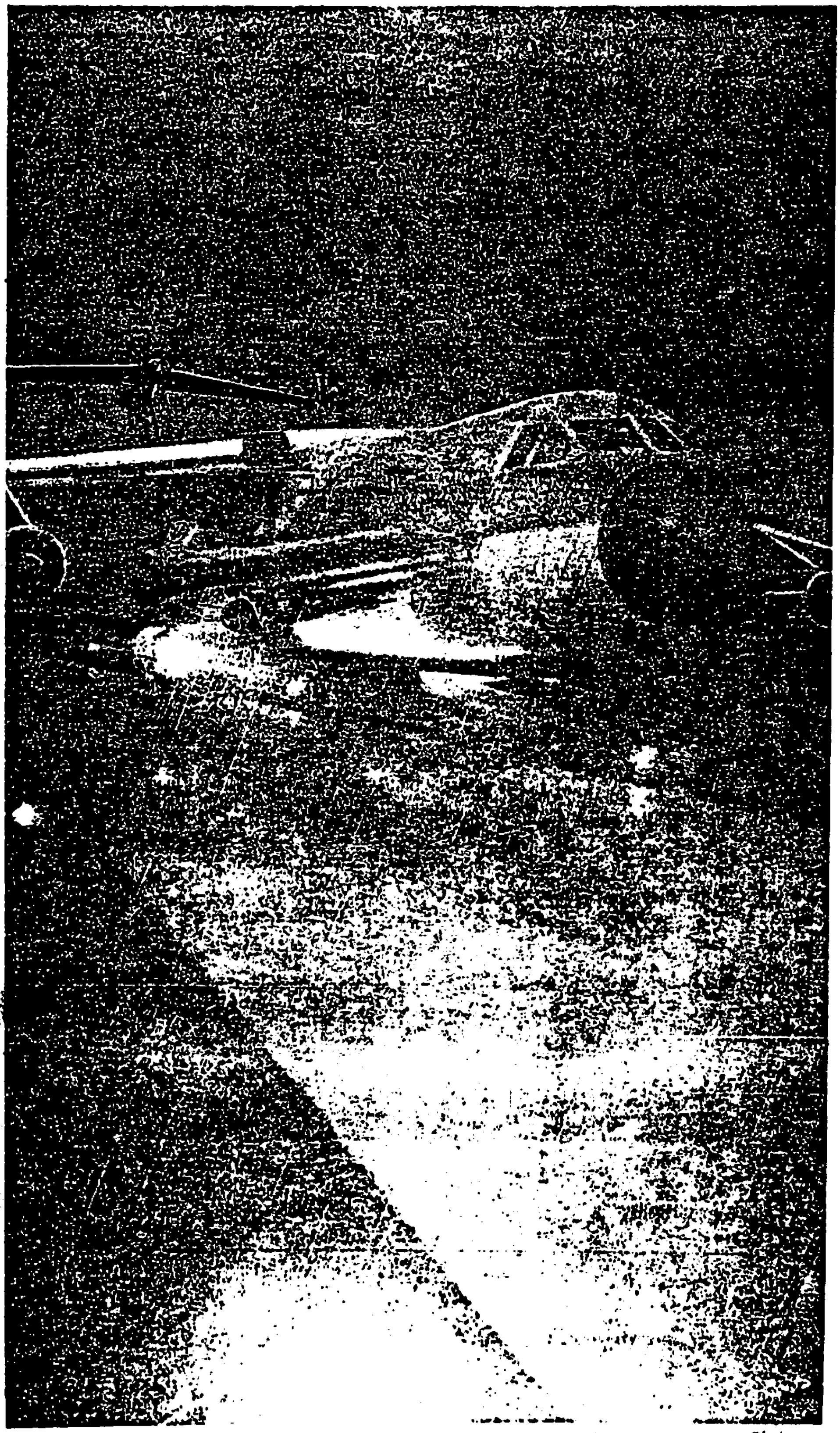
STARLIFTER TRAINING MANUAL • VOLUME IX



ALL WEATHER LANDING SYSTEM

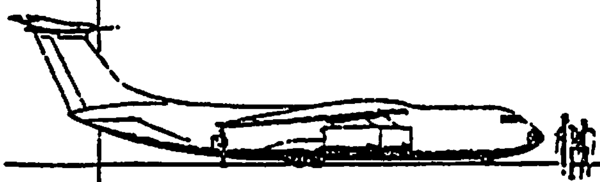
Customer Training Department • Lockheed - Georgia Company • Marietta, Georgia 3006

CONTENTS

CONTENTS (Continued)

ILLUSTRATIONS

ILLUSTRATIONS (Continued)



INTRODUCTION

The All Weather Landing System (AWLS)* installed on the StarLifter provides the capability of safely landing the aircraft under Category II (CAT II) weather conditions (100-foot altitude and 1200-foot visual range). Either automatic or manual aircraft control can be used during AWLS landing approaches down to a minimum decision altitude of 100 feet. As an added capability, if adequate visual contact with the runway is established at a 100-foot altitude, the AWLS controlled approach can be continued to runway touchdown. Control of the flare maneuver and automatic retarding of the throttles are provided. The pilot, however, must decrab the aircraft as necessary and must control the roll-out by using visual references.

Progress display lights, as shown in Figure 1-1, illuminate as certain milestones are reached to inform the pilots of progress of the AWLS approach. The lights are labeled LOC, G/S, APPR ARM, LAND ARM, and FLARE. The localizer (LOC) light illuminates to indicate localizer beam engagement. The Glide Slope (G/S) light illuminates to indicate G/S beam engagement. An automatic preland test is then initiated. The APPR ARM (AA) light illuminates at test completion. The LAND ARM light illuminates to indicate minimum descision altitude of 100 feet. The FLARE light illuminates to indicate flare engagement.

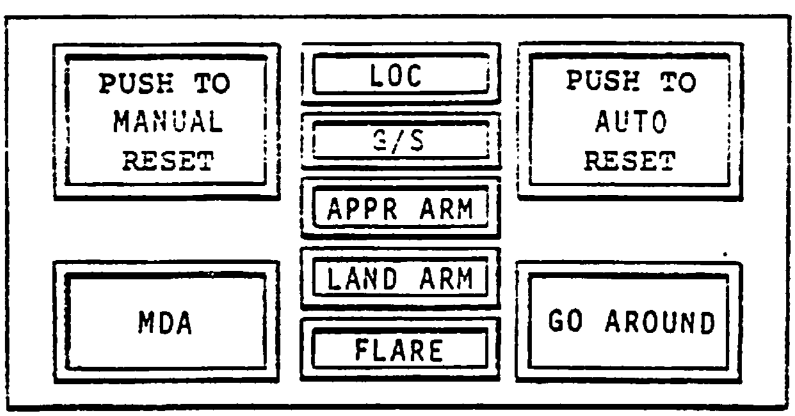


FIGURE 1-1. AWLS FLIGHT PROGRESS DISPLAY & AWLS CAUTION PANEL

*A complete list of abbreviations and symbols is shown in the Appendix.

In the event an approach abort is necessary, optimum pitch steering commands and wings level commands are displayed to the pilots for performing the go-around maneuver through manual control of the aircraft. A GO-AROUND mode light on each panel illuminates when go-around is selected.

Automatic failure monitoring of the AWLS is provided during the final phases of the AWLS approach (beginning at APPR ARM progress). If an AWLS failure should occur that affects the approach, appropriate failure indication lights illuminate to alert the pilots of the malfunction. The fault is announced by the illumination of either the MANUAL or the AUTO (or both) caution lights on the flight progress display panels as shown in Figure 1-1. The pilots' attention should then be directed to the AWLS fault identification panel, Figure 1-2, where the appropriate fault light or lights further identify the faulted area.

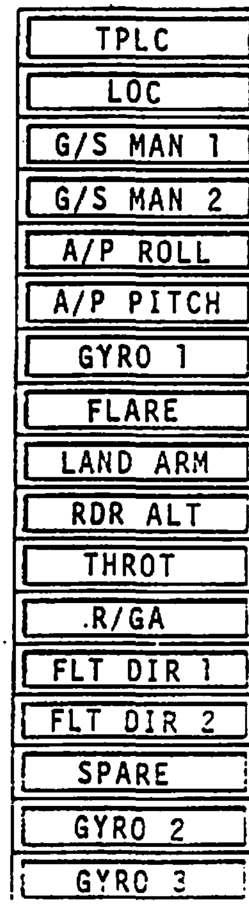


FIGURE 1-2. FAULT IDENTIFICATION

The AWLS also contains a manually operated, "enroute" test capability. It can be run during cruise or at any time prior to the beginning of an AWLS approach. If a failure is detected during the test, appropriate failure lights, as shown in Figure 1-3, illuminate to provide the pilot with enough information to make the descision to continue to his destination or to proceed to his alternate.

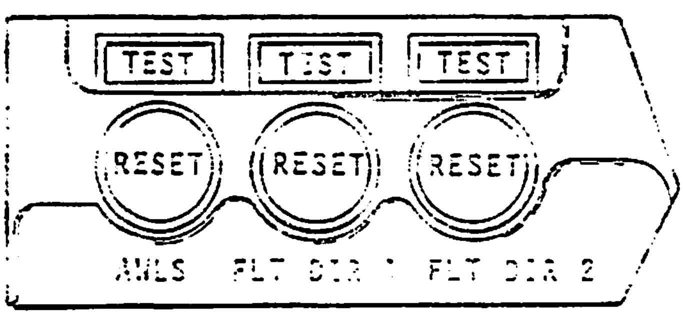


FIGURE 1-3. AWLS AND FLIGHT DIRECTOR TEST PANEL

Even though the Flight Director Systems (FDS's) are tested during the AWLS test, they are provided with separate test capabilities. An FDS test can be run at any time. The enroute test and the FDS test can also be used as preflight items and as an aid when maintenance is being performed. A second test, the "preland" test, is automatically run during the AWLS approach just after G/S interception. This test is a confidence one which checks the failure monitors to assure that failures would not go undetected. The flare computer is also functionally tested at this time.

If a failure is announced during test, or occurs during the AWLS approach that would prohibit or cause termination of the CAT II approach, a normal CAT I Instrument Landing System (ILS) approach (200-foot altitude and 2400-foot visual range) can be executed. The CAT II approach cannot be made if any of the following is under failure conditions:

- o Low-Range Radar Altimeter
- o Either Glide Slope Receiver
- o Either Localizer Receiver
- o Two of three Attitude Gyros

Automatic control of the approach is provided through operation of the Autopilot (A/P) system. Manual control of the approach is provided through guidance displays on the FDS's Attitude Director Indicators (ADI's).* The A/P, or either of FDS, can be used to make the AWLS approach. Normally, the A/P is used and the FDS's serve as monitoring and backup systems.

The FLARE computer provides the flare maneuver commands to the A/P and FDS. It also provides LAND ARM (minimum decision altitude of 100 feet), FLARE engage (45-foot altitude) and throttle retard (30-foot altitude) signals. The Automatic Throttle System (ATS) is used to automatically maintain the desired Indicated Air Speed (IAS) during the approach and to retard the throttles during flare. However, an AWLS approach can be accomplished with the pilot controlling the throttles manually. The ATS can also be used during normal cruise for maintaining a desired IAS.

The Rotation/Go-Around (R/GA) computer provides go-around pitch commands. During go-around, it provides the maximum safe pitch steering commands (angle-of-attack) for arresting the descent and optimum pitch commands for performing the climb-out. The computer can also be used to provide commands during takeoff.

Two G/S receivers and two VHF Navigation (LOC) receivers provide signals for making the AWLS ILS approach. It is necessary that they be tuned to the same ILS frequency and be functioning normally for the AWLS approach to be accomplished.

The low-range radar altimeter (Chapter 2) provides absolute altitude which is displayed on the radar altitude indicator, and on the altitude pointer of each ADI.

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* Shown on Figure 1-4, main instrument panel

Radar altitude is also furnished to the flare computer for developing the LAND ARM, FLARE, and throttle retard signals. The low-range radar altitude indicator furnishes a minimum radar altitude warning signal to the Minimum Decision Altitude (MDA) light on each flight progress display panel.

The Test Programmer and Logic Computer (TPLC) (Chapter 10) is a central point which interconnects all other AWLS subsystems. Basically it provides monitoring, programmed testing, gyro attitude intermediate signal selection (using three vertical gyro inputs), and automatic disengage functions.

The Vertical Navigation (VER NAV) system (Chapter 3) is a natural supplement to AWLS as are lateral navigation systems. Lateral navigation systems provide lateral guidance for getting the aircraft within 20 miles of the terminal area, and VER NAV provides descending vertical guidance to the A/P and FDS's for getting the aircraft down from cruise altitude to terminal area altitude (1500 to 2500 feet) where the normal ILS or AWLS approach begins. VER NAV can also be used to provide ascending vertical guidance to a desired cruise altitude.

AIRCRAFT INSTALLATION

The AWLS consists of multiple equipment, subsystems and associated systems, and equipment all integrated to perform as a single functioning unit to safely guide the aircraft to the runway. All displays associated with AWLS are arranged on the main instrument panel, as shown in Figure 1-4, to provide the pilots with easily interpreted information pertinent to the AWLS approach. All controls, with the exception of the AWLS ARM switch which is on the A/P control panel, are located on the center console, as shown in Figure 1-5, to permit maximum usability with minimum pilot effort and distraction.

AWLS EQUIPMENT, SUBSYSTEMS, AND ASSOCIATED SYSTEMS

AWLS equipment and subsystems are as follows:

- AWLS Master Caution System
- Flare Computer
- Rotation/Go-Around Computer
- Test Programmer and Logic Computer

AWLS associated systems are as follows:

- Autopilot System
- Two Flight Director Systems
- Low-Range Radar Altimeter System

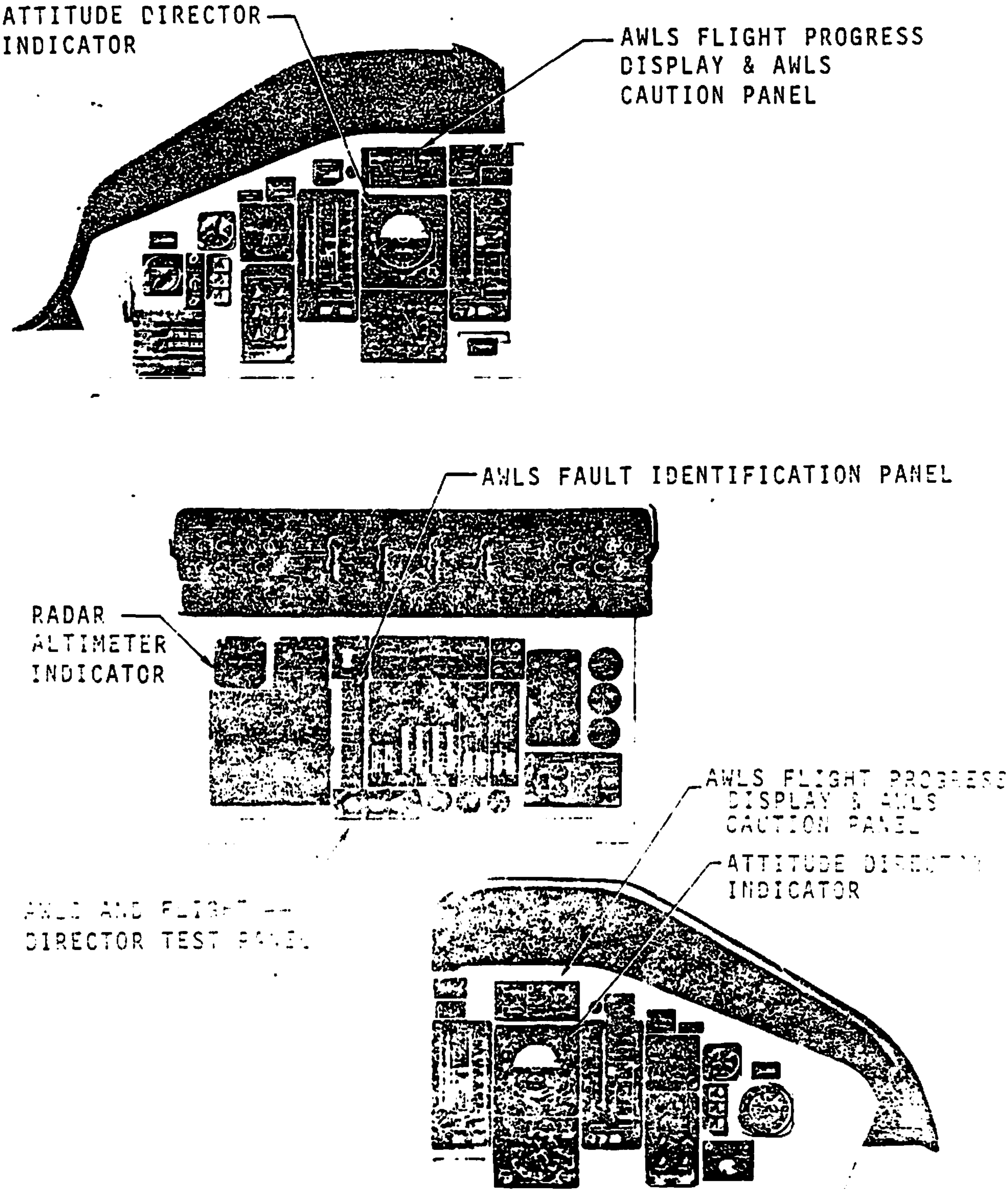


FIGURE 1-4. MAIN INSTRUMENT PANEL

- Automatic Throttle System
- Two Glide Slope Systems
- Two VHF Navigation Systems (LOC)
- Two C-12 Compass Systems
- Two Central Air Data Computers
- Vertical Navigation System (not used during actual AWLS approach)

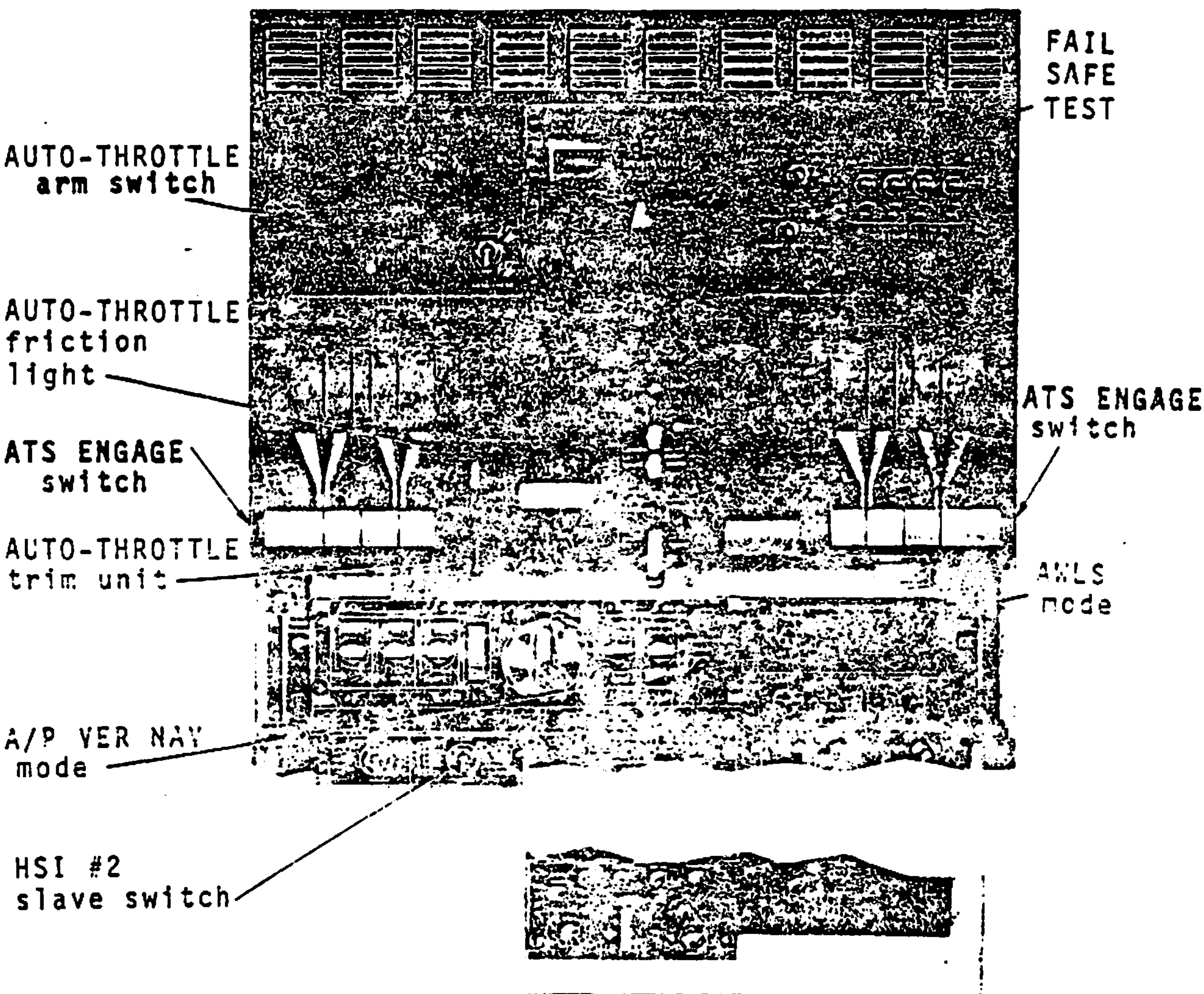


FIGURE 1-5. CENTER CONSOLE

Most of the rack-mounted components of the above systems are located in the avionics equipment racks under the flight station floor.

NOTE

Refer to specific system chapters for a breakdown of components and locations.

SYSTEM OPERATION

General operation of each AWLS subsystem and associated system is given first and then followed by an explanation of total AWLS operation. More detailed operation of subsystems and associated systems are found in appropriate system chapters. A block diagram of the AWLS is shown in Figure 1-6.

Autopilot System

The Autopilot (A/P) system (Chapter 9) is an associated system of the AWLS. It provides automatic control of the aileron and elevator axes and also controls the rudder through the yaw damper system (Chapter 8) to provide turn coordination. Automatic control of the AWLS approach is provided by fail-safe operation of the A/P, i.e. if a failure occurs that affects automatic control, the appropriate axes (roll or pitch) automatically disengage.

Sensor inputs for the A/P are provided from No. 2 systems when dual systems are on the aircraft.

The A/P is interconnected with the copilot's (No. 2) FDS. Navigation signal inputs and navigation mode selections are provided from the copilot's navigation selector panel. Heading and course select signals are supplied from the copilot's Horizontal Situation Indicator (HSI). Autopilot panels and controls are shown in Figure 1-7.

The AWLS switch, located on the A/P control panel, is used to arm the AWLS. Its operation does not require the A/P to be engaged.

Built-In Test Equipment (BITE) is implemented in the A/P to serve two functions; First, it enables functional testing of the A/P circuits that are used for an AWLS approach. Second, it provides monitoring of the A/P during the AWLS approach after APPR ARM progress. If faults are detected during test or during the AWLS approach, the appropriate failure lights (A/P ROLL or A/P PITCH) illuminate. The A/P test is automatically run as part of the AWLS "enroute" test. The A/P disengages when the enroute test is initiated. Manual re-engagement at completion of the test is required if A/P operation is desired.

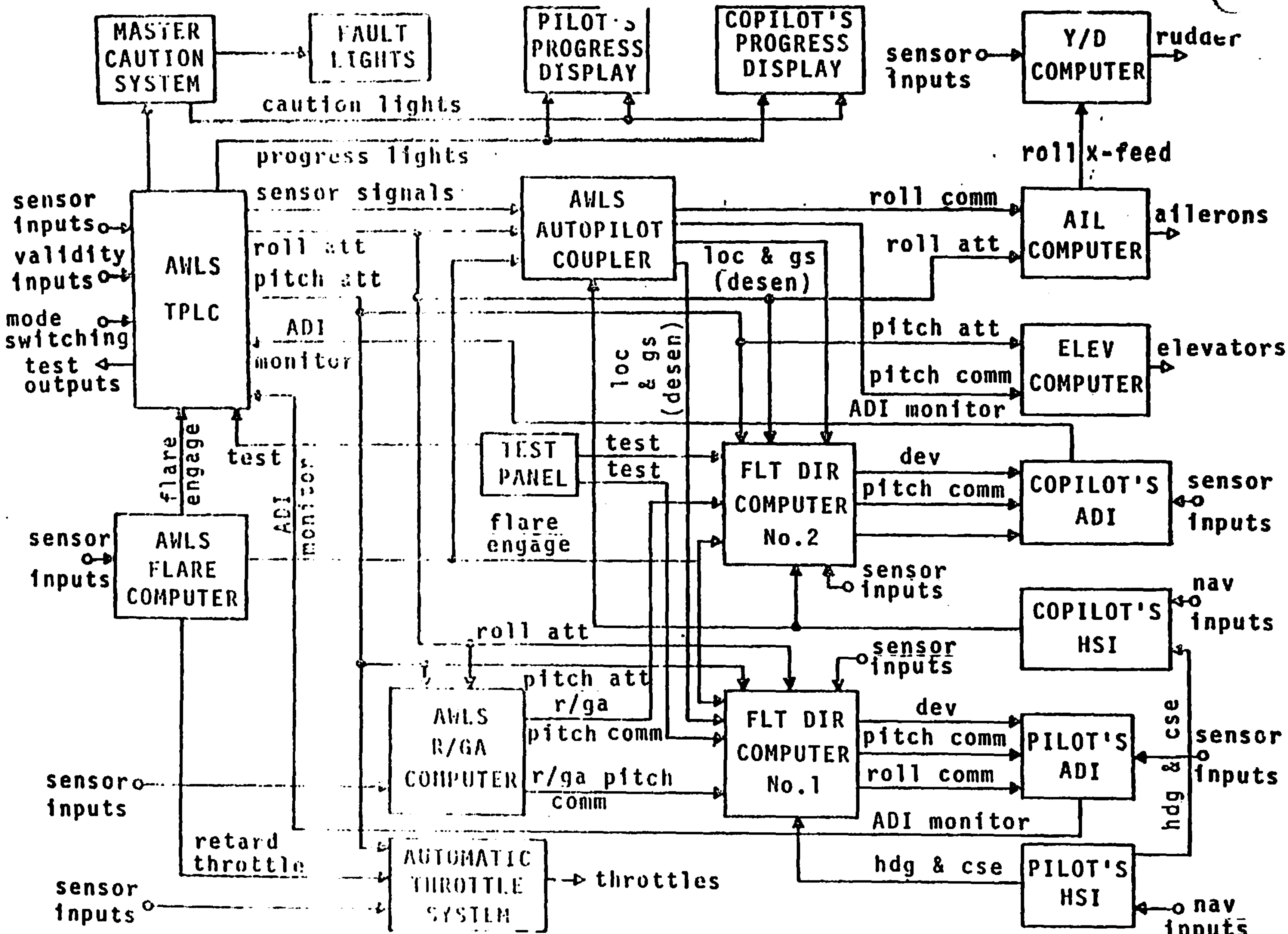


FIGURE 1-6. AWLS DATA FLOW

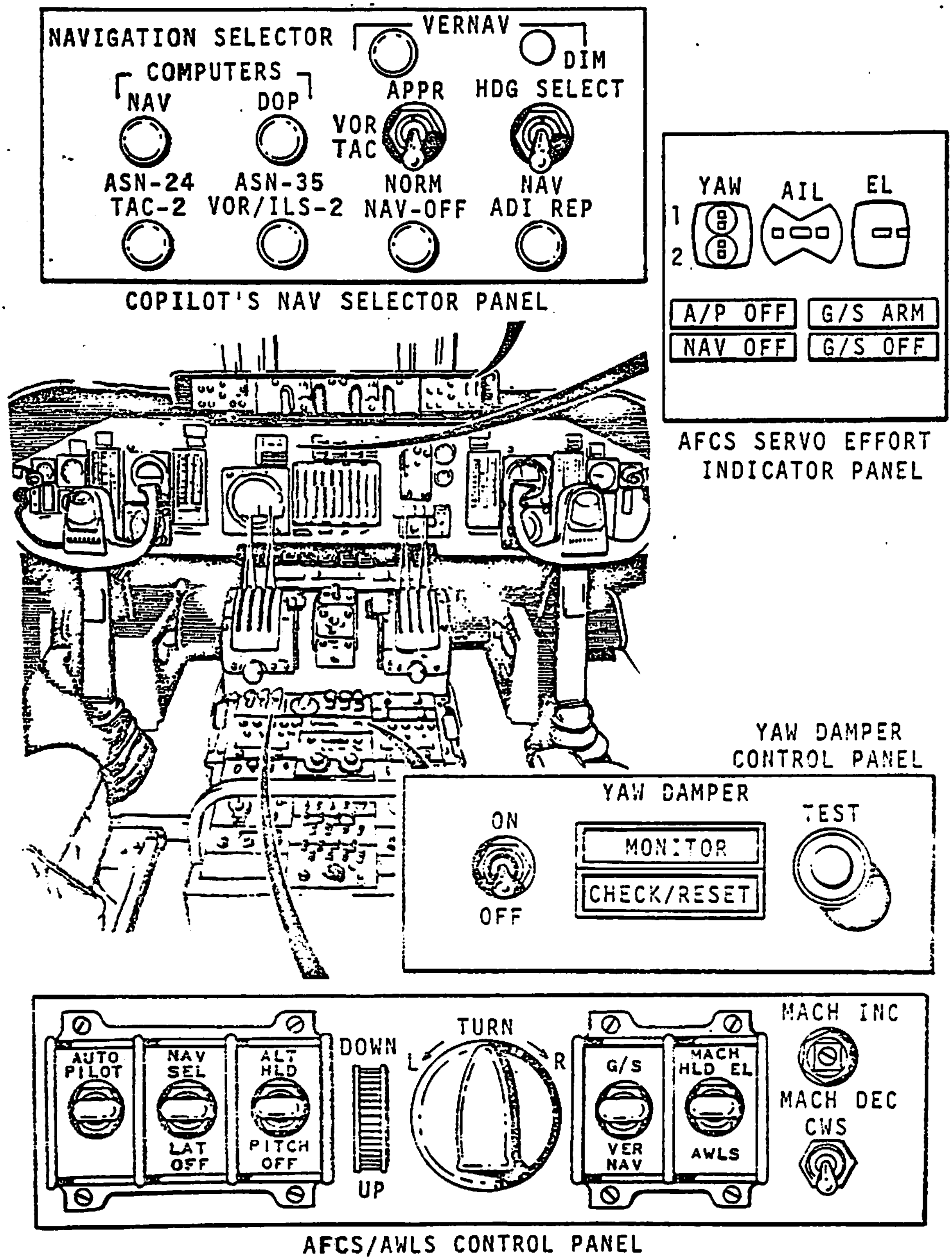


FIGURE 1-7. AWLS/AFCS CONTROL PANELS

Flight Director System

The two FDS's (Chapter 7) are associated systems of the AWLS. FDS No. 1 is the pilot's system, and FDS No. 2 is the copilot's system. They provide a visual display of guidance commands to the pilots to enable manual control of the aileron, elevator, and rudder axes for flying the desired navigational path. During A/P control, the FDS's serve as monitoring or back-up systems.

Separate sensor inputs for the two systems are provided in all cases where dual systems are on the aircraft. For example, the FDS No. 1 receives inputs from G/S No. 1, VOR/LOC No. 1, Tacan No. 1, and C-12 compass No. 1. FDS No. 2 uses the same sensor's inputs as the A/P.

Roll and pitch attitude for the pilot's and copilot's ADI attitude display (artificial horizon) is supplied directly from gyro No. 1 and gyro No. 2, respectively. Roll and pitch attitude used in FDS steering computations is supplied from the TPLC attitude intermediate signal selectors.

Radar altitude is displayed on the ADI altitude pointer. The pointer represents a miniature runway. For altitudes above 200 feet, the pointer is deflected down and out-of-view. At 200-foot altitude the pointer is in view at the bottom of the indicator. As altitude decreases, it moves toward the miniature aircraft symbol. At zero-foot altitude, it will have moved to the aircraft symbol.

Most FDS modes are selected from the No. 1 and No. 2 navigation selector panels. Normal FDS modes not new to AWLS are discussed in the FDS chapter.

The VER NAV mode is selected when the navigator places the vertical navigation system to active operation. Selection of this mode is indicated by the illumination of the VER NAV mode light on the navigation selector panels. An intensity control for the VER NAV mode light is adjacent to the light. In this mode, the ADI's display displacement and pitch steering from the vertical navigation system.

The R/GA mode is selected when the go-around button on either pilot's outer control wheel grip is pressed. The ADI's display pitch steering commands from the R/GA computer and wings level commands from the FDS computer.

The FLARE mode is automatically selected during an AWLS approach at about 45 feet, radar altitude. The mode select signal comes from the flare computer by way of the TPLC to the FDS's. Flare maneuver pitch steering commands from the flare computer (Chapter 5) are displayed on the ADI's in this mode.

The A/P mode is provided in the copilot's FDS only. Switching to this mode is automatic and is done during an AWLS approach by the TPLC at APPR ARM progress. Both the lateral and vertical channels switch to the A/P mode for displaying A/P roll and pitch steering commands on the copilot's ADI. The copilot's ADI therefore provides A/P steering displays for monitoring purposes.

The VOR/TAC APPR mode is manually selected on the navigation selector panel and is used with VOR or Tacan after on course tracking is established. Its purpose is to increase bank steering sensitivity for low-speed, close-in VOR or Tacan approaches.

FDS mode priorities are established as follows from highest to lowest priority:

LATERAL CHANNEL	VERTICAL CHANNEL
HDG Select	R/GA
R/GA	A/P (FDS No. 2 only)
A/P (FDS No. 2 only)	ILS (Glide Slope)
VOR/ILS, TAC, ASN-24, ASN-35 (as manually selected)	VER NAV

The TPLC performs the following signal or validity monitoring functions associated with the FDS's:

<u>ADI Attitude Display</u> - The TPLC monitors the position of the pilot's and copilot's attitude display (artificial horizon). If either differs appreciably from the gyro input, the appropriate GYRO 1 or GYRO 2 fault light illuminates.

NOTE

Invalid gyro attitude signal outputs also illuminate the GYRO fault lights.

<u>TPLC Attitude Outputs</u> - The TPLC monitors TPLC attitude outputs signals. If those supplied to either FDS computer fail, the appropriate ADI steering pointer is pulled out-of-view.

<u>Localizer 1 and 2, Glide Slope 1 and 2, and Flare Validities</u> - If either of these are invalid during the AWLS approach after APPR ARM, the appropriate ADI steering pointer is pulled out of view and the appropriate AWLS fault light will illuminate. A flare signal failure would have no effect until the flare mode occurs.

<u>Flare System</u>

The flare computer is a subcomponent of the AWLS. During the AWLS approach, it provides flare maneuver commands for display on the FDS's ADI pitch steering bars and to the autopilot. In addition, it provides the LAND ARM signal (100-foot radar altitude), the FLARE engage signal (45-foot radar altitude) and the throttle retard signal (30-foot radar altitude). The AWLS LAND ARM and FLARE progress lights illuminate as a result of the first two. The flare engage signal is also used for AWLS mode switching. The throttle retard signal is sent to the ATS to initiate automatic retarding of the throttles.

The flare computer has BITE capabilities which enable automatic testing and AWLS monitoring. Testing is performed during the AWLS enroute and preland test. If a flare fault occurs during test or during AWLS approach, the FLARE fault light illuminates. If the land arm circuits fail, the LAND ARM fault light illuminates.

Rotation/Go-Around (R/GA)

The R/GA system (Chapter 6) is a subsystem of the AWLS. It provides optimum commands (angle-of-attack) for arresting the descent and making the climb-out in case of an approach abort. It also provides optimum pitch rotation angle for use during take-off and optimum pitch commands for making the climb-out lift-off. In the R/GA mode, these commands are displayed to both pilots simultaneously on their ADI pitch steering bars.

Either pilot selects the rotation or go-around mode by pressing the GO-AROUND select button on his outer wheel grip. Pressing either button a second time removes the R/GA mode.

BITE is implemented in the R/GA system to provide continuous self-test and for testing its monitors during the AWLS enroute test. If a system failure occurs, the following things happen:

- The R/GA fault light illuminates.
- The Instantaneous Vertical Velocity (IVV) to the Vertical Velocity Indicator (VVI) is replaced with vertical velocity from the central air data computers.
- The VER NAV system is rendered inoperative.

Test Programmer and Logic Computer (TPLC)

The TPLC (Chapter 10) is a subcomponent of the AWLS. It interconnects all AWLS equipment and performs various functions for AWLS and non-AWLS operation. Four major functions are performed by the TPLC:

1. Gyro Attitude Intermediate Signal Selection - The TPLC receives roll and pitch from three vertical gyros. These attitude signals are processed through

six Intermediate Signal Selector (ISS) circuits. The middle amplitude signals are selected by each ISS and distributed to all using equipment except the ADI's. Roll and pitch attitude for the attitude display (artificial horizon) on the ADI No. 1 and No. 2 is supplied directly from gyro No. 1 and No. 2 respectively. Bank versine is also supplied directly from gyro No. 3 to the A/P elevator computer.

2. <u>Monitoring</u> - The TPLC performs analog signal and binary signal monitoring of all critical AWLS functions as well as some non-AWLS functions.

Analog signal monitoring includes monitoring of the following:

- ISS Inputs and Outputs
- Five Accelerometers
- Two ADI Roll and Pitch Attitude Displays
- TPLC Power Supplies

Binary signal monitoring functions include monitoring of validities from AWLS equipment, as follows:

- A/P System
- A/P Coupler (includes localizer, glide slope, and radar altitude validities)
- Flare System
- Radar Altimeter
- TPLC (warning, switching, test, and monitor locations

If a failure is detected during the AWLS approach, or during AWLS test, the TPLC causes the associated failure lights to illuminate, ADI steering pointers to be pulled out of view, A/P axes (roll or pitch) to be disengaged as appropriate, and AWLS unit fault indicators on the front of the TPLC to be activated as appropriate.

NOTE

The FDS's computers and the ATS contain self-monitoring. Illumination of their failure lights is not a function of TPLC monitoring.

3. <u>Programmed Test</u> - Two tests are programmed by the TPLC: the manually initiated enroute test and the automatically initiated preland test. During these tests, the TPLC performs the test on some equipment and commands other equipment to test itself. In either case, the TPLC monitors the test results and causes failure lights to illuminate if failures are detected.

To initiate the enroute test, the AWLS arm switch on the A/P control panel must be placed to the "AWLS" position. At test initiation, which is started when the AWLS TEST/RESET button is pressed, the following action occurs:

- o The A/P disengages (if it was engaged).
- o The AWLS TEST (test in progress) light and both FDS TEST (test in progress) lights illuminate.

NOTE

FDS test is also initiated with the AWLS TEST/RESET button. (The FDS test is described fully in Chapter 7 of this manual.)

- o All fault lights except the THROT, FLT. DIR. 1 and FLT. DIR 2 fault lights illuminate (flashing) and go out after 2 seconds.
- o Both caution lights (AUTO and MANUAL) illuminate and then go out after 2 seconds.
- o The LOC and G/S progress lights illuminate and remain on for the duration of the test.

After two seconds, the preland test is then run as part of the enroute test program. At completion of the preland test the APPR ARM progress light illuminates and remains on for the rest of the test.

The test program then runs 14 additional test steps to check the functional ability of all AWLS equipment and to check all AWLS monitoring circuits. During the test steps, various fault lights are turned on and off (either individually or in groups). Some come on more than once. The remaining progress lights also come on and remain on for the rest of the test.

At test completion, the AWLS TEST (test in progress) light goes out. If the test is satisfactorily completed, the following results are obtained:

- o The AWLS TEST/RESET button light illuminates.

- All progress lights are illuminated.
- All fault lights are out.

If the test is not satisfactorily completed, the following results are obtained:

- The AWLS TEST/RESET button is not illuminated.
- All progress lights may not illuminate.
- Various fault lights with the associated caution light is illuminated, depending on type of failure.

In either case, the test results should be acknowledged by pressing the TEST/ RESET button again. This action takes the AWLS out of the test mode. All illuminated lights then go out, and the FDS reverts to its original mode of operation. The A/P may now be re-engaged.

Satisfactory completion of the enroute test verifies the operational integrity of the following AWLS equipment:

- A/P
- FDS's
- TPLC
- FLARE.
- R/GA
- AWLS Master Caution System

4. Automatic Disengage - Disengagement of the A/P, ATS, and AWLS is provided when R/GA is selected. A/P disengagement also occurs when the AWLS enroute test is initiated.

AWLS Master Caution System

The AWLS master caution system is a subsystem of the AWLS which receives failure signals from the TPLC, FDS's, and ATS. A failure signal causes a fault light to flash and the appropriate caution light to illuminate.

Two flasher circuits are used, one each for the manual and auto fault lights. If either flasher fails, a fail-safe circuit bypasses the flasher to illuminate the fault lights in response to AWLS failure signals.

All lights on the AWLS progress display panels, fault identification panel, and test panel are tested with the AWLS caution lights test switch, as shown in Figure 1-8. It is a three-position switch spring-loaded to the center off position. "TEST" and "FAIL SAFE TEST" are the two momentary positions. When held in the "TEST" position, all lights come on. The fault lights are flashing. Those associated with manual AWLS monitoring and those associated with auto AWLS monitoring most probably have slightly different flash rates and therefore will not flash together. The reset function is checked by pressing the MANUAL and AUTO caution lights. As each is pressed, it goes out and the associated fault lights burn steadily. The FAIL SAFE TEST results are the same except the fault lights do not flash.

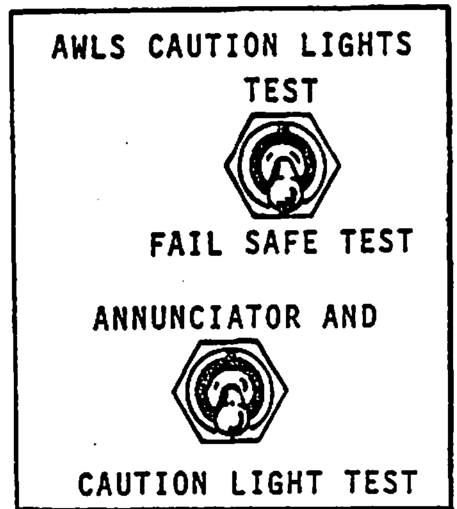


FIGURE 1-8. ANNUNCIATOR AND CAUTION LIGHT TEST PANEL

Low-Range Radar Altimeter

The low-range radar altimeter (Chapter 2) is an associated system of the AWLS. Its operation is necessary during the AWLS approach. The system provides a display of absolute altitudes between minus 1.5 and 2500 feet on the radar altitude indicator located on the left side of the center instrument panel. Altitude is displayed by the altitude pointer and the altitude turns counter.

All system controls, which are on the indicator, follow:

- ON-OFF/TEST Knob
- Altitude Set Knob

When the system is turned OFF and system power is applied, the indicator drives to 2500 feet and a mask comes into view which covers the altitude display. Any time system power is removed, the mask goes out of view.

For all altitudes above 2500 feet, the altitude display (pointer) is covered by the mask. On the ground with normal compression of the landing gear struts, the indicator reads approximately minus 1.5 feet. With struts fully extended as an initial landing touchdown, the indicator reads approximately zero feet. The indicator provides a minimum radar altitude warning signal to the MDA lights on the flight progress display panel when the aircraft goes below a desired minimum altitude.

Radar altitude is supplied for display on the radar altitude pointer of both flight director ADI's. The range of this display is zero to 200 feet. Radar altitude is also sent to the flare computer and A/P coupler. The minimum radar altitude warning signal, in addition to turning on the MDA light, is sent to the VER NAV system.

BITE is implemented in the system to provide monitoring and manually initiated system test. A striped fail flag appears over the altitude turns counter, and the ADI altitude pointers deflect out of view if the system fails, is turned OFF, or is in self-test. The RDR ALT fault light on the AWLS fault identification panel also illuminates if a system failure occurs during the AWLS approach after APPR ARM progress.

The radar altimeter is not tested during the AWLS test.

Automatic Throttle System (ATS)

The ATS (Chapter 4) is an associated system of the AWLS. It provides automatic throttle control for maintaining a desired indicated airspeed during the AWLS approach or during normal cruise, and to retard the throttles during AWLS flare. The IAS present at ATS engage is maintained. All four throttles are controlled together by one ATS servo motor. Slip clutches are provided for each throttle to permit manual adjustment by the pilots for power trimming. Also, the pilots can manually override all four throttles simultaneously with the ATS engaged if the need should arise.

Maximum throttle limits are provided to prevent the ATS from exceeding engine parameters. Minimum throttle limits are provided to prevent throttle travel to the idle position. Two throttles must reach the limits before further travel in that direction is prohibited. During throttle retard during flare, the minimum limits are bypassed, which allows the throttles to be retarded to idle. When two throttles reach idle, the ATS disengages.

BITE is implemented in the ATS to provide monitoring and maintenance testing. If a failure occurs, the system automatically disengages. This action causes the throttle fault light to illuminate. The ARM switch is then turned off and the light goes out.

Test switches and lights are located on the front of the ATS computer for performing system fault isolation test procedures. The ATS is not tested during the AWLS test.

Vertical Navigation (VER NAV) System

The VER NAV system (Chapter 3) is an associated system of the AWLS. Its control panel is located at the navigator's station. The VER NAV system provides vertical guidance for descending or ascending from one altitude to another. VER NAV information is displayed on both flight director ADI's for manually flying the VER NAV path and is furnished to the A/P for automatically flying the VER NAV path.

Two stages of operation are provided and controlled with the STANDBY/STAGE 1/

STAGE 2 selector switch. Various manual inputs must be set in on the VER NAV control panel to obtain proper operation.

Manual inputs follows:

- STAGE 1 and STAGE 2 aimpoint altitudes
- STAGE 1 and STAGE 2 VER NAV angles
- STAGE 2 offset (moves ASN-35 or ASN-24 destination point in a straight line toward the aircraft)
- Barometric correction (used with both stages)

Distance-to-go is supplied from the ASN-35 or the ASN-24. Selection of one or the other is made with the AUX/ASN-35/ASN-24 switch on the control panel. The "AUX" position is not used.

A VER NAV ANGLE indicator on the control panel displays the actual angle from the aircraft to the aimpoint. During operation (STAGE 1 or STAGE 2), the system provides VER NAV displacement signals and VER NAV pitch steering commands. Both FDS's are automatically switched to the VER NAV mode when STAGE 1 or STAGE 2 is selected.

The A/P must be manually switched to the VER NAV mode if automatic control of the VER NAV flight is desired.

If the problem is properly set up, the VER NAV system is initially in the VER NAV arm submode. As the aircraft progresses along the VER NAV path as shown in Figure 1-9, submode switching occurs in sequence as follows:

- Arm
- Capture
- Track
- Altitude hold capture/track

The VER NAV displacement signals are displayed on the ADI's displacement pointers during all four submodes. The VER NAV pitch steering commands are displayed on the ADI's pitch steering bars and sent to the A/P during the capture, track, and altitude hold capture/track submodes.

BITE is implemented in the VER NAV system to provide monitoring and manually initiated system test. The control panel fail light comes on if the VER NAV system fails, the R/GA fails, ASN-35 is selected and <u>is not</u> in the DROP mode,

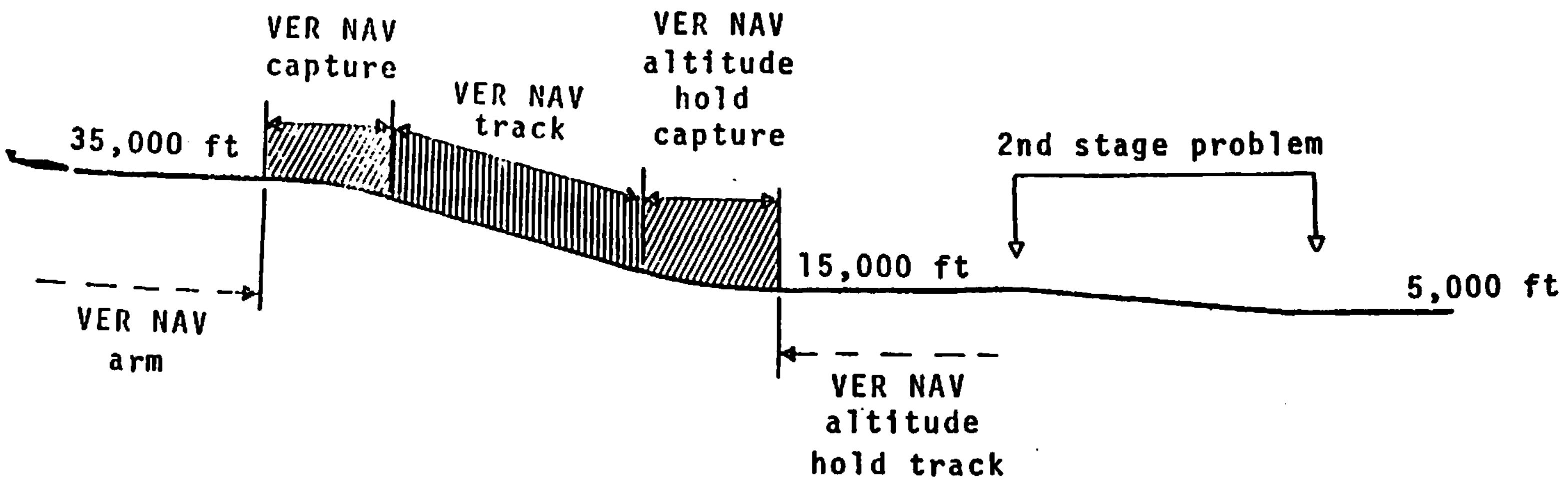


FIGURE 1-9. VER NAV FLIGHT PATH

or if the system is in self-test. The BITE also provides system maintenance testing. A test meter, test switches, and a FAIL light are located on the front of the computer for performing system fault isolation test procedures.

The VER NAV system is not tested during the AWLS test.

Total AWLS Operation

During an approach, the AWLS can be operated in two general modes: manual and/or automatic. In the manual mode, the AWLS provides visual guidance to both pilots through FDS ADI displays for manual control of the aircraft. In the automatic mode, the AWLS provides guidance commands to the autopilot for automatic control of the aircraft. In the automatic mode, visual AWLS guidance continues to be displayed to the pilot. Visual guidance is also displayed to the copilot until APPR ARM occurs (about 30 seconds after G/S intercept); then autopilot roll and pitch commands are displayed for the rest of the automatic AWLS approach. There is no manual/automatic mode selection switch as such. If manual control is desired, only the FDS's are set up for making the approach. If automatic control is desired, the A/P is also engaged and set up for making the approach.

Total AWLS operation is presented in the order in which events would occur during a normal flight from cruise altitude to landing including a vertical navigation let-down.

The AWLS enroute test would be performed while at cruise altitude with the aircraft flying essentially straight and level. The A/P and FDS's should be operating in their normal cruise modes and the radar altimeter must be turned on. The AWLS is armed with the switch on the A/P control panel. The AWLS TEST/RESET button is pressed to start the test. (The TPLC section contains a description of the enroute test.) At completion of the test, the AWLS arm switch is placed in the center "OFF" position. The A/P can be re-engaged, if desired.

The VER NAV system let-down may now begin using one or both stages (both are normally used for let-down from high altitudes). The navigator may have already set up the VER NAV problem (or problems); if not, he may do so at this time. He will use the ASN-24 or ASN-35 destination as the termination of aimpoint 1 and set in the proper terminal altitude and descent angle. The appropriate barometric setting (enroute or destination) should be obtained and set in. Now the stage switch should be moved from "STANDBY" to "STAGE 1." The VER NAV system is now in the arm submode and the VER NAV mode lights on both pilots navigation selector panels illuminate. Both ADI's displacement pointers begin displaying VER NAV deviation and continue to display VER NAV deviation for the duration of the STAGE 1 problem. The A/P VER NAV mode may also be engaged. If the altitude hold or the mach hold mode is engaged, it remains engaged during the VER NAV arm mode. VER NAV pitch steering signals are not sent to either the A/P or FDS's until VER NAV capture occurs.

As the VER NAV path is approached, the VER NAV system switches to the capture mode far enough in advance of actual intercept to allow a comfortable nose over without overshooting. VER NAV pitch steering commands are now sent to the A/P and FDS's and continue to be sent for the rest of the STAGE 1 problem. The ADI's pitch steering bars come into view at this time to provide VER NAV flight path guidance. The autopilot ALT HOLD or MACH HOLD mode, if engaged, automatically disengages, and the A/P provides automatic pitch steering for flying the VER NAV path.

As the aimpoint altitude is approached, the VER NAV switches to the altitude hold capture mode far enough in advance of actual altitude intercept to allow smooth termination of the descent. Subsequent tracking of the aimpoint altitude is provided until commanded to do otherwise, e. g. switching to STAGE 2 or turning VER NAV to standby.

In response to the VER NAV pitch steering commands, the aircraft captures and tracks the selected VER NAV path and aimpoint altitude if either the FDS, for manual control, or the A/P for automatic control, are used.

The navigator may now program the second VER NAV stage if he has not already done so, including an offset distance if desired. A fix or course change could be made at this time. When STAGE 2 is selected, the second VER NAV path would be flown in the same manner as the first. The aimpoint would probably be in the terminal area (within 20 miles of the runway) with an aimpoint altitude of 1500 to 2500 feet.

<u>AWLS Operation</u>

The aircraft should now be in the terminal area and the crew would prepare for the AWLS approach, as shown in Figure 1-10. Both VHF navigation receivers would be tuned to the same ILS frequency, and both VOR/ILS buttons on the navigation selector panels would be pressed. This action establishes HSI No. 2 heading and course set slaving. The pilot would set in the localizer beam intercept heading and the inbound runway course on his HSI. The copilot's HSI would be slaved to the pilot's settings. The pilot would select "G/S" on the A/P control panel which illuminates the G/S ARM light on the A/P servo effort indicator panel.

NOTE

Assume that the radar altimeter is turned on, and arm the AWLS.

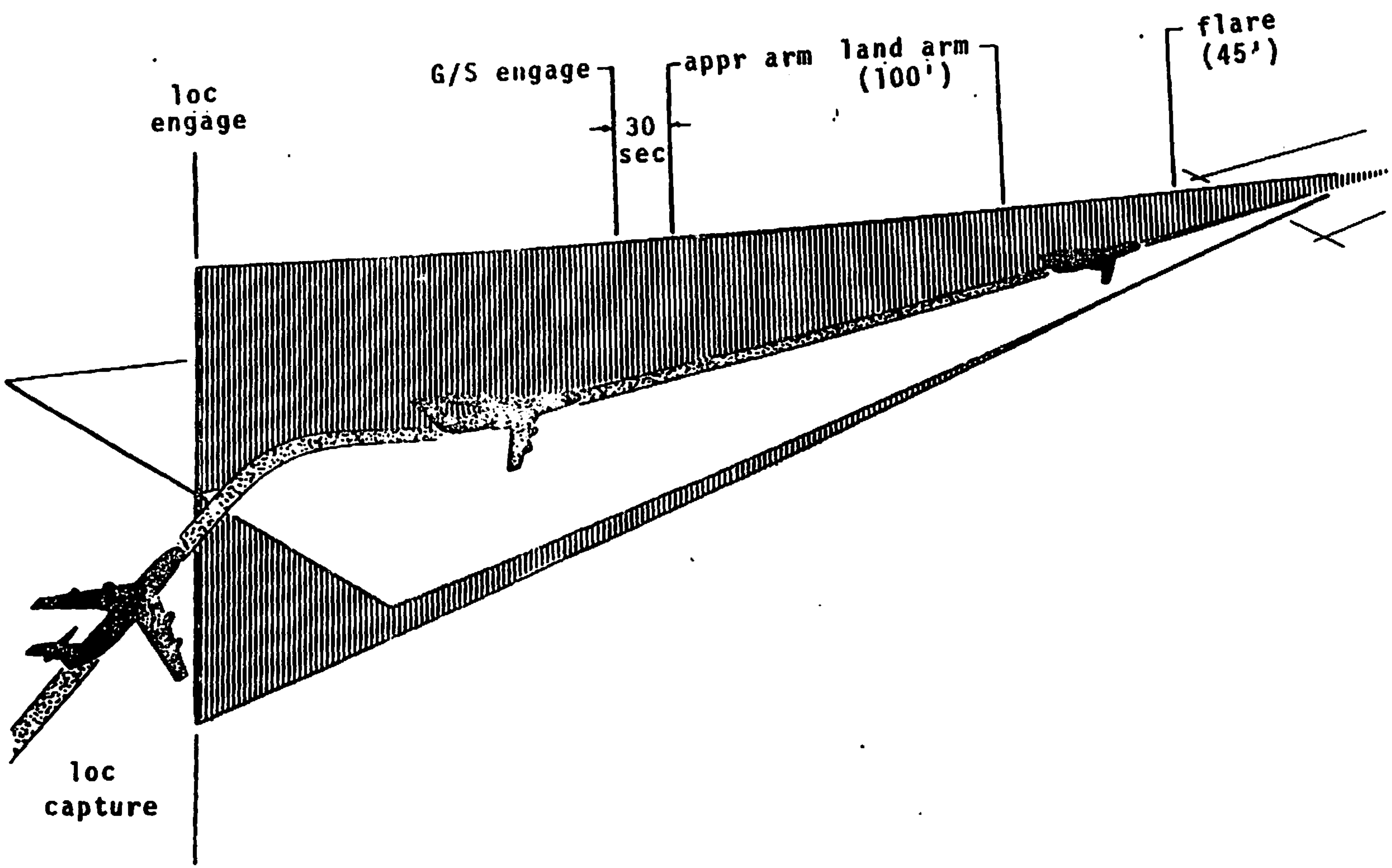


FIGURE 1-10. TYPICAL AWLS LANDING APPROACH

LOC Engage - The aircraft progresses from localizer intercept heading through localizer capture to the point of localizer beam intercept at which time the LOC progress light illuminates. Tracking of the localizer continues from this point on. The ATS system may be engaged for controlling the approach speed.

G/S Engage - At glide slope intercept, the G/S progress light illuminates, the A/P G/S ARM light goes out, the A/P begins tracking the glide slope beam, the AWLS preland test is automatically initiated, and the ADI's pitch steering bars come into view to display pitch steering.

APPR ARM - Upon completion of the preland test, the APPR ARM progress light illuminates, which indicates that full AWLS monitoring and warning is then in operation. The copilot's FDS ADI begins displaying A/P roll and pitch commands for monitoring purposes.

LAND ARM - At 100-foot radar altitude, the LAND ARM light illuminates. This altitude is the AWLS approach minimum decision altitude. If visual contact with the runway is established, the approach is continued. (If visual contact is not established, an abort would be initiated when the control wheel go-around button is pressed.)

FLARE - At approximately 45-foot radar altitude, the FLARE progress light illuminates, which indicates that the flare mode is engaged. The control column moves aft to provide up elevator control. The pilot's ADI pitch steering bar displays the flare steering commands while the copilot's ADI pitch steering bar continues to display the A/P pitch commands which are now flare maneuvering commands.

At approximately 30-foot altitude, the throttles begin to retard to the idle position at which time the ATS automatically disengages. At about 12 feet, the pilot should manually decrab the aircraft, as necessary. The touchdown occurs at about 750 feet beyond the glide slope transmitter with a sink rate of about 150 feet-per-minute. As soon as contact is established, the pilot must take control of the aircraft for the roll-out.

The above approach procedure utilizes the A/P for an automatic controlled AWLS approach. If the A/P is not used, the aircraft would be controlled manually in response to roll and pitch commands displayed on the ADI's. All other functions would be the same.

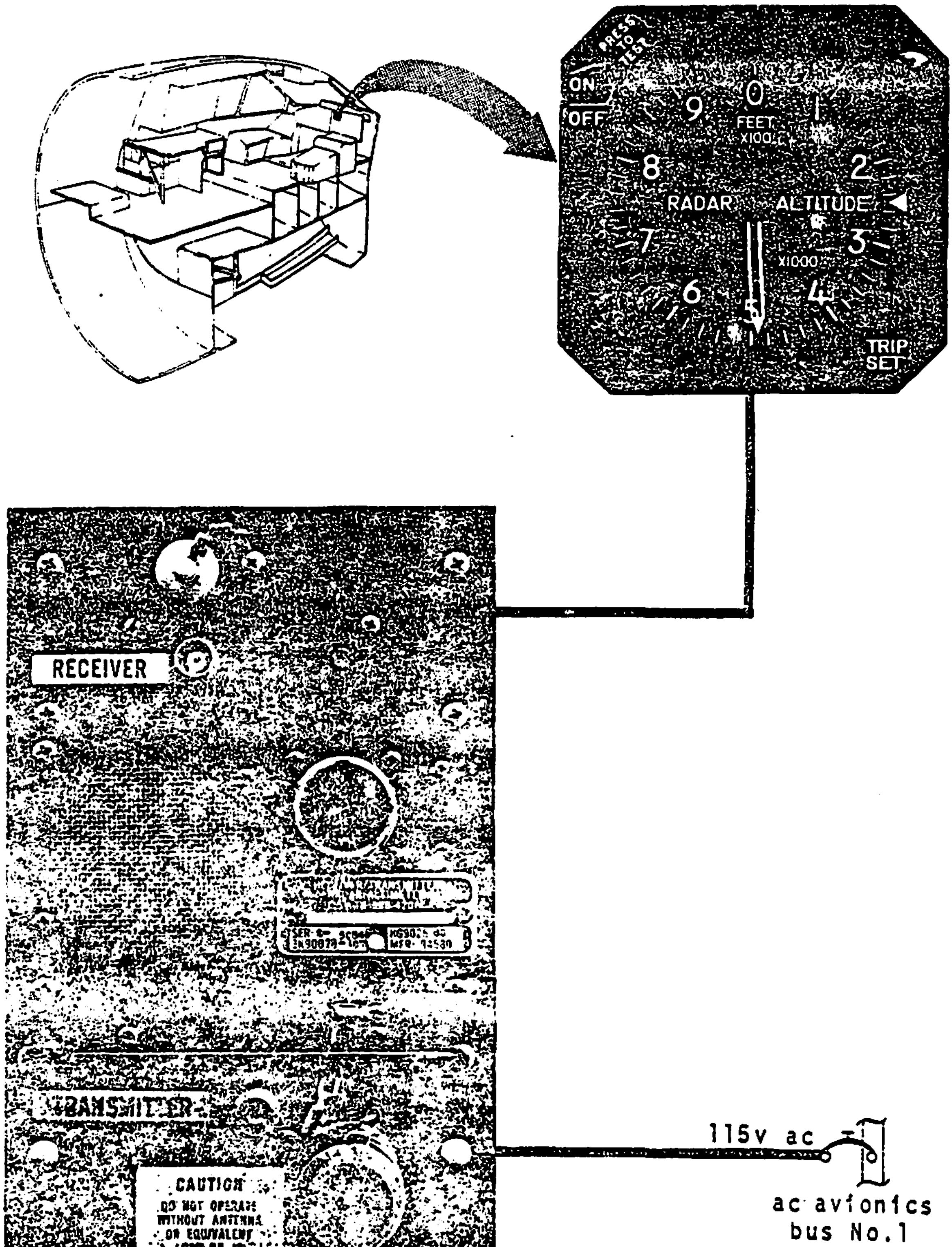


FIGURE 2-1. LOW-RANGE RADAR ALTIMETER AND RECEIVER/TRANSMITTER

LOW-RANGE RADAR ALTIMETER

The radar altimeter, which is interconnected with the All Weather Landing System, is shown in Figure 2-1. The altimeter is a low-level altitude tracking and indicating radar. It instantaneously senses absolute altitude above the terrain within the range of -10 to 2500 feet. Radar altimeter operations are not affected by atmospheric or barometric conditions. Figure 2-2 shows the relationship between radar and barometric altitude

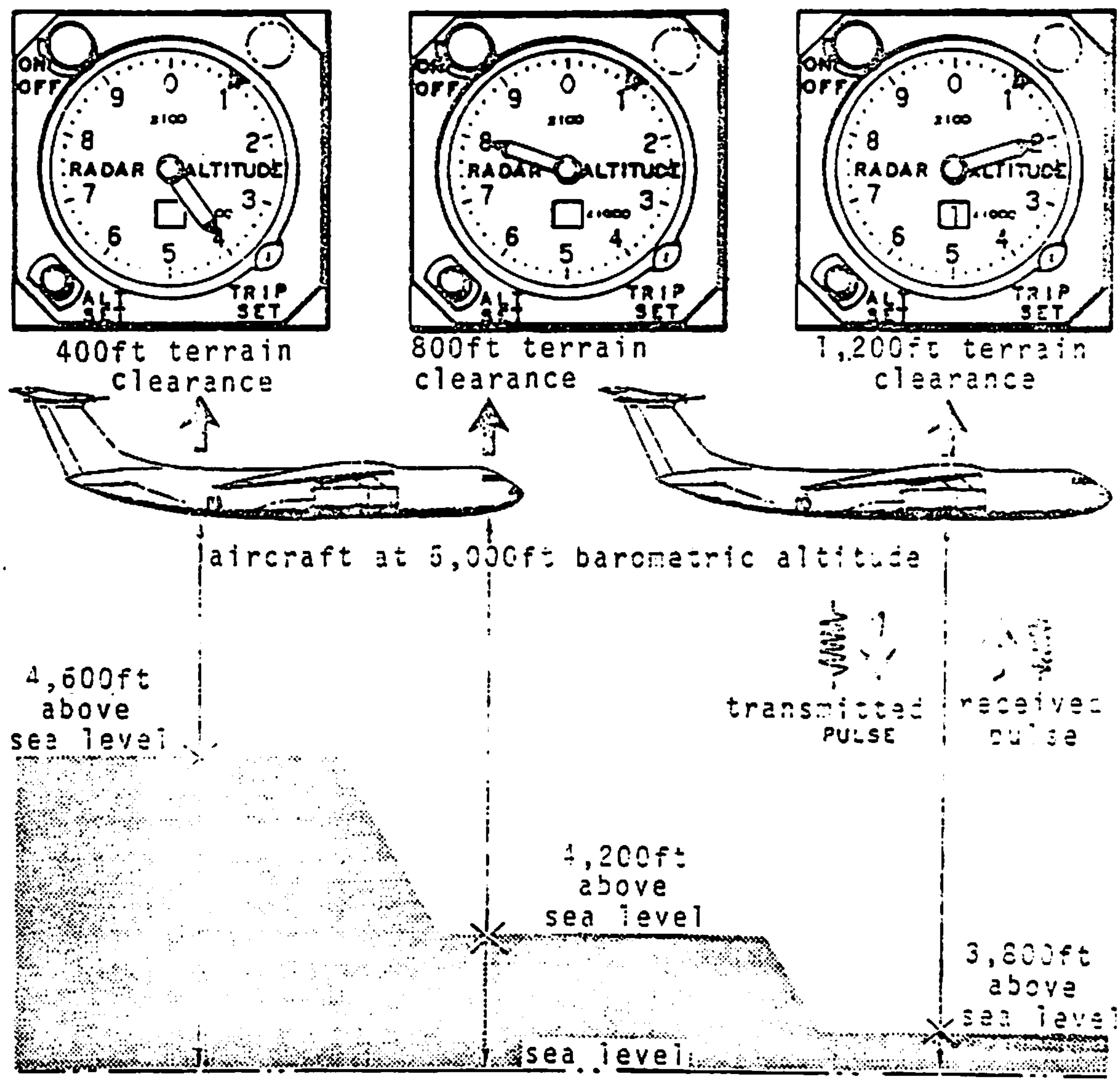


FIGURE 2-2. RADAR ALTITUDE VERSUS BAROMETRIC ALTITUDE

indications.

SYSTEM OPERATION

All controls necessary for operating the altimeter are located on the indicator as shown in Figure 2-3.

When the ON-OFF switch (A) is placed in "OFF," the fail flag (B) appears, and the indicator pointer (H) drives to 2500 feet where a mask (D) appears and covers the pointer. The system is deactivated only after the mask has appeared. The ON-OFF switch is also used to self-test the altimeter by pressing on the knob. The indicator shows a 40-foot altitude in self-test.

The ALT-SET knob (E) is used to preset a Minimum Decision Altitude (MDA). When the aircraft descends below the preset altitude, an MDA signal is produced and sent to the pilot's and copilot's MDA lights and to the [illegible] NAV system. The ALT-SET index cursor (F), located on the outer periphery of the dial scale, can be set to any altitude between zero and 2500 feet with the ALT-SET knob. The ALT-SET turns counter (G) is used in conjunction with the index cursor to indicate the altitude at which the MDA is set. The turns counter (H) consists of a disk segment carrying digits 2 and 1 and a blank. When the index cursor (F) is set below

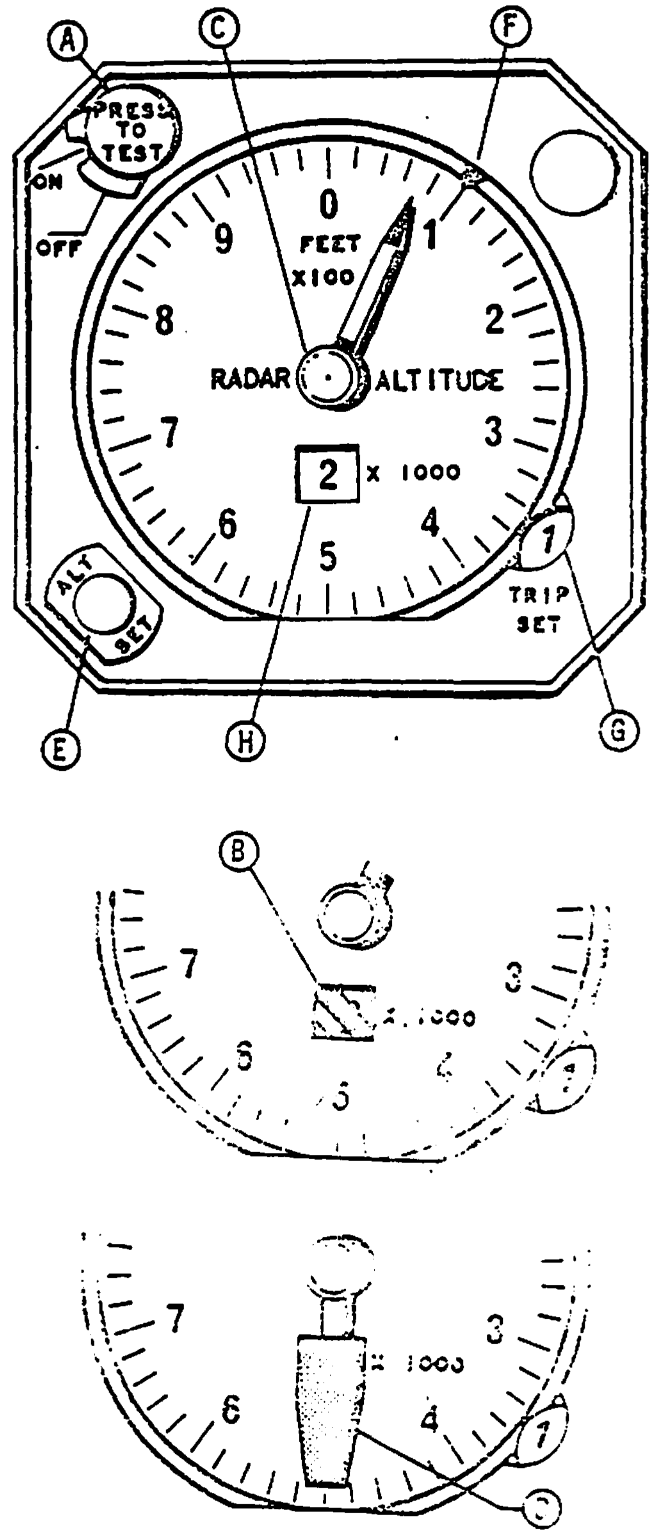


FIGURE 2-3.
RADAR ALTIMETER INDICATOR

1000 feet, the turns counter shows the blank. No. 1 is in view when the index cursor is set between 1000 and 2000 feet. No. 2 is shown when the index cursor is set above 2000 feet.

A flag alarm (B) provides an indication of altimeter failure. It is also displayed during self-test operation. The mask (D) on the indicator is spring-loaded to the out-of-view position. It is held in view to obscure the pointer and altitude window when the system has been turned off and primary power is still available, or when the altimeter is operating above 2500 feet. The mask is removed from view and the flag alarm is in view when the system is in self-test operation.

The indicator displays absolute altitude by using a servo-driven pointer and a turns counter. The pointer (C) indicates in hundreds of feet and the turns counter (H) indicates in thousands of feet. The scale of the indicator is graduated in 20-foot increments.

