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ORION

editor DON WADSWORTH

assistant editors WAYNE CRADDUCK : EUGENE DRICKEY

art GLEN COCHRAN

FRONT AND BACK COVERS Patrol Squadron 26 was first commissioned in 1943 as Bombing Squadron 114, later added "Patrol" to this title, then switched to Heavy Patrol Squadron 6 and finally emerged as VP-26.

Although their mission has always been primarily ASW, a task ably carried out in the Mediterranean and Caribbean areas during the latter part of WWII, their versatility and mobility made them a "natural" for many diverse and unrelated tasks. These included their assignment as one of the first Navy units to conduct hurricane reconnaissance and their task of delivering tons of medical supplies in the massive Berlin airlift.

Originally assigned to fly PB 4Y "Liberators", VP-26 transitioned to the land based SP-2 "Neptune" in 1951. Shortly thereafter they were the first squadron ordered aboard the recommissioned Naval Air Station at Brunswick, Maine, a spot they still use as their home base. VP-26 deployed to many North Atlantic outposts including Thule, Greenland in 1956. While there, all twelve crews established another first by flying their Neptunes as a squadron over the North Pole.

The second consecutive "E" for Battle Efficiency awarded to VP-26 in 1956 and their 1963 record for having the largest number of designated "aircrewmen" in the Atlantic Fleet attest to their continued excellence and concentration on assigned tasks.

A detachment of VP-26 was directly involved in the Cuban crisis in 1962 where during more than 1000 hours of flying in support of the quarantine, squadron units sighted four Soviet vessels carrying medium range ballistic missiles and three transporting IL-26 bombers.

On an unusual assignment for a low altitude ASW aircraft at Pope Air Force Base, North Carolina, the squadron's Neptunes were rated "excellent" as a jump platform by the Army Special Forces. This task was carried out far inland at altitudes above 20,000 feet.

Fifteen years of outstanding service in P-2 "Neptunes" came to a close in 1965, when VP-26 transitioned to the Navy's latest ASW aircraft, the P-3 Orion. Early in 1966 Commander James H. Cullen, the squadron's Commanding Officer at the time, ferried the first of the Navy's "B" model Orions to Brunswick, Maine.

The cover picture shows one VP-26 aircraft operating near Surtsey, an active volcano south east of Iceland and probably the "hottest" spot in the squadron's icy North Atlantic ASW surveillance area.

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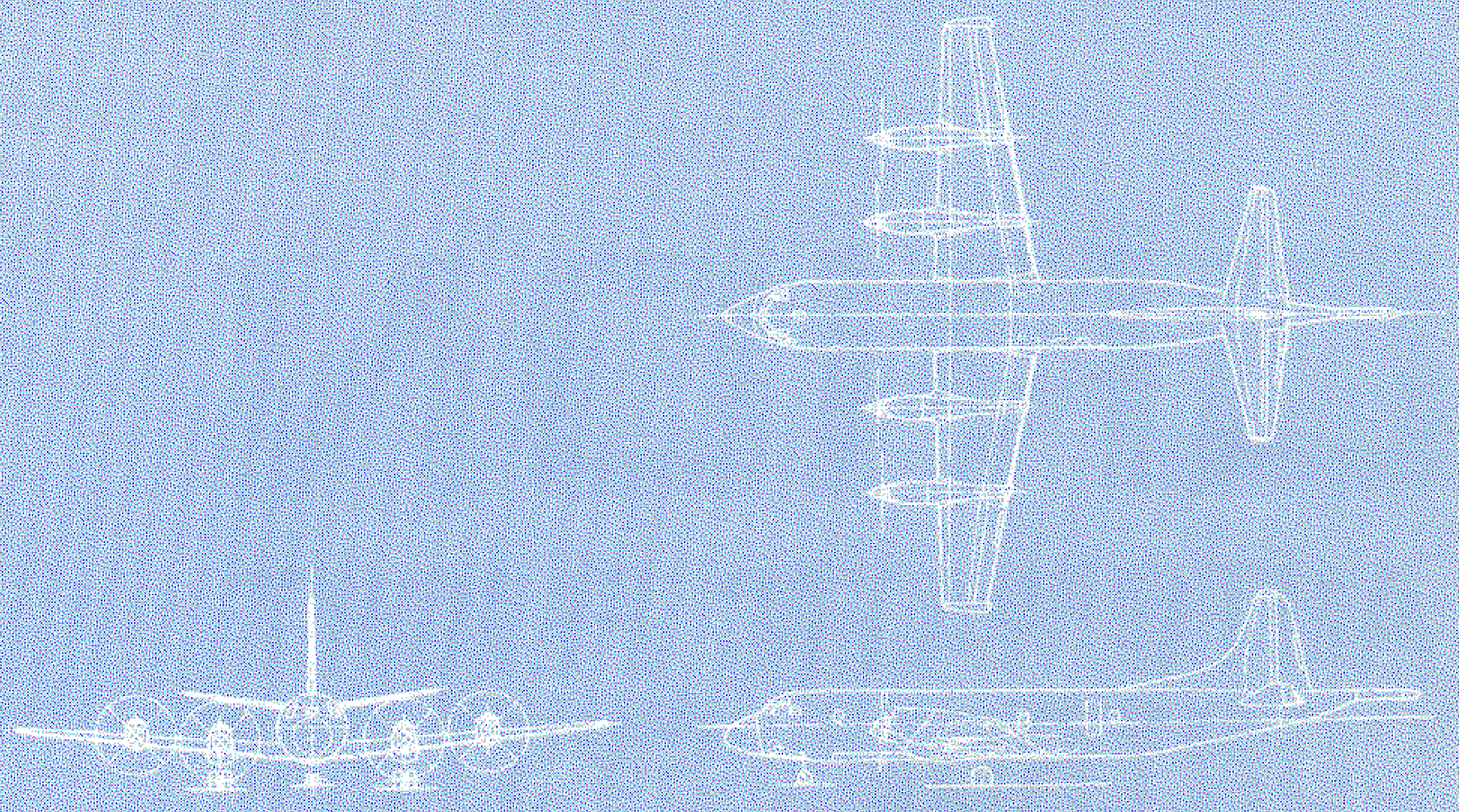


FOREWORD

The Orion Automatic Flight Control System (AFCS) information in this article is closely related to the "Flight Controls" described in Digest Issues 8 and 10 and indirectly to the "Hydraulic Power System" covered in Issue 7.

Our discussion describes the system currently being installed on new aircraft (BUNO 152718 and subsequent) and those modified by AFC No. 71. However, **all** AFCS units except the Amplifier/Computer are interchangeable with those used in unmodified aircraft.

A few of the P-3 aircraft in service have already been modified, and all will eventually be retrofitted by Airframe Change No. 71 which, in addition to the installation of a reworked Amplifier/Computer, provides the following changes to components which are closely allied to the AFCS:

1) Replacement of the APN-117 Radar Altimeter (range 0 to 1000 ft.) with the more accurate and expanded-scale APN-141 Radar Altimeter (range from 0 to 5000 ft.).

2) Installation of an Altimeter/Autopilot Coupler (CU-1503) which "memorizes" the radar altitude signal at the time RADAR ALT. HOLD is selected and thereafter provides a signal to the autopilot that represents deviation from the engaged altitude. In the radar hold mode the coupler also activates a flasher that causes the dual AUTO PILOT/RADAR ALTM red warning lights on the glare shields to flash when the altitude deviates more than 75 feet from the memorized reference altitude.

3) Installation of the APQ-107 Radar Altitude Warning Set (RAWS) which alerts the Pilot and Copilot — with a pulsating 1000-cps tone in their headsets or speakers and with the flashing red lights on their glare shields — to one or more of the following: the radar altitude signal is unreliable or its power is off; the aircraft is either descending through a 380-foot altitude (3-sec. duration warning) or is operating at 160 feet or less (continuous warning).

The RAWS system is a universal warning set, designed for use in both fixed and rotary wing aircraft. In the P-3, the RAWS principal function is to report on the status of the radar altimeter and its operation is independent of the autopilot.

The internal modification of the PB-20N Amplifier/Computer deletes those functions that are now provided by the new Altimeter/Autopilot Coupler. Our text contains some further explanation of the Amplifier/Computer modification, and for more detailed information about these changes, personnel should consult the latest revision of the official NAVAIR manuals.

ORION

AUTOMATIC FLIGHT CONTROL SYSTEM

THE BENDIX PB-20N AFCS supplied by their Eclipse-Pioneer Division is designed to control the flight path of the P-3 with a minimum of supervision by the pilot. In flight prior to engagement, it generates signals representing aircraft attitude and computes a synchronizing (cancelling) signal to provide a smooth transition from standby (no-signal) to "autopilot engage." After engagement, it continues generating attitude signals, accepts signals from other systems aboard the aircraft, combines this data, limits its value, and applies computed signals to the hydraulic boosters of the appropriate flight controls. These convert the electrical signals into the amount of hydraulic/mechanical force — within the autopilot limits — needed to return the aircraft smoothly to the flight profile selected by the Pilot.

Actually, each of the AFCS channels — rudder, aileron, and elevator — is a servo system which is practically independent of the other two channels. Each channel includes several sensors, amplifiers for precise signal control, boosters, and feedback devices to limit the extent and rate at which the work is done. In each channel, the P-3's hydraulically operated control surface booster provides the working portion of the loop to make transitory corrections — replacing the electric motors utilized by some designs.

As explained in Digest Issues 7, 8, and 10, the Orion's primary flight control surfaces are usually operated with the hydraulic boosters "on" to accomplish a large part of the work required to move the control surfaces. The power section of each booster assembly consists of a dual hydraulic cylinder incorporating twin pistons on a single actuating rod that is linked mechanically with its associated flight control surface. The pistons work in concert, normally, utilizing power from independent hydraulic systems, but either piston alone can furnish sufficient force for normal maneuvers. A dual control valve meters hydraulic forces at the dual power cylinder in response to mechanical signals from the pilot via the primary flight controls, or in response to electrical signals from the autopilot Amplifier/Computer. Direct "booster off" manual control of each surface can be selected individually by pulling the appropriate booster shift control (See Figure 1) but to provide the necessary increase in the pilot's mechanical advantage during "boost-off" operation it is necessary to appreciably reduce the "throw" of the control surfaces when the shift handle is pulled.

The autopilot can effect aircraft control only during "boost-on" operation, the case in which control surfaces are permitted maximum throw. Actually, however, it is unlikely that the autopilot will realize full throw on any control under flight conditions, for autopilot authority to overcome control surface airloads via the boosters is mechanically limited to only a small fraction of the force which can be obtained by the pilot in the manual "boost-on" configuration. The pilot can apply manual control at will without disengaging the autopilot, either overriding the autopilot signal authority, by applying an opposite manual signal, or adding a manual signal to supplement that authority if he wishes to facilitate a maneuver beyond the autopilot's built-in limitations.

CONTROL SURFACE BOOSTERS

It is not within the scope of this article to provide a complete detailed explanation of P-3 booster operation. However, since the boosters are not only the "working" portion of the system but are also equipped with components which receive information from, and transmit information to the autopilot, we are including a brief generalized discussion and some rudimentary schematics to illustrate their operating principles.

Operationally the boosters for all three channels are identical, although their installations differ in the mechanical details of force application to the control surfaces. While "boost-on" is selected, full system pressure reaches the booster control valve and, if the valve is neutral, a reduced pressure, equal at all points, reaches all of the boost actuator ports. This is due to the design of the boost control valve which permits a small flow from pressure — to both actuator ports — to return. The continual small flow that returns to each system reservoir is an effective means of "bleeding" the booster components,

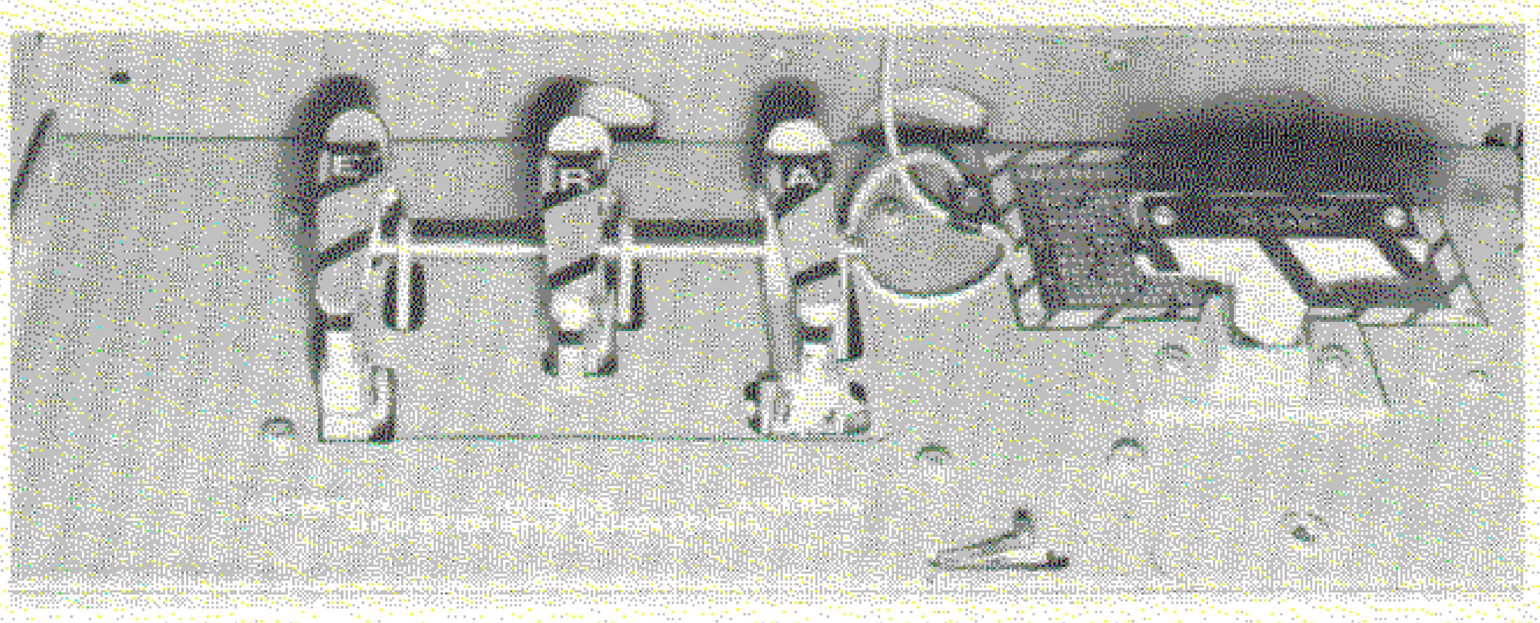


Figure 1 Booster Shift Controls

and explains the reduced, equalized pressure throughout the boost actuator shown in Figure 12.

It should be noted that a surge-damping orifice valve and an additional hydraulic load sensor are peculiar to the elevator booster. The additional load sensor generates an electrical signal which is utilized for automatic control of the elevator trim tabs (Automatic Pitch Trim) to be discussed later in this article.

The orifice valve assembly depicted in Figure 3 is utilized only while the elevators are operating "boost-on" under manual control. It provides internal porting (calibrated leakage) of hydraulic fluid across both pistons of the dual actuating cylinder. Its pur-

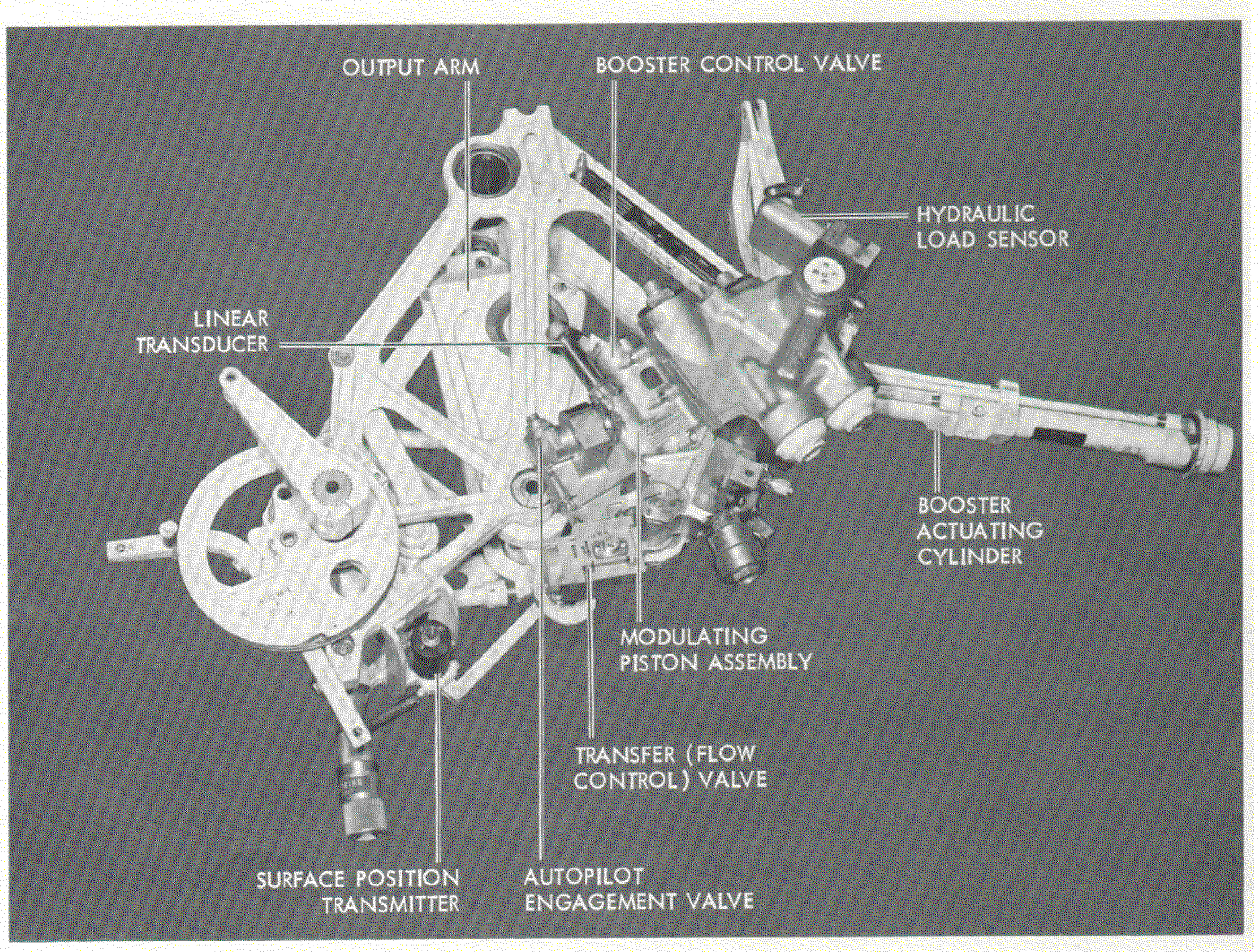


Figure 2 Typical Booster Assembly — Aileron Shown

pose is to damp out the hydraulic pressure surges caused by "pumping" action of the cylinders (induced by transient airloads) which would otherwise tend to start slow speed porpoising of the aircraft under manual control. As noted in Figure 3, during autopilot operation hydraulic system No. 1 pressure is directed to the area between the orifice pistons, and piston action closes the interconnecting ports. The autopilot can then develop the full differential pressure needed for speedy response to autopilot control.

As might be expected, autopilot operation involves additional system complexity, incorporating devices to establish restricted authority over the control surfaces.

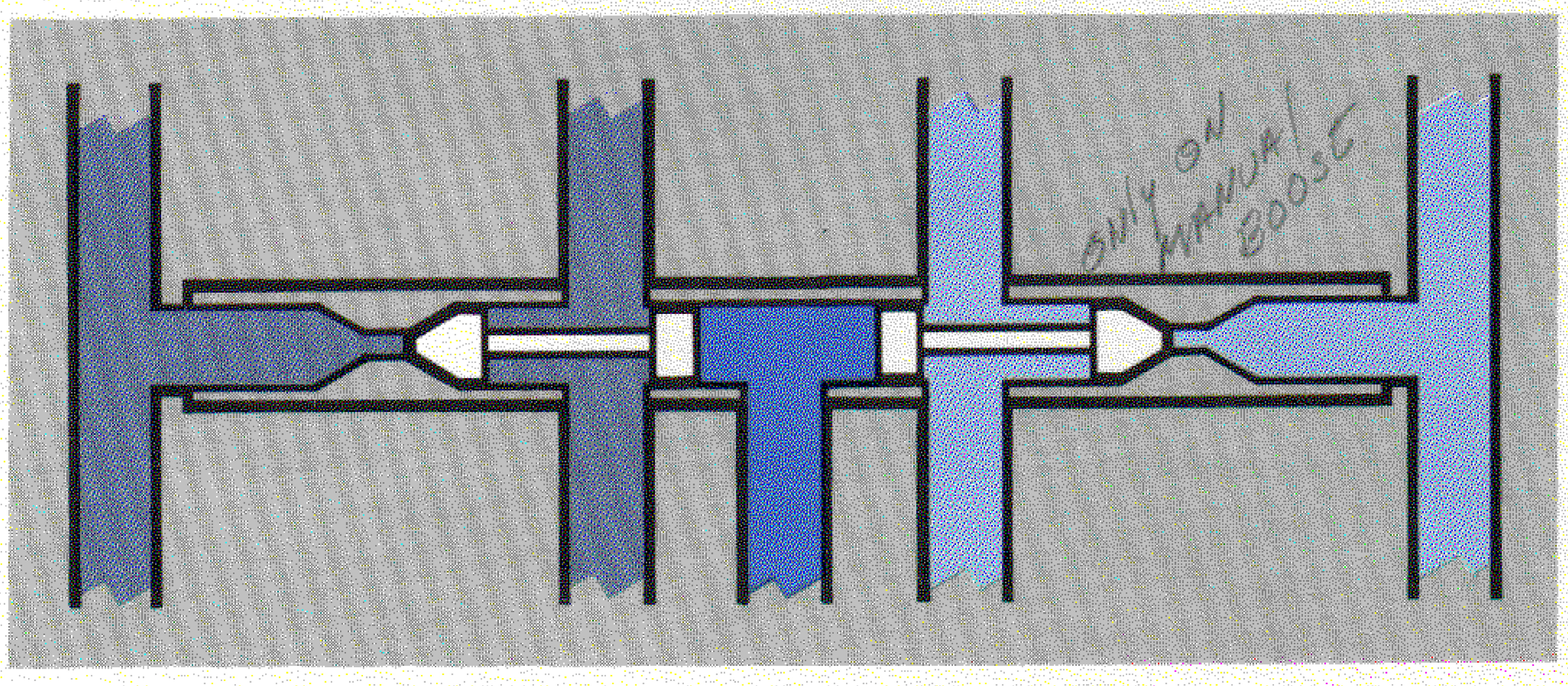
Selecting "ENGAGE" at the autopilot ENGAGE-OFF SWITCH actuates the *autopilot engagement valve,* a simple solenoid operated shutoff valve that ports full No. 1 hydraulic system pressure* to the transfer (flow control) valve** and to the *autopilot engagement actuator* at each booster. A lower balanced pressure also reaches both sides of the modulating piston in the booster control valve.

The autopilot engagement actuator receives full system No. 1 hydraulic pressure when the autopilot is engaged and its piston moves a roller into a V-shaped ramp detent at one end of the booster control valve actuating arm. The roller retains the arm in a stationary position, except for the unusual case in which the pilot applies sufficient force to the flight station control to override the autopilot. Since the effective area of the engagement piston is quite small, override of the autopilot requires only a relatively small pilot force on a flight control. As shown in Figure 4, the valve actuating arm acts as a pivot point for the small auxiliary arm that links the modulating piston to the booster control valve spool. It is the offset pivot point of this small arm that limits the autopilot control of the booster valve and therefore the degree of force the actuating cylinder can apply to move the control surface.

* *Note that this use of No. 1 System pressure, exclusive of No. 2 System, makes it impossible to utilize the autopilot if No. 1 System is de-pressurized.*

***The transfer valves permit a "bleed" flow similar to the flow at the boost control valve.*

Figure 3
Surge Damping Orifice Valve is Utilized on Elevator Booster Only



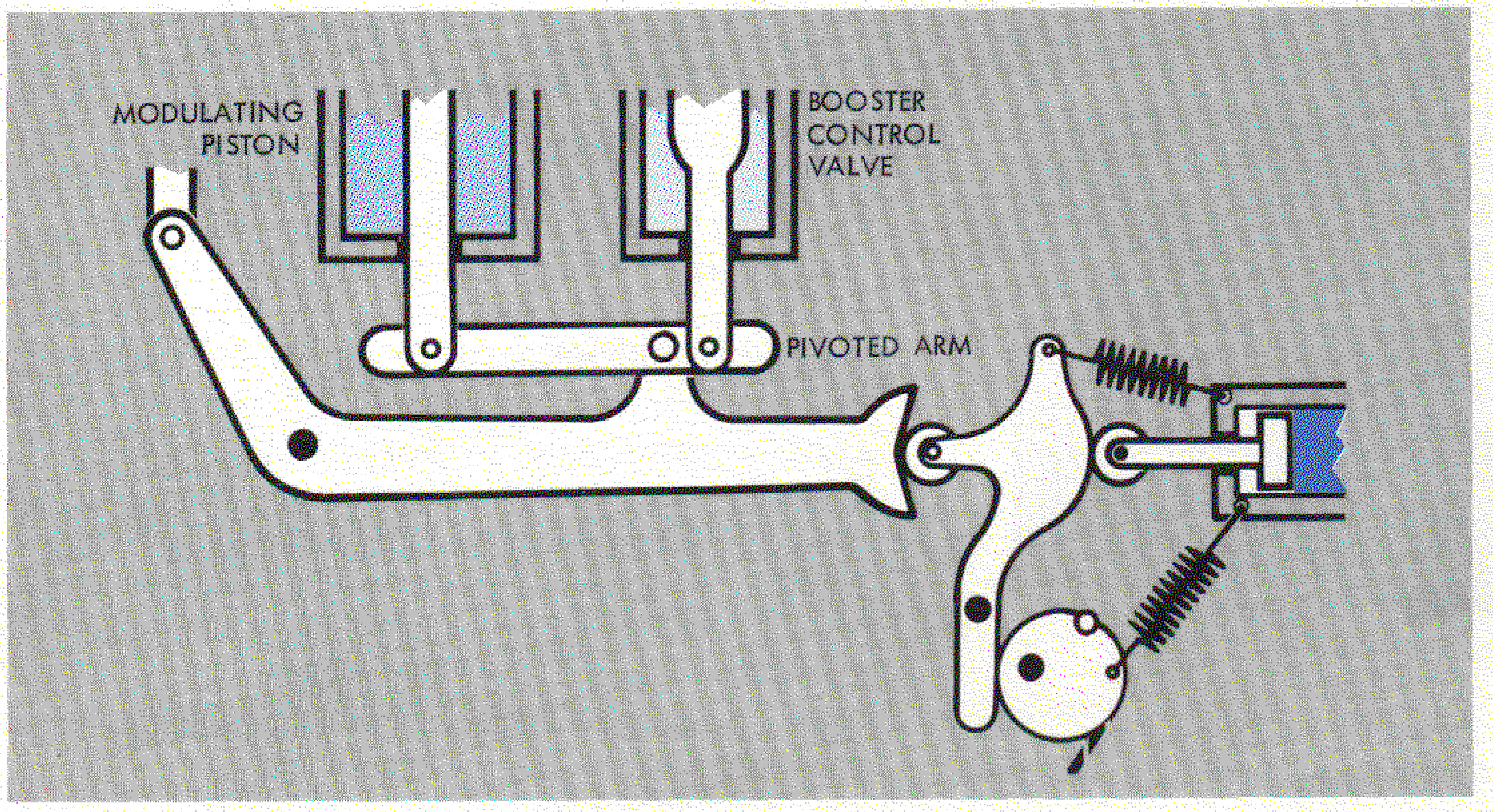


Figure 4
Pivoted Auxiliary Arm Linking Modulating Piston and Booster Control Valve

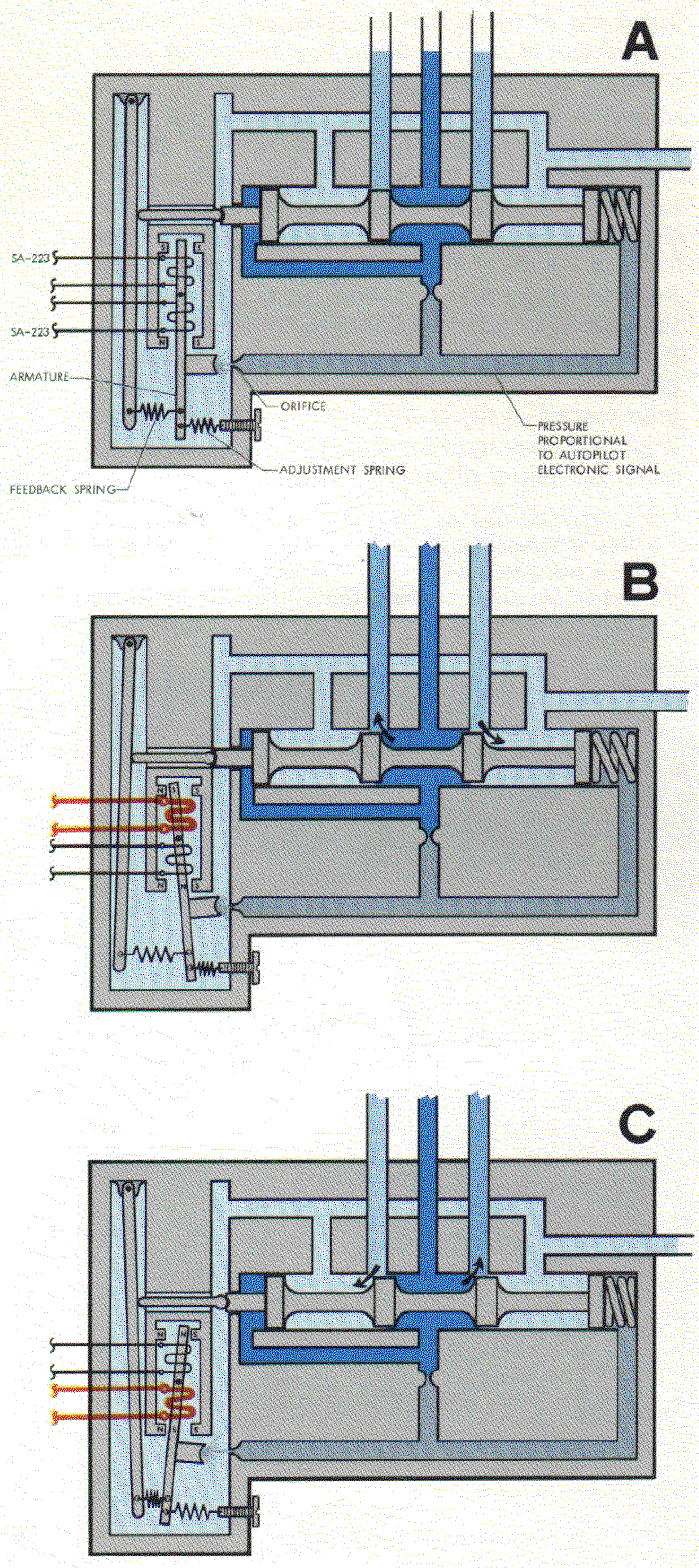


Figure 5 Effect of Applying Autopilot Signals to Transfer Valve

The transfer valve in each channel contains two coils directly interconnected to its valve amplifier and, when No. 1 hydraulic system pressure is applied by the autopilot engagement valve, the transfer valve converts any incoming autopilot electronic signal into a proportional hydraulic pressure — a "pilot" pressure that moves the modulating piston. As shown in Figure 5 a, b, and c these "pilot" pressures are determined by precise positioning of the slide valve in response to differential forces exerted at the slide ends.

