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THREE-AXIS FLUIDIC STABILITY AUGMENTATION SYSTEM FLIGHT TEST REPORT

By

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September 1971

EUSTIS DIRECTORATE
U. S. ARMY AIR MOBILITY RESEARCH AND DEVELOPMENT LABORATORY
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13. ABSTRACT

This report covers the flight test of a three-axis hydrofluidic stability augmentation system for a UH-1-type helicopter. The design goal was to improve the performance of the aircraft, without stabilizer bar, in the speed range of 60 to 120 kn.

The system was installed in a UH-1C helicopter. The helicopter's hydraulic power supply was used to power the FSAS controllers and servoactuators. The motions of the helicopter were measured by recording instruments installed in the vehicle. The performance of the helicopter was determined by comparing the recordings of the measuring instruments with the size and type of helicopter command. This type data was obtained with the helicopter in following four conditions:
1) stabilizer bar attached FSAS off with spool valve servoactuators,
2) stabilizer bar off, FSAS off with spool valve servoactuators,
3) stabilizer bar off, FSAS on, with spool valve servoactuator, and
4) stabilizer bar off, FSAS on, with vortex valve servoactuators.

The system improved the performance of the UH-1C helicopter in all three axes by increasing the damping, increasing the phugoid mode period, and producing a constant vehicle rate proportional to cyclic stick input.

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Spool valve servoactuators						
Vortex valve servoactuators						
Pitch-axis controller						
Roll-axis controller						
Yaw-axis controller						
Aircraft handling qualities						

DEPARTMENT OF THE ARMY
U. S. ARMY AIR MOBILITY RESEARCH & DEVELOPMENT LABORATORY
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This report has been reviewed by the Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory and is considered to be technically sound. The purpose of the program was to install the three-axis hydrofluidic stability augmentation system in a UH-1 helicopter and conduct a flight test evaluation of the system. The report is published for the exchange of information and appropriate application. The technical monitor for this contract was Mr George W. Fosdick, Aeromechanics Division.

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The system was installed in a UH-1C helicopter. The helicopter's hydraulic power supply was used to power the FSAS controllers and servoactuators. The motions of the helicopter were measured by recording instruments installed in the vehicle. The performance of the helicopter was determined by comparing the recordings of the measuring instruments with the size and type of helicopter command. This type data was obtained with the helicopter in following four conditions:
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The system improved the performance of the UH-1C helicopter in all three axes by increasing the damping, increasing the phugoid mode period, and producing a constant vehicle rate proportional to cyclic stick input.

FOREWORD

This document is the final report on a flight test program authorized by the Eustis Directorate, U.S. Army Air Mobility Research and Development Laboratory under Contract DAAJ02-70-C-0017. The technical monitor of this program was Mr. G. W. Fosdick.

This program is part of the U.S. Army's continuing effort to develop stability augmentation systems for helicopters. The object was to flight test a UH-1-type helicopter with a hydrofluidic stability augmentation system. The work presented started 1 January 1970 and was completed 17 December 1970.

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LIST OF SYMBOLS

db	decibel - 20 log (output/input)
e	constant = 2.71828
S	LaPlace operator
T_{HP}	high-pass time constant, sec
T_{lag}	lag time constant, sec
K_{ψ}	tail rotor angle/aircraft yaw-axis turning rate, deg/deg/sec
$K_{\delta_{\psi}}$	augmented servoactuator, command/mechanical tail rotor command, in./in.
δ_{ψ_m}*	mechanical tail rotor command from pilot's pedals, deg
$\delta_{\psi_{aug}}$	yaw-axis augmented servoactuator command from pilot's pedals, deg
δ_{ψ_s}	yaw-axis servoactuator displacement, deg
δ_{ψ}	total tail rotor deflection, deg
$\dot{\psi}$	aircraft yaw-axis turning rate, deg/sec
ψ	yaw attitude, deg
K_{ϕ}	roll blade angle/aircraft roll-axis turning rate, deg/deg/sec
δ_{ϕ_m}*	mechanical roll cyclic command from pilot's stick, deg
δ_{ϕ_s}	roll-axis servoactuator displacement, deg
δ_{ϕ}	total blade deflection (roll), deg

*Note: These items are in inches in Appendix II.

LIST OF SYMBOLS (CONCLUDED)

$\dot{\phi}$	aircraft roll-axis turning rate, deg/sec
ϕ	roll attitude, deg
K_{θ}	pitch blade angle/aircraft pitch-axis turning rate, deg/deg/sec
δ_{θ_m}*	mechanical pitch cyclic command from pilot's stick, deg
δ_{θ_s}	pitch-axis servoactuator displacement, deg
δ_{θ}	total blade deflection (pitch), deg
$\dot{\theta}$	aircraft pitch-axis turning rate, deg/sec
θ	pitch attitude, deg
ζ	damping ratio
τ	transport delay, sec
ω	natural frequency, rad/sec

*Note: These items are in inches in Appendix II.

LIST OF DEFINITIONS

Definitions for the abbreviations used in this report are as follows:

- SV Servo, W/O Stab - UH-1C helicopter without mechanical stabilizer bar; FSAS engaged in pitch, roll, and yaw axes coupled to spool valve servoactuators.
- VV Servo, W/O Stab - UH-1C helicopter without mechanical stabilizer bar; FSAS engaged in pitch, roll, and yaw axes coupled to vortex valve servoactuators.
- W/O Stab - UH-1C helicopter without mechanical stabilizer bar; FSAS disengaged.
- With Stab - UH-1C helicopter with mechanical stabilizer bar; FSAS disengaged.
- RS - Right step
- LS - Left step
- RP - Right pulse
- LP - Left pulse
- US - Up step
- DS - Down step
- Control Sensitivity:
  - a) $\dot{\phi}_{ss}/\delta_{\phi}$ - Helicopter's steady-state roll rate attained for a roll cyclic step input, deg/sec/in.
  - b) $\dot{\phi}_{peak}/\delta_{\phi}$ - Helicopter's peak roll rate attained for a roll cyclic step input, deg/sec/in.
- Response Time, $t_{90\%}$:
  - a) Time required for helicopter to achieve 90 percent peak rate for a step input command, when helicopter's response is predominantly a peak rate response.
  - b) Time required for helicopter to achieve 90 percent of steady-state rate for a step input command.
  - c) When helicopter's response is predominantly overshoot less than approximately 30 percent.

LIST OF DEFINITIONS (CONCLUDED)

- Control Power, ϕ/δ_{ϕ} - Helicopter's roll attitude achieved after 1 sec for a roll cyclic step input, deg/in.
- LVDT - Linear variable displacement transducer

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SECTION I
INTRODUCTION

This report presents results of flight testing a three-axis fluidic stability augmentation system (FSAS) using a UH-1C helicopter as the test vehicle. The primary objective of the flight test program was to demonstrate the feasibility of using hydrofluidic sensors and shaping networks in flight control system applications.

The FSAS, developed under Contracts DAAJ02-68-C-0039 and DAAJ02-69-C-0036, was designed to optimize the damping versus control response of the UH-1B without mechanical stabilizer bar for the high-speed, fire-support mission. The FSAS was developed from the viewpoint of increasing vehicle damping and augmenting the free vehicle's short-term response characteristics to provide a rate response proportional to control stick inputs.

FSAS performance was evaluated analytically and comparatively. The analytic evaluation consisted of measuring the short-period damping and control sensitivity of (1) the FSAS-augmented helicopter without mechanical stabilizer bar, and (2) the unaugmented helicopter with and without mechanical stabilizer bar.

The comparative evaluation consisted of qualitatively comparing flight recordings for (1) the FSAS-augmented helicopter, (2) the unaugmented helicopter without mechanical stabilizer bar, and (3) the unaugmented helicopter with mechanical stabilizer bar. Attempts were made to correlate pilot comments (see Appendix I) with the specific tests.

FSAS flight test results were recorded (see Appendix II) for the stability augmentation system coupled to spool valve servoactuators and fluidic vortex valve servoactuators.

SECTION II
SUMMARY

SYSTEM DESIGN AND INSTALLATION

System Design

A previous study phase contract, DAAJ02-68-C-0039, mathematically defined a hydrofluidic stability augmentation system for a UH-1B helicopter during the high-speed, gun-firing mission. A set of design goals was established which stated that the FSAS must be simple, provide a stable gun platform, and augment the vehicle in a manner similar to the mechanical stabilizer bar. Using six-degree-of-freedom equations of motion, an FSAS was designed under Contract DAAJ02-69-C-0036, fabricated, and closed-loop simulation tests were conducted. An analog computer was used to simulate the vehicle and was connected through a servo-driven oscillating table to the controller. To close the loop the output of the servoactuator was then routed back to the computer. Simulated flight conditions from hover to 130 kn were run. Results of the closed-loop tests showed that the FSAS was nearly equivalent to that defined during the study phase analysis.

Installation

Mechanical Installation

The servoactuators were located below the floor in series with the control tubes of the vehicle. It was necessary to enlarge the control tube bulkhead openings to permit installation of the servoactuators. Also, the copilot throttle control and the heater ducts were removed for ease of installation.

New control tubes, used in conjunction with the servoactuators, replaced the standard control tubes. The control tubes had swaged ends with included angles of 30 deg instead of the specified 10 deg, so strength tests in compression and tension were successfully conducted.

To control the pilot's control input steps and pulses, adjustable stops were installed on the copilot's stick and pedals. These stops were fabricated with shear pins so if a 25-lb force on the stick or a 50-lb force on the pedals was applied, the pins would break away, returning unlimited control of the vehicle to the pilot.

The mechanical stabilizer bar was removed and replaced with fixed brackets that connected the control tubes directly to the main rotor blades.

Hydraulic Installation

The FSAS was connected to the No. 2 boost hydraulic system during the spool valve servoactuator flight test. During the flight test of the vortex valve servoactuators, the controllers were connected to the No. 2 boost hydraulic system, while the servoactuators were connected to the No. 1 boost hydraulic system. This was necessary because of the increased flow requirements of the vortex valve servoactuators.

Electrical Installation

Instrumentation was installed in the vehicle to record linear acceleration, rate of turn, angular displacement, pilot input, servoactuator motion, and controller output for all three axes. Time, controller supply pressure, controller flow, collective position, and an event marker were also recorded.

It was possible to control the FSAS and recorders from the pilot console. Each axis could be energized in any combination with the other two axes.

MODIFICATION AND PROBLEMS DURING FLIGHT TEST

Servoactuators

The first checkout flight of the FSAS revealed a pull on the stick which caused a left roll. This was traced to the hoses supplying the roll-axis servoactuactor. The problem was corrected by rerouting two of the hoses.

Performance Changes

The FSAS was changed during the flight test because a UH-1C helicopter was used rather than a UH-1B, the vehicle originally studied. Also, the servoactuators and vortex rate sensors had different characteristics from those analyzed during the study phase. These changes did not affect yaw-axis performance, but roll-axis controller gain had to be increased 50 percent, the pitch-axis shaping network was changed to a lag-lead, and pitch-axis controller gain was reduced 33 percent.

Vortex Valve Servoactuators

The vortex valve servoactuators did not perform as well as the spool valve servoactuators. For example, they did not move for low-frequency, low-amplitude signals. Subsequent tests showed that they would move 0.050 in. when a 20-lb load was applied, while the spool valve servoactuators moved only 0.002 in. A static friction level of 20 lb in the control linkages would cause the friction effects that were observed.

Data Reduction

Two factors made it extremely difficult to accurately and consistently determine quantitative aircraft performance. One was the weather, which was poor and required that flights be made on windy, gusty days. The other was the brackets that controlled the size of the pilot's inputs during steps and pulses. These brackets were required to break if too high a force was applied, resulting in a bracket with insufficient stiffness. They would bend, especially during the pulse-type inputs; also, the aircraft had a tendency not to return to the original trim condition after the input was removed.

UH-1C/FSAS FLIGHT TEST RESULTS

Flight testing was conducted to obtain both qualitative and quantitative data with the aircraft in various configurations and at a number of speeds and altitudes. Table I lists these conditions and those at which data were recorded.

Yaw FSAS Performance

The FSAS virtually eliminated the undamped yaw rate response, thus making the vehicle easier to control, and the control sensitivity of the vehicle was not changed significantly by the FSAS. The project pilot could not detect any difference in the handling characteristics in response to step inputs with the FSAS engaged or disengaged. The damping was increased from 0.15 to 0.7 with the FSAS engaged.

The vortex valve servoactuator portion of the flight test program was conducted after completing the flight test with the spool valve servoactuators. Yaw rate response with the vortex valve servoactuator system was under-damped and jerky, which was caused by the low pressure gain in the servoactuators. The servoactuators could not overcome the static friction in the control linkage until a large input signal was applied. The damping ratio with the vortex valve servoactuator system ranged from 0.3 to 0.7.

Roll FSAS Performance

The FSAS greatly improved the roll characteristics of the aircraft without stabilizer bar by holding the roll rate constant for a given roll cyclic command. Large rate overshoots were eliminated which allowed pilots to more precisely and easily control the aircraft. Control sensitivity and power with the FSAS engaged were approximately the same as those with the stabilizer bar.

TABLE I. UH-1C/FSAS FLIGHT TEST ENVELOPE

Flight Condition	UH-1C Flight Test Configuration: UH-1C, Without Mechanical Stabilizer Bar, With Spool Valve Servoactuators	UH-1C, Without Mechanical Stabilizer Bar, With Vortex Valve Servoactuators	UH-1C, Without Mechanical Stabilizer Bar	UH-1C With Mechanical Stabilizer Bar
Response to Pulse and Step Inputs:*				
Hover, 3000 ft	X	X	X	X
60 kn, 3000 ft	X	X	X	-
90 kn, 3000 ft	X	X	X	X
120 kn, 3000 ft	X	X	X	-
60 kn, 10,000 ft	X	X	X	-
Five-Minute Stabilized Flight:**				
Hover in ground effect	X	X	X	-
60 kn, 3000 ft	X	X	X	-
Engage/Disengage Transient Climbing, Left and Right Turns:				
60 kn	X	X	-	-
90 kn	X	X	-	-
120 kn	X	X	-	-
Autorotation Entries:				
60 kn	X	X	-	-
90 kn	X	X	-	-
120 kn	X	X	-	-
Qualitative Evaluation:				
Two Government pilots	X	X	X	-
One contractor pilot	X	X	X	X
Two airframe manufacturer pilots	-	X	X	-

*Pitch, roll, and yaw pulses and steps in each direction. Pulses and steps range in magnitude from 0.25 to 1.0 in.

**For these tests, project pilot shall minimize all induced inputs.

The pilots commented, in particular, about the constant roll rate proportional-to-cyclic stick position control provided by the FSAS. It was easier to maneuver the vehicle with this kind of response, as opposed to the response provided by the stabilizer bar.

Pitch FSAS Performance

The FSAS reduced the control sensitivities to more controllable levels, while still providing adequate control power. The FSAS at hover provided nearly a constant rate response-to-cyclic input, making the vehicle easier to hover. The phugoid characteristics were significantly improved, along with an increase in damping ratio to approximately 0.6.

Stabilized Flight, Autorotations, and Turns

Pilot inputs were reduced with the FSAS engaged during stabilized flight and autorotations. The FSAS reduced yaw excursion during autorotations when the throttle was cut. During turns it was easier to control to a given turning rate with the FSAS engaged.

CONCLUSIONS

The following conclusions are presented:

- The FSAS decoupled external cross-coupling disturbances.
- Pilots could fly with more precision.
- Turn coordination was not affected.
- The phugoid mode was stabilized.
- The UH-1C can be flown faster without the stabilizer bar and with the FSAS.
- The aircraft with FSAS still meets required handling characteristics of MIL-H-8501A.
- Oil temperature did not adversely affect FSAS performance over the operating range encountered.
- The FSAS stabilized the UH-1C in a manner comparable to or better than the mechanical stabilizer bar.
- Gain scheduling to improve FSAS performance over the complete flight envelope should be analyzed.
- Angular measuring sensors, synchronizers, and automatic trim functions should be developed.
- A rotor/pylon stabilization system for the UH-1 series vehicle using hydrofluidics should be explored.
- Further development of temperature compensation techniques should be pursued.

In general, the flight test program results demonstrated that hydrofluidic technology can adequately mechanize the damping functions required by single-rotor helicopters.

SECTION III
SYSTEM DESIGN AND INSTALLATION

This section reviews system design efforts and tasks that lead to the hardware development of the flight-test FSAS and installation of the FSAS in the test vehicle. Areas covered are system design goals, computer simulation development, closed-loop hardware plus computer simulation evaluation, and mechanical, hydraulic, and electrical installation of the instrumentation, servoactuators, and controllers.

SYSTEM DESIGN

FSAS Design Goals

The computer simulation analysis program of the study phase mathematically defined system block diagrams for a simple stability augmentation system to augment the roll, pitch, and yaw axes of the UH-1B helicopter using hydraulic fluidics as the control medium. The performance requirements specified were that the fluidic stability augmentation system (FSAS) must improve vehicle damping and handling qualities of the UH-1B helicopter during the high-speed, gun-firing mission. With these general system requirements in mind, a set of detailed design goals was generated that permitted FSAS performance to be evaluated.

The detailed design goals used as guidelines during development of the FSAS control system are presented in USAAMRDL Technical Report 71-30. These design goals were generated to be in agreement with helicopter flying and ground handling-quality requirements detailed in Military Specification MIL-H-8501A.

Those design goals needed to establish FSAS performance are included in the performance and evaluation summaries of Section V.

The primary analysis goals were that the resulting FSAS must be a simple system and it must provide a more stable gun-firing platform for the high-speed UH-1B gun-firing mission. Second, the system should augment the vehicle in pitch and roll comparable to the manner presently accomplished by the mechanical stabilizer bar.

Computer Simulation Development

Six-degree-of-freedom, linear perturbation equations of motion mathematically representing the UH-1B helicopter were used to analytically define the FSAS. The complete math model, including equations of motion, aerodynamic data, and computer simulation diagrams of the UH-1B helicopter and FSAS, are presented in Appendix I of USAAMRDL

Technical Report 71-30. Time histories showing UH-1B response characteristics with and without the FSAS engaged are also presented.

Although the UH-1B was used as the analytical model, the UH-1C was selected as the test vehicle. A qualitative comparison of the helicopter responses for the UH-1B (USAAMRDL Technical Report 71-30) with those recorded for the UH-1C (this report), leads to the following conclusions:

- At hover, the pitch and roll axes of the UH-1C helicopter have predominantly a rate response proportional to stick input, while the simulated UH-1B was predominantly an attitude response proportional to control stick input.
- The yaw axis of the UH-1C is much more underdamped than that of the simulated UH-1B (0.15 as opposed to 0.35).
- The roll-axis spiral divergence mode is more pronounced and occurs much sooner on the UH-1C.

With the exception of the above three items, the gross aerodynamic characteristics of the UH-1B and UH-1C are similar.

Closed-Loop Hardware Evaluation

Performance of the three-axis FSAS was investigated by evaluating system transient response behavior in closed-loop tests. The closed-loop test setup checked the pitch and roll-yaw axes separately. Analog computer simulations represented the uncoupled pitch and roll-yaw equations of motion of the UH-1B helicopter at representative forward speeds ranging from hover to 130 kn. A servo-driven oscillating table was used to provide appropriate motion inputs to the FSAS rate sensors. The three-axis FSAS was mounted on the oscillating table and used to drive servoactuators similar to those provided on the UH-1B helicopter. Thus, the closed-loop tests were functionally equivalent to those anticipated during flight testing except for any nonlinear characteristic present in the primary control system of the UH-1B helicopter. For complete details concerning test setup and test results, refer to USAAMRDL Technical Report 71-30. Highlights of the closed-loop test results are presented in the following paragraph.

The closed-loop simulation results obtained on the three-axis FSAS indicate that nominal transient response performance at the design oil temperature (120°F) is nearly equivalent to that indicated in the study phase analysis studies. However, the closed-loop simulation tests pointed out two problem areas: (1) high oil temperatures (185°F) produce undesirably high system noise levels; (2) low oil temperatures (60°F) result in almost no system noise but also reduce damping of the short-period and dutch roll modes at high speeds to nearly that of the free aircraft.

During the UH-1C/FSAS flight test program, servoactuator travel and oil temperature were monitored. The oil temperature stayed between 100°F and 150°F. There was no evidence of noise in either the roll- or pitch-axis servoactuators. The yaw-axis servoactuator did have a small amount of pedal motion caused by the controller, but it did not at any time become objectionable to the pilot or to overall yaw-axis performance. The cause of this pedal motion is the pilot input device and the lack of friction in the linkage between the pedal and the servoactuator. When the pedal moves forward, the action of the pilot input device causes the servoactuator to retract, which pulls the pedal rearward, and when the pedal moves rearward, the servoactuator extends, pushing the pedal forward. This closed-loop connection causes the pedal to oscillate, but does not become excessive because of a lag network in the pedal input circuit. If the pedals were kept from moving by the pilot or by increasing friction, the motion could be stopped.

INSTALLATION

Mechanical Installation

Servoactuator and Linkages

Figure 1 is a schematic of the servoactuator installation, showing the approximate location of the three servoactuators. During installation it became necessary to cut a larger hole in the bulkhead at station 66 so that roll and pitch servoactuators could be installed. Figure 2 shows the hole in this bulkhead with the stiffeners and doubler that were added. To provide enough clearance for the roll-axis servoactuator, it was necessary to remove the throttle linkage that connects the copilot's throttle to the pilot's throttle linkage. This is discussed further in Appendix III. To install the yaw-axis servoactuator, the control rod hole in the bulkhead at station 52 was enlarged and a stiffener and doubler were added, which is shown in Figure 3. The pitch jackshaft that connects the pilot and copilot controls was modified. The original installation used a swivel-type bearing in the arm assembly. The arm assembly was replaced with a newly fabricated part using a fixed bearing. This prevents the servoactuator, which connects through a control tube that attaches to the arm assembly, from rotating and possibly hitting the roll servoactuator or bulkhead.

It was necessary to fabricate and test new control tubes for all three axes. The report covering the strength testing of these assemblies is given in Appendix IV. It was necessary to test the strength of these control tubes because the half-angle taper on the tube was 15 deg instead of the specified 5 deg. To provide increased safety, the tube wall thicknesses were increased. Tubes

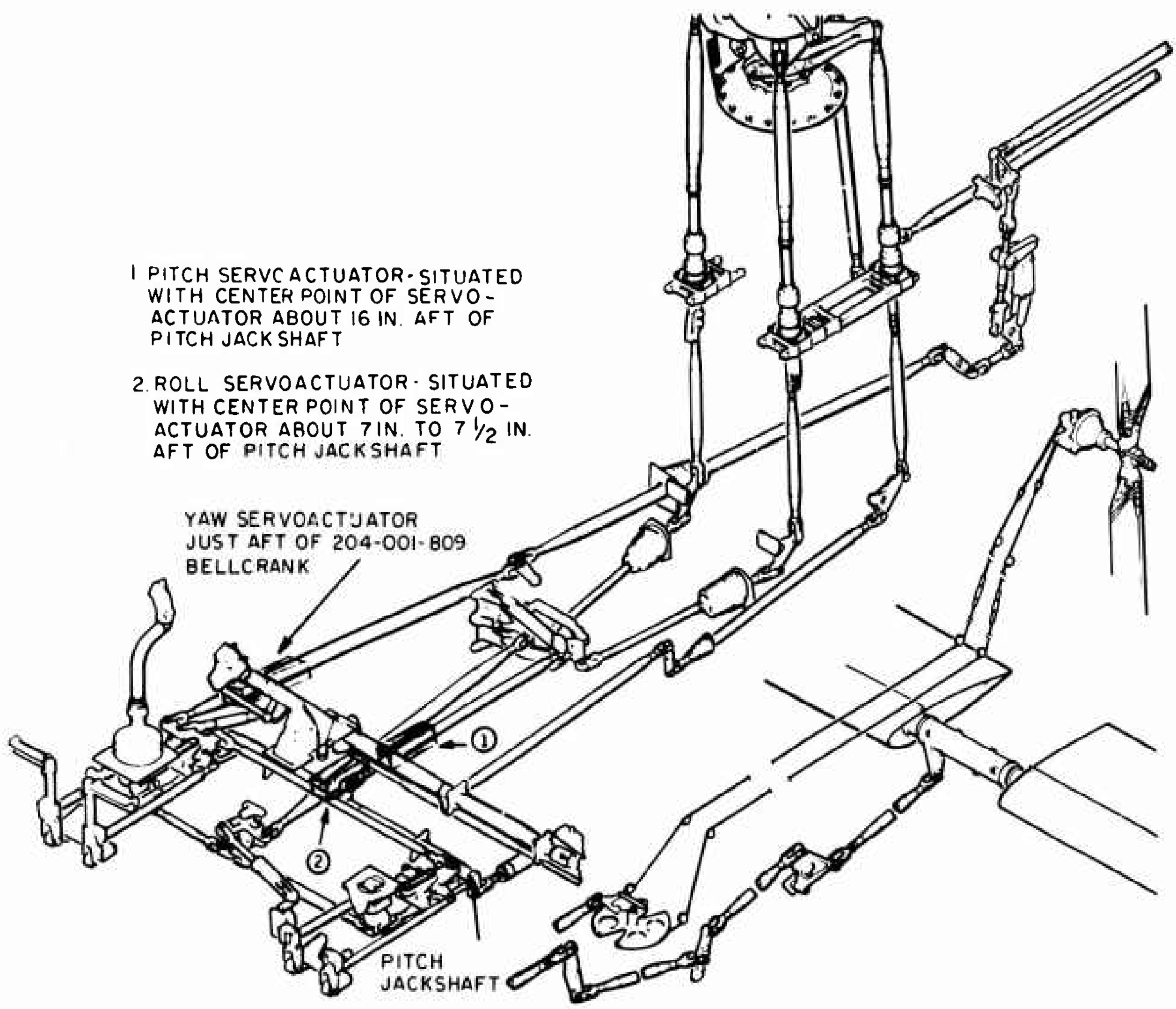


Figure 1. Servoactuator Installation Schematic.

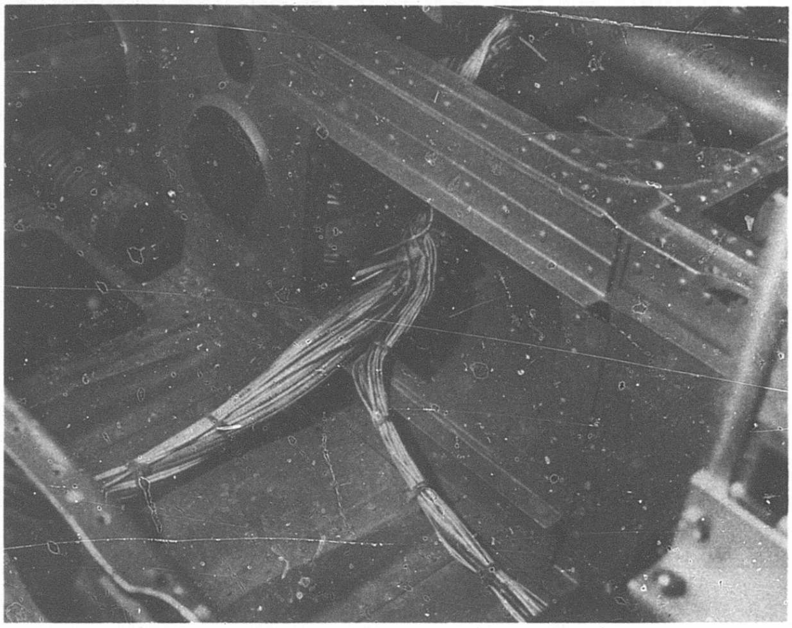


Figure 2. Fuselage Station 66 Looking Forward After Modification.

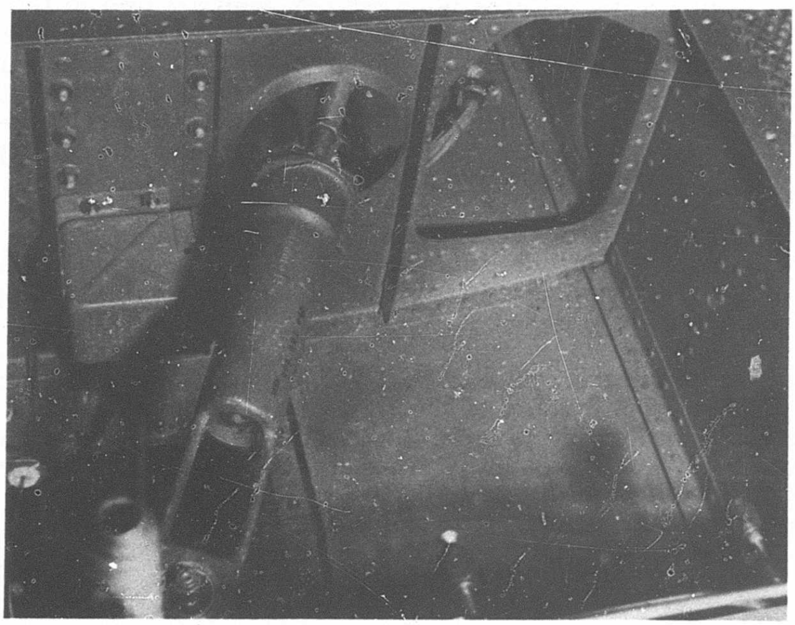


Figure 3. Fuselage Station 52 Looking Forward After Modification.

with the larger angle were purchased because the vendor would have had to acquire special fixturing to fabricate tubes with the smaller angle, and an unnecessary delay in the program would have resulted.

Figures 4, 5, 6, and 7 show the installation of the roll- and pitch-axis servoactuators beneath the floor of the vehicle. The servoactuator support brackets had adjustable stops installed to limit the total travel of the idler arms. The support brackets for the bellcranks located at station 123 also had adjustable stops installed to limit travel of the control tubes aft of the roll-pitch mixing linkage. These are shown in Figures 8 and 9. Figures 10 and 11 show the yaw-axis servoactuator installed below the floor under the pilot's seat. The support bracket and bellcrank at station 161.61 are shown in Figure 12.

During installation, it was necessary to remove the heater ducts in the area of the servoactuators to make installation easier. Appendix III presents a method of installation resulting in retention of the heater ducting for any future permanent installation.

Pilot Input Fixtures

Breakaway fixtures were installed on the copilot side of the aircraft to control the motion of the cyclic stick and rudder pedals during the data flights. These fixtures are shown in Figure 13. The arm extending from the instrument panel was held with rivets that would shear when a load of 25 lb was applied to the movable rod. The same type device was used on the rudder pedals, except 50 lb could be applied before shearing. This was required as a safety precaution in case the adjustment fixture could not be removed from around the stick or pedals when the pilot needed to make a quick correction to the aircraft. This capability was utilized three times during the flight test.

The "fingers" around the stick could be adjusted during flight to restrict pilot control inputs to a predetermined safe amount. This also provided repeatability of fixed step and pulse-type pilot control inputs.

The pedal fixture was eventually changed slightly from that shown in Figure 13. Instead of the fixture being placed on top of the pedal, it was held up under the pedal so that if it was released in any way it would fall away from the pedal. Also, the bracket that held the rod with the adjustable fingers was bent 90 deg down so the rod pivot line was parallel with the pedal foot rest.

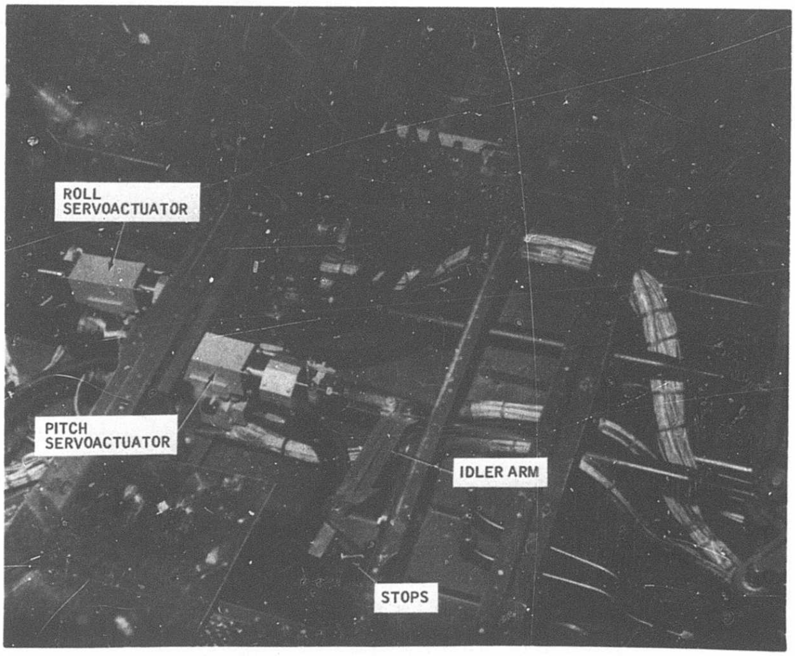


Figure 4. Roll- and Pitch-Axis Spool Valve Servoactuators Installation.

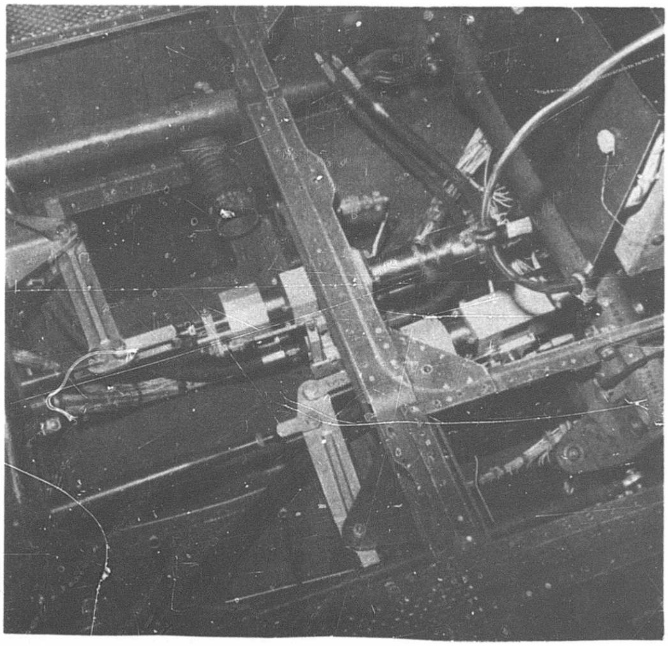


Figure 5. Roll- and Pitch-Axis Spool Valve Servoactuators From Right Side of Aircraft.

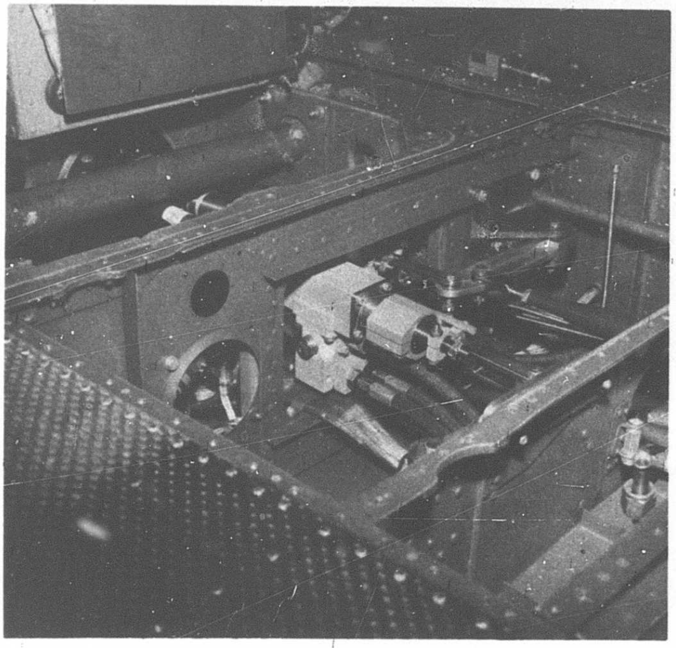


Figure 6. Roll- and Pitch-Axis Spool Valve Servoactuators From Left Side of Aircraft.

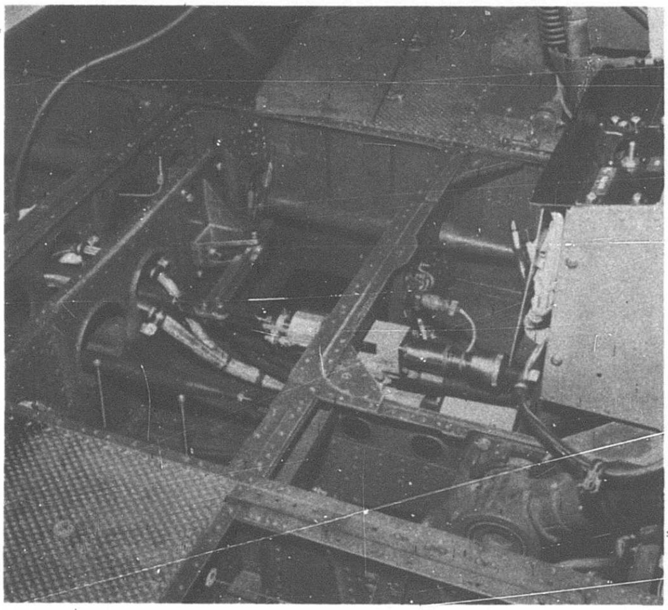


Figure 7. Roll- and Pitch-Axis Spool Valve Servoactuators From Right Front Side of Aircraft.

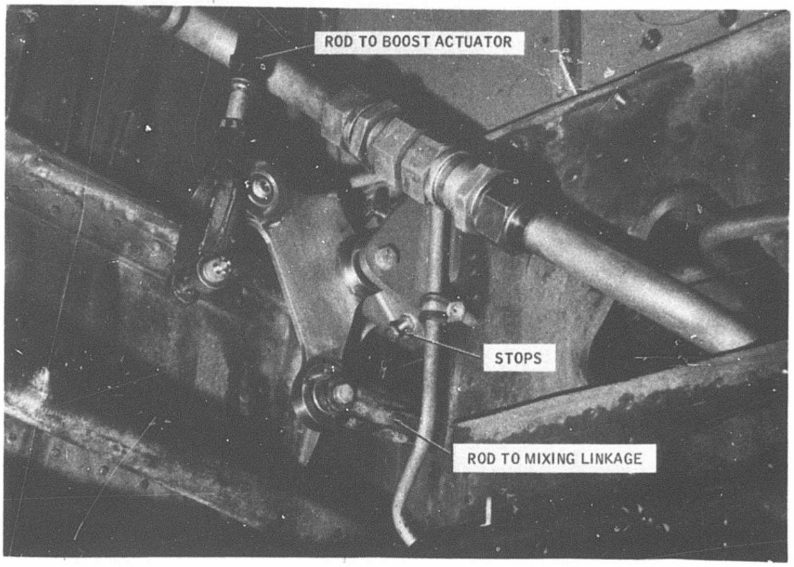


Figure 8. Left Pitch-Roll Bellcrank Installation (Fuselage Station 123).

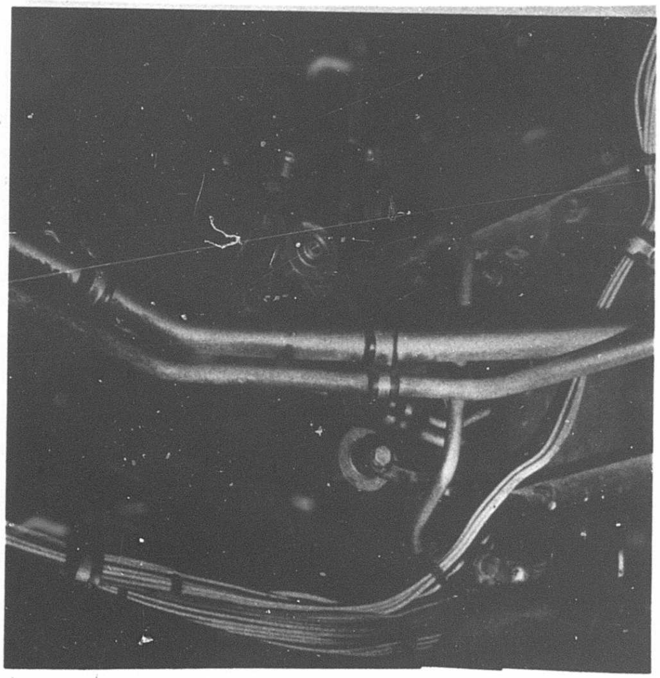


Figure 9. Right Pitch-Roll Bellcrank Installation (Fuselage Station 123).

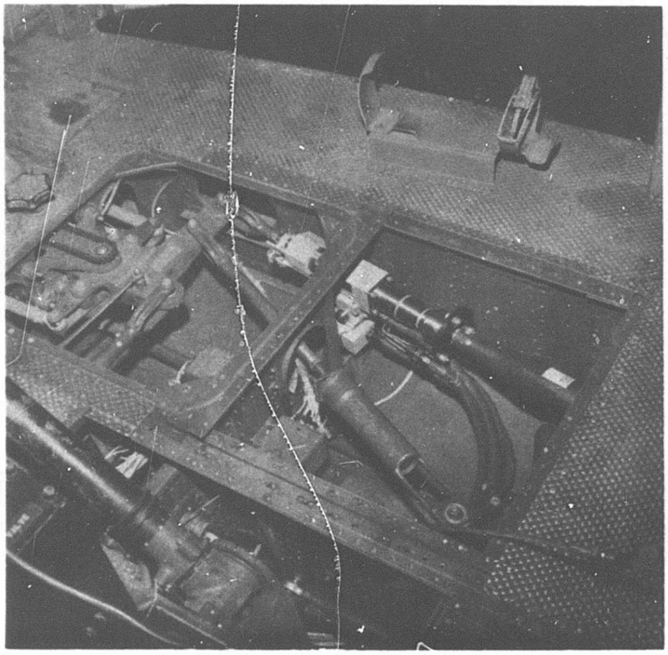


Figure 10. Yaw-Axis Spool Valve Servoactuator Installation.

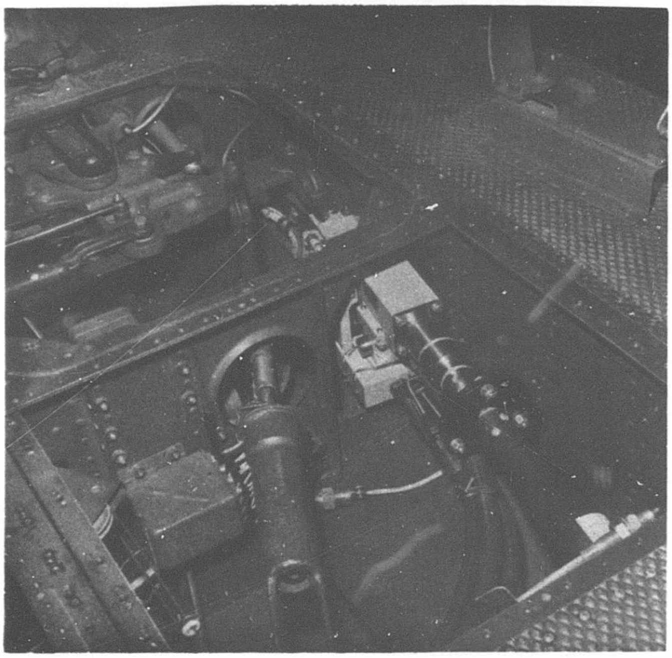


Figure 11. Yaw-Axis Spool Valve Servoactuator From Rear of Crew Compartment.

