the control panel (Fig. 10). When the system select switch (when installed) is in either position A or B (B position on airplanes that do not have system select switch), the voltage is applied to one side of an AND gate circuit. Also, 28-volt dc roll control power is applied to one side of an AND gate in the control panel.

3) When the AIL engage switch is placed in the engage position and all other inputs to the roll channel (Fig. 10) are good, the roll channel output is an interlocks logic input to an AND gate in the control panel. This causes the AND gate logic function to be HIGH or 1. With both conditions 1 to the second AND gate, an output energizes the roll engage holding coil. The AIL engage switch is locked mechanically when the holding coil is not energized. When power is applied to the roll engage holding coil, the mechanical lock is removed. By engaging the AIL engage switch, the aileron force limiter switch contacts and the roll engage holding coil contacts are switched to a ground path through the circuit interrupter. The aileron force limiter switch contacts operate when the AIL engage switch is engaged.



Roll Control Channel Block Diagram
Figure 9 (Sheet 1)



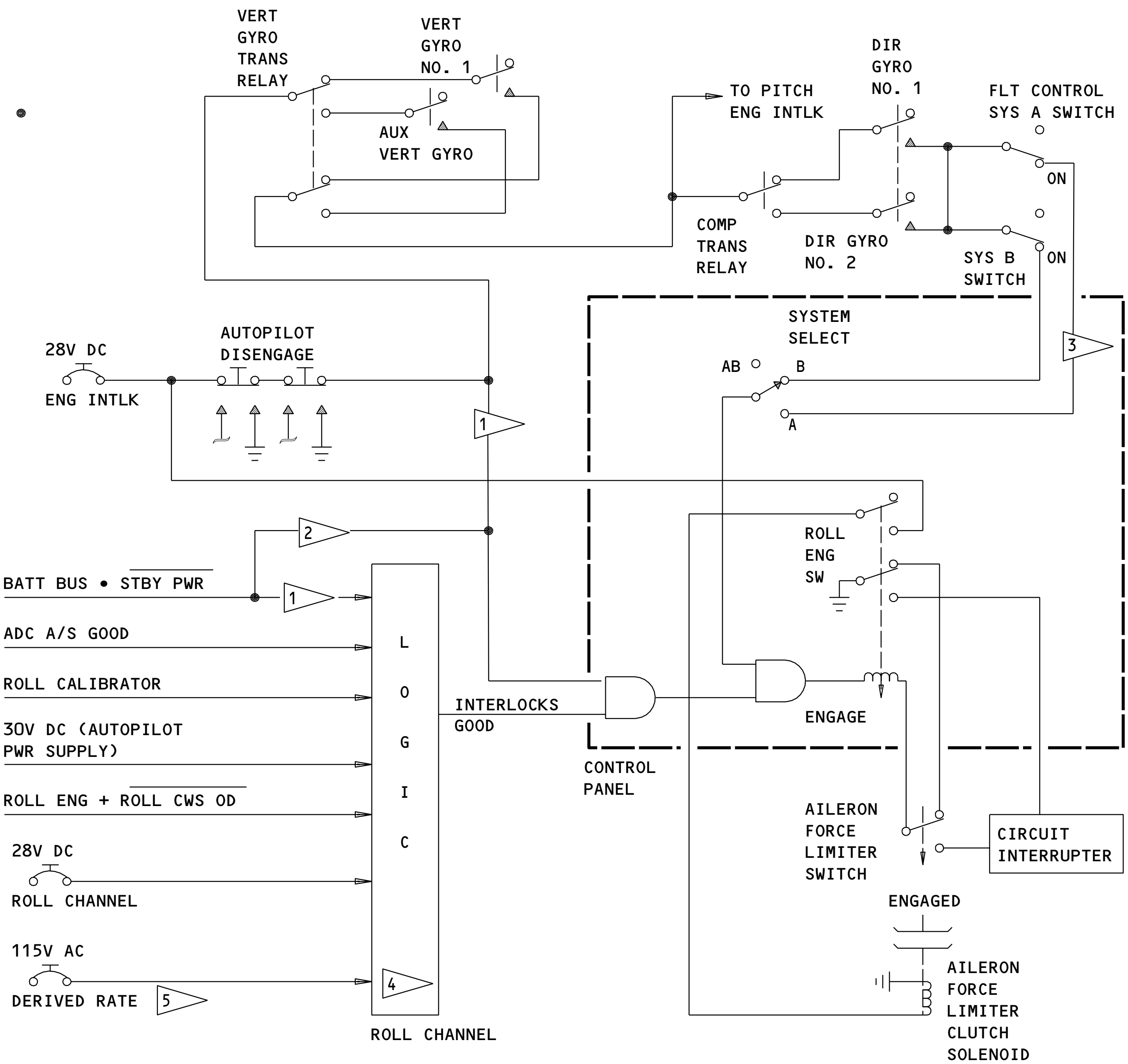
Roll Control Channel Block Diagram
Figure 9 (Sheet 2)

4) Failure of any interlock signal will cause the AIL engage switch to disengage. Also, interruption of the interlock power sources, or instrument switching (as applicable) will cause the AIL engage switch to disengage. However, CWS inputs do not cause disengagement.

(b) Roll Attitude Hold

1) The roll channel operates in the roll attitude hold mode whenever the autopilot roll axis is engaged, and the following switching is selected the heading select switch is in HDG OFF, nav mode select switch is in MAN, and the roll CWS is in detent. Upon engagement of the roll attitude hold mode, the roll computer is clamped by a motor clamp circuit, and the bank angle existing at the time of mode engagement is maintained. The bank angle limit for this mode is 32 degrees.

2) When the roll computer is clamped, changes to the airplane bank angle are sensed by the vertical gyro which will produce an error signal in the roll computer CT. This error signal is fed to summing point 9 and to an integrator (Fig. 9). (There is no other input to the integrator at this time because the synchronization feedback signal is grounded by energized switch RS-9.) The effect of the integrator is to increase the low-frequency gain of the roll computer CT signal into summing point 9 to decrease system standoff errors. Roll computer CT error signals and integrated error signals are summed with roll rate gyro signals at summing point 9. The rate gyro signals provide rate damping of the autopilot roll axis.



1 AR LV-JMW THRU LV-JMZ
LV-JND AND LV-JNE
IC VT-EAG THRU VT-EAL
TZ CF-TAN AND CF-TAO
NZ ZK-NAC THRU ZK-NAM

2 ALL EXCEPT 1

3 REMOVED ON AR LV-LIU AND ON
IC VT-ECP AND ON
TZ C-GTAQ AND ON
NZ ZK-NAP AND ON

4 DERIVED RATE LOGIC ON ROLL CHANNEL PART NO. 2588812-902

5 ON AIRPLANES INCORPORATING SB 22-1018

Roll Engage Logic Circuit
Figure 10

3) The signal resulting from the summation at summing point 9 is amplified by a preamplifier in the valve amplifier circuit and fed to an air data computer (ADC) Q-potentiometer. The ADC Q-potentiometer changes the forward gain of the signal fed to summing point 10 as an inverse function of airspeed. Gain scheduling as a function of airspeed provides for uniform autopilot control over the operational range of the airplane. The output signal from the ADC Q-potentiometer is fed to summing point 10 where it serves as the aileron command signal. The aileron command signal is summed with a feedback signal from the position transducer in the selected (B or A when autopilot system switching is installed) aileron actuator. Therefore, there will be an output from summing point 10 whenever the aileron command is not cancelled by a position feedback signal of opposing phase. The output signal from summing point 10 is amplified and demodulated by the valve amplifier and the resulting dc signal is applied to the electrohydraulic transfer valve in the selected aileron actuator. The aileron actuator will drive the airplane ailerons until the actuator position is such that the feedback signal from the actuator position transducer cancels the aileron command at summing point 10. Aileron displacement will be in the direction which causes the airplane to roll back toward the bank angle at which the roll computer is clamped. As the reference bank angle is approached, the roll computer CT error signal diminishes and, correspondingly, the aileron displacement decreases. The net result is zero aileron displacement when the reference bank angle is restored. The preceding sequence of events occurs whenever the airplane deviates from the clamped bank angle reference with the result that a reference roll attitude is maintained. Roll attitude hold mode automatically disengages if any of the previously mentioned modes or conditions are initiated.

(3) Control Wheel Steering (Bank Angle Maneuvering)

(a) Roll CWS mode can be initiated at any time after the autopilot roll channel is engaged, regardless of roll attitude, by the application of control wheel force. A force transducer, located in the aileron control quadrant, supplies an output signal proportional to the force applied to the control wheel. The force, applied to the control wheel in excess of detent level, will cause the roll computer to command a rate of change of bank angle proportional to the applied force. When operating CWS, the resolver (RS) feedback is removed by action of switch RS-2 allowing the roll computer to function as an integrator. The roll computer then produces a roll rate command proportional to control wheel force above the detent level.

(b) The output signal from the CWS force transducer is applied to deadzone amplifier (Fig. 9). The function of the deadzone amplifier is to produce zero output (deadzone level) for applied forces ranging between approximately 0 and 4 pounds. Above this detent level, the output of the deadzone amplifier is proportional to the force applied to the control wheel. An output of the deadzone amplifier is applied to the CWS detent sensor in the deadzone amplifier. The function of the detent sensor is to detect control wheel force transducer output levels corresponding to those which are above the deadzone level and to activate interlock circuits accordingly. Switch RS-16 increases the threshold detection level of the detent sensor during the on-course phase of lateral radio beam operation and during the heading select mode to prevent nuisance disconnects of these modes. The other output of the deadzone amplifier is applied to a low-pass filter. The function of the low-pass filter is to filter and attenuate control system resonances. Switch RS-17 is paralleled to RS-16 and prevents CWS signals from affecting the autopilot when in on-course or heading select modes. The output of the low-pass filter is applied to summing point 7 and to an amplifier which provides an input to summing point 9.

(c) The signal at summing point 7 is applied to the bank rate limiter. The bank rate output drives the roll computer up to its maximum rate of 13 degrees per second through the motor amplifier. Bank angle command is limited to 32 degrees by the bank angle stops in the roll computer. When the roll computer is driven by the CWS signal, the roll computer CT develops an error signal. This error signal results in aileron displacement as described in roll attitude hold mode. The CWS signal which is amplified and fed directly to summing point 9 provides compensation for velocity error between actual and commanded bank angles. This signal path is blocked by RS-18 when the lateral beam sensor is pulled in. The CWS signal gain and the roll resolver feedback into summing point 7 are switched (RS-8 and RS-2) as a function of LBS dropout, thereby enabling those airplanes with a supervisory override option to use CWS inputs into the roll computer when the airplane is on course.

(4) Heading Select Mode

(a) The heading select mode of operation can be selected any time after roll axis engagement except during the CWS mode or after lateral radio beam engagement. The heading select mode is obtained by placing the control panel heading switch to HDG SEL. With the heading switch in the HDG SEL position, 28 volts dc power is applied to an AND gate in the control panel (Fig. 11). An additional power input (28 volts dc from the engage interlock circuit) and a logic input (lateral beam sensor (LBS) has not dropped out and control wheel steering is not out of detent) trigger the AND gate to hold the heading select switch engaged. The normally closed set of contacts of the heading switch open to remove power from a heading select NOT logic circuit and heading computer motor clamp circuit in the roll channel.

(b) Heading error signals from the heading select synchro in the course deviation indicator (CDI) are applied past de-energized switch RS-5 through a bus synchronizer to summing point 1 of Fig. 9. The bus synchronizer consists of a demodulator-modulator which performs a quadrature rejection function (rejection of extraneous 90-degree components of inphase or out-of-phase signals) and synchronizes the signal with the autopilot reference signal. Heading error signals at summing point 1 are amplified by an amplifier, fed through the heading Q-potentiometer, and then applied to summing point 2. The output signal from the heading Q-potentiometer does not drive the lagged roll integrator since the lagged roll integrator is in a reset condition. The heading error signal at summing point 2 is applied to summing point 7 through the bank angle limiter. The bank angle limiter provides a bank angle command limit of 25 degrees. The limited heading signal is summed with the output of the roll resolve (RS) at summing point 7 and applied to summing point 8 through bank rate limiter. The bank rate limiter limits the roll rate command to 4 degrees per second in this mode.

(c) The heading error signal at summing point 8 is summed with tachometer feedback signals and applied to the roll computer through the motor amplifier and the power amplifier. The amplified signal drives the roll computer in a direction to allow the roll resolver RS to cancel the heading error at summing point 7. When the roll computer is driven by the heading error signal, the roll CT develops an error signal. The roll CT error signal results in aileron displacement as described in roll attitude hold mode. The aileron displacement produces a banked turn in the direction required to reduce the heading error and roll CT signals to a null.

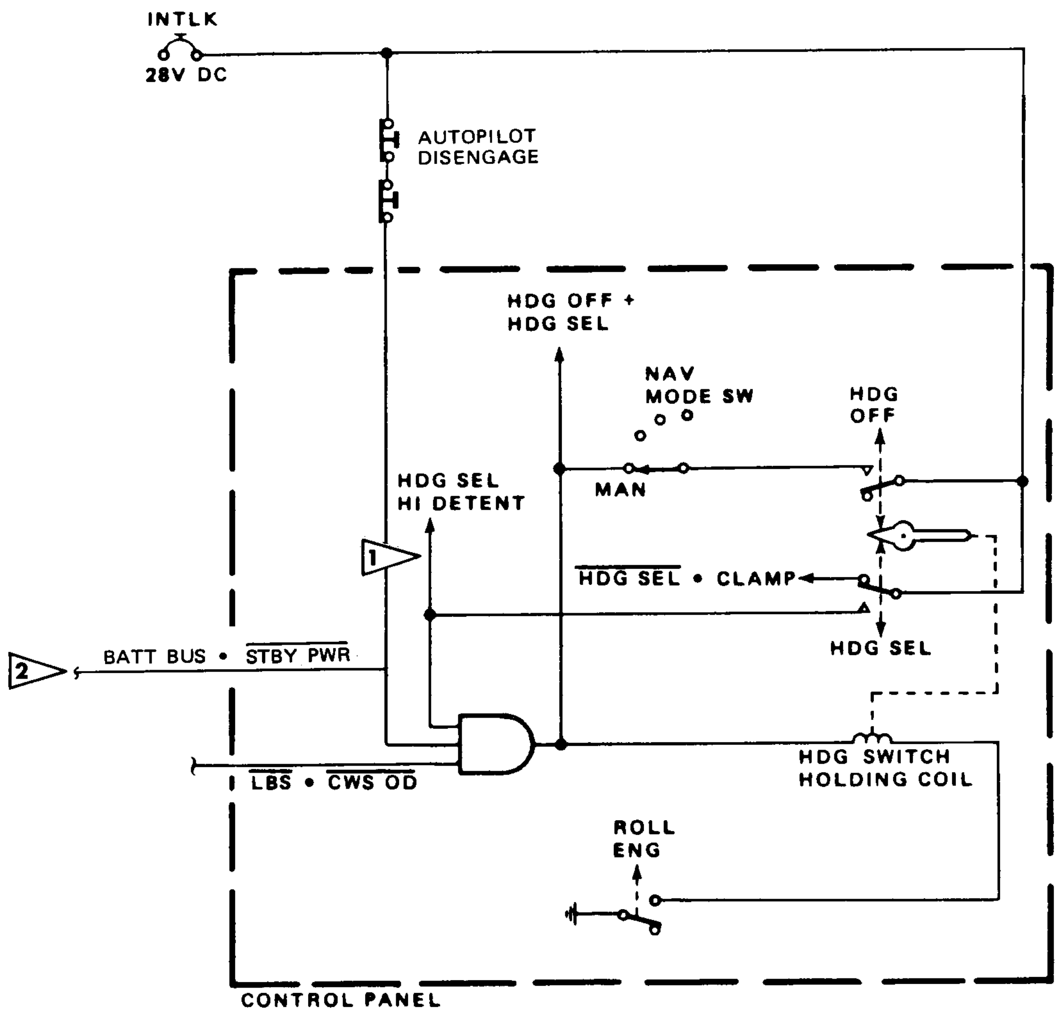
(5) Heading Hold and Heading Off Modes

(a) The heading hold mode is selected on the control panel whenever the heading switch is in the center position. Heading hold mode may be used during the intercept phase of lateral beam operation (VOR/LOC) but will be automatically disengaged at lateral radio beam capture. Heading hold will be in effect after engaging the roll system except during CWS out of detent, after lateral beam engagement or whenever HDG SEL or HDG OFF are selected. An additional requirement for heading hold mode engagement is that the bank angle sensed by the roll sensor must be less than 5 degrees. However, once the mode is engaged, CWS bank angles greater than 5 degrees will not cause mode disengagement. The heading switch will be in the center position (heading hold) whenever the heading logic circuit (Fig. 11) removes voltage from the heading switch holding coil thus allowing the spring-loaded switch to return to center from the HDG SEL position. The heading off mode can be used only when the mode select switch is in MAN; 28 volts dc is then applied directly to the heading switch holding coil.

(b) The heading hold mode is selected on the control panel whenever the heading switch is in the center position. Heading hold mode may be used during the intercept phase of lateral beam operation (VOR/LOC) but will be automatically disengaged at lateral radio beam capture. Heading hold will be in effect after engaging the roll system except during CWS out of detent, after lateral beam engagement or whenever HDG SEL or HDG OFF are selected. An additional requirement for heading hold mode engagement is that the bank angle sensed by the roll sensor must be less than 5 degrees. However, once the mode is engaged, CWS bank angles greater than 5 degrees will not cause mode disengagement. The heading switch will be in the center position (heading hold) whenever the heading logic circuit (Fig. 11) removes voltage from the heading switch holding coil thus allowing the spring-loaded switch to return to center from the HDG SEL position. The heading off mode can be used only when the mode select switch is in MAN; 28 volts dc is then applied directly to the heading switch holding coil.

(c) During the heading hold mode, the autopilot will maintain the airplane compass heading that exists one second after the mode is engaged. Any heading error output of the heading CT (which is clamped during heading hold) is applied past de-energized switch RS-6 to summing point 1 of Fig. 9 and amplified. The heading error signal is applied to an ADC q-potentiometer (heading q-potentiometer) which schedules signal gain as a direct function of airspeed to provide optimum control at all airspeeds. The signal is then applied to summing point 2.

(d) The heading error signal is applied to summing point 7 through the bank angle limiter which provides bank angle commands limiting of the heading error signal. For the heading hold mode, the bank angle command limit established by the bank angle limiter is 25 degrees. The mode of operation determines whether a bank angle command limit of 4, 8, or 25 degrees is utilized, and the corresponding bank angle command limit is applied by the bank angle limiter. The heading error signal, which is now a bank angle command, is summed with the output of the roll computer resolver at summing point 7. The result of the summation is applied to the bank rate limiter. The bank rate limiter limits the roll rate command through the roll computer. For heading hold mode, the rate command limit is 7 degrees per second. The mode of operation determines whether a rate command limit of 1.3, 4, 7, or 13 degrees per second is utilized. The output signal from the bank rate limiter is summed with tachometer feedback signals at summing point 8. These tachometer feedback signals provide damping for the roll computer servo loop. The signal from summing point 8 is amplified by the motor amplifier. The amplified signal drives the roll computer in a direction which causes the resolver (RS) signal to cancel the heading error signal at summing point 7.



1 NH JA8401 THRU JA8411
ML 9M–AOU THRU 9M–AOW, 9V–BBC, 9V–BBE
AR LV–JMW THRU LV–JMZ, LV–JND, LV–JNE
TM CR–BAA AND CR–BAB
IN EI–ASA THRU EI–ASH
CP C–GCPM AND ON

2 ALL EXCEPT 1

Heading Select Logic Circuit
Figure 11

(e) When the roll computer is driven by the heading error signal, the computer CT develops an error signal. The CT error signal results in aileron displacement as described in roll attitude hold mode. The aileron displacement produces a banked turn in the direction required to reduce the heading error and roll CT signals to a null.

(f) When the autopilot enters the localizer approach on-course submode of operation, the heading computer is clamped (heading hold configuration). Error signals from the heading CT are fed to summing point 2 as previously described except that the signal gain is reduced by energized switch RS-7 and an integrated heading term is supplied to summing point 2 by the lagged roll-integrator. The summation of heading error signals with the inverted integral of heading error signals provides mode damping for the localizer on-course submode of lateral radio beam operation.

(g) The heading switch will hold in the heading off (HDG OFF) position only when the roll axis is engaged and the navigation mode select switch is set to the manual (MAN) position. When the heading switch is set to HDG OFF, the heading computer is unclamped and allowed to synchronize continuously. Therefore, no heading information is provided to the roll control channel when the heading switch is in the HDG OFF position.

(6) Very High Frequency Omnirange (VOR) Control

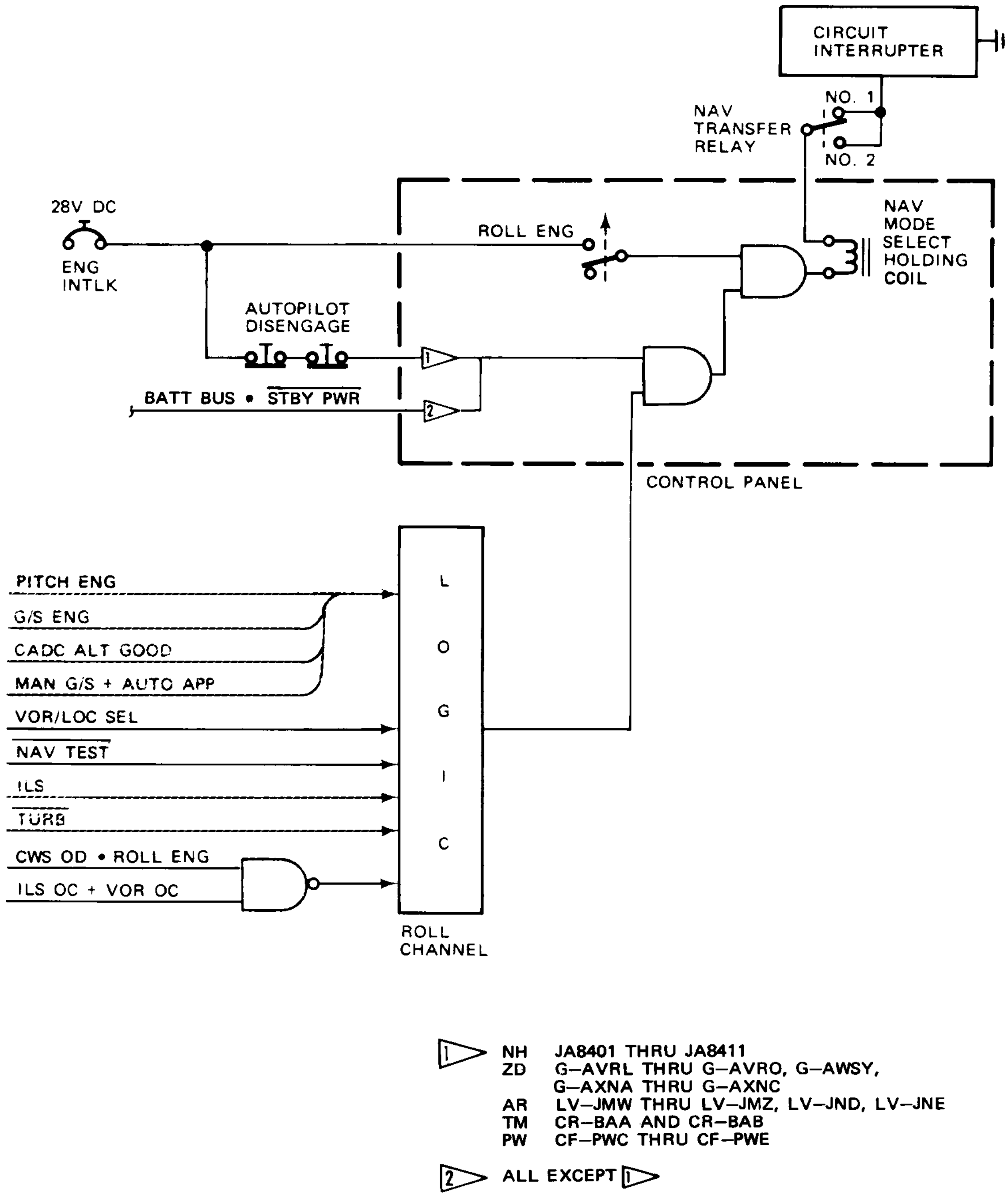
(a) The roll control channel operates in the VOR mode whenever VOR/LOC is selected (NAV mode select switch in VOR/LOC position) and the NAV receiver is tuned to a VOR frequency. In this mode, 28 volts dc from the engage interlock circuit breaker or 28 volts dc through the standby power transfer relay is applied to one input of an AND gate in the control panel (Fig. 11, Fig. 12, effectivity). With interlocks in the roll channel satisfied, and a NAV test NOT signal available or when the NAV TEST is inhibited when VOR/LOC mode is selected (Fig. 9), a logic 1 is applied to the other input of the AND gate. An additional logic 1 input is necessary to satisfy the NAV mode select switch interlocks. This input is obtained when any one of the remaining functions on figure 9 are available. The logic 1 inputs trigger the first AND gate and apply an input to a second AND gate. The 28 volts dc from the INTLK circuit breaker, through the roll engage switch, triggers the second AND gate to engage the NAV mode select holding coil. The holding coil is held to ground through the NAV transfer relay and a circuit interrupter.

(b) The CWS OD * ROLL ENG and ILS OC + VOR OC circuit is responsible for disengaging the NAV mode select switch and causing it to return to MAN position when the airplane is in the ILS or VOR on course mode and the control wheel is moved out of detent. When the control wheel is out of detent and the airplane is in ILS or VOR on course, a logic 1 is applied to a gate in the roll channel logic, is inverted to a logic 0 and applied to the first AND gate in the control panel. Operation of the NAV test switch on the NAV control panel or the NAV select switch on overhead panel P5 will also cause the NAV mode select switch to return to MAN position thus disengaging the VOR mode. The VOR mode is divided into intercept, capture, on-course, and over-station submodes.

(c) The VOR intercept submode can operate in conjunction with other roll axis modes such as heading select or heading hold. In this submode, radio beam deviation signals from the VOR/ILS receiver are modulated and amplified by a modulator amplifier and applied to the lateral beam sensor (LBS). The associated circuitry is armed. At this point, the approach progress display VOR/LOC annunciator illuminates amber. The LBS will automatically change state when the amplitude of the modulated radio beam deviation signal drops below a preset threshold. For the VOR mode, the threshold is established at a level of approximately 30 millivolts by the action of energized switch RS-23 which removes a bias voltage from the LBS (Fig. 13). When the LBS changes state (LBS drop-out), a signal is fed to the roll interlock circuits which terminates the intercept submode and initiates the capture submode and the VOR/LOC annunciator illuminates green.

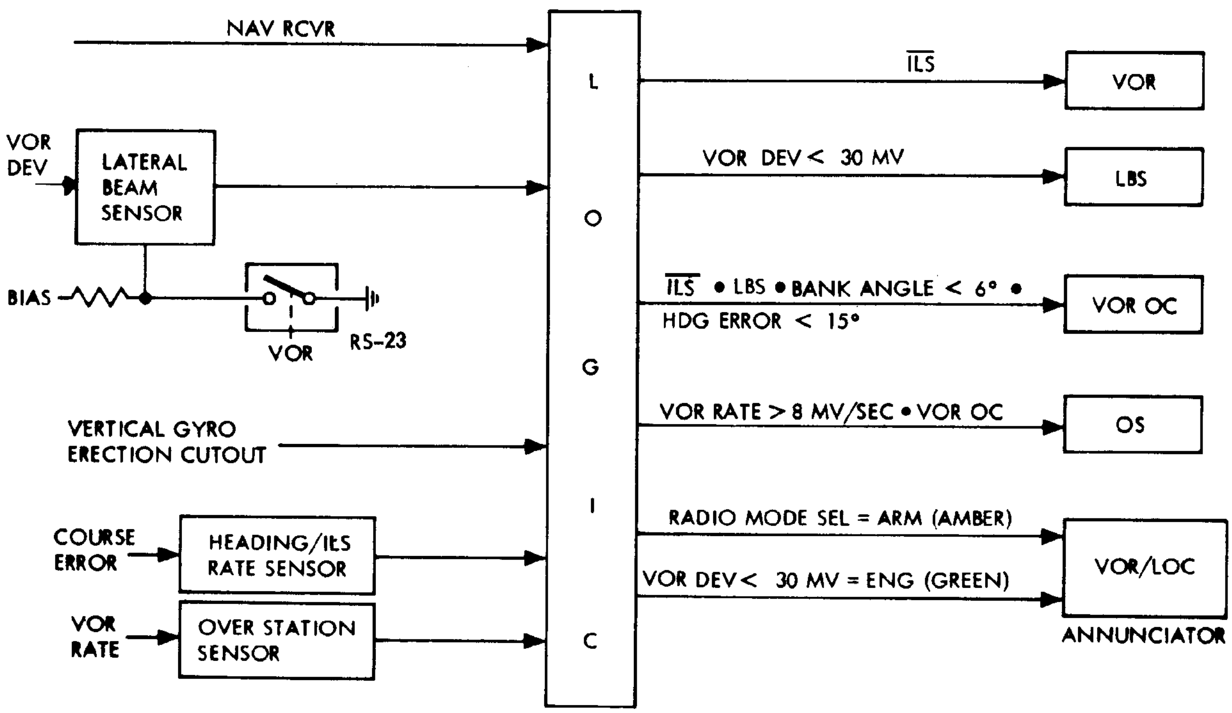
(d) VOR Capture Submode

1) When the capture submode is initiated by LBS drop-out, the radio beam is engaged and used to control the roll axis. The modulated radio beam deviation signal is amplified by a gain programmer which is in the maximum gain condition at this time, and applied to summing point 6 past de-energized switch RS-14 of Fig. 9. Because there are no other signals present at summing point 6 at this time, the amplified radio beam deviation signal is fed, essentially unaltered, to an amplifier limiter. The amplifier limiter amplifies the signal and the amplified signal is applied to summing point 2.



Nav/Mode Select Holding Coil Logic Circuit for VOR Mode
Figure 12

2) Course error signals from the heading course error synchro in the CDI are fed through a bus synchronizer. The bus synchronizer rejects extraneous quadrature components of the heading course error signal and synchronizes the signal with the autopilot 400-cps reference. The output from the synchronizer is fed to summing point 12 past de-energized switches RS-19 and RS-20. Heading course error signals from summing point 12 are amplified and fed to summing point 2. At summing point 2, the heading course error signals are summed with radio beam deviation signals. Radio beam deviation signals alone would result in the generation of a bank command which would turn the airplane toward the selected VOR radial, while course error signals alone would result in a bank command which would turn the airplane toward the course selected on the CDI. The result of summing the radio beam deviation and heading course error signals is the production of a continuously decreasing bank command which will cause the airplane to approach the selected VOR radial in an exponentially decreasing turn. The signal resulting from the summation at summing point 2 is fed to summing point 7 through bank angle limiter which establishes a bank angle command limit of 25 degrees for this submode. The output signal from the bank angle limiter is summed with the resolver sine output signal at summing point 7. Because this signal is proportional to bank angle, the output from the bank angle limiter is established as a bank angle command. Output signals from summing point 7 are fed to summing point 8 through the bank rate limiter which establishes its inherent roll rate command limit of 4 degrees per second during this submode. Bank angle command signals are summed with tachometer feedback signals at summing point 8 and the resulting signals are applied to the motor amplifier. The motor amplifier drives the roll computer whenever there is an output signal from summing point 8. When the roll computer is driven, the CT develops an error signal. The CT error signal results in aileron displacement as described in roll attitude hold mode. This aileron displacement produces banked turns in the direction required to decrease the CT output toward a null. Unless the VOR mode is disengaged, the roll channel operates in the capture submode from the time the submode is initiated until the conditions are satisfied for the VOR on-course submode.



VOR Mode Switching Logic Diagram
Figure 13

(e) VOR On-Course Submode

1) The roll control channel makes an automatic transition from the capture submode to the VOR on-course submode when the airplane bank is less than 6 degrees and the heading course error is less than 15 degrees (Fig. 13).

2) Bank angles less than 6 degrees are sensed by the roll erection cutout switch in the vertical gyro switch supplies a signal to the roll channel interlock circuits. Heading course error is detected by the rate sensor portion of the over-station rate sensor which receives heading course error signals past switch RS-21, summing point 11, and an amplifier (Fig. 9). When the heading course error signal decreases to a level corresponding to a heading error of less than 15 degrees, the rate/heading sensor supplies the interlock circuits with a signal which initiates the on-course submode when the bank angle is simultaneously less than 6 degrees (Fig. 13).

3) The on-course submode operates essentially the same as the capture submode except that radio beam deviation signals are applied to summing point 5 past deenergized switch RS-13 (Fig. 9). Heading course error signals are also applied to summing point 5 but past deenergized switch RS-24. The radio beam deviation and heading course error signals are summed and applied to the heading/beam integrator which is released from the reset state during this submode. The heading/beam integrator effectively increases the long-term gain of the radio beam deviation signals and provides long-term washout of the heading course error signals into summing point 2, which the heading/beam integrator feeds. The increase in long-term (low frequency) radio beam deviation gain provides for crosswind drift compensation with minimum standoff error. Output signals from the heading/beam integrator are summed with radio beam deviation and heading course error signals, and the resulting signal is applied to summing point 7 through the bank angle limiter. The bank angle limiter establishes a bank angle command limit of 8 degrees for this submode. Output signals from the limiter are summed with RS signals at summing point 7. The output signals from summing point 7, which will be present whenever the bank angle associated with the RS rotor position fails to correspond to the bank angle command, are fed to summing point 8 through the bank rate limiter. In this submode, bank rate limiter operates at the lowest limit level (normally 4 degrees per second); however, the roll rate command limit is reduced further to 1.3 degrees per second by the application of additional tachometer feedback to summing point 8 past deenergized switch RS-4. The tachometer feedback signals are summed with outputs from the bank rate limiter and signals resulting from the summation are applied to the motor amplifier. The roll computer will provide an output signal when it receives an error signal from summing point 7. When the roll computer is driven, the CT develops an error signal. This error signal results in aileron displacements as described in roll attitude hold mode. Aileron displacements produce banked turns in the direction required to reduce the output from summing point 2 and the CT error signal to a null. The roll channel continues to operate in the on-course submode until either the over-station submode is initiated or the VOR mode is disengaged.

(f) VOR Over-Station (OS) Submode

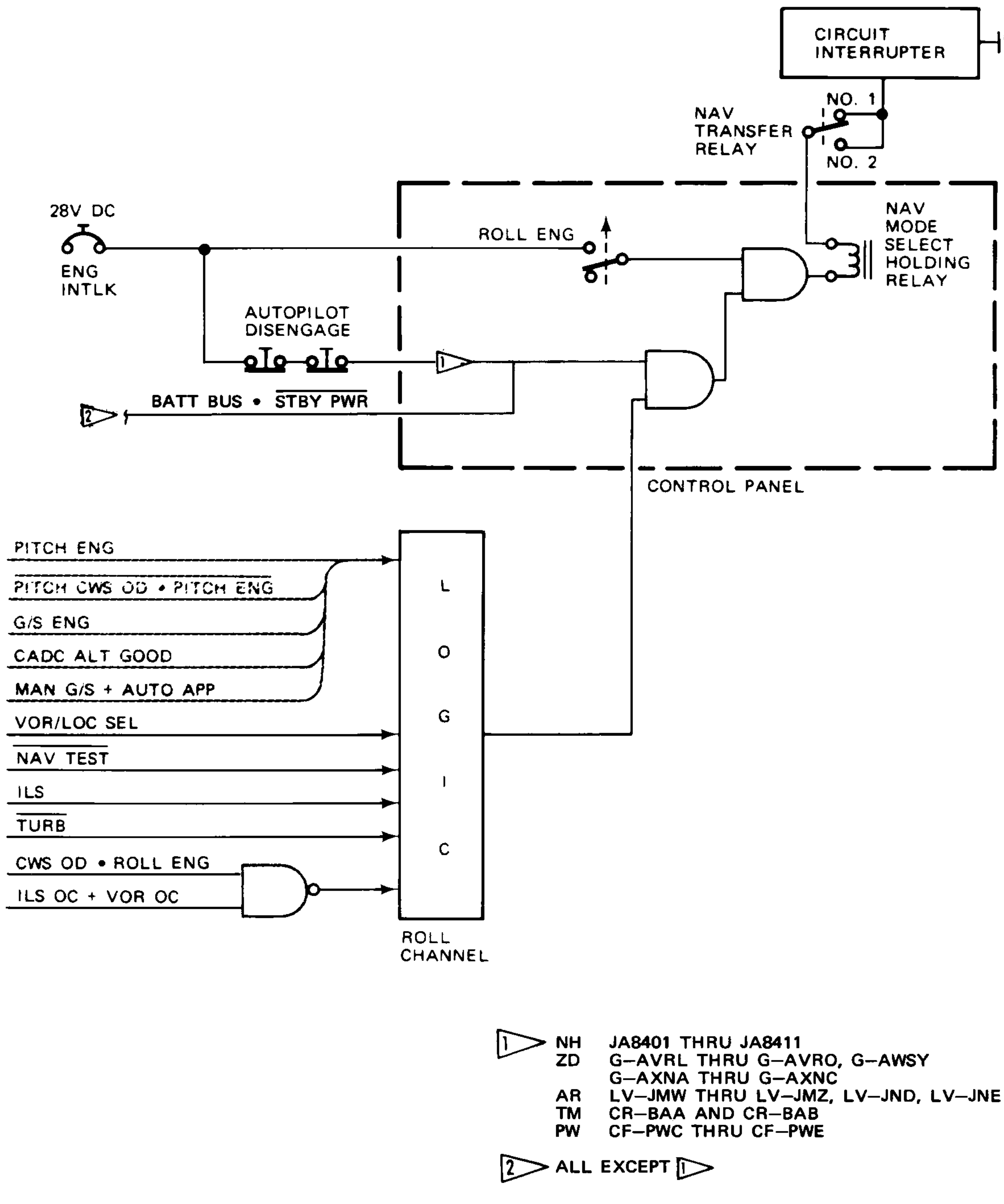
1) The roll channel automatically switches to the over-station submode when the OS portion of the over-station rate sensor senses the rapid fluctuations in radio signal intensity associated with the zone of confusion. (Zone of confusion refers to the area in proximity to VOR stations in which the erratic nature of the radio signals renders them unsuitable for controlling the autopilot.) When the zone of confusion is entered, the roll channel automatically decouples from the radio beam and controls the airplane on the drift-corrected heading existing when the radio signals are decoupled.

2) The OSS receives radio signals which have been processed by the rate network and an amplifier. Because of the effect of the rate network, the amplitude of these signals

3) The OSS receives radio signals which have been processed by the rate network and an amplifier. Because of the effect of the rate network, the amplitude of these signals is proportional to the time rate of change in the amplitude of the radio deviation signal. During the on-course submode, the sensitivity of the OSS is increased by switch RS-25 which provides a ground return for a diode which acts as part of a voltage-doubler circuit in the OSS. During the on-course submode, the threshold sensitivity of the OSS is approximately 8 millivolts per second. When the radio rate exceeds the threshold level, the OSS initiates the over-station mode.

4) During the over-station submode, the radio deviation and heading course error signals are removed from summing point 5 by energized switches RS-13 and RS-24. The radio deviation signal is removed from summing point 6 (and summing point 2) by energized switch RS-14. Because the radio signals are disconnected, the roll channel is controlled by heading course error signals while the airplane is in the zone of confusion. It should be noted that the heading/beam integrator is not reset during this submode and therefore stores drift-correction information. The roll channel remains in this submode until the airplane passes out of the zone of confusion. At this time, the radio rate drops below the threshold level and the OSS causes the roll channel to revert to the on-course submode. The bank angle command and roll rate command limits for the over-station submode are the same as those for the on-course submode(Fig. 13).

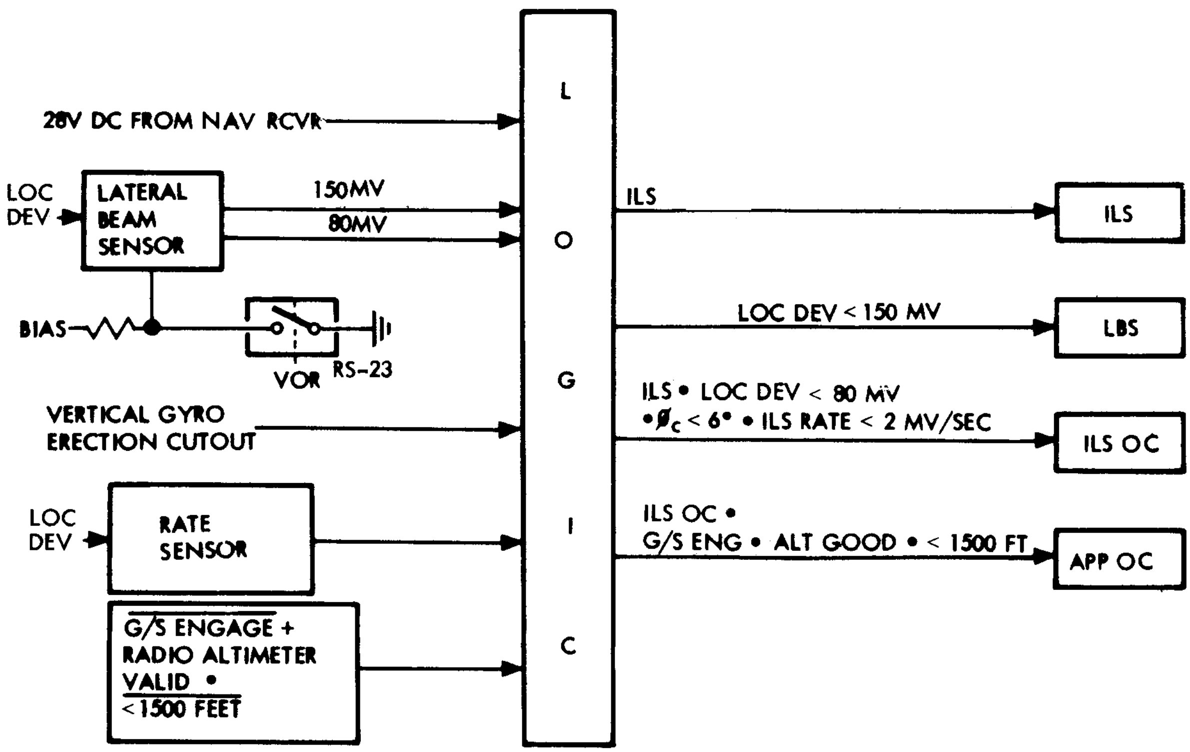
(7) Localizer Mode
 (a) The roll channel operates in the localizer mode whenever either VOR LOC or AUTO APP is selected on the control panel and the VOR/ILS receiver is tuned to an ILS frequency. When the NAV mode select switch is in VOR/LOC (localizer) position, the following inputs are required to engage the NAV mode select switch. The pitch mode select switch must not be in the TURB position. Selection of the TURB (turbulence penetration) mode will prevent selection of the localizer mode or will cause the localizer mode to drop out by providing a logic 0 input to the first AND gate in the control panel (Fig. 14). An additional logic 1 input from any of the functions in the remaining logic circuit on figure 11 is necessary to satisfy the NAV mode select switch interlocks. This circuit operates in the same manner as in the VOR mode.
 (b) The localizer mode is divided into intercept, capture, on-course, and approach on-course (AUTO APP position only) submodes.
 (c) The localizer intercept submode functions the same as the VOR INTERCEPT SUBMODE EXCEPT THAT SWITCH RS-23 is deenergized and the LBS threshold is changed to approximately 150 millivolts (Fig. 15).
 (d) The localizer capture submode functions approximately the same as the VOR capture submode. The differences between these submodes are as follows:
  1) The radio beam deviation signal path to summing point 6 is applied past switch RS-15 of Fig. 9 instead of switch RS-14. This provides a gain change between the VOR and localizer submodes.
  2) Radio rate (cross-beam rate) signals from the rate network are applied to summing point 6 past switch RS-26. As a result of the summation of cross-beam rate signals with radio beam deviation signals, the airplane radio beam intercept angle will remain essentially constant as long as the output from summing point 6 exceeds the limit point of the amplifier limiter. When the output from summing point 6 drops below the limit point, the intercept angle decreases exponentially.
  3) Heading course error signals are applied to summing point 12 past deenergized switch RS-19.
  4) The bank angle command limit is the same as for VOR capture (+ 25 degrees) but the roll rate command limit is increased to 7 degrees per second.



Nav/Mode Select Holding Coil Logic Circuit for LOC Mode
Figure 14

(e) The localizer on-course submode functions in approximately the same manner as the VOR on-course submode. The differences are as follows:
   1) The radio beam deviation signals are applied to summing point 5 past deenergized switch RS-12, instead of RS-13, to provide a gain change.
   2) No heading course error signals are applied to the heading/beam integrator through summing point 5. Therefore, radio beam deviation signals alone are applied to the heading/beam integrator during the localizer on-course submode.
   3) Cross-beam rate signals are applied to rate sensor past de-energized switch RS-22, summing point 11, and an amplifier. The rate sensor serves the same function as for the VOR on-course submode except the sensor trips when the cross-beam rate drops to 2 millivolts per second rather than tripping for a heading course error of less than 15 degrees, which is the case during the VOR mode (Fig. 15).
   4) The bank angle command limit for this submode is the same as that for the VOR on-course submode (± 8 degrees), but the roll rate command limit is increased to 7 degrees per second.

(f) The localizer approach on-course submode is initiated during the on-course submode after the glide slope is engaged by the pitch axis and the airplane altitude (sensed by the radio altimeter) is less than 1500 feet. This submode differs from the on-course submode as follows:
   1) Gain programmer decreases the amplitude of the radio beam deviation signal as a function of altitude (gain programming), starting with the initiation of the approach on-course submode. The program control signals are supplied by the pitch channel in which radio deviation gains are also programmed to compensate for beam convergence as the transmitter is approached.
   2) Heading course error signals are removed from summing point 12 and therefore from summing point 2 by energized switch RS-19 of Fig. 9. Heading information for this submode is provided by the heading computer which reverts to the heading hold (clamped) configuration. Heading error signal gain is decreased during this submode by energized switch RS-7. Heading error signals are applied to summing point 2, both directly and through summing point 3 and the lagged roll integrator. Summing heading error signals with the inverted integral of signal from the lagged roll integrator produces washed-out heading error signals to provide mode damping without contributing to beam standoff in the presence of wind shear. Roll attitude signals from the vertical gyro are also fed to the lagged roll integrator to produce the lagged roll signals which provide additional damping and improve the wind shear performance during the approach on-course submode.



LOC Switching Logic Diagram
Figure 15

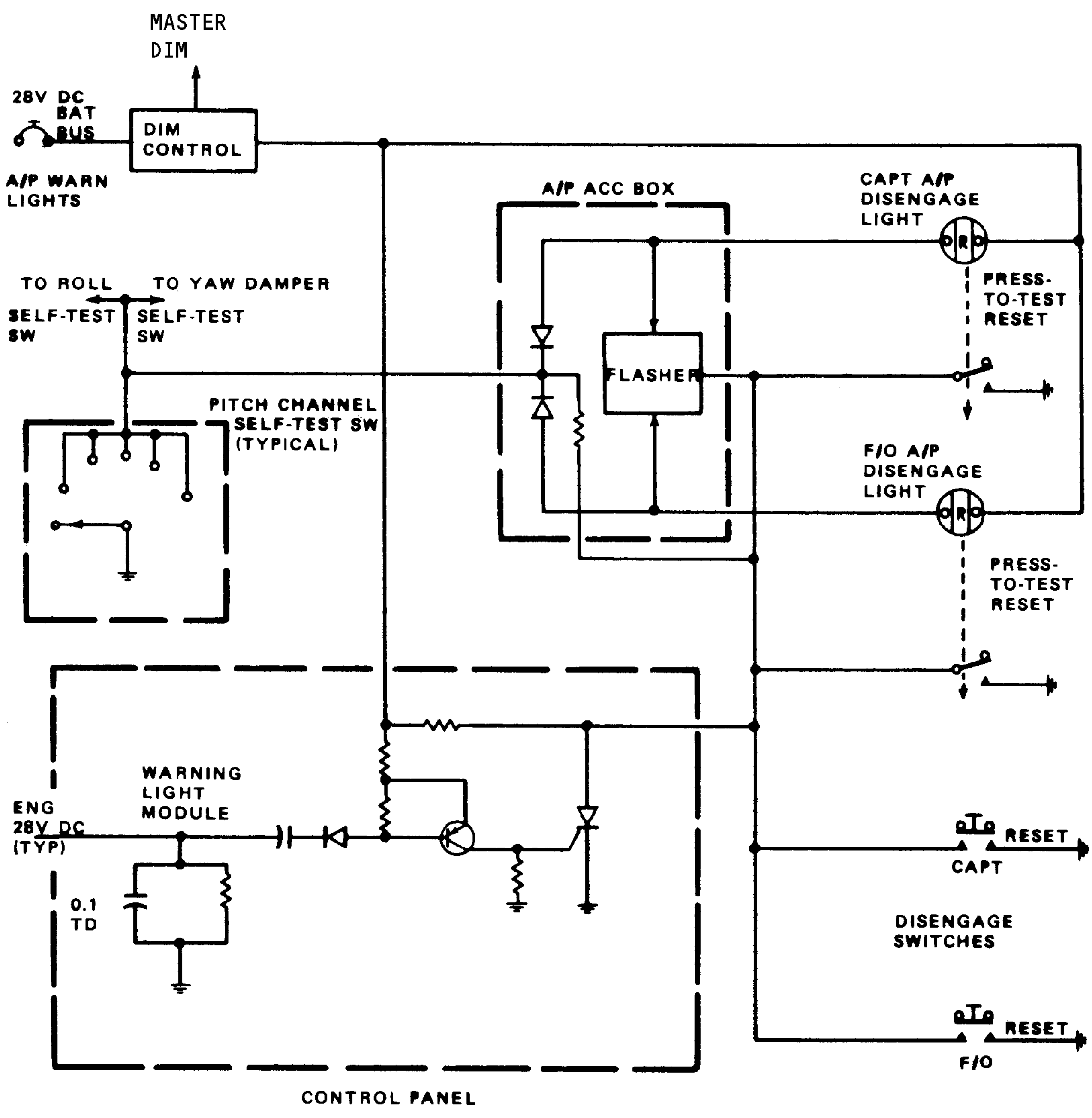
3) The bank angle command and roll rate command limits remain the same for this submode as for localizer on-course.

(8) Turbulence Penetration Mode

(a) Although the turbulence penetration mode is primarily an autopilot pitch axis mode, it also affects the roll axis. Whenever the pitch channel is placed in turbulence penetration, the roll axis bank angle command limit is 8 degrees and the localizer mode is made inoperative.

(9) Warning Lights

(a) The disengage warning light circuit operates in the following manner. When electrical power is applied to the airplane, power is applied to the disengage lights, flasher and warning light module (Fig. 16). If the warning lights illuminate steadily, one or more of the pitch, roll, or yaw self-test switches is in a test position other than OFF. Proper operation of the autopilot is dependent upon all self-test switches being in the OFF position. Upon autopilot channel engagement, 28 volts dc engage interlock voltage is applied through a nuisance suppressor time delay (0.1 second) to a capacitor in the base circuit of a gating transistor. When interlock voltage is interrupted the gating capacitor is grounded and a momentary negative pulse developed across the capacitor triggers the transistor which fires the SCR. The SCR conducts and provides a ground for the flasher, thereby causing the warning lights to illuminate and flash. Positive voltage on the transistor base keeps the transistor turned off. However, the SCR continues to conduct and the warning lights continue to flash. The SCR ceases to conduct and the lights are extinguished when the light circuits are reset by depressing either warning light cap or by depressing either disengage switch. Once the lights are reset, they are rearmed to monitor the remaining interlock. If the autopilot is disengaged through operation of either disengage switch, the warning lights will illuminate momentarily while the switch is depressed and then go out.



Autopilot Warning Lights Circuit
Figure 16

6. Autopilot Pitch Channel Operation

A. General

(1) The pitch channel provides pitch axis control by amplifying, shaping, computing and coupling error and command signals to drive the elevator control surface through an actuator (Fig. 16). Error and command signals originate in the input sections, are summed and are processed in the pitch processing section. Intelligence from the pitch processing section is fed to the output section. The output section together with the position feedback and stabilizer trim section further refines pitch channel signals to the proper DC voltage levels necessary to actuate the transfer valve in the appropriate elevator power unit. A lift compensation signal, from the roll channel is also fed directly to the output section.

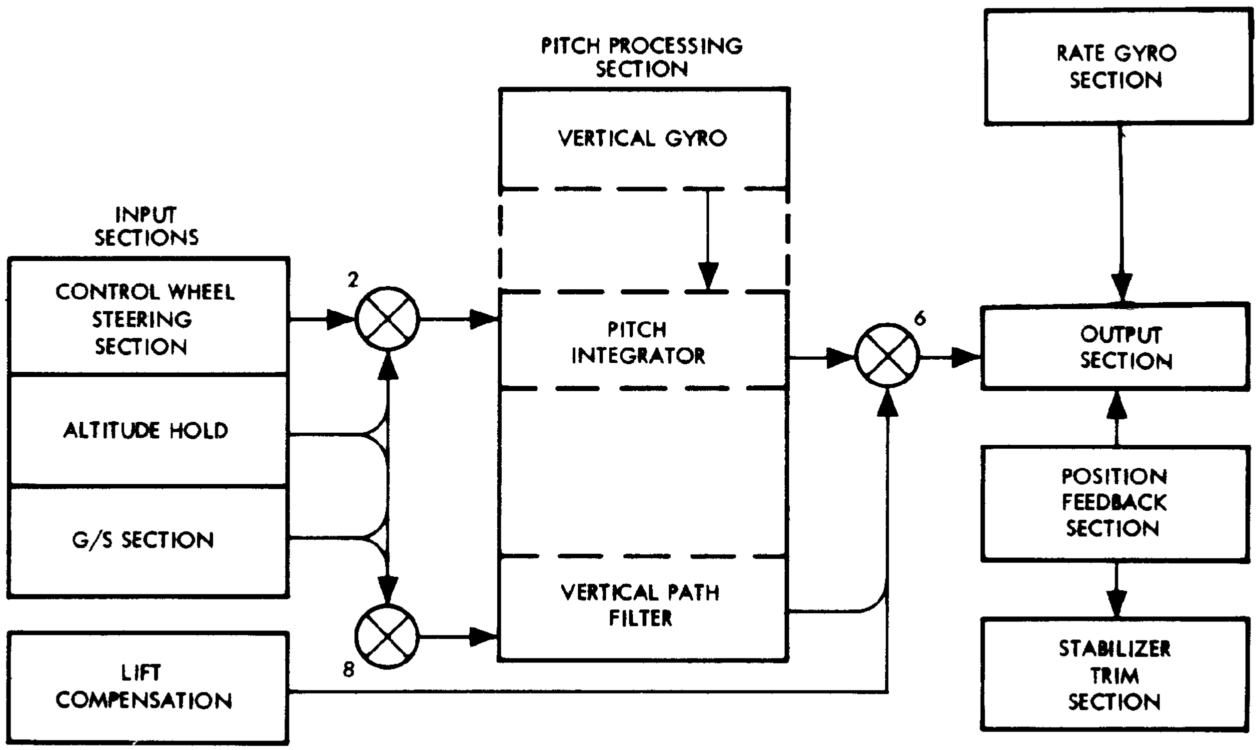
(2) The pitch channel provides pitch axis control by amplifying, shaping, computing and coupling error and command signals to drive the elevator control surface through an actuator (Fig. 17). Error and command signals originate in the input sections, are summed and are processed in the pitch processing section. Intelligence from the pitch processing section is fed to the output section. The output section together with the position feedback and stabilizer trim section further refines pitch channel signals to the proper DC voltage levels necessary to actuate the transfer valve in the appropriate elevator power unit. A lift compensation signal, from the roll channel is also fed directly to the output section.

(3) Discussion of the pitch channel operating modes is based on the block diagram (Fig. 18).

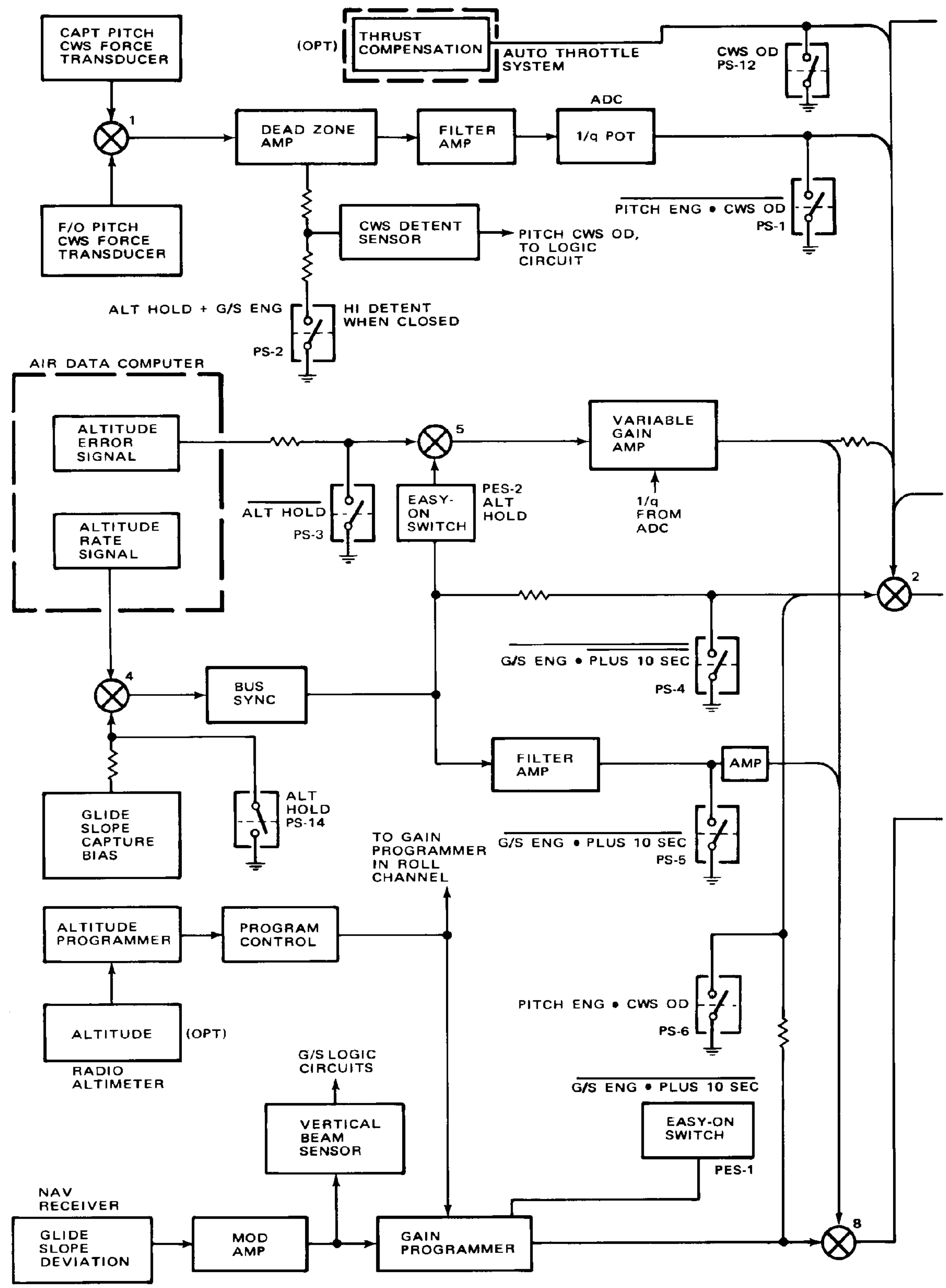
B. Operating Modes

(1) Synchronization Mode

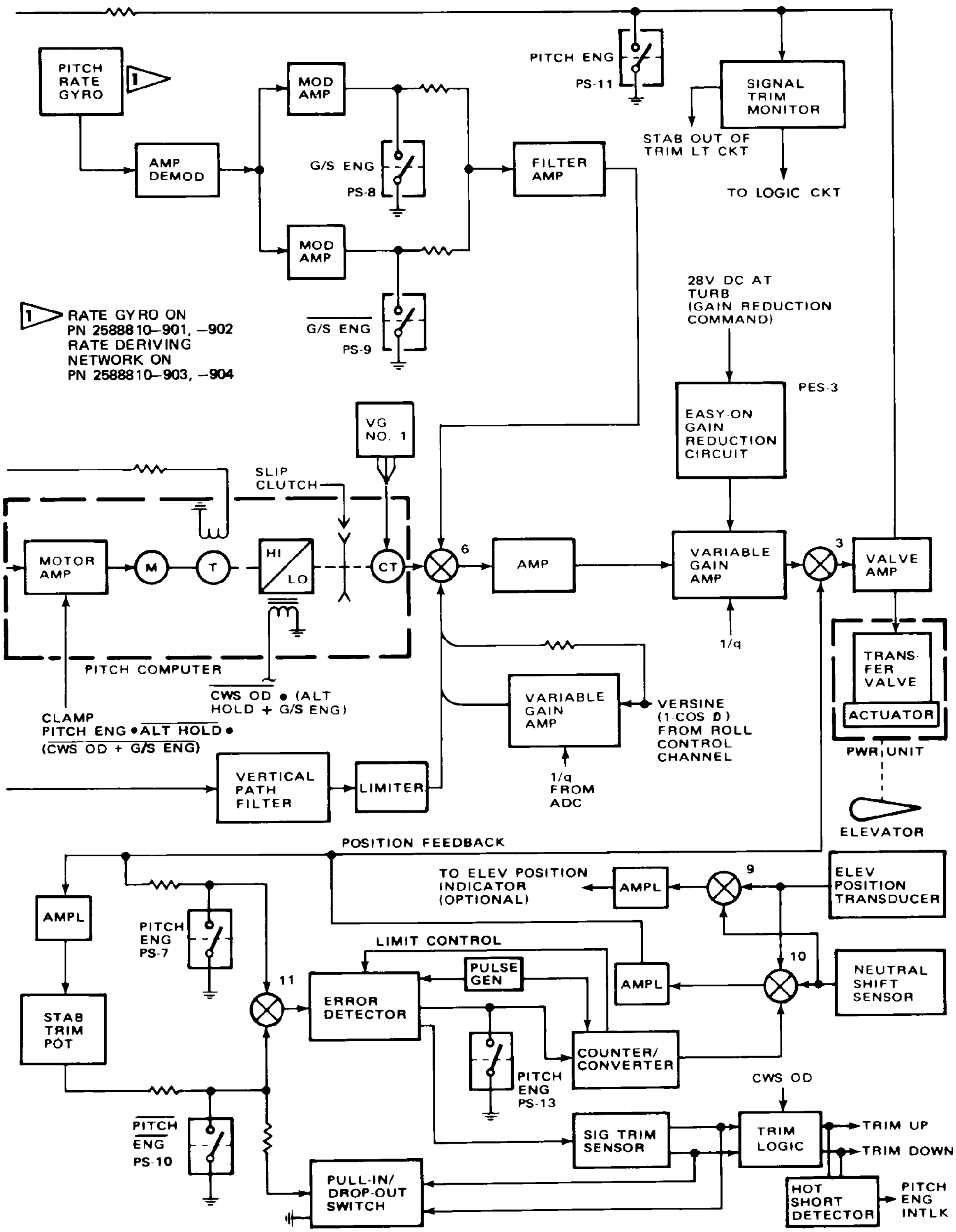
(a) Prior to pitch channel engagement, the following functions are provided in the pitch control channel: pitch attitude synchronization, valve amplifier synchronization, and elevator position synchronization. Pitch attitude and valve amplifier synchronization null the valve amplifier output signal, while synchronizing the pitch servo control transformer (CT) to the airplane attitude. The valve amplifier output signal is modulated within the valve amplifier and the resulting 400-cps signal is routed back past open engage switch PS-11 to summing point 2 and the motor control amplifier in the pitch computer.



Pitch Control Channel Simplified Block Diagram
Figure 17



Pitch Control Channel Block Diagram
Figure 18 (Sheet 1)



Pitch Control Channel Block Diagram
Figure 18 (Sheet 2)

(b) Output from the motor control amplifier rotates the motor-tachometer generator. This drives the CT through the numerically lower ratio (LO) of a dual ratio gear train. Rotation speed is governed by output from the motor-tachometer generator applied to summing point 2. The CT rotates until the valve amplifier output is nulled. When this output is eliminated, motor-tachometer generator and CT rotation cease, and the rotor position of the CT is synchronized with the airplane pitch attitude sensed by the vertical gyro. Modulated valve amplifier output is also applied to the signal trim monitor. Until pitch attitude and valve amplifier synchronization are complete, the signal at the input to the trim monitor will cause the trim monitor to prevent pitch control channel engagement. Synchronization in this manner eliminates transient pitch commands up on engagement of the pitch control channel.

(c) Elevator position synchronization is accomplished as follows: Signals from the stabilizer neutral shift sensor and the elevator position transducer are summed at summing points 9 and 10. Output signals from summing point 9 are amplified and fed to the elevator position indicator. The position signals at summing point 10 are summed with the output of a counter/converter (part of a digital synchronizer), and the resultant signal is fed to summing point 3 and past de-energized switch PS-7 to summing point 11. Signals from summing point 11 are sensed by an error detector (part of a digital synchronizer) which feeds clock pulses past de-energized switch PS-13 to the counter/converter. The phasing of the error signal determines whether the clock pulses are added to or subtracted from the count stored in the counter/converter. This is analogous to driving an electromechanical servomechanism in either direction as a function of error signal phase. The counter/converter sends a feedback signal to the error detector when the counter is full to inhibit the clock pulses. This corresponds to driving an electromechanical servomechanism to its mechanical limit. The amplitude of the counter/converter output signal is proportional to the count in the counter. This signal is fed to summing point 10 where it is summed with elevator position and neutral shift sensor signals. During synchronization, the counter/converter output will automatically adjust to a value which reduces the error signal inputs to summing points 11 and 3 to approximately zero. Elevator position synchronization establishes a reference trim position for computation of automatic stabilizer trim commands when the pitch channel is engaged. The reference position information is stored in the counter/converter which does not receive clock pulses when the pitch channel is engaged because of the action of a toggle which is represented (Fig. 18) by PS-13.

(2) Manual Mode

(a) The basic pitch channel operating mode is the manual (pitch attitude hold) mode. To engage the pitch channel in the manual mode (NAV mode select switch in MAN position and pitch mode select switch in OFF position), the flight control system A or B switches must be on, for airplanes with A and B system switching on autopilot control panel, or the flight control B switch must be on for airplanes with B system only on autopilot control panel, the vertical gyro must be up to speed, the main electric stabilizer trim not in operation, the stabilizer trim cutout switch in the normal position, a valid pitch calibrator must be installed, the flaps must be synchronized and stabilizer trim servo speed change relay must be good (2-second time delay), 30-volt dc pitch channel power must be good, a hot short NOT signal must be available, a pitch control wheel steering out of detent NOT, stabilizer mistrimmed NOT signal must be available and air data computer airspeed interlock must be good. On airplanes incorporating SB 22-1018 with pitch channel part No. 2588810-903,-904 installed, derived rate 115 vac must be good (Fig. 19).

(b) The interlock signals are processed through a logic chain in the pitch channel, summed, and fed to one input of an AND gate in the pitch control panel. The 28 volts dc from the autopilot disengage switches provides a corresponding 1 input to the AND gate which triggers the AND thus providing a 1 output to one input of a second AND gate in the control panel. When integrity of the autopilot disconnect circuits are satisfied, 28 volts dc from the engage interlock circuit breaker triggers the second AND gate. This action allows the pitch engage switch to unlatch and energizes the pitch engage switch solenoid. The pitch engage switch solenoid is held to ground through the circuit interrupter. Except for the trim monitor circuit and the CWS circuit, failure of any of the prescribed interlocks or deliberate operation of the disconnect switches will disengage the pitch channel. After engagement, trim monitor signal will not cause the pitch channel to disengage. This is also true for control wheel steering inputs.

(c) Upon engagement, if no other mode is selected, the pitch channel automatically operates in the pitch attitude hold mode. The motor clamp circuit for the pitch computer is activated, locking the servo assembly in the position corresponding to the vertical gyro sensed pitch attitude existing upon engagement (Fig. 18). Changes in airplane pitch attitude are sensed in the vertical gyro. The CT, directly coupled to the vertical gyro, provides an output signal to summing point 6 proportional to the attitude change. This signal is routed to the valve amplifier which commands elevator movement to restore original pitch attitude. When the elevator moves as a result of the command signal, the elevator position transducer signal is fed to summing point 3 to null the command signal and stops the elevator movement. As the airplane returns to the original pitch attitude, the CT error signal diminishes to zero. The elevator position signal, at summing point 3, restores the elevator to neutral position. Damping of airplane movement is provided by the signal from pitch rate gyro which is shaped in the wipeout (bandpass) filter in the rate gyro filter. Output from the pitch rate gyro is routed through a filter and summing point 6, where it is added to the pitch attitude error signal. Gain of the variable gain amplifier is inversely proportional to the signal from the air data computer to compensate pitch command signals for changes in airplane control dynamics as a function of airspeed.

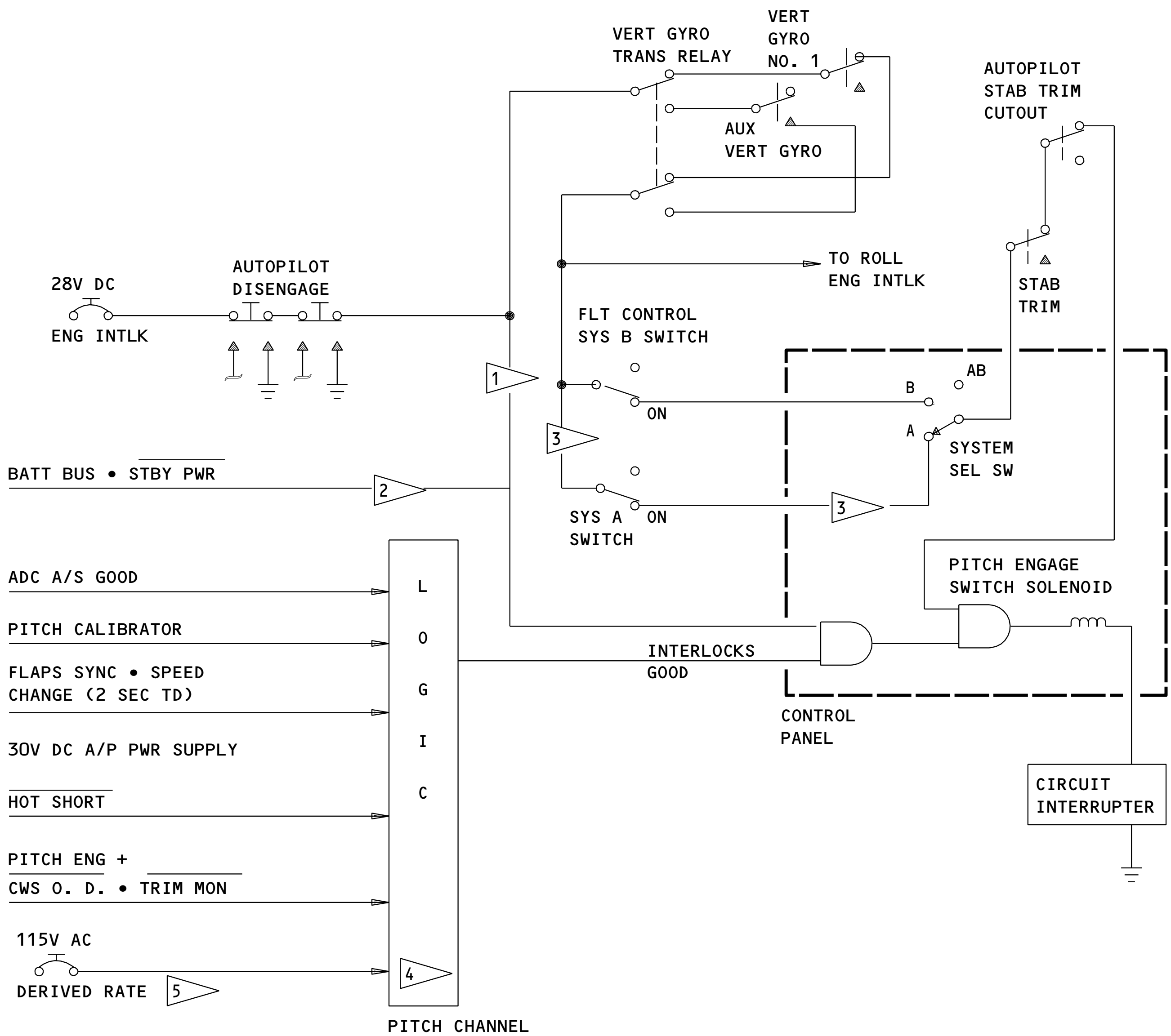
(3) Lift Compensation

(a) During turn maneuvers, a lift compensation (pitch up) signal, proportional to the versine (1-cosine) of the commanded bank angle is routed through a variable gain amplifier to summing point 6. At summing point 6, this signal is summed with the pitch computer CT output to provide an elevator pitchup command from the valve amplifier. Elevator motion stops when position transducer signal at summing point 3 cancels the pitch command signal. The amplifier gain is inversely proportional to a signal from the air data computer to compensate the lift compensation signals for changes in airplane control dynamics.

(4) Thrust Compensation (if installed)

(a) An input from the auto throttle system is fed to summing point 2 to give down elevator command for acceleration and to give up command for deceleration. This action complements the auto throttle system.

(b) Thrust compensation is not effective when the pitch computer is clamped (pitch attitude hold) or when the pitch CWS is out of detent. The thrust compensation input to summing point 2 is grounded by energized switch PS-12 when the CWS is out of detent.



1 AR LV-JMW THRU LV-JMZ
LV-JND AND LV-JNE
IC VT-EAG THRU VT-EAL
TZ CF-TAO AND CF-TAN
PV CF-EPL, CF-EPO AND CF-EPR

2 ALL EXCEPT 1

3 REMOVED ON AR LV-LIU AND ON
IC VT-ECP AND ON
PV C-GEPA AND ON
TZ C-GTAQ AND ON

4 DERIVED RATE LOGIC ON PITCH CHANNEL
PART NO. 2588810-903, -904

5 ON AIRPLANES INCORPORATING SB 22-1018

Pitch Engage Logic Circuit
Figure 19

(5) Pitch (Stabilizer) Trim

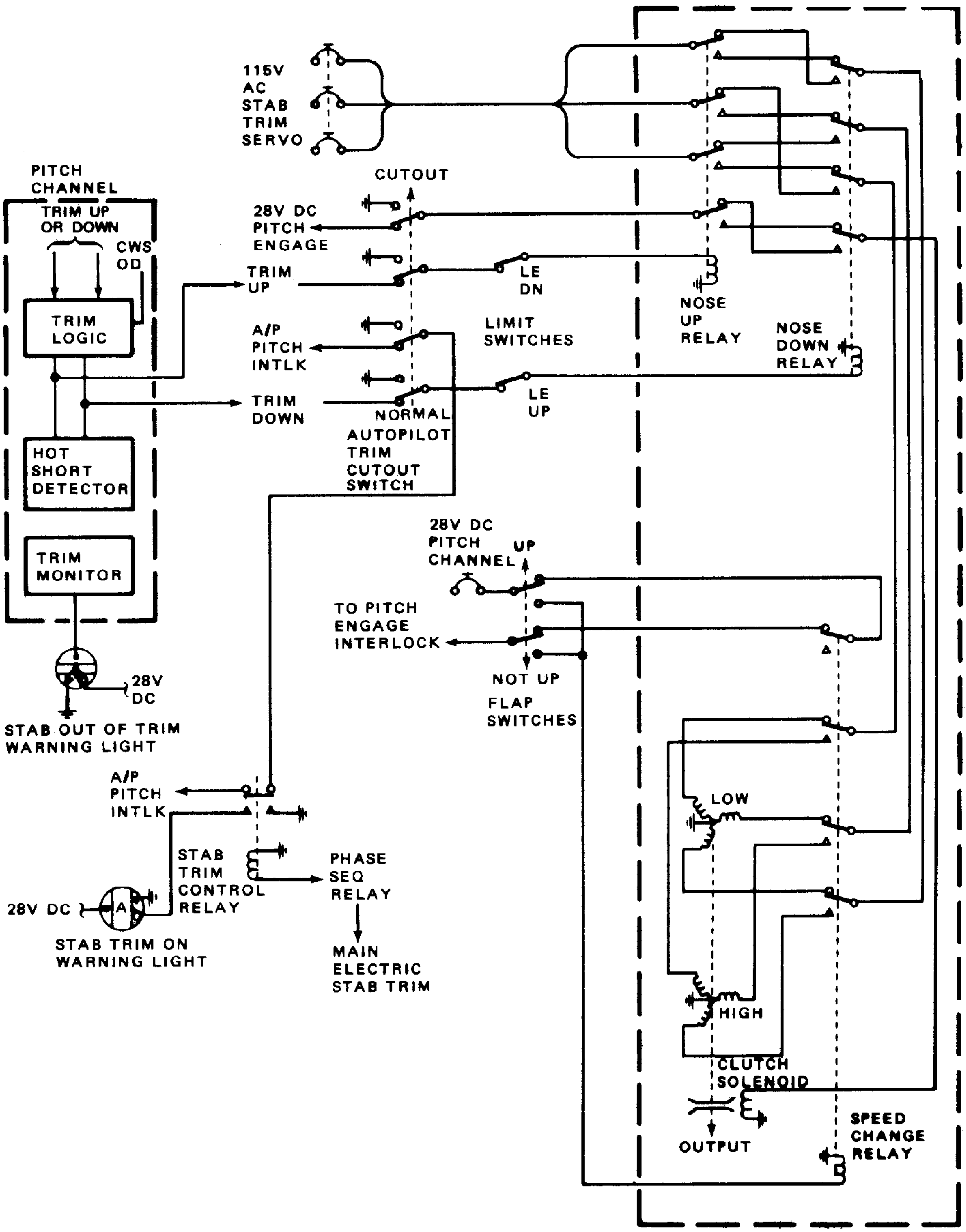
(a) Automatic pitch (stabilizer) trim positions the stabilizer to compensate for airplane trim changes. The stabilizer trims remove sustained elevator displacements from neutral. Automatic stabilizer trim is disabled during control wheel steering or when the pitch channel is disengaged. The pitch channel is disengaged if the main electric trim switches on the pilots' control wheels are operated. The stabilizer trim on warning light illuminates whenever the main electric switches are operated. Elevator position signals from the elevator position transducer and stabilizer position from the neutral shift sensor are combined with the counter/converter output at summing point 10. The output from summing point 10 is amplified and fed to the automatic stabilizer trim potentiometers which vary the trim threshold as a function of stabilizer position. Output signals from the automatic stabilizer trim potentiometers are fed to summing point 11, past de-energized switch PS-10. (The output from the automatic trim potentiometers is the only input to summing point 11 when the pitch channel is engaged because of the action of energized switch PS-7 which shunts the other input to ground.) The output from summing point 10 is fed to the error detector which produces a train of pulses when its input signal exceeds the trim threshold. This train of pulses, which constitutes a trim command, is routed to the signal trim sensor which sends trim up or trim down signals to the trim logic on the pitch interlock card. The trim logic provides power to energize one of the trim servo relays and provides interlock voltage to prevent simultaneous trim-up and trim-down (hot short) commands. Output from the signal trim sensor is also routed to a pull-in/ drop-out switch in the amplifier. In its normal state, this switch grounds one end of a resistor to attenuate the input signal to the error detector. When a trim command is given, the pull-in/drop-out switch removes the ground from the resistor increasing the sensitivity of the error detector. Increased error detector sensitivity causes the stabilizer to be driven beyond the position corresponding to the trim threshold, eliminating hunting in the servo loop.

(b) When a trim-down command is given and the airplane is at cruising speed, power is applied through the autopilot trim cutout switch and limit switches to a nosedown relay in the stab trim servo (Fig. 20). The nosedown relay applies 115 volts ac 3-phase power to the low speed windings of the servo and 28 volts dc power from pitch engage circuits to the clutch solenoid. This action energizes the clutch to drive the stabilizer toward a trimmed condition. A trim-up command follows a parallel path to the stab trim servo; however, the noseup relay is energized. The relays change the phase relationship of the 115 volts ac inputs to the low or high speed windings thus reversing the output to the stabilizer. When the airplane is not in the cruising mode (flaps not up), 28 volts dc is supplied to the speed change relay which, when energized, routes power to the high speed windings of the servo.