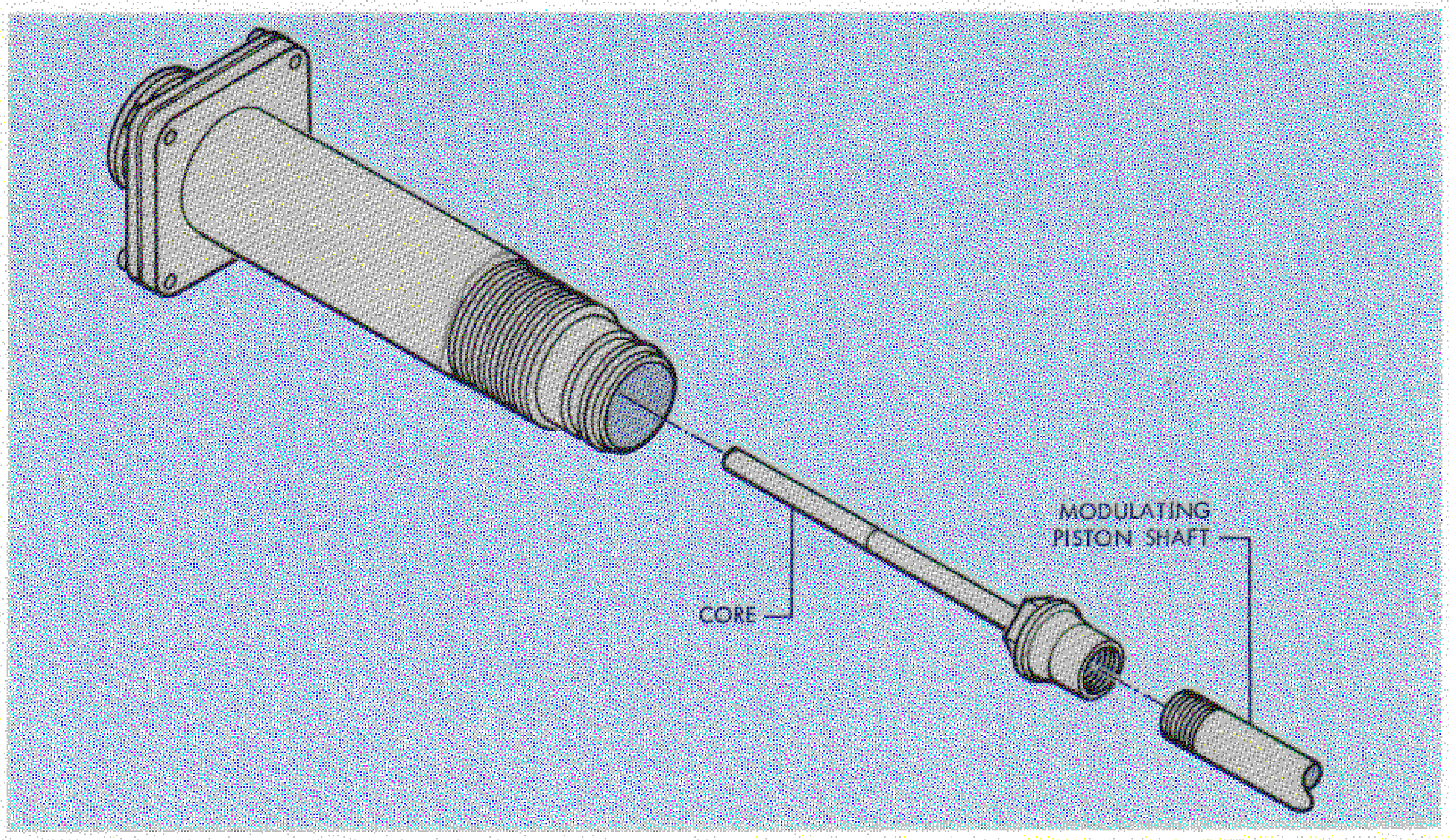
Initially, the slide is held at "hydraulic null" by spring action, and remains there when full system pressure (approx. 3000 PSI) is applied on a small piston at one end while a reduced pressure (approx. 1500 PSI) is applied on the opposite large-area piston. Note that as mentioned previously, at the null point the calibrated "internal leakage" of the transfer valve supplies a balanced pressure via its output lines to both ends of the modulating piston.

When an autopilot signal is applied to one transfer valve coil the armature is displaced from its spring-centered position in the direction shown in Figure 5b and the target is moved closer to the nozzle orifice. This restricts the flow of hydraulic fluid into the low pressure return chamber and results in an increase in pressure in the 1500 PSI line. This increase is reflected on the slide valve large-area piston and the slide is forced out of null to the left.

The displaced slide valve increases tension on a *feedback* spring through a push-rod lever arrangement which tends to return the armature toward its null position. This has the overall effect of slowing the valve's response to autopilot commands and its purpose is to ensure a smooth change in hydraulic pressure to the modulating piston.

As the slide moves in the direction shown in Figure 5b, the pressure rises in the left hand and falls in the right hand output line. The increasing pressure on one side and the decreasing pressure on the other force the modulating piston down against its centering spring. When the modulating piston moves, it displaces the core of the linear transducer attached to the piston shaft and the transducer generates an electrical signal that is proportional to the core's linear motion. This signal is applied to the autopilot command to the transfer valve amplifier where it tends to oppose the initial signal, thus inhibiting the modulating piston's influence on the booster control valve through the offset arm. The effect of this feedback signal is to damp the original signal, thus controlling the *rate* at which booster force is applied to the control surface.

Figure 6
Linear Transducer
with Core Removed



The booster control valve is a dual unit that effects simultaneous pressure changes of No. 1 and No. 2 hydraulic systems across the actuating cylinder. In its initial "balanced" state the 3000 PSI system pressures are applied to the input ports and internal leakage reduces this to approximately 2000 PSI at the four output ports leading directly to the actuating cylinder. Referring to the condition shown in Figure 5b, the interconnecting link forces the booster control valve spool up and thus increases the pressure to the upper output ports and decreases that to the lower ports. This causes the actuating piston to move to the left (see Figure 7).

Since both sections of the actuating cylinder are symmetrical, the total differential pressure used to move the piston is additive, half being supplied by each hydraulic system. The resultant movement is transferred to the respective control surface by way of the power arm and the output push rod.

Each booster assembly incorporates a surface position (synchro) transmitter with its rotor linked to the booster power arm by a lever and rod assembly which is spring-loaded to prevent backlash. The synchro output provides a surface position follow-up signal to the valve amplifier input to complete the servo loop.

As a control surface moves out of its trimmed position in response to autopilot commands, the output of the position transmitter increases and eventually balances the autopilot command signal with an equal and opposite signal. As the corrective maneuver returns the aircraft to the desired flight path, the initial command signal diminishes and the transmitter signal (which is now the larger of the two) takes effect as an opposite command at the transfer valve amplifier, returning the control surface to the original position.

The rudder and the aileron boosters are equipped with dual pressure-differential sensors (one for each of the two hydraulic systems). The pick-off coils of the dual units are connected in series to provide an electrical signal equivalent to the *total* hydraulic pressure imbalance across the booster actuating cylinder. The ac excitation voltage is supplied to the load sensors when power is applied to the autopilot but the "force-proportional" signal is not interconnected to its trim indicator meter until the autopilot is engaged. The elevator booster has *two* dual sensors, one for operating the elevator portion of the trim indicator and the other for supplying input to the automatic pitch trim circuit.

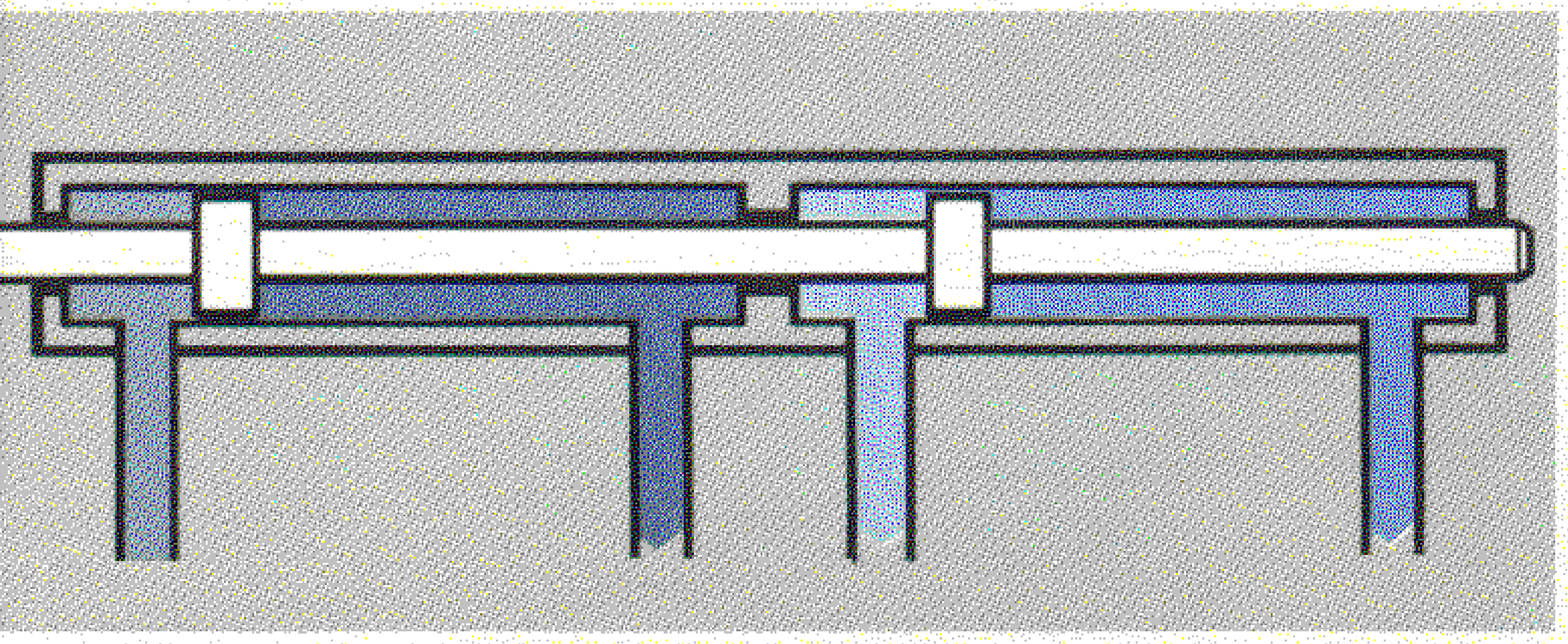


Figure 7 Booster Actuating Piston Displaced to the Left

Figure 8 Surface Position Synchro Transmitter

CONTROL PANELS

The AFCS operating controls are incorporated on the two panels shown in Figure 9 as they appear on the flight station center pedestal. It should be noted that the AFCS power control circuitry is intercon-nected with the "ground-air" sensing circuit, which automatically energizes the AFCS at takeoff, de-energizes it at touch-down. Thus, the AFCS will be operational in "standby" almost immediately after takeoff, it can be engaged at will, and the elevator auto-trim system becomes operational automatically at the time of engagement.

The "ground test control panel" shown in Figure 10 is mounted on the Pilot's side console and in-corporates controls to operate the system during ground check out operations.

The solenoid-held GROUND POWER switch shown in Figure 10 de-energizes the AFCS power relay, just as the ground-air sensing circuit does at lift off, thereby applying 115-V, 3 phase ac power to the autopilot system for ground test purposes. Although the solenoid will be de-energized (allow-ing the switch to return to "OFF") at takeoff, when ground tests are completed the switch should be re-turned to the "OFF" position manually to minimize AFCS total operating time.

Operation with ground power differs from air-borne operation in that the auto-trim system cannot be made to function by merely engaging the auto-pilot so long as the ground power switch is at "GROUND POWER." The spring-loaded AUTO TRIM GROUND TEST switch must be manually held in the test position after the autopilot is engaged to conduct a ground check of the automatic trim circuit. This switch energizes the auto trim servo clutch and retracts the AUTO TRIM OFF flag on the face of the three-axis trim indicator. The auto trim is functioning properly if the elevator trim control wheels rotate NOSE UP when the control column is pulled full aft — and vice versa.

The CONT WHEEL STEERING switch shown in Figure 9 activates the Control Wheel Steering mode by applying a 15-V ac excitation to the pitch and to the bank wheel-force sensors on both control column wheels. This provides a means whereby the pilot can temporarily assume full control of the elevator and/or the aileron boosters while the re-mainder of the AFCS is still engaged.

The "NORM" position of the AUTO TRIM LT OVERRIDE/NORM switch allows the "AUTO TRIM" warning light to illuminate if the auto trim system malfunctions. Thereafter, selecting the sole-noid-held "AUTO TRIM LT OVERRIDE" position will extinguish the light, provided that the wing flaps are in the 0-to-60% range. Beyond 60%, the flap position relay opens the ground circuit of the switch solenoid, and the switch will not override the "AUTO TRIM" warning. Appreciable pitch mom-

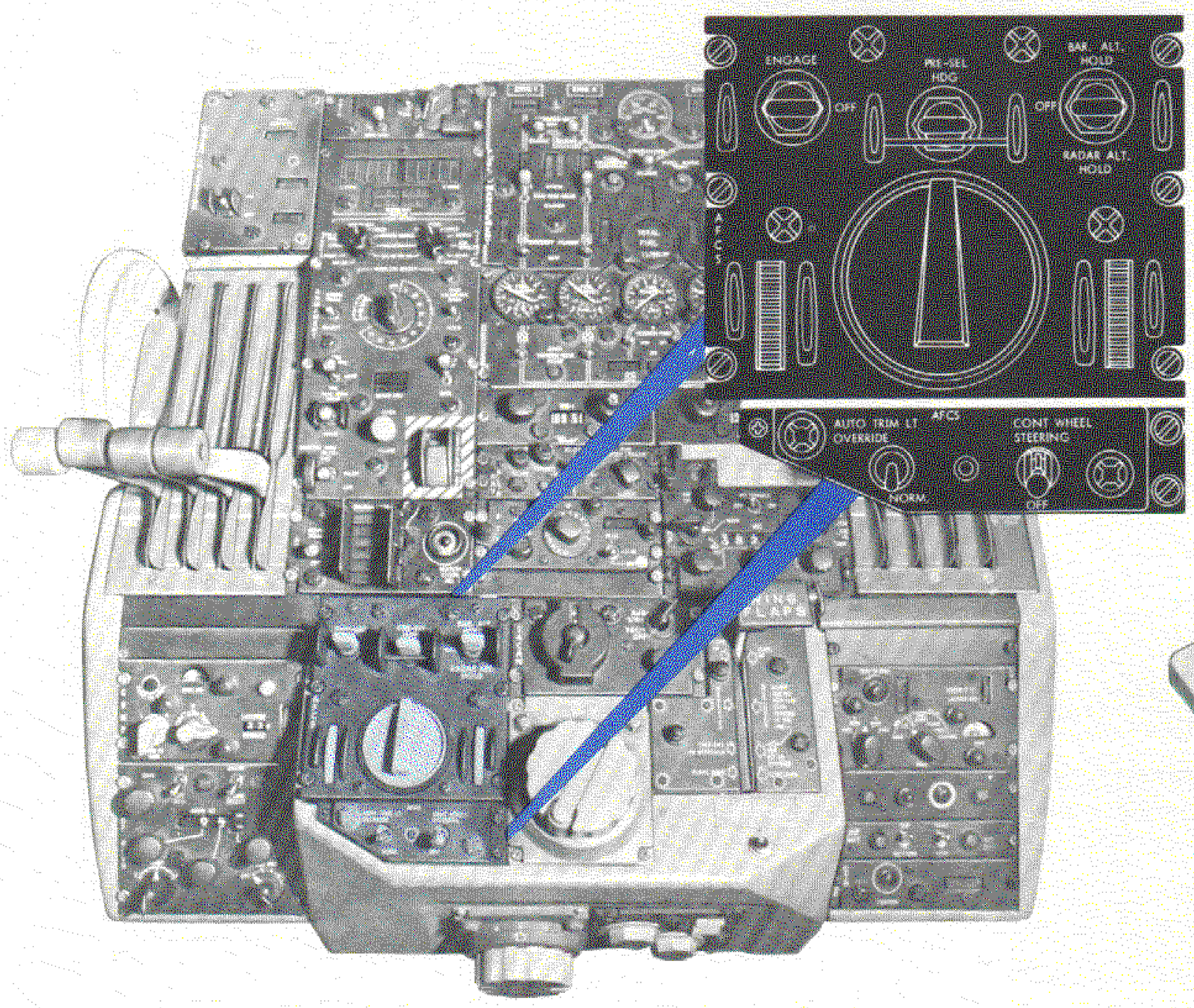


Figure 9 AFCS Control Panel and AFCS Auxiliary Panel

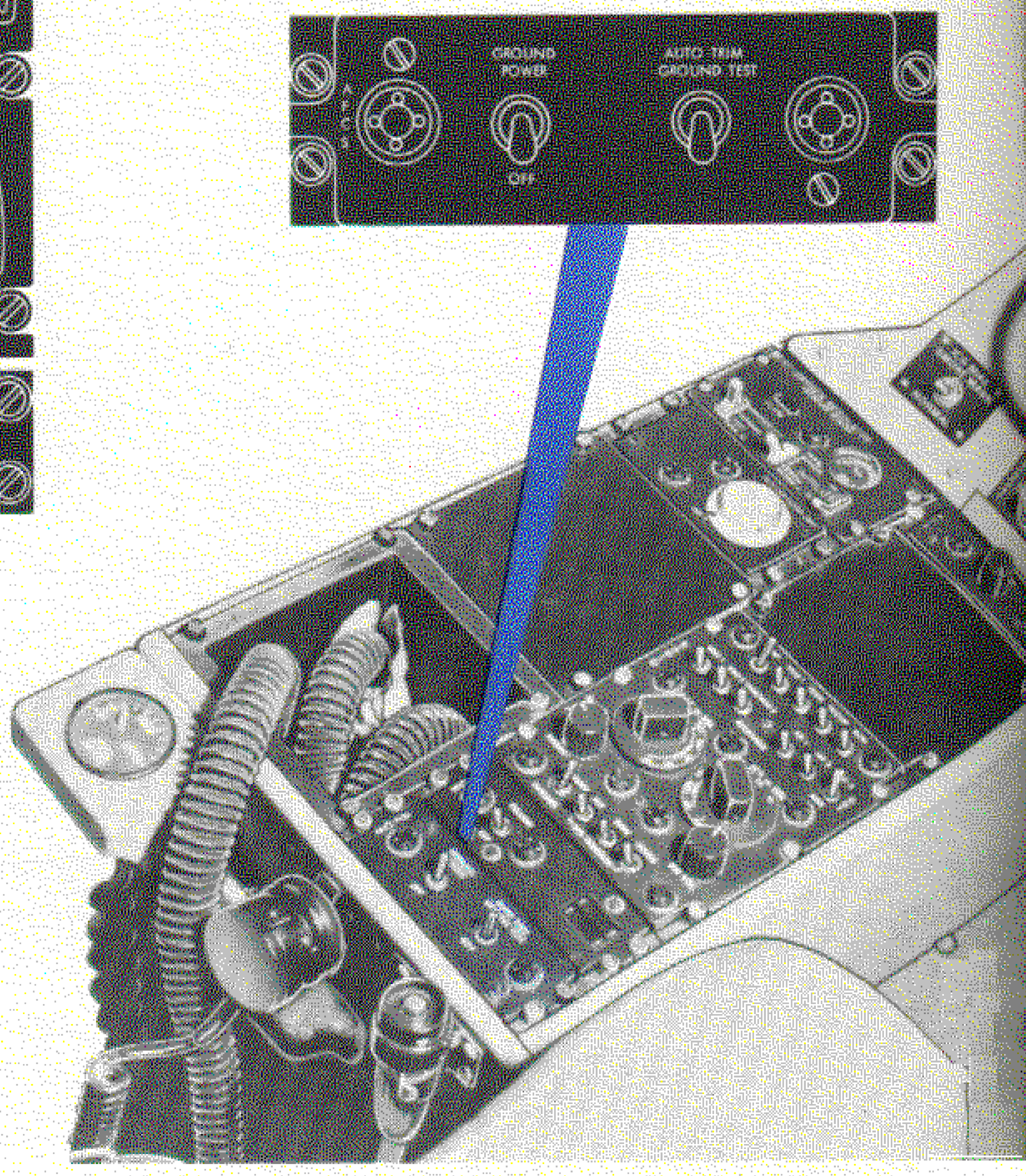


Figure 10 AFCS Ground Test Control Panel

ents are induced by extreme flap extension, which the autopilot, while engaged, will counter with a continual booster force. The "AUTO TRIM" light, then, serves as a reminder that manual setting of the elevator trim tabs is necessary before the autopilot is disengaged.

The AFCS control panel energizes and controls all other modes of the system. Since a more detailed explanation of the various modes is presented in the text covering the operation of the three channels, only the basic functions of the controls are given here.

The manually-operated, solenoid-held ENGAGE switch is self-explanatory in that it engages all three channels of the PB-20 system, provided of course that certain power and signal interlock requirements have been met.

The Preselect Heading switch activates circuitry enabling the pilot to steer the aircraft to an optional heading by rotating the HEADING SET knob which positions the "Selected Heading Marker" on the Horizontal Situation Indicator. The Preselect Heading switch is of the "solenoid-hold-in" variety, and is ineffective unless the autopilot is engaged, the turn control knob is centered, and the channel is not temporarily disengaged by the use of control wheel steering.

The Altitude Hold switch is a normal "OFF" switch, solenoid held in both manually selected positions ("BAR. ALT. HOLD" and "RADAR ALT. HOLD"). Neither position will operate unless the autopilot is engaged prior to selection. The radar altimeter must also provide a valid input logic (reliable signal) before "RADAR ALT. HOLD" can be successfully selected.

The large circular knob on the control panel is the "turn" controller by which the pilot is able to command the autopilot to make a coordinated turn to the left or right. This knob is motor driven to its center (detent) either by aileron channel control wheel steering operation or by autopilot disengagement.

The thumb wheel pitch control knobs on opposite sides of the AFCS Panel are mounted on a common shaft that turns a potentiometer to "set in" a manual nose up or nose down pitch signal to climb or descend. Note, however, that disengaging the Elevator channel as a result of control wheel steering operation or autopilot disengagement will de-energize the pitch centering solenoid, which allows the spring loaded centering lever to return the pitch signal potentiometer to the null position. In addition, to prevent a manual pitch input from interfering with operation in either altitude-hold mode, the thumb wheel knobs are disengaged from the potentiometer — by means of a clutch — when either "BAR. ALT. HOLD" or "RADAR ALT. HOLD" is selected.