SPECIFICATIONS
(Minneapolis Honeywell HG9025B-1)

Power Requirements	115 volts, ac, 400 hertz, single-phase
Altitude Range	-10 to 2500 feet
Altitude Accuracy	2 feet or ± 2 percent of altitude
Transmitter	
Carrier Frequency	4300 megahertz
Peak Output Power	100 watts
Pulse Width	
Zero to 500 feet	25 nanoseconds ± 10
500 to 2500 feet	125 nanoseconds ± 25
Pulse Repetition Frequency	10,000 pulses/second

Continued

SPECIFICATIONS (Continued)

Receiver	
Receiver Frequency	4300 megahertz
Local Oscillator Frequency	4300 megahertz
Bandwidth	
Zero to 500 feet	30 megahertz
500 to 2500 feet	10 megahertz
Track Rate (maximum)	2000 feet/second
Indicator	
Track Rate (maximum)	600 feet/second

THEORY OF OPERATION

The low-range radar altimeter is a high-resolution pulse radar operating at 4300 megahertz. Its purpose is to automatically locate the closest terrain returns. It also has the ability to precisely track the rate of altitude change (terrain irregularities or changes in actual aircraft altitude above ground). A block diagram of the system is shown in Figure 2-4.

Operation of the radar altimeter is based on the precise measurement of the time required for an rf energy pulse to travel from the aircraft to the nearest ground and to return. Arrival time of the received pulse is compared with the transmittal time. The time differential is processed in the tracking circuits to provide range (altitude) information. The Receiver-Transmitter (RT) unit is the main component of the system. All electronic circuits are contained in this unit with the exception of the indicator servo system.

The RT unit is divided into three sections:

- Transmitter
- Receiver
- Tracker

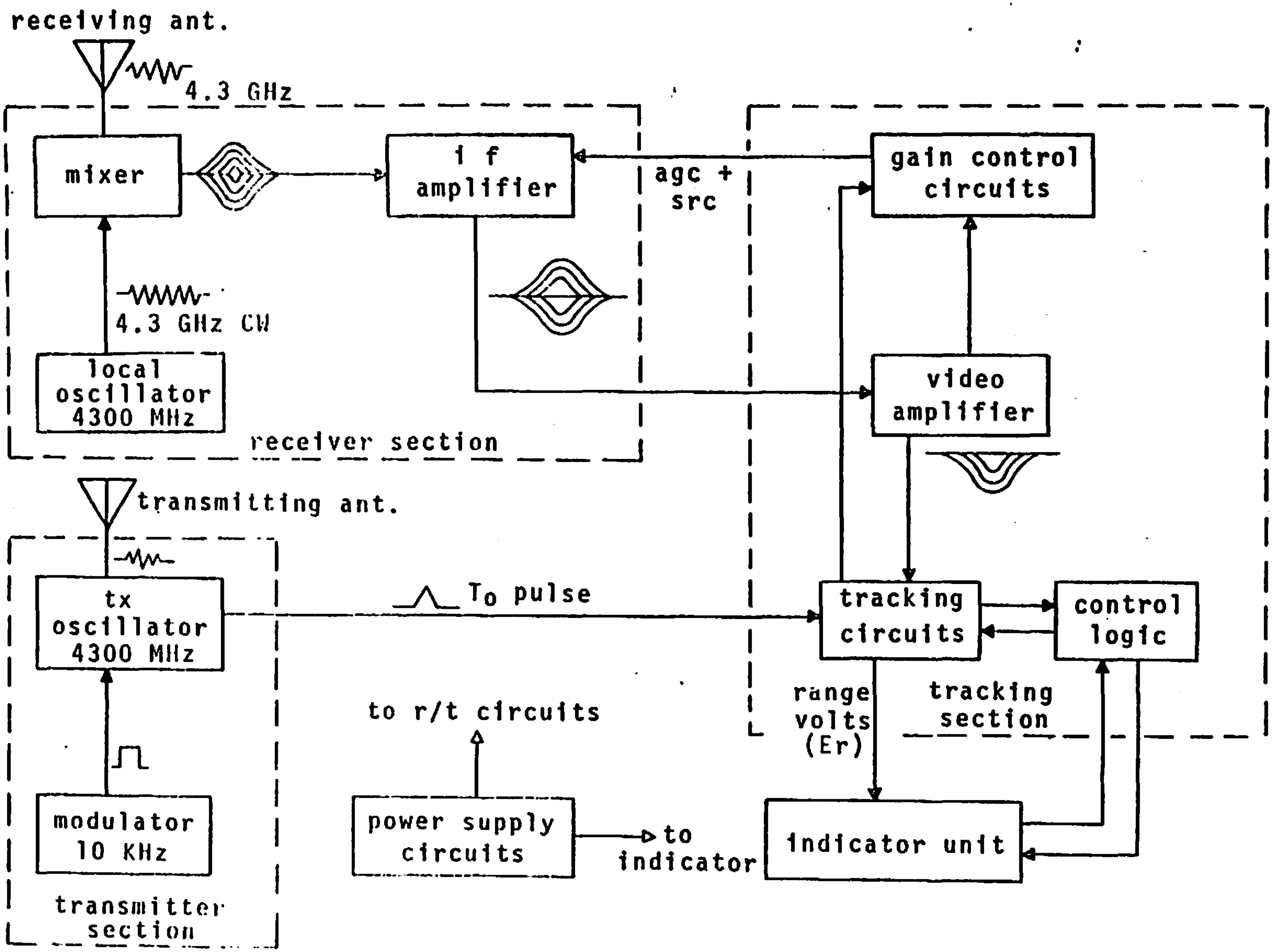


FIGURE 2-4. WAVEFORMS BLOCK DIAGRAM

Transmitter

The transmitter section, as shown in Figure 2-5 consists of a transmitter oscillator and a modulator (PRF generator). The modulator is a free-running multivibrator operating at 10 kilohertz. Each pulse from the modulator triggers the transmitter oscillator which emits a short rf pulse with a carrier frequency of 4300 megahertz. The rf pulse is sent to the transmitter antenna. A portion of the transmitted pulse is detected and fed to the tracking section to start the timing circuits. This signal is called the "time reference" (T_0) pulse.

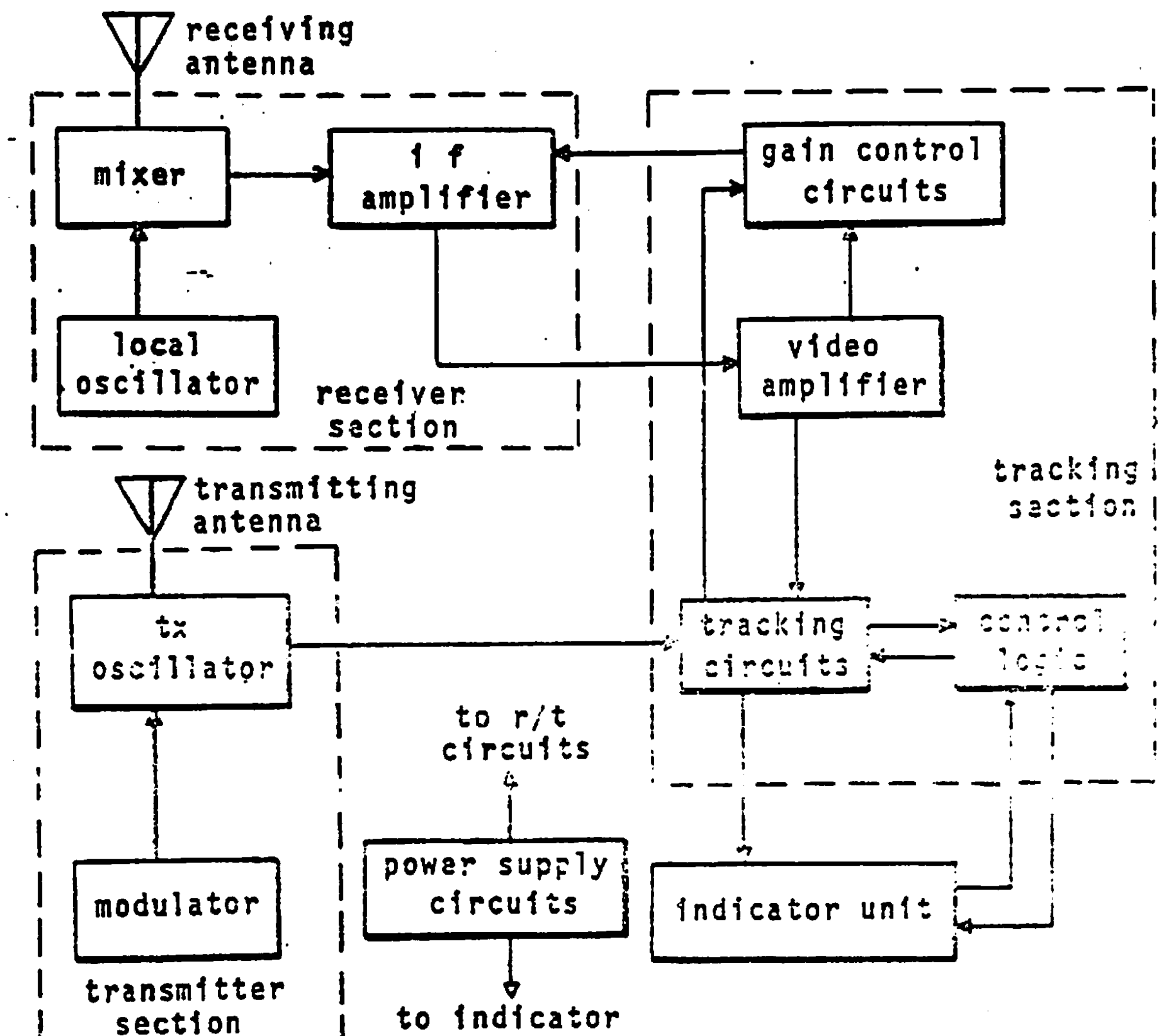


FIGURE 2-5. RADAR ALTIMETER -- SIMPLIFIED BLOCK DIAGRAM

Receiver

When the rf pulse emitted by the transmitter antenna reaches the ground, it is partially reflected back to the aircraft where it is picked up by the receiving antenna and applied to the receiver. The receiver section, shown in Figure 2-5, consists of a balanced mixer, a local oscillator, and an i-f amplifier circuit. The type of receiver used in the altimeter is a "zero i-f" superhetrodyne receiver. In this type system, the receiver's local oscillator is operated at approximately the same frequency as the transmitter. Both the transmitter and the local oscillator are of the same basic design and use planar triodes. They inherently have identical frequency tracking characteristics and require no Automatic Frequency Control (AFC) circuit. The local oscillator signal is mixed with the received rf signal (echo pulse) in the balanced mixer. The output is bipolar (ac) pulses similar to video. The i-f amplifier is then, for all practical purposes, a video amplifier. The bipolar pulses are amplified and detected to a unipolar (single direction or dc) pulse. The detected video pulses are then sent to the tracking circuits where they are used to measure altitude.

Tracker

The function of the tracker is to produce a current equal to a reference by overlapping edges of the transmitted pulse and the echo pulse. The tracker consists of a video amplifier, tracking circuits, Automatic Gain Control (AGC) circuits, and logic circuits. The tracking circuit is a "leading edge" tracker. The circuit is designed so that the current (I_e) is proportional to the overlap of the edges of the tracking gate and the return as shown in Figure 2-6. Further, the circuit is designed to sense this current and use it to position the tracking gate on the

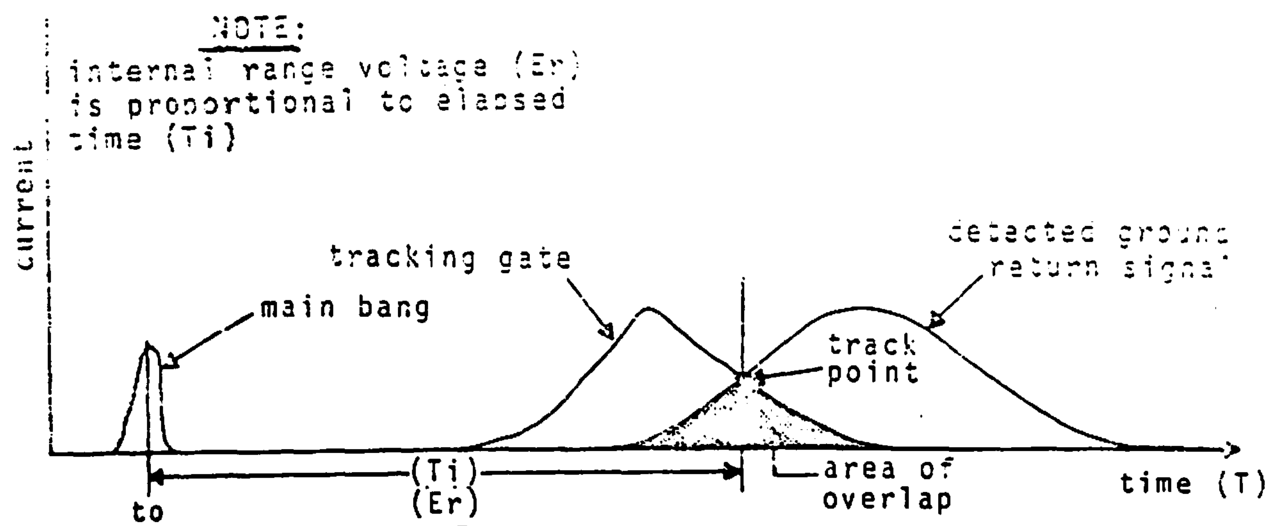


FIGURE 2-6. TRACKING GATE — SIGNAL RELATIONSHIP

leading edge of the ground return pulse as it moves back and forth with changes in altitude. Thus, it is the function of the tracking loop to move the track gate at whatever rate necessary to keep it in coincidence with the leading edge of the ground return pulse. The elapsed time (T_1) from the transmission of a pulse until it returns from the ground is thus measured and presented as an analog voltage (E_a). A radar range measurement is sent to the indicator to drive the altitude pointer.

Modes of Operation

The altimeter has four different modes of operation:

- o Tracking Mode
- o Search Mode
- o Confidence Check Mode (self-test)
- o Failure Monitor Mode

The different modes of operation pertain mainly to the tracker circuits.

Tracking Mode - When the altimeter is operating in the tracking mode, its output represents the actual distance downward to the nearest object as a function of time. Figure 2-7 shows the circuits that are operational in the tracking mode. As stated earlier, each time the transmitter is triggered by the modulator, a time reference pulse (T_o) is applied to the T_o amplifier. The T_o pulse triggers a sawtooth generator which supplies a calibrated linear voltage ramp. The instantaneous voltage of this precision ramp is directly proportional to the elapsed time since T_o. The sawtooth voltage is fed to a voltage comparator circuit where it is compared to the internal range voltage (E_r).

At the instant of time when the sawtooth voltage equals E_r, the comparator applies a short pulse to the gate generator circuit. At this time, the tracking gate is generated by the tracking gate generator circuit.

When the rf pulse emitted by the transmitter at T_o reaches the ground, it is partially reflected back to the receiving antenna. The antenna feeds the echo pulse into the receiver where it is amplified, detected, and presented to the Track Gate Amplifier (TGA) as a video signal.

The TGA functions as an "and" gate. The time coincidence between the video signal and the tracking gate allows the video signal to pass as a current pulse to the rate intergrator. Since E_r determines the sawtooth level at the time of comparison, and since the time position of the video signal is a direct function of the distance to the ground, coincidence occurs when the sawtooth reaches

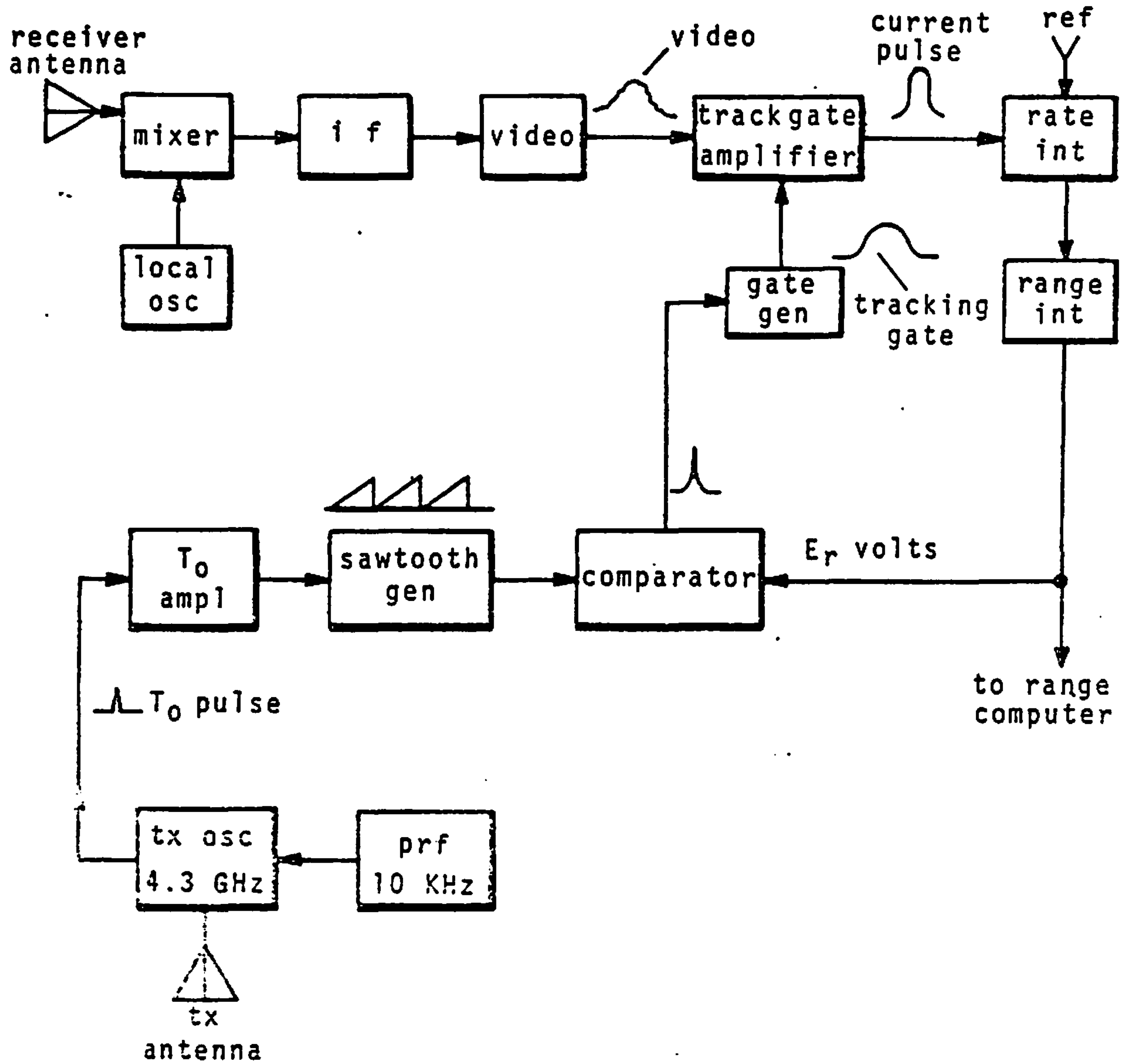


FIGURE 2-7. TRACK MODE — SIMPLIFIED BLOCK DIAGRAM

the coincidence voltage (E_c) level, as shown in Figure 2-8.

Coincidence of the video and tracking gate at the TGA means that E_r is proportional to range. It is important to note that since this is a "leading-edge" tracker, E_r represents the shortest range to ground. The TGA produces a current into the rate integrator proportional to the overlapped area of the video pulse and gate. A reference input equal to the average overlap current holds the gate positioned on the leading edge of the video pulse. The leading edge is composed of the earliest ground return.

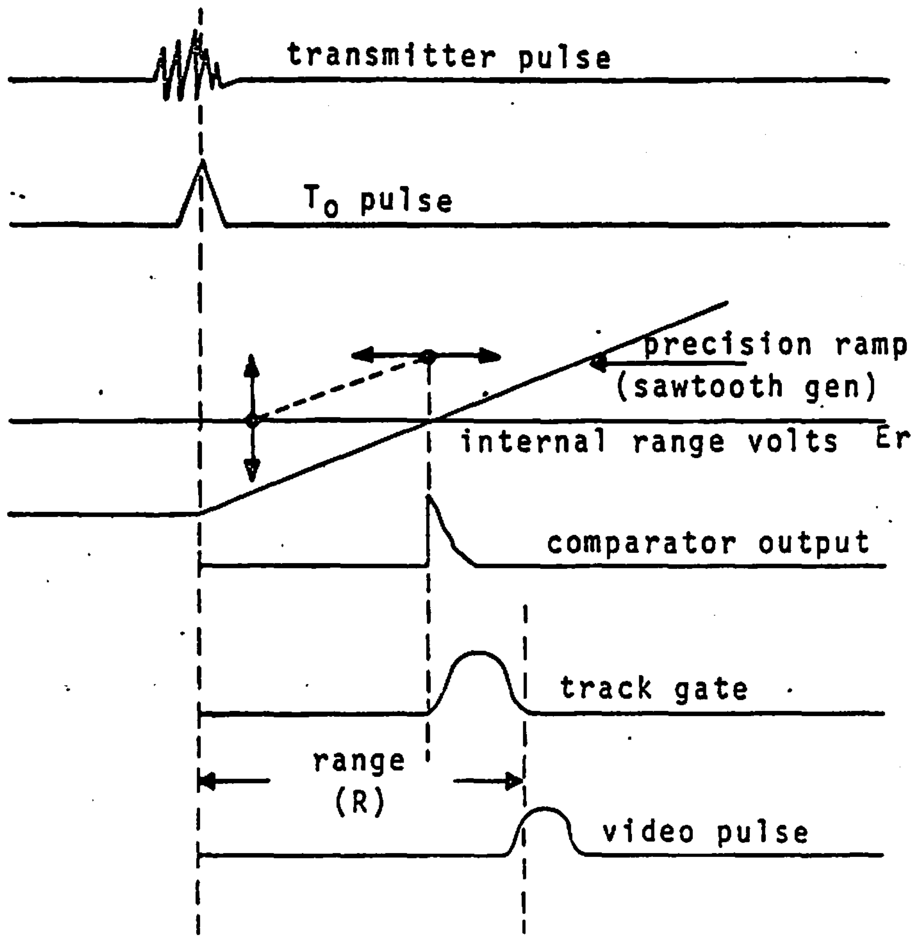


FIGURE 2-8. TRACKING GATE GENERATION

Search Mode - If the return signal falls below a minimum level required to maintain the track point current to the rate integrator, the video pulse and the tracking gate separate and the the tracker "loses track." Tracking can only be re-established when the tracking gate overlaps acceptable video. Therefore, it is necessary to provide a means to establish this condition, which is the purpose of the search mode. The tracking gate is caused to "sweep" from minimum to maximum range until it intercepts a video signal. If the video signal is acceptable, the tracking gate locks on it; thus tracking is re-established.

The simplified block diagram, Figure 2-9, shows the circuits that are operational in the search mode.

The following additional blocks are used in this mode of operation:

- o Tracking AGC (TAGC) Gate Amplifier

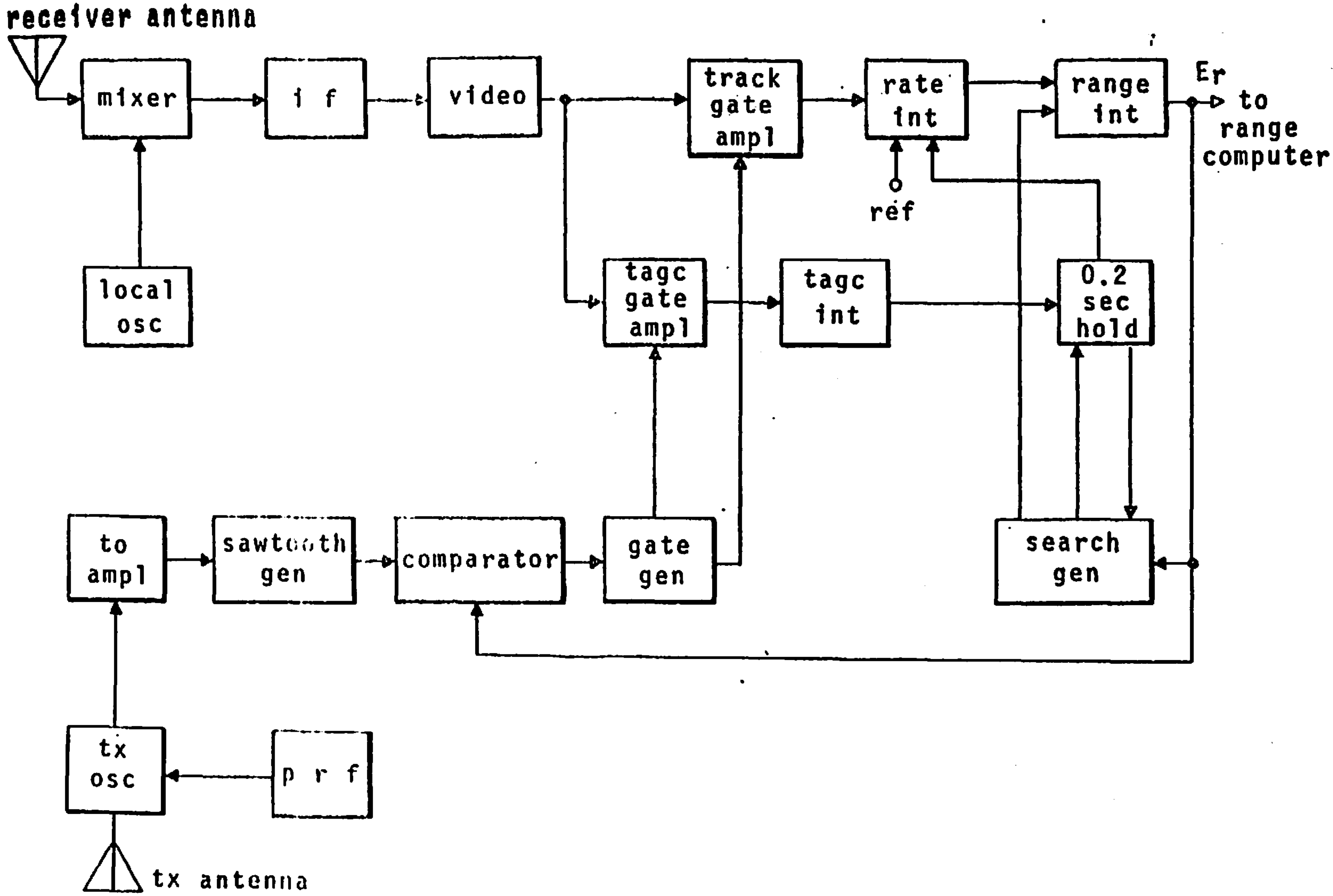


FIGURE 2-9. SEARCH MODE -- SIMPLIFIED BLOCK DIAGRAM

- o TAGC Integrator
- o 0.2-Second Hold Circuit
- o Search Generator

The TAGC gate amplifier is similar to the tracking gate amplifier. It is an "and" gate for the video signal and the TAGC gate from the gate generator. Its output is proportional to the video and gate overlap. An important difference is that the TAGC gate is about twice as wide as the tracking gate. The leading edge of the two gates are in time coincidence. The output of the TAGC gate amplifier is applied to the TAGC intergrator.

The TAGC integrator integrates negative pulses from the TAGC gate amplifier. Amplitude of the pulses is equal to the TAGC gate and the video overlap current. These pulses occur at the Pulse-Repetition Frequency (PRF) rate. Pulses are integrated in the TAGC integrator to establish a threshold voltage (E_O). For normally good video, about 20 pulses per second maintain E_o at the tracking threshold. If a loss of video occurs, the integrator maintains E_o for approximately 2 milliseconds (ms).

The 0.2-second hold circuit provides an additional delay time of 200 ms to the decay time of the TAGC integrator. Together they allow a short-term loss of video (200 ms) to pass without losing track. If a signal fade lasts for more than 200 ms, the 0.2-second hold trips and turns the search generator on. A signal is also sent to the rate integrator to clamp the output to zero. This action prevents E_r from changing during the 0.2-second hold period.

The search generator is a controlled multivibrator governed by the 0.2-second hold circuit. If the 0.2-second hold trips, the search generator supplies input current to the range integrator which causes E_r to sweep back and forth from minimum range to maximum range. This search cycle continues until it is stopped by the 0.2-second hold circuit. In addition, the search generator provides an inhibit signal to the 0.2-second hold during inbound sweeps. The 0.2-second hold can only stop the search generator on outbound sweeps. The diagram shown in Figure 2-10 clarifies the sequence of events during search and acquisition of the video target. The horizontal numbers at the top of the illustration depict the following sequence of events:

- o Loss of video. The current drive from the track gate amplifier to the rate integrator vanishes. The TAGC integrator loses its drive and discharges. The 0.2-second hold initiates its hold cycle.

- o After 200 milliseconds, the search generator is activated which drives E_r to minimum altitude.

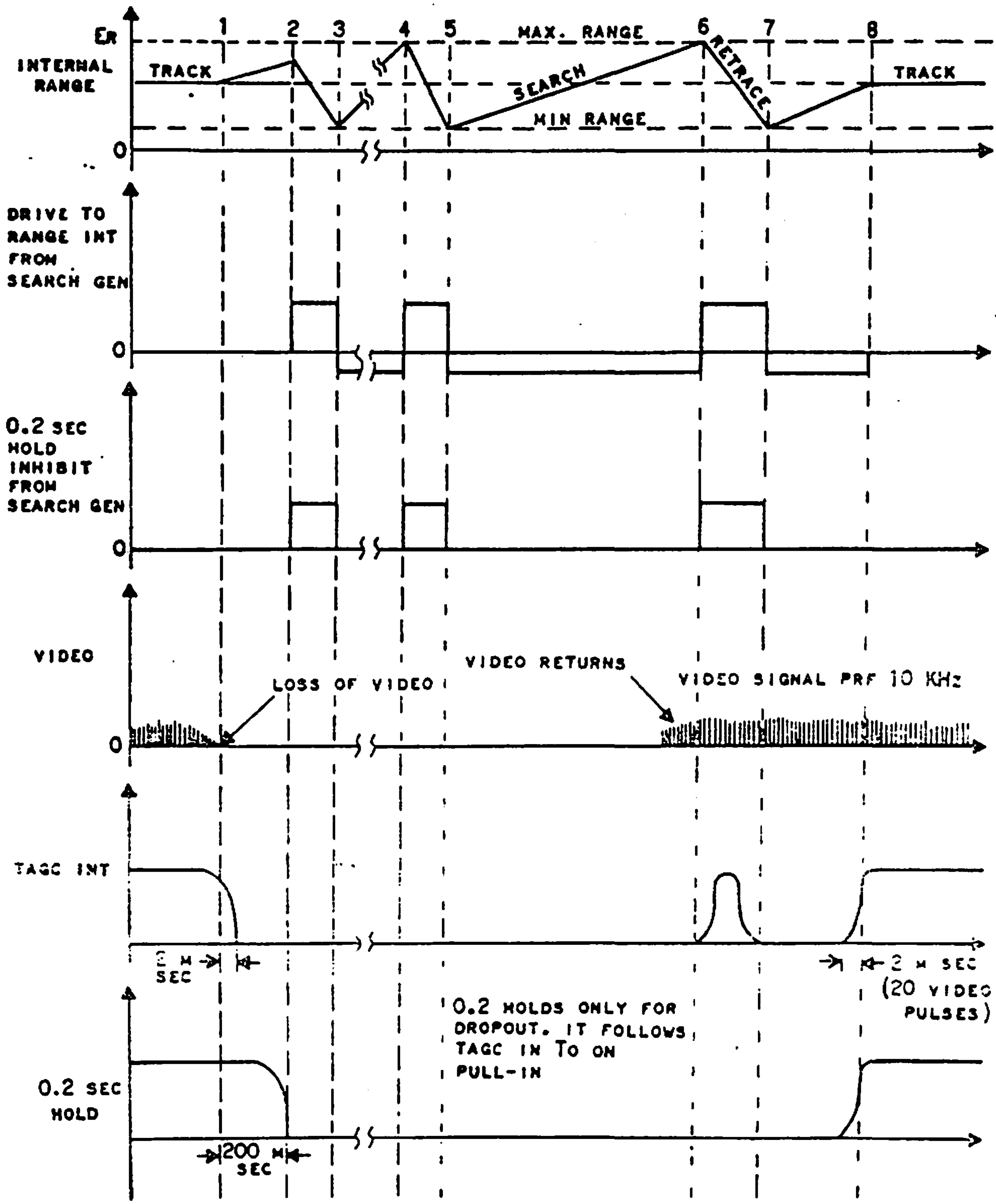


FIGURE 2-10. SEARCH AND ACQUISITION WAVEFORMS

- The drive to the range integrator is reversed and E_r starts toward 2500 feet.
- The upper limit reverses E_r and inhibits the 0.2-second hold.
- Repeat of search cycle.
- Reversal of E_r occurs. Video is regained. The gates overlap but because the 0.2-second hold is inhibited, lock-on does not occur.
- E_r reverses again and sweeps outward.
- Assuming that the actual range remains constant during the loss of video, the gates overlap again. The TAGC integrator charges which resets the 0.2-second hold, thus stopping the search generator. The tracking loop is now closed and the altimeter is again in the tracking mode. The above search cycle occurs at approximately a 3-hertz rate.

Confidence Check (CC) Mode - The CC (self-test) mode is designed and built into the system to provide a quick comprehensive test of system operation. The block diagram in Figure 2-11 shows the circuits used in this mode of operation. Two additional circuits used are the CC-PRF generator and the CC level set.

When the CC command is given the transmitter is disabled, a flag alarm signal is given, the sawtooth level is set, and the CC-PRF generator is enabled. The CC-PRF generator applies the T_o pulse to the T_o amplifier and a confidence test pulse to the mixer. The test pulse acts as a gate to open the mixer. This action allows a portion of the Local Oscillator (LO) energy through the i-f amplifier where it is amplified, detected, and presented as a synthetic video target to the video amplifier.

Disabling the transmitter eliminates the normal video from the TGA input, and the tracker goes to the search mode. The synthetic video is passed to the gate amplifiers and the loop searches until E_r corresponds to the test altitude. The tracker then acquires and tracks the synthetic video pulse. The amplitude of the pulse fed to the mixer is set to a minimum level for reliable track. Thus, the CC mode also checks the system sensitivity.

Failure Monitor Mode/Built-In Test Equipment (BITE) - The failure monitor mode is used to detect failures in the track and search loops and in the output circuits. A simplified block diagram in Figure 2-12 shows the circuits used in the failure mode.

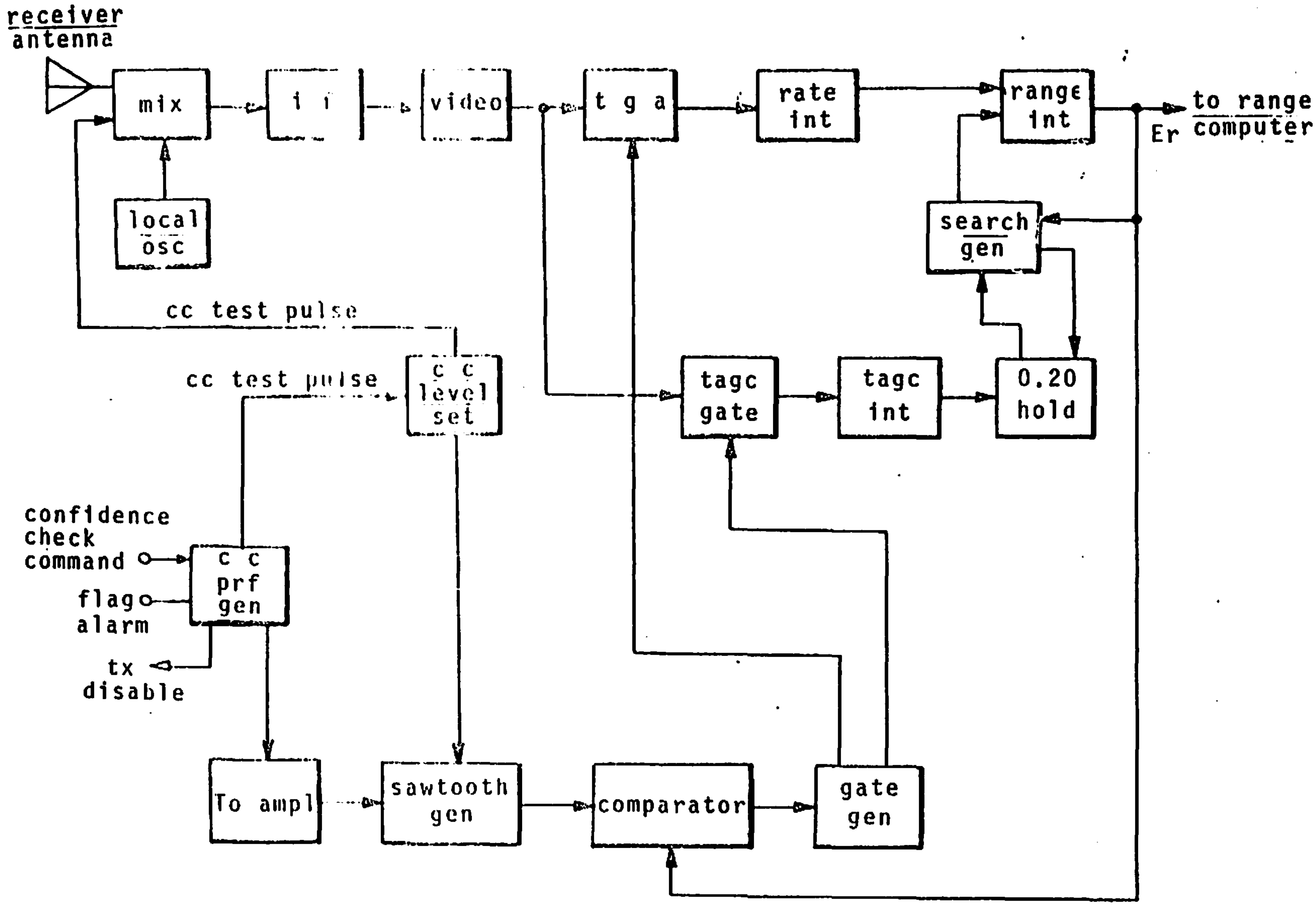


FIGURE 2-11. CONFIDENCE CHECK MODE — SIMPLIFIED BLOCK DIAGRAM

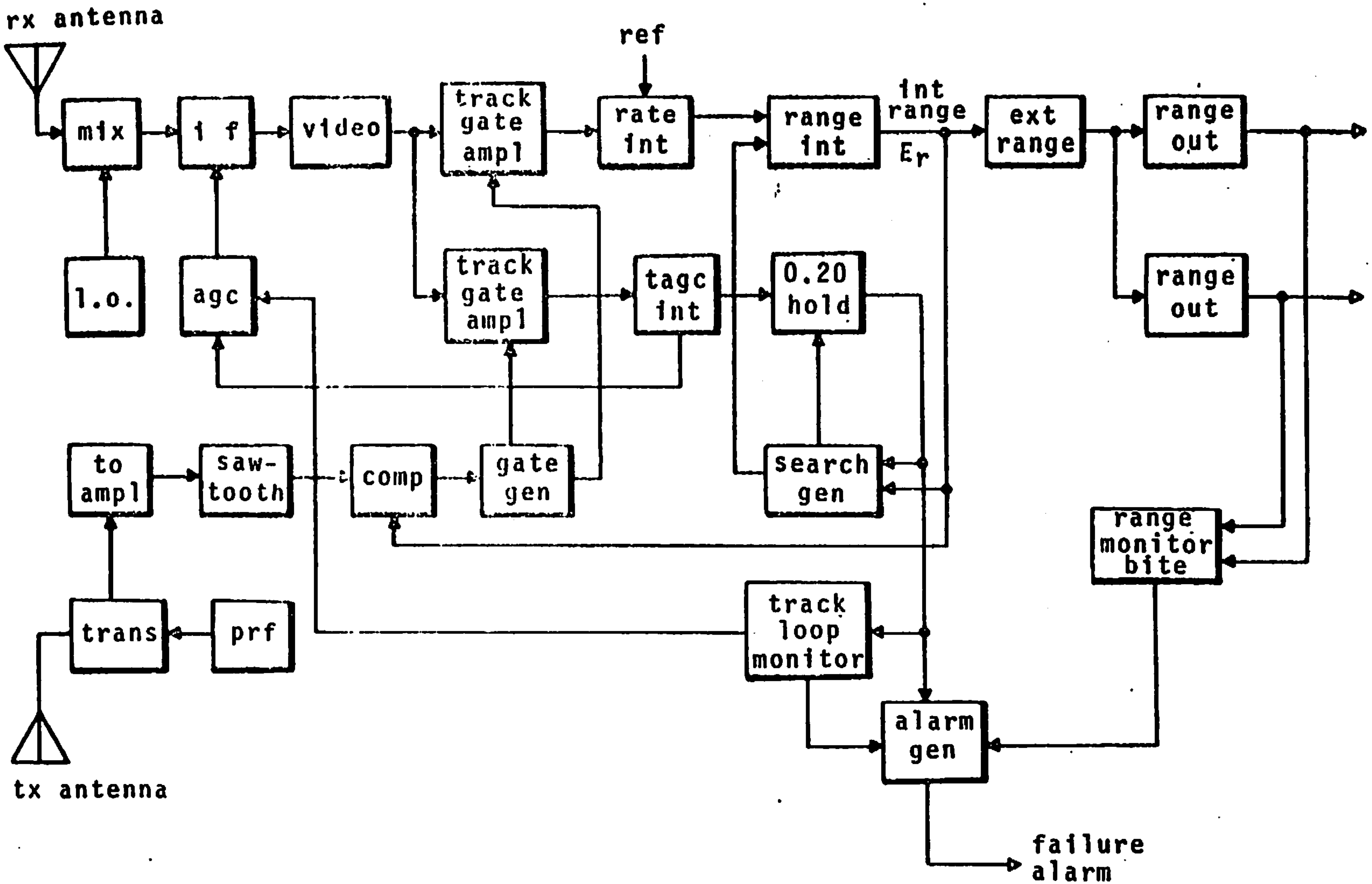


FIGURE 2-12. FAILURE MONITOR MODE — SIMPLIFIED BLOCK DIAGRAM

In the normal track condition, the two range output signals and the internal range voltage (E_r) are continually compared to verify the accuracy of the output circuits. If either of these differs from the other by a voltage equivalent to 15 feet or more, a signal is sent to the alarm generator and a failure indication is provided. The track loop monitor is activated at any time that the tracking loop cannot maintain a reliable track condition for more than 0.2 second (time period of the 0.2-second hold) and above a 2500-foot indicator reading.

When the 0.2-second hold deenergizes, the system goes into the search mode. The track loop monitor decreases the AGC signal to the i-f for approximately 0.5 second. This action allows the rf leakage signal from the transmitting antenna to the receiving antenna to pass through the i-f amplifier. If the track loop is functional, this appears as a video signal. The track loop regains a track condition and no failure alarm is provided. If a fault has occured in the track loop or if the transmitter power or receiver sensitivity has reached an unsafe level, a track condition is not obtained within the 0.5 second. During this time, an rf leakage signal is provided and a signal is sent from the track loop monitor to the alarm generator which provides a failure indication to the indicator. If no failure is indicated at the end of this 0.5-second period, the system returns to the search mode and attempts to reacquire the ground return for up to 4.5 seconds. In the absence of a ground return, the failure monitor continues to gate-out the AGC signal for 0.5 second at 5-second intervals, thus continuously checking the tracking loop operation.

Antennas

Two identical antennas are used with the radar altimeter system. Each antenna consists of a flared horn and a coax-to-waveguide adapter. The units are sealed by flush, non-reflecting radomes. They have a db beamwidth of 40 degrees in the "H" plane and 50 degrees in the "E" plane. Antenna locations are shown in Figure 2-1.

Indicator

The following inputs are supplied to the indicator, shown in Figure 2-18, from the radio-transmitter unit:

- DC Altitude
- Flag Alarm
- Mask Drive
- Indicator Reference Voltage

The mask drive voltage is used to operate the mask when the system is turned off or when operating above 2500 feet. The dc altitude voltage is compared in

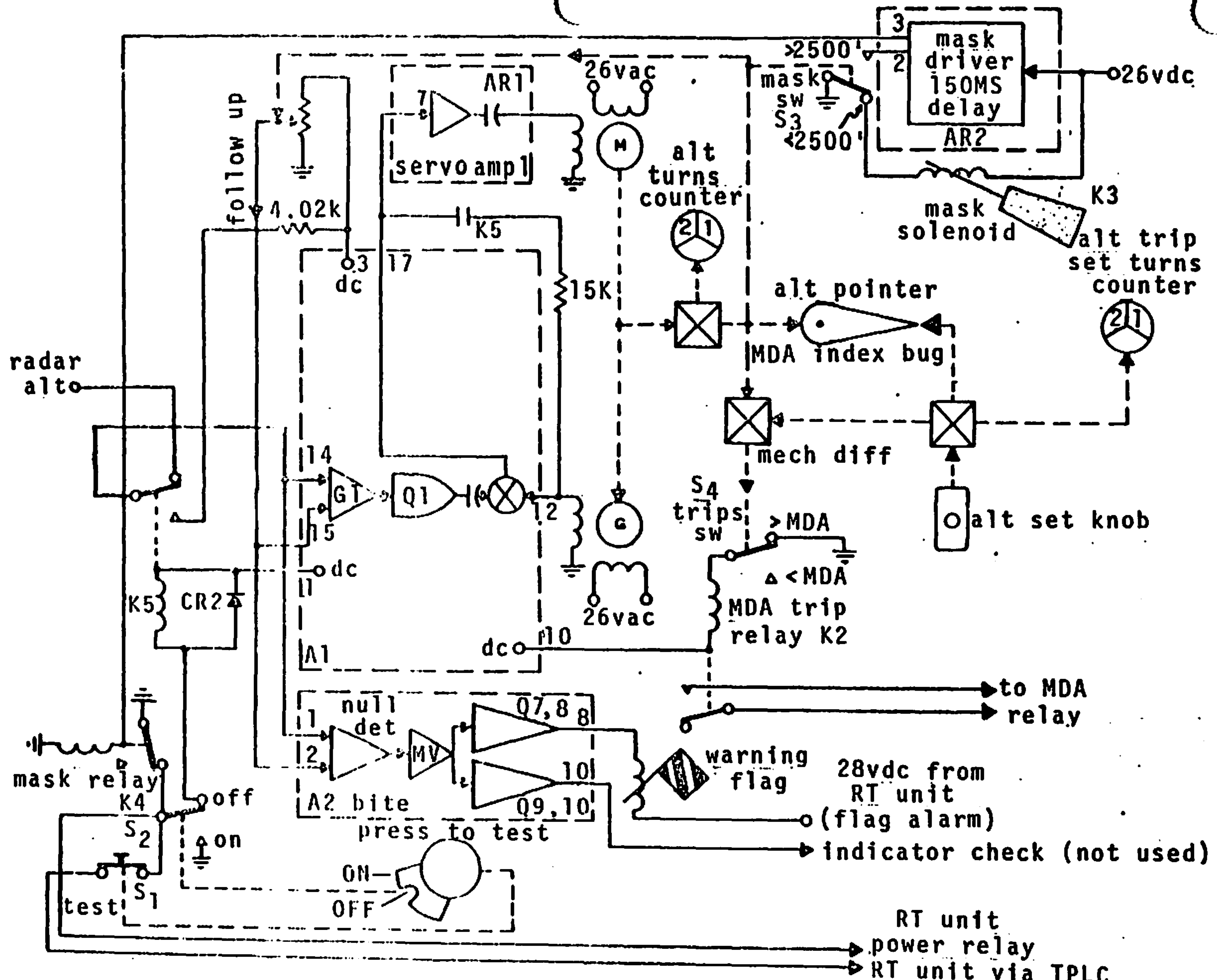


FIGURE 2-13. ALTIMETER INDICATOR BLOCK DIAGRAM

the servo amplifier with the indicator reference voltage. The resultant dc voltage polarity determines the phase of the ac used to drive the servo motor. As the motor drives the altitude pointer and turns counter, it also positions the follow-up potentiometer to null the servo loop. When the servo loop is at a null, the proper altitude reading is displayed on the indicator. The flag alarm signal activates the indicator failure circuit. Whenever the altimeter is not functioning properly for any reason (such as unreliable altitude signal, loss of power, etc.), the null monitor (BITE) initiates the flag alarm.

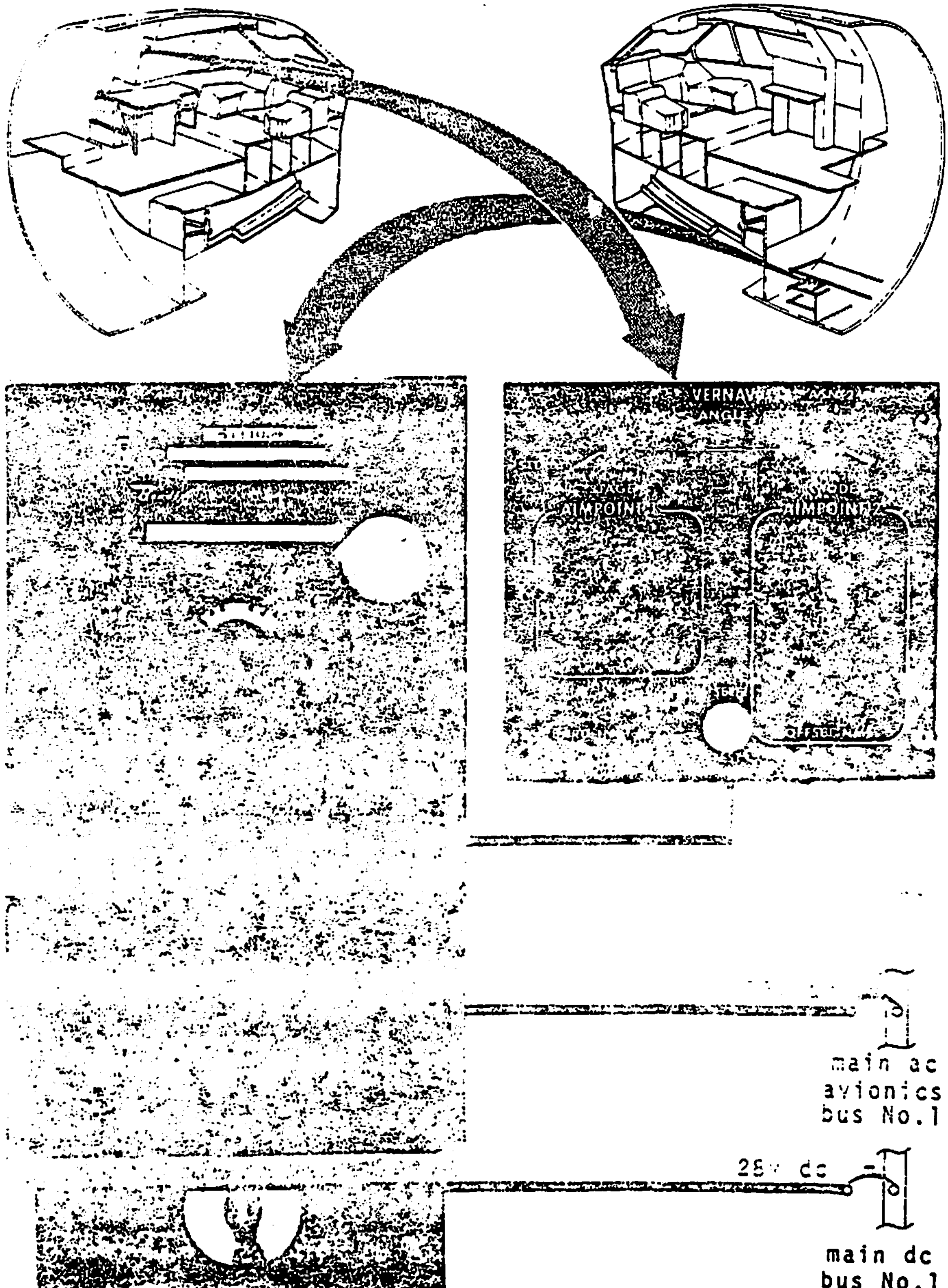
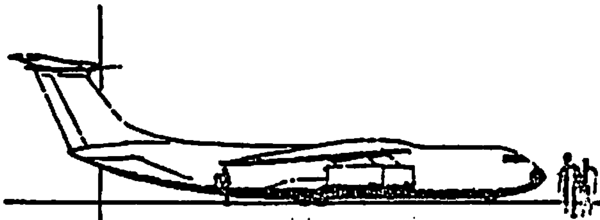


FIGURE 3-1. VER NAV COMPUTER AND CONTROL PANEL



VERTICAL NAVIGATION (VER NAV)

The Vertical Navigation (VER NAV) subsystem is used to command a sink or ascent rate and path deviation to arrive at a predetermined position and altitude.

Two separate vertical flight profiles can be established by the VER NAV computer. Each path terminates in altitude hold at the predetermined aimpoint altitude. Two stages are set to determine the flight paths: Stage 1 may be used for ascent or descent while Stage 2 can be used for descent only.

VER NAV command and deviation signals are transmitted to the autopilot coupler, elevator computer, and flight director computers for steering and display.

SYSTEM OPERATION

The VER NAV system is remotely controlled from the navigator's instrument panel as shown in Figure 3-1. The VER NAV system is placed in operation when the stage selector switch is placed in "STDBY" (standby) and either the ASN-24 or ASN-35 controls are set for the proper aimpoint data.

VER NAV Control Unit Data

Aimpoint 1 Altitude	zero to 40,000 feet
Aimpoint 2 Altitude	zero to 10,000 feet
Aimpoint 1 Angle	15 to -15 degrees in 0.5 degree increments
Aimpoint 2 Angle	-0.1 to -5.0 degrees in 0.1 degree increments

Continued

VER NAV Control Unit Data (Continued)

Barometric Correction	28 to 31 inches mercury
Distance Offset	zero to 20 nautical miles
Angle Meter	± 15 degrees

Setup Procedure

NAV Computer Control Settings	
Step 1	
A. ASN-24 Mode	For a one-stage letdown, only one D-position need be used, but if the full two-stage letdown is planned, two D-positions should be set up in advance. For example, the latitude and longitude or Aimpoint 1 may be set into D1, and the latitude and longitude of Aimpoint 2 may be set into D2. When D1 is selected, the ASN-24 calculates the distance from the aircraft's present position set into the selected D position. This distance-to-go calculation is made once per second, and the signal is provided to the VER NAV computer. Selection of D2 and VER NAV Stage 2 causes similar distance-to-go to Aimpoint 2 to be provided.
B. ASN-35 Mode	The airdrop of 100-nautical mile mode of the ASN-35 must be selected if the ASN-35 mode of the VER NAV computer is to be used. In this mode, two stages are available for separate setting of distance to go. Only after the first stage has been selected as an operating mode can the

Continued

Setup Procedure (Continued)

	second ASN-35 stage be set up for the next VER NAV stage. This VER NAV Stage 2 leg must be set up on the ASN-35's control panel just as before.
Step 2	Select ASN-24 or ASN-35.
Step 3	Set aimpoint altitude.
Step 4	Set VER NAV angle.
Step 5	Set in barometric correction.
Step 6	If Stage 2 is to be used, set in offset distance from touchdown point to Aimpoint 2.
Step 7	Press self test. When this test is completed (assuming no malfunction indication), the VER NAV system is now ready for operation. (Test switch on the computer at normal.)
Step 8	Select Stage 1 or Stage 2.* This action automatically places the flight director in the VER NAV mode (pitch axis), and the displacement pointer on the ADI displays deviation. A VER NAV mode light on the navigator's selector panels also illuminate.
Step 9	Autopilot may be engaged after Step 8.

* Stage 1 and Stage 2 may be set up together or individually and while operating on the other stage.

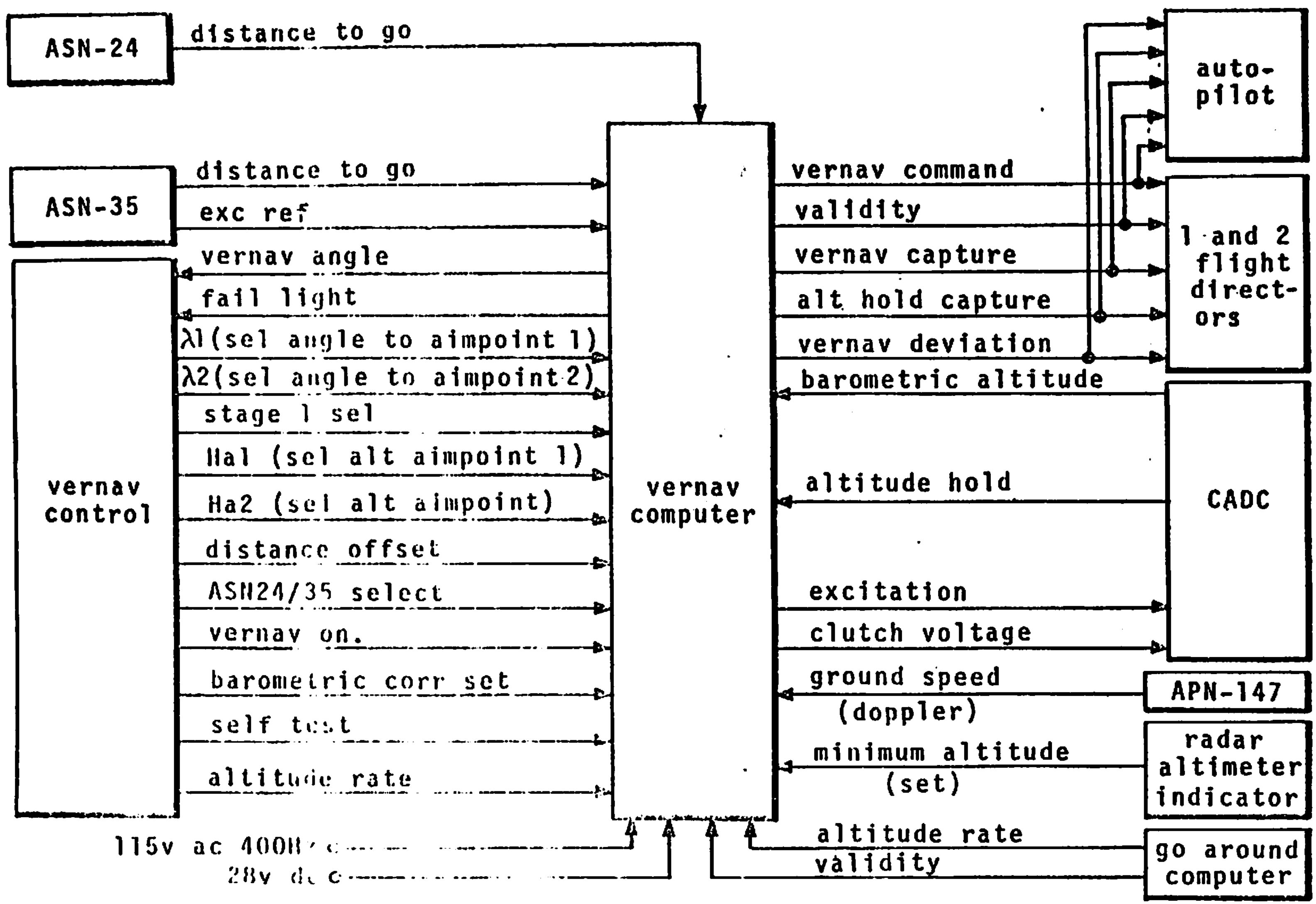


FIGURE 3-2. VERNAV DATA FLOW

Figure 3-2 illustrates the data flow from control to computer.

SPECIFICATIONS

<u>Inputs</u>	
Power	115-volt ac. 400 hertz, single phase and 28-volt dc.
Distance-To-Go	ASN-24: Digital Nav computer; updated once each second: 360 degrees equals 100 nautical miles, analog of X coordinate ASN-35: Doppler computer; updated every 0.1 nautical mile: 36 degrees equals 10 nautical miles, analog of X coordinate.
Radar Altimeter	Switch control at preset minimum altitude. When activated, places VER NAV in altitude hold.
Rotation/Go-Around	Complemented altitude rate (IVV) validity, 28-volts dc.
<u>Outputs</u>	
Altitude Deviation	Aircraft above path (plus) Aircraft below path (minus) Scale: 0.15 millivolts per foot between zero and 500 feet 0.0019 millivolts per foot between 500 and [illegible] feet
VER NAV Command	Aircraft above path, zero-degree phase (steer down) Aircraft below path, 180-degree phase (steer up) Scale: 10 millivolts per foot, altitude error
Malfunction	VER NAV mode light
Validity	28-volt dc valid; zero volts, non-valid

Continued

SPECIFICATIONS (Continued)

VER NAV Capture	28-volt dc, at 150 milliamperes initiation of maneuver to the new vertical path from level light
Altitude Hold Capture	28-volt dc, at 150 milliamperes initiation of maneuver to horizontal path from VER NAV angle
VER NAV Mode Light	VER NAV provides ground for light

THEORY OF OPERATION

The VER NAV system is an electromechanical analog computer that uses ac signals. One of the main inputs to the VER NAV is the Distange-To-Go (DTG) signal from an ASN-24 digital navigation computer or the ASN-35 doppler navigation unit.

Difference in position of the navigation computer synchro and a VER NAV servo control transformer causes an error voltage to exist at the servo amplifier. The signal is amplified and causes the servomotor to turn the control transformer, through a 1200:1 reduction gear, to null. A rate generator and autotransformer are also turned by the motor through a 100:1 reduction gear. A portion of the rate generator is fed back to the amplifier input to prevent hunting, as shown in Figure 3-3

One output of the servo system warns of a servo malfunction. The null detector senses absence of a null at the servo summing junction and trips the VER NAV malfunction circuit.

An output from the DTG autotransformer is supplied to the VER NAV angle select autotransformers for summation with corrected barometric altitude. This signal (h_c) is command altitude. Barometric altitude, augmented altitude, and aimpoint altitude voltages are summed to provide an altitude of aircraft above aimpoint signal. This signal is applied to the VER NAV angle generator, X tan transformers, and a demodulator. Aircraft altitude and DTG signals are summed in the tan function autotransformers. Excitation of the tan function autotransformers is proportional to DTG. This excitation is multiplied by the tangent of the VER NAV angle (λ) in the autotransformers, and the output is the DTG (X) times the tangent of λ plus h_c. The summation result is command altitude (h_c) minus aircraft altitude above aimpoint which is an error signal to be zeroed at VER NAV capture (VC).

The altitude error (h_e), which is the algebraic summation of h_a and $h_p \pm h_b$, is connected, through the stage selector switch, to a buffer amplifier and demodulator. This now demodulated signal (h_e) is amplified and applied to a scaling/summing amplifier and a sine function amplifier. The scaling/summing amplifier output goes through contacts of the validity and altitude hold relays to the VER NAV deviation driver. From the sine function amplifier, which develops a non-linear voltage, the signal (h_e) is applied to the altitude error amplifier along with vertical speed error (h_e). This composite signal is then applied to the VER NAV capture trip and steering command modulator circuits. The non-linear output of the sine function amplifier adjusts the capture maneuver signals for VER NAV capture anticipation. At high aircraft speed, capture is made earlier than at slow speeds.

Groundspeed from the doppler system is applied with corrective bias to the commanded vertical speed potentiometers. Commanded vertical speed, which is the speed (V) times the tangent of λ, goes through switching to a demodulator. The demodulated signal is mixed with an actual vertical (IVV) signal, which produces vertical speed error. This error goes through a 30-second switched washout filter. Thirty seconds after VER NAV capture, the washout filter removes the average value of vertical speed error from the steering signals. The altitude rate error signal and altitude error signal are added in a summation amplifier. One of the two outputs of this summation amplifier goes to VER NAV trip logic; the other goes to VER NAV steering command modulator through the VER NAV capture logic relay contacts. These two signals produce a new error voltage that commands the flight down the VER NAV path. The error is still a function of DTG and altitude rate.

As the aircraft approaches the aimpoint, a signal composed of IVV (from R/GA) and aircraft altitude above aimpoint is switched into the steering command signal chain, by logic, e.g. AHC to provide a smooth flare to level flight. The IVV signal from the R/GA computer is demodulated and applied to the commanded vertical speed amplifier and the flare smoothing amplifier.

The summation of stage 1 or stage 2 altitude set and barometric altitude correction, minus aircraft altitude voltages, provides the altitude of the aircraft above aimpoint. This signal is amplified and demodulated. One output of the demodulator goes through AHC relay contacts to the VER NAV deviation driver. Another output goes through a sine amplifier to the altitude rate amplifier. The third demodulated voltage goes to clutched altitude trip logic, which operates

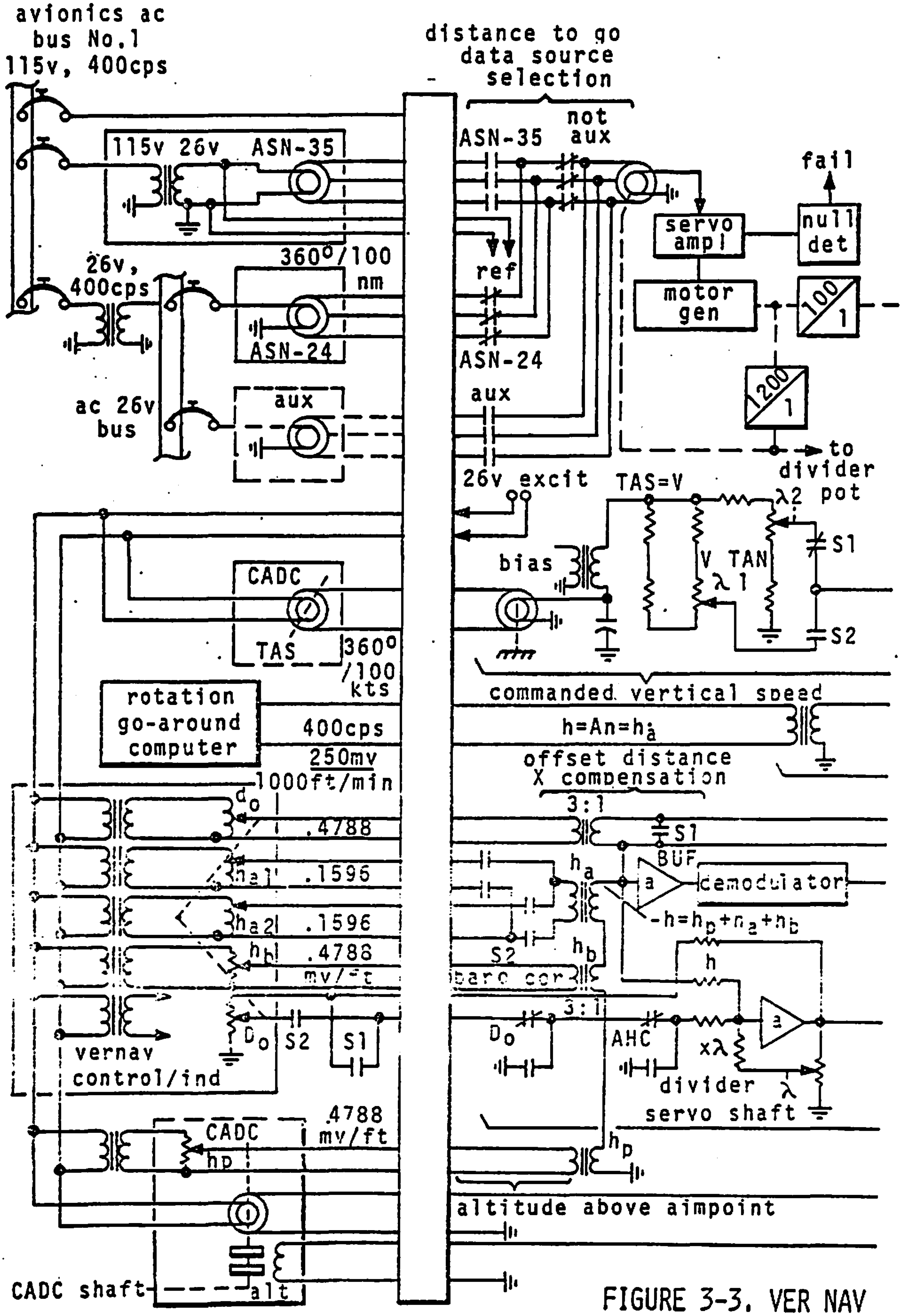
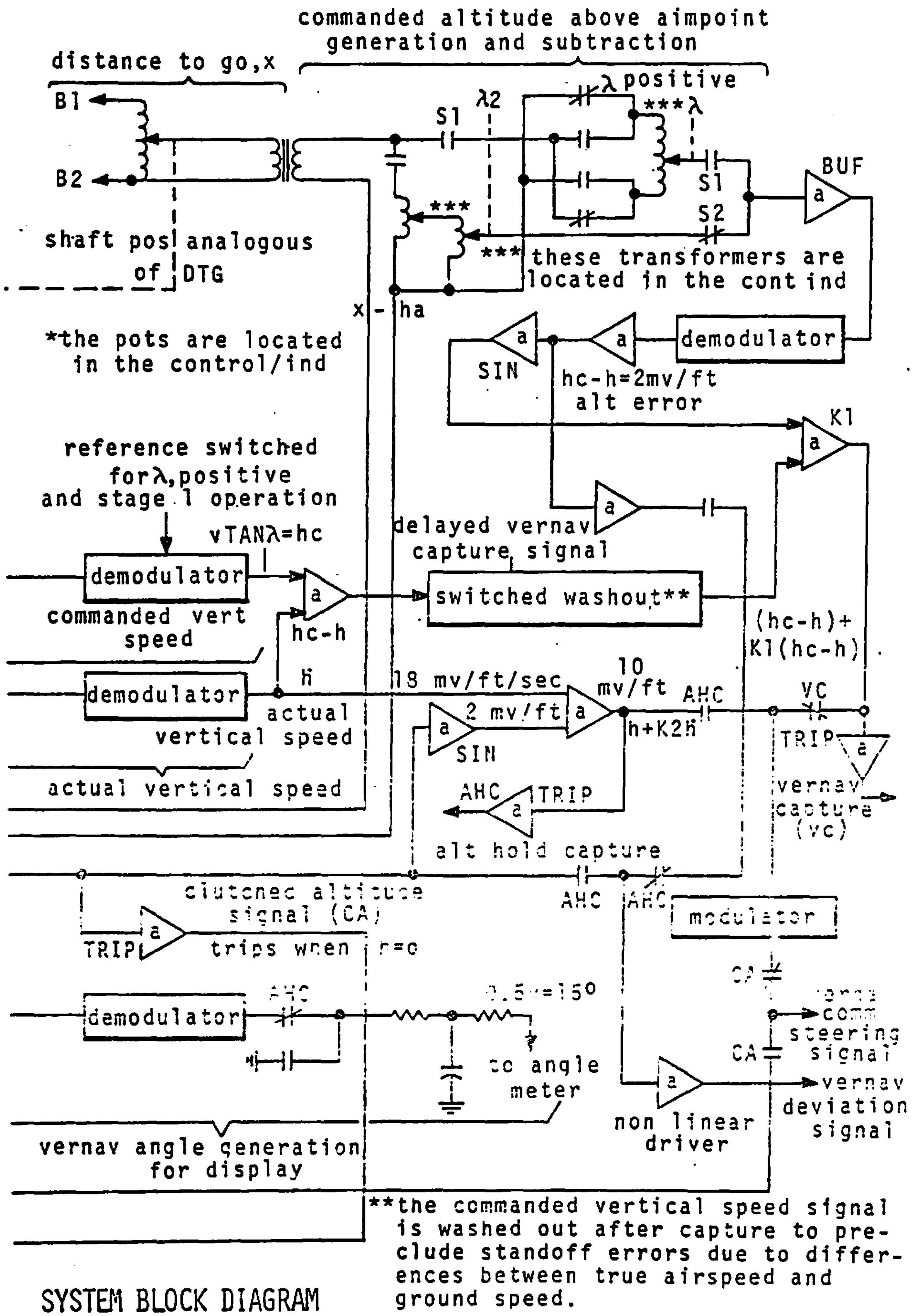


FIGURE 3-3. VER NAV



SYSTEM BLOCK DIAGRAM

when aimpoint is reached as (h_e) equals aimpoint altitude. Clutched altitude, through logic, keeps the aircraft on a horizontal flight path until commanded otherwise.

