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**AN/ASW-31 AUTOMATIC FLIGHT CONTROL SYSTEM**

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### AN/ASW-31 AUTOMATIC FLIGHT CONTROL SYSTEM



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**FRONT AND BACK COVERS**
Patrol Squadron Fifty-Six, currently based at NAS Jacksonville, Florida, was the first operational squadron in the Navy to fly the P-3C. The squadron's history goes back more than 20 years to when it was commissioned Patrol Squadron Nine Hundred on 1 July 1946, as part of the Naval Reserve Program. While flying PV-2 aircraft at NAS Anacostia, the squadron was redesignated VPML-71, then VP-661. In 1950, VP-661 was called to active duty and, following transition to PBM aircraft, joined the Atlantic Fleet at NAS Norfolk. On 2 March 1953, the squadron was redesignated VP-56.

In January 1961, VP-56 began its transition from seaplanes to P2V-7 Neptunes. In October 1962, PATRON FIFTY-SIX deployed five aircraft to the U.S. Naval Air Station, Guantanamo Bay, Cuba; and, during the Cuban Quarantine the entire squadron deployed to Cuba. For this concerted effort the squadron received a "Well Done" from several senior commands and a letter of commendation from the Commander-in-chief, U.S. Atlantic Fleet.

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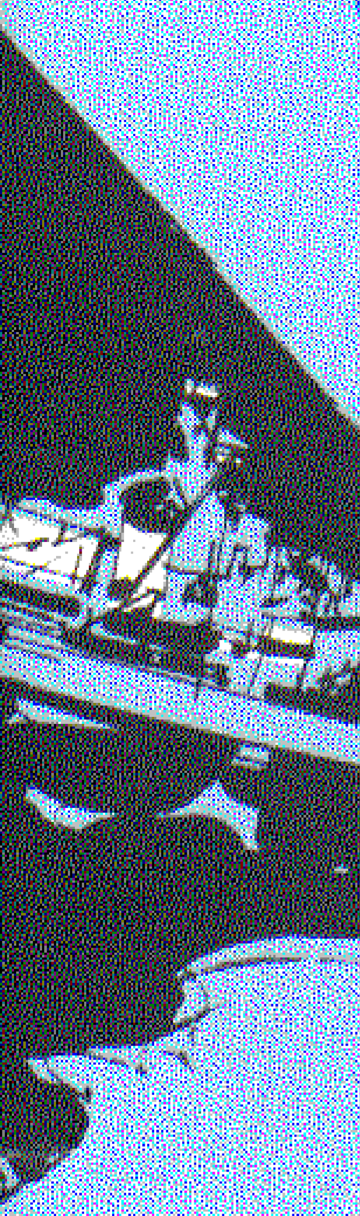

The squadron joined Task Group Delta in 1963 to develop advanced ASW tactics and evaluations. The next years saw VP-56 participate in many joint NATO, Canadian, and U. S. exercises, including yearly deployments to Puerto Rico for "Operation Springboard" and split deployments to Rota, Spain, and Keflavik, Iceland. At NAS Patuxent River, the Dragons became the first operational P-3C squadron in October of 1969, and also assumed duties as the Task Group Delta patrol squadron, developing advanced antisubmarine warfare tactics and sensor equipment.

In 1970, while on deployment to Keflavik, VP-56 earned a Meritorious Unit Commendation. In July 1971, the squadron changed its homeport to NAS Jacksonville, becoming the first P-3C squadron in Fleet Air Wing Eleven. In December 1971, the squadron returned to Iceland for six months, and in June 1972, provided the air element for ASW exercise, "Squeezeplay XI". Also in 1972, the squadron provided support for ASW operations from Lajes, Azores, becoming the first P-3C squadron to support sustained operations from that site.

The Dragons set new records in P-3C availability, crew qualifications, and ASW effectiveness while on deployment to Iceland in March 1973. Ten months later the squadron supported the 6th Fleet in the Eastern Mediterranean during the Yom Kippur crisis.

Deployed to Sigonella, Sicily, in 1974, the squadron flew 4851 hours of accident-free support of the 6th Fleet. The following month a citation for Accident Free Aircraft Operations was awarded to the squadron for 1964 to 1974.

During its successful career, Patrol Squadron Fifty-Six has also been awarded the Captain Arnold J. Isbell ASW Trophy for Excellence in Air Anti-Submarine Warfare and the Battle Readiness Efficiency "E".

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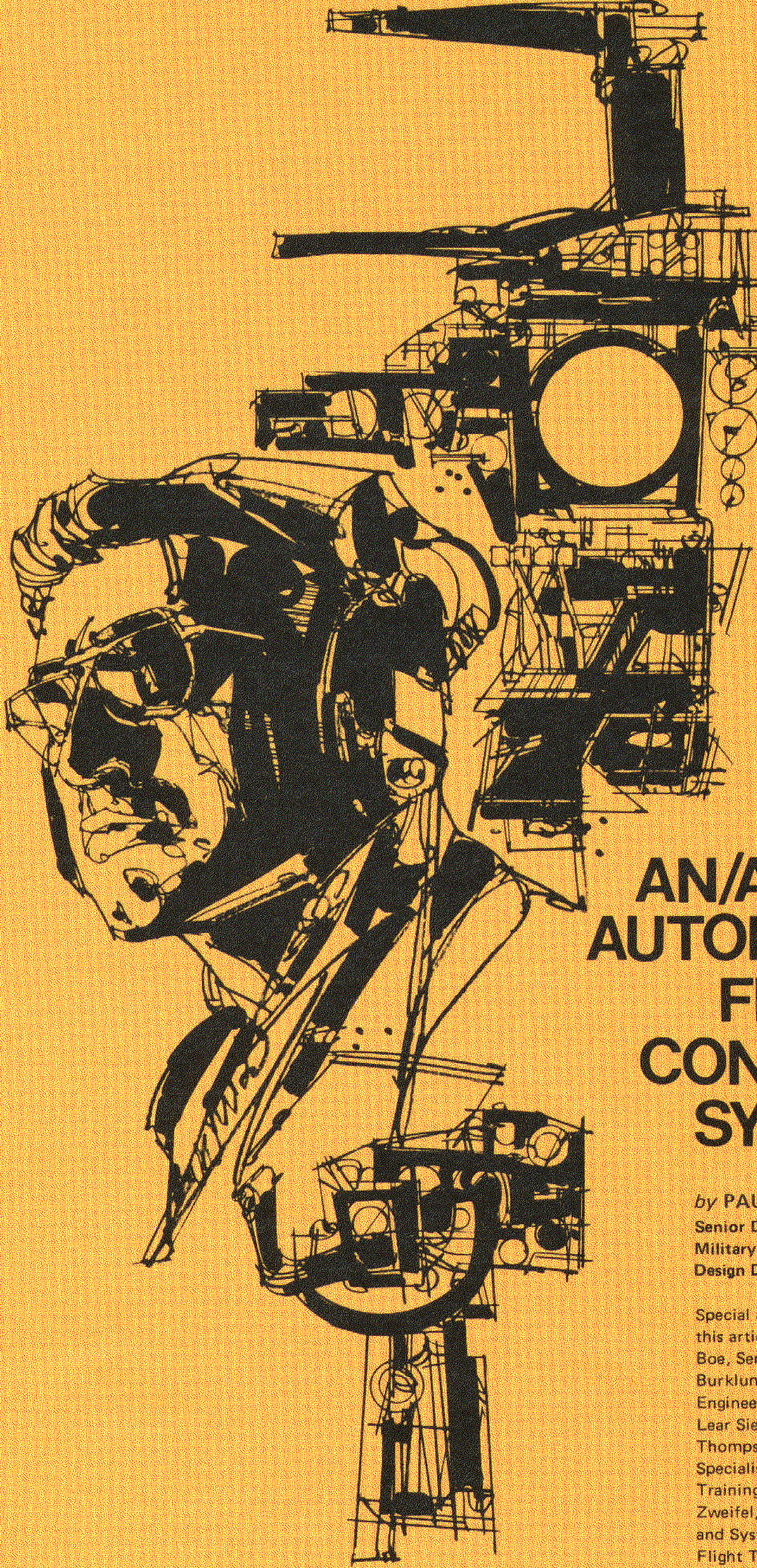

# AN/ASW-31 AUTOMATIC FLIGHT CONTROL SYSTEM

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## INTRODUCTION

The tedious long hours of ASW flight missions under adverse weather conditions and low-level operations require continuing advancement in associated system technology. Sophisticated avionics platforms and highly trained operators aid in producing the desired result in ASW tactics. The data achieved and the success of the missions depend upon the accuracy in the flight navigation profile as well as the crew fatigue factor.

The AN/ASW-31 Automatic Flight Control System (AFCS) supplied by the Astronics Division of Lear Siegler, Inc., was designed into the P-3C with crew comfort and safety in mind. The crew comfort gained from improved rideability, especially in turbulence and ASW maneuvering (due to improved directional damping), decreases the fatigue time factor which directly affects the operators' ability to sort out ASW tactics. Flight safety is enhanced by the use of dual channel fail-passive design which eliminates the aerodynamic transients that can and have occurred in earlier autopilots. The AN/ASW-31 Automatic Flight Control System puts the pilot in the loop for complete AFCS checkout prior to flight and provides all of the necessary monitor functions for cues and alerts.

This article discusses the features and modes of operation of the AN/ASW-31 Automatic Flight Control System.

## FEATURES

**SINGLE AND DUAL CHANNEL OPERATION WITH INDIVIDUAL SEPARATE AXIS ENGAGEMENT** The AN/ASW-31 Automatic Flight Control System is a dual-channel, fail-passive system. A dual-channel system with the capability of single-channel operation was chosen in order to obtain high operational system reliability (availability). During dual-channel operation (normal operation), single-channel malfunctions in the AFCS (excluding the servos) are sensed by comparison monitoring circuitry. Should a failure occur in either channel of any AFCS axis, the failed axis is automatically disconnected without deflecting the aircraft from its flight path, offering a major increase in safety for ASW low-level overwater tactics. Thus, the system, in dual-channel operation, is fail-passive for sensor and electronics faults.

**FULL-TIME PROPORTIONAL CONTROL WHEEL STEERING** This feature is provided to give the pilot/copilot more accurate control over the aircraft and to furnish "pilot-assist" in maneuvers required in execution of the ASW mission. Free aircraft handling characteristics are augmented to provide constant stick force per "g" in pitch and constant roll rate per pound of force applied at the control wheel. In addition, the use of proportional control wheel steering facilitates pilot commanded maneuvers, since aerodynamic disturbances such as gusts, etc., are damped through the rate stabilization networks in the AFCS.

**YAW DAMPING** Yaw damping is improved to a damping factor of approximately 0.7 (essentially, dead beat). In addition, separate axis engagement enables use of yaw damping while other AFCS functions are disengaged.

**BACKUP MODE IN EVENT OF ATTITUDE REFERENCE SYSTEM FAILURE** In the AN/ASW-31 system each channel of the AFCS is normally connected to each of the two inertial systems. In the event of failure of either inertial system, the alternate inertial system may be selected to provide attitude data to both channels of the AFCS. Furthermore, in the event of total loss of attitude reference sources, the AFCS remains engaged but reverts to a "control augmentation mode" in which rate damping is provided and the control wheel steering (pilot assist) mode is still available.

**FULL-TIME ALTITUDE CONTROL** This feature is provided to simplify pilot/copilot tasks required in AFCS operations for capture, holding, or changing altitude. Operation of this feature is accomplished in conjunction with pitch control wheel steering.

**CONTROL AUGMENTATION** A stability augmentation feature is provided in the pitch and roll axes (in addition to yaw damping described previously) under the following conditions:

1. When flying with control wheel steering.

*Table 1. AN/ASW-31 AFCS Monitor Performance*

| DESCRIPTION | FUNCTION | EFFECT |
|---|---|---|
| Channel Comparison Monitor (one provided per axis) | Compares channel signals at input to servo amplifier; operates when failures occur in one channel during dual operation. | Disengages affected axis without aircraft transient. Illuminates flashing red warning lights, AFCS lights on the AFCS status light assemblies, and axis WARN light on AFCS control panel. |
| Servo Monitor (one for each axis in dual operation, and one for each axis in single channel operation) | Operates in event of electrical, hydraulic, or mechanical failure in actuators. | Disengages affected axis with limited aircraft transient in dual operation:<br>Pitch axis - 0.25 G<br>Roll axis - 5.0 deg bank<br>Yaw axis - 2.0 deg sideslip<br>Illuminates flashing red warning lights, AFCS and SERVO lights on the AFCS status light assemblies, and axis WARN light on AFCS control panel. |
| Attitude Signal Comparison Monitor (one for pitch axis, and one for roll axis) | Compares attitude reference input signals to AFCS; operates when discrepancy exceeds 3 deg of attitude; monitors attitude reliability signal. | Warns of attitude reference system mistrack, or failure. Illuminates ATTD lights on the AFCS status light assemblies. If pitch attitude affected while altitude hold is engaged, also causes illumination of the ALT lights on the AFCS status light assemblies and illumination of the flashing red warning lights.<br><br>AFCS affected axis (axes) switched to augmentation mode; attitude loops switch to synchronizing mode. |
| Altitude Deviation Monitor | Compares aircraft altitude performance against engaged reference. Operates when deviation exceeds 60 ft. | Warns of altitude system malfunction. Illuminates flashing red warning lights and ALT lights on the AFCS status light assemblies. |
| Trim Monitor | Compares dual surface load sensor signals and trim drive signal with motor command signal. | Warns of pitch auto-trim malfunction. Disconnects trim servo drive; locks out drive if malfunction persists over 7 seconds. Illuminates AUTOTRIM light on the AFCS status light assemblies.<br><br>Auto-trim OFF flag appears on the three-axis trim indicator. |

2. When attitude reference data are invalid or not available.

With the AN/ASW-31 system, an attitude reference failure will automatically cause the affected axis/axes of the AFCS to revert to the "augmentation mode," with maneuvering capability still available through the control wheel steering feature.

**SYSTEM PERFORMANCE MONITORS** The system performance monitors in the AN/ASW-31 include: Channel Comparison Monitor, Servo Monitor, Attitude Signal Monitor, Altitude Deviation Monitor, and Trim Monitor. These monitors and their functions are summarized in Table 1.

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## DESCRIPTION

The AN/ASW-31 Automatic Flight Control System was derived from the AN/ASW-26 system used in U.S. Navy A-7 Series Aircraft. The general design of the pitch, roll, and yaw control amplifiers and the three-axis rate gyros are taken directly from the AN/ASW-26. Approximately 75 percent of the modules in the above amplifiers, as well as the rate gyro modules, share commonality with their AN/ASW-26 counterparts.

The remaining system elements, monitor control amplifier, barometric altitude control, AFCS control, AFCS test, and MAD maneuver programmer panels, as well as the accelerometers, were specially designed to meet P-3 requirements.

Additional monitoring and redundancy are provided, as well as the capability to select single channel operation. A new servo-actuator interface with in-line monitoring was also developed.

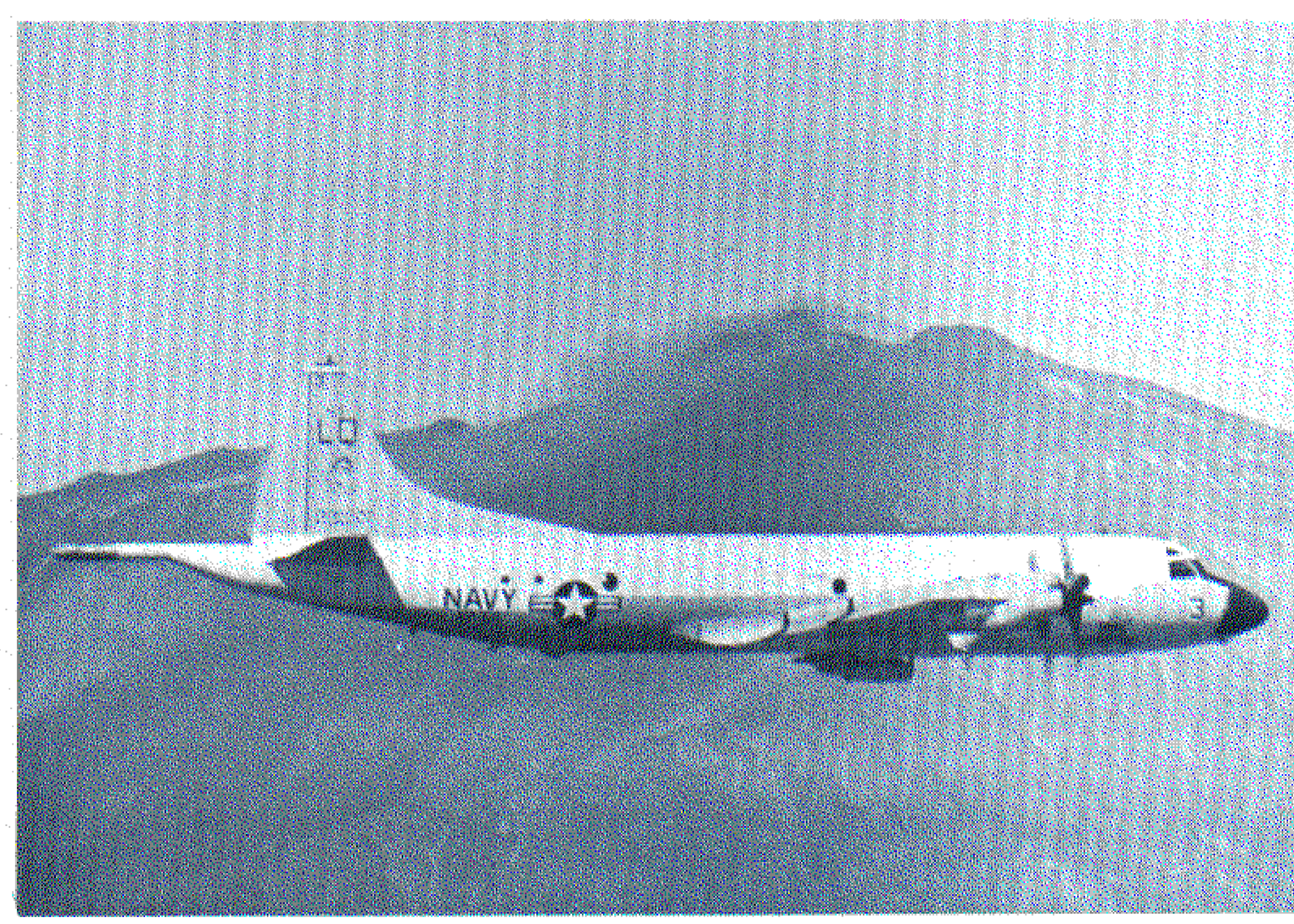
The yaw, roll, and pitch control amplifiers receive sensor signals and develop dc drive voltages to control the rudder, aileron and elevator hydraulic boost packages. The automatic pitch trim control electronics are located within the monitor control amplifier.