(c) The stabilizer out of trim warning light, located on the center panel is controlled by the trim monitor circuit. The light will illuminate when an out-of-trim condition exists between the stabilizer and elevator for approximately 12 seconds.

(6) Control Wheel Steering (Pitch Rate Maneuvering)

(a) Force transducers (sensors) in the control column's elevator linkage provide an output proportional to force applied to the control column. When this output reaches a predetermined level, the deadzone amplifier and the control wheel steering (CWS) detent sensor portion of the deadzone amplifier provides a CWS out-of-detent (OD) interlock voltage. This interlock voltage deactivates the pitch computer motor clamp circuit and provides a path for the composite force transducer signal out of summing point 1 (Fig. 18). The summation performed at summing point 1 gives both control wheels equal authority. The force transducer signal is routed to summing point 2 through the deadzone amplifier, low pass filter amplifier, and l/q potentiometer in the air data computer. The low pass filter amplifier attenuates resonant frequencies from the deadzone amplifier output. The deadzone amplifier causes the pitch channel to ignore small nuisance signals from the force transducers. The 1/q potentiometer compensates for changes in airplane dynamics due to airspeed. The output from summing point 2 is applied to the unclamped motor amplifier in the pitch computer, causing the amplifier to drive the motor-tachometer generator, which drives the pitch CT through the low ratio gear train. Vertical gyro signal is applied to the pitch computer resulting in CT output which is routed to the valve amplifier to command a change in elevator position.



Stabilizer Trim Circuits
Figure 20

(b) Movement of the pitch computer servo assembly and elevator continues until control wheel force is released or until the mechanical limit in the servo assembly is reached. When control wheel force is released, CWS OD interlock is removed, and the servo assembly is clamped at the new reference position. Output from the CT continues and causes airplane pitch attitude changes until the vertical gyro output results in a CT output null. Switch PS-2, when closed, increases the control wheel steering detent level to hi-detent.

(7) Altitude Hold Mode

(a) The altitude hold mode is selected at the control panel. To engage the pitch channel in the altitude hold mode, the pitch mode select switch must be in the ALT HOLD position and 28 volts dc flag voltage must be available from the air data computer altitude monitor, a glide slope engage NOT signal must be available, a CWS OD NOT and a pitch engage signal must be available (Fig. 21).

(b) The mode interlock signals are processed through a logic chain in the pitch channel and fed to one input of an AND gate in the pitch control panel. 28 volts dc from the engage interlock circuit breaker provides a corresponding 1 input to the AND gate which triggers the AND thus providing a 1 output to one input of a second AND gate. The 28 volts dc from the pitch channel circuit breaker through the system select switch (A or B) and the closed pitch engage switch triggers the second AND gate which energizes the pitch mode select switch solenoid.

(c) Selection of this mode disengages pitch attitude hold by unclamping the pitch computer. Altitude error signals from the ADC are applied to summing point 5 past de-energized switch PS-3 (Fig. 18). Altitude rate signals from the ADC are applied to a bus synchronizer through summing point 4. The bus synchronizer removes quadrature components from the rate signals and synchronizes the rate signals to the autopilot 400-Hz reference. The output of the bus synchronizer is fed to easy-on switch PES-2 which applies the altitude rate signals to summing point 5 approximately 3 seconds after engagement of the altitude hold mode. Altitude rate signals are summed with altitude error signals at summing point 5. The altitude rate signals are summed with the error signals to provide mode damping which eliminates "porpoising" about the reference barometric altitude. Output from summing point 5 is fed to a variable gain amplifier, the gain of which is varied as an inverse function of differential pressure to compensate for changes in airplane control dynamics as a function of airspeed. The output of the variable gain amplifier is routed through summing point 8 and the vertical path filter to summing point 6.

(d) Output from summing point 2 results in rotation in the pitch computer in which the gear changer has selected the numerically higher gear ratio to form an electromechanical integrator. Integrator movement results in CT output which is routed to summing point 6. Output from summing point 6 causes the valve amplifier to command elevator movement. The control wheel steering detent level is increased while in the altitude hold mode. If force transducer output reaches this higher detent level, the altitude hold mode is disengaged. The increased detent level prevents unintentional disengagement of the mode.

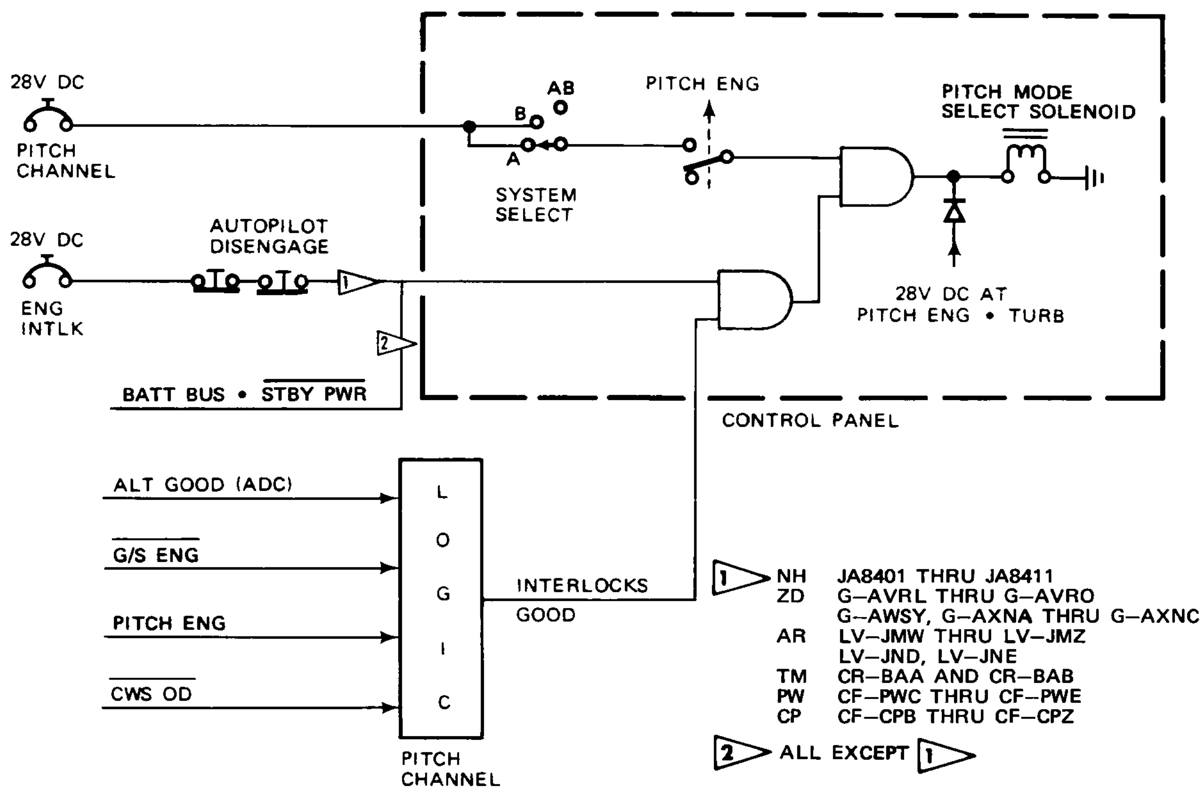
(8) Turbulence Penetration Mode

(a) The turbulence penetration mode is selected at the Control Panel (pitch mode select switch in TURB position). In this mode, pitch channel and pitch control panel logic is bypassed and 28-volt dc power is fed directly to the pitch mode select switch solenoid (Fig. 21). Selection results in reduction in amplification of the pitch rate and pitch attitude gains because of the effect of easy-on gain reduction circuit.

(b) PES-3 on the variable gain amplifier and application of 28 h1 volts dc to clamp the pitch computer (Fig. 15). These changes effectively provide attitude hold mode with softened commands for pitch control movements. Glide slope mode is disabled during use of turbulence penetration mode, but control wheel steering functions normally.

(9) Automatic Approach Mode

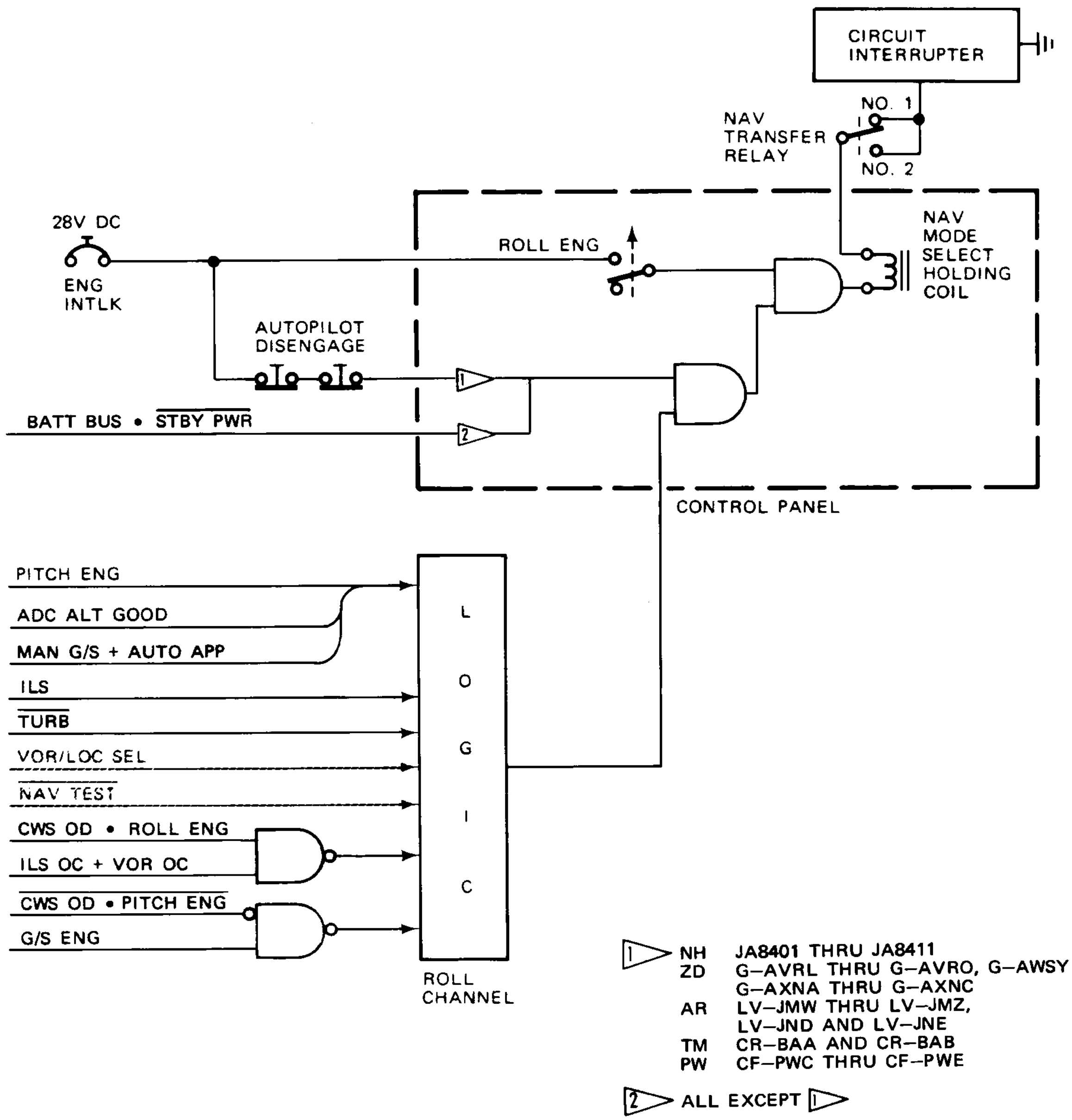
(a) In automatic approach mode (NAV mode select switch in AUTO APP position), the pitch channel commands aircraft pitch attitude to follow a glide slope radio beam. This mode is initiated at the control panel (Fig. 22). To engage the NAV mode select switch in the AUTO APP position, 28 volts dc from the pitch engage circuits must be available, 28 volts dc from the air data computer altitude flag monitor, a pitch control wheel steering not out of detent and a roll engage signal must be available, 28 volts dc from the ILS receiver must be available and the pitch mode select switch must not be in TURB position. The G/S ENG (ILS OC) input state has no effect on the initial AUTO APP mode engage interlock requirements. However, once the vertical beam sensor has changed state (G/S ENG), this circuit is responsible for causing the NAV mode select switch to return to MAN position when pitch control wheel steering is moved out of detent. The pitch CWS OD input is combined with ILS OC input, summed at an effective AND gate, and inverted until a logic 0 is applied to one input of the first AND gate in the roll control panel, thereby causing the NAV selector switch to return to MAN. Control wheel steering detent level is increased during glide slope mode to prevent unintentional disengagement of the mode. The glide slope automatic mode is executed in three phases: intercept, capture, and track.



Pitch Mode Select Logic Circuit
Figure 21

(b) In the intercept phase, the glide slope receiver signal is applied to a modulator amplifier (Fig. 18). The approach progress display GLIDE SLOPE annunciator is illuminated amber. This indicates that the vertical beam sensor is armed. The VBS, connected to the output of modulator amplifier, senses glide slope level. When this output signal level decreases to a predetermined value, the VBS changes state. This action terminates the intercept phase and is termed glide slope engaged. At this point the approach progress display GLIDE SLOPE annunciator illuminates green.

(c) The capture phase begins when the vertical beam sensor changes state to provide an interlock voltage output and starts a 10-second timer. The vertical beam sensor changes state when the glide slope receiver signal decreases to a predetermined level. During the capture phase, an ac glide slope capture bias signal is applied to summing point 4 past de-energized switch PS-14 to command a predetermined rate of descent. An altitude rate signal from the air data computer is also applied to summing point 4 so that output from summing point 4 is proportional to deviation from the commanded rate of descent. With switch PS-4 open, the output from summing point 4 is routed through the electrical bus synchronizer to summing point 2. Output from summing point 2 causes movement in the pitch computer servo assembly in which the gear changer has selected the numerically higher gear ratio. The resultant output from the CT causes the valve amplifier to command elevator movement. Ten seconds after the vertical beam sensor changes state, switches PS-4 closes and PS-5 opens. This change terminates the capture phase.



NAV/Mode Select Holding Coil Logic Circuit (Auto App-Man G/S)
Figure 22 (Sheet 1)

(d) The tracking phase of the automatic glide slope control mode begins when the 10-second timer, started by VBS state change, causes switches PS-4 and PS-5 to change state. This change removes altitude rate deviation signal (derived at summing point 4) from summing point 2, and applies this signal, after additional filtering to summing point 8. The glide slope deviation signal is applied to the gain programmer as well as to the vertical beam sensor. This deviation signal does not pass through the gain programmer until the 10-second timer turns on easy-on switch PES-1 in the programmer control circuit. The easy-on switch causes the gain programmer to gradually allow passage of the glide slope signal to summing point 8 and, past open switch PS-6 to summing point 2. The altitude rate signal is combined with the glide slope deviation signal at summing point 8 to provide damping for the glide slope mode. Output from summing point 8 is routed through the vertical path filter to summing point 6. The vertical path filter limits commanded pitch attitude changes while in the glide slope mode. Output from summing point 2 provides rotation of the pitch computer in which the gear changer has selected the numerically higher gear ratio to form an electromechanical integrator. Movement in this integrator results in output from the CT which is applied to summing point 6. Summing point 6 output causes the valve amplifier to command elevator movement necessary to maintain flight path in the glide slope beam center. Integrator operation results in high long-term gain used to eliminate beam standoff errors. Use of control wheel steering after termination of the intercept phase, disengages the glide slope mode. As the airplane descends along the glide slope beam, a radio altimeter (when installed) signal is routed to the altitude programmer. If the radio altimeter signal is valid, it is routed to the programmer control. A switch within the programmer control changes state when signal level corresponds to a 1500-foot altitude. This change of state grounds an interlock voltage and routes the radio altimeter signal through the programmer control to the gain programmer, through which the glide slope signal is passing. Amplification of the glide slope signal in the gain programmer is reduced as a function of radio altimeter signal. This gain reduction compensates for the increase in glide slope beam intensity as the transmitter is approached. If the radio altimeter fails or is not installed, amplification of glide slope deviation signal in the gain programmer is then reduced as a function of time, starting at the beginning of the capture phase.

(10) Manual Glide Slope Mode

(a) Glide slope manual mode operation is identical to automatic glide slope operation, except that the intercept phase is bypassed. Glide slope manual mode is selected at the control panel (NAV mode select switch in MAN G/S position), and the capture phase begins immediately. Selection of glide slope manual mode simulates change of state in the VBS. Subsequent events are explained under automatic approach mode.

7. Approach Progress Display Operation

A. The approach progress display provides annunciator light (VOR/LOC and GLIDE SLOPE) indication when a navigation mode is selected (VBS or LBS armed) on the autopilot control panel and when the vertical or lateral beam sensors have operated to capture a VOR radial, LOC beam or glide slope beam. The displays on each pilot's instrument panel are paralleled together to the pitch and roll channel logic circuits. Two annunciator lights are used on the display; amber for VBS or LBS armed and green for VBS or LBS captured. Other annunciator lights on the display are spares or are associated with the flight director system. (Refer to Flight Director System, Chapter 34.)

B. The approach progress display lights can be tested by pressing the display; power from the autopilot warning lights circuit breaker is then completed through all lights, thereby indicating if all lights are operable (Fig. 22). The ground side of the annunciator lights is connected to a dimming module in the flight instrument accessory box. A photocell on each pilot's instrument panel senses background light conditions and, through the dimming circuit, keeps the annunciator lights at the same relative brightness. The master dimming switch on the center instrument panel also controls the annunciator lights. When the switch is positioned to BRT, the dimming circuit is bypassed and the annunciator lights remain at maximum intensity.

C. The VOR/LOC annunciator is controlled by logic in the roll channel. The VOR/LOC light illuminates amber whenever the NAV selector switch is in any position except MAN or MAN G/S. If the VOR/LOC super flag logic is available, the VOR/LOC annunciator light turns green whenever the LBS drops out (VOR/LOC capture). The GLIDE SLOPE annunciator is controlled by the pitch channel. Selection of AUTO APP or MAN G/S completes 28 volts dc to illuminate the glide slope annunciator amber. Power for the glide slope annunciator lights is completed through the glide slope relay in the autopilot accessory box. If the glide slope super flag logic is available, the glide slope relay is energized and the glide slope annunciator turns green whenever the VBS drops out (glide slope capture). Once the annunciator has turned green, loss of LBS/VBS or superflag logic does not return the light to amber or off. MAN must be selected to extinguish the light.

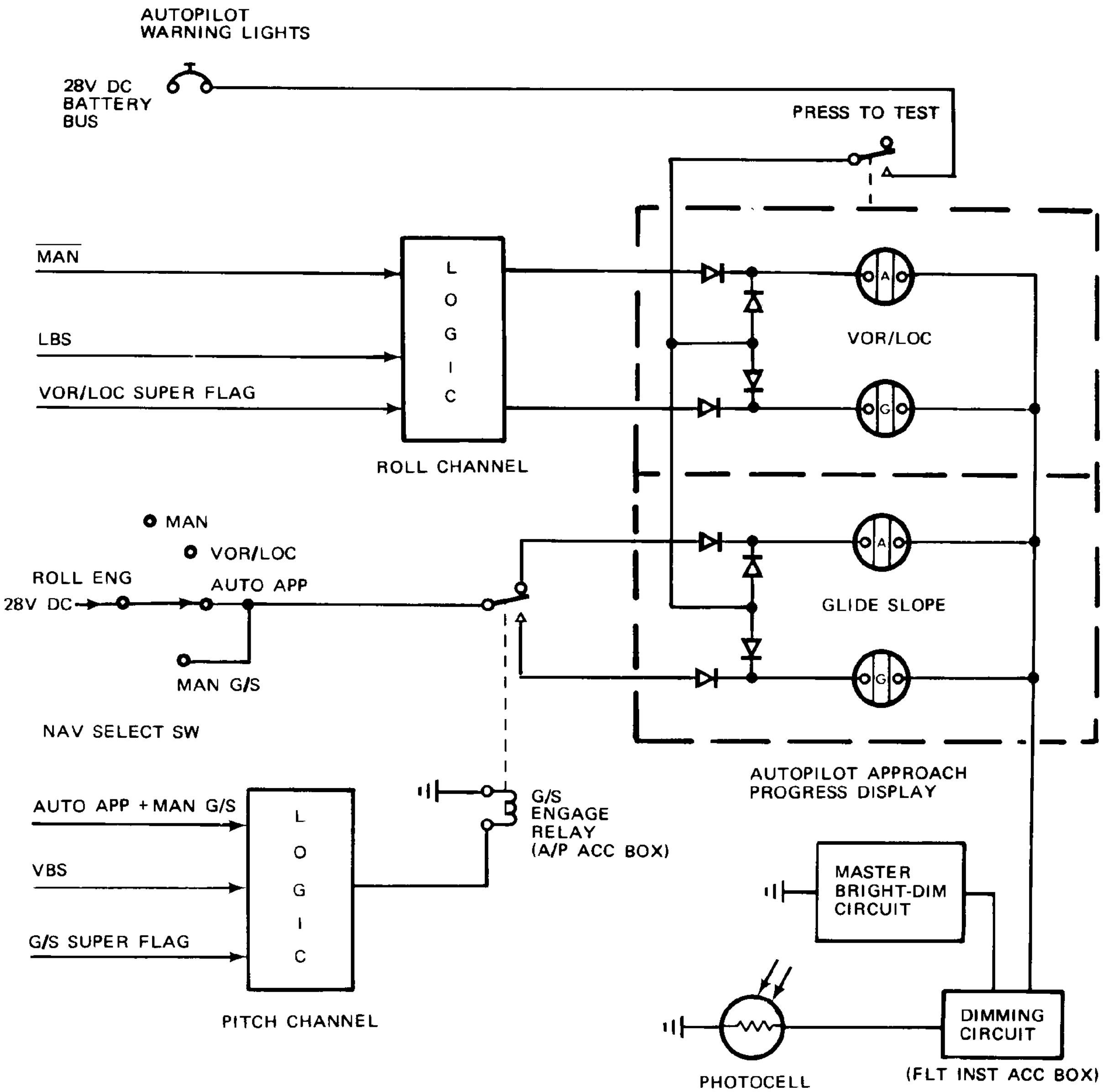
8. Autopilot Mode Select Operation

A. General

(1) The various autopilot modes are selected on the autopilot control panel. Some modes require only the pitch or roll channels to be engaged, others require both channels to be engaged (Fig. 23). The logic required to keep a selected mode engaged has been previously covered. The following discussion covers typical uses of the different switches and when they can be used in relation to each other. It is assumed all logic inputs are available for switch engagement. Figure 21 provides a summation of the different switch positions, where the primary signals originate and the associated roll rates and bank limits. The charts may be useful when studying the roll and pitch channel block diagrams and the operation of the autopilot system.

B. NAV Mode Select Switch

(1) The basic autopilot mode is MAN. The NAV mode select switch must be in MAN to engage either the pitch or the roll channels. The airplane may be banked or pitched at any angle, but as long as control wheel steering is in detent, the pitch and/or roll channels can be engaged. The heading mode select switch must be centered and the pitch mode select switch off. The pitch channel maintains the airplane at the same attitude existing at the time ELEV switch is engaged. The roll channel maintains the airplane at the same attitude existing at the time the AIL switch is engaged if the bank angle is greater than 5 degrees. If the bank angle is less than 5 degrees, the wings roll level and, after 1 to 2 seconds, the airplane flies on established heading. If the roll channel is engaged at a bank angle greater than 32 degrees, the airplane rolls upright to a 32 degree bank angle and then maintains the established attitude (attitude hold). Low detent is used by control wheel steering when MAN is selected, provided the pitch mode select switch is not in ALT HOLD and the heading mode switch is not in HDG SEL. Control wheel steering enables the pilot to manually control the airplane through the autopilot pitch and roll channels. The control wheel is moved and the airplane flown just as it would be without the autopilot system engaged. The airplane will remain at the attitude or bank angle as established by the pilot (and modified by the mode limitations) whenever the control wheel or column is returned to center (in detent). The detent levels are established to ensure positive control inputs by the pilot and to ensure against any accidental control inputs which may occur.



NAV/Mode Select Holding Coil Logic Circuit (Auto App-Man G/S)
Figure 22 (Sheet 2)

(2) The VOR/LOC, AUTO APP and MAN G/S positions of the NAV mode select switch are the radio modes. The VOR/LOC position requires only the roll channel to be engaged; however, the pitch channel can also be engaged. The AUTO APP and MAN G/S positions require both pitch and roll channels to be engaged. Selection of VOR/LOC allows the airplane to capture a VOR radial on localizer beam. Course error and radio deviation then cause the autopilot to fly the airplane onto the selected course indicated on the course deviation indicator.

(3) Control wheel steering can be used to aid putting the airplane on-course. The roll control wheel steering goes to high detent when the airplane is on-course. Use of roll control wheel steering with the airplane on-course trips the NAV mode select switch back to the MAN position. Operation of the autopilot is similar for the AUTO APP and MAN G/S positions except if either the pitch or roll control wheels are moved out of high detent when roll on-course or glide slope capture, the switch trips back to MAN. Selection of MAN G/S manually operates the vertical beam sensor to establish glide slope capture modes. MAN G/S is normally selected if flying within 1/2 dot above the glide slope beam and manual capture is desired. Operation of the pitch and roll channels is then identical to that used in AUTO APP. If a localizer frequency is selected, selection of TURB trips the NAV mode switch back to MAN.

C. Heading Mode Select Switch

(1) The heading mode select switch is normally centered. When centered, the autopilot controls the airplane in either attitude hold or heading hold (if MAN is selected). The heading switch is always centered if the autopilot has captured a VOR radial, localizer beam or glide slope beam.

(2) The HDG SEL position can be selected provided the roll channel is engaged and the LBS or VBS has not dropped out (capture), regardless of the position of the NAV mode select switch. The autopilot flies the airplane to whatever heading is selected on the course deviation indicator. Control wheel steering cannot be used in HDG SEL without tripping the HDG SEL switch back to center.

(3) Selecting HDG OFF locks out the automatic wings leveling feature of the roll channel when the bank angle is less than 5 degrees, thereby allowing the airplane to be maneuvered with control wheel steering and allowing the airplane to fly at any attitude. The HDG OFF position can only be selected if the NAV mode select switch is in MAN and the roll channel is engaged. The pitch channel does not have to be engaged for either HDG SEL on HDG OFF; however, it can be engaged.

MODE SELECT AND PURPOSE	HEADING SWITCH	PITCH MODE SELECTOR	AXES CONTROLLED	CONTROL SIGNAL SOURCE	CWS DETENT LEVEL	MAXIMUM ANGLE DEGREES	MAXIMUM ANGLE METHOD	MAX RATE 1°/SEC	MODE TRIP
MAN 1. PITCH AND ROLL ENGAGE IF CWS NOT OUT OF DETENT. 2. MANEUVER IN PITCH AND ROLL. NO RADIO INPUTS.	CENTERED	OFF	PITCH	CWS AND VG	LOW	+27/-10	MECH	10	PITCH OR ROLL CHANNEL DISENGAGED WITH AIL OR ELEV SWITCH OR LOGIC INTERLOCK FAILURE
			ROLL	CWS	LOW	32	MECH	13	
				BA<5° DG,VG		25	LIMITER	7	
				BA>5° VG					
		ALT HOLD	PITCH	VG AND ADC	HIGH	+27/-10	MECH	1	PITCH MODE REVERTS TO OFF IF PITCH CWS EXCEEDS HIGH DETENT OR ALT ADC FAILS.
			ROLL	SAME AS OFF, BA< 5°					
		TURB	PITCH	SAME AS OFF EXCEPT SIGNALS DAMPED 50 %					PITCH MODE MUST BE MANUALLY POSITIONED TO OFF
			ROLL	CWS	LOW	32	MECH	13	
				DG AND VG		8	LIMITER	7	
	HDG OFF (MAN MODE ONLY)	OFF	PITCH	CWS AND VG	LOW	+27/-10	MECH	10	HEADING SWITCH REVERTS TO CENTER IF MODE SELECT CHANGED.
			ROLL	CWS	LOW	32	MECH	13	
				VG		25	MECH	7	
		ALT HOLD AND TURB	SAME AS CENTERED						
	HDG SEL	OFF	PITCH	CWS AND VG	LOW	+27/-10	MECH	10	HEADING SWITCH REVERTS TO CENTER IF ROLL CWS EXCEEDS HIGH DETENT
			ROLL	CDI-HE AND VG	HIGH	25	LIMITER	4	
		ALT HOLD	PITCH	VG AND ADC	HIGH	+27/-10	MECH	1	PITCH TO OFF AND HEADING SWITCH CENTERED IF CWS EXCEEDS HIGH DETENT OR ALT ADC FAILS.
			ROLL	SAME AS OFF					
		TURB	PITCH	SAME AS OFF EXCEPT SIGNALS DAMPED 50%					HEADING SWITCH REVERTS TO CENTER IF ROLL CWS EXCEEDS HIGH DETENT
			ROLL	CDI-HE AND VG	HIGH	8	LIMITER	4	

Autopilot Mode Select Operational Chart
Figure 23 (Sheet 1)

MODE SELECT AND PURPOSE	HEADING SWITCH	PITCH MODE SELECTOR	AXES CONTROLLED	CONTROL SIGNAL SOURCE		CWS DETENT LEVEL	MAXIMUM ANGLE DEGREES	MAXIMUM ANGLE METHOD	MAX RATE 1°/SEC	MODE TRIP
VOR/LOC	CENTERED	OFF	PITCH	CWS AND VG		LOW	+27/-10	MECH	10	MODE SELECT RETURNS TO MAN IF ROLL CWS EXCEEDS HIGH DETENT WHEN ON COURSE
1. INTERCEPT, CAPTURE AND TRACK VOR WITH ROLL CHANNEL			ROLL (SAME AS MAN UNTIL VOR OR LOC CAPTURE)	$\overline{VOROC}$ + $\overline{LOCOC}$	CWS	LOW	32	MECH	13	
					CDI-CE NAV, VG		25	LIMITER	4 (VOR) 7 (LOC)	
2. INTERCEPT, CAPTURE AND TRACK LOC WITH ROLL CHANNEL				VOROC	CDI-CE NAV VG	HIGH	8	LIMITER	1.3	
				LOCOC	CDI-CE (OC) NAV, VG DG (APP OC)	HIGH	8	LIMITER	7	
NOTE: VOR OR LOC FREQUENCY MUST BE SELECTED OR SWITCH WILL NOT ENGAGE. TURB WILL PREVENT ENGAGEMENT IF LOC FREQUENCY SELECTED		ALT HOLD	PITCH	VG AND ADC		HIGH	+27/-10	MECH	1	PITCH MODE REVERTS TO OFF IF PITCH CWS EXCEEDS HIGH DETENT OR ALT ADC FAILS
			ROLL	SAME AS OFF						
		TURB	PITCH	SAME AS OFF EXCEPT SIGNALS DAMPED 50%						MODE SELECT REVERTS TO MAN IF TURB SELECTED WHEN ON LOC OR IF ROLL CWS EXCEEDS HIGH DETENT
			ROLL	$\overline{VOROC}$	CWS	LOW	32	MECH	13	
					CDI-CE NAV, VG		8	LIMITER	4	
				VOROC	CDI-CE NAV, VG	HIGH	8	LIMITER	1.3	
	HDG OFF	CAN NOT BE ENGAGED								
	HDG SEL	OFF	PITCH	CWS AND VG		LOW	+27/-10	MECH	10	HEADING SWITCH REVERTS TO CENTER ON LBS ENGAGEMENT OR ROLL CWS EXCEEDS HIGH DETENT
			ROLL	$\overline{VORLBS}$ + $\overline{LOC LBS}$	CDI-HE VG	HIGH	25	LIMITER	4	
		ALT HOLD	PITCH	VG AND ADC		HIGH	+27/-10	MECH	1	PITCH MODE REVERTS TO OFF IF CWS EXCEEDS HIGH DETENT OR ALT ADC FAILS
			ROLL	SAME AS OFF						
		TURB	PITCH	SAME AS OFF EXCEPT SIGNALS DAMPED 50%						HEADING SWITCH REVERTS TO CENTER ON LBS ENGAGEMENT OR ROLL CWS EXCEEDS HIGH DETENT
			ROLL	$\overline{VORLBS}$	CDI-HE VG	HIGH	8	LIMITER	4	

Autopilot Mode Select Operational Chart
Figure 23 (Sheet 2)

MODE SELECT AND PURPOSE	HEADING SWITCH	PITCH MODE SELECTOR	AXES CONTROLLED	CONTROL SIGNAL SOURCE		CWS DETENT LEVEL	MAXIMUM ANGLE		MAX RATE 1°/SEC	MODE TRIP
							DEGREES	METHOD		
AUTO APP 1. INTERCEPT, CAPTURE AND TRACK LOC WITH ROLL CHANNEL. 2. INTERCEPT, CAPTURE AND TRACK GLIDE SLOPE WITH PITCH CHANNEL. NOTE: ILS FREQUENCY MUST BE SELECTED OR SWITCH WILL NOT ENGAGE. TURB WILL PREVENT ENGAGEMENT. BOTH ROLL AND PITCH AXES MUST BE ENGAGED	CENTERED	OFF	PITCH	$\overline{\text{VBS}}$	CWS&VG	LOW	+27/-10	MECH	10	MODE SELECT RETURNS TO MAN IF CWS EXCEEDS HIGH DETENT WHEN ON GLIDE SLOPE OR ON COURSE
				VBS	GLIDE SLOPE, RADIO ALT, PROGRAM, VG	HIGH	700 FT/MIN DESCENT RATE & 6° LIMITER, 1°/ SEC RATE			
			ROLL (SAME AS MAN UNTIL LOC CAPTURE)	$\overline{\text{LOCOC}}$	CWS	LOW	32	MECH	13	
					CDI-CE NAV, VG		25	LIMITER	7	
				LOCOC	CDI-CE (OC) NAV, VG DG (APP OC)	HIGH	8	LIMITER	7	
		ALT HOLD	PITCH	$\overline{\text{VBS}}$	VG&ADC	HIGH	+27/-10	MECH	1	PITCH MODE TRIPS TO OFF ON G/S ENGAGE
				VBS	SAME AS OFF					
			ROLL	SAME AS OFF						
		TURB	TRIPS MODE SELECT TO MAN							
	HDG OFF	CAN NOT BE ENGAGED								
	HDG SEL	OFF	PITCH	$\overline{\text{VBS}}$	CWS&VG	LOW	+27/-10	MECH	10	HEADING SWITCH REVERTS TO CENTER ON VBS ENGAGEMENT OR CWS EXCEEDS HIGH DETENT
			ROLL	$\overline{\text{LBS}}$	CDI-HE VG	HIGH	25	LIMITER	4	
		ALT HOLD AND TURB	SAME AS CENTERED							

Autopilot Mode Select Operational Chart
Figure 23 (Sheet 3)

MODE SELECT AND PURPOSE	HEADING SWITCH	PITCH MODE SELECTOR	AXES CONTROLLED	CONTROL SIGNAL SOURCE	CWS DETENT LEVEL	MAXIMUM ANGLE		MAX RATE 1°/SEC	MODE TRIP
						DEGREES	METHOD		
MAN G/S 1. INTERCEPT GLIDE SLOPE FROM ABOVE WITHIN 1/2-DOT. 2. INTERCEPT, CAPTURE, TRACK LOC WITH ROLL CHANNEL NOTE: ILS FREQUENCY MUST BE SELECTED OR SWITCH WILL NOT ENGAGE. TURB WILL PREVENT ENGAGEMENT. BOTH ROLL AND PITCH AXES MUST BE ENGAGED.	CENTERED	OFF	PITCH	PITCHDOWN FOR 10 SEC AT 700 FT/MIN RATE THEN SAME AS AUTO APP					
			ROLL	SAME AS AUTO APP					
		ALT HOLD	TRIPS TO OFF ON MAN G/S MODE						
		TURB	TRIPS MODE SELECT TO MAN						
	HDG OFF	CAN NOT BE ENGAGED							
	HDG SEL	TRIPS TO CENTERED ON MAN G/S MODE							

Autopilot Mode Select Operational Chart
Figure 23 (Sheet 4)

D. Pitch Mode Select Switch

(1) The normal position of the pitch mode select switch is OFF. The pitch channel maintains the airplane attitude at time of engagement. Pitch control wheel steering can then be used to maneuver the airplane. With the control wheel back in detent, the pitch channel keeps the airplane at the new established attitude.

(2) Positioning the pitch mode select switch to ALT HOLD requires the pitch channel to be engaged and a valid altitude good logic from the air data computer. The pitch channel keeps the airplane flying at the established altitude at the time of selecting ALT HOLD.

(3) The NAV mode select switch can be in any position. The pitch mode select switch remains in ALT HOLD provided the glide slope has not been captured. Pitch control wheel steering can not be used when in ALT HOLD without tripping the switch back to OFF. The roll channel does not have to be engaged to select ALT HOLD.

(4) Selection of TURB requires only the pitch channel to be engaged. The TURB position trips the NAV mode select switch back to MAN if VOR/LOC and a localizer frequency is selected, or if AUTO APP or MAN G/S is selected. With TURB selected, pitch channel control signals are reduced approximately 50 percent. If the roll channel is engaged, the roll command bank angle is limited to 8 degrees. Control wheel steering can be used; however, airplane response is reduced approximately 50 percent.

9. Yaw Damper BITE Operation (Fig. 24) (Coupler P/N 4030952)

A. The BITE system simultaneously checks the overall operation of each electronic unit and its interface with other components. Each series of tests can be completed in spite of individual test failures.

B. The BITE portion utilizes both analog and digital circuitry to test a high percentage of the computation and logic circuits. Manually initiated automatic go/no go testing checks dynamic circuits for gain and frequency and mode sensors for proper threshold levels. Each test is initiated with a single push button switch and evaluated by one of two colored indicators. These PASS and FAIL indicators provide the test status of each test without requiring any meter movement interpretation.

C. Prior to self-testing, a preliminary cockpit setup and a check of the BITE system are completed. Upon completion, self-testing may begin.

D. Typical BITE Control

(1) The BITE provides a check of a majority of the signal computation circuitry. This is accomplished by comparing a programmed (pre-determined and scaled) signal with a fixed reference signal. Tolerance and polarity of each signal is also checked.

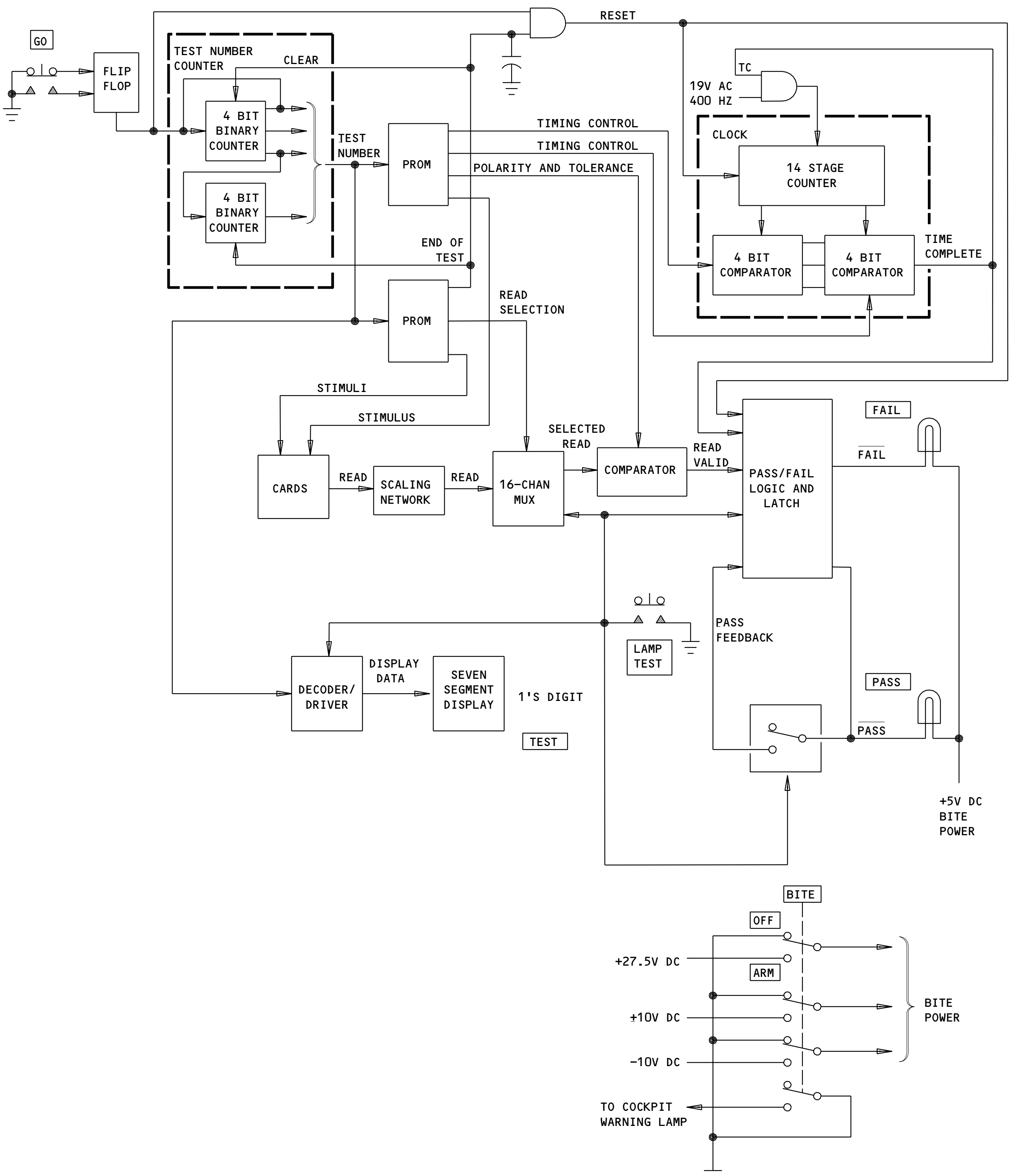
(2) The BITE controls are on the front panel of the yaw damper coupler. At the top center of the panel is the test number window which indicates the specific test being computed. Below, is a row of two colored indicators which illuminate to indicate the status of the particular test shown in the test window. The left indicator is green to indicate a PASS condition; the right indicator is red to indicate FAIL condition. Directly below the PASS light is the GO switch used to trigger the sequence of tests. Directly below the FAIL light is the LAMP TEST switch used to check the PASS-FAIL lamps as well as the seven-segment display indicators behind the test number window. At the bottom of the panel, directly below the test number window, is the BITE (OFF-ARM) switch. The ARM position supplies BITE voltages to the multiplexer (MUX), comparator, and card input stimuli, and prepares circuits within the unit for testing (initializes the unit).

E. Operation

(1) Selecting BITE-ARM applies + 10 volts dc to the MUX, comparator, and card input stimuli circuits, and 5 volts dc to the integrated circuits. With these voltages applied, a power reset and power reset removal initializes the unit during test 0. The power reset: resets the 14 stage counter to zero; time complete not logic starts the clock; the sequence counter is cleared to 0; the FAIL latch is reset to PASS; and a PASS signal is injected into the MUX channel and the comparator. The power reset removal: removes the reset from the 14 stage counter and starts a .64 second pulse; removes the clear from the sequence counter; and removes the reset from the FAIL latch. With time completed for test 0: the 14 stage counter stops and the injected comparator signal (PASS) causes the PASS output to the latch to illuminate the PASS indicator. This output is fed back to prevent a fail light at a later time. This terminates test 0.

(2) To check the PASS-FAIL lamps and the seven-segment digital display, the LAMP TEST switch is pushed and held. Pressing the switch injects a fail signal into the MUX and comparator; the pass light feed back path is opened, allowing a comparator fail signal to trip the fail latch; the FAIL indicator illuminates; and 8 appears at the test number window as long as the switch remains pressed.

(3) To select the next test, the GO switch is pressed and released. This action triggers a pulse which: resets the 14 stage counter and the FAIL latch; increments the sequence counter; and repeats power reset and power reset removal action as stated previously, except the sequence counter is not cleared and the memory outputs inject the proper stimuli for the particular test. When the GO switch is pressed after the last test, a dedicated memory output simulates a power reset back to test 0.



Typical BITE Block Diagram
Figure 24

(4) During normal testing procedures, the action of the GO switch, in conjunction with a flip-flop circuit, triggers a dual binary counter to provide consecutive test number data. This data is routed to the address inputs of two programmable read-only memories (PROM). The PROM outputs (binary form) representing the particular test number, is used to control a major portion of the BITE circuitry. The PROM outputs control test timing, provide tolerance and polarity information to the analog comparator, determines proper BITE stimuli for the cards being tested, pass appropriate read signal from the multiplexer to the comparator, and supply essential test number data to activate the front panel seven-segment digital display. A 14 stage counter and two, four-bit comparators are combined to form a clock circuit. The output of these combined circuits is one of five available test times. The selected read outputs, tolerance and polarity signals are applied to the comparator to produce a single output. This output and the selected timing signal are applied to several logic gates to determine either a pass or fail logic signal: A pass signal activates the PASS indicator; a FAIL signal activates the FAIL indicators.

(5) Yaw Damper Coupler BITE Tests (Fig. 4)

NOTE: The following test descriptions are typical and may not represent the latest test procedures due to design improvements.

(a) Unless otherwise noted, the following conditions are assumed for each test:
1) Yaw damper is engaged.
2) Hydraulics are on.

(b) Test 1 - Rate Gyro Filter 1
1) GO is pressed. A stimulus is applied to the rate gyro input at 1 which drives the valve amplifier. Rudder position is read at 6 Test time is approximately 1 second.

(c) Test 2 - Engage Null
1) GO is pressed. The stimulus at input 1 is removed and the valve amplifier output is read at 3. Test time is approximately 5 seconds.

(d) Test 3 - Rate to Rudder (Displacement)
1) GO is pressed. A stimulus is applied to input 3 at the valve amplifier which drives the rudder. Rudder position feedback is read at 4. Test time is approximately 1 second/

(e) Test 4 - Rate to Rudder (Integral)
1) GO is pressed. A stimulus is applied to input 3 at the valve amplifier which drives the rudder. Rudder position feedback is integrated and read at 1. Test time is approximately 45 seconds.
(f) Test 5 - Trim Meter
1) Go is pressed. The stimulus at input 3 is removed and the rudder position feedback signal to trim meter amplifier is read at 5. Test time is approximately 8 seconds.
(g) Test 6 - Disengage Synchronization
1) The yaw damper is disengaged, hydraulics are turned off and GO is pressed. A stimulus is applied to input 3 at the valve amplifier. The valve amplifier output is synchronized and the integrator output read at 1. Test time is approximately 5 seconds.
(h) Test 7 - Rate Gyro
1) The yaw damper is disengaged, hydraulics are turned off, and GO is pressed. The stimulus at input 3 and the rate gyro is torqued. The rate gyro demodulator output is read at 2. Test time is approximately 1 second. Torquing stimulus is removed from the rate gyro at the end of the test.
2) The BITE switch is returned to OFF at the end of the tests. Hydraulics are applied and the yaw damper coupler is engaged for normal operation.
(6) Yaw Damper Coupler (P/N 4030952-906) BITE Tests (Fig. 4)

NOTE: The following test descriptions are typical and may not represent the latest test procedures due to design improvements.

(a) Unless otherwise noted, the following conditions are assumed for each test:
1) Yaw damper is engaged.
2) Hydraulics are on.
(b) Test 1 - Rate Gyro Filter 1
1) GO is pressed. A stimulus is applied to the rate gyro input at 1 which drives the valve amplifier. Rudder position is read at 5. Test time is approximately 1 second.
(c) Test 2 - Engage Null
1) GO is pressed. The stimulus at input 1 is removed and the valve amplifier output is read at 3. Test time is approximately 5 seconds.

(d) Test 3 - Rate to Rudder (Displacement)
   1) GO is pressed. A stimulus is applied to input 3 at the valve amplifier which drives the rudder. Rudder position feed back is read at 4. Test time is approximately 1 second.
(e) Test 4 - Rate to Rudder (Integral)
   1) GO is pressed. A stimulus is applied to input 3 at the valve amplifier which drives the rudder. Rudder position feedback is integrated and read at 1. Test time is approximately 40 seconds.
(f) Test 5 - Disengage Synchronization
   1) The yaw damper is disengaged, hydraulics are turned off and GO is pressed. A stimulus is applied to input 3 at the valve amplifier. The valve amplifier output is synchronized and the integrator output read at 1. Test time is approximately 5 seconds.
(g) Test 6 - Derived Acceleration First Stage
   1) The yaw damper is disengaged, hydraulics are turned off, and GO is pressed. The stimulus at input 3 is removed and a 10v dc stimulus is applied to input 4 at the demodulation amplifier. The derived acceleration output is read at 6. Test time is approximately 1 second.
(h) Test 7 - Rate Gyro Filter 2
   1) The yaw damper is disengaged, hydraulics are turned OFF, and GO is pressed. The stimulus at input 4 is removed and a 10v dc stimulus is applied to input 5 at the airspeed gain circuit and airspeed gain is read at 7. Test time is approximately 1 second.
(i) Test 8 - Rate Gyro
   1) The yaw damper is disengaged, hydraulics are off and GO is pressed. The stimulus at input 3 is removed and the rate gyro is torqued. The rate gyro demodulator output is read at 2. Test time is approximately 1 second. Torquing stimulus is removed from the rate gyro at the end of the test.
   2) The BITE switch is returned to OFF at the end of the tests. Hydraulics are applied and the yaw damper coupler is engaged for normal operation.

AUTOPILOT SYSTEM - TROUBLESHOOTING

1. General

A. The following troubleshooting procedures are intended to help trouble shoot the yaw damper system and autopilot system down to a component level. Troubleshooting charts are based on possible failure reports which may occur and the corresponding isolation procedure to detect and replace the faulty component. The charts are listed under the system heading and are separated into engage problems or operational problems.

B. Yaw Damper System (All except coupler P/N 4084042)

NOTE: For aircraft with coupler P/N 4084042 installed (POST-SB 27A1206), refer to AMM 22-12-01.

(1) Troubleshooting the yaw damper system operational faults can be accomplished fairly quickly through use of the system test circuits. The yaw damper test switch on the center instrument panel torques the rate gyro in the yaw damper coupler. With hydraulic and electrical power on and the system engaged, a complete check of the system is made as indicated on the yaw damper indicator on the center instrument panel and observed rudder movement. If the system is faulty, the yaw damper coupler is usually at fault. The yaw damper coupler has a self-test or BITE switch on the front panel. Operation of the self-test switch or performing BITE sequence checks the majority of the signal paths in the system (Ref Autopilot and Yaw Damper System - Adjustment/Test).

(2) Proper analysis of the self-test readings, while performing a self-test, or BITE failure analysis after completion of BITE sequence, helps indicate whether the coupler is at fault or a fault is at the rudder hydraulic power pack (Ref Yaw Damper System Operation Troubleshooting Charts). It should be noted that the autopilot warning lights illuminate steadily if a self-test or BITE switch on the yaw damper coupler (and roll channel or pitch channel) is in any position other than off or if the rate gyro switch on the coupler (when installed) is operated.

CAUTION: OPERATION OF THE YAW DAMPER TEST SWITCH AND THE RATE GYRO TEST SWITCH ON THE COUPLER MUST BE LIMITED TO 10 SECONDS OR LESS TO PREVENT OVERHEATING OF THE RATE GYRO TORQUING COILS. TEST SWITCH MAY BE OPERATED AGAIN AFTER ALLOWING RATE GYRO TO COOL FOR APPROXIMATELY 1 MINUTE.

(3) Troubleshooting yaw damper system engage faults involves checking the operation of the autopilot accessory box, autopilot switching accessory box (when installed) and yaw damper coupler. If the yaw damper system does not engage or has nuisance trips, the fault will probably be in one of the accessory boxes (Ref Yaw Damper System Engage Troubleshooting Charts).

(4) The solenoid valves, transfer valves and position transducers on the hydraulic power pack have proven to be very reliable. Operational problems will more than likely be traced to relays in one of the autopilot accessory boxes. Hydraulic power pack problems are usually indicated by operational faults when the yaw damper system is not engaged, such as sloppy centering of the rudder due to hydraulic wear or leaking seals (AMM Chapter 27, Flight Controls).

(5) Before troubleshooting the yaw damper system, ensure hydraulic and electrical power is applied to the airplane and the rudder operates normally without the yaw damper system engaged. The flight control systems A and B switches must be on to engage the yaw damper system. Ensure the yaw damper system, master caution, master dimming bus indicator lights, dim and test, autopilot engage interlock and warning light circuit breakers are closed before attempting to engage the yaw damper system. To prevent misleading indications, test operation of the autopilot disengage lights by pressing lenses to ensure bulbs illuminate. It should be noted that the yaw damper engage light is normally illuminated whenever the yaw damper system is not engaged. The central air data computer does not have to be energized for self-test of the yaw damper system since no interlock voltages are provided to the yaw damper systems. However, the air data computer must be installed since the yaw damper Q-potentiometer is used in the system.

(6) Voltage through the Q-potentiometer is provided by the yaw damper system. Readings taken on the yaw damper coupler self-test meter are calibrated with no airspeed inputs to the central air data computer. Removal of black boxes and system components requires normal safety and maintenance precautions such as removal of power prior to removal of components and capping of all exposed electrical connectors. Trouble shooting procedures are indicated with the system A selected on the autopilot control panel. Troubleshooting procedures with system B selected are similar. Airplanes delivered with system B only plus airplanes POST-SB 27-1074 must select system B for yaw damper operation.

CAUTION: DO NOT REMOVE YAW DAMPER COUPLER FOR AT LEAST 3 MINUTES AFTER REMOVING POWER FROM THE SYSTEM. RATE GYRO MAY BE DAMAGED IF COUPLER IS MOVED BEFORE GYRO IS ALLOWED TO SPIN DOWN.

C. Autopilot System

(1) Autopilot system troubleshooting requires a good knowledge of the pitch and roll channel operation. The pitch and roll channels operate independently in the MAN, VOR/LOC modes. However, both pitch and roll channels must be engaged to operate in the AUTO APP or MAN G/S modes. Failures which occur when operating in the MAN or VOR/LOC mode can usually be traced to the individual channels since the pitch channel does not depend upon roll channel signals except for versine. Failures which occur when operating in the AUTO APP or MAN G/S modes may be more difficult to isolate. A good failure report stating which mode and submode the autopilot system was in at the time of failure is a necessity.

(2) Determination of what submode failure and recognizing the signals and engagement requirements for operation in the submode will help isolate the failure to the pitch or roll channels quickly or help determine whether it was really a problem at all. Failures which occur due to faulty outputs from the gyros, navigation receiver, air data computer or radio altimeter (if installed) will usually show up as flag indications on the flight director indicator, course deviation indicator or on the black box due to self-monitoring circuits and associated flag voltages within the components (AMM Chapter 34, Navigation). Troubleshooting procedures given assume the signals from associated systems are valid and analyze only the autopilot system failures (Ref Pitch and Roll Channel Operation Troubleshooting Charts).

(3) The pitch and roll channels have self-test switches on the front panels of their boxes. Each self-test switch is capable of testing the majority of the signal paths within the box. If a signal path malfunction exists within the box, it will usually become apparent by completion of the self-test procedures (Ref Autopilot and Yaw Damper System - Adjustment/Test). The self-test procedures for the pitch and roll channels do not test any system components outside of the black box. Failures which occur outside of the black boxes are usually apparent when operating in the manual mode. It should be noted that the autopilot warning lights illuminate steadily if a self-test switch on the roll or pitch channel (or yaw damper coupler) is left in any position other than OFF or if the rate gyro on a channel is operated.

<u>CAUTION</u>: OPERATION OF THE RATE GYRO TEST SWITCH MUST BE LIMITED TO 10 SECONDS OR LESS TO PREVENT OVERHEATING OF THE TORQUING COILS. TEST SWITCH MAY BE OPERATED AGAIN AFTER ALLOWING RATE GYRO TO COOL FOR APPROXIMATELY 1 MINUTE.

(4) Test points on the front of the boxes may be useful when trouble shooting the system. Expected voltages for some of the test points are given in the adjustment/test procedures as part of system test. The following list indicates each test point and where it is connected on the circuit. Refer to vendor overhaul manual for specific signal for the applicable test point:

CAUTION: TEST POINT CIRCUITS ARE NOT PROTECTED AGAINST INADVERTENT SHORTS.

PITCH CHANNEL	
Test Points	Description
TP-1	DC ground
TP-2	Pitch 28-volt dc circuit breaker
TP-3	Stab trim down signal
TP-4	Stab trim up signal
TP-5	Gain programmer output
TP-6	Pitch integrator clamp
TP-7	Engage interlock (flaps and stab trim servo)
TP-8	Rate gyro filter output
TP-9	Vertical path filter output
TP-10	Elevator position
TP-11	Rate gyro filter input
TP-12	Pitch control transformer output
TP-13	CWS Q-pot out
TP-14	Output to equalizer
TP-15	Neutral shift position
TP-16	Derived rate input *[1]
TP-17	Derived rate Compensated output *[1]
TP-18	Derived rate output *[1]
TP-19 thru TP-21	Spare

ROLL CHANNEL	
Test Points	Description
TP-1	DC ground
TP-2	Roll 28-volt dc circuit breaker
TP-3	Aileron position feedback
TP-4	Aileron position feedback
TP-5	Gain programmer output
TP-6	Intercept limiter output
TP-7	Engage interlock (ADC airspeed monitor)
TP-8	Spare
TP-9	Erection cutoff
TP-10	Lagged roll integrator output
TP-11	Forward path integrator output
TP-12	Forward path integrator reset
TP-13	Heading beam integrator output
TP-14	Roll control transformer clamp
TP-15	Roll control transformer output
TP-16	Heading Q-pot output
TP-17	Roll rate Q-pot output
TP-18	CWS out of detent
TP-19	Derived rate input *[2]
TP-20	Derived rate compensated output *[2]
TP-21	Derived rate output *[2]

*[1] Test points used only on pitch computers 2588810-903, 904
*[2] Test points used only on roll computers 2588812-902

(5) Troubleshooting pitch or roll channel engage problems involves checking the interlock circuits for each channel to determine if the interlock voltages are available to the autopilot control panel. If interlock voltages are available, the autopilot control panel is probably faulty. However, engage circuits are also in the autopilot accessory box and pitch or roll channels. Therefore, replacement of the accessory box or a channel may be required. The engage circuit for the roll channel is also routed through a switch in the aileron force limiter. Faulty operation of the force limiter clutch may prevent engagement of the roll channel. Improper rigging or operation of the control wheel steering transducers may cause engagement difficulties even though they do not cause the pitch or roll channel to disengage once it is already engaged. It should be noted that transferring from system A to B (when A and B systems are on autopilot control panel), VG-1 to AUX VG (if installed), or DG-1 to DG-2 causes the roll channel to disengage.

(6) Transferring from system A to B or VG-1 to AUX VG causes the pitch channel to disengage (AMM Chapter 34, Navigation). Airplanes which have glide slope and localizer superflag and glide slope relays in the interlock circuits will experience disengagement of the pitch and roll channels when on glide slope and a superflag is lost. This is normal. Airplanes without the glide slope and localizer superflag interlocks do not normally disengage the pitch and roll channels when a superflag is lost (refer to Pitch and Roll Channel Engage Troubleshooting Charts). To assist in trouble shooting, following is a list of the inputs required for pitch and roll channel engagement. Loss of any one of the inputs (except as noted) causes the channel to disengage.

PITCH CHANNEL
1. Elex bus-1 energized
2. Battery bus energized (not on standby) (if interlock connected)
3. Air data computer airspeed good
4. 30 volt dc monitor good
5. Correct pitch calibrator installed
6. Control wheel steering not out of detent (engage only)
7. No trim monitor (engage only)
8. No hot short - stab trim signal (disengage only)
9. Flaps synchronized; no stabilizer trim speed change relay energized when flaps are up
10. Autopilot stabilizer trim cutout switch in NORMAL
11. Main electric stabilizer trim motor not operating
12. Vertical gyro flag out of view
13. Vertical gyro transfer relays good
14. Autopilot control panel good
15. Autopilot accessory boxes good
16. Flight control system A, B switches on
17. Control wheel autopilot disengage switches not open

ROLL CHANNEL
1. Elex bus-1 energized 2. Battery bus energized (not on standby) 3. Air data computer airspeed good 4. 30-volt dc monitor good 5. Correct roll calibrator installed 6. Control wheel steering not out of detent (engage only) 7. Aileron force limiter clutch engaged - switch closed 8. Vertical gyro flag out of view 9. Directional gyro flag out of view (if interlock connected) 10. VG and DG transfer relays good 11. Autopilot control panel good 12. Autopilot accessory boxes good 13. Flight control system A, B switches on 14. Control wheel autopilot disengage switches not open

(7) Selection of autopilot operational modes requires certain interlocks to be satisfied. During normal operation of the autopilot system, the navigation mode select switch, pitch mode select switch and heading mode select switch may be expected to return to MAN or OFF (heading switch to center) if certain interlocks fail. Failure reports involving abnormal operation of the switches require recognition of the interlocks required to keep the switch operative.

(8) Failure of any one of the interlocks may require replacement of the autopilot control panel, roll channel, pitch channel or autopilot accessory box (refer to Autopilot Mode Select Troubleshooting Chart). Following is a list of interlocks required to keep each mode select switch coil energized:

Navigation Mode Select Switch	
Position	Interlock
MAN	1. None (Spring loaded to MAN)
VOR/LOC	1. Roll engage 2. VOR/LOC selected 3. No VOR/ILS control panel test (if connected) 4. No NAV transfer (if third navigation receiver installed) NOTE: TURB selected when ILS frequency selected trips NAV mode switch to MAN. When on course, roll control wheel steering out of detent trips NAV mode switch to MAN.

AUTO APP or MAN G/S	1. Roll engage 2. Pitch engage 3. AUTO APP or MAN G/S selected 4. TURB not selected 5. ILS frequency selected 6. Air data computer altitude good 7. No VOR/ILS control panel test (if connected) 8. No NAV transfer (if third navigation receiver installed) NOTE: When on course, roll control wheel steering out of detent, trips NAV mode switch to MAN. When glide slope is captured, pitch control wheel steering out of detent trips NAV mode select switch to MAN.

Heading Mode Select Switch	
Position	Interlock
Centered	1. None (Spring loaded to center)
HDG OFF	1. Roll engaged 2. MAN selected 3. HDG OFF selected
HDG SEL	1. Roll engaged 2. HDG SEL selected 3. VOR or LOC not captured 4. Roll control wheel steering not out of detent

Pitch Mode Select Switch	
Position	Interlock
OFF	1. Spring-loaded to OFF from ALT HOLD
ALT HOLD	1. Pitch engaged 2. Air data computer altitude good 3. Pitch control wheel steering not out of detent 4. Glide slope not captured
TURB	1. Pitch engaged 2. TURB selected

(9) The solenoid valves, transfer valves and position transducers on the hydraulic power packs have proven to be very reliable. Operational problems will more than likely be traced to relays in one of the autopilot accessory boxes. Isolation of faulty components can be quickly done by comparing autopilot operation between systems A and B on airplanes with A and B systems selection on autopilot control panel. A faulty solenoid valve circuit is indicated if the control wheel manually moves easily with the associated channel engaged.

(10) The control wheel/column is normally more difficult to move when the autopilot system is engaged due to the control wheel transducers providing signal output and the autopilot system in turn driving the control wheel. If the transducer valve on position transducers fail, the control wheel and associated control surface will move to the extreme positions (Ref Hydraulic Power Pack Component Troubleshooting Charts). Hydraulic power pack problems are usually indicated by operational faults when the autopilot system is not engaged such as sloppy centering of the control surfaces due to hydraulic wear or leaking seals (AMM Chapter 27, Flight Controls).

(11) Faults involving the stabilizer trim circuits can usually be isolated between the pitch channel and the stabilizer trim servo circuits by use of the pitch channel self-test circuits. During airplane flight, the stabilizer trim circuits should cause the airplane to trim out any sustained output from the valve amplifier within approximately 12 seconds. If not, the stabilizer out of trim warning light illuminates; however, the pitch channel does not disengage.

(12) It should be noted that with the airplane on the ground and pitch channel engaged, operation of control wheel steering or any other signal input causes the stabilizer to trim to maximum up or down position unless the autopilot stabilizer trim circuit breaker is pulled. The stabilizer trim warning light also illuminates. It should also be noted that operation of the main electric stabilizer trim system causes the pitch channel to disengage (AMM Chapter 27, Horizontal Stabilizer Trim Control System). Stabilizer trim problems outside of the pitch channel must be traced down on a wire to wire basis until the fault is traced to a switch or stabilizer trim servo (Ref Stabilizer Trim Troubleshooting Chart).

(13) Troubleshooting the autopilot annunciator lights on the captain's or first officer's approach progress display is relatively easy since circuitry which controls the lights is primarily in the pitch and roll channels. A glide slope engage relay in the autopilot accessory box is controlled by the pitch channel vertical beam sensor circuits. Failure of both amber and green glide slope annunciator lights indicates lack of 28 volts dc from the autopilot control panel.

(14) Failure of only the green light indicates the vertical beam sensor, glide slope superflag circuits or the glide slope relay is inoperative. The VOR/LOC annunciator lights are controlled by the lateral beam sensor and navigation receiver superflag circuits. Replacing the roll channel usually corrects VOR/LOC annunciator faults. Both annunciators are also controlled by an automatic dimming circuit in the flight instrument accessory box and by photo cells on each pilot's instrument panel. Failure of all amber and green VOR/LOC and glide slope annunciator lights indicates a faulty dimming circuit in the flight instrument accessory box (Ref Autopilot Annunciator Lights Troubleshooting Chart).

(15) Autopilot disengage lights are controlled by sensing circuits in the autopilot control panel and the self-test switches on the yaw damper coupler, pitch channel and roll channel. Flashing of the lights is controlled by a flasher module in the autopilot accessory box. Improper operation of the lights will therefore involve one of the three control sources (Ref Autopilot Disengage Lights Troubleshooting Chart).

(16) Before troubleshooting the autopilot system, ensure hydraulic and electrical power is applied to the airplane and the control surfaces operate normally without the pitch or roll channels engaged. Ensure the master caution, master dimming bus indicator lights, dim and test, autopilot warning light, pitch and rol1 channel circuit breakers are closed. The autopilot stabilizer trim circuit breakers should be open. Ensure the navigation receivers, air data computer and low range radio altimeter (if installed) are operative; the compass system and attitude reference systems must also be operative (AMM Chapter 34, Navigation). To prevent misleading indications, test operation of all system lights by pressing lenses to assure bulbs illuminate.

(17) The autopilot stabilizer trim warning light can be ignored when the pitch channel is engaged. Readings taken on the pitch and roll channel self-test switches are calibrated with no airspeed inputs to the central air data computer. Removal of black boxes and system components requires normal safety and maintenance precautions such as removal of power prior to removal of components and capping of all exposed electrical connectors. Trouble shooting procedures are indicated with the system A selected on the autopilot control panel. Troubleshooting procedures with system B are similar.

CAUTION: DO NOT REMOVE PITCH OR ROLL CHANNELS FOR AT LEAST 3 MINUTES AFTER REMOVING POWER FROM THE SYSTEM. RATE GYRO MAY BE DAMAGED IF COUPLER IS MOVED BEFORE GYRO IS ALLOWED TO SPIN DOWN.

2. Yaw Damper Components (All except coupler P/N 4084042)

NOTE: For aircraft with coupler P/N 4084042 installed (POST-SB 27A1206), AMM 22-12-01.

A. Yaw Damper System Engage Troubleshooting Chart (Effective for airplanes with systems A and B selection on autopilot control panel)

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Yaw damper system does not engage with system A selected	Coupler	On coupler P/N 2588880 check meter on front of coupler. If meter does not read in red, coupler is faulty. On coupler P/N 4030952 press lamp test switch on front of coupler. If PASS/FAIL lamps do not light, coupler is faulty	Replace yaw damper coupler
	Autopilot switching accessory box	Select system B with system select switch and attempt to engage yaw damper system. If system engages, autopilot switching accessory box is faulty. If system does not engage, autopilot accessory box is faulty	Replace autopilot switching accessory box. Replace autopilot accessory box
Yaw damper system disengages when transfer bus is switched to different generator	Autopilot accessory box	Engage yaw damper system and verify faulty by switching generator supply to 115V ac transfer bus. If yaw damper disengages, autopilot accessory box is faulty	Replace autopilot accessory box

		NOTE: Yaw damper disengage light illuminates for 2 seconds when power is switched.	

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Yaw damper system disengages intermittently	Autopilot accessory box	Engage and manually disengage yaw damper system several times, allowing approximately 30 seconds for each engagement. If system does not remain engaged during 30-second interval, autopilot accessory box is faulty	Replace autopilot accessory box
Rudder moves more than 1/2 degree when system is engaged or disengaged	Coupler or autopilot switching accessory box	Select system B and engage yaw damper system. If rudder deflection is excessive, coupler is probably faulty. If system is normal, reselect A system and engage yaw damper system. If rudder deflection is excessive or problem is intermittent, replace coupler. If replacing coupler does not correct fault, autopilot switching accessory box is probably faulty	Replace coupler. Replace autopilot switching accessory box

B. Yaw Damper System Operation Trouble Shooting Chart
(Effective for airplanes with A and B system selection on autopilot control panel)

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Yaw damper system fails cockpit test. Yaw damper indicator bar and rudder remain centered	Coupler or solenoid valve	Select system B and engage yaw damper system. Conduct yaw damper system cockpit test. If system fails to test satisfactorily, coupler is faulty. If system tests OK, system A solenoid valve is faulty	Replace coupler. Replace solenoid valve (Ref 22-11-61, Rudder Solenoid Valve)

Autopilot and Yaw Damper System - Troubleshooting
Figure 101 (Sheet 1)

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Yaw damper indicator bar and rudder locked in left or right position, yaw damper system engaged	Coupler or transfer valve	Select system B and engage yaw damper. If rudder remains centered, select system A and conduct self-test of coupler. If coupler P/N 2588880 fails self-test 1,2,3 or 4, coupler is faulty. If coupler fails self-test 5, coupler may be faulty. If replacing coupler does not correct fault, system A transfer valve is faulty. If coupler P/N 4030952-902 fails BITE 1,3,4, and 5, or coupler P/N 4030952-906 fails BITE 1,3, and 4, one of rudder PCU components is faulty. If other BITE sequence failed, coupler is faulty	Replace coupler. Replace transfer valve (Ref 22-11-71, Rudder Transfer Valve)
Rudder locked in left or right position, yaw damper indicator bar centered, yaw damper system engaged	Coupler or position transducer	Select system B and engage yaw damper. If rudder does not remain centered, coupler is faulty. If rudder remains centered, select system A and conduct self-test. If coupler P/N 2588880 fails self-test 1,2 3 or 4, coupler is faulty. If coupler fails self-test 5, system A position transducer is faulty. If coupler P/N 4030952-902 fails BITE 1,3,4, and 5, or coupler P/N 4030952-906 fails BITE 1,3, and 4, one of rudder PCU components is faulty. If other BITE sequence failed, coupler is faulty	Replace coupler. Replace position transducer

Autopilot and Yaw Damper System - Troubleshooting
Figure 101 (Sheet 2)

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Yaw damper indicator bar does not move when system tested - rudder moves OK	Indicator	Engage yaw damper system and conduct cockpit test. Observe that rudder moves but indicator bar does not move. Indicator is probably faulty	Replace yaw damper indicator
Rudder oscillates or airplane dutch rolls. Yaw damper engaged	Coupler	Engage yaw damper and conduct cockpit test. If rudder does not center when switch movement stops, conduct coupler self-test. If coupler fails self-test, coupler is faulty. If coupler OK, hydraulic power pack is probably faulty. (Refer to Chapter 27, Rudder and Rudder Trim Control System.)	Replace coupler
System operates intermittently on A or B system select	Autopilot switching accessory box or autopilot control panel	Select system A and engage yaw damper system. Test system and coupler. Disengage and repeat for system B. If coupler tests OK but system operation is intermittent, switching accessory box may be faulty. If replacing autopilot switching accessory box does not correct fault, autopilot accessory box may be faulty. If replacing accessory box does not correct fault, autopilot control panel is probably faulty	Replace autopilot switcing accessory box. Replace autopilot accessory box. Replace autopilot control panel

Autopilot and Yaw Damper System - Troubleshooting
Figure 101 (Sheet 3)

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Experiencing high rudder pedal forces. Rudder pedal kick back or yaw damper backdrives rudder pedals during flight. Erratic yaw damper operation or rudder oscillates with yaw damper engaged. Erratic rudder pedal steering on ground with yaw damper engaged.	Binding within the standby actuator	Shutoff all hydraulics. Disconnect input control rod from the standby input lever. Measure the force required to move the lever throughout its range of motion. This force, when applied tangentially (90°) to the input crank arm and at approximately the bolt/clevis centerline, shall not exceed one pound.	Replace the standby actuator.

Autopilot and Yaw Damper System - Troubleshooting
Figure 101 (Sheet 4)

C. Yaw Damper System Engage Trouble Shooting Chart (Effective for airplanes with B system only in autopilot control panel)

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Yaw damper system does not engage	Coupler	On coupler P/N 2588880, check meter on front of yaw damper coupler. If meter does not read in red, coupler is faulty. On coupler P/N 4030952, press lamp test switch on front of coupler. If PASS FAIL lamps do not light, coupler is faulty	Replace yaw damper coupler
	Autopilot accessory box *[1]	If system does not engage, autopilot accessory box is faulty *[1]	Replace autopilot accessory box *[1]
Yaw damper system disengages when transfer bus is switched to different generator	Autopilot accessory box	Engage yaw damper system and verify fault by switching generator supply to 115-volt ac transfer bus. If yaw damper disengages, autopilot accessory box is faulty <u>NOTE</u>: Yaw damper disengage light illuminates for 2 seconds when power is switched.	Replace autopilot accessory box
Yaw damper system disengages intermittently	Autopilot accessory box	Engage and manually disengage yaw damper system several times, allowing approximately 30 seconds for each engagement. If system does not remain engaged during 30-second interval, autopilot accessory box is faulty	Replace autopilot accessory box

*[1] On airplanes incorporating SB 27-1074 autopilot switching accessory box can cause this problem and should be replaced if autopilot accessory box does not remedy the problem.

Autopilot and Yaw Damper System - Troubleshooting
Figure 101 (Sheet 5)

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Rudder moves more than 1/2 degree when system is engaged or dis- engaged	Coupler		Replace coupler

D. Yaw Damper System Operation Trouble Shooting Chart (Effective for airplanes with B system only on autopilot control panel)

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Yaw damper system fails cockpit test. Yaw damper indicator bar and rudder remain centered	Coupler or solenoid valve	Conduct self-test of coupler. If coupler P/N 2588880 fails self-test 1, 2,3 or 4, coupler is faulty. If coupler fails self-test 5, coupler may be faulty. If replacing coupler does not correct fault, system solenoid valve is faulty. If coupler P/N 4030952-902 fails BITE 1,3,4, and 5, or coupler P/N 4030952-906 fails BITE 1,3, and 4, one of rudder PCU components is faulty. If other BITE sequence failed, coupler is faulty	Replace coupler. Replace solenoid valve (Ref 22-11-61, Rudder Solenoid Valve)
Yaw damper indi- cator bar and rudder locked in left or right position, yaw damper system engaged	Coupler or transfer valve	Conduct self-test of coupler. If coupler P/N 2588880 fails self-test 1, 2,3 or 4, coupler is faulty. If coupler fails self-test 5, coupler may be faulty. If replacing coupler does not correct fault, system transfer valve is faulty. If coupler P/N 4030952-902 fails BITE 1,3,4, and 5, or coupler P/N 4030952-906 fails BITE 1,3, and 4, one of rudder PCU components is faulty. If other BITE	Replace coupler. Replace transfer valve (Ref 22-11-71, Rudder Transfer Valve)

Autopilot and Yaw Damper System - Troubleshooting
Figure 101 (Sheet 6)

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Rudder locked in left or right position, yaw damper indicator bar centered, yaw damper system engaged	Coupler or position transducer	Conduct self-test. If coupler P/N 2588880 fails self-test 1,2,3 or 4, coupler is faulty. If coupler fails self-test 5, system position transducer is faulty. If coupler P/N 4030952-902 fails BITE 1,3, 4, and 5, or coupler P/N 4030952-906 fails BITE 1, 3, and 4, one of rudder PCU components is faulty. If other BITE sequence failed, coupler is faulty	Replace coupler. Replace position transducer
Yaw damper indicator bar does not move when system tested - rudder moves OK	Indicator	Engage yaw damper system and conduct cockpit test. Observe that rudder moves but indicator bar does not move. Indicator is probably faulty	Replace yaw damper indicator
Rudder oscillates or airplane dutch rolls. Yaw damper engaged	Coupler	Engage yaw damper and conduct cockpit test. If rudder does not center when switch movement stops, conduct coupler self-test. If coupler fails self-test, coupler is faulty. If coupler OK, hydraulic power pack is probably faulty (Ref Chapter 27, Rudder and Rudder Trim Control System)	Replace coupler
System operates intermittently	Autopilot accessory box		Replace autopilot accessory box

Autopilot and Yaw Damper System - Troubleshooting
Figure 101 (Sheet 7)

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Experiencing high rudder pedal forces. Rudder pedal kick back or yaw damper backdrives rudder pedals during flight. Erratic yaw damper operation or rudder oscillates with yaw damper engaged. Erratic rudder pedal steering on ground with yaw damper engaged.	Binding within the standby actuator	Shutoff all hydraulics. Disconnect input control rod from the standby input lever. Measure the force required to move the lever throughout its range of motion. This force, when applied tangentially (90°) to the input crank arm and at approximately the bolt/clevis centerline, shall not exceed one pound.	Replace the standby actuator.

Autopilot and Yaw Damper System - Troubleshooting
Figure 101 (Sheet 8)

3. Autopilot System Trouble Shooting Charts *[1]

A. Pitch Channel Engage Trouble Shooting Chart

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Pitch channel does not engage (Air data computer, vertical gyro, autopilot cutout switch, main electric stabilizer trim and power OK)	Pitch channel	Check meter on front of pitch channel. If meter does not read in red, pitch channel is faulty Conduct self-test of pitch channel. If channel fails self-test, pitch channel is faulty Check for voltage at TP-3 and TP-4 on pitch channel. If voltage, pitch channel is faulty	Replace pitch channel
	Flaps up limit switch, mach trim flap switch or autopilot stabilizer trim servo	Check for 28 volts dc at TP-7 on the pitch channel. If voltage available, interlock OK. If no voltage, test operation of mach trim flap switch (Ref 22-21-31, Mach Trim Flap Switch). Test operation of flaps up limit switch (Ref Chapter 27, Flap Limit Switches). If switches test OK, lower flaps 2 units and attempt to engage system. If system engages, stabilizer trim servo is faulty. If system does not engage, a flap switch is faulty	Replace or adjust flaps up limit switch (Ref Chapter 27, Flap Limit Switches). Replace or adjust mach trim flap switch (Ref 22-21-31, Mach Trim Flap Switch). Replace autopilot stabilizer trim servo (Ref 22-11-81, Autopilot Stabilizer Trim Servo)
	Autopilot accessory box or autopilot control panel	Replace autopilot accessory box and attempt to engage system. If system engages, autopilot accessory box was faulty. If system does not engage, autopilot control panel is probably faulty	Replace autopilot accessory box. Replace autopilot control panel

*[1] Airplanes delivered with B system only on autopilot control panel use separate procedures as indicated. Airplanes incorporating SB 27-1074 (B system only) use procedures for A and B system switching. System switching is retained but inflight operation is restricted to B system only to provide operable yaw damper system.

