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V/STOLAND Digital Avionics System For XV-15 Tilt Rotor

PREPARED BY: SAM P. LIDÉN

JANUARY 1980

SPERRY FLIGHT SYSTEMS
PHOENIX, AZ 85036

CONTRACT NO. NAS2-10326
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FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Ames Research Center
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SECTION I

INTRODUCTION AND PROGRAM OBJECTIVES

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INTRODUCTION AND PROGRAM OBJECTIVES

This final report describes the XV-15 V/STOLAND integrated digital avionics system, the program under which this system was designed, developed and delivered, and the performance obtained with the delivered system. This system was designed for the NASA/Army XV-15 Tilt Rotor aircraft (shown in Figures 1-1 through 1-3) in its three configuration modes: helicopter, tilt, and airplane.

The XV-15 V/STOLAND system was produced as a part of a joint NASA/Army V/STOLAND Flight Research Program which develops operational and design criteria for civil and military operations utilizing advanced V/STOL vehicles. The objective of the XV-15 V/STOLAND part of that program was to produce a digital flight control system for the NASA/Army XV-15 Tilt Rotor research aircraft that provides sophisticated navigation, guidance, control, display and data acquisition capabilities for performing terminal area navigation, guidance and control research.

The V/STOLAND system accomplishes these objectives by a system of software and hardware components that allow liberal modification of the navigation, guidance, control and display functions so that extensive and effective experimental research can be undertaken. Two Sperry 1819B general-purpose digital computers are provided. One, designated as the "Basic" computer, contains the Sperry-developed software that performs all of the specified system flight computations. The second computer, designated as the "Research" computer, is available to NASA for experimental programs that run simultaneously with the Basic computer programs, and which may at the push of a button replace selected Basic computer computations. Other features that provide research flexibility include keyboard-selectable gains and parameters, and software-generated (by the Basic or the Research computer) alphanumeric and CRT displays.

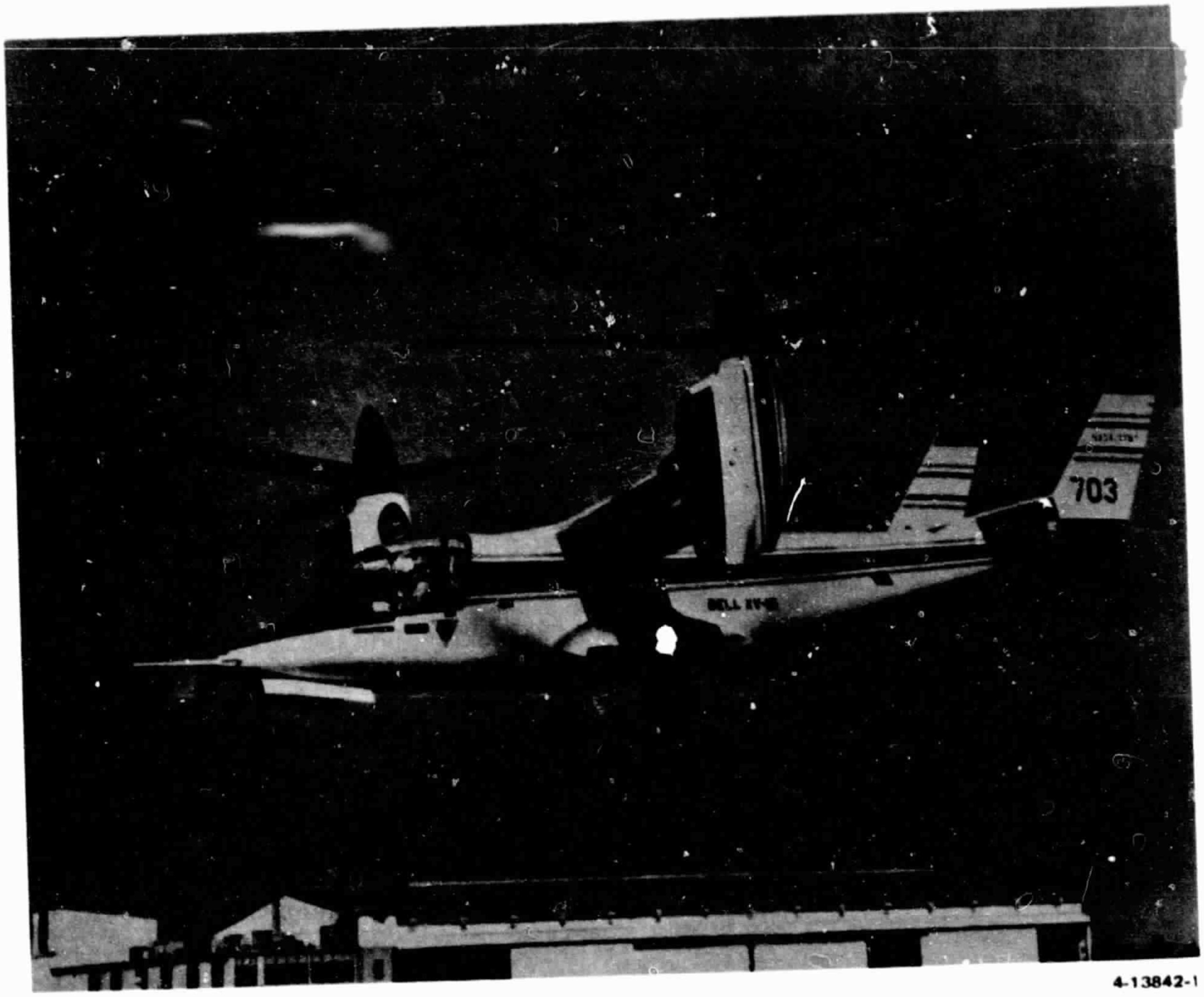

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Figure 1-1
NASA/Army XV-15 in the Helicopter Mode

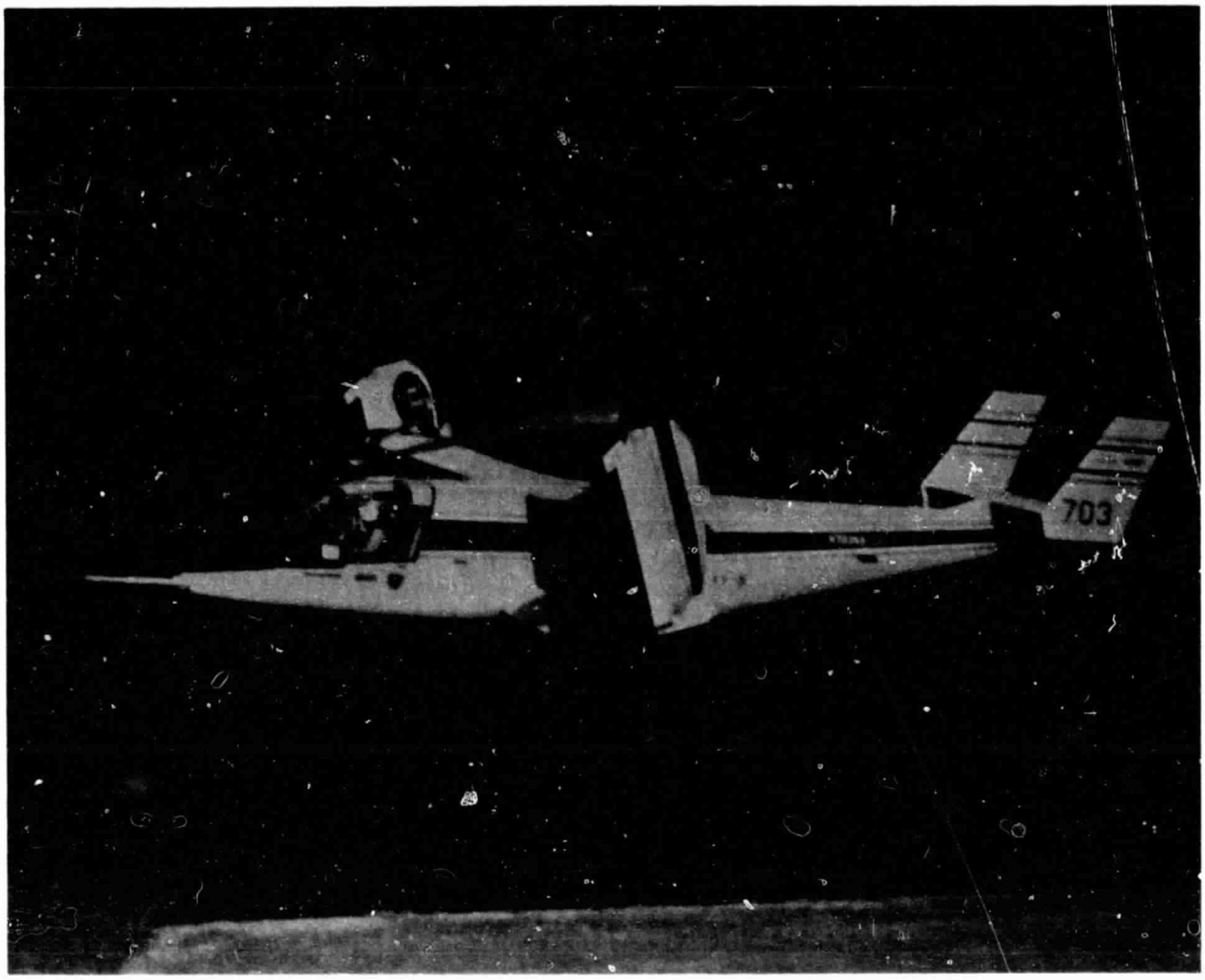

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Figure 1-2
NASA/Army XV-15 in the Tilt Mode

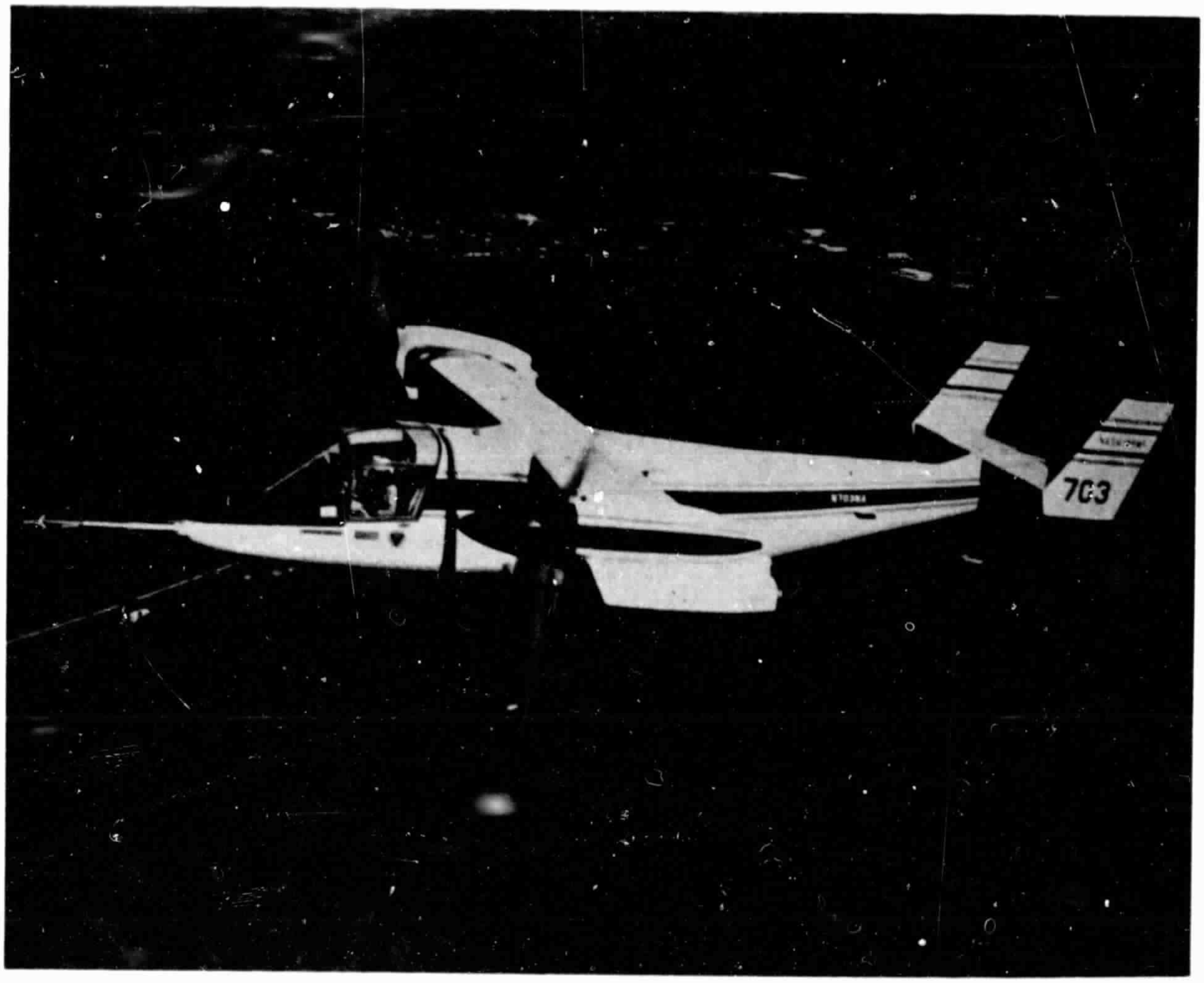

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Figure 1-3
NASA/Army XV-15 in the Airplane Mode

The Basic system performance objectives include:

- Automatic attitude stabilization and control via commands to the Force-Feel system (which is part of the basic aircraft equipment).
- Automatic control of the engine pylon angle (rotor tilt angle) over the entire range, from helicopter mode to airplane mode, as a function of velocity and flight conditions.
- Terminal area navigation, utilizing navaids (VOR/DME, TACAN, MLS) plus conventional body-mounted inertial sensors (attitude and rate gyros, accelerometers) for smoothing.
- Automatic guidance along VOR, TACAN and Waypoint radials, including captures and automatic transitions between such radials.
- Automatic 3D guidance along a prescribed reference flight path.
- Automatic landing guidance to touchdown for straight-in and helix trajectories.
- Flight director guidance.
- Moving map CRT display.
- Extensive in-flight failure monitoring, with diagnostic reporting.
- Air data computation.
- Exhaustive preflight testing and reporting.
- Data acquisition facilities.
- Pilot warning annunciations (display messages, caution lights, audible alarms).

The system that accomplishes these and many other objectives is of necessity large and complex for an airborne system. (These units are listed in Appendix A.)

The Basic computer software utilizes 16,208 18-bit words of memory, and the Research computer is supplied with software utilizing 5178 words of memory (which includes executive and I/O functions, the preflight test program, and general use routines). The software is thoroughly documented in 10 volumes as listed in Appendix B.

The performance of the system was formally demonstrated via the Dynamic Acceptance Test on the S-19 simulator at NASA ARC in February 1978. The simulation cab included servos and control linkages, including hydraulic boost actuators. The aircraft dynamics were simulated on an EAI 8400 computer. Substantial effort was expended in integrating and validating the V/STOLAND equipment on the S-19 simulation facility. The test facility and test results are described in Section VI.

The remainder of this document begins with a section that briefly reviews the history of the program. It is followed by Section III which summarizes capabilities of the V/STOLAND system. Section IV describes the equipment that comprises the system, and Section V describes the system software, covering the executive, control, guidance, navigation, and the failure monitoring and diagnostics functions with extra detail. The validation and testing of the system is then described in Section VI, followed by conclusions and recommendations in Section VII.

SECTION II

BRIEF CONTRACT HISTORY

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BRIEF CONTRACT HISTORY

The XV-15 V/STOLAND system was produced under an amendment to an ongoing contract for a similar system for the UH-1H helicopter (V/STOLAND Digital Avionics System for UH-1H, Final Report, NASA CR-152179).

Both V/STOLAND programs were preceded and overlapped by several NASA/Sperry programs for STOL aircraft, referred to as STOLAND. Three STOLAND systems were delivered. Two systems were installed in aircraft and one in a simulation facility. One of the aircraft, the Augmentor Wing, is a modified de Havilland C-8A Buffalo, fitted with jet engines, rotatable hot thrust nozzles, augmentor wings and a hydraulic powered elevator. The other is a de Havilland DHC-6 Twin Otter fitted with spoilers. The original contract was authorized in June 1971.

The UH-1H V/STOLAND program incorporated much of the technology and experience achieved under the STOLAND effort, but also took advantage of newer technology (for example, the newer 1819B computer replacing the 1819A).

The XV-15 system has many components in common with the UH-1H system, but a few were redesigned to accommodate the smaller cockpit space available on the XV-15, and the Data Adapter was redesigned for the XV-15 requirements.

The system design and requirements changed considerably during the course of the program, as might be expected for a large and complex research-oriented system. Specific dates are given in Table 2-1 for the preliminary and final design reviews; however, several follow-up design reviews were also conducted for further definitions and revisions of components and software modules. Many of the modifications were of major impact, requiring contract modifications.

TABLE 2-1

MAJOR MILESTONES

Contract Start	September 1974
Preliminary Design Review	July 1975
Final Design Review	March 1976
System Delivery to NASA	November 1977
Installation at NASA Completed	February 1978
Dynamic Acceptance Testing Completed	April 1978
Contract Completed	September 1978

Table 2-2 lists the significant XV-15 V/STOLAND contract changes from the original contract (NAS 2-7306) which became effective on December 15, 1972 for the UH-1H program. The XV-15, Phase IV, of the V/STOLAND contract was initially authorized by Contract Amendment No. 12, effective September 1, 1974. The table is in chronological order, and each change is identified by a modification number and a brief description of the change. Note that the table refers only to XV-15 hardware, software, documentation, and design studies, and specifically does not cover the changes associated with the UH-1H or XV-15 cost and schedule changes.

The S-19 simulator at NASA Ames Research Center was configured and mechanized to represent a comprehensive simulation of the V/STOLAND XV-15 aircraft system in flight. The equipment included a cab with a complete instrument panel, control sticks and pedals, control linkages and booster servos, an AHS (Airborne Hardware Simulator) that simulates the interfaces of airborne hardware (navaids, sensors, etc) to the Data Adapter, the EAI 8400 simulation computer facility with the associated Redifon visual terrain display system, and miscellaneous computer terminals and peripherals. The extensive electrical interfaces were checked out by a formal Static Acceptance Test which utilized special software in the Basic, Research, and Simulation computers. The integration of the V/STOLAND system in this simulation facility was a significant task, but proved to be a very effective tool in validating the system and in responding to the research pilot's evaluation.

TABLE 2-2

TECHNICAL CONTRACT CHANGES

Modification Number	Effective Date	Description
12	9/1/74	Study, System Design, and Preliminary Software Development for V/STOLAND to be used in XV-15 Research Aircraft.
18	12/16/74	Authorized Phases V and VI of XV-15 Program: • Hardware and Software for XV-15 Aircraft • XV-15 Simulator Hardware and Software Changes • Ground-Support Equipment and Software to Support XV-15 V/STOLAND System.
23	5/14/75	Added four new items and deleted nine items as a result of Phase IV Study: Added: Data Adapter; Flight Mode Annunciator; Mode Select Panel; Data Entry Panel Deleted: Basic Data Adapter; Aux Data Adapter; Mode Select Panel; Status Panel; Keyboard; Mode Status Panel; MFD Control Panel; Panel Power Supply; Servo Interlock Unit
32	1/28/76	Deletes the HSI Signal Conditioner.
33	2/9/76	Authorized work on XV-15 Simulation Bench at Phoenix.
39	3/15/77	Deletes a portion of Phase VI, Task XI.
41	5/3/77	Deletes XV-15 GFE Flight Rack Checkout.
45	9/30/77	Deletes XV-15 Model Specification.
46	10/7/77	Authorizes Message Alert Lights.
54	7/1/78	Deletes XV-15 Final Report.

SECTION III

SUMMARY OF SYSTEM CAPABILITIES

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SUMMARY OF SYSTEM CAPABILITIES

The V/STOLAND system was designed to be a research tool with a high degree of flexibility and capability for experimentation. Furthermore, a software package was developed for the Basic computer which implements sophisticated guidance and control functions, navigation computations, display data, self-test monitoring and diagnosis, and numerous other functions. This section summarizes the most noteworthy capabilities of this system in general operational terms.

3.1 RESEARCH-ORIENTED ARCHITECTURE

Several characteristics of the V/STOLAND system architecture make it particularly suitable for research and experimental projects. Figure 3-1 illustrates the general architectural configuration. The principal features are described in the following paragraphs.

Integrated Computations - A single 1819B computer, designated as the Basic computer, performs all computations for the complete set of prescribed airborne functions under programs running in core memory. This integrated approach provides for a high degree of flexibility in designing and modifying all computed functions, including guidance and control laws, flight reference trajectories, displayed data, monitoring, instrumentation, and data interchange between the various functions. A new program may be loaded into the computer memory before each flight.

Dedicated Research Computer - A second identical and interchangeable computer, designated as the Research computer, may be programmed by experimenters to perform specific experimental functions without being burdened by the I/O processing or other computations not directly related to the experiment.

All input data to the Basic computer from sensors, control panels, etc, is immediately decoded and then transferred to the Research computer with minimum delay, at 40 Hz sampling rate. Also transferred at this rate is an extensive list of data computed by the Basic computer programs which may be utilized in experimental routines in the Research computer, just as if they had been

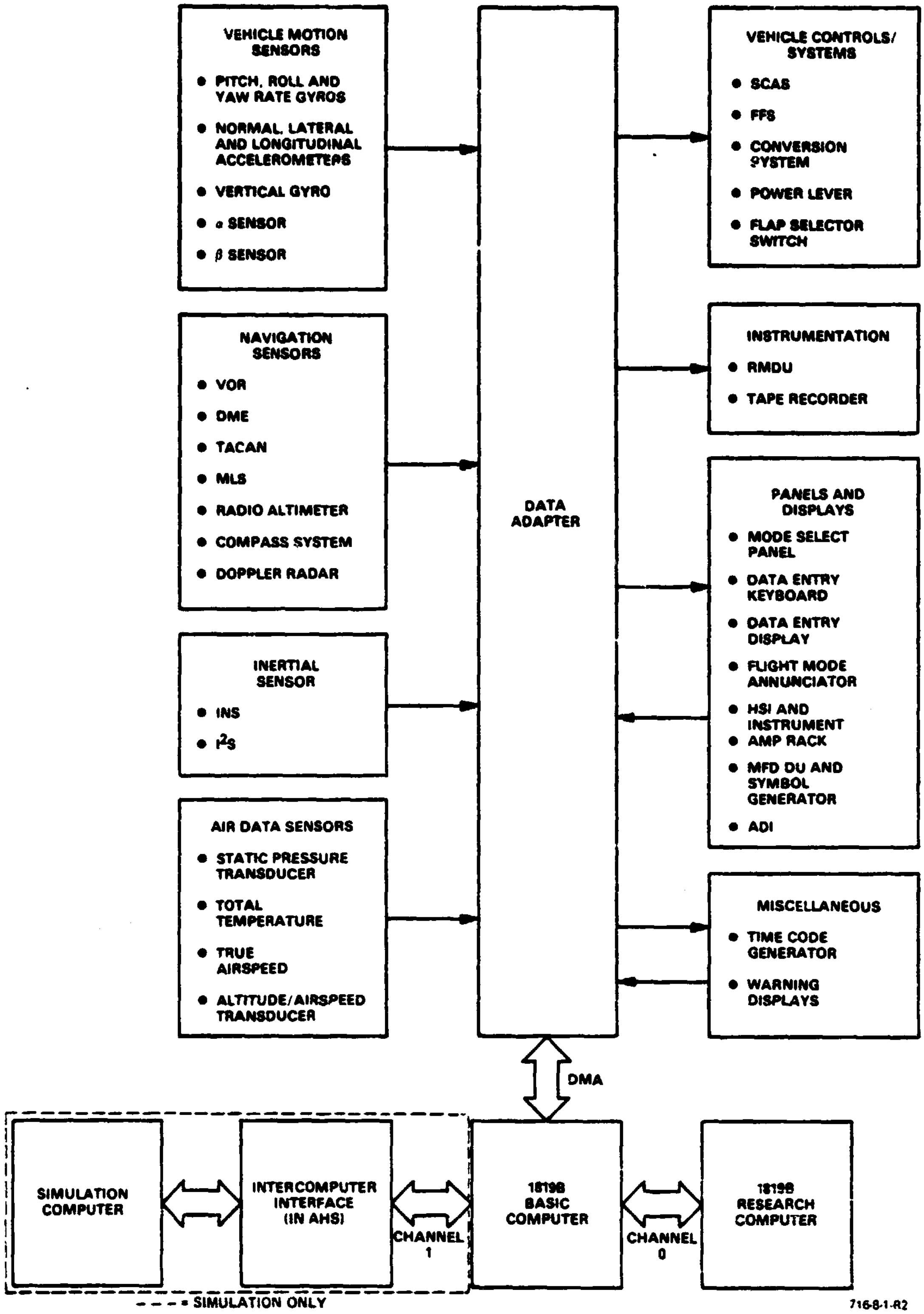

Figure 3-1
V/STOLAND System Architecture

computed there. Research modes, described in Paragraph 3.6, may be selected which substitute for Basic computer modes without incurring additional transport lag.

Large Complement of Sensors - The V/STOLAND system includes interfacing to an extensive set of inertial, radio, and air data sensors to permit a variety of experiments in navigation and guidance. Such sensors include:

- Pitch, roll, and yaw rate gyros
- Normal, lateral, and longitudinal accelerometers
- Vertical gyro
- Compass system
- Static pressure transducer
- True airspeed sensor (J-TEC)
- LORAS airspeed sensor
- Radio altimeter
- VOR receiver (digital)
- DME
- TACAN
- ILS
- MLS
- LTN-51 inertial navigation system
- Doppler radar system
- Tetrad ring-laser gyro

Software-Driven Displays - Considerable flexibility and research capacity in display techniques are provided by the following software-driven displays. Figure 3-2 shows the displays as installed in the Sperry simulator.

- CRT Multifunction Display (MFD) - This stroke-written CRT display is suitable for maps or any figures that can be constructed of lines. It also displays alphanumerics and other symbols, and is completely controlled by a data stream generated in the computer.

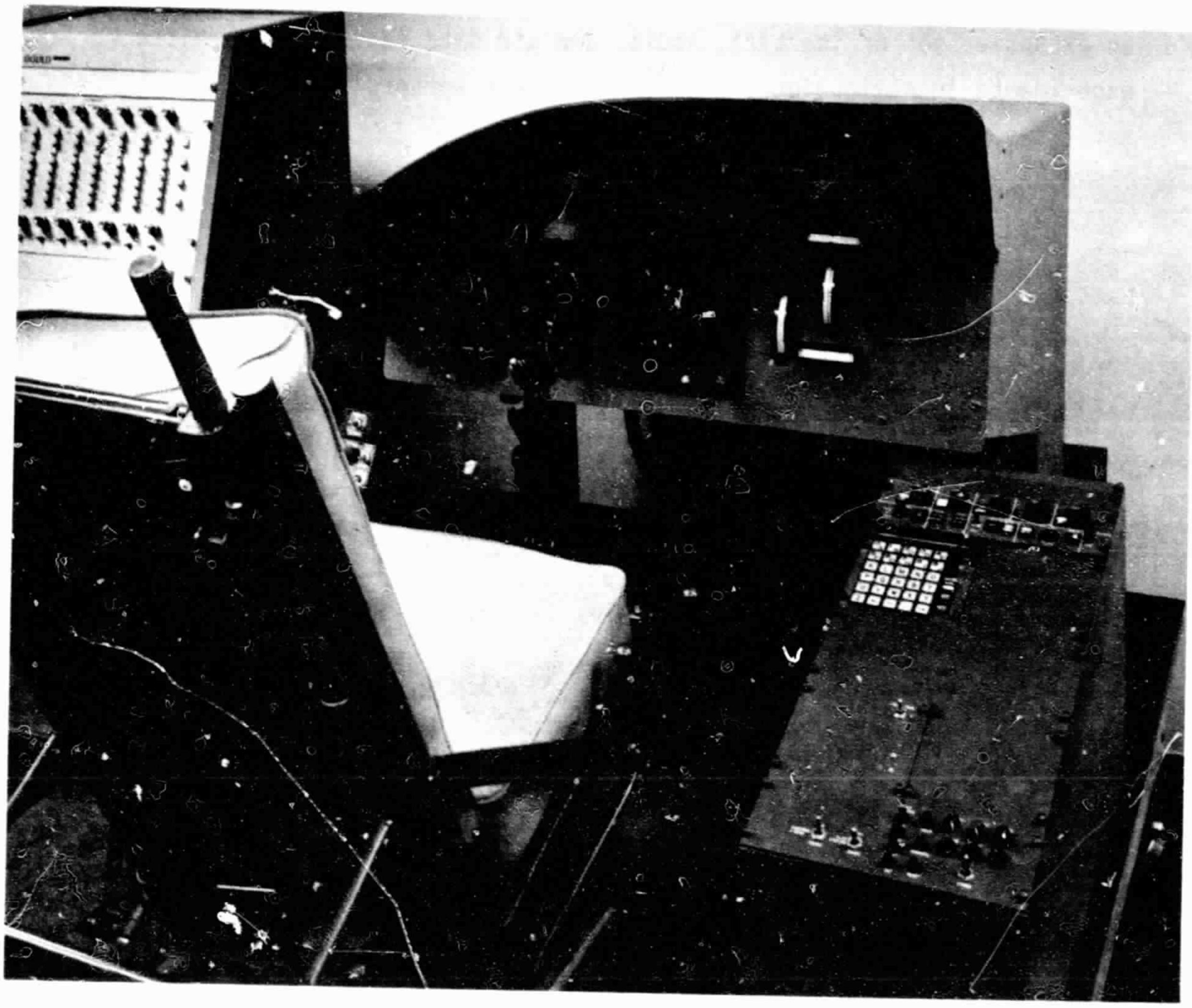

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Figure 3-2
Panels installed in Sperry Simulator

- Alphanumeric Displays - The 24-character Flight Mode Annunciator (above the ADI), the 24-character Data Entry Display on the Status Panel (on the center console), and the five numeric reference displays on the Mode Select Panel are completely controlled by software, permitting flexibility in displaying messages to the pilot.

- ADI/HSI Indicators - All but the primary attitude displays on the ADI (pitch and roll attitude) and on the HSI (heading) are controlled by software, allowing flexibility in the presentation of deviations, flight directors, flags and the two numeric range displays on the HSI (see Figures 4-51 and 4-52 in Section IV).

Software-Sensed Control Panels - Two control panels contain buttons and other switches which are sensed directly by software. The pushbuttons are also illuminated under software control to provide mode annunciation. All buttons are currently dedicated and labeled for prescribed functions; however, such dedication is totally under software control.

- Mode Select Panel - 12 dedicated pushbuttons, each with up to 5 illumination states (including off); 5 slew switches (5-position, spring-return to center); one 4-way map slew switch.

- Data Entry Keyboard - A 30-button (6 x 5) alphanumeric keyboard plus 3 auxiliary buttons (letter/number, enter, clear). The keyboard software allows entry of data, such as gains and other parameters, by keying in 3-letter mnemonics plus the desired associated numeric data. This panel provides a high level of flexibility for modifying parameters or modes in flight.

Dedicated Parameter Assignments - The Basic computer programs are written to facilitate modifications of gains, thresholds and other parameters by assigning a distinct and labeled word in a data table for each parameter that could conceivably want to be changed under future program experimentation and development. Such parameters are thereby also accessible for modification via the keyboard.

Data Recording Capability - An on-board digital recorder records 96 words as defined by computer software at a rate of 20 times per second. Recorded flight test data may subsequently be processed to produce graphs as desired.

Simulation/Validation Facilities - The system is configured to facilitate development and validation of new or revised software on NASA's S-19 simulation facility before it is taken on a flight test. Equipment racks, interface equipment, and a cab that includes a full set of instruments and servos are provided which allow mounting the airborne V/STOLAND equipment in the simulation facility for a thorough checkout before a flight test is conducted.

3.2 GUIDANCE AND CONTROL FEATURES

The Basic computer software includes guidance and control programs that provide total hands-off automatic control, or flight director guidance via the ADI flight director indicators. The guidance modes and associated reference values are selectable on the Mode Select Panel. The following guidance modes may be selected:

Independent Vertical-Longitudinal Modes:

- Airspeed Select/Hold
- Flight Path Angle Select/Hold - to ±15 deg
- Altitude Select/Hold - Automatic transition through Flight Path Angle select and hold

Independent Lateral-Directional Modes:

- Heading Select/Hold
- TACAN Radial Capture/Hold
- VOR Radial Capture/Hold
- Waypoint Radial Capture/Hold - The waypoint is a virtual TACAN/VOR station which may be placed at any point on a runway-referenced x-y coordinate frame (within a 100-nmi square) via the keyboard.

The reference angles for these modes are independently selectable on the Mode Select Panel, and transition between selected modes is automatic when capture conditions are satisfied.

Three-Dimensional Modes:

- Reference Flight Path - A path composed of straight and circular line segments connecting a sequence of waypoints defined in an x-y-z coordinate frame (see Figure 5-4). Lateral and vertical capture is automatic when respective capture conditions are satisfied.

- Straight-In Land (LAND-1) - A path lined up with the runway laterally, but with a selectable glide slope. The aircraft is also automatically decelerated in accordance with a prescribed velocity profile until it comes to a hover 10 feet above the designated touchdown point. Finally, the aircraft descends to touchdown under a prescribed letdown profile.

- Helix Land (LAND-2) - A land path that includes a 2-turn helical trajectory to provide 1600 feet of descent within a confined area (helix radius = 1189 feet). This path terminates on a "pad" offset from the main runway.

Capture of the Land modes may be set up by flying the Reference Flight Path, or any path that intersects the Land path, and then pushing the appropriate buttons to arm the desired Land mode.

3.3 NAVIGATION

The Basic Computer Navigation Software Module computes the aircraft position and ground velocity with respect to the Crow's Landing runway coordinate frame by using ground-based navaid position data augmented with acceleration data derived from a strapdown system. The available navaid data sources are the VOR/DME at Stockton and the TACAN MLS at Crow's Landing. The acceleration data is supplied by the outputs of three body-axis-mounted accelerometers.

The pilot may control the navigation function from the Mode Select Panel by the switch and buttons in the NAV SOURCE area (see Figure 4-42). With the AUTO/MAN switch in the MAN position, the pilot may manually select any valid navigation source by pushing a button illuminated amber. The TAC/VOR button alternately selects the indicated nav source. Green illumination indicates the selected source. With the AUTO/MAN switch in the AUTO position, the system automatically selects the navaid source based on priority and validity.

The navigation outputs are used to position the aircraft symbol on the MFD map display. The aircraft symbol will be displayed on the map any time the navigation module has a reasonably accurate value for the aircraft position. This will be the case when a valid navigation mode is selected or during a two-minute, dead-reckoning period after valid navigation data is lost where the aircraft position is updated only from accelerometer values and the best estimate of wind velocity.

During normal operation, the raw navaid inputs are prefiltered to smooth the data and eliminate "dropouts" which often occur. A "dropout" is defined as an excursion of the data to an incorrect and unreasonable value for a short period of time while the associated valid remains high. The raw data prefilter technique involves continuous estimation of the rate of change of the raw data based on aircraft ground speed and heading. When a data dropout occurs, the condition is sensed by comparing the raw data with the estimated value, i.e., the output of the prefilter. The estimated value is then updated by integrating the rate estimate until the raw data returns to a reasonable value. If the dropout lasts more than ten seconds, the estimate is declared invalid and the navigation reverts to dead reckoning unless a valid navigation source is selected manually or automatically.

The prefiltered raw data estimates are used to compute the aircraft position in the x-y runway axis system. This value of aircraft position is input to the navigation x and y complementary filters along with runway axis acceleration values derived from the body axis acceleration values. The complementary filters generate the filtered estimates for aircraft position and velocity, and the prefilters isolate the complementary filters from the effects of the raw data dropouts. This technique has proven effective in maintaining reasonable navigation outputs in the presence of source data dropout conditions.

The Navigation software module also computes the barometric altitude of the aircraft from static pressure sensor data and the aircraft height above the runway at Crow's Landing. The height complementary filter output is derived from barometric, MLS, and radio altitude position information augmented with vertical acceleration derived from the body-axis-mounted accelerometers. Barometric position data is used until MLS height information is available. When MLS height information is available, it is blended into the height filter input while barometric height is blended out. Below 400 feet AGL, radio altimeter height is blended into the height complementary filter position input. Below 100 feet, the height complementary filter gains are increased significantly to allow precise height tracking to the touchdown point.

Other outputs of the Navigation software module include wind components in the runway axis system and ground speed. Further details of the navigation software module are presented in Paragraph 5.4.

3.4 MOVING MAP CRT DISPLAY

The Multifunction Display (MFD) program presents a map window on the MFD screen that may be moved over a 100 x 100 nmi area centered at Crow's Landing. The map scale is selectable at 5, 1.5 or .5 nmi per inch, and the map window may be slewed by a 4-way slew switch which causes the map to move at 4 inches per second in the selected direction. The map may also be displayed in the North-Up mode or the Heading-Up mode. These selections are made in the MFD area of the Mode Select Panel.

In the North-Up mode (where north is up on the screen) the map is stationary and the aircraft symbol (a triangle) moves on the screen. It may move off the screen edge if the map is not slewed appropriately. In the Heading-Up mode the aircraft symbol remains stationary at the center of the screen, pointing up, and the map moves correspondingly. A representative display is shown in Figure 3-3.

The location of the aircraft on the map is obtained from the navigation computations. When the navigation goes into the dead-reckoning mode, the aircraft symbol flashes for 2 minutes after valid navigation data has dropped to warn the pilot of this condition. If valid navigation is not restored by the end of the 2-minute period, the aircraft symbol then disappears from the screen.

As shown in Figure 3-3, a series of track history dots are displayed behind the aircraft, each dot representing a 10-second interval. Similarly, two lines ahead of the aircraft symbol predict where the aircraft will be, up to 40 seconds in the future, assuming that the current aircraft attitude and velocity are maintained.

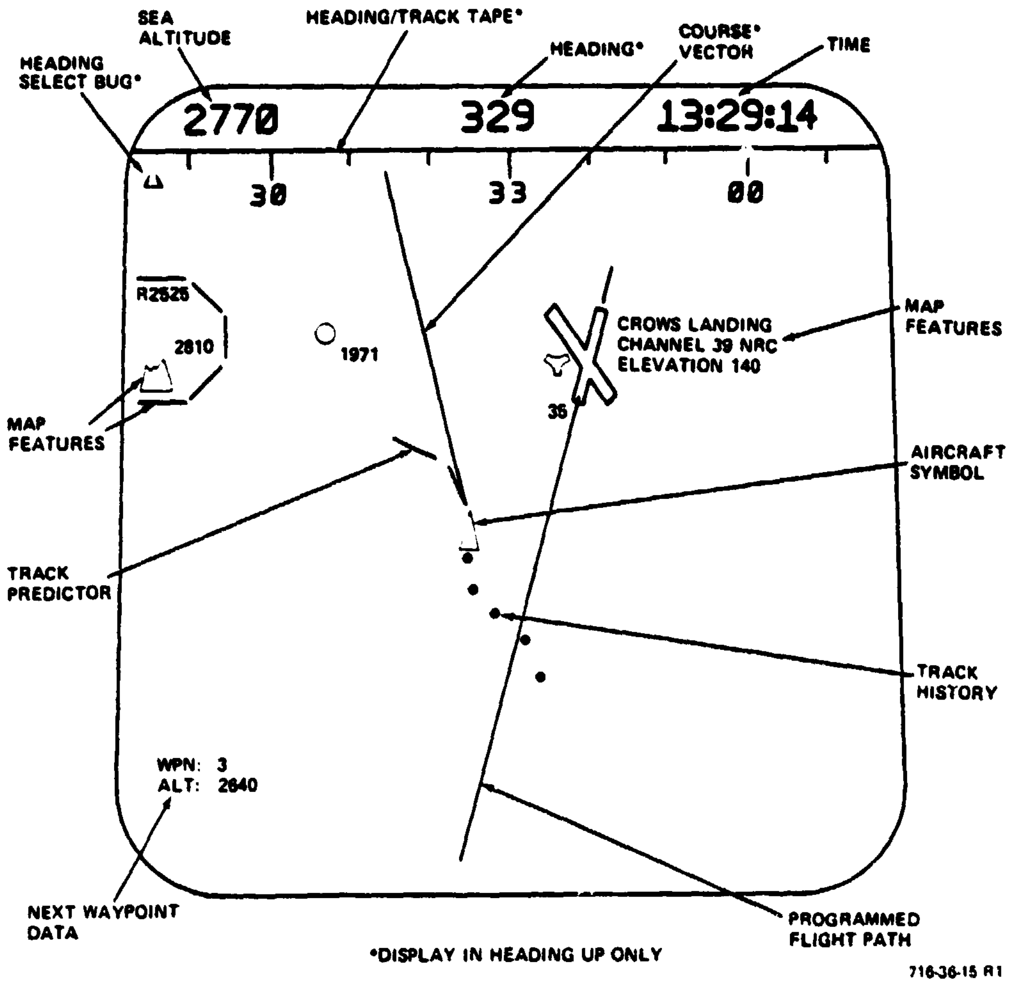

Figure 3-3
MFD Face Showing Moving Map Display
Features in the Heading-Up Mode

3.5 FAILURE MONITORING AND DIAGNOSTICS

The Basic computer program includes a failure monitoring and diagnostics program module which monitors the V/STOLAND system operation and, upon detection of a malfunction, causes the appropriate disconnect actions and reporting to occur. The module monitors data valid signals from the sensors and subsystems that provide such valids, before this data is used in the flight program. It also calculates software monitors to augment the hardware monitors as described in Paragraph 5.5.

If a failure occurs in a component required for an engaged Autopilot mode, the Autopilot will disconnect, the red V/STOLAND caution light on the instrument panel will flash, and a diagnostic message will be displayed on the Data Entry Display Panel at the center console. This message will also be logged in the computer memory, and may be reviewed via the keyboard at any later time (until the failure lagging buffer is cleared).

If a failure occurs in a component that is not critical to the Autopilot (for example, the MFD display), the V/STOLAND light will go on (but not flash), and a diagnostic message will be displayed and logged as described above. Paragraph 5.5 describes the monitoring and diagnostics functions in more detail.

3.6 RESEARCH MODES

The system has the capability of operating with software modules in the Research computer which replace functions in the Basic computer. The Research mode is enabled by setting RES to a non-zero number on the Data Entry Keyboard. When this is done, any of the following Research sub-modes may be engaged by the Research computer software by setting associated flags which are part of the data transmitted from the Research to the Basic computer:

- Vertical/Longitudinal Guidance and Control - When RVLGAC is set (by the Research computer) the commands from the Research computer vertical and longitudinal guidance and control computations are used instead of the analogous Basic computer commands.

- Lateral/Directional Guidance and Control - When RLDGAC is set, the lateral and directional Research commands are used as described above.

• Navigation - When RESNAV is set, the computed estimates for aircraft position and velocity obtained from the Research computer are used by the Basic computer instead of the Basic-computer-derived estimates.

• Multifunction Display (MFD) - When RESMFD is set, the output to the MFD is channeled from the Research computer instead of from the Basic MFD computations.

• HSI/ADI Course and Vertical Deviation - When RESAM is set, the course and vertical deviation displayed on the HSI and the ADI come from the Research computer instead of from the Basic computer computations.

The Research mode must be enabled to engage the above Research sub-modes. The setting of the eight sub-mode flags (RVLGAC, etc) is controlled entirely by the Research computer software. Setting these flags may be initiated from the keyboard by the number entered for mnemonic RES. This number is transmitted to the Research computer. If there is a failure in the Research computer, in the Basic/Research I/O, or in the Research software as indicated by the computed Research software valid flag (RSWVAL), the Research mode will automatically disengage.

A large data buffer transferred to the Research computer every compute cycle (25 ms) includes formatted input data from sensors, switches, etc, plus the major computed variables. The Research software may use this data as desired, and may return any portion of the computed data without modification to reduce the amount of computations performed in the Research computer. The Research-to-Basic buffer is 250 words long, and this input buffer is open until approximately 5.5 milliseconds after the Basic-to-Research output buffer is completed. Therefore, the Research computer computations performed within this period will not suffer any transport lag due to being cycled through the Research computer. This time period is adequate to complete the high-speed computations.

3.7 PREFLIGHT TESTING

A comprehensive preflight test function is provided by a software module that exercises the V/STOLAND flight system hardware and associated interfaces, and verifies that the system is in proper working condition. Alphanumeric messages guide the test conductor in conducting the tests, and diagnostic messages are provided for tests that fail. The preflight test software is resident in the Research computer, and operates in conjunction with the Basic computer airborne software.

The preflight test is divided into six sections, and each section is further divided into its associated subsections. This sectionalization makes it possible to bypass sections not of interest, and the tests for a specific section or LRU can be accessed easily. The preflight test sections are as follows:

- Section 000 - Valids and Nulls
- Section 100 - Panels
  - Data Entry Display (DED)
  - Flight Mode Annunciator (FMA)
  - Data Entry Keyboard (DEK)
  - Mode Select Panel (MSP)
- Section 200 - Displays
  - Horizontal Situation Indicator (HSI)
  - Attitude Director Indicator (ADI)
  - Multifunction Display (MFD)
- Section 300 - Pilot Controls and Control Systems
  - A/P Engage Logic
  - FFS Servos
  - Flap-Select Actuator
  - Conversion System
  - RPM Control
  - Power Lever
- Section 400 - Sensors
  - Force Transducers
  - Inertial Sensors
  - Air Data Sensors

- Section 500 - Navaids
  - Navaid Initialization
  - Radio Altimeter
  - VOR
  - DME
  - TACAN
  - MLS
  - INS
  - Doppler

The preflight test program offers the user each of the major test sections listed above, one at a time. If a section is selected, the program then offers the user the subsections in the order listed until a subsection is chosen, or until all subsections and sections have been offered.

The test operator interfaces with the preflight test program through Data Entry Keyboard (DEK) entries and through Data Entry Display (DED) and Flight Mode Annunciator (FMA) messages. As a general rule, the FMA is used to display the program status (i.e., what is presently under test) while the DED is used as the working display for instructing the operator to do certain procedures, for informing the operator of the results of a specific test or procedure, and for asking the operator to respond to some types of test results.

The preflight test engagement is interlocked with weight-on-wheels. If weight-on-wheels is present and the V/STOLAND system is powered, the preflight program is initiated by the operator typing PFT=1 on the keyboard. After the operator has pressed the ENTER button, the introductory message XV-15 V/STOL PFT TEST will be displayed on the FMA for 5 seconds. This message indicates that the preflight program has been successfully engaged and that the various preflight flags and program directors have been initialized. The program then begins to offer sections for testing as described above.

An example of display messages for section selection is as follows for the two display panels:

FMA	DED
* PANELS *	SECTION 100
SECTION 100	PUSH T OR →

If testing in a subsection of Section 100 is desired, the operator pushes T on the keyboard; otherwise he pushes → to go on to the next section. The format is similar for selecting or bypassing subsection tests.

Some tests require operator verification. For example, in a test of the HSI heading-select cursor, the cursor is driven to 45 degrees and the DED displays:

HDG SEL = 45°
VERIFY: F/→

If the operator observes the cursor at 45 degrees, he pushes on the keyboard to go on to the next test item. Otherwise he pushes F to log a test failure. This logging occurs automatically for tests that do not require operator verification.

When a test failure has been logged, the DED identifies the failed component and offers to provide additional detail in the form of a diagnostic number. An example of such a display is:

MODE SEL PNL
FAULT: D/→

If D is pressed, the display will be

MODE SEL PNL
DIAG NO. 146

or any other of 44 diagnostic numbers devoted to the MSP. The diagnostic 146 means that a fault was detected when the HDG HLD/SEL pushbutton discrete was being tested. The operator's manual* lists failures by diagnostic numbers (Table 3-3), as well as all FMA and DED messages for all failure modes under each section and subsection (Table 3-2).

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*V/STOLAND Digital Avionics System for XV-15 Tilt Rotor Aircraft, Operator's Manual, Sperry Flight Systems, Pub. No. 71-1255-00, August 1978.

At the end of the preflight test, the program control is transferred to the failure summary mode. If there have been no failures, the message V/STOL PFT TEST PASSED appears on the FMA and the message NO FAILURES appears on the lower DED (upper DED is blank). These messages are displayed for 3 seconds and then the preflight program executes an automatic transfer to the preflight exit logic.

If there have been failures, the failure summary logic displays the total number of failures that have occurred on the upper DED and instructs the operator to PUSH D OR P via the lower DED. The operator may then continuously review the diagnostic numbers of the failed tests on the upper DED by repeatedly pressing the D button on the Data Entry Keyboard (DEK). The numbers will appear in ascending order, one number for each button depression. After the highest numbered diagnostic has been reviewed and the operator presses the D, the DED will read as follows:

REPEAT DIAG
PUSH D OR P

If the D button is pressed, the diagnostic review procedure is repeated. The operator may initiate a program exit at any time by pressing the P button.

The operator may also elect to terminate the preflight test at any time during the test (except during the DEK K through T test) by pressing the P button on the DEK. If the end of the preflight test is reached and no failures have occurred, the exit will be initiated automatically. The preflight exit logic displays the message

EXITING PFT
PFT = 0

on the DED for 3 seconds to inform the operator that the preflight program is being terminated. The preflight logic then drops the preflight engage flag and program contol is returned to the basic airborne mode.

SECTION IV

DESCRIPTION OF SYSTEM HARDWARE

SECTION IV

DESCRIPTION OF SYSTEM HARDWARE

4.1 GENERAL DESCRIPTION

The principal contractor-furnished components comprising the V/STOLAND system, and their signal flow relationships, are shown in Figure 4-1. The cockpit equipment used or viewed by the pilot is configured as shown in Figure 4-2, and includes:

- Flight Mode Annunciator (FMA)
- Mode Select Panel (MSP)
- Data Entry Display (DED)
- Data Entry Keyboard (DEK)
- Attitude Director Indicator, Sperry HZ-6F (ADI)
- Horizontal Situation Indicator, Sperry RD-202 (HSI)
- Multifunction Display (MFD)
- Miscellaneous Controls and Indicators

Other equipment is mounted in a flight rack in the cargo compartment as shown in Figures 4-3 through 4-5. This equipment includes:

- Basic Computer
- Research Computer
- Data Adapter
- Inertial Sensors
- Air Data Sensors
- Navigation Sensors
- Instrumentation System (DDAS)

Appendix A contains a complete list of system components.

4.2 THE AIRCRAFT SCAS AND FFS INTERFACES

The V/STOLAND Autopilot operates through the aircraft Force-Feel System (FFS), and only with the SCAS engaged. A schematic illustration of the mechanical controls is given in Figure 4-6. Figures 4-7 and 4-8 are block diagrams of the XV-15 flight control system, showing all pilot and autopilot inputs, and all controlled outputs.

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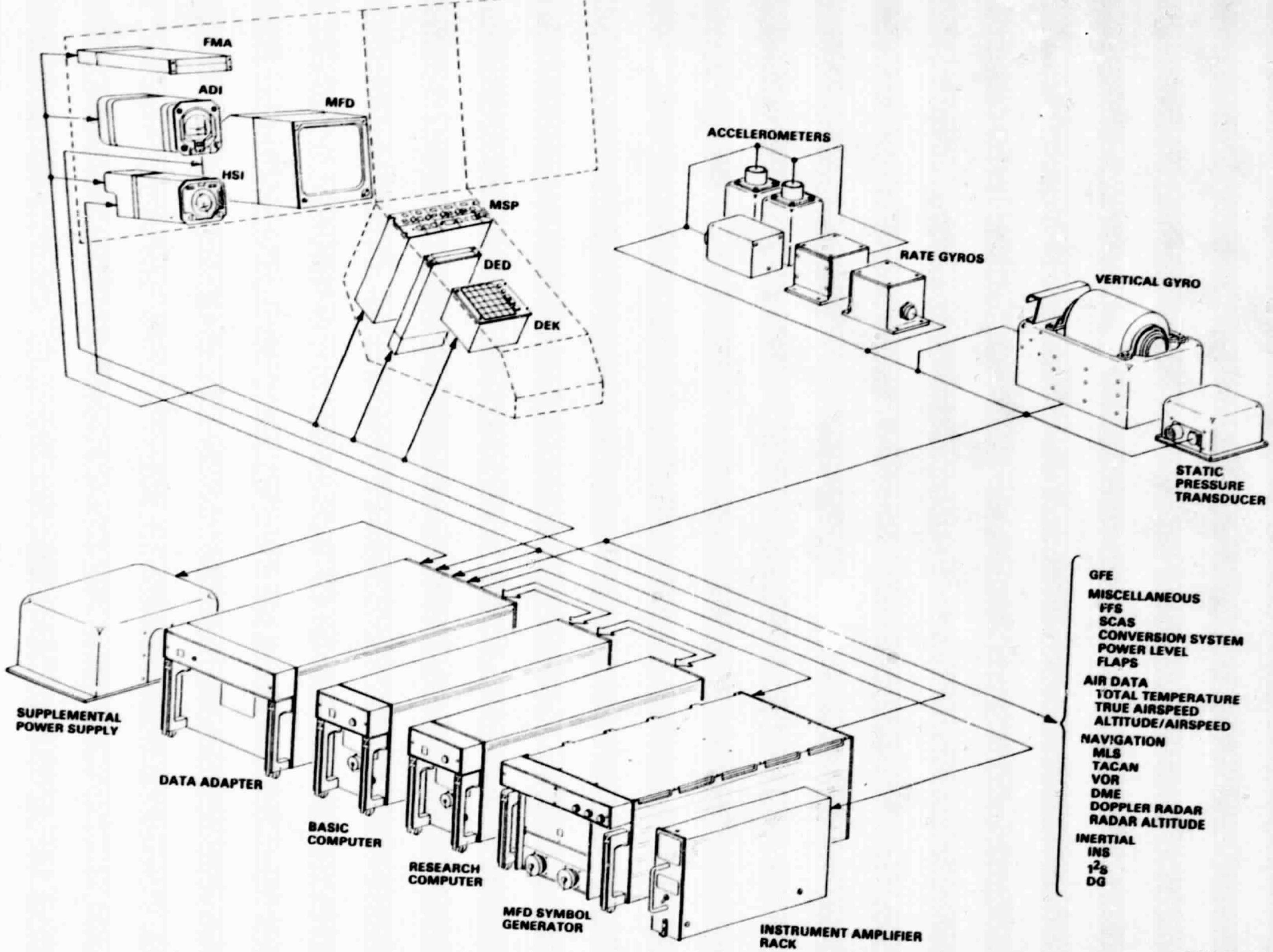


Figure 4-1
Component Configuration Diagram
of XV-15 V/STOLAND System

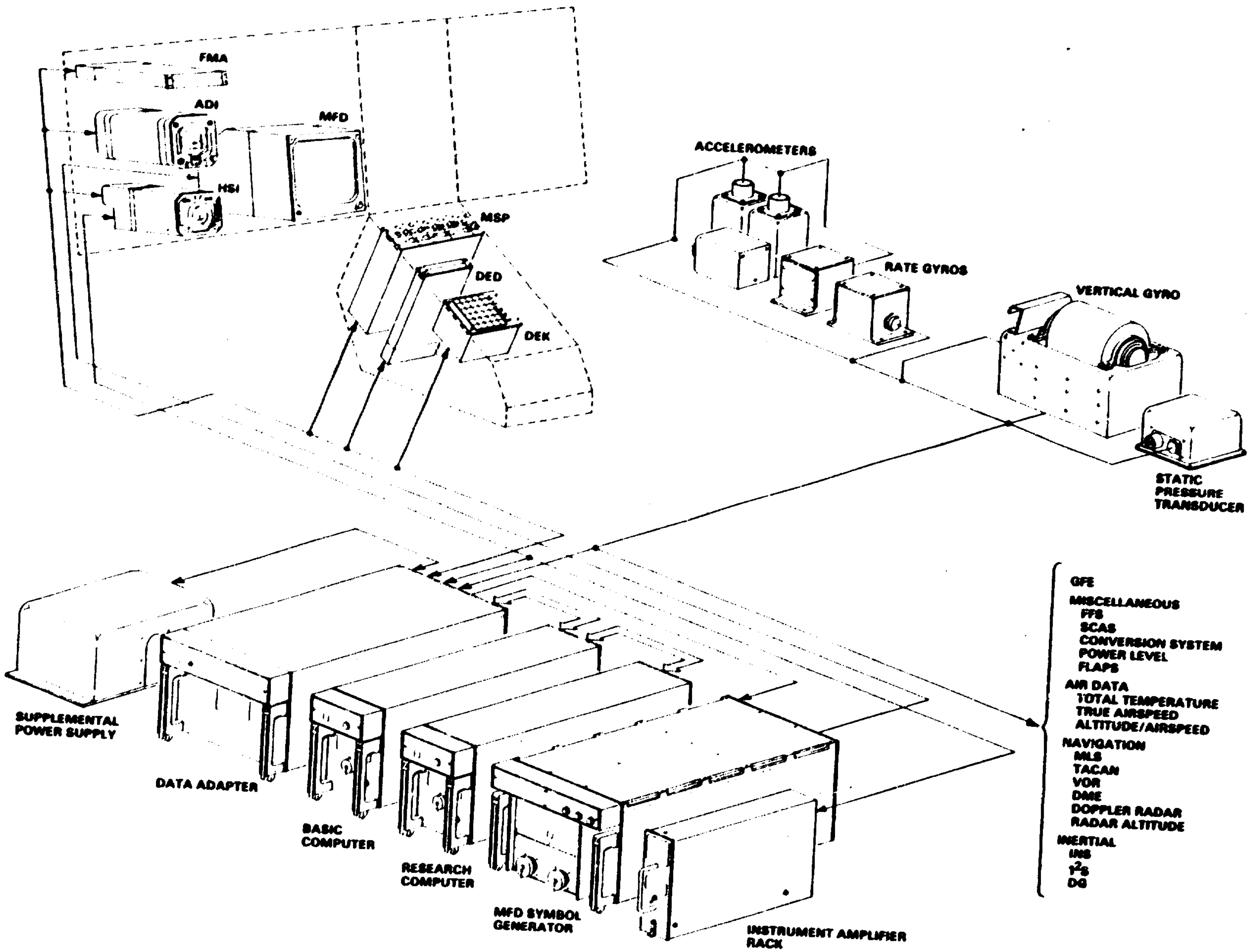


Figure 4-1
Component Configuration Diagram
of XV-15 V/STOLAND System

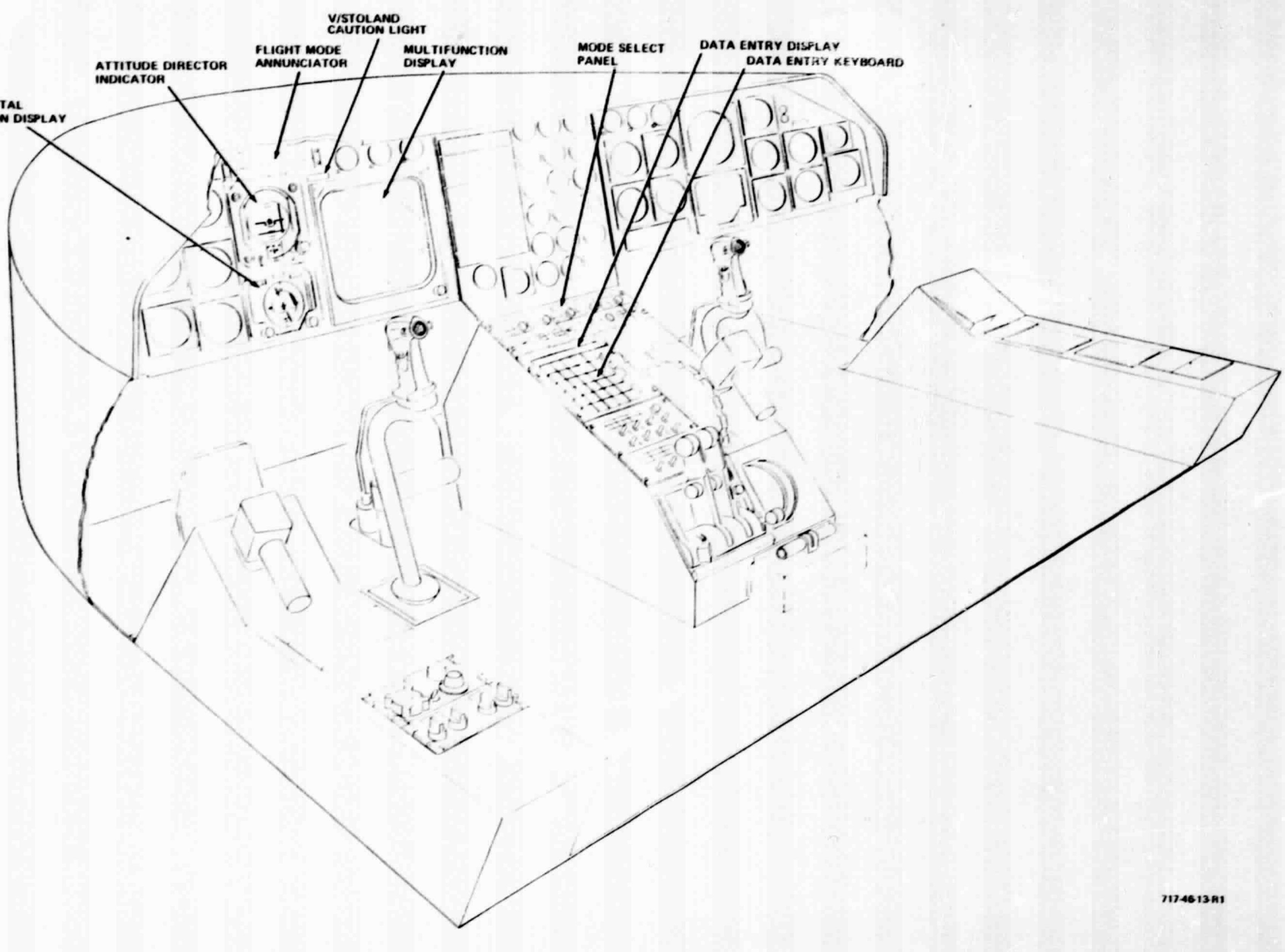


Figure 4-2
The XV-15 Cockpit with V/STOLAND Instruments

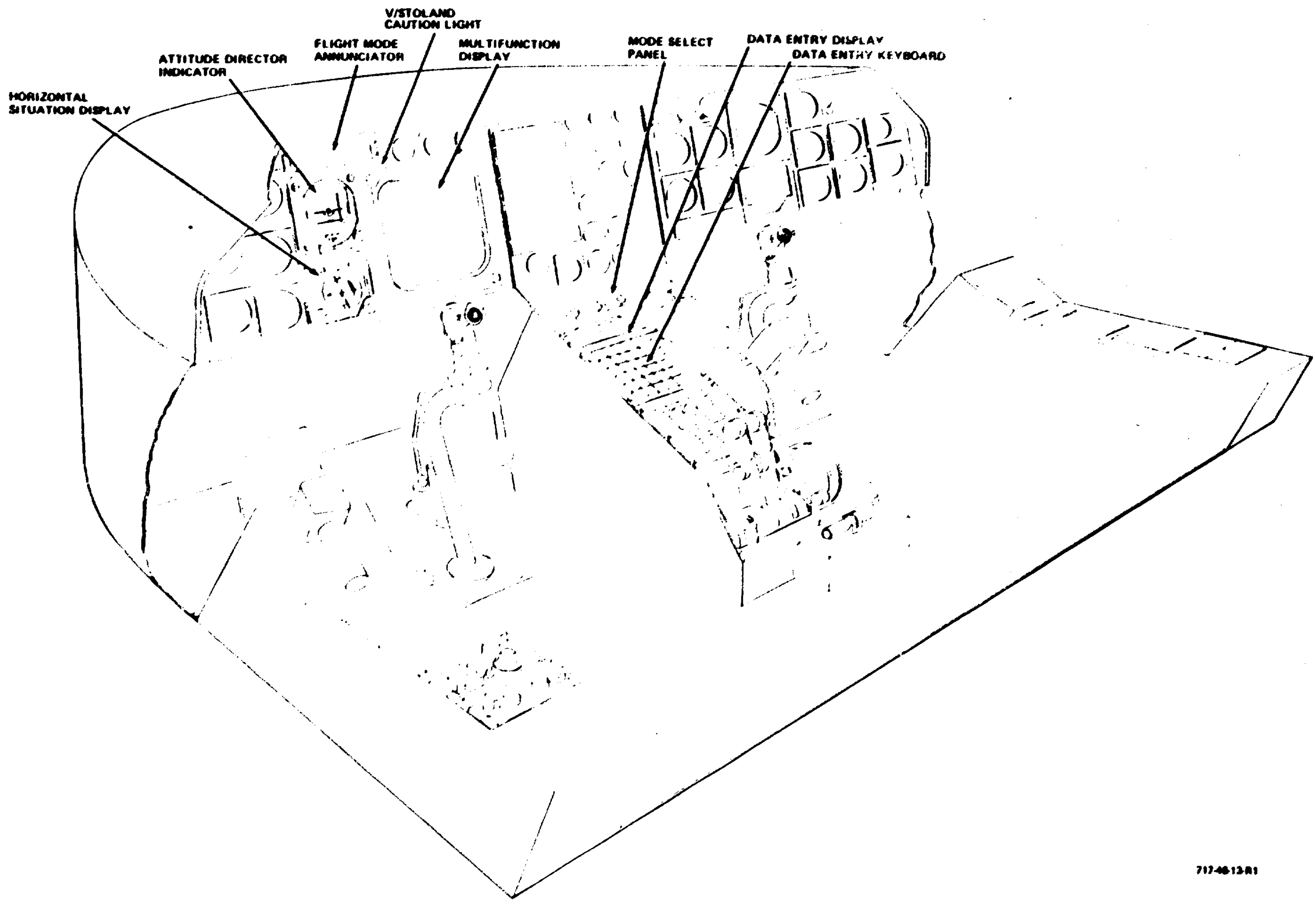
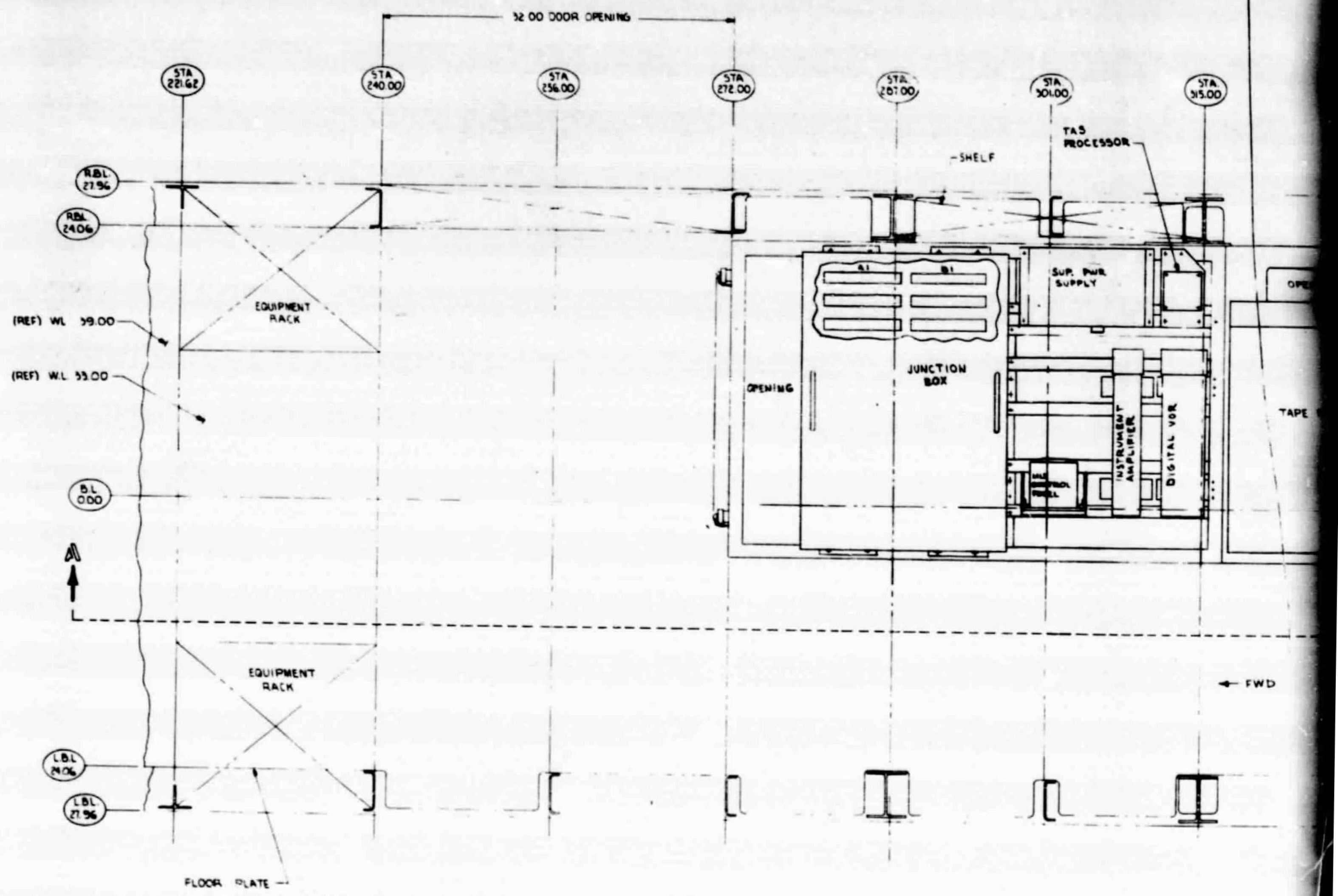
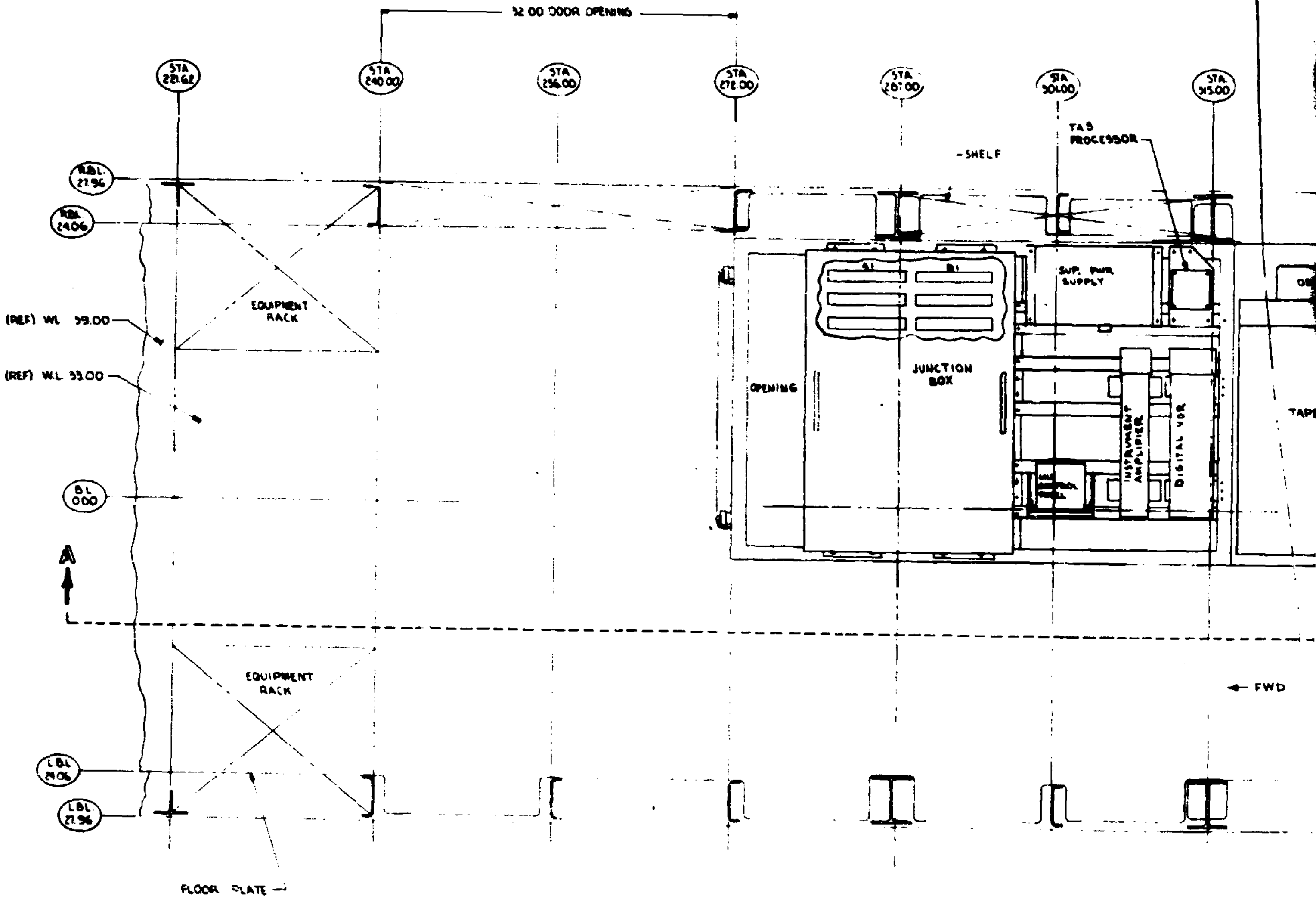


Figure 4-2
The XV-15 Cockpit with V/STOLAND Instruments



CARGO COMPARTMENT
PLAN VIEW
SCALE 1/4

FOLDOUT FRAME



CARGO COMPARTMENT
PLAN VIEW
SCALE 1/4

FOLDOUT FRAME

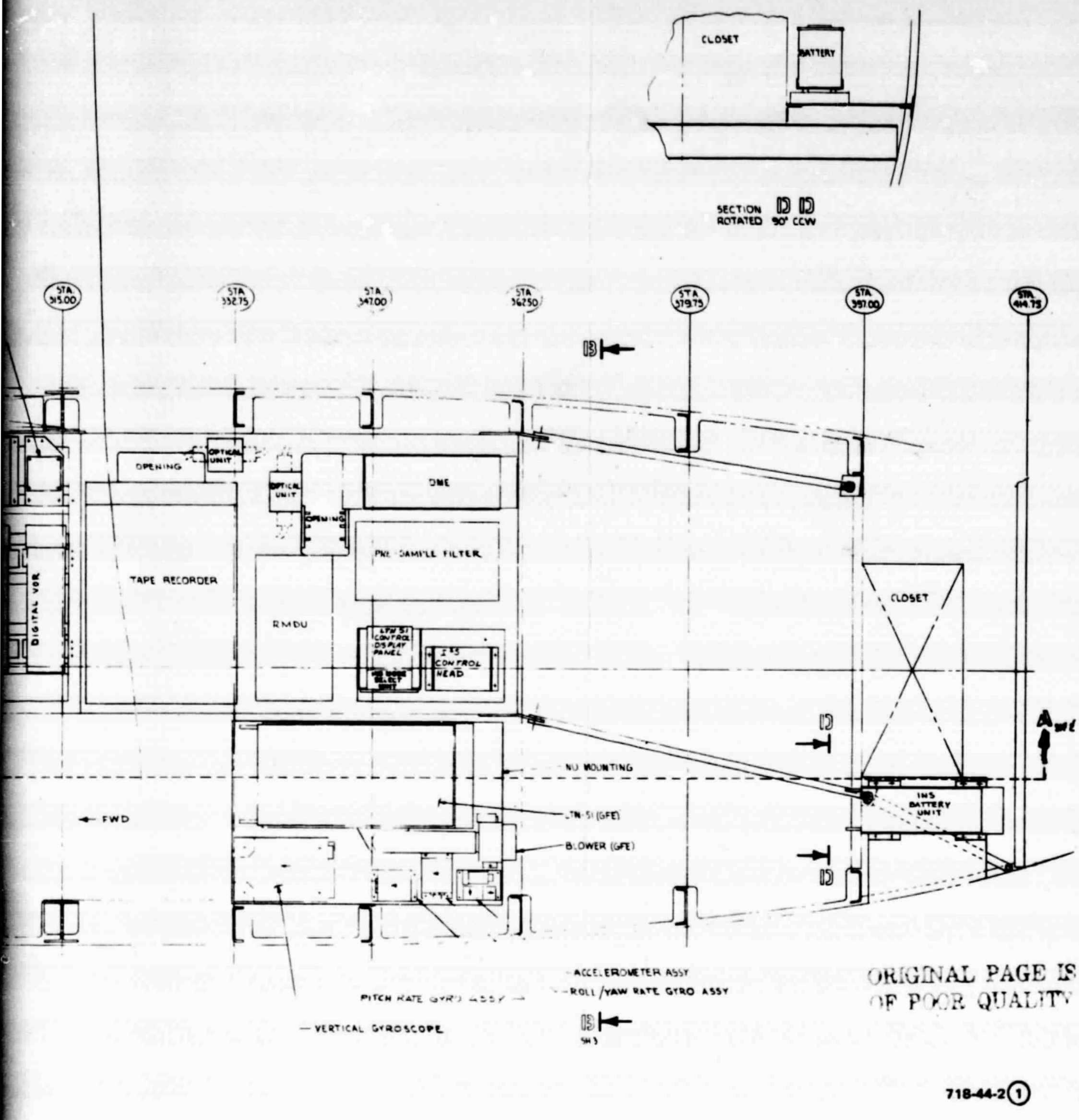


Figure 4-3
Plan Views of the Location of
XV-15 V/STOLAND Flight
Rack Components

FOLDOUT FRAME 2

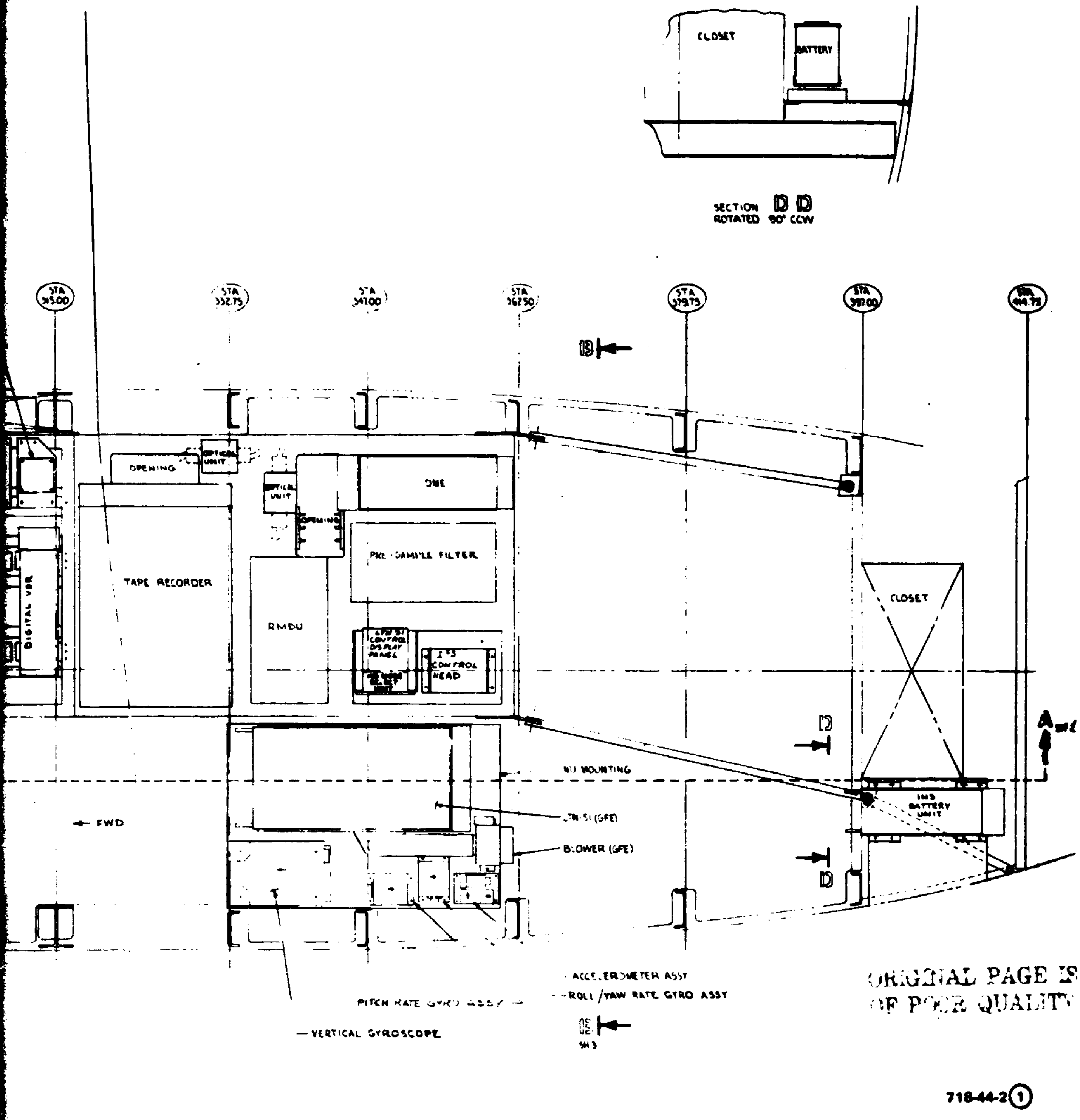
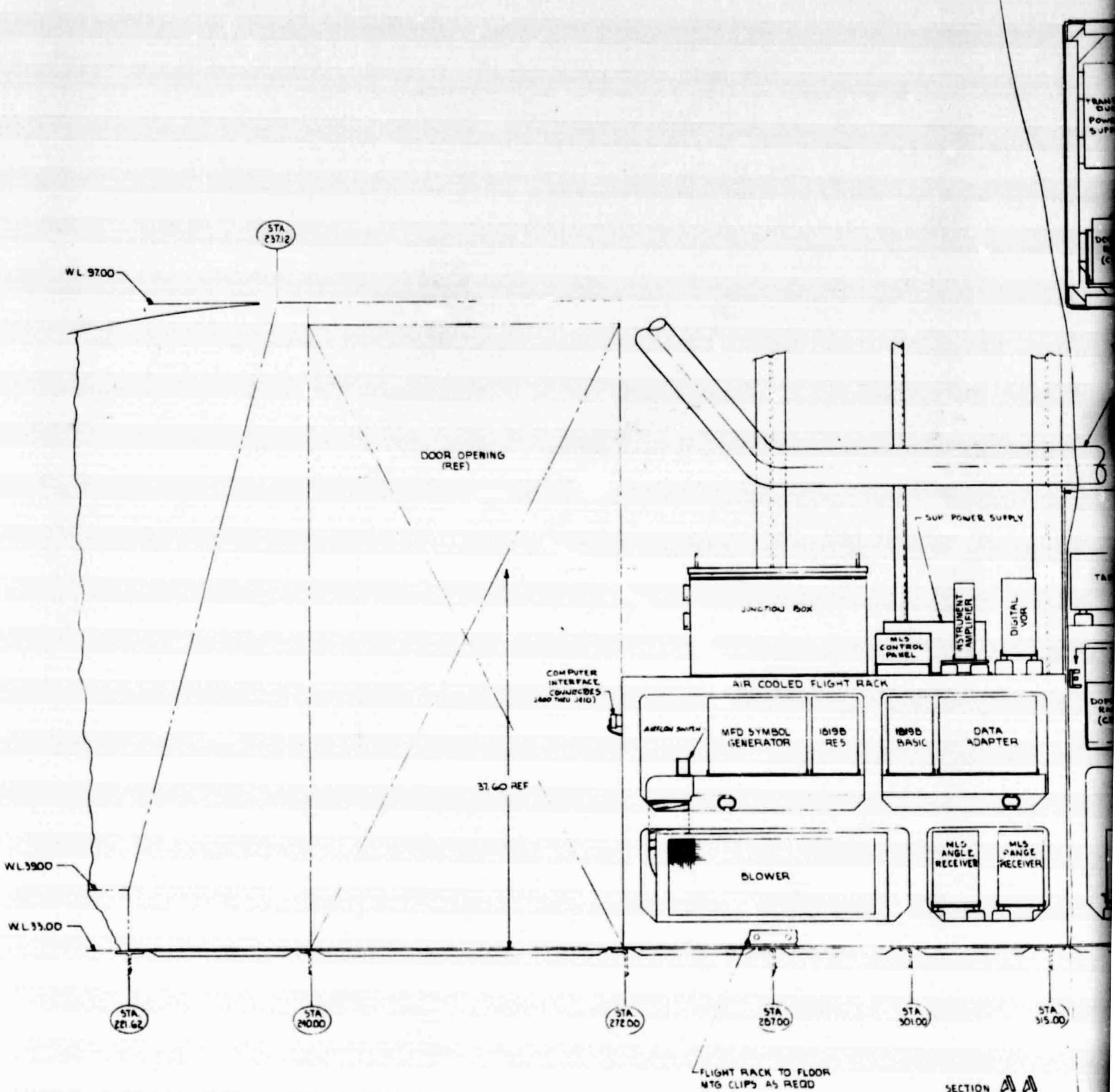
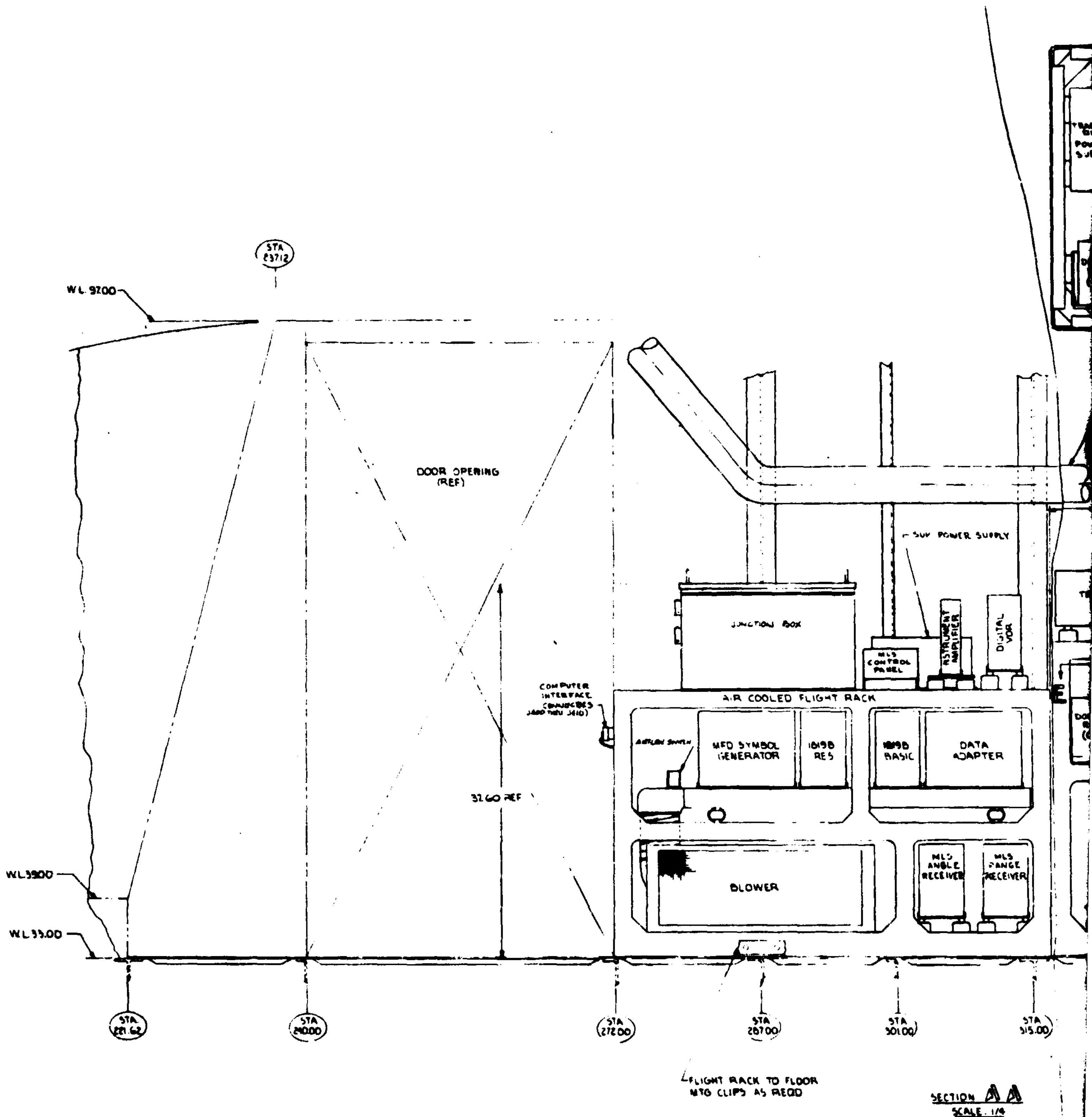


Figure 4-3
Plan Views of the Location of
XV-15 V/STOLAND Flight
Rack Components

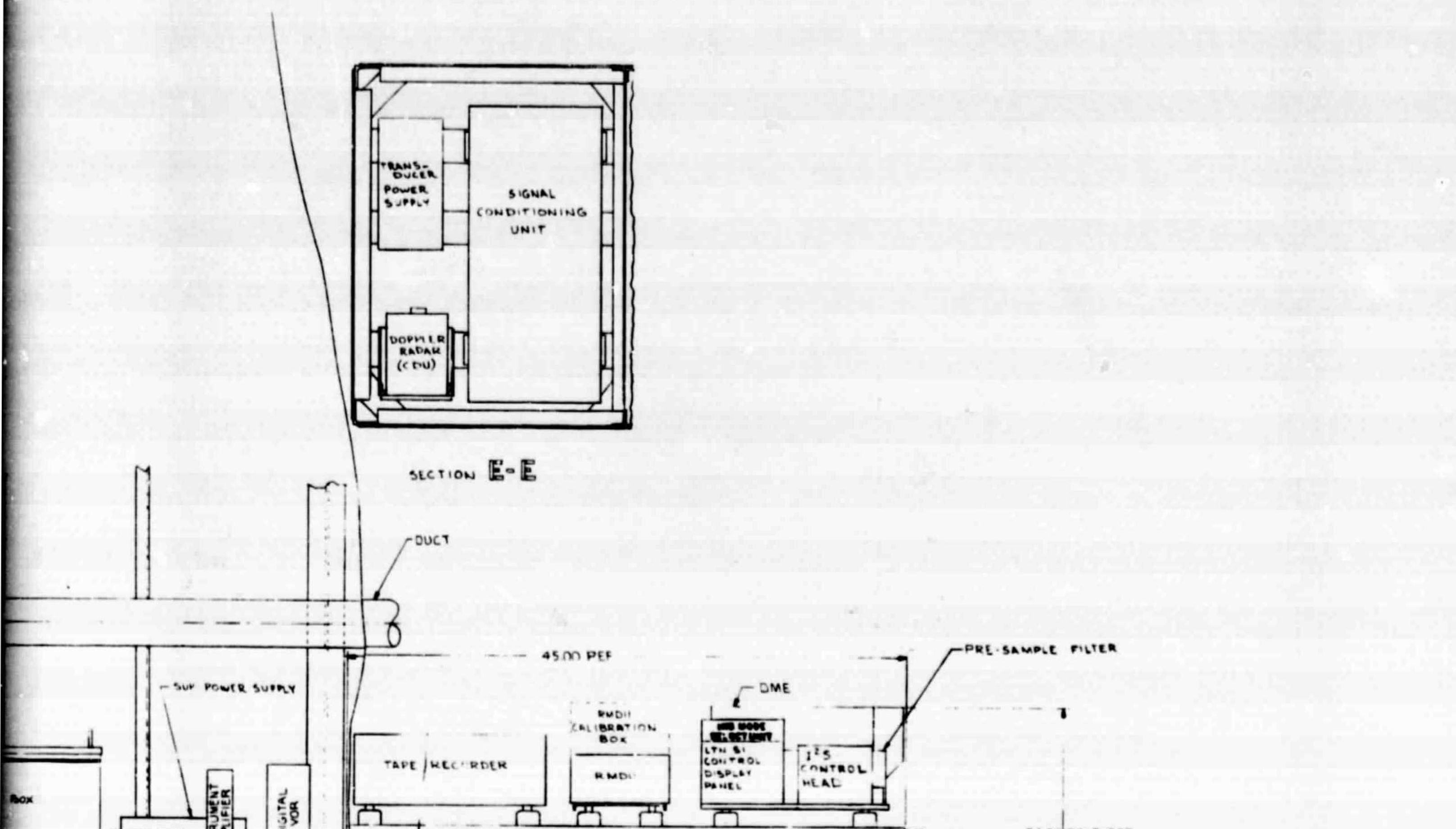


SECTION A A
SCALE 1/4

FOLDOUT FRAME



FOLDOUT FRAME



FOLDOUT FRAME 2

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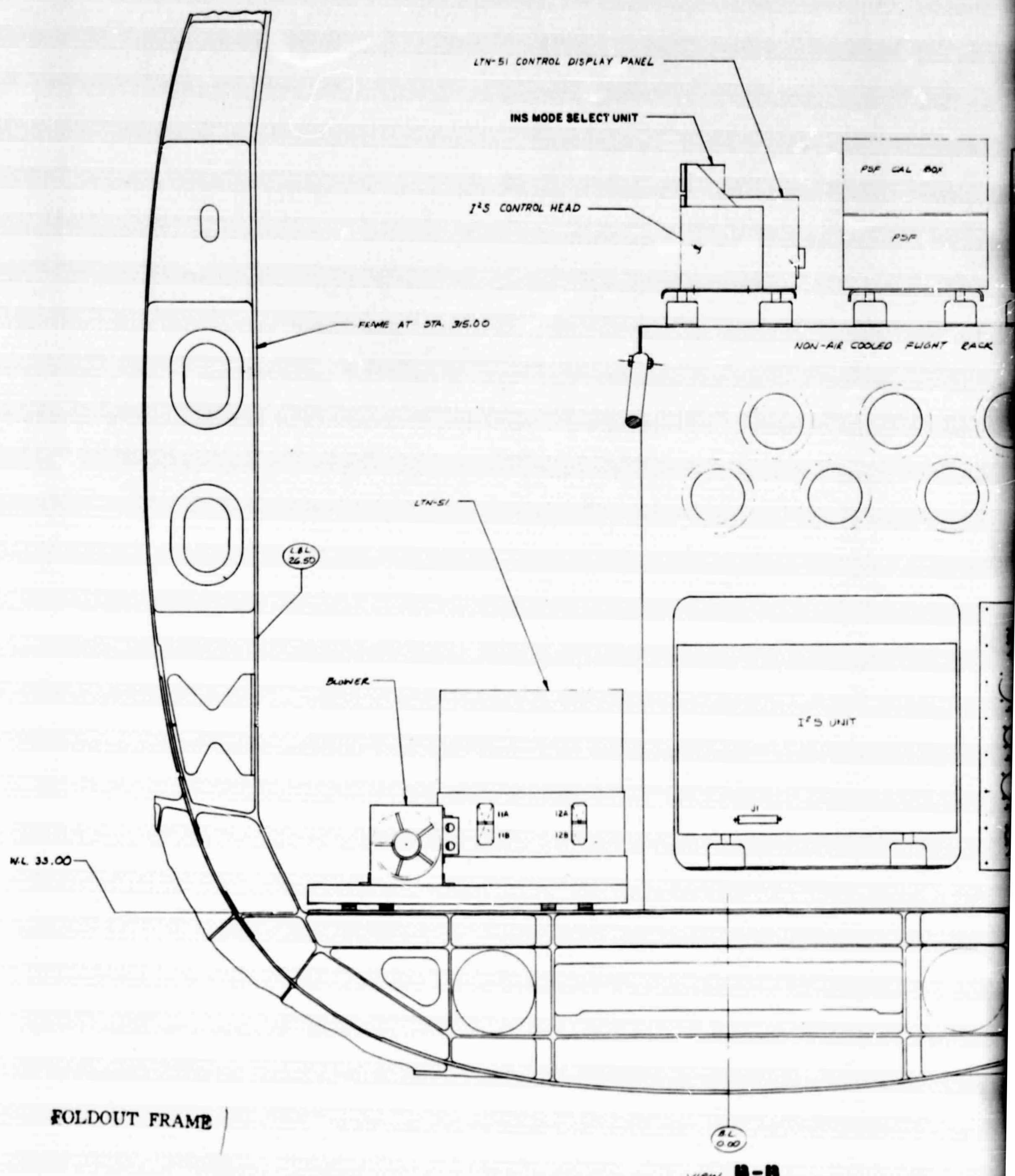
Figure 4-4
Plan Views of the Location of XV-15 V/STOLAND Flight Rack Components



FOLDOUT FRAME

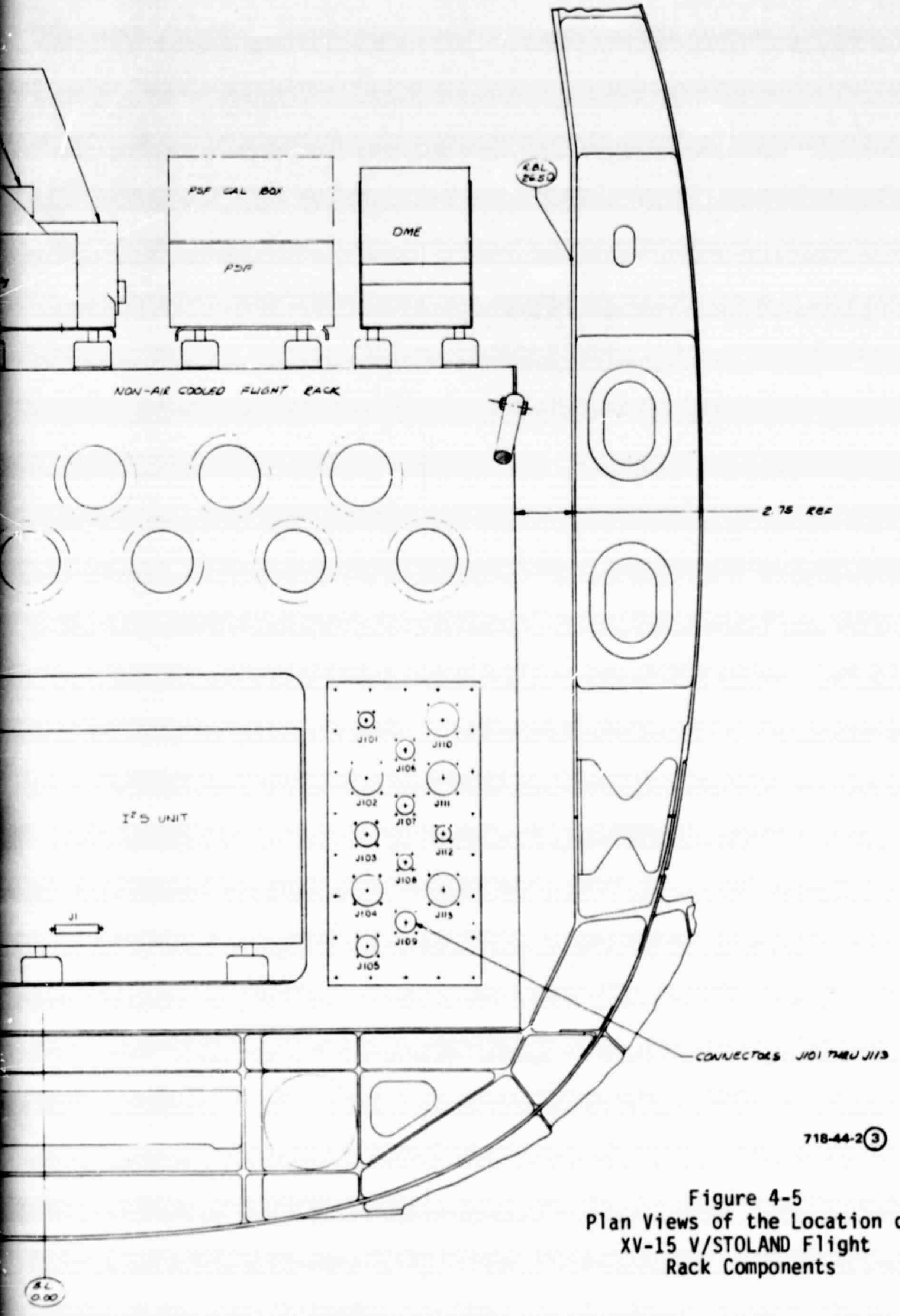


Figure 4-5
Plan Views of the Location of
XV-15 V/STOLAND Flight
Rack Components

FOLDOUT FRAME 2

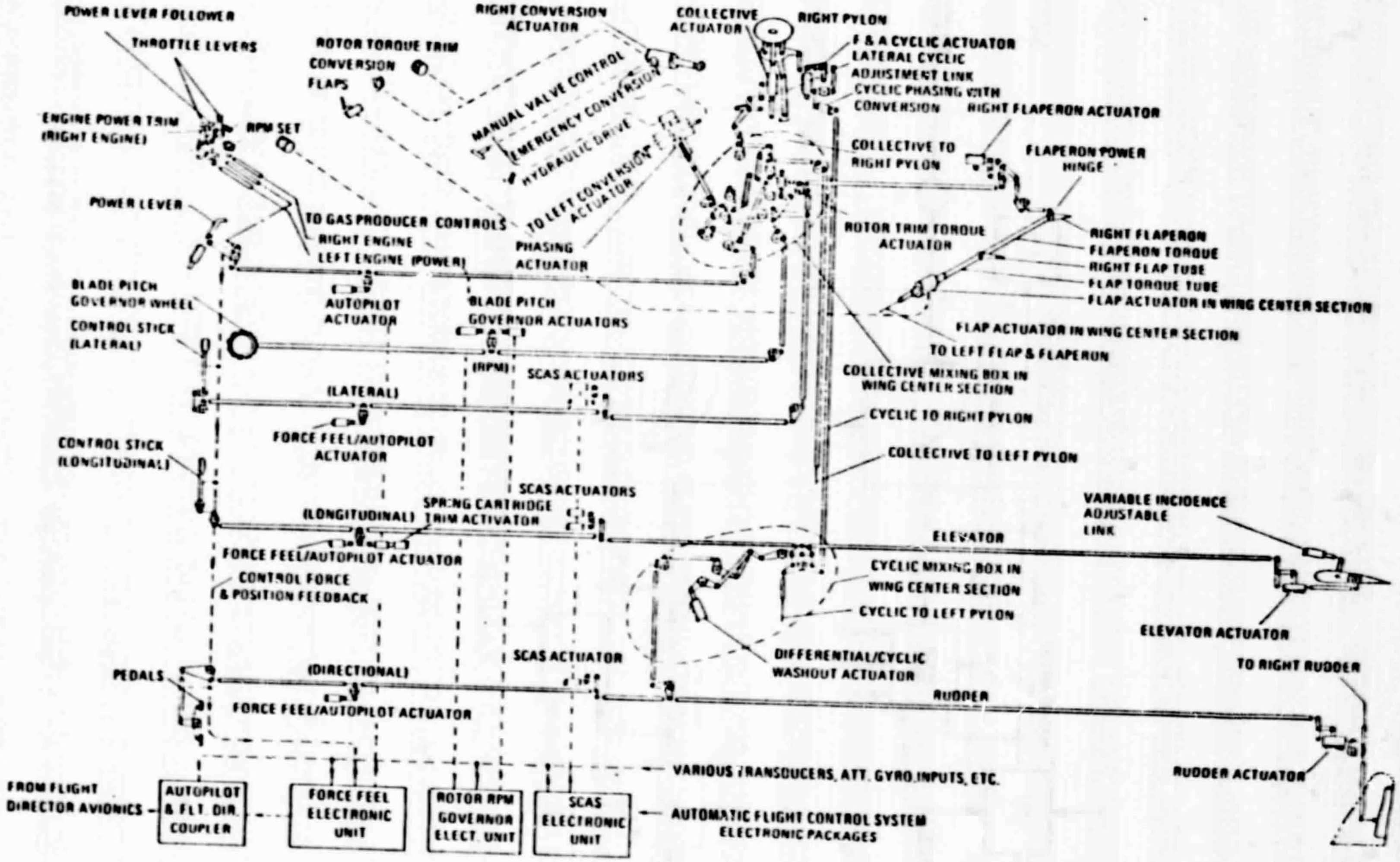


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Figure 4-6
Flight Control Mechanical Schematic

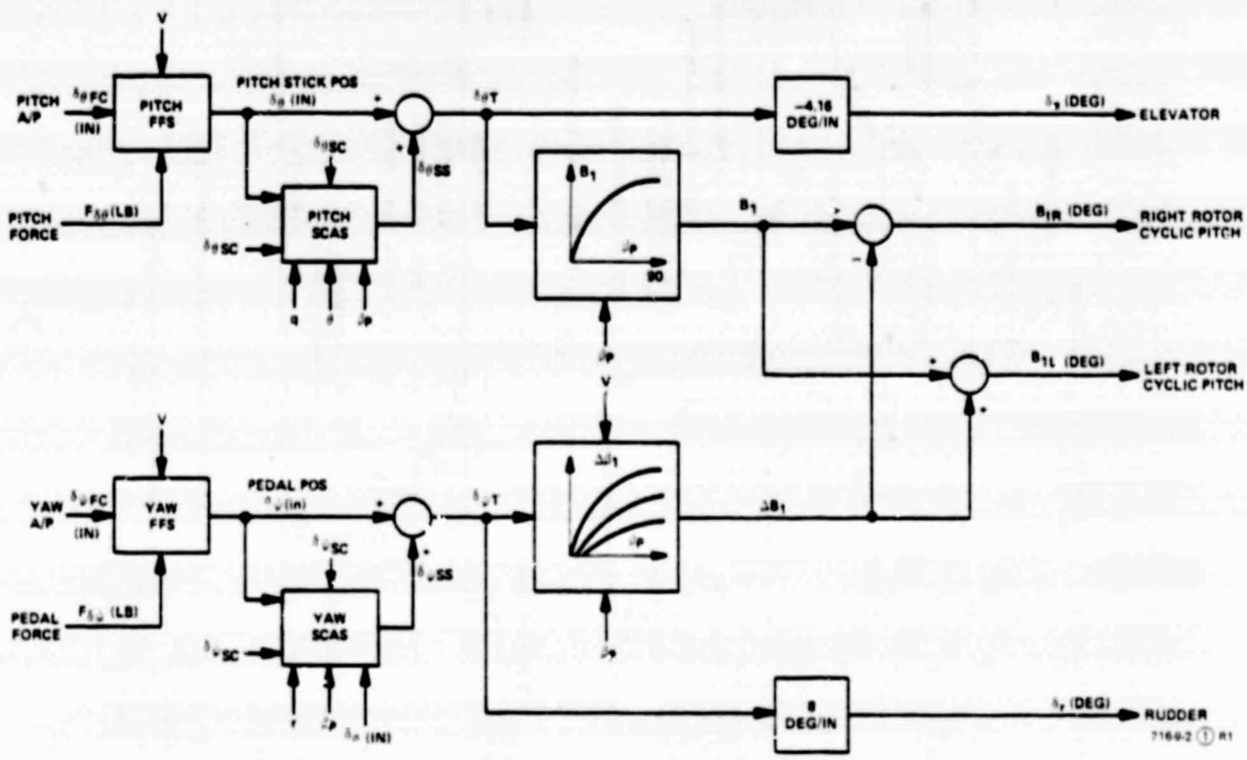


Figure 4-7
XV-15 Flight Control System Block Diagram

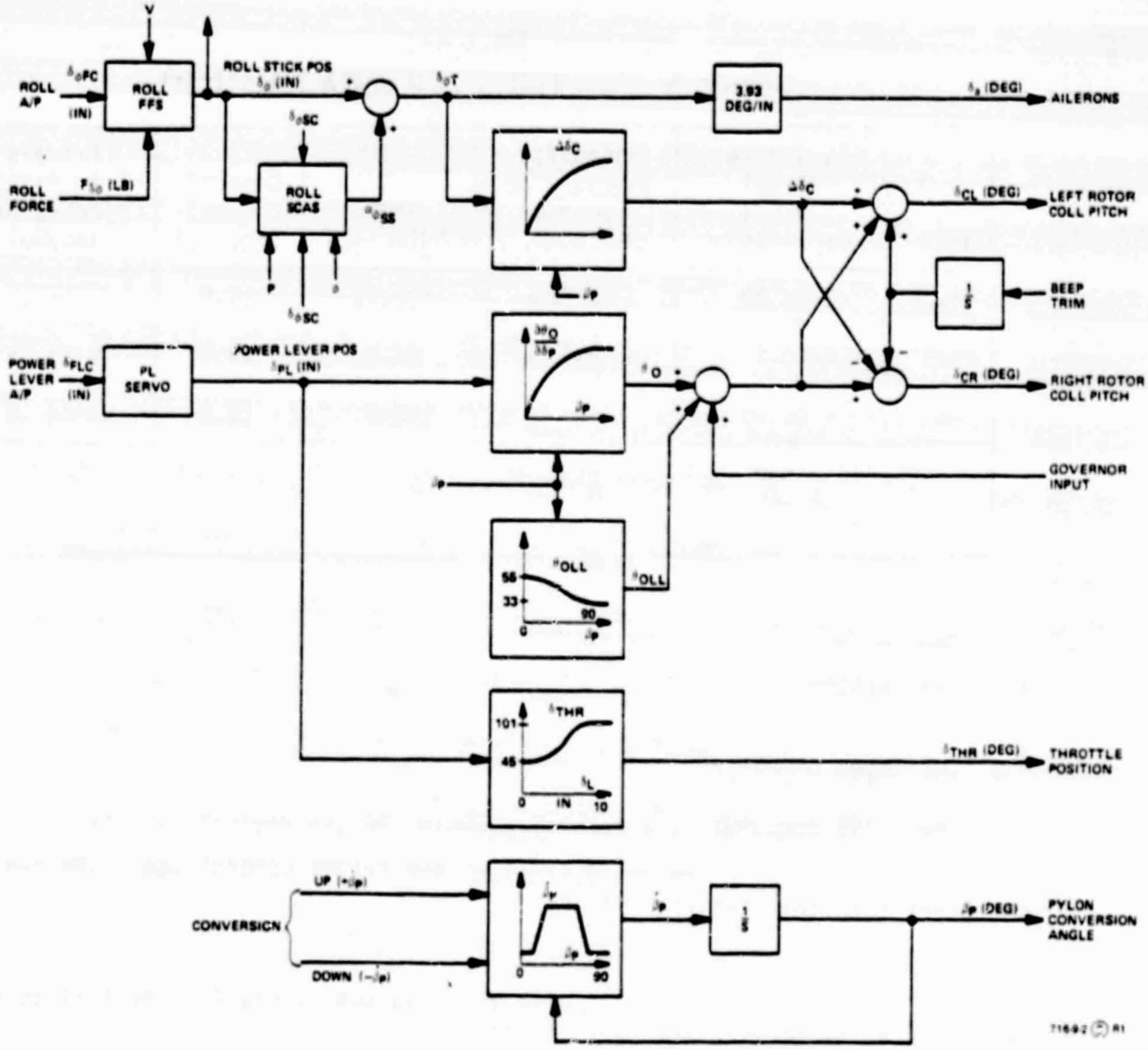


Figure 4-8
XV-15 Flight Control System Block Diagram

Block diagrams for the pitch, roll, and yaw FFS systems are given in Figures 4-9 through 4-11, which include equations for the scheduled gains as functions of airspeed. Table 4-1 summarizes the FFS characteristics. Block diagrams for the pitch, roll, and yaw SCAS systems are given in Figures 4-12 through 4-14.

