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DEVELOPMENT AND FLIGHT TEST OF AN ADVANCED
HYDROFLUIDIC STABILIZATION SYSTEM

Honeywell, Incorporated
Minneapolis, Minnesota

February 1976

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**DEVELOPMENT AND FLIGHT TEST OF AN ADVANCED
HYDROFLUIDIC STABILIZATION SYSTEM**

**Honeywell Inc.
Government and Aeronautical Products Division
Minneapolis, Minn. 55413**

February 1976

Final Report for Period 23 November 1971 - 10 June 1975

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Prepared for

**EUSTIS DIRECTORATE
U. S. ARMY AIR MOBILITY RESEARCH AND DEVELOPMENT LABORATORY
Fort Eustis, Va. 23604**

EUSTIS DIRECTORATE POSITION STATEMENT

This report describes the design, development, and flight test of the Advanced Hydrofluidic Stabilization System, which successfully demonstrated the application of hydrofluidic aircraft state detectors and control command and conditioning devices to perform automatic aircraft stability augmentation and pilot relief functions. The stabilization system developed provides the helicopter pilot with stability augmentation in three axes over the total flight envelope, and with pitch and roll attitude hold, heading hold, altitude hold, and airspeed hold over the normal range of cruise airspeeds. Definition and analysis of this advanced hydrofluidic stabilization system were completed during a previous program. No further USAAMRDL R&D effort is anticipated for this equipment, as the technology is sufficiently developed to be applied to specific user applications.

Mr. George W. Fosdick of the System Support Division served as project engineer for this effort.

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**DEPARTMENT OF THE ARMY
U. S. ARMY AIR MOBILITY RESEARCH & DEVELOPMENT LABORATORY
EUSTIS DIRECTORATE
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ERRATA

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TITLE: Development and Flight Test of an Advanced Hydrofluidic Stabilization System

- p. 85** In the first sentence under "SYNCHRONIZER AND STICK TRIM CIRCUITS", change "system modifitions" to "system modifications".
- p. 136** Change "torque charge" to "torque change" at the end of the first sentence after the second bullet.
- p. 146** The paragraph under "3.5 System Open-Loop Performance" should read "Normal operating conditions are defined as: ambient temperature, $70^{\circ}\text{F} \pm 10^{\circ}\text{F}$; hydraulic fluid temperature, $120^{\circ}\text{F} \pm 10^{\circ}\text{F}$; hydraulic fluid pressure; 1000 to 1500 psig ahead of flow regulator, with a maximum of 20 psig return pressure."

PREFACE

This document is the final report on the design, development, and flight test of an advanced hydrofluidic SAS. The program was conducted under Army Contract DAAJO2-72-C-0019 and was administered under the direction of the Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, with Mr. G. W. Fosdick as the Project Engineer. The work was conducted during the period 23 November 1971 through 10 June 1975.

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SECTION I

INTRODUCTION

The feasibility and advantages of using a hydraulic fluidic (hydrofluidic) control system to provide short-period stability augmentation for single-rotor helicopters has been demonstrated by actual flight tests on a number of vehicles, including the UH-1 and OH-58 helicopters. These results have warranted continued development of more advanced fluidic flight control systems and hybrid fluidic/electronic flight controls with the objective of adding other pilot relief modes to the basic SAS.

The ultimate goal of this program is to obtain a reliable, low-cost flight control system that provides pilot relief control modes. The definition and analysis of this advanced hydrofluidic stabilization system were completed in 1972 under Contract DAAJ02-71-C-0040. The result of this phase of the program was the detailed system specification presented in Appendix A of this report. The flight control system defined provides the pilot with stability augmentation in three axes over the total flight envelope, and with pitch and roll attitude hold, heading hold, and altitude hold over the normal range of cruise airspeeds (50 knots to maximum cruise). Later, during flight testing, a modification was incorporated to add air-speed hold to these modes.

The defined system design and mechanization are the result of two trade-offs: (1) between increased pilot relief and system flexibility and the desire to obtain a simple, low-cost system, and (2) once the system was defined, between fluidic or electronic mechanization of the component parts. Design features or constraints that are incorporated to reduce the complexity of the system are:

- Limiting the service flight envelope of certain system modes. This eliminated the need for system control signals into the helicopter collective axis and for switching or blending of heading control signals between the roll and yaw axes.
- Requiring the pilot to trim the aircraft to the desired flight condition prior to engaging the hold modes. This makes it possible to use just three limited authority series servo-actuators for all system modes, simplifies sensor designs, and eliminates the need for synchronization of reference signals.

- **Use of constant gain in all control modes. This eliminated the need for complex gain scheduling networks and additional sensors for scheduling parameters.**
- **Use of control panel switching for engage/disengage of modes. This eliminates the need for the stick force sensors, synchronization circuits, and engage/disengage logic associated with conventional control stick steering modes.**

The fluidic versus electronic mechanization decision was based on the criterion of using fluidic mechanizations as long as satisfactory performance could be obtained with present state-of-the-art components without compromising the complexity of the system. As a result, all of the system components with the exception of the aircraft attitude references, the control panel hardware, and the synchronization circuits added during a later system modification are fluidic.

This report presents the results of the second phase of the program, which includes: (1) the building and test in the laboratory of a flightworthy system suitable for controlling the UH-1M helicopter and (2) the installation, flight test, and evaluation of the system in a UH-1M helicopter at the Honeywell flight test facility. The system evaluation consisted of obtaining and analyzing quantitative data at representative flight conditions for selected inputs and steady-state flight and qualitative evaluation by the contractor's pilot and two Government test pilots.

SECTION II SYSTEM DESIGN

DESIGN OBJECTIVE

The system was designed to provide a highly reliable, low-cost automatic flight control system (AFCS) incorporating pilot relief modes.

AFCS modes aid the pilot in four ways:

- The AFCS helps the pilot maintain precise aircraft control through stability augmentation and attitude hold during reconnaissance and armament firing missions.
- The AFCS relieves the pilot of the routine task of maintaining a given flight path such that he can concentrate on the overall mission task. Attitude hold and heading hold modes are typical examples.
- The AFCS can relieve pilot fatigue by providing such modes as altitude hold or airspeed hold.
- The AFCS can perform certain tasks significantly better than the pilot can.

As increasing the degree of pilot relief and system flexibility is generally in contrast to keeping the system simple and low cost, trade-offs had to be made. The AFCS control modes and features that are included are those that are of a major benefit to the pilot for typical observation and transportation missions performed by the UH-1 helicopter. The final selection of the system design was based on the use of the mode or feature, the frequency of use, and the functional and performance limitations of a hydrofluidic mechanization. The system design mechanized and flight tested on this program was defined and analyzed on an earlier design study performed for the Eustis Directorate of the U. S. Army Air Mobility Research and Development Laboratory under Contract DAAJ02-71-C-0040. The results of that program are presented in USAAMRDL Technical Report 72-46, "Advanced Hydrofluidic Stabilization System."

Features incorporated in the system to reduce cost and to increase reliability are:

- Use of control panel switching for system engage/disengage, mode selection, and maneuver commands
- Use of aircraft standard display gyros to obtain attitude and heading references

- Use of three limited-authority, series servoactuators for all system output motions
- Limitation of the service flight envelope of certain system modes

MODES OF OPERATION

Selection of the modes of operation for the system was based on an analysis of mission tasks and the free-vehicle handling qualities of the UH-1-type helicopter, keeping in mind the guidelines of simplicity and low cost.

The hydrofluidic flight control system is capable of controlling the helicopter in the following modes:

- Stability augmentation (three axis)
- Attitude hold (pitch and roll)
- Heading hold and heading select
- Altitude hold
- Airspeed hold

The flight envelope for operation of the various system modes is based on the flight envelope of the UH-1 during typical observation and transportation missions and the desired use of the modes during the various segments of the mission profile. In general, the baseline system is intended for operation over the range of airspeeds and altitudes at which the UH-1 is flown the majority of the time. Information obtained on the typical flight profiles for the UH-1-type helicopter indicated that over 90 percent of the flying was done at airspeeds above 50 knots and at less than 6000 feet density altitude. The following service flight envelope was then defined for the system:

- Airspeed: Stability augmentation - Hover to maximum cruise
Attitude hold, heading hold, altitude hold, and - 50 kn to maximum cruise airspeed hold
- Altitude: All modes - 0 to 6000 ft

SYSTEM DESCRIPTION

The system consists of five modes of operation: SAS, attitude hold, heading hold, altitude hold, and airspeed hold. Block diagrams of the system's three control axes are shown in Figures 1, 2 and 3. There are various interlocks between each of these modes. The SAS mode becomes operable when the system is engaged by energizing the three series servoactuators with aircraft hydraulic power. The other individual modes or combinations of modes can then be selected. These modes operate through the same three series servoactuators (pitch, roll, and yaw) as the SAS. The need for system inputs into the aircraft's collective control is eliminated by limiting the service flight envelope of these modes to normal aircraft cruise conditions, which also makes it possible to eliminate the need for switching or blending of control signals between the yaw axis and the roll axis when in the heading hold mode.

The system has three additional features: (1) turn control, by which a constant rate of turn can be introduced using a turn control knob on the control panel; (2) manual trim, using either the trim knobs on the control panel or the trim switch on the stick; and (3) synchronization (automatic trim). The synchronization feature along with the stick trim switch was added later in the program to simplify the trimming requirements of the system. The synchronizer circuits, shown in Figures 1 and 2, were mechanized and tested in the laboratory with good results prior to installation in the helicopter, but an electromagnetic interference (EMI) problem was encountered upon installation. Simple filtering techniques proved to be unsuccessful; therefore, the synchronization feature was not flight tested.

Normal operating procedure is to engage the SAS prior to takeoff, and upon reaching the desired cruise flight conditions, the attitude hold modes (pitch and/or roll) are engaged, thus controlling the aircraft to local vertical. The pilot then may select the additional modes (heading hold, altitude hold, and airspeed hold) depending on the desired flight path. For example, the system will assist the pilot in (1) flying at a fixed altitude while maneuvering the aircraft, such as in holding over a fixed ground location; (2) flying in a fixed direction at a constant airspeed while varying altitude, such as in making instrument-type approaches; or (3) using both heading hold and altitude hold or airspeed hold together to provide pilot relief for long flights from point to point. Because of their incompatibility, the altitude hold mode and the airspeed hold mode can not be used simultaneously.

The remainder of this section describes the individual modes of the system and the synchronization feature. A more detailed description of the system and its performance is presented in the system specification, Appendix A; pilot operating instructions for the system are presented in Appendix B.

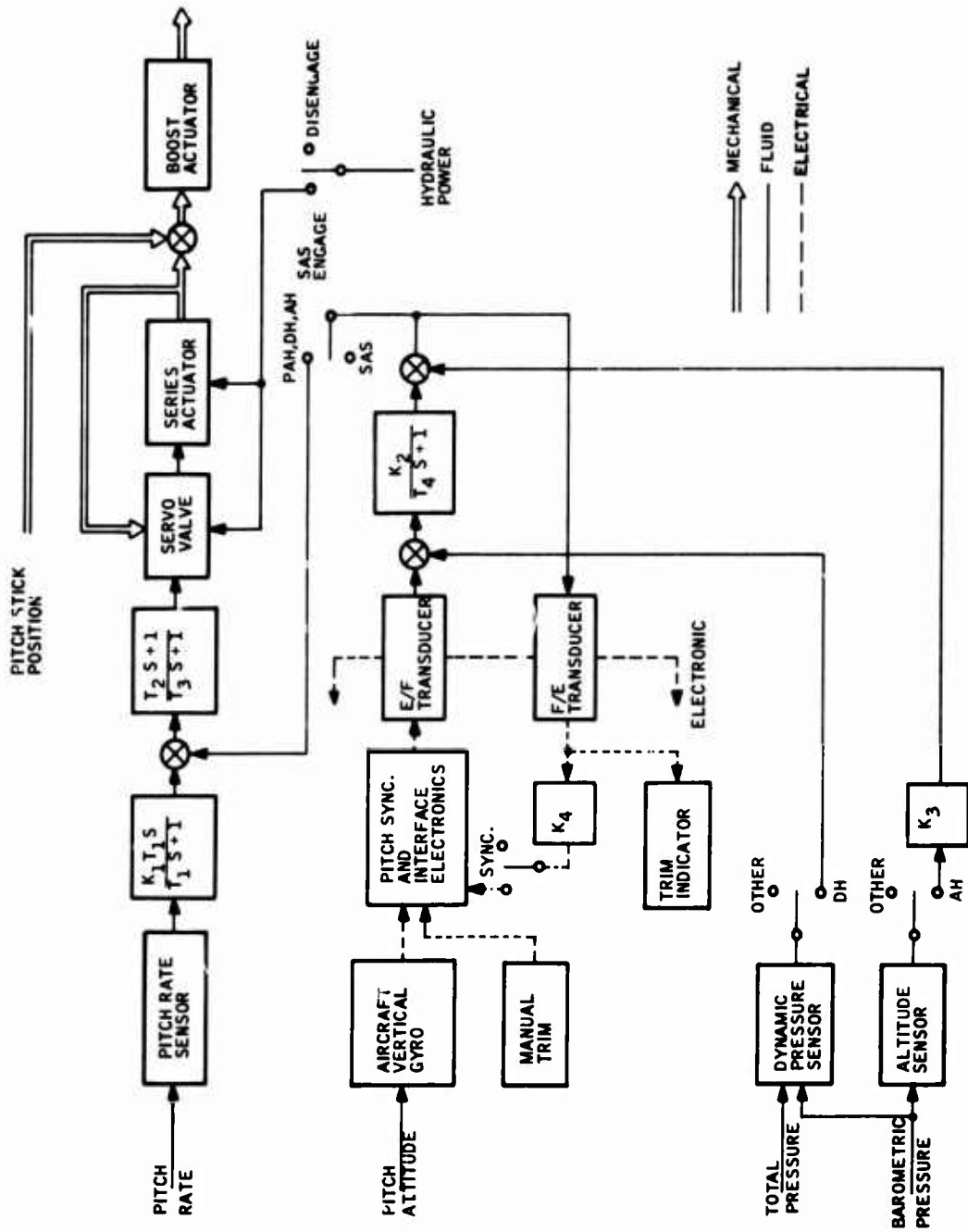


Figure 1. Pitch Axis Functional Block Diagram

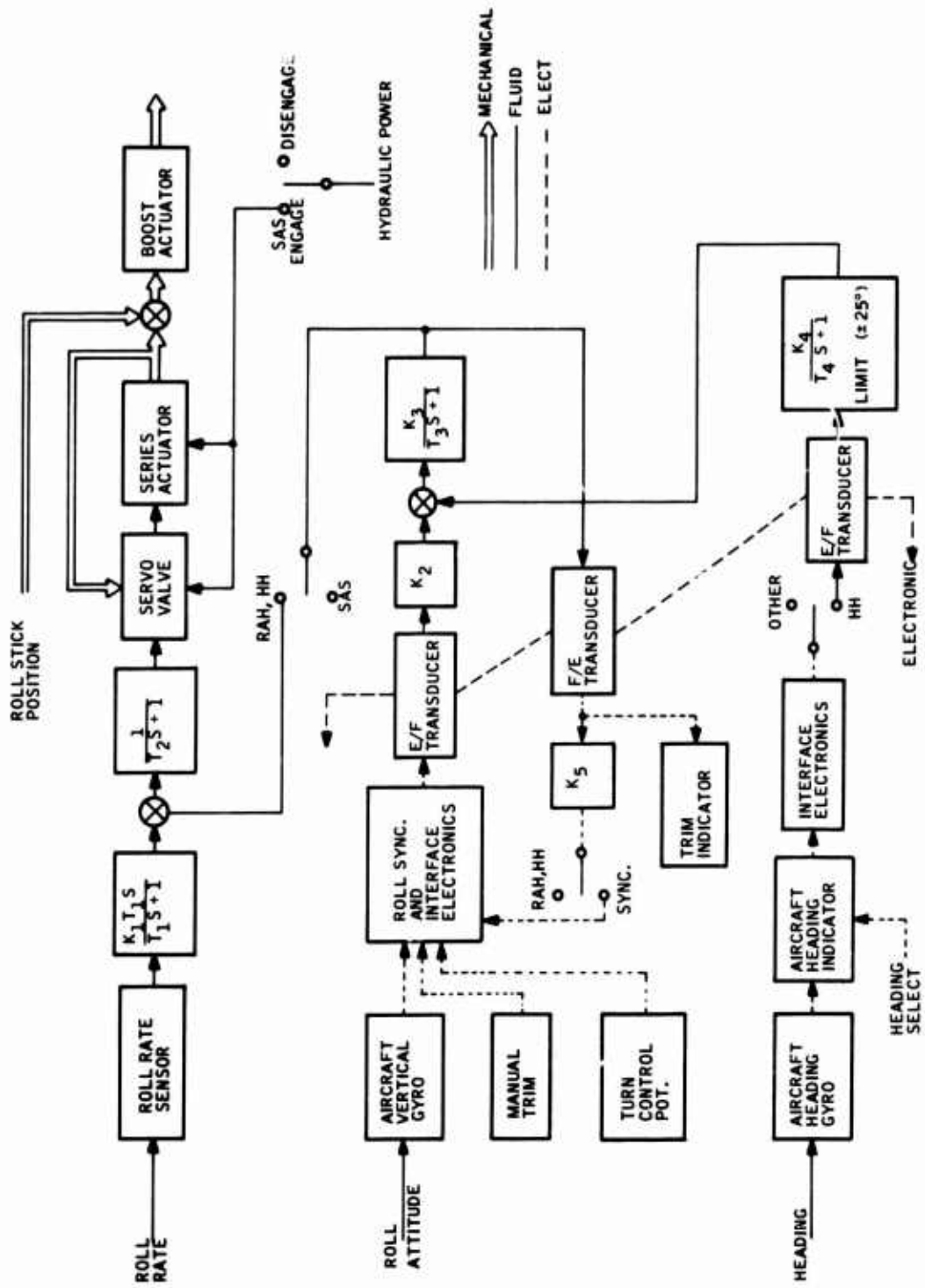


Figure 2. Roll Axis Functional Block Diagram

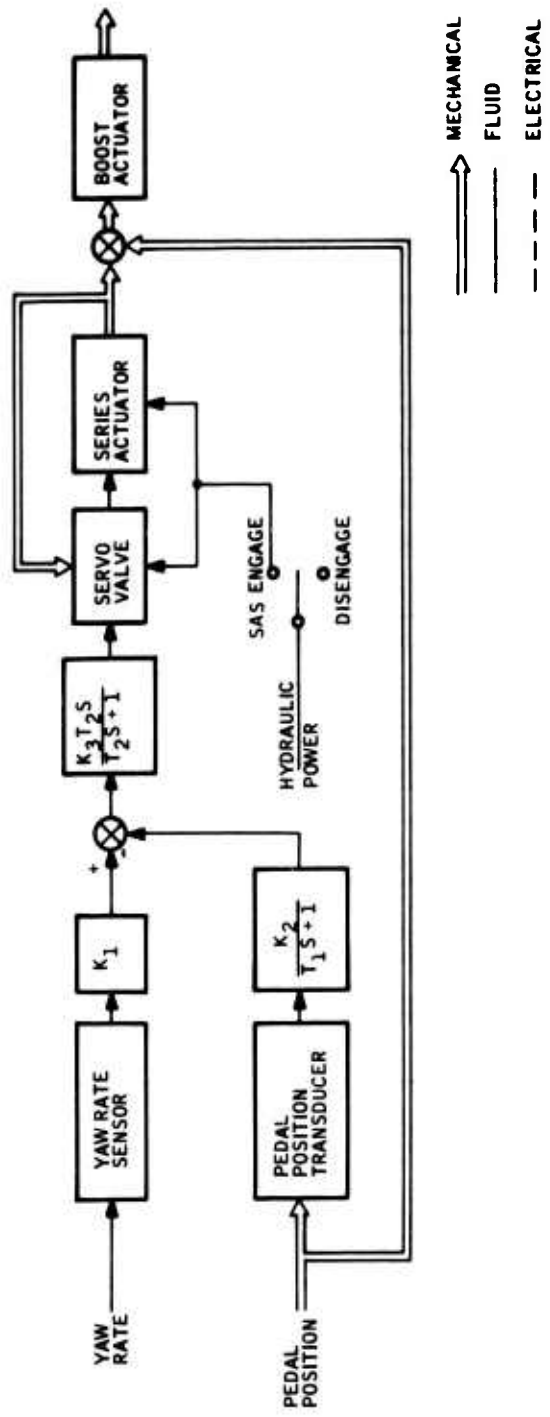


Figure 3. Yaw Axis Functional Block Diagram

Stability Augmentation System

The SAS consists of a vortex rate sensor in each axis, the necessary gain and shaping networks, an engage switch in the flight controller, and a solenoid valve and a series servoactuator in each axis. The servoactuators are mounted directly into the aircraft flight control linkage. In the yaw axis (Figure 3), a flight control pedal transducer signal with the proper gain and shaping is summed with the rate sensor signal to give the pilot complete authority at all times. This in essence cancels the rate feedback resulting from aircraft motion due to the pilot's input command. Damping of external inputs is still available at all times. The SAS mode is energized by a switch on the console-mounted control panel, which opens the solenoid valve supplying power to the series servoactuators.

Attitude Hold Modes

The attitude hold modes consist of a vertical gyro, gain and shaping networks, trim indicators, roll and pitch trim controls, a turn command control, engage valves, and an engage switch. The output of the roll attitude channel is summed with the roll SAS channel signals through a shaping network into the roll series servoactuator. The output of the pitch attitude channel is summed with the pitch SAS channel signals in a similar manner. In operation, the pilot comes to the desired attitude, trims the aircraft through the cyclic stick, and if desired, engages the stick force trim. He then checks and nulls, if necessary, the pitch and/or roll attitude control loops by monitoring the trim indicators while operating the pitch and/or roll trim controls. He then engages the attitude hold modes with the mode select switches. If a turn is desired, it can be commanded either with the stick or with the turn control knob on the function selector, which commands roll attitudes up to approximately 15 degrees. If a trim attitude change is required due to a change in flight conditions, the pilot disengages the attitude hold mode, re-trims the aircraft, checks the attitude control loop nulls, and reengages the modes. With the synchronization feature added, as shown in Figures 1 and 2, manual trimming of the system prior to engaging these modes is eliminated.

Heading Hold Mode

The heading hold mode consists of a heading gyro, gain networks, a heading select control, an engage valve, and a mode select switch. The heading channel signal is summed with the attitude channel signal and the SAS channel signal of the roll axis. This limits the heading hold mode to airspeeds above approximately 50 KIAS, due to the inability to satisfactorily hold heading through the roll axis at low air speeds.

To use the heading hold mode, either the pilot can fly to the desired heading, adjust the heading select knob to the existing heading, and engage the mode; or he can simply turn the heading select knob to the desired heading, engage the mode, and allow the control system to fly the aircraft to the selected heading. The system will then maintain the aircraft on the selected heading until the mode is disengaged or a new heading is commanded.

Altitude Hold Mode

The altitude hold mode consists of an altitude error sensor, which senses changes in the static pressure, amplification, and a mode select switch. The altitude error signal is summed with the pitch attitude and pitch SAS signals. As in the case of heading hold, this mechanization of altitude hold (through the pitch axis only) limits the effective use of this mode to airspeeds above approximately 50 KIAS.

To use the altitude hold mode, the pilot flies to the desired altitude, trims the aircraft, and engages the mode. The system will then maintain the aircraft at the selected altitude until the mode is disengaged. When an altitude change is desired, the pilot disengages the mode, flies to the new altitude, and reengages the mode.

Airspeed Hold Mode

The airspeed hold mode is similar to the altitude hold mode except that the sensor in this case senses changes in dynamic pressure rather than static pressure. The dynamic pressure error signal is summed with the pitch attitude and pitch SAS signals. The control panel engage logic is designed such that the pitch attitude and pitch SAS modes must be engaged and the altitude hold mode disengaged before the airspeed hold mode can be engaged.

To use the airspeed hold mode, the pilot establishes the desired airspeed and engages the mode. The system will then maintain the aircraft at the selected airspeed until the mode is disengaged. If an airspeed change is desired, the pilot disengages the mode, establishes the new airspeed, and reengages the mode.

Synchronization

The synchronization feature was added to the pitch and roll axis channels to simplify the task of trimming the attitude hold modes prior to engaging the system.

The original attitude hold mode mechanization included a manual trim function that was operated by the pilot to trim the attitude signal to neutral prior to attitude hold engage. The amount of attitude mistrim was determined from a differential pressure sensing trim indicator. Figure 4 is a block diagram that shows the basic configuration.

An automatic synchronizer was needed to eliminate the manual trimming function prior to engagement. It was determined that the synchronization function could be added with only minor modifications to the existing system hardware; however, the modified system would still require manual disengagement of the attitude hold mode prior to flight changes and reengagement when the new flight condition was attained.

The synchronization point is the fluidic output that supplied the original trim indicator. Figure 5 is a block diagram of the attitude input with a synchronizer loop. The trim indicator has been changed to an electrical indicator.

SERIES SERVOACTUATOR AUTHORITY

Because the series servoactuators are used for the outer-loop pilot-assist modes as well as the inner-loop SAS, an analysis was performed during the earlier design study program to define the required servoactuator displacement limits. Three factors were taken into account in selecting the desired servoactuator displacement limits: (1) series servoactuator range requirements for normal operation, (2) aircraft angular rates resulting from a servoactuator hardover failure, and (3) the effects of the series servoactuator hitting the limit during severe maneuvers or disturbances. The following is a summary of these results.

Range Requirements

The main rotor swash plate and tail rotor control limits for the UH-1 are:

Roll axis = ± 10.0 deg

Pitch axis = ± 13.75 deg

Yaw axis = ± 13.0 deg

The range requirements for the roll, pitch, and yaw series servoactuators are shown in Table 1. These values are calculated by combining the displacements resulting from maneuvers, wind gusts, engage transients, trim changes, and component tolerances. Maneuver, wind gust, and engage transient displacements were obtained from the computer simulation of system operation, and displacements from trim changes and component tolerances were calculated based on aircraft data and component null-shift data. The displacements were separated into transient and

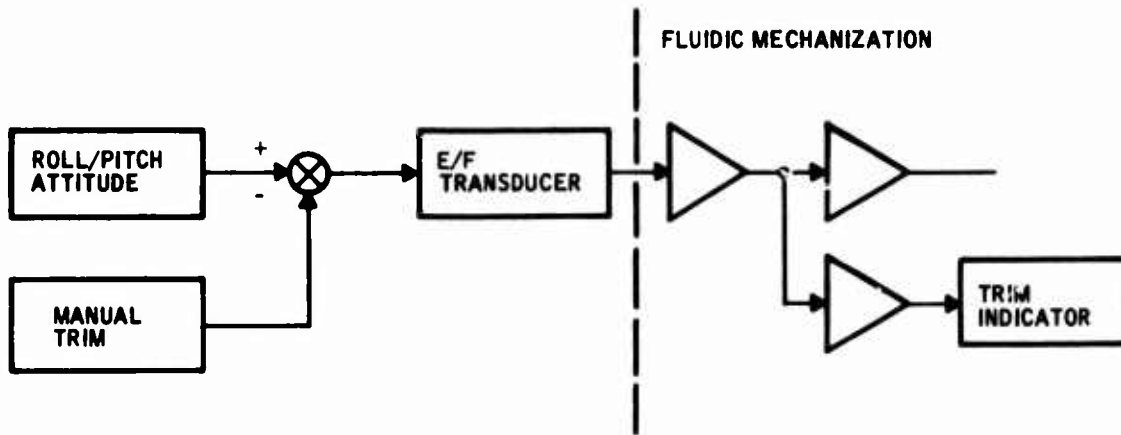


Figure 4. Attitude Input Block Diagram

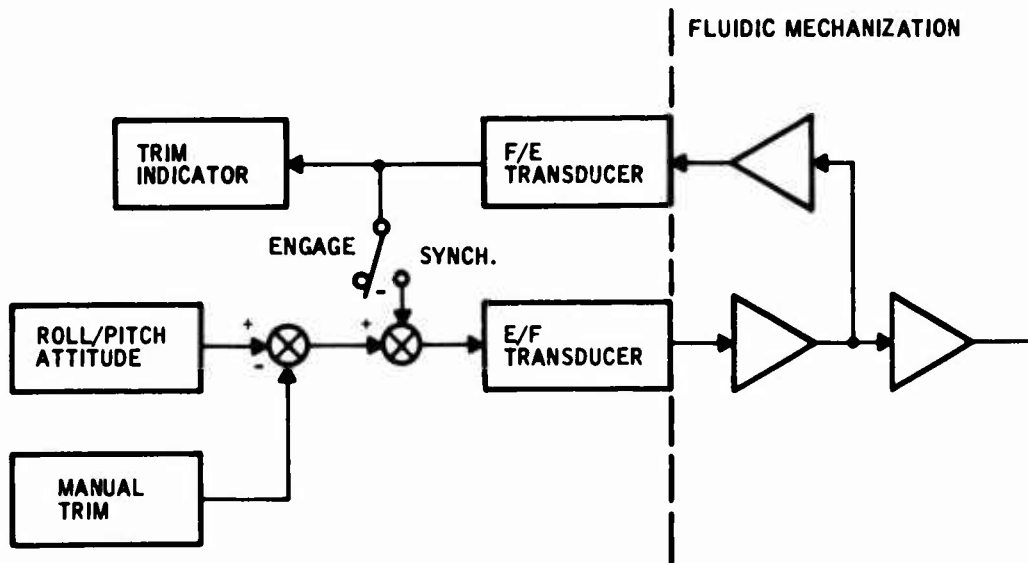


Figure 5. Attitude Synchronizer Block Diagram

Table 1. Series Servoactuator Range Requirements

Cause of Displacement	Displacement (deg swash plate/control)	
	Transient	Steady State
Roll Servoactuator		
1. Turn maneuver (15-deg roll angle)	0.75	0
2. Lateral gust (20 ft/sec)	0.80	0
3. Engage transient	0.10	0
4. Trim changes	0	0
5. Component tolerances	0	0.24
	1.10 (RSS)	0.24
	1.34 deg = 13.4% total	
Pitch Servoactuator		
1. Turn maneuver (15-deg roll angle)	0.20	0
2. Lateral gust (20 ft/sec)	0.40	0
3. Vertical gust (10 ft/sec)	0.65	0
4. Engage transient	0.30	0
5. Trim changes (2 hours fuel consumption and ±20 knots airspeed change)	0	0.52
6. Component tolerances	0	0.54
	0.84 (RSS)	1.06
	1.90 deg = 13.8% total	
Tail Rotor (Yaw) Servoactuator		
1. Turn maneuver (15-deg roll angle)	0.15	0
2. Lateral gust (20 ft/sec)	1.20	0
3. Component tolerances	0	0.24
	1.21 (RSS)	0.24
	1.45 deg = 11.2% total	

steady-state displacements. The transient values were evaluated by the root sum of the squares technique and added to the steady-state values. The total displacement required was then obtained by adding the transient and steady-state totals. All displacements are shown in terms of degrees of swash plate/control. The total displacements for the three axes are then divided by their respective swash plate/control limits to obtain servoactuator authority required in terms of percentage of control surface motion. As shown in Table 1, the required authorities for the three axes are:

Roll servoactuator = 13.4%

Pitch servoactuator = 13.8 %

Yaw servoactuator = 11.2%

Failure Effects

Servoactuator failures were simulated on the computer to determine the resulting helicopter angular rates. Servoactuator failures were assumed to be ramp control linkage displacements at a maximum servoactuator slew rate of 10 inches/second. The simulation results are presented in Table 2 for both 15 percent series servoactuator authority and for the authority actually used in the system (FSAS). The authorities of these servoactuators, which were available from the three-axis fluidic SAS program (Contract DAAJ02-70-C-0017), are:

Pitch - 18.2%

Roll - 25%

Yaw - 19.3%

The simulation results were compared with flight test results obtained for the UH-1C without the stabilizer bar and without the SAS operating, when a step stick input of equivalent swash-plate angle was applied. This information was taken from the final report for Contract DAAJ02-70-C-0017, USAAMRDL Technical Report '71-34, "Three-Axis Fluidic Stability Augmentation System Flight Test Report". In general, the flight test results almost matched the simulation results. Where differences occurred, the flight test results showed generally lower angular rates. This is attributed to other factors, such as wind, which were present during the flight test.

Table 2. Angular Rates for Series Servoactuator Failure (Degrees/Second)

Airspeed	Servo-actuator Authority	Pitch		Roll		Yaw	
		At 1.0 sec	Peak	At 1.0 sec	Peak	At 1.0 sec	Peak
Hover	15%	10.7	20.7	18.0	18.7	17.0	30.7
	FSAS	13.0	25.1	30.0	31.2	21.9	39.6
80 knots	15%	6.0	6.0	13.5	14.6	7.0	7.3
	FSAS	7.4	7.4	22.5	24.4	9.0	9.4

Series servoactuator authorities for the FSAS were: Pitch - 18.2%, roll - 25.0%, and yaw - 19.3%.

Effects of Servoactuator Hitting Limits

A computer simulation study was made to investigate aircraft motions for commands and disturbances that saturate the series servoactuators. Time responses were recorded for the various flight control modes with servoactuator authority limits set at 10 percent of full swash plate/control travel. In general, relatively large disturbances were required to saturate the servoactuators. When a servoactuator did saturate, the response became that of the free aircraft, except that the peak rate was reduced by the limited action of the saturated servoactuator. These results indicate that no serious performance degradation occurs for transient saturation of the series servoactuators as a result of severe disturbances and commands.

Based on this analysis, an authority of at least 15 percent was recommended for the roll, pitch, and yaw series servoactuators. As servoactuators from the earlier three-axis FSAS program were available, these units were selected for use with the Advanced Hydrofluidic Stabilization system.

SECTION III SYSTEM MECHANIZATION

The system consists of five major assemblies: the hydrofluidic package, which contains all of the fluidic components; the control panel, which includes the interface electronic circuits; and the three series servoactuators, which mount in the helicopter control linkages. Figure 6 shows the hydraulic and electrical interconnection of these assemblies. A more detailed description of these assemblies is presented in this section.

HYDROFLUIDIC PACKAGE

Figure 7 shows the hydrofluidic package with cover installed, and Figure 8 shows the assembly with the cover removed. The package has two levels: the lower level contains the SAS engage valves and the hydraulic power conditioning equipment, and the upper level contains the fluidic control components.

Figure 9 is a picture of the package lower level showing the primary solenoid valve, the three SAS solenoid valves, a pressure regulator that reduces the aircraft supply pressure to 1000 psi for the servoactuators, a filter, the fluidic system flow control valve (temperature schedule), and a back-pressure regulator to isolate servoactuator-induced return line surges from the fluidic controllers. The package upper level, which can be separated from the lower level for testing, is shown in Figure 10. The dynamic pressure sensor, which was a later addition to the system, is shown mounted on top of the altitude sensor in the right foreground of the picture. Figure 8 shows the package before the addition of the dynamic pressure sensor.

All hydraulic connections to the package are made through fittings on one end, and electrical connections are made through a signal connector on the same end. The flexible cable, which transmits the pedal motion from the pedals to the position transducer on the yaw controller, is shown extending from the package in Figure 7.

The control components are divided between the controllers for the three axes. Hydraulic power to the three controllers is provided in series to minimize total system flow consumption. Following is a description of these subassemblies.

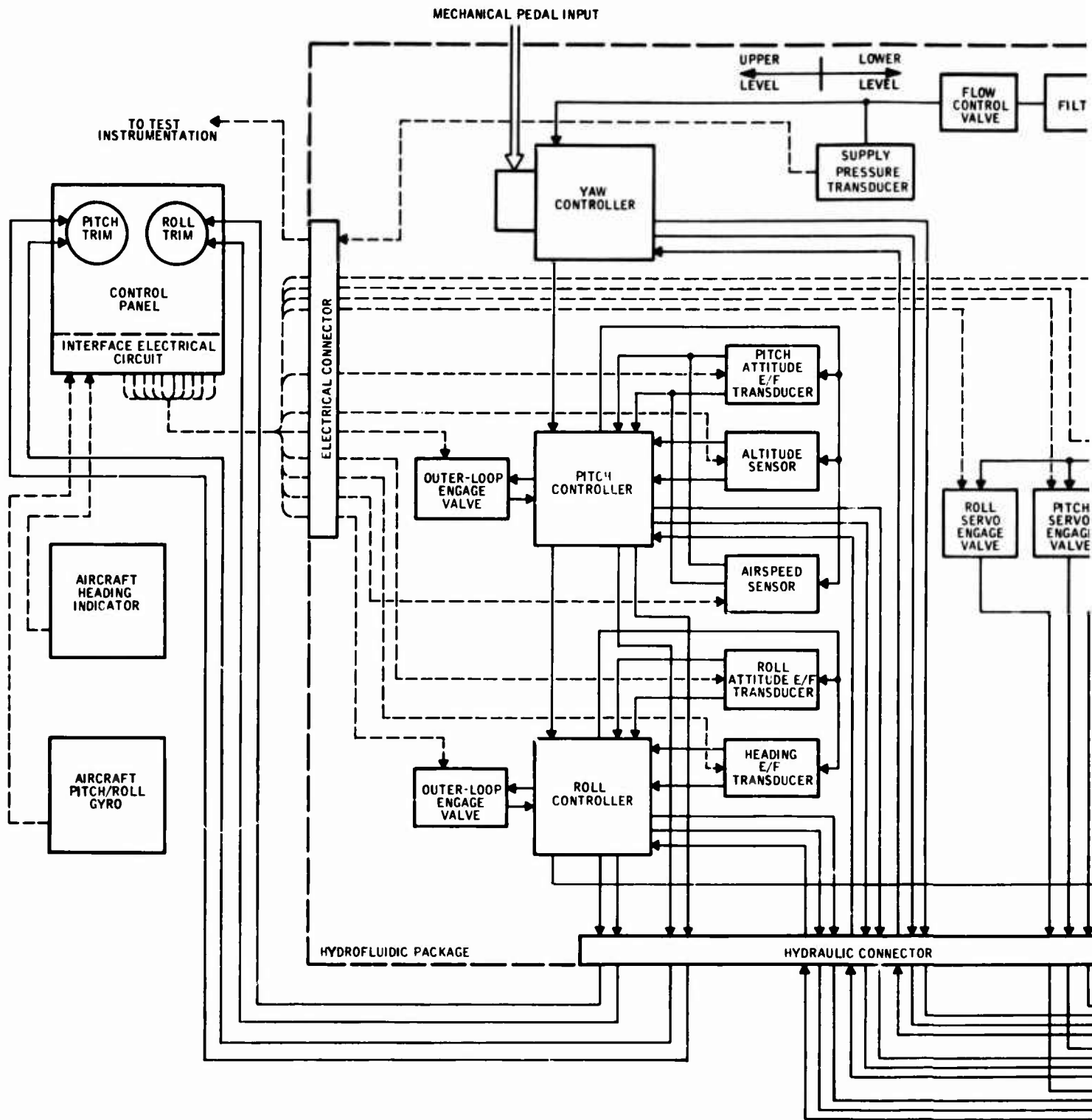
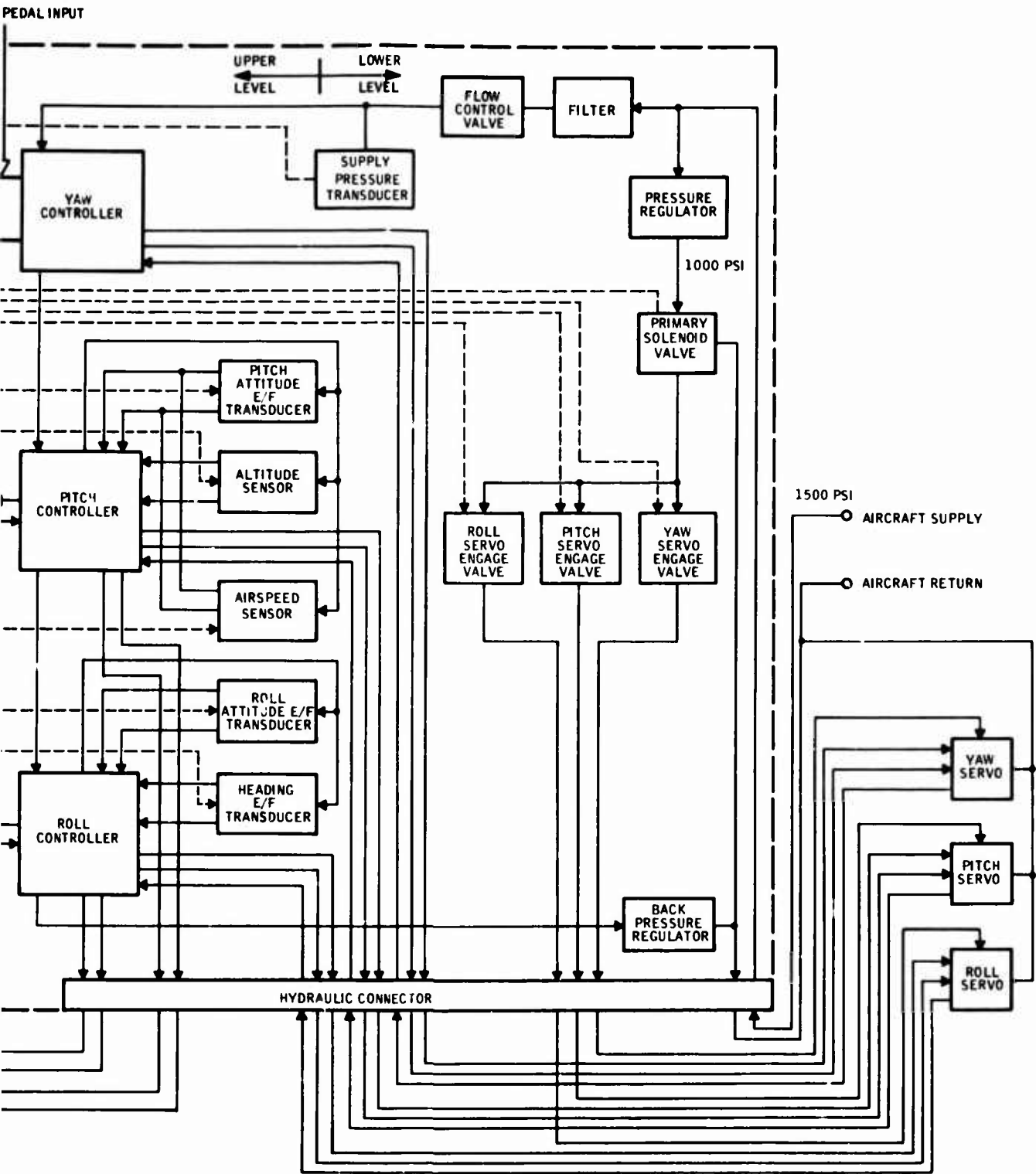


Figure 6. System Interconnection Diagram

H



on Diagram

T-3

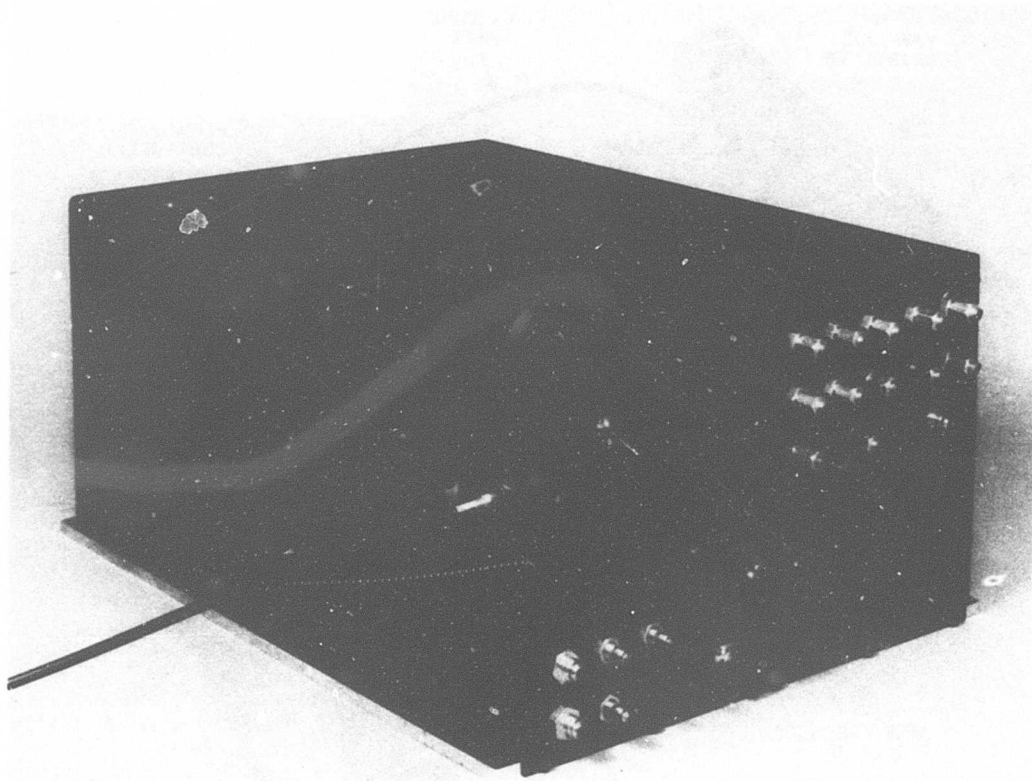


Figure 7. Hydrofluidic Package (Cover Installed)

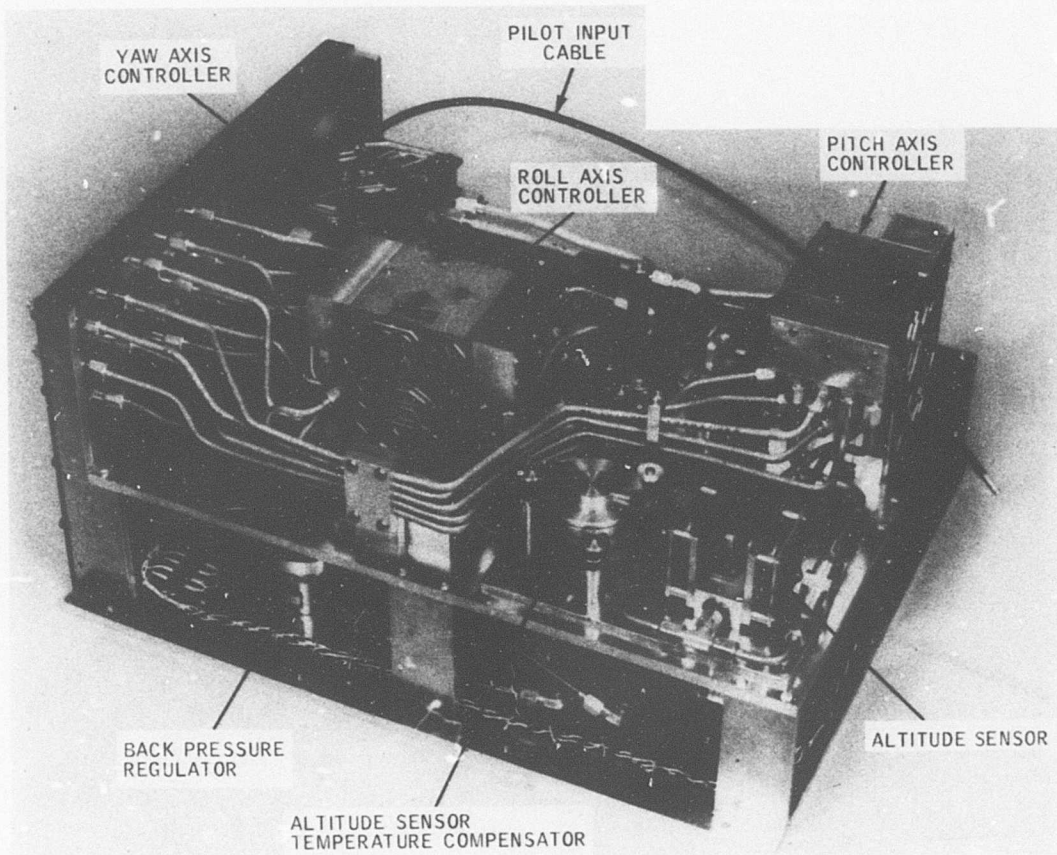


Figure 8. Hydrofluidic Package (Cover Removed)

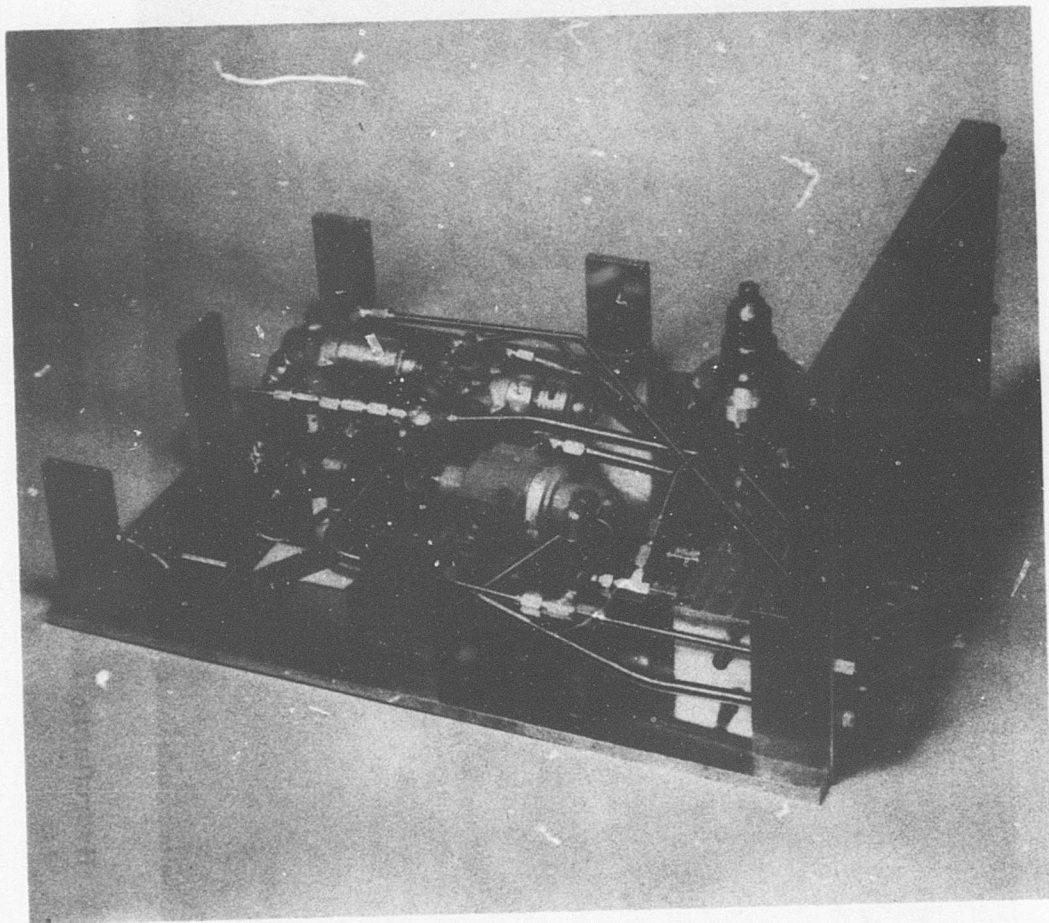


Figure 9. Hydrofluidic Package Lower Level (SAS Engage Valves and Power Conditioning Components)

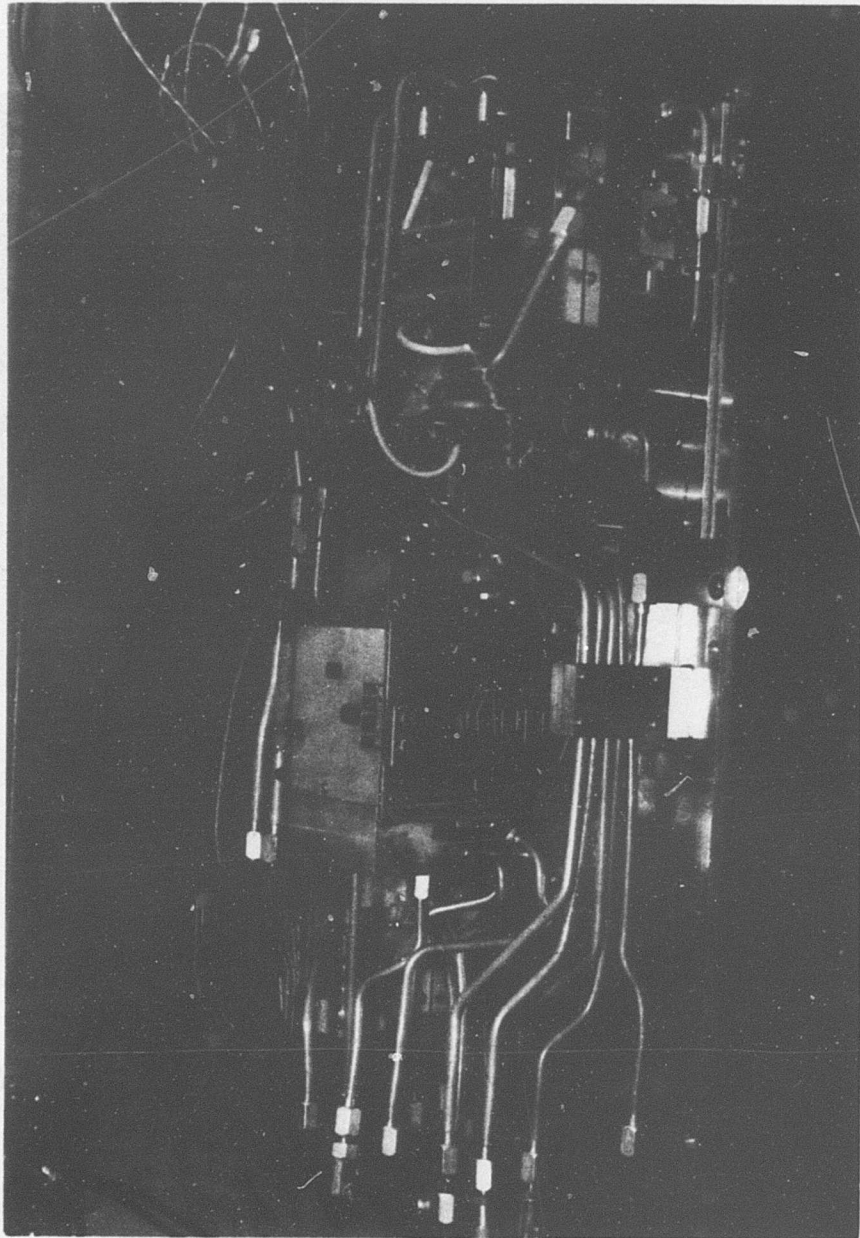


Figure 10. Hydrofluidic Package Upper Level (Control Components)

Yaw Controller

A schematic diagram of the yaw controller circuit is shown in Figure 11, and a picture of the subassembly in Figure 12. The only mode operating through the yaw axis is the yaw SAS. This mechanization with the pedal displacement input is sometimes referred to as a yaw SCAS (stability and control augmentation system). The yaw controller subassembly includes a vortex rate sensor, a pedal displacement transducer, and an amplification and dynamic shaping (high-pass) circuit. A description of the individual components of the system is presented in Section IV of this report, and performance data are included in Section V.

Pitch Controller

A schematic diagram of the pitch controller circuit is shown in Figure 13, and a picture of the controller with the altitude error sensor and electric-to-fluidic (E/F) transducers in Figure 14. The pitch SAS, pitch attitude hold, altitude hold, and airspeed hold modes operate through the pitch axis.

The pitch attitude hold mode is engaged by closing the solenoid valve in the amplifier circuit, and the mode is disengaged by opening the solenoid valve, thereby shorting out the signal at that point. In the original manual trim mechanization of the system, a trim indicator, which is a differential pressure gauge, was included upstream of the solenoid valve to provide a monitor of the circuit null. Later in the program, the pressure transducer, which is part of the synchronization circuit, was substituted for the trim indicator.

The altitude sensor and the dynamic pressure sensor are both fluidic devices that provide an output signal proportional to the difference between the reference altitude/airspeed and the actual value. A double bellows driving a flapper-nozzle device is used to convert the differential air pressure to a hydraulic pressure signal. Both the altitude and airspeed modes are engaged by closing solenoid valves within the respective sensors that trap the reference pressure(s).

A description of the individual components is presented in Section IV of this report. Figure 15 shows input/output curves for the pitch attitude, altitude, and airspeed loops, where the input signals are dc voltage to the E/F transducer for pitch attitude and the pneumatic pressure difference for altitude and airspeed, and where the output is differential pressure to the servoactuator. Frequency response data for the pitch controller loops are presented in Section V.

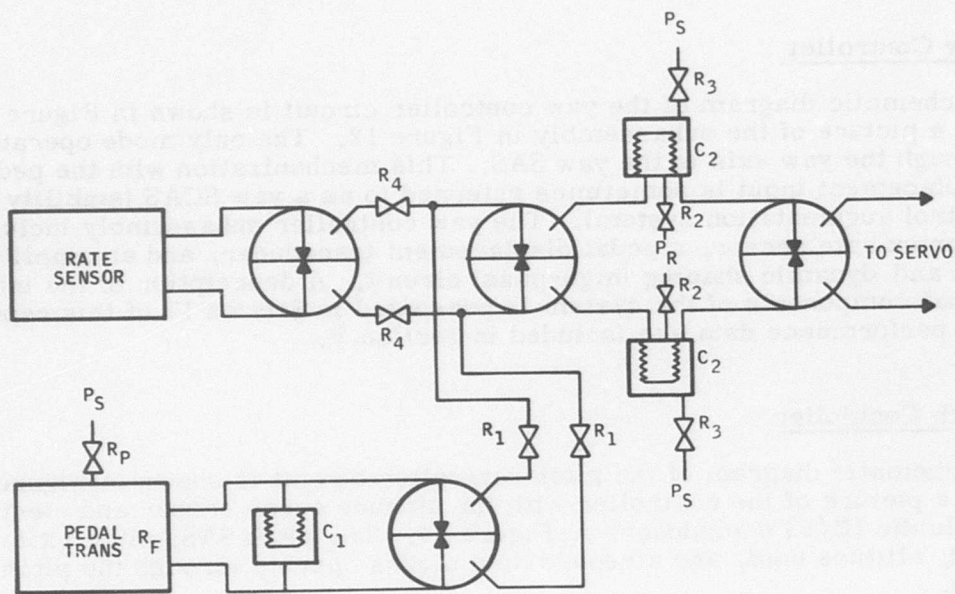


Figure 11. Yaw Controller Schematic

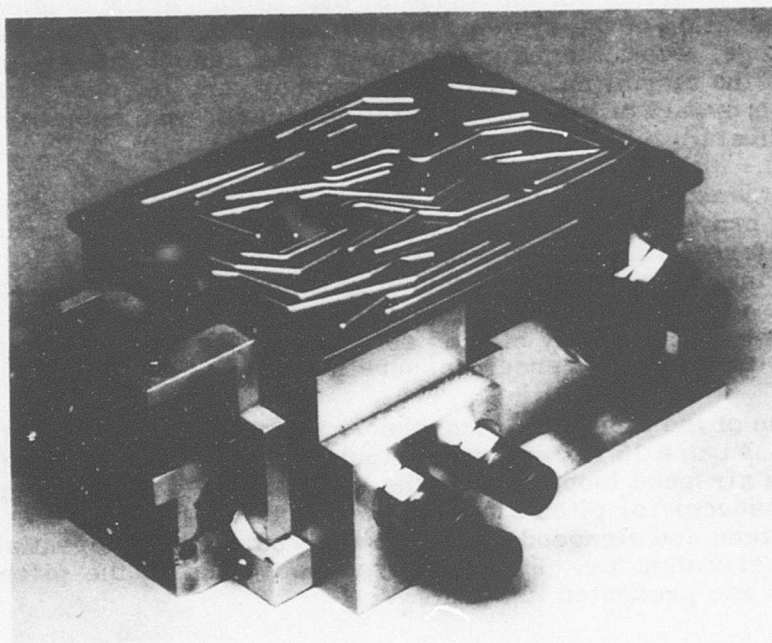


Figure 12. Yaw Controller Assembly

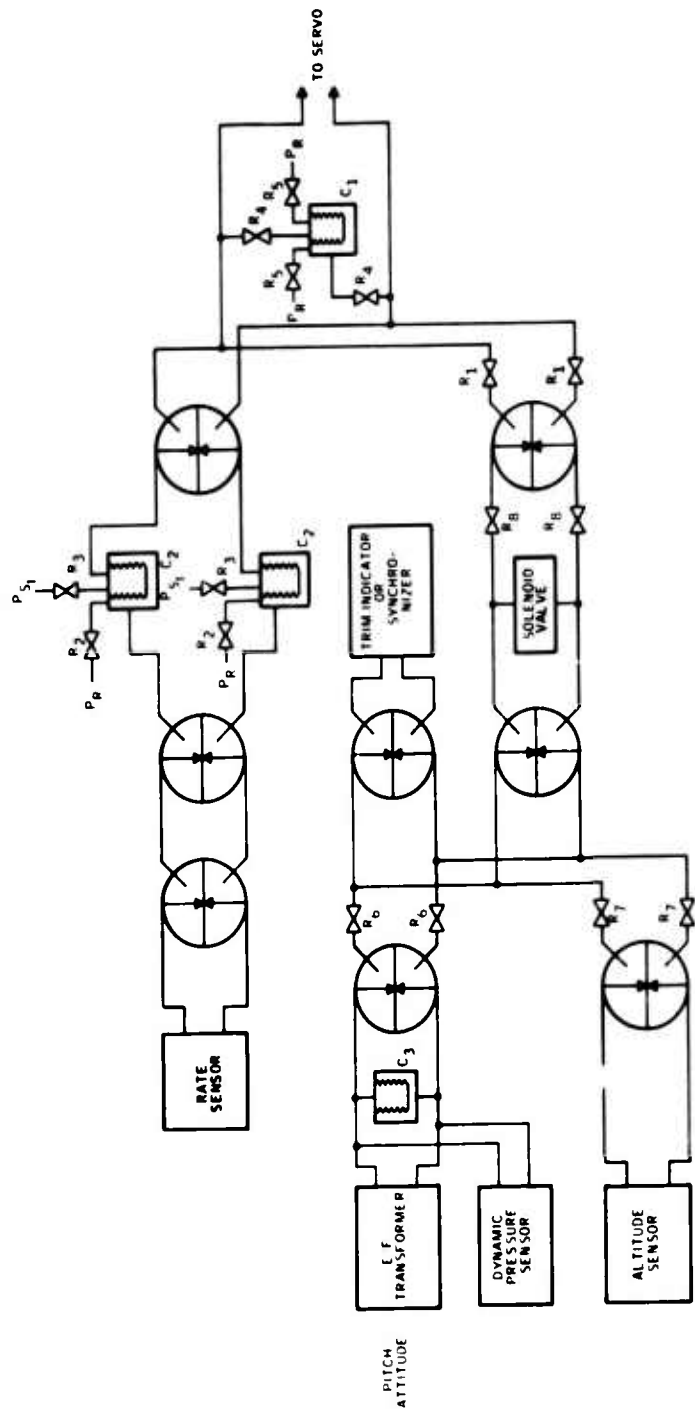


Figure 13. Pitch Controller Schematic

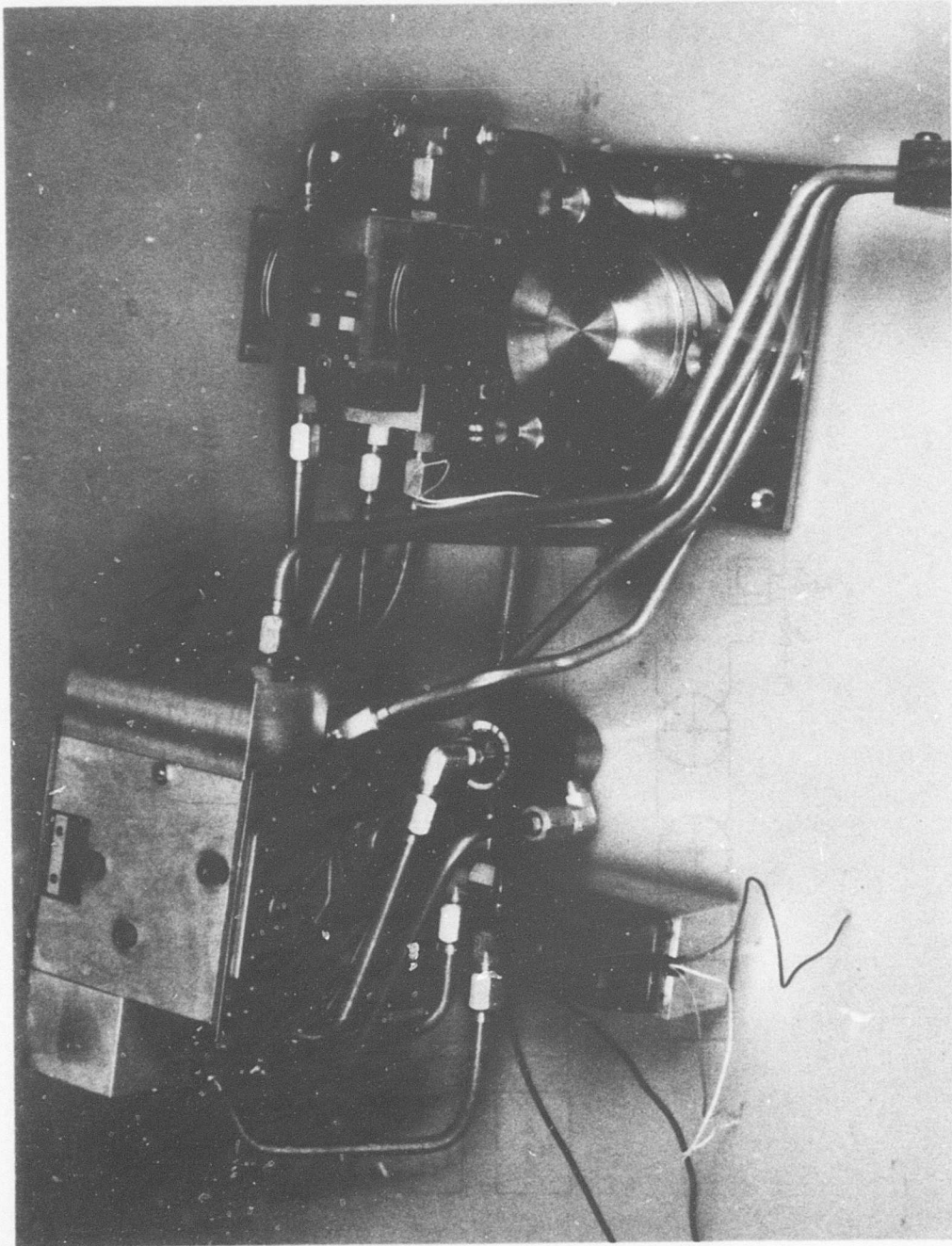


Figure 14. Pitch Controller with Altitude Sensor

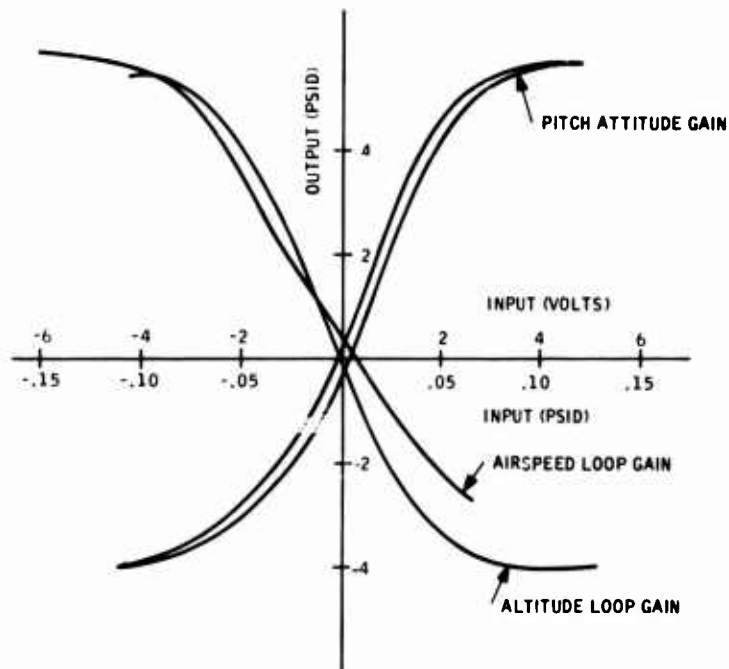


Figure 15. Pitch Axis Controller Performance

Roll Controller

Figure 16 is a schematic diagram of the roll controller, and Figure 17 is a picture of the controller with the E/F transducers attached. The roll SAS, roll attitude hold, and heading hold modes operate through the roll axis.

The heading input limit was originally incorporated in the heading E/F transducer. Later when the synchronization feature was added, it was moved to the electronic circuit to provide more flexibility.

The roll attitude hold mechanization is similar to that of the pitch attitude hold. The mode is engaged by closing the solenoid valve in the fluidic circuit. The trim indicator is located in a similar position as in the pitch axis. An electrical switch in the interface electronic circuit is used to engage the heading hold mode.

Figure 18 shows the input/output curves for the roll attitude and heading signals, where the input signals are dc voltage to the E/F transducers and the output is differential pressure to the servoactuator. Frequency response data for the roll axis modes are presented in Section V of this report.