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TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Pitch channel does not engage (Cont)	Pitch CWS force transducer	Check pitch control wheel steering transducers null voltage (Ref 22-11-91, Pitch Control Wheel Steering Force Transducer). If voltage unsatisfactory, transducer is faulty	Adjust or replace pitch control wheel steering force transducer (Ref 22-11-91, Pitch Control Wheel Steering Force Transducer)

B. Pitch Channel Operation Trouble Shooting Chart

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Elevators kick sharply on engagement (Control column movement limit is 0.5 inch)	Pitch channel or control panel	Conduct self-test of pitch channel. If pitch channel fails self-test, pitch channel is faulty. If pitch channel OK, control panel is probably faulty	Replace pitch channel. Replace control panel
Pitch channel disengages intermittently (Interlocks known good)	Pitch channel or control panel	Replace pitch channel. Engage pitch channel. If fault persists, control panel is probably faulty	Replace pitch channel. Replace control panel
Pitch channel inoperative or intermittent operation on system A or B *[1]	Pitch channel	Select system A, engage pitch channel and move control column. If elevators do not respond, select system B and engage pitch channel. Move control column. If system OK, see Hydraulic Power Pack Component Trouble Shooting Chart. If system does not operate, pitch channel is probably faulty	Replace pitch channel
	Autopilot switching accessory box or autopilot accessory box	If pitch channel has intermittent operation on system A or B, replace autopilot switching accessory box. If fault is not corrected, replace autopilot accessory box	Replace autopilot switching accessory box
	Control panel	If intermittent fault persists, autopilot control panel is faulty	Replace autopilot control panel

*[1]
panel.

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TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Pitch channel inoperative or intermittent operation *[2]	Pitch channel	Replace pitch channel and move control column. If elevators do not respond, see Hydraulic Power Pack Component Trouble Shooting Chart. If system does not operate, pitch channel is probably faulty	Replace pitch channel
	Autopilot accessory box	If pitch channel has intermittent operation, replace autopilot accesory box	Replace autopilot accessory box
	Control panel	If intermittent fault persists, autopilot control panel is faulty	Replace autopilot control panel

*[2] Effective on airplanes with B system only on autopilot control panel.

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Pitch axis stabilization is poor (Vertical gyro known good)	Pitch channel	Conduct self-test of pitch channel. If pitch channel fails self-test pitch channel is faulty	Replace pitch channel
Pitch channel does not respond to control wheel steering inputs	Pitch channel, CADC or control wheel steering transducer	Conduct pitch channel self-test. If pitch channel fails self-test 2 or more, push on control column and check for 1.3 to 2.4 V ac at TP-13 on pitch channel. If voltage, pitch channel is faulty. If no voltage, CADC is faulty. If pitch channel ok, control wheel steering transducer is faulty	Replace pitch channel Adjust or replace pitch CWS transducer (Ref 22-11-91, Pitch Control Wheel Steering Force Transducer)
Pitch control wheel steering oversensitive or erratic	Control column detent breakout forces out of tolerance (pitch force transducers misrigged)	Engage autopilot. Apply 14-pound push then pull to control column while monitoring voltage at pitch channel TP-13. Check voltage is between 1.3 and 2.4 VAC	Voltage erratic. Replace pitch channel

		If voltage incorrect disconnect captain's force transducer connector. Apply 28-pound push then pull to control column while monitoring voltage at pitch channel TP-13, Check voltage is between 1.3 and 2.4 VAC	If voltage correct replace first officers pitch CWS force transducer (Ref 22-11-91)
		If voltage incorrect repeat above procedure with captains force transducer connected and first officers force transducer disconnected	If voltage correct replace captain is pitch CWS force transducer (Ref 22-11-91)

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Pitch control wheel steering oversensitive or erratic (Cont)		If voltages incorrect for all checks, perform pitch channel self-test	If self-test fails, replace pitch channel
		If pitch self-test OK, replace CADC and repeat voltage checks above. If OK CADC is faulty	Replace CADC
		If still not OK, check elevator rigging for excessive force and binding	Perform elevator rigging check (Ref Chapter 27)
Intermittent high pitch control wheel steering detent forces	Approach progress display	If abnormally high forces are required to initiate a control wheel steering pitch input, check the approach progress display by shining a light on the approach progress display photocell sensor. If control wheel steering forces are reduced to the proper level, then the approach progress display is faulty	Replace approach progress display
Control wheel steering trips ALT HOLD with low forces or excessively high forces	Pitch channel or control wheel steering transducer	Conduct pitch control wheel steering test (Ref 22-11-91, Pitch Control Wheel Steering Force Transducer). If forces required to trip ALT HOLD are low, pitch channel is faulty. If forces required to trip ALT HOLD are high, control wheel steering transducer is faulty	Replace pitch channel. Adjust or replace pitch CWS transducer (Ref 22-11-91, Pitch Control Wheel Steering Force Transducer)
Elevators drive to limits	Pitch channel	Conduct self-test of pitch channel. If pitch channel fails self-test, pitch channel is faulty. If pitch channel ok, refer to Hydraulic Power Pack Component Trouble Shooting Chart	Replace pitch channel

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TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Autopilot stabilizer trim warning light illuminated (flight fault)	Pitch channel	Replace pitch channel. Engage pitch channel. Close autopilot stabilizer trim circuit breaker. Pull or push on control column and return to neutral. Stabilizer should drive. If stabilizer does not drive, see Stabilizer Trim Trouble Shooting Chart	Replace pitch channel
	Neutral shift sensor	Check voltage at TP-15 and TP-10 on pitch channel with elevators in neutral. If voltage at TP-15 is not 70 ±20 percent of voltage at TP-10, neutral shift sensor is faulty	Adjust or replace neutral shift sensor (Ref 22-11-121, Elevator Neutral Shift Sensor)
	Elevator power control unit (PCU)	Replace elevator PCU and check for sufficient actuator force output with autopilot input per overhaul manual instructions	Replace elevator PCU
Airplane porpoises when on ALT HOLD (ADC known good)	Pitch channel	Conduct self-test of pitch channel. If pitch channel fails self-test 3 or more, pitch channel is faulty	Replace pitch channel
Airplane porpoises (autopilot and ADC known good)		Conduct feel force system and autopilot authority (Ref 27-31-0, elevator and tab control system). Check that none of the column force measurments are at or exceed the maximum allowed values	Identify source of excess friction and replace appropriate components NOTE: The most likely sources of friction is faulty support bearings on the input (lower) elevator torque tube
Autopilot does not capture glide slope when on approach (Nav good)	Pitch channel	Conduct self-test of pitch channel. If pitch channel fails self-test 4 or more,	Replace pitch channel

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TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Airplane does not follow glide slope	Pitch channel	Conduct self-test of pitch channel. If pitch channel fails self-test 4 or more, pitch channel is faulty	Replace pitch channel
Airplane response is not reduced approximatly 50 percent when TURB is selected	Pitch channel or control panel	Conduct self-test of pitch channel. If pitch channel fails self-test 4 or more, pitch channel is faulty. If pitch channel ok, control panel is faulty	Replace pitch channel. Replace autopilot control panel
Airplane does not maintain pitch attitude when in a turn (roll engaged)	Pitch channel or roll channel	Conduct self-test of pitch channel. If pitch channel fails self-test 5 or more, pitch channel is faulty. If pitch channel ok, roll channel is faulty	Replace pitch channel. Replace roll channel
Elevator neutral not correct for stabilizer posi- tion (pitch engaged)	Neutral shift sensor	Engage pitch channel. Energize mach trim system and press mach trim cock- pit test button. If con- trol column does not move back and elevators move up, neutral shift sensor is faulty. (Refer to 22-21-0, Mach Trim System.)	Replaace neutral shift sensor (Refer to 22-11- 121, Elevator Neutral Shift Sensor.)

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TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Glide slope annunciator lights do not indicate armed or engaged modes	Autopilot accessory box	Replace autopilot accessory box and test lights (Refer to Test Approach Mode in Adjustment/Test). If fault not corrected, see Autopilot Annunciator Trouble Shooting Chart	Replace autopilot accessory box
NAV mode select switch does not return to MAN when CWS used on glide slope	Pitch channel or roll channel	Replace pitch channel and test system. (Refer to Test AUTO/APP Mode Control in Adjustment/Test.) If fault is not corrected, roll channel is faulty	Replace pitch channel. Replace roll channel

B. Stabilizer Trim Troubleshooting Chart

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Stabilizer trim warning light illuminated (flight fault) (interlocks ok)	Pitch channel or stabilizer trim servo	Replace pitch channel and engage pitch channel. Close autopilot stabilizer trim circuit breakers. Move control column forward or back and neutralize. Stabilizer should drive to maximum nose up or nose down position. If stabilizer does not move, stabilizer trim servo is faulty	Replace pitch channel. Replace autopilot stabilizer trim servo (Ref 22-11-81, Autopilot Stabilizer Trim Servo)
	Elevator power control unit (PCU)	Replace elevator PCU and check for sufficient actuator force output with autopilot input per over-haul manual instructions	Replace elevator PCU
Stabilizer drives in only one direction	Pitch channel or stabilizer trim servo	Repeat above isolation procedure	Replace pitch channel. Replace autopilot stabilizer trim servo (Ref 22-11-81, Autopilot Stabilizer Trim Servo)
	Stabilizer trim limit switches	Manually trim stabilizer to 3 units of trim. Open autopilot trim cutout switch. Check continuity through S144 and S145 limit switches. No continuity, limit switch is faulty.	Replace stabilizer trim limit switch

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
		NOTE: Terminals 2 and 5 are stowed.	

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Stabilizer speed does not increase with flaps down	Stabilizer trim servo	Test autopilot stabilizer trim servo (Refer to 22-11-81, Autopilot Stabilizer Trim Servo).	Replace autopilot stabilizer trim servo (Refer to 22-11-81, Autopilot Stabilizer Trim Servo).
Stabilizer light does not illuminate with airplane out of trim	Pitch channel	Press autopilot stabilizer trim warning light. If light illuminates, pitch channel is faulty	Replace pitch channel

C. Roll Channel Engage Troubleshooting Chart

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Roll channel does not engage (air data computer vertical gyro, directional gyro and power OK)	Roll channel	Check meter on front of roll channel. If meter does not read in red, roll channel is faulty	Replace roll channel
		Conduct self-test of roll channel. If channel falls self-test, roll channel is faulty	
	Aileron force limiter	Disconnect D443 on force limiter; apply P8-v dc to pin 1 and ground pin 2 of force limiter. Check for continuity between pins 3 and 4. If no continuity, force limiter is faulty	Replace aileron force limiter (Refer to 22-11-131, Aileron Force Limiter).
	Autopilot accessory box or control panel	Replace autopilot accessory box. Attempt to engage roll channel. If roll channel engages, accessory box was faulty. If roll channel does not engage, autopilot control panel is probably faulty	Replace autopilot accessory box. Replace autopilot control panel

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
	Roll CWS force transducer	Check roll control wheel steering transducer null voltage (22-11-111, Roll Control Wheel Steering Force Transducer.) If voltage unsatisfactory transducer is faulty	Adjust or replace roll control wheel steering force transducer (Refer to 22-11-111, Roll Control Wheel Steering Force Transducer).

E. Roll Channel Operation Trouble Shooting Chart

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Ailerons kick sharply on engagement NOTE Ailerons may kick when airplane on ground due to wings leveling circuit. Maximum allowable control wheel rotation is 6 degrees.	Roll channel or autopilot switching accessory box (if installed)	Conduct self-test of roll channel. If roll channel fails self-test, roll channel is faulty. If roll channel OK, switching accessory box is probably faulty NOTE: Switching accessory box is removed on airplanes delivered with B system only on autopilot control panel.	Replace roll channel. Replace autopilot switching accessory box (if installed)
Roll channel disengages intermittently (interlocks known good)	Roll channel or control panel	Replace roll channel. Engage roll channel. If fault persists, control panel is probably faulty	Replace roll channel. Replace control panel
Roll channel inoperative or intermittent on system A or B *[1]	Roll channel or autopilot switching accessory box	Select system A. Engage roll channel and move control wheel. If ailerons do not respond, select system B and engage roll channel. Move control wheel. If system OK, switching accessory box is probably faulty. If trouble persists, see Hydraulic Power Pack Component, Trouble Shooting Chart. If system does not operate, roll channel is probably faulty	Replace roll channel. Replace autopilot switching accessory box
	Autopilot switching accessory box	If roll channel has intermittent operation on system A or B, replace autopilot switching accessory box. If fault is not corrected, replace autopilot accessory box	Replace autopilot switching accessory box. Replace autopilot accessory box
	Control panel	If intermittent fault persists, autopilot control panel is faulty	

*[1] panel.

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TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Roll channel inoperative or intermittent *[2]	Roll channel	Engage roll channel and move control wheel. If ailerons do not respond, see Hydraulic Power Pack Component, Trouble Shooting Chart. If system does not operate, roll channel is probably faulty	Replace roll channel
	Autopilot accessory box	If roll channel has intermittent operation, replace autopilot accessory box	Replace autopilot accessory box
	Control panel	If intermittent fault persists, autopilot control panel is faulty	Replace control panel
Roll axis stabilization is poor (vertical gyro known good)	Roll channel	Conduct self-test of roll channel. If roll channel fails self-test, roll channel is faulty.	Replace roll channel
Roll channel does not respond to control wheel steering inputs	Roll channel or control wheel steering transducer	Conduct roll channel self-test. If roll channel fails self-test 3 or more, roll channel is faulty. If roll channel ok, roll channel wheel transducer is faulty	Replace roll channel. Adjust or replace roll CWS transducer (Ref 22-11-111, Roll Control Wheel Steering Force Transducer)
Roll control wheel steering oversensitive or erratic	Control wheel detent breakout forces out of tolerance (roll force transducer misrigged)	With autopilot disengaged and hydraulic pressure off apply 90 pound-inches of torque on control wheel while monitoring at roll channel TP-18. Check voltage is between 4.0 and 6.0 VAC	Voltage erratic. Replace roll channel.

*[2] Effective on airplanes with B system only on autopilot control panel.

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Roll channel wheel steering oversensitive or erratic (Cont)		If voltage incorrect in either direction or both. Perform roll channel self-test	If roll channel self-test fails, replace roll channel
		If self-test OK replace CADC and recheck voltage check above. If OK CADC is faulty	Replace CADC
		If still not OK roll CWS transducer is misrigged or faulty, or aileron system has excessive force or binding	Rerig or replace roll CWS transducer (Ref 22-11-111). If not OK - rerig aileron control system (Ref Chapter 27)
Control wheel steering trips high detent modes with low forces or excessively high forces	Roll channel or control wheel steering transducer	Conduct roll control wheel steering test (Ref 22-11-111, Roll Control Wheel Steering Force Transducer). If forces required to trip HGD SEL are low, roll channel is faulty. If forces required to trip HDG SEL, are high, control wheel steering transducer is faulty	Replace roll channel. Adjust or replace roll CWS transducer (Ref 22-11-111, Roll Control Wheel Steering Force Transducer)

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Ailerons drive to limits NOTE: Ailerons may slowly drive to limits due to forward path integrator	Roll channel or autopilot switching accessory box (if installed)	Disengage channel or 10 seconds and re-engage. Ground TP-11 or roll channel. If ailerons stop driving, system OK. If ailerons continue to drive, connect self-test of roll channel. If roll channel fails a self-test, roll channel is faulty. If roll channel OK, switching accessory box is probably faulty. If trouble persists, see Hydraulic Power Pack Compnt. TS Chart NOTE: Switching accessory box removed on airplanes delivered with B system only on autopilot control panel.	Replace roll channel. Replace autopilot switching accessory box (if installed)
Force limiter forces or bank limits are not correct for cam-out	Force limiter	Conduct aileron force limiter test (Ref 22-11-131, Aileron Force Limiter). If forces and limits are not correct, force limiter is faulty	Replace aileron force limiter (Ref 22-11-131, Aileron Force Limiter).
Airplane rolls wings level when in attitude hold and bank angle exceeds 5°	Roll channel	Conduct self-test of roll channel. If roll channel fails self-test 2 or more, roll channel is faulty	Replace roll channel
Airplane does not return to wings level when bank angle less than 5° and in heading hold mode	Roll channel	Conduct self-test of roll channel. If roll channel fails self-test 2 or more, roll channel is faulty	Replace roll channel

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Autopilot does not capture VOR radial or LOC beam (Nav receiver known good)	Roll channel	Conduct self-test of roll channel. If roll channel fails self-test 1 or more, roll channel is faulty	Replace roll channel
Heading switch does not go to center on VOR/LOC capture	Roll channel or control panel	Engage roll channel and select HDG SEL. Rotate control wheel until heading switch centers. If switch centers, roll channel is faulty. If switch dos not center, control panel is faulty	Replace roll channel. Replace autopilot control panel
Airplane does not roll out on-course (Nav receiver and CDI known good)	Roll channel	Conduct self-test roll channel. If roll channel fails self-test 5 or more, roll channel is faulty	Replace roll channel
Airplane flies with standoff from VOR radial and LOC beam NOTE: Could be caused by mistrimmed airplane or faulty navigation receiver.	Roll channel or navigation receiver	Conduct self-test of roll channel. If roll channel fails self-test, roll channel is faulty. If roll channel ok, navigation receiver may be faulty. If replacing receiver does not correct fault, check vertical gyro and directional gyro installation (Refer to Chapter 34, Navigation.)	Replace roll channel. Replace navigation receiver

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Airplane oscillates when over VOR station	Roll channel	Conduct self-test of roll channel. If roll channel fails self-test 1 or more, roll channel is faulty	Replace roll channel

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Bank angle limits or rates are not correct for a mode or submode	Roll channel	Conduct self-test of roll channel. If roll channel fails self-test, roll channel is faulty	Replace roll channel
NAV mode switch does not return to MAN when on localizer and TURB is selected	Roll channel or control panel	Replace roll channel. Engage roll and pitch channels, select ILS frequency and VOR/LOC. Select TLJRB If fault is not corrected, control panel is faulty	Replace roll channel. Replace autopilot control panel
VOR/LOC lights do not indicate armed or engaged mode	Roll channel	Replace roll channel and test lights (Refer to Test VOR/WC Mode in Adjustment/Test) If fault is not corrected, see Autopilot Annunciator Trouble Shooting Chart	Replace roll channel
NAV mode select switch does not return to MAN when CWS used with on-course submode	Roll channel or control panel	Engage roll channel and or control panel pitch channel. Select an ILS frequency. Select AUTO APP. Change from ILS to a VOR frequency. If the NAV mode switch returns to MAN, the roll channel is faulty. If the NAV mode switch does not return to MAN, the control panel is faulty	Replace roll channel. Replace autopilot control panel

D. Autopilot Mode Select Troubleshooting Chart

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
NAV mode switch does not remain in VOR/LOC, AUTO APP or MAN/GS	Autopilot accessory box (if NAV mode circuit interrupter is installed) or roll channel	Replace autopilot accessory box. Engage roll channel and select VOR/LOC. If NAV mode switch remains in VOR/LOC, autopilot accessory box was faulty. If switch does not remain in VOR/LOC, roll channel may be faulty	Replace autopilot accessory box. Replace roll channel
	Control panel	If replacing roll channel does not correct fault, control panel is faulty	Replace control panel
NAV mode switch does not remain in VOR/LOC (AUTO APP and MAN G/S ok)	Roll channel	Engage roll channel. Select VOR/LOC. If VOR/LOC does not remain engaged, roll channel is faulty	Replace roll channel
NAV mode switch does not remain in AUTO APP or MAN G/S (NAV receiver and ADC known good)	Roll channel or pitch channel	Replace roll channel. Engage roll and pitch channel. Select an ILS frequency. Select AUTO APP. If AUTO APP does not remain engaged, pitch channel is faulty	Replace roll channel. Replace pitch channel

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Heading switch does not remain in HDG OFF	Control panel	Engage roll channel. Select HDG OFF. If switch does not remain in HDG OFF, control panel is faulty.	Replace autopilot control panel
NOTE: NAV mode select switch must be in MAN.			
Heading switch does not remain in HDG SEL	Roll channel or control panel	Replace roll channel. Engage roll channel and select HDG SEL. If switch remains in HDG SEL, roll channel is faulty. If switch returns to center, control panel is faulty	Replace roll channel. Replace autopilot control panel
NOTE: Control wheel steering must be in detent			
Pitch mode switch does not remain in TURB	Control panel or pitch channel	Engage pitch channel. Select TURB. If switch does not remain in TURB, control panel may be faulty. If replacing control panel does not correct fault, pitch channel is faulty.	Replace control panel. Replace pitch channel
Pitch mode switch does not remain in ALT HOLD (ADC known good)	Pitch channel or control panel	Replace pitch channel. Engage pitch channel and select ALT HOLD. If switch remains in ALT HOLD, pitch channel was faulty. If switch returns to OFF, control panel is faulty.	Replace pitch channel. Replace autopilot control panel.

G. Autopilot Hydraulic Power Pack Trouble Shooting Chart (Effective on airplanes with A and B system selection on autopilot control panel)

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Roll or pitch channel inoperative on system A or B	Roll or pitch channel	Select system A. Engage channel and move control wheel/column. If control surface does not respond, select system B and engage channel. Move control wheel/column. If control surface moves, channel is ok. If control surface does not move, channel is faulty	Replace pitch or roll channel
	Solenoid valve	Select system A or B and and check on which system pitch or roll channel is inoperative. If inoperative on system A, solenoid valve on A power pack is faulty. If inoperative on system B, solenoid valve on B power pack is faulty	Replace solenoid valve (Ref 22-11-11, Aileron Solenoid Valve or 22-11-31, Elevator Solenoid Valve).
Roll or pitch channel intermittent on system A or B (roll and pitch channel known good)	Autopilot switching accessory box or autopilot accessory box	Replace autopilot switching accessory box. Select system A, engage channel and move control wheel/column. Observe control surface movement is correct. Select system B, engage channel and move control wheel/column. If control surfaces move satisfactorily for both systems, autopilot switching accessory box was faulty. If control surface movement is not satisfactory on both systems, autopilot accessory box is probably faulty	Replace autopilot switching accessory box. Replace autopilot accessory box
	Control panel	Observe if autopilot warning lights flash but channel is engaged. Press the warning light. The light should extinguish. Disengage channel and re-engage, if lights continue to flash, control panel is faulty	Replace control panel

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TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Roll or pitch channel appears to have not mechanically disengaged from system A or B power pack (hydraulic pressure on packs)	Solenoid valve	Remove hydraulic power from control systems (AMM Chapter 27, Flight Controls). Manually cycle control surfaces serveral times. Energize hydraulic system B. Manually move control wheel or column. If control surfaces cannot be moved or are difficult to move, system B solenoid valve is leaking. If control surfaces move freely, apply hydraulic power to system A. Manually move wheel or column.	Replace solenoid valve (AMM 22-11-11, Aileron Solenoid Valve or 22-11-31, Elevator Solenoid Valve)
Ailerons move to limits when roll channel is engaged (roll channel known good)	Position transducer or transfer valve	Select system A or B. Engage roll channel. If ailerons move to limits, check for voltage between TP-3 and TP-4 on roll channel. If no voltage position transducer is faulty. If voltage is available, transfer valve is faulty	Replace aileron position transducer (AMM 22-11-231, Aileron Position Transducer). Replace aileron transfer valve (AMM 22-11-21, Aileron Transfer Valve).
Elevators move to limits when pitch channel is engaged (pitch channel known good)	Position transducer or transducer transfer valve	Select system A or B. Engage pitch channel. If elevators move to limits, check for voltage at TP-10 on pitch channel. If no voltage, position transducer is faulty. If voltage is available, transfer valve is faulty	Replace elevator position transducer (AMM 22-11-241, Elevator Position Transducer). Replace elevator transfer valve (AMM 22-11-41, Elevator Transfer Valve).

E. Autopilot Hydraulic Power Pack Trouble Shooting Chart (Effective on airplanes with B system only on autopilot control panel)

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Roll or pitch channel inoperative	Roll or pitch channel or solenoid valve	Engage channel and move control wheel/column. If control surface does not move, channel is faulty. If fault persists after channel is replaced, solenoid valve on B power pack is faulty	Replace pitch or roll channel Replace solenoid valve (AMM 22-11-11, Aileron Solenoid Valve or AMM 22-11-311, Elevator Solenoid Valve)
Roll or pitch channel intermittent (roll and pitch channel known good)	Autopilot accessory box or control panel	Replace autopilot accessory box. Observe if autopilot warning lights flash but channel is engaged. Press the warning light. The light should extinguish. Disengage channel and re-engage, if lights continue to flash, control panel is faulty	Replace autopilot accessory box. Replace control panel

F. Autopilot Disengage Warning Lights Trouble Shooting Chart

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Roll or pitch channel appears to have not mechanically disengaged from power pack (hydraulic pressure on packs)	Solenoid valve	Remove hydraulic power from control systems (Ref Chapter 27, Flight Controls). Manually cycle control surfaces several times. Energize hydraulic sytem B. Manually move control wheel or column. If control surfaces cannot be moved or are difficult to move, system B solenoid valve is leaking	Replace solenoid valve (Ref 22-11-11, Aileron Solenoid Valve or 22-11-31, Elevator Solenoid Valve)
Ailerons move to limits when roll channel is engaged (roll channel known good)	Position transducer or transfer valve	Engage roll channel. If ailerons move to limits, check for voltage between TP-3 and TP-4 on roll channel. If no voltage position transducer is faulty. If voltage is available, transfer valve is faulty	Replace aileron position transducer (Ref 22-11-231, Aileron Position Transducer). Replace aileron transfer valve (Ref 22-11-21, Aileron Transfer Valve)
Elevators move to limits when pitch channel is engaged (pitch channel known good)	Position transducer or transducer transfer valve	Engage pitch channel. If elevators move to limits, check for voltage at TP-10 on pitch channel. If no voltage, position transducer is faulty. If voltage is available, transfer valve is faulty	Replace elevator position transducer (Ref 22-11-241, Elevator Position Transducer). Replace elevator transfer valve (Ref 22-11-41, Elevator Transfer Valve)

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Figure 101 (Sheet 17)

I. Autopilot Annunciator Trouble Shooting Chart

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Both captain's and first officer's VOR/LOC lights do not indicate armed or engaged modes (lights known good)	Roll channel or flight instrument accessory box	Replace roll channel. Test lights. (Refer to VOR/LOC mode in Adjustment/Test) If fault is not corrected, flight instrument accessory box box is faulty	Replace roll channel. Replace flight instrument accessory box
Both captain's and first officer's glide slope lights do not indicate armed or engaged modes	Autopilot accessory box or pitch channel	Replace autopilot accessory box. Test lights. (Refer to Test Approach Mode in Adjustment/Test) If fault is not corrected, the pitch channel is faulty	Replace autopilot accessory box. Replace pitch channel
Either captain's or first officer's VOR/LOC and glide slope lights do not indicate armed and engaged modes	Flight instrument accessory box	Replace flight instrument accessory box. Test lights by pressing light covers. If lights illuminate flight instrument accessory box was faulty	Replace flight instrument accessory box
Either captain's or first officer's VOR/LOC and glide slope lights do not dim automatically	Flight instrument accessory box or photocell	Replace flight instrument accessory box. Dim cockpit lights (Ref Chapter 33, Control Cabin Lighting). If annunciator lights do not dim automatically, the associated photocell on the captain's or first officer's instrument panel is faulty	Replace flight instrument accessory box. Replace photocell

Autopilot and Yaw Damper System - Troubleshooting
Figure 101 (Sheet 18)

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Lights illuminate steady	Self-test switch left in a test position	Check pitch channel, roll channel and yaw damper coupler self-test switches. Ensure all self-test switches are off	Return self-test switch to OFF
Lights illuminate steady when autopilot is disengaged	Autopilot accessory box	Press warning light or autopilot disengage switch. If lights do not flash, autopilot accessory box is faulty	Replace autopilot accessory box
Lights do not illuminate when autopilot is disengaged	Control panel or autopilot accessory box	Press warning light. If lights flash, control panel is faulty. If lights do not flash, autopilot accessory box is faulty	Replace autopilot control panel. Replace autopilot accessory box

3. Yaw Damper Coupler Part No. 4030952 BITE Test Troubleshooting Charts
   A. The BITE tests have been designed primarily for yaw damper system verification. The tests are not intended for isolating failures to the line replaceable unit (LRU); however, failed BITE tests or combination of tests provide some clues useful or troubleshooting to the most likely faulty LRU.
   B. All BITE tests should be run first and failed tests noted (refer to Yaw Damper System - Adjustment/Test). Components involved in each test have an X listed under the appropriate test. In most cases, failure of any of the BITE requires replacement of the unit being tested. If additional tests are failed after replacement of the unit, or if all test fail involving the engage interlock circuits, the interlock circuits should be checked and verified before replacing additional LRUs. Hydraulic power control unit components are assumed to be part of the engage interlocks. The following charts assume only single failures with the most likely failures listed from the top down for each test.
   C. Yaw Damper Coupler BITE Test Chart
      ON AIRPLANES WITH YD COUPLER P/N 4030952-902
      ON AIRPLANES WITH YD COUPLER P/N 4030952-906

YAW DAMPER COUPLER BITE TEST FAILED									PROBABLE CAUSE	REMEDY
1	2	3	4	5 [1]	5 [2]	6	7	8 [2]		
X	X	X	X	X	X	X	X	X	Yaw Damper Coupler	Replace yaw damper coupler
X		X	X	X					Engage Interlocks	Check interlock continuity and correct fault
X									ADC	Replace ADC (Chapt 34)
X		X	X	X					Y/D PCU Components	Y/D PCU components (22-11-51) (22-11-61) (22-11-71)

[1] ON AIRPLANES WITH YD COUPLER P/N 4030952-902

[2] ON AIRPLANES WITH YD COUPLER P/N 4030952-906

Autopilot and Yaw Damper System - Troubleshooting
Figure 101 (Sheet 19)

AUTOPILOT AND YAW DAMPER SYSTEM - ADJUSTMENT/TEST

1. General
   A. This series of tests ensures that the autopilot and yaw damper system is not malfunctioning. It is assumed that all systems furnishing signal inputs to the autopilot and yaw damper system are in proper working order and that the stabilizer, elevator, and aileron control systems have been properly rigged.
   B. The following sources supply input signal data to the autopilot and yaw damper system (AMM Chapter 34).
      (1) VHF navigation
      (2) Glide slope
      (3) Vertical gyro
      (4) Compass
      (5) Radio altimeter
      (6) Air data computer
   C. To ensure complete system operability, all tests contained herein must be performed. However the autopilot system or the yaw damper system may be tested separately without degrading overall system performance. Further, tests contained within each system test are arranged to allow testing on a selective basis.
   D. The pitch, roll and yaw test points (TP) referenced throughout the tests, refer to test points located on the front panel of the pitch, roll and yaw control channels.
   E. Throughout the autopilot test, on airplanes with A-B system select switching and POST-SB 27-1074, use system select B position and flight control B switch as the primary system instead of system select A position (inop) and flight control A switch. On airplanes without system select switching, disregard references to system A and B switching.
2. Autopilot System Test
   A. Equipment and Materials
      (1) Signal Generator - Collins, 972Q-4; Instrument Flight Research, NAV401; Tel Instruments, T30B/B; Cossor, CRM 555A
      (2) Portable Pressure Equipment - To simulate 0 to 35,000 feet altitude and 0 to 400 knots airspeed and 400 to 1000 feet per minute altitude rate signals
      (3) Tilt Table (for vertical gyro)
      (4) Protractor - F-52485-500 (4MIT65B80307-1) plus Adapter F-72790
      (5) Vertical Gyro Extension Cable
      (6) Radio Signal Simulator (Fig. 501)
      (7) Radio Altimeter Test Set - Collins Radio Co., Model 980N-1 or Bendix Radio Corp., Model AMT-51A as applicable
      (8) VTVM, Hewlett Packard - Model 410C (with test leads and adapter to fit test points on face of roll and pitch channels)
      (9) Chatillon Spring Scale - Model DPP-50
      (10) Steel Measuring Tape

(11) Adapter Assemblies (2), Control Wheel Torque Tool - F72867-1

B. Prepare to Test Autopilot System

(1) Mount vertical gyro(s) on tilt table as required to measure roll and pitch displacements. Connect gyro(s) to system using vertical gyro extension cable. Use gyro No. 1 for all aileron tests.

(2) Ensure that autopilot trim cutout switch is in NORMAL.

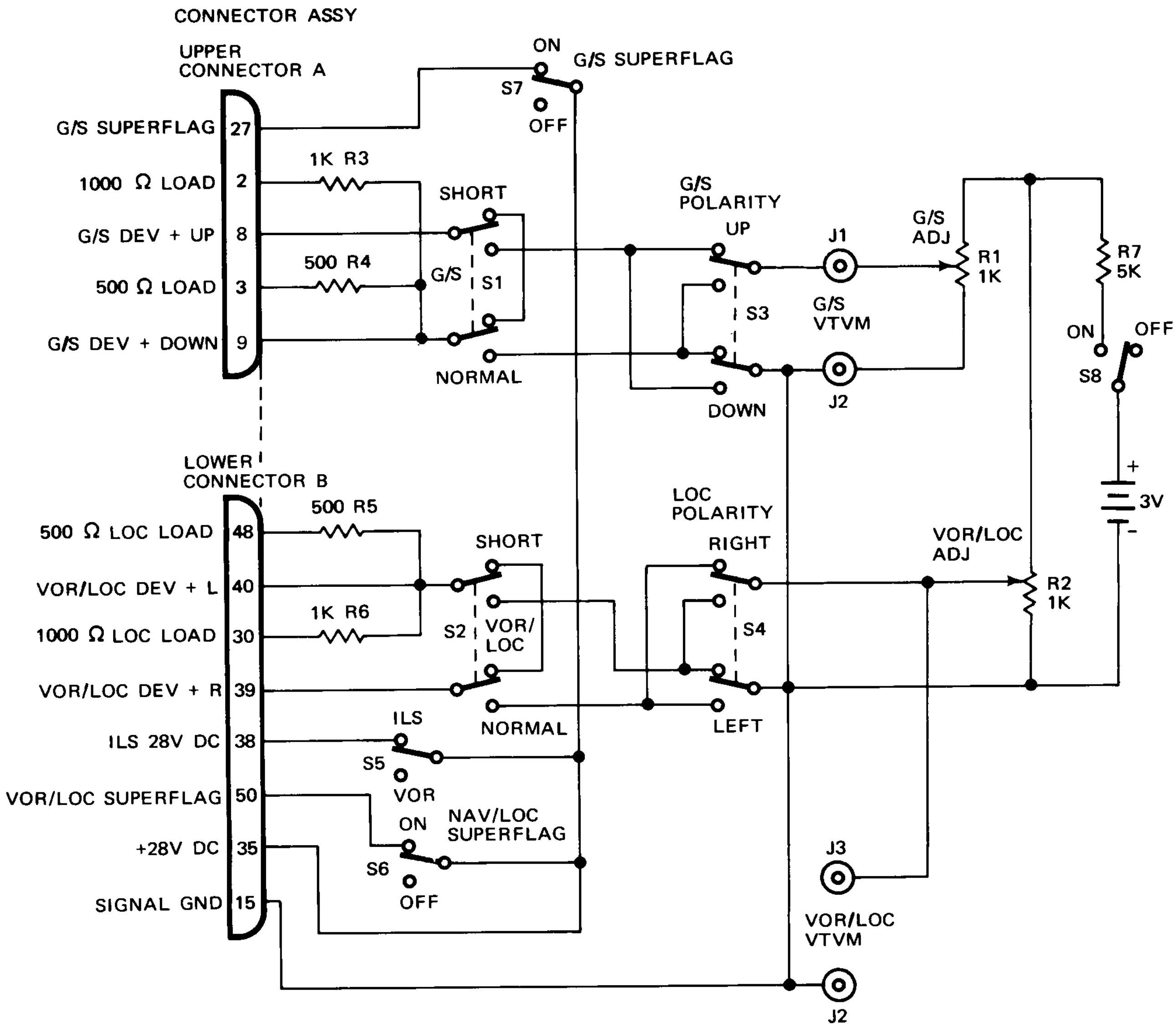
(3) Connect pitot static pressure test equipment to air data computer pitot and static sources as required. Maintain ambient static pressure and pitot pressure to produce 120 knots indicated airspeed (KIAS).

CAUTION: INSTRUMENT DAMAGE MAY RESULT IF PRESSURE TEST EQUIPMENT IS NOT CONNECTED PROPERLY. REFER TO AMM CHAPTER 34, AIR DATA INSTRUMENTS, FOR DETAILED INSTRUCTIONS.

(4) Use the radio signal simulator to generate required localizer and omnirange signals. Equivalent localizer and omnirange signals may be generated using signal generators. Place localizer and omnirange signal generators 75 feet from VOR antennas.

NOTE: Whenever VHF navigation receivers are required to be energized and no localizer or VOR signal is required, receivers should be detuned from any local station.

NOTE: Omniradial, as used in this test, is the signal as it exists along a line in the magnetic direction from the VOR station, specified by the number of the radial; i.e., 0 degree radial is north of the station, 90 degrees radial is east of the station. As presented on the course deviation indicator (CDI), it is the signal which will produce a FROM indication when the course selector is placed on the radial number.



LIST OF MATERIALS

BATTERY	2 DRY CELL, 1-1/2 VOLTS
J1, J2, & J3	BANANA JACK
S1, S2, S3, & S4	DPDT SWITCH
S5, S6, S7, & S8	SPST SWITCH
R1 &R2	1K, 0.25%, 2W
R3 & R6	1K (± 5%)
R4 & R5	500 OHM (± 5%)
R7	5K (± 1%)
CONNECTOR ASSEMBLY, CANNON	DPX2CA32W2P57P34B0002

TO BE FABRICATED BY THE AIRLINE USING ANY SUITABLE CHASSIS AND CABLE LENGTH

Radio Signal Simulator
Figure 501

(5) Place glide slope signal generators approximately 50 feet from glide slope antenna with the test set antenna at a right angle to airplane centerline.

NOTE: VOR, localizer, and glide slope deviation indicators have either two or five scale divisions left and right or up and down with full scale deflection to 10 degrees, 2 degrees and 0.7 degree of OFF COURSE signal respectively. These divisions are referred to as dots. Two-dot VOR/LOC indicators are assumed. The following deflections or equivalent should be substituted on airplanes with 5-dot instruments:

2-Dot Indicators	5-Dot Indicators	Equivalent Input Signal 1000-ohm load (mv)	Deviation (Degrees)		
			VOR	LOC	G/S
1/4	1/2	18.8	1.25	0.25	0.087
1/2	1-1/2	37.5	2.5	0.5	0.175
1	2-1/2	75.0	5	1	0.35
2	5	150.0	10	2	0.7

(6) Ensure that any VOR, localizer or glide slope signals, which are required to perform any of the tests, are provided with the respective flag out of view, with an on-course, zero deviation display prior to starting the test.
(7) Obtain simulated compass headings by setting up desired heading in DG mode on the appropriate compass system.
(8) Observe the following whenever control wheel or control column displacements are being measured:
(a) Ensure that controls are in detent or neutral position.
(b) Before each test, ensure that hydraulic power is applied to the control system being tested.

(c) Ensure that stabilizer trim motor is turned off for all tests requiring control column travel with no reference to stabilizer travel. This may be accomplished by opening autopilot stabilizer trim circuit breaker.

CAUTION: TO PREVENT STABILIZER TRIM MOTOR FROM BURNING OUT, DO NOT OPERATE MORE THAN ONE MINUTE IN EVERY 10 MINUTES.

NOTE: Performance of tests in accordance with the sequence presented herein is required for items within a given paragraph. All required sequences start from energized systems, with the autopilot disengaged and the respective controls in a neutral condition.

NOTE: When the aileron engage switch is placed in the ENGAGE position, there may at times be an apparent roll engage transient observed by watching the control wheel move away from center. This is the action of the wings leveling circuit in the autopilot and is entirely normal providing the control wheel error does not exceed 6 degrees. The wheel will recenter after engagement (Ground TP-11 of the roll control channel, while the platform is being recentered) by offsetting the tilt table base (bubble level platform) when the tilt table protractor is set at zero.

(9) Provide electrical power to airplane and ensure that the following systems are energized and operating.
  (a) Hydraulic systems A and B (AMM 29-11-0/201 and AMM 29-12-0/201).

  WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

  (b) Autopilot and yaw damper systems.
  (c) Attitude reference system (AMM Chapter 34).
  (d) Compass system (AMM Chapter 34).
  (e) VOR/GS navigation system (AMM Chapter 34).
  (f) Low range radio altimeter system (if installed) (AMM Chapter 34).
  (g) Air data computer (AMM Chapter 34).
(10) Using altimeter test set, energize radio altimeter at 2000 feet indicated altitude (if installed).

(11) Check that all control surface areas are cleared of obstructions and personnel.
(12) Connect adapter assemblies to control wheel hubs.
(13) Check that flight control A and B switches are on.

<u>NOTE</u>: The flight control A and B switches are in the autopilot and yaw damper engage interlock circuit on airplanes with A and B system selection on the autopilot control panel. The flight control B switch is in the autopilot and yaw damper engage circuit on airplanes with B system only on the autopilot control panel. Hydraulic pressure does not have to be applied for autopilot or yaw damper systems engagement; however, it does have to be applied for systems operation.

C. Test Autopilot System
(1) Test roll engage interlocks (effective on airplanes with A and B system selection on autopilot control panel).
(a) Place autopilot system select switch in A position.
(b) With control wheel and column in detent position, engage AIL (roll axis) engage switch on autopilot control panel.
(c) Depress autopilot disengage button on captain's control wheel. Observe that roll axis disengages as evidenced by dropping out of the AIL engage switch. Check the autopilot disengage warning lights on the pilot's main panels illuminate momentarily.
(d) Engage AIL engage switch.
(e) Repeat (c) using first officer's disengage button.
(f) Engage AIL engage switch.
(g) Open autopilot engage interlock circuit breaker. Observe that AIL engage switch disengages and autopilot disengage warning lights illuminate and flash.
(h) Depress captain's warning light on left instrument panel. Observe that warning lights extinguish and the AIL engage switch cannot be engaged.
(i) Close autopilot engage interlock circuit breaker.
(j) Engage AIL engage switch.
(k) Repeat (f) thru (i) for the roll channel ac, roll channel dc and derived rate ac (when derived rate roll channel is installed) circuit breakers on the P6 load control center and the compass No. 1, vertical gyro No. 1, and air data computer No. 1, 115-volt ac circuit breakers on the P18 load control center.
(l) Engage AIL engage switch. Open compass No. 2 (P6 load control center) and auxiliary vertical gyro (P18 load control center) circuit breakers. Observe that AIL engage switch does not disengage. Close circuit breakers.

(m) Place VERTICAL GYRO select switch in CAPT ON AUX position. Observe that AIL engage switch disengages and that autopilot warning lights illuminate and flash.
(n) Extinguish lights by depressing first officer's light on right-hand panel and engage AIL engage switch.
(o) Repeat (f) thru (i) for auxiliary vertical gyro circuit breaker (P18 load control center).
(p) Engage AIL engage switch and place COMPASS select switch in BOTH ON COMP-2 position. Observe that AIL engage switch disengages.
(q) Engage AIL engage switch. Repeat steps (f) thru (i) for compass No. 2 circuit breaker.
(r) Engage AIL engage switch. Open compass No. 1, and vertical gyro No. 1 circuit breakers. Observe that AIL engage switch does not disengage.
(s) Close circuit breakers.
(t) Place COMPASS select switch in NORMAL position and VERTICAL GYRO switch in NORMAL position. Observe that AIL engage switch disengages. Reset the warning lights.
(u) Engage AIL engage switch and place system select switch in AB position. Observe that AIL engage switch disengages, autopilot warning lights illuminate and flash, and the AIL engage switch cannot be engaged.
(v) Place system select switch in A position. Turn control wheel clockwise out of detent using minimum of 3 pounds-feet of torque. Observe that the AIL engage switch cannot be engaged.
(w) Repeat (v) for counterclockwise control wheel movement.
(x) Engage autopilot and move STANDBY POWER switch on forward overhead panel to OFF position.
(y) AR LV-JMW thru LV-JMZ, LV-JND, LV-JNE;
IC VT-EAG thru VT-EAL;
PV CF-EPL, CF-EPO, CF-EPR;
WE N2711R, N4906, N4907, N4902W;
AQ N21SW thru N23SW;
Observe that AIL engage switch disengages. On ALL EXCEPT airplanes listed above, observe that both AIL and ELEV engage switches disengage. Return STANDBY POWER switch to AUTO position.
(z) Engage AIL engage switch and place system select switch in B position. Observe that AIL engage switch disengages.

(aa) Engage AIL engage switch. On airplanes with Honeywell (Analog) ADC, select IAS SLEW on No. 1 air data computer and press toggle switch marked IAS-M for 10 seconds. On airplanes with Kollsman (Analog) ADC, press IAS SLEW button on No. 1 air data computer for 10 seconds. Observe that AIL engage switch disengages. On airplanes with Honeywell (digital) ADC, set Test Select Switch to FAIL and depress and hold PUSH TO TEST switch for 10 seconds. Release switch and return ADC to normal.
(ab) Check that system select switch is on A and FLT CONTROL A switch is on.
(ac) Position FLT CONTROL B switch to OFF and engage AIL engage switch.
(ad) Move control wheel and observe that ailerons respond to control wheel steering inputs.
(ae) Position FLT CONTROL B switch to ON and system select switch to B.
(af) Position FLT CONTROL A switch to OFF and engage AIL engage switch.
(ag) Move control wheel and observe that ailerons respond to control wheel steering inputs.
(ah) Position FLT CONTROL A switch to ON and system select switch to A.
(ai) Test flight control switches interlock.
1) Engage AIL engage switch and position FLT CONTROL A switch to OFF. Observe AIL engage switch disengages.
2) Return FLT CONTROL A switch to ON.
3) Position system select switch to B and engage AIL engage switch.
4) Position FLT CONTROL A switch to OFF. Observe AIL engage switch does not disengage.
5) Return FLT CONTROL A switch to ON.
6) Position FLT CONTROL B switch to OFF. Observe AIL engage switch disengages.
7) Return FLT CONTROL B switch to ON.
8) Position system select switch to A and engage AIL engage switch.
9) Position FLT CONTROL B switch to OFF. Observe AIL engage switch does not disengage.
10) Position FLT CONTROL B switch to ON and disengage AIL engage switch.

(2) Test roll engage interlocks (Effective on airplanes with B system only on autopilot control panel).
(a) Place autopilot system select switch in B position.
(b) With control wheel and column in detent position, engage AIL (roll axis) engage switch on autopilot control panel.

(c) Depress autopilot disengage button on captain's control wheel. Observe that roll axis disengages as evidenced by dropping out of the AIL engage switch. Check the autopilot disengage warning lights on the pilots' main panels illuminate momentarily.
(d) Engage AIL engage switch.
(e) Repeat (c) using first officer's disengage button.
(f) Engage AIL engage switch.
(g) Open autopilot engage interlock circuit breaker. Observe that AIL engage switch disengages and autopilot disengage warning lights illuminate and flash.
(h) Depress captain's warning light on left instrument panel. Observe that warning lights extinguish and the AIL engage switch cannot be engaged.
(i) Close autopilot engage interlock circuit breaker.
(j) Engage AIL engage switch.
(k) Repeat (f) thru (i) for the roll channel ac, roll channel dc and derived rate ac (when derived rate roll channel is installed) circuit breakers on the P6 load control center, compass No. 1, vertical gyro No. 1, and air data computer No. 1, 115 volts ac circuit breakers on the P18 load control center.
(l) Engage AIL engage switch. Open compass No. 2 (P6 load control center) and auxiliary vertical gyro (P18 load control center) circuit breakers. Observe that AIL engage switch does not disengage. Close circuit breakers.
(m) Place VERTICAL GYRO select switch in CAPT ON AUX position. Observe that AIL engage switch disengages and that autopilot warning lights illuminate and flash.
(n) Extinguish lights by depressing first officer's light on right-hand panel and engage AIL engage switch.
(o) Repeat (f) thru (i) for auxiliary vertical gyro circuit breaker (P18 load control center).
(p) Engage AIL engage switch and place COMPASS select switch in BOTH ON COMP-2 position. Observe that AIL engage switch disengages.
(q) Engage AIL engage switch. Repeat steps (f) thru (i) for compass No. 2 circuit breaker.
(r) Engage AIL engage switch. Open compass No. 1, and vertical gyro No. 1 circuit breakers. Observe that AIL engage switch does not disengage.
(s) Close circuit breakers.
(t) Place COMPASS select switch in NORMAL position and VERTICAL GYRO switch in NORMAL position. Observe that AIL engage switch disengages. Reset the warning lights.

(u) Engage AIL engage switch and place system select switch in AB position. Observe that AIL engage switch disengages, autopilot warning lights illuminate and flash, and the AIL engage switch cannot be engaged.

(v) Place system select switch in B position and engage AIL engage switch and place system select switch in A position. Observe that AIL engage switch disengages, autopilot warning lights illuminate and flash, and the AIL engage switch cannot be engaged.

(w) Place system select switch in B position. Turn control wheel clockwise out of detent using minimum of 3 pounds-feet of torque. Observe that the AIL engage switch cannot be engaged.

(x) Repeat (w) for counterclockwise control wheel movement.

(y) Engage AIL engage switch and move STANDBY POWER switch on forward overhead panel to OFF position. Observe that AIL and ELEV engage switches disengage. Return STANDBY POWER switch to AUTO position.

(z) On airplanes with CADC, engage AIL engage switch.

1) On No. 1 air data computer, depress and hold IAS-SLEW button (Kollsman ADC) for 10 seconds, or select IAS SLEW on rotary switch and press and hold toggle marked IAS-M SLEW (Honeywell ADC) for 10 seconds.

a) Observe that AIL engage switch disengages.

(aa) On airplanes with DADC, engage AIL engage switch.

1) On No. 1 air data computer, set Test Select Switch to FAIL and depress and hold PUSH TO TEST switch for 10 seconds.

a) Observe that AIL engage switch disengages.

(ab) Position FLT CONTROL A switch to OFF and engage AIL engage switch.

(ac) Move control wheel and observe that ailerons respond to control wheel steering inputs.

(ad) Test flight control switches interlock.

1) Engage AIL engage switch and position FLT CONTROL B switch to OFF. Observe AIL engage switch disengages.

2) Return FLT CONTROL A and B switches to ON and engage AIL engage switch.

3) Position FLT CONTROL A switch to OFF. Observe AIL engage switch does not disengage.

(3) Test Pitch Engage Interlocks (Effective on airplanes with A and B system on autopilot control panel)

(a) Place system select switch in A position.

(b) Place ELEV engage switch in ENGAGED position.

NOTE: For the remainder of the Autopilot and Yaw Damper tests, the term engage autopilot will imply that both aileron and elevator engage switches must be engaged and the term disengage autopilot will imply that both aileron and elevator engage switches must be disengaged.

(c) Depress autopilot disconnect button on captain's control wheel. Observe that the ELEV engage switch disengages.
(d) Engage autopilot.
(e) Disengage AIL engage switch. Observe that pitch engage switch does not disengage.
(f) Open autopilot engage interlock circuit breaker (P6 load control center). Observe that ELEV engage switch disengages and warning lights illuminate and flash.
(g) Extinguish warning lights. Observe that the ELEV engage switch cannot be engaged.
(h) Close circuit breaker.
(i) Engage ELEV engage switch and repeat (f) thru (h) for pitch ac, dc and derived rate ac (when derived rate pitch channel installed) circuit breakers on the P6 load control center, air data computer No. 1 and vertical gyro No. 1 115-volt ac circuit breakers on the P18 load control center. Observe that the ELEV engage switch disengages (within 3 seconds when opening the pitch dc circuit breaker).
(j) Engage ELEV engage switch and open auxiliary vertical gyro circuit breaker. Observe that ELEV engage switch does not disengage.
(k) Close circuit breaker and open compass No. 2 circuit breaker. Observe that ELEV engage switch does not disengage. Close circuit breaker.
(l) Place AUX VERTICAL GYRO select switch in CAPT ON AUX position. Observe that ELEV engage switch disengages and that autopilot warning lights illuminate and flash.
(m) Extinguish lights by depressing first officer's light on right-hand panel and engage ELEV engage switch.
(n) Repeat (f) thru (h) for auxiliary vertical gyro circuit breaker (P18 load control center).
(o) Engage ELEV engage switch. Open vertical gyro No. 1 circuit breaker. Observe that ELEV engage switch does not disengage. Close circuit breaker.
(p) Engage ELEV engage switch and place the system select switch in AB position. Observe the ELEV engage switch disengages and that autopilot warning lights illuminate and flash. Observe that pitch axis cannot be engaged.

(q) Place system select switch in A position. Engage ELEV engage switch and place system select switch in B position. Observe that the ELEV engage switch disengages.

(r) Engage ELEV engage switch.

1) On airplanes with Honeywell (analog) ADC, select IAS SLEW on No. 1 air data computer and press toggle switch marked IAS-M for 10 seconds.
2) On airplanes with Kollsman (analog) ADC, press and hold IAS SLEW button for 10 seconds.
3) On airplanes with Honeywell (digital) ADC, set Test Select Switch to FAIL and press and hold PUSH TO TEST switch for 10 seconds.

(s) Pull and push either control column out of detent using a minimum of 10 pounds of force. Observe that pitch axis cannot be engaged.

(t) Engage autopilot.

(u) Extend flaps to 2 units and retract. Observe that ELEV engage switch does not disengage.

(v) Move stabilizer trim switch on captain's control wheel to NOSE UP trim. Observe that the ELEV engage switch disengages.

(w) Engage autopilot and repeat (v) for NOSE DOWN trim.

(x) Repeat step (v) and (w) for first officer's switch.

(y) Engage pitch axis. Move cruise trim cutout switch to CUTOUT. Observe that pitch axis disengages. Check that it is not possible to engage pitch axis. Return switch to NORMAL.

(z) Place system select switch in A position, and place VERTICAL GYRO switch in NORMAL position.

(aa) Check that system select switch is on A and FLT CONTROL A switch is on.

(ab) Position FLT CONTROL B switch to OFF and engage ELEV engage switch.

(ac) Move control column and observe that elevators respond to control wheel steering inputs.

(ad) Position FLT CONTROL B switch to ON and system select switch to B.

(ae) Position FLT CONTROL A switch to OFF and engage ELEV engage switch.

(af) Move control column and observe that elevators respond to control wheel steering inputs.

(ag) Position FLT CONTROL A switch to ON and system select switch to A.

(ah) Test flight control switches interlock.

1) Engage ELEV engage switch and position FLT CONTROL A switch to OFF. Observe ELEV engage switch disengages.
2) Return FLT CONTROL A switch to ON.
3) Position system select switch to B and engage ELEV engage switch.

4) Position FLT CONTROL A switch to OFF. Observe ELEV engage switch does not disengage.
5) Return FLT CONTROL A switch to ON.
6) Position FLT CONTROL B switch to OFF. Observe ELEV engage switch disengages.
7) Return FLT CONTROL B switch to ON.
8) Position system select switch to A and engage ELEV engage switch.
9) Position FLT CONTROL B switch to OFF. Observe ELEV engage switch does not disengage.
10) Position FLT CONTROL B switch to ON and disengage ELEV engage switch.

(4) Test Pitch Engage Interlocks (Effective on airplanes with B system only on autopilot control panel)
(a) Place system select switch in A position.
(b) Place ELEV engage switch in ENGAGED position.

<u>NOTE</u>: For the remainder of the Autopilot and Yaw Damper tests, the term engage autopilot will imply that both aileron and elevator engage switches must be engaged and the term disengage autopilot will imply that both aileron and elevator engage switches must be disengaged.

(c) Depress autopilot disconnect button on captain's control wheel. Observe that the ELEV engage switch disengages.
(d) Engage autopilot.
(e) Disengage AIL engage switch. Observe that pitch engage switch does not disengage.
(f) Open autopilot engage interlock circuit breaker (P6 load control center). Observe that ELEV engage switch disengages and warning lights illuminate and flash.
(g) Extinguish warning lights. Observe that the ELEV engage switch cannot be engaged.
(h) Close circuit breaker.
(i) Engage ELEV engage switch and repeat (f) thru (h) for pitch ac, pitch dc, and derived rate ac (when derived rate pitch channel is installed) circuit breakers on the P6 load control center, vertical gyro No. 1 and air data computer No. 1 115-volt ac circuit breakers on the P18 load control center. Observe that the ELEV engage switch disengages (within 3 seconds when opening the pitch dc circuit breaker).
(j) Engage ELEV engage switch and open auxiliary vertical gyro circuit breaker. Observe that ELEV engage switch does not disengage.
(k) Close circuit breaker.

(l) Place AUX VERTICAL GYRO select switch in CAPT ON AUX position. Observe that ELEV engage switch disengages and that autopilot warning lights illuminate and flash.
(m) Extinguish lights by depressing first officer's light on right-hand panel and engage ELEV engage switch.
(n) Repeat (f) thru (h) for auxiliary vertical gyro circuit breaker (P18 load control center).
(o) Engage ELEV engage switch. Open vertical gyro No. 1 circuit breaker. Observe that ELEV engage switch does not disengage. Close circuit breaker.
(p) Engage ELEV engage switch and place the system select switch in AB position. Observe the ELEV engage switch disengages and that autopilot warning lights illuminate and flash. Observe that pitch axis cannot be engaged.
(q) Place system select switch in B position. Engage ELEV engage switch and place system select switch in A position. Observe that the ELEV engage switch disengages and that autopilot warning lights illuminate and flash. Observe that pitch axis cannot be engaged.
(r) Place system selector in B position and engage ELEV engage switch. On airplanes with Honeywell (analog) ADC, select IAS SLEW on No. 1 air data computer and press toggle switch marked IAS-M for 10 seconds. On airplanes with Kollsman (analog) ADC, press and hold IAS SLEW button on the air data computer for 10 seconds. On airplanes with Honeywell (digital) ADC, set Test Select switch to FAIL and depress and hold PUSH TO TEST switch for 10 seconds. Observe that ELEV engage switch disengages.
(s) Pull and push either control column out of detent using a minimum of 10 pounds of force. Observe that pitch axis cannot be engaged.
(t) Engage autopilot.
(u) Extend flaps to 2 units and retract. Observe that ELEV engage switch does not disengage.
(v) Move stabilizer trim switch on captain's control wheel to NOSE UP trim. Observe that the ELEV engage switch disengages.
(w) Engage autopilot and repeat (v) for NOSE DOWN trim.
(x) Repeat step (v) and (w) for first officer's switch.
(y) Engage pitch axis. Move cruise trim cutout switch to CUTOUT. Observe that pitch axis disengages. Check that it is not possible to engage pitch axis. Return switch to NORMAL.
(z) Place VERTICAL GYRO switch in NORMAL position.
(aa) Check that system select switch is on B and FLT CONTROL A and B switches are on.
(ab) Position FLT CONTROL A switch to OFF and engage ELEV engage switch.

(ac) Move control column and observe that elevators respond to control wheel steering inputs.
(ad) Test flight control switches interlock.
1) Engage ELEV engage switch and position FLT CONTROL B switch to OFF. Observe ELEV engage switch disengages.
2) Return FLT CONTROL B switch to ON.

(5) Test VOR/LOC Mode Control
(a) Place system select switch in B position and engage autopilot.
(b) Place navigation (NAV) mode select switch (on autopilot control panel) in VOR/LOC position.
(c) Disengage AIL engage switch. Observe that NAV mode select switch returns to MAN position.
(d) Engage AIL engage switch and place NAV mode select switch in VOR/LOC position.
(e) Place NAV select switch in CAPT ON AUX position or BOTH ON -2 position (as applicable). Observe that NAV mode select switch returns to MAN position and can be rotated back to VOR/LOC position.
(f) Return NAV select switch to NORMAL or 1 (as applicable). Observe that NAV mode select switch returns to MAN position.
(g) Select an ILS frequency, place NAV mode select switch in VOR/LOC position, and turn control wheel out of detent to stop. Observe that NAV mode select switch returns to MAN position.
(h) Place NAV mode select switch in VOR/LOC position and actuate test switch on No. 1 NAV control panel (when test feature installed). Observe NAV mode select switch returns to MAN.
(i) Select an ILS frequency on No. 2 NAV control panel or auxiliary NAV control panel, if installed.
(j) Place NAV mode select switch in VOR/LOC position and actuate ILS test switch on No. 2 NAV control panel or auxiliary NAV control panel (when test feature installed). Observe the NAV mode select switch does not return to MAN.
(k) Place NAV select switch in BOTH ON NAV 2 or CAPT ON AUX, if auxiliary navigation receiver is installed.
(l) Repeat step (h) using No. 2 NAV control panel or auxiliary control panel as applicable.
(m) Return NAV select switch to NORMAL.
(n) Disengage autopilot.

(6) Test AUTO/APP Mode Control
(a) Place system select switch in B position and engage autopilot.
(b) Apply an ILS frequency and place NAV mode select switch in AUTO/APP position.
(c) Disengage ELEV engage switch. Observe that NAV mode select switch returns to MAN position. Engage ELEV engage switch.

(d) Place NAV mode select switch in AUTO/APP position and with a minimum of 6 pounds-feet of torque, turn either control wheel out of detent. Observe that NAV mode select switch returns to MAN position and cannot be rotated to AUTO/APP position.
(e) Repeat (d) using either control column and applying a minimum of 12 pounds of force.
(f) Place NAV mode select switch in AUTO/APP position and select a VOR frequency. Observe that NAV mode select switch returns to MAN position and cannot be rotated to AUTO/APP position.
(g) Place NAV mode select switch in AUTO/APP position and select an ILS frequency.
(h) Place pitch mode select switch (on autopilot control panel) in TURB position. Observe that NAV mode select switch returns to MAN position and cannot be rotated to AUTO/APP position.
(i) Return pitch mode select switch to OFF position.
(j) Place NAV mode select switch in AUTO/APP position and place NAV select switch in CAPT AUX position or BOTH ON -2 position (as applicable). Observe that NAV mode select switch returns to MAN position and that it can be rotated back to AUTO/APP position. Return NAV select switch to NORMAL or 1 position (as applicable).
(k) Place NAV mode select switch in AUTO/APP position.
(l) On airplanes with CADC, place selector switch (on face of No. 1 computer) in ALT MON position. Press toggle switch to MON for approximately 2 seconds and observe that the autopilot disengages.
(m) On airplanes with DADC, place selector switch (on face of No. 1 computer) in FAIL position. Depress PUSH TO TEST button for approximately 2 seconds and observe that the autopilot disengages.
(n) Return computer selector switch to normal operating position.

(7) Test MAN G/S Mode Control
(a) Select a ILS frequency, and repeat (6)(a) thru (c) with NAV mode select switch in MAN G/S position.
(b) Disengage autopilot.

(8) Test HDG OFF and HDG SEL Mode Control

NOTE: This test assumes zero VOR deviation signal and valid NAV superflag (as applicable).

(a) Apply a VOR frequency, place system select in B position and engage AIL engage switch.
(b) Engage HDG SEL switch.
(c) Disengage AIL engage switch. Observe that HDG SEL switch drops out to center position.
(d) Engage AIL engage switch and operate HDG SEL switch.

(e) Place NAV mode select switch in VOR/LOC position. Observe that HDG SEL switch drops out. Return NAV mode select switch to MAN.
(f) Engage HDG SEL switch.
(g) Turn either control wheel out of detent. Observe that HDG SEL switch drops out and that HDG SEL mode cannot be selected.
(h) Disengage autopilot.
(i) Engage AIL engage switch, engage HDG OFF switch, and then disengage AIL engage switch. Observe HDG OFF switch returns to center position.
(j) Engage AIL engage switch, engage HDG OFF switch, and place NAV mode select switch to VOR/LOC. Observe HDG OFF switch returns to center position.
(k) Place NAV mode select switch in MAN position and engage HDG OFF switch. Observe that rotating the control wheel out of detent requires 1.5 to 2.8 pound-feet of torque.

NOTE: Torque valves in excess of 2.8 pounds-feet may indicate an incorrectly installed aileron force limiter.

(l) Disengage autopilot.

(9) Test ALT HOLD Mode Control
(a) Place system select switch in B position and engage autopilot.
(b) Place pitch mode select switch in ALT HOLD position.
(c) Disengage ELEV engage switch. Observe that pitch mode select switch returns to OFF and that ALT HOLD mode cannot be selected.
(d) Engage ELEV engage switch and place pitch mode select switch in ALT HOLD position.
(e) Pull either control column out of detent. Observe that pitch mode select switch returns to OFF when there is 15 to 23 pounds of force on the control column.
(f) Place pitch mode select switch in ALT HOLD position.
(g) Place NAV mode select switch out of MAN position. Observe that pitch mode select switch returns to OFF when NAV mode select switch reaches MAN G/S position.
(h) Return NAV mode select switch to MAN position.
(i) Place pitch mode select switch in ALT HOLD position.
(j) On airplanes with CADC, place air data computer selector switch in ALT MON position. Depress air data computer ALT-SLEW test switch for approximately 2 seconds and observe that pitch mode select switch returns to OFF.
(k) On airplanes with DADC, place air data computer selector switch to FAIL position. Depress air data computer PUSH TO TEST switch for approximately 2 seconds and observe that pitch mode select switch returns to OFF.

(l) Return air data computer selector switch to normal operating position.
(m) Disengage autopilot.

(10) Test TURB Mode Control
(a) Place system select switch in B position and engage autopilot.
(b) Place pitch mode select switch in TURB position.
(c) Disengage ELEV engage switch. Observe that pitch mode select switch returns to OFF and the TURB mode cannot be selected.
(d) Engage ELEV engage switch. With either control wheel or control column out of detent, observe that TURB position can be selected.
(e) Place pitch mode select switch in TURB position. Observe that VOR/LOC position can be selected.
(f) Select an ILS frequency. Observe that NAV mode select switch returns to MAN position and the VOR/LOC mode cannot be reselected. Observe that HDG SEL or HDG OFF can be engaged.
(g) Disengage autopilot.

(11) Test Autopilot Annunciation
(a) Rotate roll control channel self-test switch through five test positions and then to OFF. Observe that autopilot warning lights illuminate steadily for each test position.
(b) Repeat (a) for pitch control channel.
(c) Open 28-volt dc autopilot warning light battery bus circuit breakers and depress captain's and first officer's warning lights. Observe that neither light illuminates.
(d) Close circuit breaker and press each light. Observe that lights flash when pressed.

NOTE: Pressing cap on A/P (right) side causes the annunciators to illuminate green. Pressing the cap on the F/D (left) side causes the applicable annunciators to illuminate amber.

(e) Depress captain's and then first officer's approach progress display indicator caps. Observe that VOR/LOC and GLIDE SLOPE lights illuminate evenly.
(f) Engage autopilot and open stab trim three-phase ac circuit breaker.
(g) Push either control column forward and release. Observe that STAB OUT OF TRIM light on center instrument panel illuminates after approximately 12 seconds.
(h) Place LIGHTS switch on center instrument panel to DIM position. Observe that STAB OUT OF TRIM light dims.
(i) Place LIGHTS switch to BRT position. Observe that STAB OUT OF TRIM light returns to original brightness.

(j) Disengage autopilot, and place the LIGHTS switch, located on center instrument panel, to DIM position. Observe the autopilot warning light flashes at reduced intensity.

(12) Autopilot Self-Test

(a) General

1) Steps for any one channel must be performed in sequence starting with the channel disengaged and the self-test switches in OFF position.
2) Autopilot warning lights must come on steadily when any self-test switch is in any position except OFF.
3) Do not apply rate gyro test signals for longer than 10 seconds. On airplanes with computers using derived rate circuits, ignore steps using rate gyro switch.
4) Open radio altimeter circuit breaker (as applicable).
5) Maintain air data computer pitot and static inputs at ambient pressure.

(b) Roll Control Channel Self-Test

STEP	ACTION	OBSERVATION
1)	Engage AIL engage switch and select a VOR frequency	
2)	Move rate gyro self-test switch to the left (-)	Control wheel moves counterclockwise, then returns to center position.
3)	Move rate gyro self-test switch to the right (+)	Control wheel moves clockwise, then returns to center position.
4)	Disengage AIL engage switch	
5)	Place self-test switch in OFF position	Self-test meter reads in RED range.
6)	Rotate self-test switch from OFF to position 1	Self-test meter slowly increased to YELLOW range, after 10 seconds in YELLOW range, rapidly decreases to zero, then rapidly increases with a reduction in rate of increase in the YELLOW range, then slowly and continuously increases to right stop.
7)	Rotate self-test switch from position 1 to 2	Self-test meter initially deflects to left, then settles in YELLOW range.

STEP	ACTION	OBSERVATION
8)	Rotate self-test switch from position 2 to 3	Self-test meter initially deflects to left, then settles in YELLOW range within 1 minute.
9)	Move rate gyro switch to left (-)	Self-test meter reading increases slightly.
10)	Move rate gyro switch to the right (+)	Self-test meter reading decreases slightly.
11)	Rotate self-test switch from position 3 to 4	Self-test meter initially deflects to zero then increases, momentarily settles in the YELLOW range, then slowly increases to right stop.
12)	Rotate self-test switch from position 4 to 5	Self-test meter initially deflects left, then increases with 2 rate changes. The second rate change should occur in the YELLOW range. Indicator then increases toward right stop.
13)	Rotate self-test to OFF	

(c) Pitch Control Channel Self-Test

STEP	ACTION	OBSERVATION
1)	With self-test switch in OFF position	Meter reads in RED range
2)	Rotate self-test switch from OFF to position 1	Meter moves to right at a constant rate for approximately 10 seconds. Rate of movement changes in the YELLOW range.
3)	Rotate self-test switch from position 1 to 2	Self-test meter initially deflects to right stop, then decreases and settles in YELLOW range. range
4)	Move rate gyro switch to the right (+)	Self-test meter deflects right momentarily.
5)	Move rate gyro switch to the left (-)	Self-test meter deflects left momentarily.
6)	Rotate self-test switch from position 2 to 3	Self-test meter initially deflects left, then increases, with a decreasing rate in the YELLOW range, to the right stop.
7)	Rotate self-test switch from position 3 to 4	Self-test meter initially reads zero for approximately 10 seconds, increases, then peaks in the GREEN range for approximately 5 seconds, then slowly decreases left for 110 to 170 seconds.
8)	Rotate self-test switch from position 4 to 5	Self-test meter settles in the YELLOW range.

STEP	ACTION	OBSERVATION
9)	Rotate self-test switch to OFF position	Meter reads in RED range.
10)	Engage autopilot	
11)	Move rate gyro switch to right (+)	Control column moves aft and stops at neutral.
12)	Move rate gyro switch to left (-)	Control column moves forward and stops at neutral.

(13) Test roll axis manual mode.
  (a) Test wings leveling.
    1) Ground roll TP-11, roll TP-13, roll TP-16, and roll TP-18, and center control wheel. Tilt the vertical gyro for 6.5 degrees of right roll.
    2) Engage the AIL engage switch. Observe the control wheel remains centered.
    3) Disengage the AIL engage switch, tilt the vertical gyro for 3.5 degrees of right roll, and engage the AIL engage switch. Observe the control wheel rotates counterclockwise.
    4) Disengage the AIL engage switch.
    5) Repeat 2) thru 4) for left roll and clockwise wheel rotation.
    6) Remove grounds and level the vertical gyro.
  (b) Test Aileron Gain
    1) With control wheel centered, engage AIL engage switch and ground roll TP-11, roll TP-13, and roll TP-18.
    2) With control wheel centered, tilt No. 1 vertical gyro approximately 2 degrees to simulate a right roll until control wheel is rotated counterclockwise 15 degrees. Observe that required gyro tilt is not less than 1.3 degrees and not more than 2.9 degrees.

3) Relevel vertical gyro, disengage AIL engage switch, and remove ground from roll TP-11.
4) Repeat 1) thru 3) with No. 1 vertical gyro tilted to simulate a left roll.
5) On airplanes with A and B system switching on autopilot control panel, place system select switch in A position and repeat 1) and 2).
6) Disengage AIL engage switch, remove grounds, level vertical gyro and place system select switch in B position.

(c) Test Heading Hold (compass heading displacement)
1) Ground roll TP-13, engage AIL engage switch and ground roll TP-11.
2) Increase compass heading by 10 degrees. Observe that control wheel rotates counterclockwise.
3) Ground roll TP-14.
4) Adjust No. 1 vertical gyro to center wheel. Ensure that vertical gyro bank angle is not less than 7.5 degrees and not more than 12.5 degrees.
5) Disengage AIL engage switch, remove grounds, level vertical gyro, and reset heading.
6) Repeat 1) thru 5) for 10-degree decrease in heading.

(d) Test CWS Gain
1) Remove hydraulic pressure from control surfaces.
2) Apply 7.5 pound-feet of torque in a clockwise direction to the captain's control wheel. Observe voltage at TP-18 of the roll channel is no less than 4.00 and no greater than 6.00 volts ac.
3) Repeat step 2) in a counterclockwise direction.
4) Reapply hydraulic power to control surfaces.

(e) Test Aileron Camout Force
1) Slowly apply a clockwise torque to control wheel until wheel begins to move.
2) Record torque value. Repeat 1) for counterclockwise torque.
3) Engage AIL engage switch and ground roll TP-11 and roll TP-14.

4) Maintaining control wheel position at neutral (in detent) with torque wrench, slowly tilt vertical gyro for a left bank. Observe applied torque value builds up to a maximum value and then levels off. Record maximum value. The combined values of 2) and recorded maximum value shall be greater than 7.7 pounds-feet but less than 12.8 pounds-feet. Relevel the gyro.
5) Maintaining control wheel position at neutral (in detent) with torque wrench, slowly tilt vertical gyro for a right bank. Observe applied torque value builds up to maximum value and then levels off. The combined values of 2) and the recorded maximum value shall be greater than 7.7 pounds-feet but less than 12.8 pounds-feet.
6) Disengage AIL engage switch, remove grounds, and level vertical gyro.
7) Place system select switch in A position and repeat 3) thru 6) on airplanes with A and B system switching on autopilot control panel.
8) Place system select switch in B position.

(f) Test Aileron Force Limiter
1) Engage AIL engage switch.
2) Tilt vertical gyro to simulate a 30-degree right bank. Observe that control wheel rotates counterclockwise and stops between 24 and 28 degrees.

NOTE: If control wheel stops between 18 to 24 degrees, disconnect CWS Force Transducer at D397. Apply a torque wrench at the control wheel and apply up to 10 inch-pounds maximum. If the control wheel stops between 24 to 28 degrees, the test is acceptable.

3) Level vertical gyro and disengage AIL engage switch.
4) Repeat 1) thru 3) for opposite direction (left bank).
5) Place system select switch in A position and repeat 1) thru 4) on airplanes with A and B system switching on autopilot control panel.
6) Place system select switch in B position.

(g) Test Heading Off
1) Ground roll TP-11, TP-13, and TP-18, engage AIL engage switch, and increase the compass heading by 3.5 degrees. Observe the control wheels rotate counterclockwise.
2) Engage HDG OFF switch and return compass heading to 0 degrees. Observe the control wheels do not move.
3) Tilt the vertical gyro to center the wheel. Observe 2.1 to 3.9 degrees of left roll is required to center the wheel.
4) Place HDG OFF switch to center position. Observe the control wheel rotates clockwise.
5) Disengage AIL engage switch, remove grounds, and level gyro.

(h) Test Derived Rate (Roll computers without rate gyro test switch)
1) Ground roll TP-15, and TP-18, engage AIL engage switch, and tilt vertical gyro approximately 5 degrees right roll. Observe that control wheel rotates left then returns to center.
2) Return vertical gyro to level. Observe that control wheel rotates right then returns to center.
3) Disengage AIL engage switch and remove grounds.

(14) Test HDG SEL Mode
(a) Test Preset Heading Gain
1) Engage AIL engage switch and ground roll TP-11.
2) Engage HDG SEL switch.
3) Move preset heading bug 10 degrees to right of airplane heading. Observe that control wheel turns clockwise and stops.
4) Tilt No. 1 vertical gyro to simulate a right roll until control wheel is centered. Observe that the required gyro tilt is not less than 7.0 degrees and not more than 13.5 degrees.
5) Relevel vertical gyro, disengage AIL engage switch, remove ground from roll TP-11.
6) Apply 350 KIAS and repeat 1) thru 4) moving preset heading bug 5 degrees. Observe that required gyro tilt is not less than 13 degrees and not more than 21 degrees. Apply 120 KIAS.

(15) Test VOR/LOC Mode
(a) Test Lateral Beam Sensor
1) Apply 1-dot fly-right deviation signal.
2) Place NAV mode select switch in VOR/LOC position. Observe that VOR/LOC approach progress display indicators illuminate amber (VOR ARM).

3) Fail the VOR/LOC superflag and decrease radio signal to zero. Observe that VOR/LOC approach progress display indicators illuminate amber (VOR ARM).
4) Restore superflag. Observe approach progress display indicators illuminate green (engaged).
5) Fail the VOR/LOC superflag. Observe approach progress display indicator remains illuminated green. Disengage AIL engage switch.

(b) Test VOR Displacement Gain
1) Engage AIL engage switch, ground roll TP-9 (erection cutoff) and place NAV mode select switch in VOR/LOC position.
2) Ground roll TP-11, roll TP-13 and roll TP-5 (gain programmer output).
3) Set course selector to airplane heading (0 degree of wheel displacement).
4) Apply 1/2-dot of fly-right VOR deviation signal.
5) Remove ground from roll TP-5. Observe that control wheel turns clockwise.
6) Tilt vertical gyro to center control wheel. Observe that required gyro tilt is not less than 12 degrees and not more than 22 degrees of right roll.
7) Remove ground from TP-9. Turn the control wheel out of detent. Observe the mode selector switch returns to MAN.
8) Disengage AIL engage switch.
9) Remove grounds, reduce VOR deviation signal to zero, and level vertical gyro.

(16) Test ILS Mode
(a) Test Heading Hold and Lateral Beam Sensor (150 mv)
1) Engage AIL engage switch, apply an ILS voltage using the radio simulator, and fail the NAV superflag.
2) Apply 2-dots or more fly-right deviation signal.
3) Place NAV mode select switch in VOR/LOC position. Observe that VOR/LOC approach progress display indicators illuminate amber (LOC ARM).
4) Ground roll TP-11 and engage HDG SEL switch. Observe that control wheel responds to heading selector inputs.
5) Set heading selector to airplane heading (center the control wheel) and ground roll TP-13.
6) Decrease deviation input to 1/2-dot fly-right. Observe that approach progress display VOR/LOC indicator remains amber.

7) Restore the NAV superflag. Observe the approach progress display changes to VOR/LOC engaged (green), HDG SEL switch returns to center position and the control wheel rotates clockwise. Observe that not less than 4 degrees and not more than 12 degrees of left course selector command are required to center the control wheel.
8) Disengage AIL engage switch, remove grounds, remove ILS voltage and remove radio signal.

(b) Test Gain Programmer
1) Engage autopilot, set ILS deviation to zero, and ground roll TP-10, roll TP-11, and roll TP-13.
2) Using radio altimeter test set, adjust radio altimeter altitude to 2000 feet, apply ILS voltage, and place NAV mode select switch to AUTO/APP position.
3) Adjust LOC deviation for 1/2-dot fly-right. Center the control wheel with the vertical gyro.
4) Decrease radio altimeter to 500 feet. Observe that control wheel rotates counterclockwise.
5) Disengage autopilot.
6) Repeat steps 1) thru 3) and open radio altimeter No. 1 circuit breaker. Observe that the control wheel rotates counterclockwise.
7) Disengage autopilot, remove grounds and remove ILS voltage. Close circuit breaker and adjust simulated altitude to 2000 feet.

(17) Test Pitch Control Wheel Steering

NOTE: Protractors are not required for pitch channel tests. They should be removed from the control wheel and column.

Check, and adjust if necessary, null voltage of captain's and first officer's pitch transducers before conducting pitch control wheel steering tests (AMM 22-11-91).

Set stabilizer to 16 units.

(a) Test CWS Detent
1) Engage ELEV engage switch.
2) Slowly push captain's control column until column starts to move. Observe that 3.8 to 7.5 pounds of force are required. Record force value.
3) Repeat 2) by pulling on control column. Observe that the difference in required force is less than 2 pounds.
4) Repeat 2) and 3) for flight officer's column.
5) Disengage autopilot.

(b) Test CWS Gain and Camout
1) Slowly push control column until column starts to move. Record required force.

2) Repeat 1) for pull on control column.
3) Engage ELEV engage switch and ground pitch TP-6. Cycle column back and forth 15 or 20 times leaving it in a vertical position.
4) Apply 14 pounds of pulling force on captain's control column. Observe that voltage at pitch TP-13 is 1.3 to 2.4 volt ac.
5) Repeat 4) for control column push.
6) Repeat 4) and 5) for first officer's control column.
7) Apply 350 KIAS and increase altitude to 20,000 feet.
8) Repeat 4). Observe that voltage at pitch TP-13 is 0.43 to 0.80 volt ac.
9) Decrease airspeed to 120 KIAS and altitude to ambient altitude.
10) Remove ground and disengage ELEV engage switch.
11) Using spring scale, maintain force on control column to prevent it from moving. Engage ELEV engage switch and ground pitch TP-6.
12) Apply noseup tilt to vertical gyro until force on spring scale levels out. Disengage ELEV engage switch, level the vertical gyro, and remove ground. Observe that force required in step 1) plus the force in this step is greater than 20 pounds and less than 26 pounds.
13) Repeat 11) and 12) for nosedown vertical gyro tilt. Observe that required force plus the force in 2) is greater than 19 pounds and less than 27 pounds.
14) Place system select switch in A position and repeat 11) thru 13) on airplanes with A and B system selection on autopilot control panel.
15) Disengage autopilot, remove ground, and level vertical gyro.
16) Place system select switch in B position.