Another use of the summed altitude voltage is in the VER NAV angle generator. The summed altitude and distance-offset wiper voltages are inputs to the VER NAV angle generator. Part of the output is fed back to the input of the generator. This feedback is controlled by the 1200:1 gear system in the DTG servo system. The X servo shaft multiplies X in the feedback path. Division of altitude is made in the amplifier. (Proper degenerative feedback causes a decrease through an amplifier.) The VER NAV capture voltage (VC) is used to bring the horizontal pointer on the ADI's into view through the flight director and to unlock the autopilot altitude hold. VER NAV deviation is displayed by the ADI's, and VER NAV command signals are routed to the autopilot and flight directors for vertical steering. When the aircraft has completed a stage 1 maneuver, command steering is controlled by the clutched altitude voltage from the CADC until stage 2 capture. VER NAV angle voltage is used to display the aircraft's progress on the VER NAV path on the angle meter located on the control unit.

The following relationship applies during stage 2 function:

$$\lambda = \arctan h/X \text{ (zero distance offset)},$$

where

λ = VER NAV angle,

h = altitude

X = DTG.

If D_o is not zero,

$$\lambda = \arctan h/X - D_o,$$

where

D_o = distance from a_o to target.

The amplifier is nulled when,

$$X\lambda = h.$$

When D_o is not zero,

$$X\lambda - D_o = h.$$

The D_o input is grounded when the D_o potentiometer is operated to the zero position to prevent instability. The VER NAV angle generator ac output is demodulated, filtered, scaled, and then sent to the angle meter on the control unit where 0.5 volt represents 15 degrees. The meter is grounded by AHC and is kept grounded after capture.

SELF TEST (BITE)

Position 1 on the test selector switch (computer) and the stage to be tested (control panel) are selected. Predetermined aimpoint, VER NAV angle, and distance offset should be preset. The self-test button (control panel) is pressed and released to begin the test.

A light in the test button illuminates and the computer goes into an automatic test sequence. The sequence checks performance of a complete sample problem and a time history of response of the VER NAV system to predetermined characteristics. Any discrepancy in time of computations is sensed by BITE. This condition initiates a "no-go" status and illuminates the fail light (control panel), which stops the test sequence. The fail light stays "ON." If there are no failures, the test is completed, and the test button light is extinguished.

Position 1 tests the system computation and logic switching. Positions 2 through 9 check the functions of the control panel.

A specific test is run for each stage. To completely test the two stages, a dual run of the test sequence must be made. BITE consists of the following:

- A reference source for inputs
- Relay switches to enter the simulated inputs into the proper computational channels
- A computational time counter
- A comparator which compares the actual time of VC, AHC, and AH against pre-calculated ideal VC, AHC, and AH times.

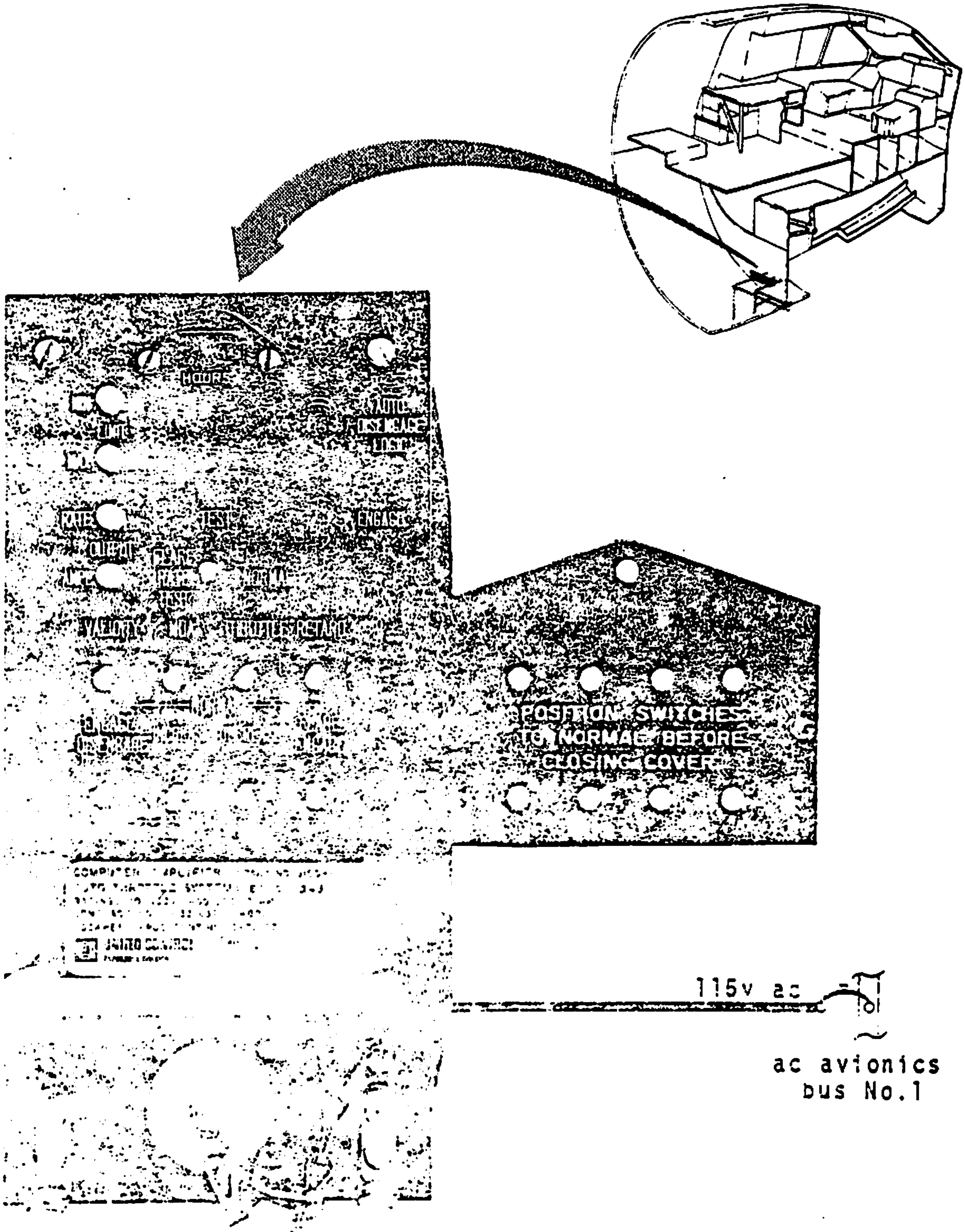


FIGURE 4-1. COMPUTER/AMPLIFIER

AUTOMATIC THROTTLE SYSTEM (ATS)

The Automatic Throttle System (ATS) is used to maintain a constant indicated airspeed. It also provides an automatic throttle retard rate when the system is used in an AWLS approach. The system consists of four major components:

- Computer/Amplifier
- Servomotor
- Clutch Pack
- Speed Trim Assembly

COMPONENTS

Computer/Amplifier

The computer/amplifier, shown in Figure 4-1, is the most complex of the four components. It receives input and validity signals from the CADC No. 1 and 2, the TPLC, and the flare computer. It also receives feedback signals from the ATS. An error, representing the algebraic sum of these signals, is developed in the computer and routed to the servomotor assembly as a command signal.

Automatic disengage logic in the computer/amplifier uses validity signals that are developed by the ATS as well as validity inputs from the CADC's, the TPLC, and the flare computer.

If, during certain phases of auto throttle operation, a validity signal goes invalid, it would normally cause the ATS to disengage, thereby assuring safe control of the aircraft. Another feature of this unit is that it has its own Built-In Test Equipment (BITE), which is initiated and monitored by means of switches and lights on the front of the computer and is used by the ground crew as an aid in troubleshooting.

Servomotor Assembly

The servomotor assembly, shown in Figure 4-2, receives its input command from the computer/amplifier which drives the servomotor to the commanded position. When the servomotor drives, it drives the clutch pack assembly through a gear train and also produces a feedback signal for the computer/amplifier.

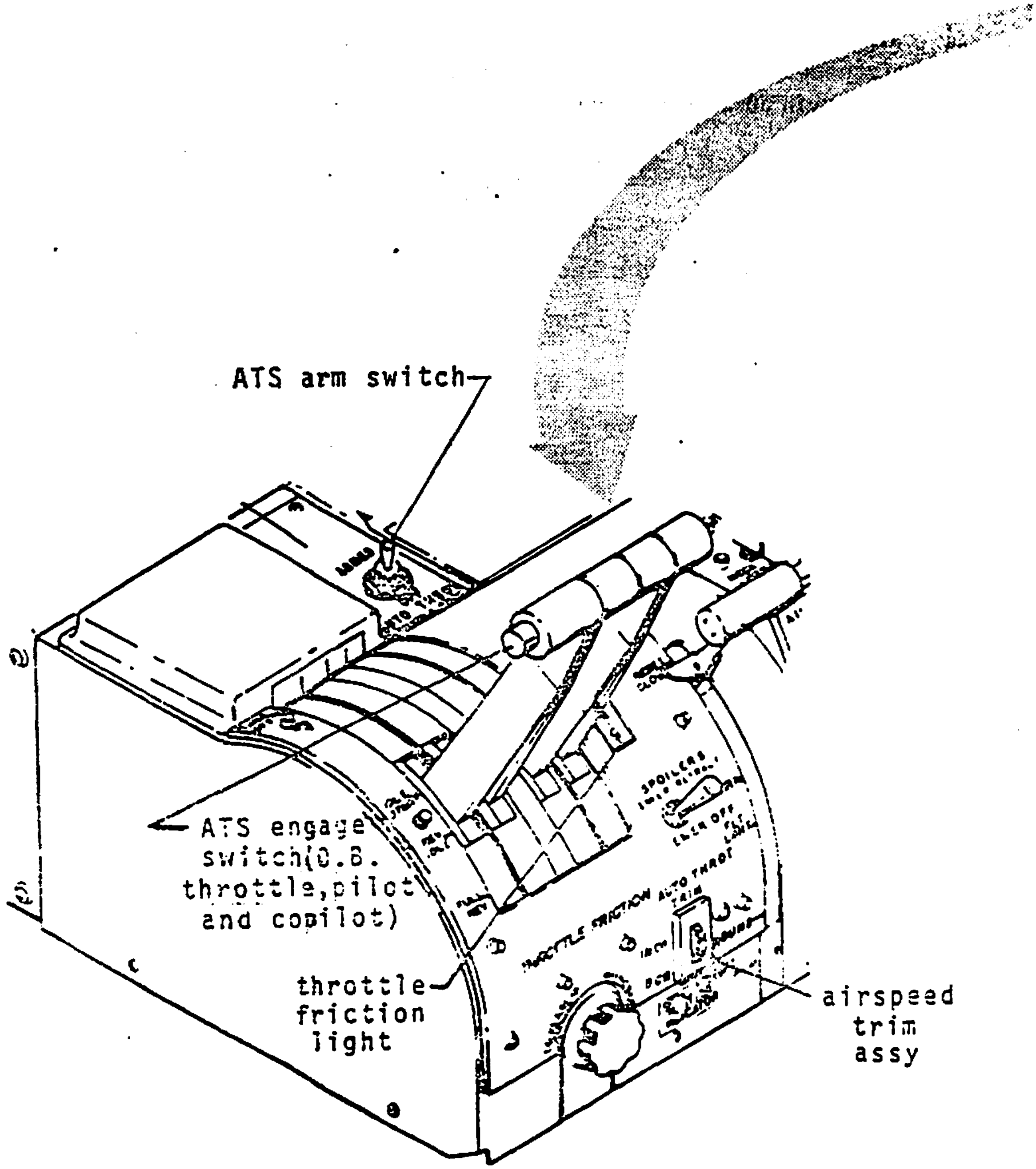
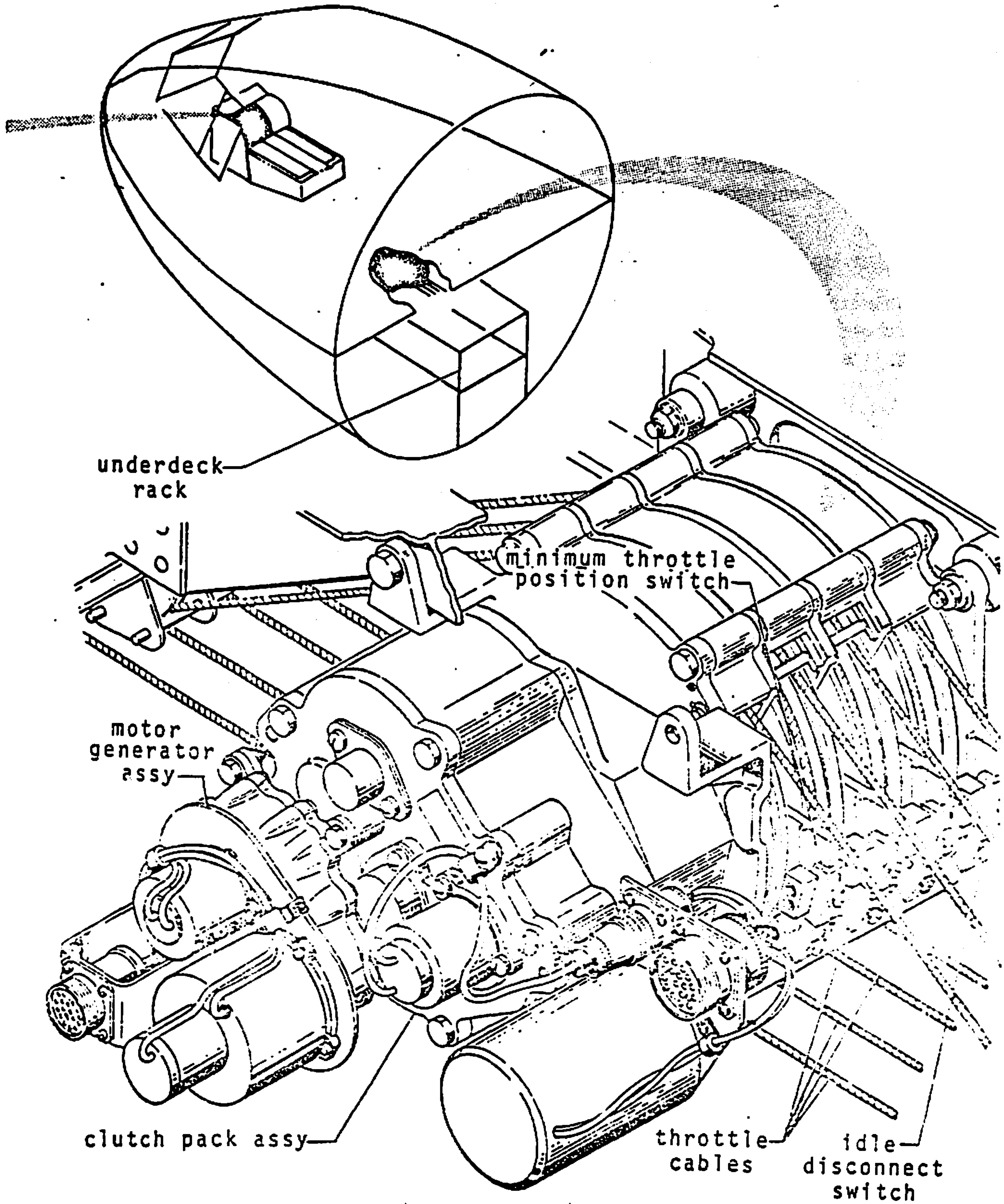


FIGURE 4-2. AUTOMATIC THROTTLE



SYSTEM COMPONENTS LOCATIONS

Clutch Pack Assembly

The clutch pack assembly, also shown in Figure 4-2, receives a mechanical input command from the servomotor. This action causes the throttles to be repositioned by cable linkage. It also changes the setting of the fuel control valves of the engines, thereby correcting for changes in indicated airspeed. The clutch pack assembly also has a position synchro which produces a feedback signal for the computer amplifier that represents throttle position.

Speed Trim Assembly

The speed trim assembly, also shown in Figure 4-2, allows the pilot to make low-authority speed trim corrections, which provides precise control of the aircraft's clutched airspeed.

Cabling

The cabling diagram, Figure 4-3, shows the manner in which all these components are connected.

SPECIFICATIONS

Speed Trim Authority	± 5 knots
Throttle Rates	
Manual/ATS Mode	1.7 degrees/second or 3.3 degrees/second if airspeed error is 3 knots or more for over 2 seconds
Autopilot/ATS mode	3.3 degrees/second
At FLARE (30-foot altitude)	2.5 degrees/second ± 0.7
Servomotor-to-power gearing ratio	15:1
Power gear-to-clutch pack	100:1
Clutch pack-to-throttle control driving ratio	1.8:1

Continued

Pounds of force required to move throttle with ATS engaged		
One throttle lever		13 to 16 pounds
All four throttle levers		52 to 64 pounds
Limit switch actuation from rig pin position	<u>Clutch Pack Quadrant</u>	<u>Throttle Handle Position</u>
Maximum limit	52.4 degrees ± 1	62 degrees ± 1
Minimum limit	3 degrees ± 1	33 degrees ± 1
Idle disconnect	- 6.1 degrees ± 1	29 degrees ± 1
Power supply output voltages		7 volts, dc - 7 volts, dc 10 volts, dc - 10 volts, dc 35 volts, dc 26 volts, ac 115 volts, ac 6 volts, ac

THEORY OF OPERATION

To select the ATS mode, the pilot positions the ATS ARM switch to the "ARM" position. The THROTTLE light on the fault identification panel illuminates to indicste that the ATS mode is armed but not engages. If the friction knob setting is too high, the ATS FRICT light, mounted near the throttle friction knob, illuminates. When the friction knob is repositioned to a lower setting, the system is in the armed position and the THROTTLE light remains illuminated. The airplane is then flown to the desired airspeed. The ATS may then be engaged

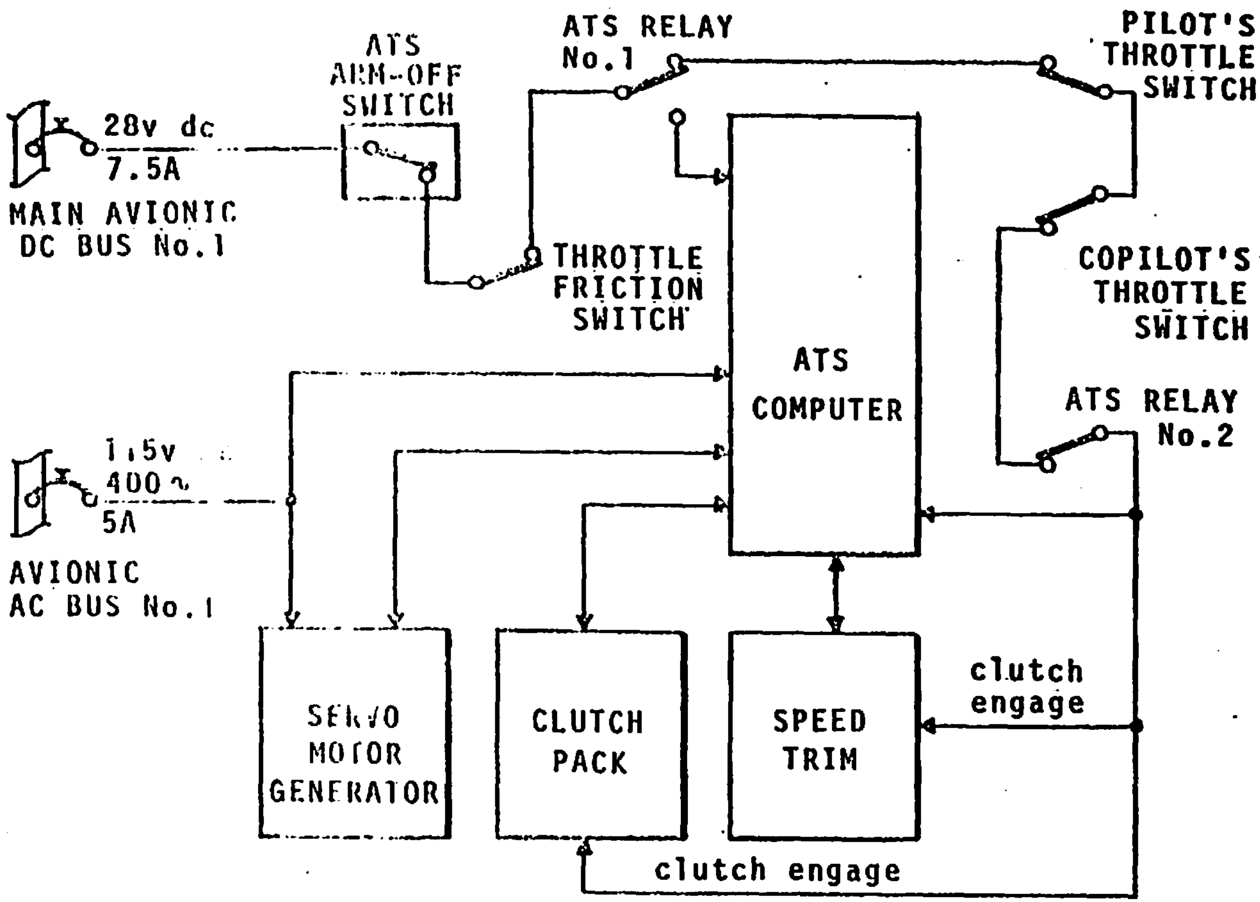


FIGURE 4-3. AUTOMATIC THROTTLE SYSTEM CABLING

by pressing the switch mounted in either the pilot's No. 1 throttle lever or the copilot's No. 4 throttle lever as shown in Figure 4-4. The pulse signal from the throttle switch causes the ratchet relay to close its switch contacts and energize the engage circuits in the CADC's, throttle clutch pack, speed trim unit, and the computer/amplifier.

The ATS mode can be disengaged by once more pressing either the pilot's or copilot's throttle lever switch. This pulse signal causes the ratchet relay to open its switch contacts to disconnect the engage circuitry. The THROTTLE light illuminates since the ATS mode is still armed by the ATS ARM switch. Re-engagement of the ATS by the throttle lever switches or disengagement of the ATS ARM switch puts the light out. The ATS mode can be directly disengaged by positioning the ATS ARM switch to the "OFF" position without pressing the throttle lever switch. This circuitry is designed to automatically advance the ratchet relay to the proper position so that the next time the arming switch is engaged, the ATS mode is armed but not engaged.

The ATS uses the airspeed hold signal from the CADC as the airspeed error source, as shown in Figure 4-5. This signal is generated in the CADC by a clutch-operated synchro connected to the indicated airspeed shaft. When the synchro is declutched, a spring-loaded cam maintains the synchro at its electrical null position, thus presenting a zero error signal to the ATS computer prior to engagement. When the pilot engages the ATS system, a signal is sent to the CADC to engage the synchro clutch to the indicated airspeed shaft. Thereafter, until disengaged, the synchro output signal is proportional to aircraft speed changes from the clutched-in speed setting. Within the computer/amplifier, the airspeed error signal from CADC No. 1 is always switched into the computer circuitry as long as a validity signal from the CADC No. 1 self-monitoring circuit is received. Should the CADC No. 1 malfunction, the loss of the monitoring signal automatically switches the CADC No. 2 airspeed error signal into the computer circuitry. Should CADC No. 2 also malfunction, the loss of its monitoring signal then automatically disengages the ATS system and illuminates the THROTTLE light on the AWLS display panel. The THROTTLE light is always illuminated when the ATS is armed but not engaged.

Airspeed error signal from the CADC is sent to the CADC demodulator in the computer/amplifier. The demodulator is a phase-sensitive, dual-bridge circuit, as shown in Figure 4-6.

This circuit is designed to generate a positive dc output voltage when it receives an input that is in-phase with the reference voltage. When the input is 180 degrees out-of-phase with the reference voltage, a negative dc output is developed.

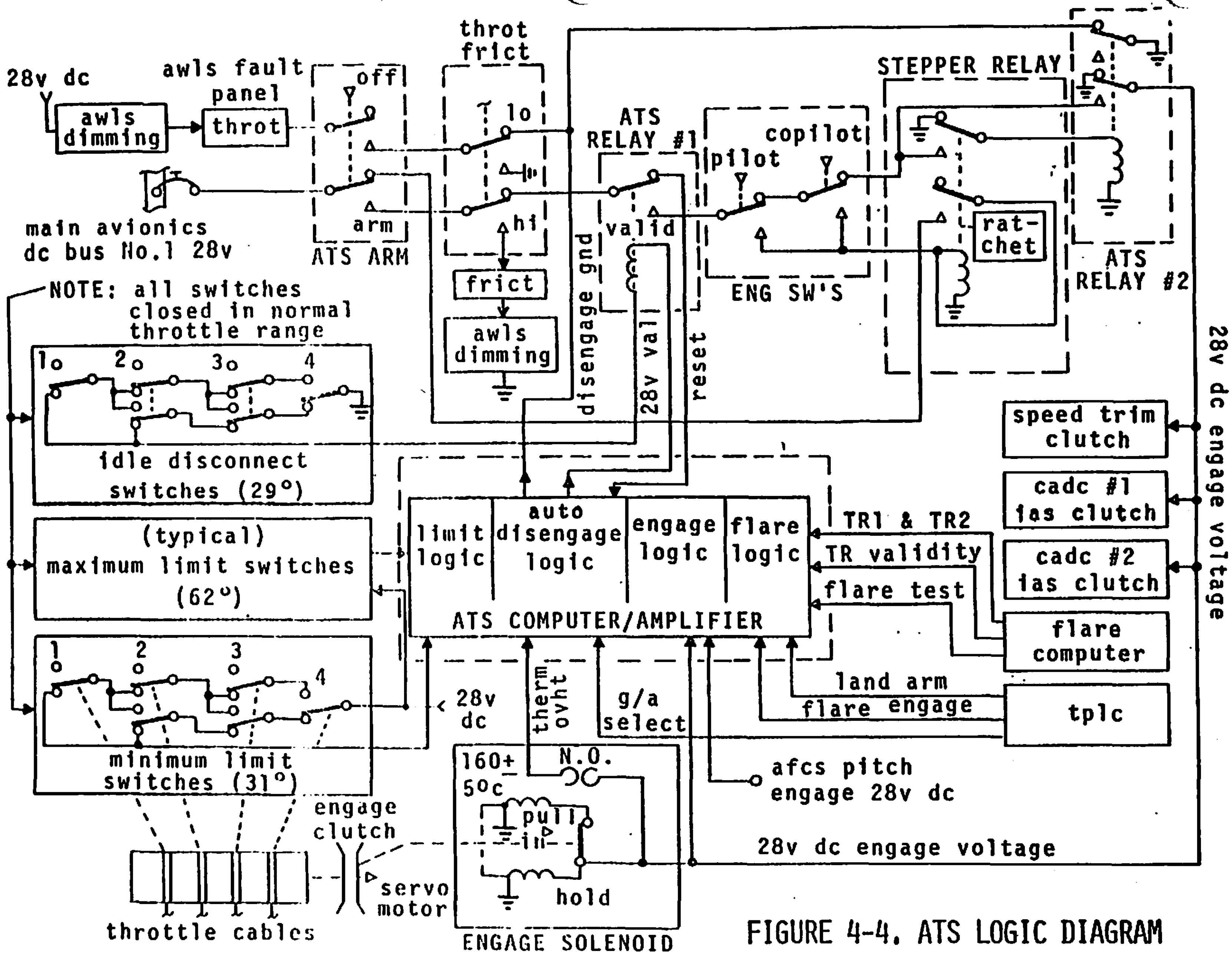


FIGURE 4-4. ATS LOGIC DIAGRAM

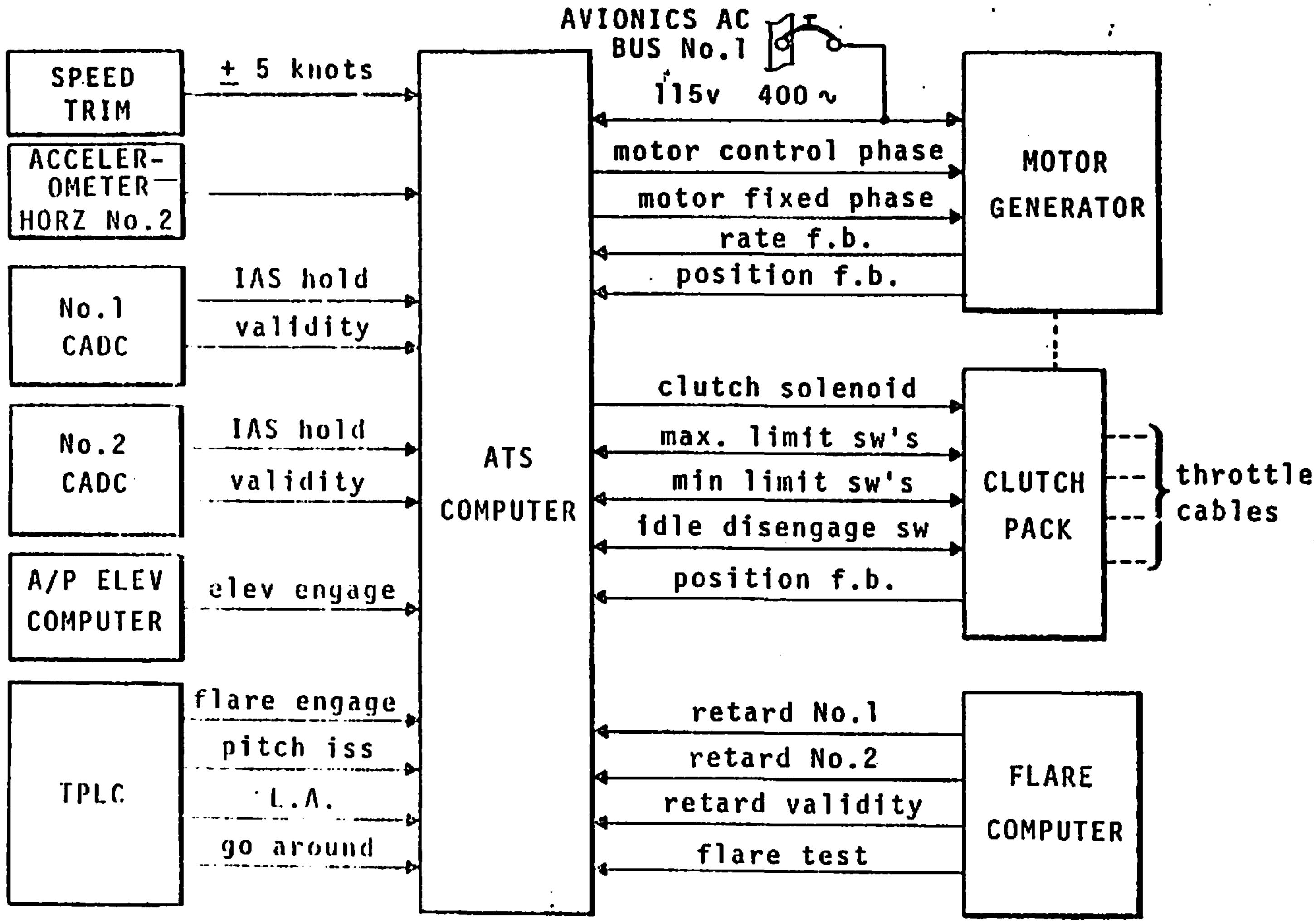


FIGURE 4-5. AUTOMATIC THROTTLE SYSTEM DATA FLOW

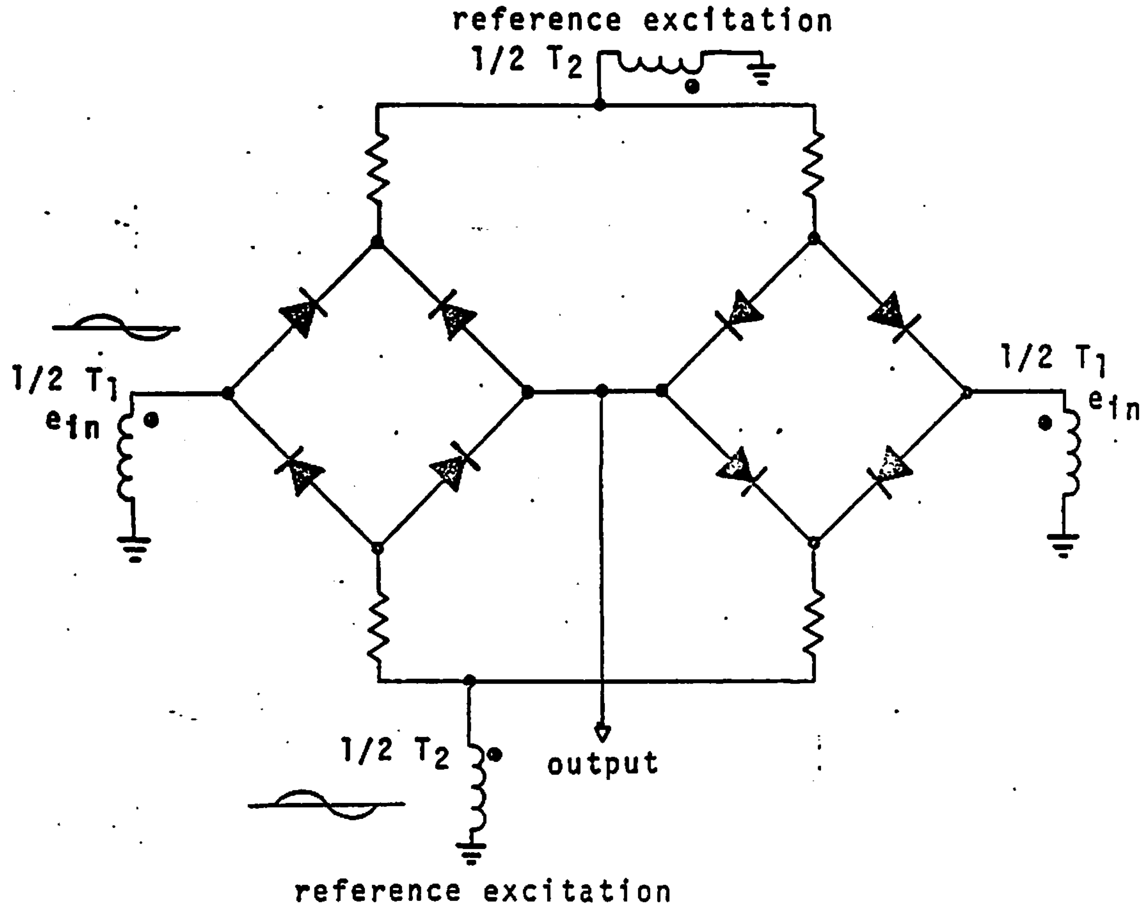


FIGURE 4-6. TYPICAL DUAL BRIDGE DEMODULATOR

The CADC demodulator output is sent to four places:

- A summing junction at the input of the integrator.
- A summing junction at the gust filter and limiter
- A lead circuit.
- The input of a dual-rate sensing circuit.

The speed trim signal permits the pilot to command a change in aircraft speed, in 1-knot increments up to 5 knots of the original engaged speed without disconnecting the ATS system. The speed trim control, as shown in Figure 4-7, consists of a position transducer controlled by a thumb wheel which is detented for each 1-knot increment. It is spring-loaded to automatically reset itself to the null position when the ATS system is disengaged. This 1- to 5-knot error signal is then added to the airspeed error signal at two summing junctions: one at the input to the electronic integrator, and the other at the input to the gust

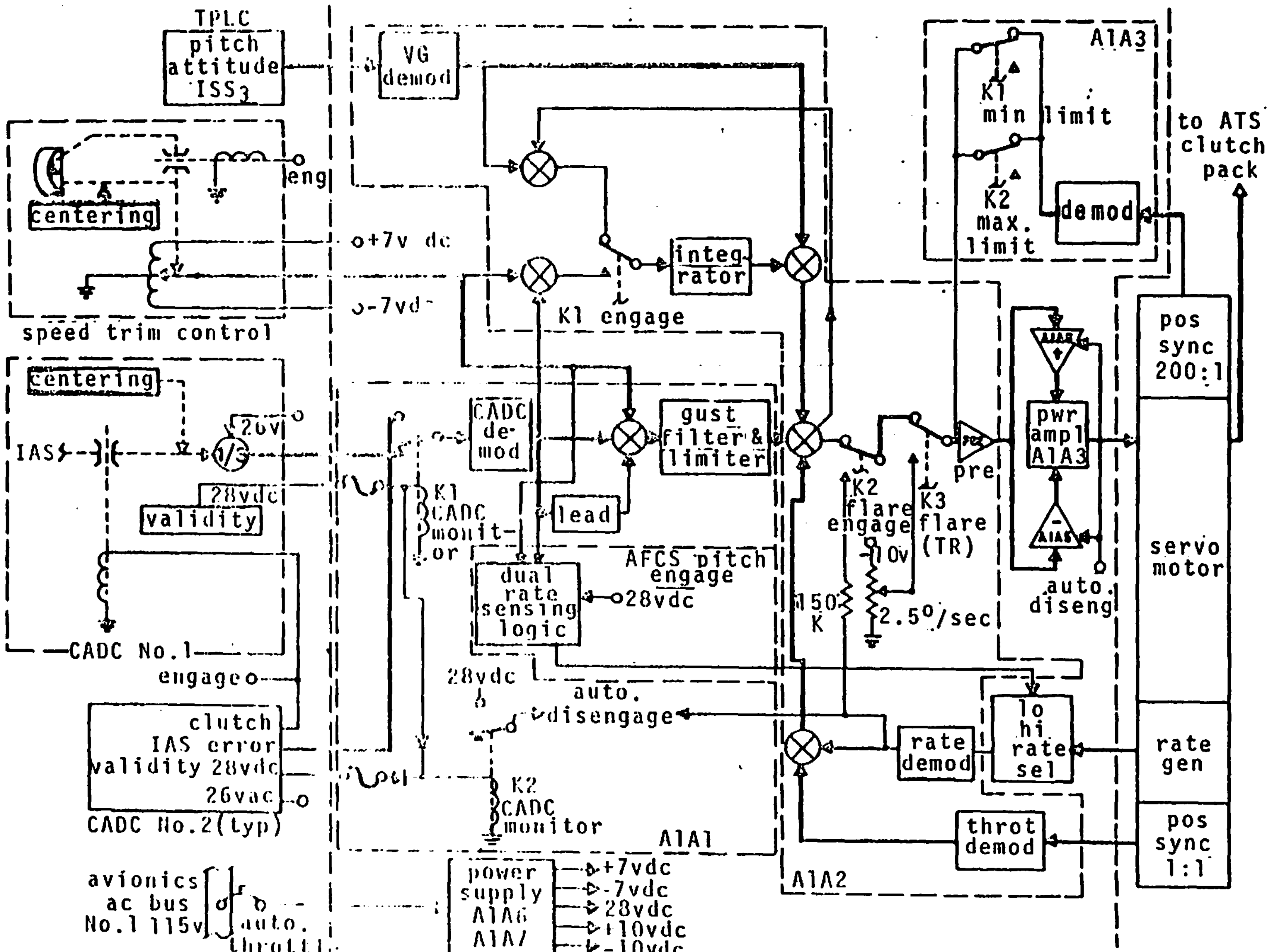


FIGURE 4-7. ATS BLOCK DIAGRAM

filter and limiter. Speed trim is also sent to the dual-rate sensing circuit where it is summed with airspeed error at the input of the sensing circuit.

CADC error signal is sent through a lead circuit and then applied to the summing junction at the input of the gust filter and limiter. The purpose of this lead circuit is to provide optimum anticipation and damping, by proper gain adjustment, for smooth and rapid correction of airspeed error.

The limiter limits the signal value leaving it while the gust filter removes higher frequency signals that would cause excessive throttle activity. This signal is sent to a summing junction on the output of the gust filter and limiter.

Combined airspeed error and speed trim signal, at the input summing junction of the integrator, are applied through the now closed contacts of relay K1 (K1 energized when ATS is engaged) to the integrator to increase system accuracy by eliminating small, long-term errors. The output of the integrator is then summed with the output of the vertical gyro demodulator.

Pitch attitude signals are provided by the TPLC vertical gyro Intermediate Signal Selector (ISS). It is used to achieve a lead signal to move the throttle in anticipation of a speed change caused by a change in aircraft pitch attitude.

The summed outputs of the integrator and vertical gyro demodulator are then sent to the summing junction at the output of the gust filter and limiter. There, they are combined with the summed outputs of the rate generator and throttle position demodulators as well as the output of the gust filter and limiter.

These summed outputs are then sent through the deenergized contacts of the flare engage relay K2 (energized at radar altitude of 45 feet) and through the deenergized contacts of the flare relay K3 (energized at radar altitude of 30 feet) to a pre-amplifier.

The pre-amplifier has a threshold of ± 0.2 knot which reduces throttle activity due to the effects of backlash in the system. The combined error signals are amplified and fed to the power amplifier which determines the direction of servomotor rotation. This error signal is then routed to the servomotor generator, which is mounted on the clutch pack assembly, and commands the desired throttle movement to correct for the airspeed error.

A rate feed-back generator, in the servomotor, developes an output that is fed to the high-low rate selector portion of the dual rate sensing circuit. When the combined error signals of airspeed and speed trim are greater than 3 knots for 2 seconds or more, the dual rate sensing circuit causes the high-low rate selector to switch to a higher rate. This rate is fed to the rate generator demodulator and then to a summing point. In manual pilot/ATS mode, the normal maximum throttle rate is 1.7 degrees/second or 3.3 degrees/second. In autopilot/ATS mode, the normal maximum throttle rate is 3.3 degrees/second regardless of error signal's magnitude or duration.

The output of the servomotor generator drives a power gear at a 15:1 ratio, and the power gear drives the clutch pack at a 100:1 ratio. This gear arrangement develops the torque to drive the throttles to the commanded position and generates a throttle position feedback signal by means of a synchro. This position signal is demodulated and summed with demodulated rate. Maximum and minimum limit switches, as shown in Figure 4-8, mounted on the periphery of the clutch pack cable drums, are provided to restrict throttle activity over a specified range of travel. These limit switches also function to lockout the airspeed error integrator when two or more throttles reach their limits.

Grounding the integrator prevents it from going to a larger value which would otherwise delay the throttles from moving out of their limits when commanded by a pitching maneuver or airspeed error in the opposite direction. This

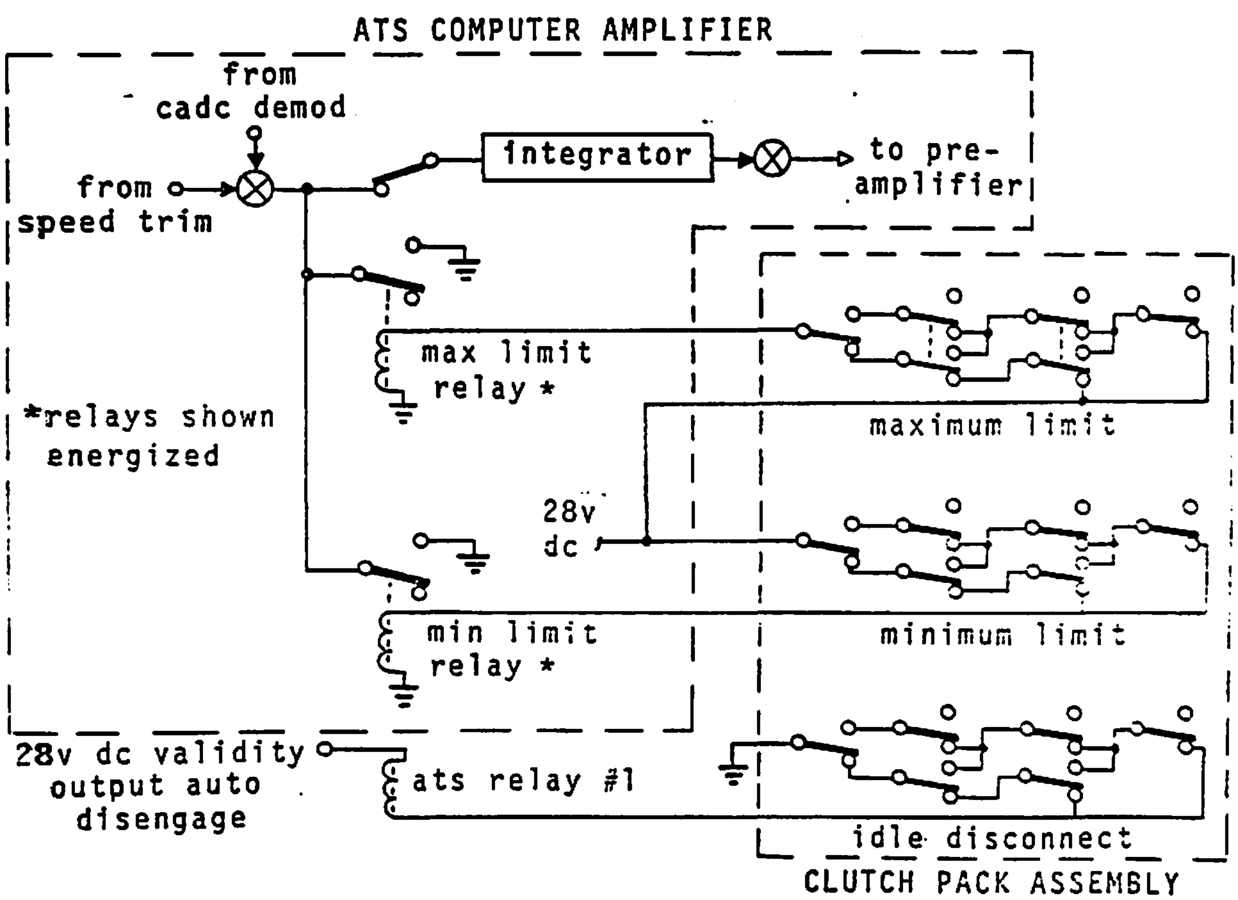


FIGURE 4-8. MAX-MIN LIMIT SWITCHES & IDLE DISCONNECT SWITCHES

switch arrangement permits the pilot to shut down a malfunctioning engine, then to position the dead throttle lever to the maximum forward thrust position to get it out of the way, and to continue ATS operation with three throttles even though one lever exceeds the limit switch settings. Idle disconnect switches are provided to automatically disconnect the ATS system when two or more throttles are moved to the idle position.

When the ATS is used in an AWLS approach, a 28-volt, dc LAND ARM (LA) signal is transmitted at a radar altitude of 100 feet by the TPLC. The ATS is automatically disconnected at this time if the throttle retard validity signal, from the flare computer, is not present. This action provides the pilot with an early warning that the flare mode is not going to operate properly. If the flare computer is working, then the LA signal has no effect. At flare engage altitude (45 feet), flare engage relay K2 energizes, which switches the input to the pre-amplifier to the output of the rate generator demodulator only, as shown in Figure 4-6. This action has the effect of clamping the throttles at their existing position to prevent them from increasing as the aircraft pitches up. At approximately 30-foot altitude, dual throttle retard signals are received. One signal (TR1) overrides the minimum throttle limit switch logic and other command signals and causes the throttles to drive to the idle position at a rate of 2.5 degrees/second. This action is accomplished by energizing the flare relay K3 connecting a portion of -10-volt, dc to the input of the pre-amplifier. The second throttle retard signal (TR2) is compared with the rate generator output in a manner that automatically disconnects the ATS if the rate signal is not within 2.5 ± 0.7 degrees/second with 1 second. Once the throttle retard mode is properly initiated, the logic is locked in to assure that the throttles continue to drive to the idle position in the event of loss of any flare computer signal output.

If conditions warrant and the pilot elects abort the approach, he depresses the go-around button on his control wheel. This action produces a continuous 28-volt, dc signal which is transmitted by the TPLC and causes automatic disengagement of the ATS by tripping the computer's internal automatic disengage logic. This signal prevents re-engagement of the ATS as long as it is present. Actuating the go-around switch on the control wheel a second time removes the 28-volt, dc go-around signal.

When the ATS computer is functioning properly, the automatic disengage logic circuit provides a 28-volt, dc validity signal to ATS relay No. 1. When the logic is tripped, the validity signal is lost and the servomotor control phase voltage is interrupted. This interrupts power to all ATS clutches and eliminates servomotor backdriving resistance whould the clutch pack clutch fail to disengage. The disengage logic, once tripped, remains tripped until a reset signal is applied. The external wiring circuit produces an automatic reset signal whenever the system is disengaged and the throttles are positioned in the forward thrust regime. If a fault which caused an automatic disconnect still exists, or the go-around signal is still present, the reset signal is overriden, thus preventing reset of the computer. When the go-around signal is absent or the fault has healed itself, re-engagement of the ATS can only be accomplished by actuation of the pilot's or copilot's throttle lever switches.

The ATS power supply receives 115-volt, ac input and develops all necessary voltages for the system.

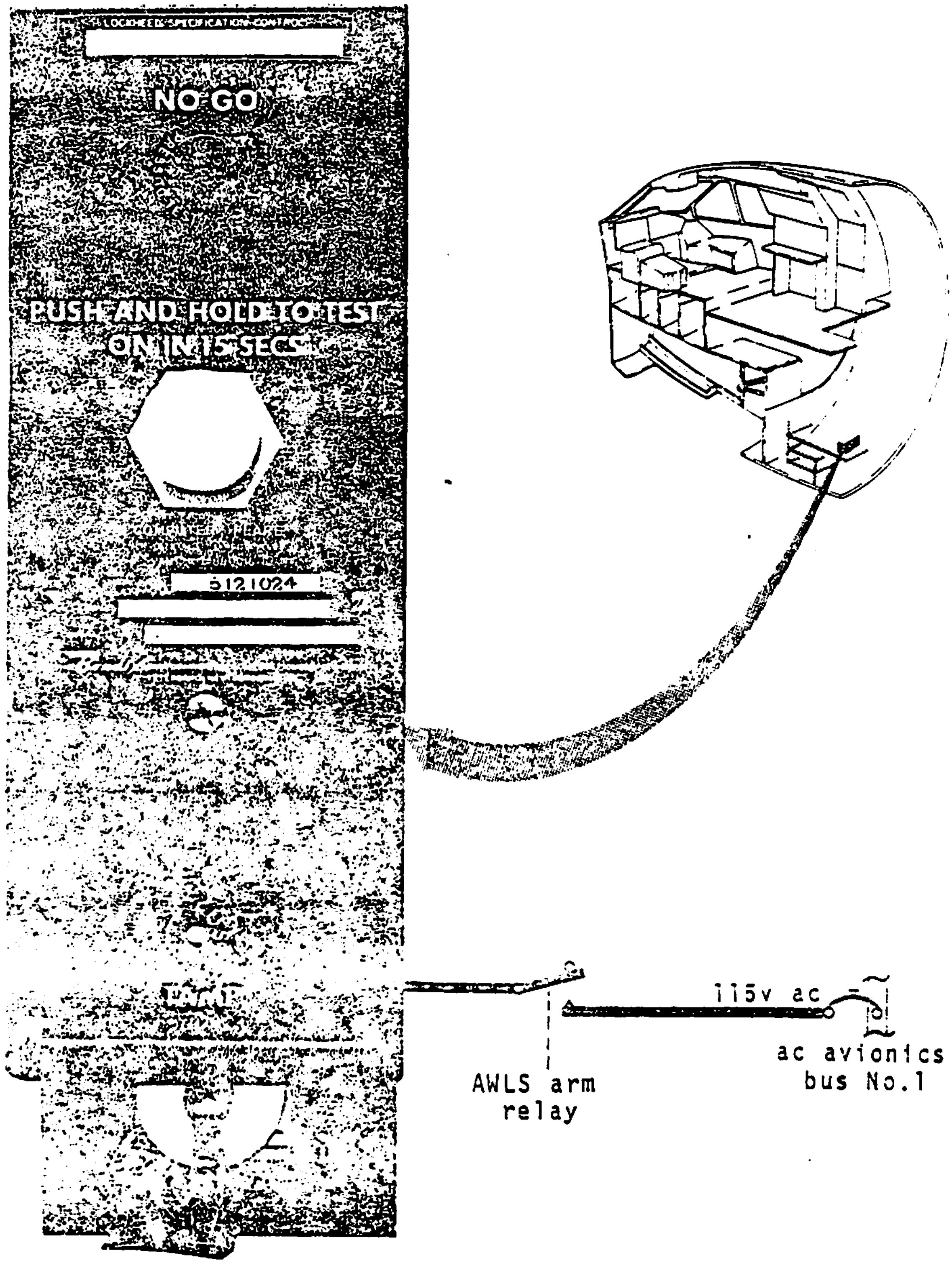


FIGURE 5-1. FLARE/LAND COMPUTER

FLARE/LAND COMPUTER

The flare computer is a single, self-contained unit as shown in Figure 5-1. It provides an altitude rate error voltage to the automatic flight control system and flight director system. This signal is used to automatically or manually control the aircraft in the pitch axis on a flight path that is a function of programmed altitude rate during a landing. It provides pitch steering commands to the touchdown point. Absolute altitude from a radar altimeter (Chapter 2) and vertical velocity rate from a normal accelerometer (Chapter 11) are used in performing the flare computation.

Altitude circuits trip three detectors:

- LAND ARM (LA) at 100 feet
- Flare Engage (FE) at 45 feet
- Throttle Retard (TR) at 30 feet

Validity signals go to the TPLC for validity functions. Error voltages and switch functions are provided to properly initiate indications and related functions during a flare maneuver. Any unsafe fault in the computer is detected and the computer outputs are indicated as invalid. The computer has built-in test circuits that may be initiated by enroute, preland, or self-test. Dual channels are continuously compared for validity.

SYSTEM OPERATION

The flare computer system is turned on through the AWLS arm relay which is energized by the AWLS switch on the autopilot control panel. A voltage of 115 volts, ac, through the arm relay is sent through a 1-ampere fuse to the flare computer power supply. All dc power used by the computer is furnished by the computer power supply. The flare computer is connected to other AWLS components as shown in Figure 5-2.

A front panel test button is used with a "no-go" light for self test. Pressing the test button starts a programmed test of the computer. The no-go light goes on and in 15 seconds the "test complete" light in the test switch illuminates. If a failure is detected, the no-go light illuminates. Releasing the test button extinguishes the test complete light. The no-go light stays illuminated if a fault is detected. Enroute and preland tests are initiated by the TPLC and cause identical performance of the self test. Invalid outputs illuminate the master caution and FLARE lights, disengage the AFCS and ATS, and bias the ADI pitch command pointer.

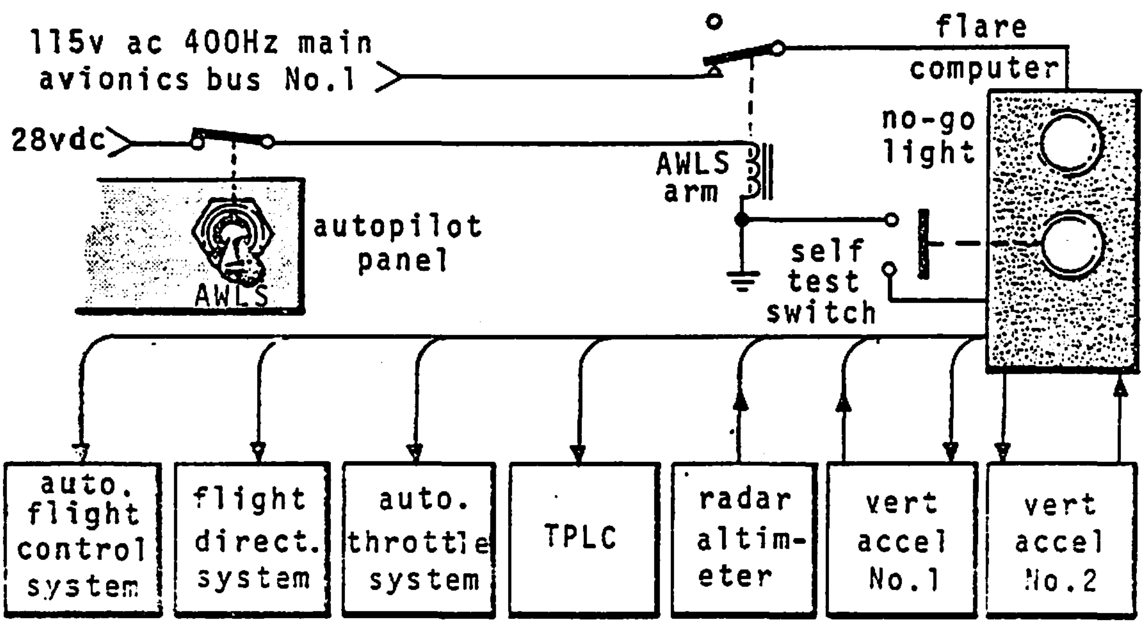


FIGURE 5-2. INTERCONNECTED SYSTEMS

SPECIFICATIONS

Input Voltage	115-volt, ac, 400 hertz avionics ac bus No. 1
Signal Input	Radar altimeter and accelerometer No. 1 and No. 2
Outputs	Fail-safe designated, off-line monitored Dual vertical velocity error ($\dot{h}_e$) Dual Throttle Retard (TR)

Continued

	Dual Flare Engage (FE) Dual LAND ARM (LA) Validity Test complete Accelerometer test voltages
Self Test	When initiated, programmed testing of computer is accomplished by BITE
Validity	$\dot{h}_e$, 2 channels compared automatically, LA, FE, TR: Main and model outputs compared in reference to time of trip.

THEORY OF OPERATION

The computer uses the summation of the altimeter and the accelerometer input rates above a biased reference to form the altitude rate error signal ($\dot{h}_e$). This error voltage is used in the automatic flight control and flight director systems and to automatically control the throttles during the flare maneuver. The altitude circuit has three altitude detectors, one each for the 100-foot, 45-foot, and 30-foot levels. Normal acceleration channels limits accelerometer misalignment and trim changes by a high-pass filter. The filter performs as a dc washout circuit.

When the aircraft descends below 200 feet, detectors indicate aircraft descent through the land arm altitude, the flare engage altitude, and the throttle retard altitude.

Radar altimeter signal is processed through an isolation circuit to provide a proper load to the altimeter as shown in Figure 5-3. The output, controlled by a logic switch, inhibits altitude voltage during self test. A following limiter amplifier functions to limit the altimeter signals to 200 feet and below. The limiter output goes to detectors of land arm altitude, flare engage altitude, throttle retard, altitude rate filter, and summation amplifiers. These detectors provide logic signals as the aircraft descends through each decision altitude.

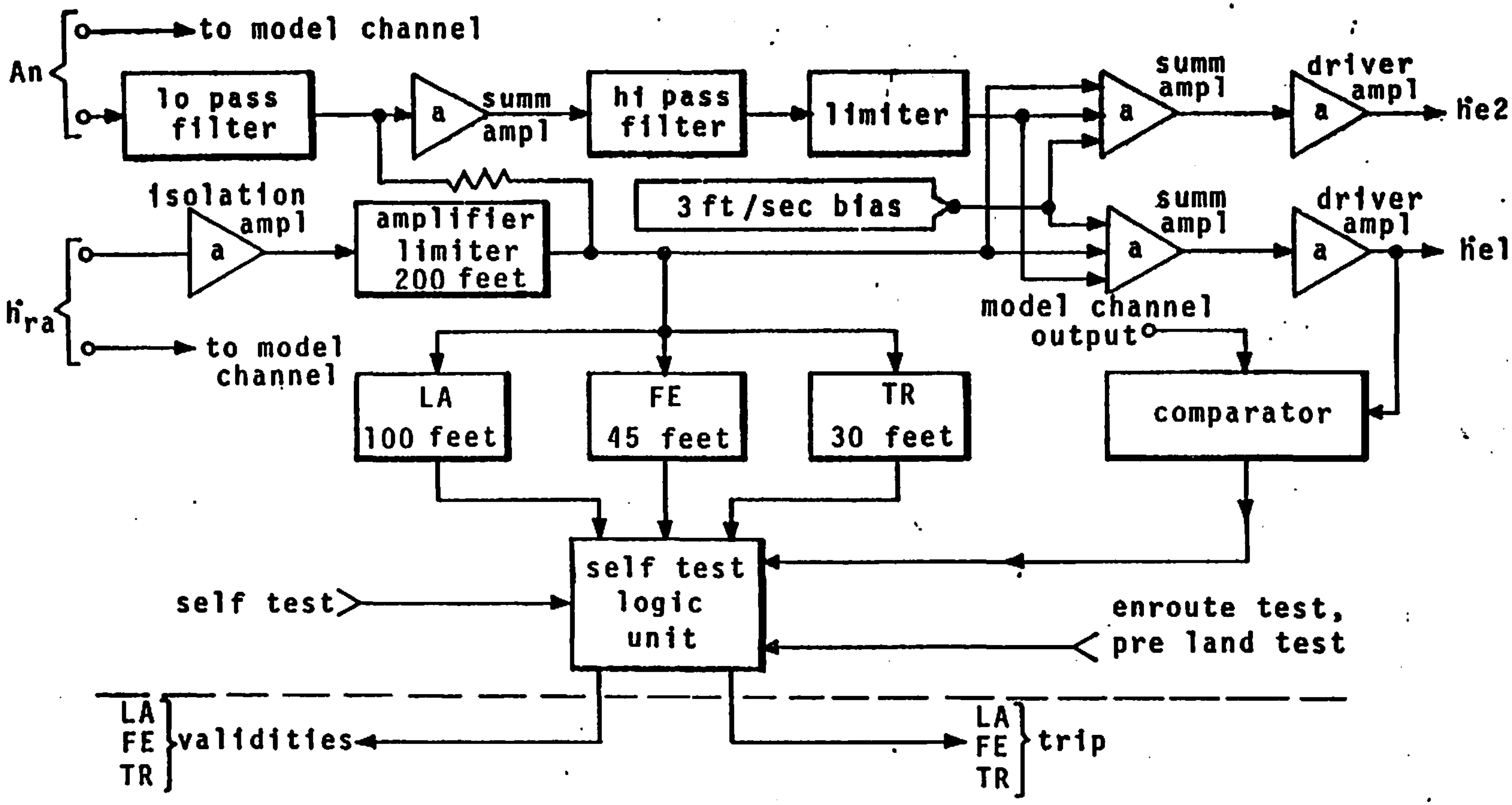


FIGURE 5-3. ACTIVE CHANNEL BLOCK DIAGRAM

Normal acceleration voltage goes in and out of a 20-second lag filter and is summed with the radar altitude voltage.

A voltage equivalent to the sink rate of the aircraft is produced in the acceleration channel by the summing amplifier, filter, and limiter. This voltage is augmented altitude rate ($\dot{h}_e$ aug).

Summing amplifiers at the end of the channels control the output drivers. Driver outputs are altitude rate error one and two, which go to the automatic flight controls and throttle control systems.

A parallel channel identical to the active channel is used for comparison. A single summing amplifier output is compared with No. 1 active channel driver output for validity of the computer channels. Level detector outputs are compared by their trip times. A long-time difference between main and model LA, FE, or TR causes a fault indication.

SELF TEST -- BUILT-IN TEST EQUIPMENT (BITE)

The computer circuits may be tested without external AGE equipment. A test button on the front panel initiates a 15-second self-test program. The program is divided into five 3-second program steps. These steps test conditions of maximum signal levels, null, state of deliberate faults, and normal operation. Program steps T1 and T5 are valid; T2 and T3 are invalid. T4 is a delay period. Flare computer outputs are valid if the logic counters read "5." BITE does not test two radar altitude isolation amplifiers and one buffer amplifier or differentiate an accelerometer failure from computer failure. External AGE equipment has provisions to test the complete flare system circuits. TPLC initiates the self-test during enroute and preland tests.

The flare computer BITE sequence is shown in the following table.

Flare Computer BITE Sequence

Test Step	Test Action	Test Result	Functions Tested
1	a) Inhibit both h_{RA} signals and substitute test signals b) Torque accelerometers	a) Comparator Safe b) All MDA, FE, TR detectors-no trip c) All MDA, FE, TR drivers-off d) MDA, FE, TR logic-valid	a) Computation channel agreement b) Amplitude detectors-no trip c) Comparator safe condition d) Drivers off e) Validity and logic
2	a) Inhibit the h_{RA} signals and test signal in channel No. 1, maintain test signal in channel No. 2 b) Maintain accelerometer torquing	a) Comparator failed b) No. 1 MDA, FE, TR detectors-trip c) No. 2 MDA, FE, TR detectors- no trip d) MDA, FE, TR drivers-on e) MDA, FE, TR logic-non valid	a) Computation channel disagreement b) No. 1 amplitude detectors-trip c) No. 2 amplitude detectors-no trip d) Drivers on e) Validity and logic
3	a) Inhibit both h_{RA} signals and substitute test signal in channel No. 1, remove test signal from channel 2. Remove torque signal	a) Comparator failed b) No. 1 MDA, FE, TR detectors-no trip c) No. 2 MDA, FE, TR detectors-trip d) MDA, FE, TR drivers-on e) MDA, FE, TR logic-non valid	a) Computation channel disagreement b) No. 1 amplitude detectors-no trip c) No. 2 amplitude detectors-trip d) Drivers on e) Validity and logic

Flare Computer BITE Sequence (Continued)

Test Step	Test Action	Test Result	Functions Tested
4	a) Inhibit both h_{RA} signals b) No test signals present (Note-step 4 is a 3-sec. stabilization period.)	a) Comparator alarm b) MDA, FE, TR detectors-trip c) MDA, FE, TR drivers-on d) MDA, TR logic-valid e) FE Logic invalid	a) Computation channel null agreement b) Amplitude detectors c) Drivers on d) Validity and logic
5	a) Remove both h_{RA} inhibits b) System normal	a) Comparator safe b) MDA, FE, TR-concurrence c) MDA, FE, TR-function of b d) MDA, FE, TR logic-function of previous test results e) Test complete signal f) Go-No/Go light function of test results	a) Computation channel normal b) Amplitude detectors c) Drivers d) Validity and logic

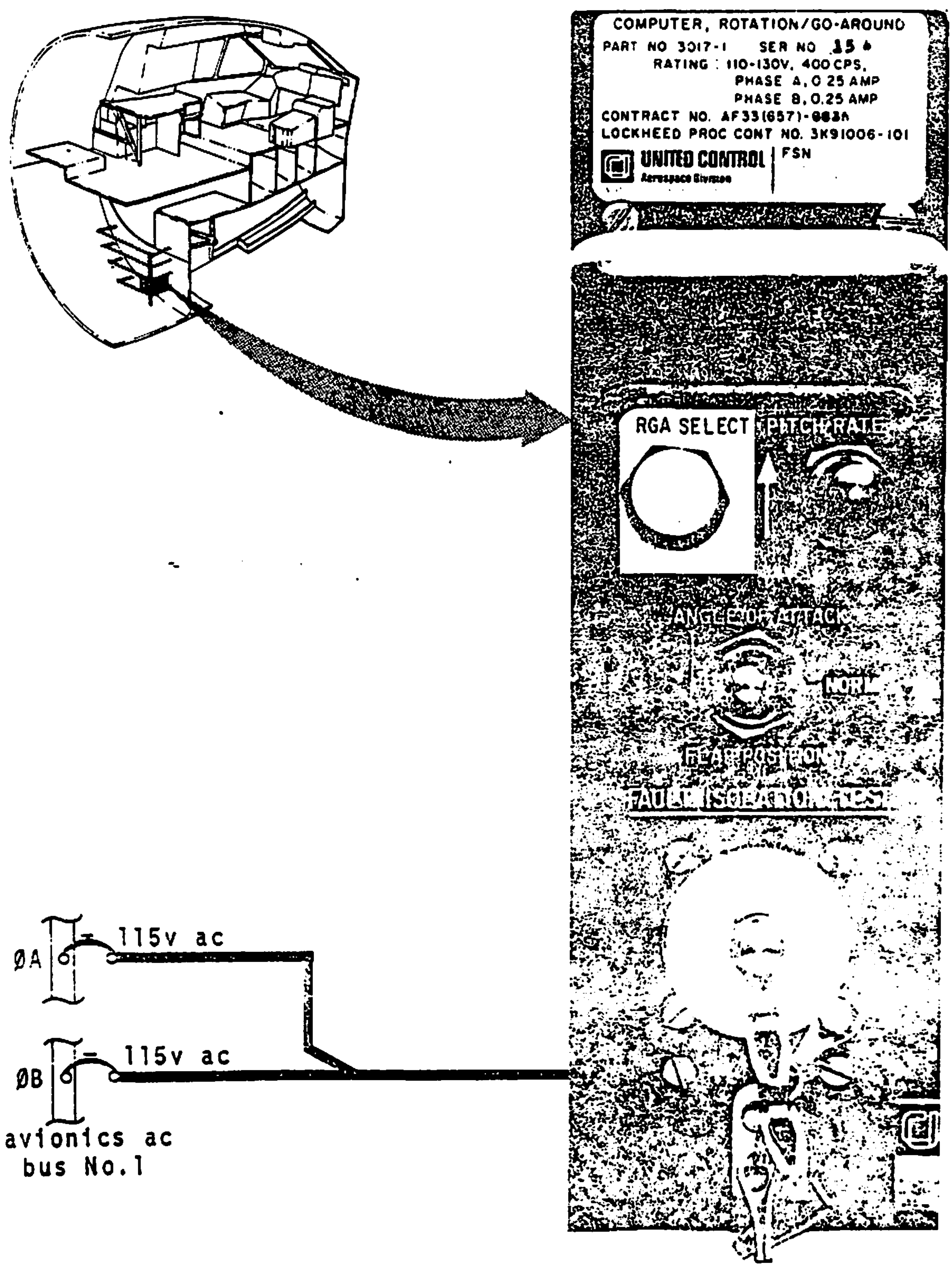


FIGURE 6-1. ROTATION/GO-AROUND COMPUTER

ROTATION GO-AROUND (R/GA)

A Rotation Go-Around (R/GA) computer, shown in Figure 6-1, is one of the associated components of AWLS. The R/GA system provides pitch steering commands to the pilot's ADI's for use during takeoff or go-around maneuvers. In addition the R/GA computer develops a complemented altitude rate ($\dot{h}_c$) signal for display on the pilot's Vertical Velocity Indicators (VVI) and for use by the VER NAV system (Chapter 3).

A programmer in the R/GA computer develops an angle-of-attack signal which is the proper one for takeoff or go-around maneuvers based on aircraft configuration.

AIRCRAFT INSTALLATION

The following components are utilized in the R/GA system:

- Rotation/Go-Around Computer
- Test Programmer and Logic Computer
- Angle-of-Attack Transducer, Left and Right
- Autopilot Two-Rate Axis Gyro
- No. 1 Horizontal Accelerometer
- No. 3 Vertical Accelerometer
- CADC No. 1 and 2
- Touchdown Relays No. 9 and 10
- Flight Directors No. 1 and 2 (Computer and ADI)
- Pilot and Copilot Altitude VSI

The VER NAV system utilizes complemented altitude rate (IVV) and validity from the R/GA computer but is not directly related to the R/GA system.

The R/GA computer and Test Programmer and Logic Computer (TPLC) are located in the left underdeck equipment rack. The 115-volt, 400-hertz, ac, (phases A and B) required for each system is taken from avionics ac bus No. 1. The 5-ampere circuit breakers which protect the systems are located on the avionics circuit breaker panel.

SYSTEM OPERATION

To engage the R/GA computer, either pilot may operate a GO-AROUND switch on his outside yoke handle. The next switch operation disengages the R/GA and returns the pitch steering to the previously selected mode.

When the R/G mode is initiated, the ATS is disengaged, wings level is commanded by the flight director computer, and the pitch steering is switched to the R/GA computer output for display on the ADI pitch steering bars. The GO-AROUND mode light on the AWLS progress display panel also illuminates at this time.

Manual "enroute test" of AWLS initiate a test command through the TPLC to the R/GA computer which tests the fault detector circuit only. This test is usually conducted at cruise altitudes. The R/GA mode must be selected or the APPROACH ARM point of the approach reached, to test the R/GA computer.

A continuous self-test of the R/GA computer is in progress while power is applied. An invalid output goes to the TPLC which, in other than test mode, initiates fault indications on the AWLS fault panel. A fault in R/GA mode prevents pitch steering displays on the ADI's, and the VVI's display barometric altitude in lieu of complemented altitude rate.

SPECIFICATIONS

Outputs:	
Valid	Complemented altitude rate ($\dot{h}_c$) Angle-of-attack error Validity voltage (28 volts, dc)
Invalid	Barometric altitude rate ($\dot{h}_b$) Angle-of-attack error (inhibited by logic switching) Invalid voltage

Continued

Test:	Manual "enroute" Built-in self test (BITE) while power is on
Inhibit:	Disable bias inhibits programmed angle-of-attack by squat relays No. 9 and 10 contacts AOA vane comparator inhibited unless R/GA selected or APPROACH ARM point reached Pitch-up limited to 15 degrees (maximum) Steep bank maneuver during R/GA, faults the computer but heals toward wings level attitude
Input (See Figure 6-2):	
Angle-of-Attack Vanes No. 1 and 2	3-wire synchro: left synchro for signal, right for comparison
Longitudinal Acceleration	From No. 1 horizontal accelerometer: dc output, 8.0 volts per g, output biased for 1.0 g
Normal acceleration	From No. 3 vertical accelerometer: dc output, 8.0 volts per g, biased for 1.0 g
Flap Position	Signal proportional to flap position from right inboard position transmitter
Barometric Altitude	From No. 1 and 2 CADC: 250 millivolts-per-thousand-feet-per-minute, in-phase = climb, out-of-phase = dive
Pitch Rate	Autopilot 2-axis rate gyro: 400 hertz, 200 millivolts-per-degree-per-second, in-phase, pitch-up, out-of-phase, pitch-down
Pitch Angle	From TPLC, ISS: 400-hertz, in-phase, pitch-up, out-of-phase, pitch-down

Continued

SPECIFICATIONS (Continued)

Bank Angle	From TPLC ISS: 400 hertz, in-phase, right bank; out-of-phase, left bank
Squat Relays	No. 0 and 10 contacts used for disable bias in angle-of-attack programmer
Approach Arm	Ground from TPLC
R/GA Mode Select	28-volt dc from TPLC
Test Command	28-volt dc from TPLC

THEORY OF OPERATION

The R/GA computer generates two outputs:

- o AOA Error
- o Complemented Altitude Rate

AOA error voltage is a summation of augmented AOA and programmed AOA voltages. This summation results in either a negative AOA error voltage (producing a pitch-up command) or a positive AOA error voltage (producing a pitch-down command) to the pilot's ADI pitch steering bars.