TABLE 4-1

SUMMARY OF FORCE-FEEL SYSTEM CHARACTERISTICS**

Axis	FFS Gradient Range* (pounds per inch)	Mechanical Spring Grad. (pounds per inch)	Servo Authority	Control Travel (inches)	Primary Trim Rate* (inches per second)
Pitch	2 to 20	11.25	±15.75	±4.8	.1 to .5
Roll	1 to 5	3.4	±4.8	±4.8	.1 to .5
Yaw	7 to 56	---	±45.0	±2.5	.1 to .5

*Airspeed range (0 to 300 kts)

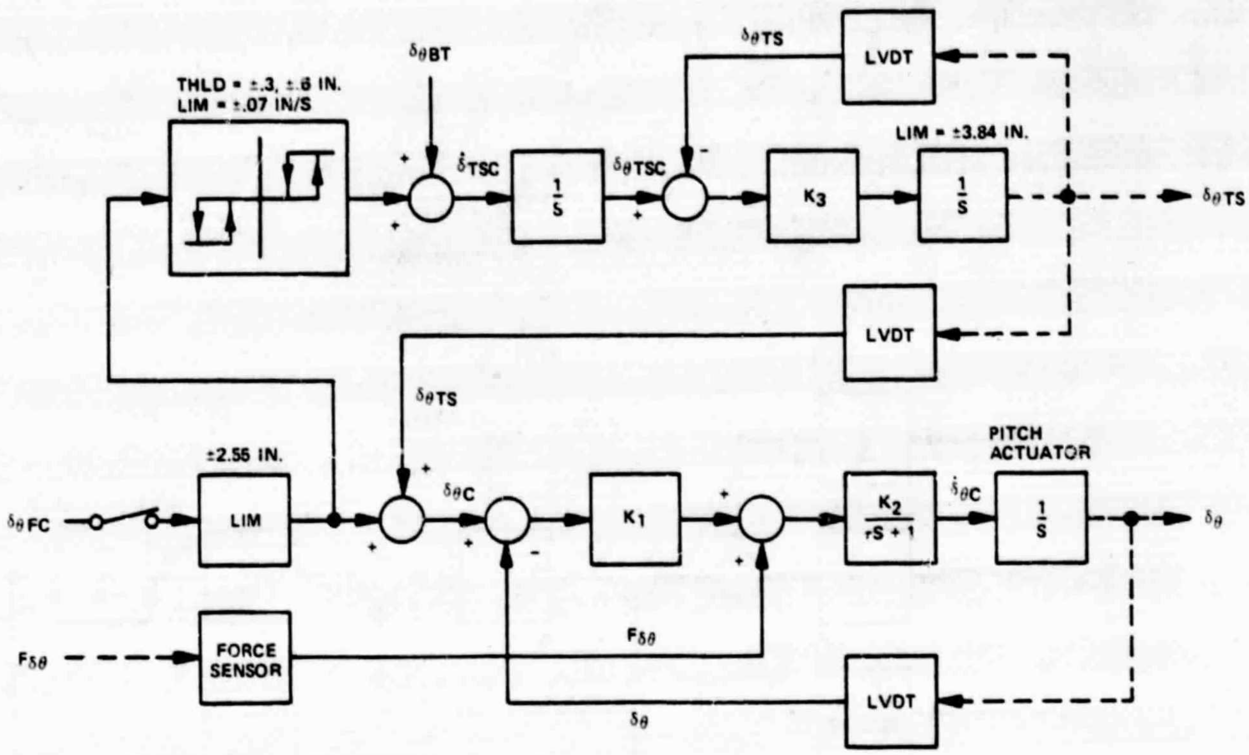
**Referred to top of stick

The following paragraphs describe the major hardware elements of the V/STOLAND system.

4.3 THE 1819B COMPUTERS

The 1819B computer is a general-purpose, 18-bit digital computer designed expressly for airborne real-time avionics and flight control use. The two such computers supplied with the V/STOLAND system are identical and interchangeable.

An ARINC long, one-half ATR case with two dual 106-pin connectors at the rear is used to package the basic 1819B Computer as illustrated in Figure 4-15. A plenum chamber under the computer supplies forced-air cooling; exhausted air is vented throuh the front panel housing. Interconnection for the control panel illustrated in Figure 4-16 is made through a front-mounted circular connector that is accessible when the computer is mounted in the in-service configuration.



$K_1 = 20\ K_p$ LB/IN.

$K_2 = \frac{1}{1.4\sqrt{K_p}}$ (IN/S/LB

$\tau = \frac{1}{49.8\sqrt{K_p}}$ S

$K_p = \frac{1}{1 + 1.69S}\left(\frac{.5 + q_F}{5}\right)$

$q_c = \frac{1}{2}\rho_o\ V_c^2$ (IN. HG)

$K_3 = 10.4$ (IN/S)/IN

$$\frac{\delta_\theta}{\delta_{\theta C}} \approx \frac{1}{\frac{S}{K_1K_2} + 1} = \frac{1}{\left(\frac{.07}{\sqrt{K_p}}\right) S + 1}$$

$$\frac{\delta_\theta}{F_{\delta\theta}} \approx \frac{1}{K_1}\ \frac{1}{\left(\frac{.07}{\sqrt{K_p}}\right) S + 1} \approx \frac{1}{K_1}$$

$$\frac{F_{\delta\theta}}{\delta_\theta} \approx K_1 = \frac{1}{1 + 1.69S}\ (2 + 4q_c)$$

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Figure 4-9
Pitch FFS

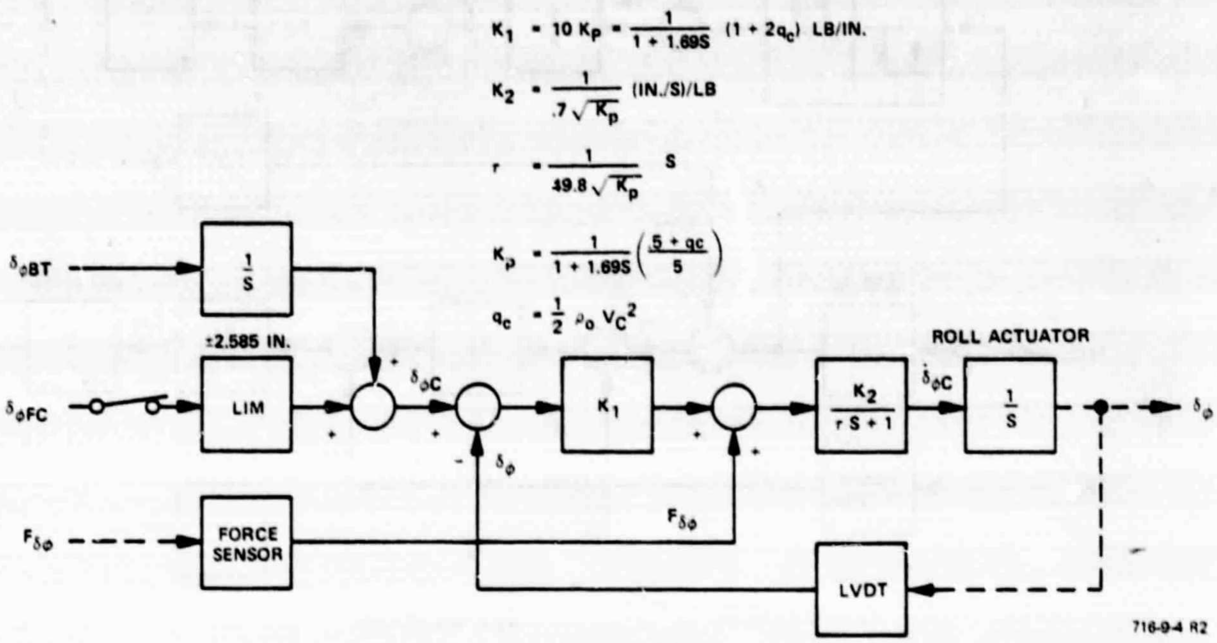


Figure 4-10
Roll FFS

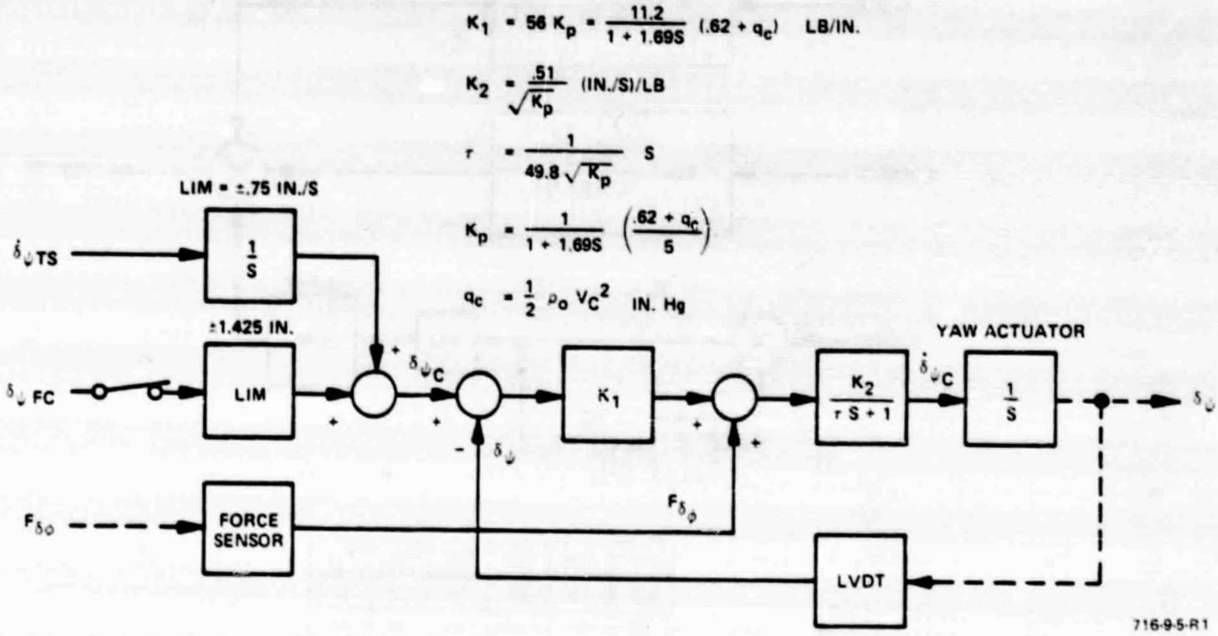
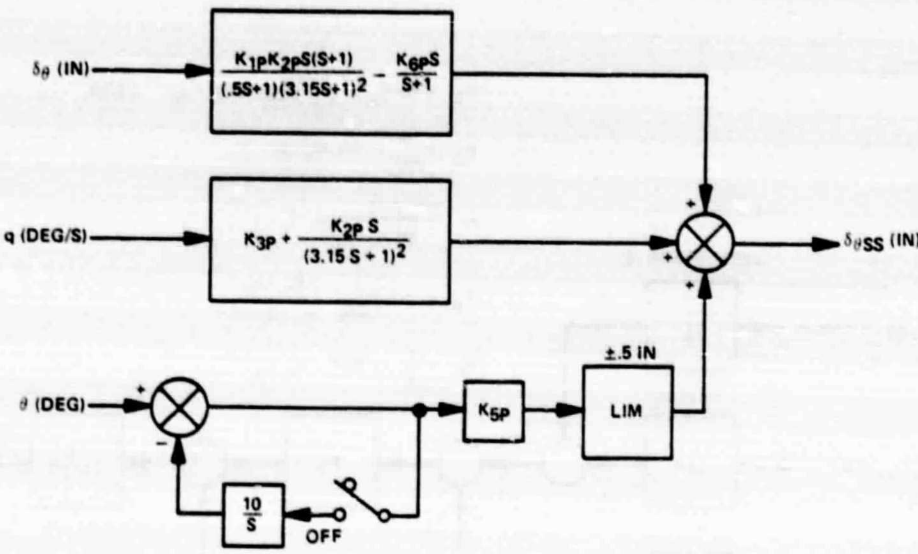


Figure 4-11
Yaw FFS



β_P	K_{1P}	K_{2P}	K_{3P}	K_{5P}	K_{6P}
90°	7.5	.47	.10	.20	0
0°	4.5	1.105	.06	.10	.60

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Figure 4-12
Pitch SCAS Block Diag:am

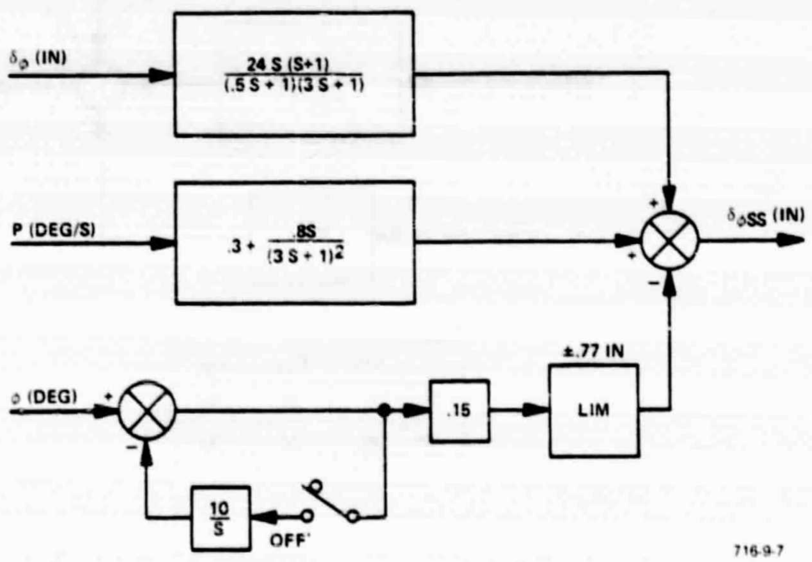
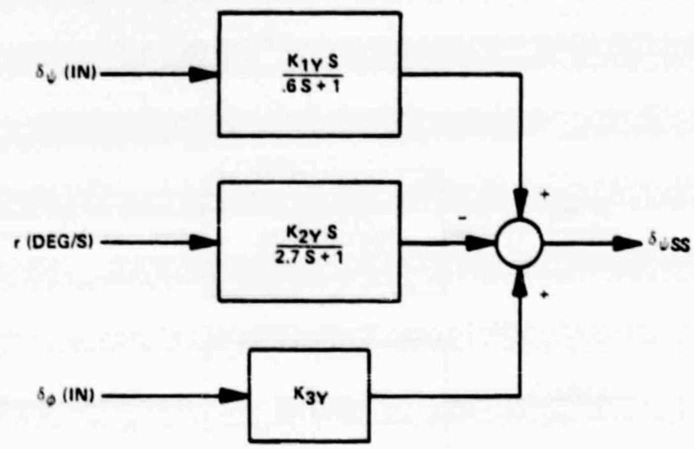


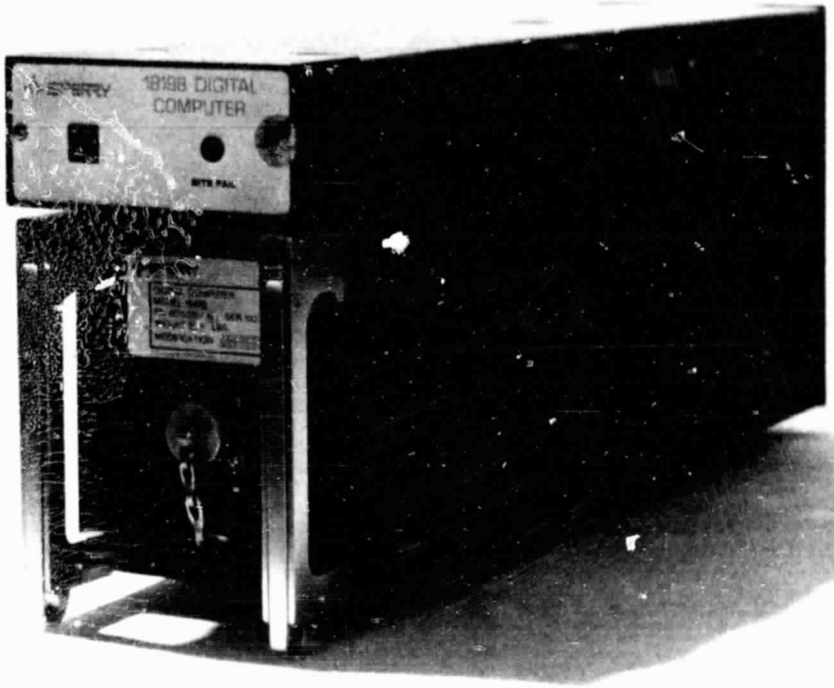
Figure 4-13
Roll SCAS Block Diagram



β_P	K_{1Y}	K_{2Y}	K_{3Y}
0°	0	.07	.20
90°	.90	.30	.80

716-9-8

Figure 4-14
Yaw SCAS Block Diagram



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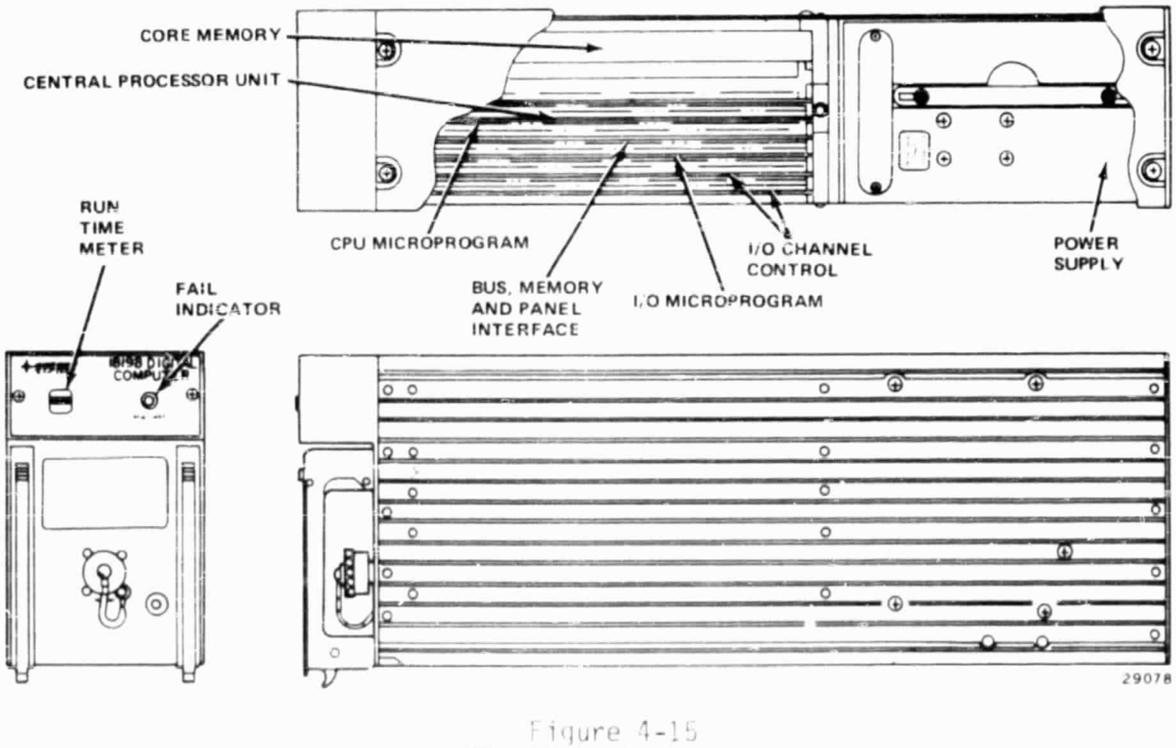

Figure 4-15
The 1819B Computer

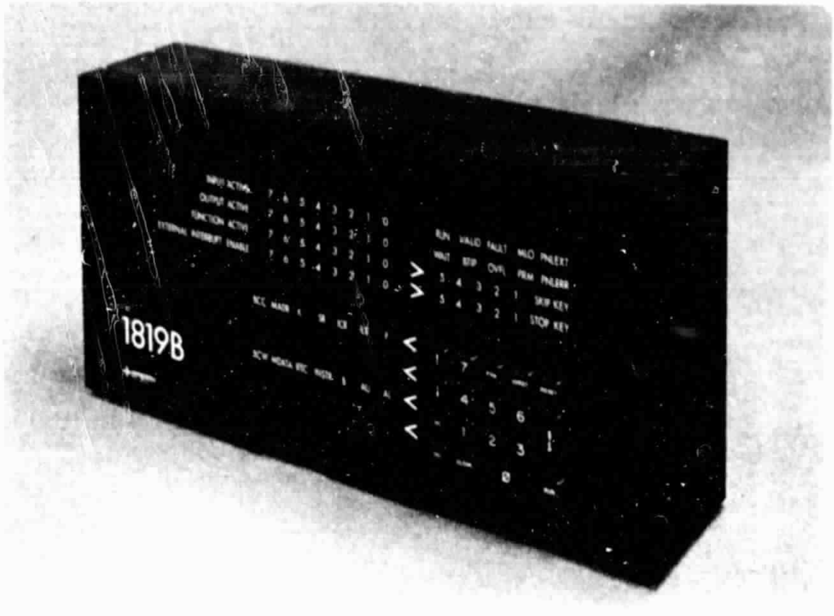

Figure 4-16
1819B Control Panel

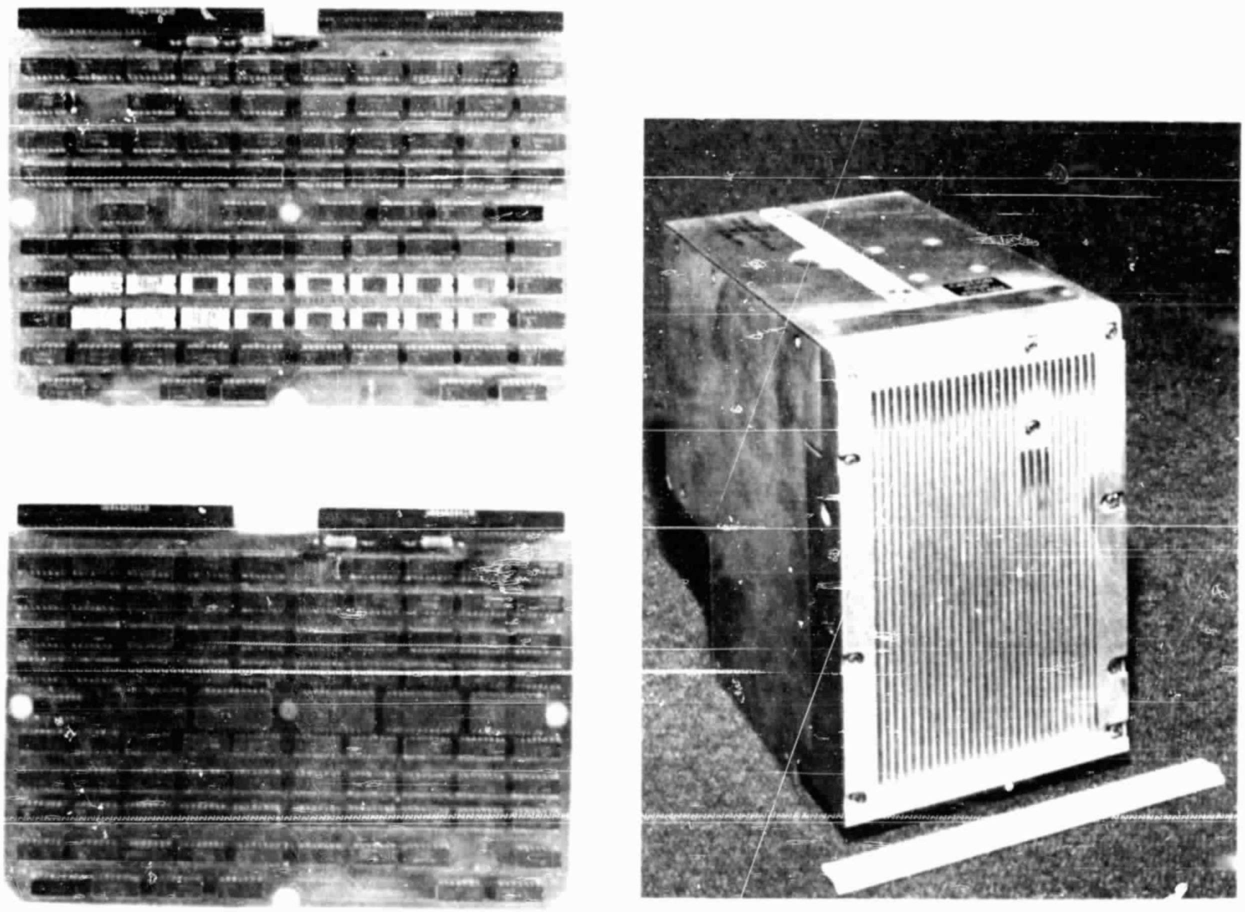

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Figure 4-17
1819B Subassemblies

Also located on the front housing are a failure indicator to indicate the status of the computer after conducting a built-in test, and a run-time meter to maintain a chronological record of the computer usage. A chassis grounding point is provided on the front for attachment of a bonding strap.

All removable components are securely mounted to eliminate vibration. Printed Wiring Boards (PWBs) are held by card guides at each end, and are secured by the top cover. These cards plug into a wire-wrapped PWB at the bottom of the chassis. This same PWB contains all memory and power supply interconnections. The top and bottom covers are secured with cross-head screws that are easily removed for service.

The electronics for the CPU and the I/O are contained on six 6-1/2 inch by 10 inch, ten-layer printed wiring boards as illustrated in Figure 4-17. The functions of the six cards are:

(1) CPU microprogram controller and associated circuitry

(2) Central processing unit

(3) Bus control, panel interface, 1K semiconductor ROM, power sequence, and system clocks

(4) I/O microprogram controller

(5),(6) Identical PWBs containing the odd and even numbered I/O channel controls with channel-oriented circuitry.

Both boards (5) and (6) must be installed to allow 36-bit I/O data transfers. Each board is keyed to prevent installation in the incorrect position. The Power Supply Unit, also illustrated in Figure 4-17, converts primary aircraft power to regulated dc voltages of +15 volts, ±5 volts, and -12 volts.

The memory module is an Electronic Memories SEMS-9 planar memory system configured for 16,384 18-bit words of magnetic core storage. The memory system has an access time of 420 nanoseconds and a full cycle time of 1.2 microseconds. A coincident-current (3D), three-wire organization is used in the memory. The maximum power required is 94 watts, with nominal power dissipation of 42 watts in a "half-zero" pattern at 50 percent duty cycle.

Three semiconductor read-only memories are used in the computer for transfer and control of data and instructions. Within the main memory, a 1024 by 18-bit ROM is used for macrcinstructions which typically include bootstraps, loaders, and self-test programs. The Central Procesor Unit (CPU) employs a 512 by 52-bit ROM for microinstructions. A 256 by 40-bit ROM contains the I/O section microinstructions.

The 1819B control panel allows the programmer or maintenance operator full access to the CPU and I/O registers and memory. This panel is configured to allow display of all I/O active states, pertinent information about the status of the I/O and processor, and also allows setting the sense and stop keys. Using this panel, a programmer may examine, load, and control all CPU states and control execution of any program.

Table 4-2 summarizes the principal characteristics of the 1819B computer.

TABLE 4-2
1819B CHARACTERISTICS

Parameter	Characteristic
Word Length	18 bits (full set of 36-bit data instructions and 36-bit I/O).
Instruction Set	150 instructions including I/O.
Arithmetic	Parallel, binary, fixed point, one's complement.
Memory	16,384 18-bit words of core memory, expandable in 16,384-word increments to 65,536 words. 512 18-bit words of semiconductor read-only memory expandable to 1024 words.
Registers	Two 18-bit accumulators which may be linked to form a 36-bit accumulator. Eight 18-bit index registers that have limited accumulator functions.
Addressing	Page size of 4096 words for direct addressing; 65,536 words for indirect addressing.

TABLE 4-2 (cont)

1819B CHARACTERISTICS

Parameter	Characteristic
Typical Execution Speeds	• Register-to-Register Addition - 1.6 microseconds • Memory-to-Register Addition - 2.4 microseconds • Double Precision Memory-to-Register Addition - 4.8 microseconds • Memory times register multiplication - 7.4 microseconds
Operating Speed	100 nanosecond microcycle time; 1.2 microsecond memory cycle time.
Input/Output and Control	• 8 independent I/O channels • 417 kHz I/O transfer rate • External, maskable interrupts, 1 per I/O channel • Internal buffer termination interrupts, 1 for input and 1 for output on each channel • 18 or 36 bits available on each channel • Data transfer without interrupting processor • Entire I/O removable in 4-channel increments
Real-Time Clock	Automatic internal clock generates 1000 counts per second - accurate to 1 count in 10 seconds. Interrupt rate under software control can be varied from 1 to 2^{16} counts in one count increments.
Interrupts	• Power fail • Fault • Overflow • Real-Time Clock • 8 External • 8 Input buffer termination • 8 Output buffer termination

TABLE 4-2 (cont)
1819B CHARACTERISTICS

Parameter	Characteristic
Interrupt Priority	Each interrupt has assigned priority, and all but power fail and fault may be masked out.
Input Power	200 V line-to-line, 3-phase, 400-Hz per MIL-STD-704 for Category B equipment. 150 watts nominal
Temperature	• Operating - Normal Operating - 100°F; Severe Operating - 130°F; Intermittent, 30 minutes - 160°F • Storage - -50°F to 185°F • Thermal Control - Direct forced-air cooling (1.76 pounds per minute)
Environmental (MIL-E-5400 Class 2)	• Altitude - 20,000 feet • Vibration - 10 - 40 Hz: +6 dB/octave; 40 - 250 Hz; $.02G^2$ Hz; 250 - 2000 Hz; -3 dB/octave • Shock - 6g - 3 shocks along 6 directions; crash safety: 15g • Humidity - Category A, 48 hours • EMI - Per MIL-STD-467A
Dimensions	4.9 in. X 7.6 in X 19.5 in (124mm X 194mm X 495mm) (half ATR long)
Weight	25 pounds (11.5 Kg)

4.4 THE DATA ADAPTER

The Data Adapter, illustrated in Figure 4-18, is a multipurpose unit that provides interfacing between the Basic digital computer and the other airborne equipment. It controls the information transfer between the different subsystems, sensors, displays, and the computer, and also provides the necessary signal conditioning for all associated equipment. The interface between the computer and the Data Adapter is a fast, parallel, Direct Memory Access (DMA) transmission system. The Data Adapter provides multiplexed A/D, D/A and digital-to-digital conversions, and lends itself functionally to division into six subsystems, as illustrated in the functional block diagram in Figure 4-19.

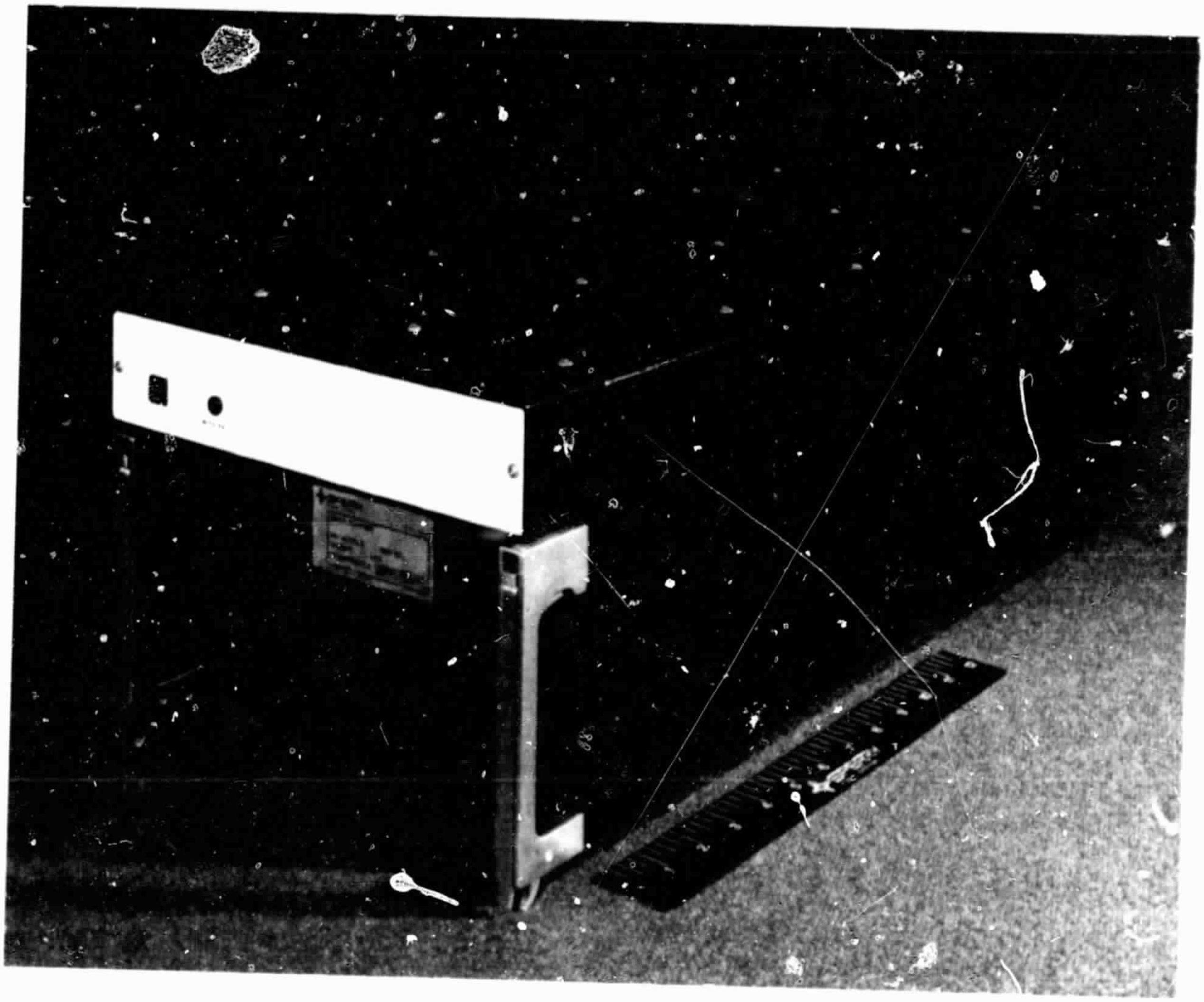

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Figure 4-18
Data Adapter

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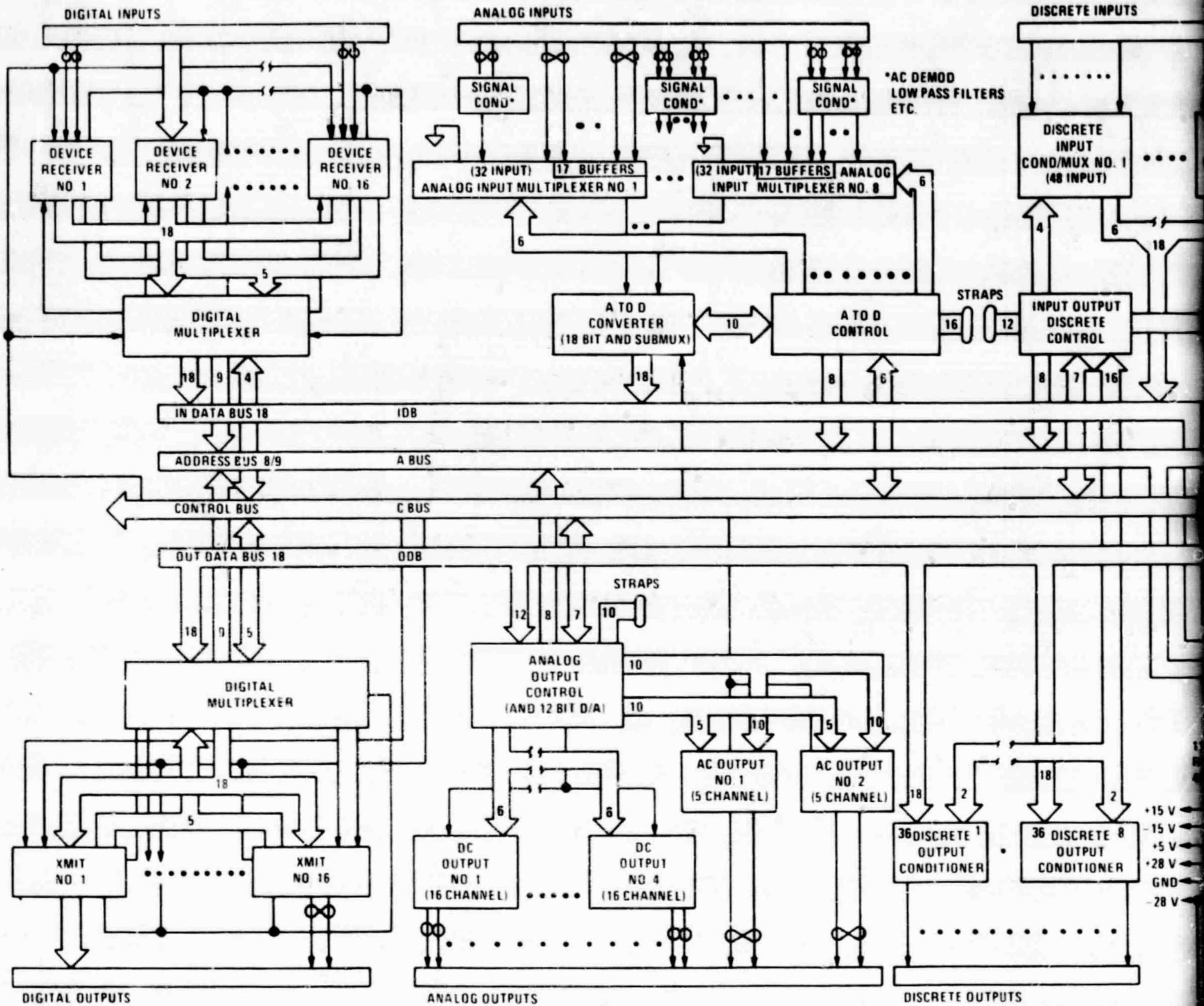
OF POOR QUALITY



Functiona
V/STOLAND

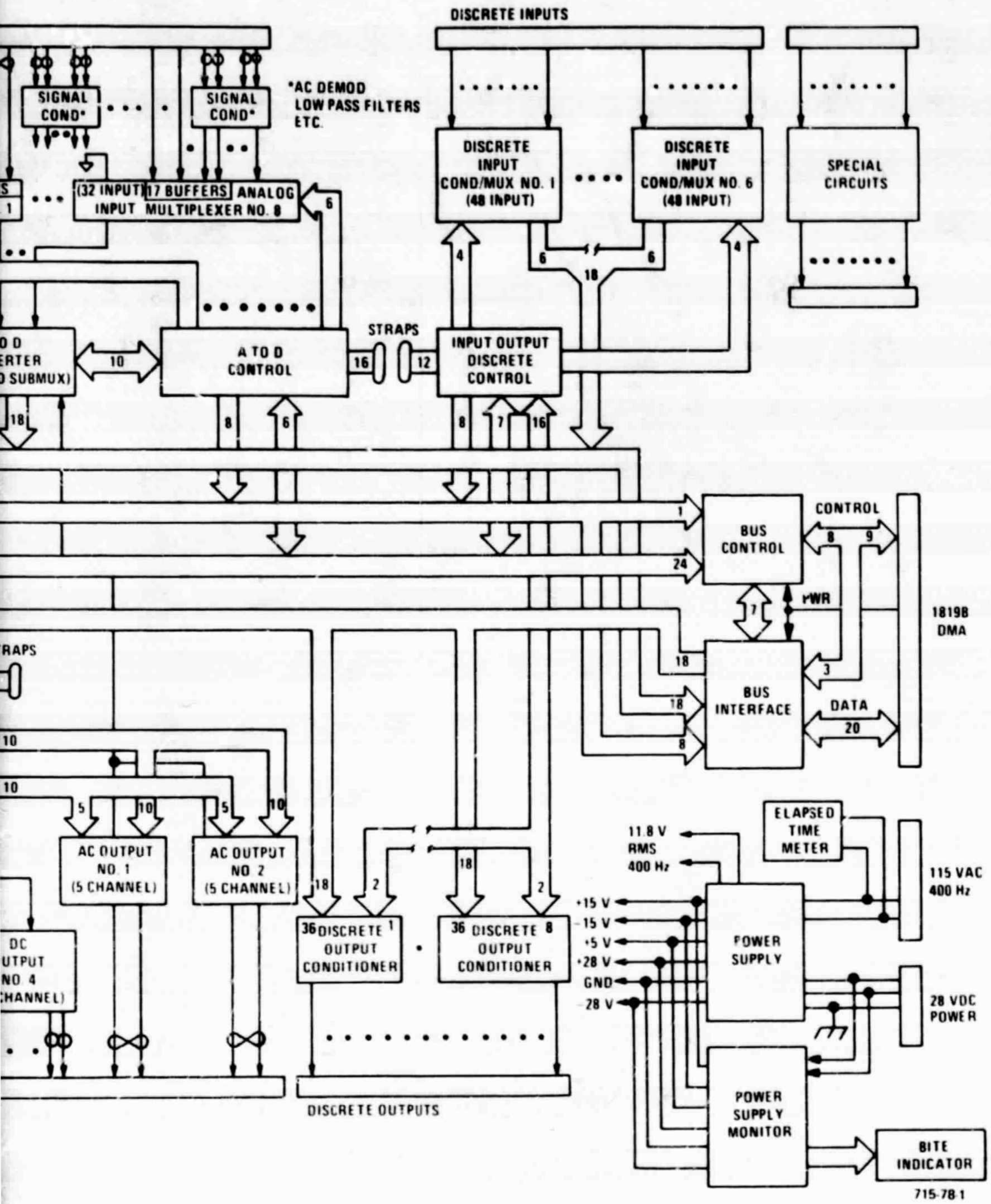


Figure 4-19
Functional Block Diagram of
V/STOLAND XV-15 Data Adapter

FOLDOUT FRAME 2

These subsystems are completely independent and operate asynchronously, using a common power supply and clock. The six subsystems are:

- analog input channel
- discrete input channel
- analog output channel
- discrete output channel
- digital input channel
- digital output channel

Data is transferred between the Data Adapter and the digital computer via the digital computer Direct Memory Access bus. Use of DMA reduces both the software overhead required for transferring data and the number of interconnecting lines as compared to an I/O channel type of interface. All data transfers are initiated by the Data Adapter and require no software in the digital computer to complete the transfer. However, the Data Adapter may be configured so that the computer can control the start of most of the different data interfaces. Table 4-3 summarizes the I/O types, word counts, and address assignments of Data Adapter DMA.

TABLE 4-3

DATA ADAPTER DMA I/O SUMMARY

I/O Type	No. Words		Address Area (octal)
	Available	Used	
Status Word	1	1	4000
Digital Inputs	511	51	4001 - 4777
Digital Outputs	512	444	5000 - 5777
Analog Inputs	256	64	6000 - 6377
Analog Outputs	64	26	6400 - 6477
Discrete Inputs	144*	107*	6500 - 6517
Discrete Outputs	72*	64*	6520 - 6537

*bits (18 bits per word available)

Internally the Data Adapter is organized around a central bus structure. This configuration allows the six major interface subsystems to move data to and from the bus interface and bus control section with a minimum amount of hardware (see Figure 4-19). Each major subsystem has a separate control unit (except for the discrete inputs and outputs which share a control unit) which controls the data flow to/from its section in response to commands from the bus interface and bus control section. The control units may be set up to operate in a continuous mode, or they may be set up to cycle through their complete set of inputs/outputs only upon command from the computer. Each control unit is assigned a priority for the purpose of resolving the conflict that occurs when more than one device requests use of the internal bus at the same time.

4.4.1 DMA Interface

The DMA interface section of the Data Adapter is implemented on two boards and performs the following functions:

- Priority Decoding - Resolves contention problems between up to six requesting devices. In order of priority, they are:
  - Digital Input Request (highest priority)
  - Digital Output Request
  - Analog Input Request
  - Analog Output Request
  - Discrete Input/Output Request
  - Status Request

- Status Monitoring - Up to six inputs may be monitored for changes. If any one of the six changes state, a status request is generated. When the priority decoder honors this request, the current status information is sent to the digital computer as an 18-bit word with the lower 6 bits containing status data. The inputs of the status monitor are the busy signals from the major interface subsystems.

- Address Generation - The memory location accessed for a data transfer by the Data Adapter is selected by an 18-bit address that is formed by the Bus Control and Bus Interface unit and the requesting interface unit control. Bits 9 through 17 are generated by the Bus Control and Bus Interface unit, with the lower bits supplied by the control units.

- Data Transfer - Transfers data at rates up to the maximum DMA rate of one 18-bit word every 1.2 microseconds, which is the core memory cycle time. The actual data rate will depend on the data transfer rate between the Data Adapter and its input/output devices.

Bus Control - Figure 4-20 is a block diagram of the Bus Control section. When the power is first applied to the Bus Control board, the input to the Power-On Clear (POC) circuit clears memory type devices. A special signal (BICLR), generated by the POC circuitry, also clears memory elements on the Bus Interface board. The Bus Control board contains a 5 MHz clock (RCLK) for the Bus Control circuitry. The clock circuit also provides an 833 kHz clock for the A/D circuits and the digital multiplexer.

The Bus Control board also contains three decoders. The Data Request priority decoder determines the priority of the requests from the different data interface subsystems of the Data Adapter. The second decoder (Status Change Detector) detects when a change in status of the device busy lines occurs, such as Digital Input Busy (DIGIBZY, etc). The third decoder is a priority decoder which determines which request is honored when both a Status Request (SREQ) and a Data Request (DREQ) are active.

Finally, the Bus Control also contains an address encoder to reduce the number of signal connections to the Bus Interface.

Figure 4-21 shows the Bus Interface block diagram. This section of the Data Adapter transfers data to and from the computer under the control of the Bus Control board and the computer. Data transfers require both an address and data for a write operation, or an address only for a read operation. The address and data are multiplexed on the DMA bus under control of the computer.

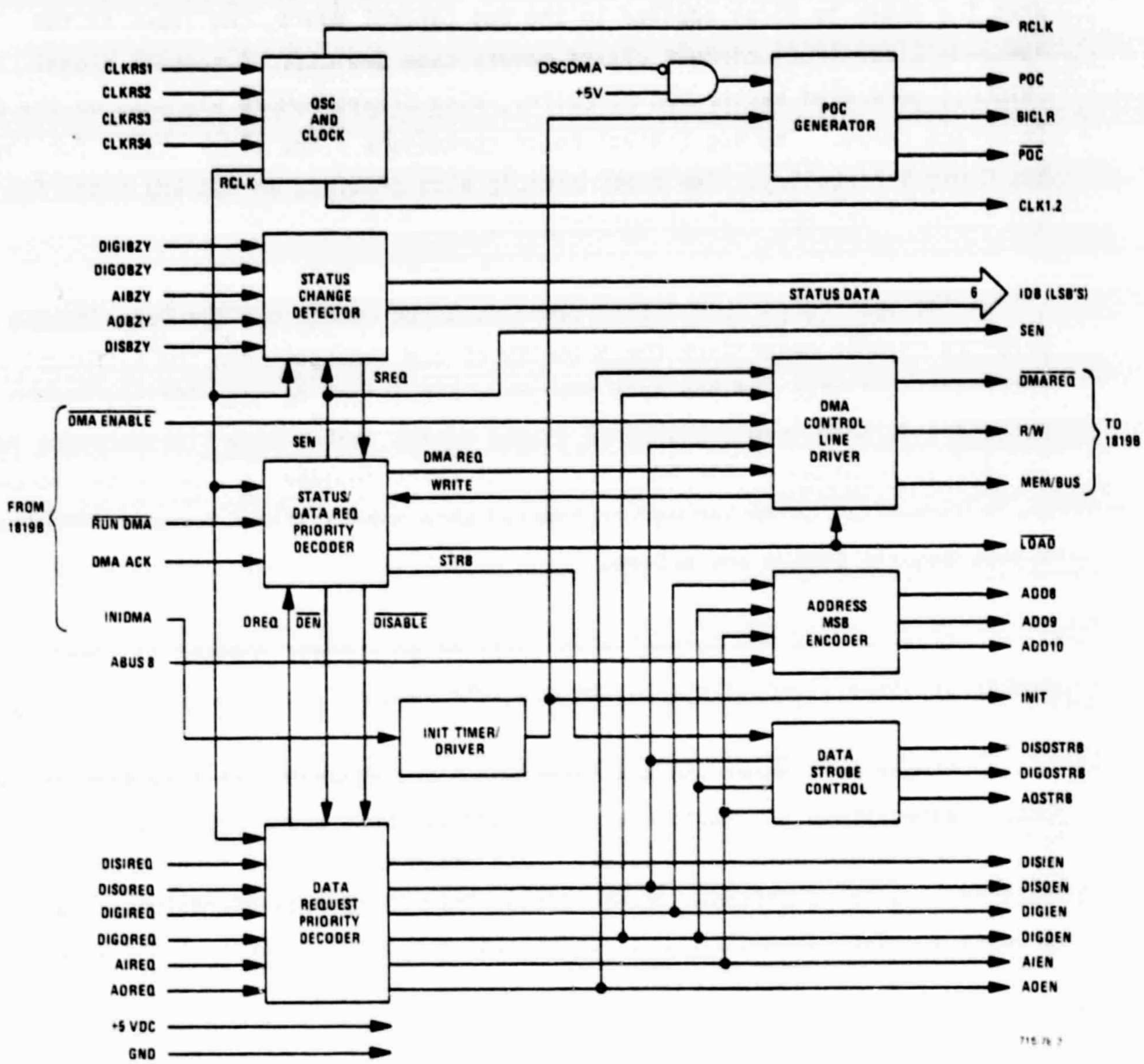


Figure 4-20
Bus Control Block Diagram

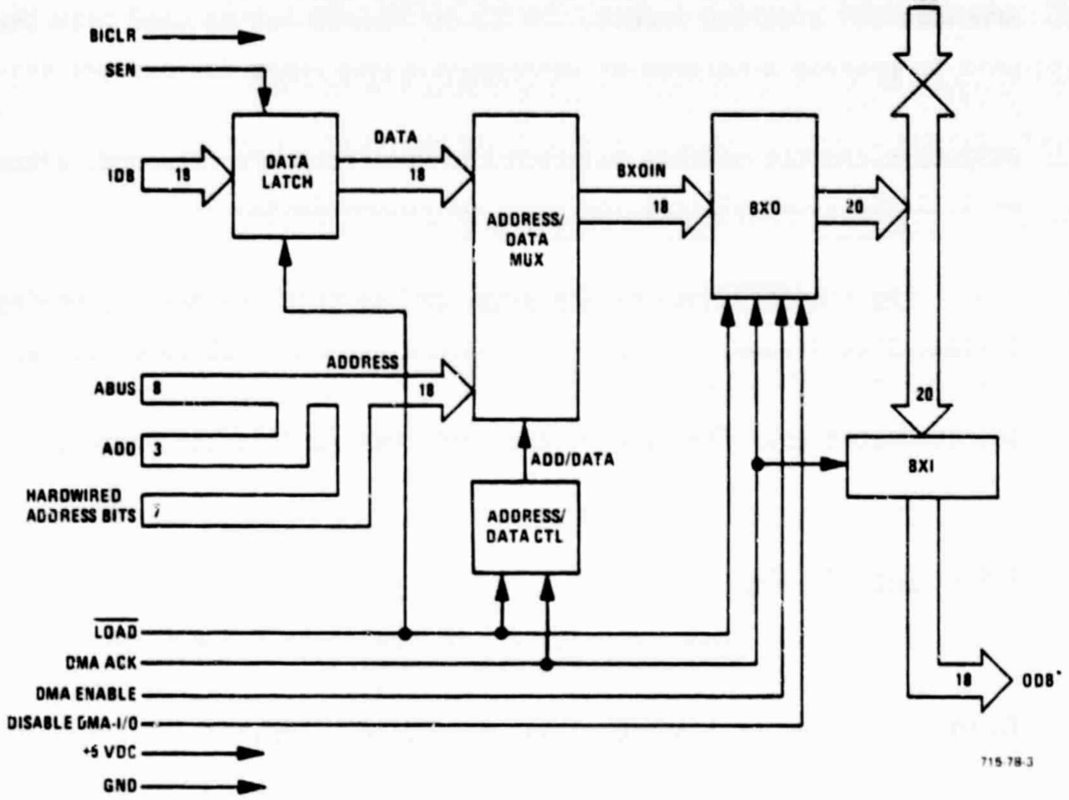


Figure 4-21
Bus Interface Block Diagram

4.4.2 Discrete Input/Output

The Discrete Input and Discrete Output sections share a common control unit which is shown in the block diagram of Figure 4-22. Thus, for every discrete input word transferred, one discrete output word will be transferred. A tri-state buffer is used to drive the address bus. The discrete inputs are organized so that one board has 48 inputs which are broken up into eight groups or words of six inputs each (see Figure 4-23). Thus, when the discrete inputs are active, one board will have six inputs enabled to drive six bits of the DA internal data bus and, in turn, six bits of the DMA bus. The Data Adapter provides 144 discrete inputs. Up to six boards may be used with the control unit to provide a maximum of 288 inputs. The input devices are tri-state CMOS buffers with resistive dividers on their inputs. By changing the value of the resistors and the voltage to which the resistors are returned, almost any type of discrete input voltage level may be accommodated.

The discrete outputs are organized so that one board contains 36 outputs divided into groups of three which have a common input power pin as shown in Figure 4-24. Four of the 36 discretes are sink-type discretes, and 32 are source discretes. The Data Adapter provides 72 discrete outputs (2 cards), with the capability for expansion to 288.

4.4.3 Analog Input

The analog input provides all of the control functions and signal conditioning necessary to convert a variety of ac and dc analog signals into a binary format suitable for transfer into the digital computer.

The analog input subsystem is made up of three standard board types plus special conditioner boards that are required to meet the signal conditioning requirements of the system. The three standard boards are: A/D Control, A/D Converter, and Analog Input Multiplexer. The A/D control board provides the logic to control the A/D converter and up to eight analog input multiplexer boards. A single analog-to-digital converter is used with the analog input multiplexers to form a multi-input A/D.

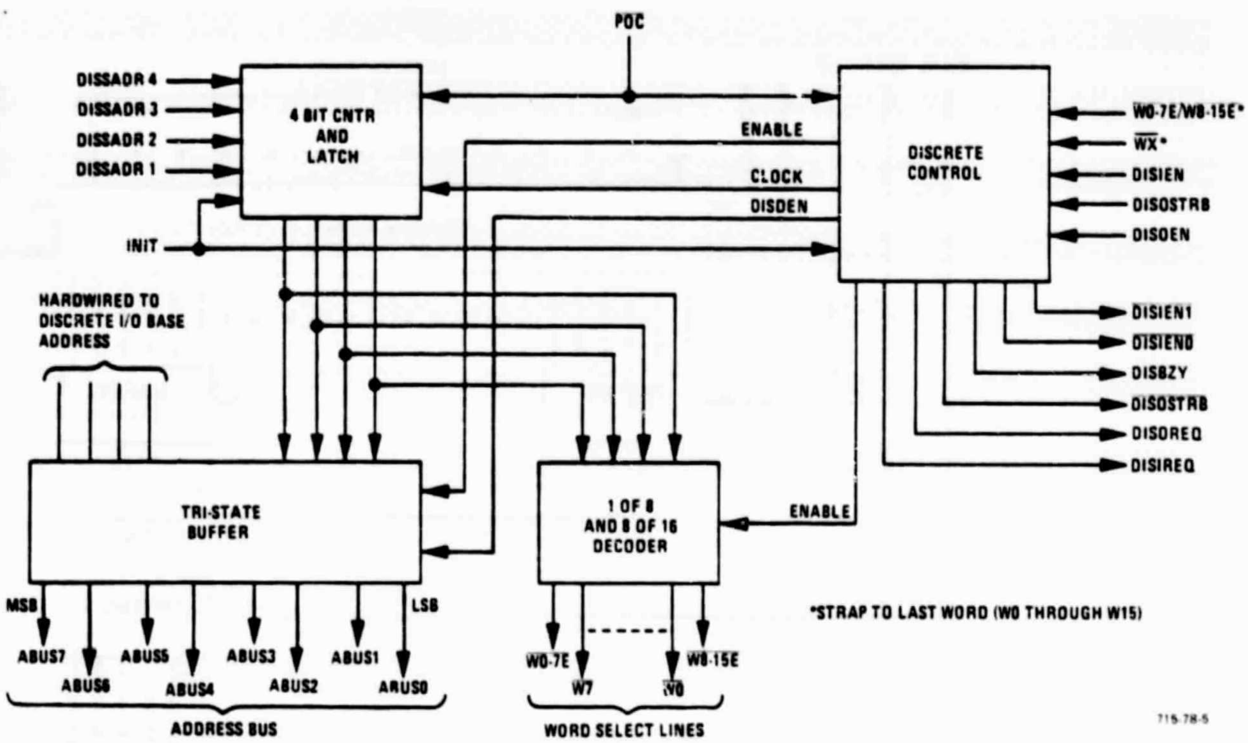


Figure 4-22
Input/Output Discrete Control Block Diagram

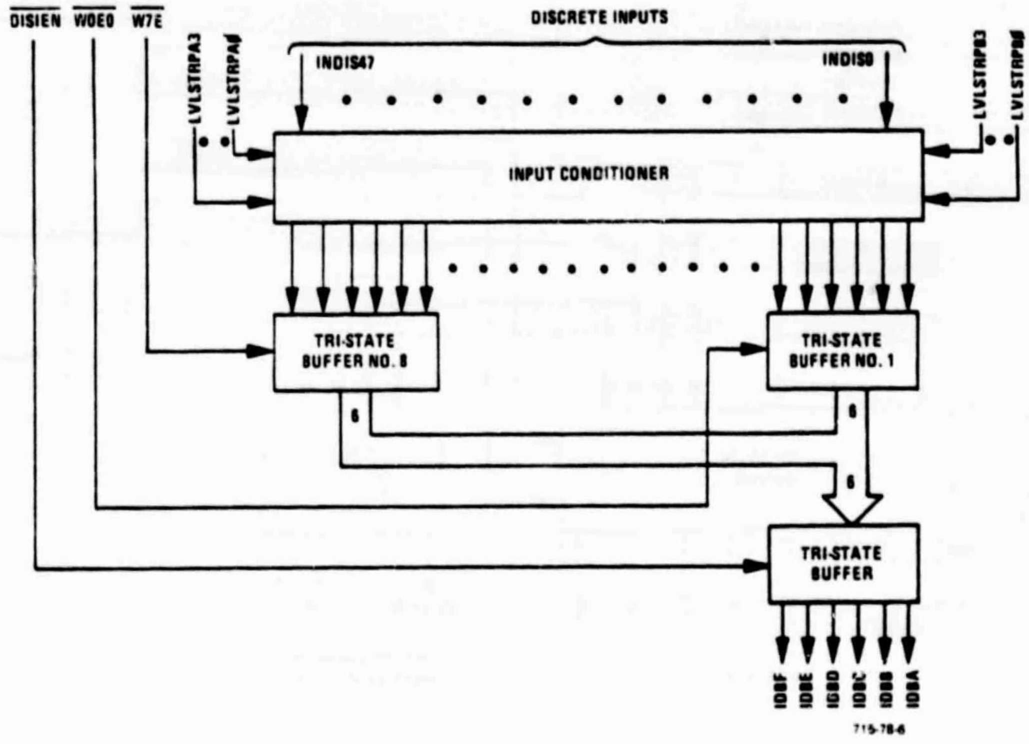


Figure 4-23
Input Discrete Conditioner/Multiplexer Block Diagram

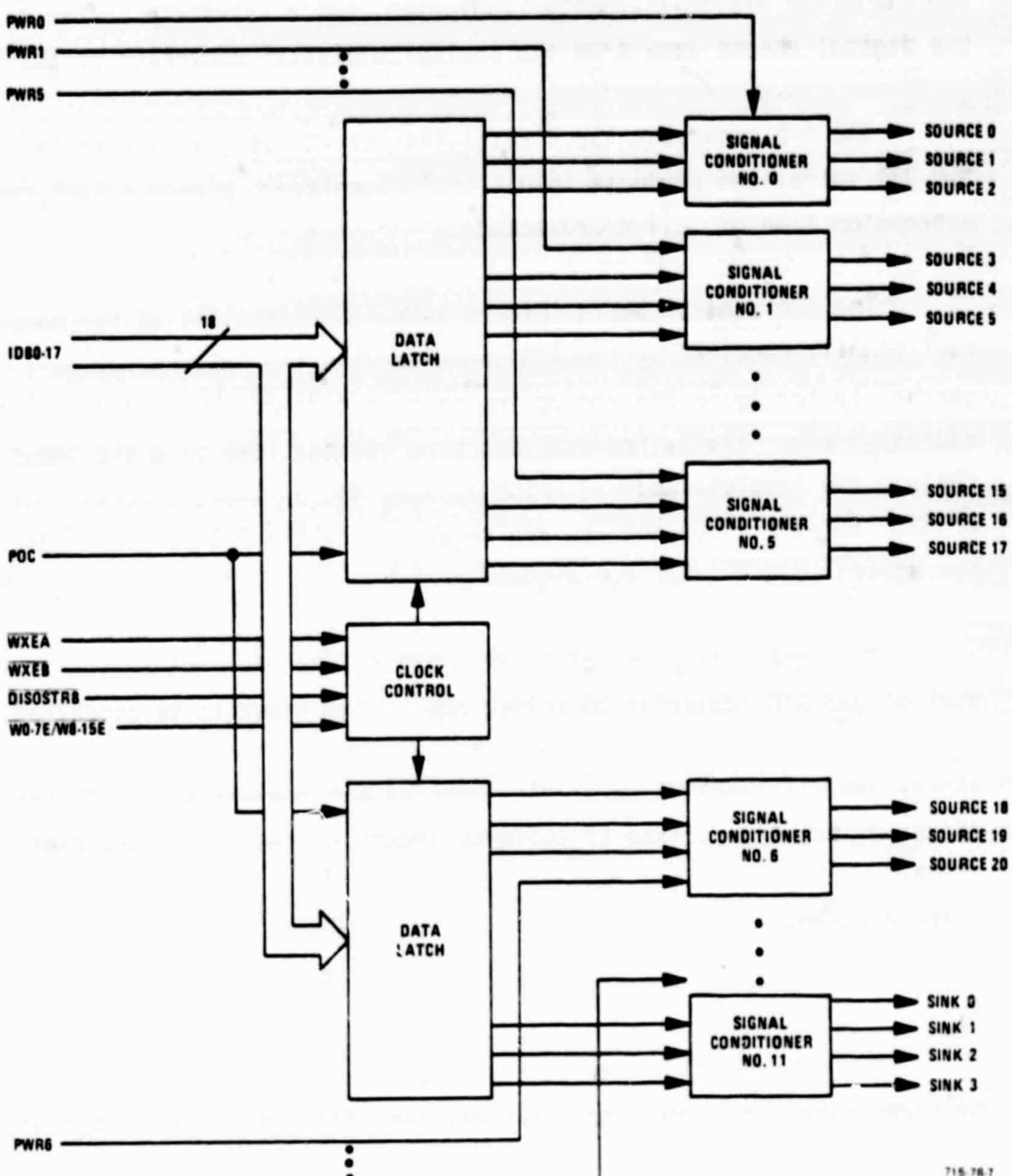


Figure 4-24
Discrete Output Source Block Diagram

The A/D converter, shown in Figure 4-25, is used in conjunction with the analog input multiplexer and the A/D control to convert a selected analog input voltage to a digital data word. It contains an eight-input multiplexer for selecting one of eight analog input multiplexer board outputs, a successive approximation analog-to-digital converter, and a tri-state buffer for placing the digital output data from the analog-to-digital converter on the data bus.

The A/D converter has a resolution of 12 bits, a full-scale range of ±10.235 volts, and produces 18-bit one's-complement binary output code at a conversion time of 32.4 microseconds.

The A/D control board (Figure 4-26) generates all of the necessary control signals required by the A/D converter and the A/D multiplexer. Addressing capability for up to 256 analog inputs is provided. The analog inputs are addressed sequentially from the starting address (set by eight input straps) through the complete address which is also set by input straps. The control signals necessary for the SAR are generated by the A/D control board, as are the bus control board interface signals.

The analog input multiplexer board is used in conjunction with the A/D control and A/D converter to select the analog input to be converted by the A/D converter. The Analog Multiplexer contains 17 buffer amplifiers for use as gain blocks and differential to single-ended signal conversion. A 32-input multiplexer selects one of the 17 buffered inputs or one of 15 single-ended non-buffered inputs for conversion. Figure 4-27 shows the analog input multiplexer block diagram.

4.4.4 Analog Output

Three board types are used in the analog output section (Figure 4-28). 64 words total in 1819B memory are reserved for the dc and ac outputs.

The function of the analog output control board is to convert 12-bit one's complement numbers retrieved from computer memory into corresponding analog voltages (1 LSB = 5 millivolts). The conversion of the digital data into analog voltages is done by a multiplexing scheme which selects the dc output storage (sample/hold) device for which each output is intended. A second

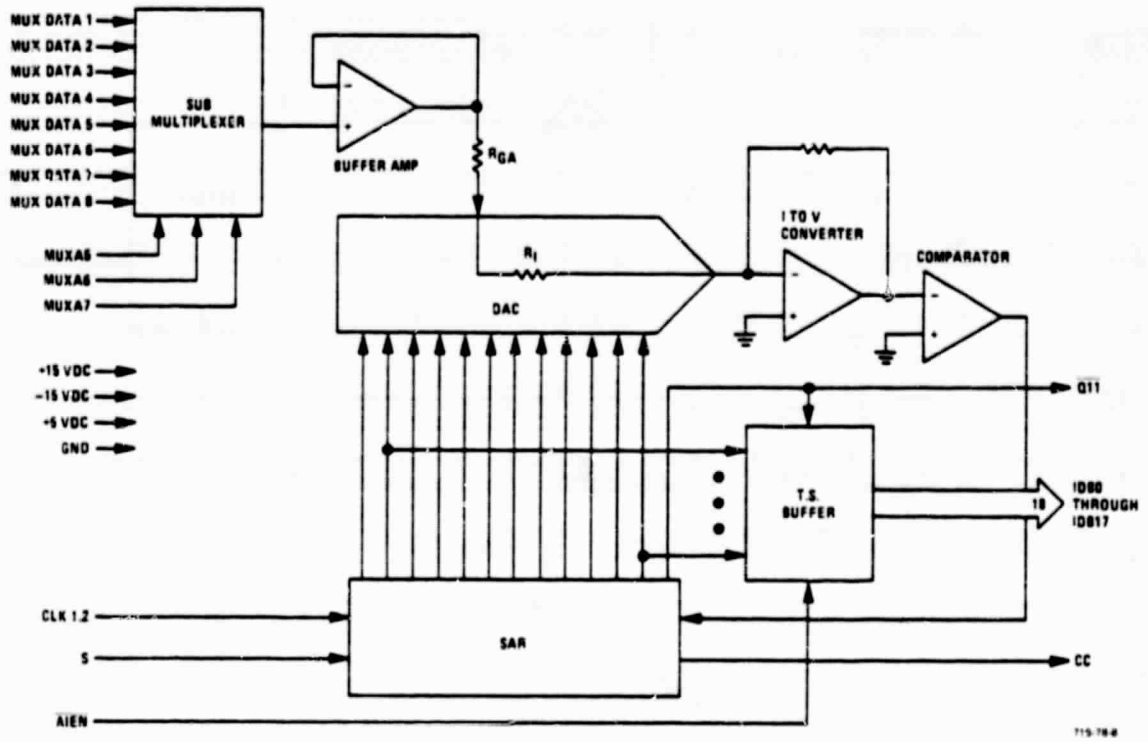


Figure 4-25
A/D Converter Block Diagram

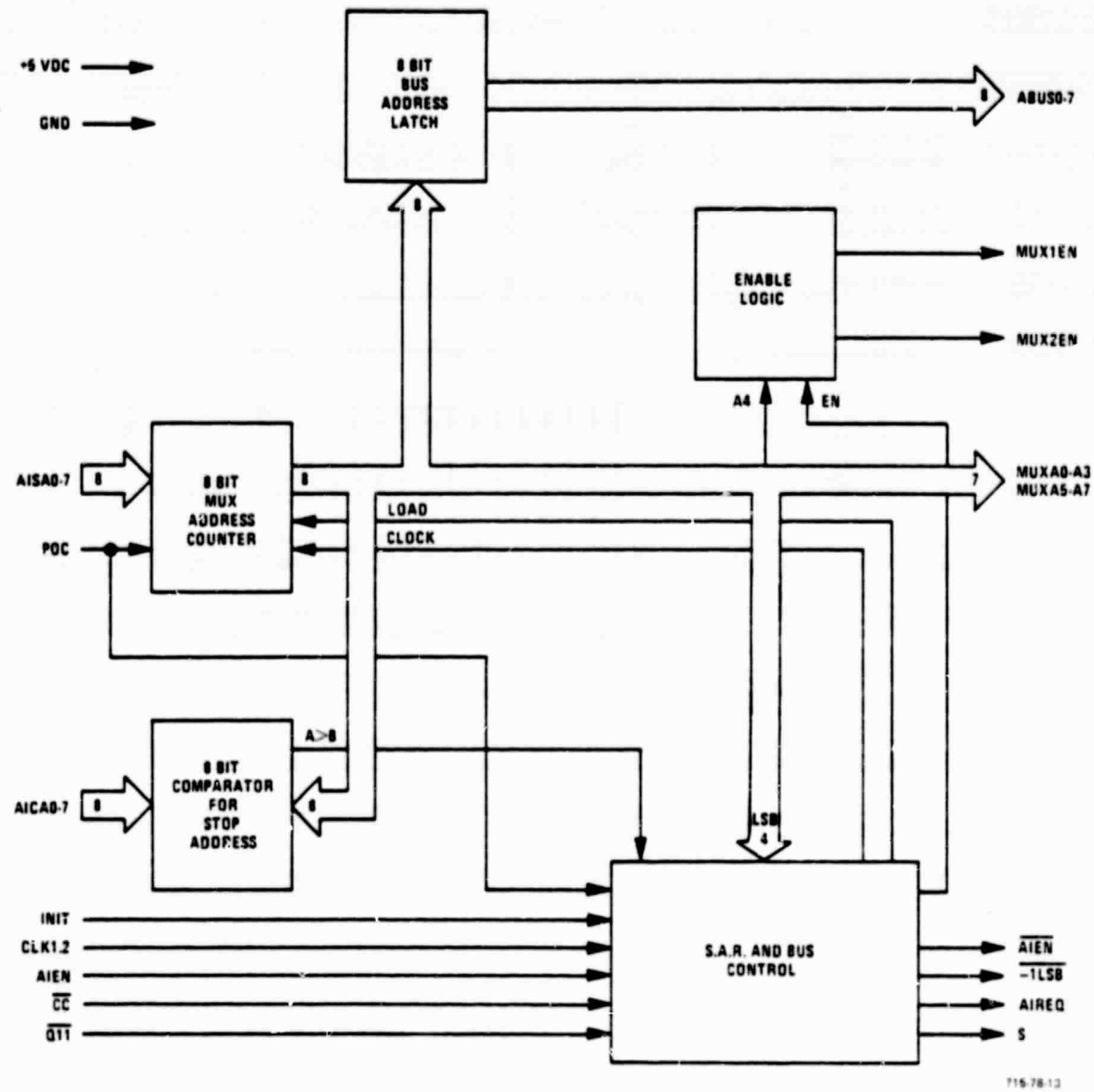


Figure 4-26
A/D Control Block Diagram

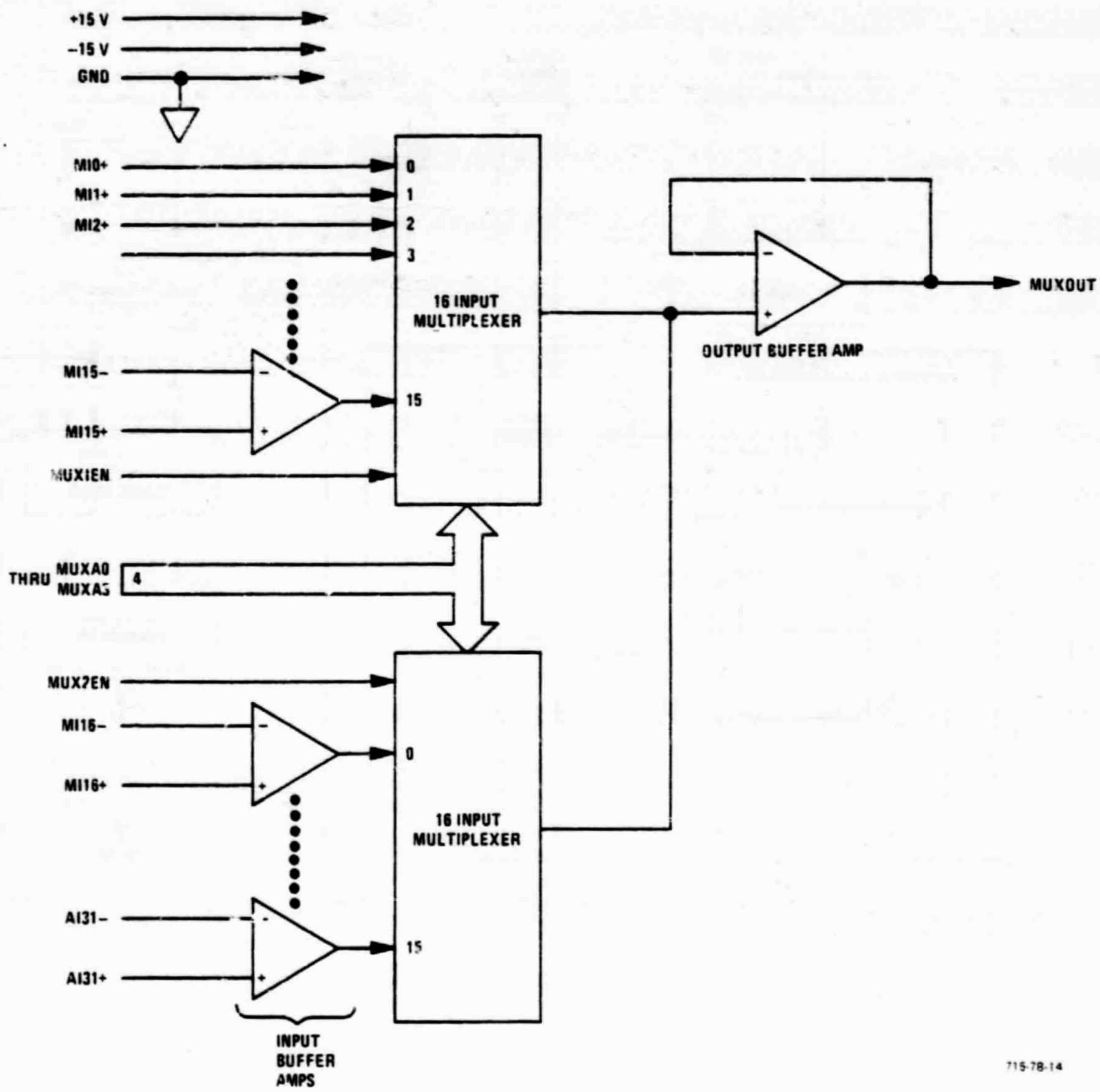


Figure 4-27
Analog Input Multiplexer Block Diagram

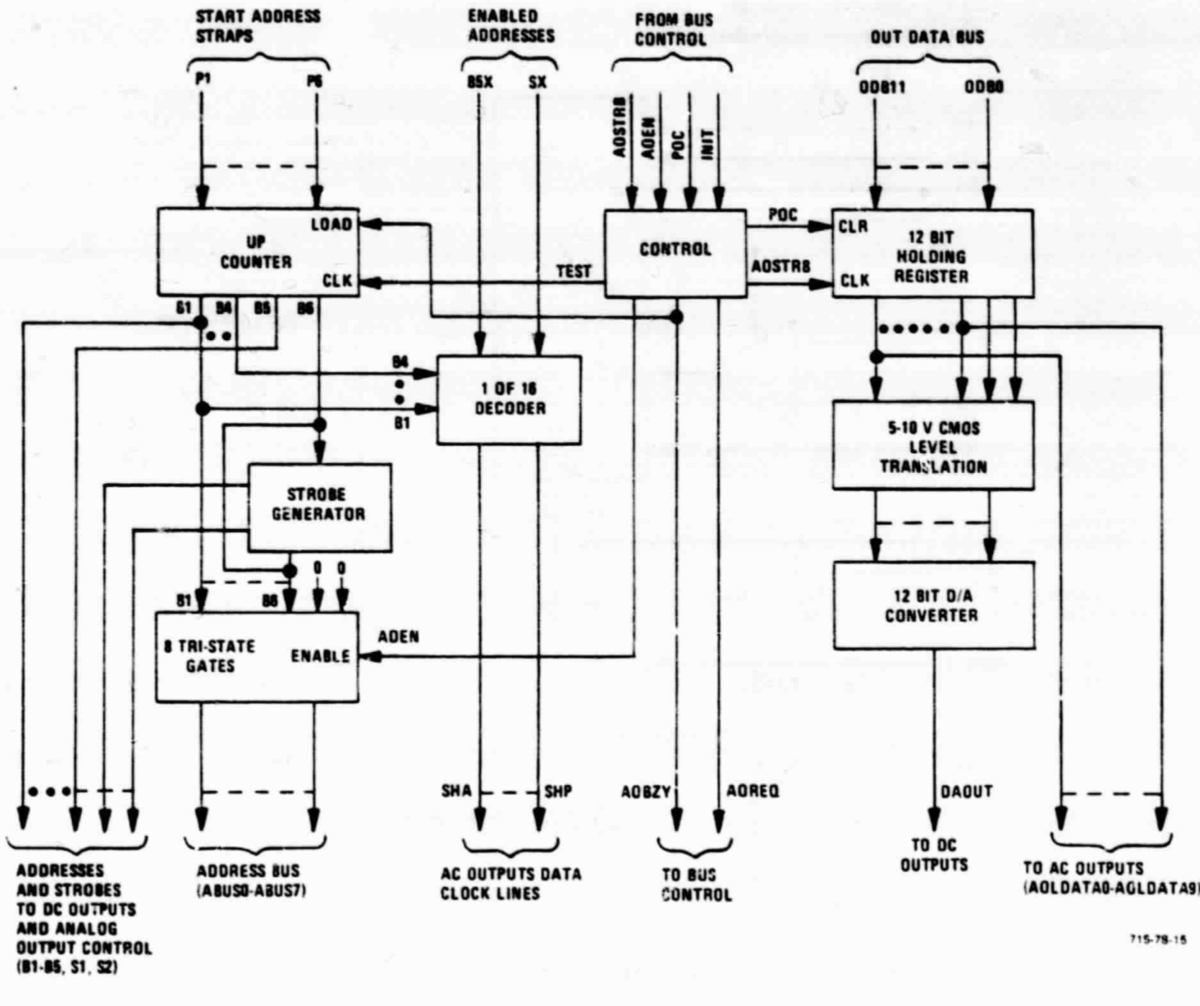


Figure 4-28
Analog Output Control Block Diagram

function of the analog output control board provides digital (rather than analog) information for up to 16 ac output devices using the same multiplexing scheme. The total number of dc and ac outputs which can be serviced is 64.

The analog output section converts up to 64 words in 12 bit one's complement form (1 LSB = 5 millivolts for dc outputs; 1 LSB = 5.77 millivolt rms for ac outputs). The update period is the number of devices times 160 microseconds, so the 26 devices require 4.16 milliseconds update time. The output voltage range is ±10.235 volts, with slew rate of .5 volt per microsecond and output current of ±1 milliampere. A 25 millivolt error is typical, with 40 millivolts maximum. The resolution is 5 millivolts.

Each dc outputs board (Figure 4-29) contains a decoder which has its inputs wired to the proper address lines from the analog output control board. The decoder enables the proper sample/hold during the sample period of the sample/hold update cycle. Each sample/hold is connected to the output of the DAC on the control board so that when placed in the sample mode, the sample/hold will acquire the current value of the DAC output. Each dc outputs board contains 16 sample/holds, 16 sample/hold capacitors, and an address decoder. Up to four dc outputs boards may be controlled by the analog output control board.

The dc output is accurate to 5 millivolts error typical, 15 millivolts maximum for a 5 millisecond period (no scaling). The output voltage is ±10.235 volts full scale, with ±5 milliamperes maximum output current.

400-Hz ac outputs are provided for driving synchros. These outputs may be varied in amplitude and phase (either 0 or 180 degrees with respect to the reference) by the digital data that is also used to drive the DAC on the analog output control board. The decoded address from the control board is used to latch the digital data in the latches on the ac outputs board. Each of the five outputs on each ac outputs board (Figure 4-30) has a common 400-Hz reference driving a 10-bit multiplying DAC which in turn drives a power amplifier. The 10 bits used are the upper 10 bits of the 12-bit one's complement data sent to the analog output control. The ac outputs are short-circuit protected for shorts to ground.

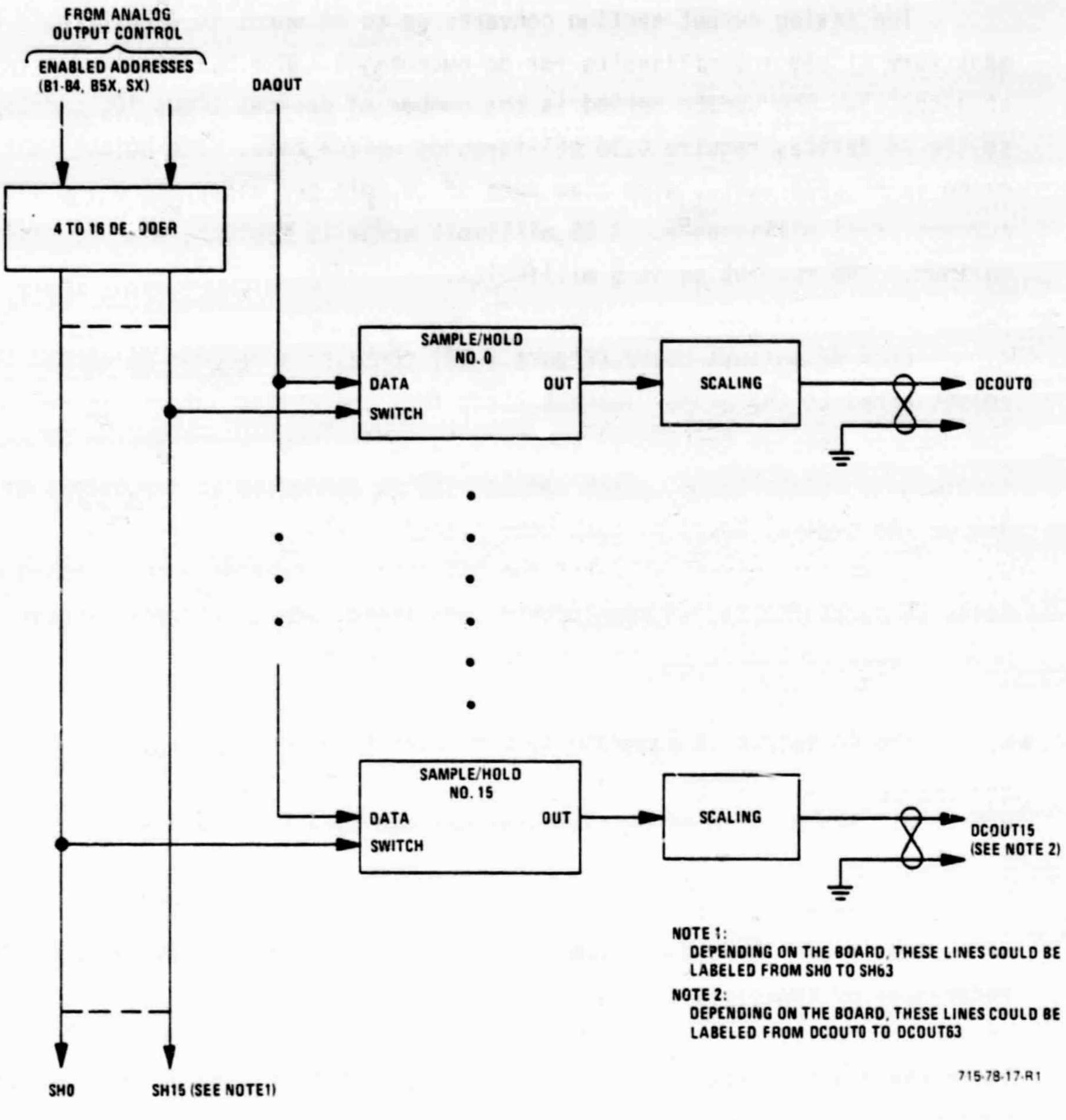


Figure 4-29
DC Outputs Block Diagram

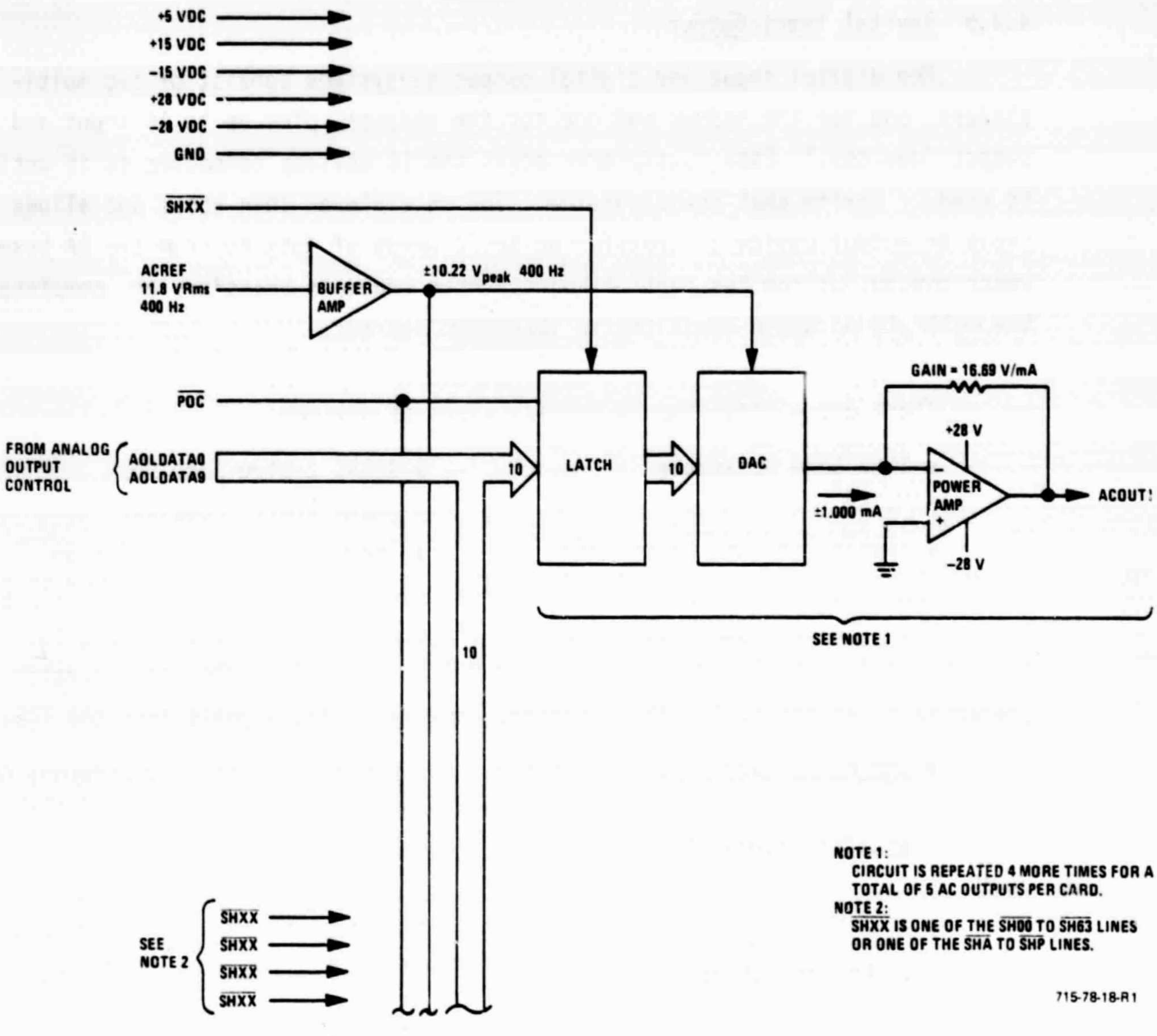


Figure 4-30
AC Outputs Block Diagram

The output voltage is adjustable from 0 to 11.8 vrms, 0 or 180 degrees, with short-circuit output current of 35 milliamperes rms minimum, 55 milliamperes rms maximum.