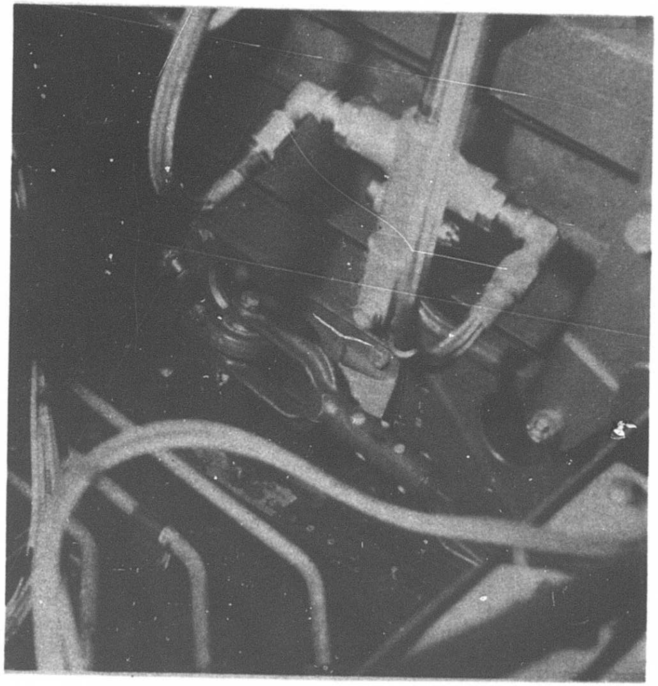


Figure 12. Yaw Bellcrank With Stops (Fuselage Station 161.61).

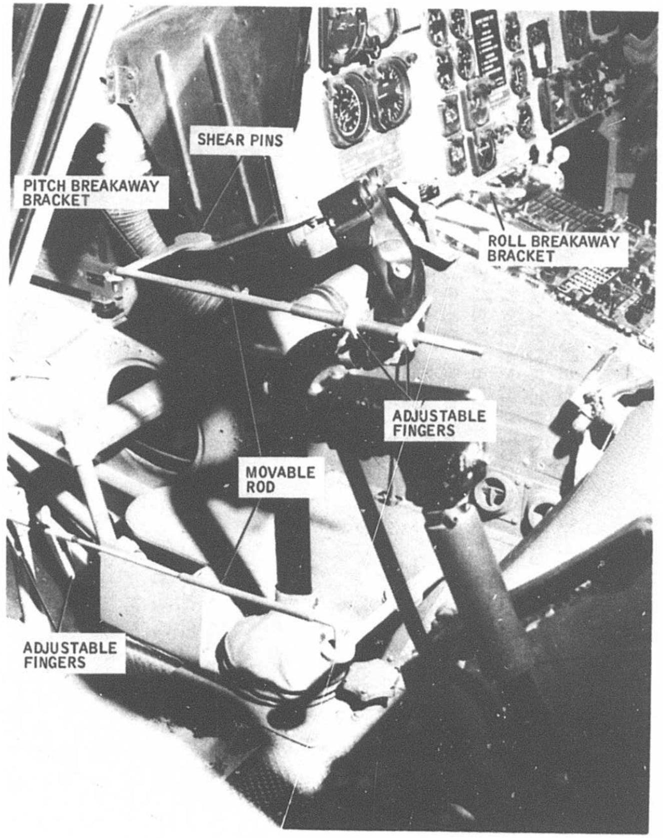


Figure 13. Pilot Control Input Fixtures.

Stabilizer Bar

Flight was conducted with four aircraft configurations: normal aircraft with stabilizer bar on; free aircraft with stabilizer bar off; FSAS engaged with spool valve servoactuators (stabilizer bar off); and FSAS engaged with vortex valve servoactuators (stabilizer bar off). Figure 14 shows the aircraft in its normal configuration with the stabilizer bar in place. Figure 15 shows the aircraft with the stabilizer bar removed and fixed brackets connecting the control arm and rotor mast added.

Hydraulic Installation

During the first series of flight tests using the spool valve servoactuators, the complete fluidic installation was accomplished using only the No. 2 aircraft hydraulic power supply. This was done for safety reasons, which left the No. 1 hydraulic power supply unchanged from the normal aircraft configuration. The system was connected as shown schematically in Figure 16. The hardware installation is shown in Figures 17 (bench) and 18 (in aircraft). The lines connecting the controllers to the servoactuators passed through the aircraft floor using bulkhead fittings. From the bulkhead fittings, flexible lines were run to the servoactuators.

The second portion of the flight test program was to evaluate the vortex valve servoactuators. It was necessary at this time to use the No. 1 and the No. 2 aircraft hydraulic power supplies, as the No. 2 aircraft hydraulic power supply could not supply enough flow for both the FSAS and vortex valve servoactuators connected in parallel. The vortex valve servoactuators used a steady-state flow of approximately 0.45 gpm each, while the FSAS used 2.3 gpm. Previous plans to operate the servoactuators and FSAS in series to conserve flow were discarded to expedite the program. Since the first phase of the flight test program had shown that the FSAS was a reliable system, it was considered safe to connect into both hydraulic power supplies. Figure 19 shows the schematic of the vortex valve hydraulic circuit. The only difference between the spool valve and vortex valve servoactuator circuits is in the servoactuator portion of the system. The fact that the FSAS circuit was not changed also ensured that the operating conditions of the FSAS would be unchanged and that a more valid comparison of the two types of servoactuators could be made. Figures 20 through 23 show the vortex valve servoactuators installed in the aircraft.

Electrical Installation

Two inverters were installed in the aft storage compartment to provide 115-volt, 400-Hz power for the instrumentation equipment, including a 24-channel visicorder, which was used to record the following parameters:

Figure 14. UH-1 With Stabilizer Bar.

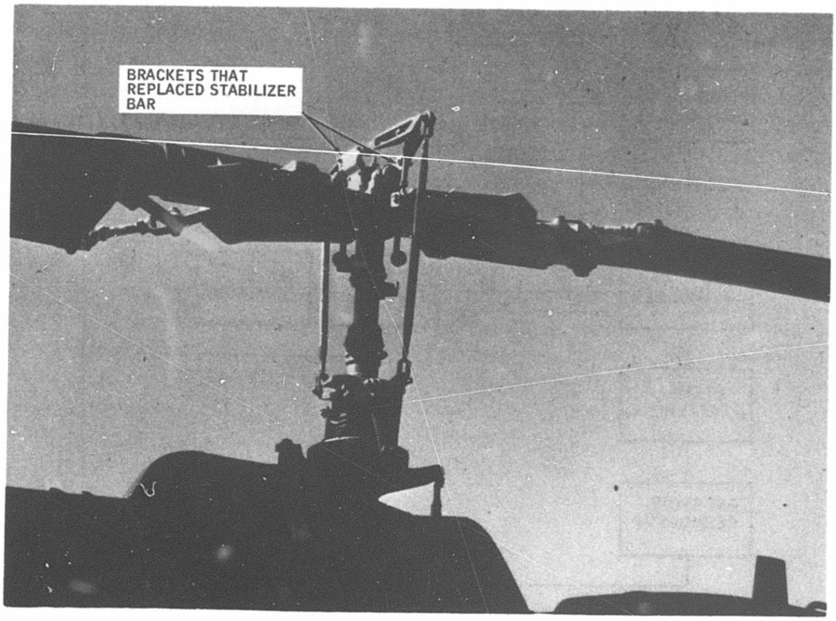


Figure 15. UH-1 Without Stabilizer Bar.

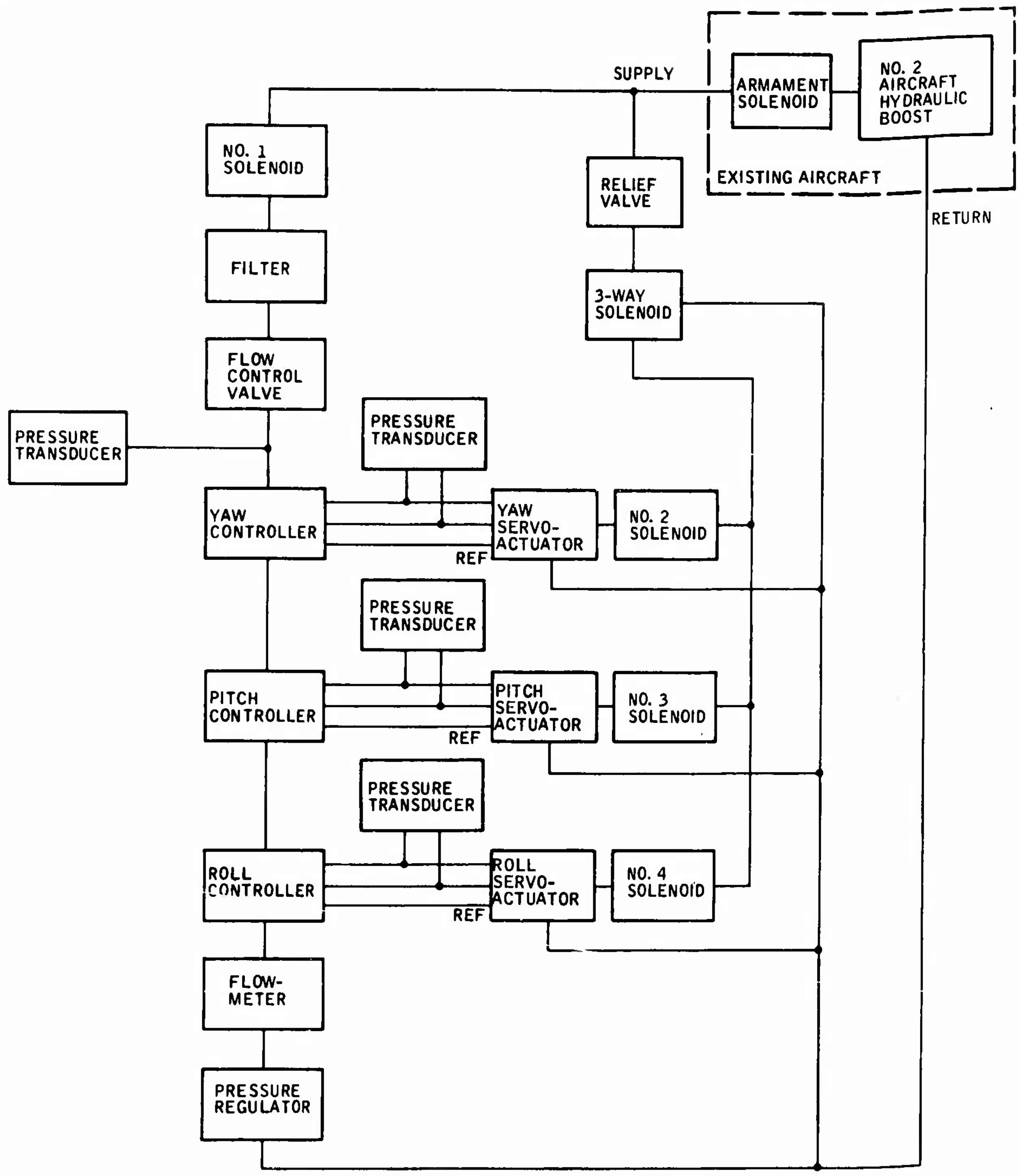


Figure 16. Spool Valve Servoactuator Hydraulic Circuit Diagram

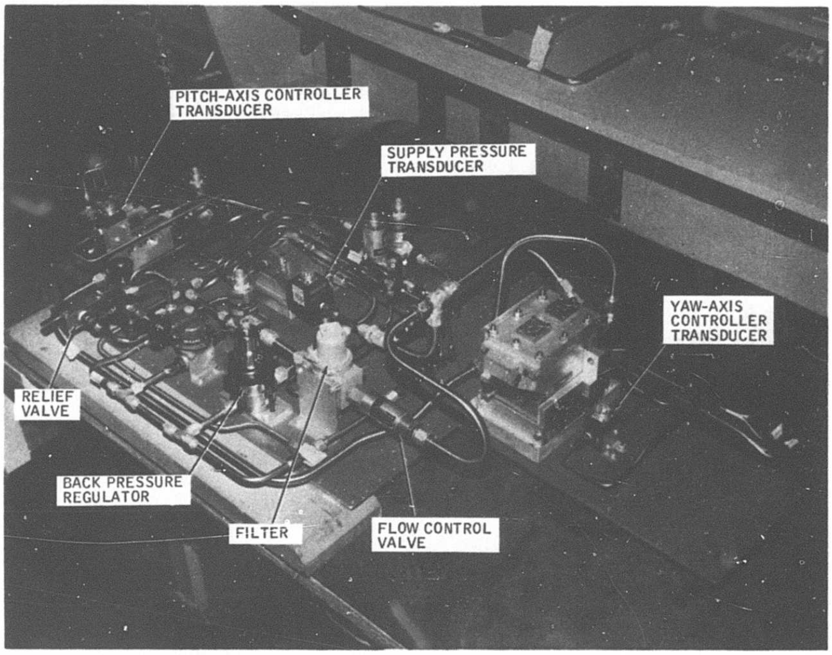


Figure 17. FSAS Ready for Installation in Aircraft.

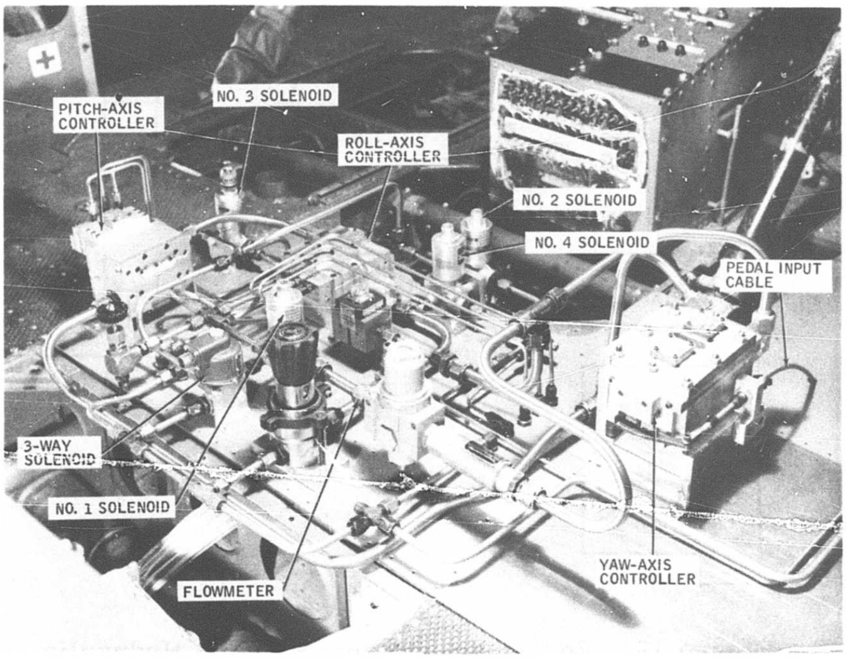


Figure 18. FSAS Installation From Right Side of Aircraft.

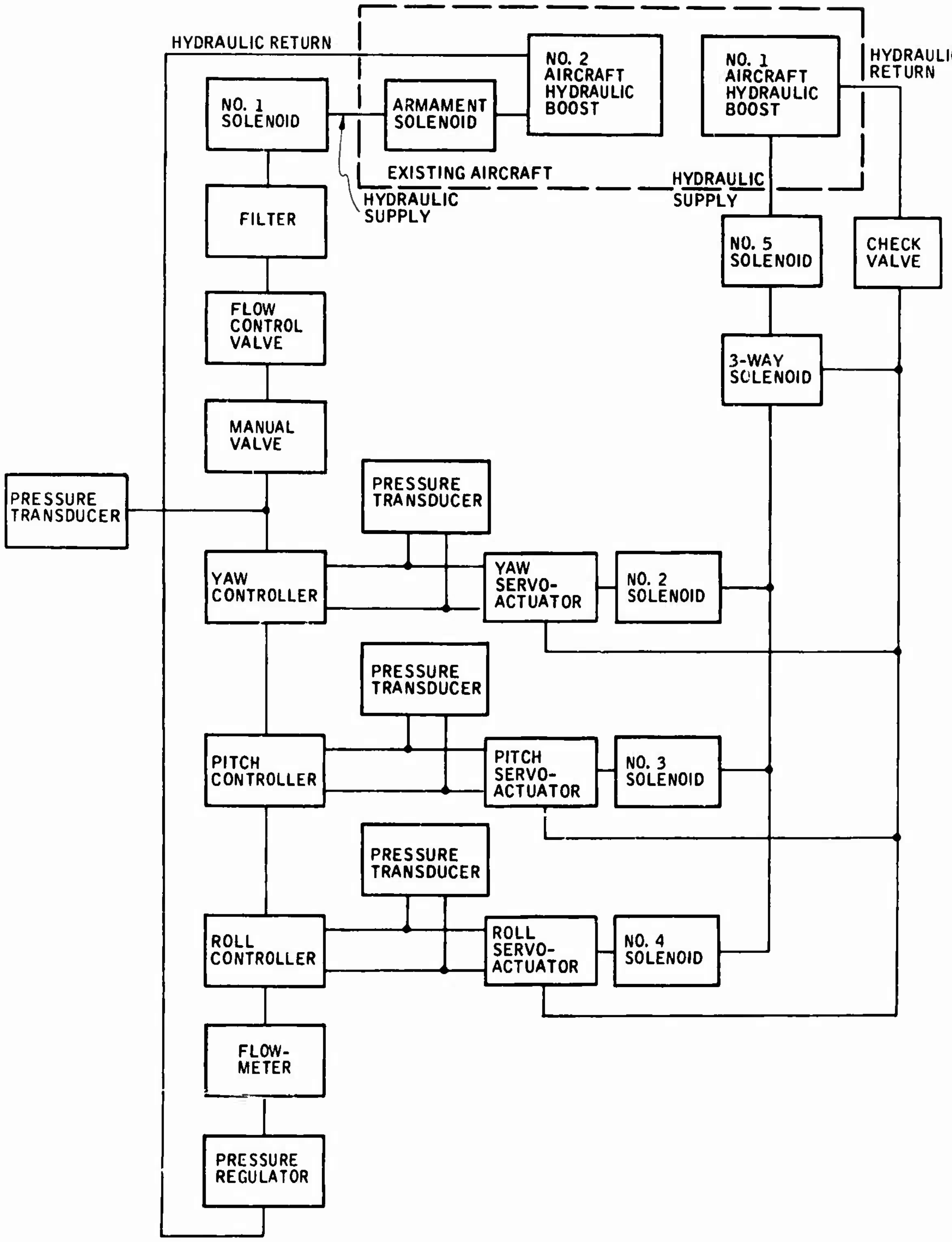


Figure 19. Vortex Valve Servoactuator Hydraulic Circuit Diagram.

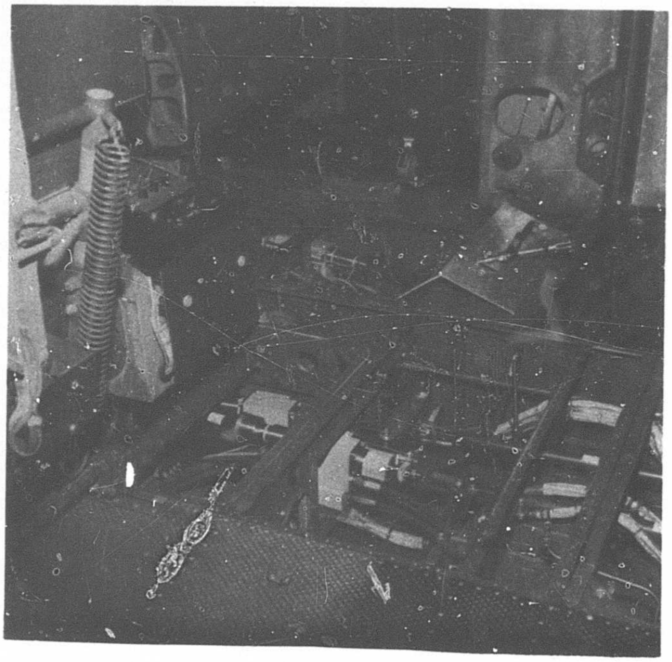


Figure 20. Full View of Vortex Valve Servoactuators Installation.

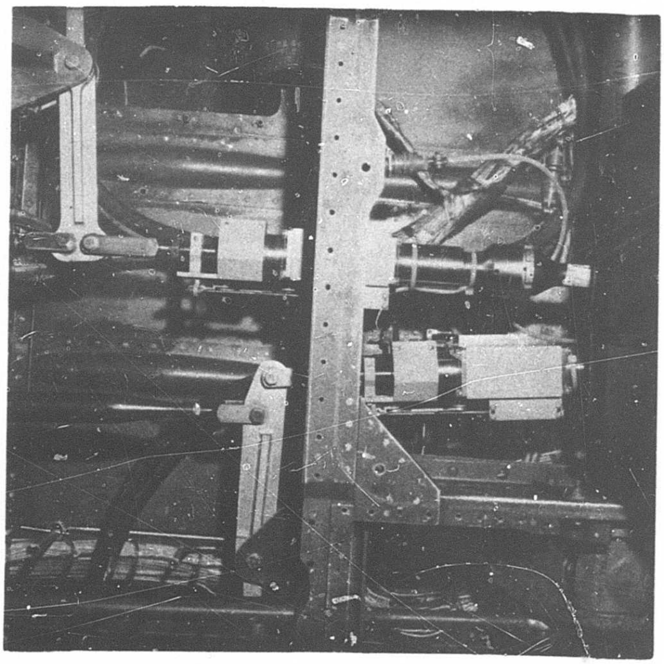


Figure 21. Top View of Vortex Valve Servoactuators Installation.

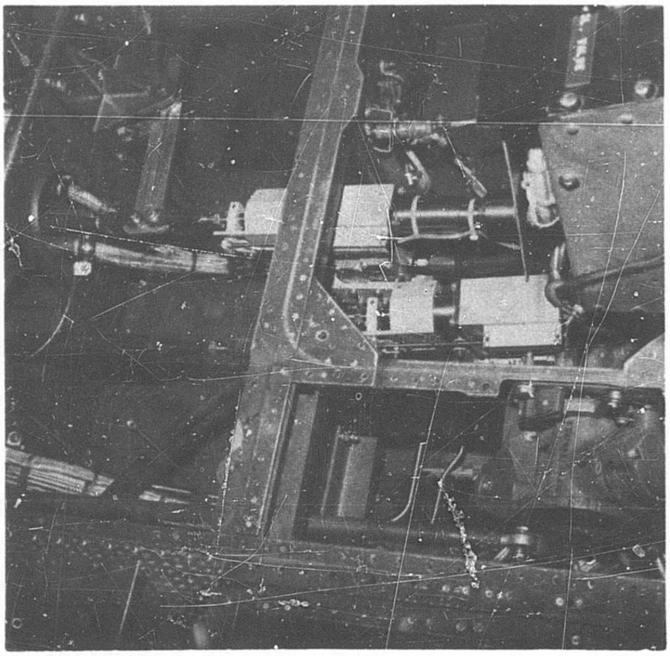


Figure 22. Roll- and Pitch-Axis Vortex Valve Servo-actuators Installation.

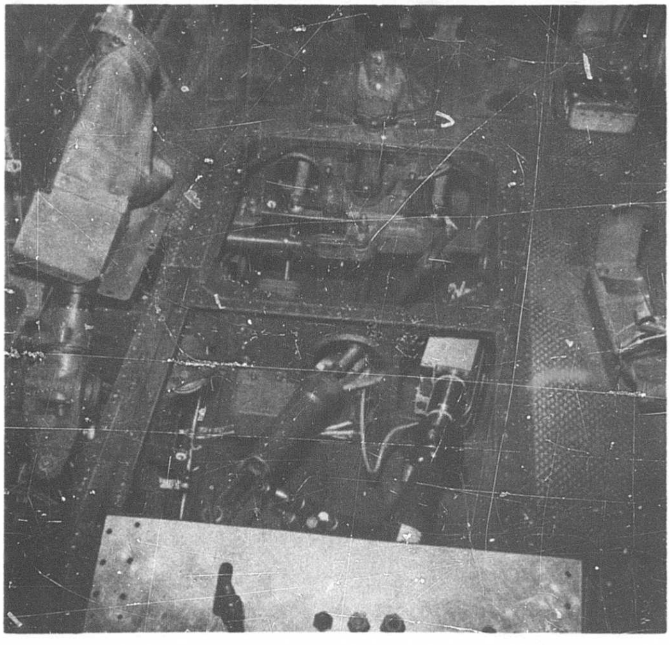


Figure 23. Yaw-Axis Vortex Valve Servoactuator Installation.

Yaw attitude
Yaw-axis servoactuator motion
Pedal position
Yaw-axis controller output
Yaw rate
Lateral acceleration
Pitch attitude
Pitch-axis servoactuator motion
Pitch cyclic position
Pitch-axis controller output
Pitch rate
Vertical acceleration
Roll attitude
Roll-axis servoactuator motion
Roll cyclic position
Roll axis controller output
Roll rate
Collective position
Controller supply pressure
Controller flow
Longitudinal acceleration
Timing marker
Event marker

Figure 24 shows the inverters installed in the aircraft.

Yaw attitude was measured by a heading gyro mounted in the instrument panel. Roll and pitch attitudes were measured by a vertical gyro. The accelerations in all three axes were measured using a package containing three accelerometers. Three electronic rate gyros measured aircraft turning rates. These instruments are shown in Figures 25 and 26. (These photos were taken during installation with the pilot's and copilot's seats removed.)

Transducers were incorporated in the servoactuators to measure actuator piston motion; the spool valve servoactuators used linear variable displacement transducers (LVDTs), and the vortex valve servoactuators used potentiometers. The LVDTs were located within the servoactuator housing, while the vortex valve servoactuator potentiometers were mounted on the side. These can be seen in Figures 27 and 28.

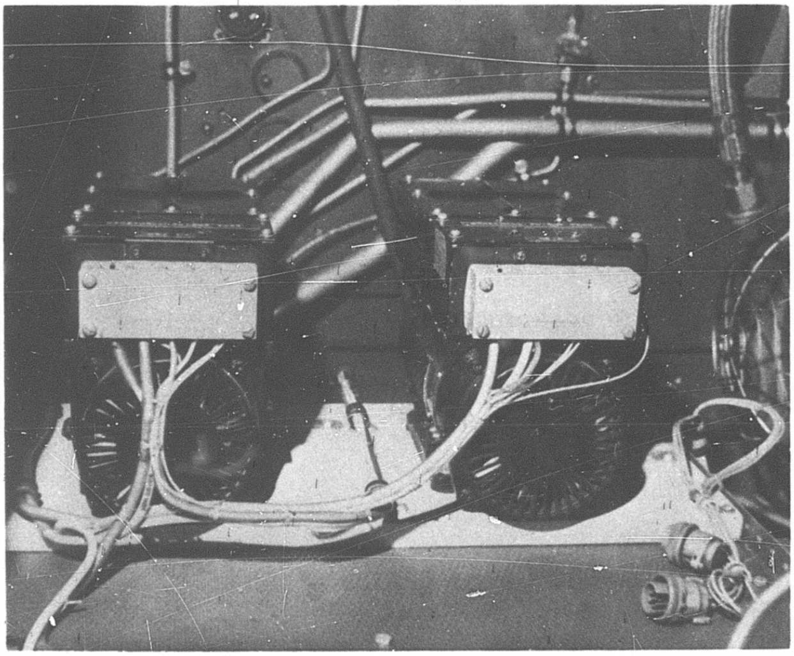


Figure 24. Inverters Mounted in Aft Compartment.

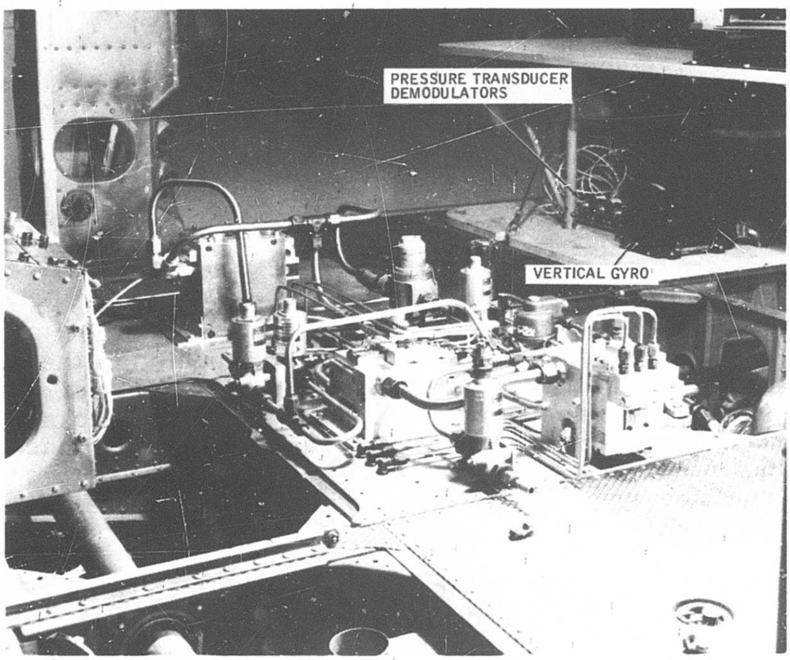


Figure 25. Instrumentation Viewed From Left Side of Aircraft.

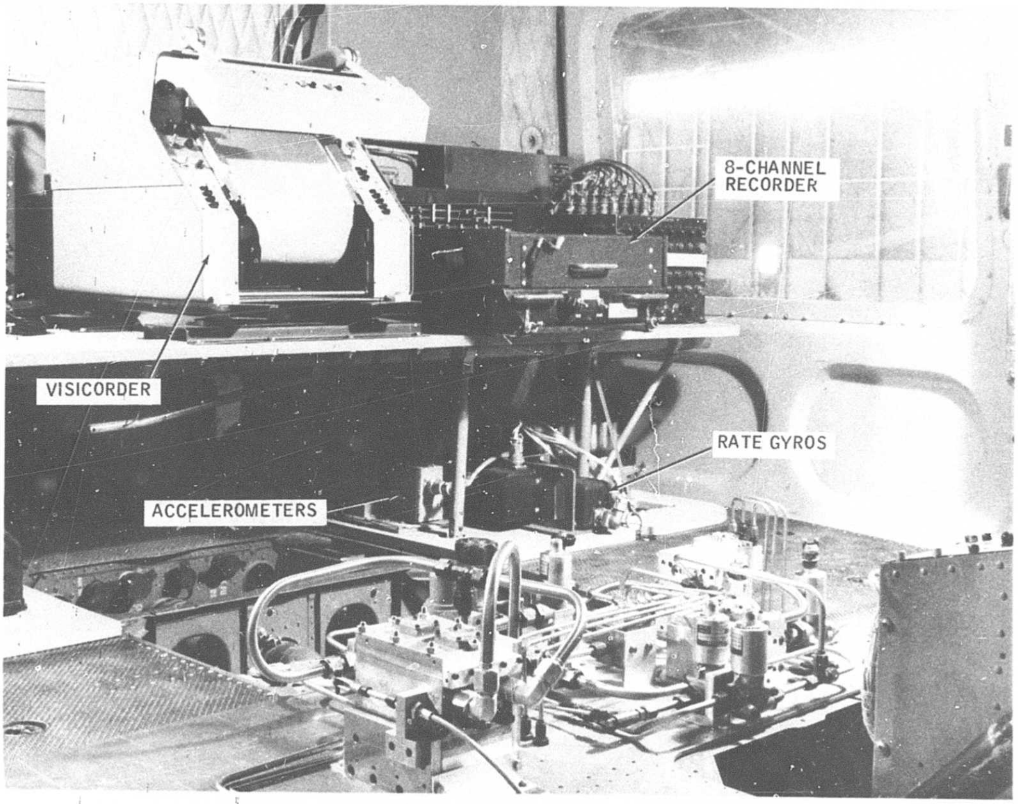


Figure 26. Instrumentation Viewed From Right Side of Aircraft.

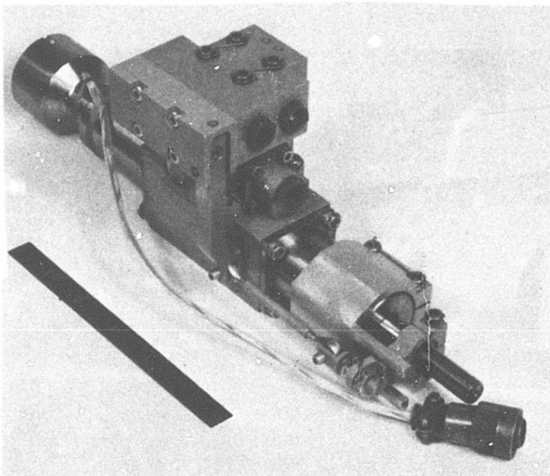


Figure 27. Spool Valve Servoactuator.

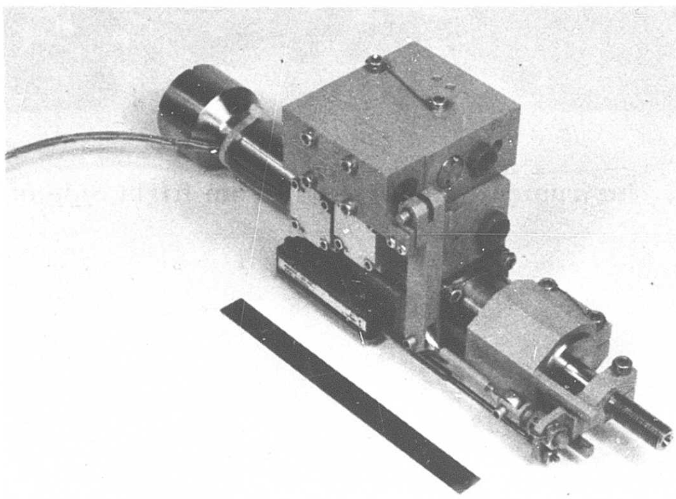


Figure 28. Vortex Valve Servoactuator.

Pilot control input motions were measured by potentiometers located under the aircraft floor and connected to the stick and pedal control linkage. The pedal position and pitch cyclic position potentiometer can be seen in Figure 29.

Pressure transducers used to measure controller fluidic signals and supply pressure can be seen in Figure 17. Also shown in Figure 17 is the turbine flowmeter used to measure the total flow used by the FSAS controllers.

Solenoid valves were used in the power circuit to provide the capability of shutting off portions of the system, or the complete system, in case of a catastrophic failure such as a line rupturing. The electrical schematic of this engage and disengage provision is shown in Figure 30. (Refer also to Figures 16 and 19.) With this circuit it was possible to engage or disengage any servoactuator independently of the others, to actuate the disengage provisions of the servoactuators from the pilot seat, copilot seat, or control panel locations, and to shut off the complete system. An additional solenoid was added when the vortex valve servoactuators were installed so that the power supply could be shut off at the discharge of the No. 2 aircraft hydraulic power supply. Figure 31 shows the layout of the switches located on the pilot console.

In addition to the 24-channel recorder that provided a record of all instrumentation outputs when the aircraft was in flight, an 8-channel, hot-pen recorder was installed to record selected parameters, including aircraft turning rates, pilot control inputs, and roll and pitch attitudes. These recordings were used to evaluate FSAS performance at the various flight conditions.

The control of the electrical supply for the instrumentation equipment, the recorder on-off switches, the 8-channel recorder speed control, and the vertical gyro cage-uncaging switch were all located on the pilot console shown in Figure 31.

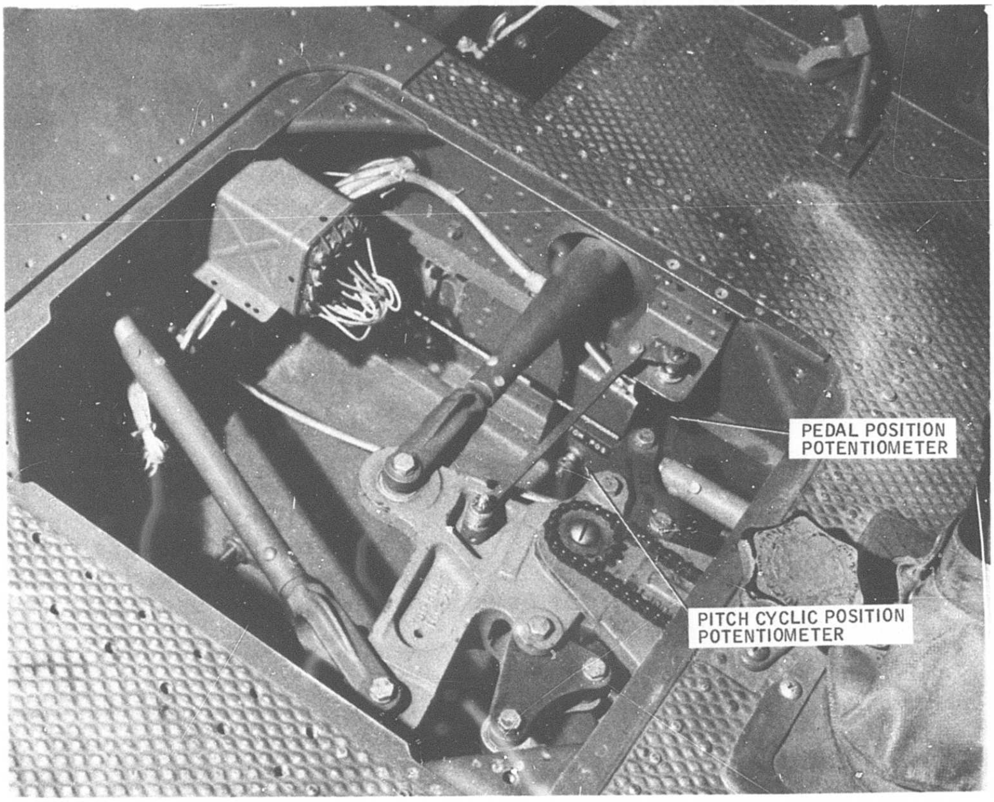


Figure 29. Pilot Control Position Potentiometers Under Pilot's Seat.

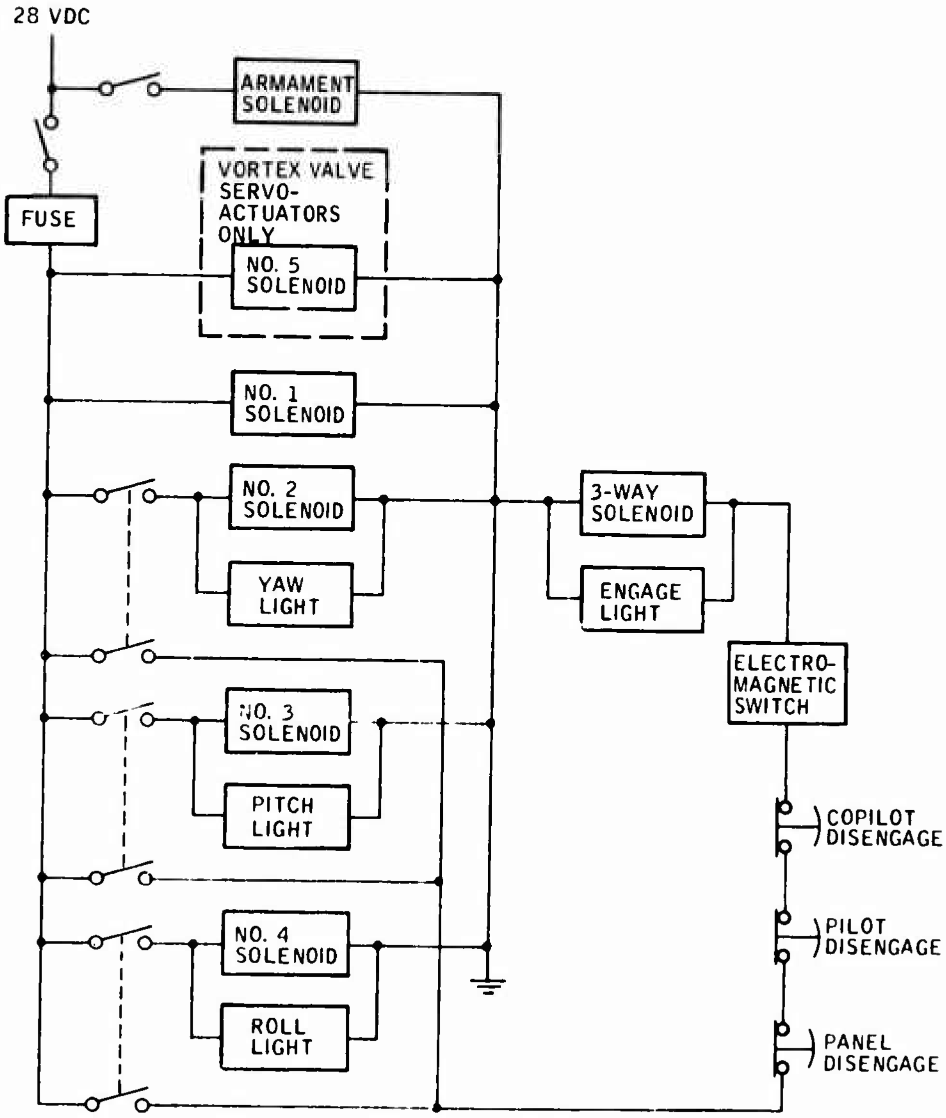


Figure 30. FSAS Operation Electrical Circuit Diagram.