(c) Test Pitch Synchronization Prior to Engage
1) Ground pitch TP-6 and tilt the vertical gyro 5 degrees in nosedown direction.
2) Wait 15 seconds, then attempt to engage the ELEV engage switch. Observe that the ELEV engage switch does not engage.
3) Remove the ground from pitch TP-6.
4) Wait 10 seconds, then attempt to engage the ELEV engage switch. Observe that the ELEV engage switch does engage.
5) Disengage the ELEV engage switch, remove ground, and level the vertical gyro.

(18) Test Attitude Hold
   (a) Test Q Control and Turbulence
      1) Engage ELEV engage switch, ground pitch TP-6, and observe control column in neutral position.
      2) Apply 2 degrees noseup tilt to vertical gyro. Observe that control column moves forward. Record displacement.
      3) Increase airspeed to 350 KIAS. Observe that control column moves aft to approximately 60 to 80% of its displacement in 2).
      4) Decrease airspeed to 120 KIAS. Observe that control column moves forward.
      5) Place pitch mode select switch in TURB position. Observe that control column moves aft to approximately 40 to 70% of its displacement in 2) within 4 seconds.
      6) Disengage autopilot, remove ground and level vertical gyro.
   (b) Test Altitude Hold Versine
      1) Engage autopilot and observe control column in neutral position, and verify vertical gyro is level.
      2) Engage HDG SEL switch. Set heading selector for a 45-degree increase in heading. Observe that control column moves aft.
      3) Slowly increase airspeed to 350 KIAS. Observe that control column moves forward.
      4) Disengage autopilot, level vertical gyro, decrease airspeed to 120 KIAS and return heading selector to zero.
   (c) Test Derived Rate (Pitch computers without rate gyro test switch)
      1) Ground pitch TP-12, engage ELEV engage switch, and tilt vertical gyro approximately 5 degrees noseup. Observe that control column moves forward then returns to neutral.
      2) Return vertical gyro to level. Observe that control column moves aft then returns to neutral.
      3) Disengage ELEV engage switch and remove ground.
   (d) Test Altitude Displacement Gain and Limit
      1) Engage ELEV engage switch and ground pitch TP-6.

         NOTE: Observe control column in neutral before each sequence.

      2) Set altitude at 500 feet above ambient.
      3) Place pitch mode select switch in ALT HOLD position and increase altitude to 580 feet. Observe that control column moves forward, and that 2.5 to 4.2 degrees of nosedown gyro tilt centers control column.

4) Decrease altitude to 420 feet. Observe that control column moves aft, and that 2.5 to 4.2 degrees noseup gyro tilt centers control column.
5) Disengage autopilot, level vertical gyro, and remove ground.

(19) Test Approach Mode

(a) Test Glide Slope Annunciator

1) Engage autopilot ground pitch TP-6, place radio simulator switch to ILS position.
2) Apply one-dot fly-up glide slope deviation signal.
3) Fail the glide slope superflag and place NAV mode select switch in AUTO/APP position. Observe that approach progress display GLIDE SLOPE annunciator illuminates amber (G/S ARM).
4) Decrease deviation signal to zero. Observe that approach progress display GLIDE SLOPE annunciator remains illuminated amber (G/S ARM).
5) Restore the glide slope superflag. Observe that approach progress display GLIDE SLOPE annunciator illuminates green (G/S ENG).
6) Fail the glide slope superflag. Observe that approach progress display remains green (G/S ENG).
7) Restore the glide slope superflag and fail the NAV superflag. Observe the AIL engage switch does not disengage.
8) Disengage autopilot, remove ground and remove radio signal.

(b) Test Altitude Rate Gain

1) Engage autopilot and observe control column in neutral.
2) On airplanes with CADC, set in a 600 feet per minute rate of descent on the ADC by depressing 600 FPM pushbutton or toggle (as applicable) on face of computer. On airplanes with DADC, set air data computer selector switch to SLEW position. Depress (and hold down) air data computer PUSH TO TEST switch.
3) Place NAV mode select switch in MAN G/S position. Observe that after 15 seconds, from 0 to 2 degrees of vertical gyro is required to center control column.
4) Disengage autopilot and level vertical gyro.
5) Return ADC to normal (release button or toggle).

(c) Test Beam Displacement Gain

1) Engage autopilot, set radio altimeter to 2000 (if installed), ground pitch TP-6, and observe control column in neutral.
2) Place NAV mode select switch in MAN G/S position.

3) Wait 15 seconds and apply 1/4-dot fly-up glide slope signal. Observe that 2.2 to 6.0 degrees of noseup gyro tilt centers control column.
4) Disengage autopilot, remove ground, level vertical gyro and remove glide slope signal.

(20) Test Stabilizer Trim

(a) Test Trim Threshold

NOTE: The mach trim actuator must be fully extended for this test (AMM 22-21-11/501).

1) Set 13 units of trim on stabilizer trim wheel and open stabilizer trim three-phase ac circuit breaker. Move control column back and forth through neutral several times.
   a) Engage pitch axis. Hand crank the stabilizer to 3 units of trim. Check that stabilizer out-of-trim warning light does not illuminate.
   b) Use the control wheel stabilizer trim (pickle) switch to return stabilizer to 13 units of trim.
2) Apply an airspeed of 350 KIAS, engage the autopilot, ground pitch TP-6, and with a VTVM observe and note ac voltage at pitch TP-10.
3) Slowly apply nosedown tilt to vertical gyro until 28 volts dc appears at pitch TP-4. With a VTVM, monitor changing ac voltage at pitch TP-10 for a minimum and final maximum voltage while tilting the vertical gyro. Observe sum of reducing voltage added to increasing voltage (which is total voltage change) is 2.2 ±0.6 volts ac.

   NOTE: The total voltage change must be measured. Going from 0.5 volt ac to 0.0 and from 0.0 to 1.8 volts ac is a total change of 2.3 volts ac.

4) Slowly apply noseup tilt to vertical gyro until 28 volts dc disappears at pitch TP-4. Measure and record ac voltage at pitch TP-10. The change between this voltage and voltage noted in step 2) should be between 50 and 90% of the voltage change noted in step 3). Disengage autopilot. Remove ground from pitch TP-6.
5) Repeat steps 2) thru 4) for a noseup vertical gyro tilt while monitoring 28 volts at pitch TP-3. Observe total voltage change between step 2) and 3) is 2.2 ±0.6 volts ac.
6) Set 3 units of trim on stabilizer trim wheel and ensure that stabilizer trim three-phase ac circuit breaker is open.

7) Move control column through neutral several times; engage autopilot, and ground pitch TP-6.
8) Observe voltage at pitch TP-10 is 1.75 +1.0/-0.5 volts ac and record.
9) Ground pitch TP-15. Observe voltage at pitch TP-10 decreases to 0.0 ±0.5 volt ac.
10) Disengage autopilot and remove grounds from pitch TP-6 and TP-15.
11) Move column back and forth through neutral several times, engage autopilot, and ground pitch TP-6 and TP-12. Observe and record ac voltage at pitch TP-10.
12) Engage HDG SEL and slowly move heading selector knob for increasing heading until 28 volts dc appears at pitch TP-4. Observe voltage change at pitch TP-10 from value recorded in 11) is 0.7 ±0.2 volt ac.
13) Slowly move heading selector knob toward airplane heading, and record voltage at pitch TP-10 when voltage at pitch TP-4 nulls. Observe that voltage change between this voltage and that recorded in 11) is between 50 and 90% of voltage in 12).
14) Return heading selector to the airplane heading.
15) Disengage autopilot, center stabilizer, close circuit breaker, remove grounds, level vertical gyro and set airspeed to 120 KIAS.

(b) Test Trim Monitor
1) Engage autopilot and ground pitch TP-6.
2) Open stabilizer trim three-phase ac circuit breaker.
3) Apply 3 degrees noseup tilt to vertical gyro. Observe that STAB OUT OF TRIM light on center instrument panel illuminates after 8 to 16 seconds.
4) Disengage autopilot.
5) Engage autopilot and repeat 3) for nosedown vertical gyro.
6) Disengage autopilot, remove ground, close stabilizer trim three-phase ac circuit breaker, and position stabilizer to zero degree trim using trim switch on captain's control wheel.

(c) Test Trim Speeds

NOTE: 1 unit of trim equals 9.5 revolutions of trim wheel.

1) Manually trim the stabilizer to its maximum leading edge down (airplane noseup) position. Set flaps to 2 units. Engage ELEV engage switch. Ground TP-6 on the pitch channel. Close autopilot 3 phase stabilizer trim circuit breaker. Tilt vertical gyro approximately 2 degrees nosedown. Observe that stabilizer does not move.

2) Tilt vertical gyro approximately 2 degrees noseup and observe that stabilizer leading edge moves up (airplane nosedown) at a rate of 1 unit of trim in 4 to 7 seconds. Tilt vertical gyro in opposite direction until trim direction reverses. Allow stabilizer to run until it is automatically stopped by limit switches. Disengage ELEV engage switch. Check that dimension B is between 26.90 and 27.26 inches.
3) Engage ELEV engage switch. Tilt vertical gyro 2 degrees noseup. Observe that stabilizer trims in a leading edge up (airplane nosedown) direction at a rate of 1 unit in 4 to 7 seconds. Let stabilizer run until it is automatically stopped by limit switches. Disengage ELEV engage switch. Check that dimension B is between 42.47 and 42.83 inches.
4) Set flaps to 0 unit. Engage ELEV engage switch. Hand crank stabilizer to airplane nosedown limit.
5) Tilt vertical gyro approximately 2 degrees nosedown. Observe that stabilizer moves in the leading edge down (airplane noseup) direction at a rate of 1 unit in 30 to 40 seconds.
6) Tilt vertical gyro in either noseup or nosedown direction. With stabilizer trim operating, place autopilot trim cutout switch in CUTOUT position. Observe that stabilizer stops immediately and does not operate when vertical gyro is tilted in either direction. Observe that pitch axis disengages. Return autopilot trim cutout switch to NORMAL position. Engage ELEV engage switch and check that stabilizer operates when vertical gyro is tilted.
7) Disengage ELEV engage switch, remove grounds and level gyro.

D. Restore Airplane to Normal Configuration
   (1) Disengage autopilot.
   (2) Remove vertical gyro(s) from tilt table (Ref Chapter 34, Removal/Installation).
   (3) Turn off all systems.
   (4) Disconnect and remove altimeter test set.
   (5) Disconnect pressure source from pitot static system (Ref Chapter 34, Air Data Instruments).
   (6) Remove radio signal generators.
   (7) Remove adapter assembly and restore control wheel hubs to normal configuration.
   (8) Determine whether there is further need for electrical or hydraulic power on airplanes; if not, remove power.

3. Yaw Damper System Test (All except coupler P/N 4084042)

NOTE: For aircraft with coupler P/N 4084042 installed (incorporating SB 27A1206), see 22-12-01

A. Equipment and Materials
  (1) Portable pressure equipment to simulate 0 to 35,000 feet altitude and 0- to 400-knot indicated airspeed (KIAS) type MB-1 Pitot Static System Tester, MIL-T-8076, Aircraft Products Part Number 381-100 or equivalent
  (2) DC Volt-Ohmmeter, Simpson Model 260 or equivalent (with test leads and adapter to fit test points on face of yaw damper coupler)

B. Prepare to Test Yaw Damper System
  (1) Connect pitot static pressure test equipment to air data computer pitot and static sources as required. Maintain ambient static pressure and pitot pressure to produce 120 KIAS.

CAUTION: INSTRUMENT DAMAGE MAY RESULT IF PRESSURE TEST EQUIPMENT IS NOT CONNECTED PROPERLY. REFER TO CHAPTER 34, FOR DETAILED INSTRUCTIONS.

  (2) Provide electrical power to airplane and energize load control center P6.
  (3) Energize hydraulic systems A and B and ensure flight control system A and B switches are positioned to ON.

WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

  (4) Ensure following circuit breakers on load control center P6 are closed.
    (a) MASTER CAUTION BUS (two)
    (b) MASTER DIMMING BUS INDICATOR LIGHTS (nine)
    (c) DIM AND TEST
    (d) YAW DAMPER AC
    (e) YAW DAMPER DC
    (f) ENG INTLK

C. Test Yaw Damper System
  (1) Test Yaw Damper Indicator Lights
    (a) Observe that YAW DAMPER warning light located on overhead panel and two MASTER CAUTION lights and FLT CONT annunciator light located on light shield are illuminated.

(b) Place LIGHTS switch on center instrument panel first in DIM position then in BRT position. Observe that YAW DAMPER warning light dims and then illuminates brightly.
(c) Depress either MASTER CAUTION light. Observe that both MASTER CAUTION lights and FLT CONT annunciator light go out and YAW DAMPER warning light remains illuminated.
(d) Momentarily depress annunciator light. Observe that both MASTER CAUTION lights and FLT CONT annunciator light comes on.

NOTE: Disregard other master caution annunciator lights that illuminate.

(2) Test Yaw Damper Interlocks
(a) Place system select switch on autopilot control panel to B and engage YAW DAMPER engage switch. Observe that YAW DAMPER warning light extinguishes after approximately 2 seconds and both MASTER CAUTION lights extinguish.
(b) Open YAW DAMPER AC circuit breaker. Observe that YAW DAMPER warning light illuminates, YAW DAMPER engage switch disengages after approximately 2 seconds, and rudder does not deflect more than 1/2 degree.
(c) Close YAW DAMPER AC circuit breaker and engage YAW DAMPER engage switch. Within 2 seconds momentarily open and close ac circuit breaker. Observe that engage switch remains engaged, YAW DAMPER warning light illuminates immediately and remains on for approximately 2 seconds.
(d) On airplanes with A and B system on autopilot control panel, rotate system select switch to A position. Observe that YAW DAMPER engage switch disengages.
(e) On airplanes with A and B system on autopilot control panel, engage YAW DAMPER engage switch and return system select switch to B position. Observe that YAW DAMPER engage switch disengages.
(3) Test Flight Control Switches Interlock (Effective on airplanes with A and B system switching on autopilot control panel)
(a) Rotate system select switch to A position and engage YAW DAMPER engage switch and position FLT CONTROL A switch to OFF. Observe that YAW DAMPER engage switch disengages.
(b) Return FLT CONTROL A switch to ON.
(c) Position system select switch to B and engage YAW DAMPER engage switch.
(d) Position FLT CONTROL A switch to OFF. Observe that YAW DAMPER engage switch does not disengage.
(e) Return FLT CONTROL A switch to ON.
(f) Position FLT CONTROL B switch to OFF. Observe that YAW DAMPER engage switch disengages.

(g) Return FLT CONTROL B switch to ON.
(h) Position system select switch to A and engage YAW DAMPER engage switch.
(i) Position FLT CONTROL B switch to OFF. Observe that YAW DAMPER engage switch does not disengage.
(j) Position FLT CONTROL B switch to ON and disengage yaw damper system.

(4) Test Flight Control Switches Interlock (Effective on airplanes with B system only on autopilot control panel)
(a) Rotate system select switch to B position and engage YAW DAMPER engage switch and position FLT CONTROL B switch to OFF. Observe that YAW DAMPER engage switch disengages.
(b) Return FLT CONTROL B switch to ON.
(c) Engage YAW DAMPER engage switch.
(d) Position FLT CONTROL A switch to OFF. Observe that YAW DAMPER engage switch does not disengage.
(e) Return FLT CONTROL A switch to ON.
(f) Disengage yaw damper system.

(5) Test Yaw Damper System Response
(a) Station observer facing tail of airplane to monitor rudder deflection.
(b) Position system select switch to B and engage YAW DAMPER engage switch.
(c) Place and hold YAW DAMPER TEST switch, on center instrument panel, in R (right) position. Observe that yaw damper indicator deflects right momentarily and returns to center (zero) and rudder momentarily deflects right and returns to center.
(d) Release YAW DAMPER TEST switch. Observe that rudder momentarily deflects left and returns to center.
(e) Place and hold YAW DAMPER TEST switch in L (left) position. Observe that yaw damper indicator deflects left momentarily and returns to center (zero) and rudder momentarily deflects left and returns to zero.
(f) Release YAW DAMPER TEST switch. Observe that rudder momentarily deflects right and returns to center.
(g) On airplanes with A and B system on autopilot control panel, repeat (a) thru (f) with system select switch in A position.

(6) Test Yaw Damper Continuity
(a) Position system select switch to B and engage yaw damper. Position FLT CONTROL A switch to OFF and repeat steps (5)b) thru (5)f). Observe that results are satisfactory.
(b) Position FLT CONTROL A switch to ON and position FLT CONTROL B switch to OFF. Observe YAW DAMPER disengage switch disengages.

(c) On airplanes with A and B system on autopilot control panel, position system select switch to A and engage YAW DAMPER engage switch. Repeat steps (4)c) thru (4)f). Observe that results are satisfactory.

(d) On airplanes with A and B system on autopilot control panel, position FLT CONTROL B switch to ON and position system select switch to B. Observe that YAW DAMPER engage switch disengages.

(7) Test Yaw Damper System Gain (Coupler P/N 2588880)

(a) With system select switch in B position and YAW DAMPER engage switch engaged, jumper yaw TP-9 (rate gyro washout input) to yaw TP-10 (rate gyro washout output). Ground yaw TP-6 (integrator output).

(b) Place and hold YAW DAMPER TEST switch to R (right) position. Observe that voltage across yaw TP-4 and yaw TP-5 (LVDT) is 2.3 ±0.8 volts ac and that indicator shows right rudder deflection.

(c) Release TEST switch and wait 30 seconds.

(d) Repeat (b) for left rudder. Observe that voltage difference for left and right rudder is not greater than 0.8 volt ac.

(e) Using pitot static source, increase airspeed to 350 KIAS.

(f) Place and hold YAW DAMPER TEST switch to R (right) position. Observe that voltage across yaw TP-4 and yaw TP-5 is 0.92 ±0.32 volts ac and that indicator shows right rudder deflection.

(g) Release TEST switch and wait 30 seconds.

(h) Repeat (f) for left rudder. Observe that voltage difference for left and right rudder is not greater than 0.32 volt ac.

(i) Decrease airspeed to 120 KIAS and disengage yaw damper.

(j) On airplanes with A and B system on autopilot control panel, place system select switch in B position and engage YAW DAMPER engage switch. Repeat (b) thru (d).

(k) Disengage yaw damper, place system select switch in B position, remove jumper and ground.

(8) Yaw Damper Self-Test (Coupler P/N 2588880)

(a) General

1) Ensure that line voltages are 115 volts ac ±5% and 28 volts dc ±5% and air data computer is at zero airspeed.

2) Observe that autopilot disengage warning lights illuminate steadily when rate gyro test switch is moved or when self-test switch is in any position except off.

3) Perform steps in sequence.

4) Ensure that yaw damper engage switch is in OFF position for self-test positions 1, 2, and 3.

(b) Self-Test

STEP	ACTION	OBSERVATION
1)	Place self-test switch in OFF position	Meter reads in RED range
2)	Place self-test switch in position 1	Meter reads in GREEN range
3)	Place self-test switch in position 2	Meter reads in YELLOW range
4)	Place self-test switch in position 3	Deleted
5)	Move rate gyro self-test lever to the right (+) and then left (-)	Meter reads in YELLOW range for either position
6)	Place self-test switch in position 4 and place YAW DAMPER engage switch to ON	Meter reads between left stop and left of REMOVED range
7)	Place self-test switch in position 5 and place YAW DAMPER engage switch to ON	Meter reads in YELLOW range
8)	Return self-test switch to OFF position	

(9) Yaw Damper Self Test (Coupler P/N 4030952)

(a) General

1) Line voltage must be 115 ±5.8 volts ac and 28 ±1.4 volts dc.
2) Autopilot warning lights will come on steady when the BITE switch is positioned to ARM.
3) Airspeed must be at zero knot.
4) When self-testing, the yaw damper is engaged unless stated otherwise during the test. Steps (tests) do not have to be completed in sequence unless stated otherwise in a test. When a step says "repeat" or a particular test is desired, cycle BITE switch to off then to ARM and press GO switch repeatedly until one less than desired test is indicated; allow to run to completion; ignore fail indication and then press GO to advance to desired test.

(b) Self-Test (Coupler P/N 4030952-902)

STEP			TEST NO.	OBSERVE LIGHTS PASS	FAIL
1)	BITE Self-Test Arm				
	a)	Ensure yaw damper is disengaged.			
	b)	Set BITE switch to ARM.		ON	OFF
	c)	Press LAMP TEST switch and hold.	8	ON	ON
	d)	Release LAMP TEST switch.	0	ON	OFF
2)	Test 1 - Rate Gyro Filter				
	a)	Engage yaw damper and ensure hydraulics are on.			
	b)	Press GO switch. Observe after 1 second.	1	ON	OFF
3)	Test 2 - Engage Null				
	a)	Engage yaw damper and ensure hydraulics are on (from Test 1).			
	b)	Press GO switch. Observe after 6 seconds.	2	ON	OFF

STEP			TEST NO.	LIGHTS PASS	LIGHTS FAIL
4)	Test 3 - Rate to Rudder (Displacement)				
	a)	Ensure yaw damper is engaged and hydraulics are on (from Test 2).			
	b)	Press GO switch. Observe after 1 second.	3	ON	OFF
5)	Test 4 - Rate to Rudder (Integral)				
	a)	Ensure yaw damper is engaged and hydraulics are on (from Test 3).			
	b)	Press GO switch. Observe after 40 seconds.	4	ON	OFF
6)	Test 5 - Trim Meter				
	a)	Ensure yaw damper is engaged and hydraulics are on (from Test 4).			
	b)	Press GO switch. Observe after 8 seconds.	5	ON	OFF
7)	Test 6 - Disengage Synchronization				
	a)	Disengage yaw dampers and ensure hydraulics are off.			
	b)	Press GO switch. Observe after 6 seconds.	6	ON	OFF
8)	Test 7 - Rate Gyro				
	a)	Ensure yaw dampers are disengaged and hydraulics are off (from Test 6).			
	b)	Press GO switch. Observe after 1 second.	7	ON	OFF
9)	Return BITE switch to OFF.				

(c) Self-Test (Coupler P/N 4030952-906)

STEP			OBSERVE		
			TEST NO.	LIGHTS	
				PASS	FAIL
1)	BITE Self-Test Arm				
	a)	Ensure yaw damper is disengaged.			
	b)	Set BITE switch to ARM.		ON	OFF
	c)	Press LAMP TEST switch and hold.	8	ON	ON
	d)	Release LAMP TEST switch.	0	ON	OFF
2)	Test 1 - Rate Gyro Filter 1				
	a)	Engage yaw damper and ensure hydraulics are on.			
	b)	Press GO switch. Observe after 1 second.	1	ON	OFF
3)	Test 2 - Engage Null				
	a)	Engage yaw damper and ensure hydraulics are on (from Test 1).			
	b)	Press GO switch. Observe after 5 seconds.	2	ON	OFF
4)	Test 3 - Rate to Rudder (Displacement)				
	a)	Ensure yaw damper is engaged and hydraulics are on (from Test 2).			
	b)	Press GO switch. Observe after 1 second.	3	ON	OFF
5)	Test 4 - Rate to Rudder (Integral)				
	a)	Ensure yaw damper is engaged and hydraulics are on (from Test 3).			
	b)	Press GO switch. Observe after 40 seconds.	4	ON	OFF

STEP			OBSERVE		
			TEST NO.	LIGHTS	
				PASS	FAIL
6)	Test 5 - Disengage Synchronization				
	a)	Disengage yaw dampers and ensure hydraulics are off.			
	b)	Press GO switch. Observe after 5 seconds.	5	ON	OFF
7)	Test 6 - Derived Accel First Stage				
	a)	Ensure yaw damper is disengaged and hydraulics are off (from Test 5).			
	b)	Press GO switch. Observe after 1 second.	6	ON	OFF
8)	Test 7 - Rate Gyro Filter 2				
	a)	Ensure yaw damper is disengaged and hydraulics are off (from Test 6).			
	b)	Press GO switch. Observe after 1 second.	7	ON	OFF
9)	Test 8 - Rate Gyro				
	a)	Ensure yaw dampers are disengaged and hydraulics are off (from Test 7).			
	b)	Press GO switch. Observe after 1 second.	8	ON	OFF
10)	Return BITE switch to OFF.				

D. Restore Airplane to Normal Configuration
(1) Disengage yaw damper.
(2) Disconnect pressure source from pitot static system. Refer to Chapter 34, Air Data Instruments.
(3) Determine whether there is further need for electrical power on airplane, if not, remove external power.

AILERON SOLENOID VALVE - REMOVAL/INSTALLATION

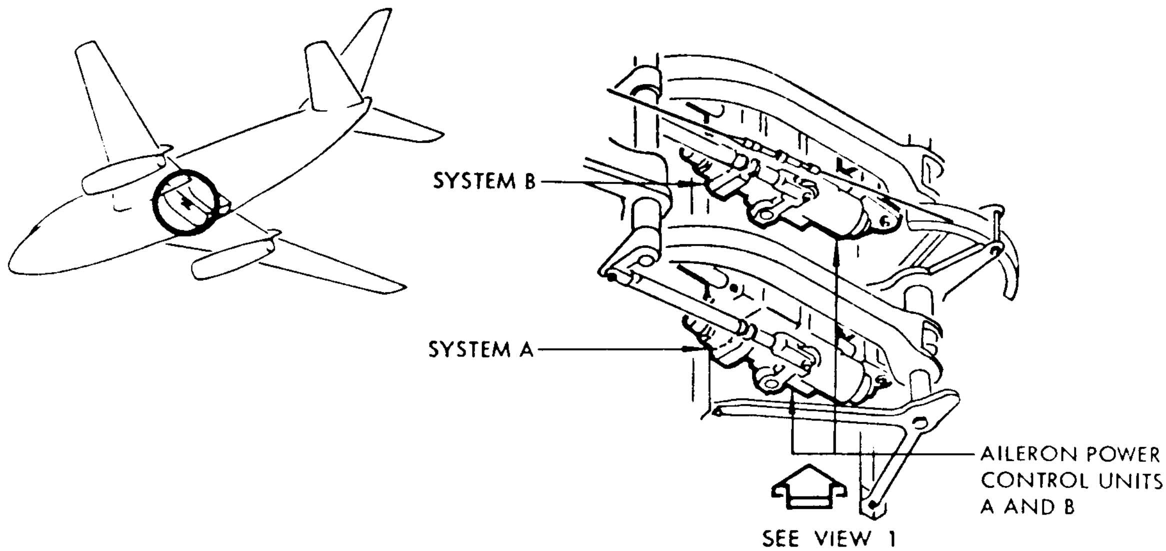
1. General
   A. Aileron solenoid valve is externally mounted on left end of aileron power unit A and B respectively (Fig. 401).
2. Remove Solenoid Valve
   A. Depressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0 MP).

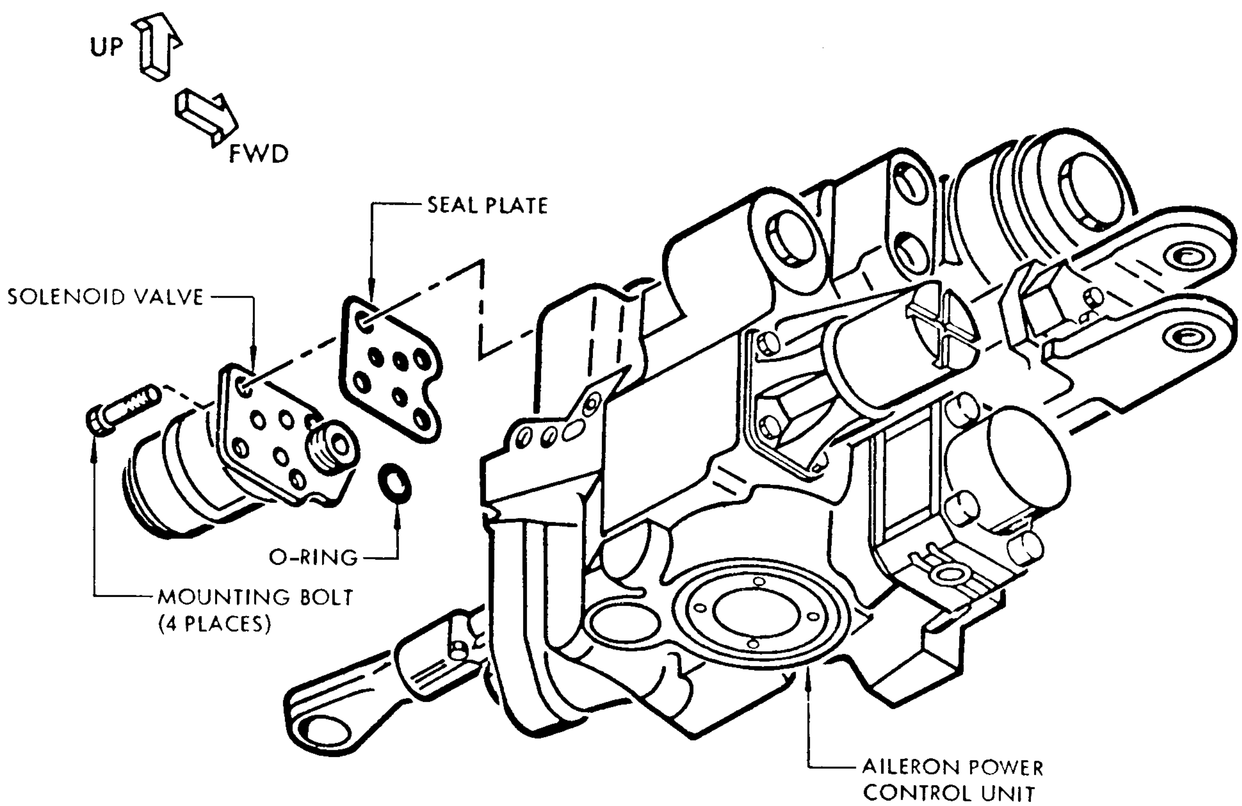
      WARNING: PRESSURIZING HYDRAULIC SYSTEMS WILL SUPPLY POWER TO RUDDER AILERON, ELEVATOR, SPOILER AND SPEED BRAKE SYSTEMS. CARE SHOULD BE TAKEN TO ISOLATE OR TAG SYSTEMS NOT BEING TESTED TO PREVENT INJURY TO PERSONNEL OR DAMAGE TO AIRPLANE AND EQUIPMENT.

   B. Remove mounting bolts and lift valve and seal plate from power unit (Fig. 401).
   C. Cover all openings
3. Install Solenoid Valve
   A. Verify seal plate and O-rings are satisfactory or replace.
   B. Lubricate O-rings with BMS 3-11 fluid and install valve and seal plate on power unit (Fig. 401).
   C. Install mounting bolts and tighten to 40-50 pound-inches torque.
   D. Lockwire bolts.
   E. Pressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0).

      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

   F. Check for leakage and operation of valve (Ref Aileron Solenoid Valve - A/T).





VIEW 1

Aileron Solenoid Valve Installation
Figure 401

AILERON SOLENOID VALVE - ADJUSTMENT/TEST

1. Aileron Solenoid Valve Test
   A. General
      (1) This test ensures that the aileron solenoid valve operates properly and assumes that the rest of the autopilot is operating normally.
   B. Prepare to Test Aileron Solenoid Valve
      (1) Pressurize hydraulic systems A and B (Ref 29-11-0, MP and 29-12-0, MP).

      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

      (2) Assure that all autopilot and associated circuit breakers are closed.
      (3) Place autopilot system select switch to appropriate position.
   C. Test Aileron Solenoid Valve
      (1) Engage autopilot AIL engage switch.
      (2) Move roll rate gyro test switch, located on face of roll control channel, to the left. Observe movement of ailerons and observe control wheels move counterclockwise then return to center.

      CAUTION: DO NOT ACTUATE RATE GYRO TEST SWITCH FOR MORE THAN 10 SECONDS.

      (3) Move control wheel to operate ailerons. Observe force required to move control wheel.
      (4) With AIL engage switch disengaged, move roll rate gyro test switch to the left and observe control wheels and ailerons do not move.
      (5) Move control wheel and check that less force is required than in step (3).
      (6) Remove hydraulic pressure and electrical power if no longer required.

AILERON TRANSFER VALVE - REMOVAL/INSTALLATION

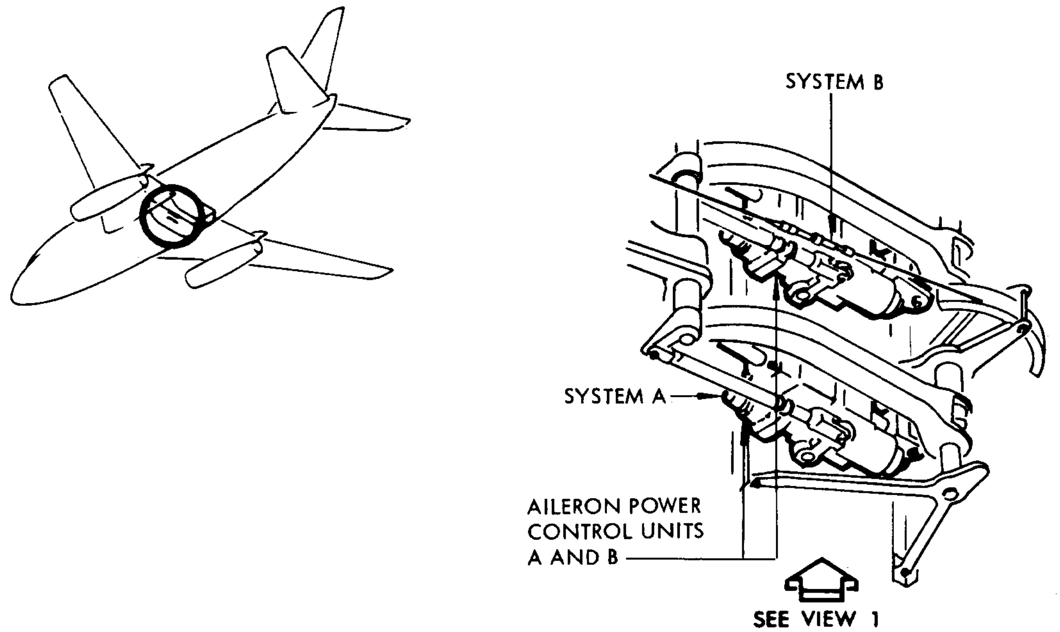
1. General
   A. Aileron transfer (servo) valve is externally mounted on right end of aileron power unit A and B respectively (Fig. 401).
2. Remove Transfer Valve
   A. Depressurize aileron hydraulic systems A and B (Ref 29-11-0, MP and 29-12-0, MP).

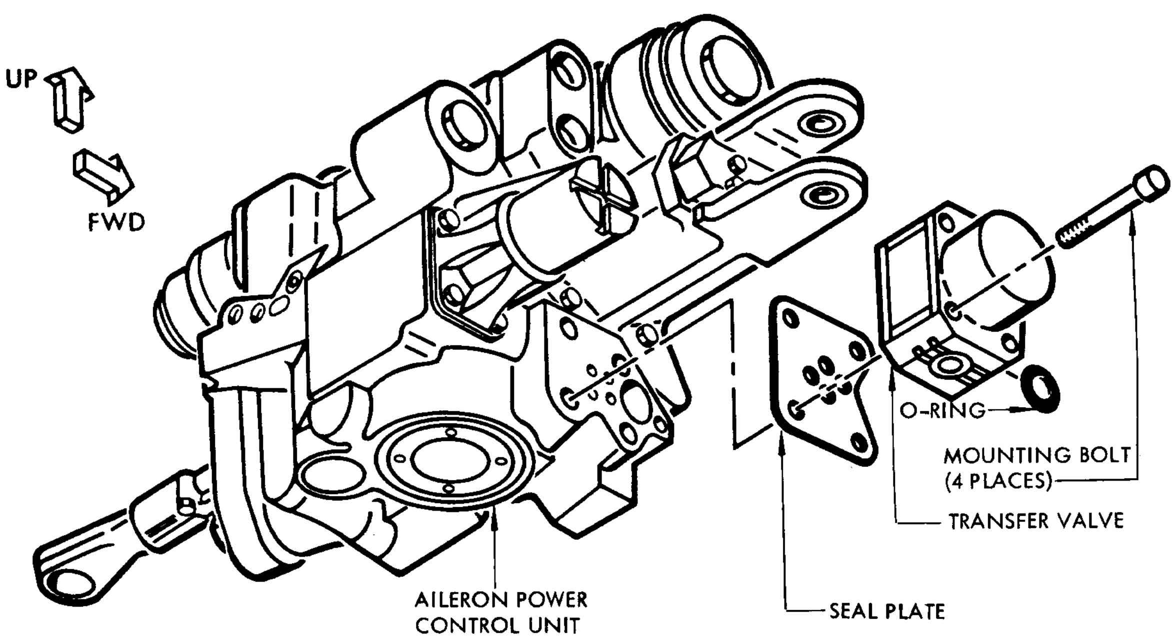
      WARNING: PRESSURIZING HYDRAULIC SYSTEMS WILL SUPPLY POWER TO RUDDER, AILERON, ELEVATOR, SPOILER AND SPEED BRAKE SYSTEMS. CARE SHOULD BE TAKEN TO ISOLATE OR TAG SYSTEMS NOT BEING TESTED TO PREVENT INJURY TO PERSONNEL OR DAMAGE TO AIRPLANE AND EQUIPMENT.

   B. Remove mounting bolts and lift valve and seal plate from power unit (Fig. 401).
   C. Cover all openings.
3. Install Transfer Valve
   A. Verify seal plate and O-rings are satisfactory or replace.
   B. Lubricate O-rings with BMS 3-11 fluid and install valve and seal plate on power unit (Fig. 401).
   C. Install mounting bolts and tighten to 40-50 pound-inches torque.
   D. Pressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0).

      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

   E. Check for leakage and operation of valve (Ref Aileron Transfer Valve - Adjustment/Test).





VIEW 1

Aileron Transfer Valve Installation
Figure 401

AILERON TRANSFER VALVE - ADJUSTMENT/TEST

1. Aileron Transfer Valve Test
   A. General
      (1) This test ensures that the aileron transfer valve operates properly and assumes that the rest of the autopilot is operating normally.
   B. Prepare to Test Aileron Transfer Valve
      (1) Pressurize hydraulic systems A and B (Ref 29-11-0, MP and 29-12-0, MP).

      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

      (2) Assure that all autopilot and associated circuit breakers are closed.
      (3) Place autopilot system select switch to appropriate position.
   C. Test Aileron Transfer Valve
      (1) Engage autopilot AIL engage switch.
      (2) Move roll rate gyro test switch, located on face of roll control channel to the left.

      CAUTION: DO NOT ACTUATE RATE GYRO TEST SWITCH FOR MORE THAN 10 SECONDS.

      (3) Observe movement of aileron surfaces and observe counterclockwise movement of control wheel then return to center.
      (4) Move roll rate gyro test switch to the right.
      (5) Observe movement of aileron surfaces and observe clockwise movement of control wheel then return to center.

ELEVATOR SOLENOID VALVE - REMOVAL/INSTALLATION

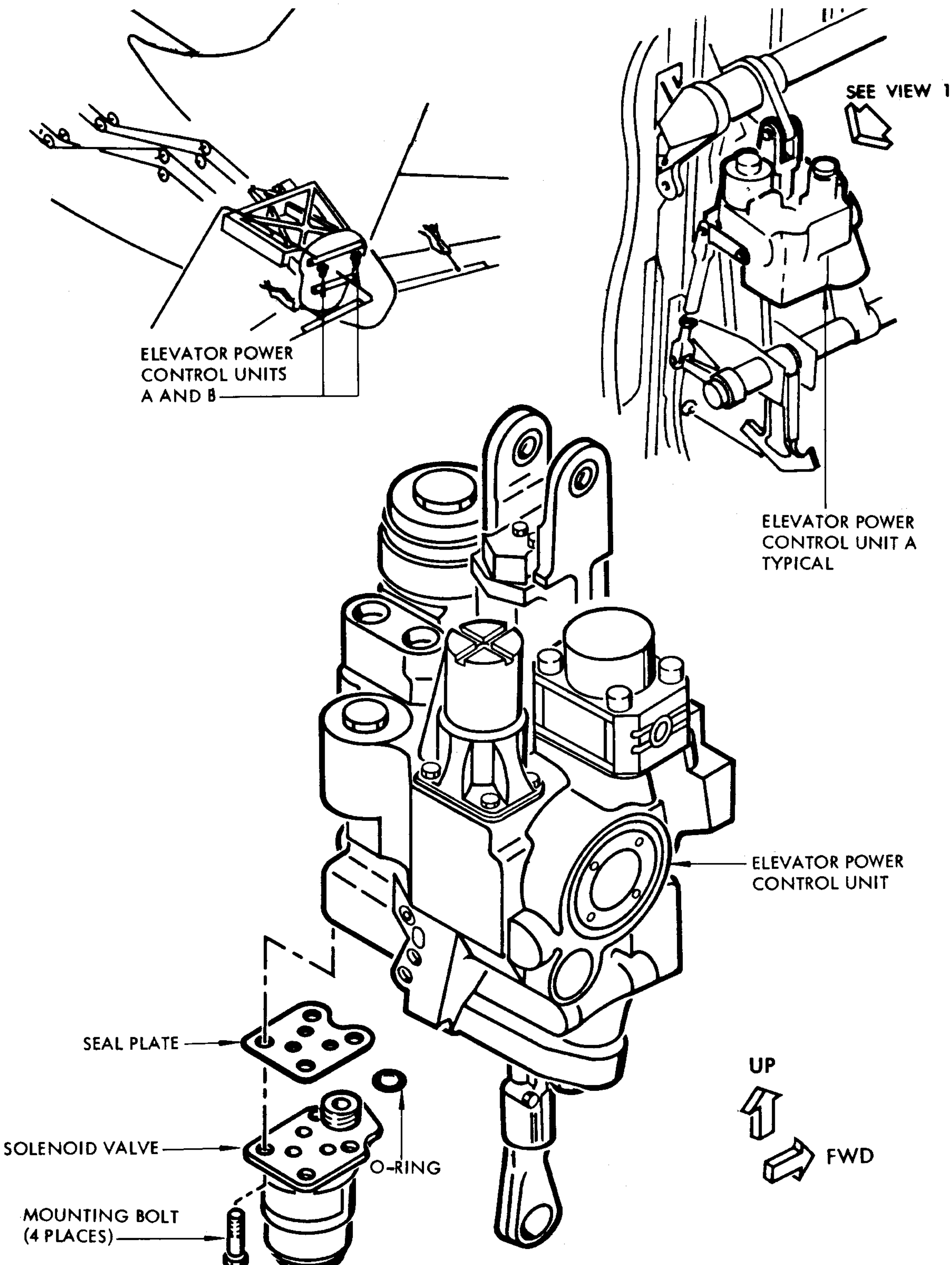
1. General
   A. Elevator solenoid valve is externally mounted on lower end of elevator power unit A and B respectively (Fig. 401).
2. Prepare to Remove Solenoid Valve
   A. Depressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0 MP).

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

   B. Remove access panel 3802 for both left and right elevator power units (Ref Chapter 12, Access Doors and Panels).
3. Remove Solenoid Valve
   A. Remove mounting bolts and lift valve and seal plate from power unit (Fig. 401).
   B. Cover all openings.
4. Install Solenoid Valve
   A. Verify seal plate and O-rings are satisfactory or replace.
   B. Lubricate O-rings with BMS 3-11 fluid and install valve and seal plate on power unit (Fig. 401).
   C. Install mounting bolts and tighten to 40-50 pound-inches torque.
   D. Lockwire bolts.
   E. Pressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0).

      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

   F. Check for leakage and operation of valve (Ref Elevator Solenoid Valve - A/T).
   G. Replace access panel.



Elevator Solenoid Valve Installation
Figure 401

ELEVATOR SOLENOID VALVE - ADJUSTMENT/TEST

1. Elevator Solenoid Valve Test
   A. General
      (1) This test ensures that the elevator solenoid valve operates properly and assumes that the rest of the autopilot is operating normally.
   B. Prepare to Test Elevator Solenoid Valve
      (1) Pressurize hydraulic systems A and B (Ref 29-11-0, Hydraulic System A - MP and 29-12-0, Hydraulic System B - MP).

      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

      (2) Assure that all autopilot and associated circuit breakers are closed.
      (3) Place autopilot system select switch to appropriate position.
   C. Test Elevator Solenoid Valve
      (1) Engage autopilot.
      (2) Move pitch rate gyro test switch, located on the face of pitch control channel, to the right. Observe movement of elevators and observe movement of control columns aft and return to neutral.

      CAUTION: DO NOT ACTUATE RATE GYRO TEST SWITCH FOR MORE THAN 10 SECONDS.

      (3) Move control column to operate elevators. Observe force required to move control column.
      (4) With autopilot disengaged, move pitch rate gyro test switch to the right and observe that elevators and control columns do not move.
      (5) Move control column and check that less force is required than in step (3).
      (6) Remove hydraulic pressure and electrical power if no longer required.

ELEVATOR TRANSFER VALVE - REMOVAL/INSTALLATION

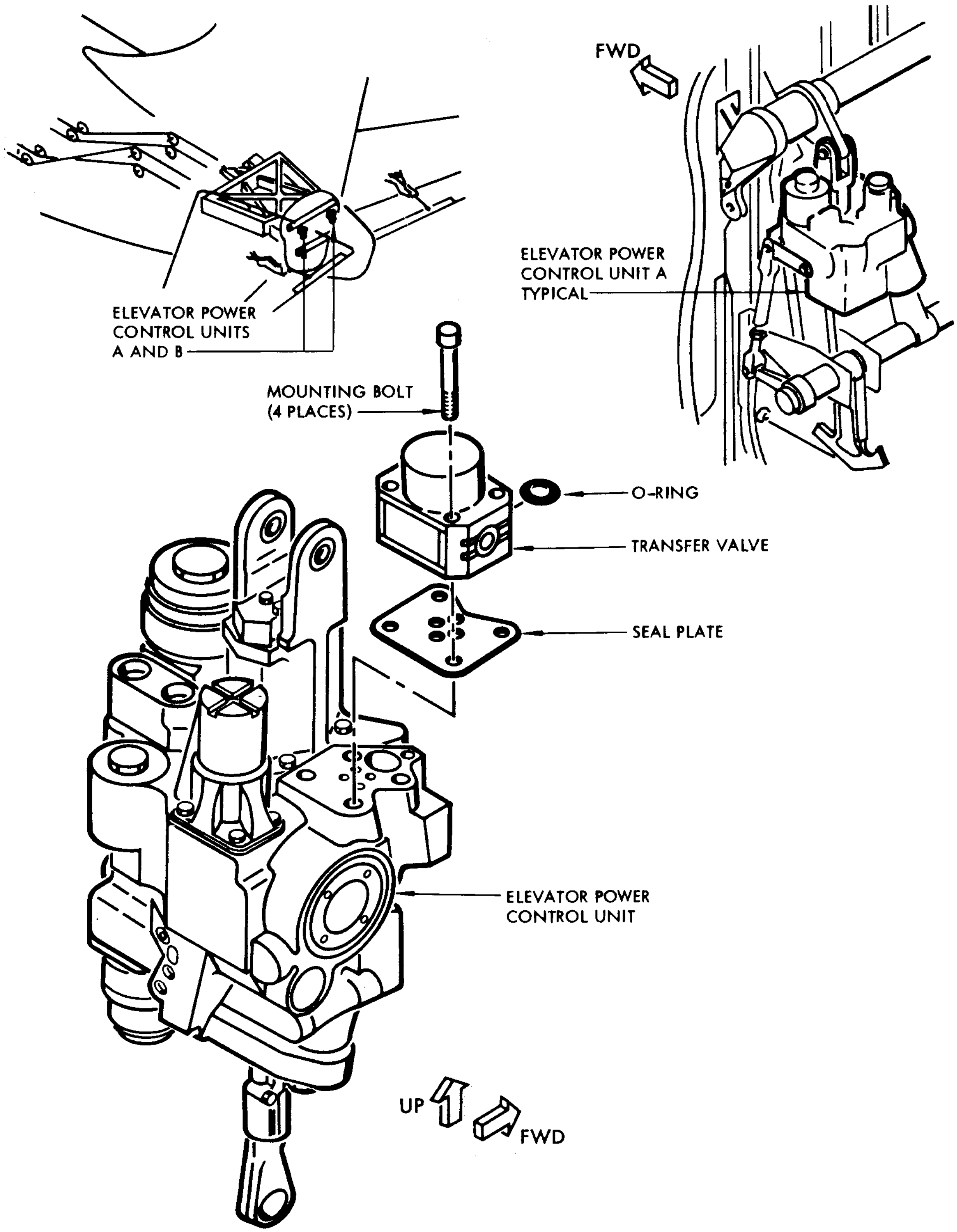
1. General
   A. Elevator transfer valve is externally mounted on upper end of elevator power unit A and B respectively (Fig. 401).
2. Prepare to Remove Transfer Valve
   A. Depressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0, MP).

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

   B. Remove access panel 3802 for both left and right elevator power units (Ref Chapter 12, Access Doors and Panels).
3. Remove Transfer Valve
   A. Remove mounting bolts and lift valve and seal plate from power unit (Fig. 401).
   B. Cover all openings.
4. Install Transfer Valve
   A. Verify seal plate and O-rings are satisfactory or replace.
   B. Lubricate O-rings with BMS 3-11 fluid and install valve and seal plate on power unit (Fig. 401).
   C. Install mounting bolts and tighten to 40-50 pound-inches torque.
   D. Pressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0).

   WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

   E. Check for leakage and operation of valve (Ref Elevator Transfer Valve - Adjustment/Test).
   F. Replace access panel.



Elevator Transfer Valve Installation
Figure 401

ELEVATOR TRANSFER VALVE - ADJUSTMENT/TEST

1. Elevator Transfer Valve Test
   A. General
      (1) This test ensures that the elevator transfer valve operates properly and assumes that the rest of the autopilot is operating normally.
   B. Prepare to Test Elevator Transfer Valve
      (1) Pressurize hydraulic systems A and B (Ref 29-11-0 MP and 29-12-0 MP).

      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

      (2) Assure that all autopilot and associated circuit breakers are closed.
      (3) Place autopilot system select switch to appropriate position.
   C. Test Elevator Transfer Valve
      (1) Engage autopilot.
      (2) Move pitch rate gyro test switch, located on face of pitch control channel, to the right.

      CAUTION: DO NOT ACTUATE RATE GYRO TEST SWITCH FOR MORE THAN 10 SECONDS.

      (3) Observe movement of elevators and observe movement of control columns aft and return to neutral.
      (4) Move pitch rate gyro test switch to the left.
      (5) Observe movement of elevators and observe movement of control columns forward and return to neutral.

RUDDER SOLENOID VALVE - REMOVAL/INSTALLATION

1. General
   A. Rudder solenoid valves are externally mounted on each end of rudder power unit located in the lower area of the vertical fin (Fig. 401). Two valves are installed on airplanes with A and B switching on the autopilot control panel and one valve (aft) is installed on airplanes with B system only on the autopilot control panel.
2. Prepare to Remove Solenoid Valve
   A. Depressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0 MP).

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

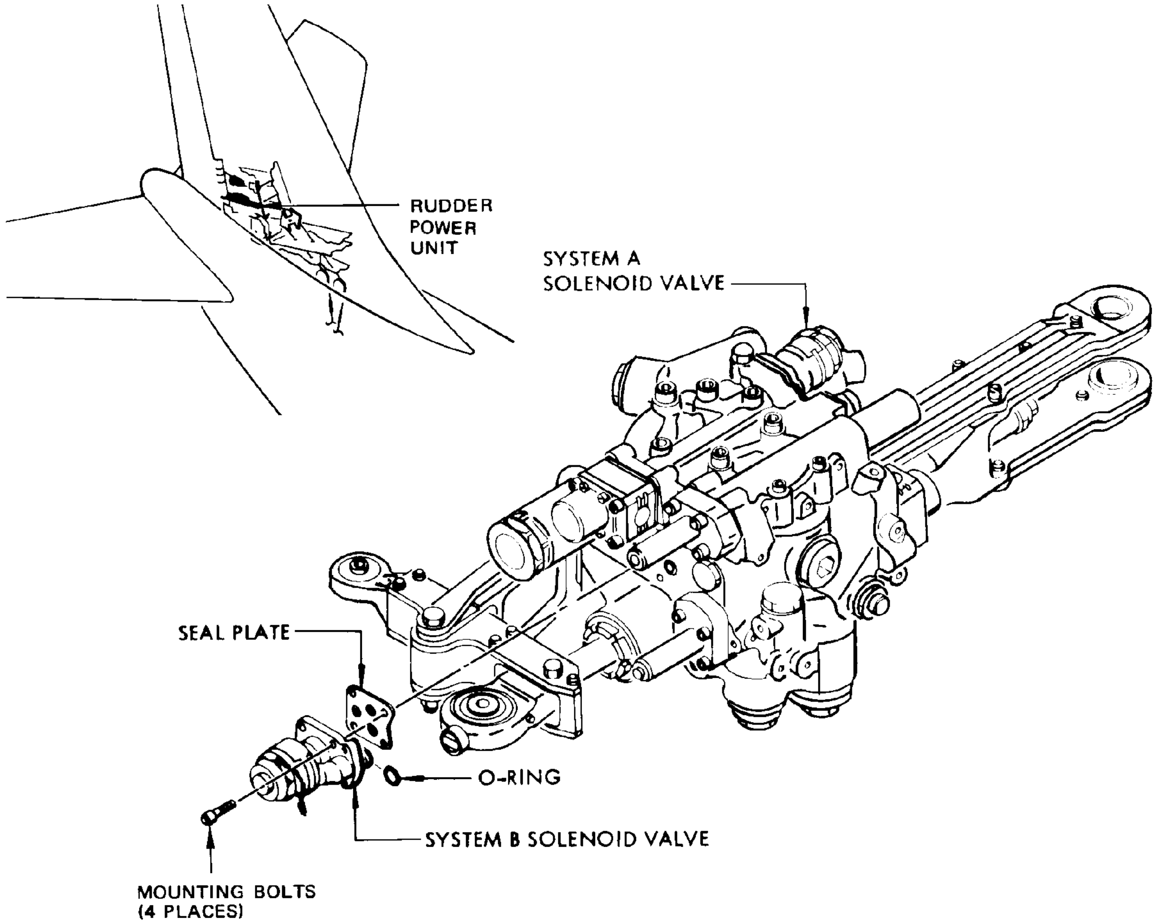
   B. Remove access panels 9512 and 9514 (Ref Chapter 12, Access Door and Panels).
3. Remove Solenoid Valve
   A. Remove mounting bolts and lift valve and seal plate from power unit (Fig. 401).
   B. Cover all openings.
4. Install Solenoid Valve

   NOTE: Do not install rudder solenoid valves with these part numbers: Boeing P/N 10-60811-1, or -9 Boeing P/N 10-60811-3 with Parker P/N 59600-5007 Refer to the IPC for correct parts usage.

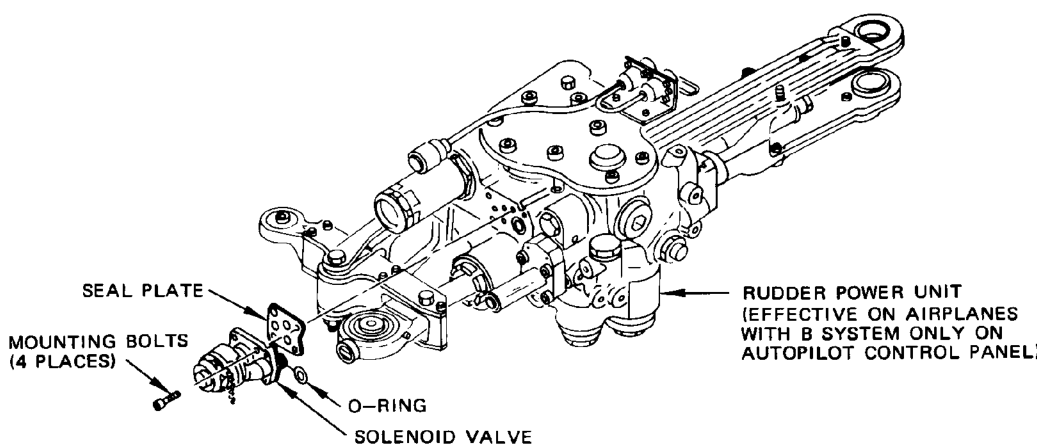
   A. Verify seal plate and O-rings are satisfactory, or replace.
   B. Lubricate O-rings with BMS 3-11 fluid and install valve and seal plate on power unit (Fig. 401).
   C. Install mounting bolts and tighten to 40-50 pound-inches torque.
   D. Lockwire bolts.
   E. Pressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0).

      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

   F. Check for leakage and operation of valve (Ref Rudder Solenoid Valve - A/T).
   G. Replace access panels.



AIRPLANES WITH A AND B SWITCHING ON THE AUTOPILOT CONTROL PANEL



AIRPLANES WITH B SYSTEM ONLY ON AUTOPILOT CONTROL PANEL
RUDDER POWER UNIT

Rudder Solenoid Valve Installation
Figure 401

RUDDER SOLENOID VALVE - ADJUSTMENT/TEST

1. Rudder Solenoid Valve Test
   A. General
      (1) This test ensures that the solenoid valve operates properly and assumes that the rest of the yaw damper system is operating normally.
   B. Prepare to Test Rudder Solenoid Valve
      (1) Pressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0 MP).

      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

      (2) Close appropriate circuit breakers.
      (3) Place autopilot system select switch to appropriate position.
   C. Test Rudder Solenoid Valve (Aircraft not incorporating SB 27A1206)
      (1) Place yaw damper engage switch to ON.
      (2) Move the yaw damper test switch, located on center instrument panel, to the right. Observe yaw damper test meter needle and rudder moves to the right.

      CAUTION: DO NOT ACTUATE TEST SWITCH FOR MORE THAN 10 SECONDS.

      (3) Place yaw damper engage switch to OFF.
      (4) Move yaw damper test switch to the right and left. Observe yaw damper test meter needle and rudder do not move in either direction.
   D. Test Rudder Solenoid Valve (Aircraft incorporating SB 27A1206, Yaw Damper Coupler P/N 10-62253, 4084042)
      (1) Perform Yaw Damper Coupler Installation Test (Ref 22-12-01/401).

RUDDER TRANSFER VALVE - REMOVAL/INSTALLATION

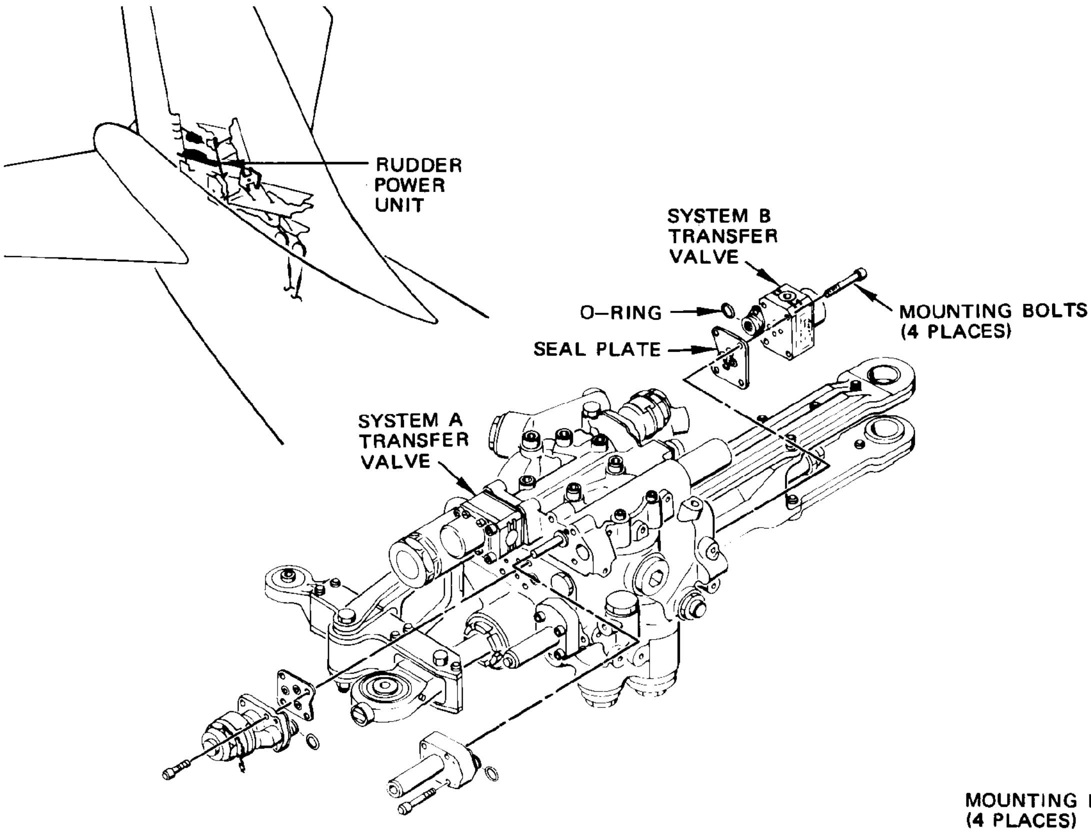
1. General
    A. Rudder transfer valves are externally mounted on each end of rudder power unit located in the lower area of the vertical fin (Fig. 401). Two valves are installed on airplanes with A and B switching on the autopilot control panel and one (fwd) valve is installed on airplanes with B system only on the autopilot control panel.
2. Prepare to Remove Transfer Valve
    A. Depressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0 MP).

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

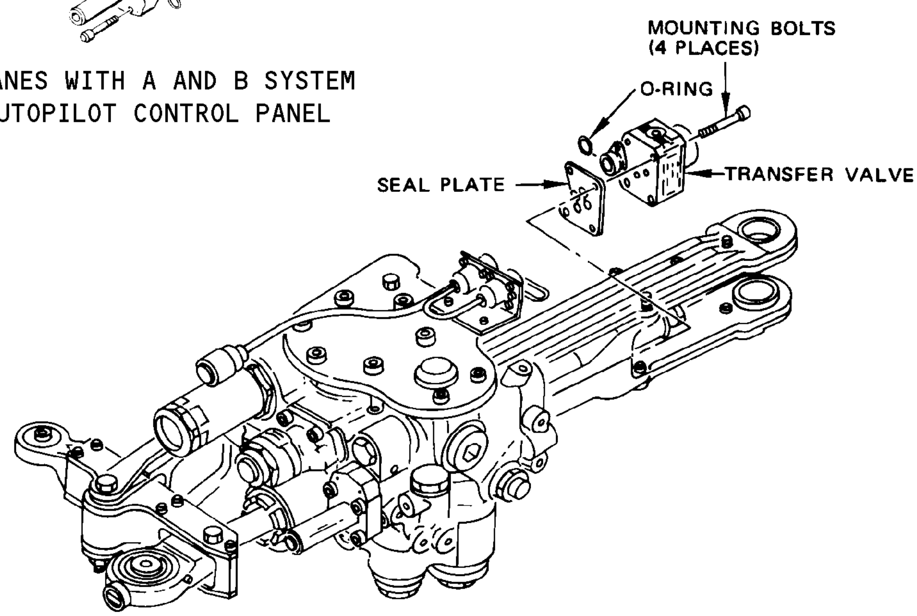
    B. Remove access panels 9512 and 9514. Refer to Chapter 12, Access Doors and Panels.
3. Remove Transfer Valve
    A. Remove mounting bolts and lift valve and seal plate from power unit (Fig. 401).
    B. Cover all openings.
4. Install Transfer Valve
    A. Verify seal plate and O-rings are satisfactory or replace.
    B. Lubricate O-rings with BMS 3-11 fluid and install valve and seal plate on power unit (Fig. 401).
    C. Install mounting bolts and tighten to 40-50 pound-inches torque.
    D. Lockwire bolts.
    E. Pressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0).

    WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

    F. Check for leakage and operation of valve (Ref Rudder Transfer Valve - Adjustment/Test).
    G. Replace access panels.



AIRPLANES WITH A AND B SYSTEM
ON AUTOPILOT CONTROL PANEL



AIRPLANES WITH B SYSTEM ONLY
ON AUTOPILOT CONTROL PANEL
RUDDER POWER UNIT

Rudder Transfer Valve Installation
Figure 401

RUDDER TRANSFER VALVE - ADJUSTMENT/TEST

1. Rudder Transfer Valve Test
   A. General
      (1) This test ensures that the transfer valve operates properly and assumes that the rest of the yaw damper system is operating normally.
   B. Prepare to Test Rudder Transfer Valve
      (1) Pressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0 MP).

      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

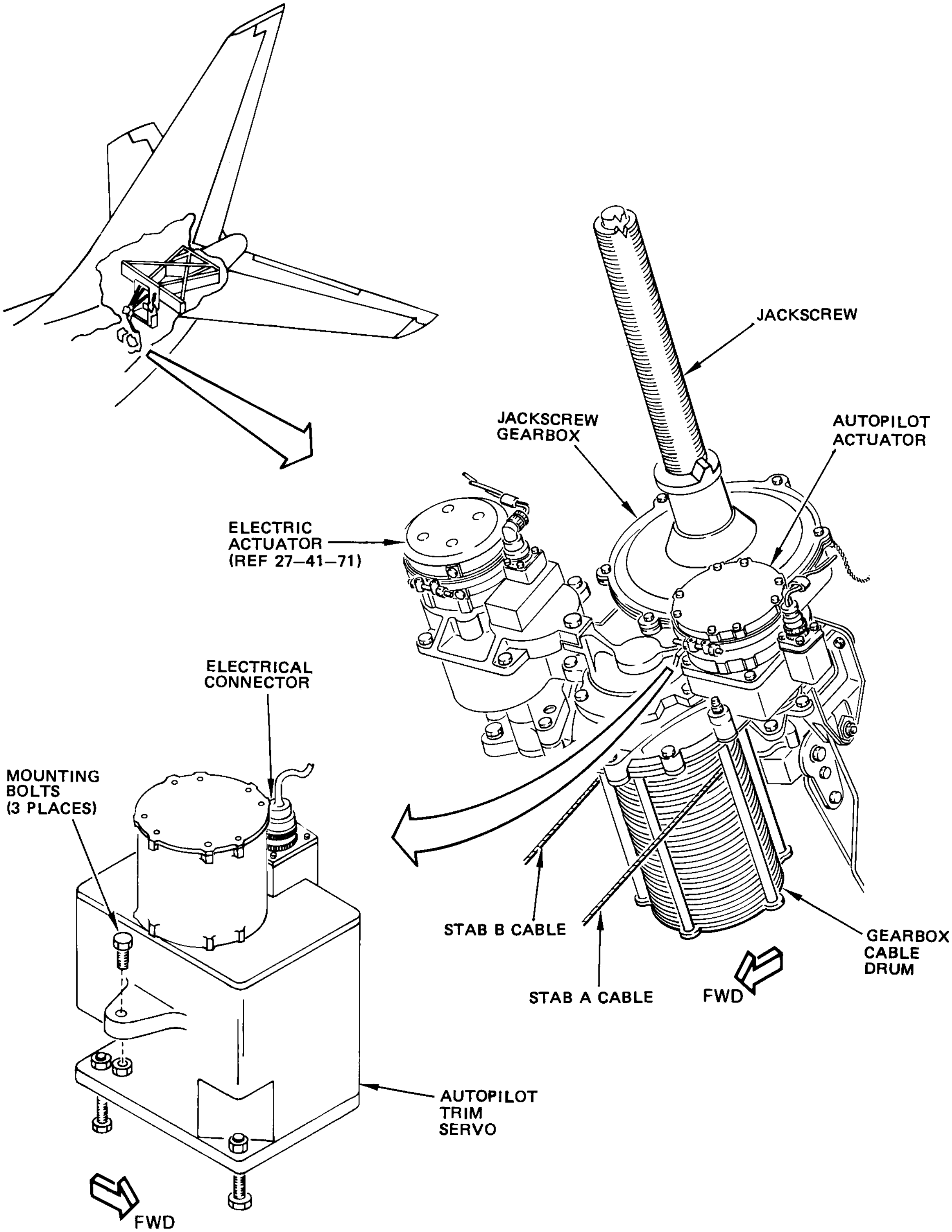
      (2) Close appropriate circuit breakers.
      (3) Place autopilot system select switch to appropriate position.
   C. Test Rudder Transfer Valve (Aircraft not incorporating SB 27A1206)
      (1) Place yaw damper engage switch to ON.
      (2) Move yaw damper test switch, located on center instrument panel, to the right. Observe yaw damper test meter needle moves to the right, then returns to center. Observe rudder moves approximately 2 degrees to the right, then returns to center.

      CAUTION: DO NOT ACTUATE TEST SWITCH FOR MORE THAN 10 SECONDS.

      (3) Move yaw damper test switch to the left. Observe yaw damper test meter needle moves to the left, then returns to center. Observe rudder moves approximately 2 degrees to the left, then returns to center.
   D. Test Rudder Transfer Valve (Aircraft incorporating SB 27A1206, Yaw Damper Coupler P/N 10-62253, 4084042)
      (1) Perform Yaw Damper Coupler Installation Test (Ref 22-12-01/401).

AUTOPILOT STABILIZER TRIM SERVO - REMOVAL/INSTALLATION

1. Remove Autopilot Stabilizer Trim Servo (Fig. 401)
   A. Remove access panel 3701 (Ref Chapter 12, Access Doors and Panels),
   B. Disconnect electrical connector and wire bundle clamp from autopilot trim servo.
   C. Remove mounting bolts and remove trim servo.
   D. Remove foreign matter from motor mounting pad.
   E. Position autopilot trim servo on mounting pad and install mounting bolts.
   F. Connect electrical connector to servo assembly.
   G. Install access panel.
   H. Check operation of stabilizer trim servo (Ref Autopilot Stabilizer Trim Servo - Adjustment/Test).



Autopilot Stabilizer Trim Servo Installation
Figure 401

AUTOPILOT STABILIZER TRIM SERVO - ADJUSTMENT/TEST

1. Autopilot Stabilizer Trim Servo Test
   A. Prepare to Test Autopilot Stabilizer Trim Servo
      (1) Check that horizontal stabilizer and elevator are free to travel.
      (2) Ensure ground power is applied and close circuit breakers for autopilot system and stabilizer trim circuit on load control center P6.
      (3) Pressurize hydraulic systems A and B (Ref 29-11-0 MP and 29-12-0 MP).

      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

      (4) Position the stabilizer to 5 units of trim and place autopilot system select switch to A position.
   B. Test Autopilot Stabilizer Trim Servo
      (1) Engage the autopilot ELEV engage switch, and pull the control column out of detent. Observe the elevator moves up.
      (2) Return the column to neutral. Observe the stabilizer leading edge moves down and trims at a rate of 1 unit in 30 to 40 seconds.

      CAUTION: TO PREVENT STABILIZER TRIM MOTOR FROM BURNING OUT, DO NOT OPERATE MORE THAN ONE MINUTE IN EVERY 10 MINUTES.

      (3) Disengage the autopilot engage switches (this terminates trimming action) and return the stabilizer to 5 units of trim.
      (4) Extend the flaps to 5 degrees, engage the autopilot ELEV engage switch, and push the control column out of detent. Observe the elevator moves down.
      (5) Return the column to neutral. Observe the stabilizer leading edge moves up and trims at a rate of 1 unit in 4 to 7 seconds.
      (6) Disengage autopilot.
   C. Restore Airplane to Normal Configuration
      (1) Raise the flaps.
      (2) Determine whether there is any further need for electrical power on the airplane; if not, remove external power.

PITCH CONTROL WHEEL STEERING FORCE TRANSDUCER - REMOVAL/INSTALLATION

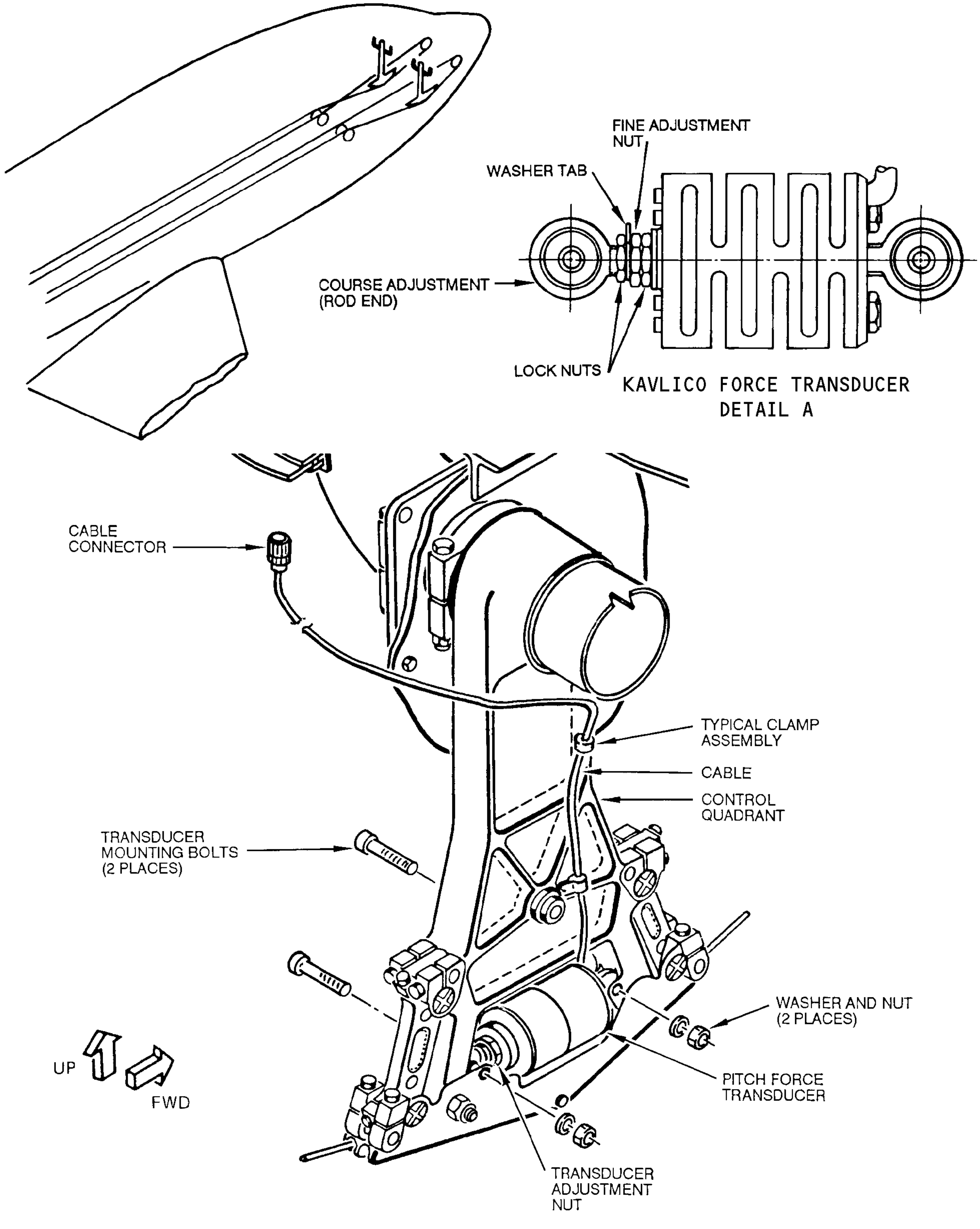
1. General
   A. Each control column has a pitch force transducer installed on the elevator control quadrant assembly. The two force transducers are installed opposite to each other; however, removal/installation of the transducers is similar.
2. Equipment and Materials
   A. Rigging Pins E1 and E5 - 0.309 (+0.003/-0.000)-inch diameter, 6-inch minimum length (MS20392-4)
3. Remove Pitch Force Transducer
   A. Open pitch channel circuit breakers on load control center P6.
   B. Open lower nose access door 1103. Refer to Access Doors and Panels, Chapter 12.
   C. Disconnect force transducer connector from airplane wiring (Fig. 401).
   D. Remove force transducer cable from clamps.
   E. Remove two bolts, washers, and nuts and remove force transducer from quadrant assembly.
4. Prepare to Install Pitch Force Transducer
   A. Open lower nose access door 1103. Refer to Access Doors and Panels, Chapter 12.
   B. Remove tail cone access panel 3802. Refer to Access Doors and Panels, Chapter 12.
   C. Insert rigging pin E5 in elevator right aft quadrant (Fig. 402).

      NOTE: Rigging pins should fit freely.

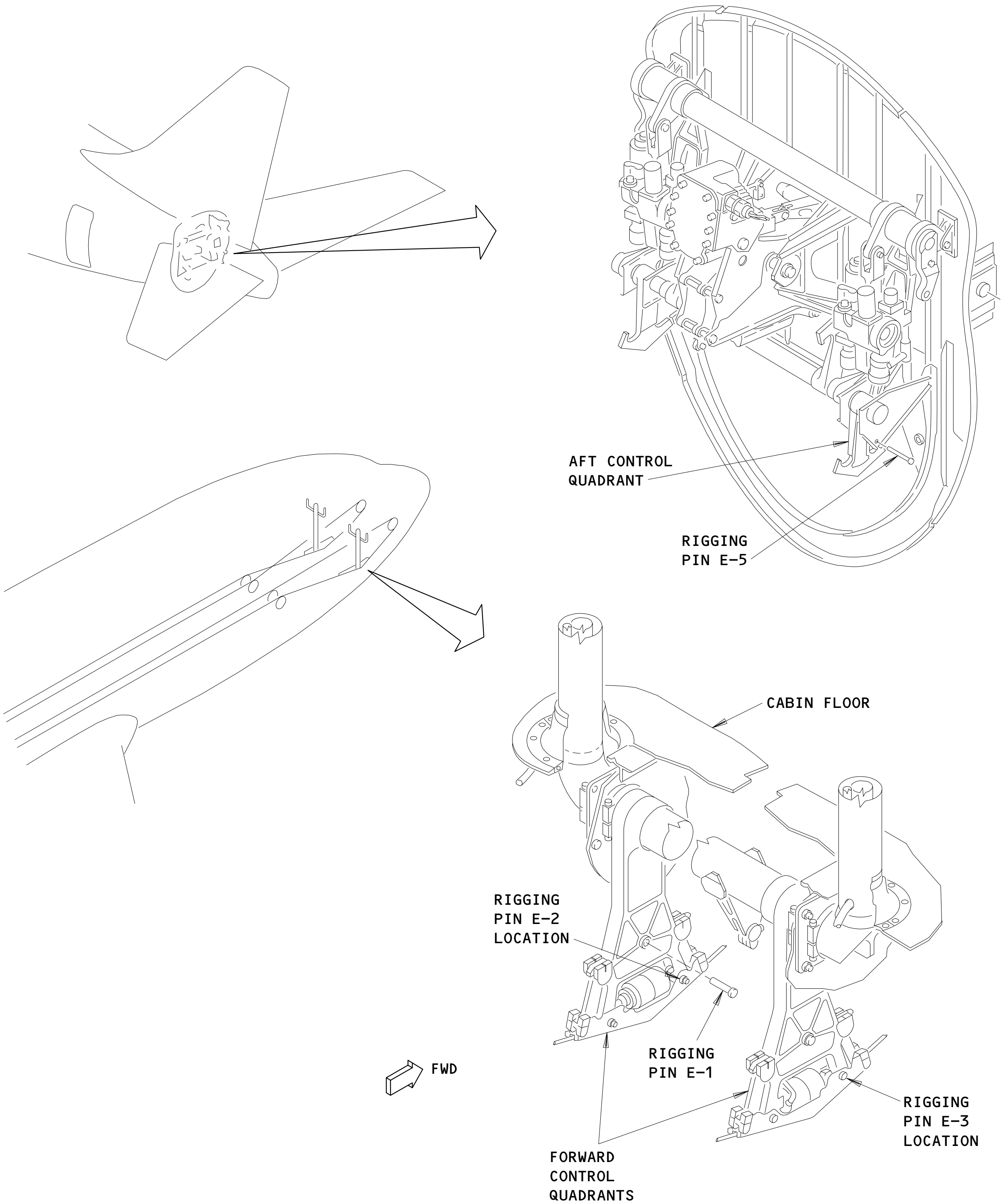
   D. Insert rigging pin E1 in elevator left forward quadrant.
5. Install Pitch Force Transducer
   A. Loosen the adjustable rod end and locknuts to allow fine adjustment nut and rod end to be rotated with fingers.
   B. Hold pitch force transducer in position and connect to quadrant assembly.