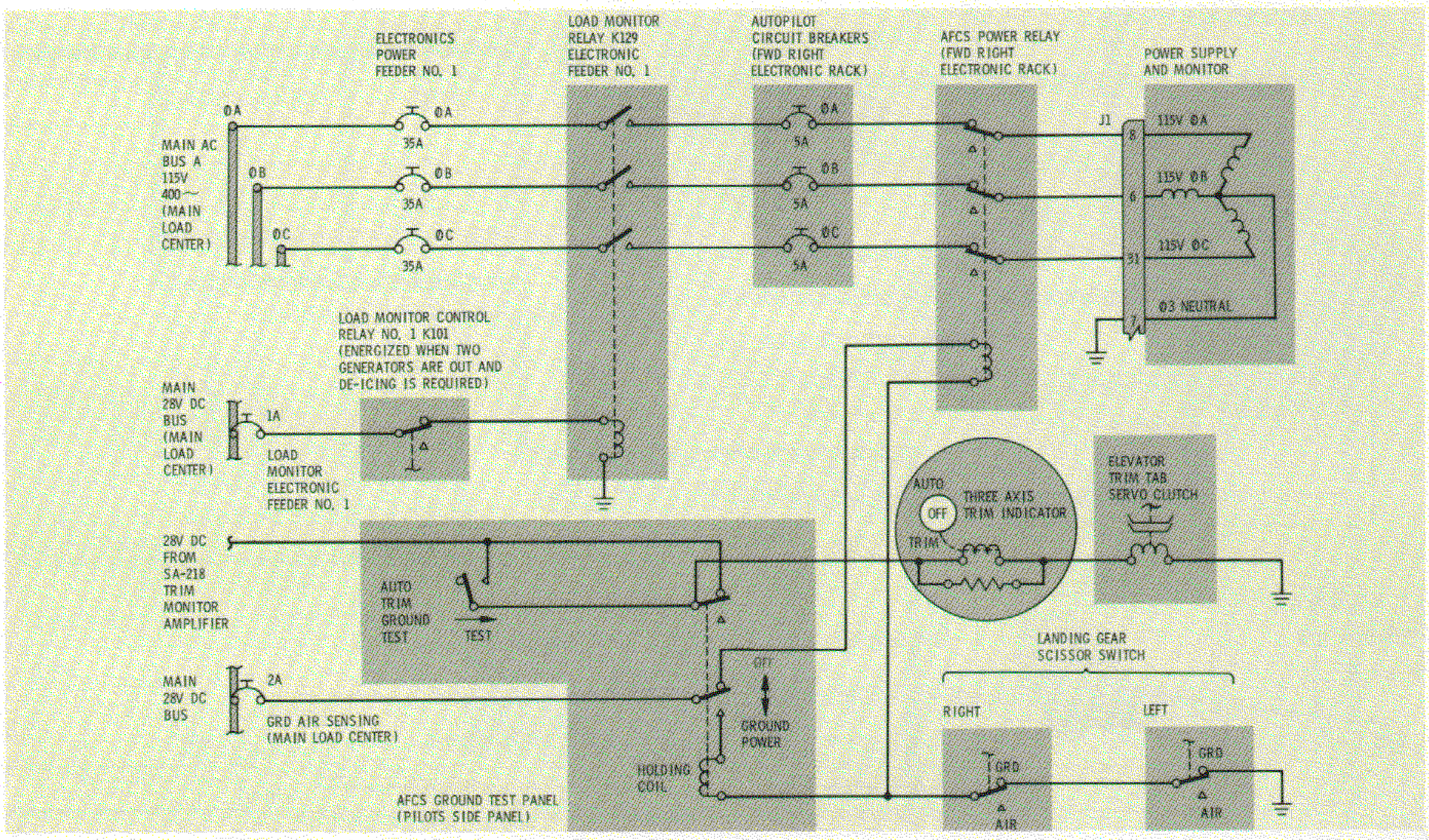


Figure 11 Autopilot Ground Power and Auto Trim Test Circuits

RUDDER CHANNEL

Automatic control is less intricate for the rudder than that for either the aileron or elevator channel. As shown in Figure 12, only six inputs are utilized in the series-connected signal input circuit of the rudder transfer valve amplifiers.

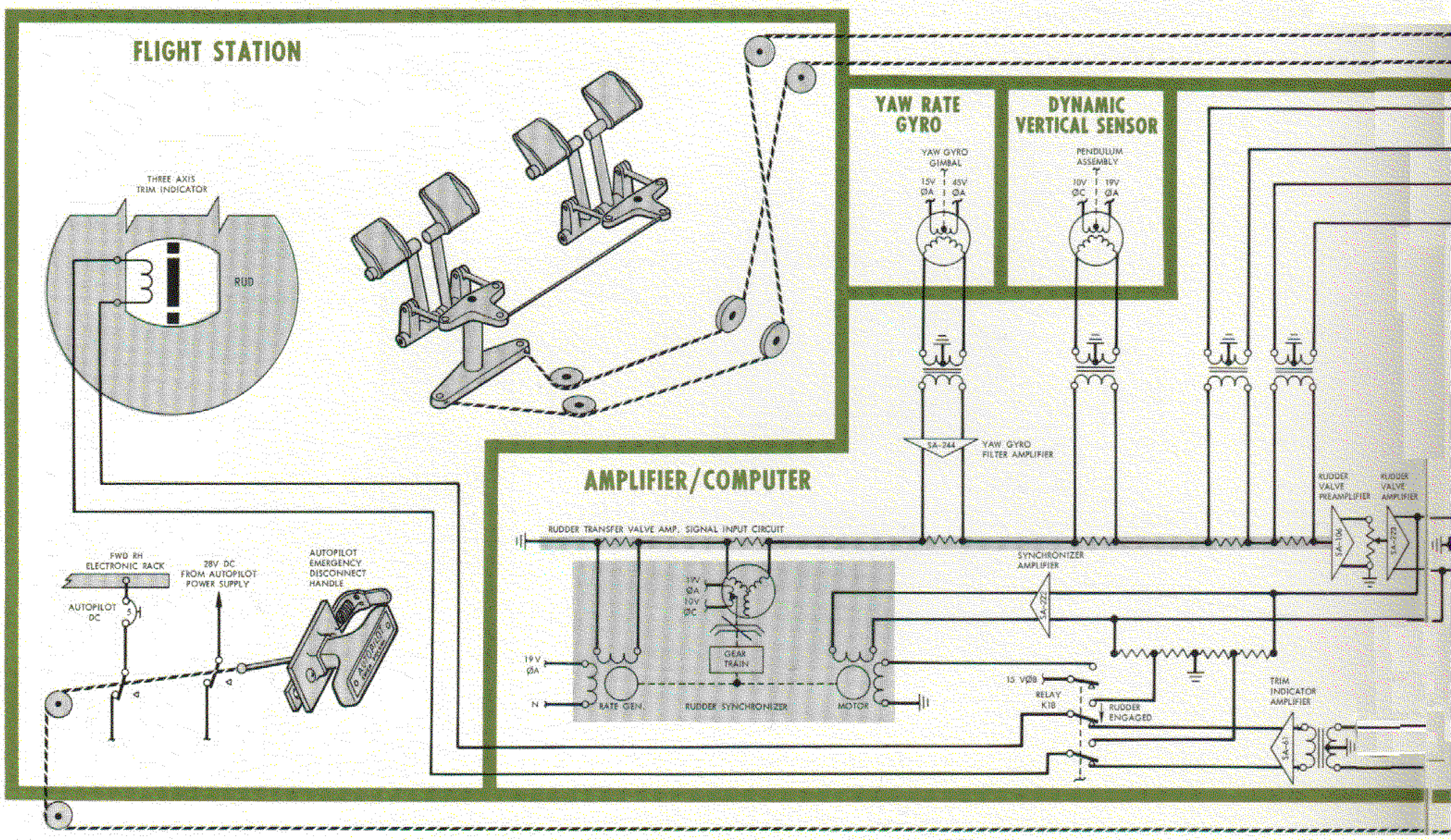
Reading from left to right as they appear in Figure 12 the first two summing resistors in the signal input are fed by outputs from the GL-14 rudder synchronizer. Each of the three autopilot channels is provided with this device to cancel all transfer valve inputs that develop during standby operation before the autopilot is engaged. In this way, unwarranted control surface deflections are avoided as the autopilot assumes control of the boosters.

As shown in Figure 12 the synchronizer amplifier input is connected directly to the rudder valve amplifier output. This loop arrangement ensures that any signal appearing in the signal chain — while the autopilot is in standby — will drive the synchronizer until its output cancels the original input. A portion of the valve amplifier output is also applied to the RUD bar on the three-axis trim indicator where it serves to advise the pilot of the pre-engagement status of the autopilot.* The output of the synchronizer rate generator is utilized to damp synchronizer action and is proportional to motor speed. Its output cancels a portion of the valve amplifier input thus controlling the synchronizer motor speed and minimizing overshoot at the zero signal point.

Relay K18 contacts remove fixed-phase ac power from the synchronizer motor, and the motor stops when the autopilot engages. The synchro output now provides a fixed, engagement-reference signal for the rudder channel. Note also that relay K10 contacts simultaneously transfer the RUD trim indicator input to the rudder boost hydraulic load sensors. Connected in this manner, the sensors continuously measure — and the trim indicator displays — the hydro/mechanical forces imposed on the rudder by the booster. Conversely, the aerodynamic loads on the rudder surfaces are sensed and indicated.

**Any steady state reading on the trim indicator of more than ½-bar width while the autopilot is in "standby" indicates either a failure of the synchronizer in that particular channel or a large, probably faulty, input signal beyond the cancelling capability of the synchronizer.*

Figure 12 Rudder Channel Master Diagram — Autopilot Engaged



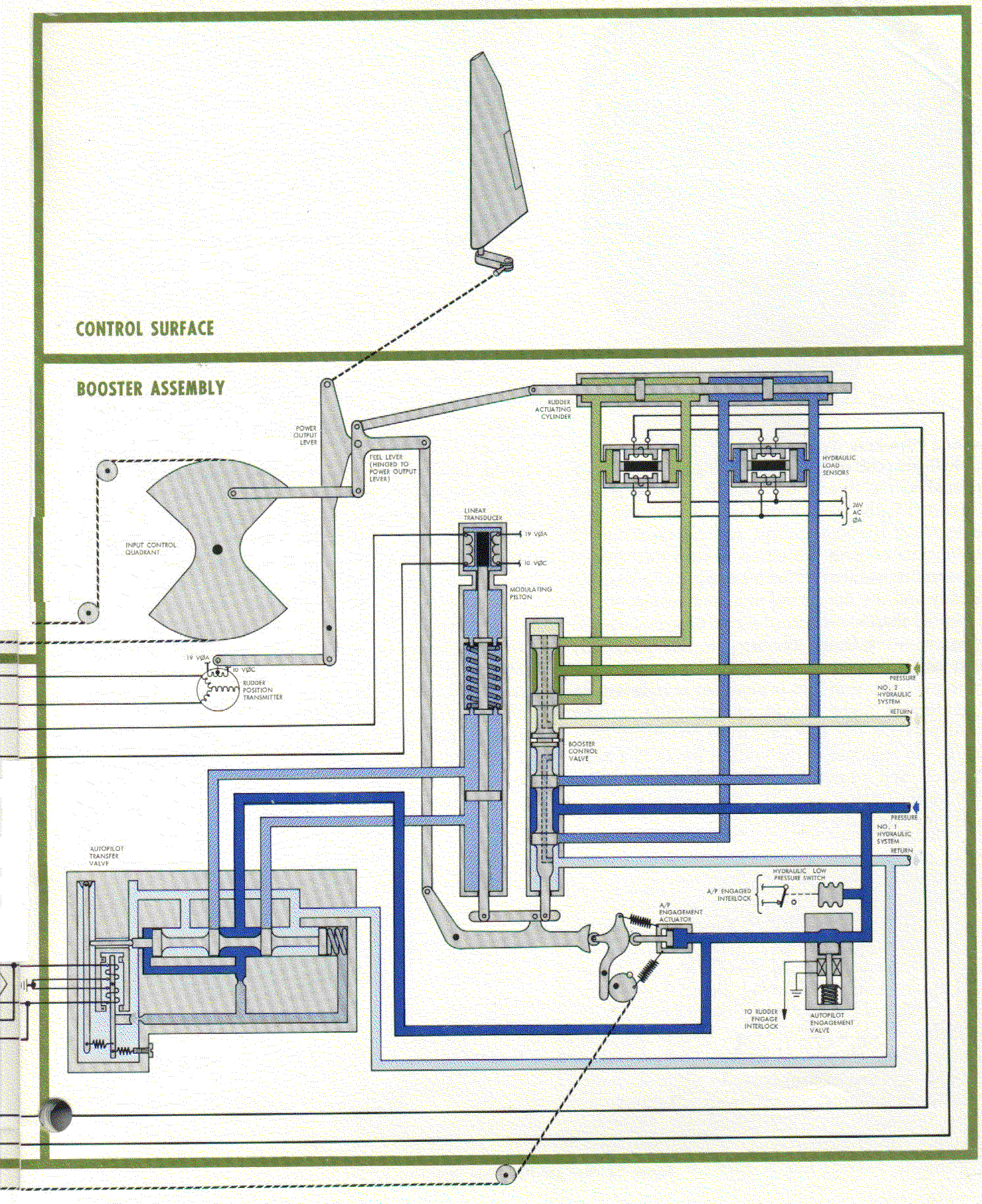

CONTROL SURFACE
BOOSTER ASSEMBLY
POWER OUTPUT LEVER
RUDDER ACTUATING CYLINDER
FEEL LEVER (HINGED TO POWER OUTPUT LEVER)
HYDRAULIC LOAD SENSORS
26V AC ØA
INPUT CONTROL QUADRANT
LINEAR TRANSDUCER
19 VØA
10 VØC
MODULATING PISTON
19 VØA
10 VØC
RUDDER POSITION TRANSMITTER
PRESSURE
NO. 2 HYDRAULIC SYSTEM
RETURN
BOOSTER CONTROL VALVE
PRESSURE
NO. 1 HYDRAULIC SYSTEM
RETURN
AUTOPILOT TRANSFER VALVE
HYDRAULIC LOW PRESSURE SWITCH
A/P ENGAGED INTERLOCK
A/P ENGAGEMENT ACTUATOR
TO RUDDER ENGAGE INTERLOCK
AUTOPILOT ENGAGEMENT VALVE

Figure 13 Three-Axis Trim Indicator

Yaw and Pitch Rate Gyro The yaw gyro portion of this dual purpose unit produces a signal that is proportional to the *rate* of the aircraft yaw. As shown in Figure 15, the Yaw Rate Gyro is mounted in a one-degree-of-freedom gimbal which makes use of the principle of gyro precession to turn the rotor of a synchro transmitter. The air-escapement dashpot assembly and the flat spring mounted on one end of the gimbal shaft determine the rate of gyro response to (and recovery from) aircraft yaw, and practically negate any vibration tendencies of the gimbal.

The yaw gyro spin axis is mounted at right angles to the line of flight and its one-degree-of-freedom gimbal prevents the spin axis from precessing horizontally, (in response to the vertical forces of a bank maneuver) and allows the gyro shaft ends to move in the vertical plane when the gyro senses the horizontal forces of a yaw maneuver.

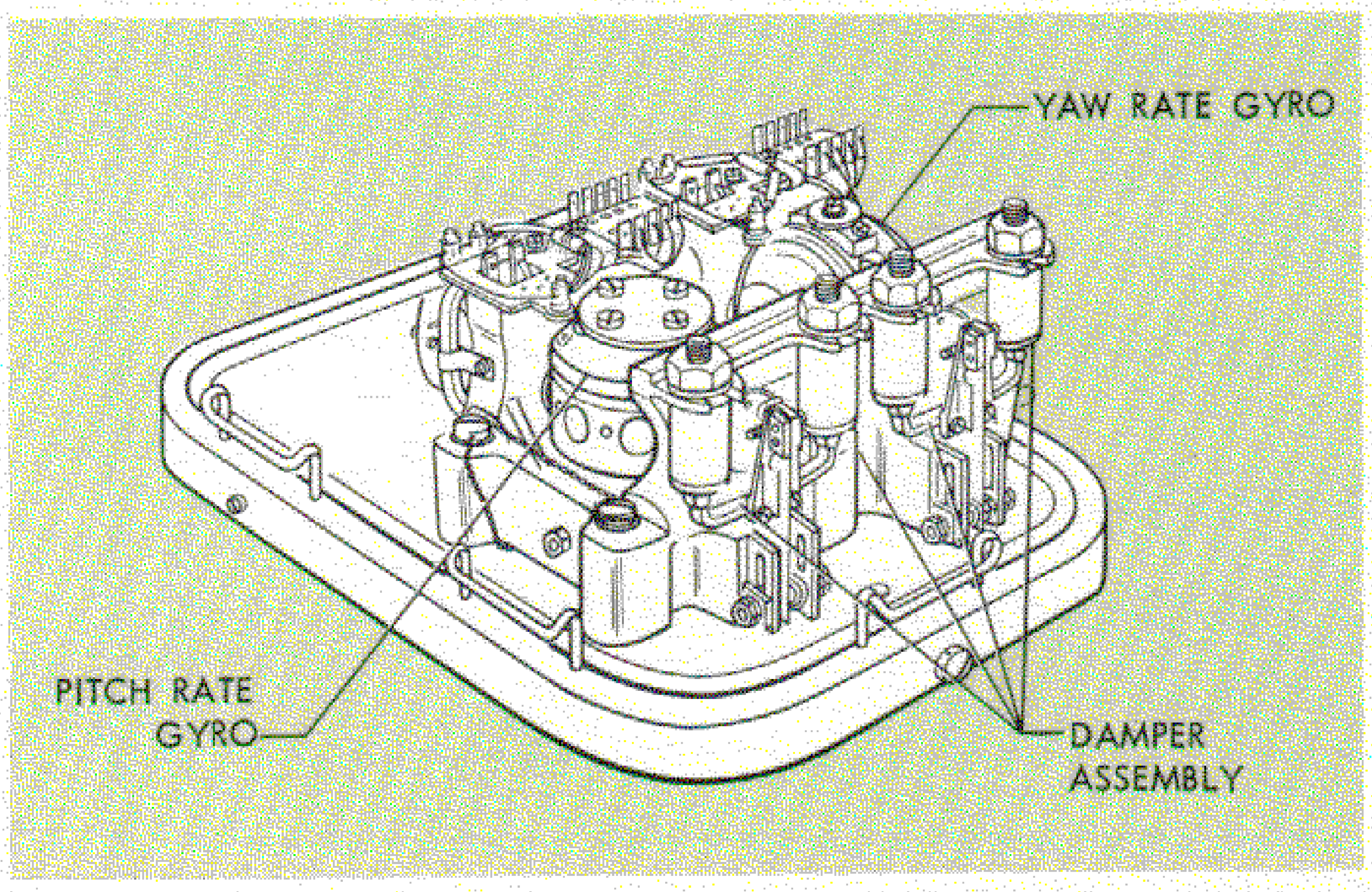
As the gimbal tilts, it turns the rotor of the synchro transmitter and its electrical output is a signal equivalent to the rate of yaw. The signal is applied to the rudder valve amplifier input and instigates a corrective rudder deflection before an excessive yaw angle can develop.

Although the rate gyro responds to all yaw motion of the aircraft, the SA-244 filter amplifier suppresses the steady-state or slowly changing signals that develop in a sustained turn and passes most readily those resulting from a sudden yawing motion, as for example, that induced by a gust of wind. This long-term "washout" feature of the amplifier is necessary to ensure proper rate damping.

Dynamic Vertical Sensor The cut-away drawing of the Dynamic Vertical Sensor in Figure 17 shows the fluid-damped, vertically-suspended pendulum attached to the rotor of a synchro transmitter. The unit is installed with the pendulum axis parallel to the fore and aft axis of the aircraft, and the pendulum responds only to lateral accelerations such as side slip or skid.

In flight, side slip — travel towards the center of the turning arc — will occur as the aircraft turns with insufficient rudder application to coordinate the turn rate with the bank angle. A skid — motion away from the center of the turning arc — results when the aircraft turns with too much rudder application for the rate of turn. In either case, the vertical sensor detects the steady state transverse drift and applies a corrective signal to the rudder signal chain. Effectively, the dynamic vertical sensor measures the combined effect of gravity plus the centrifugal force of the turn and signals for a rudder correction that adjusts the turn rate until the pendulum returns to a centered position (perpendicular to the floor of the aircraft).

The two remaining inputs to the rudder signal chain are feedback or follow-up signals. Both devices



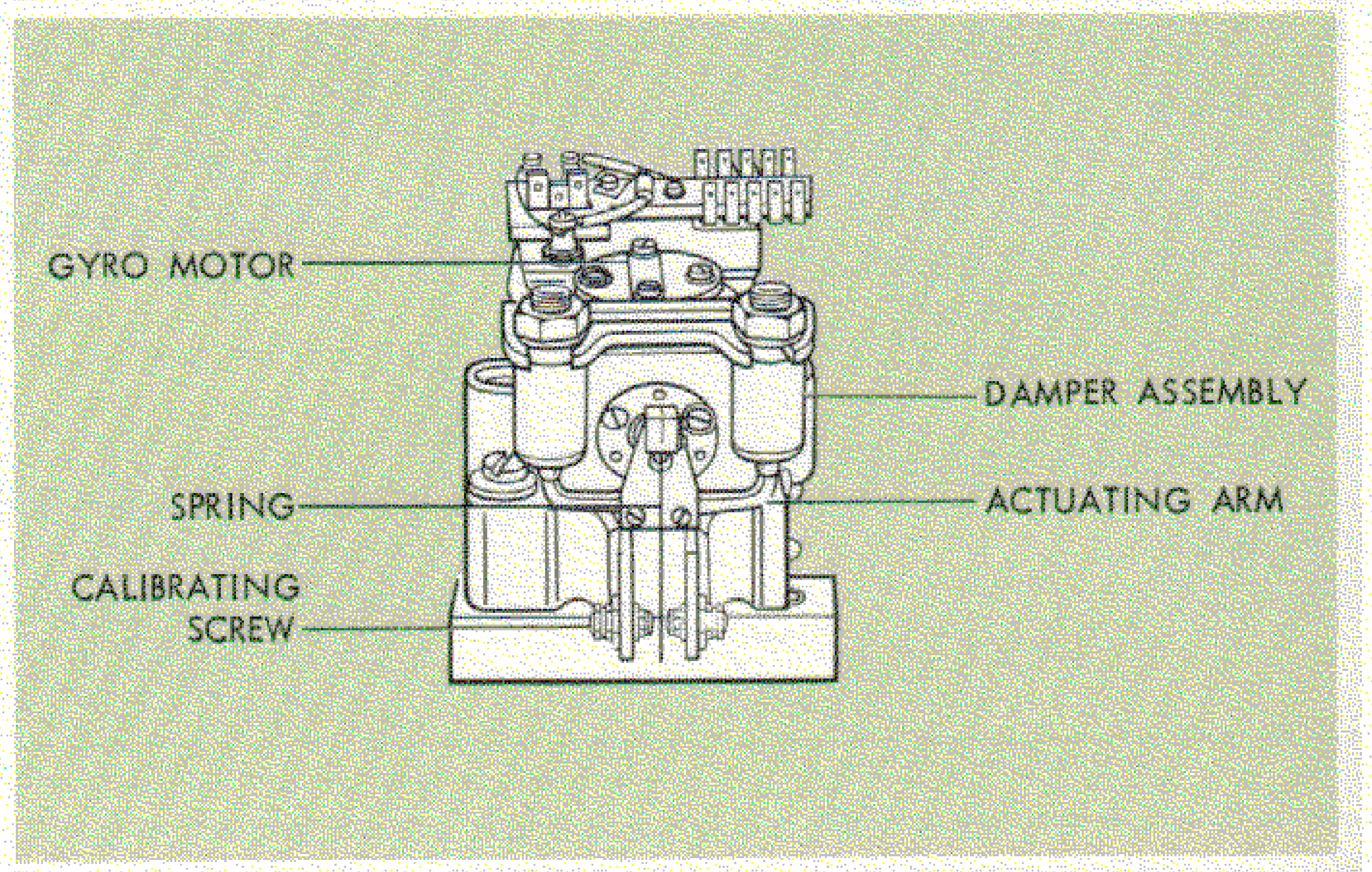


Figure 14 Yaw and Pitch Rate Gyro

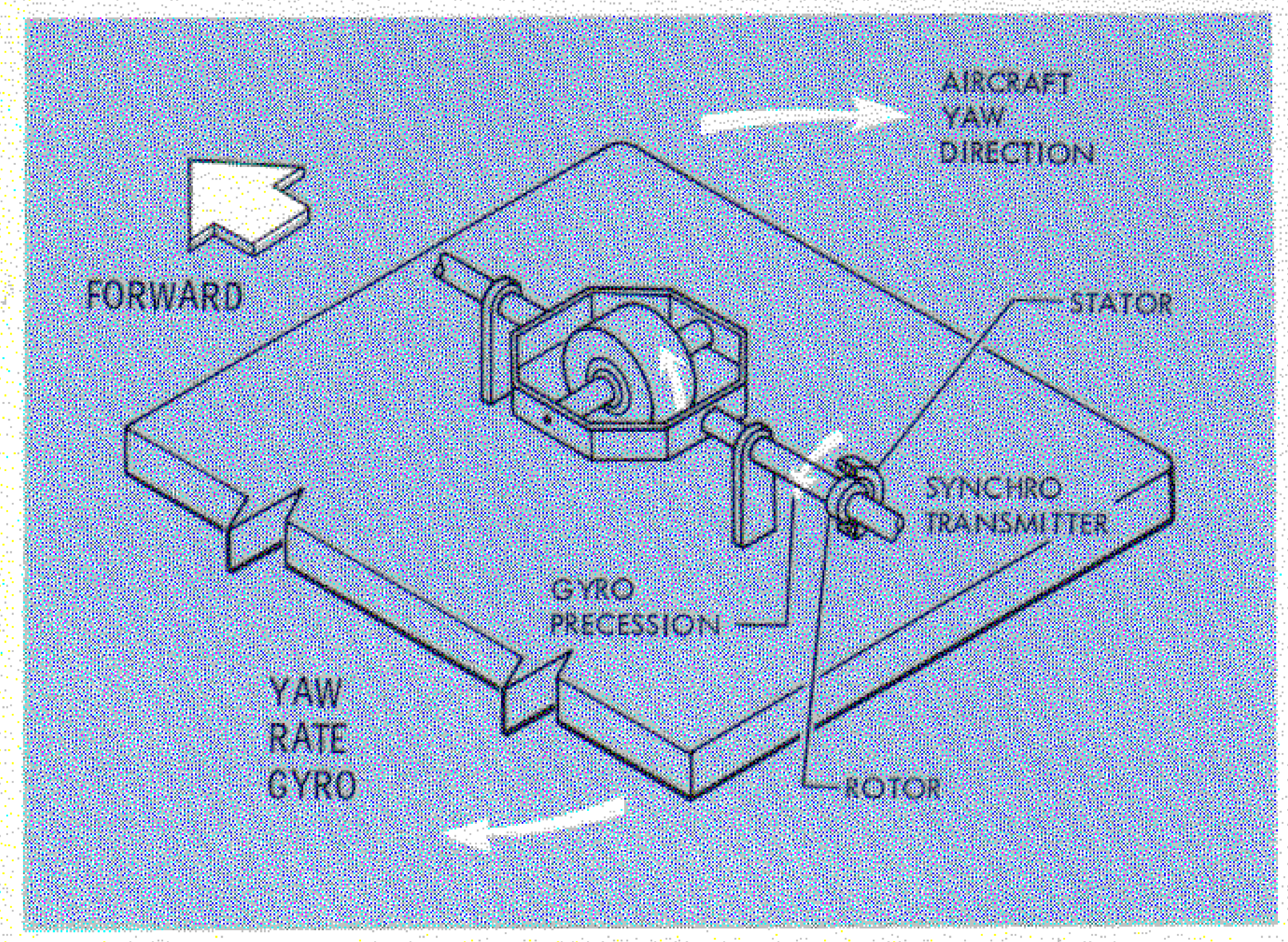


Figure 15 Yaw Rate Gyro Gimbal Rotation Resulting from Aircraft Clockwise Yaw Motion

that furnish these inputs — the modulating piston linear transducer and the rudder position transmitter — appear on the rudder boost package in the same relative position as that shown in Figure 2.