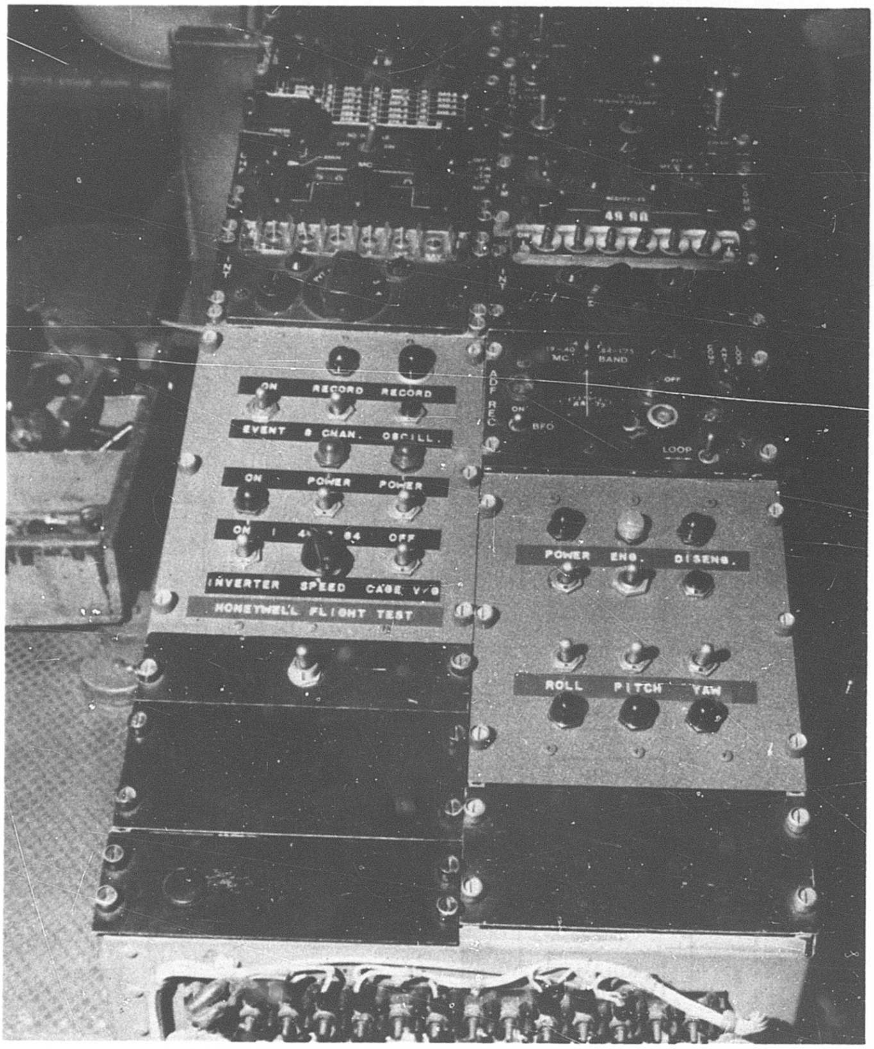


Figure 31. Pilot's Console.

SECTION IV
MODIFICATION AND PROBLEMS DURING FLIGHT TEST

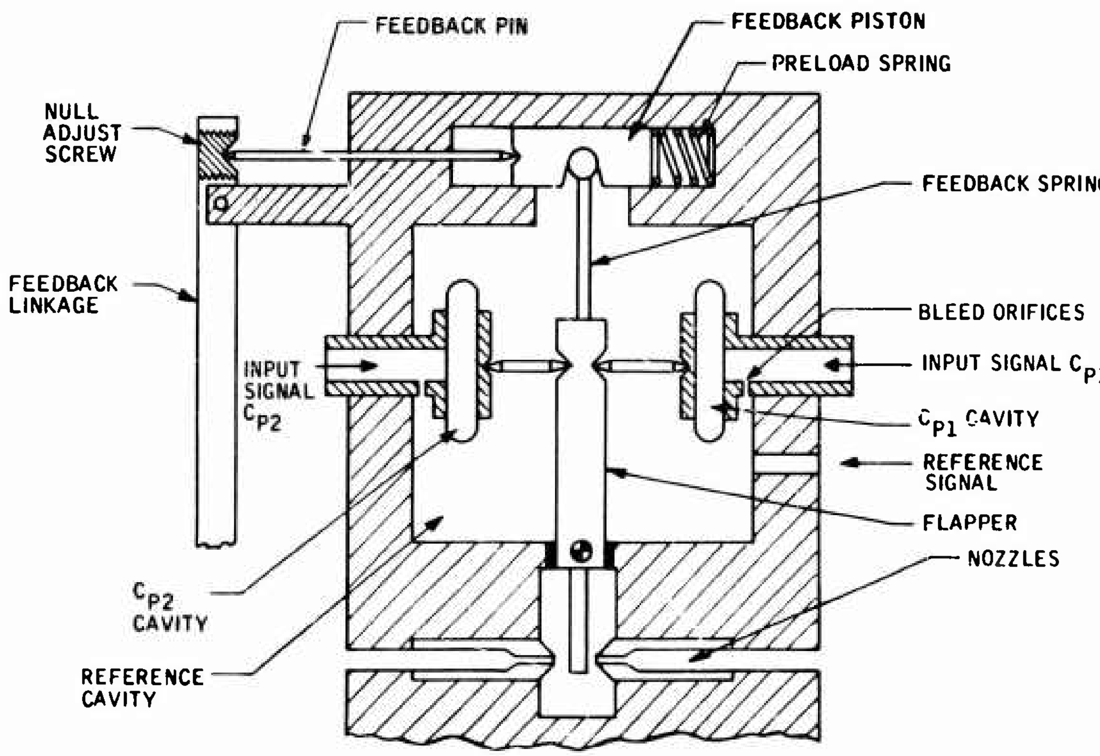
This section discusses changes and modifications made after flight testing had started.

FORCE ON SERVOACTUATOR DUE TO FLEXIBLE HOSES

During the first checkout flight, after all flight test equipment had been installed, a residual force on the stick was encountered that resulted in an aircraft roll to the left. The force was traced to five flexible hoses connected to the roll servoactuator and routed from the servoactuator toward the rear of the aircraft. It was possible to reroute two of the flexible lines forward to balance the load on the servoactuator. This reduced the force to a satisfactory level.

PRESSURE SURGES ON SPOOL VALVE SERVOACTUATORS

Initial test flights were made with the stabilizer bar on to obtain basic aircraft data for this configuration. Some preliminary checks of the FSAS were also made. It was noted that during preflight checkout of the aircraft, electrical power was interrupted twice. Because of the solenoid valves in the circuit, rather large pressure surges were being imposed on the servoactuator force capsules. (It had been assumed that pressure on the FSAS would be increased gradually by being connected to the hydraulic pump while the engine was started and run up.) These pressure surges on the servoactuators caused the yaw and roll servoactuators to fail. The failures were traced to the feedback pin (see Figure 32). During the pressure surge, the servoactuator would move in such a way that the feedback pin would move out of the null adjust screw seat. Most of the time it would slide back into position, up the sloped wall of the seat, and function satisfactorily. Examination of the pin and seat showed that the pin had gouged the seat quite badly and failed to reseat. The seat and pin were cleaned with grinding compound after which the servoactuators functioned properly. To eliminate the reoccurrence of the pressure surges during startup of the aircraft and preflight checkout, a manual-operated valve was added to the FSAS supply line just downstream of the flow control valve shown in Figure 17. This valve was opened only after the aircraft was ready for flight.



NOTE: INPUT SIGNAL AND REFERENCE SIGNAL PORTS CONNECT TO THE CONTROLLER

Figure 32. Schematic of Spool Valve Servoactuator -- Input Portion.

FSAS PERFORMANCE CHANGES WITH SPOOL VALVE SERVOACTUATORS

Analytic block diagrams of the yaw, roll, and pitch axes of the FSAS are shown in Figures 33, 34, and 35. Each axis consisted of a hydraulic fluid vortex rate sensor with the fluid output amplified, shaped, and then fed directly to the series servoactuators. The feedback signals were high-passed to eliminate damper opposition to pilot input commands. and also to minimize the effects of any component drift. Pedal position input was incorporated in the yaw-axis SAS, as shown in Figure 33, to eliminate the decrease in vehicle response to pedal inputs caused by the yaw damper. Series servoactuator authority was set at 20 percent.

The FSAS briefly described above was designed to augment the stability of the UH-1B helicopter. In addition, the FSAS was designed with the following sensor and servoactuator characteristics:

- A servoactuator with a fluid interface having a natural frequency (90-deg phase lag point on amplitude response) of 10 Hz and a damping ratio of 0. 7 was initially specified.
- System dynamics capable of generating gain and phase shift characteristics were assumed to be accounted for in a double-lagged sensor transport delay as follows:

$$T(S) = e^{-\tau S} \frac{1}{(0.05S + 1)^2}$$

During the FSAS program, the test vehicle was changed from a UH-1B to a UH-1C, and the dynamic characteristics of the sensor and servoactuator components of the flight-tested FSAS were as follows:

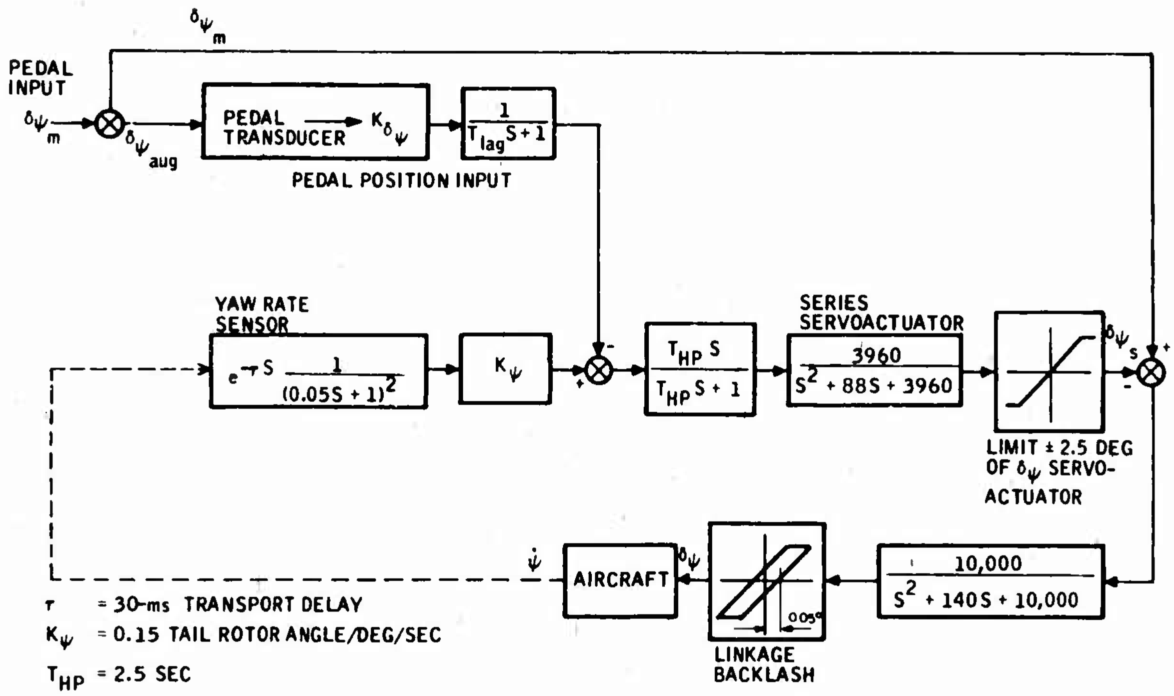
- The series servoactuator had a natural frequency (90-deg phase lag point) of 5. 5 Hz and a damping ratio of 0. 32. The actual servoactuator transfer function was:

$$T(S) = \frac{(34.6)^2}{S^2 + 22.1S + (34.6)^2} \quad \frac{1}{(0.029 + 1)}$$

- The rate sensor dynamics were a pure transport delay as follows:

$$T(S) = e^{-\tau S}$$

Changing the test helicopter to a UH-1C and using the rate sensor and servoactuator with the dynamic characteristic described above required modification of the roll and pitch controllers. These changes and the final FSAS characteristics are explained in the following sections.



τ = 30-ms TRANSPORT DELAY

K_ψ = 0.15 TAIL ROTOR ANGLE/DEG/SEC

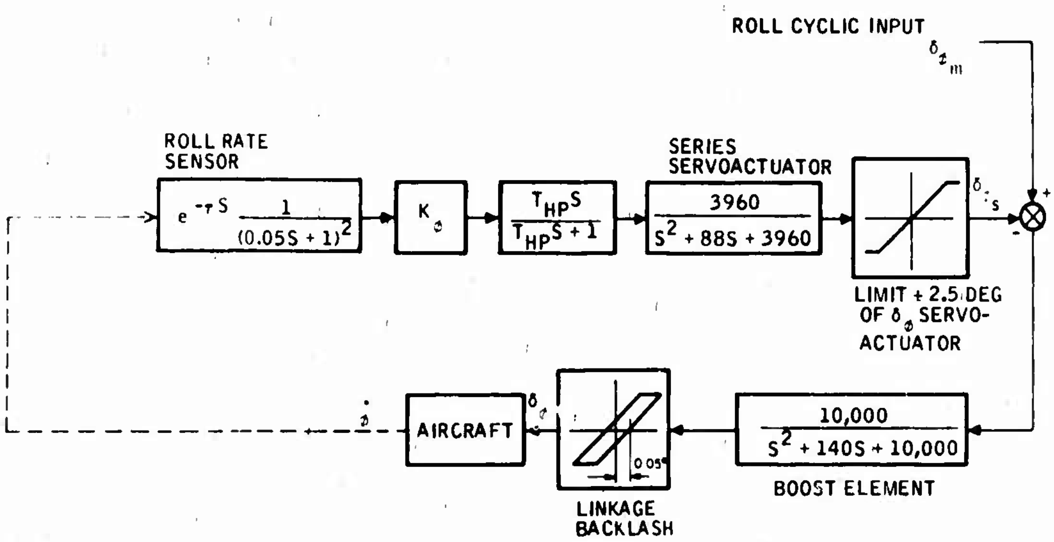
T_{HP} = 2.5 SEC

K_{δ_ψ} = 1.5 $\delta_{\psi_{aug}}/\delta_{\psi_m}$

T_{lag} = 1.0 SEC

PEDAL SENSITIVITY = 2 DEG OF TAIL ROTOR/IN. PEDAL

Figure 33. Analytic Block Diagram for UH-1B Yaw-Axis SAS.



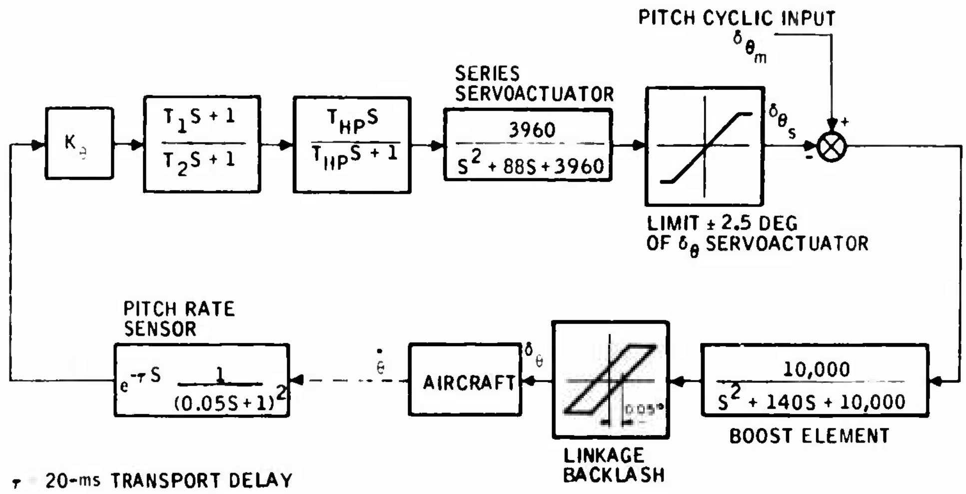
τ 20-ms TRANSPORT DELAY

K_ϕ = 0.083 DEG ROLL BLADE ANGLE/DEG/SEC

T_{HP} 11.5 SEC

ROLL CYCLIC STICK SENSITIVITY = 1.54 DEG ROLL BLADE ANGLE/IN. STICK

Figure 34. Analytic Block Diagram for UH-1B Roll-Axis SAS.



τ 20-ms TRANSPORT DELAY

K_θ 0.173 DEG PITCH BLADE ANGLE/DEG/SEC

T_1 = 0.04 SEC

T_2 = 0.107 SEC

T_{HP} 2.5 SEC

PITCH CYCLIC STICK SENSITIVITY = 1.84 DEG PITCH BLADE ANGLE/IN. STICK

Figure 35. Analytic Block Diagram for UH-1B Pitch-Axis SAS.

<u>Yaw-Axis FSAS</u>

The yaw axis was designed to improve damping of the external flight path disturbance during steady maneuvers without opposing the pilot's commands. The original control equation of the yaw axis was:

$$\delta_\psi = \left\{\delta_{\psi_m} + \left[K_{\delta_\psi}\left(\frac{1}{T_{lag}\,S+1}\right)\delta_{\psi_{aug}} - (K_\psi)\,\dot{\psi}e^{-\tau S}\left(\frac{1}{0.05\,S+1}\right)^2\right]\left(\frac{T_{HP}\,S}{T_{HP}\,S+1}\right)\left(\frac{(62.8)^2}{S^2+2(0.7)(62.8)S+(62.8)^2}\right)\right\}$$

$$\frac{(100)^2}{S^2+2(0.7)(100)S+(100)^2}$$

where

δ_{ψ_m} = mechanical tail rotor command from pilot's pedals, deg

δ_ψ = total tail rotor deflection, deg

$\delta_{\psi_{aug}}$ = augmented servoactuator command from pilot's pedals, deg

$K_{\delta_{\psi}}$ = 1.5 augmented servoactuator command/mechanical tail rotor command, in./in.

T_{lag} = 1.0 sec

K_{ψ} = 0.15 tail rotor angle/aircraft yaw-axis turning rate, deg/deg/sec

τ = 0.02 sec

$\dot{\psi}$ = yaw rate, deg/sec

T_{HP} = 2.5 sec

The yaw-axis aerodynamic characteristics of the UH-1B and UH-1C are similar; therefore, changes in the series servoactuator and sensor dynamics mentioned above did not significantly deteriorate the gain bandwidth of the yaw-axis FSAS. As a result, no design modifications were required to the yaw system during the flight test program.

Roll-Axis FSAS

The roll-axis stability augmentation system was designed to decrease the control sensitivity of the UH-1B helicopter without mechanical stabilizer bar to the level of control sensitivity characterized by the UH-1B helicopter with mechanical stabilizer bar. The roll rate feedback was high-passed to provide a long-term trim on the roll FSAS output. The original control equation for the roll FSAS was:

$$\delta_{\phi} = \left\{ \delta_{\phi_m} - K_{\phi\dot{\phi}}\, e^{-\tau S} \left(\frac{1}{0.05\ S+1} \right)^2 \left(\frac{T_{HP}\ S}{T_{HP}\ S+1} \right) \left(\frac{(62.8)^2}{S^2+2(0.7)(62.8)^2} \right) \right\} \frac{(100)^2}{S^2+2(0.7)(100)S+(100)^2}$$

where

K_{ϕ} = 0.055 deg roll cyclic blade angle/deg/sec

T_{HP} $\geq$ 10 sec

τ = 0.02 sec

δ_{ϕ_m} = mechanical roll cyclic command from pilot's stick, deg

δ_{ϕ} = total roll cyclic blade deflection, deg

The roll-axis aerodynamic characteristics of the UH-1B and UH-1C are quite similar, and the desired level of control sensitivity was achieved by increasing the roll rate gain by 50 percent. This was accomplished by reducing the size of the bleed orifices in the preamplifier. The schematic of the roll-axis controller is shown in Figure 36.

Frequency response of the controller after the gain was increased is shown in Figure 37. The shaping network was not changed. The final value of FSAS roll rate gain established during flight test was:

$$K_{\phi} = 0.083 \text{ deg roll cyclic blade angle/deg/sec}$$

The changes in the sensor and series servoactuator dynamics mentioned previously did not significantly affect roll FSAS performance.

Pitch-Axis FSAS

The pitch-axis FSAS was designed to increase vehicle damping ratio without significantly affecting the control power of the UH-1B. The original control equation of the pitch FSAS was:

$$\delta_{\theta} = \left\{ \delta_{\theta_m} - K_{\theta}\dot{\theta}\ e^{-\tau S} \left(\frac{1}{0.05S+1}\right)^2 \left(\frac{T_{HP}S}{T_{HP}S+1}\right) \left(\frac{T_1 S+1}{T_2 S+1}\right) \left(\frac{(62.8)^2}{S^2+2(0.7)(62.8)S+(62.8)^2}\right) \right\} \frac{(100)^2}{S^2+2(0.7)(100)S+(100)^2}$$

where

K_{θ} = 0.25 pitch blade angle/aircraft pitch-axis turning rate, deg/deg/sec

T_{HP} = 1.5 sec

T_1 = 0.25 sec

T_2 = 0.1 sec

τ = 0.02 sec

δ_{θ} = total pitch cyclic blade angle, deg

δ_{θ_m} = mechanical pitch cyclic command from pilot's stick, deg

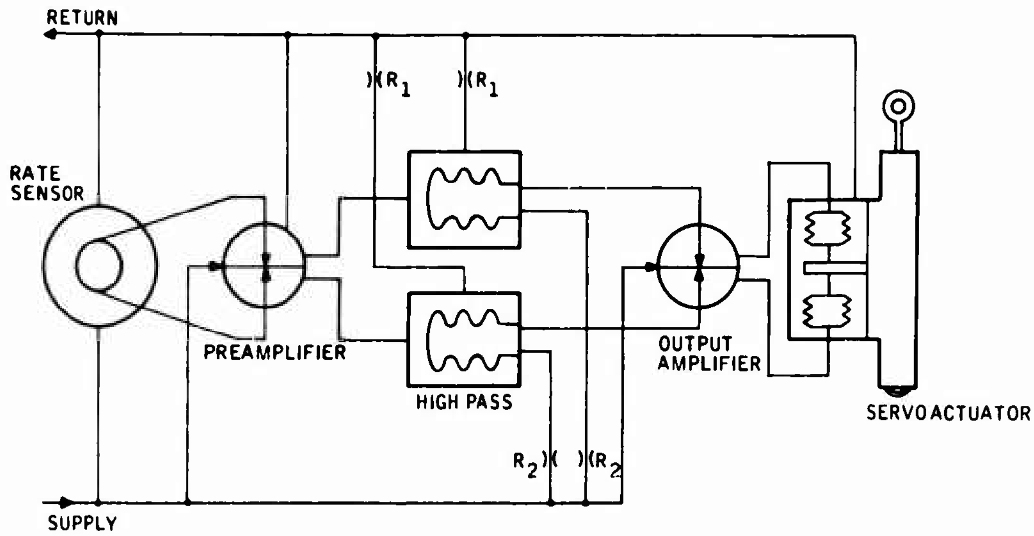


Figure 36. Roll-Axis Controller Schematic.

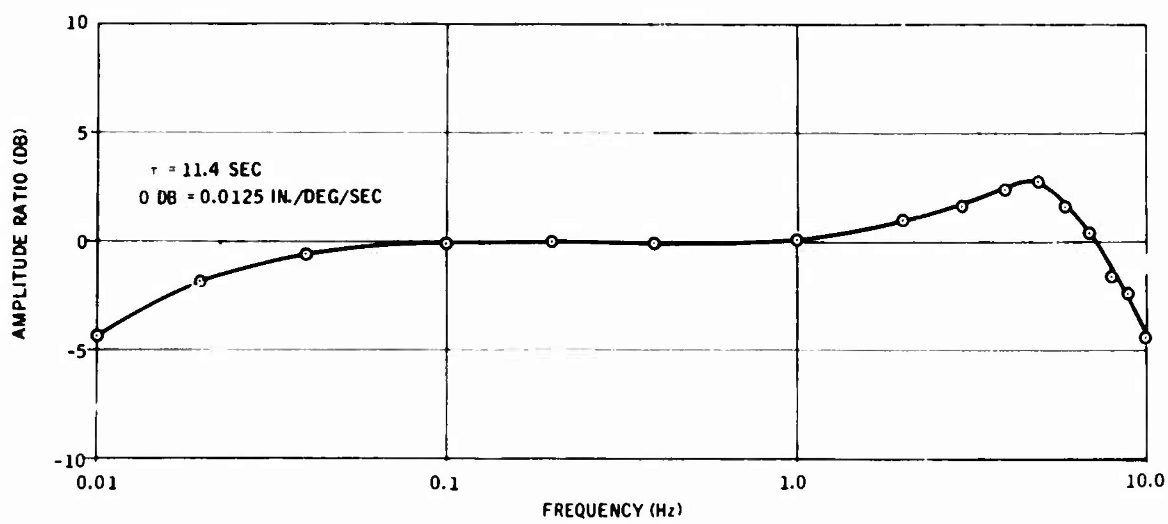


Figure 37. Frequency Response of Roll-Axis Controller After Gain Increase.

The pitch rate feedback was high-passed to avoid damper opposition to the relatively low-frequency pilot control inputs. Also, the original pitch FSAS was designed so that the high-frequency gain was sharply attenuated above 3 Hz. This gain attenuation function was performed by the double-lag dynamics of the rate sensor. Absence of the double-lag sensor dynamics caused the high-frequency gain of the pitch FSAS to be increased rather than attenuated. In addition, the 0.32 damped, 5.5-Hz servoactuators caused a further increase in the high-frequency gain, especially in the area of 6 Hz, the UH-1C's rotor frequency. Because of these dynamic conditions, the pitch-axis controller could not be engaged above an airspeed of approximately 70 kn, where a severe vertical oscillation resulted. The Phase II closed-loop testing had failed to show this problem, as the rotor characteristics were not in the simulation. The solution was to change the lead-lag shaping network to a lag-lead. This circuit was examined on the analog computer to verify its effect on aircraft performance. The change was then incorporated in the controller.

Schematics of the controller before and after modification are shown in Figures 38 and 39, respectively. Frequency response of the controller before and after modification is shown in Figures 40 and 41, respectively. This entire operation was accomplished in one 8-hour day, demonstrating the versatility of the controller package.

With this modification it was then possible to fly at speeds up to 110 kn without any adverse effects. Above 110 kn, the vertical oscillation again became excessive. The controller-servoactuator gain was then reduced 30 percent to 0.026 in. servoactuator/deg/sec. Frequency response data are shown in Figure 42.

The pitch-axis FSAS control equation then became:

$$\delta_\theta = \left\{ \delta_{\theta_m} - K_\theta \dot{\theta}\, e^{-\tau S} \left(\frac{T_{HP}S}{T_{HP}S+1}\right)\left(\frac{T_1S+1}{T_2S+1}\right)\left[\frac{(34.6)^2}{S^2+2(0.32)(34.6)S+(34.6)^2}\right] \frac{1}{0.029S+1} \right\} \frac{(100)^2}{S^2+2(0.7)(100)S+(100)^2}$$

where

K_θ = 0.173 deg/deg/sec

T_1 = 0.04 sec

T_2 = 0.107 sec

τ = 0.02 sec

T_{HP} = 2.5 sec

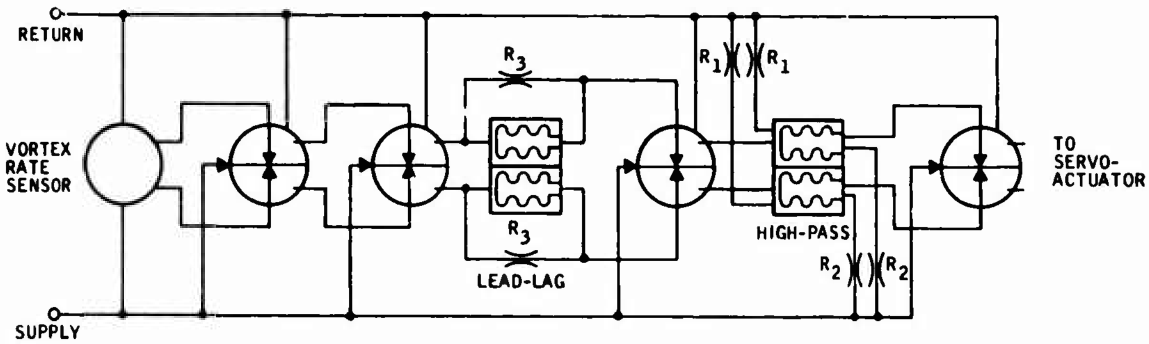


Figure 38. Schematic of Pitch-Axis Controller Before Modification.

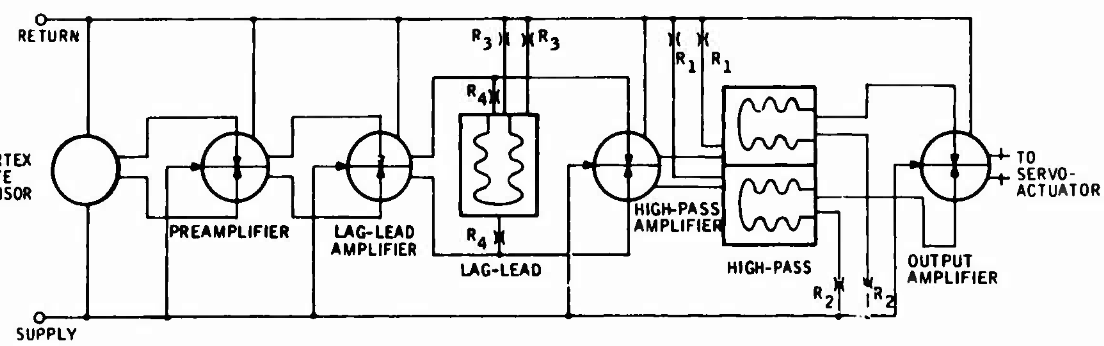


Figure 39. Schematic of Pitch-Axis Controlle After Modification

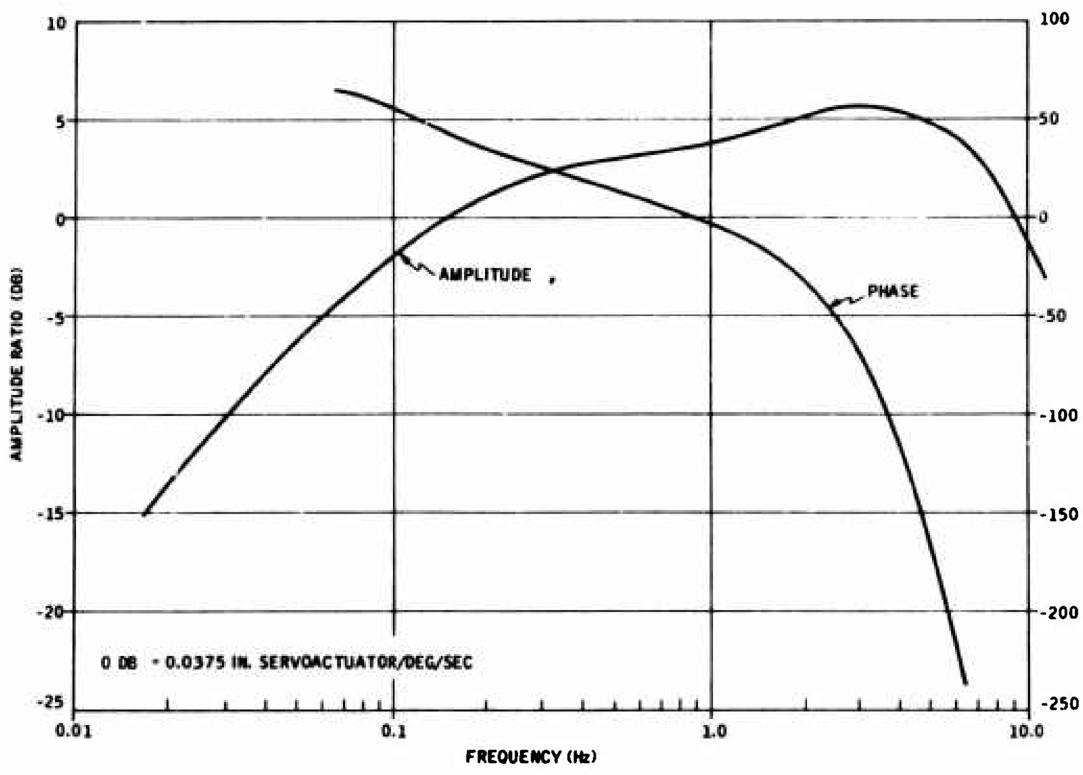


Figure 40. Frequency Response of Pitch-Axis Controller Before Modification.

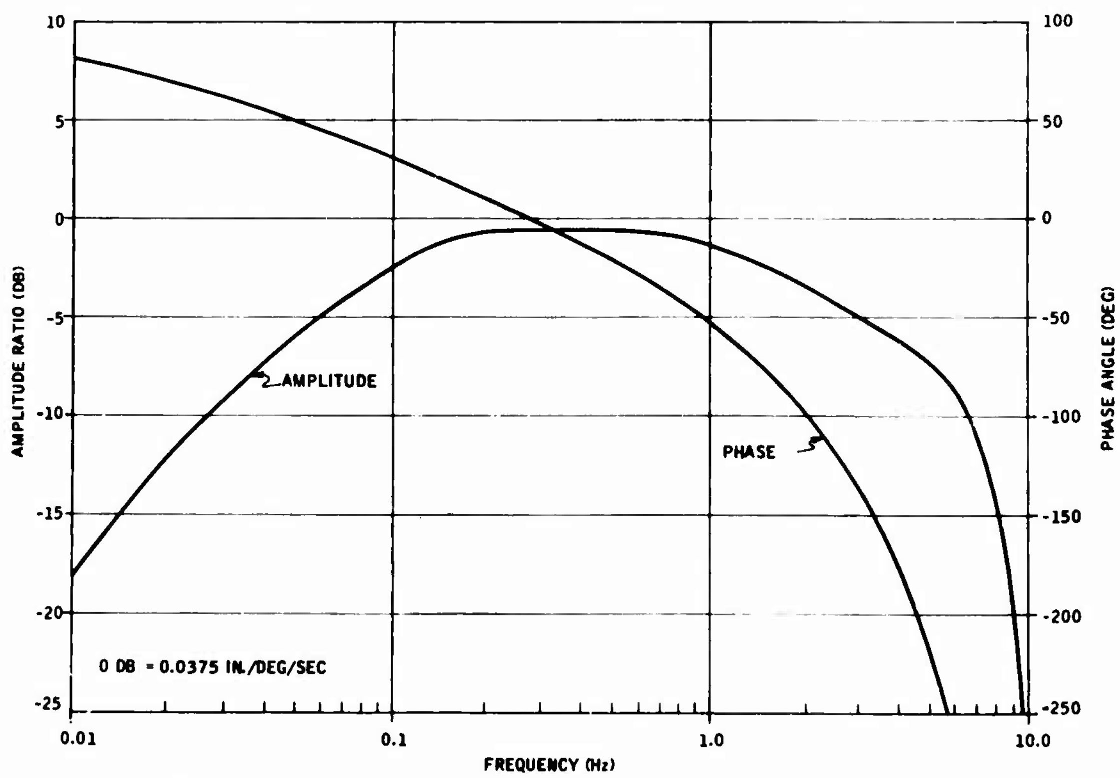


Figure 41. Frequency Response of Pitch-Axis Controller After Modification

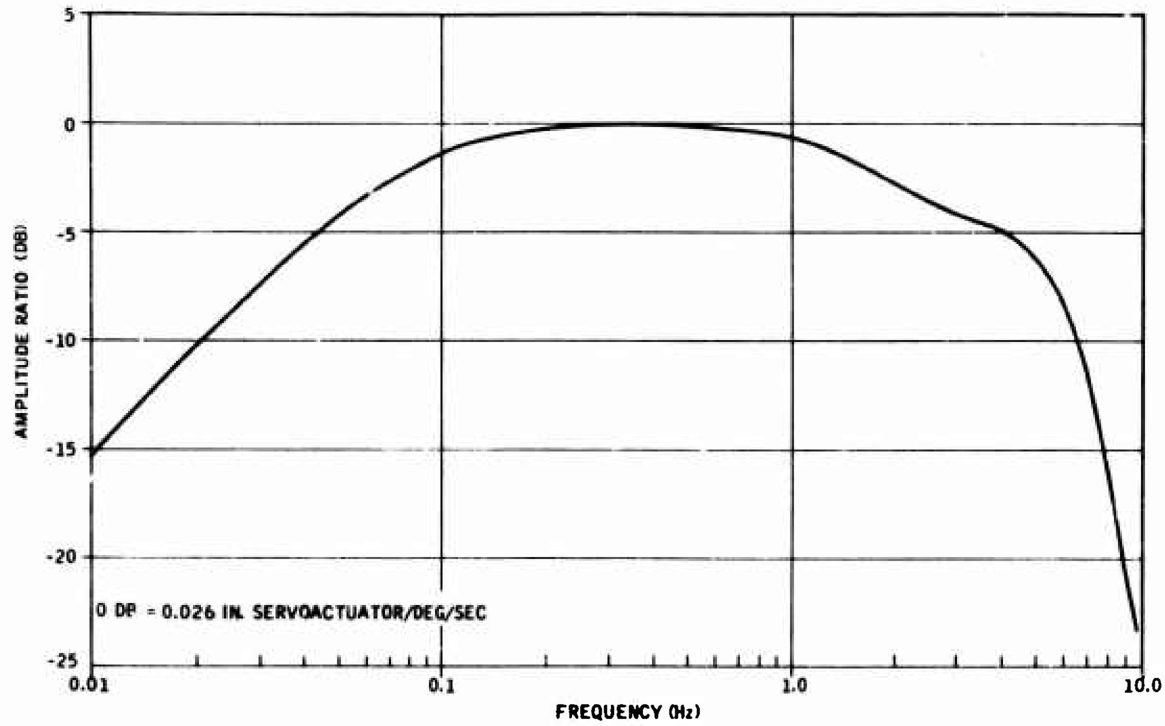


Figure 42. Frequency Response of Pitch-Axis Controller After Gain Reduction.

TABLE II. FINAL FSAS CONTROLLER - SERVOACTUATOR PERFORMANCE

Controller	Gain	Equation
Pitch-Axis	0.26 psi/deg/sec	$e^{-0.02S}\left(\frac{2.5S}{2.5S+1}\right)\left(\frac{0.04S+1}{0.107S+1}\right)$
Roll-Axis	0.125 psi/deg/sec	$e^{-0.02S}\left(\frac{11.5S}{11.5S+1}\right)$
Yaw-Axis	0.22 psi/deg/sec	$e^{-0.02S}\left(\frac{2.5S}{2.5S+1}\right)$
	6.9 psi/in. cable	$\frac{1}{S+1}$
Servoactuator	0.1 in./psi	$\left(\frac{1}{0.029S+1}\right)\left(\frac{(34.6)^2}{S^2+2(0.32)(34.6)S+(34.6)^2}\right)$

These changes resulted in pitch-axis FSAS performance that satisfied the requirements of the design goals. The FSAS was engaged without difficulty up to the 140-kn limit.

The flight test program was completed with the FSAS controllers and series servoactuator characteristics shown in Table II.

NULL OFFSETS

The only problem encountered with the hydrofluidic controllers was a null offset occurring in the roll-axis controller. The offset was caused by shreaded bits of Teflon tape lodging in the output bleed orifices. The tape had been used to seal the threaded output fittings. After the Teflon tape was completely removed from the fittings, the null problem did not reoccur.

FLIGHT TEST USING VORTEX VALVE SERVOACTUATORS

After the flight test program with spool valve servoactuators was completed, these servoactuators were removed and replaced with vortex valve servoactuators supplied as GFE by USAAMRDL.

Pressure Surge Problems

The yaw-axis servoactuator was damaged during startup of the aircraft when part of the aircraft preflight was done after the manually operated valve had been opened (see Figure 19 for circuit details). The aircraft has two hydraulic systems, and it is possible to use each one independently to supply the boost servoactuator power. In so doing, the No. 2 system is shut off with a solenoid valve while the No. 1 system is being checked, and vice versa. This checkout procedure resulted in a pressure surge being applied to the servoactuator input force capsule, thus rupturing it.

The vortex valve servoactuators are slightly different in their input configuration than the spool valve servoactuators. As shown in Figure 43, the cavities for the signal inputs for the vortex valve servoactuators are dead ended, while the spool valve servoactuator (Figure 32) has a bleed orifice between the input signal line and reference signal line. There is no steady-state flow through the lines from the controller to the vortex valve servoactuator, as in the spool valve servoactuator, due to the lack of bleed orifices. Also, the cavity designated C_{p2} is quite large compared with cavity C_{p1}. These differences make it extremely difficult in the aircraft installation to prevent the accumulation of air, or removal of trapped air from the lines and cavities. Therefore, assuming trapped air in cavity C_{p2} of the vortex valve servoactuator, prior to startup, a finite time will be needed to compress this air to the operating pressure (350 psi in the yaw-axis servoactuator), while cavity C_{p1}, which has no

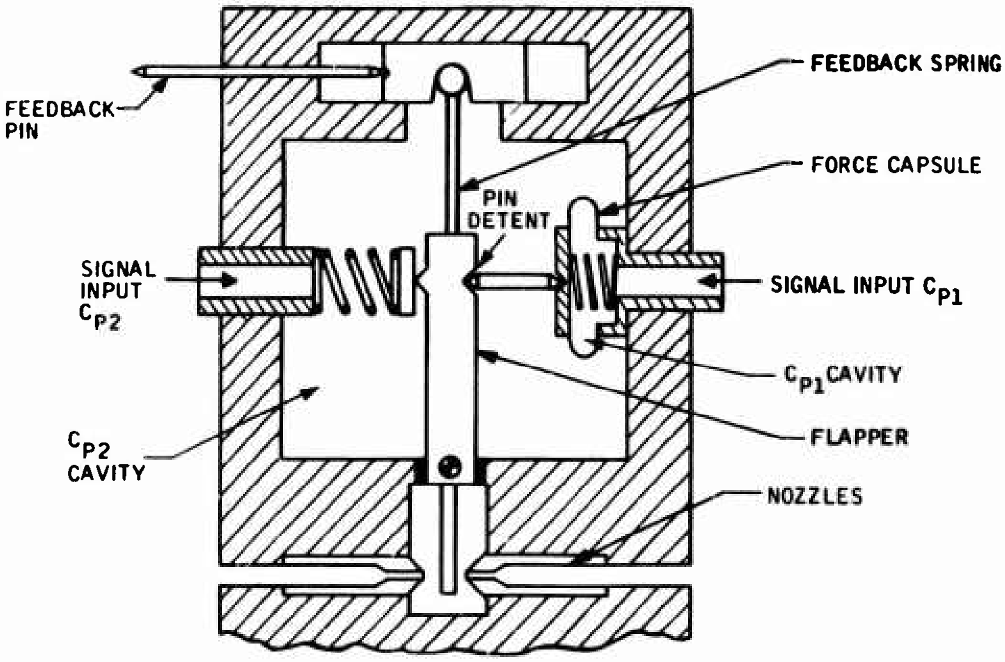


Figure 43. Schematic of Vortex Valve Servoactuator -- Input Portion.

trapped air, will charge up almost immediately upon pressurization. The differential pressure that develops across the force capsule can destroy it.

One other failure occurred with the vortex valve servoactuators. During pressurization of the system using the hand valve, the pitch servoactuator went hard-over in the retracted direction. The servoactuator was removed and tested in the laboratory and exhibited the same hard-over with zero differential pressure applied to the signal ports. Removal of the bias springs, shown in Figure 43, did not eliminate the hard-over condition. The pin connecting the force capsule to the flapper was ultimately found to be exerting considerable force on the flapper, thus causing the hard-over. The pin was moved and it snapped to a new position. It appeared that the pin had been hung up on the side of the detent in the flapper. With more care taken during system pressurization, the servoactuator performed properly during the rest of the flight test program.

The pins and seats on future servoactuators should be polished so that if the pin slides partially out of the detent, it will slide back to the proper position again.