The monitor electronics are housed primarily within the monitor control amplifier. Altitude error monitoring and inadvertent disconnect (through the engage switches) monitoring are exceptions; these functions are accomplished within the barometric control and AFCS control panels, respectively.

**POWER DISTRIBUTION** The electric power distribution is configured (to the extent permitted by system design constraints) to minimize loss of unrelated functions resulting from failures in the primary and secondary power distribution system.

**Primary Power** Use of cross-channel signal comparison monitoring and the requirement for signal phase compatibility with interfacing systems, as well as the design of the aircraft electric power system, requires that primary ac power to the AFCS be furnished from a common source which also furnishes power to the interfacing systems.

Thus, primary power to the AFCS is taken from Main AC Bus A (in the Main Load Center), through three single-phase circuit breakers, and through the AFCS power interrupt relay, which is controlled by ground-air sensing logic and the ground power



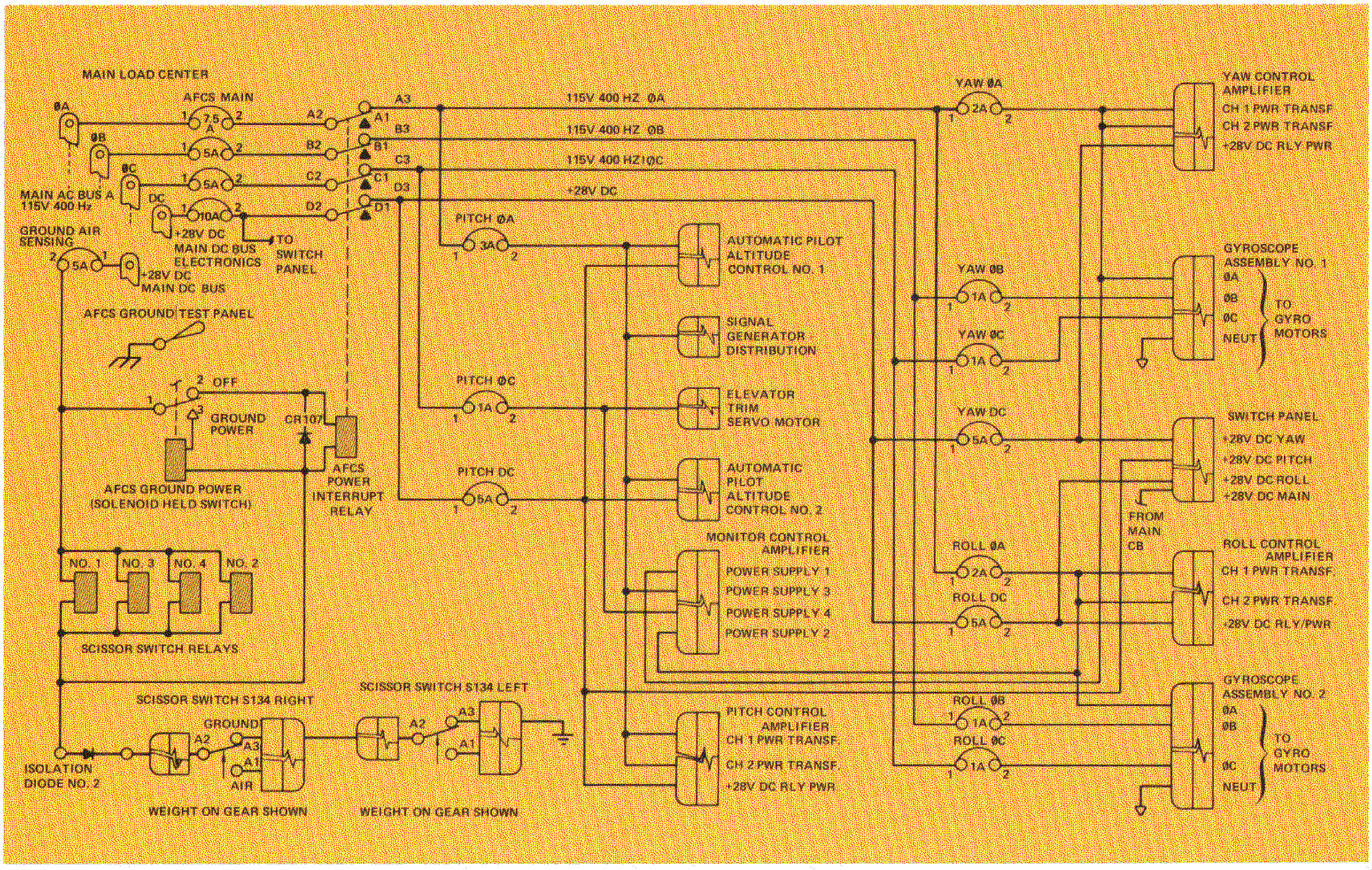



*Figure 1. Primary Power Distribution to AFCS Components*

switch on the AFCS ground power panel. Dc power for relay operation is taken through the AFCS power interrupt relay from the Main DC Bus, which is also in the Main Load Center.

The main AFCS power is then subdivided through separate circuit breakers into a sub-feeder distribution system for each axis, thereby preventing secondary power faults from affecting other axis operation. Figure 1 shows the primary power distribution to the AFCS components.

**Secondary Power** After primary power is routed to pitch, roll, and yaw axis control amplifiers, the ac power is further subdivided through separate power supplies within these units for operation of the electronics of each channel. Each amplifier contains two identical 400-Hz power transformers, one for each channel. Each transformer provides 13 v ac and 26 v ac excitation voltages, including excitation for sensors for each axis and channel, and an isolated 55 v ac from which regulated 39 v dc is developed to power internal modules.

In the monitor control amplifier, space constraints precluded provisioning separate power supplies for each monitor function or for complete axis and channel power isolation. This unit contains four power supplies. The AFCS monitoring functions are subdivided among three of these power supplies, each of which receives ØA 115 v ac 400 Hz current from the primary power distribution source for each axis as shown in Figure 1. The fourth supply receives ØC 115 v ac 400 Hz current from the pitch axis primary power distribution and serves as supply for the pitch autotrim monitor, amplifier, and trim drive. This distribution of the monitoring functions was designed to minimize loss of AFCS functions resulting from failures of any of the supplies. The monitor secondary power distribution is given in Table 2.

**AN/ASW-31 SYSTEM COMPONENTS** The hardware which makes up the P-3C AN/ASW-31 Automatic Flight Control System (AFCS) is listed in Table 3. Each unit is identified by its common and official designations. Figure 2 shows locations of the AFCS components.

*Table 2. Monitor Secondary Power Distribution*

| NO. 1 POWER SUPPLY |
|---|
| Yaw Dual Servo Monitor<br>Pitch Single Servo Monitor<br>Yaw Cross Channel Monitor<br>Yaw No. 1 Gyro Speed Monitor<br>Pitch No. 2 Gyro Speed Monitor<br>Servo Lamp Drive |
| **NO. 2 POWER SUPPLY** |
| Roll Dual Servo Monitor<br>Yaw Single Servo Monitor<br>Roll Cross-Channel Monitor<br>Roll Attitude Monitor<br>Roll No. 1 Gyro Speed Monitor<br>Yaw No. 2 Gyro Speed Monitor |
| **NO. 3 POWER SUPPLY** |
| Pitch Dual Servo Monitor<br>Roll Single Servo Monitor<br>Pitch Cross-Channel Monitor<br>Pitch Attitude Monitor<br>Pitch No. 1 Gyro Speed Monitor<br>Roll No. 2 Gyro Speed Monitor |
| **NO. 4 POWER SUPPLY** |
| Pitch Trim Monitor<br>Pitch Trim Drive |

*Table 3. LSI Astronics AN/ASW-31 AFCS Units*

| ITEM NO. | COMMON NAME | OFFICIAL NAME |
|---|---|---|
| 1* | Panel, AFCS Control | Switch Panel |
| 2* | Amplifier, Pitch Control | Amplifier, Pitch Control |
| 3* | Amplifier, Roll Control | Amplifier, Roll Control |
| 4* | Amplifier, Yaw Control | Amplifier, Yaw Control |
| 5* | Amplifier, Monitor Control | Amplifier, Monitor Control |
| 6 | Rack, Electrical Equipment, AFCS | Base, Shock Mount, Electrical Equipment |
| 7 | Rack, Electrical Equipment Altitude Control | Rack, Electrical Equipment |
| 8* | Control, Barometric Altitude | Control, Altitude, Automatic Pilot |
| 9* | Gyroscope, Three Axis Rate | Gyroscope Assembly |
| 10 | Wheel, Copilot Control | Control Wheel, Copilot |
| 11 | Wheel, Pilot Control | Control Wheel, Pilot |
| 12 | Accelerometer, Normal | Accelerometer, Normal |
| 13 | Accelerometer, Lateral | Accelerometer, Lateral |
| 14 | Panel, AFCS Test | Panel, Test, Electrical |
| 15 | Panel, MAD Maneuver Programmer | Generator-Distribution, Signal |

* Units which have elapsed time indicators.

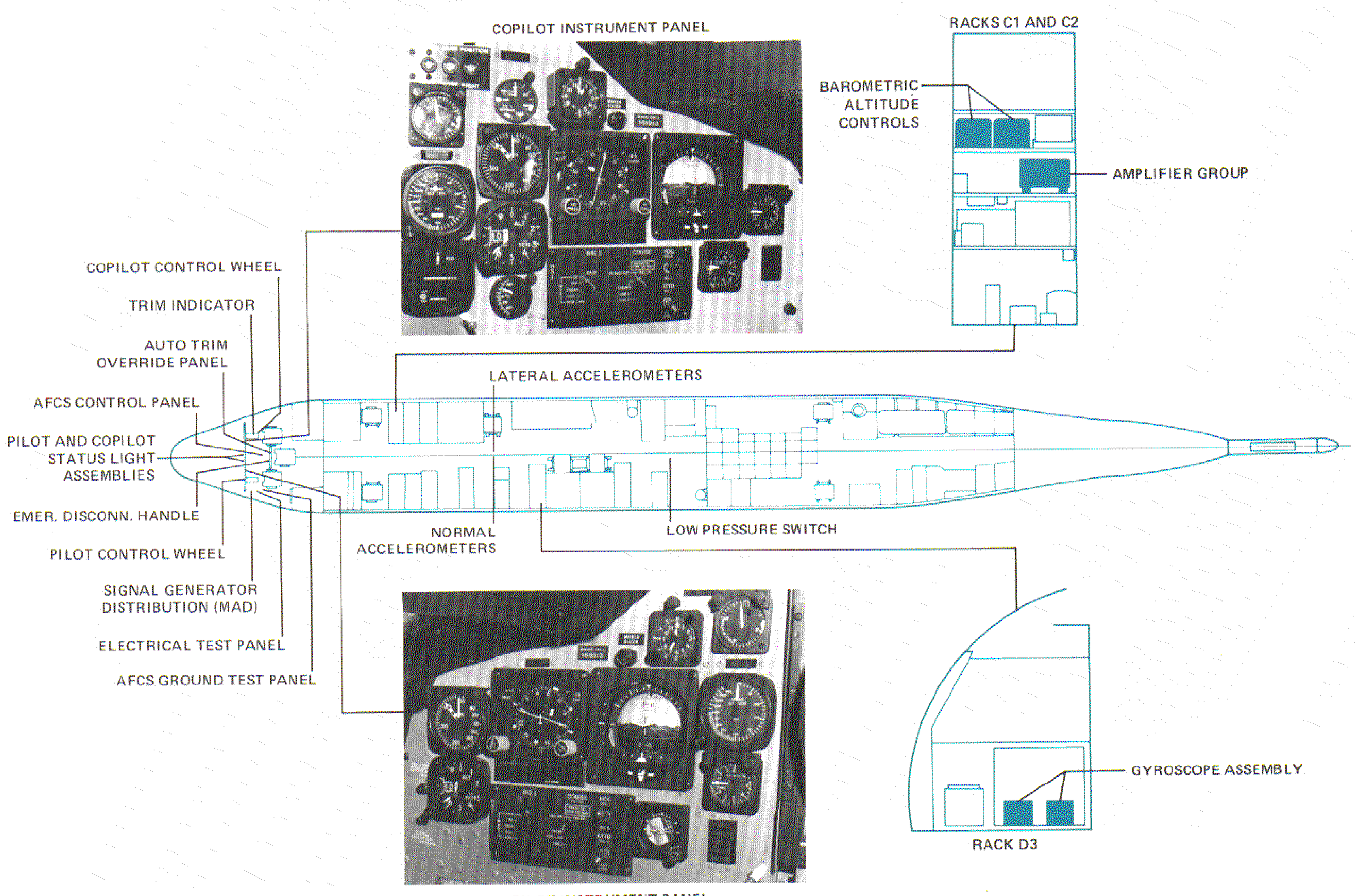



*Figure 2. AFCS Component Locations*

The amplifiers, barometric sensors, and monitor units are installed on shockmounts and are located in the C1 and C2 equipment racks. In the P-3C aircraft the rate gyros are mounted on the floor in the D3 rack* and the accelerometers are located under the aisle floor, near Sensor Station No. 3.

**AFCS Control Panel** The main AFCS control panel (Figure 3) is located on the center console and contains the controls for engaging each control axis, selecting individual channels, and engaging the Heading Select and Altitude Hold modes. It also contains WARN/TEST illuminated pushbutton switches for each control axis. The upper WARN half illuminates if there is a malfunction in the axis circuitry; the lower TEST half illuminates while a test is in progress.