Augmented angle-of-attack is the result of summations of pitch rate, pitch angle, and AOA and logitudinal acceleration sensor voltages. These voltages are a dc analog of the sensor outputs as shown in Figure 6-3. Longitudinal acceleration voltage is applied through a 1-g bias network to (A4)J5 as shown in the illustration. Pitch angle voltage after demodulation is divided into two channels in (A7)J5. One channel is used to washout the effect of gravity field change on the longitudinal accelerometer. Washout depends on pitch angle phase and amplitude. The other pitch channel is further divided into two identical channels: One channel is the active; the other is for malfunction monitoring. Identical channel outputs are in-phase, relative to the inputs. Active channel voltage goes to the summation inputs of (A5)J5, while monitor voltage goes to the summation inputs of (A6)J5. Active channel (A7) limits aircraft pitch-up to 16 degrees maximum.

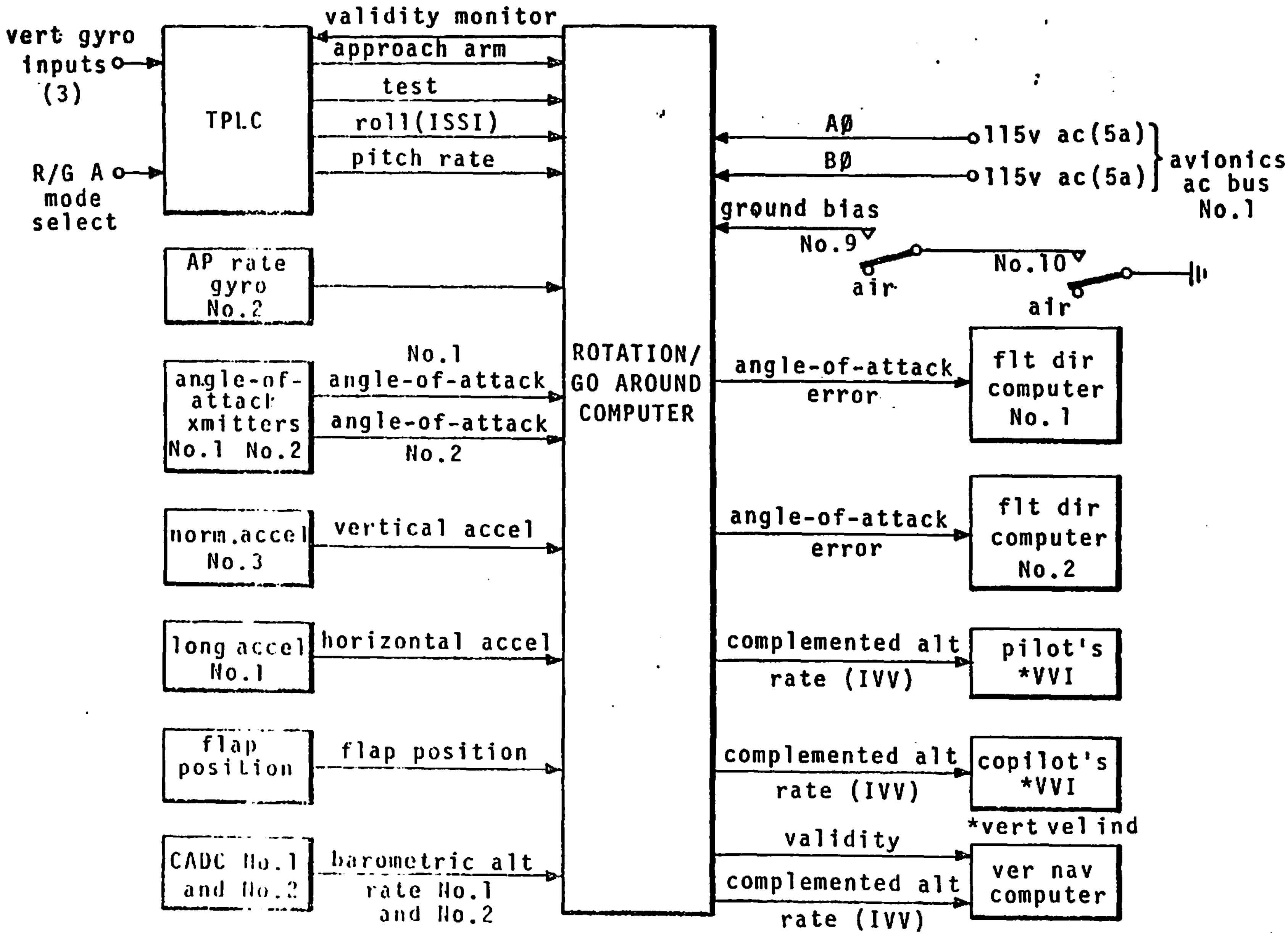


FIGURE 6-2. ROTATION / GO AROUND COMPUTER DATA FLOW

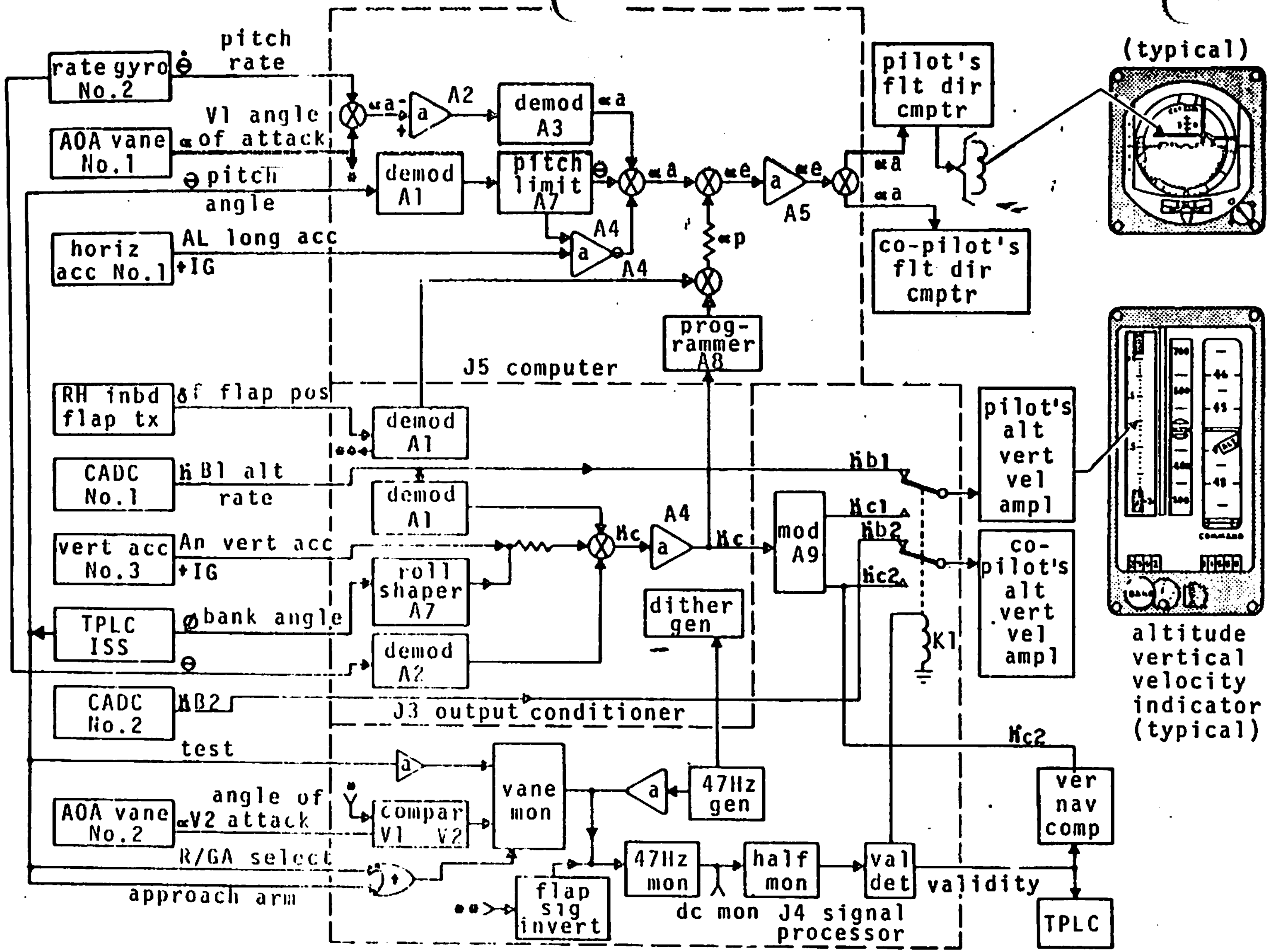


FIGURE 6-3. ROTATION / GO AROUND SYSTEM BLOCK DIAGRAM

Pitch rate voltage is applied to the summation input of (A2)J5 with vane No. 1 AOA. An AOA voltage is extracted for comparison with vane No. 2 AOA where a five degree difference causes a fault condition. The input of (A2)J5 is amplified, inverted and then demodulated. This demodulated voltage is summed with pitch angle and modified longitudinal acceleration voltages. Summation of these voltages generates augmented AOA voltage at the input of (A5)J5. Augmented AOA and programmed AOA, when summed and amplified, is the AOA error voltage. ($\propto$ E)

Complemented altitude rate is the result of the summation of barometric altitude rate, normal acceleration, bank angle, and pitch rate. Barometric altitude rate and pitch rate voltages are demodulated and applied to the summing input of (A4)J3. Aircraft ascent or descent determines the phase of the altitude rate voltage. Normal acceleration voltage is modified by bank angle voltage to wash-out "g" forces produced by bank attitude and is applied to summing input of (A4)J3. This voltage is amplified and inverted. One output of A4 is applied to modulator (A9)J3 for the vertical velocity amplifiers and VER NAV system; the other goes to the angle of attack programmer (A8)J5.

Complemented altitude rate ($\dot{h}_c$) is applied to two identical channels in the AOA programmer. One is active, the other is for fault monitoring. The programmer has a preset AOA (approximately +1.5 degrees). To prevent over-rotation, programmed AOA is limited in the takeoff schedule by a disable bias through the squat switches. Complemented altitude rate modifies the programmed AOA, and flap position further modifies the programmed AOA voltage. This modified, programmed AOA is applied to the input summing of (A5)J5. Summed, augmented, and programmed AOA is the AOA error voltage into (A5)J5. AOA error voltage is amplified, inverted, and sent to the flight director computers and may be switched in by the TPLC for display on the ADI's.

An example of AOA error change is to assume the aircraft is in the R/GA maneuver with longitudinal acceleration and instantaneous vertical velocity (IVV) increasing. Since pitch-up command depends on aircraft speed and altitude rate, the AL and IVV voltage increase would provide commands for more pitch-up. Pitch-up command is represented by a negative AOA error [illegible]. [illegible] sensor input changes to the computer, such as flap position, roll, and pitch, would also alter pitch commands.

TEST AND MONITOR

Two modes of test are used in relation to the R/GA computer: manual AWLS test by the pilots (enroute) and an integrated self test. The self test tests the R/GA fault monitor only. Integrated self test is always functioning during the time power is applied to the R/GA computer.

The R/GA computer is monitored by either an in-line or dual-channel comparator technique as shown in Figure 6-4. The in-line method is accomplished by adding a 400-hertz modulated 47-hertz (dither) signal to the sensor inputs. The dither signal is acted upon by the operational circuitry in the same manner as the sensor inputs. Following this, the dither signal is removed at a point downstream. If the signal chain is intact and functioning properly, the level of the dither signal is known. The dither signal, along with another signal (47 hertz) of proper phase and amplitude are both applied to an ac monitor. The monitor does not trip as long as the dither signal characteristics have not been altered by the signal chain. If the dither signal has been altered by the signal chain, the ac fault monitor applies a signal to the validity detector.

Parallel channel monitor voltages are summed and applied to the input of (A6)J5, dc monitor. Part of the active channel voltage (AOA error) which has been inverted is also applied to the summing input of A6. The two voltages cancel if there is no malfunction in any of the active or monitor channels. The dc monitor applies any fault signal to the validity director. This is shown on Figure 6-5.

A ground maintenance test of the computer may be initiated after the AWLS is on and the R/GA mode is selected. A validity light on the R/GA computer is illuminated at this time. Activating the AOA-flap positioner (angle-of-attack) switch located on the computer front panel, shown on Figure 6-1 alternately between the two positions causes the light to go out. The switch should be released and the light illuminates. Placing the pitch rate switch to "PITCH RATE" has the same effect.

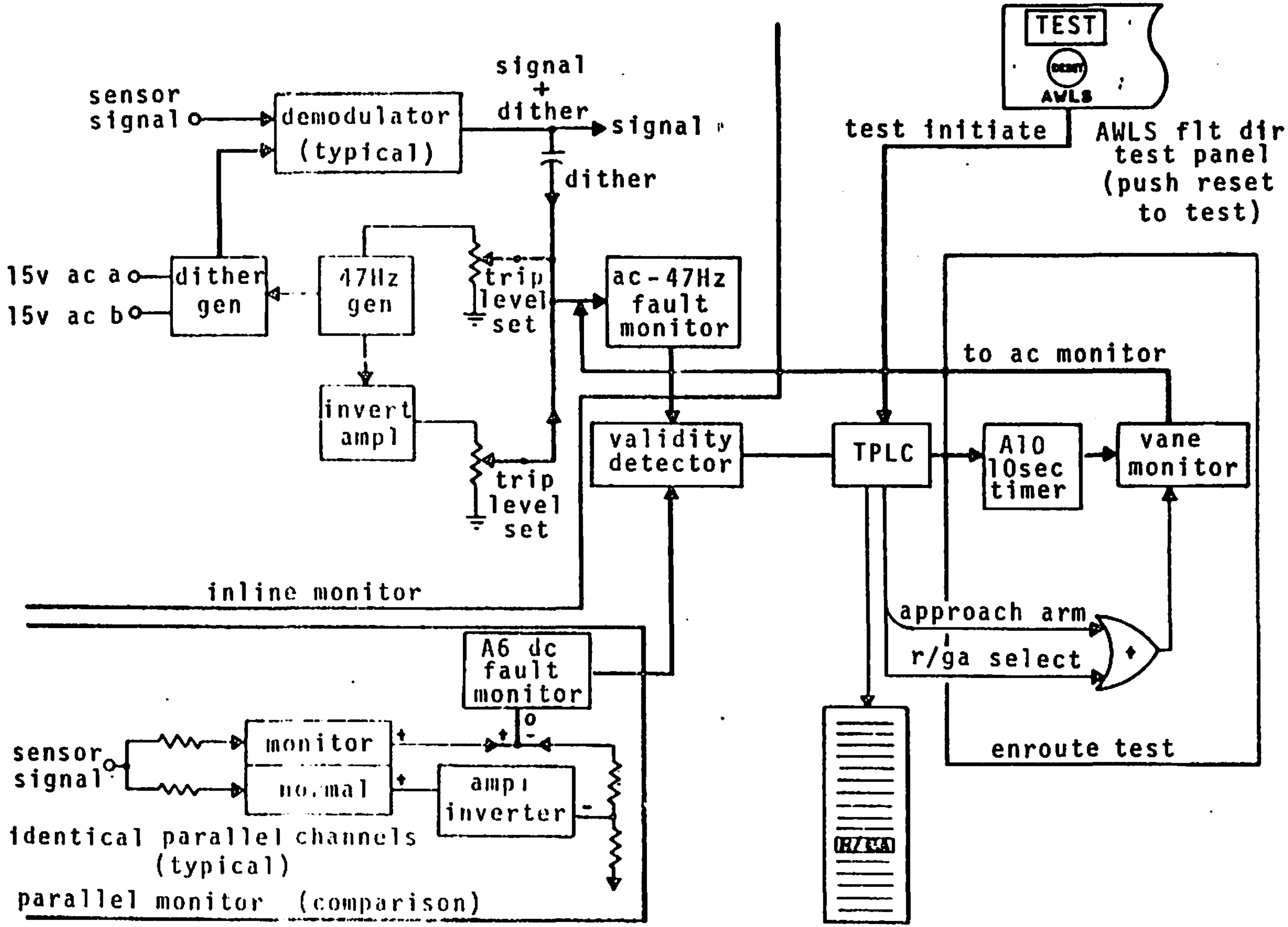


FIGURE 6-4. TEST AND MONITOR BLOCK DIAGRAM

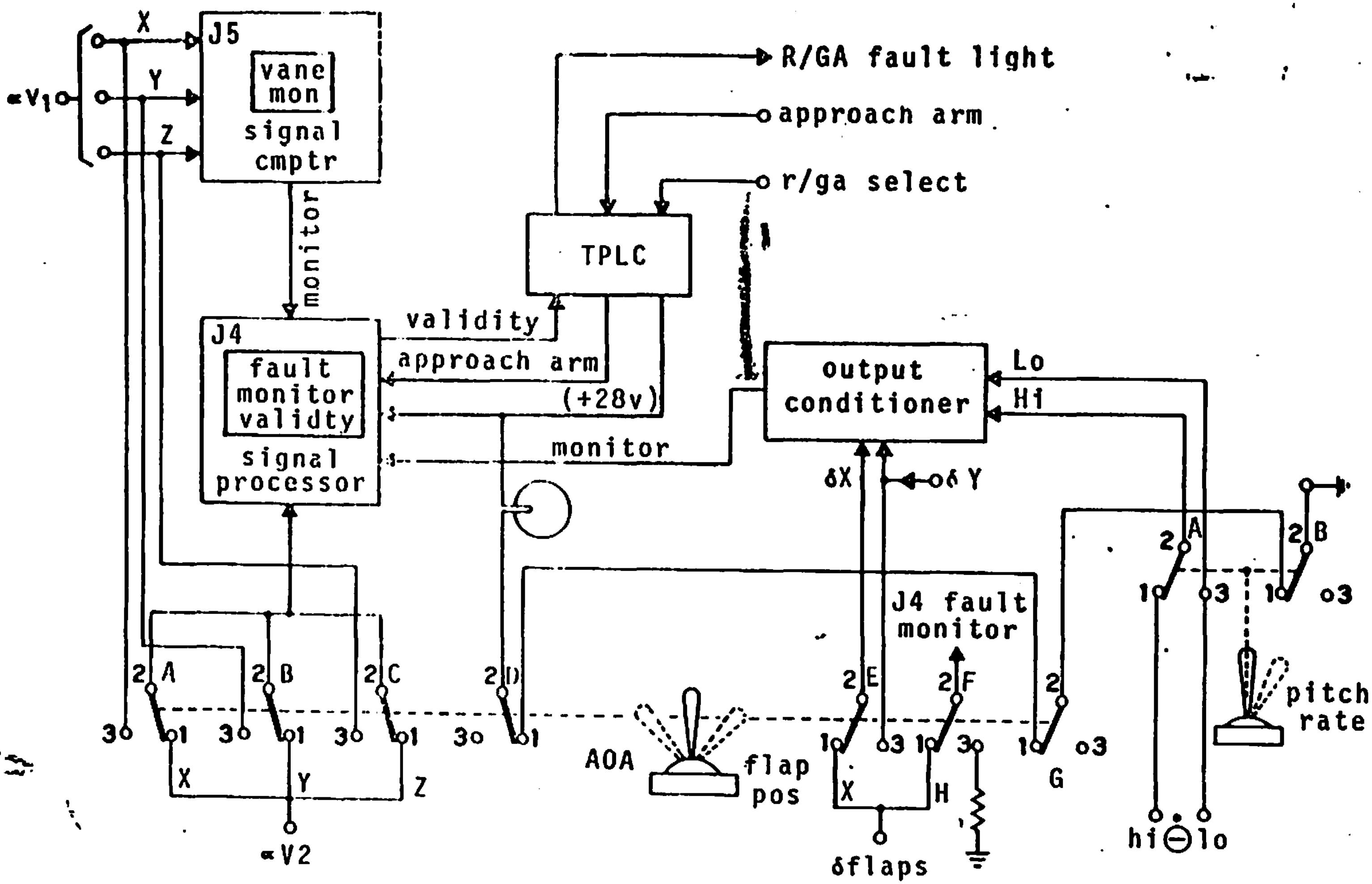


FIGURE 6-5. GROUND TEST BLOCK DIAGRAM

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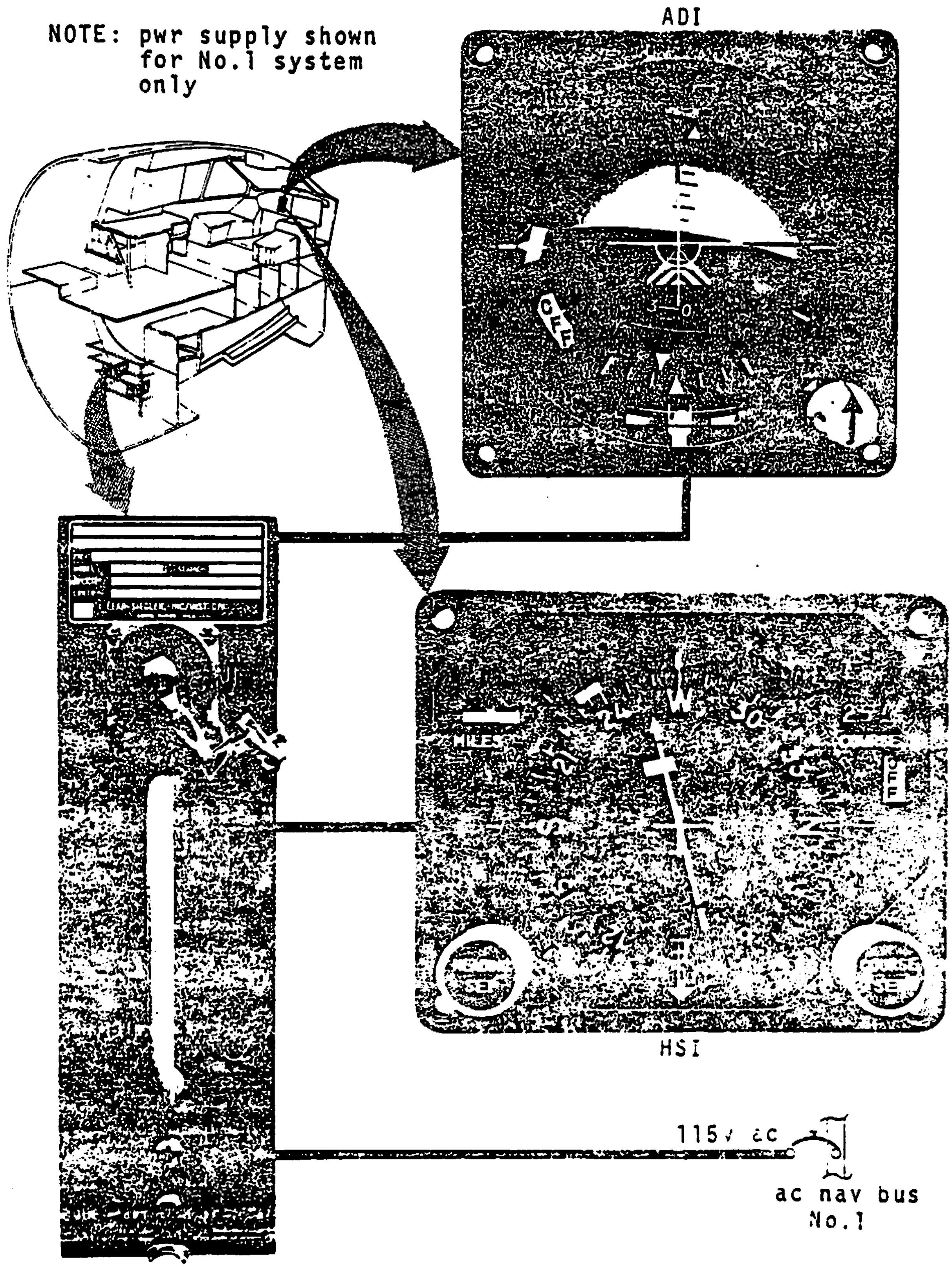


FIGURE 7-1. FLIGHT DIRECTOR COMPUTER, ADI, AND HSI

FLIGHT DIRECTOR SYSTEM (FDS)

The Flight Director System (FDS) provides a pictorial display of the data required to perform instrument flight maneuvers. The flight director receives information from various electronic navigation systems and from AWLS components located onboard the aircraft. Selected input signals are processed and displayed on two multipurpose flight director indicators. These signals may be selected both manually by the pilot or copilot or automatically by computers within the AWLS.

This system provides continuous monitoring of itself and of the subsystems supplying data to it. Failures within the flight director or loss of reliability signals from the feeding subsystems cause immediate warnings to the pilots. This monitoring enables the FDS to supply the pilots with steering commands to direct the aircraft to the minimum limits of an FAA-approved, Category II landing. For a Category II landing, the aircraft must have broken through any overcast by 100 feet altitude and must have visibility of one-quarter of a mile. These minimums are a significant improvement over the previous minimums of 200-foot altitude and half-mile visibility (Category I minimums).

The FDS indicators, Horizontal Situation Indicator (HSI) and Attitude Director Indicator (ADI), as shown in Figure 7-1, display both raw (non-computed) navigation information and computed steering information, which has been processed by, or developed within, the flight director system. The HSI displays primarily the raw information and the ADI displays primarily the computed steering information. Combining or integrating the displays of several related types of information provides the pilots with a much more meaningful presentation on two instruments, which eliminates the need to scan numerous instruments.

AIRCRAFT INSTALLATION

Two complete and independent FDS's, designated as the No. 1 (or pilot's) flight director and the No. 2 (or copilot's) flight director, are installed in the C-141A. Each system contains two display indicators: the HSI and the ADI. The two indicators for each system are located on the corresponding instrument panels. An elevator position transmitter for each system is located on an elevator quadrant beneath the pilot's and copilot's floorboards in the flight station.

In the left-hand underdeck avionics rack is a computer for each system. For each system there is an attitude gyro, rate switching gyro, and rate transmitter gyro located in the center underdeck equipment rack. A power adequacy indicator (rate off) for each rate transmitter gyro is located on the main instrument panels above the ADI's. A navigation selector panel is located on the glare shield above each pair of indicators. An HSI slaving panel, located on the center pedestal, and a test panel, located on the center instrument panel, are all common to both flight director systems.

The Circuit Breakers (C/B) for the FDS as shown in Figure 7-2 are located on the avionics C/B panel at the navigators station and on the emergency C/B panel aft of the pilot's side console.

SYSTEM OPERATION

The system has no on-off switch, as such. Anytime aircraft power is supplied and the system C/B's are in, the system is energized. There is a one-minute time delay incorporated to allow time for the gyros to come up to speed and stabilize. After the time delay has elapsed, a voltage is supplied to mask the POWER OFF flag on the ADI.

In order to understand the operation of this system, it is first necessary to become familiar with the indicators and the purpose of each of their indicating movements.

Horizontal Situation Indicator (HSI)

The AQU-4/A HSI presents a pictorial plan view display of aircraft heading and position with respect to a selected course. Heading and course error signals developed in the indicator are used by the flight director computer and the autopilot system. (No. 2 HSI supplies the A/P and No. 2 flight director.)

Miniature Aircraft Symbol

A miniature aircraft symbol, fixed to the center of the instrument, represents the actual aircraft as viewed from above. The symbol establishes a reference for presentation of navigational information.

Lubber Line

A lubber line provides an index for accurate reading of aircraft heading.

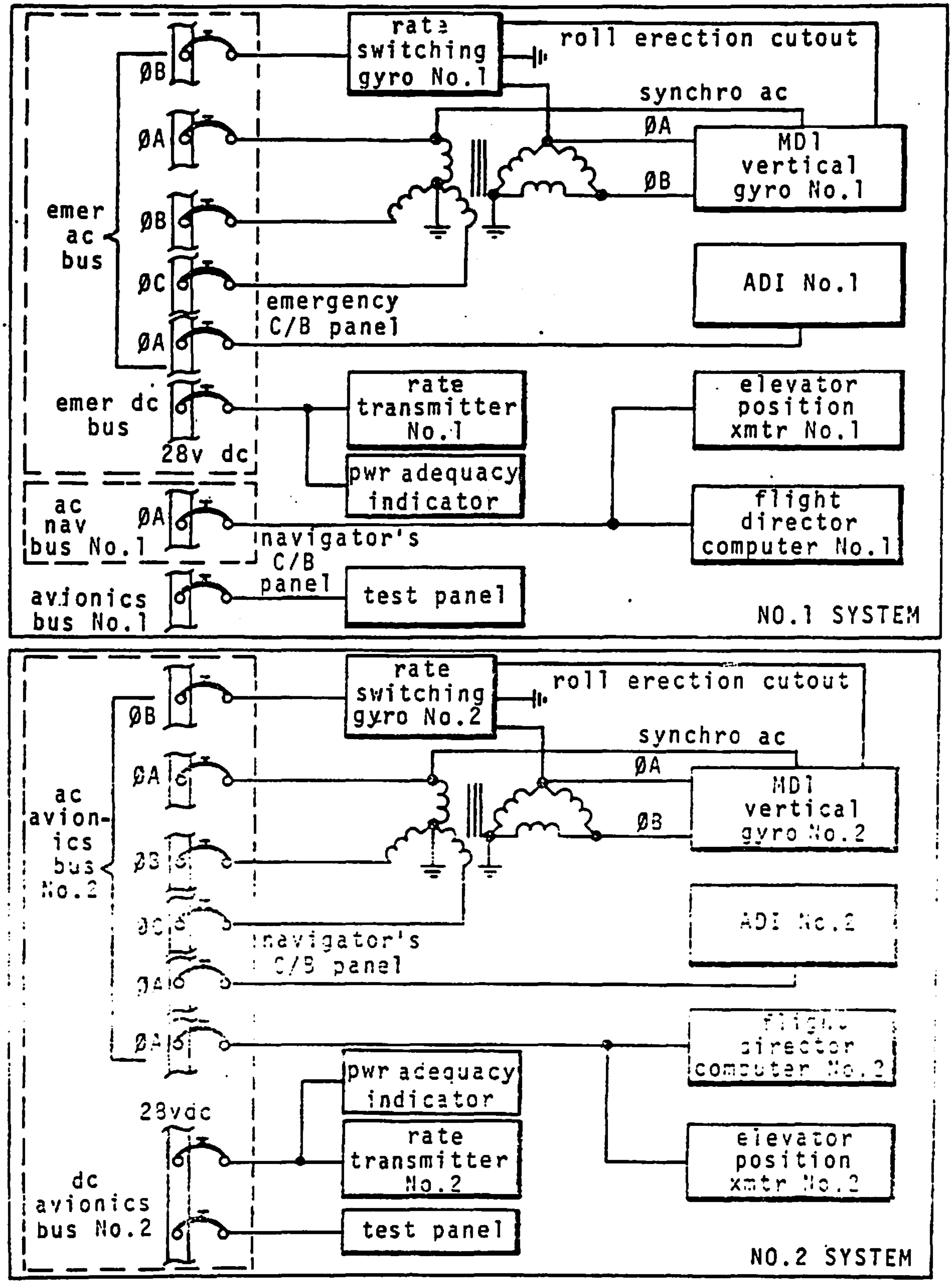


FIGURE 7-2. POWER DISTRIBUTION

Compass Card

The compass card is driven by signals from the No. 1 and No. 2 C-12 compass system. When read against the lubber line, the compass card indicates aircraft heading.

Bearing Pointer

The bearing pointer indicates bearing to a radio navigation station or Track Angle Error (TAE) depending on whether a radio mode or a computer mode is selected. When NAV OFF mode is selected, the bearing pointer is automatically positioned to the bottom of the lubber line.

Course Arrow

The course arrow is a movable pointer which is positioned manually to the desired course by the COURSE SET knob. Selected course is read by the position of the course arrow relative to the compass card. When NAV OFF mode is selected, the course arrow is slaved to aircraft heading.

Course Deviation Bar

The course deviation bar indicates deviation to the left or right of the selected course. Movement of the bar indicates degrees off course in VOR/TACAN, and miles crosstrack in ASN-35, ASN-24 computer modes, relative to the graduated course deviation scale.

To-From Arrow

The to-from arrow functions during Tacan or VOR modes only. The arrow indicates whether the aircraft is flying inbound or outbound (toward, or away from) on radial.

Deviation Bar Flag

The deviation bar flag indicates a malfunction in the system providing course deviation information to the indicator. The flag is masked when the input signal is valid.

Heading Marker

The heading marker is a movable marker which is manually positioned by the HEADING SET knob to the desired aircraft heading. Once set, the heading marker rotates with the compass card.

Range Indicator

The range indicator provides slant range distance to a selected Tacan facility or distance-to-go along a selected computer course.

Range Warning Flag

The range warning flag masks the range indicator when distance information is unreliable or when a mode other than Tacan, doppler, or navigation computer is selected.

Power Off Flag

The power off flag, when in view, indicates a malfunction in the compass card circuits or in the compass system.

Course Readout

The course readout provides a digital readout of the selected course.

Attitude Director Indicator (ADI)

The ADI presentation is a pictorial display of aircraft attitude and computed steering commands by means of a three-dimensional, forward-looking display. Glide slope deviation, turn and slip information, and radar altitude are also presented on this indicator.

Miniature Aircraft Symbol

The miniature aircraft symbol, fixed to the center of the indicator, represents the actual aircraft as viewed from the rear. It provides a reference for pitch and roll information, computer steering commands, and aircraft radar altitude.

Attitude Sphere

The attitude sphere is positioned in the roll and pitch axis by signals from the attitude gyro. Displacement of the sphere indicates actual aircraft attitude relative to straight and level flight.

Pitch Attitude Scale

The pitch attitude scale consists of specific increments of pitch attitude. Pitch displacement of the aircraft is determined by the position of the scale relative to the miniature aircraft.

Bank Attitude Scale

The bank attitude scale consists of graduated marks located around the attitude sphere. Bank attitude of the aircraft is read by comparing the position of the band index with the fixed bank scale.

Glide Slope Displacement Pointer

The glide slope displacement pointer is driven vertically along the G/S deviation scale by signals from the VER NAV computer or the G/S receiver. The center of the G/S beam or the VER NAV flight path is represented by the pointer. Graduations on the G/S deviation scale represent specific increments of deviation of displacement.

Bank Steering Bar

The bank steering bar (vertical pointer) is driven laterally across the face of the instrument by signals from the flight director computer. Movement of the bank steering bar away from the center of the aircraft symbol represents lateral steering commands. The pilot is directed to fly the aircraft toward the deflected pointer to satisfy the command.

Pitch Steering Bar

The pitch steering bar (horizontal pointer) is driven vertically up and down the face of the instrument by signals from the flight director computer. Movement of the pointer away from the center of the aircraft symbol represents vertical steering commands. The pilot is directed to fly the aircraft toward the deflected pointer to satisfy the command.

Rate-of-Turn Indicator

The rate-of-turn indicator is driven against the rate-of-turn scale at the bottom of the ADI by signals from the rate transmitter. Displacement of the pointer from center indicates rate-of-turn in degrees-per-minute or in minutes per 360-degree turn.

Slip Indicator

The slip indicator consists of a weighted ball contained in a liquid-filled transparent tube. An uncoordinated turn is indicated by lateral displacement of the ball.

Pitch Trim Knob

The pitch trim knob provides a means of adjusting the pitch reference (miniature aircraft) of the indicator to coincide with the desired flight attitude of the aircraft.

The vertical pointer flag (when in view) represents loss of validity from the guidance system supplying information to the computer or an invalid signal from the flight director computer or TPLC.

Displacement Pointer Flag

The displacement pointer flag indicates loss of G/S reliability.

Radar Altitude Pointer

The radar altitude pointer has a 200-foot range. At zero feet (when the aircraft wheels touch the runway), the pointer should touch the wheels of the miniature aircraft. This pointer is biased out of view above 200 feet and/or when altimeter information becomes unreliable.

ADI Adjustments

ADI adjustments include four meter movement zeroing-type adjustment devices located on the rear of the indicator. The ADI amplifier must be removed to gain access to them. These adjustments are for the vertical steering pointer, horizontal steering pointer, displacement pointer, and rate-of-turn pointer. They are used to center the meter movements during power-off conditions. ADI amplifier adjustments are provided on the side of the amplifier. These adjustments vary an input bias to the altitude indicator amplifier and to the roll servo amplifier to ensure proper zeroing of the altitude pointer and ADI sphere.

Enroute Guidance Systems

Certain of the aircraft guidance systems are utilized only during enroute flight and not during landings. These are the Tacan, VOR, ASN-24, ASN-35, compass, VER NAV, and LOC prior to glide slope intercept.

The basic method flight with the FDS is Manual Heading (MH). This mode is initiated by positioning the HDG SELECT/NAV switch to "NAV." In this mode the FDS commands a heading marker on the compass card. When on the desired heading, the heading marker is beneath the lubber line and the ADI vertical pointer is centered.

Manual Heading Presentation (Typical)

In the example shown in Figure 7-3, the aircraft is heading north and the selected heading is 060 degrees, which represents a 60-degree heading error. The steering command to correct for this error is shown in position No. 1

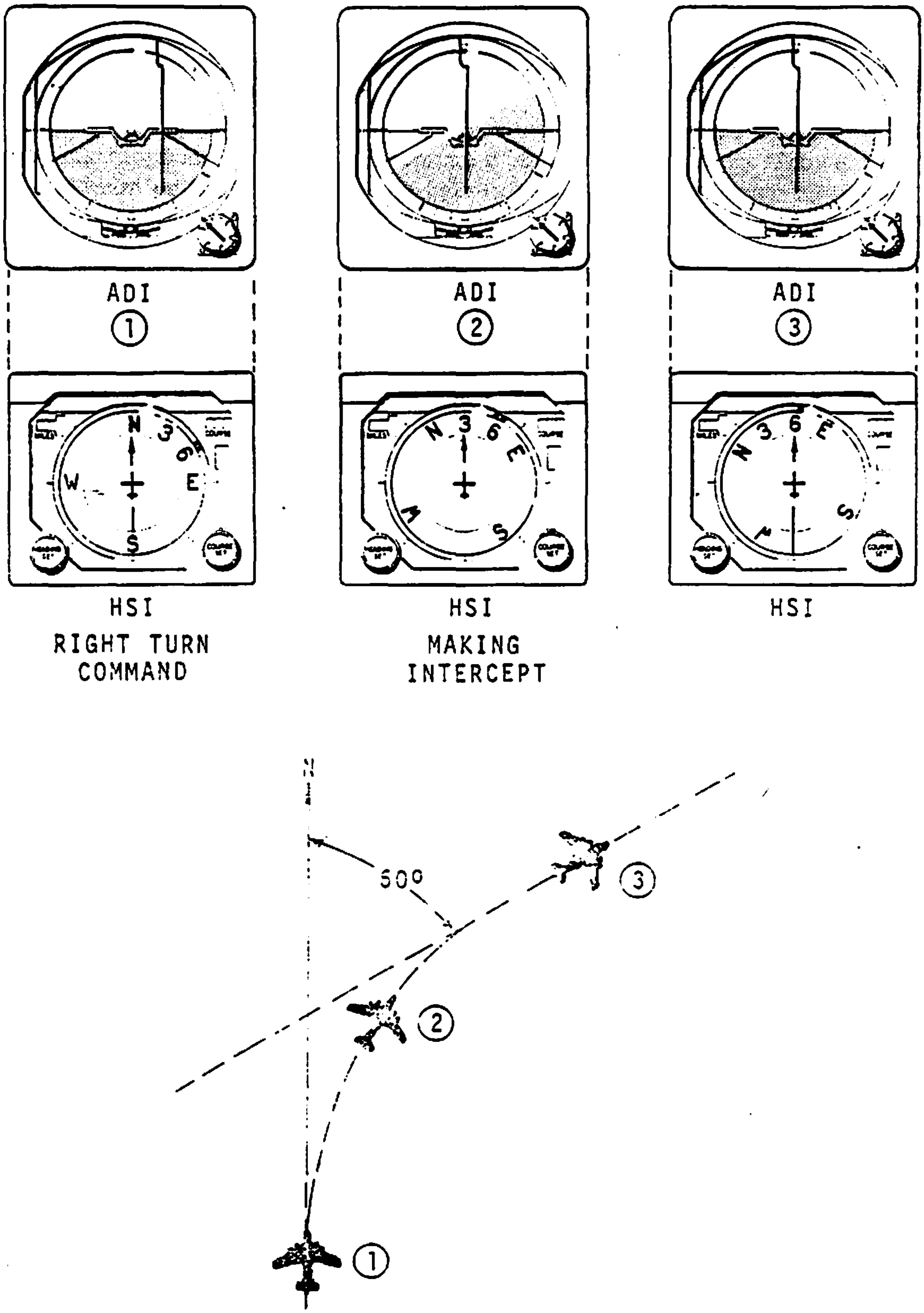


FIGURE 7-3. MANUAL HEADING PRESENTATION

(ADI pointer to right). In position No. 2, a proper turn of correct magnitude to bring the aircraft onto the desired heading has caused the ADI pointer to center. The aircraft has rolled out on course in position No. 3. The ADI pointer and HSI heading marker are centered on their respective indicators. After rolling out on the desired heading, only minor corrections should be necessary to keep the aircraft on the desired heading.

VOR/Tacan Flight

Since VOR and Tacan flight are identical with the exception of the magnitude of radio deviation necessary to initiate capture, only VOR flight is discussed. The VOR capture initiation point is a fixed amount of deviation (75 millivolt), while Tacan capture point varies inversely with distance to the station (150 millivolts at 50 miles or less, to 50 millivolts at 150 miles or more).

When initiating a VOR course, the desired course (radial) is selected with the COURSE SET knob on the HSI. The desired heading to intercept this course is set with the HEADING SET knob. The mode (Tacan or VOR) is then selected on the navigation selector panel. The HDG SELECT/NAV switch is placed in "NAV" and the FDS logic switches to automatic manual heading mode. The FDS commands flight on the selected heading until the aircraft is within the threshold of radio deviation required for capture. The FDS then commands a course cut intercept of the desired radial and provides steering for an asymptotic intercept and roll-out on the desired radial, as shown in Figure 7-4. In position No. 1 the aircraft is maintaining a heading of 350 degrees. The ADI pointer is centered since the steering commands are satisfied. A right steering command shown in position No. 2 is the required capture steering maneuver. In this position, radio deviation has dropped to the capture threshold. Position No. 3 shows the aircraft banking right satisfying the steering command. Rolled-out and on-course are shown in position No. 4. In this condition, the FDS is tracking the radio beam. Cross-wind compensation is provided. Any steady state course errors are washed-out and only radio deviation errors and course and roll rate signals cause course corrections. In position No. 5, the aircraft is turned into the wind (crabbed) to maintain beam tracking. The course error (deviation bar deflected) displayed on the HSI is washed-out since it is a steady-state error. The ADI pointer is centered since no errors are being supplied to it, i. e. the aircraft is on the radio beam and the course error is washed out.

VOR/TAC approach mode is initiated by placing the VOR/TAC APPR/NORM switch to "APPR." This mode is a tracking mode and can be switched in only after on-course (normally only within 50 miles of ground station). The displayed information is the same as in VOR/TAC mode except gains are optimized for close low-speed tracking of the radio beam.

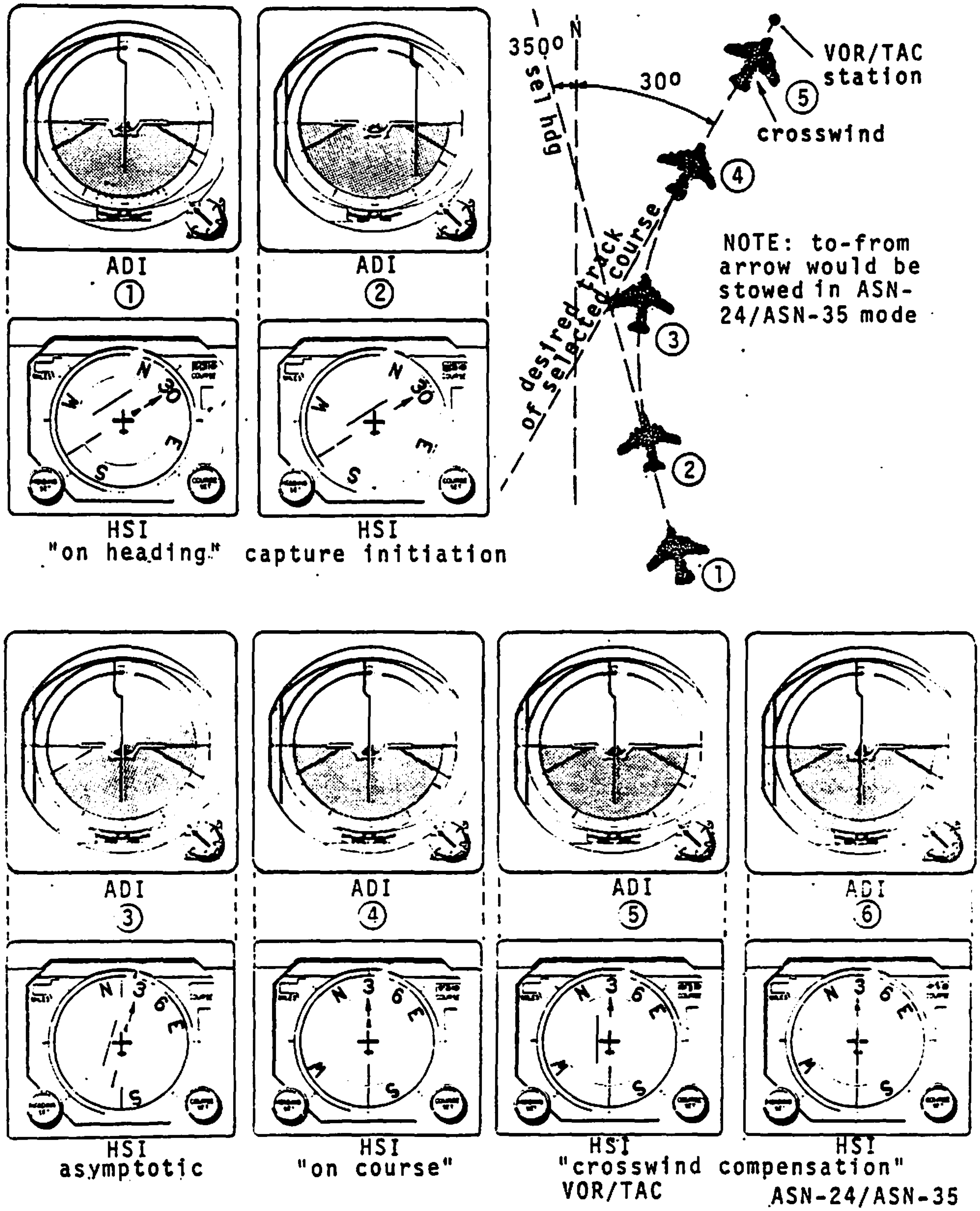


FIGURE 7-4. VOR/TAC ASN-24/35 PRESENTATION

Computers: NAV (ASN-24) and DOP (ASN-35)

Since navigation and doppler computer modes are basically the same, only doppler mode will be discussed, with the differences noted.

The computer modes are similar to the VOR/TAC modes in that the computers simulate a radio beam, i. e. the desired flight path or track is seen by the FDS as a radio signal. The computer mode utilizes automatic manual heading to approach the desired track, a capture mode to intercept the desired track, and a tracking mode to maintain the desired track. Computer crosstrack (distance from the desired track) to the FDS appears the same as radio deviation. The capture initiation point is a constant value (150 millivolt).

ASN-24/ASN-35/VOR/TAC Flight - As shown in Figure 7-4, the desired track is along the 30-degree radial. However, the radial has no bearing on the desired track. Desired track is determined only by data stored in the computers. In position No. 1, the aircraft is flying the desired intercept heading of 350 degrees. At position No. 2 the crosstrack has decreased to capture level, and the FDS has commanded a course cut on the ADI. At position No. 3 the aircraft is banked and is satisfying the command to turn onto the desired track. Position No. 4 shows the aircraft rolled-out and on course. In position No. 5, the aircraft is crabbing in a crosswind. The difference in position No. 5 for VOR/TAC and position No. 5 for ASN-24/ASN-35 should be noted. The deviation bar is deflected in position No. 5 for VOR/TAC indicating a course error. However, in position No. 5 for ASN-24/ASN-35, the bar (representing crosstrack deviation) is centered.

o Doppler ASN-35 - The doppler ASN-35 computer mode has two additional FDS submodes. They are paradrop and high-speed paradrop and should not be used until after capture. These modes are used when it is desired to maintain tracking much closer to the desired track than is possible with normal ASN-35. When these submodes are selected, the normal scale factor of deviation (3 miles crosstrack = 150 millivolts) is switched to a factor 10 times greater (0.3 mile crosstrack = 150 millivolts). The only difference in the two modes (paradrop and high-speed paradrop) is the speed of the aircraft. If the aircraft is flying faster than 190 knots, the FDS is switched into high-speed paradrop. This action causes gains to be reduced within the FDS so that ADI pointer movements are not erratic and difficult for the pilots to follow. Below 190 knots, the FDS is a normal paradrop mode. Paradrop is selected when the ASN-35 is tracking the desired course by setting the NAV/X10 switch on the doppler auxiliary control panel to "X10."

The C-141A aircraft is restricted to an airspeed of 190 knots when the flaps are not full up. A flap position transmitter senses flap position and this signal is used to switch the FDS to high-speed paradrop when paradrop mode is selected and flaps are up.

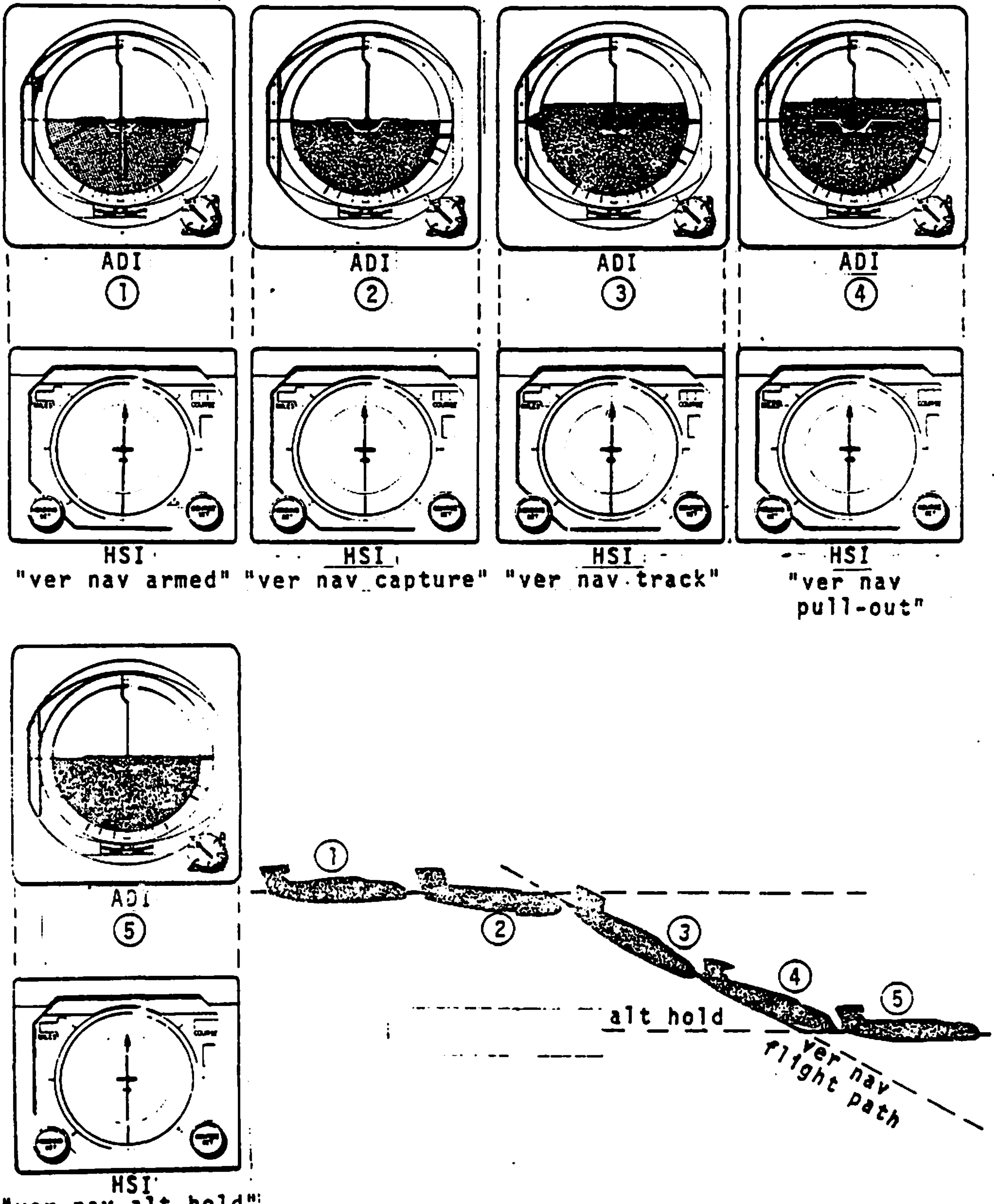


FIGURE 7-5. VER NAV PRESENTATION

VER NAV Mode

The VER NAV computer, which is completely described in Chapter 3, supplies the FDS with signals to perform vertical navigation of the aircraft concurrently with any lateral radio mode (except normal ASN-35 or heading select). There are three sub-fractions in VER NAV mode: VER NAV ARMED, VER NAV capture and track, and VER NAV altitude hold.

When the switch on the VER NAV control panel is turned from "STBY to "STAGE I" or "STAGE II," the VER NAV light on the navigation selector panel illuminates, which is the armed phase of VER NAV.

The VER NAV flight path may be thought of as a high altitude G/S since essentially that is what the FDS sees.

In the armed state, the G/S displacement pointer displays VER NAV error or deviation from the VER NAV flight path. The horizontal steering pointer is out of view until initiation of VER NAV capture. At this point the horizontal pointer comes into view and gives a fly-down command to place the aircraft on the VER NAV path. After flying the path to the desired altitude, the VER NAV computer commands pull-out and goes to altitude hold. These commands are fed to the horizontal pointer and maintain the aircraft at the altitude commanded by the VER NAV computer until capture of the next VER NAV stage or G/S intercept.

In figure 7-5, position No. 1, the presentation during VER NAV arm is shown. The displacement pointer is on scale and showing deviation from VER NAV path. At position No. 2, fly-down command is given to capture the path. At No. 3, the VER NAV path is being tracked. At No. 4, pull-out altitude is reached and a command is being given to arrest the descent at the desired altitude. At position No. 5, the horizontal pointer is displaying commands to hold the aircraft at the commanded altitude. If an ILS frequency is tuned and VOR ILS selected on the navigation selector panel, VER NAV is switched out.

Localizer (LOC)

Prior to G/S intercept, desensitized localizer deviation from the A/P coupler is utilized by the FDS. Thirty seconds after G/S engage, the preland test program should be completed. At this point, the AA light on the progress display panel illuminates, and an AA signal is applied to the FDS. This signal causes the FDS to switch from the desensitized localizer to raw localizer directly from the receiver. Localizer mode incorporates an automatic manual heading mode, capture mode, and track mode. The manual heading mode flies the aircraft on an intercept heading. When radio deviation decreases to capture level, the FDS commands a course cut intercept of the localizer beam. After capture, the FDS commands roll-out and on-course tracking. The ADI and HSI presentations are the same as VOR mode except there is no bearing pointer or to-from information in the localizer mode.

Glide Slope (G/S)

The FDS utilizes only desensitized G/S deviation from the A/P coupler to compute steering information. Raw localizer is displayed on the G/S deviation pointer. The displacement flag is connected to the G/S reliability flag during G/S mode prior to flare engage.

ILS Guidance Systems

Certain aircraft guidance and control systems are utilized only during landings. During an AWLS approach, the following systems may be utilized: ILS system, flare computer, R/GA, and A/P.

During a normal AWLS approach, only the ILS system and R/GA are fully active.

AWLS Approach

During a normal AWLS approach, the AFCS provides steering control for the aircraft. The pilot monitors his FDS indicators to assure that the aircraft is being maintained on the proper flight path. The copilot monitors his ADI which is displaying steering commands to the A/P. The horizontal steering pointer displays A/P pitch channel steering commands and the vertical steering pointer displays A/P lateral channel steering commands. If either A/P axis is disengaged, the steering pointer begins to display whatever mode the FDS would have been in if A/P had not been engaged. If the A/P is not engaged, the copilot's presentation is the same mode as the pilot's.

The ILS provides pitch and lateral guidance to maintain the aircraft on the desired flight path for a Category II approach, i. e. to deliver the aircraft to 100-foot altitude with one-quarter mile visibility and runway in sight.

An AWLS ILS landing begins with the FDS commanding a manual heading localizer intercept course. When radio deviation decreases to capture level (150 millivolts), the FDS commands a course-cut intercept of the localizer beam. The pilot's steering commands roll the aircraft out on course and the FDS begins tracking localizer. The FDS continues tracking localizer while G/S deviation is steadily decreasing, and the G/S deviation pointer is moving steadily downward. When G/S deviation has decreased to the G/S capture level, the horizontal steering pointer comes into view and a 1.7-degree positive bias is fed into the pitch channel. This action causes the horizontal steering pointer to command a steer down attitude and G/S tracking begins. The A/P steering commands are displayed on the copilot's ADI after AA if the A/P is engaged. If for some reason the A/P is not engaged, both FDS's operate in the same mode and their presentations should be identical. The FDS's continue tracking the localizer and G/S signals until flare engage.

The flare computer provides commands to the A/P and to both FDS's to arrest the sink rate of the aircraft to a rate at which the aircraft is not damaged at touchdown, which is done by commanding an exponential flare maneuver, i. e. a nose-up attitude to provide added lift to the aircraft and to allow ballooning to occur. This ballooning lets the aircraft settle or touchdown at a sink rate within the structural design limits of the aircraft.

At flare engage, the FDS is switched to flare mode. The horizontal steering pointer displays flare error, and the displacement pointer and flag are biased out of view. Localizer error continues to be displayed on the vertical steering pointer. Flare engage normally occurs at 45-foot radar altitude.

At 15-foot radar altitude, the pilot must manually decrab (align the aircraft with the runway) the aircraft and touchdown.

A typical AWLS approach is illustrated in Figure 7-6. In position No. 1, the aircraft is maintaining an intercept manual heading of 60 degrees. At position No. 2, localizer deviation has decreased to a capture initiation level, and the FDS has commanded a course cut intercept. Position No. 3 shows the aircraft properly banked to satisfy the steering command. Position No. 4 shows the steering command satisfied and the aircraft rolled out on course. Position No. 5 shows localizer commands satisfied and a fly-down command to capture the G/S beam. The aircraft is pitched down and tracking the G/S beam in position No. 6. Flare engage has occurred at position No. 7, and the horizontal steering pointer is commanding a pitch up attitude to execute the required exponential flare maneuver. Position No. 8 shows the flare command satisfied (horizontal pointer centered) and the aircraft in the flared attitude.

NOTE

Displacement pointer and flag are biased out of view in positions No. 7 and 8. It would be impossible to predict autopilot steering commands, so the indicators shown depict a normal AWLS FDS display with the A/P engaged.

<u>Non-AWLS Approach</u>

During a non-AWLS approach, the visibility limits for landing are greater than in an AWLS approach (Category I instead of Category II). Category I limits are

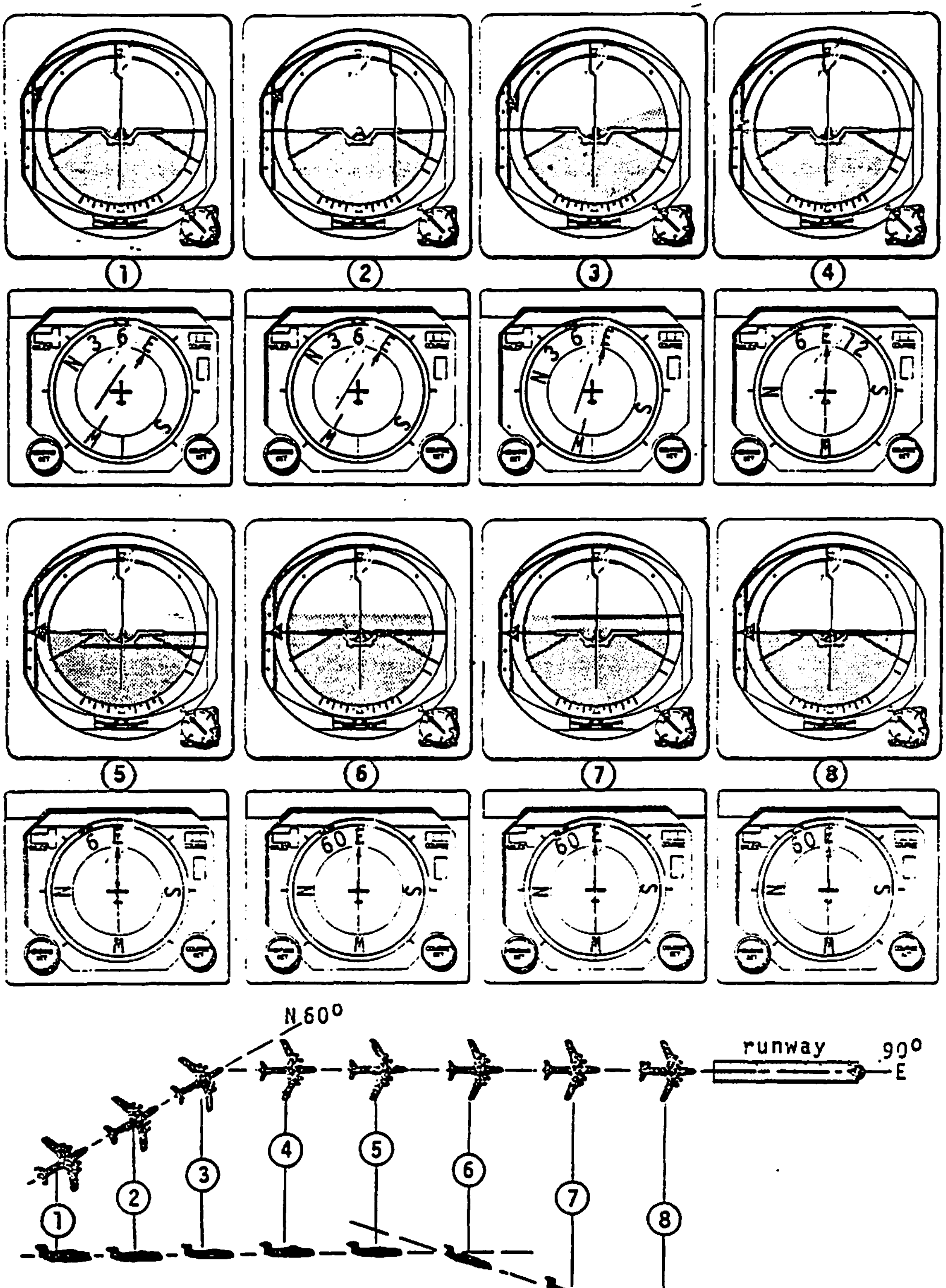


FIGURE 7-6. AWLS LANDING DISPLAYS

200-foot altitude and one-half mile visibility. A non-AWLS approach or conventional approach begins with a localizer intercept manual heading. When radio deviation drops to capture initiation level (150 millivolts), capture is accomplished, and tracking begins just as it would in an AWLS approach. There are no localizer switching or land arm signals. The computers (FDS) utilize desensitized localizer for the entire approach. The pilot must manually flare and manually decrab the aircraft in a non-AWLS approach.

Figure 7-7 illustrates a typical non-AWLS approach. The indicator displays and aircraft positions are identical to those in Figure 7-6 except there is no flare maneuver commanded by the FDS.

Rotation/Go-Around (R/GA) Mode

The R/GA computer provides pitch steering commands to the flight director computer. When the pilot makes the decision to abort the landing, the R/GA computer provides optimum pitch steering commands to arrest the aircraft descent and climb-out. This steering information is necessary as the aircraft altitude could be as low as 100 feet when the abort decision is made.

In R/GA mode, the horizontal steering pointer displays angle-of-attack error, and the vertical steering pointer displays a wings level command. The G/S displacement pointer may be in view displaying G/S deviation.

NOTE

Selection of the R/GA mode overrides all vertical modes and all lateral modes except heading select.

When R/GA is selected, the GO-AROUND light on the progress display panel illuminates. ADI display, modes, and navigation selector panel switch functions would be as shown in Figure 7-8.

Self Test (ST)

During self test operation, known signals are injected into the FDS signal paths. The computer is forced into ILS mode by logic switching. When the signals have washed out in the pitch and lateral channels, the steering pointer driver outputs should be at null. These outputs are sensed, and, if not at null, cause the test to fail. Any voltage failutes also cause the test to fail.

When the test button is depressed, the steering pointers initially deflect away from center; 4 or 5 seconds later the error signals have washed out, the

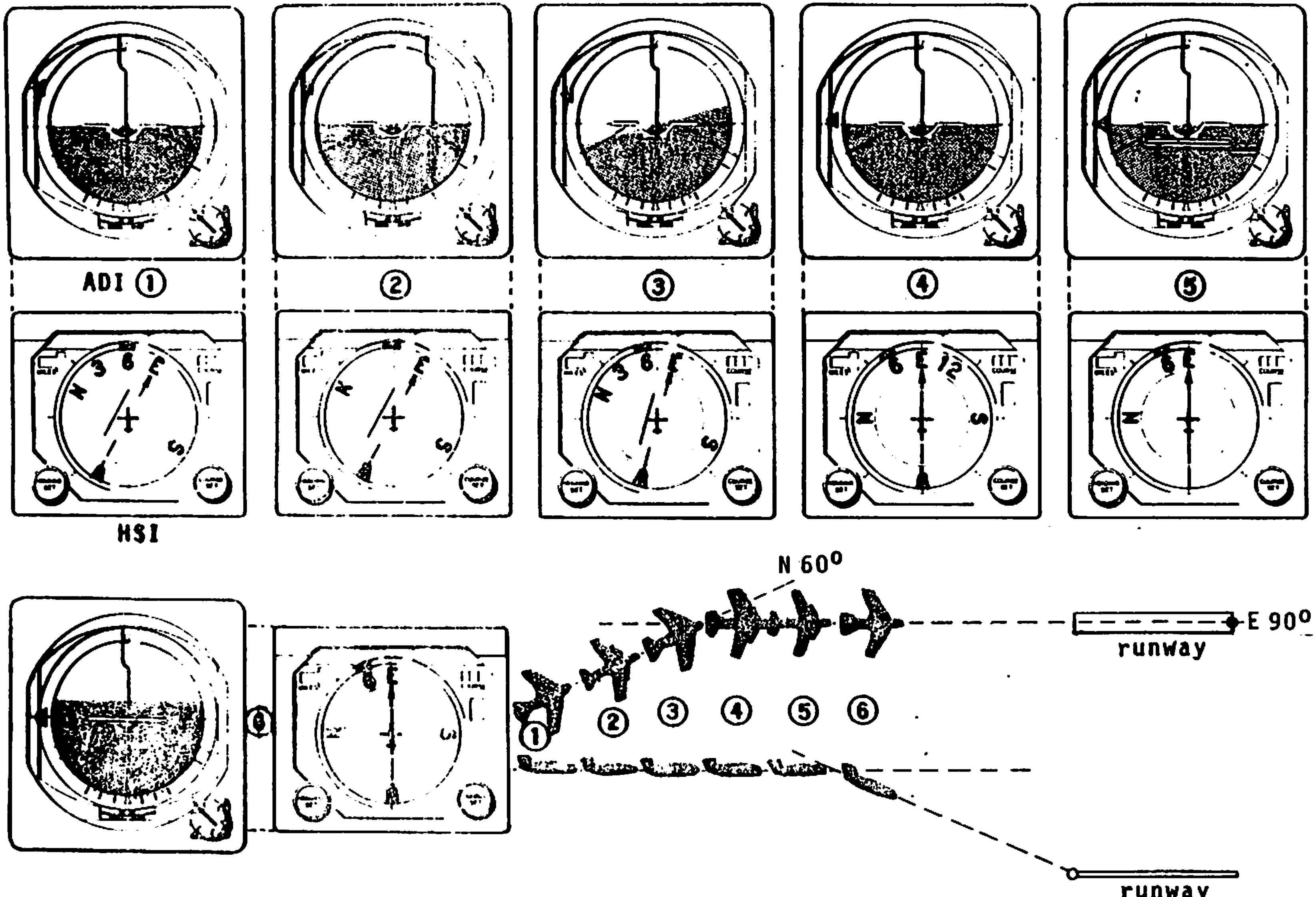


FIGURE 7-7. NON-AWLS LANDING DISPLAYS

MODE	FUNCTION	NAV SEL PANEL	OTHER MODE EXCITATION
1. Manual Heading	Select Heading Hold Automatic Heading	Select "HDG" Select "NAV"	None Selected Nav Mode Outside Lat. Beam Sense
2. VOR	Capture Track	Select "NAV," VOR/TAC IN "NORM," "VOR/ILS" IN	Inside VOR Lat. Beam Sense VOR On-Course Sense
3. TACAN	Capture Track	Select "NAV," VOR/TAC IN "NORM," "TAC" IN	Inside TACAN Lat. Beam Sense TACAN On-Course Sense
4. VOR TAC Approach	Low-speed/low alt. Close-in Approach	Select "NAV," VOR/TAC IN "APPR," "VOR," or "TAC" IN	VOR or TACAN On-Course Sense
5. ASN-24	Capture Track	Select "NAV" "ASN-24" IN	NAV Cmptr "NOT" On-Course ⚠5 NAV Cmptr On-Course Sense
6. ASN-35 (Normal)	Capture Track	Select "NAV" "ASN-35" IN	Inside NAV Cmptr Beam Sense NAV Comptr On-Course Sense
7. PARADROP (ASN-35)	High-Sensitivity Low-Speed Track	Select "NAV" "ASN-35" IN	ASN-35 in "DROP," Flaps, NAV Cmptr On-Course Sense
8. HIGH-SPEED PARADROP (ASN-35)	High-Sensitivity High-Speed Track	Select "NAV" "ASN-35" IN	ASN-35 in "DROP," No Flaps NAV Cmptr On-Course Sense
9. ILS (LOCALIZER)	Capture Track	Select "NAV" "VOR/ILS" IN	Loc Freq. Inside Beam Sense Loc Freq. On-Course Sense
10. ILS APPROACH	Tracking of ILS Beams	Select "NAV" "VOR/ILS" IN	ILS Mode Engaged Vertical Beam Sense
11. AUTOPILOT	Autopilot Command Monitor	Select "NAV" "VOR/ILS" IN	A/P Engaged, AWLS Armed, "APPROACH ARM"
12. FLARE	Landing Flare	Select "NAV" "VOR/ILS" IN	Flare Engage from TPLC, AWLS Armed
13. ROTATION/GO-AROUND	Go-Around or Take-Off Climbout	Select "NAV" Select "HDG"	R/G-A Mode Signal from TPLC R/G-A Mode Signal from TPLC
14. VERNAV	Armed Capture/Track Altitude Hold	Select "HDG" or "NAV" Do Not Select ILS Loc. or "ASN-35" Normal	VERNAV Mode Signal VERNAV and VERNAV Capture Mode Signals
15. NAV-OFF	Attitude Reference	Select "NAV" "NAV-OFF" IN	No Mode Signals
16. SELF-TEST	Flight Director Self-Test	Select "HDG" ⚠	Self-test Mode Signal
17. AWLS TEST	AWLS Enroute System Test	Select "HDG" ⚠	AWLS Enroute Test Mode Signal
18. ADI REP	Display Opposite Computer Outputs	"ADI REP" IN	None

Continued

NOTES:

1. May be active in Rotation/Go-Around Mode
2. May be active in VERNAV MODE
3. May be active in ILS Approach (Glideslope) Mode
4. VERNAV Mode usable only in lateral tracking mode
5. ASN-24 NAV Mode always inside Lateral Beam Sense
6. Raw glideslope information may be in view during Rotation/Go-Around Mode
7. Conduct test in Heading Select Mode to assure proper resetting following test
8. Display depends on inputs and mode of opposite flight director

FIGURE 7-8. FLIGHT DIRECTOR MODES AND DISPLAYS

MODE	ATTITUDE DIRECTOR INDICATOR DISPLAYS				
	BANK STRG. BAR	LATERAL STRG. FLAG	PITCH STRG. BAR	DISPLACEMENT PTR.	DISPL. PTR. FLAG
1. Manual Heading	Heading Strg. Heading Strg.	Active Active	△1 △7 △2	△2 △7 △2 △1	△1 △1
2. VOR	Capture Strg. Track Strg.	Active Active	Masked △4 △2	Masked △4 △2	Masked & Inactive Masked & Inactive
3. TACAN	Capture Strg. Track Strg.	Active Active	Masked △4 △2	Masked △4 △2	Masked & Inactive Masked & Inactive
4. VOR/TAC Approach	Track Strg.	Active	△2	△2	Masked & Inactive
5. ASN-24	Capture Strg. Track Strg.	Active Active	Masked △4 △2	Masked △4 △2	Masked & Inactive Masked & Inactive
6. ASN-35 (Normal)	Capture Strg. Track Strg.	Active Active	Masked △4 Masked	Masked △4 Masked	Masked & Inactive Masked & Inactive
7. PARADROP (ASN-35)	Track Strg.	Active	△2	△2	Masked & Inactive
8. HIGH-SPEED PARADROP (ASN-35)	Track Strg.	Active	△2	△2	Masked & Inactive
9. ILS (LOCALIZER)	Capture Strg. Track Strg.	Active Active	Masked Masked	G/S Deviation G/S Deviation	Active Active
10. ILS APPROACH	Loc Trk Strg.	Active	G/S Steering	G/S Deviation	Active
11. AUTOPILOT	A/P Roll Strg. Err.	Active	A/P Pitch Strg. Er.	G/S Deviation	Active
12. FLARE	Loc Track Strg.	Active	Flare Strg.	Masked	Masked & Inactive
13. ROTATION/GO-AROUND	Wings Level Cmd. Heading Strg.	Active Active	R/G-A Strg. R/G-A Strg.	△6 △6	△6 △6
14. VERNAV	Selected Lateral Track Steering	Active	Masked VERNAV Strg. Alt. Hold Strg.	VERNAV Error	Masked & Inactive
15. NAV-OFF	Masked	Masked & Inactive	Masked	Masked	Masked & Inactive
16. SELF-TEST	Test Bank Cmd.	Masked & Inactive	Test Pitch Cmd.	In View Centered	Masked & Inactive
17. AWLS TEST	Test Bank Cmd.	Masked & Inactive	Test Pitch Cmd.	In View Centered	Masked & Inactive
18. ADI REP	△8	△8	△8	△8	△8

NOTES:

△1 May be active in Rotation/Go-Around Mode
△2 May be active in VERNAV MODE
△3 May be active in ILS Approach (Glideslope) Mode
△4 VERNAV Mode usable only in lateral tracking mode
△5 ASN-24 NAV Mode always inside Lateral Beam Sense
△6 Raw glideslope information may be in view during Rotation/Go-Around Mode
△7 Conduct test in Heading Select Mode to assure proper resetting following test
△8 Display depends on inputs and mode of opposite flight director

FIGURE 7-8. FLIGHT DIRECTOR MODES AND DISPLAYS (CONTINUED) pointers are centered, and the flags are out of view. The test-in-progress light illuminates when the button is depressed. It remains illuminated until the test is complete. At the end of 7 seconds, if the test is completed successfully, the test-in-progress light goes out, the test complete (reset) light comes on, and no fault lights should be illuminated. If the test is not successful, the test complete light will not illuminate and the appropriate FDS fault light will illuminate.