4.4.5 Digital Input/Output

The digital input and digital output subsystems consist of two multiplexers, one for the inputs and one for the outputs, plus up to 16 input and 16 output "devices." Each multiplexer scans the 16 devices connected to it until it finds a device that needs service. The multiplexer then stops and allows the input or output device to transfer up to 32 words of data to/from the DA buses under control of the bus control board. When the data transfers are complete, the multiplexer again scans the input/output devices.

There are digital inputs for six types of devices:

- Time Code Generator
- TACAN
- Static Pressure Transducer
- ARINC 2-Wire Receiver
- ARINC 6-Wire Receiver
- Inertial Navigation System

The time-code generator interface (Figure 4-31) is a special digital input board for the Datametrics Model SP-375 Airborne Synchronized Generator (referred to as the TCG). The interface receives three signals from the TCG:

- Serial DC Code IRIG-B - A pulse train containing the time information (seconds, minutes, hours, and days) in the format specified by IRIG standard format B.

- Modulated Code - The serial dc code IRIG-B amplitude modulated on a 1-kHz sine signal.

- 1PPS - A one-per-second pulse train synchronized to within 100 microseconds of the pulse PR occurring in the serial dc code IRIG-B.

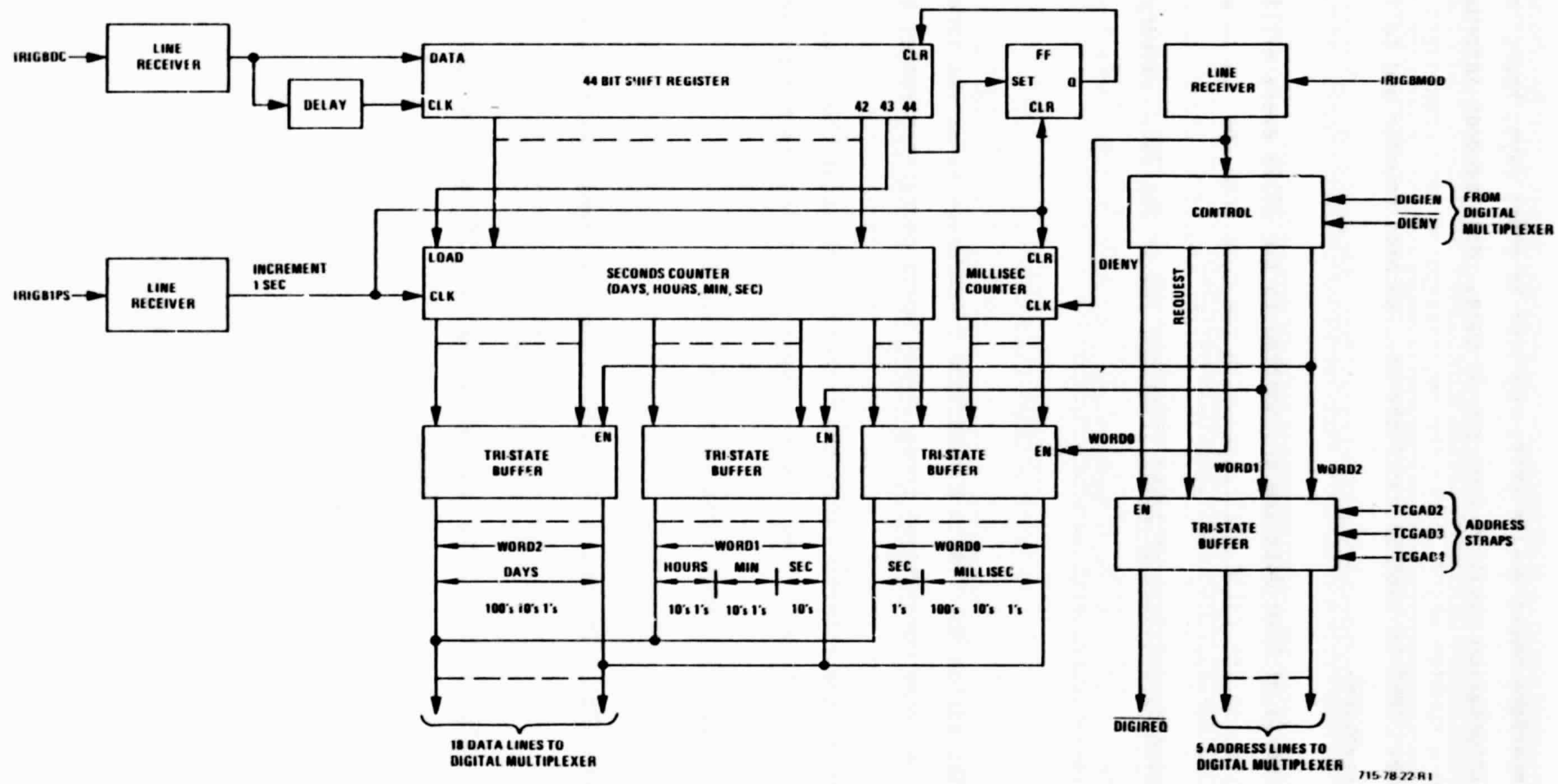


Figure 4-31
Time Code Generator Interface Block Diagram

The interface board uses a shift register to store days, hours, minutes, and seconds information obtained from the dc code. Milliseconds information is obtained from a counter which counts the positive-going zero crossovers of the modulated code. 1PPS is used to reset the milliseconds counter and to increment the time by 1 second.

Three words of time data are transmitted to the 1819B every millisecond. Time information in the 1819B will agree with the code from the TCG to within an accuracy of one millisecond. The leap-year straps should be wired correctly if days information is critical at 23:59:59 on day 365 or day 366. January 1 is considered Day 1. Valid time information will be sent to the 1819B, starting 1.5 seconds after application of system power.

The TACAN Interface (Figure 4-32) is a special digital input board for the Hoffman Navigation Set, TACAN AN/ARN-103( ) (referred to as the TACAN). The output from the TACAN consists of three 14-bit words containing range, bearing, and control information, respectively. One of these three words is transmitted in serial form to the interface when the interface generates the proper code on the two TACAN data request lines (TACDRA/TACDRB). The interface also generates the 1 MHz TACAN Input Shift Clock (TACISC) which the TACAN uses to properly synchronize the serial transmission of the data word.

The Static Pressure Transducer interface is a frequency-to-digital converter (Figure 4-33). The Static Pressure Transducer generates a squarewave with frequency a function of the barometric pressure. Conversion of the sensor frequency is done by counting the number of clock pulses (generated by a highly accurate 5-MHz clock) which occur during 100 periods of the squarewave. This number, called the PS count, is related to the sensor frequency by the equation

$$\text{PS count} = \frac{5 \times 10^8}{\text{Sensor Frequency}}$$

The sensor period varies from 320 to 420 microseconds (between 0 and 20,000 feet) so the PS count typically will vary from 160,000 to 210,000. This data is converted to barometric altitude by computer programming.

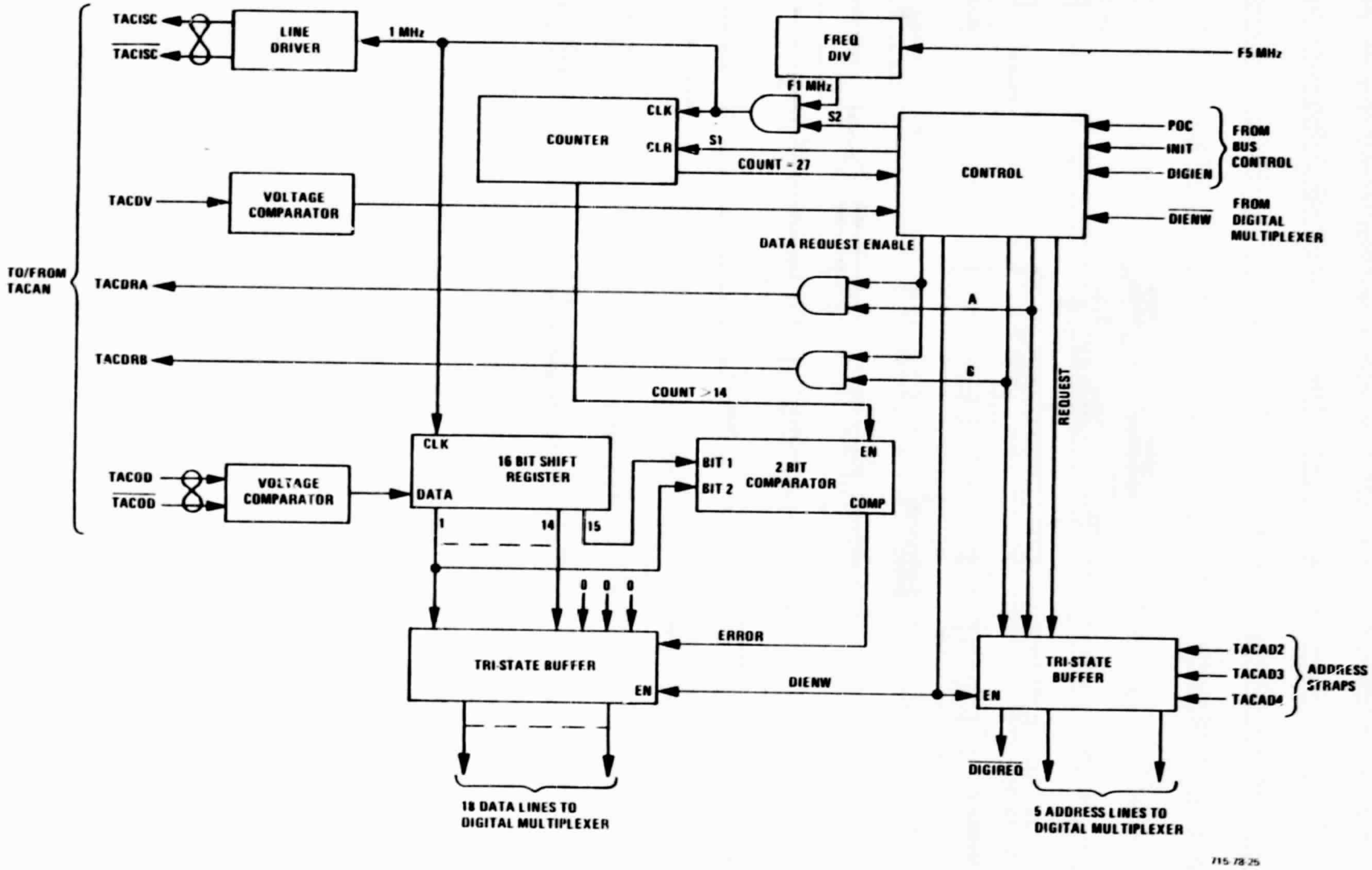


Figure 4-32
TACAN Interface Block Diagram

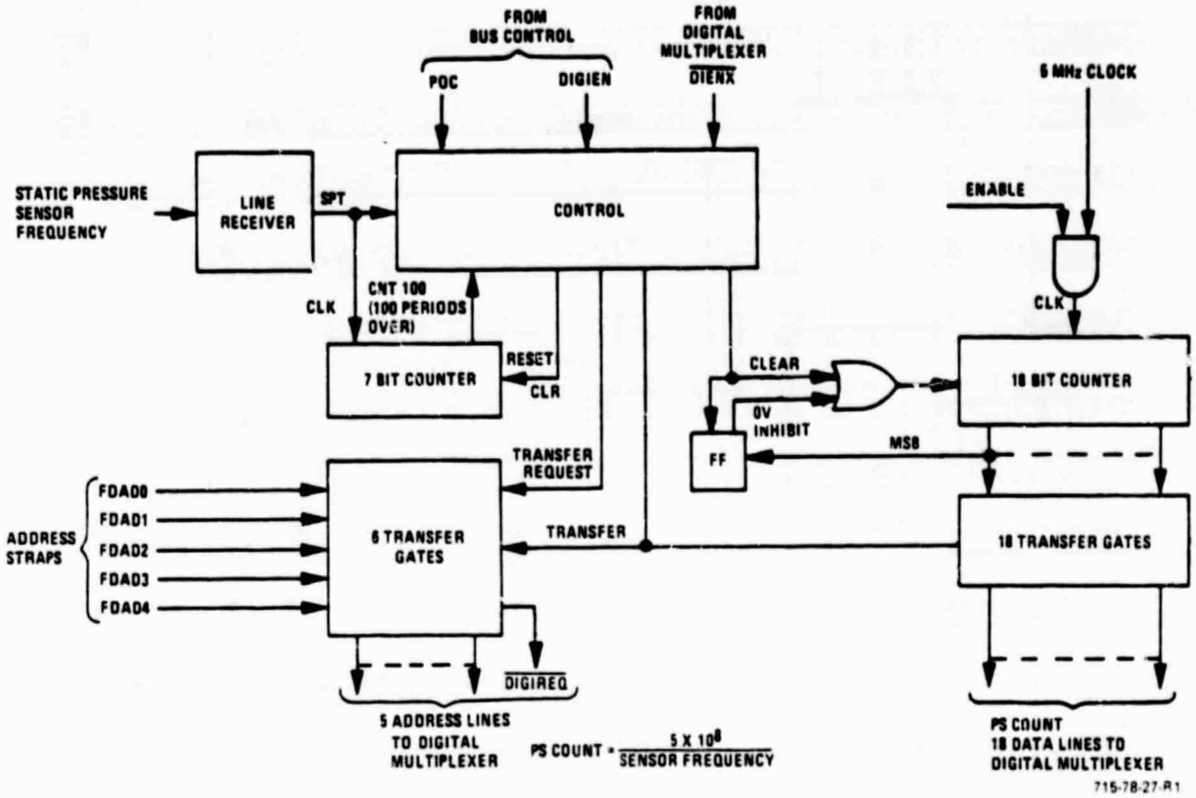


Figure 4-33
F/D Converter Block Diagram

The digital ARINC 2-wire receiver interfaces with a VOR receiver or an MLS receiver and digital multiplexer of the data adapter. The input signals from the VOR or MLS units are received serially and transferred to the multiplexer in parallel. The VOR transmits one 32-bit word which is stored as two 18-bit words in the 1819B memory. The receiver is sectioned into nine parts as shown in the block diagram, Figure 4-34.

The ARINC 6-wire receivers interface with INS, DME, and the digital multiplexer of the data adapter. The data received is a serial 32-bit data format at bit rates from 7.5 kHz to 14.5 kHz. The receiver components are shown in Figure 4-35.

The Inertial Navigation System Delta V Interface (INS Delta V) is a special digital input board (Figure 4-36) for the I_2S Electronic Interface Unit (EIU). The output of the EIU consists of three 12-bit two's complement numbers (words) containing acceleration data in the X, Y and Z axes, respectively. These three words (36 bits) are transmitted in parallel form to the interface when the INS Interrupt (INSINT) line becomes active. INSINT is triggered by the Initiate (INIT) line from the bus control; INIT, in turn, is controlled by the software. An INIT pulse should occur every 50 milliseconds, although none of the control circuitry depends on that rate.

There are two types of digital outputs: the Instrumentation Output and the Split-Phase Bipolar Transmitter. 512 words total are reserved for them in computer memory.

The Instrumentation Output (Figure 4-37) is a special digital output board which interfaces with the Remote Multiplexer Demultiplexer Unit (RMDU). Each time the RMDU pulses its REQUEST line, the Instrumentation Output board generates a digital output request to the bus controller in the data adapter. Within 50 microseconds the request is serviced and 16 bits of data are latched and transmitted in parallel form to the RMDU. An 8-bit word counter provides the address of the word pulled from memory; this means that up to 256 different words can be transmitted to the RMDU.

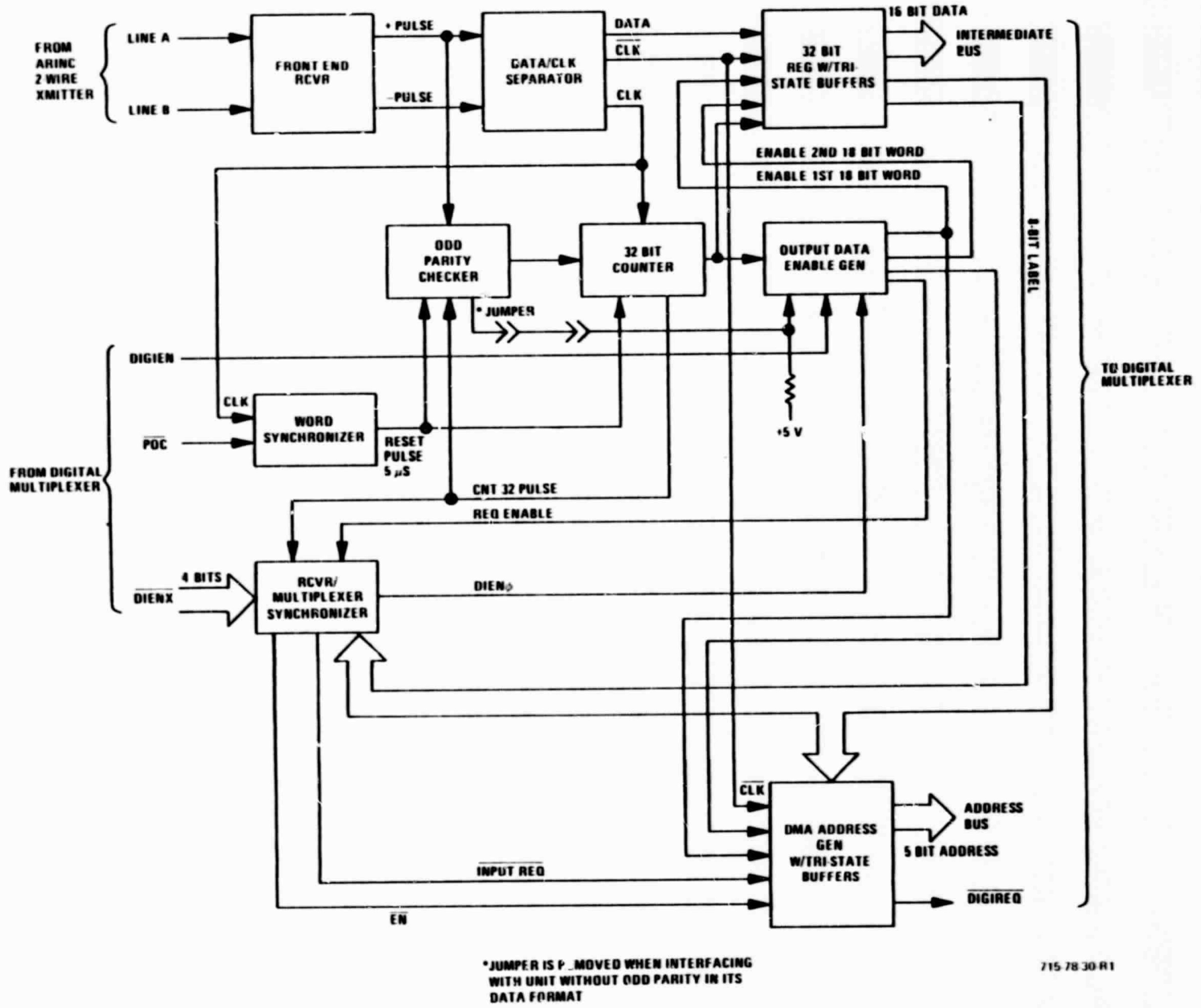


Figure 4-34
ARINC 2-Wire Receiver Block Diagram

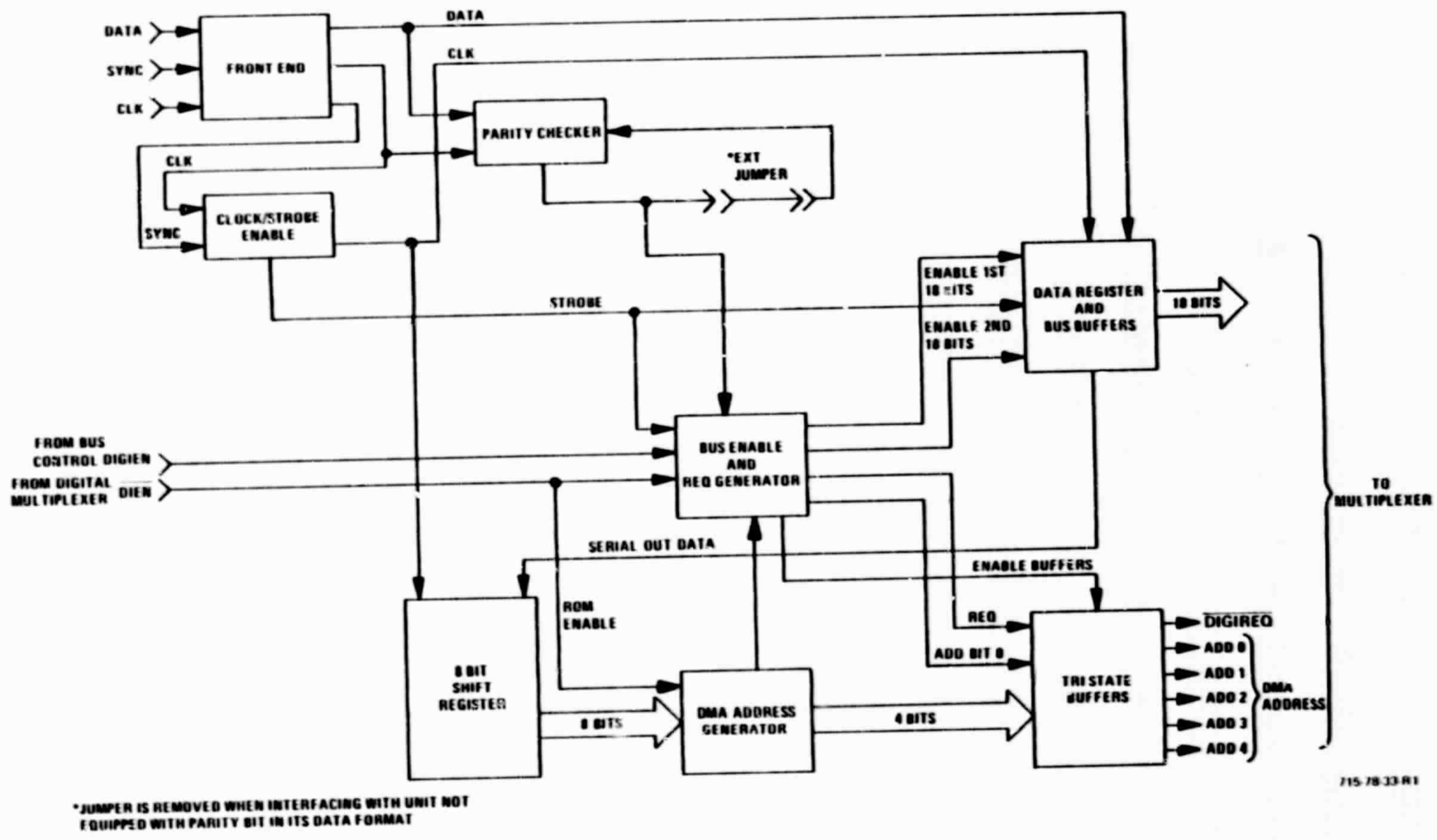


Figure 4-35
ARINC 6-Wire Receiver Block Diagram

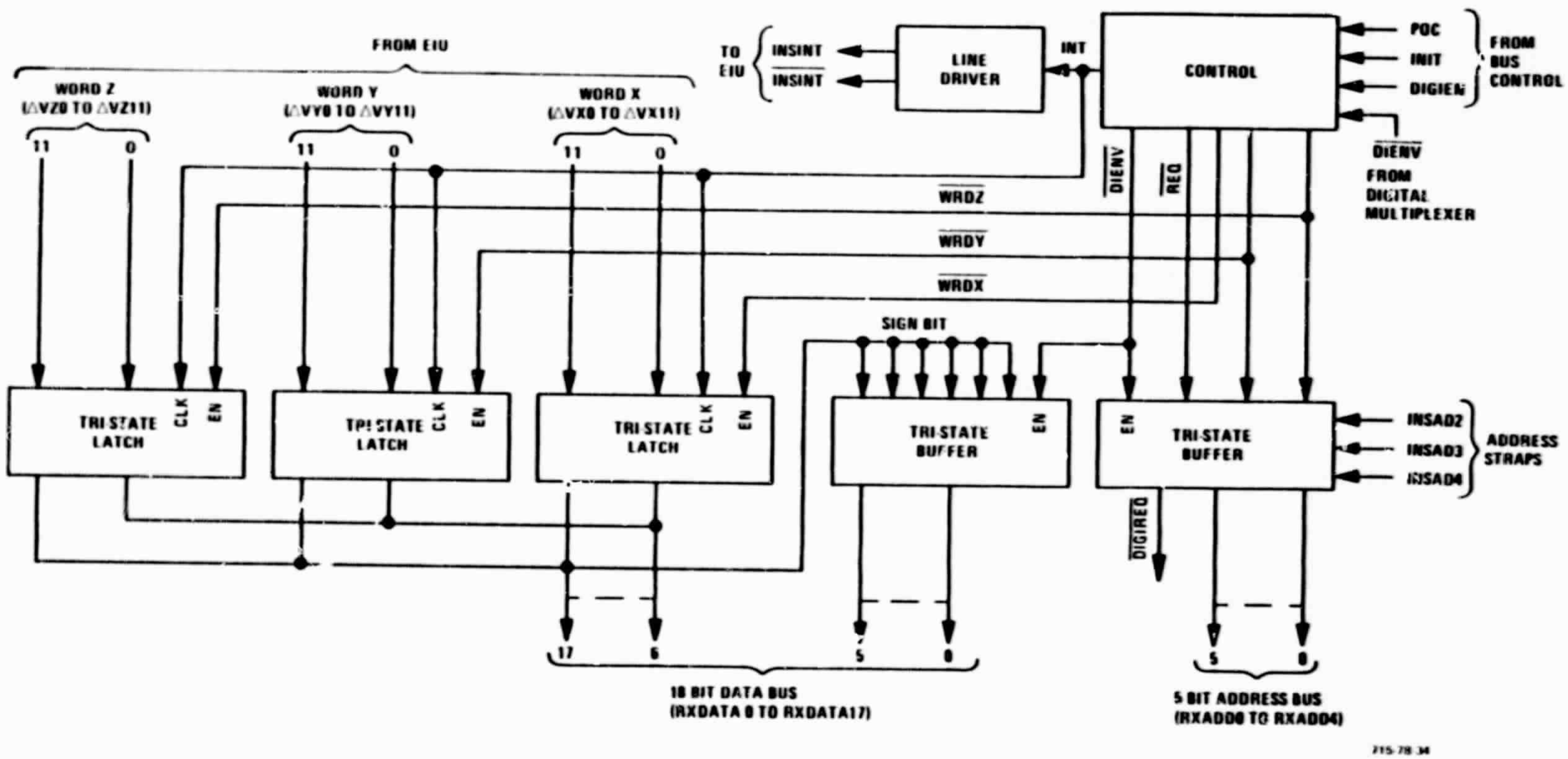


Figure 4-36
INS Delta V Interface Block Diagram

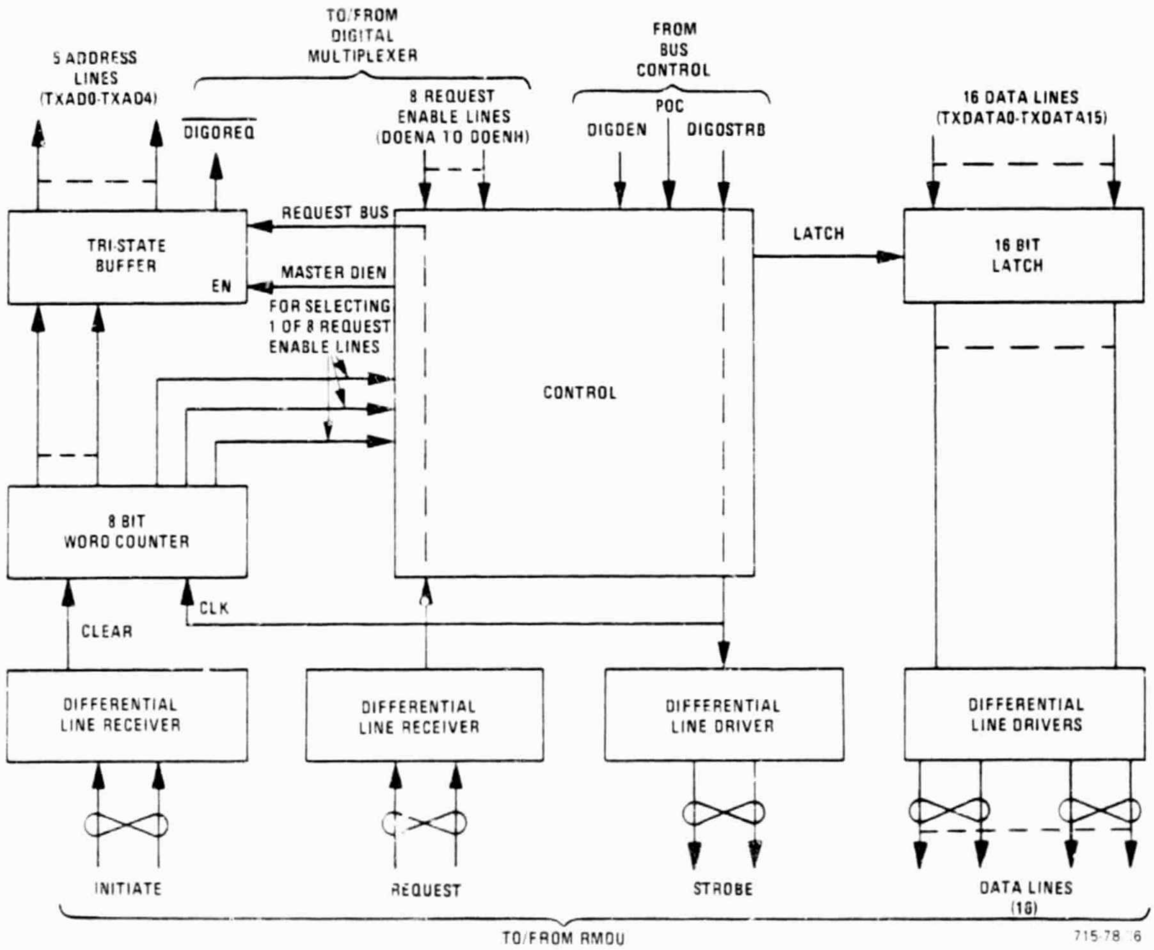


Figure 4-37
Instrumentation Output Block Diagram

The Split-Phase Bipolar Transmitter board is a general purpose board which interfaces with any device which can accept Split-Phase Bipolar (SPBP) data. During each update cycle, (delineated by an INIT pulse from the bus controller), the transmitter transmits up to 128 32-bit words in SPBP serial form (see Figures 4-38 and 4-39). The exact number of words transmitted is determined by straps on the wire-wrap plate, and is limited by the input clock frequency and the number of words being used by other digital output devices.

A transmission cycle starts when the board requests a double word digital output transfer from the bus control. (See Figure 4-40.) The two words (whose addresses in 1819B memory are determined by an 8-bit word counter) are loaded in two 16-bit shift registers. The 32 bits are then transmitted serially in the format shown in Figure 4-38. Two double-width one's are transmitted in front of each word, and bit 32 is determined by an odd parity generator located on the board. After all 32 bits are transmitted, the word counter increments and the board requests that two new words be loaded into the shift registers. This cycle continues until the word counter equals the end address strapped on the backplane. At that time the board goes into a do-nothing state, waiting for an INIT pulse to start another update cycle.

The normal SPBP transmission rate is 50K bits per second. At this rate, 64 32-bit words can be transmitted during a 50 millisecond update cycle. By doubling the input clock frequency, the transmitter is capable of transmitting 128 32-bit words at a 100K bits-per-second transmission rate.

4.4.6 Special Circuits

Several special circuits are provided to handle special interface requirements of the data adapter.

Six single-pole, low-pass filters are provided with 10 rad/s (1.59 Hz) break frequency. They are used to inhibit frequency aliasing on analog sensor inputs.

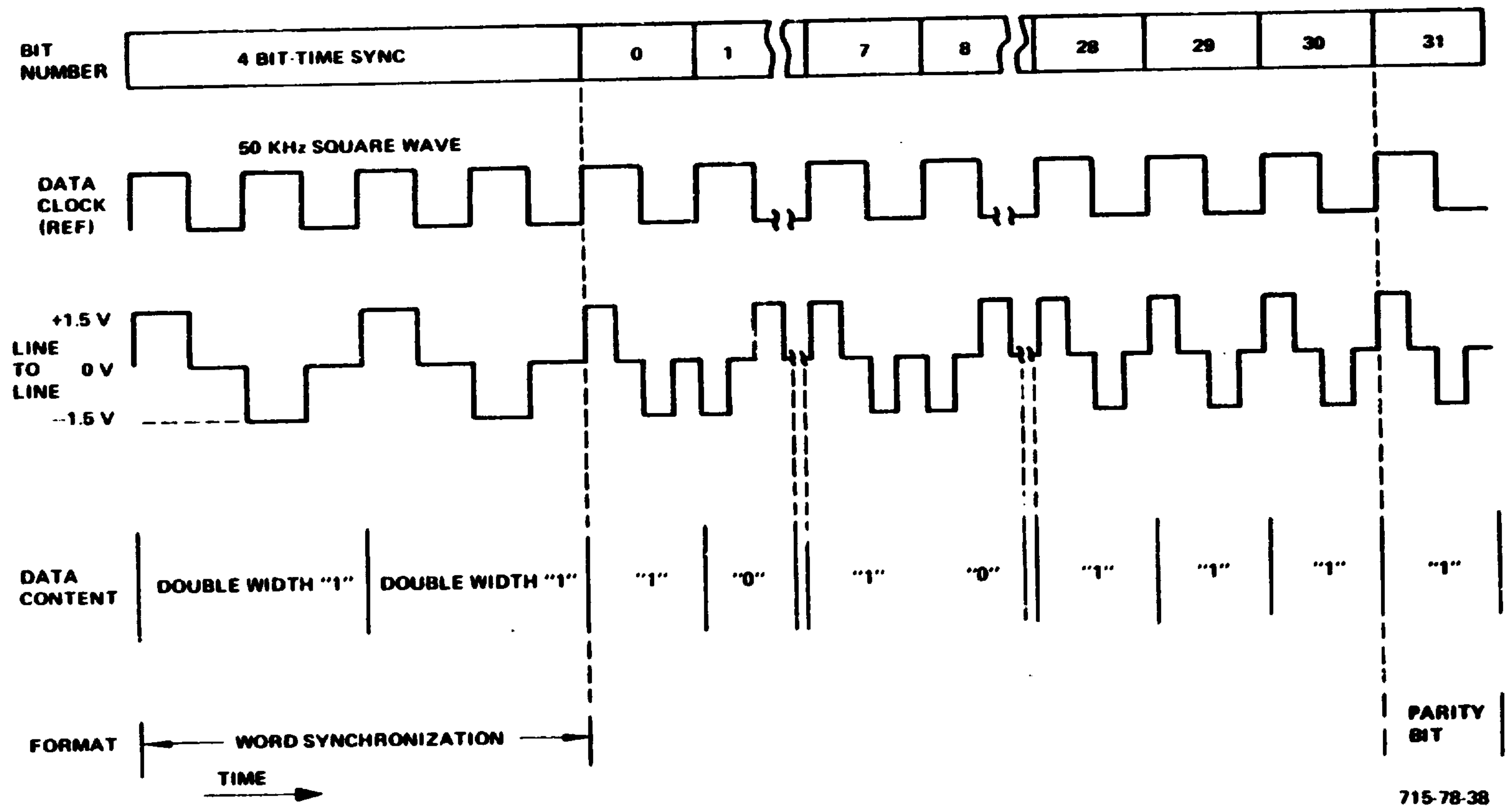


Figure 4-38
Basic Word Format and Split-Phase Bipolar Modulation

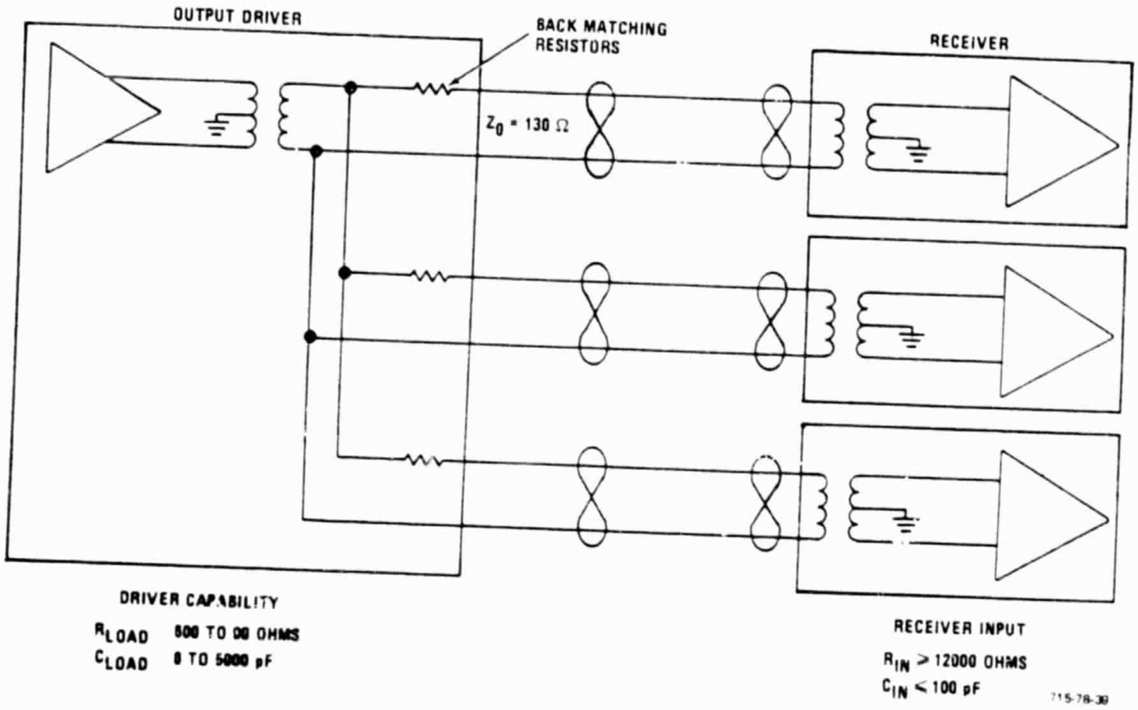
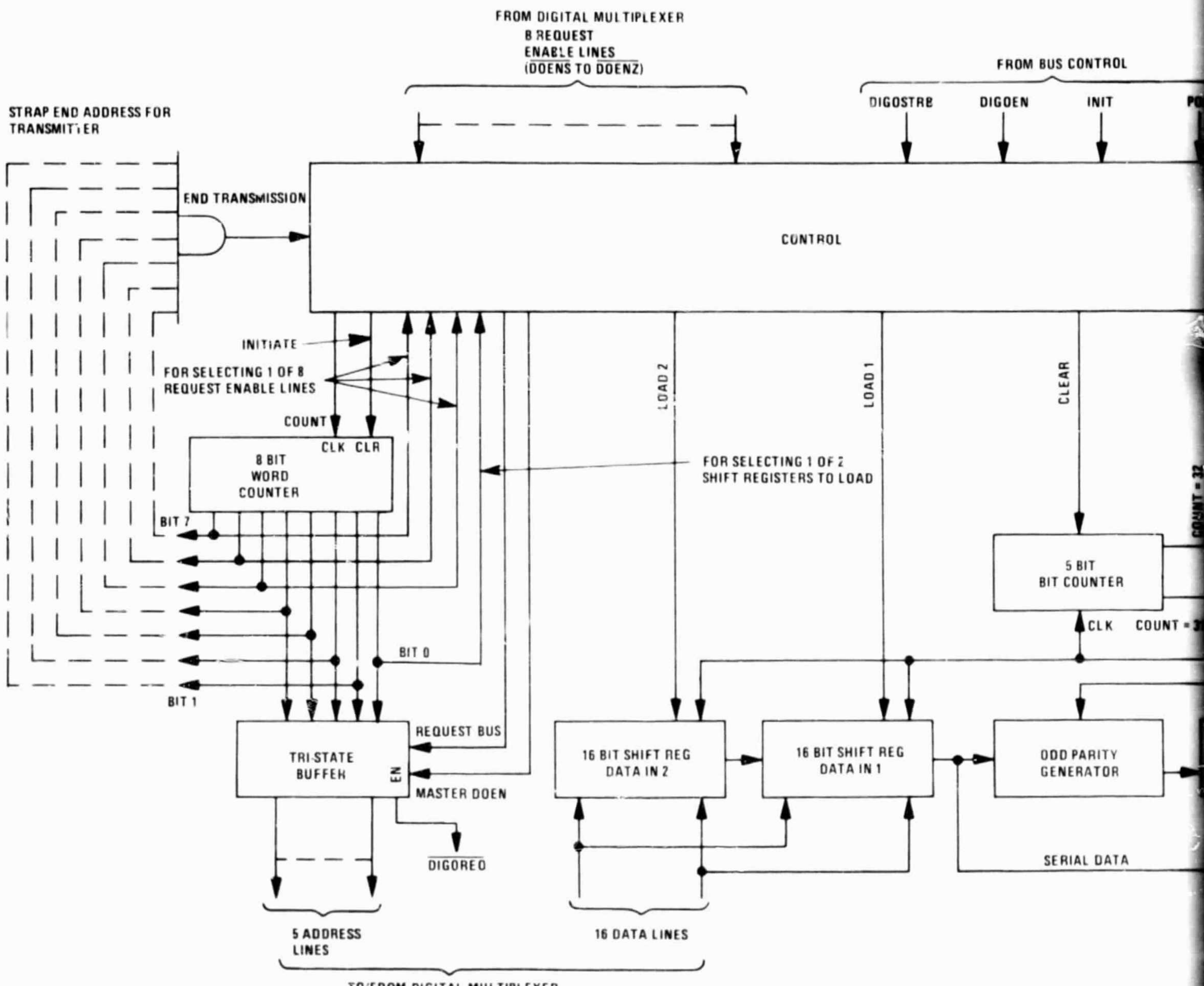


Figure 4-39
Split-Phase Bipolar Transmission Electrical Standards



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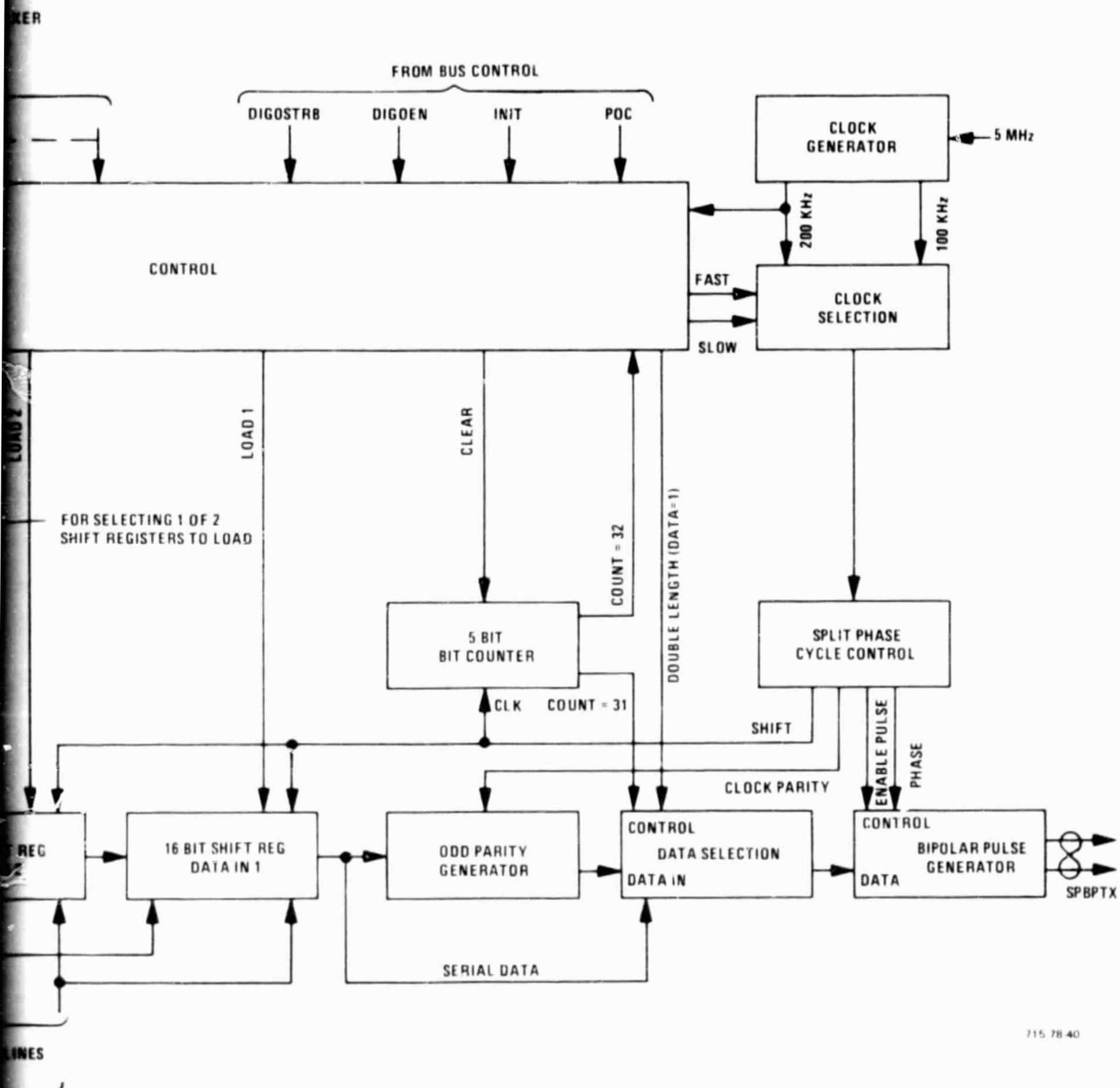

Figure 4-40
SPBP Transmitter Block Diagram

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Accelerometer Self-Test circuit is provided. The current generator uses an isolated low-voltage ac source to generate an isolated dc current. This current is switched by a relay network to apply the current to one of three accelerometer self-test ports.

An ac demodulator board provides eight separate demodulators plus four references for ac input signals. The demodulators are the sampling type, which eliminates ac output ripple under steady-state conditions.

The system logic board provides servo engage logic to engage the various servos and servo interfaces on the XV-15. Maximum versatility is achieved by using a combination of hardware and software interlocking. It also provides logic that drives the pilot and copilot warning lights. V/STOLAND WARNING lights annunciate servo-related failures while in RES or AUTO mode, and software-detected failures anytime. Failures are stored in a flip-flop until reset by the pilot or copilot by pressing either V/STOLAND WARNING light assembly or by depressing either V/STOLAND DISCONNECT switch.

Flap-select and power-lever interface circuitry drives the valves and solenoids associated with the power lever. A monitor circuit supplies a valid signal to the engage logic as long as the power lever is positioned within the proper limits. Also a flap-select switch interface provides the drive for the flap-selector mechanism motor and clutch solenoid. Four discrete outputs provide software control of flap position through the flap-select switch interface. The software commands are compared with the output of the flap-selector mechanism status switch to generate up and down commands to the motor drive relays.

Buffering of the dc outputs is provided for driving the Force-Feel System (FFS). These buffers generate a pseudo differential output from a single-ended output, and are necessary because the FFS analog inputs are single-ended.

The data adapter power supply operates from a +28-volt dc input and provides regulated +5 volts dc, +15 volts dc and -15 volts dc. Also provided are +28 volts dc and -28 volts dc unregulated. Power supply ratings are summarized in Table 4-4, and Figure 4-41 shows the data adapter power supply modules.

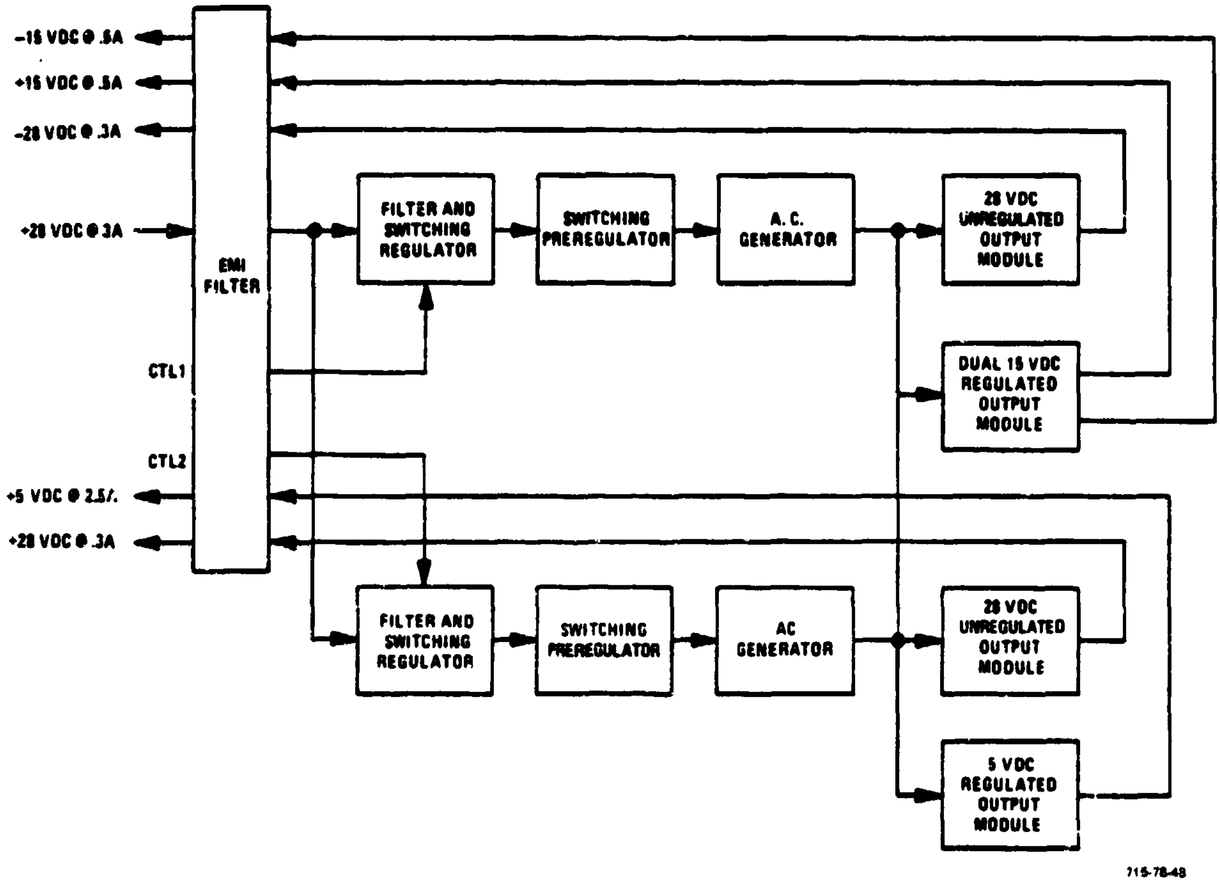

Figure 4-41
Power Supply Block Diagram

TABLE 4-4

POWER SUPPLY RATINGS

	Nominal Current (amperes)	Maximum Current (amperes)	Regulation (percent)	Ripple (millivolts)
+5 V dc	2.5	3.0	±3	50
+15 V dc	.5	1.0	±3	50
-15 V dc	.5	1.0	±3	50
+28 V dc	.3	1.5	±10	100
-28 V dc	.3	1.5	±10	100
28 V dc Input Power	3.0	10.0		

A power supply monitor section monitors each output voltage from the power supply and sets its output if any of these voltages exceed a preset deviation from nominal. Input aircraft power is also monitored and used to inhibit the setting of the BITE indicator if the input power is out of limits. A crowbar driver circuit provides the gate drive to a Silicon Controlled Rectifier (SCR) across each power supply output. In the event of the failure of one output, all outputs are shorted to ground by the SCRs. A power conditioner circuit supplies power to the power supply monitor and also maintains power long enough after aircraft power is removed to ensure proper circuit operation.

4.5 THE MODE SELECT PANEL

The Mode Select Panel (MSP), illustrated in Figure 4-42, is the primary control panel for engaging V/STOLAND modes. Except for the A/P ENG switch in the lower left corner, which is hardwired to enable/disable the autopilot inputs to the aircraft force-feel system, and the dim control in the lower right corner, all switches interface directly to the Basic computer so that their functions are completely under software control. The A/P ENG toggle switch is solenoid-latched, and is disengaged by software as well as by the hardwired monitoring logic. The five identical rotary switches have five positions, with spring return to the center position, and are programmed to provide slow and fast slew rates in each direction for the associated references.

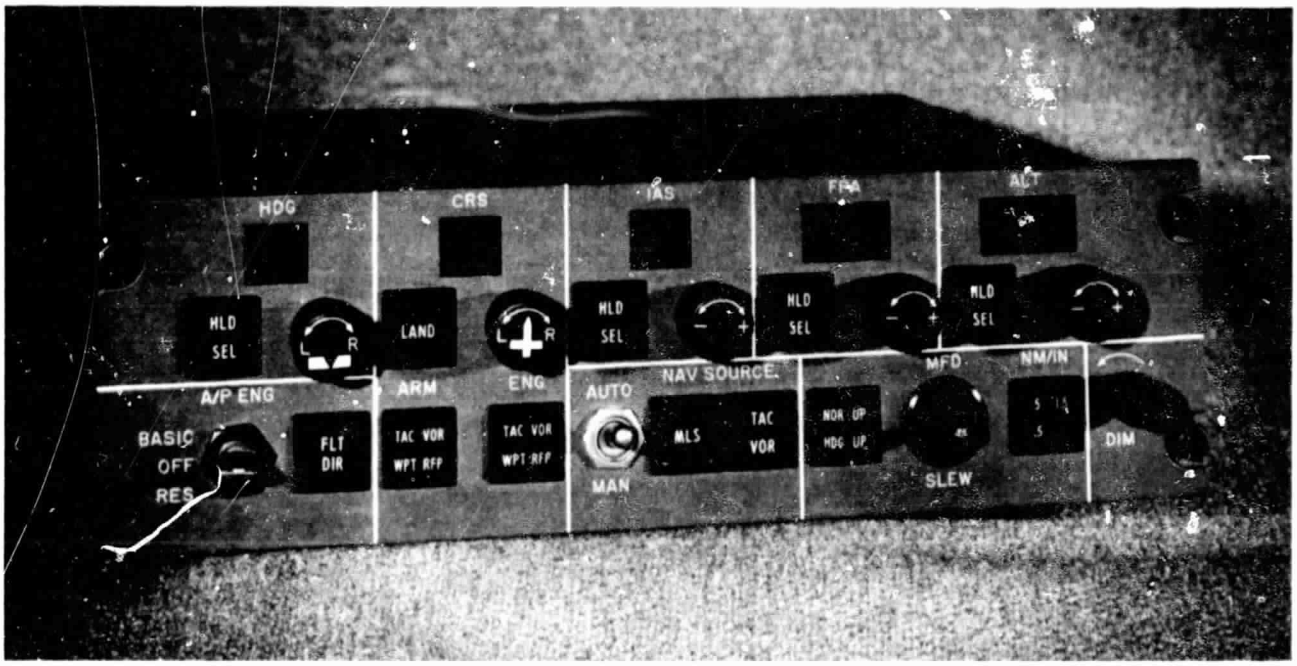

Figure 4-42
Mode Select Panel

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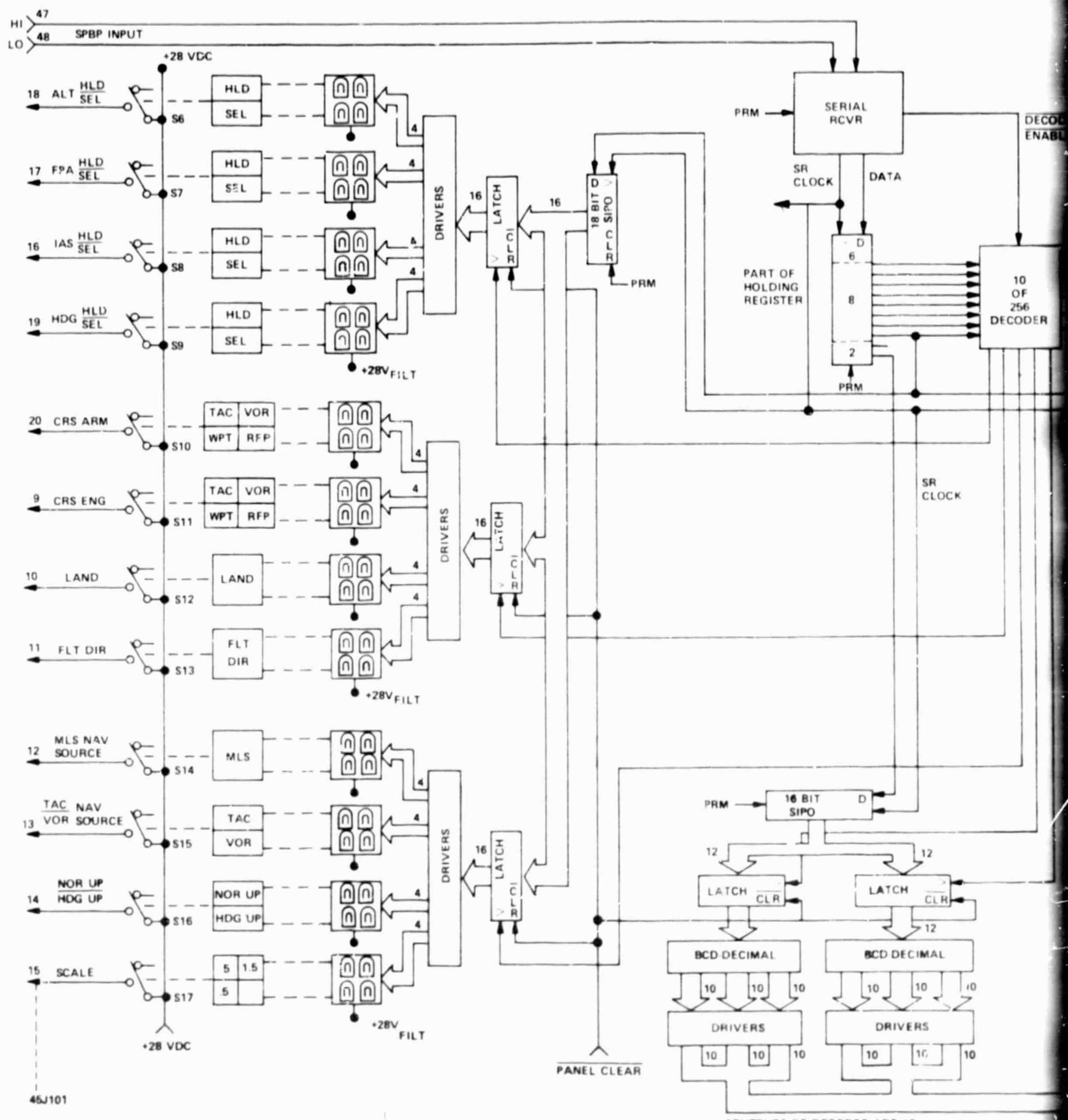
The MSP allows the pilot to engage the autopilot and/or flight director and to select the aircraft flight mode. In addition, the MSP is used to control the MFD display and the navigation source modes. The illuminated segments of the pushbuttons and the alphanumeric displays indicate the state of the V/STOLAND system. The illuminated pushbuttons indicate what flight modes are being armed, selected, held, or engaged, as well as the mode and scale of the MFD display and what navigation sources are providing valid data to the navigation computation. Since the operation of the MSP is primarily determined by software, the description of its operation is contained in Section V, "Description of System Software."

A block diagram of the Mode Select Panel is shown in Figures 4-43 and 4-44. A general overview of Figure 4-43 shows the basic scheme for receiving, decoding and routing input data to the proper storage/display device for annunciation. Figure 4-44 gives the basic layout for the internal power supply and unlighted switches.

The MSP electronics are contained on four plug-in circuit cards, using mostly CMOS logic for low power dissipation. The cards contain the following functions:

A1 - Receiver Logic and Decimal Data Storage/Drivers
A2 - ALT, FPA Numeric Display Storage/Drivers
A3 - Switch Lights Storage/Drivers
A4 - IAS, HDG, CRS Numeric Display Storage/Drivers

Basically, the MSP electronics drive the ALT, FPA, IAS, HDG and CRS lighted numerics and panel switch lights in response to information received via the SPBP input lines. Sixty decimal data lines (6 sets of 10 each) are also driven in response to serial input words. The decimal data lines can be used for driving external indicators.



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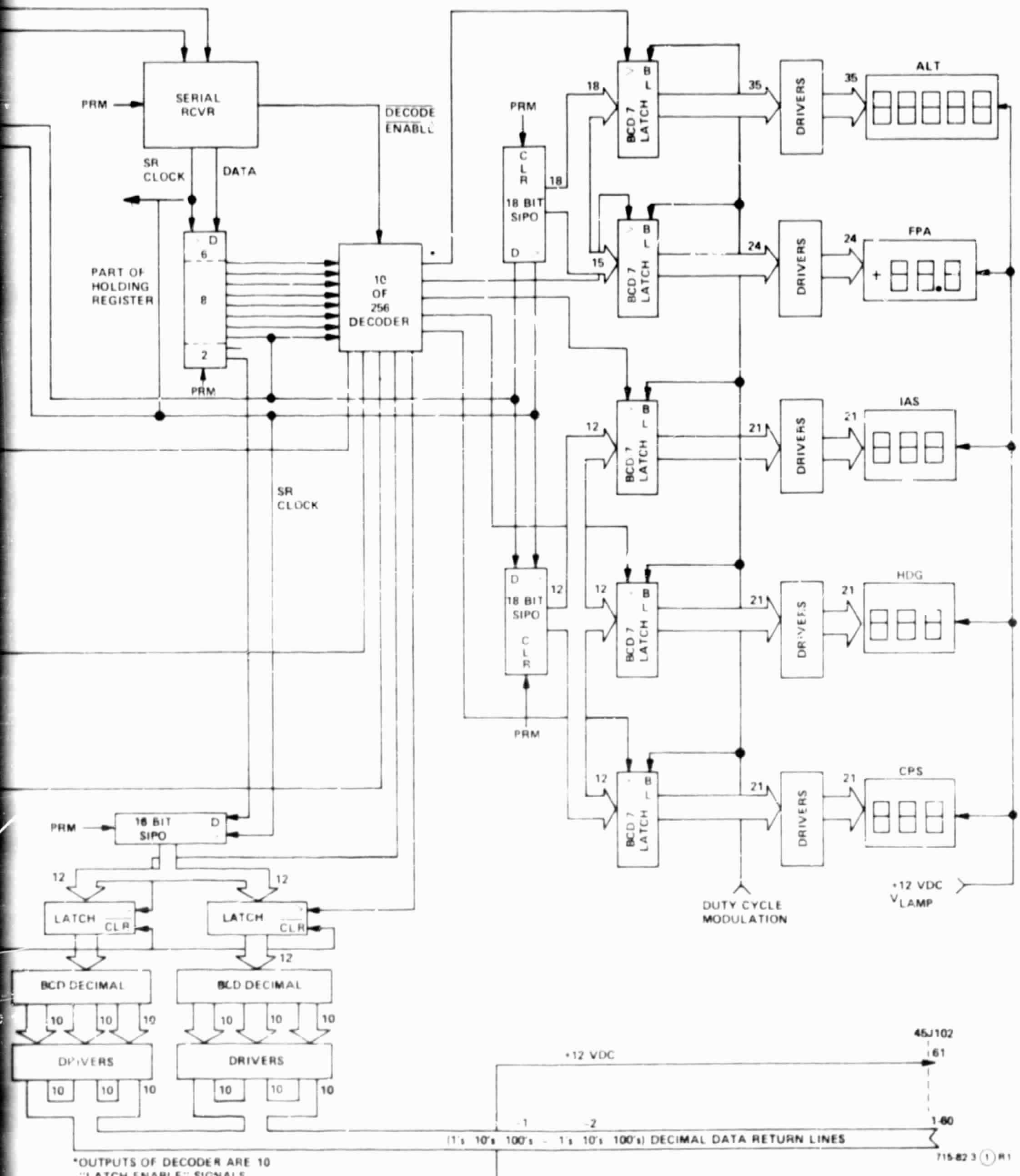
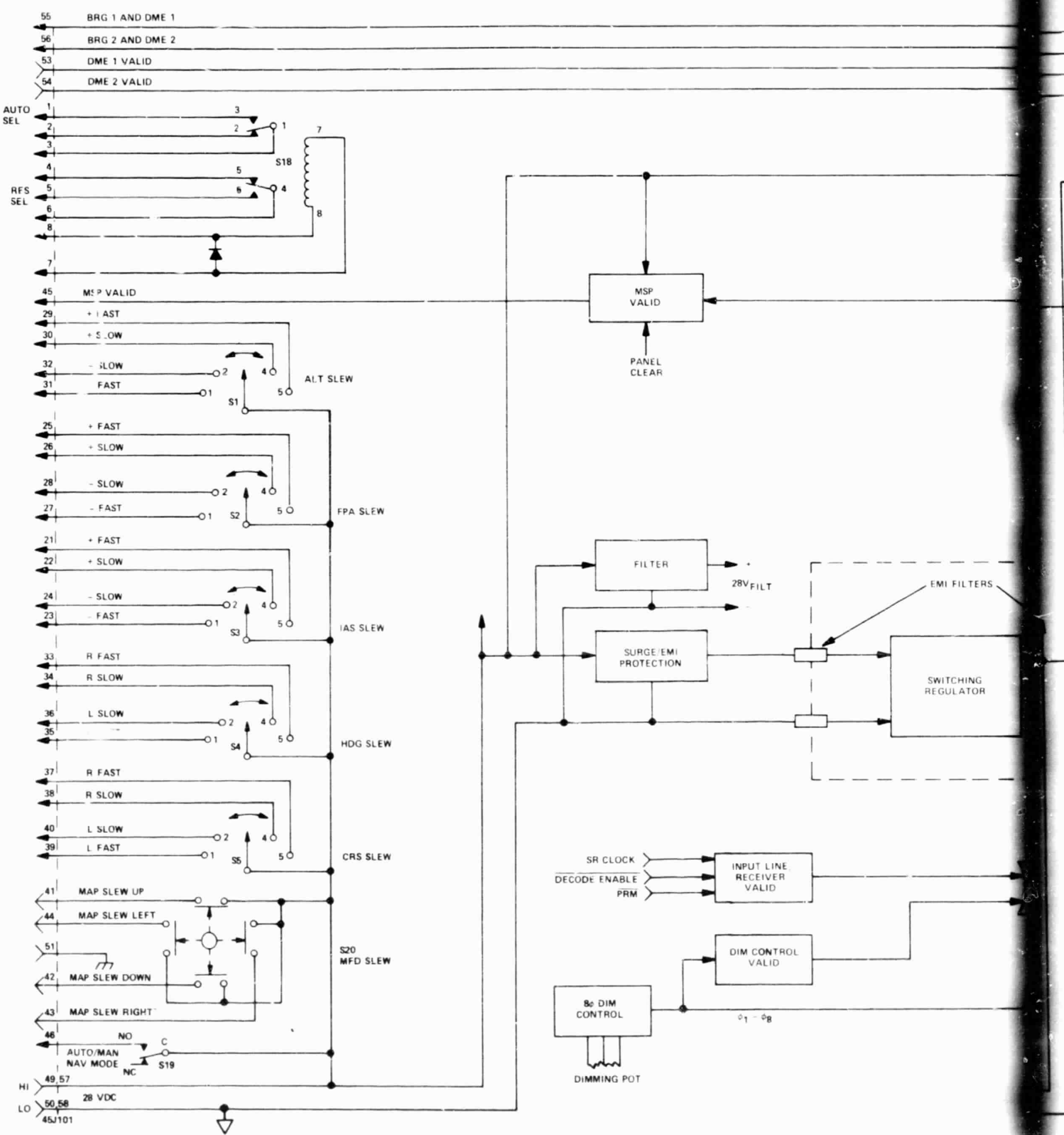
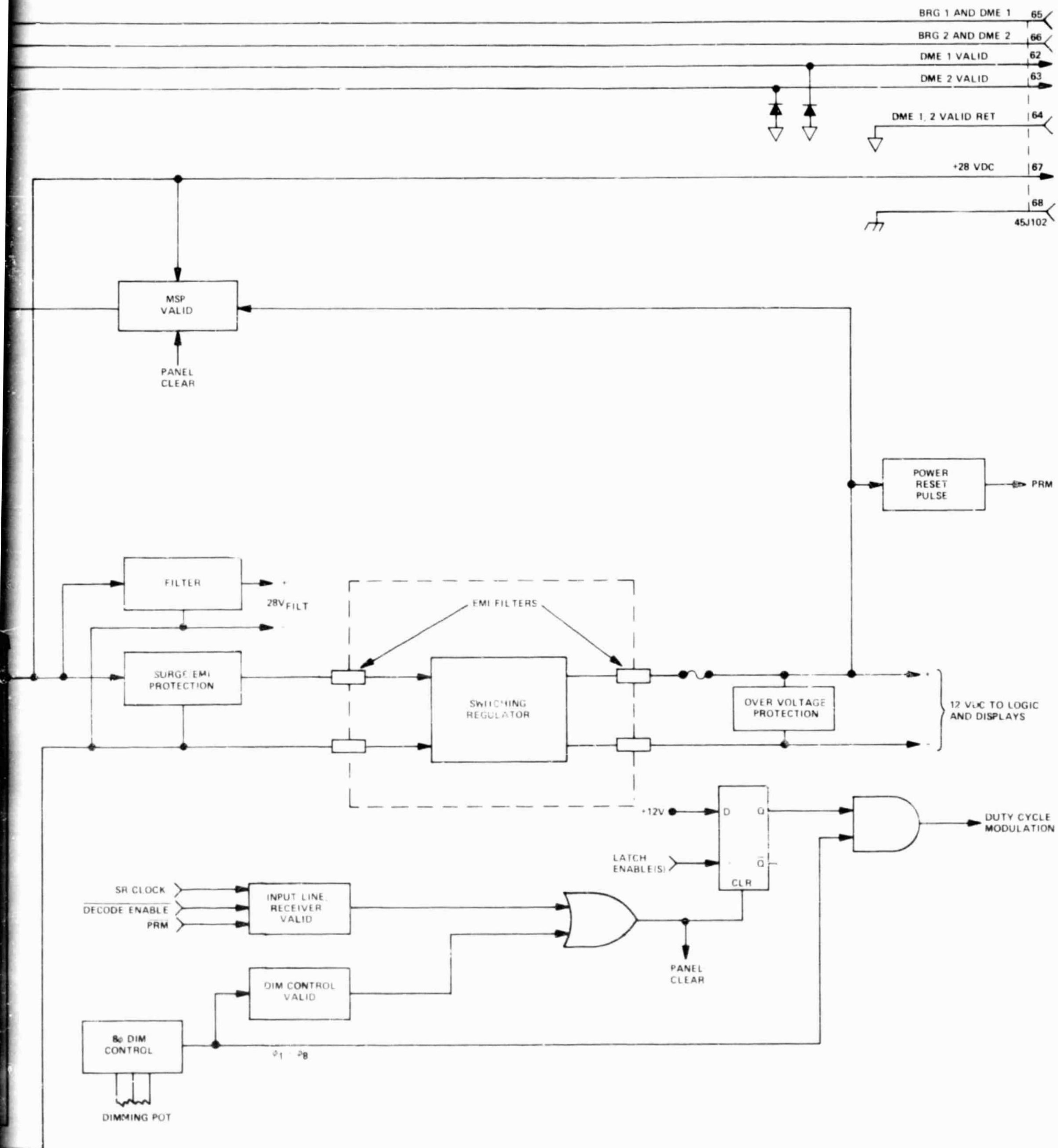


Figure 4-43
V/STOLAND XV-15
Mode Select Panel Block Diagram

FOLDOUT FRAME



FOLDOUT FRAME



715-82-3(2)R1

Figure 4-44
V/STOLAND XV-15
Mode Select Panel Block Diagram

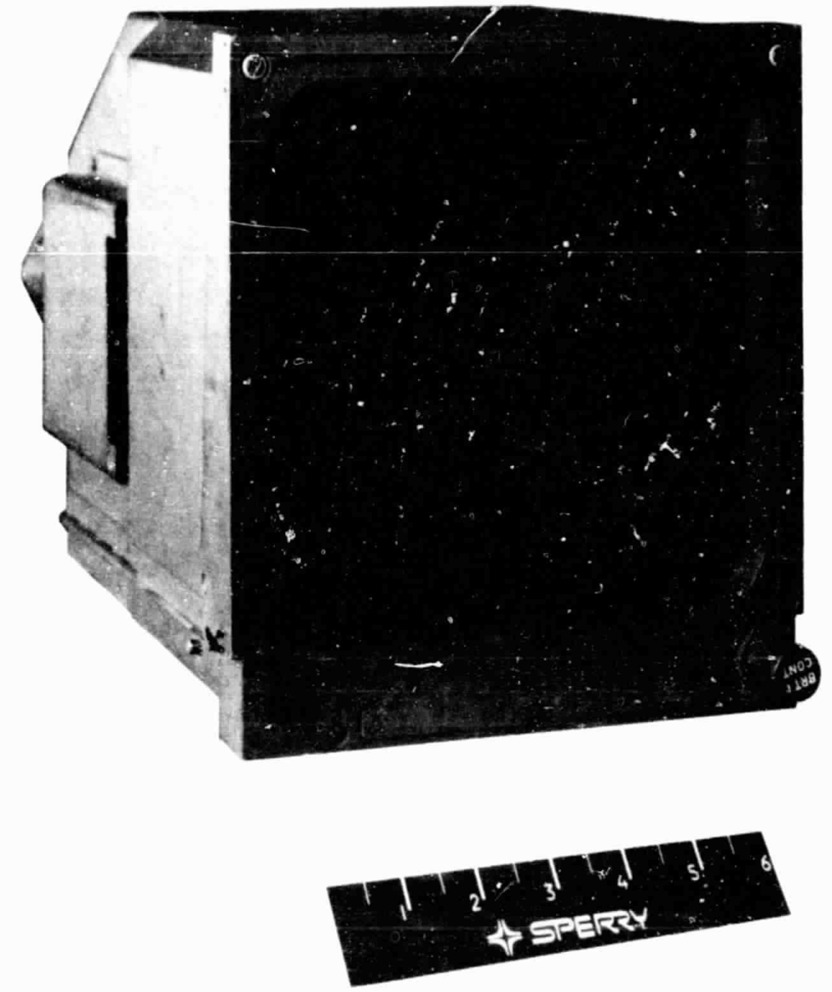
FOLDOUT FRAME

4.6 THE MULTIFUNCTION DISPLAY

The Multifunction Display (MFD) system is composed of the MFD Display Unit illustrated in Figure 4-45 and the MFD Symbol Generator illustrated in Figure 4-46. This system is capable of a broad range of graphic and alphanumeric displays, but is specifically intended and programmed (in the Basic computer) to provide mainly horizontal situation information. The stroke-written MFD Display Unit displays a map based on a data stream from the Basic computer (or from the Research computer if the Research MFD mode is selected) to the MFD Symbol Generator. The Symbol Generator decodes and transforms this data, representing lines, alphanumeric characters, map symbols, aircraft position and heading, etc, to x-y deflection and video signals for the MFD Display Unit.

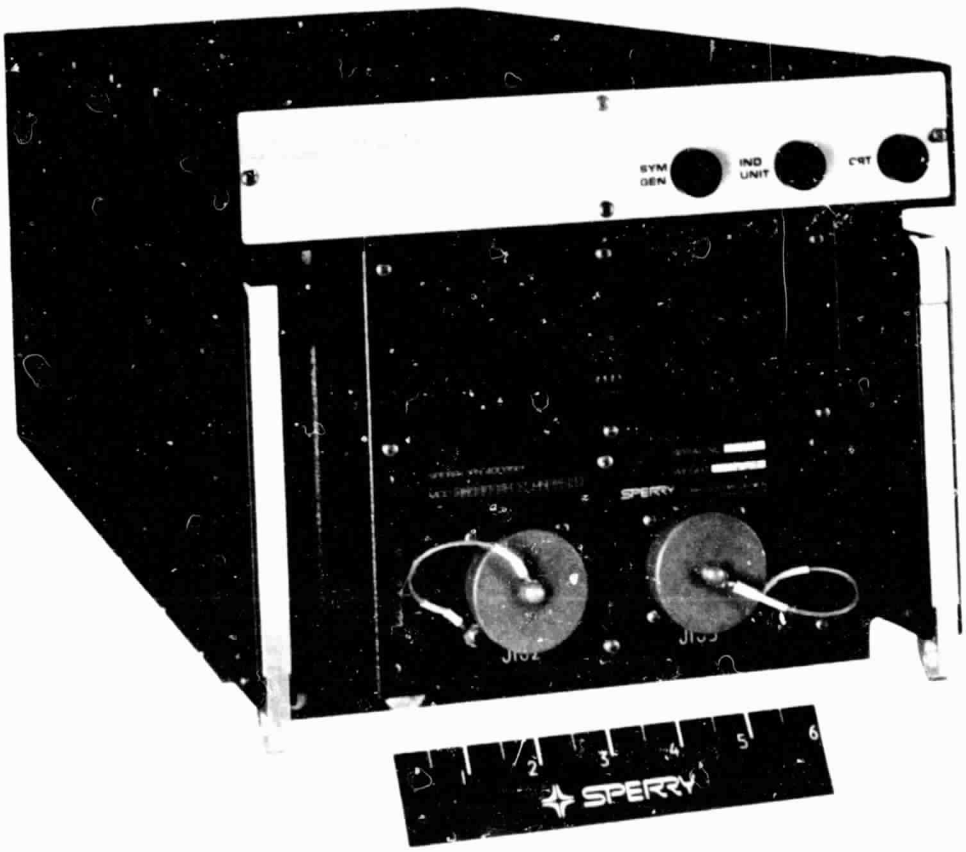
Under control of the supplied MFD software, the MFD screen displays an aircraft symbol on a topographical map. The position of the aircraft symbol on this map shows the horizontal situation of the aircraft with respect to the topographical features. In addition, the MFD displays future track, past track, waypoint, course vector, sea altitude, time of day, heading, navaid course line, MLS localizer, and topographical and navigational data. The MFD map display may be shown in two formats (heading-up or north-up), and in three scales (5, 1.5, and .5 nautical miles per inch). Also, the map may be slewed to convenient viewing positions, ad a map warning light annunciates the presence of invalid data.

The MFD map covers an area of 10,000 square nautical miles as shown in Figure 4-47. The center of this map is approximately 15 nautical miles west of Crow's Landing, California. An MFD display in the heading-up format is shown in Figure 4-48.



718-51-13

Figure 4-45
The MFD Display Unit



718 51-14

Figure 4-46
The MFD Symbol Generator

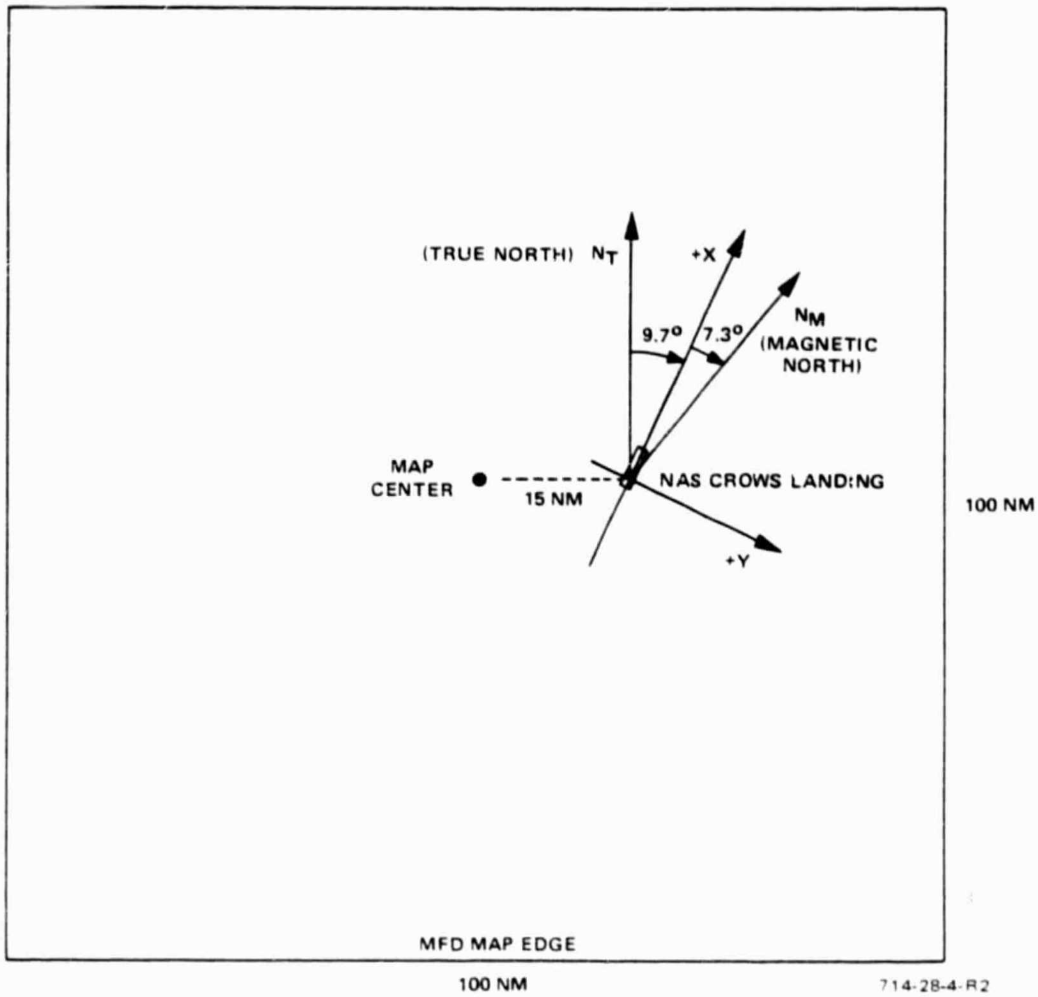


Figure 4-47
Total Navigable Map of MFD

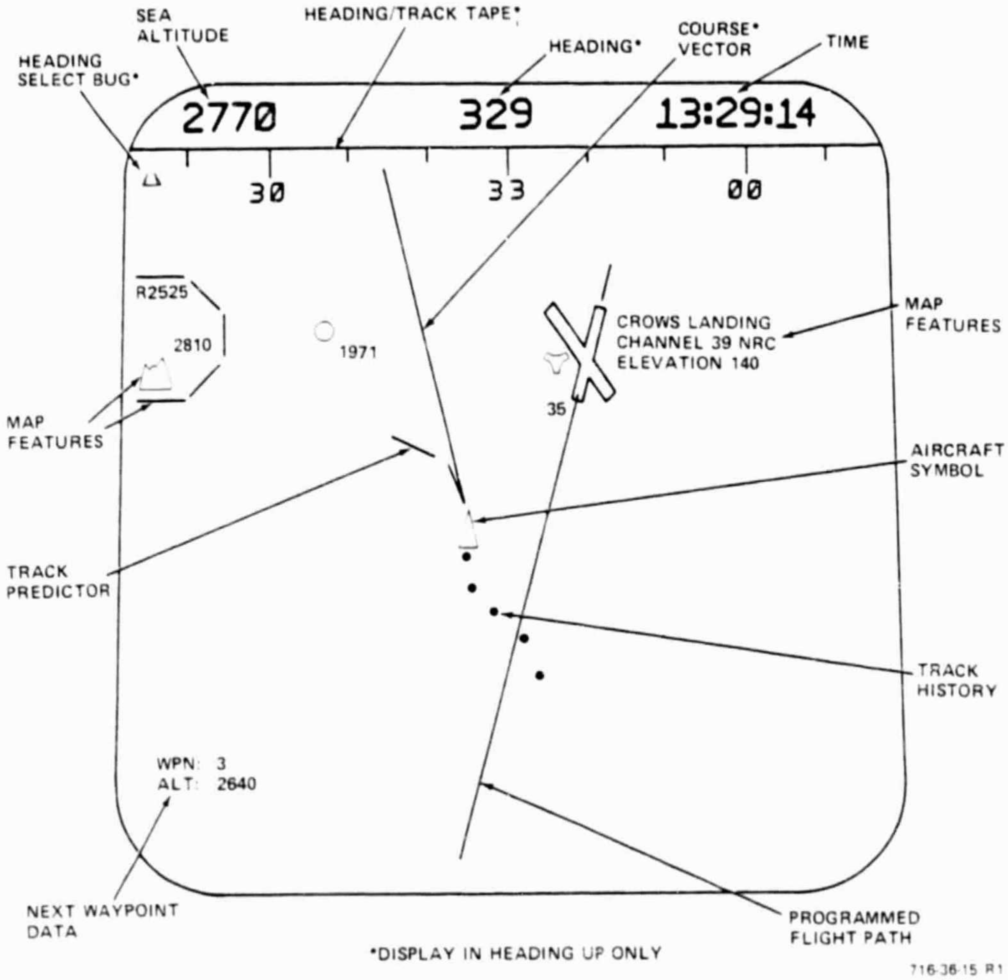


Figure 4-48
Typical Heading-Up MFD Display

When the V/STOLAND system is initially powered, the MFD is initialized in the north-up display mode, at 5 nautical mile/inch scale and with the map window centered near the Crow's Landing airport. The MFD map is composed of distinctive symbols for navaids, airports, restricted areas, mountains, and other obstacles. The symbology used is as follows:

MAP FEATURE	SYMBOL	MAP FEATURE	SYMBOL
VORTAC		WATER TANK AND HEIGHT	1971
VOR		RESTRICTED AREA	R 2525
DME		LANDING SITE SYMBOL	CROWS LND
WAYPOINT		TACAN CHANNEL 39NRC	CHAN 39NRC
MOUNTAIN		RUNWAY ELEVATION	ELEV 140

714-28-26

In addition to the map display, the MFD also shows several other types of data via other types of displays as described in the subsections which immediately follow. A sample of such a display is shown in Figure 4-48, and the following paragraphs describe the features.

• Aircraft Symbol - The aircraft symbol is displayed on the MFD as an isosceles triangle pointing in the direction of the aircraft heading. This symbol appears continuously (non-flashing) when there is a valid navigation data source (TACNAV, VOR, or MLS) from which the aircraft position may be computed. The symbol flashes when the V/STOLAND navigation computations go to the dead-reckoning mode as a result of a loss of valid navigation data. When no valid navigation data is available and the dead-reckoning period is exhausted, the aircraft symbol is not displayed. In the nortn-up map mode, the aircraft symbol moves relative to a map fixed with respect to the MFD CRT screen. In the heading-up mode, the aircraft symbol is stationary, pointing up on the MFD CRT screen, and the MFD map rotates and translates.

• Track Predictor - The track predictor consists of two-line segments that appear in front of the aircraft symbol, indicating the predicted horizontal flight path for the next 40 seconds. The length of, and space between, each segment indicates 10 seconds. The prediction is based on the current rate of turn and aircraft velocity. The track predictor is displayed whenever the keyboard mnemonic MTR is set non-zero: otherwise, this display will not be shown. Upon system power-up, this display is enabled.

- <u>Track History</u> - A series of dots tracking previous positions of the aircraft symbol on the MFD represent the aircraft flight-path history for the immediate past 90 seconds. Each dot represents the aircraft position at 10-second intervals. The track-history dots are displayed whenever the keyboard mnemonic MHI is set non-zero: otherwise, this display will not be shown. Upon system power-up this display is enabled.

- <u>Course Vector</u> - In the heading-up mode only, the direction of the aircraft course is displayed by a straight line emanating from the aircraft and intersecting one of the screen boundaries. The course vector display is shown in the MFD heading-up display mode only whenever the keyboard mnemonic MCV is set non-zero: otherwise, this display will not be shown. Upon system power-up, the MCV mnemonic will be set to zero, thereby disabling this display.

- <u>Sea-Level Referenced Altitude</u> - The aircraft altitude referenced to sea level and resolved to the nearest 10 feet is displayed in the upper left corner of the MFD screen. This display is unaffected by scale, slew, or mode of display selections.

- <u>Time of Day</u> - Real time in hours, minutes, and seconds, derived from the time-code generator aboard the aircraft, is displayed in the upper right corner of the MFD screen. This display is unaffected by scale, slew, or mode of display selections.

- <u>Heading Tape</u> - In the heading-up mode only, the heading tape is displayed across the top of the MFD display, centered on and extending to 50 degrees either side of the current heading. The tape is marked in 5-degree increments, with 2-digit heading numbers at every 30-degree increment. This display is unaffected by scale or slew changes.

- <u>Heading</u> - In the heading-up mode only, the aircraft heading is displayed as a three-digit number in a window at the top center of the MFD display. This display is unaffected by the scale or slew changes.

• Heading Select Bug - In the heading-up mode only, a heading select bug tracks the heading reference in exactly the same manner as described for the operation of the heading-select cursor on the HSI. It is displayed slightly below the heading tape if the selected heading reference is within ±50 degrees of the current heading. The heading select bug display is unaffected by scale or slew changes.

• Reference Flight Path - If the reference flight path or helix land (LAND-2) course modes are armed or engaged, the reference flight path will be shown on the MFD. Figure 4-49 shows the reference flight path and helix land displays. The reference flight path consists of four waypoints: two straight-line segments, and two semicircular segments forming a closed loop. The reference flight-path loop is positioned so that the straight-line segment between Waypoints 3 and 4 coincides with the center line of Crow's Landing Runway 35.

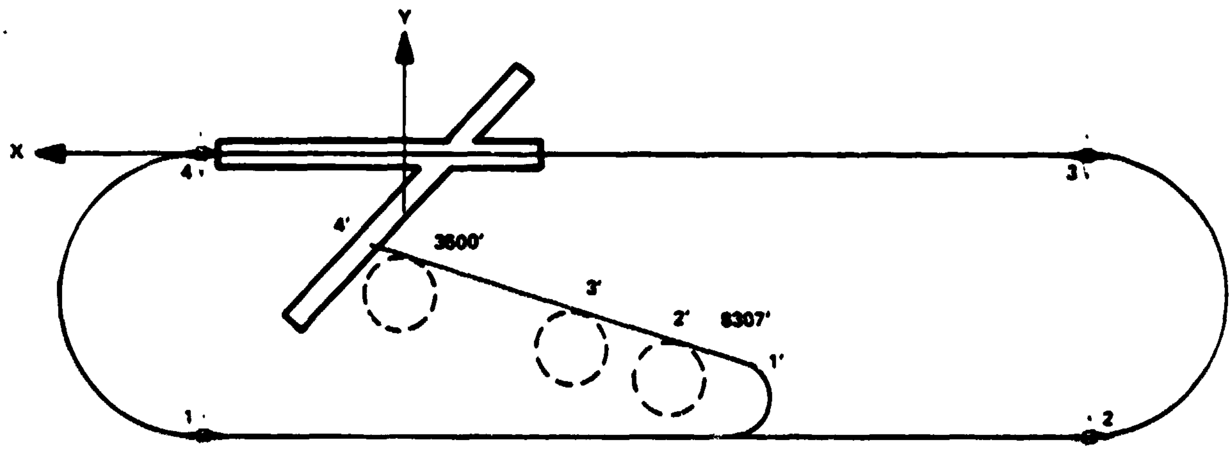
• Next Waypoint Data - When the reference flight-path course mode is armed or engaged and the keyboard mnemonic MFP is set non-zero, the next waypoint number and its altitude (above sea level) will be displayed in the lower left corner of the MFD in the following format.

WPN = X

ALT = XXXX

As the aircraft passes the currently displayed waypoint on the reference flight path, the next waypoint number and its altitude are displayed. The next waypoint data display may be deleted by zeroing the mnemonic MFP via the keyboard. Upon system power-up, this display is enabled. This display is unaffected by scale or slew selections.

• MLS Localizer Line - If the straight-in land course mode (LAND-1) is armed or engaged, the MLS localizer line is displayed on the MFD as a line segment extending from the straight-in land touchdown point on Crow's Landing Runway 35 south to the boundary of the MFD map displayed on the screen. This line corresponds to the course reference for straight-in land approaches made by V/STOLAND. The MFD will display this line even when the touchdown point is not visible in the area of the MFD map currently in view. In this case, the MLS localizer line will be extrapolated into the area of the map being displayed.



NOTES: 1. REFERENCE FLIGHT PATH IS DEFINED BY THE WAYPOINTS 1, 2, 3 AND 4.
2. LAND MODE 2 PATH DEFINED BY THE WAYPOINTS 1', 2', 3' AND 4'. A HELIX IS DEFINED AT THE WAYPOINT 3'. THE HELIX IS SELECTABLE IN TERMS OF THE DISTANCE TO THE TOUCHDOWN POINT 4' WITHIN THE LIMITS OF 3600' AND 8307'. WAYPOINT 2' DEFINES THE BEGINNING OF THE -6.11° GLIDE SLOPE TO THE TOUCHDOWN POINT.

716-11-83

Figure 4-49
MFD Reference Flight Path and Helix Land Displays

• Helix Land Path - The helix land path will be shown with the reference flight path whenever the helix land course mode (LAND-2) is armed or engaged. Figure 4-49 shows this display. The helix land path consists of a circular segment tangent to the reference flight path and to the 18° course line leading to the LAND-2 touchdown point on the Crow's Landing Runway 30, a straight segment between this latter point of tangency and the LAND-2 touchdown point, and a circle which represents the helix tangent to this line. The circle representing the helix of the LAND-2 flight path may be moved along the straight-line segment when the pilot selects the distance of the point of tangency of the helix from the LAND-2 touchdown point via the keyboard mnemonic HLX. The helix may be placed from 3600 feet to 8307 feet from the LAND-2 touchdown point. Upon system power-up, this distance is set to 5000 feet.

• Waypoint - If the waypoint course mode is armed or engaged, the location of a selectable waypoint, with coordinates set via keyboard mnemonics WPX and WPY, is shown on the MFD map by a waypoint star symbol and the letters WPT printed nearby. Until otherwise selected by the pilot, the coordinates of this waypoint will be at the initial values of

$$X = +30{,}000 \text{ feet}$$

$$Y = -20{,}000 \text{ feet}$$

• Navaid or Waypoint Course Lines - If a radial course mode is armed or engaged, the MFD will display the reference course vector on the MFD screen. The course vector will be displayed as a line passing through the appropriate navigation station or waypoint, oriented along the course azimuth, and having an arrow associated with its direction. The navigation station or waypoint position does not have to be included in the area currently in view on the MFD. In such cases, the course line will be extrapolated into the screen area if it crosses this area.

If a radial course mode (TAC, VOR or WPT) is armed while another is engaged, the navaid course line will pass through the armed course mode station, taking priority over the engaged mode. The direction of the navaid course line displayed on the MFD will correspond to the azimuth shown by the course-select pointer on the HSI.

• Map Orientation Modes - The MFD map may be displayed in two orientations: north-up or heading-up. In the north-up mode the aircraft symbol moves relative to the displayed MFD map which is fixed with respect to the display screen. In the heading-up display mode the aircraft symbol is stationary relative to the display screen, pointing up, and the displayed MFD map rotates and translates with respect to it. The orientation mode is selected via the NOR UP/HDG UP button on the MSP.

• Map Scale - In either orientation mode (north-up or heading-up), the MFD map may be displayed in three scales: 5, 1.5, or .5 nautical miles per inch. Upon system power-up, the MFD map will be displayed with a scale of 5 nautical miles per inch. Pushing the scale button on the MSP causes the scale to change cyclically in the order 5, 1.5, .5, and the associated segment on the button to light green.

• Map Slewing - The MFD map may be slewed by the five-position, return-to-center switch labeled MFD SLEW on the MSP. The map will move in the direction of the slew switch at 2 inches per second. Hence, if the map is displayed with a scale of 1.5 nautical miles/inch the map will slew at 3 nautical miles/second. Similarly, MFD map distance slew rates for the 5 and .5 nautical miles/inch scales will be 10 and 1 nautical miles/second, respectively. If an attempt is made to slew the map beyond the map limits (100 nautical mile square area as shown in Figure 4-47), a dotted line with the word CUT is shown at the map limit, and further slewing is inhibited.