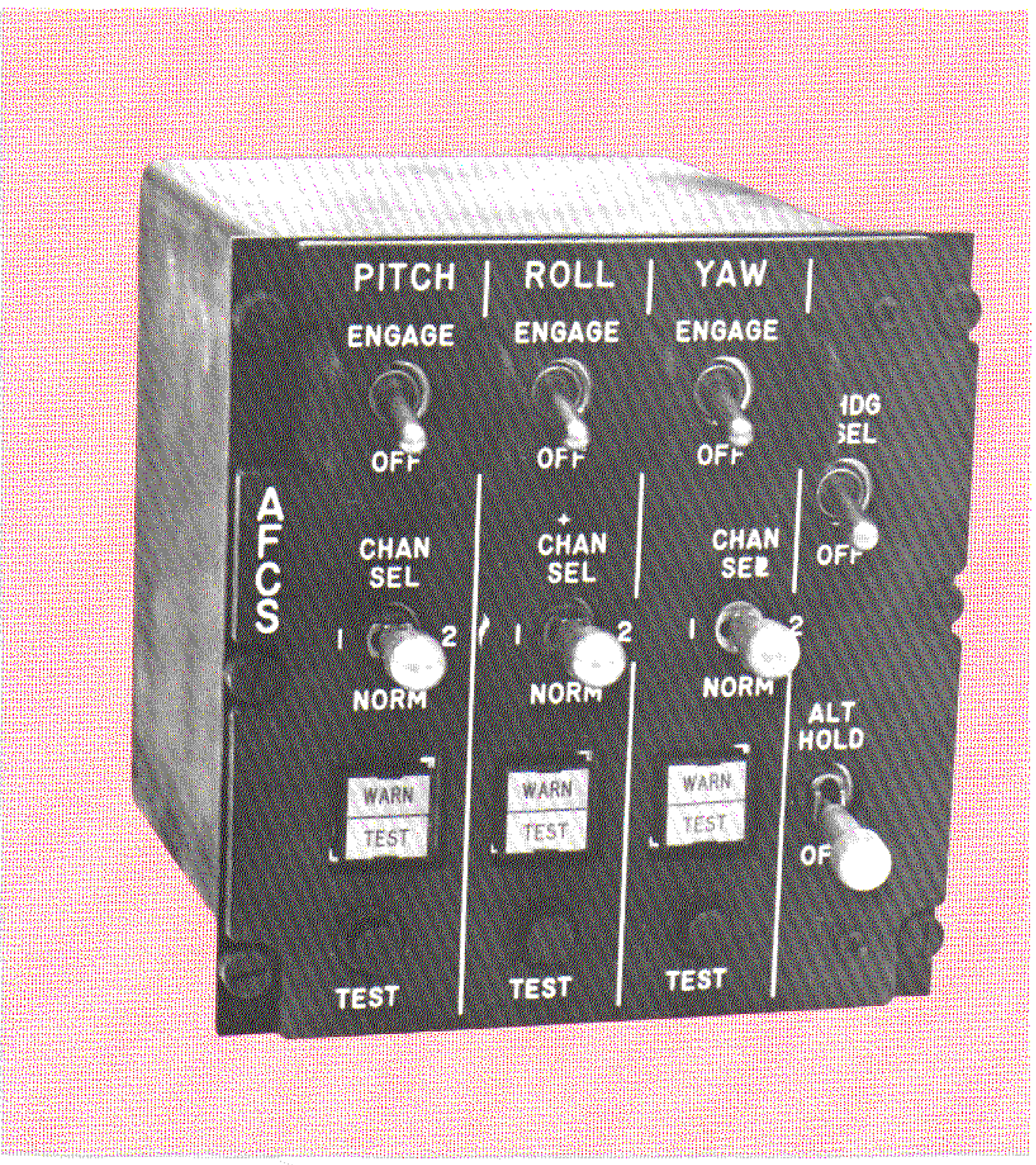



*Figure 3. AFCS Control Panel*

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**In other models of P-3 aircraft equipped with the AN/ASW-31 AFCS, the rate gyros are located elsewhere; however, the function of the gyros remains the same.*

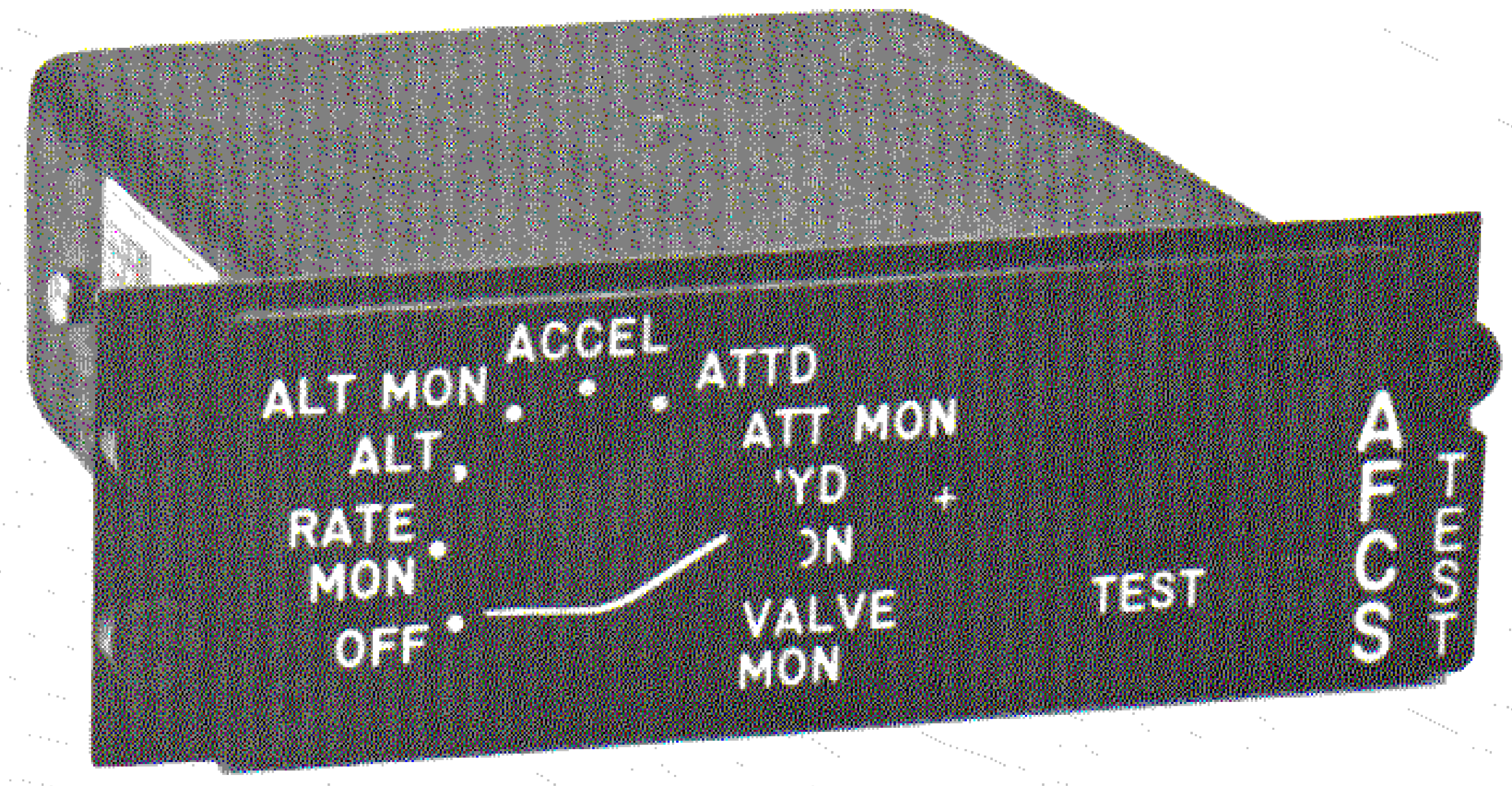



*Figure 4. AFCS Test Panel*

Three momentary TEST pushbuttons at the bottom of the control panel are used for in-flight and ground testing and fault isolation for the control axes. The CHAN SEL switches are level-locked three-position switches: in the center NORM position, the axis is in the dual-channel selected configuration; when moved to the left, only Channel 1 is selected; when moved to the right, only Channel 2 is selected.

When a WARN condition is indicated, the indicator light can be extinguished by depressing the WARN/TEST switch. This also resets the AFCS monitors, extinguishes the flashing red warning lights and the AFCS status lights on the AFCS Status Light Panels. (See "Flashing Red Warning Lights" and "AFCS Status Light Panels" discussed in this section.)

**AFCS Test Panel** The AFCS test panel has a nine-position rotary switch. (See Figure 4.) Any of eight AFCS system functions can be selected and tested by moving the switch to a position and depressing the TEST pushbutton for 1 to 10 seconds.

The AFCS test panel is used exclusively for ground testing. An interlock prevents its use during flight.

**Control Wheels** The pilot's and copilot's control wheels are shown in Figure 5. In addition to providing control wheel steering (CWS) via sensors in the hub, the outboard horn of each wheel contains a dual-detent pushbutton switch on the forward side. The first detent position disconnects the AFCS Altitude Hold mode; the second position disconnects the AFCS entirely.

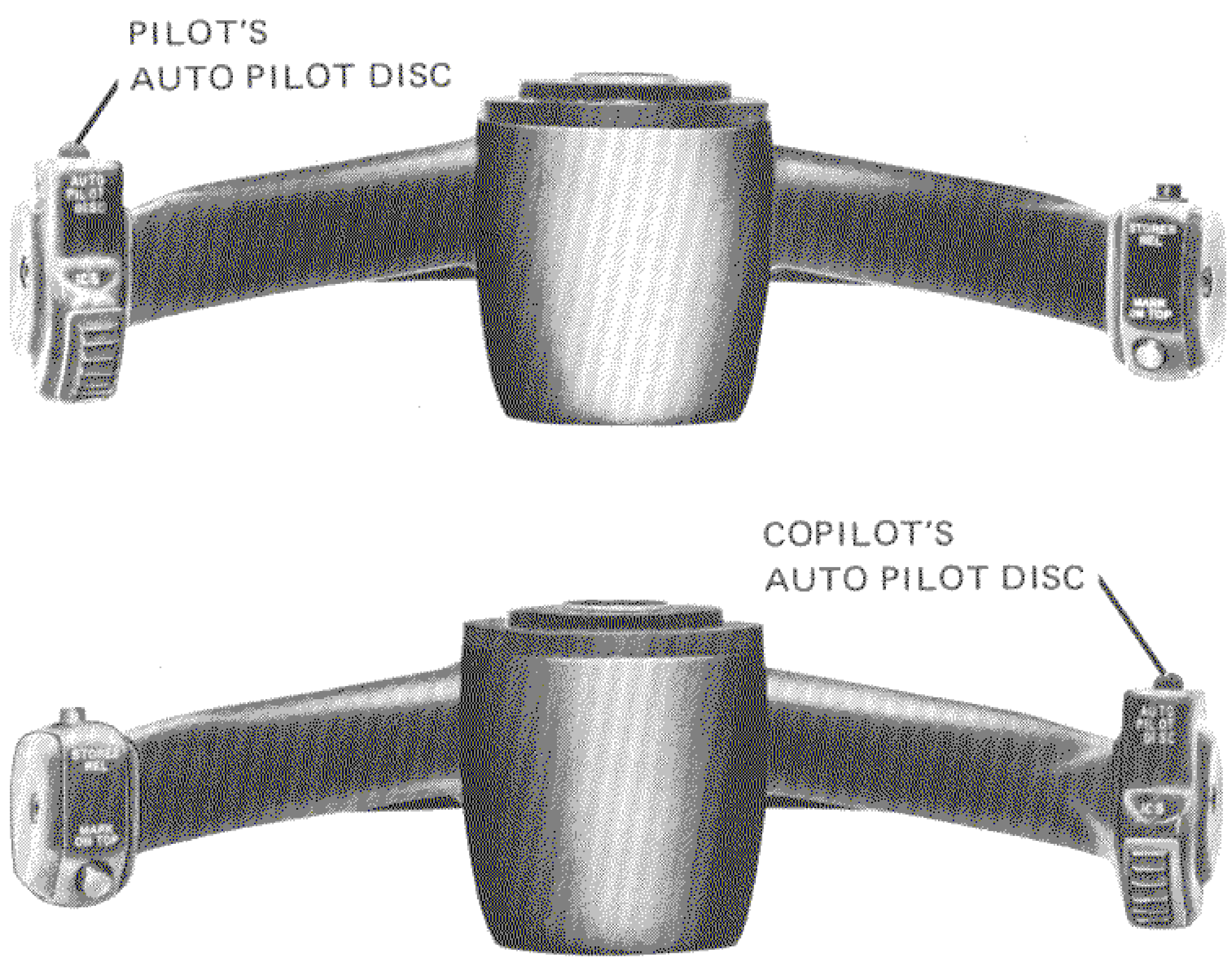
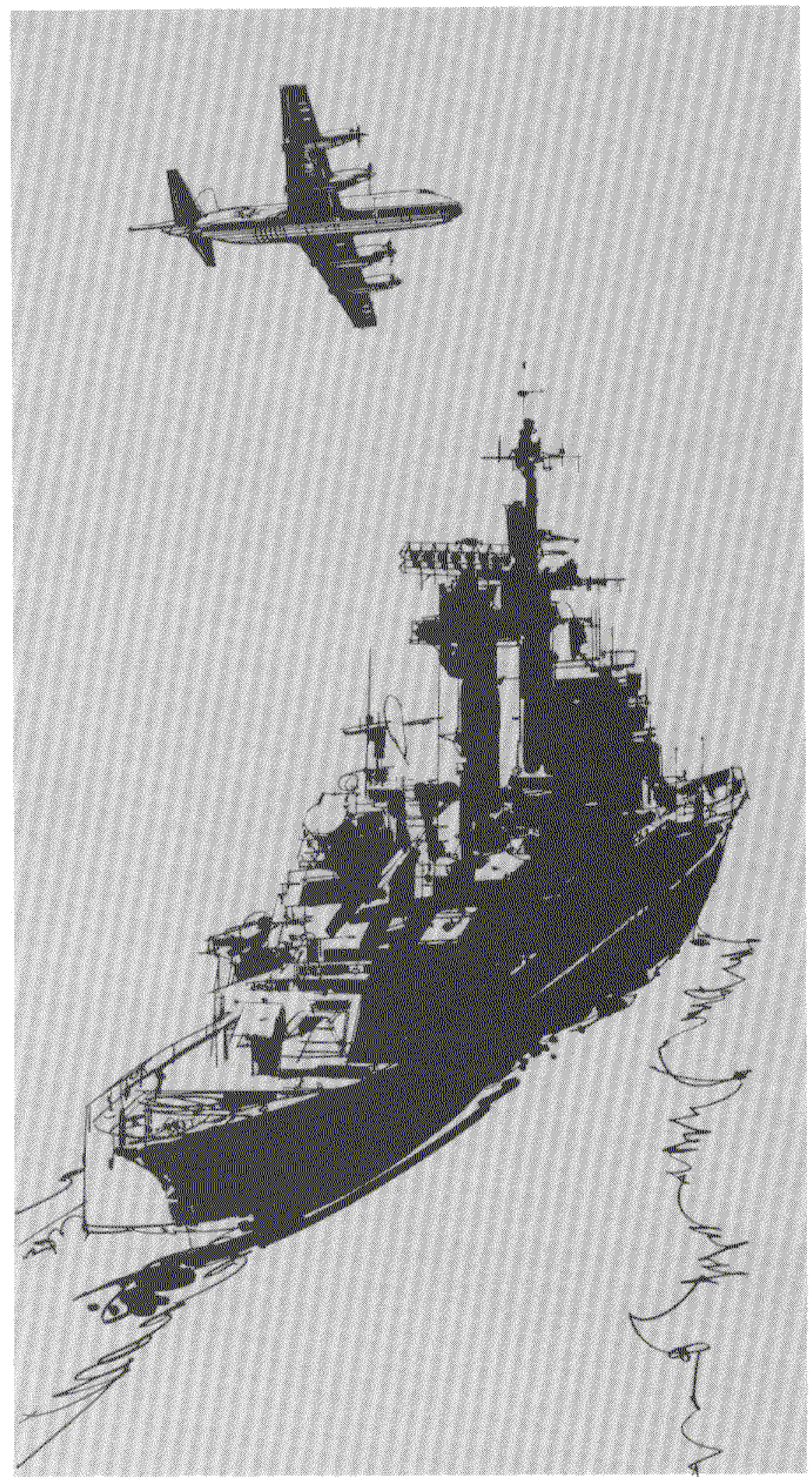



*Figure 5. Pilot's and Copilot's Control Wheels*

**AFCS Status Light Panels** The five AFCS status panel lights (Figure 6) indicate disengagement of the AFCS and malfunctions in the attitude reference system, servo booster system, altitude hold function, and the automatic pitch trim system. The AFCS light illuminates when automatic disengagement occurs following a failure; when manual disengagement occurs by positioning the ENGAGE switch OFF or by moving the CHANNEL SELECT to a new position; or when disengagement occurs due to hydraulic system No. 1 failure, or electric power failure. The ATTD light illuminates whenever there is a difference of more than 3 degrees (approx.) between the two attitude reference systems or if either reference system has failed. The SERVO light indicates malfunctions in the servo amplifiers or the boosters, and operates in conjunction with the AFCS light and the flashing red warning lights. The ALT light indicates malfunctions in the altitude hold function system or disengagement of the Altitude Hold mode. The AUTOTRIM light indicates a pitch autotrim malfunction.