Fault Light

Fault lights identified as FLT DIR 1 and FLT DIR 2 are the lights mentioned above. Loss of voltage validities within the computer is the only malfunction which turns on a fault light.

In the ADI repeat mode, it is possible to display No. 2 computer information on No. 1 ADI, and No. 1 computer information on No. 2 ADI. The ADI REP buttons on the navigation selector panels determine the switching.

NOTE

If a FDS test is run with ADI REP selected, the opposite ADI will be exercised.

NAV OFF Mode

The NAV OFF mode, initiated by depressing the NAV OFF button on the navigation selector panel, removes navigation information from the ADI. Only R/GA or heading select mode signals can be applied in NAV OFF mode. In this mode, the ADI continues to supply attitude, rate-of-turn, and turn coordination information. The bearing pointer on the HSI drives to the bottom of the lubber line. The course arrow is slaved to compass heading and remains at the upper lubber line. A digital readout of compass heading is provided.

HSI Slaving

In order for the pilot to have control of the A/P preset heading and preset course, the No. 2 HSI heading and course set are slaved to the No. 1 HSI, i. e. when the pilot positions his heading marker or course arrow, the copilot's HSI follows to the same position. This capability is provided by switching the HSI No. 2 HEADING and COURSE switch on the heading and course set panel to the "SLAVE" position. Selecting VOR/ILS on both navigation selector panels and tuning both VOR/ILS receivers to localizer frequencies accomplishes slaving automatically.

Summary of Modes Operation

The FDS provides an integrated display of navigation and other flight information required for control of the aircraft. Flight information displayed consists of aircraft attitude displayed on the attitude sphere of the ADI, rate-of-turn displayed on the rate-of-turn pointer of the ADI, turn coordination information displayed on the slip, and skid indicator on the ADI.

The other flight information and the navigation information consists of signals developed within the FDS from data furnished by Tacan, VOR/ILS, ASN-34, ASN-35 systems; compass; R/GA; flare; VER NAV; and A/P system. The above systems data may be summed with, or complimented by, other information: pitch rate from the two-axis rate gyros; flap position from the flap position transmitters; roll and pitch attitude from the attitude gyros and from the ISS in the TPLC; vertical acceleration from the normal accelerometers; and elevator position from the elevator position transmitter. The resultant information may then be displayed on the FDS indicators.

The information displayed is determined by the switching of the navigation selector panel accomplished manually by the pilots or by automatic switching performed externally by the TPLC or the appropriate AWLS subsystem.

Mode Priority

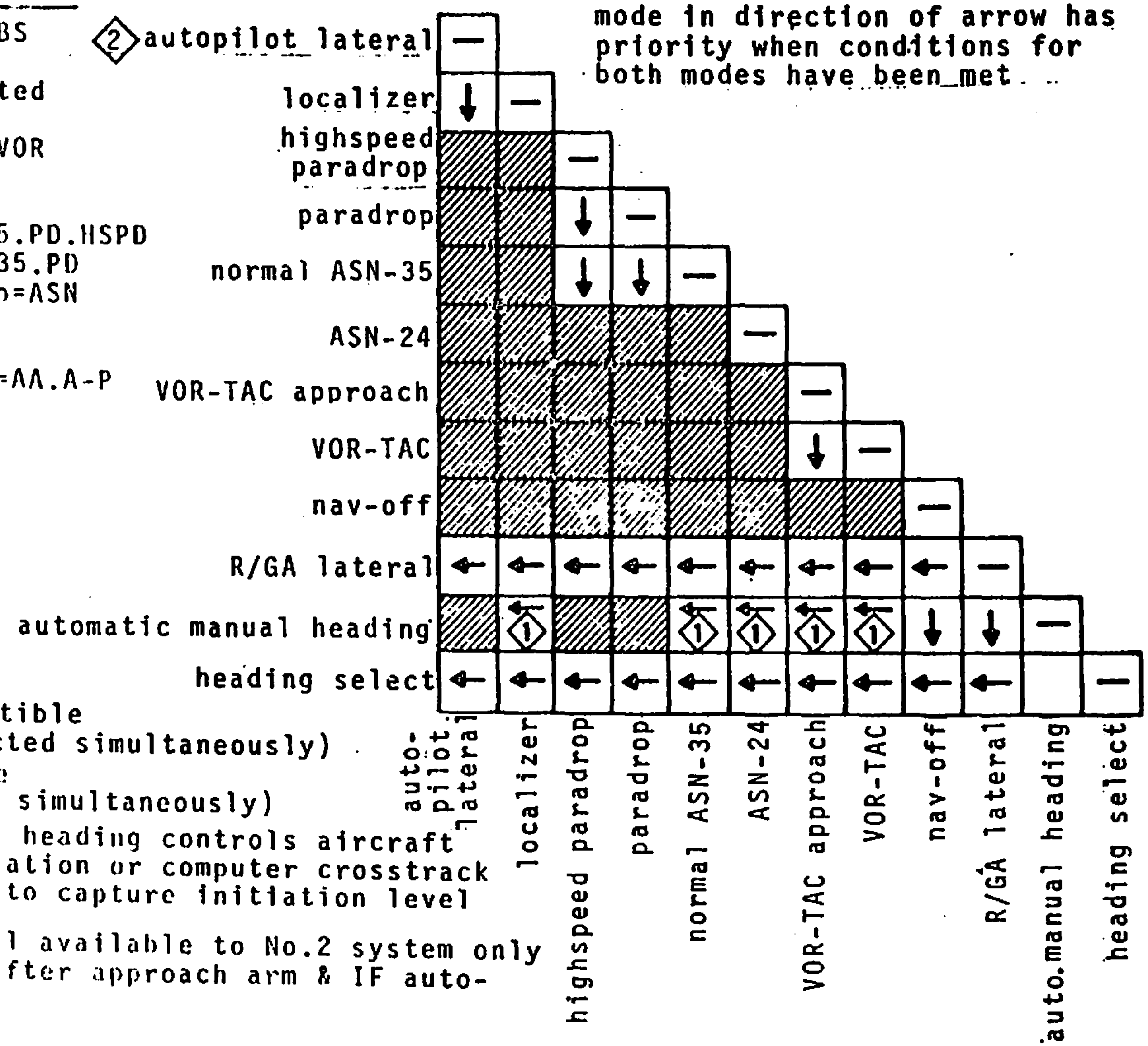
A complete tabulation of modes and priorities is provided in Figure 7-9 for lateral channel and Figure 7-10 for vertical channel.

Heading Select - This mode overrides all lateral channel modes and provides the vertical steering pointer with commands to maintain a preset heading. It is selected by moving the HDG SELECT/NAV switch to "HDG SELECT."

VOR, Tacan, ASN-24 ASN-35 - These modes are all called radio modes. None of them, or the NAV OFF mode, may be selected at the same time since the mode push-buttons on the navigation selector panel are mechanically interlocked. When any one is pushed in and another one is pushed in, the first one pops out. In the radio modes, a radio beam (real or artificially produced by the computers) is tracked by the FDS. Each of these modes consists of an automatic manual heading, capture, and track mode. VOR and Tacan have a submode (VOR/TAC APPROACH) which optimizes gains for close tracking of the beam after capture. It is initiated by switching the VOR/TAC APPR/NORM switch on the navigation selector panel to "APPR." The ASN-35 has two submodes which provide close tracking of the desired track. These are paradrop and high-speed paradrop. The paradrop is initiated by switching the NAV/X10 switch on the doppler auxiliary control to "X10." The system is in paradrop below 190 knots and high-speed paradrop above 190 knots.

COMPUTER LOGIC EQUATIONS

- auto. MH=RT+ILS.LBS
- HS=HS
- R/GAL=R/GAL selected
- VOR-TAC=VOR+TAC
- VOR-TAC approach=VOR +TAC.VOR-TAC app
- ASN-24=ASN-24
- norm ASN-35=ASN-35.PD.HSPD
- paradrop(PD)=ASN-35.PD
- highspeed paradrop=ASN -35.PD.HSPD
- loc= loc
- autopilot lateral=AA.A-P engage



modes not compatible (cannot be selected simultaneously)

modes compatible (can be selected simultaneously)

(1) automatic manual heading controls aircraft until radio deviation or computer crosstrack error decreases to capture initiation level

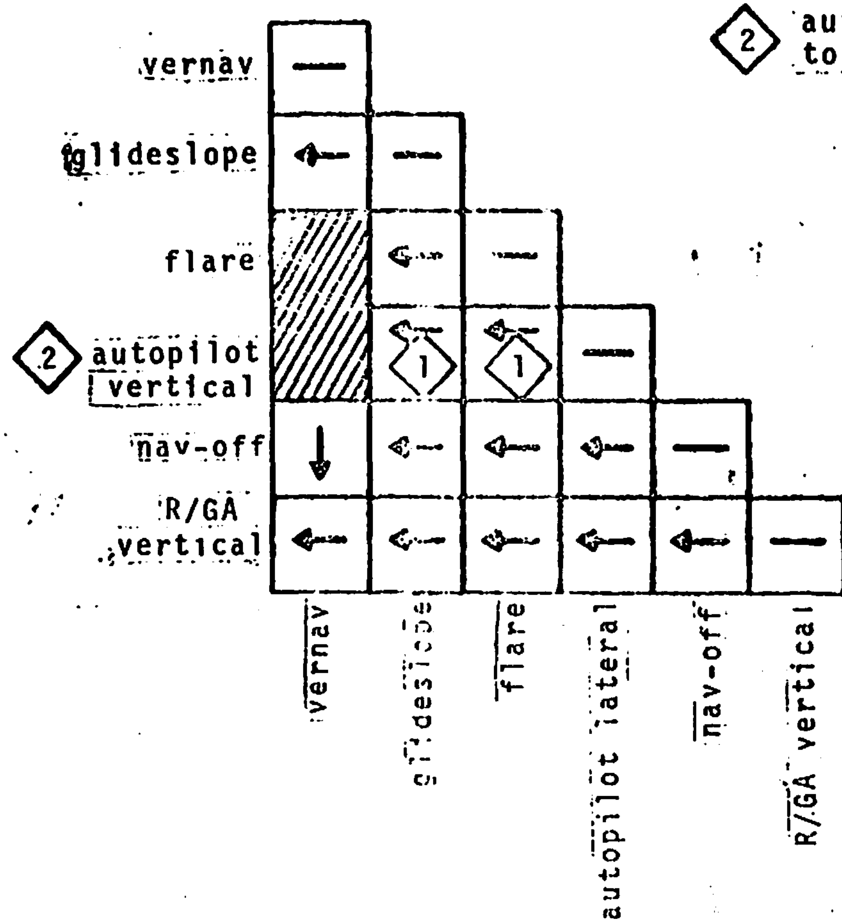
(2) autopilot lateral available to No.2 system only (available only after approach arm & IF auto-pilot engaged)

FIGURE 7-9. LATERAL CHANNEL MODE PRIORITY modes not compatible

1 autopilot vertical available only after approach arm & IF autopilot engaged

2 autopilot vertical available to No.2 system only

NOTE: mode in direction of arrow has priority when conditions for both modes have been met



COMPUTER LOGIC EQUATIONS

VVN=VN.ILS

FL=FE.ILS

APV=AA.AP engage

R/GA V=R/GA selected

FIGURE 7-10. VERTICAL CHANNEL MODE PRIORITY

The HSI bearing pointer indicates bearing to the station in VOR or Tacan and TAE in ASN-24 or ASN-35 mode. The course deviation bar indicates degrees off course in VOR/TAC and miles off the desired track (crosstrack) when in ASN-24 or ASN-35 modes.

Rotation/Go-Around (R/GA) - The R/GA mode is activated by depressing the go-around button on either the pilot's or copilot's control wheel. This mode overrides all vertical modes and all lateral modes except heading select. Angle-of-attack error and wings level commands are displayed in this mode.

ILS - The ILS mode is initiated by selecting VOR/ILS on the navigation selector panel and tuning the VOR/ILS system to a localizer frequency. Localizer utilizes an intercept heading, capture, and track modes similar to other radio modes. G/S utilizes a capture and a track mode. In G/S mode, the vertical and horizontal steering pointers provide guidance to the runway. Raw localizer is displayed on the G/S displacement pointer.

Flare - The flare mode is initiated automatically at 45-foot radar altitude in an AWLS approach. Flare supplies vertical velocity error to the horizontal steering pointer and biases the displacement pointer and flag out of view. Localizer is still active in the lateral channel.

VER NAV - The VER NAV mode is initiated automatically by a VER NAV capture signal from the VER NAV computer when the aircraft is sufficiently close to the VER NAV path. In this mode, the horizontal steering pointer displays VER NAV steering, and the displacement pointer displays VER NAV error. The vertical pointer displays whatever lateral channel mode is selected concurrently with VER NAV.

If the A/P is engaged, the flight director automatically goes into the A/P mode at the AA point. In this mode, the horizontal steering and vertical steering pointers monitor the A/P pitch and roll steering commands, respectively. The G/S deviation pointer and flag display deviation and reliability respectively in this mode.

THEORY OF OPERATION

Although, as previously mentioned, C-141A aircraft contain two complete flight director systems, only one is discussed here since both systems are the same. The following system diagram illustrates the inter-relationship of the various flight director components and signal sources. By breaking this diagram down into its various information loops, the following can be noted:

- Heading information is supplied from the C-12 compass. The signals are amplified and used to position the servo-driven compass card. Heading information is also applied to other circuits in the HSI which are covered later in the text.

- Roll and pitch attitude information is supplied to the ADI and to the computer from two different sources. The ADI is supplied by the MD1 attitude gyro. The computer is supplied by the ISS circuits in the TPLC. The signals furnished to the ADI are amplified and used to position the attitude sphere. Bank attitude is used in the computer signal mixing network to compute vertical steering pointer information. Pitch attitude is utilized in the computer signal mixing network in ILS, flare, and VER NAV modes to compute horizontal steering pointer information.

- Course deviation signals are sent to the computer. The deviation signal is also sent to the HSI course deviation indicator. The computer mixes the deviation signal with other signals to provide computer guidance information to the ADI.

- Distance information is supplied to the HSI window. This indicator is made up of three synchro movements and dc meters. The synchro movements position the units, tens, and hundreds counters. The dc meter movement positions the thousands digit. Only two indications can appear on the shutter and thousands digit: the figure "1" or a blank. Since this digit is not used in C-141A aircraft, it is always blank. The shutter appears if signals are unreliable or not available.

- To-from information is supplied to the to-from arrow on the HSI in VOR and Tacan.

- Bearing signals are supplied to the bearing indicator on the HSI. This movement is a servo-driven pointer.

- Course command signals (supplied only when the system is in the NAV OFF mode) are used to position the servo-driven course arrow. The course arrow is manually positioned by the course set knob in all other modes of operation except slave mode.

- The course error signal sent to the computer is derived in the HSI course datum synchro's stator; its rotor is positioned by the course set knob. The signal developed across the rotor is the difference between aircraft heading and selected course and is called course error. When the course arrow is under the lubber line, the selected course is the same as aircraft heading, and the course datum synchro is nulled with zero course error output.

- Heading error signals sent to the computer are derived in the HSI heading datum synchro. This is done in the same manner as the course error signal except that its rotor is positioned by the heading set knob.

- Erection cutout to the attitude gyro to prevent erection during turns is supplied by the rate switching gyro.

- Rate-of-turn is sensed by the rate transmitter and sent to the rate-of-turn indicator on the ADI.

- Navigation systems, such as VOR/ILS, Tacan, ASN-24 ANS-35, are selected by the navigation selector panel which connects these inputs to the HSI and computer.

- Normal acceleration is supplied to the computer and is used in computing pitch steering during ILS or flare modes. No. 1 FDS is supplied by the No. 4 vertical accelerometer, and No. 2 FDS is supplied by the No. 3 vertical accelerometer.

- Pitch rate is supplied to the computer by a two-axis rate gyro. No. 1 two-axis rate gyro supplies No. 1 FDS, and No. 2 gyro supplies FDS No. 2. This information is used in the same manner as normal acceleration.

- Elevator position is furnished to the computer for pitch steering computation during flare mode only.

- AWLS computer signals supply the FDS computer with signals commanding optimum angle of attack (R/GA), vertical velocity error (flare), and VER NAV deviation and steering. R/GA is switched in manually by the go-around button on either control wheel. VER NAV is armed manually when the VER NAV control is moved from "STDBY" to "STAGE 1" or "STAGE 2" and switches itself in when close to the VER NAV path. Flare is switched in automatically as a function of altitude. All connecting systems are shown in Figure 7-11.

<u>Heading Loop and Slaving Operation</u>

Compass heading information is supplied at all times to the azimuth ring control transformer synchro in the HSI. Any error voltage is amplified by the compass servo amplifier which drives the servo motor turning the compass card, CT rotor, and rate generator as shown in Figure 7-12.

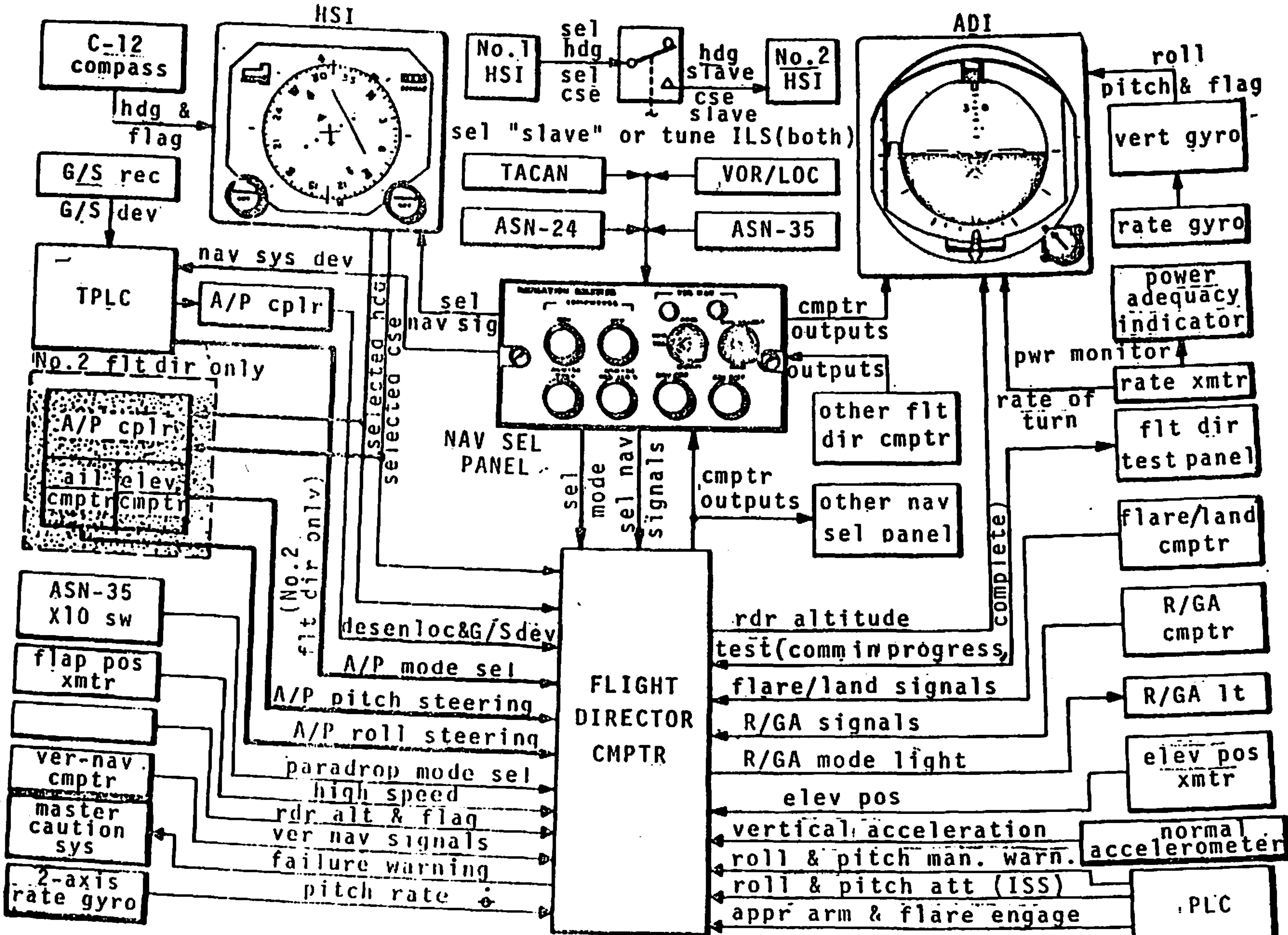


FIGURE 7-11. FLIGHT DIRECTOR SYSTEM DATA FLOW

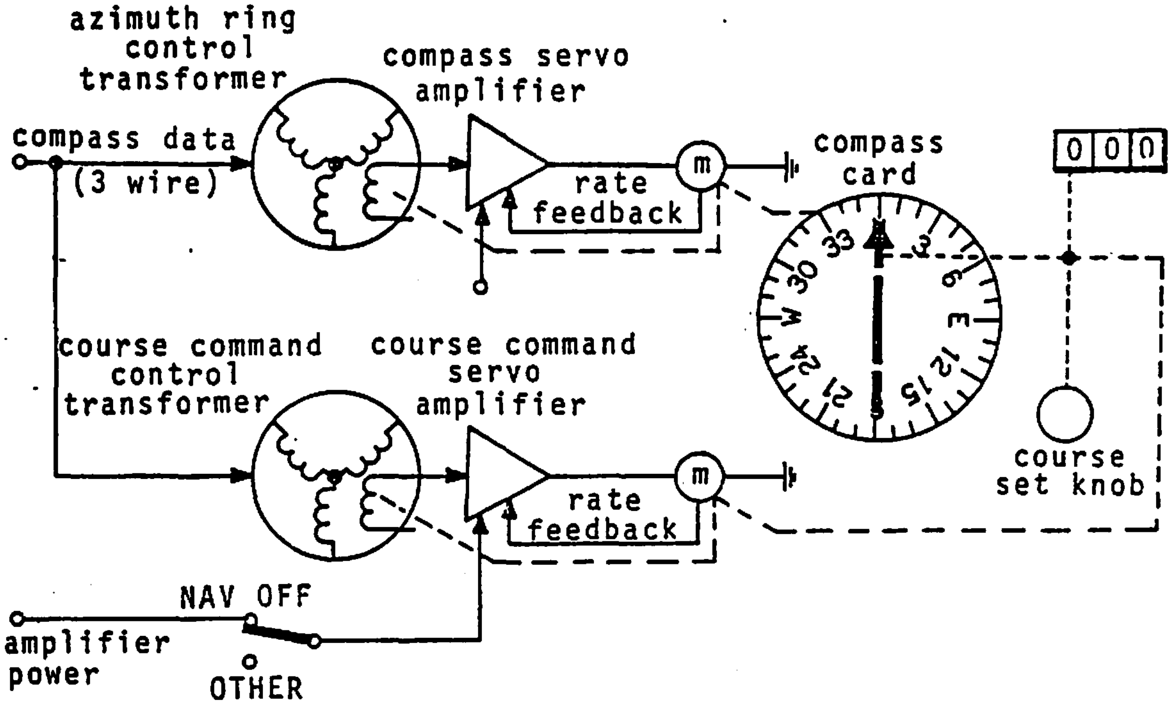


FIGURE 7-12. HSI HEADING LOOP

The rate generator feedback is used to stabilize operation and prevent overshooting or oscillation. Whenever the system is in the NAV OFF mode, power is supplied to the course command servo amplifier.

The course loop is similar to the heading loop and is used to drive the course arrow and course digital readout to the aircraft heading. In both loops, the motor nulls out error signals by repositioning the CT rotor as well as the various indicators.

The HSI course and heading slaving circuits are shown in Figure 7-13. Although feedback from the motor to the amplifier is used, it is not shown here for clarity.

NOTE

There are two rotors in each of the CT's of the pilot's HSI. (There are two in the copilot's also but they are not shown here since they are not connected.)

During normal operation of the heading set marker, the only way of moving the marker is by manually turning the HEADING SET knob. Neither of the motors can be driven since there is no power input to the pilot's heading amplifier,

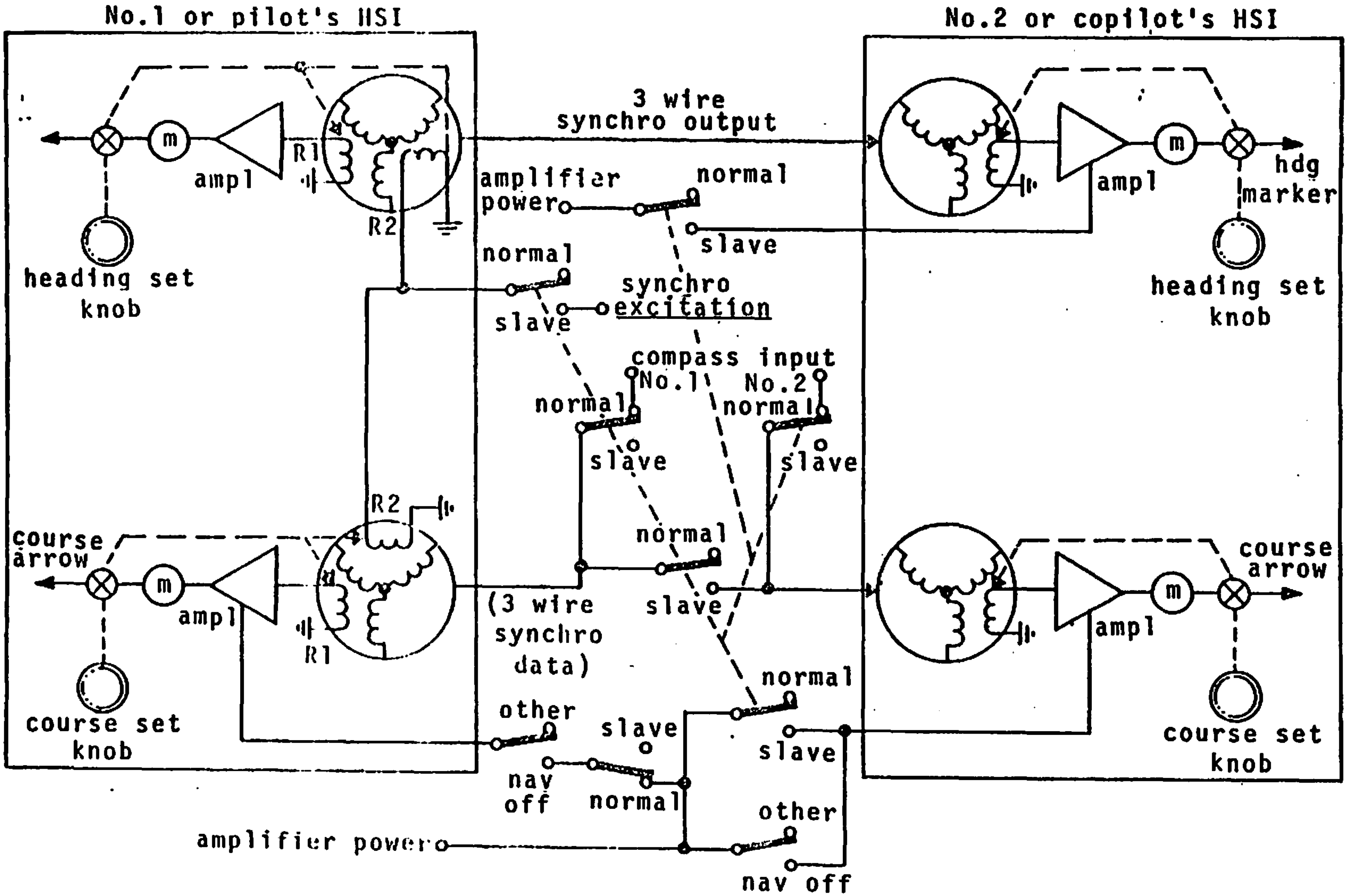


FIGURE 7-13. HSI HEADING SET AND COURSE SET SLAVING

and the switch in the copilot's heading amplifier power line is open. Thus, during normal operation, each of the heading markers is independent of the other. However, when the HSI No. 2 HEADING and COURSE switch is placed to "SLAVE," amplifier power is applied to the copilot's amplifier, and synchro excitation is applied to CT rotor R_2. This CT then becomes a synchro transmitter feeding the No. 2 HSI CT. If both heading markers are not positioned to the same point on the compass card, the difference in position, or error, is felt on the copilot's CT rotor. Since there is no power to the pilot's amplifier, it does not matter if some error is felt across R_1. The No. 2 heading set amplifier, however, senses the error and drives the motor turning the shaft until the error is cancelled. When the error is nulled, the No. 1 and 2 heading markers are at the same point on their respective compass cards. Then, anytime the No. 1 marker is moved, an error is developed in No. 2's CT, which is nulled only when its marker is coincident with the No. 1 marker.

The functioning of the course arrow during slave mode is identical to that of the heading marker. In Figure 7-13, it is seen that synchro excitation is supplied to R_2 of No. 1 HSI's course CT. Amplifier power is also applied to the No. 2 course amplifier through slave switch contacts, and No. 1 amplifier power supply is open in slave.

In slave, the switches disconnect compass information from the CT's and connect the CT's together, which gives identically the same operation as the heading marker. Normal operation is without synchro excitation or amplifier power. With these conditions, the course arrow can only be moved manually.

Although compass information is being fed in, lack of amplifier driving power prevents the motor from driving. If NAV OFF is selected, amplifier power is fed to the amplifiers and the motors slave the course arrows to compass heading. This action occurs since the output of the CT rotors is the difference or error between course arrow position and actual compass heading. This error drives the motor until the error is nulled which occurs only when course arrow is coincident with compass heading.

Roll and Pitch Attitude

The artificial horizon section of the flight director consists of two conventional servo loops. The error-sensing synchros (roll and pitch) are attached to the gymbals of the attitude gyro. The synchro stators are connected to the stators of their respective control transformer synchros in the ADI. Error signals are amplified to drive a servo motor which positions the sphere, provides a stabilizing feedback signal, and nulls out the rotors of the control transformer synchros as shown in Figure 7-14.

The CT's have two rotors fixed 90 degrees apart. One of these rotors senses

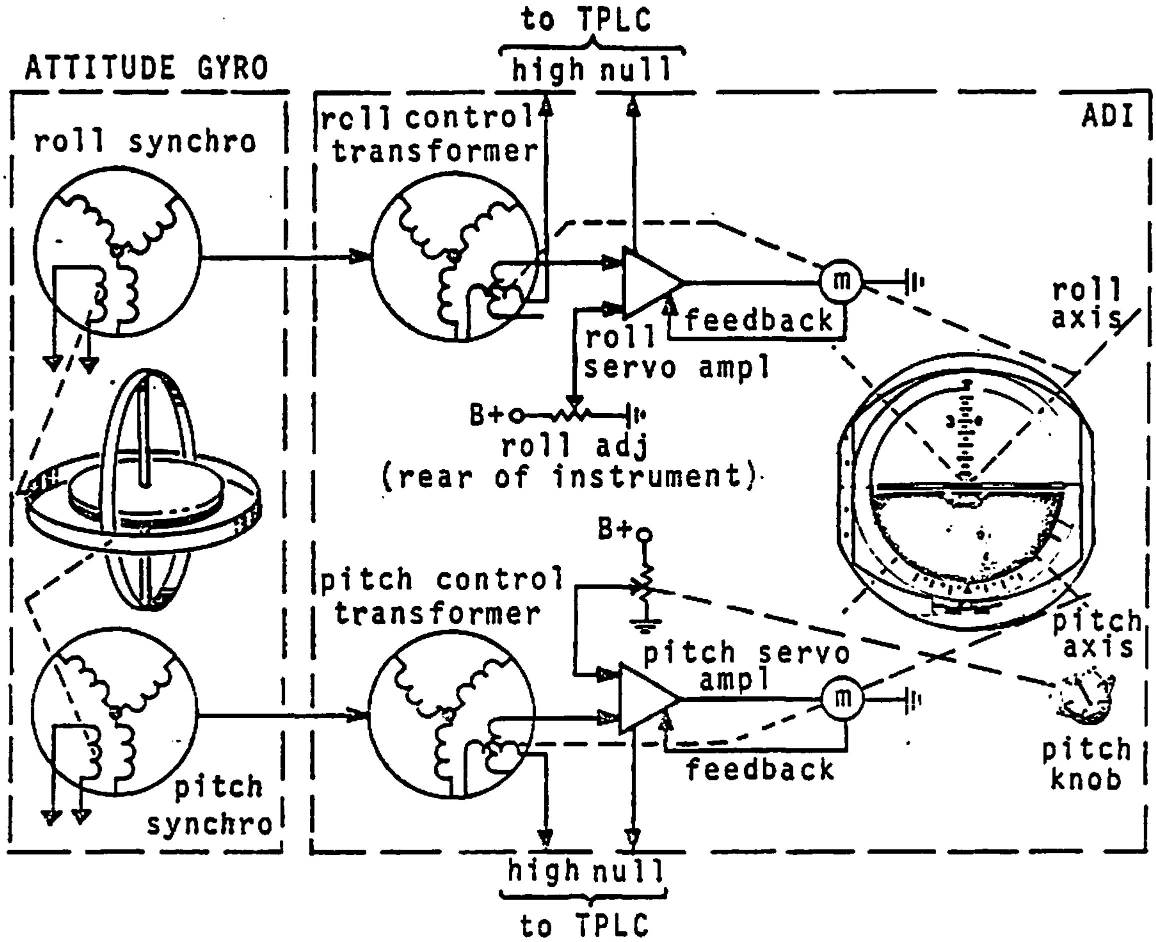


FIGURE 7-14. ARTIFICIAL HORIZON LOOP

sphere position error. The second rotor is referred to as the "high winding." Since the error sensing rotor should be held near null and the high winding is fixed 90 degrees away, the high windings will have a nearly constant output. Also, driving voltage to the amplifiers are monitored. If the loop remains nulled (and it should), the output should be very small. Both the null and the high are fed to the TPLC as a sphere position monitoring signal. If the high or null deviate from their normal values, the TPLC senses this and turns on the appropriate gyro fault light on the fault identification panel.

The pitch knob varies a bias voltage to the pitch servo amplifier, which varies the position of the sphere in the pitch axis to correspond to actual flight pitch attitude. The ADI amplifier contains a potentiometer to adjust the roll axis of the attitude sphere for any inherent synchro unbalance in the roll loop. This adjustment is not needed in the pitch axis since this axis is varied by the pitch knob.

ISS circuits in the TPLC select the intermediate signals from the two FDS attitude gyros and the A/P displacement gyro. The ISS circuits have 6 outputs (3 roll and 3 pitch). The No. 1 FDS uses ISS output No. 1, and No. 2 FDS uses ISS output No. 2, as shown in Figure 1-15.

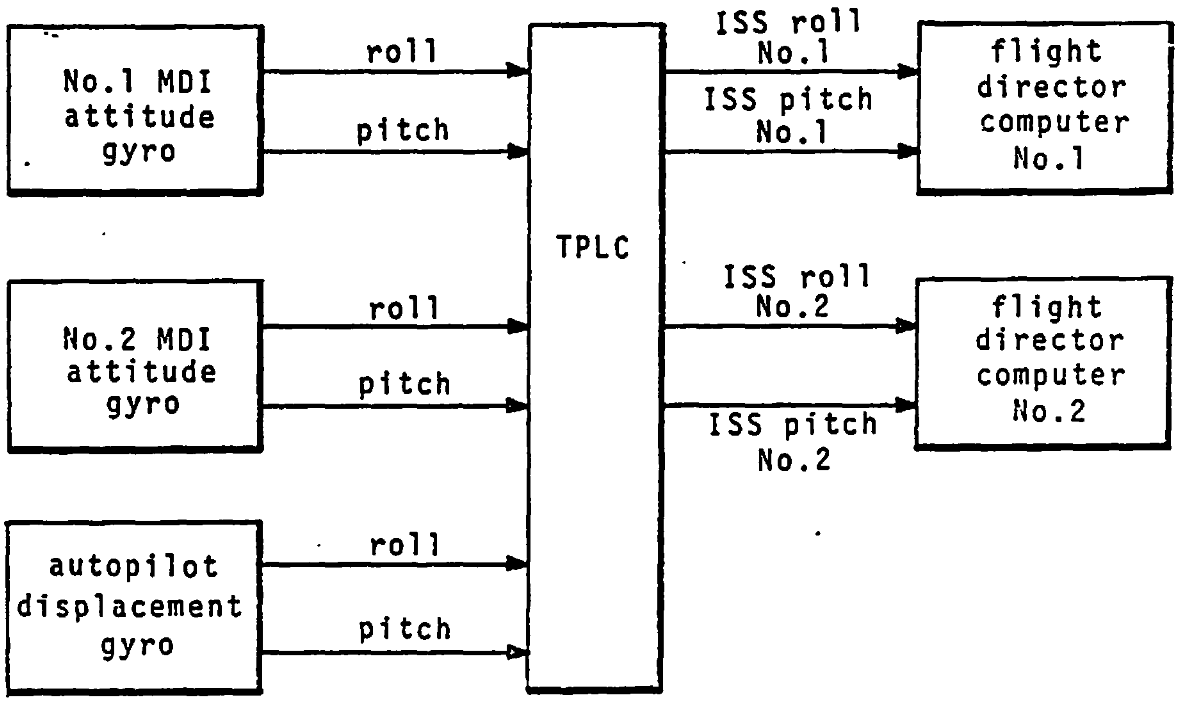


FIGURE 7-15. COMPUTER ROLL AND PITCH INPUT

Attitude Gyro Loop

The attitude gyro loop employs a type MD-1 attitude gyro for vertical reference to the artificial horizon. An MC-1 rate gyro is used to cut out the roll erection torque motor during turns when the gyro might be erected to a false vertical. The motors used in these gyros for drive and torque are two-phase types.

A typical phase-sensitive motor, used to run the gyros, is shown in Figure 7-16. In a motor of this type, it is necessary for the currents through the two windings to have some finite phase difference (other than zero and 180 degrees) before the motor will run. Normally, the windings are connected to a single-phase power source with a capacitor in series with one winding. The insertion of the capacitor in series with one winding causes sufficient

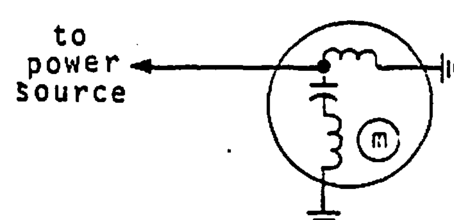


FIGURE 7-16. PHASE-SENSITIVE MOTOR

phase shift in winding current to enable the motor to run. As shown here, the motor runs anytime power is applied but always in the same direction.

In some applications it is desirable for the motor to be capable of operating in either direction. Such is the case of the torque motors used to keep the MD-1 attitude gyro erect. In this application, two capacitors and some type of switching-device are used to insert the capacitance in series with one winding and connect the other winding directly to the power source. A typical method of switching used in gyro erection systems employs a liquid-level switch. When the gyro is level, the liquid switch, as shown in Figure 7-17, bypasses both capacitors so that no phase difference exists between the currents through the two windings. In this case, the motor would not operate. If the gyro drifts off level, the electrolyte in the level switch would be displaced. If the electrolyte varies the resistance to either of the two top contacts, the appropriate capacitor becomes effective as it is no longer bypassed, which gives the necessary phase difference and the motor would operate. If the gyro tilted the other way, the same action would occur on the other side, and the motor would operate in the opposite direction.

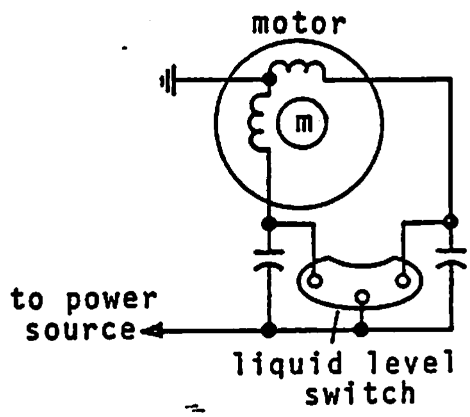


FIGURE 7-17. LIQUID LEVEL SWITCHING

The erection system in the MD-1 gyro uses a similar arrangement. The switches are electrolytic types whose action is to change resistance. In essence, the operation is as described in the preceding paragraph. Operation of either torque motor causes a force to be exerted on the gyro gimbal which, in turn, causes the gyro to precess in the proper direction to return to a vertical position. This vertical position is relative to the earth's surface. During turns of the aircraft, it is likely that the liquid levels would sense a false vertical due to centrifugal force and cause the gyro to erect to a false vertical. For this reason, an MC-1 rate gyro is used to disable the erection during turns that exceed a predetermined rate.

Figure 7-18 shows the connection between the MD-1 attitude and MC-1 rate gyros. If the turn rate is high enough to close the switch in the rate gyro, the amplifier signal closes relay K-101. When K-101 closes, both winding of the roll torquer are connected together to AØ-bus 26-volt ac power. The essential action is both roll torque windings tied together and no phase difference can exist. With zero phase difference, the motor does not operate. After rolling out of the turn, the rate gyro switch opens to deenergize K-101 and allow the roll erection system to operate.

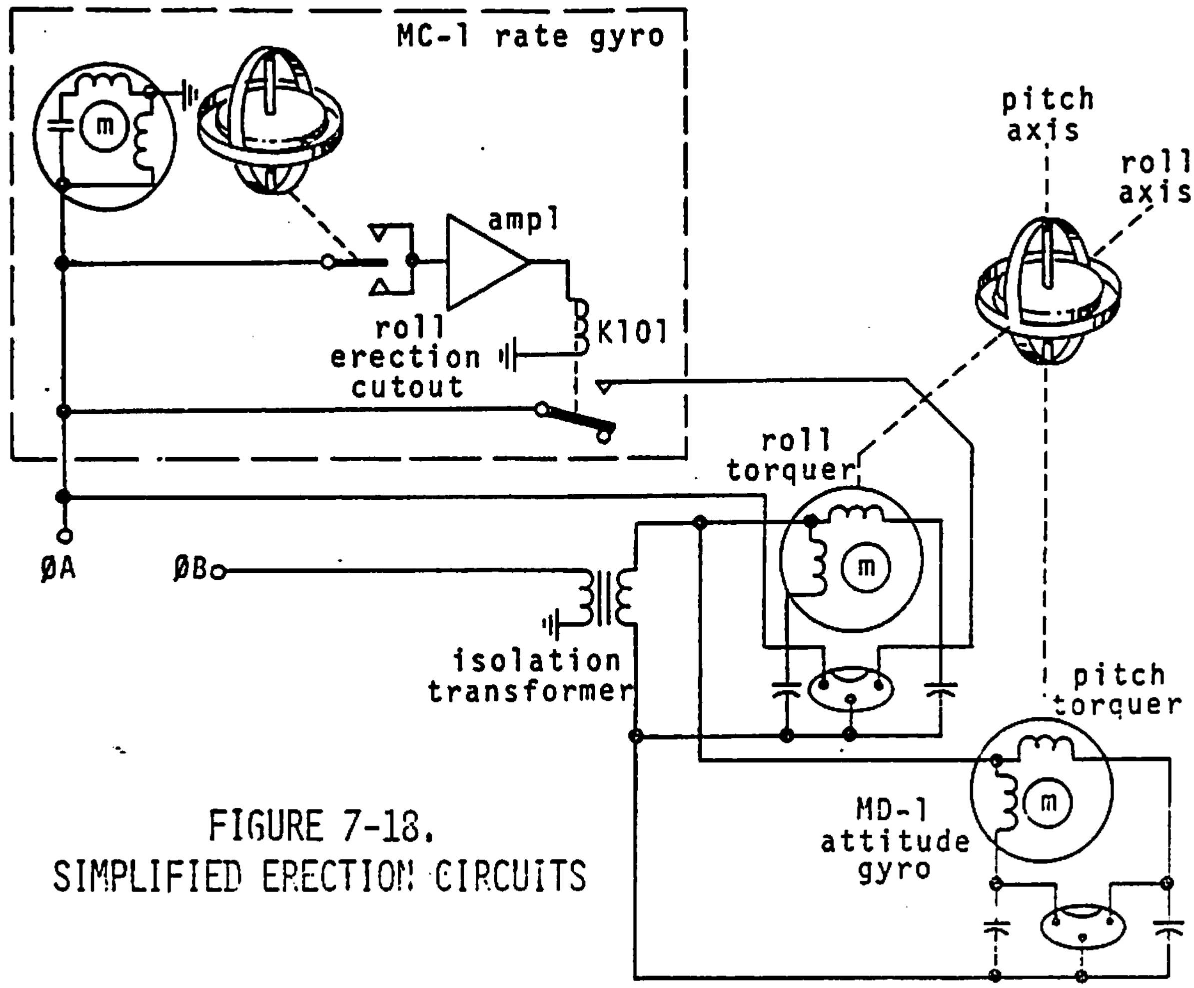


FIGURE 7-18.
SIMPLIFIED ERECTION CIRCUITS

Rate-of-Turn Indicator

The gyro in the rate transmitter, as shown in Figure 7-19, positions the arm on a potentiometer. This potentiometer is in a typical wheatstone bridge circuit. The rate-of-turn indicator is a dc meter movement, which measures the relative unbalance of the bridge. It is calibrated in minutes (number of minutes to complete a 360 degree turn).

Slip Indicator

The slip indicator is a method of indicating the direction of the vector sum of gravitational/centrifugal forces which act on the aircraft as shown in Figure 7-20. In straight and level flight, there is no centrifugal force and gravity centers the inclinometer ball.

During a turn, centrifugal force acts at right angles with gravity, and the ball is positioned as the vector sum of the forces. A coordinated turn is one during

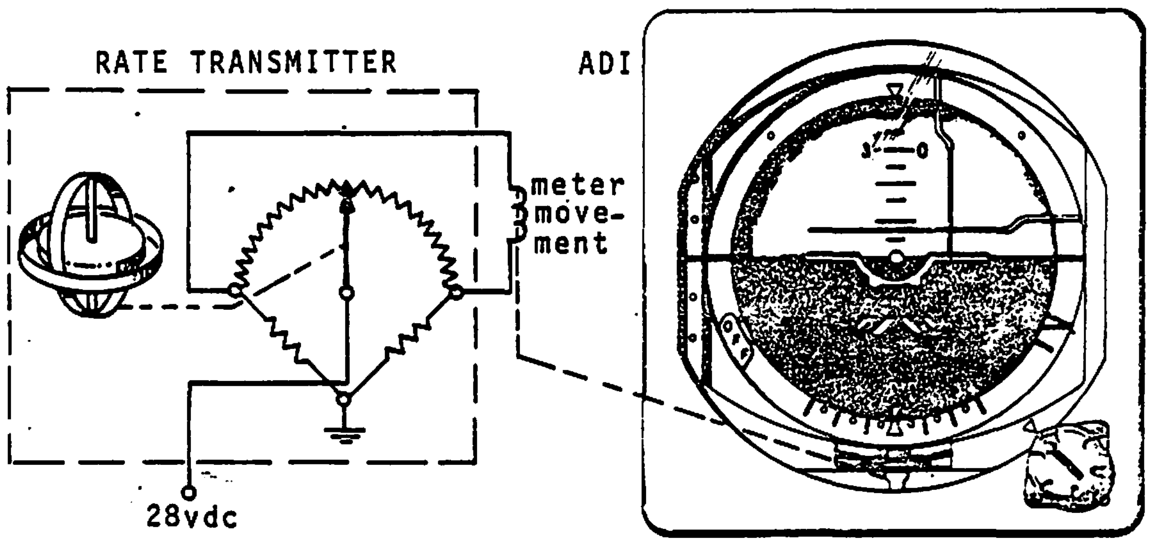


FIGURE 7-19. RATE-OF-TURN DISPLAY

which the direction of the net force is at right angles (perpendicular) with respect to the cargo floor. As long as this force is perpendicular, the ball remains centered. With inadequate bank, the centrifugal force is greater than gravity. The net result is an aircraft skid toward the outside of the turn, and the ball is displaced in a direction toward the outside of the turn (high wing).

With too much bank, the effect is just the opposite. The centrifugal force is less than the gravity force, and the net force is not perpendicular to the wings. The aircraft would then slip downward toward the inside of the turn, and the ball would be displaced in the same direction (low wing).

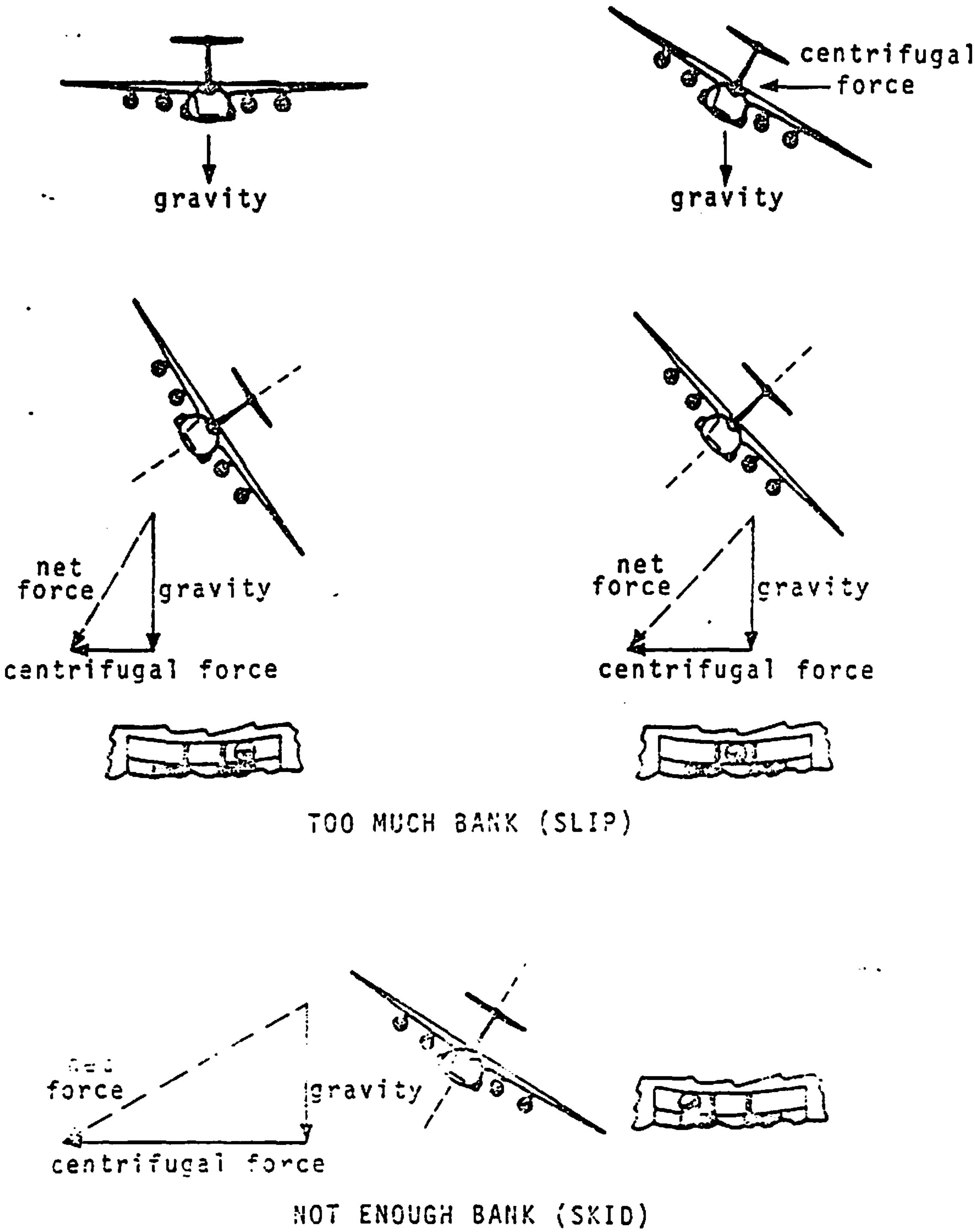


FIGURE 7-20. GRAVITATIONAL / CENTRIFUGAL FORCES

Navigational Information Loop

Course deviation signals are sent to the FDS computer. The deviation signal is also sent to the HSI course deviation indicator as shown in Figure 7-21 which is a simple dc meter movement. The computer mixes the deviation signal with other signals to supply computed guidance information to the ADI bank steering bar. The course warning flag signal is monitored in the computer, which provides an output to the course warning flag dc meter movement. To-from signals are also applied to the HSI. The signals position a to-from dc meter movement.

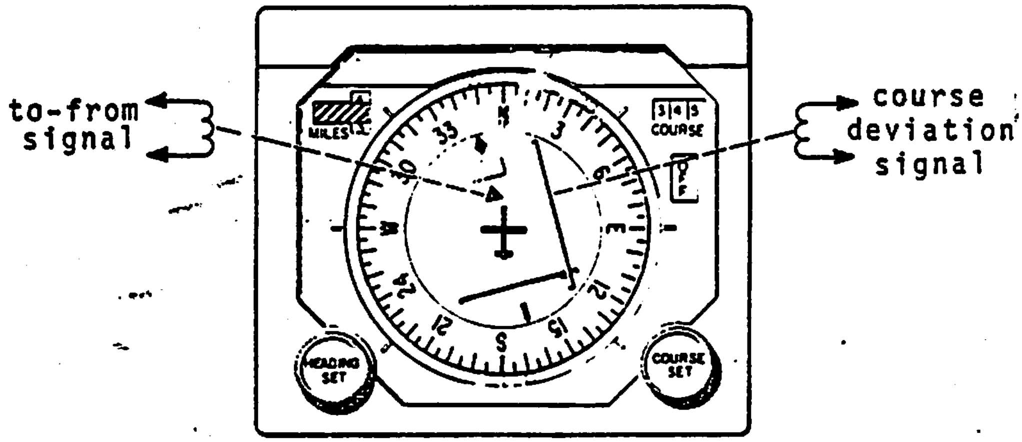


FIGURE 7-21. HSI

Distance Information

Distance information is supplied to the HSI range indicator as shown in Figure 7-22. This indicator is made up of three synchro movements and a dc meter

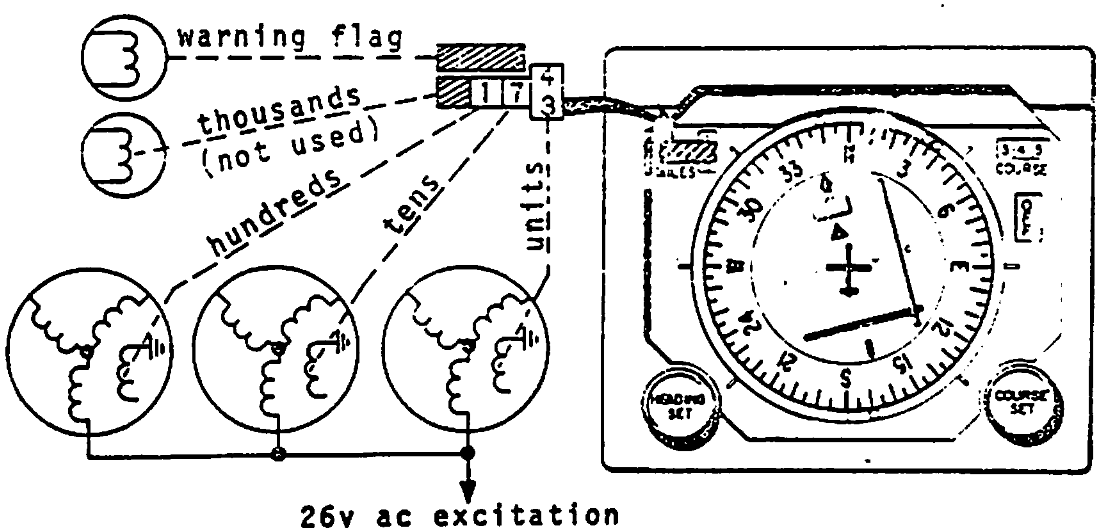


FIGURE 7-22. DISTANCE DISPLAY

movement. The three synchro movements position the "units," "tens," and "hundreds" counter dials. The dc meter movement positions a shutter over the thousands digit. The shutter is not connected since the thousands indicator is not used. A second dc meter positions the distance warning flag shutter.

Distance information displayed is either Tacan distance from a Tacan ground station or distance remaining on a selected doppler or navigation computer course.

<u>Bearing Indication Loop</u>

The bearing pointer, as shown in Figure 7-23, displays either VOR or Tacan station bearing or track angle error during doppler or ASN-24 guidance. The loop is a typical servo loop using a control transformer synchro, amplifier, and motor the same as the heading loop, previously discussed.

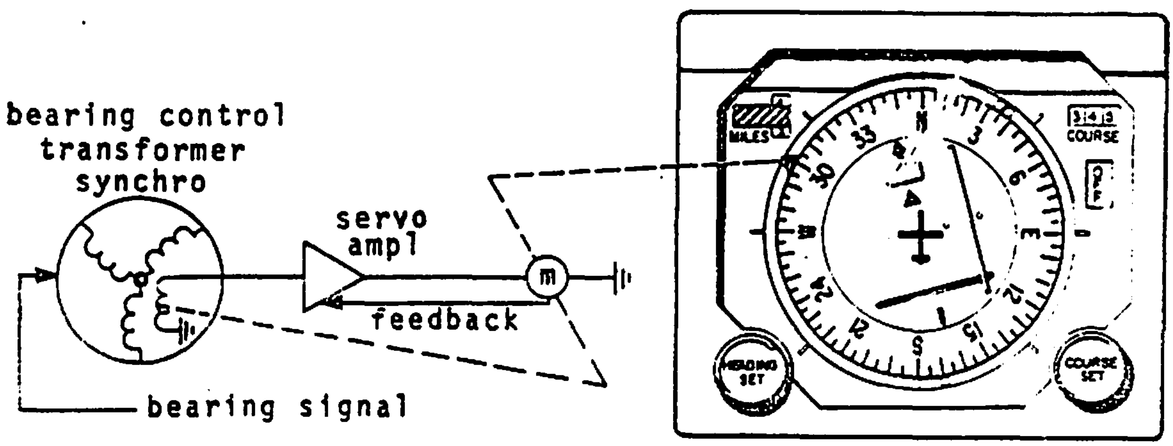


FIGURE 7-23. BEARING DISPLAY

COMPUTER - THEORY OF OPERATION

The AWLS FDS computer utilizes 19 distinct modes. These modes, along with the ADI presentations for each, are shown in Figure 7-8. The manual switching, with two exceptions, is accomplished from the pilot's or copilot's navigation selector panel. These exceptions are R/GA and VER NAV.

Lateral Channel Modes

<u>Heading Select</u>

The basic mode of the FDS is manual heading mode, selected by positioning the HDG SELECT/NAV switch to "HDG SELECT." In this mode, the signal supplied to the vertical steering pointer is a composite signal of bank angle from the ISS circuits and HSI heading error, as shown in Figure 7-24. Logic switching prevents any other lateral mode from being displayed when the FDS is in "HDG SELECT." A bank command limiter prevents command exceeding 23 degrees. A bank rate command limiter limits the rate at which bank commands are given (10 degrees/second). The automatic manual heading mode is initiated whenever a radio track mode is selected and lateral beam sense is not satisfied. Lateral beam sense is a signal term used in the FDS to initiate capture. When radio deviation drops to the capture initiation point, Lateral Beam Sense (LBS) is satisfied, and the computer switches to capture mode. The bank command limits follow:

Automatic Manual Heading	Heading Select	Radio Capture	Radio Track	LOC Track	Approach Arm
30 degrees	23 degrees	30 degrees	15 degrees	15 degrees	7.5 degrees

Automatic Manual Heading (MH) is very silimar to heading select except for bank limits. Automatic MH is switched in whenever a radio track mode is selected with the HDG SELECT/NAV switch to "NAV" and the aircraft outside LBS.

Radio Track Mode

The radio track mode is initiated by selecting a radio mode on the navigation selector panel (VOR, Tacan, ASN-35, and ASN-24). In radio track, there are three submodes: Automatic MH, Radio Track Capture (RTC), and Radio Track Track (TRK). RTC occurs at LBS. In RTC mode, radio deviation is filtered and summed with bank angle and course error; it is then sent to the course cut limiter which limits course cut commands to a maximum of 45 degrees. (The course inputs to the signal mixing networks are applied through resistive networks whose gains are changed by logic switching to make them compatible with the appropriate mode.) The signal is then fed through the energized contacts of the MH switch. (At this point it should be noted that a logic term identified with a bar over it is the negation or inversion of the term itself. Also, the logic terms shown with each switch is the logic required to energize that particular switch.)

From this point, the signal is fed through the limiters to the meter driver to the vertical steering pointer. The VP logic would be energized and off-scale bias removed from the pointer. At this time, if the signal is reliable, the vertical pointer is in view, and the vertical flag is out of view. Later, the

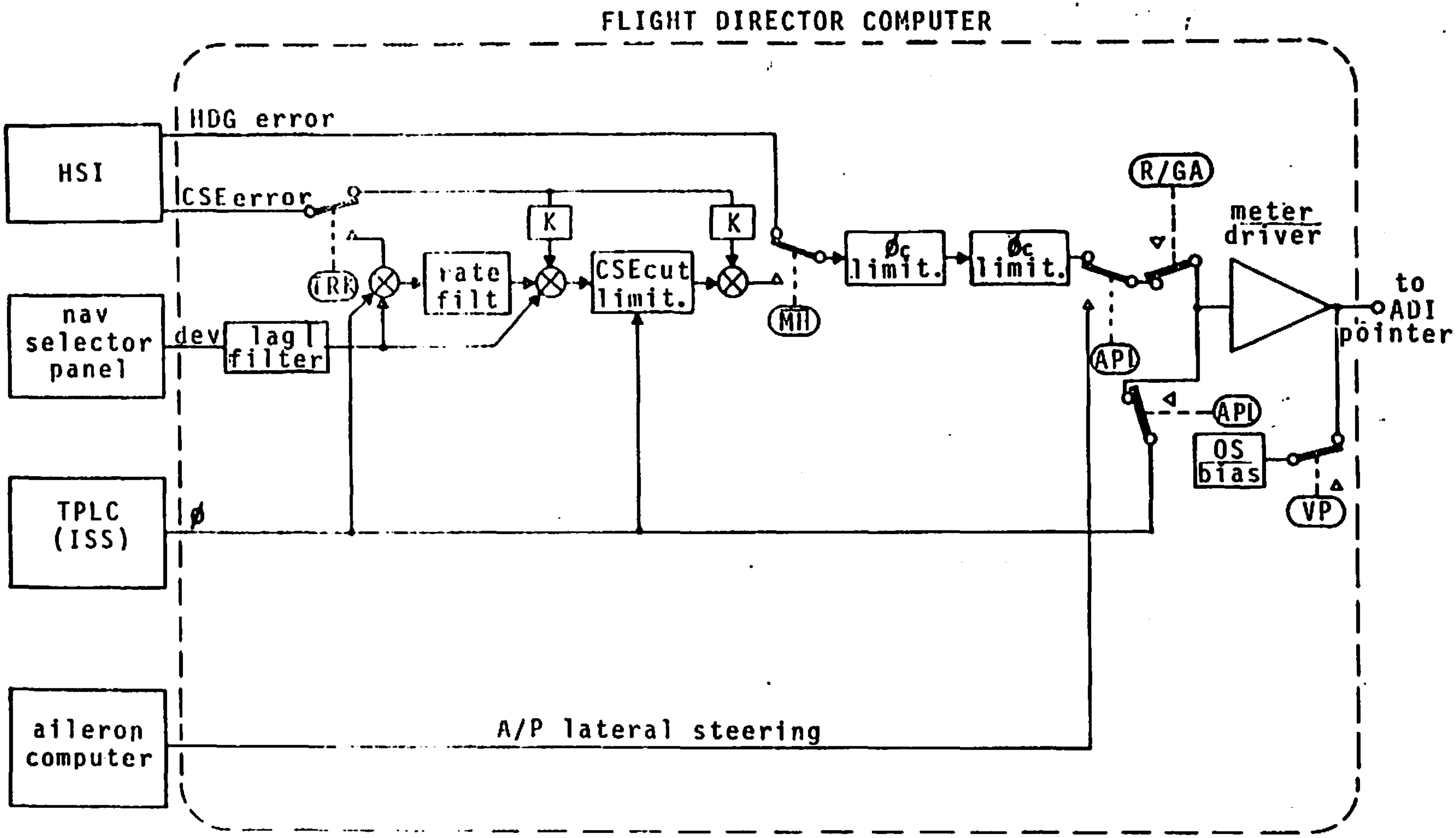


FIGURE 7-24. SIMPLIFIED LATERAL CHANNEL

specific logic to bias the pointer and flag off scale is discussed. After capture is complete and requirements for TRK are complete, the TRK switch energizes, which allows only course rate information to be summed with bank angle and radio deviation. Since only rate information (rate-of-change) is passed by the rate filter, long-term course errors such as those existing due to cross wind effects would be washed out, and only radio deviation and bank angle could command changes in aircraft heading. In TRK, the gains are changed to optimize tracking. In this mode, as well as all radio modes, the pointer output signal is the difference between the bank command signal and bank angle.

NOTE

Gain changes are made by logic switching dependent upon the mode selected.

VOR/ILS Mode

Whenever VOR/ILS is selected, the ADI is supplied a lateral steering signal composed of the difference between the bank angle and bank command signals. The radio deviation is provided by the VOR receiver.

Tacan Mode

Tacan mode is identical to VOR mode with two exceptions: Tacan LBS varies with distance to the station and the Tacan slant range distance is displayed on the HSI.

ASN-24 or ASN-35

ASN-24 or ASN-35 mode signal process is similar to VOR or Tacan modes. The computer modes, however, utilize crosstrack error rather than radio deviation although the computer treats this signal exactly as it would a radio deviation signal. Also, TAE is displayed on the HSI rather than bearing to a ground station.

Paradrop mode is initiated concurrently with normal ASN-35 track mode by positioning the NAV/X10 switches on the doppler auxiliary control panel to "X10." Circuits for signal processing are the same as in normal ASN-35; however, gain changes are optimized to more fully utilize the X10 scale factor switched in by the paradrop mode logic switching.

Highspeed paradrop mode is a submode of normal ASN-35. In order to keep the steering commands within controlling limits of the pilot and the aircraft, highspeed paradrop mode is switched in whenever the aircraft is above 190 knots or when flaps are fully up. Since the aircraft is limited by the flight manual to speeds of 190 knots or less unless flaps are up, a switch in the flap position transmitter closes whenever the flaps are lowered greater than 3 degrees. This switch is, in effect, an airspeed detector. In highspeed paradrop mode, gains are reduced further to keep pointer commands with the response ranges of the pilot and the aircraft.

In Figure 7-25, it can be seen that 28-volt, dc is fed to the NAV computer logic input whenever either computer is selected. If ASN-35 is selected and the NAV/X10 switch on the ASN-35 auxiliary control panel is in the "X10" position, the FD computer switches to paradrop mode. If the flaps are up, 28-volt, dc is fed from the ASN-35, NAV computer input through the deenergized contacts of the relay to the FD computer highspeed paradrop input. This is highspeed paradrop mode. If however, the flaps are lowered (and if they are, it will be 3 degrees or more) the relay energizes removing the 28-volt, dc signal from the High-Speed Paradrop (HSPD) input. This action causes the FD computer to revert to normal paradrop mode.

VOR/TAC Approach Mode

VOR/TAC approach mode is a submode of both VOR and Tacan. The approach mode is for use within 50 miles of the ground station to provide optimum tracking by altering computer gains. The VOR/TAC approach mode is selected from the navigation selector panel by placing the VOR TAC APPR/NORM switch to "APPR."

R/GA Lateral Mode

The R/GA lateral mode consists of bank angle information fed to the steering pointer for a wings level command, which is to avoid altitude loss since lift is maximum when the wings are level. As shown in Figure 7-24, when go around is selected, the R/GA switch energizes, leaving only bank angle to be fed into the meter driver through the deenergized contacts of the APL (A/P Lateral) switch. If the autopilot is engaged after approach arm, the autopilot steering commands are passed through the energized APL switch, through the deenergized R/GA switch, and the meter driver to the ADI pointer. Bank angle is not mixed with APL steering since the energized APL switch disconnects bank angle from the meter driver input.