The linear transducer core is attached to, and moves with, the modulating piston. As the piston is displaced by the transfer valve pressure changes, the core moves within its coils and generates a pickoff signal which is applied to the rudder signal chain. This signal opposes and partially cancels the auto-

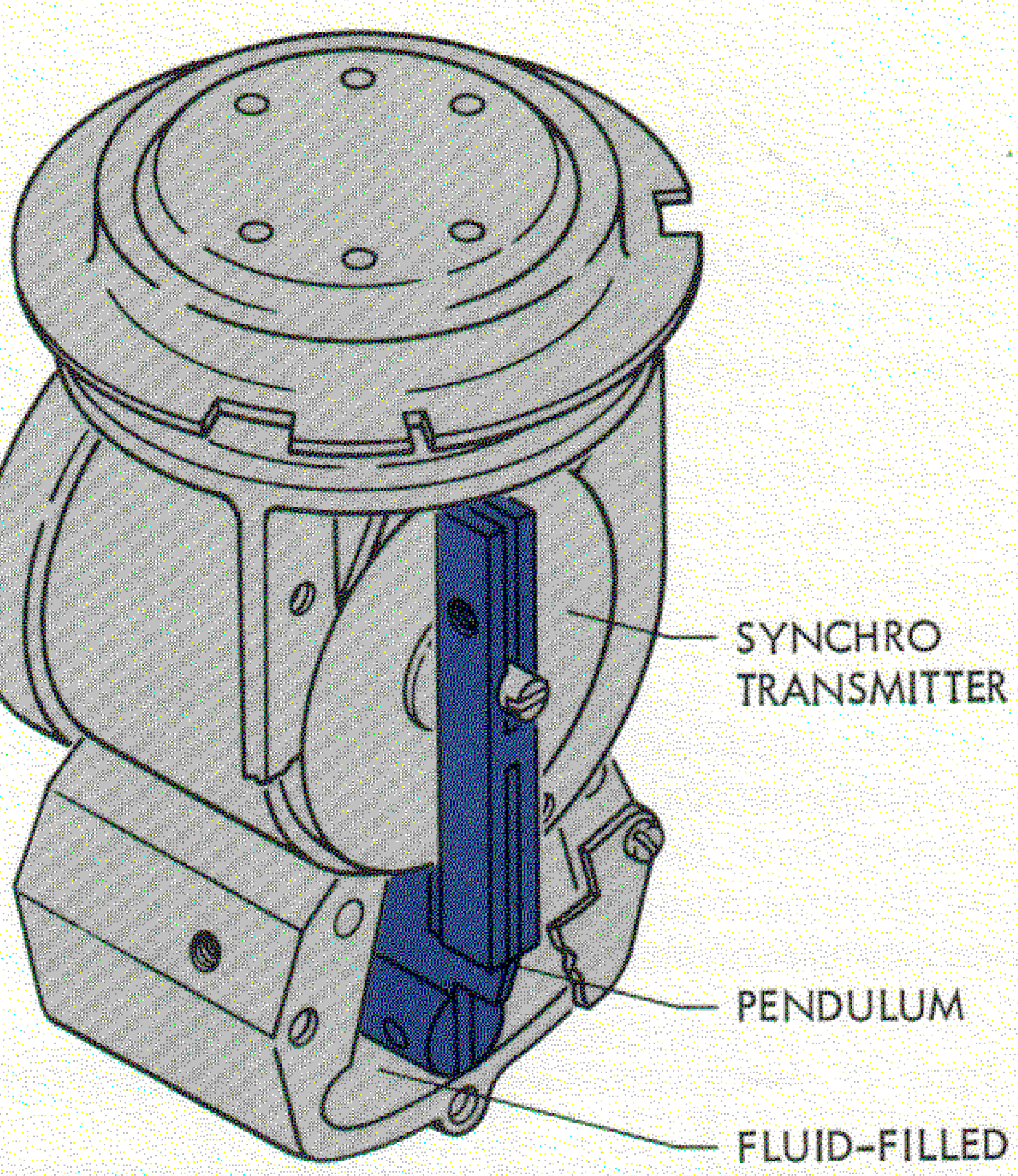


Figure 17 Cutaway View of Dynamic Vertical Sensor

pilot command to the transfer valve amplifier, and, as mentioned previously, determines the *rate* of pressure change that is ultimately applied to the booster cylinder.

The Rudder Position Transmitter rotor is actuated by a mechanical coupling link attached to the booster power arm. As the arm moves, it displaces the rotor as well as the rudder and the electrical output of the transmitter is a direct measure of control surface displacement. This displacement signal is applied to the signal chain in opposition to the initiating signal to prevent over-control of the rudder and serves to return the rudder to a trimmed position as the original error signal decreases to zero. The coupling arm is spring-loaded in one direction to prevent mechanical "backlash" — due to wear at the link joints — that would result in erratic feedback signals.

Automatic control of the rudder system can be summed up by the following statement: Rudder signals will be initiated and the rudder displaced only when an uncoordinated turn causes side slip or skid or a sudden air disturbance initiates an aircraft yaw condition. No provision is made for inserting a long-term command to the autopilot rudder circuit as is the case in the aileron and elevator channels.

Figure 16 Dynamic Vertical Sensor

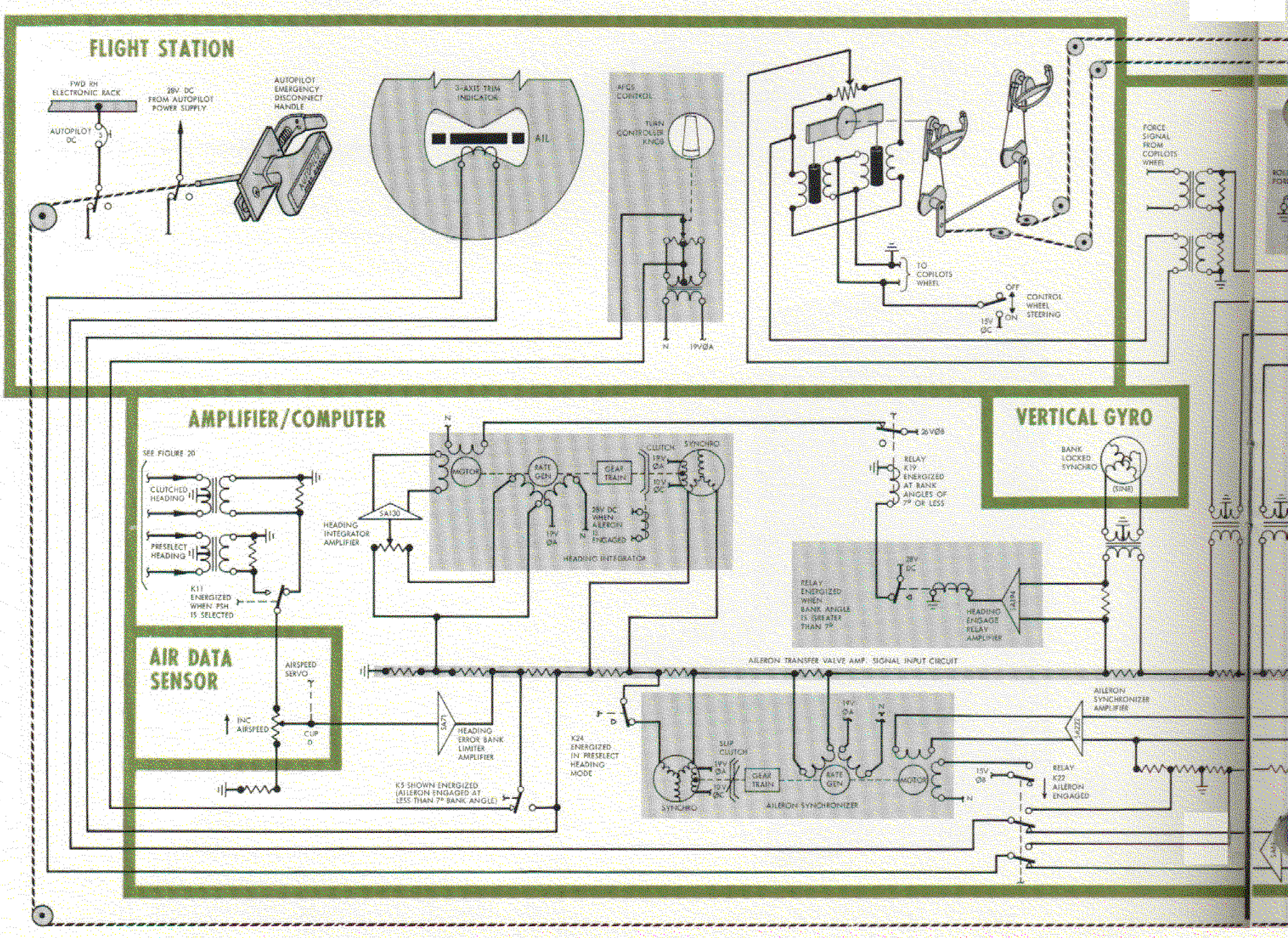
AILERON CHANNEL

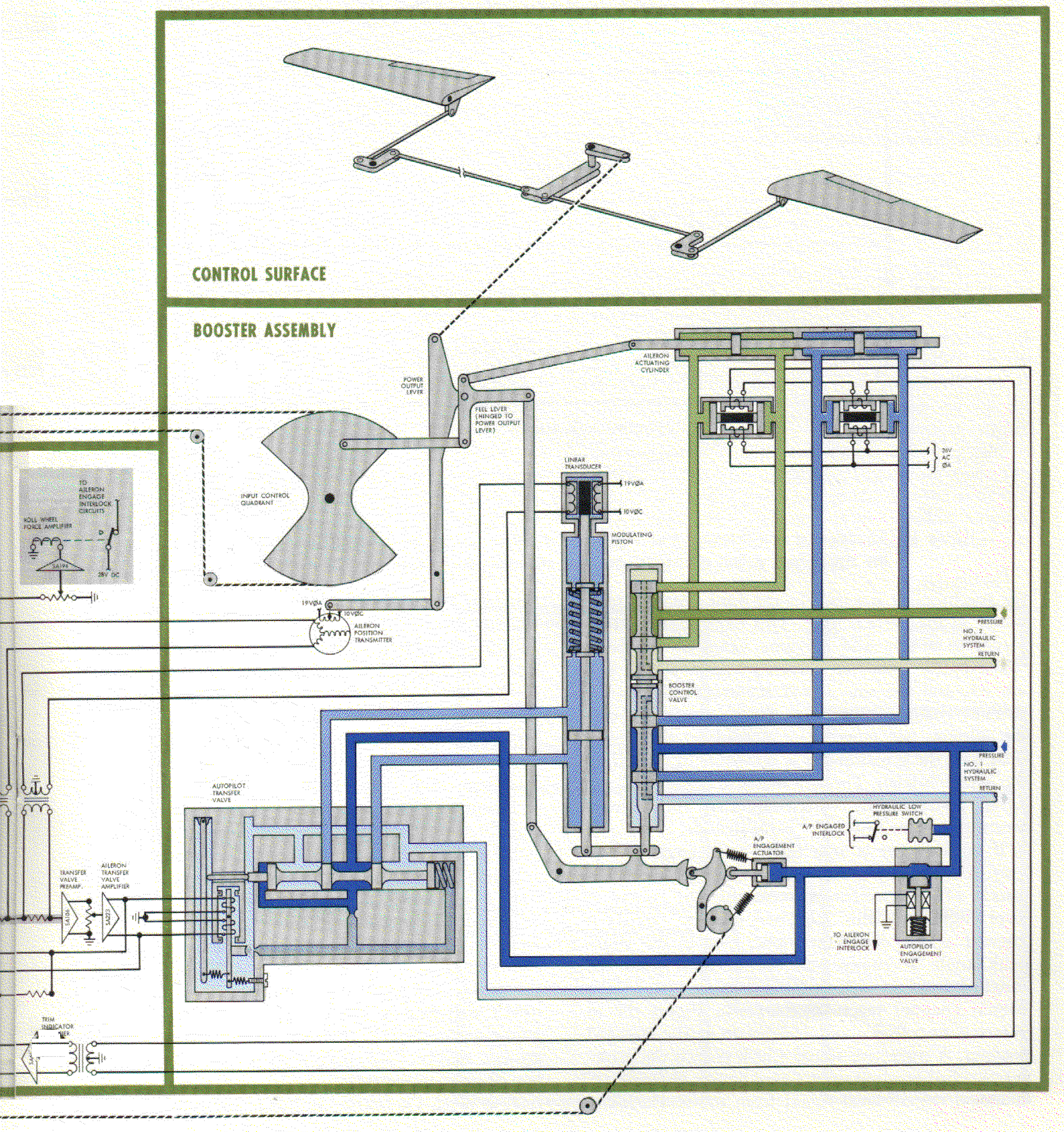
Automatic control of the Aileron channel provides the aileron action required to attain and maintain both the magnetic heading and the aircraft attitude. However, as shown in Figure 18, the amplitude of the heading signals (and the resulting aileron displacement) is modified by the Air Data Sensor as a function of air speed. In addition, the heading error bank limiter amplifier (SA71) prevents the autopilot from inducing a bank angle in excess of 25° to eliminate heading errors.

With the autopilot energized but prior to engagement, the vertical gyro generates signals indicative of aircraft bank, but the aileron synchronizer functions in the same manner as that discussed for the rudder, i.e., its synchro output continually counters this input signal (up to a value equivalent to a 45° bank angle) and no significant signal reaches the aileron transfer valve amplifier. This averts the possibility of a sudden aileron displacement at the time the autopilot is engaged. Coincident with aileron engagement, the 15V ϕ B power is removed from one winding of the synchronizer motor by relay K22 contacts and thereafter its synchro output remains fixed as an operating reference indicative of the bank angle at which the ailerons are to be returned to trimmed position.

If the autopilot is engaged with the aircraft at any bank angle between approximately 7° and 45°, a continual turn is obviously intended, and no heading signal is provided. Any random bank attitude excursion away from the operating reference stored in

Figure 18 Aileron Channel Master Diagram — Autopilot Engaged





CONTROL SURFACE
BOOSTER ASSEMBLY
POWER OUTPUT LEVER
FEEL LEVER (HINGED TO POWER OUTPUT LEVER)
AILERON ACTUATING CYLINDER
TO AILERON ENGAGE INTERLOCK CIRCUITS
ROLL WHEEL FORCE AMPLIFIER
28V DC
INPUT CONTROL QUADRANT
LINEAR TRANSDUCER
19VØA
10VØC
MODULATING PISTON
26V AC ØA
AILERON POSITION TRANSMITTER
PRESSURE
NO. 2 HYDRAULIC SYSTEM
RETURN
BOOSTER CONTROL VALVE
PRESSURE
NO. 1 HYDRAULIC SYSTEM
RETURN
AUTOPILOT TRANSFER VALVE
HYDRAULIC LOW PRESSURE SWITCH
A/P ENGAGED INTERLOCK
A/P ENGAGEMENT ACTUATOR
TRANSFER VALVE PREAMP.
AILERON TRANSFER VALVE AMPLIFIER
SA223
TO AILERON ENGAGE INTERLOCK
AUTOPILOT ENGAGEMENT VALVE
TRIM INDICATOR

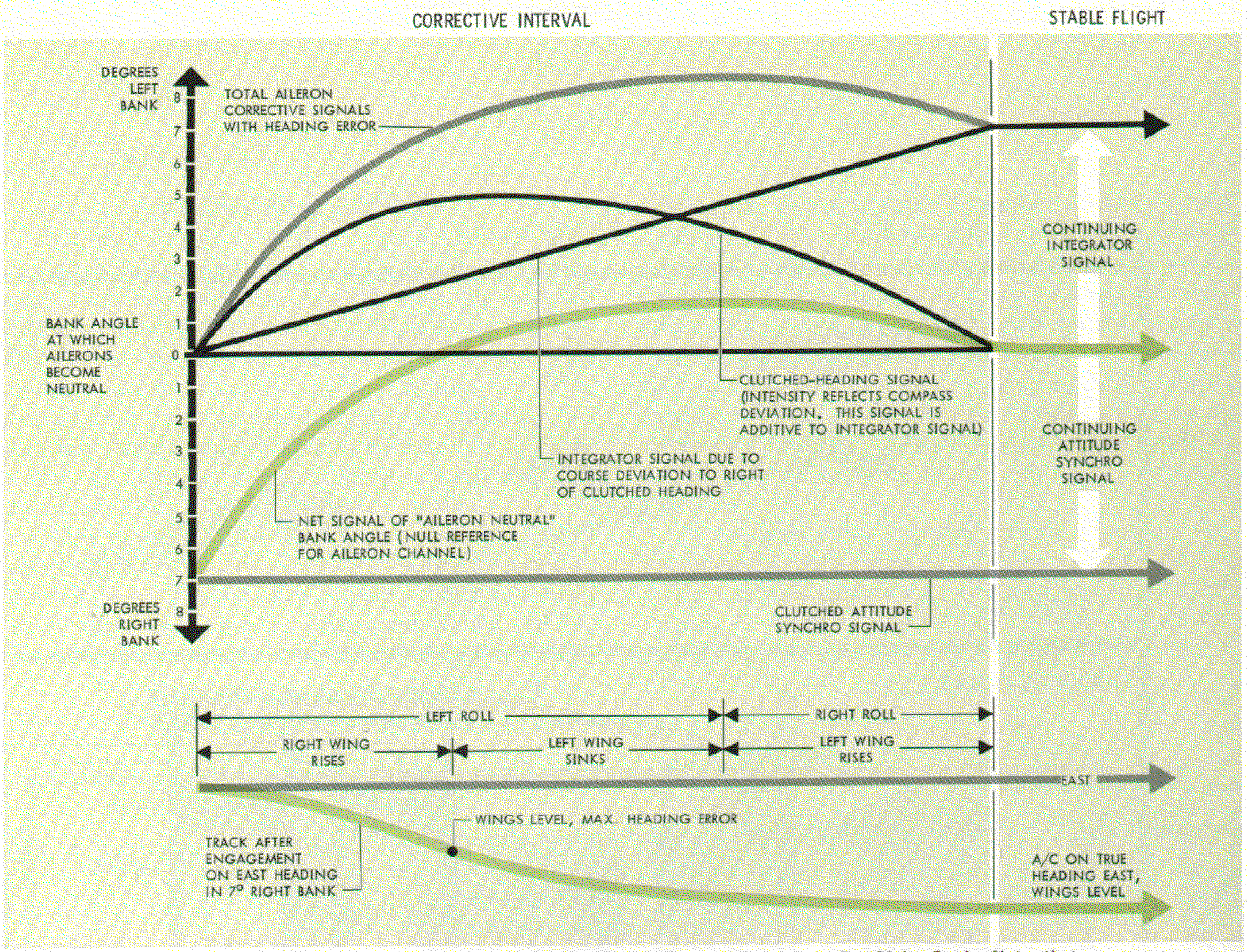


Figure 19 Adjustment of Null Reference After Engagement of Aileron Channel at 7° Right Bank. Note that corrective signals from vertical gyro result when bank angle does not coincide with "net signal" reference index.

the synchronizer is sensed by the *bank* synchro transmitter in the vertical gyro. This bank signal is converted to a sine and a cosine value by the bank "locked-rotor" resolver, shown schematically in Figure 31, and the sine signal is introduced into the Aileron channel to counter the excursion. In case the ailerons are engaged at a bank angle greater than 45°, the synchronizer is incapable of cancelling the entire signal. The uncancelled portion of the signal automatically commands that the ailerons return the aircraft to a 45° bank attitude.

If engagement is at a lesser bank angle (7° or less), it is presumed that the aircraft heading at the instant of engagement is to become the course in level flight, and this "clutched heading" signal is inserted as the dominant input. Due to the slight bank angle at engagement, a deviation from the clutched heading will ensue immediately, and the aileron channel reacts as shown in Figure 19 to induce a smooth roll-out and return to the engaged heading. A corrective signal which reflects the growing clutched heading error signal begins immediately to cancel the off-level signal from the synchronizer synchro. The heading error signal also drives an integrator motor which produces a second, steadily growing corrective signal to aid in leveling the aircraft. The aircraft will roll to level rapidly as a result of the concerted corrections, but note that the heading error will be maximum when the wings come level, and the integrator signal will continue to grow, causing the aircraft to continue its roll past level, and bank back towards the clutched heading.

As the aircraft comes back towards clutched heading, the corrective signal which reflects heading error will diminish, of course, but the integrator signal will continue to grow enough to sustain a slight bank as long as *any* heading error exists. Eventually, the net amplitude of the combined corrections will diminish, and the aircraft will roll-out level when its heading coincides with the clutched heading. No heading error signal will exist, of course, and the integrator motor will stop in the position at which the integrator signal exactly counters the off-level engaged attitude signal. Thus, both signals remain, but their net effect is to neutralize the ailerons when the aircraft is level. Of course, deviation between the null references and the aircraft attitude after engagement will produce a corrective signal at the aileron transfer valve. This has occurred repeatedly in the course of the adjustment period, (with the result that the right wing tip followed a trajectory similar to the curve plot of null-reference) and will recur thereafter if random bank excursions are sensed. Any tendency to veer off the selected heading is immediately cancelled by a momentary excursion from the heading null reference, which elicits immediate counteracting signals from the vertical gyro.

Preselect Heading (PSH) is an alternate mode for control of the ailerons and can be utilized even after the autopilot is engaged at high bank angles (7° to 45°). The signal is inserted into the aileron channel through an airspeed attenuator potentiometer (cup D) in the same manner as the clutched heading signal. However, the PRE-SEL HDG switch will not hold in the engaged position prior to aileron engagement and will be de-energized if the turn controller knob is rotated from its center detent position. Relay K24 is also operated by the PSH switch and its contacts disconnect the synchronizer reference signal. Power (26v ϕ B) to the heading integrator amplifier and its motor is removed by roll out relay K19 if a bank angle greater than 7° is required to effect a turn to the new heading. This prevents an undesirable buildup of integrator output during long sustained turns.

The pilot can set the desired heading on the Horizontal Situation Indicator (HSI) with the HEADING SET knob either before or after selecting "PRE-SEL HDG". As depicted in Figure 20, this turns a control transformer in the HSI which modifies the magnetic heading signal thereby causing the heading signal to the autopilot to "rotate" to the selected point.

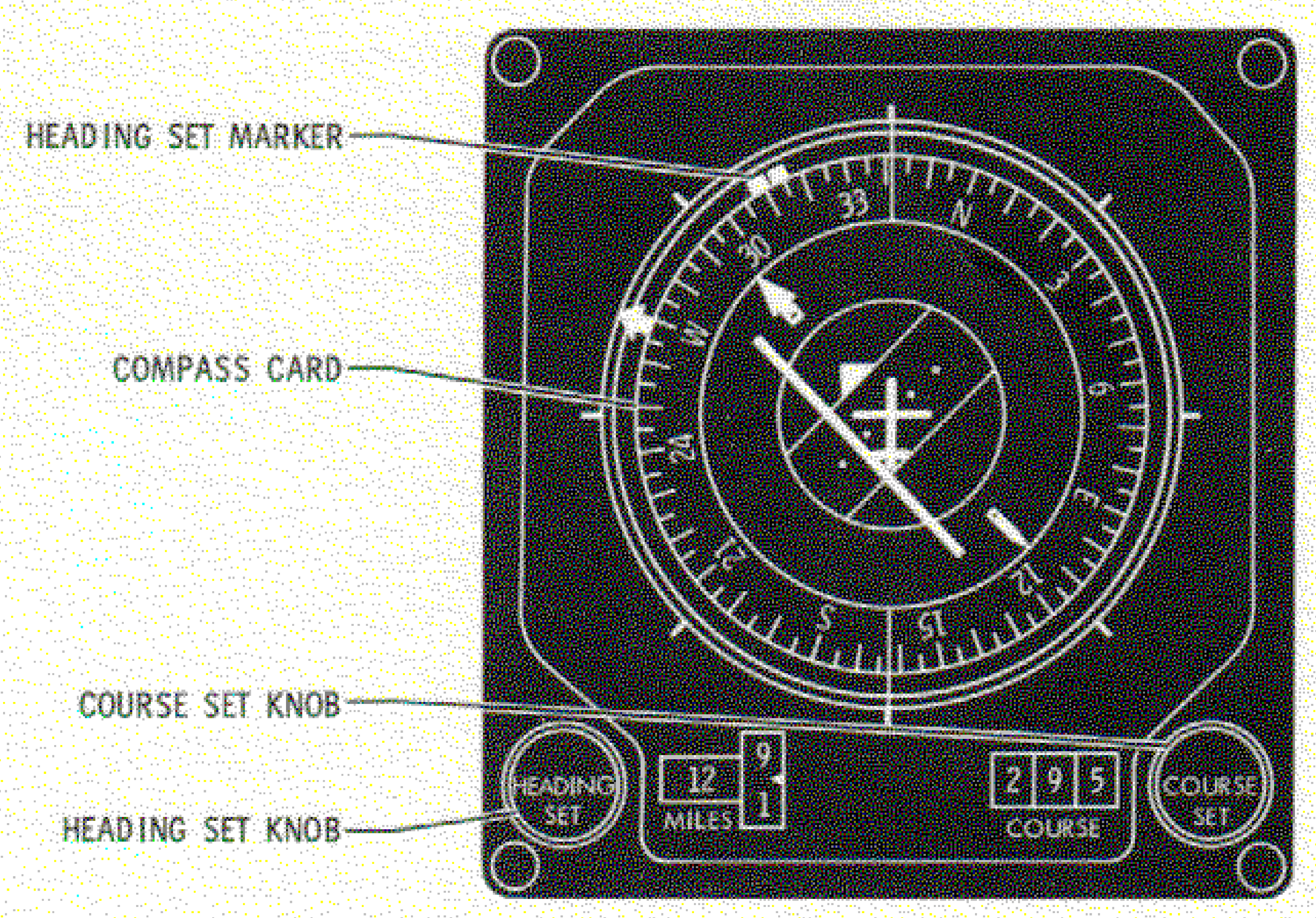
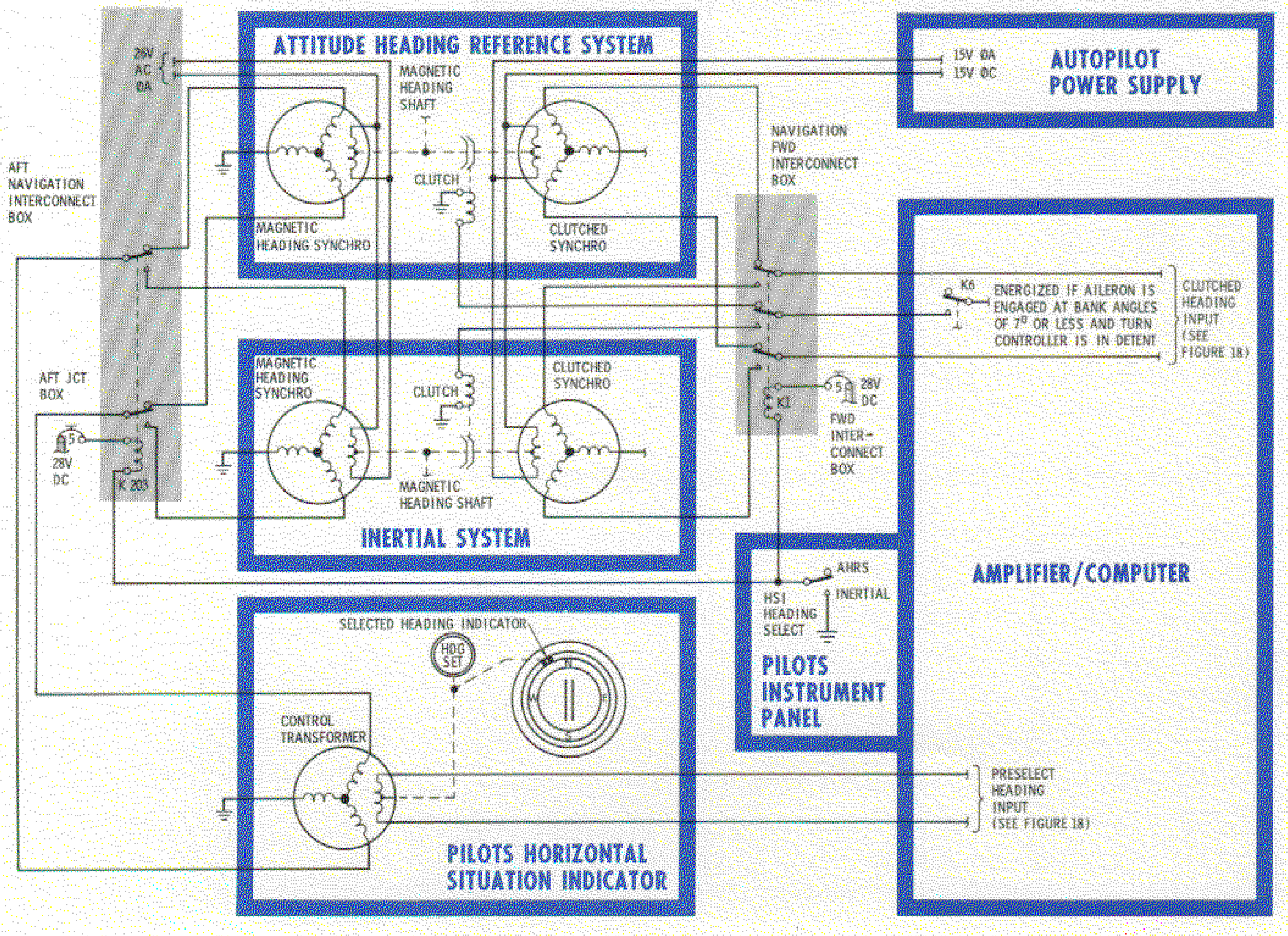


Figure 21 Horizontal Situation Indicator

Both the preselect and the clutched heading signals shown in Figure 20 originate either in the Attitude Heading Reference System (AHRS) or in the Inertial System, whichever is selected by the pilot's HSI HDG switch. In both systems the synchro that supplies the preselect heading is coupled directly to, and rotates with, the magnetic heading shaft. The "clutched" heading synchro of the selected system is driven only when the clutch is energized by an autopilot "engaged" 28 V dc signal and is otherwise spring-held in a null position.