Vortex Valve Servoactuator Performance

The flight test recordings disclosed that the vortex valve servoactuators did not linearly follow the SAS fluidic output commands. It could be seen that the servoactuators were not moving for low-amplitude, low-frequency signals until the input differential pressure from the FSAS controller increased to approximately 0.5 psi. This is equivalent to a servoactuator motion of approximately 0.05 in. In some flight conditions, the duration of this deadband would be 1 sec or longer, depending on the rate buildup of the FSAS controllers. Servoactuator thresholds of this type cause the helicopter to respond as though it were augmented up to the time the servoactuator starts to move. However, even after the initial servoactuator movement took place, it usually continued to move in discrete steps rather than smoothly proportional to the controller commands.

These servoactuator characteristics cause the helicopter to take on response characteristics which are, in general, different or variable from test to test or maneuver to maneuver. Stability augmentation systems containing servoactuators with the characteristics described above will provide aircraft control responses that lie somewhere between the free aircraft and the nominally augmented aircraft. This variable response-type of control will cause the pilot to fly with less precision and control during tracking maneuvers and may result in unsafe flight conditions when the helicopter is flown at low altitudes.

Because the vortex valve servoactuators were not functioning in a manner required by the FSAS design specifications, a detailed evaluation of the FSAS/vortex valve servoactuactor performance in compliance with the design goals was not attempted.

The vortex valve servoactuators were tested in the laboratory after the flight test program was completed. A load of approximately 20 lb would make the ram move approximately 0.05 in. If the friction level were this amount, the servoactuator would be prevented from moving until it developed enough force to overcome this load. The spool valve servoactuators were also tested and did not exhibit this problem because they move only 0.0025 in. for a load of approximately 20 lb. If the pressure gain of the vortex valve servoactuators were higher, such as in the spool valve servoactuators, the problem would probably be eliminated.

UH-1C HELICOPTER CHARACTERISTICS

During some of the flights conducted at 60 kn, high-level rotor vibrations at about one-half per revolution were detected. These vibrations occurred most frequently when flying the UH-1C without the mechanical stabilizer bar.

A check with the helicopter manufacturer disclosed that this problem was called "pylon rock" and may be caused by:

- Deteriorating pylon friction mounts; worn pylon dampers
- Cracked support structure
- Loose collective sleeve device on the mast causing intermittent one-per-revolution random hop.

Preventive maintenance was performed to remedy this problem. However, this problem reoccurred, and it was decided that those tests scheduled for 60 kn should be flown at 70 kn to avoid this portion of the flight envelope where the helicopter is susceptible to the pylon rock problem.

DATA REDUCTION

Several factors complicated evaluation and correlation of FSAS performance with free-aircraft performance:

- Weather conditions at the time of test. To complete the program on schedule, flight testing was conducted on all flyable days. Data flights were conducted in weather conditions ranging from clear, with winds less than 3 kn, to freezing drizzle with wind 20 kn gusting to 30 kn.
- Consistent and repeatable pilot inputs could not be made for each test. The pilot input test brackets were made of lightweight aluminum and would bend or flex under pilot-applied forces. This required the conversion of all performance data to a common base of "per in. of stick or pedal."
- Test-bracket bending had a serious impact on the pitch pulse tests. Control stick pulses were not made with the magnitudes specified for each test, and the control stick was not returned to the trim position that existed at the start of the test. This caused the helicopter to fly to a new trim position, and this commanded maneuver masked the damping action of the pitch FSAS. Similar characteristics were noted for the yaw-axis tests; but, because of the underdamped characteristics of the free helicopter's yaw axis and because the velocity and attitude of the helicopter were not grossly affected by the tests, yaw-axis damping performance could be measured.

These factors made it extremely difficult to accurately and consistently determine quantitative data for evaluating certain performance measures. Performance measures most seriously affected were response time ($t_{90\%}$), control power, and pitch-axis damping.

Cross-coupled pilot-commanded inputs in the two axes, which were not the test axes, also had a masking effect on the performance response in the axis being tested.

SECTION V
UH-1C/FSAS FLIGHT TEST RESULTS

TEST PROCEDURE

The three-axis FSAS was installed in a UH-1C helicopter and was evaluated both quantitatively and qualitatively over the flight envelope and conditions shown in Table I.

FSAS quantitative performance was established by analyzing flight test tracings recorded at each flight condition for the following tests:

- Pilot-commanded pulses and steps in pitch, roll, and yaw axes
- Stabilized flight
- FSAS engage/disengage transients
- Autorotation entries
- Trim maneuvers

These tests were performed for four helicopter/SAS configurations as follows:

- UH-1C helicopter without mechanical stabilizer bar, with FSAS engaged in pitch, roll, and yaw axes, and with spool valve servoactuators
- UH-1C helicopter without mechanical stabilizer bar, with FSAS engaged in pitch, roll, and yaw axes, and with vortex valve servoactuators
- UH-1C helicopter without mechanical stabilizer bar, with FSAS disengaged
- UH-1C helicopter with mechanical stabilizer bar, with FSAS disengaged

Qualitative evaluation was conducted by the project pilot, two Army pilots, and pilots from two airframe manufacturers.

All flight testing was conducted on a UH-1C helicopter with a gross weight of less than 8000 lb and a mid-center-of-gravity loading condition.

YAW FSAS PERFORMANCE

Yaw FSAS performance was evaluated for the four vehicle configurations and flight conditions listed in Table I. The principal criteria for evaluating yaw FSAS 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 FSAS is engaged. A small decrease in control sensitivity is permissable providing it has the effect of giving pilots 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 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 which simulate wind disturbances acting on the helicopter.

Pedal Step Inputs - Yaw FSAS With Spool Valve Servoactuator

Flight recordings for the pedal step input tests are presented in Appendix II, Figures 44 through 53. Performance results taken from these recordings are presented in Table III.

A comparison of the recordings for the FSAS with those of the free helicopter, with and without the mechanical stabilizer bar, shows that the FSAS virtually eliminated the underdamped yaw rate response. The augmented helicopter responds with a yaw rate which is easy to control and exhibits one overshoot to pedal step input commands. This is generally true for the complete operational flight envelope of the UH-1C.

Control sensitivity data taken from Figures 44 through 53 are presented in Table III. These data show that the control sensitivity of the UH-1C helicopter is retained when the FSAS is engaged. Step input response time data are also presented in Table III. Response time is defined as the time, in seconds per inch of pedal, to reach 90 percent of the peak yaw rate. The summarized data show that yaw rate response time was not changed significantly by the FSAS.

The project pilot commented that he could not detect any difference in helicopter handling characteristics in response to step inputs with the FSAS engaged or disengaged. This is the way it should be; the yaw FSAS was designed with a pedal feed forward loop which cancels the damping function of the SAS for pilot command inputs.

TABLE III. YAW FSAS/UH-1C PERFORMANCE -- STEP INPUTS

UH-1C/FSAS Configuration	Test (Right Step or Left Step)	Control Sensitivity, $\dot{\psi}_{peak}$/in. of pedal (deg/sec/in.)	Response Time, $t_{90\%}$ (sec)
Hover, 3000 Ft			
SV Servo, W/O Stab	RS	13.0	0.54
	LS	16.5	0.83
VV Servo, W/O Stab	RS	15.0	0.64
	LS	12.7	0.45
W/O Stab	RS	14.5	0.70
	LS	16.7	1.00
With Stab	RS	13.0	0.60
	LS	13.5	0.70
60 Kn, 3000 Ft			
SV Servo, W/O Stab	RS	10.5	0.50
	LS	10.0	0.50
VV Servo, W/O Stab	RS	11.7	0.25
	LS	10.4	0.26
W/O Stab	RS	15.0	0.36
	LS	19.0	0.57
60 Kn, 5000 and 10,000 Ft			
SV Servo, W/O Stab	RS*	9.0	0.42
	LS*	8.9	0.45
VV Servo, W/O Stab	RS*	7.6	0.25
	LS*	9.5	0.40
W/O Stab	RS*	14.0	0.59
	RS**	13.4	0.58
	LS**	10.6	0.59
90 Kn, 3000 Ft			
SV Servo, W/O Stab	RS	12.5	0.40
	LS	9.4	0.30
VV Servo, W/O Stab	RS	9.6	0.20
	LS	8.7	0.20
W/O Stab	RS	13.6	0.32
	LS	13.7	0.46
With Stab	RS	13.2	0.43
	LS	11.0	0.50
120 Kn, 3000 Ft			
SV Servo, W/O Stab	RS	8.1	0.50
	LS	8.8	0.33
VV Servo, W/O Stab	RS	7.4	0.30
	LS	7.7	0.26
W/O Stab	RS	10.2	0.69
	LS	9.4	0.35

*Altitude 10,000 ft

**Altitude 5000 ft

Pedal Step Inputs - Yaw FSAS With Vortex Valve Servoactuator

Flight recordings for the vortex valve servoactuators plus FSAS combination pedal step input tests are also presented in Appendix II. Figures 44 through 53 show that yaw rate response for these tests was underdamped and jerky. These undesirable response characteristics were caused by servoactuator threshold nonlinearities. That is, the servoactuator did not linearly follow FSAS commands, but moved in discrete steps. As the flight test program progressed, this servoactuator problem became worse, and FSAS performance, with vortex valve servoactuators, deteriorated significantly. This problem area is covered in greater detail in Section IV.

Pedal Pulse Inputs - Yaw FSAS With Spool Valve Servoactuator

Flight recordings for the pedal pulse tests are presented in Figures 54 through 63. Damping and performance results taken from these recordings are presented in Table IV.

The flight recordings show that with the FSAS engaged, the oscillatory or underdamped yaw rate response of the free helicopter is changed to a well-damped rate response. Damping ratio data summarized in Table IV show the following improved yaw-axis performance:

- At hover and 120 kn, yaw-axis damping ratio was increased from approximately 0.25 to 0.6 or 0.7.

- At 60 and 90 kn, yaw-axis damping ratio increased from 0.15 to 0.7.

This improved yaw-axis damping performance was noted by all pilots who participated in the flight test program.

Pedal Pulse Inputs - Yaw FSAS With Vortex Valve Servoactuator

Flight recordings for the vortex valve servoactuator FSAS tests are shown in Figures 54 through 63. The recordings show that, at times, yaw rate response was underdamped (refer to Figures 54, 58, 60, and 62). This is attributed to sticking of nonlinear servoactuators. Because of these servoactuator thresholds, a yaw rate disturbance was not damped in a linear manner, and the helicopter's yaw axis was allowed to drift until yaw rate reached a level that resulted in a FSAS output command that overcame the servoactuator thresholds.

TABLE IV. YAW FSAS/UH-1C PERFORMANCE -- PULSE INPUTS

UH-1C/FSAS Configuration	Test (Right Pulse or Left Pulse)	Frequency, ω (rad/sec)	Damping (ζ)
Hover, 3000 Ft			
SV Servo, W/O Stab	RP	0.00	>0.70
	LP	-	0.60
VV Servo, W/O Stab	RP	2.00	0.35
	LP	-	0.40
W/O Stab	RP	-	*
	LP	1.60	0.26
With Stab	RP	1.65	0.35*
	LP	1.90	0.23
60 Kn, 3000 Ft			
SV Servo, W/O Stab	RP	-	0.78
	LP	-	0.65
VV Servo, W/O Stab	RP	-	>0.70
	LP	-	0.70**
W/O Stab	RP	1.70	0.20
	LP	2.00	0.15
60 Kn, 5000 and 10,000 Ft			
SV Servo, W/O Stab	RP†	-	0.68
	RP†	-	>0.70
	LP†	-	0.62
VV Servo, W/O Stab	RP†	-	>0.70
	LP†	-	0.40
W/O Stab	RP††	1.70	0.18
	LP††	2.00	0.19
	LP††	1.80	0.17
90 Kn, 3000 Ft			
SV Servo, W/O Stab	RP	0	>0.70
	LP	-	0.60
VV Servo, W/O Stab	RP	-	0.70
	LP	-	0.4-0.5
W/O Stab	RP	1.85	0.17
	LP	2.19	0.22
With Stab	RP	2.10	0.17
	LP	2.00	0.22
120 Kn, 3000 Ft			
SV Servo, W/O Stab	RP	-	0.55
	RP	-	0.70
	LP	-	0.70
	LP	-	0.70
VV Servo, W/O Stab	RP	-	-
	LP	-	0.40
W/O Stab	RP	2.29	0.20
	LP	2.40	0.24

*During this test, cross-coupled commands in pitch and roll axes are masking yaw rate response.

**Because of underdamped vehicle characteristic in response to right yaw rates, damping response is masked by vehicle drift characteristic. Damping is estimated to be 0.7.

†Altitude 10,000 ft

††Altitude 5000 ft

Damping ratio data of Table IV show that an FSAS with vortex valve servoactuators would provide random damping augmentation to the helicopter yaw axis, with damping ratios ranging from 0.3 to 0.7.

ROLL FSAS PERFORMANCE

Roll FSAS performance was evaluated for the four vehicle configurations and flight conditions listed in Table I.

Roll FSAS performance was evaluated against the following criteria:

- Control sensitivity of the FSAS/UH-1C without mechanical stabilizer bar should be equal to or comparable to that of the UH-1C with 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 which prevents pilot maneuvering difficulty.

Flight recordings demonstrating improved roll FSAS/UH-1C performance satisfying these criteria are presented in Figures 64 through 73. Roll FSAS performance data taken from these recordings are presented in Table V.

Roll FSAS With Spool Valve Servoactuator

A review of these recordings shows that when the roll rate responses of the FSAS-augmented UH-1C and the helicopter with and without the mechanical stabilizer bar are compared, the roll-axis handling qualities are greatly improved by the FSAS. This improvement is due to the FSAS holding roll rate constant for a given roll cyclic input command. This gives the pilots the capability to perform more precise and easily controlled roll maneuvers without chasing a wandering roll rate. In addition, large rate overshoots were eliminated, which allows the pilots to achieve and maintain a desired roll attitude. Flight recordings of Figures 64 through 73 show that this improved roll-axis performance was provided over the entire UH-1C operational airspeed envelope.

At the hover flight condition, the control sensitivity of the FSAS-augmented UH-1C was 7.1 and 5.6 deg/sec/in. roll cyclic. This compares favorably with the control sensitivities of 6.7 and 7.0 for the UH-1C with mechanical stabilizer bar. A review of the control powers recorded for these two vehicle combinations shows that the inherent control power of approximately 7.5 was maintained on the FSAS-augmented helicopter.

TABLE V. ROLL FSAS/UH-1C PERFORMANCE -- STEP INPUTS

UH-1C/FSAS Configuration	Test (Right Step or Left Step)	Control Sensitivity ϕ_{ss}/ in. of stick (deg/sec/in.)	Control Sensitivity $\dot{\phi}_{peak}$/ in. of stick (deg/sec/in.)	Response Time, $t_{90\%}$ (sec)	Over-shoot (%)	Control Power, ϕ/ in. of stick (deg/in.)
Hover, 3000 Ft						
SV Servo, W/O Stab	RS	7.1	-	0.94	0	7.5
	LS	5.6	8.1	0.63	40.0	7.5
VV Servo, W/O Stab	RS	-	7.4	0.33	-	5.3
	LS	-	7.5	0.31	-	6.9
W/O Stab	RS	-	8.5	1.40	-	7.5
	LS	-	12.0	1.60	-	13.3
With Stab	RS	-	6.7	1.00	-	6.3
	LS	-	7.0	0.85	-	7.5
60 Kn, 3000 Ft						
SV Servo, W/O Stab	RS	5.9	6.5	0.60	-	7.0
	LS	7.0	-	0.80	0	8.0
VV Servo, W/O Stab	RS	4.6	8.6	0.40	87.0	7.2
	LS	4.7	8.8	0.59	87.0	7.1
W/O Stab	RS	6.9	11.0	0.90	-	12.7
	RS	7.2	8.3	-	-	11.7
	LS	-	14.5	1.00	-	10.0
	LS	-	14.2	1.30	-	11.7
60 Kn, 10,000 Ft						
SV Servo, W/O Stab	RS	8.4	-	1.20	10.0	8.0
	LS	10.0	-	1.30	0	7.5
VV Servo, W/O Stab	RS	4.8	7.8	0.39	60.0	8.4
	LS	5.3	9.3	0.40	75.0	8.0
W/O Stab	RS	12.5	-	1.00	13.5	12.0
	RS	18.0	-	2.40	-	12.0
	LS	-	10.5	1.27	60.0	8.0
	LS	-	10.5	0.72	120.0	9.0
90 Kn, 3000 Ft						
SV Servo, W/O Stab	RS	7.5	-	1.10	0	8.8
	LS	7.6	-	1.00	0	6.7
VV Servo, W/O Stab	RS	5.8	10.3	0.77	82.0	9.6
	LS	5.4	10.0	0.71	79.0	10.0
W/O Stab	RS	-	20.0	2.50	-	10.2
	LS	-	16.0	1.00	-	10.4
With Stab	RS	4.2	8.3	0.50	100.0	13.3
	LS	6.7	9.5	0.83	45.0	11.5
120 Kn, 3000 Ft						
SV Servo, W/O Stab	RS	7.9	10.3	1.60	35.0	8.0
	LS	14.7	-	1.30	0	10.7
VV Servo, W/O Stab	RS	-	13.4	1.20	-	15.0
	LS	14.7	-	1.00	0	12.0
W/O Stab	RS*	-	16.7	-	-	-
	LS*	-	26.8	2.80	-	9.3

*Flight recorder not feeding paper at constant speed.
Response time ($t_{90\%}$) and control power cannot be determined.

At the 90-kn flight condition, the steady-state control sensitivity of the FSAS-augmented UH-1C was increased slightly over that of the UH-1C with mechanical stabilizer bar. However, rate overshoots on the order of 45 to 100 percent were eliminated, which has the effect of providing a more precisely controlled roll axis at these airspeeds.

Control sensitivities for the UH-1C without mechanical stabilizer bar are also presented in Table V for all flight conditions studied. These control sensitivities are nearly double those of the UH-1C with mechanical stabilizer bar. The FSAS reduced these high-peak rate control sensitivities to steady-state rate control sensitivities with magnitudes comparable to those provided by the stabilizer bar.

Roll FSAS With Vortex Valve Servoactuator

Flight recordings showing helicopter roll-axis response characteristics for the FSAS with vortex valve servoactuator are also shown in Figures 64 through 73. Roll rate responses for this system are characterized by large rate overshoots and, at some flight conditions, such as hover and 90 kn, a wandering roll rate response. The gross shape of the rate responses is similar to that of the unaugmented helicopter. These response characteristics are caused by the high thresholds in the vortex valve servoactuator. The control sensitivity data of Table V does show, however, that this system reduces control sensitivities of the UH-1C without mechanical stabilizer bar to comparable levels obtained with spool valve servoactuators.

Roll FSAS Pilot Comments

One of the airframe manufacturer pilots commented after flying the FSAS-augmented UH-1C that (1) roll-axis control response is a constant rate proportional to cyclic stick input, and (2) the helicopter response due to cyclic inputs does not have the rate control lag as characterized by the UH-1C with stabilizer bar. This made it easier to maneuver the helicopter because it was easier to achieve and maintain a desired rate maneuvering command. He further commented that the FSAS-augmented helicopter could be controlled with more precision and ease over a larger maneuver envelope. The extent to which helicopter maneuverability was increased was not put in quantitative terms. Improvement in the UH-1C roll rate response becomes apparent when the roll rate traces of Figures 64 through 73 are compared for the various FSAS/ UH-1C configurations evaluated.

PITCH FSAS PERFORMANCE

Pitch FSAS performance was evaluated for the vehicle configurations and flight conditions listed in Table I.

Pitch FSAS performance was evaluated against the following criteria:

- Control sensitivity of the FSAS/UH-1C without mechanical stabilizer bar should be comparable to that of the UH-1C with 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 2.43 deg/in. 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-kn flight condition.

- The pitch FSAS should provide a rate proportional to control-stick deflection.

Flight recordings demonstrating improved pitch FSAS/UH-1C performance satisfying these criteria are presented in Appendix II, Figures 74 through 93.

Cyclic Step Inputs - Pitch FSAS With Spool Valve Servoactuator

Flight recordings showing pitch FSAS performance in response to a pitch cyclic step input command are presented in Figures 74 through 83. These recordings show that for a short time period, the pitch FSAS changes the free-helicopter characteristics to provide the desired rate response proportional to cyclic stick deflection. The pitch FSAS complements the free-aircraft characteristics rather than changing them significantly. That is, the system produces a rate response proportional to stick input rather than the free helicopter's peak rate response proportional to stick inputs. This peak rate response is characteristic of the UH-1C with and without the mechanical stabilizer bar. The pitch rate response of the FSAS-augmented helicopter provides the pilots with a vehicle that is easier to control when performing tracking tasks.

Pertinent data showing that the pitch FSAS does not adversely affect the helicopter's control sensitivity (control power) are shown in Table VI. Control power at the hover flight condition was 7.0 deg/in. This value more than satisfies the control power criterion of 2.43 deg/in.

Control sensitivity data at the hover, 60-, and 90-kn flight conditions show that the high sensitivities of the UH-1C without mechanical stabilizer bar are reduced to more controllable levels with the FSAS engaged.

TABLE VI. PITCH FSAS/UH-1C PERFORMANCE -- STEP INPUTS

UH-1C/FSAS Configuration	Test (Down Step or Up Step)	Control Sensitivity $\dot{\theta}_{ss}$/ in. of stick	$\dot{\theta}_{peak}$/ in. of stick (deg/sec/in.)	Response Time, $t_{90\%}$ (sec)	Overshoot (%)	Control Power θ/ in. of stick (deg/in.)
Hover, 3000 Ft						
SV Servo, W/O Stab	DS	9.6	-	1.20	0	7.2
	US	9.1	-	1.20	0	7.0
VV Servo, W/O Stab	DS	10.7 at 1.5 sec	-	-	-	-
	US	10.0	-	1.67	0	8.3
W/O Stab	DS	13.5 at 1.5 sec	-	-	-	11.0
	US	9.1 at 1.5 sec	-	-	-	7.3
With Stab	DS	12.0 at 1.5 sec	-	-	-	7.0
	US	8.3 at 1.5 sec	-	-	-	6.5
60 Kn, 3000 Ft						
SV Servo, W/O Stab	DS	-	5.7	0.60	-	4.5
	US	-	5.5	0.98	-	4.8
VV Servo, W/O Stab	DS	-	6.2	0.77	-	6.1
	US	-	6.5	0.86	-	7.2
W/O Stab	DS	-	10.4	1.00	-	9.0
	US	-	10.2	1.50	-	9.0
60 Kn, 10,000 Ft						
SV Servo, W/O Stab	DS	-	7.0	0.70	-	7.0
	US	-	8.4	1.00	-	7.0
VV Servo, W/O Stab	DS	4.3	7.9	0.71	90	6.0
	US	5.7	-	0.86	-	6.0
W/O Stab	DS	-	9.7	0.80	-	9.7
	US	-	11.0	1.40	-	11.5
	US	-	11.3	1.10	-	9.0
90 Kn, 3000 Ft						
SV Servo, W/ Stab	DS	5.6	-	0.91	-	4.8
	US	-	6.4	0.82	-	3.9
VV Servo, W/O Stab	DS	-	6.5	0.67	-	6.7
	US	6.5	8.9	0.70	40	8.2
W/O Stab	DS	-	9.5	1.20	-	5.7
	US	-	13.2	1.50	-	6.4*
With Stab	DS	-	9.3	1.00	-	7.4
	US	-	10.0	0.90	-	6.3
120 Kn, 3000 Ft						
SV Servo, W/O Stab	DS	7.6	-	1.20	0	7.0
	US	10.0	-	1.20	-	6.1
VV Servo, W/O Stab	DS	-	9.0	1.00	-	8.5
	US	-	10.8	1.10	-	8.2
W/O Stab	DS	-	9.6	1.40	-	10.5
	US	10.0	-	1.30	-	6.7

*Because of large pitch-to-roll rate cross coupling, it was difficult for the pilot to hold roll cyclic stick to its trim position and yet maintain safety of flight. This is further evidenced by comparing the helicopter's pitch and roll attitude excursions for these tests as shown on Figure 82.

Significant improvement in hovering performance can be seen by reviewing the recordings of Figures 74 and 79. Without the FSAS engaged, the helicopter rate response is nearly a ramp with a magnitude that ranges from 8 to 14 deg/sec at the end of 1.5 sec. A pitch rate response of this shape makes the hovering task difficult. With FSAS engaged, helicopter response is nearly a constant pitch rate proportional to cyclic inputs. All pilots commented that the hovering task was easier with FSAS engaged.

Cyclic Step Inputs - Pitch FSAS With Vortex Valve Servoactuator

Flight recordings showing FSAS with vortex valve servoactuator response to pitch cyclic step input commands are presented in Figures 74 through 83. These recordings show that at some flight conditions the FSAS had very little effect on the shape of the helicopter's short-term response. This can be seen by referring to Figures 74, 76, 77, 78, and 82. On the other hand, at other flight conditions the vortex valve servoactuators were effective in changing the unaugmented helicopter's peak rate response to rate response proportional to cyclic inputs. Examples of this desired performance are shown in the recording of Figures 75, 80, and 81. This nonrepeating performance is due to the vortex valve servoactuator threshold problem previously discussed.

Cyclic Pulse Input - Pitch FSAS With Spool Valve Servoactuator

The original criterion for determining FSAS performance for the pitch cyclic pulse input tests was to increase the damping ratio to 0.5 or greater. During the flight test program it was discovered that quantitative values of damping ratio could not be obtained from the flight test recordings. This is due to the following:

- The short-period response of the UH-1C is moderately damped and slow over a large portion of the operational flight envelope, and any improvement in damping is masked by the outside disturbances acting on the helicopter.

- During the pulse inputs, the pitch cyclic stick was not returned to the trim position that existed at the start of the test. This caused the helicopter to fly to a new trim position, and the damping function of the FSAS was superimposed with the rate command from the cyclic stick.

In light of these difficulties in determining quantitative values to evaluate pitch FSAS performance, the evaluation was performed qualitatively.

Pitch FSAS/UH-1C response to pitch cyclic pulse inputs is shown in the flight recordings of Figures 84 through 93.

At the hovering flight condition, Figures 84 and 89, a significant improvement in the helicopter's phugoid characteristics is noted. With FSAS disengaged, a pitch-axis disturbance excites the divergent damping phugoid mode which the pilot must overcome to perform a precision hovering task. With FSAS engaged, the hover phugoid mode is stabilized over the short-term control response of the helicopter. Experience with other analysis programs has shown that a near-deadbeat rate response to a longitudinal-axis disturbance is accompanied by a pitch-axis damping ratio of 0.7. (This can be verified by comparing pitch SAS response to a vertical gust, as shown in Figure 98 of USAAMRDL Technical Report 71-30, with those figures referenced above.)

At the 60-, 90-, and 120-kn flight conditions, the FSAS improved helicopter performance by extending the period of the phugoid and eliminating or minimizing overshoot in pitch rate following the pulse input. This indicates that the damping function is being performed, but a firm damping ratio value cannot be given. By the shape of the pitch rate recordings, damping is estimated to be 0.5 to 0.6.

FSAS-STABILIZED FLIGHT

Five-minute stabilized flights of the FSAS-augmented UH-1C helicopter without mechanical stabilizer bar were conducted at hover (in ground effect) and at 3000 feet pressure altitude and 60-kn indicated airspeed (IAS). For these tests the pilot minimized all induced inputs. This test was also conducted at hover, in ground effect, with FSAS disengaged.

The flight recordings with the spool valve servoactuators for these tests showed that with FSAS engaged, the pilot's ability to fly in gusty and turbulent wind conditions was noticeably improved. The improvement noted was that pilot input commands to maintain straight and level flight were of smaller magnitude and lower in frequency. Pitch, roll, and yaw rate and attitude excursions from the trimmed flight path were of smaller magnitude and lower in frequency.

Attempts to describe the performance improvement quantitatively were not successful because to do so required the recording of the instantaneous outside disturbances (winds and gusts) that the FSAS is designed to minimize. Without a quantitative model of the outside disturbances, a quantitative performance measure to evaluate these tests cannot be developed.

Therefore, the flight recording for these tests was evaluated by correlating pilot comments to pitch, roll, and yaw rates and attitudes and control stick and pedal input commands. The pilot comments verified that when flying with FSAS disengaged, more control inputs of slightly higher magnitude were required to maintain trimmed straight and level flight.

Flight recordings for these tests are not included in this report for the following reasons:

- The 5-minute time duration of these tests and the need to run the recorder at a speed to prevent data compression resulted in recordings of considerable length.

- To take a small portion of a test recording where the FSAS was disengaged and compare it with a small portion of a test recording where the FSAS was engaged requires a common reference for making a performance comparison. For these tests, this common reference is a measure of the outside disturbances acting on the helicopter. Without this common reference, a fair and unbiased selection of a small portion of a test recorded with FSAS engaged and the selection of another small portion of a test recorded with FSAS disengaged would be very difficult.

AUTOROTATION ENTRIES

Autorotation entries were performed at 60, 90, and 120 kn with FSAS engaged and at 60 kn with FSAS disengaged. The autorotations performed with FSAS engaged had similar flight characteristics and received similar comments by the flight test pilots, so only the tests performed at 60 kn are included in this report.

An autorotation was performed by cutting the throttle to simulate engine failure and then dropping collective control at the appropriate time to increase rotor rpm. The flight recordings for these tests are presented in Figure 94. The most significant comments made by the flight test evaluation pilots were that with FSAS engaged, the high yaw rate kick experienced on cutting the throttle was eliminated. After dropping the collective control, the pitch and roll rate and attitude excursions were easier to control. These comments can be visualized graphically by reviewing the recordings of Figure 94.

TURN MANEUVERS

Turn maneuvers of 3 deg/sec to a heading change of 90 deg were performed at two sets of flight conditions:

- Hover, 60, 90, and 120 kn (IAS) at a pressure altitude of 3000 ft
- 60 kn (IAS) at a pressure altitude of 10,000 ft

These maneuvers were performed in both the left and right directions. During these tests, the UH-1C helicopter was flown without mechanical stabilizer bar, and the tests were repeated for both the free aircraft and the FSAS-augmented helicopter. At hover, the turns were performed using the pedals as the principal command function; at 60 kn and above, the maneuvers were performed using the lateral cyclic stick as the principal command function.

The criteria for evaluating FSAS performance during these maneuvers are:

- At hover, does the FSAS fight the turn, or is turning maneuver performance improved by the FSAS?
- At 60 kn (IAS) and above, does the FSAS have any adverse effect on turn coordination, or is turning maneuver performance improved by the FSAS?

During the hovering out-of-ground effect turns, the pilots commented that there was no noticeable difference in performance when performing these maneuvers with FSAS engaged or disengaged. The FSAS did not fight the turn.

When performing unscheduled pedal turns while hovering in ground effect, maneuvering performance was improved when the FSAS was engaged.

During turn maneuvers performed at 60 kn (IAS) and above, the pilots commented that when performing a steady cyclic-only turn, a given turning rate was easier to achieve and maintain when the FSAS was engaged. The FSAS does not degrade turn coordination.

Flight test recordings for these maneuvers are not included in this report, because the criterion for judging FSAS performance was not a measurable quantity. The recordings would be of little value in determining FSAS performance during the above-mentioned turn maneuver tests.

SECTION VI
CONCLUSIONS

The following conclusions regarding FSAS performance were drawn from evaluation of the flight data and from the pilot's comments:

- The FSAS significantly decoupled external disturbance cross coupling. That is, the drift or divergence from trim rates was much less in the two nondisturbed axes for a disturbance in a third axis.
- For the FSAS/UH-1C without mechanical stabilizer bar, a given test or control input could be held in much longer (2 or 3 times) due to stability augmentation control action of the FSAS.
- Pilots could fly with more precision, that is, rate excursions from trim were much less, with the FSAS engaged.
- The roll-yaw control function of the FSAS did not reduce the turn coordination capability of the aircraft to a level detectable by the pilots. One of the FSAS design ground rules was that it would not contain turn coordination sensors and shaping networks. The FSAS was designed to provide the desired stability functions and have minimum effect on turn coordination.
- The longitudinal divergent phugoid mode was stabilized by the FSAS.
- The FSAS/UH-1C without mechanical stabilizer bar can be flown at a continuous speed 10 to 20 kn faster with the FSAS engaged.
- The FSAS augments damping and control response characteristics of the UH-1C in a manner that satisfies the requirements of the design goals. These design goals were generated in light of Military Specification MIL-H-8501A requirements and good engineering judgment.
- There were no detectable effects on FSAS performance due to changes in gains, time constants, or noise levels, which are functions of oil temperature, for the range of oil temperature changes that occurred during flight test.
- The FSAS augmented the stability of the UH-1C helicopter in a manner comparable to or better than that provided by the mechanical stabilizer bar.

The following conclusions are items that become particularly critical in the development of outer-loop flight path control modes:

- Gain scheduling of the control system, to improve aircraft performance and handling qualitites over the entire flight envelope, is required.

- Development of flight path control system components, such as aircraft angular measurement devices, angular measurement synchronizers, and automatic trim systems, is required.

- Development of a rotor/pylon stabilization system to minimize or eliminate vibration modes that exist in the UH-1 series helicopter is necessary.

- Development is required of design and hardware build techniques to eliminate system gain and noise level changes caused by hydraulic oil temperature changes.

- Elimination of signal drift in the fluidic components is necessary.

This analysis study, hardware build, and flight test program have demonstrated the feasibility of using hydrofluidic sensors and shaping networks in stability augmentation flight control applications. This work should be continued to further advance the state of the art in hydrofluidic controls to permit development of control systems to automatically stabilize the flight of an aircraft, that is, automatic flight control system (AFCS) capability.

APPENDIX I
PILOT FLIGHT TEST REPORTS OF THREE-AXIS HYDROFLUIDIC STABILITY AUGMENTATION SYSTEM

HONEYWELL PILOT REPORT OF FSAS FLIGHT TEST

Summary

The FSAS-augmented UH-1C without stabilizer bar exhibited excellent damping and handling characteristics. The FSAS provided better roll stabilization than the mechanical stabilizer bar. The handling characteristics of a UH-1C hovering in a strong downwind condition were improved by the FSAS. The stability of the aircraft at all flight test conditions was improved by the FSAS.

Introduction

The following paragraphs constitute the project pilot's qualitative flight test report of the FSAS.

Twenty-two data flights were conducted during the period from 2 September 1970 to 2 December 1970. The test vehicle was a UH-1C helicopter, Serial No. 64-14101.

The flight test program was initiated with a series of FSAS engage and disengage tests in hovering and translational flight. Limited pulse and step response tests were conducted in the roll and yaw axis while hovering in ground effect. A series of control pulses and steps was used at all other flight conditions to evaluate FSAS performance. Stabilized flight conditions were also evaluated. Finally, a series of turns and autorotation entries was evaluated.

The data flights were conducted in weather conditions ranging from clear with 15 miles visibility and wind less than 3 kn to 600 ft overcast with 3 miles visibility and light freezing drizzle with wind 20 kn gusting to 30 kn.

Results and Discussion

Stick Forces

During flight No. 1 (maintenance test flight), a left lateral cyclic stick pull was noted. It was estimated that a 1.5-lb pull to the right was required to maintain level flight. The magnitude and direction of the force on the cyclic stick was subsequently measured and verified during ground tests. The pilot considered the force objectionably high. The existence of a lateral force made accurate pitch-axis control difficult. Postflight investigations revealed that the lateral force was caused by the geometry of the flexible hydraulic lines connected to the FSAS roll servoactuator. The lateral force was removed by rerouting the flexible lines in a manner to nearly balance the forces on the roll servoactuator. In general, throughout the flight test program, small forces detectable at the cyclic stick were caused by the flexible lines.

Built-In-Test System Checks

FSAS built-in-test (BIT) buttons were actuated during ground running at 6600 engine rpm. FSAS response to the BIT system was just barely detectable at the cyclic stick and pedals. Unfortunately, the BIT system, as presently designed, does not afford the pilot a convenient preflight check of the FSAS. Even though the FSAS is not a safety-of-flight item, it would be desirable to know the operational status of the system before takeoff.

Engage and Disengage Tests

Engage and disengage tests were conducted on the FSAS both in-flight and during ground operation. No objectionable transients were noted during ground operation. Initial tests disclosed an objectionable transient in flight when the yaw FSAS was engaged or disengaged. Upon engaging the yaw FSAS, the aircraft would yaw to the right approximately 5 deg and about the same amount to the left at disengage. Subsequent investigation revealed that a high null signal in the yaw FSAS caused the transient. The null signal was reduced, eliminating the objectionable transient for the remainder of the flight test program.

Thereafter, very small nonobjectionable transients were detectable at the cyclic stick and pedals as the FSAS was engaged and disengaged. The pilot considered these transients desirable as an indication that the FSAS had in fact engaged or disengaged.

At various times during the flight test program, a roll cyclic stick offset resulted when the roll FSAS was engaged. Disassembly of the roll controller revealed Teflon particles in the roll output block which caused the offset.

No change in FSAS engage-disengage characteristics was noted during maneuvering flight, including hovering, climbing, or descending turns, and autorotation.

Yaw-Axis Performance

1) Hovering In Ground Effect - A very limited number of dynamic yaw-axis tests were conducted while hovering in ground effect. The pilot input feature of the FSAS yaw axis was very effective in that no difference in yaw-axis response could be detected with the FSAS on or off.

   There was no tendency for the FSAS to "fight" the pilot. During initial flights, yaw FSAS performance was not impressive in a hover. During the latter part of the program, extensive downwind hovering was performed with adverse conditions of strong, gusting winds. Under these conditions, the contribution of the yaw FSAS was evident. The pilot's control task was reduced slightly. It is recognized that the FSAS was not designed to improve the handling qualities at a hover, but evaluation at this flight condition was inevitable because the control task in a hover is considerable.

   Airframe manufacturer test pilot H spent a considerable amount of time evaluating the performance of the FSAS while hovering in ground effect. He stated that he was particularly impressed with improvement of the handling qualities during lateral translation.

2) Hovering Out of Ground Effect - Dynamic yaw-axis tests were conducted at approximately 3000 ft pressure altitude. A series of yaw step inputs, pulse inputs, and pedal turns was executed.

   FSAS improvement of the aircraft handling qualities in response to yaw step inputs was not obvious to the pilot. The flight records may show an improvement in damping or response characteristics.

   On the other hand, FSAS improvement of the aircraft damping and handling qualities in response to yaw pulse inputs was clearly evident. Without stabilization in the yaw axis, the aircraft exhibited 4 or 5 overshoots in response to a left yaw pulse input, while 2 or 3 overshoots resulted from a right yaw pulse input. With FSAS engaged, overshoots to the left were reduced to 2, while overshoots to the right were eliminated. The adverse roll developed during yaw pulse inputs was also reduced when the roll FSAS was engaged.

Consecutive FSAS on and off pedal turns were not executed during out-of-ground-effect tests; therefore, improvements were not noted. Consecutive FSAS on and off pedal turns were executed during in-ground-effect hovering, and slight improvement was noted, probably due to the combined effect of roll and yaw FSAS.

3) Translational Flight (60, 90, and 120 KIAS) - Yaw step inputs and pulse inputs were used to evaluate the performance of the yaw FSAS at 60, 90, and 120 KIAS.

   Yaw FSAS performance was not detectable in response to step inputs at the three test airspeeds. Here again, though, yaw FSAS performance was readily detectable in response to pulse input. The improvement in helicopter handling qualities becomes more evident with increasing airspeed. Yaw FSAS performance was enhanced when combined with the roll FSAS. Yaw FSAS performance during simulated rocket runs was evaluated. The action of the yaw FSAS did not impair the maneuverability of the aircraft and aided the pilot's ability to maintain a line of sight to the target.

Roll-Axis Performance

Roll step inputs and "cyclic-only" turns were used to evaluate roll FSAS performance at the following flight conditions:

1) Hovering Out of Ground Effect - Roll rate sensitivity of the unaugmented aircraft without a stabilizer bar was too high. The roll FSAS markedly decreased roll rate sensitivity to a more desirable value. The roll FSAS provided a smoother roll response to step inputs and provided a definite stiffening of the roll axis.

2) Translational Flight (60, 90, and 120 KIAS) - The performance of the roll FSAS is immediately obvious upon engagement. The augmented aircraft without a stabilizer bar exhibited a strong tendency to maintain a residual roll oscillation when induced by the pilot or external disturbances. The roll FSAS dramatically damped this roll oscillation. The aircraft exhibited the same tendency, to a lesser degree, with stabilizer bar attached. The roll FSAS very definitely provided better handling qualities in roll than did the stabilizer bar.

   At higher airspeeds, the spiral instability of the aircraft without a stabilizer bar was readily detected in response to roll step input. The aircraft had a tendency to rapidly increase bank angle in response to a fixed roll input command when the bank angle exceeded approximately 20 deg. The action of the roll FSAS was to delay the onset of this instability.

During the steady "cyclic-only"turns, a given turning rate was easier to achieve and maintain when using the FSAS.

Pitch-Axis Performance

Pitch step inputs and pulse inputs were used to evaluate pitch FSAS performance at the following flight conditions:

1) Hovering in Ground Effect - During initial flights, pitch FSAS gain was too high. The excessive gain overcontrolled the pitch axis as evidenced by a strong vertical bounce of the test aircraft.

   Eventually, pitch FSAS gain was reduced to a value which eliminated the overcontrol. The handling qualities of the test aircraft without a stabilizer bar were very acceptable, and the pitch FSAS did not provide detectable improvement.

   The pitch FSAS aggravated the pylon rock problem in the test aircraft. Apparently, the pylon isolation mounts were deteriorated, as pylon rock could easily be induced by a fore and aft motion of the cyclic stick. Washing out the cockpit with the cyclic stick also produced severe pylon rock while in a hover.

2) Translational Flight (60, 90, and 120 KIAS) Pitch FSAS performance at the 60-, 90-, and 120-KIAS flight conditions was investigated with a series of pulse inputs and step inputs. The handling qualities of the aircraft in the pitch axis without a stabilizer bar were good. The pitch FSAS provided slight improvement, especially at airspeeds above 120 KIAS.

Autorotation Entries

Autorotations were initiated at each of the flight test conditions (60, 90, and 120 KIAS). The unaugmented aircraft without the stabilizer bar possesses satisfactory handling characteristics in autorotation. The FSAS functioned to reduce the yaw rate during autorotation and the yaw rate and yaw angle produced at the time of throttle chop. The FSAS further improved handling qualities by stiffening the roll axis.

Vortex Valve Servoactuator Evaluation

There is a complete lack of engage or disengage transients when using vortex valve servoactuators. There is a detectable difference between the performance of an FSAS using the spool valve servoactuators and one using the vortex valve servoactuators. There appears to be a time lag before the vortex valve servoactuators arrest an induced rate. This action is readily detectable when executing yaw step inputs. The initial yaw rate response is reduced after approximately 2 seconds when the vortex valve servoactuators respond to the rate signal. This effect was not noted during the initial flights with the vortex valve servoactuators, which leads one to suspect a degradation of performance.

Conclusions

The FSAS-augmented UH-1C without a stabilizer bar exhibited excellent damping and handling characteristics. The FSAS improved handling characteristics of the UH-1C aircraft over the speed range from hover to 140 KIAS. The roll FSAS provided better roll stability than the mechanical stabilizer bar.