Localizer Mode

Localizer mode is discussed here only through its progress to approach arm and picked up again in the discussion of ILS approach mode. Except for gain changes and the lack of a bearing pointer signal, LOC mode is much like any other lateral radio mode. Desensitized localizer is supplied to the FDS from

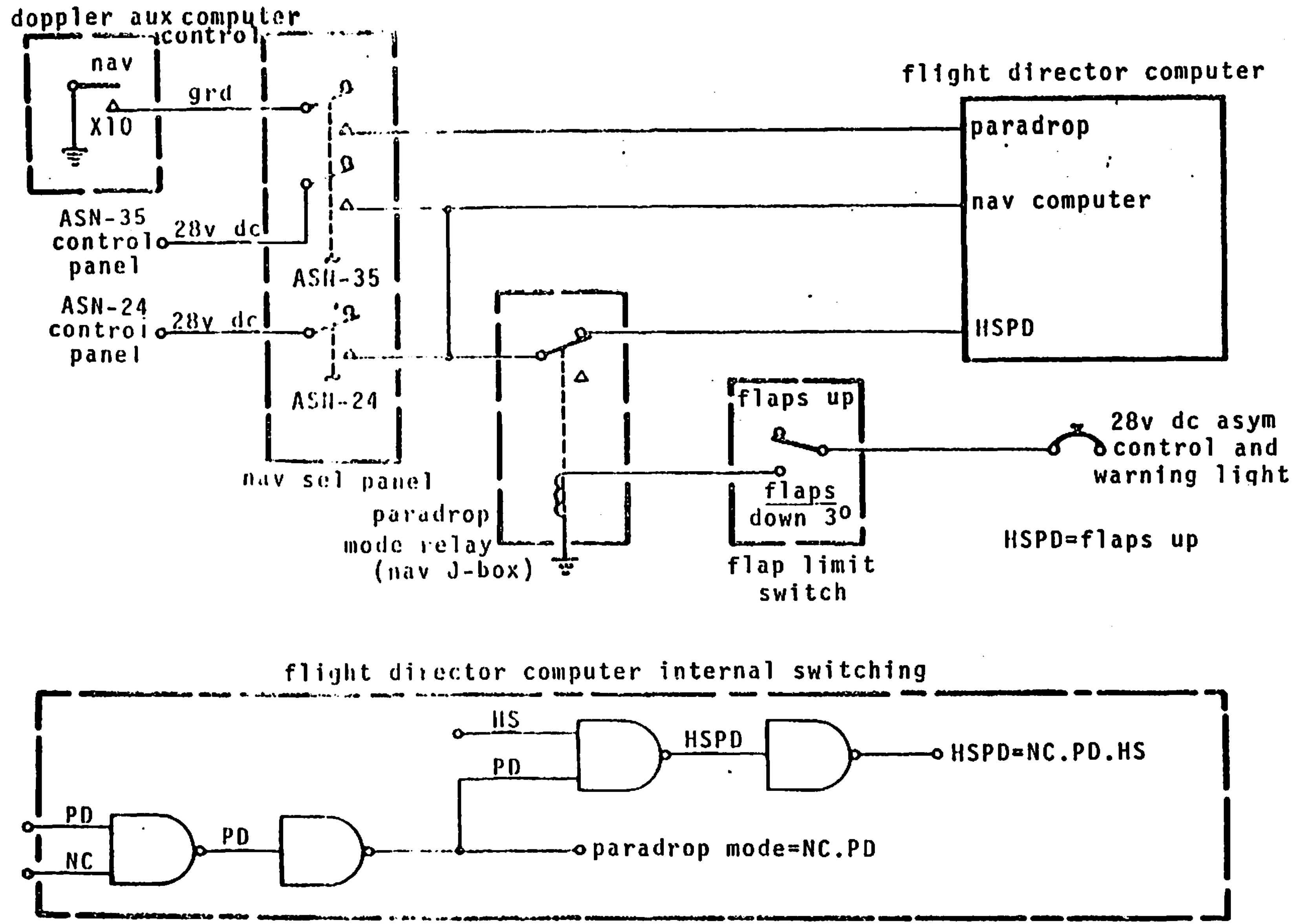


FIGURE 7-25. PARADROP MODE SWITCHING

the A/P coupler.

Desensitization of G/S and LOC begins at G/S engage. Prior to G/S engage, the LOC and G/S signal gains from the coupler are unity (1). During an AWLS approach, however, the FDS switches from A/P LOC information to raw LOC from the LOC receiver, at AA. As shown in Figure 7-26, desensitized LOC is fed through the deenergized contacts of relay K6. K6 is held deenergized in LOC mode by removal of a ground when K4 energizes; K4 is energized when a localizer frequency is selected. K6 has 28-volt, dc when any radio mode is selected. When any radio mode other than LOC is selected, K6 is energized by the 28-volt, dc RT signal, and the ground is provided through the deenergized contacts of K4. When a localizer frequency is tuned, with VOR/ILS selected from the navigation selector panel, and the VHF navigation receiver turned on, the VHF navigation relay energizes, which provides 28-volt, dc to energize K4 which removes the ground from K6 switching the lateral channel input to localizer from the A/P coupler.

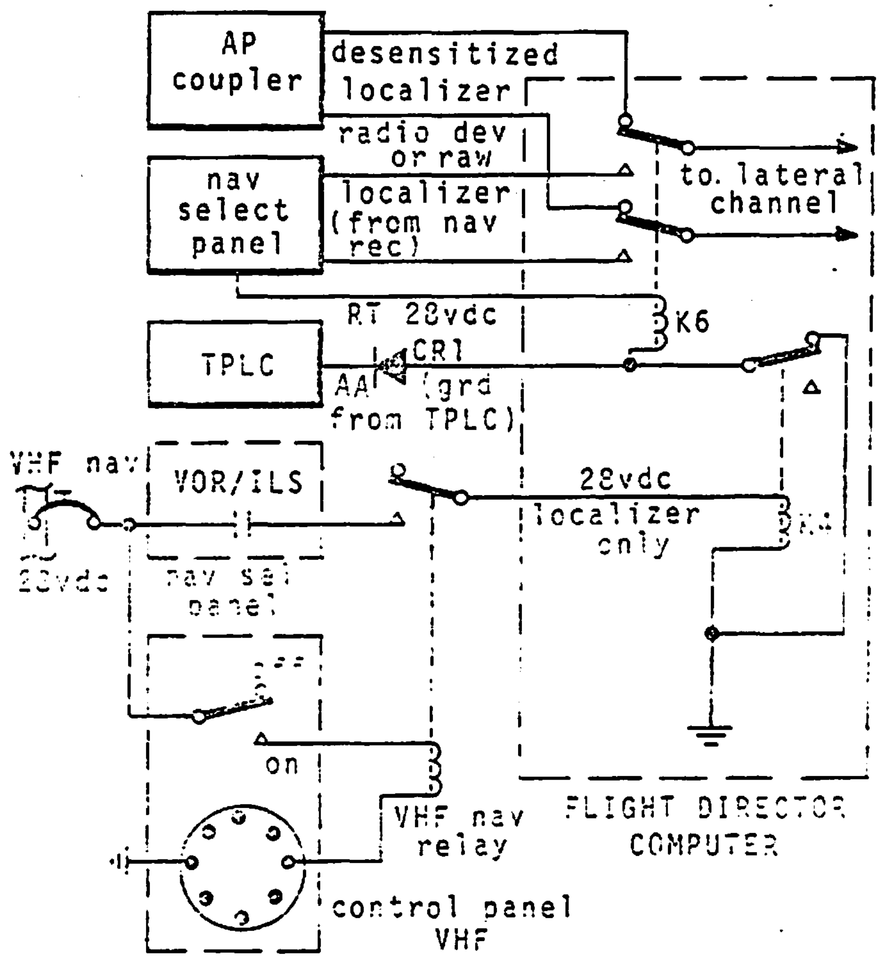


FIGURE 7-26. LOCALIZER SWITCHING

At AA, the TPLC provides a ground through CR-1 allowing K6 to energize which switches the lateral channel input from desensitized localizer to raw localizer from the receiver. Thus, in a non-AWLS approach, desensitized localizer would be used exclusively since there would be no AA signal to accomplish the switching.

A/P Lateral Mode

A/P lateral mode is initiated if the autopilot is engaged after AA. In this mode, autopilot lateral or aileron commands from the aileron computer are displayed on the vertical steering pointer as shown in Figure 7-24.

NOTE

This mode is available only to No. 2 system.

When the APL mode becomes activated, two APL switches close, which allow only APL steering to be passed to the meter driver. The upper APL switch passes APL steering and opens the signal path for other lateral channel information, while the lower APL switch removes bank angle from the meter driver input. This mode provides an expanded scale display of the aileron servo effort indicator.

NAV OFF

NAV OFF mode is selected with the NAV OFF button on the navigation selector panel. In this mode, all NAV inputs normally fed through the navigation selector panel are removed from the FD computer through logic switching; all pointer and flags are biased out of view. Only attitude information and aircraft control information, such as rate-of-turn and slip and skid are displayed in the mode.

Lateral Beam Sensing

There are two beam-sensing circuits used in the lateral channel. These are the lateral beam sensor and the track sensor.

Lateral Beam Sensor - The lateral beam sensor consists of a voltage level monitor which has two outputs. The outputs are either + 6 volts, dc or grounds (1 or 0). Output is a logic 0 until LBS occurs and a logic 1 after LBS. LBS occurs at different thresholds for different modes of operation, i. e. LBS occurs in VOR mode at 75 millivolts deviation; in NAV computer or ILS at 150 millivolt, and in the case of Tacan, LBS occurs at a threshold which varies inversely with the distance from the Tacan ground station. At 150 miles or more from the Tacan ground station, LBS occurs at 50 millivolts deviation. At 100 miles range, LBS occurs at 100 millivolts. At 125 miles, LBS occurs at 75 millivolts and at 75 miles, LBS occurs at 125 millivolts deviation. This sensor output, when combined with RT or ILS logic, initiates capture mode. Inputs to the LB sensor are radio deviation, Tacan range information, and logic switching to vary LBS levels as a function of mode.

Track Sensor - To switch into track mode requires two logic outputs from the track sensor as well as other logic inputs. The track sensor is composed of two individual sensors, each of which is similar in operation to the LBS. Each of the TRK sensors has two functions: one in normal operation, and one in self-test. In normal operation, the A track sensor monitors the radio deviation level and supplies a logic 1 output when deviation has decreased to the level required to switch into track mode (25 millivolts). In self-test operation, the A track sensor monitors the output of the horizontal steering pointer meter driver and supplies a logic 1 output when the pointer is driven to a null position. The B track sensor monitors course error during normal operation and provides a logic 1 output when course error decreases to 15 degrees. The B track sensor functions identically to the A sensor except it monitors the vertical steering pointer meter driver.

Vertical Channel Modes

There are five modes in the vertical or pitch channel. These are ILS approach, flare, R/GA, VER NAV, and APV. ILS approach mode begins when the computer satisfies its logic requirements for VBS. These are G/S valid, LOC valid, ILS selected, LBS, and Glide Slope Window (GSW). VBS is responsible for bringing the horizontal pointer into view. GSW is a term used when the aircraft is within a predetermined proximity of the G/S beam. (45 millivolts to latch GSW in or 150 millivolts to unlatch GSW). Once VBS has occurred, loss of any one of the requirements for initiating VBS may not bias the Horizontal Steering Point (HP) off scale. Unlatching VBS requires loss of LBS, switching out of ILS mode, or going to Heading Select (HS).

In G/S mode, there are four inputs to the computer as shown in Figure 7-27. They are desensitized G/S, normal acceleration, pitch angle, and pitch rate. Desensitized G/S from the A/P coupler passes through deenergized contacts of the Flare Engage (FE) switch to the pitch command limiter. Here, pitch commands are limited to ± 6 degrees. From the command limiter, the signal is passed to the command rate limiter, which limits the rate of change which can be felt at the summing junction. This action keeps pointer movement within the response capabilities of the pilot and the aircraft. At VBS (or ILS approach - both terms mean the same within the computer), a 1.7-degree positive bias is fed into the signal chain. Pitch angle, pitch rate, and normal acceleration are also fed into the signal chain. It is at this point in the approach that the HP comes into view. The + 1.7-degree bias causes the pointer to deflect downward, which commands capture of the G/S beam. As the bias washes out in the washout filter and G/S deviation decreases, the aircraft approaches the G/S beam.

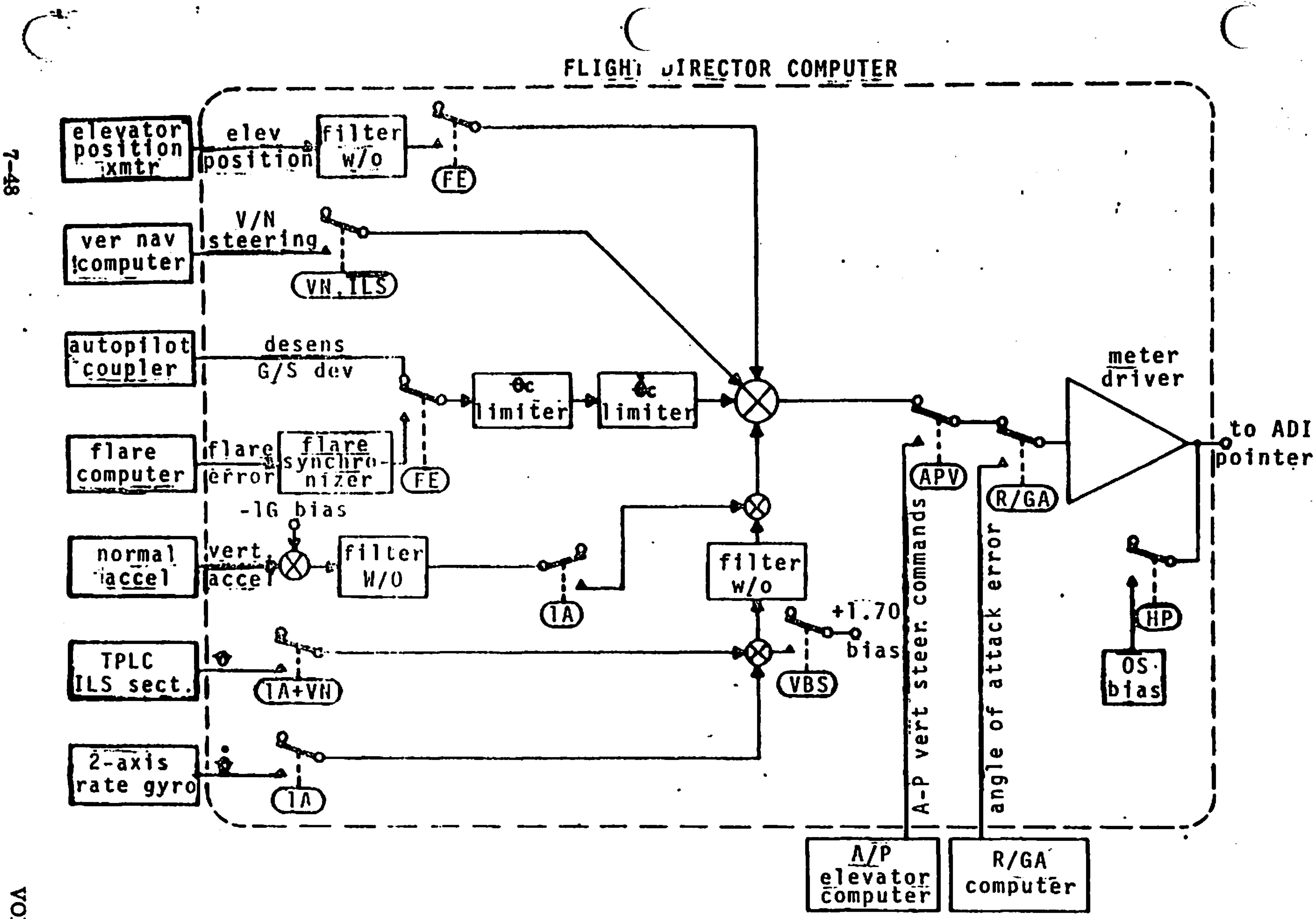


FIGURE 7-27. SIMPLIFIED PITCH CHANNEL

Normal acceleration is fed into the signal chain to anticipate changes in aircraft attitude. This is to say, if there is a change in acceleration, a corresponding deviation from the G/S beam might be expected. The same is true for pitch rate information. Pitch angle, however, is the signal summed with G/S deviation to aid steering command changes only. All of this supplementary or anticipatory information is applied to wash-out filters, which allows signals to be applied for a given length of time only. This allows steady state signals to build up without commanding steering to cancel them, while rate signals are passed directly to the signal chain. The summation of these signals occurs at a summing junction downstream of the pitch rate limiter. The composite signal is then fed through the deenergized contacts of the APV and R/GA switches to the meter driver and onto the horizontal steering pointer.

VER NAV mode provides steering information to capture, to track a VER NAV path, and to maintain altitude as commanded by VER NAV settings. VER NAV and ILS modes are not compatible; if both are selected simultaneously, the FDS switches out VER NAV and presents ILS information, as shown in Figure 7-27. In VER NAV mode, the VER NAV steering signal passes through the VN ILS switch to the summing junction at the pitch rate limiter output. No other signals except pitch angle are present at this junction. When VER NAV steering commands a pitch change, pitch angle information in opposition to the command is fed into the junction as the aircraft pitches to satisfy the command. Then, as the aircraft comes closer to the VER NAV path, the pitch signal commands the FDS to reduce the pitch command, thus enabling a smooth intercept of the VER NAV path and also smoother tracking when on the path. From the summing junction, the VER NAV composite signal follows the same path as the G/S composite signal.

APV Mode

In this mode, steering commands from the elevator computer are applied to the meter driver and on to the ADI through the energized APV switch (which also disconnects other inputs from the signal path) and through the deenergized contacts of the R/GA switch. This action provides the copilot with an expanded scale view of the elevator servo effort indicator.

R/GA Mode

R/GA mode overrides all other modes in the pitch channel. If the R/GA switch energizes, the paths from all other sources except R/GA are opened. The R/GA signal in the pitch channel is angle-of-attack error. This angle-of-attack error means that the aircraft is not at the optimum angle-of-attack for maximum lift in its present configuration. The computer angle-of-attack error is fed through the energized R/GA switch to the meter driver and the ADI steering pointer. The FDS makes no changes to the R/GA signals; it merely displays them.

Flare Mode

Flare mode is energized at 45 feet radar attitude in an AWLS approach. In this mode, vertical velocity error is fed into the FDS to provide commands for reducing the sink rate to a value within the structural limits of the aircraft. As shown in Figure 7-27, flare error is applied to the flare synchronizer, which keeps the input to the signal chain from commanding a very abrupt fly up signal at FE.

The synchronizer feedback path (within the synchronizer) is broken at FE, and the synchronizer begins to washout its built-up error to allow the complete signal to pass. This action is necessary since the FE at 45 feet is so great it would command nose-up of the aircraft into a stall attitude. The flare signal passed by the integrator is limited in the pitch command limiter to + 6 degrees and - 1 degree and to a rate of change within response times of the pilots and of the aircraft. The signal is then applied to the summing junction along with the same signals present in G/S mode (except the + 1.7-degree bias which is washed out) plus elevator position feedback. Elevator position is used to prevent over-commanding in flare mode.

Since elevator position changes almost immediately in response to a command, the elevator position feedback cancels the pointer command immediately to prevent further response. The signal (elevator position) is washed out after a fixed time delay so that the final command or the horizontal steering pointer is a result of error only. The signal path from the summing junction is through deenergized contacts of the APV and R/GA switches and through the meter driver to the ADI pointer.

NOTE

If the autopilot is engaged, flare will be displayed on No. 1 FDS only and APV steering will be displayed on No. 2 FDS.

Vertical Beam Sensing

Vertical channel beam sensing consists of a G/S level detector called the GSW. This detector senses proximity to the G/S beam and its output (logic 1 if within the threshold level and a logic 0 if outside) is utilized to develop vertical beam sense logic which brings the pointer into view to initiate capture and perform other logic switching. GSW latches in when beam error becomes less than 15 millivolts and unlatches if beam error exceeds 150 millivolts. The inputs to the GSW detector are G/S deviation and VBS logic. Before VBS becomes valid, GSW initiation level is less than 15 millivolts. After VBS becomes valid, GSW initiation level becomes 150 millivolts; therefore, to unlatch GSW requires deviation of more than 150 millivolts.

Vertical Pointer and Flag Logic

Previously, it was said that if the system supplying lateral steering information to the FDS was valid, the VP would be in view displaying steering commands and the Vertical Flag (VF) would be out of view indicating reliability. While this statement is sufficient for data flow explanation purposes, the actual monitoring of validities is somewhat more complicated than this implies. As shown in Figure 7-28, the VP output is connected through a logic switch to a minus 12-volt, dc off-scale bias source. This logic switch is gated open which biases the VP out of view or off-scale on the ADI.

This switch opens when $\overline{\text{VP}}$ logic is applied. Figure 7-27 shows the equation for VP logic. Two terms of the VP logic have not been previously discussed; they are System Validity (SV) and Lateral Manual Warning (LMW). The term SV represents an FDS validity composed of FDV and ISS. The FDV term is an internal voltage monitor, and ISS is a validity monitor of the attitude signals from TPLC. The remainder of the logic terms in the vertical pointer channel are primarily mode selection/validity and level switching.

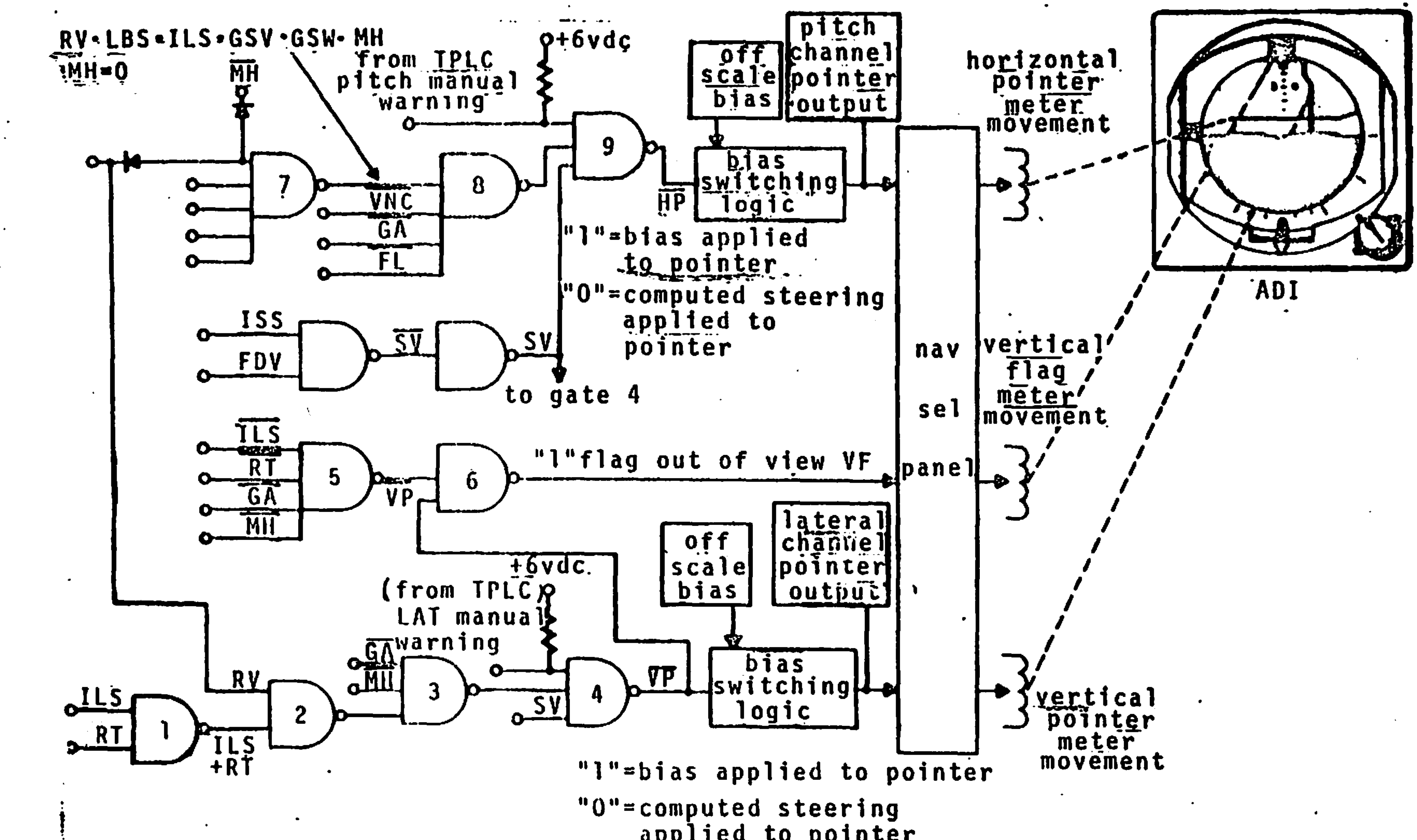
Horizontal Pointer and Flag Logic

The HP logic is similar to VP logic, i. e. if the system supplying pitch steering information is reliable, the HP is in view at the appropriate time. The equation for HP logic is also shown in Figure 7-28. The only new term is Pitch Manual Warning (PMW), which is analogous to LMW as used in the vertical pointer logic.

Displacement Pointer and Flag

The DF, as shown in Figure 7-29, is connected to G/S reliability signals when the computer is in ST + AT, ILS mode, and not in flare mode. Conversely, the flag is biased off scale in FL mode, ST + AT modes, and all other modes except ILS.

The DP is in view in VER NAV and ILS modes until FE. The logic equation shown in Figure 7-30 means that the DP is biased out of view during flare mode, ST + AT, and all other modes except VER NAV and ILS.



$\overline{HP}$=horizontal pointer out of view
HP=FL+GA+VNC+(GSV·GSW·RV·ILS.Lbs.MH).PMW·SV
VF=vertical flag out of view=VP+(ILS.RT.GA.MH)
VP=vertical pointer out of view
VP=GA+MH+[RV.(RT+ILS)].SV.LMW

FIGURE 7-28. VERTICAL FLAG, VERTICAL AND HORIZONTAL POINTER LOGIC

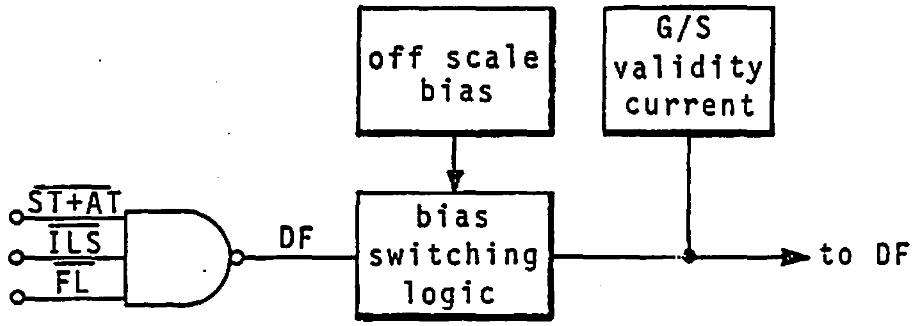


FIGURE 7-29. DISPLACEMENT FLAG LOGIC

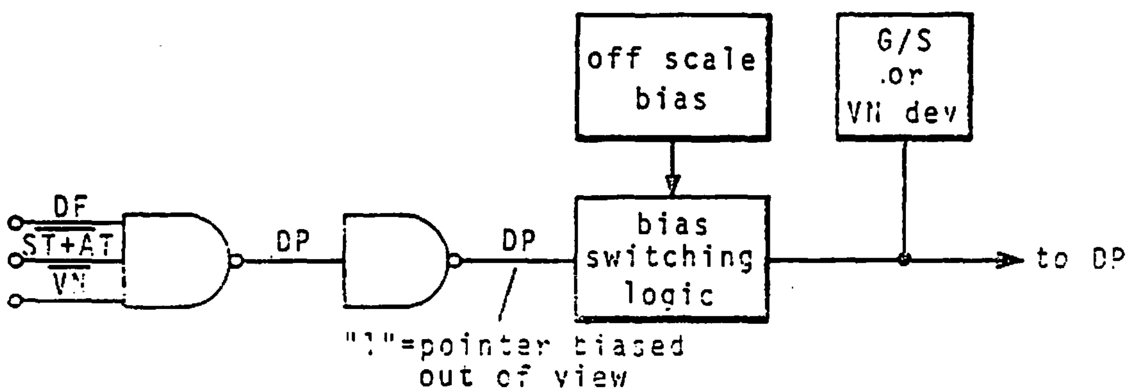


FIGURE 7-30. DISPLACEMENT POINTER LOGIC

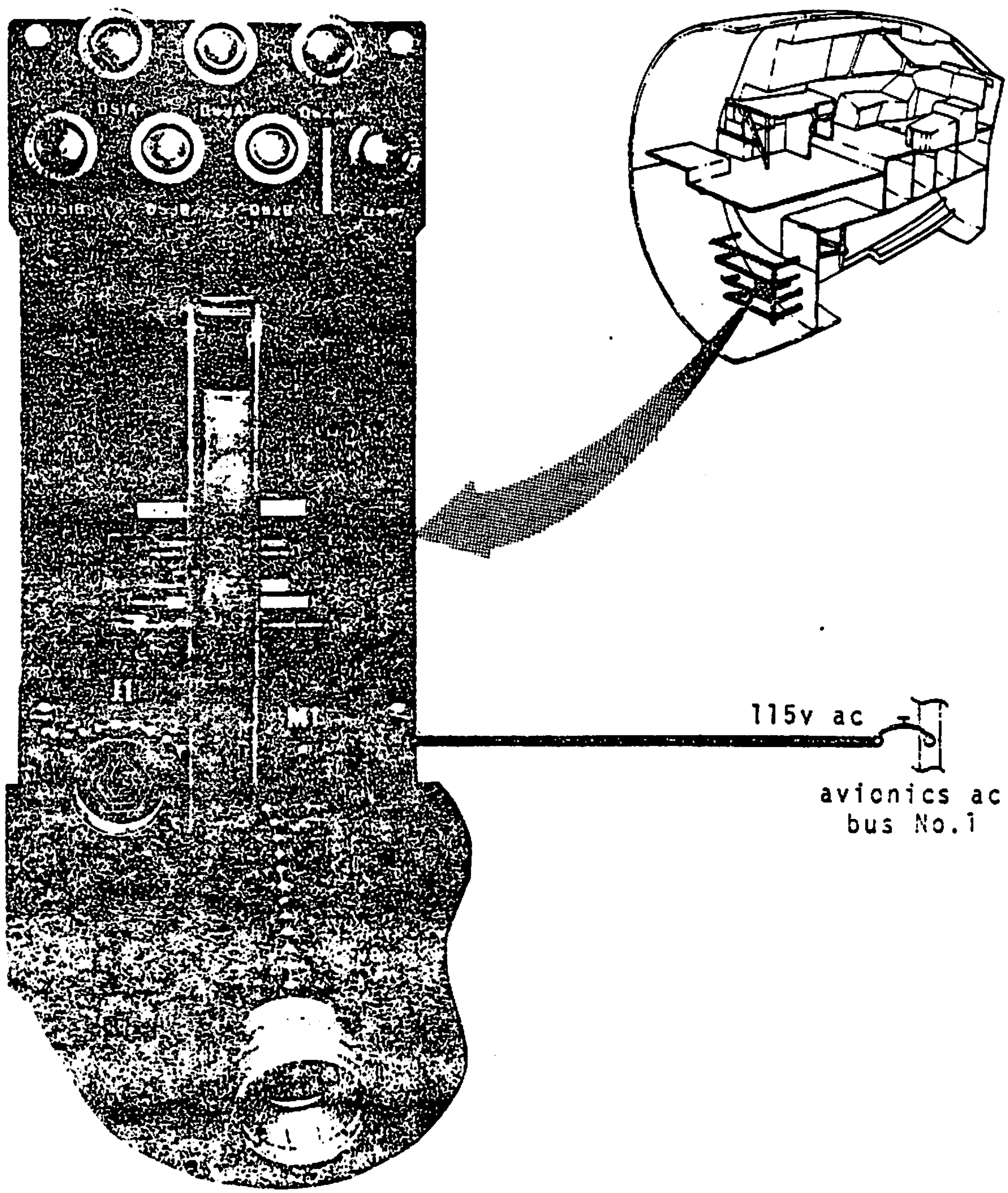


FIGURE 8-1. YAW DAMPER COMPUTER

YAW DAMPER (Y/D) SYSTEM

The Yaw Damper (Y/D) system is considered part of the Automatic Flight Control System (AFCS). It is a full-time system which supplies yaw stability augmentation as a function of yaw rate. The system also provides turn coordination.

COMPONENTS

The yaw damper system consists of the following seven units:

- o Yaw Damper Computer (Figure 8-1)
- o Yaw Damper Control Panel
- o Three Single-Axis Rate Gyros
- o Two Yaw Damper Servos

The manner in which these components are connected is shown in Figure 8-2.

SYSTEM OPERATION

The yaw damper control panel, also shown in Figure 8-2, contains an OFF-ON toggle switch, a TEST pushbutton, and two lights: one marked MONITOR and the other CHECK/RESET.

To energize the system, the ON-OFF switch is placed in the "ON" position, which applies necessary power and relay switching to fully energize the system.

When the TEST pushbutton on the right side of the panel is depressed, which allows a complete functional test of the system as shown in Figure 8-3, a holding coil is energized which holds the test button down. Then, for 10 seconds, the single-axis rate sensors are torqued off null, thus simulating a yaw condition. After the 10 seconds, the system is

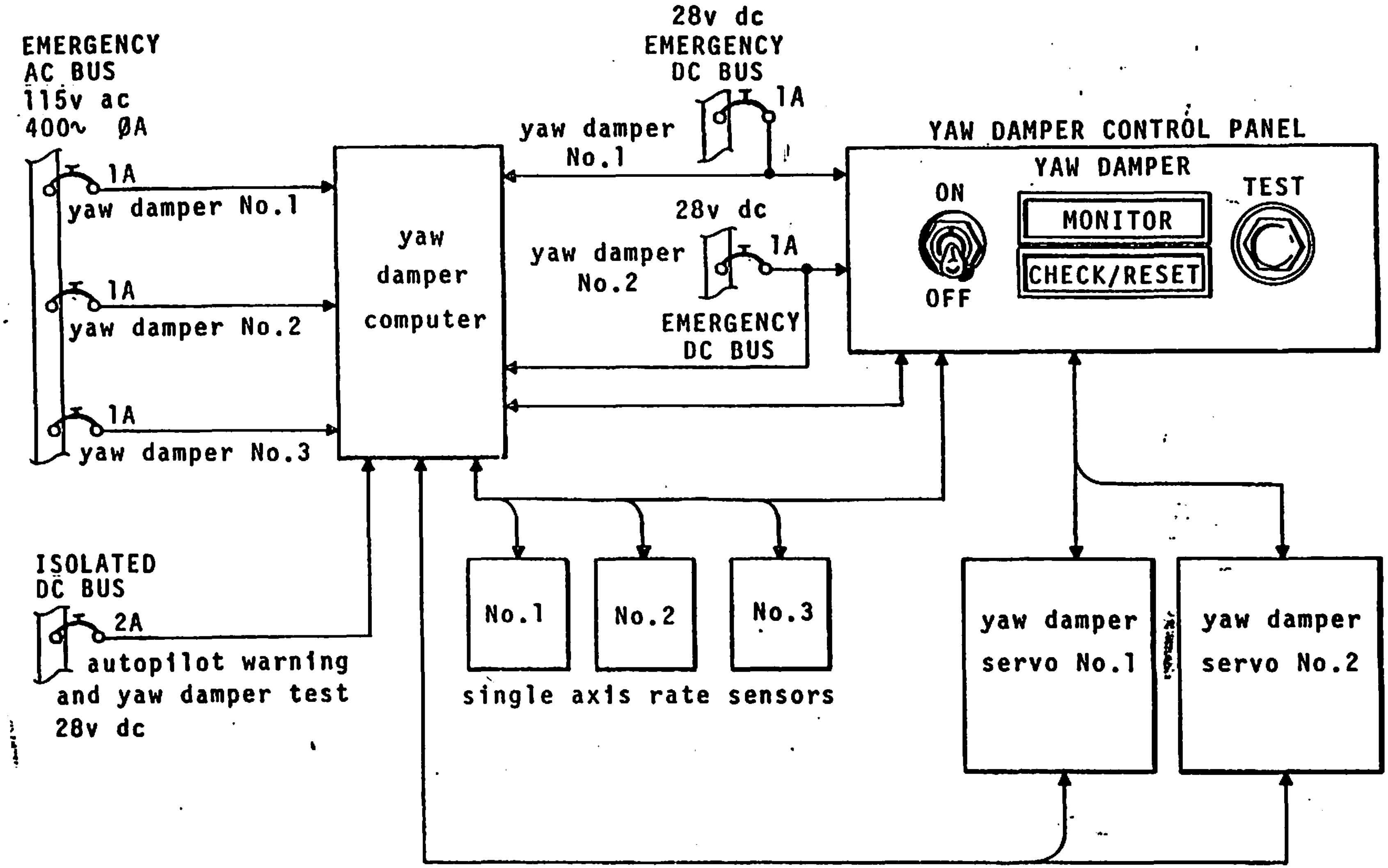


FIGURE 8-2. CABLING DIAGRAM

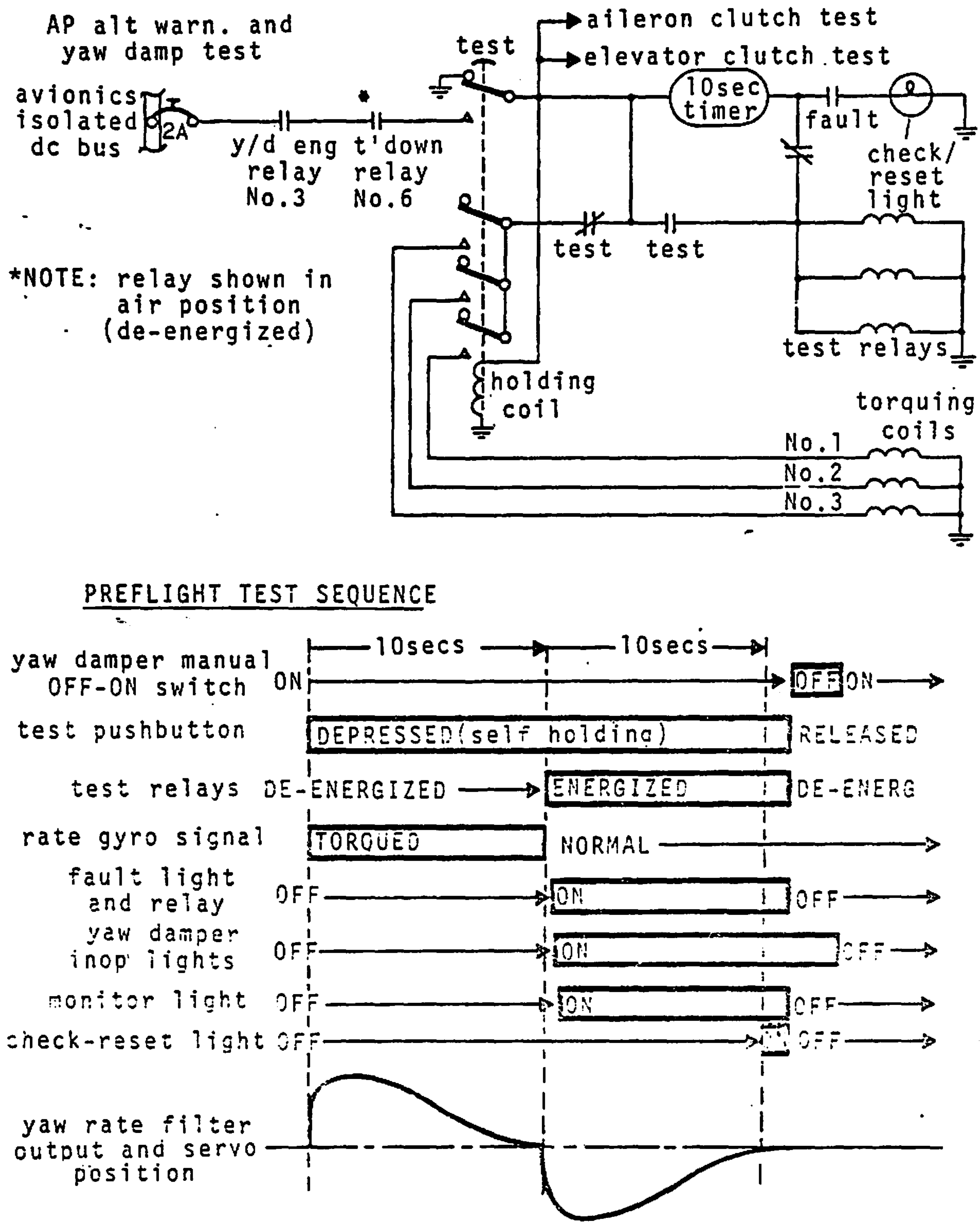


FIGURE 8-3. PRE-FLIGHT TEST CIRCUIT SEQUENCE

automatically faulted, which is done to determine if the system detects the fault. If such a fault exists, the following lights illuminate:

- o MONITOR light on the yaw damper control panel
- o YAW DAMPER INOPERATIVE lights on the pilot and copilot's instrument panel (outboard side)
- o YAW DAMPER FAULT light on the master caution annunciator panel.

Then, after 10 more seconds, the CHECK/RESET light on the yaw damper control panel illuminates, signifying the successful completion of the test. The system must be turned "OFF" and then back to "ON" in order to have it operational again.

The three single-axis rate sensors (gyros) measure the angular velocity of the aircraft in the yaw axis. These gyros are the only sensors used in the yaw damper mode. Any change of the yaw rate is sensed by the gyros and results in output voltage signals whose phase represent the direction of yaw and whose magnitude represent the yaw rate. These three signals are fed to the yaw damper computer. The two gyro inputs are used by the active channels of the computer while the third gyro input is used for the model channel (comparison channel within the computer) as shown in Figure 8-4.

The yaw damper computer receives the three gyro inputs, and, after processing them, sends command signals to the servomotors.

The servomotor, upon receiving the command signals from the yaw damper computer, cause the rudder to deflect, through electromagnetic clutches, in the proper direction to stabilize the aircraft and thus null the yaw rate signals. This action is accomplished through the rudder power package by means of mechanical linkage and hydraulic valves.

THEORY OF OPERATION

A block diagram of the yaw damper system is shown in Figure 8-5 which illustrates the essentially triple system configuration that provides adequate information to achieve a "fail operative" system, i. e. no single failure can disable the system. In the event of multiple failures that would disable the system, it will "fail-safe" by means of automatic disengagement.

Each of the three, single-axis rate gyro outputs are sent to a summing point at the input of three identical yaw rate filters. This yaw rate is then summed with a roll crossfeed signal (if the roll axis of the autopilot is engaged) to assist in turn coordination and aircraft control by means of rudder operation.

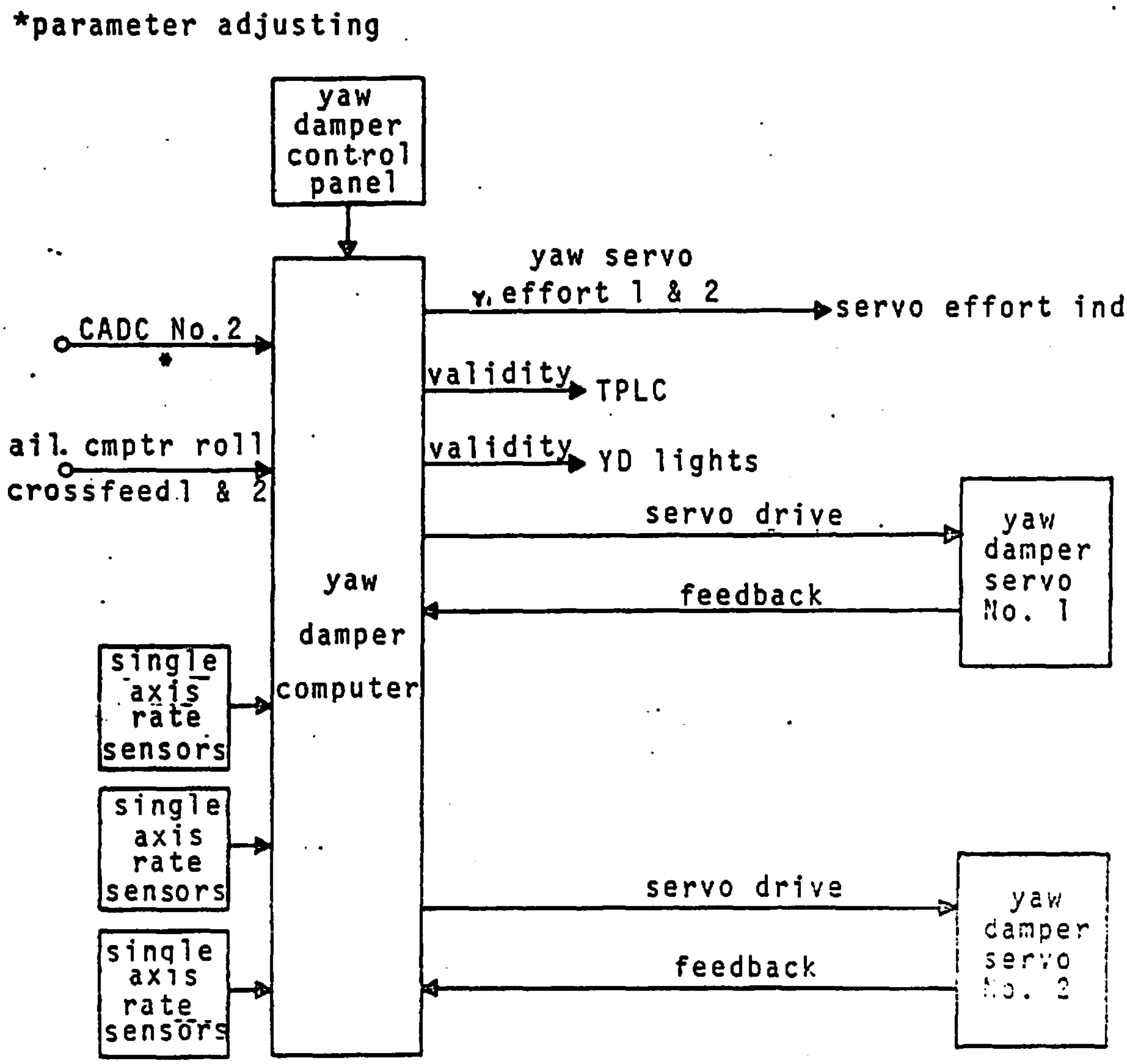


FIGURE 8-4. SYSTEM DATA FLOW

This summed signal is then fed to the yaw rate filters (filter and washout circuit). Each filter and washout circuit converts the ac input signal to a dc signal that can be shaped by the filter. After this shaping, a modulator changes the modified dc signal back to an ac signal. The filter adjusts the yaw damper gain (degrees of rudder command per degree of yaw rate) as required. At lower airspeeds, when dutch roll frequencies are low, the filter supplies a high gain. At normal cruise and high speeds, when dutch roll frequencies are higher, the filter provides low gain.

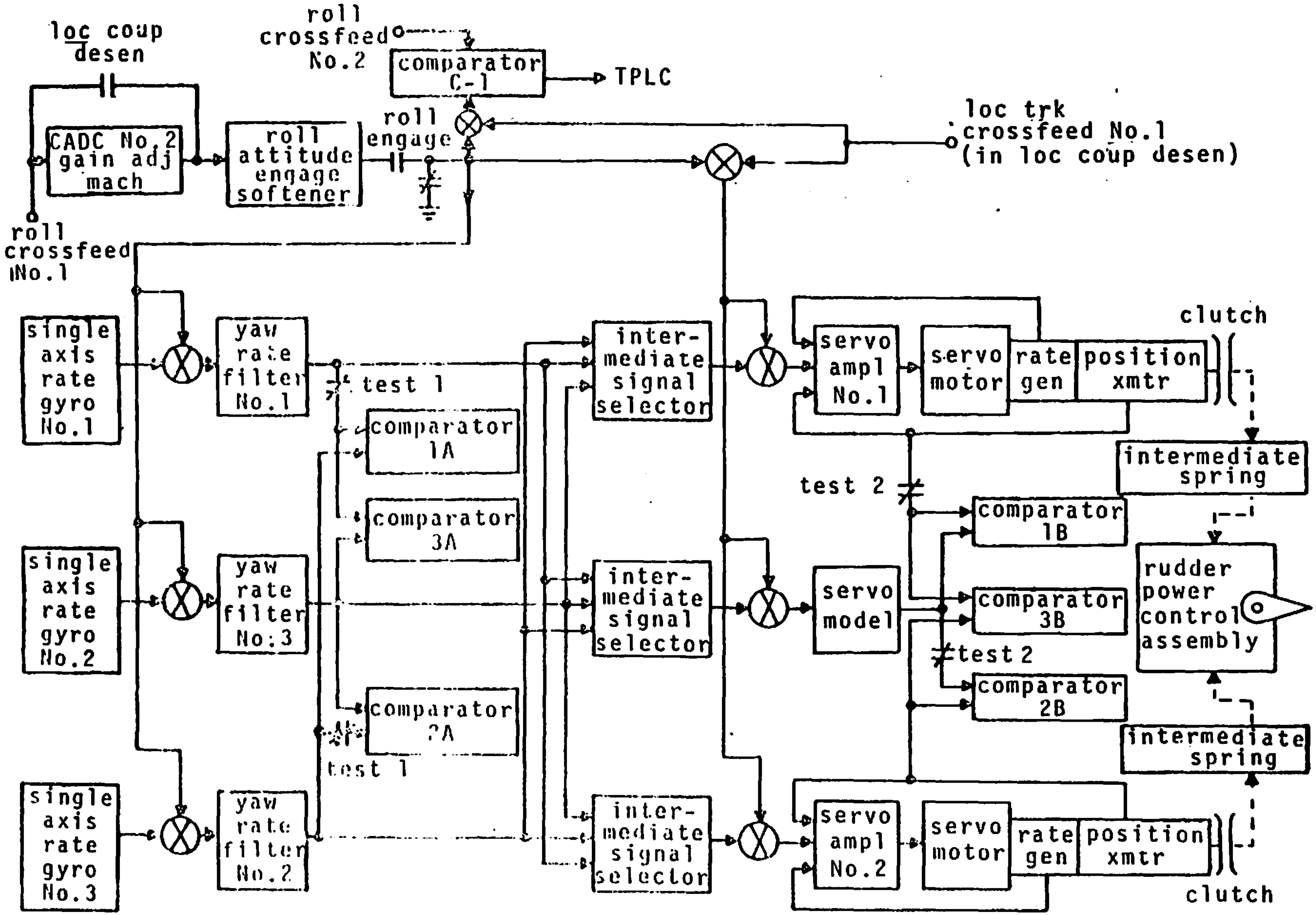


FIGURE 8-5. SYSTEM BLOCK DIAGRAM

The output of the filters is then sent to each of the three Intermediate Signal Selectors (ISS) and also to the "A" comparators. The comparators cross-compare the outputs of the three filters such that if these outputs differ by a fixed amount the comparator alarms, thus turning on the YAW DAMPER FAULT light. If the fault is in one of the active channels, it causes that channel to disengage. If no fault is detected, the comparators remain in a safe condition.

The ISS receives inputs from each of the yaw rate filters where the ISS then selects the intermediate signal of the three inputs; therefore, the outputs of all three ISS's are essentially equal. These outputs are then routed to a summing point where again, if the autopilot roll axis is engaged, the signal is summed with roll crossfeed. This summed signal is then fed to a servoamplifier where it is amplified and drives the servomotor. Another output of the servoamplifier is also a low-level dc signal that is sent to the servo effort indicator on the pilot's instrument panel. This dc signal represents the amount of servo effort but does not represent rudder displacement.

The servomotor input signal causes the motor to turn in the proper direction to compensate for the yaw condition. When the servomotor turns, it also turns a rate generator whose output is fed back to the servoamplifier for damping. Servoamplifier nulling is accomplished by a position transmitter which is driven by the servomotor. This signal is also routed to the "B" comparators which perform the same function as the "A" comparators. The output of each servomotor is connected to the rudder power pack through an electromagnetic clutch and an intermediate spring.

Roll crossfeed is developed in the aileron computer, when the roll axis of the autopilot is engaged, and sent to the yaw damper computer after first being gain-adjusted as a function of Mach in the No. 2 CADC. The signal is then passed through an "easy-on, easy-off" (engage softener) circuit and then is applied to the input of the yaw rate filters and the servoamplifier. Engage softening is necessary since the roll axis may be engaged or disengaged while the aircraft is in a bank.

This circuit prevents sudden rudder commands from being introduced which would cause undesirable aircraft maneuvers. Due to circuit design, the output of the engage softener that is fed to the yaw rate filters is 10 times greater than the output that is fed to the servoamplifier. This large command signal is introduced at the filter to give an initial rudder command. After the filter has washed out the initial large command signal (bank angle established), the small command signal that is applied at the servoamplifier is enough to hold the rudder at the desired position.

In LOC mode (autopilot engage), the roll crossfeed signal bypasses the Mach gain adjust. A signal representing LOC track is fed to the yaw damper computer as a strong crossfeed signal to the input of the servoamplifiers. This is done to give the autopilot tighter control over the aircraft and also to produce near flat turns for better beam tracking.

The comparators within the yaw damper computer have associated fault lights on the computer's front panel. These lights come on if the comparators alarm. The following table lists these lights and comparators along with cause of the fault:

Yaw Damper System	Has 7 comparators: "A" comparator will alarm if difference is greater than a signal proportional to approximately 1 degree/second yaw rate.
Comparator No. 1A, No. 3A	Failute of Rate Gyro or Yaw Rate Filter No. 1
Comparator No. 2A, No. 1A	Failure of Rate Gyro or Yaw Rate Filter No. 2
Comparator No. 3A, No. 2A	Failure of Rate Gyro of Yaw Rate Filter No. 3
Comparator No. 1A, No. 2A, No. 3A	If any one alarms, causes Y/D FAULT light to come "ON." If all 3 alarm causes total system disengagement.
Comparator No. 1B, No. 3B	ISS_1, Amp 1, or Servo 1; Fault and disengage
Comparator No. 2B, No. 3B	ISS_2, Amp 2, or Servo 2; Fault and disengage
Comparator No. 2B, No. 1B	ISS_3 or Servo model; Fault
Comparator No. 1B, No. 2B, No. 3B	If any one alarms, causes Y/D FAULT light to come "ON." If all 3 alarm, will cause both servos to disengage.
"YAW DAMPER INOPERATIVE" light	Will come on if current in both servo clutches is stopped by an automatic disconnect - manually turning the

Continued

	system off - or any other cause. The Y/D INOP light will not light in the event of disengagement of a single servo.
Fault Lights on front of Computer are "ON."	

DS1A	-	Comparator 1A alarmed.
DS2A	-	Comparator 2A alarmed.
DS3A	-	Comparator 3A alarmed.
DS1B	-	Comparator 1B alarmed.
DS2B	-	Comparator 2B alarmed.
DS3B	-	Comparator 3B alarmed.
DS4	-	Roll crossfeed comparator alarmed.

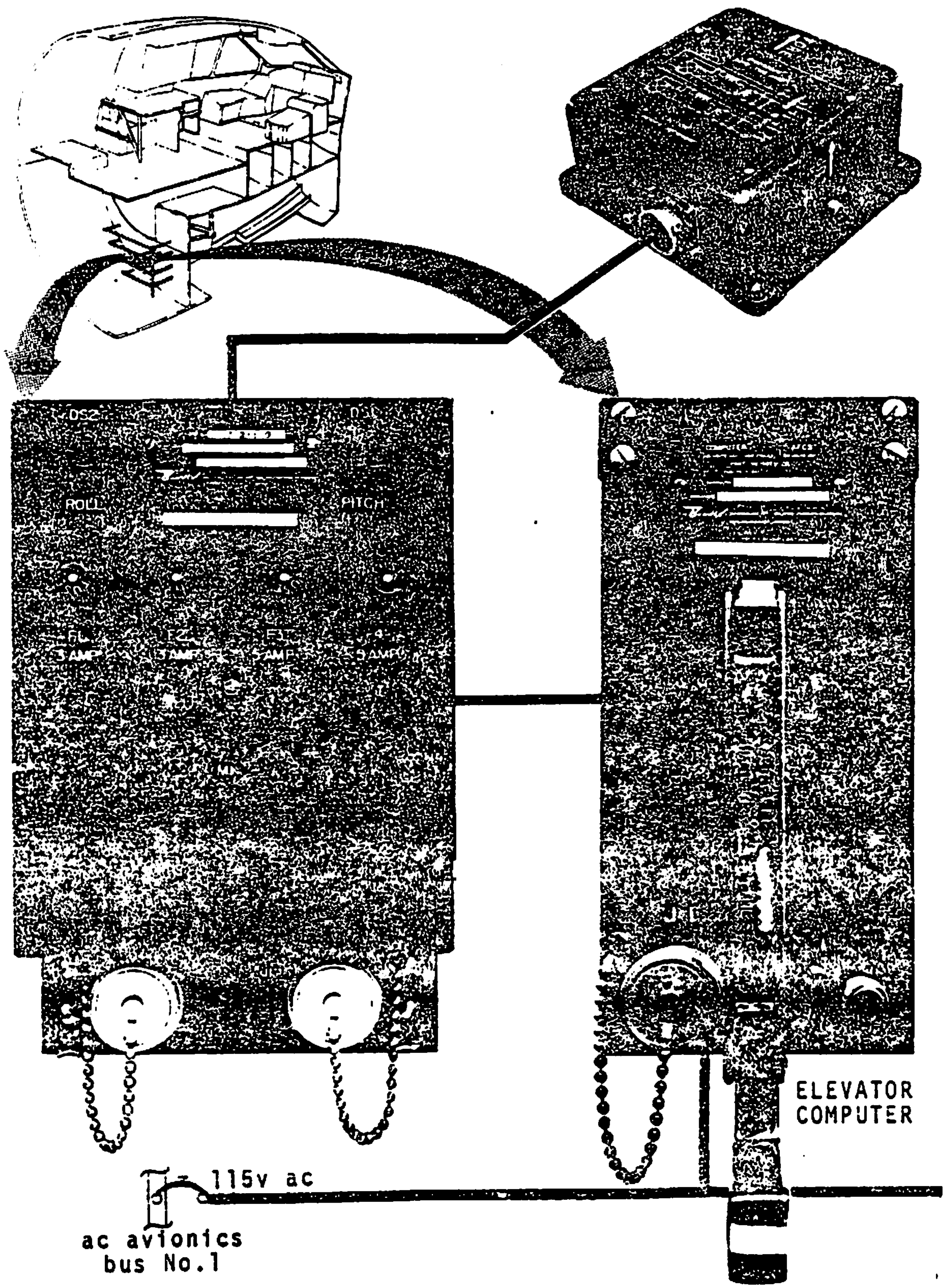


FIGURE 9-1. AUTOPILOT

AUTOPILOT (A/P

The primary function of the Autopilot (A/P), shown in Figure 9-1, is to provide basic aircraft stability in the selected flight attitude. It also controls the aircraft to fly and maintain a selected navigation mode and, in AWLS approach, to touchdown.

SYSTEM OPERATION

The following components make up the A/P system:

- Control Panel
- Trim Indicator
- Two Dual-Axis Rate Gyros
- Vertical Gyro
- Coupler
- Aileron Computer
- Elevator Computer
- Control Wheel (pilot's) Hub and Sensor

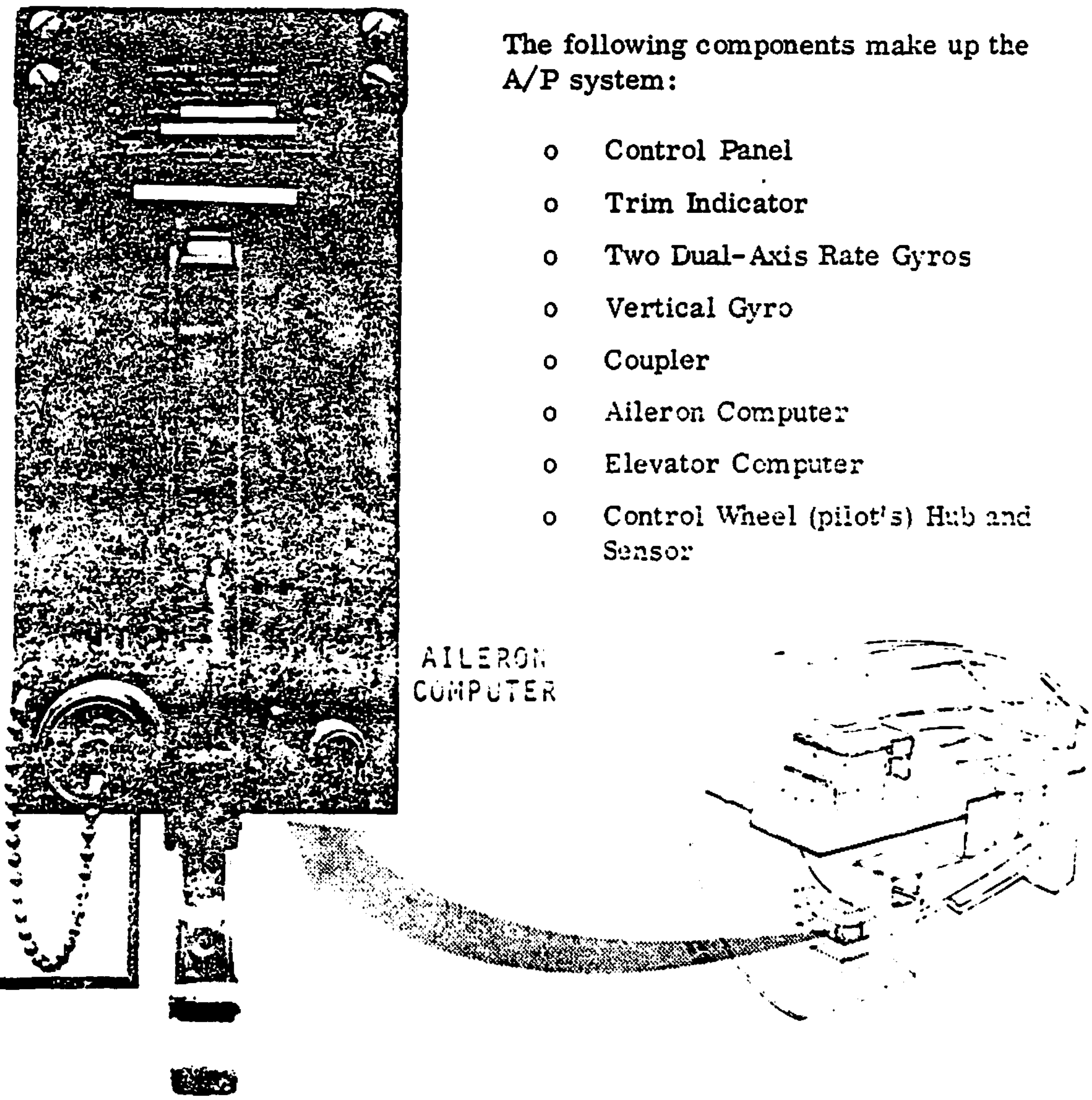


FIGURE 9-1. AUTOPILOT (CONTINUED)

- o Aileron Servo
- o Elevator Servo

A description of each of these components is given in the following paragraphs.

Control Panel

The control panel, shown in Figure 9-2, contains the following items, each of which is discussed separately:

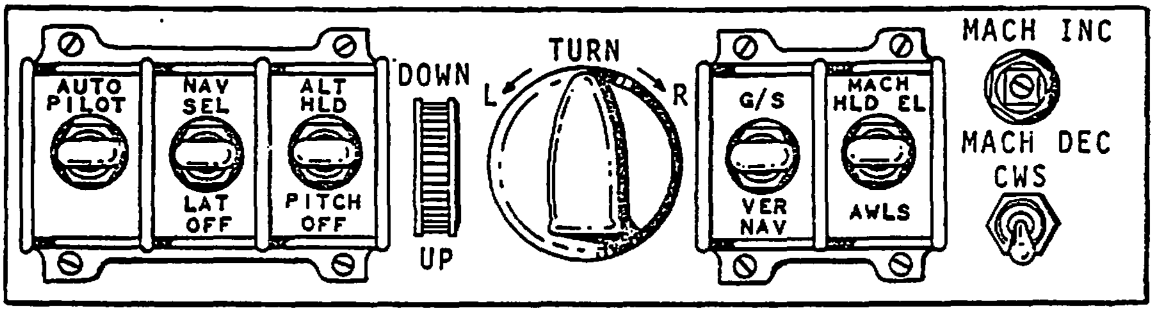


FIGURE 9-2. AFCS/AWLS CONTROL PANEL

Autopilot Engage Switch

The self-holding autopilot engage switch is used to engage or disengage the autopilot. When engaged and when no other mode is selected, this switch provides basic autopilot functions, i.e. pitch and roll stability. If the autopilot is engaged when the aircraft is flying at a bank angle of less than six degrees, the aircraft rolls down to wings level. If, however, the autopilot is engaged when the aircraft is flying at a bank angle of greater than six degrees, the aircraft maintains that bank angle. Returning the autopilot engage switch to the center position disengages the autopilot.

NAV SEL/LAT OFF Switch

The self-holding NAV SEL/LAT OFF switch has three positions:

- o "NAV SEL" (forward)
- o Off (center)
- o "LAT OFF" (aft)

In the "NAV SEL" position and with the autopilot engaged, the autopilot flies the navigation mode that was selected by the copilot (VOR, TACAN, DOPPLER, slope function. Selection of "VER NAV" position allows the autopilot to fly the vertical navigation flight path in much the same manner as glide slope. Selection of "VER NAV" while in altitude hold function does not cause the ALT HLD/PITCH OFF switch to drop out until VER NAV intercept is reached. Then selecting "ALT HLD" again causes the VER NAV switch to drop out. Returning the VER NAV switch to the off position disengages the VER NAV function.

MACH HLD EL/AWLS Switch

This switch has three positions:

- o "MACH HLD EL" (forward)
- o Off (center)
- o "AWLS" (aft)

Selection of "MACH HLD EL" (autopilot engaged) causes the autopilot to maintain the selected Mach number by means of a clutched mach synchro in the CADC No. 2. Engagement of "MACH HLD EL" when in altitude hold causes the ALT HLD switch to drop out and vice versa. Returning the MACH HLD EL switch to off (center position) disengages the mach hold function. Selection of the "AWLS" position may be made with or without the autopilot engaged since an AWLS approach may be made manually (Flight Director System) or automatically (Flight Director and Autopilot). This position arms the Test Programmer and Logic Computer to perform the necessary monitoring for an AWLS approach.

MACH INC/MACH DEC Switch

This switch, which is spring-loaded to off, has three positions:

- o "MACH INC" (forward)
- o Off (center)
- o "MACH DEC" (aft)

This switch is used to give small mach correction (± 0.05 mach) to the selected mach number when in MACH HOLD ELEVATOR function.

"CWS" (Control Wheel Steering) Switch

This is a two position toggle switch with off being aft, and "CWS" forward. Selection of the "CWS" position allows small forces, (2.5 pounds) applied to the pilot's control wheel to command changes of pitch or roll attitude of the aircraft. Returning the CWS switch to the off position disengages the control wheel steering mode.

Pitch Knob

As shown in Figure 9-2, the control panel also contains a pitch knob to command pitch attitude changes when in the basic autopilot mode: The adjustment is made by rotating the thumbwheel either fore or aft. This knob has no authority when any pitch mode of the autopilot is selected.

TURN Knob

The TURN knob provides turn (roll) commands when the autopilot is engaged. The knob rotates left or right and at its limits commands a bank angle of 38 degrees. This control has authority over any lateral mode selected.

Servo Effort Indicator

This panel contains YAW, aileron (AIL), and elevator (EL) servo effort indicators plus four autopilot system indicators as shown in Figure 9-3: A/P OFF, NAV OFF, G/S ARM, and G/S OFF warning lights.

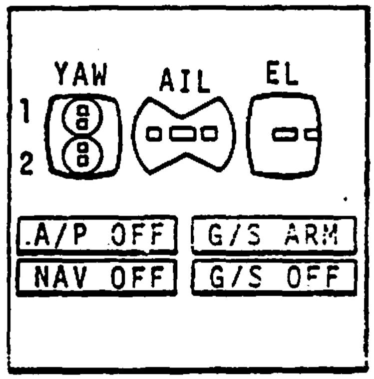


FIGURE 9-3. AFCS SERVO EFFORT INDICATOR PANEL

Yaw Trim Indicator

This unit indicates the amount of servo effort being applied to the rudder power pack when the yaw damper system is engaged. It is a dual indicator since there are two separate and independent yaw damper servos. Deflection of the indicators is left or right (about a center reference point) depending upon magnitude and direction of servo movement.

Aileron Trim Indicator

This unit indicates amount of servo effort being applied to the aileron servo when the autopilot is engaged. Deflection of the indicator is clockwise or counter-clockwise (center pivot point) depending on magnitude and direction of servo movement. Reference points (two) of the indicator are on the outside edge of the horizontal aileron bar. If the autopilot is engaged, "LAT OFF" selected, and the aileron clutch has been disengaged, the aileron trim indicator does not normally move since that axis has been disengaged; however, since the servo motor may still be displaced, some movement of the indicator may be noticed.

Elevator Trim Indicator

This unit indicates the amount of servo effort being applied to the elevator servo when the autopilot is engaged. Deflection of the indicator is up or down from a horizontal reference mark on the right side of the indicator depending on magnitude and direction of servo movement. If the autopilot is engaged and "PITCH OFF" selected, the elevator trim indicator may be displaced as described in the previous paragraph.

A/P OFF Light

This amber light comes on when the autopilot is disengaged by any means other than the control wheel disconnect switches, which are the normal means of disengaging. When the light comes on, if either disconnect button is depressed, the light goes out.

G/S OFF Light

This amber light comes on should the glide slope mode be selected, and a mode switching malfunction occurs such that the G/S switch on the autopilot control panel does not return to off.

NAV OFF Light

This amber light comes on if the NAV SEL/LAT OFF switch on the autopilot control panel is in the "NAV SEL" position and switching is not completed to engage a navigation mode.

G/S ARM Light

This green light comes on when proper mode switching has occured to engage the glide slope beam for an automatic fly-down by the autopilot. The light goes out upon engagement of the flide slope beam.

Two Axis Rate Gyros

There are two gyros installed in each aircraft to meet AWLS requirements. They provide roll rate to the aileron computer, pitch rate to the elevator computer, and roll and pitch rate to the R/GA computer.

Vertical Gyro

Although this gyro is called the autopilot vertical gyro, in the AWLS it is known as Vertical Gyro (VG) No. 3. It is self-contained having its own power supply, interlock, and provisions for roll and pitch erection cutout. Outputs of the gyro are roll angle and pitch angle which are supplied to the ISS input of the TPLC. This input, along with the two inputs from two flight director MD-1 vertical gyros, is supplied to the ISS circuit of the TPLC where the ISS selects the intermediate signal of the three roll and pitch attitudes. The TPLC then routes this signal to all using systems. Roll attitude is sent to the aileron computer, and pitch attitude is sent to the elevator computer. Another output of VG No. 3 is roll angle versine (1-consine of roll angle) which is sent directly to the elevator computer.

Pilot's Control Wheel Hub and Sensor

These units provide roll and pitch commands to the autopilot by means of an electromagnetic transducer within the pilots (L-H only) control wheel. The CWS mode is selected on the autopilot control panel.

Aileron and Elevator Servos

These units are ac split-phase motors having reduction gearing, a tachometer generator, a synchro position transmitter, and an electromechanical clutch. Positioning the servo motor causes appropriate deflection of the respective control surfaces.

Coupler

The autopilot coupler, shown in Figure 9-1, contains separate pitch and roll channels.

Roll channel input signals follow:

- No. 2 C-12 Gyro Heading
- Preset Course and Heading from No. 2 HSI
- No. 2 VOR and No. 2 TACAN
- ASN-35, ASN-24 Deviation
- No. 1 and 2 LOC Deviation
- Radar Altitude No. 1 and 2 from TPLC
- Roll Attitude from TPLC (ISS 2 and 3)

These input signals are processed by the coupler (depending upon NAV mode selected) to develop a steering command signal for the aileron computer.

Pitch channel input signals follow:

- No. 2 CADC Altitude Rate, Clutched Altitude and Clutch MACH
- VER NAV Command
- Flare Error from Flare Computer

- No. 1 and 2 Glide Slope Deviation
- Vertical Acceleration (AN)
- MACH BEEPER from A/P Control Panel

These input signals are processed by the coupler (depending upon mode selection) to develop a pitch command signal for the elevator computer.

Both channels of the coupler provide dual output signals to the respective computers. Each computer then contains two channels: one active and one model. During an AWLS approach, the active and model channels are monitored for system faults by comparators. The coupler contains eight comparators and two associated fault lights. The fault lights on the front panel of the coupler are labeled ROLL and PITCH. These lights illuminate if any of the associated comparators detect a fault and switch to an alarm state. The comparator fault logic also goes to the TPLC.