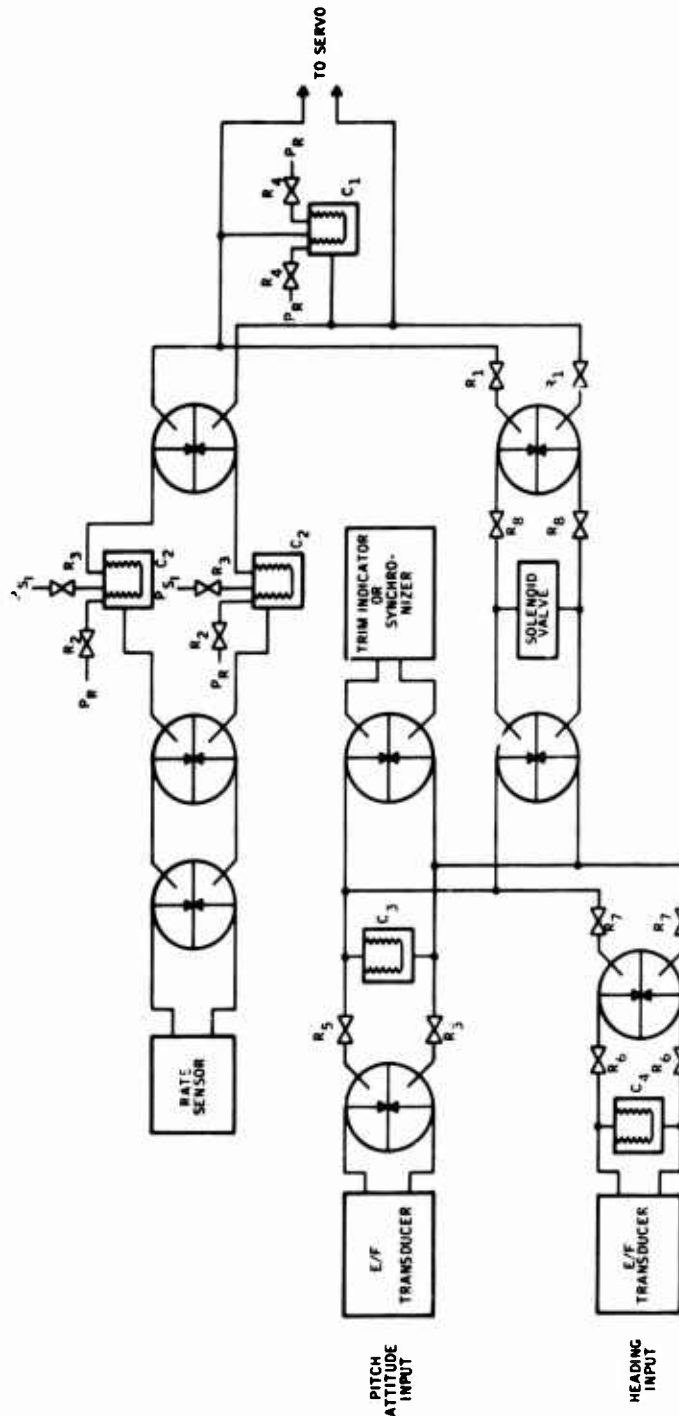


Figure 16. Roll Controller Schematic

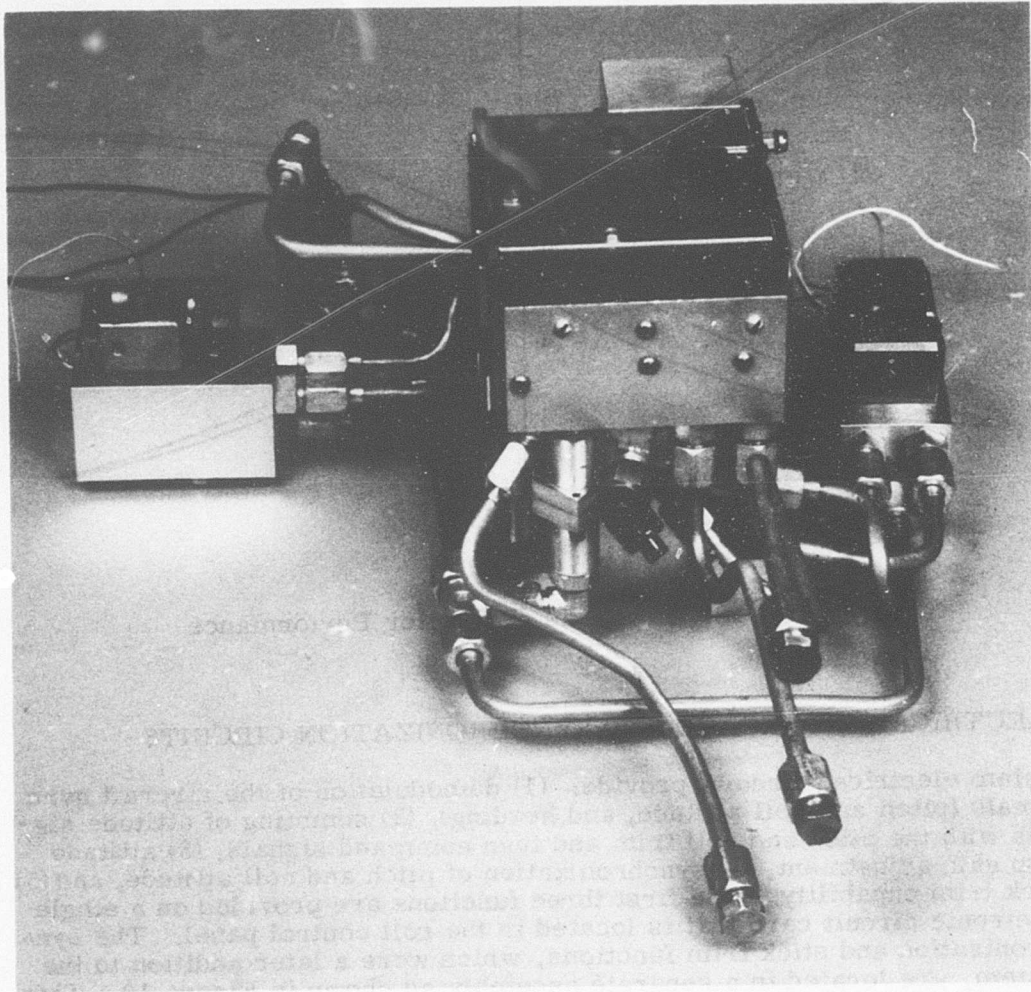


Figure 17. Roll Controller with E/F Transducers

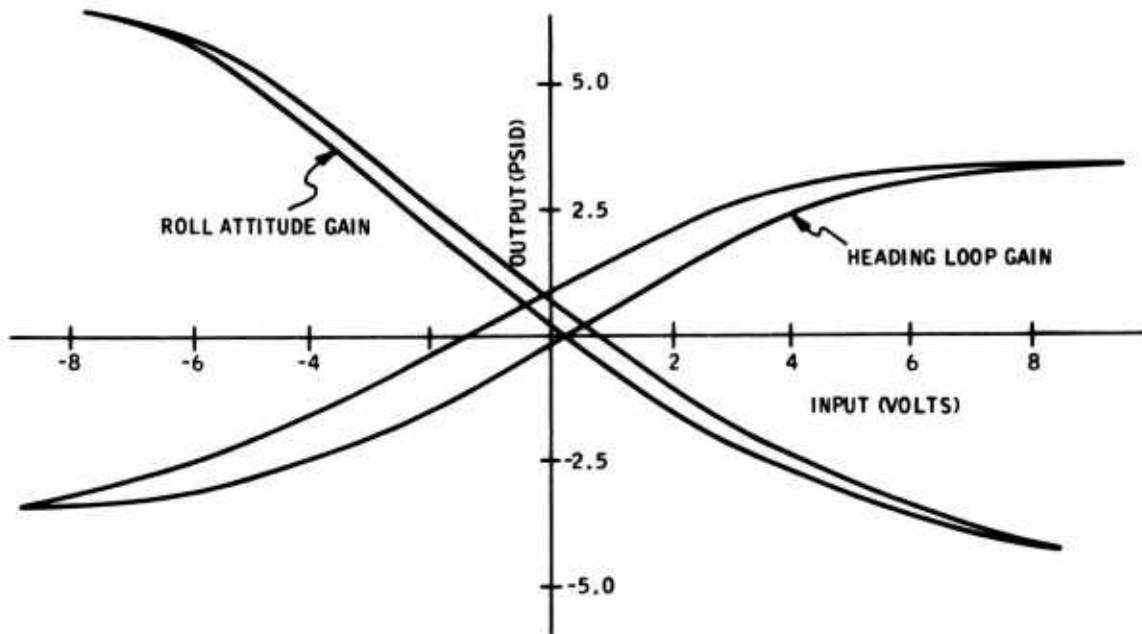


Figure 18. Roll Axis Controller Performance

ELECTRICAL INTERFACE AND SYNCHRONIZATION CIRCUITS

System electrical circuits provide: (1) demodulation of the aircraft gyro signals (pitch and roll attitude, and heading), (2) summing of attitude signals with the pitch and roll trim and turn command signals, (3) attitude loop gain adjustment, (4) synchronization of pitch and roll attitude, and (5) stick trim capability. The first three functions are provided on a single electronic circuit card that is located in the roll control panel. The synchronization and stick trim functions, which were a later addition to the system, are located in a separate assembly as shown in Figure 19. This package also contains the pressure transducers used in the two synchronizer loops.

Figure 20 shows the basic synchronizer configuration for the attitude input signals to the advanced hydrofluidic stabilization system. The demod-amp on the input is part of the original circuit configuration that was connected directly to the E/F transducer. Additional electronic circuitry has now been added to mechanize the synchronizer function. The output of the demod-amp is fed through a polycarbonate capacitor to the

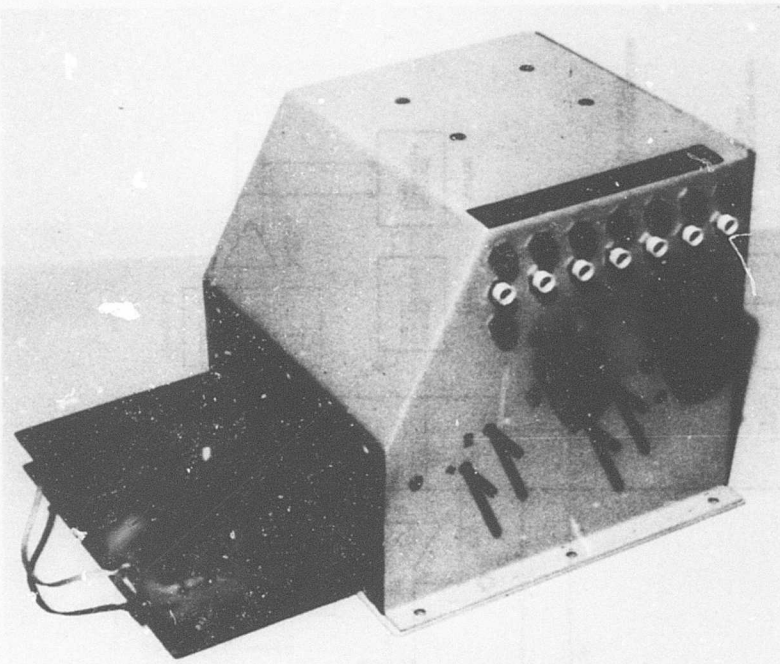


Figure 19. Synchronizer Circuit Assembly

gate of an insulated gate field effect transistor (FET). The input impedance of the gate is 1×10^{14} ohms. The FET output is then fed into an op-amp that drives the E/F transducer. The balance loop of the synchronizer takes the fluidic signal from the point in the system where the trim indicator was connected, and feeds this signal to an F/E transducer to convert it back to an electrical signal. The F/E transducer is a standard variable reluctance-type pressure transducer. The output of the transducer carrier-demodulator is further amplified in an op-amp to get the desired loop gain. The output of the op-amp is connected to the gate of the insulated gate FET when synchronizing action is taking place.

When the attitude hold mode is engaged, the output from the feedback op-amp is disconnected from the FET gate. Because of the high impedances of this circuit, the charge on the gate is maintained, and attitude error signals are coupled electrostatically through the capacitor to the FET to develop normal control error signals in the system. The limited voltage range over which the FET can operate linearly requires that the input voltage be reduced from normal op-amp output levels. The gain in the E/F transducer drive amplifier is scaled to provide the proper forward loop gain.

The synchronizer circuit using the MOS FET is a relatively new circuit concept that provides a savings in cost and size over previously used electronic synchronizers.

The stick trim circuit, which was added to the system configuration, is also shown in Figure 20. The input from the pilot's stick trim button supplies 28 Vdc to voltage divider circuits as well as a mode control relay. One signal input is inverted to produce the proper trim command into the trim integrator. The integration output is fed through an FET clamp circuit that holds the integrator output steady when on attitude hold. When attitude hold is disengaged, the trim integrator will be re-centered in 10 seconds or less.

Relay logic circuits are used for switching the synchronizer and trim circuits. The open relay contacts provide a much higher impedance than solid-state switching elements at the FET gate to minimize drift during integrator clamp or attitude hold operation.

PILOT CONTROL PANEL

Figure 21 shows the four assemblies that make up the pilot's control panel. Control panel functions are provided in two panel sections. One panel contains the pitch and power functions listed below:

- 115-V 400-Hz power switch
- 28-V dc power switch
- Pitch SAS engage switch
- Pitch attitude engage switch
- Altitude hold engage switch
- Manual pitch attitude trim control

The second panel contains the roll and yaw functions listed below:

- Roll SAS engage switch
- Yaw SAS engage switch
- Master engage switch
- Roll attitude engage switch
- Heading hold engage switch
- Manual roll attitude trim control
- Turn control

Lighted push-button switches are used for all switching functions. The power switches and the SAS engage switches are push ON - push OFF type switches; the other switches are solenoid-held ON switches that can also be disengaged by pushing OFF. When engaged, the switch position is held partly depressed and the internal light is energized.

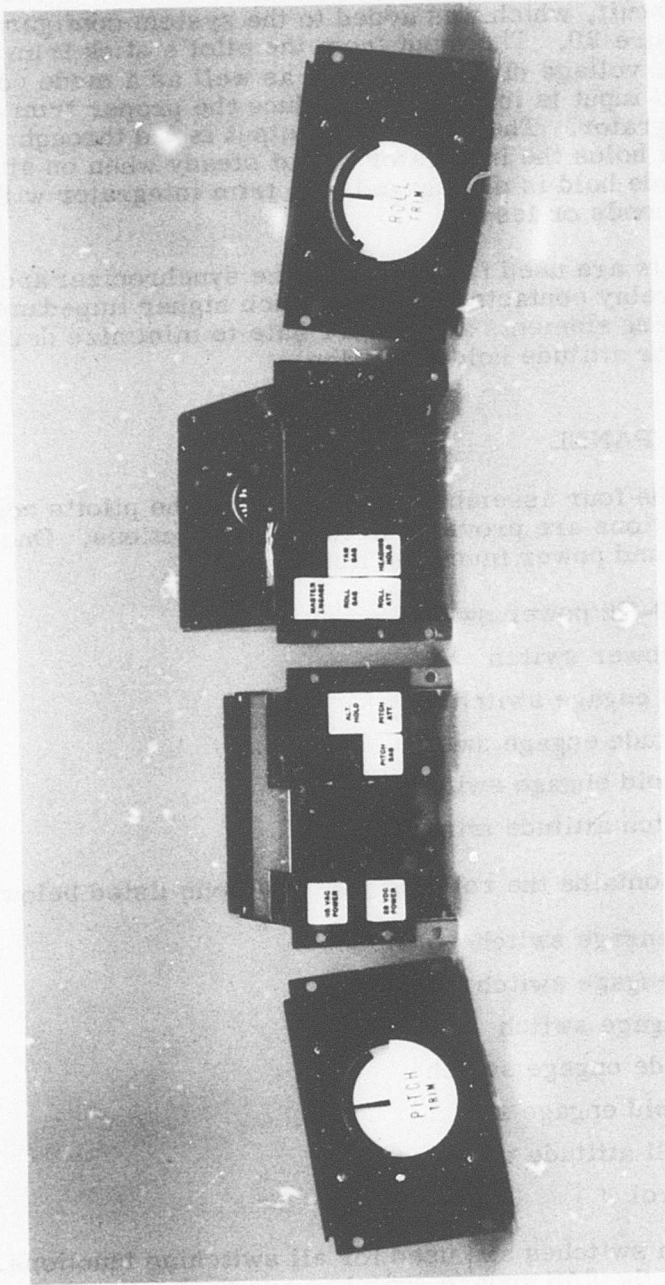


Figure 21. Control Panel Assemblies with Fluidic Trim Indicators

Switch interlocks are provided to mechanize the proper logic functions. The SAS engage switches can be in the ON position when the +28 V power is OFF such that when the system power (master engage switch) is turned on the SAS functions will also be turned on. Roll and pitch attitude hold switching is interlocked with the master switch, which is interlocked with the roll and pitch SAS engage switching and the emergency disengage switch on the control stick. When the master engage switch is ON and roll SAS is ON, roll attitude hold can be engaged. Manual synchronization of the attitude error signal was required in the original configuration using the roll trim knob on the control panel. With the roll attitude hold engaged, the interlock is closed for heading hold to be engaged. The turn control must also be centered so that the heading hold mode may be engaged, as it has priority over the heading hold mode.

In a similar manner, when the pitch SAS is engaged and the master engage switch is ON, pitch attitude hold can be engaged. Manual synchronization of the attitude error signal was required in the original configuration using the pitch trim knob on the pitch control panel. With the pitch attitude hold engaged, the interlock is closed for altitude hold to be engaged.

When pitch attitude hold, altitude hold, roll attitude hold, and heading hold modes are engaged, a single operation of the emergency disengage or master power switch will disengage all four modes.

Addition of the automatic synchronization circuits to the system eliminated the main functions of the pitch and roll trim knobs. However, these knobs have been left on the control panels and the functions remain connected electrically to the attitude input circuits as they were in the original system mechanization.

The trim indicators, shown in Figure 21, are differential pressure gauges that show the signals in the pitch and roll fluidic circuits just ahead of the attitude-loop, engage solenoid valves. Hydraulic lines are run between the hydrofluidic package and the trim indicators located on the pilot's console. With the addition of the synchronization feature, these gauges were eliminated, and the electrical trim indicators shown in Figure 22 were used to monitor system nulls.

Addition of the airspeed hold mode required another solenoid-held switch on the pitch control panel. Mode switching requirements specified that the pitch attitude mode must be engaged prior to airspeed hold engage and that airspeed and altitude hold have equal priority; that is, whichever function is selected must override the other, which required the addition of a two-way interlock between the altitude hold and airspeed hold switches. Figure 23 shows the pitch/power control panel with the airspeed switch added above the altitude hold switch.

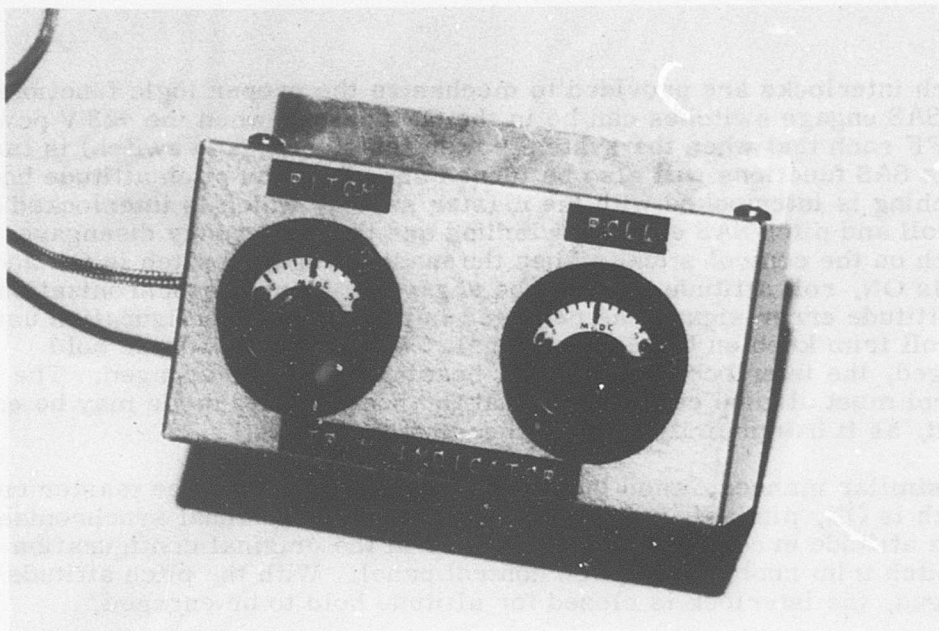


Figure 22. Electrical Trim Indicators

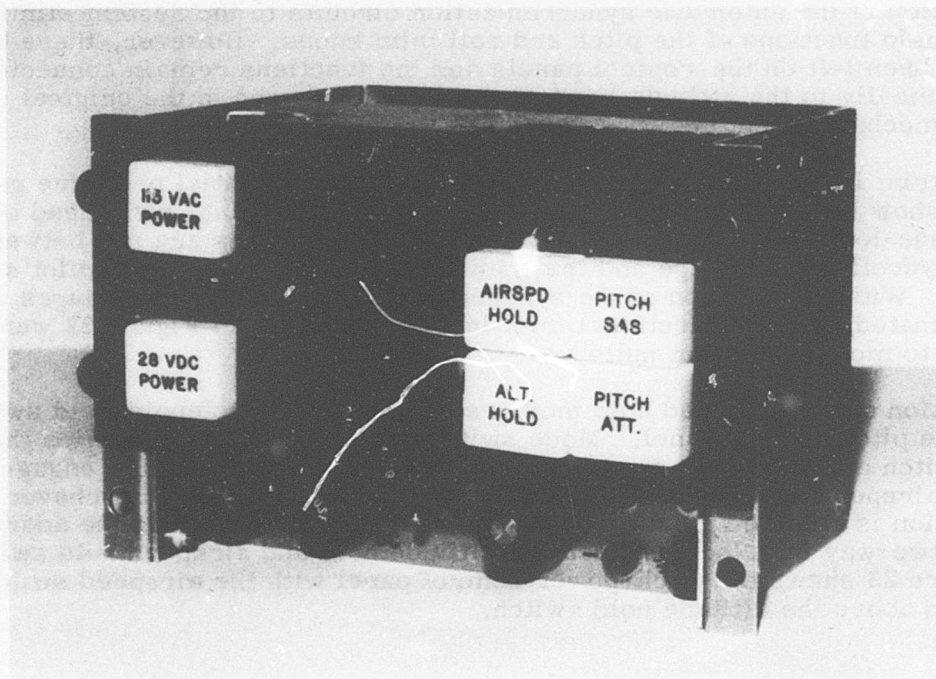


Figure 23. Pitch Control Panel

A mode compatibility chart is shown in Figure 24. Symbols are given for compatible modes, and prerequisite mode conditions are indicated, that is, those modes that must be engaged prior to the mode in question. To determine compatible modes, first select a mode in the first column. Read across the row to the diagonal bar and then down the column, noting the dot symbols that indicate compatible modes. Using roll attitude as an example, the modes indicated as being compatible are pitch SAS, yaw SAS, pitch attitude, altitude, and airspeed. To determine prerequisite modes, read across the rows and note the arrowhead symbols.

The test console shown in Figure 25 was used to check out the control panels in the laboratory. The test console contains the required power supplies and electrical interconnections to simulate the total system interconnection in the helicopter. The aircraft heading indicator is shown mounted in the console with the control panel. The aircraft vertical gyro can also be connected to the console for system testing. The large connector on the upper right of the console is where the cable to the hydrofluidic package is attached. This console was used for total system checkout and gain adjustment prior to installation in the helicopter.

SERVOACTUATORS

The servoactuators used in the system are the same units used for the flight test of a three-axis hydrofluidic stability augmentation system on Contract DAAJ02-70-C-0017. A performance summary for these units, all three of which are identical, is presented in Table 3. One of these units is shown in Figure 26.

The servoactuators are similar to conventional electromechanical designs except that the input torque motor (coil) was replaced with force capsules (bellows), and mechanical feedback from the ram is used. The differential output pressure from the fluidic controller acts on the two force capsules, causing motion of the first-stage flapper. This flapper motion closes one nozzle and opens the other of the first stage, producing an amplified differential pressure signal to the spool valve, which in turn causes the actuator ram to move. The mechanical feedback spring stops the ram at a position proportional to the differential input pressure by returning the flapper to its null position.

Bleed orifices, located in the input signal lines near the bellows, are used to help purge air from the input section. Air that enters this section passes out the reference port back to the return line of the controller.

The servoactuator also includes a centering and locking mechanism, which is activated when hydraulic pressure is removed, and a position transducer (LVDT) to measure actuator position.

PITCH SAS	/								
ROLL SAS	●	/							
YAW SAS	●	●	/						
ROLL ATTITUDE	●	▼	●	/					
PITCH ATTITUDE	▼	●	●	●	/				
HEADING	●	▼	●	▼	●	/			
TC TURN	●	▼	●	▼	●	←	/		
ALTITUDE	▼	●	●	●	▼	●	●	/	
AIRSPEED	▼	●	●	●	▼	●	●	↙	/
	PITCH SAS	ROLL SAS	YAW SAS	ROLL ATTITUDE	PITCH ATTITUDE	HEADING	TC TURN	ALTITUDE	AIRSPEED

- INDICATES COMPATIBLE MODES
- ▼ INDICATES MODE PREREQUISITE
- ← INDICATES MODE PRIORITY
- ↙ INDICATES EQUAL PRIORITY

Figure 24. Mode Compatibility Chart

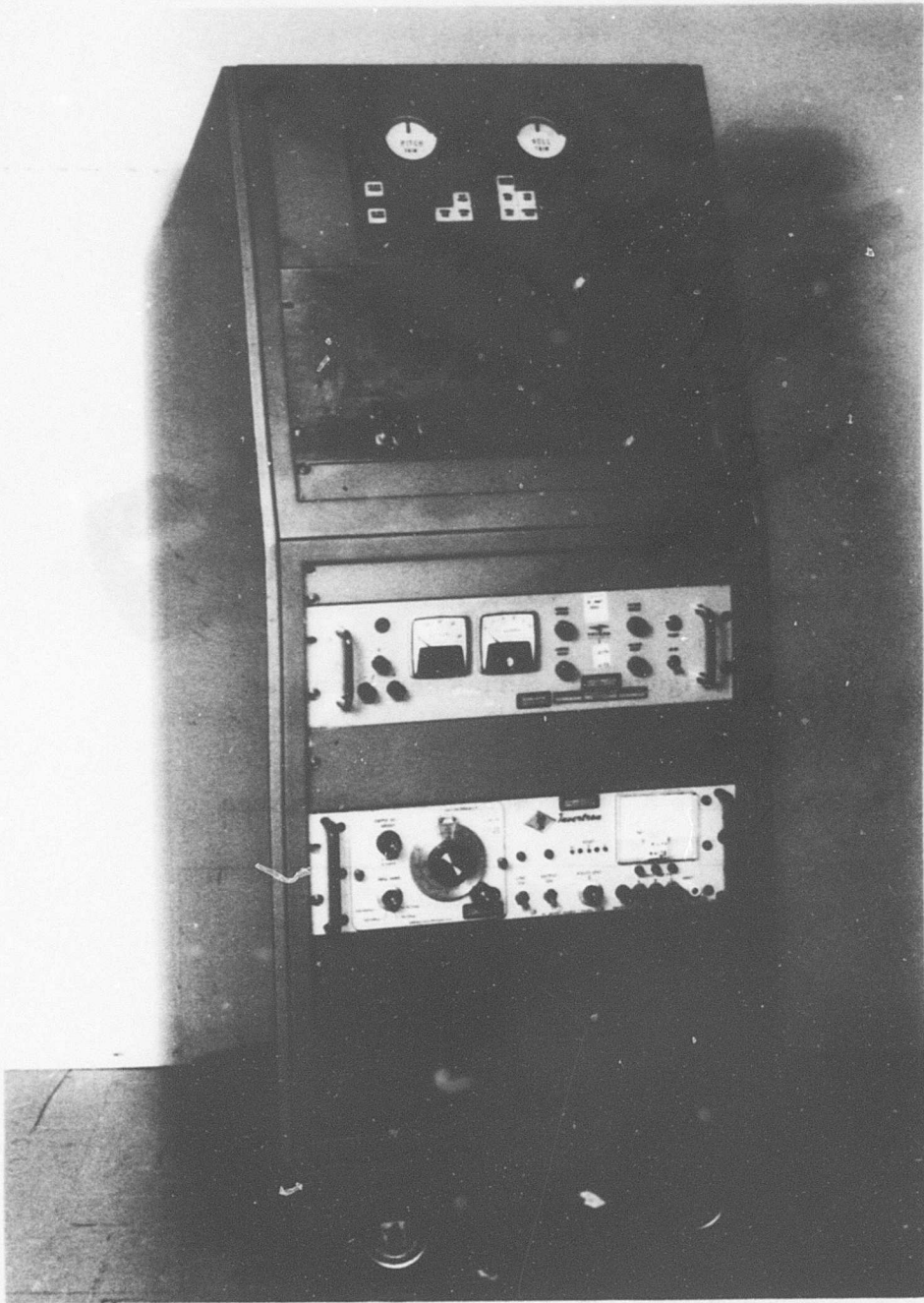


Figure 25. Control Panel Test Console

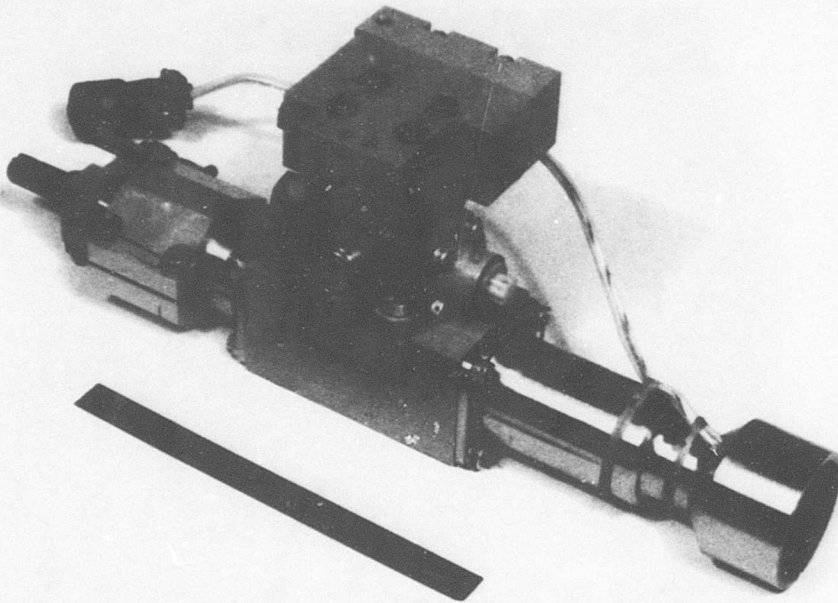


Figure 26. Series Servoactuator

Table 3. Servoactuator Performance

Scale Factor: 0.09 in. /psid
Stroke: ± 0.38 in
Supply Pressure: 1000 psi
Output Force: 160 lb (maximum)
Centering Force: 50 lb
Threshold: 1.0% (maximum)
Rated Velocity: 10 in. /sec no load
Dynamic Response: 90 deg phase lag at 5 Hz (minimum) at 25% rated input
Hysteresis: 2% of full stroke
Effective Input Capacitance: 5×10^{-4} in. ³ /psi

SECTION IV
COMPONENT DESIGN AND PERFORMANCE

ATTITUDE REFERENCES

The system attitude references are the existing electrical display gyros used for vertical attitude reference and aircraft heading reference information on the UH-1M helicopter. Figure 27 shows the ID 998/ASN course indicator which has an output proportional to the difference between actual aircraft heading and a set heading. This is standard equipment on the UH-1. Figure 27 also shows the MD-1-type roll and pitch vertical gyro used on the UH-1. The electrical outputs from these references are demodulated and amplified in the system electrical interface circuit.

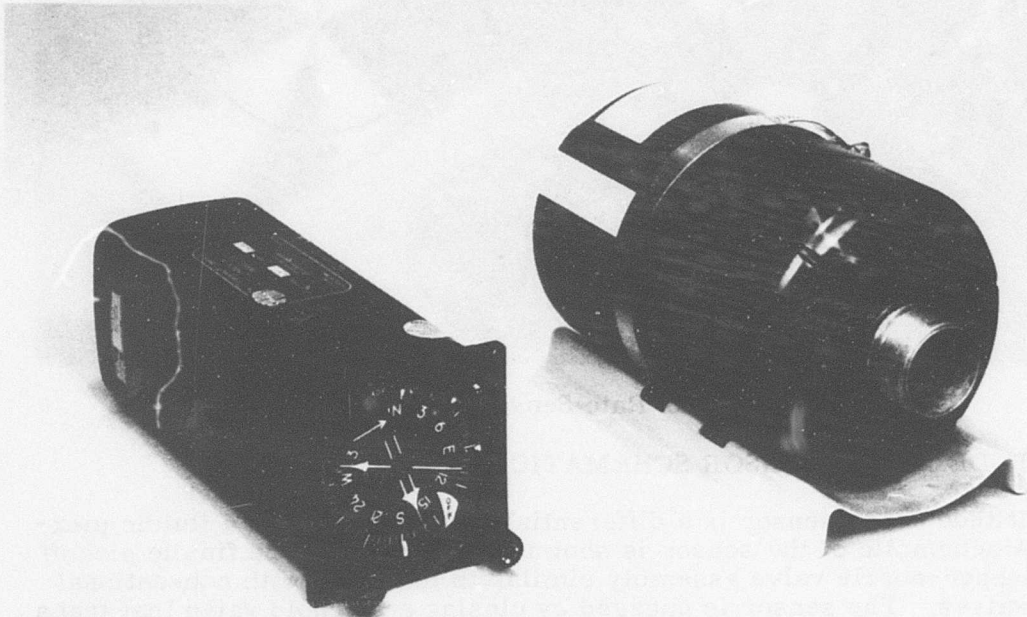


Figure 27. Aircraft Vertical Gyro and Gyrocompass Indicator

VORTEX RATE SENSOR

Similar vortex rate sensor units are used in the three axes to provide aircraft angular rate feedback signals. Figure 28 shows a disassembled unit. The main housing contains the porting for sensor power, return,

and signal lines. The etched coupling rings, which are thin washer-type elements with integral spacers, are shown stacked on two pins within the housing cavity. The two small blades shown inside the coupling element are for sensor nulling and test input signals. The smaller blade, used for built-in-test (BIT), is retracted out of the flow stream until activated for testing. The sensor pickoff (center) is an electroformed unit that includes the sink tube, airfoil, and pressure taps. The third piece is the cover for the sensor. Figure 29 shows typical sensor scale factor and noise data with a fluidic amplifier load.

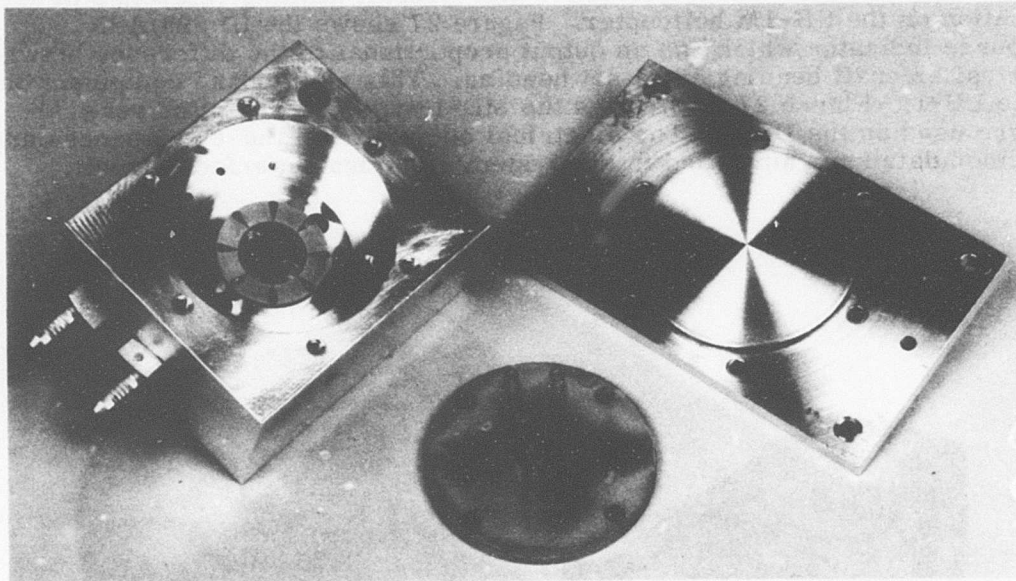


Figure 28. Vortex Rate Sensor (Disassembled)

ALTITUDE ERROR SENSOR SCHEMATIC

The altitude error sensor is a differential bellows type with a fluidic pickoff. A schematic of the sensor is shown in Figure 30. The fluidic pickoff is a flapper-nozzle valve assembly similar to that used with conventional servovalves. The sensor is engaged by closing a solenoid valve that traps a reference pressure on one side of the differential bellows. Changes in altitude produce a change in pressure on the other side of the bellows. This differential pressure becomes a force on the flapper assembly, producing a differential hydraulic pressure at the pickoff. The bellows-sensing unit of this device has an added liquid-filled capsule that acts to compensate for trapped air expansion with temperature change. It was found during flight test that because the ambient temperature variation normally occurring during the time the altitude hold mode is engaged is small, the temperature compensation capsule was not needed. If a large

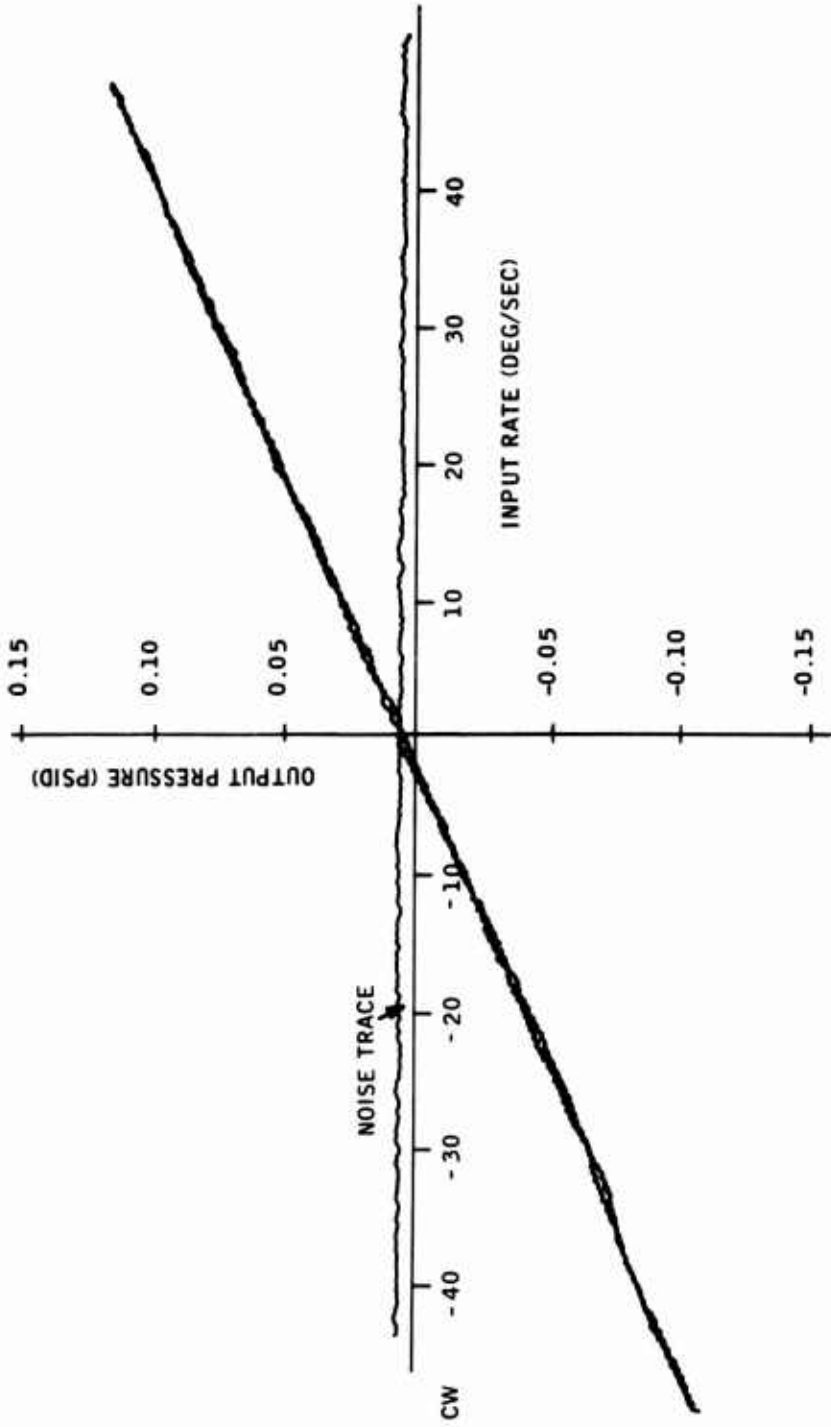


Figure 29. Vortex Rate Sensor Scale Factor and Noise

temperature change should occur during mode engagement, the sensor can be momentarily disengaged and reengaged with a new reference air sample.

The altitude error sensor is shown connected to the pitch controller in Figure 14. The large capsule in the foreground is the temperature compensation element, and the engage solenoid valve is shown in the center on the left. The assembly at the far end contains the sensing bellows and fluidic pickoff. The most forward bellows is the sensing element, and the second bellows shown at the far side is used to balance the force on the pickoff for thermal expansions in the components. Figure 30 shows a typical performance curve for the altitude error sensor.

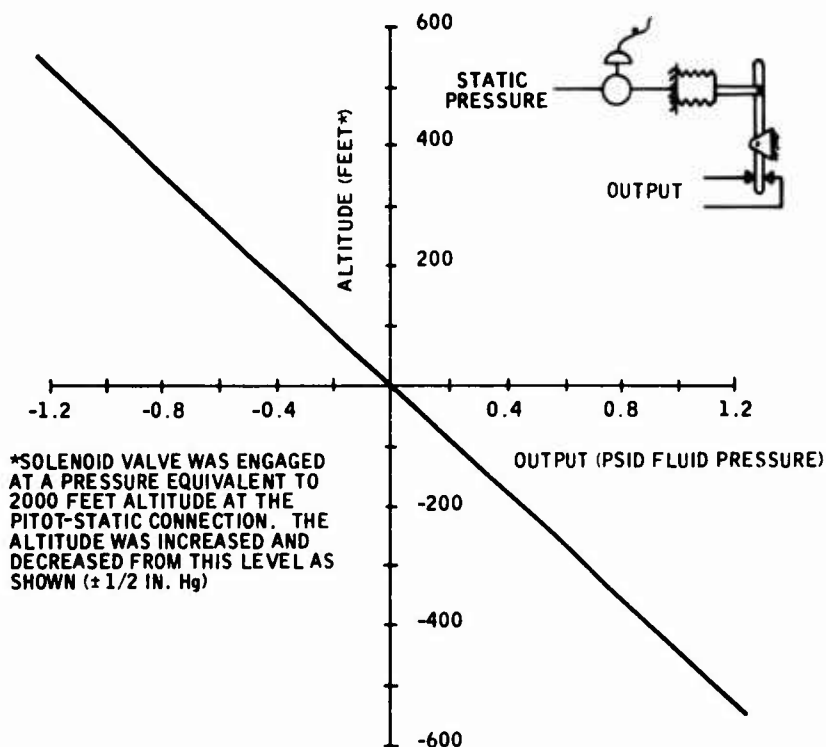
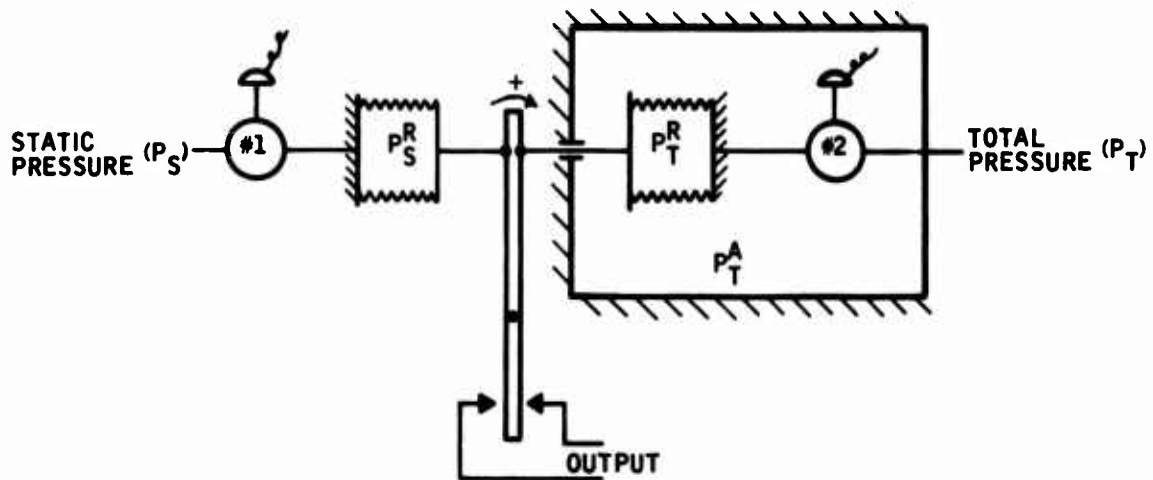


Figure 30. Altitude Error Sensor Performance

DYNAMIC PRESSURE DEVIATION SENSOR

The dynamic pressure (airspeed) sensor is very similar to the altitude error sensor design. Figure 31 is a schematic of the sensor, and Figure 32 is a picture of the unit. The sensor is the same as that used for altitude error except that (1) it has a valve on both bellows such that two reference pressures can be held and (2) one of the bellows is inside a closed



FOR SENSOR ENGAGED

VALVES NOS. 1 AND 2 CLOSED

$$(P_T^A - P_T^R) + (P_S^R - P_S^A) = \text{OUTPUT} +$$

$$(P_T^A - P_S^A) - (P_T^R - P_S^R) = q^A - q^R = \text{OUTPUT} +$$

WHERE "A" DESIGNATES ACTUAL PRESSURE AND "R" DESIGNATES REFERENCE PRESSURE

Figure 31. Dynamic Pressure Deviation Sensor Schematic

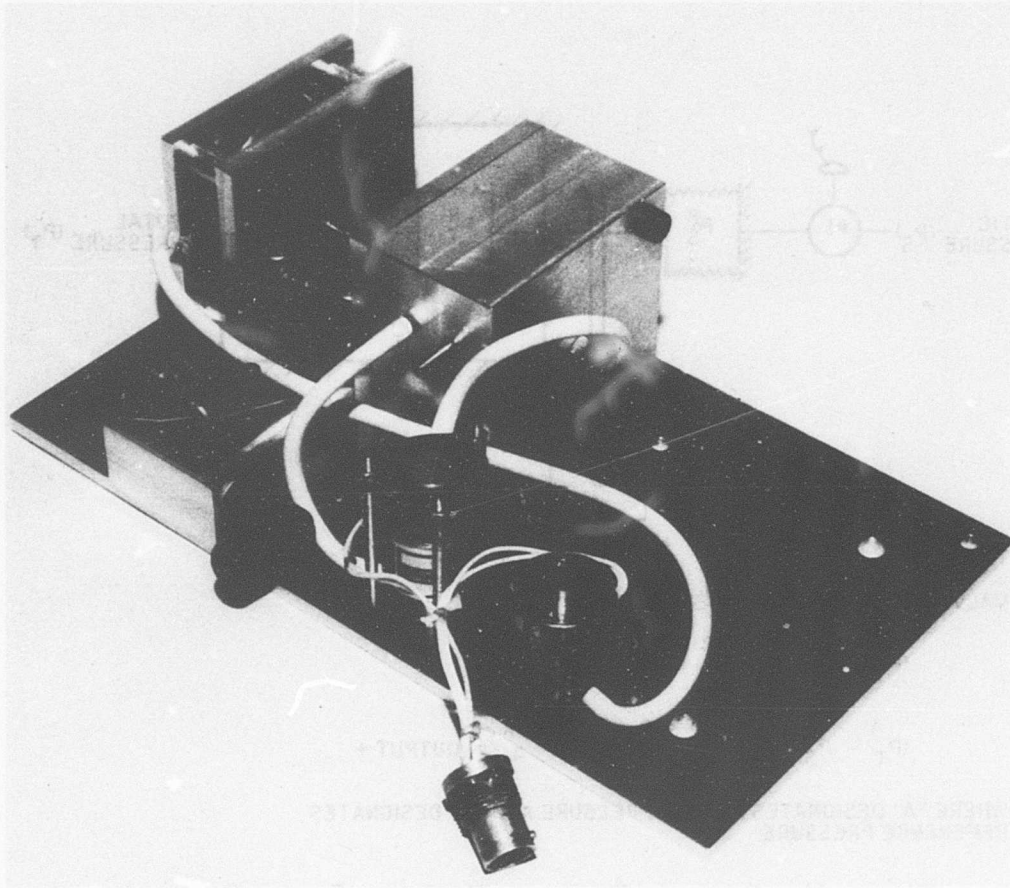


Figure 32. Dynamic Pressure Deviation Sensor

chamber such that a pressure different from ambient can be placed on both outside and inside. The chamber is attached to the total pressure manifold on the aircraft, and the opposite bellows senses cabin pressure. The valve logic and equations for operation of the sensor are also shown in Figure 31.

PEDAL INPUT TRANSDUCER

The pedal input transducer, shown in Figure 33, is the same unit used in the previous three-axis fluidic stability augmentation system. The motion of the pedals is transferred through a push-pull cable from the pedals to the position transducer mounted on the yaw controller. The cable is attached to the end of the black, motion-reducing arm. The other end of the arm operates against a spring, which in turn pushes on a flex pivot. The opposite end of the flex pivot varies the opening to two nozzles in a hydraulic flapper-nozzle circuit. A typical input/output curve for the unit is shown in Figure 34.

FLUIDIC AMPLIFIER CIRCUITS

Schematics of the amplifier circuits for the three controllers are shown in Figures 11, 13, and 16. The proportional amplifiers are the beam-deflection type, which were fabricated using the electroformed conductive wax process. The manifolds for the circuits were also fabricated by this process. Figure 35 shows a disassembled view of the pitch controller. The two electroformed manifolds, with the amplifier elements mounted on the under side, attach to the two sides of the controller block. The lower center part of the controller is the vortex rate sensor, and the upper section contains the bellows capacitors and interconnection porting between the two electroformed manifolds. Figure 36 is a picture of the capacitor block used in the yaw controller with the bellows and several orifice elements removed.

ELECTRIC-TO-FLUIDIC TRANSDUCERS

The two types of electric-to-fluidic (E/F) transducers used in the system are shown in Figure 37. The analog-unit type shown on the left is used to transform electrical pitch attitude, roll attitude, and heading signals into fluidic signals. The input signal is differential dc current, and the output is differential hydraulic pressure. The transducer consists of a torque motor (magnetic coil) and a flapper-nozzle assembly. This device is the same type used as the first stage of an electrohydraulic servo-valve. Sample performance curves for several values of supply pressure to the transducer are shown in Figure 38.

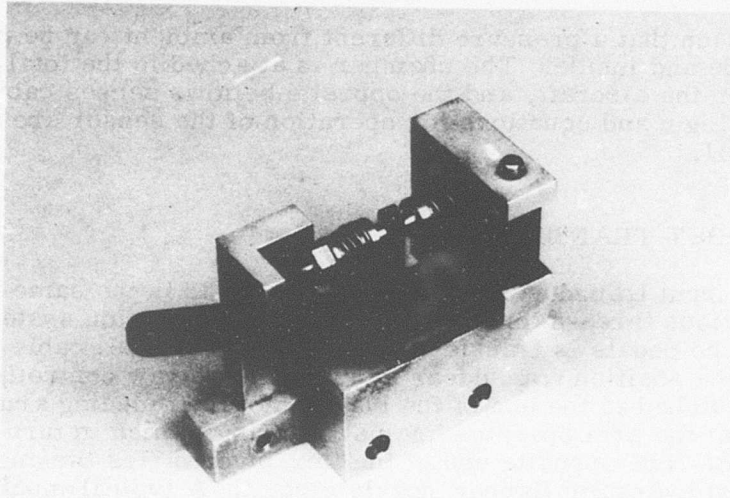


Figure 33. Position Transducer (Yaw Axis Pedal Input)

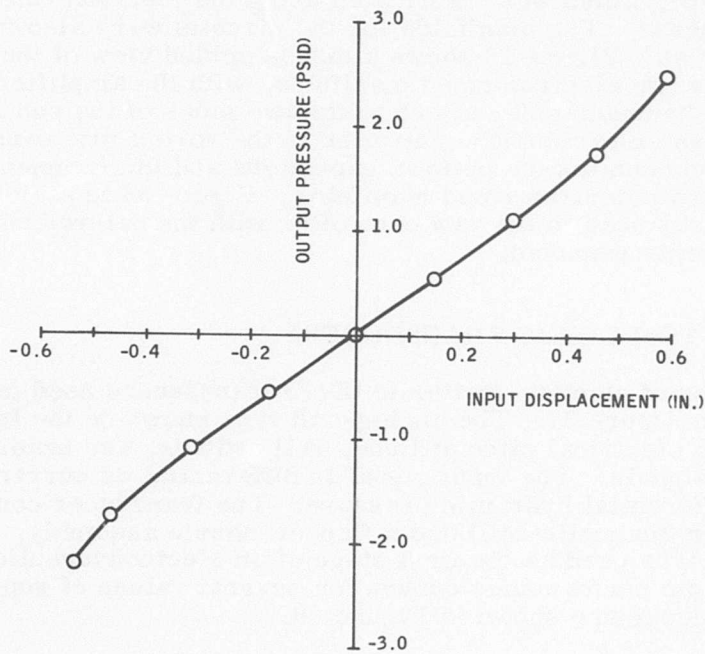


Figure 34. Pedal Input Transducer Performance

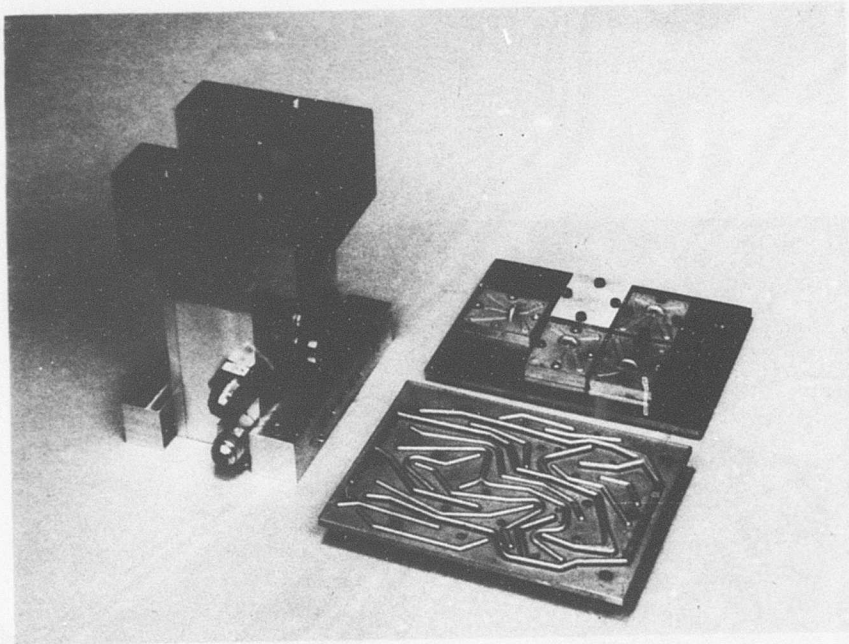


Figure 35. Pitch or Roll Controller (Disassembled)

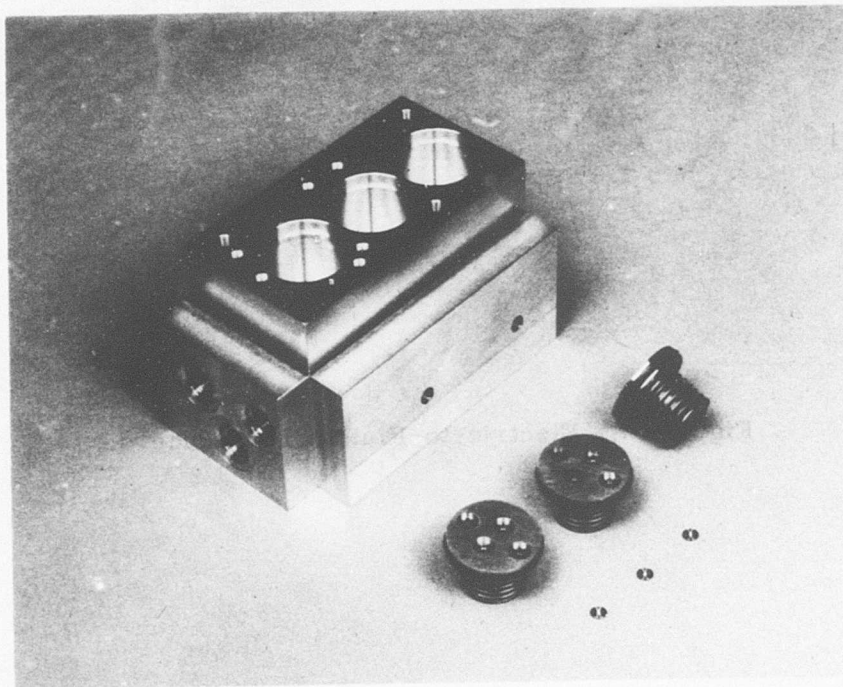


Figure 36. Yaw Capacitor Block (Bellows and Orifices Removed)

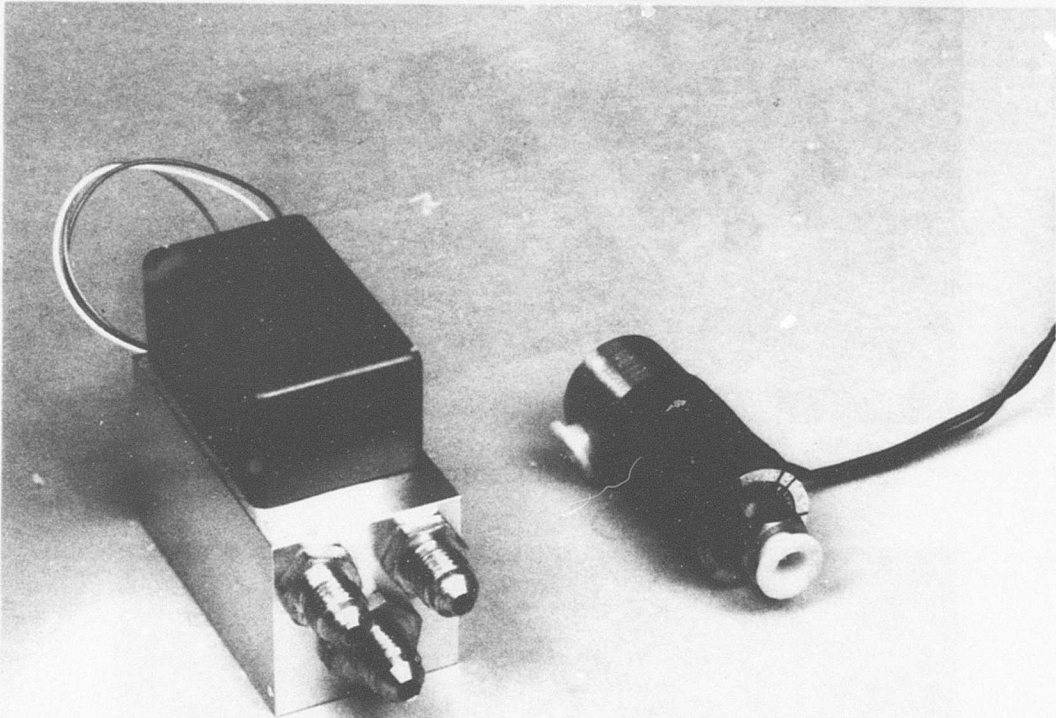


Figure 37. Electric-to-Fluidic Transducers

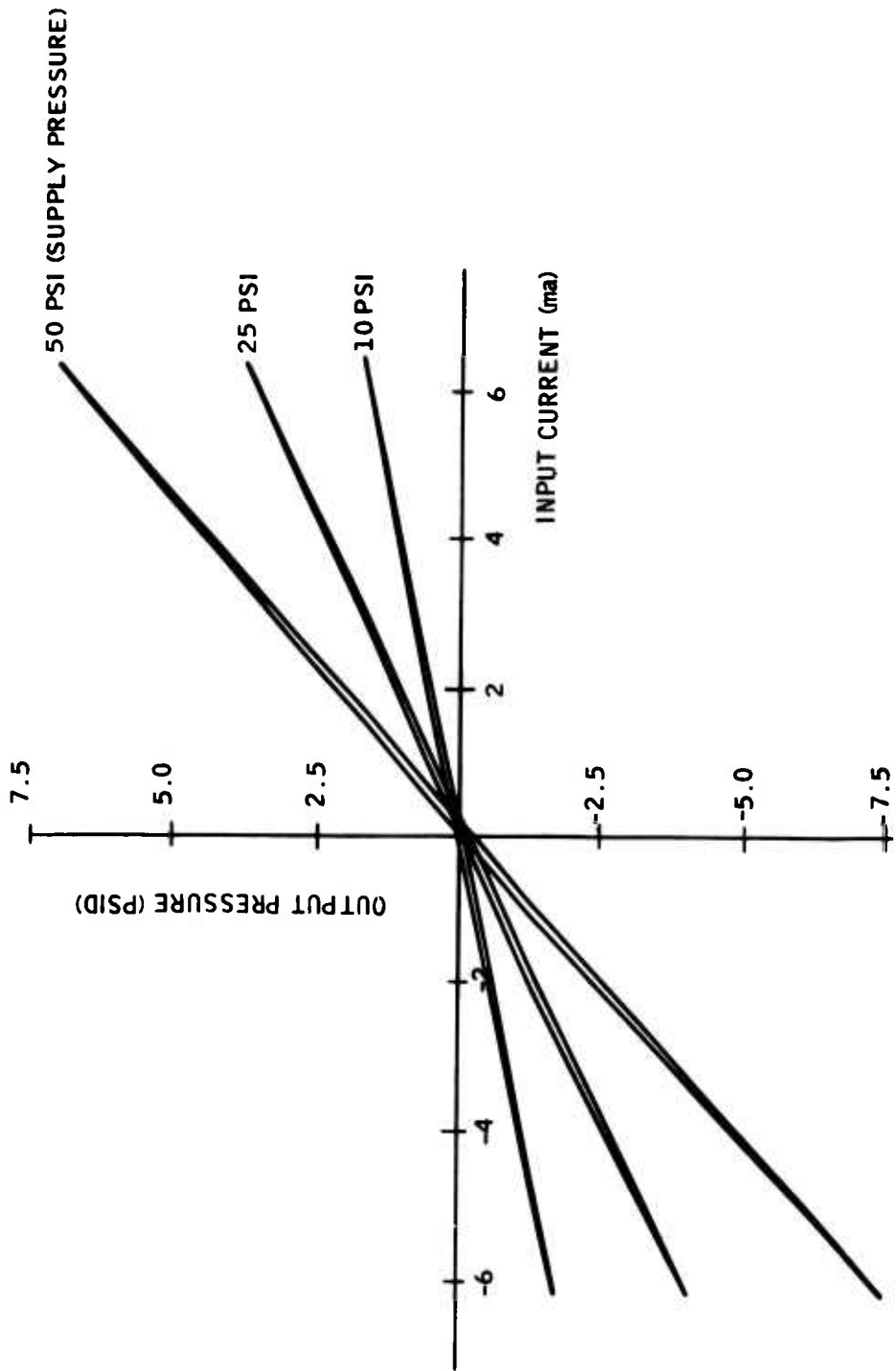


Figure 38. Electric-to-Fluidic Transducer Performance

The unit on the right of Figure 37 is a solenoid valve which is used in both the pitch and roll axes to engage and disengage the attitude loops.

ELECTRICAL CIRCUITS

Attitude Input Circuits

The original attitude control input circuit consisted of a demod-amp that also acted as the E/F valve driver. The trim pot with which the pilot manually synchronized the attitude input feeds into the amplifier and also a test input for measuring and setting the attitude gain provided. A gain adjust was provided in the amplifier feedback circuit in series with the E/F valve coil. The roll attitude input circuit is the same as the pitch attitude input circuit with the addition of a turn control input. Figure 39 shows the roll attitude input circuit as originally mechanized.

In planning the addition of the automatic synchronizing functions to these circuits, one objective was to keep as much of the original circuitry as possible and to tie in the synchronizer functions at the appropriate point. Other requirements are given below.

- The modified attitude input with electrical synchronization shall provide the same forward loop gain during the attitude hold mode as existed in the system prior to the modification.

Pitch - 0.202 mA at the E/F valve/deg attitude error.

Roll - 0.32 mA at the E/F valve/deg attitude error.

- The synchronization range shall be ± 30 deg attitude.
- Synchronizer reference drift shall be less than 1 deg/hr.
- Electrical synchronization time when switching from the attitude hold mode shall be less than 10 msec.
- The pressure transducer - carrier demodulator portion of the synchronizer loop shall have a gain capability of 0.4 V/psi with a pressure input range of ± 20 psid.

The simplest way to modify the attitude input circuit was to disconnect the E/F valve coil and gain adjust from the demodulator output and to connect them at the synchronizer output amplifier. The added circuitry was built into a separate electronics package with its own regulated ± 15 -volt power supply. The pitch and the roll synchronizer and trim circuits were built on printed-circuit cards using the same card layouts. Test

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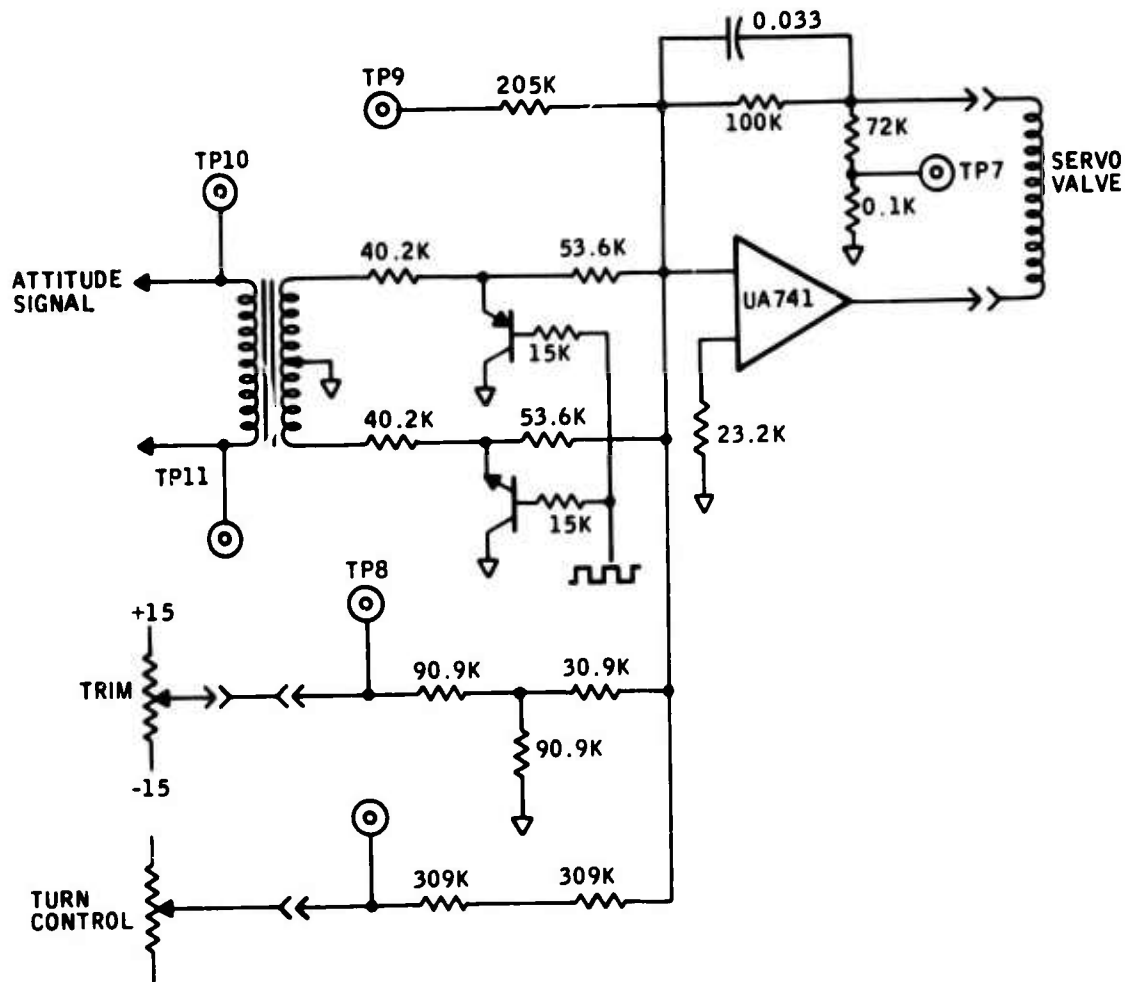


Figure 39. Roll Attitude Demodulator

points were provided at various circuit points for test monitoring and calibration purposes. The circuitry included on one circuit card is shown inside the dotted lines in Figure 20.

Figure 40 shows the trim circuit details. The input is +28 volts from the trim switch on the pilot's stick grip. This signal is the input to both the trim integrator and the switching logic to relay K1. One position of the stick input goes direct to the integrator amplifier U2, and the other input goes through inverter U1 to provide the reverse command into the integrator. The output to the system is not taken directly from the integrator output, as a bias voltage must be generated from the integrator to maintain the FET clamp circuit output at zero. The output is taken from Q1 FET to a voltage divider, which then supplies the input to the attitude demod amp.

Operation of the trim circuit is in three basic modes. With power on and when not on attitude hold, K1A is closed and the integrator is nulled by the feedback path through K2A. When attitude hold is engaged, K1A and K2A are opened and K1B is closed. This action clamps the output of the integrator to the null voltage. Any drift of the integrator output is sensed as a differential voltage at U3, and its output drives the integrator amplifier to reduce the drift. The output of U3 is a square-wave voltage of small amplitude as it switches back and forth to hold the integrator output level. An amplified output from the integrator is a small-amplitude, sawtooth waveform on the steady-state dc level needed to maintain the FET output at null. When the trim switch is closed in either direction, relay K1 is energized and both FET gates are tied to the integrator amplifier output while the integrator output is changing. Relay contact K2A remains open. When the trim switch is opened, the circuit reverts back to the clamp mode with amplifier U3 driving the integrator output to the level held by the capacitor at the gate of FET Q1.