Recommendations

- The servoactuator installation should be redesigned to preclude the introduction of control forces due to flexible hose connections.
- The built-in test (BIT) system should be redesigned to permit its use for an operational check of the FSAS during preflight checkout.
- The FSAS should be evaluated under operational conditions to include actual rocket runs.
- FSAS gains should be optimized at hover and normal cruise airspeed.
- An automatic yaw trim feature should be added to the FSAS.

REPORT OF ARMY PILOT J, FSAS FLIGHT TEST

The three-axis fluidic stability augmentation system (FSAS) (spool valve servoactuators) installed in a UH-1C helicopter was evaluated at the contractor's site on 27 October 1970.

Table VII lists the conditions tested and the pilot comments regarding the apparemt qualitative increase in aircraft damping using the three-axis fluidic damping system (FSAS). The table shows that increased damping was apparent in 10 cases, slightly apparent in five cases, and not

TABLE VII. UH-1C QUALITATIVE DAMPING EVALUATION WITHOUT STABILIZER BAR

Input	Airspeed (KIAS)	Pressure Altitude (ft)	Increased Damping
Fwd Longitudinal Step	65	2000	Not apparent
Aft Longitudinal Step	65	2000	Yes
Fwd Longitudinal Pulse	65	2000	Slight
Aft Longitudinal Pulse	65	2000	Slight
Right Roll Step	65	2000	Slight
Left Roll Step	65	2000	Slight
Left Yaw Step	65	2000	Slight
Right Yaw Step	65	2000	Not apparent
Right Yaw Pulse	65	2000	Yes
Left Yaw Pulse	65	2000	Not apparent
Fwd Longitudinal Step	110	4700	Not apparent
Aft Longitudinal Step	110	4700	Not apparent
Aft Longitudinal Pulse	110	4700	Yes
Fwd Longitudinal Pulse	110	4700	Yes
Right Roll Step	110	4700	Yes
Left Roll Step	110	4700	Yes
Left Yaw Step	110	4700	Yes
Right Yaw Step	110	4700	Yes
Right Yaw Pulse	110	4700	Yes
Left Yaw Pulse	110	4700	Yes

apparent in five cases. These results are limited by two conditions. First, aircraft damping is difficult to determine qualitatively; and second, weather conditions during the single flight were unstable, with clouds and high, gusting winds.

The following comments were noted:

- Pitch rate damping apparently was not uniform between forward and aft steps.

- Stick and pedal trim positions changed as a function of the FSAS null positions.

- UH-1C pylon rock was not affected by incorporation of the FSAS.

- FSAS gain appeared to be a function of system oil temperature.

- The FSAS did not significantly improve aircraft response following throttle chops.

- There was no objectionable feedback in the controls with the FSAS actuated.

- The FSAS did not restrict the basic mission envelope of the UH-1C helicopter, although maneuvering flight at levels below 1.0 g should receive further evaluation. In this area, the FSAS appeared to restrict rapid recovery from pushover maneuvers.

The results of this evaluation were significantly restricted by the calendar, weather, and qualitative constraints of the program. In addition, a comparative evaluation between a standard UH-1C and the FSAS helicopter was not possible because the evaluation helicopter lacked a stabilizer bar.

REPORT OF ARMY PILOT S, FSAS FLIGHT TEST NO. 1

Introduction

The three-axis fluidic stability augmentation system (FSAS) (spool valve servoactuators) installed in a modified Army UH-1C helicopter was evaluated at the contractor's site on 28 October 1970. The 2-hour flight was conducted in stable air with the wind reported at less than 3 kn. Control fixtures were used to assist in obtaining crisp, measured pulse and step control inputs. The fixtures were structurally flexible, which made it nearly impossible to reestablish the original trim position following an input. The helicopter modifications allowed for the testing of two configurations: (1) FSAS off with the stabilizer bar removed, and (2) FSAS on with stabilizer bar removed.

Separate FSAS channels (pitch, roll, yaw) could be selected by console-mounted switches. System gains and sensitivities were fixed. The takeoff gross weight was approximately 7425 lb, and the center of gravity was within 0.25 in. of the mast centerline.

Results and Discussion

Initial hovering was conducted with the FSAS off because the hydraulic fluid temperature was too low, even though the aircraft had just been rolled out of the hangar. The aircraft's low roll-axis damping was very apparent; in fact, the combination of roll sensitivity and damping was such that low-time pilots could probably experience PIO (pilot-induced oscillation) problems. Pitch was not bad, although damping seemed lower than that of the UH-1B. Nothing different was noticed in the yaw axis - as one might expect.

The flight test engineer then arrived, and by the time he finished the preflight FSAS checks and calibration (the aircraft was not shut down), the hydraulic fluid temperature was up and we proceeded to the test area with FSAS on. A 64-kn instability phenomenon was investigated enroute to the test area.

The so-called 64-kn instability phenomenon was quite apparent, although, on this flight, it did not initiate itself (stable air). Short fore and aft pitch cyclic stick pulse inputs would set it off every time; lateral roll cyclic stick pulse inputs would not. The initial body motion was a definite lateral acceleration. The following motion seemed to be predominantly lateral, although it felt as if some combination of longitudinal and lateral body motion (acceleration) was occurring - perhaps somewhat circular (whirl mode) at a frequency of 9 cycles per 5 sec. It was poorly damped as long as airspeed remained constant. Small changes in airspeed affected the damping strongly. It occurred most strongly at about 60 KIAS and persisted until the airspeed was reduced to 58 KIAS or increased to 63 KIAS. FSAS had little or no effect on its occurrence or damping. The tip path plane seemed to be oscillating (flapping) a couple of inches both ahead and to the side of the helicopter.

Hover testing was conducted over a grass-covered taxiway. All FSAS channels were turned off for a baseline qualitative feel. Then roll FSAS was switched on. Its contribution was very obvious, while no significant degradation of control power or change in sensitivity was noticed. The same technique was used to evaluate pitch and roll, and then all three channels. The effect on the pitch mode was least noticeable. It seemed that more stick was required to stop a pitch rate, although the rates developed by a given input seemed almost unchanged. In yaw, the system was more effective in stopping right yaw rates, while the initial responses seemed unchanged. With all three channels on, the aircraft was obviously easier to hover; however, the greatest improvement was in the lateral mode.

Six power-recovery, 180-deg autorotations were performed to evaluate the effect of the system (all channels on versus all channels off). The only significant effect was a reduction of the yaw excursions (a slight effect in pitch) following the throttle chop and lowering of collective pitch. The aircraft also seemed "tighter" in autorotative maneuvering flight. (Note: Engine acceleration in power recoveries was extremely slow.)

The project pilot and the flight test engineer then exchanged seats and proceeded to satisfy the contractual step and pulse control inputs using the control positioning fixtures. The same conditions that Army Pilot J flew were chosen in the interest of acquiring comparative pilot comments; i.e., 65 and 110 KIAS at 3000 ft pressure altitude. Additional comments are as follows:

- The only aspect of the step inputs that was noticeable was the reduction in angular rate (pitch and roll), the roll-pitch coupling, and the negative dihedral effect. The characteristics at 110 KIAS were much more dramatic than those at 65 KIAS.

- The low yaw damping became more apparent at the higher speed, which was expected; basic aircraft yaw damping was stronger with left pedal pulse inputs than right pedal pulse inputs by a factor of 2. The influence of FSAS yaw damping was very strong (as the time histories will indicate); i.e., 4 to 5 cycles without FSAS to damp following right pedal pulse inputs versus 1 to 1-1/4 cycles with FSAS - with only a slight overshoot. No numbers can be attached to the roll characteristics, but the FSAS improved them at least 1 to 1-1/2 Cooper Rating units. This became even more apparent with brisk rolling maneuvers at 120 KIAS. The pitch characteristics due to pulse inputs did not appear to be affected much by FSAS. The developed pitch rates seemed more constant with FSAS on for the step inputs.

Push-overs were conducted with FSAS on and off, but nothing seemed improved or adverse.

Controls-fixed throttle chops were performed at 100 KIAS. The recoveries were initiated when the rotor speed reached approximately 270 rpm. The FSAS seemed to reduce yaw excursion somewhat, with a similar but less noticeable effect in pitch.

The system was turned off in straight and level flight at 140 KIAS, with noticeable improvements in pitch and roll. Note: This would probably reach very important proportions in rough air.

The FSAS caused no apparent change to maneuvering stability.

An unsatisfactory tendency for the pedals to "float" was noted when my feet were removed from the pedals. This is reportedly due to the spring action of the hydraulic lines which serve the FSAS actuator. Also, a slight roll cyclic stick preload was reportedly being caused by the same situation. In addition, there was a slight amount of drag in the cyclic stick due to the friction involved in dragging the FSAS servoactuators across their respective sliding plates. These discrepancies are minor and can be expected in view of the research nature of the hardware.

Summary

The flight conditions were ideal for acquiring good data and the data are accurate; i.e., altitude 3000 to 3500 ft pressure altitude, airspeeds within ±2 kn, and quality pulse and step inputs (test sheets). The FSAS improved the aircraft's handling qualities significantly. The strongest improvement was in the roll mode. A lesser improvement was noticed in yaw in spite of the big improvement, quantitatively, in the directional dynamics. Little improvement was noticed in pitch. The washout scheme in yaw was apparently optimized, since no change in steady-state yaw rate was detected when engaging the system. A roll cyclic trim shift was detected when engaging and disengaging the system, and it changed "sign" from hover to forward flight; i.e., in hover, the stick trim position moved approximately 1 in. to the left when engaging the system but it moved to the right in forward flight. Engaging transients could be felt in the pedals, and they were felt ever so slightly in pitch cyclic. In forward flight, the sideslip ball moved about a "half a ball" when engaging and disengaging, but this is attributed to the coupling with the roll cyclic trim shift. No additional transients occurred when engaging or disengaging the system while imposing angular rates or accelerations to the aircraft. It is felt that some disengaging transient is desirable to signal the pilot of the situation. Other than the roll cyclic trim shift, the system transients were satisfactory.

REPORT OF ARMY PILOT S, FSAS FLIGHT TEST NO. 2

Discussion

This was the second 2-hour flight evaluation of the three-axis fluidic stability augmentation system (FSAS) installed in a modified UH-1C helicopter. The first evaluation was conducted on 28 October 1970; this one was conducted a month later, on 17 and 18 November. The purpose of the second evaluation was to investigate the performance of the system with vortex valve servoactuators installed and to compare it with the original system, which uses spool valve servoactuators.

Qualitatively there was no detectable difference except for the complete absence of any engaging or disengaging transients. The roll cyclic trim shift transient experienced during the first evaluation was reportedly due to system contamination. A small piece of a Teflon joint seal had accidentally gotten into the system. In the interest of evaluating the system as thoroughly as possible within the time allotted, considerable attention was paid to acquiring good quantitative data. Damping is often difficult to assess qualitatively; therefore, actual response characteristics must be measured and recorded.

During the contractor's flight, which immediately preceded the Government evaluation, a yaw-axis FSAS servoactuator hard-over was experienced. It posed no problem to flight safety, in that the servoactuator moved to and locked in the mid-position (as advertised) when the yaw axis was disengaged. The reason for the malfunction was structural failure within the servo valve of the servoactuator caused by a surge in the hydraulic pressure. The surge occurred as a result of aircraft system checks prior to takeoff.

Conclusions

The comments submitted in connection with the initial, 28 October 1970 flight test are, in general, unchanged. The major conclusion is that this project has quite successfully demonstrated that a three-axis rate damping system can be provided using fluidics. Additional testing would be required, however, to establish and/or demonstrate adequate reliability and maintainability.

The FSAS also demonstrated the capability of replacing the stabilizer bar on UH-1-type helicopters. The system, would appear to be competitive (cost, weight, cube, and power-wise) with electrical stability augmentation systems.

Recommendations

1. The Army should continue to pursue fluidic artificial stability system research and development by proceeding with the study, design, and testing of a fluidic autopilot (attitude hold). In addition, it would appear logical to investigate the possibility of algebraically summing the various signals fluidically (SAS, autopilot, pilot inputs, and perhaps guidance) such as the Air Force is doing electrically with their "Pilot Assist System."

2. Operational experience (R&M) should be obtained by installing a fluidic yaw damper on an Army OH-58 aircraft. It should be noted that the OH-58 is a good candidate aircraft because it possesses very poor directional control characteristics.

3. Additional effort should be expended to optimize the FSAS "gains" across the entire flight envelope; i.e., the damping desired in hover is different from that desired for brisk maneuvering or at high speed.

AIRFRAME MANUFACTURER B PILOT'S REPORT OF FSAS FLIGHT TEST

The weather at the time of flight was quite marginal, so the flight was quite short. Also, the pitch servoactuator malfunctioned before takeoff, so only the yaw and roll axes were operational. This failure is described in Section IV of the main text.

The pilot's comments are summarized as follows:

- The pylon dampers were apparently worn out because of pylon rock in hover and in the 45- to 65-kn speed range. Pylon rock is aggravated by the FSAS.
- Yaw rate with FSAS off is higher initially, but slows as the vehicle turns downwind. With the FSAS on, yaw rate is not quite as high initially, but remains nearly constant as the vehicle turns. This makes the vehicle slightly easier to turn while in hover.
- With the FSAS engaged, roll rate is constant, as compared with a change in rate after 3 or 4 sec with the stabilizer bar.
- At most forward speeds, the vehicle with FSAS engaged handled better than a vehicle with the stabilizer bar.
- The vehicle is somewhat more stable in hover, but exhibits a small reduction in control power from a UH-1C with stabilizer bar. This helped yaw but did not make roll feel any better.
- The vehicle, at approximately 120 kn with FSAS engaged, handled very much like the standard UH-1C with stabilizer bar, which was one of the design goals.

APPENDIX II
TEST DATA

This appendix contains flight recordings taken at various speeds and altitudes for yaw, roll, and pitch step and pulse inputs for the UH-1C helicopter with and without mechanical stabilizer bar, with the three-axis FSAS engaged and disengaged, and with spool valve servoactuators and vortex valve servoactuators.

The flight recordings are categorized as follows:

- Yaw step - right (Figures 44 through 48)
- Yaw step - left (Figures 49 through 53)
- Yaw pulse - right (Figures 54 through 58)
- Yaw pulse - left (Figures 59 through 63)
- Roll step - right (Figures 64 through 68)
- Roll step - left (Figures 69 through 73)
- Pitch step - down (Figures 74 through 78)
- Pitch step - up (Figures 79 through 83)
- Pitch pulse - down (Figures 84 through 88)
- Pitch pulse - up (Figures 89 through 93)
- Autorotation (Figure 94)

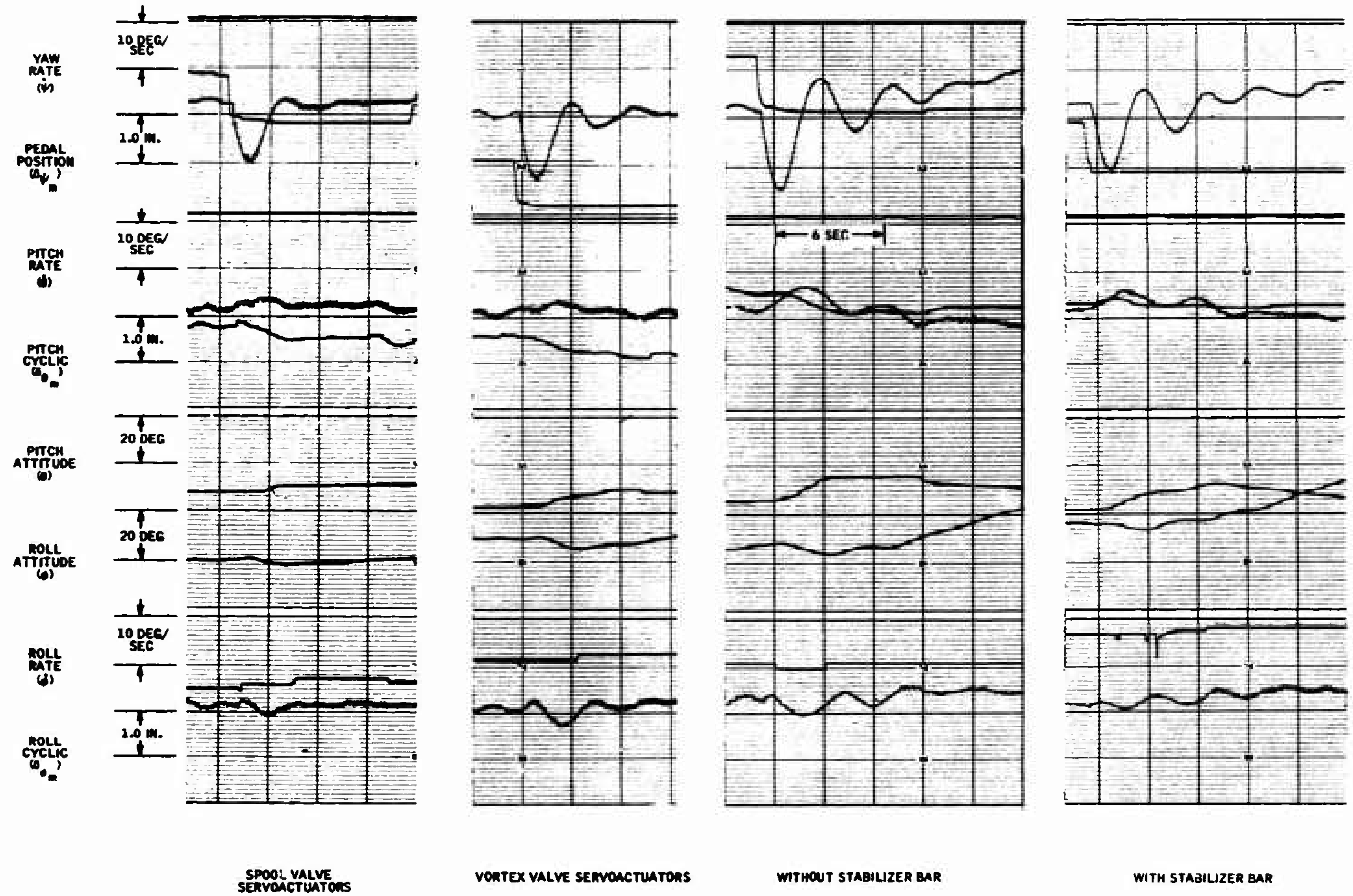


Figure 44. Yaw Step - Right (Hover, 3000 ft).

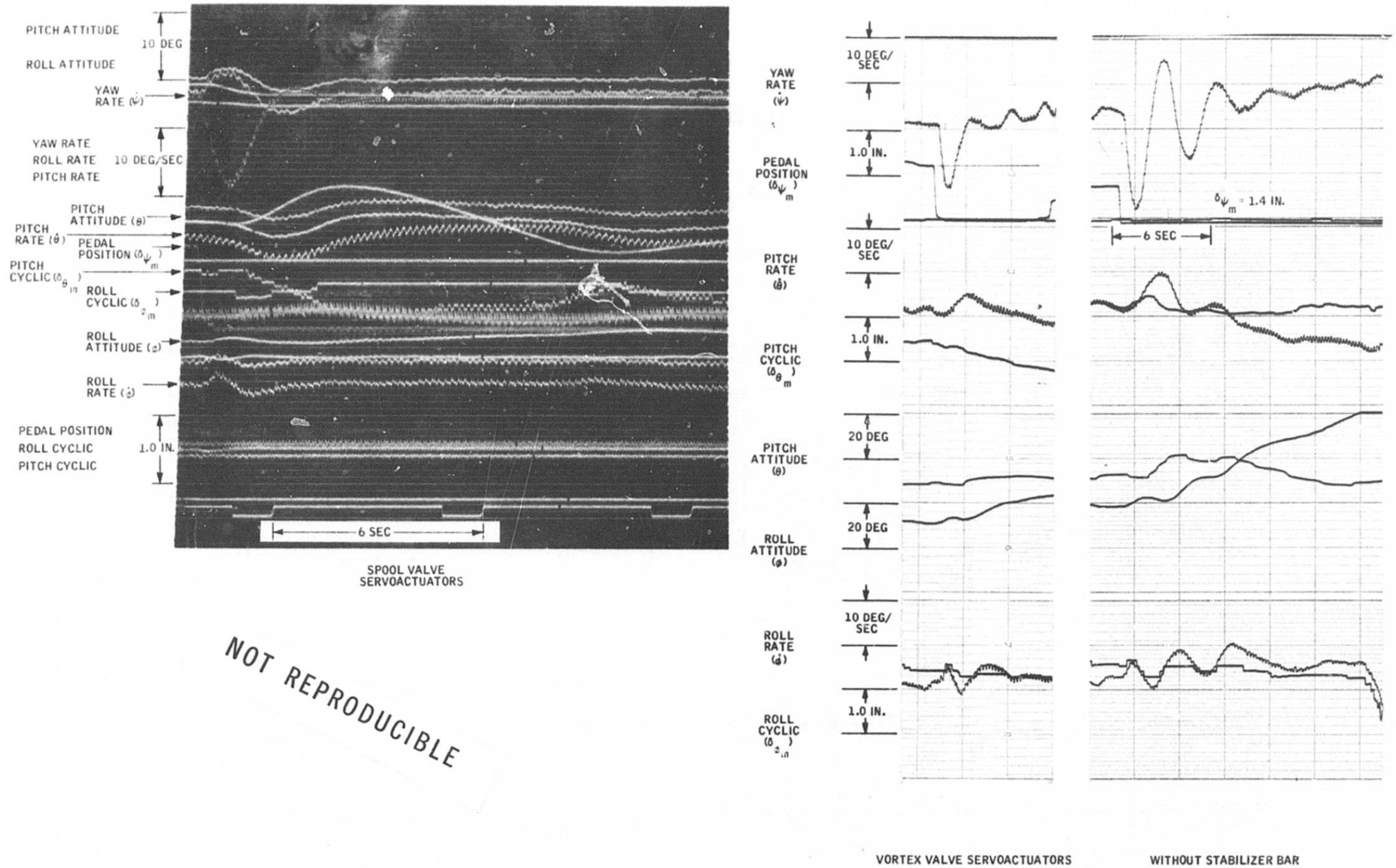


Figure 45. Yaw Step - Right (60 kn, 3000 ft).

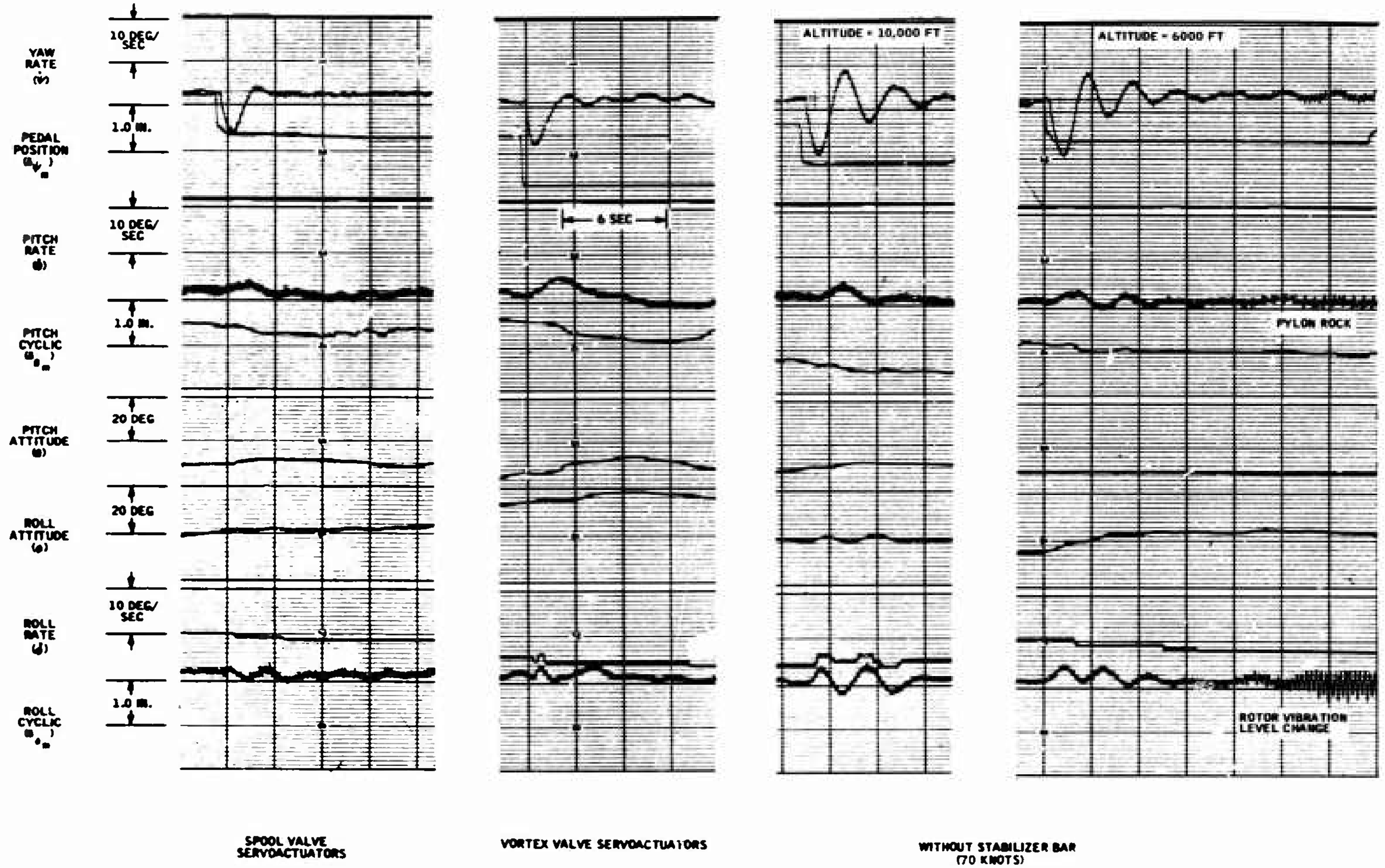


Figure 46. Yaw Step - Right (60 kn, 10,000 ft).

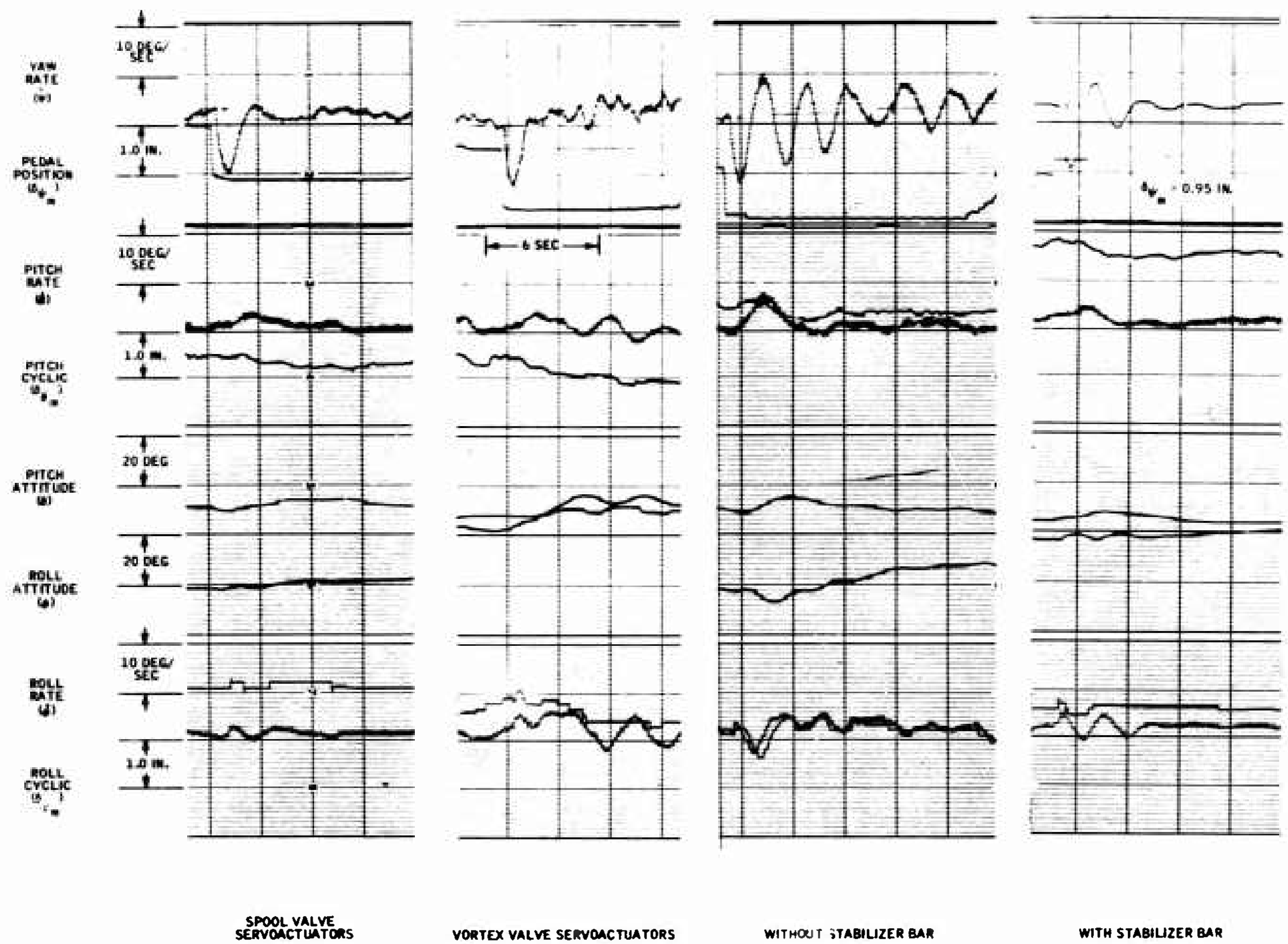


Figure 47. Yaw Step - Right (90 kn, 3000 ft).

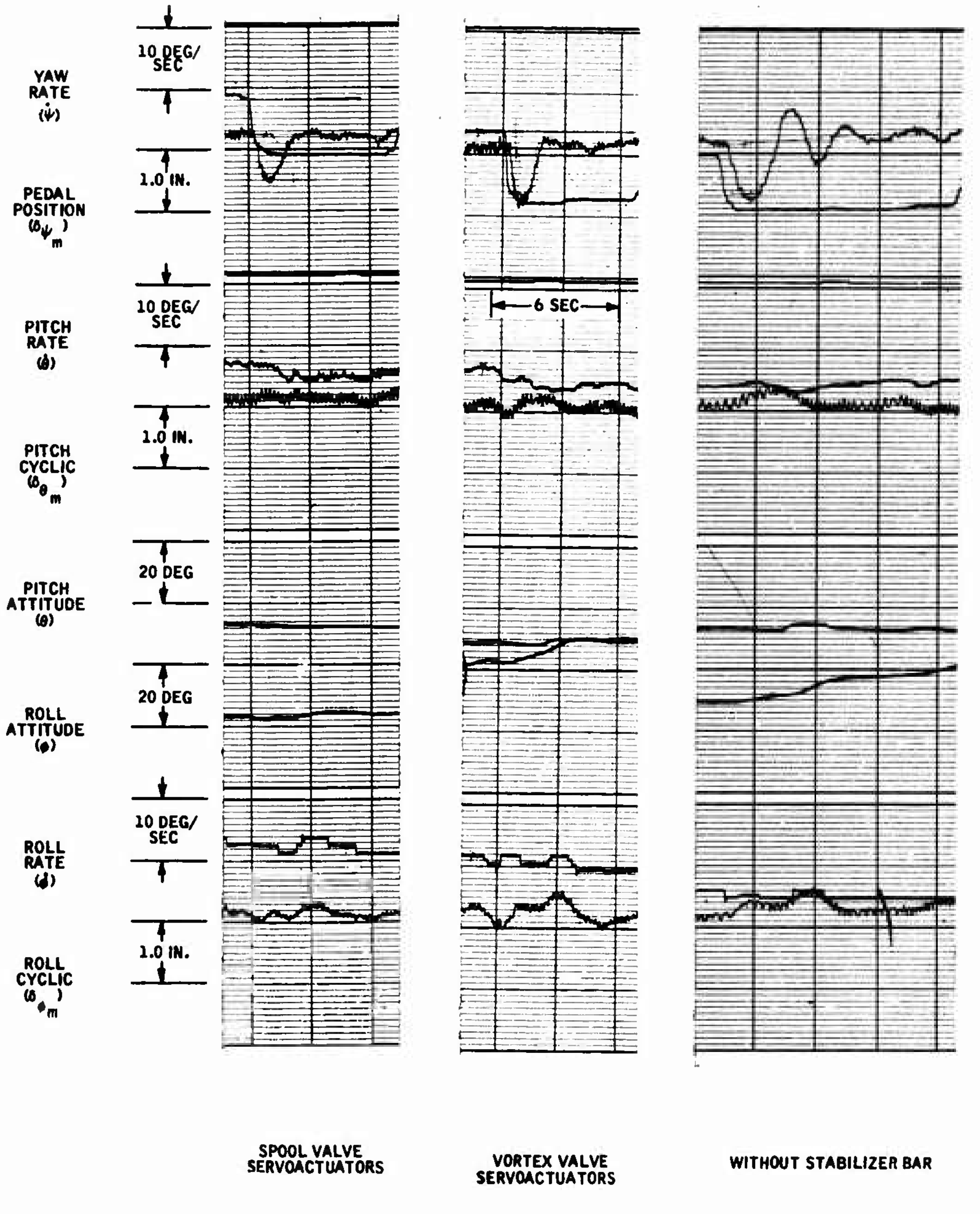


Figure 48. Yaw Step - Right (120 kn, 3000 ft).

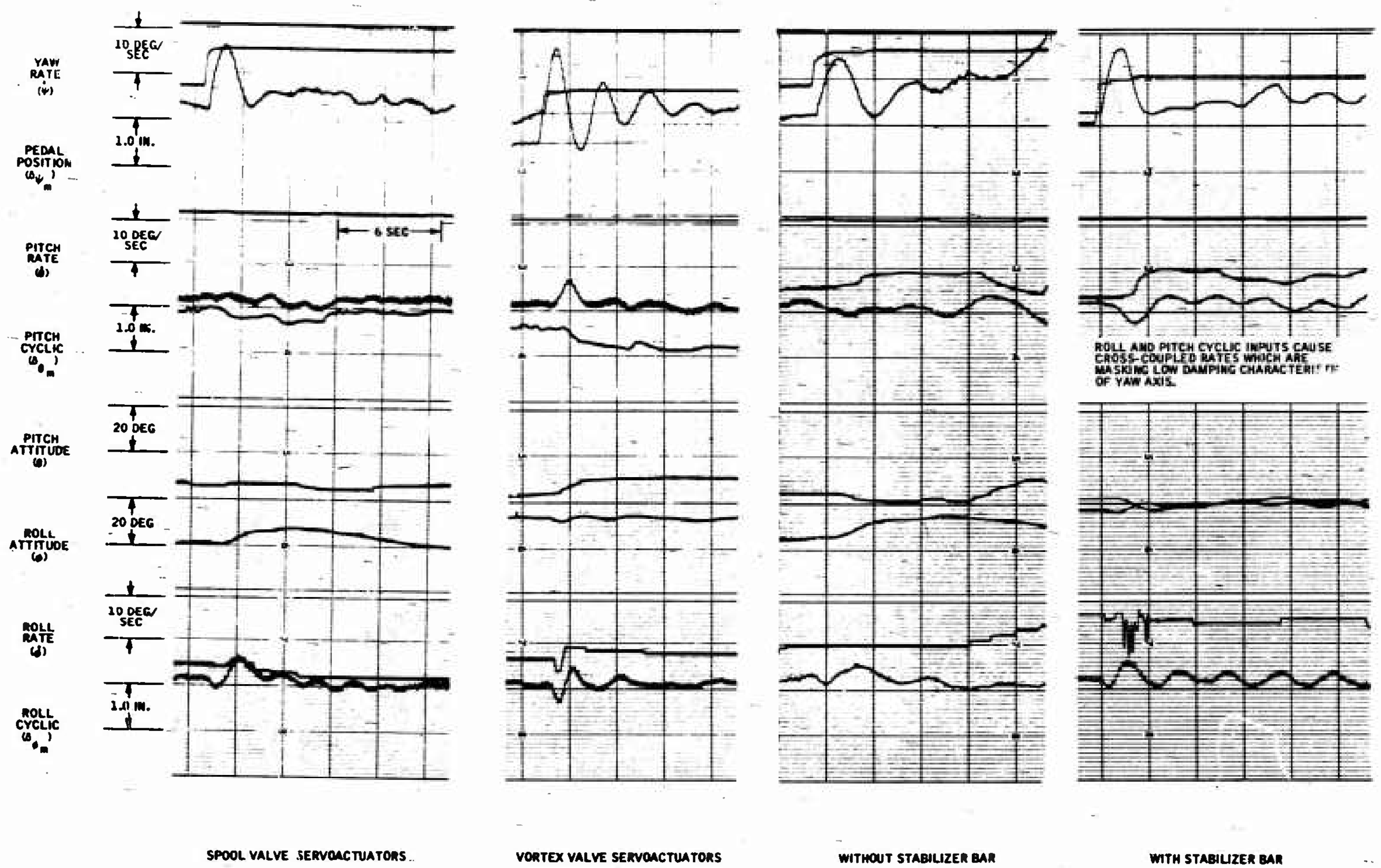


Figure 49. Yaw Step - Left (Hover, 3000 ft).

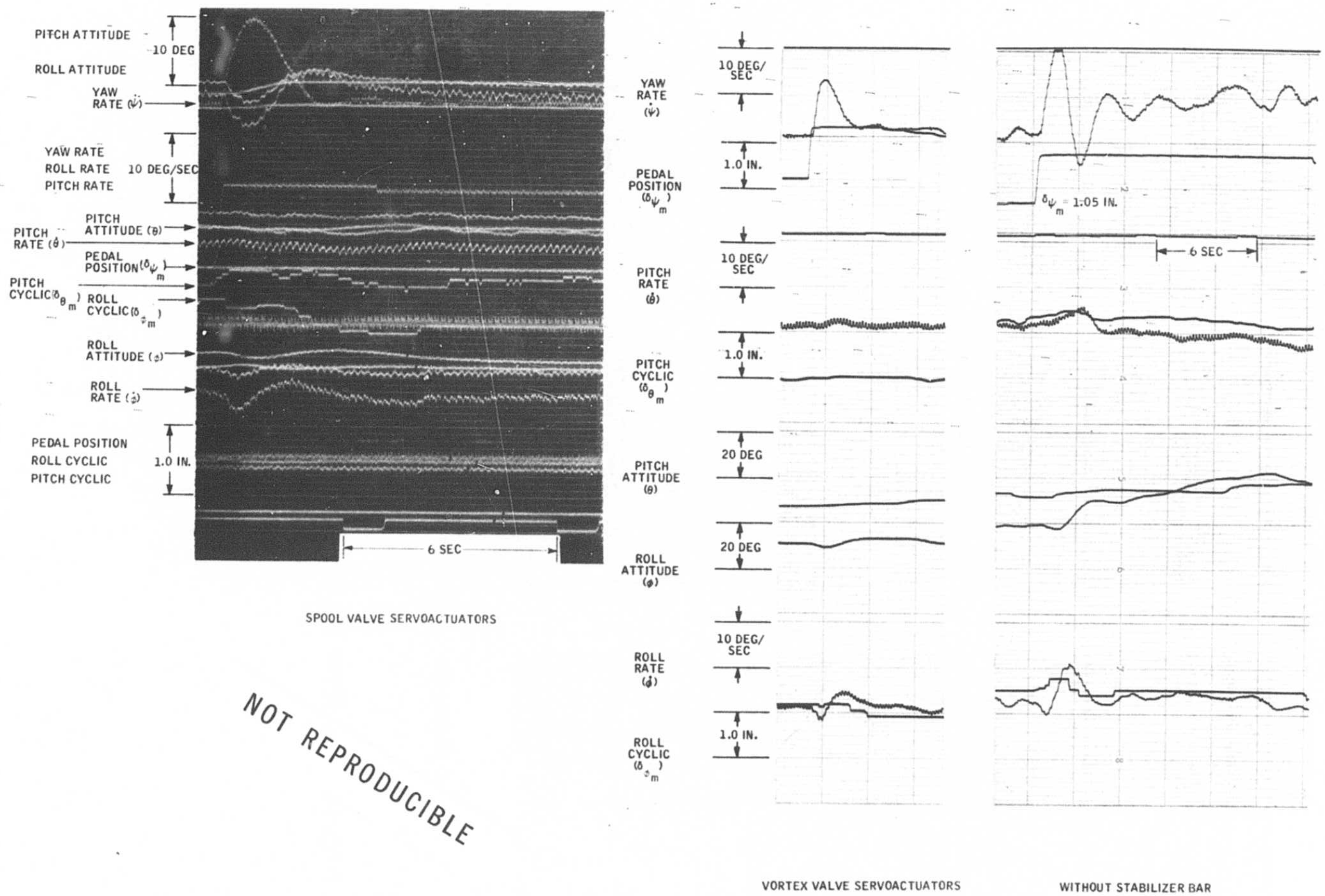


Figure 50. Yaw Step - Left (60 kn, 3000 ft).

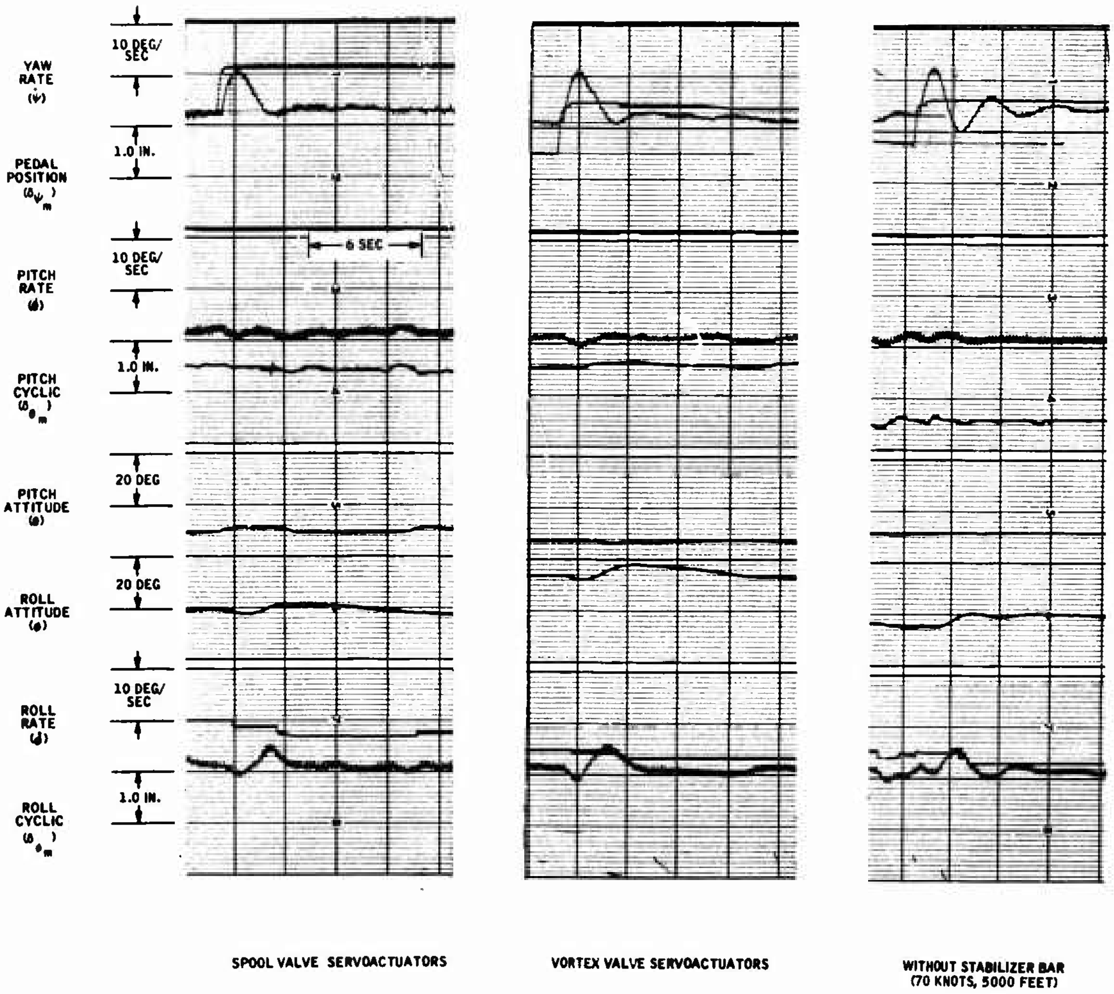


Figure 51. Yaw Step - Left (60 kn, 10,000 ft).