Aileron Computer

An aileron computer, shown in Figure 9-1, processes roll signals from the coupler. Input signals to the aileron computer follow:

- Roll Rate from No. 1 and 2 Rate Gyros
- Roll Attitude from TPLC (ISS 2 and 3)
- Roll CWS Sensor
- Active and Model Commands from Coupler
- Turn Knob Signal

Depending on mode of autopilot operation, these signals are processed by the computer which, in turn, produces four outputs as well as driving the control surfaces. The four outputs are as follow:

- Roll crossfeed (active and model) to the yaw damper computer for turn coordination when the yaw damper system is engaged.
- Localizer track crossfeed to the yaw damper computer when the autopilot is tracking the localizer beam (for flat turn control).
- Roll Command to the flight director No. 2 for display and AWLS approach monitoring.
- Servo effort to the trim indicator.

There are two AWLS comparators in this computer; if either of these comparators alarms it causes a fault light on the front of the computer to come on as well as sending this fault logic to the TPLC.

Elevator Computer

An elevator computer, shown in Figure 9-1, processes pitch signals from the coupler. Input signals to the elevator computer follow:

- Horizontal Stabilizer Position
- Pitch Rate from No. 1 and 2 Rate Gyros
- Pitch Attitude from TPLC (ISS 2 and 3)
- Band Versine from VG No. 3
- Pitch CWS Sensor
- Active and Model Commands from Coupler
- Pitch Knob Signal
- Flap Position

Depending on the mode of autopilot operation, the elevator computer processes these signals to produce four output command signals as well as driving the elevator control surfaces. These four outputs follow:

- Up and down stabilizer trim to the horizontal stabilizer actuator.
- Up and down stabilizer trim cut-out to the horizontal stabilizer actuator.
- Pitch command to the flight director No. 2 for display and AWLS approach monitoring.
- Servo effort to the trim indicator.

There is an active and model channel in the elevator computer to provide proper monitoring in an AWLS approach. There are two AWLS comparators in the computer which cause a fault light on the front of the computer to illuminate if either comparator alarms. This fault logic is also sent to the TPLC.

THEORY OF OPERATION

The overall tie-in of various aircraft systems and components is shown in Figure 9-4. A detailed description of autopilot operation by mode, follows:

Roll Axis

Basic roll stabilization of the autopilot system is shown in Figure 9-5 and is accomplished in the aileron computer. The electromechanical synchronizer unit does not contain any internal position feedback for its synchro output. Positioning of this unit and amplitude of its output are determined by external feedback loops.

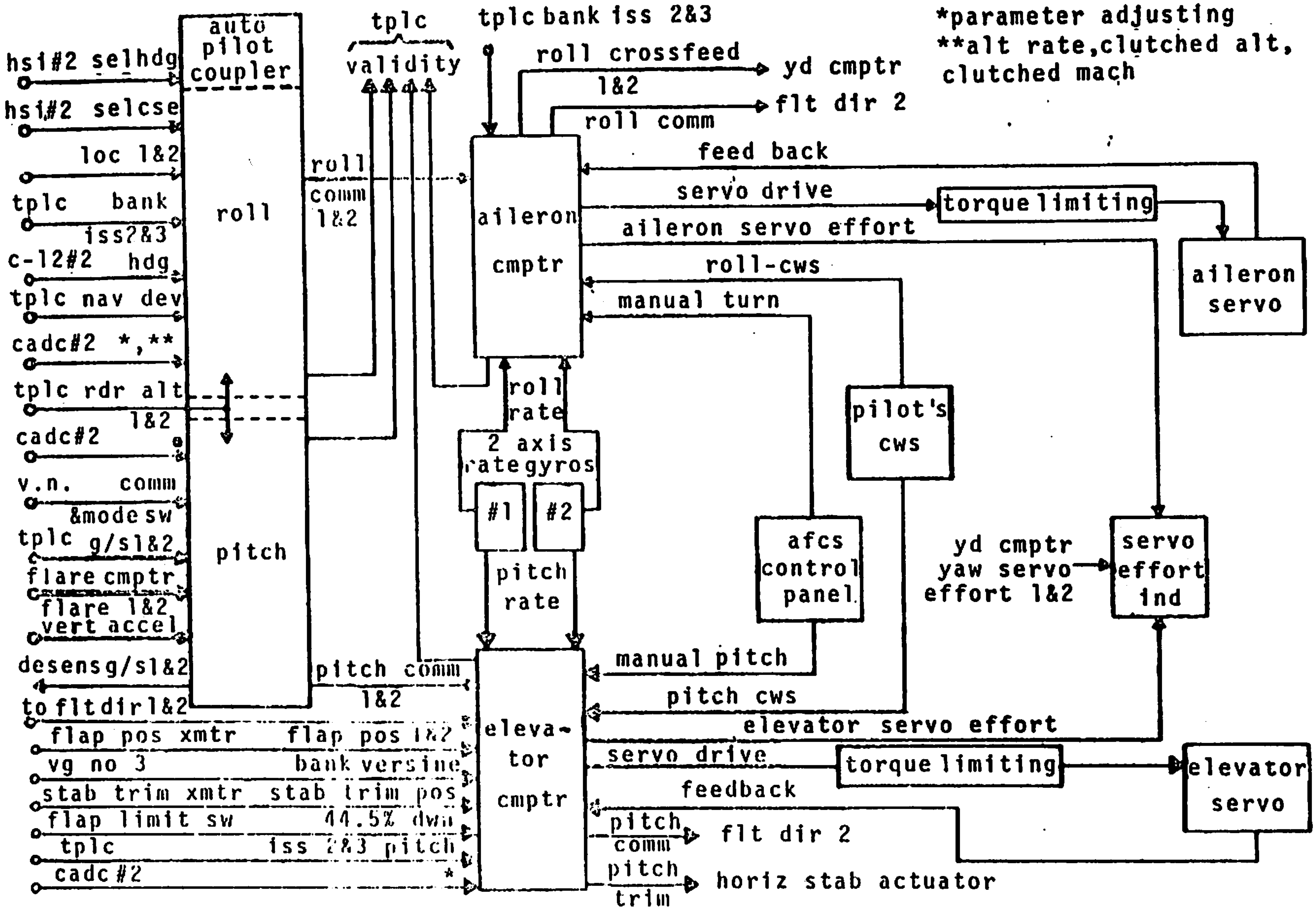


FIGURE 9-4. SYSTEM DATA FLOW

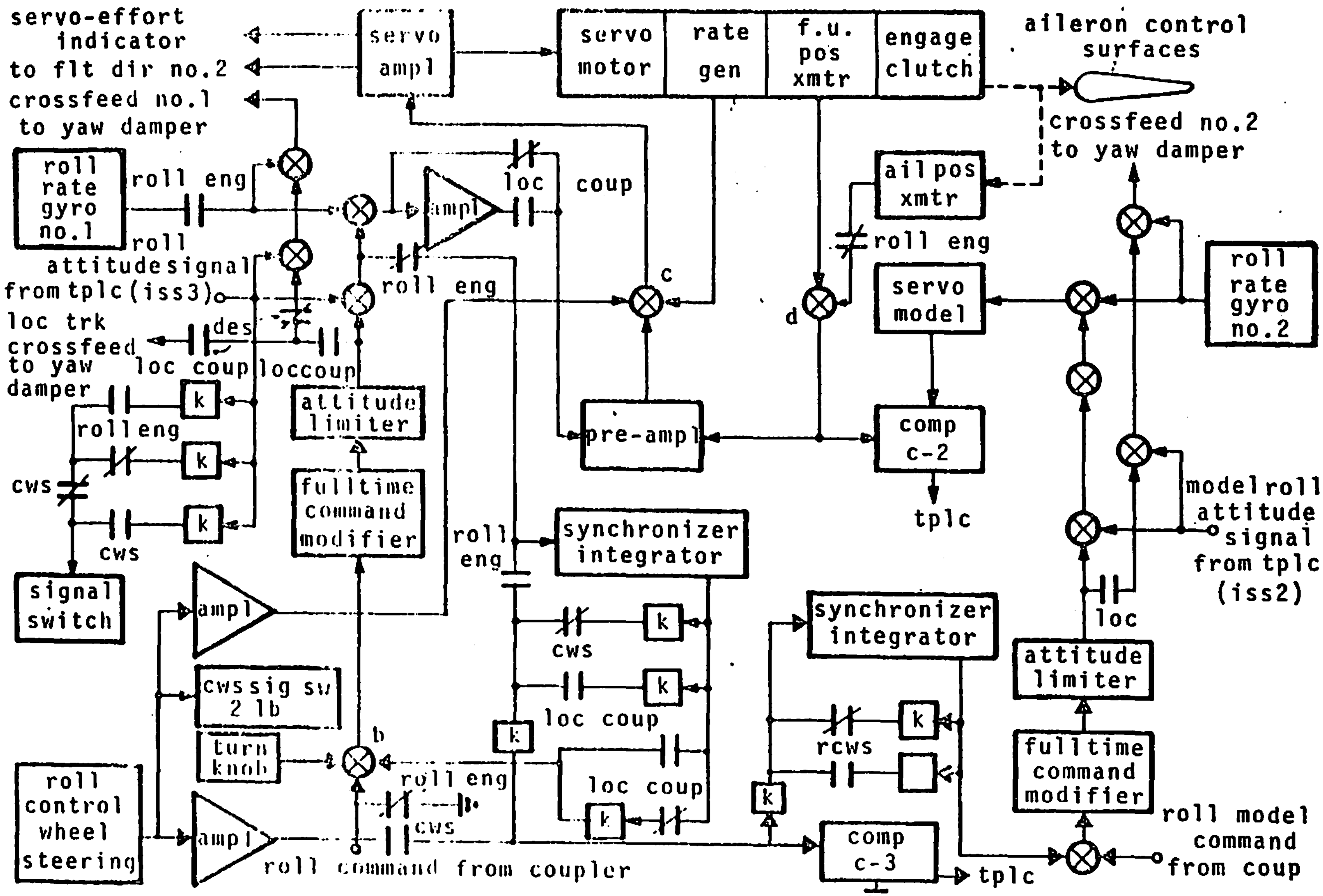


FIGURE 9-5. AILERON CHANNEL BLOCK DIAGRAM

Prior to engaging the system, the roll synchronizer, operating through the normally closed contacts of the roll engage relay, keeps the roll attitude input to the preamplifier at a null (junction "A"). It does this since its input signal is a sum of commands, roll attitude, and the synchronizer output. The synchronizer then drives until the input to the synchronizer is nulled. The preamplifier in the servo system (both aileron and elevator) is used for isolation and its output is routed to the servo amplifier which then drives the ac servo motor. The servo motor drives a rate generator, whose output is fed back to the servo amplifier input to properly damp the motor operation. It also drives a Follow-Up transmitter whose output is added to the other signals at the servo preamplifier input so that a command results in a predetermined servo rotation and surface deflection. In the disengaged condition, the servo is not clutched to the surface, and the output of the aileron position transmitter, which is coupled to the servo mount drum, is summed with the servo position follow-up (junction "D"). Prior to engagement, both inputs to the preamplifier (junction "A" and "D") are at a null; The system is therefore completely synchronized.

When the roll axis of the autopilot is engaged at bank angles of less than six degrees (as sensed by the roll signal switch), the synchronizer is allowed to follow up on its own output through the energized roll engage relay and the deenergized roll CWS relay. This synchronizer roll-down action, which is softened by the Full-Time Command Modifier (FTCM), results in the aircraft's assuming a wings level attitude, since the roll attitude signal is now unopposed (except by steering commands), and the aircraft is commanded to null the signals. If the system is engaged at a bank angle greater than six degrees, the synchronizer is locked, and the bank angle at the time of engagement is held until commanded otherwise.

All roll angle commands are softened by the FTCM introducing them as a slightly time-delayed ramp. This action eliminates abrupt command inputs and limits roll rates to 20 degrees per second (maximum) in control wheel steering mode and 4.8 degrees per second at all other times. All bank angle commands are also magnitude limited by the bank limiter which varies the maximum bank angle according to the autopilot mode in use.

Basic Aileron Control

Moving the turn knob on the autopilot control panel causes the roll synchronizer to roll down through the action of switching interlocks if it has not done so already. It also introduces a bank command signal into the FTCM, resulting in servo commands, until the roll attitude signal and the turn knob signal are at equilibrium. At which time, the airplane is flying at a steady bank angle proportional to the versine of turn knob rotation.

Roll CWS - Cruise

When CWS is selected on the autopilot control panel, the force sensor output drives the synchronizer at a speed proportional to the force applied to the pilot's wheel. Bank angle commands are then generated by the synchronizer which feeds them to the FTCM and limiter. A small part of the sensor output is applied directly to the servo amplifier input as an anticipation signal to improve the response and feel of the system. When a desired bank angle is reached and the force is removed from the control wheel, the synchronizer locks if the bank angle is over three degrees. This action causes the aircraft to remain at that bank angle until opposite force is applied or the turn knob is moved out of detent (turn knob having more authority than CWS). If control wheel force is removed at a bank angle of less than three degrees, the synchronizer rolls down, leveling the airplane, and the heading hold mode automatically engages.

Steering Inputs

All steering inputs to the aileron computer from the coupler are added to the signal chain. These inputs command aileron deflection to roll the aircraft in the appropriate direction until the steering input is nulled by the other signals in the signal chain, which, on a steady state basis, consists of roll angle signals only (with the exception of CWS in the LOC mode). Thus, the airplane assumes a bank angle that is proportional to the steering command up to the applicable bank limit. It assumes this bank angle at a rate not exceeding the allowable ramp rate from the FTCM.

Lateral Axis Model

All significant portions of the lateral axis used in AWLS are duplicated by functionally identical parallel circuits. These circuits utilize independent sensors, and the output of the model channel is monitored by a comparator. Figure 9-5 shows the roll model channel with all necessary inputs. This servo model is an electromechanical device simulating the dynamic and static characteristics of the aileron servo. A position assumed by the servo model follow-up closely corresponds to that of the actual aileron servo. The comparator (C2) monitors the two signals and will alarm if they do not agree within reasonable tolerances. Roll CWS is monitored by comparator C3, which alarms if the signal level exceeds the maximum sensor output of 0.50 volt plus tolerance. No attempt is made to provide performance monitoring of the CWS, since significant degradation of performance is apparent to the user. These comparators operate full time. However, the logic, which interprets the comparators output and the automatic disengage of functions, is contained in the TPLC, and is not armed until APPR ARM. Thus, prior to APPR ARM, the autopilot is not inherently fail-safe, and the normal servo torque limiting is the only absolute protection provided. This comparator operation is true for all the comparators in the autopilot system.

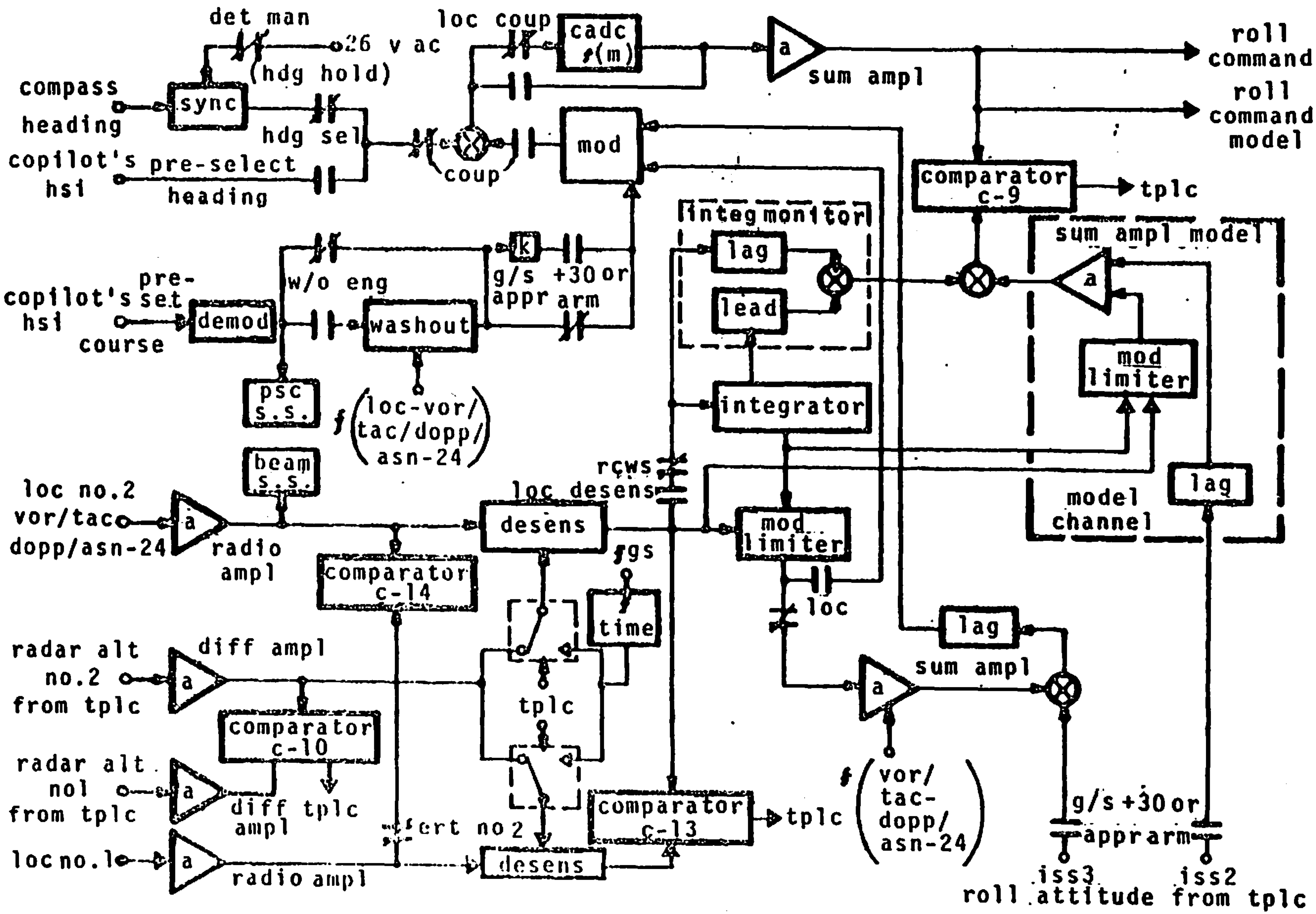


FIGURE 9-6. ROLL COUPLER BLOCK DIAGRAM

Coupler Steering Inputs

Steering signals are computed in the coupler and sent to the aileron computer through a mach gain adjustment in the No. 2 CADC. As shown in Figure 9-6, this gain adjustment is arranged to provide low gain at low speed and high gain at high speed, which causes the turn rate to be proportional to the error signal over the entire aircraft speed range. When LOC mode is engaged, the mach gain adjustment is bypassed.

Heading Hold Mode

Heading hold mode is the primary directional mode of the autopilot. However, NAV SEL, CWS, and turn knob controls are provided. Prior to engagine, the A/P DET MAN relay is deenergized (heading hold disengaged), and the compass synchronizer is operating as shown in Figure 9-6. When HEADING HOLD is engaged, the DET MAN relay energizes. The compass synchronizer output is phase-shifted, amplified, and routed through No. 2 CADC mach gain adjust to the aileron computer.

NAV SEL Modes

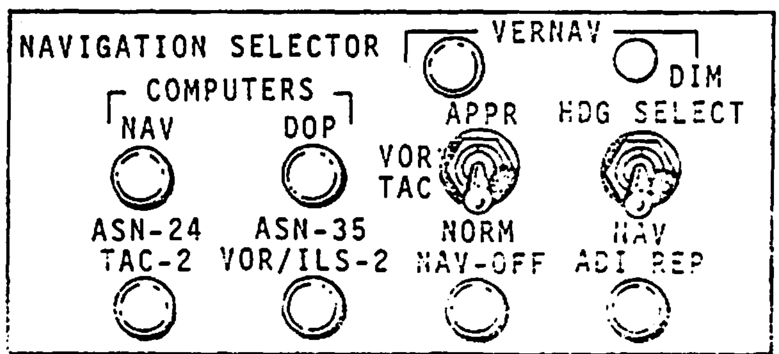


FIGURE 9-7. COPILOT'S NAV SELECTOR PANEL

The pushbuttons on the copilot's navigation selector panel, as shown in Figure 9-7, perform A/P mode switching when the HDG SELECT/NAV switch is in the "NAV" position or when the NAV SEL/LAT OFF switch is in the "NAV SEL" position. Modes that may be selected on the navigation selector panel are VOR-ILS, TACAN, ASN-35, and ASN-24. There is no autopilot mode associated with the ADI REP (ADI Repeat) button.

Heading Select Mode

When the HDG SELECT/NAV switch is in the "HDG SELECT" position, the HDG SEL relay in Figure 9-6 is energized, which allows a magnetic heading to be preselected on the copilot's HSI. Heading signals from a differential synchro in the HSI produce an error signal that is proportional to the difference between selected heading and actual heading. This is used as a heading hold signal and commands the aircraft to turn to, and hold, the preselected heading. The Pre-Select Heading (PSH) may be changed by moving the HEADING SET knob on the HSI.

NAV OFF

When the "NAV OFF" button on the navigation selector panel is pushed, the NAV SEL/LAT OFF switch on the autopilot control panel returns to the center position if it is in the "NAV SEL" position. If this switch is moved to the "NAV SEL" position when the "NAV OFF" button is pushed, the NAV OFF light on the autopilot trim indicator panel illuminates.

VOR-Tacan

Although selected separately and utilizing separate beam displacement inputs VOR and Tacan modes are identical within the autopilot. To engage either mode, the appropriate receiver tuning must be accomplished, the desired radial selected (COURSE SET knob on the HSI), and either the VOR or Tacan button on the copilot's navigation selector panel must be pushed. A dc signal proportional to the beam error is the output of the selected receiver and is measured as the difference between aircraft bearing to the station and the desired track bearing. This beam error signal is introduced, as shown in Figure 9-6, through a desensitizer, which has a gain of one at this time, to the modulator limiter. Here, it is gain-adjusted and smoothed by the data filter (lag filter with a 15-second time constant) and is added to the PRE-SET course signal in the modulator.

The track mode is established when beam displacement error is less than 12 millivolts, and the course error is less than 17.5 degrees. When the beam displacement error is greater than 12 millivolts and the course error is greater than 17.5 degrees, the PRE-SET course washout circuit is not a washout. When the two requirements are met, as sensed by the PRE-SET COURSE signal switch and the beam signal switch, the 90-second washout is activated. Steady-state course errors are reduced to zero by the washout which allows the aircraft to assume whatever heading is necessary (crab) to hold beam center. When the aircraft is flying a course to intercept the beam, initial engagement occurs at 75 millivolts (1 dot on HSI) displacement.

Over-station beam sensing is provided to prevent erratic maneuvers when the aircraft is in the cone of confusion. If, when in the track mode, beam displacement is in excess of 65 millivolts, (less than one dot displacement on HSI), logic changes the gain of the amplifier from 1.17 to 0.164. The aircraft now follows the erratic beam very slowly due to the data smoother and the very low gain with a small bank in one direction and a small bank in the opposite direction. Thus the aircraft leaves the cone of confusion reasonable close to the beam center. When the displacement is less than 65 millivolts for 15 seconds, the amplifier gain returns to normal and station passage is considered completed.

ASN-24 and ASN-35

Both of these modes are identical within the autopilot system. They are similar to the VOR/Tacan modes in operation except for the following differences:

- o Initial engagement occurs at 2.3 miles crosstrack (1 1/3 dots on the HSI).
- o Course washout criteria are 2.5 miles crosstrack and 17.5 degrees pre-set course error.
- o There is no over-station sensing in the ASN-24 or ASN-35 modes.

HSI and NAV SEL Slaving

Since the autopilot signals are from the copilot's sensors, the copilot's navigation selector panel normally controls autopilot switching. However, for the convenience of pilot operation, a No. 2 HSI heading and course set switch has been installed. When this switch is in the "SLAVE" position, the HEADING SET and COURSE SET functions of the copilot's HSI are slaved to the pilot's HSI. Thus the pilot may select course or heading inputs to the autopilot by means of his HSI. This action drives the copilot's HSI and changes the signal inputs to the autopilot.

As a further convenience to the pilot, when the HSI No. 2 HEADING and COURSE SET switch is in "SLAVE," the HDG SELECT/NAV switch on the pilot's navigation selector panel operates the autopilot switching. Pushbutton selection of the autopilot NAV modes remains on the copilot's navigation selector panel at all times. When both VHF NAV receivers are tuned to a LOC frequency and the VOR/ILS button is pushed on both navigation selector panels, the HSI SLAVE function is automatically accomplished.

LOC Mode

This mode is initiated by selecting the proper LOC frequency, selecting the inbound runway heading on the HSI, pressing the VOR/ILS button on the navigation selector panel, placing the HDG SELECT/NAV switch to the "NAV" position, and engaging the NAV SEL switch on the autopilot control panel. Autopilot LOC mode now engages when the beam error is less than 175 millivolts (slightly over 2 dots on the HSI). As shown on Figure 9-6, the LOC input signal is routed to the [illegible] where it is reduced, as a function of radar altitude, from a gain of 1 at 1000 feet to a gain of 0.45 at 0.0 feet. Desensitized LOC beam displacement signal is then applied to the modulator limiter circuit, after which it is gain adjusted, bypasses the amplifier and lag circuit (data smoother), and summed with the preset course signal in the modulator. Washout of the preset course signal does not occur until the 75-millivolt and 17.5-degree course signal sensors are satisfied. When the preset course is allowed to washout, the aircraft assumes the course required to track the beam center. At thirty seconds after glide slope engage (or at approach arm when the monitor system is used), the washout preset course signal is greatly reduced in gain, and bank angle signals are routed to the data smoother resulting in a lagged roll signal output which is used for primary beam damping. Using very low gain in the washed out preset course signal greatly enhances the system's capability of handling changing atmospheric conditions with minimum beam displacements and adequate lateral dumping is provided by the lagged roll.

A relatively low speed, low authority integration is also used to provide control of system nulls and offset errors that might arise as a result of roll error inaccuracies, which are magnified by the lagged roll circuit. This integrator is a purely electronic device and keeps itself synchronized when not in use. It provides the final degree of control required to assure that consistent performance to touchdown, close to beam center, can be achieved.

Roll CWS Approach

Roll CWS is operative in the LOC mode when the CWS switch is on. When a force over 2.0 pounds is applied to the pilot's control wheel, a roll command is generated by the synchronizer proportional to the applied force. The rate of CWS command available in the LOC mode is somewhat faster than in cruise since the full rotation of the synchronizer is 60 degrees in the LOC mode and scaled down to 38 degrees in the cruise mode. In cruise, the commanded bank angle is held (if greater than 3 degrees) until commanded otherwise, but in the LOC mode the synchronizer no longer holds command inputs indefinitely but synchronizes itself through the LOC COUP contacts on release of applied force. Bank angle commands resulting from control wheel force are not linear due to the non-linear roll CWS amplifier. A 3-pound force results in a 12-degree bank command where as a 10-pound force would require a 120-degree command output from the synchronizer, which is in excess of the synchronizer capability. However, the result is that small bank commands can easily and effectively be held, and yet rate commands are available with the application of very little more force. Bank limits in the LOC mode are 30 degrees in capture and 7.5 degrees in track. However, when a roll force over 2.0 pounds is applied, the limit reverts to 30 degrees until the force is removed. This gives roll CWS complete authority in roll throughout the coupled approach.

Roll Coupler

Inputs are provided to the roll coupler from both LOC receivers as shown in Figure 9-6. These inputs are compared by comparator C-14 to ensure that they agree. Both LOC inputs are desensitized in the coupler. These desensitizers are normally driven by a radar altitude signal; however, if the radar altimeter output goes invalid, the desensitizers are switched to a time function. Glide slope gain would then be reduced from 1 at G/S engage to 0.25 after 120 seconds. LOC gain is reduced from 1 to 0.45 after 120 seconds. After 120 seconds both signals remain constant. Outputs of the desensitizers are monitored by C-13 whose validity output is interpreted in the TPLC, and appropriate action is taken.

The monitored output of one desensitizer is used as an input to two modulator limiters: one provides signals to the operating channel, the other to the model channel. A roll input (SS 2) which is lagged and added to the beam error signal is also contained in the model channel. This channel does not include the preset-course signal, since the gain of this signal is very low during the final approach phase and does not contribute voltage in excess of the threshold of comparator C-9. Summation of the localizer integrator output is done in both the active and model signal chains in order to prevent an unbalance between the two. Integrator monitoring is provided by the summed lead and lag circuits. The integrator drives at a rate which is proportional to the input signal, and the lead-lag circuits are used to check this characteristic. The lag circuit, whose input is common to the integrator, provides an output after the lag time constant has elapsed that is proportional to the integrator input. This lag is to compensate for the rise time of the integrator. The lead circuit supplies a derived rate signal. These monitoring circuits are of relatively high gain so that by inverting the polarity of the lead circuits and summing the output of the two, a voltage offering a sensitive measure of integrator performances is introduced to the signal chain comparator. This arrangement can detect both a runaway integrator, resulting in a large lead and a small lag output, or in the event that the integrator is required to perform its function in the circuit, a passive failure, in which case the lag output is large and the lead output small.

Pitch Axis

The pitch axis provides basic longitudinal stability augmentation and pitch command control in both manual and automatic modes.

Pitch Attitude Stabilization

As shown in Figure 9-8, the basic pitch attitude stabilization is very similar to the roll axis. Prior to engaging, pitch attitude (ISS 2) drives the pitch synchronizer in order to keep the input to the servo system at null. Included in the elevator servo system is the stabilizer trim signal from the elevator servo amplifier to the auto trim drive unit. When the output of the amplifier exceeds a predetermined level (6.0 volts nominal), the aircraft stabilizer trim system is commanded to drive in a direction to relieve servo effort.

There is an interlock in aircraft wiring with the stall prevention system which interrupts the autopilot nose-up trim command in the event that the angle of attack exceeds the "pre-shaker warning" value, which is two degrees of angle of attack less than the point at which the stick shaker operates. There are internal interlocks with the autopilot automatic trim which are operated as a function of the

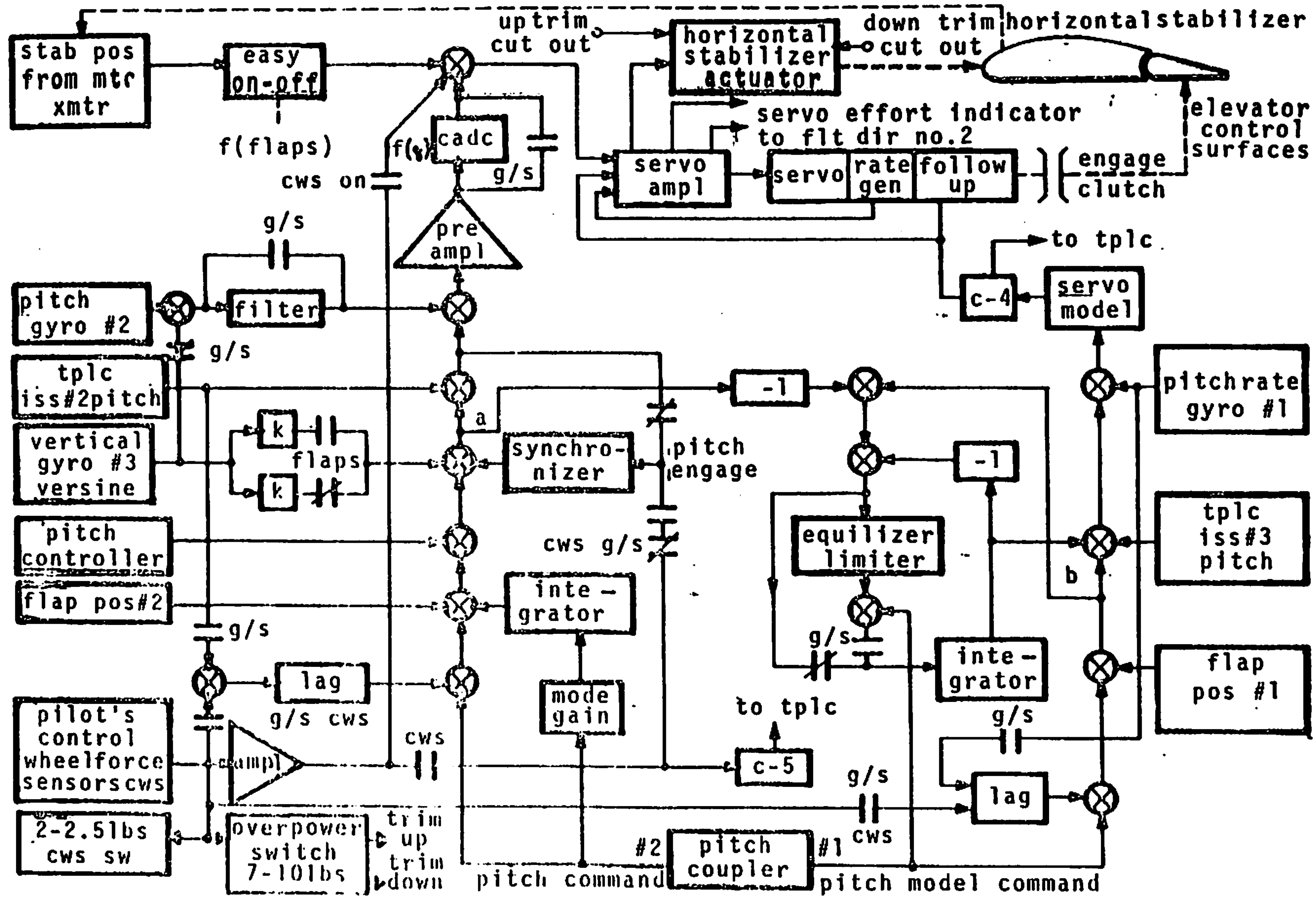


FIGURE 9-8. ELEVATOR BLOCK DIAGRAM

force applied to the control wheel sensor. This action prevents automatic trim in a direction opposite to applied force over seven pounds. Upon engagement of the pitch axis, the pitch synchronizer is locked. It does not follow-up to null as does the roll synchronizer, but rather the aircraft holds the pitch attitude at the time of engagement, utilizing error signals of pitch attitude minus synchronizer output and pitch rate signals from the rate gyro for damping. Roll versine (1-cosine) up attitude signal from the vertical gyro is added to the synchronizer output to command nose-up as a function of bank angle in order to increase lift in turns to prevent altitude loss. The versine signal is gain adjusted as a function of flap position, since the aircraft lift characheristics vary with flap.

Flap Position Signal

Lowering of the flaps increases the lift of the airplane such that it tends to rise or "balloon" until the pitch attitude is adjusted to reduce the required amount. Use of the flap position signal, which commands a change in pitch attitude, eliminates this ballooning.

Stabilizer Position

A signal proportional to stabilizer position is added to the signal chain to provide compensation for automatic trim changes at high speeds. At low speed, this signal is not required, and at flap extension the signal is removed slowly by means of the easy-off circuit.

Cruise Pitch CWS

When a force is applied fore or aft on the pilot's control wheel (CWS switch on), the pitch force sensor develops a signal and, if the force is in excess of 2.5 pounds, pitch CWS is switched in which drives the pitch synchronizer through the CWS amplifier. When the force is released, the synchronizer motor excitation is removed, thereby stopping it at that position. Since the control wheel sensor input to the synchronizer is not opposed by any position inputs in this mode, the synchronizer continues to run at a speed proportional to the applied force as long as the force is applied and the commanded attitude when the force is released is the pitch attitude which the system holds.

Pitch Axis Model

As shown in Figure 9-8, the pitch model channel uses separate sensor inputs as well as a separate integrator which, due to the differences in the two signal chains and the difficulty in achieving identical integration characteristics from two separate integrators, has a rather involved implementation. Inputs to the model integrator are determined by the state of the G/S switch input. Prior to glide slope engage, the input to the integrator is a summation of point A in the active signal chain, and point B in the model signal chain, with the integrator output used as a negative feedback. When glide slope is not engaged, the integrator speed is increased so that it functions as a synchronizer. Point A in the active signal chain consists of all signals except pitch angle and pitch rate; the same is true of point B in the model signal chain. These two signals should be identical. However, if the follow-up signal of the elevator servo is not at null when the elevator is in the detent position, the active chain is required to provide an error signal to drive the elevator servo follow-up to a position corresponding to the detent position of the surface. (Stabilizer position signal may be ignored since only a flap-down condition is under consideration.) Although subtracting signal A from B should result in zero, in practice it may not, and the model integrator then supplies an error signal (identical to that present in the active signal chain) to null the input to the integrator. This action drives the servo model output to a signal corresponding to the active servo position signal. When the glide slope mode actually engages, the input to the model integrator is routed to the equalizer limiter circuit, which limits the signal to the equivalent of 0.25 degrees of pitch. Thus the synchronization process continues after the glide slope mode is engaged, but the synchronizing input voltage to the integrator is limited and integrator speed is reduced as well.

Thus any large signal change at point A results in a comparator alarm, but small differences between points A and B cause the model integrator to make up the difference in the model signal chain. In the event of a "slow over" in the active integrator or any other point in the active signal chain, the aircraft is driven off beam center and the error input from the coupler exceeds the equalizer limiter output, thus it prevents the synchronization process, and results in an eventual disagreement between the active and model signal chains and subsequent comparator alarm.

Altitude Hold

Altitude hold is selected by placing the ALT HLD/PITCH OFF switch on the autopilot control panel to the "ALT HLD" position. In this mode the pitch knob is declutched from the potentiometer so that it can be turned freely without changing pitch attitude. Barometric altitude from No. 2 CADC, as shown in Figure 9-9, is now clutched-in, and deviations from that altitude result in signal output. This output is adjusted as a function of True Airspeed (TAS) and then used to change the pitch attitude to return the aircraft to the reference altitude. After the TAS gain adjustment of the altitude signal, the altitude rate signal provides damping, allowing high gain to be used without inducing oscillation. When ALT HLD (as well as other pitch modes) is engaged, the pitch integrator, as shown in Figure 9-8, comes into operation. It is an electromechanical device, without position feedback, which drives at a rate proportional to the applied input signal until the signal is removed. This signal, being the coupler altitude error, is the input to the integrator. In modes that do not use the integrator, the fixed

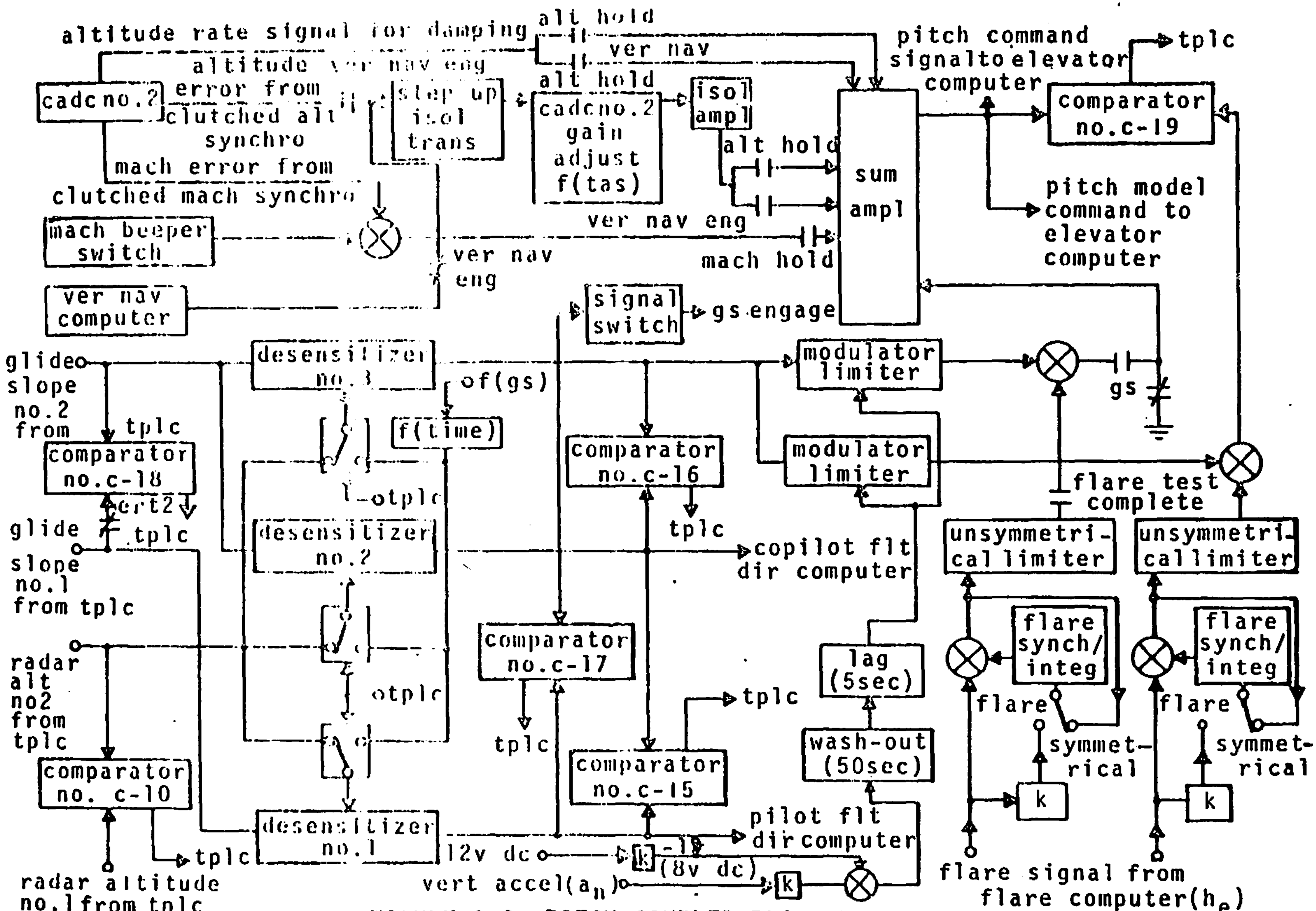


FIGURE 9-9. PITCH COUPLER BLOCK DIAGRAM

phase is removed from the motor thus preventing it from driving. When the pitch axis is disengaged, a centering clutch restores the integrator to its null position.

Pitch CWS is locked out in the ALT HLD mode, while roll CWS may be used. Engaging the MACH HLD EL mode causes ALT HLD to disengage.

Mach Hold

Moving the MACH HLD EL/AWLS switch on the autopilot control panel to the "MACH HLD EL" position engages the mach hold mode of the autopilot. In this mode, the pitch knob is declutched and the clutched mach synchro in No. 2 CADC is clutched in, which supplies a signal proportional to changes in mach number from that at which the mode was engaged. It is not necessary to adjust the mach signal as a function of an air data parameters since its use is intended for cruise only. The pitch integrator is utilized in the same manner as in ALT HLD mode. Control wheel pitch forces in excess of 2.5 pounds causes the mach mode to disengage. Engaging ALT HLD causes MACH HLD EL to disengage.

VER NAV Mode

The VER NAV mode couples the autopilot to fly the VER NAV path computed by the VER NAV system. This mode is "armed" when the G/S/VER NAV switch is placed to "VER NAV," and the computer (located at the navigator's station) is on, but the point at which the maneuver to intercept the selected path is initiated has not been reached. During the armed phase, the autopilot system (pitch) remains in the mode selected prior to VER NAV. If ALT HLD or MACH HLD is engaged, it remains engaged until the VER NAV maneuver is initiated, at which time the conflicting switches return to the off position. The VER NAV signal is switched into the signal chain in the autopilot coupler and commands appropriate pitch attitude changes to acquire and fly the selected angle, as well as flare to and hold the terminal altitude. When the terminal altitude is reached, the autopilot VER NAV ALT HOLD switches are energized, which bypasses the normal autopilot ALT HOLD mode switches to provide altitude error and altitude rate signals to the autopilot. In the elevator computer, the pitch integrator is in operation in the VER NAV mode to provide the large steady-state pitch changes required to accomplish the necessary maneuvers. Since different gains and integrator rates are required in various phases of VER NAV operation, switching functions are provided to change the gain as a function of VER NAV track and VER NAV altitude capture. There are no interlocks between the VER NAV mode and the lateral axis, which allows the pilot to utilize any lateral control, including roll CWS, even though it is intended that the VER NAV flight is accomplished essentially at wings level. Pitch CWS is locked out when the VER NAV mode is engaged. If ALT HOLD or MACH HOLD are selected, while tracking a VER NAV flight path, the VER NAV mode disengages.

Automatic Glide Slope

To arm the autopilot system to fly the glide slope, an ILS frequency must be selected on the receiver, the VOR/ILS button on the navigation selector panel must be pushed, and the G/S/VER NAV switch on the autopilot control panel must be in "G/S" before mode switching can be completed. This action is ordinarily accomplished while flying at a constant altitude under the glide slope beam. At this time, the green G/S ARM light on the autopilot trim indicator is on indicating completion of cockpit switching and a reminder that the aircraft automatically flies down the glide slope when beam center is intercepted. Upon engagement of the beam, the fly-down command is initially softened to smooth the change in pitch attitude and the G/S ARM light goes out. Desensitizing the glide slope signal is accomplished in the same manner as the LOC signals, with changes from a gain of 1 at 1000 feet to 0.25 at 75 feet, to 0.0 at 50 feet. In the elevator computer, the pitch integrator is in operation the same as in previous modes.

Additional damping in the glide slope mode is provided by normal acceleration (AN) signals. A 1-g (normal gravity) output of the accelerometer equals 8-volts. A bias is provided such that the summation of the accelerometer signal and the bias varies about zero volts. This signal goes through a long time-constant washout to eliminate any steady-state error, is lagged in a 5-second lag circuit, and summed with the glide slope signal. Engagement of the glide slope beam is subject to the following requirements:

- An ILS frequency must be selected on the copilot's navigation receiver.
- On the copilot's navigation selector panel, the VOR/ILS button must be depressed and the HDG SELECT/NAV switch must be in the NAV position.
- Glide slope deviation greater than 30 millivolt in either direction must be encountered, followed by deviation less than 20 millivolt fly-up or some deviation fly-down. Pitch CWS may be used in the glide slope mode.

Pitch CWS - Glide Slope

When the CWS switch is turned on, the glide slope mode is engaged, and pitch CWS may be used to alter the commanded pitch attitude. Pitch attitude changes proportional to applied force up to approximately 12 degrees at 10 pounds may be made, but the pitch synchronizer, which is used in the cruise pitch CWS, is not used in the glide slope CWS. An electronic lag circuit with appropriate gain and 3.5-second time-constant is used instead, as shown in Figure 9-8. An attitude change remains only so long as a force is applied to the wheel, and the autopilot commanded attitude is assumed upon removal of the force. Pitch CWS, as used in the glide slope mode, is to assist in smoothing beam irregularities or to reduce the effects of a noisy beam.

Flare Mode

In an AWLS configuration, the autopilot flare mode may be used. One flare computer provides dual flare signal inputs to the coupler. Figure 9-9 shows the altitude rate error signal from the flare computer as the input to the flare synchronizer integrator. Prior to flare engage, the device is operating as a synchronizer and keeps the input to the unsymmetrical limiter at a null. At flare engage, the synchronizer is converted to an integrator, and the input is the flare error signal. This error signal is routed to the elevator signal chain (the pitch integrator is locked during flare). Thus a combination of flare altitude rate and integrated flare altitude rate command inputs to the elevator. These signals are routed through the unsymmetrical limiter, which allows nose up commands only, to be applied to the elevator axis. Normal acceleration signals remain in the system during flare, and glide slope CWS may be used.

Pitch Coupler

In Figure 9-9, the pitch coupler receives glide slope signals in a manner similar to LOC in the roll coupler; however, the pitch coupler contains three desensitizers. Desensitizers No. 2 and 3 both receive No. 2 G/S signal. Desensitizer No. 2 drives No. 2 flight director, and No. 3 desensitizer output is used by the active and model channels. Desensitizer No. 1 receives No. 1 G/S signal, which is used by No. 1 flight director.

The outputs of all three desensitizers are cross-compared by comparators C-15, C-16, and C-17 which provide adequate information to the TPLC to identify a faulty desensitizer. There is a normal acceleration signal input to both the active and model signal chains that is summed with the glide slope signals, and flare outputs are compared at C-19 after which the active signal chain is routed to the elevator computer.

Engage Interlocks

Requirements for initially engaging the autopilot follow:

a. All autopilot circuit breakers must be closed.

b. Autopilot computers must be installed.

c. Disconnect switches on the control wheels must not be depressed.

d. The go-around switch on the control wheel must not be depressed.

e. No. 2 CADC must be installed.

f. Aircraft dc power must be available.

g. Turn knob in detent.

h. NAV/SEL/LAT OFF switch centered.

i. Aileron computer B + supply adequate.

j. Automatic trim not calling for up-trim and down-trim at the same time.

k. TPLC must be installed and ISS attitude output valid.

l. Power available to electric pitch trim system.

m. Elevator computer B + power available.

n. Autopilot engage switch to "AUTOPILOT."

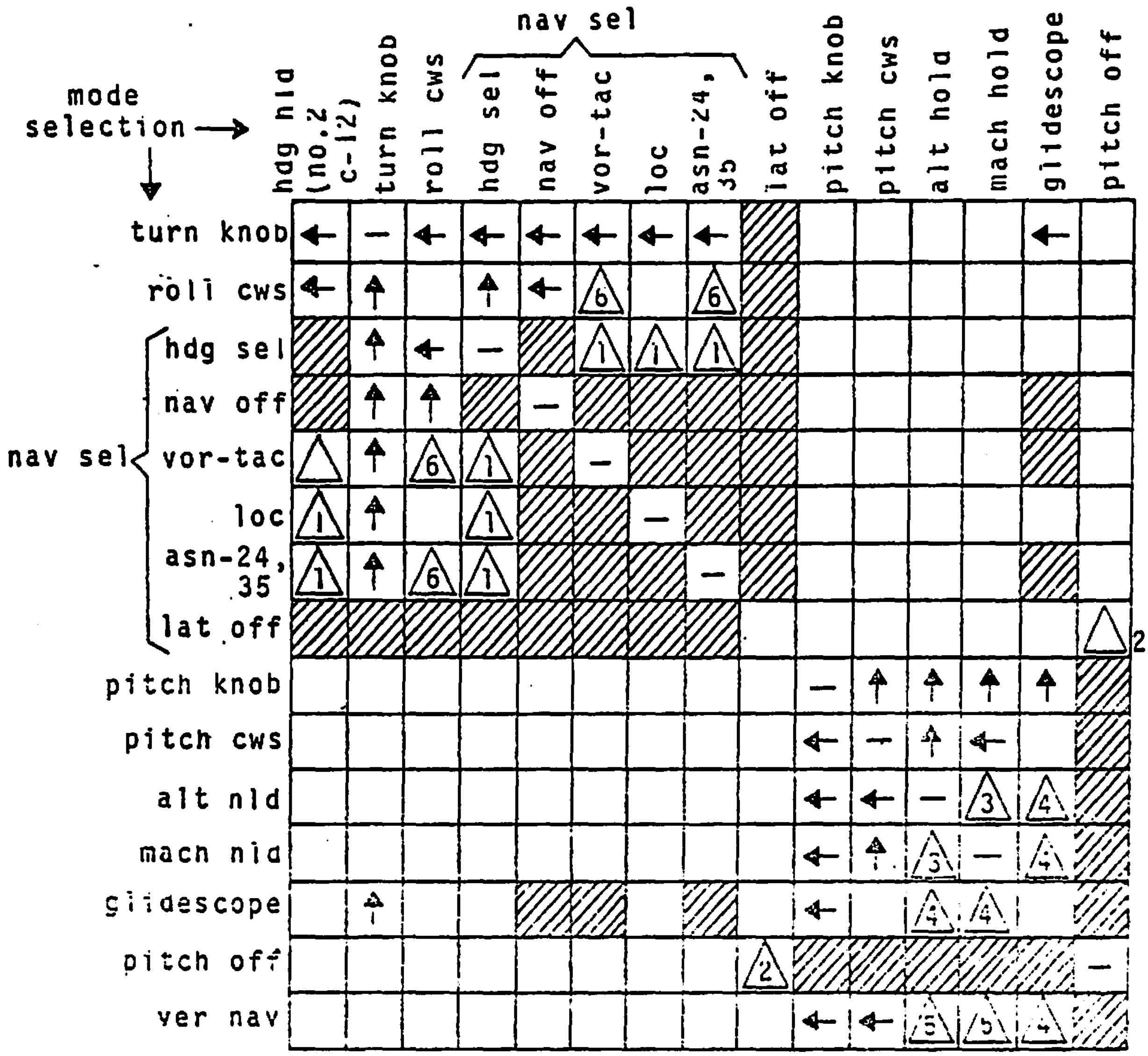
When the NAV SEL/LAT OFF switch is in "LAT OFF," requirements g, h, and i do not apply. The autopilot can also be initially engaged with one or more of these requirements missing if the switch is held to "LAT OFF" while engaging. When the ALT HLD/PITCH OFF switch is in "PITCH OFF," requirements j, k, and l do not apply. Holding the switch in "PITCH OFF" while engaging the autopilot eliminates these requirements on initial engagement.

Mode Switching and Compatibility

Some, but by no means all, mode switching and compatibility has been previously mentioned. Figure 9-10 shows a more comprehensive picture.

Disconnect Logic

The servo clutch wires are routed through the TPLC. Prior to APPR ARM, the clutch power cannot be interrupted in the TPLC and the autopilot operates normally, even though comparators alarm, which they do, because the model circuits are in the LCC configuration and the rest of the system is not. Upon activation of the APPR ARM function in the TPLC, appropriate axes of the autopilot are disengaged as a result of comparator alarms. In the event that both the roll and pitch axes of the autopilot are automatically disengaged, the autopilot engage logic is interrupted, and the AUTOPILOT engage switch falls to the off position.



△1 System will remain on "HDG HLD," or "HDG SEL" if HDG SEL was selected immediately before switching, until a/p capture is reached. △2 If both are chosen, a/p disengages. △3 Last one chosen has priority as other mode disegages. △4 A/C remains in "ALT HLD," "MACH HLD," or "VER NAV" until glideslope beam is reached. △5 "MACH HLD" or "ALT HLD" is compatible with the armed phase of the "VER NAV" mode. When "VER NAV" goes to the tracking phase "MACH HLD or "ALT" HLD" disengages. Selection of "MACH HLD" or "ALT HLD" while in ver nav tracking disengages the "VER NAV" mode. △6 CWS has priority until a/p capture is reached if HDG HOLD was established immediately prior to VOR-TAC or ASN 24, 35 switching.

☐ compatible (both can be selected simultaneously) ▨ not compatible

← mode on left has priority
↑ mode above has priority

FIGURE 9-10. AWLS/AFCS MODE SELECTION CHART

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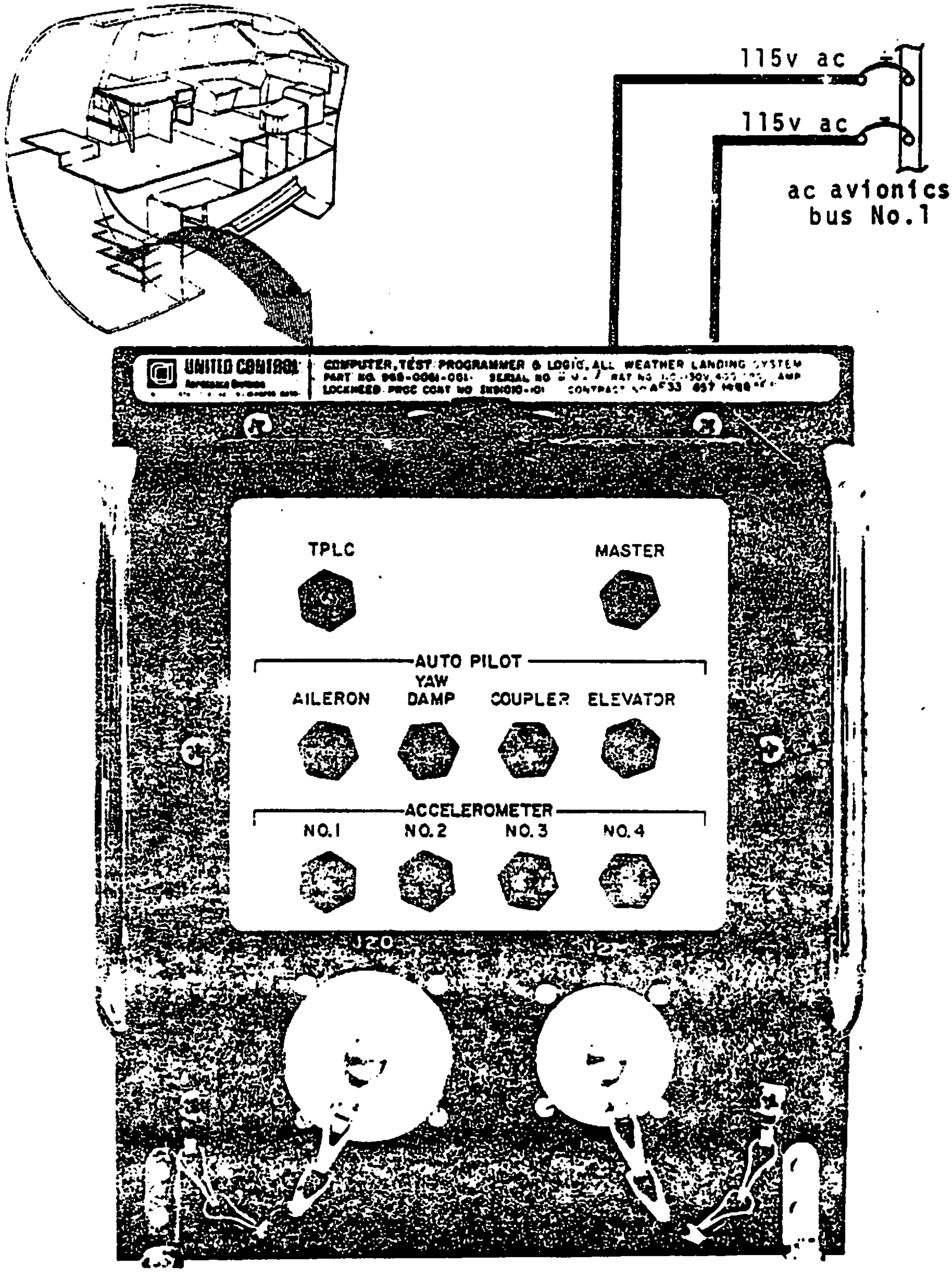


FIGURE 10-1. TEST PROGRAMMER AND LOGIC COMPUTER

TEST PROGRAMMER AND LOGIC COMPUTER (TPLC) AND MASTER CAUTION SYSTEM

TEST PROGRAMMER AND LOGIC COMPUTER (TPLC)

The Test Programmer and Logic Computer (TPLC) integrates the AWLS components into a useable coherent system by providing failure monitoring and ensuring additional reliability of the aircraft attitude signals. In addition, the test programmer guarantees internal and AWLS subsystems functional integrity.

Aircraft Installation

The TPLC is located in the left-hand avionics underdeck rack as shown in Figure 10-1. Power is supplied by avionics bus No. 1 through two 5-ampere circuit breakers. Interface with other AWLS components is provided by four rectangular plugs at the rear of the shock mount. Two plugs on the front of the TPLC furnish external connections for line test equipment.

System Operation

The TPLC is operable upon application of aircraft power and actuation of the AWLS arm switch on the AFCS control panel. Gyro processing and some fault monitoring functions are available if these conditions are met. Full-time fault monitoring becomes active once the approach arm milestone of an AWLS approach is met. During either a manual or automatic approach, the TPLC guards against an undetected failure by identifying the discrepant condition on the fault identification panel. At the same time, each event of the approach appears on the flight progress display panel.

In order to ensure fault-free operation, an enroute test is performed during the preflight checkout and prior to descent to approach altitude. Command for the enroute test originates at the AWLS test panel on the center instrument panel.

Pressing the AWLS test command pushbutton switch sets the enroute test sequence in progress and also illuminates the AWLS, Flight Director 1, and Flight Director 2 TEST in-progress lamps. During the enroute test, the navigation and attitude signal paths are checked by injecting test signals generated by the TPLC and verifying a proper response. At the same time, approach events are simulated and the associated lamp on the flight progress display panel announces the test progression. Upon successful completion of an enroute test, all flight progress display, AWLS, Flight Director 1, and Flight Director 2 TEST in-progress lights go out, and the test RESET light comes on. If the RESET light fails to come on at this time and one or more of the fault lights illuminate, the aircraft is incapable of accomplishing a category II automatic approach.

Specifications

The TPLC contains low-power, highly reliable diode-transistor micro logic integrated circuits. These circuits are moulded into sealed modules which provide normal performance under varying environmental conditions. The NAND gate is most frequently used in the TPLC because of its high power output characteristics. The logic circuitry in Figure 10-2(A) shows two NAND gates and the equivalent "AND" gate.

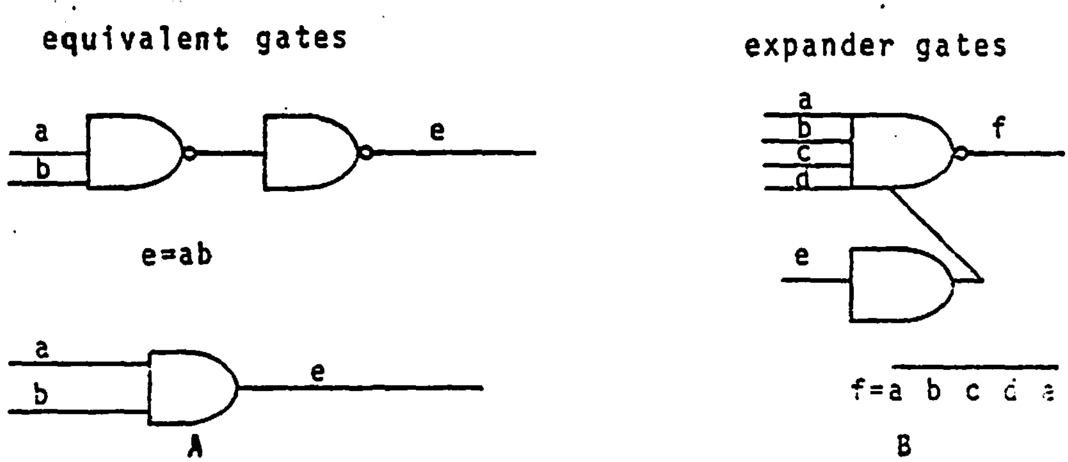


FIGURE 10-2. LOGIC SYMBOLS

Figure 10-2(B) illustrates an expander gate frequently employed to handle multiple logic inputs. A logic "0" represents a voltage level near zero while a logic 1 represents a voltage near positive 5 volts. A driver is often used in the output of a gate to increase the level of logic 1 from zero through 5 volts to zero through 28 volts.

TPLC Abbreviations*

The following symbols and abbreviations are peculiar to the TPLC and are listed for reference.

AA	Approach Arm
CLK	Clock
DGF	Double Gyro Failure
FIP	Fault Identification Panel
FLAG	Fail-Safe Latch and Gate
FPD	Flight Progress Display
ISS	Intermediate Signal Selection
LA	Land Arm
LVFF	Latch Verify Flip Flop
MCS	Master Caution System
PM	Power Monitor
STA	Strobe A
STB	Strobe B

(*A complete list of all abbreviations and symbols is shown at the end of this volume.)

Validity Symbols

The following symbols represent logic inputs from AWLS subsystems along with the TPLC's own monitoring logic.

a. Validity 1 is from comparator C-1 in the yaw damper computer. This comparator senses the two (2) roll-crossfeed inputs from the A/P aileron computer.

b. Validity 2 is from comparator C-2 in the aileron computer. This comparator senses the active and model aileron servo positions.

Continued

c. Validity 3 is from comparator C-3 in the aileron computer. This comparator compares the roll CWS sensor input to the A/P aileron computer against a fixed voltage.

d. Validity 4 is from comparator C-4 in the elevator computer. This comparator senses the active and model elevator servo positions.

e. Validity 13 is from comparator C-13 in the autopilot coupler. This comparator senses the outputs from the active and model A/P localizer desensitizers which are in the A/P coupler.

f. Validity 14 is from comparator C-14 in the A/P coupler. This comparator senses the outputs from the No. 1 localizer radio and No. 2 localizer radio amplifiers which are in the A/P coupler.

g. Validity 15 is from comparator C-15 in the A/P coupler. This comparator senses the outputs from the pilot (No. 1) and copilot (No. 2) glide slope desensitizers which are in the autopilot coupler.

h. Validity 16 is from comparator C-16 in the A/P coupler. This comparator senses the outputs from the copilot (No. 2) and A/P (No. 3) glide slope desensitizers which are in the A/P coupler.

i. Validity 17 is from comparator C-17 in the A/P coupler. This comparator senses the outputs from the pilot (No. 1) and A/P (No. 3) glide slope desensitizers in the A/P coupler.

j. Validity 31 is from the R/GA computer.

k. Validity 18 is from comparator C-18 in the A/P coupler. This comparator senses the outputs from the No. 1 glide slope radio and No. 2 glide slope radio amplifiers, which are in the autopilot coupler.

l. Validity 22 is from the comparator C-22 in the aileron computer. This comparator senses the torque capability of the aileron servo.

m. Validity 19 is from comparator C-19 in the A/P coupler. This comparator senses the active and model pitch coupler output signals.

n. Validity 20 is the flare computation, flare engage and land arm validity which is produced in the flare computer.

Continued

o. Validity 23 is from comparator C-23 in the elevator computer. This detector senses the torque capability of the elevator servo.

p. Validity 5 is from comparator C-5 in the elevator computer. This comparator compares the pitch CWS sensor input to the A/P elevator computer against a fixed voltage.

q. Validity 10 is from comparator C-10 in the A/P coupler. This comparator senses the two altitude inputs from the radar altimeter.

r. Validity 24 is the radar altimeter validity from the radar altimeter's R/T unit.

s. Validity 9 is from comparator C-9 in the A/P coupler. This comparator senses the active and model roll coupler output signals.

SVP (Super Validity Pitch) is a redundant pitch validity for the autopilot pitch channel.

SVR (Super Validity Roll) is a redundant roll validity for the autopilot roll channel.

SVF (Super Validity Flare) is a redundant flare validity for the autopilot flare channel.

RI_{12} validity is from comparator RI_{12} in the TPLC. Comparator RI_{12} senses roll signal inputs from the No. 1 and 2 gyros.

RI_{23} validity is from comparator RI_{23} in the TPLC. Comparator RI_{23} senses roll signal inputs from the No. 2 and 3 gyros.

RI_{13} validity is from comparator RI_{13} in the TPLC. Comparator RI_{13} senses roll signal inputs from the No. 1 and 3 gyros.

PI_{12} validity is from comparator PI_{12} in the TPLC. Comparator PI_{12} senses pitch signal inputs from the No. 1 and 2 gyros.

PI_{23} validity is from comparator PI_{23} in the TPLC. Comparator PI_{23} senses pitch signal inputs from the No. 2 and 3 gyros.

PI_{13} validity is from comparator PI_{13} in the TPLC. Comparator PI_{13} senses pitch signal inputs from the No. 1 and 3 gyros.

Continued

RO_{12} validity is from comparator RO_{12} in the TPLC. Comparator RO_{12} senses ISS roll signal outputs from the No. 1 and 2 ISS circuits.

RO_{23} validity is from comparator RO_{23} in the TPLC. Comparator RO_{23} senses ISS roll signal outputs from the No. 2 and 3 ISS circuits.

RO_{13} validity is from comparator RO_{13} in the TPLC. Comparator RO_{13} senses ISS roll signal outputs from the No. 1 and 3 ISS circuits.

PO_{12} validity is from comparator PO_{12} in the TPLC. Comparator PO_{12} senses ISS pitch signal outputs from the No. 1 and 2 ISS circuits.

PO_{23} validity is from comparator PO_{23} in the TPLC. Comparator PO_{23} senses ISS pitch signal outputs from the No. 2 and 3 ISS circuits.

PO_{13} validity is from comparator PO_{13} in the TPLC. Comparator PO_{13} senses ISS pitch signal outputs from the No. 1 and 3 ISS circuits.

ADI_1R is the validity output from the No. 1 ADI roll attitude monitor in the TPLC.

ADI_1P is the validity output from the No. 1 ADI pitch attitude monitor in the TPLC.

ADI_2R is the validity output from the No. 2 ADI roll attitude monitor in the TPLC.

ADI_2P is the validity output from the No. 2 ADI pitch attitude monitor in the TPLC.

Pwr Mon_2 is the power monitor in the TPLC.

Z is the validity output (power monitor) from the No. 3 gyro (A/P gyro).

Theory of Operation

Basic Functions

The TPLC can be broken down into four basic elements as shown in Figure 10-3:

- The vertical gyro signal processing section
- Monitoring logic section
- Flight progress and AWLS switching section
- Test programming section

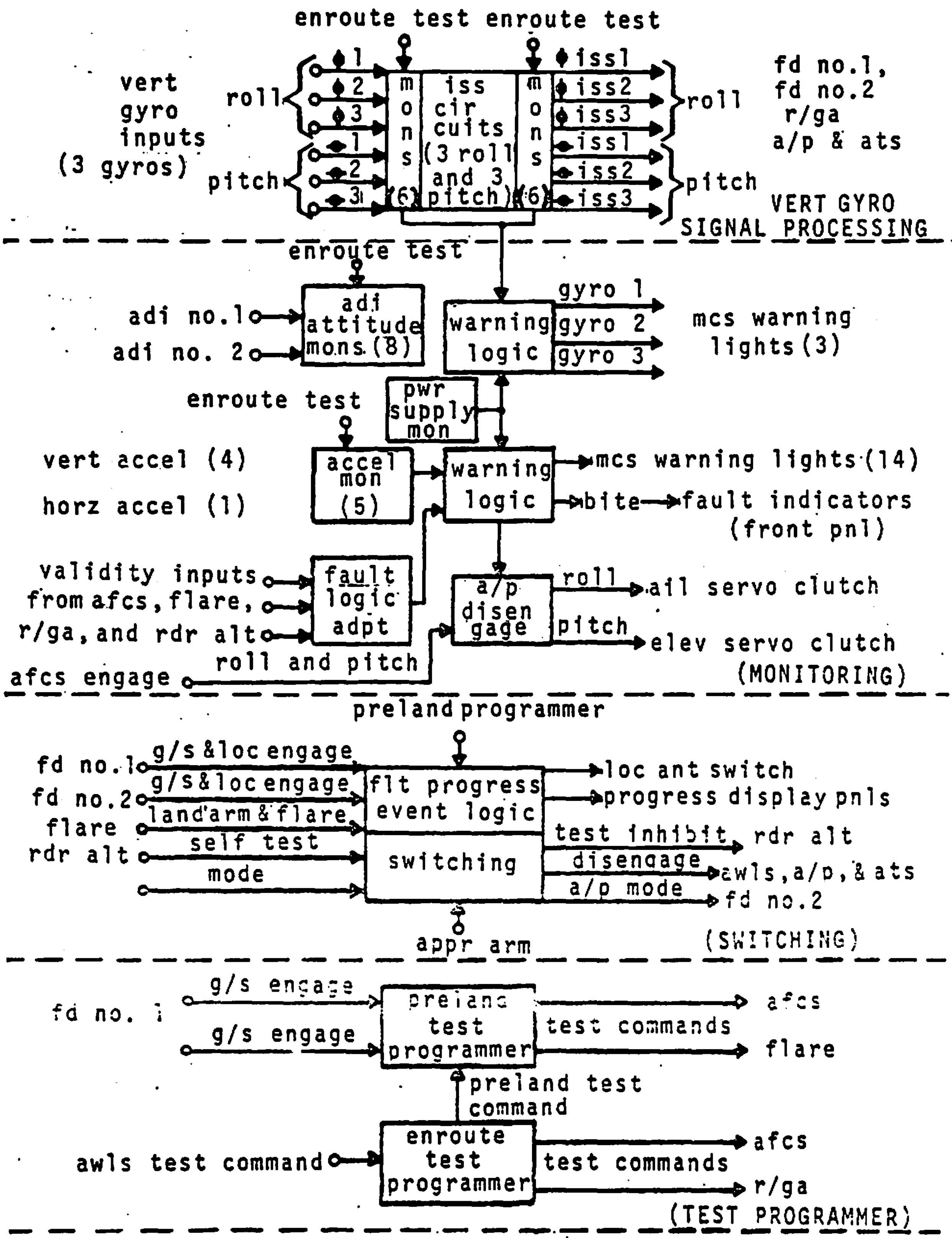


FIGURE 10-3. TPLC BASIC FUNCTIONS

The vertical gyro signal processing section consists of two channels: roll and pitch, which receive signals from three separate vertical gyros. An ISS circuit selects the intermediate or median of the three input signals and provides this intermediate signal to AWLS using components. Monitoring logic prevents an undetected failure from jeopardizing a successful AWLS approach. The ADI attitude and accelerometer analog monitors compare the input voltage level against an established level. Any extreme deviation yields an alarm logic signal output. Each component necessary for either manual or automatic AWLS approach is monitored by the TPLC. The warning logic receives validity information from the AWLS components in the form of binary signals. Voting logic interprets these signals in terms of AWLS operability and presents this information to the master caution system.