Heading Select/Hold Input

The electrical input for the heading error signal was the same as the basic attitude input circuit previously shown in Figure 39. A heading error limiter was mechanized as part of the fluidic control elements to limit heading error to a nominal of 15 degrees. The actual limit varied with fluidic oil temperature. Along with the other circuit modifications, a modification was made to the heading input to provide an adjustable limit in the E/F drive circuit. Figure 41 shows the addition of a 2000-ohm potentiometer in series with the E/F coil. This causes an increase in the voltage output of the amplifier for a given current flow. The maximum desired current is then adjusted to coincide with the voltage saturation level of the amplifier to provide the desired limit.

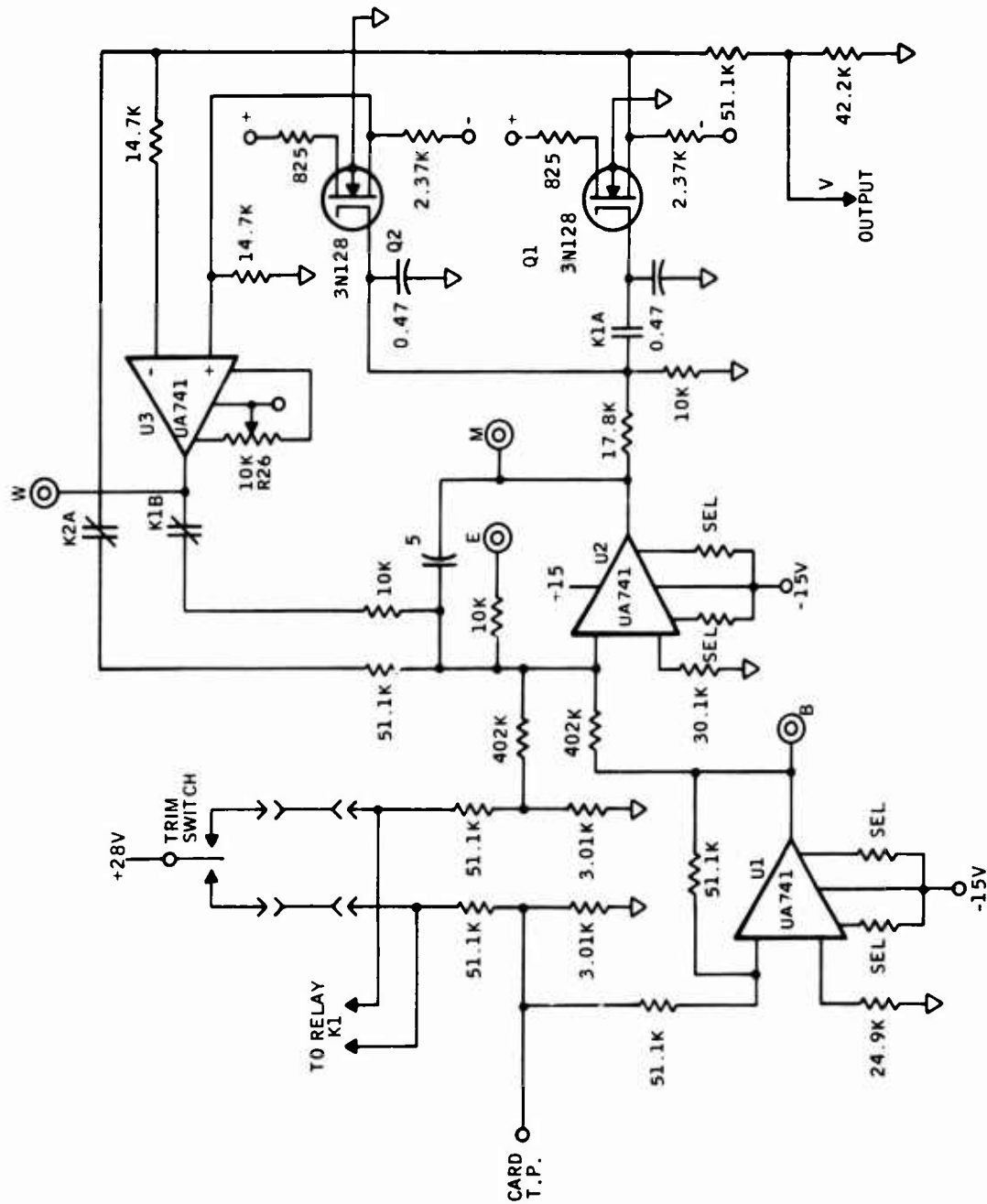


Figure 40. Stick Trim Circuit

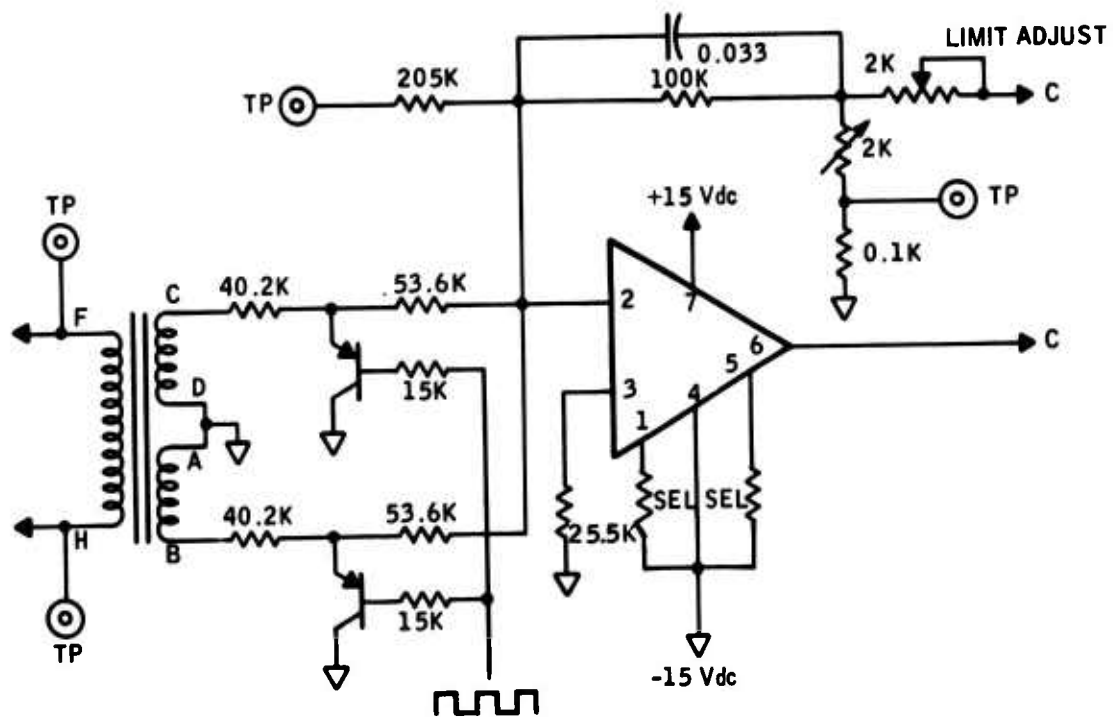


Figure 41. Modified Heading Input Circuit

Stick Trim

The stick trim circuit function that was added to the system was mechanized with the following requirements:

- The stick trim function shall provide an electrical trim rate of 0.4 deg/sec.
- The trim drift rate shall be less than 2 deg/hr.
- The trim authority shall be approximately ± 8 deg attitude change.
- When the attitude mode is disengaged, the trim output shall return to null in less than 10 sec.

The trim circuit uses the capacitor - FET storage configuration similar to that used in the synchronizer to form a clamp circuit on the integrator output. The variations in the MOS-FET operating characteristics require that reasonably close matching of the FET operating point is needed for proper operation.

CONTROL PANEL

Two control panels provide the manual trim, turn control, and switching functions required for pilot control of the advanced stabilization system. One panel provides the pitch and power control functions, and the second panel provides the roll/yaw control functions.

Pitch/Power Control Panel

Clare-Pendar lighted push-button switches give the pilot means for selecting power and pitch control functions. They also provide status indication by means of the depressed position of the push button when the switch is engaged and by internal illumination of the push button. The push buttons are $3/4$ by $7/8$ inch with legend lettering. They are illuminated by a single 28-volt, 40-milliamperere type T1- $3/4$ lamp. The push buttons are removable from the front of the panel for lamp replacement.

The two power switches are located on the left side of the panel, the four mode switches are located on the right side of the panel, and the pitch trim knob is located in the middle. No scale marks are provided for the pitch trim knob because the potentiometer controlled by the knob is spring returned to center when the 28-volt power is removed from the trim clutch such that no absolute reference exists between potentiometer position and knob position. The pitch trim potentiometer is energized with ± 15 -volt

power obtained from the roll/yaw control panel. Trim authority is controlled by the summing circuit in the demod-amplifier for the pitch attitude signal.

The schematic diagram of the pitch/power panel is shown in Figure 42. Switching interlocks between the pitch attitude, altitude, and airspeed control switches are shown.

A test stimulus potentiometer is provided at the bottom of the control panel for use in calibration and troubleshooting of the system. A switch allows selection of positive or negative input voltages.

Roll/Yaw Control Panel

The roll/yaw control panel shown in Figure 43 contains five lighted push-button switches, the roll attitude trim knob, and the turn control knob. The push-button switches are the same as described for the pitch/power panel with appropriate legends for the roll/yaw functions.

The roll trim knob and trim pot are mechanized the same way as the pitch trim knob, with the pot spring returned to center upon release of the holding clutch.

The turn control is a knob-operated potentiometer and switch that provides roll attitude commands proportional to the knob displacement. The detent switch is closed when the turn control is centered such that heading hold can be engaged.

The roll/yaw control panel is the chassis for the electronics circuit card that contains the power supply and the pitch, roll, and heading demod amplifier circuits. The power transformer is also mounted in the chassis.

FLOW CONTROL VALVE

During the earlier design study, a concept for reducing system sensitivity to oil temperature change was formulated and tested. The concept involves varying flow to the system as a function of oil temperature (increased flow at low oil temperature and decreased flow at high oil temperature) to compensate in part for the gain change caused by change in fluid viscosity.

To obtain this change in oil flow with variation in oil temperature, a flow control valve was designed that schedules flow with temperature, in addition to maintaining constant flow at a given temperature. Figure 44 shows a schematic of the flow control valve design. The standard flow

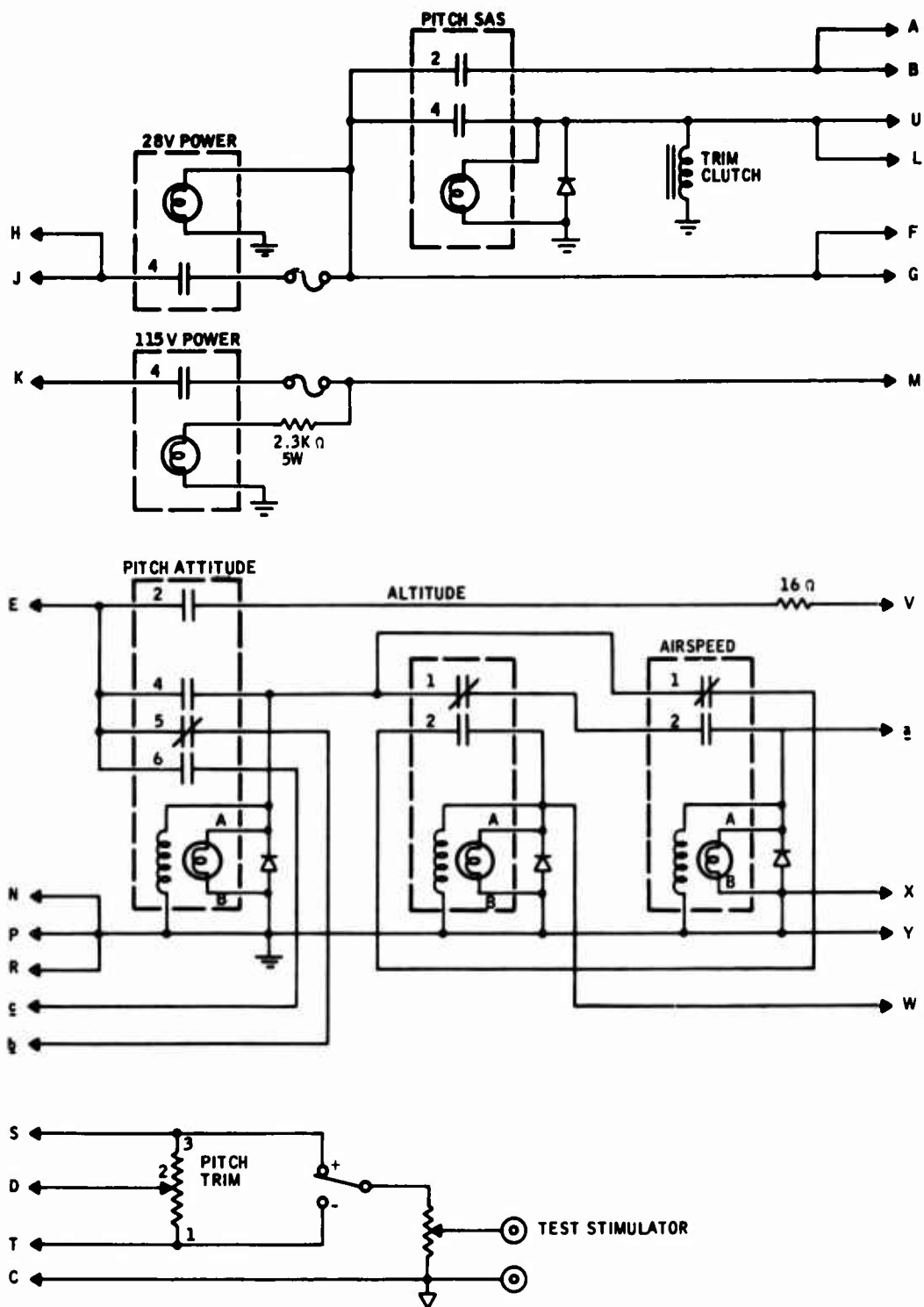


Figure 42. Pitch/Power Control Panel

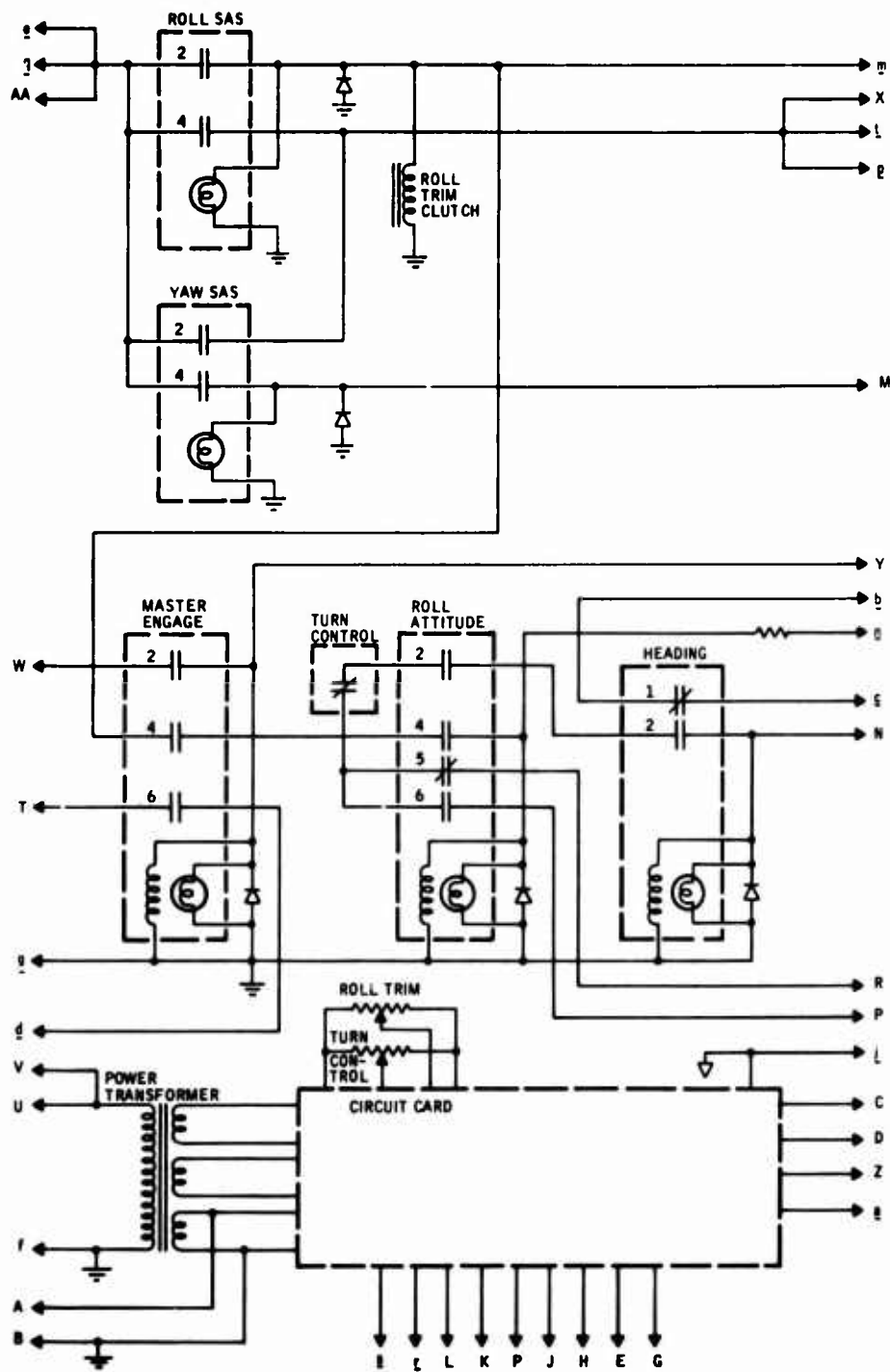


Figure 43. Roll/Yaw Control Panel

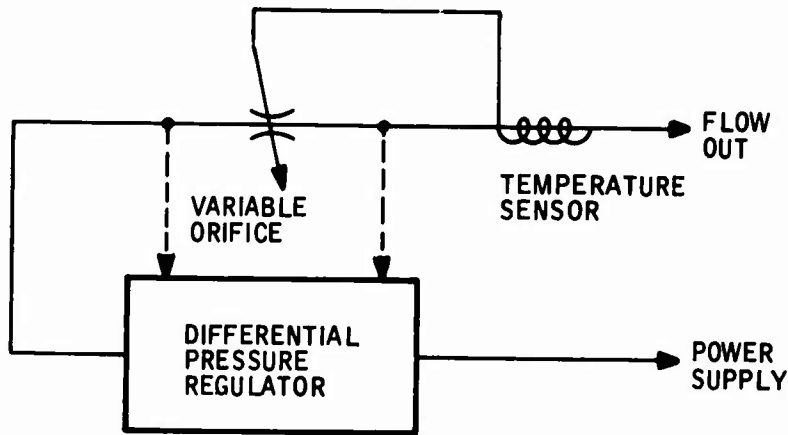


Figure 44. Temperature-Scheduled Flow Control Valve Schematic

control valve consists of a differential pressure regulator that maintains a constant pressure drop across a sharp-edged orifice, and, therefore, constant flow. This basic design was modified such that the size of the orifice is varied as a function of oil temperature.

Figure 45 shows the internal components of the valve. The small inner spool and spring make up the pressure regulator, and the larger outer spool and sleeve make up the variable orifice mechanism. The oil temperature is sensed by two bimetal beams, which rotate the outer sleeve with respect to the spool to vary a shaped orifice. The orifice shape is designed such that the desired variation in flow with fluid temperature is obtained.

Figure 46 shows system flow versus oil temperature for this unit. The valve produces a flow increase of 2.5 to 1 over the oil temperature range of 40°F to 185°F.

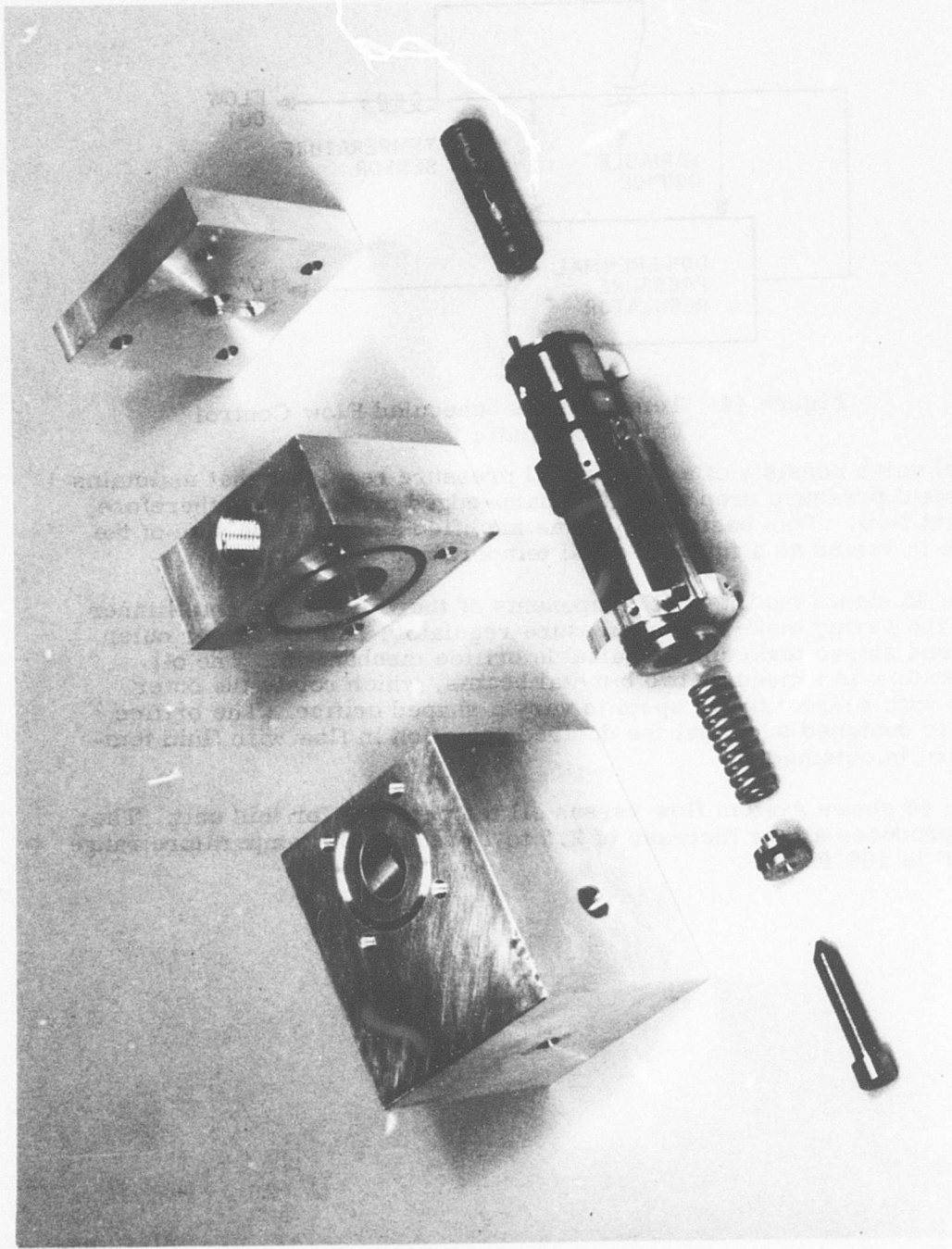


Figure 45. Temperature - Scheduled Flow Control Valve

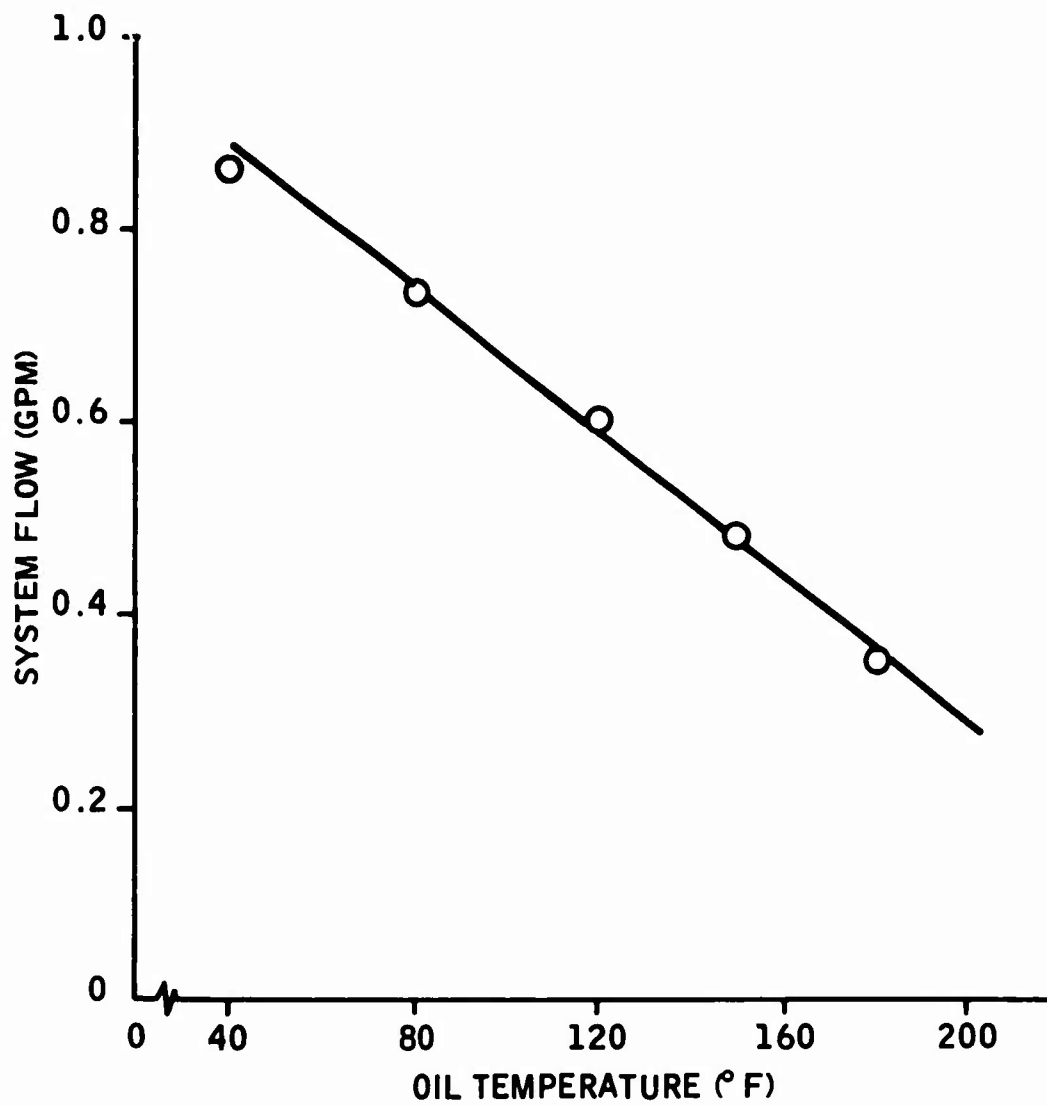


Figure 46. Temperature-Scheduled Flow Control Valve Performance

SECTION V SYSTEM FLIGHTWORTHINESS TESTS

After component development, the system was assembled and calibrated, and then subjected to temperature and vibration tests to ensure the integrity of the system for the flight test environment. In this section are described the tests and results of the tests on various parts of the system.

HYDROFLUIDIC CONTROLLER PACKAGE

The controller package was tested by determining the various loops' performance (gain and response) at oil temperatures of 40°, 80°, 120°, 150°, and 180°F. Cyclic inputs at various frequencies were applied to the system loops to determine response and gain. Figures 47 through 54 show the response test results for the various loops at 120°F oil temperature, and Table 4 lists the system gains at the test oil temperatures. The design goals are given in Appendix A, System Specification. Even though the system did not meet the design goals at the oil temperatures of 40° and 180°F, the results at 80°, 120°, and 150°F were satisfactory. The normal oil temperature range expected in a UH-1 helicopter during flight testing was 80° to 150°F; therefore, system performance was considered to be satisfactory for the tests.

Table 4. System Loop Temperature Test Results

Loop	Nominal Gain	Oil Temperature (F)					
		40	80	120	150	180	120
Yaw SAS	0.023 in /deg/sec	0.00024	0.0140	0.0228	0.0152	0.0082	0.0226
Yaw Pedal Input	2.35 in. /in pedal	0.02700	1.2100	2.3000	0.8200	0.3400	2.3000
Roll SAS	0.012 in. /deg/sec	0.00130	0.0110	0.0150	0.0110	0.0085	0.0155
Roll Attitude	0.015 in. /deg	0.00400	0.0110	0.0174	0.0114	0.0040	0.0178
Heading Hold	0.009 in /deg	0.00460	0.0080	0.0100	0.0066	0.0017	0.0092
Pitch SAS	0.026 in /deg/sec	0.00410	0.0157	0.0320	0.0221	0.0080	0.0320
Pitch Attitude	0.038 in. /deg	0.00100	0.0500	0.0380	0.0370	0.0068	0.0390
Altitude	7.60 in /psi	0.75000	7.4500	7.5000	7.5000	1.0000	7.6000

Post Vibration

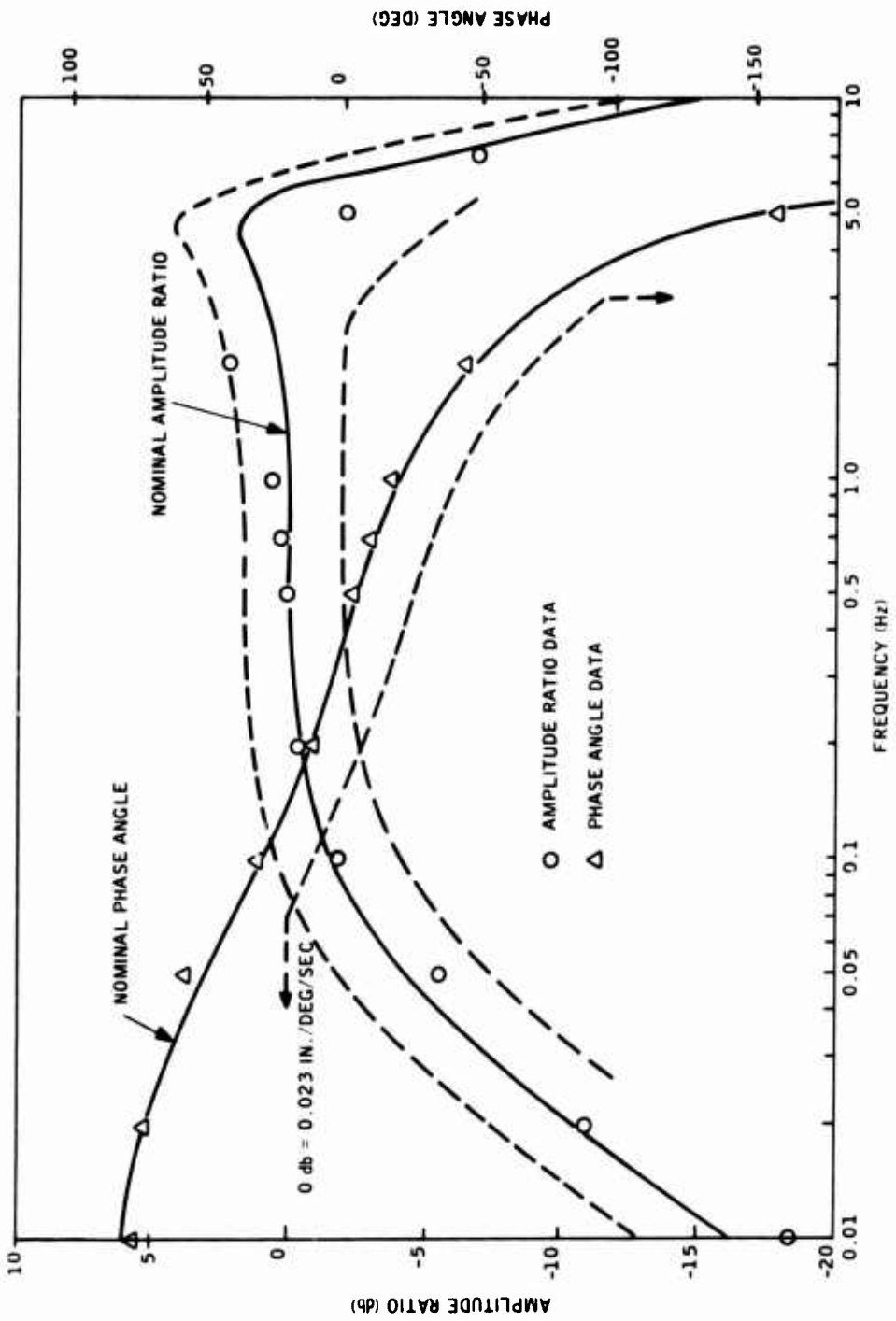


Figure 47. Yaw SAS Dynamic Response at 120°F Oil Temperature

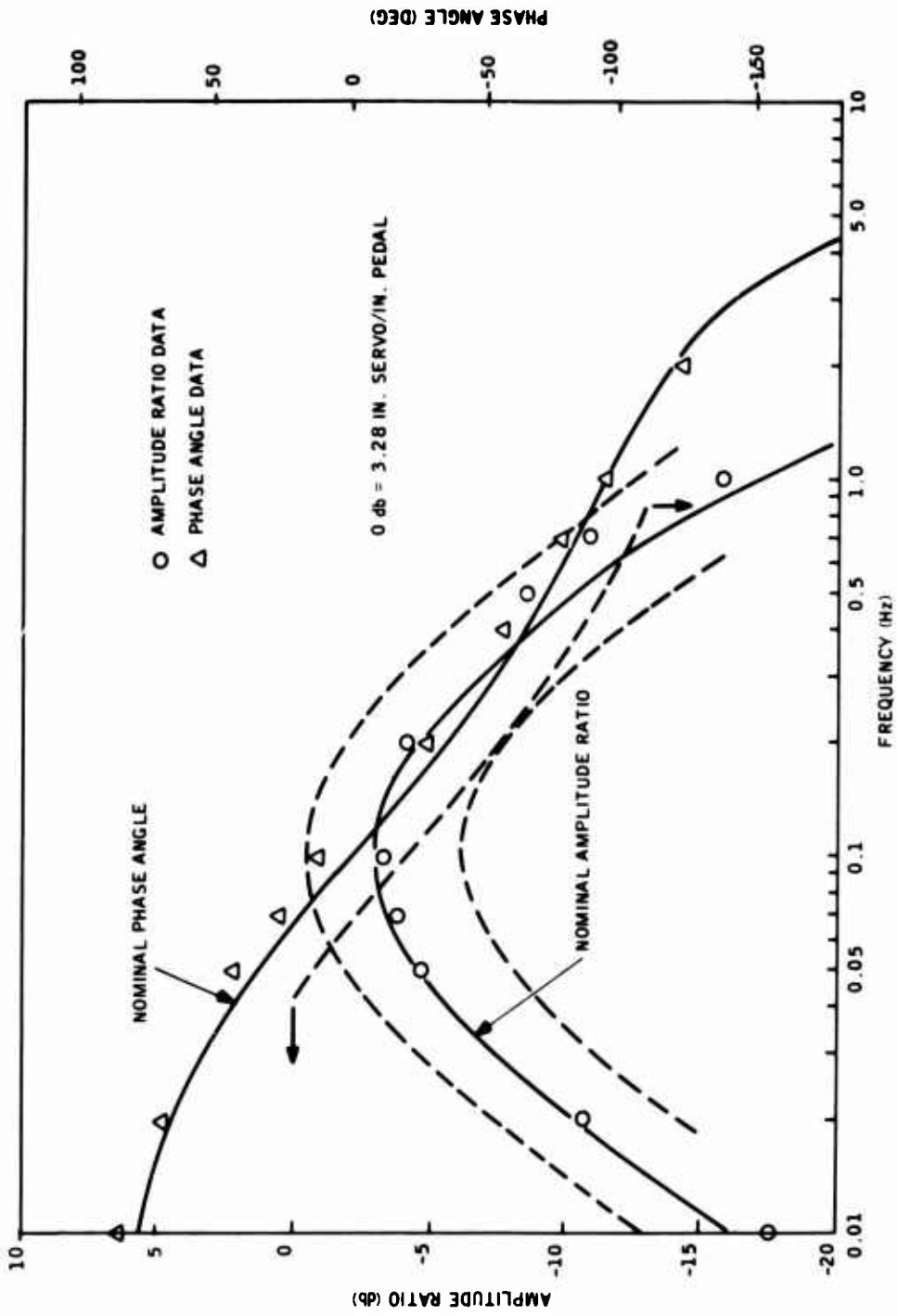


Figure 48. Yaw Pedal Input Dynamic Response at 120°F Oil Temperature

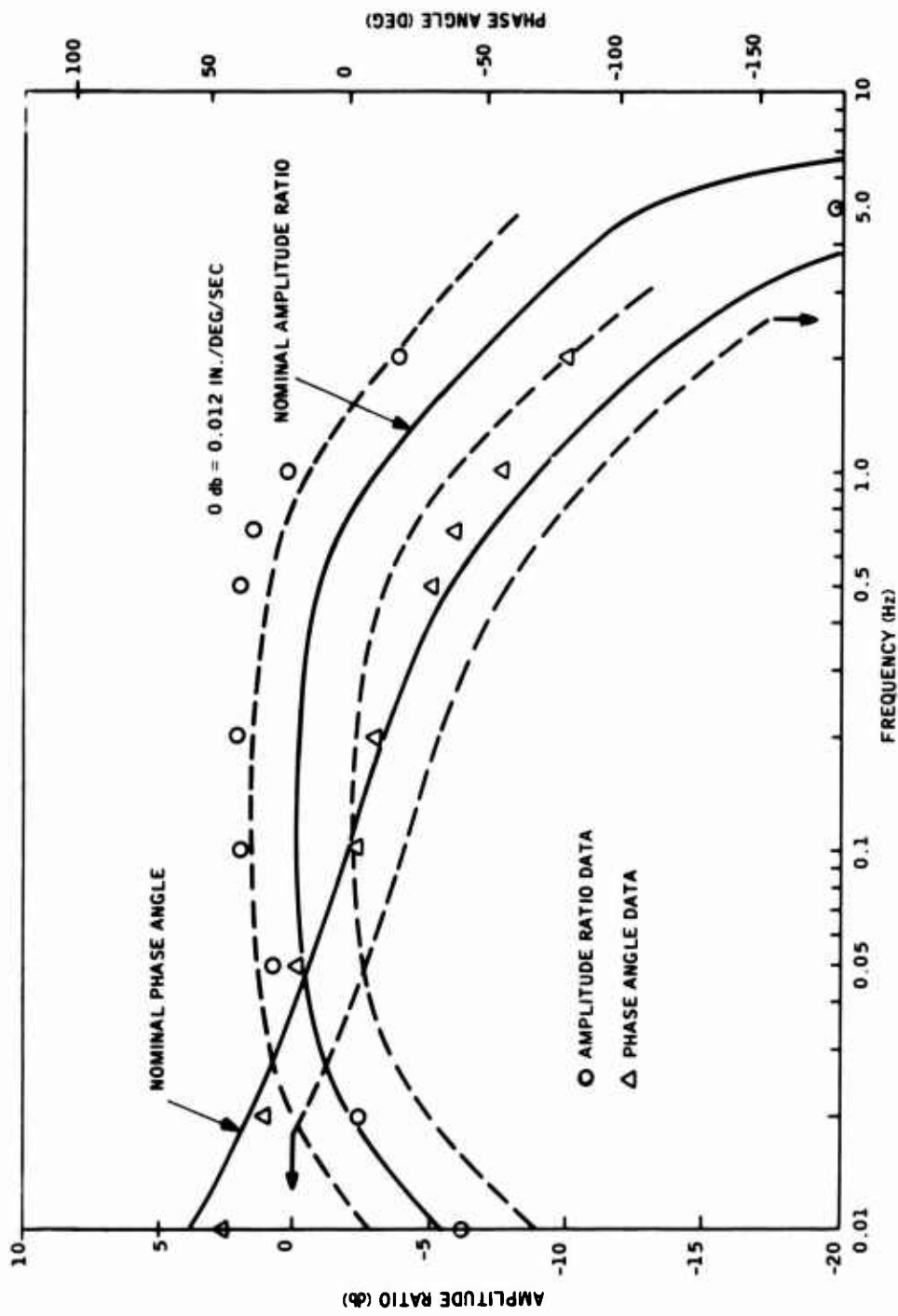


Figure 49. Roll SAS Dynamic Response

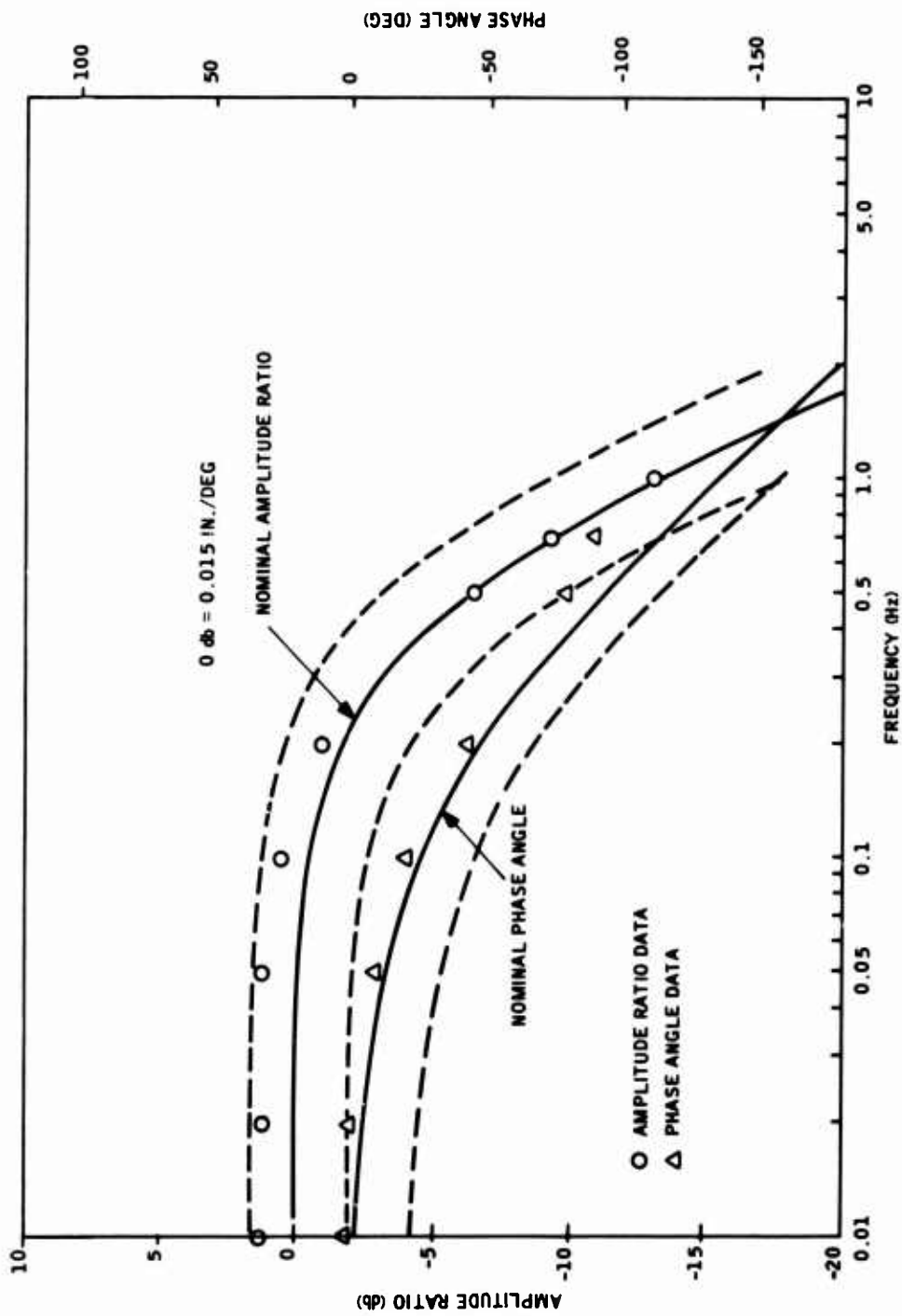


Figure 50. Roll Attitude Dynamic Response at 120°F O11 Temperature

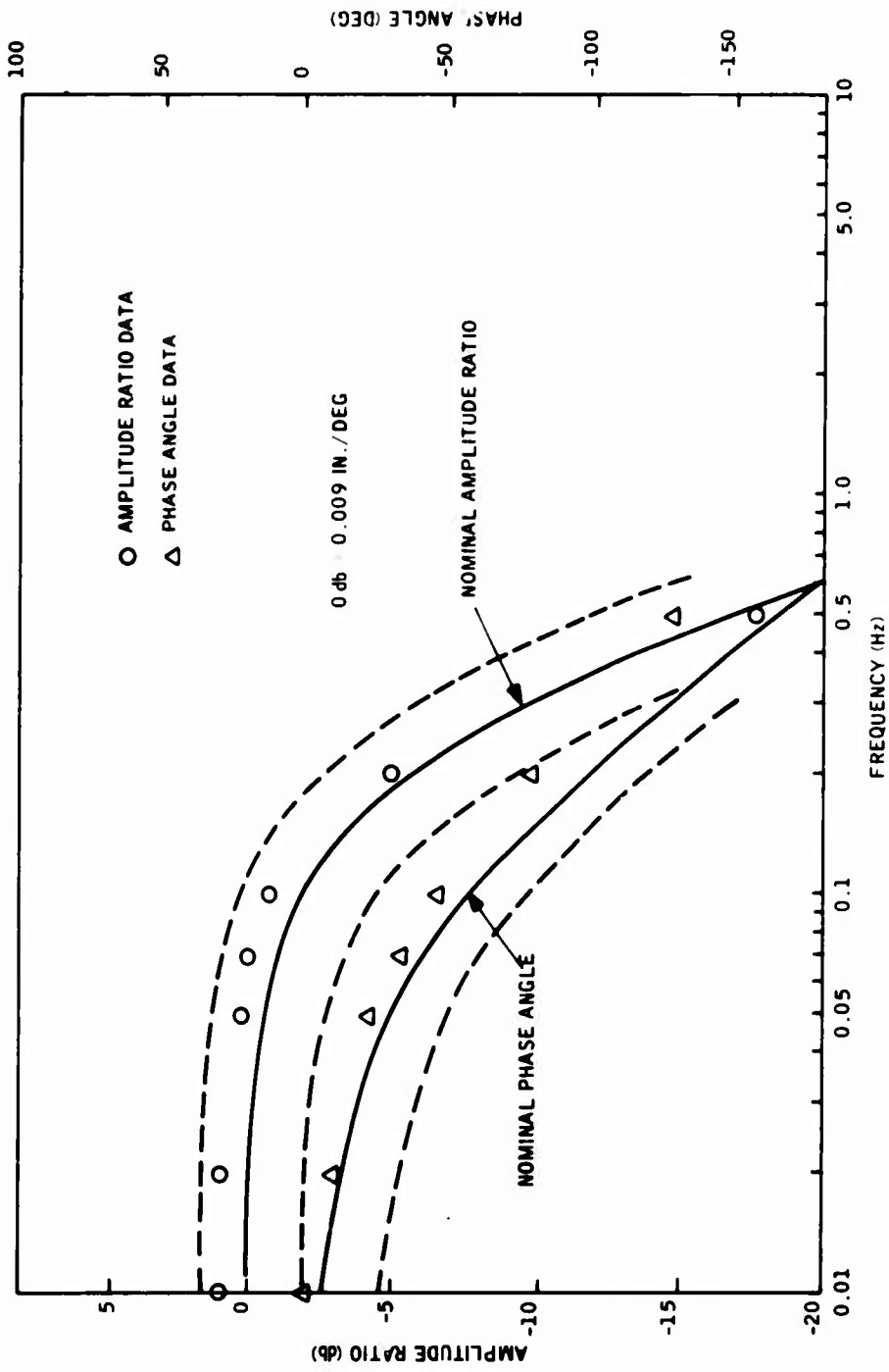


Figure 51. Heading Hold Dynamic Response at 120°F Oil Temperature

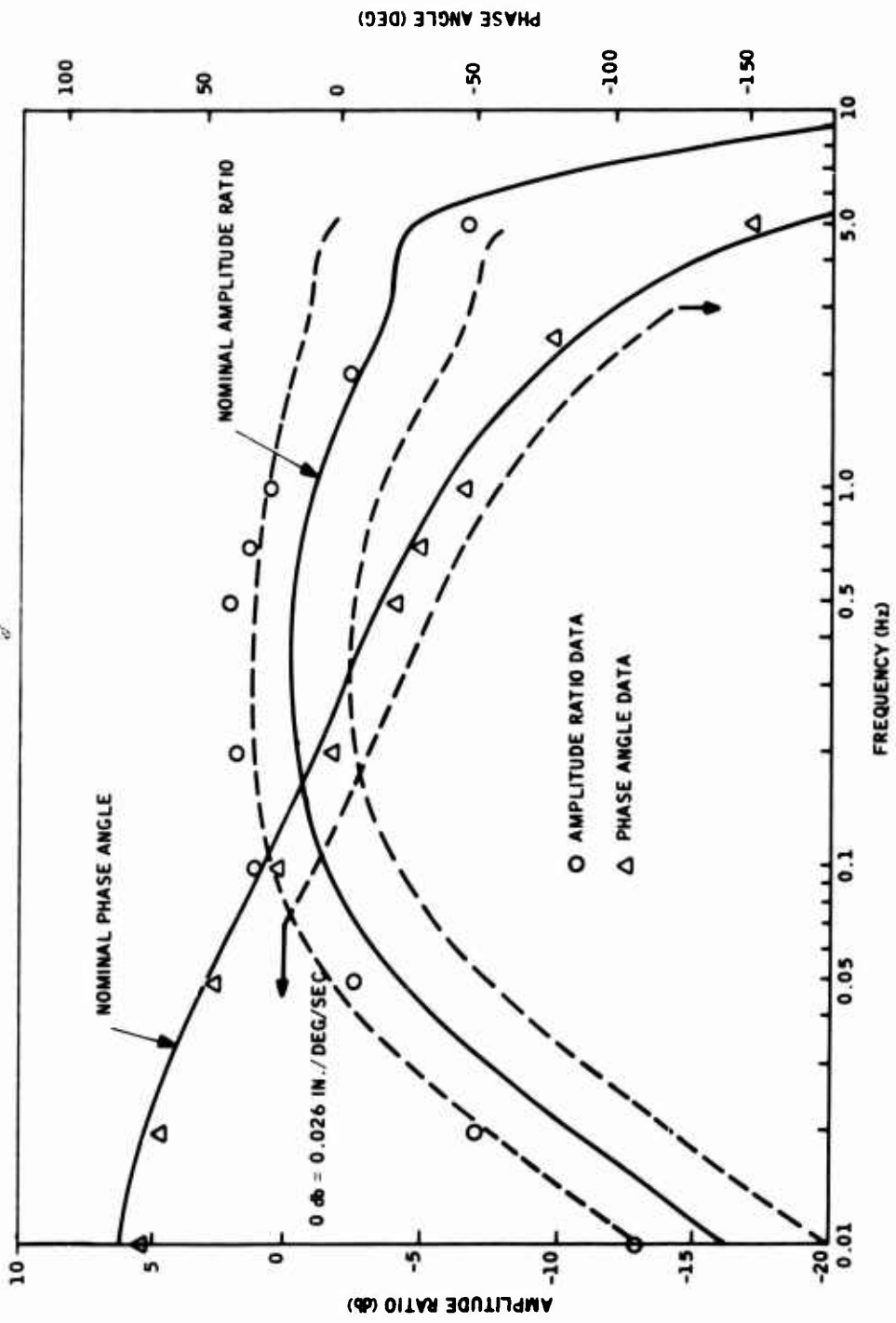


Figure 52. Pitch SAS Dynamic Response at 120°F Fluid Temperature

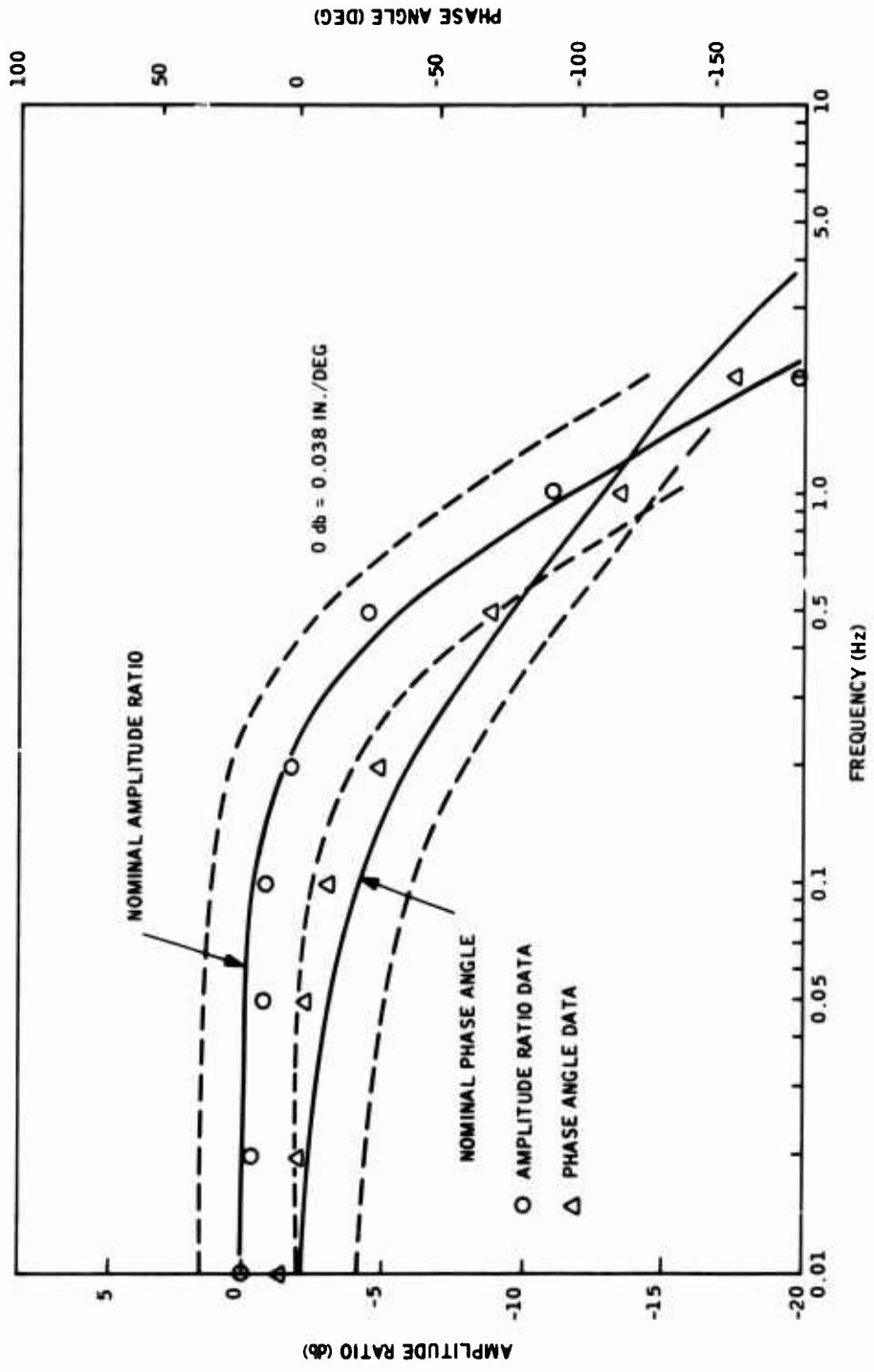


Figure 53. Pitch Attitude Dynamic Response at 120°F Fluid Temperature

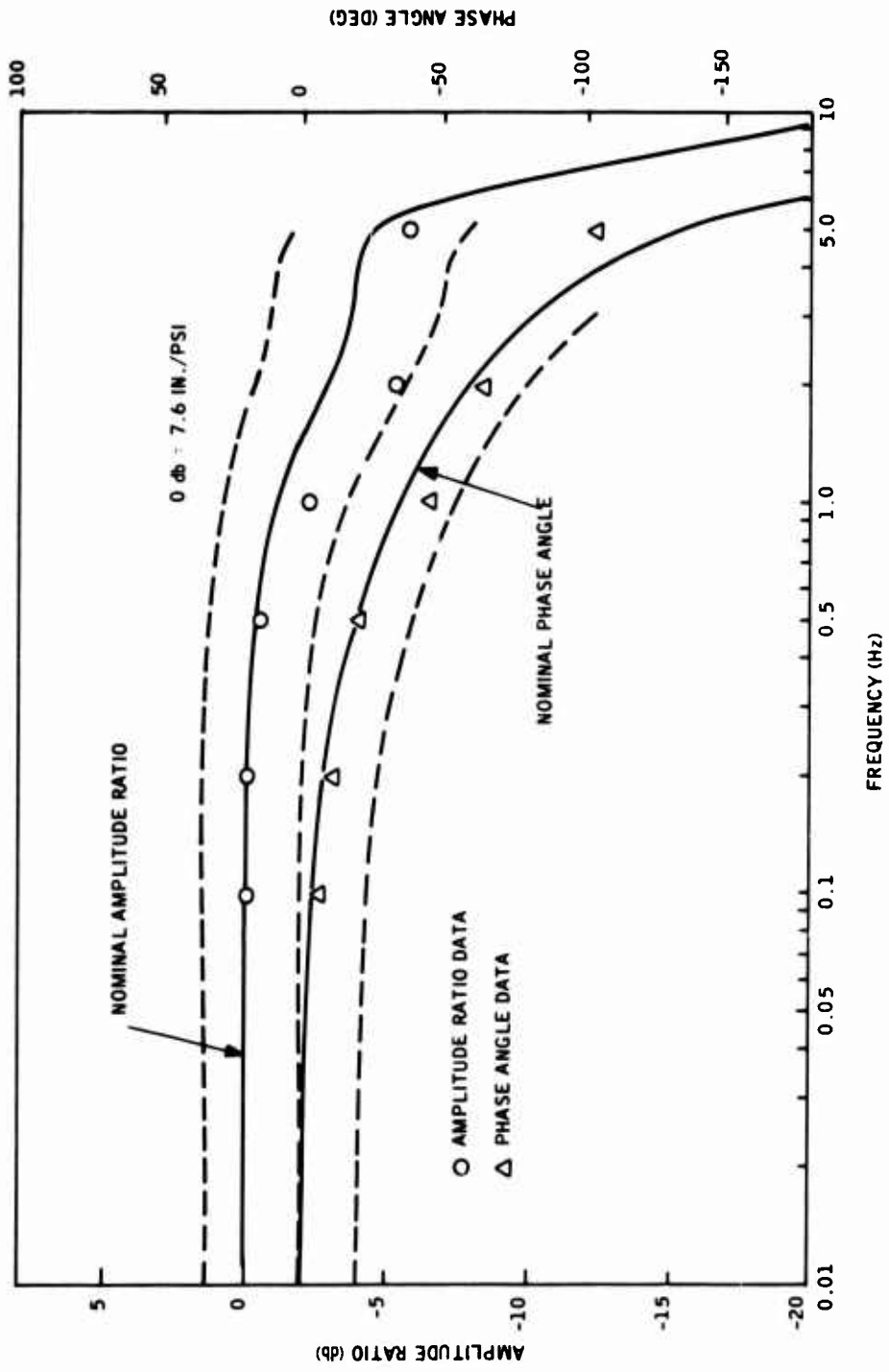


Figure 54. Altitude Dynamic Response at 120°F Fluid Temperature

The controller package was then vibrated per Curve B of Figure 514. 1, MIL-STD-810B, with a 15-minute scan 5-500-5 Hz in each of the three axes. The initial vibration scan revealed some loose bolts. These were tightened and the vibration scan repeated. No physical damage was noted from the vibration exposure, and servoactuator output motion was minimal. After the vibration test, the controller package was tested at 120°F oil temperature to determine its performance. The results, given in Table 5, were satisfactory; the only problem noted during the post-vibration tests was low gain of the yaw axis pedal input loop. This was traced to contamination, and after a valve cleaning, the performance was restored to the original condition.

CONTROL PANELS

The control panels were subjected to a vibration exposure per Curve M of Figure 514. 1, MIL-STD-810B. The vibration scan was over 5-500-5 Hz for 15 minutes in all three axes. Performance tests were conducted before and after vibration to ensure flightworthiness. They consisted of checking the panel switch functions and the corresponding voltage outputs and testing the pitch attitude, roll attitude, and heading hold electronic circuits for gain and function. Inputs were from a vertical gyro and a heading indicator; outputs were the voltages that drive the E/F transducers in the controller package.

Table 5 shows the results of the panel switch function test. The test results of the roll attitude, pitch attitude, and heading hold loops are shown in Table 6. Figures 55, 56, and 57 show the output curves of the three circuits before and after vibration. No structural or performance problems resulted from the vibration tests; the units were judged flight-worthy.

SYNCHRONIZER AND STICK TRIM CIRCUITS

Because the synchronizer and stick trim circuit were system modifications, they were subjected to a flightworthiness test as a component. The package, as shown in Figure 19, was vibrated per Curve M of Figure 514. 1, MIL-STD-810B with a 15-minute scan of 5-500-5 Hz in three axes.

Functional checks of the pressure transducer (carrier demodulator loop, synchronizer loop, and trim loop) performed before and after vibration showed no change in performance.

Table 5. Panel Switch Function Test

Test No.	Master Engage	Pitch SAS	Roll SAS	Yaw SAS	Pitch Attitude	Roll Attitude	Altitude Hold	Board Hold	Results		Voltage Checks			
									Before Vibration	After Vibration	Pin Voltage	Should Be	Results	
													Before	After
1	0 0	0 0	0 0	0 0	0 0	0 0	0 0	0 0	OK OK	OK OK	None			
2	X 0	0 0	0 0	0 0	0 0	0 0	0 0	0 0	OK OK	OK OK	None			
3	0 0	X X	0 0	0 0	0 0	0 0	0 0	0 0	OK OK	OK OK	1 H	28V 28	OK OK	OK OK
4	X X	X X	0 0	0 0	0 0	0 0	0 0	0 0	OK OK	OK OK	1 H	28 28	OK OK	OK OK
5	X X	X X	0 0	0 0	X X	0 0	0 0	0 0	OK OK	OK OK	1 H	28 28	OK OK	OK OK
6	X X	X X	0 0	0 0	X X	0 0	X X	0 0	OK OK	OK OK	1 H N	28 28 28	OK OK OK	OK OK OK
7	X 0	0 0	0 0	0 0	X 0	0 0	X 0	0 0	OK OK	OK OK	1 H N	0 0 0	OK OK OK	OK OK OK
8	0 0	0 0	X X	0 0	0 0	0 0	0 0	0 0	OK OK	OK OK	1 H	28 28	OK OK	OK OK
9	X X	0 0	X X	0 0	0 0	0 0	0 0	0 0	OK OK	OK OK	1 H	28 28	OK OK	OK OK
10	X X	0 0	X X	0 0	0 0	X X	0 0	0 0	OK OK	OK OK	1 H G	28 28 28	OK OK OK	OK OK OK
11	X X	0 0	X X	0 0	0 0	X X	0 0	X X	OK OK	OK OK	None			
12	X 0	0 0	0 0	0 0	0 0	X 0	0 0	X 0	OK OK	OK OK	1 H G	0 0 0	OK OK OK	OK OK OK
13	0 0	0 0	0 0	X X	0 0	0 0	0 0	0 0	OK OK	OK OK	1 H	28 28	OK OK	OK OK
14	X X	0 0	0 0	X X	0 0	0 0	0 0	0 0	OK OK	OK OK	1 H	28 28	OK OK	OK OK
15	X X	X X	X X	X X	X X	X X	X X	X X	OK OK	OK OK	None			
16	X X	0 0	X X	X X	X 0	X X	X 0	X X	OK OK	OK OK	None			
17	X X	0 0	0 0	X X	X 0	X 0	X 0	X 0	OK OK	OK OK	None			
18	X 0	0 0	0 0	0 0	X 0	X 0	X 0	X 0	OK OK	OK OK	None			

Notes:

Function shown in first row: 0 = off, X = actuated
Required output indication shown in second row

Press the master engage switch after the SAS switches when both types are indicated.