**Flashing Red Warning Lights** The red flashing warning lights, located on the pilot's and copilot's glareshield, alert the pilot and copilot to AFCS malfunctions. The warning lights are illuminated whenever the AFCS is disconnected due to a system malfunction. They also illuminate when



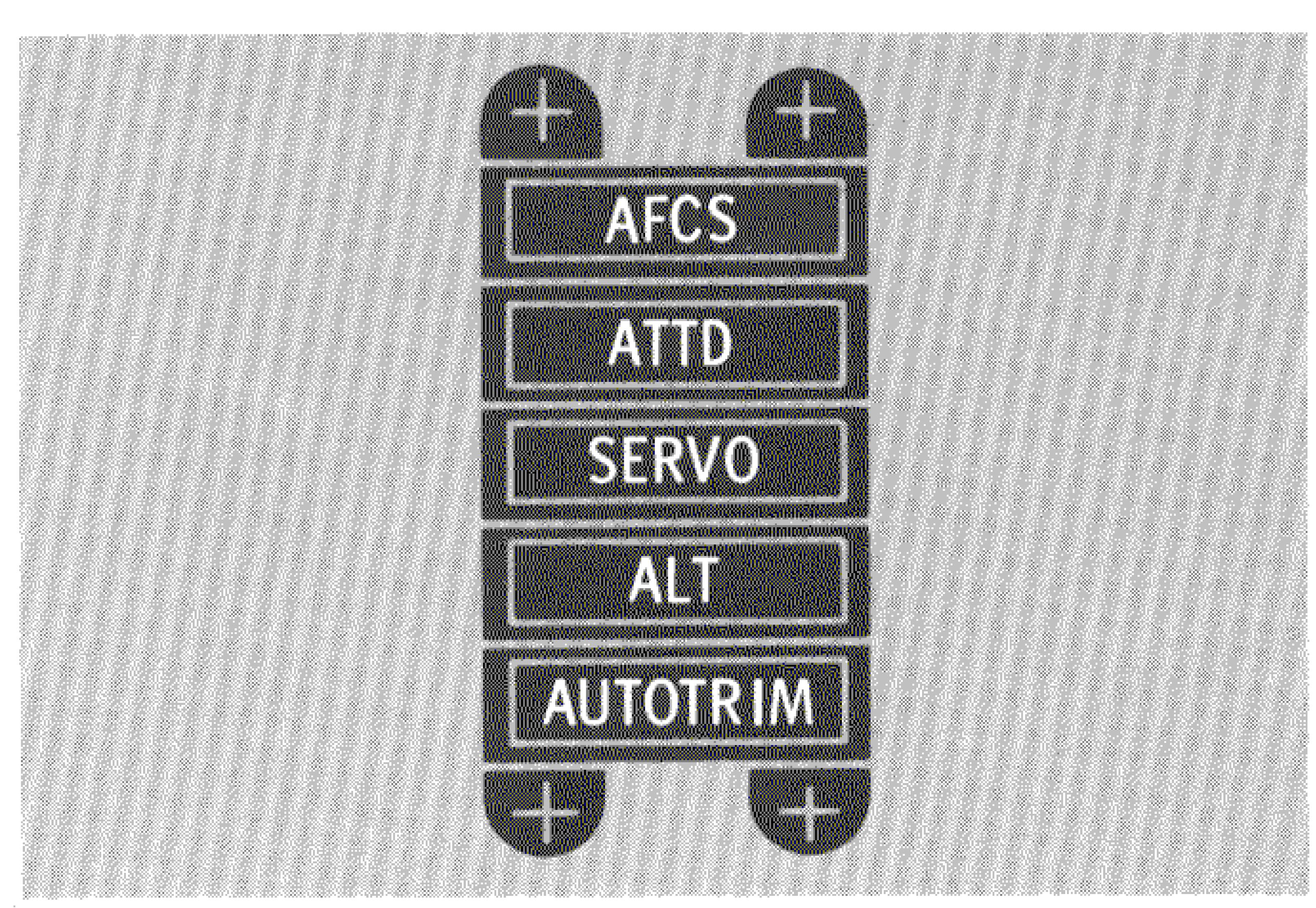



*Figure 6. AFCS Status Panel Lights*

there is a deviation of more than 60 ft from initial altitude engagement, or when there is loss of the pitch attitude reference when the AFCS is in the Altitude Hold mode. Manual disconnect of the AFCS at the control wheel or by use of the AUTOPILOT EMER DISCONN handle will not energize the red warning lights. Conditions annunciated by these lights and the AFCS status lights are summarized in Table 4.

**MAD Maneuver Programmer Panel** The magnetic anomaly detection (MAD) maneuver programmer panel (Figure 7) is used to select an axis about which maneuvers are performed to allow calibration of the MAD equipment. The four-position selector switch applies cyclical roll, pitch, or yaw commands to the selected axis. A toggle switch controls the operation of the programmer. The roll and yaw crossfeed adjustment is factory preset but is accessible for maintenance purposes.

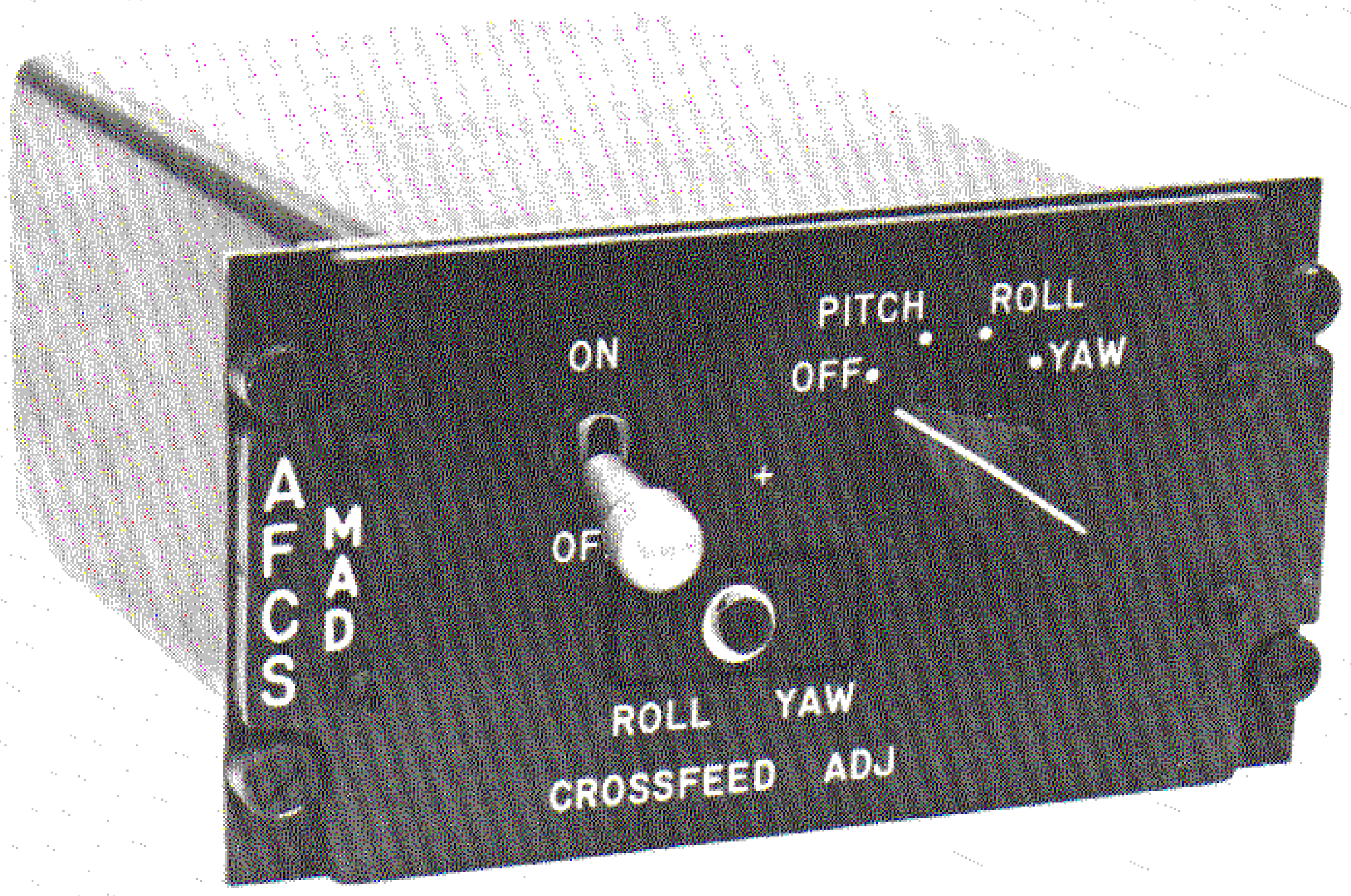



*Figure 7. MAD Maneuver Programmer Panel*

**Three-Axis Trim Indicator** A three-axis trim indicator is installed on the copilot instrument panel (Figure 8). This instrument serves two purposes:

1. When the automatic pilot is disengaged, each

*Table 4. AFCS Status and Warning Lights*

| CONDITIONS | RED FLASHER | AXIS WARN | STATUS LAMP | | | | |
|---|---|---|---|---|---|---|---|
| | | | AFCS | ATTD | SERVO | ALT | AUTO-TRIM |
| AXIS AUTOMATIC DISENGAGE | X | X | X | | | | |
| AXIS AUTOMATIC DISENGAGE DUE TO SERVO FAILURE | X | X | X | | X | | |
| AXIS MANUALLY DISENGAGED VIA ENGAGE SWITCH | X | X | X | | | | |
| AXIS DISENGAGED BY MOVING CHANNEL SELECT SWITCH | X | X | X | | | | |
| LOSS OF ELECTRIC OR HYDRAULIC POWER | X | X | X | | | | |
| ATTITUDE REFERENCE FAILURE | | | | X | | | |
| ATTITUDE REFERENCE FAILURE WITH ALT HOLD ENGAGED | X | | | X | | X | |
| ALTITUDE DEVIATION | X | | | | | X | |
| TRIM SYSTEM FAILURE | | | | | | | X |
| RADIO ALTITUDE WARNING SYSTEM | X | | | | | | |
| PITCH AXIS ENGAGED WITH ALT HOLD NOT ENGAGED | | | | | | X | |


*Figure 8. Three-Axis Trim Indicator*

axis element of the indicator receives its input signals from the respective axis autopilot output (transfer valve signal). Under steady-state, normal conditions the three-axis trim indicator bars (elements) should be centered. Element displacement serves to alert the pilot of the existence of an unsynchronized offset error which will cause an aircraft transient if the automatic pilot were to be engaged under this condition. Thus, the indicator serves as a malfunction indicator while the system is in its standby (disengaged) mode.

2. When the automatic pilot is engaged, each element receives its input signals from hydraulic load sensors in the respective axis flight control boosters. In this case, the instrument indicates the relative force (hinge moment) held on the respective surface by the automatic pilot. The EL (Elevator) bar is normally maintained at null (centered) by the automatic pitch trim system, while manual trim is required to center the AIL (Aileron) and RUD (Rudder) bars. Automatic or manual disengagement under mis-trimmed condition (displaced bar) will be accompanied by an aircraft transient.

An OFF flag appears when the automatic pitch trim system is not operating.

**Horizontal Situation Indicator** The pilot's Horizontal Situation Indicator (HSI) provides a heading signal to the roll control amplifier when the AFCS is in the Heading Select mode. The pilot selects the reference heading with the heading set knob on the HSI.

**Manual Pitch Trim Wheels** When the AFCS pitch axis is ENGAGED in the Altitude Hold or Attitude Hold modes, the AFCS automatically trims the elevator to suit power and configuration changes. The manual pitch trim wheels move as these changes take place. The pilot can use these trim wheels to manually override automatic pitch trim, if desired.

**Autopilot Emergency Disconnect** The autopilot emergency disconnect handle, located under the aft end of the center control pedestal, is used to disconnect the AFCS if all other attempts should fail. Pulling the handle out about 6 inches disengages all three AFCS axes and physically disconnects the AFCS from the servos.

**Autotrim Light Override and Attitude Select Panel** The autotrim light override and attitude select panel (Figure 9) is located on the center pedestal adjacent to the bottom of the AFCS control panel. The magnetically held AUTOTRIM OVERRIDE switch extinguishes the AUTOTRIM status light on the AFCS status panels. This switch is solenoid operated and is controlled by wing flap logic to turn the light back on when the flaps are in take-off, approach, or land positions.


*Figure 9. Autotrim Light Override and Attitude Select Panel*

The ATTD SEL switch serves to connect both channels of both the roll and pitch axes to either of the two attitude reference systems. In the center NORM position, AFCS Channel 1 is connected to INS 1 and Channel 2 is connected to INS 2.

**Ground Test Control Panel** Power is automatically removed from the AFCS when there is weight on the landing gear. The ground power switch on the ground test control panel bypasses the "weight-on-gear" function to enable ground checks of the AFCS. Figure 10 shows the ground test control panel.

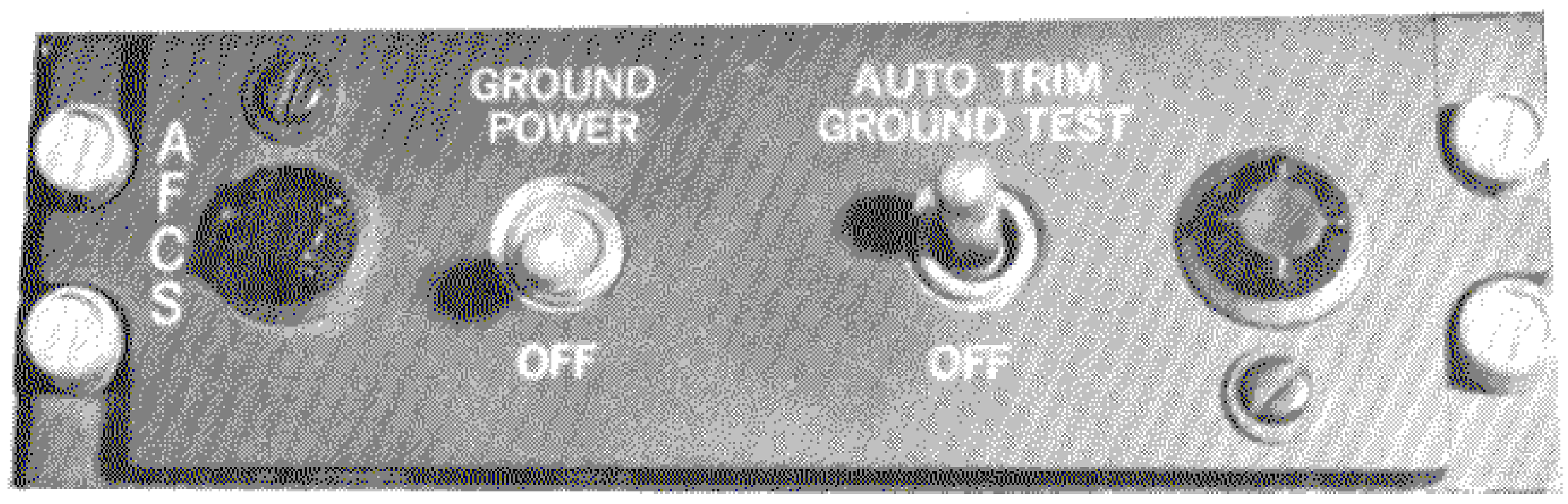



*Figure 10. AFCS Ground Test Control Panel*

During normal ground operation, interlocks prevent operation of the pitch autotrim system when the AFCS pitch axis is engaged and the Ground Power switch is in the ON position. This prevents the trim motor from driving against the stops when the aircraft is on the ground. Ground operation of the autotrim system is obtained by placing and holding the spring-loaded AUTOTRIM GROUND TEST SWITCH in the TEST position.

**Three-Axis Rate Gyro** The three-axis rate gyro assembly which is rigidly mounted to the airframe, contains three individually-mounted rate gyros – a yaw, roll, and pitch gyro in each. All three gyros contain torquers that are used for self-test. Two of these gyro assemblies, mounted side-by-side, give the system dual redundancy.

**Accelerometer** Two types of accelerometers (normal and lateral) are used. They are identical except for operating range and connector keying. Both accelerometers are excited by 26 v ac 400 Hz and both present a full scale output on application of a 28 v dc self-test signal.

**Barometric Control** The barometric altitude control unit (Figure 11) has the primary function of providing, as a pitch axis input, an ac signal proportional in amplitude and phase to the change in altitude from that at which the ALT HOLD mode is engaged. In addition, the unit contains a self-test circuit; a monitor to determine deviations greater than 60 ft from reference altitude when in the ALT HOLD mode; and circuitry to derive the altitude rate.

**Amplifier Group** The amplifier group consists of the yaw, roll, pitch, and monitor control amplifiers (Figure 12). They are housed in a common, EMI shielded and gasketed, shock-mounted equipment rack. Rear-mounted connectors are used for all system interconnects. Connectors on the front allow access to important test points for flight line troubleshooting. An elapsed time indicator is on the front panel of each amplifier to record operating time.

**Hydraulic Servo Actuator (Boost Package)** There are three hydraulic servo actuators, one for each axis of control. Each boost package accepts an electrical signal from the autopilot and directs hydraulic pressure to move its control surface. The autopilot command signal is a dc differential current applied to the windings of the boost package transfer valve. The transfer valve directs fluid flow to the modulating piston which in turn controls the main ram through the booster control valve.