      NOTE: Adjustable end should face forward on right quadrant. Adjustable end should face aft on left quadrant.

   C. Adjust force transducer. Refer to Pitch Control Wheel Steering Force Transducer - Adjustment/Test.
   D. Install lockwire through locknuts, fine adjustment nut, and washer tab on adjustable rod end.
   E. Clamp force transducer cable to structure.
   F. Connect force transducer to airplane wiring.
   G. Remove rigging pins.
   H. Close access doors.



Pitch Force Transducer Installation
Figure 401



Pitch Transducer Rigging Pin Location
Figure 402

PITCH CONTROL WHEEL STEERING FORCE TRANSDUCER - ADJUSTMENT/TEST

1. General
   A. Each control column has a pitch force transducer installed on the elevator control quadrant assembly. The two force transducers are installed opposite to each other and are electrically 180 degrees out of phase to reduce null error voltage to the autopilot system. The following procedures are intended to verify the force transducers are adjusted and operating properly. The adjustment of the force transducers is fairly critical and if not adjusted properly, the pitch channel may not engage.
2. Pitch Force Transducer Adjustment
   A. Equipment and Materials
      (1) Test box - F80218-42
      (2) Adapter cables (2 required) - F80218-48
      (3) Rigging Pins E1 and E5 - 0.309 (+0.003/-0.0)-inch diameter, 6.0-inch minimum length (MS20392-4)
      (4) Rigging pins E2 and E3 (2) - 0.4941 (±0.0004)-inch diameter, 1.5-inch minimum length
      (5) VTVM - Hewlett Packard Model 410C
   B. Prepare for Adjustment
      (1) Provide electrical power to the airplane and energize load control center P6.
      (2) Ensure elevators are properly rigged. Refer to Elevator and Tab Control System, Chapter 27.
      (3) Open pitch channel circuit breakers on load control center P6.
      (4) Open lower nose access door 1103 and tailcone access door 3802. Refer to Access Doors and Panels, Chapter 12.
   C. Adjust Pitch Force Transducer (Fig. 501)
      (1) Install rigging pin E1 in forward left control quadrant and rigging pin E5 in right aft control quadrant (Fig. 502).

         NOTE: Rigging pins should fit freely.

      (2) Disconnect both force transducer cable connectors from airplane wiring.
      (3) Remove bolt, washer, and nut from adjustable end of force transducers.
      (4) Remove shoulder bolt from each quadrant assembly and install rigging pins E2 and E3.

         NOTE: Rigging pin locations are labeled on quadrant assembly. If cables are rigged properly, the rigging pins should be easily installed.

(5) Connect captain's force transducer to the captain's connector on the test box with adapter cable.
(6) Connect first officer's force transducer to the first officer's connector on the test box with adapter cable.
(7) Energize test box and allow force transducers to warm up and stabilize for 15 minutes.

CAUTION: USE ONLY 115 VOLTS, 400 Hz AC. USE OF 60 Hz VOLTAGE WILL DAMAGE TRANSDUCERS.

(8) Position the SYSTEM A - SYSTEM B switch in the SYSTEM A position.
(9) Position test box SELECT switch to CAPT. Measure and record output voltage at static conditions. The output should be no more than 80 mv. Check electrical cable installation to see if null can be lowered by loosening cable a small amount. Tighten cable clamps.

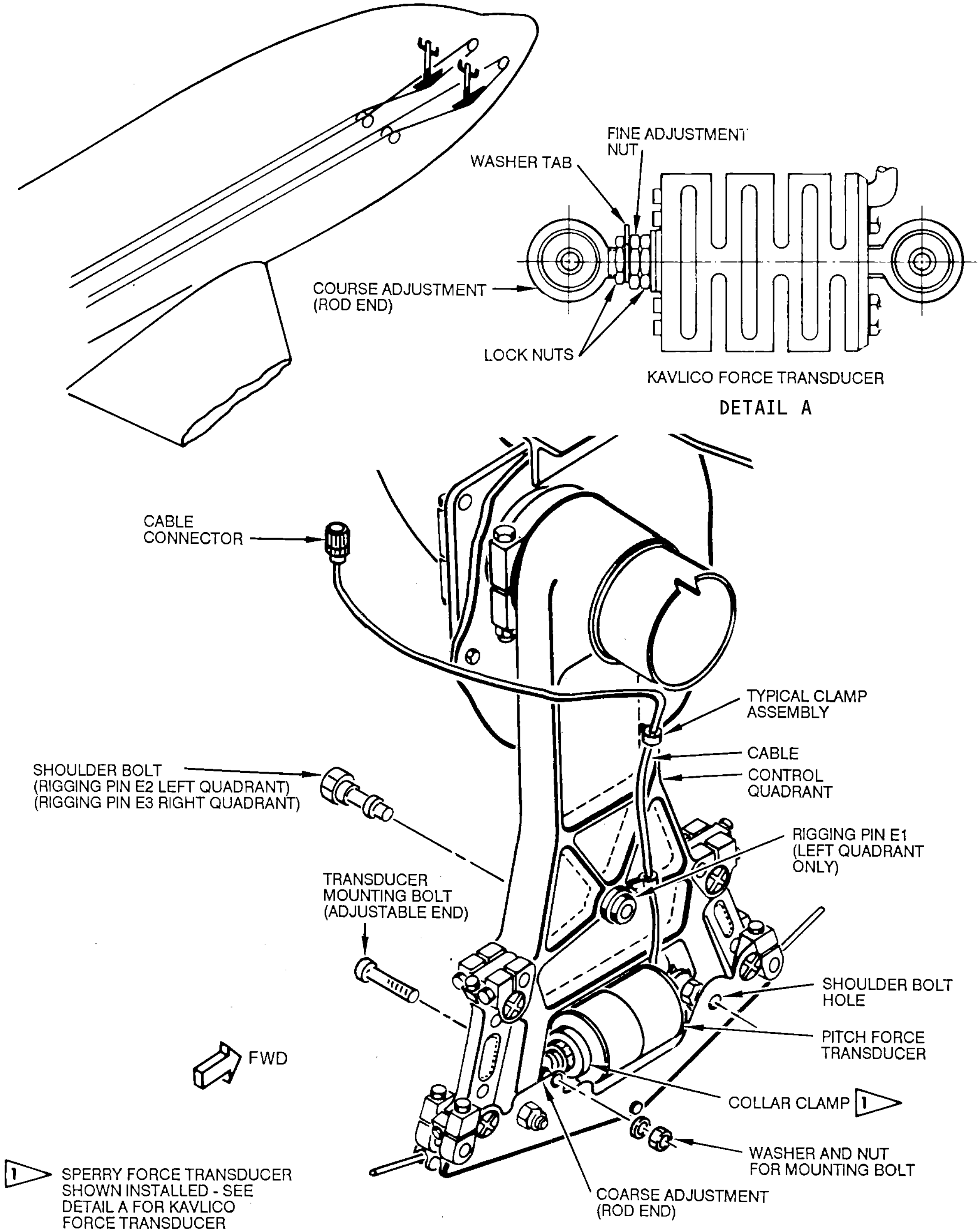
NOTE: Hold transducer level to minimize error in reading. The fine and coarse adjustments should be checked for some threads showing.

(10) Position test box SELECT switch to F/O. Measure and record output voltage at static conditions. The output should be no more than 80 mv.
(11) On airplanes with Sperry force transducers, remove lockwire on collar clamp, loosen lockscrew and adjust rod end (coarse adjustment) on each force transducer until mounting bolt can be easily inserted. Install bolt, washer, and nut. On airplanes with Kavlico force transducers, remove lockwire, loosen locknut on adjustable rod end, loosen locknut on adjustable sleeve and adjust rod end (coarse adjustment) on each force transducer until mounting bolt can be easily inserted. Install bolt, washer, and nut.

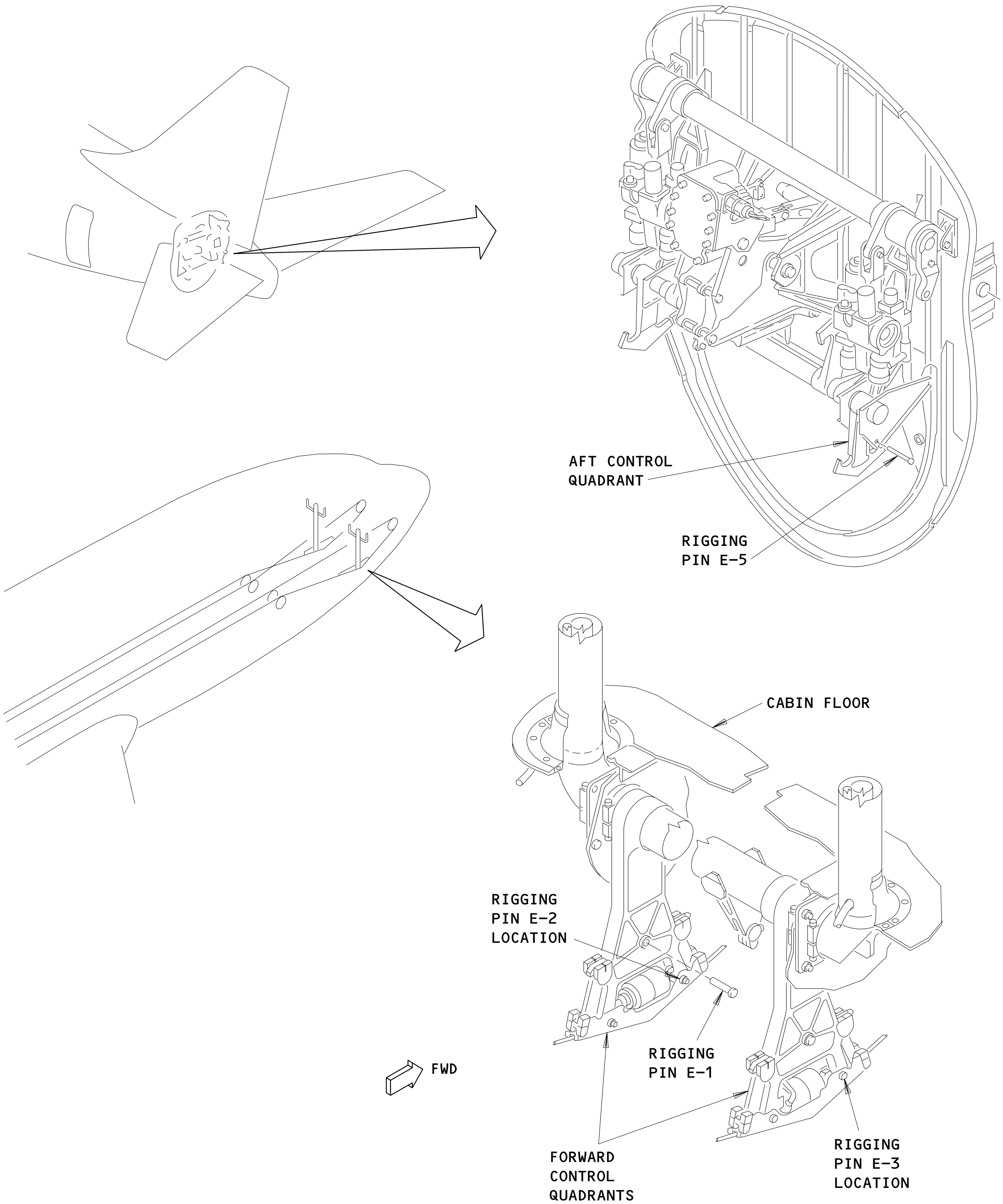
NOTE: Coarse adjustment is approximately 0.016 inch per turn. Fine adjustment is approximately 0.006 inch per turn.

(12) Adjust fine adjustment until voltage obtained for each force transducer is within ± 20 mv of the values measured in steps (9) and (10). To ensure transducer is not preloaded, turn coarse adjustment until mounting hole is parallel with mounting hole at opposite end of transducer after each fine adjustment and before taking reading.

NOTE: Use the CAPT and F/O positions on the test box. More than one null may be obtained.



Pitch Force Transducer Adjustment
Figure 501



Pitch Transducer Rigging Pin Location
Figure 502

(13) Position test box SELECT switch to TOTAL. The indicated voltage should be greater than either individual force transducer null voltage. If the indicated voltage is less than either individual force transducer null voltage, the transducers are not nulled correctly. Position test box SELECT switch to CAPT or F/O and readjust the corresponding force transducer to obtain a new null and then check voltage with test box switch in TOTAL position. Repeat as necessary until an acceptable voltage is obtained.
(14) On airplanes with Sperry force transducers, tighten lockscrew on collar clamp and install lockwiring. On airplanes with Kavlico force transducers, tighten locknut on adjustable sleeve (pressing washer against end cap of transducer), tighten locknut on rod end and install lockwiring.
(15) Remove rigging pins E2 and E3, and install shoulder bolt and nut in each quadrant so that bolt head is facing outboard.
(16) Remove rigging pins E1 and E5 from quadrants.
(17) Cycle control column several times gradually decreasing to zero.
(18) Position test box SELECT switch to CAPT and record voltage. Position test box SELECT switch to F/O and record voltage. The voltages must be less than 350 mv.
(19) Check that the difference between CAPT and F/O voltages does not exceed 150 mv.

NOTE: If voltage is excessive, check control quadrant and bearings for structural interference, excess friction in support bearings or links and/or misalignment of links.

(20) Turn off test box and reconnect force transducers to airplane wiring.
(21) Close access door.
(22) Test pitch control wheel steering force transducer.

3. Pitch Force Transducer Test

A. Equipment and Materials
(1) Spring Scale - 0 to 50 pounds, John Chatillon and Sons, Model DPP-50

B. Prepare to Test Pitch Force Transducer
(1) Verify control surfaces are free of obstructions and personnel.
(2) Provide electrical power to the airplane and energize load control center P6.
(3) Pressurize hydraulic systems A and B (Ref 29-11-0 MP and 29-12-0 MP).

WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

(4) Ensure elevators are operable.
(5) Ensure autopilot, vertical gyro, and air data computer circuit breakers are closed.

C. Test Pitch Force Transducers
(1) Place autopilot system select switch on A or B.
(2) Connect spring scale to center hub of captain's control column.
(3) Engage pitch channel.

NOTE: If the force transducers are not adjusted properly, the pitch channel may not engage.

(4) Slowly push on the captain's column until the column starts to drive. It should require 3.8 to 7.5 pounds. Record value.
(5) Repeat step (4) except pull on column. The recorded values should differ by less than 2 pounds.
(6) Repeat steps (2) thru (5) for first officer's column. Position pitch mode select switch to ALT HOLD.
(7) Repeat steps (2) thru (6) except apply a steady pull on the control column until the pitch mode select switch returns to the OFF position. The pull required should be greater than 15 and less than 23 pounds.

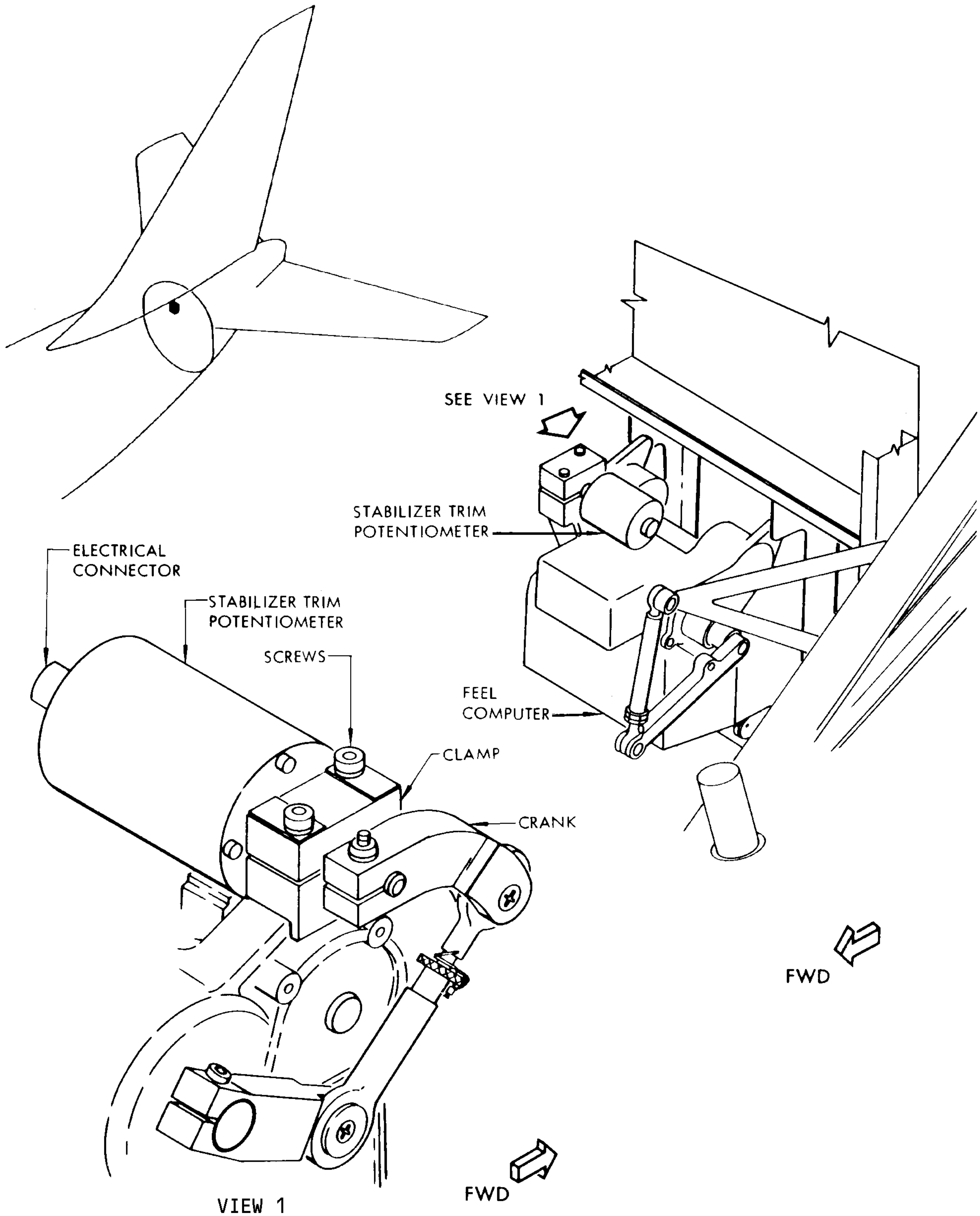
D. Restore Airplane to Normal Configuration
(1) Remove spring scale from control wheel.
(2) Disengage pitch channel.
(3) Remove electrical and hydraulic power from airplane if no longer required.

STABILIZER TRIM POTENTIOMETER - REMOVAL/INSTALLATION

1. General
   A. Stabilizer trim potentiometer is mounted above the feel computer, next to the stabilizer jackscrew (Fig. 401).
2. Remove Stabilizer Trim Potentiometer
   A. Remove access door 3701. Refer to Access Doors and Panels, Chapter 12.
   B. Remove electrical connector from potentiometer.
   C. Loosen nut on potentiometer crank bolt and remove crank from potentiometer shaft (Fig. 401).
   D. Loosen two screws in clamp and remove potentiometer.
3. Install Stabilizer Trim Potentiometer
   A. Set horizontal stabilizer at 0 degree, equal to B dimension of 41.57 ±0.10 inches (Ref Chapter 27, Horizontal Stabilizer Trim Control System).
   B. Align index marks on potentiometer housing and shaft.
   C. Install potentiometer with index mark in vertical position pointing upward.
   D. Tighten two screws in clamp to 40 to 50 pound-inches torque (Fig. 401).
   E. Without disturbing position of potentiometer, install crank on potentiometer shaft.

      CAUTION: ENSURE THAT THE POTENTIOMETER CRANK IS CORRECTLY ORIENTED AS SHOWN IN FIG. 401 FOR PROPER AUTO STABILIZER TRIM OPERATION. DAMAGE TO CRANK COULD RESULT IF INSTALLED IN AN INCORRECT ORIENTATION.

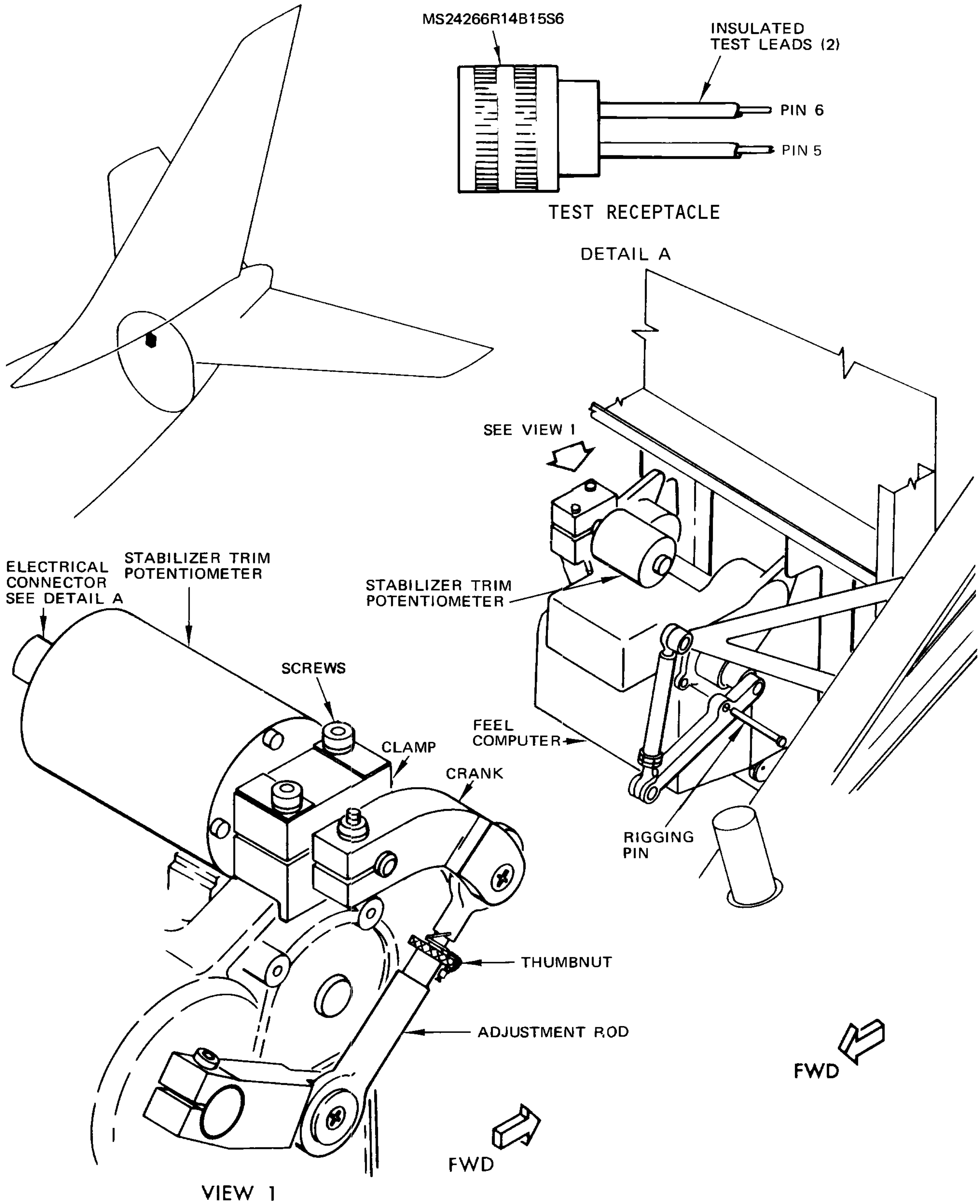
   F. Tighten nut on potentiometer crank bolt to 12 to 20 pound-inches torque.
   G. Adjust potentiometer. Refer to Stabilizer Trim Potentiometer - Adjustment/Test.
   H. Check operation of stabilizer trim potentiometer.
   I. Replace access door.



Stabilizer Trim Potentiometer Installation
Figure 401

STABILIZER TRIM POTENTIOMETER - ADJUSTMENT/TEST

1. Stabilizer Trim Potentiometer Adjustment
   A. Equipment and Materials
      (1) Test Receptacle - MS24266R14B15S6, with insulated test leads from pins 5 and 6
      (2) Rigging Pin E3 - 0.309 (+0.003/-0.000)-inch diameter (MS20392-4)
      (3) Wheatstone Bridge - Model 5430-A, Leeds and Northrup
   B. Adjust Stabilizer Trim Potentiometer
      (1) Remove access door 3701. Refer to Chapter 12, Access Doors and Panels.
      (2) Set horizontal stabilizer at 0 degree, equal to B dimension of 41.57 ±0.01 inches. Refer to Chapter 27, Horizontal Stabilizer Trim Control System.
      (3) Install rigging pin E3 (Fig. 501).
      (4) Disconnect electrical receptacle and install test receptacle on trim potentiometer.
      (5) Connect Wheatstone bridge to test leads.
      (6) Measure resistance with respect to zero index resistance stamped on potentiometer nameplate. Check that deviation is less than ± 10 ohms. If deviation is less than ± 40 ohms, adjust per step (a). If deviation is greater than ± 40 ohms, adjust per step (b).
         (a) Remove lockwire and turn knurled thumbnut on microadjustment rod until resistance measured is zero index resistance. Lockwire thumbnut to microadjustment rod and check that resistance is within ± 10 ohms tolerance (Fig. 501).
         (b) Loosen two potentiometer clamp screws enough to allow position potentiometer housing to be rotated. Rotate potentiometer housing slowly until resistance deviation from zero index is less than ± 40 ohms. Tighten clamp screws to 40-50 pound-inches torque, being careful not to disturb adjustment. Perform step (a) for fine adjustment.
      (7) Remove rigging pin.
      (8) Remove test receptacle and replace electrical receptacle.
      (9) Replace access door.



Stabilizer Trim Potentiometer Installation
Figure 501

ROLL CONTROL WHEEL STEERING FORCE TRANSDUCER - REMOVAL/INSTALLATION

1. General
   A. The roll force transducer is located on the aileron drum at the bottom of the captain's control column.
2. Equipment and Materials
   A. Shims (2) - 0.021 +0.003/-0.0 inch
3. Remove Roll Force Transducer
   A. Open roll channel circuit breakers on load control center P6.
   B. Open lower access door 1103 (Ref Chapter 12, Access Doors and Panels).
   C. Disconnect force transducer cable from airplane wiring and clamps (Fig. 401).
   D. Unwind force transducer cable from shaft assembly spool.
   E. Remove two bolts and nuts and remove force transducer from aileron drum.
4. Prepare to Install Roll Force Transducer
   A. Provide electrical power to the airplane and energize load control center P6.
   B. Pressurize hydraulic systems A and B (Ref 29-11-0 MP and 29-12-0 MP).

      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

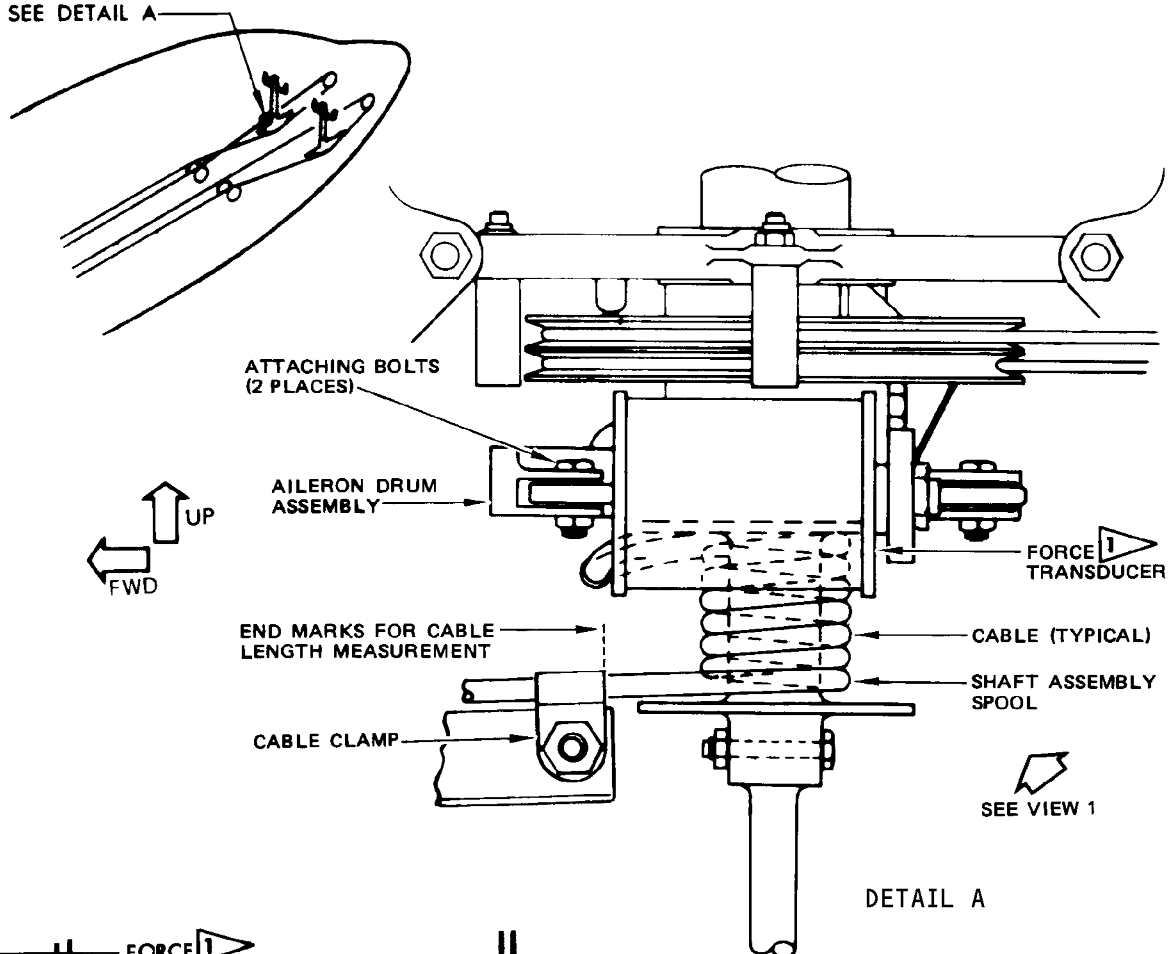
   C. Ensure hydraulic power is applied to the aileron power control units. Center control columns and tag indicating they are not to be moved.

      NOTE: Power control units must be operative to ensure cables are aligned properly in relation to force transducer.

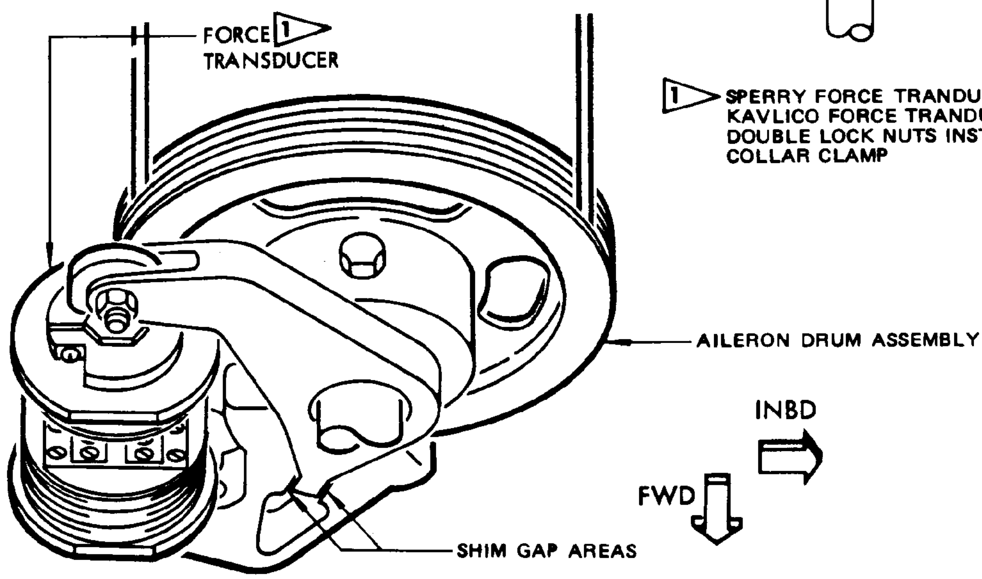
5. Install Roll Force Transducer
   A. Insert shims between shaft and drum (Fig. 401). Gap setting should be 0.021 +0.003/-0.0 inch.
   B. Prior to installing new force transducer mark cable length as follows by measuring from edge of transducer clamp along cable.

      NOTE: Do not disturb cable clamp on transducer.

      (1) On 10-61072-1, -3, or -4 transducer, mark cable for length of 33.75 ±0.25 inches.
      (2) On 10-61072-6 transducers, mark cable for length of 40.75 ±0.25 inches.
   C. Hold force transducer in position and install two bolts and nuts.
   D. Adjust force transducer (Ref Roll Control Wheel Steering Force Transducer - Adjustment/Test).
   E. Install lockwire on adjustable rod end of the force transducer.



DETAIL A



1 SPERRY FORCE TRANDUCER SHOWN– KAVLICO FORCE TRANDUCERS HAVE DOUBLE LOCK NUTS INSTEAD OF COLLAR CLAMP

VIEW 1

Roll Force Transducer Installation
Figure 401

F. Remove shims.
G. Check that gap dimensions are 0.021 +0.003/-0.0 inch.
H. Rotate captain's control wheel full left and hold.
I. Coil force transducer cable firmly around shaft spool (detail A).
   (1) On 10-61072-1, -3, or -4 transducer, wrap cable around spool seven turns.
   (2) On 10-61072-6 transducer, wrap cable around spool nine turns.
J. Hold cable and secure cable with clamps, at cable mark.
K. Check for free movement of control wheel in both directions.
L. Connect force transducer to airplane wiring.
M. Close access door.
N. Remove control column tag and test roll control wheel steering force transducer (Ref Adjustment/Test).

ROLL CONTROL WHEEL STEERING FORCE TRANSDUCER - ADJUSTMENT/TEST

1. General
   A. These procedures are intended to verify the force transducer is adjusted and operating properly. The adjustment of the force transducer is fairly critical. If not adjusted properly, the roll channel may not engage.
2. Roll Force Transducer Adjustment
   A. Equipment and Materials
      (1) Test Box - F80218-42
      (2) Adapter Cable - F80218-48
      (3) VTVM - Hewlett Packard, Model 410C
      (4) Shims (2) - 0.021 +0.003/-0.0 inch
   B. Prepare for Adjustment
      (1) Provide electrical power to the airplane and energize load control center P6.
      (2) Pressurize hydraulic systems A and B (Ref 29-11-0, MP and 29-12-0, MP).

      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

      (3) Ensure hydraulic power is applied to both aileron power control units.
      (4) Rotate control wheel ± 30 degrees and gradually center control wheel. Tag control wheel indicating it should not be moved.

      NOTE: Power control units must be operative to ensure cables are aligned properly in relation to force transducer.

      (5) Remove system A and B hydraulic power (Ref 29-11-0 and 29-12-0).
      (6) Open roll channel circuit breakers on load control center P6.
      (7) Open lower nose access door 1103 (Ref Chapter 12, Access Doors and Panels).
   C. Adjust Roll Force Transducer
      (1) Disconnect force transducer cable connector from airplane wiring (Fig. 501).
      (2) Insert shims between shaft and drum. Gap settings should be 0.021 +0.003/-0.0 inch (View 1, Fig. 501).
      (3) Remove bolt from adjustable end of force transducer.

(4) Connect force transducer to captain's connector with adapter cable and energize test box. Allow transducer to warm up and stabilize for 15 minutes.

<u>CAUTION</u>: USE ONLY 115 VOLTS, 400 HZ AC. USE OF 60 HZ VOLTAGE MAY DAMAGE TRANSDUCER.

(5) Position the SYSTEM A - SYSTEM B switch in the SYSTEM A position.
(6) Position test box SELECT switch S1 to CAPT and measure output voltage at static conditions. The output should not exceed 80 mv. If the null is higher, refer to removal/installation then loosen electrical cable to determine if null can be lowered by an adjustment to the cable installation. Tighten cable clamps.

<u>NOTE</u>: The fine and coarse adjustments should be checked for some threads showing.

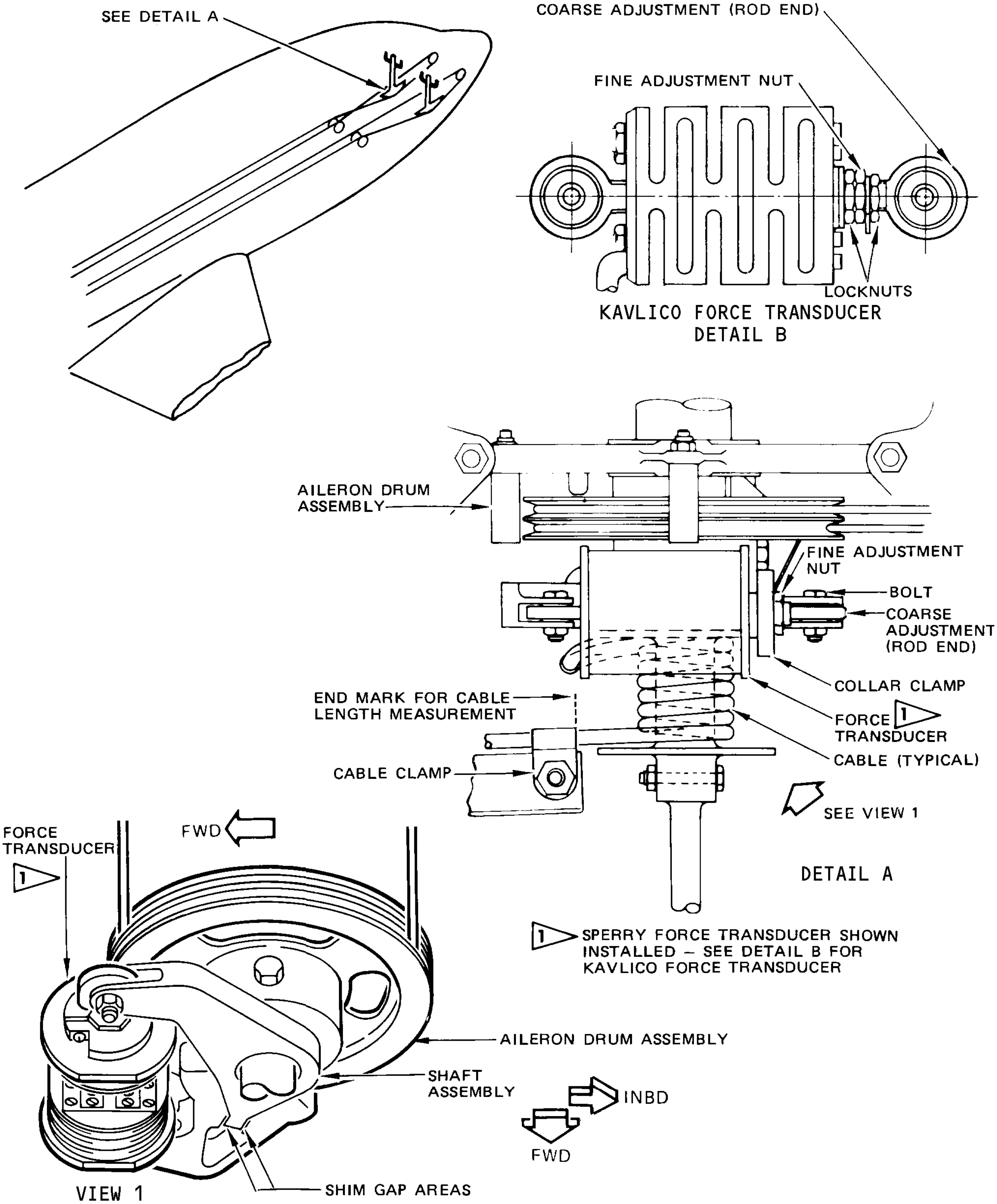
(7) On airplanes with Sperry force transducers, remove lockwire on collar clamp, loosen lockscrew and adjust rod end (coarse adjustment) on each force transducer until mounting bolt can be easily inserted. Insert bolt. On airplanes with Kavlico force transducers, remove lockwire, loosen locknut on adjustable rod end, loosen locknut on adjustable sleeve and adjust rod end (coarse adjustment) on each force transducer until mounting bolt can be easily inserted. Insert bolt.

<u>NOTE</u>: Coarse adjustment is approximately 0.016 inch per turn. Fine adjustment is approximately 0.006 inch per turn.

(8) Measure force transducer null voltage after installation and rotate fine adjustment nut until previous null ± 10 mv is obtained.
(9) On airplanes with Sperry force transducers, tighten lockscrew on collar clamp and install lockwiring. On airplanes with Kavlico force transducers, tighten locknut on adjustable sleeve (pressing washer against end cap of transducer), tighten locknut on rod end and install lockwiring.
(10) Remove shims.
(11) Pressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0).

<u>WARNING</u>: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

(12) Cycle control wheel ± 30 degrees. Gradually center control wheel.
(13) Measure null voltage. Check that null voltage is less than 150 mv.



Roll Force Transducer Adjustment
Figure 501

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(14) Turn off test box and reconnect force transducer to airplane wiring.
(15) Close access panel.
(16) Remove control column tag and test roll control wheel force transducer.

3. Roll Force Transducer Test

A. Equipment and Materials

(1) Adapter Assembly, Control Wheel Torque Tool - F80075-1

B. Prepare to Test Roll Force Transducer

(1) Verify control surfaces are free of obstructions and personnel.
(2) Provide electrical power to the airplane and energize load control center P6.
(3) Pressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0).

WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

(4) Ensure ailerons are operable.
(5) Ensure autopilot, vertical gyro, compass systems and air data computer circuit breakers are closed.

C. Test Roll Force Transducer

(1) Place autopilot system select switch on A or B.
(2) Engage roll channel.

NOTE: If the force transducer is not adjusted properly, the roll channel may not engage.

(3) Position autopilot heading select switch to HDG SEL.
(4) Connect adapter and torque wrench to hub of captain's control wheel.
(5) Turn the control wheel out of detent. The heading select switch should return to the center position when the scale indicates 5 to 13 pound-feet.

NOTE: If the transducer is not operating, the heading select switch will remain in HDG SEL.

D. Restore Airplane to Normal Configuration

(1) Disengage roll channel.
(2) Remove torque wrench and adapter from control wheel.
(3) Determine whether there is any further need for electrical or hydraulic power on the airplane; if not, remove power.

ELEVATOR NEUTRAL SHIFT SENSOR - REMOVAL/INSTALLATION

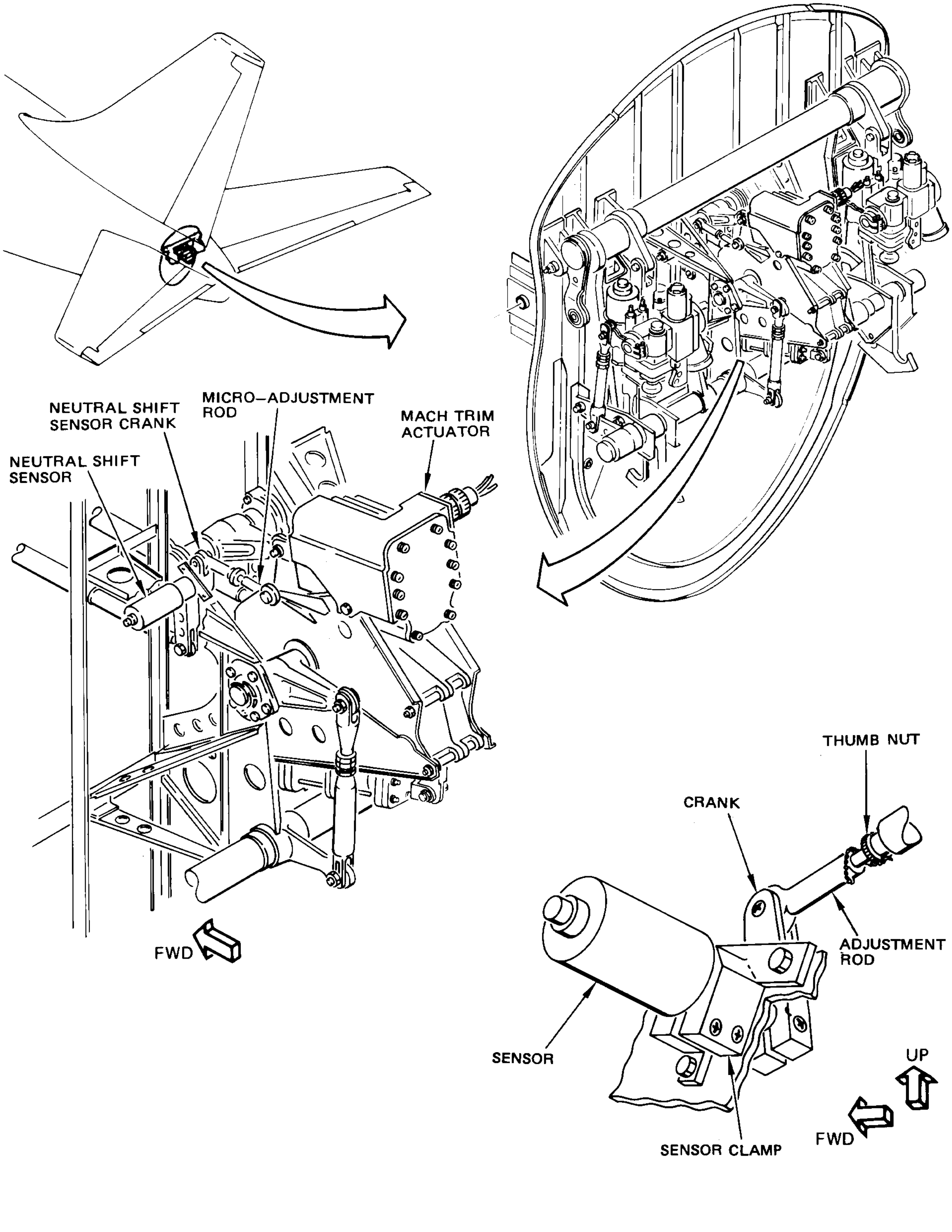
1. General
   A. The elevator neutral shift sensor, mounted on the feel and centering unit, is located immediately aft of bulkhead station 1156 (Ref Chapter 27, Elevator and Tab Control System).
2. Remove Elevator Neutral Shift Sensor
   A. Remove tail cone access panel 3802 (Ref Chapter 12, Access Doors and Panels).
   B. Remove electrical connector from sensor (Fig. 401).
   C. Loosen screw in sensor crank and remove crank from sensor shaft.
   D. Loosen two screws in sensor clamp and remove sensor.
3. Install Elevator Neutral Shift Sensor
   A. Set horizontal stabilizer at 0 degrees, equal to B dimension of 41.57 ±0.05 inches (Ref Chapter 27, Horizontal Stabilizer Trim Control System).
   B. Ensure mach trim actuator is extended (Ref Chapter 27, Elevator and Tab Control Systems).
   C. Align index marks on sensor housing and sensor shaft.

      NOTE: If index marks on sensor are difficult to identify, see Fig. 402 for positioning sensor to electrical zero position.

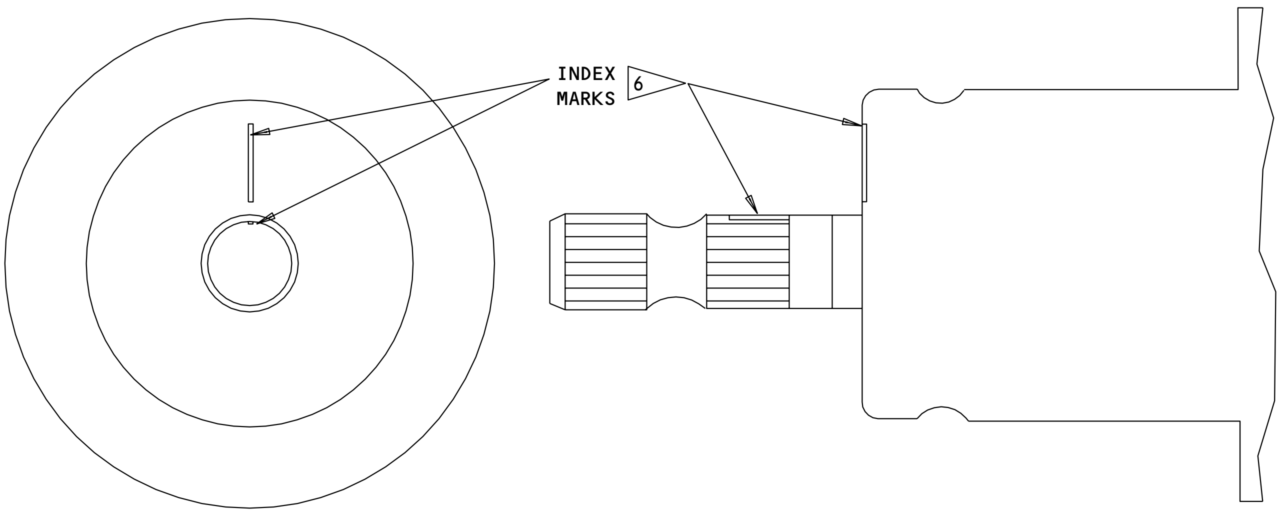
   D. Install sensor with index mark in vertical position pointing upward.
   E. Tighten two screws in sensor clamp and tighten nuts within torque range of 8 to 12 pound-inches (Fig. 401).

      CAUTION: ENSURE THAT ELEVATOR NEUTRAL SHIFT POSITION TRANSMITTER CRANK IS CORRECTLY ORIENTED IN THE UP DIRECTION.

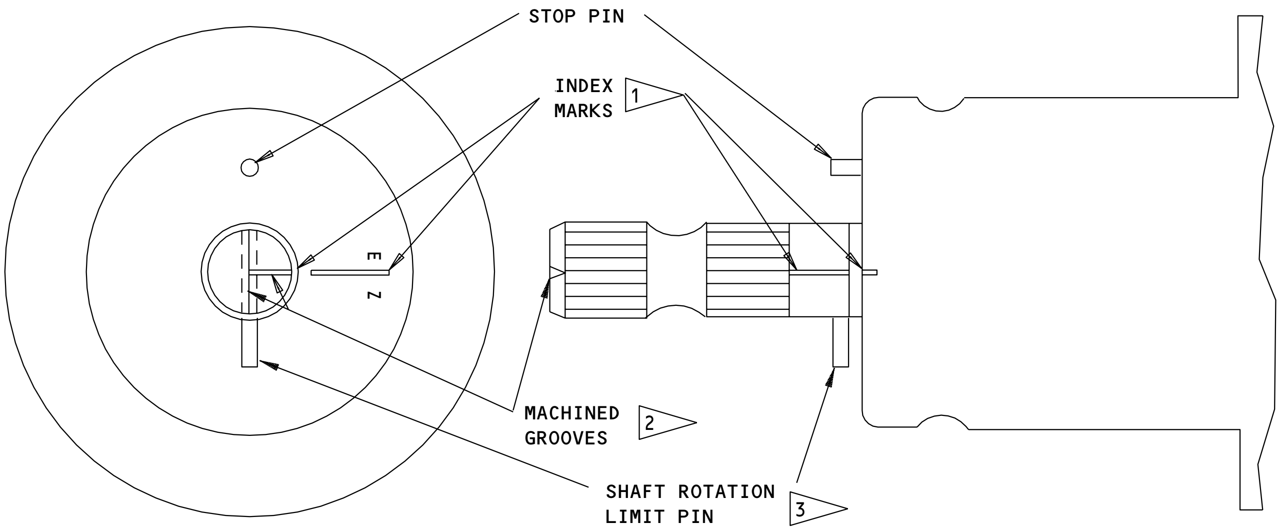
   F. Without disturbing position of sensor, install sensor crank on sensor shaft.
   G. Tighten screw in sensor crank and tighten nut within torque range of 8 to 12 pound-inches.
   H. Adjust and test neutral shift sensor (Ref Elevator Neutral Shift Sensor - Adjustment/Test).
   I. Replace access panel.



Elevator Neutral Shift Sensor Installation
Figure 401



POSITION SENSOR INDEXING MARKS 4



POSITION SENSOR INDEXING MARKS 5

1 INDEX MARKS MAY BECOME OBLITERATED. AT ELECTRICAL ZERO POSITION, SHAFT ROTATION LIMIT PIN IS 180 DEGREES FROM STOP PIN

2 MACHINED GROOVE ACROSS DIAMETER OF SHAFT END IS ALINGED WITH SHAFT ROTATION LIMIT PIN. RADIUS GROOVE IS ALIGNED WITH SHAFT INDEX MARK AND 90 DEGREES FROM SHAFT ROTATION LIMIT PIN

3 PIN IS PRESSED COMPLETELY THRU SHAFT. FLUSH END OF PIN IS ADJACENT TO STOP PIN AT ELECTRICAL ZERO POSITION

4 SPERRY

5 ASTRO

6 INDEX MARKS INSCRIBED ON SHAFT AND CASE

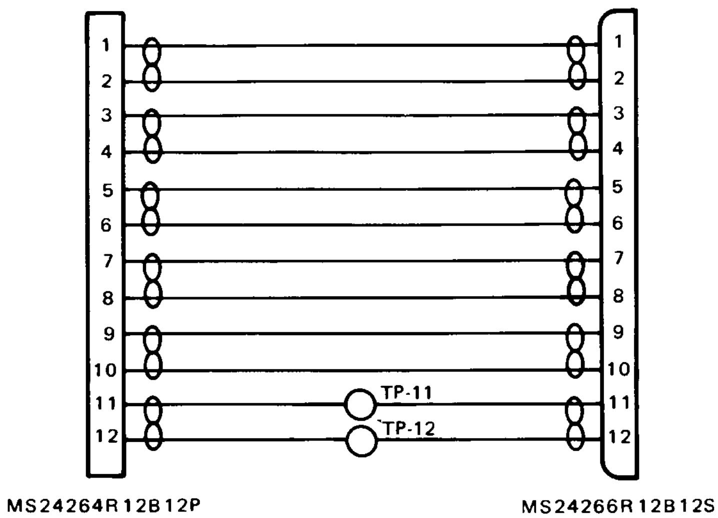
Position Sensor Indexing Marks
Figure 402

ELEVATOR NEUTRAL SHIFT SENSOR - ADJUSTMENT/TEST

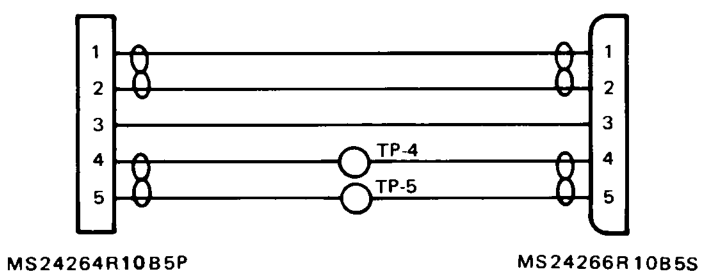
1. Elevator Neutral Shift Sensor Adjustment
   A. General
      (1) The entire pitch channel of the autopilot system and all elevator rigging must be fully adjusted and operable before the following adjustment can be performed. The mach trim system must also be operable. Two breakout cables must be fabricated (Fig. 501).
      (2) Elevator neutral shift sensor voltage is proportional to the elevator position transducer voltage; therefore, on airplanes with A and B system switching on the autopilot control panel, the average output from both elevator position transducers must be determined and multiplied by 0.707 to determine the neutral shift sensor voltage. The following procedures determine the voltage and establish the null point voltage of the neutral shift sensor.
   B. Equipment and Materials
      (1) VTVM - Hewlett Packard, Model 410C
      (2) Breakout cables (Fig. 501)
   C. Prepare to Adjust Elevator Neutral Shift Sensor
      (1) Apply electrical power to the airplane and ensure load control center P6 is energized.
      (2) Depressurize hydraulic systems A and B (Ref 29-11-0, MP and 29-12-0, MP).

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

      (3) Open pitch channel circuit breakers and ensure pitch axis is disengaged.
      (4) Ensure MACH TRIM circuit breakers are closed.
      (5) Ensure mach trim actuator is fully extended (Ref Chapter 27, Elevator and Tab Control System).
      (6) Remove tail cone access panel 3802 (Ref Chapter 12, Access Doors and Panel).
      (7) Set horizontal stabilizer at 0 degrees (3 units trim), equal to B dimension of 41.57 ±0.05 inches (Ref Chapter 27, Horizontal Stabilizer Trim Control System).
   D. Adjust Elevator Neutral Shift Sensor (On airplanes with A and B system switching on autopilot control panel)
      (1) Disconnect D-373 from left power control unit and connect the power control unit breakout cable between D-373 and the power control unit.



POWER CONTROL UNIT BREAKOUT CABLE



NEUTRAL SHIFT SENSOR BREAKOUT CABLE

NOTE: USE BMS 13-11, TYPE I, CLASS A, NO. 22 WIRE OR EQUIVALENT

Breakout Cables
Figure 501

(2) Ensure the autopilot INTLK circuit breaker is closed and close the pitch channel circuit breakers.
(3) Position the system select switch on the autopilot control panel to A.
(4) Pressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0 MP).

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(5) Jiggle aft control quadrant to ensure that system is centered.
(6) Connect VTVM to test points 11 and 12 and record voltage. Voltage should be approximately 2 volts.
(7) Depressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0 MP). Open pitch channel circuit breakers.
(8) Disconnect breakout cable and VTVM from D-373 and power control unit; reconnect D-373 to power control unit.
(9) Disconnect D-375 from right power control unit and connect breakout cable between D-375 and the power control unit.
(10) Disconnect D-377 from neutral shift sensor and connect neutral shift sensor breakout cable between D-377 and neutral shift sensor.
(11) Select B with the system select switch on the autopilot control panel.
(12) Close the pitch channel circuit breakers and pressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0 MP).
(13) Jiggle aft control quadrant to ensure that system is centered.
(14) Connect VTVM to test points 11 and 12 and record voltage.
(15) Average out voltages obtained in steps (6) and (14) by adding together and dividing by two.
(16) Determine neutral shift sensor voltage by multiplying answer obtained in step (15) by 0.707.
(17) Connect VTVM to test points 4 and 5 on neutral shift sensor breakout cable.
(18) Loosen two screws on neutral shift sensor clamp only enough to allow the sensor to be rotated in the support.
(19) Rotate neutral shift sensor housing slowly in both directions until voltage is as calculated in step (16), ± 0.3 volts.

NOTE: Index marks cannot be seen with crank installed. When facing sensor, sensor housing should be rotated clockwise approximately 7 degrees.

(20) Tighten sensor clamp screws and nuts within torque range of 8 to 12 pound-inches.

(21) Remove lockwiring and fine adjust sensor by using knurled thumbnut on microadjustment rod until VTVM reading is within ± 0.03 volts of calculated voltage.
(22) Drive stabilizer leading edge down and verify that voltage decreases. This is to ensure that neutral shift sensor is set to the correct side of null. If not, repeat steps (18) thru (22) until correct null is obtained.
(23) Lockwire knurled thumbnut to microadjustment rod.

E. Adjust Elevator Neutral Shift Sensor (On airplanes with B system only on autopilot control panel)
(1) Perform steps 1.D.(2), 1.D.(9) thru 1.D.(14), 1.D.(16) thru 1.D.(23).

NOTE: In step 1.D.(16) use answer obtained in 1.D.(14).

F. Return Airplane to Normal
(1) Depressurize hydraulic systems A and B (Ref 29-11-0 and 29-12-0).
(2) Open pitch channel circuit breakers.
(3) Disconnect breakout cables and VTVM.
(4) Connect D-375 to right power control unit and D-377 to neutral shift sensor.
(5) Replace access panel.
(6) Close pitch channel circuit breakers.
(7) If no longer required, remove electrical power from the airplane.

2. Elevator Neutral Shift Sensor Test

A. Equipment and Materials
(1) VTVM - Hewlett Packard, Model 410C, or equivalent with suitable leads and adapter to use test points on front of pitch channel.

B. Prepare to Test Elevator Neutral Shift Sensor
(1) Apply electrical power to the airplane and energize load control center P6.
(2) Provide systems A and B hydraulic power (Ref 29-11-0, MP and 29-12-0, MP).

WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

(3) Ensure the following systems are energized and operating (Ref Chapter 34):
(a) Attitude Reference System
(b) Compass System
(c) Air Data Pressure Instruments
(4) Ensure autopilot pitch channel and INTLK circuit breakers are closed.

C. Test Elevator Neutral Shift Sensor
(1) Set the stabilizer for three units of trim.
(2) Open autopilot stabilizer trim circuit breaker.
(3) Move the control column back and forth through neutral a few times to ensure elevators are operating.
(4) Engage the autopilot pitch axis.
(5) Ground TP-6 on the pitch channel.
(6) Connect VTVM to TP-10. Indicated voltage should be 1.75 +1.0/-0.5 volts ac.
(7) Ground TP-15. The voltage at TP-10 should decrease to 0.0 ±0.5 volts ac.
(8) Remove grounds from TP-6 and TP-15.
(9) Remove VTVM from TP-10.
(10) Disengage the pitch axis.

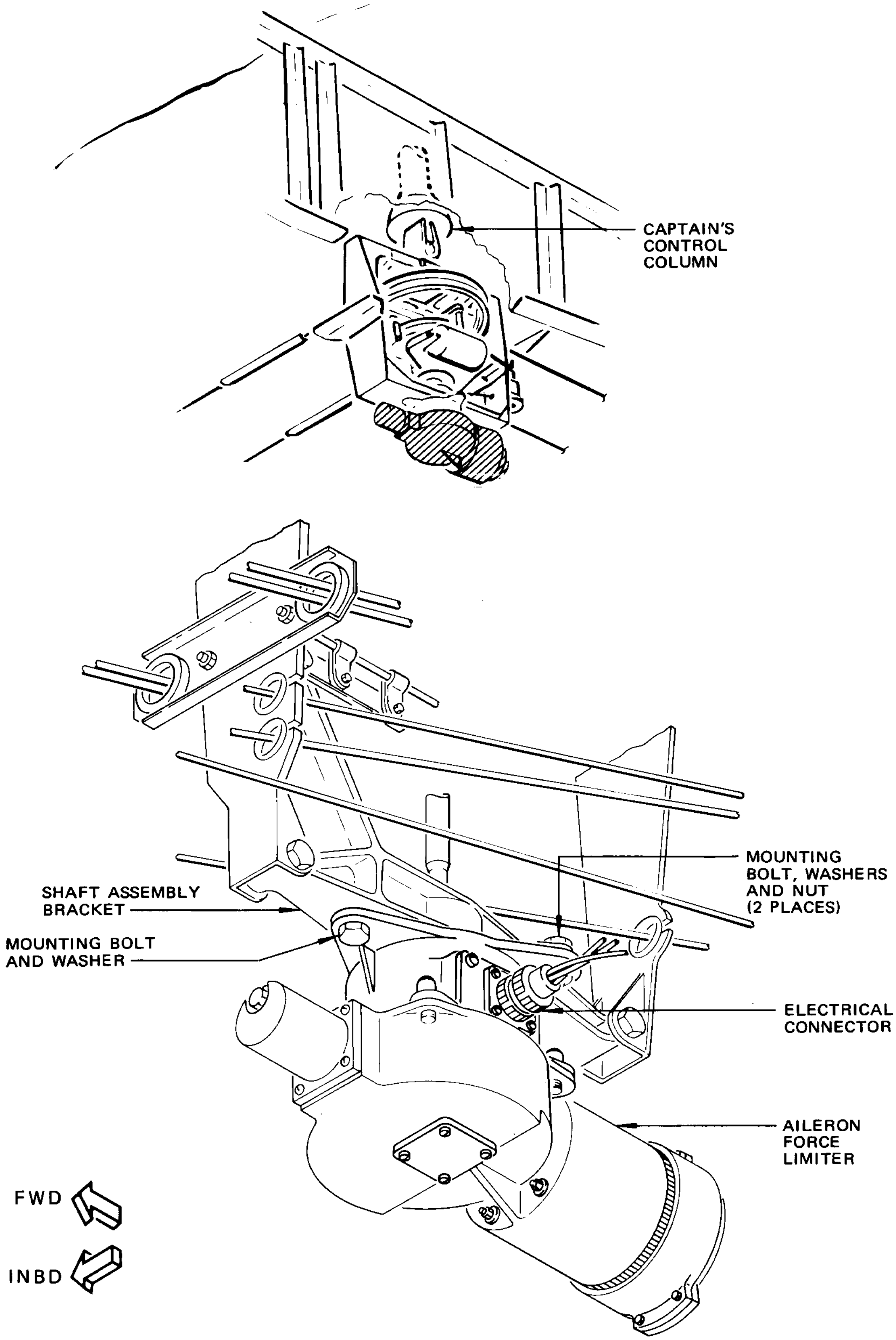
D. Return Airplane to Normal
(1) Depressurize hydraulic systems A and B. Refer to 29-11-0 and 29-12-0.
(2) Close autopilot stabilizer trim circuit breaker.
(3) Determine whether there is any further need for electrical power on the airplane; if not, remove power.

AILERON FORCE LIMITER - REMOVAL/INSTALLATION

1. General
   A. The aileron force limiter is located at the bottom of the captain's control column on the aileron drum shaft assembly (Fig. 401).
2. Remove Aileron Force Limiter
   A. Open lower nose access door 1103. Refer to Access Doors and Panels, Chapter 12.
   B. Disconnect force limiter electrical connector.
   C. Placing palm of hand underneath force limiter, remove three bolts, washers and nuts, and remove force limiter.
3. Install Aileron Force Limiter
   A. Grasping force limiter firmly, mate spline with shaft assembly and align three holes in force limiter fitting with holes in shaft assembly bracket.

      NOTE: Ensure that smaller diameter of force limiter is facing towards nose of airplane.

   B. Install bolt and washer in forward hole of limiter and bracket and two bolts, washers and nuts in rear holes (rear bolt shafts point down). Tighten bolts.
   C. Connect electrical connector to force limiter.
   D. Check operation of force limiter. Refer to Aileron Force Limiter - Adjustment/Test.
   E. Close access doors.



Aileron Force Limiter Installation
Figure 401

AILERON FORCE LIMITER - ADJUSTMENT/TEST

1. Aileron Force Limiter Test
   A. General
      (1) This test ensures that the aileron force limiter operates properly and assumes that the rest of the autopilot system is operating normally.
   B. Prepare to Test Aileron Force Limiter
      (1) Provide electrical power to the airplane and energize load control center P6.
      (2) Pressurize hydraulic system A or B (Ref 29-11-0 MP and 29-12-0 MP).

      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

      (3) Ensure that all autopilot, vertical gyro, compass system, and air data computer circuit breakers are closed.
   C. Test Aileron Force Limiter
      (1) Engage autopilot AIL engage switch.
      (2) Center heading bug on the pilot's HSI and engage heading select mode.
      (3) Rotate heading bug 45 degrees to the left.
      (4) Observe that the control wheel rotates CCW and stops between 24 and 28 degrees.
      (5) Re-center heading bug and then rotate 45 degrees to the right.
      (6) Observe that the control wheel rotates clockwise and stops between 24 and 28 degrees.
      (7) Observe that the difference between the two control wheel reading is no greater than 4 degrees.
      (8) Remove electrical and hydraulic power from airplane if no longer required.

YAW DAMPER COUPLER - REMOVAL/INSTALLATION

1. General

NOTE: For aircraft with coupler P/N 4084042 installed (incorporating SB 27A1206), see 22-12-01.

A. The yaw damper coupler is retained in the electronic rack by cam-action levers which hook over forks on the shelf.

2. Remove Coupler

A. Open yaw damper AC and DC circuit breakers on P6 panel. Wait 3 minutes for rate gyro rundown (if required).

B. Release coupler by pressing triggers then pull ejection levers to release unit.

C. Remove coupler.

3. Install Coupler

A. Place coupler in rack and simultaneously push ejection levers up until levers are latched per 20-10-111, Electrical/Electronic Black Box - Maintenance Practices.

B. Check that yaw damper self-test switch is positioned to OFF.

C. Close yaw damper circuit breakers.

4. Installation Test

A. Provide electrical power to airplane and energize load control center P6.

B. Pressurize hydraulic system A and B then apply hydraulic pressure to control surfaces (Ref 29-11-0 and 29-12-0).

WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, SPOILER SPEED BRAKE AND RUDDER SYSTEM. SURFACES MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

C. Check that flight control A and B switches are on.

D. Engage yaw damper and check that yaw damper disengage warning light is off.

E. Position yaw damper engage switch to OFF position. Verify that yaw damper disengaged light illuminates.

F. Engage yaw damper, check that rudder is centered and remains steady.

G. Position cockpit yaw damper test switch to left and hold. Observe rudder deflects left and returns to center. Center test switch and observe that rudder deflects right then returns to center. Position test switch to right and hold. Observe rudder deflects right and returns to center. Center test switch and observe that rudder deflects left then returns to center.

NOTE: Do not hold test switch longer than 10 seconds, torquing coil will overheat.

H. Remove hydraulic and electrical power if no longer required.

AUTOPILOT PITCH COMPUTER - REMOVAL/INSTALLATION

1. General
   A. The autopilot pitch computer is retained in the electronic rack by cam-action levers which hook over forks on the shelf.
2. Remove Computer
   A. Open autopilot circuit breakers on P6 or P18 panel. Wait 3 minutes for rate gyro rundown (if applicable).
   B. Release computer by pressing triggers then pull ejection levers to release unit.
   C. Remove computer.
3. Install Computer
   A. Place computer in rack and simultaneously push ejection levers up until levers are latched per AMM 20-10-111/201, Electrical/Electronic Black Box.
   B. Check that pitch computer self-test switch is positioned to OFF.
   C. Close autopilot circuit breakers.
4. Installation Test
   A. Provide electrical power to airplane buses.
   B. Open radio altimeter circuit breakers.

      NOTE: Do not apply rate gyro test signals for longer than 10 seconds. On airplanes with computers using derived rate circuits, ignore steps using rate gyro switch.

C. Perform Pitch Control Channel Self-Test:

STEP	ACTION	OBSERVATION
(1)	With self-test switch in OFF position	Meter reads in RED range.
(2)	Rotate self-test switch from OFF to position 1	Meter moves to right at a constant rate for approximately 10 seconds. Rate of movement changes in the YELLOW range.
(3)	Rotate self-test switch from position 1 to 2	Self-test meter initially deflects to right stop, then decreases and settles in YELLOW range.
(4)	Move rate gyro switch to the right (+)	Self-test meter deflects right momentarily. Note meter indication and gyro are smooth.
(5)	Move rate gyro switch to the left (-)	Self-test meter deflects left momentarily. Note meter indication and gyro are smooth.
(6)	Rotate self-test switch from position 2 to 3	Self-test meter initially deflects left, then increases, with a decreasing rate in in the YELLOW range, to the right stop.
(7)	Rotate self-test switch from position 3 to 4	Self-test meter initially reads zero for approximately 10 seconds, increases, then peaks in the GREEN range for approximately 5 seconds, then slowly decreases left for 110 to 170 seconds.
(8)	Rotate self-test switch from position 4 to 5	Self-test meter settles in the YELLOW range.

STEP	ACTION	OBSERVATION
(9)	Rotate self-test switch to OFF position	Meter reads in RED range.
(10)	Engage autopilot	
(11)	Move rate gyro switch to right (+)	Control column moves aft and stops at neutral.
(12)	Move rate gyro switch to left (-)	Control column moves forward and stops at neutral.

D. Disengage autopilot and restore airplane systems to normal.
E. Remove electrical power, if no longer required.

AUTOPILOT ROLL COMPUTER - REMOVAL/INSTALLATION

1. General
   A. The autopilot roll computer is retained in the electronic rack by cam-action levers which hook over forks on the shelf.
2. Remove Computer
   A. Open autopilot circuit breakers on P6 or P18 panel. Wait 3 minutes for rate gyro rundown (if applicable).
   B. Release computer by pressing triggers then pull ejection levers to release unit.
   C. Remove computer.
3. Install Computer
   A. Place computer in rack and simultaneously push ejection levers up until levers are latched per AMM 20-10-111/201, Electrical/Electronic Black Box.
   B. Check that roll computer self-test switch is positioned to OFF.
   C. Close autopilot circuit breakers.
4. Installation Test
   A. Provide electrical power to airplane buses.
   B. Open radio altimeter circuit breakers.

      NOTE: Do not apply rate gyro test signals for longer than 10 seconds. On airplanes with computers using derived rate circuits, ignore steps using rate gyro switch.

C. Perform Roll Control Channel Self-Test:

STEP	ACTION	OBSERVATION
(1)	Engage AIL engage switch and select a VOR frequency	
(2)	Move rate gyro self-test switch to the left (-)	Control wheel moves counterclockwise, then returns to center position.
(3)	Move rate gyro self-test switch to the right (+)	Control wheel moves clockwise, then returns to center position.
(4)	Disengage AIL engage switch	
(5)	Place self-test switch in OFF position	Self-test meter reads in RED range.
(6)	Rotate self-test switch from OFF to position 1	Self-test meter slowly increased to YELLOW range, after 10 seconds in YELLOW range, rapidly decreases to zero, then rapidly increases with a reduction in rate of increase in the YELLOW range, then slowly and continuously increases to right stop.
(7)	Rotate self-test switch from position 1 to 2	Self-test meter initially deflects to left, then settles in YELLOW range.
(8)	Rotate self-test switch from position 2 to 3	Self-test meter initially deflects to left, then settles in YELLOW range within 1 minute.
(9)	Move rate gyro switch to left (-)	Self-test meter reading increases slightly.

STEP	ACTION	OBSERVATION
(10)	Move rate gyro switch to the right (+)	Self-test meter reading decreases slightly.
(11)	Rotate self-test switch from position 3 to 4	Self-test meter initially deflects to zero then increases, momentarily settles in the YELLOW range, then slowly increases to right stop.
(12)	Rotate self-test switch from position 4 to 5	Self-test meter initially deflects left, then increases with 2 rate changes. The second rate change should occur in the YELLOW range. Indicator then increases toward right stop.
(13)	Rotate self-test to OFF	

D. Disengage autopilot and restore airplane systems to normal.
E. Remove electrical power, if no longer required.

AUTOPILOT ACCESSORY UNIT - REMOVAL/INSTALLATION

1. General
   A. The autopilot accessory unit is retained in the electronic rack by a cam-action lever which hooks on a shelf mounted fork.
2. Remove Accessory Unit
   A. Open autopilot, yaw damper, A/P stabilizer trim and autothrottle (when installed) circuit breakers.
   B. Release unit by pressing trigger to release, then pull ejection lever to release unit.
   C. Remove accessory unit.
3. Install Accessory Unit
   A. Examine mounting tray and connectors for distortion of tray and bent or pushed back connector pins. Repair as required.
   B. Place accessory unit in rack and push ejection handle up until lever is latched per 20-10-111, Electrical/Electronic Black Box - Maintenance Practices.
   C. Close circuit breakers opened in step 2.A.
4. Installation Test
   A. Provide electrical power to airplane buses.
   B. Provide attitude valid and compass valid signals.
   C. Engage autopilot (pitch and roll). Operate stab trim switch on aisle stand to CUTOUT position. Check that autopilot or pitch channel disengages and that A/P disengage warning lights flash.
   D. Disengage autopilot, if engaged. Restore airplane to normal.
   E. Remove electrical power if no longer required.

AUTOPILOT MODE CONTROL PANEL - REMOVAL/INSTALLATION

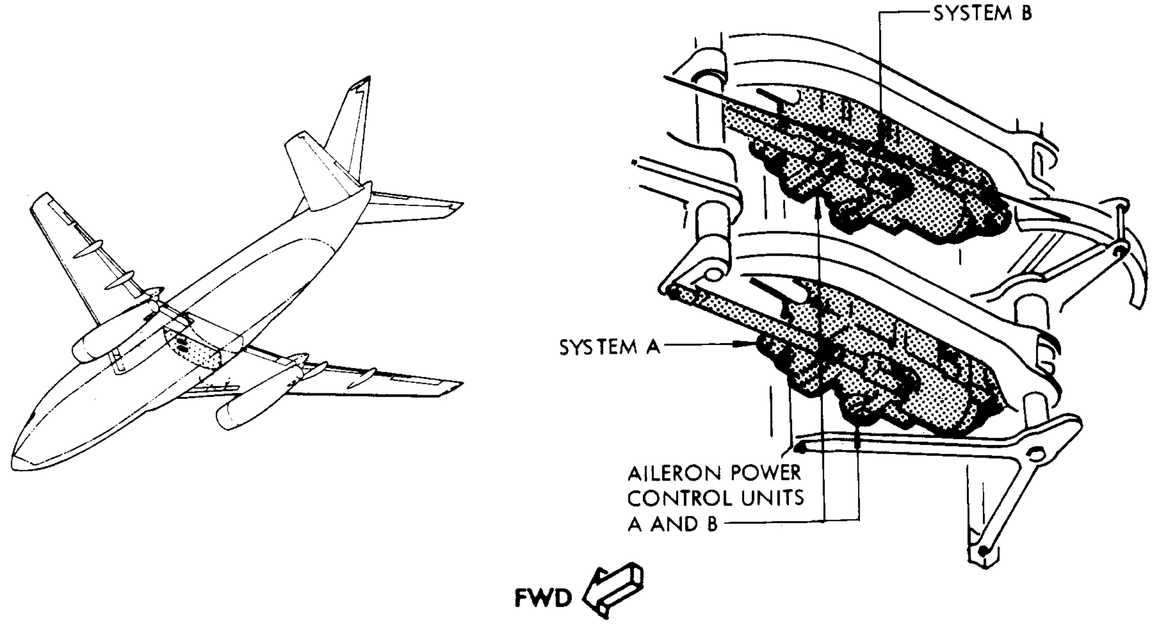
1. General
   A. The autopilot mode control panel is located on the pilot's lightshield. The mode control panel is held in place by four quick release fasteners.
2. Remove Mode Control Panel
   A. Open autopilot and panel lights circuit breakers on P6 and P18 panels.
   B. Release mode control panel by unlatching quick release fasteners.
   C. Withdraw panel from lightshield and disconnect electrical connectors.
3. Install Mode Control Panel
   A. Examine electrical connectors for bent or pushed back pins or sockets and repair as required.
   B. Connect electrical connectors to mode control panel, position panel in place and latch quick release fasteners.
   C. Close autopilot and panel lights circuits breakers.
4. Installation Test
   A. Provide electrical power to airplane buses.
   B. Pressurize hydraulic system A and B then apply hydraulic pressure to control surfaces (Ref 29-11-0 and 29-12-0).