A fault prohibiting an automatic approach disengages the appropriate control surface axis. In addition to the warning lights on the fault identification panel, the indicator panel, located on the front of the TPLC, allows maintenance personnel to locate a failed AWLS unit quickly.

The switching section at the TPLC performs the necessary operations to coordinate an AWLS approach. These operations include localizer antenna switching, radar altimeter self-test inhibit, flight progress display switching, R/GA switching, and A/P mode switching.

Aircrew confidence in AWLS components necessitates verification of functional integrity by enroute testing. Also, a preland test is initiated automatically after glide slope beam engagement. The programmers consist of shared binary counters, decode matrix, and program controllers. Programmers generate the test and heal steps used to fault logic chains, disturb navigation and altitude signal paths, and check the TPLC's internal warning logic.

Vertical Gyro Signal Processing - This section of the TPLC provides additional reliability by precluding total loss of Category II capabilities because of a single gyro failure. Loss of one gyro signal does not affect the validity of the three ISS output signals.

Gyro's No. 1 and 2 are the flight director MD-1 displacement gyros, while gyro No. 3 is commonly referred to as the A/P vertical gyro. As shown in Figure 10-4, two of the three gyro input signals are fed into a delta-connected load of which one element is a transformer primary. Gyro No. 3 feeds a two-wire signal directly into the properly loaded transformer primary. Signal isolation and impedance matching are provided by the coupling transformer and unity gain amplifiers. The output of each amplifier is coupled into two of three gyro input comparators. Comparator alarm occurs if signal differential is greater than five degrees. Six comparators are used to detect gyro input deviation since three comparators are required in both the pitch and the roll channel. Comparator alarm outputs are fed into a binary micrologic circuit which identifies the

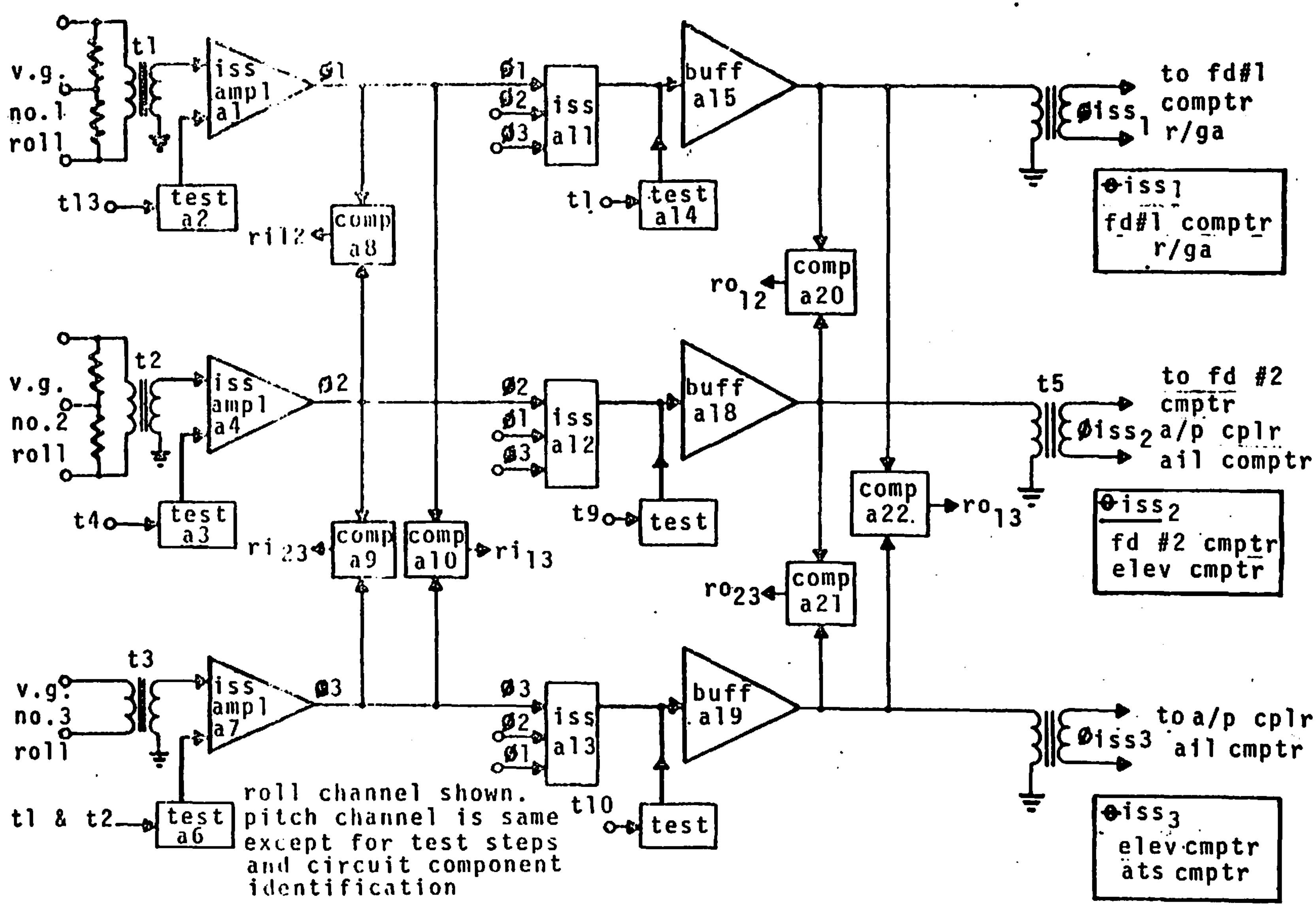


FIGURE 10-4. VERTICAL GYRO SIGNAL PROCESSING BLOCK DIAGRAM

gyro causing comparator alarm. The three gyro signals are coupled to intermediate signal selections where the intermediate amplitude signal is selected. Each intermediate signal is again compared in case the intermediate signal selection has blocked or distorted the displacement voltage, or if the output is overloaded. A deviation of more than 1.5 degrees causes a comparator alarm and a resultant trip of the TPLC voting logic. Components using the 200-millivolt/degree ISS outputs are shown in Figure 10-4.

The self-test modules are normally off and perform no function except during a program test. During the programmed test, each of the input lines undergo an insertion of 400 hertz deviation signals through the self-test modules. A separate set of testing modules disturb the output comparator signals. This disturbance causes the appropriate gyro or ISS validity in the warning logic to fault. If a warning indication fails to appear during test, a memory latch is set indicating that the comparators are unreliable.

o ISS - Figure 10-5 illustrates one of six identical intermediate signal selectors used in the TPLC. The phase relationship of the three, 400-hertz signals remains constant; however, the amplitude may vary according to gyro performance. Each of the three input diode pairs select the minimum voltage while the three output diodes select the maximum signal. In Figure 10-5, ISS No. 2 is selected as the intermediate signal and appears at the output.

Logic Monitoring (Analog) - ADI resolvers are monitored continuously for a null (LO) on one winding and a high (HI) signal on the other winding. The null winding voltage must always be less than 300 millivolts, and the high winding must always be greater than 16 volts. Two circuits are used for each ADI axis as shown in Figure 10-6 to provide the monitoring capability.

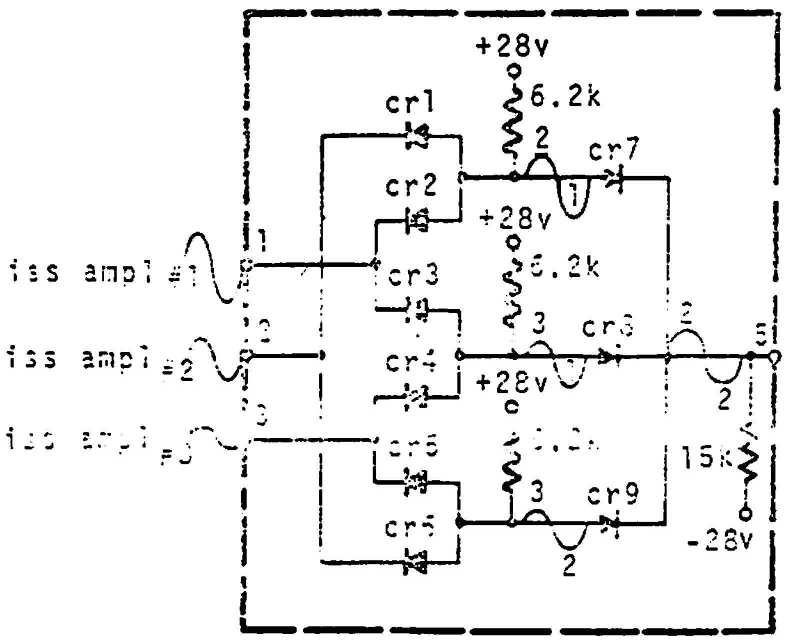


FIGURE 10-5. INTERMEDIATE SIGNAL SELECTOR

A voltage divider in the ADI power HI monitor establishes a dc trip level. Displacement variations

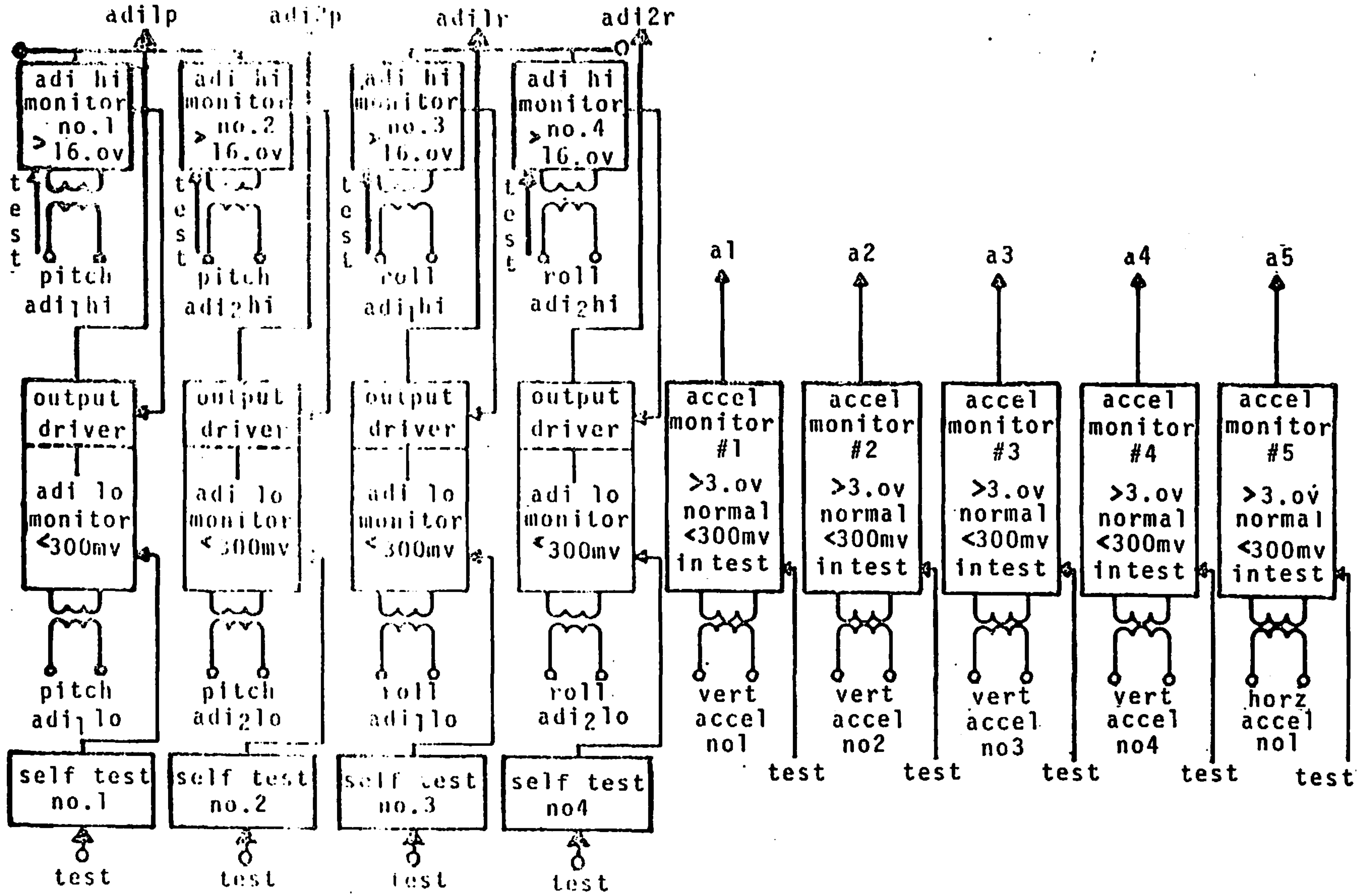


FIGURE 10-6. ADI AND ACCELEROMETER MONITORS BLOCK DIAGRAM

are coupled into the monitor, rectified, and compared with the set level. A voltage less than 16 volts at the input causes a diode to change state. The end result is a binary 1 output when the comparator alarms. Monitor test is accomplished by a logic 1 from program control which trips the comparator.

Accelerometer and ADI LO monitoring is provided by similar ac comparators, the only exception being the trip level. Test signals to the ADI LO monitors are similar to those used in the vertical gyro section, i. e. a preset voltage is inserted into the input line. An alarm appears as a binary 1 output after passing through a driver stage.

Accelerometer monitors are set at 3.0 volts except during test when the accelerometers are torqued to less than 300 millivolts and then allowed to return to greater than 3 volts. If an accelerometer fails to heal, the comparator trips and a logic 1 is sent to the TPLC warning logic. Gain change commands are initiated in the TPLC program control for the 300-millivolts monitoring period.

o Warning Logic - Each component necessary for either manual or automatic AWLS approach is monitored by the TPLC. Warning logic interprets validity inputs and translates them into binary "go," "no-go" information.

The monitor logic breaks down into sublogic in accordance with the indicator or group of indicators that are controlled by the logic, as follows: (Underlined indicators are located on the fault identification panel.)

1. Gyro logic	Gyro 1, 2, 3 ISS Validities 1, 2, 3
2. Manual longitudinal logic	G/S MAN 1 and 2 Pitch manual warning 1, 2
3. Manual lateral logic	LOC Roll manual warning 1, 2
4. Auto logic	A/P ROLL, A/P PITCH
5. Land arm logic	LAND ARM
6. Flare logic	FLARE

Continued

7.	Radar altimeter logic	<u>RDR ALT</u>
8.	AWLS disengage	Aileron, elevator, autopilot disengage
9.	R/GA logic	<u>R/GA</u> R/GA mode signal R/GA mode select
10.	TPLC indicator logic	<u>TPLC</u> A/P command roll, pitch unit fault indicators

Sublogics are interdependent in that they have many input signals in common, and they generate logic functions for each other. Most of these logic functions are not active until after AA, which means that there has to be an AWLS armed, and glide slope beam engage condition present, and that the TPLC has completed a preland test sequence. These conditions are necessary to produce the AA signal, which is generated within the TPLC. Exceptions to the AA condition are the gyro logic, R/GA logic, and AWLS disengage logic which have full-time monitoring capabilities. Both flight directors and the ATS actuate warning indications on the fault identification panel independent of the TPLC.

The following logic equations show the conditions necessary for a fault warning. Lower case alphabetical validity symbols are from AWLS subsystems and are a logic "O" when a fault exists. Comparator alarms within the TPLC, such as PO_{12}, RI_{23}, etc. are logic 1 upon alarm. In the case of the logic equations, a warning logic is true when the term on the right equals one.

o <u>TPLC Logic Equations</u> -

oo <u>Progress Display Lights</u> -

1. LOC = Antenna Switch in nose position = AWLS ARM (FD1 LOC beam engage + FD2 LOC beam engage)

2. G/S: (Pilots) = AWLS ARM · FD1 G/S beam engage
   (Copilots) = AWLS ARM · FD2 G/S beam engage

3. APPR ARM (AA) = preland test complete

4. LAND ARM (LA) = AA · LAND ARM trip (100 ft) · u

5. FLARE (FE) = AA · flare engage

oo <u>Fault Lights</u> -

1. $TPLC = (PI_{12} + PI_{23} + RI_{12} + RI_{13} + RI_{23} + PO_{12} + PO_{13} + PO_{23} + RO_{12} + RO_{13} + RO_{23} + DG_{F2} + DG_{F3})$

2. $LOC = AA \cdot (\bar{h} + \text{latch})$

3. $G/S\ MAN\ 1 = AA \cdot [(\bar{i} \cdot \bar{k}) + (\bar{m} \cdot \overline{FE}) + \text{latch}]$

4. $G/S\ MAN\ 2 = AA \cdot [(\bar{i} \cdot \bar{j}) + (\bar{m} \cdot \overline{FE}) + \text{latch}]$

5. $A/P\ ROLL = AA \cdot (\bar{b} + \bar{v} + \bar{h} + \bar{g} + \bar{c} + \bar{n} + \bar{a} + \text{latch})$

6. $A/P\ PITCH = AA \cdot [\bar{d} + \bar{o} + (\bar{j} \cdot \bar{k}) + (\bar{m} \cdot \overline{FE}) + \bar{s} + \bar{g} + \overline{LA} + (\bar{u} \cdot \bar{t} \cdot \bar{p} \cdot \overline{FSV}) + \text{latch}]$

7. $GYRO\ 1 = (\overline{RI}_{12} \cdot \overline{RI}_{13}) + (\overline{PI}_{12} \cdot \overline{PI}_{13}) + (\overline{ADI_1 R} + \overline{ADI_1 P}) + \text{latch}$

8. $\overline{FLARE} = AA \cdot (\bar{p} + \bar{u} + \bar{t} + \text{latch})$ Flare Self Test Complete

9. $LAND\ ARM = AA \cdot (\bar{p} + \bar{u} + \bar{t}\ \text{latch})$

10. $RDR\ ALT = AA \cdot (\bar{u} + \text{latch})$

11. $THROT = ATS\ ARM \cdot \overline{ATS\ engage}$

12. $R/GA = (R/GA\ select \cdot \bar{e}) + \text{latch}$

13. $FLT\ DIR\ 1 = \overline{FD1\ Valid}$

14. $FLT\ DIR\ 2 = \overline{FD2\ Valid}$

15. Spare

16. $GYRO\ 2 = (\overline{RI}_{12} \cdot \overline{RI}_{23}) + (\overline{PI}_{12} \cdot \overline{PI}_{23}) + (\overline{ADI_2 R} + \overline{ADI_2 P}) + \text{latch}$

17. $GYRO\ 3 = (\overline{RI}_{13} \cdot \overline{RI}_{23}) + (\overline{PI}_{13} \cdot \overline{PI}_{23}) + \bar{Z} + \text{latch}$

A simplified diagram of one type of warning logic used in the TPLC is illustrated in Figure 10-7.

It should be noted that all validity inputs from the autopilot and the flare computer are associated with an automatic AWLS approach. If validity 4, 5, 19, or 23 go to a

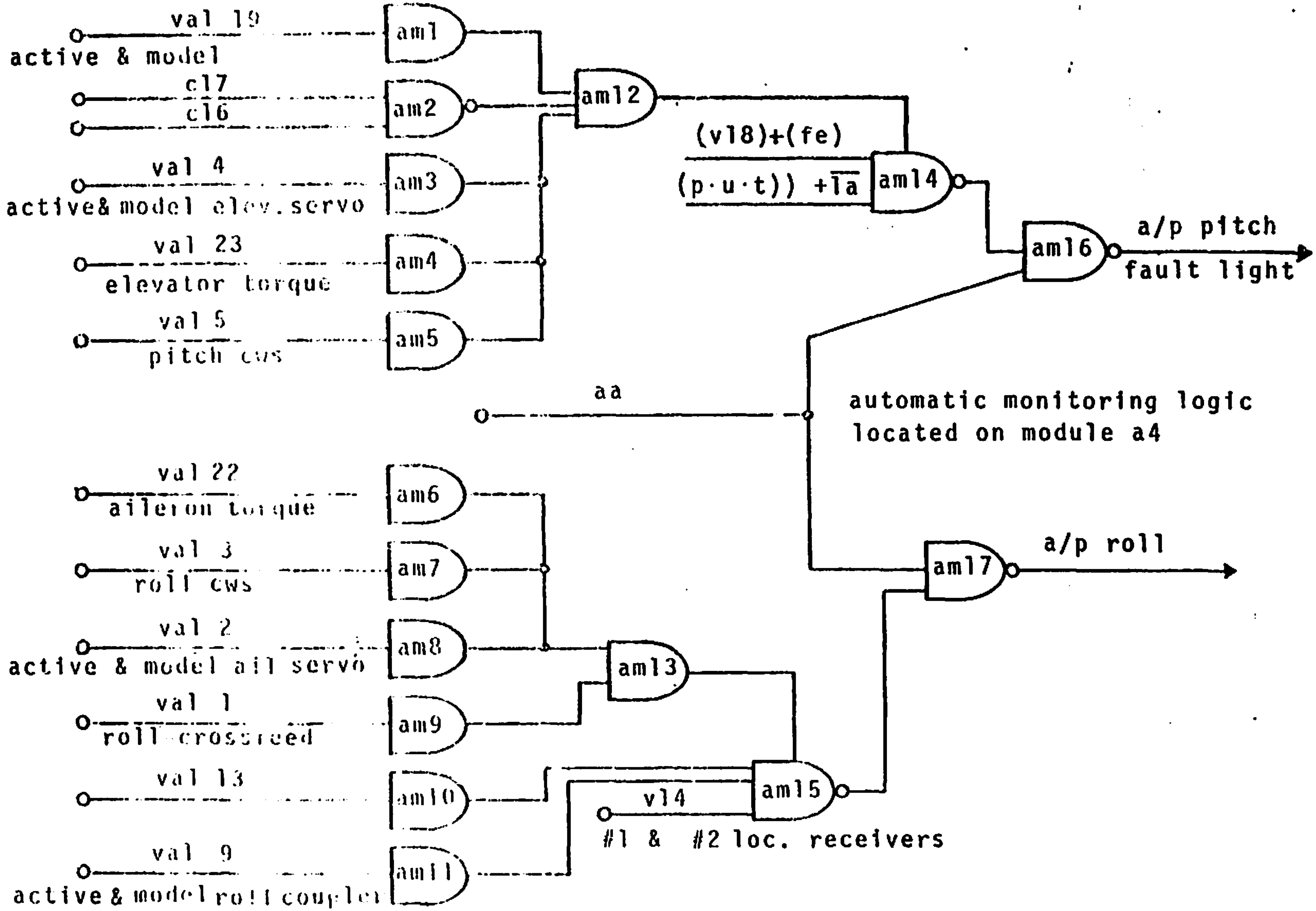


FIGURE 10-7. AUTOMATIC MONITOR

logic "O," the expander gate formed by AM 12 and AM 14 produce a logic "O" output to the fault warning system, providing that the AA signal is present. Should the autopilot desensitizer channel (No. 3) fail within the A/P coupler, both C-17 and C-16 would go invalid ultimately causing an A/P PITCH warning. Validity 18 has no effect on the output of AM 14 if the aircraft has reached the flare engage (FLARE) step of the approach simply because glide slope signals are no longer used at this point. Likewise, validities P, U, and T are blocked out until LA.

Validities 1, 2, 3, 9, 13, 14, and 22 may cause A/P ROLL warning. Here again the warning logic is inhibited until AA.

Power Monitor - The TPLC power supply system contains two redundant transformer-rectifier units with multiple regulators and protector circuits, two redundant 115-volt rf filters, and two power voltage monitor circuits. Full TPLC functional capability is maintained with either of the power supplies operational or with either one of the two ac-input power lines energized.

Each of the dc voltage sources in both supplies is monitored for validity through two non-redundant monitor circuits. One monitor circuit checks all power unit voltages with the exception of those voltages used in the vertical gyro section. The other circuit monitors the vertical gyro voltages. A failure within either power unit produces an alarm in the common monitor which causes a TPLC indicator and fault light indication. A total loss of vertical gyro voltage must occur before the gyro monitor circuit is tripped. Therefore, since the gyro section is fully operable on one supply, a gyro alarm is not produced unless the gyro power voltage is inoperable.

Both monitor circuits consist of integrated operational amplifiers which drive Schmidt trigger circuits. A deviation of ± 30 percent from any single source causes an alarm.

o Fault Indications - Fault indicator buttons, located on the front panel of the TPLC as shown in Figure 10-1, pop out when a fault exists in the following components.

- o TPLC
- o Autopilot coupler
- o Vertical Accelerometer No. 1
- o Vertical Accelerometer No. 2
- o Vertical Accelerometer No. 3
- o Vertical Accelerometer No. 4

- Horizontal Accelerometer No. 1
- Aileron Computer
- Elevator Computer
- Yaw Damper Computer
- Master (Operated any time one of the above indicators trip.)

These indicators simply aid maintenance personnel in locating a faulty component; no circuit is opened or closed because of a popped indicator. An accelerometer alarm (logic 1) from the monitoring circuits trips their respective accelerometer indicator buttons during a programmed test only. Any gyro failure or any ISS comparator alarm pops the TPLC button. Validity 22 (aileron torque monitor) and validity 23 (elevator torque monitor) control the aileron and elevator indicators respectively. Roll crossfeed validity 1 sets the yaw damper button. All nine validity inputs from the A/P coupler are instrumental in controlling the A/P coupler indicator.

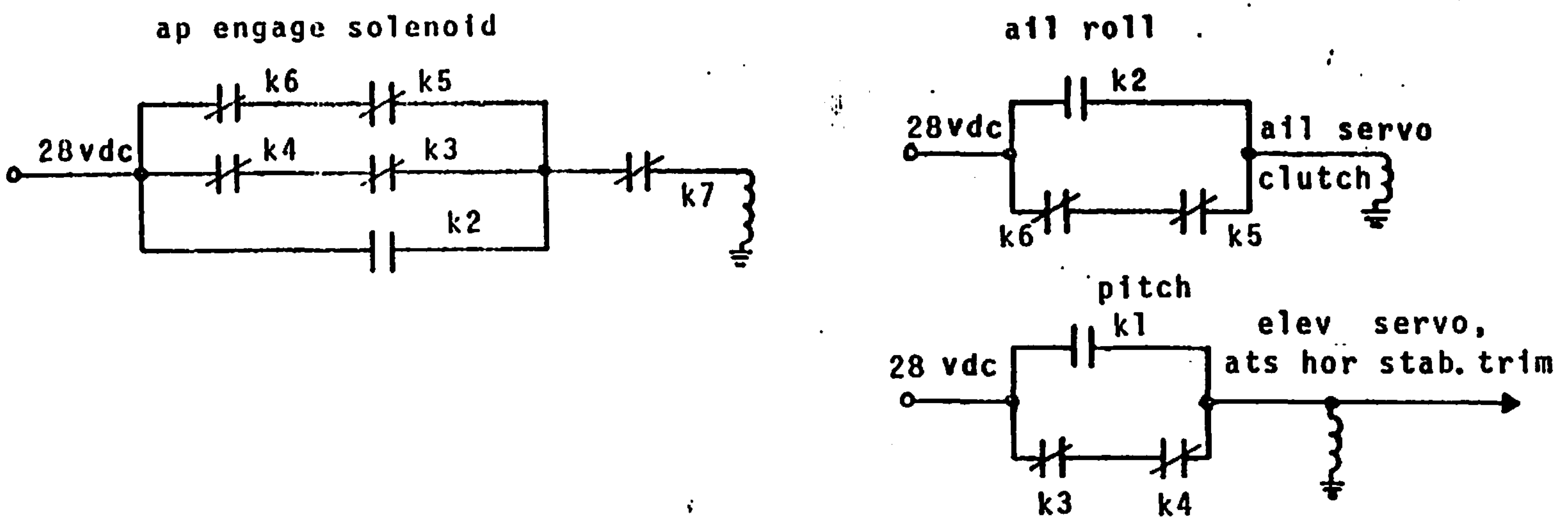
All indicators are operable during the enroute test; therefore, if one of the listed units should fail to test properly, the indicator for that unit is tripped. During an AWLS approach, all indicators, with the exception of the accelerometer monitoring, are operable.

- AWLS Disengage - To ensure against an A/P commanded hard-over control signal during a critical portion of an approach, disengage relays are provided to remove A/P control should a fault occur. Redundant disengage relays have been used to assure that an axis will be disengaged should a failure occur in that channel.

Figure 10-8 illustrates a simplified relay logic diagram. As an example, an invalid condition, which causes the auto roll logic to alarm, is assumed. Relay K6 would energize opening the loop to the aileron servo clutch. In case the relay contacts failed to open, the Super Validity Roll (SVR) signal from the A/P would cause K5 to energize. The end result would be aileron servo clutch disengagement. Pitch disengagement is similar except that in addition to auto pitch and SVP, a Super Validity Flare (SVF) failure from the A/P flare channel causes interruption of elevator servo clutch, ATS, and horizontal stabilizer trim interlock.

When both roll and pitch channels become unreliable, the holding voltage to the A/P engage solenoid is removed, which causes complete A/P disengagement. In addition, the solenoid drops out when R/GA is selected or when DGF_3 occurs.

The disengage relays are tested for proper operation during the preland test. Since an AWLS disengagement is undersirable during that portion of the approach, relays K1 and K2 are energized to provide an alternate path for current. If the



k1=aa2
k2=k1 aa1
k3=auto pitch
k4=svp+svf
k5=svr
k6=auto roll
k7=dgf3+r/ga select

note: aa1 and aa2 provided during preland test only.

FIGURE 10-8. AWLS DISENGAGE RELAYS

disengage relays fail to respond properly to test commands, the AA signal is inhibited to the flight progress display panel and a Category II approach is not possible.

AWLS Switching - The TPLC provides the external switching necessary for an AWLS approach and the progress information to the pilot and copilot.

o Localizer Antenna Switch - Upon reception of the localizer beam engagement signal from either flight director, the TPLC switches the localizer receivers from the tail to the nose antenna. Reception at the nose antenna provides more accurate lateral guidance. When the switching is completed, the LOC lights on the progress display panel are illuminated.

o Radar Altimeter Self-Test Inhibit - When the localizer beam engage signal is received, the self-test feature of the radar altimeter is switched out of the circuit by the TPLC.

o Flight Progress Display Switching - In addition to the LOC light, the TPLC controls the illumination of G/S, APPR ARM, LAND ARM, and FLARE lights on the flight progress display panel. G/S annunciation occurs when the glide slope beam engage signal is received from the flight directors. APPR ARM is announced after a successful preland test. LAND ARM is a function of radar altitude and occurs at 100 feet. Illumination of the FLARE light occurs when the flare computer begins providing guidance commands to the A/P and the ADIs.

o R/GA Switching - Upon reception of a ground signal from either the pilot's or the copilot's R/GA engage switch, the TPLC disengages the A/P and switches the flight directors into the R/GA mode.

o A/P Mode Switching - At AA, the copilot's flight director is switched by the TPLC into the A/P mode. In this mode, the copilot's ADI displays the A/P servo error. If an A/P axis is disengaged, the associated steering bar (bank or pitch depending on which axis is disconnected) reverts to flight director control.

Test Programming - In order to verify the functional integrity of the AWLS components, both an enroute and a preland test are programmed by the TPLC. Initiation of a preland test occurs automatically at glide slope beam engagement, while an enroute test is initiated manually prior to descent to approach altitude. In fact, the enroute test encompasses the preland test since the first step of the enroute test is the preland test.

The normal programmer, used for both the preland test and the enroute test, consists of 5 flip-flops (divide by two networks) and a decode matrix as shown in Figure 10-9.

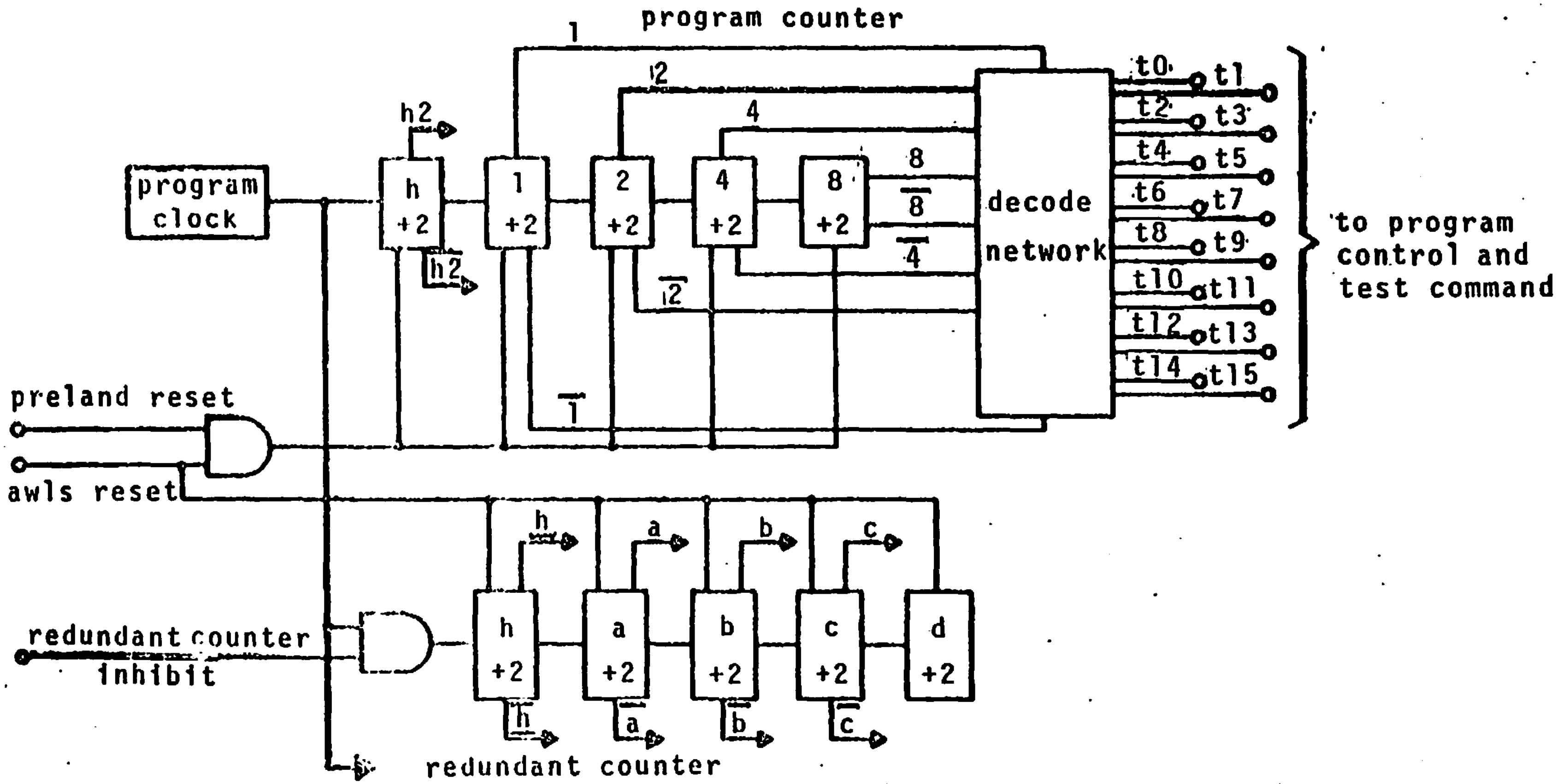


FIGURE 10-9. TEST PROGRAMMER

A redundant counter is used to check the program to assure that the proper sequence has occured.

During the preland test, the program clock generates a pulse each second. The binary counter and the decode network combine to produce 15 test-heal steps, each step having a duration of 2 seconds. Therefore, the preland test requires 30 seconds to complete. Twelve clock pluses to the redundant counter are inhibited so that the binary output from this counter at the end of the preland test should be 10010 or 18.

If the counter is not at 10010, the preland test is ruled unsuccessful and a fault exists in the system. When the preland test is being run as part of the enroute test, the normal program counter is reset, and the program clock again produces counting pulses. The clock, however, now produces a pulse every 1.5 seconds except for three test steps where 12 seconds are required to complete a test-heal sequence. The test steps are fed to program distribution where test commands are generated.

o <u>Preland Test</u> - The preland programmer performs a test of the internal TPLC warning logic, and lateral and longitudinal comparators. It also initiates an integrity test of the flare computer. Prior to exercising any external validity, the TPLC performs a preland programmer verification test to assure that no fault exists in the programmer itself. This takes about 12 seconds to complete. After the programmer validity has been verified, a step-by-step test of each validity is initiated. If a validity fails to test or heal properly during the external test, a failure warning latch is set. At APPR ARM the applicable warning light illuminates.

Following is a brief description of the preland test.

T0 - T5	Internal TPLC test.
T6	Preland test command 3 trips comparators 9 and 10 in the A/P coupler; flare test command [illegible] the flare computer self-test cycle.
T7	Flare and super validity flare tested by applying an internal land arm signal, since the flare computer is in test a flare failure warning should occur.

Continued

T8	Preland test command 2 trips desensitizers comparators C-13, C-15, C-16, and C-17 in the A/P coupler; validity 18 simulated invalid; flare engage internal signal is simulated.
T9	Preland test command 1 trips radio input comparators C-14 and C-18 in the A/P coupler; simulated flare engage signal is removed.
T10	Preland test command 4 tests A/P CWS comparators C-3 and C-5.
T11	Preland test command 5 trips A/P servo follow-up comparators C-2 and C-4
T12	No test performed.
T13	The two high torque validity inhibits removed; flare test command removed.
T14	Preland test command 6 tests the A/P coupler radar input comparator C-10; roll crossfeed comparator C-1 tested; accelerometers are tested.
T15	Test complete.

o Enroute Test - The tests performed in the enroute test follow:

T0	Preland test.
T1	Radar altimeter trips to the copilot's and A/P's glide slope desensitizers are removed; radar altimeter altitude inputs to the A/P coupler are simulated at 40 feet by the TPLC's analog simulation circuits; navigation signals generated by the TPLC cause

Continued

	comparators C-13, C-15, and C-17 in the A/P coupler to trip; No. 1 roll and No. 2 pitch ISS outputs and No. 3 roll and pitch ISS inputs are tripped.
T2	Radar altimeter trips to A/P glide slope and localizer desensitizers causing C-16, C-17, and C-13 to alarm; test signals injected into No. 3 roll and No. 1 pitch ISS inputs, TPLC power monitors tripped.
T3	Copilot, pilot, and A/P G/S desensitizers trip; signals injected into all three pitch ISS inputs and all three pitch ISS outputs.
T4	Error signals injected into aileron and elevator servo models; signals injected into No. 2 roll and No. 3 pitch ISS inputs; TPLC power monitors are tripped.
T5	Gain of deviation signals to the pitch and roll models in A/P coupler is lowered causing pitch and roll output comparators C-9 and C-19 to trip; ADI roll HI monitor trips.
T6	CWS comparators tripped; ADI pitch HI monitors are tripped.
T7	Roll crossfeed comparator is tripped by decreasing level of crossfeed signal from aileron computer to the Y/D computer. Test signal injected into ADI No. 1 and No. 2 roll LO monitors.
T8	Pilots (No. 1) simulated navigation deviation signals are removed failing V14 and 18; ADI No. 1 and 2 pitch LO monitors trip.
T9	Auxiliary radar altimeter input returned to control of radar altimeter, A/P radar altitude input is simulated at 1000 feet; signals injected into ISS No. 2 roll, and No. 3 pitch outputs.

Continued

T10	Test command sent to R/GA tripping AOA comparator; signals injected into ISS No. 1 pitch and No. 3 roll outputs.
T11	FE1 and FE2 signals simulated by TPLC, integraters in flare computers allowed to function 12 seconds; test commands to No. 1 ADI pitch and roll LO monitors; gyro 3 power monitor is simulated invalid.
T12	No. 1 FE signal is removed, No. 1 flare channel switches to synchronizer mode, with the simulated flare signal present, the No. 1 flare integrator drives to zero tripping the pitch output comparator failing validity V-19; signals sent to ADI No. 2 pitch and roll LO monitors.
T13	Radar altimeter placed in self-test mode, V-24 fails and FSV fails; signals sent to ISS tripping RI 12, RI 13, PI 12, PI 23.
T14	All FIP lights which have been latched are illuminated and the test in progress light extinguishes. If the enroute test was successful no fault lights illuminate, and the AWLS reset comes on.

MASTER CAUTION SYSTEM (MCS)

The MCS controls the illumination of the 17 lamps on the fault identification panel and the master caution lights on the two flight progress display panels. Both flight directors, the TPLC, and the ATS supply fault logic to the MCS. If an AWLS fault occurs, the MCS causes the appropriate lamp on the fault identification panel to flash and either the auto or manual caution lamp on the flight progress display to illuminate.

Either the pilot or the copilot may acknowledge the fault by actuating the reset push button switch on the flight progress display. After reset, the fault light is on steady, and the manual or auto master caution light goes out.

The major system component is the master caution controller located in the left hand under deck rack. Contained within this unit are 11 manual fault light drivers, 6 auto fault light drivers, manual and auto flashers, plus auto and manual fail-safe circuits. Twenty-eight volt dc power is supplied to the controller from the main dc avionics bus No. 1 through two 5-ampere circuit breakers on the avionic circuit breaker panel.

System Operation

Seventeen light drivers located in the master caution controller cause illumination of an associated fault light, as shown in Figure 10-10. Eleven channels are supplied manual logic while six channels receive logic from subsystems whose functional integrity is required to perform an automatic AWLS approach. A logic "O" on any input causes an associated fault light and a master caution lamp to illuminate.

The master caution controller contains a manual flasher and an automatic flasher as shown in Figure 10-10. A fault logic "O" on any input causes a relaxation oscillator to begin operation. The oscillator output biases the light driver on and off at a rate of 75 to 135 hertz.

Once the fault is acknowledged by either the pilot and copilot, depression of the master caution reset switch applies 28-volt dc to a Silicon Controlled Rectifier (SCR) in the light driver circuit. The SRC fires to apply a steady-on-bias to the light driver. At the same time, it interrupts the current flow through the master caution lamp. The fault light stays on as long as the logic input indicates an undesirable condition.

Two fail-safe circuits are employed to ensure a fault light illumination should the flasher fail to operate. One second after a fault is detected, a fail safe circuit assumes bias control over the light driver. If the flasher begins to operate normally at a later time, the fail safe circuits are deactivated, and driver bias control depends on the flasher.

The three-position AWLS annunciator test switch applies a ground to the light drivers in "TEST" (simulating a fault logic "O") and 28-volt dc to the fail safe circuits in "FAIL SAFE TEST" (simulating a flasher fault).

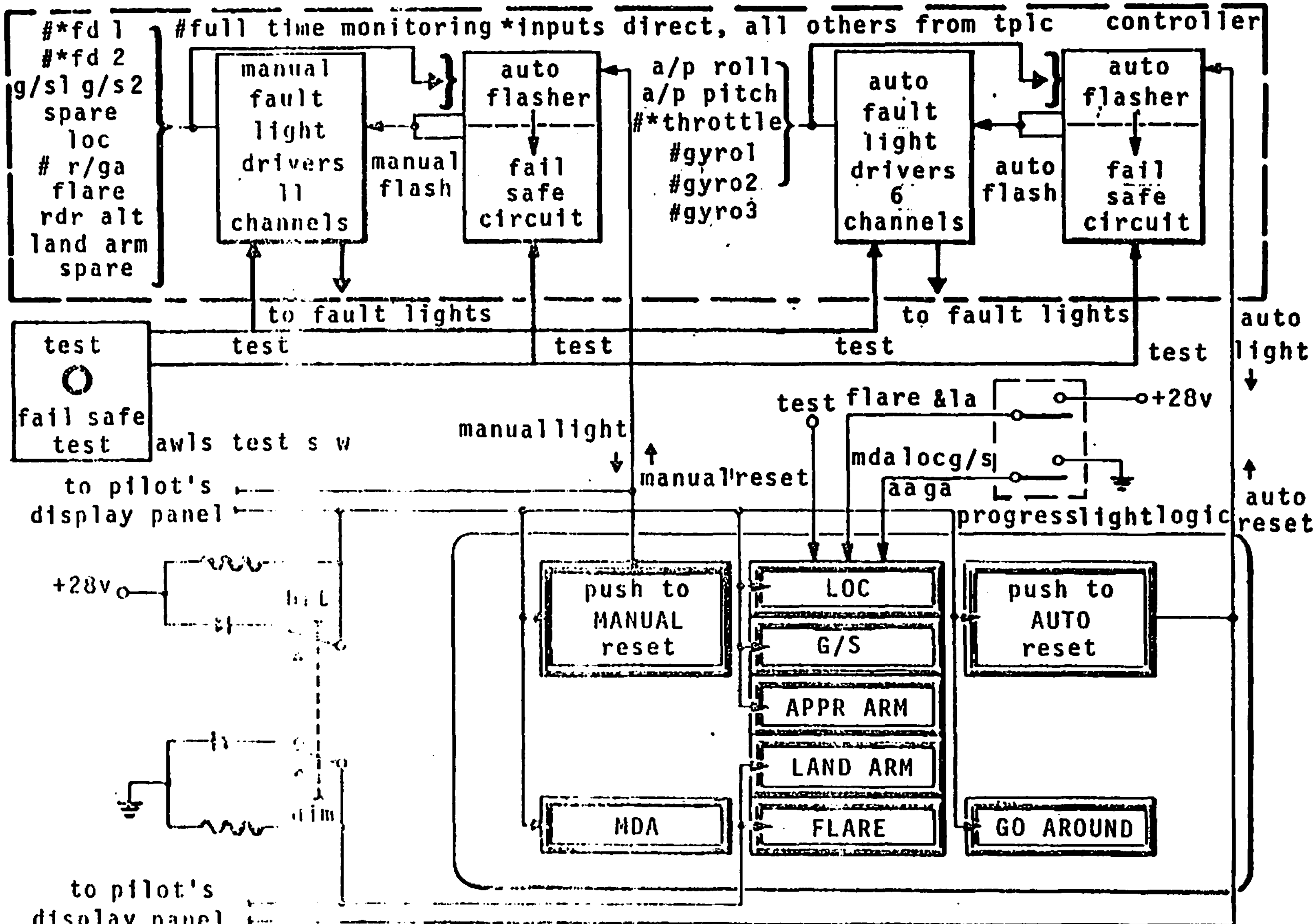


FIGURE 10-10. MASTER CAUTION SYSTEM BLOCK DIAGRAM

HORIZONTAL AND VERTICAL ACCELEROMETERS

An accelerometer is a self-contained unit and consists of a floating vane, an acceleration sensing mechanism, and solid state circuits. Electronic circuits perform the generating, sensing, and modifying functions necessary to produce ac and dc outputs that are directly proportional to acceleration applied in the sensitive axis. The sensing mechanism and electronic circuits form a closed-loop system. Output signals produced by the mechanism in response to acceleration are amplified and fed back to a torque coil in the mechanism to return the vane to the neutral position. Torque current required to null the loop is directly proportional to the acceleration force and is, therefore, a measure of that force. This current is used to produce the output voltages. Built-In Test Equipment (BITE) protection is incorporated in the unit. There are four vertical and two horizontal units installed in the aircraft.

Accelerometers are located in the center wing above the cargo compartment as shown in Figure 11-1.

SYSTEM OPERATION

The accelerometers are automatically operated by using system power and aircraft acceleration. Their outputs are monitored for proper operation. Self-test by external command checks the accuracy of the accelerometer operation and also checks the fault logic circuit.

SPECIFICATIONS

Modules	Servoamplifier, active filter 400-hertz modulator, power supply Fault logic sensing mechanism
Voltage Input	26-volt ac, 400-hertz

Continued

SPECIFICATIONS (Continued)

Outputs	Acceleration ac Acceleration dc DC dummy load
Test	By test command, accelerometer torqued by a fixed voltage Current test provisions

THEORY OF OPERATION

The sensing mechanism provides an output that is proportional to the acceleration acting on the sensitive axis.

The servoamplifier consists of a 4-megahertz oscillator, a current doubling detector, an amplifier, and a feedback network as shown in Figure 11-2. The vane reduces the oscillator's Q, thus reducing the rf output. Oscillator output is rectified, doubled, filtered, and then amplified in three stages. This voltage is applied to the torque coil in the proper phase to null the vane change. Torque voltage is sent to the dc output terminal and the modulator board. Negative feedback through a thermistor compensates for coil damping and temperature changes.

The pickoff coil is mounted near one end of the vane and is positioned so that movement of the vane varies the distance from the vane to the coil. A torque coil is mounted in a recess in the other end of the vane and is in the field of a permanent magnet as shown in Figure 11-3. The coil is oriented so an electric current through it tends to move the coil and thus the vane. Vane movement depends on the direction of current flow, which, in turn, causes a torque to move the vane. The amount of torque depends on the amount of current in the coil.

Vertical and horizontal accelerometer units are similar except that the horizontal unit requires a correction to add a fixed 1-g output. A fixed bias from a voltage divider is used. The vertical accelerometer derives 1 g from the earths gravity.

An active filter is connected across the dc output to determine the frequency response and to limit transient and vibration responses.

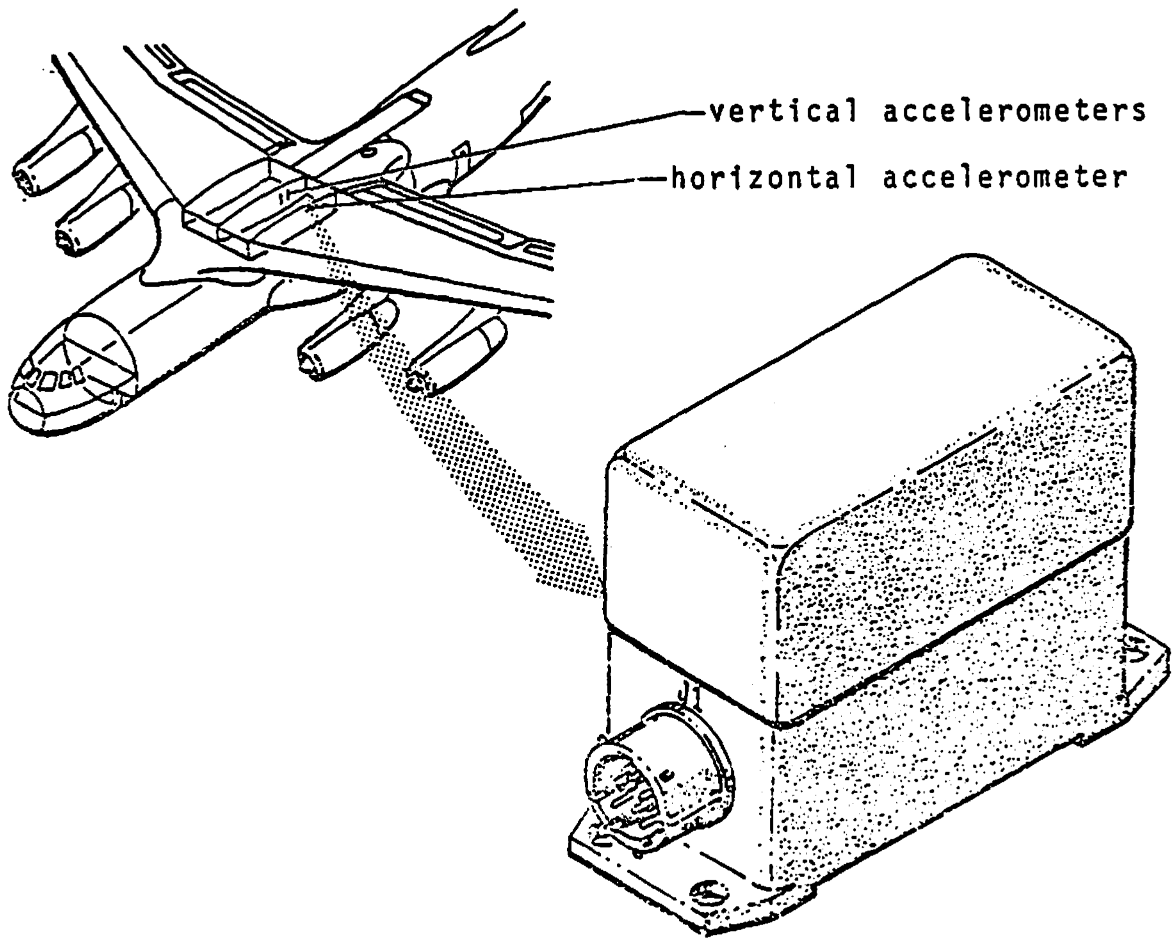


FIGURE 11-1. ACCELEROMETER LOCATION AND INSTALLATION

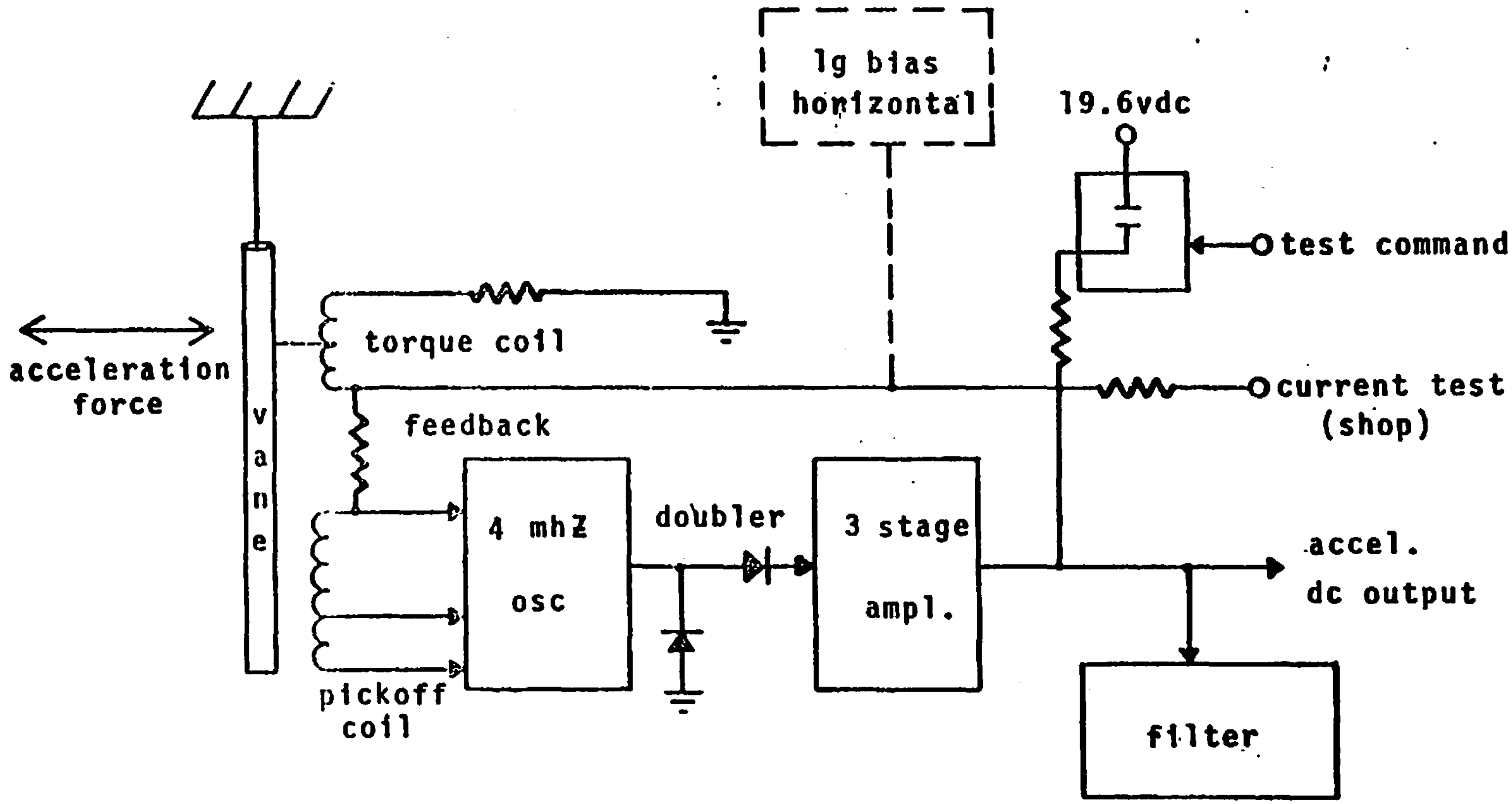


FIGURE 11-2. ACCELEROMETER DC OUTPUT BLOCK DIAGRAM

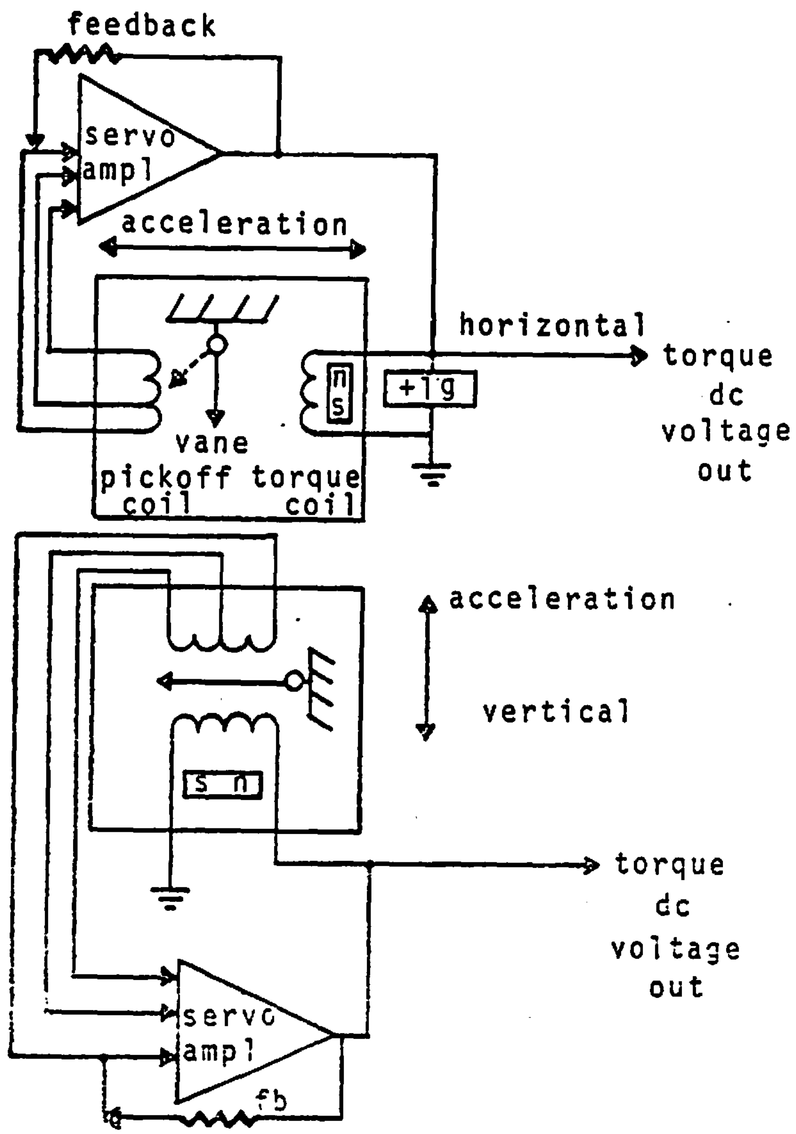


FIGURE 11-3. SENSING CIRCUIT

The modulator consists of an input transformer, amplifiers, a filter network, an output transformer, a diode demodulator, a differential amplifier, and a feedback network as shown in Figure 11-4.

Ac input is amplified and applied to an output transformer. One output goes to the fault logic and accelerometer putput; the other is rectified and applied to an integrated circuit differential amplifier with torque dc output. Torque dc and power dc difference is applied to a Field Effect Transistor (FET).

The FET output is coupled back to the power input amplifier. The FET acts as a variable resistor in the emitter circuit which varies the gain in the ac power

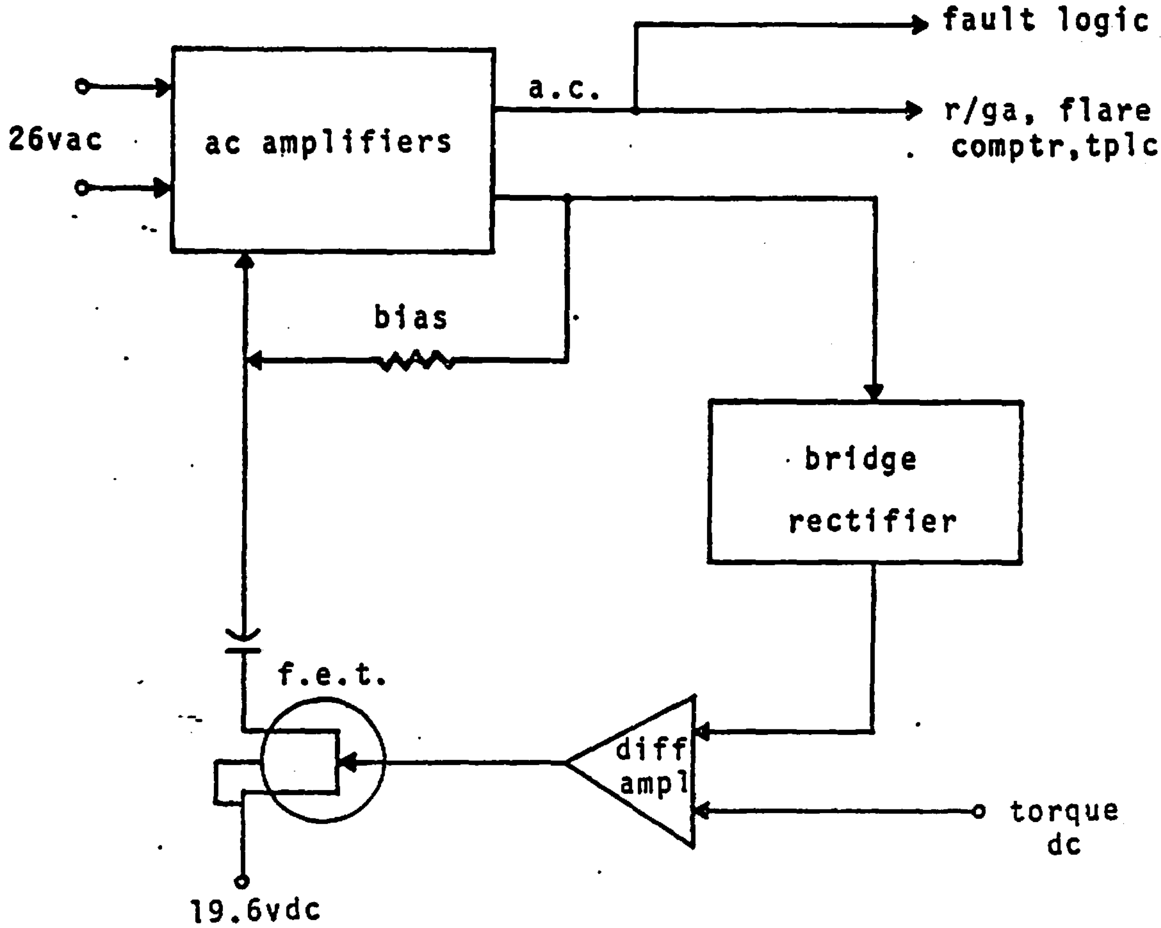


FIGURE 11-4. MODULATOR UNIT (TYPICAL) BLOCK DIAGRAM

circuit. Gain variations keeps the ac RMS value the same as the torque dc voltage.

Fault detection logic detects any condition that results in positive saturation of the torque servo system for more than 1.5 seconds. Self-test circuits provide dynamic testing of the torque servo circuits. If the modulator ac exceeds 13.1 volts, RMS in the logic circuit, power is removed from the modulator. Any fault is indicated at remote points (TPLC). If the voltage regulator allows the output to exceed a zener voltage, the fault relay energizes, which removes power from the modulator and changes the dc output to ground.

The power supply receives 26-volt ac through an rf filter and an isolation transformer to the bridge rectifier. Rectified ac is applied through a regulator circuit set at 19.6-volt dc. Fixed, selected bleeder resistors set the output voltage. Regulated dc output goes through the fault relay contacts to other circuits as shown in Figure 11-5.

Self-test provisions are made to test the accelerometer by means of BITE. A 28-volt dc through the test command relay causes 19.6 volts, dc to be applied to the torque coil and to a large capacitor as shown in Figure 11-2. This voltage to the torque coil causes an over-torque condition. The fault logic will trip causing a fault indication at the TPLC which indicates correct fault logic functioning. With the test off, the fault relay deenergizes, removing the test voltage from the torque coil. The charged capacitor, when returned to a transistor base, provides a large overrange current in the torque coil to return it to normal faster. A current test (in shop) injects a 1-milliampere current into the torque coil. The current causes the amplifier to change to maintain the correct torque coil current. A dc change caused by the test current is equivalent to an acceleration of 0.5 g.

Systems using the accelerometers are the R/GA, Flare Computer, and TPLC.

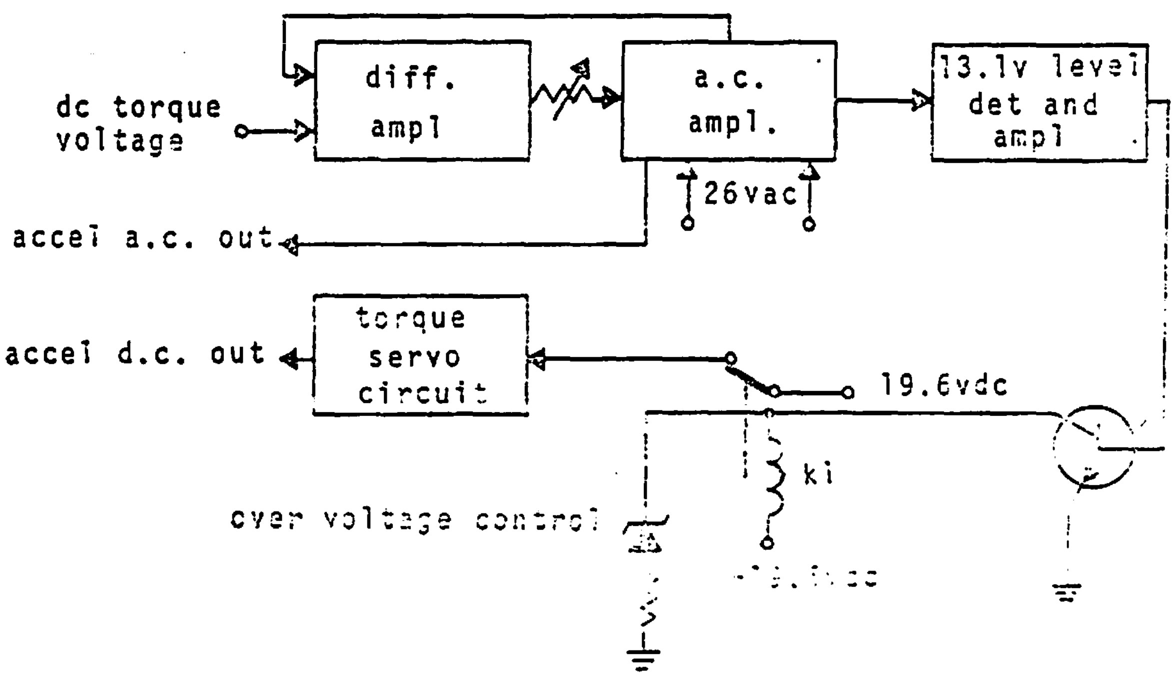


FIGURE 11-5. FAULT LOGIC

APPENDIX

APPENDIX

ABBREVIATIONS AND SYMBOLS

a_1	aimpoint No. 1
a_2	aimpoint No. 2
AA	Approach Arm
ADI	Atitude Director Indicator
AFC	Automatic Frequency Control
AGC	Automatic Gain Control
AGE	Aerospace Ground Equipment
AH	Altitude Hold
AHC	Altitude Hold Capture
AIL	aileron
AL	longitudinal (horizontal) acceleration
AN	vertical acceleration
AOA	Angle of Attack
A/P	Autopilot
APL	Autopilot Lateral
APPR ARM	Approach Arm (AA)
APV	Autopilot Vertical
AT	AWLS Test
ATS	Automatic Throttle System
AUTO	Automatic
AWLS	All-Weather Landing System
BITE	Built-In Test Equipment
BSS	Beam Signal Switch
CADC	Central Air Data Computer
CAT I	Category I
CAT II	Category II
C/B	Circuit Breaker

CC	Confidence Check
CLK	Clock
Coup	Coupler
CR	Crystal Diode
CT	Control Transformer
CWS	Control Wheel Steering
db	decibel
DESENS	desensitizer
DET	turn control in detent
DF	Displacement Flag
DGF	Double Gyro Failure
D_o	distance from a_2 to target (offset)
DOP(P)	doppler computer (ASN-35)
DP	Displacement Pointer
DTG	Distance-To-Go (X)
E_a	analog voltage
E_c	coincidence voltage
EFDC	Elevator Filter dc (28 volts)
EI	Engage Interlock
EIAP	Engage Interlock Autopilot
EL	elevator
E_o	threshold voltage
E_r	range voltage
ERT	En-Route Test
ET	test voltage
f	flap position
FD	Flight Director
FDS	Flight Director System
FDV	Flight Director Validity

FE	Flare Engage
FET	Field Effect Transistor
Fil	Filter
FIP	Fault Identification Panel
FLAG	Fail-Safe Latch And Gate
FPD	Flight Progress Display
FTCM	Full-Time Command Modifier
FU	Follow-Up
grnd	ground
G/S	Glide Slope
GSHC	Glide Slope Holding Coil
GSSS	Glide/Slope Signal Switch
GSW	Glide Slope Window
H	Altitude from sea level
h_a	altitude of aimpoint
h_b	barometric correction
h_c	command altitude
$\dot{h}_c$	complemented altitude rate (IVV)
HDG	heading
$\dot{h}$	Alt Rate
$\dot{h}$ aug	augmented altitude rate
h_e	altitude error (actual)
$\dot{h}_e$	altitude rate (vertical speed) error
HP	Horizontal Pointer
h_p	CADC altitude
h_{RA}	radar altitude
HS	Heading Select
HSI	Horizontal Situation Indicator

HSPD	High-Speed Paradrop
Hz	Hertz (cycles)
IAS	Indicated Air Speed
i-f	intermediate frequency
ILS	Instrument Landing System
ISS	Intermediate Signal Selector
IVV	Instantaneous Vertical Velocity (altitude rate)
LA	LAND ARM
LBS	Lateral Beam Sense
LF	LOC Frequency
LMW	Lateral Manual Warning
LO	Lateral Off
LO	Local Oscillator
LOC	localizer
LVFF	Latch Verify Flip Flop
MCS	Master Caution System
MDA	Minimum Decision Altitude
MH	Mach Hold
MH	Manual Heading
ms	milliseconds
NAV	(ASN-24)
NSN	Navigation Select Navigation
PCPD	Pilot/Co-Pilot Disconnect
PCWS	Pitch Control Wheel Steering
PIT	Pitch Integrator Test
PM	Power Monitor
PMW	Pitch Manual Warning
PO	Pitch Off

TGA	Track Gate Amplifier
T_o	Time Reference (zero time)
TPLC	Test Programmer and Logic Computer
TR	Throttle Retard
TRK	radio track track
V	speed
VBS	Vertical Beam Sense
VER NAV	Vertical Navigation
VF	Vertical Flag
VG	Vertical Gyro
VI	Variable Intercept
VN	VER NAV
VNC	VER NAV Capture voltage
VP	Vertical Pointer
$\overline{\text{VP}}$*	Negated VP
VVI	Vertical Velocity Indicator
WL	Warning Light
X	Distance To Go (DTG)
X_c	capture maneuver pointer
Y/D	Yaw Damper

SYMBOLS

α a	actual AOA signal
α E	AOA error
δ h	altitude error (calculated)
δ F	Flap Position

* overlining of letters indicates negative

PRF	Pulse-Repetition Frequency
PSC	Pre-Select Course
PT	Preland Test
PTT	Press-To-Test
Q	Transistor
RCWS	Roll Control Wheel Steering
RE	Roll Engage
rf	radio frequency
R/GA	Rotation Go-Around
RLY	relay
RMS	Root Mean Square
RT	Receiver-Transmitter
RTC	Radio Track Capture
SCR	Silicon Controlled Rectifier
SEL	select
SS	Signal Switch
ST	Self Test
STA	Strobe A
STB	Strobe B
SV	System Validity
SVF	Super Validity Flare
SVP	Super Validity Pitch
SVR	Super Validity Roll
SW	switch
T_1	elapsed time
TAE	Track Angle Error
TAGC	Tracking Automatic Gain Control
TC	Turn Control

θ	pitch angle signal
$\dot{\theta}$	pitch rate signal
λ	VER NAV angle
δ p	programmed angle of attack
ϕ	bank angle signal
$\dot{\phi}$	bank rate
$\otimes$	signal summation points