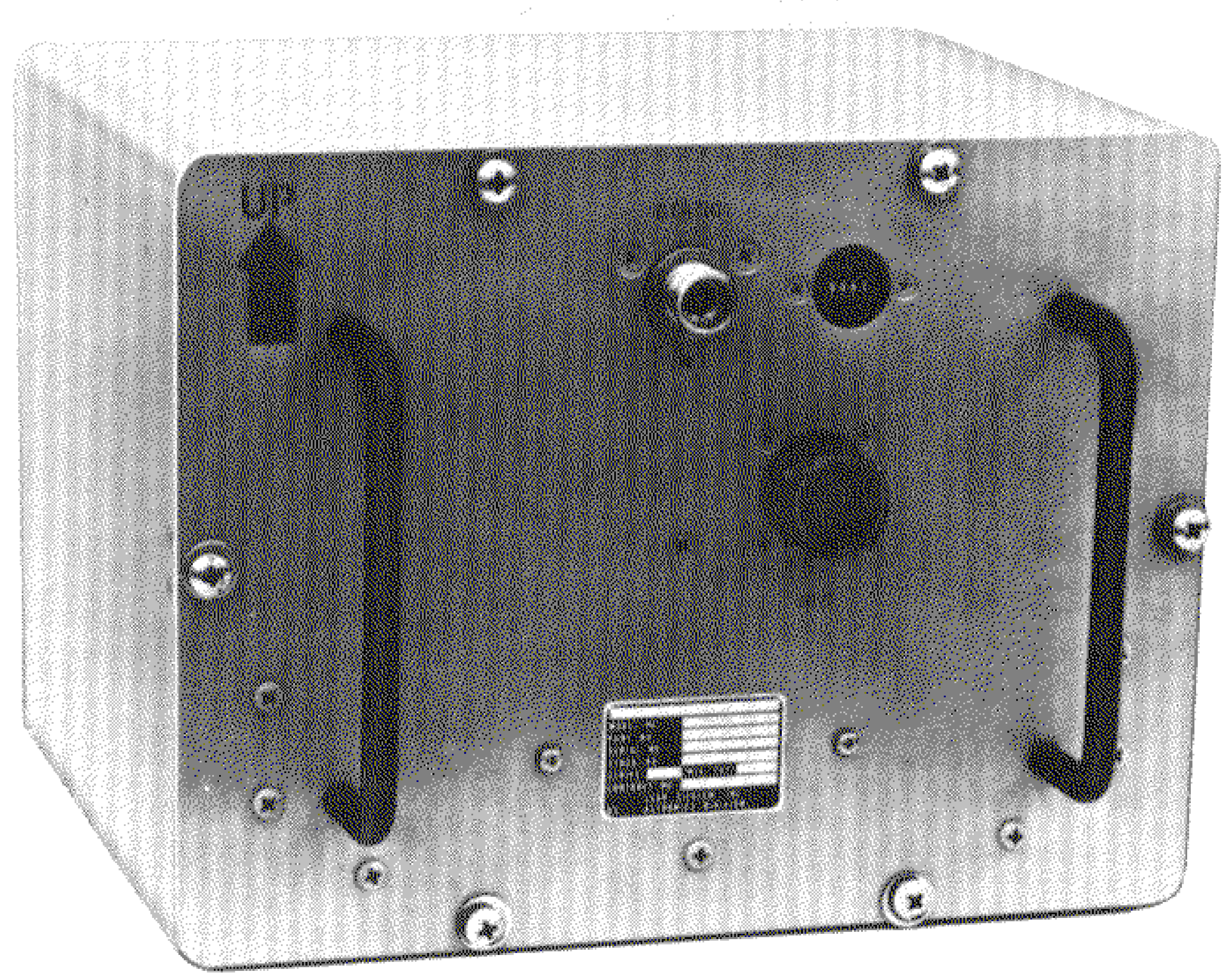



*Figure 11. Barometric Altitude Control Unit*

The surface position control loop requires two feedback signals from each servo – surface position and surface rate. Surface position is provided by a dual synchro pickoff on the power ram. The surface rate signal is provided by a dual linear variable differential transformer (LVDT) pickoff of modulating piston position. These dual-channel transducers replace single-channel transducers previously used in the PB-20N.

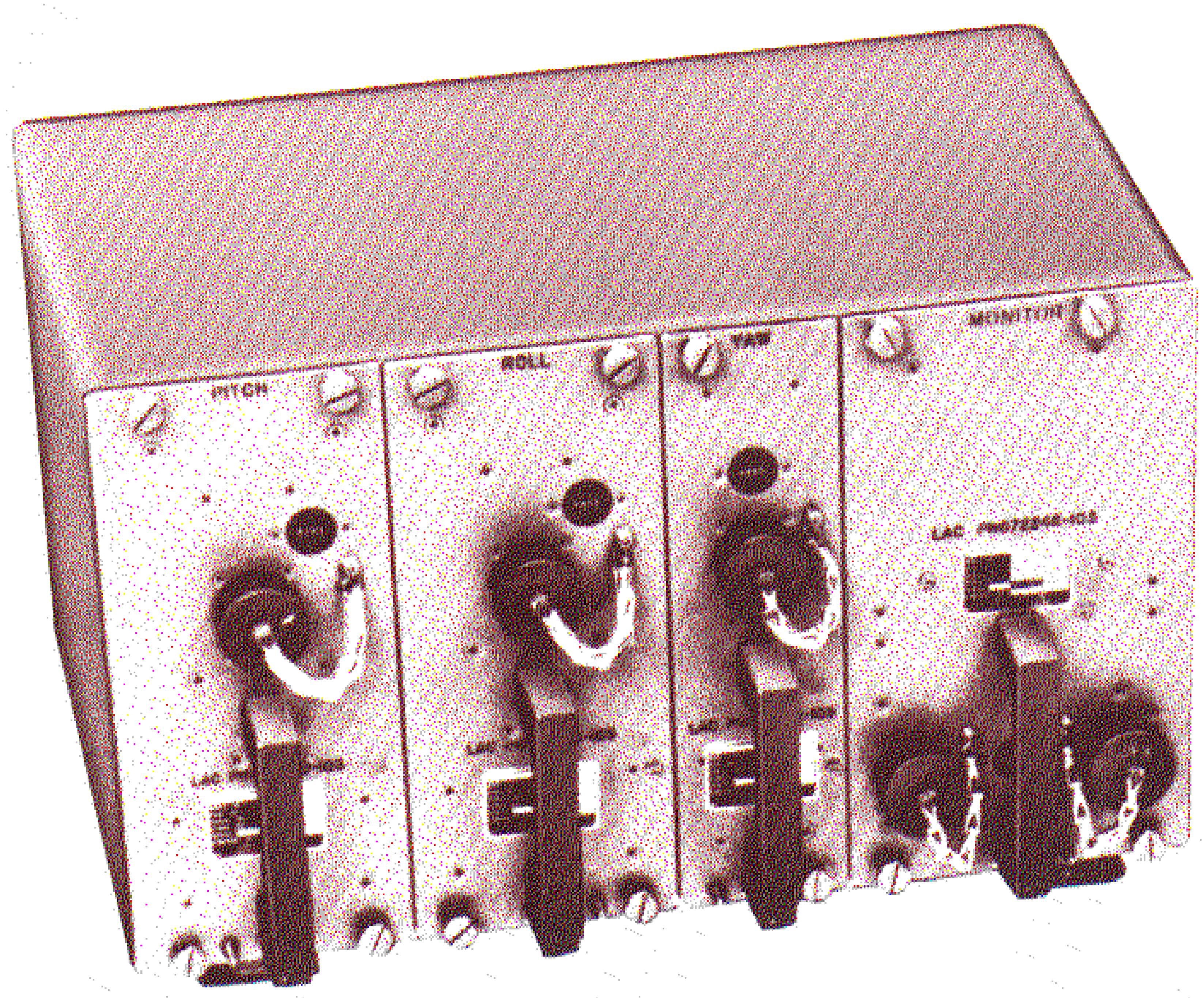



*Figure 12. Yaw, Roll, Pitch, and Monitor Control Amplifiers*

**Trim Servo** The pitch trim servo and disconnect (clutch) used in the PB-20N were retained in the AN/ASW-31. The electronics to drive and monitor the trim servo are located in the monitor control amplifier and utilize the elevator hydraulic load sensor signals from the elevator boost to control the trim tab position.

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## MODES OF OPERATION

**PITCH AXIS** The AFCS pitch axis operates in the following modes: Attitude Hold, Control Wheel Steering, Altitude Hold, Automatic Pitch Trim, and MAD Maneuver. The following reference signals are used to control pitch axis operation: pitch attitude, normal acceleration, pitch rate, barometric altitude, control wheel forces, versine (1 – cos $\phi$), and hydraulic servo feedback signals.

**Attitude Hold** The Attitude Hold mode of operation is enabled if the pitch axis is engaged when the pitch attitude of the aircraft is less than 22 degrees. The AFCS will then maintain the aircraft at the attitude at engagement. If the attitude is greater than 22 degrees, the AFCS commands the aircraft to return to and maintain an attitude of 22 degrees. The pilot may alter the aircraft attitude through use of control wheel steering.

**Control Wheel Steering** Pitch control wheel steering is engaged by exerting a pitch force on the control wheel in excess of 2 lb. The pitch maneuver is then proportional to the force exerted on the control wheel. When the force on the wheel is reduced below 2 lb, pitch attitude hold is enabled again. Pitch control wheel steering is not available while altitude hold is engaged.

**Altitude Hold** Altitude hold is the basic pitch mode, so when the AFCS pitch axis is engaged, normally the ALT HOLD/OFF switch will be placed in the ALT HOLD position. The Altitude Hold mode will be enabled and will maintain the aircraft at a desired altitude if the following conditions are satisfied:

- The pitch axis is engaged.
- Vertical speed is less than 300 ft/min.
- ALT HOLD/OFF switch is positioned to ALT HOLD.
- Pitch wheel force is less than 2 lb.
- Neither control wheel disengage switch is depressed to the first detent.

Once altitude hold is enabled, subsequent application of pitch wheel force will not disable this mode. If a different altitude reference is desired, depress the control wheel disconnect switch to the first detent position. The altitude can then be altered by application of a pitch wheel force in excess of 2 lb. (This action enables pitch control wheel steering.) If the vertical speed is greater than 300 ft/min, the control wheel disconnect switch may be released. (If less than 300 ft/min the switch must be held depressed until the desired altitude is reached.) At the desired altitude, reduce the vertical speed to less than 300 ft/min and the force on the wheel to less than 2 lb. The AFCS will then capture and maintain the new altitude. If the altitude of the aircraft deviates from the reference altitude by more than 60 ft, the flashing red cockpit warning lights and the ALT status light will illuminate; however, the AFCS will not disengage. The ALT status lights will also illuminate if the pitch axis is engaged when the Altitude Hold mode is not enabled.

**Automatic Pitch Trim** The AFCS will automatically trim the elevator as required to compensate for power or configuration changes when the Pitch Attitude or Altitude Hold modes are enabled. In the event of a malfunction in the autotrim circuitry, the trim monitor will deenergize the servo clutch to disconnect the autotrim servo, and a warning flag in the three-axis trim indicator will become visible. Also, the AUTOTRIM lights on the AFCS status panel will illuminate. This indication can be extinguished by use of the AUTOTRIM OVERRIDE switch on the autotrim override panel. However, this switch is inoperative (will not extinguish the light) if the wing flaps are in the takeoff, land, or approach positions.

**MAD Maneuver** To enable MAD maneuver in the pitch axis, the pitch axis must be engaged, the pitch position selected on the MAD maneuver panel, and the ON/OFF switch on the MAD panel positioned to ON. The Altitude Hold mode should also be engaged. The AFCS will then command the aircraft to pitch ±3 degrees in an oscillatory maneuver, the period of which is 4 to 6 seconds.

**ROLL AXIS** The roll axis operates in the following modes: Heading Hold, Attitude Hold, Heading Select, Control Wheel Steering, and MAD Maneuver. The following reference signals are used to control roll axis operation: roll rate, roll attitude, heading signals, control wheel forces, and hydraulic servo feedback signals.

**Heading Hold** The Heading Hold mode is enabled if the roll axis and yaw axis are both engaged and the roll attitude is less than 2 degrees. At this time the autopilot will maneuver the aircraft to the wings level attitude and maintain the heading at engagement.

**Attitude Hold** The Attitude Hold mode is enabled if the roll axis is engaged when the roll attitude of the aircraft is greater than 2 degrees but less than 45 degrees. When in this mode the autopilot will maintain the reference attitude. If the bank angle exceeds 45 degrees at engagement, the autopilot will roll the aircraft back to 45 degrees and maintain this attitude.

**Heading Select** The Heading Select mode is enabled by placing the HDG SEL/OFF switch to the HDG SEL position. However, both the roll and yaw axes must be engaged first. In this mode the autopilot will direct the aircraft to fly the heading selected on the pilot's HSI.

**Control Wheel Steering** The Control Wheel Steering mode is enabled by exerting a force on the control wheel in excess of 2 lb. The roll maneuver will then be proportional to the force applied to the control wheel. This mode overrides any of the above discussed modes that have been engaged previously.

**MAD Maneuver** With the roll axis engaged and in Heading Hold mode (less than 2 degrees of roll attitude), and the roll MAD maneuver selected on the MAD programmer panel, the autopilot will roll the aircraft ±10 degrees while maintaining the heading within ±1 degree. The yaw axis must also be engaged to prevent excessive yaw excursions.

**YAW AXIS** The yaw axis operates in the following modes: Yaw Damping, Turn Coordination, and MAD Maneuver. The following reference signals are used to control yaw axis operation: yaw rate, roll attitude (for turn coordination), lateral acceleration, and hydraulic servo feedback signals.

**Yaw Damping** Yaw damping is provided when the yaw axis is engaged. This mode damps out (smoothes) undesirable excursions around the yaw axis.

**Turn Coordination** When the yaw axis is engaged, it provides proper rudder deflection to ensure the performance of coordinated turns.

**MAD Maneuver** With the yaw axis engaged and the yaw MAD maneuver selected on the MAD programmer panel, the autopilot will yaw the aircraft ±5 degrees while maintaining a roll attitude of ±1 degree or less. The roll axis must also be engaged to prevent excessive roll excursions.

## AFCS SIGNAL PATHS

**SERVO AMPLIFIER AND EQUALIZERS** The servo interface typical for each of the yaw, roll, and pitch axes is shown in Figures 13, 14, and 15. Each axis contains two independent equalization networks; one for axis command equalization, the other for servo loop equalization. The purpose of the equalizers is to eliminate allowable static and dynamic channel and servo loop mismatches, which may be present in any system due to sensor and electronic tolerance, null offsets, etc. Equalization permits use of more sensitive servo monitors than would otherwise be possible.

The servo interface is completely dual, down to and including the power demodulators which drive the single hydraulic transfer valve.

In single channel operation, one of the power demodulators is selected to drive the hydraulic transfer valve alone. Both equalizers are switched out in single channel operation.