Table 6. Test Results of the Roll Attitude, Pitch Attitude, and Heading Hold Loops

	Test Activity	Test Point (Performance Goal)	Test Result	
			Before Vibration	After Vibration
Heading Circuit	Zero Heading Indicator Signal	Input (< 50 mV) Output (Zero)	< 50 mV 0	< 50 mV 0
	Engage Heading Hold Switch	Output (< 50 mV)	< 50 mV	< 50 mV
	Circuit Gain ($\pm 40^\circ$)	Output	Ref. Figure 57	Ref. Figure 57
	Zero Indicator Signal. Apply ± 5 Vdc to TP 13.	TP 12 (≈ 0.5 Vdc) Output (≈ 5 Vdc)	0.45 4.8	0.47 4.8
	Disengage heading hold switch.	Output (< 50 mV)	< 50 mV	< 50 mV
Pitch Attitude Circuit	Zero gyro pitch attitude signal	Input (< 50 mV) Output (Zero)	< 50 mV 0	< 50 mV 0
	Engage Pitch Attitude	Output (< 50 mV)	14 mV	5 mV
	Circuit Gain ($\pm 30^\circ$)	Output	Ref. Figure 56	Ref. Figure 56
	Zero gyro signal. Apply ± 5 Vdc to TP 3.	TP4 (≈ 0.5 Vdc) Output (≈ 5 Vdc)	± 0.52 ± 5.25	± 0.52 ± 5.2
	Remove ± 5 Vdc signal. Turn pitch trim wheel to both extremes.	TP5 (≈ 15 Vdc) Output (≈ 2.4 Vdc)	± 15 ± 2.4	± 15 ± 2.4
Roll Attitude Signal	Zero gyro roll attitude signal	Input (< 50 mV)	< 50 mV	< 50 mV
	Engage Roll Attitude	Output (< 50 mV)	< 50 mV	< 50 mV
	Circuit Gain ($\pm 30^\circ$)	Output	Ref. Figure 55	Ref. Figure 55
	Zero gyro. Apply ± 5 Vdc to TP9.	TP7 (≈ 0.5 Vdc) Output (≈ 5.0 Vdc)	± 0.54 ± 5.45	± 0.54 ± 5.45
	Remove ± 5 Vdc signal. Turn Roll Trim wheel to both extremes.	TP8 (≈ 15 vdc) Output (≈ 2.4 Vdc)	± 15 ± 2.55	± 15 ± 2.55
	Turn "Turn Control" knob to both extremes.	TP6 (≈ 15 Vdc) Output (≈ 5 Vdc)	± 15 ± 5.35	± 15 ± 5.35

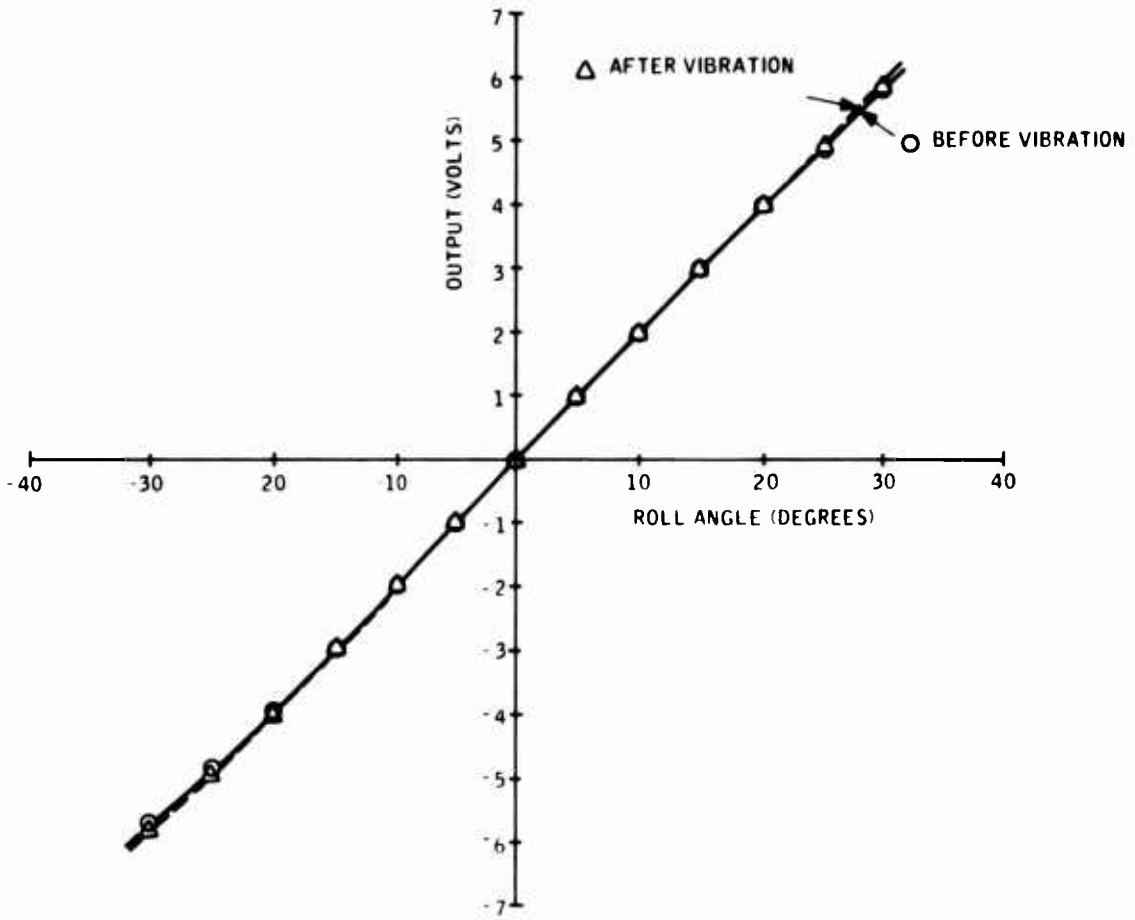


Figure 55. Roll Attitude Electronic Circuit Gain

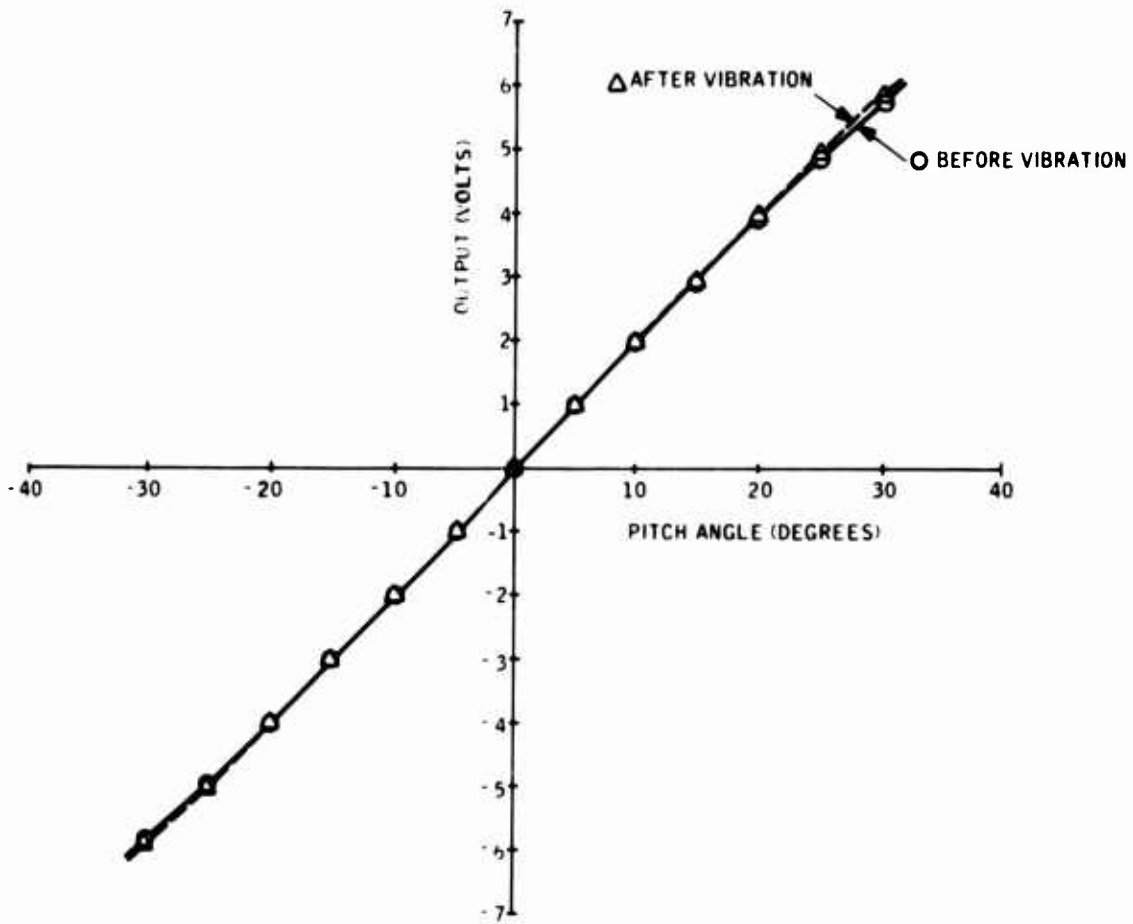


Figure 56. Pitch Attitude Electronic Circuit Gain

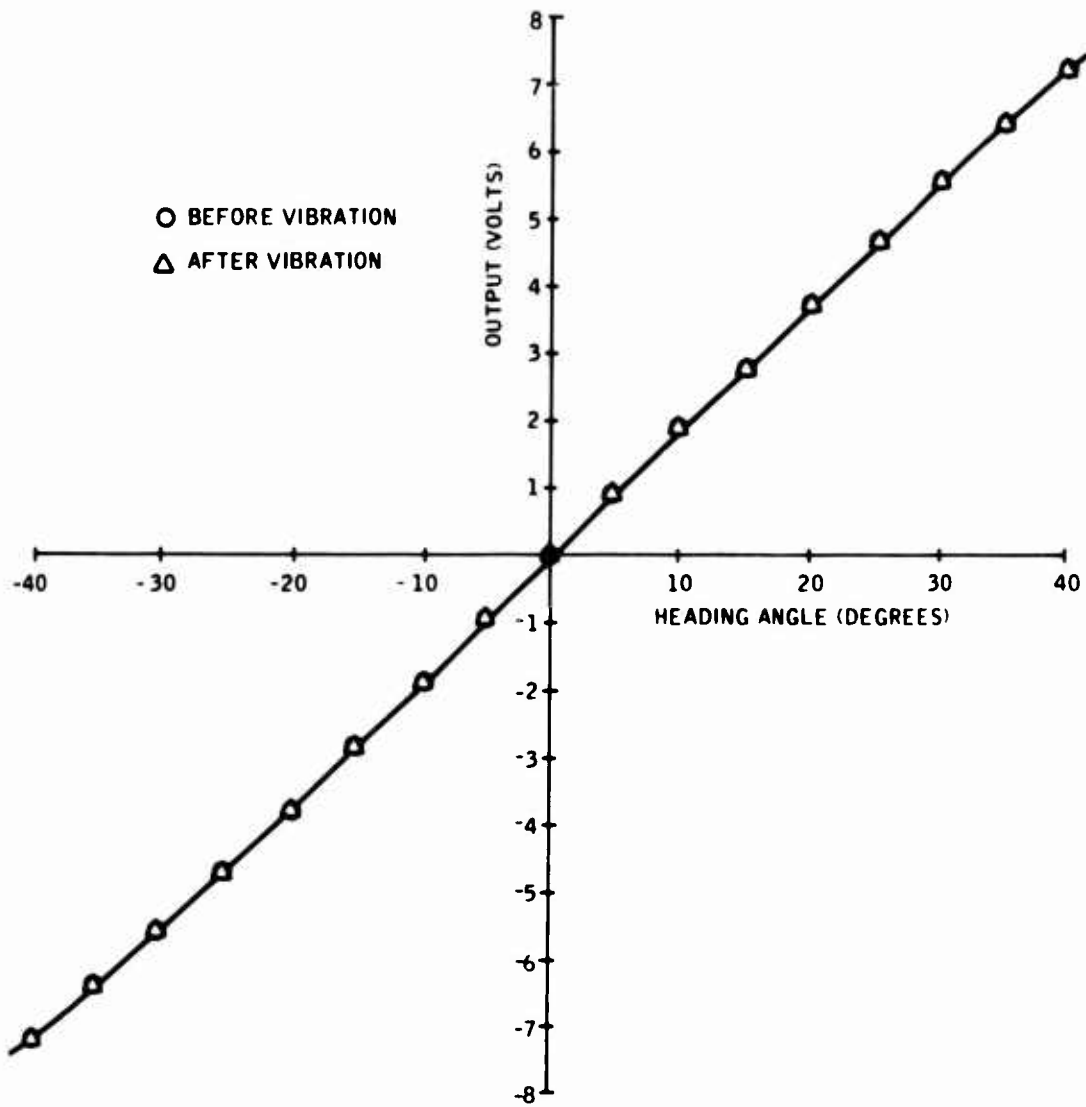


Figure 57. Heading Hold Electronic Circuit Gain

DYNAMIC PRESSURE VARIATION (AIRSPEED) SENSOR

The airspeed hold sensor shown in Figure 32 (the other system modification) was also subjected to flightworthiness tests as a component. The sensor was subjected to vibration per Curve M of Figure 514.1, MIL-STD-810B with a 15-minute scan between 5-500-5 Hz in three axes. Both performance tests conducted before and after vibration and a physical examination showed the sensor to be functionally and structurally sound.

SECTION VI

SYSTEM INSTALLATION AND INSTRUMENTATION

This section describes the installation of the system package, servo-actuators, and instrumentation in the test vehicle and the mechanical, hydraulic, and electrical work necessary for installation. Installation of the system and subsequent flight test were performed on a U. S. Army UH-1M helicopter, S/N 66-15054.

MECHANICAL INSTALLATION

Servoactuators

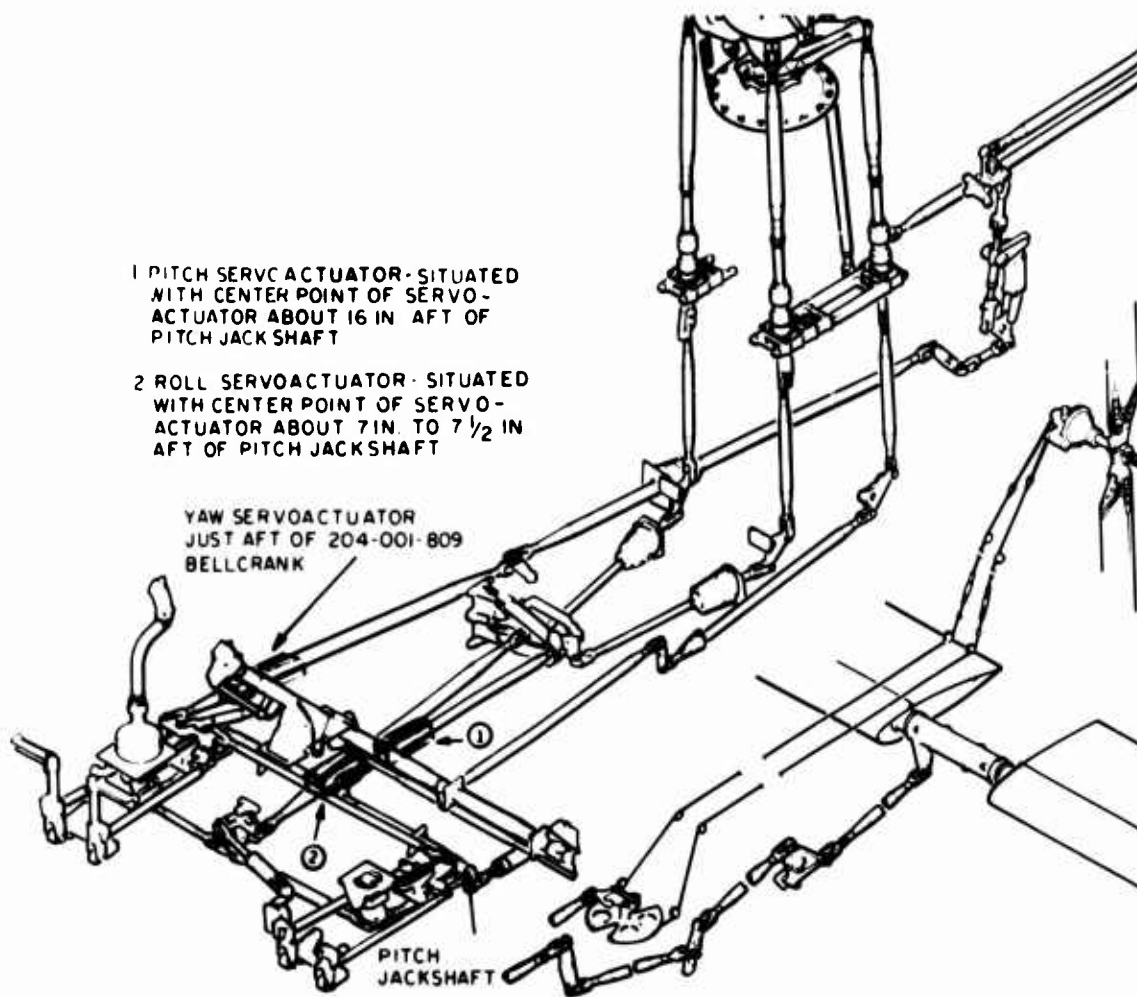
Series servoactuators were installed in the yaw, roll, and pitch axis control linkages. Figure 58 is a schematic showing the location of the servoactuators in the control linkages. Bulkheads below the cabin floor at Stations 52.00, 66.00, and 78.00 had to be modified to provide clearance for installation of the servoactuators. To support the roll and pitch servo-actuator, idler arm assemblies were mounted at Stations 66.00 and 78.00, and stop assemblies were installed at Stations 161.61 and 123.00 to limit travel of the control tubes.

Installation hardware, such as control tubes, idler assemblies, and stop assemblies, and definition of the aircraft modification were obtained from Contract DAAJ02-70-C-0017, which was a flight test of a three-axis hydro-fluidic SAS. More detailed information on the servoactuator installation can be obtained from USAAMRDL Technical Report 71-34, Three-Axis Fluidic Stability Augmentation System Flight Test Report.

As in the previous program, the major problem was interference between the full forward position of the roll axis servoactuator supply hoses and the link rod that connects the pilot's and copilot's throttle control. As before, the link rod was removed and a placard was installed to note the inoperative copilot's throttle.

Stabilizer Bar

The UH-1 helicopter normally has a stabilizer bar as shown in Figure 59 to augment pitch and roll control. The system flight test was conducted with the stabilizer bar removed and brackets used to connect the control arms to the rotor mast. This configuration is shown in Figure 60.



- 1 PITCH SERVO ACTUATOR - SITUATED WITH CENTER POINT OF SERVO-ACTUATOR ABOUT 16 IN. AFT OF PITCH JACKSHAFT
- 2 ROLL SERVOACTUATOR - SITUATED WITH CENTER POINT OF SERVO-ACTUATOR ABOUT 7 IN. TO 7 1/2 IN. AFT OF PITCH JACKSHAFT

YAW SERVOACTUATOR
JUST AFT OF 204-001-809
BELLCRANK

PITCH
JACKSHAFT

Figure 58. Servoactuator Installation Schematic



Figure 59 UH-1 With Stabilizer Bar.

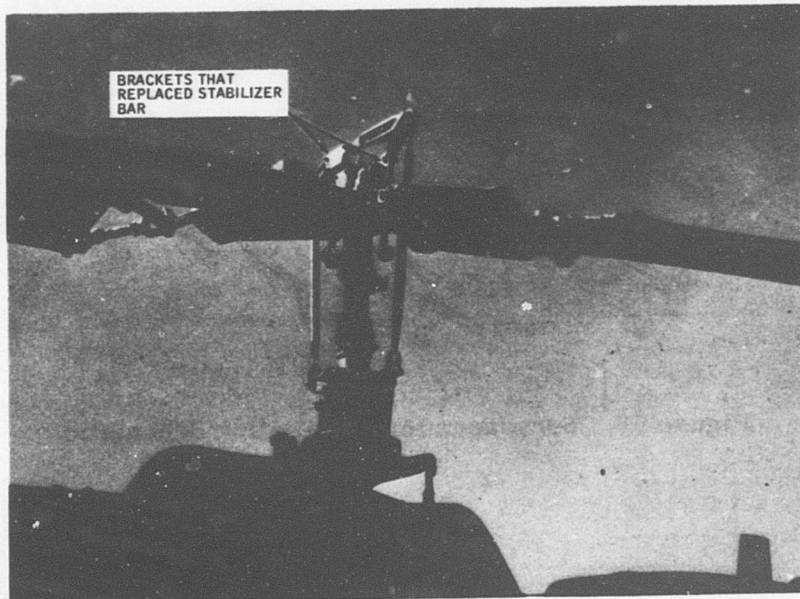


Figure 60 UH-1 Without Stabilizer Bar.

Pilot Input Fixture

A fixture was designed and installed to provide controlled pilot step and pulse inputs into the pedals and the cyclic stick during data flights. Figure 61 shows the fixture base that was mounted on the cabin floor around the copilot's cyclic stick. This provided a base to hold the input fixture. The input fixture has movable jaws to allow for various sizes of step and pulse inputs. Also incorporated into the input fixture was a thin wire that would act as a shear pin. This was a safety precaution in case the input fixture restrained the pilot from making a recovery maneuver control application; he could apply enough force to break the wire and free the stick or pedals.

System

The system was mounted on the cabin floor behind the pilot's seat as shown in Figure 62. This allowed for easy accessibility to the system for calibration checks and performance changes. The system control panels and trim indicators were mounted on the center pedestal and are shown in Figure 63.

HYDRAULIC INSTALLATION

Hydraulic power for the system was obtained from the No. 2 aircraft hydraulic supply through the armament solenoid. A schematic of the system connections is shown in Figure 64. Several cabin floor panels were removed and replaced with high-strength aluminum alloy sheet. This allowed for the mounting of bulkhead fittings in the floor to provide connections for the hydraulic signals between the system and servoactuators. The new floor panels and some of the hydraulic fittings are shown in Figure 62. Flexible hoses were used to connect the floor panels and the servoactuators to allow the servos to move. These hoses had to be positioned so as to eliminate any external forces on the control linkage.

ELECTRICAL INSTALLATION

Electrical power to the system was provided from the aircraft's 28-volt dc nonessential bus and the 115-volt ac, 400-Hz instrument inverter. Circuit breakers protect each power supply. Signals from the aircraft's attitude gyro and heading indicator were obtained by splicing wires into the appropriate connector terminals and routing the wires to the system.

Emergency disengage switches were mounted on both pilot's and copilot's cyclic stick for a quick turn-off of the system. Wires were run from these switches to the system.



Figure 61. Cyclic Stick and Pedal Input Fixture

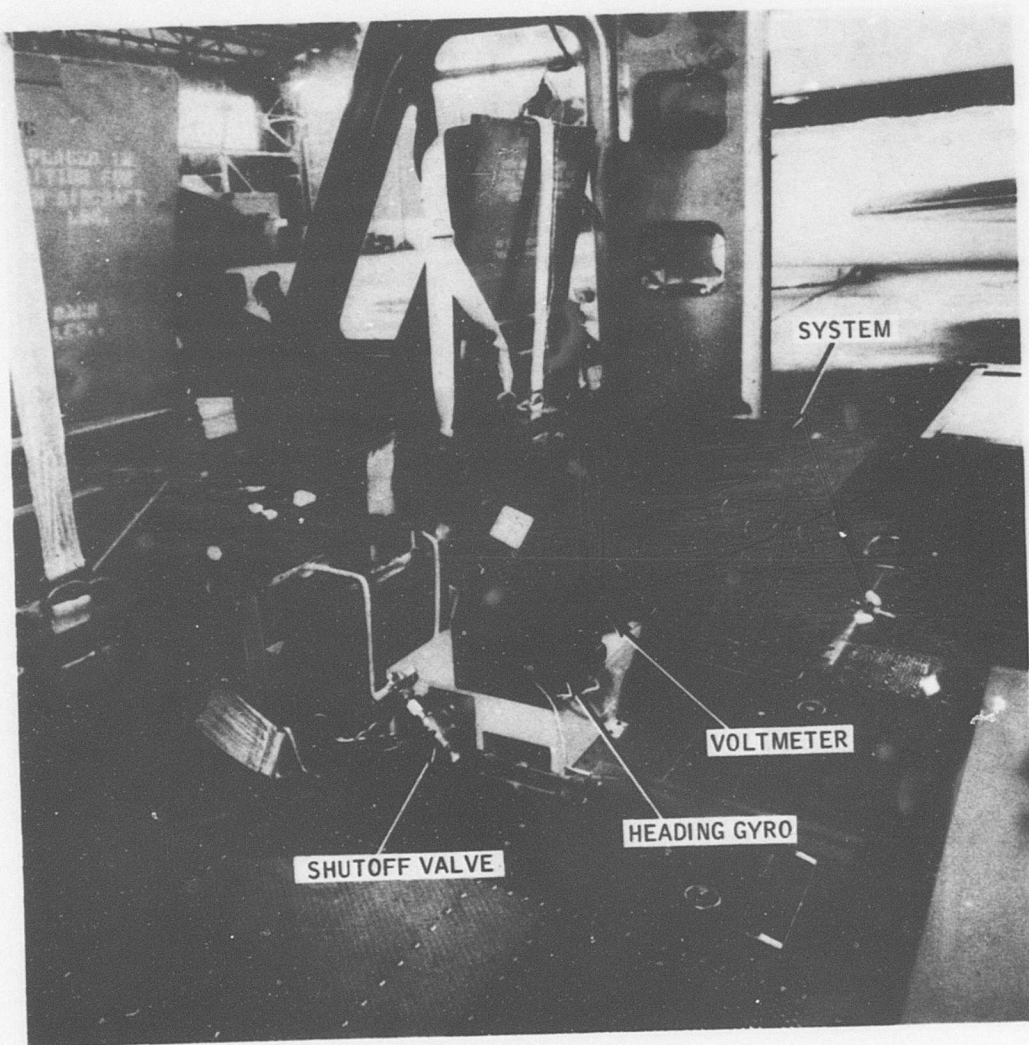


Figure 62. System Mounted on Cabin Floor

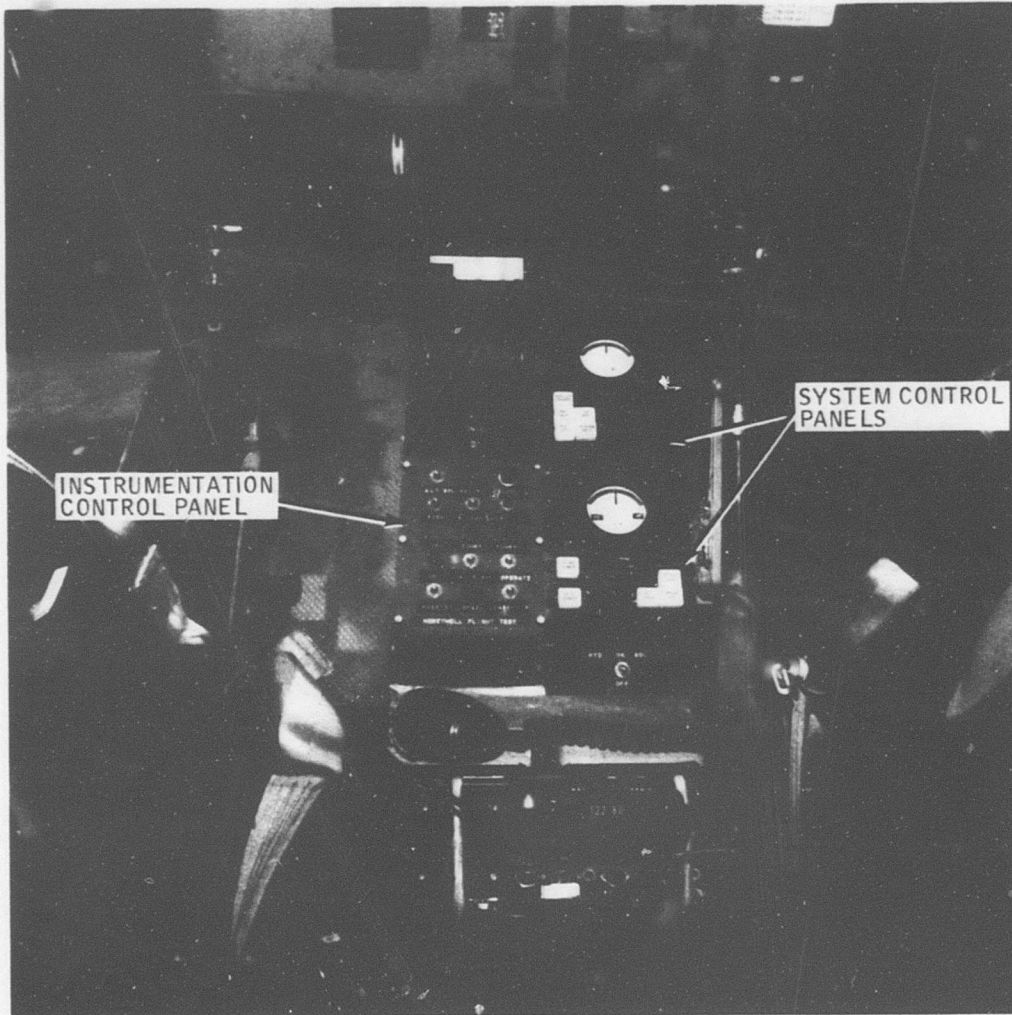


Figure 63. Control Panels Mounted on Pedestal

Two 1500-watt, 115-volt ac, 400-Hz inverters were mounted in the aft storage compartment to provide electrical power for the test instrumentation. Power for the inverters was obtained from the 28-volt dc non-essential bus, and overload protection was provided by two circuit breakers.

FLIGHT TEST INSTRUMENTATION

Test instrumentation was mounted in the aft portion of the cabin as shown in Figures 65 and 66. Figure 66 shows the 24-channel visicorder that recorded the following parameters:

- a/c roll rate
- a/c pitch rate
- a/c yaw rate
- a/c heading
- a/c roll attitude
- a/c pitch attitude
- Pedal position
- Time correlation
- Lateral acceleration
- a/c altitude
- Pitch attitude hold mode engaged
- Heading hold mode engaged
- a/c airspeed
- Collective position
- Lateral cyclic position
- Longitudinal cyclic position
- Roll axis servo motion
- Pitch axis servo motion
- Yaw axis servo motion
- Event marker
- Longitudinal acceleration
- Vertical acceleration
- Hydrofluidic controller supply pressure
- Roll attitude hold mode engaged
- Altitude hold mode engaged
- Airspeed hold mode engaged

Three electronic rate gyros measured the aircraft turning rates in the three axes, a directional gyro measured the aircraft heading, and a vertical gyro measured the roll and pitch attitude. A three-axis accelerometer package measured the aircraft accelerations, and the aircraft airspeed was measured by a low-speed airspeed sensor mounted on the outside of the aircraft. These instruments are shown in Figures 65 and 66. The aircraft control linkage positions were measured by potentiometers mounted under the cabin floor and connected to the linkages. Servoactuator motions were measured by linear variable displacement transducers that were built into the actuator housing. In addition to the 24-channel visicorder, an 8-channel recorder was used to record selected parameters for data analysis. An instrumentation control panel was mounted on the center pedestal to provide off/on control, recorder speed control, gyro caging, altitude sensor engagement, and event marking. The control panel is shown in Figure 63.



Figure 65. Instrumentation Mounted in Cabin - Viewed from Left Side



Figure 66. Instrumentation Mounted in Cabin - Viewed from Right Side

SECTION VII FLIGHT TEST

TEST PROCEDURES

The system was quantitatively and qualitatively evaluated at the flight conditions given in Table 7. At each flight condition, the SAS and outer-loop responses were recorded and the outer-loop steady-state performance was obtained. The test inputs for each flight condition are given in Table 8. The test procedures used to obtain the flight test data are explained below.

- Aircraft response with and without SAS was obtained by step and pulse inputs into the yaw and pitch axes and step inputs into the roll axis. These inputs were controlled by the input fixture shown in Figure 61. Pulses were maintained for about one second. Steps and pulses were made in both directions, and the aircraft responses were recorded for both unaugmented (free) aircraft and augmented (SAS on) aircraft. Free aircraft was defined as one being without the mechanical stabilizer bar.
- Roll and pitch attitude loop response tests were performed by applying a bias voltage into the demodulator and trim electronics to represent an error signal. The aircraft response to this error signal was then recorded. Pitch and roll loop response tests were conducted separately with both attitude hold loops engaged and the aircraft in a trim condition.
- Heading hold response data was obtained by moving the "bug" on the aircraft's heading indicator to introduce a step error signal. The aircraft's response was then recorded. The aircraft's longitudinal axis was controlled by the altitude hold loop.
- A step input into the altitude hold loop was accomplished by changing the pressure in the reference chamber of the altitude sensor while the altitude hold was engaged. This pressure change was introduced by an external bellows that could be compressed or extended manually to provide the step input pressure. Heading hold was engaged during these tests to control the lateral axis.

Table 7. Flight Test Condition

Altitude (ft-MSL)	Airspeed (KIAS)
2000 +1000 -0	Hover
2000 +1000 -0	60
2000 +1000 -0	120
5000 +1000 -0	60
10,000 +1000 -0	60

Table 8. Flight Test Inputs

Mode	Input
SAS'S Engaged	Pitch Cyclic Step Pitch Cyclic Pulse Roll Cyclic Step Yaw Pedal Step Yaw Pedal Pulse
Attitude Hold's Engaged	Roll Attitude Step Pitch Attitude Step 15- Minute straight and level accuracy flight*
Heading Hold and Altitude Hold/Airspeed Hold Engaged	Heading Step Altitude Step Airspeed Step 15-Minute straight and level accuracy flight*

*15- Minute straight and level accuracy flight test data was not taken at the hover flight condition.

- **Airspeed hold step inputs were accomplished by changing the pressure in the total pressure reference chamber of the airspeed sensor. Like the step inputs to the altitude loop, the step inputs were introduced by an external bellows while the heading hold loop was engaged.**
- **Steady-state data was obtained by engaging outer-loop control, trimming the aircraft, and recording the aircraft's performance over a 15-minute period. Flight path deviations and drifts were noted. The steady-state data was obtained with the following combination of outer loops engaged:**
 - **Roll attitude and pitch attitude**
 - **Roll attitude, pitch attitude, heading hold, and altitude hold**
 - **Roll attitude, pitch attitude, heading hold, and airspeed hold**

In addition to the previously defined tests, the SAS performance was qualitatively evaluated under autorotational entries and for engage and disengage transients at 60, 90, and 120 knots at 3000 feet. Aircraft response data to yaw and pitch steps and pulses and to roll steps was also obtained at hover at 3000 feet and at 60 knots at 3000 feet with the mechanical stabilizer bar for comparison.

FLIGHT TEST RECORDINGS

The flight test recordings presented in Appendix C show aircraft response to various inputs under the different flight conditions. Analysis of these flight recordings is discussed in the following sections.

SAS PERFORMANCE

Yaw SAS Performance

Yaw SAS performance was evaluated at the flight conditions listed in Table 7. The principal criteria for evaluating yaw SAS performances were control sensitivity and damping ratio. Yaw-axis control sensitivity is defined as the ratio of peak yaw rate attained for a given pedal step input command. Desired yaw-axis performance is achieved when the control sensitivity of the helicopter is not significantly decreased when the yaw SAS is engaged. A small decrease in control sensitivity is permissible providing it has the effect of giving a pilot the capability

to perform yaw-axis maneuvers in a smooth and more controlled manner. Maintaining the inherent augmented helicopter's control sensitivity is especially important at hover and at low-speed flight conditions.

The damping ratio criterion is satisfied if the underdamped characteristics of the helicopter's yaw axis are eliminated and the yaw-axis damping ratio is increased from approximately 0.15 to 0.7. This criterion was investigated by observing the yaw-rate response to pedal pulse commands that simulate wind disturbances acting on the helicopter.

Pedal Step Inputs - Yaw SAS -- Flight recordings of aircraft responses for the pedal step input tests are presented in Appendix C, Figures 72 through 76. Performance results taken from these recordings are presented in Table 9.

A comparison of the recordings for the SAS with those of the free helicopter with and without the mechanical stabilizer bar shows that the SAS provides a significant increase in the damping ratio of the yaw rate response. The augmented helicopter is easy to control and responds to a step input with a yaw rate that exhibits one overshoot. This response is generally true for the complete flight envelope of the aircraft, even though flight records were obtained for a limited number of conditions.

Control sensitivity data taken from the flight recordings are presented in Table 9. The requirements that provide the limits of yaw angle per inch of pedal deflection at 1 second shown in the requirements column of the table were taken from MIL-H-8501A.

Paragraph 3.3.5 of MIL-H-8501A states: "Directional control power shall be such that when the helicopter is hovering in still air at the maximum overload gross weight or at rated take-off power, a rapid 1.0 inch step displacement from trim of the directional control shall produce a yaw displacement at the end of 1.0 second which is at least $\frac{110}{\sqrt[3]{W + 1000}}$ degrees." W is the maximum overload gross weight of the aircraft. For the UH-1 the minimum heading angle that must be commanded at the end of 1 second is 5 degrees when using 9650 pounds as the maximum overload gross weight.

Paragraph 3.3.7 states: "The sensitivity shall be considered excessive if the yaw displacement is greater than 50 degrees in the first second following a sudden pedal displacement of 1 inch from trim while hovering at the lightest normal service loading." The data in Table 9 show that the directional control power at hover is increased with the yaw SAS or with the stabilizer bar over that provided in the free aircraft.

Other data in Table 9 are the peak yaw rate obtained per inch of pedal deflection and the response time to reach 90 percent of the peak rate value. The data show that the response times and the peak rates were not

Table 9. Yaw SAS - Step Inputs

Flight Condition	Test	$\dot{\psi}_{\text{peak}}$ ($\frac{\text{deg/sec}}{\text{in. pedal}}$)		Time to 90% $\dot{\psi}_{\text{peak}}$ (sec)		ψ at 1 sec ($\frac{\text{deg}}{\text{in. pedal}}$)	
		Actual	Requirement	Actual	Requirement	Actual	Requirement
Hover, 3000 ft							
With SAS, without STAB	RS	17.1	*	0.85	None	8.78	5 to 50
	LS	16.1	*	0.77		9.46	
Without SAS, without STAB	RS	16.6	*	0.85		6.30	
	LS	11.4	*	0.87		6.10	
Without SAS, with STAB	RS	17.1	*	0.92		8.40	
	LS	16.0	*	0.92		8.35	
60 kn, 3000 ft							
With SAS, without STAB	RS	14.5	None	0.51		9.36	None
	LS	13.6		0.75		7.60	
Without SAS, without STAB	RS	15.9		0.51		10.80	
	LS	16.1		0.76		9.07	
Without SAS, with STAB	RS	13.6		0.60			
	LS	13.8		0.85			
60 kn, 5000 ft							
With SAS, without STAB	RS	12.6		0.54			
	LS	12.7		0.62			
Without SAS, without STAB	RS	15.0		0.52			
	LS	14.5		0.78			
60 kn, 10,000 ft							
With SAS, without STAB	RS	12.8		0.65			
	LS	13.3		0.80			
Without SAS, without STAB	RS	12.4		0.67			
	LS	12.8		0.84			
120 kn, 3000 ft							
With SAS, without STAB	RS	11.5		0.46			
	LS	11.0		0.53			
Without SAS, without STAB	RS	10.5	▽	0.61	▽		▽
	LS	7.2		0.40			

* $\dot{\psi}$ /in. shall not be so high as to cause pilot to overcontrol unintentionally.

**STAB is an abbreviation for stabilizer bar.

significantly different between the free aircraft and the augmented aircraft. There is a consistent difference, however, in the response time between left and right step commands for both augmented and free aircraft conditions that is evidently related to the basic characteristics of the aircraft.

Pedal Pulse Inputs - Yaw SAS -- Flight recordings of aircraft responses for pedal pulse input are given in Figures 77 through 83. Natural frequency and damping ratio values obtained from the flight recordings are given in Table 10. In most recordings with the yaw SAS engaged, no natural frequency values are given because the damping ratio was high enough to eliminate oscillatory motion that prevented natural frequency determination.

It appears that the natural frequency of the free aircraft increases as the forward speed increases, but no significant differences in damping ratio are evident. With the SAS engaged, the damping ratio is increased from the 0.22 to 0.46 range up to the 0.6 to 0.7 range. With the stabilizer bar, the damping ratio is approximately the same as the free aircraft.

Roll SAS Performance

Roll SAS performance was evaluated for the flight conditions listed in Table 7 according to the following criteria:

- Control sensitivity of the SAS/UH-1C without the mechanical stabilizer bar should be equal to or comparable to that of the UH-1C with the mechanical stabilizer bar.
- The roll axis should provide a steady-state roll rate in response to a roll cyclic step input with an overshoot that is acceptable to the pilots.
- Control power should be maintained at a level that prevents pilot maneuvering difficulty and shall also meet the requirements of MIL-H-8501A.

Flight recordings showing roll control and aircraft responses to a rapid roll stick input are given in Figures 84 through 89. Roll SAS performance data taken from these recordings are presented in Table 11.

The most significant improvement in roll response by the SAS was made at the hover condition where a much smoother response to the stick command is obtained compared with either the SAS off or the stabilizer bar conditions. The control sensitivity (degrees/second roll rate per inch of roll stick deflection) with SAS on is less than with SAS off, but approximately the same as with the stabilizer bar. The response time to

Table 10. Yaw SAS - Pulse Inputs

Flight Condition		Natural Frequency (Rad/Sec)		Damping Ratio			
		Actual	Requirement	Actual	Requirement		
Hover, 3000 ft							
With SAS, without STAB*	RP		None	0.60	None		
	LP			0.60			
Without SAS, without STAB	RP	1.80	↓	0.33	↓		
	LP	1.07		0.22			
Without SAS, with STAB	RP	1.00		0.27			
	LP	1.40		0.46			
60 kn, 3000 ft							
With SAS, without STAB	RP					0.70	0.6
	LP					0.70	
Without SAS, without STAB	RP	1.66				0.38	
	LP	1.73				0.24	
Without SAS, with STAB	RP	1.59				0.35	
	LP	1.84					
60 kn, 5000 ft							
With SAS, without STAB	RP			0.65			
	LP			0.65			
Without SAS, without STAB	RP	1.51		0.39			
	LP	1.47		0.46			
60 kn, 10,000 ft							
With SAS, without STAB	RP	1.79		0.70			
	LP			0.62			
Without SAS, without STAB	RP	1.59		0.33			
	LP	1.50		0.30			
120 kn, 3000 ft							
With SAS, without STAB	RP			0.70			
	LP			0.60			
Without SAS, without STAB	RP	2.30	↓	0.28	↓		
	LP	1.91		0.22			

*STAB is an abbreviation for stabilizer bar.

Table 11. Roll SAS - Step Inputs

Flight Condition		$\dot{\phi}_{peak}$ (deg/sec) (in. stick)		Time to 90% $\dot{\phi}_{peak}$ (sec)		ϕ at 0.5 sec (deg) (in. stick)				
		Actual	Requirement	Actual	Requirement	Actual	Requirement			
<u>Hover, 3000 ft</u>										
With SAS, without STAB	RS	5.1	20 deg/sec maximum	0.67	None	1.00	1.2 minimum ↓			
	LS	13.7		0.80		2.20				
Without SAS, without STAB	RS	11.9		1.30		1.10				
	LS	13.3		1.20		1.20				
Without SAS, with STAB	RS	6.1		0.84		0.75				
	LS	4.2		0.81		0.55				
<u>60 kn, 3000 ft</u>										
With SAS, without STAB	RS	4.8		0.76		0.84		None ↓		
	LS	6.8		1.26		0.80				
Without SAS, without STAB	RS	12.9		1.40		1.30				
	LS	12.3		1.53		0.77				
Without SAS, with STAB	RS	5.5		0.76		0.65				
	LS	6.0	0.66	1.05						
<u>60 kn, 5000 ft</u>										
With SAS, without STAB	RS	10.0	0.95	0.61						
	LS	5.6	0.86	0.50						
Without SAS, without STAB	RS	10.0	1.70	0.62						
	LS	10.4	1.20	1.30						
<u>60 kn, 10,000 ft</u>										
With SAS, without STAB	RS	7.8	0.80	1.23						
	LS	7.3	1.10	0.50						
Without SAS, without STAB	RS	11.5	1.28	0.40						
	LS	11.4	1.20	0.78						
<u>120 kn, 3000 ft</u>										
With SAS, without STAB	RS	7.5	0.93	1.20						
	LS	10.1	1.23	1.30						
Without SAS, without STAB	RS	14.4	1.15	1.10						
	LS	20.0	1.63	0.75						

*Shall not be so high as to cause pilot to overcontrol unintentionally

**STAB is an abbreviation for stabilizer bar.

90 percent of peak rate was reduced with SAS on compared to the free aircraft response time, partly due to the reduced peak rate achieved. In no case was the peak rate in excess of 20 degrees per second, the maximum value specified in paragraph 3.3.15 of MIL-H-8501A.

The control power requirements for lateral control are specified in paragraph 3.3.18 of MIL-H-8501A, which states: "Lateral control power shall be such that when the helicopter is hovering in still air at the maximum overload gross weight or at the rated power, a rapid 1-inch step displacement from trim of the lateral control shall produce an angular displacement at the end of one-half second of at least $\frac{27}{\sqrt{W + 1000}}$ degrees."

Using a gross weight of 9650 pounds, the computed angular displacement requirement is 1.2 degrees minimum. To evaluate this requirement, it was necessary to integrate the roll rate trace to obtain the roll angular displacement, as the attitude reference was not accurate enough at small displacements to discern tenths of degrees. The free helicopter and the helicopter with roll SAS engaged were marginal in meeting this requirement, with angular displacements at the end of 0.5 second of 1.0 to 2.2 degrees. The helicopter with the stabilizer bar was definitely below the required response level, with a measured response of 0.55 to 0.75 degree at the end of 0.5 second.

The other requirement from MIL-H-8501A found in paragraph 3.3.16 states: "The angular acceleration shall be in the proper direction within 0.2 second after control displacement." Roll angular acceleration was not a recorded parameter, but the roll rate trace response was measured consistently in the time range of 0.05 to 0.16 second after command input.

Pitch SAS Performance

Pitch SAS performance was evaluated for the flight conditions listed in Table 7 according to the following criteria:

- Control sensitivity of the SAS/UH-1C without the mechanical stabilizer bar should be comparable to that of the UH-1C with the mechanical stabilizer bar. Control sensitivity should not be so high that it causes the pilot to experience maneuvering difficulties.
- Longitudinal control power should not be less than 2.04 degrees/inch when the helicopter is hovering in still air at the maximum overload gross weight or at the rated power.
- Pitch axis damping ratio should be increased from approximately 0.3 to approximately 0.5 or greater at or near the 100-knot flight condition.
- The pitch SAS should provide a rate proportional to control-stick deflection.

Cyclic Stick Step Input Commands -- Flight recordings of aircraft responses for the pitch stick step inputs are given in Figures 90 through 94. The pitch rate response to the pilot-commanded stick deflection for the free aircraft is a 3- or 4-second pulse of pitch rate. This corresponds to the results obtained during system analysis. The change in airspeed affects the rate response such that it is not proportional to stick deflection. Repeating the step responses with the pitch SAS engaged shows that at hover and at high speed (120 knots) the response is more like a pitch rate proportional to stick displacement. At the 60-knot conditions there is some improvement over the free-aircraft responses in that pitch rate does not drop back toward zero as fast as with the SAS engaged. Step responses with the stabilizer bar are similar to the responses obtained with the SAS engaged.

Quantitative data obtained from the flight recordings are given in Table 12. These include the peak pitch rate per inch of stick deflection, the response time to 90 percent of peak rate (T_{90}), and the control power values of angular displacement per inch of stick at one second after stick command. The peak rate per inch of stick for the free aircraft is consistently higher than that of the SAS augmented aircraft. The aircraft with the stabilizer bar achieves about the same pitch rate at hover as does the aircraft with SAS engaged; but at the 60-knot, 3000-foot condition, it achieves a peak pitch rate somewhat greater than the free aircraft.

The T_{90} response time is consistently longer for the free aircraft than for the augmented aircraft because higher peak rates are achieved with the free aircraft and greater time is required to achieve the higher rates. There is a general indication that "up" responses are somewhat slower than "down" responses, but this is not true for all conditions tested.

The minimum control power requirement is determined from paragraph 3.2.13 of MIL-H-8501A, which states: "Longitudinal control power shall be such that when the helicopter is hovering in still air at the maximum overload gross weight or at rated power, a rapid 1.0 inch step displacement from trim of the longitudinal control shall produce an angular displacement at the end of 1.0 second which is at least $\frac{45}{\sqrt{W + 1000}}$ degrees." The limit value for a maximum overload gross weight of 9650 pounds is 2.04 degrees. Both the free aircraft and the SAS-augmented aircraft meet this requirement. The aircraft with stabilizer bar did not meet the requirement for an "up" step command where an angular displacement of 1.65 degrees was measured.

Paragraph 3.2.9 of MIL-H-8501A states: "The angular acceleration shall be in the proper direction within 0.2 second after longitudinal control displacement." As angular acceleration is not a recorded parameter, the rate response time was evaluated. The rate traces show a proper response from 0.12 to 0.2 second after the step command. The SAS augmentation does not affect this initial response.

Table 12. Pitch SAS - Step Inputs

Flight Condition		$\dot{\theta}_{peak}$ (deg/sec) (in. stick)		Time to 90% $\dot{\theta}_{peak}$ (sec)		θ at 1 sec (deg) (in. stick)	
		Actual	Requirement	Actual	Requirement	Actual	Requirement
<u>Hover, 3000 ft</u>							
With SAS, without STAB*	DS	5.50	None	1.10	None	3.10	2 minimum
	US	5.50		0.90		2.50	
Without SAS, without STAB	DS	6.66	None	1.40	None	3.00	None
	US	7.80		1.40		4.00	
Without SAS, with STAB	DS	5.60	None	0.84	None	2.45	None
	US	5.55		0.94		1.65	
<u>60 kn, 3000 ft</u>							
With SAS, without STAB	DS	3.75	None	0.87	None	1.90	None
	US	4.25		1.00		1.80	
Without SAS, without STAB	DS	4.75	None	1.10	None	1.90	None
	US	5.20		1.50		2.50	
Without SAS, with STAB	DS	5.80	None	1.06	None	2.40	None
	US	5.70		1.25		1.80	
<u>60 kn, 5000 ft</u>							
With SAS, without STAB	DS	3.30	None	1.09	None	1.30	None
	US	4.10		0.95		1.70	
Without SAS, without STAB	DS	5.00	None	1.30	None	1.70	None
	US	6.30		1.28		2.10	
<u>60 kn, 10,000 ft</u>							
With SAS, without STAB	DS	4.20	None	1.07	None	2.00	None
	US	5.70		1.20		2.60	
Without SAS, without STAB	DS	6.50	None	1.20	None	1.90	None
	US	7.60		1.31		2.60	
<u>120 kn, 3000 ft</u>							
With SAS, without STAB	DS	5.40	None	0.55	None	4.30	None
	US	5.30		0.63		5.50	
Without SAS, without STAB	DS	8.10	None	0.87	None	4.60	None
	US	8.20		1.00		4.00	

*STAB is an abbreviation for stabilizer bar.

Pitch Cyclic Stick Pulse Input Commands -- Aircraft pitch response to pitch cyclic pulse inputs are given in flight recordings shown as Figures 95 through 101. The evaluation of these flight records was difficult, and not all the desired data was obtained because of the basic pitch characteristics of the helicopter. Because the effect of speed changes on the responses was more pronounced than the effects of the pulse inputs, the data needed to evaluate damping and natural frequency characteristics was lost in the overall response. Table 13 shows the natural frequency and damping ratio values that were obtained. Some of these are approximations because of the previously mentioned problems. The SAS provides an increase in the damping ratio of the aircraft, thereby improving handling characteristics, particularly at the hover condition.

Engage/Disengage Transients

Part of the system evaluation consisted of making specific pitch, roll, and yaw SAS engage and disengage operations in three different flight configurations at three airspeed conditions. The flight configurations were:

- Straight and level
- Climbing right turn
- Climbing left turn

The flight conditions were:

- 60 kn at 3000 ft
- 90 kn at 3000 ft
- 120 kn at 3000 ft

The flight recordings shown in Figures 102, 103, and 104 were made at the three flight conditions. Recorded parameters are yaw rate, pedal position, pitch rate, pitch stick position, pitch attitude, roll attitude, roll rate, and roll stick position. From these traces it was not immediately evident where the engage and disengage events occurred. The 24-channel recording showed traces of the servo position signals that indicated the engage and disengage events. Correlation of the two recordings was made, and the engage and disengage times have been indicated on the recordings of Figures 102, 103, and 104. The yaw rate recordings show no significant or consistent changes related to the engage or disengage operation at any of the flight conditions. The pitch rate trace shows higher rate excursions than yaw during the test runs, but there appears to be no correlation between the engage and disengage functions and the rate variations.

Table 13. Pitch SAS - Pulse Inputs

Flight Condition		Natural Frequency (Rad/Sec)		Damping Ratio			
		Actual	Requirement	Actual	Requirement		
Hover, 3000 ft							
With SAS, without STAB*	UP	-	None	-	None		
	DN	1.9		0.40			
Without SAS, without STAB	UP	-		0.20			
	DN	2.1		-			
Without SAS, with STAB	UP	-		-			
	DN	-		-			
60 kn, 3000 ft							
With SAS, without STAB	UP	-		0.60		0.5	
	DN	2.1		0.40			
Without SAS, without STAB	UP	1.7		0.20			
	DN	2.1		0.15			
Without SAS, with STAB	UP	1.5		0.50			
	DN	-	0.70				
60 kn, 5000 ft							
With SAS, without STAB	UP	-	0.80	0.70			
	DN	-	0.80				
Without SAS, without STAB	UP	1.0	0.70				
	DN	1.1	0.70				
60 kn, 10,000 ft							
With SAS, without STAB	UP	-	0.80		0.50		
	DN	-	0.80				
Without SAS, without STAB	UP	1.3	0.50				
	DN	1.2	0.50				
120 kn, 3000 ft							
With SAS, without STAB	UP	-	0.60			0.70	
	DN	-	0.70				
Without SAS, without STAB	UP	1.6	0.30				
	DN	1.5	0.70				

*STAB is an abbreviation for stabilizer bar.

The roll rate trace shows a right roll rate disturbance after every engage operation and a left roll rate disturbance after every disengage operation, possibly introduced by a null offset in the roll control channel. Rate magnitude varied from 1.4 to 4 degrees per second at engage, and from 1.2 to 4.5 degrees per second at disengage.

Pilot reaction to these disturbances is evident by comparing the roll stick motion traces with the rate traces. The amount of pilot motion was not greater than the range of stick excursions to correct for normal aircraft deviations. No adverse reactions or comments were made by the pilot about the roll disturbances.

The lateral and normal acceleration traces on the 24-channel recordings show no deviations that can be associated with the engage and disengage operations.

Autorotation Tests

Autorotation entries were performed at 60, 90, and 120 knots, 3000 feet altitude, with SAS engaged and with SAS off. No specific maneuvers were conducted during autorotation. The pilot was primarily holding airspeed and maintaining directional stability. An autorotation was performed by cutting the throttle to simulate engine failures and then dropping collective control at the appropriate time to maintain rotor rpm. The flight recordings for these tests are presented in Figures 105, 106, and 107. The most noticeable difference between the autorotation entry with SAS off and with SAS on is in the amount of pedal displacement required to hold the aircraft heading. With the SAS engaged, the pedal displacement was reduced.

OUTER-LOOP MODE PERFORMANCE

Roll Attitude Hold

The roll attitude hold was evaluated by applying 10-degree roll attitude step commands at the flight conditions listed in Table 7. Aircraft time responses showing the vehicle-recorded parameters for roll attitude step commands of 10 degrees are given in Figures 108 through 111. Performance parameters obtained from the flight recordings are given in Table 14.

The T_{90} response times obtained in flight compare favorably with the T_{90} response times obtained during the system analysis. Table 15 shows performance parameters obtained during the analysis. In-flight response times were between 2.6 and 3.0 seconds, and analysis response times were between 2.2 and 2.8 seconds.

Table 14. Roll Attitude Hold Response to 10-Degree Step Command During Flight Test

Flight Condition		Response Time T ₉₀ (sec)	Overshoot (pct)	Damping	Peak Roll Rate (deg/sec)
Airspeed (kn)	Altitude (ft)				
60	3,000	2.6	12	0.20	4.0
60	5,000	2.6	24	0.20	5.0
60	10,000	2.8	16	0.45	4.5
120	3,000	3.0	50	0.25	5.5

Table 15. Roll Attitude Hold Response from Analysis (10-Degree Step Command)

Airspeed (kn)	Response Time T ₉₀ (sec)	Overshoot (pct)
60	2.8	3.0
80	2.4	6.0
100	2.2	12.0

The percentage of overshoot obtained in the flight evaluation of the system was greater than that obtained by analysis, ranging from 12 to 50 percent in flight compared with 3 to 12 percent by analysis. The 120-knot flight condition, which had the 50 percent overshoot, was not analyzed; therefore, no direct comparison can be made of that flight condition.

Oscillations induced by the step response damped out in two or three cycles, indicating a damping ratio in the range of 0.2 to 0.45, which was lower than indicated by the analysis. The greater percentage of overshoot and lower damping results determined from the flight test resulted from increasing the roll attitude loop gain 50 percent to improve the heading hold performance. This is discussed in greater detail in the section of the report on problems encountered. The overshoot and damping presented no problems, and the pilots commented favorably on the loop response.

The last column of Table 14 shows the peak roll rate achieved during the step responses. Rates of 4 to 5.5 degrees per second are not excessive and are lower than might be expected. During the system analysis phase, however, emphasis on small servo displacement was given a high priority, resulting in lower rates.

Pitch Attitude Hold

The pitch attitude hold was evaluated by applying 2-degree pitch attitude step commands at the flight conditions previously listed under roll attitude hold. Aircraft time responses showing the vehicle-recorded parameters for these commands are given in Figures 112 through 115. Performance parameters obtained from the flight recordings are given in Table 16.

Table 16. Aircraft Pitch Attitude Hold Responses to 2-Degree Step Commands During Flight Test

Flight Condition		Time to Peak Attitude (sec)	Peak Pitch Rate / 2-deg Attitude (deg/sec/deg)
Airspeed (kn)	Altitude (ft)		
60	3,000	3.40	1.2
60	5,000	3.25	1.1
60	10,000	3.40	1.0
120	3,000	2.70	1.2

The flight test recordings of step responses when compared with the analysis recordings indicate that the commanded attitude step is achieved only momentarily. The attitude then droops back toward the value existing before the step was applied. This type of response occurs because the airspeed changes when the aircraft attitude is changed. Analysis recordings made with controlled airspeed show that commanded attitude changes can be maintained when the airspeed is held constant.

The measured response time to the peak attitude values obtained from the flight recordings produced values from 2.7 to 3.4 seconds as shown in Table 16. Time to reach peak pitch attitude, as obtained from the analysis recording, provides values from 3.5 to 5.0 seconds as indicated in Table 17.

Table 17. Aircraft Pitch Attitude Hold Response From Analysis

Airspeed (kn)	Time to Peak Attitude (sec)	Peak Pitch Rate / 2-deg Attitude (deg/sec/deg)
60	3.5	1.0
80	4.0	0.8
100	5.0	0.7

The response times from the analysis data show an increase in time to achieve peak attitude as airspeed is increased. The flight recordings indicate that a faster response is obtained at the higher airspeed. An exact comparison cannot be made of the flight recordings and the analysis results, as the same conditions were not evaluated in the flight spectrum as were analyzed, but the general trend can be noted.

The peak pitch rates obtained during the flight test step responses varied between 1.0 and 1.2 seconds compared to 0.7 to 1.0 second measured from the analysis recordings.

Heading Select and Hold

The heading hold mode provides heading control by using heading error signals to command a bank angle. The gain is fixed with a bank to heading ratio of 0.4 to 1. The bank angle command is limited to approximately 15 degrees. A heading select capability is also provided that allows the pilot to select a new heading and then to engage the heading select switch. This applies the heading error as a step command filtered by a 0.5-second first-order lag.

The heading select performance was evaluated using step heading commands of 10 degrees and 45 degrees. Aircraft responses for 10- and 45-degree heading steps are shown in Figures 116 through 119. Performance parameters obtained from the flight recordings are given in Table 18.

The response time to the 10-degree change in heading varied between 8.5 and 14.2 seconds, with definite time response differences between left and right commands. There was little or no overshoot, and the damping ratio is approximately 1.0. The analysis recordings show a 12.5-percent overshoot at 60 knots for a 10-degree heading command. The peak roll rate experienced in the command maneuvers varied from 1.2 to 3.0 degrees per second.

Table 18. Aircraft Response to Heading Step Commands

Flight Condition		Command (deg)	Response Time T ₉₀ (sec)		Overshoot (pct)	Damping	Peak Roll Rate (deg/sec)
Airspeed (kn)	Altitude (ft)		Right	Left			
60	3,000	10	12.0	10.4	0	1.0	1.2
60	5,000	10	12.4	10.4	0	1.0	1.5
60	10,000	10	11.0	11.5	5	0.8	2.0
120	3,000	10	14.2	8.5	0	1.0	3.0
60	3,000	45	13.8	12.0	10	0.6	6.0
60	5,000	45	17.3	17.4	0	1.0	4.0
60	10,000	45	14.0	14.5	5	0.8	5.0
120	3,000	45	17.0	15.8	0	1.0	6.5

With the 0.4 to 1 bank-to-heading ratio, a 10-degree heading error does not command a bank angle large enough to reach the bank limit of 15 degrees. However, when a 45-degree heading error is applied, the bank angle commanded would be 18 degrees if it were not limited. Limiting the bank angle also slows down the total response time.

The aircraft heading response for 45-degree step heading changes is also shown in Figures 116 through 119. The responses are not symmetrical between left and right step changes partly because of a difference between the left and right roll attitude limits and partly because of aircraft response differences. The limits were mechanized by limiting the E/F valve motion, and it was difficult to get symmetrical operation with this type of limiting action.

The T₉₀ response times for the 45-degree step changes varied from 14.0 to 18.5 seconds. The percentage of overshoot exceeded 9 percent, only for the right step response at the 60-knot, 3000-foot condition. Damping ratio was determined to be in the range of 0.7 to 1.0. Because of the larger command, the peak rate values were approximately 2.5 times those achieved with the 10-degree bank step commands, ranging in magnitude from 2.8 to 6.5 degrees per second.

Effects of Bank Angle Limit Change

The effect of changing the bank angle limits can be shown by comparing two sets of response data taken at 60 knots, 3000 feet for a 45-degree step heading change. Table 19 shows the response data.

Table 19. Aircraft Response to Heading Step Commands with Varying Bank Angle Limit

Bank Limit (deg)	Response Time (sec)		Overshoot (pct)	Damping Ratio	Peak Roll Rate (deg/sec)
	Right	Left			
15	15.0	18.5	9	0.7	2.8
20	13.8	12.0	10	0.6	6.0

The main differences are that response time is reduced and the peak roll rate is increased when a larger bank limit is used.

Altitude Hold

The altitude hold mode provides automatic control of a reference barometric altitude through control of the pitch axis of the aircraft. It is designed for use at speeds above 50 knots and for operation with the mode engaged from a trimmed pitch attitude hold condition. Pitch mistrims cause an altitude error, the ratio being about 4 feet of altitude per degree.

Altitude hold performance was evaluated using step altitude commands of 50 feet. Aircraft time responses for 50-foot altitude steps are shown in Figures 120 through 123. Performance parameters obtained from the flight recordings are given in Table 20.

The altitude responses obtained in flight were similar to those obtained in the analysis of the system. The initial response to the step command is rapid, and then there is a gradual approach to the new reference altitude. The T_{90} response times varied from 6.0 to 9.0 seconds.

Overshoots did not exceed 20 percent, and damping was in the range of 0.5 to 0.7 except for the 120-knot, 3000-foot condition, which was somewhat oscillatory. Damping at this condition is approximately 0.2. Peak pitch rate values measured during the altitude response maneuvers were between 2.4 and 4.0 degrees per second.

Table 20. Aircraft Response to 50-Foot Altitude Step Commands

Flight Condition		Response Time T ₉₀ (sec)	Overshoot (pct)	Damping	Peak Pitch Rate (deg/sec)
Airspeed (kn)	Altitude (ft)				
60	3,000	9.0	0	0.7	2.5
60	5,000	6.0	20	0.5	3.2
60	10,000	8.5	20	0.7	3.0
120	3,000	6.6	20 down 0 up	0.2	2.4 down 4.0 up

Airspeed Hold

The airspeed hold mode provides automatic control of a reference dynamic pressure through control of the pitch axis of the aircraft. It is designed for use at speeds above 50 knots. The mode is interlocked with the altitude hold mode such that both altitude and airspeed hold cannot be engaged at the same time. The airspeed hold mode is designed to be engaged from a trimmed pitch attitude hold condition. Pitch mistrims cause an airspeed error.

Airspeed hold performance was evaluated using step airspeed commands of 5 to 15 knots. Aircraft time responses for the airspeed step commands are shown in Figures 124 through 127. Performance parameters obtained from the flight recordings are given in Table 21. The recorded airspeed on the flight recordings was obtained from an independent airspeed sensor and not from the flight control hardware.

Airspeed responses obtained in flight were quite similar to those obtained during analysis of the system. In the analysis study, airspeed step changes of 10 feet/second (5.9 knots) were introduced at speeds from 60 to 100 knots. The response time (T₉₀) obtained was 7.0 seconds with no overshoot.

The T₉₀ response times shown in Table 21 (obtained from the flight recordings) show time variations from 6.6 to 14.2 seconds at the 60-knot conditions, and from 4.0 to 6.6 seconds at the 120-knot conditions. The variations in response time are due in part to the noise on the traces, the difference in step magnitudes, and the variations in the responses caused by aerodynamic disturbances. There was no significant overshoot

Table 21. Aircraft Response to Airspeed Step Commands

Flight Condition		Step Command (kn)	T90 Response Time (sec)	Peak Attitude Change (deg)	Peak Pitch Rate (deg/sec)
Airspeed (kn)	Altitude (ft)				
60	3,000	Increase airspeed 14, 14	8.8, 6.6	11.0, 11.5	6.0, 6.0
60	3,000	Decrease airspeed 15, 15	9.7, 7.8	6.0, 7.0	5.0, 3.5
60	5,000	Increase airspeed 11.0, 10	9.1, 14.2	8.5, 6.5	4.7, 3.0
60	5,000	Decrease airspeed 7, 15	8.3, 12.6	3.0, 4.0	2.5, 2.0
60	10,000	Increase airspeed 5, 7	9.1, 10.3	4.0, 5.0	3.0, 3.0
60	10,000	Decrease airspeed 7, 7	8.6, 8.3	5.0, 4.7	2.0, 4.0
120	3,000	Increase airspeed 5, 5	4.3, 5.7	6.0, 6.0	4.5, 4.8
120	3,000	Decrease airspeed 4, 6	6.6, 4.0	7.0, 5.0	3.5, 3.2

of the commanded airspeed. Noise on the recorded airspeed traces tends to mask minor parameter changes. The peak pitch rate, which occurred during the airspeed step transient in the analysis study, was 2.5 degrees per second at 60 knots and 2.0 degrees per second at 100 knots. The peak pitch rates from the flight recordings, as shown in Table 21, varied from 2.0 to 6.0 degrees per second.

The transient pitch attitude deviations obtained from the flight recordings varied from 3.0 to 11.5 degrees from the steady-state attitude. Maximum pitch attitude occurred approximately 2 to 3 seconds after initiation of the step. In general, the airspeed hold mode performed very much like that shown in the analysis study.

Figure 128 shows aircraft responses to airspeed hold during a 15-degree bank, 180-degree turn with turn entries of 15 degrees per second at 60 knots, and of 10 degrees per second at 120 knots. In each case, a minor airspeed transient of about 3 knots was introduced, and a slower roll rate was used when returning to level flight. While this did not diminish the magnitude of the airspeed transient, it was not abrupt and was therefore probably not noticeable to the pilot.

Pilot Override Capability

Data were taken at the hover condition at 3000 feet to record the effect on the aircraft of outer-loop input steps and to demonstrate the capability of the pilot to override the pilot-assist functions. Aircraft responses at hover are given in Figures 129 through 132.

Because the outer loops were not designed to perform at hover, the pilot had to control the aircraft during hover and during the step inputs. The flight recordings indicate that at the hover condition, the aircraft response to a given step command results in motions in more than one axis, and the attitude changes resulting from the inputs, in some cases, larger than those obtained with forward velocity. The pilot was able to bring the vehicle under control in all cases by overriding the outer-loop controls.

Steady-State Accuracy

Outer-loop hold mode accuracies were measured for steady-state flight conditions. Deviations were measured over time intervals of 15 minutes to determine the range of parameter variations. Because of air turbulence encountered in flying, very short-period disturbances that produced some peak deviations were averaged out to the nominal control level. A summary of the data derived from the flight recordings is given in Table 22. Some flight conditions were evaluated twice under different conditions of turbulence, as indicated by two sets of numbers in some blocks of the table.

Table 22. Steady-State Accuracy Data Summary

Flight Condition	Roll Attitude Hold (deg)	Pitch Attitude Hold (deg)	Heading Hold (deg)	Altitude Hold (ft)	Airspeed Hold (kn)
60 kn, 3,000 ft	±3.0	±1.8	±4.0	±40.0	±3.2
	±1.0	±0.8	±1.8	±18.0	
60 kn, 5,000 ft	±2.0	±2.0	±2.5	±22.5	±2.5
60 kn, 10,000 ft	±1.0	±1.0	±1.0	±10.0	±3.2
120 kn, 3,000 ft	±3.0	±2.2	±2.5	±40.0	±4.0
			±1.7	±30.0	
Design Goal	±1.0	±1.0	±1.0	±30.0	±3.0

Roll and pitch attitude hold accuracies were mainly in the ± 1.0 - to ± 2.0 -degree attitude range. When conditions were more turbulent, roll attitude showed a greater variation than did pitch attitude. Heading hold accuracies were mostly in the range of ± 1.0 to ± 2.5 degrees with deviations up to ± 4.0 degrees in turbulence.

Altitude hold accuracies were mostly in the range of ± 10 to ± 30 feet from reference. Part of the altitude deviations at the 60-knot, 5000-foot condition and the 120-knot, 3000-foot conditions were produced by a phugoid oscillation. The phugoid amplitude was approximately 25 feet peak to peak with a time period of about 3.3 minutes at the 60-knot, 5000-foot condition (shown in Figure 133). At the 120-knot, 3000-foot condition the phugoid amplitude was 30 feet peak to peak with a time period of approximately 2.5 minutes. At the other flight conditions no definite phugoid period was indicated on the flight recordings. In some places it appeared that a phugoid oscillation was starting, but turbulence effects interrupted the control response to prevent a cycle from being completed.

Airspeed hold accuracies were in the range of ± 2.5 to ± 3.2 knots, with deviations going as high as ± 4.0 knots in more turbulent conditions. As a comparison of achieved performance, the design goals for control accuracy as determined from the analysis study are summarized in Table 22.

PILOT EVALUATIONS

The system was evaluated by a Honeywell test pilot and by two U. S. Army test pilots. Their reports are given below. It should be noted that the U. S. Army pilot evaluations were made prior to the incorporation of a system modification to add the airspeed hold mode and the cyclic stick trim switch feature.

Mr. Donald Sotanski (Honeywell)

The system flight test established that the design goal of pilot relief was attained. The system provides accurate, smooth response to pilot-commanded inputs. Long-term, hands-off flight capability is provided and hovering performance is improved. It is evident that the pilot's work load is dramatically decreased, even though specific tests were not performed to establish this feature.

The cyclic stick forces of the test aircraft were not significantly altered by installation of the system. The forces were symmetrical and without discontinuities in all axes. Only a very slight pull to the left (approximately 0.25 pound) could be detected during cruise flight; it was less detectable at lower speeds and hovering flight. The pull was not objectionable, but should be eliminated in any production installation. Also, no change in pedal forces could be detected as a result of system installation.

An outstanding feature of the system is the absence of engage and disengage transients. The SAS was engaged and disengaged during maneuvering flight without any objectionable "bumps." Initially, there were minor transients upon system engagement, but these transients were eliminated by adjusting the system null. Throughout the program, the engage/disengage characteristics remained unchanged, with no objectionable transients during maneuvering flight, hovering, or autorotations. The engage/disengage characteristics were checked over a wide range of ambient temperatures (approximately -10°F to 90°F).

The dynamic stability of the pitch axis of the UH-1M aircraft is excellent even without a stabilizer bar. Pitch-axis response is also good and presents no control problems. The addition of a stabilizer bar or SAS provides a perceptible increase in damping with insignificant control power loss.

The SAS definitely improves the spiral stability of the roll axis. The task of rapidly rolling up to a given bank angle and maintaining this angle is greatly simplified by the SAS. No ratcheting tendency is noted during execution of this maneuver. A loss of control power is readily sensed during SAS operation, but the roll response is still well within desirable limits for all tactical and operational flight maneuvers. The damping afforded by the SAS is superior to that provided by the stabilizer bar: the SAS considerably reduces the pilot work load associated with establishing and maintaining a given bank angle, especially above 30 degrees. The yaw axis of the UH-1M aircraft lacks dynamic stability over the entire flight regime. The SAS provides near deadbeat damping at cruise speeds and considerably improves handling in the yaw axis at hover and at low speed. The pilot-induced yaw/roll motion is dramatically reduced with SAS operation, especially in a hover. With SAS operation, no loss of control power is readily evident. The yaw response is considerably reduced, and at high speeds may be considered inadequate for some tactical maneuvers. However, for the majority of tactical flying and overall operational flying, the yaw response provided by the SAS is near ideal.

The roll-attitude hold mode provides excellent, steady-state accuracy. The aircraft roll attitude is maintained at the trimmed attitude to within approximately ± 1.0 degree or better in nonturbulent air and at the steady-state power setting. Response to turn control knob inputs is satisfactory. Smooth, proportional response is attained without excessive overshoot. Bank-angle limits are symmetrical. In turbulent air, the excursions from the desired attitude are somewhat greater than ± 3 degrees; however good average-commanded attitude is maintained.

The system provides excellent pitch-attitude hold under steady-state conditions. The selected pitch attitude is maintained within approximately ± 1.0 degree in nonturbulent air and at the steady-state power setting.

The desired pitch attitude is easily attained by mode engagement and by the pitch-trim function. The response to trim commands is slightly slow, but adequate over the range of 60 knots to 140 knots. In turbulent air, the pitch attitude varies approximately ± 3 degrees; however, good average-commanded attitude is maintained.

The heading-hold mode provides the selected heading within approximately ± 1.0 degree in steady-state flight and in nonturbulent air. Frequently, a steady-state offset from the selected heading is evident and appears to be a direct function of roll- and yaw-axis mistrim. Other than the offset problem, the heading-hold mode is completely satisfactory. The response to selected heading changes is smooth with bank angle proportional to the commanded heading change. The maximum bank angle of 15 degrees is comfortable and provides a good turning rate.