• Map Warning Light - A map warning light is located in the bottom center portion of the MFD display unit on the bezel. When illuminated, the light is red and displays the word MAP. When not illuminated, the word MAP is generally not visible. The MFD map warning light is illuminated under any of the following three conditions:

- The V/STOLAND failure monitoring software module detects a failure in the MFD Symbol Generator via the Symbol Generator Valid.
- The V/STOLAND failure monitoring software module detects a failure in the MFD Display Unit via the Display Unit Valid.
- The self-test button on the MFD display unit is pushed.

When the MFD map warning light is illuminated, the pilot is to disregard the MFD map data displayed.

• Self-Test Button - The MFD display unit is equipped with a self-test button mounted on the lower left corner of the instrument on the bezel. When this button is pressed, the test pattern shown in Figure 4-50 will appear if both the MFD Display Unit and Symbol Generator are operating correctly.

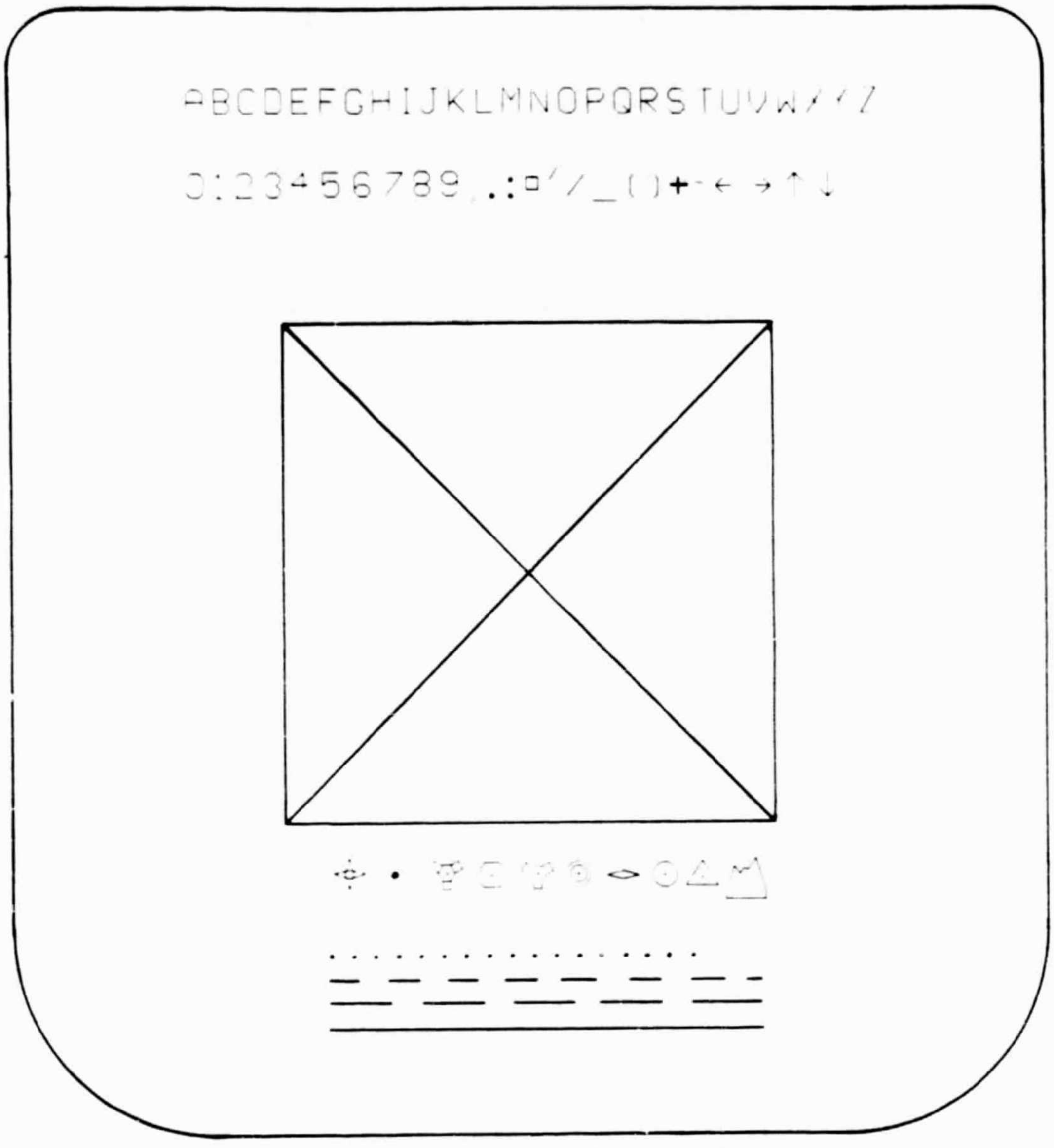
4.7 THE ATTITUDE DIRECTOR INDICATOR

The Attitude Director Indicator (ADI) illustrated in Figure 4-51, displays attitude, flight director commands, vertical deviation (glide slope), course deviation (localizer), radio altitude and rate of turn. The ADI also has Decision Height (DH), Flare (FLR) annunciation lights and failure warning flags for the vertical deviation (GS), and the flight director (FD).

Pushing the TEST button causes the pitch attitude to change (from the current attitude) by +10 ± 5 degrees (nose up), the roll attitude to change by +20 ± 5 degrees (right bank), and the attitude flag (G) to come into view.

The attitude display is driven directly from the pitch and roll outputs of the vertical gyro (VG). The attitude flag (G) comes into view if the VG or ADI power is lost. The pitch scale sensitivity on the sphere is electrically expanded to provide approximately .070 inch per degree at zero pitch attitude, progressively decreasing to approximately .030 inch per degree at 90 degrees pitch attitude. The roll scale has a one-to-one relationship with aircraft attitude. The attitude sphere has approximately ±85 degrees of range in pitch, and full freedom in roll.

The rate-of-turn indicator is driven by an input from the yaw rate gyro. The deflection of the pointer to two pointer widths represents 5 degrees per second (= 2 mA). It is possible to drive the pointer to four pointer widths to represent 10 degrees per second before it goes out of view.

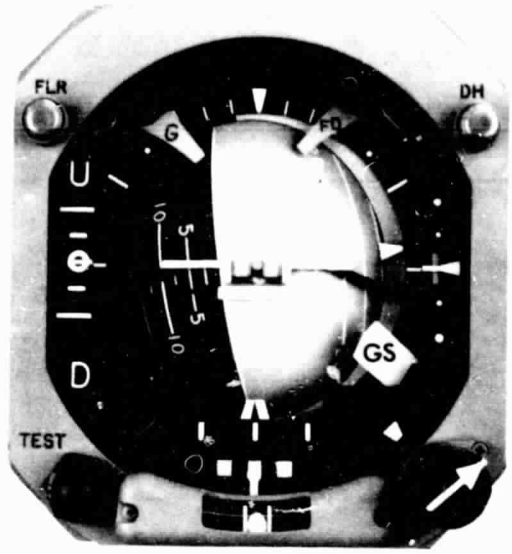


NOTES: 1. THE CROSS INTENSITY SHALL BE DIMMER THAN THE BOX

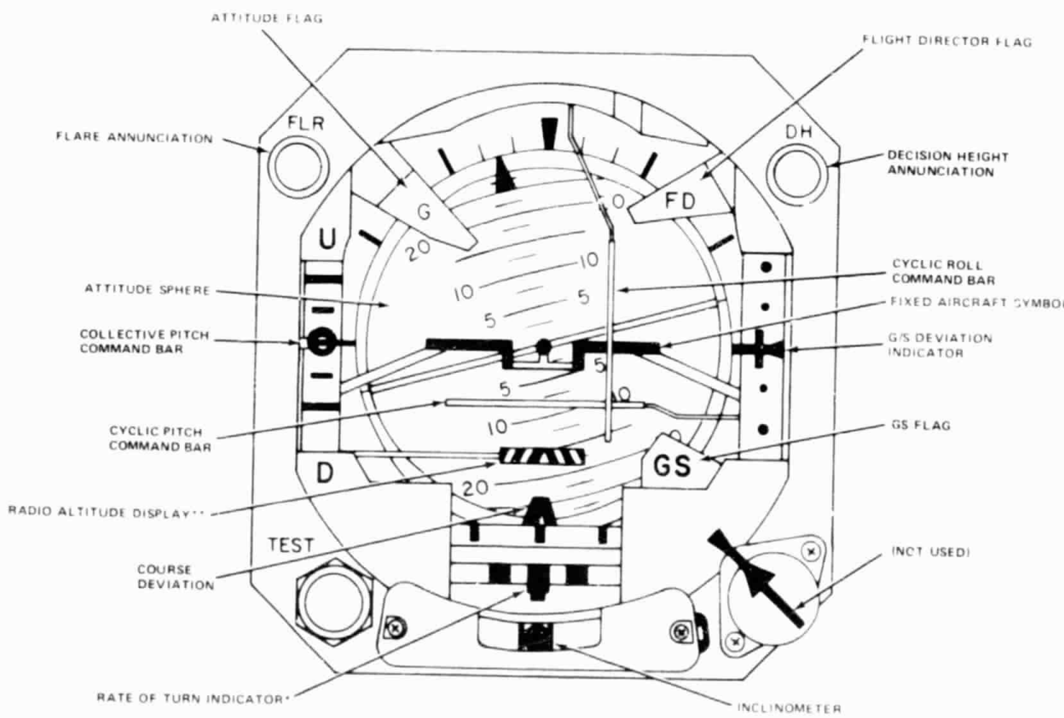
2. THE BOX INTENSITY SHALL BE DIMMER THAN THE CHARACTERS.

715-14-12

Figure 4-50
MFD Test Pattern



718-51-16



NOTES * INPUT FROM ROLL/YAW RATE GYRO ASSEMBLY
** INPUT FROM RADIO ALTIMETER

716 36 7

Figure 4-51
The Attitude Director Indicator

ORIGINAL PAGE IS OF POOR QUALITY

The radio altitude display is driven directly from the radio altimeter. It is displayed by a horizontal bar that moves from the area of the expanded localizer (200-foot altitude) to the bottom of the miniature aircraft symbol (touchdown).

The remaining displays are based on software data, and the following paragraphs describe operation under the supplied Basic computer software.

• Flight Director Commands - When the flight director is engaged by pushing FLT DIR on the MSP, but A/P is off, the three flight director bars give the pilot command instructions for the cyclic and collective sticks. For a pitch-up command, the pitch command bar moves above the aircraft symbol, instructing the pilot to pull back on the cyclic stick until the bar is centered. Similarly, for a roll-right command, the roll command bar moves right, instructing the pilot to move the cyclic stick to the right until the bar is centered. For a positive-up altitude rate command, the collective command bar moves up, instructing the pilot to pull up on the collective stick until the indicator is centered.

If A/P and FLT DIR are both engaged, the flight director bars monitor the performace of the autopilot. The guidance modes under the flight director are identical to those under the A/P modes.

If research guidance modes are selected, the commands generated in the Research computer are displayed by the command bars. If FLT DIR is not selected at the MSP, the pitch, roll, and collective command bars are biased out of view.

• Vertical Deviation - Vertical deviation is displayed on the right vertical scale when the REF FP or LAND guidance mode is selected. In all other modes, the indicator is biased out of view. In the REF FP mode the full-scale deflection from center represents 500 feet (= 150 microamperes = 2 dots) above or below the desired path. In the LAND modes the full-scale deflection from the center represents 100 feet (= 150 microamperes = 2 dots) above or below the reference glide slope. In the ILS LAND mode, with both AUTO and Flight Director disengaged, the vertical deviation indicator displays raw angular deviation data where full-scale deflection represents .7 degree (= 150 microamperes = 2 dots).

• Course Deviation - Course deviation is displayed (on the indicator conventionally used for expanded localizer) when a radial or LAND guidance mode is selected. In all other modes, it is biased out of view. In the LAND modes full-scale deflection from the center represents 100 feet (= 20 microamperes) to the right or left of the runway centerline. In the VOR/TACAN modes, using raw data, the full-scale deflection represents ≃ 3.3 degrees. In all other guidance modes full-scale deflection from the center represents 1000 feet (= 20 microamperes) to the right or left of the desired path.

• Flare Annunciation - When the flare mode is engaged, it is indicated by green annunciation marked FLR in the upper left corner of the HZ-6F.

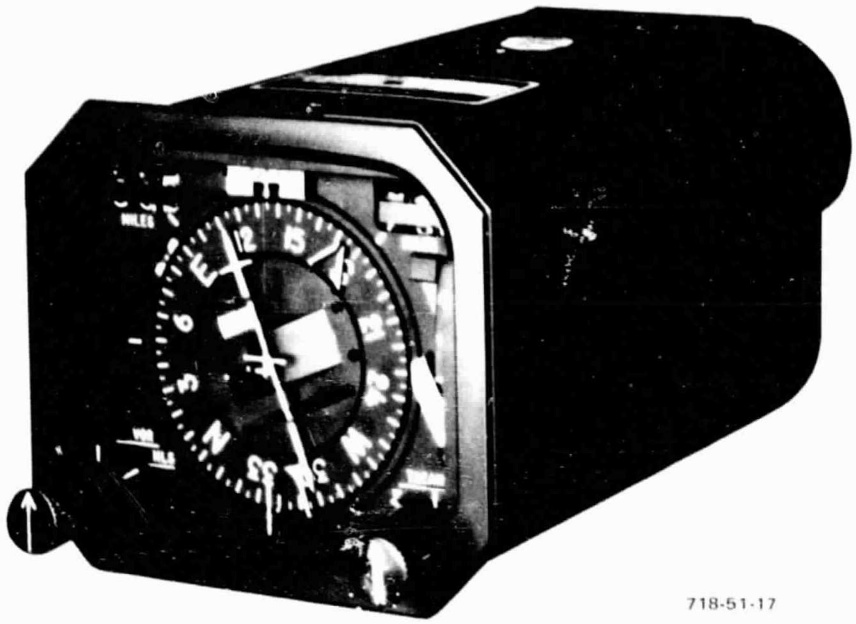
• DH Annunciation - The decision height (DH) annunciator in the upper right corner of the HZ-6F indicates amber when the aircraft altitude is less than or equal to the decision height. This height is normally set to 100 feet (above the runway), but may be changed by the pilot via the keyboard with mnemonic DHT.

• Flight Director Flag - The flight director flag is not used during V/STOLAND operation since the flight director cannot be engaged if there is a failure. It only comes into view when the system power is off.

• GS Flag - The GS flag comes into view when the navigation during MLS becomes invalid. It is biased out of view for all other modes.

4.8 THE HORIZONTAL SITUATION INDICATOR

The Horizontal Situation Indicator is illustrated in Figure 4-52. It combines ten radio and compass navigational displays in a single 4 by 5-inch instrument, and permits integration of the complete horizontal situation display into the central scanning area. The radio navigational displays include two independent servo-driven radio bearing pointers, two independent distance measuring equipment readouts, vertical deviation indication, course deviation



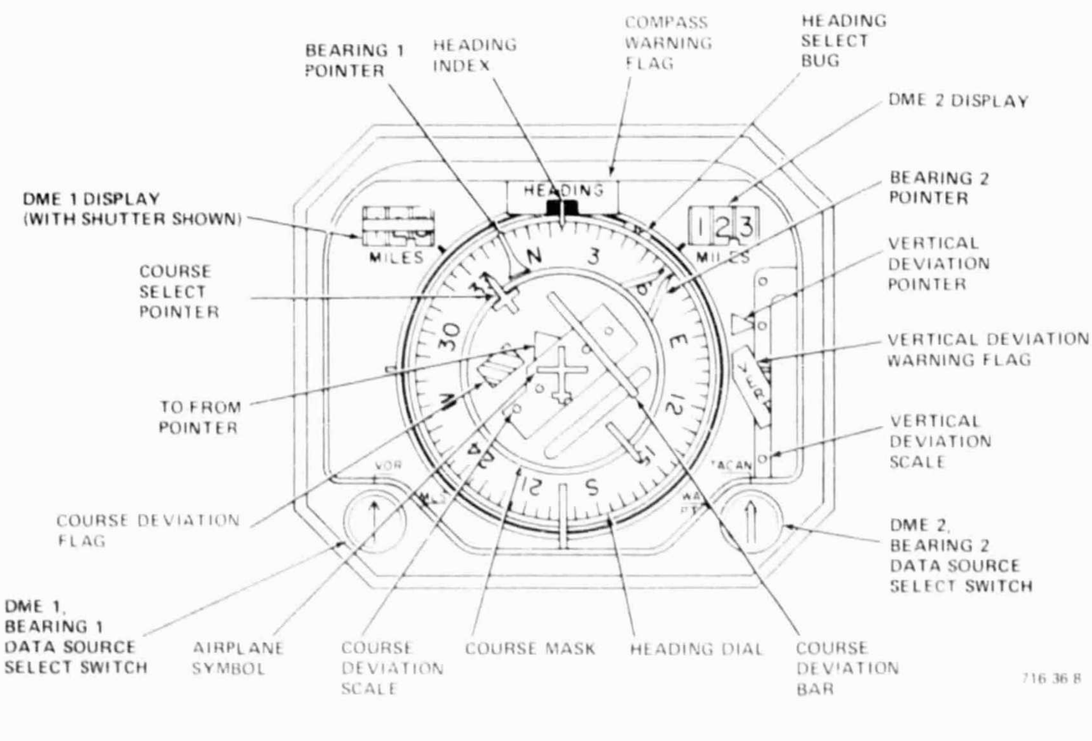


Figure 4-52
The Horizontal Situation Indicator

indication, remotely selected radio course indication, and VOR/TACAN/ WPT TO/FROM indication.

The compass heading display data and the compass warning flag come directly from the directional gyro. The remaining displays are based on software data, and the following paragraphs describe their operation under the supplied Basic computer software.

- Heading-Select Bug - The heading-select bug indicates the heading reference, which may be selected by the pilot by the MSP heading slew switch. It aligns with the heading index (lubber line) at the top of the instrument when heading-hold engages.

- Course Select Pointer - The course-select pointer indicates the course selected by the pilot for radial guidance. The position of this pointer is controlled either by the MSP course slew switch or by keying in the mnemonic CRR at the keyboard. The course-deviation indicator displays the aircraft's lateral displacement from the selected course, as described below. The course-select pointer becomes aligned with the aircraft heading, assuming no crab angle, when the aircraft is tracking the selected course. In the standby mode, the course-select pointer is stowed to the North position.

- Bearing 1 Pointer and DME 1 Display - If VOR is selected by the Data Source-select switch, and if VOR is valid (determined by VOR status, bits 30 and 31), the bearing of the aircraft to the VOR station is indicated by the Bearing 1 pointer. If VOR is not valid, the bearing pointer is stowed at North. The bearing display is independent of MSP mode selection. The DME range is indicated by the DME 1 display. The range of the DME display is .1 to 86.2 nautical miles. If DME is not valid, the DME 1 display is obscured by a shutter.

If MLS is selected by the Nav Source-select switch, and if MLS azimuth is valid, the bearing pointer indicates the MLS azimuth. If not valid, the pointer is stowed at North. MLS range is displayed by the DME 1 display if valid; otherwise, the shutter is activated to obscure the display. The range of MLS range data is .1 to 9.9 nautical miles.

• Bearing 2 Pointer and DME 2 Display - If TACAN is selected by the Data Source-select switch, and if TACAN bearing is valid, the bearing to the TACAN station is displayed by the Bearing 2 pointer. If not valid, the bearing pointer is stowed at North. TACAN range is displayed by the DME 2 display if TACAN range is valid; otherwise, the DME display is obscured by activation of the shutter. The range of the TACAN range data is .1 to 86.2 nautical miles.

If WPT is selected by Data Source-select switch and the REF FP mode is engaged, the bearing and range to the next waypoint of the reference flight path are displayed by the Bearing 2 pointer and DME 2 display, respectively. Otherwise, the bearing and range to the selected WPT are displayed. However, if the Research mode flag is set (RESMDE) and the HSI mode flag (RESARF) from the Research computer is set, then the bearing and range from the Research computer are displayed. If navigation is not valid, then the Bearing 2 pointer is stowed at North and DME 2 display is obscured by the shutter.

• TO/FROM Indicator - The TO/FROM indicator indicates the direction of the selected radial in relation to the navaid during radial guidance modes, i.e., Waypoint, VOR, TACAN. The TO/FROM indicator indicates TO (arrow in the same direction as course-select pointer) if the absolute value of the angle between the course-select pointer and the bearing to the selected navaid is less than 90 degrees. It indicates FROM (arrow in opposite direction to course-select pointer) if the absolute value of the angular difference is greater than 90 degrees. It goes out of view when the aircraft is over the station.

• Course and Vertical Deviation Indicators - The lateral and vertical deviations from the selected paths are displayed by the deviation indicators in the HSI. Full-scale deviations are represented by two dots which represent different units (degrees or feet), depending on whether the autopilot and/or flight director or the raw data modes are engaged. The course deviations are displayed during VOR, TACAN, WPT, REF FP and LAND modes. The vertical deviation is displayed during REF FP and LAND modes only; it is biased out of view during other guidance modes.

When either the autopilot or flight director is engaged, the course deviations represented by the indicator are in feet. In LAND mode, one dot represents 100 feet, and in the other guidance modes (VOR, TACAN, WPT and REF

FP), one dot represents 1000 feet. In the VOR and TACAN raw data modes (AUTO and Flight Director off) the course deviations are displayed with a scaling of 5 degrees per dot; the angular deviation represents the difference between the selected course reference and the bearing to the navaid.

When either the autopilot or flight director is engaged, one dot of vertical deviation has the following meaning: LAND mode, one dot equals 50 feet; REF FP mode, one dot equals 250 feet. The indicator is biased out of view in other guidance modes and the vertical deviation flag is out of view. If the research mode is engaged and the research mode flag for HSI and ADI course and vertical deviation is set, the deviation data from the Research computer is displayed by both deviation indicators.

• Course and Vertical Deviation Flags - The course and vertical deviation flags indicate the validity of the displayed deviation data. The course deviation flag comes into view when:

- TACAN bearing becomes invalid during TACAN guidance.
- VOR bearing becomes invalid during VOR guidance.
- Navigation becomes invalid during REF FP or WPT guidance.
- MLS azimuth becomes invalid during LAND guidance.
- Research course deviation becomes invalid (RESLVF=0).

The vertical deviation flag comes into view when the research vertical deviation becomes invalid (RESGVF=0).

4.9 THE DATA ENTRY KEYBOARD AND DISPLAY

The Data Entry Keyboard (DEK) and Data Entry Display (DED) illustrated in Figure 4-53 are mounted below the Mode Select Panel in the center column of the Sperry simulator cab. The DEK provides, in conjunction with the 24- character alphanumeric display on the DED, a general purpose interactive interface between the pilot and the V/STOLAND software. The pilot may use the keyboard to insert and retrieve data which is in turn displayed on the DED alphanumeric display.

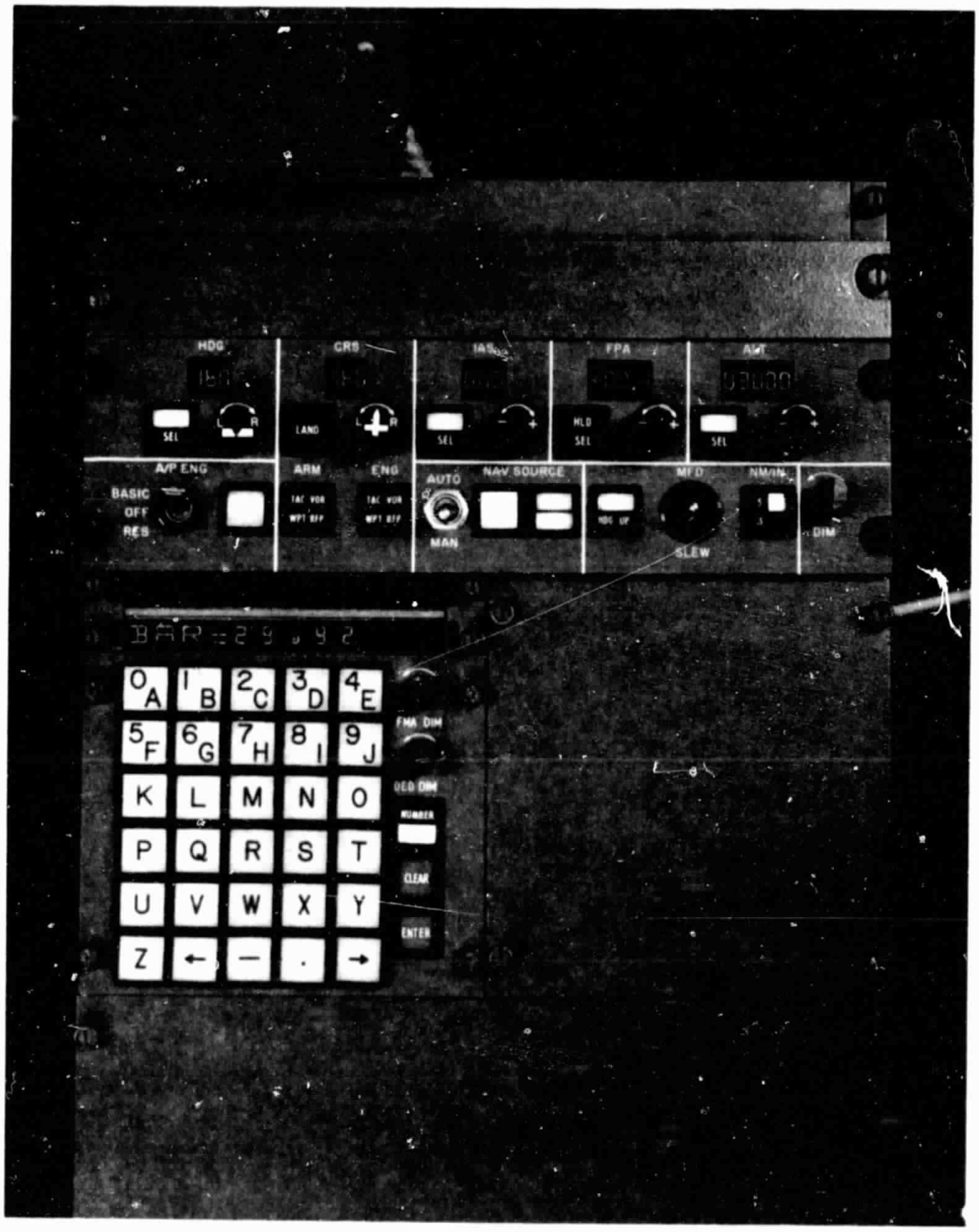
The DEK is in the data entry and review mode whenever a failure message is not displayed on the DED. The failure message display format is described in the following subsection (4.9.2). If a failure message is being displayed, the pilot may depress the CLEAR pushbutton on the DEK to enter the data entry and review mode.

4.9.1 Data Entry and Review Mode

In the DATA ENTRY and REVIEW mode, the pilot may examine or change specific parameters within the V/STOLAND software modules. Each such parameter is identified by a three-letter mnemonic as listed in Table 4-5. This table also indicates the default value of the parameter, the upper and lower limits of the value to which the parameter may be changed, and an indication of the error message that will be displayed in the event the pilot attempts to enter a disallowed value for a particular parameter.

At power-up, the DEK and DED are initialized in the data entry and review mode, with all parameters set to their default values. Also, the DEK is initialized in the LETTER mode, which is annunciated on the NUMBER/LETTER button of the DEK. To examine the value of a parameter, the pilot keys in the associated three-letter mnemonic. As he keys in the letters, they appear in the lower display row of the DED. As the third letter of a valid mnemonic is keyed in, it is followed by an equal sign and the current value of the parameter. The following is an example of a typical display on the DED.

ALT = 12762



4-12799-1

Figure 4-53
The Keyboard

TABLE 4-5

KEYBOARD MNEMONICS

Description	Mnemonic and Default Value[1]	Limits		Over-Range Condition[2]
		Hi	Lo	
Programmable Parameters (% of nominal values):				
KTHT	AAA = 100	1000	10	L
KHDTHT	BBB = 100	1000	10	L
KHDPLH	CCC = 100	1000	10	L
KHDDOT	DDD = 100	1000	10	L
KCRSTH	EEE = 100	1000	10	L
KVTHTH	FFF = 100	1000	10	L
KVTHTA	GGG = 100	1000	10	L
KVPL	HHH = 100	1000	10	L
KVPLFF	III = 100	1000	10	L
KVI	JJJ = 100	1000	10	L
KRPSI	KKK = 100	1000	10	L
KAYPSI	LLL = 100	1000	10	L
KCSPSI	MMM = 100	1000	10	L
KPSICS	NNN = 100	1000	10	L
KDYCRS	OOO = -0[3]	1000	10	L
TAUDD	PPP = 100	1000	10	L
KPHI	QQQ = 100	1000	10	L
KPHID	RRR = 100	1000	10	L
KVEPHI	SSS = 100	1000	10	L
KYDDPH	TTT = 100	1000	10	L

1. The default value is on the program tape, but may have been altered by previous keyboard or MSP action.
2. L = Limit Value, E = Entry Error.
3. Programmable parameters with an absolute default value of zero will display a percentage default value of -0.

TABLE 4-5 (cont)

KEYBOARD MNEMONICS

Description	Mnemonic and Default Value	Limits Hi	Limits Lo	Over-Range Condition
Programmable Parameters (% of nominal values):				
KVPHI	UUU = -0[3]	1000	10	L
KH	VVV = 100	1000	10	L
TROLL	WWW = 100	1000	10	L
GTANP1	XXX = 100	1000	10	L
DF1	YYY = 100	1000	10	L
HDOTCL	ZZZ = 100	1000	10	L
Altitude Reference (ft)	ALT = XXXXX[1]	20,000	0	E
Barometer Reference Setting (in. Hg)	BAR = 29.92	31	28	E
Decision Height (ft)	DHT = 100	400	0	L
Glide Slope Reference (deg)	GSR = -4.0°	-2°	-8°	L
Land Mode Director	LND = 1	2	1	E
Helix Location (ft from TD)	HLX = 5000	8307	3600	L
RFP Waypoint Number	WPN = 2	4	0	E
X-Coordinate of Way-point (ft)	WPX = 30000[2]	1,310,710	-1,310,710	L
Y-Coordinate of Way-point (ft)	WPY = -20000[2]	1,310,710	-1,310,710	L
Programmable Research Parameter (degree)	RAA = 0°	359°	0°	E
Programmable Research Parameter (degree)	RAB = 0°	359°	0°	E

1. The value equals the value on the MSP ALT display at the time the ALT mnemonic is keyed in.
2. Rounded to nearest 10 feet.
3. See note 3 on the previous page.

TABLE 4-5 (cont)
KEYBOARD MNEMONICS

Description	Mnemonic and Default Value	Limits		Over-Range Condition
		Hi	Lo	
Programmable Research Parameter (degree)	RAC = 0°	15°	-15°	L
Programmable Research Parameter	RAD = 0	131071	-131071	E
Programmable Research Parameter	RAE = 0	131071	-131071	E
Programmable Research Parameter	RDA = 0	131071	-131071	E
Programmable Research Parameter	RDB = 0	131071	-131071	E
Programmable Research Parameter	RDC = 0	131071	-131071	E
Programmable Research Parameter	RDD = 0	131071	-131071	E
Programmable Research Parameter	RDE = 0	131071	-131071	E
Research Mode Engage Flag	RES = 0	131071	-131071	E
MFD Course Vector	MCV = 0	131071	-131071	E
MFD Next RFP Waypoint Data	MFD = 1	131071	-131071	E
MFD Flight Path History DOTS	MHI = 1	131071	-131071	E
MFD Trend Vector	MTR = 1	131071	-131071	E
Failure Review Number	FAL = Last diagnostic number ($\leq$99)	Last diagnostic number ($\leq$99)	0	L
Preflight Test Flag	PFT = 0	$\begin{cases} 1 \text{ with weight-on-wheels} \\ 0 \text{ otherwise} \end{cases}$	0	E
Altitude of Runway (ft)	ARW = 140	131,071	-131,071	E

If an invalid three-letter mnemonic (one not shown in Table 4-5) is keyed according to the above procedure, the message ENTRY ERROR will be displayed in the upper DED when the third letter is depressed on the DEK. The mnemonic in the lower DED will remain unchanged. The pilot may delete characters, one at a time, by depressing the ← key, and then retype the correct characters. The pilot may also depress the CLEAR pushbutton, causing the display to blank, after which he may retype the correct mnemonic from the beginning.

To change the value of the associated parameter, the NUMBER mode is first entered by pushing the NUMBER/LETTER key. As the numeric value is keyed, the corresponding characters appear in the upper DED display, starting with the fifth character from the left. A typical display is shown below.

-14.2
RAC = 13.9

When the pilot has typed the value to which the parameter associated with the currently displayed mnemonic is to be changed, he must depress the ENTER pushbutton on the DEK to assign the data within the V/STOLAND system. When this is done, the upper DED display will blank and the DEK will automatically revert to the LETTER mode. The lower DED display will continue to display the mnemonic, but will now indicate the new value to which its associated parameter has been set.

If the entered numerical data is beyond the limits assigned for the mnemonic (see Table 4-5), the value entered is either the limit value or it won't be changed. For example, if a value of -20 degrees is inserted for the mnemonic GSR, the DED display will appear as follows:

LIMIT VALUE
GSR = -8.0

The DEK then reverts to the LETTER mode. For some parameters, the upper DED display will show the message ENTRY ERROR and the lower DED display will remain unchanged. For example, the mnemonic RAA is limited to the range 0 to 350 degrees. If the current value of this parameter is 200 degrees and an input of 400 degrees is attempted, the DED display will appear as follows immediately after the ENTER button is depressed:

ENTRY ERROR
RAA = 200

The DEK remains in the NUMBER data entry mode in this case, and the pilot may immediately re-enter a revised numeric value.

The ← and → keys on the DEK may also be used to display mnemonics and their associated values without having to key in the alphabetic characters. When the DED is blank, depressing the ← numeric key will cause the last displayed mnemonic and its associated value to re-appear on the DED. When the DED is not blank, each depression of the ← or → pushbuttons on the DEK causes the mnemonic adjacent to the last displayed mnemonic to be displayed, in the order shown in Table 4-5.

4.9.2 Failure Message Review Mode

The FAILURE MESSAGE REVIEW mode annunciates the occurrence of failures detected by the V/STOLAND system and allows the pilot to review up to 99 failure messages stored in the memory of the Basic computer. The DEK and DED are in the FAILURE MESSAGE REVIEW mode whenever a failure message is being displayed on the DED. A typical failure message is shown in the example below.

FAILURE 13
ADI ATTITUDE

The general format of a failure message displays the failure number on the upper DED display, and a diagnostic message on the lower DED display. The failures are numbered in the order in which they occur. A → symbol is displayed after the failure number if there are failures logged with a higher number than the one currently displayed. No → symbol is displayed with the last failure number.

The FAILURE MESSAGE REVIEW mode may be entered manually by the pilot, or is entered automatically whenever a failure is detected by the V/STOLAND system. The MESSAGE ALERT light on the main instrument panel will turn on in this case, indicating that there are unreviewed failure messages. The FAILURE MESSAGE REVIEW mode is entered manually by keying in mnemonic FAL and the number of the failure message to be reviewed. After the mnemonic is keyed in, the last failure number will be displayed. Assuming 15 failures have occurred, the display would be as follows:

FAL = 15

The pilot may then key in the number of any failure message he wishes to review. For example, if he wishes to review the eighth failure, the display would show:

8
FAL = 15

Depressing the ENTER pushbutton will result in the eighth failure message being displayed in the format shown in the first part of this section. If the pilot enters a failure number equal to or exceeding the value associated with the FAL mnemonic, the last stored failure message will be displayed.

The failure message buffer may be initialized (cleared) only while the aircraft is on the ground by keying the DEK mnemonic FAL and entering "0" for the new value to be assigned to this mnemonic. In this case the DED display, just prior to depressing the ENTER pushbutton, will be

0
FAL = 15

assuming 15 failures have occurred so far. Then, after depressing the ENTER pushbutton, the display will revert to showing

FAL = 0

indicating that there are no failure messages stored. If the aircraft is not on the ground, the failure message buffer cannot be cleared. In this case, keying in a value less than or equal to zero for the mnemonic FAL will cause the first failure message to be displayed.

If more than 99 failures occur during a flight and the failure message buffer cannot be cleared, all failures beyond the 99th will be overstored in the 99th failure buffer location, and the failures occurring between the 98th and last failure will be lost.

The ← and → keys may be used to scan the failure messages stored in the failure message review buffer. Depressing the → key on the DEK will display the next higher numbered failure; depressing the ← key on the DEK will display the next lower numbered failure. If the first or last failure is displayed, depressing the ← or → keys on the DEK has no effect, respectively.

The failure message review mode may be canceled only by depressing the CLEAR pushbutton on the DEK. This action will return the DEK to the DATA ENTRY and REVIEW mode, blank the DED, and turn off the message alert light if it was on.

4.10 THE FLIGHT MODE ANNUNCIATOR

The Flight Mode Annunciator (FMA) is located immediately above the ADI on the instrument panel. It is physically identical to the DED, with 24 16-segment alphanumeric characters arranged in two rows. The upper and lower rows have separate functions, as described below (unless the PREFLIGHT mode is engaged as was described in Paragraph 3.7).

The upper FMA messages, displayed on the top row of characters on the FMA, are of two types: advisory and command. Advisory messages on the FMA are displayed as non-flashing messages. Command messages on the FMA are displayed as flashing messages. Table 4-6 lists the messages that may be displayed on the upper FMA (arranged in decreasing order of priority).

TABLE 4-6

UPPER FMA MESSAGES

1	2	3	4	5	6	7	8	9	10	11	12	Notes
(research message, if any)												
T	O	U	C	H	D	O	W	N				
L	E	T	D	O	W	N						
L	O	C	K		P	Y	L	O	N			1
U	N	L	O	C	K		P	Y	L	O	N	1
P	Y	L	O	N	S		T	O		(X	X)	2
F	L	A	P	S		T	O		(X	X)		2
R	P	M		T	O		(X	X	X)			2
G	O		A	R	O	U	N	D				
R	E	S		A	P	,						3
B	A	S		A	P	,						3
R	E	S		F	D	,						3
B	A	S		F	D	,						3
R	E	S		(X	X)	,						3
M	A	N	U	A	L	,						3

Notes:

1. Message flashes

2. Message flashes in flight director mode only

3. The message AIRPL, HELCP, or TILT is displayed after the comma when the pylon angle is less than 5 degrees, greater than 85 degrees, or between 5 and 85 degrees, respectively.

The highest priority messages (for both rows) are any non-blank messages from the Research computer (that is, the input buffer for such messages is not cleared). Any message in this buffer will preempt all other messages. Hence, the Research computer can have full control of the FMA. In the absence of a research message, the following messages will be displayed under the condition described:

- TOUCHDOWN - Advisory message displayed any time weight-on-wheels is detected.

- LETDOWN - Advisory message displayed when the aircraft ceases to hover above the touchdown point and begins the descent of the final 10 feet to the ground.

- LOCK PYLON - Command message displayed when the indicated airspeed of the aircraft exceeds 190 knots and the pylons are unlocked.

- UNLOCK PYLON - Command message displayed prior to any commanded pylon movement if the pylons are locked.

- PYLONS TO (XX)° - Advisory message if the autopilot is engaged, indicating that the V/STOLAND system is converting the pylon angle to the position indicated. Command message if the manual flight director is engaged, indicating that the pilot, at his discretion, is to convert the pylon angle to the position indicated.

- FLAPS TO (XX)° - Advisory message if the autopilot is engaged, indicating that the V/STOLAND system is moving the flap-selector switch to the indicated position. Command message if the manual flight director is engaged, indicating that the pilot, at his discretion, is to move the flap-selector switch to the position indicated.

- RPM TO (XXX)% - Advisory message if the autopilot is engaged, indicating that the V/STOLAND system is slewing the RPM command to the value indicated. Command message if the manual flight director is engaged, indicating that the pilot, at his discretion, is to slew the RPM command to the value indicated.

- <u>GO AROUND</u> - Advisory message indicating that the go-around submode has been engaged.

- <u>RES AP, BAS AP, RES FD, BAS FD, RES (XX), MANUAL</u> - In the absence of any one of the above upper FMA messages, the upper FMA will annunciate the flight configuration of the V/STOLAND system. In the order listed, these advisory messages indicate that the V/STOLAND's system is engaged in the research autopilot, basic autopilot, research flight director, basic flight director, research mode number XX, or the manual mode. These messages are always left-justified on the top line of the FMA, and are always followed by one of the following three titles which indicates the range of the current pylon angle.

- <u>AIRPL, HELCP, TILT</u> - One of this set of advisory messages is always displayed immediately to the right of one set of advisory messages listed immediately above. In the order listed, these messages indicate the pylon angle is less than 5 degrees (nacells almost aligned with wing chord), greater than 85 degrees (nacells almost perpendicular to wing chord), or between 5 and 85 degrees.

The bottom row of characters of the FMA is dedicated to displaying the flight guidance modes in which the V/STOLAND system is armed or engaged. Table 4-7 lists all of the messages which may be displayed on the lower FMA. As in the case of Table 4-6, the messages are in the order of decreasing priority. In the absence of a research mode message, the lower FMA will display some combination of the other messages listed in Table 4-7.

The lower FMA is divided into three sections as shown in Table 4-7, corresponding to categories of modes. Characters 13 through 15 are used to annunciate the engaged vertical or longitudinal guidance mode. Characters 17 through 19 annunciate the engaged lateral-directional guidance mode. Characters 21 through 24 annunciate any armed lateral-directional or vertical flight modes. If no guidance mode is engaged for a given category, that section will be blank. The messages displayed on the lower FMA may be a combination from each category, including blanks.

TABLE 4-7
LOWER FMA MESSAGES

13	14	15	16	17	18	19	20	21	22	23	24	Notes
(research message, if any)												
F	L	R	,	T	A	C	,	T	A	C	A	
L	-	(X)	,	L	-	(X)	,	L	-	(X)	A	1
R	F	P	,	R	F	P	,	R	F	P	A	1
A	L	T	,	W	P	T	,	W	P	T	A	
F	P	A	,	V	O	R	,	V	O	R	A	
I	A	S	,	H	D	G	,	A	L	T	A	
(BLANK)				(BLANK)				L	-	(X)	A	2
								R	F	P	A	2
									(BLANK)			

Notes:

1. For LAND or Reference Flight Path Lateral-Directional Modes
2. For LAND or Reference Flight Path Vertical Modes

The messages listed in Table 4-7 are described below:

- FLR - Indicates that the flare longitudinal submode of the land mode is engaged.

- L-(X) - Indicates that LAND mode X is engaged in the lateral-directional axes if the message is in the center section, or the vertical axis if it is in the left section. If the message is L-1, a straight-in approach is engaged. If the message is L-2, a helix approach is engaged.

- RFP - Indicates that a Reference Flight Path mode is engaged, either in the lateral-directional axes or the vertical axis, depending upon the position of the message on the FMA, as explained above.

- ALT - Indicates that the altitude vertical mode is engaged.

- FPA - Indicates that the flight path angle vertical mode is engaged.

- IAS - Indicates that the airspeed longitudinal mode is engaged.
- WPT - Indicates that the waypoint lateral-directional radial course mode is engaged.
- VOR - Indicates that the VOR lateral-directional radial course mode is engaged.
- TAC - Indicates that the TACAN lateral-directional radial course mode is engaged.
- HDG - Indicates that the heading lateral-directional mode is engaged.
- L-(X)A - Indicates that LAND mode X is armed in the lateral-directional if not already engaged in this axis. If it is already engaged in this axis (as indicated by the center section message), this message indicates that LAND mode X is armed in the vertical axis. If the message is L-1A, a straight-in approach is armed. If the message is L-2A, a helix approach is armed.
- RFPA - Indicates that a reference flight path mode is armed, either in the lateral-directional axes or the vertical axis under the conditions described above for L-(X)A.
- WPTA - Indicates that the waypoint lateral-directional radial course mode is armed.
- VORA - Indicates that the VOR lateral-directional radial course mode is armed.
- TACA - Indicates that the TACAN lateral-directional radial course mode is armed.
- ALTA - Indicates that the altitude vertical mode is armed.

4.11 INERTIAL SENSORS

The following inertial sensors supply data to the V/STOLAND system:

- Vertical Gyro
- Rate Gyros
- Accelerometers
- Compass System

The inertial sensors supply aircraft angular position, angular rate, and translational acceleration data to the V/STOLAND system. A brief discussion of each unit is given in the following paragraphs.

4.11.1 Vertical Gyro

The VG-14H vertical gyro is Government Furnished Equipment (GFE) on the XV-15 aircraft, and is located on the inertial platform assembly. The gyro provides aircraft pitch and roll attitude reference data for use in the V/STOLAND ADI and Basic computer. It has high inertia, providing an angular momentum of 3.8 million gm-cm^2 per second at 22,000 rpm to stabilize itself about both gimbal axes. The gyro has ±80 degrees of freedom in pitch and unlimited (360 degrees) freedom in roll. In addition, it is designed to operate in the vibration levels encountered in helicopter environments by virtue of its improved suspension system, antifrothing liquid levelers, and redundant connections for the pitch and roll synchro transmitters. Performance specifications for this gyro follow:

PERFORMANCE DATA

Gyro Rotor Speed	22,000 rpm (nominal within 5 minutes after power is applied
Gyro Angular Momentum	3.8 million gm-cm^2 per second
Gyro Erection	Vertical within 3 minutes after power is applied
Verticality (Alignment of spin axis with true vertical)	.25 degree (bench)
Roll Erection Cutoff Threshold	.10g (6-degree roll angle nominal)
Erection Rate	
Fast Erection (Roll)	20 degrees per minute, minimum
Fast Erection (Pitch)	20 degrees per minute, minimum
Slow Erection (Roll Pitch)	2.5 degrees per minute, nominal

The VG-14H produces a valid which, when set, indicates that the unit is operating properly. This valid is monitored in the V/STOLAND failure monitoring software module, and will cause a system disconnect if it is not detected. Also, failure of the VG is annunciated by a flag on the ADI.

4.11.2 Rate Gyros

The pitch, roll, and yaw rate gyros are contained in two separate assemblies mounted on the inertial platform assembly. The yaw/roll assembly contains two rate gyros mounted in a precision-machined block and attached to the unit's side. The pitch assembly contains the pitch rate gyro mounted in a block similar to the yaw/roll assembly.

The rate gyros sense and process the aircraft angular rates about each of three orthogonal aircraft body axes. The signal gradient from each rate gyro is .25 volt-second per degree. The yaw rate gyro produces an additional output which drives the rate-of-turn indicator on the ADI. The gradient of this signal is .333 volt-second per degree. Each gyro also provides a valid signal that is dependent upon the speed of the gyro. The valid of a rate gyro is lost when the gyro's speed decreases to 75 percent of its nominal value. Each gyro has a self-test feature which is exercised during the preflight test. The self-test feature is initiated by setting a discrete from the Data Adapter to the rate gyro. This introduces a torquing current that simulates a 5-degree-per-second angular rate which is checked by preflight software.

4.11.3 Accelerometers

The longitudinal, lateral, and normal accelerometers are mounted in a single assembly, with the proper orientations to measure the aircraft linear accelerations in each of the three orthogonal aircraft body axes. Each accelerometer is mounted separately on a precision base with an arrow indicating its sensitive axis. All units are mounted on the inertial platform assembly.

The accelerometers are force-balanced and have a closed-loop configuration. The accelerometer electronics contained in each unit consist of an electrical pickoff and a servo amplifier that provide the closed-loop operation. The output of each accelerometer is a dc voltage that is proportional to the aircraft acceleration in that axis. The longitudinal and lateral accelerometer signal gradients are 8 volts per g of acceleration. The signal gradient of the normal accelerometer is 2 volts per g of acceleration. In addition, the normal accelerometer has a 1g bias so that it produces no output voltage when at rest.

Eacn accelerometer contains self-test provisions in which a torquing current from the Data Adapter is injected. This results in accelerometer outputs that are proportional to the test current applied and which are used as a system check during preflight test.

4.11.4 Compass System

The C-14 gyromagnetic compass system provides accurate heading information referenced to a free directional gyro heading or slaved to the earth's magnetic field. This heading information drives the HSI, and is also transmitted to the Data Adapter for use in the V/STOLAND system software.

The compass set consists of a C-14 directional gyro, flux valve, compensator, and controller. The flux valve and compensator provide the direction of the earth's magnetic field, corrected for the aircraft's disturbing field. In addition, the compensator provides the error signal between the earth's magnetic field and the heading of the directional gyro. This latter signal slaves the directional gyro to the earth's magnetic field when the compass system is used in this mode.

When power is applied to the aircraft, the pilot must synchronize the heading of the directional gyro to the earth's magnetic field. This is accomplished by turning the compass synchronization knob located on the compass controller until the annunciator window is clear. Once this is done the pilot selects the mode (free or slaved gyro) in which he wishes the compass system to operate for the flight.

4.12 AIR DATA SENSORS

The following air data sensors supply data to the V/STOLAND system:

- Static Pressure Transducer
- True Airspeed Sensor
- Altitude/Airspeed Transducer
- Total Temperature Probe

The air data sensors supply data on airspeed, air pressure, and air temperature to the V/STOLAND system. A brief discussion of each unit is given in the following paragraphs.

4.12.1 Static Pressure Transducer

The static pressure sensor is capable of measuring pressures in the range of 20 to 31.5 inches of mercury. The pressure data is used for deriving pressure altitude.

The pressure sensor has a rigidly supported metal diaphragm subjected to vacuum on one side and static pressure from the pitot tube on the other side. The natural resonance frequency of the diaphragm is directly related to the applied static pressure and the ambient temperature. The resonant frequency is sensed by a coil located in the field of a permanent magnet attached rigidly to the vibrating diaphragm. A count proportional to the period of the vibrating diaphragm is transmitted to the Data Adapter.

Pressure altitude is derived from the calibration data of the sensor which is stored in the Basic computer as a table whose independent variables are proportional to the period of the vibrating diaphragm and the ambient temperature of the sensor. The V/STOLAND navigation software module computes barometric altitude from the pressure altitude derived from this table and the pilot-supplied baro setting input via the DEK mnemonic BAR.

4.12.2 True Airspeed Sensor

The true airspeed sensor is a J-TEC VA-210 model capable of measuring true airspeed in the range of 2 to 200 knots. The sensor utilizes an ultrasonic technique to count the frequency of the vortex sheet behind a strut in the airflow. The frequency at which the vortices are shed is directly proportional to the airspeed.

The sensor and associated electronics/display unit are Government Furnished Equipment (GFE). The electronics/display unit provides a dc analog output signal whose gradient is .025 volt per knot.

4.12.3 Altitude/Airspeed Transducer

The altitude/airspeed transducer measures total and static pressure to compute altitude (-1000 to 40,000 feet), indicated airspeed (75 to 350 knots), and impact pressure (.27 to 6.29 inches of Hg). The outputs are dc analog signals whose gradients are:

- Impact Pressure: 1 volt per inch Hg
- Indicated Airspeed: .02 volt per knot
- Altitude: .0002 volt per foot (-1000 foot bias)

All equipment associated with the altitude airspeed transducer is GFE.

4.12.4 Total Temperature Probe

Total temperature is sensed via a platinum resistance type sensor and conditioned with a signal conditioning amplifier to output a 0 to 5 volt dc signal. The gradient of this signal is .033 v/°C with a 50°C bias. The total temperature probe and signal conditioning amplifier are GFE.

4.13 NAVIGATION SENSORS

Five sets of navigation units are provided (GFE) on the XV-15 aircraft:

- TACAN
- VOR/DME
- MLS
- INS
- Doppler Radar

The navigation sensors supply aircraft translational position data to the V/STOLAND system. A brief discussion of each unit is given in the following paragraphs.

4.13.1 TACAN

The TACAN navigation set consists of the following items of equipment:

- TACAN Receiver/Transmitter
- TACAN Signal Data Converter
- TACAN Navigation Control Unit

This system is a MIL-spec, all solid-state, airborne navigation sensor which supplies data to the V/STOLAND system regarding the range and bearing of the aircraft to a TACAN or VORTAC station. Each station with which this navigation sensor can communicate is identified by a channel number that is selected at the TACAN set control unit.

TACAN is tuned by the channel selector switch located on the control unit mounted on the port console. The control unit is shown in Figure 4-54. The channel selector has two controls - a circular disc for selecting the first two digits of the channel number, and a lever for selecting the third digit. The ECM WARN annunciator is not used for V/STOLAND. The BIT button is pressed during preflight test, and the test outcome is indicated by the GO and NO-GO lights.

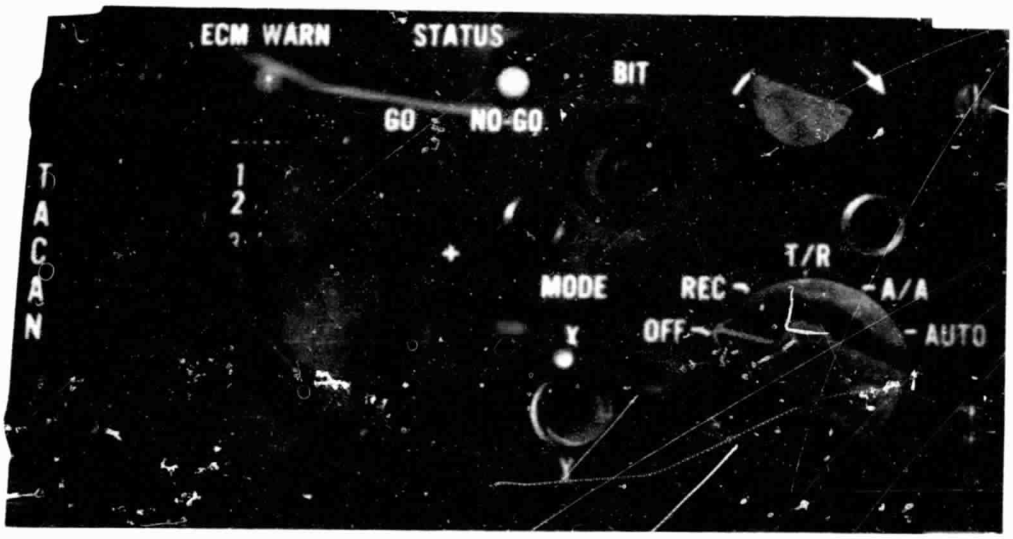
For V/STOLAND use, the X mode is selected via the two-position toggle switch on the panel, and the five-position selector switch is moved into the T/R detent in order to receive range and bearing data. The VOL control, located in the upper right corner, adjusts the ident tone volume.

4.13.2 VOR/DME

The VOR/DME navigation set consists of the following items of equipment:

- VOR Navigation Receiver
- DME Receiver
- VHF NAV Control Unit

The VOR navigation receiver accepts VOR ground-station transmission signals in the frequency range 108.00 to 117.95 MHz to supply the digital VOR bearing of the aircraft to the station. Selection of the frequencies associated with each VOR station is provided by the control unit. The DME receiver interrogates the DME ground station by transmitting pulse pairs, and the station responds to a number of these interrogations. For a colocated VOR and DME, the VOR/LOC frequencies in the range of 108.00 to 117.95 MHz and corresponding DME frequencies are paired. Therefore, selecting the VOR frequency will automatically provide the corresponding DME frequency for the range data.



716 36 16

Figure 4-54
TACAN NAV Control Unit

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The VHF navigation control unit shown in Figure 4-55 is mounted on the port console. The COMM portion of this unit, with the associated dual control knob located on the left side, is not used for V/STOLAND operations. The dual control knob on the right side provides the means for selecting the VOR frequencies which are displayed in the NAV window. Colocated DME frequencies are also selected by this control. The VOL knob controls the navigation ident audio selected by this control. The VOL knob controls the navigation ident audio level. The selectable positions of the NAV switch have the following meaning:

- <u>NAV</u> - Only the VOR (digital) receiver is on.

- <u>STBY</u> - The VOR (digital) and DME receivers are on, but the DME cannot transmit.

- <u>DME</u> - Same as in STBY, but the DME can transmit. Range search is limited to 200 miles. This position is used for V/STOLAND navigation.

- <u>DVRD</u> - This position allows the DME to search over a 400 mile range. This position is not generally used since it extends the DME acquisition time.

The NAV test switch is a three-position switch, spring-loaded to return to center. It is used for self-test of the VOR and DME receivers during preflight.

4.13.3 <u>MLS</u>

The MLS navigation set consists of the following items of equipment:

- MLS Angle Receiver (azimuth and elevation)
- MLS Receiver (range)
- MLS Receiver Control Unit
- MLS C-Band Antenna Switch Module
- MLS Frequency Select Switch
- MLS Range BIT Switch

The MLS angle receiver accepts MLS azimuth and elevation signals from these respective ground stations, providing localizer and glide slope information to the V/STOLAND system. Selection of the channel to which the angle receiver is

716-20-39

Figure 4-55
VHF NAV Control Unit

tuned to receive this data is provided by the MLS receiver control unit. The MLS DME receiver interrogates the MLS range transmitter ground station by transmitting pulse pairs, and the station responds to a number of these interrogations. The MLS DME receiver on the XV-15 aircraft may be tuned only to two frequencies via the MLS frequency-select switch (location undetermined at the time of this writing). The two frequencies to which the MLS range receiver may be tuned are associated with the MLS DME transmitter at Crow's Landing and the VOR/DME station at Woodside, California near Moffet Naval Air Station. In flight, the MLS frequency select switch should be in the "Crow's Landing" position in order to receive MLS range data. During preflight at Moffet Field, this switch should be placed in the "Moffet Field" position in order to receive DME data while at the Naval Air Station.

The MLS receiver control unit controls the operation of the MLS angle receivers. A line drawing of this unit is shown in Figure 4-56. Table 4-8 describes the functions of the controls on this panel. During V/STOLAND operation, the mode-control switch should be in the ON position. During preflight test, the test button on this panel will be pressed. Otherwise, the state of this panel is irrelevant to V/STOLAND operation.

4.13.4 Inertial Navigation System

The INS aboard the XV-15 aircraft is a Litton LTN-51 system consisting of the following items of equipment:

- Inertial Navigation Unit
- Mode-Selector Unit (MSU)
- Control/Display Unit (CDU)

All INS equipment is located on the flight rack of the XV-15 aircraft, and is inaccessible by the pilots from the cockpit.

The MSU is located on the non-aircooled rack, and is shown in a line drawing in Figure 4-57. This unit consists of a five-position, mode-selector switch and two annunciator lights. The mode-selector switch is detented in the NAV position to prevent the INS from being inadvertently switched out of the

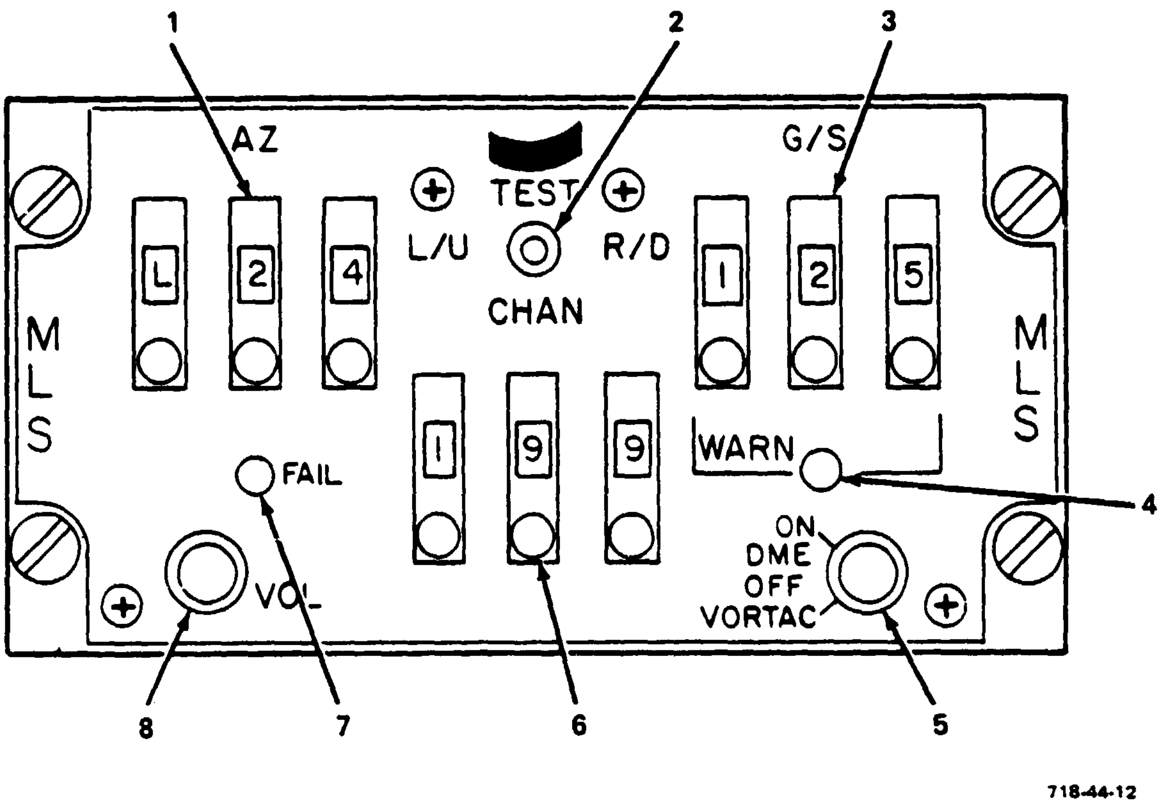


Figure 4-56
MLS Control Panel Controls/Indicators

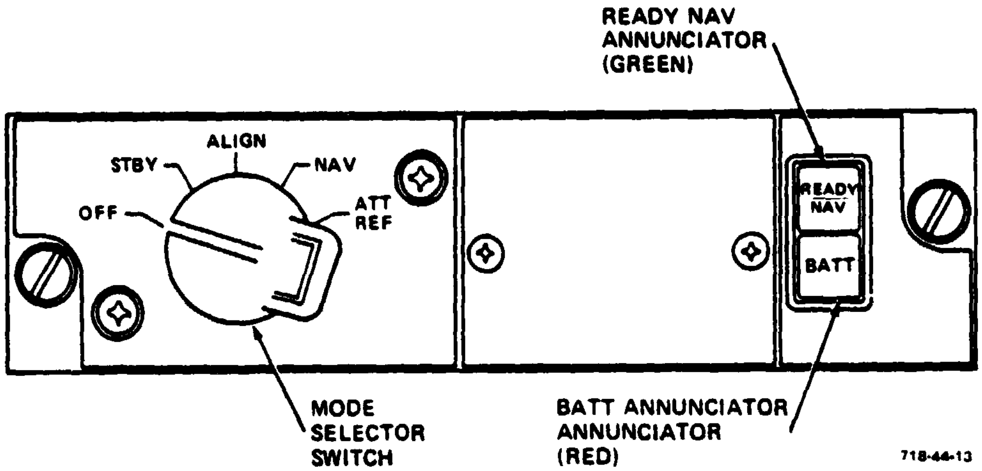


Figure 4-57
INS Mode Select Unit

navigate mode. The knob must be pulled away from the MSU panel before the mode-selector switch can be set out of the NAV position. The mode-selector switch and annunciator lights perform the following functions:

- OFF - Power is applied only to the MSU and CDU edge lighting. INS primary power is off.

- STBY - In standby, INS primary power is ON and the CDU is operated to perform display test and to insert the aircraft's present position. An automatic alignment sequence starts, during which the platform cages to the aircraft's axes and platform temperature stabilization and gyro run-up are initiated.

- ALIGN - The automatic alignment sequence continues and a platform alignment sequence is initiated. When the automatic alignment sequence is completed, the READY NAV (Ready to Navigate) annunciator comes on to indicate that the navigate mode may be selected by setting the mode selector switch to NAV. The aircraft must not be moved when the mode-selector switch is set to ALIGN. However, gusty wind conditions and movement caused by fueling or cargo and passenger loading do not significantly affect alignment.

- NAV - In navigate, the aircraft may be moved and normal in-flight operations are performed. Note that if the mode selector switch is moved out of the NAV position, the INS navigational capability is lost and an alignment must be performed on the ground.

- ATT REF - In attitude reference, the INS provides pitch, roll, and platform heading outputs only. No navigational capability exists, and CDU numerical displays are blank.

- READY NAV ANNUNCIATOR (GREEN) - Comes on when the INS has completed alignment and is ready for navigating.

- BATT ANNUNCIATOR (RED) - Comes on when backup power is less than the minimum required to operate the INS. Must be pressed to reset to OFF when power is restored, or will remain on as long as sufficient power to light the annunciator lamp is available.

TABLE 4-8

MLS CONTROL PANEL CONTROLS/INDICATORS

No.	Function
1	These three pushbutton digital switches select the azimuth radial used to compute azimuth angle deviation. They range from L00 to L49 and R00 to R49 in 1 degree increments. Only angles between L40 and R40 are usable. Angles that exceed this range are converted to either L40 or R40. "L/R" means left or right.
2	Test junction that generates a L/U (left/up) or R/D (right/down) indicator deflection on the CDI.
3	These three pushbutton digital switches select the glide slope used to compute elevation angle deviation. They range from 00.0 to 19.5 degrees in increments of .5 degree. Only angles between 2 and 15.5 are usable. Angles exceeding this range are converted to either 2 or 15.5 degrees.
4	If a glide slope is selected that is below the minimum recommended for the landing site, the minimum glide slope indicator will flash at a 1 Hz rate.
5	This switch in VORTAC sets the DME in a standard operating mode. OFF removes all power. DME is reserved for future use. ON applies power to angle receiver and transfers the DME from the VORTAC mode to a precision approach MLS mode.
6	These three pushbutton digital switches select the channel that the MLS angle receiver is on. Their range is from 000-199, dedicated to commercial channels. These are coded in standard 2 of 5 coding.
7	Indicates a failure in the angle receiver.
8	This 500-ohm potentiometer adjusts the audio volume of the Morse code (station identification).

The INS Control Display Unit (CDU) is located on the non-aircooled rack and is illustrated in Figure 4-58. This unit controls the operation of the INS during flight and allows the examination and entry of data in the INS computer.

Data generated by the INS is not used in the Basic computer. Hence, the states of the MSU and CDU are irrelevant to V/STOLAND operation while in a Basic mode. However, in preflight the INS system is tested and aligned, and the preflight procedure indicates the selections that must be made on the MSU and CDU.

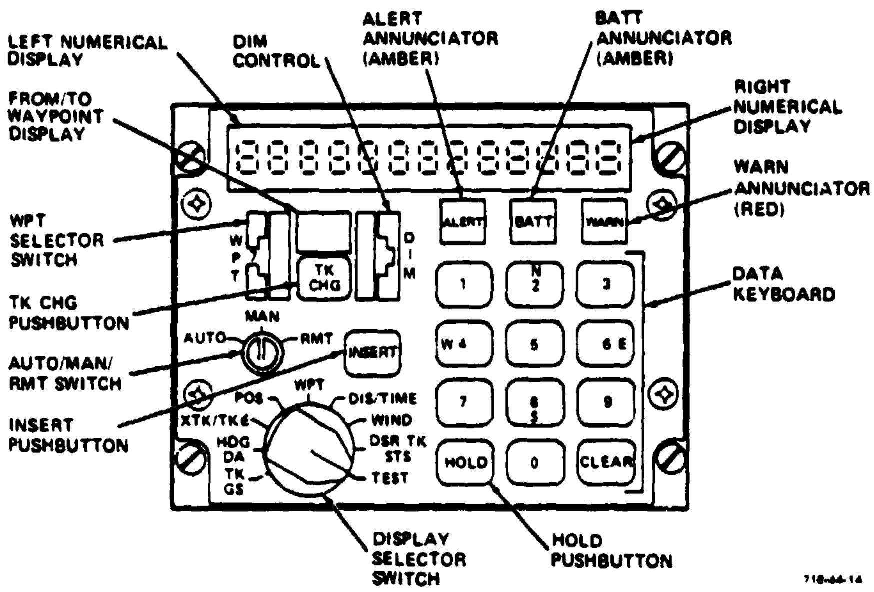


Figure 4-58
INS Control Display Unit

4.13.5 Radio Altimeter

The Radio Altimeter System consists of the following units:

- Radio Altimeter (Bendix, ALA-51A)
- Altimeter Indicator (Bendix, INA-51A)
- Antenna (2) (Bendix, ANA-51D)

This system provides aircraft altitude in the range 0 to 2500 feet, and is displayed by the altimeter indicator. The radio altitude data is blended with barometric altitude between 400 and 200 feet of altitude. Below 200 feet, radio altitude is used exclusively for navigation.

4.13.6 Doppler Radar

At the present time, the equipment associated with the Doppler radar has not been defined by the Government.

4.14 THE INSTRUMENTATION SYSTEM

The Research Instrumentation System consists of all instrumentation, including transducers, data acquisition system, and tape recorder to measure and record the aircraft systems' performance during flight.

The Research Instrumentation System, illustrated by the block diagram of Figure 4-59, operates as follows. Data from the transducers are forwarded to the Remote Multiplexer/Digitizer Unit (RMDU) which provides the signal conditioning for the transducer, adjusts signal gain to programmed value, converts analog data to digital form, and encodes the data into a Pulse Code Modulation (PCM) serial bit stream. Transducer excitation (if required) is supplied from a separate low-voltage (±3V) power source. Two 64-channel preamplifier filters are available to condition the transducers which require special filtering or amplification. An additional active network panel may be used to condition or process transducer signals that have characteristics which do not readily match the interface requirements of the RMDU. A time-correlation base for the total system is supplied from a time-code generator, with a remote time display mounted on the pilot/copilot instrument panel. All data are recorded on a standard airborne magnetic tape recorder. An interface is available for inflight transmission of data from one RMDU via L-band telemetry. The following paragraphs describe the principal components of the system.

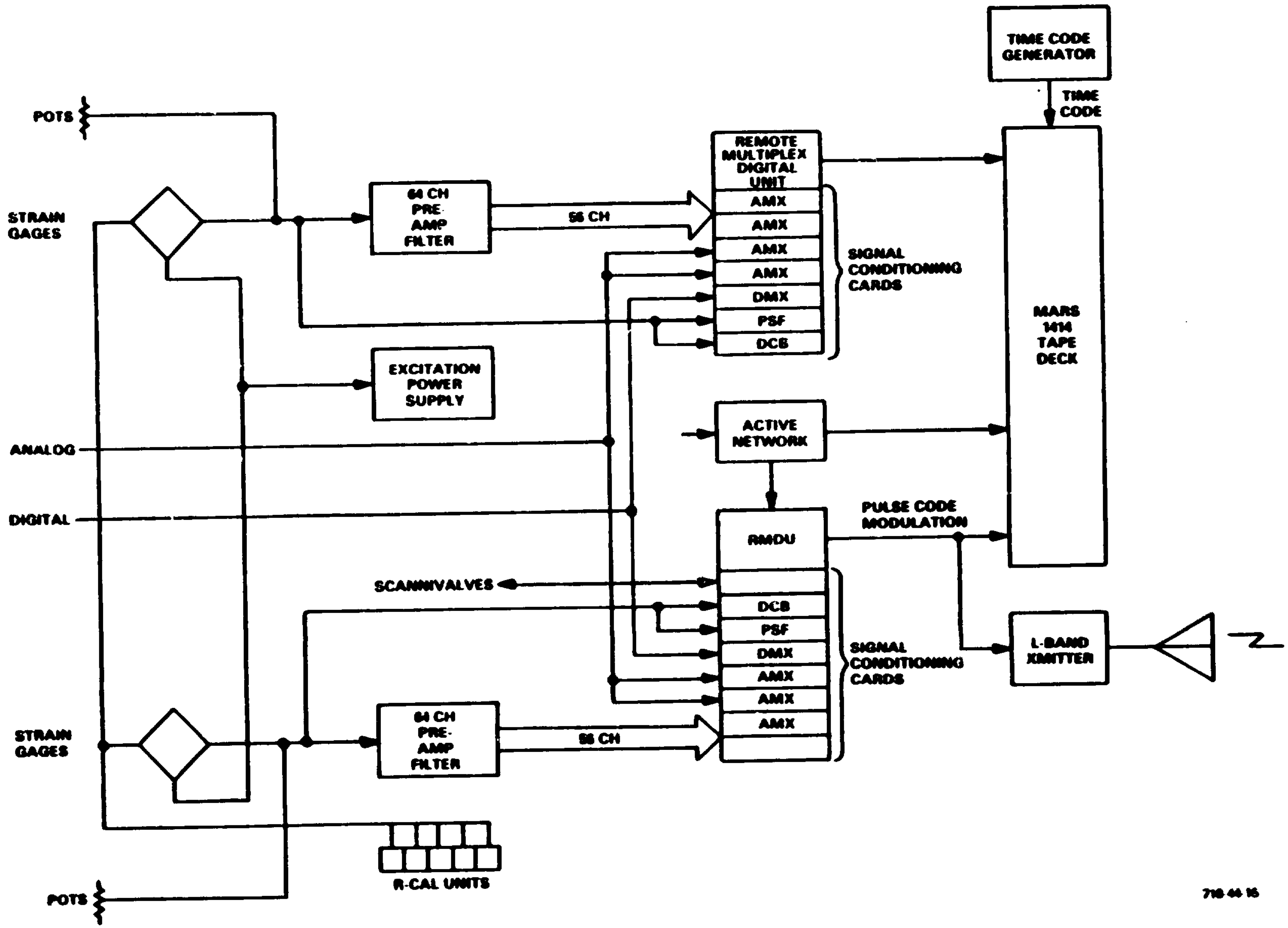


Figure 4-59
Research Instrumentation Block Diagram

4.14.1 Remote Multiplexer/Digitizer Unit

The Remote Multiplexer/Digitizer Unit (RMDU), designed and manufactured by Teledyne Controls, receives transducer signals in analog or discrete/digital form, conditions/normalizes and multiplexes the input data, converts the analog data to a digital format, and outputs the data in a PCM format. Each unit can accept up to 256 channels of data, with a serial output of up to 131,000 words per second. This word rate cannot be utilized by the system due to a tape recorder limitation (single-track capacity of 40,000 words per second at 30 inches per second tape speed).

The RMDU is configured for flight by inserting printed circuit cards which interface with the transducers into any of ten card slots. A wide variety of interface cards are available which are compatible with most aircraft transducer signals. Included in the system is a programmable gain amplifier which provides for eight preselected and programmable gains to amplify signal strength to ±5 volts, full scale (gains from ±10 MV to 10V full scale).

The RMDU is programmed for frame format, word rate, and gain by a Standalone Timing Module (STM) which is preprogrammed on a ground-based PROM programmer. The STM provides PCM outputs of serial NRZ to a telemetering transmitter and serial biphase level to the tape recorder.

4.14.2 Preamplifier Filter Unit

A preamplifier filter unit is also available for the system. This unit is used for low-level signals that require extensive filtering. The unit provides for 64 channels, with a gain from 128 to 1024 and a three-pole active low-pass filter.

4.14.3 Tape Recorder

The tape recorder is an Astro-Science (Bell & Howell) Airborne wideband FM recorder Model MAR S 1414 (LT)-3D. This unit is a 14-track analog recorder which takes a 14-inch reel of magnetic tape. A PCM bit stream from each RMDU, the time code generator output, and the pilot's voice are recorded on separate tracks. The remaining tracks are used to record active network channel outputs, if required. The tape recorder is the limiting item in the system due to its bit-packing density limitation related to the amount of tape available, speed of recording, and length of record required for flight testing. The above recorder may be operated at 30 inches per second, a record rate of 40,000 words (12-bit words) per second, which allows approximately 1 hour of full-time data recording. A Miller code is used to achieve this word rate capacity. However, the tape recorder may also be operated at seven inches per second, thereby extending inflight recording time to 4 hours by reduced data rates.

4.14.4 Time-Code Generator

The time-code generator is a Datametrics Model SP-375 Airborne Synchronized Generator with integral battery pack which produces an IRIG-B output for recording on the tape recorder and provides both a local and remotely mounted pilot's display. This unit can be synchronized with Radio Station WWV (time-standard station), and acts as the time base for the research instrumentation system.

4.15 SUPPORT EQUIPMENT

Several items of ground-support equipment are ancillary to the operation of the V/STOLAND airborne system. They are:

- Airborne Hardware Simulator (AHS)
- Peripheral Controller (PERCON)
- Portable Loader
- 1819B Control Panel
- Infoton Terminal (CRT)
- Simulation Bench

4.15.1 Airborne Hardware Simulator

The XV-15 AHS is designed to interface a Simulation computer with the V/STOLAND Data Adapter, thereby allowing the V/STOLAND system to operate in the simulation environment. Basically, the AHS converts digital data from the Simulation computer into the appropriate analog, discrete, serial digital, synchro, or parallel digital data formats accepted by the Data Adapter. In addition, the AHS converts data in the analog and discrete data formats from the Data Adapter into parallel digital data which is sent to the Simulation computer. The AHS also has the capability for the Simulation and Basic computers to communicate directly with each other via a digital-to-digital interface.

Figure 4-60 shows front, side, and rear views of the AHS. The electronics are contained in two drawers in the cabinet. The upper drawer has a control panel mounted on its face which allows the operator to examine data being received or transmitted by the AHS on the data lines between it and the Simulation computer. Detailed specifications for AHS operation, maintenance, and signal formats are provided in the following documents, respectively.

Document	Title
5720-1068	Performance Specification for Airborne Hardware Simulator
5720-MM 1068	Component Maintenance Manual for Airborne Hardware Simulator
--	V/STOLAND XV-15 Laboratory Simulation Document

4.15.2 Multiport Peripheral Controller

The XV-15 Peripheral Controller (PERCON), shown in Figure 4-61, is designed to interface two 1819B computers to the following peripheral devices:

- HP7970 Reel Magnetic Tape Transport
- Kennedy 4345 Cartridge Tape Transport
- Data Products 2310 Line Printer
- Documentation M-300 Punched Card Reader
- Infoton or Teletype

These five interfaces are mounted in the upper drawer labeled "Multiport Peripheral Controller," shown in Figure 4-62. Five lights mounted on this drawer have the following labels:

- Line Printer
- CRT KYBD
- Card Reader
- Cartridge Mag Tape
- Reel Mag Tape

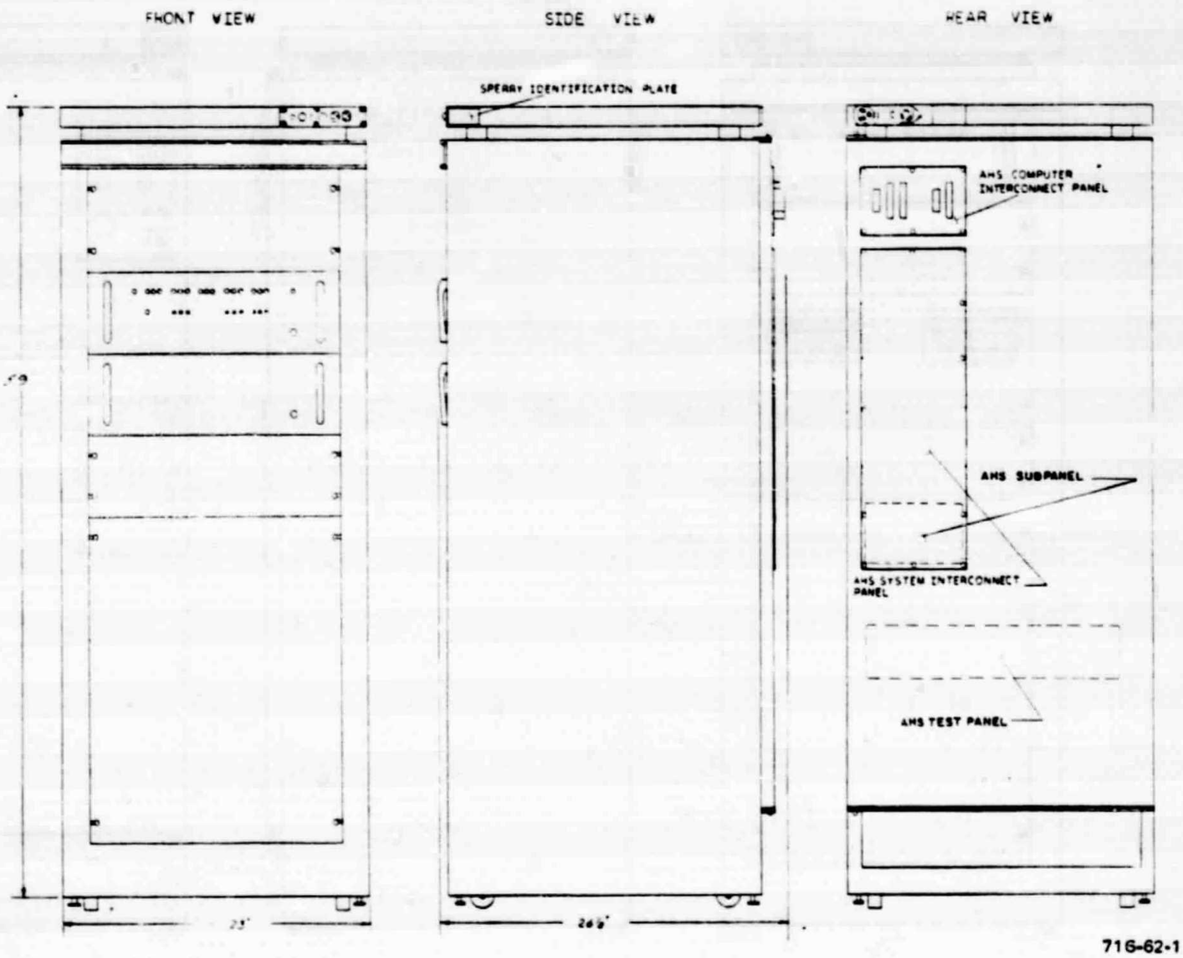


Figure 4-60
XV-15 Airborne Hardware Simulator

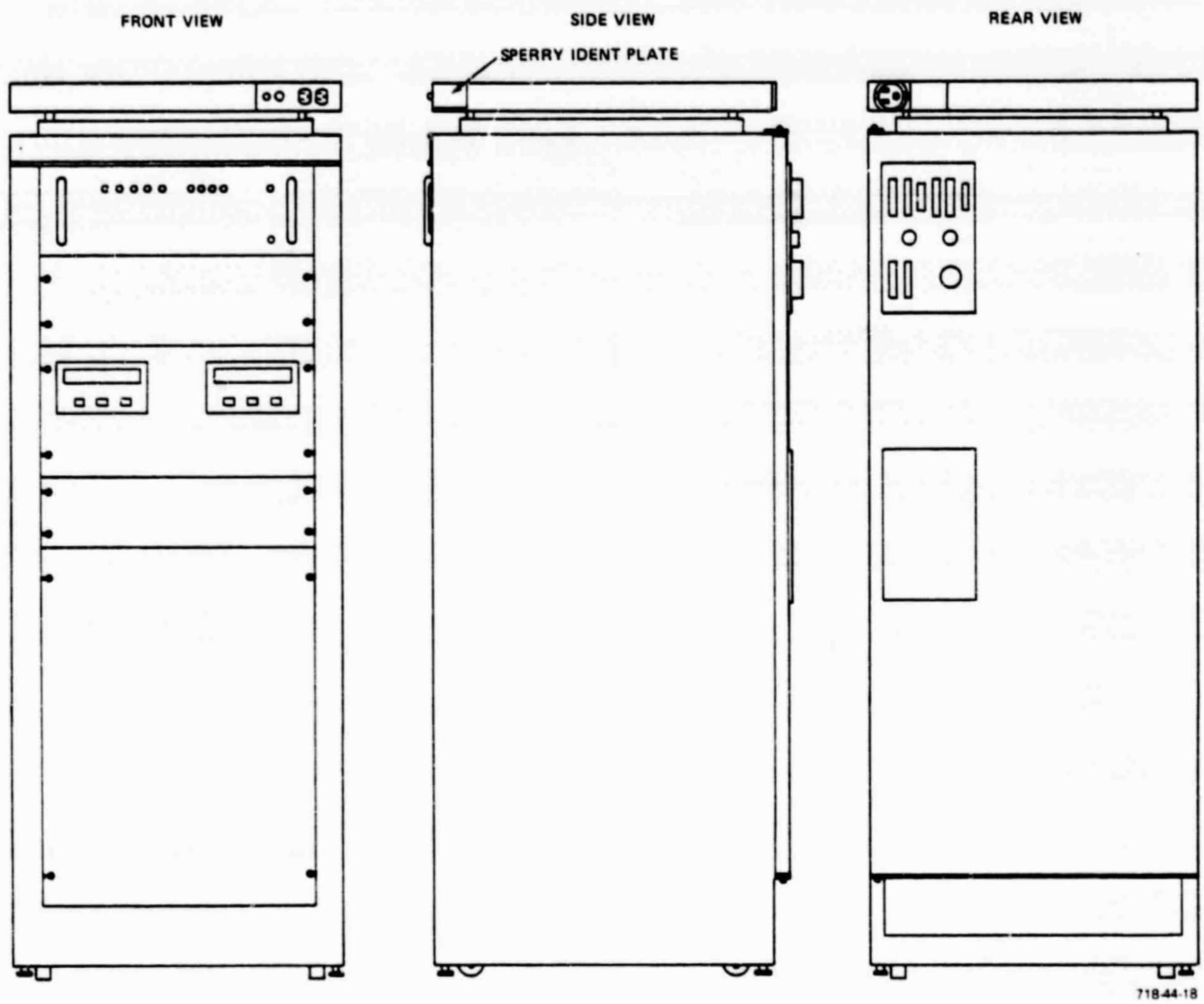


Figure 4-61
XV-15 Multiport Peripheral Controller

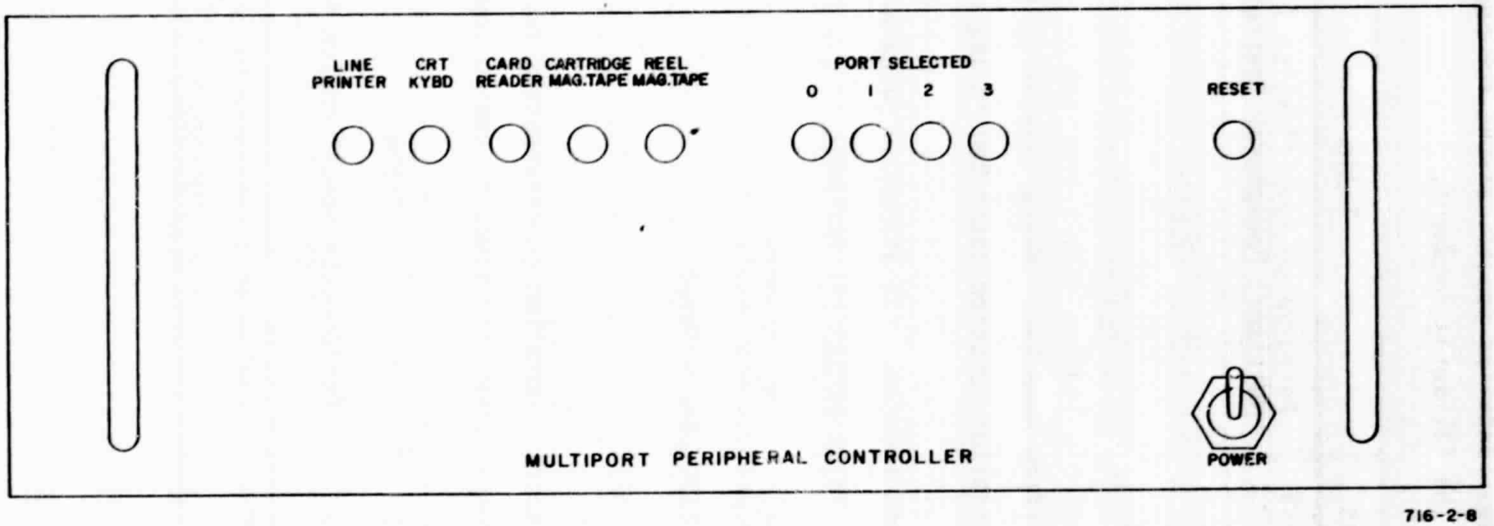


Figure 4-62
XV-15 Multiport Peripheral Controller Interface
Electronics Drawer, Front Panel

These lights indicate the active peripheral device. Four additional pushbutton lights mounted on this drawer annunciate to the computer which PERCON is communicating, and allow the operator to manually select the port that has access to the peripheral devices. Detailed specifications for the operation of the PERCON are contained in the following document.

Document	Title
5720-1069	Performance Specification for Multiport Peripheral Controller

4.15.3 Portable Loaders

Two available portable loaders enable programs to be loaded into or dumped from an 1819A or 1819B computer. These loaders perform essentially the multiport peripheral controller function described in the previous paragraph. The larger portable computer loader (Figure 4-63) is mounted on a truck for conveyance to the aircraft location. It allows an 1819A or 1819B computer to communicate with the following peripheral devices:

- HP7970 Reel Magnetic Tape Transport
- Kennedy 4345 Cartridge Tape Transport
- Data Products 2310 Line Printer
- Infoton or Teletype

Space is provided to mount an 1819A or 1819B control panel and a Kennedy 4345 dual cartridge tape transport. Detailed specifications for the operation and maintenance of this portable loader are found in the following documents:

Document	Title
5442-1050	Performance Specification for 1819A/1819B Computer Loader
544-CMM1050	Component Maintenance Manual for 1819A/1819B Computer Loader

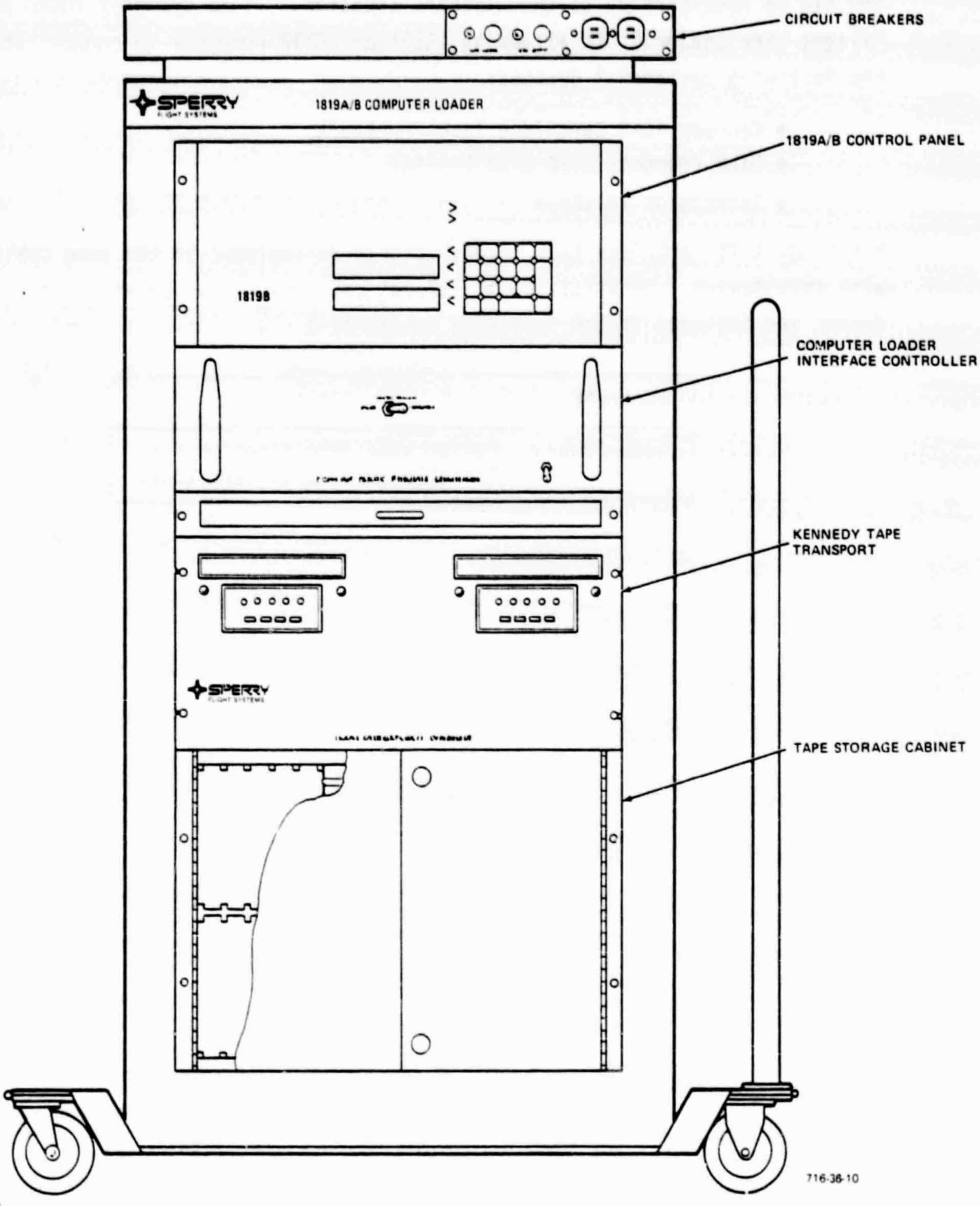


Figure 4-63
1819A/1819B Computer Loader, Front View

The smaller portable loader (Figure 4-64) weighs less than 100 pounds, and can be hand-carried to the aircraft location. It is commonly known as the "Flight Line Loader." It allows an 1819A or 1819B computer to communicate with the following peripheral devices:

- Kennedy 4345 Cartridge Tape Transport
- Data Products 2310 Line Printer
- Infoton or Teletype

A Kennedy 4345 cartridge tape transport unit is included in the same cabinet with this device. Detailed specifications concerning the operation of this device are included in the following document:

Document	Title
5710-1065	Performance Specification for Portable Peripheral Controller

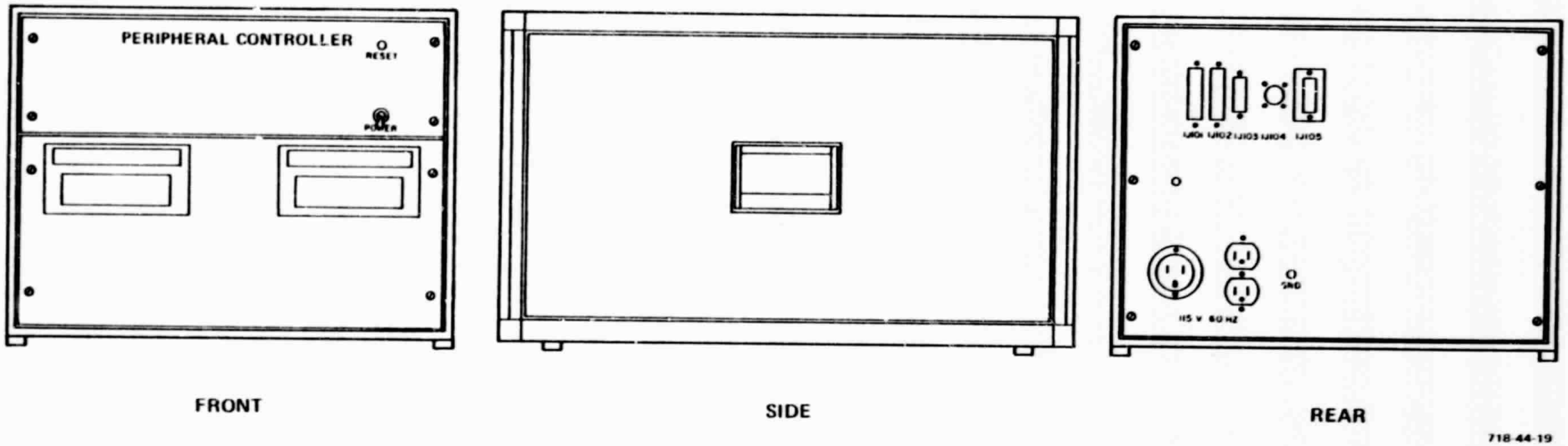


Figure 4-64
Portable Peripheral Controller

SECTION V

DESCRIPTION OF SYSTEM SOFTWARE

SECTION V

DESCRIPTION OF SYSTEM SOFTWARE

5.1 GENERAL DESCRIPTION

The XV-15 V/STOLAND software package is divided into the following set of software modules for the purposes of software specification, documentation, development, and test.

- Basic Executive and General Use Routines
- Data Files and I/O Processing
- Guidance and Control
- Navigation
- Panels and Displays
- Failure Monitoring
- Preflight Test
- Research Computer Executive and I/O

Also supplied with the package are general-use library routines (filters, integrators, etc), a short utility program, and the data files utilized by all modules. All but the Research Computer Executive and I/O, and the major share of the Preflight Test program, are resident in the Basic computer.

The supplied Basic computer software occupies essentially all of the 16,384 words of core in the Basic computer, with only 176 words of spare memory, as summarized in Table 5-1. The distribution of the 86 software sections in the four 4K memory banks is given in Table 5-2 (for the 19 June 78 listing).

The following software modules are described in this section to a greater extent than the rest of the software.

- Basic Computer Executive
- Guidance and Control
- Navigation
- Failure Monitoring and Diagnostics

Other software functions have been described to a lesser extent in Section III, Summary of System Capabilities, or in connection with the hardware descriptions in Section IV.

TABLE 5-1

TOTAL CORE USAGE

Bank	Core Used	Spare	Bank Total
0	3995*	101	4,096
1	4,068	28	4,096
2	4,091	5	4,096
3	4,054	42	4,096
Totals	16,208	176	16,384
*Includes 128 for dedicated lower core.			