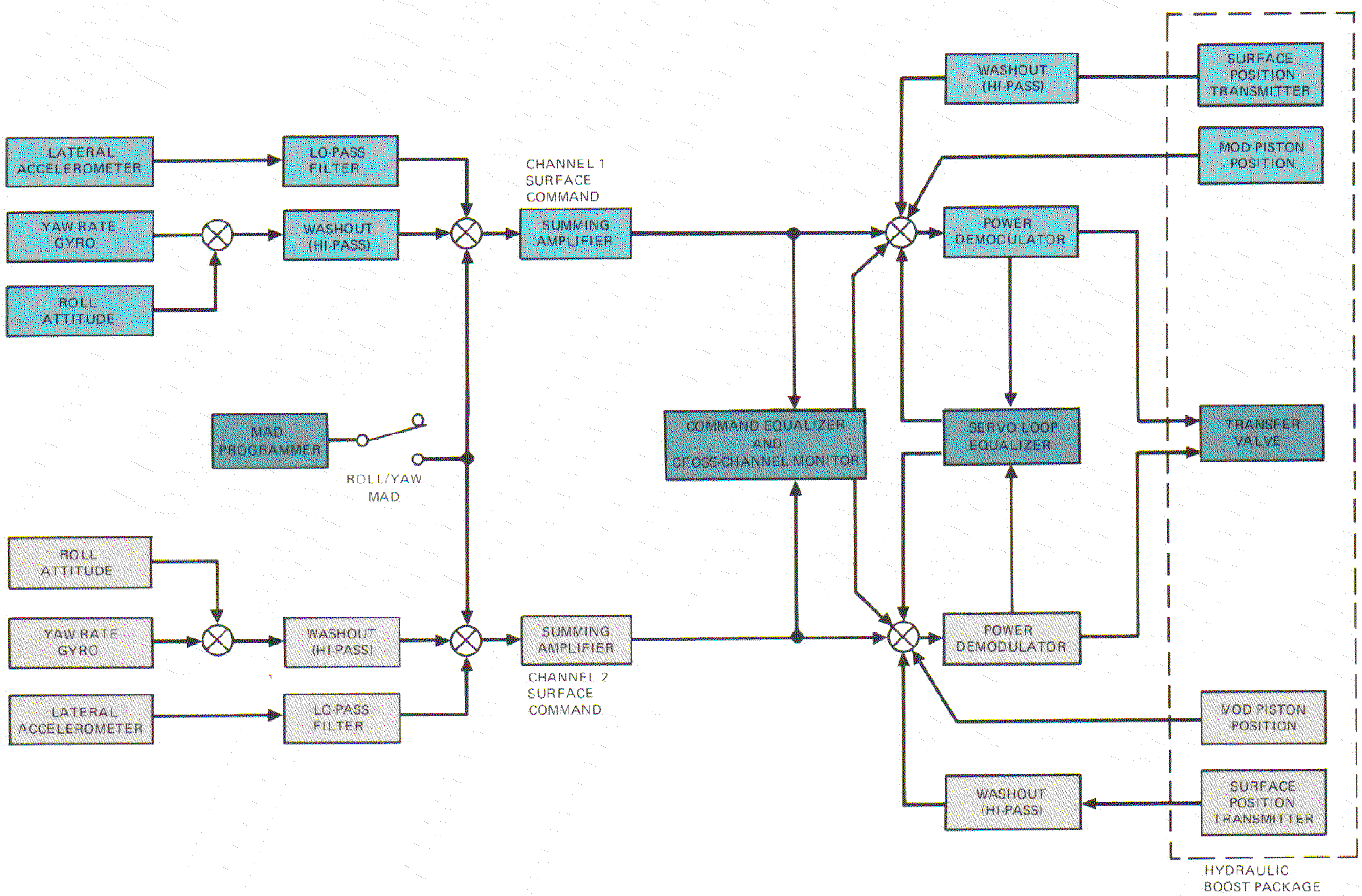



*Figure 13. Yaw Axis Functional Block Diagram*

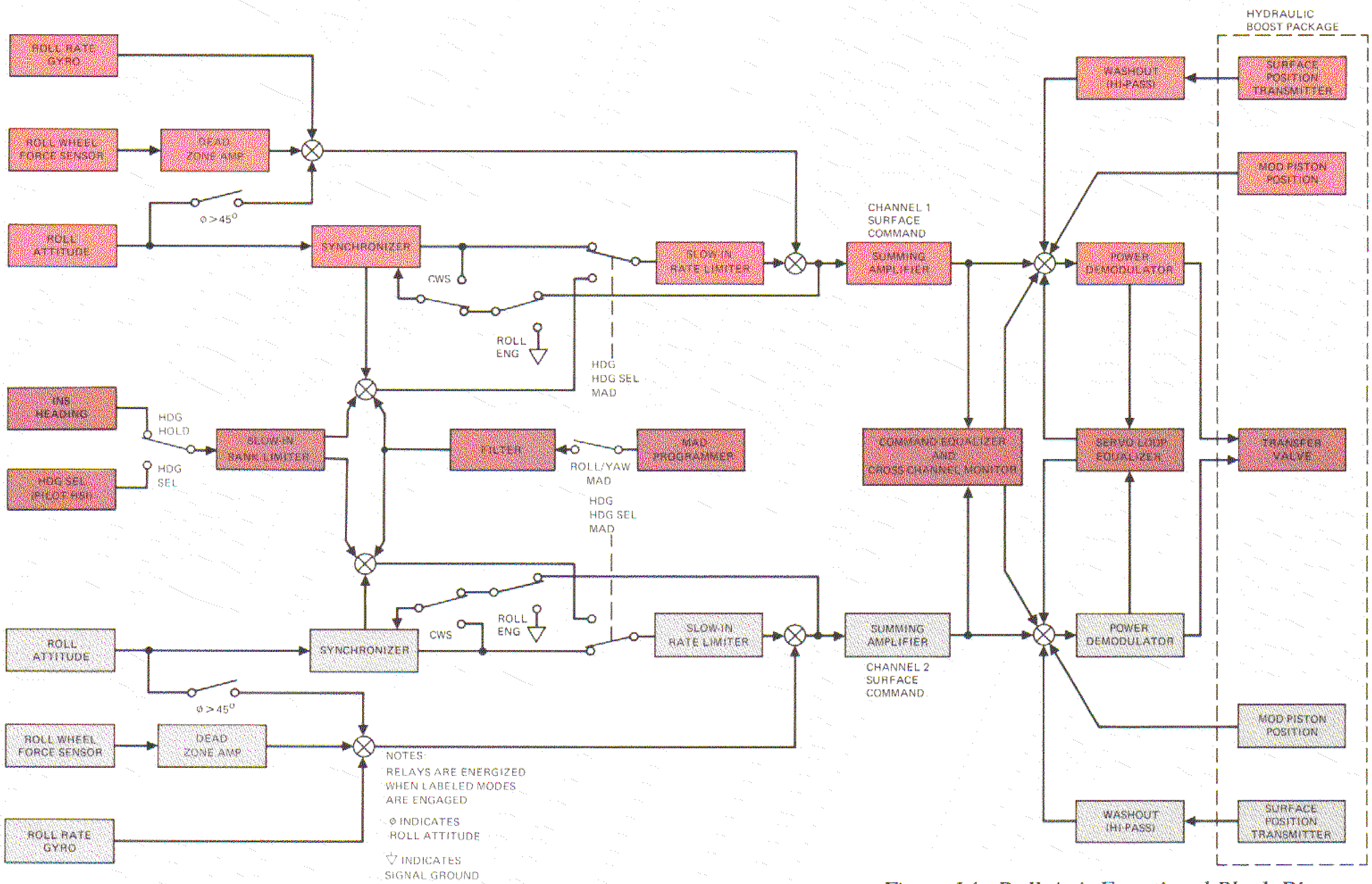



*Figure 14. Roll Axis Functional Block Diagram*

The servo loop is closed by two feedback signals. Surface position feedback is provided by the follow-up signals which are high-passed to wash-out long term surface position errors. Damping is provided by the modulator piston position signal which is proportional to surface rate.

## YAW CONTROL SIGNAL PATHS

**Yaw Damping** Channels 1 and 2 of the yaw axis signal paths are similar in operation; therefore, only Channel 1 will be described. (Refer to Figure 13.) The yaw rate gyro produces a signal proportional to yaw rate. Therefore, a rotational aircraft motion in azimuth will send a signal through the high-pass filter to the yaw Channel 1 summing amplifier and the power demodulator to move the control surface. The control surface booster reacts to this signal, causing rudder displacement to counteract the rotation in yaw. The high-pass (washout) filter is used to remove long term (constant or low frequency) signals experienced in a turn and pass short term (high frequency) signals for yaw damping.

The yaw rate gyro signal would, without the high pass, cause a constant rudder offset during turns.

**Turn Coordination** The P-3C mission requires that sideslip and lateral acceleration resulting from turns be kept at a minimum. The AN/ASW-31 utilizes two signal sources to assist in turn coordination: the lateral accelerometer and the roll bank angle.

The accelerometer signal is applied to the channel summing amplifier through a low-pass filter. This low-pass filter serves to attenuate high frequency accelerometer input and precludes aeroelastic bending inputs from the lateral accelerometer.

The roll bank angle (roll attitude) is also used to assist in turn coordination when rolling into banks. The bank signal is processed through the same path as yaw rate. The resultant signal is proportional to the roll rate which serves to anticipate lateral accelerations that accompany a banking maneuver.

**MAD Calibration Maneuvers** The MAD calibration signal for yaw only is supplied to the ASW-31 yaw amplifier through the channel summing amplifier. The yaw maneuver would normally be accompanied by a natural rolling motion. The rolling motion is counteracted, however, by supplying a crossfeed signal to the roll control amplifier.

**ROLL CONTROL SIGNAL PATHS** The roll control amplifier (Figure 14) contains two 250-degree per second synchronizers to perform pre-engage synchronization. Prior to engage, the synchronizer maintains a null at the output of the channel summing amplifier. After engagement of the roll axis, the synchronizers are supplied drive excitation for automatic heading modes and the Control Wheel Steering mode. The synchronizer memorizes the engage attitude when the roll axis is engaged at bank angles above 2 degrees and below 45 degrees. This will establish a reference for the attitude hold mode.

Each roll synchronizer also contains two function generators that provide a signal proportional to the bank angle ($\phi$). This is a signal equal to 1-cos $\phi$ and is mathematically defined as the versine. This signal is used to enhance altitude hold during banking maneuvers. It is used in the pitch axis and will be discussed in that section.

The roll axis signal paths are dual channel except for heading input and MAD maneuver program input circuits. For the sake of brevity and clarity,

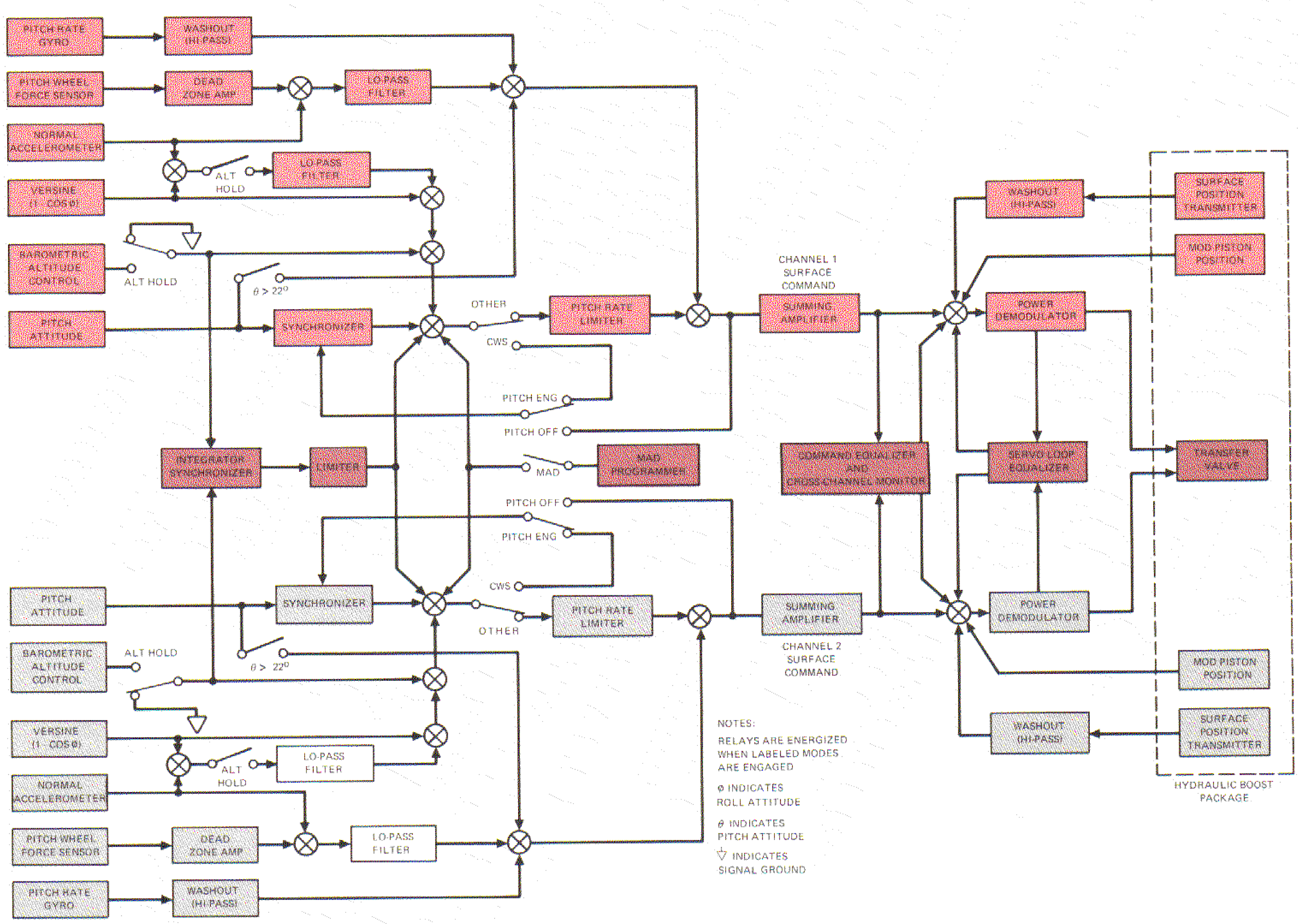



*Figure 15. Pitch Axis Functional Block Diagram*

only Channel 1 of the dual signal paths will be described.

**Heading Hold Mode** The Heading Hold mode is the basic wings-level operational mode in the roll axis. Engagement of the Heading Hold mode will automatically take place if the yaw axis is engaged, the roll axis is engaged, one or both valid INS sources are selected, the aircraft has less than 2 degrees bank, force is removed from the pilot and copilot control wheels, and the yaw MAD programming maneuver has not been selected. The heading source to be used, INS 1 or 2, is selected by the HDG switch on the pilot's HSI control panel. Only one heading source is used at a time.

The heading error signal from the selected inertial navigation system is applied to the slow-in limiter, and then it is summed with the output of the level synchronizer. This heading error, because of the rate of control of the slow-in bank limiter, is gradually applied to the slow-in rate limiter. (The slow-in rate limiter's function is to limit the maximum bank rate to approximately 8 degrees per second.)

The wings-level signal at the output of the synchronizer is zero. A heading error at wings level will then, by itself, be applied to the slow-in rate limiter. The signal will then be applied from the slow-in rate limiter to the summing amplifier, and from there to the roll power demodulator.

The roll power demodulator then commands aileron displacement. The aircraft will respond by changing its bank angle until the wings-level signal cancels out the heading error.

**Heading Select** The Heading Select mode utilizes the position of the heading bug on the pilots' HSI as a reference. This mode has two phases of operation: less than 5 degrees azimuth (low gain) and greater than 5 degrees (high gain). If the selected heading is 5 degrees greater than the present aircraft heading, a relay will energize and increase the gain of the slow-in bank limiter. This will saturate the limiter and command the aircraft to the maximum AFCS heading select bank angle (about 25 degrees). This bank angle will be maintained until the aircraft is within 5 degrees of the selected heading. At this time the low-gain phase is enabled. The low gain promotes smooth transition to the capture point of the new heading. The slow-in rate limiter provides smooth response to large heading errors.

A force of 2 lb on the control wheel will override and disconnect the heading select mode and deenergize the solenoid holding the HDG SEL switch on the AFCS control panel.

**Roll Attitude Hold** Roll Attitude Hold, together with Heading Hold (as discussed above) constitute the basic roll operating mode for the AN/ASW-31. The roll attitude is engaged whenever the roll axis is engaged, when no other roll mode is selected, when roll control wheel steering is not in use, when a valid attitude signal is present, and whenever roll attitude is between 2 degrees and 45 degrees. Upon engagement the synchronizer establishes the reference for the Roll Attitude Hold mode.

**Roll Damping and the Roll Rate Gyro** A roll rate gyro is included in the AN/ASW-31 AFCS to provide a well-damped roll axis which enhances the Roll Attitude and Heading modes and the operation of the Control Wheel Steering mode. The use of a roll rate gyro allows use of a high, constant system gain throughout the flight envelope without requiring scheduled gains.

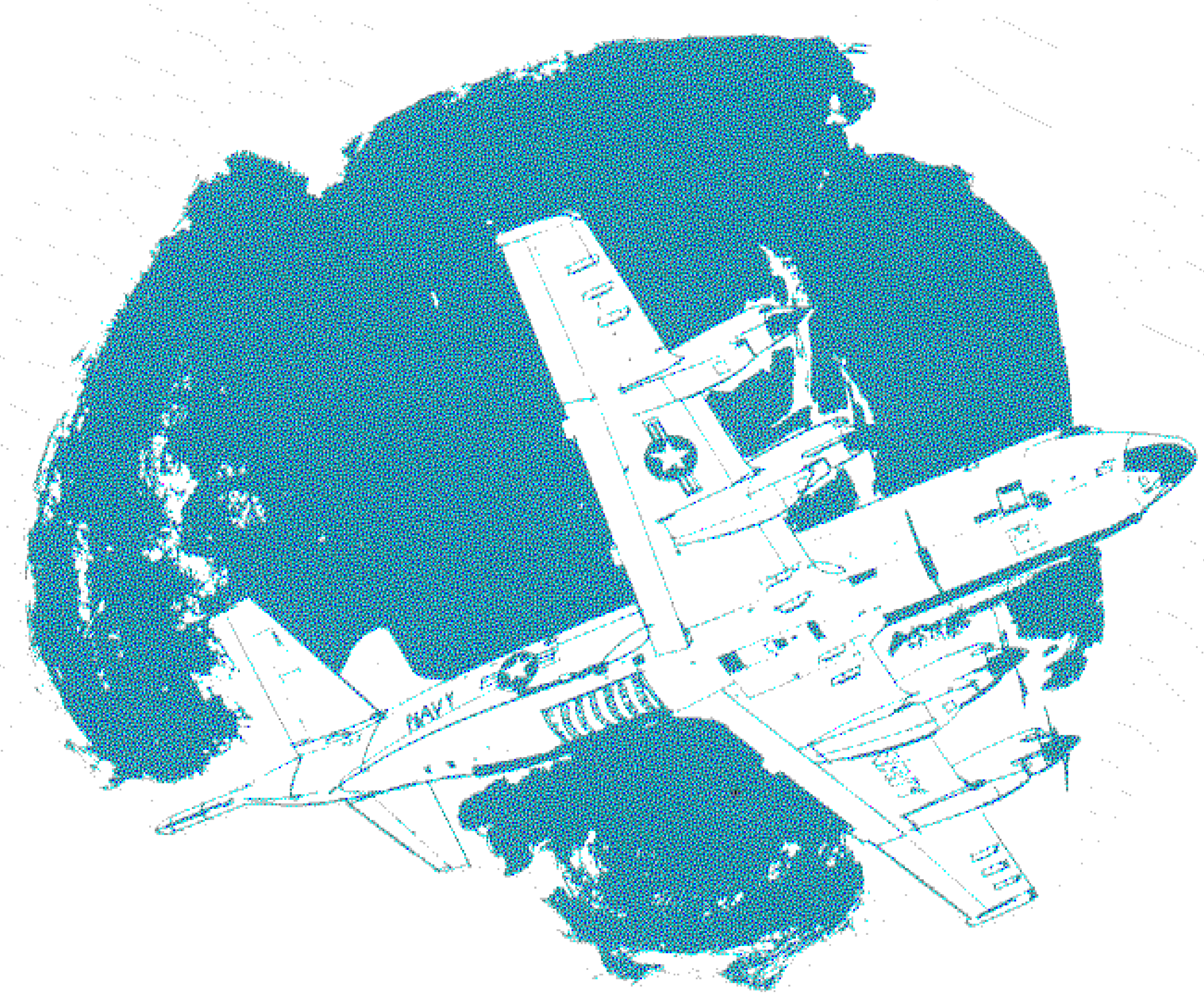
**Roll Attitude Limits and Return Command** If the roll axis is engaged, or if roll control wheel force is released at a roll attitude below 2 degrees, a wings-level signal from the synchronizer will command the autopilot to return the aircraft to wings level. If the yaw axis is also engaged the system will respond to heading error signals as previously described.