The altitude-hold mode provides adequate altitude-hold performance above 60 knots; a slow drift-down is evident with bank angles above 30 degrees. The short-term altitude-hold performance is certainly within acceptable limits, but a slow drift from reference altitude may be experienced over a 30-minute period. The response to altitude errors is somewhat soft, especially at the lower airspeed range around 60 knots. The mode provides altitude hold within ± 20 feet in smooth air with steady-state power.

The airspeed-hold mode provides barely adequate performance. The response to error is slow, with performance deteriorating rapidly at the lower airspeeds. Overshoots of increasing magnitude are experienced in response to airspeed errors, especially in the lower speed range. Repeated pitch changes to establish the selected airspeed ultimately dissipate aircraft energy and result in a complete loss of function. At the higher airspeeds, approximately 80 to 140 knots, performance is much improved, but altitude excursions to maintain airspeed are excessive.

The trim function provided by the hot switch on the cyclic is excellent. The pilot may trim to the desired attitude with a minimum of effort by observing the horizon or trim indicators. A slight lag in response is evident, but this can be readily compensated for with experience. The trim rate and sensitivity could be reduced to accommodate the critical attitude accuracy required to provide desired performance.

The system does not present a safety-of-flight hazard. The system can easily be overpowered to provide desired aircraft response. Response to a system "hardover" signal can be compensated for with ease and without structural limitations being exceeded.

The major disadvantage of the system as presently mechanized is its response to collective pitch changes. Assuming a perfectly trimmed aircraft at system engagement, a collective pitch change while in the heading-hold mode will cause a yaw-axis mistrim that will be manifested in the

roll axis of the system. Given a moderate yaw mistrim, the roll axis will compensate to its limit with a resulting large heading offset and ultimately loss of heading-hold control.

The system provides pilot relief with adequate accuracy, provided the aircraft is reasonably trimmed at engagement. The steady-state heading and altitude-hold factors offer the best pilot relief functions and are accurate for long periods of flight. The system, as presently mechanized, provides excellent pilot relief for reconnaissance missions and all VFR operational flying. It is not considered adequate for IFR because of the unsatisfactory performance in response to collective pitch changes.

System utility can be vastly improved by providing an automatic yaw-trim function. All phases of the aircraft mission could be greatly enhanced by a system with yaw-trim capability, especially IFR capability.

Mr. Duane R. Simon (Eustis Directorate, USAAMRDL)

The aircraft was flown at the Honeywell Flight Facility in Minneapolis the week of 3 December 1973. Two flights were made in the UH-1M: one in very gusty conditions, and the other in smooth, stable air. The UH-1M was modified somewhat by the removal of the gyro bar and the addition of series-type actuators in the pitch, roll, and yaw control axes. The three actuators provided angular rate damping from purely fluidic signals generated by three fluidic vortex rate sensors. Two of these same series actuators (pitch and roll) were also used to provide the outer-loop/autopilot functions of heading, pitch, roll, and altitude hold. Signals for heading hold were obtained electrically from the ship's directional gyro, converted to fluidic signals, appropriately massaged, and applied to the lateral cyclic series actuator. Likewise, pitch- and roll-attitude hold signals were obtained from the ship's vertical gyro and were applied fluidically to the respective cyclic series actuators. The altitude-hold system was entirely fluidic using barometric data to drive the longitudinal cyclic series actuator. The collective control system was unmodified.

In testing the rate damping feature as provided by the SAS, it was found that the engaging and disengaging characteristics were satisfactory. Hard-over testing was not possible. Yaw appeared to be optimized in terms of how hard the system was working in hovering flight as evidenced by the aircraft ride qualities. Because some jerkiness was noticed in hovering turns, any further increase in the yaw gain would undoubtedly become objectionable. Pitch-axis SAS gain appeared acceptable, although somewhat low. The gain in roll seemed to provide considerably less damping in hovering flight than that provided by the gyro bar on the basic UH-1M. A cursory check of the degradation of roll response caused by the current level of roll damping revealed a noticeable but insignificant loss. It would

appear that an increase in damping could be tolerated before it would become objectionable or until a washout scheme would be necessary. However, the contractor indicated that, based upon previous work with the UH-1C, any increased cyclic damping becomes objectionable at higher forward airspeeds. Notwithstanding, it would seem prudent to investigate the possibilities of either providing more roll damping or varying the levels of damping as a function of airspeed. Naturally, it would be desirable to do this fluidically.

The hybrid fluidic-electrical autopilot was designed to provide automatic control of the helicopter when flying at a constant altitude from point A to point B. The design also provided the pilot the capability of selecting different magnetic headings to which the aircraft would automatically turn to, roll out, and maintain. The system performed this task fairly well as long as the collective pitch was maintained at or very near a fixed setting. The system exhibited some directional overshoot and steady-state heading error in acquiring new commanded headings, particularly following large heading changes. However, once properly trimmed, it provided a true enroute level-flight "hands-off" capability. Trimming was actually a cumbersome two-phased task. Prior to engaging each outer-loop channel, it was necessary to null the respective signals using knurled trim wheels located on the center console. Immediately following engagement, it was then necessary to retrim for coordinated wings-level flight. Lateral trimming for a wings-level attitude usually introduced a sideslip that in turn called for disengaging heading hold, manually moving the pedals to center the ball, then reengaging the heading-hold function. The trimming characteristics were unsatisfactory and should be improved prior to demonstrating the system to anyone outside the R&D community.

As a consequence of air-traffic control procedures with Minneapolis approach control, the system was subjected to localizer-type approaches at the conclusion of each test flight. While the philosophy of providing an austere, limited pilot assist is appreciated, the terminal area operations demonstrated that the limited capability may render the system unacceptable from an operational, cost-effective standpoint. The major deficiency was the inability to effectively change altitude or enter climbing/descending flight without first disengaging the autopilot, then manually establishing the new flight condition, nulling the signals, reengaging the hold functions, and finally retrimming. Another problem associated with changing flight conditions was the system's inability to hold heading with changes in power. Because the heading-hold function relied on the inherent directional stability of the helicopter, any power change caused a side load at the tail that introduced an unacceptable heading error. As an example, a heading change of 25 degrees occurred when the power was reduced to establish a 500-fpm rate of descent at 90 KIAS. It was possible to "trim out" these errors using the console directional trim wheel, which suggests a possible fix using some sort of compensating network that would sense sideslip angle or power changes. The heading error resulting from power

changes was unsatisfactory and should be corrected before future demonstrations. The consequence of not being able to correct this deficiency reflects upon the basic design premise of controlling heading (outer loop) with lateral cyclic pitch. It may mean that heading hold control inputs must be applied through the tail rotor.

The altitude hold function worked very well in maintaining selected cruising altitudes and in capturing the MDA or DH on typical instrument approaches. It was interesting to note that the heading-hold mechanization (altitude control using longitudinal cyclic) gave the pilot the capability of controlling airspeed with only the collective pitch stick. This characteristic is viewed as only a spin-off of the design scheme and has no real useful application; in fact, it is backward to the normal helicopter control logic.

A suggested modification to the autopilot, which should expand its usefulness in terminal area-type operations, is airspeed hold either in addition to or in lieu of altitude hold. Honeywell felt that a fluidic hookup to the Pitot tube was not only feasible but relatively simple and inexpensive. If the system could be made to hold heading during power changes, then the autopilot with airspeed hold should be compatible with full-instrument approaches including missed-approach procedures. It is recommended that the airspeed-hold function be added to the system and flight tested.

In summary, the SAS/autopilot shows promise of providing the much needed automated pilot assistance in helicopters like the UH-1; however, it is felt that further exploratory work is necessary to "put the technology on the shelf." The mechanization of the outer-loop functions is relatively unique for helicopters in that the pilot can enter and exit the control loop at will without disengaging the system. When the pilot enters the cyclic control loop, the system automatically gives him attitude control as compared to rate control for the basic helicopter. Attitude control is a definite advantage during instrument flight, and conceivably should be an enhancement in the hovering flight regime; however, no effect was noted. The pitch- and roll-attitude hold functions were also ineffective in hover. The contractor contended that the system was reacting properly to the gyro signals and that different gains and shaping were required for hovering flight. It would seem worthwhile to proceed with additional work to determine whether or not series-type, outer-loop functions can be tailored to the hovering flight regime and, if so, then to determine how well it works.

Major Donald A. Couvillion (Eustis Directorate, USAAMRDL)

The Honeywell three-axis fluidic stability augmentation system (FSAS) with autopilot was test flown and evaluated at the contractor's site during 4 through 6 December 1973. Two flights, each approximately one hour in

duration, were conducted in a modified UH-1 under two different meteorological conditions. The first flight was accomplished during a period of strong gusty winds and turbulence. During this flight, the initial hovering was conducted with the FSAS disengaged to determine the handling characteristics of an aircraft from which the gyro stabilizer bar had been removed. In this condition, the aircraft was unstable, particularly in the roll and yaw axes. Engagement of the axes, one at a time, was accomplished, and the aircraft then became much more stable. Instability in the roll axis was eliminated and the aircraft handled normally; however, it was not possible to discern any particular advantage of the FSAS over the gyro stabilizer bar at this point.

A normal takeoff was accomplished with the FSAS engaged, and no unusual characteristics were noted. Upon reaching cruise altitude, the outer loops of the autopilot were engaged and the altitude hold, lateral control, and pitch control enabled "hands-off" flight. All loops seemed to function satisfactorily with the exception of the yaw control, which consistently gave an out-of-trim condition, with the turn and slip indicator indicating a need for left pedal. The indicator showed a constant half-ball width to the left. Pilot-induced attitude changes were corrected by the autopilot. In yaw, however, the system seemed to overcorrect and had to go back and forth seeking the correct heading. Turns were made to the left and to the right using the heading "bug." It was noted that during turns to the left, that the bank angle was approximately 15 degrees while those to the right were at approximately 25 degrees. At this time, it was also noted that the aircraft did not turn exactly to heading indicated by the "bug." The aircraft always returned to a position approximately 5 to 10 degrees to the right of the indicator. Pitch and roll controls were basically satisfactory unless a change was made in collective pitch. Any change in pitch immediately resulted in a change in roll attitude. Upon beginning the descent, the collective pitch was lowered and the aircraft immediately began a roll to the right. This tendency was easily overridden manually but was somewhat disconcerting at the time.

The aircraft was then hover tested with the aircraft being positioned in various headings relative to the strong prevailing winds. Even under the most adverse condition (quartering tail wind), the aircraft was controllable and relatively stable. Disengagement of each channel resulted in a very unstable condition in all axes.

Upon completion of the hover tests, another normal takeoff was conducted and an ILS approach was started using the FSAS/autopilot. Interception of the localizer was accomplished without incident by use of the lateral control of the autopilot. Upon initiation of the descent, however, the tendency of the aircraft to roll was again noted, and it was necessary to disengage the force trim and to reposition the cyclic control. This resulted in the aircraft being out of trim, and the descent to middle marker was conducted in approximately a 25-degree crab angle from the true course. Upon reaching minimums, the autopilot loops were disengaged and the landing was accomplished normally using the FSAS.

The second flight was conducted under calm wind condition but under much colder OAT. The only differences noted during this second flight were that the system seemed to be slightly more sluggish at the start and slow to correct pilot-induced oscillation. After five minutes or less, however the system operated as on the previous flight. The only other problem was a sharp lateral transient noted upon engagement of that channel. Three consecutive engagements of the lateral hold resulted in the same sharp transient. A different pilot technique was used during the ILS approach and landing for the second flight. Upon interception of the LOM, the altitude hold was disengaged and the trim wheel was used to initiate the descent. The collective was lowered to maintain the approach air-speed, and the roll trim was used to correct for the roll induced by the collective. This approach worked very well and the aircraft was completely stable and under autopilot control during the entire descent. Upon reaching minimums, the altitude hold was again engaged and the roll trim used to correct for the increase in collective pitch. No adverse tendencies were noted.

The three-axis autopilot/FSAS on the UH-1 results in a definite improvement in the handling qualities of the aircraft. The discrepancies mentioned are more annoying than serious and seem to be easily correctable.

The strongest single recommendation that can be made is that the tendency of the aircraft to change attitude drastically with any change of collective pitch be corrected. Additionally, it is felt that the controls are far too easy to override; a more positive control of the stick and pedal forces would be beneficial. As the system now operates, there feels to be very little more control than is exercised by the application of the force trim.

PROBLEMS ENCOUNTERED

During the flight test, several system problems were encountered. These problems are discussed in the following paragraphs.

Roll Attitude Gain/Heading Hold Gain

The initial system had a roll attitude loop gain of 0.015 inch/degree and a heading hold loop gain of 0.009 inch/degree. This resulted in a smooth roll attitude response with minimal overshoot, but the heading hold response resulted in a lightly damped oscillation. This oscillation was more pronounced at lower airspeeds (≈ 60 knots) and decreased as the aircraft's airspeed increased. The problem resulted from the heading hold gain being too high in relation to the roll attitude gain. The initial heading hold/roll attitude gain ratio was $0.009/0.015 = 0.6$. This ratio was reduced to 0.4 by reducing the heading gain to 0.006 inch/degree. This eliminated the oscillation, but the low loop gain resulted in large deviations around the heading set point. The heading gain was reset at

0.009 inch/degree and roll attitude gain was increased to 0.023 inch/degree, thus maintaining the 0.4 heading hold/roll attitude gain ratio. At this gain setting, the heading hold functioned well. Because of the increased roll attitude gain, its response was quicker with a 12 to 24 percent overshoot at 60 knots. The pilots liked the faster response, and the overshoot was not felt to be a problem. Other gain values were tried, but the heading hold/roll attitude gain ratio of 0.4 with the heading hold gain of 0.009 inch/degree and the roll attitude gain of 0.023 inch/degree was felt to be optimum.

Yaw Axis Oscillation

In the middle of the flight test, yaw axis oscillation started to occur upon SAS engagement. Sometimes the oscillations would not occur; other times, the yaw SAS had to be disengaged. The problem became more pronounced as the flight test proceeded. Tests indicated that the yaw SAS controller was functioning properly. The servoactuator was removed and replaced with the spare unit, which solved the problem, and the yaw SAS functioned satisfactorily for the rest of the flight test. Due to time and facility constraints, the servoactuator was not tested to determine the cause of failure.

Flow Control Valve

During one flight, the initial engagement of the SAS's resulted in random motion from the series servoactuators, which was severe enough to disengage the SAS and to abort the test. Ground tests revealed that the variable flow control valve was not controlling, and as the fluid temperature increased, the system gain and noise got too high. The flow control valve was removed, disassembled, cleaned, reassembled, and reinstalled. However, system performance was still unsatisfactory, as some loops had low gain and other loops were noisy. Ground tests revealed that flow change per temperature change of the flow control valve was four times higher than the original design, resulting in proper operation in only a very narrow temperature band. The flow control valve was again removed and disassembled, and a damaged O-ring was found and replaced. This restored the flow control valve to its proper performance.

Synchronizer EMI

The synchronizer circuits for pitch and roll attitude were tested prior to installation in the helicopter; both functioned satisfactorily. During check-out of the system modifications in the helicopter while operating from a ground power cart, the synchronization function was normal; however when the system was operated from the aircraft dc generator while the engine

was running, an engage offset was experienced both in roll and pitch attitude loops. The magnitude and direction of the offset varied from one engagement to another and sometimes drove the servoactuator hard-over.

After a review of the power and grounding circuitry and after monitoring the engage circuit transients on an oscilloscope, some ground wiring changes were made along with the addition of radio noise filters. There was about a 10 percent improvement in the attitude engagement offset. The source of the interference was considered to be the aircraft dc generator, since it was the only different piece of operating equipment between running from ground power and running from aircraft power. Limited time prevented undertaking an elaborate EMI program; therefore, to resume flight testing, the pitch and roll synchronizers were bypassed, and manual synchronization was used to null the system before engaging the attitude hold modes.

Heading Hold E/F Valve

During one flight, the heading hold loop failed. It would not hold or maintain a set heading, and the only roll axis control was from the roll attitude loop. From ground tests, the problem was traced to the electrical-to-fluidic (E/F) interface valve. The valve was removed and cleaned, restoring the valve's performance for the remainder of the flight test.

Aircraft Vertical Gyro

On two occasions, the aircraft vertical gyro malfunctioned during flight. Both times the gyros exhibited erratic behavior, indicated by the attitude indicator. This prevented the pitch and roll attitude loops from being engaged and tested. On both occasions the vertical gyro problem was verified during ground tests and the gyro was replaced.

Aircraft Gyrocompass

Another heading hold problem occurred during a test flight that resulted in erratic behavior of the heading hold loop. The heading indicator showed that the gyro compass/indicator was not following the heading, but was just drifting. Ground tests indicated that the gyrocompass had malfunctioned, and it was replaced.

Collective Power Changes

A collective power change made while the heading hold mode is engaged changes the tail rotor torque, causing the aircraft to rotate about the yaw axis, producing a heading error. The system tries to correct for the heading error by rolling the aircraft, but due to the yaw axis offset, it cannot adequately correct the heading error. The net result is that the aircraft flies at a mistrimmed condition with a sideslip angle and an offset roll attitude angle, which can be corrected only by disengaging the system, retrimming the aircraft, and reengaging the system.

This power change problem was investigated during pilot evaluations and was felt to be detrimental to system performance. The system was designed as a steady-state, cross-country pilot-relief autopilot that relied on the aircraft roll axis to provide the heading hold function. This mechanization did not provide automatic corrections for power changes. To correct this problem, heading control would have to be provided by the tail rotor (yaw axis) or some type of signal applied to the tail rotor to compensate for the torque change. This type of system modification was beyond the scope and capability of this program for two reasons:

- Extensive redesign and modification of the system would be required.
- Yaw axis servoactuator did not have sufficient authority to compensate for the torque change. At 80 knots airspeed, yaw axis compensation for a collective change, which produces a 500-fpm descent rate, required 0.9 inch of pedal motion. This corresponds to 0.48 inch of series servoactuator travel, which exceeds the maximum stroke available (± 0.38 inch).

SECTION VIII CONCLUSIONS

The conclusions drawn from this program are grouped into two categories: those related to system design and mechanization, and those related to system performance as determined from the flight test evaluation.

SYSTEM DESIGN AND MECHANIZATION

- Fluidics can be applied to more advanced automatic flight control system designs; however, completely fluidic systems are presently not feasible due to the unavailability of adequate fluidic attitude references.
- Hybrid fluidic/electronic systems are presently feasible, and they offer significant advantages in the areas of reliability and cost.
- The fluidic, electronic, and interface components for hybrid flight controls are presently available.
- The trade-off between fluidic versus electronic mechanization of the individual components and the location of the electronic-to-fluidic interfaces will depend on the particular system design and performance requirements.
- The advanced stabilization system tested provided gain stability of approximately ± 20 percent over an oil temperature range of 80°F to 150°F . This was found to be equivalent to an ambient temperature range of approximately 20°F to 90°F for the UH-1M helicopter. In general, the gain of the outer-loop hold modes varied less than the SAS gains.
- No problems were encountered in the integration and operation of the fluidic system with the UH-1M hydraulic power supply.
- The following new components and features, which will be applicable to future fluidic or hybrid systems, were successfully demonstrated:
 - Fluidic altitude error sensor
 - Fluidic dynamic pressure deviation sensor

- Temperature-scheduled flow control valve
 - Smaller size, reduced flow (approximately 50 percent) fluidic amplifiers and vortex rate sensors
 - Less complex electronic synchronizer circuit
 - Remote engage/disengage capability within fluidic circuits
 - Interfacing of fluidic and electronic control modes within a single system.
- The following features of the advanced stabilization system have potential for significant cost savings for future helicopter flight control systems:
 - Use of aircraft instrument gyros for flight control attitude references
 - Use of limited-authority series servoactuators for outer-loop hold modes as well as SAS modes

SYSTEM PERFORMANCE

- The system provides the intended pilot assistance for the typical observation and transportation mission for which it was designed, including the capability for extended hands-off flight during cross-country flying.
- In general, system performance coincided with the design goals established in the earlier system analysis. Steady-state performance (accuracy) was dependent on the amount of air turbulence encountered during a particular flight and the accuracy with which the pilot trims the aircraft prior to engaging the hold modes. For this reason, the system did not meet the design goals during some flight tests, but general performance was considered to be acceptable.
- The SAS mode stability augmentation of the UH-1M helicopter was comparable to or better than that provided by the mechanical stabilizer bar. The SAS increased yaw axis damping ratio from approximately 0.3 to 0.65.
- The limited-authority, series servoactuators provided satisfactory performance for the flight conditions tested.

- **Operation of any of the system modes outside of their design flight envelope did not present a flight safety problem. Pilot override capability was available at all conditions.**
- **The system hold modes will not compensate for the effects of changes in collective pitch on the UH-1 helicopter, therefore resulting in unsatisfactory performance of these hold modes for maneuvers involving collective pitch changes.**
- **The pilots objected to the manual engage, disengage, and trimming requirements under certain conditions where changes in flight path are being made (i. e. , approaches). It is believed that the synchronization functions added to the roll and pitch controls would greatly reduce this problem. Unfortunately, due to EMI problems, it was not possible to reevaluate the system with this added feature.**
- **Additional system design studies are needed to:**
 - **Develop a method to compensate for the effects of changes in collective pitch.**
 - **Determine whether series servoactuator outer-loop hold modes can be tailored to the helicopter hovering flight regime.**
- **Component development is needed to:**
 - **Reduce the electromagnetic interference (EMI) susceptibility of the new synchronizer circuit.**
 - **Obtain lower-cost E/F and F/E transducers with performance compatible with hybrid flight control system interface requirements.**
 - **Reduce the temperature sensitivity of fluidic components and circuits with emphasis on optimizing the temperature-scheduled flow control concept and on improved fluidic amplifier designs.**

APPENDIX A
PERFORMANCE, DESIGN, AND QUALIFICATION
REQUIREMENTS FOR AN ADVANCED
HYDROFLUIDIC STABILIZATION SYSTEM*

1.0 SCOPE

This specification defines the design requirements for the Advanced Hydrofluidic Stabilization System, hereafter referred to as the "system." The objective of this system is to provide a reliable, low-cost automatic flight control system incorporating pilot relief modes.

2.0 APPLICABLE DOCUMENTS

The following documents and the applicable specifications referenced therein shall apply to the extent specified herein:

- a. MIL-H-8501A, Helicopter Flying and Ground Handling Qualities, General Requirements for.
- b. MIL-H-5606, Hydraulic Fluid, Petroleum Base, Aircraft, Missile and Ordnance.
- c. MIL-STD-810B, Military Standard Environmental Test Methods for Aerospace and Ground Equipment.

3.0 REQUIREMENTS

3.1 General

3.1.1 Modes of Operation

The system shall be capable of controlling the helicopter in the following modes:

- a. Stability augmentation (three axes)
- b. Attitude hold (pitch and roll)
- c. Heading hold and select
- d. Altitude hold
- e. Airspeed hold

***Note:** This specification is an updated version of the preliminary specification presented in the Design Study report (USAAMRDL Technical Report 72-46). It reflects the system changes made during the hardware program and the performance obtained.

3.1.2 Service Flight Envelope

The flight envelope for operation of the various system modes shall be as defined below.

a. Airspeed:

- Stability augmentation -- Hover to maximum cruise
- Attitude hold -- 50 knots to maximum cruise
- Heading, altitude, and airspeed hold -- 50 knots to maximum cruise

b. Altitude: All modes -- 0 to 6000 feet

3.1.3 Functional Components

The system shall consist of the following functional units:

- a. Attitude Sensor -- The system shall have two attitude sensors, one for sensing aircraft yaw attitude (heading) and a second for sensing aircraft pitch and roll attitude. These sensors will also be used for display of aircraft attitude.
- b. Rate Sensor -- Each axis shall have a vortex rate sensor which provides a signal that is proportional to the aircraft angular rate in the specific axis.
- c. Altitude Error Sensor -- The system shall have an altitude sensor which provides a signal proportional to the difference between aircraft actual altitude and a particular reference altitude (altitude existing at time of engaging).
- d. Dynamic Pressure Sensor -- A dynamic pressure sensor will be used for the airspeed hold mode. The sensor shall provide a signal proportional to the difference between aircraft actual airspeed (dynamic pressure) and a particular reference airspeed (airspeed existing at time of engaging).
- e. Rudder Input Device -- This device provides a signal which is a function of rudder pedal displacement. This signal reduces the tendency of the rate damper to "fight" pilot inputs in the yaw axis.

- f. Electronic Circuit -- An electrical circuit shall be provided to process the outputs of the panel display attitude sensors, so as to transform them into a form compatible with the electrical-to-fluidic transducers. It shall also include the electrical circuits for the synchronization function, electronic power supply, and specific engage/disengage switches.
- g. Electric-to-Fluidic Transducer -- E/F transducers shall be used to transform the electrical attitude signals to fluidic signals.
- h. Fluidic-to-Electric Transducer -- F/E transducers shall be used to transform fluidic attitude signals to electrical signals as part of the synchronization function.
- i. Amplifier Circuits -- Fluidic amplifier circuits shall be used to sum, amplify, and limit differential pressure signals in the system.
- j. Shaping Networks -- A combination of resistors (orifices) and capacitors (bellows) shall be used to provide the following functions:
 1. Lag -- With a characteristic of $1/(TS + 1)$
 2. High-pass -- With a characteristic of $TS/(TS + 1)$
 3. Lag-lead -- With a characteristic of $(TS + 1)/(NTS + 1)$ where T is time constant and N is greater than 1.0.
- k. Servoactuator -- The servoactuator, mounted in series with the aircraft power boost servoactuators, accepts differential pressure signals and converts them to displacements of the power boost servoactuator pilot valve.
- l. Trim Indicator -- Trim indicators shall be provided in the system outer loops to monitor signal nulls prior to engagement of these modes.
- m. Engage Valve -- Solenoid-operated hydraulic valves will be remotely controlled from the cockpit to engage all or part of the system.
- n. Flow Control Valve -- This device shall maintain a constant flow to the fluidic system with changes in hydraulic system pressure. It shall also include a feature that schedules the flow set point, as a function of fluid temperature, providing increased flow at low oil temperature and reduced flow at high oil temperature to compensate for temperature/viscosity effects on the fluidic circuits.

- o. Back Pressure Regulator -- This device shall isolate servoactuator-induced return pressure surges from the fluidic system.
- p. Control Panel -- This component provides the controls needed to engage and operate the system modes. It contains a turn control knob; switches to engage or disengage the SAS, pitch attitude hold, roll attitude hold, heading hold, altitude hold, and airspeed hold modes; pitch and roll trim knobs; and pitch and roll trim indicators.

3.2 Environment

The system shall perform satisfactorily when exposed to the following environmental conditions:

- a. Vibration -- MIL-STD-810B, Figure 514-1, Curve B.
- b. Temperature -- The system shall be able to start at -25°F ambient temperature with the fluid at -25°F , and then operate satisfactorily with an ambient temperature of -25°F with the fluid temperature at $+40^{\circ}\text{F}$, up to an ambient temperature of $+100^{\circ}\text{F}$ with the fluid temperature at $+185^{\circ}\text{F}$.
- c. Supply Flow and Pressure Variations -- The system shall be able to withstand the supply flow and pressure variations determined from previous programs to be normally occurring in the selected aircraft hydraulic supply system.

3.3 Power Supplies

Input power to the system shall be hydraulic fluid per Specification MIL-H-5606 at a pressure of 1500 psig (nominal), which is obtained from the aircraft No. 2 hydraulic power system. The system (except augmentation servoactuators) shall not require more than 1.5 gpm.

Electrical power for relays and solenoids will be 28 Vdc. Power for the electrical interface circuit will be 115 V, 400 Hz.

3.4 Detailed Performance Requirements

The performance requirements for the system are defined in Table 23. These requirements are applicable to the UH-1C helicopter operating within the flight envelope as specified in 3.1.2.

Table 23. Performance Requirements

System Mode or Function	Characteristic	Requirements
Pitch Stability Augmentation	Dynamic stability	<p>The helicopter shall exhibit satisfactory dynamic stability characteristics following longitudinal disturbances in forward flight. Specifically, the stability characteristics shall be unacceptable if the following are not met for a single disturbance in smooth air:</p> <ul style="list-style-type: none"> a) Any oscillation having a period of less than 5 seconds shall damp to one-half amplitude in not more than 2 cycles, and there shall be no tendency for undamped small-amplitude oscillations to persist. b) Any oscillation having a period greater than 5 seconds but less than 10 seconds shall be at least lightly damped. c) Any oscillation having a period greater than 10 seconds but less than 20 seconds shall not achieve double amplitude in less than 10 seconds.
	Control power	<p>Longitudinal control power shall be such that when the helicopter is hovering in still air at the maximum overload gross weight or at the rated power, a rapid 1.0-inch step displacement from trim of the longitudinal control shall produce an angular displacement at the end of 1.0 second, which is at least 2.0 degrees.</p>
	Damping	<p>The pitch axis damping ratio of the UH-1 helicopter shall be increased from approximately 0.3 to approximately 0.5 or greater at or near the 100-knot flight condition. This value may be demonstrated by a simulated vertical gust input and measuring the aircraft performance in damping the gust.</p> <p>The vertical gust input shall be damped to within 20 percent of its maximum value within 1.5 seconds following the gust.</p> <p>At the hover flight condition, the time to damp the gust may be significantly longer to account for the free vehicle damping characteristics.</p>
Roll Stability Augmentation	Response	<p>The response of the helicopter to lateral control deflection, as indicated by the maximum rate of roll per inch of sudden control deflection from the trim setting, shall not be so high as to cause a tendency for the pilot to overcontrol unintentionally. In any case, at all levels of flight speeds, including hovering, the control effectiveness shall be considered excessive if the maximum rate of roll per inch of stick displacement is greater than 20 degrees per second.</p>
	Control power	<p>Lateral control power shall be such that when the helicopter is hovering in still air at the maximum overload gross weight or at the rated power, a rapid 1-inch step displacement from trim of the lateral control shall produce an angular displacement at the end of one-half second (0.5) of at least 1.2 degrees.</p>
Yaw Stability Augmentation	Response	<p>The response of the helicopter to directional control deflection, as indicated by the maximum rate of yaw per inch of sudden pedal displacement from trim while hovering, shall not be so high as to cause a tendency for the pilot to overcontrol unintentionally. In any case, the sensitivity shall be considered excessive if the yaw displacement is greater than 50 degrees in the first second following a sudden displacement of 1.0 inch from trim while hovering at the lightest normal service loading.</p>
	Control power	<p>Directional control power shall be such that when the helicopter is hovering in still air at the maximum overload gross weight or at rated takeoff power, a rapid 1.0-inch step displacement from the trim of the directional control shall produce a yaw displacement at the end of 1.0 second which is at least 5.0 degrees.</p>
	Damping	<p>The directional axis damping ratio of the UH-1 helicopter shall be increased from approximately 0.3 to approximately 0.6 or greater at the high-speed flight conditions. This value may be demonstrated by a simulated lateral gust input and measuring the aircraft performance in damping the gust.</p>

Table 23. Performance Requirements (Concluded)

System Mode or Function	Characteristic	Requirements
Roll Attitude Hold	Accuracy (steady state)	The referenced roll attitude shall be maintained within ± 1.0 degree.
	Response	Response to an attitude command shall be smooth and rapid with less than 20 percent overshoot. Response time shall be less than 3 seconds.
	Solution time	Solution time shall be less than 5 seconds.
	Command maneuver limit	± 30 degrees (using flight controller).
Heading Hold and Select	Accuracy (steady state)	The reference heading shall be maintained to within ± 1.0 degree.
	Response	Response to a heading error shall be smooth and rapid with one overshoot which shall not exceed 20 percent or 5 degrees, whichever is smaller. Response time shall be between 5 and 20 seconds.
	Heading select range	Existing heading ± 180 degrees.
	Maximum bank angle limit	± 15 degrees.
Pitch Attitude Hold	Accuracy (steady state)	The referenced pitch attitude shall be maintained within ± 1.0 degree.
	Response	Response to an attitude command shall be smooth and rapid with less than 20 percent overshoot. Response time shall be less than 4 seconds.
	Command maneuver limit	No provisions for maneuvering in pitch are provided in the flight controller.
Altitude Hold	Accuracy (steady state), straight and level and in turns within roll limits of ± 15 degrees	Referenced altitude shall be maintained within ± 30 feet. (± 50 feet for ± 30 degree roll attitude.)
	Response	Response to an altitude step shall be smooth with less than 20 percent overshoot.
Airspeed Hold	Accuracy (steady state) straight and level, and in turns within roll limits of ± 15 degrees	Referenced airspeed shall be maintained within ± 3 knots.
	Response	Response to an airspeed step shall be smooth with less than 20 percent overshoot.
Turn Coordination	Steady-state sideslip during turns within roll limits of ± 15 degrees	Steady-state sideslip angle shall be less than 2.0 degrees.
System Authority	Series servoactuator displacement limits	Series servoactuator displacement limits shall be limited to the following values expressed in percent of total surface limits. <ul style="list-style-type: none"> ● Pitch - 20 percent ● Roll - 25 percent ● Yaw - 20 percent
Engagement/Disengagement	Switching transients	Engagement/disengagement of any of the system modes at steady-state conditions shall not result in helicopter transients in excess of 0.05g.

3.5 System Open-Loop Performance

All performance requirements in this section pertain to normal operating conditions. Normal operating conditions are defined as: ambient temperature - $70^{\circ}\text{F} \pm 10^{\circ}\text{F}$; hydraulic fluid temperature - $120^{\circ}\text{F} \pm 10^{\circ}\text{F}$; hydraulic fluid pressure - 1000 to 1500 psig ahead of flow regulator, with a maximum of 20 psig return pressure.

3.5.1 Lateral-Directional Axes

The block diagrams for this part of the system are given as Figures 67 and 68. The performance requirements for the individual modes are summarized below.

3.5.1.1 Yaw Axis SAS -- Yaw axis SAS requirements are summarized in the block diagram of Figure 67. Additional performance requirements are listed below.

- a. **Transfer Functions:** The nominal transfer functions for the yaw rate feedback loop (yaw rate to series servoactuator displacement) and rudder pedal position input loop (pedal position to series servoactuator displacement) shall be the form shown below.

$$\frac{\text{Servoactuator Displacement}}{\text{Yaw Rate}} = 0.023 e^{-0.02S} \left(\frac{2.5S}{2.5S + 1} \right) \left[\frac{1200}{(0.03S + 1)(S^2 + 22S + 1200)} \right] \frac{\text{in.}}{\text{deg/sec}} \quad (1)$$

$$\frac{\text{Servoactuator Displacement}}{\text{Pedal Displacement}} = 0.84 \left(\frac{1}{S + 1} \right) \left(\frac{2.5S}{2.5S + 1} \right) \left[\frac{1200}{(0.03S + 1)(S^2 + 22S + 1200)} \right] \frac{\text{in.}}{\text{in.}} \quad (2)$$

- b. **Range:** The range shall be at least ± 40 deg/sec ahead of the high pass and ± 100 percent actuator stroke downstream of the high pass.
- c. **Noise:** Peak-to-peak noise at the actuator shall not exceed an equivalent of ± 0.5 deg/sec yaw rate at maximum dynamic gain.
- d. **Accuracy:** Gain and time constants shall be maintained within ± 20 percent of the nominal requirements.

3.5.1.2 Roll Axis SAS -- Roll axis SAS requirements are summarized in the block diagram of Figure 68. Additional performance requirements are listed below.

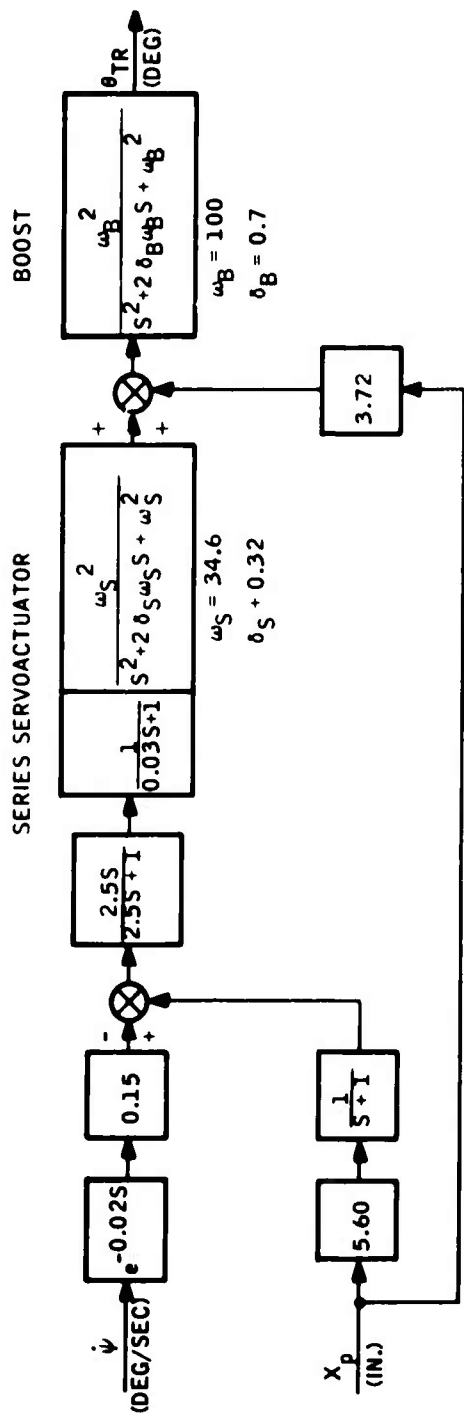


Figure 67. Yaw Axis Requirements

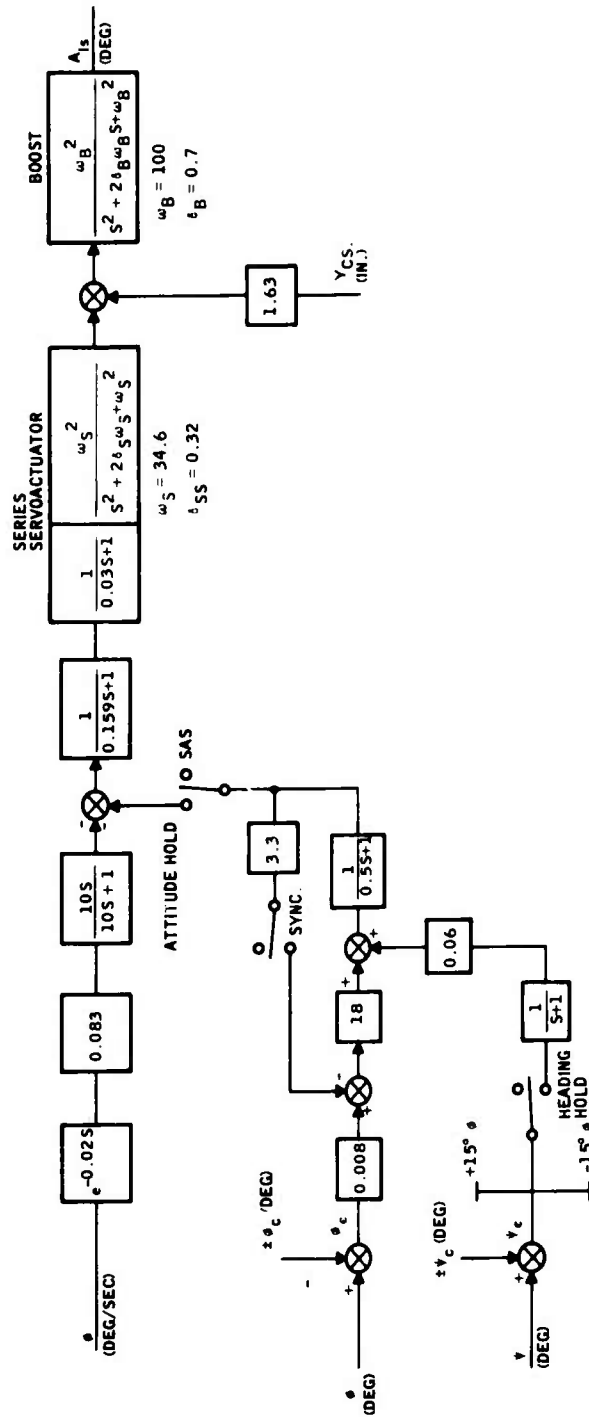


Figure 68. Roll Axis Requirements

- a. **Transfer Function:** The nominal transfer function for the roll rate feedback loop (roll rate to series servo-actuator displacement) shall be of the form shown below.

$$\frac{\text{Servoactuator Displacement}}{\text{Roll Rate}} = 0.012 e^{-0.02S} \left(\frac{10S}{10S + 1} \right) \left(\frac{1}{0.16S + 1} \right) \left[\frac{1200}{(0.03S + 1)(S^2 + 22S + 1200)} \right] \frac{\text{in.}}{\text{deg/sec}} \quad (3)$$

- b. **Range:** The range shall be at least ± 40 deg/sec ahead of the high pass and ± 100 percent actuator stroke downstream of the high pass.
- c. **Noise:** Peak-to-peak noise at the actuator shall not exceed an equivalent of ± 0.5 deg/sec roll rate at maximum dynamic gain.
- d. **Accuracy:** Gain and time constants shall be maintained within ± 20 percent of the nominal requirements.

3.5.1.3 Roll Attitude Hold -- Roll attitude hold requirements are summarized in the block diagram of Figure 68. Additional performance requirements are listed below.

- a. **Transfer Function:** The nominal transfer function for the roll attitude feedback loop (roll attitude to series servo-actuator displacement) shall be the form shown below.

$$\frac{\text{Servoactuator Displacement}}{\text{Roll Attitude}} = 0.023 \left(\frac{1}{0.5S + 1} \right) \left(\frac{1}{0.16S + 1} \right) \left[\frac{1200}{(0.03S + 1)(S^2 + 22S + 1200)} \right] \frac{\text{in.}}{\text{deg}} \quad (4)$$

- b. **Range:** The range shall be at least ± 100 percent actuator stroke (25 degrees equivalent roll attitude) through the roll attitude loop.
- c. **Noise:** Peak-to-peak noise at the actuator shall not exceed an equivalent of ± 0.5 degree roll attitude.
- d. **Accuracy:** Gain and time constants shall be maintained within ± 20 percent of the nominal requirements.

3.5.1.4 Heading Hold -- Heading hold requirements are summarized in the block diagram of Figure 68. Additional performance requirements are listed below.

- a. **Transfer Function:** The nominal transfer function for the heading loop (heading error to series servoactuator displacement) shall be the form shown as follows.

$$\frac{\text{Servoactuator Displacement}}{\text{Heading Error}} = 0.009 \left(\frac{1}{s+1} \right) \left(\frac{1}{0.55s+1} \right) \left(\frac{1}{0.16s+1} \right) \left[\frac{1200}{(0.03s+1)(s^2+22s+1200)} \right] \frac{\text{in.}}{\text{deg}} \quad (5)$$

- b. **Range:** The range of the heading loop shall be limited to an equivalent of ± 15 degrees roll attitude ahead of the point where the heading and roll attitude signals are summed.
- c. **Noise:** Peak-to-peak noise at the actuator shall not exceed an equivalent of ± 1.0 degree heading error.
- d. **Accuracy:** Gain and time constants shall be maintained within ± 20 percent of the nominal requirements.

3.5.2 Longitudinal-Vertical Axes

Figure 69 is a block diagram for this part of the system. The performance requirements for the individual modes are summarized below.

3.5.2.1 Pitch Axis SAS -- Pitch axis SAS requirements are summarized in the block diagram of Figure 69. Additional performance requirements are listed below.

- a. **Transfer Function:** The nominal transfer function for the pitch rate feedback loop (pitch rate to series servo-actuator displacement) shall be the form shown below.

$$\frac{\text{Servoactuator Displacement}}{\text{Pitch Rate}} = 0.026 e^{-0.02s} \left(\frac{2.5s}{2.55s+1} \right) \left(\frac{0.04s+1}{0.15s+1} \right) \left[\frac{1200}{(0.03s+1)(s^2+22s+1200)} \right] \frac{\text{in.}}{\text{deg/sec}} \quad (6)$$

- b. **Range:** The range shall be at least ± 40 deg/sec ahead of the high pass and ± 100 percent actuator stroke downstream of the high pass.
- c. **Noise:** Peak-to-peak noise at the actuator shall not exceed an equivalent of ± 0.5 deg/sec pitch rate at maximum dynamic gain.
- d. **Accuracy:** Gain and time constants shall be maintained within ± 20 percent of the nominal requirements.

3.5.2.2 Pitch Attitude Hold -- Pitch attitude hold requirements are summarized in the block diagram of Figure 69. Additional performance requirements are as follows.

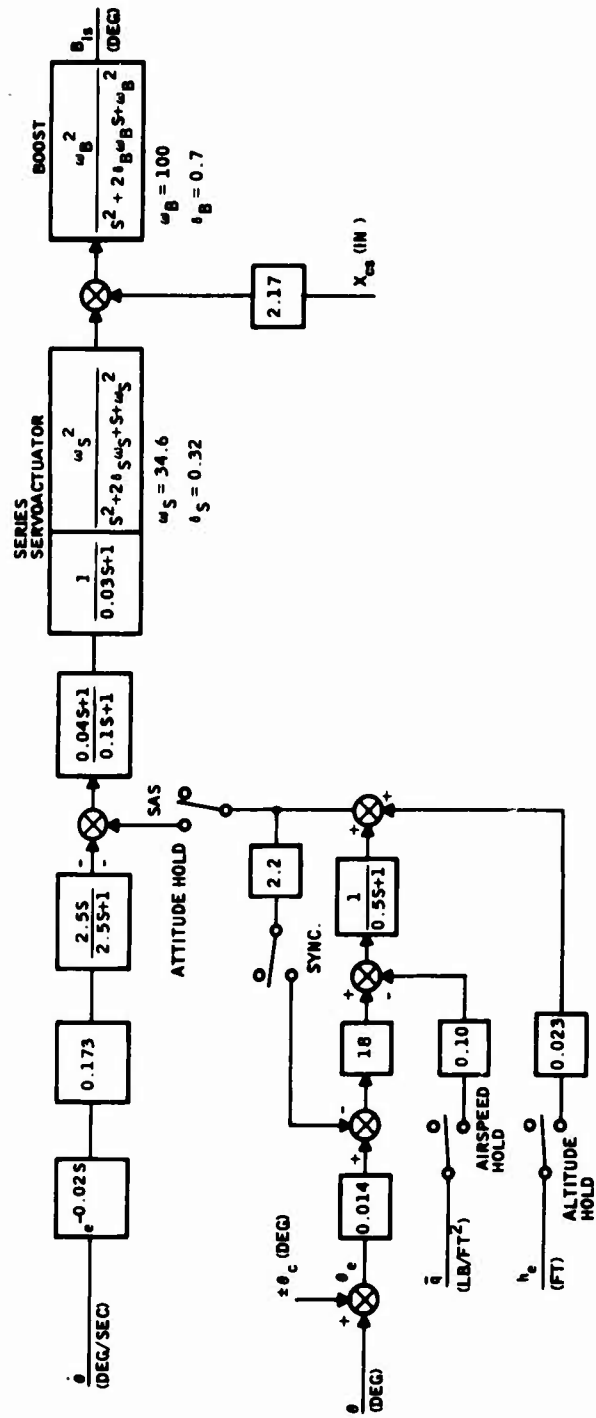


Figure 69. Pitch Axis Requirements

- a. **Transfer Function:** The nominal transfer function for the pitch attitude feedback loop (pitch attitude to series servoactuator displacement) shall be the form shown below.

$$\frac{\text{Servoactuator Displacement}}{\text{Pitch Attitude}} = 0.035 \left(\frac{1}{0.5S + 1} \right) \left(\frac{0.04S + 1}{0.1S + 1} \right) \left(\frac{1200}{(0.03S + 1)(S^2 + 22S + 1200)} \right) \frac{\text{in.}}{\text{deg}} \quad (7)$$

- b. **Range:** The range shall be at least 100-percent actuator stroke (10 degrees equivalent pitch attitude) through the pitch attitude loop.
- c. **Noise:** Peak-to-peak noise at the actuator shall not exceed an equivalent of ± 0.5 degree pitch attitude.
- d. **Accuracy:** Gain and time constants shall be maintained within ± 20 percent of the nominal requirements.

3.5.2.3 Altitude Hold -- Altitude hold requirements are summarized in the block diagram of Figure 69. Additional performance requirements are listed below.

- a. **Transfer Function:** The nominal transfer function for the altitude loop (altitude error to series servoactuator displacement) shall be in the form shown below.

$$\frac{\text{Servoactuator Displacement}}{\text{Altitude Error}} = 0.0035 \left(\frac{0.04S + 1}{0.1S + 1} \right) \left(\frac{1200}{(0.03S + 1)(S^2 + 22S + 1200)} \right) \frac{\text{in.}}{\text{ft}} \quad (8)$$

- b. **Range:** The range shall be at least 100-percent actuator stroke (100 feet equivalent altitude error) through the altitude error loop.
- c. **Noise:** Peak-to-peak noise at the actuator shall not exceed an equivalent of ± 10 feet altitude error.
- d. **Accuracy:** Gain and time constants shall be maintained within ± 20 percent of the nominal requirements.

3.5.2.4 Airspeed Hold -- Airspeed hold requirements are summarized in the block diagram of Figure 69. Additional performance requirements are listed below.

- a. **Transfer Function:** The nominal transfer function for the airspeed loop (dynamic pressure to series servoactuator displacement) shall be in the form shown as follows.

$$\frac{\text{Servoactuator Displacement}}{\text{Dynamic Pressure}} = 2.19 \left(\frac{1}{0.5S + 1} \right) \left(\frac{0.04S + 1}{0.1S + 1} \right) \left[\frac{1200}{(0.03S + 1)(S^2 + 22S + 1200)} \right] \frac{\text{in.}}{\text{psi}}$$

(9)

- b. **Range:** The range shall be at least 100-percent actuator stroke (0.171 psi equivalent dynamic pressure error) through the airspeed loop.
- c. **Noise:** Peak-to-peak noise at the actuator shall not exceed an equivalent of ± 0.0086 psi airspeed error.
- d. **Accuracy:** Gain and time constants shall be maintained within ± 20 percent of the nominal requirements.

3.5.3 Pitch/Roll Synchronization and Stick Trim Requirements.

- a. **Engage Transient:** Engagement of the roll and pitch attitude loops shall not result in a transient greater than 0.050 inch of actuator motion when engaged within a range of ± 30 degrees attitude.
- b. **Range:** Synchronization range shall be ± 30 degrees attitude.
- c. **Reference Drift:** Synchronizer reference drift shall be less than 1 degree per hour.
- d. **Synchronization Time:** Electrical synchronization time when switching from the attitude hold mode shall be less than 10 milliseconds.
- e. **Trim Rate:** The stick trim functions added to the attitude hold mode shall provide an electrical trim rate of 0.4 deg/sec.
- f. **Trim Drift:** The trim drift rate shall be less than 2 deg/hr.
- g. **Trim Authority:** The trim authority shall be approximately ± 8 degrees attitude change.
- h. **Trim Recentering:** When the attitude mode is disengaged, the trim shall recenter in less than 10 seconds.

3.5.4 Control Panel Assembly

The control panel assembly shall provide the controls necessary for system engagement, selection of mode of operation, and introduction

of pilot trim and turn commands. The functions of the controller switches and knobs are shown in Figure 70 and are defined below.

- a. **28 V Power Switch:** Controls 28-Vdc power to the flight controller, logic circuits, and portions of the synchronizer electronics.
- b. **115 V Power Switch:** Controls 115-Vac power to the interface electronics circuitry.
- c. **Master Engage Switch:** Activates the system primary solenoid valve and provides 28-Vdc power to the outer-loop mode switches. For the switch to remain engaged, at least one of three SAS engage switches (pitch, roll, or yaw) must be engaged.
- d. **Emergency Disengage Switches:** Deactivates the primary solenoid valve by opening the master engage switch. An emergency disengage switch shall be located on the pilot's control stick and on the copilot's control stick. (This function is used on the flight test system only.)
- e. **SAS Engage Switches:** Activates the individual axis solenoid valve, thereby engaging the series servoactuator and, therefore, the SAS mode in that axis. Separate switches shall be provided for the pitch, roll, and yaw axes.
- f. **Attitude Hold Switches:** Activates the solenoid valve in the fluidic attitude loop, thereby engaging the attitude hold mode. Separate switches shall be provided for pitch attitude and roll attitude. These switches shall be the latching type, which disengage the mode when the master engage switch is opened.
- g. **Heading Hold Switch:** Activates the electrical heading bridge switch in the interface electronic circuit, thereby engaging the heading hold mode. For the switch to remain engaged, the roll attitude hold switch must be closed and the turn control knob must be in its center detent position.
- h. **Altitude Hold Switch:** Activates the solenoid valve in the altitude error sensor, thereby engaging the altitude hold mode. For the switch to remain engaged, the pitch attitude hold switch must be closed and the airspeed hold mode must not be engaged.

- i. **Airspeed Hold Switch:** Activates the solenoid valve in the airspeed error sensor, thereby engaging the airspeed hold mode. For the switch to remain engaged, the pitch attitude hold switch must be closed and the altitude hold mode must not be engaged.
- j. **Turn Command Knob:** Allows the pilot to establish bank angles up to 15 degrees as a function of displacement from a center detent position. The knob also operates a switch that disengages the heading hold mode when the knob is out of the detent position.
- k. **Trim Knobs and Indicators:** Trim knobs and indicators shall be provided in the roll and pitch attitude loops to provide the pilot with a trim indication and manual trim capability prior to and after engagement of the outer-loop modes.
- l. **Indicator Lights:** Provide an indication of the status of the following switches:
 - Power switches
 - Master engage switch
 - SAS engage switches (each axis)
 - Attitude engage switches
 - Heading hold switch
 - Altitude hold switch
 - Airspeed hold switch

3.5.5 System Interconnection

The system hydraulic interconnection is defined in Figure 70; the system electrical circuit interconnection is defined in Figure 71.

3.6 Performance Under Environmental Conditions

The system shall be compensated to minimize changes in performance with changes in environment over the range of conditions specified in

3.2 The following tolerance values are desired.

- a. System gains: ± 20 percent.
- b. System time constants: ± 20 percent.

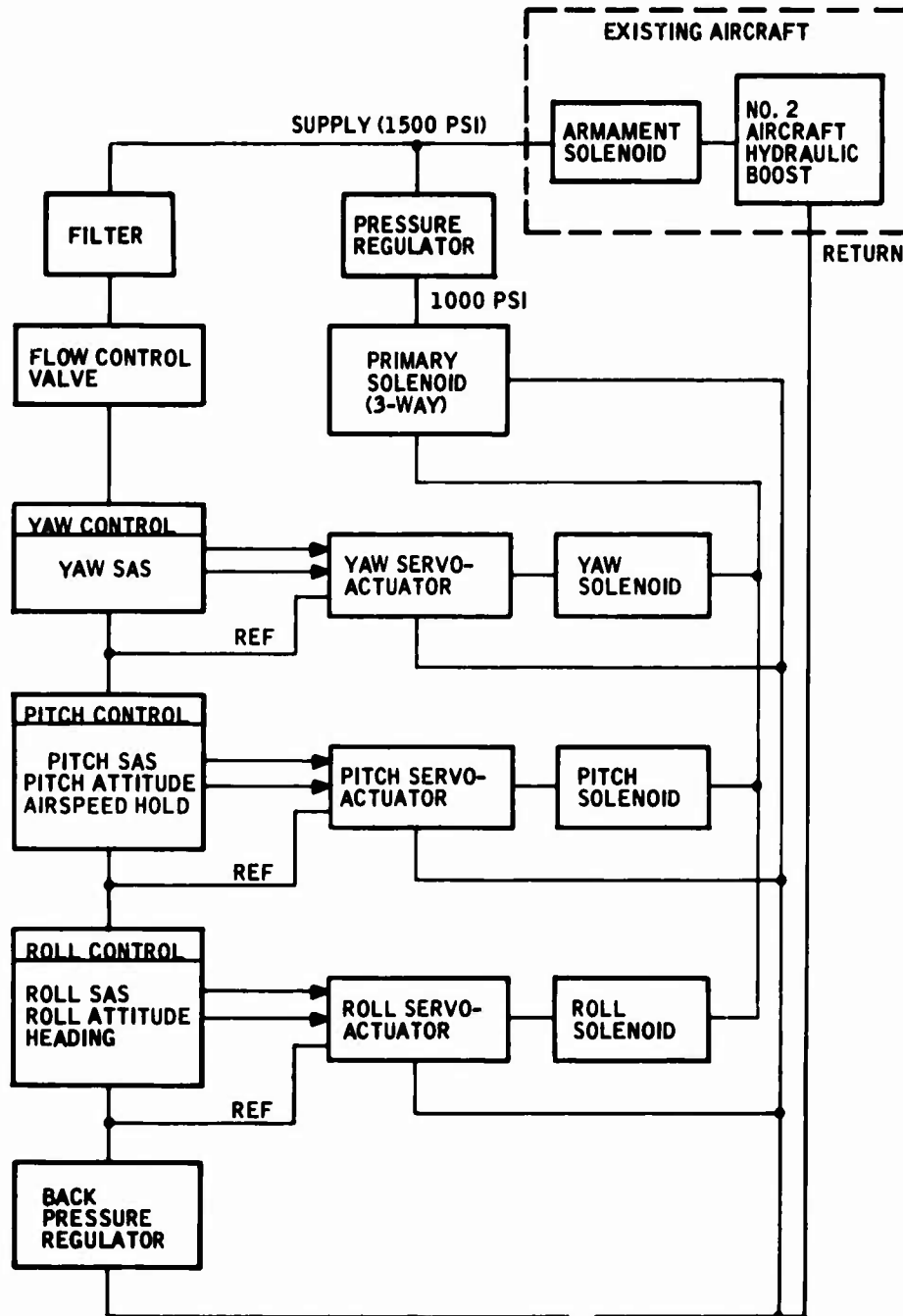


Figure 70. System Hydraulic Interconnection Diagram

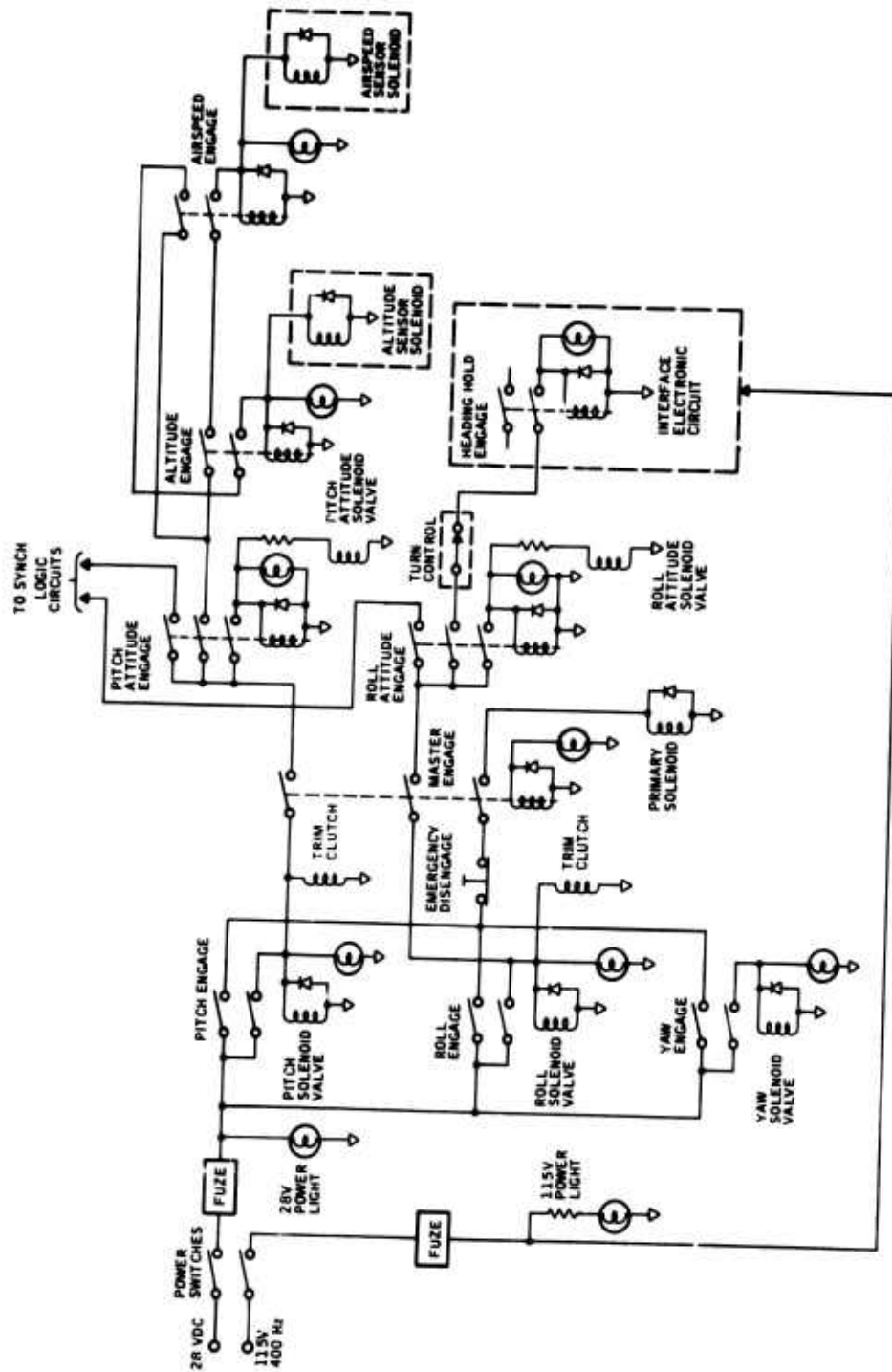


Figure 71. System Electrical Interconnection Diagram

4.0 QUALITY ASSURANCE

Conformance of the hardware to the program objective shall be evaluated with the tests described below. Vibration tests should be completed before performance tests are conducted.

4.1 Vibration

A vibration scan with the system energized and operating shall be conducted at the amplitudes and frequencies of Figure 514-1, Curve B, of MIL-STD-810B. A sinusoidal vibration cycling, per the test envelope, shall be conducted at a rate sufficiently slow to allow adequate identification and evaluation of the resonant frequencies or functional phenomena that may occur. Sinusoidal vibration cycle times shall be not less than 15 minutes per each of the three axes. System vibration testing shall be conducted with the hydraulic supply and connections simulating the actual aircraft installation as nearly as practicable.

4.2 Open-Loop Tests

4.2.1 Yaw Axis SAS

Yaw SAS gain and response shall be determined by measuring channel output versus frequency at two input amplitudes. Response shall be measured with fluid temperatures of 40°F and 185°F. Results are to be compared with the requirements of Paragraphs 3.5.1.1 and 3.6.

4.2.2 Roll Axis SAS

Roll SAS gain and response shall be determined by measuring channel output versus frequency at two input amplitudes. Response shall be measured with fluid temperatures of 40°F, 120°F, and 185°F. Results are to be compared with the requirements of Paragraphs 3.5.1.2 and 3.6.

4.2.3 Roll Attitude Hold

Roll attitude hold gain shall be determined by measuring channel output versus gyro roll angular displacement from vertical. Response shall be determined by measuring output versus frequency for simulated (electrical) input angular displacement signals. Gain and response shall be measured with fluid temperatures of 40°F, 120°F, and 185°F. Results are to be compared with the requirements of Paragraphs 3.5.1.3 and 3.6.

4.2.4 Heading Hold

Heading hold gain shall be determined by measuring channel output versus heading error. Response shall be determined by measuring output versus frequency for simulated (electrical) heading error signals. Gain and response shall be measured with fluid temperatures of 40°F, 120°F, and 185°F. Results are to be compared with the requirements of Paragraphs 3.5.1.2 and 3.6.

4.2.5 Pitch Axis SAS

Pitch SAS gain and response shall be determined by measuring channel output versus frequency at two input amplitudes. Response shall be measured with fluid temperatures of 40°F, 130°F, and 185°F. Results are to be compared with the requirements of Paragraphs 3.5.2.1 and 3.6.

4.2.6 Pitch Attitude Hold

Pitch attitude hold gain shall be determined by measuring channel output versus gyro pitch angular displacement from vertical. Response shall be determined by measuring output versus frequency for simulated (electrical) input angular displacement signals. Gain and response shall be measured with fluid temperatures of 40°F, 120°F, and 185°F. Results are to be compared with the requirements of Paragraphs 3.5.2.2 and 3.6.

4.2.7 Altitude Hold

Altitude hold gain shall be determined by measuring channel output versus pneumatic pressure differential to the altitude sensor. Response shall be determined by measuring output versus frequency for simulated (fluidic) altitude error signals. Gain and response shall be measured with fluid temperatures of 40°F, 120°F, and 185°F. Results are to be compared with the requirements of Paragraphs 3.5.2.3 and 3.6.

4.2.8 Airspeed Hold

Airspeed hold gain shall be determined by measuring channel output versus pneumatic pressure differential to the airspeed sensor. Response shall be determined by measuring output versus frequency for simulated dynamic pressure error signals. Gain and response shall be measured with fluid temperatures of 40°F, 120°F, and 185°F. Results are to be compared with the requirements of Paragraphs 3.5.2.4 and 3.6.

4.2.9 Pitch/Roll Synchronization and Stick Trim

The pitch/roll synchronization and stick trim loop shall be tested at ambient temperature to measure:

- attitude gain
- synchronization drift for 1/2 hour
- synchronization null and range
- stick trim rate and authority
- stick trim drift for 1/2 hour
- trim recenter time

The results shall meet the requirements of Paragraph 3.5.3.

4.2.10 Control Panel Assembly

The control panel assembly function switches, indicator lights, and control knobs shall be tested at room ambient temperature. The results shall meet the requirements of Paragraph 3.5.4.

4.3 Verification

1. Inspect system for quality of workmanship and conformance to installation drawings.
2. Determine that system contains all features described in this specification.
3. Establish that power required does not exceed amount specified in Paragraph 3.3.

APPENDIX B
SYSTEM OPERATING INSTRUCTIONS

1. Power

Turn on "28 Vdc Power" and "115 Vac Power" switches.

2. SAS Engagement

Press desired SAS switch; "Pitch SAS", "Roll SAS", and/or "Yaw SAS".

Press "Master Engage" switch for SAS engagement.

3. Pitch or Roll Attitude Hold Engagement*

Engage corresponding SAS.

Trim aircraft.

Center needle in trim indicator by turning proper trim knob (on console) or by use of stick trim switch.

Engage desired attitude hold by pressing "Pitch Att." and/or "Roll Att." switches.

Trim changes can be made using stick trim switch.

4. Altitude or Airspeed Hold Engagement*

Trim aircraft at desired altitude/airspeed.

Engage pitch SAS and pitch attitude modes.

Press "Alt. Hold" switch for altitude hold engagement or "Airspd Hold" switch for airspeed hold engagement (mode engage switch logic prevents these two modes from being operated simultaneously).

5. Heading Hold Engagement*

Trim aircraft at desired heading.

Engage roll SAS and roll attitude modes.

Press "Heading Hold" switch for heading hold engagement.

Heading changes can be made automatically by moving heading bug on heading indicator to desired heading.

6. Turn Control

A constant rate of turn can be introduced using turn control knob.

* Engage at forward speeds of 50 knots or greater

Heading hold mode will automatically disengage when turn control is activated.

7. System Disengage

Complete system may be disengaged at any time by pressing "Master Engage" on panel or disengage switch on stick.