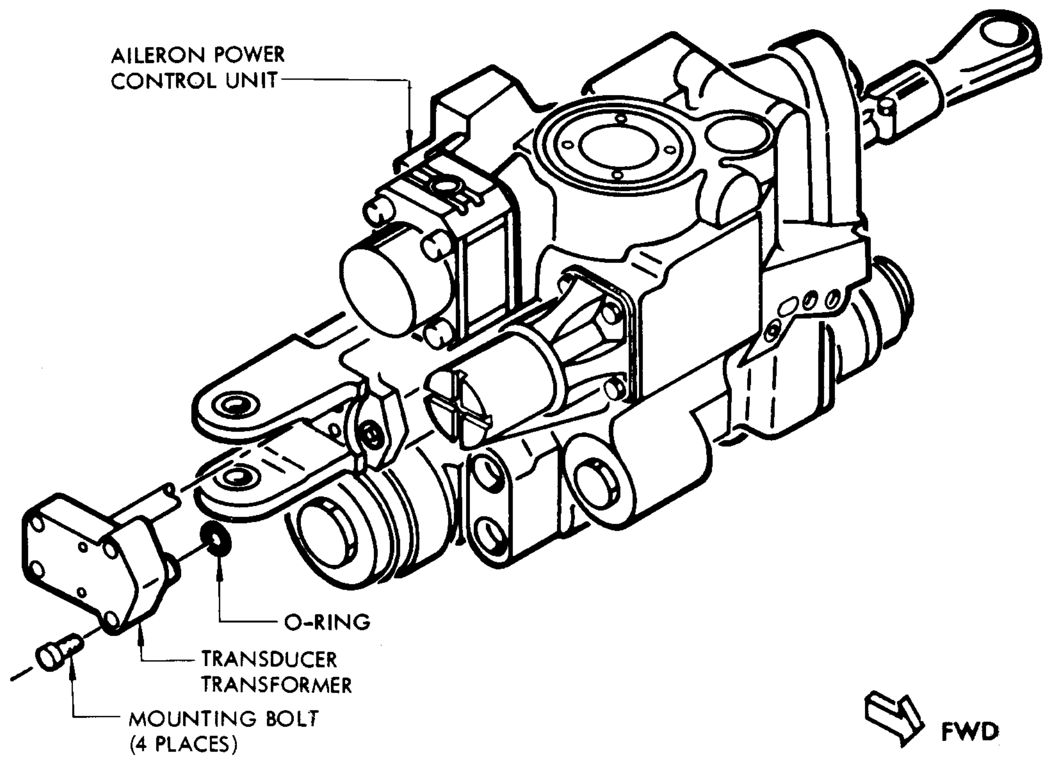
      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, SPOILER SPEED BRAKE AND RUDDER SYSTEMS. SURFACES MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

   C. Check that flight control A and B switches are on.
   D. Verify that one yaw damper is engaged.
   E. Provide attitude and compass valid signals.
   F. Place autopilot stab trim switch on aisle stand to NORMAL position.
   G. Engage AIL engage switch.
   H. Place navigation (NAV) mode select switch (on autopilot control panel) in VOR/LOC position.
   I. Disengage AIL engage switch. Observe that NAV mode select switch returns to MAN position.
   J. Engage autopilot.
   K. Place pitch mode select switch in ALT HOLD position.
   L. Disengage ELEV engage switch. Observe that pitch mode select switch returns to OFF and that ALT HOLD mode cannot be selected.
   M. Engage ELEV engage switch and place pitch mode select switch in ALT HOLD position.
   N. Remove hydraulic power and restore airplane to normal.
   O. Remove electrical power if no longer required.

AILERON POSITION TRANSDUCER - REMOVAL/INSTALLATION

1. General
   A. Aileron position transducer is externally mounted between ends of clevis on right end of aileron power units A and B respectively.
   B. This procedure covers transformer portion of transducer only. Detail disassembly of the power unit is required to remove the transducer rod. (See figure 401.)
2. Remove Transducer Transformer
   A. Remove power unit from wheel well area. Refer to Aileron Power Control Unit, Chapter 27.
   B. Remove mounting bolts and lift transducer transformer from power unit. (See figure 401.)
   C. Cover all openings.
3. Install Transducer Transformer
   A. Verify O-ring is satisfactory or replace.
   B. Install transducer transformer. (See figure 401.)
   C. Install mounting bolts and tighten to 40 to 50 pound-inches.
   D. Lockwire bolts.
   E. Install aileron power unit. Refer to Aileron Power Unit, Chapter 27.
   F. Check for leakage and operation of transducer. Refer to Aileron Position Transducer - Adjustment/Test.





Aileron Position Transducer Installation
Figure 401

AILERON POSITION TRANSDUCER - ADJUSTMENT/TEST

1. Aileron Position Transducer Test
   A. Prepare to Test Aileron Position Transducer
      (1) Verify control surfaces are free to move.
      (2) Provide ground power to the airplane and energize buses on load control center P6.
      (3) Ensure autopilot circuit breakers, located on load control center P6, are closed.
      (4) Pressurize hydraulic systems A and B (Ref 29-11-0 MP and 29-12-0 MP).

      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

   B. Test Aileron Position Transducer
      (1) Place autopilot system select switch to appropriate position.
      (2) Engage AIL engage switch and observe control wheel does not rotate from neutral.
      (3) Move rate gyro self-test switch, located on roll control channel, to the left and observe control wheel moves counterclockwise, then return to center position.

      CAUTION: TO AVOID DAMAGE, DO NOT OPERATE RATE GYRO SELF-TEST SWITCH FOR LONGER THAN 10 SECONDS.

      (4) Move rate gyro self-test switch to the right and observe control wheel moves clockwise, then return to center position.
   C. Restore Airplane to Normal Configuration
      (1) Disengage autopilot engage switch.
      (2) Determine whether there is any further need for electrical power on the airplane; if not, remove external power.

ELEVATOR POSITION TRANSDUCER - REMOVAL/INSTALLATION

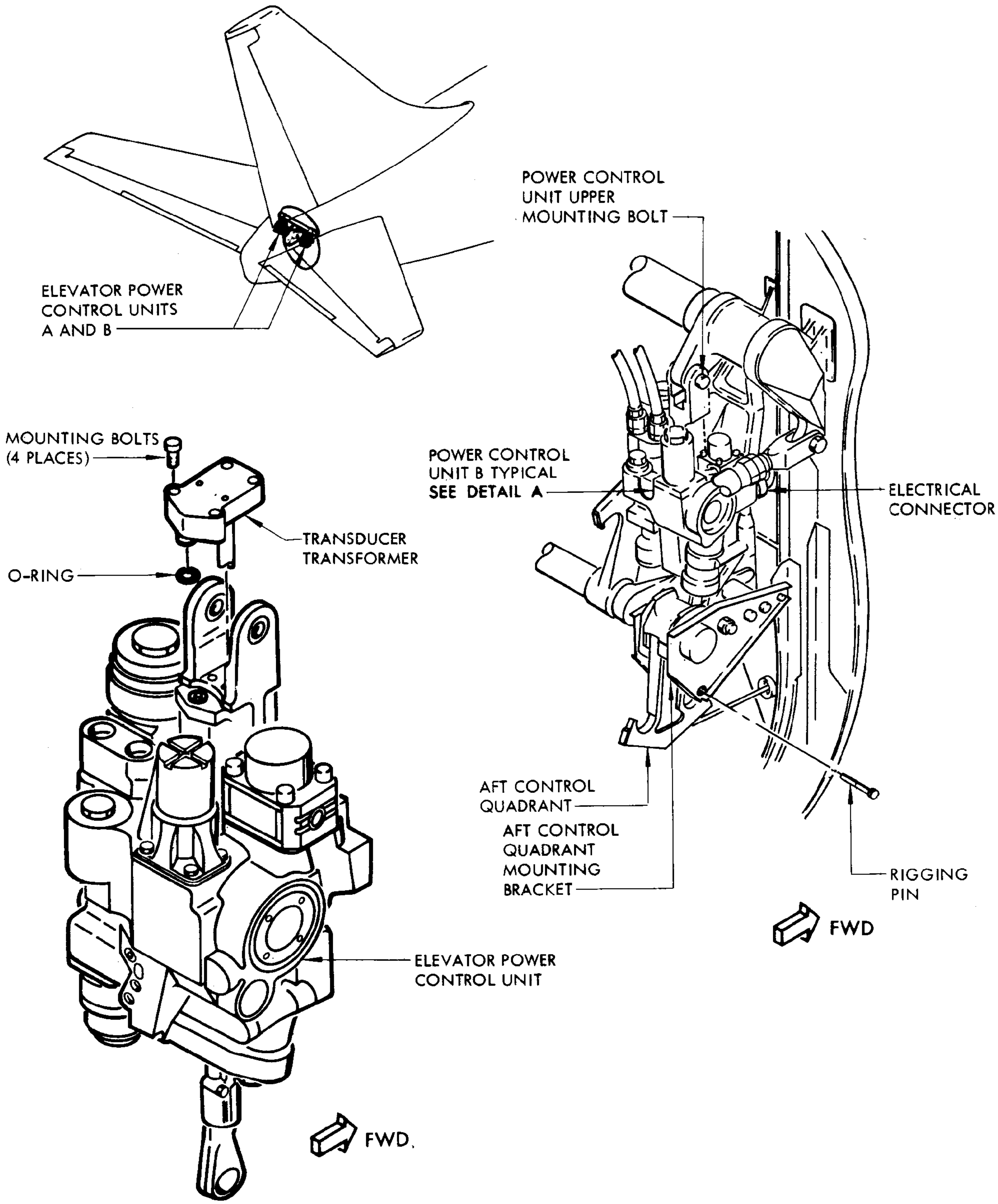
1. General
   A. Elevator position transducer is externally mounted between ends of clevis on upper end of elevator power unit A and B respectively.
   B. This procedure covers transformer portion of transducer only. Detail disassembly of the power unit is required to remove the transducer rod. (See figure 401.)
2. Equipment and Materials
   A. Rigging Pin - 0.309 (+0.003/-0.000)- inch diameter (MS20392-4)
   B. Grease - BMS 3-33 (Perferred)
   C. Grease - MIL-PRF-23827 (Supercedes MIL-G-23827)
3. Prepare to Remove Transducer Transformer
   A. Depressurize hydraulic system A and B. Refer to 29-11-0, Hydraulic System A - Maintenance Practices and 29-12-0, Hydraulic System B - Maintenance Practices.

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

   B. Remove access panel 3802 for both left and right elevator power units. Refer to Chapter 12, Access Doors and Panels.
   C. Install rigging pin in aft control quadrant. (See figure 401.)
4. Remove Transducer Transformer
   A. Remove power control unit upper mounting bolt from output torque tube. (See figure 401.)
   B. Remove transducer mounting bolts and lift transducer from power unit.
   C. Cover all openings.
5. Install Transducer Transformer
   A. Verify O-ring is satisfactory or replace.
   B. Install transducer transformer. (See figure 401.)
   C. Install transducer mounting bolts and tighten to 40 to 50 pound-inches.
   D. Lockwire bolts.
   E. Apply grease to entire shank of mounting bolt and install power control unit upper mounting bolt.
   F. Remove rigging pin from control quadrant.
   G. Pressurize hydraulic system A and B (Ref 29-11-0 and 29-12-0).

      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

   H. Check for hydraulic leakage and operation of transducer. Refer to Elevator Position Transducer - Adjustment/Test.



Elevator Position Transducer Installation
Figure 401

I. Replace access panel.

ELEVATOR POSITION TRANSDUCER - ADJUSTMENT/TEST

1. Elevator Position Transducer Test
   A. Prepare to Test Elevator Position Transducer
      (1) Verify control surfaces are free to move.
      (2) Provide ground power to the airplane and energize buses on load control center P6.
      (3) Ensure autopilot circuit breakers, located on load control center P6, are closed.
      (4) Pressurize hydraulic systems A and B (Ref 29-11-0 MP and 29-12-0 MP).

      WARNING: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

   B. Test Elevator Position Transducer
      (1) Place autopilot system select switch to appropriate position.
      (2) Engage autopilot engage switch and observe control column does not drift forward or aft from neutral.
      (3) Move the rate gyro switch, located on the pitch control channel, to the right and observe control column moves aft, then returns to center position.

      CAUTION: TO AVOID DAMAGE, DO NOT OPERATE RATE GYRO SELF-TEST SWITCH FOR LONGER THAN 10 SECONDS.

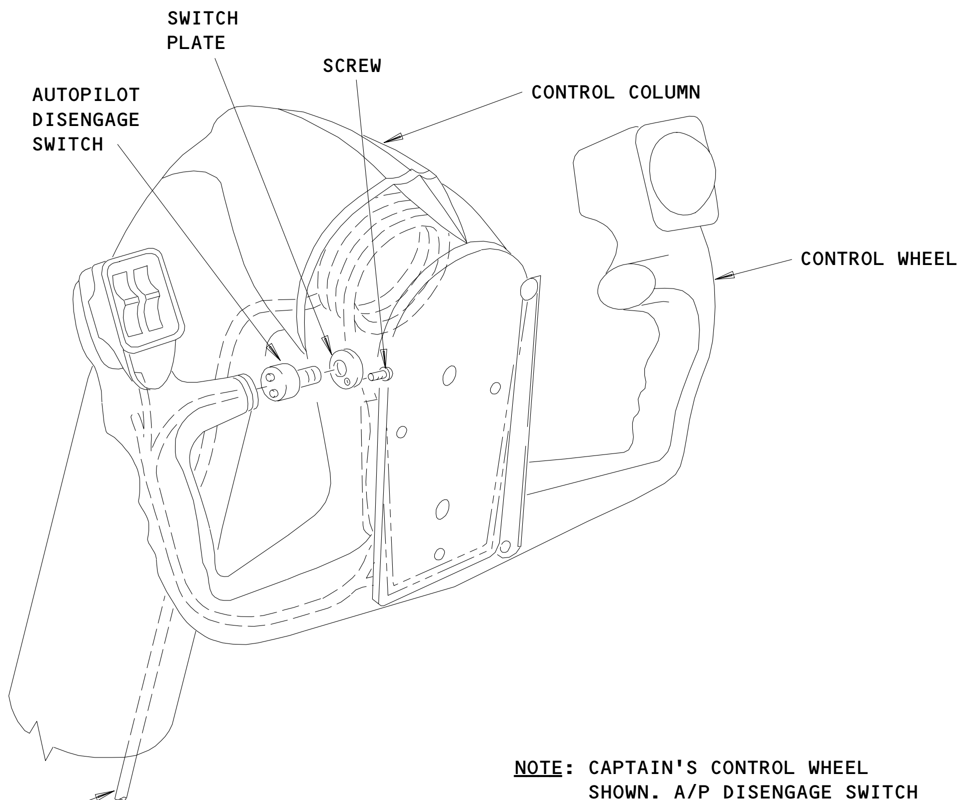
      (4) Move the rate gyro switch to the left and observe control column move forward, then returns to center position.
   C. Restore Airplane to Normal Configuration
      (1) Disengage autopilot engage switch.
      (2) Determine whether there is any further need for electrical power on the airplane; if not, remove external power.

AUTOPILOT DISENGAGE SWITCH - REMOVAL/INSTALLATION

1. General (Fig. 401)
   A. Two autopilot disengage switches, one on each outboard horn, are located on the pilots' control wheels. Each switch is a single-action, multipole, pushbutton type switch which is held in place with a switch plate by a single screw.
2. Remove Switch
   A. Open all autopilot circuit breakers on P6 and P18 panel.
   B. Remove screw and switch plate from switch and carefully pull switch assembly from control wheel.
   C. Note and mark arrangement of color coded wires and disconnect wires from terminals of switch. See Fig. 402 for wiring configuration.
   D. Remove switch.
3. Install Switch
   A. Ensure that all autopilot circuit breakers on P6 and P18 panel are opened.
   B. Locate and connect proper color coded wires to switch (Fig. 402).

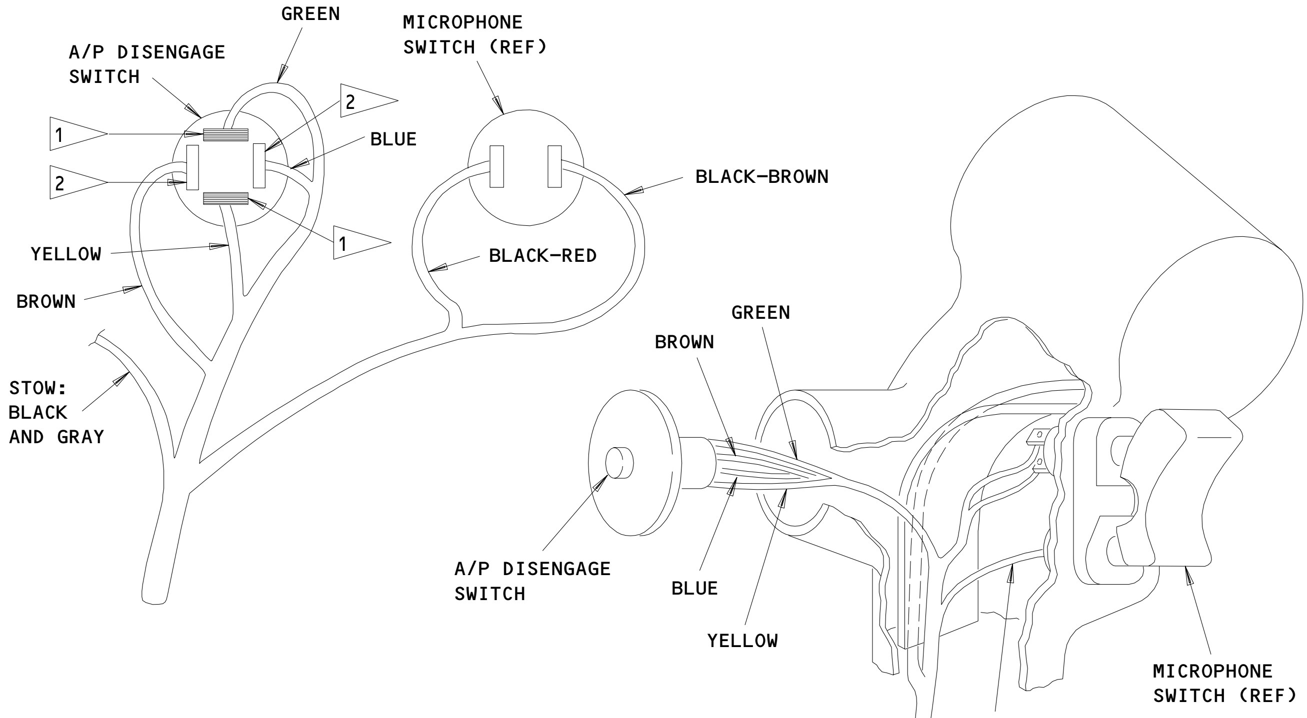
      NOTE: Pull wires away from switch holes and install teflon sleeves before connecting. Tape and stow unused wires.

   C. Position switch in control wheel and secure switch with switch plate and one screw.
   D. Close all autopilot circuit breakers on P6 and P18 panel.
   E. Test autopilot disengage switch (Ref 22-11-411 A/T).

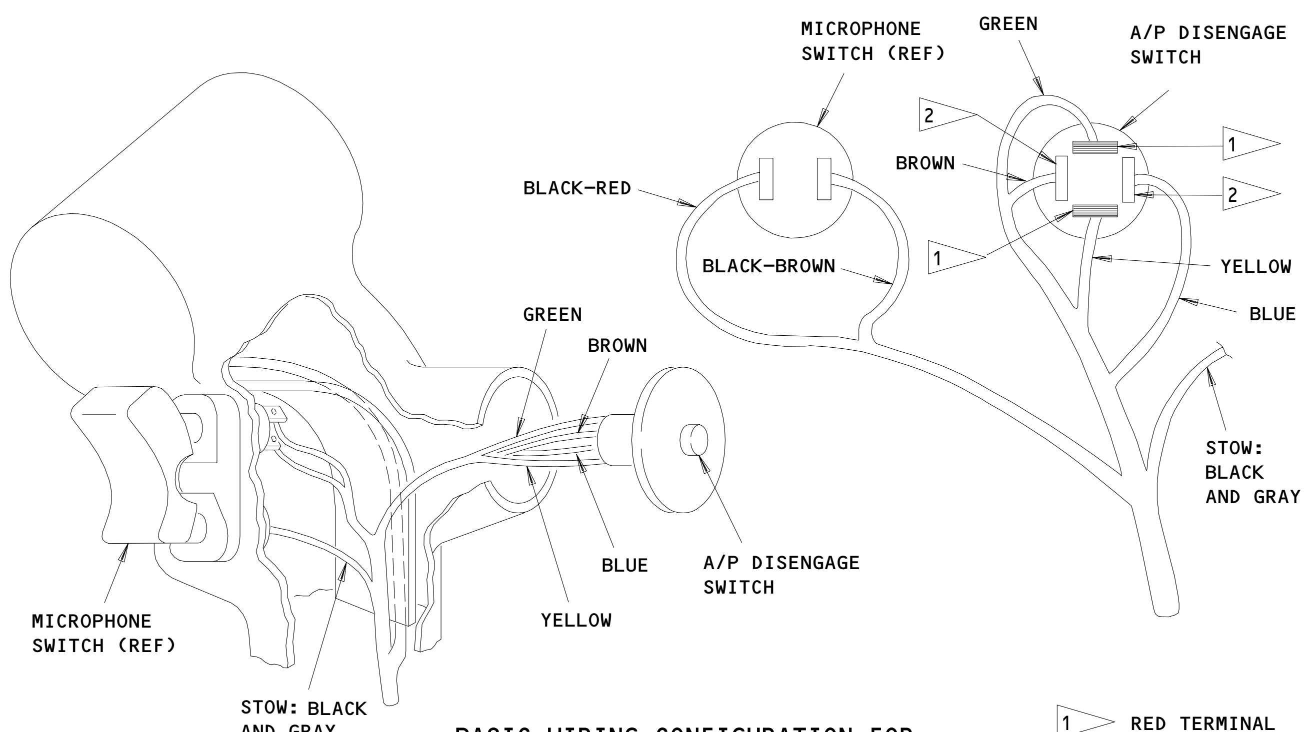


NOTE: CAPTAIN'S CONTROL WHEEL SHOWN. A/P DISENGAGE SWITCH ON FIRST OFFICER'S CONTROL WHEEL IS LOCATED ON RIGHT OUTBOARD HANDLE.

Autopilot Disengage Switch Installation
Figure 401



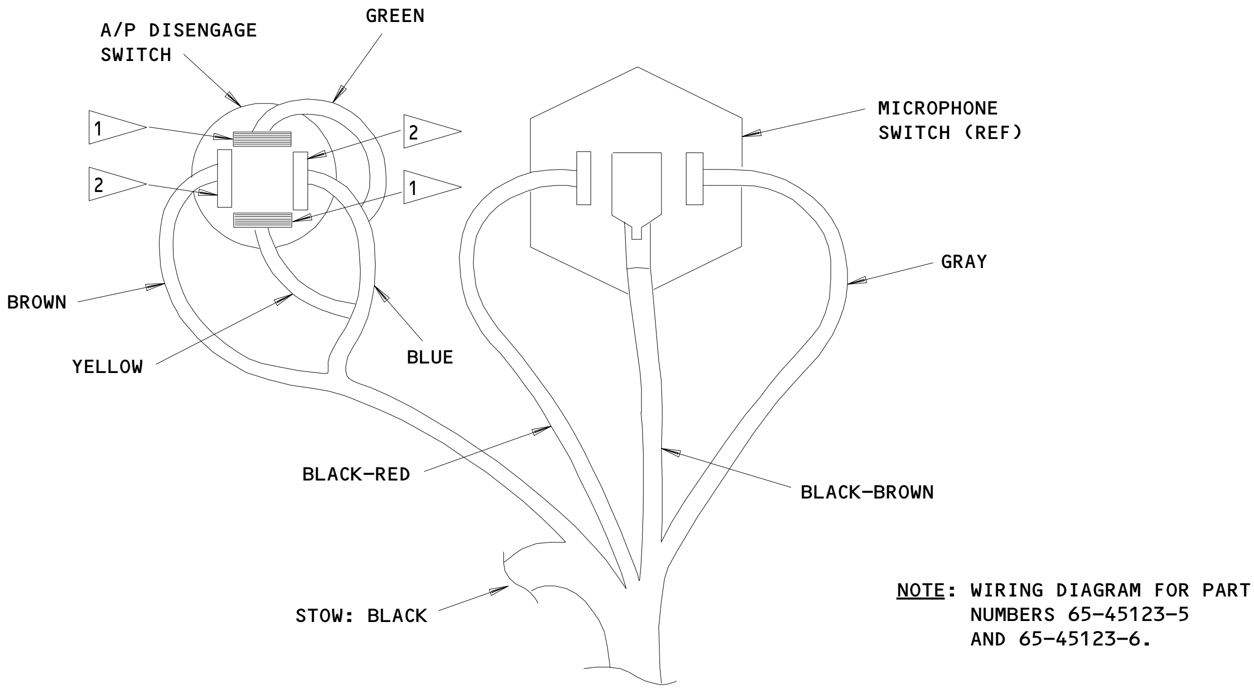
BASIC WIRING CONFIGURATION FOR
LEFT-HAND CONTROL COLUMN

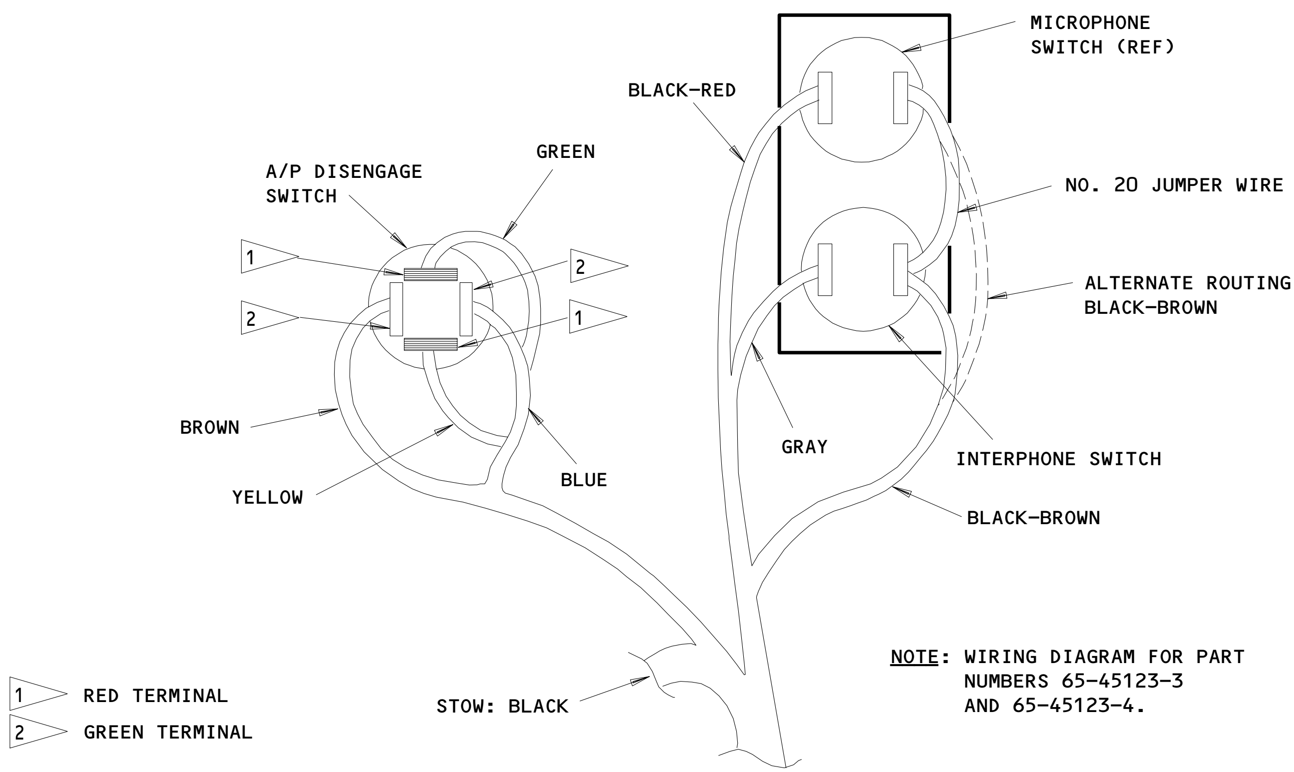


BASIC WIRING CONFIGURATION FOR
RIGHT-HAND CONTROL COLUMN

1 RED TERMINAL
2 GREEN TERMINAL

Autopilot Disengage Switch Wiring Configuration
Figure 402 (Sheet 1)





ALTERNATE SWITCH WIRING FOR LEFT AND RIGHT-HAND CONTROL
COLUMN (SEE NOTES) LEFT-HAND SHOWN, RIGHT-HAND OPPOSITE

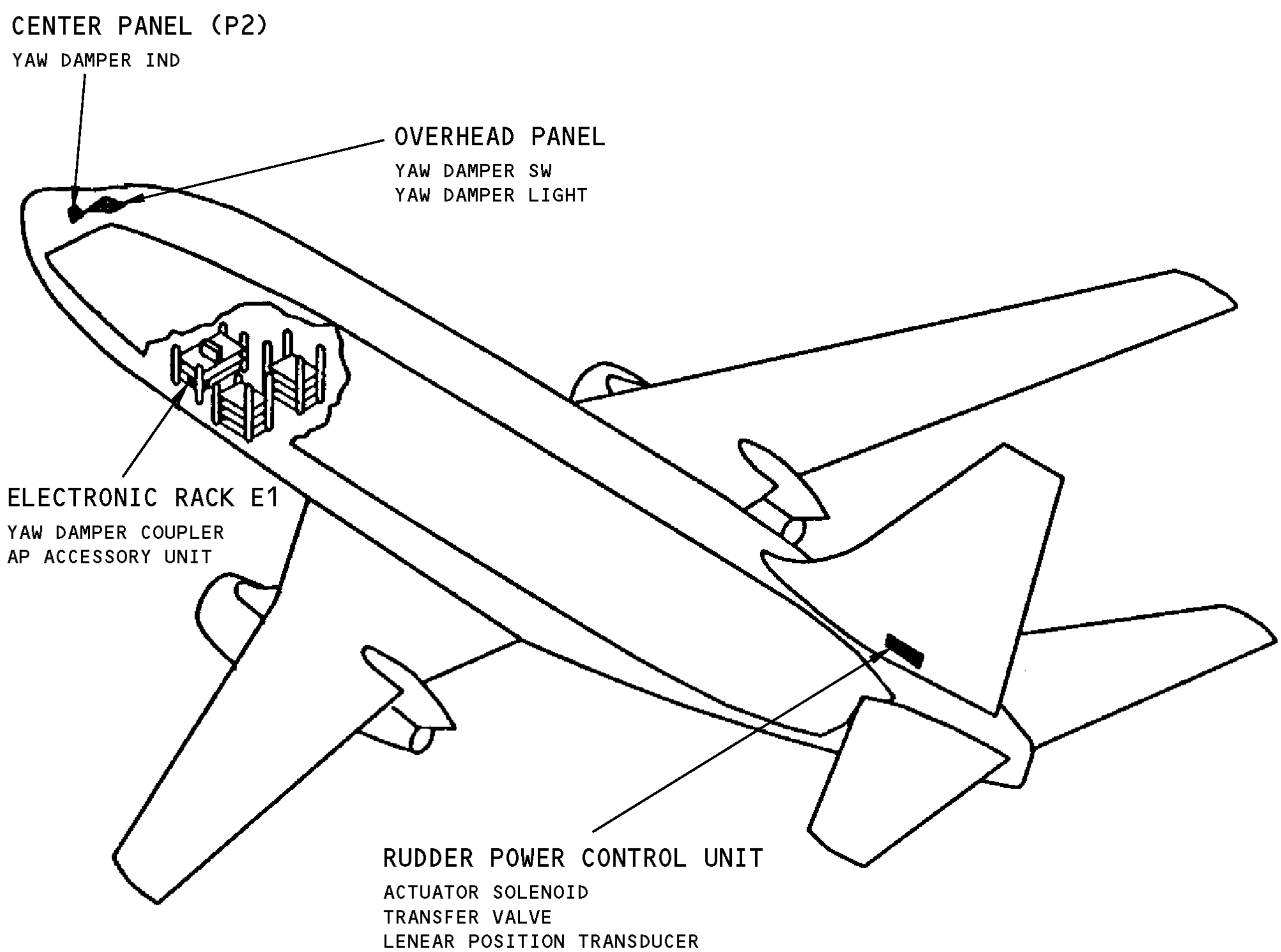
Autopilot Disengage Switch Wiring Configuration
Figure 402 (Sheet 2)

AUTOPILOT DISENGAGE SWITCH - ADJUSTMENT/TEST

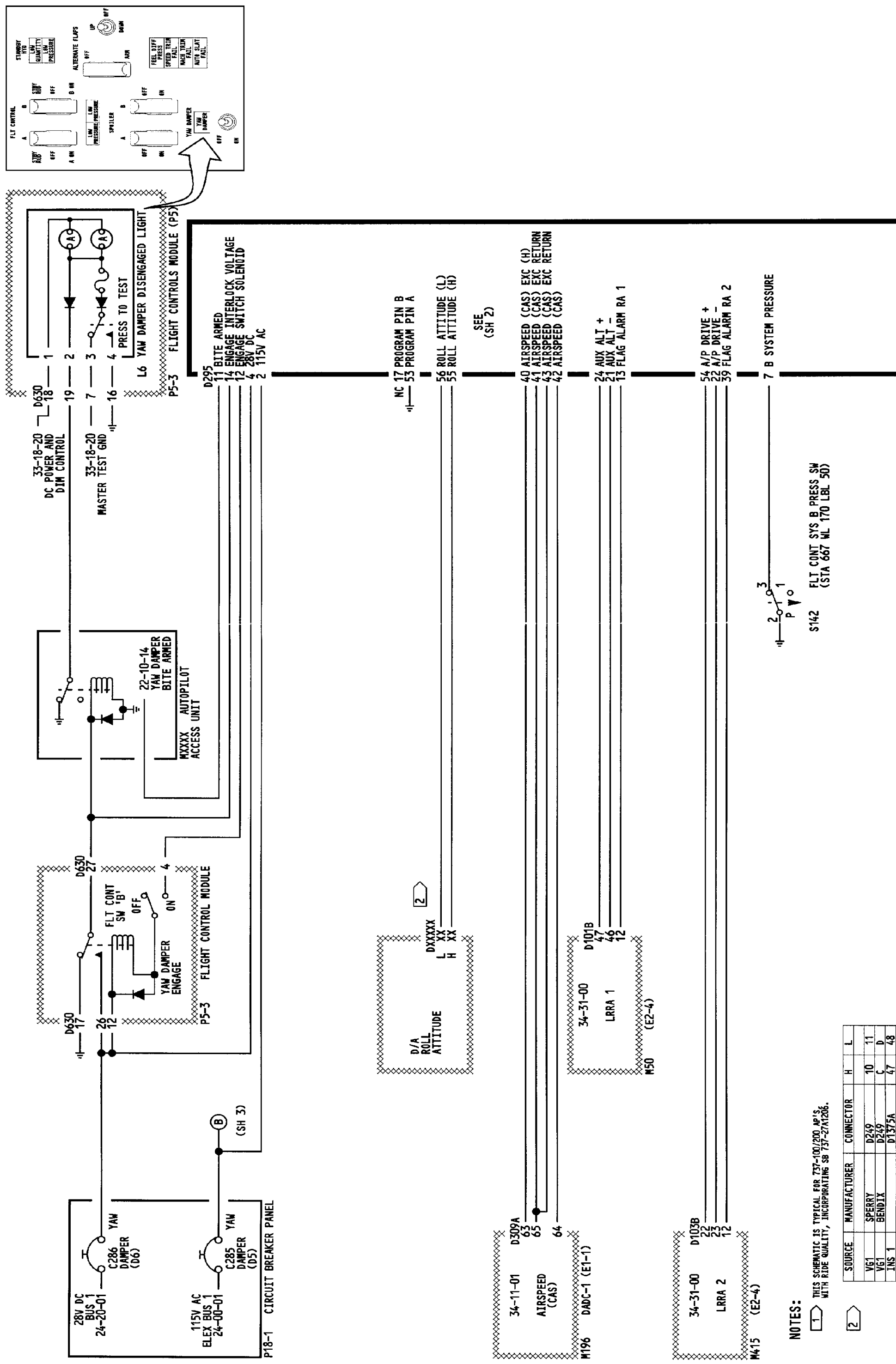
1. General
    A. The autopilot disengage switches, one on each outboard horn, are located on the pilots' control wheel.
    B. The switch must be tested whenever it is replaced.
2. Test Autopilot Disengage Switch
    A. Provide electrical and hydraulic power.
    B. Provide attitude valid and compass valid signals.
    C. Ensure that all autopilot circuit breakers P6 and P18 panels are closed.
    D. Engage autopilot. Operate autopilot disengage switch. Check that autopilot disengages and A/P warning lights flash. Reset A/P warning lights.
    E. Remove electrical and hydraulic power if no longer required.

YAW DAMPER SYSTEM - DESCRIPTION AND OPERATION

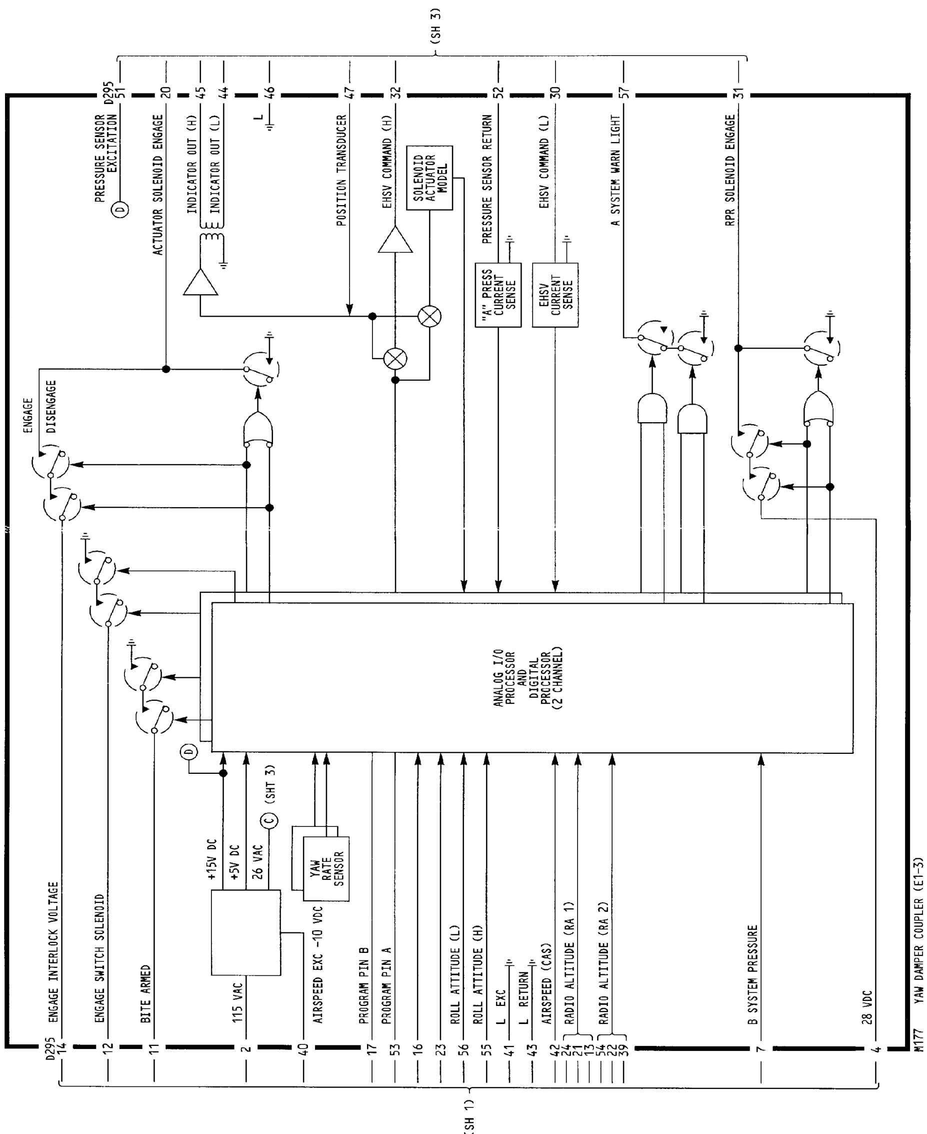
1. General
    A. Yaw axis stability augmentation is provided by the yaw damper system. The yaw damper system stabilizes the airplanes yaw motions during manual and autopilot controlled flight by providing rudder displacement proportional to and opposing the yaw rate of the airplane. Since a series hydraulic actuator tie-in is used, there is no damper control feed back to the rudder pedals. This allows the pilot to maneuver the airplane in a normal manner without having an opposing force from the yaw damper system.
    B. The yaw damper system consists of a solenoid held yaw damper engage switch, yaw damper disengage light, Flight Control B switch, yaw damper coupler (YDC), yaw damper actuator with transfer and solenoid valves and position transducer located in the rudder power control unit, yaw damper actuator position indicator, autopilot warning light, and the YDC BITE control panel.
    C. Yaw damper electrical power is supplied through two circuit breakers located on the main load control center P18. One circuit breaker supplies 115 volts ac and the other 28 volts dc.
    D. The B hydraulic system powers the yaw damper actuator portion of the rudder power control unit (Ref Chapter 27).
    E. The YDC performs monitoring of the yaw damper system and provides passive response or automatic disengagement of the yaw damper system upon detection of system failures or malfunctions.
2. Yaw Damper Coupler
    A. The YDC is installed on the electronic equipment racks in the electrical and electronic compartment. See figure 1.
    B. The YDC is a dual processor architecture utilizing an active channel (Lane 1) and monitor channel (Lane 2) to drive and monitor the yaw damper actuator and other outputs from the YDC. The dual-lane architecture provides redundant signals for monitoring the yaw damper and rudder pressure reducer systems.
    C. The YDC provides control and monitoring of the yaw damper system. The YDC senses airplane motion in the yaw axis using dual solid state quartz rate sensors located inside of the YDC. The YDC provides control signals to the yaw damper actuator solenoid valve and transfer valve, provides 26 VAC 400 Hz excitation to the actuator position transducer, and receives actuator position signals from the yaw damper actuator position transducer. The YDC automatically disengages the yaw damper engage switch when its monitors detect a malfunction in the yaw damper system. The YDC disables the yaw damper actuator in response to manual or automatic disengagement of the yaw damper engage switch. The YDC illuminates the autopilot disengage warning lamps when the YDC is performing ground maintenance functions.



Yaw Damper Component Locations
Figure 1



Yaw Damper System Functional Schematic Diagram
Figure 2 (Sheet 1)

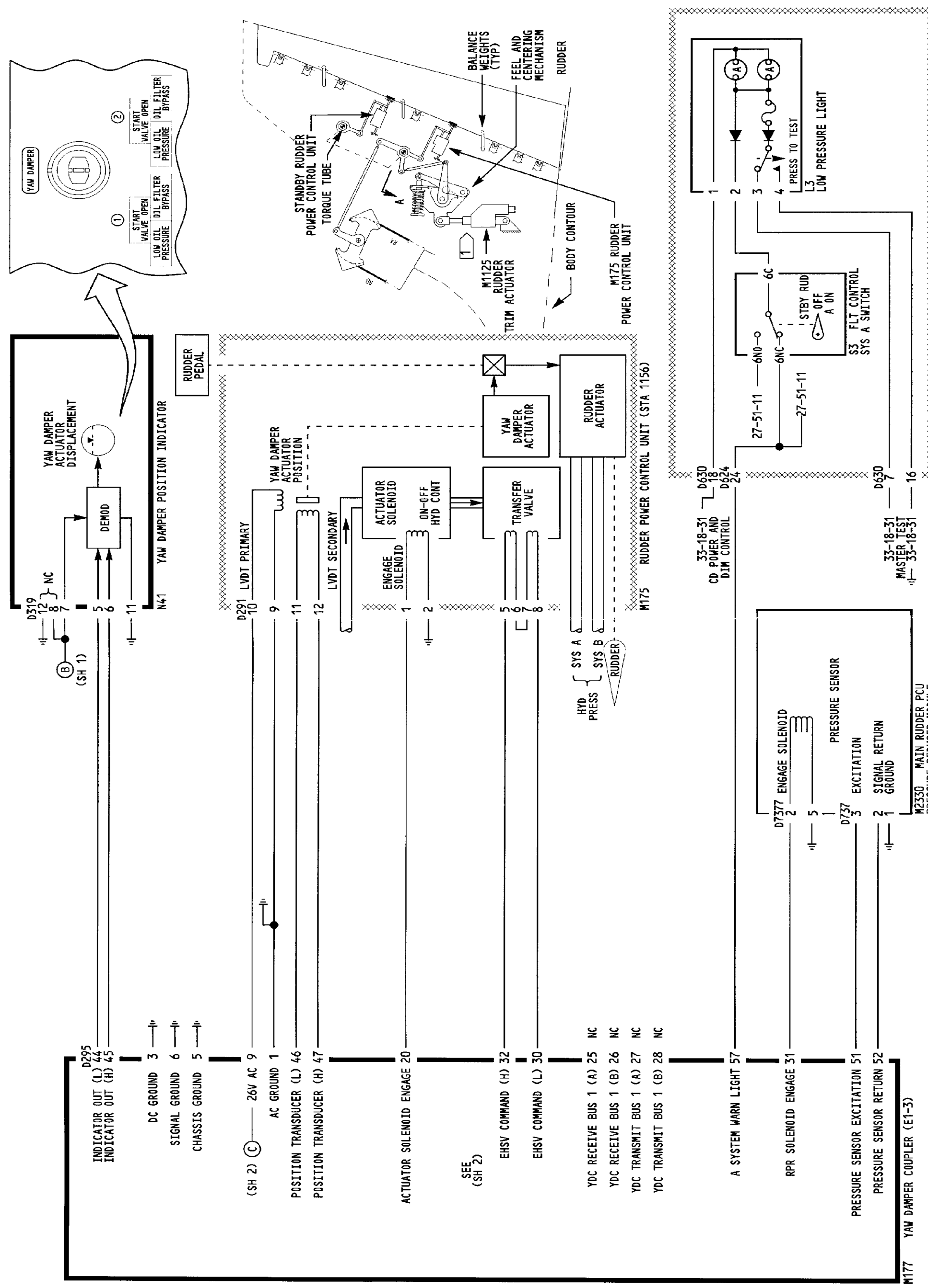


NOTES:

1. THIS SCHEMATIC IS TYPICAL FOR 737-100/200 AP'S, WITH RIDE QUALITY, INCORPORATING SB 737-27A1206.

Yaw Damper System Functional Schematic Diagram
Figure 2 (Sheet 2)

451889

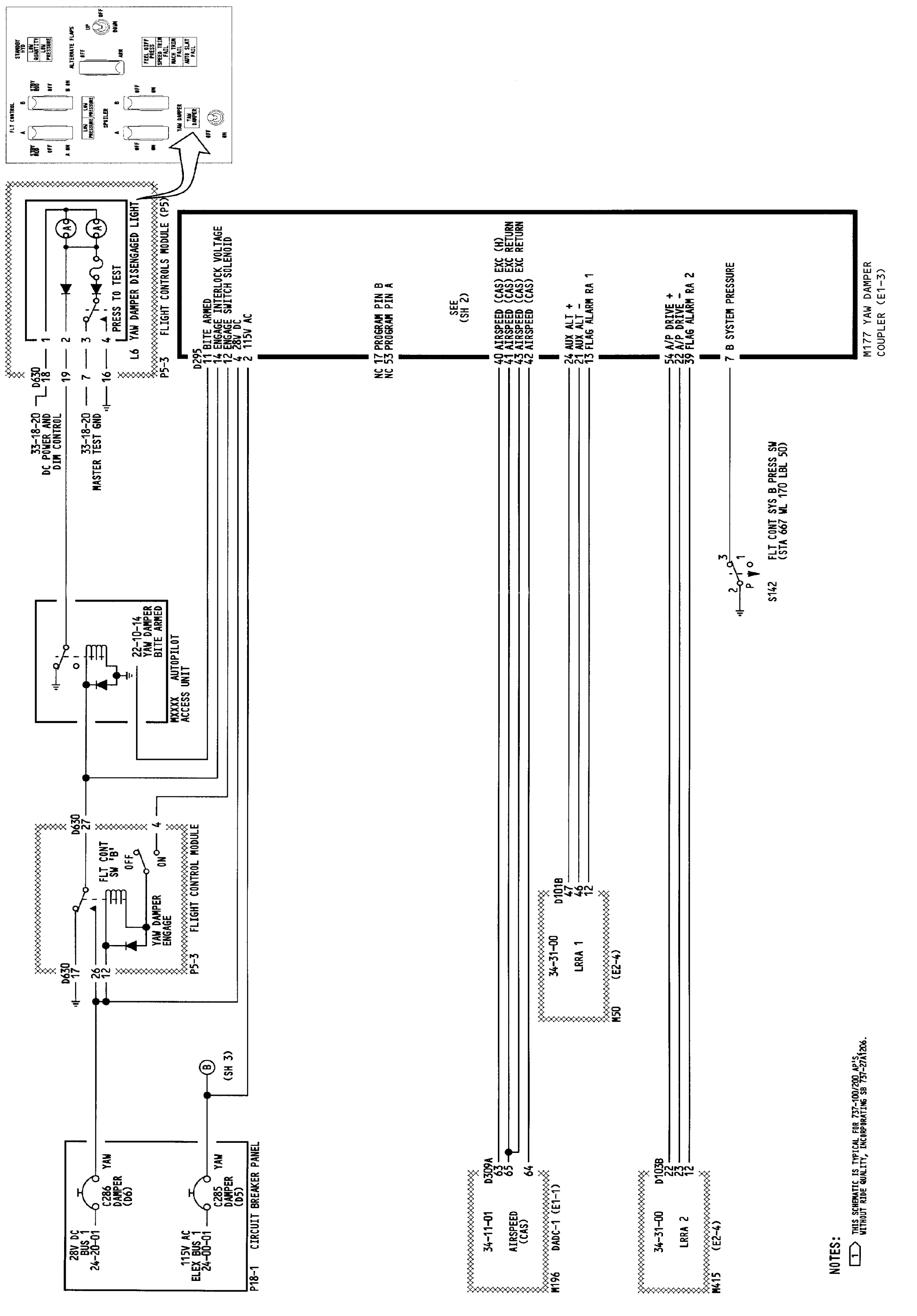


NOTES:

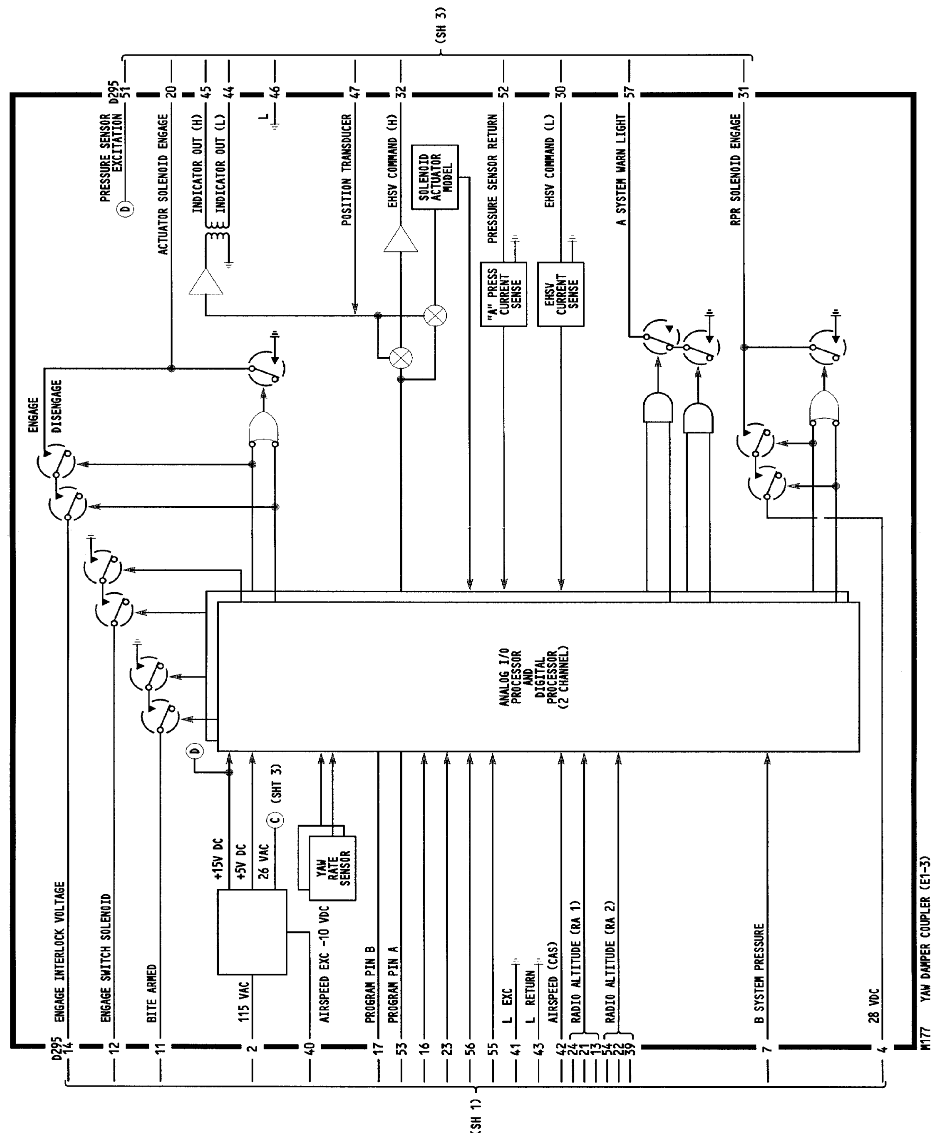
1 THIS SCHEMATIC IS TYPICAL FOR 737-100/200 AP'S, WITH OR WITHOUT RIDE QUALITY, INCORPORATING SB 737-27A1206.

Yaw Damper System Functional Schematic Diagram
Figure 2 (Sheet 3)

451890



Yaw Damper System Functional Schematic Diagram
Figure 2 (Sheet 4)

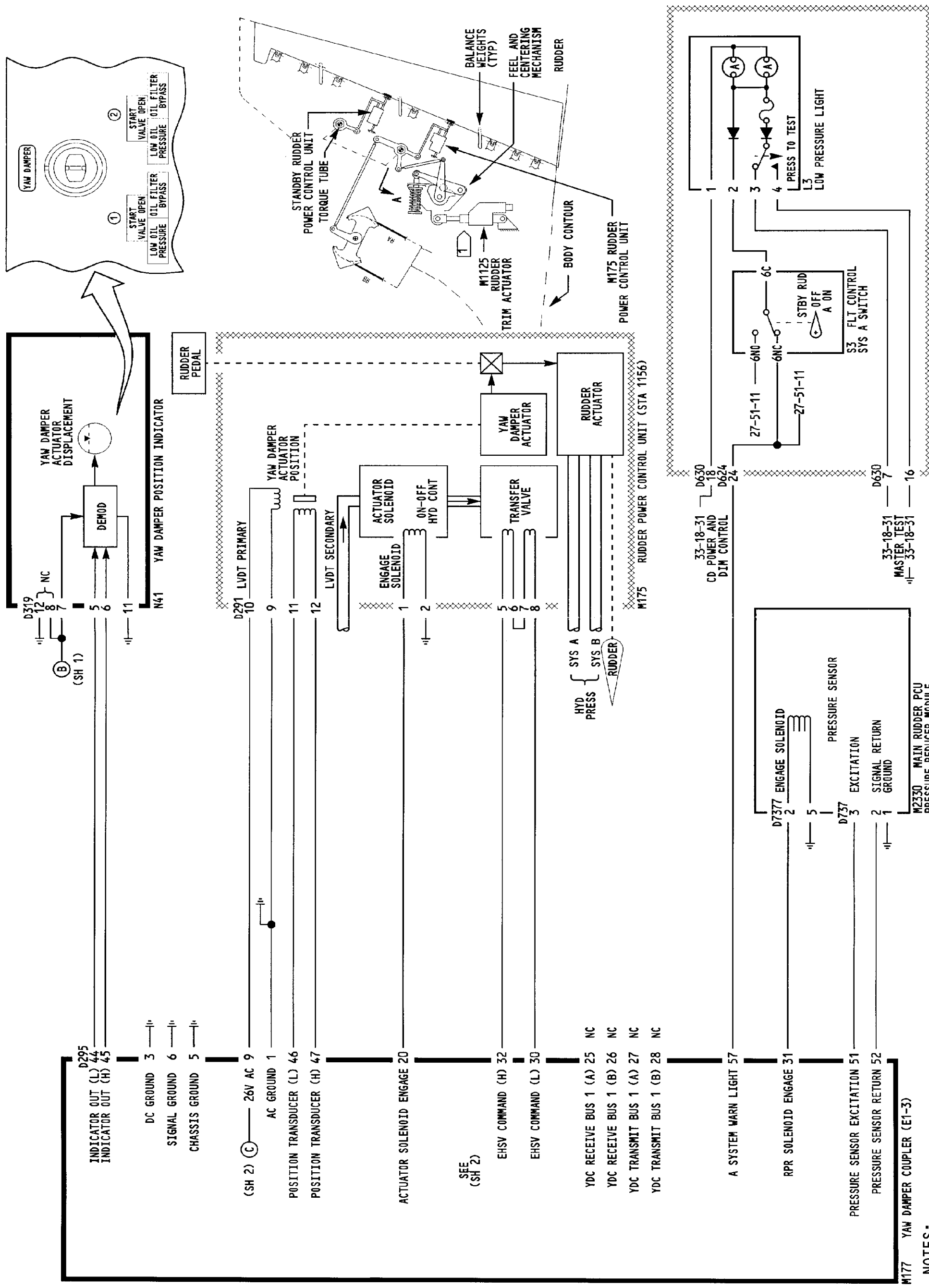


NOTES:

1 THIS SCHEMATIC IS TYPICAL FOR 737-100/200 AP'S, WITHOUT RIDE QUALITY, INCORPORATING SB 737-27A1206.

Yaw Damper System Functional Schematic Diagram
Figure 2 (Sheet 5)

451892



NOTES:

1 THIS SCHEMATIC IS TYPICAL FOR 737-100/200 AP'S, WITH OR WITHOUT RIDE QUALITY, INCORPORATING SB 737-27A1206.

Yaw Damper Systems Functional Schematic Diagram
Figure 2 (Sheet 6)

451893

D. The YDC provides control and monitoring of the main rudder PCU pressure reducer (RPR) system. The YDC provides control signals to the RPR solenoid, provides pressure sensor excitation, and receives signals from the pressure sensor to measure A system hydraulic pressure to the main rudder PCU. The YDC interfaces to Radio Altimeter #1 and Radio Altimeter #2, if installed, to determine when the airplane transitions above 1,000 feet and below 700 feet to activate and deactivate the RPR solenoid. The YDC interfaces to the Flight Control System B pressure switch and deactivates the rudder pressure reducer solenoid when B hydraulic system is depressurized. The YDC interfaces to the Flight Control System A Low Pressure Light and provides illumination when A system pressure to the main rudder PCU is below 500 PSI or when full pressure is commanded to the RPR and system pressure is below 2,200 PSI. There is a 5 second delay from when the YDC detects a change in system pressure to when the YDC turns on or off the low pressure light.

3. Component Description

A. Yaw Damper Engage Switch, Solenoid, and Disengage Light

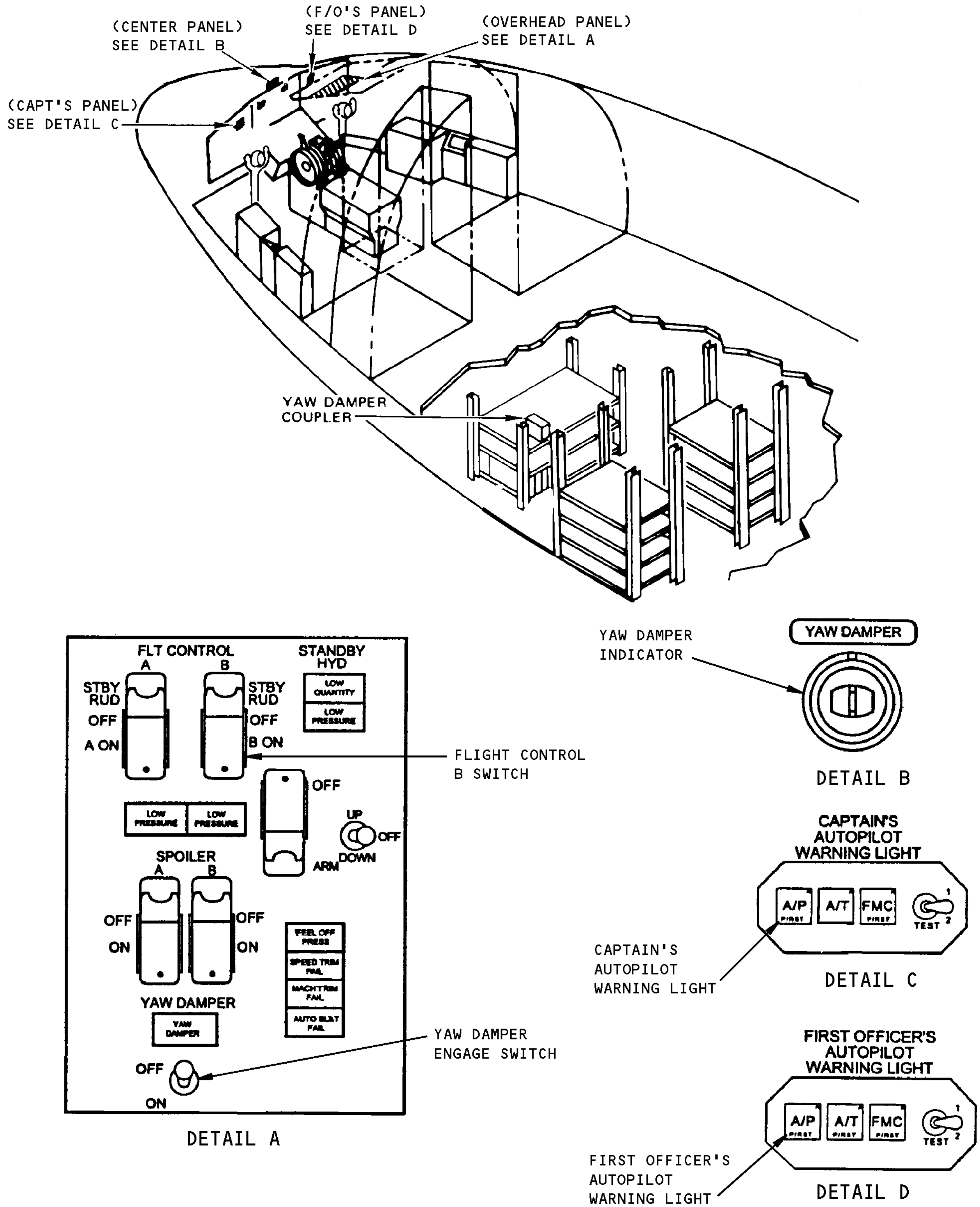
(1) The Yaw Damper Engage switch is located on overhead panel P5. It is a solenoid held switch. When the engage switch is placed to the ON position, and if the yaw damper engage interlocks are satisfied, the solenoid holds the switch in the engaged position and the YAW DAMPER light extinguishes. The yaw damper is then in the yaw damping mode.

(2) Whenever the yaw damper is disengaged, the amber light labeled YAW DAMPER, on overhead panel P5, illuminates steadily. The MASTER CAUTION lights and the FLT CONT master caution annunciator light, located on the light shield, also illuminate.

(3) Failure of the YDC or selected interfacing systems causes the yaw damper to automatically disengage.

B. Flight Control B Switch

(1) The flight control B switch is located on the pilot's overhead panel (see figure 2). It is used to active the B system hydraulics. For airplanes LN 213 and beyond and airplanes with SB 737-22-1009, the Flight Control B switch must be in ON position for the yaw damper to remain engaged. The yaw damper actuator will not engage until the yaw damper engage switch is placed to the ON position and B system hydraulics is pressurized. The YDC is designed to remain engaged when the B system hydraulics is depressurized, although the yaw damper actuator will be disabled.



Yaw Damper System Flight Deck Interfaces
Figure 3

C. Yaw Damper Indicator

(1) The yaw damper indicator, located on the center instrument panel, operates when 115 volts ac is applied to the YDC. The yaw damper indicator displays movements of the yaw damper actuator. A linear position transducer mounted on the rudder power control unit supplies a yaw damper actuator position feedback signal to the yaw damper coupler. The yaw damper coupler provides a 400 Hz signal to the indicator in proportion to the signal received from the actuator position transducer. The transducer is a variable reluctance transformer which is excited by 26 volts 400 Hz from the YDC. The magnitude of the sensors output signal varies directly with the length of the input stroke and the phase of the signal changes with the direction of the stroke from the center null position.

D. Rudder Transfer and Solenoid Valves

(1) The transfer and solenoid valves control hydraulic flow for yaw damper control of the rudder power control unit.

(2) The yaw damper solenoid valve is mounted on the rudder power control unit. It is a solenoid actuated valve and is energized by the YDC when the yaw damper is engaged and B system hydraulics is pressurized. When the valve is energized, hydraulic power is supplied to the transfer valve.

(3) The transfer valve, also mounted on the rudder power control unit, converts the yaw damper electrical signals into hydraulic flow to move the rudder. The transfer value consists of a torque motor which moves a jet pipe assembly to regulate the hydraulic flow to the control valve.

E. Autopilot Warning Light

(1) The autopilot warning light illuminates when the YDC ground maintenance mode is engaged. This annunciation provides an indication that the airplane should not be dispatched. The autopilot warning lights are illuminated by the YDC providing a ground to the autopilot accessory unit. The autopilot accessory unit provides diode isolation between the Captains and First Officers Autopilot warning lights to the YDC ground.

F. Ground Maintenance BITE (Built in Testing Equipment)

(1) The BITE provides an integrated ground maintenance capability for the YDC. The BITE minimizes on-airplane maintenance time, reduces unconfirmed removal rates of Line Replaceable Units (LRUs), and facilitates identification of failed major components and associated interfaces.

(2) The BITE controls are on the front panel of the yaw damper coupler (Fig. 4). At the top of the panel is the output display. The output display actually consists of two individual displays, one upper and one lower. Each display contains eight 7-segment characters. The display provides menu options and test results. In the center of the panel are two rows of selector buttons. In the top row, there is the MENU, up arrow and the ON/OFF selector buttons. In the bottom row, there is the YES, down arrow and NO buttons. The up and down arrow buttons are used to scroll through menu options. The YES and NO buttons are used to respond to menu inquiries and pass test verifications. Pressing the MENU button displays the main menu. Pressing the ON/OFF button turns the BITE mode on and off.

(3) Upon entering BITE, the autopilot warning lights illuminate to provide indication that the YDC is in ground maintenance and that the airplane should not be dispatched.

4. Yaw Damper Operation

A. General

(1) The yaw damper augments the airplane's yaw axis stability by shaping, amplifying, and coupling quartz rate sensor and yaw damper actuator position signals to control the rudder through the hydraulic actuator.

B. Operating Modes

(1) Initialization

(a) Prior to yaw axis engagement, the yaw damper coupler command is zero. Upon engagement, the actuator command will easy-on over a 5 second period providing a transient free transition.

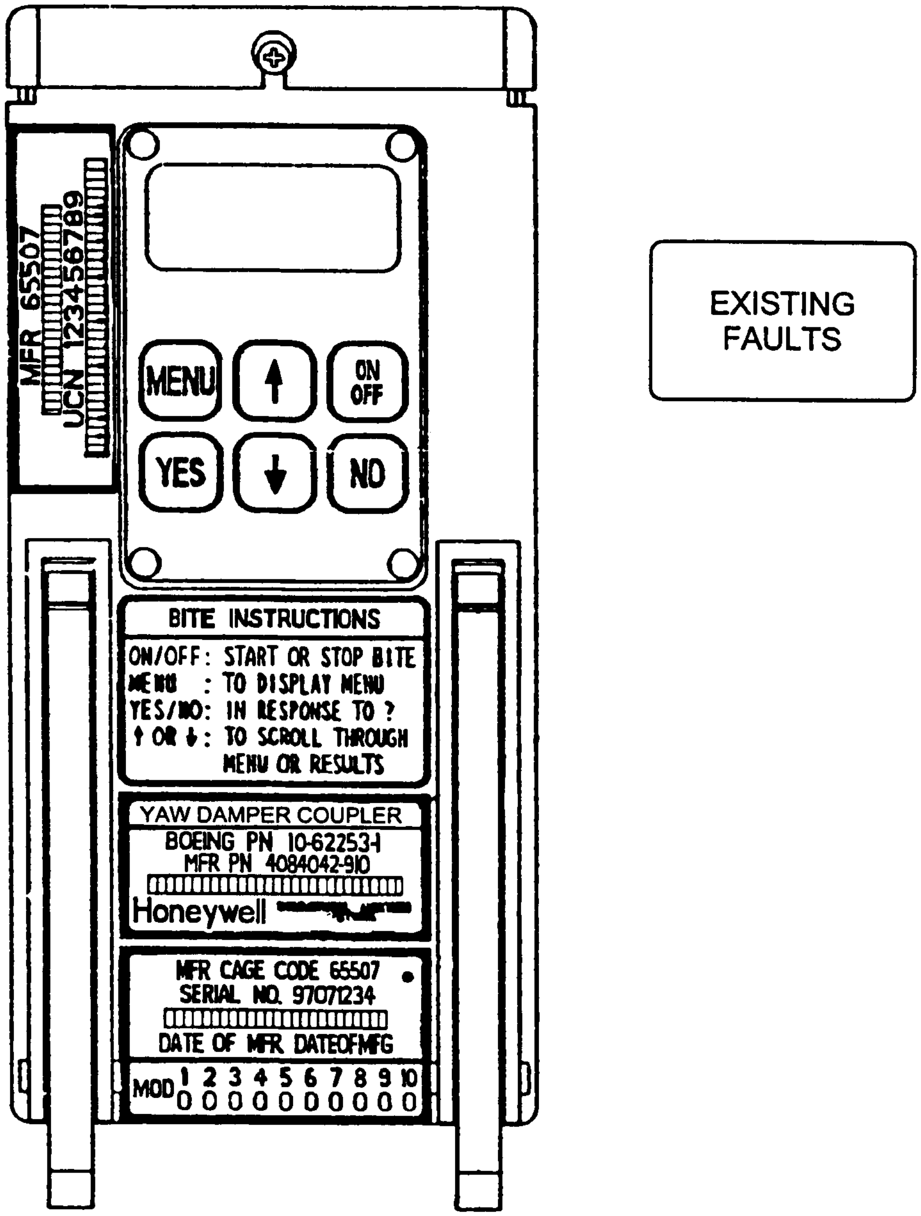
(2) Engage Interlocks and Logic

(a) Prior to engaging the yaw damper system, the yaw damper disengage warning light is illuminated. For the yaw damper engage switch to remain engaged, the 115 volt ac and 28 volt dc must be applied to the YDC, the flight control B switch must be on, and YDC monitors must not be tripped.

(b) The yaw damper engage switch remains in the ON position after holding the switch to ON for at least 0.1 second.

(c) The yaw damper engage switch hold-in solenoid is energized by a ground through the YDC. A set of relay contacts in the autopilot accessory unit open to extinguish the yaw damper warning light when 28 volts is applied to the relay from the yaw damper engage switch contacts.

(d) A 2-second delay timer in the YDC allows the yaw damper engage switch solenoid to remain energized during transfers between 115 volt ac generators.



Yaw Damper Coupler BITE Control Panel
Figure 4

(3) Engaged Mode
(a) When the yaw damper coupler is engaged, its output is applied to the series yaw damper actuator transfer valve to command rudder movement. Movement of the yaw damper series actuator results in rudder movement and an output from the position transducer.

C. Yaw Damper Test
(1) Following application of 115 VAC power to the YDC, after it was off for more than 2.5 seconds, a series of self tests and system tests are performed to verify that the YDC hardware, software, and disconnect mechanisms are operating. The power up self tests are completed within 12 seconds and verify correct operation of the YDC hardware, software, and configuration of the airplane program pins. Following the YDC self test, when the yaw damper engage switch is placed to the on position, the YDC performs a test of the engage switch solenoid to verify that it can be disconnected electrically. Following the engage solenoid test, and upon application of B system hydraulic pressure, the YDC performs a test of the yaw damper actuator to verify that the actuator will not move with the actuator solenoid de-energized. After these tests are complete, the actuator solenoid is energized and the yaw damper system is fully operational. If the power up self test or the engage switch tests fail, the yaw damper engage switch will not hold in the ON position. Failure of the actuator solenoid test will cause the yaw damper to disconnect when B system hydraulics is pressurized. If any of these tests fail, it is necessary to correct the system malfunction and to perform YDC BITE to reset the YDC monitors before the yaw damper system can be engaged.
(2) Application of 115VAC power to the YDC after it was off for less than 2.5 seconds, causes the YDC to verify the correct configuration of the airplane program pins. After this test is complete, the actuator solenoid is energized and the yaw damper system is fully operational.

5. <u>Yaw Damper BITE Operation</u>
A. Yaw damper system ground maintenance BITE is accessed through the YDC BITE control panel. The BITE display consists of two eight character lines of data (Fig. 4). The BITE mode is engaged by pressing the ON/OFF button followed by the YES button within 10 seconds. BITE entry will not be allowed if the YES push button is not selected with 10 seconds after ON/OFF push button is selected.
B. Entry into BITE is enabled when the autopilot warning lights are illuminated. BITE entry will be prevented if the current to the ground path from the 28 VDC autopilot warning light is less than 20 mA.
C. Once engaged, BITE can be disengaged at any time by pressing the ON/OFF button followed by the YES button. BITE will disengage automatically if no keys are pressed within a 5 minute interval.

D. Upon entry into BITE, the first option in the menu is displayed (Existing Faults). Four different modes of operation can be selected; Existing Faults, Fault History, Ground Tests, and Other Functions.
   (1) When the EXISTING FAULTS option is selected, the YDC will display a list of all existing fault messages on the BITE Control Panel. As the operator scrolls through the list, the option of seeing more details is offered for each fault. At the end of the list is the option to reset all latches.
   (2) If the operator selects more details for any particular fault, a further list of details shows the message number, flight deck effect (if any), the likely LRU or LRUs at fault, and whether the fault is latched or not. A further option is then available to view the fault number or numbers that gave rise to the fault message. Note that there may be more than one fault causing any particular fault message, but in this case only the single fault message is displayed.
   (3) If the operator chooses to reset latches, all YDC monitor latches are reset. After a short delay the software assesses whether any faults are still present or have reoccurred. If there are any faults then the list of fault messages is displayed.

E. Fault History
   (1) When the FAULT HISTORY option is selected the YDC scans the records of continuous monitor and power-up test faults in the YDC non-volatile memory (NVM). A list is displayed of those power cycles during which one or more faults were recorded. A new power cycle occurs whenever 115 VAC power is removed for more than 2.5 seconds. The power cycles are numbered in descending order relative to the current cycle. The operator has the option of displaying the faults for the currently displayed power cycle. If selected, the faults are displayed in a similar manner to the Existing Faults option. The only differences are firstly that in place of the latched indication the operator is told whether the fault was a hard or intermittent fault, and if intermittent, how many times it repeated during that power cycle. Secondly, there is no option to reset latches (fault history can only be cleared in the shop)
   (2) YDC fault history has storage capacity for 256 fault messages. A maximum of 64 faults can be stored during a single power cycle. Fault history will be stored for approximately 32,000 power cycles, provided the total number of fault messages does not exceed 256.

F. Ground Test
   (1) Self Test
      (a) The purpose of the self test is to run a test of the YDC hardware, primarily to detect latent faults and to confirm faults found by continuous monitor and power-up tests.

(b) Since there is the possibility of rudder movement in the presence of a failure, a warning is given. The yaw damper has to be engaged for this test to run, even though it is not necessary to have hydraulic power supplied.

(c) If the test fails, the YDC displays a list of the faults that caused the failure.

(d) If any existing faults are present that would prevent the test from completing, they are displayed as inhibit conditions before starting the test. The operator has the option of removing the YDC at that point or trying to fix the problem if it is external to the YDC.

(2) Servo Test

(a) The purpose of the servo test is to verify correct operation of the yaw damper actuator, verify that the actuator LVDT is rigged within acceptable limits, and to determine the actuator travel limits for programming of the YDC servo loop monitor.

(b) First, a test is run to ensure that the yaw damper actuator will not move when the actuator solenoid is de-energized. The yaw damper actuator is then driven to its maximum travel in both directions. During this sweep the actuator servo command response monitor is set to the lowest possible authority to avoid trips if the actuator is lower in authority compared with the YDC previously stored actuator authority. At this point, the following can cause test failure and exit: command response monitor trip, other internal YDC monitor trips, actuator limits or rigging not correct, or actuator authority of 2 degrees/0.15 inches in an airplane configuration where ride quality control laws will command up to 3 degrees/0.225 inches. If none of these are applicable then another sweep is performed with the correct authority limits applied, thus testing the command response monitoring over its entire range.

(c) If the test fails at any point the YDC displays a list of the faults that caused the failure. These are displayed in a similar manner to the Existing Faults option. The only difference is that there is no option to reset latches.

(d) If any existing faults are present that would prevent the test from completing, they are displayed as inhibit conditions before starting the test. The operator has the option of removing the YDC at that point or trying to fix the problem if it is external to the YDC.

(3) Pressure Reducer Test

(a) The pressure reducer test is a functional test of the rudder pressure reducer system, including the pressure reducer solenoid, the pressure sensor and the A System Low Pressure warning lamp. The pressure reducer solenoid is taken through a no-reduction, reduction and no-reduction cycle, and the resultant pressure is checked at each stage. The Low Pressure warning is taken through an on-off cycle. Tests of internal YDC circuitry and monitors are also performed.

(b) If the test fails at any point, the YDC displays a list of the faults that caused the failure. These are displayed in a similar manner to the Existing Faults option. The only difference is that there is no option to reset latches.

(c) If any existing faults are present that would prevent the test from completing, they are displayed as inhibit conditions before starting the test. The operator has the option of removing the YDC at that point or trying to fix the problem if it is external to the YDC.

(4) Program Pin Test

(a) This test confirms that the two YDC program pins (D295, pins 17 and 53) are correct. The settings are stored to NVM at the end of the test. If the YDC is moved to a different type airplane, or the program pins fail to a different setting, then this test has to be run before that yaw damper will engage, since the YDC detects a difference between the program pins and the values stored in NVM.

(b) If the operator does not confirm the apparent settings of the program pins, the test fails and the YDC displays the fault.

(c) If any existing faults are present that would prevent the test from completing, they are displayed as inhibit conditions before starting the test. The operator has the option of removing the YDC at that point or trying to fix the problem if it is external to the YDC.

(5) Display Test

(a) This test illuminates all segments of the BITE display in a sequence. The operator determines whether the test passes or fails. The YDC does not determine or display any faults.

G. Other Functions

(1) System Configuration

(a) When the System Configuration option is selected, the YDC displays a list of configuration parameters. These are software version number, airplane type as determined by the program pins and program pin test, yaw damper actuator limit as determined by the servo test, elapsed time indication in hours, as stored in NVM, and the number of power cycles that the YDC has experienced.

(2) I/O Monitor
   (a) When the I/O Monitor option is selected the YDC displays a list of options, Analog Inputs/Outputs or Discrete Inputs/Outputs.
      1) Analog Inputs and Outputs
         a) When the Analog Inputs/Outputs option is selected, the YDC displays a list of analog type YDC inputs and outputs in engineering units. The values are updated at a rate of 1 Hz. If a particular input parameter is currently unknown due to a detected input fault, it is displayed as invalid instead. Only inputs and outputs that are relevant to the current airplane type are displayed.
      2) Discrete Inputs and Outputs
         a) When the Discrete Inputs/Outputs option is selected, the YDC displays a list of discrete type YDC inputs and outputs in engineering units. The values are updated at a rate of 1 Hz. If a particular input parameter is currently unknown due to a detected input fault, it is displayed as invalid instead. Only inputs and outputs that are relevant to the current airplane type are displayed.
(3) Output Control
   (a) When the Output Control option is selected, the YDC displays a list of the output control functions. The control functions are yaw damper servo control, yaw damper engage switch, pressure reducer, and hydraulic lamp control for the A system low pressure warning.
   (b) The purpose of this function is to provide direct control of the YDC outputs to assist in airplane system installation and troubleshooting. This supplements the I/O monitoring provided elsewhere. Note that all control outputs except the autopilot warning lights are selectable. The Ground Maintenance mode (Autopilot Warning Lights) can simply be toggled by going in and out of BITE.
      1) Yaw Damper Servo Control
         a) When the Yaw Damper Servo Control option is selected, the operator can change the yaw damper actuator position using the up-arrow and down-arrow keys in steps of 0.05 deg. If a key is held pressed for more than one second, the command changes by 0.5 deg. The command can be moved over a range exceeding the largest actuator authority. The actuator command is displayed on the upper display segment and the actuator position is displayed on the lower display segment.

2) Yaw Damper Engage Switch Control
   a) When the Yaw Damper Switch Control option is selected, the operator is able to enable or energize the yaw damper engage switch solenoid by pressing the up-arrow and down-arrow keys.
3) Pressure Reducer Control
   a) When the Pressure Reducer Control option is selected, the operator is able to energize the pressure reducer solenoid using the up-arrow and down-arrow keys.
4) Low Hydraulic Pressure Lamp Control
   a) When the Hydraulic Lamp Control option is selected, the operator is able to turn the A system Low Pressure Warning lamp on and off using the up-arrow and down-arrow keys.
   b) Since this lamp can be driven by the A system flight controls low hydraulic Pressure switch, the YDC will not be able to extinguish the lamp if the A-System is not pressurized.