If the roll axis is engaged, or roll control wheel force released at a roll attitude above 45 degrees, a return to 45-degree command signal is applied to the summing amplifier. The synchronizer remains in an active state and the controls return the aircraft to a bank angle of 45 degrees. At this time the synchronizer hold mode is established and a roll attitude of 45 degrees will be maintained. A force of about 8 pounds on the control wheel will overcome the 45-degree return signal and allow control wheel steering maneuvering to any desired bank angle. The return to 45 degrees command will increase gradually as the bank angle increases above 45 degrees.

**Pilot and Copilot Control Wheel Steering** The Control Wheel Steering mode is available whenever the roll axis is engaged, even if roll attitude reference is invalid. The control wheel force sensor for the roll axis in each wheel is a "dual" sensor. The control Wheel Steering mode is initiated when a force greater than 2 lb is applied to the control wheel. The signal is applied to a dead zone amplifier stage. The output is summed with roll rate and applied to the summing amplifier. The signal is then applied to the booster through the power demodulator. Roll rate and roll wheel force signals are summed to provide a constant roll rate per pound of wheel force (approximately 0.8 degree per second per pound of force). While in the Control Wheel Steering mode, the roll synchronizer is driven to follow the aircraft bank position. This allows a reference to be established when the control wheel is released.

**MAD Calibration Maneuvers** Engagement of roll MAD maneuvers causes the aircraft to roll ±10 degrees with a period of 8 seconds.

The MAD command signals are routed through the roll channel slow-in rate limiters. The roll rate limit is increased during roll MAD maneuver because maximum roll servo authority is needed.



**PITCH CONTROL SIGNAL PATHS** The pitch control amplifier of the AN/ASW-31 AFCS has much the same hardware makeup as that of the roll axis. It has three synchronizers; two 30-degree per second synchronizers for pre-engage synchronization, and one 5-degree per second synchronizer for use as an altitude error integrator. To assist in following the pitch signal flow description, refer to Figure 15. Prior to pitch axis engage, the synchronizer is configured to reduce the summing amplifier output signal to null. The basic mode of operation for the pitch axis is Altitude Hold.

**Altitude Hold Mode** The Altitude Hold mode will engage when the following initial conditions are satisfied:

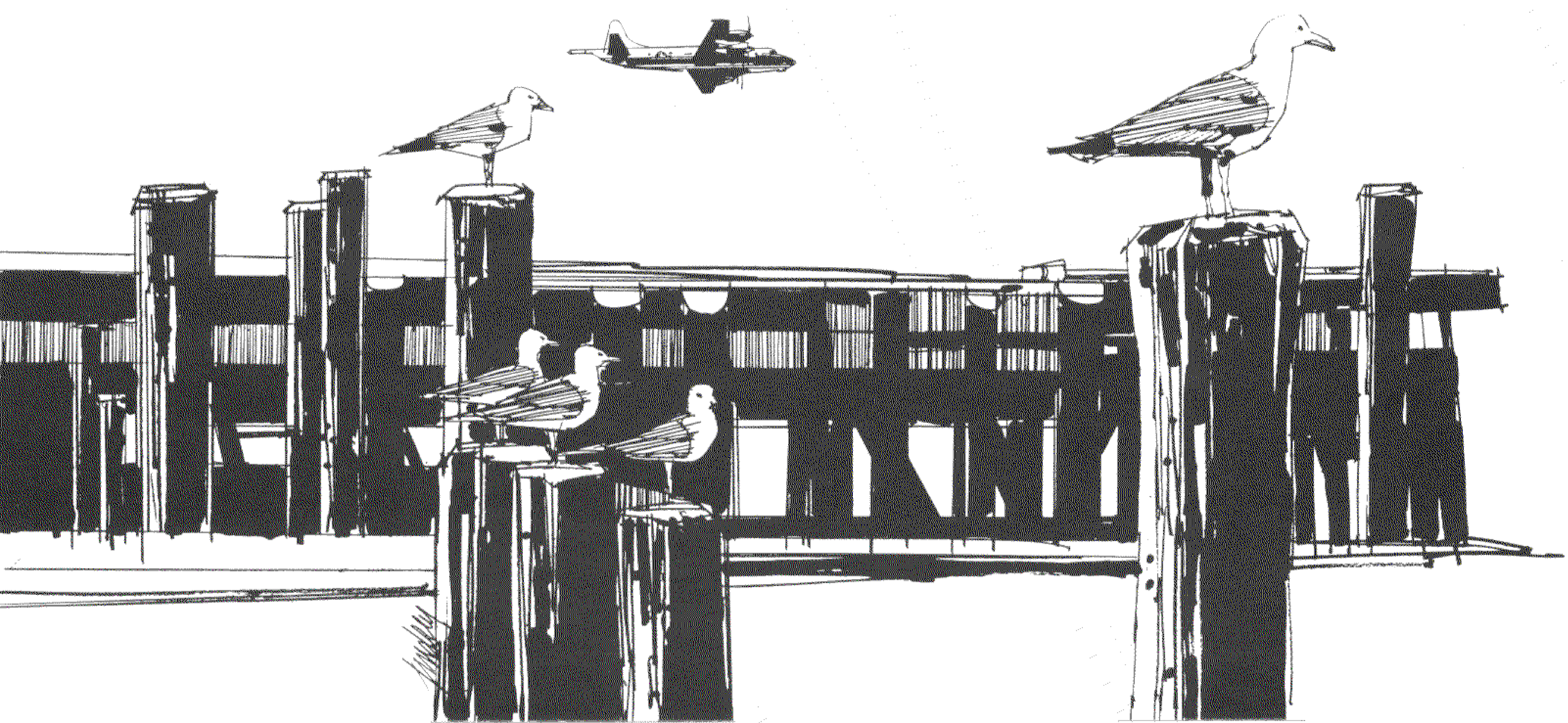
- Pitch axis is engaged.
- Valid single or dual INS signals are selected.
- ALT HOLD/OFF switch on the AFCS control panel is in ALT HOLD position.
- Neither AFCS disconnect switch is depressed to the first detent.
- Fore-aft forces on the control wheel are less than 2 lb.
- Rate of climb is below 300 ft per minute.

When engaging the Altitude Hold mode, the following changes occur:

- All pitch control wheel signals are switched out.
- The integrator (synchronizer) functions as an error integrator.
- The barometric altitude control switches to its reference hold mode and begins supplying altitude error signals to the pitch amplifier if the altitude deviates from the reference altitude.
- The pitch attitude synchronizers are placed in a hold mode.
- Normal accelerometer and versine signal gains are increased to improve altitude hold stability and accuracy during gusts and turns.
- The altitude deviation monitor is activated.

The control wheel signals are removed in the Altitude Hold mode to ensure altitude hold priority over control wheel steering, thereby precluding the possibility of loss of the Altitude Hold mode due to inadvertent application of pitch forces to the control wheel.

**ALTITUDE INTEGRATOR** The altitude integrator (synchronizer) is activated to assist in altitude hold



during bank maneuvers and flight configuration changes. For example, when airspeed is decreased, the angle of attack and pitch attitude must increase to maintain the same lift at the same altitude. Without the integrator, the change in pitch attitude would result in a nose-down command since the attitude synchronizers are in a hold configuration. This would result in altitude loss, so the altitude error is used to command the integrator. The altitude error is applied to the altitude integrator. The integrator output is applied to both channels. The integrator will drive continuously until the aircraft has recovered its original altitude (altitude error zero) and, therefore, the integrator output electronically equals the change in pitch attitude. The integrator will then remain inactive unless an altitude error reappears at the input.

The integrator synchronizer stops completely in Attitude Hold mode. The integrator will reset to zero output when the pitch axis is in Pre-engage mode or Control Wheel Steering mode.

**ACCELEROMETER IN ALTITUDE HOLD** The accelerometer is used to improve response of the pitch attitude hold sub-loop of altitude hold, and also to damp the low frequency pitch oscillation (phugoid) of the aircraft.

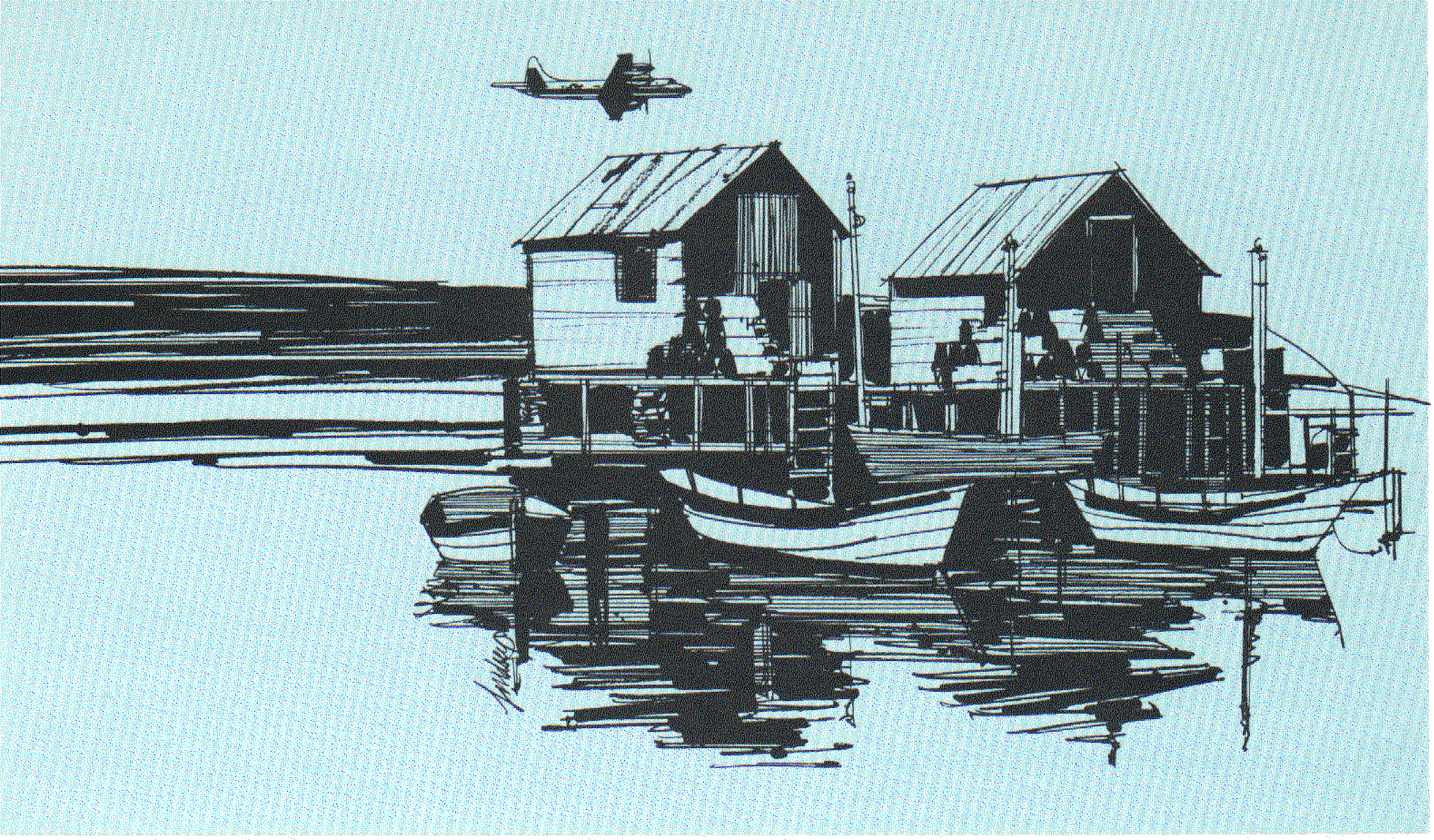
**BAROMETRIC ALTITUDE CONTROL** Each pitch channel has a barometric altitude control unit. Each unit houses two circuit cards, one synchronizer, four relays, power supply components, and a barometric altitude transducer. The heart of the control is the barometric altitude transducer. The transducer is powered by external positive and negative 15 v dc supplies housed in the control unit and has two altitude signal outputs. A high range output provides a linear profile between -1000 ft and +40,000 ft. The voltage is +10 v dc at 40,000 ft and 0 v dc at -1000 ft, or about a quarter of a millivolt per foot. The low range output is linear between -1000 ft and +2000 ft, and is 10 v dc at 2000 ft and 0 v dc at -1000 ft. The low range signal (3.3 millivolts per foot) is used below approximately 1700 ft.

Before altitude hold engage occurs, the synchronizer is controlled to continuously cancel out the altitude signal. After Altitude Hold mode

engagement, the synchronizer maintains the reference position at which altitude hold was engaged.

Deviation from the reference altitude causes a change in transducer output. The resulting signal is directed to the pitch control amplifier and the altitude deviation monitor in the barometric control unit. If the deviation exceeds 60 feet, the monitor will announce altitude system malfunction by illuminating the altitude status lamp and the flashing red warning lights. The lights will extinguish automatically when altitude error is reduced to less than 60 feet.

**VERSINE ($1-\cos\phi$) LIFT COMPENSATION IN ALTITUDE HOLD** An aircraft flying at a constant altitude with the wings level must develop sufficient lift



force to exactly balance its weight. When the aircraft enters a banked turn, the lift force is tilted, making it necessary to increase the total lift so that the vertical component equals the lift developed when the wings are level and thus balance the weight. Since the lift force acts perpendicular, or "normal," to the wing plane, the required increase in lift force acts on the aircraft mass to produce a normal acceleration that is sensed by the ASW-31 normal accelerometers. The resultant signal from the accelerometer would cause a pitch down command and loss of altitude if not counteracted.

The increase in lift required for a constant altitude turn equals the total lift developed in the turn minus the vertical component of that lift. (The vertical component is equal to lift required when the wings are level as discussed above.) The vertical component is, in geometric terms, equal to the total lift times the cosine of the bank angle. Thus, the increase in lift required for the turn is equal to the total lift minus the total lift times the cosine of the bank angle or, total lift times ($1 - \cos \phi$). This latter expression is defined mathematically as the versine of the bank angle.

By electronically deriving the versine function and selecting proper gains, it can be subtracted from the normal accelerometer output to counteract the pitch down signal sensed in a turn, thus maintaining the desired altitude. This is accomplished by the pitch amplifier.

Lift compensation is taken from specially designed function generators that produce a signal proportional to one minus the cosine of the roll bank angle ($1 - \cos \phi$). The signal is a null for 0 degrees bank and increases in proportion to $1 - \cos \phi$ up to 75 degrees of bank.

**Attitude Hold Mode** The Attitude Hold mode is enabled upon engagement of the pitch axis when the pitch attitude is less than 22 degrees. At that time the synchronizer is placed in a hold mode, thus establishing the attitude reference. The pitch axis will maintain this attitude reference unless changed by applying pitch wheel force. The attitude error traverses from the synchronizer output, to the limiter, through the summing amplifier, and to the power demodulator. The limiter performs primarily as an acceleration command limit, since the rate gyro signal is high-passed for damping as it is in the yaw amplifier. (The reason pitch rate is not a factor in a turn is that pitch rate is applied through the high-pass filter as discussed previously.)

While the Pitch Attitude Hold mode is not expected or required to maintain constant altitude, some versine compensation is supplied as a pilot assist to avoid excessive altitude loss when using Pitch Attitude hold during roll maneuvering.

**Pitch Attitude Limit and Return Command** If the pitch attitude exceeds 22 degrees of pitch upon engaging the pitch axis, the pitch attitude signal from the INS will be applied directly to the summing amplifier. This signal will command a brisk return to 22 degrees. Control wheel force can override the return signal, however, and command increased pitch attitudes.