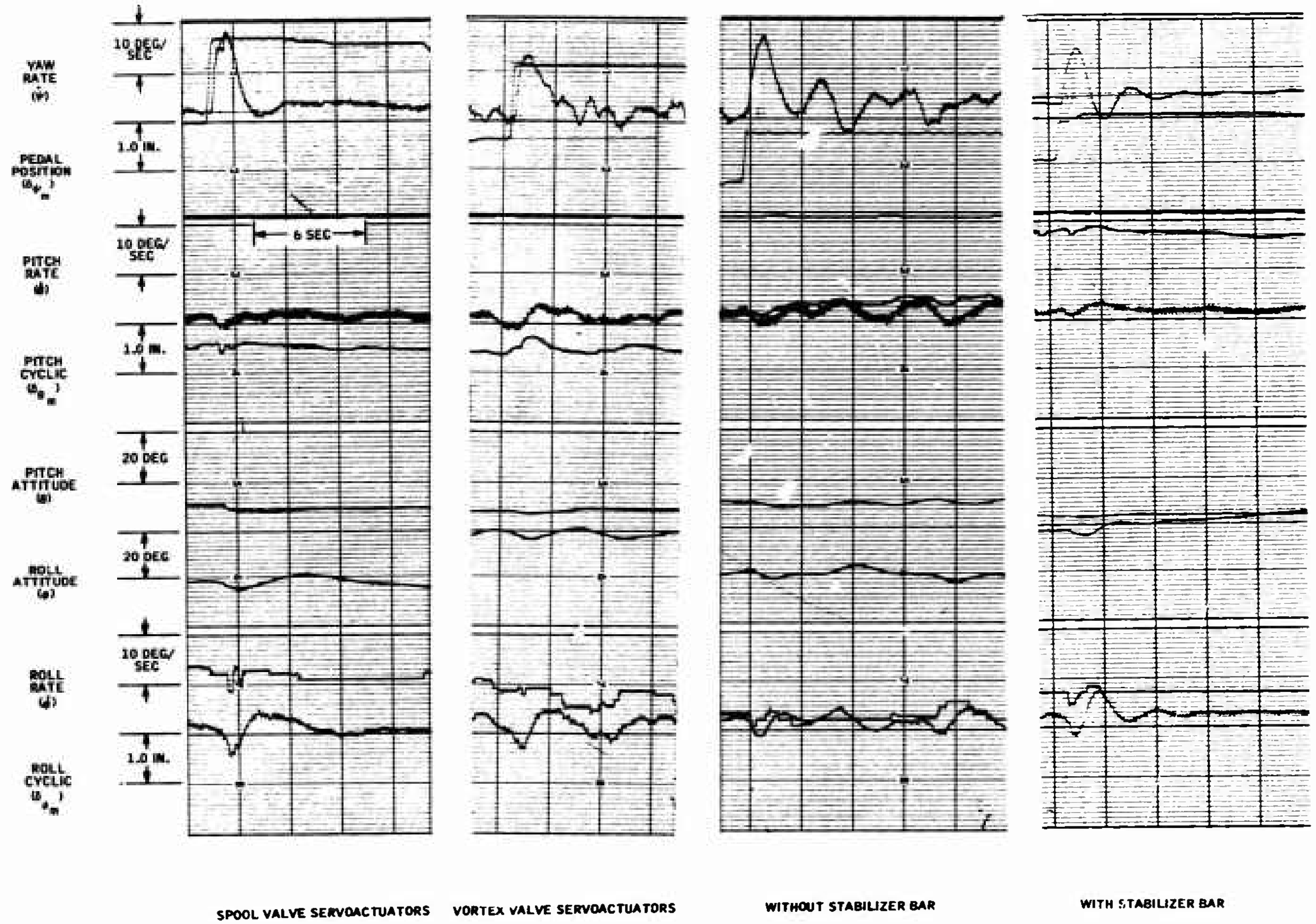


Figure 52. Yaw Step - Left (90 kn, 3000 ft).

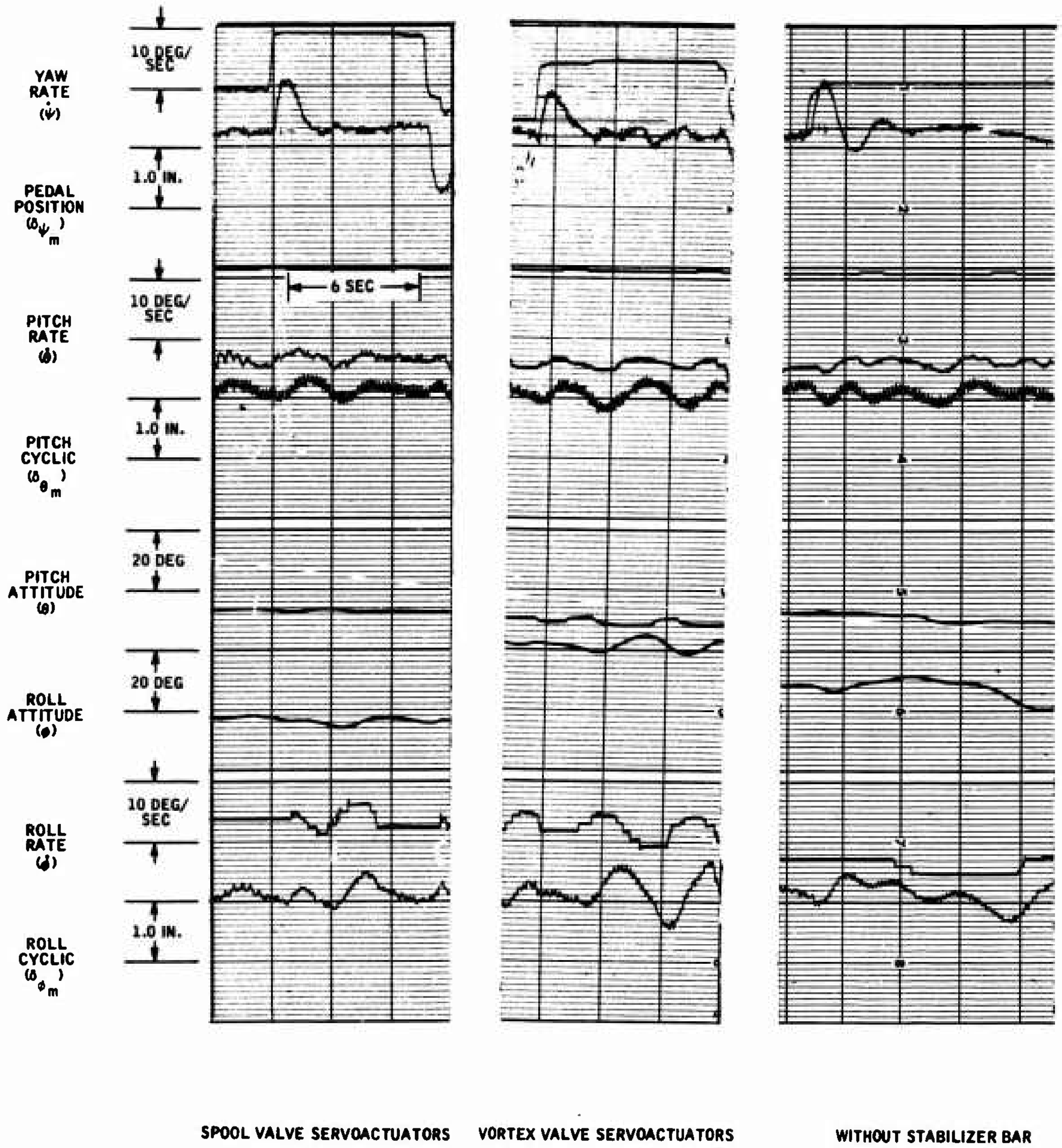


Figure 53. Yaw Step - Left (120 kn, 3000 ft).

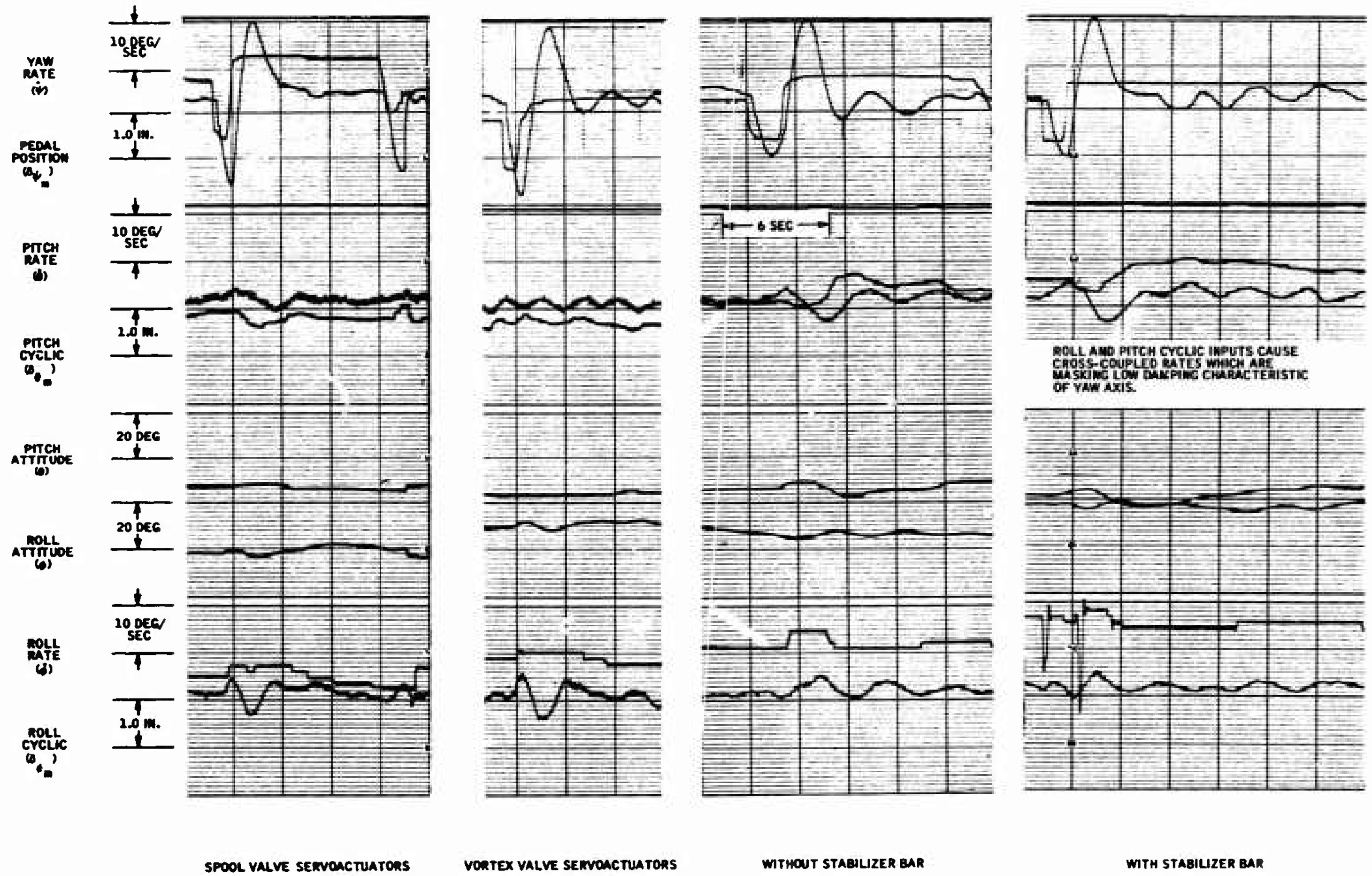


Figure 54. Yaw Pulse - Right (Hover, 3000 ft).

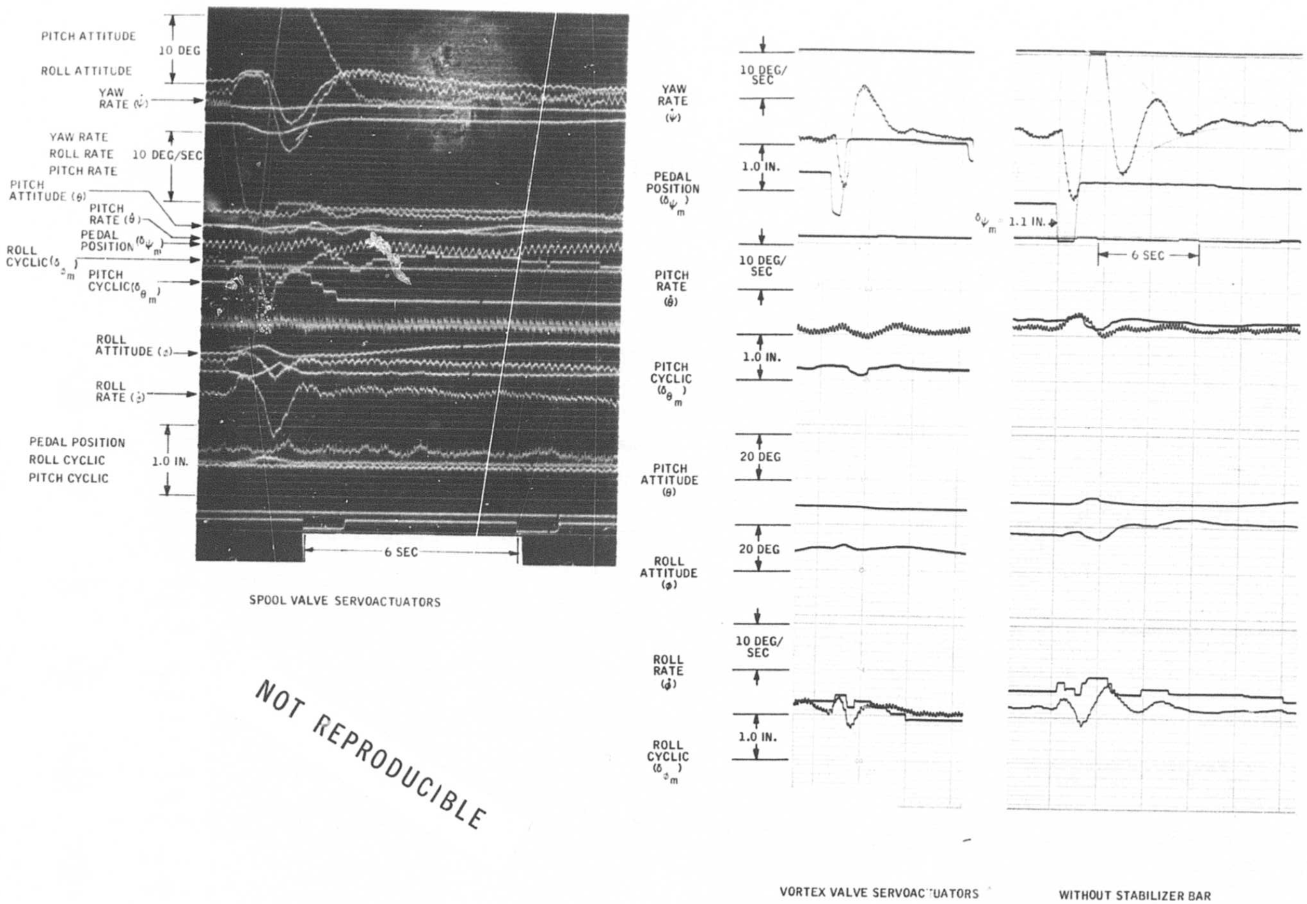


Figure 55. Yaw Pulse - Right (60 kn, 3000 ft).

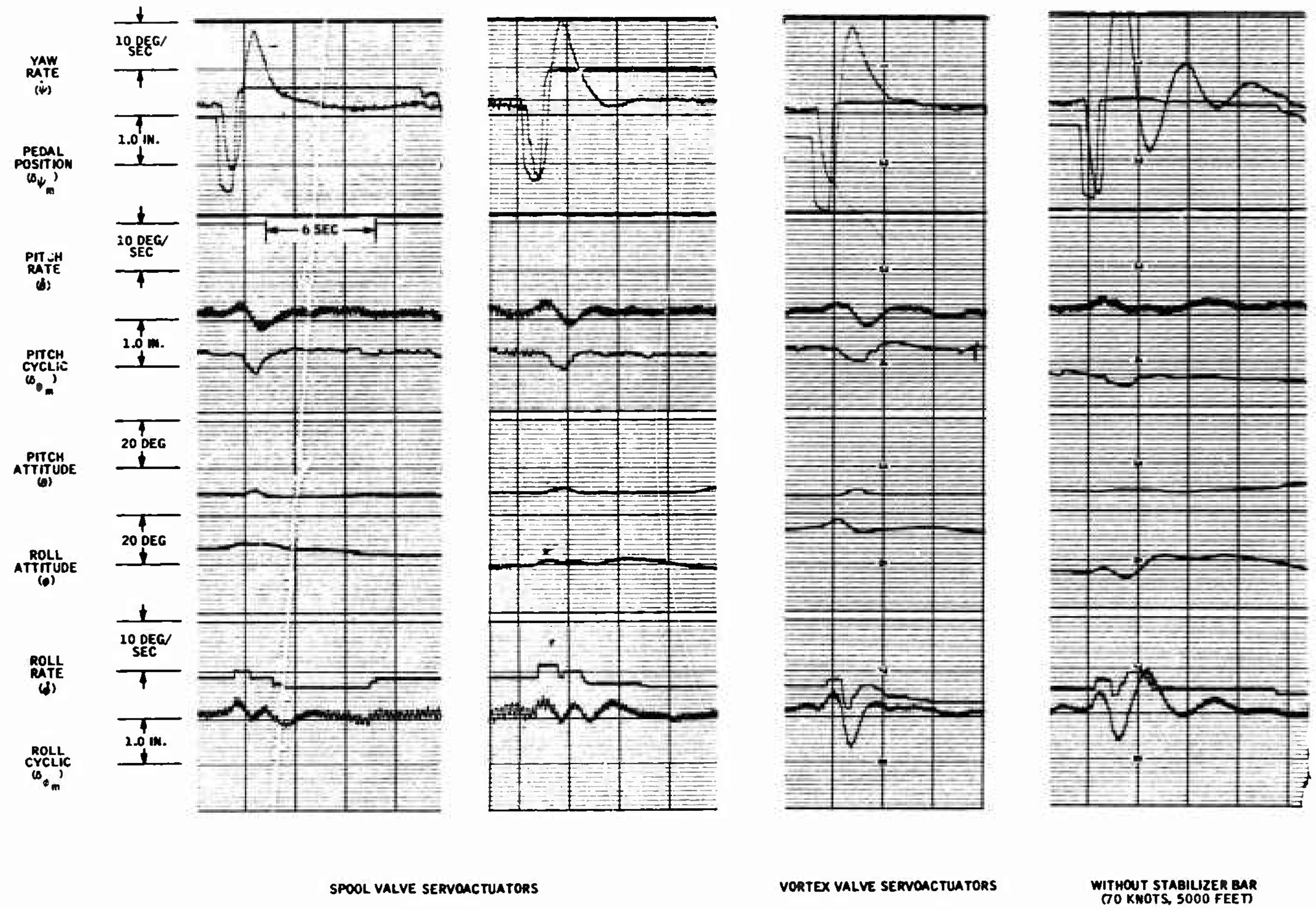


Figure 56. Yaw Pulse - Right (60 kn, 10,000 ft).

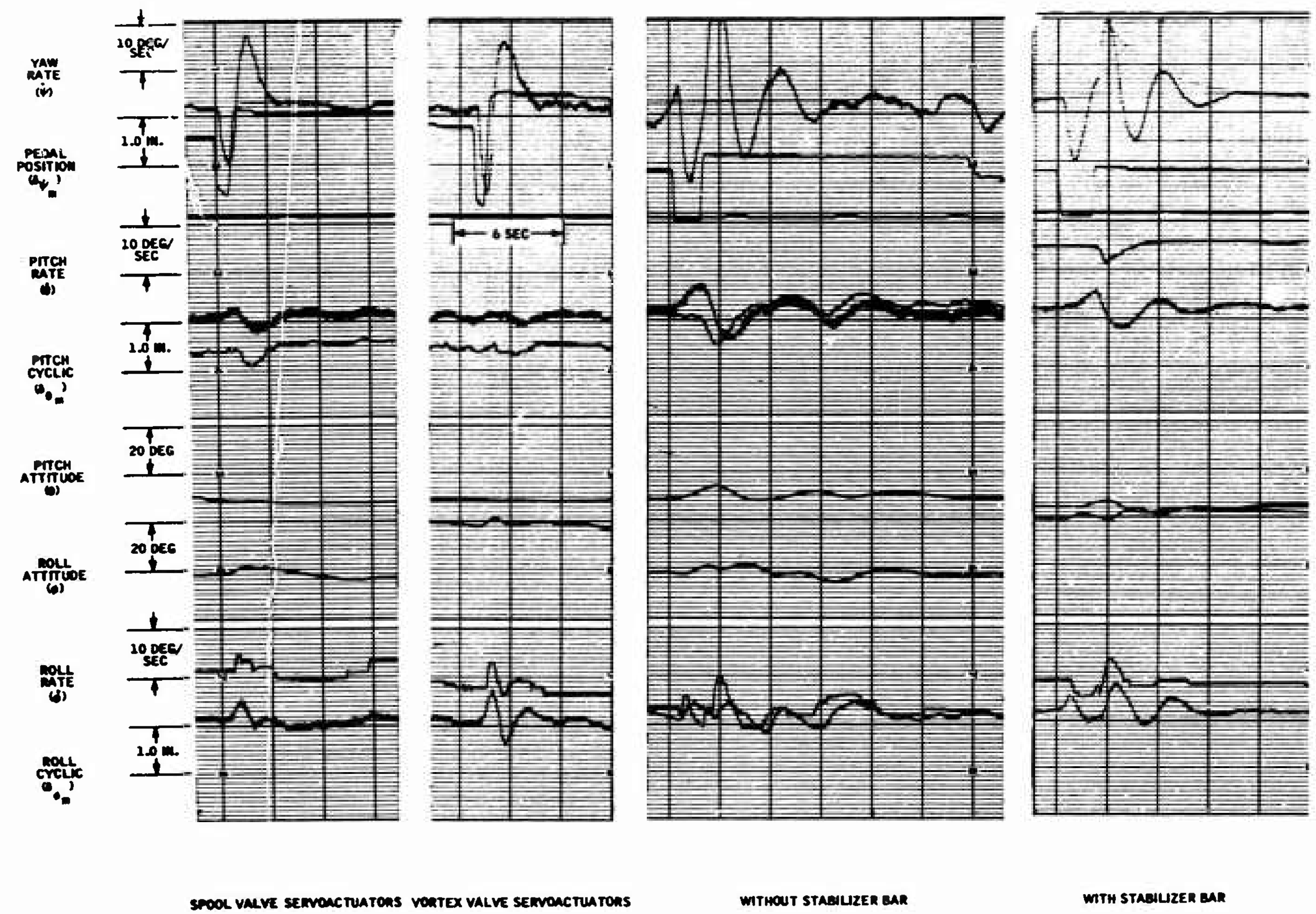


Figure 57. Yaw Pulse - Right (90 kn, 3000 ft).

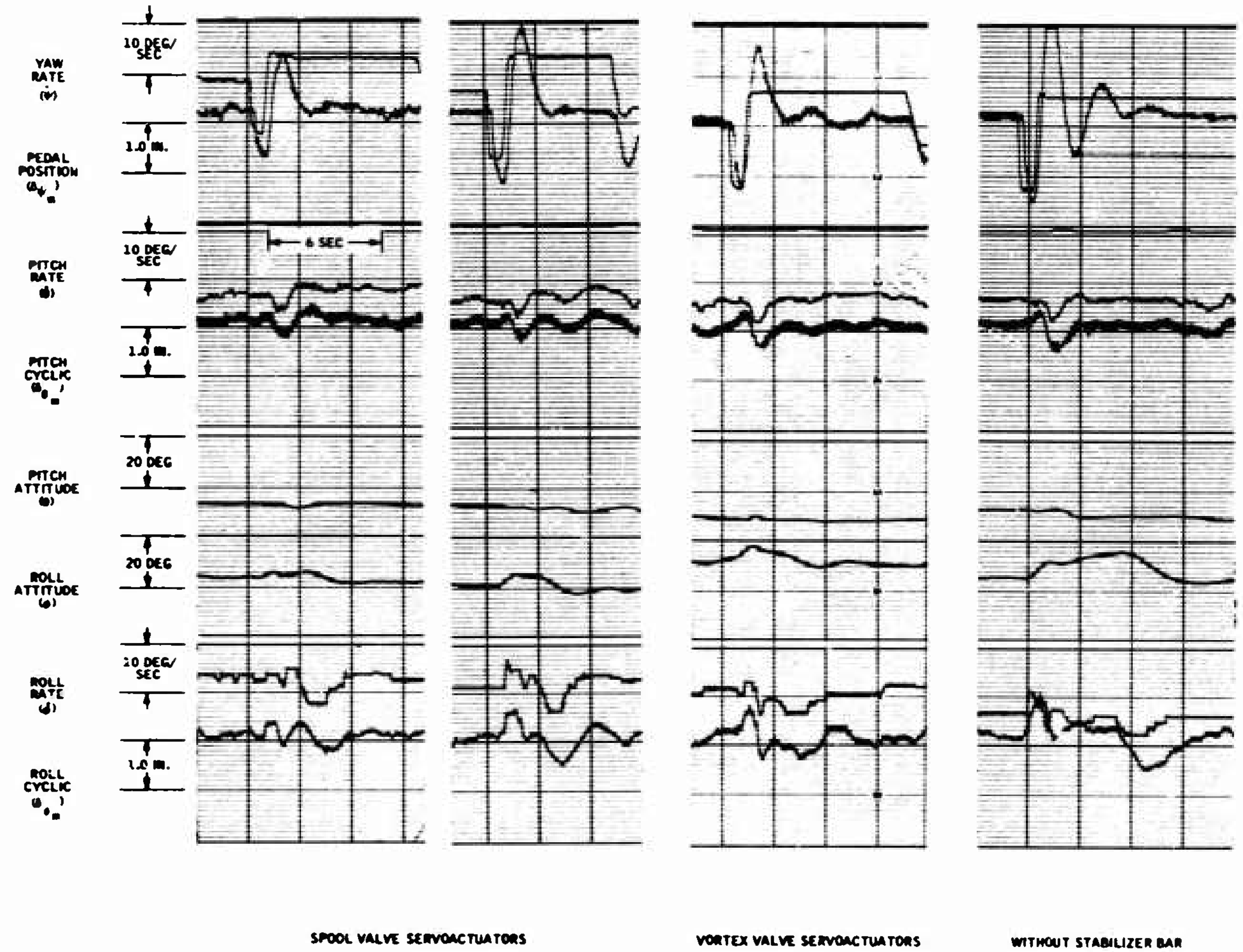


Figure 58. Yaw Pulse - Right (120 kn, 3000 ft).

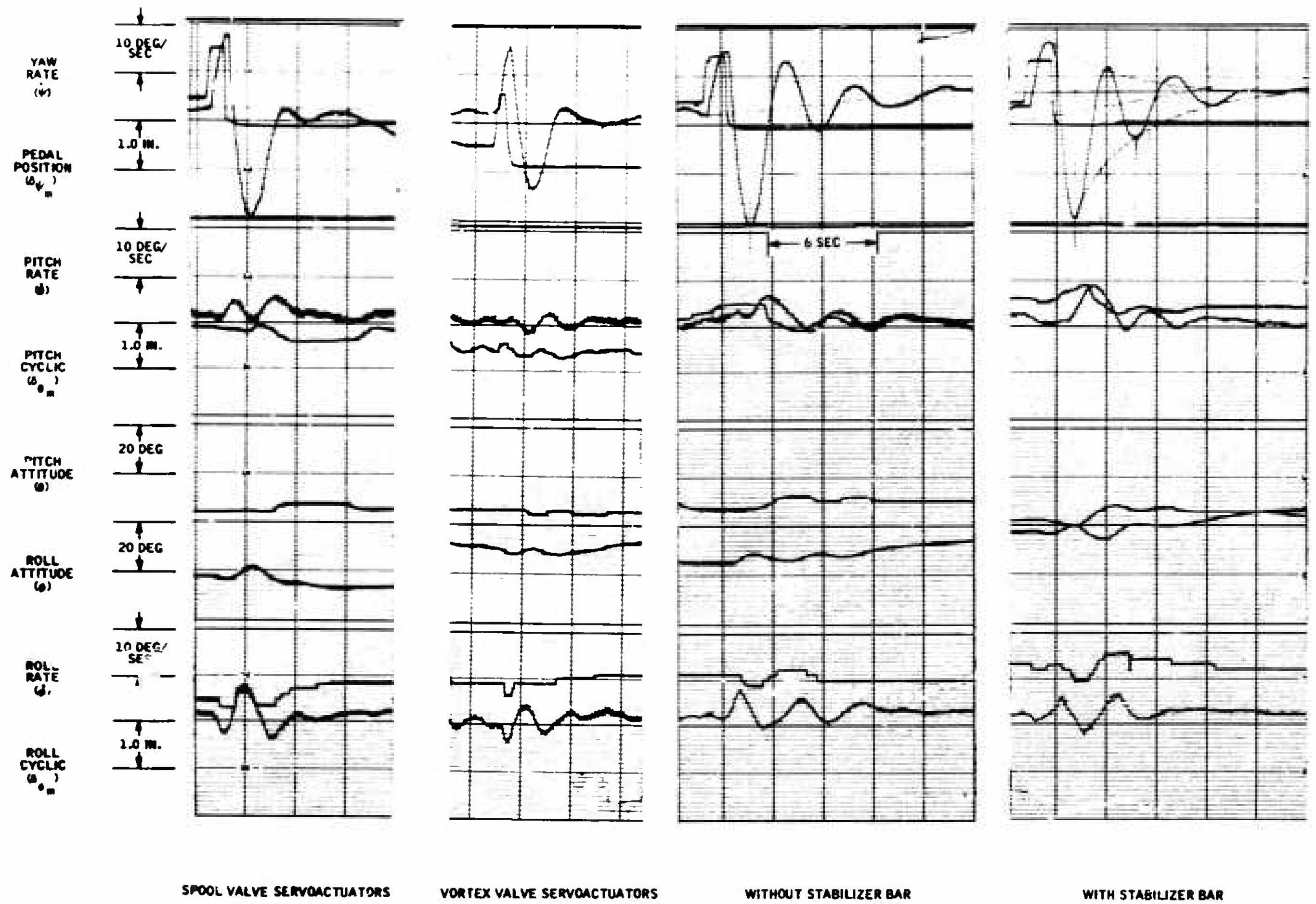


Figure 59. Yaw Pulse - Left (Hover, 3000 ft).

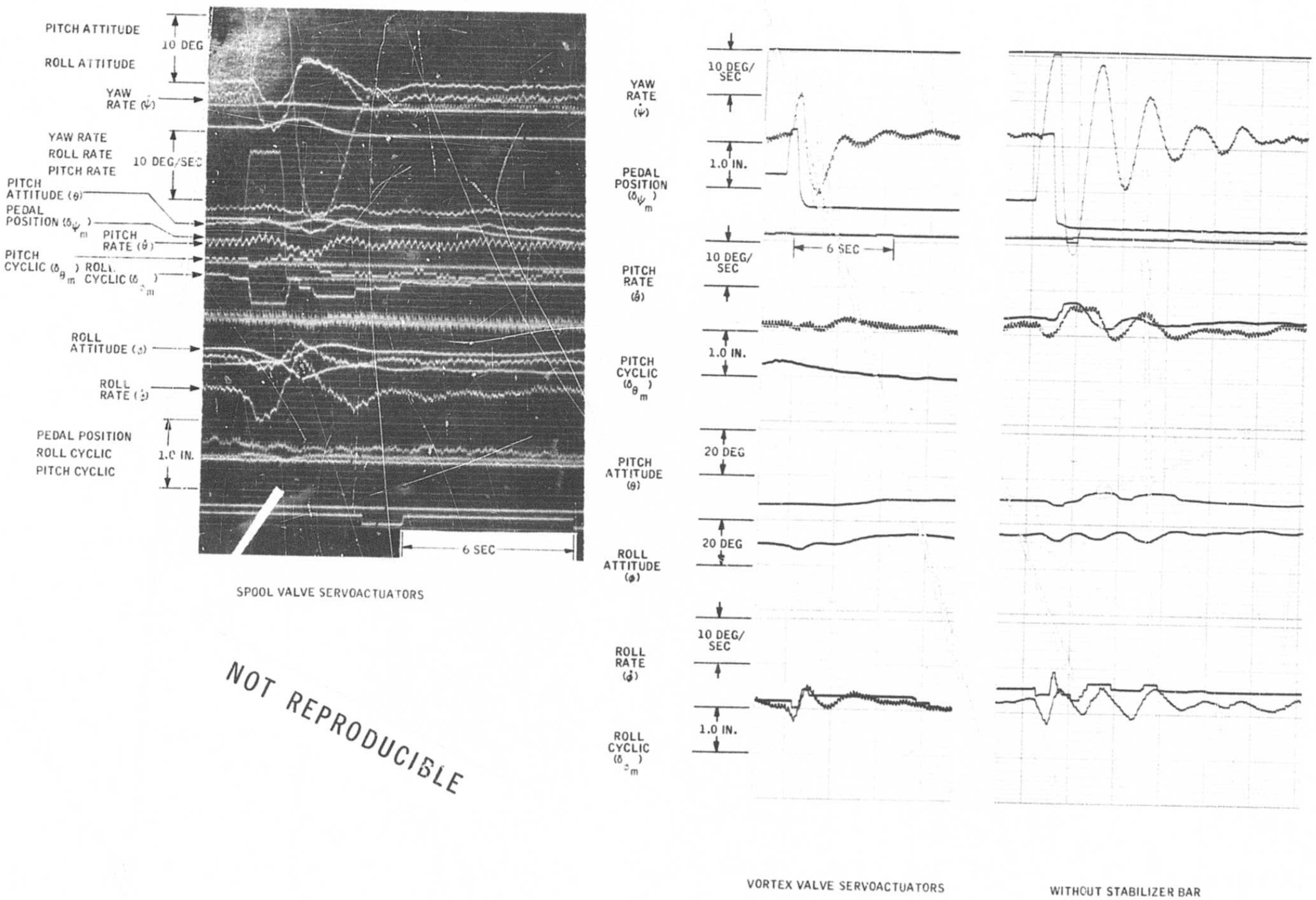


Figure 60. Yaw Pulse - Left (60 kn, 3000 ft).

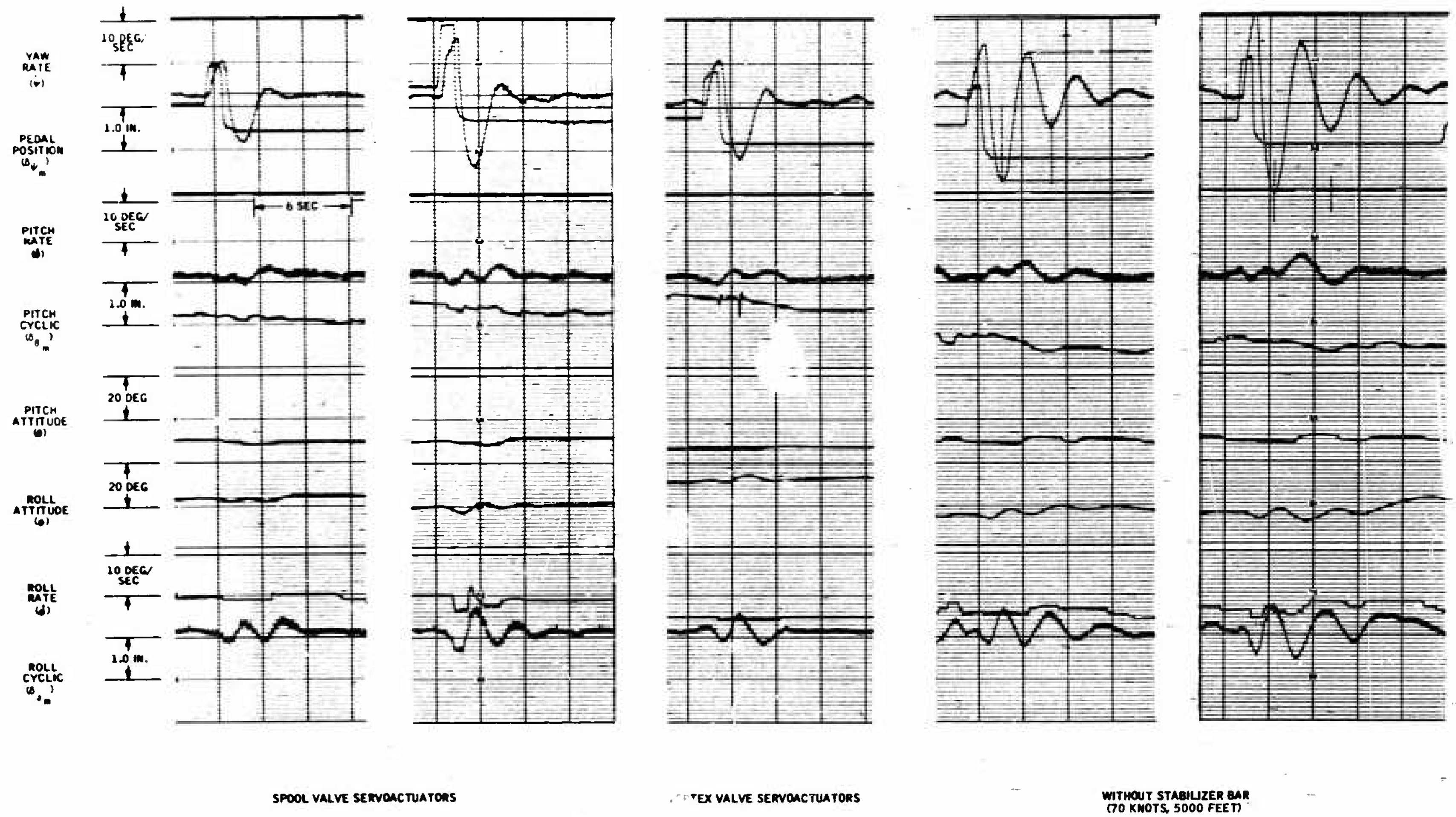


Figure 61. Yaw Pulse - Left (60 kn, 10,000 ft).

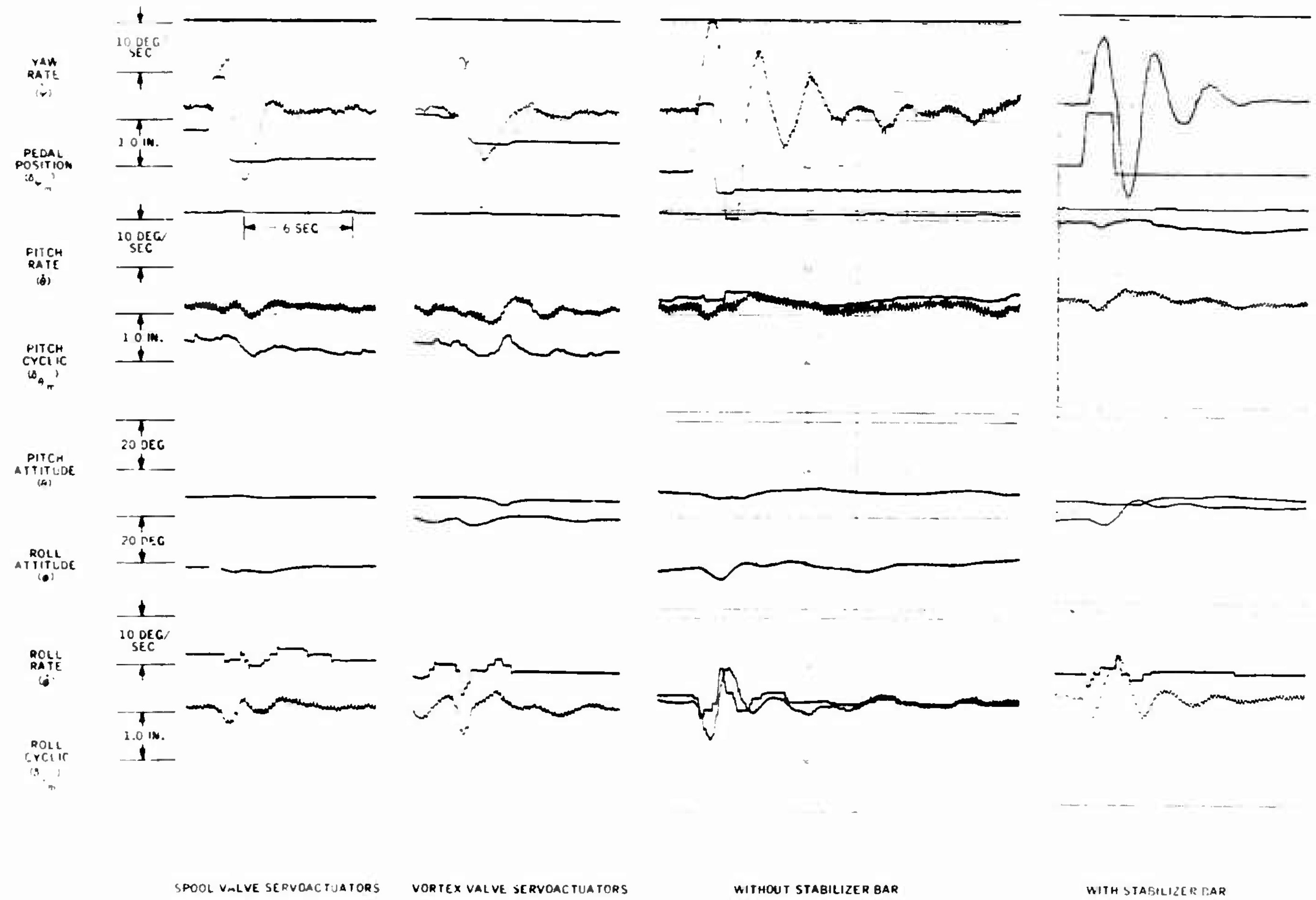


Figure 62. Yaw Pulse - Left (90 kn, 3000 ft).