TABLE 5-2

XV-15 BASIC COMPUTER CORE MAP
(19 June 78 Listing)

Bank	Section	Description	Location Range (octal)	Core Used
0	3	Basic Executive and Associated Routines	200 - 707	328
	4	Assigned Memory Locations Executive Program Variables	710 - 722	11
	5	Tabular Data Filter Tables Integrator Tables Math Tables	723 - 1524	386
	6	Keyboard Tables (Refer)	1525 - 1730	132
	7	Kayboard Tables (Modify)	1731 - 2014	52
	8	Input Data From Research Computer (Refer)	2015 - 2404	248
	9	Input Data From Research Computer (Modify); Output Data to Research Computer	2405 - 3000	252
	10	Discrete Output Data	3001 - 3100	64
	11	Simulation I/O Buffers	3101 - 3155	45

TABLE 5-2 (cont)

XV-15 BASIC COMPUTER CORE MAP
(19 June 78 Listing)

Bank	Section	Description	Location Range (octal)	Core Used
	12	Messages	3156 - 3605	280
	13	Configuration Speed Control Tables	3606 - 3631	20
	14	DDAS Buffer Variable Address and Shift Code Table	3632 - 3771	96
		(Spare Core)	3772 - 3777	(6)
	15	Data Adapter Status Word	4000 - 4000	1
	16	Executive Reference Addresses	4001 - 4002	2
	17	General Temporaries	4003 - 4016	12
	18	Math Temporaries	4017 - 4024	6
		(Spare Core)	4025 - 4027	(3)
	19	Digital Inputs (Modify)	4030 - 4157	88
	20	Digital Inputs (Refer)	4160 - 4402	147
	21	Bank 0 Indirect Table	4403 - 4437	29
		(Spare Core)	4440 - 4441	(2)
	22	Doppler Radar DMA Inputs	4442 - 4467	22
		(Spare Core)	4470 - 4475	(6)
	23	Doppler Radar DMA Inputs	4476 - 4477	2
	24	Addresses for Guidance Land and FMA Display	4500 - 4532	27
		(Spare Core)	4533 - 4535	(3)
	26	Theta and Power Lever Predict Tables	4540 - 4564	21
		(Spare Core)	4565 - 4575	(9)
	27	Spare DMA (Spare 6-Wire)	4576 - 4577	2
	28	INS No. 2	4600 - 4637	32

TABLE 5-2 (cont)

XV-15 BASIC COMPUTER CORE MAP

(19 June 78 Listing)

Bank	Section	Description	Location Range (octal)	Core Used
0	29	INS No. 3	4640 - 4663	20
		(Spare Core)	4664 - 4675	(10)
	30	INS No. 3	4676 - 4677	2
	31	Land Data	4700 - 4757	48
	32	Configuration Analog Command Table	4760 - 4771	10
		(Spare Core)	4772 - 4777	(6)
	33	Digital Instrumentation Buffers	5000 - 5277	192
	34	16-Segment Display Character Set	5300 - 5377	64
	35	MFD Output Buffer	5400 - 5577	128
	36	FMA, DED and MSP Output Buffer	5600 - 5763	116
	37	Theta and Power Lever Predict Variables	5764 - 5773	8
		(Spare Core)	5774 - 5777	(4)
	38	Analog Input Data	6000 - 6077	64
	39	Octal Constants; Decimal Constants; Degree Constants	6100 - 6363	180
		(Spare Core)	6364 - 6377	(12)
	40	Analog Output Data	6400 - 6431	26
	41	Reference Flight Path Data	6432 - 6471	32
		(Spare Core)	6472 - 6477	(6)
	42	Discrete Input Data	6500 - 6507	8
		(Spare Core)	6510 - 6517	(8)
	43	Discrete Output Words	6520 - 6523	4

TABLE 5-2 (cont)

XV-15 BASIC COMPUTER CORE MAP
(19 June 78 Listing)

Bank	Section	Description	Location Range (octal)	Core Used
0	44	Variables	6524 - 6655	90
	45	Flags, Valids, Timers, Directors, and Intermediates	6656 - 7553	446
	46	Labeled Constants	7554 - 7745	122
		(Spare Core)	7746 - 7777	(26)
1	47	Failure Monitoring	10000 - 11222	659
	48	Fast Navigation	11223 - 11640	270
	49	Slow Navigation	11641 - 13433	891
	50	General Use Routines	13434 - 13736	195
	51	Data Entry Keyboard	13737 - 16251	1227
	52	Mode Select Panel	16252 - 17313	546
	53	FMA	17314 - 17665	234
	54	Guidance Basic/Res Command Reference Selection	17666 - 17742	45
	55	Bank 1 Indirect Table	17743 - 17743	1
		(Spare Core)	17744 - 17777	(28)
2	56	HSI and ADI	20000 - 20166	119
	57	MFD Temporaries	20167 - 20174	6
	58	MFD Data (Modify)	20175 - 20366	122
	59	MFD Constants	20367 - 20423	29
	60	MFD Data (Refer)	20424 - 22013	760
	61	Movable Helix Trajectory Data	22014 - 221267	76
	62	MFD Program	22130 - 25313	1652
	63	I/O Routines (Interrupts and DMA Initiates)	25314 - 25464	105

TABLE 5-2 (cont)

XV-15 BASIC COMPUTER CORE MPA
(19 June 78 Listing)

Bank	Section	Description	Location Range (octal)	Core Used
2	64	I/O Temporaries	25465 - 25467	3
	65	I/O Decode and Encode	25470 - 26375	454
	66	DDAS Routine	26376 - 26447	42
	67	Guidance Basic/Res Control Selection	26450 - 26717	168
	68	Guidance Research MSP Switch Servicing	26720 - 26760	33
	69	Guidance Research Control Storage	26761 - 27027	39
	70	Guidance Basic/Res Mode Director Selection	27030 - 27112	51
	71	Guidance Research Mode Director Selection	27113 - 27146	28
	72	Simulation Interrupt Service Routine	27147 - 27161	11
	73	Simulation I/O Routines	27162 - 27240	47
	74	Buffer of Strip Chart Variable Addresses	27241 - 27304	36
	75	Barometric Altitude Table	27305 - 27766	306
		(Spare Core)	27767 - 27773	(5)
	76	Bank 2 Indirect Table	27774 - 27777	4
3	77	Guidance and Control Executive	30000 - 30013	12
	78	Guidance Preset Initialization	30014 - 30046	27
	79	Guidance MSP Switch and Button Servicing	30047 - 30713	421
	80	Go-Around Routines	30714 - 30773	48
	81	Guidance MSP Display Servicing	30774 - 31455	306
	82	Guidance Computations	31456 - 35254	1919

TABLE 5-2 (cont)

XV-15 BASIC COMPUTER CORE MAP
(19 June 78 Listing)

Bank	Section	Description	Location Range (octal)	Core Used
3	83	Guidance ADI and HSI Deviation Computations	35255 - 35413	95
	84	Control Computations	35414 - 37513	1088
	85	Flight Director Computations	37514 - 37706	123
	86	Bank 3 Indirect Table	37707 - 37725	15
		(Spare Core)	37726 - 37777	(42)

5.2 THE BASIC COMPUTER EXECUTIVE

The Basic Computer Executive module controls and sequences all other software modules. Specifically, it has programming for:

- Power-up initialization of the total system
- Initiating data transfers on the available channels of the 1819B
- Sequencing the different software modules
- Interfacing with the simulation computer
- Interfacing with the 1819B control panel
- Handling the different types of interrupts
- Initiating the BITE in the Basic computer during preflight and in-flight, and processing the other preflight-related data from the Research computer.

The various modules are assigned priorities for execution, based on module data requirements and individual execution times. For example, the output to the MFD occurs over most of the 50-millisecond computation cycle and, hence, the output to the MFD is initiated before the execution of other modules. The Executive is designed to operate in both the airborne and simulation environments. Prior to execution of the modules, a power-up initialization is performed.

5.2.1 Power-Up Initialization

A power-up initialization routine is called when power is first applied to the computer. This routine sets up the interrupt lockout mask and the interrupt entrance addresses, initializes the register save push/pull stacks, zeros the I/O buffers, blanks displays, and initializes flags, filters, timer, intermediates, and all other remaining data that requires initialization.

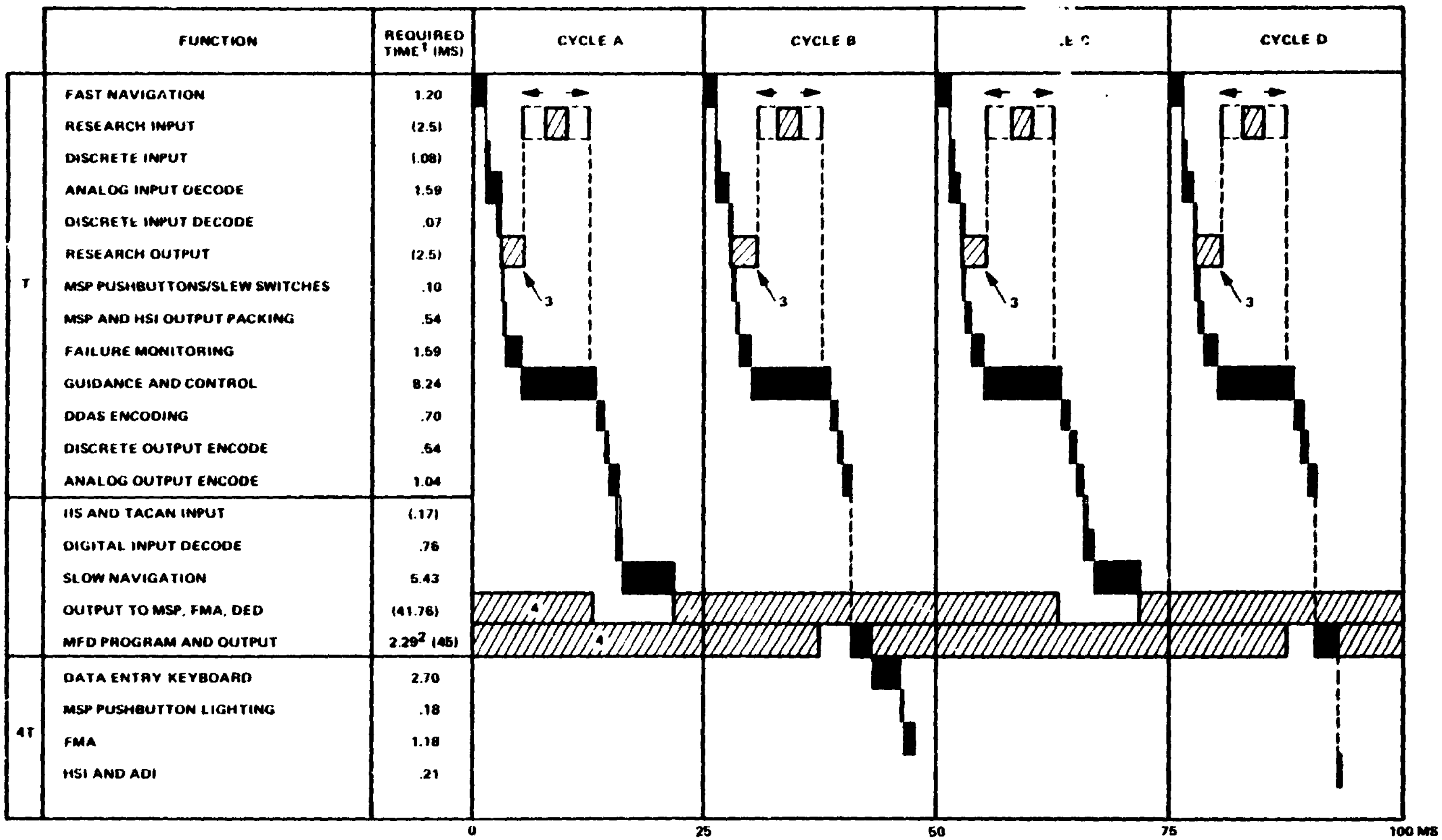
For example, the Data Entry Keyboard (DEK) letter/number pushbutton is initialized in the LETTER mode at power-up. The AUTO and Flight Director Guidance and Research mode flags are initialized to zero. The MFD flags are initialized at North-up and 5 nautical miles per inch. The real-time Clock is initialized to T-1, and the Executive sequencing flags are initiated to zero. Also, the interrupt lockout mask is set so that only real-time Clock interrupts are enabled.

When initialization is completed, the program will then proceed to the deadtime routine to await a real-time Clock interrupt. In the simulation environment, the deadtime routine consists of a real-time Utility program. In the aircraft environment, the program goes to a wait loop.

5.2.2 Software Sequencing and Timing

The control and sequencing of the execution of the various modules of the XV-15 V/STOLAND software are implemented in the module TIMER. The various modules are assigned priorities for execution, based on module data requirements and individual execution times. For example, the output data transfer to the MFD requires approximately 45 milliseconds and, hence, the data should not be computed any more often than that.

The basic program cycle time T has been implemented as an explicit parameter so that it can be easily modified without requiring revision of parameters in integrators and filters (which are programmed as functions of T). The nominal cycle time T is 25 milliseconds. Sequencing programming is designed so that various module iteration rates may be at cycle time periods of T, 2T, or 4T. An overall timing diagram is shown in Figure 5-1.



1 TIMES OBTAINED BY TRACKING ROUTINES UNDER WORST CASE (LONGEST PATH) CONDITIONS; 10 μs PER WORD ASSUMED FOR I/O TRANSFERS
2 AVERAGE TIME FOR MFD COMPUTATIONS
3 START OF THE RESEARCH COMPUTE CYCLE
4 NOT PRESENT ON POWER UP CYCLE

= COMPUTATION   = I/O TRANSFER

716 07 50

Figure 5-1
Computation Timing Diagram

The various iteration rates are provided by routines that are structured so that the execution time can be evenly distributed over several cycles. Subroutine RTN1T is executed in the fastest loop and used to call routines requiring minimum turnaround time. Functions included in this loop are:

- Fast Navigation
- Open Research Input Buffer
- Initiate Discrete Inputs
- Analog Input Decode
- Discrete Input Code
- Output to Research Computer
- Service MSP Pushbuttons and Slew Switches
- Pack MSP and HSI Digital Outputs
- Failure Monitoring
- Guidance and Control
- Instrumentation Output Encoding
- Discrete Output Encoding
- Analog Output Encoding

Subroutines RTN2TA and RTN2TB provide cycle time of 2T, and are alternately called every execution cycle. Routines not requiring minimum turnaround time are distributed between these two routines. Functions included in the 2T loops are:

- Initiate ILS and TACAN Inputs
- Digital Input Decode
- Slow Navigation
- Initiate Ouptuts to MSP, FMA and DED
- MFD

Subroutines RTN4TA, RTN4TB, RTN4TC, and RTN4TD are contained in the 4T loop, and are evenly spaced through a period of four execution cycles. Routines involving extra computation time and/or slow iteration rates are called from one of these subroutines. Functions included in the 4T loops are:

- Data Entry Keyboard
- MSP Pushbutton Lighting
- FMA
- HSI and ADI

The Executive has a provision which allows individual cycle times to exceed the basic cycle time T. As long as program execution does not fall more than one basic cycle time behind, the program will continue to cycle. If it falls more than the one basic cycle time behind, program execution will stop in the simulation environment. In the aircraft environment, EXCESSIVE CYCLE TIME will appear on the FMA, the autopilot will disengage (if engaged), and the program will start a new compute cycle. The average cycle time is monitored in this manner.

Routines may also be executed during "deadtime," that is, during the time left over at the end of each execution cycle. Deadtime routines are entered into automatically at the end of each execution cycle, provided there is time remaining during that particular compute cycle. When a deadtime routine is interrupted by the real-time Clock, the state and location of the program at the interrupt point is saved so that the deadtime routine may be entered at the same point when additional time is available in the next cycle.

For simulation and development purposes, a real-time Utility program is called during deadtime when the system is in the simulation environment (Skip Key 2 set). In the aircraft environment, deadtime consists of a simple wait loop.

5.2.3 Interrupt Handling

Real-time Clock, Fault, Power, Research Input Monitor, and Simulation Computer interrupts are handled by the Basic Executive.

- Real-Time Clock Interrupts - Real-time Clock interrupts are serviced by the TIMER routine at a frequency determined by the cycle time period T as previously described. This servicing routine is the beginning of the main flow of the Executive program. When preflight test is not in progress, normal software module sequencing is initiated.

- Fault Interrupts - When a Fault interrupt occurs the STRSTP routine is called which first stores the processor registers in designated core locations and then stops execution.

• Power-Fail Interrupts - Power-Fail interrupts are also serviced by STRSTP. In addition, the computer powers itself down (after a delay). When power is again acceptable, the machine will power itself back up and then go through the power-up initialization.

• Research Input Monitor Interrupts - The Input Monitor interrupt on Channel 0, indicating completion of the research input buffer into the Basic computer, is enabled during power-up. When such an interrupt occurs during program execution it is serviced by the CHØMNI routine. If the preflight is engaged (PFTENG≠0), the CHØMNI routine will initiate one of three additional input buffers from the Research computer on each of the first and subsequent input monitor interrupts. These input buffers from the Research computer consist of (1) output to the panels (DED, FMA, MSP), (2) analog outputs, and (3) discrete outputs.

The first two buffers are initiated with monitor to cause additional interrupts. The discrete output buffer is initiated without monitor, and no further input monitor interrupts will occur until the next compute cycle. Once these buffers are input in any given compute cycle, no more research input buffers are initiated until the research input with monitor is initiated in the next compute cycle.

• Simulation Computer Interrupts - When the airborne program is executed in the simulation environment, Channel 1 interrupts from the Simulation computer are enabled. When these interrupts occur (approximately at T_S, the simulation fast-loop cycle time), they are serviced by the SIMIO routine which performs the following functions:

- Stops all ongoing Channel 1 I/O activity
- Generates an external function transfer out on Channel 1 to clear the D/D interface between the Airborne and the Simulations computers.
- Initiates input and output buffers on Channel 1 for D/D transfer between computers.
- Forces the first output word to the Simulation computer (only required if Simulation computer is an 1819B).

5.2.4 Preflight Test

At the beginning of the Basic Computer Executive routine, TIMER, the preflight test executive is called if the preflight test has been engaged in the Research computer. The preflight test executive calls the program modules required in the Basic computer. In this branch the real-time airborne BIT is called. Should real-time BIT fail during preflight test, program execution will stop. During preflight test, the normal Basic software module sequencing is not executed.

5.2.5 MFD Output Timing

The Data Adapter requires approximately 45 milliseconds to transmit the MFD data computed in one cycle to the MFD Symbol Generator, and initiation of new MFD output before the previous output is completed results in undesirable "jumps" in the MFD display unit's image. The Basic Executive incorporates logic to circumvent this problem in both the Basic and Research MFD modes.

5.3 GUIDANCE AND CONTROL

5.3.1 General Description

The Guidance and Control (G&C) function of the V/STOLAND system provides visual commands to the pilot and/or automatic commands to the aircraft control system which guide the aircraft in accordance with the guidance modes described in the following paragraphs. The visual commands go to the Attitude Director Indicator (ADI) and Horizontal Situation Indicator (HSI) instruments, displaying vertical and lateral deviations from the reference paths, and Flight Director commands to aid the pilot in obtaining coordinated and safe capture and tracking of the paths. The automatic (autopilot) commands go to the aircraft Force-Feel System (FFS) for pitch, roll, and yaw control, to the power lever servo, the pylon conversion system, the RPM control system, and to the flap controls to provide total hands-off automatic control. Selection of the operating configuration and guidance mode is made via the Mode Select Panel (MSP).

The G&C function may be operated in three basic flight control configurations: Manual, Flight Director, and Autopilot. The Flight Director and Autopilot configurations may be engaged independently on the MSP, and the Manual configuration is in effect when neither the Flight Director nor the Autopilot have been engaged.

In the Manual configuration, deviations from the selected heading or navaid references (radial, glide slope) are displayed on the ADI and HSI. The Autopilot or Flight Director must be engaged to be able to display deviations from the programmed 3D flight-path references.

The Basic computer G&C software performs the mode control and initialization necessary to transition between the various configurations and modes, based on commands from the MSP and automatic mode transition tests. It also performs the basic control law computations to generate the Autopilot, Flight Director, and Indicator commands.

G&C computations may also be performed by the Research computer.

An important objective in designing the G&C software was to provide flexibility for modification of the control laws, which is highly desirable for a research system. Therefore, the software is structured to facilitate control-law modifications by maintaining explicit parameter storage for all gains, limits, thresholds, etc, and also by adhering to structured programming methods. Inflight modification of parameters via the keyboard and research mode capability also contribute to this objective.

The top-level organization of the Guidance and Control program is shown in Figure 5-2. Major signal paths as well as inputs and outputs to each module are also depicted in this diagram.

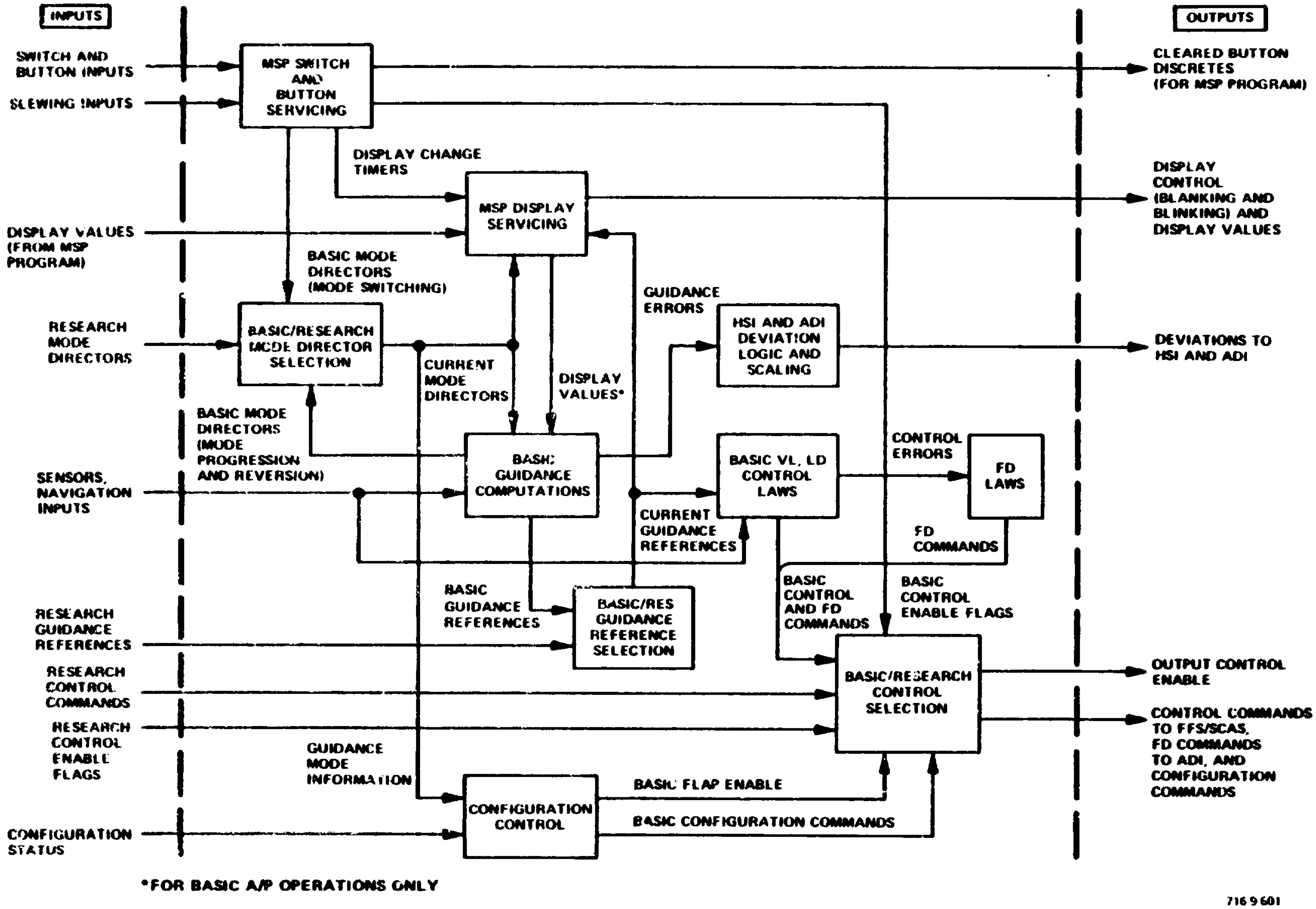


Figure 5-2
Guidance and Control Top-Level Organization

Proper operation of the program depends on the calling sequence of the basic modules, listed in calling order as follows:

1. MSP Switch and Button Servicing
2. MSP Display Servicing
3. Basic/Research Mode Director Selection
4. Guidance Computations
5. HSI and ADI Deviations
6. Basic/Research Command Reference Selection
7. Control Computations (includes configuration control)
8. Flight Director Command Computations
9. Basic/Research Control Selection

This calling order was selected to guarantee minimum delay from inputs to outputs for the Guidance and Control program.

The pilot's inputs to the MSP panel are detected and serviced in the MSP switch and button servicing module which sets flags that are used throughout the G&C program to indicate the baseline status of the program, i.e., Basic or Research Autopilot engaged (positive or negative APENG), Flight Director engaged (FLTDIR), or manual Flight Director (FDONLY). The basic mode directors are changed by this module in response to modes selected at the MSP. When the slew switches on the MSP are activated, the keyboard program changes the appropriate MSP display values. The MSP switch and button servicing module detects the changes in the MSP displays (via display change flags set by MSP program) and sets a 30-second timer used by the MSP display servicing module (described in the following paragraph) which causes the MSP displays to blink. The MSP switch and button servicing module also sets the basic control ENABLE flags when the autopilot is engaged.

The MSP display servicing module contains MSP display control and storage logic. In the manual mode, the displays are blanked. When a future guidance reference (selected via slew switch) is displayed, the associated display blinks. If a mode is engaged or armed (associated MSP button lit), the MSP display servicing module stores an associated reference value in the appropriate MSP display window. Otherwise, an actual value is stored in the display window.

The Basic/Research mode director selection module stores basic or research mode directors into the current mode directors as a function of Basic/Research mode operation. The current mode directors determine which basic guidance modes are called in the Basic guidance computation module. It is therefore possible for the Research computer, in a research mode, to control which basic guidance laws are computed.

The Basic guidance computations module computes the VL and LD guidance laws described in the following paragraphs. Each guidance mode computation subroutine contains first-pass mode synchronization logic and mode progression or reversion (fallback) logic, if any, and may therefore modify the basic mode directors.

The HSI and ADI deviations module picks up guidance errors computed in the Basic guidance module and rescales the error for output to the HSI and ADI course and vertical deviation indicators. Logic is also included that gives armed mode deviations higher priority when armed and engaged modes coexist.

The Basic/Research command reference selection module stores basic or research guidance references into current references which are used by the Basic VL, LD control laws module and the MSP display servicing module. Split or combined VL/LD reference storage (as a function of the Research computer-controlled RVLGAC and RLDGAC flags) is possible. These flags allow the Research computer to drive the basic control laws, as a research mode of operation.

The Control computations module (includes the Configuration control module) computes the basic VL/LD control and configuration control laws as described in subsequent paragraphs. Outputs of this module consist of basic commands to the force-feel system (pitch, roll, and yaw) and power lever, and basic configuration commands (rpm, pylons, flaps). This module also computes the system airspeed limits as a function of aircraft configuration.

The Flight Director command computations module computes the flight director control laws as described in subsequent paragraphs. Outputs of this module consist of basic pitch, roll, and power-lever indicator deviations which go to the ADI whenever the basic Flight Director mode is on.

The Basic/Research control selection module stores basic or research VL/LD control commands, configuration commands, flight director commands and control enable flags into output values to the vehicle control system. In Research Autopilot or Flight Director modes, split or combined VL/LD command storage is controlled by the research-generated RVLGAC and RLDGAC flags. VL, in this case, includes all configuration commands. When the Autopilot and Flight Director modes are off, the Basic/Research control selection module clears all autopilot and flight director output commands.

The system is operational whenever power is engaged at the V/STOLAND circuit breaker panel. The Multifunction Display (MFD) comes up in the North-Up mode, with a scale factor of 5 nautical miles per inch, and the Manual system configuration is in effect. The FMA annunciates the aircraft configuration as well as the system operating configuration.

The operating configurations and the guidance modes are selected on the Model Select Panel (MSP) located at top of the center console, and illustrated in Figure 5-3.

The Basic Autopilot mode is engaged by moving the solenoid-held A/P ENG switch to the BASIC position. However, the Autopilot will engage and remain engaged only if all of the following conditions are met:

1. All FFS channels are engaged and valid
2. All SCAS channels are engaged and valid
3. SCAS attitude retention is not engaged
4. The RPM governor is valid
5. There is no manual pylon conversion command
6. The computer monitor discrete is valid
7. The software monitor discrete (ENAUTO) is valid

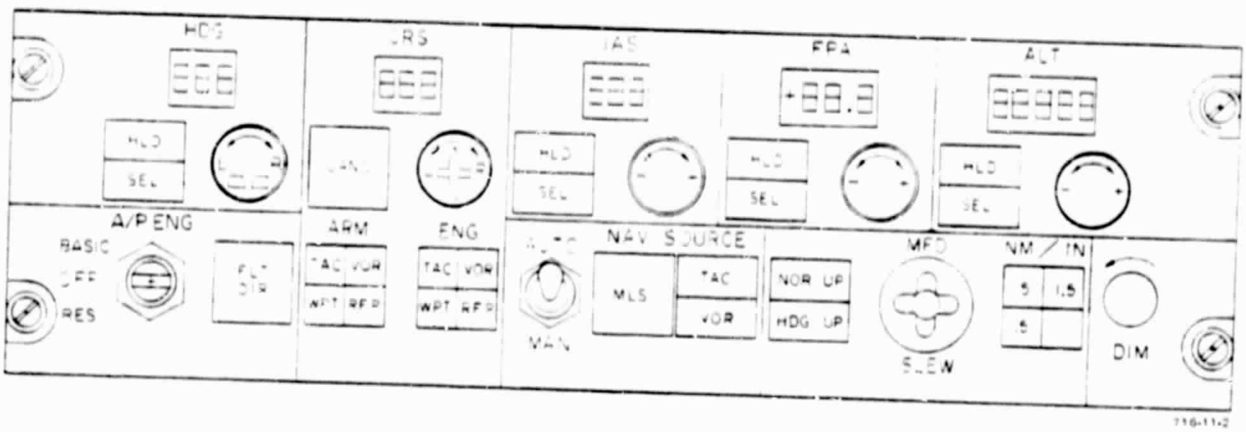


Figure 5-3
Mode Select Panel

In order to enable any Research mode, RES = n must be entered via the keyboard, where n is a non-zero number. This number is transmitted to the Research computer, and may be used to select among several research modes in the Research computer. If RES ≠ 0, the following research modes become engaged when the associated flags are set by the Research computer:

- RESAH — Reserach ADI/HSI Computed Display Data
- RESNAV — Research Navigation
- RESMFD — Research MFD

Research Guidance and Control is enabled (but not engaged) when RVLGAC or FLDGAC is set by the Research computer. The Research A/P mode may then be enaged, if RES ≠ 0 and RVLGAC and/or RLDGAC is set, by moving the A/P ENG switch to RES. The Research Flight Director mode is engaged when the FLT DIR button is pushed, if RES ≠ 0 and RVLGAC and/or FLDGAC is set, and if the A/P ENG switch is set to OFF or RES. If the A/P ENG switch is set to BASIC, the Basic FLT DIR is engaged.

Table 5-3 lists all of the guidance modes included in the Basic computer G&C software, and also presents a summary of how the modes are engaged and annunciated on the MSP. The modes are also annunciated on the Flight Mode Annunciator (FMA), as defined in Paragraph 4.10. The guidance modes are classified as independent VL modes, independent LD modes, and 3D modes which include both VL and LD submodes. The following discussions of guidance modes refer to Basic modes only, unless Research modes are specifically mentioned.

Some of the guidance modes may be armed only. Pushing the associated button on the Mode Select Panel will cause the button to light amber, indicating the armed condition. Such modes engage automatically when the engagement criteria are met. The remaining modes (except for the primary modes) are engaged when the button is pushed. Mode engagement is annunciated by the green illumination of the associated button. Initialization and synchronization of variables is accomplished as part of the mode engagement computations.

TABLE 5-3

XV-15 V/STOLAND GUIDANCE MODES

Guidance Mode	Engagement Method	MSP Annunciation
Primary VL (VL PRM)	A/P switch to Basic position; Automatic fallback mode	None
Airspeed Hold (IAS HLD)	Push IAS HLD/SEL; automatic transition from IAS SEL	IAS HLD segment green
Airspeed Select (IAS SEL)	Push IAS HLD/SEL when display blinks	IAS SEL segment green
Flight-Path Angle Hold (FPA HLD)	Push FPA HLD/SEL; Automatic transition from FPA SEL	FPA HLD segment green
Flight-Path Angle Select (FPA SEL)	Push FPA HLD/SEL while display blinks	FPA SEL segment green
Altitude Hold (ALT HLD)	Push ALT HLD/SEL; automatic transition from ALT SEL	ALT HLD segment green
Altitude Select (ALT SEL)	Armed by pushing ALT HLD/SEL; engagement is automatic	ALT SEL segment amber for armed, green for engaged
Primary LD (LD PRM)	A/P switch to Basic position; Automatic fallback mode	None
Heading Hold (HDG HLD)	Push HDG HLD/SEL; automatic transition from HDG SEL, and from primary LD if $\phi \leq 5$ deg	HDG HLD segment green
Heading Select	Push HDG HLD/SEL when display blinks	HDG SEL segment green
TACAN Course (TAC CRS)	Armed by pushing CRS ARM; engagement is automatic	TAC ARM segment amber for armed, TAC ENG segment green for engaged
VOR Course (VOR CRS)	Armed by pushing CRS ARM; engagement is automatic	VOR ARM segment amber for armed, VOR ENG segment green for engaged
Waypoint Course (WPT CRS)	Armed by pushing CRS ARM; engagement is automatic	WPT ARM segment amber for armed, WPT ENG segment green for engaged

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TABLE 5-3 (cont)

XV-15 V/STOLAND GUIDANCE MODES

Guidance Mode	Engagement Method	MSP Annunciation
LD Reference Flight Path (LD RFP)	Armed by pushing RFP ARM; engagement is automatic	RFP ARM segment amber for armed, RFP ENG segment green for engaged
VL Reference Flight Path (VL RFP)	Automatically armed when LD RFP is engaged; engagement is automatic	None
LD LAND1	Enter LND = 1 on keyboard. Push LAND to arm; engagement is automatic	LAND button amber for armed, green for engaged
VL LAND1	Automatically armed when LD LAND1 is engaged; engagement is automatic	None
LD LAND2	Enter LND = 2 on keyboard. Push LAND to arm; engagement is automatic	LAND button amber for armed, green for engaged
VL LAND2	Automatically armed when LD LAND2 is engaged, engagement is automatic	None
Go-Around	Armed automatically when any LD LAND mode is armed or engaged; Push Go-Around button on control stick to engage	None

5.3.2 Guidance Mode Descriptions

5.3.2.1 Mode Control

The guidance and control software performs all of the mode control logic associated with guidance and control, including initiation of filters, integrators and flags. Guidance mode control is achieved with the mode directors shown in Table 5-4. The state of guidance at any time is specified by the values of the "current" mode directors. These values are stored from the basic values if basic guidance is selected, or from the research values if research guidance is selected. The RGACVL and RGACLD flags also permit mixed Basic/Research mode operation, under Research computer control. The mode director assignments are shown in Table 5-5.

TABLE 5-4
GUIDANCE MODE DIRECTORS

Current	Basic	Research	Director Description
VRTENG	BVRTEN	RVRTEN	Vertical Mode Engaged
VRTARM	BVRTAR	RVRTAR	Vertical Mode Armed
LONENG	BLONEN	RLONEN	Longitudinal Mode Engaged
LONARM	BLONAR	RLONAR	Longitudinal Mode Armed
LDENG	BLDEN	RLDEN	Lateral/Directional Mode Engaged
LDARM	BLDARM	RLDARM	Lateral/Directional Mode Armed

TABLE 5-5

MODE ASSIGNMENTS

Engaged Modes			Armed Modes		
Director	Value	Mode	Director	Value	Mode
VRFTENG	0	OFF	VRTARM	0	OFF
	1	FPA HLD		1	(Not Assigned)
	2	FPA SEL		2	(Not Assigned)
	3	ALT HLD		3	(Not Assigned)
	4	ALT SEL		4	ALT SEL
	5	Not Assigned		5	Not Assigned
	6	VRT RFP		6	VRT RFP
	7	VRT LND		7	VRT LND
LONENG	0	VL PRIMARY	LONARM	0	OFF
	1	IAS HLD		1	(Not Assigned)
	2	IAS SEL		2	Go-Around Velocity
	3	LND VEL		3	LND VEL
LDENG	0	LD PRIMARY	LDARM	0	OFF
	1	HDG HLD		1	(Not Assigned)
	2	HDG SEL		2	(Not Assigned)
	3	TAC		3	TAC
	4	VOR		4	VOR
	5	WPT		5	WPT
	6	LD RFP		6	LD RFP
	7	LD LND		7	LD LND

The following paragraphs briefly describe the various Guidance modes. Details on each mode are presented in Paragraph 5.3.4, "Guidance Computations."

5.3.2.2 Independent Vertical/Longitudinal Guidance Modes

• Primary VL Mode - The primary VL mode is a fallback guidance mode which is in effect when no other VL mode has been selected. It goes in effect when A/P or FLT DIR is initially engaged, or when the other VL modes are disengaged. In this mode, attitude hold is commanded to the pitch controls at the value coincident with mode engagement. The power lever receives no command, and may be operated manually.

• Indicated Airspeed Hold (IAS HLD) - IAS HLD is engaged in transition from IAS SEL. If the IAS HLD/SEL button is pushed when the IAS modes are off, the coincident indicated airspeed shown on the IAS display becomes the control reference. Since the capture conditions are met immediately, the IAS HLD mode immediately engages and the HLD segment lights green. The IAS HLD mode also engages, via select, similarly whenever any other VL mode engages, unless IAS SEL is already engaged.

If IAS HLD is the only engaged VL mode, the IAS command goes to the pitch axis. The power lever then receives no command and may be operated manually.

If other VL modes are engaged, IAS commands go to both the pitch- and power-lever controls as a function of the aircraft configuration.

IAS HLD can be disengaged by pushing the IAS HLD/SEL button if no other vertical mode is engaged. The IAS display then reverts to actual airspeed. IAS HLD also disengages by engagement of IAS SEL or by the automatic velocity control in the LAND modes.

• Indicated Airspeed Select (IAS SEL) - IAS SEL is engaged by first slewing the IAS display to a new desired reference value and then pushing the IAS HLD/SEL button. The SEL segment will light green and the HLD segment will be blanked if it was lit. The IAS display will blink for 30 seconds after the reference has been changed and before IAS SEL is engaged. This indicates that the selected reference is not yet the control reference. After 30 seconds, the IAS display reverts to actual IAS. When IAS SEL is enaged, the IAS commands go to the control axes described for IAS HLD and cause the aircraft to approach the selected IAS reference. IAS SEL converts to IAS HLD when the selected speed is reached.

If the IAS HLD/SEL button is pushed while IAS SEL is engaged, the mode will disengage. If another VL mode is engaged, IAS HLD will simultaneously engage.

• Flight Path Angle Hold (FPA HLD) - FPA HLD is engaged in transition from FPA SEL when the reference FPA is attained. When FPA HLD engages, the HLD segment lights green.

The guidance laws in this mode compute altitude rate commands to both the pitch- and power-lever controls and/or directors to hold the aerodynamic flight-path angle, $\gamma = \tan^{-1}(\dot{h}/V_T)$.

If ALT SEL is not armed, FPA HLD may be disengaged by pushing the FPA HLD/SEL button. VL guidance then reverts to an IAS-only mode on pitch. If ALT SEL is armed and FPA HLD is engaged, pushing the FPA HLD/SEL button has no effect. FPA HLD is also disengaged by engagement of the other vertical modes.

• Flight Path Angle Select (FPA SEL) - FPA SEL is engaged by first slewing the FPA display to a new desired value and then pushing the FPA HLD/SEL button. The SEL segment will light green and the HLD segment will be blanked if it was lit. The displayed value may be changed while FPA SEL is engaged.

In general, FPA SEL engages when the FPA HLD/SEL button is pushed and FPA SEL is not already engaged. If FPA SEL is already engaged, it will disengage when the FPA HLD/SEL button is pushed. When FPA SEL engages, the value displayed in the FPA window becomes the FPA reference. The reference value of FPA may be preselected via the FPA slew switch. The FPA display will blink for 30 seconds after the reference has been changed and before FPA SEL is engaged. This indicates that the selected reference is not yet the control reference. After 30 seconds the FPA display reverts to actual FPA, if FPA SEL is not engaged. For this reason, the FPA HLD/SEL button should be pushed within

30 seconds after slewing if FPA SEL at the preselected reference is desired. FPA SEL will engage at the current actual (displayed) value of FPA if the FPA HLD/SEL button is pushed when a reference has not been preselected via the slew switch.

FPA SEL also engages when ALT SEL is armed. In this case the computer automatically computes and displays the flight-path angle required to reach the selected altitude in 1 minute, subject to a maximum limit of ±8 degrees, a minimum limit of ±1 degree, and a maximum $\dot{h}$ limit of ±1000 feet per minute. However, this angle may be modified by manually slewing the displayed flight-path angle. Engagement of FPA SEL also causes IAS HLD to engage unless IAS SEL is already engaged.

The FPA commands go to the pitch- and power-lever controls and/or flight directors, and cause the aircraft to smoothly approach the selected FPA. FPA SEL transitions to FPA HLD when the selected FPA is reached to within specified limits.

FPA SEL may be disengaged manually by pushing the FPA SEL button. FPA SEL also disengages in the normal transition to FPA HLD.

- <u>Altitude Hold (ALT HLD)</u> - ALT HLD is engaged in transition from ALT SEL. When the ALT HLD/SEL button is pushed, the coincident altitude shown on the ALT display becomes the reference altitude. Prior to engaging the ALT HLD/SEL button, the displayed value on the ALT display may either represent a selected reference, distinguished by a blinking display, or it may continuously display actual data in a non-blinking fashion. When the ALT HLD/SEL button is pushed while it is not lit, the ALT SEL mode temporarily becomes armed and then engaged. However, since the altitude capture conditions are quickly satisfied, ALT HLD soon engages and the HLD segment lights green. The control commands go to the pitch- and power-lever controls and/or flight directors to hold the selected altitude.

ALT HLD may be disengaged by pushing the ALT HLD button when the display is not blinking. THe ALT display then reverts back to displaying actual data. VL guidance then reverts to the primary VL mode and IAS HLD (or IAS SEL if engaged). ALT HLD is also disengaged by engagement of the other vertical modes.

• Altitude Select (ALT SEL) - ALT SEL is armed by first slewing the ALT display to a new desired altitude reference and then pushing the ALT HLD/SEL button. The SEL segment will light amber, indicating that the mode is armed, and the FPA SEL mode will engage as described above. When the capture criteria are met, ALT SEL engages, FPA SEL or HLD disengages, and the ALT SEL segment changes to green.

When ALT SEL is engaged, the guidance laws command the aircraft to gradually level out the flight path to the selected non-flashing altitude on the digital altitude display.

When the altitude comes within specified limits of the selected altitude, ALT SEL converts to ALT HLD and the digital altitude display shows the altitude reference that is currently being used.

If the ALT/HLD SEL button is pushed while ALT SEL is armed, the mode is disengaged. Also, if the ALT HLD/SEL button is pushed while ALT SEL is engaged, or ALT HLD is engaged and the display indicates a non-flashing active reference, the mode is disengaged. However, if the ALT HLD/SEL button is pushed when in either the ALT SEL or HLD engaged modes and the display indicates a flashing non-active reference, the system will revert to the ALT SEL armed mode.

5.3.2.3 Independent Lateral-Directional Guidance Modes

• Primary LD MODE - The primary LD mode is a fall-back guidance mode which is in effect when no other LD mode has been selected or automatically engaged. It goes into effect when

- A/P becomes engaged
- FLT DIR becomes engaged while AUTO is not engaged
- All other LD guidance modes become disengaged while AUTO or FLT DIR remains engaged

If the roll attitude is less than ±5 degrees when this mode is engaged (and the airspeed is not below 30 knots if manual flight director is on), HDG HLD becomes engaged as described in the next paragraph. If the roll angle is greater than ±5 degrees (or the airspeed is below 30 knots when manual Flight Director is on), the guidance laws command a constant roll angle equal to the value coincident with the primary LD mode engagement.

• Heading Hold (HDG HLD) - HDG HLD is engaged in transition from HDG SEL. If HDG SEL is called from primary LD, or by pushing the HDG HLD/SEL button when the HDG modes are off, the coincident heading displayed on the HDG display becomes the heading reference. HDG SEL temporarily becomes engaged and transitions to HDG HLD when specified capture criteria are met. The HLD segment then lights green.

If the airspeed is above 60 knots, the heading command goes to the roll-axis control and/or Flight Director and the yaw axis controls provide turn coordination. In the hover condition, the heading command goes to the yaw-axis controls and no command goes to the roll-axis controls. (This is equivalent to a wings-level command.) Between 60 knots and hover, the heading command is blended from roll-axis control to the yaw axis.

HDG HLD disengages when other LD modes engage, or when the HDG HLD/SEL button is pushed while the HDG display is not blinking and the roll attitude is greater than ±5 degrees (roll attitude condition does not apply for manual Flight Director below 30 knots). The HDG display then reverts to actual heading.

• Heading Select (HDG SEL) - HDG SEL is engaged by first slewing the HDG display to a new desired heading and then pushing the HDG HLD/SEL button. The SEL segment will light green and the HLD segment will be blanked, if it was lit. The display value may be changed while HDG SEL is engaged, and HDG display operation is as described for IAS and FPA SEL modes. The heading commands to the controls and/or Flight Director described in the previous paragraph cause the heading to approach the selected value and, when within capture limits, the mode transitions to HDG HLD.

The mode may be disengaged by pushing the HDG HLD/SEL button or by engaging other LD modes, thereby causing the HDG display to revert to actual heading.

• Course Guidance (TAC CRS, VOR CRS, WPT CRS) - The three course guidance modes (TAC CRS, VOR CRS, and WPT CRS) are functionally identical, and differ only in the reference data and associated processing. These modes, as well as the Reference Flight-Path mode, are armed by the single CRS ARM button. When the button is blank, a single push arms the TAC CRS mode and lights the TAC segment amber. Additional pushes sequentially arm VOR CRS, WPT CRS, RFP, and then blank the button. Hence, up to four pushes may be required to arm the desired mode.

Course reference values may be slewed in at the MSP before or after the desired CRS mode is armed. The CRS display will blink for 30 seconds after the last CRS reference change, and then revert to displaying the actual aircraft course if no CRS mode has been armed. If a CRS mode is armed, the CRS display will blink continuously, indicating a future reference, until the CRS mode engages automatically.

An armed course mode engages when the capture conditions are met. However, a time-delay function inhibits engagement for 3 seconds after the CRS ARM button or CRS slew switch were last activated. This prevents inadvertent engagement to undesired radials. When a course mode engages, the associated ENG segment lights green, the ARM segment is blanked, and the CRS display stops blinking.

When a CRS mode is engaged:

- A second CRS mode may be armed, as described above, without disenaging the engaged mode. The CRS display and the ADI/HSI course deviations will be associated with the armed mode.

- Pushing the ENG button disengages the engaged mode, blanks the ENG segment, and reverts the display to actual course if no CRS mode is armed.

- If the waypoint location is moved (via the keyboard) while WPT CRS is engaged, this mode reverts to the armed condition and primary LD engages.

- When the data for the engaged CRS mode becomes invalid, the mode reverts to the armed condition and primary LD engages.

When no CRS mode is engaged, pushing the CRS ENG button has no effect.

5.3.2.4 Three-Dimensional Guidance Modes

- Reference Flight Path Guidance (RFP) - The reference flight path illustrated in Figure 5-4 is a three-dimensional flight path defined by a series of waypoints that are connected by a straight or circular line. The waypoints are referenced to the runway coordinate frame and stored in the Basic computer data bank. This data is also accessed by the MFD program which displays the reference flight path on the MFD as part of the map display.

The LD RFP submode is armed as described above for the CRS modes. The desired entrance waypoint number must also be entered via the keyboard. The course reference associated with a waypoint is fixed and not selectable as for the other CRS modes. The RFP course reference is displayed in the CRS display when the LD RFP mode is armed, or when engaged with no other course or LAND mode armed.

When LD RFP submode has engaged, the VL RFP submode is armed and engages independently when the vertical capture conditions are met.

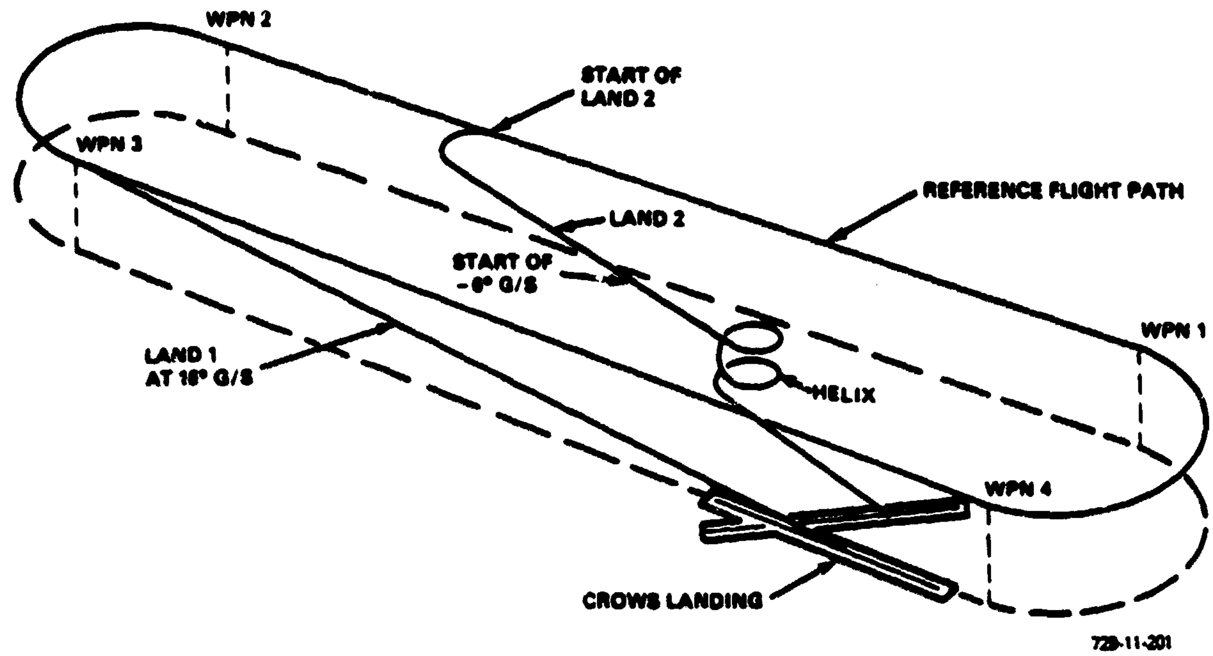


Figure 5-4
Reference Flight Path and LAND Trajectories

• <u>Automatic Landing Guidance (LAND)</u> - If the MLS navaid data is available and valid, there is a choice between two approach trajectories for LAND, which must be selected via the keyboard before arming LAND. The selection and a brief description of the approach modes are as follows:

LAND1 - Selected via the keyboard as LND = 1 before LAND is armed. With MLS, the glide slope is selectable (GSR) via the keyboard between -2.0 and -8.0 degrees. However, to capture this approach from the reference flight path, the selected glide slope must be less than -6.0 degrees. Figure 5-4 illustrates the LND1 trajectory with GSR = -6 degrees.

LAND2 - Selected via the keyboard as LND = 2 before LAND is armed. The trajectory consists principally of an 18-degree localizer approach to a touchdown point on Runway 30. The approach path includes a 2-revolution helix as illustrated in Figure 5-4.

LAND may be armed manually by pushing the LAND button on the Mode Select Panel. If MLS navaid data is valid and the navigation is valid, LAND engages automatically when the capture criterion for the selected approach is satisfied.

A LAND velocity submode is activated for the final portion of LAND, where ground velocity is controlled, decelerating the vehicle to touchdown. This submode is armed when VL LAND engages. It engages automatically, and is annunciated on the FMA when it engages.

• <u>Go-Around</u> - The Go-Around mode is armed whenever a LAND mode is armed or engaged. The mode will engage when the Go-Around button, located on the research pilot's control stick, is pushed while the Go-Around mode is armed.

When the Go-Around mode is engaged, a GO-AROUND annunciation appears on the upper FMA unless it is temporarily overridden by a configuration message. When the Go-Around mode engages, an automatic altitude-select to 2500 feet ASL is initiated and airspeed is held. After a 5-second delay, an airspeed-select to 60 knots will be initiated if the airspeed is below 60 knots; otherwise, airspeed hold will remain in effect. These modes are annunciated on the MSP and lower FMA.

If Go-Around is initiated from a LAND1 approach, an immediate heading-select to the runway heading of 353 degrees occurs. If Go-Around is initiated from a LAND2 approach, the existing roll attitude will be held until the aircraft heading falls within a ±20-degree range of the runway heading (353 degrees). When this condition is satisfied, a heading-select to the runway heading will be initiated and annunciated on the MSP and lower FMA.

After Go-Around has been initiated, it may be cancelled at any time by pushing the Go-Around button again, or by pushing any other button on the MSP. When this occurs, the Go-Around annunciation on the upper FMA will be cancelled and the modes currently engaged on the MSP will remain engaged.

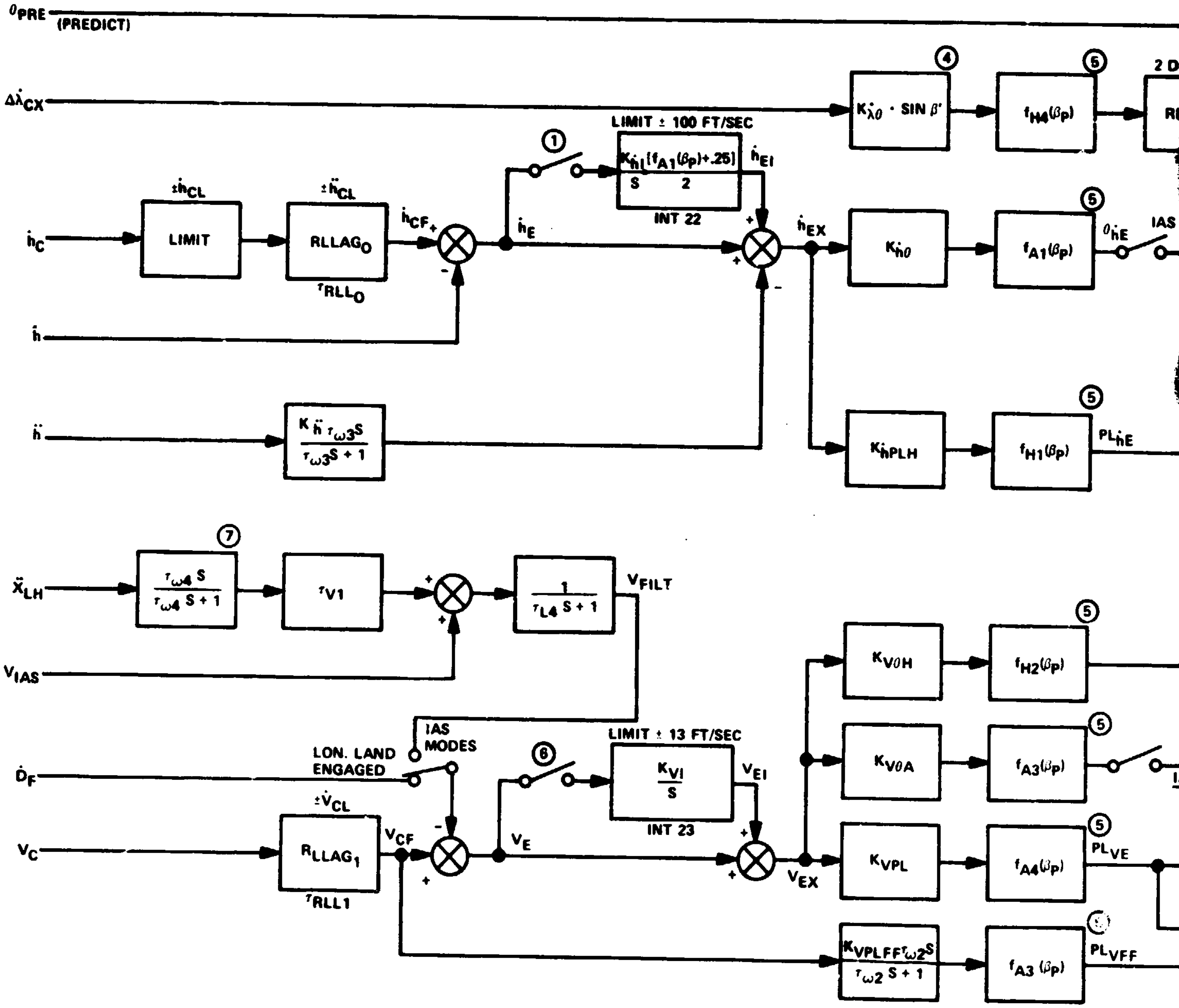
5.3.3 Control Computations

5.3.3.1 Vertical-Longitudinal Control

The VL control laws generate autopilot commands, θ_{FC} and δ_{PLC}, to the pitch FFS and the power-lever position servo, respectively. The input commands to the VL control laws, illustrated in Figure 5-5, depend on the selected guidance mode as follows:

- Primary VL mode: θ_0
- IAS only modes: V_C
- Other VL modes: V_C, $\dot{h}_C$

where θ_0, V_C and $\dot{h}_C$ are pitch, velocity, and altitude rate commands, respectively. For the primary VL and IAS-only modes, the power-lever servo system is not engaged, and the pilot may operate the power lever manually. For the IAS-only mode, the power lever then becomes a rate-of-climb actuator in both the airplane and helicopter configurations. For the other VL modes, $\delta\theta_{FC}$ and δ_{PLC} are based on both $\dot{h}_C$ and V_C, with degree-of-blend dependent on the pylon conversion angle, β_p.



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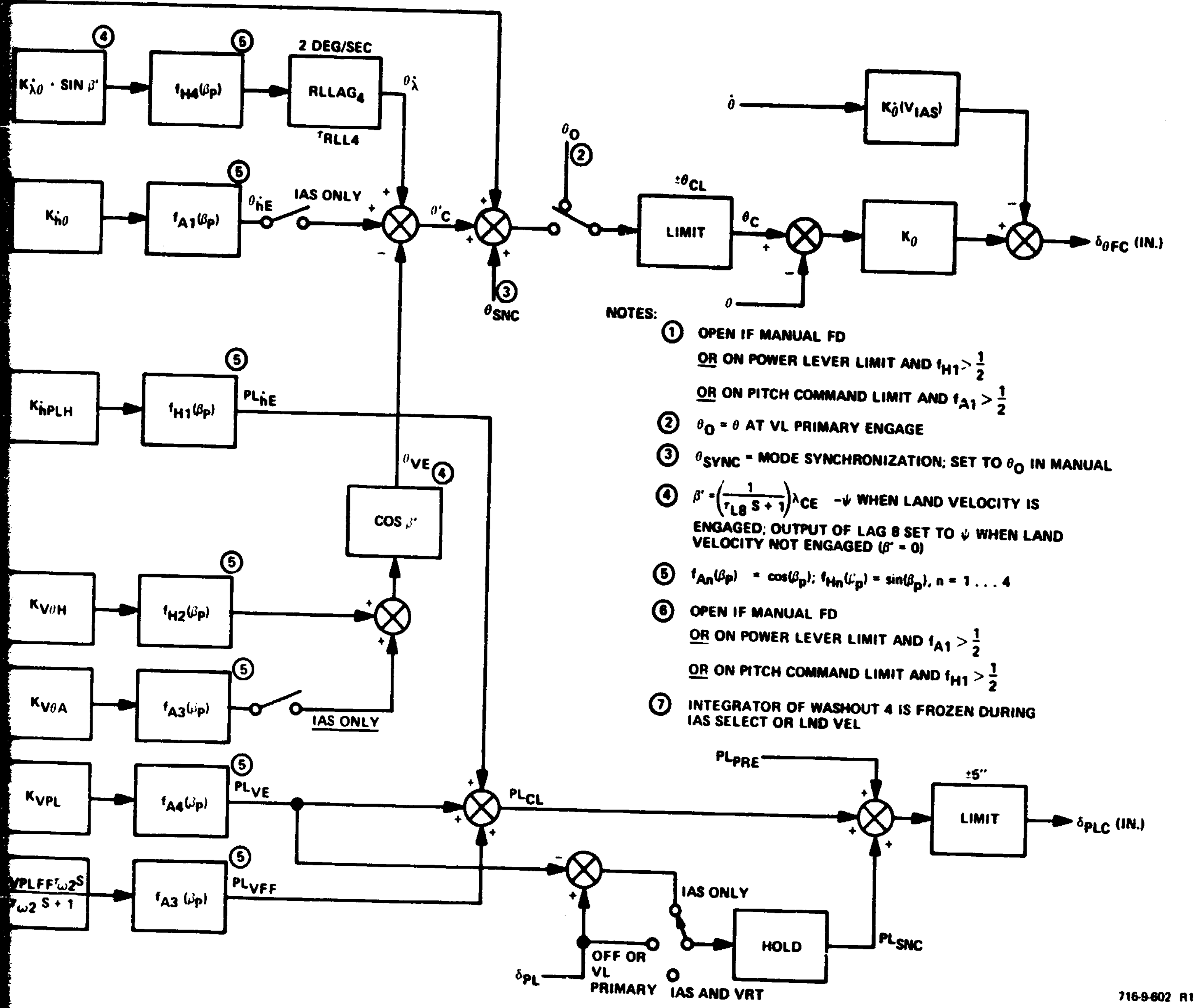


Figure 5-5
Vertical/Longitudinal
Control Block Diagram

FOLDOUT FRAME 2

Definitions of the symbols used in Figure 5-5, Vertical/Longitudinal Control Block Diagram, are given in Table 5-6 which follows. Associated gains, limits, and time constants are specified in Table 5-7.

TABLE 5-6
VL CONTROL-RELATED SYMBOLS

Symbol	Description	Units
$\dot{h}_C$	Altitude Rate Command (unfiltered)	ft/s
$\dot{h}$	Altitude Rate Feedback	ft/s
$\ddot{h}$	Vertical Acceleration	ft/s^2
$\dot{h}_{CF}$	Filtered Altitude Rate Command	ft/s
$\dot{h}_{CL}$	Altitude Rate Command Limit	ft/s
$\ddot{h}_{CL}$	Altitude Rate Command Rate Limit (acceleration limit)	ft/s^2
$\dot{h}_E$	Altitude Rate Error	ft/s
$\dot{h}_{EI}$	Altitude Rate Control Integral Term	ft/s
$\dot{h}_{EX}$	Altitude Rate Error plus Integral Term	ft/s
$\theta\dot{h}_E$	Pitch Command for $\dot{h}$ Error in Airplane Mode	deg
$\theta_{\dot{\lambda}}$	Pitch Command for Lateral Error (crossed control)	deg
θ_{PRE}	Pitch Predict	deg
θ_C'	Total Closed-Loop Portion of Pitch Command	deg
θ_{SNC}	Pitch Command Synchronization Term (mode transitions)	deg
θ_O	VL Primary Pitch Attitude Hold Command	deg
θ	Pitch Attitude Feedback	deg
θ_C	Total Pitch Attitude Command	deg
θ_{VE}	Pitch Attitude Command for Speed Error	deg
$\dot{\theta}$	Pitch Rate Feedback/Euler Axis, from Navigation	deg/s
PL_{CL}	Total Closed-Loop Portion of Power-Lever Command	in

TABLE 5-6 (cont)
VL CONTROL-RELATED SYMBOLS

Symbol	Description	Units
PL_{SNC}	Power-Lever Command Synchronization Term	in
PL_{VE}	Power-Lever Command for Speed Error in Airplane Mode	in
PL_{VFF}	Power-Lever Speed Reference Feedforward in Airplane Mode	in
$PL_{\dot{h}E}$	Power-Lever Command for Altitude Rate Error in Helicopter Mode	in
PL_{PRE}	Power-Lever Predict	in
V_{IAS}	Indicated Airspeed (from air data)	ft/s
V_{FILT}	Airspeed Complementary Airspeed Output	ft/s
V_C	Speed Command	ft/s
V_{CF}	Filtered Speed Command	ft/s
V_E	Speed Error	ft/s
V_{EI}	Speed-Control Integral Term	ft/s
V_{EX}	Speed Error plus Integral Term	ft/s
$\Delta\dot{\lambda}_{CX}$	Course-Rate Command plus Course-Rate Integral (from LD control)	deg/s
$\ddot{X}_{LH}$	Longitudinal Acceleration in Local Horizontal (from navigation)	ft/s^2
$\dot{D}_F$	Along-Axis Approach Horizontal Ground-Speed Feedback	ft/s
δ_{PL}	Power-Lever Position	in
$\delta_{\theta FC}$	Pitch Force Feel Command	in
δ_{PLC}	Power-Lever Force-Feel Command	in

TABLE 5-7

VL CONTROL GAINS, LIMITS AND TIME CONSTANTS

Symbol	Description	Value	Units
K_{θ}	Pitch Attitude Gain	.35	in/deg
$K_{\dot{\theta}}(V_{IAS})$	Pitch Attitude Damping Gain	$.1732(\frac{90}{V_{IAS}+90})$	in(deg/s)
$K_{\dot{\lambda}\theta}$	Course to Pitch Helicopter Mode Gain	1.0	in/(deg/s)
$K_{\dot{h}\theta}$	Altitude Rate to Pitch Airplane Mode Gain	.20	deg/(ft/s)
$K_{\dot{h}PLH}$	Altitude Rate to Power Lever Helicopter Mode Gain	.25	in/(ft/s)
$K_{\dot{h}I}$	Altitude Rate Integral Gain	1.0	None
$K_{\ddot{h}}$	Altitude Rate Damping Gain	.3	s
$K_{V\theta H}$	Speed Error to Pitch Helicopter Mode Gain	1.0	deg/(ft/s)
$K_{V\theta A}$	Speed Error to Pitch Airplane Mode Gain	1.0	deg/(ft/s)
K_{VPL}	Speed Error to Power-Lever Airplane Mode Gain	.3	in/(ft/s)
K_{VPLFF}	Speed Command Feedforward to Power Lever (Airplane)	.3	in/(ft/s)
K_{VI}	Speed Integral Gain	.1	None
$\dot{h}_{CL}$	Altitude Rate Command Limit	16.67	ft/s
h_{CL}	Altitude Rate Command Rate Limit	1.5	ft/s^2
$\dot{V}_{CL}$	Speed-Command Rate Limit	1.5*	ft/s^2
*$\dot{V}_{LND}$ in Long. Land; Now set at 1.5 also.			

TABLE 5-7 (cont)

VL CONTROL GAINS, LIMITS AND TIME CONSTANTS

Symbol	Description	Value	Units
τ_{RLL0}	Altitude Rate Command RLL Time Constant	.5	s
τ_{RLL1}	Speed Command RLL Time Constant	.5	s
τ_{RLL4}	Course Error to Pitch Roll Time Constant	.5	s
τ_{L4}	Airspeed Complementary Filter Time Constant	4	s
τ_{V1}	Airspeed Complementary Filter Time Constant	4	s
τ_{W2}	Speed-Command Feedforward Washout Time Constant	3	s
τ_{W4}	Local Horizontal Acceleration Washout Time Constant	15	s
τ_{L8}	β' Phase-in Time Constant	10	s

VL control initialization and mode synchronization are accomplished as defined by the following equations:

a) When the Autopilot and Flight Director are off or in VL primary:

$\dot{h}_C = \dot{h}$ ft/s

$\dot{h}_{CF} = \dot{h}$ ft/s

$\dot{h}_{EI} = 0$ ft/s

$V_{FILT} = V_{IAS}$ ft/s

$BV_C^* = V_{IAS}$ ft/s

$V_{CF} = V_C$ ft/s

$V_{EI} = 0$ ft/s

Washout 2 Integrator = V_C ft/s

Washout 2 Remainder = 0 ft/s

*Basic value for V_C

$\theta_{SNC} = \theta_o$ deg

$PL_{SNC} = PL$ in

Washout 3 Integrator = h ft/s^2

Washout 4 Integrator = X_{LH} ft/s^2

$$\left.\begin{array}{l}\theta_{PRE} \text{ synchronized to } 0 \\ \\ PL_{PRE} \text{ synchronized to } 0\end{array}\right\} \quad \text{See PRESNC Subroutine}$$

b) In IAS-Only Mode:

$\dot{h}_{EI} = 0$ ft/s

$\dot{H}_{CF} = \dot{h}$ ft/s

$PL_{SNC} = PL - PL_{VE}$ in.

PL_{PRE} synchronized to 0 (See PRESNC subroutine, B = 1)

c) At transition from VL primary or IAS-only modes to IAS and VRT modes:

$\theta_{SNC} = \theta_{SNC} - K_{V\theta A} \cdot f_{A3}(\beta_p) \cdot V_{EX} \cdot \cos\beta' - \theta_{\dot{h}E}$ deg

d) At transition form IAS and VRT modes to IAS-only mode:

$\theta_{SNC} = \theta_{SNC} + K_{V\theta A} \cdot f_{A3}(\beta_p) \cdot V_{EX} \cdot \cos\beta' + 0$

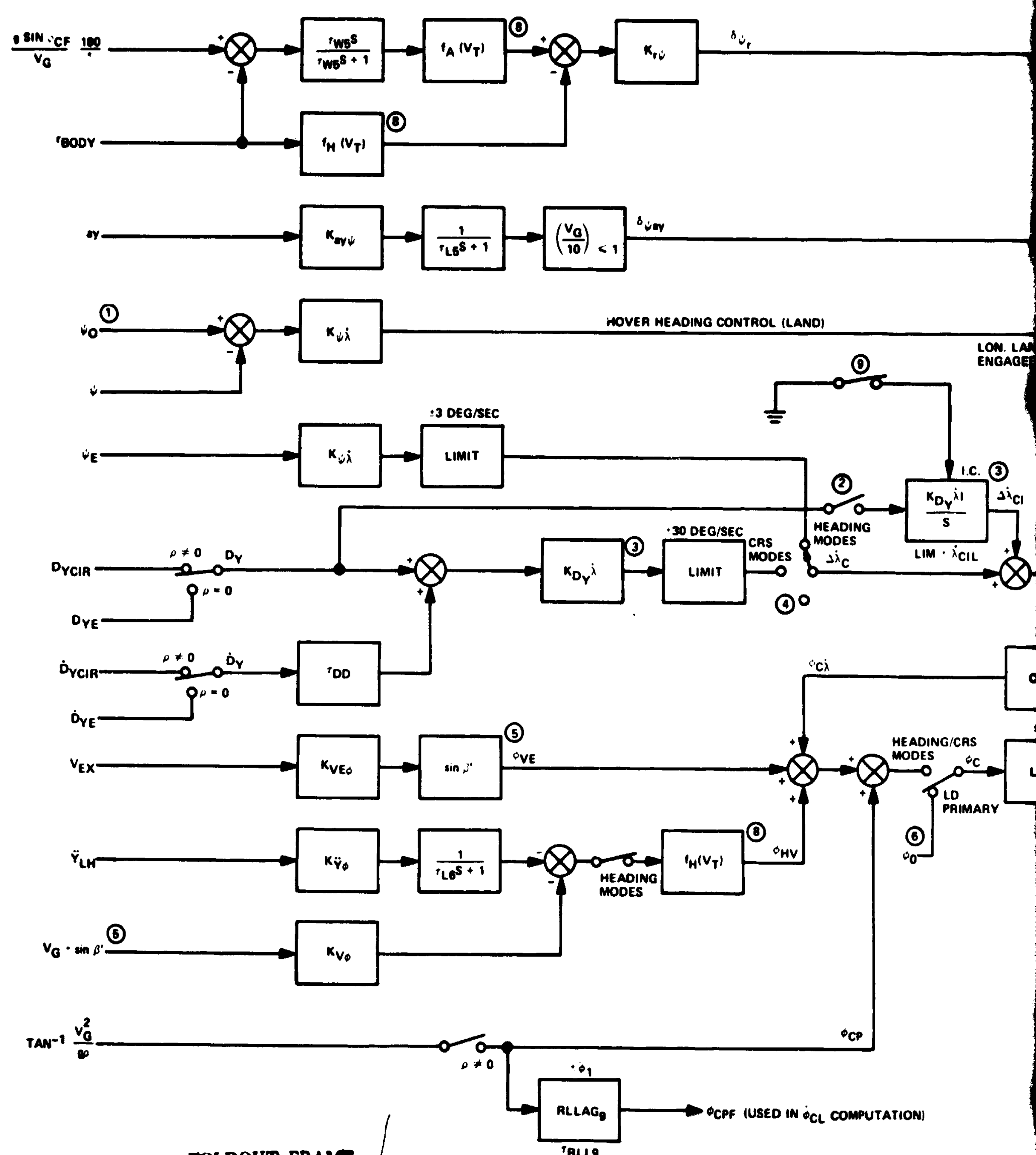
5.3.3.2 Lateral-Directional Control

The LD control laws generate autopilot command $\delta_{\phi FC}$ and $\delta_{\psi FC}$ to the roll and yaw FFS inputs. The LD control laws are illustrated in Figure 5-6. Symbol definitions are given in Table 5-8. Associated gains, limits, and time washouts are specified in Table 5-9.

The roll FFS command is the output of a roll attitude control loop closed through K_{ϕ} and $K_{\dot{\phi}}$. Other guidance loops are closed around the roll control loop. When course modes are engaged, roll-control commands are generated from crosstrack guidance errors. During LAND approaches at low speeds in crosswind, when the drift angle (β') becomes large, speed error, generated in VL control, is cross-coupled into the roll command and crosstrack control is phased out of the roll command.

When the autopilot is in heading-hold or select, heading control is achieved with roll and yaw command mixing as a function of airspeed. As the aircraft slows from slow helicopter flight to hover, heading control is shifted from the roll axis to the yaw axis.

The directional control command is used to achieve yaw damping, turn coordination, and low-speed heading control. At speeds close to hover, the turn coordination term is phased out. During the final portion of a deceleration to hover in a LAND mode, a heading-hold function is automatically switched into the yaw control law.



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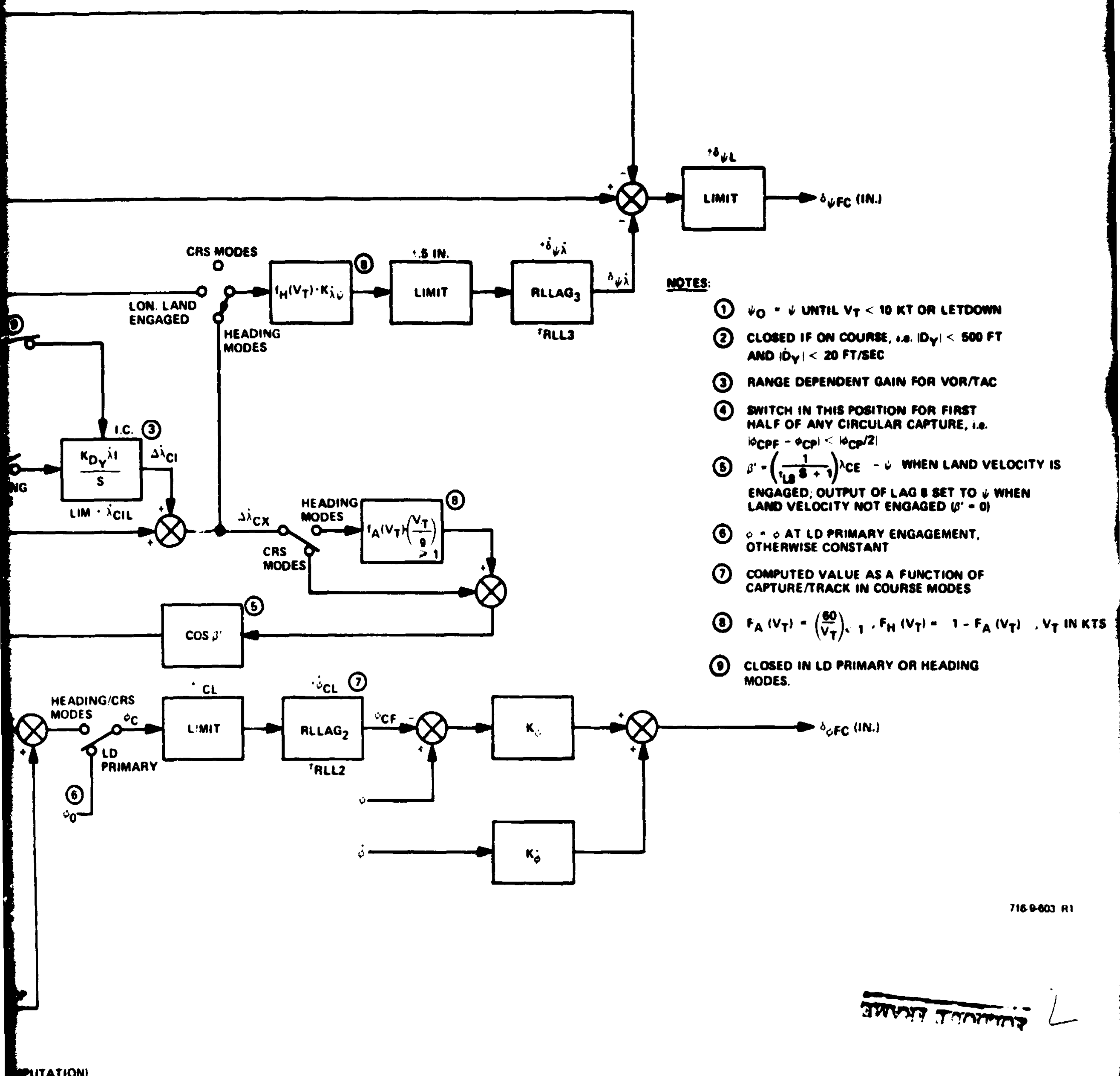


Figure 5-6
Lateral-Directional
Control Block Diagram

TABLE 5-8

LD CONTROL-RELATED SYMBOLS

Symbol	Description	Units
ϕ_{CF}	Filtered Roll-Attitude Command	deg
ϕ_C	Roll-Attitude Command (unfiltered)	deg
ϕ	Roll Attitude	deg
ϕ_{CL}	Roll-Attitude Command Limit	deg
$\phi_{C\dot{\lambda}}$	Roll-Attitude Command for Course Rate Error	deg
ϕ_{VE}	Roll-Attitude Command for Speed Error (cross-coupled)	deg
ϕ_{HV}	Roll-Attitude Command for Helicopter Turn Coordination	deg
ϕ_o	LD Primary Roll-Attitude Command	deg
ϕ_{CP}	Roll-Attitude Predict	deg
ϕ_{CPF}	Filtered Roll-Attitude Predict	deg
$\dot{\phi}$	Roll-Rate Euler Axis (from navigation)	deg/s
$\dot{\phi}_{CL}$	Roll-Attitude Command Rate Limit	deg/s
$\dot{\phi}_1$	Desired Roll Rate to Predicted Bank Angle	deg/s
$\Delta\dot{\lambda}_C$	Course Rate Command for Heading or Crosstrack Error	deg/s
$\Delta\dot{\lambda}_{CI}$	Crosstrack Integral Term	deg/s
$\Delta\dot{\lambda}_{CX}$	Course Rate Error plus Integral Command	deg/s
$\delta_{\psi r}$	Yaw Rate Damping Directional Command	in
$\delta_{\psi ay}$	Turn Coordination Directional Command	in
$\delta_{\psi\dot{\lambda}}$	Directional Command for Heading Error	in
$\delta_{\psi FC}$	Total Directional Command to Force-Feel System	in
$\dot{\delta}_{\psi\dot{\lambda}}$	Rate Limit Value of Directional Heading Command	in/s
ψ_E	Heading Error from Guidance	deg
ψ_o	Hover Heading Command in LAND	deg

TABLE 5-8 (cont)

LD CONTROL-RELATED SYMBOLS

Symbol	Description	Units
ψ	Aircraft Heading	deg
D_{YE}	Engaged Crosstrack Error for Straight Segments (guidance)	ft
D_{YCIR}	Engaged Crosstrack Error for Circular Segments (guidance)	ft
D_Y	Combined Crosstrack Control Error	ft
$\dot{D}_{YE}$	Engaged Crosstrack Rate for Straight Segments (guidance)	ft/s
$\dot{D}_{YCIR}$	Engaged Crosstrack Rate for Circular Segments (guidance)	ft/s
$\dot{D}_Y$	Combined Crosstrack Rate	ft/s
V_G	Navigation-Derived Ground Speed	ft/s
V_{EX}	Speed Error plus Integral from VL Control	ft/s
a_y	Lateral Acceleration, Body Axis	ft/s^2
$\ddot{Y}_{LH}$	Lateral Acceleration, Local Horizontal (navigation)	ft/s^2
β'	Aircraft Drift Angle, Using Course Reference	deg
ρ	Control Radius	ft
V_T	True Airspeed (air data)	ft/s
λ_{CE}	Engaged Course Mode Course Reference, wrt M.N.	deg

TABLE 5-9
LD CONTROL GAINS, LIMITS, AND TIME CONSTANTS

Symbol	Description	Value	Units
K_{ϕ}	Roll-Attitude Gain	.35	in/deg
$K_{\dot{\phi}}$	Roll-Rate Gain	.15	in/(deg/s)
$K_{DY\dot{\lambda}}$	Path Error to Course-Rate Gain	-.05**	(deg/s)/ft
$K_{DY\dot{\lambda}I}$	Path Error Integral Term	-.001**	(deg/s)/ft
$K_{VE\phi}$	Speed-Error Crossfeed to Roll Gain	.25	deg/(ft/s)
$K_{\ddot{Y}\phi}$	Helicopter Lateral Acceleration Gain (Turn Coordination)	0	deg/(ft/s^2)
$K_{V\phi}$	Helicopter Side Velocity Gain (Turn Coordination)	0	deg/(ft/s)
$K_{\psi\dot{\lambda}}$	Heading Error to Course-Rate Command Gain	.15	(deg/s)/deg
$K_{\dot{\lambda}\psi}$	Helicopter Mode Yaw-Heading Control Gain	.1	in/(deg/s)
$K_{a_y\psi}$	Lateral Acceleration, Turn Coordination Gain	2.5	in/(ft/s^2)
$K_{r\psi}$	Yaw-Rate Damping Gain	.1	in/(deg/s)
ϕ_{CL}	Roll-Attitude Command Limit	10° for CRS Mode and ρ = 0, otherwise 30°	deg
$\dot{\phi}_{CL}$	Roll-Attitude Command Rate Limit	5 or $\dot{\phi}_1$**	deg/s
$\dot{\phi}_1$	Nominal Roll Rate during Capture Roll Maneuver	3.5	deg/s

*Multiply by $L_{\infty,1}$ $(\frac{6}{R})$ for VOR/TAC; where R = Range to STN (mi)

**Use $\dot{\phi}_1$ if $|\phi_{CPF}-\phi_{CE}|$ > 1° (rolling maneuver)

TABLE 5-9 (cont)

LD CONTROLS, GAINS, LIMITS, AND TIME CONSTANTS

Symbol	Description	Value	Units
$\dot{\delta}_{\psi}$	Directional Heading Command Rate	.5	in/s
$\delta_{\psi L}$	Total Force-Feel Directional Command Limit	2.55	in
$V_{T_{LHH}}$	Airspeed Below Which LAND Heading Hold Mode Engages	16.9	ft/s
τ_{RLL2}	Roll Command RLL Time Constant	.2	s
τ_{RLL3}	Directional Heading Command RLL Time Constant	.2	s
τ_{RLL9}	Filtered Roll-Predict RLL Time Constant	.1	s
τ_{L5}	Lateral Acceleration Filter Time Constant	.5	s
τ_{L6}	Lateral Acceleration, Local Horizontal Filter Time Constant	.5	s
τ_{DD}	Crosstrack Rate Damping Time Constant	10	s
τ_{W5}	Yaw Rate Damping Washout Time Constant	5	s
τ_{L8}	β' Phase-in Time Constant	10	s

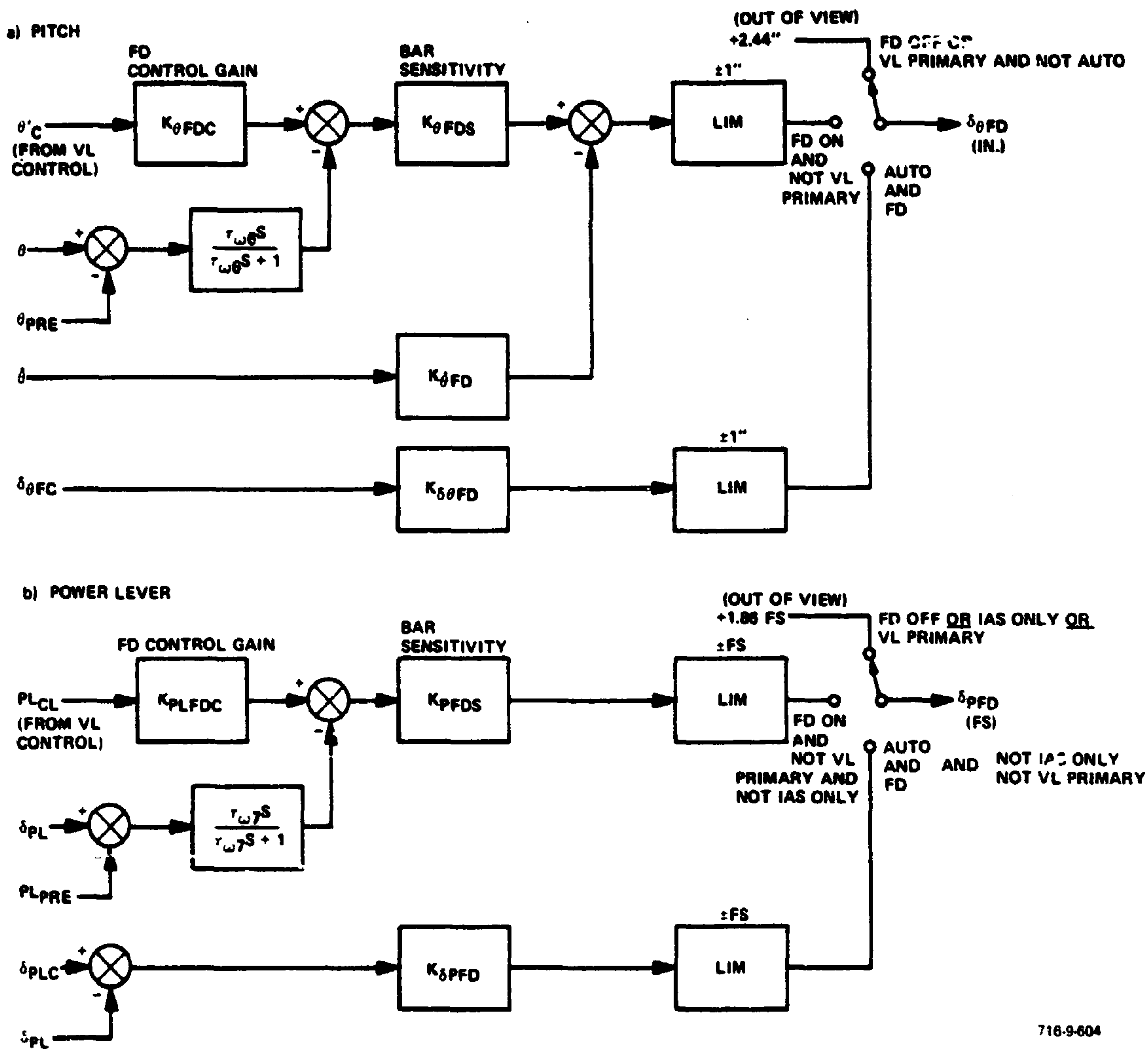
5.3.3.3 Flight Director Equations

- Pitch Flight Director - The pitch flight director law is illustrated in Figure 5-7. The law uses θ_C^L, the closed-loop portion of the pitch autopilot attitude command from VL control. Note that pitch altitude is washed out to provide an integration effect. Also, $\dot{h}_{EI}$ and V_{EI} (refer to Figure 5-5) are set to zero when the Flight Director is ON and Autopilot OFF. The flight director law incorporates (via θ_C^L), the same control blending as a function of pylon angle as the Autopilot. Gains and time constants used for the pitch Flight Director are shown in Table 5-10. (Refer to Table 5-6 for symbol definitions.)

The pitch flight-director command, $\delta_{\theta FD}$, drives the horizontal flight-director bar on the ADI. Conventional polarity is used.

- Power Lever Flight Director - The power lever flight-director law is illustrated in Figure 5-7. The law uses PL_{CL}, the closed-loop portion of the power-lever autopilot command. The power-lever position feedback is washed out to provide an integration effect. As mentioned above for the pitch Flight Director, the autopilot control integrators are set to zero when the Flight Director is on and the Autopilot is off. The same pylon angle blending used by the autopilot control law is incorporated in the power-lever flight director via the PL_{CL} term. Gains and time constants used for the power-lever flight director are shown in Table 5-10. (Refer to Table 5-6 for symbol definitions.)

The power-lever flight director command, δ_{PFD}, drives an indicator which moves over a vertical scale on the left side of the ADI. The UP direction commands an increase in power lever.



NOTE: $\dot{h}_{EI}$ AND V_{EI} SET TO 0 IF FD ONLY

Figure 5-7
XV-15 VL Flight-Director Computations

TABLE 5-10

PITCH AND POWER-LEVER FLIGHT DIRECTOR GAINS AND TIME CONSTANTS

Symbol	Description	Value	Units
$K_{\theta FDC}$	Pitch Flight-Director Control Error Gain	$2.5 \quad 1 - L_{.8397, \infty} \frac{\beta p}{45}$	deg/deg
$K_{\theta FDS}$	Pitch Flight-Director Bar Sensitivity Gain	0.1	in/deg
$K_{\dot{\theta} FD}$	Pitch Flight-Director Attitude Rate Gain	0	in/(deg/s)
$K_{\delta \theta FD}$	Auto Mode Pitch FD Gain	0.5	in
τ_{W6}	Pitch Attitude Feedback Washout Time Constant	20	s
K_{PLFDC}	Power-Lever Flight Director Control Error Gain	1.0	in/in
K_{PFDS}	Power-Lever Flight Director Bar Sensitivity Gain	0.3	in/in
$K_{\delta PFD}$	Auto Mode Power-Lever FD Gain	0.5	in/in
τ_{W7}	Power-Lever Position Feedback Washout Time Constant	5	s

● Roll Flight Director - The roll flight-director law is illustrated in Figure 5-8. The law uses the autopilot filtered roll attitude command (refer to Figure 5-6) derived in the LD control law. When the Flight Director is ON and the Autopilot is OFF, the autopilot integral control term, $\Delta\dot{\lambda}_{CI}$, is set to zero. The roll flight director gains are summarized in Table 5-11. (Refer to Table 5-8 for symbol definitions.)

The roll flight-director command, $\delta_{\phi FD}$, drives the vertical flight-director bar on the ADI. Conventional polarity is used.

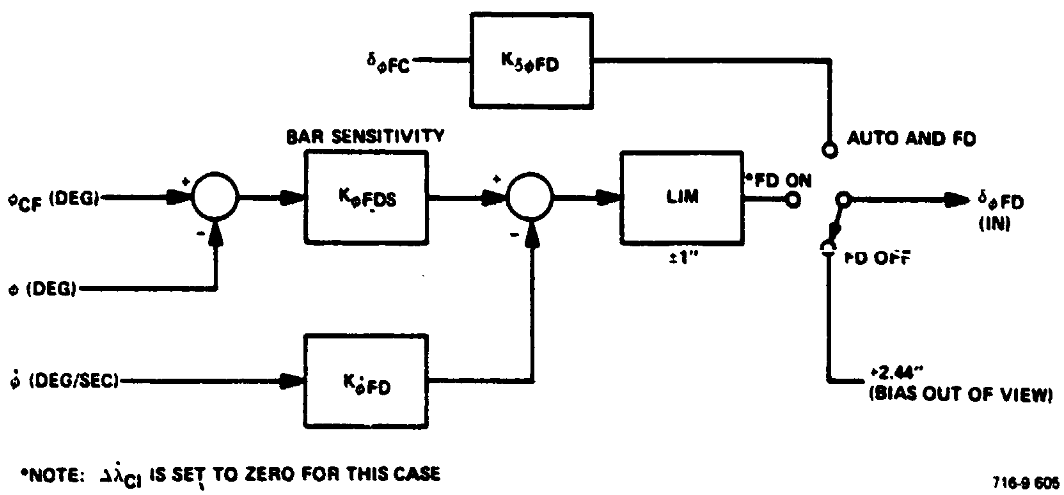


Figure 5-8
Roll Flight-Director Computations

TABLE 5-11

ROLL FLIGHT-DIRECTOR GAINS

Symbol	Description	Value	Units
$K_{\phi FDS}$	Roll Flight-Director Bar Sensitivity Gain	.0165	in/deg
$K_{\dot{\phi} FD}$	Roll Flight-Director Roll Rate Gain	0	in/(deg/s)
$K_{\delta\phi FD}$	AUTO Mode Roll Flight-Director Gain	.5	in/in

5.3.3.4 Vertical/Longitudinal Predict Equations

The VL control predict equations are defined in Figure 5-9. The pitch attitude and power-lever predict outputs, θ_{PRE} and PL_{PRE}, are inputs to the VL control laws described in Paragraph 5.3.3.1. These outputs, derived from control reference values and configuration information, are used to reduce the control errors of the closed-loop control paths during control reference or configuration changes (thereby improving control performance at a given gain level). The outputs are added to the autopilot and flight-director pitch- and power-lever commands. They represent estimates of required steady-state pitch attitude and power-lever changes due to control reference and configuration changes.

In the VL primary mode, the predict terms are synchronized to zero by continuous updating of θ_{P0} and PL_{P0}. In the IAS-only mode, the power-lever predict term continues to be synchronized while the pitch-predict term becomes active (θ_{P0} is no longer updated). In the IAS and VRT mode, both predict terms are active.

5.3.3.5 Airspeed Envelopes

The system airspeed limits in effect when the Autopilot or Flight Director are ON are defined in Figure 5-10. The minimum and maximum airspeed limits are a function of the pylon angle, as shown in the figure. The system will not allow the airspeed-command reference to go outside or exist outside the region defined by the minimum and maximum airspeed limits. If the pilot attempts to engage the Autopilot or Flight Director while the airspeed is outside the allowable airspeed region, or if he attempts to fly from a speed within the

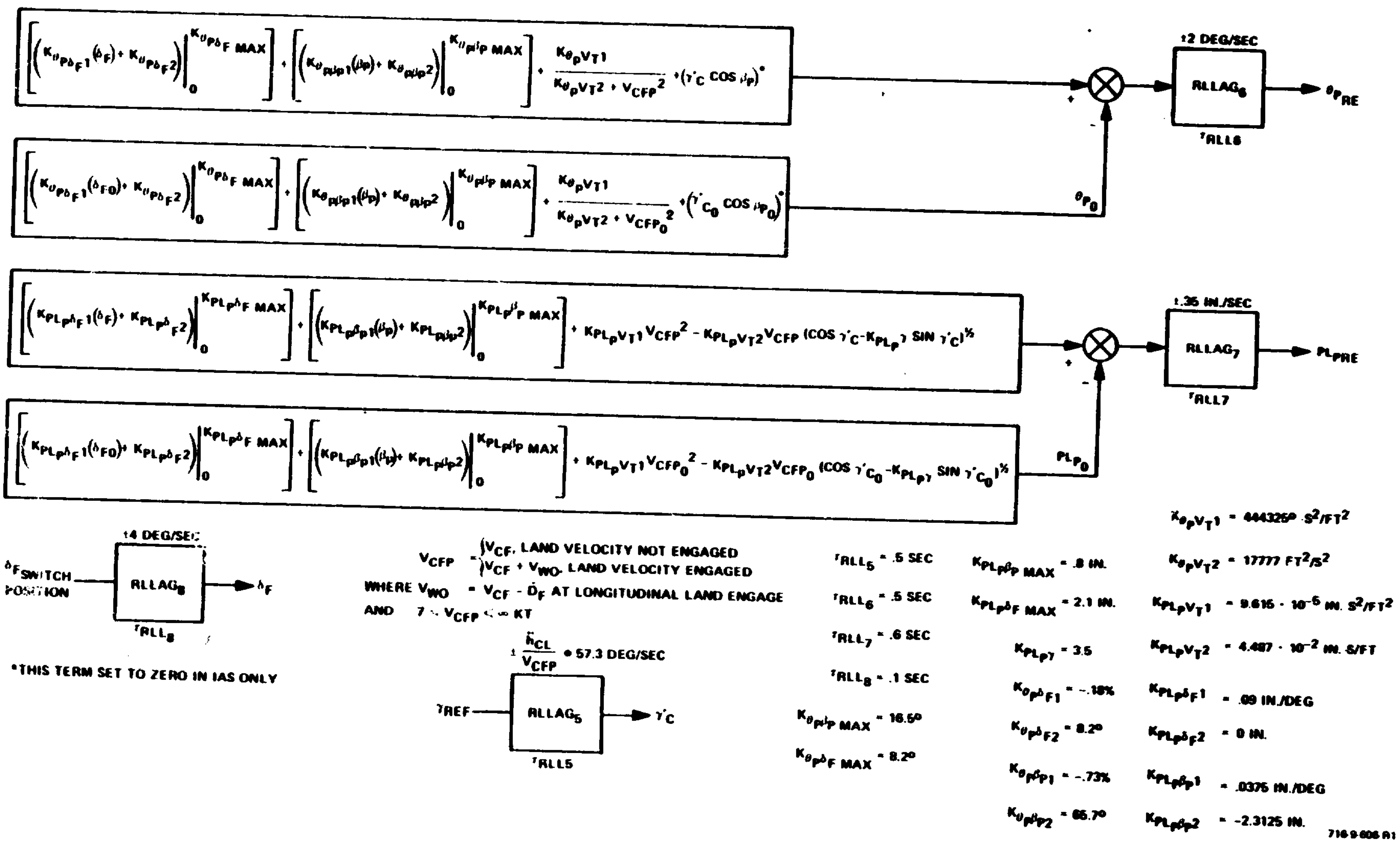


Figure 5-9
VL Control Predicts

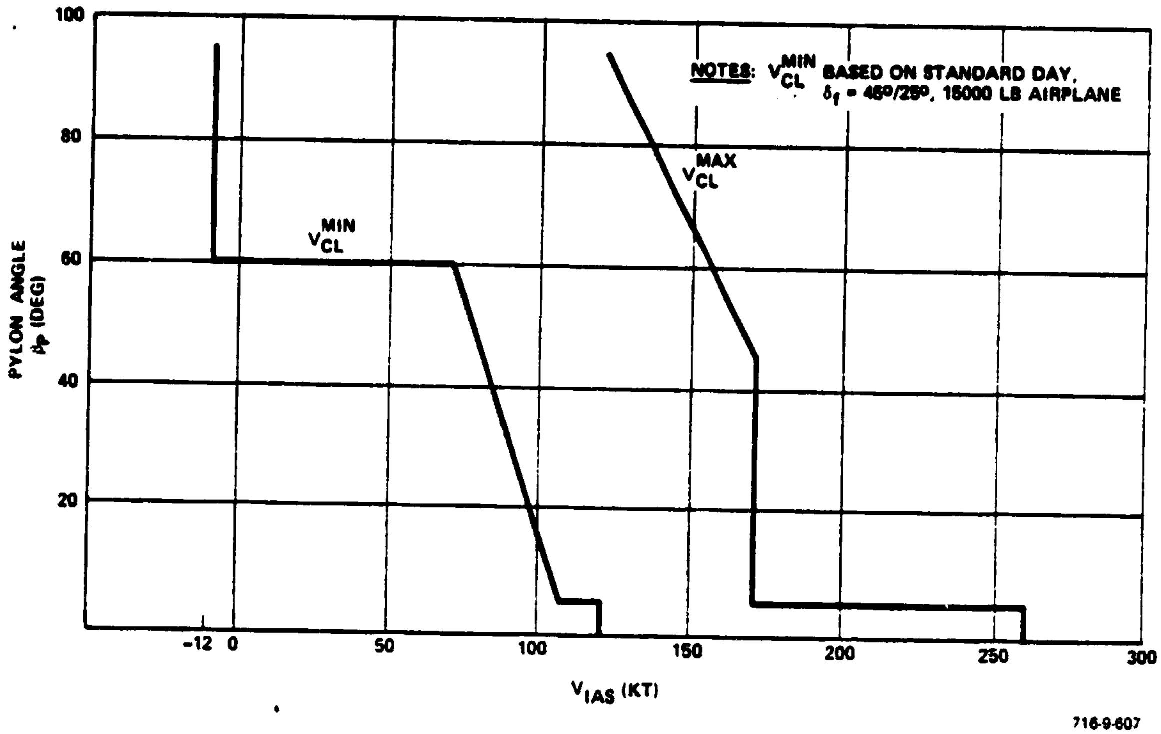


Figure 5-10
Airspeed Limits

region to one outside the region, the Autopilot and/or Flight Director will immediately disengage and a TOO FAST or TOO SLOW message will appear on the data entry display.