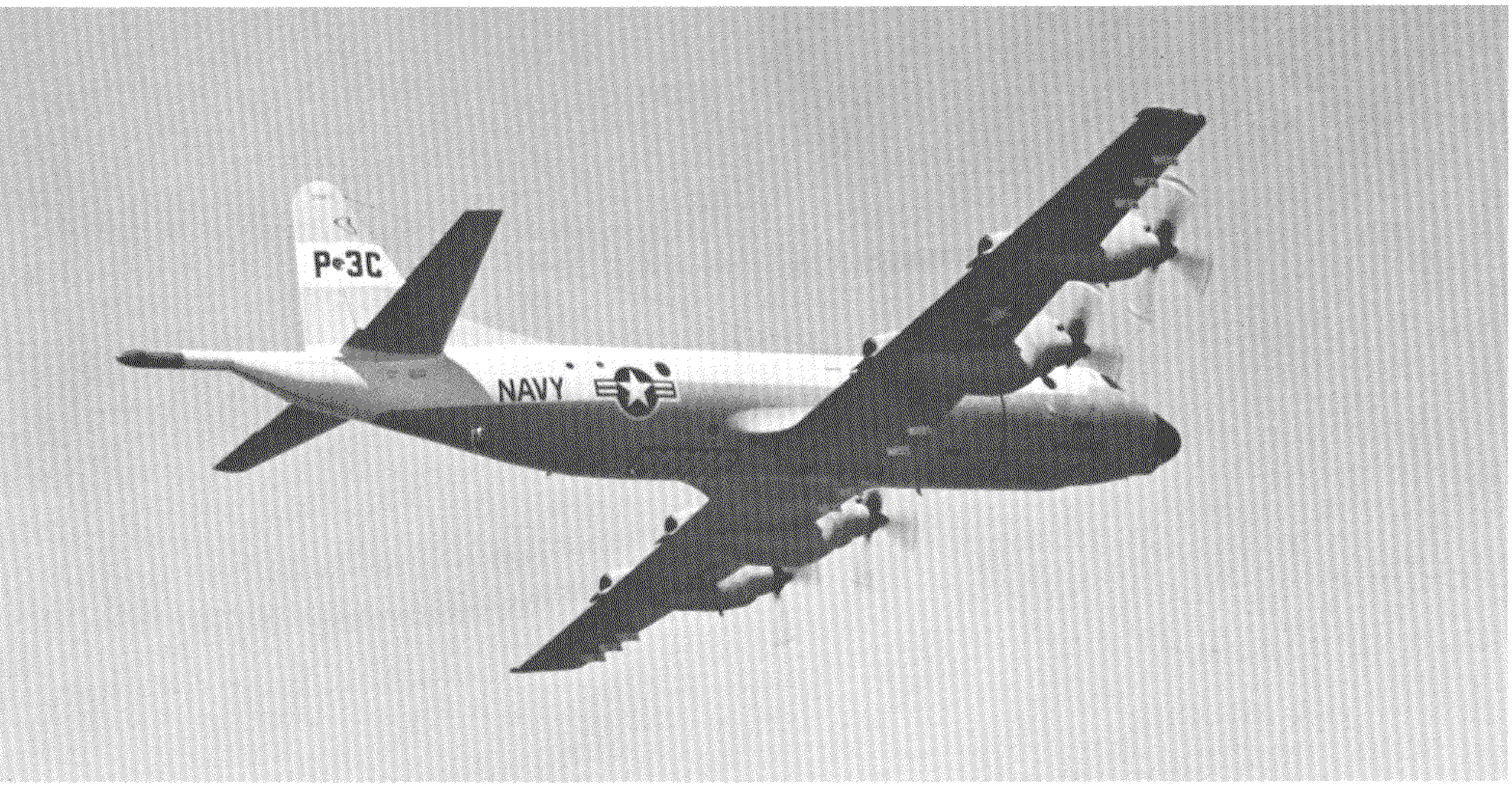
**Control Wheel Steering** When a fore or aft force is applied to the control wheel, activity similar to that in roll axis takes place. The control wheel steering (CWS) signal is applied to the dead zone amplifier. A signal equal to a force greater than 2 lb is required to produce an output from this amplifier.

The output is algebraically summed with pitch rate and then routed to the summing amplifier and the power demodulator. The control surface then responds to control wheel forces. While in the CWS mode, the synchronizer keeps the attitude signal at zero. In CWS, the only signals being sent to the power demodulator are the high-passed pitch rate signals for pitch damping, the normal accelerometer signals and the control wheel steering force sensor inputs. Wheel force and accelerometer inputs are used for constant force-per-g mechanization of control wheel steering.

**MAD Calibration Maneuvers** The pitch axis MAD calibration signal is similar to that of the yaw and roll axes. There is no mode-engage logic in the pitch computer since the signal is switched on and off in the MAD panel. Engagement of the pitch MAD calibration signal causes the aircraft to pitch ±3 degrees in a cyclical manner. The period is approximately 5 seconds.

**MONITOR CONTROL AMPLIFIER** The monitor control amplifier contains four sets of power supplies, an automatic pitch trim control, a set of cross-channel monitors (one monitor for each axis), two sets (three monitors per set) of servo (in-line) monitors for dual/single channel operation, pitch and roll attitude comparison monitors, and six rate gyro rotor speed monitors.

**Channel Comparison Monitors** The purpose of the channel comparison monitor is to protect against sensor failure and malfunctions in the AFCS electronics (with the exception of the MAD maneuver, heading hold, and heading select which are not dual) upstream of the servo command amplifiers. One channel comparison monitor is provided in each axis. The basic design and operation of this monitor is identical in all three axes, although the comparison levels (trip levels) vary between the axes. Each channel comparison monitor operates on the difference and the rate-of-difference between the servo command signals in the redundant channels. This design feature allows the trip levels to be set at a relatively high value, ensuring a low probability for nuisance disconnects while still providing adequate protection against failures which cause aircraft transients (such as a hardover). The selection of monitor rate gain, trip levels, and time delay ensures fail-passive performance




coupled with axis disconnect without any objectionable aircraft transients.

The trip levels, time delay, and rate gain were selected on the basis of essentially three different criteria:

1. Channel mismatch due to expected tolerance buildup should not cause axis disconnect; i.e., the probability for nuisance disconnects should be minimized.

2. All possible failures or combinations of failures should be detected.

3. Any failure should result in fail-passive performance with axis disconnect without any discernible aircraft transient.

The rate gain and time delay are the same for all three axes; namely 3 and 0.24 seconds, respectively, while a different trip level is employed in each axis.

During straight and level flight, or mild maneuvers in calm air; while sensor error signals are at null or low level values, some sensor failures (open circuit or reduced output level) will be compensated for by cross-channel equalization. This precludes detection and disconnect by cross-channel monitoring, thereby permitting continued dual-channel operation with only a slight performance degradation. However, at higher maneuver rates such failures will cause channel signal differences that exceed the equalizer authority and result in failure detection by the cross-channel monitor. Execution of the BITE test in NORM (dual channel) and in each of the two channels will detect this type of failure. Fail-passive disconnect is ensured in the cases where an open sensor causes an autopilot mode instability.

**Servo Monitor** Two servo (in-line) monitors are provided for each axis; one for dual-channel operation, the other for single-channel operation. The servo monitor senses electrical, hydraulic, and mechanical failures in the respective surface control actuators and electrical malfunction between the AFCS and the control actuators and the accessories. Each servo monitor consists of a modulator piston or hydraulic monitor and a coil current or electronic monitor.

The function of the hydraulic monitor is to detect the following failures:

- Servo valve spool jam and hardover.
- Modulator piston jam or hardover.
- Loss of hydraulic power.
- Loss of modulating piston LVDT.

The primary function of the electronic monitor is to detect failures in the following signals and components:

- Servo command amplifiers and demodulators.
- Power servo follow-ups.
- Transfer-valve coils.
- Signal paths between the demodulators and transfer valve.

The electronic monitor serves a secondary function as backup to the above described cross-channel monitor by detecting failures in command input signals to the servo.

**HYDRAULIC MONITOR** The hydraulic monitor verifies that the modulating piston is responding properly to the transfer valve inputs. Modulating piston position is a function of differential coil current. Thus, a measure of modulating piston performance can be obtained by comparing total coil current with the modulating piston position LVDT signal. Any significant difference between these two signals, is therefore, an indication of a modulating piston or transfer valve malfunction.

**ELECTRONIC MONITOR** The electronic monitor compares the current in the two transfer valve coils. Ideally, these currents are of opposite sign but of equal magnitude. Any malfunction which causes a difference in the coil currents results in a voltage proportional to this difference to be generated across a resistor connected between the common point of the transfer valve coils and ground. This signal is fed to a level detector and time delay circuit.

**Cross-Channel Monitor Backup** Even though the cross-channel comparison monitor is in itself essentially fail-safe, there are a few very rare failure modes in the monitor that are not self-betraying. In order to meet and maintain the fail-passive requirement in this case, the limit on the command equalizer is set just above that of the cross-channel comparator. Thus, in normal operation, the cross-channel monitor trips before the servo-loop equalizer limit level is reached. The purpose of the servo-loop equalizer is to prevent command signal unbalance from tripping the electronic valve/monitor. The electronics valve monitor is approximately 10 times more sensitive than the cross-channel monitor in terms of equivalent signal. Assuming that the cross-channel monitor fails to function, the equalizer limit will be reached and the very slight increase in cross-channel mismatch will cause the electronic valve monitor to trip. This failure mode is self-indicating during ground test by illumination of the SERVO lamp when the RATE MON self-test shows two good single channels but a fault in dual operation.

**Attitude Signal Comparison Monitors** Two attitude signal comparison monitors are provided in the AN/ASW-31 AFCS; one for pitch axis and one for roll axis. These monitors compare the difference between Channel 1 and Channel 2 attitude data when the applicable axis (roll or pitch) is operating in dual (NORM) mode. If a mis-track of 3 degrees or greater occurs, the monitor will disconnect the Attitude Hold mode in the affected axis. If the roll axis is affected, the heading mode will also be disconnected; if the pitch axis is affected while the Altitude Hold mode is engaged, this mode will also be disconnected. The affected axis will revert to its "control augmentation" configuration as long as the mis-track is present. Attitude reference failure disclosed by the INS validity signals will also trip the attitude monitors. Functional operation of the monitors is determined by AFCS INS input selection and pitch and roll channel selection as shown in Table 5.

Attitude reference malfunction sensed by the monitors is indicated by illumination of the ATTD status lights. If pitch attitude is affected while altitude hold is engaged, the ALT status lights and the red flashing warning lights will also illuminate.

Table 5. *Attitude Comparison Monitor Functional Operation*

| NO. | INS | AFCS CHANNEL SELECTED | | MONITOR FUNCTION |
|---|---|---|---|---|
| | SELECT | PITCH | ROLL | |
| 1 | Norm (INS 1 & 2) | Norm (Dual) | Norm (Dual) | Pitch and roll monitors compare tracking between INS 1 and INS 2 respective attitude reference input signals and monitor INS 1 and 2 validity signals. |
| 2 | INS 1 or INS 2 | Norm | Norm | The comparison portion of each monitor is still active, but only one INS is supplying both AFCS channels and each monitor. Thus, the comparison function will only sense input errors caused by short or open circuits between the switching in the NAV junction box and the AFCS. Each monitor checks validity signal from selected INS. |
| 3 | Norm | 1 or 2 | Norm | Pitch comparator inoperative; pitch monitor checks validity from INS 1 or 2 depending on pitch channel selection. Roll monitor functions as in 1 above. |
| 4 | Norm | Norm | 1 or 2 | Pitch monitor functions as in 1 above. Roll comparator inoperative; roll monitor checks validity from INS 1 or 2 depending on roll channel selection. |
| 5 | Norm | 1 or 2 | 1 or 2 | Pitch and roll comparators inoperative; pitch and roll monitors check validity from INS 1 or 2 depending on respective axis channel selection. |
| 6 | INS 1 or INS 2 | 1 or 2 | 1 or 2 | Pitch and roll comparators inoperative; pitch and roll monitors check validity from selected INS. |

Normal operation is restored upon disappearance of the discrepancy or by selection of valid attitude reference input.

The trip level of 3 degrees was selected to ensure detection of attitude input discrepancies by the attitude monitors, rather than the AFCS cross-channel monitors. However, due to this relatively low setting, there may be cases where tolerance buildup in the roll attitude signals at high bank angles may momentarily trip the roll attitude monitor without an actual failure present.

**Trim Monitor** The trim monitor performs two basic functions: verification of the inputs to the trim drive, and verification of the trim drive itself. Verification of the inputs to the trim drive is accomplished by comparing the two pitch booster load sensor signals. Only one of these signals is used for the input to the trim drive. The two signals are applied to the input of a differential amplifier. The output is zero, provided there is no mis-track between the load sensor signals. If there is a mis-track exceeding 300 millivolts and 100 milliseconds, the monitor will energize the trim disconnect relay. This relay applies +12 v dc to a time delay circuit. If the mis-track persists for a period of time greater than 7 seconds, the trim disconnect relay is latched, disabling the autotrim function and illuminating the AUTOTRIM status lights.

The trim drive is an inverting power amplifier.

Therefore, the operation of the trim drive may be monitored by amplifying the hydraulic load sensor input to the trim drive and summing it with the output of the trim drive. This is done at the input of a level detector. If the trim drive fails, the trim disconnect relay will be energized. The 7-second latch will operate as described previously, disabling the autotrim function and illuminating the AUTO-TRIM status lights.

**Rate Gyro Speed Monitor** Each rate gyro spin motor generates an ac signal whose amplitude and frequency are proportional to the speed of the motor. This signal is applied to a level detector to determine if the gyro is up to normal speed. When the rate gyro motor comes up to speed, a relay is energized signaling proper operation of the gyro motor.

The speed monitors prevent axis engagement in the single-channel configuration until the appropriate rate gyro is up to speed, or dual axis engagement until at least one rate gyro is up to speed.

## AN/ASW-31 AFCS IMPROVEMENT PROGRAM

Since the introduction of the AN/ASW-31 AFCS in mid-1970, the Navy, Lear Siegler, Inc. (Astronics Division), and Lockheed have been engaged in a cooperative effort to provide effective system improvements. At present, this effort has resulted in initiation of Engineering Change Proposal (ECP) action for incorporation of improved, solid-state synchronizers and installation and testing of a group of proposed improvement changes (identified informally as "Modification Block III") in P-3C Serial No. 158204.

**IMPROVED SYNCHRONIZERS** An ECP has been submitted to the Navy recommending production installation of all-electronic, solid-state synchronizers in place of electro-mechanical synchronizers in AN/ASW-31 roll and pitch computers, and the barometric altitude controls. The new synchronizer units are direct replacements and would have full forward and backward interchangeability. The test procedures in the NAVAIR manuals would not be affected, and the computers would not be reidentified in any way. A similar ECP has already been approved for incorporation in the AN/ASW-26 and AN/ASW-30 systems in A-7 Series aircraft.

**AN/ASW-31 AFCS MODIFICATION BLOCK III** A group of ten improvement changes are under consideration following joint Navy (NASC), Lear Siegler, Inc. (LSI), and Lockheed studies of actual AN/ASW-31 AFCS service performance experience. Studies included review of fleet maintenance actions, Unsatisfactory Reports (UR's), and personnel interviews with various P-3C squadron personnel.

NASC established a test program which was performed by NATC and ASAT (NAS Patuxent River) utilizing P-3C Serial No. 158204 with its AN/ASW-31 AFCS modified to incorporate Modification Block III. The changes comprising this modification are summarized in Table 6.

Results of the evaluation by ASAT/NATC were favorable and it was recommended by ASAT/NATC that production be incorporated after a simplified revision of the attitude monitor has been made. ▲▲

*Table 6. Summary of AN/ASW-31 AFCS Modification Block III Improvement*

| MODIFICATION TITLE | IMPROVEMENT PROVIDED |
|---|---|
| In-Line Servo Monitor and Self-Test Modified | Prevent random AFCS servo disconnects during Control Wheel Steering operation. |
| Servo Light Latching/Modified | Insures consistent indication of AFCS Servo Disconnect. |
| Attitude Signal Monitor - Single Source Added - Comparator Modified | Prevent hardover due to attitude reference failures when AFCS operating from single source; prevent nuisance attitude failure indication at high bank angles. |
| AFCS Engage/MAD Logic Isolation Added | Provides consistent AFCS emergency disconnect function in all modes. |
| Servo Command Limiting Added | Prevent nuisance disconnects during ground bite operation. |
| Altitude Control Engage Logic and Synchronizing Modified | Optimize synchronizer loop damping; reduce altitude hold disengage transients; prevent nuisance pitch disconnects when attitude disconnect switch is pulsed; improve engagement logic. |
| MAD Programming Axis Select Logic Modified | Prevents occasional transients when MAD programmer axis select switch is moved between positions. |
| Roll MAD Oscillator Modified | Ensures compatible functioning of AN/ASQ-81 MAD and AN/ASA-65 during roll MAD calibration. |
| Control Wheel Wiring Harness Modified | Protects sensors from static bias loading caused by wire harness. |
| Rate Gyros Relocated | Increases life of booster modulator piston seal rings. |

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