(4) BITE Control Panel Deactivation
   (a) At any time when the BITE control panel is in use, the operator can press the ON-OFF button to exit BITE and ground maintenance state. Confirmation is requested before the exit actually occurs. Any test or other operation that was in progress at the time is stopped before BITE is exited. If the operator does not provide confirmation, the YDC displays the last menu option that was selected and waits for further input.

YAW DAMPER SYSTEM - TROUBLESHOOTING

1. General
   A. The following troubleshooting procedures are intended to help trouble shoot the yaw damper system down to a component level.
   B. Troubleshooting charts are provided based on possible fault codes that can be reported by the BITE and the corresponding isolation procedure to detect and replaced the faulty component. The charts are separated into categories of fault types. Among the details listed are:
      (1) Fault
      (2) Applicability
         (a) Existing faults
         (b) Fault history
         (c) Ground test fault display
      (3) Fault message
      (4) Fault details
         (a) Detail 1 - message number
         (b) Detail 2 - flight deck effect
         (c) Detail 3 - most likely LRU(s) causing the fault
         (d) Detail 4
            1) 4a - fault latched or not
            2) 4b - hard or intermittent fault type
   C. Before troubleshooting the yaw damper system, ensure hydraulic and electrical power is applied to the airplane and the rudder operates normally without the yaw damper system engaged. The flight control system B switch must be on to engage the yaw damper system. Ensure the yaw damper system, master caution, master dimming bus indicator lights, dim and test, autopilot engage interlock and warning light circuit breakers are closed before attempting to engage the yaw damper system. To prevent misleading indications, test operation of the autopilot disengage lights by pressing lenses to ensure bulbs illuminate. It should be noted that the yaw damper engage light is normally illuminated whenever the yaw damper system is not engaged. The air data computer does not have to be energized for self-test of the yaw damper system since no interlock voltages are provided to the yaw damper system. However, the air data computer must be installed since the yaw damper q-potentiometer is used in the system. Voltage through the q-potentiometer is provided by the yaw damper system. The yaw coupler BITE is calibrated with no airspeed inputs to the air data computer. Removal of black boxes and system components requires normal safety and maintenance precautions such removal of power prior to removal of components and capping of all exposed electrical connectors.
2. Yaw Damper BITE
   A. BITE entry
      (1) Press ON/OFF on the YDC BITE panel. Observe the ENTER BITE? is displayed on the panel.

(2) Within 10 seconds press YES. Observe that PLEASE WAIT is temporarily displayed followed by EXISTING FAULTS? on the YDC BITE panel. Observe the AUTOPILOT DISENGAGE WARNING lights illuminate steadily.

NOTE: If no keypad entries are made for 5 minutes, the YDC BITE display will turn off. It will be necessary to start over it this occurs.

B. Existing Faults provides detail information on currently existing faults.

(1) Ensure that the YDC BITE panel displays EXISTING FAULTS? from the main menu. If necessary, press MENU until the main menu appears and scroll using up or down arrows until EXISTING FAULTS? is displayed.

NOTE: Pressing MENU will take the BITE display to the next higher level until the main menu is reached.

(2) Press YES.

(3) If no faults are present, the panel displays NO FAULTS. If one or more faults exist, the panel will display the first fault message.

(a) To view details of the fault, press down arrow. MORE DETAILS? will be displayed on the panel. Press YES.

NOTE: Four lines of data are displayed in the MORE DETAILS? screen.
Line 1 - Message Number
Line 2 - Flight Deck Effect
Line 3 - LRU(s) responsible for the fault
Line 4 - Fault latched or not latched

1) Press the down arrow key to cycle thru the fault detail lines.

2) Record the maintenance message for future reference.

3) After the last fault detail line, SHOP DETAILS? is displayed on the panel. Press NO.

4) Observe that the panel displays END OF LIST. Press MENU to return to the EXISTING FAULTS screen.

a) If the same fault message is displayed, there are no more faults.

b) If a different message is displayed, steps 3(a) thru 3(a)-2 can be repeated to view details of the next fault.

(b) To bypass the details and view the next fault, scroll until MORE DETAILS? is displayed on the panel. Press NO.

(4) To reset latches, ensure the main menu is displayed on the panel.

(a) Scroll until EXISTING FAULTS? is displayed. Press YES.

(b) Cycle thru all (if any) faults until RESET LATCHES is displayed on the panel. Press YES.
(c) Observe that ARE YOU SURE? is displayed on the panel. Press YES.
(d) Observe that RESET IN PROGRESS is displayed on the panel. After 2 seconds, ensure that the panel displays NO FAULTS.
(e) Press MENU to return to the main menu.

C. Fault History provides detail information about faults that occurred prior to the last power cycle. A power cycle is the time between power being applied to the plane and the power being disconnected.
(1) Ensure that the YDC BITE panel displays EXISTING FAULTS? from the main menu. If necessary, press MENU until the main menu appears and scroll using up or down arrows until EXISTING FAULTS? is displayed.
(2) Press the down arrow and observe that FAULT HISTORY? displays on the panel.
(3) Press YES.
(a) If there are no faults in Fault History, NO FAULT HISTORY will display on the panel.
(b) If any faults are present in Fault History, CYCLE N? will display on the panel. N is the number of the last power cycle to incur a fault.
1) To view fault details for this cycle, press YES.
a) The panel will display the faults that occurred during the cycle.
b) To view fault details, follow the procedures to view faults as shown in 3(a) thru 3(a)-4.

NOTE: Four lines of data are displayed in the MORE DETAILS? screen.
Line 1 - Message Number
Line 2 - Flight Deck Effect
Line 3 - LRU(s) responsible for the fault
Line 4 - Fault was hard or intermittent.

c) To view other faults, press the down arrow, observe that MORE DETAILS? displays on the panel, and then press NO.
(c) To view other power cycles, press menu until CYCLE X is displayed (where X is the last cycle displayed). Press the down arrow to display the next power cycle with a fault.
(d) If END OF LIST is displayed on the panel, no further power cycles exist in the Fault History.

D. Ground Tests
(1) Perform Check Self-Test Switches and Warning Lights (Ref 22-12-01)
(2) Perform Yaw Damper Configuration and System Tests (Ref 22-12-01)

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
Internal YDC faults requiring or resulting from YDC Self Test						
YDC Fault	msg no. 22-21000	FDE: YD disenga	YDC	LATCHED: Note 3	TYPE: Note 2	Possible Causes: • Yaw damper coupler Troubleshooting: • Pressurize A and B system hydraulics • Perform the yaw damper coupler installation test, ref AMM 12-12-01, page 401 step 4 • If the problem persists replace the yaw damper coupler
Internal YDC faults requiring or resulting from Servo Test						
YDC YD Fault	msg no. 22-21001	FDE: YD disenga	YDC	LATCHED: Note 3	TYPE: Note 2	Possible Causes: • Yaw damper coupler Troubleshooting: • Pressurize A and B system hydraulics • Perform the yaw damper coupler installation test, ref AMM 12-12-01, page 401 step 4 • If the problem persists replace the yaw damper coupler
Internal YDC faults requiring or resulting from Pressure reducer Test						
YDC YD Fault	msg no. 22-21001	FDE: YD disenga	YDC	LATCHED: Note 3	TYPE: Note 2	Possible Causes: • Yaw damper coupler Troubleshooting: • Perform the yaw damper coupler installation test, ref AMM 12-12-01, page 401 step 4 • If the problem persists replace the yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
YD engage solenoid faults						
YD sol Fault	msg no. 22-21003	FDE: YD disenga	YDC sol/ wrg/YDC	LATCHED: Note 3	TYPE: Note 2	Possible Causes: • Rudder power control unit actuator solenoid • Yaw damper coupler Troubleshooting: • Verify resistance from D295 socket 20 to ground is 60 ohms • Verify wiring between D295 socket 20 and D291 socket 1 at the rudder power control unit actuator solenoid • Verify connection to ground from D291 socket 2 at the rudder power control unit actuator solenoid • Inspect electrical connections for bent or broken pins, contami- nation, or damage • If problem persists, replace the rudder power control unit • If problem persists, replace the yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
PR solenoid faults						
PR sol Fault	msg no. 22-21004		PR sol/ wrg/YDC	LATCHED: Note 3	TYPE: Note 2	Possible Causes: • Pressure reducer bypass valve • Yaw damper coupler Troubleshooting: • Verify resistance from D295 socket 31 to ground is 60 ohms • Verify wiring between D295 socket 31 and D7377 socket 2 at the pressure reducer bypass valve solenoid • Verify connection to ground from D7377 socket 5 at the pressure reducer bypass valve solenoid • Inspect electrical connections for bent or broken pins, contami- nation, or damage • If problem persists, replace the pressure reducer bypass valve • If problem persists, replace the yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
YD switch faults						
YD sw Fault	msg no. 22-21005	FDE: YD disenga	YD sw/ wrg/YDC	LATCHED: Note 3	TYPE: Note 2	Possible Causes: • Yaw damper engage switch • Yaw damper coupler • Autopilot accessory unit Troubleshooting: • Verify resistance from D295 socket 14 to ground is less than 2.0 ohms with yaw damper engage switch in the OFF position • Verify wiring between D295 socket 14, the yaw damper engage switch, and autopilot accessory unit • Inspect electrical connections for bent or broken pins, contami- nation, or damage • If problem persists, replace the yaw damper engage switch • If problem persists, replace the yaw damper coupler • If problem persists, replace the autopilot accessory unit

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
Airplane configuration faults						
APL type error	msg no. 22-21006	FDE: YD disenga	APL/pgm pin/YDC	LATCHED: YES	TYPE: Note 2	Possible Causes: • Yaw damper coupler replacement • Aircraft wiring • Yaw damper coupler Troubleshooting: • This fault may be present when a yaw damper coupler is initally installed into an aircraft. Perform the yaw damper coupler installation test, ref AMM 12-12-01, page 401 step 4 • If problem persists, inspect D295 socket 17, D295 socket 53, and YDC electrical connector for bent or broken pins, contamination, or damage • Verify D295 socket 17 and D295 socket 53 resistance to ground is consistent with aircraft wiring diagrams, summarized 737 MODEL D295-53 D295-17 -100/200/ GROUND OPEN 200 ADV RIDE QUALITY -100/200/ OPEN OPEN 200 ADV • Perform the yaw damper coupler installation test, ref AMM 12-12-01, page 401 step 4 • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
28 VCD faults						
YDC 28V DC high	msg no. 22-21007	FDE: YD disenga	28V src/ wrg/YDC	LATCHED: NO	TYPE:	Possible Causes: • Excessive 28 VDC voltage applied to yaw damper coupler • Yaw damper coupler Troubleshooting: • Verify voltage to D295 socket 4 is between 22.0 and 29.5 volts DC with the yaw damper DC circuit breaker closed • If problem persists, replace yaw damper coupler
YDC 28V DC low	msg no. 22-21008	FDE: YD disenga	CB/wire/ 28V src	LATCHED: Note 3	TYPE: Note 2	Possible Causes: • Low or No. 28 VDC voltage applied to the yaw damper coupler • Yaw damper coupler Troubleshooting: • Verify yaw damper DC circuit breaker is closed • Verify voltage to D295 socket 4 is between 22.0 and 29.5 volts DC • Inspect electrical connectors for bent or broken pins, contami-nation, or damage • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
YDC/solenoid/wiring faults						
YD sol Fault	msg no. 22-21009	FDE: YD disenga	YD sol/ YDC/wrg	LATCHED: Note 3	TYPE:	Possible Causes: • Rudder power control unit actuator solenoid • Yaw damper coupler Troubleshooting: • Verify resistance from D295 socket 20 to ground is 60 ohms • Verify wiring between D295 socket 20 and D291 socket 1 at the rudder power control unit actuator solenoid • Verify connection to ground from D291 socket 2 at the rudder power control unit actuator solenoid • Inspect electrical connections for bent or broken pins, contami- nation, or damage • If problem persists, replace the rudder power control unit • If problem persists, replace the yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
PR sol Short	msg no. 22-21010		PR sol/ YDC/wrg	LATCHED: Note 3	TYPE:	Possible Causes: • Pressure reducer bypass valve • Yaw damper coupler Troubleshooting: • Verify resistance from D295 socket 31 to ground is 60 ohms • Verify wiring between D295 socket 31 and D7377 socket 2 at the pressure reducer bypass valve solenoid • Verify connection to ground from D7377 socket 5 at the pressure reducer bypass valve solenoid • Inspect electrical connections for bent or broken pins, contamination, or damage • If problem persists, replace the pressure reducer bypass valve • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
YDC/solenoid/wiring/power/faults						
YD sol low volt	msg no. 22-21011	FDE: YD disenga	28V/sol/ YDC/wire	LATCHED: Note 3	TYPE: Note 2	Possible Causes: • Low 28 VDC voltage applied to yaw damper coupler • Yaw damper engage switch • Yaw damper coupler • Rudder power control unit actuator solenoid Troubleshooting: • Verify voltage to D295 socket 4 is between 22.0 and 29.5 volts DC with the yaw damper DC circuit breaker closed • Verify resistance from D295 socket 14 to ground is less than 2.0 ohms with yaw damper engage switch in the OFF position • Verify resistance from D295 socket 20 to ground is 60 ohms • Verify wiring between D295 socket 20 and D291 socket 1 at the rudder power control unit actuator solenoid • Verify connection to ground from D291 socket 2 at the rudder power control unit actuator solenoid • Inspect electrical connectors for bent or broken pins, contami- nation or damage • If problem persists, re- place yaw damper coupler • If problem persists, replace the rudder power control unit actuator solenoid

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
PR sol low volt	msg no. 22-21012		28V/sol/ YDC/wire	LATCHED: Note 3	TYPE: Note 2	Possible Causes: • Low 28 VDC voltage applied to yaw damper coupler • Pressure reducer bypass valve • Yaw damper coupler Troubleshooting: • Verify voltage to D295 socket 4 is between 22.0 and 29.5 volts DC with the yaw damper DC circuit breaker closed • Verify resistance from D295 socket 31 to ground is 60 ohms • Verify wiring between D295 socket 31 and D7377 socket 1 at the pressure reducer bypass valve solenoid • Verify connection to ground from D7377 socket 2 at the pressure reducer bypass valve solenoid • Inspect electrical connectors for bent or broken pins, contami- nation or damage • If problem persists, re- place yaw damper coupler • If problem persists, replace the pressure reducer bypass valve

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
EHSV faults						
YD EHSV fault	msg no. 22-21013	FDE: YD disenga	YD EHSV /wrg/YDC	LATCHED: Note 3	TYPE: Note 2	Possible Causes: • Rudder power control unit transfer valve • Yaw damper coupler Troubleshooting: • Verify resistance between D295 socket 30 and D295 socket 32 is 2K ohms • Verify wiring between D295 and the rudder power control unit transfer valve: D295 socket 30 and D291 socket 8 D295 socket 32 and D291 socket 5 • Verify wiring between D291 socket 7 and D291 socket 6 at the rudder power control unit transfer valve • Inspect electrical connectors for bent or broken pins, contamination, or damage • If problem persists, remove and replace rudder power control unit transfer valve • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
26 VAC faults						
YDC 28V AC fault	msg no. 22-21014	FDE: YD disenga	LVDT/DAA /wrg/YDC	LATCHED: Note 3	TYPE: Note 2	Possible Causes: • Rudder power control unit position transducer • Yaw damper coupler Troubleshooting: • Verify resistance between D295 socket 9 and D295 socket 1 is 100 ohms • Verify wiring between D295 socket 9 and D295 socket 10 at the rudder power control unit position transducer • Verify connection to ground at D295 socket 1 • Verify connection to ground at D291 socket 9 at the rudder power control unit position transducer • Inspect electrical connectors for bent or broken pins, contami-nation, or damage • If problem persists, remove and replace rudder power control unit • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
Ground Test various faults not covered elsewhere						
YD not engaged	msg no. 22-21015		YD sw/ wrg/YDC	LATCHED: NO	TYPE:	Possible Causes: • Yaw damper engage switch not engaged during YDC BITE self test • Yaw damper coupler • Autopilot accessory unit Troubleshooting: • Perform the yaw damper coupler installation test, ref AMM 12-12-01, page 401 step 4 • Verify resistance from D295 socket 14 to ground is less than 2.0 ohms with yaw damper engage switch in the OFF position • Verify wiring between D295 socket 14, yaw damper engage switch, and autopilot accessory unit • Inspect electrical connectors for bent or broken pins, contamination, or damage • If problem persists, replace the yaw damper engage switch • If problem persists, replace yaw damper coupler • If problem persists, replace the autopilot accessory unit

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
B hyd not on	msg no. 22-21016		B hyd/ sensor	LATCHED: NO		Possible Causes: • B system hydraulics not pressurized during YDC BITE servo test • Flight control system B pressure switch • Yaw damper coupler Troubleshooting: • Verify wiring between D295 socket 7 and D816 socket 3 at the flight control system B pressure switch • Verify connection to ground at D816 socket 2 at the flight control system B pressure switch • Inspect electrical connectors for bent or broken pins, contami-nation, or damage • If problem persists, replace flight control system B pressure switch • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
YD actr limit	msg no. 22-21017		YD actr	LATCHED: NO		Possible Causes: • Excessive yaw damper actuator travel • Rudder power control unit position transducer • Yaw damper coupler Troubleshooting: • Verify resistance between: D295 socket 46 and D295 socket 47 is 80 ohms D295 socket 9 and D295 socket 1 is 100 ohms • Verify wiring at the rudder power control unit position transducer D295 socket 46 and D295 socket 11 D295 socket 47 and D291 socket 12 D295 socket 9 and D291 socket 10 • Verify connection to ground at: D295 socket 1 D291 socket 9 at the rudder power control unit position transducer • Inspect electrical connectors for bent or broken pins, contamination, or damage • If problem persists, replace rudder power control unit • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
YD actr rigging	msg no. 22-21018		YD actr/ rigging	LATCHED: NO		Possible Causes: • Asymmetric yaw damper actuator travel • Rudder power control unit position transducer • Yaw damper coupler Troubleshooting: • Verify resistance between: D295 socket 46 and D295 socket 47 is 80 ohms D295 socket 9 and D295 socket 1 is 100 ohms • Verify wiring at the rudder power control unit position transducer D295 socket 46 and D295 socket 11 D295 socket 47 and D291 socket 12 D295 socket 9 and D291 socket 10 • Verify connection to ground: D295 socket 1 D291 socket 9 at the rudder power control unit position transducer • Inspect electrical connectors for bent or broken pins, contamination, or damage • If problem persists, replace rudder power control unit • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
YDC,PDU in-compat	msg no. 22-21019		YD actr	LATCHED: NO		Possible Causes: • Aircraft wiring • Restricted yaw damper actuator travel • Rudder power control position transducer • Yaw damper coupler Troubleshooting: • Inspect D295 socket 17, D295 socket 53, and YDC electrical connector for bent or broken pins, contamination, or damage • Verify D295 socket 17 and D295 socket 53 resistance to ground is consistent with aircraft wiring diagrams, summarized: 737 MODEL D295-53 D295-17 -100/200/ GROUND OPEN 200 ADV RIDE QUALITY -100/200/ OPEN OPEN 200 ADV • Verify resistance between: D295 socket 46 and D295 socket 47 is 80 ohms D295 socket 9 and D295 socket 1 is 100 ohms

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
						• Verify wiring at the rudder power control unit transducer : D295 socket 46 and D291 socket 11 D295 socket 47 and D291 socket 12 D295 socket 9 and D291 socket 10 • Verify connection to ground: D295 socket 1 D291 socket 9 at the rudder power control unit position transducer • Inspect electrical connectors for bent or broken pins, contamination, or damage • If problem persists, replace rudder power control unit. Note that rudder power control units -3,-4,-5,-8 is not compatible with -100/200/200 ADV RIDE QUALITY yaw damper installation, reference Boeing 737 Service Letter 737-SL-27-050 • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
A hyd not on	msg no. 22-21020		A hyd/ PR/PR snsr	LATCHED: NO		Possible Causes: • Low pressure in A system hydraulics during YDC BITE pressure reducer test • Pressure reducer bypass valve • Yaw damper coupler Troubleshooting: • Verify resistance from D295 socket 31 to ground is 60 ohms • Verify wiring between D295 socket 31 and D7377 socket 2 at the pressure reducer bypass valve solenoid • Verify connection to ground from D7377 socket 5 at the pressure reducer bypass valve solenoid • Verify wiring at the pressure reducer bypass valve: D295 socket 51 and D737 socket 3 D295 socket 52 and D737 socket 2 • Verify connection to ground from D737 socket 1 at the pressure reducer bypass valve solenoid • Inspect electrical connectors for bent or broken pins, contamination, or damage • If problem persists, replace the pressure reducer bypass valve • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
PR off fault	msg no. 22-21021		PR/A hyd/PR snsr	LATCHED: NO		Possible Causes: • Low pressure in A system hydraulics during YDC BITE pressure reducer test • Pressure reducer bypass valve • Yaw damper coupler Troubleshooting: • Verify resistance from D295 socket 31 to ground is 60 ohms • Verify wiring between D295 socket 31 and D7377 socket 2 at the pressure reducer bypass valve solenoid • Verify connection to ground from D7377 socket 5 at the pressure reducer bypass valve solenoid • Verify wiring at the pressure reducer bypass valve: D295 socket 51 and D737 socket 3 D295 socket 52 and D737 socket 2 • Verify connection to ground from D737 socket 1 at the pressure reducer bypass valve solenoid • Inspect electrical connectors for bent or broken pins, contami- nation, or damage • If problem persists, replace the pressure reducer bypass valve • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
PR on fault	msg no. 22-21022		PR/PR snsr	LATCHED: NO		Possible Causes: • Hydraulc pressure not reduced in A system hydraulics during YDC BITE pressure reducer test • Pressure reducer bypass valve • Yaw damper coupler Troubleshooting: • Verify that the rudder pressure reducer valve manual lever is in the normal position • Verify resistance from D295 socket 31 to ground is 60 ohms • Verify wiring between D295 socket 31 and D7377 socket 2 at the pressure reducer bypass valve solenoid • Verify connection to ground from D7377 socket 5 at the pressure reducer bypass valve solenoid

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
						• Verify wiring at the pressure reducer bypass valve: D295 socket 51 and D737 socket 3 D295 socket 52 and D737 socket 2 • Verify connection to ground from D737 socket 1 at the pressure reducer bypass valve • Inspect electrical connectors for bent or broken pins, contamination, or damage • If problem persists, replace the pressure reducer bypass valve • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
prgm pin fault	msg no. 22-21023		prgm pin/YDC	LATCHED: NO		Possible Causes: • Program pins do not match aircraft type selected during YDC BITE program pin test • Operator error • Aircraft wiring • Yaw damper coupler Troubleshooting: • Inspect D295 socket 17, D295 socket 53, and YDC electrical connector for bent or broken pins, contamination, or damage • Verify D295 socket 17 and D295 socket 53 resistance to ground is consistent with aircraft wiring diagrams, summarized: 737 MODEL D295-53 D295-17 -100/200/ GROUND OPEN 200 ADV RIDE QUALITY -100/200/ OPEN OPEN 200 ADV • Perform the yaw damper coupler installation test, ref AMM 12-12-01, page 401 step 4 • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
A hyd lo on fault	msg no. 22-21025		Lamp/YDC /wiring	LATCHED: NO		Possible Causes: • Flt control A low pressure light did not illuminate during YDC BITE pressure reducer test • Operator error • Flt control A low pressure light inoperative • Yaw damper coupler Troubleshooting: • Verify that flt control A switch is in the ON position • Verify that flt control A light illuminates, press to test • Verify wiring between D295 socket 57 and flt control A low pressure light • Inspect D295 socket 57 and YDC electrical connector for bent or broken pins, contamination, or damage • Perform the yaw damper coupler installation test, ref AMM 12-12-01, page 401 step 4 • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
A hyd lo off fault	msg no. 22-21026		A hyd off/wire	LATCHED: NO		Possible Causes: • Flt control A low pressure light did not illuminate during YDC BITE pressure reducer test • Operator error • Flt control system A pressure switch • Yaw damper coupler Troubleshooting: • Pressurize A system hydraulics • Verify that flt control A switch is in the ON position • Verify that flt control A low pressure light is off • Verify wiring between D295 socket 57 and flt control A low pressure light • Inspect D295 socket 57 and YDC electrical connector for bent or broken pins, contami-nation, or damage • Perform the yaw damper coupler installation test, ref AMM 12-12-01, page 401 step 4 • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
A hyd off	msg no. 22-21027		B hyd off	LATCHED: YES		Possible Causes: • Low pressure B system hydraulics • Yaw damper coupler Troubleshooting: • Pressurize B system hydraulics • Verify that flt control B switch is in the ON position • Verify that flt control B low pressure light is off • Perform the yaw damper coupler installation test, ref AMM 12-12-01, page 401 step 4 • If problem persists, verify wiring between D295 socket 7 and D816 socket 3 at the flight control system B pressure switch • If problem persists, verify connection to ground at D816 socket 2 at the flight control system B pressure switch • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
AS in fault	msg no. 22-21031	FDE: YD disenga	AS src/ wrg/YDC	LATCHED: NO	TYPE	Possible Causes: • Air data computer-1 • Yaw damper coupler Troubleshooting: • Verify that air data computer-1 is installed • Verify wiring between D295 socket 40, D295 socket 41, D295 socket 42, D295 socket 43, and air data computer-1 • If problem persists, replace air data computer-1 • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
PR snsr in fault	msg no. 22-21034		PR snsr/ wrg/YDC	LATCHED: NO	TYPE:	Possible Causes: • Pressure reducer bypass valve • Yaw damper coupler Troubleshooting: • Verify resistance from D295 socket 31 to ground is 60 ohms • Verify wiring between D295 socket 31 and D7377 socket 2 at the pressure reducer bypass valve solenoid • Verify connection to ground from D7377 socket 5 at the pressure reducer bypass valve solenoid • Verify wiring at the pressure reducer bypass valve: D295 socket 51 and D737 socket 2 D295 socket 52 and D737 socket 3 • Verify connection to ground from D737 socket 1 at the pressure reducer bypass valve • Inspect electrical connectors for bent or broken pins, contami- nation, or damage • If problem persists, replace the pressure reducer bypass valve • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
YD sw in fault	msg no. 22-21035	FDE: YD disenga	YD sw/ wrg/YDC	LATCHED: NO	TYPE:	Possible Causes: • Open circuit between D295 socket 14 and yaw damper engage switch contacts • Yaw damper engage switch • Yaw damper coupler Troubleshooting: • Verify nominal resist
LRRA 1 fault	msg no. 22-21036	FDE: YD disenga	LRRA 1/ wrg/YDC	LATCHED: NO	TYPE:	Possible Causes: • Radio altimeter-1 • Yaw damper coupler Troubleshooting: • This is a normal condition if radio altimeter-1 is not powered or not installed • If radio altimeter-1 is installed, close radio altimeter-1 AC circuit breaker • If problem persists, verify wiring between D295 socket 13 and D101B socket 12, flag alarm from radio altimeter-1 • Inspect electrical connectors for bent or broken pins, contami- nation, or damage • If problem persists, troubleshoot radio altimeter-1 • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
LRRA 2 fault	msg no. 22-21037	FDE: YD disenga	LRRA 2/ wrg/YDC	LATCHED: NO	TYPE:	Possible Causes: • Radio altimeter-2 • Yaw damper coupler Troubleshooting: • This is a normal condition if radio altimeter-2 is not powered or not installed • If radio altimeter-2 is installed, close radio altimeter-2 AC • If problem persists, verify wiring between D295 socket 39 and D103B socket 12, flag alarm from radio altimeter-2 • Inspect electrical connectors for bent or broken pins, contamination, or damage • If problem persists, troubleshoot radio altimeter-2 • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
LRRA mis-compr	msg no. 22-21041 [1]		LRRA #1/ LRRA #2	LATCHED: NO	TYPE:	Possible Causes: • Radio altimeter-1 and radio altimeter-2 radio height disagree • Yaw damper coupler Troubleshooting: • Troubleshoot radio altimeter-1 and radio altimeter-2 • Verify wiring between D295 and radio altimeter-1 D295 socket 13 and D101B socket 12 D295 socket 21 and D101B socket 46 D295 socket 24 and D101B socket 47 • Verify wiring between D295 and radio altimeter-2 D295 socket 39 and D103B socket 12 D295 socket 22 and D103B socket 23 D295 socket 54 and D103B socket 22 •If problem persists, replace yaw damper coupler

[1] For airplanes with one LRRA installed and full provisions for LRRA-2, FaultCode 22-21041 will be seen.

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
Uncorrelated response monitor trips						
YD actr fault	msg no. 22-21042	FDE: YD disenga	YD actr/ LVDT/wrg	LATCHED: YES	TYPE:	Possible Causes: • Rudder power control unit actuator solenoid • Rudder power control unit transfer valve • Rudder power control unit position transducer • Yaw damper coupler Troubleshooting: • Verify resistance: D295 socket 20 to ground is: 72 to 76 ohms (P/N 881600-1001) 49 to 80 ohms (P/N 45080) 79 to 115 ohms (P/N 45080-1) D295 socket 30 to D295 socket 32 is 2K D295 socket 9 to D295 socket 1 is 100 ohms D295 socket 46 to D295 socket 47 is 80 ohms • Verify wiring at the rudder power control unit actuator solenoid D295 socket 20 and D291 socket 1 D295 socket 30 and D291 socket 8 D295 socket 32 and D291 socket 5 D291 socket 7 and D291 socket 6

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
						• Verify wiring at the rudder power control unit position transducer D295 socket 46 and D291 socket 11 D295 socket 47 and D291 socket 12 D295 socket 9 and D291 socket 10 • Verify connection to ground at D291 socket 2 at the rudder power control unit actuator solenoid D295 socket 1 D291 socket 9 at the rudder control unit position transducer • Inspect electrical connectors for bent or broken pins, contamination, or damage • If problem persists, replace rudder power control unit • If problem persists, replace yaw damper coupler

Fault Message	Fault Detail 1	Fault Detail 2	Fault Detail 3	Fault Detail 4a	Fault Detail 4b	Isolation Procedure
PR fault	msg no. 22-21043		PR sol/ PR snsr	LATCHED: YES	TYPE:	Possible Causes: • Pressure reducer bypass valve • Yaw damper coupler Troubleshooting: • Verify that the rudder pressure reducer valve manual lever is in the normal position • Verify resistance from D295 socket 31 to ground is ohms • Verify wiring between D295 socket 31 and D737 socket 2 at the pressure reducer bypass valve solenoid • Verify connection to ground from D737 socket 5 at the pressure reducer bypass valve solenoid • Verify wiring at the pressure reducer bypass valve: D295 socket 51 and D737 socket 3 D295 socket 52 and D737 socket 2 • Verify connection to ground from D737 socket 1 at the pressure reducer bypass valve • Inspect electrical connectors for bent or broken pins, contamination, or damage • If problem persists, replace the pressure reducer bypass valve • If problem persists, replace yaw damper coupler

YAW DAMPER COUPLER - REMOVAL/INSTALLATION

1. General
   A. The yaw damper coupler is retained in the electronic rack by cam-action levers which hook over forks on the shelf.
2. Remove Coupler
   A. Open yaw damper AC and DC circuit breakers on P6 panel

      CAUTION: DO NOT TOUCH THE CONNECTOR PINS OR OTHER CONDUCTORS OF THE YAW DAMPER. IF YOU TOUCH THESE CONDUCTORS, ELECTROSTATIC DISCHARGE CAN CAUSE DAMAGE TO THE COUPLER.

   B. Release coupler by pressing triggers then pull ejection levers to release unit.
   C. Remove coupler.
3. Install Coupler
   A. Place coupler in rack and simultaneously push ejection levers up until levers are latched (Ref 20-10-111).
   B. Close yaw damper circuit breakers.
4. Installation Test

   NOTE: The following test must be completed to release the aircraft for further flight with the Yaw Damper Coupler active. A portion of the test configures the Yaw Damper Coupler to the airplane configuration.

   A. Prepare for the Yaw Damper Coupler test.
      (1) Provide electrical power to the aircraft and energize load control center panels P6 and P18 (Ref 24-22-00).
      (2) Energize hydraulic systems A and B and ensure FLT CONTROL A and FLT CONTROL B switches are positioned to ON (Ref 29-15-00).
      (3) Ensure the following circuit breakers on load control center panels are closed (Reference 22-12-01/501, step 2.C for circuit breaker list.).
   B. Yaw Damper Configuration and System Tests

      NOTE: If no keypad entries are made for 5 minutes, the YDC BITE display will turn off. It will be necessary to start over if this occurs.

      NOTE: A fault message "LRRA 2 FAULT" is normal if No. 2 LRRA (Low Range Radio Altimeter) is not installed or powered.

      (1) Press ON/OFF on the YDC BITE panel. Observe that ENTER BITE? Is displayed on the panel.

(2) Observe that PLEASE WAIT is temporarily displayed followed by EXISTING FAULTS? on the panel. Observe the AUTOPILOT DISENGAGE WARNING lights illuminate steadily.
(3) Within 10 seconds, press YES on the YDC BITE panel. Ensure that the panel displays NO FAULTS.
   (a) If the YDC BITE panel does not display NO FAULTS, then press the down arrow until RESET LATCHES? is displayed.
      1) Press YES.
      2) When ARE YOU SURE? is displayed on the panel, press YES again. Observe that RESET IN PROGRESS is displayed on the panel.
      3) Wait 10 seconds. The YDC BITE panel will display NO FAULTS or it will display an existing fault. For the purpose of this test, it is irrelevant if a fault is displayed or not.
(4) Press MENU. Observe EXISTING FAULTS? is displayed on the panel.
(5) Press down arrow until GROUND TEST? is displayed on the panel.
(6) Press YES and observe that SELF TEST? is displayed on the panel.
(7) Press down arrow until PRGM PIN TEST? is displayed on the panel.
(8) Press YES and observe that APL TYPE 100/200? is displayed on the panel.

NOTE: If the airplane type is not the same as the airplane you are on, the program pins are not correct. The program pins must be changed.

(9) Press YES. Observe that CONFIRM APL type followed by 737 -100 or -200? is displayed on the panel.
(10) Press YES. Observe that the panel displays STORING CONFIG followed by TEST PASS.
(11) Ensure the FLT CONTROL B switch is on.
(12) Press ON/OFF on the YDC BITE panel. Observe that TURN OFF DISPLAY? is displayed on the panel.
(13) Press YES and pause 15 seconds. Observe DISPLAY OFF is temporarily displayed followed by the panel blanking. Observe the AUTOPILOT DISENGAGE WARNING lights are not illuminated.
(14) Press ON/OFF. Observe that ENTER BITE? is displayed on the panel.
(15) Within 10 seconds, press YES. Observe that PLEASE WAIT is temporarily displayed followed by EXISTING FAULTS? on the YDC BITE panel. Observe the AUTOPILOT DISENGAGE WARNING lights illuminate steadily.
(16) Press down arrow until GROUND TESTS? is displayed on the panel.
(17) Press YES and observe that SELF TEST? is displayed on the panel.
(18) Station an observer facing the tail to monitor rudder deflection.

(19) Press YES on the YDC BITE panel. Observe that WARNING! WARNING!, then RUDDER MOVEMENT, then RUDDER CLEARED? displays on the panel.

<u>WARNING</u>: ENSURE PERSONNEL ARE CLEAR OF RUDDER PRIOR TO PERFORMING THESE STEPS. DEATH OR INJURY TO PERSONNEL MAY RESULT.

(20) Press YES. Observe that YD ENGA SW ON? is displayed on the panel.
(21) Engage the Yaw Damper Engage switch and observe that the Yaw Damper indicator light extinguishes within 3 seconds.
(22) Press YES on the YDC BITE panel and observe that the panel displays TEST IN PROGRESS followed by TEST PASS after 70-150 seconds.

<u>NOTE</u>: TEST IN PROGRESS is displayed on the YDC BITE panel and the AUTOPILOT DISENGAGE WARNING lights flash off and on during this test.

(23) Press ON/OFF on the YDC BITE panel. Observe that TURN OFF DISPLAY? is displayed on the panel.
(24) Press YES and pause 15 seconds. Observe that DISPLAY OFF is temporarily displayed followed by the panel blanking. Observe the AUTOPILOT DISENGAGE WARNING lights are not illuminated.

C. Servo Test
(1) For aircraft prior to L/N 213 not incorporating SB 737-22-1009, disengage the Yaw Damper.
(2) Position FLT CONTROL B switch to OFF. Observe that YAW DAMPER engage switch is in the OFF position and the YAW DAMPER warning light, the FLT CONT annunciator light, and both MASTER CAUTION lights illuminate.
(3) Press either MASTER CAUTION light. Press ON/OFF on the YDC BITE panel. Observe that ENTER BITE? is displayed on the panel.
(4) Within 10 seconds, press YES.
(5) Observe that PLEASE WAIT is temporarily displayed followed by EXISTING FAULTS? on the panel. Observe the AUTOPILOT DISENGAGE WARNING lights illuminate steadily.
(6) Press down arrow until GROUND TESTS? is displayed on the panel.
(7) Press YES. Observe that SELF TEST? is displayed on the panel.
(8) Press down arrow. Observe that SERVO TEST? is displayed.
(9) Press YES. Observe that WARNING! WARNING!, then RUDDER MOVEMENT, then RUDDER CLEARED? is displayed on the panel.

<u>WARNING</u>: ENSURE PERSONNEL ARE CLEAR OF RUDDER PRIOR TO PERFORMING THESE STEPS. DEATH OR INJURY TO PERSONNEL MAY RESULT.

(10) Press YES. Observe that FLT CTRL B SW ON? is displayed on the panel.

(11) Position FLT CONTROL B switch to ON. Engage the Yaw Damper engage switch. Observe that the Yaw Damper indicator light extinguishes within 3 seconds.

NOTE: During the following steps, the Yaw Damper indicator on the P2 center instrument panel will move. Ensure that it moves left, then right and then returns to the center position. This movement should occur twice during the test.

(12) Press YES on the YDC BITE panel. Observe that TEST IN PROGRESS is temporarily displayed followed by TESTING LIMITS.

(13) Observe that, after approximately 2 seconds, the following data is displayed. X's are numeric and their actual values are irrelevant.
(a) YD cmd positive
(b) LVDT pos XXXXX deg
(c) YD cmd negative
(d) LVDT pos XXXXX deg
(e) YD cmd zero
(f) LVDT pos XXXXX deg
(g) PCU LIM +/- X deg

(14) Observe, after approximately 5 seconds, that the panel displays the following data:
(a) SECOND SWEEP
(b) YD cmd XXXXX deg
(c) LVDT pos XXXXX deg
(d) YD cmd XXXXX deg
(e) LVDT pos XXXXX deg
(f) YD cmd XXXXX deg
(g) LVDT pos XXXXX deg

(15) Ensure that the panel displays TEST PASS after approximately 60 seconds.

(16) Press ON/OFF. Observe that TURN OFF DISPLAY? is displayed on the panel.

(17) Press YES. Observe that DISPLAY OFF is temporarily displayed followed by the panel blanking. Observe the AUTOPILOT DISENGAGE WARNING lights are not illuminated.

(18) Position the YAW DAMPER engage switch to OFF. Observe that the YAW DAMPER warning light, the FLT CONT annunciator light, and both MASTER CAUTION lights illuminate.

D. Pressure Reducer Test

(1) Position FLT CONTROL A switch to OFF. Observe that the FLT CONTROL A LOW PRESSURE light illuminates.

(2) Press ON/OFF on the YDC BITE Panel. Within 10 seconds, press YES. Observe that PLEASE WAIT is temporarily displayed followed by EXISTING FAULTS? on the YDC BITE panel. Observe the AUTOPILOT DISENGAGE WARNING lights illuminate steadily.

(3) Press down arrow until GROUND TESTS? is displayed on the panel.
(4) Press YES. Observe that SELF TEST? is displayed on the panel.
(5) Press down arrow until REDUCER TEST? is displayed on the panel.
(6) Press YES. Observe that WARNING! WARNING!, then RUDDER MOVEMENT, then RUDDER CLEARED? is displayed on the panel.

WARNING: ENSURE PERSONNEL ARE CLEAR OF RUDDER PRIOR TO PERFORMING THESE STEPS. DEATH OR INJURY TO PERSONNEL MAY RESULT.

(7) Press YES on the YDC BITE panel. Observe that SYS A HYD ON? is displayed.
(8) Position FLT CONTROL A Switch to ON. Observe that the FLT CONTROL A LOW PRESSURE light extinguishes within 6 seconds.
(9) Press YES on the YDC BITE panel. Observe the following data is displayed. The X's are numeric values and their actual values are to be checked.
(a) Reducer = off
(b) A hyd XXXX psi
(c) Make sure that the value displayed in step (b) is greater than 2200 psi.
(d) Reducer = on
(e) A hyd XXXX psi
(f) Make sure that the value displayed in step (e) is between 1250 and 1650 psi.
(g) Reducer = off
(h) A hyd XXXX psi
(i) Make sure that the value displayed in step (g) is greater than 2200 psi.
(10) Observe that Lo Press (A) Off? is displayed on the YDC BITE panel and the FLT CONTROL A LOW PRESSURE light is OFF.
(11) Press YES on the YDC BITE panel. Observe that LO PRESS (A) ON? is displayed.
(12) Observe that the FLT CONTROL A LOW PRESSURE light is illuminated.
(13) Press YES on the YDC BITE panel. Observe that the panel displays TEST PASS.
(14) Press ON/OFF. Observe that TURN DISPLAY OFF? is displayed on the panel.
(15) Press YES. Observe that the DISPLAY OFF is temporarily displayed followed by the panel blanking. Observe the AUTOPILOT DISENGAGE WARNING lights are not illuminated.
(16) Position the FLT CONTROL A switch to OFF.

5. Restore Aircraft to Normal Configuration
A. Disengage yaw damper.
B. Disconnect electrical power. (Ref 24-22-00)
C. Disconnect hydraulic power. (Ref 29-15-00)

YAW DAMPER SYSTEM - ADJUSTMENT/TEST

1. General
   A. The successful completion of these tests ensures that the yaw damper system is fully operable. It is assumed that all systems furnishing signal data to the yaw damper are in proper working order and that the rudder control systems are properly rigged.
2. Prepare to Test Yaw Damper System
   A. Provide electrical power to aircraft and energize load control center P6 and P18 (Ref 24-22-00).
   B. Energize hydraulic systems A and B and ensure FLT CONTROL A and FLT CONTROL B switches are positioned to ON (Ref 29-15-00).
   C. Ensure following circuit breakers on load control center are closed.

      NOTE: Circuit breakers may vary slightly (or even not be used) from airplane to airplane.

      (1) Panel P6-3
         (a) Indicator Master Dim Bus Bat -- (C310)
         (b) Indicator Master Dim Bus No. 1 DC -- (C311)
         (c) Indicator Master Dim Bus No. 2 DC -- (C312)
         (d) Dim and Test -- (C133)
         (e) Indicator Master Dim Bus Sect 1 -- C313)
         (f) Indicator Master Dim Bus Sect 2 -- (C314)
         (g) Indicator Master Dim Bus Sect 3 -- (C315)
         (h) Master Caution Air Cond -- (C571)
         (i) Master Caution Anti-Ice -- (C573)
         (j) Master Caution Fuel -- (C572)
         (k) Master Caution Bus No. 1 -- (C132)
         (l) Master Caution Bus Bat -- (C131)
         (m) Master Caution Bus Sect 4 -- (C316)
         (n) Master Caution Bus Sect 5 -- (C317)
         (o) Master Caution Bus Sect 6 -- (C318)
         (p) Yaw Damper AC -- (C285)
         (q) Yaw Damper DC -- (C286)
      (2) Panel P6-1
         (a) Radio Altimeter-2 AC (if airplane has second altimeter installed) -- (C321)
      (3) Panel P18-1
         (a) Radio Altimeter-1 AC -- (C302)
      (4) Panel P18-2
         (a) Air Data Computer No. 1 (115 V AC) -- (C300)
         (b) Air Data Computer No. 1 (26 V AC) -- (C425)

3. Test Yaw Damper System

A. Yaw Damper Configuration and System Tests

NOTE: If no keypad entries are made for 5 minutes, the YDC BITE display will turn off. It will be necessary to start over if this occurs.

(1) Press ON/OFF. Observe that ENTER BITE? is displayed on the panel.
(2) Within 10 seconds, press YES. Observe that PLEASE WAIT is temporarily displayed followed by EXISTING FAULTS? on the YDC panel. Observe that the AUTOPILOT DISENGAGE WARNING lights illuminate steadily.
(3) Press YES on the YDC BITE panel. Ensure that the panel displays NO FAULTS.

NOTE: For aircraft equipped with only one Radio Altimeter System, a "LRRA #2° fault code, message number 22-21037, is a normal condition if Radio Altimeter-2 is not powered or not installed.

(a) If the YDC BITE panel does not display NO FAULTS, then press the down arrow until RESET LATCHES? is displayed.
1) Press YES.
2) When ARE YOU SURE? is displayed on the panel, press YES again. Observe that RESET IN PROGRESS is displayed on the panel.
3) Wait 10 seconds. The YDC BITE panel will display NO FAULTS or it will display an existing fault. For the purpose of this test, it is irrelevant if a fault is displayed or not.

(4) Press MENU. Observe EXISTING FAULTS? is displayed on the panel.
(5) Press down arrow until GROUND TEST? is displayed on the panel.
(6) Press YES and observe that SELF TEST? is displayed on the panel.
(7) Press down arrow until PRGM PIN TEST? is displayed on the panel.
(8) Press YES and observe that APL TYPE 100/200? is displayed on the panel.
(9) Make sure that the airplane type shown on the display is the same as the airplane you are on.

NOTE: If the airplane type is not the same as the airplane you are on, the program pins are not correct. The program pins must be changed.

(10) Press YES. Observe that CONFIRM APL type followed by 737 -100 or -200? is displayed on the panel.
(11) Press YES. Observe that the panel displays STORING CONFIG followed by TEST PASS.
(12) Ensure the FLT CONTROL B switch is on.
(13) Press ON/OFF on the YDC BITE panel. Observe that TURN OFF DISPLAY? is displayed on the panel.

(14) Press YES and pause 15 seconds. Observe DISPLAY OFF is temporarily displayed followed by the panel blanking. Observe the AUTOPILOT DISENGAGE WARNING lights are not illuminated.
(15) Press ON/OFF. Observe that ENTER BITE? is displayed on the panel.
(16) Within 10 seconds, press YES. Observe that PLEASE WAIT is temporarily displayed followed by EXISTING FAULTS? on the YDC BITE panel. Observe the AUTOPILOT DISENGAGE WARNING lights illuminate steadily.
(17) Press down arrow until GROUND TESTS? is displayed on the panel.
(18) Press YES and observe that SELF TEST? is displayed on the panel.
(19) Station an observer facing the tail to monitor rudder deflection.
(20) Press YES on the YDC BITE panel. Observe that WARNING! WARNING!, then RUDDER MOVEMENT, then RUDDER CLEARED? displays on the panel.

WARNING: ENSURE PERSONNEL ARE CLEAR OF RUDDER PRIOR TO PERFORMING THESE STEPS. DEATH OR INJURY TO PERSONNEL MAY RESULT.

(21) Press YES. Observe that YD ENGA SW ON? is displayed on the panel.
(22) Engage the Yaw Damper Engage switch and observe that the Yaw Damper indicator light extinguishes within 3 seconds.
(23) Press YES on the YDC BITE panel and observe that the panel displays TEST IN PROGRESS followed by TEST PASS after 70-150 seconds.

NOTE: TEST IN PROGRESS is displayed on the YDC BITE panel and the AUTOPILOT DISENGAGE WARNING lights flash off and on during this test.

(24) Press ON/OFF on the YDC BITE panel. Observe that TURN OFF DISPLAY? is displayed on the panel.
(25) Press YES and pause 15 seconds. Observe that DISPLAY OFF is temporarily displayed followed by the panel blanking. Observe the AUTOPILOT DISENGAGE WARNING lights are not illuminated.

B. Check Self-Test Switches and Warning Lights
(1) Press either MASTER CAUTION light. Position FLT CONTROL B switch to ON if not already engaged.
(2) Close YAW DAMPER DC circuit breakers.
(3) Station an observer in the Electrical/Electronic bay.
(4) Ensure that the AUTOPILOT DISENGAGE WARNING lights are not illuminating. Press the ON/OFF button on the YDC BITE Panel. Observe that ENTER BITE? displays on the YDC BITE panel.
(5) Within 10 seconds, press YES. Observe that PLEASE WAIT temporarily displays followed by EXISTING FAULTS? on the YDC BITE panel.
(6) Ensure that the AUTOPILOT DISENGAGE WARNING lights (located on the Captain's and First Officer's panel) illuminate steadily.

C. Servo Test
(1) For aircraft prior to L/N 213 not incorporating SB 737-22-1009, disengage the Yaw Damper.
(2) Position FLT CONTROL B switch to OFF. Observe that YAW DAMPER engage switch is in the OFF position and the YAW DAMPER warning light, the FLT CONT annunciator light, and both MASTER CAUTION lights illuminate.
(3) Press down arrow until GROUND TESTS? is displayed on the panel.
(4) Press YES. Observe that SELF TEST? is displayed on the panel.
(5) Press down arrow. Observe that SERVO TEST? is displayed.
(6) Press YES. Observe that WARNING! WARNING!, then RUDDER MOVEMENT, then RUDDER CLEARED? is displayed on the panel.

WARNING: ENSURE PERSONNEL ARE CLEAR OF RUDDER PRIOR TO PERFORMING THESE STEPS. DEATH OR INJURY TO PERSONNEL MAY RESULT.

(7) Press YES. Observe that FLT CTRL B SW ON? is displayed on the panel.
(8) Position FLT CONTROL B switch to ON. Engage the Yaw Damper engage switch. Observe that the Yaw Damper indicator light extinguishes within 3 seconds.

NOTE: During the following steps, the Yaw Damper indicator on the P2 center instrument panel will move. Ensure that it moves left, then right and then returns to the center position. This movement should occur twice during the test.

(9) Press YES on the YDC BITE panel. Observe that TEST IN PROGRESS is temporarily displayed followed by TESTING LIMITS.
(10) Observe that, after approximately 2 seconds, the following data is displayed. X's are numeric and their actual values are irrelevant.
(a) YD cmd positive
(b) LVDT pos XXXXX deg
(c) YD cmd negative
(d) LVDT pos XXXXX deg
(e) YD cmd zero
(f) LVDT pos XXXXX deg
(g) PCU LIM +/- X deg
(11) Observe, after approximately 5 seconds, that the panel displays the following data:
(a) SECOND SWEEP
(b) YD cmd XXXXX deg
(c) LVDT pos XXXXX deg
(d) YD cmd XXXXX deg
(e) LVDT pos XXXXX deg
(f) YD cmd XXXXX deg

(g) LVDT pos XXXXX deg

(12) Ensure that the panel displays TEST PASS after approximately 60 seconds.

(13) Press ON/OFF. Observe that TURN OFF DISPLAY? is displayed on the panel.

(14) Press YES. Observe that DISPLAY OFF is temporarily displayed followed by the panel blanking. Observe the AUTOPILOT DISENGAGE WARNING lights are not illuminated.

(15) Position the YAW DAMPER engage switch to OFF. Observe that the YAW DAMPER warning light, the FLT CONT annunciator light, and both MASTER CAUTION lights illuminate.

D. Pressure Reducer Test

(1) Position FLT CONTROL A switch to OFF. Observe that the FLT CONTROL A LOW PRESSURE light illuminates.

(2) Press ON/OFF on the YDC BITE Panel. Within 10 seconds, press YES. Observe that PLEASE WAIT is temporarily displayed followed by EXISTING FAULTS? on the YDC BITE panel. Observe the AUTOPILOT DISENGAGE WARNING lights illuminate steadily.

(3) Press down arrow until GROUND TESTS? is displayed on the panel.

(4) Press YES. Observe that SELF TEST? is displayed on the panel.

(5) Press down arrow until REDUCER TEST? is displayed on the panel.

(6) Press YES. Observe that WARNING! WARNING!, then RUDDER MOVEMENT, then RUDDER CLEARED? is displayed on the panel.

WARNING: ENSURE PERSONNEL ARE CLEAR OF RUDDER PRIOR TO PERFORMING THESE STEPS. DEATH OR INJURY TO PERSONNEL MAY RESULT.

(7) Press YES on the YDC BITE panel. Observe that SYS A HYD ON? is displayed.

(8) Position FLT CONTROL A Switch to ON. Observe that the FLT CONTROL A LOW PRESSURE light extinguishes within 6 seconds.

(9) Press YES on the YDC BITE panel. Observe that the following data is displayed. The X's are numeric values and their actual values are irrelevant.

(a) Reducer = off

(b) A hyd XXXX psi

(c) Make sure that the value displayed in step (b) is greater than 2200 psi.

(d) Reducer = on.

(e) A hyd XXXX psi.

1) AIRPLANES PRE SB 737-27-1252;
Make sure that the value (A hyd XXXX psi) is between 1250 and 1650 psi.

2) AIRPLANES POST SB 737-27-1252;
Make sure that the value (A hyd XXXX psi) is more than 1600 and less than 2000 psi.

(f) Reducer - off.

(g) Make sure that the value displayed in stp (g) is greater than 2000 psi

(10) Observe that Lo Press (A) Off? is displayed on the YDC BITE panel and the FLT CONTROL A LOW PRESSURE light is OFF.

(11) Press YES on the YDC BITE panel. Observe that LO PRESS (A) ON? is displayed.

(12) Observe that the FLT CONTROL A LOW PRESSURE light is illuminated.

(13) Press YES on the YDC BITE panel. Observe that the panel displays TEST PASS.

(14) Press ON/OFF. Observe that TURN DISPLAY OFF? is displayed on the panel.

(15) Press YES. Observe that the DISPLAY OFF is temporarily displayed followed by the panel blanking. Observe the AUTOPILOT DISENGAGE WARNING lights are not illuminated.

(16) Position the FLT CONTROL A switch to OFF.

E. Yaw Damper Interface Test

(1) Check system configuration.

(a) Press ON/OFF on the YDC BITE panel. Observe that ENTER BITE? is displayed on the panel.

(b) Within 10 seconds press YES. Observe that PLEASE WAIT is temporarily displayed followed by EXISTING FAULTS? on the YDC BITE panel. Observe the AUTOPILOT DISENGAGE WARNING lights illuminate steadily.

(c) Press down arrow until OTHER FUNCTNS? is displayed.

(d) Press YES. Observe that SYSTEM CONFIG? is displayed on the panel.

(e) Press YES. Observe that S/W VRSN XXXXXXXX is displayed on the panel. The actual values for XXXXXXXX is irrelevant.

(f) Press down arrow.

1) Record the data that shows on the panel.

NOTE: It will show APL TYPE 100/200, or APL TYPE 1/200 RQ. You will do some steps below only if you have an airplane with APL TYPE 1/200 RQ.

(g) Press ON/OFF. Observe that TURN DISPLAY OFF? is displayed on the panel.

(h) Press YES. Observe that DISPLAY OFF is temporarily displayed followed by the panel blanking. Observe the AUTOPILOT DISENGAGE WARNING lights are not illuminated.

(2) AIRPLANES WITH APL TYPE 1/200 RQ;
Prepare for roll simulation.

(a) If dual Inertial Navigation System (INS) has been installed on the aircraft

1) Ensure that the MSU mode selector switch is set to the OFF position.

2) Install the Attitude Slew Test Plug (Litton Part Number 456058-01) on INU test connector J4.
3) Position the MSU switch to STBY. The INS CDU Warn lamp will start continuous flashing.
4) For LTN-72RL models, key in the date including leading zeros (e.g., 100780 for July 10, 1980) on the INS CDU. Press ENT to accept date.
5) For LTN-72 and LTN-74R models, position the INS CDU Display selector switch to POS.
6) For LTN-72RL models, press POS.

(b) If dual INS has not been installed
1) Unbolt the Captain's VG-1 vertical gyro from mount.

(3) Prepare YDC BITE for interface test.
(a) Press ON/OFF on the YDC Bite Panel. Observe that ENTER BITE? is displayed on the panel.
(b) Within 10 seconds, press YES. Observe that PLEASE WAIT is temporarily displayed followed by EXISTING FAULTS? on the YDC BITE panel.
(c) Press down arrow until OTHER FUNCTIONS? is displayed on the panel.
(d) Press YES. Observe that SYSTEM CONFIG? is displayed on the panel.
(e) Press down arrow. Observe that I/O MONITOR? is displayed on the panel.
(f) Press YES. Observe that ANALOG I/O? is displayed on the panel.
(g) Press YES.

<u>NOTE</u>: Step (h) should be omitted for aircraft with APL TYPE 100/200, identified in Yaw Damper Interface test step (1) (f).

(h) Press down arrow until the panel displays roll att XXX.X deg.

<u>NOTE</u>: It should be noted that step (4)(a)1) will take approximately 2 minutes to complete. The YDC BITE panel will shut off with approximately 5 minutes of inactivity. Therefore, if step (4)(a)1) is to be performed, it should occur as soon as possible.

(4) AIRPLANES WITH APL TYPE 1/200 RQ;
Simulate roll.

(a) If dual Inertial Navigation System (INS) has been installed on the aircraft

1) Enter a simulation roll angle of 15° ±2° on the INS CDU. Approximately 2 minutes after achieving Status 80, the gimbals will start to slew.

2) To enter desired roll angle:

a) For LTN-72 and LTN-72R models, press 6E or W4 on the INS CDU keypad.

b) For LTN-72RL models, press N/6E or L/W4.

NOTE: If the INU is facing aft (INU handles facing aft) and 6E or N/6E is pressed, the INU will provide right wing down or roll right. If the INU is facing forward, left wing down or roll left will be provided. The opposite applies to W4 and L/W4.

c) Enter degrees and minutes of roll. Each segment must have two digits (e.g., 010203 for 1°2.03')

d) Press INSERT to accept or CLEAR to re-enter value.

e) If an invalid value was entered, the left display will flash. Press CLEAR and re-enter the value.

3) Verify the following conditions:

a) INS CDU WARN lamp flashing

b) Position INS CDU Display switch to DSR TK STS and observe that action code 07 is displayed.

NOTE: If neither condition occurs, position MSU switch to OFF, remove Attitude Slew Plug and verify that Pin A is jumpered to Pin U. Reinstall test plug and restart test.

4) Roll angles may be changed while the MSU switch is in the ALIGN position:

a) Press HOLD on the INS CDU. Ensure that the button illuminates.

b) Enter the roll angle on the INS CDU panel using same method as step (4)(a)1)a).

c) Press HOLD.

5) Observe that both the ADI and the YDC BITE panel displays a nonzero roll attitude. Observe that ADI displays a nonzero value and the YDC BITE panel displays roll att ± 15° (with a tolerance of 5°).

6) At the conclusion of the simulation, position the MSU switch to OFF and remove the Attitude slew Test plug to restore the system to normal operation.

(b) If dual INS has not been installed

1) Manually tilt the vertical gyro left and observe that the ADI displays a nonzero reading and the YDC BITE panel displays nonzero roll att.
2) Manually tilt the vertical gyro right and observe that the ADI displays a nonzero reading and the YDC BITE panel displays nonzero roll att.
3) At the conclusion of the simulation, return the Captain's VG-1 vertical gyro to its original state prior to the test.

(5) Perform radio altimeter test.

NOTE: Connect the radio altimeter test set to Radio Altimeter #1 and Radio Altimeter #2, if installed.

(a) If Radio Altimeter #2 is installed, open Radio Altimerter-2 AC circuit breaker.
(b) Press down arrow on YDC BITE panel until the panel displays a reading for RA 1.
(c) Using radio altimeter test set, simulate an altitude of 1400 feet. Observe that the panel displays RA1 of 1400 ±200 feet.
(d) Decrease simulated altitude to 750 feet. Observe that the YDC BITE panel displays RA 1 of 750 ±200 feet.
(e) AIRPLANES WITH SECOND RADIO ALTIMETER;
Do these steps:

1) Press down arrow until the panel displays reading for RA 2.
2) Observe that the panels displays RA 2 invalid.
3) Close Radio Altimeter-2 circuit breaker.
4) Increase simulated altitude to 1400 feet. Observe that the panel displays RA 2 of 1400 ±200 feet.
5) Decrease simulated altitude to 750 feet. Observe that the YDC BITE panel displays RA 2 of 750 ±200 feet.

(6) Perform radio altimeter flag test.

(a) Press MENU.
(b) Press down arrow until DISCRETE I/O? displays on the panel.
(c) Press YES.
(d) Press down arrow until the panel displays a reading for RA1 flag.
(e) Observe that the panel displays RA1 flag = set.
(f) Open Radio Altimeter 1 circuit breaker. Observe that the YDC BITE panel displays RA1 flag = not set.

(g) Close Radio Altimeter 1 circuit breaker.
(h) AIRPLANES WITH SECOND RADIO ALTIMETER;
Do these steps:
1) Open Radio Altimeter 2 circuit breaker.
2) Press down arrow. Observe that the YDC BITE panel displays RA2 flag = not set.
3) Close Radio Altimeter 2 circuit breaker. Observe that the YDC BITE panel displays RA2 flag = set.
4) Turn off radio altimeter test set.

(7) Perform airspeed test.
(a) Press MENU. Observe that DISCRETE I/O? is displayed on the panel. Press up arrow and observe that ANALOG I/O? is displayed on the panel.
(b) Press YES.
(c) Press down arrow until the YDC BITE panel displays reading for CAS qpot.
(d) Using air data test set, simulate airspeed of 130 KTS to Air Data Computer #1 (Ref 34-11-0, Adjustment/Test). Observe that the panel displays CAS qpot between -6.4 V and -8.4 V.
(e) Simulate airspeed of 280 KTS to Air Data Computer #1. Observe that the panel displays CAS qpot between -3.4 and -4.8.
(f) Turn off air data test set.

(8) Shutdown BITE
(a) Press ON/OFF on the YDC BITE panel. Observe that TURN OFF DISPLAY? displays on the panel.
(b) Press YES. Observe that DISPLAY OFF temporarily displays followed by the panel blanking. Observe the AUTOPILOT DISENGAGE WARNING lights are not illuminated.

NOTE: Disregard other Master Caution annunciator lights that illuminate.

F. Test Yaw Damper Interlocks
(1) Engage YAW DAMPER engage switch. Observe that YAW DAMPER warning light and FLT CONT annunciator light extinguish within 3 seconds.
(2) Press either MASTER CAUTION light. Open YAW DAMPER AC circuit breaker. Observe that YAW DAMPER engage switch moves to the OFF position after approximately 2 to 5 seconds. Observe that the YAW DAMPER warning light, the FLT CONT annunciator light, and both MASTER CAUTION lights illuminate.

(3) Press either MASTER CAUTION light. Close the YAW DAMPER AC circuit breaker. Wait 15 seconds and then engage the YAW DAMPER engage switch. Open the YAW DAMPER DC circuit breaker. Observe that YAW DAMPER engage switch moves to the OFF position and that the YAW DAMPER warning light, the FLT CONT annunciator light, and both MASTER CAUTION lights illuminate.

G. Test Yaw Damper Indicator Lights
   (1) With the yaw damper disengaged, observe that the YAW DAMPER warning light is illuminated on P5-3 Flight Controls Module, that the FLT CONT annunciator light is illuminated on the Left Master Caution Annunciator on the P7 Light Shield Panel, and that the Master Caution, left, and Master Caution, right, warning lights are illuminated on the P7 Light Shield Panel.
   (2) Place LIGHTS switch on center instrument panel first in DIM position, then in BRT position. Observe that YAW DAMPER warning light dims and then illuminates brightly.
   (3) Press either MASTER CAUTION light. Observe that both MASTER CAUTION lights and FLT CONT annunciator light go out, and that YAW DAMPER warning light remains illuminated.
   (4) Momentarily press Left Master Caution Annunciator. Observe that both MASTER CAUTION lights and FLT CONT annunciator light illuminate.

H. Hold Down Forces Test
   (1) Open Handle Levers on the YDC assembly.
   (2) Close Handle Lever to the point where the bottom corner of the lever just makes contact with the Fork Assembly.
   (3) Measure the distance between the tip of the lever and the position of the tip of the lever when fully closed. Distance should be between 0.35 and 0.45 inch.
   (4) Do these steps until the dimension is in tolerance:
      (a) If necessary, adjust the jamnut fork adjustment point:
         1) With the plug on the rear of the unit fully engaged and the handle lever on the front of the unit at a dimension within tolerance.
         2) Adjust forks by rotating to a position where they start to exert pressure on the locking lever.
      (b) Fully close the Handle Levers to properly seat the YDC in the rack.
      (c) Ensure proper adjustment by unlatching Handle Levers, closing to the point of contact with the Fork Assembly, and verifying the dimension is within tolerance.
   (5) Fully close Handle Levers.

4. <u>Restore Aircraft to Normal Configuration</u>
   A. Disengage yaw damper.
   B. Disconnect pressure source from pitot static system (Ref 34-12-00).

C. Determine whether there is further need for electrical power on aircraft; if not, remove external power (Ref 24-22-00).
D. Determine whether there is further need for hydraulic power on aircraft; if not, remove hydraulic power (Ref 29-15-00).

MACH TRIM SYSTEM - TROUBLE SHOOTING

1. General
   A. The mach trim system can be checked for proper operation by using the mach trim test switch on overhead panel P5 or the self-test switch on the front of the mach trim coupler. (Refer to Mach Trim System - Adjustment/Test.) Failure of the mach trim system, as indicated by the mach trim failure light on overhead panel P5 and the meter and light on the coupler, usually indicates the coupler is faulty. The mach trim failure light will also be illuminated if the coupler self-test switch is left in any position other than OFF. Replacement of the coupler will correct the majority of any faults which may occur. The mach trim system operates with the autopilot system either engaged or not engaged. If no elevator movement results when the mach trim system is operated and the pitch channel is engaged, the neutral shift sensor is faulty (assuming the pitch channel operates normally). Connectors may also be a source of trouble. If replacement of the coupler does not correct the trouble, connectors and wiring should be checked for damage.
   B. The air data computer supplies mach signals and a mach good logic voltage to the mach trim coupler. Failure of one of the mach potentiometers in the air data computer will cause loss of the mach good logic to the mach trim coupler, thereby causing the mach trim failure light to illuminate. If a failure originates in the air data computer, a module failure indicator will be operated on the front panel of the computer, thereby indicating a need for replacement of the air data computer. Checking for mach good logic can be done by looking for voltage at TP-8 or by positioning coupler self-test switch to 5. The meter should indicate in the red, if not, the air data computer is faulty. Pitot static inputs for the air data computer are not required for basic trouble shooting. Faults which may occur when pitot pressure is applied during a test of the system may be corrected by replacing the air data computer.
2. Mach Trim Components
   A. Mach Trim Coupler
      (1) The mach trim coupler is located on the El equipment rack in the electrical and electronics compartment. The mach trim coupler consists of a rack assembly and five plug-in card assemblies. The card assemblies are two amplifier-generator assemblies, the amplifier-
      monitor assembly, control amplifier assembly, and the power supply.
      (2) The rack assembly acts as a base for the mach trim coupler and supplies the necessary support and interconnection for the plug-in card assemblies. The unit houses self-test components and test points. The coupler contains automatic self-monitoring circuitry to electrically decouple the actuator and illuminate the warning light in the event of failure in electronic circuitry or actuator.

(3) The coupler receives signals which are computed as an elevator command signal. This signal is compared with the actuator feedback signal and then drives the actuator motor.

B. Mach Trim Actuator

(1) The mach trim actuator consists of a servomotor, a position sensor, limit stops, gearbox, brake, and actuator screw. The actuator has a 115 volt, 400 cps, single-phase motor which drives the gearbox;

(2) the gearbox then drives the actuator screw. The direction of screw movement is determined by the application of the proper voltage polarity to the motor control winding. The mach trim coupler provides 28 volts dc to the actuator brake mechanism to unlock the actuator and allow its movement when the actuator is engaged and operating. When power is applied to the motor and brake, actuator motion is limited to fully extended and fully retracted positions by both electrical and mechanical limit stops.

C. Mach Trim Controls

(1) The mach trim flap switch, located in the aft upper right portion of wheel well area, applies 28 volts dc to the mach trim coupler when the flaps are lowered. This signal prevents illumination of the mach trim warning light. See figure 1 for flap switch location.

(2) The mach trim cockpit test switch is a pushbutton switch on overhead panel P5. This switch provides a means of testing the mach trim system from the cockpit.

(3) The mach trim failure light is located adjacent to the mach trim test switch on the overhead panel. If the flaps are up, the light will illuminate whenever the cockpit test switch is depressed or whenever there is a malfunction in the mach trim system. The light will also illuminate if the coupler self-test switch is left in any position other than OFF.

3. <u>Operation</u>

A. Engagement

(1) The mach trim system engages automatically if airplane speed is in the mach tuck region, air data computer information is valid, system power is present, and computed actuator position and actual position agree.

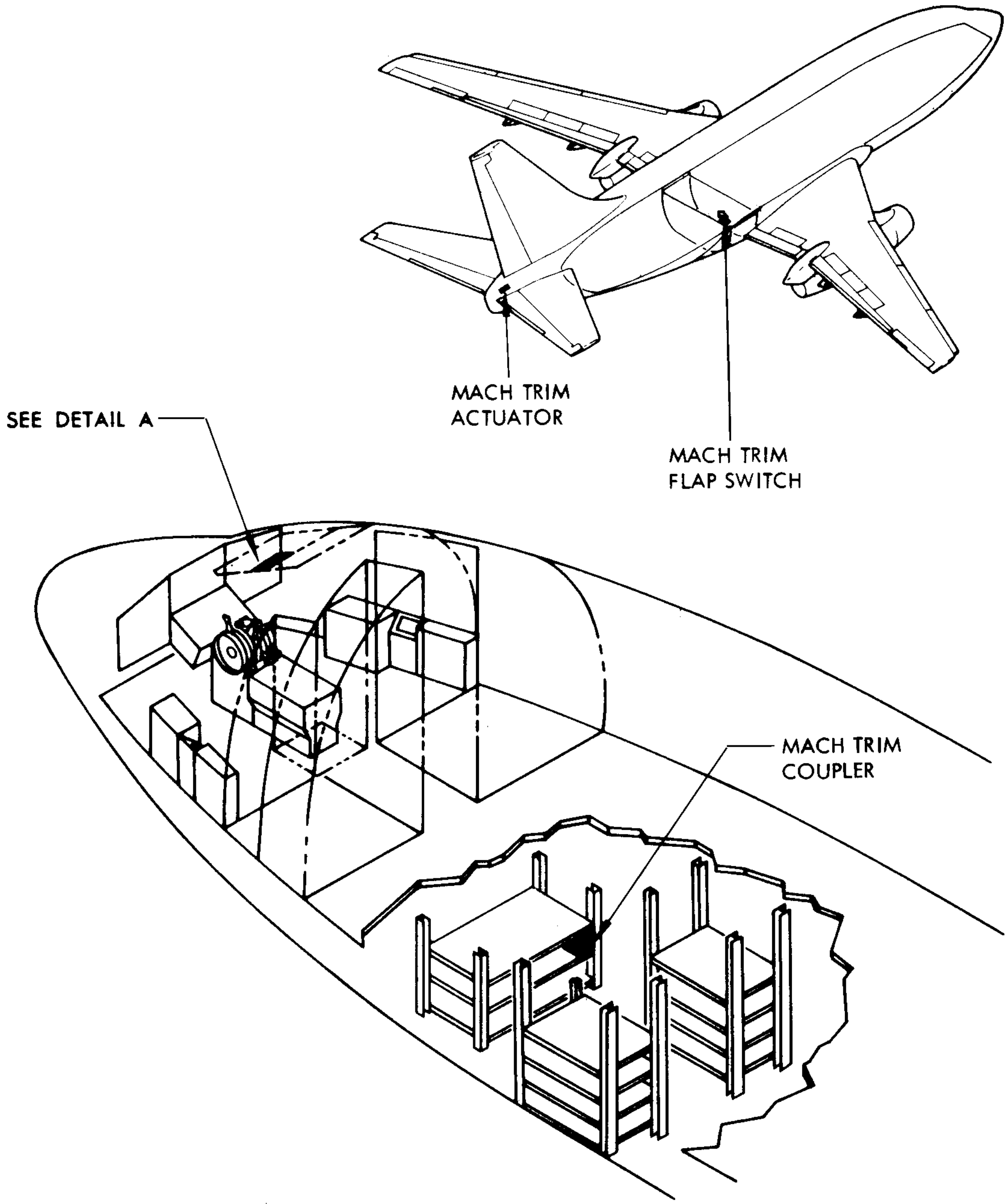
B. Disengagement

(1) The system automatically disengages when airspeed is below mach tuck region. When disengaged, the warning light remains out and an electromechanical brake locks the actuator to prevent changes in actuator length. Automatic disengagement with warning light illumination takes place when the following conditions exist.

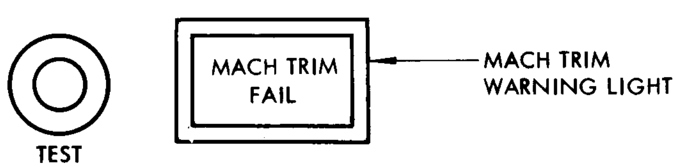
(a) The computed actuator position disagrees with the actual actuator position for more than 1 second.

(b) Air data computer information is invalid.

(c) Failure of electromechanical brake to actuate below mach tuck region.



MACH TRIM



DETAIL A

Mach Trim System Components Location
Figure 1

(d) E;ectrical power failure to mach trim system.

(2) When disengagement is caused by any of the above causes, the warning light will illuminate. The warning light will extinguish when the flaps are lowered.

C. Mach Trim Circuit Operation

(1) Signal Flow (See figure 2.)

(a) The air data computer (ADC) supplies parallel dc signals from two mach potentiometers. The two signals are applied to the mach trim coupler command channel and comparator channel respectively. These channels operate in a similar manner.

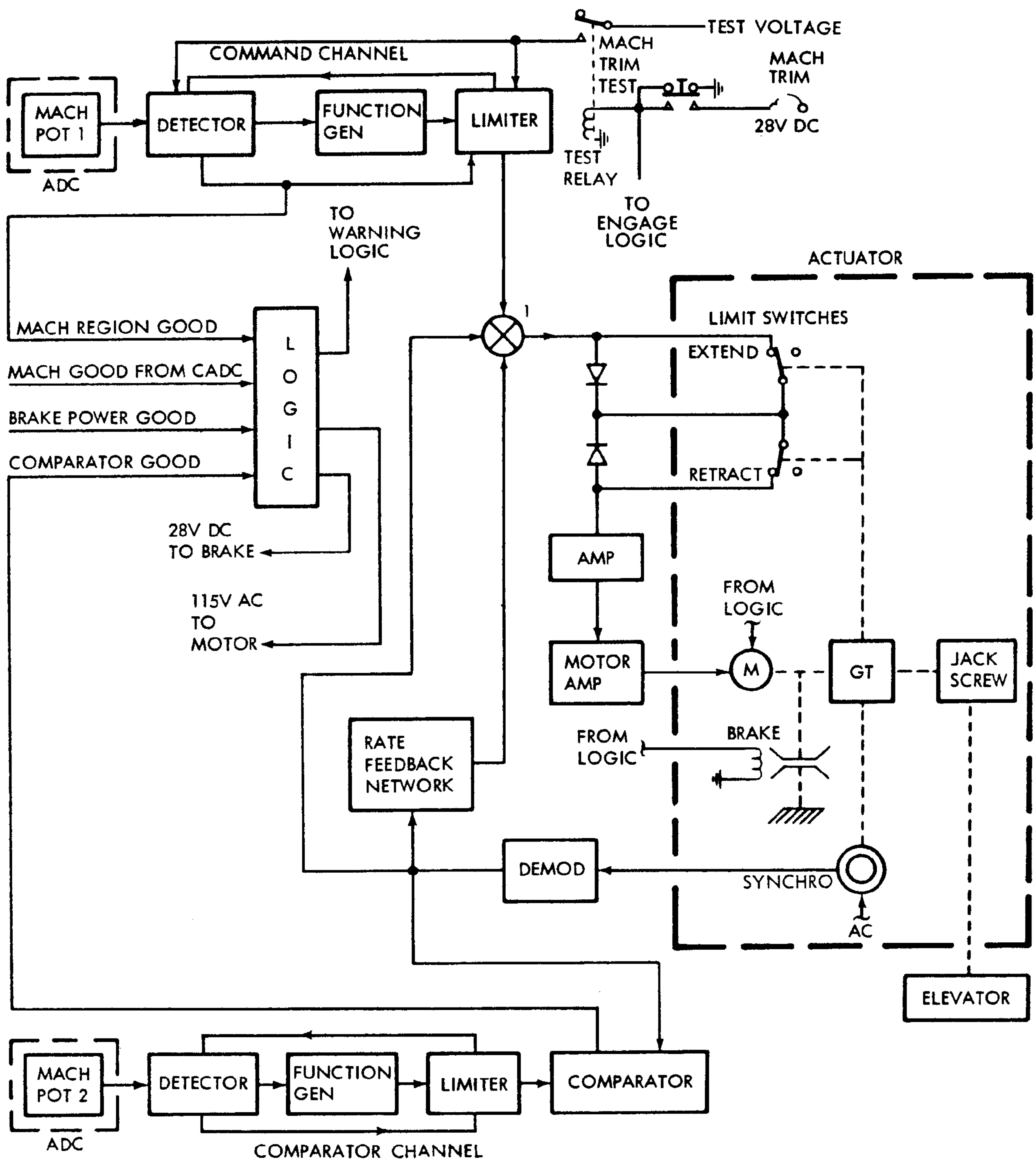
(b) The detector circuit receives signal voltage from the ADC mach potentiometer which is a function of mach number (speed). The received signal is monitored and compared for an amplitude that is in the mach tuck region. When in the mach region, one output from the detector is applied to a logic circuit for control of power to the motor and brake. A second output is applied to the limiter which removes a ground from the input signal to summing point 1. The third output is the mach signal which is fed to the function generator.

(c) The function generator amplifies and shapes the signal to provide proper gain and magnitude which is proportional to the signal level.

(d) The limiter amplifies, limits, and controls the input signal. One function of the limiter is to monitor the ADC mach potentiometer output for control of mach tuck upper voltage limits. The second function of the limiter is to ground the output signal when the output of the detector is below mach tuck region. The limiter also shapes and amplifies the output signal which is then applied to summing point 1.

(e) The output signal from summing point 1 goes through two limit switches in the actuator. When the actuator rod (jackscrew) is driven by the gear train to either fully extended or retracted position, the appropriate switch opens and stops the motor. Diodes then allow a signal of opposite polarity to drive the motor in the opposite direction. When below the mach tuck region, the actuator is in the fully extended position.

(f) The signal from the limit switches is applied to an amplifier. The signal is also modulated and chopped at 90 degree phase angle with respect to the actuator motor excitation. The resultant output has a pulse width which is a function of the magnitude of the dc mach signal input. The dc mach signal input polarity determines the phase of the pulse train.



Mach Trim Block Diagram
Figure 2

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(g) The motor amplifier has two split secondary transformers operating on the input signal to produce a switching signal. This signal applies 40 volts dc to the motor control winding for the pulse duration and also switches the 40 volt dc polarity at the pulse frequency.

(h) An ac position feedback from the synchro is routed to the demodulator. The demodulator chops the synchro signal at opposing phases and amplifies to supply a dc output. This output is used in the comparator, the rate feedback network, and at summing point 1. The rate feedback network acts as a damper to the mach signal by providing an input to summing point 1. The composite signal from summing point 1 is used to drive the actuator motor.

(i) The comparator circuit operates from information received from the demodulator and the limiter. This circuit supplies a 12 volts dc signal to a logic circuit when the difference between its two input signals is less than a predetermined level.