5.3.3.6 Configuration Control

When the Autopilot is OFF, the system does not control any configuration functions. When the Autopilot is ON, the total vehicle configuration is under autopilot control. In VL primary (pitch-attitude hold) the Autopilot holds the configuration existing at the time of VL primary engagement. In all other VL modes the configuration is controlled as described in the following paragraphs. All autopilot configuration changes are annunciated on the upper FMA for 5 seconds, or until the control position satisfies the command to within the control-in-motion tolerance specified in Table 5-12, whichever is longer.

In the manual flight-director mode (Autopilot OFF), the pilot changes the vehicle configuration in response to flashing configuration commands on the upper FMA. An annunciated command will go away only when the pilot satisfies the command to within the control-in-motion tolerance specified in Table 5-12.

In either the Autopilot or manual Flight Director modes, the tolerances are widened on the control-position error when the configuration controls are not being commanded to move. These control-not-in-motion tolerances are also listed in Table 5-12.

• RPM Control - The RPM command schedule is shown in Figure 5-11. The RPM value is commanded to change only during acceleration or deceleration (changing speed reference). When the RPM is commanded to a new value, an RPM TO XX% message appears on the upper FMA.

The three-knot hysteresis zones identified in Figure 5-11 have been incorporated for noise protection since the actual airspeed used for command generation may be noisy.

The pilot is allowed to override the RPM command in VL primary (pitch attitude hold). However, if he attempts to override the commanded RPM while any airspeed mode is in effect, the system will drive the RPM back to the commanded value.

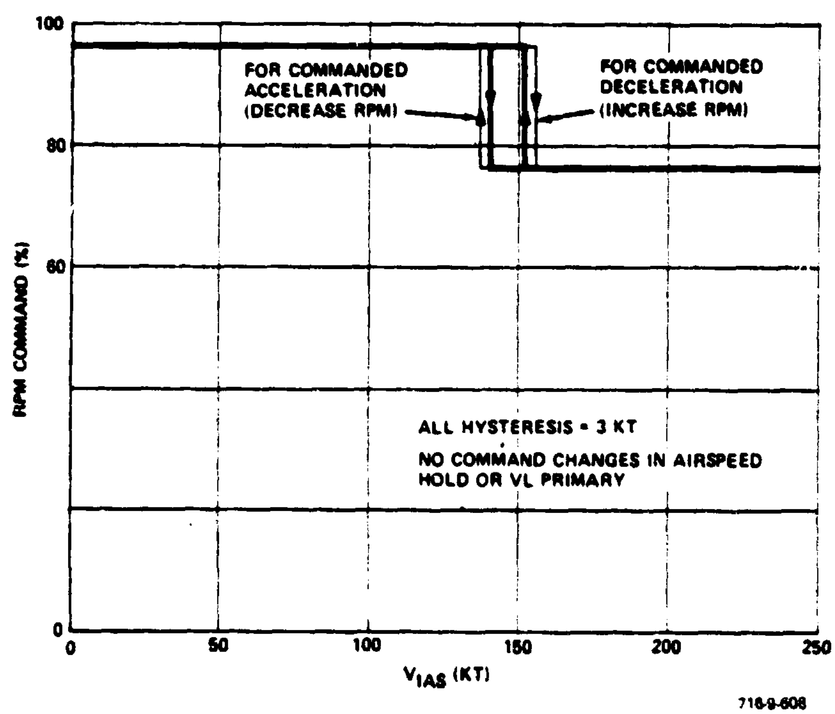


Figure 5-11
RPM Control

TABLE 5-12
CONFIGURATION CONTROL TOLERANCES

		RPM (percent)	Flaps	Pylons (degree)
Autopilot	Control-In-Motion	±.696	*	±.5
	Control-Not-In-Motion	±.696	*	±1
Manual Flight Director	Control-In-Motion	±2	*	±3
	Control-Not-In-Motion	±2	*	±5

*In flap configuration control, commands are satisfied when the flap-lever position corresponds to the commanded value.

• Flap Control - The flap command is shown in Figure 5-12. The flap position is commanded to change only during accelerations or decelerations (changing speed reference).

When the flaps are being commanded to a new position, a FLAPS TO XX° message appears on the upper FMA. The three-knot hysteresis zones identified in Figure 5-12 have been incorporated for noise protection since the actual air-speed used for command generation may be noisy.

If the pilot attempts to override the commanded flap position, the system will command the flaps back toward the commanded value and the Autopilot may disconnect, depending on the time the flap lever is not in the commanded position.

• Pylon Control - The pylon control schedule is shown in Figure 5-13. It has been designed to minimize pitch attitude excursions for speed change and to use the pylon angle for assistance during speed changes in the helicopter or transition configurations.

The command generated to the aircraft conversion system is a function of whether the Autopilot is commanding an acceleration, a deceleration, or airspeed hold. When airspeed hold is ON, pylon commands are derived from the curve labeled FOR AIRSPEED HOLD in Figure 5-13. In this case the reference

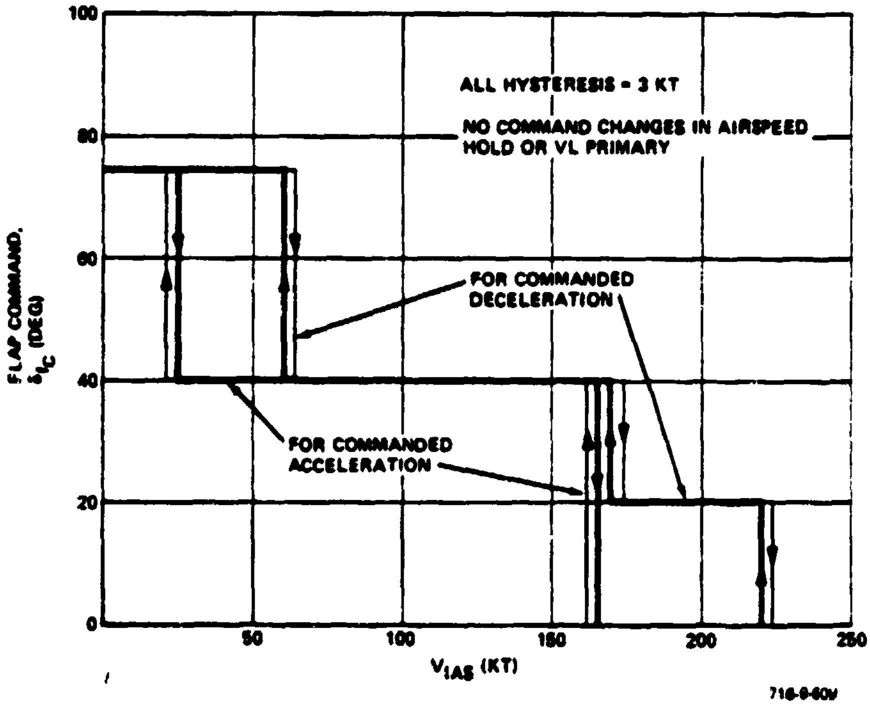


Figure 5-12
Flap-Configuration Control

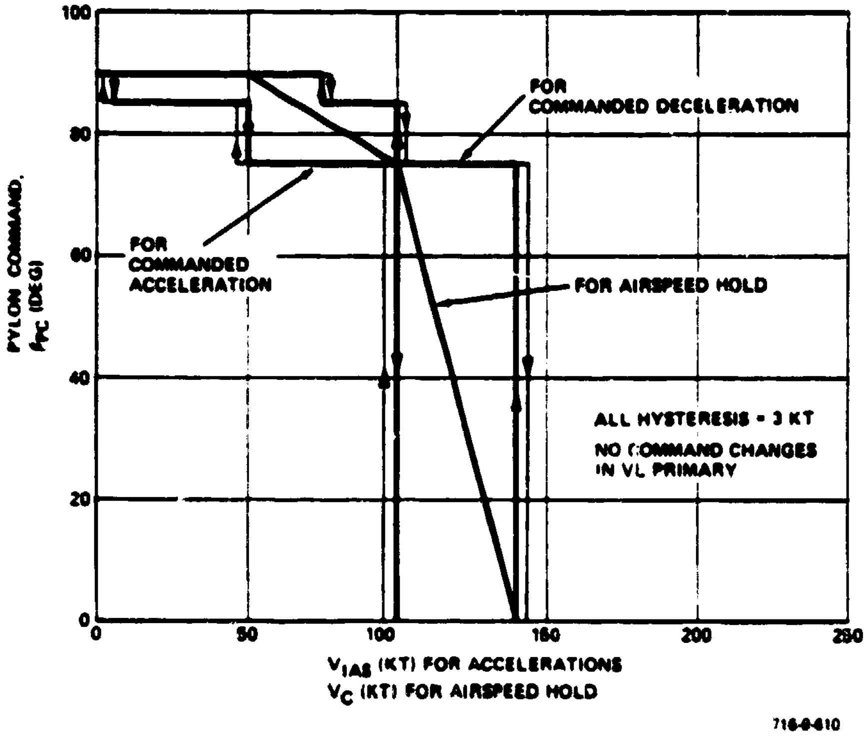


Figure 5-13
Pylon Configuration Control

airspeed V_C is used instead of the actual airspeed V_{IAS} to determine the pylon command. This eliminates the need for hysteresis in the airspeed-hold curve. For a commanded acceleration or deceleration, one of the other two curves in Figure 5-13 are used. The three-knot hysteresis shown protects against possible noise and turbulence on the airspeed signal V_{IAS} which is used to generate the pylon command.

To prevent the pylon command from "back-tracking" at transitions from airspeed-select to hold, the pylon command, during acceleration or deceleration, is limited to the airspeed-hold pylon command (determined by V_C) which will occur when the mode transition from select to hold occurs. During acceleration, no pylon angle will be commanded that is less than the airspeed-hold pylon angle at the final selected speed. Similarly, during deceleration no pylon angle will be commanded which is greater than the airspeed hold pylon angle at the final selected speed. For example, if an airspeed-select is initiated from 30 to 80 knots, V_C is 80 knots during and after the airspeed-select and the acceleration pylon command is limited to a value greater than the airspeed hold value for 80 knots, or about 81 degrees. Therefore, when 50 knots is reached during the acceleration maneuver, the pylons will be commanded to 81 degrees instead of the acceleration value of 75 degrees. Thus, when the airspeed-hold mode engages at 80 knots, no further pylon motion is required.

When the pylons are commanded to a new position, a PYLON TO XX° message appears on the upper FMA. If the pylons are locked when the Autopilot is attempting to command them to move, a flashing UNLOCK PYLON message appears on the upper FMA. Whenever the aircraft airspeed is above 190 knots and the pylons are unlocked, a flashing LOCK PYLON message appears on the upper FMA. If the pilot attempts to move the pylons to a position other than the commanded value, the Autopilot will disconnect.

5.3.4 Guidance Computations

The guidance computations generate commands to the control equations as required for each guidance mode described in Paragraph 5.3.2. The following paragraphs are grouped by vertical/longitudinal and lateral/directional guidance modes. Since some modes require little or no computations, such paragraphs are brief.

5.3.4.1 Vertical/Longitudinal Guidance

5.3.4.1.1 Primary VL Mode

In this fall-back mode the pitch-attitude command θ_C is set to a constant value equal to the pitch attitude θ coincident with the most recent of:

- Primary VL mode engagement if A/P is not engaged
- A/P engagement
- FLT DIR engagement if A/P is not engaged

Velocity is not controlled in this mode.

5.3.4.1.2 Airspeed Hold (IAS HOLD)

In IAS HLD the raw velocity command V_C is the value V_{DSP} displayed on the MSP IAS display when the IAS HLD becomes engaged. If V_{DSP} is subsequently changed via the IAS slew switch to differ from V_C, the displayed value will blink. The change in V_{DSP} does not affect V_C.

The filtered airspeed V_{FILT} is subtracted from the filtered velocity command V_{CF} by the basic VL control computations to obtain the velocity error V_E.

In IAS HLD, zero steady-state velocity error is obtained by integration of V_E in the VL control law.

5.3.4.1.3 Airspeed Select (IAS SEL)

In IAS SEL the raw airspeed command is given by

$$V_C = V_{DSP} = \text{displayed airspeed} \quad (\text{ft/s})$$

IAS SEL automatically transitions to IAS HLD when

$$|V_{DSP} - V_T| \text{ becomes } \leq 2 \quad \text{ft/s}$$

5.3.4.1.4 Flight Path Angle Hold (FPA HLD)

Vertical speed command is computed from flight path angle command and true airspeed by

$$\dot{h}_C = V_T \tan \gamma_C \quad \text{ft/s}$$

γ_{DSP} may be previously selected FPA (by slewing) or an actual FPA given by

$$\gamma_A = \tan^{-1} \left(\frac{\dot{h}}{V_T}\right) \quad \text{deg}$$

When FPA HLD engages in transition from FPA SEL, the commanded flight path angle is,

$$\gamma_C = \text{the } \gamma_{DSP} \text{ coincident with FPA HLD engagement}$$

where γ_{DSP} is the FPA value displayed on the MSP.

The FPA SEL mode will engage automatically if ALT SEL is armed and FPA SEL is not already engaged with a FPA reference which ensures convergence to the selected altitude. When FPA SEL engages automatically, the system selects an FPA reference to guarantee convergence to the selected altitude in 1 minute. This computed reference is displayed on the MSP.

$$\gamma_C = \gamma_{DSP} = 57.3 \tan^{-1} \left(\frac{h_{DSP} - h_{SEA}}{60\ V_T}\right) \quad \text{deg}$$

where

$$1 \leq |\gamma_C| \leq 8 \quad \text{deg}$$

If the FPA display is changed (via the slew switch) so that it differs from γ_C, the display will blink for 30 seconds and then revert to the current value of γ_C.

5.3.4.1.5 Flight Path Angle Select (FPA SEL)

Also in this mode,

$$\dot{h}_C = V_T \tan \gamma_C$$

where $\gamma_C = \gamma_{DSP}$ = the FPA value displayed on the MSP. FPA SEL automatically transitions to the FPA HLD if

$$\gamma_{DSP} - \gamma_A \text{ becomes } \leq .2 \text{ deg}$$

and the FPA display has not been slewed within the past 1 second, where γ_A is the actual flight-path angle.

5.3.4.1.6 Altitude Hold (ALT HLD)

In ALT HLD,

$$\dot{h}_C = K_h (h_C - h_{SEA}) + \frac{K_{hI}}{s} (h_C - h_{SEA}) \quad \text{ft/s}$$

where

$K_h = K_{ho} (1 + f_{H1})$ (ft/s)/ft, for Autopilot mode

f_{H1} = Sin β_p

β_p = Pylon angle (deg)

K_{ho} = .2 (ft/s)/ft

K_h = .1 (ft/s)/ft for manual Flight Director

k_{hI} = path integral gain = .005 s^{-2} for $|h_C - h_{SEA}| \leq 25$ ft and not manual FD

= 0 s^{-2} otherwise

h_{SEA} = altitude above sea level (ft)

If the ALT HLD/SEL button is pushed when ALT HLD/SEL is not previously engaged, ALT SEL becomes armed, and h_C is set by the G&C program to

$h_C = h_{DSP}$, coincident with pushing the ALT HLD button

where h_{DSP} is the altitude displayed on the MSP. While in the ALT SEL mode described below, the active reference may be changed by slewing the h_{DSP} to the desired reference via the ALT slew switch.

When ALT SEL automatically transitions to ALT HLD, h_C is set to

$$h_C = \text{the } h_{DSP}\text{, coincident with ALT HLD engagement}$$

If h_{DSP} is changed (via the slew switch) while ALT HLD is engaged so that it differs from h_C rounded to the nearest foot, the display will blink, indicating that it is a non-active future reference. If no action is taken within 30 seconds to re-arm the ALT SEL mode, the display will revert to the active reference (non-blinking).

5.3.4.1.7 Altitude Select (ALT SEL)

If the ALT HDG/SEL button is pushed when this mode is not previously engaged, ALT SEL becomes armed, the altitude display on the MSP blinks, and FPA SEL engages. A value for γ_{DSP} is first computed by the G&C program, which results in altitude capture in approximately 1 minute, but is subject to the following limits:

$$1° \leq |\gamma_{DSP}| \leq 8°$$

The computed γ_{DSP} may be altered manually, however, via the IAS slew switch while FPA SEL remains engaged. FPA SEL then transitions to FPA HLD, as previously described. FPA SEL may be re-engaged at any time, while ALT SEL is armed, to alter the flight-path angle for capturing the selected altitude.

ALT SEL engages automatically when

$$|h_{DSP} - h_{SEA}| \text{ becomes } < \max \left[5, \frac{\dot{h}^2}{2h_{CL}}\right] \text{ ft}$$

where

$$\ddot{h}_{CL} = \pm \text{ rate limit on } \dot{h}_C = 1.5 \text{ ft/s}^2$$

Then

$$\dot{h}_C = K_h F_{BLND} (h_C - h_{SEA}) \text{ ft/s}$$

where

h_C = the h_{DSP} coincident with ALT SEL engagement

$\dot{h}_C$ = raw altitude rate command (ft/s)

$$F_{BLND} = \frac{t}{\Delta t} \leq 1$$

t = time after last capture (s)

$$\Delta t = \left[2 \frac{V_T |\Delta\gamma|}{57.3\ \ddot{h}_{CL}} \right], \text{ total hold gain blend-in time (s)}$$

$\Delta\gamma = \gamma_c$ before capture (deg)

K_h as defined in Paragraph 5.3.4.1.6.

However, the filtered command, $\dot{h}_{CF}$, is at the acceleration limit, $\dot{h}_{CL}$, as long as

$$\frac{1}{.5} \left| L_{\pm \ddot{h}_{CL}} (\dot{h}_C) - \dot{h}_{CF} \right| \geq \dot{h}_{CL} = 1.5 \text{ ft/s}^2$$

ALT SEL automatically transitions to ALT HLD if both

$$|h_C - h_{SEA}| \leq 50 \text{ ft for 2 seconds}$$

and

$$|\dot{h}| \leq 5 \text{ ft/s for 2 seconds}$$

5.3.4.1.8 Reference Flight Path VL Guidance (VL RFP)

For the 3D guidance modes (RFP and LAND), the VL mode is armed only after the LD mode has engaged. The LD RFP mode, described below, therefore contains the computation and logic required for entering the reference flight path and for transition between the flight path segments. Only the LD RFP mode becomes armed when the RFP button is pushed (if not previously armed or engaged). When LD RFP engages, the button turns green and the VL RFP mode becomes armed. While LD RFP is engaged but FL RFP is not, FPA HLD and FPA SEL may be used to set up a vertical intercept with the RFP reference altitude, h_{RFP}. When the capture conditions for h_{RFP} are met, VL RFP will engage, thereby disengaging any other vertical mode. When VL RFP is armed, or if it is engaged with no other vertical mode armed, the vertical deviations are displayed on the ADI and the HSI.

The VL RFP capture and track laws are essentially the same as for altitude capture and hold except that the reference altitude can have a slope and is computed as defined below. In the reference flight-path entrance mode, the reference altitude is defined by a straight backward extension from the entrance waypoint. The computation for segment n (connecting waypoints n-1 and n), or for the backward extension of segment n, is given by

$$h_{RFP} = h_n - D_{xn} \tan \gamma_n \quad \text{ft}$$

where

h_{RFP} = reference altitude wrt the runway for the reference flight path (or a backward extension) (ft)

h_n = altitude of waypoint n wrt the runway (ft)

D_{xn} = horizontal along-track distance to end of segment n (ft)

$$= \begin{cases} (x_n - x)^2 + (y_n - y)^2 & \text{for straight segment } n \\ (\lambda_n - \lambda_{CRn}) \left(\frac{\pi}{180}\right) (R_n) & \text{for circular segment } n \end{cases}$$

(x, y) = aircraft horizontal location in the runway coordinates (ft)

(x_n, y_n) = location of waypoint n in the runway coordinates (ft)

R_n = radius of curvature of circular segment n, positive for right turn (stored data), (ft)

λ_n = Course angle at the end of circular segment n relative to the X axis (stored data; may be different from $\lambda_n + 1$, the course angle at the beginning of segment n + 1).

(x_{Rn}, y_{Rn}) = location of the center of curvature of circular segment n in the runway coordinates (stored data)

λ_{CRn} = course angle on circular segment n at the point nearest to the aircraft, relative to the X axis

$$= \begin{cases} \tan^{-1} \left(\dfrac{x - x_{Rn}}{y_{Rn} - y}\right) & \text{for right turn} \\ \tan^{-1} \left(\dfrac{x_{Rn} - x}{y - y_{Rn}}\right) & \text{for left turn} \end{cases}$$

$\tan \gamma_n$ = slope of segment n

The above nonvariable reference flight path parameters [(x_n, y_n, h_n), (x_{Rn}, y_{Rn}), R_n, λ_n] are precomputed and stored as reference flight-path data.

If LD RFP is engaged, VL RFP will engage automatically if

$$|h_{RFP} - h| < \max \left[\frac{(\dot{h} - V_G \tan \gamma_n)^2}{2\,\ddot{h}_{CL}}, 5 \right]$$

where

V_G = ground speed (ft/s)

$\ddot{h}_{CL}$ = acceleration limit = 1.5 ft/s^2

Then, after the vertical capture,

$$\dot{h}_C = (F_{BLND})\, K_h\, (h_{RFP} - h) + V_G \tan \gamma_n + \frac{K_{hI}}{s} (h_{RFP} - h) \quad \text{ft/s}$$

where

$K_h = K_{ho}\,(1 + f_{H1})$ (ft/s)/ft, for Autopilot modes

$f_{H1} = \sin \beta_p$

β_p = Pylon angle (deg)

K_{ho} = .2 (ft/s)/ft

K_h = .1 (ft/s)/ft, for manual Flight Director

K_{hI} = path integral gain = .005 s^{-2} for $|h_{RFP} - h| \leq 25$ ft and not manual FD

= 0 s^{-2}, otherwise

h = aircraft altitude wrt the runway (ft)

$$F_{BLND} = \frac{t}{\Delta t} \leq 1$$

where

t = time after last capture

$$\Delta t = 2\frac{(V_T \tan|\Delta\gamma|)}{\ddot{h}_{CL}} \quad s$$

where

$\Delta\gamma = (\gamma_{n+1} - \gamma_c)$ computed before capture (deg)

However, the filtered command, $\dot{h}_{CF}$, is at the acceleration limit, $\ddot{h}_{CL}$, as long as

$$\frac{1}{.5}\left|L_{\pm \ddot{h}_{CL}}(\dot{h}_C) - \dot{h}_{CF}\right| \geq \ddot{h}_{CL} = 1.5 \quad ft/s^2$$

See Table 5-6 for relevant definitions.

5.3.4.1.9 Automatic Landing VL Guidance (VL LND)

If the LAND button is pushed when not previously armed or engaged, the LD LND mode becomes armed. The VL mode is armed when the LD mode engages, and remains armed until the vertical capture conditions for VL LND are met. After LD LND has engaged but VL LND has not, FPA HLD and FPA SEL, may be engaged for the purpose of setting up an intercept with the vertical LND reference. When VL LND is armed or engaged, the vertical deviations are displayed on the ADI and HSI if MLS elevation is valid.

When LD LND is engaged, VL LND will engage automatically if MLS elevation is valid and

$$|h_E| < \max\left[5, \frac{(\dot{h} - V_G \tan \gamma_{REF})^2}{2\ddot{h}_{CL}}\right] \quad ft$$

where

$h_E = h_{LND} - h =$ altitude error (ft)

$h_{LND} =$ altitude reference wrt the runway in the LAND mode, defined as the altitude on the LND segment to be captured (ft)

$h =$ altitude wrt the runway (ft)

$V_G =$ ground speed (ft/s)

$\gamma_{REF} =$ reference glide slope for the selected LND submode (deg)

$\ddot{h}_{CL} =$ acceleration limit $= 1.5$ ft/s^2

The altitude rate command in the capture mode is given by

$$\dot{h}_C = (F_{BLND})\, K_h\, h_E + V_G \tan \gamma_{REF} + \frac{K_{hI}}{s}\, h_e \quad ft/s$$

where F_{BLND}, K_h and k_{hI} are the same as defined in Paragraph 5.3.4.1.8.

For both LND modes, a final vertical guidance mode is defined in Paragraph 5.3.4.1.9.3. This final vertical guidance is the same for the two LND modes, and is therefore not repeated in the following corresponding LND submode paragraphs.

- <u>LAND-1 Approach (Straight-in Approach)</u>

Until the final glide slope is captured,

$$h_{LND} = (X - X_T) \tan \gamma_{REF} \quad \text{ft}$$

where

X = aircraft position along runway X-axis (ft)

X_T = X coordinate of touchdown point (ft)

γ_{REF} = reference glide slope (deg)

γ_{REF} is selected via the keyboard, and is limited to

$$-8° \leq \gamma_{REF} \leq -2°$$

- <u>LAND-2 Approach</u>

Figure 5-14 illustrates the LAND-2 approach trajectory. It consists of four segments, excluding the final vertical guidance described below.

(a) A circular, level segment of radius 1189 feet, 2500 feet above the runway and tangent to the reference flight path.

(b) A straight, level segment directed along the constant 18-degree azimuth line.

(c) A descending, straight segment at a -6.11 degree slope directed along the 18-degree azimuth line.

(d) A counterclockwise helix segment of two revolutions tangent to the 18-degree azimuth line. This segment has a 6.11 degree downward slope. The location of the helix segment is preselected prior to LAND-2 engagement by specifying D_H, the horizontal distance along the 18-degree azimuth line from the touchdown point to the helix center. X_1 must be between 1500 feet and 8307 feet, and is selectable at the keyboard via the HLX mnemonic.

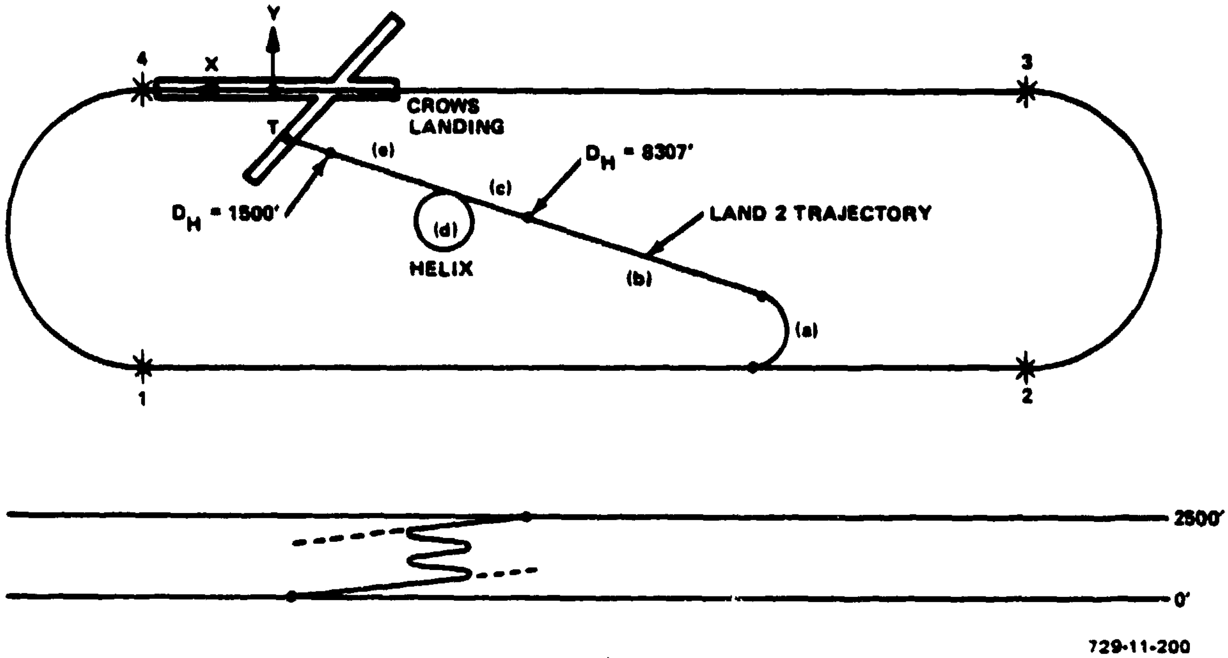


Figure 5-14
Reference Flight-Path and LAND-2 Trajectories

(e) A final straight segment along the 18-degree azimuth line with a -6.11 degree slope. This segment terminates directly above the touchdown point at an altitude of h_{FIN} where hover before the final letdown occurs.

When LAND-2 is engaged, the reference altitude h_{LND} is initially defined for segment (a), and converts to segment (b) when the capture conditions for segment (b) are met. Similarly, h_{LND} then remains defined for segment (b) until the capture conditions for segment (c) are met. Hence, the reference altitude for LND2 is defined by

$$h_{LND} = \begin{cases} 2500 \text{ ft, for segments (a) and (b)} \\ 2500 + \tan(-6.11)\,[(X_T - X)^2 + (Y_T - Y)^2]^{1/2} \text{ ft, for segment (c)} \\ 2500 - (8307 - X_1)(\sin 6.11) - |R_{HX}| \tan(-6.11)\left(\frac{\pi}{180}\right)(18^\circ - \psi_{HX}) \\ \text{ft, for segment (d)} \\ \tan(-6.11)\,[(X_T - X)^2 + (Y_T - Y)^2]^{1/2} \text{ ft, for segment (e)} \end{cases}$$

where

R_{HX} = helix radius = -1189 ft

$\gamma_{HX} = \gamma_{REF}$ = glide slope angle = -6.11 deg

ψ_{HX} = helix course angle, ranges from 720 to 0 degrees for two revolutions

X = aircraft position along the runway X-axis (ft)

Y = aircraft position perpendicular to X-axis (ft)

(X_T, Y_T) = coordinates of touchdown point

The helix course angle may be computed as:

$$\psi_{HX} = \tan^{-1}\left(\frac{X_c - X}{Y - Y_c}\right) + \psi_{HXR} \text{ degrees}$$

where

(X, Y) = aircraft horizontal position in the runway coordinates (ft)

ψ_{HXR} = initially 720, decreases by 360 degrees each time $X_H - X$ changes from positive to negative

X_H = X coordinate of helix entrance point (ft)

The capture conditions for the segments are the same as defined above for VL LND capture, where

$$\gamma_{REF} = \begin{cases} 0 \text{ for segments (a) and (b)} \\ -6.11 \text{ for segments (c), (d) and (e)} \end{cases}$$

• <u>Final Vertical Guidance</u> - Figure 5-15 illustrates the geometry for the final vertical guidance in the LAND-1 (straight-in) and LAND-2 (HELIX) modes.

The final vertical guidance consists of three straight segments:

(a) A glide slope segment of γ_{REF} value over the approach axis D_F through $D_F = 0$ and $h = h_{FIN} + h_{TD}$

(b) A horizontal segment over the approach axis at an altitude of $h_{TD} + h_{FIN}$

(c) A vertical "letdown" segment at $D_F = 0$.

The reference altitude for segments (a) and (b) is hence defined by

$$h_{LND} = \begin{cases} h_{TD} - \tan\gamma_R \left[(X_T - X)^2 + (Y_T - Y)^2\right]^{1/2}, \text{ for segment (a)} \\ h_{TD} + h_{FIN}, \text{ for segment (b)} \end{cases}$$

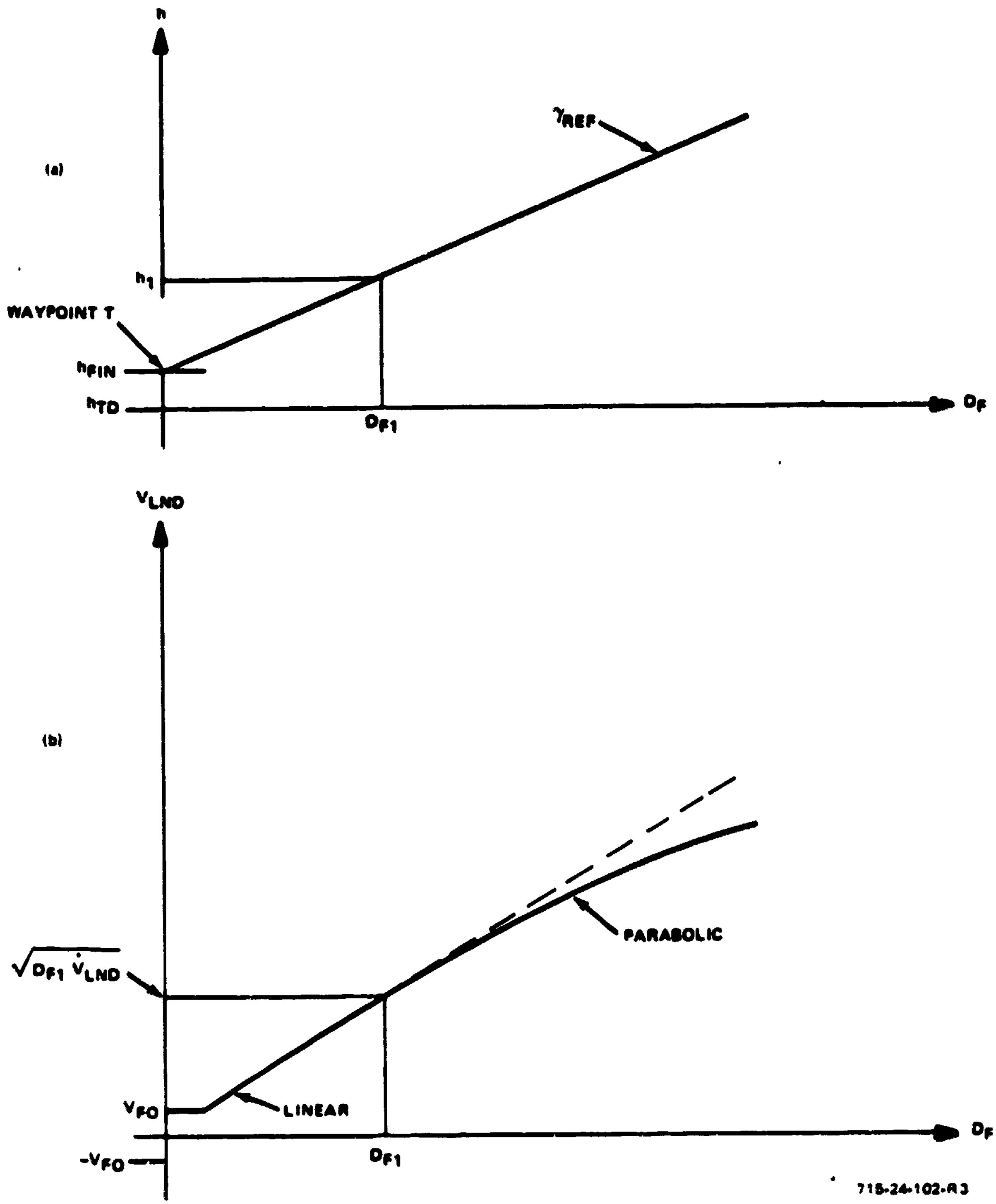


Figure 5-15
Final Vertical and Longitudinal Guidance Geometry

where

$$h_{TD} = 0 \quad (\text{ft})$$

$$X_T = \begin{cases} -1124 \text{ ft for LAND-1} \\ -686 \text{ ft for LAND-2} \end{cases}$$

$$Y_T = \begin{cases} 0 \text{ ft for LAND-1} \\ -1664 \text{ ft for LAND-2} \end{cases}$$

h_{FIN} = final hover altitude before letdown = 10 ft

The capture conditions for segments (a) and (b) are the same as for VL LND capture defined above.

where

$$\gamma_{REF} = \begin{cases} -6.11 \text{ deg for segment (a)} \\ 0 \text{ deg for segment (b)} \end{cases}$$

The letdown mode is initialized if both

$$|\dot{D}_F| \leq .5 \quad \text{ft/s}$$

and

$$|D_F| \leq 24 \quad \text{ft}$$

where

D_F = aircraft position along the approach axis (ft)

$\dot{D}_F$ = aircraft ground velocity along the approach axis (ft/s)

$$= \begin{cases} +\dot{X} \text{ ft/s for LAND-1} \\ +X \text{ (cos 18°) ft/s for LAND-2} \end{cases}$$

When the letdown mode is in effect, the vertical guidance is the following altitude rate command:

$$\dot{h}_C = -\max\left(.2, \frac{h - h_{TD}}{4} - .05\right) \quad \text{ft/s}$$

The letdown mode will remain in effect until the LND mode is disengaged by the pilot, or automatically when the weight-on-wheels discrete is received.

• LND Velocity Control - When the velocity submode of VL LND is engaged, IAS HLD or IAS SEL disengages and velocity is controlled by the following velocity profile law. If VL LND is engaged, LND Velocity engages automatically when

$$V_{LND} \text{ becomes } \leq V_G \quad \text{ft/s}$$

where V_{LND} is the computed deceleration velocity reference. If, at the time LND Velocity is armed (at VL LND engagement), the land velocity reference (V_{LND}) is more than 10 knots below the actual ground speed (V_G), then a "late" flare condition exists and the Autopilot or FD will disconnect with a TOO FAST message appearing on the DED.

When the VL LND mode is engaged,

$$V_C = V_{LND}$$

$$\text{and } \dot{V}_{CL} = 1.5 \quad \text{ft/s}^2$$

$$\text{where } \dot{V}_{CL} = \text{speed command rate limit}$$

V_{LND} is illustrated in Figure 5-15(b) and is given by

$$V_{LND} = \begin{cases} \sqrt{(2D_F - D_{F1})\,\dot{V}_{LND}}, & \text{if } D_F > D_{F1} \\ \max\left(V_{Fo}, D_F\sqrt{\dot{V}_{LND}/D_{F1}}\right), & \text{if } 0 < D_F \leq D_{F1} \\ -V_{Fo}, & \text{if } D_F \leq 0 \end{cases}$$

where

D_F = horizontal along-track distance to go (defined below) (ft)

D_{F1} = inflection point where profile changes from linear versus time to linear versus D_F; = 250 ft

$\dot{V}_{LND}$ = deceleration magnitude when $D_F > D_{F1}$; = 1.5 ft/s^2

V_{F1} = velocity at D_{F1} = 19.36 ft/s

V_{F0} = final constant velocity before letdown = .25 ft/s

For a straight-in approach (LAND-1),

$$D_F = -X \text{ ft}$$

For the 18-degree helix approach (LAND-2),

$$D_F = (4450 - X)(\cos 18°) - [(X_T - 4450)^2 + Y_T^2]^{1/2}$$

where (X_T, Y_T) are the coordinates of the touchdown point on Runway 30 at Crow's Landing.

As shown in the time-domain profile in Figure 5-16(b), V_{LND} is exponential for $D_F < D_{F1}$ until $V_{LND} = V_{F0}$. The exponential segment is incorporated to prevent excessive pitchup which would occur if constant deceleration were continued to the end. The exponential segment terminates with a low constant velocity, V_{F0}; otherwise, $D_F = 0$ cannot be attained in finite time.

5.3.4.1.10 Go-Around VL Guidance

When the Go-Around mode is engaged from the research pilot's control stick, an automatic altitude-select to 2500 feet ASL is initiated and airspeed hold at the current airspeed is engaged. After a 5-second delay, an airspeed select to 60 knots is initiated if the airspeed is below 60 knots; otherwise, airspeed hold will remain engaged.

The above described operation is achieved with logic defined in the GAPSH subroutine and the subroutines called by GAPSH. This logic generates the 5-second timer required and changes the basic VL mode directors and guidance reference to provide the desired performance.

5.3.4.1.11 Vertical Deviation Indicators

The vertical deviation from the reference altitude is displayed on the ADI and the HSI whenever the vertical RFP or LAND mode is armed or engaged and a valid vertical deviation is available. Otherwise, the indicator needle is stored out of view. If one of these modes is armed while another is engaged, the deviation for the armed mode is displayed. The deviation is proportional to vertical footage error, h_E, with scaling as follows for both indicators:

Mode	1 dot =
RFP	250 feet
LAND	50 feet

Hence,

$$VRT = \begin{cases} L\pm 2\ [\frac{1}{250}\ h_E]\ \text{dots for RFP} \\ L\pm 2\ [\frac{1}{50}\ h_E]\ \text{dots for LAND} \\ \text{out of view otherwise} \end{cases}$$

where

δ_{VRT} = ADI and HSI vertical deviation indicator value (dots)

5.3.4.2 Lateral-Directional Guidance

The lateral/directional guidance commands go to the roll axis controls and the roll flight director. Directional control is provided by the LD control laws when A/P is engaged. The pedal commands provide turn coordination in cruise conditions. In hover conditions they provide heading control.

5.3.4.2.1 Primary LD Mode

If the roll attitude is below ±5 degrees when the primary LD mode engages, this mode immediately transitions to HDG HLD, except when the manual Flight Director is on and the airspeed is below 30 knots. While primary LD is engaged, a constant roll attitude-hold mode is in effect, defined by

$$\phi_C = \text{the } \phi \text{ coincident with engagement of the primary LD mode} \quad \text{(deg)}$$

The primary LD mode engages under the conditions described below:

- Automatic reversion
- A/P engagement
- FLT DIR engagement if A/P not engaged

5.3.4.2.2 Heading Hold (HDG HLD)

In the HDG HLD mode,

$$\psi_E = \psi_C - \psi \quad \text{deg}$$

where

ψ = aircraft heading (deg)

When HDG HLD engages in transition from HDG SEL, then

$$\psi_C = \text{the } \psi_{DSP} \text{ coincident HDG HLD engagement} \quad \text{(deg)}$$

where

ψ_{DSP} is the heading displayed on the MSP.

If ψ_{DSP} is subsequently changed via the HDG slew switch so that it differs from ψ_C, the displayed value will blink for 30 seconds, or until the pilot selects the new heading. After 30 seconds, the display reverts to the current reference value of heading.

5.3.4.2.3 <u>Heading Select (HDG SEL)</u>

In heading select $\psi_E = \psi_{DSP} - \psi$ degree. HDG SEL reverts to HDG HOLD when

$$|\psi_E| \leq 5 \quad \text{deg}$$

5.3.4.2.4 <u>Lateral Circular Course Capture</u>

All lateral captures of paths defined by straight and circular segments are of the following types:

- Straight segment to straight segment
- Straight segment to circular segment
- Circular segment to straight segment
- Circular segment to circular segment

The circular capture law described in the following paragraph is a general law that computes the quantities required to define a circular path and any deviations from the path for the four types of path transitions mentioned above. The law defines the circular capture path to be tangent to the instantaneous aircraft course line and tangent to the path (circular or straight) to be captured. At the instant of capture, the circular capture path is defined and fixed in space relative to a conveniently chosen (DX, DY) coordinate system. The deviations from this circular capture path are then computed in the chosen (DX, DY) coordinate system. (See Figure 5-16 for a pictorial description of the circular capture law.)

5.3.4.2.4.1 <u>Armed Circular Course Capture</u>

When an LD path is armed for capture, the following computations are performed as a function of the data available for the capture computations. The radial paths for VOR and TACAN will use basic (prefiltered) range/bearing data to allow them to be independent of X, Y navigation. WPT radial captures as well as all RFP and LAND captures will use X, Y navigation data since this data is also required to track these paths.

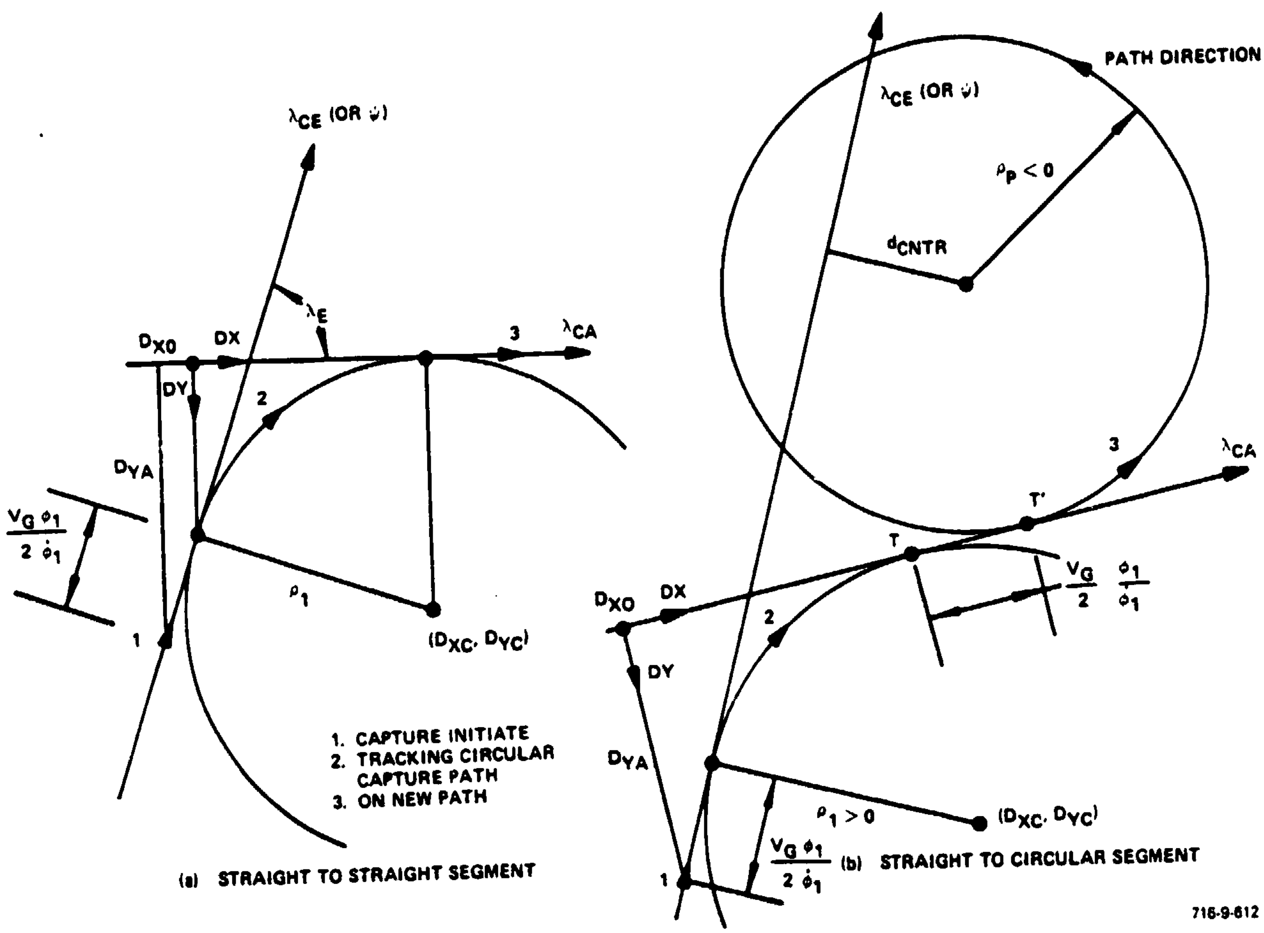


Figure 5-16
Circular Capture Geometry

(a) Preliminary computations

- Angular error to course arm reference:

$$\lambda_E = \lambda_{CA} - \lambda_{CE} \quad \text{deg}$$

where

λ_{CA} = course arm reference derived from the MSP CRS display for radial modes; computed for RFP and LAND modes.

λ_{CE} = current commanded course reference; reverts to actual heading if no CRS mode is engaged.

- Capture Radius (used only for straight segment or outside circular segment captures):

$$\rho_1 = \begin{cases} \rho \text{ sign } \lambda_E \text{ ft, if segment is circular} \\ \max \left[\dfrac{57.3\ V}{\dot{\psi}_o}, \dfrac{V_G^2}{g \tan \phi_1} \right] \text{ sign } \lambda_E \quad \text{feet, otherwise} \end{cases}$$

where

ρ = current circular path radius (ft)

V_G = ground speed (ft/s)

$\dot{\psi}_o$ = 3 deg/s

ϕ_1 = nominal bank angle during capture

- Capture circle center coordinates in (DX, DY) coordinate system:

$$D_{XC} = \rho_1 \sin \lambda_E + \frac{V_G}{2} \frac{\phi_1}{\dot{\phi}_1} \cos \lambda_E \quad \text{feet}$$

$$D_{YC} = \rho_1 \quad \text{feet}$$

where

$\dot{\phi}_1$ = nominal roll rate during roll-out for capture (deg)

- DX coordinate at capture initiation (D_{XO}):
  - If VOR/TAC radial armed:

$$D_{XO} = (R_{STN})_o \cos(\lambda_A')_o \quad \text{feet}$$

where

$(R_{STN})_o$ = Range to VOR or TACAN station at capture (ft)

$(\lambda_A')_o$ = Arm bearing error at capture from VOR or TAC bearing reference (deg)

  - If LD WPT, RFP or LAND armed:

$$D_{XO} = \cos(\lambda_{CA} - \psi_{RNY}) (X_A(_o + \sin(\lambda_{CA} - \psi_{RNY}) (Y_A)_o \quad \text{feet}$$

where

$(X_A)_o$, $(Y_A)_o$ = aircraft coordinates at capture in runway axis system, feet

and

λ_{CA} = MSP CRS display value or λ_n for straight segments and is computed as follows for non-tangent circular segments:

$$\lambda_{CA} = \lambda_{CE} + (\text{sign } \rho_1)\tan^{-1} \left\{ \frac{\left[\rho_p{}^2 - d_{CNTR}{}^2 - 2\rho_1 \rho_p + d_{CNTR}\right]^{1/2}}{|d_{CNTR} + \rho_1|} \right\} + \alpha_x \quad \text{deg}$$

where

$$\alpha_x = \frac{V_G}{2} \left(\frac{\phi p}{\dot{\phi}_1} \right) \frac{57.3}{|\rho_1| + |\rho_p|} \quad \text{deg}$$

$$d_{CNTR} = \cos(\lambda_{CE} - \psi_{RNY}) (\Delta Y) - \sin(\lambda_{CE} - \psi_{RNY}) (\Delta X) \quad \text{ft}$$

ρ_p = radius of path to be captured (ft)

α_X = adjustment angle to tangent direction to allow for finite roll-out time (deg)

ϕ_p = computed nominal bank angle for circular path to be captured (deg)

$$= \tan^{-1} \frac{V_G^2}{\rho_p\, g}$$

d_{CNTR} = distance from course error engage reference line to center of path to be captured (positive for center to left of line)

ρ_1 = capture radius as defined above <u>except</u> for inside circular captures where it is redefined as

$$\rho_1 = -d_{CNTR} \quad \text{feet}$$

$(\Delta X, \Delta Y)$ = along runway axes distances to center of circle (ft).

For RFP or LAND paths,

$$\Delta X = X_C - (X_{REF})_E \quad \text{ft}$$

$$\Delta Y = Y_C - (Y_{REF})_E \quad \text{ft}$$

otherwise,

$$\Delta X = X_C - X_A \quad \text{ft}$$

$$\Delta Y = Y_C - Y_A \quad \text{ft}$$

(X_C, Y_C) = Coordinates of center of circular path to be captured in runway axis system (ft)

(X_A, Y_A) = aircraft coordinates in runway axis system (ft)

$(X_{REF})_E$, $(Y_{REF})_E$ = coordinates of point currently flying to on RFP or LAND path (ft)

ψ_{RNY} = runway heading wrt magnetic north

All other symbols have been previously defined.

For circular RFP or LAND paths the coordinates of the tangent point T are computed as:

$$X_T = X_C - |\rho_p| \sin(\lambda_{CA} - \psi_{RNY}) \quad \text{ft}$$

$$Y_T = Y_C - |\rho_p| \cos(\lambda_{CA} - \psi_{RNY}) \quad \text{ft}$$

X_T and Y_T are then used to compute the crosstrack deviation from the tangent to the circular path to be captured:

$$D_{YA} = -\sin(\lambda_{CA} - \psi_{RNY})\ (X_A - X_T) + \cos(\lambda_{CA} - \psi_{RNY})\ (Y_A - Y_T) \quad \text{ft}$$

This quantity is used to determine the proper capture point.

(b) <u>Capture Criteria</u> - The circular capture law will transition from the armed state to the engaged state when the following conditions are satisfied:

- A/P or Flight Director is engaged and a course mode is armed or engaged.
- The 3-second inhibit course engage timer has timed out (after slewing).
- The capture angle to the new path is $\leq 90°$.
- If the path to be captured is a straight segment or a nontangent circular path, then capture will occur when:

$$K_{LC}\ [\rho_1\ (1 - \cos\ {}_E)] \leq D_{YA} \leq \rho_1\ (1 - \cos \lambda_E) + (\sin \lambda_E)\ \left(\frac{V_G}{2}\right)\ \left(\frac{\phi_1}{\dot{\phi}_1}\right) \quad \text{ft}$$

for right turning captures, or

$$\rho_1(1 - \cos \lambda_E) + \sin \lambda_E(\frac{V_G}{2}) (\frac{\phi_1}{\dot{\phi}_1}) \leq D_{YA} \leq K_{LC}\ \rho_1(1 - \cos \lambda_E) \quad \text{ft}$$

for left turning captures, where K_{LC} = late capture tolerance factor, set to .6

<u>OR</u> when the following close and shallow capture conditions are met:

$$|D_{YA}| < 500 \quad \text{ft}$$

and

$$|\dot{D}_{YA}| < 10 \quad \text{ft/sec}$$

- If the path to be captured is a tangent circle, as determined by the following conditions:

$$\cos(\lambda_{CE} - \lambda_{CNTR}) \geq 0$$

and

$$|\rho_p| - T_{TOL} < |d_{CNTR}| \leq |\rho_p| + T_{TOL}$$

where

$\lambda_{CNTR} = \text{Tan}^{-1}\left(\frac{Y_C - Y_A}{X_C - X_A}\right)$ degrees is the direction of the line from aircraft to the center of the circle, and

T_{TOL} = tangency tolerance, set to 10 feet. All other quantities previously defined,

$\underline{\text{then}}$ capture of the tangent circle will occur when

$$|\lambda_{CE} - \lambda_{CNTR}| \geq 90 - \alpha_C \quad \text{deg}$$

where

$$\alpha_C = \tan^{-1} \frac{\phi_p V_G \left(\frac{\phi_1}{\dot{\phi}_1}\right)}{2\phi_1 |\rho_p|} \quad \text{deg}$$

α_C is the predicted angle at which capture must occur to ensure proper roll-out timing. All other symbols have been previously defined.

At capture, an LD capture flag (LDCAPF) is set to designate the capture phase.

5.3.4.2.4.2 Engaged Circular Course Capture - During circular capture of a path, the crosstrack deviation and deviation rate to the circular capture path are computed as follows:

- Compute aircraft position along the DX axis:

For VOR and TACAN radial captures:

$DX = \sigma_{TF} [D_{XO} - (R_{STN}) \cos(\lambda'_E)]$ ft, DX is positive in the direction of the radial

where

D_{XO} is as defined in capture arm

R_{STN} = range to VOR or TACAN station (ft)

λ'_E = engage bearing error from VOR or TACAN bearing reference (deg)

$$\sigma_{TF} = \begin{cases} +1 \text{ when flying to the station} \\ -1 \text{ when flying from the station} \end{cases}$$

For WPT, RFP and LAND segments:

$D_X = [\cos(\lambda_{CE} - \psi_{RNY})\ (X_A) + \sin(\lambda_{CE} - \psi_{RNY})\ (Y_A)] - D_{XO}$
feet, D_X is positive in the direction of λ_{CE}

where

X_A, Y_A = aircraft position in runway axis system (ft)

λ_{CE} = engaged course reference (deg)

- Compute aircraft velocity $\dot{D}_X$ along the DX axis

For VOR and TACAN radial captures:

$\dot{D}_X = [-D_{YE}\ (\dot{\lambda}'_E) - \sigma_{TF} \cos(\lambda'_E)\ (\dot{R}_{STN})]$ ft/s

where

D_{YE} = engage crosstrack error (output of complementary filter) (ft)

λ'_E = engage bearing error (deg)

$\dot{\lambda}'_E$ = engage bearing error rate (deg/s)

$\dot{R}_{STN}$ = range rate to VOR or TACAN station

$$\sigma_{TF} = \begin{cases} +1 & \text{when flying to the station} \\ -1 & \text{when flying from the station} \end{cases}$$

For WPT, RFP and LAND segments:

$\dot{D}_X = \cos(\lambda_{CE} - \psi_{RNY})\ (\dot{X}_A) + \sin(\lambda_{CE} - \psi_{RNY})\ (\dot{Y}_A)$ ft/s

where

λ_{CE} = engage course reference

$\dot{X}_A, \dot{Y}_A$ = aircraft velocities in runway axis system

• Compute circular capture crosstrack deviation and deviation rate:

$$D_{CNTR} = [(D_{XC} - D_X)^2 + (D_{YC} - D_Y)^2]^{1/2} \text{ (sign } \rho) \quad \text{ft}$$

$$d = \rho - D_{CNTR} \quad \text{ft}$$

$$\dot{d} = \frac{(D_{XC} - D_X)\,\dot{D}_X + (D_{YC} - D_Y)\,\dot{D}_Y}{D_{CNTR}} \quad \text{ft/s}$$

where

d = crosstrack deviation to circle (ft)

$\dot{d}$ = crosstrack deviation rate to circle (ft/s)

D_{CNTR} = signed aircraft distance to capture circle center

(D_{XC}, D_{YC}) = capture circle center coordinate in (DX, DY) system

$$D_Y = \begin{cases} D_{YE}, \text{ for VOR/TAC or WPT} \\ \sin(\lambda_{CE} - \psi_{RNY})\,(X_A - X_T') + \cos(\lambda_{CE} - \psi_{RNY}) \\ (Y_A = Y_T'), \text{ for RFP or LAND segments} \end{cases}$$

where

$(X_T', Y_T') = (X_T, Y_T)$ for circular segments

$(X_T', Y_T') = (X_n, Y_n)$ for straight RFP segments

$(X_T', Y_T') = (X_L, Y_L)$ for straight LAND segments

(X_L, Y_L) = coordinate of end point of LAND segments

$$\dot{D}_Y = \begin{cases} \dot{D}_{YE}, \text{ for VOR/TAC or WPT} \\ -\sin(\lambda_{CE} - \psi_{RNY}\,(\dot{X}_A) + \cos(\lambda_{CE} - \psi_{RNY})\,(\dot{Y}_A), \\ \text{for RFP or LAND segments} \end{cases}$$

5.3.4.2.4.3 Circular Capture Progression Logic to Roll-Out and Track - The circular capture law will sequence to a roll-out phase (where the predictive roll angle is set to zero) when the following condition is satisfied:

$$|D_{XC} - D_X| \leq \frac{V_G}{2} \left(\frac{\phi_1}{\dot{\phi}_1}\right) \quad \text{ft}$$

During the roll-out phase of the circular capture, the closed-loop control to the circular capture path is still in effect.

The circular capture law will sequence to the track computations when the following condition is met:

$$D_X \geq D_{XC} \quad \text{ft}$$

5.3.4.2.5 Lateral On-Course Criterion - If one of the LD course modes is engaged, the on-course integration submode will be in effect when the following two conditions are satisfied:

$$|D_{YE}| \leq 500 \quad \text{ft}$$

$$|\dot{D}_{YE}| \leq 20 \quad \text{ft/s}$$

When the on-course criterion is lost, crosstrack integration is inhibited.

5.3.4.2.6 VOR/TACAN Course Guidance (VOR CRS, TAC CRS) - The computations for VOR and TACAN course guidance are identical except for the source of the bearing and range data. The crosstrack error is computed by

$$D_Y' = D_{STN} \sin \lambda' \quad \text{ft}$$

where

D_{STN} = horizontal range to the station (VOR or TAC)

$$= \begin{cases} \sqrt{R_{STN}^2 - (h_{SEA} - h_{STN})^2} \text{ ft, if } R_{STN} > (h_{STN} - h_{STN}) \\ 0, \text{ otherwise} \end{cases}$$

where

h_{SEA} = altitude above sea level (ft)

h_{STN} = altitude of navaid station above sea level (ft)

R_{STN} = slant range to the station (ft)

$$\lambda' = \left\{\begin{array}{l} \lambda_{REF} - \lambda_{STN}, \text{ if } |\lambda_{REF} - \lambda_{STN}| < 90 \text{ degrees} \\ \lambda_{STN} - 180° - \lambda_{REF}, \text{ otherwise} \end{array}\right\} = \text{angular course deviation (deg)}$$

$$\lambda_{REF} = \text{reference course angle to the station (deg)} = \left\{\begin{array}{l} \lambda_{CE} \text{ for engage mode} \\ \lambda_{CA} \text{ for arm mode} \end{array}\right.$$

$\lambda_{STN} = \lambda_{VOR}$ or λ_{TAC} = bearing to the VOR/TACAN station (deg)

R_{VOR}, R_{TAC}, λ_{VOR} and λ_{TAC} are computed by the navigation program. The computation of crosstrack rate and position is illustrated in Figure 5-17, and is given by

$$\dot{D}_Y = \frac{s + \frac{1}{DY} D_Y + \left(\frac{1}{2\gamma_{DY}}\right)^2 s\, D'_Y}{s^2 + \frac{1}{DY} s + \left(\frac{1}{2\gamma_{DY}}\right)^2} \quad \text{ft/s}$$

$$D_Y = \frac{\dot{D}_Y + \frac{1}{\gamma_{DY}} D'_Y}{s + \frac{1}{\gamma_{DY}}} \quad \text{ft}$$

where $\ddot{D}_Y$ is crosstrack acceleration. It is computed as

$$\ddot{D}_Y = \ddot{y}_{LH} \cos(\lambda_{REF} - \psi) - \ddot{x}_{LH} \sin(\lambda_{REF} - \psi) \quad \text{ft/s}^2$$

where

$(\ddot{x}_{LH}, \ddot{y}_{LH})$ = local horizontal aircraft acceleration (ft/s^2)

ψ = aircraft heading (deg)

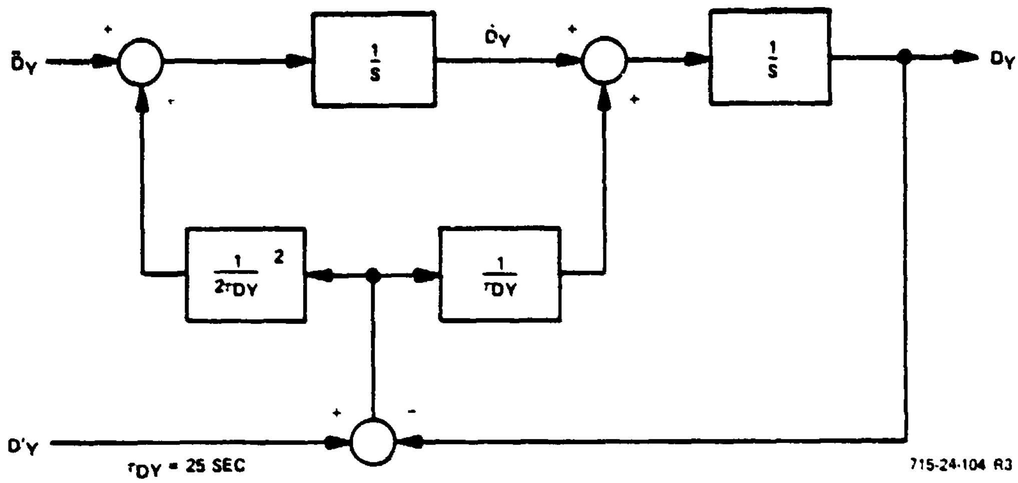


Figure 5-17
VOR and TACAN Crosstrack Complementary Filter

The horizontal acceleration is computed by the navigation program.

For engage modes,

$$D_Y = D_{YE}$$

$$\dot{D}_Y = \dot{D}_{YE}$$

$$\ddot{D}_Y = \ddot{D}_{YE}$$

$$D_Y' = D_{YE}'$$

$$\lambda' = \lambda_E'$$

and for ARM modes,

$$D_Y = D_{YA}$$

$$\dot{D}_Y = \dot{D}_{YA}$$

$$\ddot{D}_Y = \ddot{D}_{YA}$$

$$D_Y' = D_{YA}'$$

$$\lambda' = \lambda_A'$$

The initial condition value for the right integrator in Figure 5-17 is D_Y', and for the left integrator, an estimate of $\dot{D}_Y$ is used, given by

$$\hat{\dot{D}}_Y = V_G \sin(\lambda - \lambda_{REF})$$

where

λ = direction of V_G referenced to magnetic north (deg)

When the VOR or TACAN mode is armed, the displayed course angle blinks. If the applicable navigation "data good" flag is set (VORDGD for VOR, TACDGD for TACAN) and the lateral course capture criteria are met, the mode becomes engaged.

When the mode initially engages, the course display on the MSP stops blinking. Also, if the applicable "data good" flag drops while the mode is engaged, the mode reverts to the armed condition, and the primary LD mode engages. The "data good" flag is generated by the navigation program, and drops, for example, when the overstation condition occurs. After the overstation condition has passed, the armed guidance will normally re-engage if the lateral capture criteria are met.

5.3.4.2.7 <u>Waypoint Course Guidance (WPT CRS)</u>

The waypoint course guidance mode is functionally identical to VOR and TACAN course guidance, but the range and bearing are referenced to a selectable waypoint instead of a fixed navaid station. The waypoint coordinates are selected via the keyboard. However, the computations of D_Y and $\dot{D}_Y$ are based on the navigation-processed X and Y data instead of bearing and range to the station. D_Y is computed as

$$D_Y = (X_{WPT} - X) \sin(\lambda_{REF} - \psi_{RNY})$$

$$- (Y_{WPT} - Y) \cos(\lambda_{REF} - \psi_{RNY}) \quad \text{ft}$$

where

(X_{WPT}, Y_{WPT}) = waypoint location in the runway frame (ft)

(X, Y) = aircraft horizontal location in the runway frame (ft)

λ_{REF} = course reference angle (relative to magnetic north) (deg)

ψ_{RNY} = runway heading (relative to magnetic north) = -7.3 deg

$\dot{D}_Y$ is given by

$$\dot{D}_Y = \dot{Y} \cos(\lambda_{REF} - \psi_{RNY}) - \dot{X} \sin(\lambda_{REF} - \psi_{RNY})$$

where

$(\dot{X}, \dot{Y})$ = aircraft horizontal velocity in the runway frame (ft/s)

The transitions from the armed condition to the engaged condition and vice versa, and the blinking of the CRS display are the same as for the VOR and TAC CRS modes. The "data good" flag for WPT CRS is NAVVLD (good is $\neq 0$).

5.3.4.2.8 Reference Flight Path LD Guidance (LD RFP)

The reference flight path is illustrated in Figure 5-4, and reference flight-path data is described in Paragraph 5.3.5.

In order to set up the LD RFP mode, a valid entrance waypoint number must be selected via the keyboard. When LD RFP is armed, or engaged with no other LD mode armed, the reference course angle for RFP is displayed on the MSP and the course deviation to the selected waypoint is displayed on the HSI and ADI. If LD RFP is armed, it will engage when the lateral course capture criterion (Paragraph 5.3.4.2.4) is met and the navigation data is valid (NAVVLD set). A valid waypoint number is one that is included in the programmed reference flight path.

5.3.4.2.8.1 Reference Flight Path Entrance

If waypoint number n is selected for the entrance waypoint, the LD capture will be made on a backward extension of segment n, which is the segment connecting waypoint n-1 to waypoint n. For a straight segment, the backward extension is simply the straight line extending backward from segment n. If segment n is circular, a proper intercept of the circular segment must be set up. The intercept course must intercept the circular segment at an angle ≤ 90 degrees with respect to the segment direction at the intercept point.

The crosstrack displacement from the straight backward extension to waypoint n is given by

$$D_Y = (Y_A - Y_n) \cos \lambda_n - (X_A - X_n) \sin \ _n \quad \text{ft}$$

where

(X_n, Y_n) = location of waypoint n in the runway coordinates (ft)

λ_n = course angle of the backward extension from waypoint n relative to the X axis (deg)

If segment n is straight,

$$\lambda_n = \tan^{-1}\left(\frac{Y_n - Y_{n-1}}{X_n - X_{n-1}}\right) \quad \text{deg}$$

The crosstrack rate is given by

$$\dot{D}_Y = \dot{Y}\cos\lambda_n - \dot{X}\sin\lambda_n \quad \text{ft/s}$$

where

$(\dot{X}, \dot{Y})$ = aircraft velocity vector in the runway coordinates (ft/s)

5.3.4.2.8.2 Straight Segment Guidance

The crosstrack displacement from a straight segment n is computed by

$$D_Y = (Y - Y_n)\cos\lambda_n - (X - X_n)\sin\lambda_n \quad \text{ft}$$

where

(X_n, Y_n) = location of waypoint n in the runway coordinates (ft)

n = course angle of segment n relative to the X axis

$$= \tan^{-1}\left(\frac{Y_n - Y_{n-1}}{X_n - X_{n-1}}\right)$$

The crosstrack rate is computed by

$$\dot{D}_Y = \dot{Y}\cos\lambda_n - \dot{X}\sin\lambda_n \quad \text{ft/s}$$

5.3.4.2.8.3 Circular Segment Guidance

The crosstrack displacement from a circular segment n is computed as

$$D_Y = R_n - D_{An}$$

where

R_n = radius of curvature of segment n, positive for a right-turn segment (ft)

D_{An} = aircraft horizontal distance to the center of curvature of segment n, positive for a right-turn segment (ft)

$$= \pm \sqrt{(X_{Rn} - x)^2 \_ (Y_{Rn} - y)^2}$$

(X_{Rn}, Y_{Rn}) = location of center of curvature of segment n

The crosstrack rate is computed by

$$\dot{D}_Y = \frac{[(X_{Rn} - X)\ \dot{X} + (Y_{Rn} - Y)\ \dot{Y}]}{D_{An}} \quad \text{ft/s}$$

where

$(\dot{X}, \dot{Y})$ = aircraft velocity vector in the runway coordinates

5.3.4.2.8.4 Segment-to-Segment Transition

The system will perform a circular capture of segment n + 1 when the capture criteria defined in Paragraph 5.3.4.2.4.1b are met for the n + 1 segment. Whenever the system is tracking any given segment, it is always armed to capture the next segment.

5.3.4.2.9 Automatic Landing LD Guidance (LD LAND)

The selection among the two approach modes should be made before LAND is armed as described in Paragraph 5.3.2.3. If valid MLS navigation data is available, the straight-in and helical MLS LAND modes are allowed. If LD LAND is engaged when the MLS navigation data becomes invalid, the LAND mode disengages and LD LAND is rearmed.

When LD LAND is armed or engaged, the reference course angle is displayed on the MSP, and the course deviation is displayed on the HSI and the ADI with expanded sensitivity. If LD LAND is armed, it will engage when the lateral course capture criterion is met.

5.3.4.2.9.1 LAND-1 (Straight In)

The crosstrack displacement and rate from the MLS localizer are given by

$$D_Y = Y \quad \text{ft}$$

$$\dot{D}_Y = \dot{Y} \quad \text{ft/s}$$

where Y and $\dot{Y}$ are the runway Y-axis aircraft position and velocity as computed by the navigation computations.

5.3.4.2.9.2 LAND-2 (Helix)

The helix approach trajectory is described in Paragraph 5.3.4.1.9.2 and is illustrated in Figure 5-14. For segment (d), the helix, the crosstrack displacement is computed by

$$D_Y = R_{HX} + R_{AHX} \quad \text{ft}$$

where

R_{HX} = helix radius = -1189 feet

R_{AHS} = aircraft horizontal distance to the helix axis

$$= \sqrt{(X_{HX} - X)^2 + (Y_{HX} - Y)^2} \quad \text{ft}$$

The crosstrack rate is computed by

$$D_Y = \frac{[(X_{HX} - X)\,\dot{X} + (Y_{HX} - Y)\,\dot{Y}]}{R_{AHX}} \quad \text{ft}$$

where

$(\dot{X}, \dot{Y})$ = aircraft velocity vector in the runway coordinates (ft/s)

(X_{HX}, Y_{HX}) = coordinates of point on helix nearest aircraft (ft)

5.3.4.2.10 Go-Around LD Guidance

When Go-Around is initiated from a LAND-1 approach, an immediate heading-select to the runway heading of 353 degrees occurs. If Go-Around is initiated from a LAND2 approach, the existing roll attitude will be held (LD PRIMARY) until the aircraft heading falls within a ±20 degree range of the runway heading (353 degrees). When this condition is satisfied, a heading-select to the runway heading is initiated. The above described operation is achieved with logic that generates the changes to the basic LD mode directors and heading reference to provide the desired performance.

5.3.4.2.11 Course Deviation Indicator

The course deviations associated with the TACAN, VOR, WPT, RFP, and LAND guidance modes are computed whenever one of the above course modes has been selected via the Mode Select Panel, independent of whether the Autopilot or Flight Director has been engaged. These deviations are displayed on the HSI's Course Deviation Indicator (CDI).

In the manual course mode the deviations displayed on the CDI are computed directly from navigation data in the guidance software. However, when in a non-manual course mode (either Autopilot or Flight Director engaged) the course deviations are computed from guidance lateral deviations for TACAN, VOR, and WPT modes.

If one course mode is armed while another is engaged, the deviations for the armed mode are selected for display on the CDI. The deviation selected for display in the TACAN, VOR, and WPT modes is proportional to the angular course deviations, with scaling as shown in the following table. For the RFP and WPT course modes, the deviation shown on the CDI is proportional to the lateral crosstrack deviation with the following scale:

Mode	Course Deviation Display Scaling
TACAN	1 dot = 5 degrees
VOR	1 dot = 5 degrees
WPT	1 dot = 5 degrees
RFP	1 dot = 1000 feet
LAND	1 dot = 100 feet

When no course modes are selected, the CDI is centered.

Hence,

$$\delta_{CRS} = \begin{cases} 0 \text{ for centered} \\ L_{\pm 2}\left[\frac{1}{5}\left(\frac{D_Y}{D_{STN}}\right)\left(\frac{180}{\pi}\right)\right] \text{ dot, for VOR or TACAN modes} \\ L_{+2}\left[\frac{1}{5}\left(\frac{D_Y}{R_{WPT}}\right)\left(\frac{180}{\pi}\right)\right] \text{ dot, for WPT mode} \\ L_{\pm 2}\left[\frac{1}{10000} D_Y\right] \text{ dot, for RFP} \\ L_{\pm 1}\left[\frac{1}{100} D_Y\right] \text{ dot, for LAND} \end{cases}$$

where

δ_{CRS} = course deviation indicator value (dots)

D_Y = guidance lateral crosstrack deviation (ft) (D_{YE} for engaged modes, D_{YA} for armed modes)

D_{STN} = distance to NAV station (ft)

R_{WPT} = range to waypoint (ft)

5.3.4.2.12 TO/FROM Indicator

The TO/FROM indicator on the HSI is associated only with the TACAN, VOR, and WPT guidance course modes, and indicates the direction of a selected course radial in relation to a selected navaid or waypoint.

The TO/FROM indicator indicates TO (arrow in the same direction as course-select pointer) if the absolute value of the angle between the course-select pointer and the bearing to the selected navaid or waypoint is less than 90 degrees, i.e.:

$$|\lambda_{REF} - \lambda_{STN}| < 90°$$

where

λ_{REF} = selected course reference angle relative to magnetic north (deg)

λ_{STN} = bearing to selected navaid or waypoint (deg)

The TO/FROM indicator will indicate FROM (arrow in opposite direction to course-select pointer) if the absolute value of the angular difference between the course-select pointer and the bearing to the selected navaid or waypoint is geater than 90 degrees, i.e.:

$$| \ REF - \ STN| \geq 90°$$

where

λ_{REF} and λ_{STN} are defined above.

If a course mode associated with the TO/FROM indicator is armed while another is engaged, the TO/FROM indicator will be associated with the armed mode. When no TO/FROM course modes are selected, the TO/FROM indicator is stored out of view.

5.3.5 Reference Flight Path and LAND Data

The parameters for Reference Flight Path implemented in the basic program are given in Table 5-13. The table includes the following information:

- Waypoint/Segment Index - Four waypoints are used, indexed from 1 through 4 as shown in Figure 5-4. Segment n connects waypoints n-1 and n.

TABLE 5-13

REFERENCE FLIGHT PATH DATA
(segment n terminates at WPT n)

Index	n	1	2	3	4
Waypoint Location	X_n	4450	-25000	-25000	4450
	Y_n	-9000	-9000	0	0
	h_n	2500	2500	2500	2500
Radius of Curvature	R_n	-4500	0	-4500	0
Center of Curvature	X_{Rn}	4500	-	-25000	-
	Y_{Rn}	-4500	-	-4500	-
Segment FPA	γ_n	0	0	0	0
Segment Final Course Angle/RNY	λ_n	180	180	0	0

- Waypoint Location, (X_n, Y_n, h_n) - The waypoint locations are referenced to the runway coordinate system, measured in feet ($h_n = -z_n$).

- Radius of Curvature, R_n - In addition to the conventional definition of radius of curvature, R_n is defined to be negative for a left-turn circular segment, and zero for a straight segment (thereby identifying the straight and circular segments). It is measured in feet.

- Center of Curvature, (X_{Rn}, Y_{Rn}) - Conventional definition, measured in feet.

- Final Course Angle, λ_n - The course angle of the end of segment n does not have to be the same as the beginning of segment n + 1. For the case where segment n is circular and the course angle makes an abrupt change at waypoint n + 1, λ_n is needed and is given by

$$\lambda_n = \begin{cases} \tan^{-1} \dfrac{X_n - X_{Rn-1}}{Y_{Rn-1} - Y_n}, & \text{if } R_n > 0 \\ \tan^{-1} \dfrac{X_{Rn-1} - X_n}{Y_n - Y_{Rn-1}}, & \text{if } R_n < 0 \end{cases}$$

The data which specifies the LAND2 path is given in Table 5-14. The LAND2 path is specified with the same types of parameters as for the Reference Flight Path. The segment index identifies the endpoint of a segment ((a) through (e)), except for Index 3 which is on an extension of the straight segment (c), right above the touchdown point (to simplify computations). Some of the parameters in the table are computed in terms of D_H, which is the selectable (via the keyboard) horizontal distance from the helix tangency point to the touchdown point (see Figure 5-14). Distances are given in feet and angles in degrees.

TABLE 5-14
LAND-2 DATA

Index	n	1	2	3	4	5
Waypoint Location	X_n	-16120	-8583	-683	-683 - .9511 D_H	-583
	Y_n	-6680	-4231	-1664	-1664 - .3090 D_H	-1664
	h_n	2500	2500	1611	10 + .1070 D_H	10
Radius of Curvature	R_n	-1189	0	0	-1189	0
Center of Curvature	X_{Rn}	-15750	0	0	X_4 - 367	0
	Y_{Rn}	-7810	0	0	Y_4 + 1131	0
Segment FPA	γ_n	0	0	-6.11	-6.11	-6.11
Segment Final Course Angle/RNY	λ_n	18	18	18	18	18

5.4 NAVIGATION

The Basic computer navigation program computes aircraft position and ground velocity with respect to the Crow's Landing runway coordinate frame, using ground-based navaid position data augmented with acceleration data derived from a strapdown system. The available navaid data sources are the VOR/DME at Stockton and the TACAN and MLS at Crow's Landing. The acceleration data is supplied by the outputs of three body-axis-mounted accelerometers. The vertical position information is derived from barometric, MODILS and radio altitude data.

The raw navaid data is first prefiltered to reduce noise levels and "coast through" short-term data dropouts. The general prefilter block diagram is shown in Figure 5-18. The raw data value ρ is input to a rate-limiting lag filter of time constant τ The output of the filter, $\hat{\rho}$, is the estimated value of the raw data. The phase lag that is introduced by the $\hat{\rho}/\rho$ lag filter is compensated for by the summation of the raw data rate estimate $\hat{\dot{\rho}}$ at the integrator. This rate estimate is derived from aircraft heading and ground speed. When the raw data value differs from the estimated value by more than a specified amount (ϵ), the switch in the position loop opens and the raw data is ignored until it returns to a reasonable value. In this mode the estimate is updated by the integral of $\hat{\dot{\rho}}$. The position loop is also opened if the data becomes invalid. The filter time constant τ and the rate limit value are chosen for each type of Navigation source and variable to reject noise on ρ and minimize steady-state errors due to errors in the $\hat{\dot{\rho}}$ estimate.

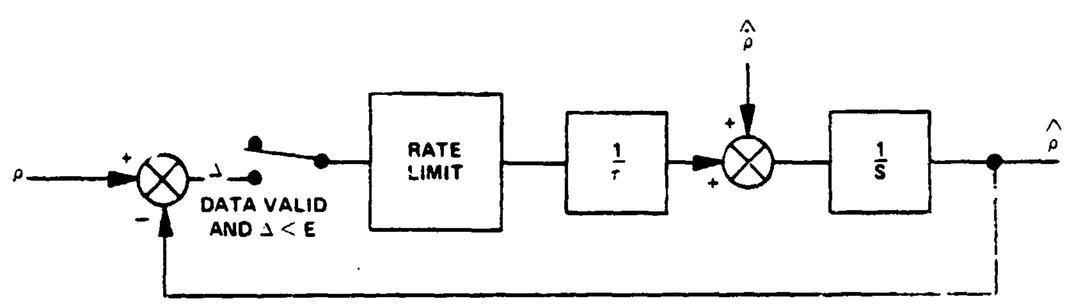


Figure 5-18
Navigation Prefilter

The filtered navaid estimates for a given navigation source are then transformed into runway-axis-referenced coordinates (X_R, Y'_R, Z'_R). Figures 5-19 through 5-21 illustrate the geometries involved in these transformations. The outputs of these transformations become the inputs to the navigation complementary filters. The X complementary filter is shown in Figure 5-22, and is discussed below. The Y complementary filter is obtained by replacing X with Y wherever it is used in the figure or the text since the two filters are functionally identical.

The X complementary filter is a third-order filter with fixed gains. The gains are chosen to give complementary weighting of the acceleration (X_R) and position (X'_R) inputs. The outputs of the filter are estimated position (X_R), inertial velocity ($\dot{X}_R$), and the wind components in the runway coordinate system ($\dot{X}_W$). The proportional and integral paths in the feedback are included for compensating bias errors in the accelerometers. The integral path is switched in only when valid navaid position data is present.

The dead-reckoning mode goes into effect when the selected navaid has remained invalid for a period of 5 seconds. In this mode the velocity loop is closed and the position loop is open. The last value of filtered position, X_R, is updated with position changes derived from the air mass velocity component ($\dot{X}_A$) and the last (just prior to switching to dead reckoning) computed value of wind velocity component ($\dot{X}_W$). The position update accurate in this mode deteriorates with time, especially with changing wind conditions, and hence dead reckoning is limited to a period of 2 minutes.

The Z complementary filter which estimates aircraft height above the Crow's Landing runway is shown in Figure 5-23. The filter is similar to the X and Y complementary filters, however the gains are a function of the aircraft altitude. The Z filter becomes a third-order filter only below 100 feet when a high- gain integral term is used to eliminate any height bias due to accelerometer biases.

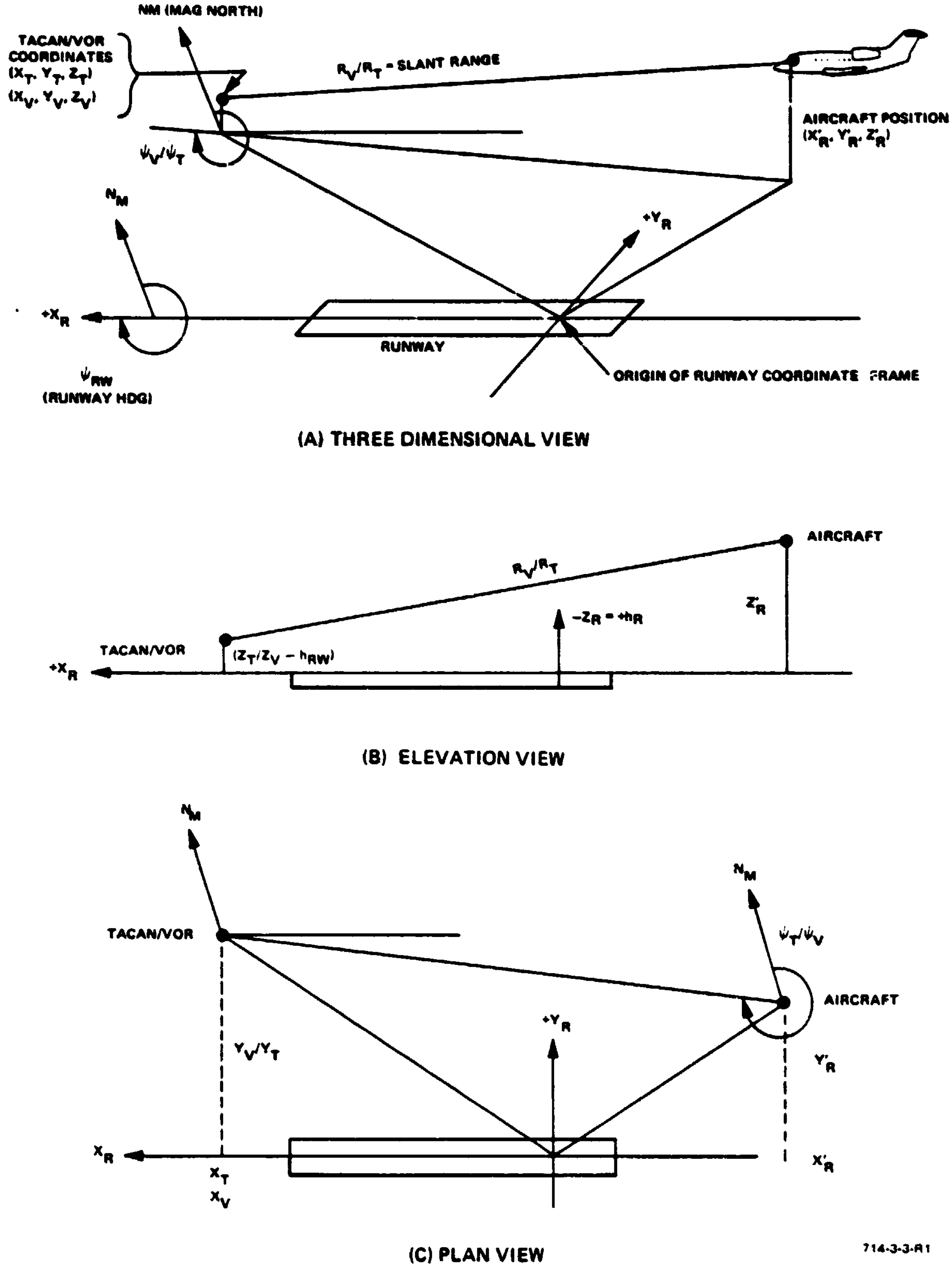


Figure 5-19
Geometry of TACAN and VOR/DME Navigation

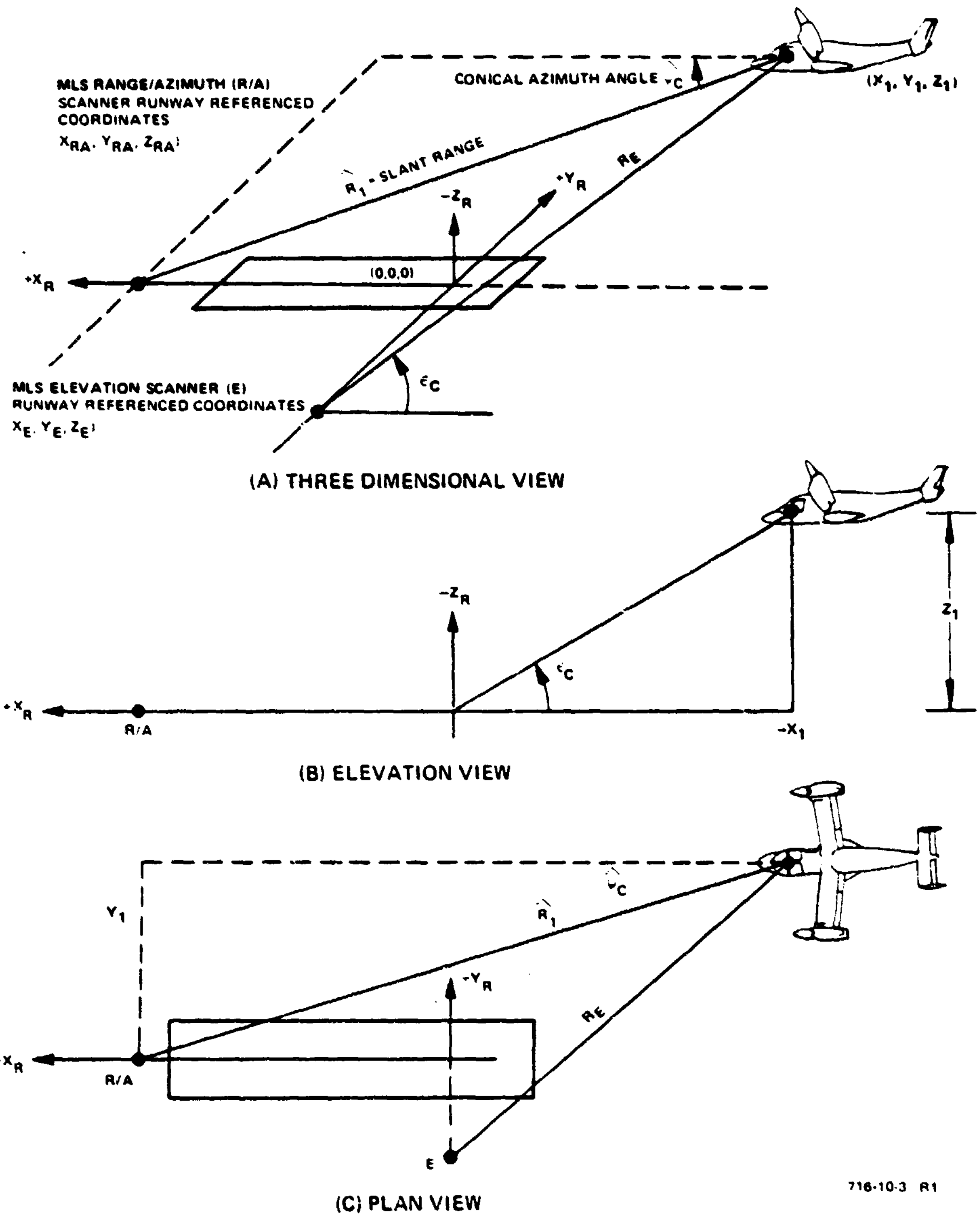


Figure 5-20
Geometry for MLS Navigation

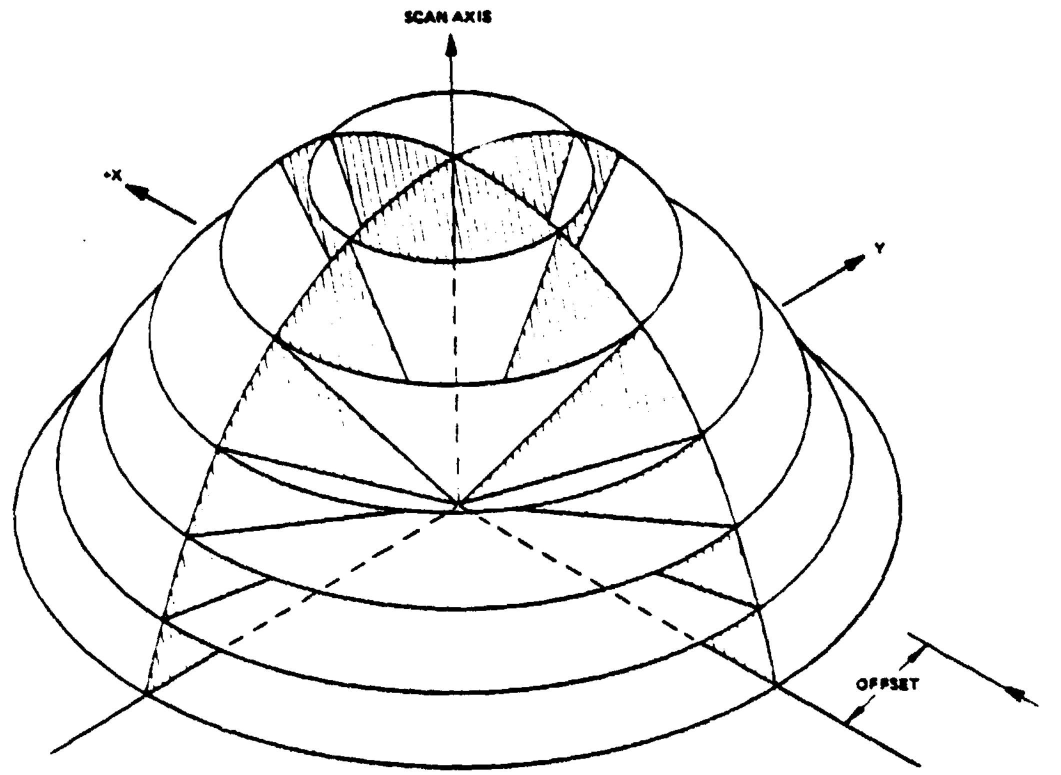
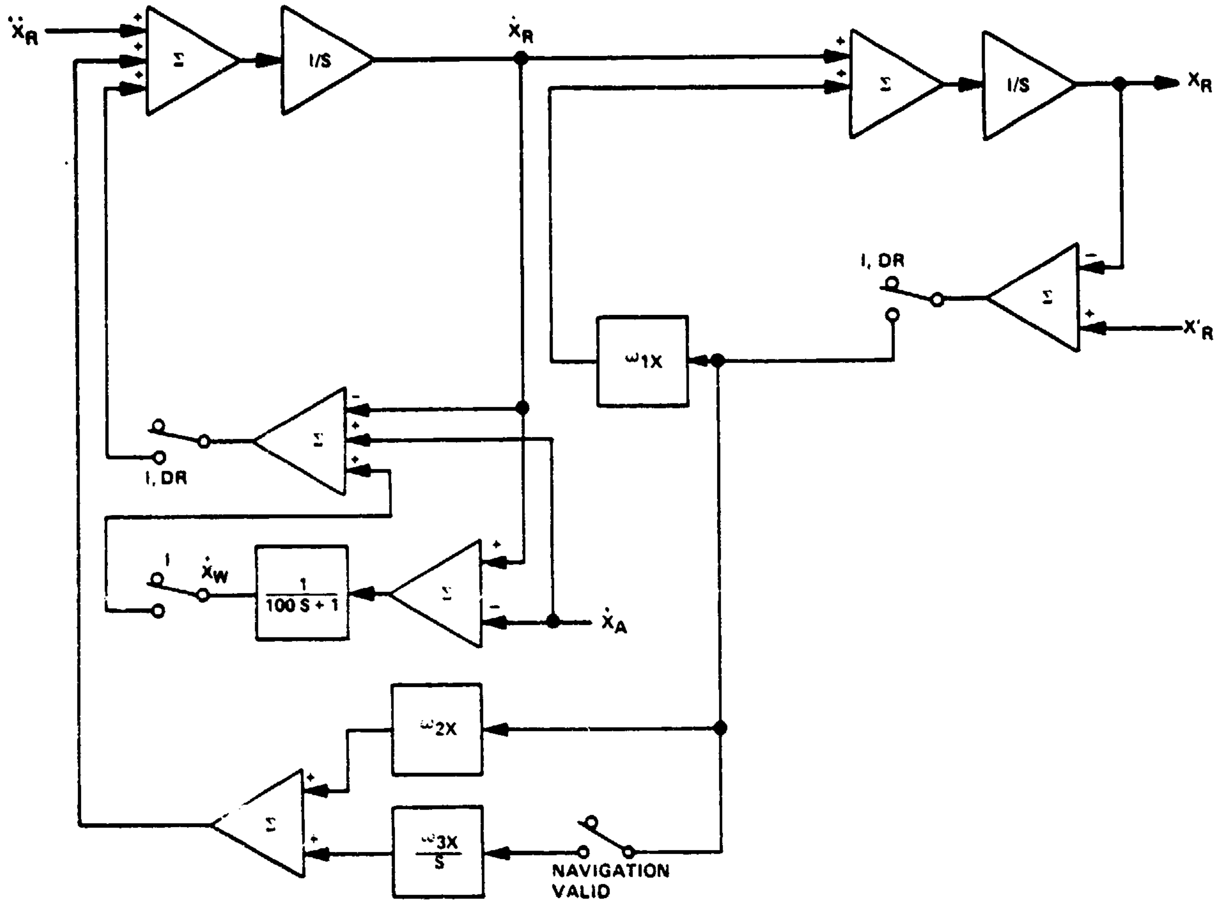


Figure 5-21
MLS Conical Elevation-Angle Radiation Pattern



NOTES:

$\ddot{X}_R$ = $f_1(\theta, \phi, \psi, a_x, a_y, a_z)$

X'_R = NAVAID DERIVED POSITION

$\dot{X}_A$ = A/C VELOCITY RELATIVE TO AIR MASS (X-COMPONENT)

$\dot{X}_W$ = WIND VELOCITY (X-COMPONENT)

DR = DEAD RECKONING MODE

I = INITIALIZATION

NOMINAL FILTER GAINS

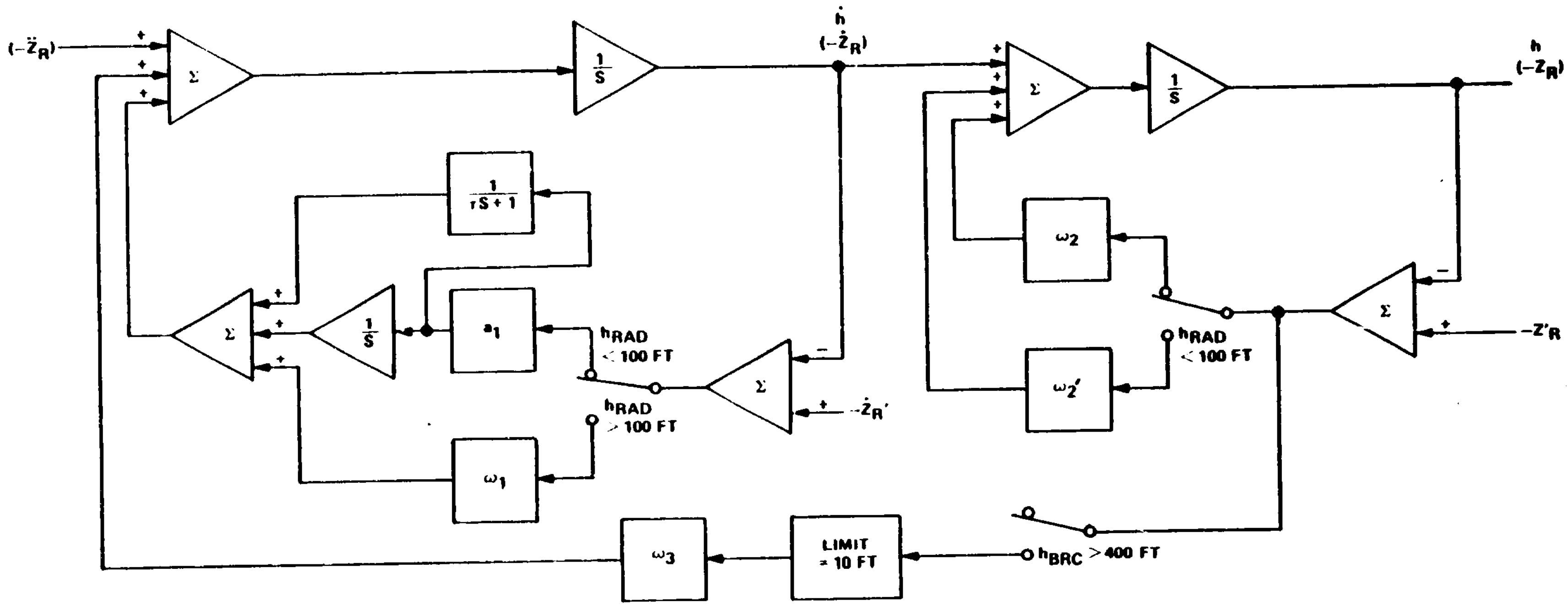
$\omega_{1X} = 0.2$

$\omega_{2X} = 0.02$

$\omega_{3X} = \omega_{2X}/120$

711-3-30-R1

Figure 5-22
X Complementary Filter



NOTES:

$Z_R = f(\theta, \phi, a_X, a_Y, a_Z)$

$Z'_R = \left[\dfrac{\dot{Z}_R(n) - \dot{Z}_R(n-1)}{\Delta T = 0.050 \text{ SECS}}\right]$

WHERE: $Z_R = h_R$ FOR $h_R < 100$ FT

$= h_B$ FOR $h_R > 100$ FT

NOMINAL GAINS:

$\omega_2 = 0.1$ (> 100')

$\omega_2' = 2$ (< 100')

$\omega_1 = 0.2$

$\omega_3 = 0.025$

$a_1 = 1/7$

$\tau = 1$ SEC

714-3-61-R2

Figure 5-23
Z Complementary Filter

5.5 FAILURE MONITORING AND DIAGNOSTICS

The failure monitoring function consists of detecting that a system failure has occurred. This is done by monitoring the hardware valids and computing software valids. Timers are provided for some of the valids. When such a valid drops, the failure is determined to be conclusive only after the valid has stayed low for a specified time. This prevents nuisance failure actions for momentary loss of a valid, such as for power transients, settling time, etc.