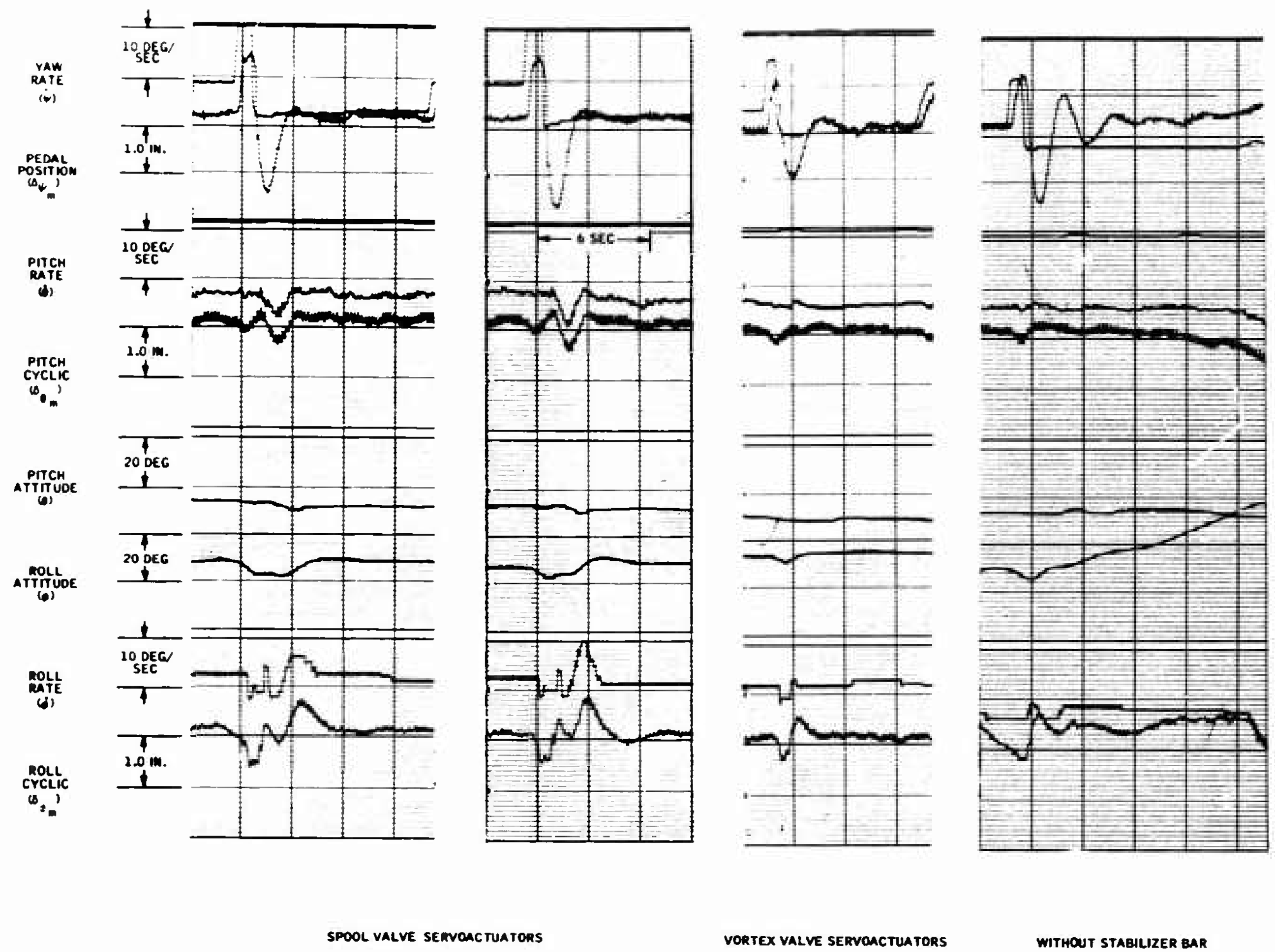


Figure 63. Yaw Pulse - Left (120 kn, 3000 ft).

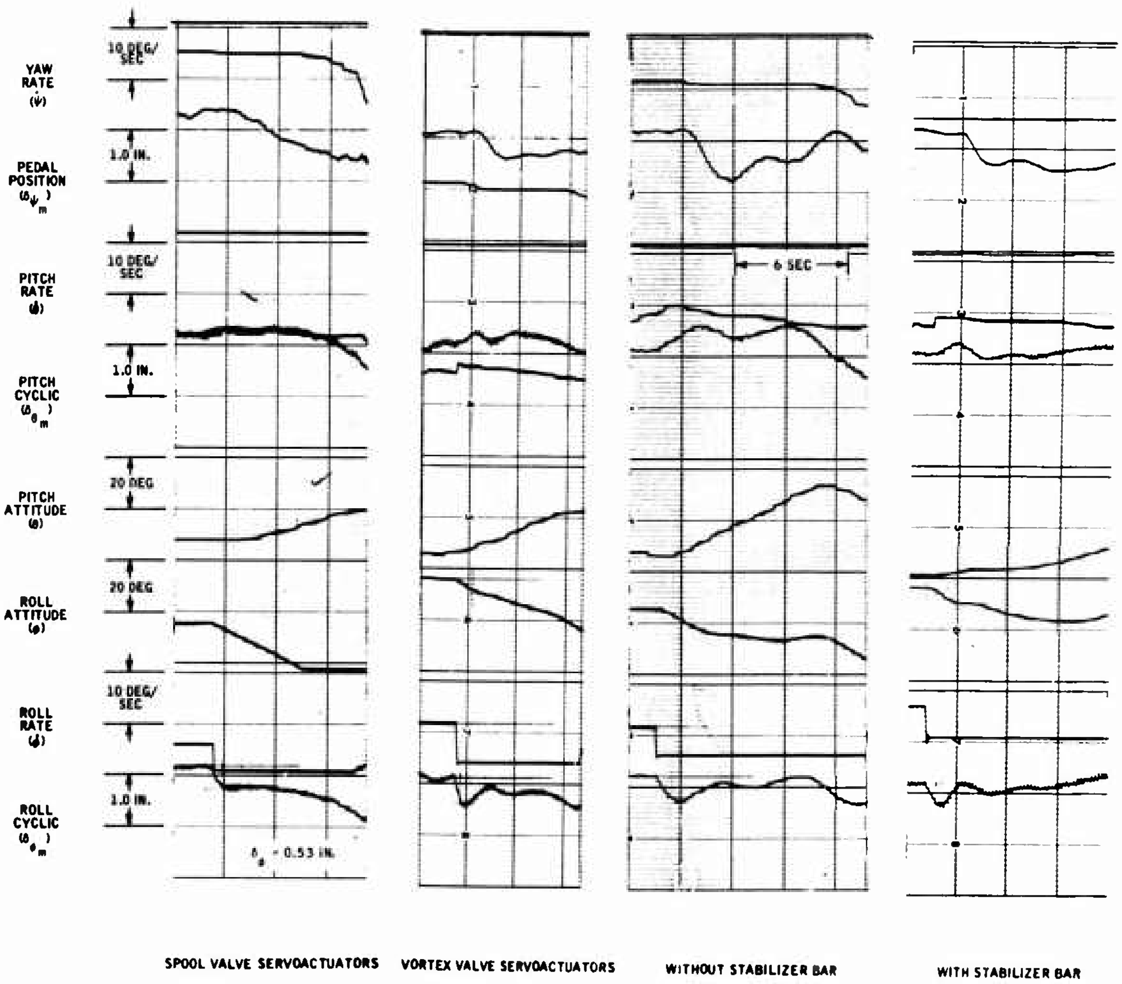


Figure 64. Roll Step - Right (Hover, 3000 ft).

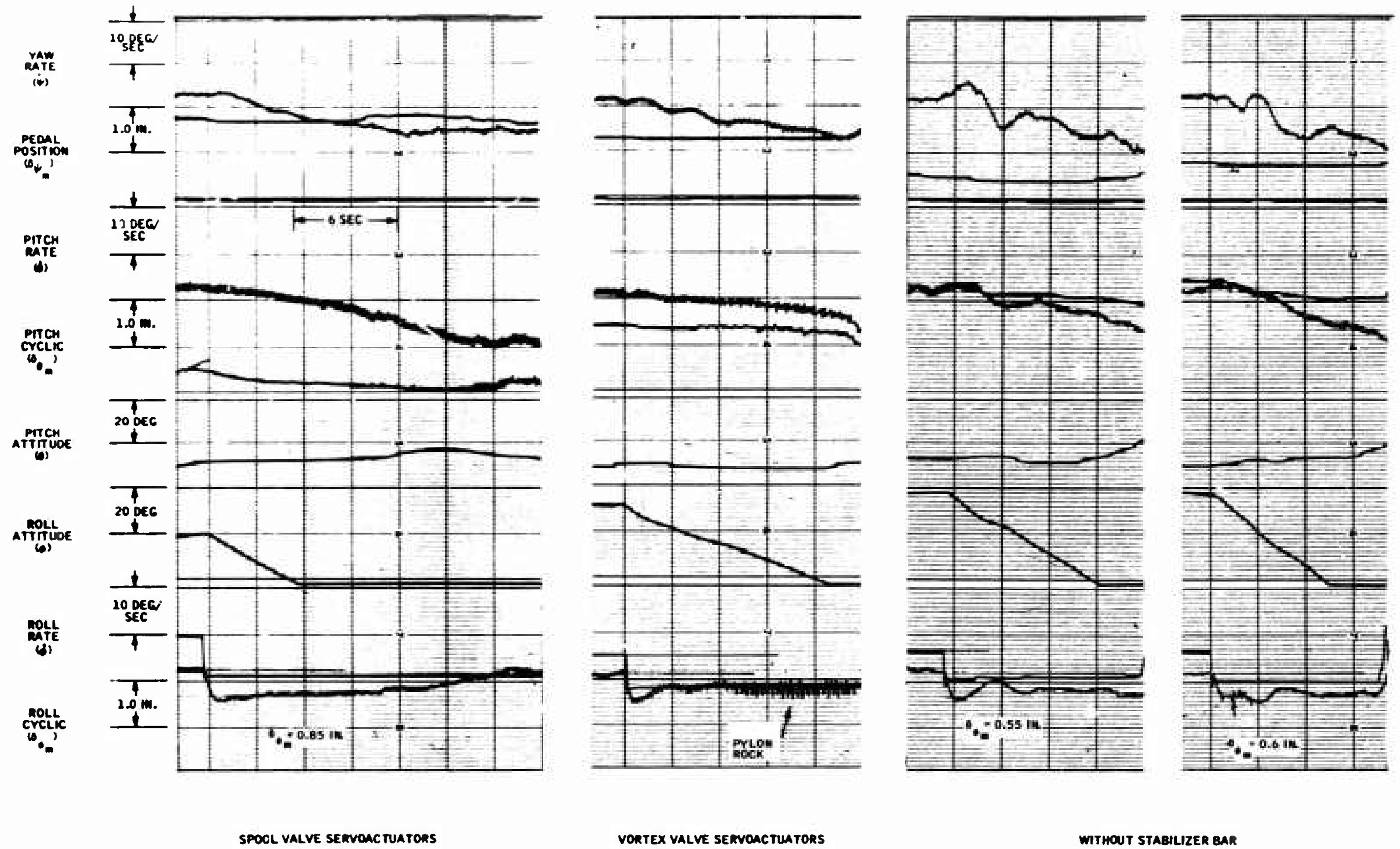


Figure 65. Roll Step - Right (60 kn, 3000 ft).

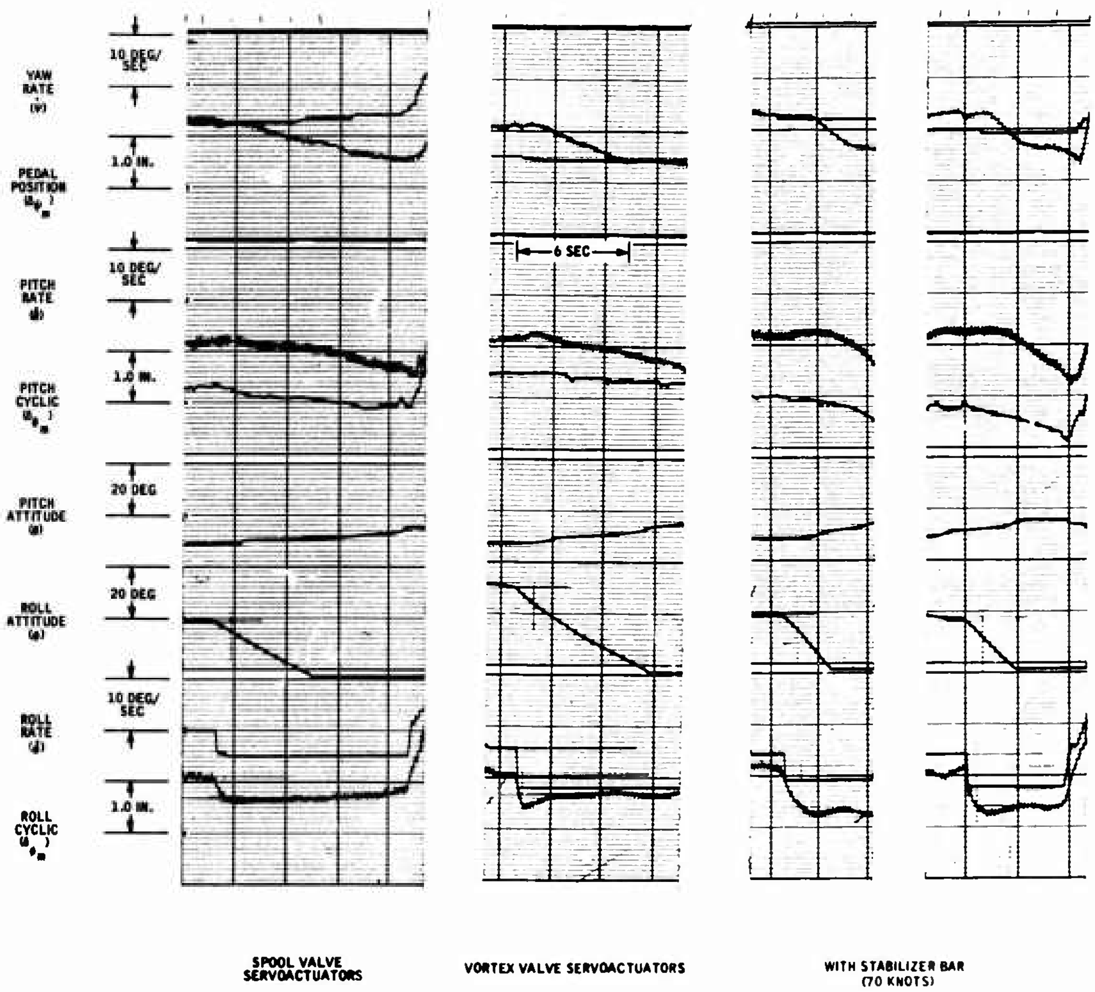


Figure 66. Roll Step - Right (60 kn, 10,000 ft).

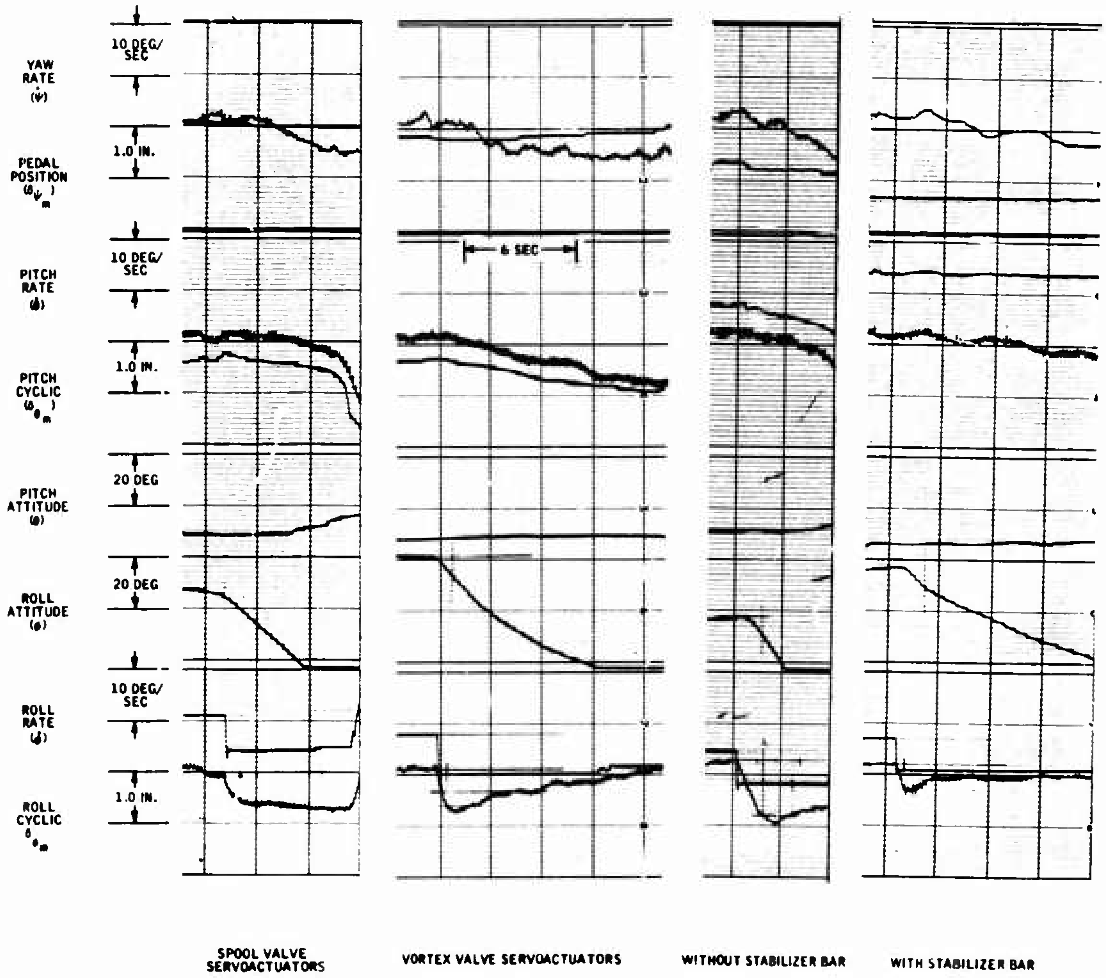


Figure 67. Roll Step - Right (90 kn, 3000 ft).

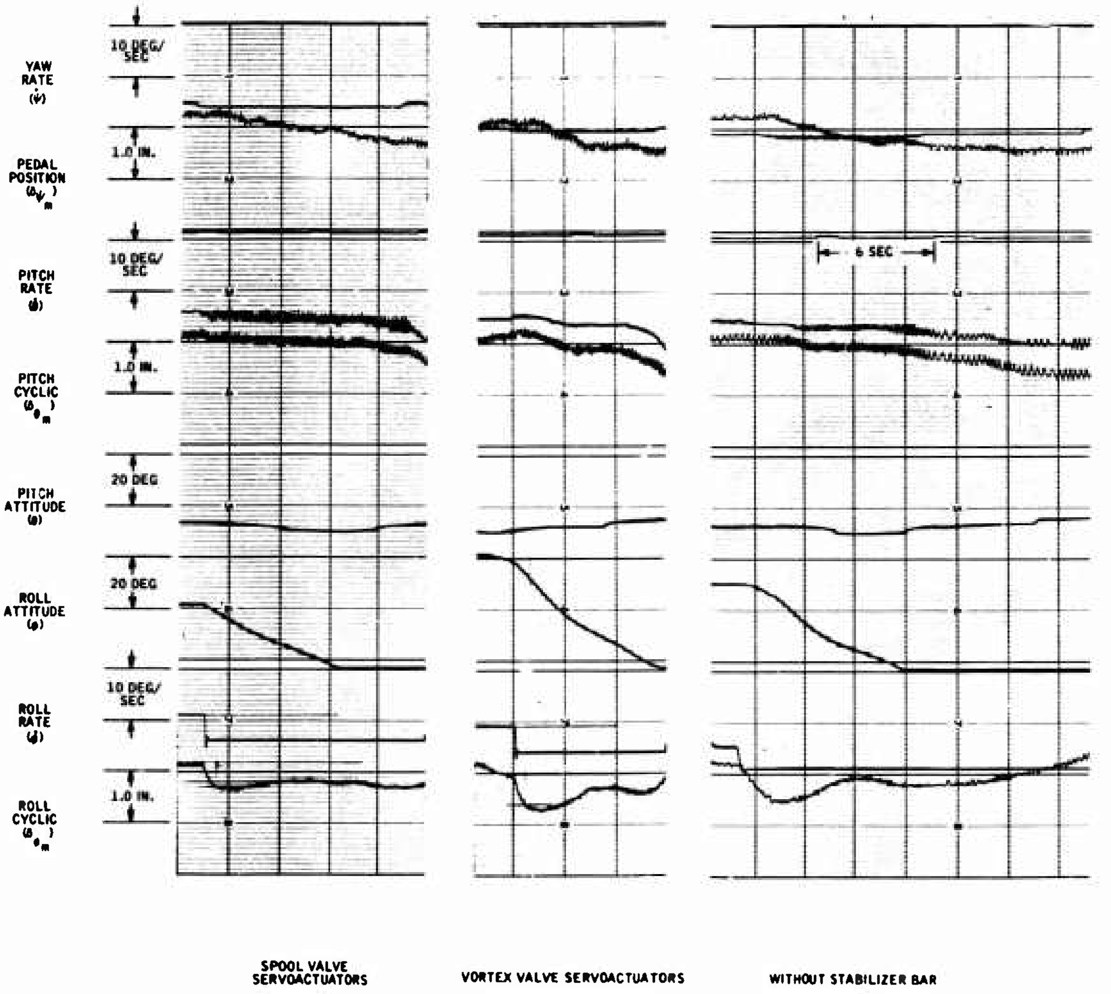


Figure 68. Roll Step - Right (120 kn, 3000 ft).

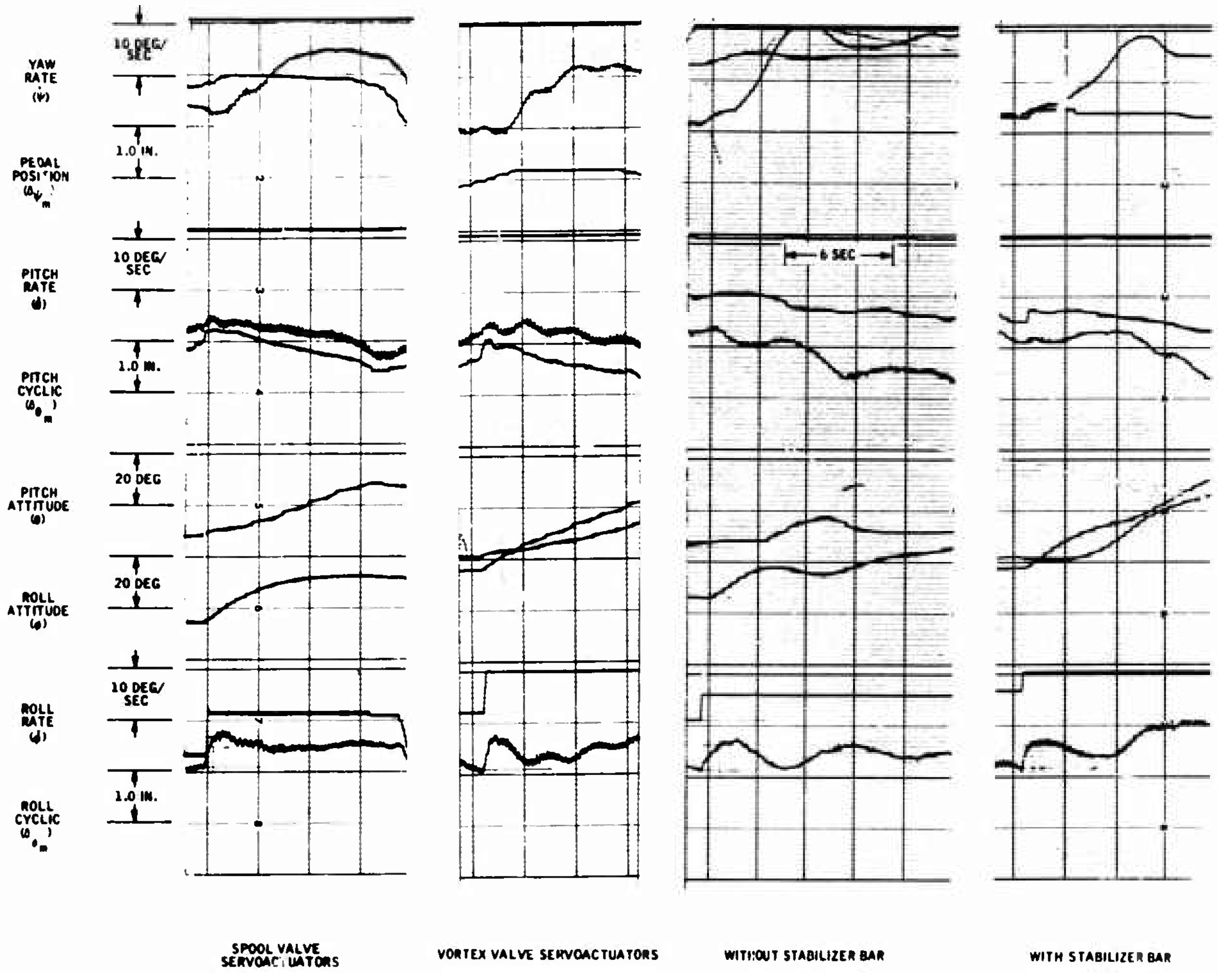


Figure 69. Roll Step - Left (Hover, 3000 ft).

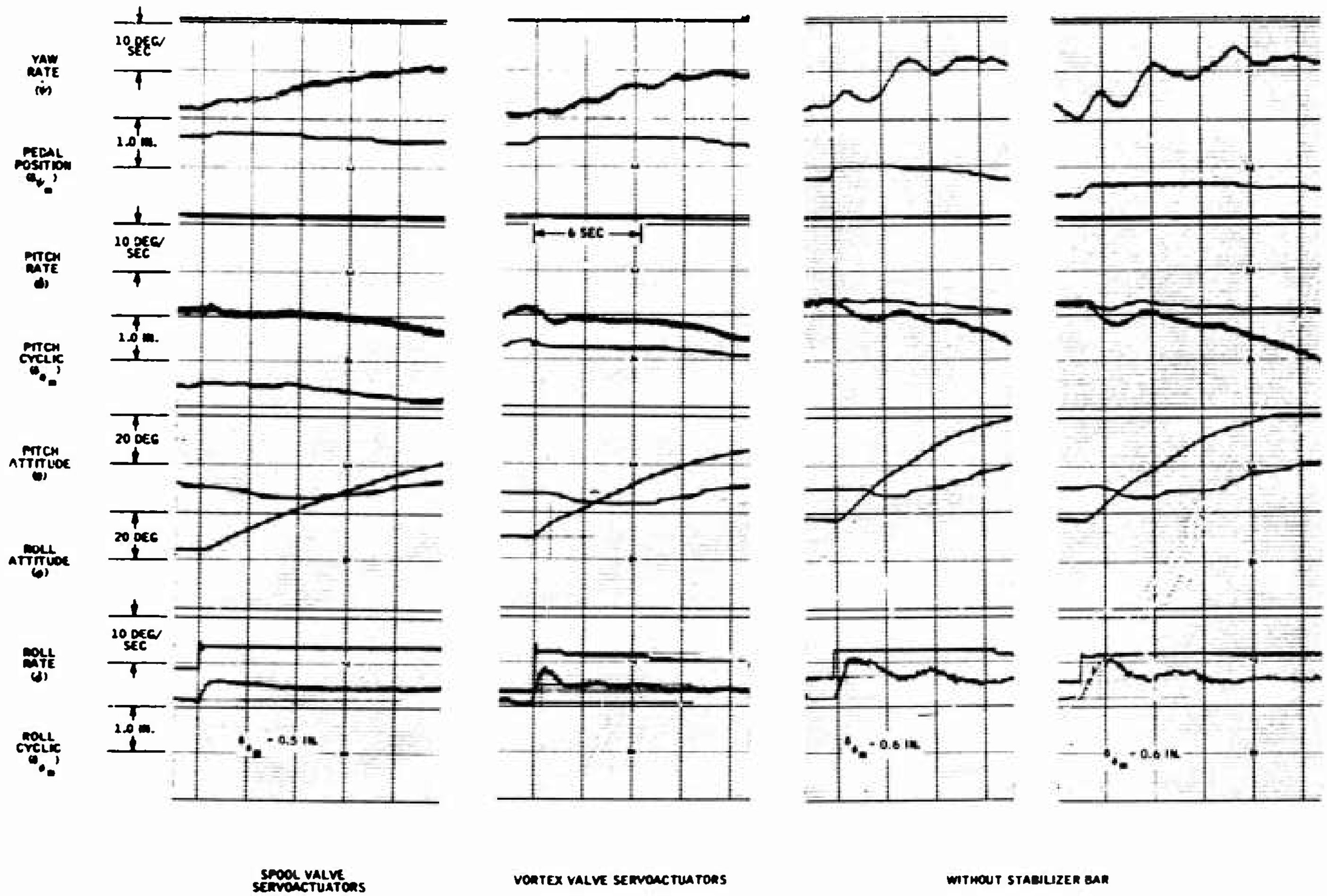


Figure 70. Roll Step - Left (60 kn, 3000 ft).

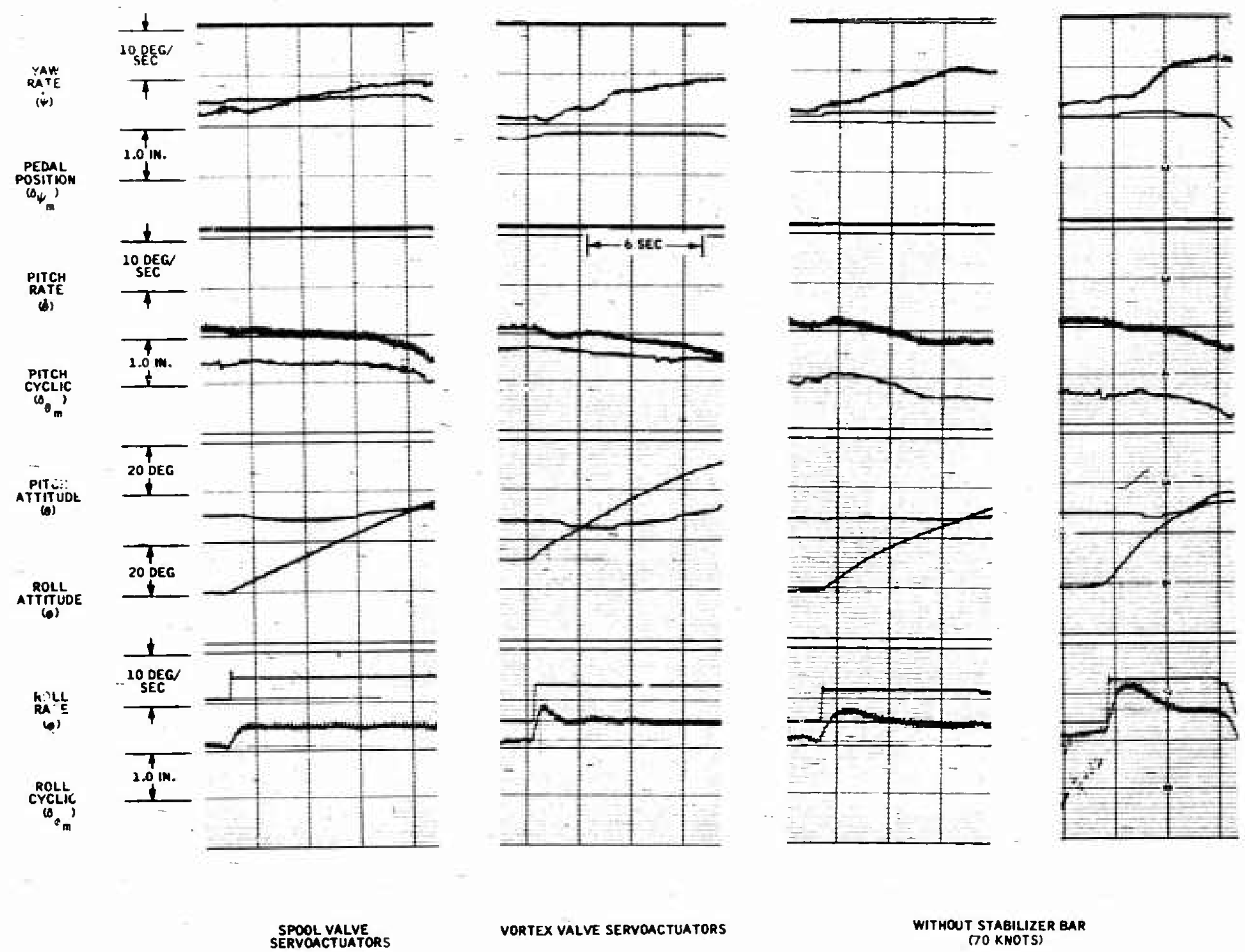


Figure 71. Roll Step - Left (60 kn, 10,000 ft).

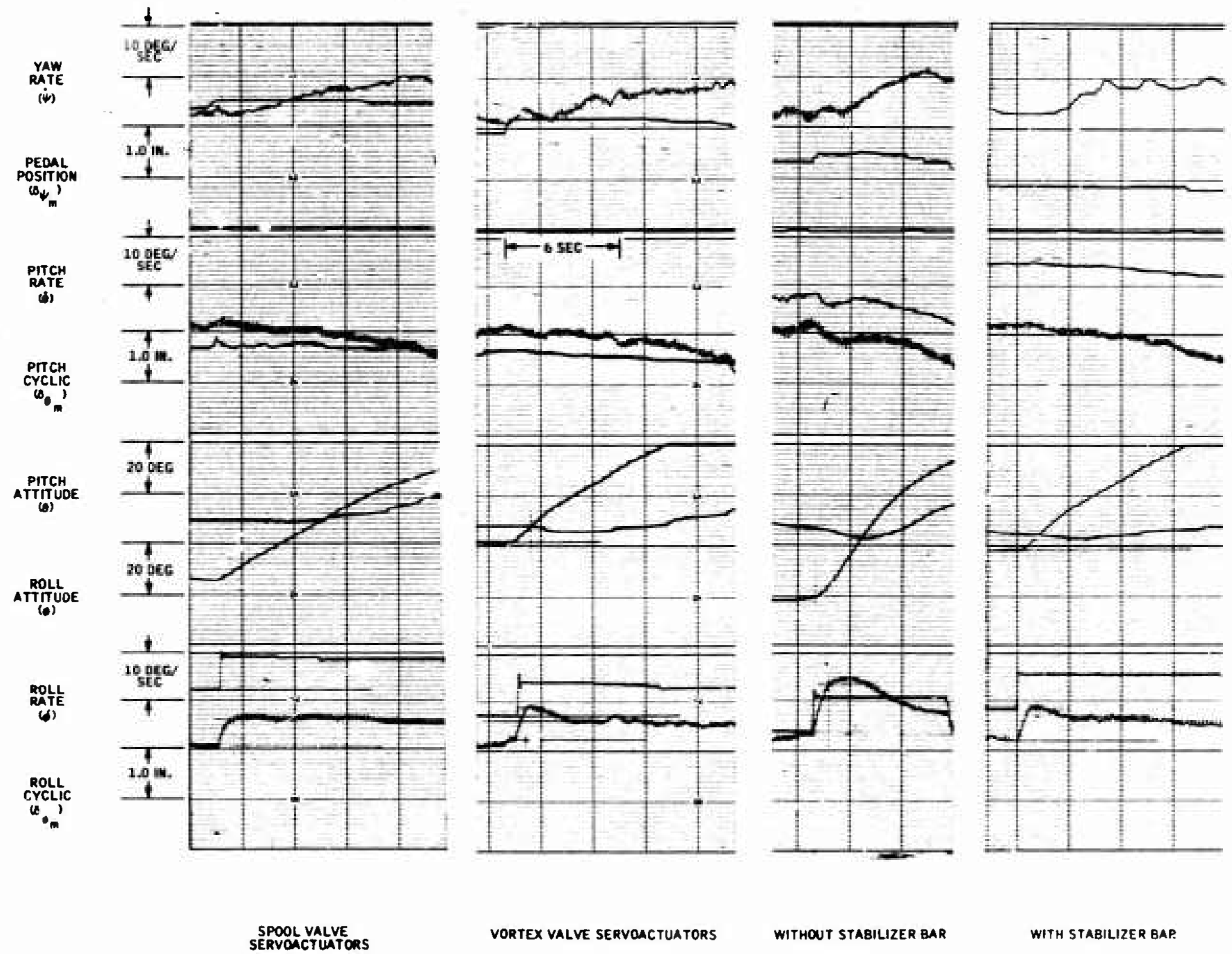


Figure 72. Roll Step - Left (90 kn, 3000 ft).

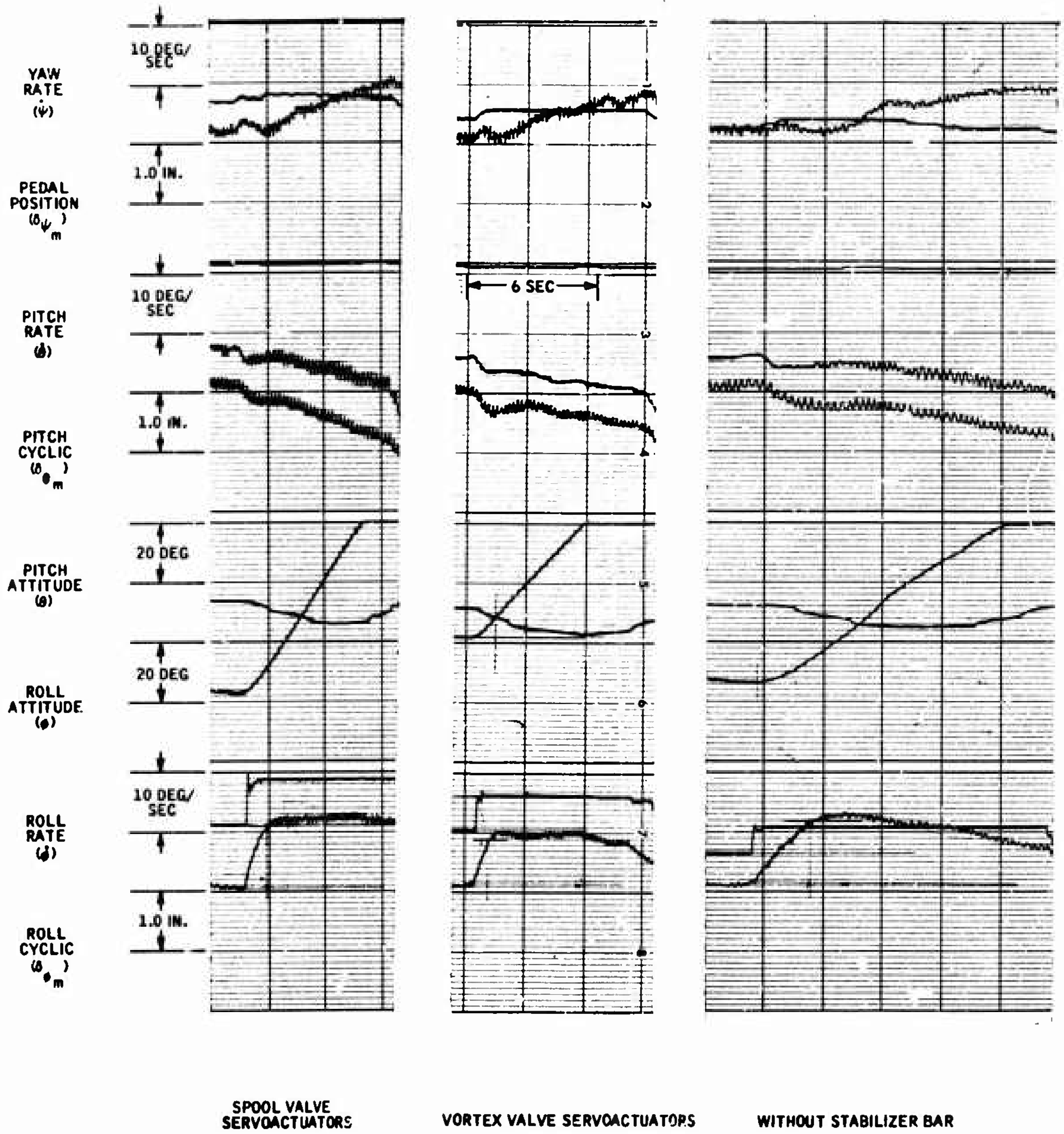


Figure 73. Roll Step - Left (120 kn, 3000 ft).