Appendix C - Flight Test Recordings

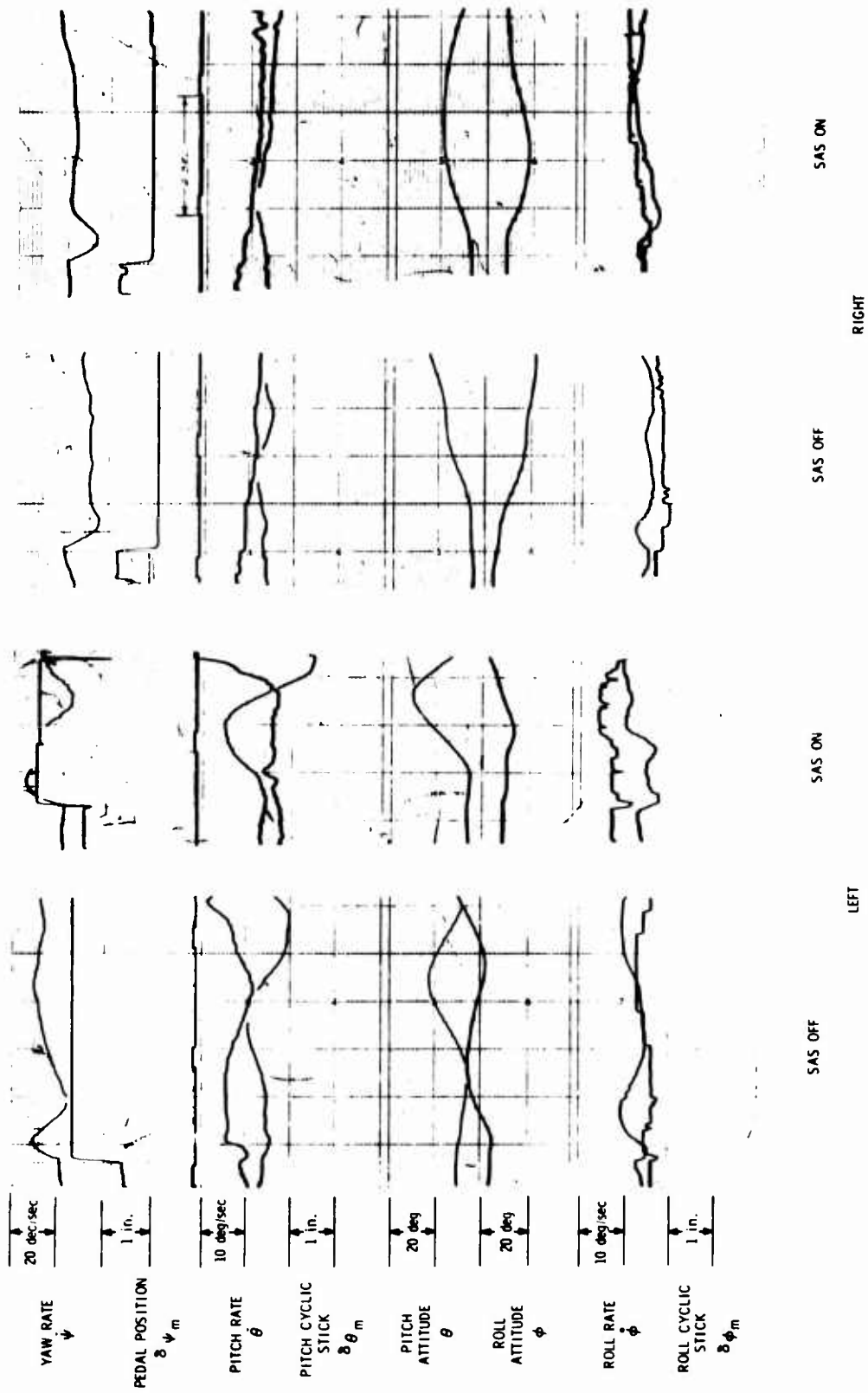


Figure 72. Aircraft Response to Pedal Steps (Hover, 3000 ft)

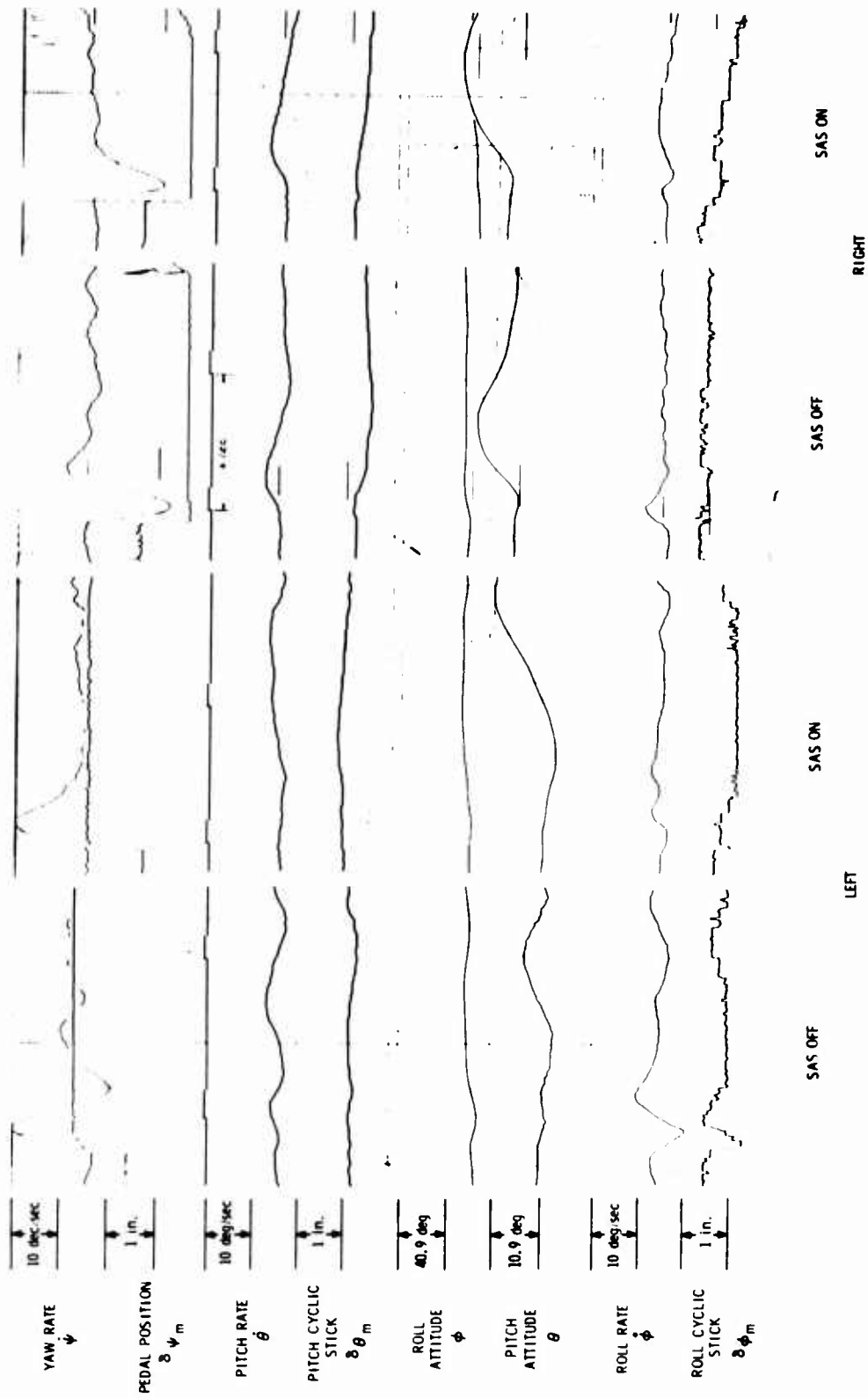


Figure 73. Aircraft Response to Pedal Steps (60 kn, 3000 ft)

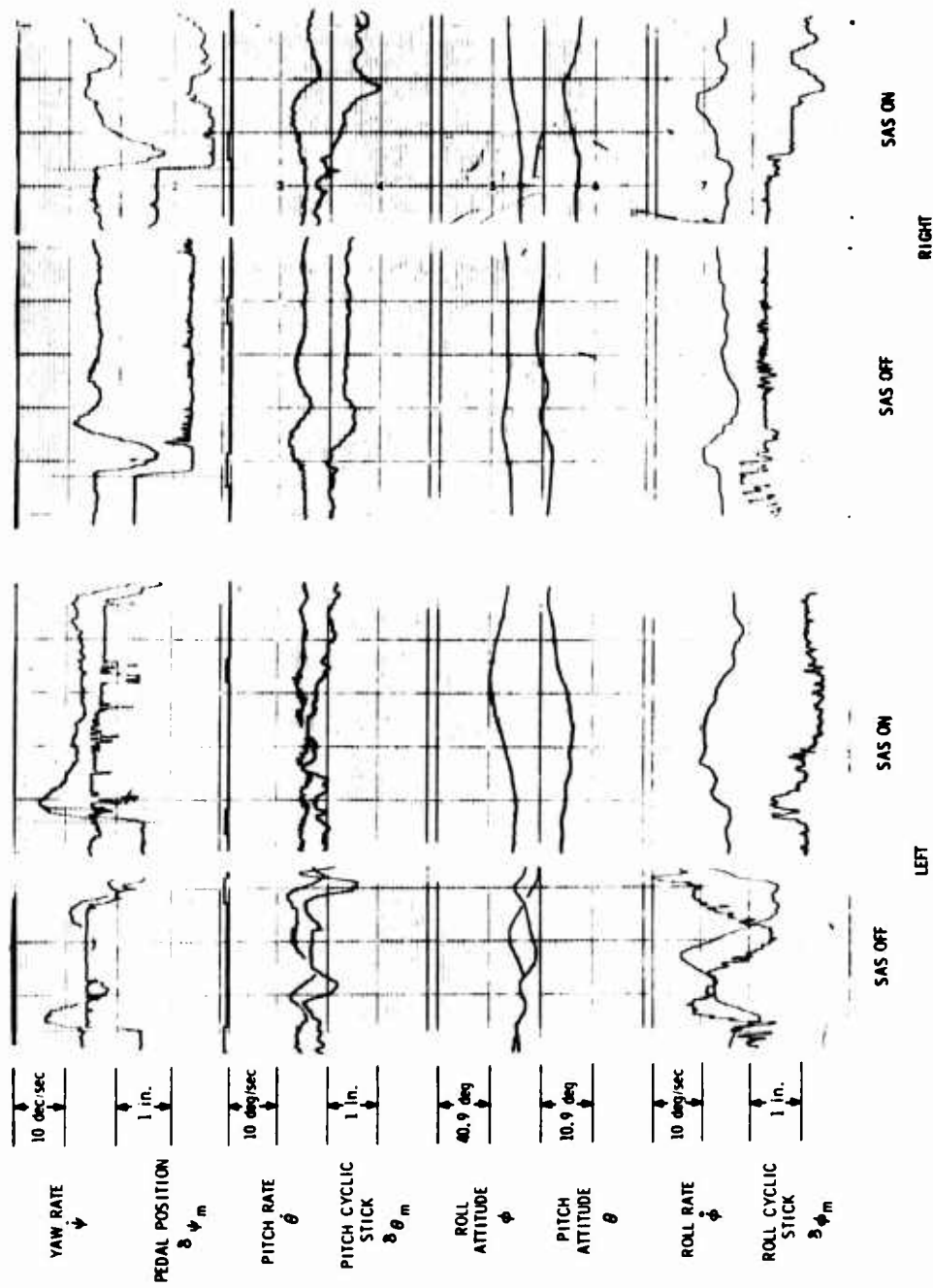


Figure 74. Aircraft Response to Pedal Steps (120 kn, 3000 ft)

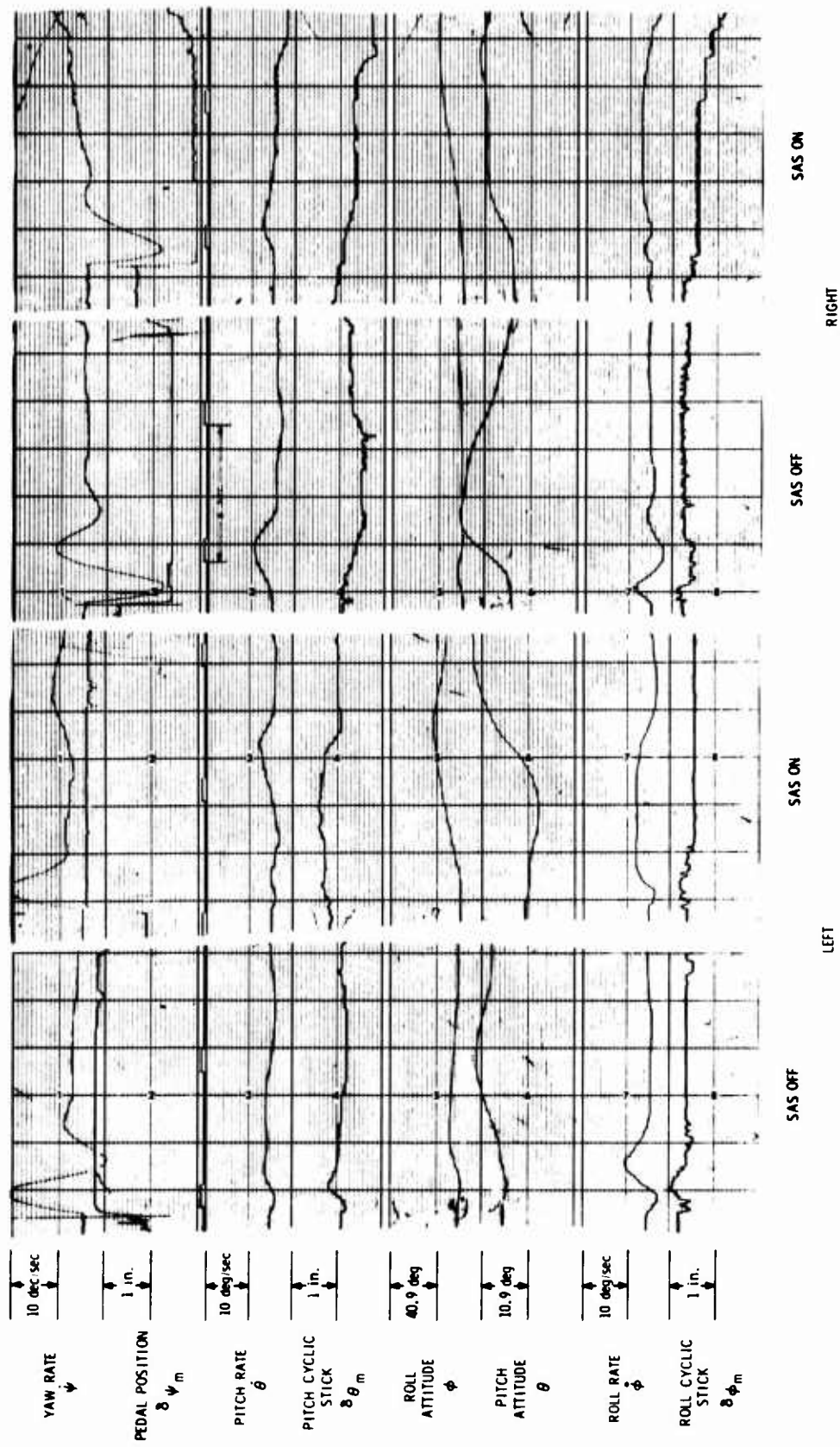


Figure 75. Aircraft Response to Pedal Steps (60 kn, 5000 ft)

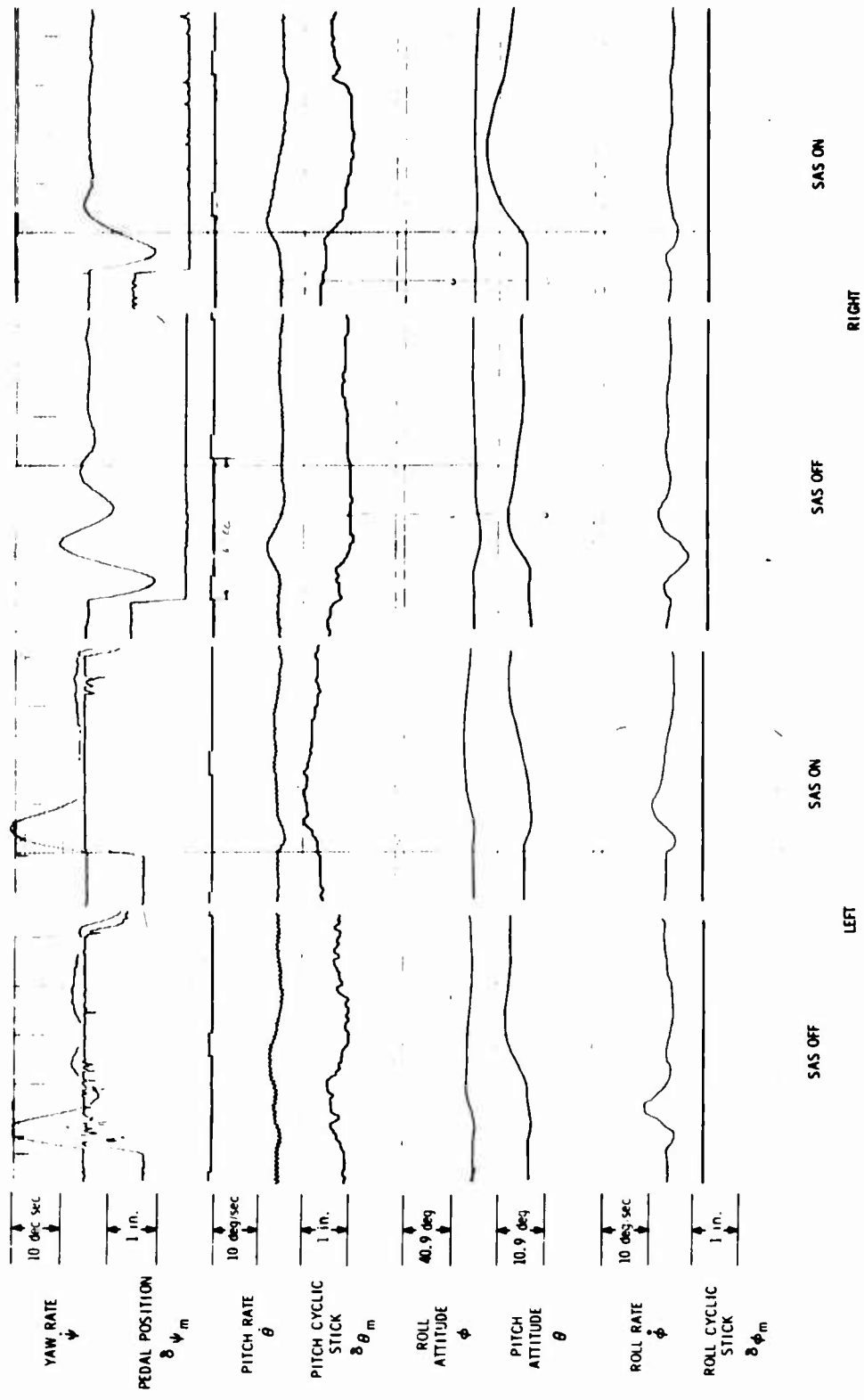


Figure 76. Aircraft Response to Pedal Steps (60 kn, 10,000 ft)

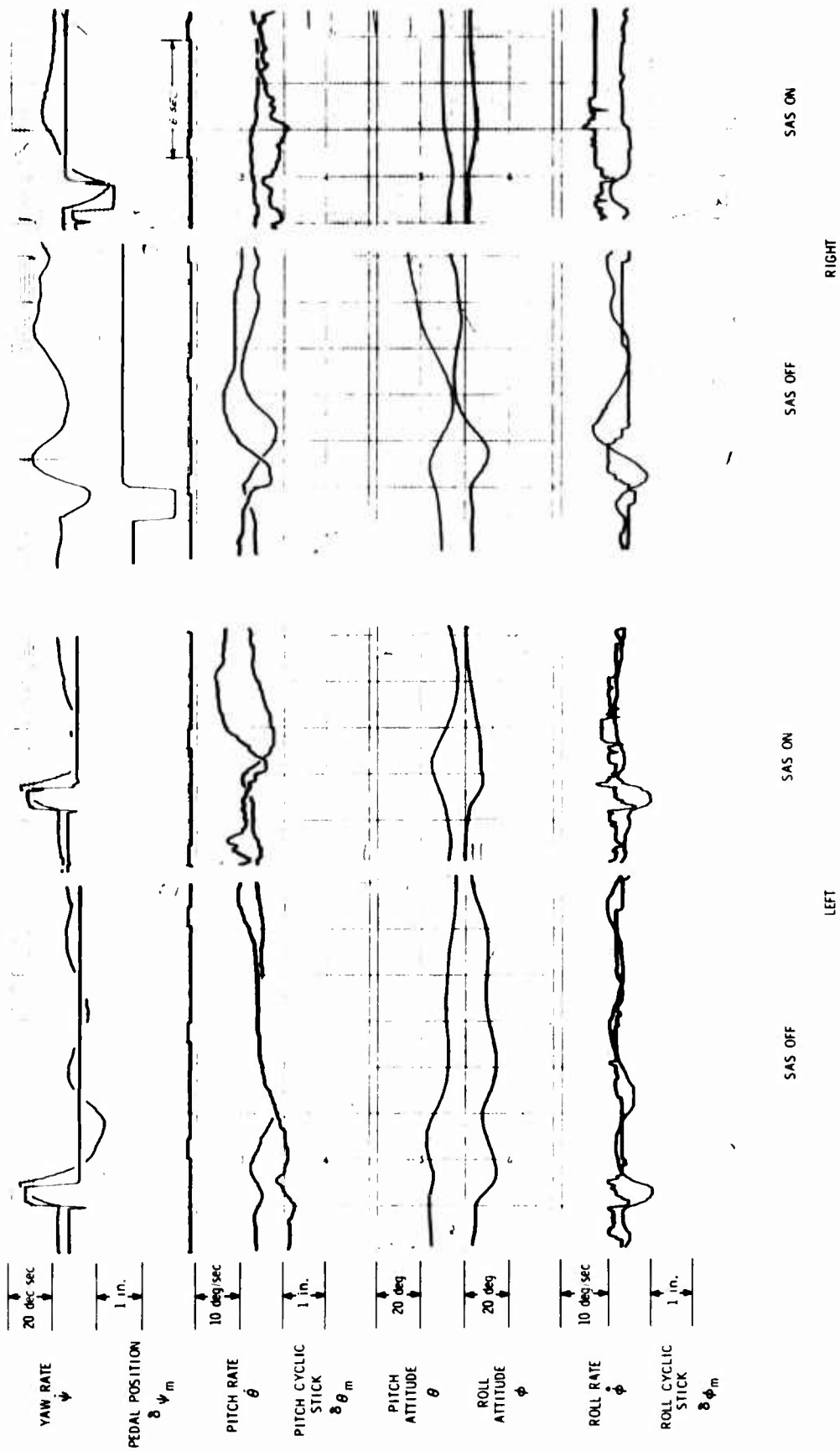


Figure 77. Aircraft Response to Pedal Pulses (Hover, 3000 ft)

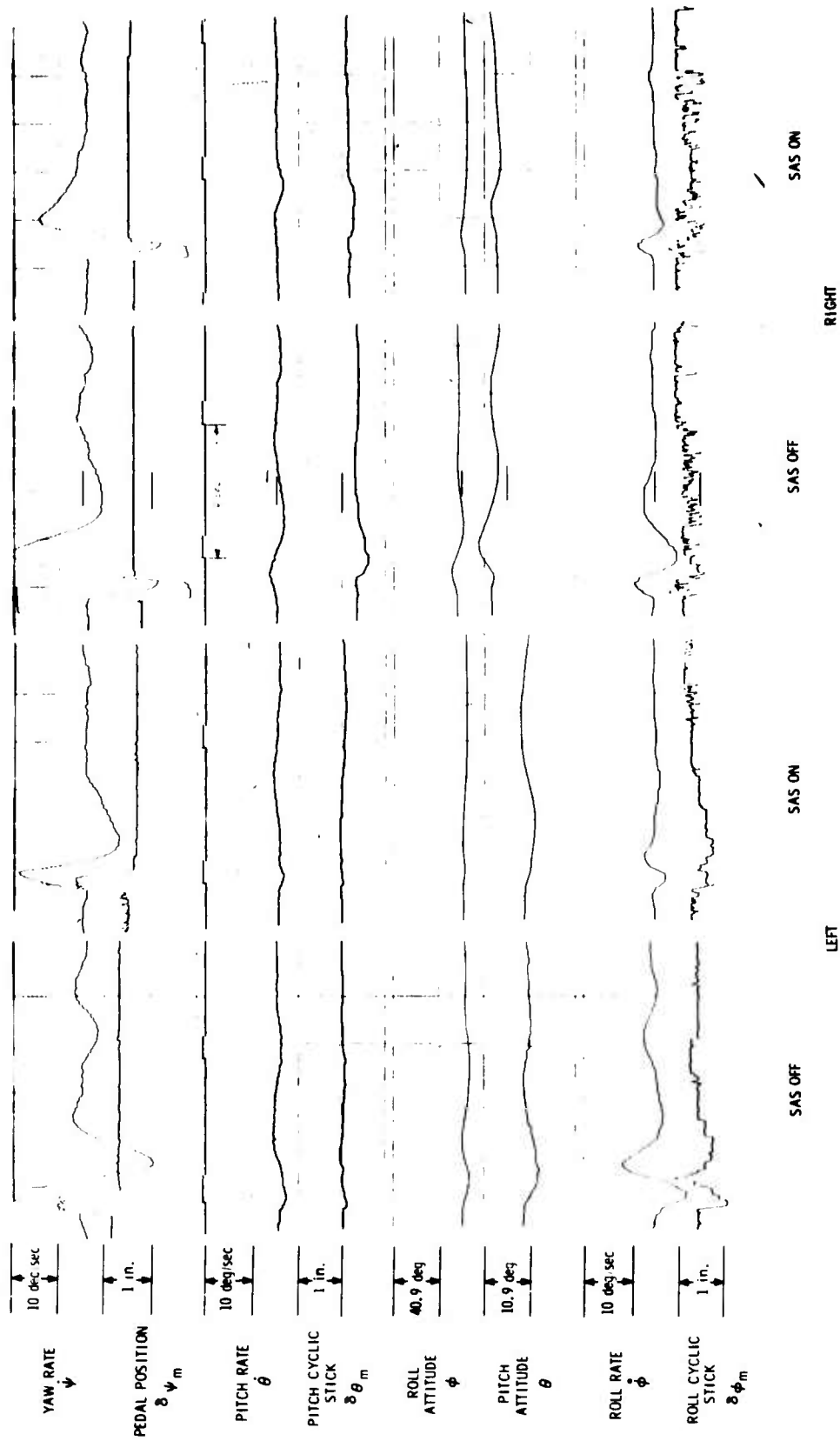


Figure 78. Aircraft Response to Pedal Pulses (60 kn, 3000 ft)

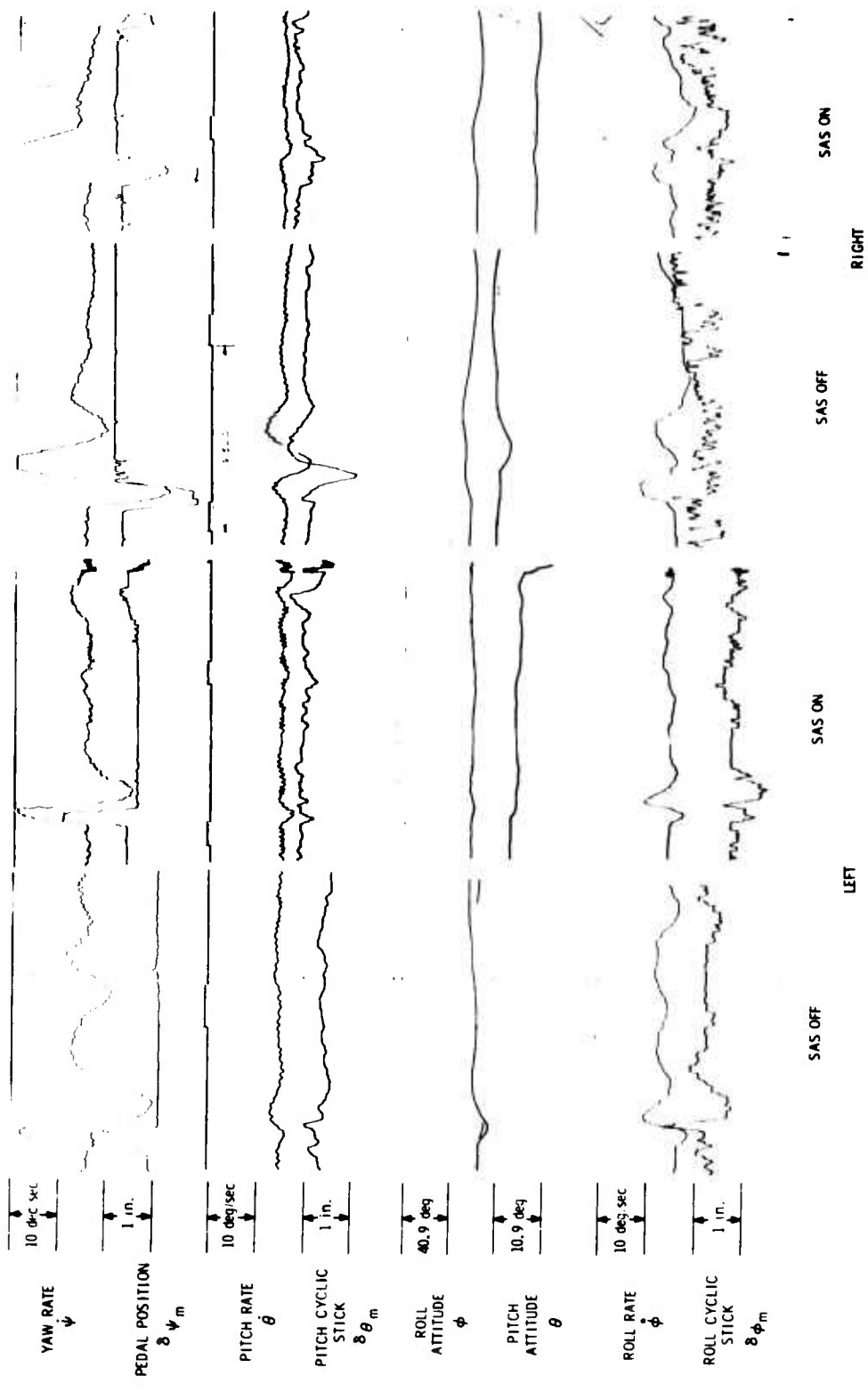


Figure 79. Aircraft Response to Pedal Pulses (120 kn, 3000 ft)

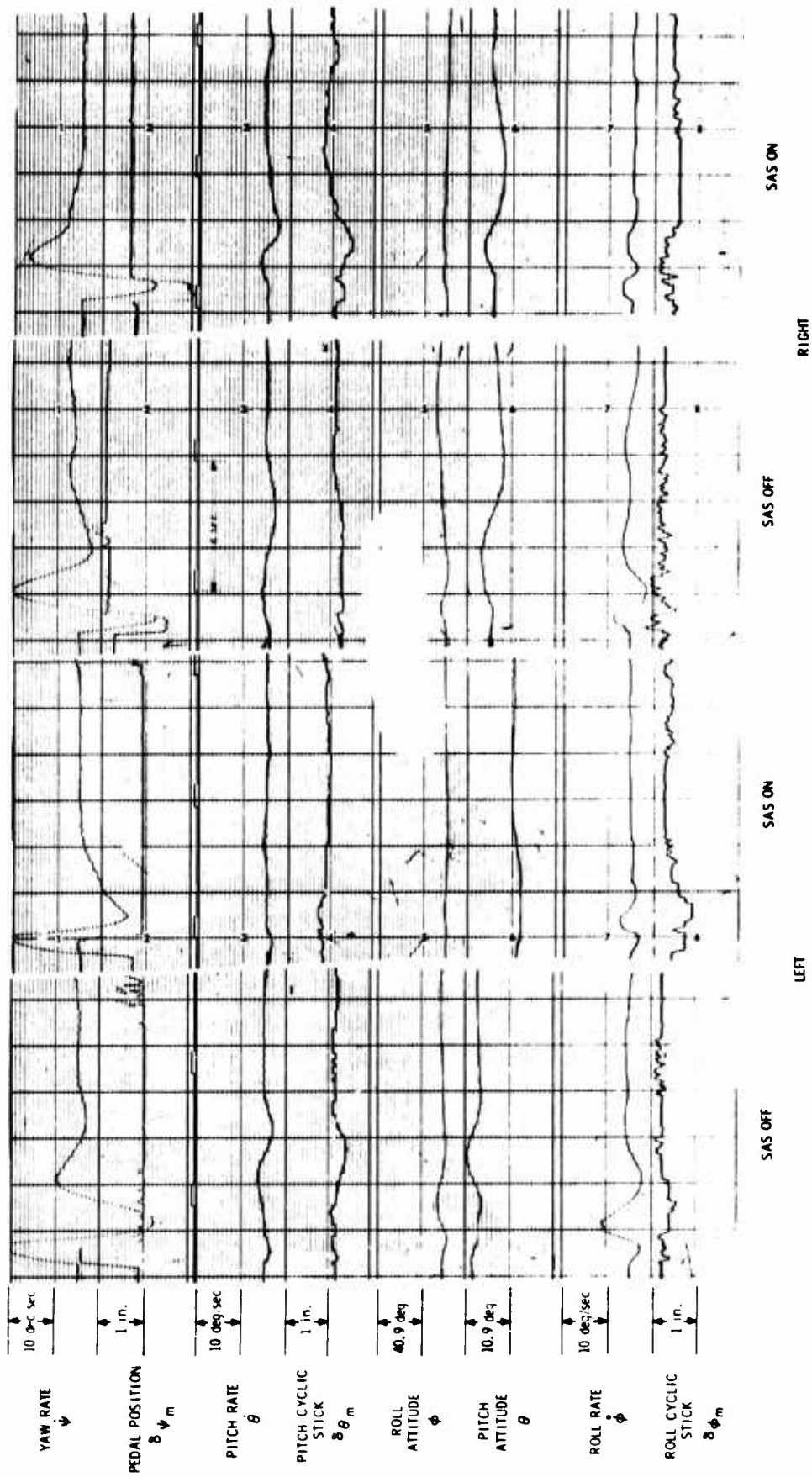


Figure 80. Aircraft Response to Pedal Pulses (60 kn, 5000 ft)

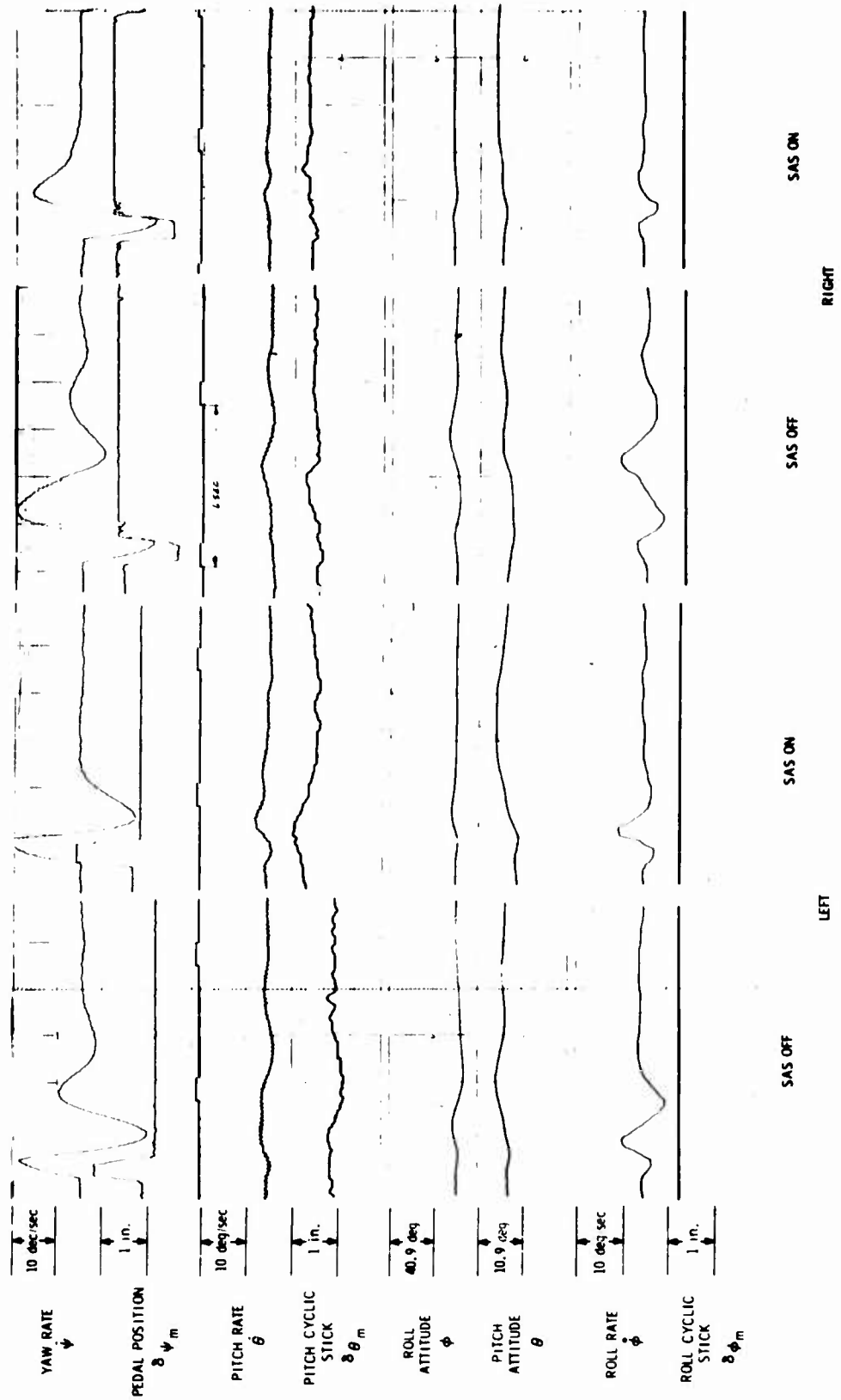


Figure 81. Aircraft Response to Pedal Pulses (60 kn, 10,000 ft)

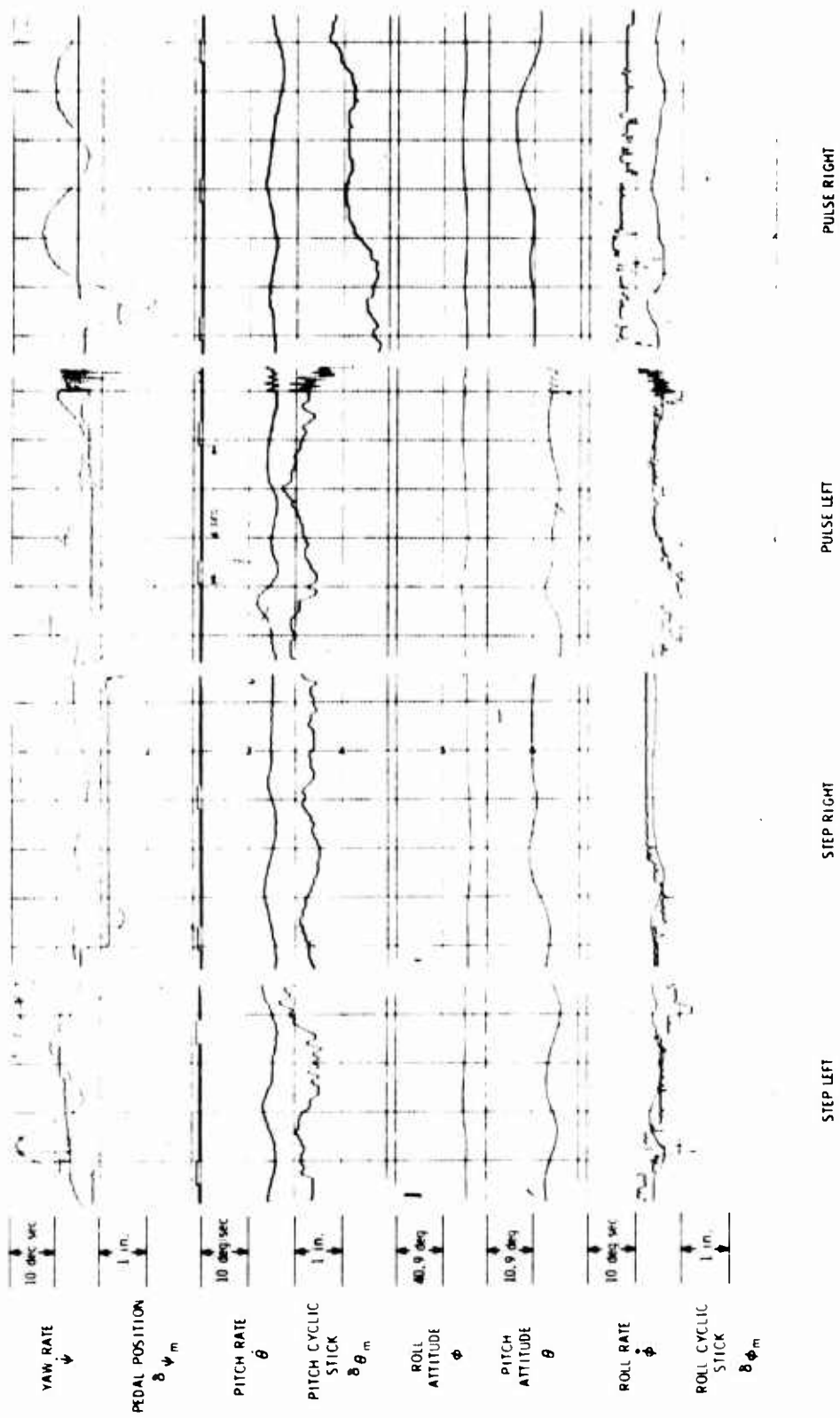


Figure 82. Aircraft Response to Pedal Inputs (Hover, 3000 ft) With Stabilizer Bar

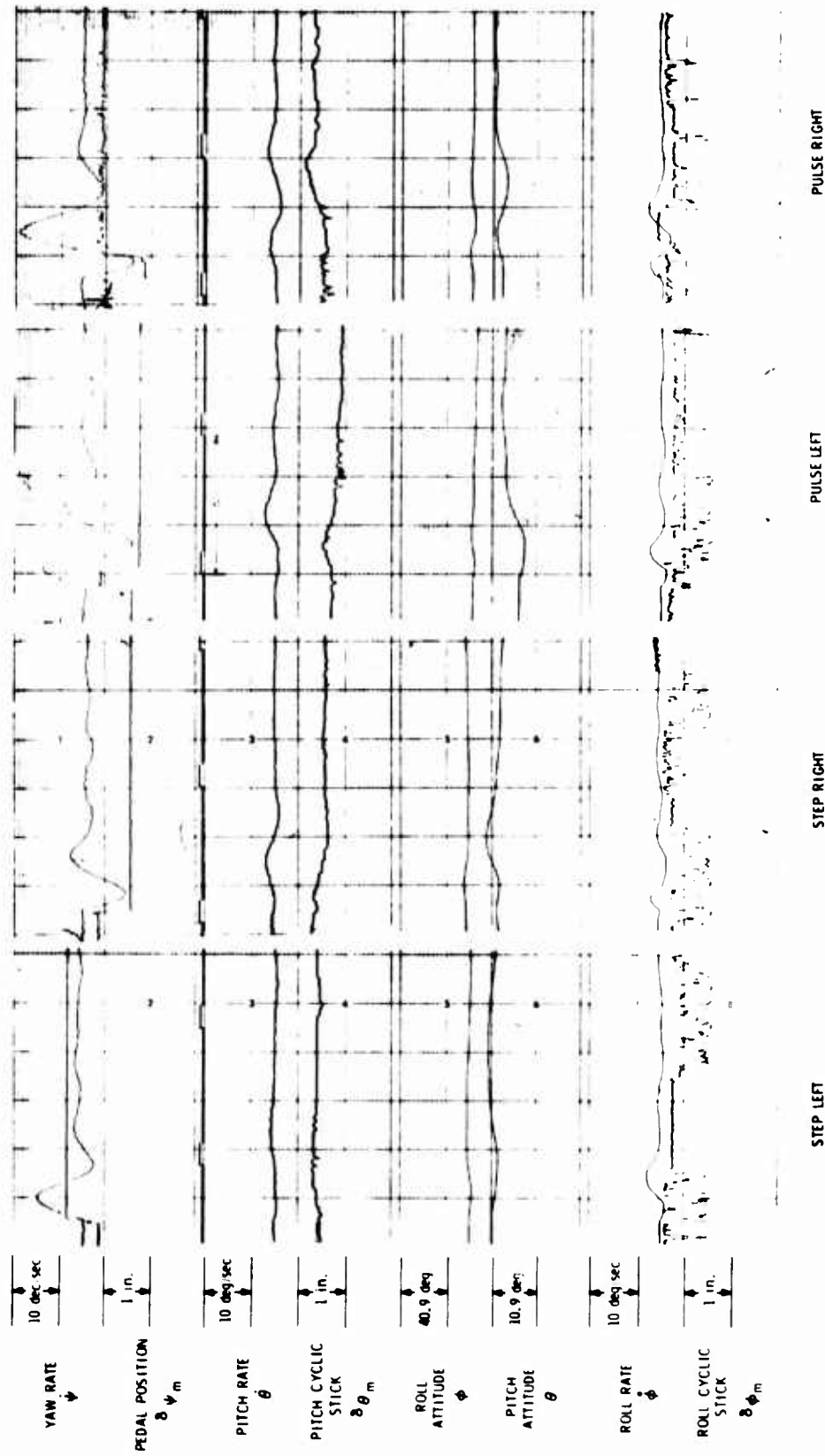


Figure 83. Aircraft Response to Pedal Inputs (60 kn, 3000 ft) With Stabilizer Bar

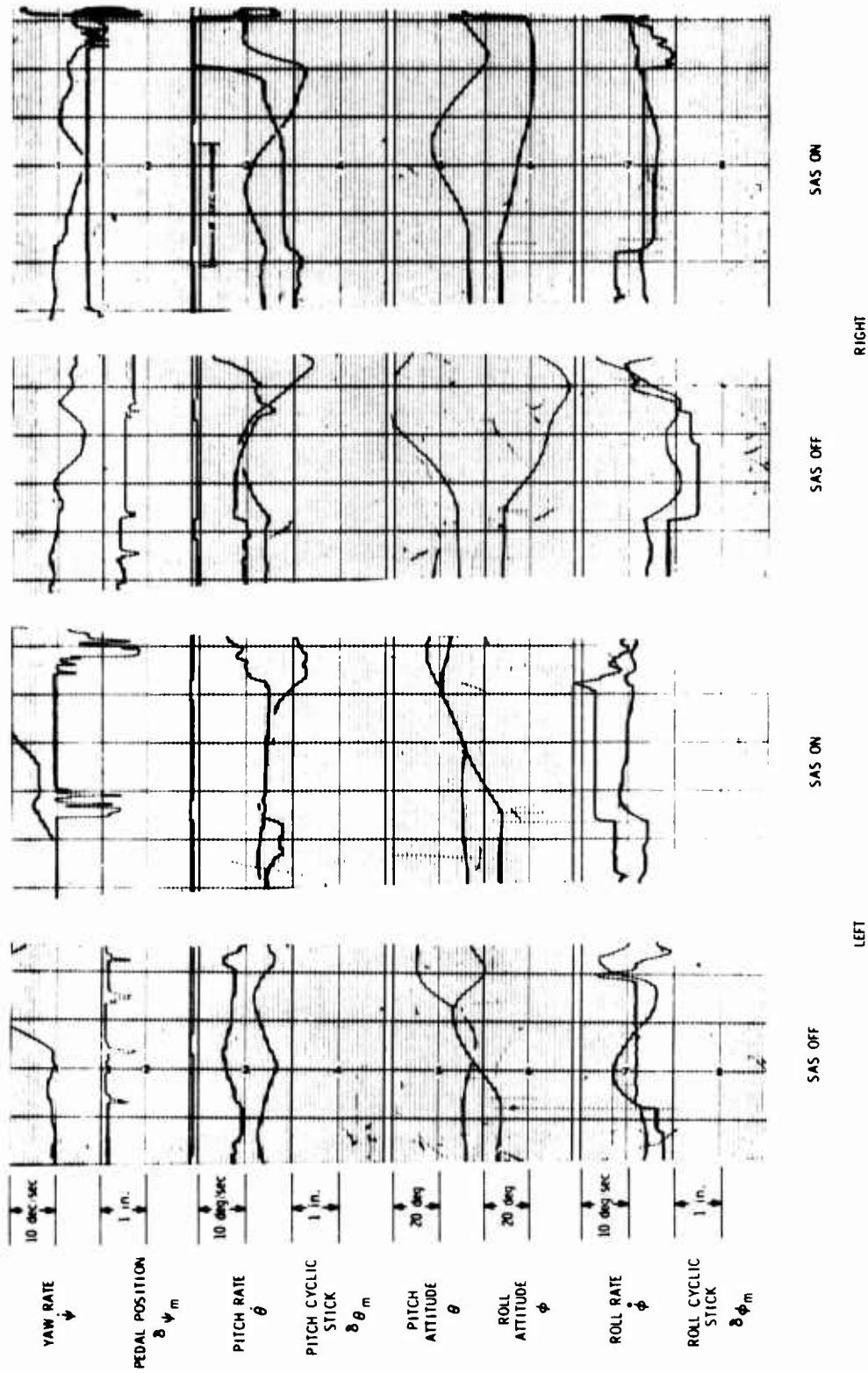


Figure 84. Aircraft Response to Roll Cyclic Steps (Hover, 3000 ft)

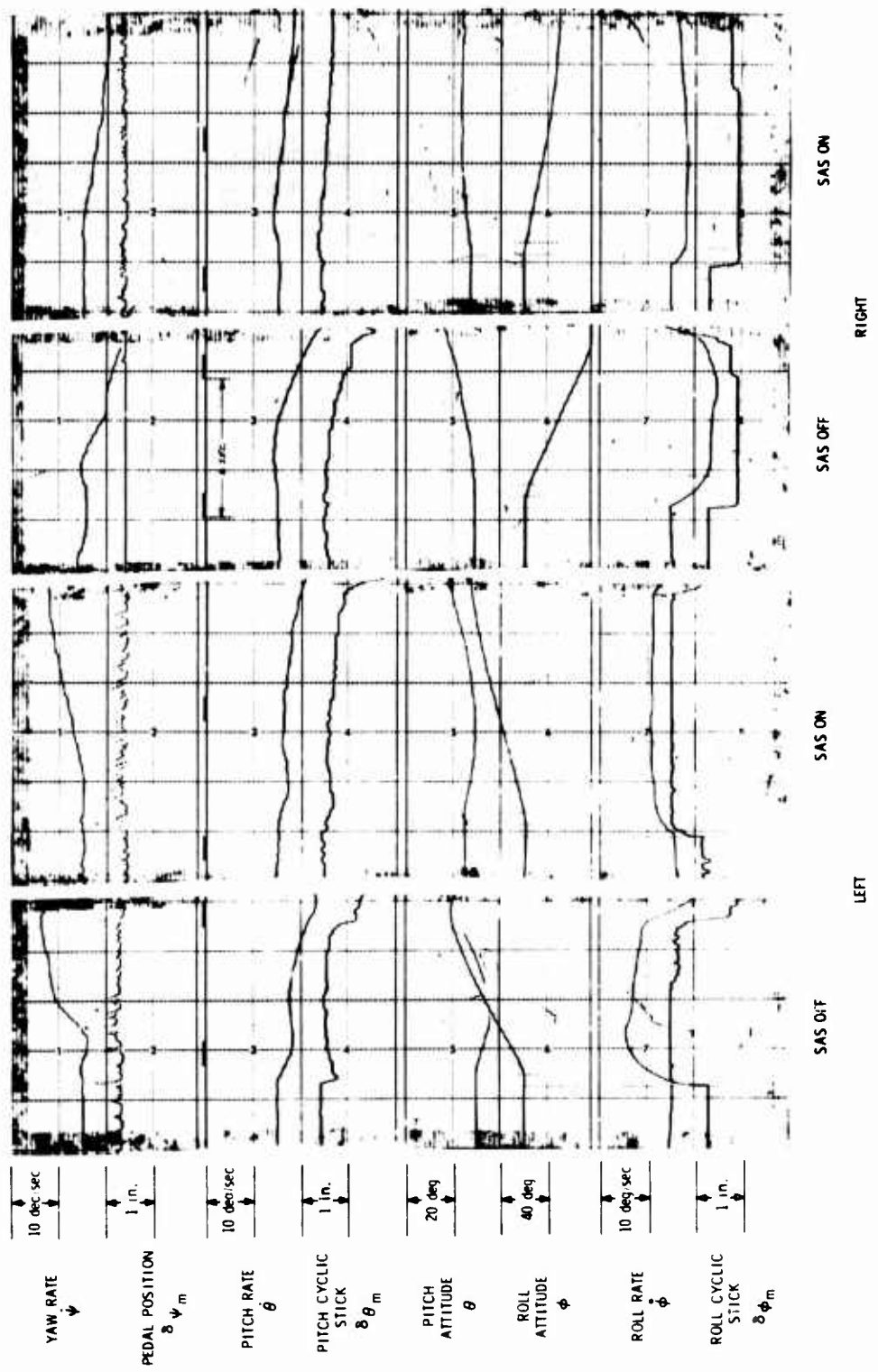


Figure 85. Aircraft Response to Roll Cyclic Steps (60 kn, 3000 ft)



Figure 86. Aircraft Response to Roll Cyclic Steps (120 kn, 3000 ft)

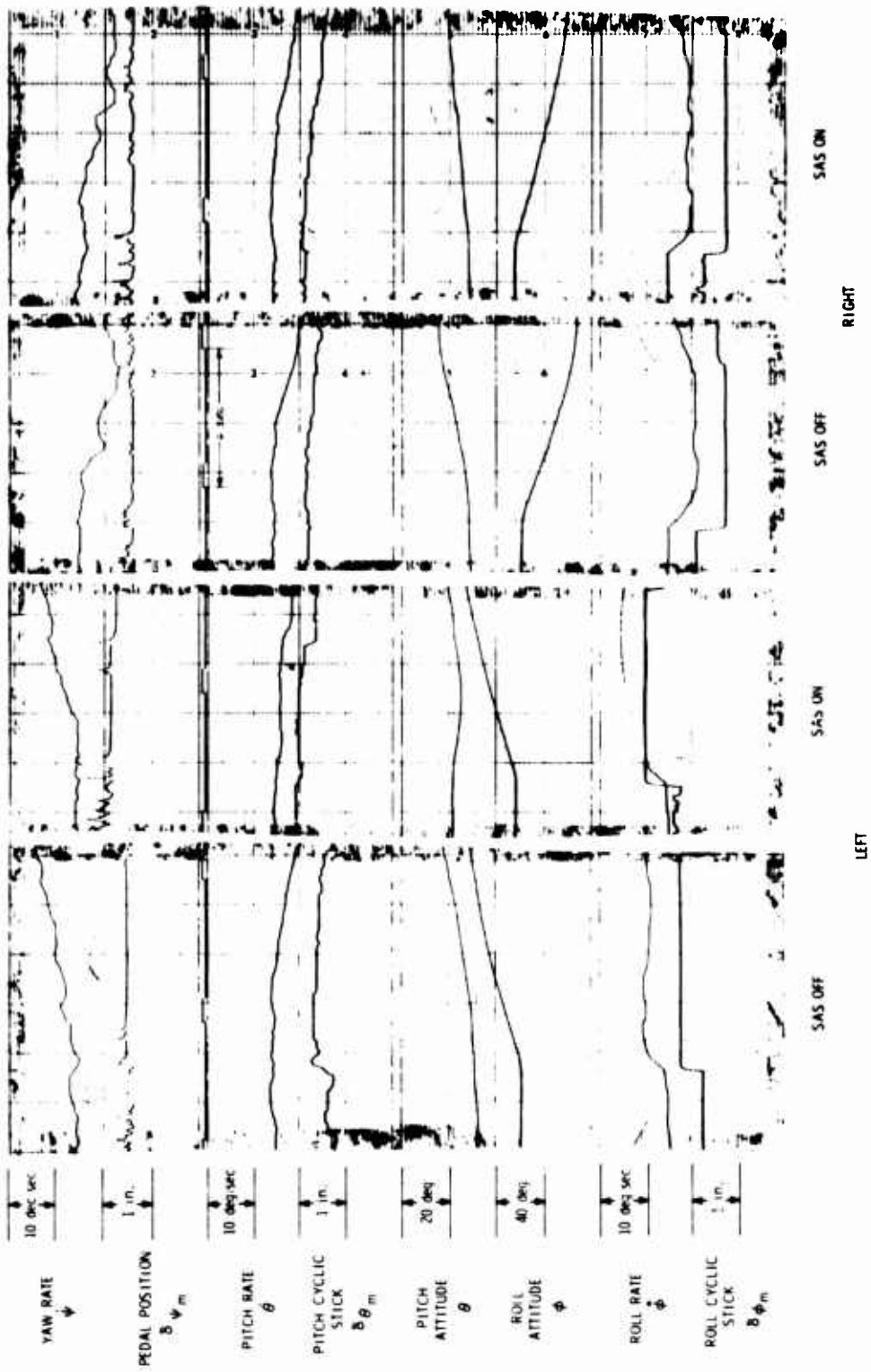


Figure 87. Aircraft Response to Roll Cyclic Steps (60 kn, 5000 ft)

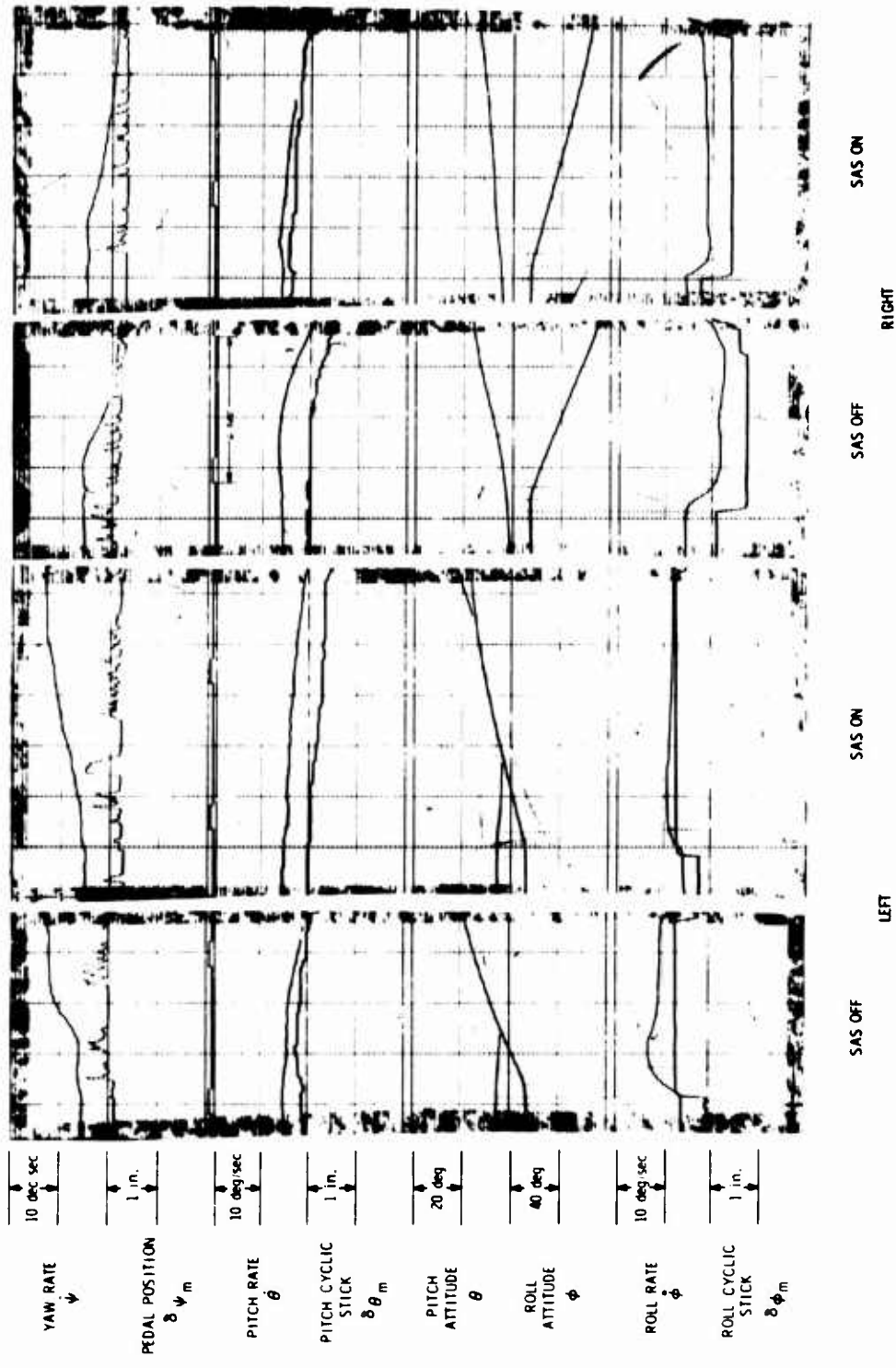


Figure 88. Aircraft Response to Roll Cyclic Steps (60 kn, 10,000 ft)

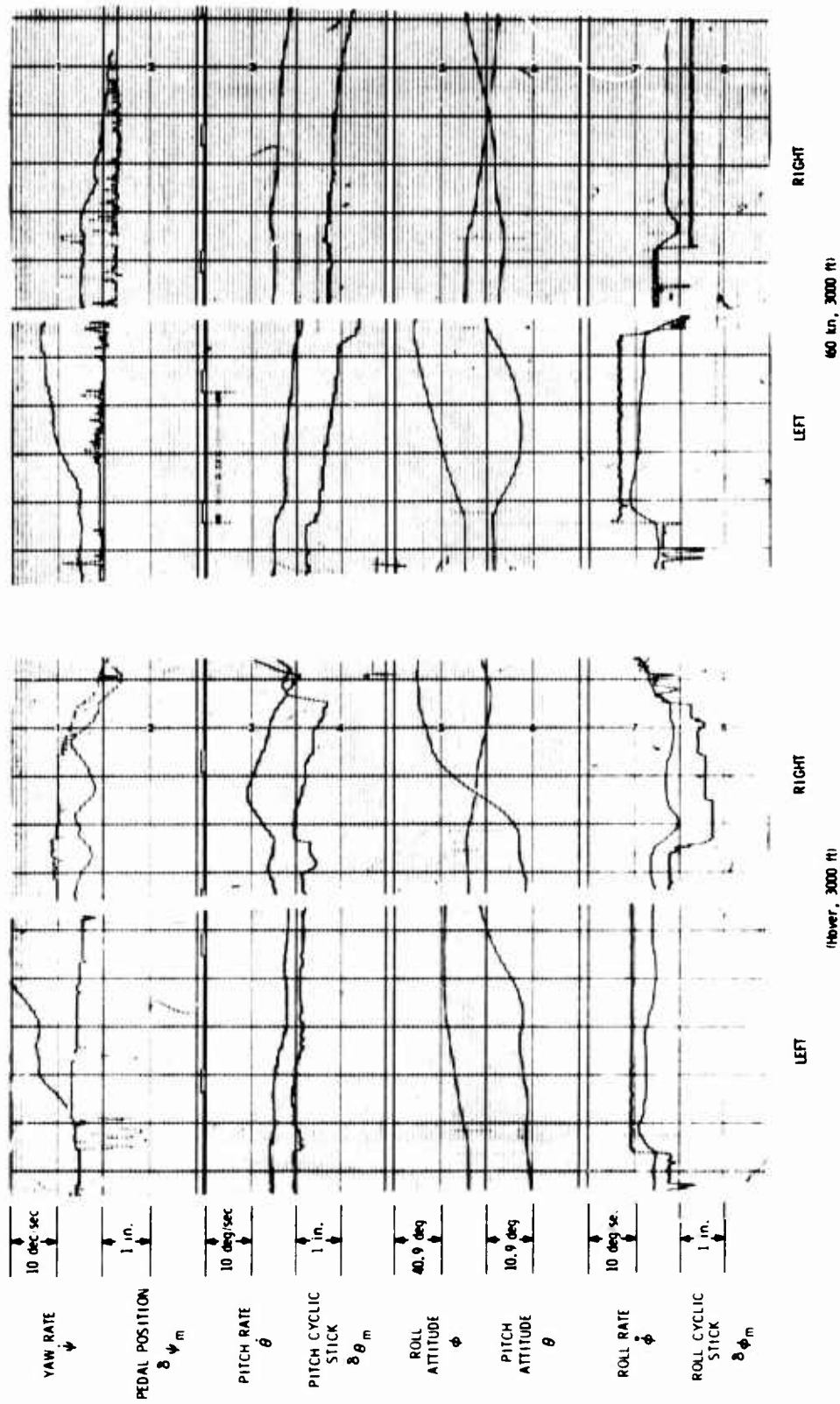


Figure 89. Aircraft Response to Roll Cyclic Steps with Stabilizer Bar

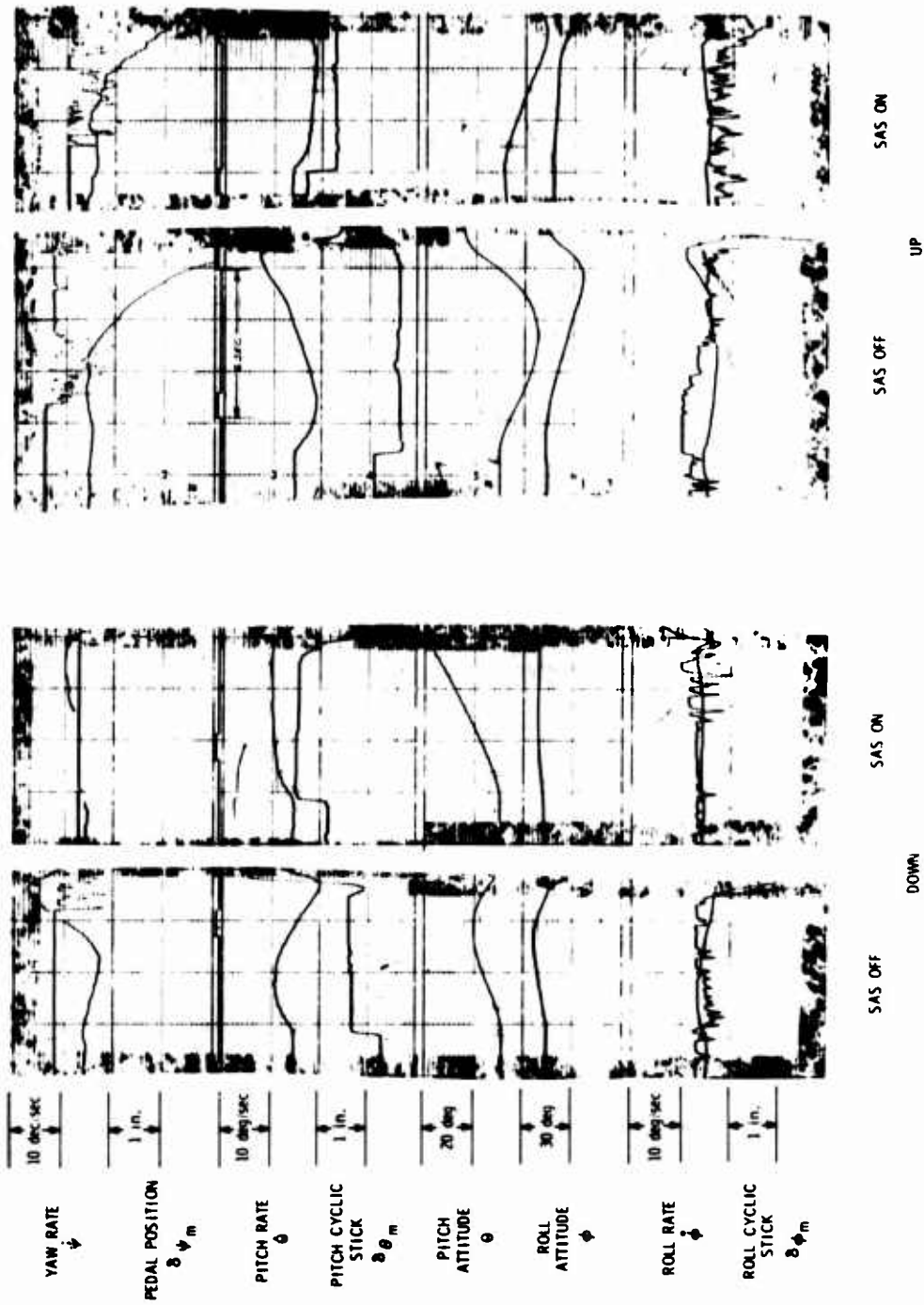


Figure 90. Aircraft Response to Pitch Cyclic Steps (Hover, 3000 ft)

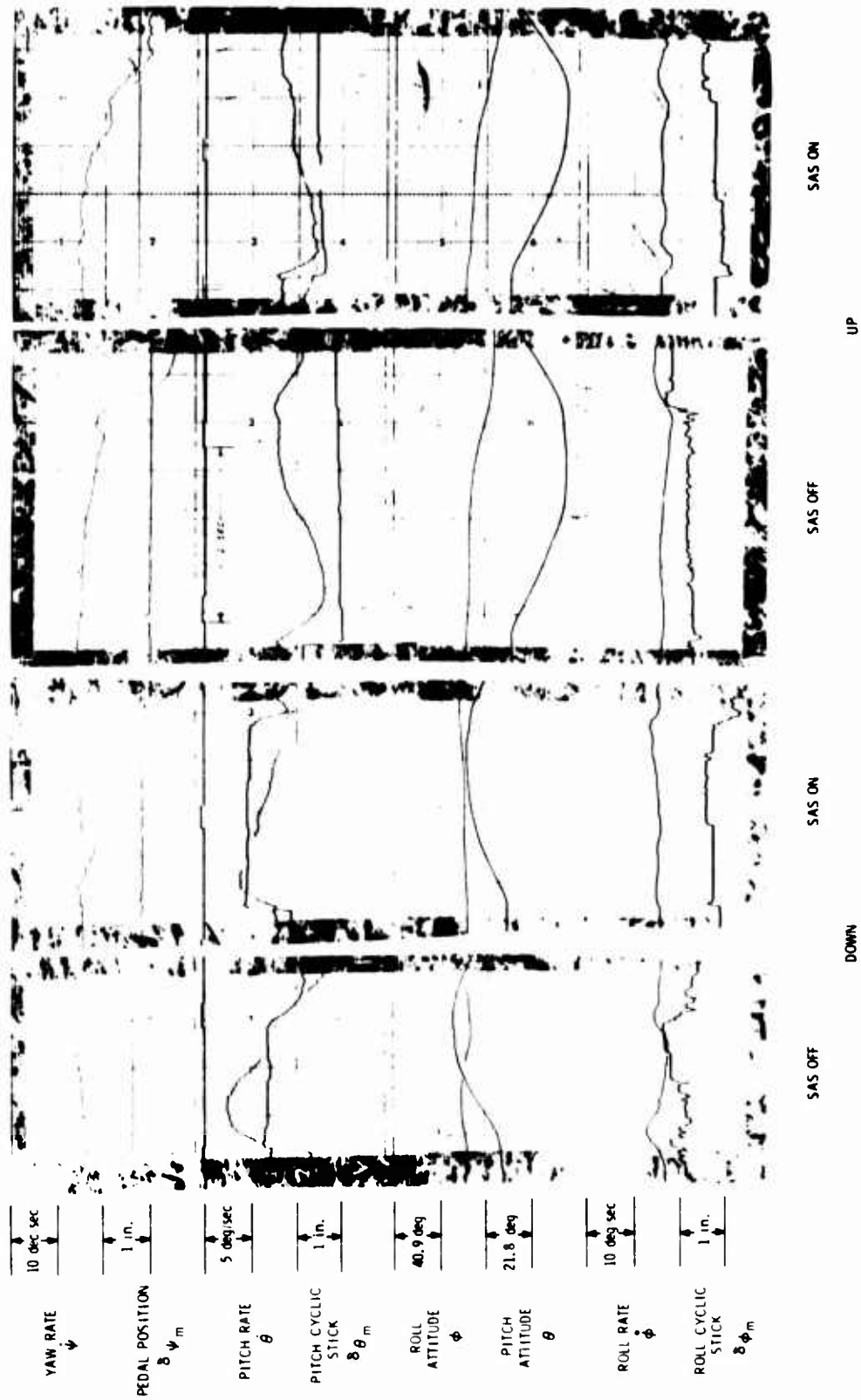


Figure 91. Aircraft Response to Pitch Cyclic Steps (60 kn, 3000 ft)