The diagnostic function consists of identifying the failure that has occurred, and displaying appropriate information to the pilot. If the engaged mode depends on a failed unit, then the mode is disengaged and the failure is annunciated.

The monitors are grouped in three categories, depending on the mode for which the failures are critical:

- Manual Monitors
- Flight Director Monitors
- Autopilot Monitors

The Manual monitors include:

- Vertical Gyro
- Radio Altimeter
- Research Computer
- Hardware Discretes

The Flight Director monitors include:

- Pitch Rate Gyro
- Roll Rate Gyro
- Yaw Rate Gyro
- Accelerometers
- Speed Limits
- Land Navigation

The Autopilot monitors include:

- Servos
- Conversion System
- Force Transucers
- RPM Governor Actuator
- Mast Torque

When neither the Flight Director nor the Autopilot is on, only the Manual monitors are checked. When only the Flight Director is on, both the Manual and the Flight Director monitors are checked. When the Autopilot is on, all monitors are checked.

Table 5-15 summarizes the failure monitoring criteria. When a failure occurs, it will be annunciated on the DED. The message associated with each failure monitor is shown in Table 5-15. A message-alert light on the cockpit instrument panel annunciates that a failure message is displayed on the DED.

TABLE 5-15

FAILURE MONITOR SUMMARY

Monitor	Failure Criteria	Failure Message	Action
<u>Vertical Gyro</u>			
a) VG Valid	VG Valid = 0	VERT GYRO	Note 1
b) Pitch Attitude	$\|\theta\| > 30$ degrees	EXCESS PITCH	Note 1
c) Roll Attitude	$\|\phi\| > 55$ degrees	EXCESS ROLL	Note 1
<u>Research</u>			
a) Computer Valid	Research computer not valid and Research mode engaged	RES COMPUTER	Note 1
b) Research to Basic I/O	Test words incorrect and Research mode engaged	BASIC/RES IO	Note 1
c) Research Software Valid	Software Valid = 0	RES SOFTWARE	Note 1

TABLE 5-15 (cont)

FAILURE MONITOR SUMMARY

Monitor	Failure Criteria	Failure Message	Action
d) Research Autopilot	No Research mode selected and Research Autopilot engaged	RES DISABLED	Note 1
	LD and VL guidance not selected and Research Autopilot engaged	RAP INVALID	Note 1
<u>Data Adapter</u>			
Data Adapter Servo Command End-Around Monitors	Difference between command output and command input > 3 percent full scale	DATA ADAPTER	Note 2
<u>Pitch Rate Gyro</u>			
Pitch Rate Gyro Valid	Pitch Rate Gyro Valid = 0	PITCH R GYRO	Note 1
Pitch Rate Error	Pitch Rate Error > 5 deg/sec (see text)	PITCH RATE	Note 1
<u>Roll Rate Gyro</u>			
Roll Rate Gyro Valid	Roll Rate Gyro Valid = 0	ROLL R GYRO	Note 1
Roll Rate Error	Roll Rate Error > 10 deg/sec (see text)	ROLL RATE	Note 1
Excessive Roll Rate	Roll Rate > 20 deg/sec	EXCES R RATE	Note 1
<u>Yaw Rate Gyro</u>			
Yaw Rate Gyro Valid	Yaw Rate Gyro Valid = 0	YAW R GYRO	Note 1
Yaw Rate Error	Yaw Rate Error > 10 deg/sec (see text)	YAW RATE	Note 1
<u>Conversion System</u>			
Conversion Valid	Conversion Valid = 0	CONVRSN SYS	Note 2
Conversion Position	Position Error > 3°	CONVRSN RATE	Note 2

TABLE 5-15 (cont)

FAILURE MONITOR SUMMARY

Monitor	Failure Criteria	Failure Message	Action
Power Lever			
Power Lever Valid	Power Lever Valid = 0	POWER LEVER	Note 2
Flap Interface			
Flap Lever Valid	Flap Lever Valid = 0	FLAP LEVER	Note 2
Flap System Valid	Flap System Valid = 0	FLAP SYS	Note 1
SCAS/FFS Interface			
SCAS/FFS Valid	SCAS/FFS Valid = 0	SCAS/FFS SYS	Note 2
RPM Governor			
RPM Governor Valid	RPM Governor Valid ≠ 0	RPM GOVERNOR	Note 2
Accelerometer			
Lateral Acceleration	a_Y sensor > .25g for 2 seconds	LAT ACCEL	Note 1
Longitudinal Acceleration	a_X sensor > .25g for 2 seconds	LONG ACCEL	Note 1
Normal Acceleration	a_Z sensor > 1g about a 1g bias for .2 second	NORMAL ACCEL	Note 1
Force Transducers			
Pitch Over-Force	Force > 3.5 pounds for 1 second Force > 7.0 pounds (no delay)	PITCH OVRFRC	Note 2
Roll Over-Force	Force > 3.0 pounds for 1 second Force > 6.0 pounds (no delay)	ROLL OVRFRCE	Note 2
Yaw Over-Force	Force > 10.0 pounds for 1 second Force > 20.0 pounds (no delay)	YAW OVRFRCE	Note 2

TABLE 5-15 (cont)

FAILURE MONITOR SUMMARY

Monitor	Failure Criteria	Failure Message	Action
Radio Altimeter			
Radio Altimeter Valid (Monitor for Altitude < 500 ft)	Radio Altimeter Valid = 0	RAD ALT RCVR	Note 1
Altitude Error (Monitor for Altitude < 500 ft)	Error > 50 feet (see text)	ALTITUDE	Note 1
ADI Indicator			
Attitude Valid**	ADI Valid = 0 for .1 second	ADI ATTITUDE	Note 3
MFD System			
MFD Display Unit Valid	MFD Display Unit Valid = 0 for 10 seconds	MFD DISPLAY	Note 3 Light Map Annunciator on MFD
MFD Symbol Generator Valid	MFD Symbol Generator Valid = 0 for 10 seconds	MFD SYMB GEN	Note 3 Light Map Annunciator on MFD
Compass			
Power Valid	Flag Alarm Warning = 0 for 10 seconds	COMPASS SYS	Note 1
HSI System			
Heading Valid		HSI HEADING	Note 1
Course-Select Valid		HSI CRS SEL	Note 3
Heading-Select Valid	Any HSI Valid = 0 for 10 seconds	HSI HDG SEL	Note 3
Bearing 1 Valid		HSI BEARING1	Note 3
Bearing 2 Valid		HSI BEARING2	Note 3
Mode Select Panel			
MSP T/R Valid	MSP Valid = 0 for 1 second	MODE SEL PNL	Note 3

TABLE 5-16 (cont)

FAILURE MONITOR SUMMARY

Monitor	Failure Criteria	Failure Message	Action
<u>Flight Mode Annun</u>			
FMA Valid	FMA Valid = 0 for 1 second	FLT MODE ANN	Note 3
<u>Data Entry Display</u>			
DED Valid	DED Valid = 0 for 1 second	DATA ENT DSP	Note 3
<u>Blower</u>			
Blower Valid	Air Flow Valid = 0 for 10 seconds	BLOWER OFF	Note 3
<u>Speed Monitor</u>			
High Speed	$V_{IAS} \geq V_{C_L}$ max (see text)	TOO FAST	Note 1
Low Speed	$V_{IAS} < V_{C_L}$ min (see text)	TOO SLOW	Note 1
<u>LAND Navigation Monitor</u>			
Navigation Director	No MLS navigation when land course mode engaged	LOST MLS NAV	Note 1

Note 1: Disengage Autopilot. Turn on V/STOLAND caution*. Disengage Flight Director. Display message.

Note 2: Disengage Autopilot. Turn on V/STOLAND caution*. Display message.

Note 3: Display Message.

*V/STOLAND caution = flashing light.

**This monitor is locked out if VG valid drops.

The pitch, roll, and yaw rate gyro errors referred to in Table 5-15 are computed by taking the derivatives of the VG and DG attitudes and comparing them to the transformed pitch, roll, and yaw rate gyro outputs. The altitude error referred to in Table 5-15 is computed by comparing the radio-altimeter derived altitude to a blended altitude derived from the barometric altimeter and MLS range and elevation data. This comparison is only performed for altitudes less than 500 feet to ensure a proper transition to and operation of radio altitude during landings. The speed limits V_{C_L} max and V_{C_L} min needed to implement the speed monitors referred to in Table 5-15 are computed in the configuration control portion of the V/STOLAND program. These speed limits are functions of the pylon angle as shown in Figure 5-24. The limits are specified in the XV-15 aircraft familiarization and operating document to define a conversion corridor within which the possibility of stalling the aircraft is minimized.

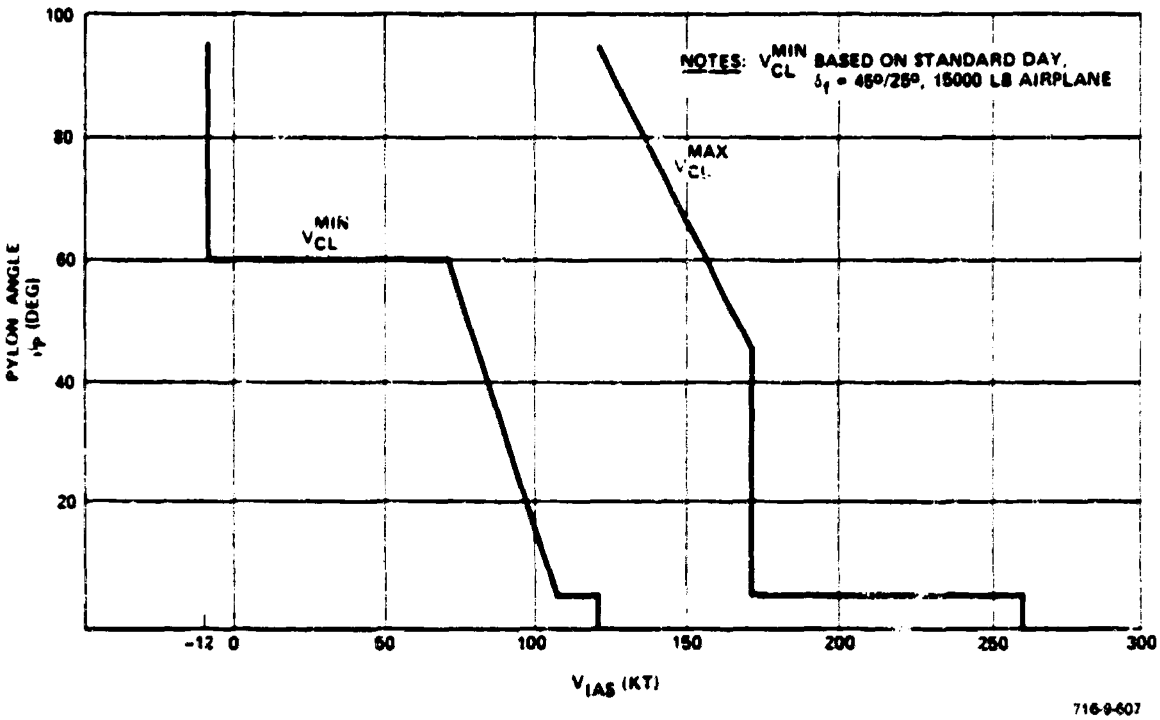


Figure 5-24
Airspeed Limits

SECTION VI

SYSTEM VALIDATION AND TESTING

C-4

SECTION VI

SYSTEM VALIDATION AND TESTING

The V/STOLAND system was subjected to the following five stages to assure the integrity of and validate the performance requirements of the system:

(1) Component Acceptance Testing

(2) Software Development Testing

(3) Static Acceptance Testing

(4) Dynamic Validation Testing at the Sperry Simulation Facility

(5) Dynamic Acceptance Testing at the ARC Simulation Facility

Formal test specifications were written and submitted for the (1), (2), and (5) test stages. Informal test procedures and results were also written for the software development testing (2). The dynamic acceptance test procedures for (5) were used for (4), but without formal sign-off. The following paragraphs describe the five test stages.

6.1 COMPONENT ACCEPTANCE TESTING

Complete electrical functional tests were performed on the following components, using Sperry Automatic Test Equipment and special test fixtures, and were formally witnessed by Sperry and government QC representatives:

- Mode Select Panel
- Data Entry Keyboard
- Data Entry Display
- Flight Mode Annunciator
- Data Adapter
- 1819B Computer

The Acceptance Test Specification published for each component details the test procedures and approval requirements.

The Sperry product-line components underwent the regular factory acceptance tests. These components include:

- HSI Display Unit (RD-202)
- HSI Amplifier Rack
- HSI Signal Conditioning Unit
- ADI (HZ-6F)
- Static Pressure Transducer
- Accelerometer Assembly (3-Axis)
- Rate Gyro Assembly, Yaw/Roll
- Rate Gyro Assembly, Pitch

Since the MFD system (Display Unit and Symbol Generator) requires a computer to generate input data, it was tested in the closed-loop simulation environment as part of the Static and Dynamic Acceptance Tests.

6.2 SOFTWARE DEVELOPMENT TESTING

After each software module had been coded, assembled and first-cut debugged, an "Informal Software Test Procedure" was written which listed the tests to be performed on the module and the desired results. This procedure was often updated as the tests were performed, and the results were recorded and signed by the engineer. This procedure thus provided a running log of any problems encountered and the changes made to solve the problems. The tests were performed on an interactive test facility utilizing utility programs that permit running any part of a program, or stepping through a program while inspecting all internal registers of the processor. Some modules were also run in a simulation environment to obtain input data.

6.3 STATIC ACCEPTANCE TESTING

The Static Acceptance Test (SAT) is a formal test for validating all electrical interfaces between all system components, and to/from all external destinations/sources. A SAT Procedure document was published which detailed test procedures, performance requirements, and acceptance requirements. Figure 6-1 shows the cabling between the system components and peripherals in the simulation environment.

The Static Acceptance Test consists of six groups of tests conducted in the order given:

(1) Basic and Research computers to peripherals, and Basic to Research Computers

(2) Basic computer to cockpit instruments, via Data Adapter; also includes computer Built-In-Test

(3) Basic computer to Airborne Hardware Simulator (AHS) via Data Adapter, and simulation cab to AHS

(4) Additional discrete Basic computer outputs

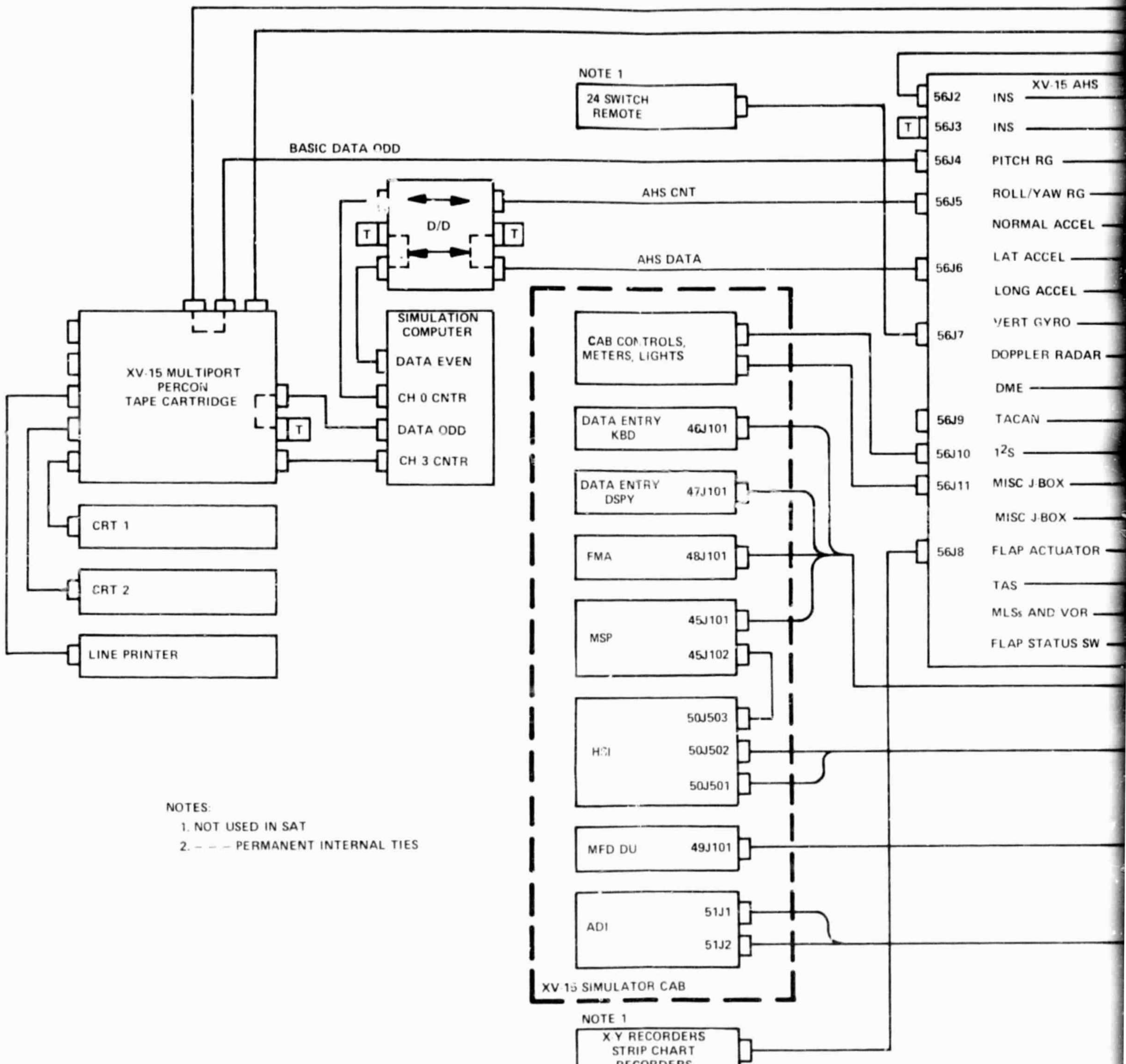
(5) Autopilot control engage logic and interlocks in Data Adapter

(6) Nonsimulation-environment interfaces (static pressure transducer)

Most of the tests are implemented by running a test program developed for this purpose in the various computers, with printouts of test results where applicable. The Preflight Test program was also used for group (2) above.

6.4 DYNAMIC VALIDATION TESTING AT SPERRY

Before the system was shipped to NASA it underwent a complete validation test in a simulation facility at Sperry. Figure 6-2 is a block diagram of this facility, and Figure 6-3 shows the facility area photographically. The simulation cab, shown in Figure 6-4, contains the V/STOLAND cockpit panels, plus several aircraft instruments simulated by galvanometer-type meters: calibrated air speed, baro altitude, vertical speed, rpm, flap angle, pylon angle, angle of attack.



NOTES:
1. NOT USED IN SAT
2. – – – PERMANENT INTERNAL TIES

FOLDOUT FRAME

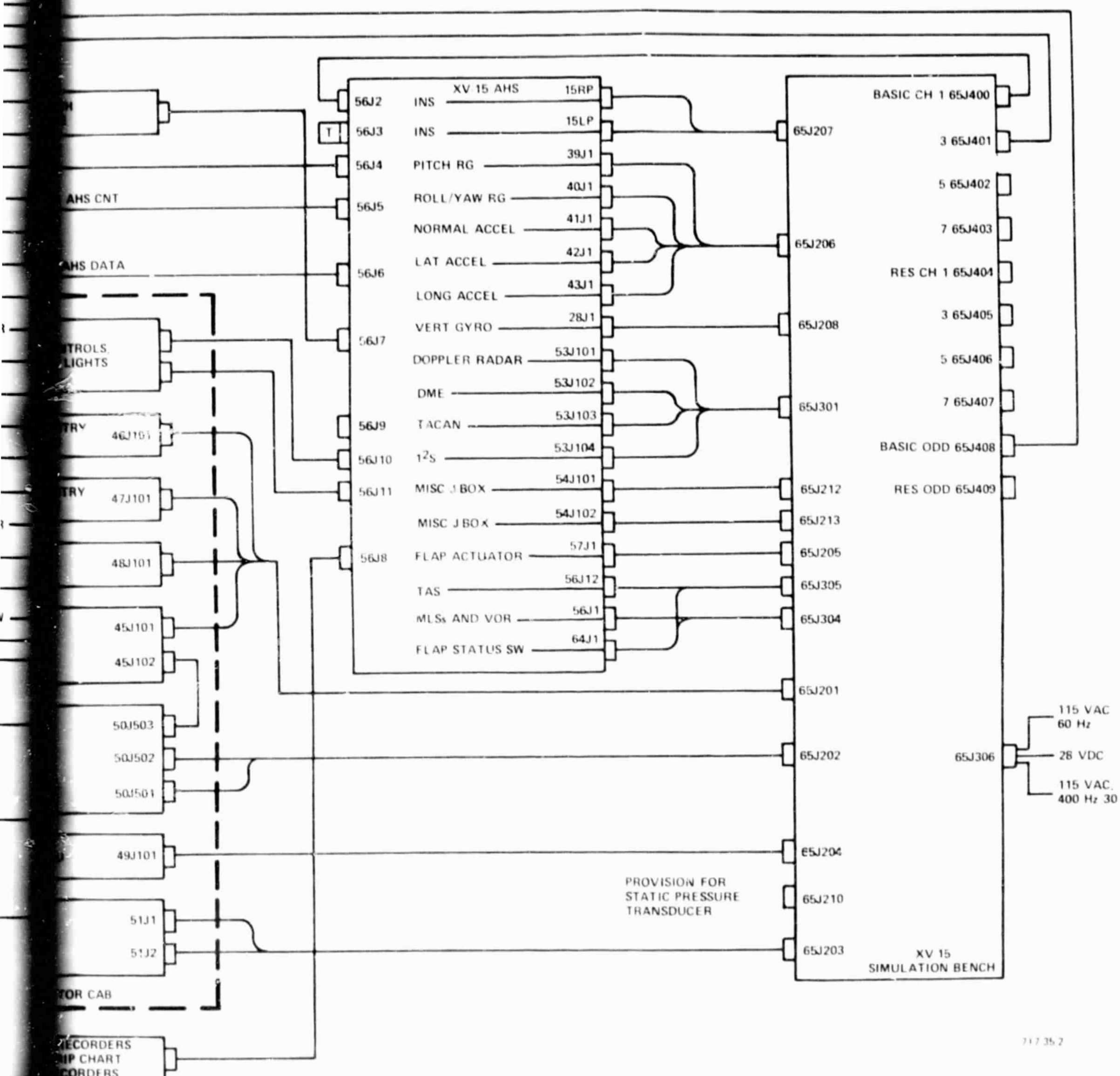


Figure 6-1
V/STOLAND XV-15 Simulation
Cabling Diagram

FOLDOUT FRAME

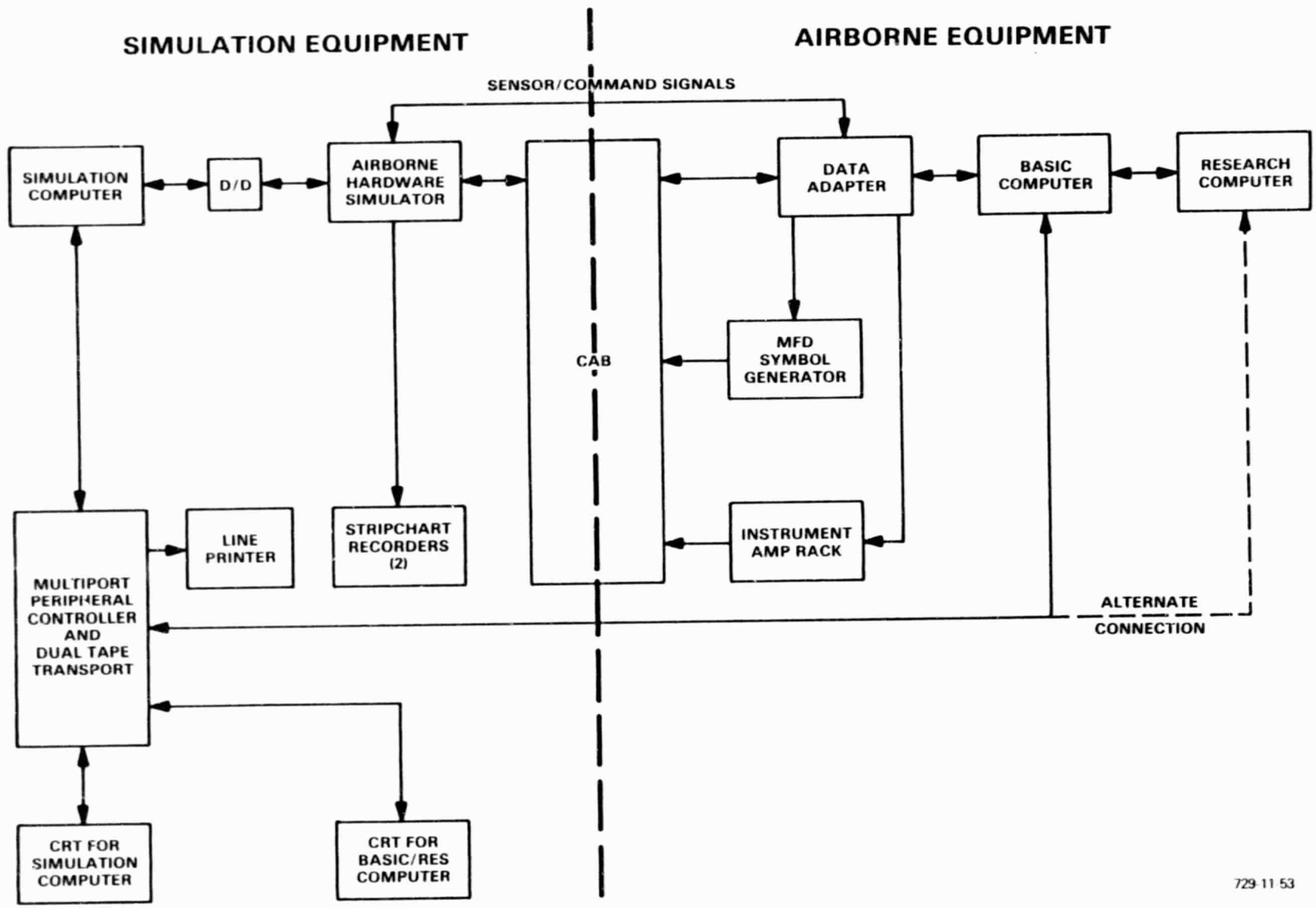


Figure 6-2
Sperry Simulation Facility Configuration

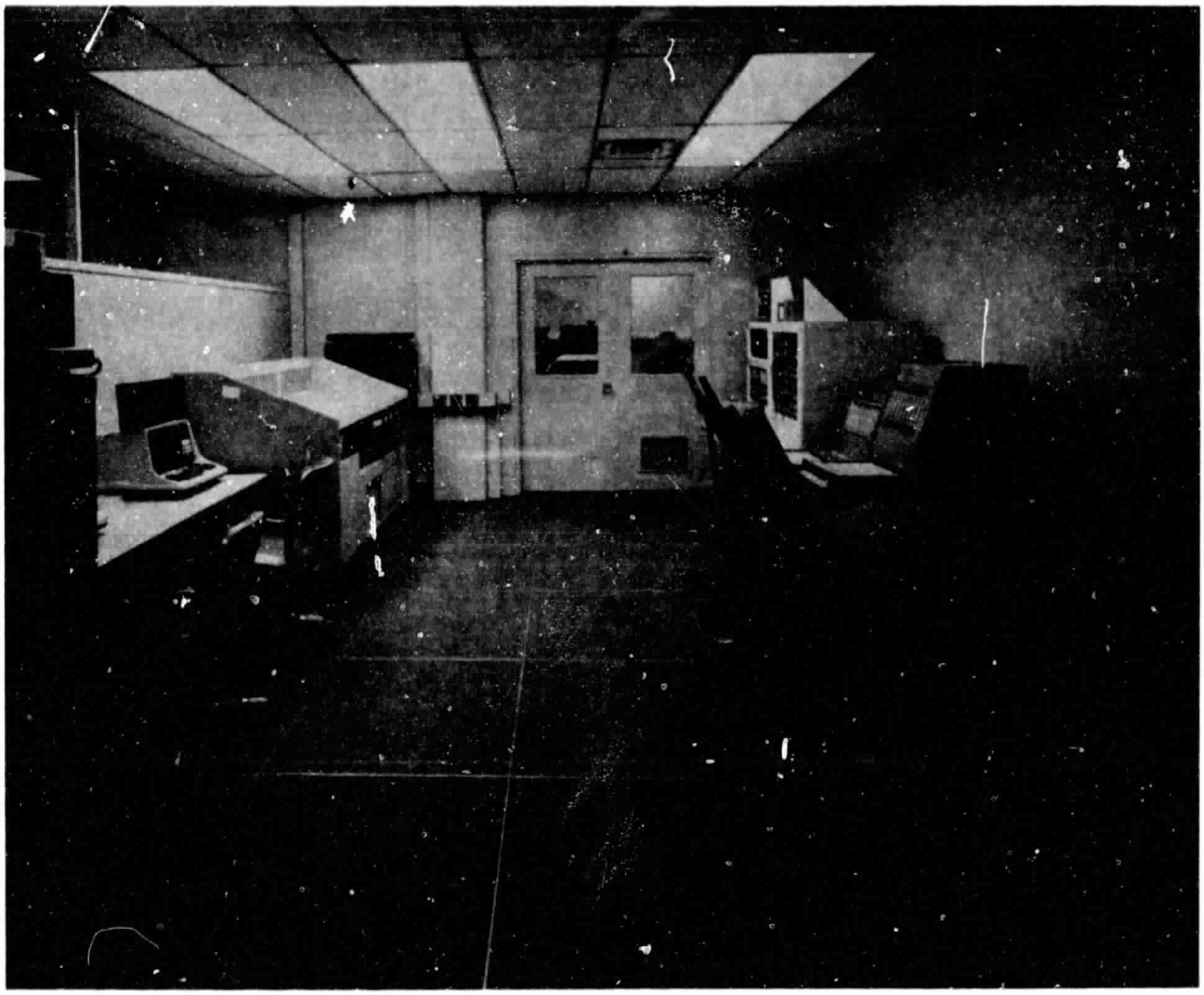


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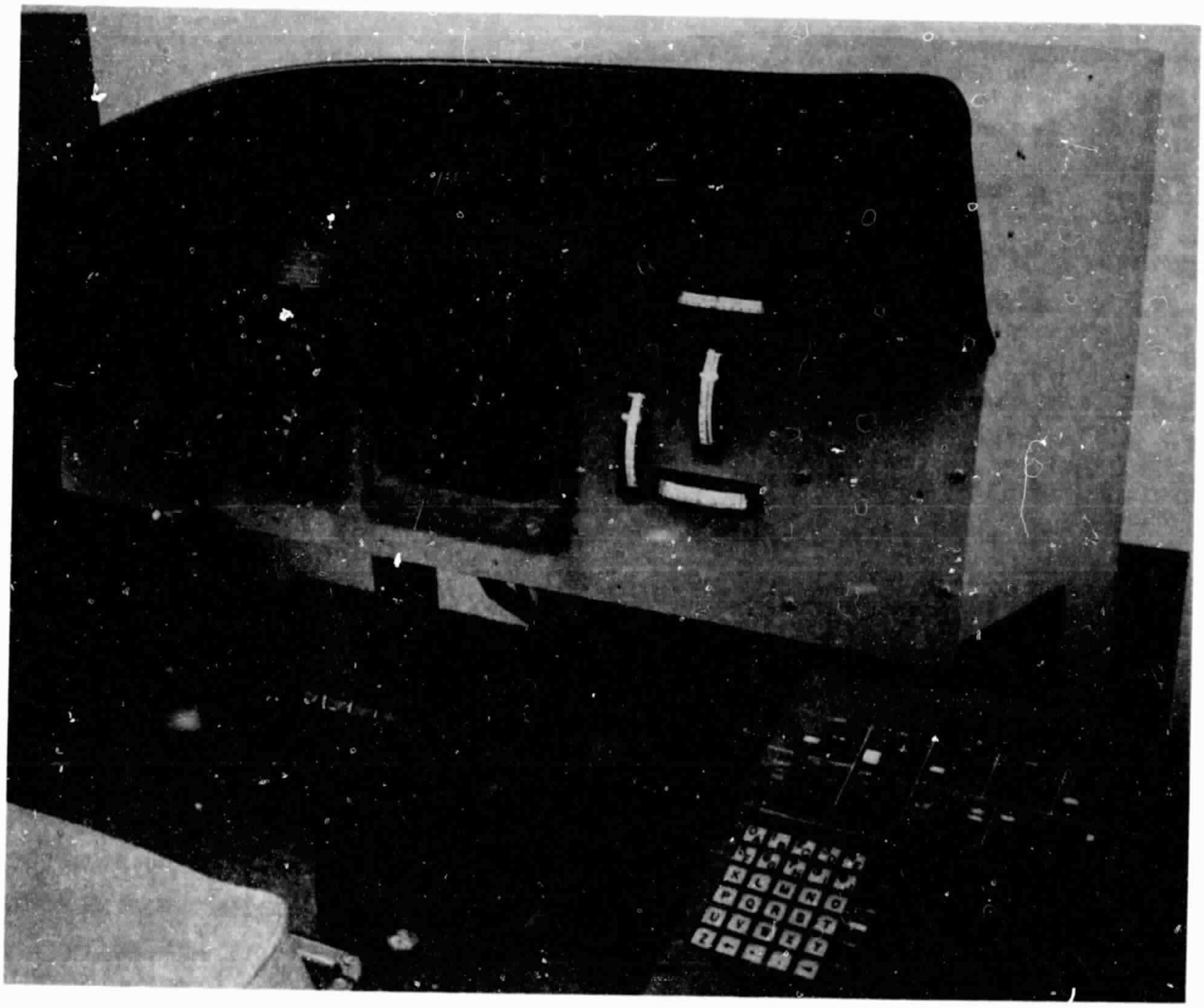
Figure 6-3
Sperry V/STOLAND Simulation Facility



4-12799-6

Figure 6-4
Sperry V/STOLAND Simulation Cab

Meters shown on the right side of the instrument panels also indicate the positions of the four basic control actuators - roll (ailerons and differential collective), pitch (elevator and cyclic pitch), yaw (rudder and differential cyclic pitch), and power lever (power and collective pitch). They are part of the Sperry simulation environment only (no equivalent on the aircraft).

The cab included the following controls and switches:

- Control stick for pitch and roll
- Power lever stick
- Pedals
- A/P disconnect switch (on control stick)
- Trim release switch (on control stick)
- Go-Around switch (on control stick)
- Pylon conversion switch (on power lever)
- RPM command switch (on power lever)
- Flaps command switches (on center console)
- Hold/Operate/Reset switches for the simulation computation (on center console)

The control stick, power lever and pedals were outfitted with LVDT and synchro position sensors, bungees and magnetic brakes, as in the aircrft installation. However, the servos were simulated in the simulation computer. The above switches and sensors were interfaced to the Simulation computer via the Airborne Hardware Simulator (AHS) described in Paragraph 4.15.1 and illustrated in Figure 4-60. The AHS also provides digital-to-analog (D/A) conversion for two 8-channel strip-chart recorders, driven by software in the Simulation computer which multiplexes two signals on each channel, yielding 32 possible variables recorded simultaneously on the strip charts.

The Simulation computer was a laboratory version of the 1819B (not packaged for airborne use) with 32K memory. The simulation software includes the following functions:

- Nonlinear, six-degree-of-freedom simulation of the XV-15 aircraft motion based on Bell Helicopter's mathematical models, including the effects of pylon conversion (with associated control mixers), flaps, landing gear, and associated actuators.
- Simulation of the Calspan SCAS and Force-Feel systems (described in Paragraph 4.2).
- Simulations of the large complement of sensors, including the navaid signals from the VOR, DME, TACAN and MLS receivers, static and dynamic pressure sensors, accelerometers, rate gyros and directional gyros. Also included, noise models for the navaid signals.
- Wind and turbulence models.
- Provisions for convenient selection of aircraft initial conditions and trimming.
- Strip-chart recording routines to facilitate selection and scaling of variables to be recorded from simulation or airborne computer data.
- Utility software for peripherals handling and for inspection and modification of computer memory data via the CRT.

The basic cycle time for the simulation computations is 50 milliseconds, with some of the less time-critical functions cycling at 100 milliseconds.

The airborne flight-rack equipment was mounted in the Simulation Bench as illustrated in Figure 6-5. This bench included a blower system that supplied air to the trays for the air-cooled boxes. The bench also provided convenient breakout of all cabling contacts through a terminal panel, as shown in the figure. The following units were mounted in the Simulation Bench:

- Basic Computer
- Research Computer
- Data Adapter
- MFD Symbol Generator
- Instrument Amplifier Rack

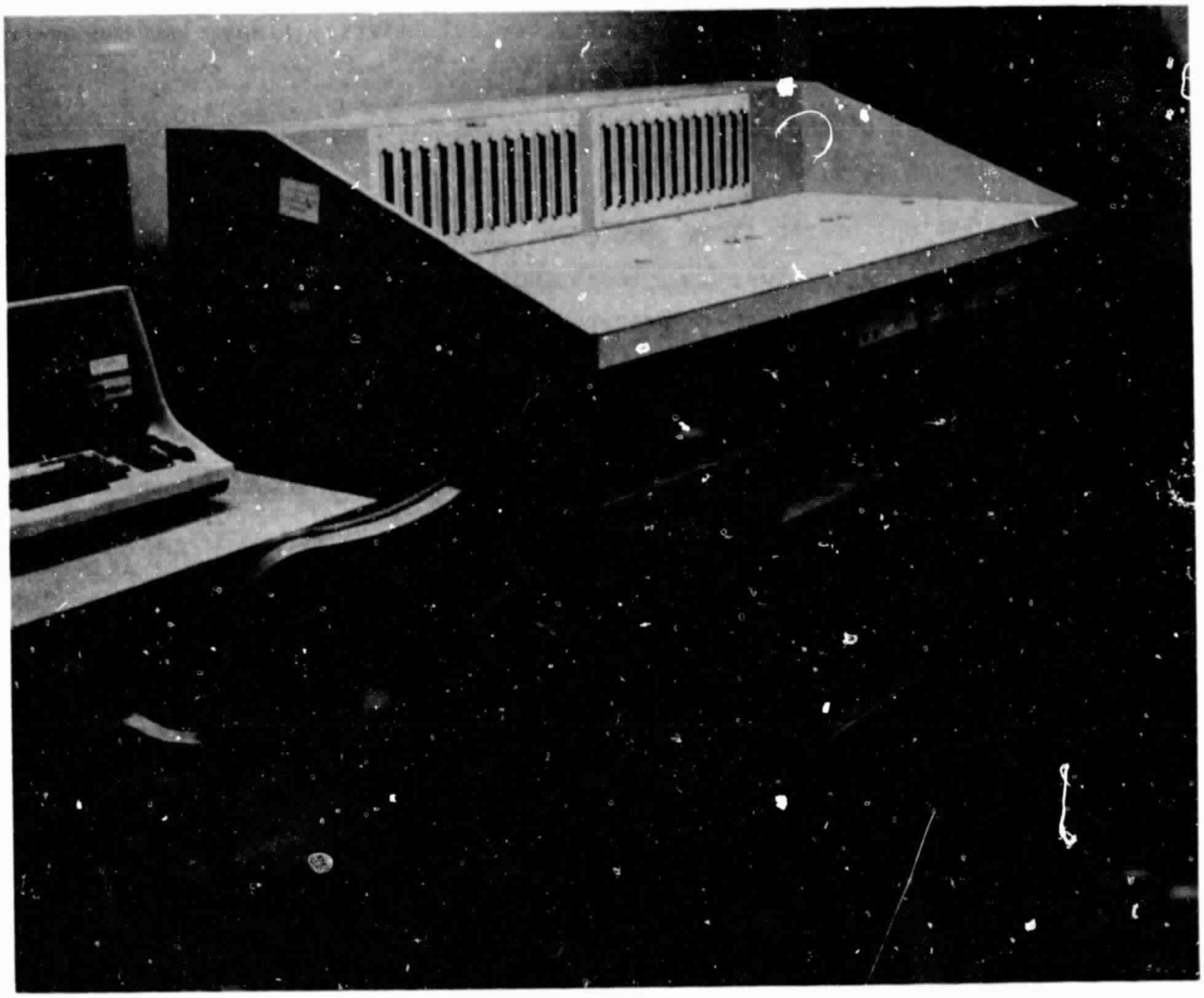


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Figure 6-5
XV-15 Simulation Bench

This bench later became part of the simulation facility at NASA/ARC as described in the next paragraph.

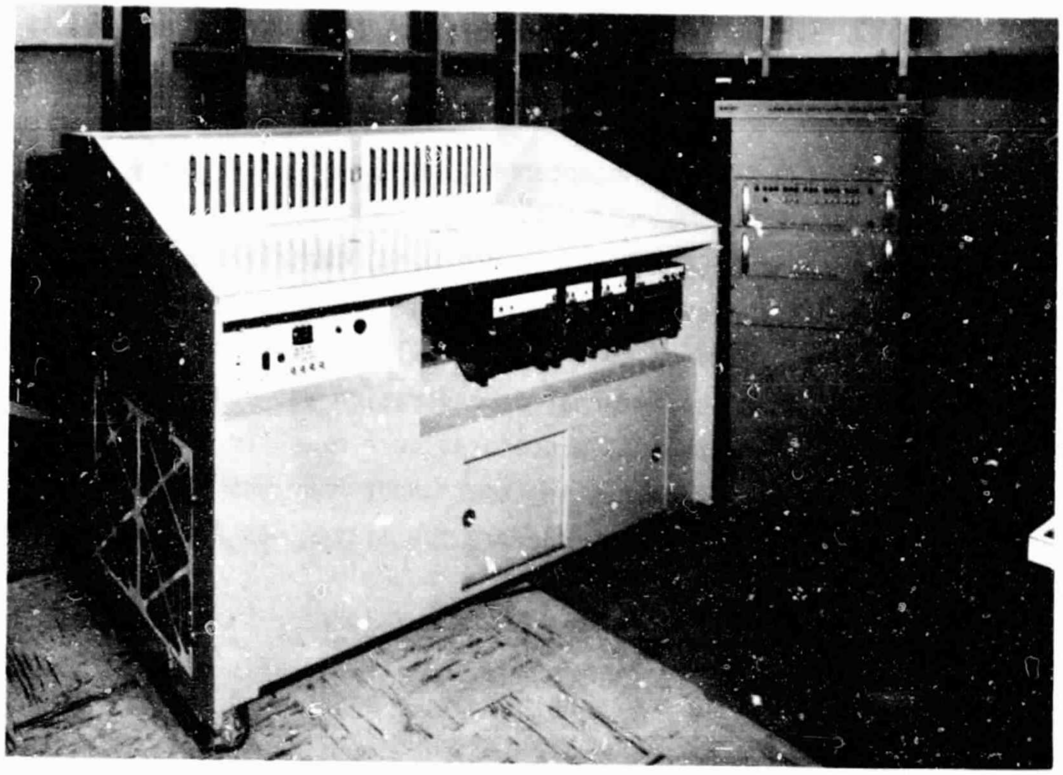
A comprehensive Dynamic Acceptance Test (DAT) Procedures document was written, with over 600 numbered tests to be officially signed off under the acceptance testing at NASA/ARC. This document, described in Paragraph 6.5, also served as the basis for system validation at Sperry before shipment. After the considerable effort required to wring out all problems and optimize performance in this complex system, an informal demonstration was conducted at Sperry, with NASA test pilots, where the DAT procedures were executed and performance was verified. The demonstration testing took about four weeks, two of which were conducted with NASA test pilots. Before the system was shipped to NASA/ARC, it had thus been thoroughly validated.

6.5 DYNAMIC ACCEPTANCE TESTING AT NASA

6.5.1 Facility Description

The S-19 simulation test facility at NASA/Ames Research Center is shared with the previous UH-1H V/STOLAND system, and includes equipment for both systems. The cab is constructed from a former UH-1 helicopter, and is complete with hydraulically driven series and booster servos, with linkages all the way to the swashplate. The facility includes the XV-15 simulation bench described in the previous paragraph, and shown with the Airborne Hardware Simulator in Figure 6-6a. Figure 6.6b shows the terminals and computer control panels that provide access to the Basic and Research computers, respectively. Proper interconnection of all units was verified by running the Static Acceptance Test described in Paragraph 6.3.

The simulation computations were done on the nearby EAI 8400 computer facility, part of which is shown in Figure 6-7.



(a) A80 0101 3

(b) A80 0101 4

Figure 6-6
S-19 Simulation Facility Equipment at NASA/ARC

A80 0101 2

Figure 6-7
Part of NASA's EAI 8400 Computer Facility

6.5.2 Test Procedures

The Dynamic Acceptance Test was organized into the following 35 groups of tests:

A Preflight Test

B Basic Executive

C Digital Data Acquisition System

D Data Entry Keyboard

E Failure Monitoring

F Mode Select Panel

G Horizontal Situation Indicator

H Multifunction Display

I Navigation Without Noise

J Navigation With Noise

K Basic A/P, Transient Responses

L Basic A/P, V-L Guidance and Control, No Wind, No Navigation Noise

M Basic A/P, V-L Guidance and Control, With Crosswinds Plus Turbulence, With Navigation Noise

N Basic A/P, L-D Guidance and Control, No Wind, No Navigation Noise

P Basic A/P, L-D Guidance and Control, No Wind, With Navigation Noise

Q Basic A/P, L-D Guidance and Control, With Constant Wind, With Navigation Noise

R Basic A/P, L-D Guidance and Control, With Constant Plus Turbulent Winds, With Navigation Noise

S Basic A/P, LAND-1 Guidance and Control, No Wind, No Navigation Noise

T Basic A/P, LAND-1 Guidance and Control, No Wind, With Navigation Noise

U Basic A/P, LAND-1 Guidance and Control, With Constant Head Wind, With Navigation Noise

V Basic A/P, LAND-1 Guidance and Control, With Constant Crosswind, With Navigation Noise

W Basic A/P, LAND-1 Guidance and Control, With Crosswind Plus Turbulence, With Navigation Noise

X Basic A/P, LAND-2 Guidance and Control, No Wind, No Navigation Noise

Y Basic A/P, LAND-2 Guidance and Control, With Constant Wind, With Navigation Noise

Z Basic A/P, LAND-2 Guidance and Control, With Constant Plus Turbulent Winds, With Navigation Noise

AA Flight Director, V-L Guidance and Control, No Wind, No Navigation Noise

BB Flight Director, V-L Guidance and Control, With Constant Plus Turbulent Winds, With Navigation Noise

CC Flight Director, L-D Guidance and Control, No Wind, No Navigation Noise

DD Flight Director, L-D Guidance and Control, With Constant Plus Turbulent Winds, No Navigation Noise

EE Flight Director, LAND-1 Guidance and Control, No Wind, No Navigation Noise

FF Flight Director, LAND-1 Guidance and Control, With Constant Plus Turbulent Winds, No Navigation Noise

GG Flight Director, LAND-2 Guidance and Control, No Wind, No Navigation Noise

HH Flight Director, LAND-2 Guidance and Control, With Constant Plus Turbulent Winds, With Navigation Noise

II Off-Nominal Tests

JJ Research Modes

Fourteen different initial conditions (as listed in Table 6-1) were set up in the aircraft simulation program which were called for in the test procedures by number (IC = 0 ... 13). Figure 6-8 illustrates the initial aircraft locations and headings near the Crows Landing runway and the Reference Flight Path. Not shown are IC 0 which is on the runway for the preflight test, and IC 9 and 10 which are identical to 1 and 6, respectively, except that they are translated to have the same relative position to the VOR station at Stockton that IC 1 and 6 have to the TAC station at Crows Landing. Also not shown is IC 13, located on the runway adjacent to the TACAN station and used for the Off-Nominal tests.

The IC positions were chosen to provide approximately 3 minutes before lateral captures occur. This is to allow the navigation filters sufficient time to wash out the errors introduced when initialization is done in wind conditions. Other initial conditions that apply to all test cases are:

- Flight Path Angle = 0 degree
- Gross Weight = 13,000 pounds
- Center of Gravity = Station 26.1
- Standard Atmosphere (Baro Set = 29.92 in. Hg)

6.5.3 Disturbance Models

Some of the test groups were performed under the disturbance conditions of navigation noise, constant winds and/or turbulent winds. The associated disturbance models programmed in the simulation computer are described in the following paragraphs.

6.5.3.1 Navigation Noise

The navigation noise disturbance includes Gaussian and drop-out models for TACAN, VOR and MLS signals, and Gaussian models for baro and radio altitudes. The Gaussian noise for the TACAN, VOR and MLS signals is composed of a

TABLE 6-1

INITIAL CONDITIONS FOR AIRCRAFT SIMULATION

	0	1	2	3	4	5	6	7	8	9	10	11	12	13
Air Speed (knots)	0	140	140	5	50	100	140	250	140	140	140	60	5	25
Heading (deg)	353	130	353	353	353	353	353	353	40	135	353	130	353	353
x (feet)	0	30,000	-10,000	-10,000	-20,000	-30,000	-45,000	-75,000	-90,000	166,406	104,406	15,000	-45,000	-2,570
y (feet)	0	-40,000	-15,000	-15,000	-15,000	-15,000	-15,000	-15,000	-30,500	-64,446	-59,446	-22,320	-15,000	0
h (feet above runway)	0	3,000	400	3,000	3,000	3,000	3,000	3,000	3,000	3,000	3,000	3,000	3,000	20
Pylon Angle (deg)	90	0	0	90	90	75	0	0	0	0	0	87	90	90
Flap Angle (deg)	0	40	40	75	75	40	40	0	40	40	40	75	75	75
Rotor RPM (percent)	0	76	76	94	94	94	76	76	76	76	76	94	94	94

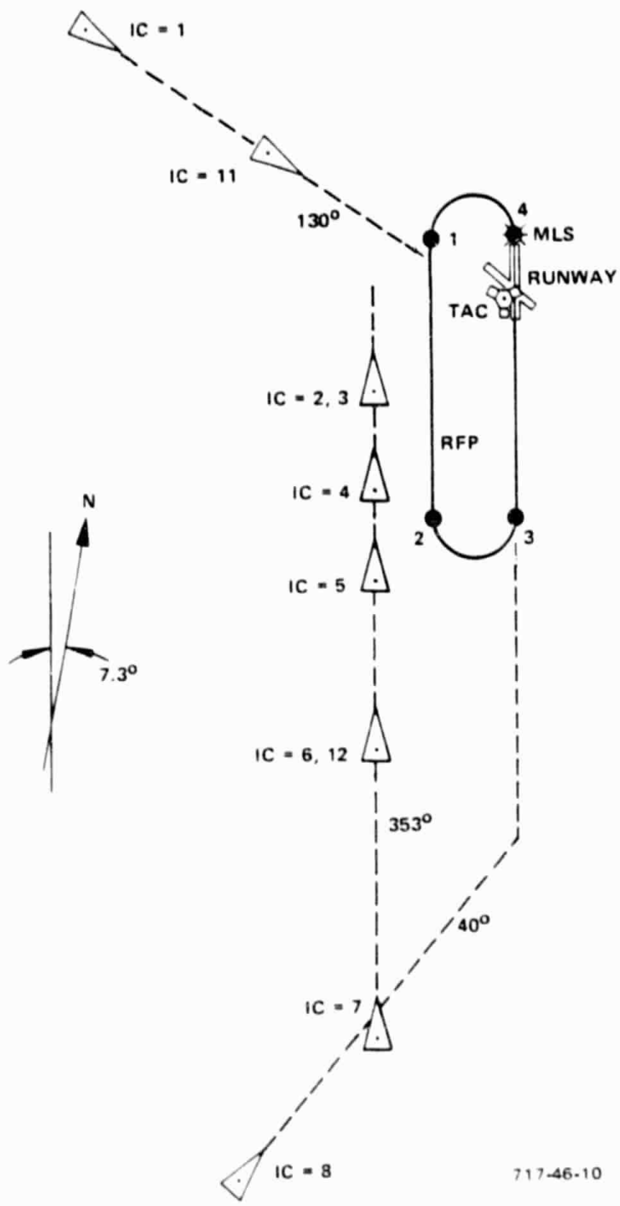


Figure 6-8
IC Locations and Headings

"bias" and a "random" component which is summed to produce the total Gaussian noise. The general expression for computing each Gaussian noise component is

$$N_i(t) = e^{-\Delta T/\tau_i} N_i (t - \Delta T) + \sigma_i \sqrt{1 - e^{-2\Delta T/\tau_i}}\, u_i(t)$$

where

i = b or r, denoting bias or random component

ΔT = computation cycle time (nominally .05 sec)

$u_i(t)$ = a sequence of statistically independent samples from a standard normal population ($\mu = 0$, $\sigma = 1$), taken each computation cycle.

The parameters, σ_i and τ_i are listed in Table 6-2 for each navigation signal.

In addition to the Gaussian noise, the TACAN, VOR and MLS signals contain a drop-out noise component which is modeled as periodic with period t_{cycle}. At time t_1 after the start of each cycle, a quantity Δ is added to each navigation signal until the end of the cycle. At time t_2, after the start of each cycle, the valid discrete is dropped until the end of the cycle. The parameters are specified in Table 6-2.

TABLE 6-2

NAVIGATION NOISE PARAMETERS FOR TACAN, VOR AND MLS

Navigation Signal	σ_b	τ_b	σ_r	τ_r	t_1	t_2	t_{cycle}	Δ
TACAN Bearing	.762 deg	500 s	.59 deg	1 s	30 s	32.5 s	35 s	36 deg
TACAN Range	850 ft	500 s	12 ft	1 s	40 s	44 s	48 s	10^5 ft
VOR Bearing	.35 deg	500 s	.3 deg	1 s	30 s	32.5 s	35 s	36 deg
VOR Range (DME)	850 ft	500 s	12 ft	1 s	40 s	44 s	48 s	10^5 ft
MLS Azimuth	.094 deg	500 s	.065 deg	.5 s	43 s	43.25 s	43.5 s	15 deg
MLS Range	40 ft	500 s	40 ft	.5 s	33 s	37 s	39 s	*
MLS Elevation	.038 deg	500 s	.055 deg	.5 s	35 s	35.25 s	35.5 s	15 deg

*If the true range is greater than 8000 ft, set the range signal to zero (i.e., $\Delta = -R_{MLS}$) when $t_1 < t < t_{cycle}$; $\Delta = 0$ otherwise.

The barometric altimeter signal has errors due to a nonstandard atmosphere (NSA) and to pitot-static (PS) errors. The model for these errors is given by

$$\Delta H_B = \sigma_{NSA} u_1 + \sigma_{PS} u_2$$

where

$\sigma_{NSA} = .02H_B$ (H_B = true barometric altitude in feet)

$\sigma_{PS} = (\frac{1}{599}) \frac{V^2}{2g}$ (V = true air speed in ft/s)

u_1 and u_2 = statistically independent random samples from a standard normal population ($\mu = 0$, $\sigma = 1$), taken at program initialization (and then held constant).

The sensed barometric altitude is then given by

$$H_B' = H_B + \Delta H_B \text{ ft}$$

The radio altimeter signal has a bias error component and a Gaussian error component. The error model is given by

$$\Delta H_R = \begin{cases} |1.3\ u_B + .25\ u_R(t)| & \text{for } H_R \leq 100 \text{ ft} \\ .013\ H_R u_B + .003\ H_R u_R(t) & \text{for } H_R > 100 \text{ ft} \end{cases}$$

where

H_R = true radio altitude in feet

u_B = a sample from a standard normal population ($\mu = 0$, $\sigma = 1$), taken at program initialization,

$u_R(t)$ = a sequence of statistically independent samples from a standard normal population, taken each computation cycle.

The sensed radio altitude is then given by

$$H_R' = H_R + \Delta H_R \text{ ft}$$

6.5.3.2 Constant Wind

The constant wind disturbance is constant as a function of time but varies as a function of altitude and heading as follows. The north and east components of the wind velocity are given by

$$V_{WN} = - V_o \cos(\theta_{RWY} + A_{MW}) \text{ ft/s}$$

$$V_{WE} = - V_o \sin(\theta_{RWY} + A_{MW}) \text{ ft/s}$$

where

$$V_o = \begin{cases} C_W e^{-h/10,000} \dfrac{D \log_{10} h + E}{D + E}, & h > 10 \text{ ft} \\ C_W e^{-10/10,000} \left(\dfrac{h}{10}\right), & h \leq 10 \text{ ft} \end{cases}$$

θ_{RWY} = magnetic heading of the runway = -7.3 deg

A_{MW} = direction from which the wind is blowing with respect to the runway

h = aircraft height above ground, ft

$C_W = A + B \cos A_{MW} + C \cos^2 A_{MW}$

A = 25.3161 ft/s (15 kt)

B = 12.6585 ft/s (7.5 kt)

C = 4.2195 ft/s (2.5 kt)

D = .43

E = .35

This model represents a 3σ wind magnitude. At 10 feet above ground, a head wind (A_{MW} = 0) is at 25 knots, a crosswind (A_{MW} = 90 or 270 deg) is at 15 knots and a tail wind (A_{MW} = 180 deg) is at 10 knots.

6.5.3.3 Turbulence

The turbulence model is based on Military Specification MIL-F-8785B (ASG), and assumes the Dryden spectral form for the turbulence. This model and its implementation on the 8400 computer is described in "Modeling Turbulence for Flight Simulations at NASA-Ames" by Benton L. Parris, January 1975, CSCR No. 4. Figure 6-9 summarizes the computation model for the six turbulence components to be added to the associated aircraft body-axis linear and angular velocities where

$u_i(t)$ = a sequence of independent samples from a standard nominal population, taken each computation cycle

ΔT = computation cycle time

V = true airspeed, ft/s

b = wing span = 32.17 ft

L_u = scale length associated with the u-axis (a parameter of the Dryden spectral form)

$$= \begin{cases} 1750 \text{ ft}, & h \geq 1750 \text{ ft} \\ 145\, h^{1/3} \text{ ft}, & 100 \leq h < 1750 \text{ ft} \\ 673.03 \text{ ft}, & h < 100 \text{ ft} \end{cases}$$

$L_v = L_u$

$$L_w = \begin{cases} 1750 \text{ ft}, & h \geq 1750 \\ h \text{ ft}, & 100 \leq h < 1750 \text{ ft} \\ 100 \text{ ft}, & h < 100 \text{ ft} \end{cases}$$

h = height of aircraft cg above the runway, ft

${}_{w}$ = 6.8 ft/s

$$\sigma_u = \sqrt{\frac{Lu}{Lw}}\,\sigma_w \text{ ft/s}$$

$$\sigma_v = \sigma_u \text{ ft/s}$$

The $\sqrt{\pi/\Delta T}$ term at each input serves to normalize the input sequence to produce unit spectral-density white noise.

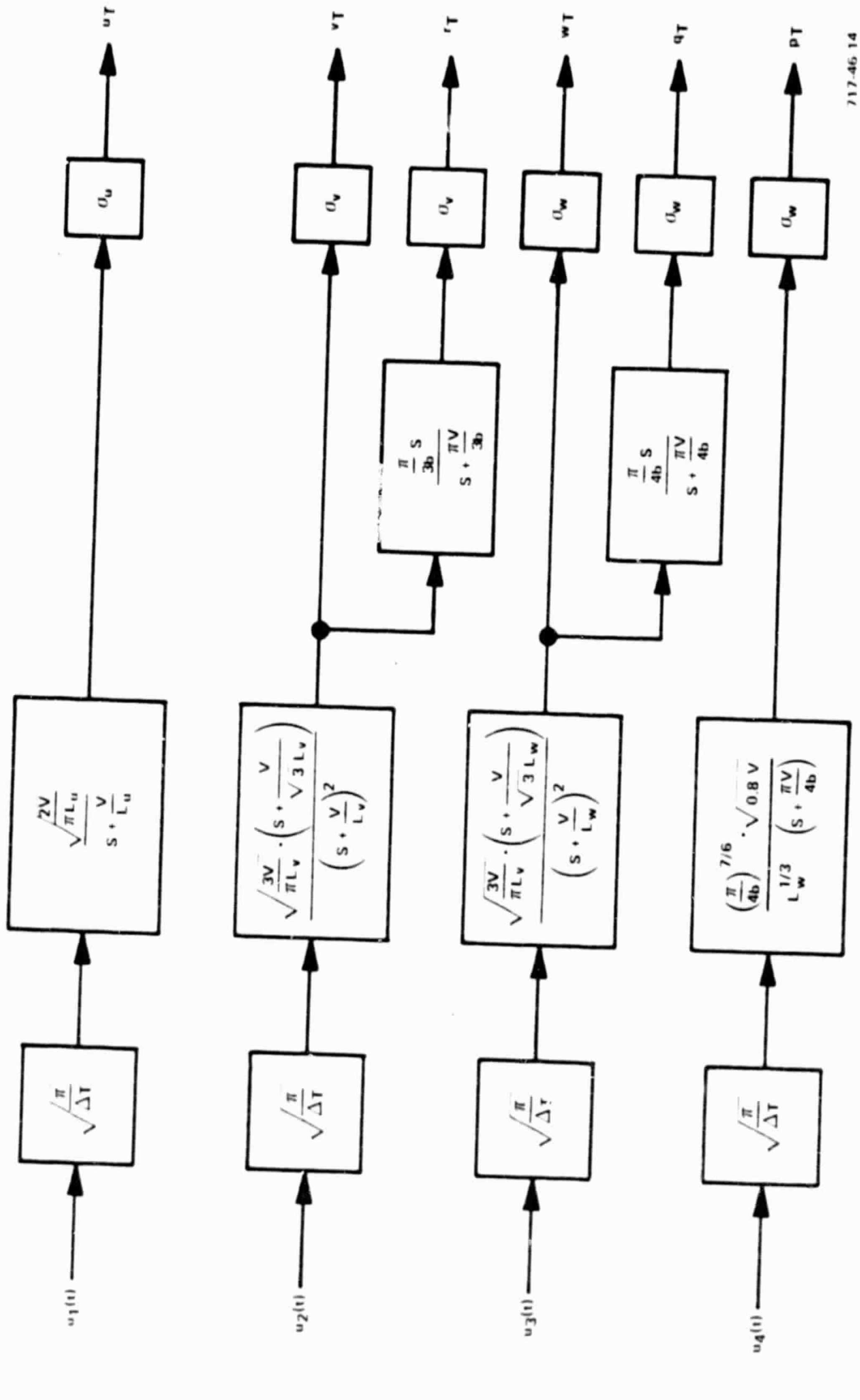


Figure 6-9
Turbulence Model

6.6 SELECTED TEST RESULTS

This paragraph presents some of the strip-chart recordings obtained in the Dynamic Acceptance Test at NASA/ARC in February 1978. Two 8-channel recorders were used, and each channel was multiplexed to record two variables, resulting in 32 variables recorded. The following figures include six or eight of those strip-chart channels, selected as appropriate for the respective tests. Table 6-3 lists the variables that are plotted in Figures 6-10 through 6-30.

The two variables recorded on a single channel are distinguishable by the relative duration of the sampling of each. The variable with the longer duration is referred to by the letter "L", and the one with the short duration by "S". "Spikes" with full-scale amplitude occasionally occur in the following graphs. These are due to a problem with the digital-to-analog converters used for the strip-chart recordings, and should not be confused with the recorded data.

The following paragraphs describe selected test sequences that illustrate the performance of the system.

6.6.1 Airspeed Select

The first sequence consists of doing an airspeed-select to 225 knots, starting from the following initial conditions:

- V_{IAS} = 5 knots
- h = 3000 feet
- β_p = 90 degrees
- Basic A/P engaged
- ALT HLD engaged
- IAS HLD engaged
- Pylons locked
- No disturbances

TABLE 6-3

VARIABLES PLOTTED IN FIGURES 6-10 THROUGH 6-30

θ_C	Pitch Attitude Command	±25 deg
θ	Pitch Attitude (sensed)	±25 deg
ϕ_C	Roll Attitude Command	±25 deg
ϕ	Roll Attitude (sensed)	±25 deg
p	Roll Rate (sensed)	±12.5 deg/s
q	Pitch Rate (sensed)	±12.5 deg/s
ψ_C	Heading Command	±250 deg
ψ	Heading (sensed)	±250 deg
ψ_E	Heading Error (ψ_C-ψ)	±12.5 deg
r	Yaw Rate (sensed)	±12.5 deg/s
h	Height Above Runway	0 to 2500 ft
h_E	Height Error, Engaged Mode (h_{CF}-h)	±50 ft
$\dot{h}_{CF}$	Vertical Rate Command, Filtered	±50 ft/s
V_{CF}	Velocity Command, Filtered	0 to 500 ft/s
V_{IAS}	Indicated Air Speed	0 to 500 ft/s
V_G	Ground Speed	0 to 500 ft/s
V_E	Velocity Error (V_{CF} - V_{IAS} or V_{CF} - $\dot{D}_F$)	±50 ft/s
D_Y	Cross-Track Displacement, Engaged Mode	±250 ft
$\dot{D}_Y$	Cross-Track Rate, Engaged Mode	±100 ft/s
δ_{PLC}	Power Lever Command	±5 in.
δ_{PL}	Power Lever Position	±5 in
θFC	Command to Pitch FFS	±5 in.
ϕFC	Command to Roll FFS	±5 in.
ψFC	Command to Yaw FFS	±2.5 in.
β_p	Pylon Angle	0 to 125°

The action begins by selecting an airspeed of 225 knots on the Mode Select Panel and then pushing the IAS HLD/SEL button. The following sequence of events will then take place:

- IAS SEL engages and the aircraft accelerates at approximately 1.5 ft/s^2.
- The pylons immediately start conversion to 85 degrees and the upper FMA displays PYLONS TO 85°.
- When the conversion is completed, the upper FMA reverts to displaying BAS AP, TILT.
- At 25 knots the flaps start moving to 40 degrees and the upper FMA displays FLAPS TO 40°. (When completed, the upper FMA reverts to the previous display.)
- At 50 knots the pylons start conversion to 75 degrees and the FMA displays PYLONS TO 75°. (When completed, the upper FMA reverts to the previous display.)
- At 100 knots the pylons start conversion to 0 degree and the upper FMA displays PYLONS TO 0°.
- When the conversion is completed, the upper FMA reverts to displaying BAS AP, AIRPL.
- At 140 knots the rotor RPM starts decreasing to 76 percent and the upper FMA displays RPM TO 76%. (When completed, the upper FMA reverts to the previous display.)
- At 165 knots the flaps start moving to 0 degree and the upper FMA displays FLAPS TO 0°. (When completed, the upper FMA reverts to the previous display.)
- At 190 knots the upper FMA flashes LOCK PYLONS. When the pylons have been locked by the pilot, the message reverts to the previous one.
- At 225 knots IAS HLD engages.

Figure 6-10 shows the profiles of selected variables during this sequence.

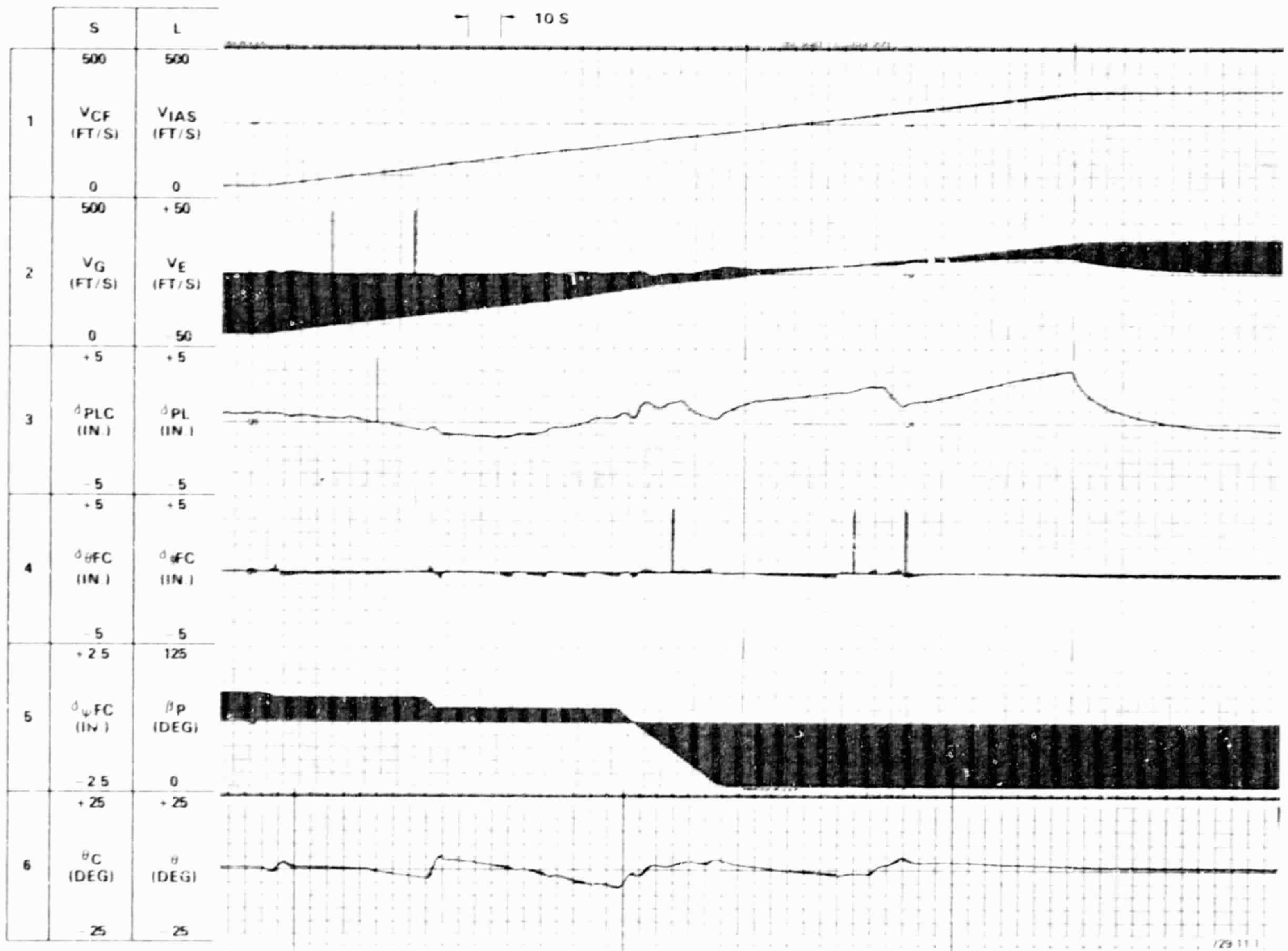


Figure 6-10
Airspeed-Select from 5 to 225 Knots,
No Disturbance

Next, an airspeed of 5 knots is selected and initiated. The following sequence then takes place:

- IAS SEL engages and the aircraft decelerates at approximately 1.5 ft/s (.89 kt/s). (The upper FMA messages for this test are similar to the previous test.)
- At 220 knots the flaps start moving to 20 degrees.
- At 160 knots the flaps start moving to 40 degrees.
- At 155 knots the RPM starts increasing to 94 percent.
- At 140 knots the upper FMA starts flashing UNLOCK PYLONS, and the pilot responds.
- The pylons then start conversion to 75 degrees.
- At 100 knots the pylons start conversion to 85 degrees.
- At 75 knots the pylons start conversion to 90 degrees.
- At 60 knots the flaps start moving to 75 degrees.
- At 5 knots IAS HLD engages.

Figure 6-11 gives the profiles for this seqeuence.

The above sequences are now repeated, but with all disturbance models in effect:

- Baro altitude noise as described in Paragraph 6.5.3.1
- 90-degree crosswind as described in Paragraph 6.5.3.2
- Turbulence as described in Paragraph 5.5.3.3

Figures 6-12 and 6-13 show the respective profiles.

6.6.2 Flight Path Angle Select

The next four figures illustrate Flight Path Angle Select performance at two speeds (helicopter mode and airplane mode, respectively) under both no disturbance and worst-case disturbance conditions. The first is at 50 knots, no disturbance, where a plus 6-degree FPA is first selected, followed by a zero-degree FPA selection, and then a minus 6-degree FPA selection. Figure 6-14

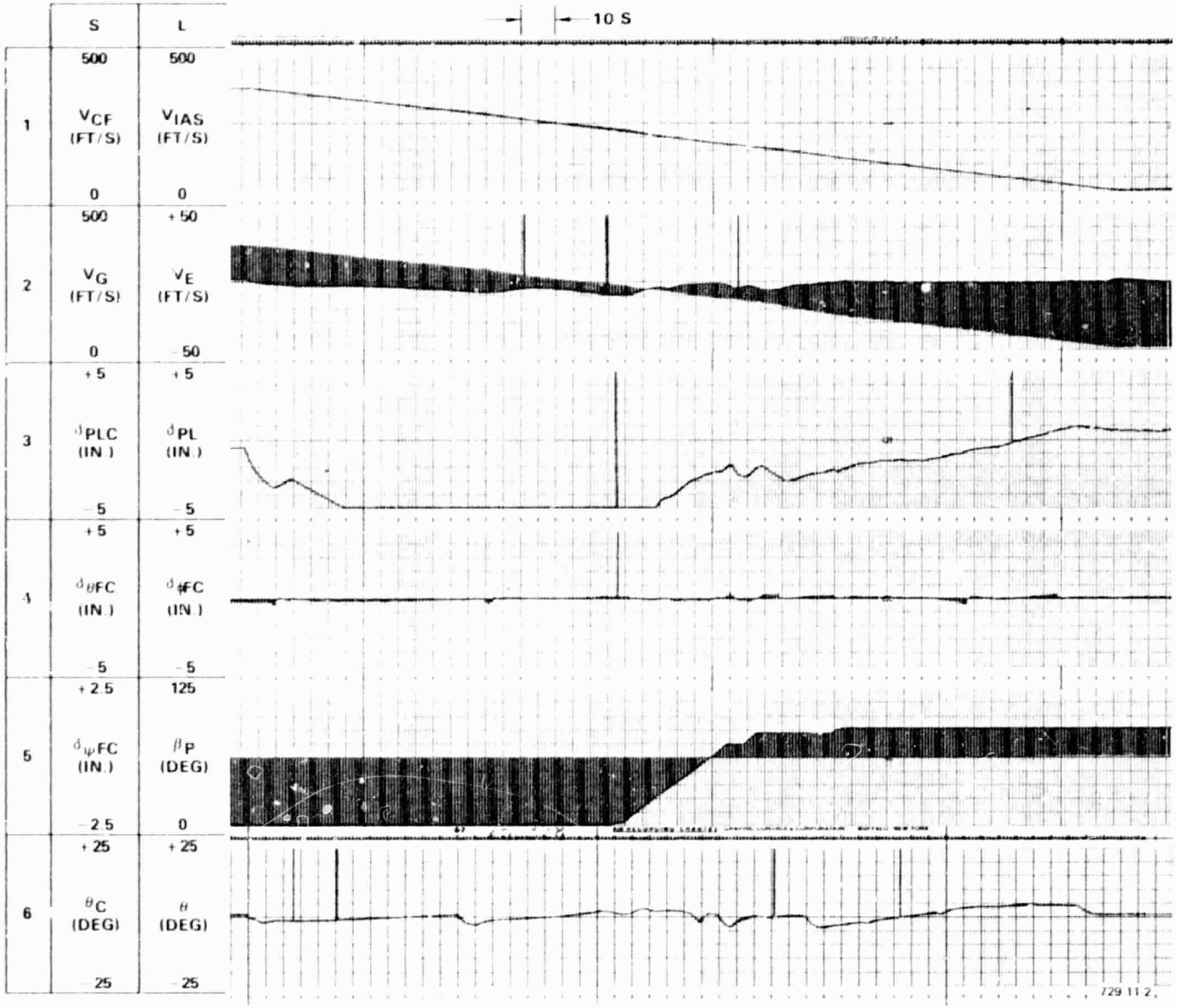


Figure 6-11
Airspeed-Select from 225 to 5 Knots,
No Distubance

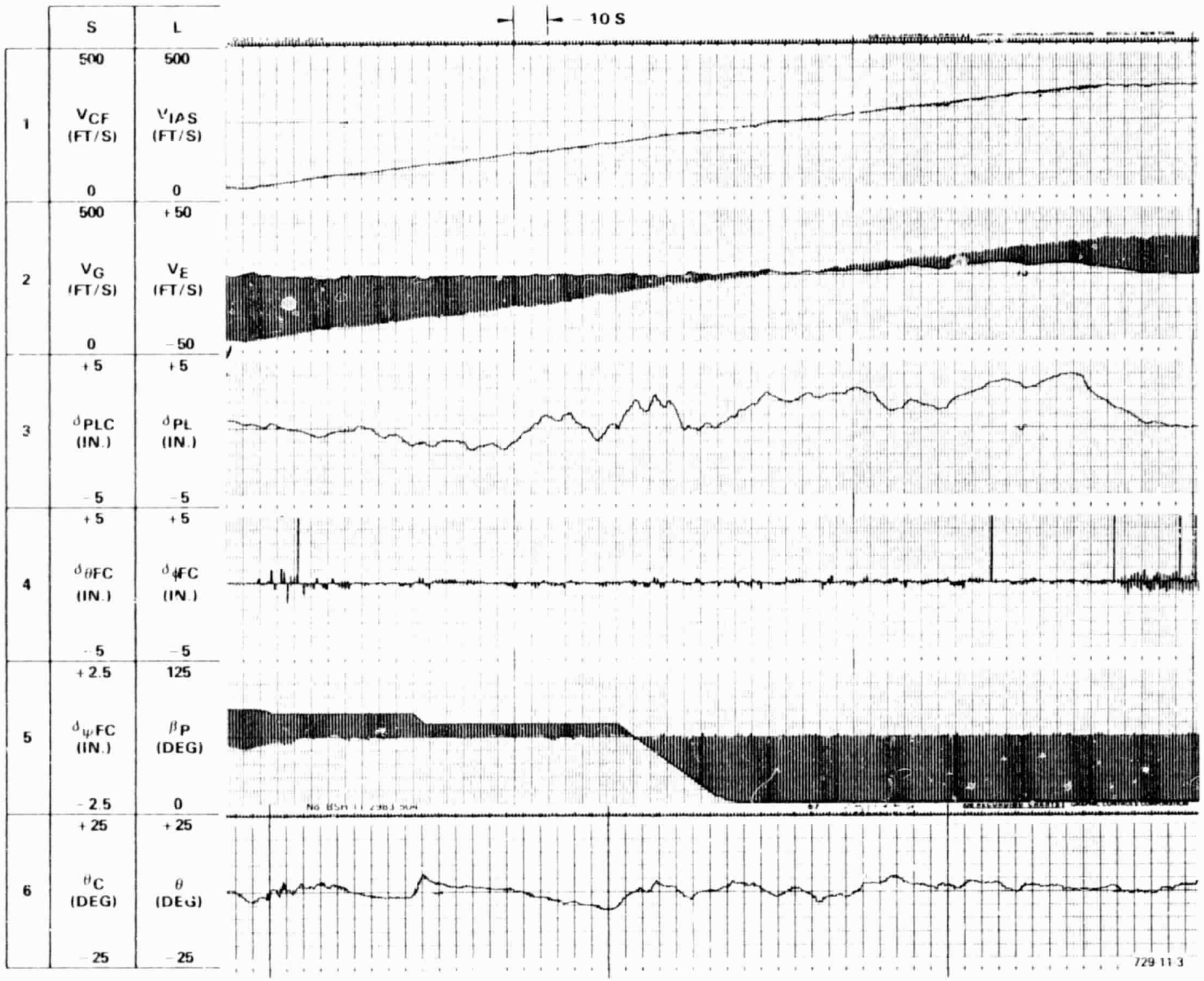


Figure 6-12
Airspeed-Select from 5 to 225 Knots, with Worst-Case Disturbances

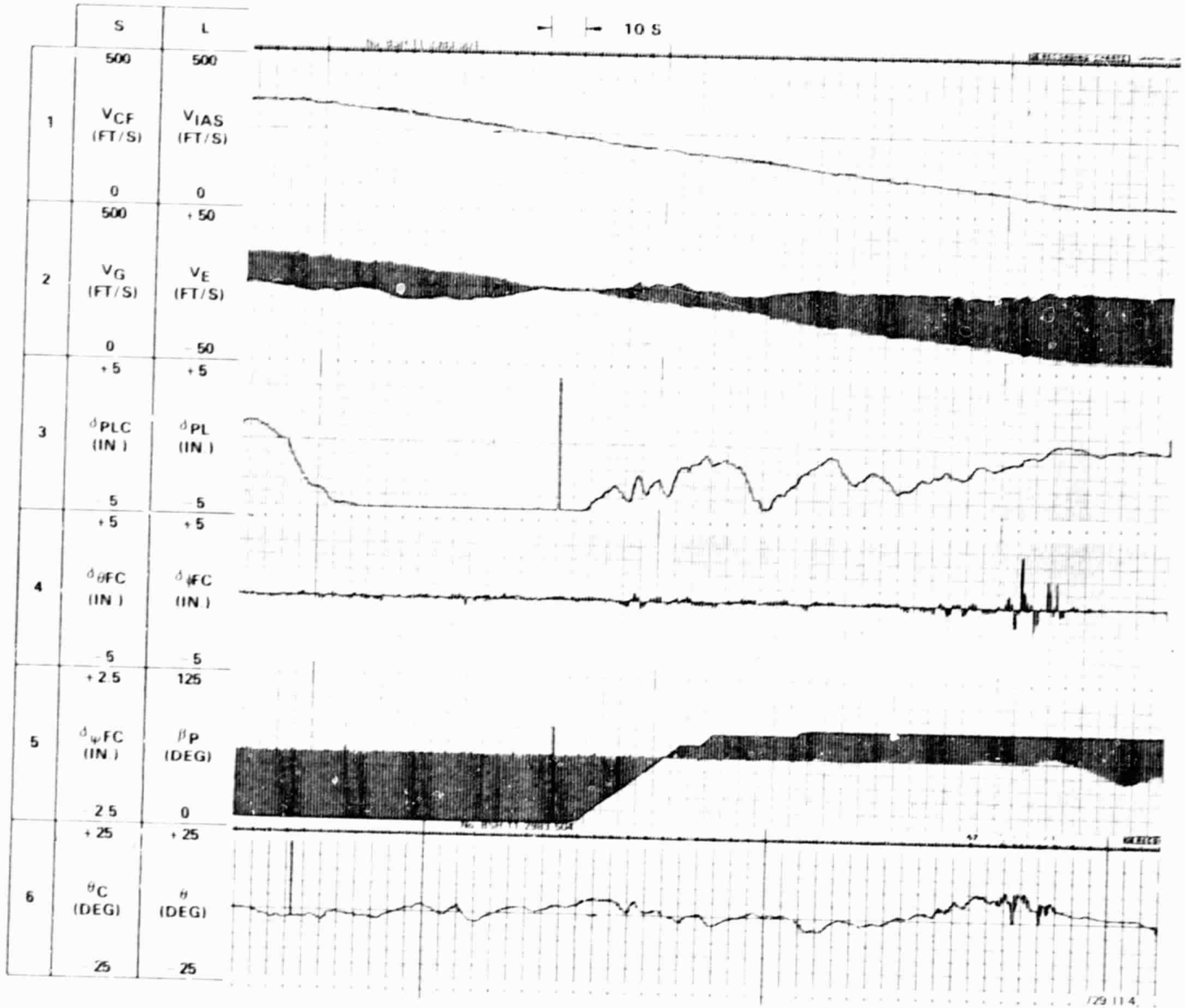


Figure 6-13
Airspeed-Select from 225 to 5 Knots,
with Worst-Case Disturbances

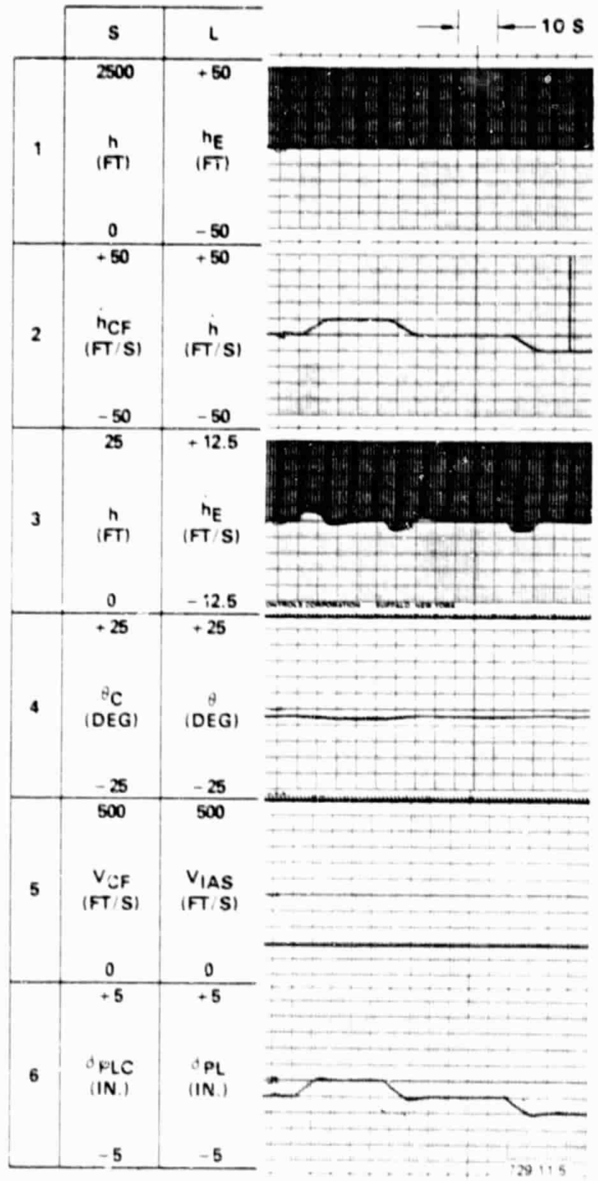


Figure 6-14
Flight-Path Angle-Select at 50 Knots,
No Disturbance

shows the relevant profiles. Figure 6-15 shows a similar sequence at 225 knots with a plus one, zero, and minus two-degree FPA select sequence. Figures 6-16 and 6-17 repeat the same sequences, but with the worst-case disturbances described above.

6.6.3 Altitude Select

The next six figures illustrate Altitude-Select performance at 50, 100, and 225 knots, without and with disturbances, respectively. For the first case, the initial altitude is 3000 feet above the runway, airspeed is 50 knots (helicopter mode), and a 4000-foot altitude is selected on the MSP. This action results in the following sequence:

- ALT SEL arms.
- FPA SEL engages at plus eight-degree (limit value).
- The lower FMA displays FPA, HDG, ALTA.
- When the FPA is captured, FPA HLD engages.
- When the vertical capture conditions are met:
  - ALT SEL engages.
  - FPA HLD disengages.
- The lower FMA displays ALT, HDG.
- When the vertical track conditions are met, ALT HLD engages.

Figure 6-18 shows the corresponding profiles (however, h is unfortunately above the recorder range).

Figure 6-19 shows a +500-foot Altitude Select, followed by a -500-foot Altitude Select, at 100 knots airspeed, where the aircraft is in the tilt mode with β_p = 75 degrees. Figure 6-20 shows a similar sequence at 225 knots (airplane mode), where a +1500-foot Altitude Select is followed by a -1500-foot Altitude Select.

Figures 6-21 through 6-23 are repetitions of the three previous sequences, but with the described worst-case disturbances applied.