Figure 20
Clutched and Preselect
Heading Schematic



The turn controller can be utilized when the aileron is engaged and the bank angle is 7° or less. When the knob is rotated from its center position it operates a detent switch that disconnects clutched heading, or if PSH mode is engaged, it de-energizes the preselect heading switch and also turns a potentiometer that inserts a left or right turn command signal (see Figure 22). Small angular displacements of the potentiometer arm about the mid-position produce a slow-rise voltage output to provide smooth entry of the aircraft into a turn attitude. Subsequently, the magnitude of the signal increases in value per degree of knob rotation. A reversible motor drives the knob and the potentiometer through a solenoid-operated clutch and recenters the control if the autopilot is disengaged or if the control wheel steering mode is utilized for aileron manipulation.

The coordinated turn induced by the turn controller involves all three control surfaces to some extent. For this reason we digress briefly to point out the significant aspects involved. Part of this discussion involves control of the elevator and will be discussed in greater detail in the ELEVATOR CHANNEL section.

The following chain of events transpire during an autopilot-controlled turn: When the Pilot sets the turn control knob to right of neutral, he has required the autopilot to fly the airplane at an angle of bank which we will assume to be 45° right.

The immediate effect is to introduce a signal for "right aileron" hinge moment at the aileron booster, and as the aircraft banks right, the vertical gyro supplies a counter-signal which reduces the initial signal to zero (ailerons faired) when the required bank angle is achieved.

While the ailerons were displaced from neutral, the mechanical inter-connect between the aileron and rudder control system will introduce a mild "right-rudder" hinge moment, and this hinge moment too is eliminated when the bank angle is achieved and ailerons return to neutral. It should be noted that the Dynamic Vertical Sensor discussed in the RUDDER CHANNEL section may supplement or detract from this mechanically-induced rudder application if the aircraft starts to slip or skid.

As the vertical gyro begins to detect the roll motion, a versine signal is generated which is modified according to the existing airspeed and grows continuously until the aircraft stabilizes at the selected angle of bank. The versine signal is introduced into the elevator booster as a requirement for "elevator-up" hinge moment. As the elevator responds, its displacement is sensed by the position transmitter, and the hydraulic force imbalance at the boost actuator runs the elevator trim tab actuator towards "nose-up" trim. The elevator position transmitter will offset the versine signal when the elevator reaches an "up" position which is predetermined to be correct for the P-3 at the existing bank angle and airspeed. The trim actuator will continue to run until the tab has provided sufficient aerodynamic hinge moment at the elevator to relieve the hydraulic force differential at the boost actuator.

The control wheel steering switch can be actuated at any time but is normally selected after the autopilot

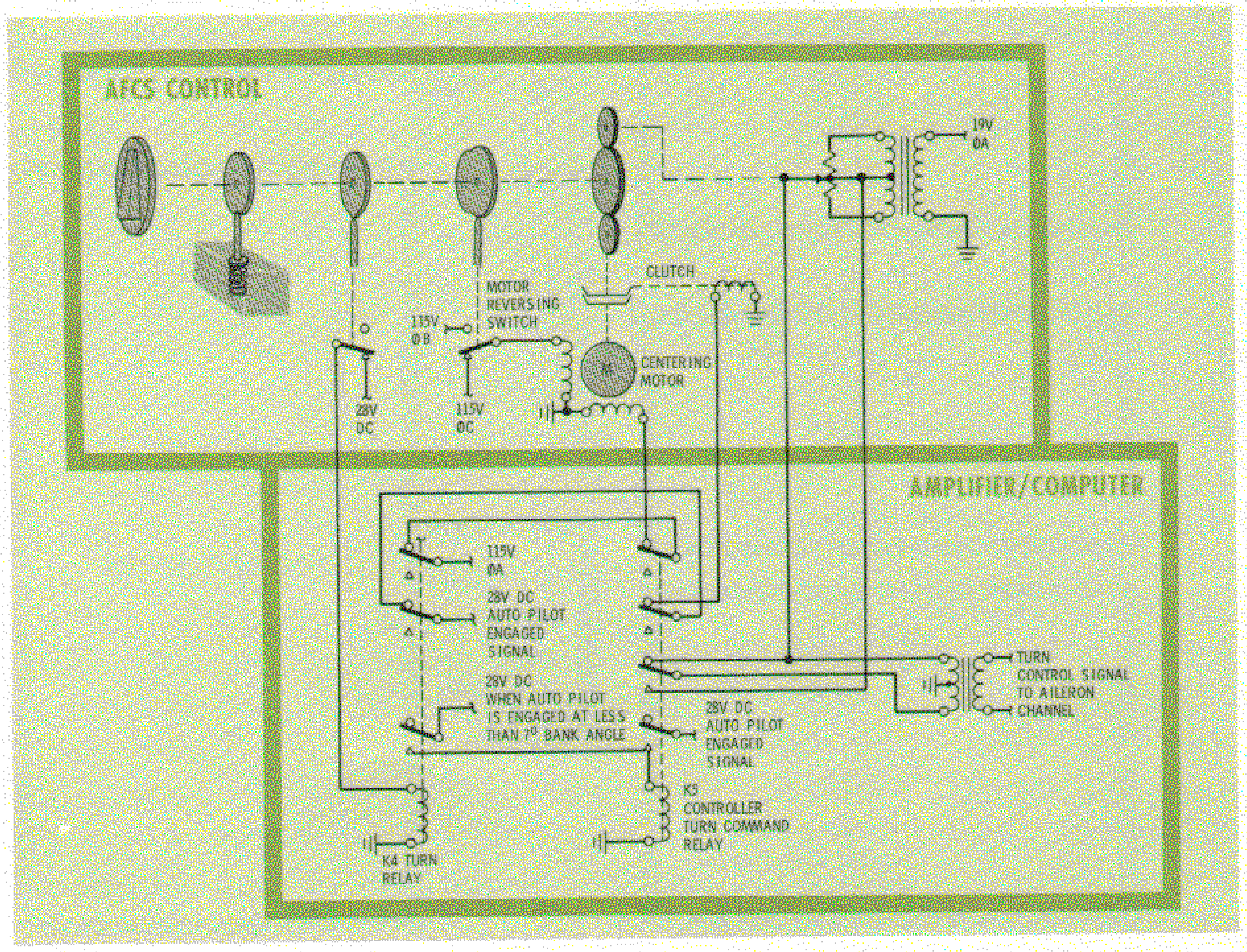
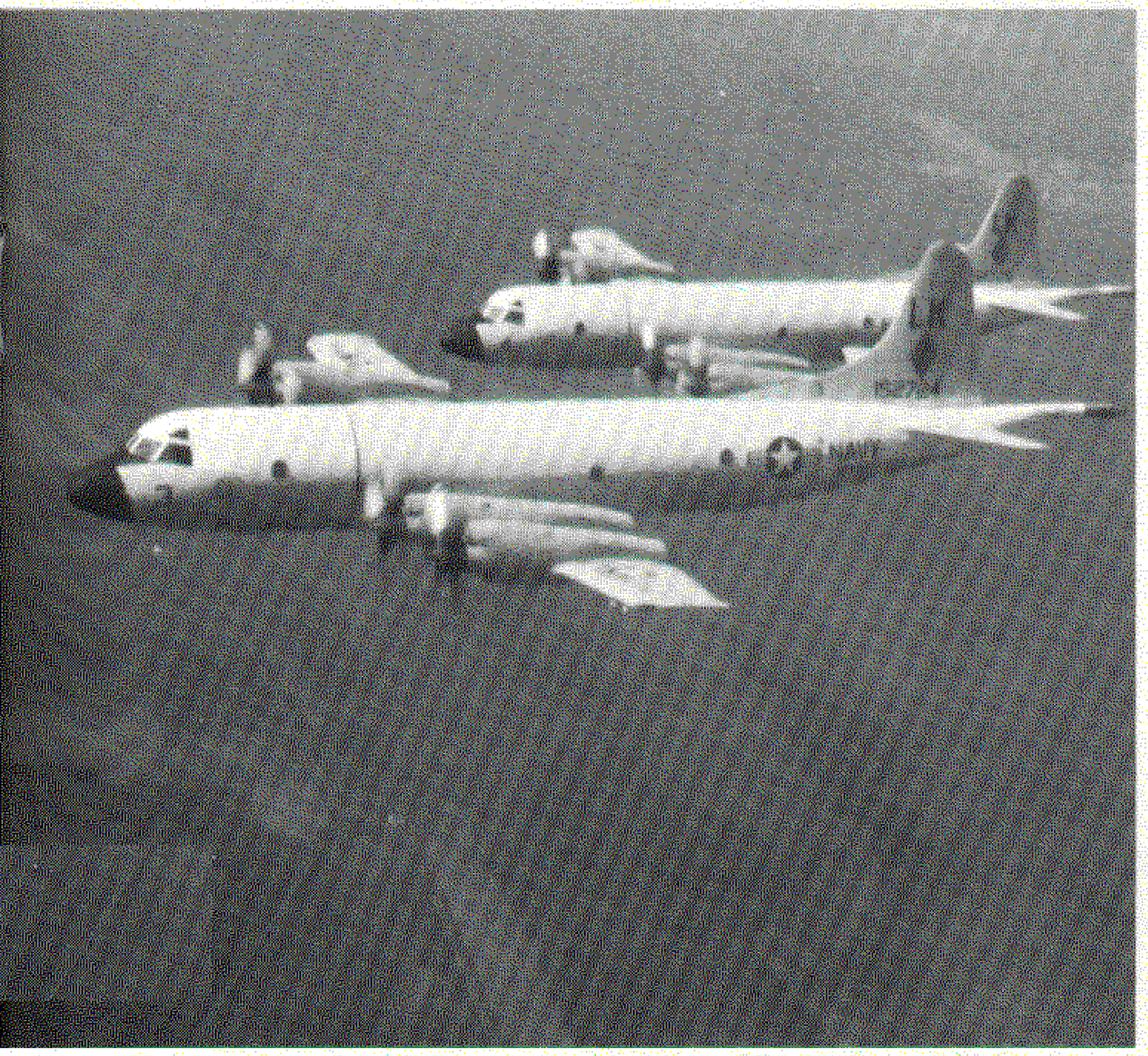


Figure 22
Turn Controller
Electro/Mechanical Diagram

is engaged and the pilot or copilot elects to manipulate one flight control surface manually without disturbing automatic control of the other two channels. The only function of the switch is to apply 15 V ac excitation to both control wheel force sensors. When the autopilot is engaged and the pilot or the copilot exerts approximately a 1½-pound rotational force on the control wheel, the force sensor applies enough signal to the "roll" wheel-force amplifier (SA-194) to energize a relay at the amplifier output. This relay disengages the aileron channel for the duration of the force application without disturbing the engaged status of the other two channels. Thus, the ailerons actually revert to boost-on manual control while force is applied to the wheel. In the event the PSH switch is engaged, it will be de-energized when force is applied to the wheel.

Note that the synchronizer is reactivated during control wheel steering maneuvers. Re-engagement when the control wheel is released is essentially the same as that at initial engagement, that is, the ailerons will re-engage and hold any bank angle (between 7° and 45°) existing when the wheel is released or the clutched heading will assume control and cause the aircraft to roll out and hold the engaged heading if the wheel is released below a 7° bank.

Inasmuch as the linear transducer, the aileron position transmitter, and the aileron bar on the three-axis indicator are interconnected in the same manner and serve essentially the same purposes as those discussed for the rudder, no additional discussion is given here.



ELEVATOR CHANNEL

Automatic operation of the AFCS elevator channel is in many respects similar to the other two control surfaces just discussed. However, the nature of an ASW mission involving both long range, high speed, high altitude flight as well as low speed, low altitude loiter establishes a need for an automatic altitude hold system to promote both efficiency (fuel conservation) and safety. Implementation of the altitude-hold provisions quite naturally adds complexity to the elevator channel.

To fulfill these requirements, automatic operation of the elevator channel is controlled in several ways: Engaged pitch attitude, manipulation of the AFCS control panel pitch thumbwheels, barometric altitude hold, radar altitude hold, and control wheel steering. Except for operation in the control wheel steering mode (which is essentially manual control), the Pitch Limiter Amplifier (SA-71) shown in Figure 23 restricts the total command signal in any of these modes to a value equivalent to 25 degrees of pitch.

The autopilot goes initially into the engaged pitch angle mode and pitch attitude deviations are sensed by the vertical gyro pitch synchro. As shown in Figure 31 this pitch signal is converted to sine and cosine values by the pitch locked-rotor resolver and the sine output determines the pitch attitude in this mode. Air currents or other disturbances that cause an altitude change without disturbing the pitch synchro will be uncorrected by the autopilot. This may be unimportant from a safety standpoint at high altitude but it does affect fuel economy and thus the maximum range of the aircraft. To alleviate this condition, the barometric hold mode utilizes electrical signals generated by the static pressure portion of the Air Data Sensor to sustain the assigned altitude.

The opposite end of the operating spectrum (low altitude ASW maneuvers) requires an even closer altitude tolerance to ensure an adequate safety margin. This is provided in the radar altitude hold mode by combining the better short term characteristics of the sensed barometric altitude with the long term stability of the sensed radar altitude. Although barometric pressure variations are generally insignificant over a short distance, they may be too large in more widely separated areas, especially at low altitude over water, for use as the "prime" signal in low-level altitude-hold operation. On the other hand, although the radar altimeter signal may fluctuate rapidly as the reflected beam strikes large wave troughs in passage over rough seas, the mean of the reflected radar signals will be consistent with the actual altitude.

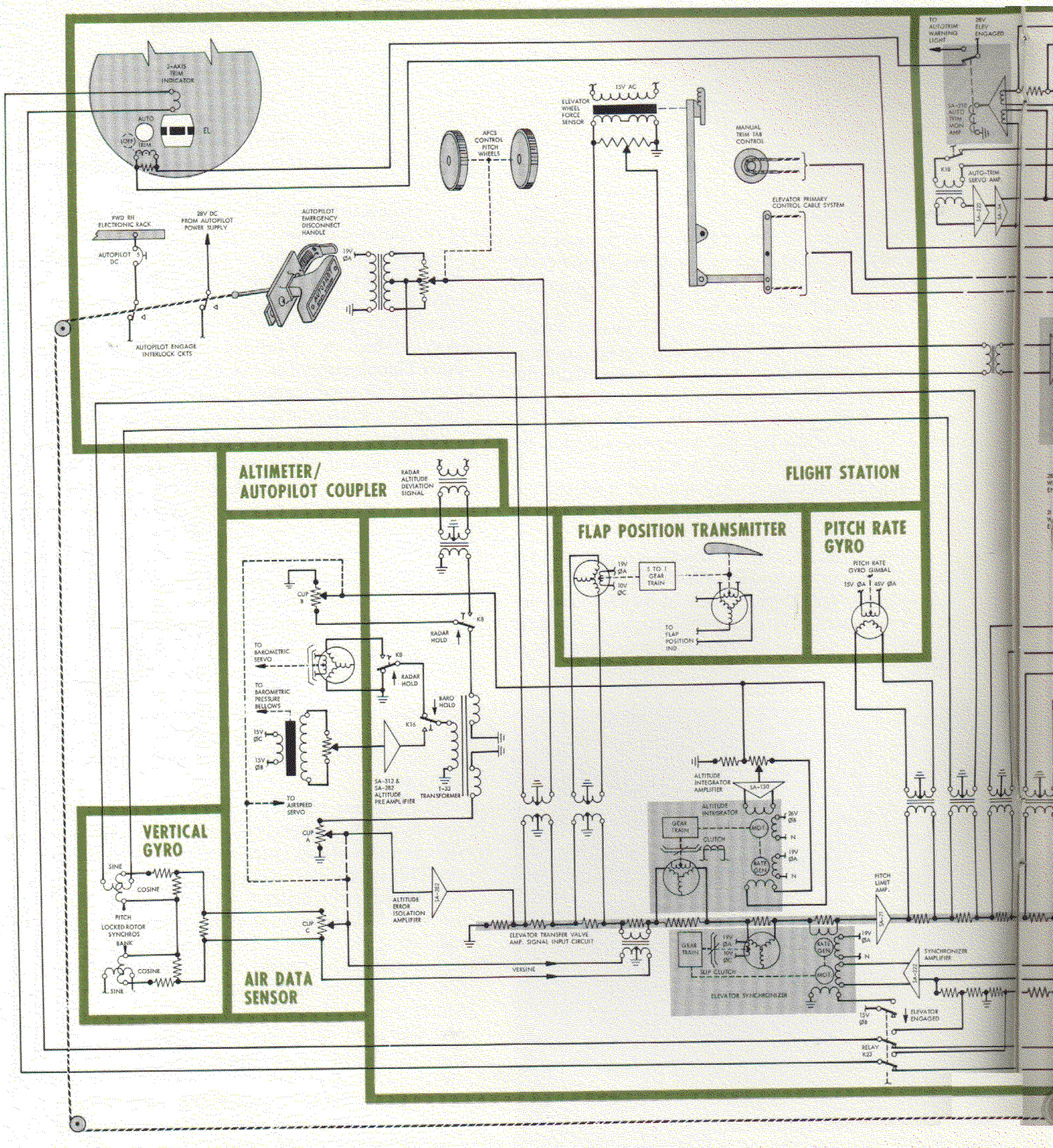
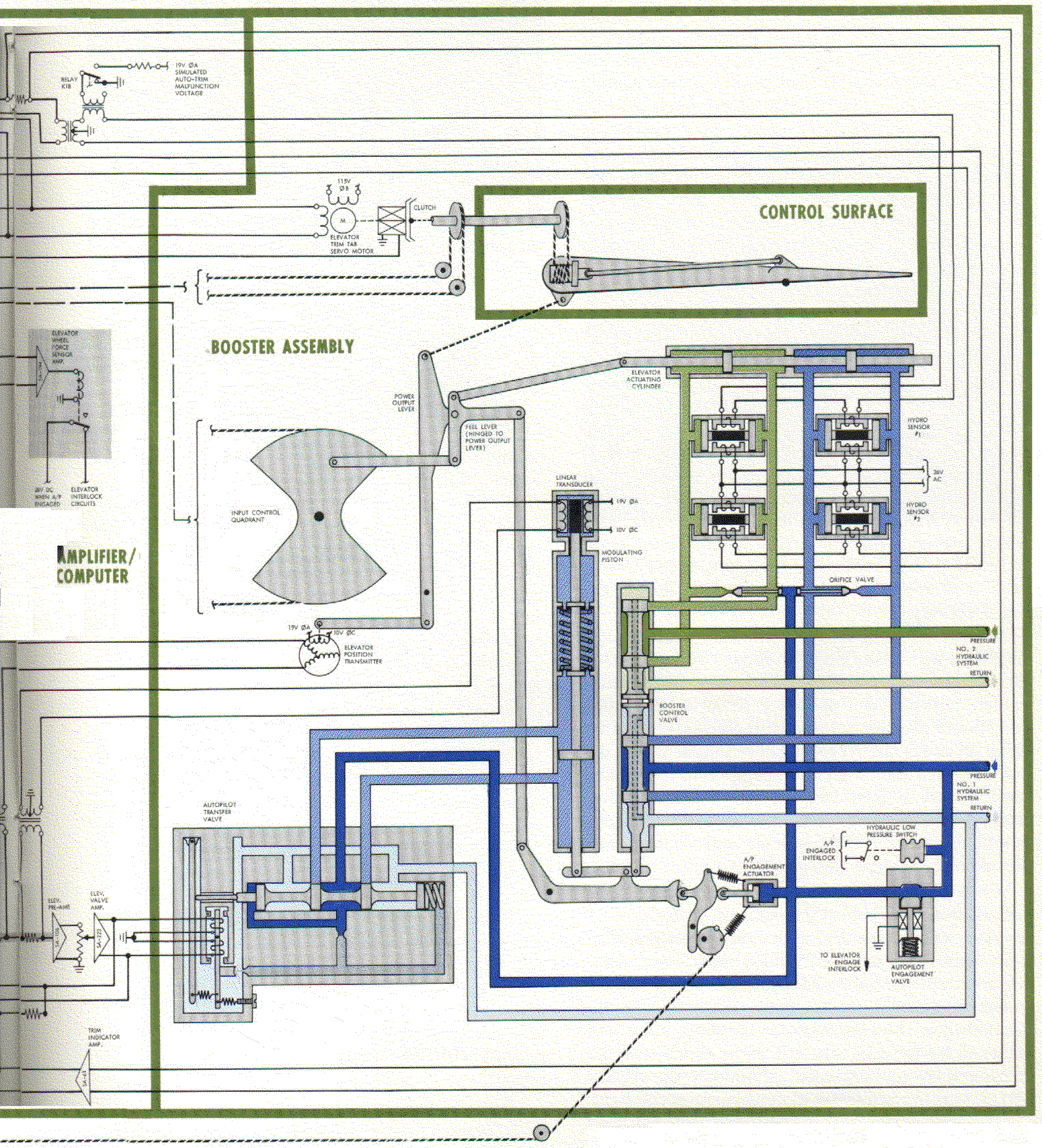


Figure 23 Elevator Channel Master Diagram — Autopilot Engaged


CONTROL SURFACE
BOOSTER ASSEMBLY
AMPLIFIER/ COMPUTER
RELAY K1B
19V ØA SIMULATED AUTO-TRIM MALFUNCTION VOLTAGE
115V ØB
CLUTCH
ELEVATOR TRIM TAB SERVO MOTOR
ELEVATOR WHEEL FORCE SENSOR AMP.
28V DC WHEN A/P ENGAGED
ELEVATOR INTERLOCK CIRCUITS
POWER OUTPUT LEVER
FEEL LEVER (HINGED TO POWER OUTPUT LEVER)
ELEVATOR ACTUATING CYLINDER
HYDRO SENSOR #1
HYDRO SENSOR #2
28V AC
INPUT CONTROL QUADRANT
LINEAR TRANSDUCER
19V ØA
10V ØC
MODULATING PISTON
ORIFICE VALVE
ELEVATOR POSITION TRANSMITTER
PRESSURE
NO. 2 HYDRAULIC SYSTEM
RETURN
BOOSTER CONTROL VALVE
PRESSURE
NO. 1 HYDRAULIC SYSTEM
RETURN
AUTOPILOT TRANSFER VALVE
HYDRAULIC LOW PRESSURE SWITCH
A/P ENGAGED INTERLOCK
A/P ENGAGEMENT ACTUATOR
ELEV. PRE-AMP.
ELEV. VALVE AMP.
TO ELEVATOR ENGAGE INTERLOCK
AUTOPILOT ENGAGEMENT VALVE
TRIM INDICATOR AMP.

The automatic pitch trim system is activated when the autopilot is engaged and automatically repositions the elevator trim tabs in the direction necessary to relieve elevator hinge moment forces. These forces develop, for example, as fuel is consumed, armament stores are released or personnel move about the aircraft, and the autopilot commands a compensating elevator change to maintain the proper pitch attitude.

The elevator command creates an imbalance in pressure across the No. 2 hydraulic load sensor and the resulting signal drives the trim tab servo (see Figure 23) until the elevator is "in trim." This relieves the booster of the necessity of maintaining a constant pressure in one direction to hold the elevator in a given position.

The 3-axis trim indicator provides a continual indication of booster forces, and the pilot can manually adjust the primary control trim tabs as required to relieve all booster force before engaging the autopilot. He has, however, no means of monitoring the auto trim system for operability prior to engagement, and if the engagement system were to permit a faulty auto trim system to assume command of the elevator tabs, an immediate and dangerous pitch excursion might occur. For this reason, the automatic Auto Trim Monitoring circuit is included which prevents the auto trim system from being activated if it is faulty on engagement and deactivates it if a persistent fault is detected during engaged operation.

The heart of this system is the auto trim monitor amplifier, which continually receives two inputs. One input derives from the No. 1 elevator force sensor (which indicates the existence of a booster force that should produce an auto trim reaction) the other input measures the servo-amplifier output to the tab motor (which should go into operation to relieve any booster force as sensed by the No. 2 elevator force sensor). The two inputs to the monitor amplifier should coincide in time, voltage and phase; when they do not, a malfunction is indicated.

While the elevator channel is not engaged, the auto trim system is kept inoperative by Relay K18 contacts which keep the auto trim servo amplifier from responding to No. 2 sensor signals and simulate a fault indication by introducing a strong voltage at one of the monitor amplifier inputs. Under this circumstance, the trim monitor amplifier relay will not energize (engage) the tab servo clutch, and the monitor relay contacts which illuminate the AUTO TRIM warning are in position to energize the light. When the elevator channel becomes engaged, load sensor signals reach the tab servo motor amplifier, the artificial fault signal is removed from the monitor amplifier, and power is supplied to the "AUTO TRIM" warning. For a brief interval until the monitor amplifier adjusts to the newly balanced inputs, the "AUTO TRIM" warning will illuminate* and the servo clutch will remain disengaged.