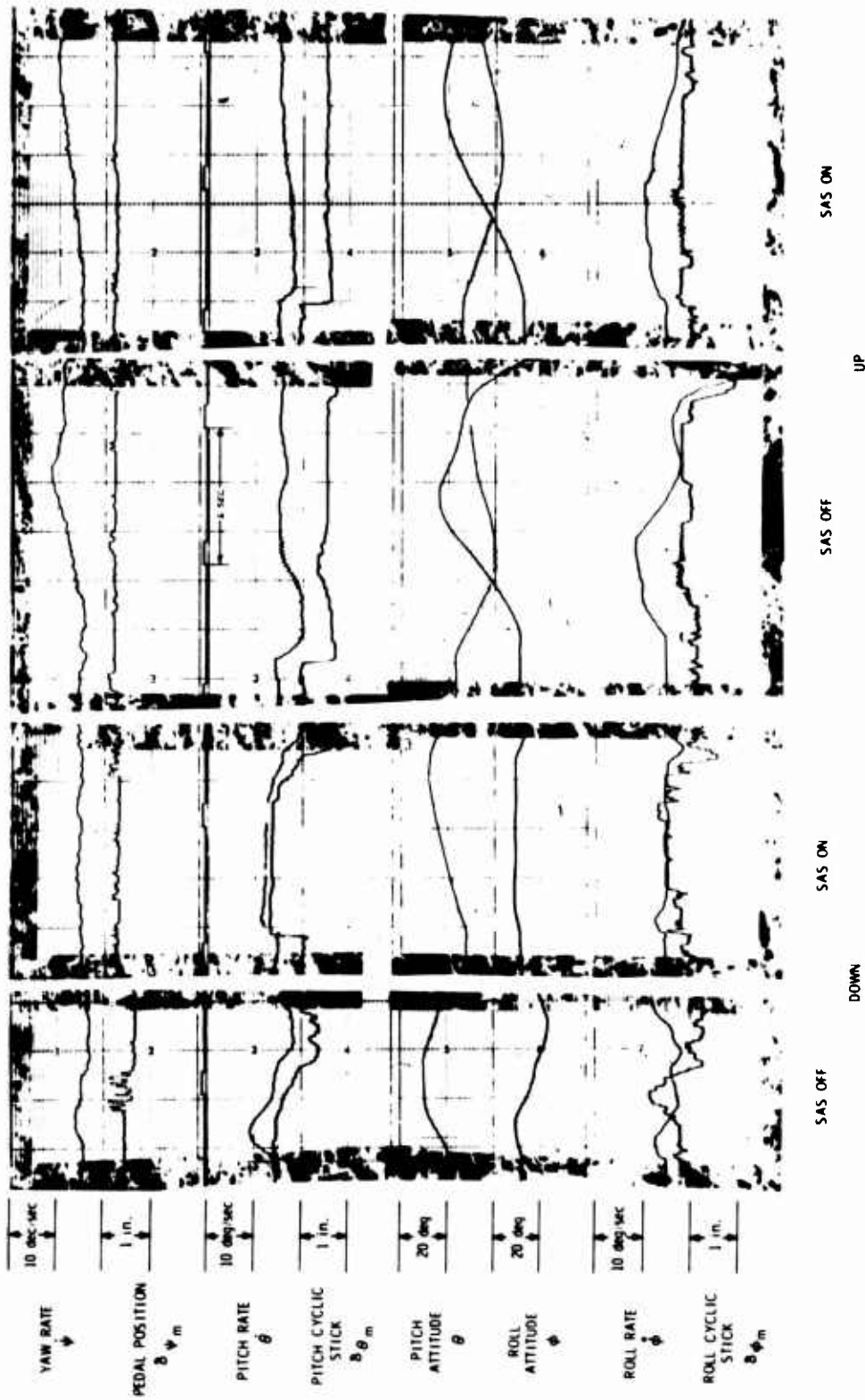


Figure 92. Aircraft Response to Pitch Cyclic Steps (120 kn, 3000 ft)

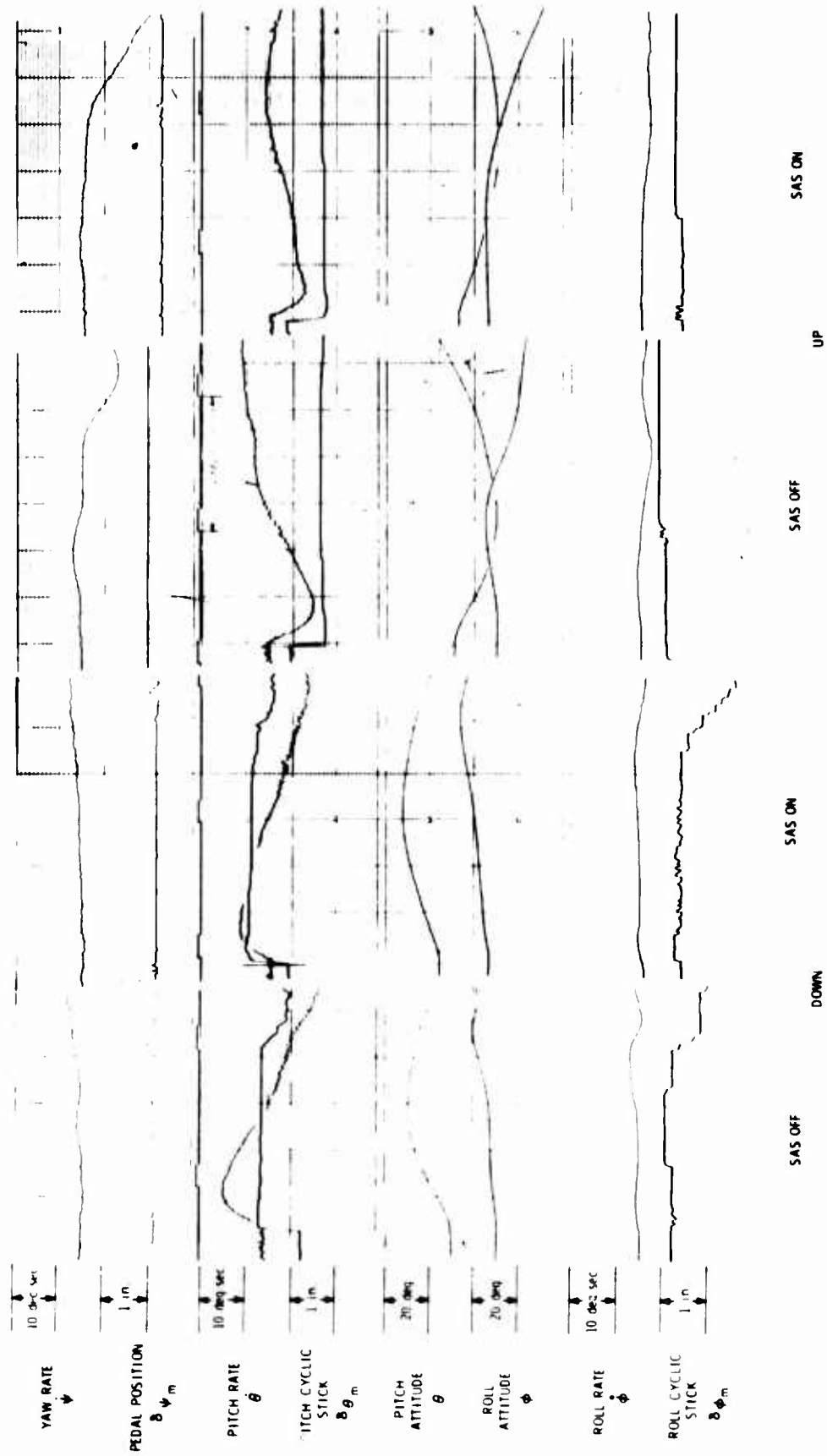


Figure 93. Aircraft Response to Pitch Cyclic Steps (60 kn, 5000 ft)

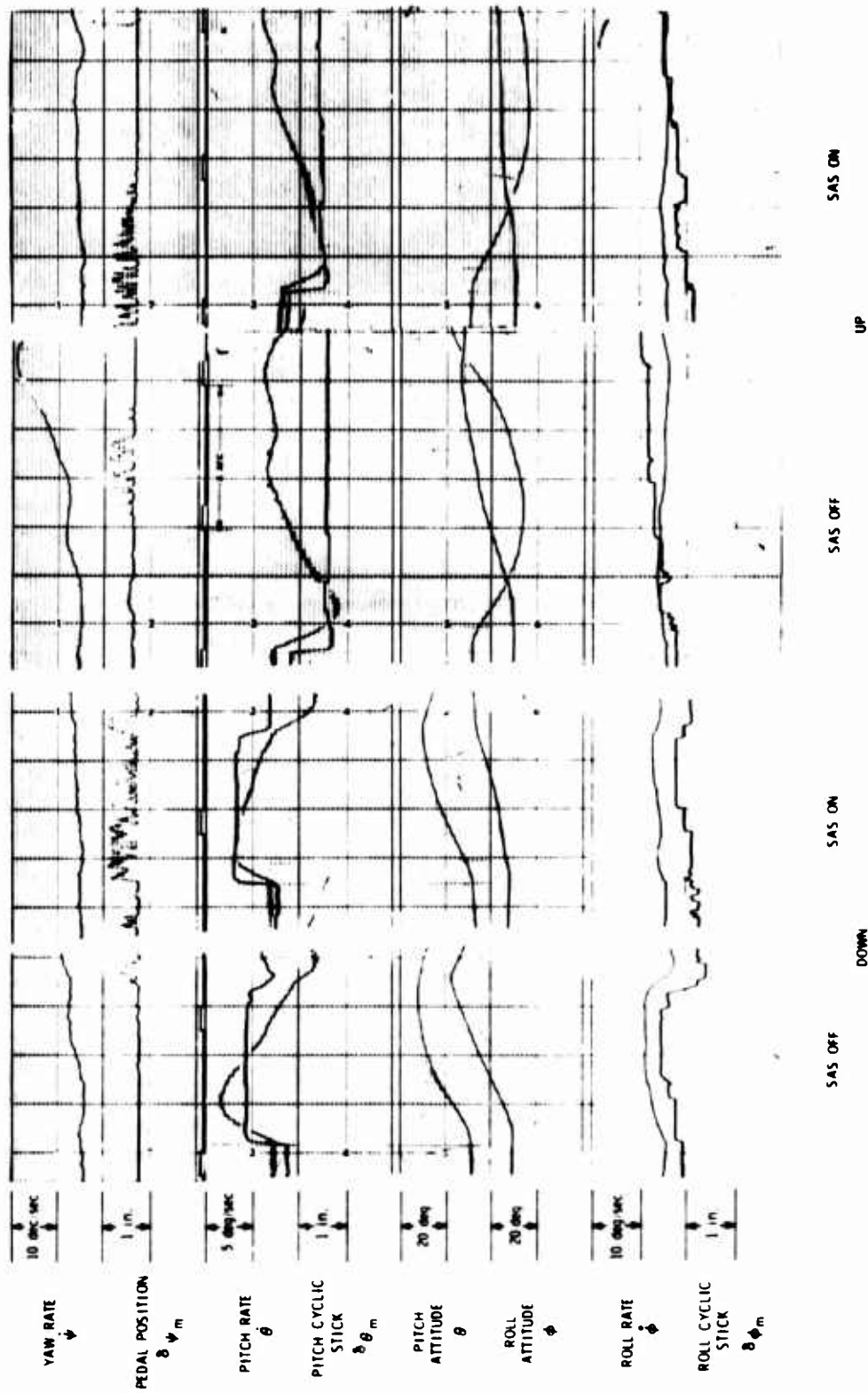


Figure 94. Aircraft Response to Pitch Cyclic Steps (60 kn, 10,000 ft)

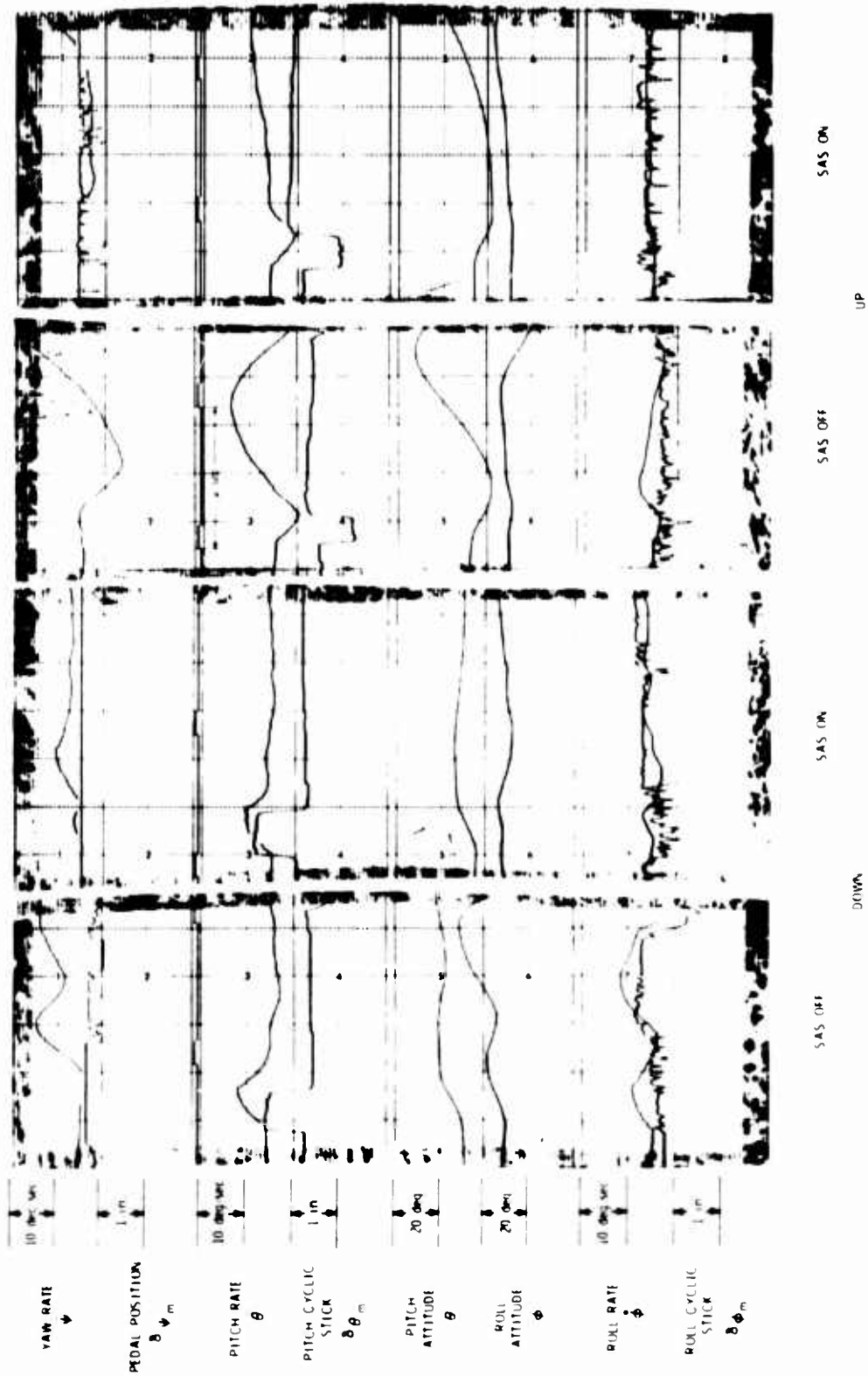


Figure 95. Aircraft Response to Pitch Cyclic Pulses (Hover, 3000 ft)

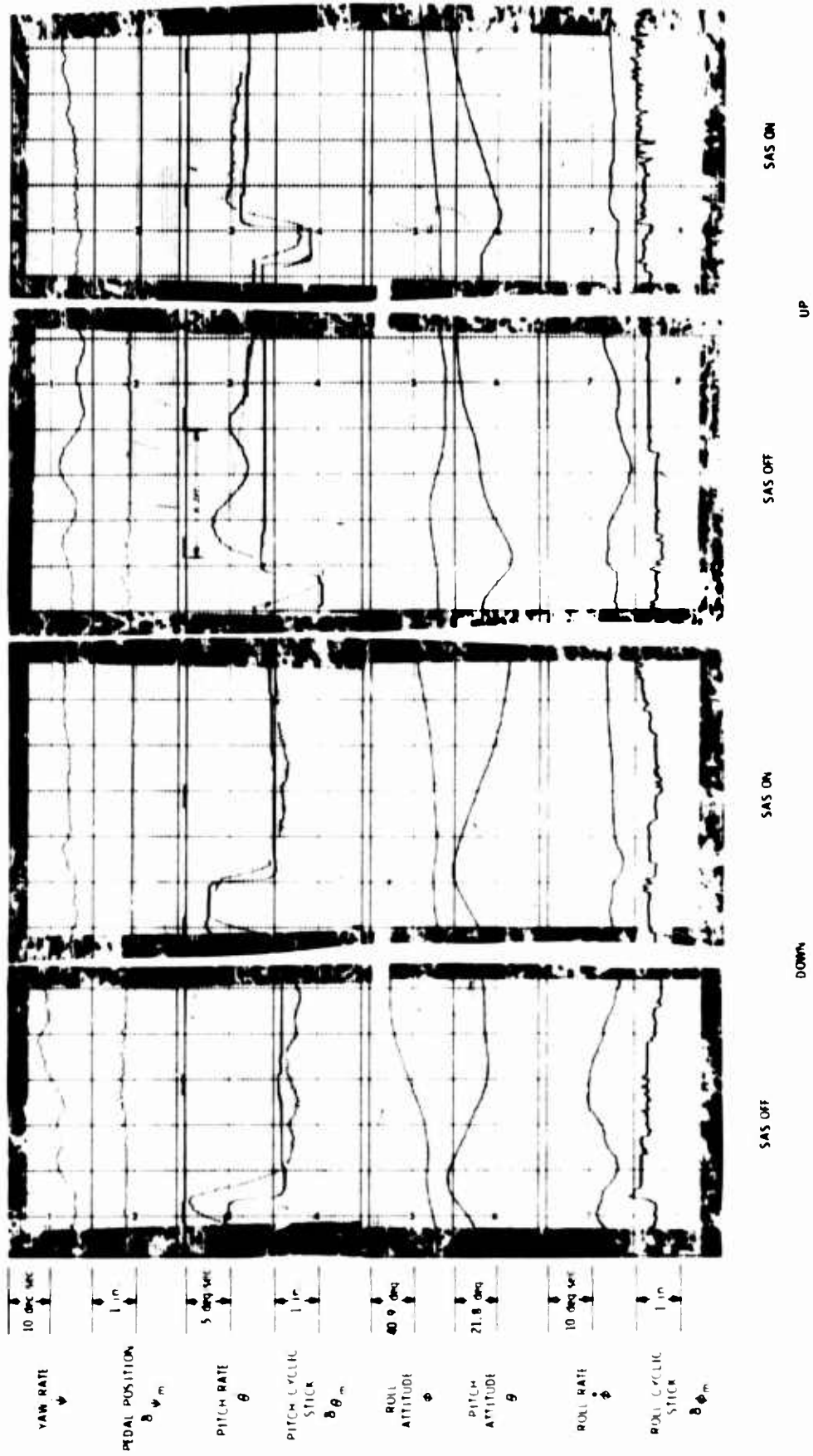


Figure 96. Aircraft Response to Pitch Cyclic Pulses (60 kn, 3000 ft)

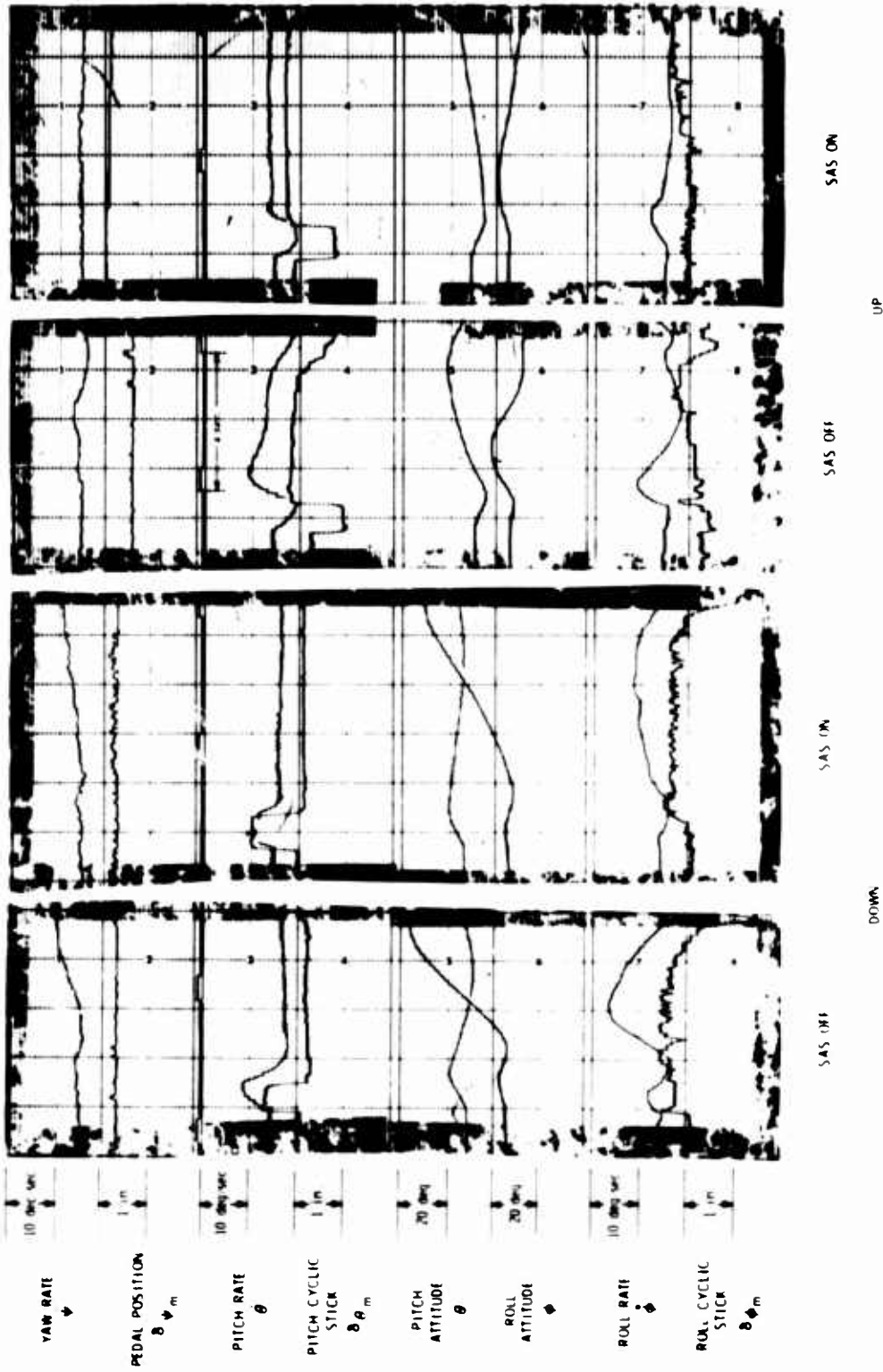


Figure 97. Aircraft Response to Pitch Cyclic Pulses (120 kn, 3000 ft)

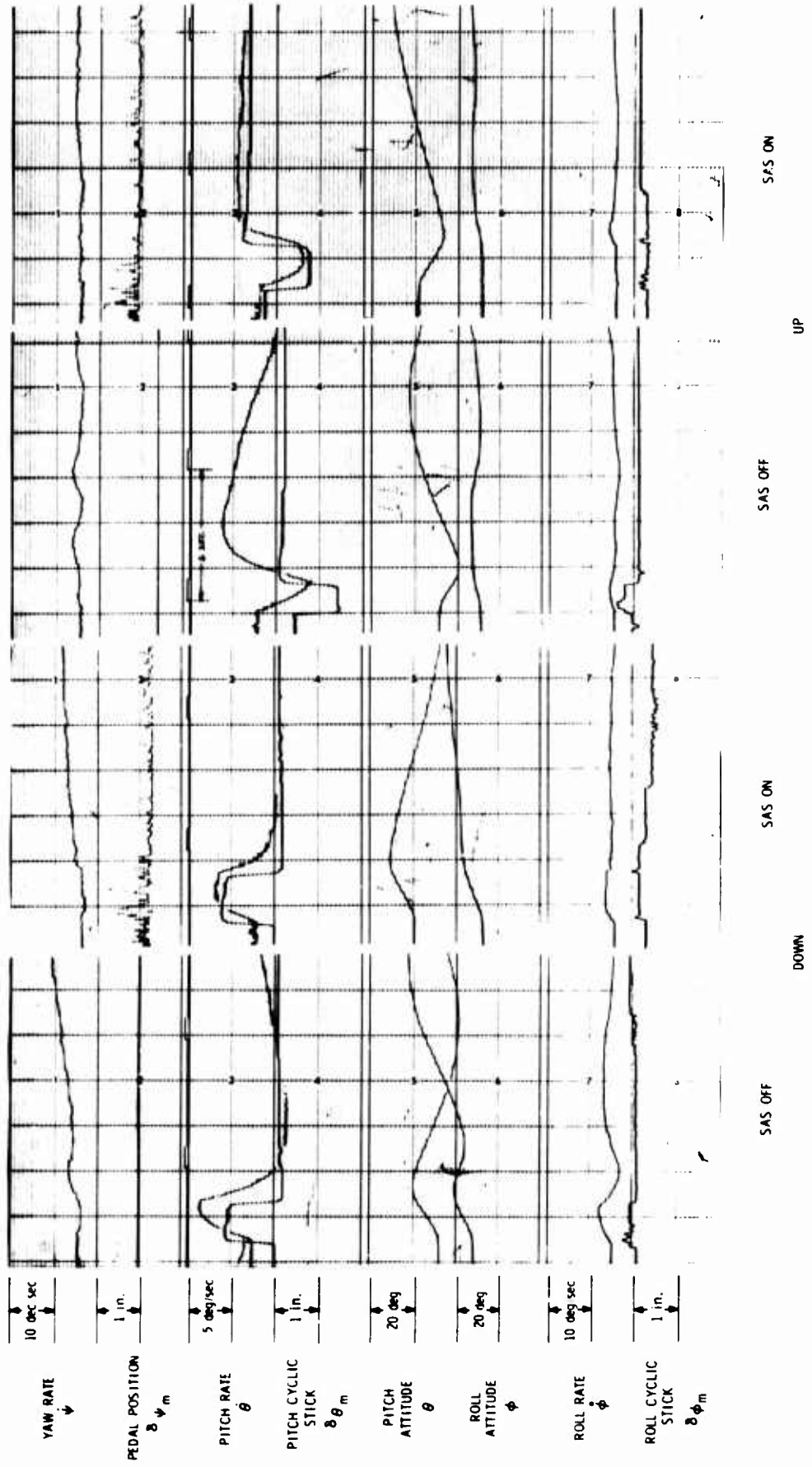


Figure 98. Aircraft Response to Pitch Cyclic Pulses (60 kn, 5000 ft)

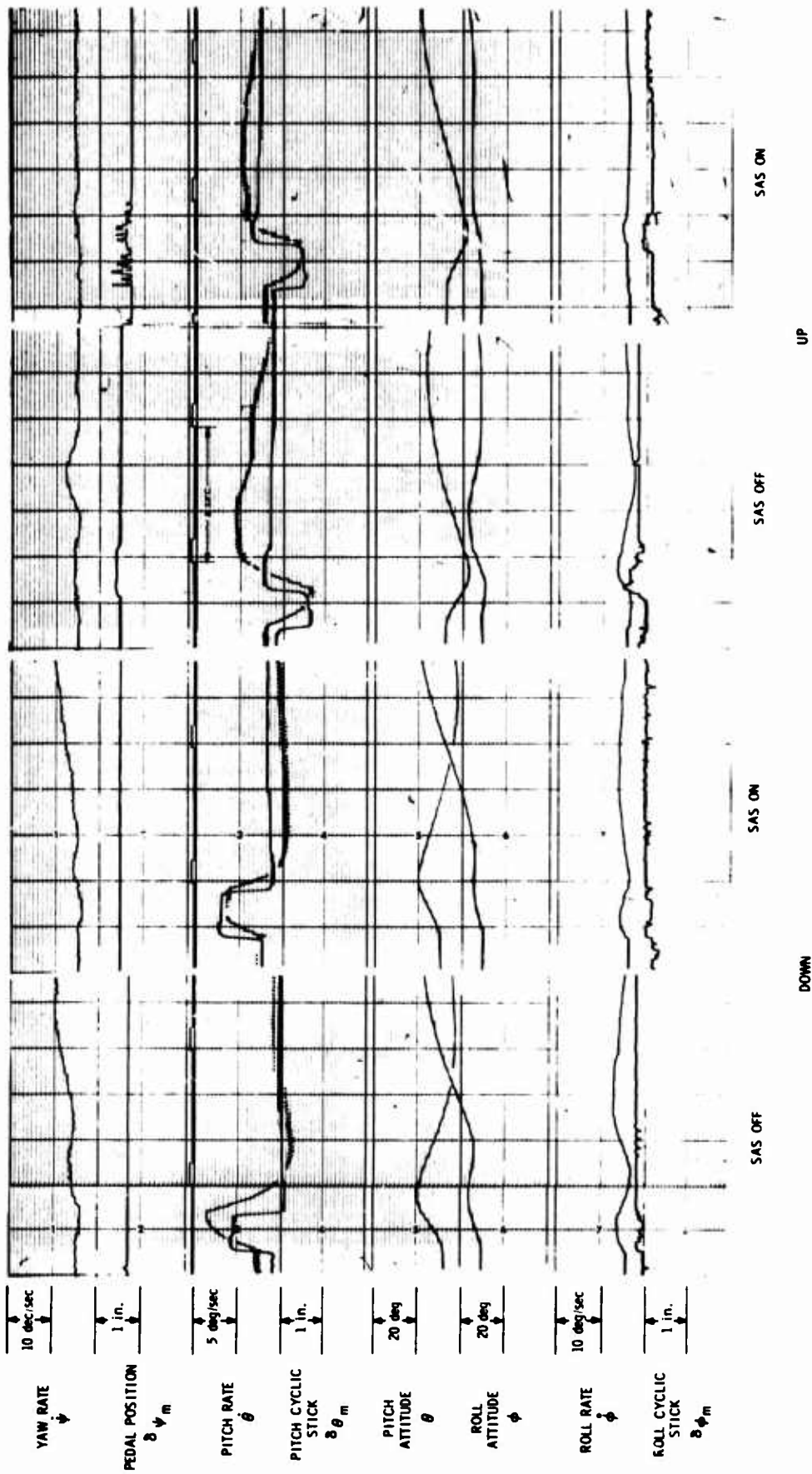


Figure 99. Aircraft Response to Pitch Cyclic Pulses (60 kn, 10,000 ft)

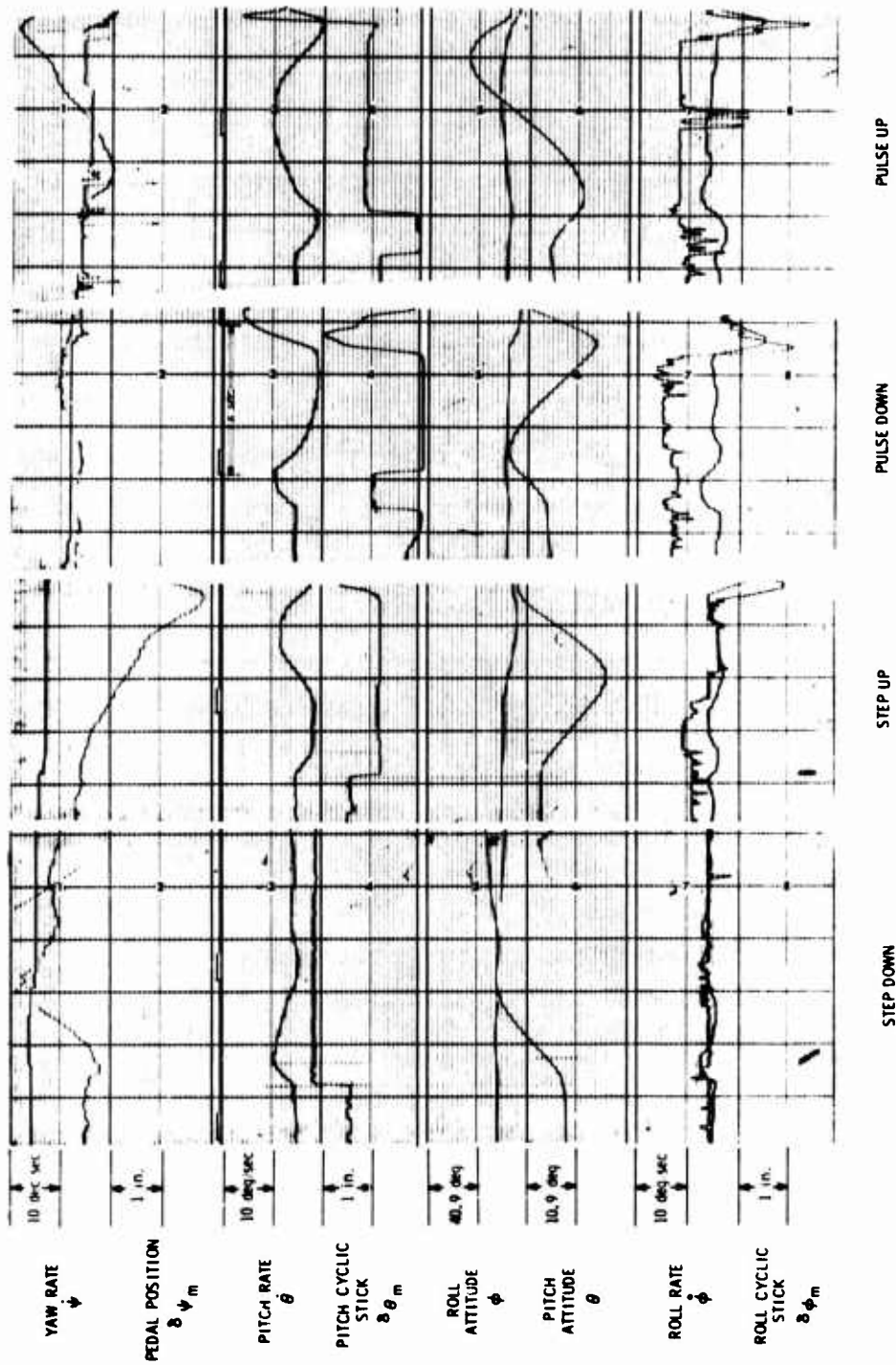


Figure 100. Aircraft Response to Pitch Cyclic Inputs (Hover, 3000 ft) With Stabilizer Bar

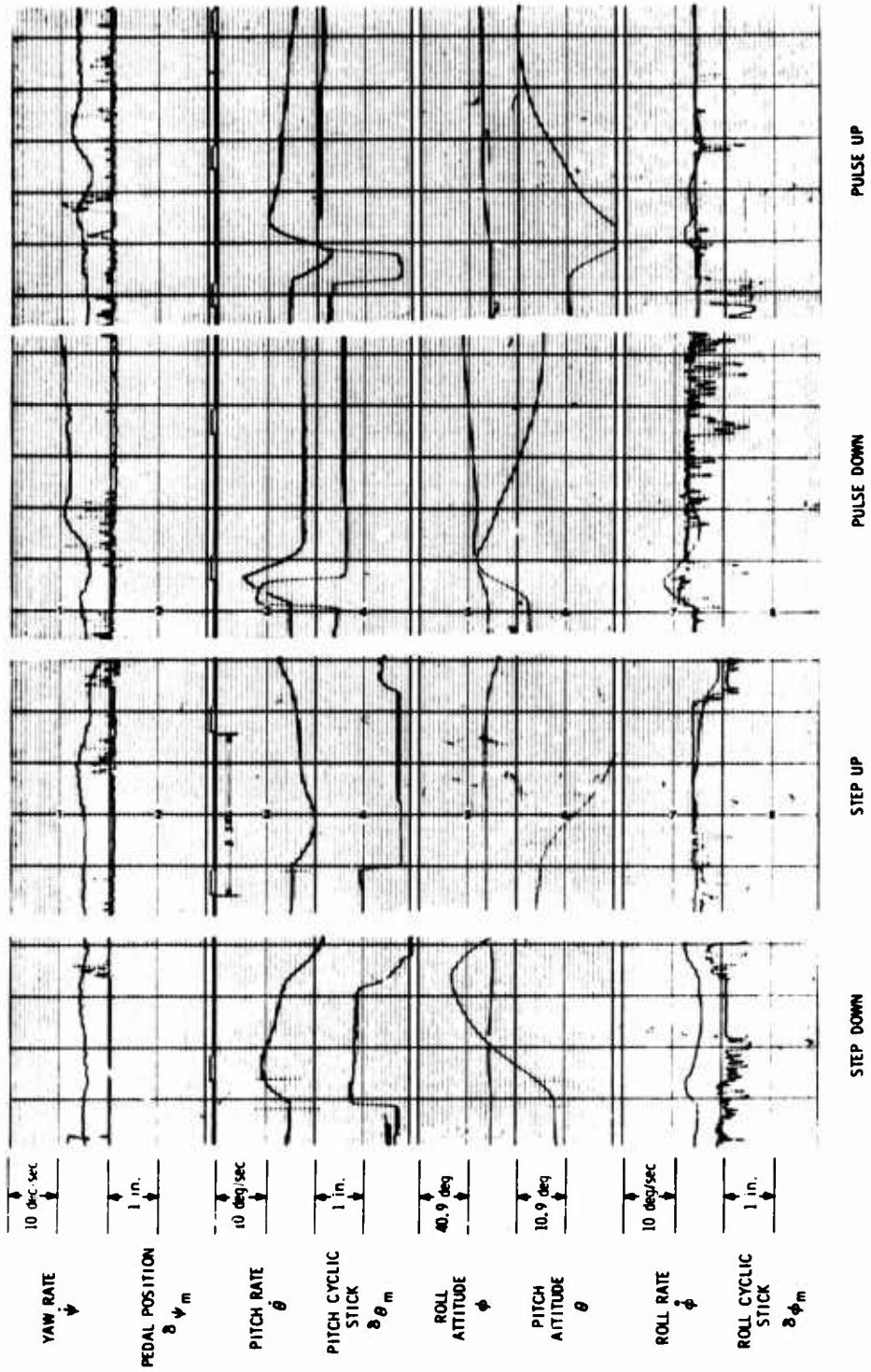


Figure 101. Aircraft Response to Pitch Cyclic Inputs (60 kn, 3000 ft) With Stabilizer Bar

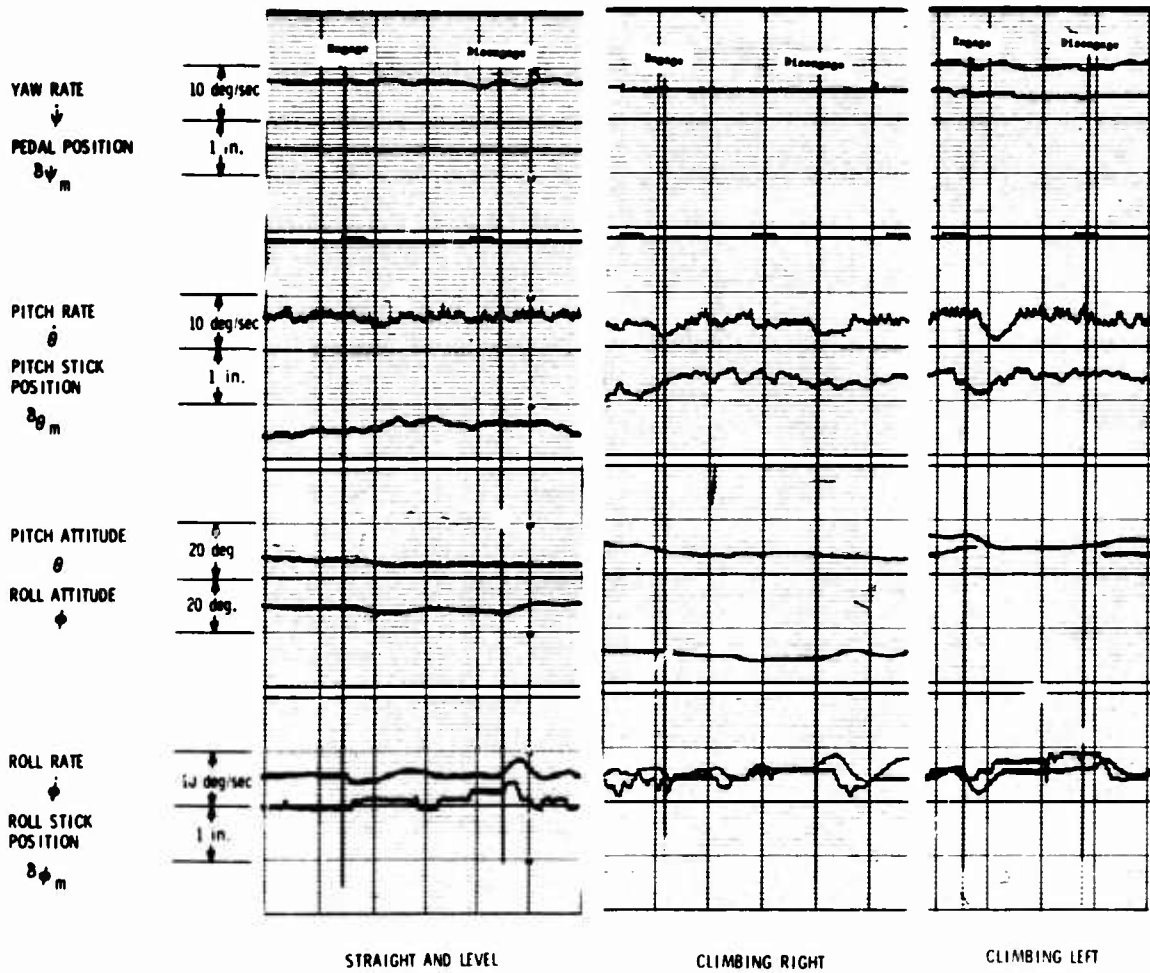


Figure 102. Aircraft Response to SAS Engage/Disengage Transients (60 kn, 3000 ft)

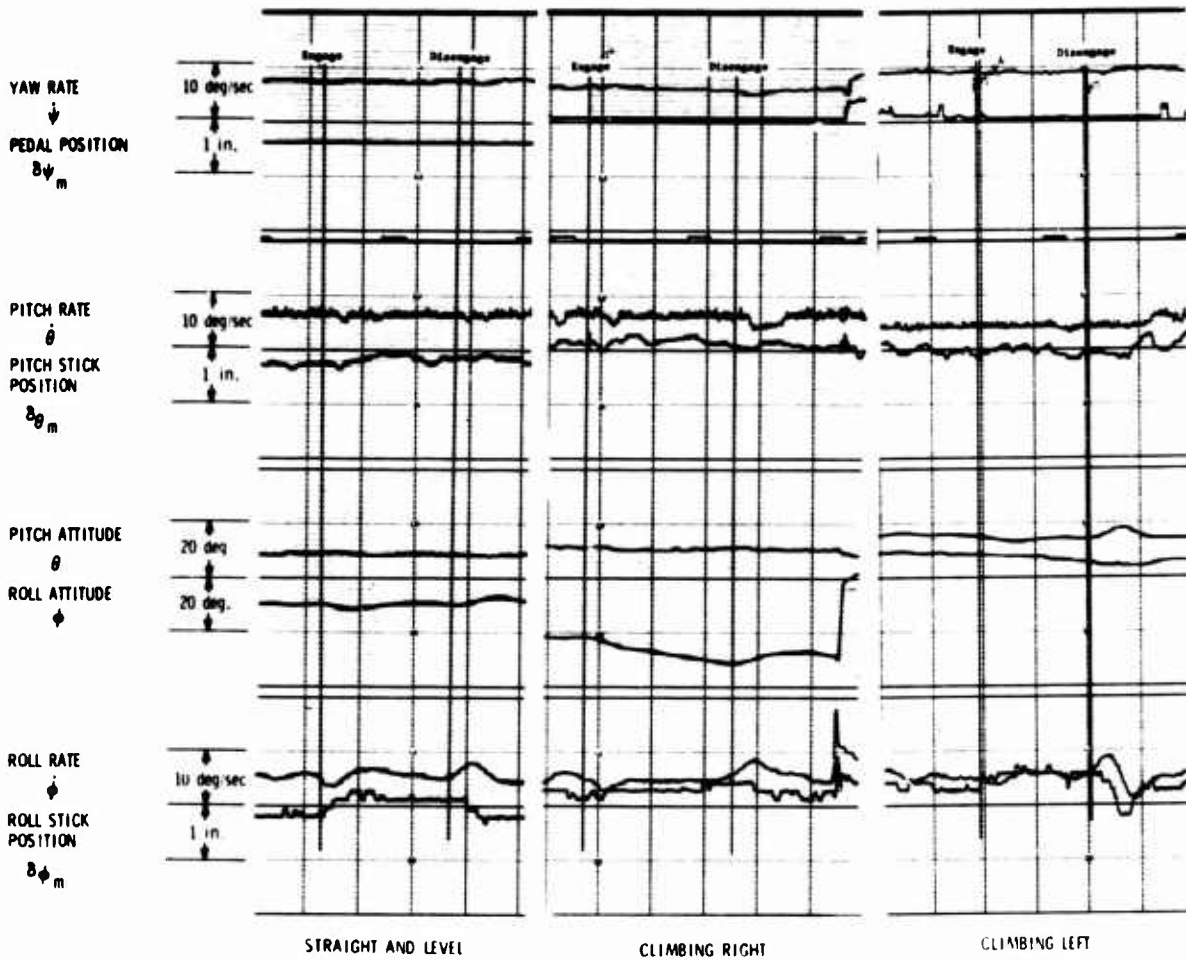


Figure 103. Aircraft Response to SAS Engage/Disengage Transients (90 kn, 3000 ft)

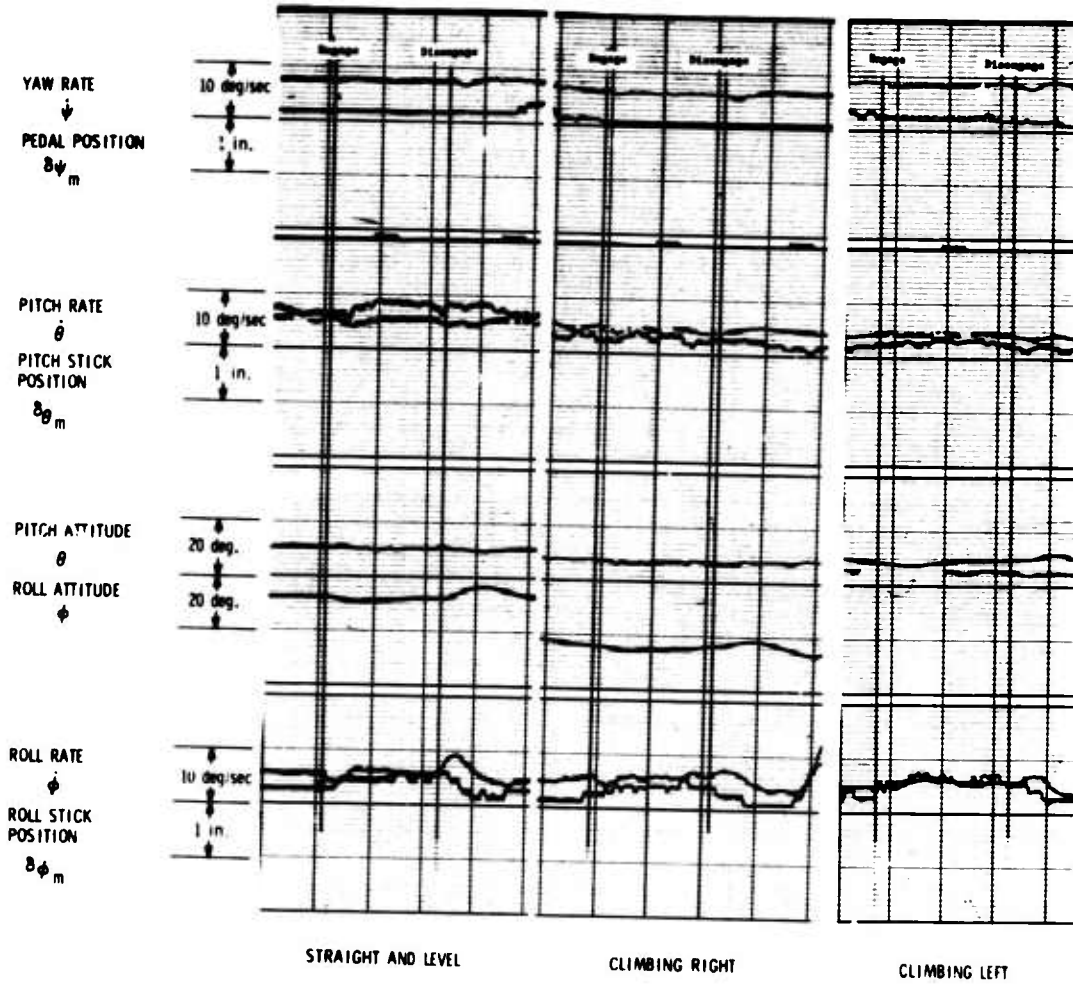


Figure 104. Aircraft Response to SAS Engage/Disengage Transients (120 kn, 3000 ft)

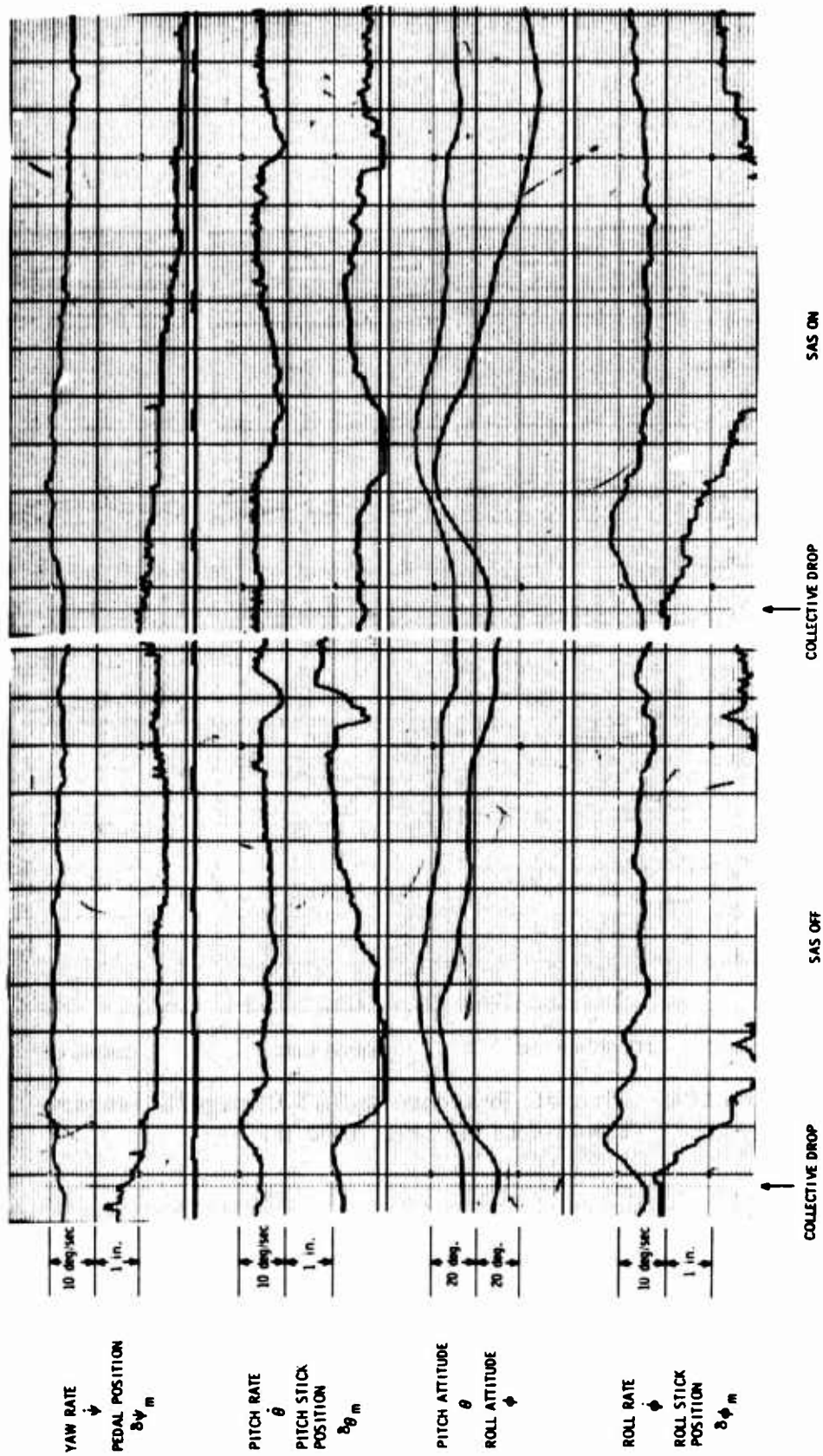


Figure 105. Aircraft Response to Autorotation (60 kn, 3000 ft)

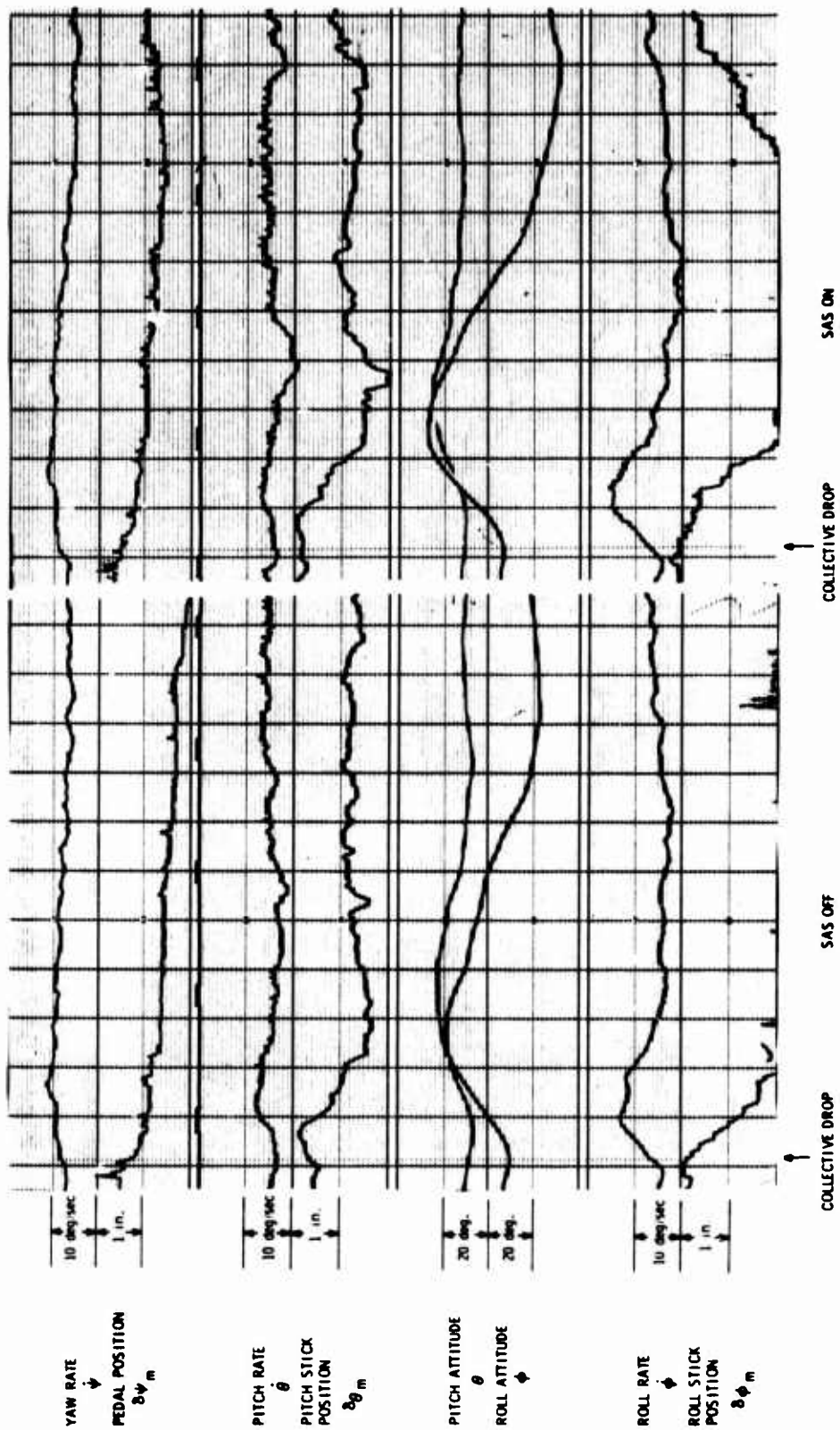


Figure 106. Aircraft Response to Autorotation (90 kn, 3000 ft)

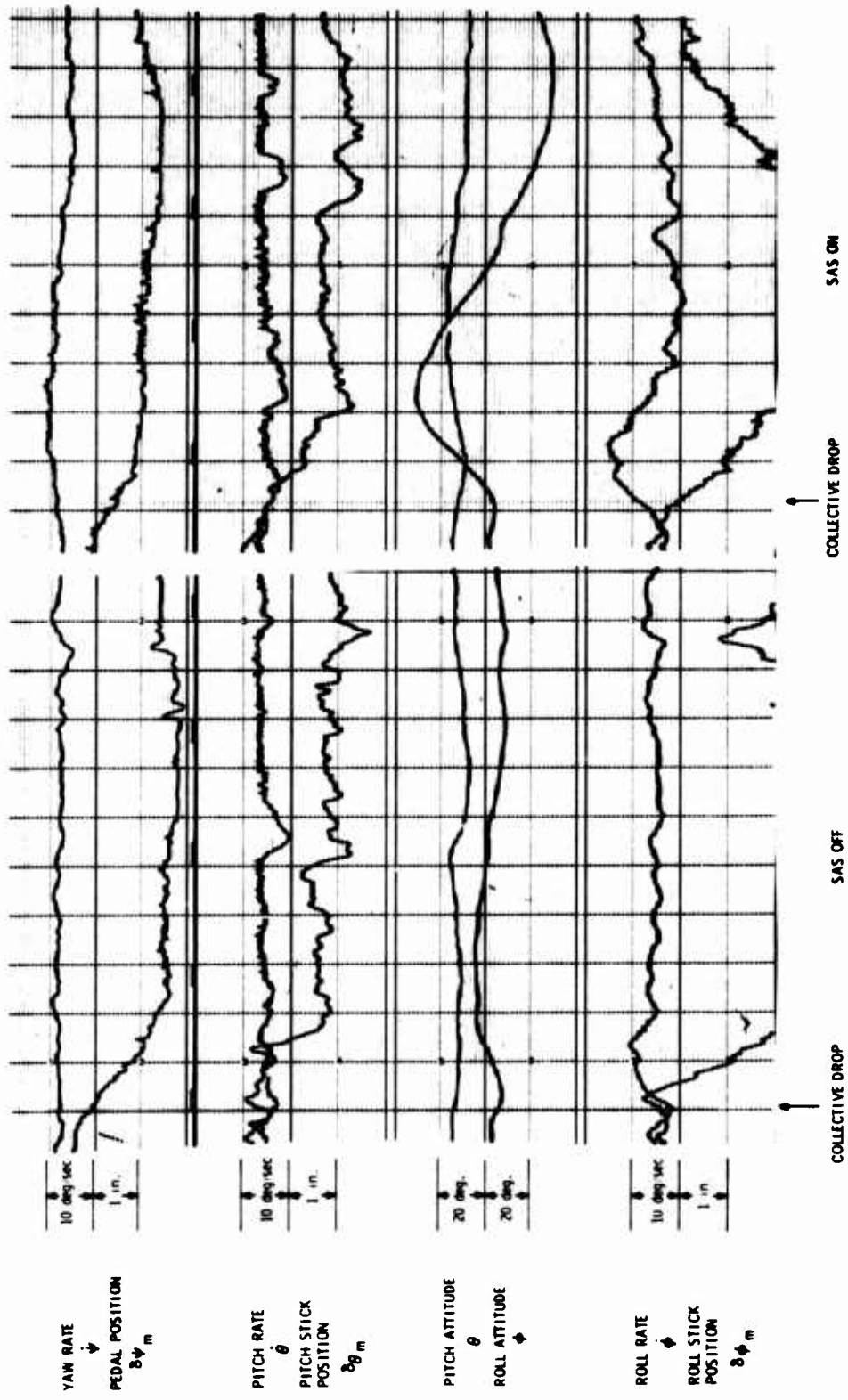


Figure 107. Aircraft Response to Autorotation (120 kn, 3000 ft)

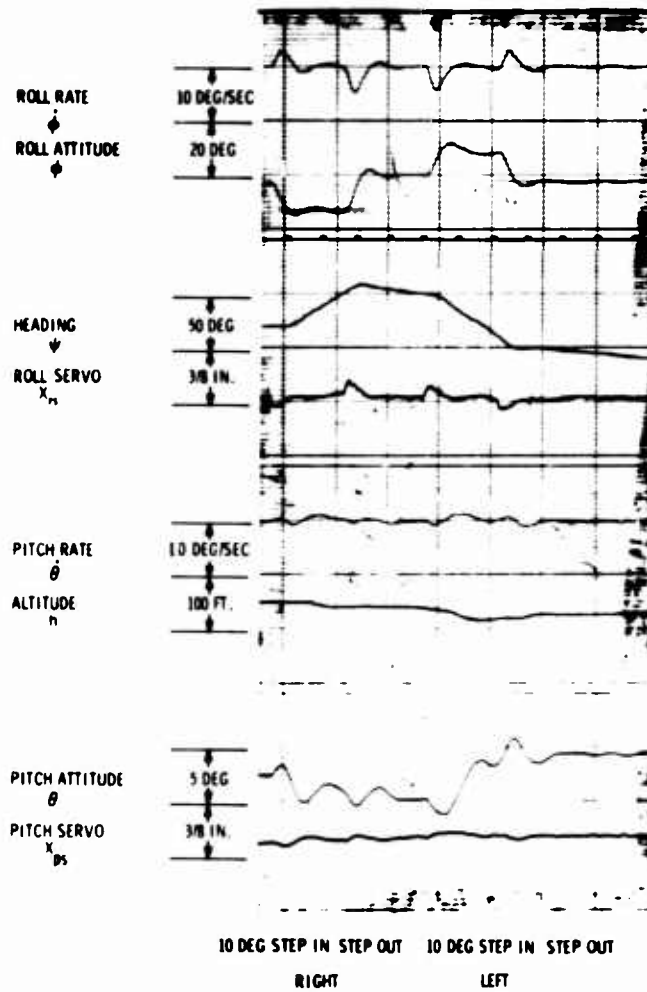


Figure 108. Aircraft Response to Roll Attitude Steps (60 kn, 3000 ft)

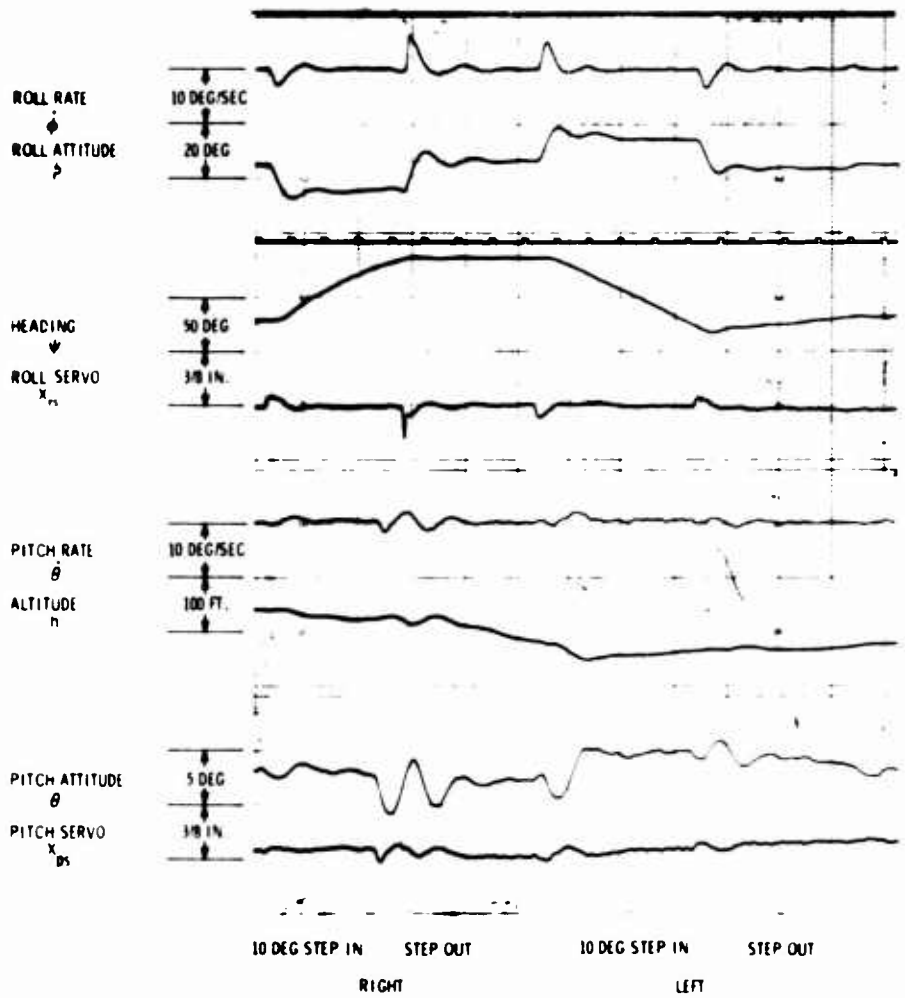


Figure 109. Aircraft Response to Roll Attitude Steps (60 kn, 5000 ft)

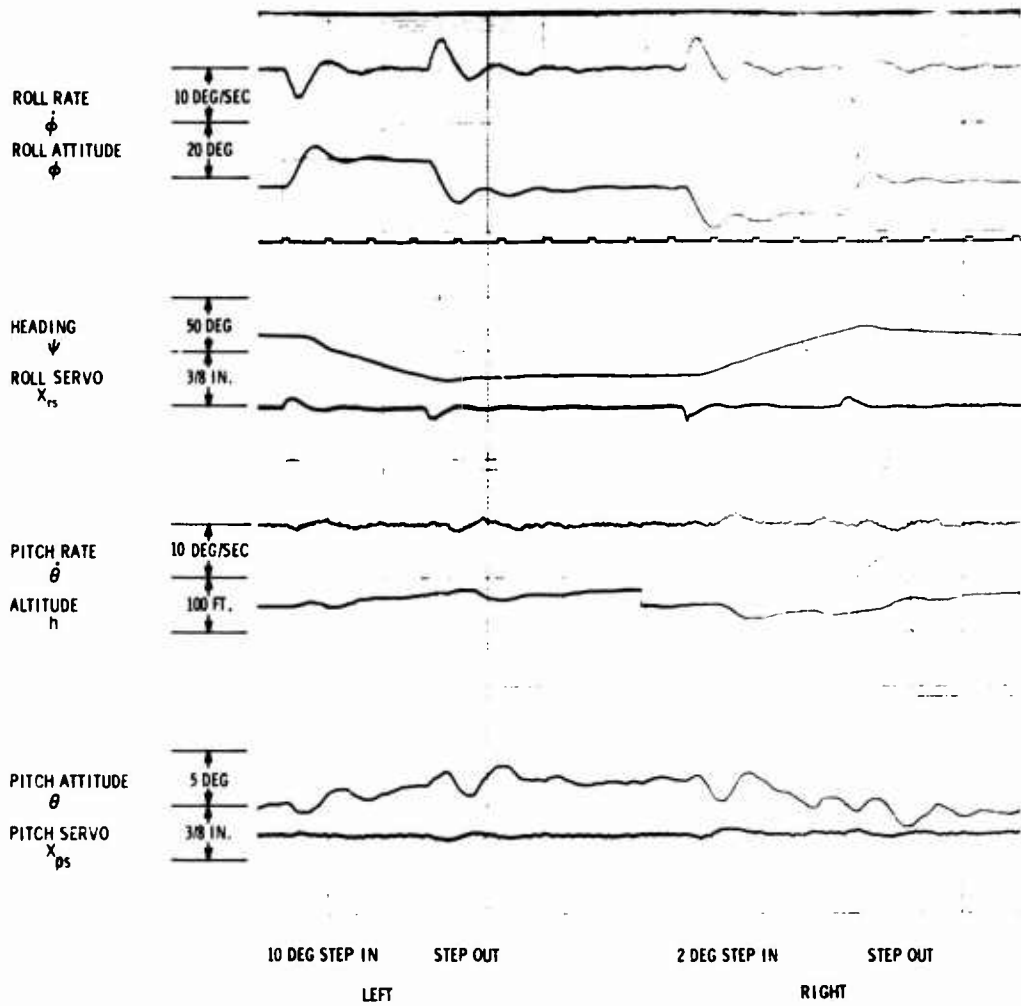


Figure 111. Aircraft Response to Roll Attitude Steps (120 kn, 3000 ft)