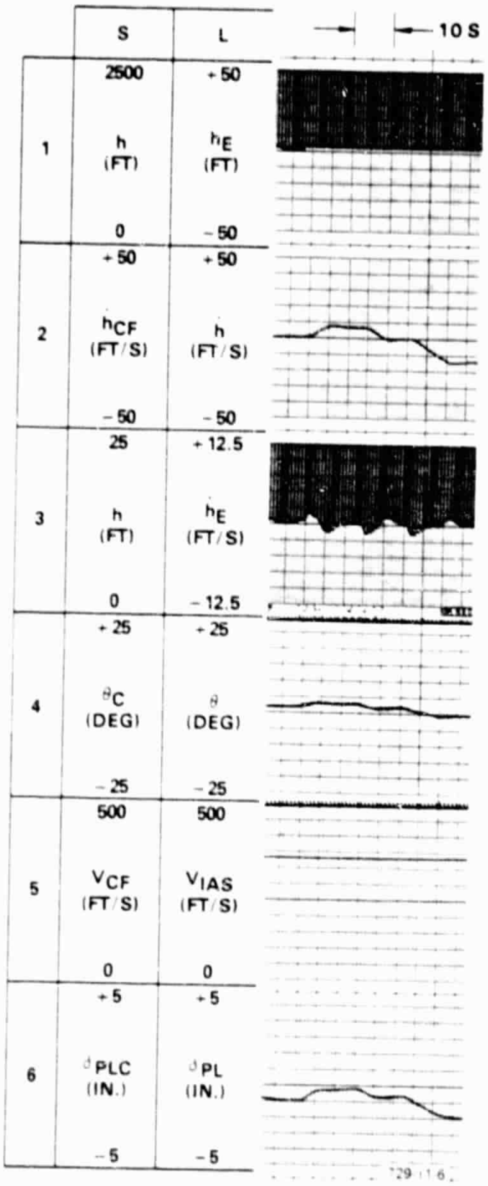


Figure 6-15
Flight-Path Angle-Select at 225 Knots,
No Disturbance

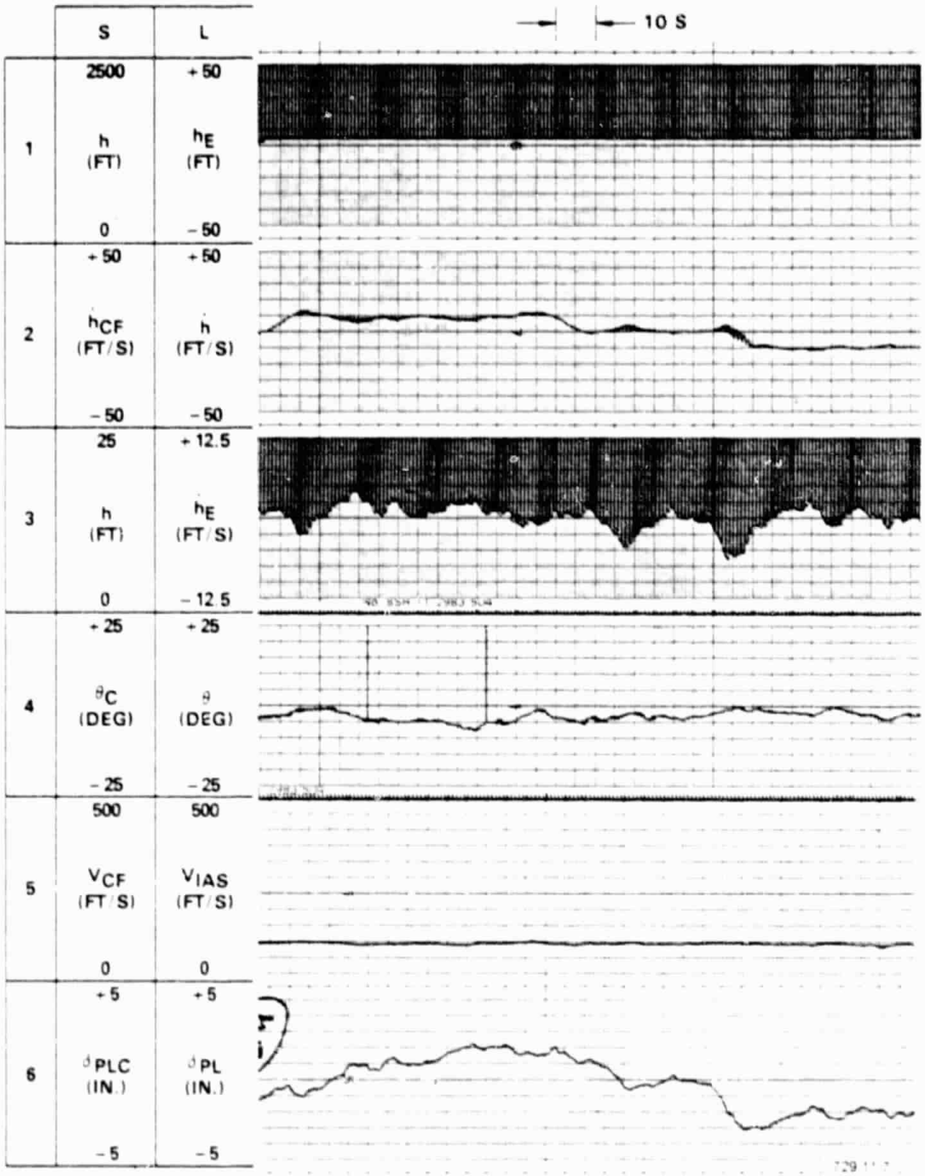


Figure 6-16
Flight-Path Angle-Select at 50 Knots,
Worst-Case Disturbances

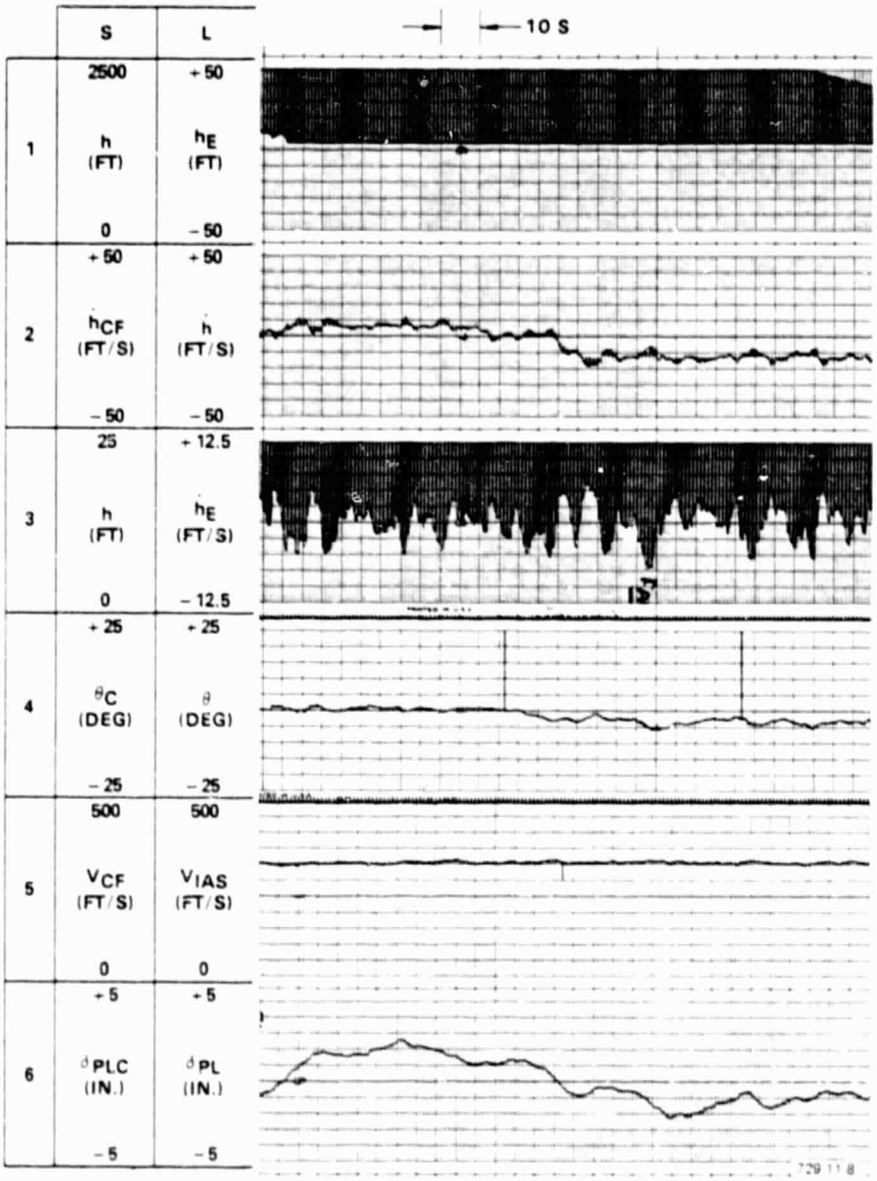


Figure 6-17
Flight-Path Angle-Select at 225 Knots,
Worst-Case Disturbances

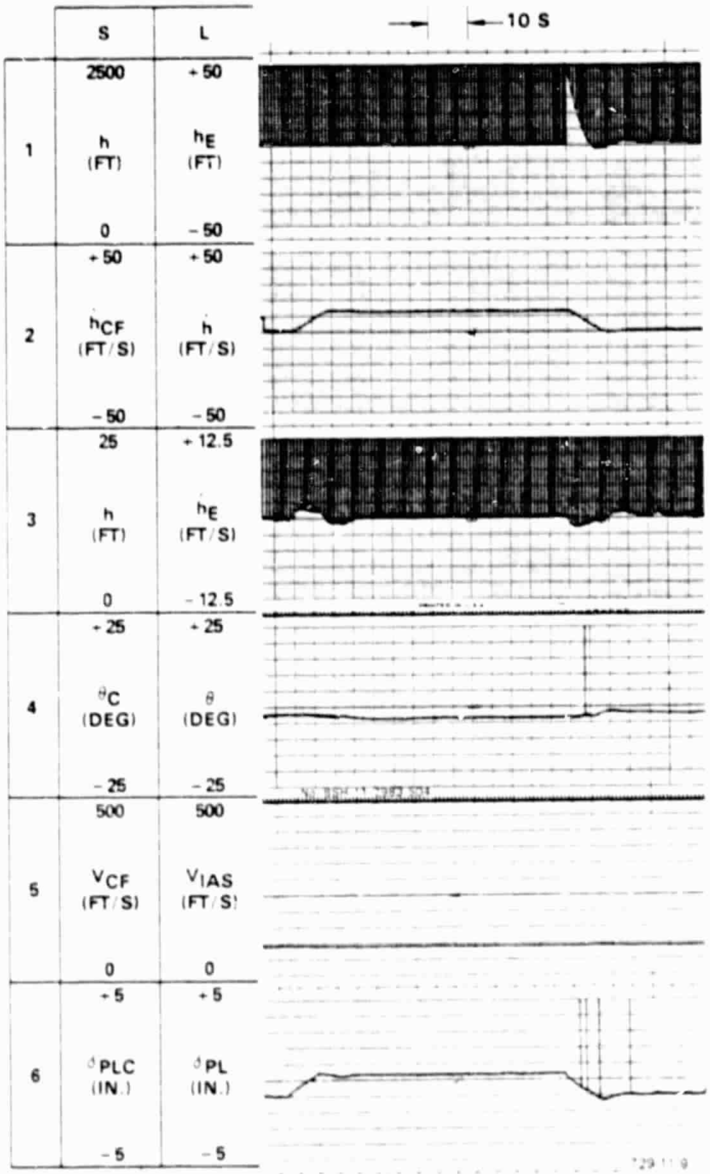


Figure 6-18
Altitude-Select at 50 Knots,
No Disturbance

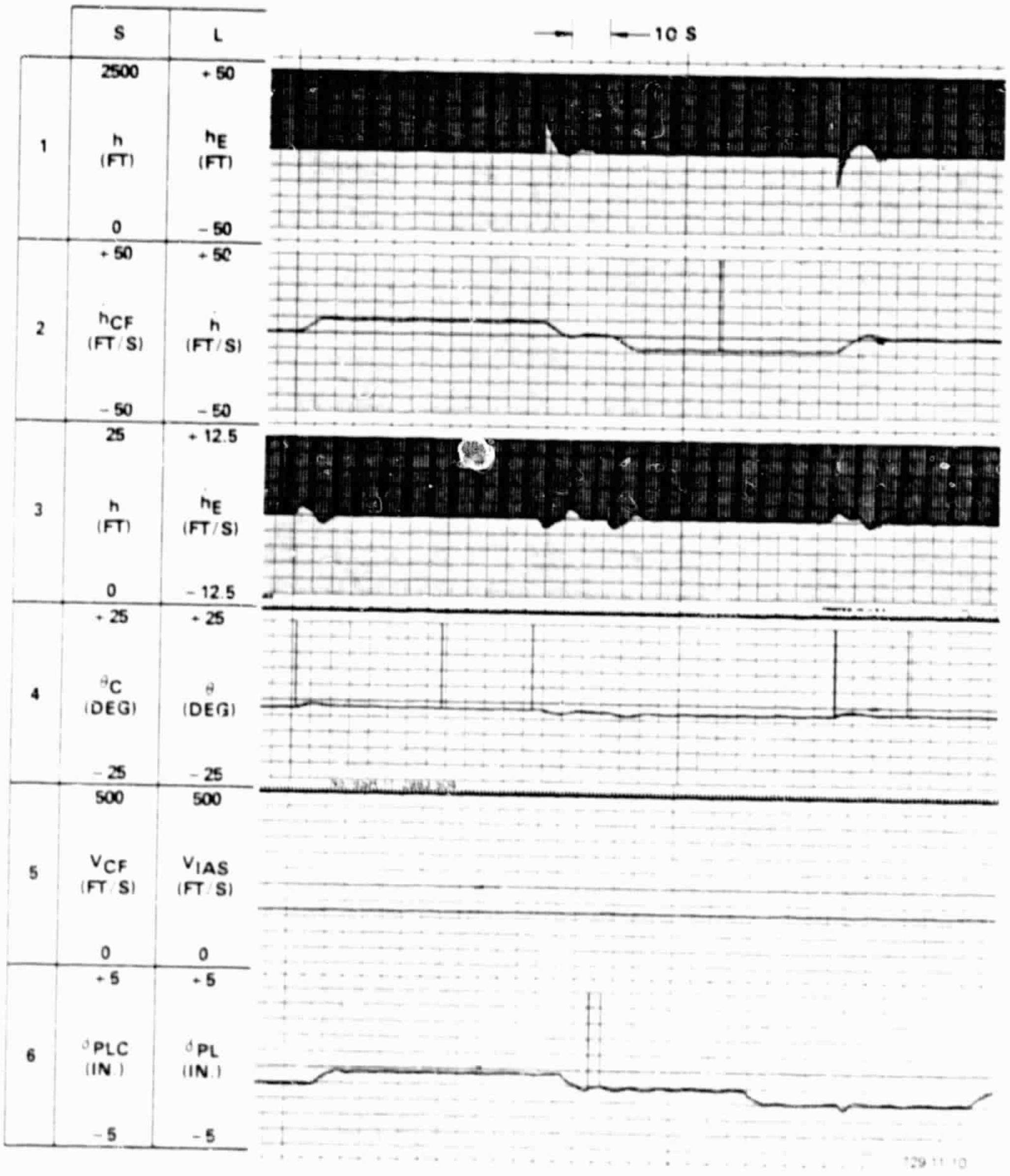


Figure 6-19
Altitude-Select at 100 Knots,
No Disturbance

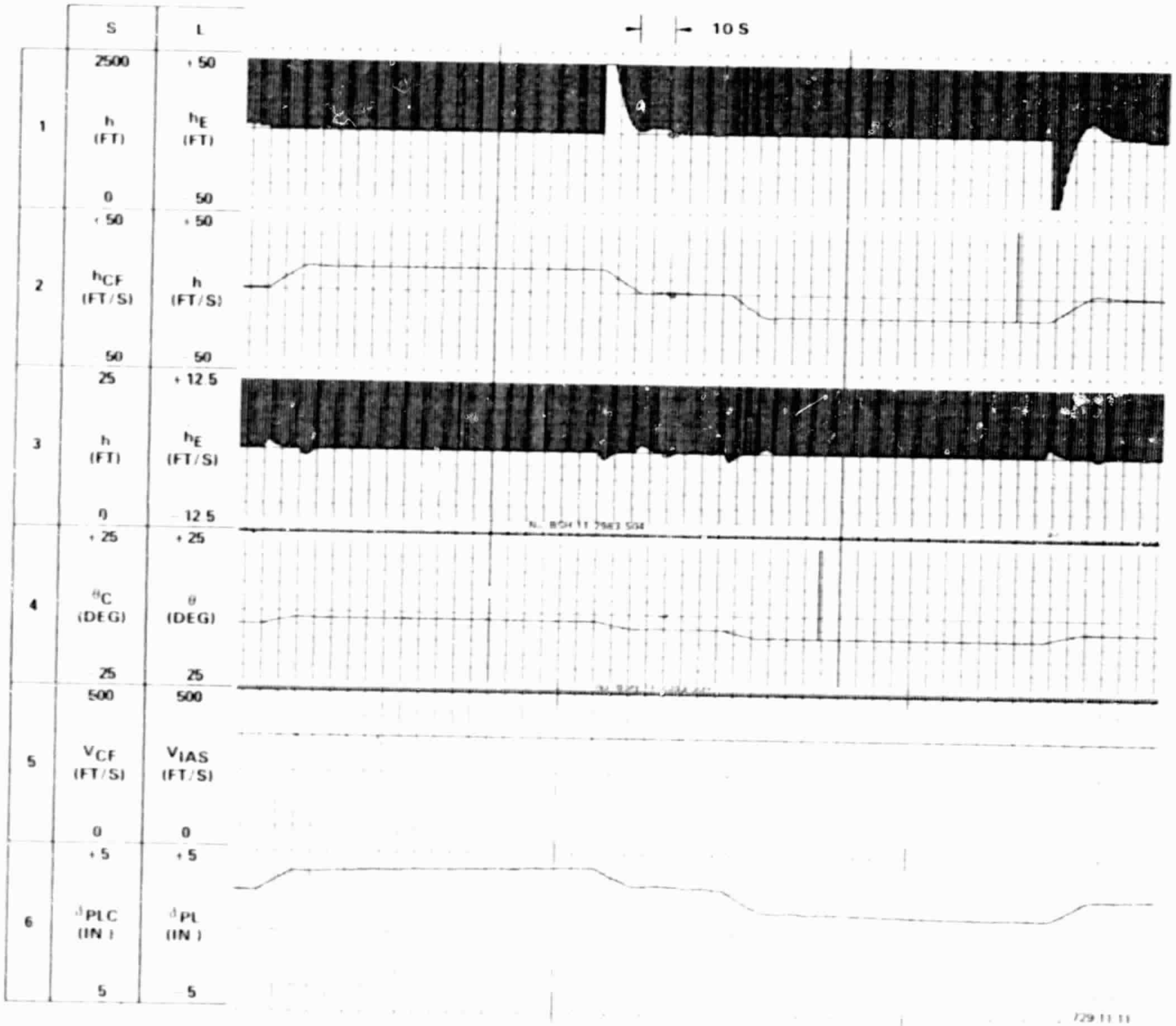


Figure 6-20
Altitude-Select at 225 Knots,
No Disturbance

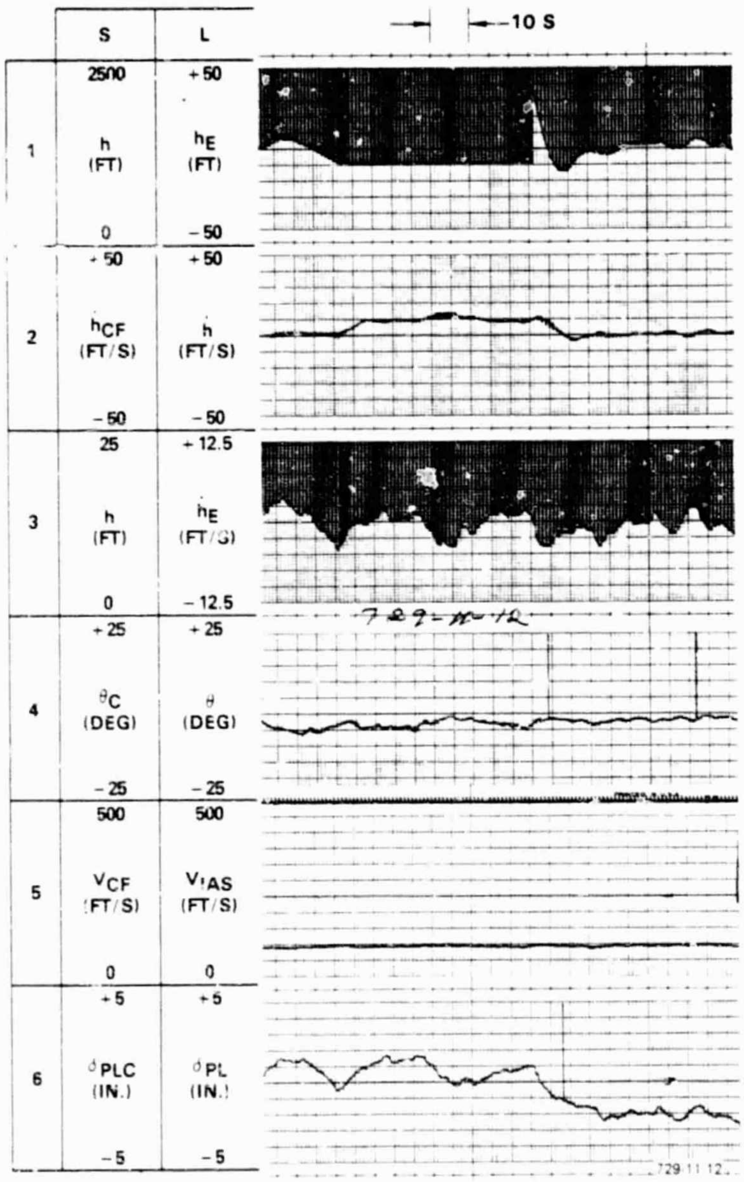


Figure 6-21
Altitude-Select at 50 Knots,
Worst-Case Disturbances

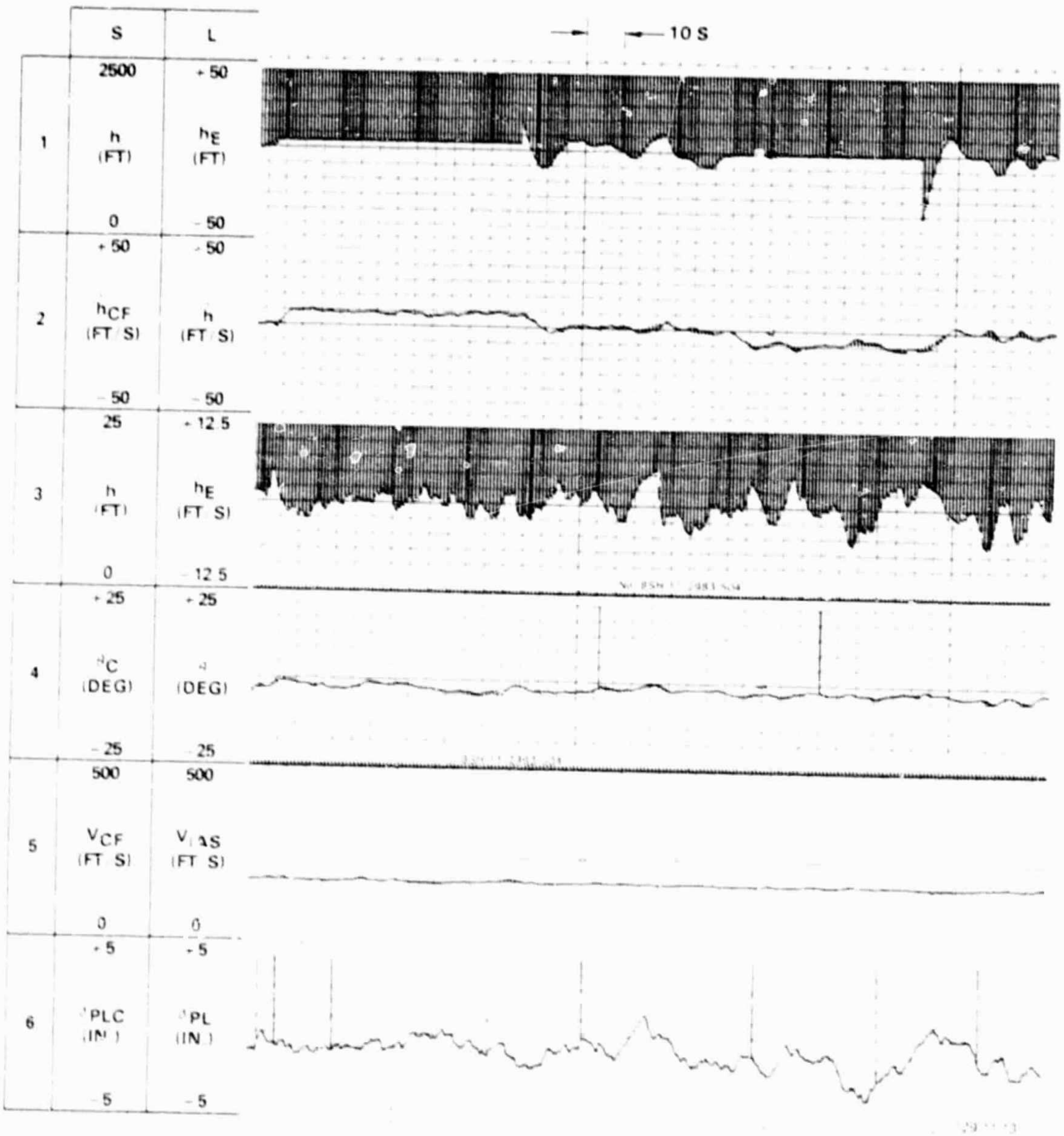


Figure 6-22
Altitude-Select at 100 Knots,
Worst-Case Disturbances

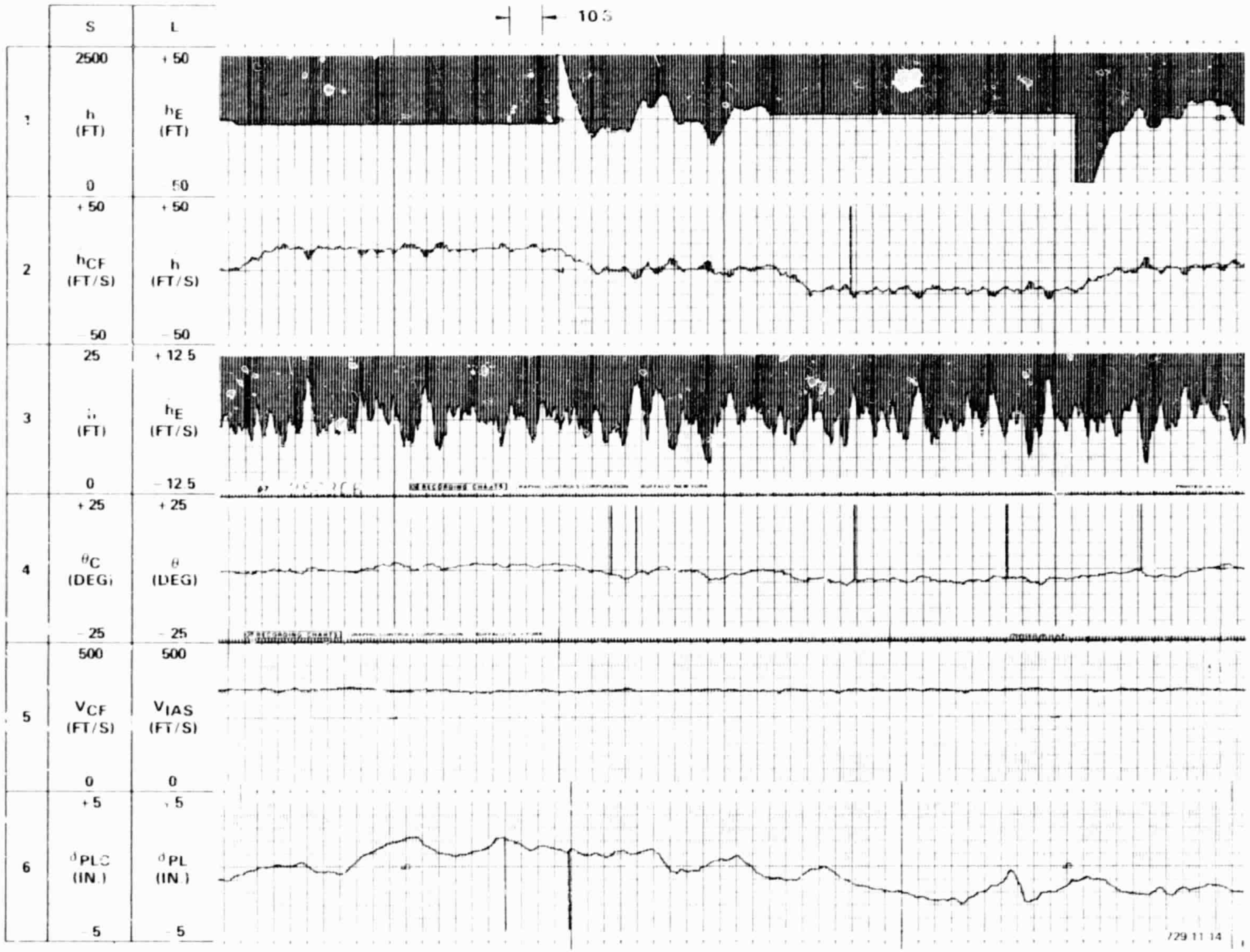


Figure 6-23
Altitude-Select at 225 Knots,
Worst-Case Disturbances

6.6.4 The Grand Tour

The "Grand Tour" is an extended test sequence that demonstrates the system performance in most of the available lateral-directional modes. It begins as shown in Figure 6-24 at IC = 6 with

- V_{IAS} = 140 knots
- Basic A/P Engaged
- HDG HLD engaged (ψ = 353 degrees)
- ALT HLD engaged (h = 3000 feet)
- AUTO NAV engaged
- TAC armed for 270-degree radial

Also, the location of the waypoint (WPT) has been set at x = 30,000 feet and y = 20,000 feet (in the runway reference frame) via the keyboard. The entire sequence takes place with no pilot input on the sticks or on the pedals. Figure 6-24A shows the profiles recorded for this sequence.

- The aircraft approaches the 270-degree onbound TAC radial and then maneuvers to capture it at a standard turn rate (3 deg/s) on a circular path.
- After the TAC course is captured, the pilot selects CRS = 353 degrees and arms VOR. The aircraft remains in the TAC CRS mode until the VOR capture conditions are met. Then it similarly captures and tracks the inbound VOR radial to Stockton. This station is approximately 50 miles away, so the cross-track estimator (D_Y and $\dot{D}_Y$) are noisier than for TAC, as evident in Figure 6-24.
- After VOR is captured, the pilot selects CRS = 70 degrees and arms WPT to capture this radial to the previously positioned waypoint. The action to capture this radial is similar to the above cases.
- Similarly, the pilot sets up to capture the outbound 140-degree radial from the waypoint.

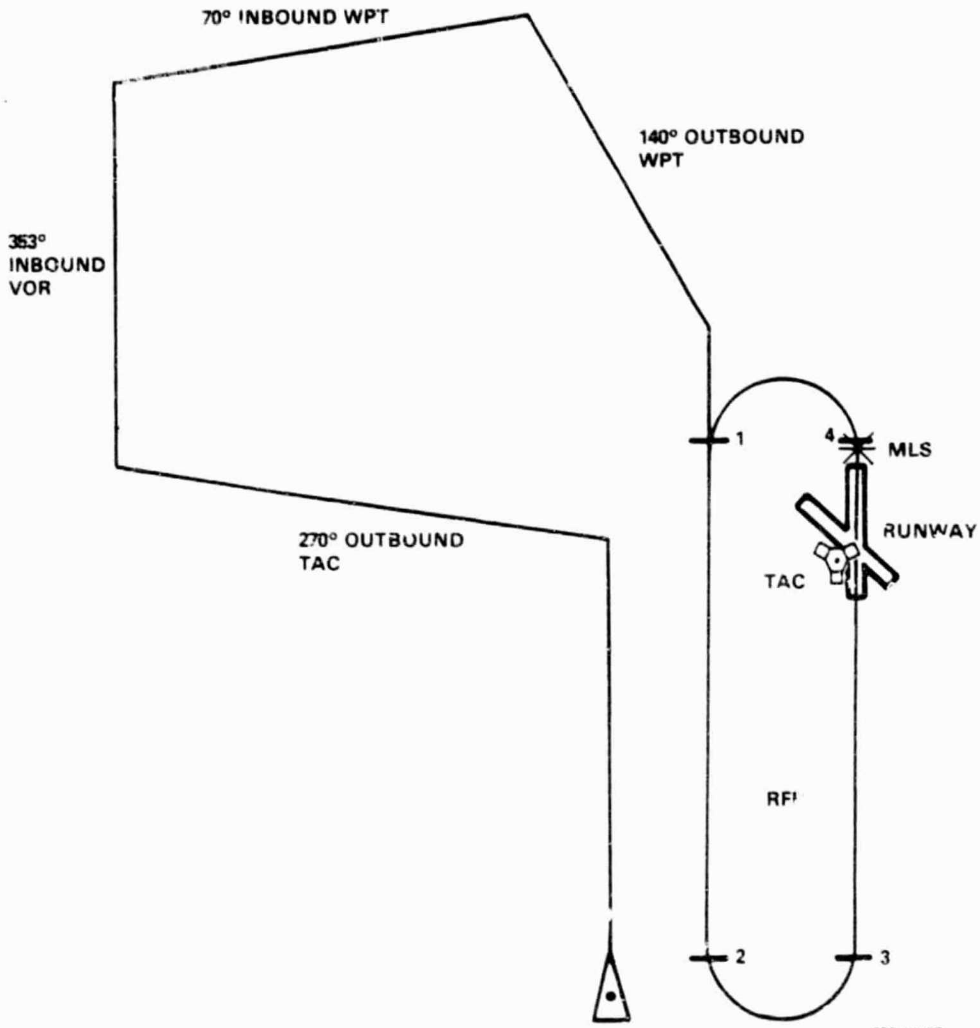
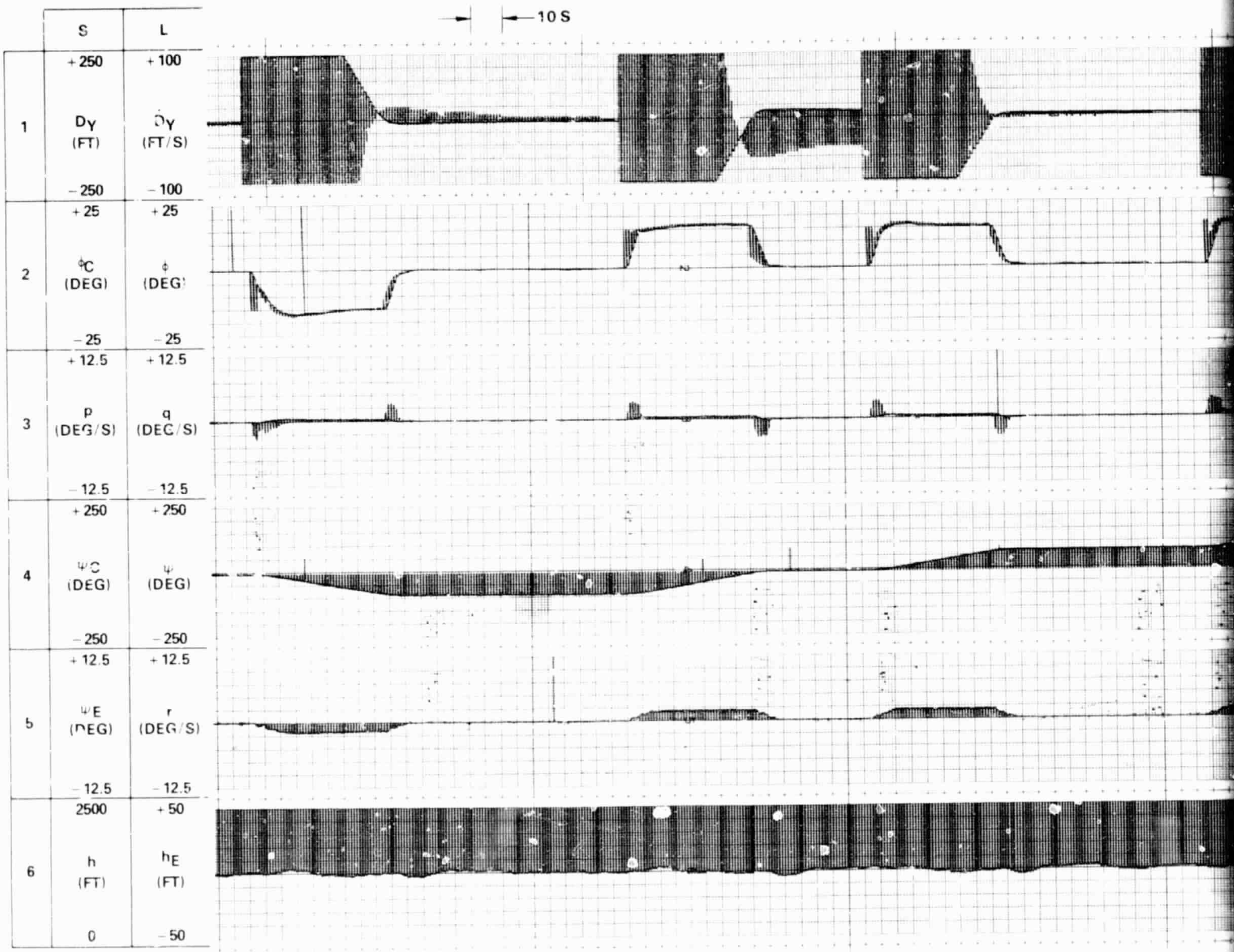
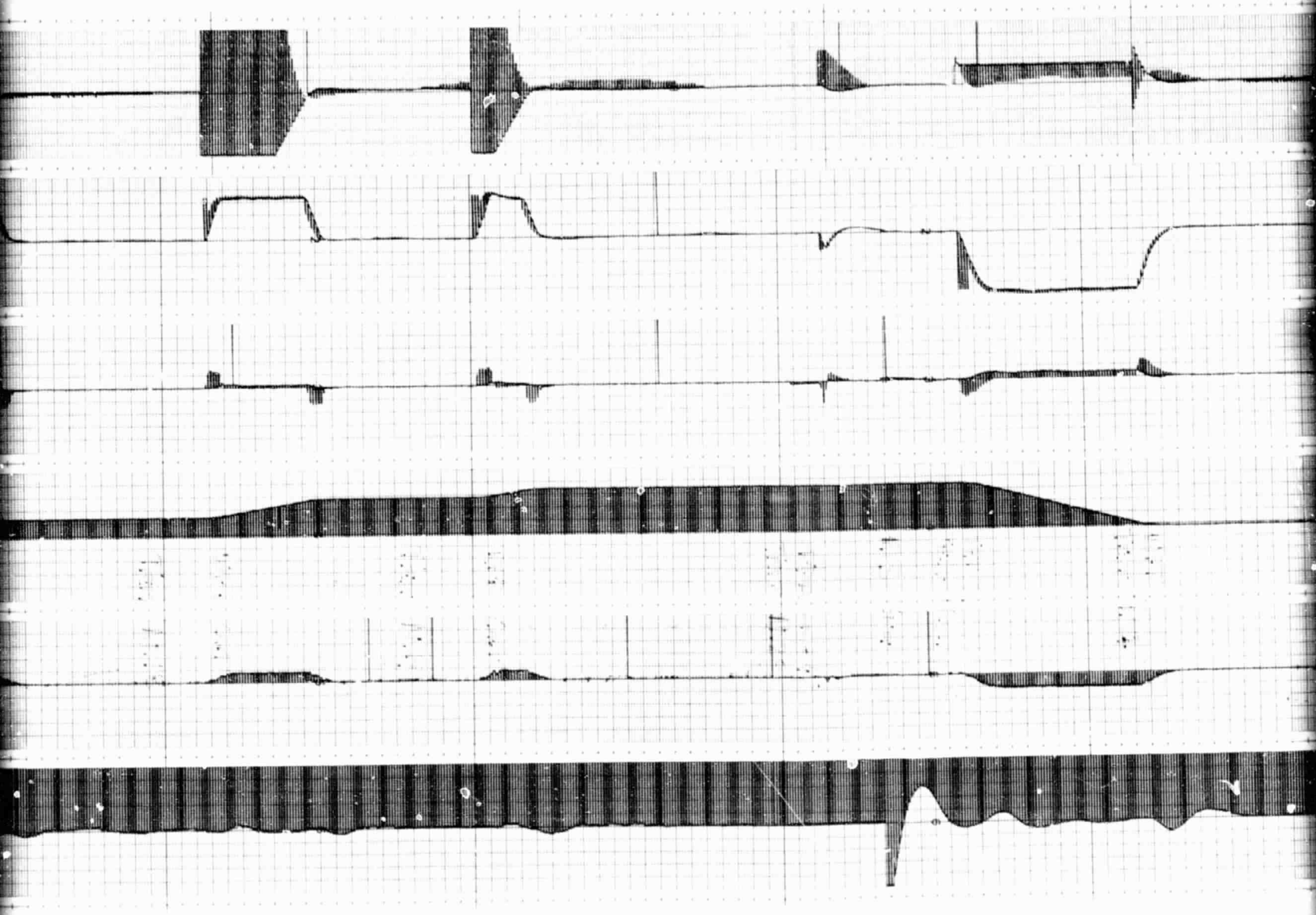


Figure 6-24
The Grand Tour Course



FOLDOUT FRAME



FOLDOUT FRAME 2

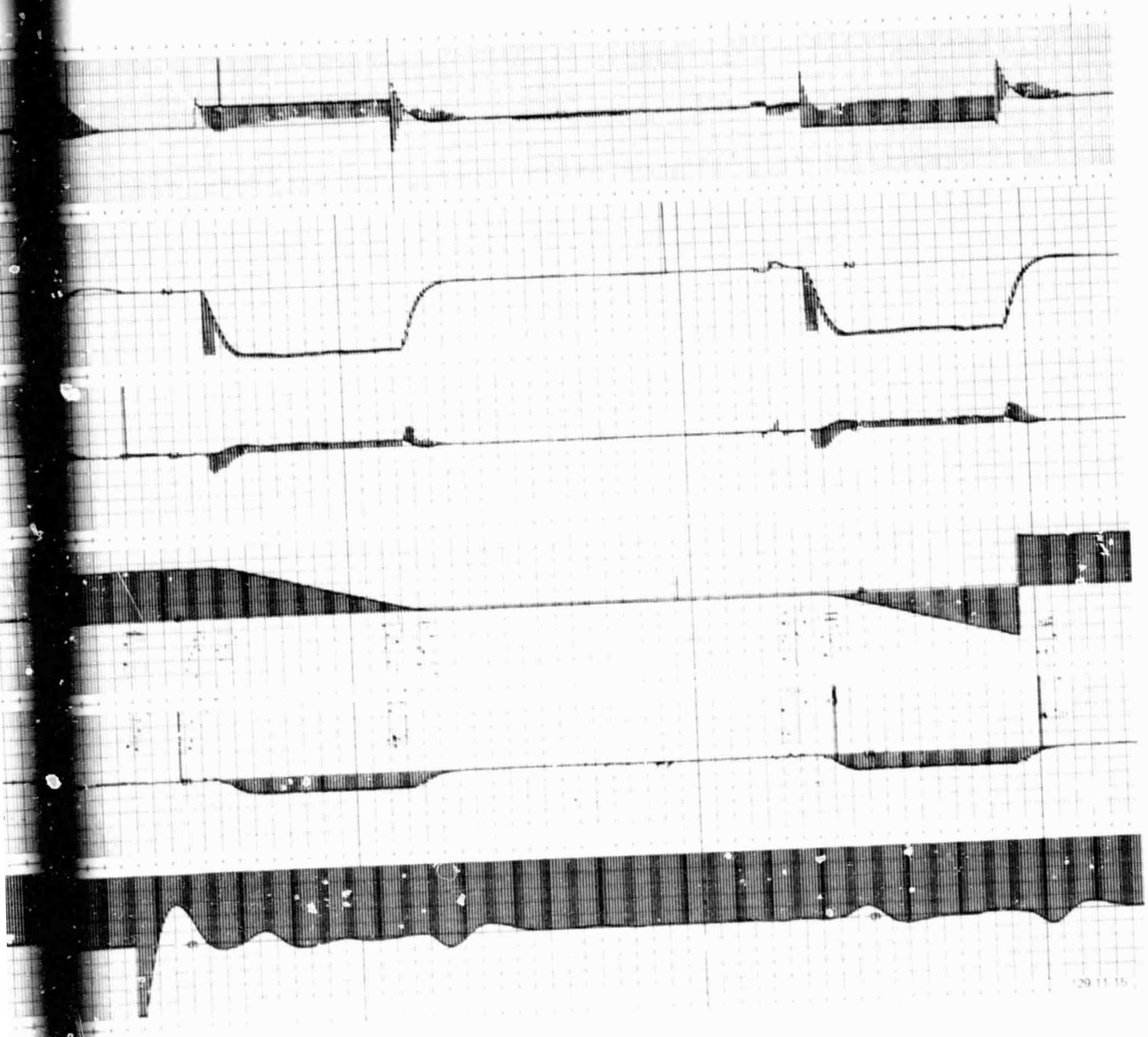


Figure 6-24A
The Grand Tour, No Disturbance

- After this radial has been captured, the pilot arranges to capture the backward extension of Segment 2 (the line between Waypoints 1 and 2) of the Reference Flight Path (RFP) by entering WPN = 2 on the keyboard and then arming RFP. The right HSI DME window then indicates direct range to Waypoint 2 of the RFP. Before the RFP is captured laterally, the lower FMA displays ALT, WPT, RFPA. After the RFP is captured laterally, the lower FMA displays ALT, RFP, and the right HSI DME window indicates along-track distance to Waypoint 2. The capture performance is the same as above.

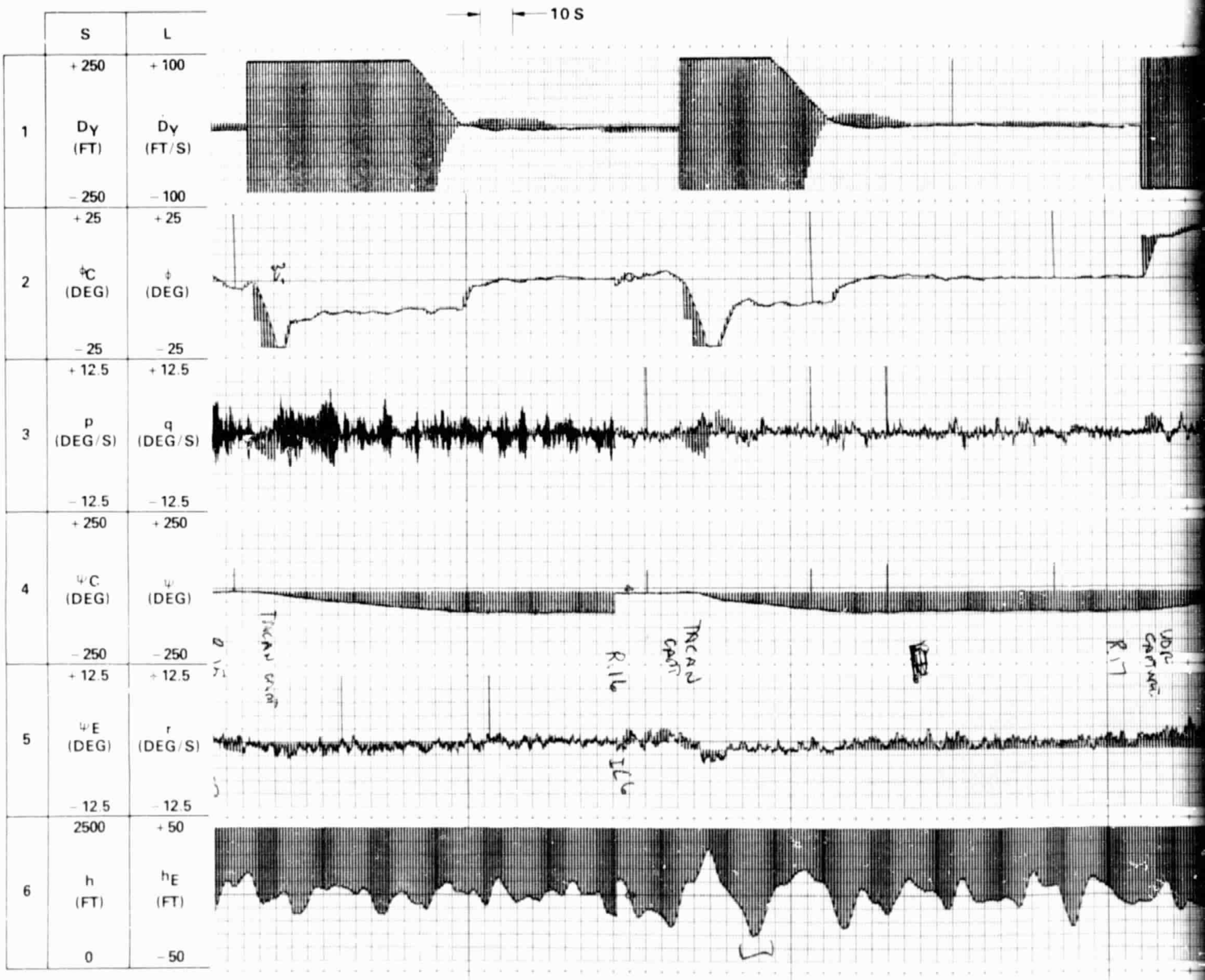
- After the lateral RFP capture, the pilot initiates a -4 degree FPA select to set up an intercept with the vertical RFP reference at 2500 feet above ground. Before the RFP is captured vertically, the lower FMA displays FPA, RFP, RFPA. The vertical deviation indicators on the ADI and the HSI show deviation from the vertical reference, scaled 250 ft/dot.

- After the RFP is captured vertically, the lower FMA displays RFP, RFP. The next waypoint number and its altitude are displayed at the lower left corner of the MFD: WPT = 2, ALT = 2640.

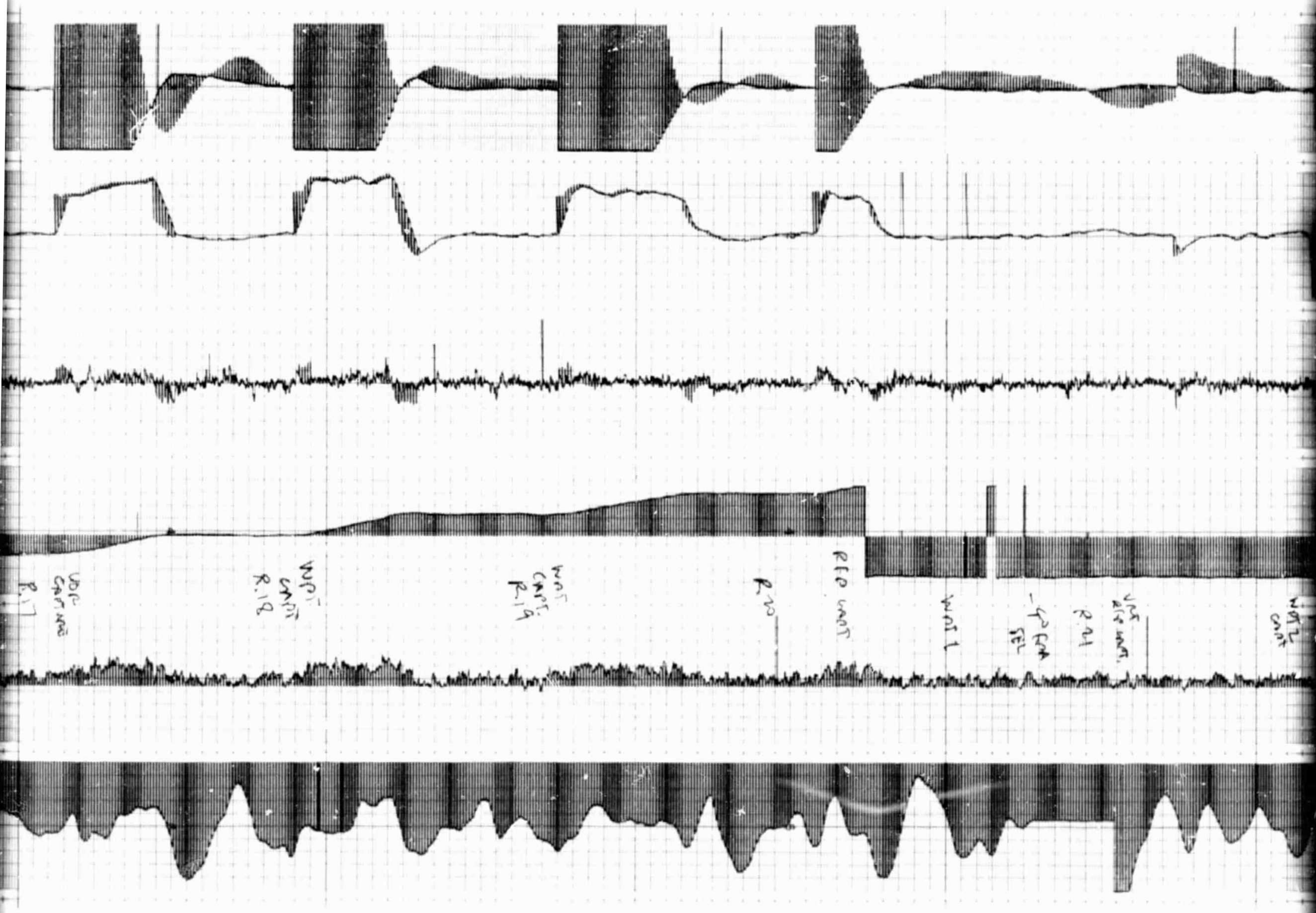
- When the aircraft gets to Waypoint 2, it captures the curved segment to Waypoint 3. The MFD then displays WPT = 3, ALT = 2640, and the right HSI DME window indicates the along-the-path distance to Waypoint 3. The aircraft rolls to a nominal roll altitude of 21 degreees (at 140 knots) in the curved segment.

- The aircraft continues to fly around the oval RFP as described above. The test is discontinued when the aircraft has again passed Waypoint 1.

The above sequence is repeated under the worst-case disturbance conditions which include, in addition to the noise models listed above, the navigation noise models described in Paragraph 6.5.3.1. The performance profiles are shown in Figure 6-25.



FOLDOUT FRAME



FOLDOUT FRAME 2

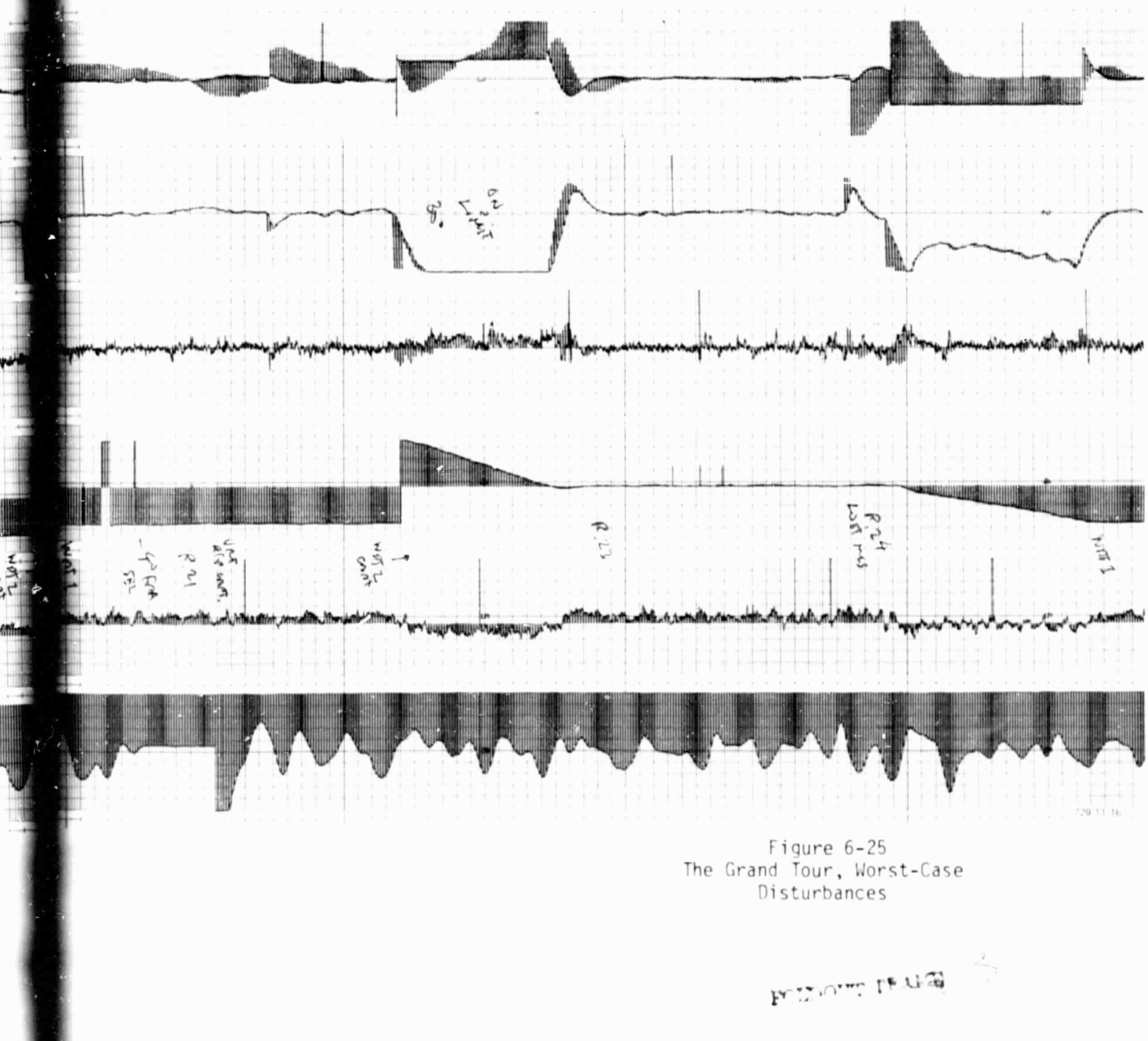


Figure 6-25
The Grand Tour, Worst-Case
Disturbances

FOLDOUT FRAME

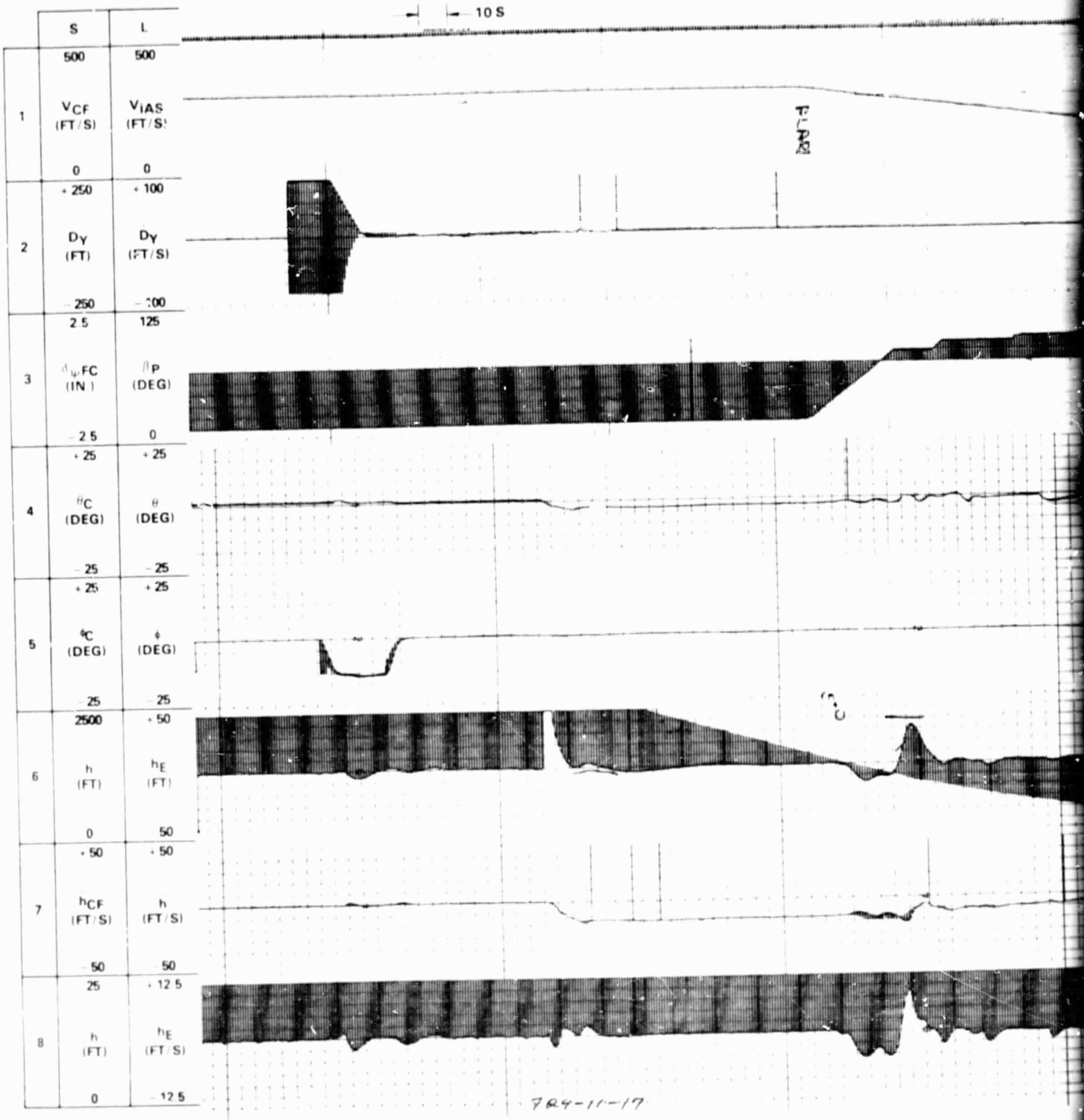
6.6.5 Straight-In Land (LAND-1)

The LAND-1 mode is the straight-in automatic land mode along Runway 35, with a selectable glide slope that aims at a hover point 10 feet above the touchdown point. The FLARE mode is a deceleration mode that brings the aircraft to zero ground velocity at the hover point. MLS navigation is required for all LAND modes. The first approach is without disturbances, and is initialized as follows:

- V_{IAS} = 140 knots
- Basic A/P engaged
- ALT HLD engaged (h = 3000 feet)
- HDG HLD engaged (ψ = 40 degrees)
- LAND-1 selected on keyboard
- G/S of -4 degrees selected on keyboard
- AUTO NAV selected
- The aircraft is positioned left of the localizer as detailed in Paragraph 6.5.2 for IC = 8.

The following events take place in the first Land sequence, which is without disturbances. Figure 6-26 shows the associated profiles.

- When the LAND mode is armed, the lower FMA displays ALT, HDG, L-1A.
- The MSP CRS display and the HSI course pointer indicate the LAND course reference (353 degrees).
- Course deviation is shown on the HSI scaled 100 ft/dot, and on the ADI scaled 100-foot full scale.
- When the lateral capture conditions are met, the aircraft maneuvers to capture the LAND-1 course reference. (The initial course intersects the LAND course at 11.3 nautical miles from the MLS station.)
- The lower FMA displays ALT, L-1, L-1A.
- The ADI and HSI show vertical deviation from the glide-slope reference scaled 50 ft/dot.



Stra
Gli

FOLDOUT FRAME

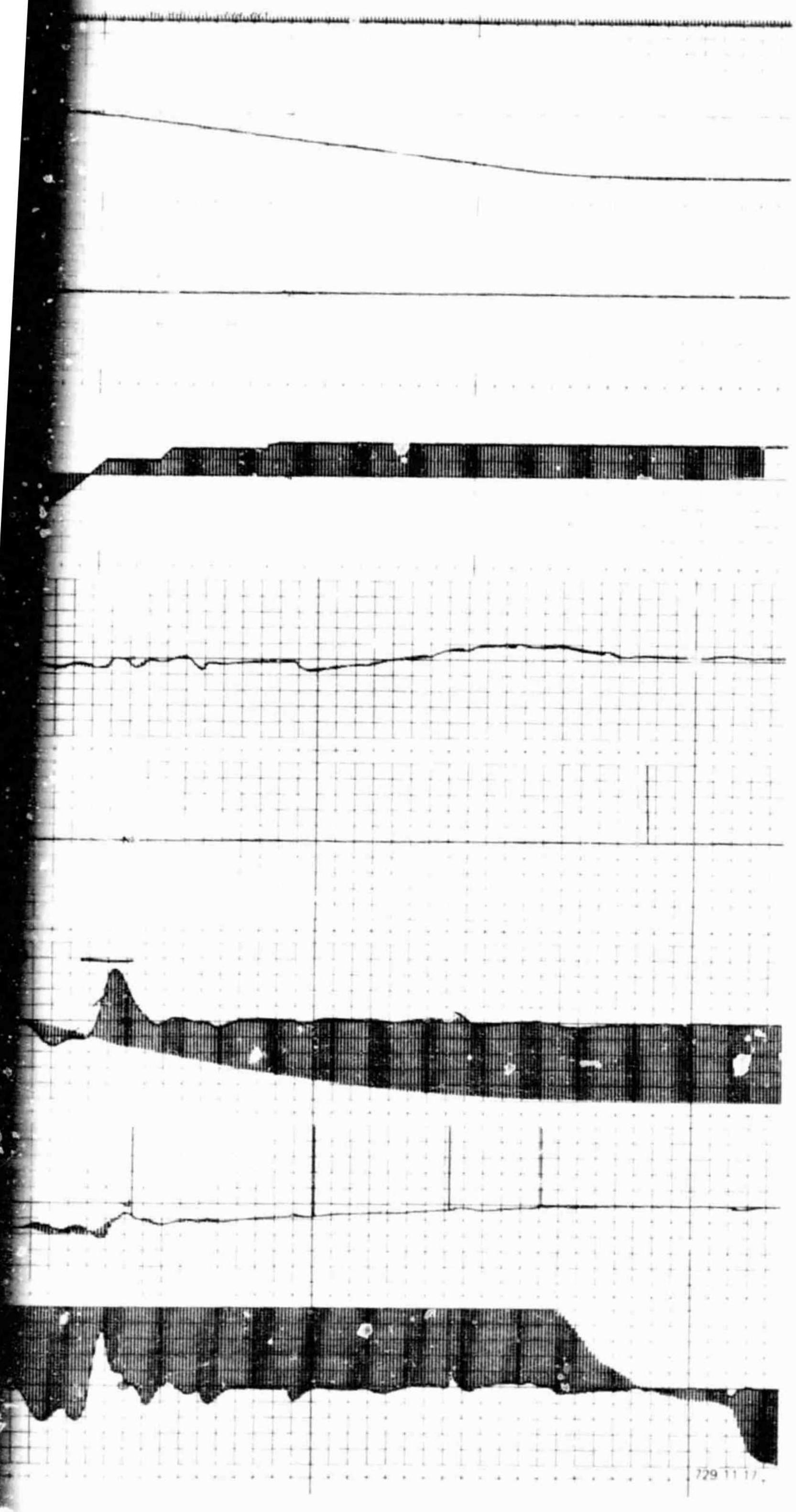
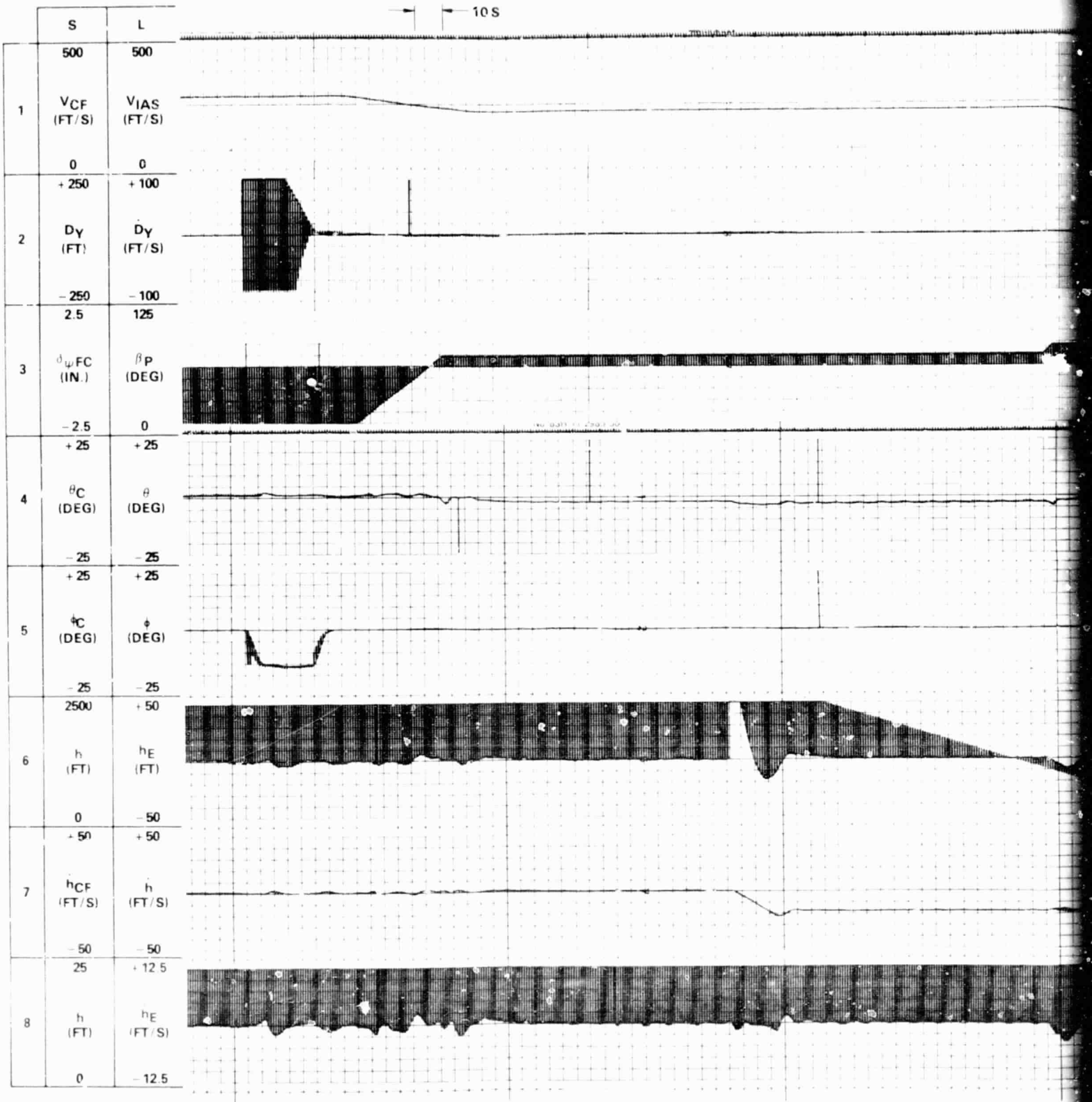


Figure 6-26
Straight-In Land at -4 Degree
Glide Slope, No Disturbance

FOLDOUT FRAME 2

- When the vertical capture conditions are met, the aircraft maneuvers to capture the -4 degree glide-slope reference (which intersects 3140 feet altitude at 7.8 nautical miles from the MLS station).
- The lower FMA displays L-1, L-1.
- At about 4 nautical miles MLS range, the FLR mode engages (assuming 140 knots ground speed). The green FLR light on the ADI then comes on, the lower FMA displays FLR, L-1, the upper FMA displays PYLONS TO 75° and the pylons start conversion to 75 degrees.
- The aircraft starts decelerating at 1.5 ft/s^2 nominally.
- At approximately 100 knots the pilot lowers the landing gear.
- At 100 knots the pylons start conversion to 85 degrees.
- At 75 knots the pylons start conversion to 90 degrees.
- At 60 knots the flaps start deployment to 75 degrees.
- At 200 feet above the runway the ADI radio altitude indicator starts rising.
- At the decision height (normally 100 feet above the runway) the amber DHT light on the ADI comes on.
- The aircraft gradually comes to a hover at 10 feet above the runway.
- When the aircraft has stabilized at a hover, the letdown mode engages and the upper FMA displays LETDOWN. The aircraft then gradually descends to touchdown.
- When the touchdown discrete is obtained, the A/P is disengaged and the upper FMA displays TOUCHDOWN.

The profiles for a similar sequence, but with a -6 degree glide slope preselected on the keyboard, are shown in Figure 6-27. However, the aircraft is not capable of decelerating on a -6 degree glide slope while it is in the airplane mode. An IAS Select to 100 knots is therefore done, right after the localizer is captured, which converts the pylon angle to 75 degrees. This conversion must be completed before the glide slope is captured.


10 S
S
L
1
500
500
VCF (FT/S)
VIAS (FT/S)
0
0
2
+ 250
+ 100
DY (FT)
ḊY (FT/S)
– 250
– 100
3
2.5
125
δψFC (IN.)
βP (DEG)
– 2.5
0
4
+ 25
+ 25
θC (DEG)
θ (DEG)
– 25
– 25
5
+ 25
+ 25
φC (DEG)
φ (DEG)
– 25
– 25
6
2500
+ 50
h (FT)
hE (FT)
0
– 50
7
+ 50
+ 50
ḣCF (FT/S)
ḣ (FT/S)
– 50
– 50
8
25
+ 12.5
h (FT)
ḣE (FT/S)
0
– 12.5

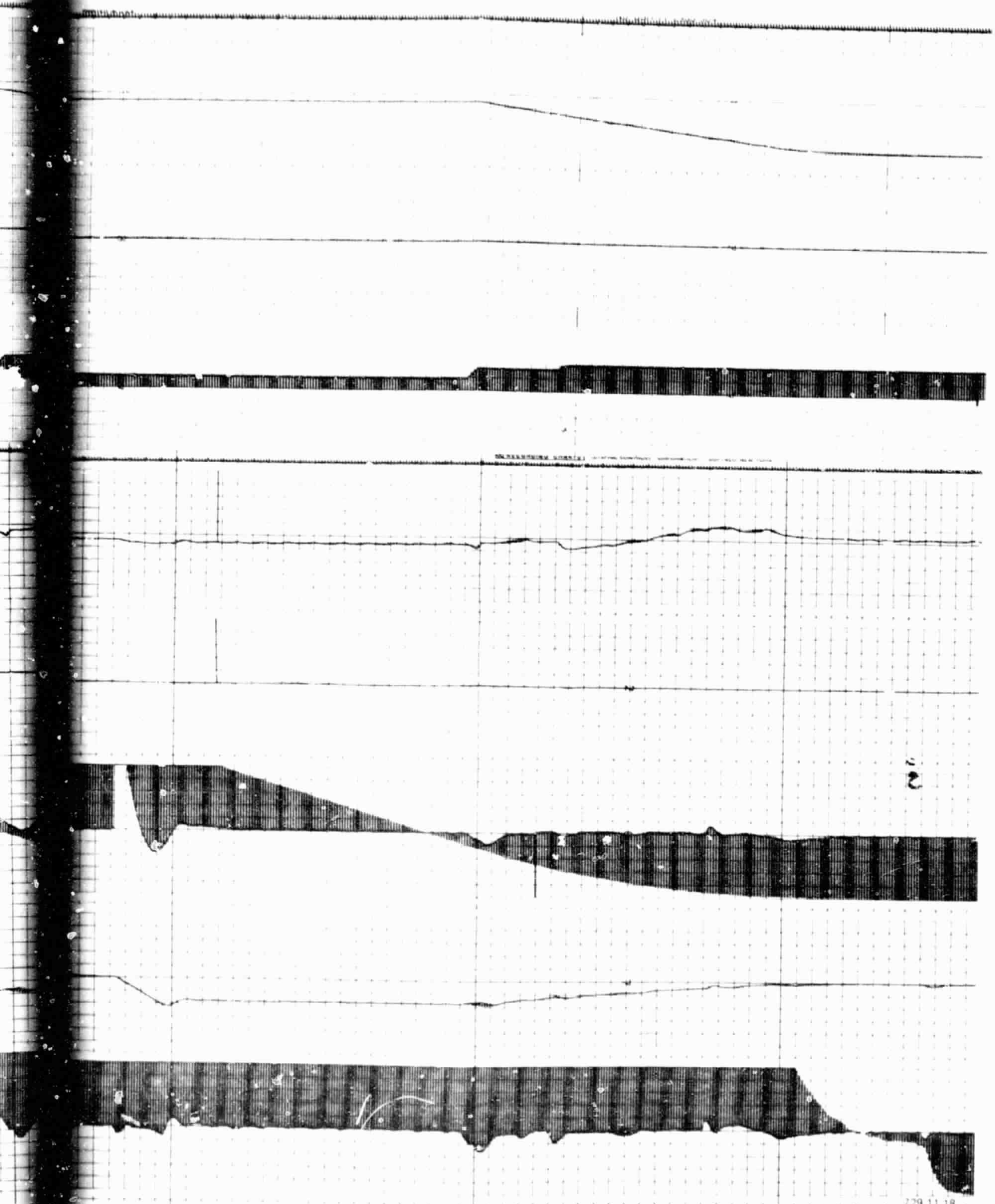


Figure 6-27
Straight-In Land at -6 Degree
Glide Slope, No Disturbance

A LAND-1 sequence with worst-case disturbances applied, including a 90-degree crosswind, was run at -4 degrees glide slope. Figure 6-28 shows the recorded profiles. A test was also attempted at -6 degrees glide slope with the worst-case disturbances, but the authority limits on the power lever were not sufficient to control the aircraft in this condition.

6.6.6 Helix Land (LAND-2)

The LAND-2 trajectory is entered from the Reference Flight Path as illustrated in Figure 6-28A. Before the start of LAND-2 is reached, the RFP must be captured laterally and vertically, MLS guidance must be engaged, and the airspeed should be at 60 knots. The -6.1 degree glide slope starts at 8307 feet from the touchdown point, and the 2-turn helix is placed between this point and 3600 feet from the touchdown point, as selected on the keyboard. The aircraft descends 1600 feet in the two helix revolutions. The FLAPE mode, which engages only after the helix is completed, and the final hover and letdown mode, are the same as in LAND-1. The initial conditions for the no-disturbances sequence are as follows:

- V_{IAS} = 60 knots
- Basic A/P engaged
- ALT HLD engaged (h = 3000 feet)
- HDG HLD engaged (ψ = 130 degrees)
- LAND = 2 selected on the keyboard
- HLX = 7000 feet selected on the keybard
- Waypoint = 2 selected on the keyboard
- AUTO NAV selected
- The aircraft is positioned as detailed in Paragraph 6.5.2 for IC = 11

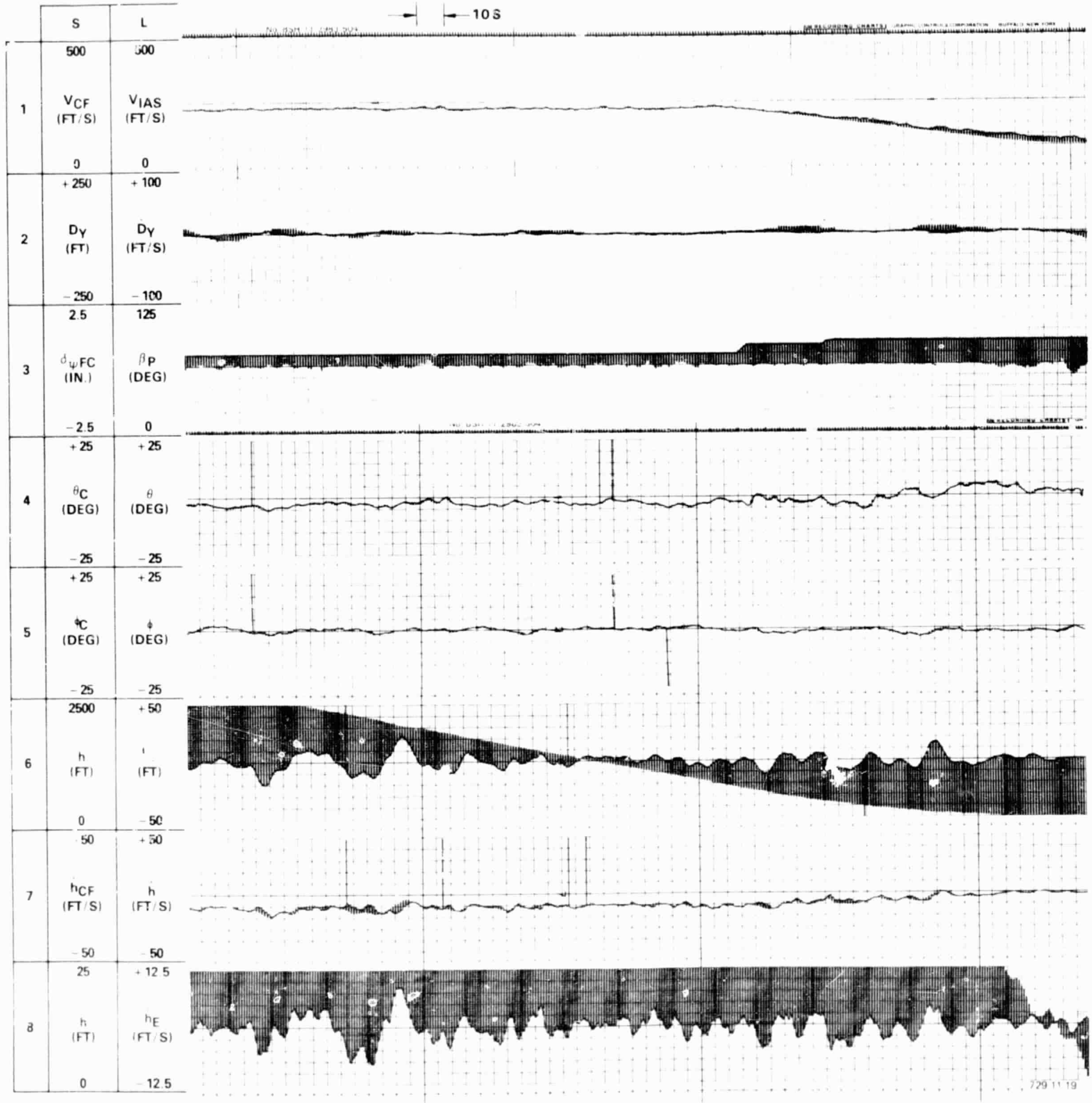


Figure 6-28
Straight-In Land at -4 Degree
Glide Slope, Worst-Case
Disturbances

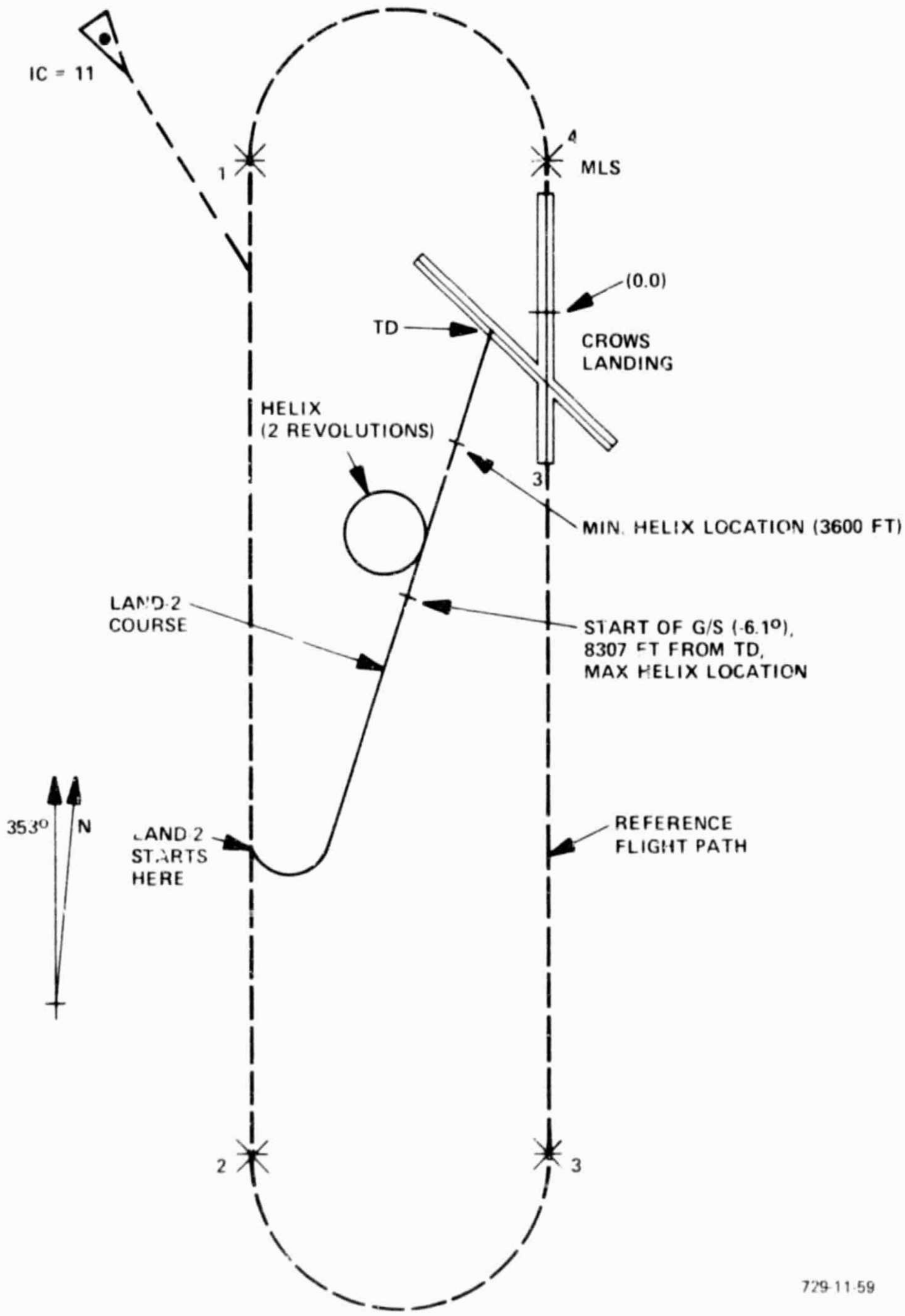


Figure 6-28A
LAND-2 Course

The following sequence of events take place. Figure 6-29 shows the associated profiles.

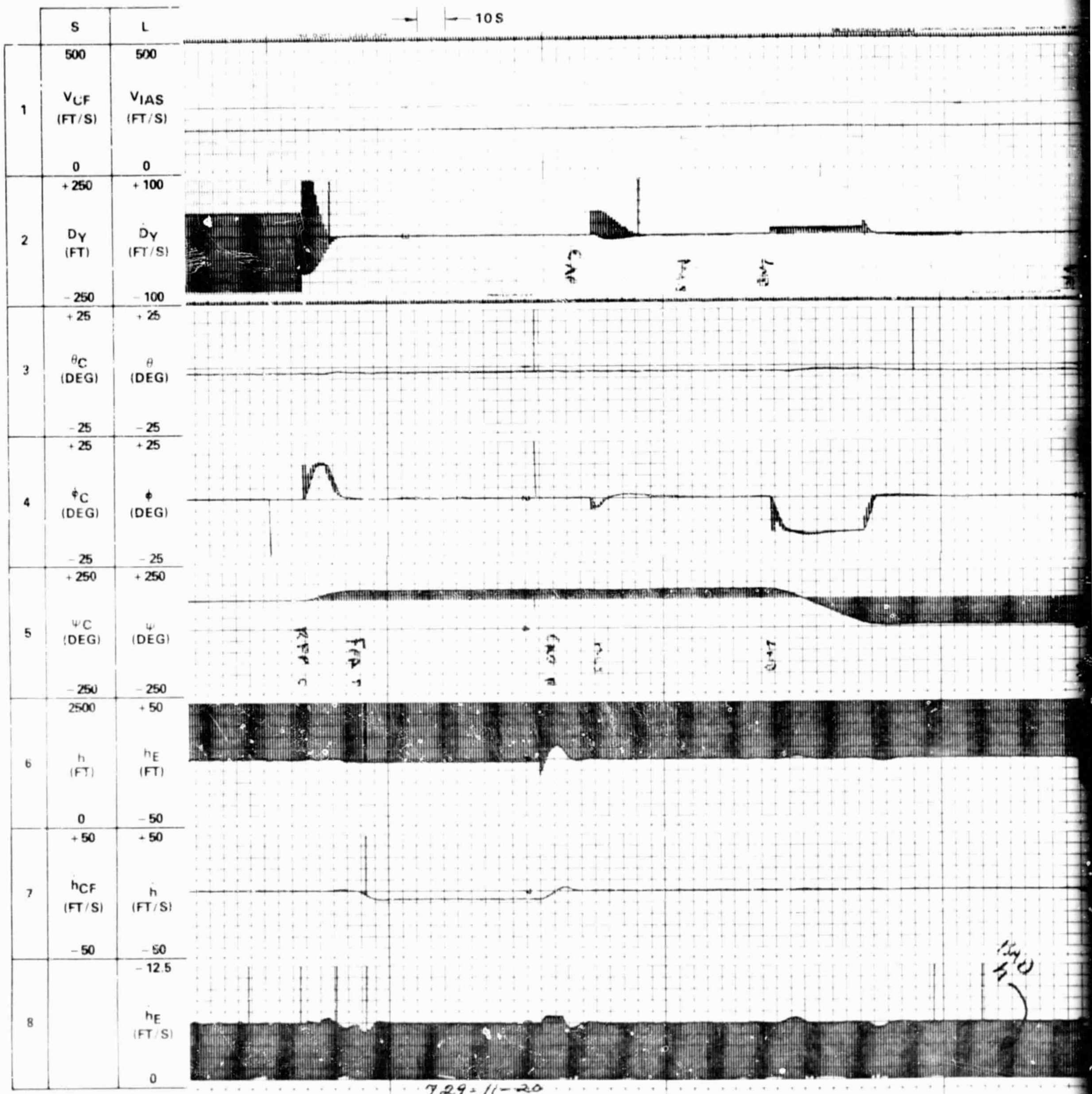
- The pilot arms RFP. After a while the RFP is captured laterally and the lower FMA displays ALT, RFP, RFPA.
- When the range to Waypoint 2 gets to 4 nautical miles, the pilot arms LAND and engages FPA Select at -4 degrees to intercept the RFP vertically. The LAND-2 course then is displayed on the MFD and the lower FMA displays FPA, RFP, L-2A.

When the capture conditions for the LAND-2 course are met:

- LAND-2 engages and the lower FMA displays L-2, L-2.
- The aircraft rolls left to 15 degrees, nominally, tracking the circular segment, then levels out to a straight segment at 18 degrees course angle which is directed to the MLS azimuth/range station.
- When the capture conditions for the LAND-2 glide slope are met, the aircraft starts to descend at a -6.1 degree FPA, nominally.
- When the capture conditions for helix are met, the aircraft rolls left to 15 degrees, nominally, tracking the helix course for two revolutions, then levels out to track the remaining straight segment.
- At about 3544 feet from the touchdown point, or about 1.5 nautical miles range, (assuming 60-knots ground speed), the FLR mode engages and the aircraft decelerates to a hover at 10 feet altitude on Runway 30 (X = -683, Y = -1664, MLS range = .9 nautical mile).
- When the aircraft has stabilized at the hover point, the aircraft lets down the touchdown as in the LAND-1 case.

This sequence is repeated under worst-case disturbances as described previously, with the wind direction at 300 degrees. The resulting profiles are shown in Figure 6-30.

The above test sequences represent only a small part of the performance tests conducted, but are sufficiently representative to demonstrate the performance of the system.



FOLDOUT FRAME

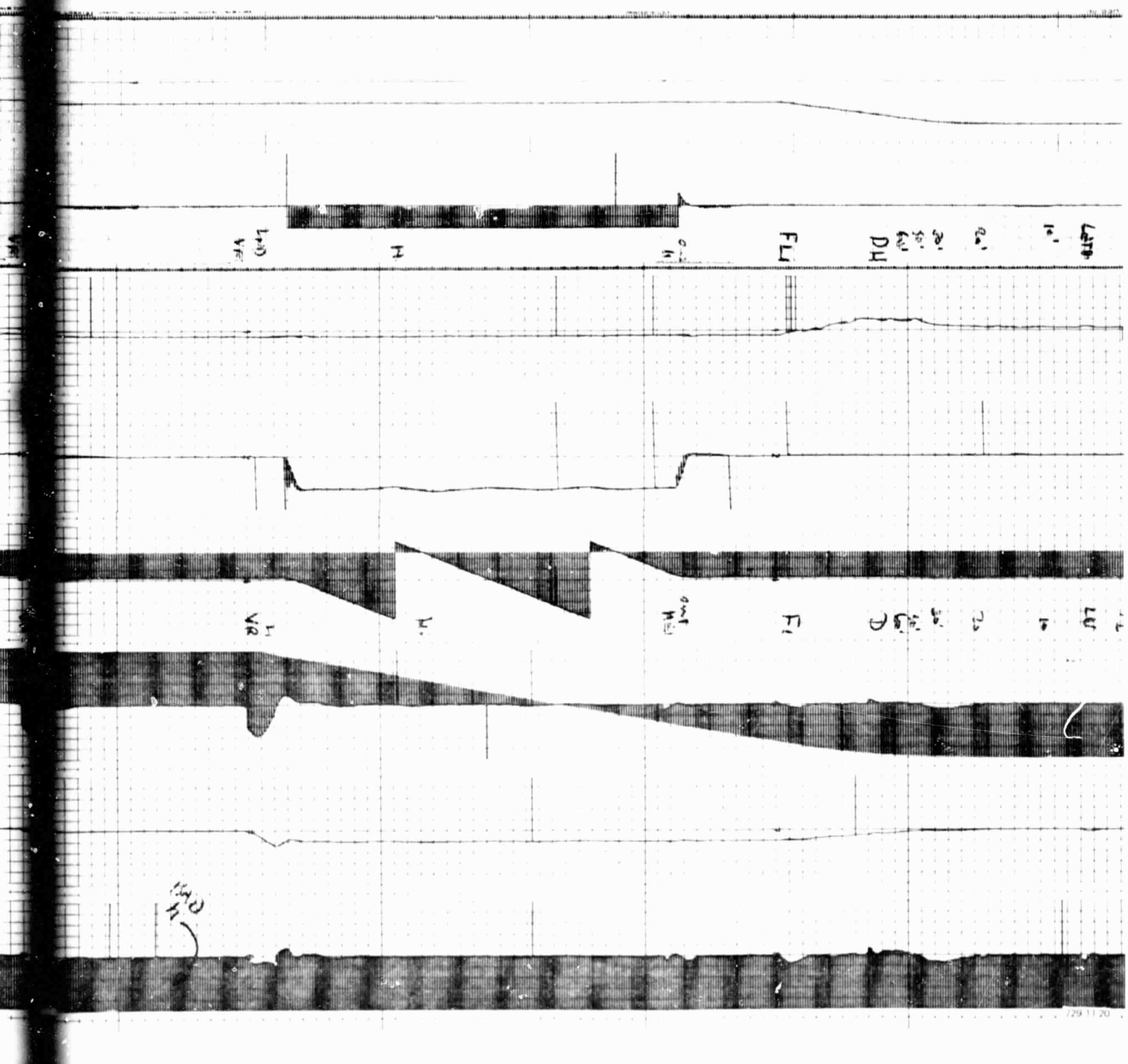
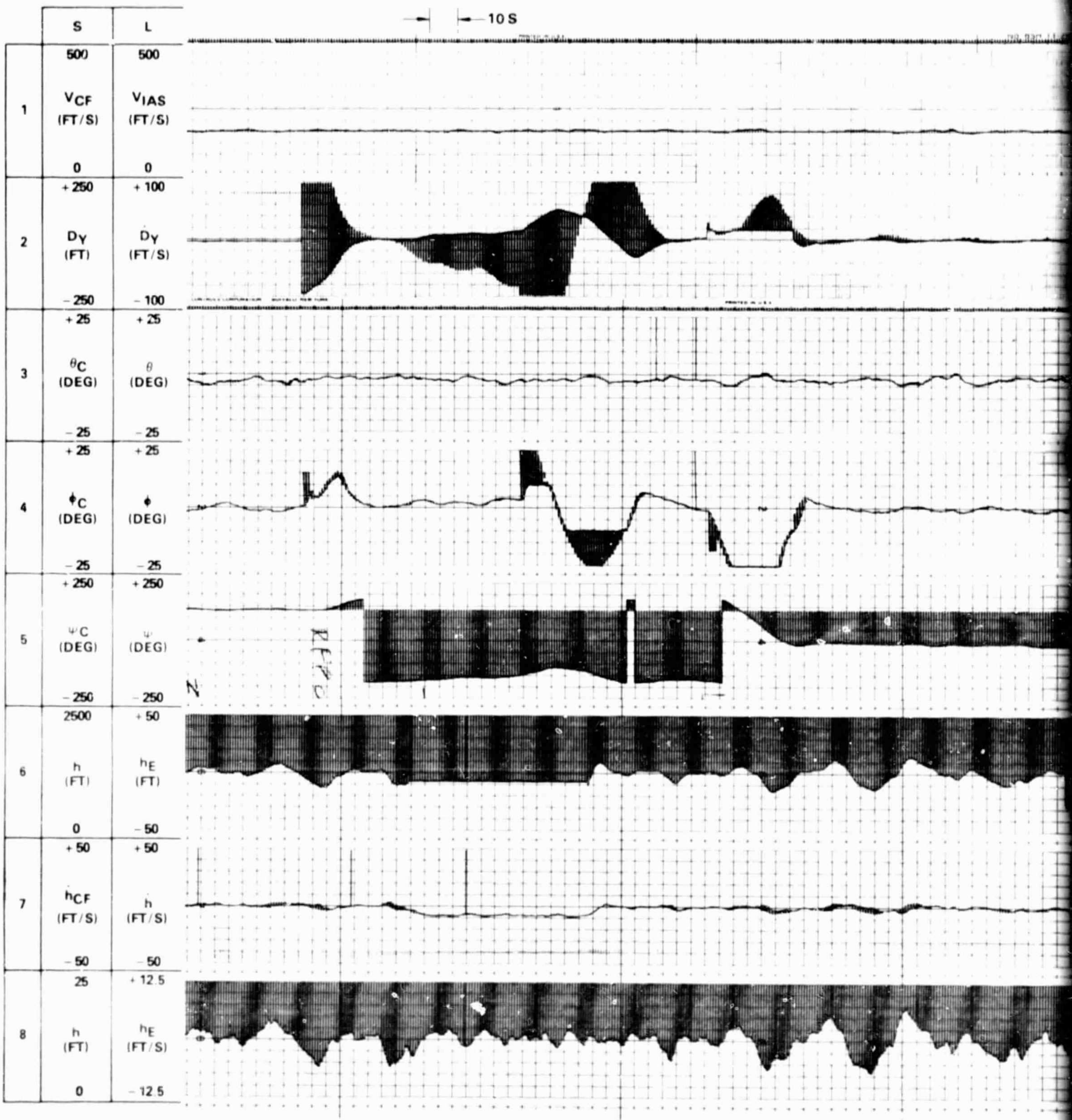


Figure 6-29
LAND-2 Sequence, No Disturbance



FOLDOUT FRAME

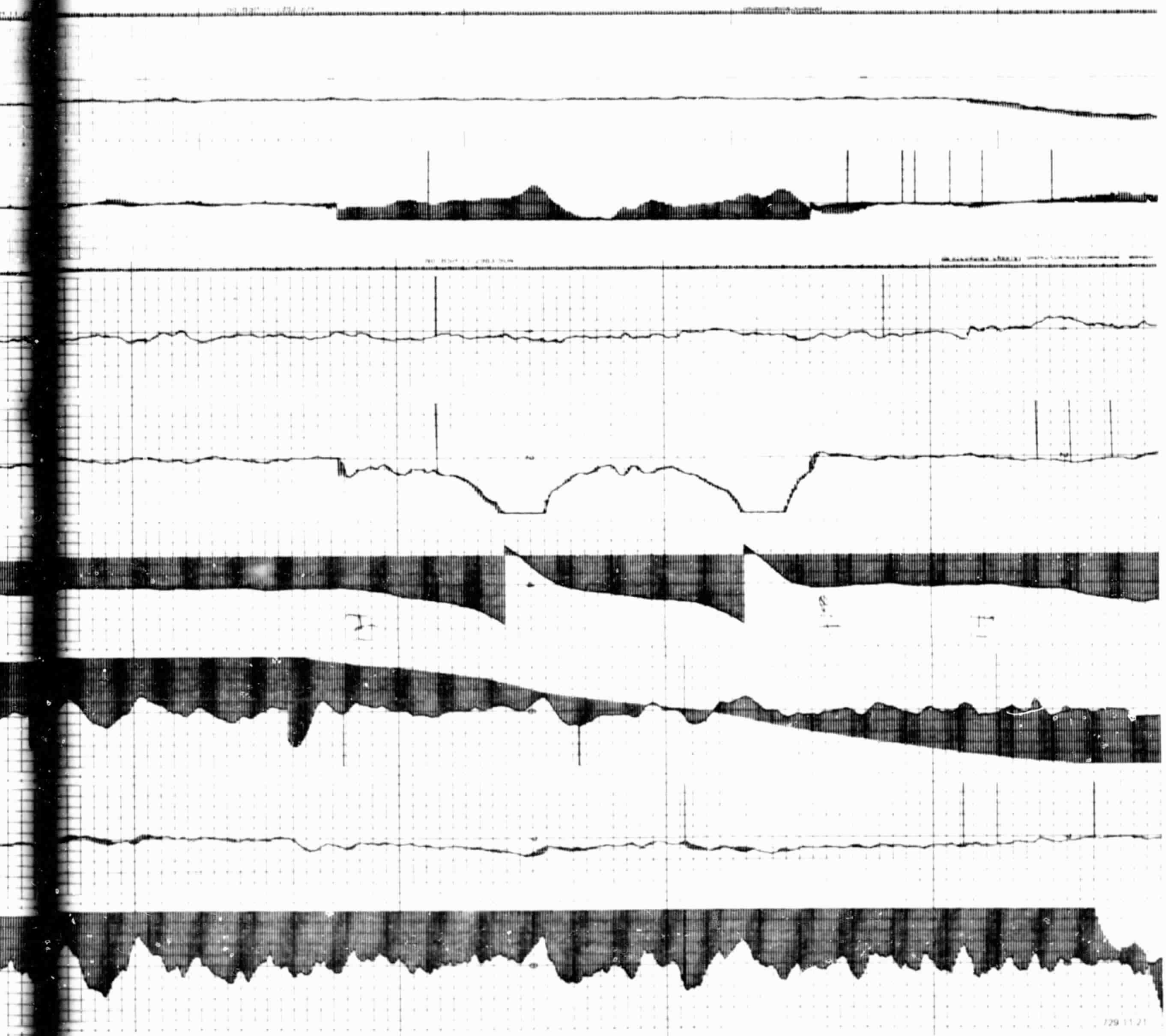


Figure 6-30
LAND-2 Sequence,
Worst-Case Disturbances

SECTION VII

CONCLUSIONS AND RECOMMENDATIONS

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CONCLUSIONS AND RECOMMENDATIONS

The XV-15 V/STOLAND system has demonstrated all of the performance objectives specified at the outset (identified in Section I), and has thereby achieved a new level of performance and automation for V/STOL aircraft. All functions were demonstrated on the NASA-ARC S-19 simulation facility under a comprehensive Dynamic Acceptance Test. The following are the most noteworthy accomplishments of the system:

- Automatic configuration control of a Tilt-Rotor aircraft over its total operating range.
- Total hands-off automatic landing to touchdown on various selectable straight-in glide slopes and on a flight path that includes a two-revolution helix.
- Automatic guidance along a programmed three-dimensional reference flight path.
- Navigation data for the automatic guidance modes computed on board, based on VOR/DME, TACAN or MLS navaid data which was filtered and blended with rate gyro and accelerometer data for smoothing and short-term performance.
- Integration of a large set of functions in a single computer, utilizing 16K words of storage for programs and data. The functions include: automatic guidance and control, flight director guidance, mode select panel logic and displays, mode status alphanumeric display, air data computation, navigation (VOR/DME, TACAN, MLS), ADI/HSI displays, MFD moving map display, failure monitoring and diagnostics reporting, keyboard interface and alphanumeric display, research moding and Research computer interface, and portions of the preflight test program (majority resident in the research computer).

A major feature of the system is its extensive research capability due to the software structure, the uncoupled dual-computer architecture, the software sensed and controlled panels and displays, and the extensive set of sensors as described in Paragraph 3.1. This makes it possible to continually develop and refine the performance of the existing functions and expand functional capabilities. The 10K-plus* of memory in the Research computer provides ample room for software expansion.

The XV-15 V/STOLAND system has of course not yet been tested in flight due to unavailability of the aircraft. Based on the experience with the UH-1H V/STOLAND flight test program, changes in control-law gains and possibly additional prefiltering of sensor signals are likely to be required when the system is tested in flight since the simulation cannot precisely duplicate the real aircraft dynamics. Hence, the primary recommendation is to proceed with installation of the system in the XV-15 aircraft as soon as it becomes available, and then conduct flight tests. The performance in the final Land phases cannot be adequately judged by the simulation, and will probably require additional development under a flight test program.

A disadvantage with the current navigation reference system is that it is only usable near Crows Landing. It would not be a major task to generalize the navigation software so that the terminal area reference frame and the associated navaid stations are relocatable in terms of latitude, longitude, and altitude, thereby making the V/STOLAND system operable at any terminal that has the necessary navaids. Sperry recommends that such improvements be made.

Because of the extensive research capability of the V/STOLAND system, the possibilities for experimentation are so vast that they cannot be easily itemized. The XV-15 V/STOLAND system should be able to serve NASA and the Army as an invaluable research tool in V/STOL research for many years to come.

*The preflight test software is resident in the Research computer, but may be over-stored for additional airborne software memory, if necessary, after the preflight test is completed.

APPENDIX A

LIST OF DELIVERABLE HARDWARE AND DOCUMENTS

TABLE A-1
DELIVERABLE HARDWARE

Description	Quantity
1819B Digital Computer and Support Software	2
Data Adapter	1
1819B Computer Control Panel	1
MFD Display Unit	1
MFD Symbol Generator	1
HSI Display Unit (RD-202)	1
HSI Instrument Amplifier Rack	1
HSI Signal Conditioning Unit	1
Mode Select Panel	1
Flight Mode Annunciator	1
Data Entry Panel	1
Static Pressure Transducer	1
HZ-6F ADI	1
Mounting Tray, MFD DU	2
Aircraft Harness (External to Equipment Racks)	1
Dimming Panel Controls	1 set
Accelerometer Assembly (3-Axis)	1
Rate Gyro Assembly, Yaw/Roll	1
Rate Gyro Assembly, Pitch	1
Transducer, Synchro	5
Transducer, LVDT	1
Airborne Software Program	1
Automatic Test Equipment Program and Adapters not common to UH-1H V-STOLAND or STOLAND	1 set
Airborne Hardware Simulator	1
Simulator Cable Set	1
Peripheral Controller	1