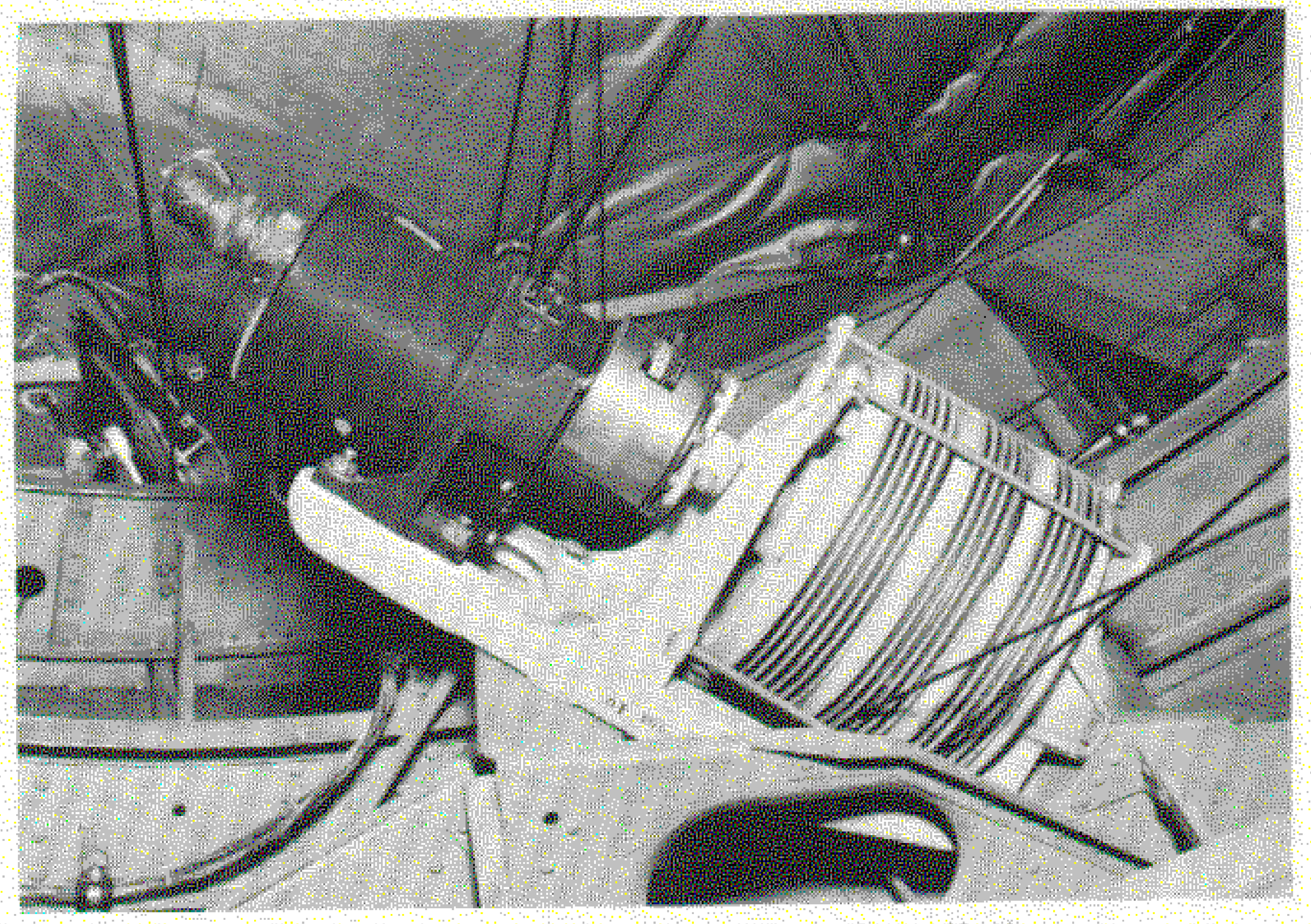


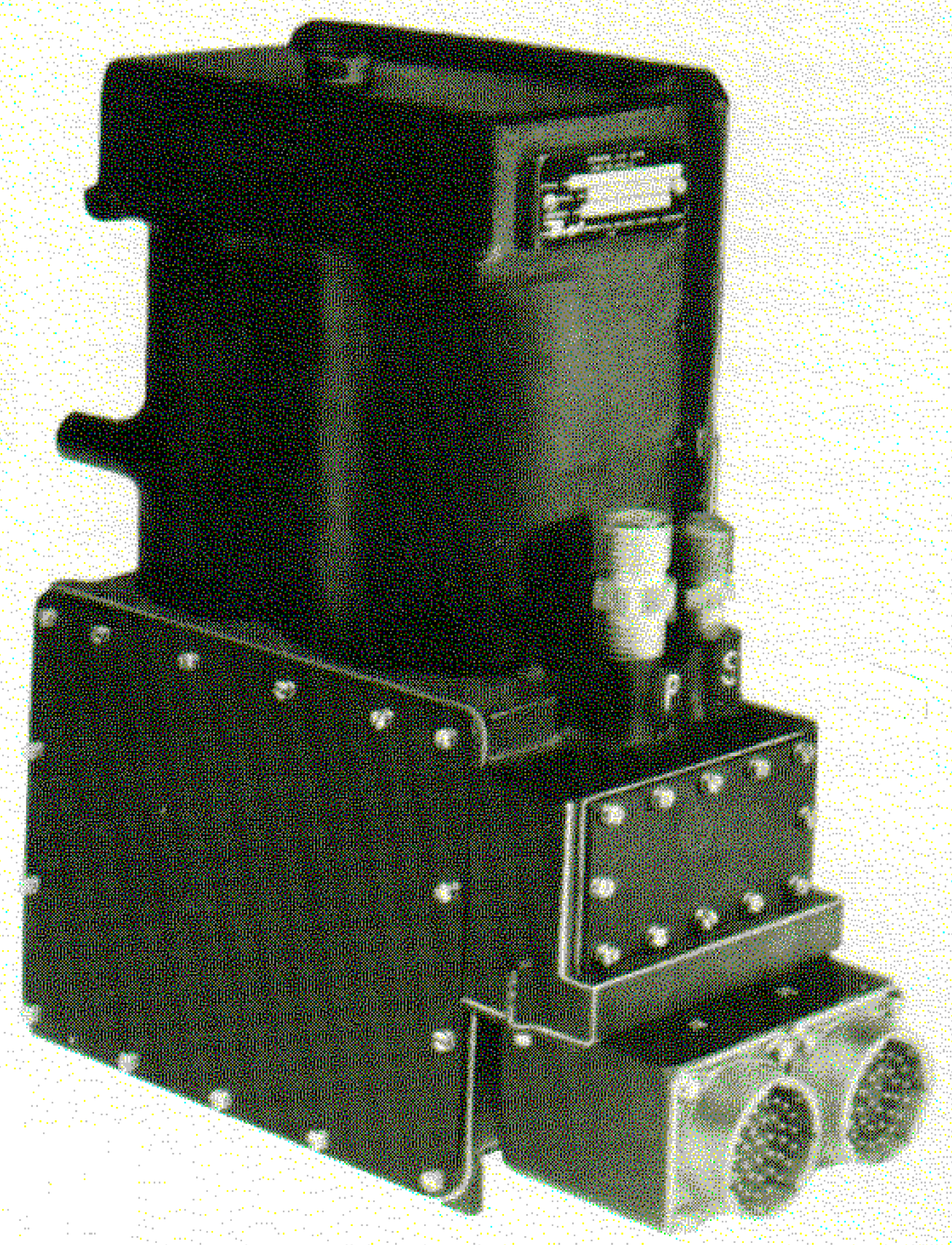
Figure 24 Automatic Trim Servo Installation

The auto trim system will become fully operational and "AUTO TRIM" will extinguish when the monitor amplifier receives, and adjusts to, balanced inputs. Note that if an unbalanced input is received (indicative of circuit malfunction) the trim servo amplifier and motor will become operational, but the "AUTO TRIM" light will remain illuminated, the servo clutch will remain disengaged, and a 7-second "trial" period begins. If during this period the two monitor amplifier inputs become equal, a condition indicating a proper response of the servo amplifier and the servo motor to the demand as sensed by the "hydraulic" load sensors, the servo clutch will engage and the AUTO TRIM light will extinguish.

On the other hand, if the servo takes no action or inappropriate action for 7 seconds, a lockout will occur which will completely deactivate the auto trim system and keep the clutch disengaged and the "AUTO TRIM" light on until 1), the fault is repaired, and 2), the three ac circuit breakers for the autopilot are opened and closed.

If an auto trim malfunction occurs while the system is operating, the "AUTO TRIM" will illuminate and the servo clutch will disengage immediately, and the system is again tested during the 7-second delay period. If the malfunction was transient, the clutch will re-engage and the warning light will be extinguished when the motor begins to take appropriate action.

**This "flash" of the "AUTO TRIM" light at the time of autopilot engagement is not indicative of malfunction, in fact, it verifies that the Trim Monitor is functioning correctly.*



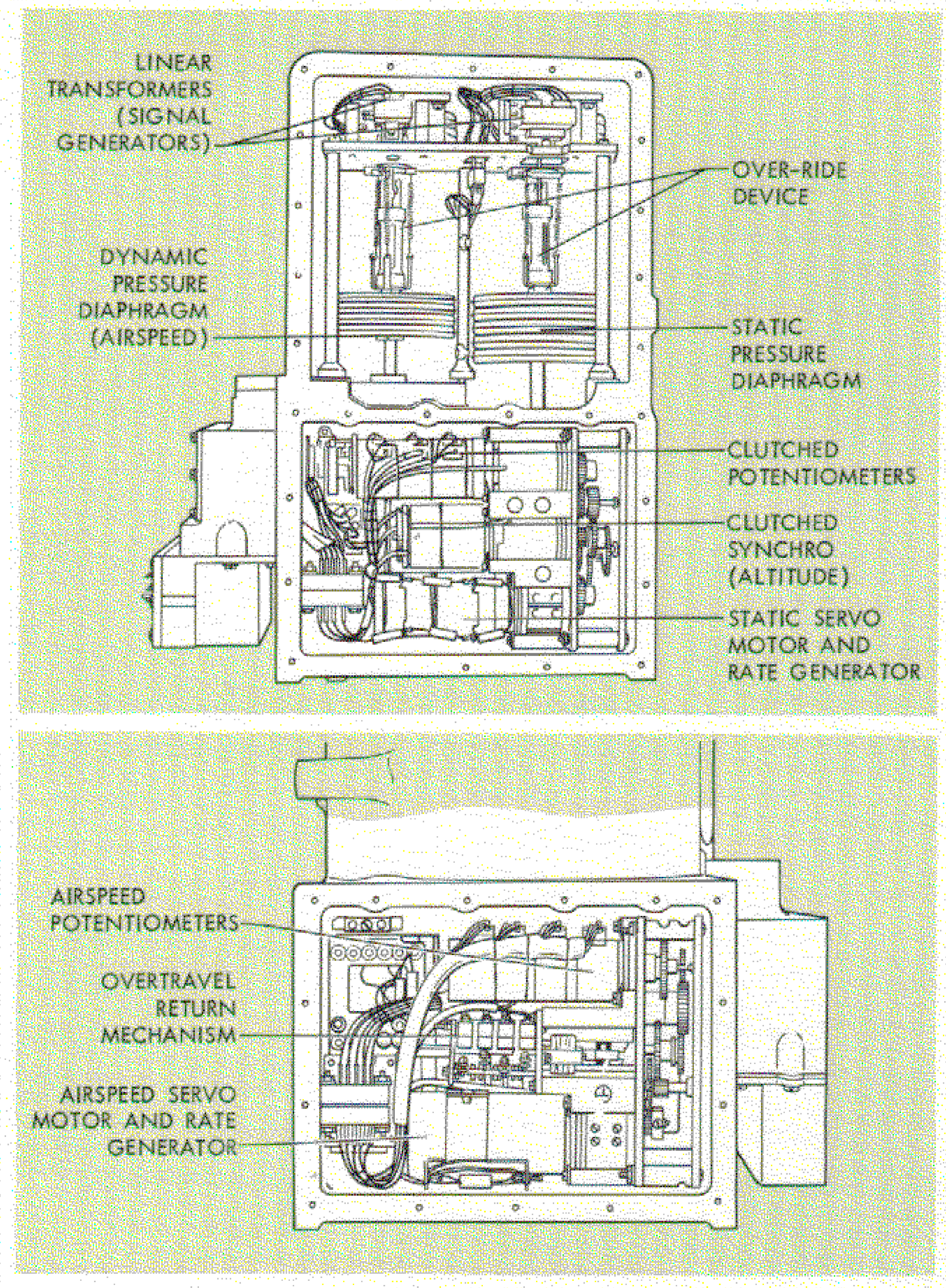


Figure 25 Air Data Sensor

The air data sensor has a dual function to perform for the Automatic Flight Control System. It detects changes in both barometric (static) pressure and dynamic (air speed) pressure and converts these to electrical signals for autopilot use. The electrical signals resulting from barometric pressure changes are mixed with other inputs to control the elevator channel but those from the dynamic section provide the controlling signal for a servo system which positions four variable resistors.

The airspeed section of the Air Data Sensor shares the copilot's pitot tube *ram* air with the copilot's airspeed indicator, the navigator's airspeed indicator, and the true airspeed system, but the ADS has its own static port just aft of the crew door.

As shown in Figure 25, the two bellows-type aneroid pressure sensors are similar. Each operates in conjunction with an electromechanical servo and is linked to a rocker arm which moves the core of a linear differential transformer (signal generator). The static bellows is evacuated and both aneroids are installed in an air tight chamber which is vented to static pressure. Thus the static aneroid expansion corresponds to a drop in barometric pressure, and since the dynamic aneroid is subjected externally to static pressure and internally to total pressure (ram air plus static pressure), its net expansion is the result of airspeed.

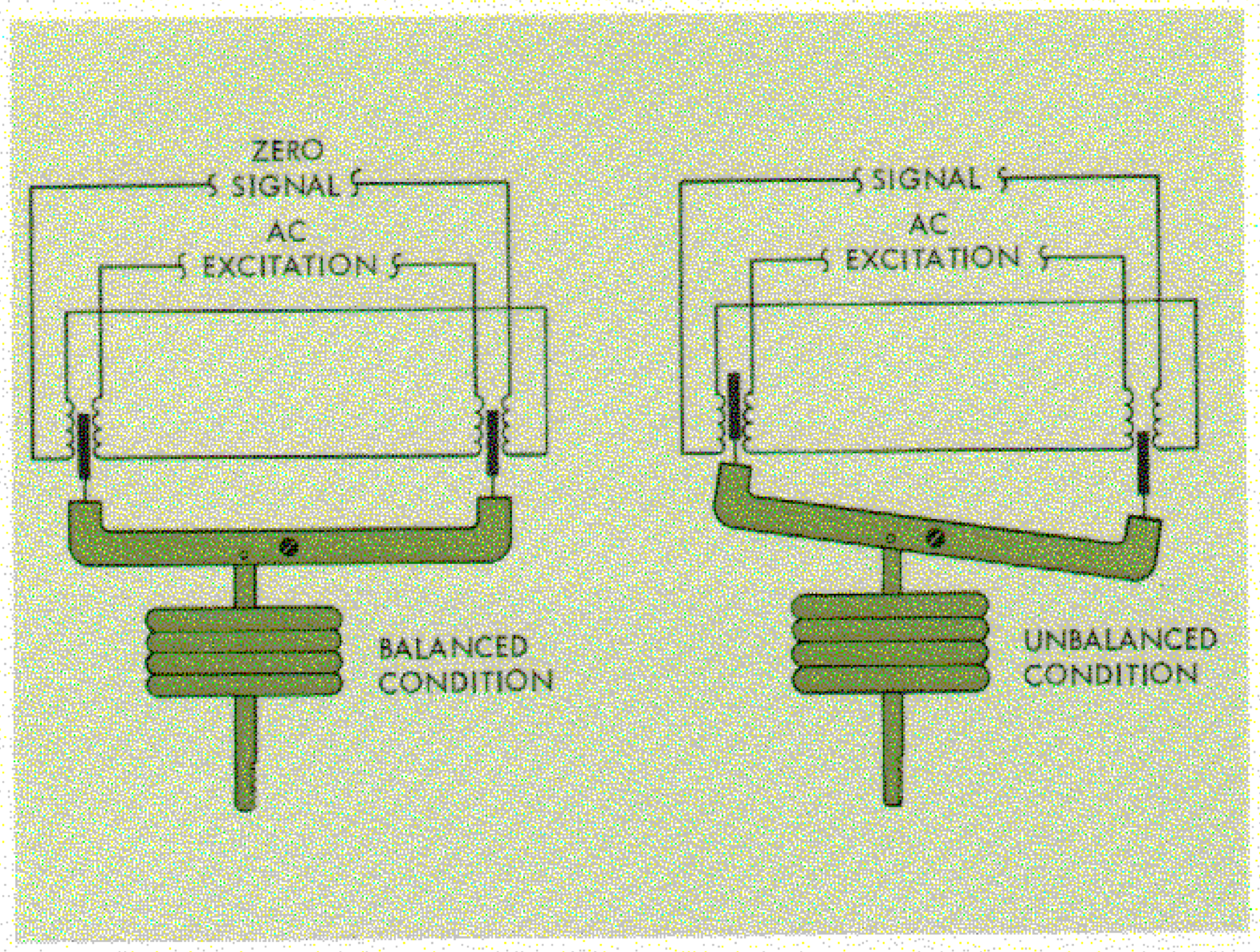


Figure 26 Pressure Change Displaces Linear Transformer Core

Both servo systems are designed to drive their linear transformer cores to a null position when power is applied to the autopilot. As shown in Figures 26 and 27 any contraction or expansion of the bellows displaces the linear transformer core and the resulting pick-off winding signal drives the servo motor to restore the bellows to its original configuration thus recentering the core.

The air speed (dynamic) servo continues to renull its core — if the airspeed changes — as long as autopilot ac power is applied. As the airspeed servo renulls the core, it repositions four airspeed attenuators (cups A, B, C, and D). Cups A, B, and C vary the "Altitude Hold" signals and the versine input to the elevator channel while cup D controls the heading input to the aileron channel.

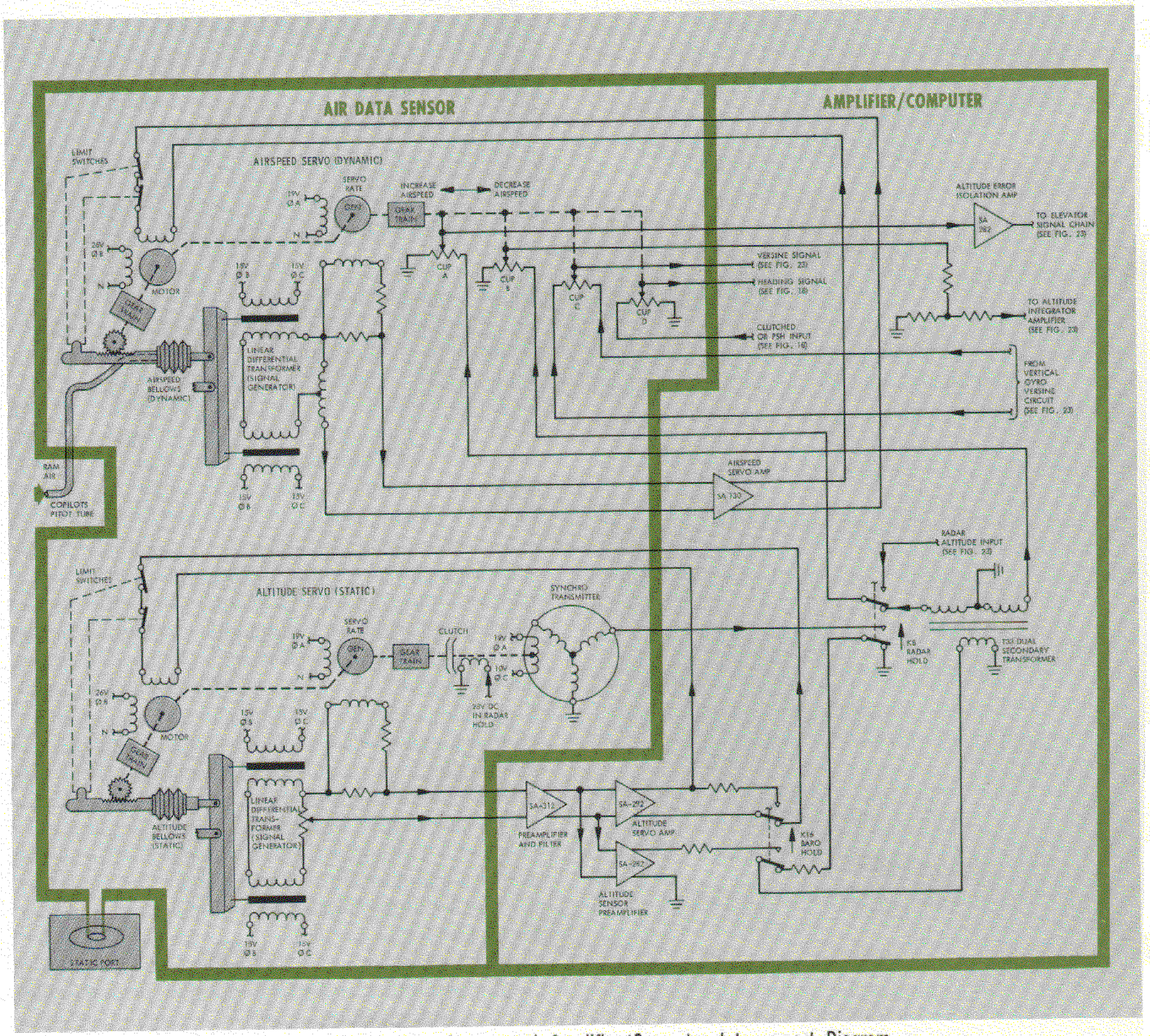


Figure 27 Air Data Sensor and Amplifier/Computer Interconnect Diagram

The barometric (static) altitude servo operates to renull its linear transformer until "BARO" hold is selected at which time relay K16 contacts disconnect the servo motor from its amplifier. In this mode the static transformer core is displaced by incremental altitude changes and the amplified signal controls the elevator channel as necessary to maintain flight at the selected pressure altitude. As shown in Figure 23 the altitude signal is introduced into the elevator signal chain through dual-secondary transformer T-33. One transformer output makes an immediate correction, via Cup A of the airspeed attenuator, at the elevator booster; the other secondary output of T-33 is applied, during Baro-Hold operation, through airspeed attenuator cup B and an amplifier to the altitude integrator, which requires a relatively longer time to develop an effective output.

Altitude integrator As shown in Figure 23, the integrator signal is applied to the signal chain in the same manner as the other elevator signals. It serves in both altitude-hold modes to establish the prime reference of pitch angle needed to maintain altitude. It utilizes signals from the T-33 transformer only during Baro-Hold operation, and we will describe its function in that connection.

Altitude-hold operation requires that a signal control channel must observe two prime references: one from the vertical gyro demanding stability at the engaged attitude, the other from the aneroid (during Baro-Hold operation) demanding stability at the engaged pressure altitude. It is predictable that these two references will conflict increasingly as continued flight reduces fuel weight. The engaged pitch angle of attack, which provided sufficient lift to offset weight at the time of engagement, will provide more than sufficient lift for a lightened aircraft, and altitude will be gained. In effect, the vertical gyro's commands to hold angle of attack constant result in the flight path being diverted upward.

Of course, as the aneroid senses the tendency to change pressure altitude, the aneroid corrective signal fed through the "direct correction" tap of the T-33 transformer will be for "nose-down" trim, but as the pitch angle is depressed, the vertical gyro will sense the deviation and begin to supply a growing "nose-up" signal. Since the normal control corrections to maintain pressure altitude are less than those which may be needed to maintain pitch, the aneroid signal is proportionately restricted in intensity. The nose-up command of the vertical gyro will shortly equal the maximum direct aneroid signal capability for "nose-down," and the aircraft would gain altitude if it were not for the contribution of the altitude integrator during altitude-hold operation.

The altitude integrator operates continually in response to aneroid corrective signals, and due to the prevalence of "nose-down" aneroid signals in the forementioned circumstance, the integrator synchro will feed to the transfer valve an increment of "nose-down" signal which grows until it surpasses the "nose-up" signal emanating from the vertical gyro. Thus, the integrator signal predominates at the transfer valve, and the transfer valve will react as it should, that is, hold the nose down just enough to maintain engaged altitude and provide a properly scaled control response to temporary signals deriving either from the aneroid or from the gyro system.

When the integrator motor is made inoperative by selecting "OFF" at the Altitude Hold switch, its synchro signal becomes ineffective, but since the elevator channel is at this time briefly disengaged and then re-engaged, the accumulated error at the vertical gyro is eliminated coincidentally with the integrator corrective input, and the flight will continue at the re-engaged attitude.

The radar hold mode utilizes both barometric and radar altimeter inputs. When this mode is selected, the air data sensor's static servo is reactivated, the synchro transmitter is clutched to the static servo shaft, and the synchro output is switched in to replace the linear transformer as the barometric (short term) input. This substitution is made primarily to take advantage of the broader operating range (± 500 ft) of the reactivated servo in preference to the limited static range (± 80 feet) of the linear transformer.

Coincident with the switching just discussed, the output from the radar Altimeter/Autopilot Coupler is substituted for the T-33 secondary as the signal source for the altitude integrator.

Signals of sensed radar altitude work exclusively through the slow-reacting integrator. A protracted tendency to climb or descend will work through the autopilot to operate the auto trim, and the aircraft will be continually trimmed to fly a level course as sensed by the radar altimeter.

Momentary elevator corrections will be made via aneroid signals through the T-33 Transformer and/or the vertical gyro system as before; but the radar-hold control via the integrator will shortly cancel any aneroid or gyro correction that persistently tends to change altitude. Thus, a gradual barometric change resulting from penetrating a weather front cannot induce gradual incline of the flight path during radar-hold operation. On the other hand, if a transient air disturbance were to deflect the flight path up or down, both the radar altimeter and the aneroid signals will coincide in supplying corrective action.

The radar altimeter/autopilot coupler is not a part of the PB-20N autopilot and, as stated previously, it is incorporated on current aircraft and added in service as a part of the AFC 71. The operating details are beyond the scope of this article but some of its basic functions are important in understanding the "Radar Hold" mode of operation.

The coupler "memorizes" the radar altitude in the form of a dc reference voltage at the time radar hold is selected. It then supplies the autopilot with an equivalent ac voltage representing any deviation from this reference. The initial coupler output to the autopilot altitude integrator is a zero signal since the servo loop in the coupler is maintained at a null until "radar hold" is engaged. When this mode is selected the servo is deactivated and the coupler develops a properly scaled ac output only if altitude deviations occur. As mentioned previously, the autopilot altitude integrator is driven by this signal and builds up a corrective output to restore the aircraft to the original altitude. Through other circuitry the coupler provides a warning signal to activate the pilot's and copilot's glare shield AUTOPILOT/RADAR ALTM warning lights when the deviation exceeds 75 feet.

The PB-20 Amplifier/Computer type 15483-1D1 utilized on the majority of the P-3 aircraft in service (Prior to BUNO 152718 or the inclusion of AFC71) incorporates a radar synchronizer to provide a constantly nulled radar altitude signal prior to radar hold

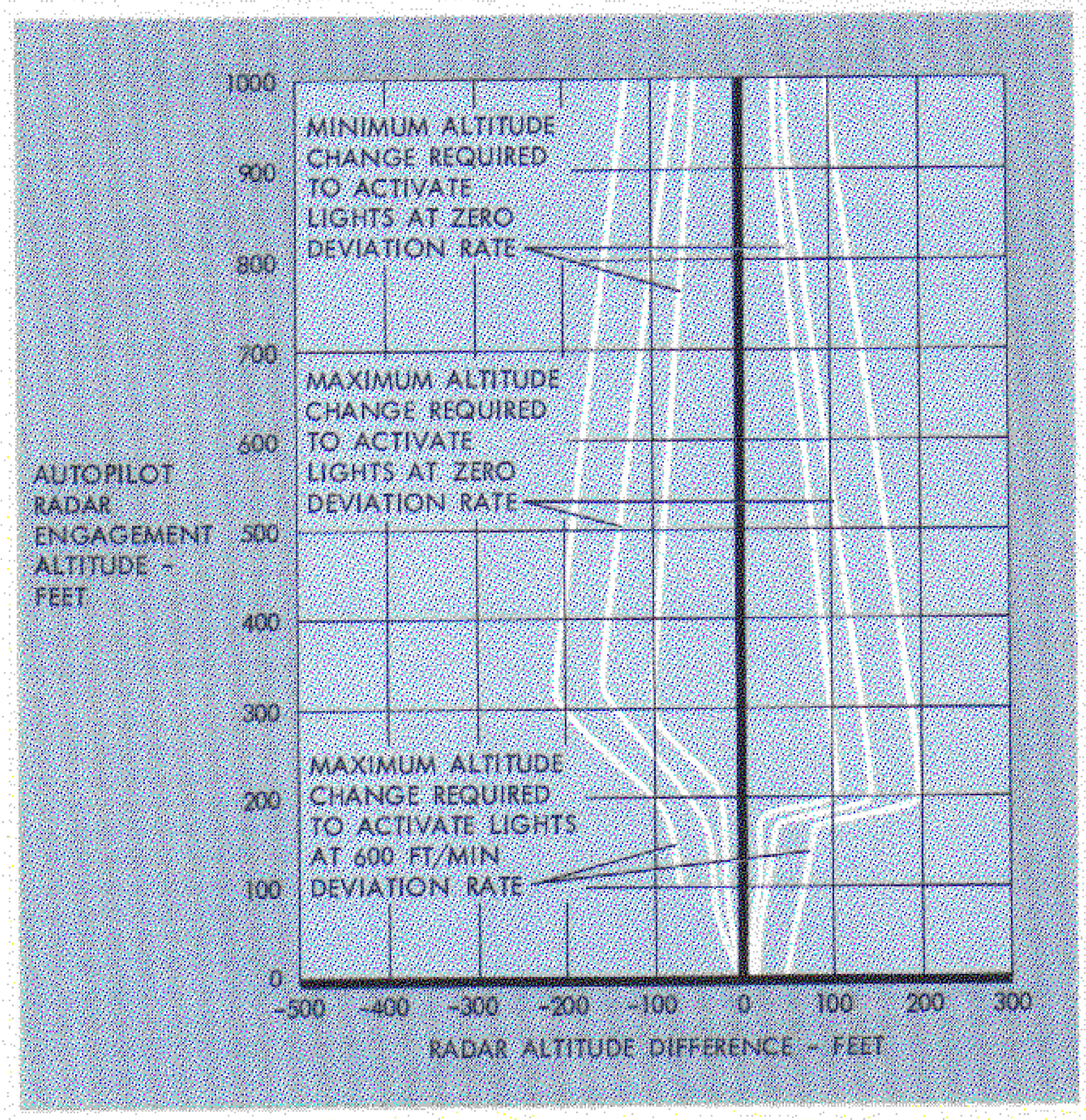


Figure 28 Graph Depicts the Difference Between Actual Aircraft Altitude and the Radar Altitude Hold Reference Signal that Occurs on Unmodified Aircraft before the Flashing Red Warning Lights are Activated

engagement and a reference signal thereafter. The Amplifier/Computer also includes a radar error monitor to sense radar altitude deviation from the engaged reference. It energizes the flight crew's flashing red lights when the deviation exceeds the threshold limit of the monitor amplifier. As shown in Figure 28, this threshold limit, and thus the operation of the warning lights, varies with altitude as well as with the rate of change of altitude.