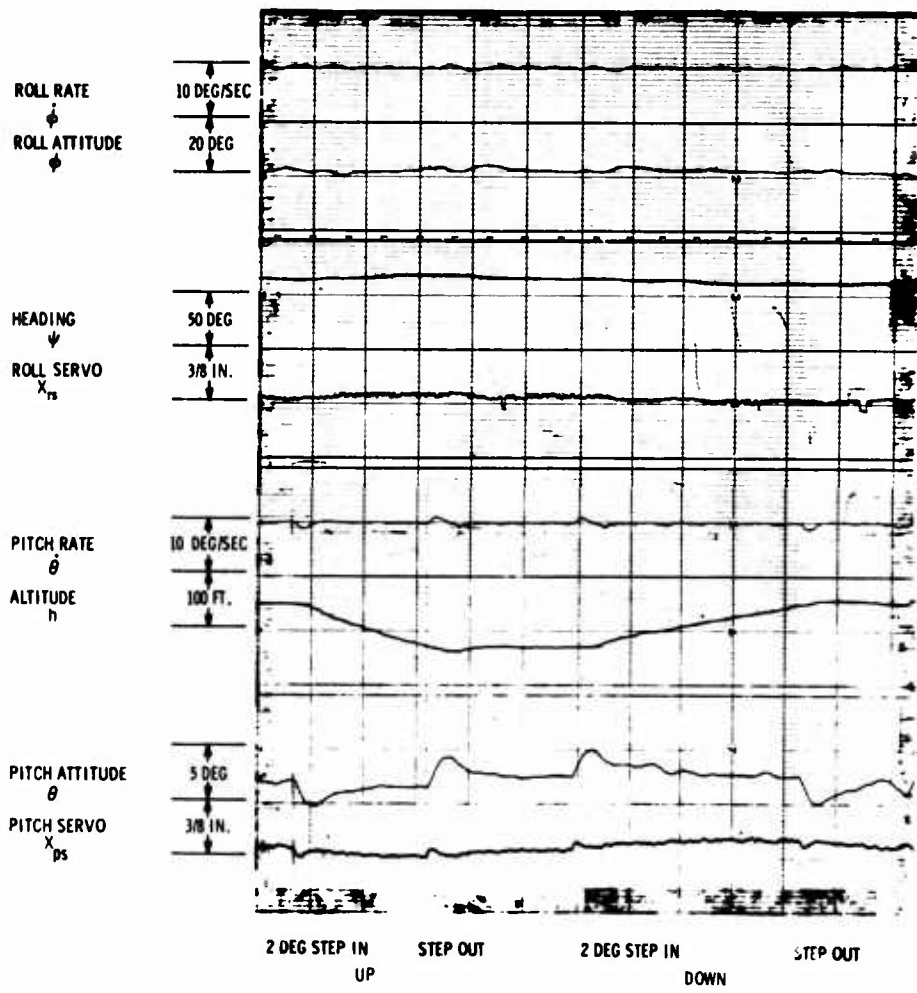


Figure 112. Aircraft Response to Pitch Attitude Steps (60 kn, 3000 ft)

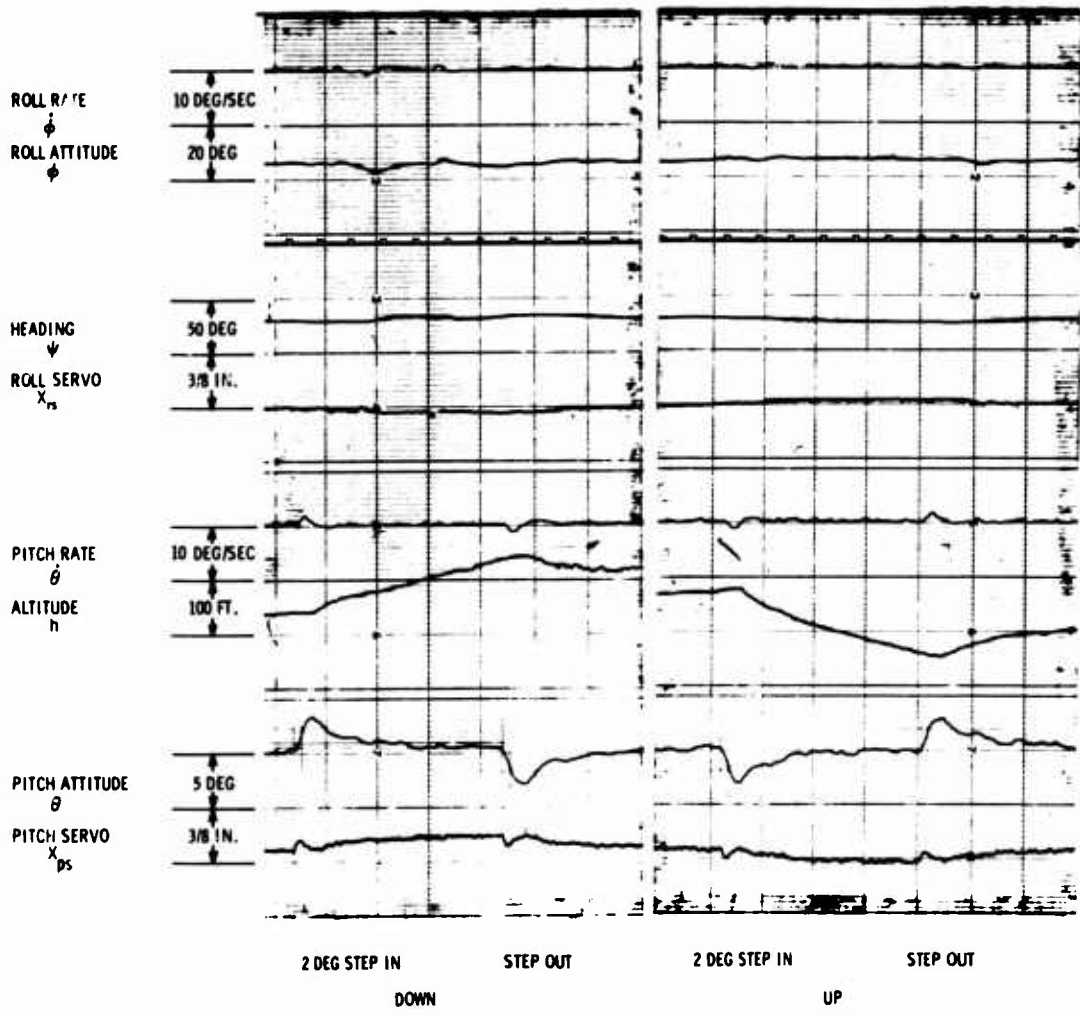


Figure 113. Aircraft Response to Pitch Attitude Steps (60 kn, 5000 ft)

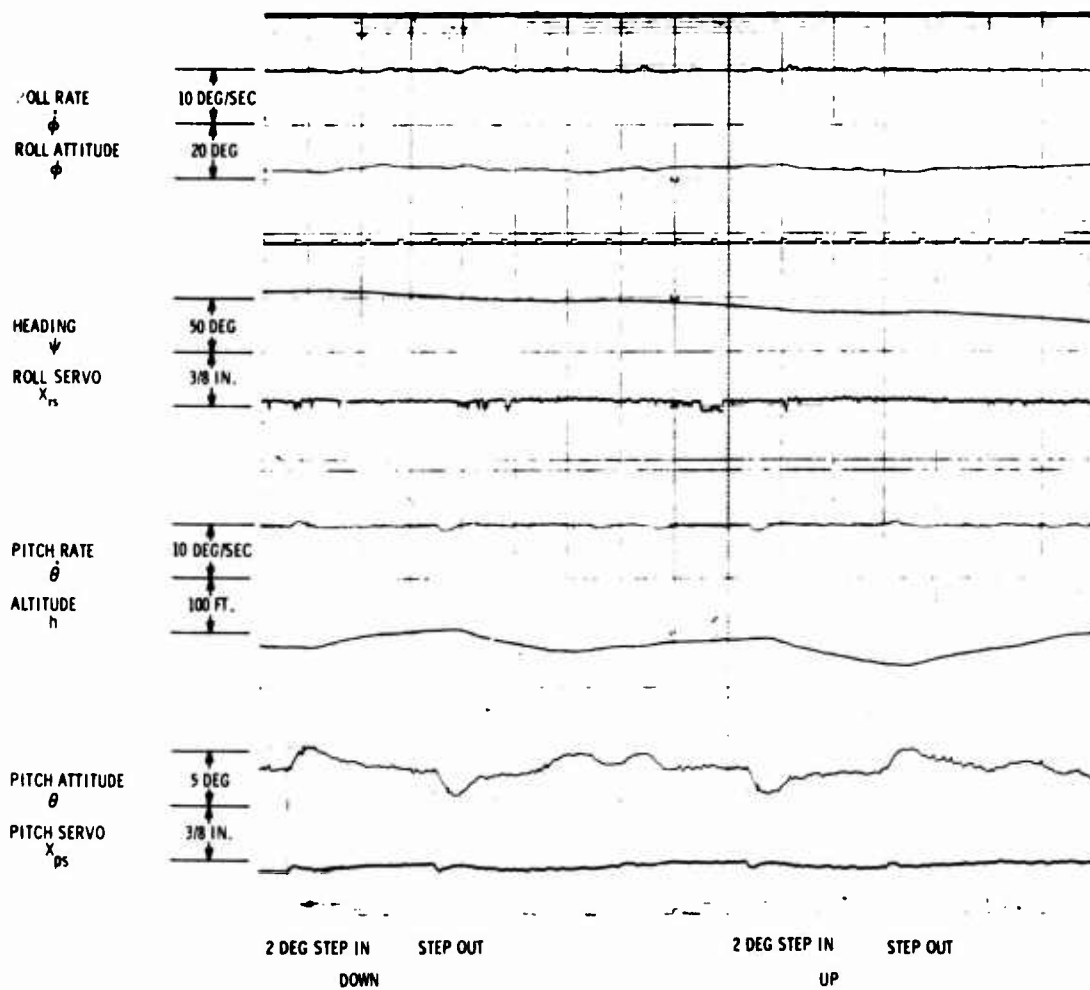


Figure 114. Aircraft Response to Pitch Attitude Steps (60 kn, 10,000 ft)

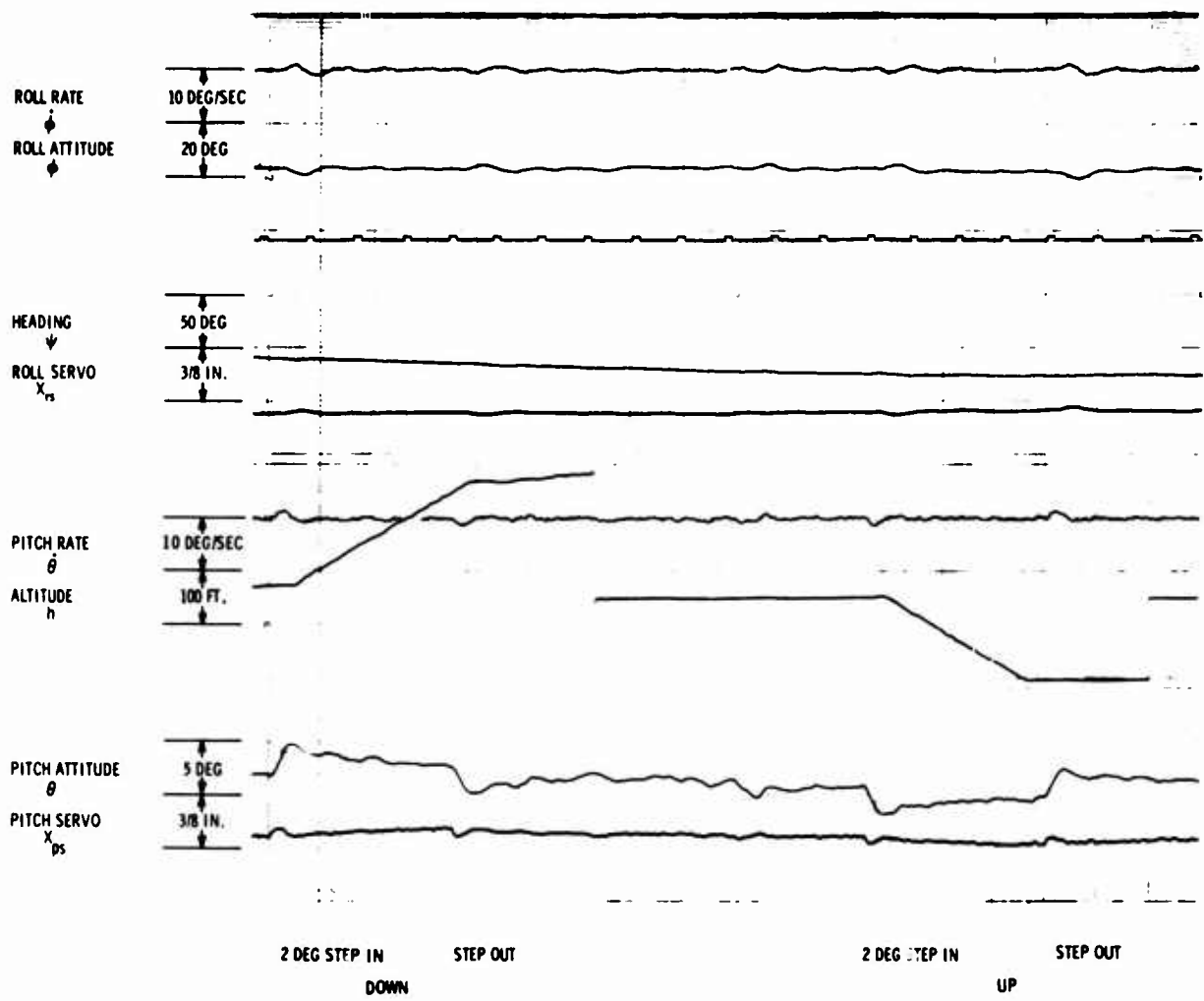


Figure 115. Aircraft Response to Pitch Attitude Steps (120 kn, 3000 ft)

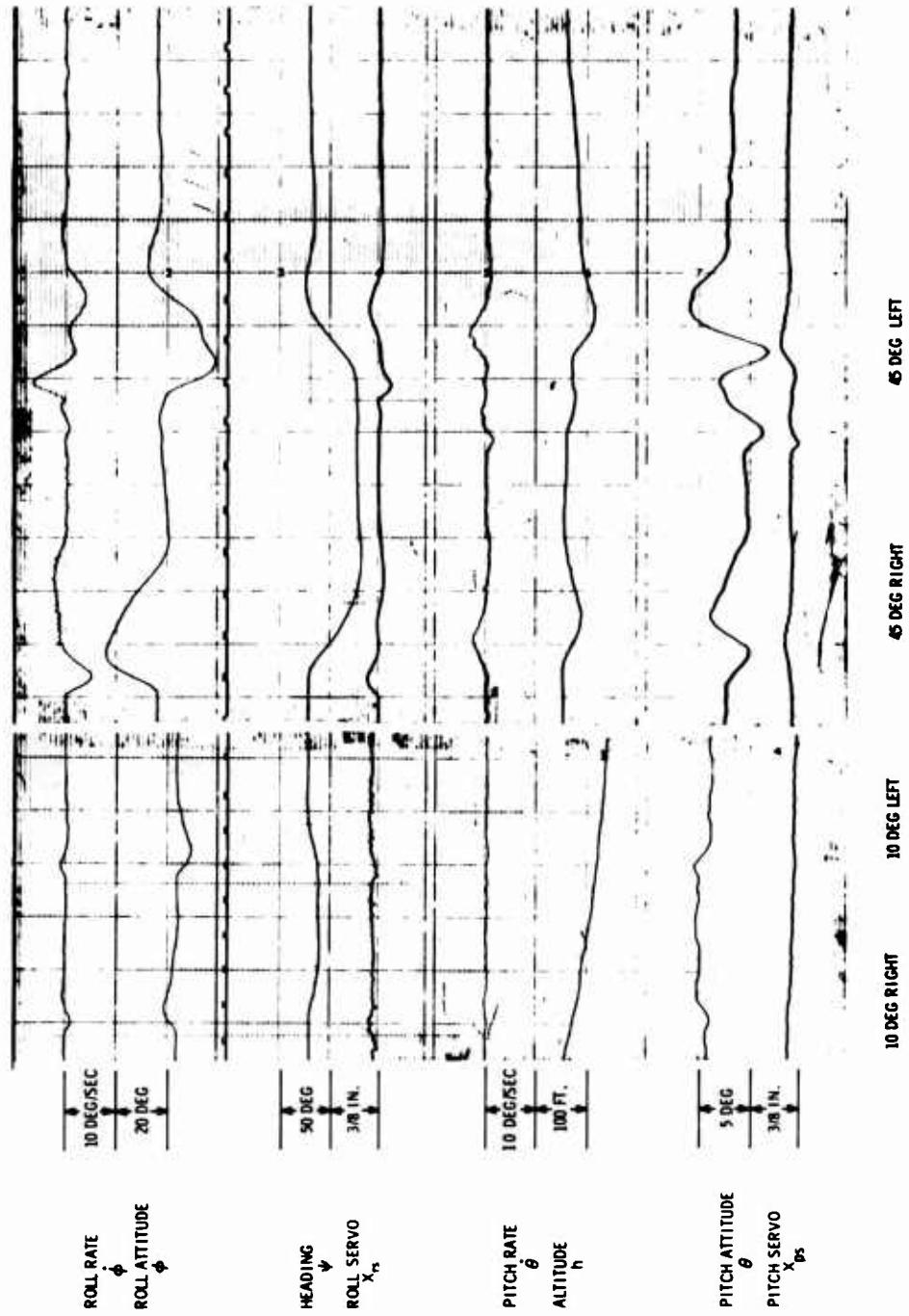


Figure 116. Aircraft Response to Heading Steps (60 kn, 3000 ft)

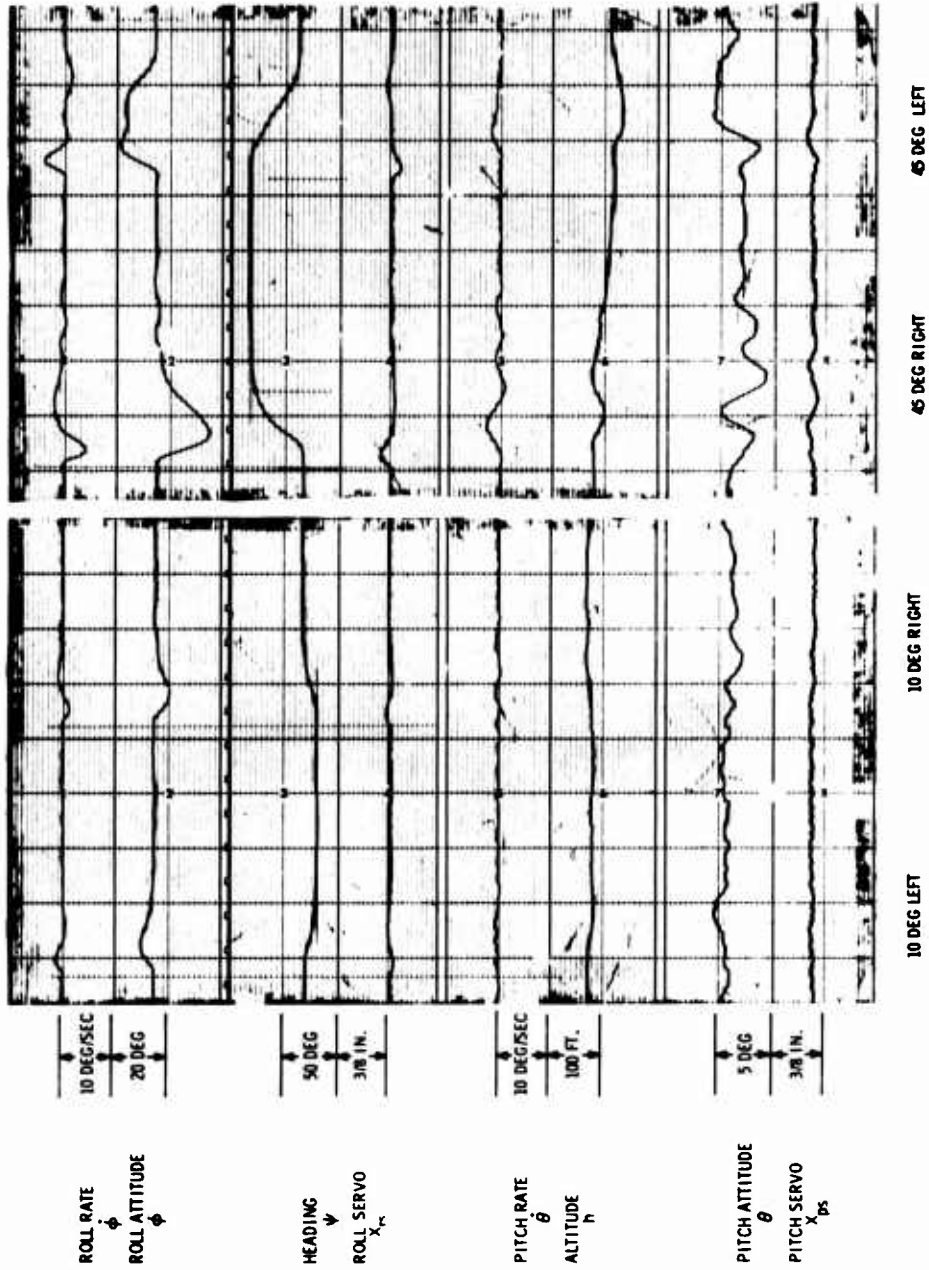


Figure 117. Aircraft Response to Heading Steps (60 kn, 5000 ft)

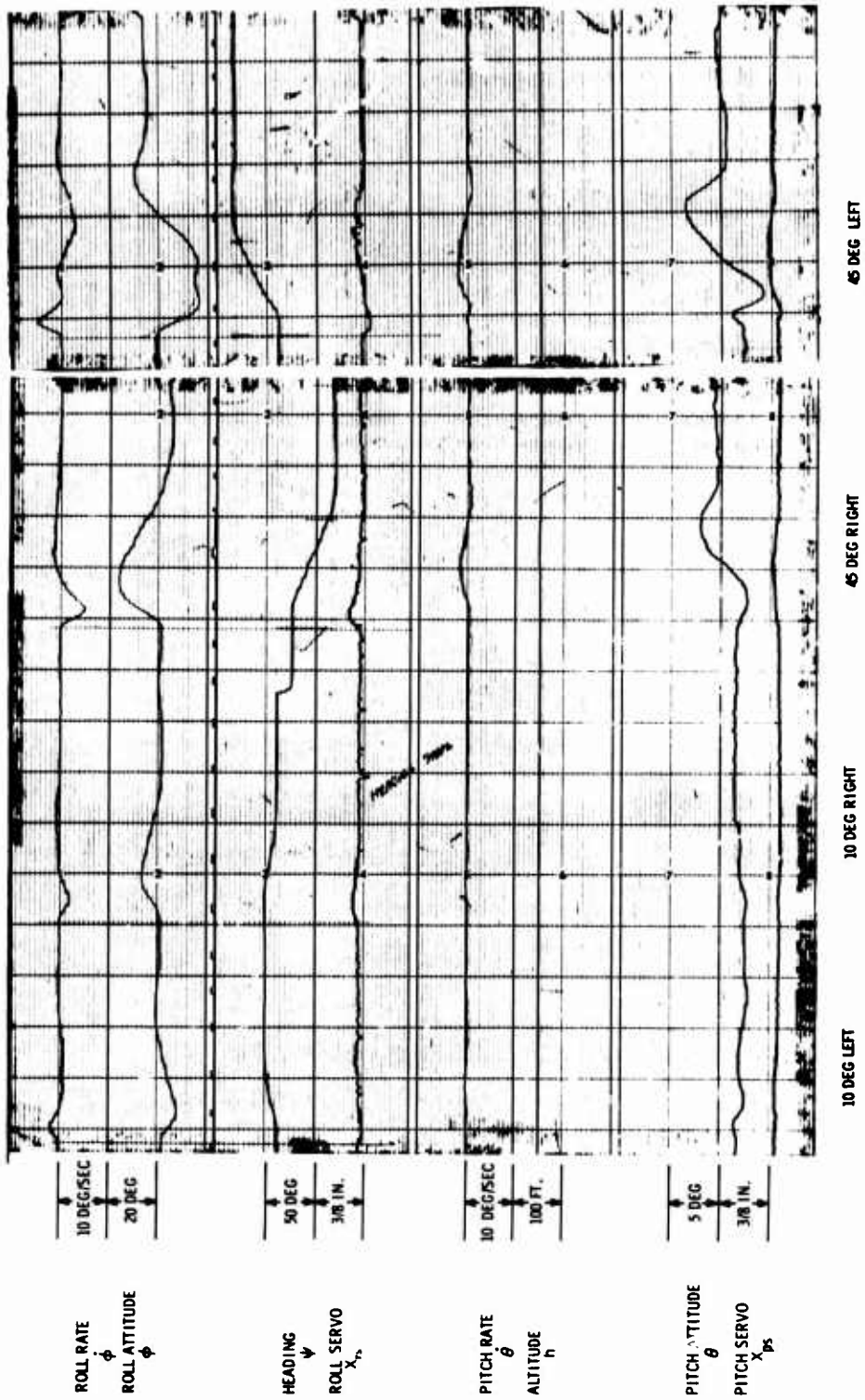


Figure 118. Aircraft Response to Heading Steps (60 kn, 10,000 ft)

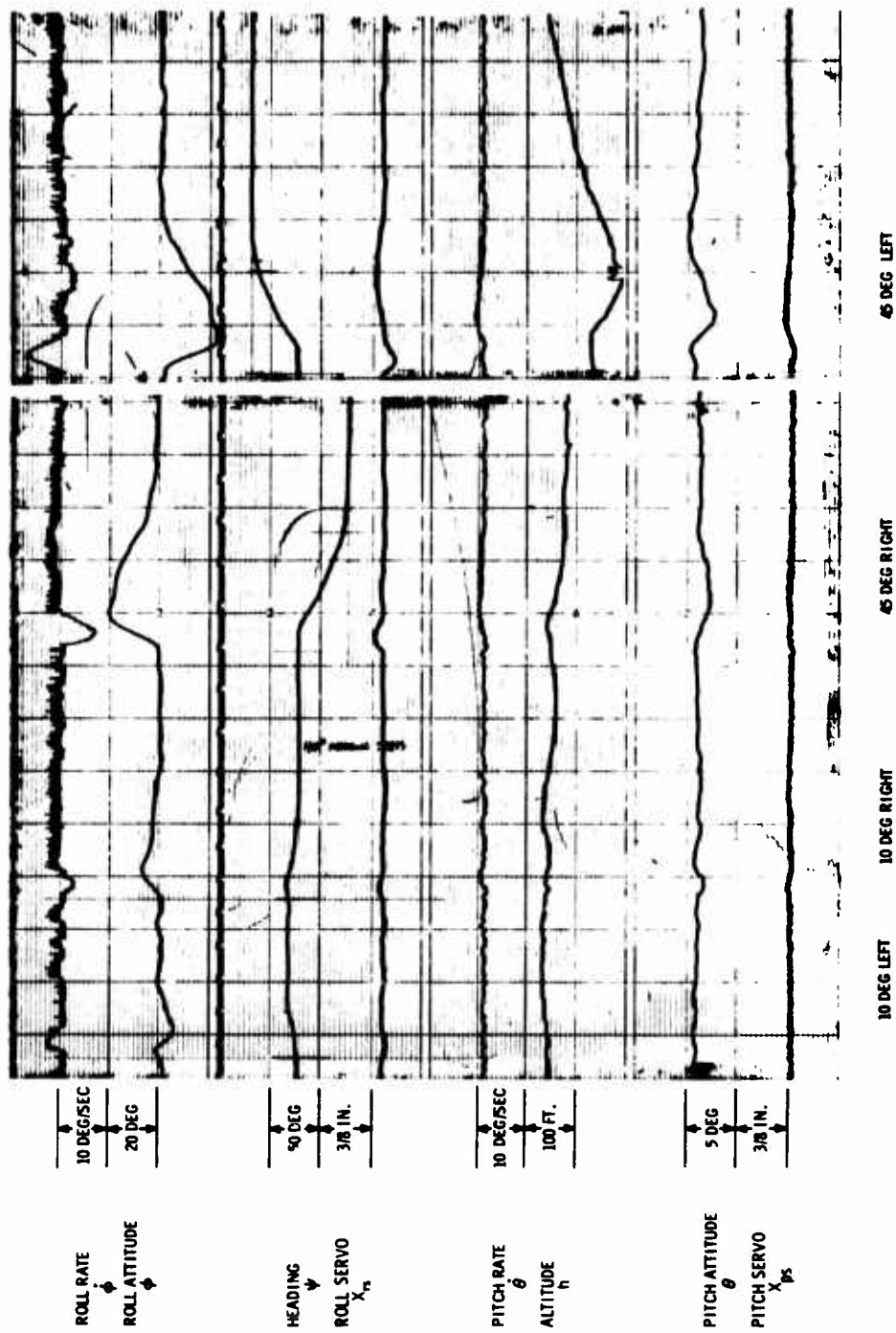


Figure 119. Aircraft Response to Heading Steps (120 kn, 3000 ft)

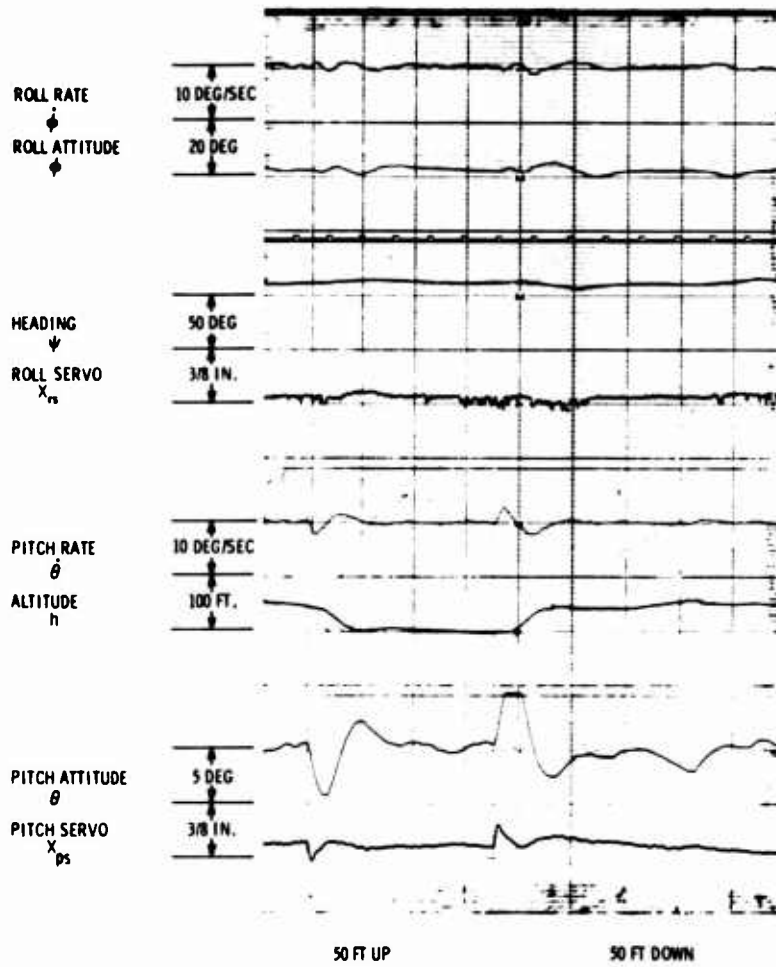


Figure 120. Aircraft Response to Altitude Steps (60 kn, 3000 ft)

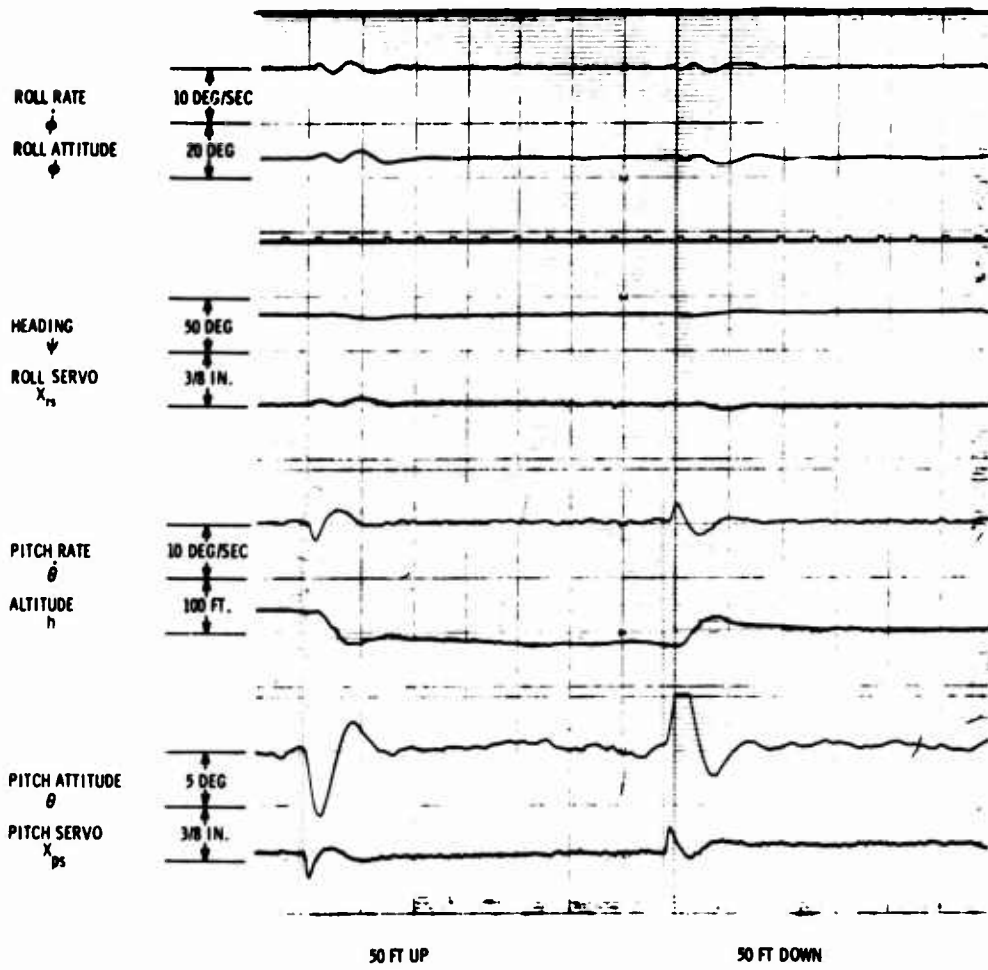


Figure 121. Aircraft Response to Altitude Steps (60 kn, 5000 ft)

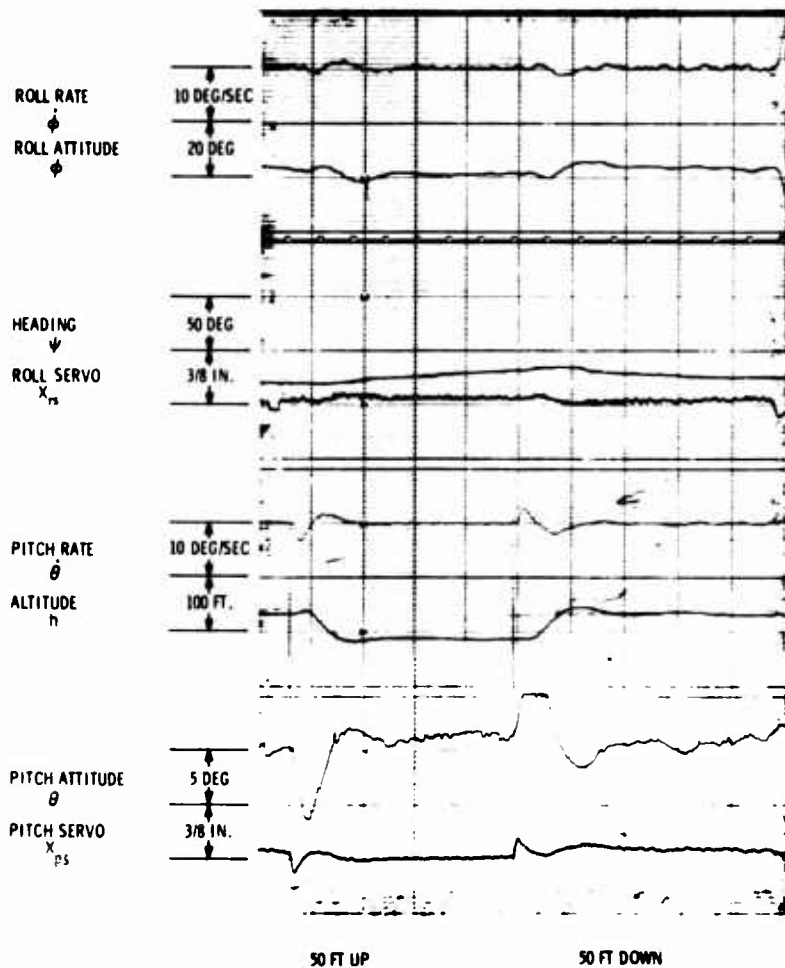


Figure 122. Aircraft Response to Altitude Steps (60 kn, 10,000 ft)

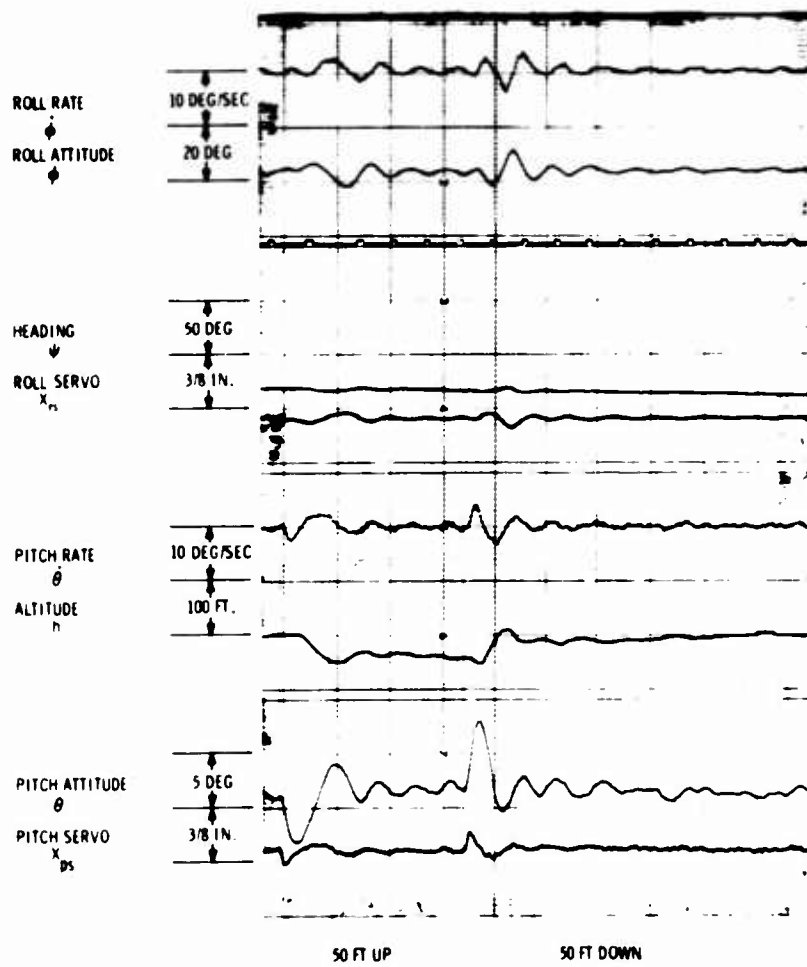


Figure 123. Aircraft Response to Altitude Steps (120 kn, 3000 ft)

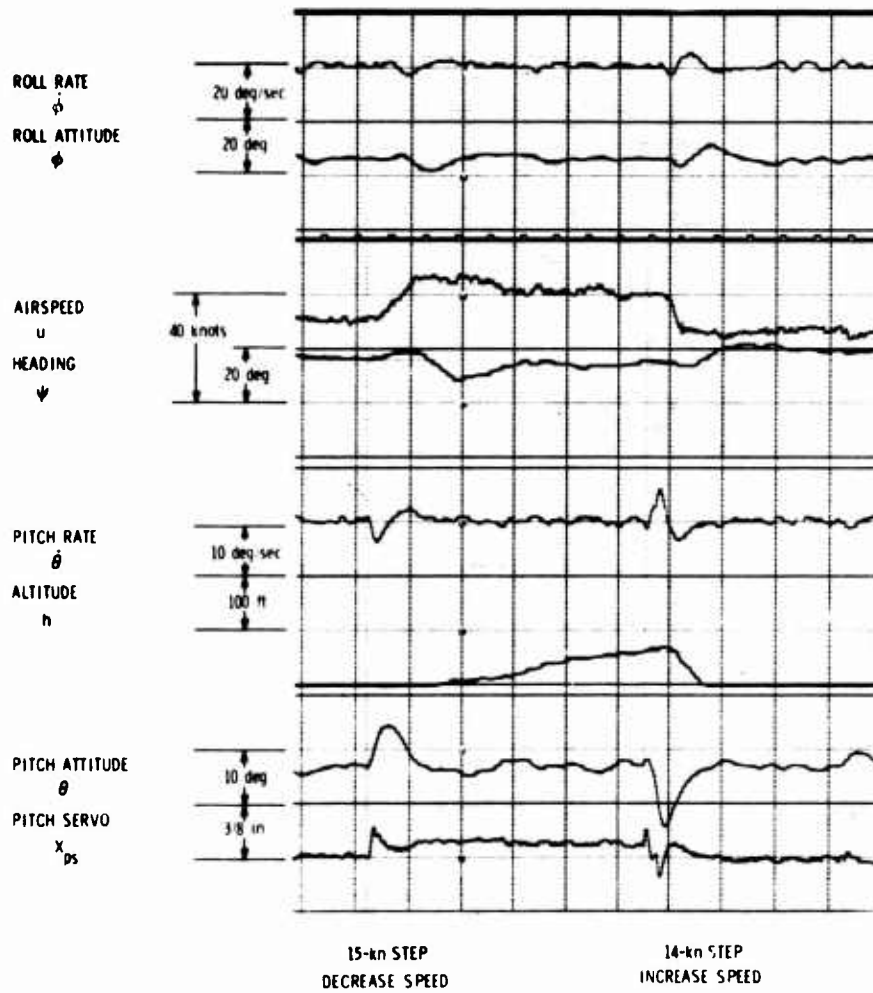


Figure 124. Aircraft Response to Airspeed Steps (60 kn, 3000 ft)

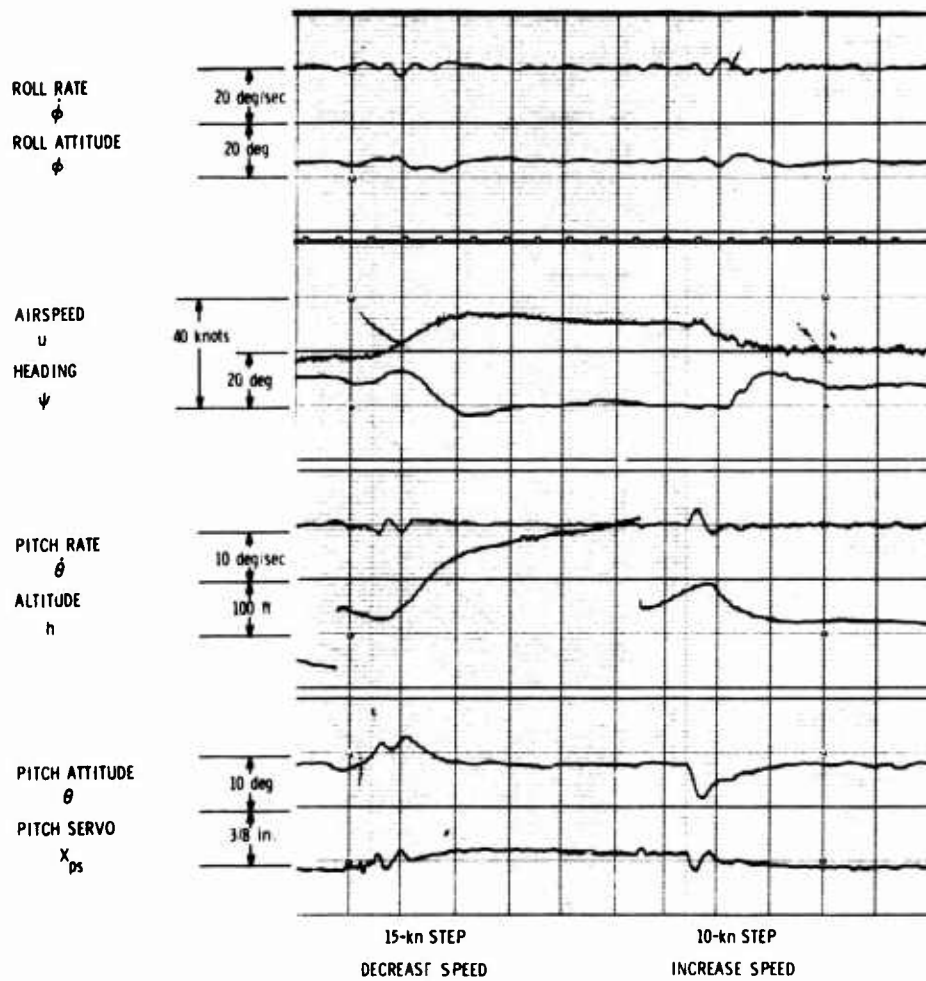


Figure 125. Aircraft Response to Airspeed Steps (60 kn, 5000 ft)

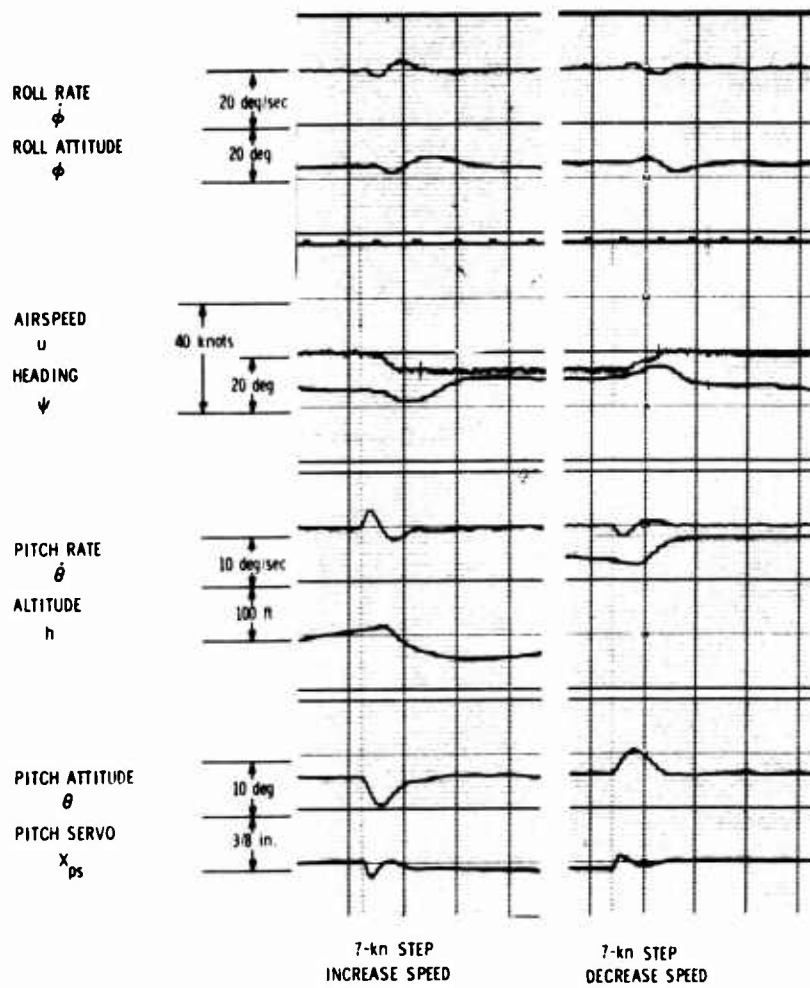


Figure 126. Aircraft Response to Airspeed Steps (60 kn, 10,000 ft)

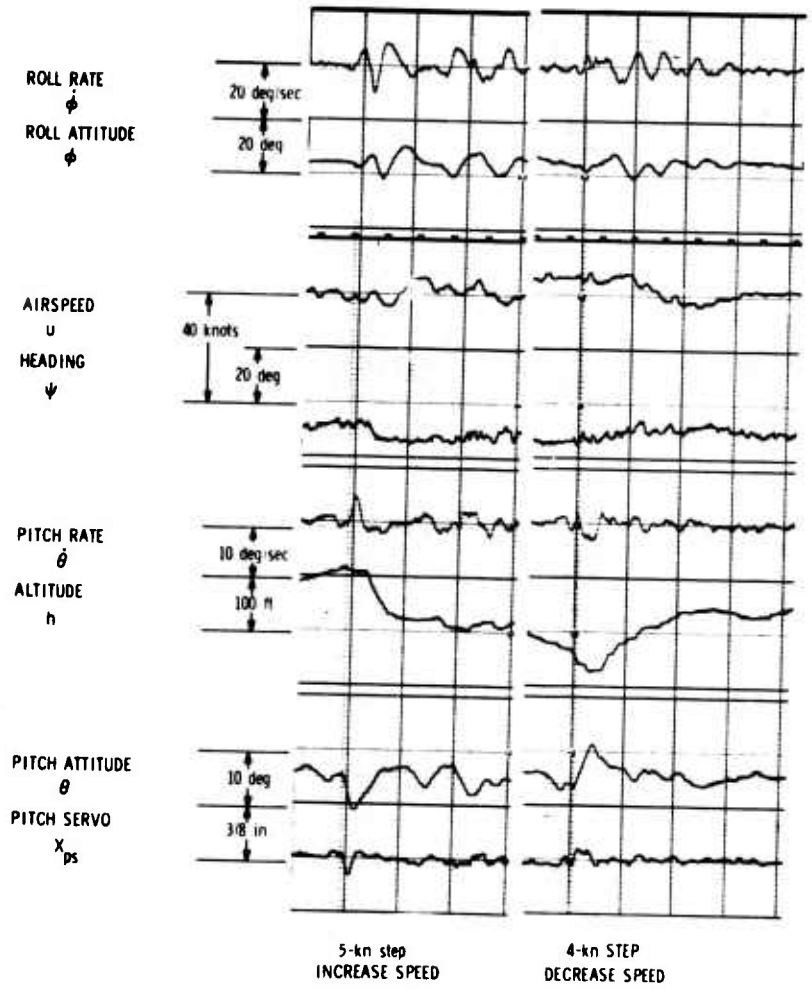


Figure 127. Aircraft Response to Airspeed Steps (120 kn, 3000 ft)

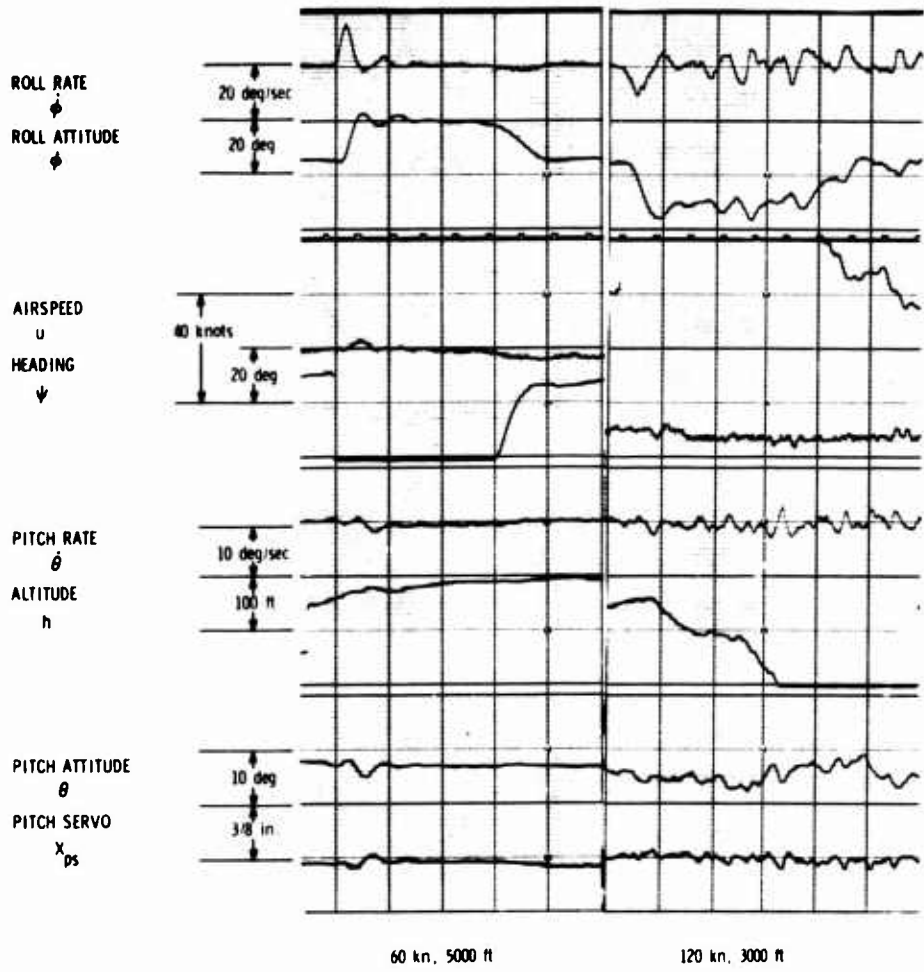


Figure 128. Aircraft Response to Airspeed Hold in Turns

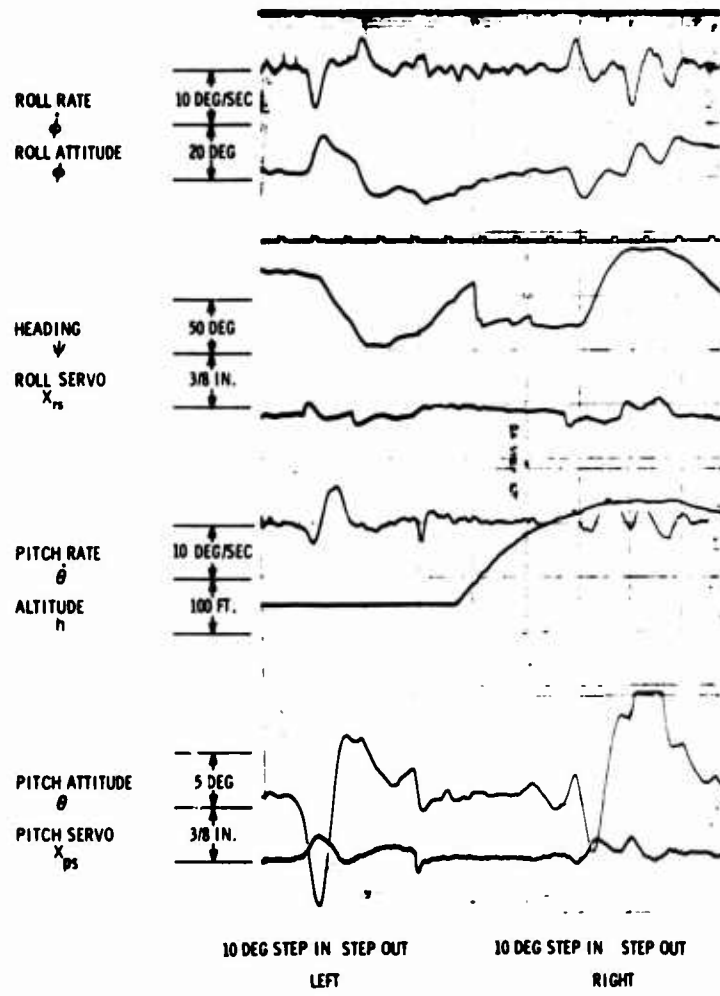


Figure 129. Aircraft Response to Roll Attitude Steps (Hover, 3000 ft)

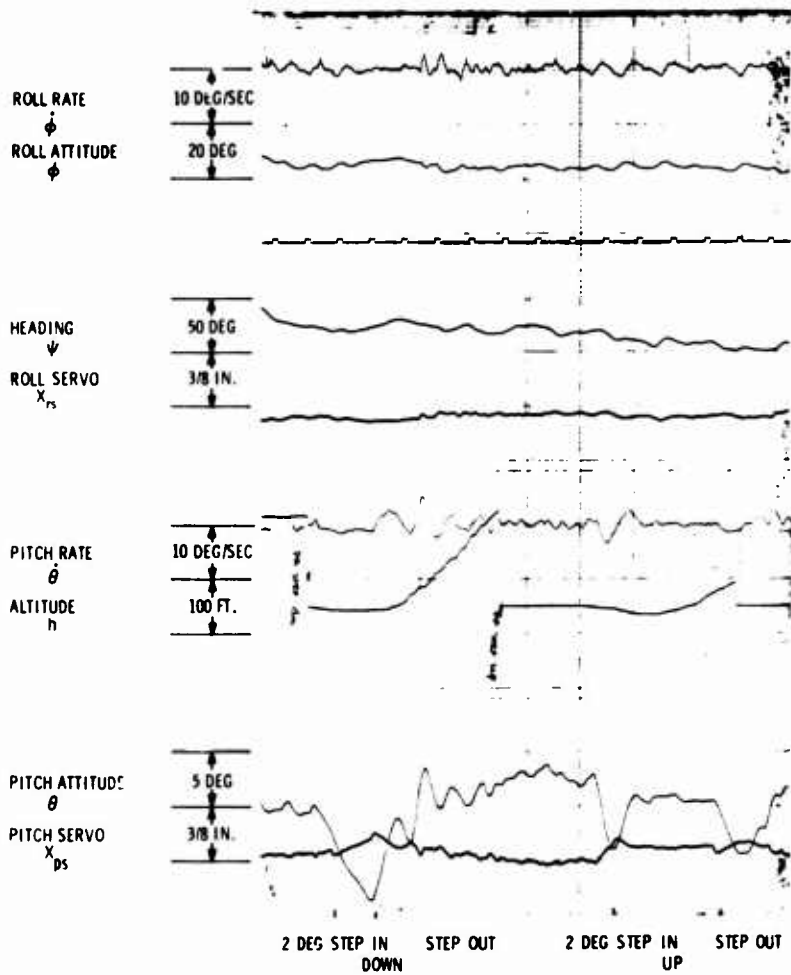


Figure 130. Aircraft Response to Pitch Attitude Steps (Hover, 3000 ft)

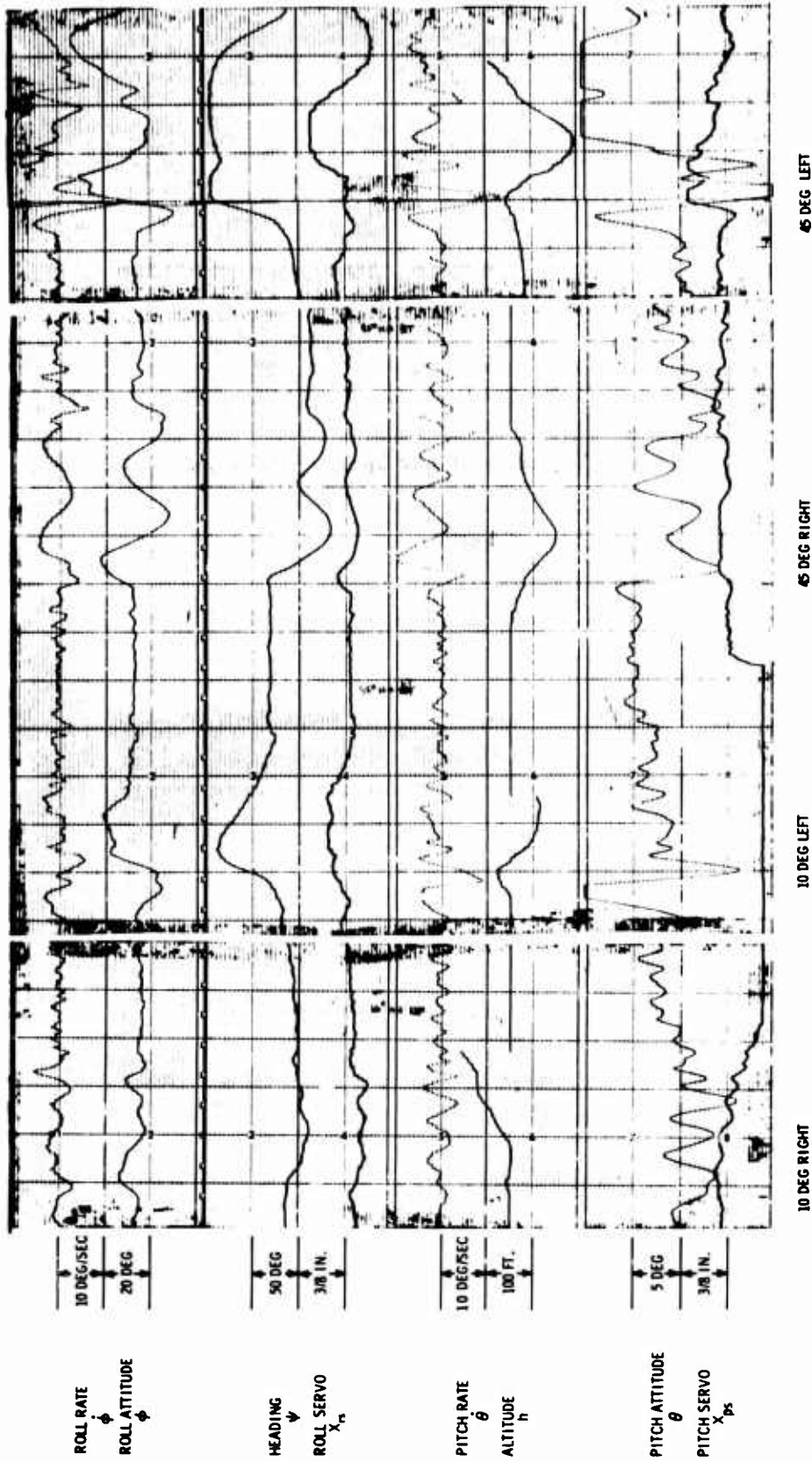


Figure 131. Aircraft Response to Heading Steps (Hover, 3000 ft)

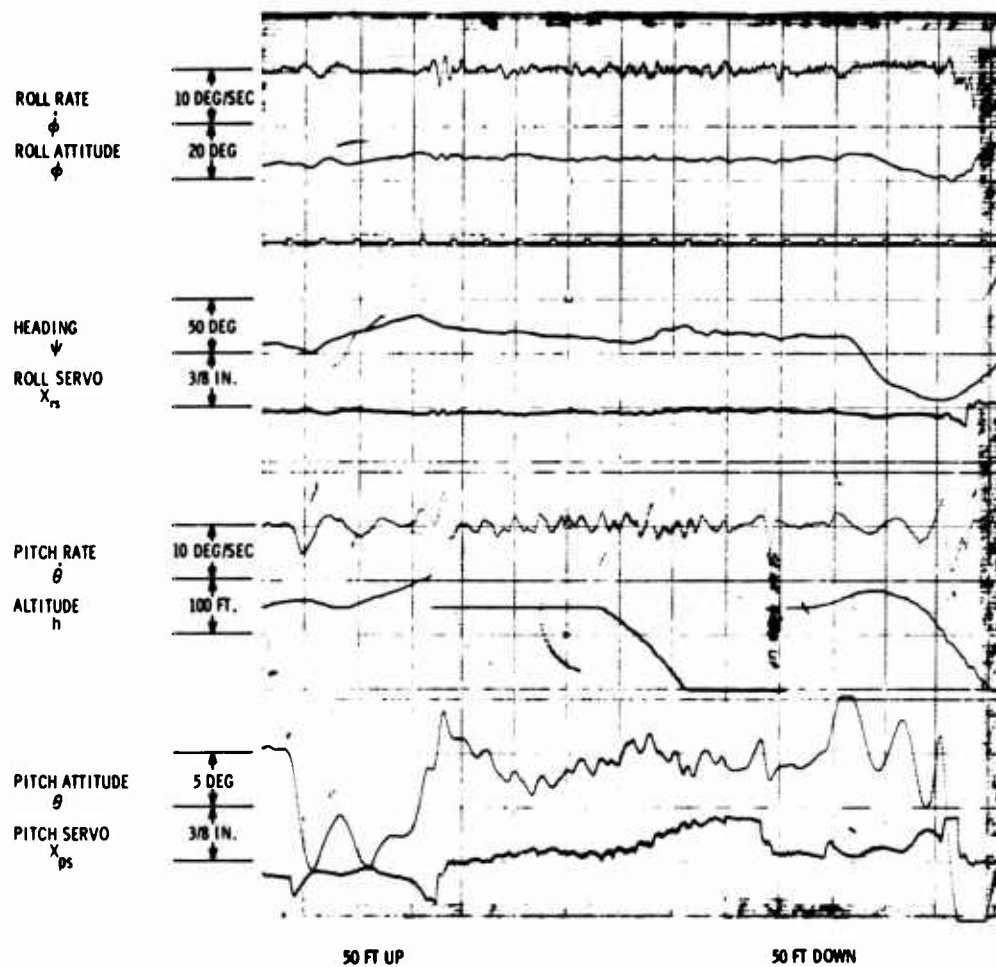


Figure 132. Aircraft Response to Altitude Steps (Hover, 3000 ft)

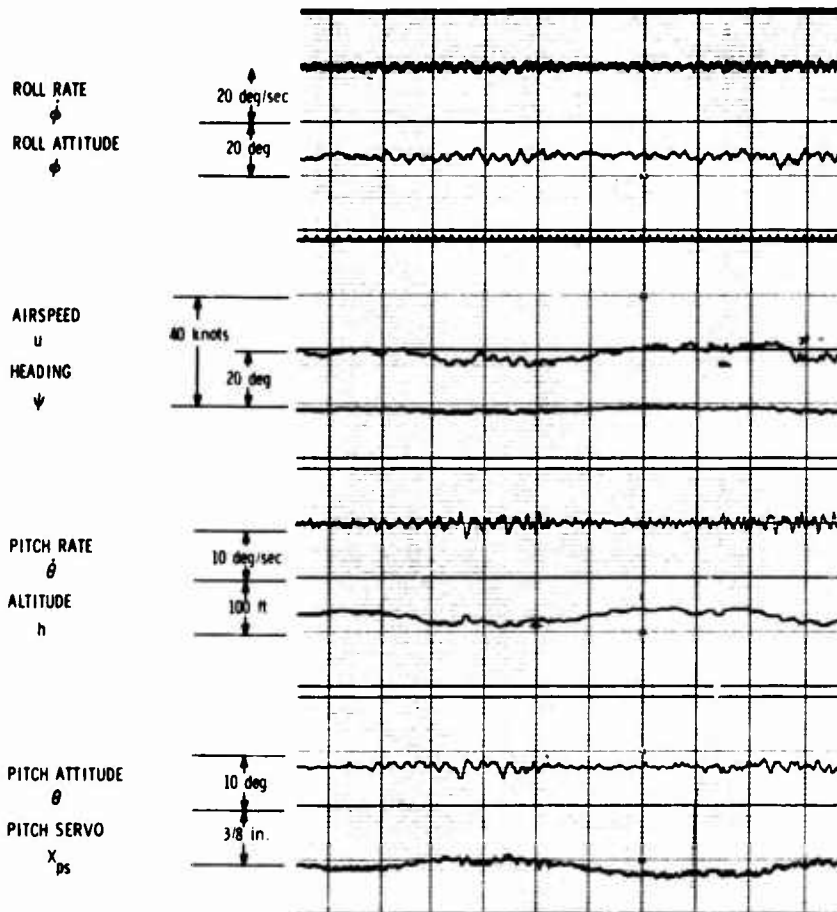


Figure 133. Aircraft Performance During Straight and Level Flight (60 kn, 5000 ft) Heading Hold and Altitude Hold Engaged

LIST OF SYMBOLS

A_{is}	roll cyclic ram displacement, deg
B_{is}	pitch cyclic ram displacement, deg
BIT	built-in test
q	dynamic pressure, lb/ft ²
e	2.7314
E/F	electric to fluidic
FSAS	fluidic stability augmentation system
F/E	fluidic to electric
FET	field effect transistor
h	altitude, ft
h_e	altitude error, ft
K	gain
K_ϕ	roll attitude gain, rad/rad
$K_{\phi\psi}$	bank to heading gain, rad/rad
LVDT	linear variable differential transformer
ma	milliampere
o	used as subscript, means initial steady-state conditions
RSS	root sum of squares
S	Laplace operator
SAS	stability augmentation system
$t_{90\%}$	time to reach 90 percent of final value, sec
T	time constant, sec

t_s	solution time, sec
u	airspeed, knots
W	maximum overload gross weight of helicopter, lb
X_{cs}	pitch axis mechanical cyclic displacement, in.
X_p	yaw axis mechanical pedal displacement, in.
X_{ps}	pitch series servoactuator position, in.
X_{rs}	roll series servoactuator position, in.
Y_{cs}	roll axis mechanical cyclic displacement, in.
ζ_β	boost actuator damping ratio
ζ_s	series servoactuator damping
θ	pitch angle, deg
θ_c	commanded pitch angle, deg
$\dot{\theta}$	pitch angular rate, deg/sec
θ_e	pitch angle error, deg
$\delta_{\theta m}$	pitch cyclic stick, in.
θ_{TR}	tail rotor ram displacement, deg
τ	vortex rate sensor transport delay, sec
ϕ	roll angle, deg
ϕ_c	commanded roll angle, deg
ϕ_e	roll angle error, deg
$\dot{\phi}$	roll angular rate, deg/sec
$\delta_{\phi m}$	roll cyclic stick, in.

ψ	yaw angle, deg
ψ_c	command yaw angle, deg
ψ_e	yaw angle error, deg
$\dot{\psi}$	yaw axis angular rate, deg/sec
$\delta_{\psi m}$	pedal position
ω_B	boost actuator natural frequency, rad/sec
ω_s	series servoactuator natural frequency, rad/sec
P_s	static pressure, psi
P_t	total pressure, psi
AH	altitude hold
DH	dynamic pressure hold
HH	heading hold
PAH	pitch attitude hold
RAH	roll attitude hold