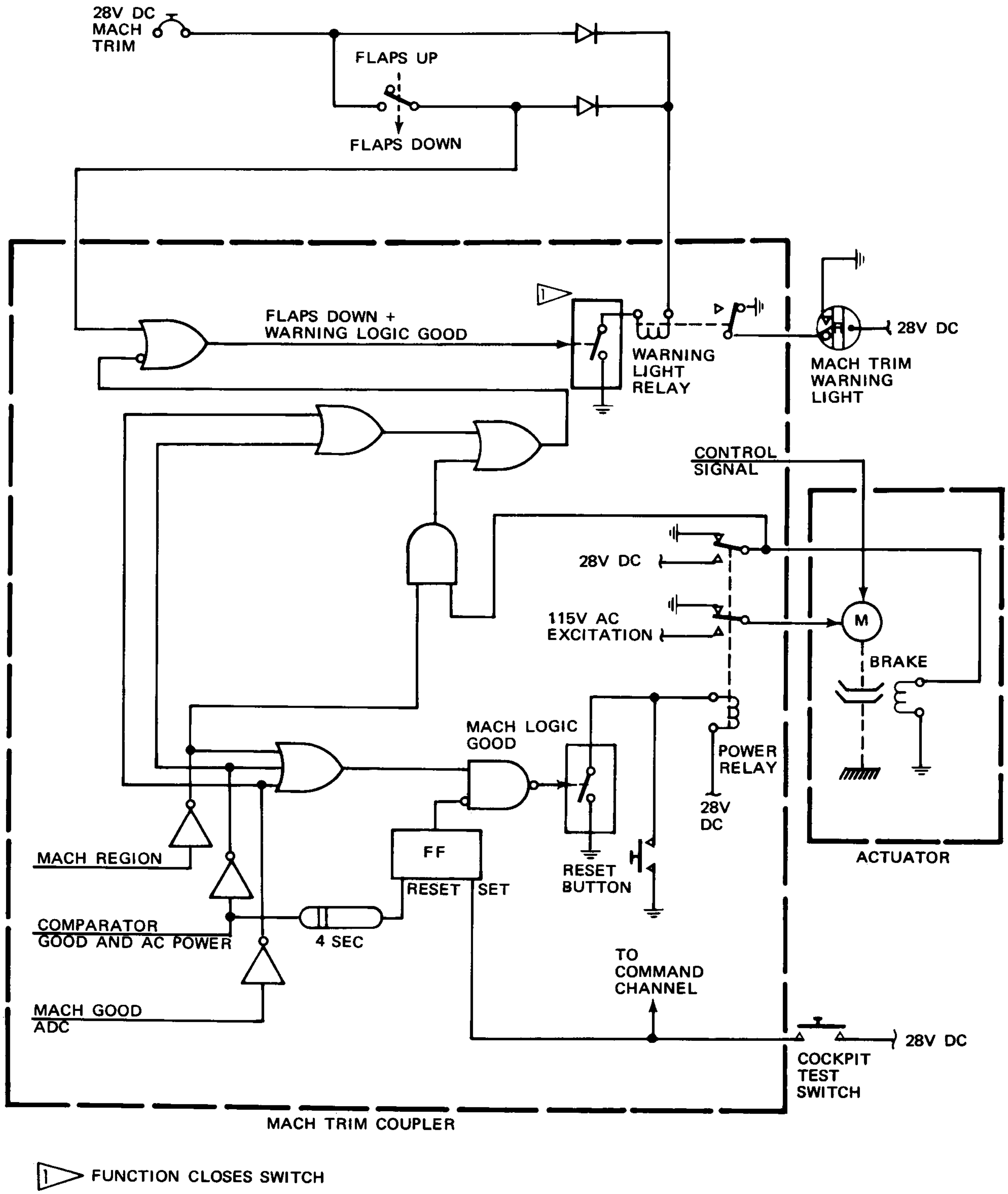
(j) The logic circuit requires mach region good input, mach good input from ADC, and comparator good input in order to complete the logic necessary to provide voltage to the brake and motor. The logic circuit monitors for 28 volts dc as brake power good from the brake circuit. With this input and the other inputs good, the logic circuit provides a warning logic good output to the warning circuit.

(2) Logic and Interlocks

(a) Power to the actuator motor and brake is provided through the power relay contacts in the coupler. (See figure 3.) The relay is operated by the following logic requirements.

1) The 28 volt dc and 115 volt ac mach trim circuit breakers are closed.
2) Airplane is operating in mach region as monitored through mach trim coupler command channel.
3) Comparator channel in coupler provides correct voltage and ac power is correct.
4) Mach signal is good as provided by the air data computer.

(b) If any one of the above requirements fail, the requirements for mach logic good are no longer satisfied. This opens the solid state switch and breaks the ground to thee e power relay. As a result, the brake clamps the actuator from further movement and the excitation voltage is removed from the actuator motor.



Mach Trim Engage and Warning Logic Circuit
Figure 3

(c) Power is present at the warning light relay when flaps are up, and circuit breakers are closed. The mach trim warning light remains off when the following requirements provided warning logic good to complete the logic circuit to ground.
   1) Power, that is applied to release the brake, is also applied to a logic circuit as an input to the warning light logic.
   2) The inputs that satisfy mach logic good for the power relay are the same requirements to satisfy warning logic good. Failure of any of these inputs will cause the mach trim warning light to illuminate.
   3) When flaps are lowered, the warning light is extinguished.

(3) Mach Trim Cockpit Test
   (a) The cockpit test circuit is activated by depressing and holding the pushbutton switch on the forward overhead panel which is labeled MACH TRIM TEST. (See figure 1.) After 3 to 4 seconds, the MACH TRIM FAIL light illuminates which indicates the actuator has retracted. (The actuator is normally extended until the mach tuck region is entered.) The MACH TRIM FAIL light remains illuminated 7 to 8 seconds after releasing the test switch, until the actuator is repositioned to its original reference position. Extinguishing of the MACH TRIM FAIL light indicates satisfactory completion of the mach trim cockpit test. The test circuit operates as follows:
      1) By depressing the MACH TRIM TEST switch, 28 volts dc is applied to a test relay which provides a test voltage to the command channel (figure 2) and to the logic circuit (figure 3).
      2) The test voltage applied to the command channel causes the command channel to provide a signal which is different from the one provided by the comparator channel. The test signal is applied to the actuator in the same manner as described in paragraph (l). As the actuator changes length, an error signal is induced in the position feedback synchro. The comparator uses the error signal to compare and detect differences from the air data computer input. Differences cause the comparator to apply comparator not good to the warning logic circuit. This circuit removes the voltage from the warning light relay which allows the MACH TRIM FAIL light to illuminate.
      3) When the MACH TRIM TEST switch is released, voltage is held by the engage logic circuit to allow the actuator to return to its original position. After 7 or 8 seconds, the warning logic and engage logic receive logic good voltage. This voltage causes the MACH TRIM FAIL light to extinguish and causes removal of power from the actuator.

MACH TRIM SYSTEM - TROUBLESHOOTING

1. General
   A. The mach trim system can be checked for proper operation by using the mach trim test switch on overhead panel P5 or the self-test switch on the front of the mach trim coupler (refer to Mach Trim System - Adjustment/Test). Failure of the mach trim system, as indicated by the mach trim failure light on overhead panel P5 and the meter and light on the coupler, usually indicates the coupler is faulty. The mach trim failure light will also be illuminated if the coupler self-test switch is left in any position other than OFF. Replacement of the coupler will correct the majority of any faults which may occur. The mach trim system operates with the autopilot system either engaged or not engaged. If no elevator movement results when the mach trim system is operated and the pitch channel is engaged, the neutral shift sensor is faulty (assuming the pitch channel operates normally). Connectors may also be a source of trouble. If replacement of the coupler does not correct the trouble, connectors and wiring should be checked for damage.
   B. The air data computer supplies mach signals and a mach good logic voltage to the mach trim coupler. Failure of one of the mach potentiometers in the air data computer will cause loss of the mach good logic to the mach trim coupler, thereby causing the mach trim failure light to illuminate. If a failure originates in the air data computer, a module failure indicator will be operated on the front panel of the computer, thereby indicating a need for replacement of the air data computer. Checking for mach good logic can be done by looking for voltage at TP-8 or by positioning coupler self-test switch to 5. The meter should indicate in the red, if not, the air data computer is faulty. Pitot static inputs for the air data computer are not required for basic trouble shooting. Faults which may occur when pitot pressure is applied during a test of the system may be corrected by replacing the air data computer.
   C. Test points on the front of the mach trim coupler may be useful in troubleshooting the system. Expected voltages for some of the test points are given in the adjustment/test procedures as part of system test. Most of the indications which are provided by the test points can also be determined by observing how the system operates. The following list indicates each test point and where it is connected in the circuit:

Test Points	Description
TP-1	Flaps Down Logic
TP-2	Mach Pot No. 1
TP-3	Mach Pot No. 2
TP-4	Signal Ground
TP-5	26 Volts AC, In Phase
TP-6	Linear Synchro Feedback
TP-7	Comparator Output
TP-8	Mach Good Logic
TP-9	Brake Release Logic
TP-10 thru TP-12	Spare

D. Before troubleshooting, check that electrical power is applied to load control center P6, hydraulic systems A and B are operating and that elevator power control units operate satisfactorily. The mach trim, air data computer, indicator lights and master dimming and test circuit breakers must be closed and the autopilot system disengaged. Flaps must be up. The mach trim coupler self-test switch must be in the OFF position.

2. Mach Trim System Trouble Shooting Chart

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
Mach trim failure light illuminated with flaps up	Coupler, Air Data Computer, actuator or connectors	Check coupler meter, if not in red, and coupler light illuminated, press reset button 10 seconds. If meter in red but light still illuminated, check self-test 5. If meer reads in red, coupler is faulty. If not in red, replace air data computer. With suspected components replaced, check coupler meter, if meer does not read in red with self-test switch off and coupler light is still illuminated, conduct self-test series. If self-test 1 or more fails, check wiring and connectors. Wiring and connectors ok, acutator is faulty.	Replace coupler. Replace air data computer. Repair connectors and wiring. Replace actuator (AMM 22-21-11, Mach Trim Actuator).

TROUBLE	PROBABLE CAUSE	ISOLATION PROCEDURE	REMEDY
System fails cockpit test	Coupler or test switch	Press cockpit test switch and hold for 15 seconds. If mach trim failure light does not illuminate during test or after switch is released, coupler is faulty. faulty. If light illuminates and remains on after switch is released, check per mach trim failure light illuminated with flaps up. If replacing coupler does not correct fault, check wiring and mach trim test switch.	Replace coupler. Replace mach trim test switch.
Mach trim failure light illuminates with falps down	Flap switch or coupler	Check for 28 volts dc at TP-1. If voltage, coupler is probably faulty. If no voltage, flap switch is faulty.	Replace coupler. Adjust mach trim flap switch (AMM 22-21-31, Mach Trip Flap Switch).

MACH TRIM SYSTEM - ADJUSTMENT/TEST

1. Mach Trim System Test
   A. General
      (1) This test is to check the mach trim system. It is assumed that associated systems are operating properly.
   B. Equipment and Materials
      (1) VTVM - Hewlett Packard, Model 410C
      (2) Portable pressure equipment to simulate 0 to 35,000 feet altitude and 0 to 450 knots airspeed, ERA 82305, Pitot Static Test Set; adapters and hoses, Nav Aids PSS22300-4-6-6 and F72823-40
      (3) Scale (for measuring elevator travel) 0 to 2 feet, graduated in inches, tenths, and hundredths
      (4) Dry air pressure source, 0 to 5 psi, with gage, fittings, hose, etc., for connecting to feel system pitot tubes, F72928-53(preferred), -52(alternate)
   C. Prepare to Test Mach Trim System
      (1) Verify control surfaces are free to move and flaps are up.
      (2) Provide ground power to the airplane and energize busses on load control centers P6 and P18.
      (3) Close mach trim, master dimming, and dim test circuit breakers.
      (4) Pressurize hydraulic Systems A and B (AMM 29-11-0/201 and AMM 29-12-0/201).
      (5) Connect pitot static source (AMM Chapter 34, Pitot-Static System).

         NOTE: Continuously energize air data computers for at least 30 minutes prior to start of test (AMM Chapter 34, Air Data Pressure Instruments). Application of pitot-static pressure required only for elevator displacement checks, par. (5) and on.

      (6) Remove the tailcone access panel 3802 (AMM Chapter 12, Access Doors and Panels).
   D. Test Mach Trim System
      (1) Test Indicator Lights
         (a) Open mach trim ac and dc circuit breakers.
         (b) Observe mach trim fail indicator light, located on P5 panel, the two MASTER CAUTION lights and FLT CONT master caution annunciator lights are illuminated.
         (c) With light switch positioned to DIM, observe mach trim fail indicator light is dim.
         (d) With light switch positioned to BRT, observe mach trim fail indicator light returns to bright.
         (e) With either master caution light depressed, observe both MASTER CAUTION light and FLT CONT master caution annunciator light extinguishes and observe mach trim light remains illuminated.

(f) Momentarily depress FLT CONT master caution annunciator light and observe both MASTER CAUTION lights and FLT CONT master caution annunciator light illuminates.

(2) Test Circuit Breakers, Warning Lights, and Mach Trim Cockpit Test Switch

(a) Close air data computer-1 ac and 26 volt ac circuit breakers on load control center P18 and mach trim computer circuit breakers ac and dc on load control center P6. Observe mach trim fail light is off.

(b) Open mach trim dc circuit breaker. Observe mach trim fail light illuminates.

(c) Close mach trim dc circuit breaker. Observe mach trim fail light extinguishes.

(d) Open mach trim ac circuit breaker. Observe mach trim fail light illuminates.

(e) Lower flaps to 1 unit. Observe mach trim fail light extinguishes.

(f) Raise the flaps. Observe the mach trim fail light illuminates.

(g) Close mach trim ac circuit breaker. Observe mach trim fail light extinguishes.

(h) Open air data computer(-1) 115-volt ac circuit breaker. Observe mach trim fail light illuminates.

(i) Close air data computer(-1) 115-volt ac circuit breaker, and observe mach trim fail light extinguishes.

(j) Open air data computer(-1) 26-volt ac circuit breaker. Observe mach trim fail light illuminates.

(k) Close air data computer(-1) 26-volt ac circuit breaker. Observe mach trim fail light extinguishes.

(l) Move self-test switch, located on mach trim coupler, to position 1 and depress the reset button for 10 seconds. Observe mach trim fail light illuminates and coupler light remains off.

(m) Return self-test switch to OFF. Observe mach trim coupler light and mach trim fail light illuminates.

(n) Press reset button for 10 seconds. Observe coupler light and mach trim fail light extinguishes.

(o) Depress and hold for at least 15 seconds the mach trim cockpit test switch which is on the overhead panel. Observe the mach trim fail light illuminates 4 seconds after depressing test switch and captain's column moves slightly aft.

NOTE: Mach trim fail light may flash on momentarily, and master caution light may come on when test switch is depressed. Observe steady illumination of mach trim fail light 4 seconds after test switch operation.

(p) Release the test switch. Observe the column moves slightly forward and the light extinguishes after 10 seconds.

(3) Test Self-Test Annunciation

(a) Operate mach trim self-test switch to each of five positions and return to off. Observe mach trim warning light illuminates for each position.

(4) Test Mach Trim Self-Test

Step	Action	Observe
(a)	Self-test switch in OFF position.	Self-test meter reads in the RED range, and coupler light is extinguished.
(b)	Place self-test switch to position 1.	Self-test meter reads in the RED range and coupler light illuminates.
(c)	Depress reset button for 10 seconds.	Coupler light extinguishes.
(d)	Place self-test switch to position 2. Depress reset button for 10 seconds.	Self-test meter reads in RED range and coupler light illuminates 4 to 7 seconds after switching from self-test position 1.
(e)	Place self-test switch to position 3.	Self-test meter reads in RED range and coupler light extinguishes.

Step	Action	Observe
(f)	Place self-test switch to position 4.	Self-test meter reads in RED range. Coupler light illuminates for approximately 5 seconds and then extinguishes.
(g)	Place self-test switch to position 5.	Self-test meter reads in RED range and coupler light illuminates.
(h)	Place self-test switch to OFF position. Depress reset button for 10 seconds.	Self-test meter reads in RED range and coupler light extinguishes.

NOTE: Verify that self-test meter is in RED range after reset to ensure actuator retraction.

(5) Apply 5 ±0.1 psig dynamic pressure to both pitot tubes on fin with static port at ambient.

CAUTION: DO NOT RAISE PRESSURE TO PITOT TUBES ON FIN ABOVE 6 PSIG AT ANY TIME.

(6) Test Below Mach Region
- (a) Position stabilizer trim to 3 units.
- (b) Jiggle the elevator right-hand quadrant until a detent position is felt (this is to ensure the system is centered).
- (c) Measure elevator trailing edge position with reference to index plate and record. Observe that the measurement is 0.0 ±0.l inch.
- (d) Apply conditions of step 1) of Fig. 501 to portable pressure equipment to simulate 0.708 mach. Observe mach trim coupler light and mach trim warning light is extinguished and self-test meter reads in the red range.

  NOTE: Increase the differential pressure slightly then drop back to specified pressure for each step in Fig. 501.

(e) Measure and observe voltage between TP-4 (signal ground) and TP-88 of 22 to 32 volts dc and between TP-4 and TP-9 of less than 1 volt dc.

(f) Measure right elevator position and note.

(7) Test Performance Curve Start Point

(a) Apply conditions of step 2) of Fig. 501 to portable pressure equipment to simulate 0.723 mach. Observe mach trim coupler light and mach trim warning light are extinguished.

(b) With a scale, measure and record position of right elevator. Observe difference of measurements from step (5)(f) is -0.02 inch to +0.22 inch maximum.

(c) Measure and observe voltage from TP-4 to TP-8 of 22 to 32 volts dc and between TP-4 and TP-9 of 28 ±1 volts dc.

(8) Test Performance Curve First Break Point

(a) Apply conditions of step 3) of Fig. 501 to portable pressure equipment to simulate 0.78 mach. Observe mach trim coupler light and mach trim warning light remain extinguished.

(b) Jiggle the elevator for right-hand quadrant until a detent position is felt.

(c) With a scale, measure and record position of right elevator. Observe difference of measurements from step (5)(f) is 0.84 inch minimum to 1.06 inch maximum. Nominal 0.95 inch.

Step No.	Static In. Hg.	Diff. Press. In. Hg.	Pitot Press In. Hg. ABS	Mach No.	Calib. A/S	Altitude Feet
1)	11.104	4.408	15.512	0.708	296k	25,000
2)	11.104	4.630	15.734	0.723	303k	25,000
3)	11.104	5.474	16.578	0.780	328k	25,000
4)	11.104	6.058	17.162	0.815	344k	25,000
5)	11.104	7.630	18.734	0.900	383k	25,000

(9) Test Performance Curve Second Break Point
- (a) Apply conditions of step 4) of Fig. 501 to portable pressure equipment to simulate 0.815 mach. Observe mach trim coupler light and mach trim warning light remain extinguished.
- (b) Jiggle the right-hand quadrant until a detent position is felt.
- (c) With a scale, measure and record position of right elevator. Observe difference of measurement from step (5)(f) is 1.72 inches minimum to 1.94 inches maximum. Nominal 1.83 inches.

(10) Test Performance Curve Limit
- (a) Apply conditions of step 5) of Fig. 501 to portable pressure equipment to simulate 0.90 mach. Observe mach trim coupler light and mach trim warning light remain extinguished and self-test meter reads far to the right.
- (b) Jiggle right-hand elevator quadrant until a detent position is felt.
- (c) With a scale, measure and record position of right elevator. Observe difference of measurements from step (5)(f) is 1.72 inches minimum to 1.94 inches maximum. Nominal 1.83 inches.

E. Restore Airplane to Normal Configuration
- (1) Disconnect pitot-static source (AMM Chapter 34, Pitot-Static System).
- (2) Replace access panel and remove air supply from pitot tubes.
- (3) If no longer required, remove electrical power from the airplane.

MACH TRIM ACTUATOR - REMOVAL/INSTALLATION

1. General
   A. The mach trim actuator is mounted on top of the elevator feel and centering unit (Fig. 401). If the actuator is replaced because of malfunction or to perform routine maintenance, the elevator does not have to be rerigged. However, if the elevator is being rerigged and the actuator is removed, personnel must ensure the actuator is extended and properly nulled out before it is reinstalled and the elevator rerigged. Ref Chapter 27, Elevator and Tab Control System.
2. Equipment and Materials
   A. Rigging Pin E-5 - 0.309/0.311-inch diameter, 3.7 ±0.25 inches long (MS20392-4)

      NOTE: Rigging pin is part of kit F70207-61 (preferred), F70207-3 (optional).

3. Prepare for Removal
   A. Remove elevator systems A and B hydraulic power (Ref 27-31-0, MP).
   B. Open and safety tag mach trim circuit breakers.
   C. Remove tail cone access panel 3802 (Ref Chapter 12, Access Doors and Panels).
   D. Install rigging pin E-5 in aft control quadrant (Fig. 402).
4. Remove Mach Trim Actuator (Fig. 401)
   A. Disconnect electrical connector from actuator.
   B. If installed, relieve tension on return springs. Unscrew eyebolt nuts on return springs each side of elevator feel and centering unit, but do not remove.
   C. Apply an upward pressure to the elevator surface or elevator feel and centering unit and remove double bolt, nuts and washers connecting actuator jackscrew to neutral shift crank.

      NOTE: Upward pressure required to rotate feel and centering unit may be relieved by moving control column aft until forward bolt can be easily removed.

   D. Remove two side mounting bolts retaining actuator and remove actuator.

5. Install Mach Trim Actuator (Fig. 401)
   A. Position actuator on top of elevator feel and centering unit and install two side mounting bolts, but do not tighten.

      CAUTION: ENSURE THAT ELEVATOR NEUTRAL SHIFT POSITION TRANSMITTER CRANK IS CORRECTLY ORIENTED IN THE UP DIRECTION.

      NOTE: The right side mounting bolt fits through two floating bearings on all installations. The left side mounting bolt may fit through a single bearing, a single bearing and a washer, or a double bearing. All bearings on the left side are sleeve retained in the centering unit. Refer to Details B and C of Fig. 401 for positioning. Be sure the floating bearings and the washer (if used) are correctly positioned before installing the side mounting bolts.

   B. Apply an upward pressure to elevator surface or elevator feel and centering unit and install double bolt connecting actuator jackscrew to neutral shift crank.

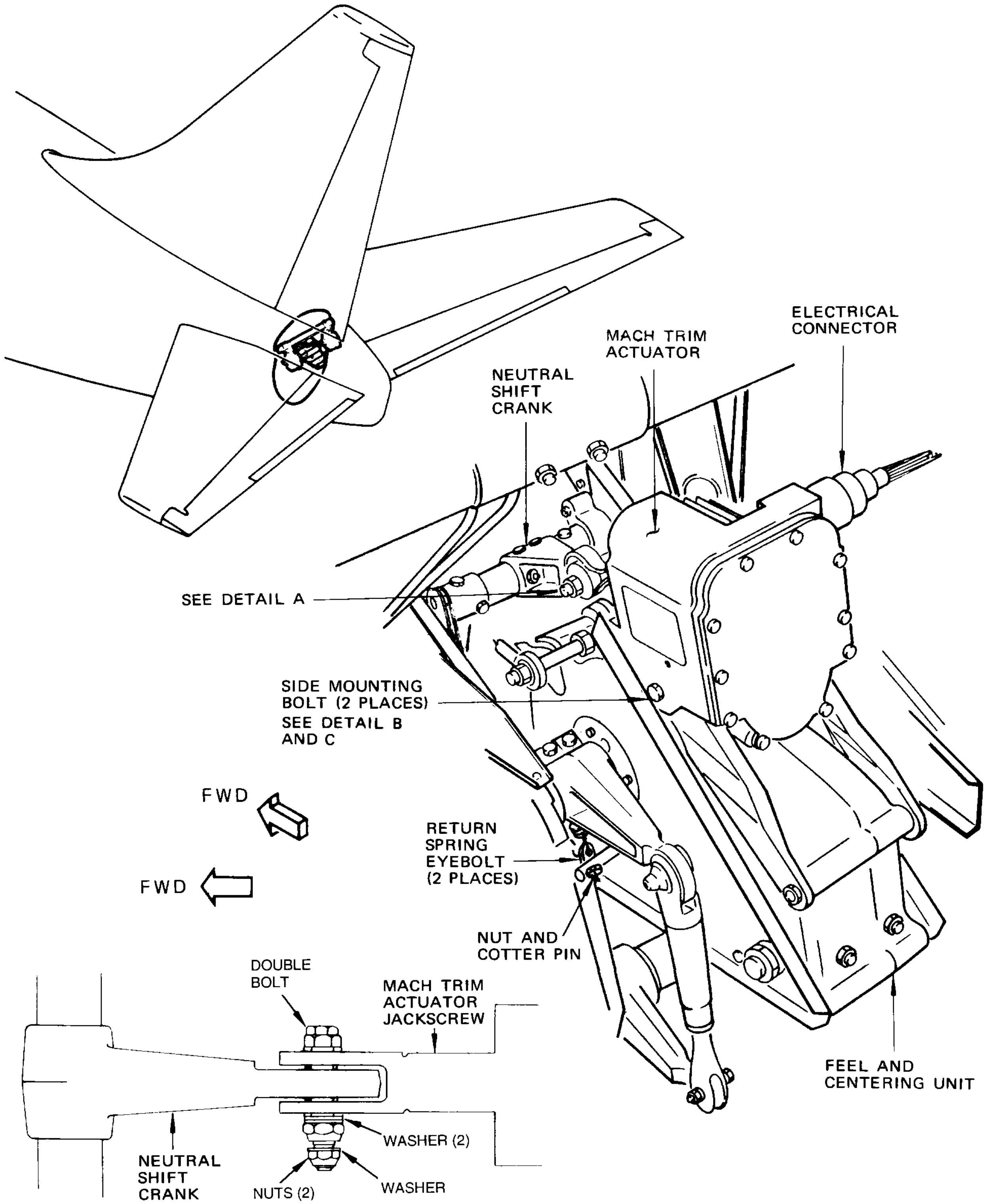
      NOTE: Upward pressure required to rotate feel and centering unit may be relieved by moving control column aft until forward bolt can be easily removed.

      If the Mach Trim Actuator does not satisfactorily align with the Neutral Shift Crank, adjust the Neutral Shift Crank per MM 27-31-91, R/I.

   C. Install washers and nuts on double bolt. Torque large nut to 85-100 pound-inches. Torque small nut to 30-35 pound-inches.

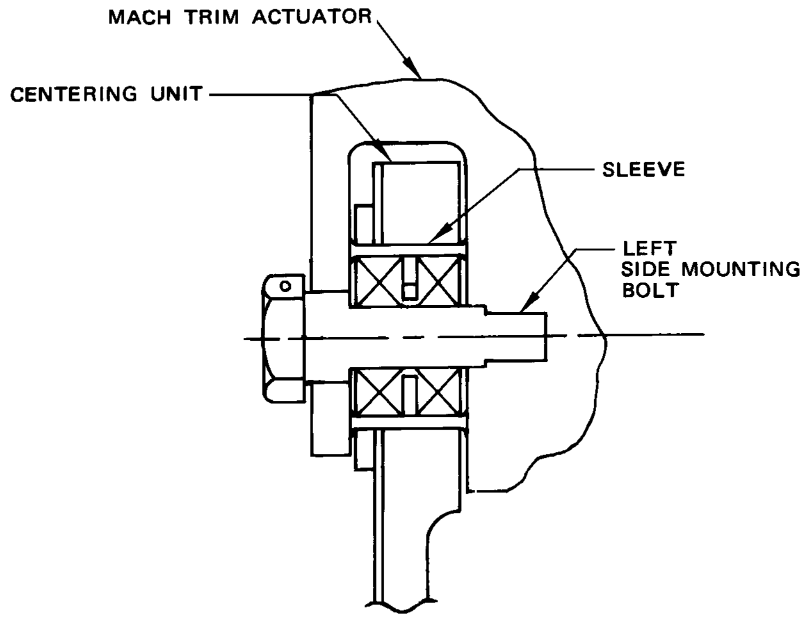
      CAUTION: TO AVOID DAMAGE, DO NOT RETORQUE LARGE NUT AFTER TORQUING SMALL NUT.

   D. Torque side mounting bolts to 20-25 pound-inches.
   E. If installed, torque eyebolt nuts to 30-40 pound-inches. Install cotter pins through each nut.
   F. Check neutral shift sensor crank alignment. Verify that crank is orientated in upward position (Fig. 402).
   G. Connect electrical connector to actuator.
   H. Remove rigging pin E-5.
   I. Test actuator. (Refer to Mach Trim Actuator - Adjustment/Test.)
6. Restore Airplane to Normal
   A. Replace access panels.
   B. Remove safety tags and close circuit breakers.

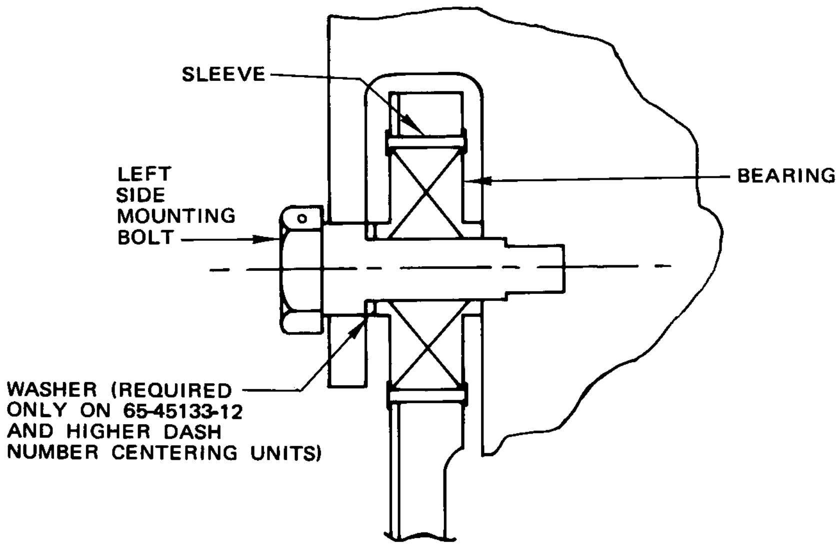


DETAIL A

Mach Trim Actuator Installation
Figure 401 (Sheet 1)

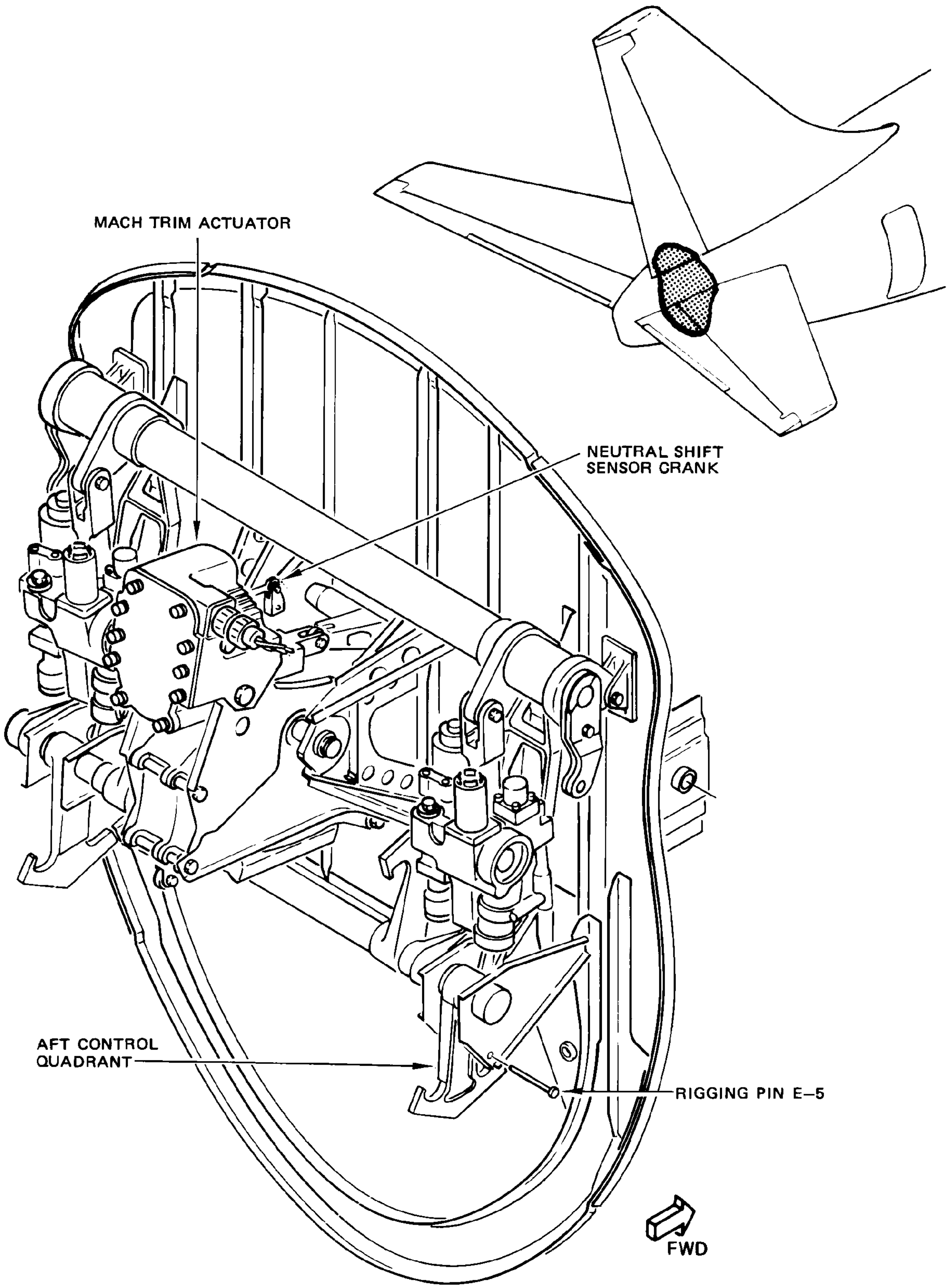


LEFT SIDE DOUBLE BEARING INSTALLATION
DETAIL B



LEFT SIDE SINGLE BEARING INSTALLATION
DETAIL C

Mach Trim Actuator Installation
Figure 401 (Sheet 2)



Rigging Pin E-5 Location
Figure 402

C. Restore airplane to normal hydraulic configuration (Ref 27-31-0, MP).
D. If no longer required, remove electrical power from airplane.

MACH TRIM ACTUATOR - ADJUSTMENT/TEST

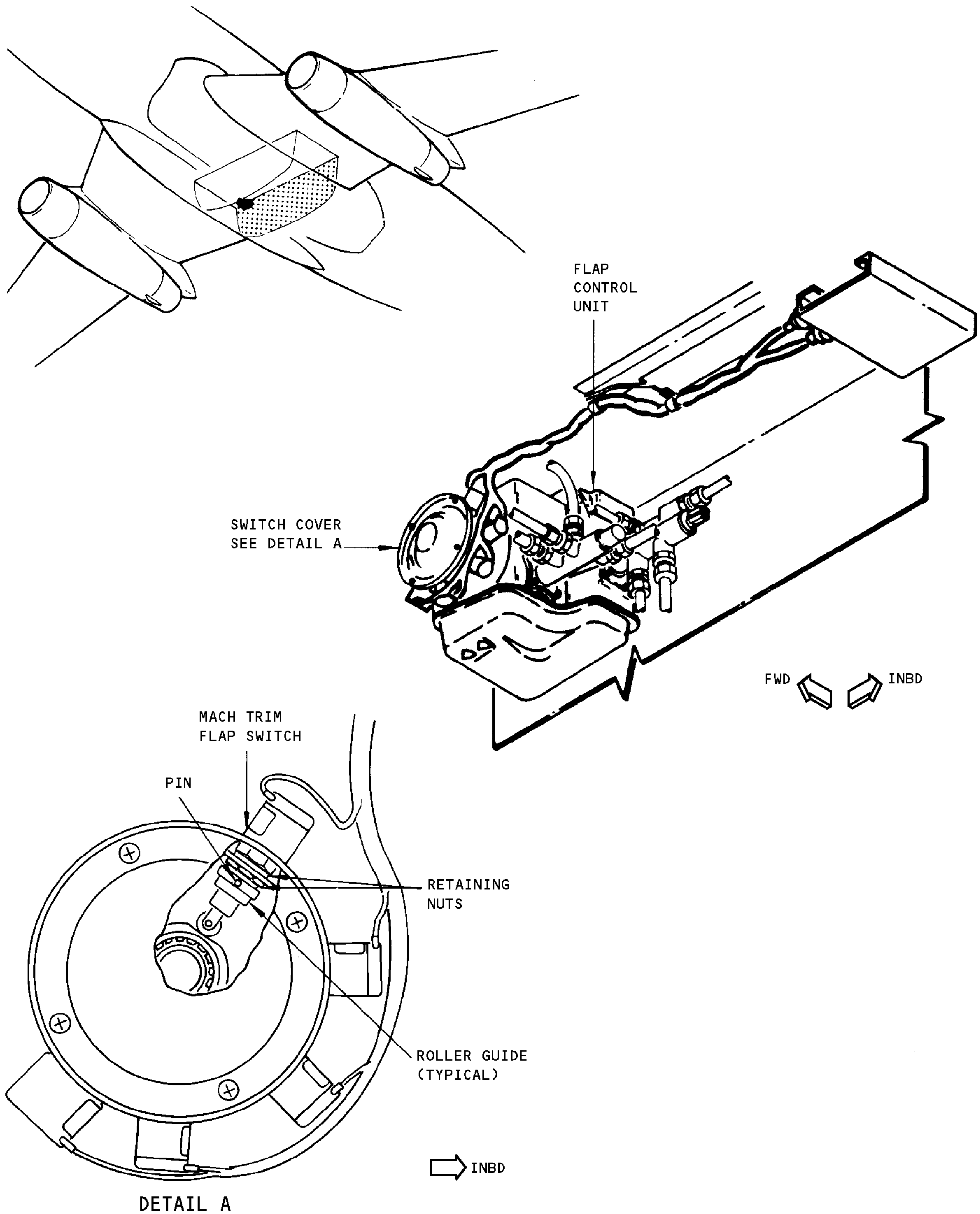
1. Mach Trim Actuator Test
    A. General
        (1) The following test assumes the elevator is rigged; if not, refer to Chapter 27, Elevator and Tab Control.
    B. Prepare to Test Mach Trim Actuator
        (1) Verify stabilizer and elevator surfaces are free to move.
        (2) Pressurize hydraulic systems A and B. Refer to 29-11-0, Hydraulic System A - Maintenance Practices and 29-12-0, Hydraulic System B - Maintenance Practices.
        (3) Ensure ground power is applied and check that mach trim and air data No. 1 circuit breakers on load control center P6 are closed.
        (4) Position stabilizer trim to three units.
    C. Test Mach Trim Actuator
        (1) Jiggle captain's control column to ensure control system is centered.
        (2) Check that mach trim coupler self-test switch is off and press reset button for 10 seconds. Note elevator trailing edge position with reference to the index plate.
        (3) Position self-test switch to 1 and press reset button for 10 seconds. Check that elevator trailing edge moved up 1.5 (+ 0.4) inches from position noted in step 2.
        (4) Position self-test switch to OFF and press reset button for 10 seconds. Check that elevator trailing edge returns to position noted in step 2.
        (5) If no longer required, remove electrical power from the airplane.

MACH TRIM COUPLER - REMOVAL/INSTALLATION

1. General
   A. The mach trim coupler is located in the E1 equipment rack in the electrical and electronics compartment. The coupler is installed with a knurled nut on swiveled bolts.
2. Remove Coupler
   A. Open the mach trim AC and DC circuit breakers on P6 panel.
   B. Remove the coupler (Ref 20-10-111, Electrical/Electronic Black Box - Maintenance Practices).
3. Install Coupler
   A. Install the coupler (Ref 20-10-111, Electrical/Electronic Black Box - Maintenance Practices).
   B. Close the mach trim AC and DC circuit breakers.
   C. Do the mach trim self-test (Ref 22-21-00, Mach Trim System - Adjustment/Test).

MACH TRIM FLAP SWITCH - REMOVAL/INSTALLATION

1. General
   A. Five switches are located on the flap control unit. The mach trim flap switch is mounted on the aft (inner) portion of the unit. (See figure 401.) The five switch leads make a single wire bundle and must be separated during the mach trim switch removal and retied when the new switch is installed.
2. Remove Mach Trim Flap Switch
   A. Remove switch cover from flap control unit. (See figure 401.)
   B. Separate wire bundle connector from receptacle.
   C. Remove wire bundle support clamps and bundle ties.
   D. Separate mach trim flap switch leads from bundle and disconnect from connector.
   E. Extract pin to release roller guide.
   F. Remove retaining nut from limit switch and remove switch from control unit.
   G. Replace nut, roller guide, and pin on switch.
3. Install Mach Trim Flap Switch
   A. Remove pin, roller guide, and one retaining nut from switch. (See figure 401.)
   B. Position switch in flap control unit and install retaining nut and roller guide. Install pin to secure roller guide.
   C. Install switch leads in connector and connect to receptacle.
   D. Replace wire bundle ties.
   E. Install wire bundle support clamps.
   F. Adjust and test mach trim switch. See Mach Trim Flap Switch - Adjustment/Test.
   G. Ensure that switch retaining nuts are lockwired.
   H. Install switch cover on flap control unit.



Mach Trim Flap Switch Installation
Figure 401

MACH TRIM FLAP SWITCH - ADJUSTMENT/TEST

1. Flap Mach Trim Switch Adjustment
   A. Prepare Flap Mach Trim Switch for Adjustment
      (1) Connect electrical power and energize load control center P6.
      (2) Position flap control lever to FLAP DOWN detent.

      NOTE: Flap control lever is moved to FLAP DOWN position as a standard procedure to minimize load on flap hydraulic motor when hydraulic system A is pressurized.

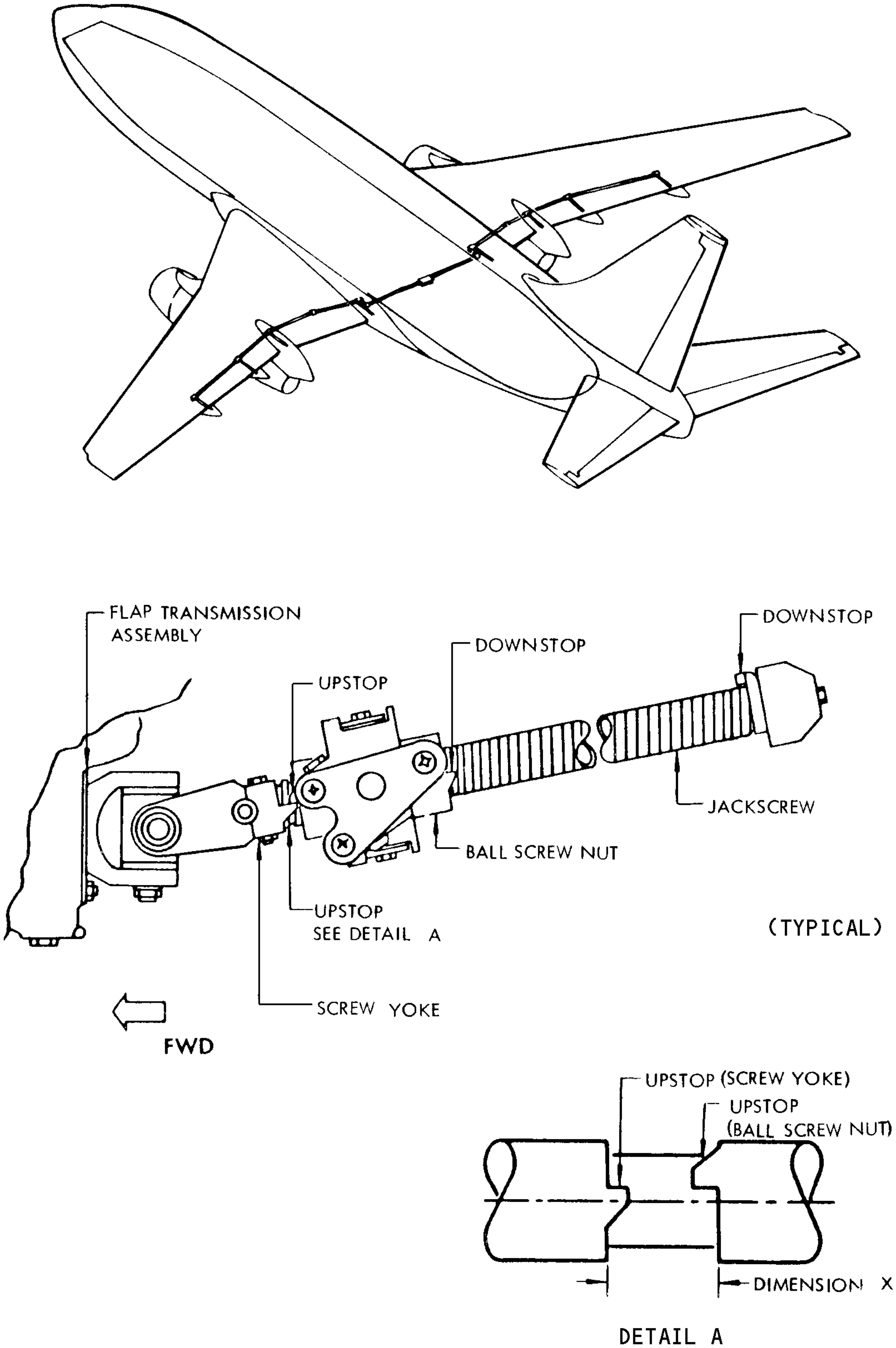
      (3) On circuit breaker panel P6-2:
         (a) Close TE ALT FLAP DRIVE MOTOR circuit breaker.
         (b) Close FLAP VALVES circuit breaker.
   B. Adjust Flap Mach Trim Switch
      (1) Position alternate flap switch to ARM. Verify that trailing edge flaps are extended 1 degree or more. If flaps are fully retracted, actuate alternate flap switch to DOWN, and hold until flaps are extended to 1 degree position.
      (2) Actuate alternate flap switch to UP and hold. When flap drive stops at full up position, release flap switch. Verify that No. 4 ball screw nut is 1-3/4 to 2-1/4 turns from up stop on ball screw yoke (Fig. 501). If this dimension is not obtained, adjust flap up limit switch per 27-51-291 prior to adjusting flap mach trim switch.

      NOTE: Do not disconnect torque tubes to position flaps.

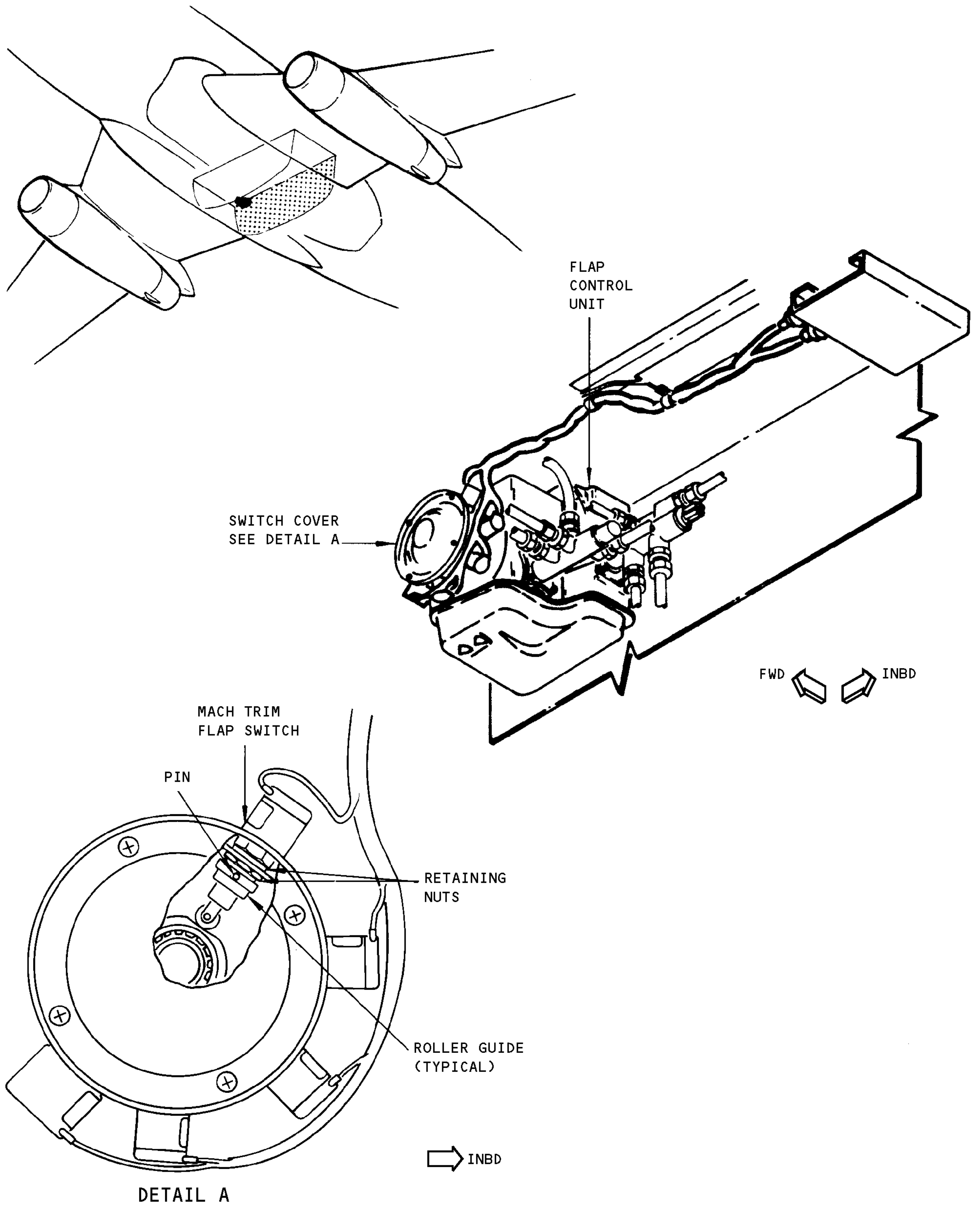
      (3) Remove switch cover from flap control unit (Fig. 502).
      (4) Measure and record dimension X (Fig. 501). Actuate alternate flap switch to DOWN for several revolutions of jackscrew. Actuate alternate flap switch to UP, release switch to stop flaps at dimension X plus 0.10 ±0.05 inch. This dimension must be obtained by moving flaps in the up direction.
      (5) Adjust retaining nuts on flap mach trim switch to position switch body as close as possible toward actuating cam, then adjust retaining nuts to move switch away from actuating cam until switch actuates (audible click).

      NOTE: Care must be taken to preposition switch so that actuation occurs when switch body is moved away from cam.

      (6) Tighten retaining nuts.
      (7) Lockwire switch retaining nuts.
      (8) Replace switch cover on flap control unit.



Flap Jackscrew Stops
Figure 501



Mach Trim Flap Switch Installation
Figure 502

2. <u>Mach Trim Flap Switch Test</u>

A. Prepare Mach Trim Flap Switch for Test

(1) Check that control surfaces are free to travel and flaps are in the up position.

(2) Provide power to the airplane and energize busses on load control center P6.

(3) Close mach trim circuit breakers on load control center P6.

(4) Pressurize hydraulic system A and B (Ref 29-11-0 MP and 29-12-0 MP).

<u>WARNING</u>: MAKE SURE ALL PERSONNEL AND OBSTRUCTIONS ARE CLEAR OF AILERON, ELEVATOR, OUTBOARD SPOILERS AND RUDDER SYSTEMS. SURFACES AND CONTROLS MAY MOVE ABRUPTLY WHEN HYDRAULIC POWER IS APPLIED.

B. Test Mach Trim Flap Switch

(1) Depress mach trim cockpit test switch and hold. Observe mach trim warning light is illuminated.

(2) Lower flaps to 1 degree. Observe mach trim warning light extinguishes.

(3) Raise the flaps. Observe the mach trim warning light illuminates.

(4) Release mach trim cockpit test switch.

(5) Determine whether there is any further need for electrical power on the airplane; if not, remove power.

AUTOTHROTTLE SYSTEM - DESCRIPTION AND OPERATION

1. General
   A. On airplanes LV-JMW thru LV-JMZ, LV-JND and LV-JNE, provisions are made for an auto throttle system installation. These provisions consist of two autothrottle clutch assemblies located in the aft ceiling of the electrical and electronics compartment.
   B. On ALL EXCEPT LV-JMW thru LV-JMZ, LV-JND and LV-JNE, partial provisions are made for an autothrottle system installation. These provisions consist of mountings for two autothrottle clutch assemblies located in the aft ceiling of the electrical and electronics compartment.
2. Auto Throttle Clutch Assemblies
   A. Each clutch assembly consists of an input quadrant, an output quadrant with a ball carrier, a ball race, clutch balls, a torque limiting clutch and a friction brake assembly. The quadrants, ball race, torque limiting clutch and friction brake are concentrically assembled. The clutch balls are retained by the ball carrier and engage the ball race.
   B. During manual throttle operation, throttle motion is transmitted to the input quadrant by the control cables. The unlock plate on the input quadrant pushed on one of the two clutch balls (depending on throttle direction), which lifts the center ball, releasing the pressure on the other clutch ball and allowing the input and output quadrants to rotate. The friction brake assembly smoothes throttle action. During autothrottle operation, throttle commands are applied through a gearbox to the ball race. Rotation of the ball race forces one of the clutch balls to wedge between the ball race and the ball carrier to drive the input and output quadrants.

AUTOTHROTTLE CLUTCH ASSEMBLY - REMOVAL/INSTALLATION

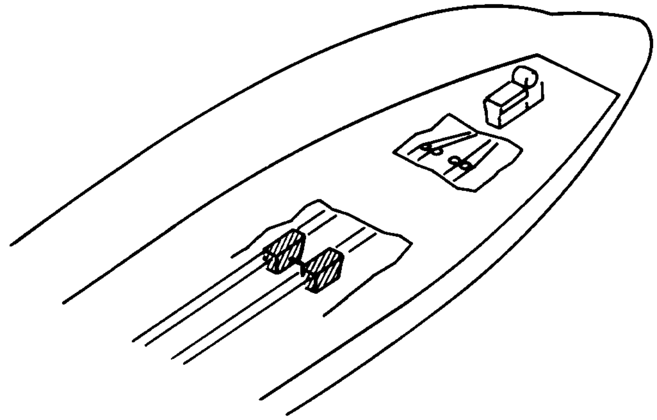
1. Remove Clutch Assembly
   A. Place thrust levers in idle position.
   B. Remove cotter pin, nut, and washer (1) from shaft of engine No. 1 clutch assembly (Fig. 401, Section A-A).
   C. Remove two bolts (4) at each end of coupling and torque tube assembly (3) which retain couplings to clutch shafts (2), (6).
   D. Remove nut, and washer (11) from engine No. 2 clutch assembly key shaft (7) (section B-B).
   E. Remove nut and washer (99) from locking arm (8) and remove locking arm.
   F. Remove engage key (10) from right end of clutch shaft (6) and key shaft (7).
   G. Fully extend flaps on leading edge to expose control cable turnbuckles at wing stations 183 and 155 (Fig. 402).
   H. Loosen turnbuckles in wings and at body station 303. Tag applicable TA and TB throttle control cables for each engine.
   I. Remove cotter pins, washers and cable guard rods from each clutch support.
   J. Remove cotter pin at end of each cable retainer on clutch assembly.
   K. Remove cables from input and output drums of clutch assembly and tag cables.
   L. Support clutch assembly. Slide clutch shaft inboard until shaft is removed and remove clutch assembly.
   M. Save bushing (12) for reinstallation (RH only).
   N. Tag clutch assemblies and clutch shafts No. 1 and No. 2 respectively.

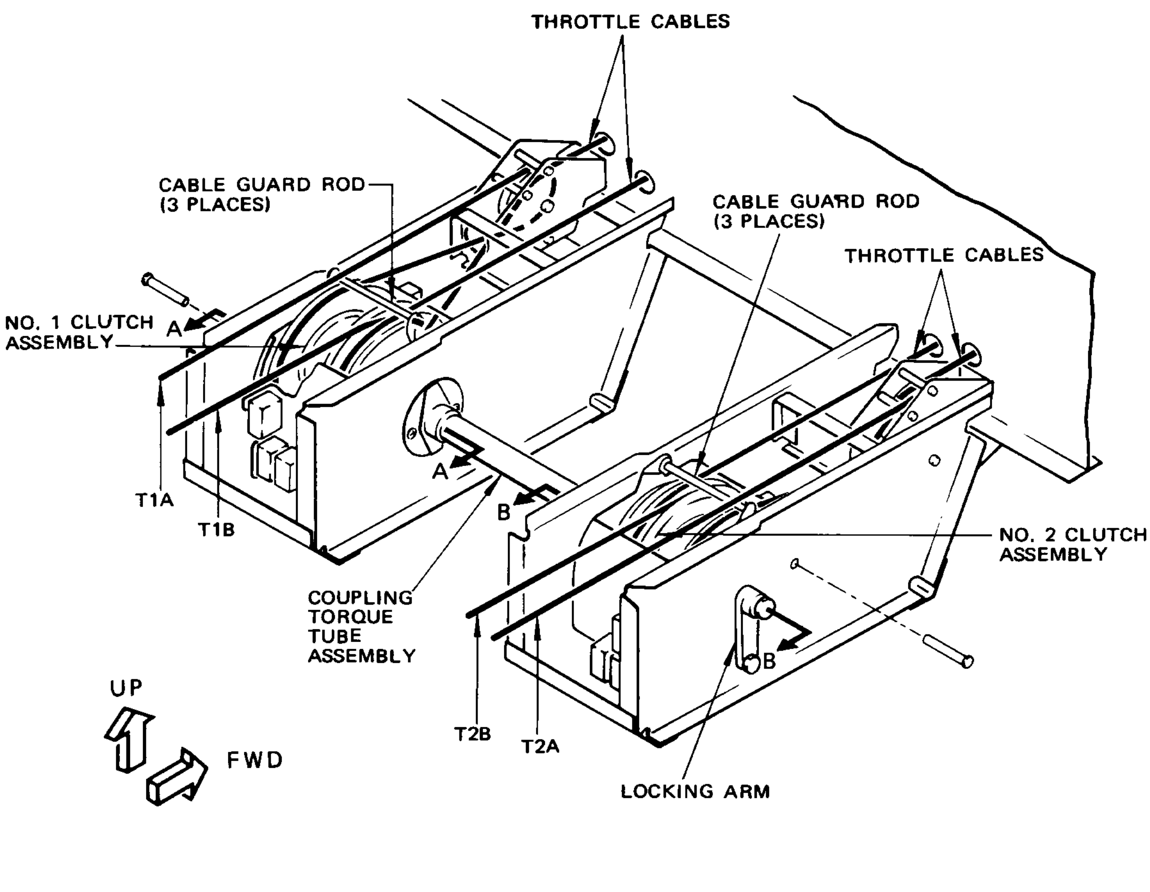
      NOTE: Units are not interchangeable.

2. Install Clutch Assembly
   A. Check clutch assembly, shafts and clutch support bearings for wear. Refer to Autothrottle Clutch Assembly - Inspection/Check.
   B. Apply a light film of grease to clutch assembly spline and clutch shaft (Fig. 401).
   C. Position clutch assembly between clutch supports and install clutch shafts and clutch assembly as follows:

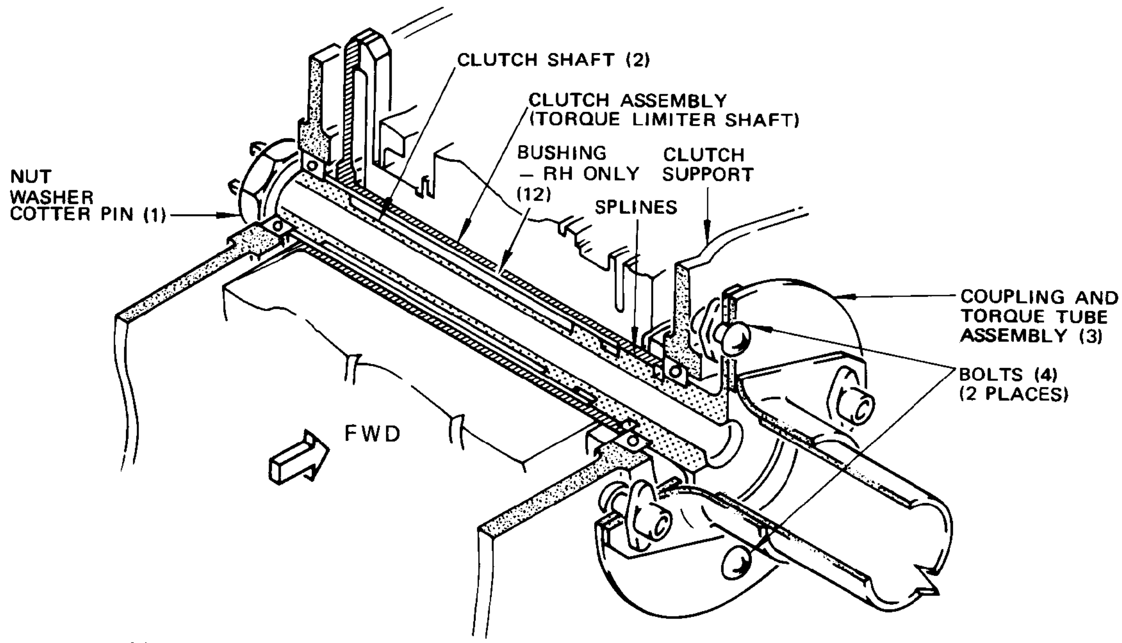
      NOTE: Clutch assembly is installed with ball race inboard.

      (1) Clutch Assembly No. 1 (section A-A).
          (a) Slide bushing removed in step 2.M. over shaft.
          (b) Slide clutch shaft (2) outboard through clutch support and clutch assembly until splines are engaged and shaft is firmly seated against clutch support bearings.

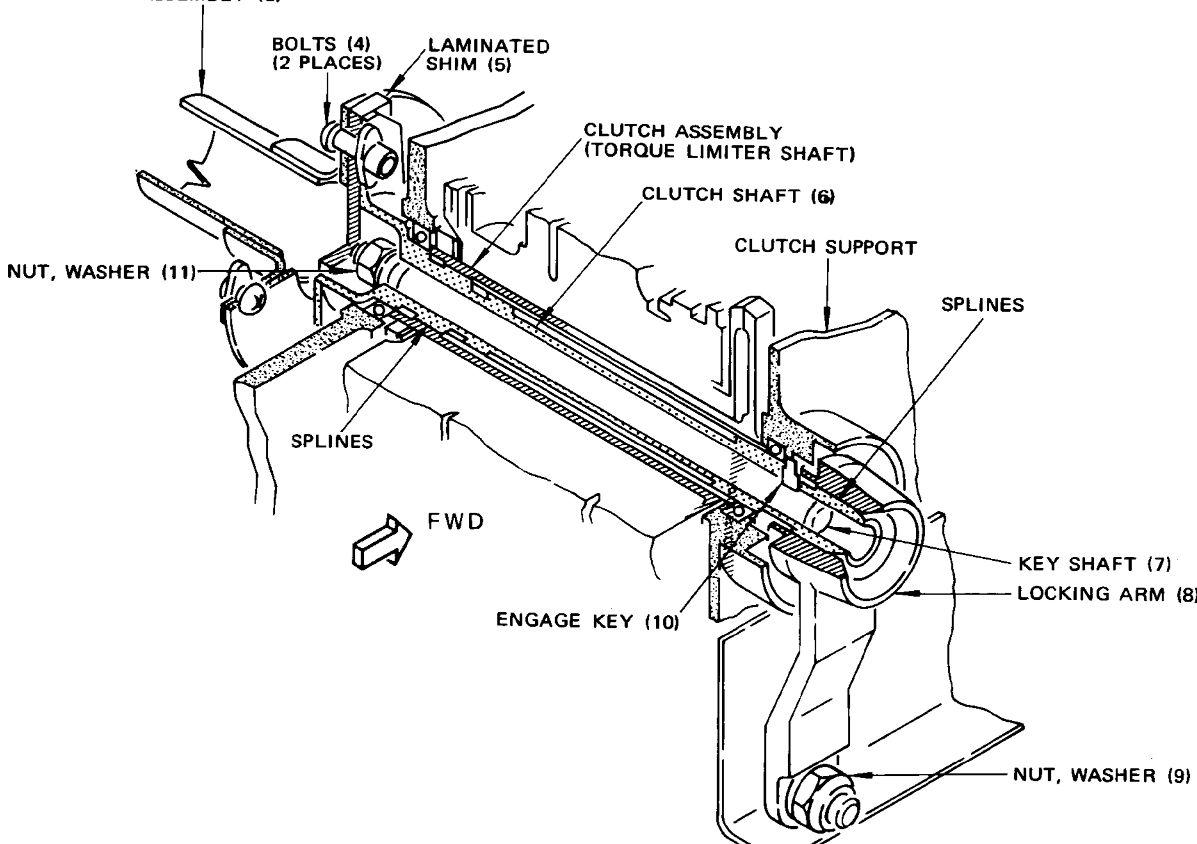




Auto Throttle Clutch Assembly Installation
Figure 401 (Sheet 1)

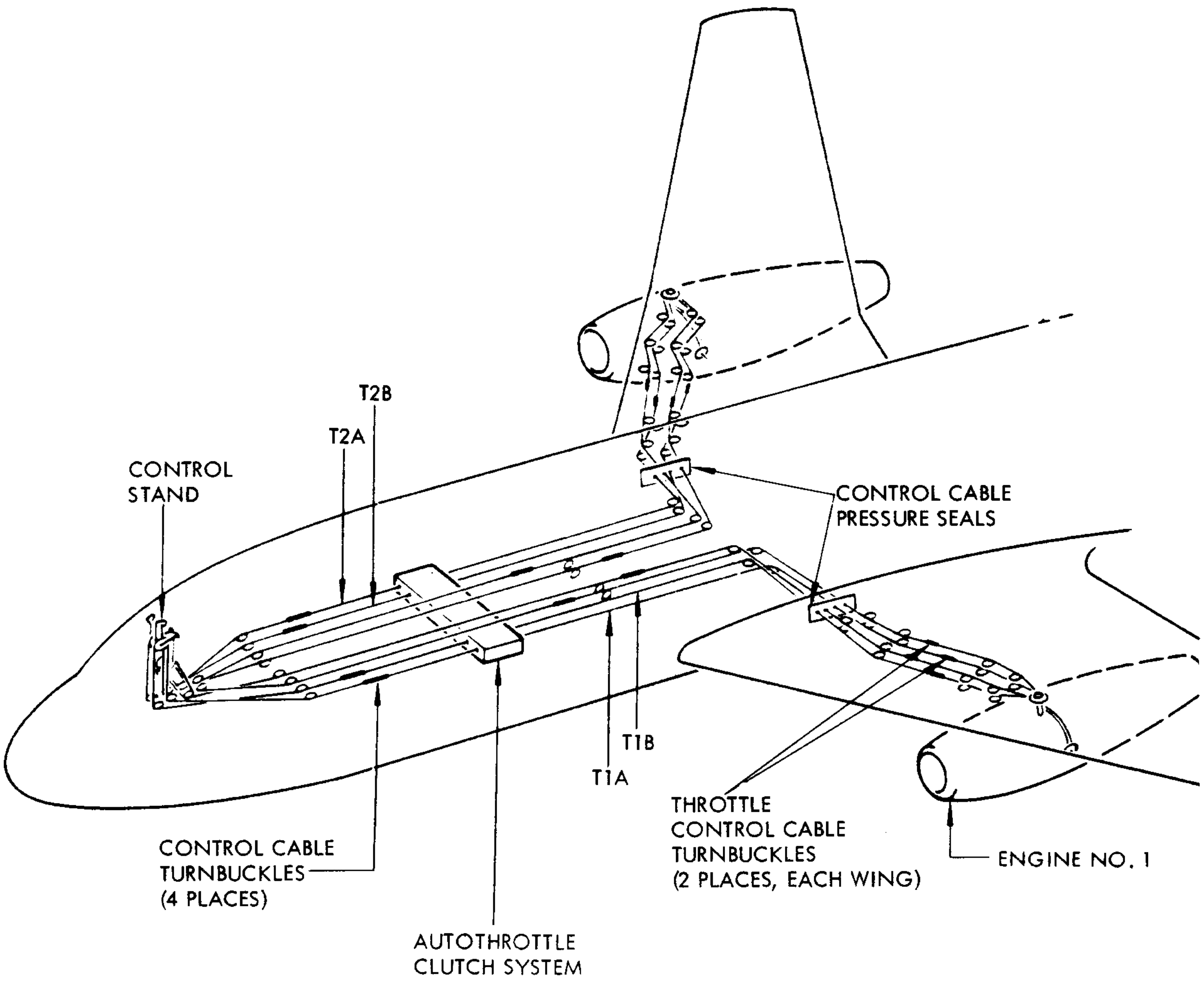


SECTION A-A



SECTION B-B

Autothrottle Clutch Assembly Installation
Figure 401 (Sheet 2)



Engine Control Cables
Figure 402

(c) Install washer and nut (1) on clutch shaft. Torque nut 40 to 50 pound-inches and install cotter pin.

(2) Clutch Assembly No. 2 (section B-B).

(a) Slide bushing removed in step 2.M. overshaft.

(b) Slide clutch shaft (6) outboard through clutch support and clutch assembly until splines are engaged and shaft is firmly seated against clutch support bearings.

(c) Slide key shaft (7) into clutch shaft keeping screw threads inboard.

(d) Align key shaft with clutch shaft and install engage key (10) through slot in outboard end of shafts.

(e) Install washer and nut (11) on key shaft. Torque nut to 20 pound-inches.

D. Connect coupling and torque tube assembly to clutch shaft No. 1 with bolts (4).

E. Position laminated shims (5) between coupling and clutch shaft No. 2 and install bolts (4). Replace shims as required for maximum gap of 0.010 inch before installing bolts.

F. Position cables on input and output drums of clutch assembly and install cotter pins.

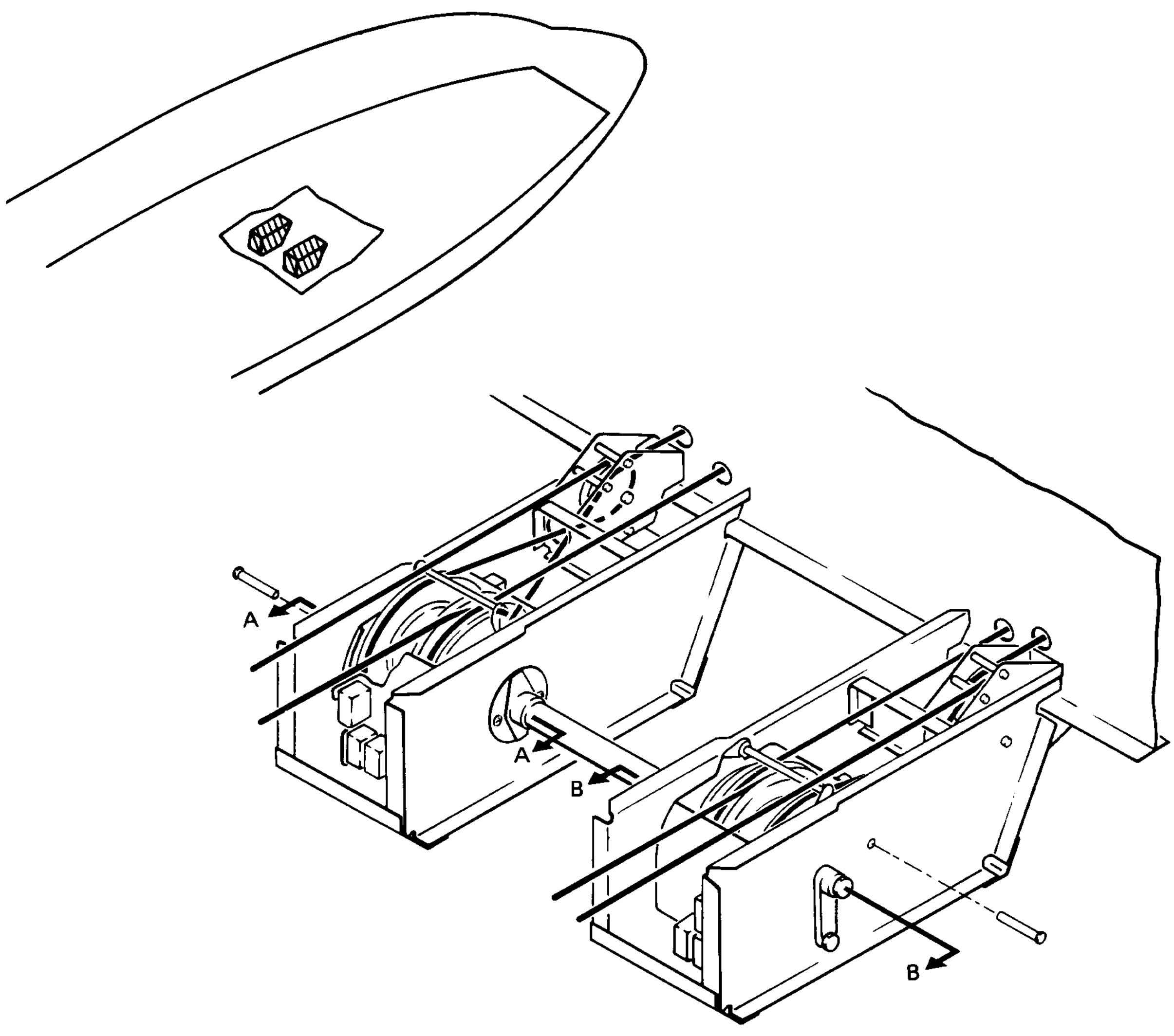
G. Install cable guard rods with washers and cotter pins.

H. Install locking arm (8) with washer and nut (9).

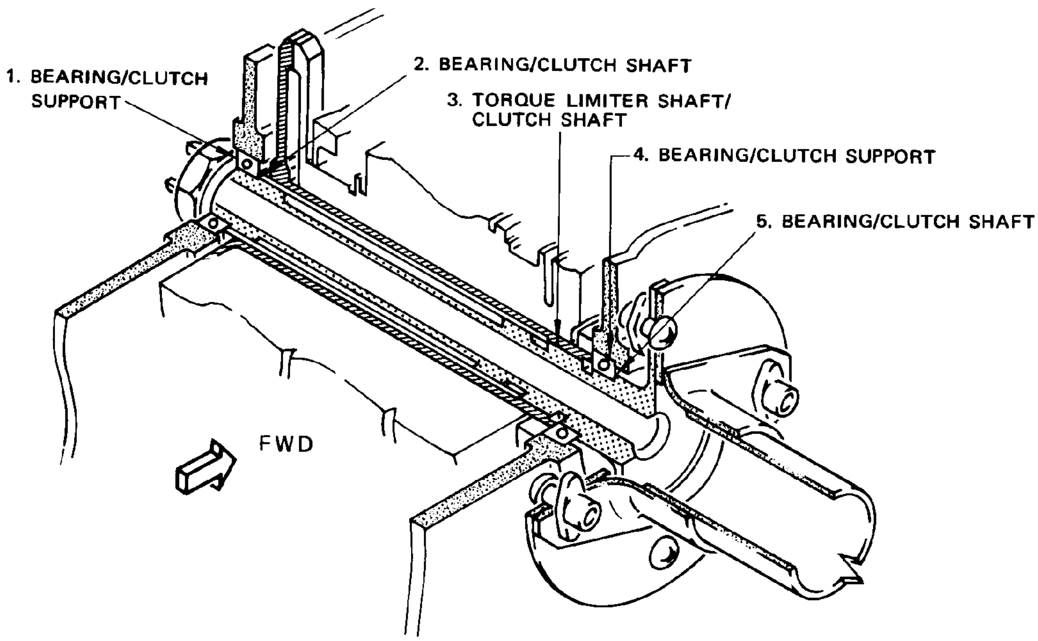
I. Adjust and test throttle control cables. See Engine Control System, Chapter 76.

AUTOTHROTTLE CLUTCH ASSEMBLY - INSPECTION/CHECK

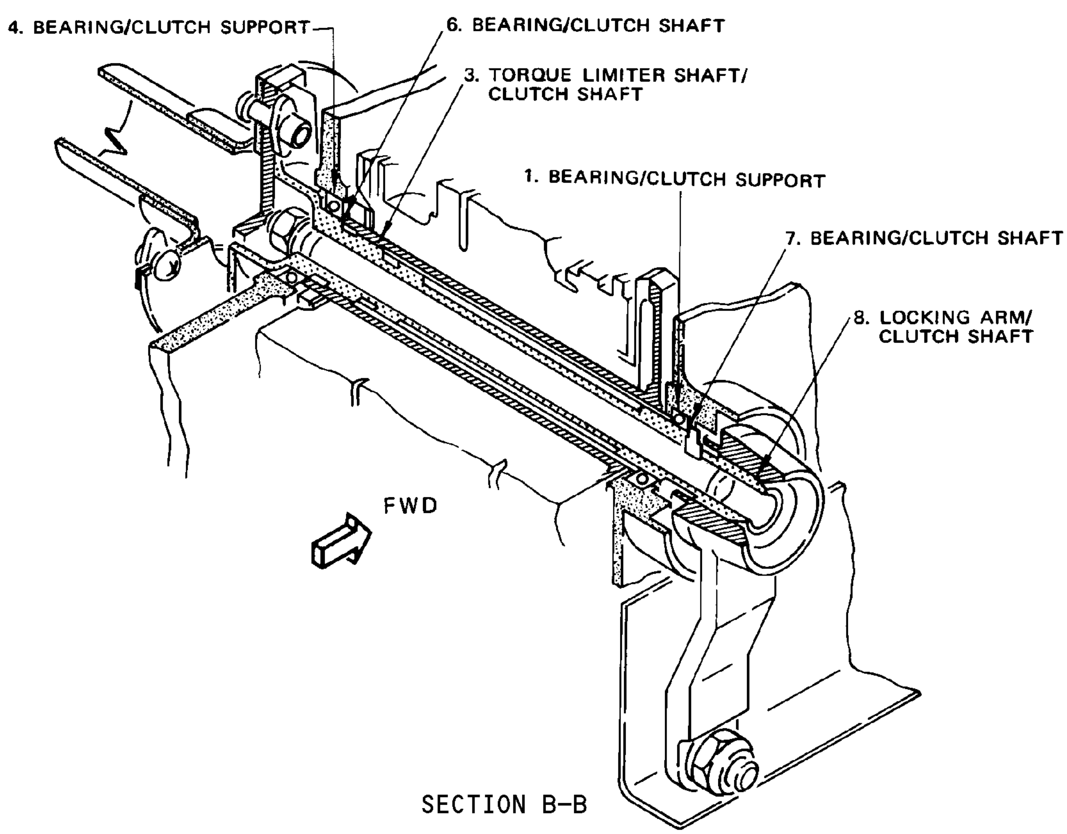
1. General
   A. The following chart and illustrations provide wear limit information for inspection of the components shown. No procedure is given for gaining access to, or removing and replacing, the component after inspection. Refer to Autothrottle Clutch Assembly - Removal/Installation procedures for this information.
2. Autothrottle Clutch Assembly Wear Limits



Auto Throttle Clutch Assembly Wear Limits
Figure 601 (Sheet 1)



SECTION A-A



SECTION B-B

Auto Throttle Clutch Assembly Wear Limits
Figure 601 (Sheet 2)

INDEX NO.	PART NAME	DIM.	DESIGN LIMITS DIAMETER MIN	DESIGN LIMITS DIAMETER MAX	WEAR LIMITS MAX WEAR DIM.	WEAR LIMITS MAX DIAM CLEARANCE	REPLACE WORN PART	REPAIR WORN PART	REPAIR INSTR.
1	Bearing	OD	1.0615	1.0625	1.0610	0.004	X		
	Clutch Support	ID	1.0630	1.0640	1.0650			X	[1]
2	Bearing	ID	0.6243	0.6257	0.6262	0.004	X		
	Clutch Shaft	OD	0.6233	0.6243	0.6222		X		
3	Torque Limiter Shaft	ID				0.005 [2]	X		
	Clutch Shaft	OD					X		
4	Bearing	OD	1.1875	1.1885	1.1870	0.003	X		
	Clutch Support	ID	1.1880	1.1890	1.1900			X	[3]
5	Bearing	ID	0.7493	0.7507	0.7514	0.004	X		
	Clutch Shaft	OD	0.7483	0.7493	0.7474		X		
6	Bearing	ID	0.7493	0.7507	0.7514	0.004	X		
	Clutch Shaft	OD	0.7483	0.7493	0.7474			X	[4]
7	Bearing	ID	0.6243	0.6257	0.6262	0.004	X		
	Clutch Shaft	OD	0.6233	0.6243	0.6222			X	[5]
8	Locking Arm	ID				0.005 [2]	X		
	Clutch Shaft	OD					X		

Auto Throttle Clutch Assembly Wear Limits
Figure 601 (Sheet 3)