On updated aircraft these functions are included in the Altimeter/Autopilot Coupler. The Amplifier/Computer is modified by removing the error monitor, the radar synchronizer, the synchronizer amplifier, and the associated circuitry. The reworked unit is reidentified as type 15483-1E1.

The pitch control thumbwheel knobs provide a means for inserting a manual pitch command to the engaged autopilot. The knobs are mounted on a common shaft and operate the arm of a potentiometer through an electrically engaged clutch (See Figure 29). The clutch is de-energized in both altitude hold modes to prevent a manual pitch command input from inducing a climb or descent in contradiction to the control action needed for maintaining altitude.

A mechanical centering device is an integral part of the pitch controller assembly and includes a solenoid — energized when the elevator channel is engaged — to lift a spring-loaded centering lever and apply a light braking pressure to one side of a heart-shaped cam which is attached to the potentiometer arm. When the elevator is disengaged, a roller on one end of the centering lever rides the edge of the heart-shaped cam thereby forcing the cam to return the potentiometer to its center (null) position.

Forward rotation of the pitch knobs inserts a nose-down signal and aft rotation a nose-up signal. The potentiometer is designed to have a linear voltage output to ensure that pitch command signals are proportional to knob rotation.

Control wheel steering for the elevator involves essentially the same control method as that for the aileron. Approximately a two-pound push or pull force on either control wheel is needed to de-energize the elevator engagement valve and allow boost-on manual manipulation of the elevators. Immediately after the fore or aft force is removed, the elevators are restored to full automatic operation at the existing pitch angle. Control wheel steering of the elevators is disabled during the "Baro" or "Radar" altitude hold modes.

The flap position transmitter is a dual-synchro transmitter unit operated by the flap drive gear box in the Hydraulic Service center. One of the two transmitters sends information to the Wing Flap Position

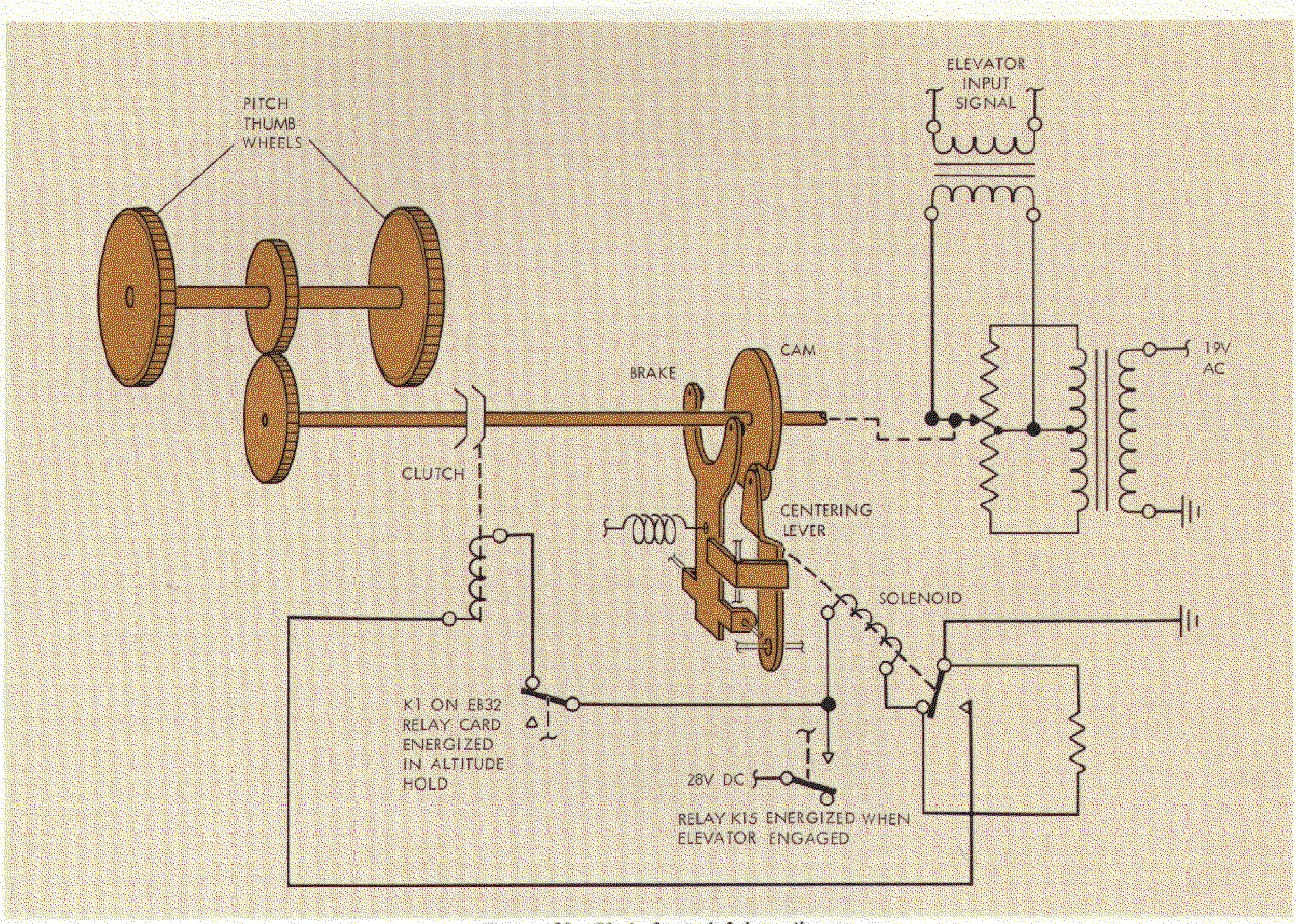


Figure 29 Pitch Control Schematic

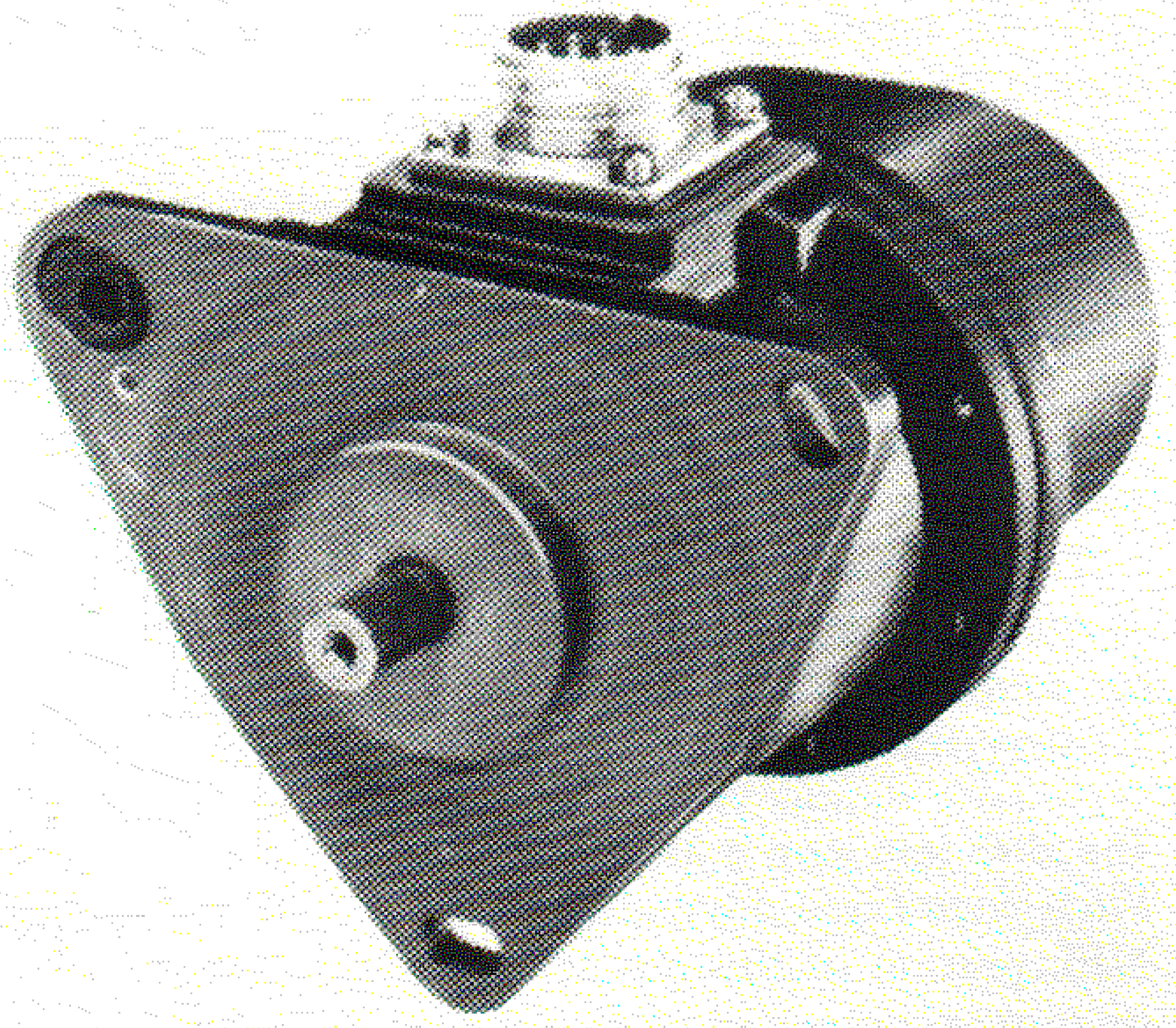


Figure 30
Dual Synchro
Flap Position
Transmitter

Indicator on the copilot's instrument panel, where it is displayed as per cent of flap extension. The other synchro is driven at a slower rate by a self-contained 5-to-1 gear reduction assembly. Its output is interconnected to the elevator automatic control circuit to counteract the "ballooning" effect of extending flaps by inducing elevator-down signal as the flaps are extended. The resulting nose-down maneuver decreases the wing's angle of attack (and therefore the lift of the fixed portion of the wing), keeping lift essentially constant as the flaps extend.

Versine Signal In contrast to the effect of the extended flaps, a bank maneuver tends to *decrease* the effective vertical lift due to the fact that lift always acts perpendicular to the wing and is not in direct opposition to gravity while in bank. Under manual control, the experienced pilot instinctively adds an elevator up force to prevent altitude loss.

Under automatic control, this corrective signal is introduced by the *versine* circuit shown in Figure 31 which provides the means for continuous computation of the required input. The circuit automatically combines the outputs of the vertical gyro's bank and

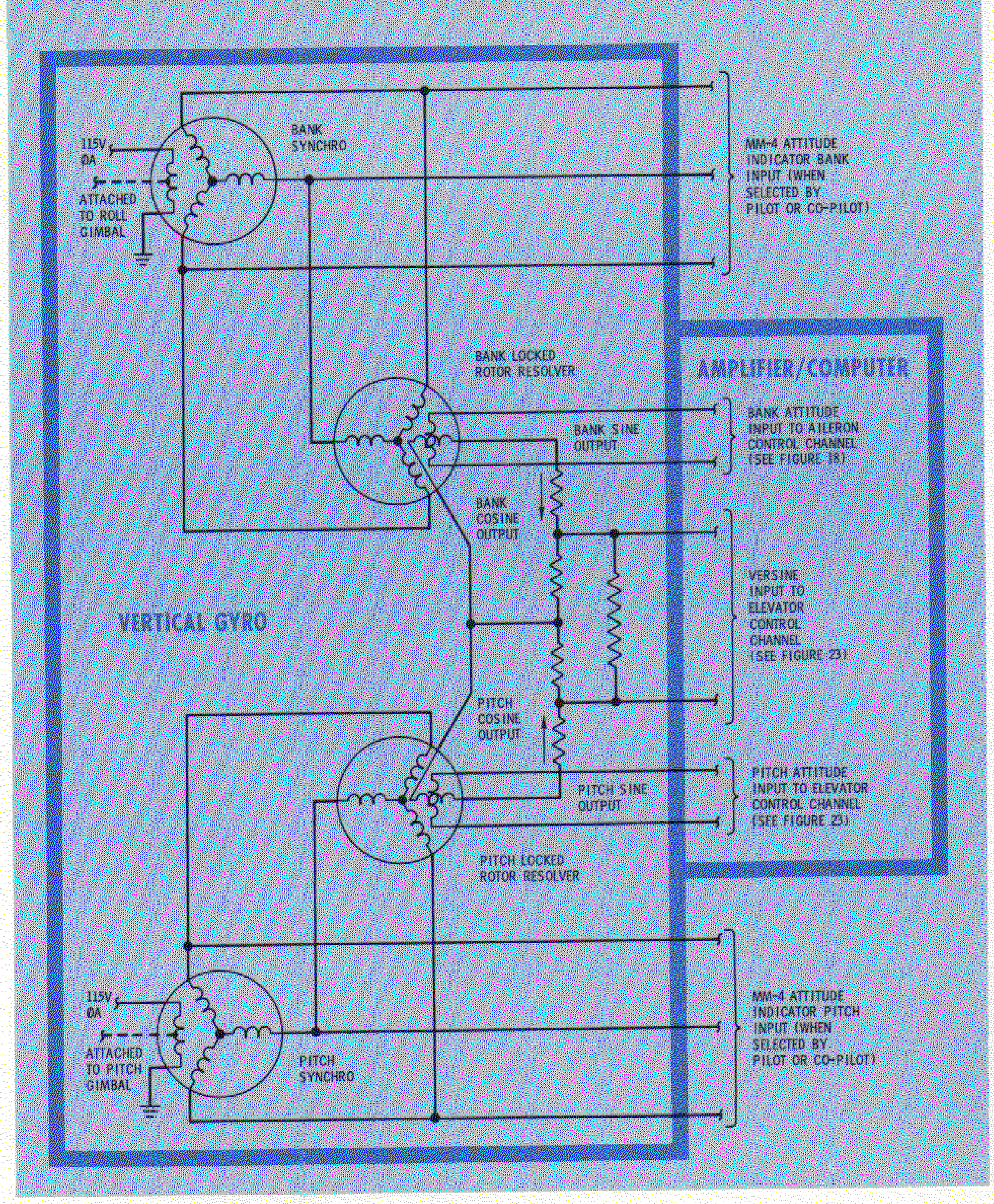


Figure 31
Versine Circuit

pitch synchros to provide a zero elevator corrective signal at zero bank angle and an increasing elevator corrective signal as the aircraft banks left or right.

This circuit is an integral part of the Vertical Gyro and consists of two locked-rotor resolvers and a resistor network. As shown schematically in Figure 31 the resolvers are basically a synchro with two rotor coils (sine and cosine) which are placed at right angles to each other on the rotor shaft. The two resolver stators receive their excitation (attitude information) in terms of angular degrees from the vertical gyro's pitch and bank synchros*, and derive from these signals the sine and cosine equivalents of those angles.

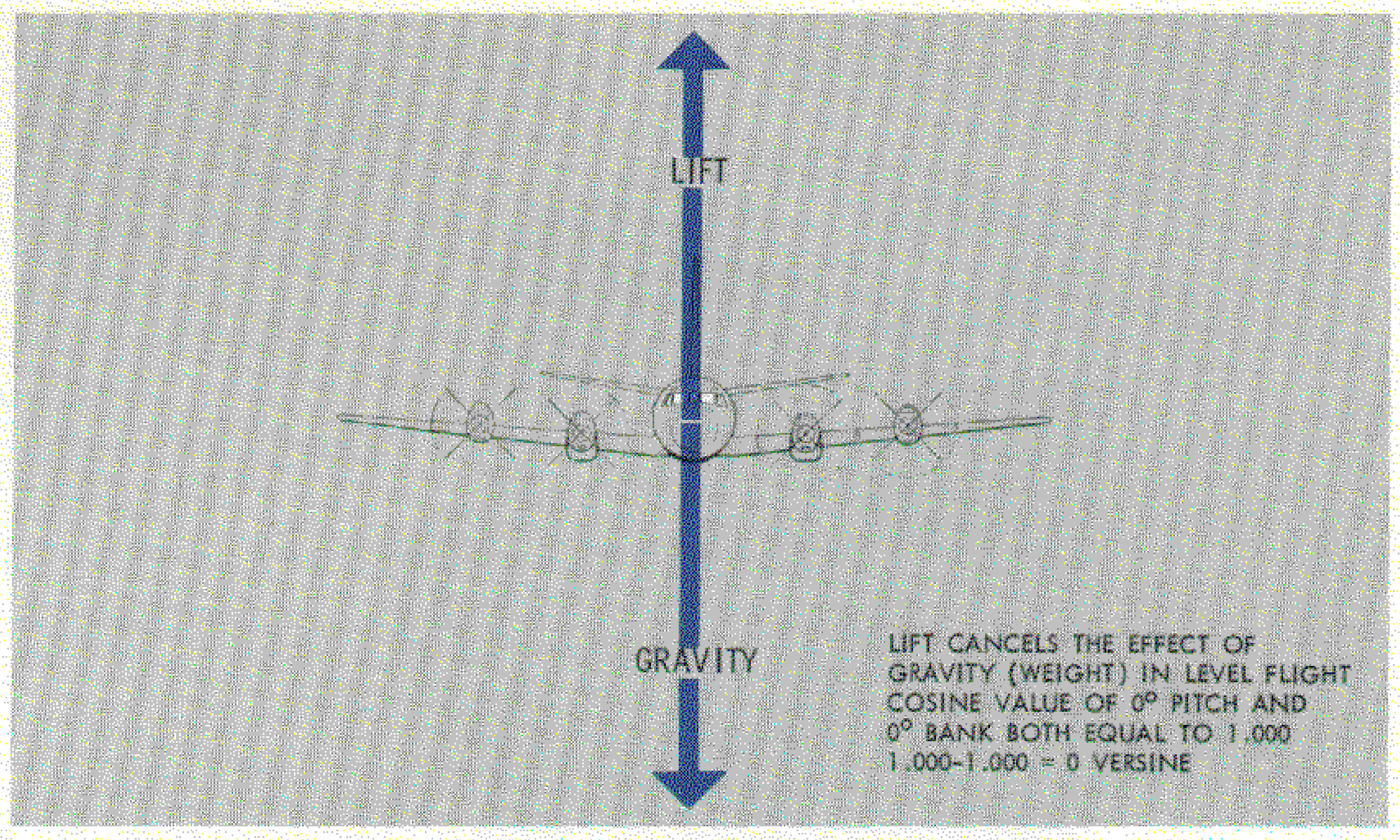
As depicted in Figure 32 the loss in lift in a bank maneuver is in direct proportion to the decrease in the cosine value of the bank angle. It follows then that the cosine output of a resolver — excited by the output of a synchro sensing bank angle — decreases at the same rate as the lift and can be utilized in computation of the necessary corrective (versine) signal.

As defined mathematically, versine equals 1 minus the cosine of an angle, and as shown in Figure 31, only the cosine outputs are used as factors in producing the versine signal. Both pitch and roll cosine winding outputs are maximum in level flight, but since the voltages are connected in electrical opposition in the resistor network, the net signal is zero. Either a left or a right bank reduces the bank input to the network, but the pitch input remains at maximum assuming, of course, that the pitch attitude is unchanged.

The remaining uncancelled portion of the pitch cosine signal is the "versine" input which is adjusted for airspeed through attenuator cup C and applied as a corrective elevator-up signal in the pitch channel.

The pitch rate gyro is an integral part of the Yaw and Pitch Rate Gyro discussed in the RUDDER CHANNEL portion of this article. The *pitch* rate gyro has its spin axis oriented vertically (see Figure 33) and is therefore unaffected by aircraft yaw motion. The ends of the gyro gimbal support shaft cannot rotate

**Either the pilot or the copilot can select these identical signal sources for his MM-4 attitude indicator by selecting his attitude switch to the STANDBY GYRO position. This provides a convenient visual check of the basic attitude information being utilized by the autopilot.*



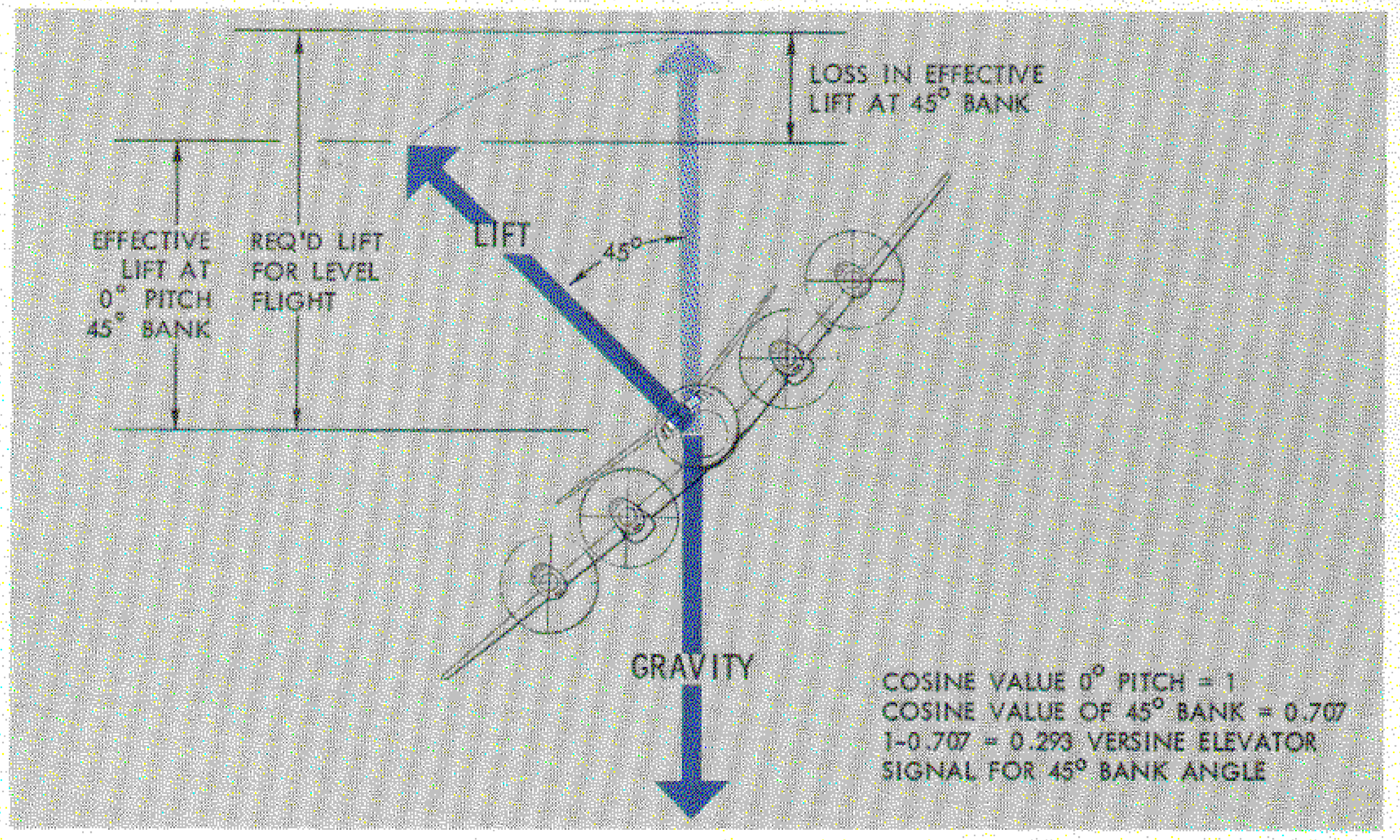


Figure 32
Decrease in Effective Lift in Bank Attitude is Proportional to the Change in the Cosine of the Bank Angle

vertically in response to an aircraft bank maneuver but can rotate in the gimbal bearings in response to a pitch maneuver. The latter rotation is restrained by an adjustable flat spring, and two restricted-flow, dashpot air cylinders on one end. A synchro transmitter is attached to the gimbal shaft and as the gimbal rotates, the synchro supplies an electrical signal which is proportional to the *rate* of the aircraft pitch maneuver. As the pitch rate decreases the spring returns the gimbal to its center position at a rate determined by the size of the air escapement orifices on the two dampener cylinders.

The output of the pitch rate synchro is filtered by the same type of amplifier (SA 244) as that used in the rudder rate circuit. As stated previously, this amplifier filters out the steady state or slowly changing signals from the pitch rate gyro — a signal of the type the gyro senses when the pitch control thumbwheels are rotated. It passes signals sensed by the gyro as a result of turbulence or other disturbances that induce an abrupt pitch change. The filtered signal is inserted in the elevator signal chain to oppose the pitch deviation signal generated by the vertical gyro. The net effect is to limit the rate at which the autopilot effects a pitch correction.

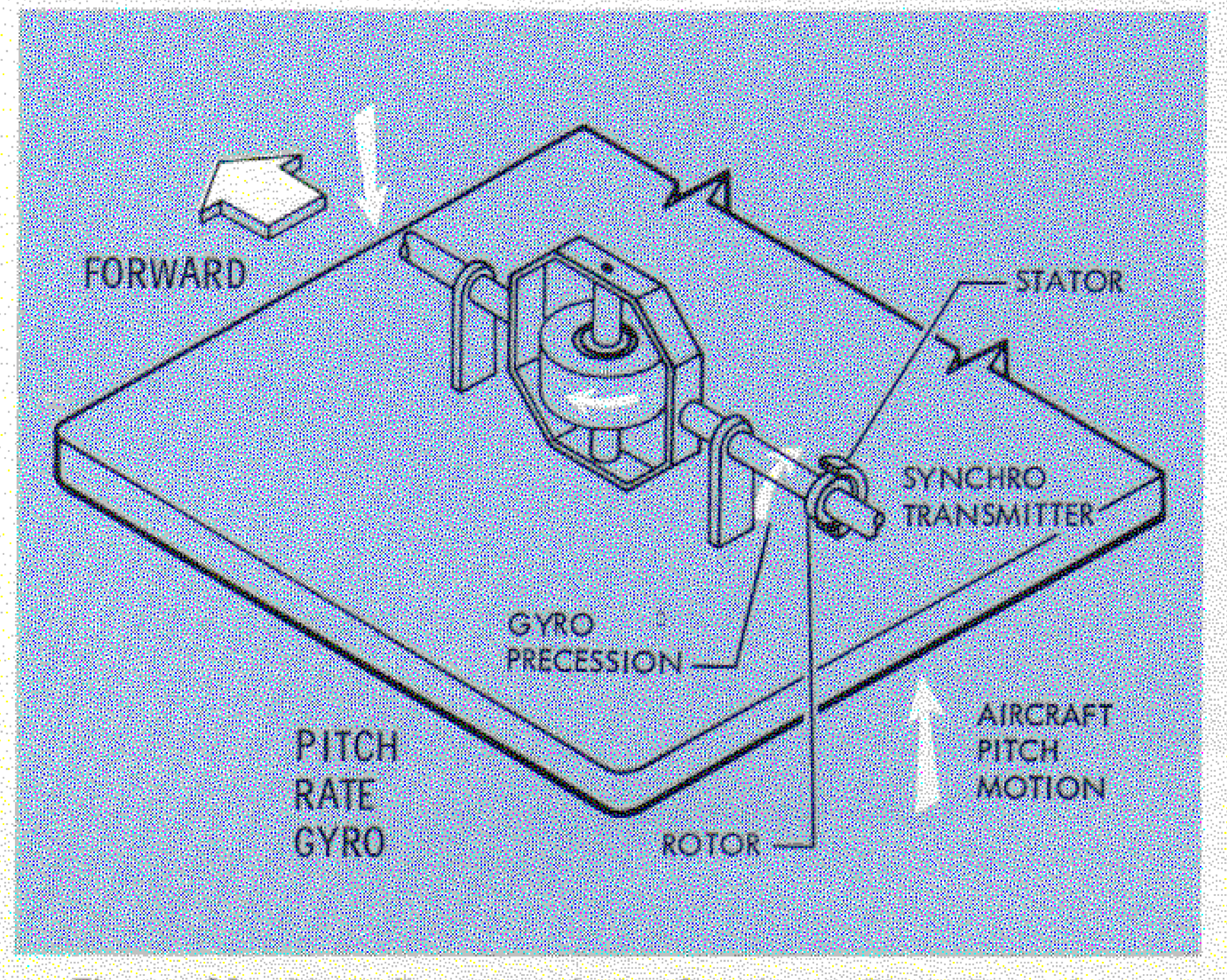


Figure 33 Pitch Forces and the Resultant Pitch Rate Gyro Gimbal Rotation

Figure 34
P-3 Vertical Gyro

VERTICAL GYRO

The vertical gyro provides the principal pitch and bank reference signals for the autopilot's control of the elevator and the aileron circuits at engagement attitudes and senses any subsequent deviation from these attitudes. As stated previously other modes will temporarily introduce overriding inputs and relegate the gyro signals to a subordinate status but the gyro signals are not disconnected.

The gyro's spin axis is automatically erected at a high rate to a vertical position during the initial 60 seconds of power application. Thereafter, the same torquers maintain the vertical status but operate at a reduced rate(lower voltage). Verticality is sensed on both the pitch and the roll axes by curved switches filled with fluid-electrolyte that vary the ac voltage to the torquer motor windings as a function of switch deviation from level position (see Figure 35).

Unfortunately the switch fluid is also affected by the centrifugal force of a turn. If ignored, this tendency would cause the gyros to be torqued off-level during a long sustained turn. To avoid this undesirable situation the Rate of Turn Gyro monitors the aircraft turn rate and removes bank torquing for the duration of any turn that exceeds 15° per minute.

The vertical gyro is interlocked electrically with the autopilot enabling circuits and with the pilot's and/or the copilots MM-4 attitude instruments if either operator has selected the vertical gyro as the indicator's signal source. The interlock relay de-energizes the autopilot and displays an OFF flag on the attitude indicator if the vertical gyro deviates more than 7° in pitch or 7° in bank and fails to recover in approximately 150 sec.

GYRO HANDLING

In spite of continued efforts on the part of designers to make equipment containing gyros more rugged, careless and improper handling continues to be the prime cause of premature failure. Gyros are delicate and expensive and should be handled accordingly.

Premature failures due to improper handling are generally due to one or more of the following types of mistreatment: lifting or transporting units by the electrical harnesses, dropping or bumping units, or moving the units while the gyro wheel is coasting down. A common failure of the P-3's vertical gyro occurs in the electrical harnesses or "pigtails" attached permanently to the case, and generally the cause can be traced directly to the use of the harness as a convenient handle for removing or transporting the units.

Gyro or gimbal bearing damage is most likely to occur when units are removed during rundown. The high "rigidity in space" characteristic that makes a gyro such a useful device tends to impose high loads on the gimbal bearings and on the bearings of the spinning wheel when units are lifted abruptly and/or tilted at extreme angles from their normal operational attitude. "Gimbal lock" may occur if an operating unit such as the P-3's vertical gyro is tilted 90 degrees. Since the roll gimbal is free to rotate 360°, it may tumble and spin rapidly enough to damage the gimbal bearings.

Insofar as maintenance procedures require that the gyros be moved or tilted while performing operational tests, care must be taken to avoid moving the units rapidly or tilting them to angles much greater than those specified.

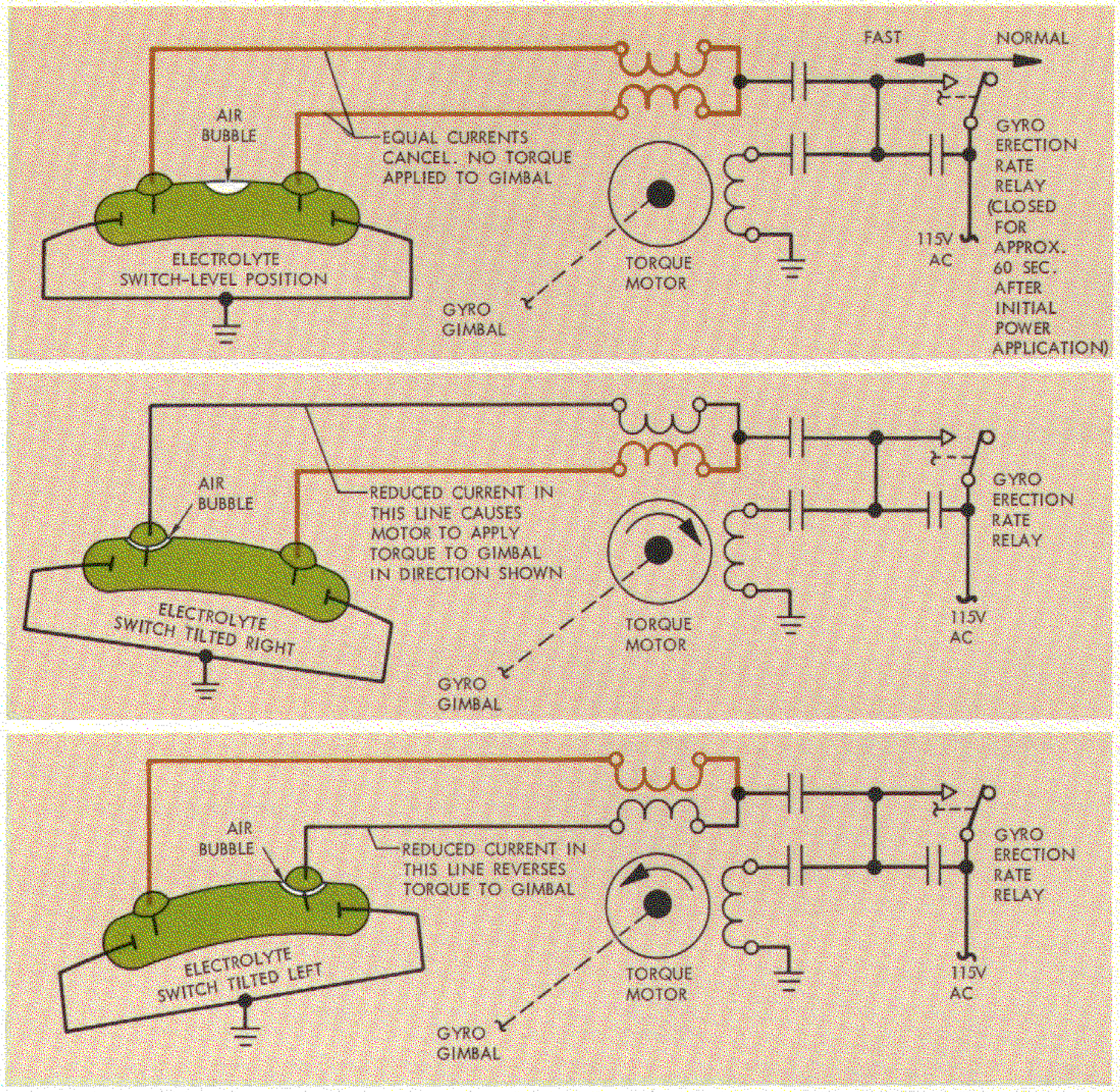
The Maintenance Instruction Manual (NAVAIR 01-75PAA-2-11) specifies minimum run-down times which should be allowed for AFCS components and these are listed here for reference.

Vertical gyro	—	25 minutes
Rate of turn	—	15 minutes
Yaw and pitch rate gyro	—	15 minutes

Since gyros in general are not designed to withstand the high shock loads encountered in dropping or in transporting them on uncushioned vehicles over rough terrain, all new units should be transported in their shipping cartons to the place of installation, and removed units should be given the same protection while in transit to shop areas.

Inasmuch as the gyro you save may be your own, handling precautions and techniques can be summarized as follows:

- Keep the gyro in the original shipping container as long as possible and transport the removed unit in the same manner.



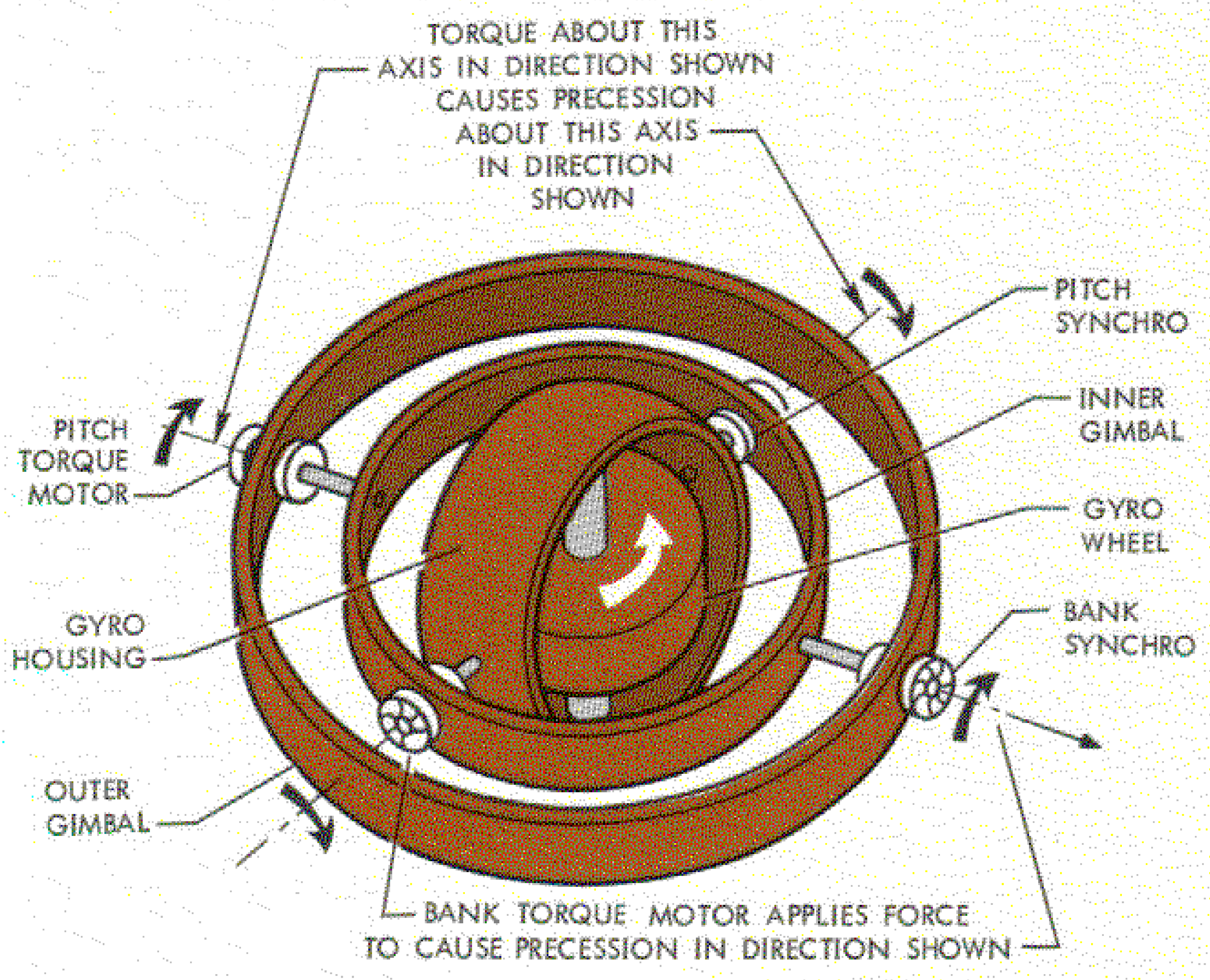


Figure 35
Gyro Erection Diagram. Pitch and bank circuits are identical.

- Wait at least the recommended time after power is shut off before removal of the gyro.
- Lift the gyro by its base (or by its handle if one is provided) and carry in an upright position.
- Avoid subjecting any gyro, old or new, to shock or vibration.

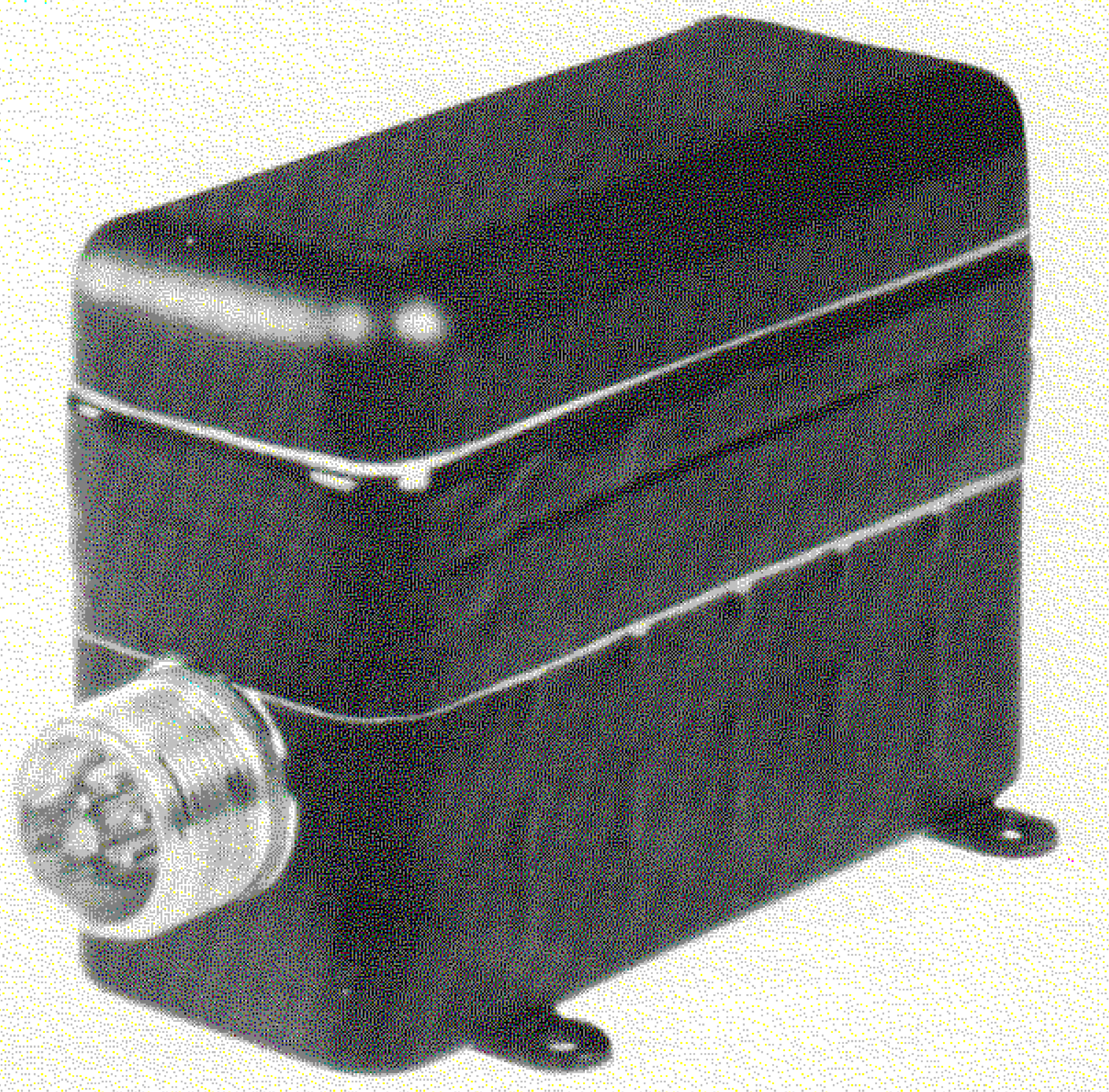


Figure 36 Rate of Turn Gyro

MALFUNCTION PROTECTION

The P-3 Automatic Flight Control System has a liberal number of protective circuits and devices — both manual and automatic — for alerting flight crews of operational discrepancies and for disabling the system in the event such malfunctions occur.

As mentioned previously, a mechanical override of any autopilot channel is accomplished simply by applying a force to the pertinent flight station control. This, in effect, partially nullifies autopilot authority by forcing the locked boost-on actuating arm to move and thereby open the boost control valve beyond the range possible under autopilot authority.

Electrical disconnect of the autopilot is effected by any one of the following series connected devices.

1. Pilots or copilots press-to-break AUTOPILOT DISC switches on the control wheels.
2. Manual actuation of the AFCS Control Panel ENGAGE-OFF switch to OFF.
3. Loss of 28-v dc, or any one or all three 115-v ac power inputs to the AFCS Power Supply. The same three Power Supply monitor relays also sense a reduced current drain by circuitry of any one of the three autopilot channels.
4. Vertical gyro malfunction relay (Part of vertical gyro) which is de-energized 150 seconds after loss of verticality or immediately after the loss of the vertical gyro's 115-v ac power.
5. No. 1 hydraulic system low pressure switch in the hydraulic service center opens when pressure drops to approximately 1100-PSI.

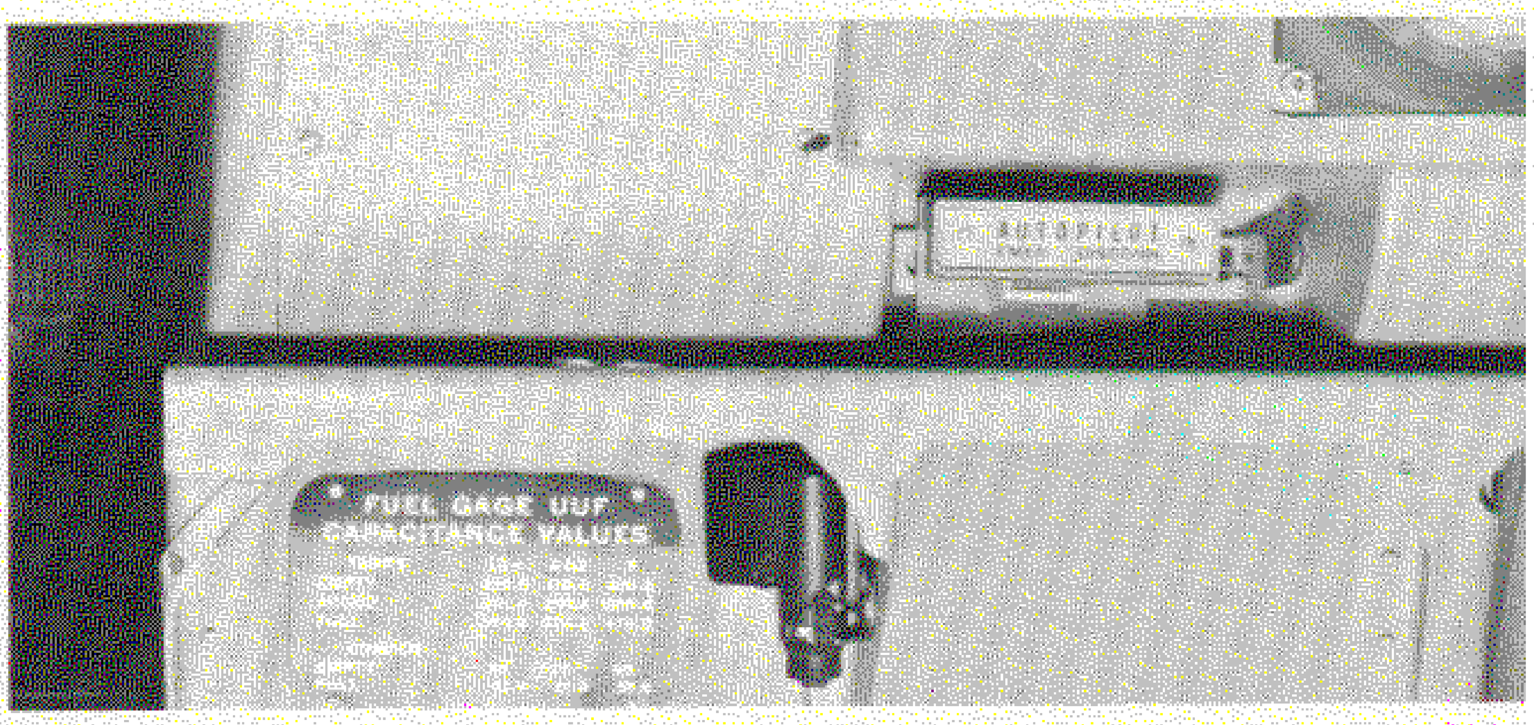


Figure 37 Autopilot Emergency Disconnect Handle

Items 2, 3, 4, or 5 are considered to be inadvertent disconnects and will cause the center instrument panel amber AUTOPILOT annunciator light to illuminate and the Pilot's and the Copilot's glare shield AUTOPILOT/RADAR ALTM red warning lights to flash. The AUTOPILOT EMER. DISCONN. handle and Item 1 above are considered to be intentional disconnects and will not activate the annunciator light or the glare shield warning lights.

If the autopilot does not disconnect electrically, the three channels can be disconnected both mechanically and electrically by pulling the AUTO PILOT EMER. DISCONN. handle (see Figure 37). As shown by Figures 12, 18, and 23 this procedure interrupts the 28-v dc interlock circuit as well as mechanically retracting the autopilot engagement actuators on all three boosters to return the system to boost-on manual operation. If this method also fails to disengage the A/P, pulling the individual BOOST OFF handle will remove hydraulic pressure from the booster actuator and lock the autopilot control mechanically from interfering with manual/mechanical operation of that surface control system. ▲▲

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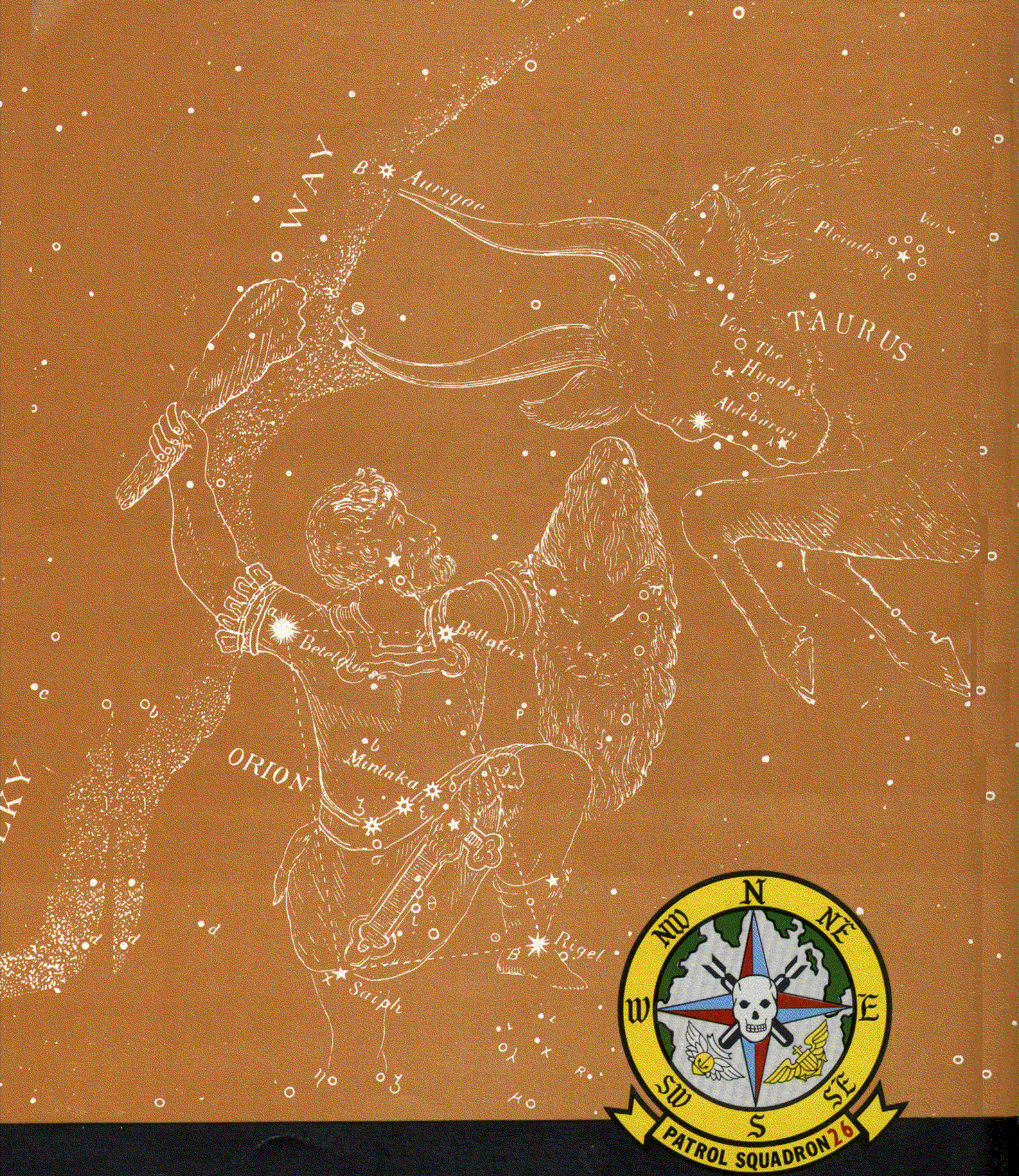
SERVICE REPRESENTATIVES, P-3 ORION

LOCATION	NAME	MAILING ADDRESS	TELEPHONE
NATC Patuxent River, Maryland	D. H. Horadam, Regional Rep.	P.O. Box 218, NATC Patuxent River, Maryland 20670	(301) 863-3111 Ext. 645 or 251
NAS Brunswick, Maine	F. Hampton, Resident Rep.	P.O. Box 38 Brunswick, Me. 04011	(207) 725-5561 Ext. 513
NAS Norfolk, Virginia	A. Barber, Resident Rep.	P.O. Box 8127, Ocean View Station Norfolk, Virginia 23503	(703) 423-8321
NAS Jacksonville, Florida	J. Gipson, Resident Rep.	P.O. Box 1300, Yukon, Florida 32230	(904) 772-2883 NAS Jacksonville
NAS Key West, Florida	W. Wilken, Resident Rep.	P.O. Box 1087, Key West, Florida 33040	(305) 294-4417, Key West
NAS Moffett Field, Calif.	F. Hays, Resident Rep.	Lockheed P-3 Office Unit One P.O. NAS Moffett Field, Calif. 94035	(415) 966-5461 Mountain View, California
NAS North Island, San Diego, Calif.	R. B. Hedin, Resident Rep.	P.O. Box 548 Coronado, California 92118	(714) 435-8910 Coronado, Calif.
NAS Barbers Point Honolulu, Hawaii	J. M. Nugent, Resident Rep.	P.O. Box 268 FPO San Francisco, Calif. 96611	(808) 683-444 Honolulu, Hawaii
Burbank, California	C. F. Wernle, Supervisor P-3 Project, Customer Service	Lockheed-California Company P.O. Box 551, Burbank, California 91503	(213) 847-4680 Burbank, Calif.



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