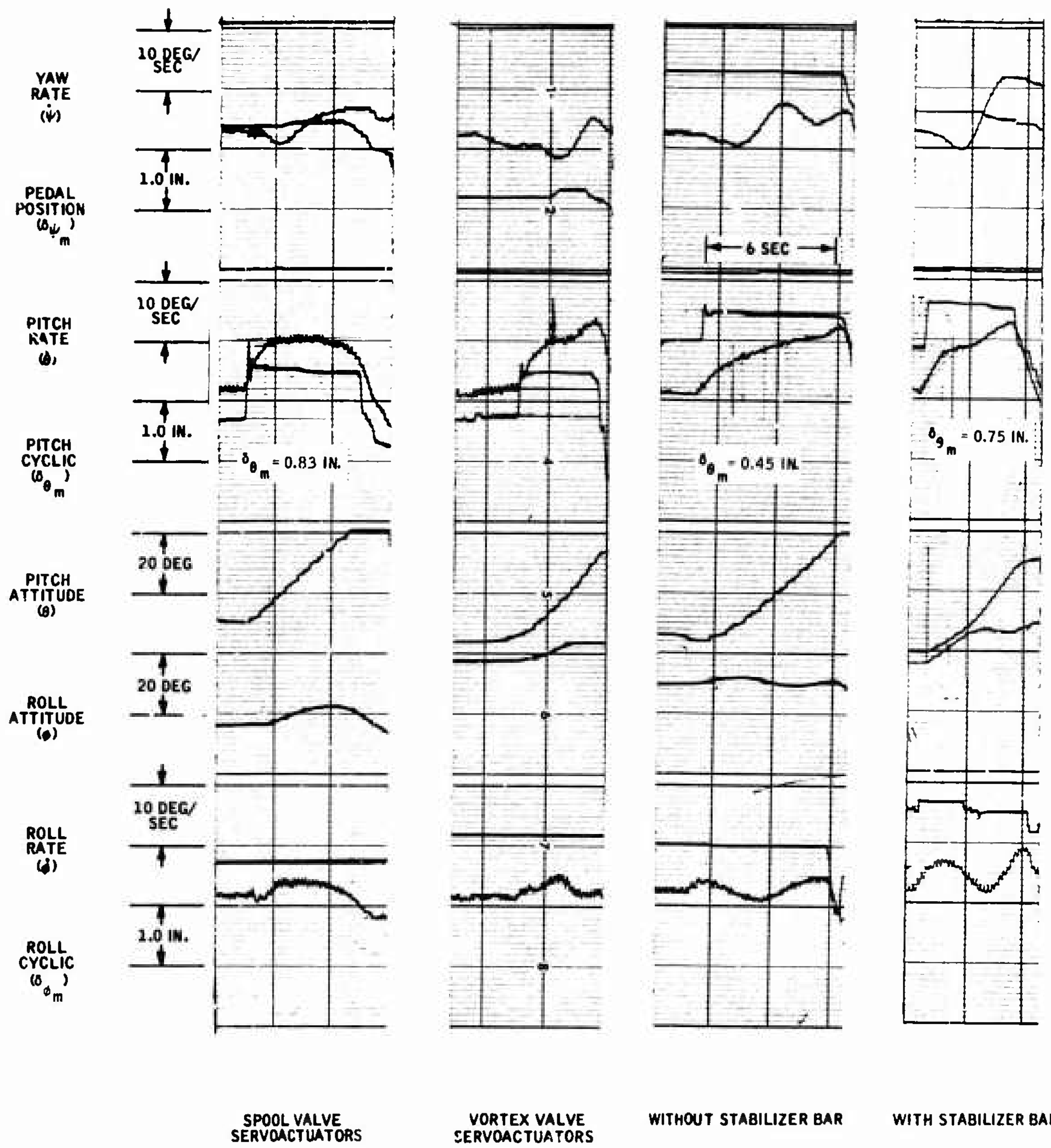


Figure 74. Pitch Step - Down (Hover, 3000 ft).

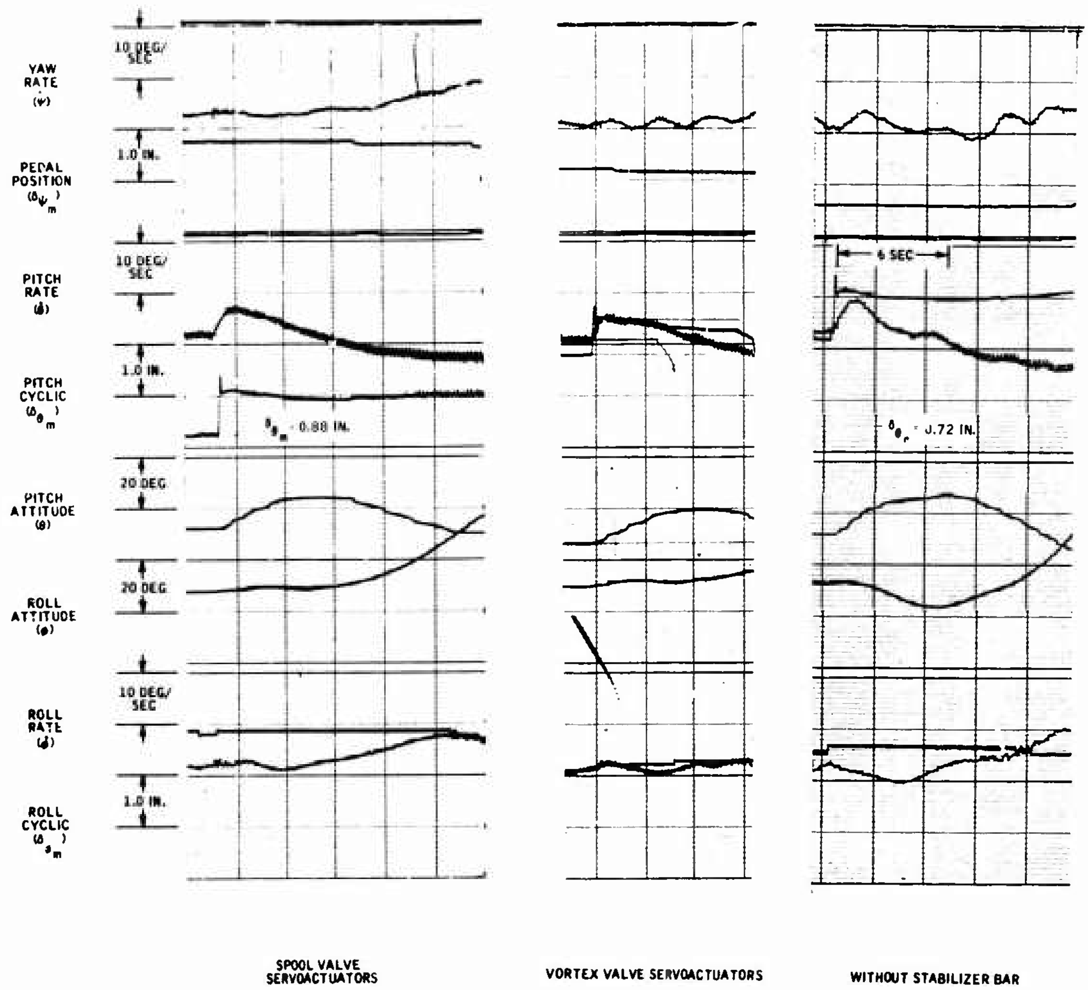


Figure 75. Pitch Step - Down (60 kn, 3000 ft).

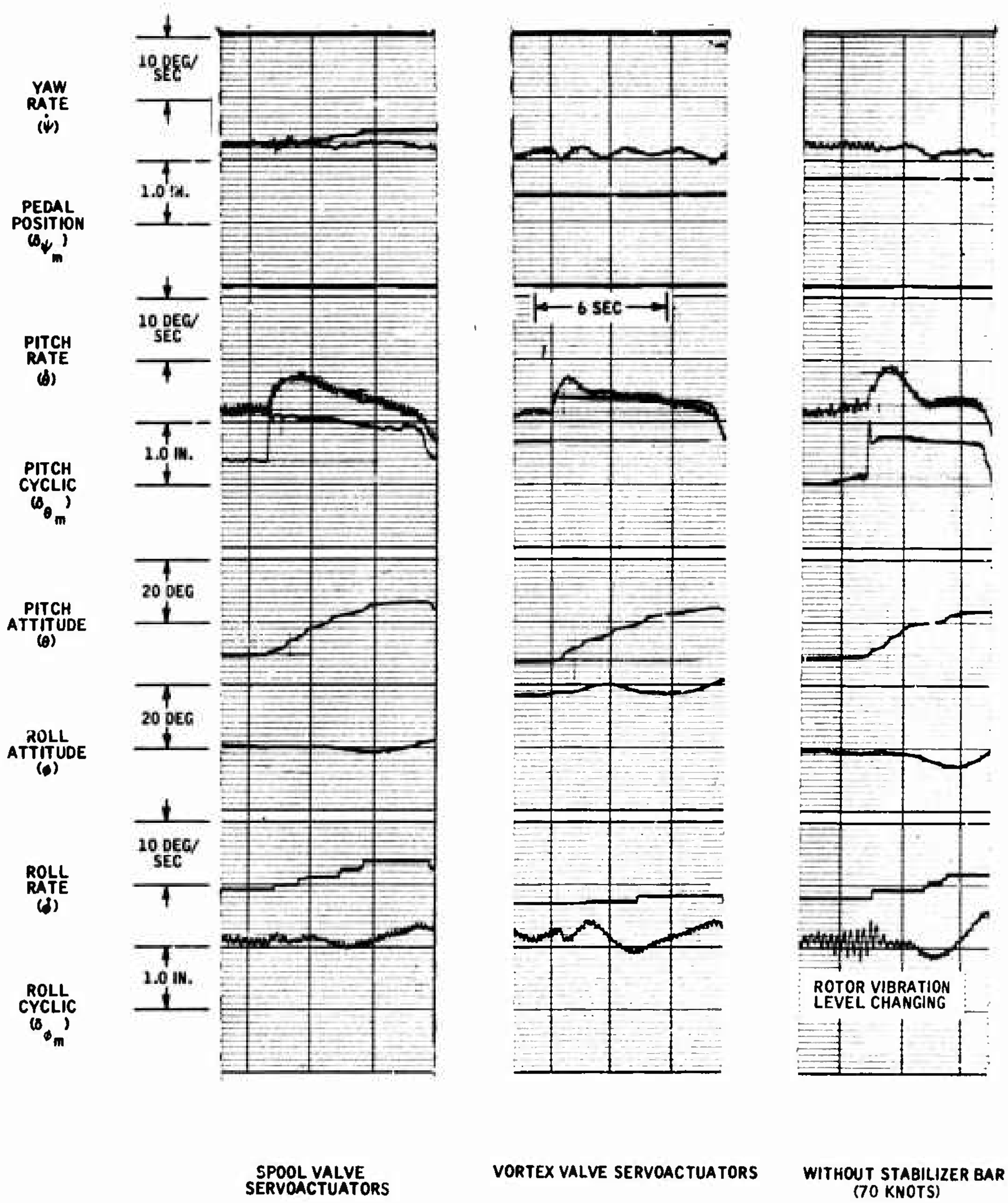


Figure 76. Pitch Step - Down (60 kn, 10,000 ft).

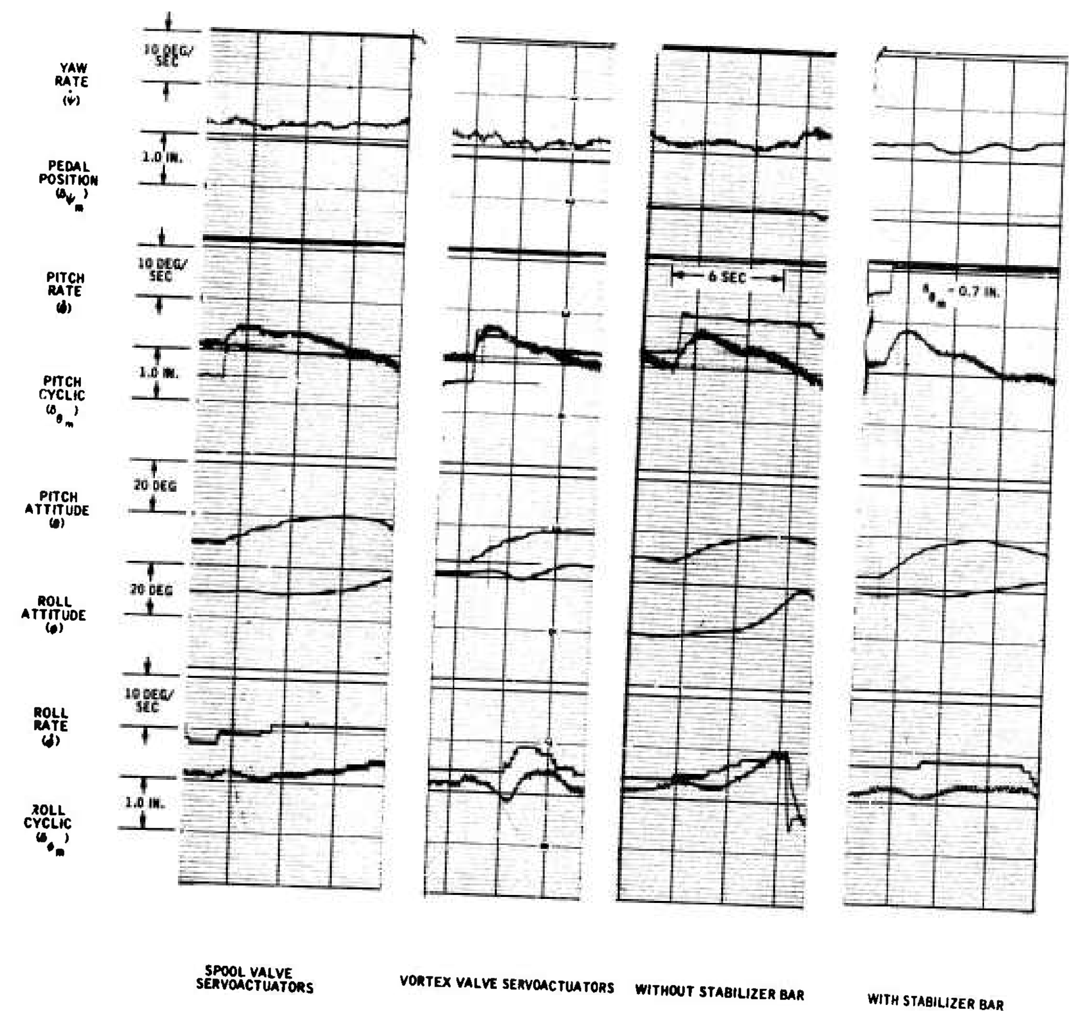


Figure 77. Pitch Step - Down (90 kn, 3000 ft).

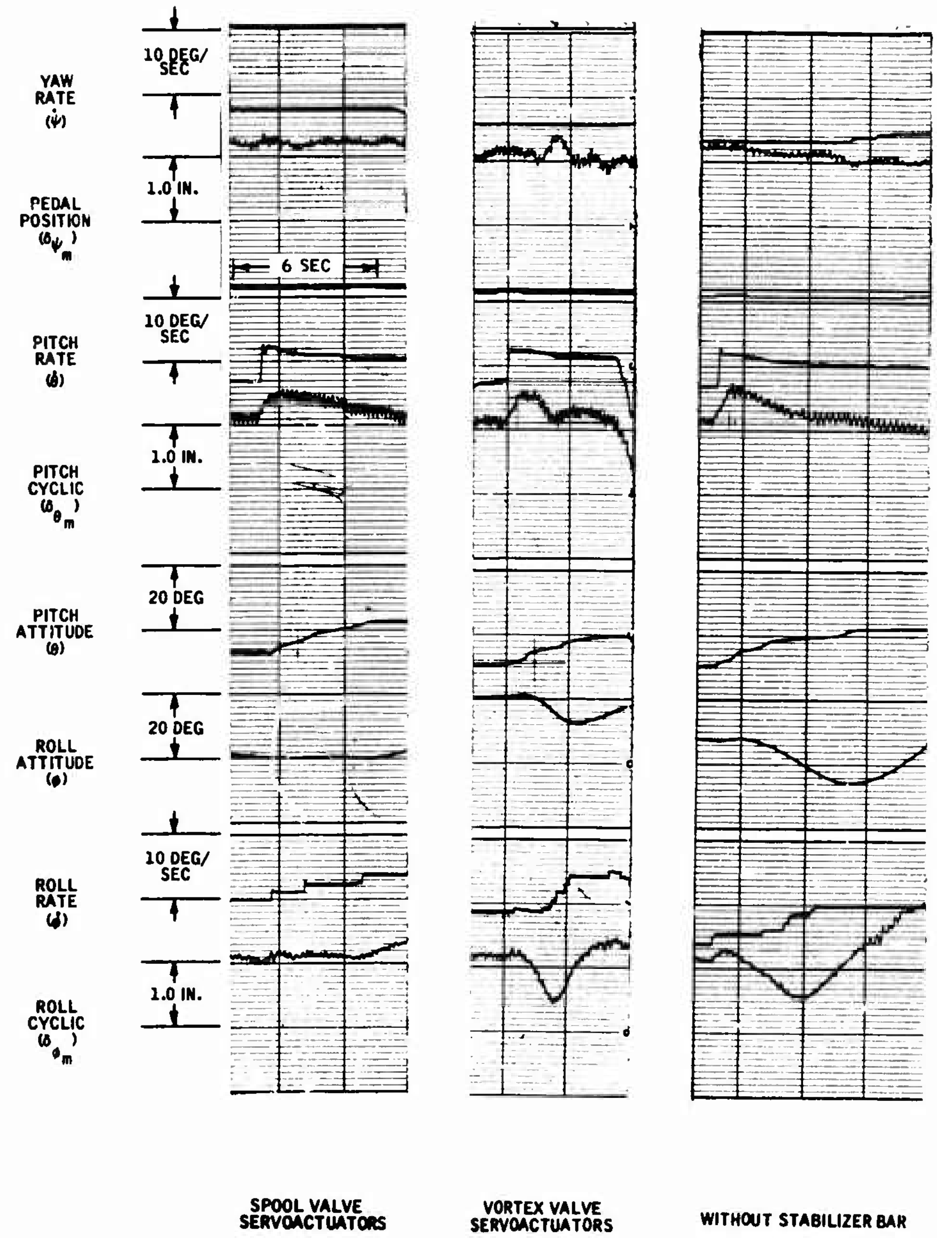


Figure 78. Pitch Step - Down (120 kn, 3000 ft).

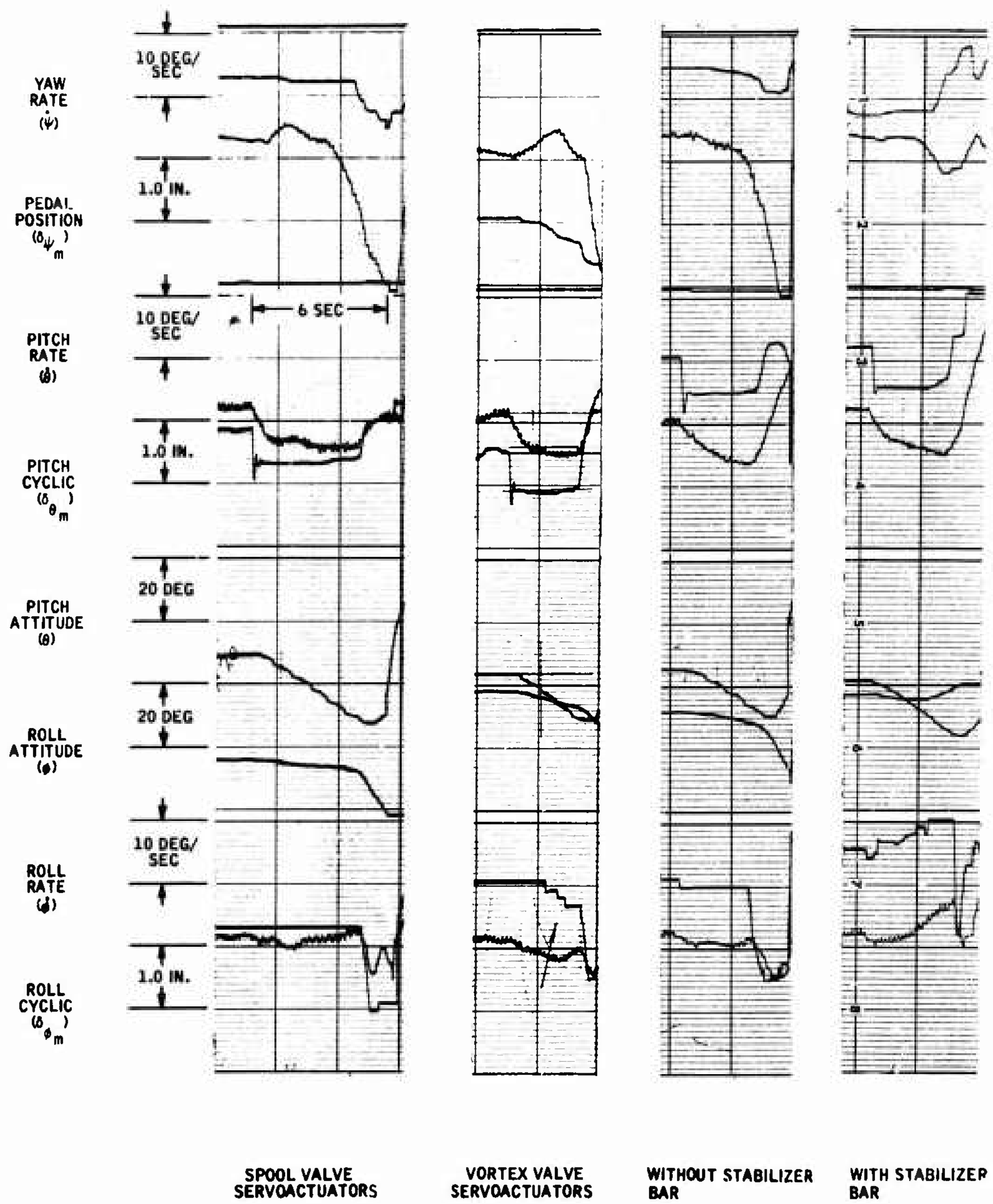


Figure 79. Pitch Step - Up (Hover, 3000 ft).

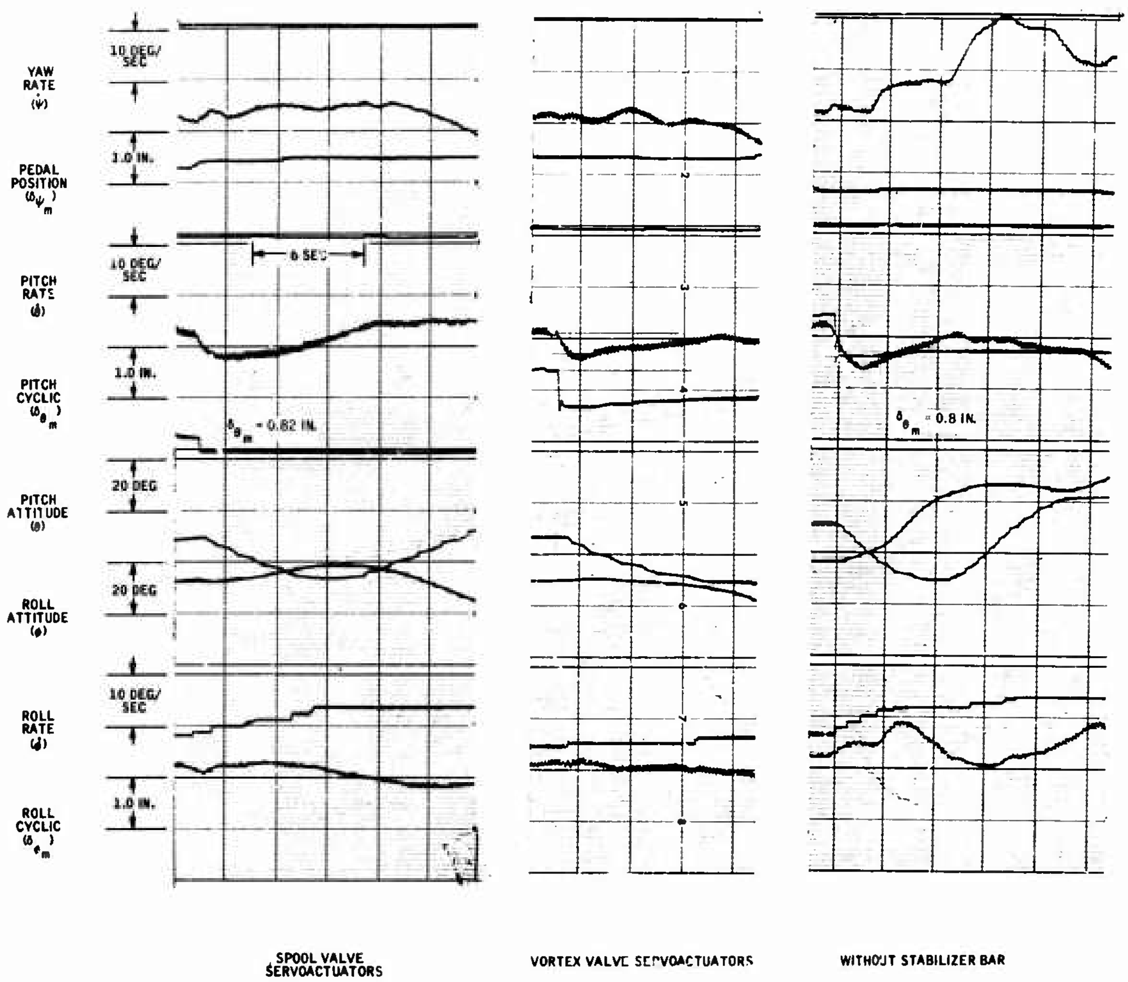


Figure 80. Pitch Step - Up (60 kn, 3000 ft).

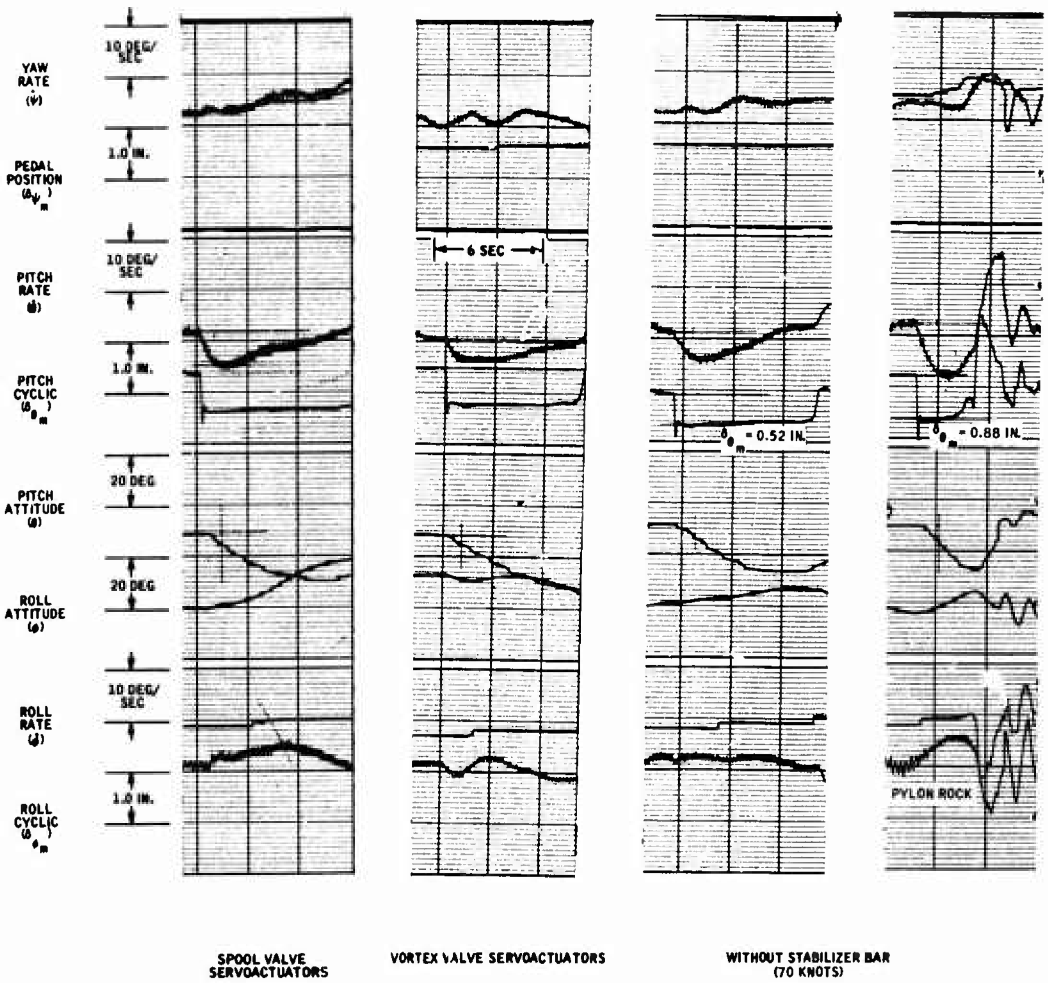


Figure 81. Pitch Step - Up (60 kn, 10,000 ft).

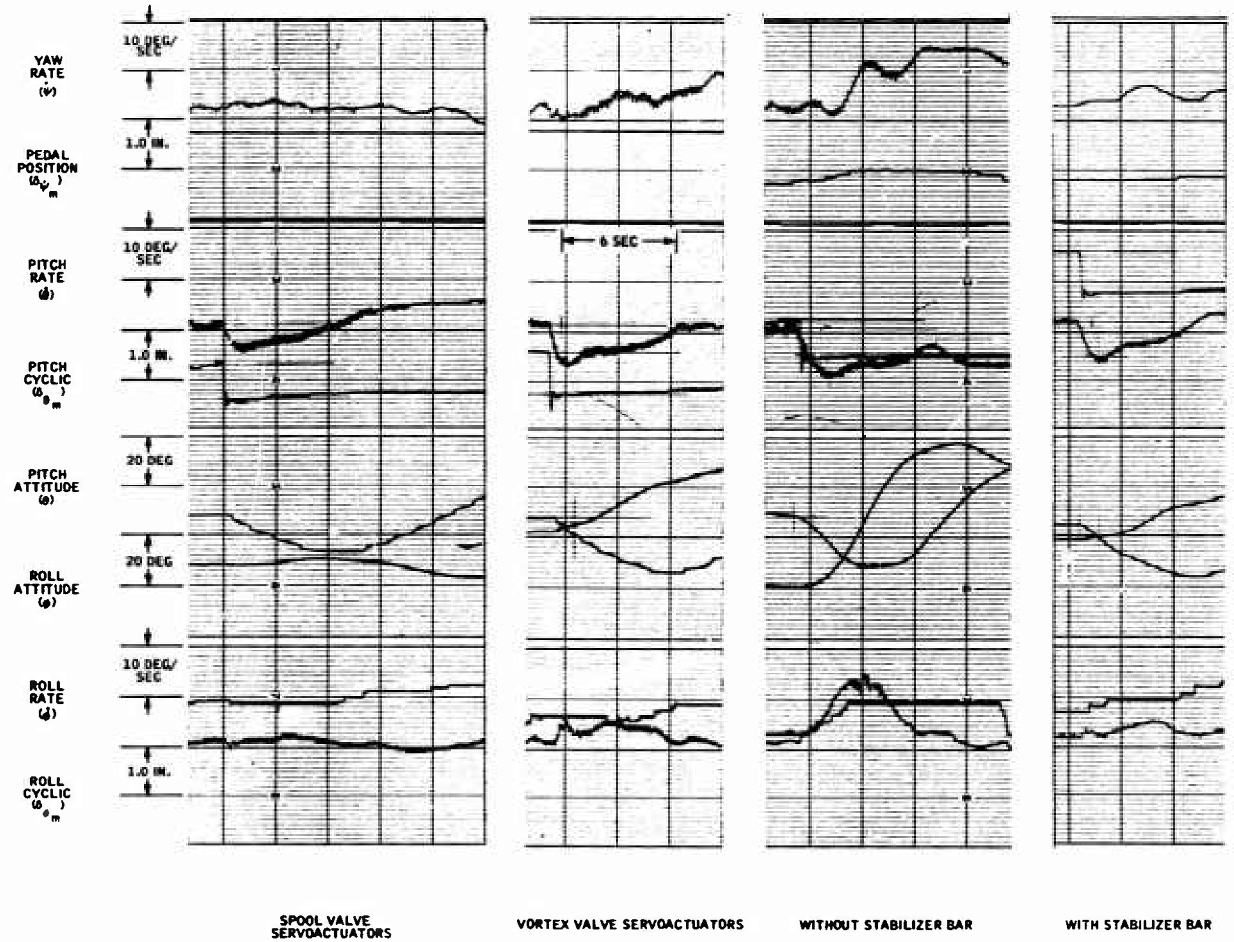


Figure 82. Pitch Step - Up (90 kn, 3000 ft).

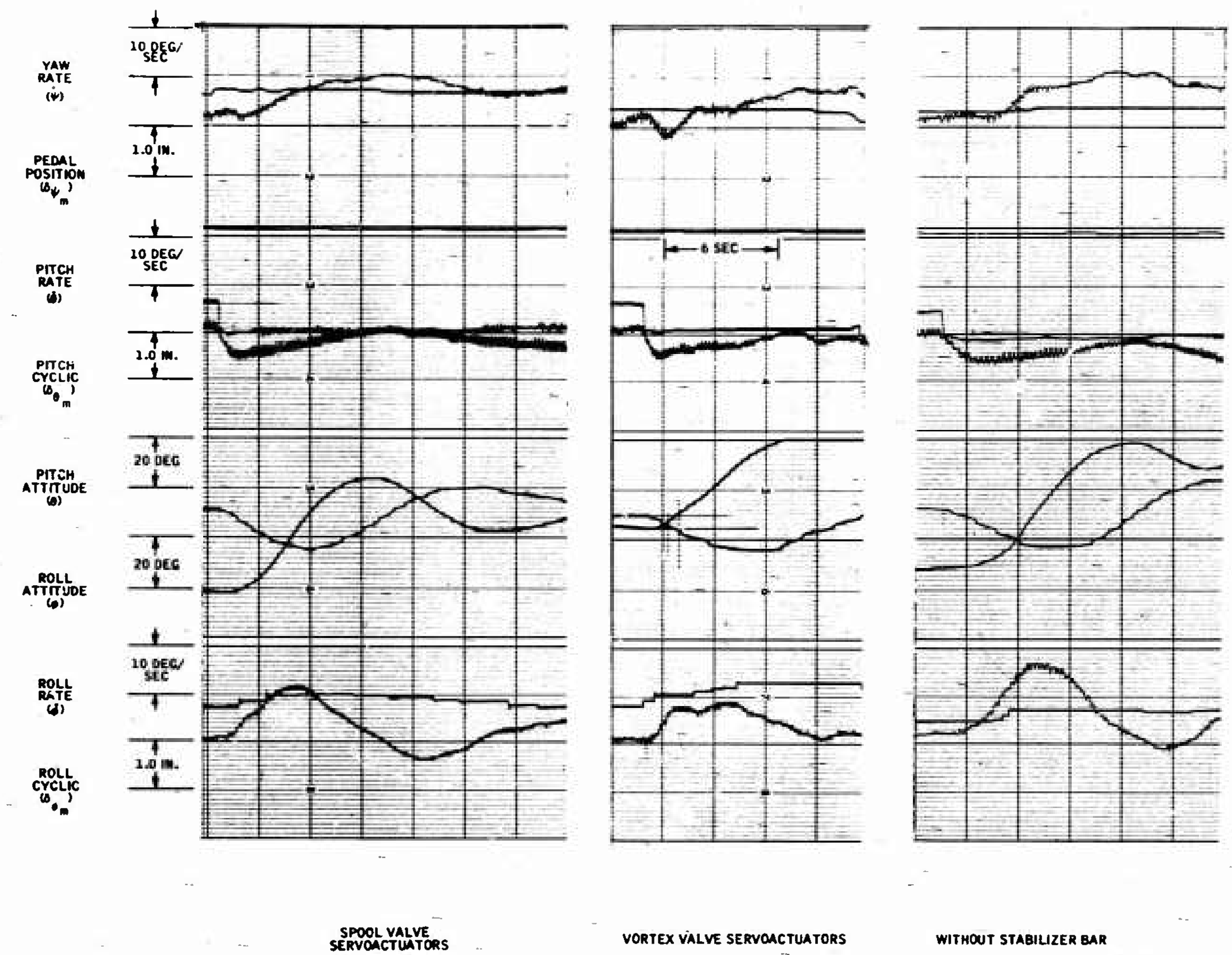


Figure 83. Pitch Step - Up (120 kn, 3000 ft).

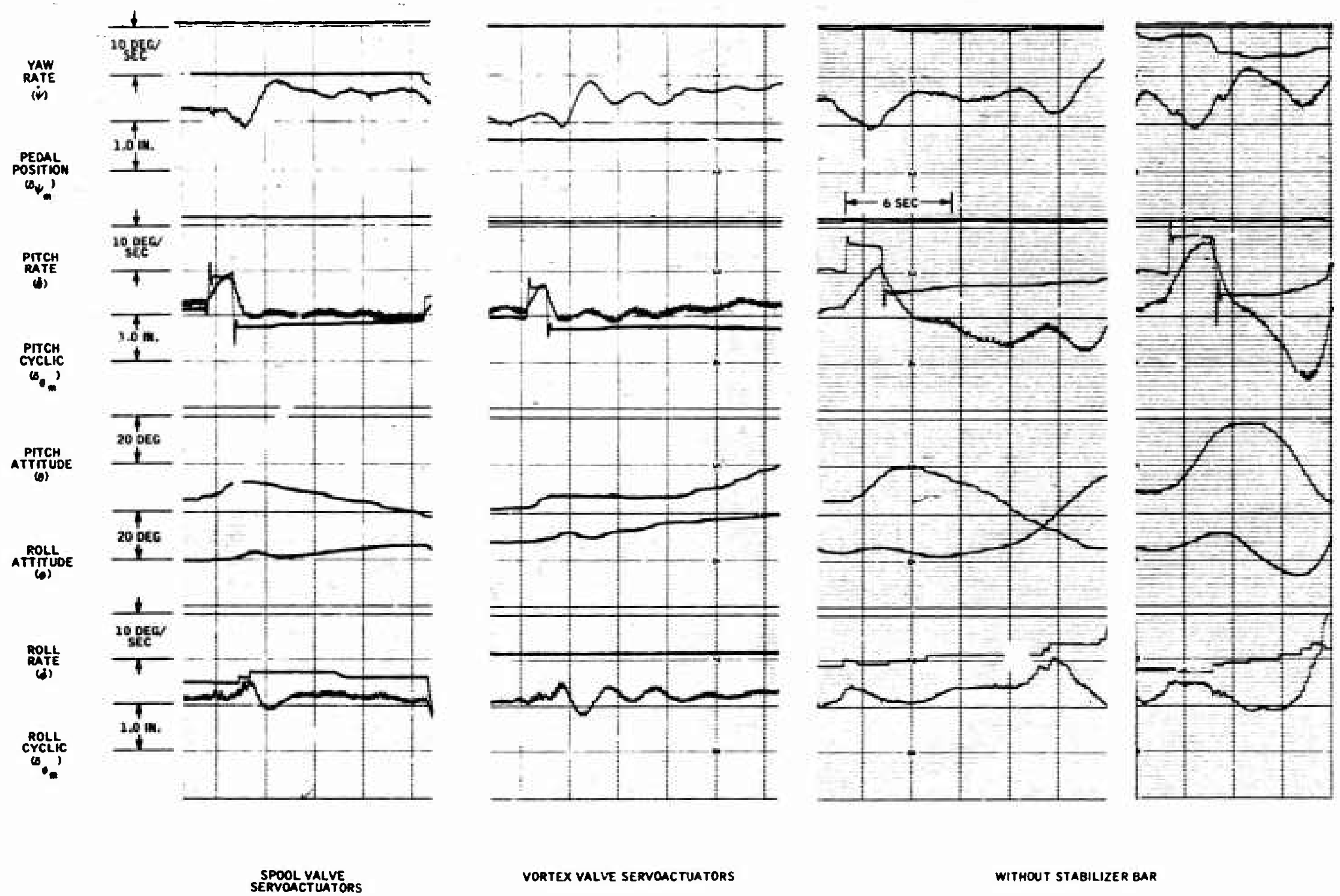


Figure 84. Pitch Pulse - Down (Hover, 3000 ft).

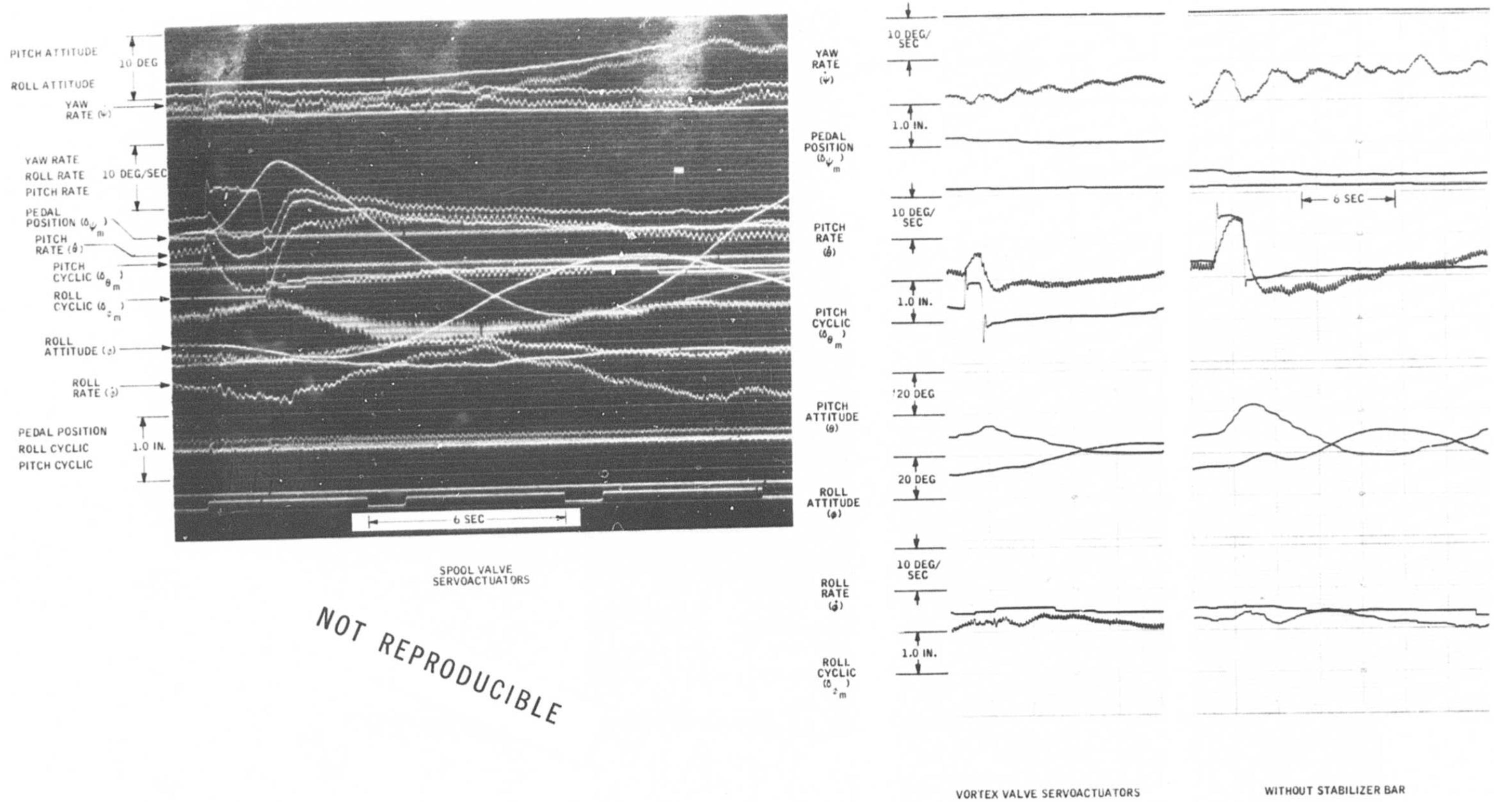


Figure 85. Pitch Pulse - Down (60 kn, 3000 ft).

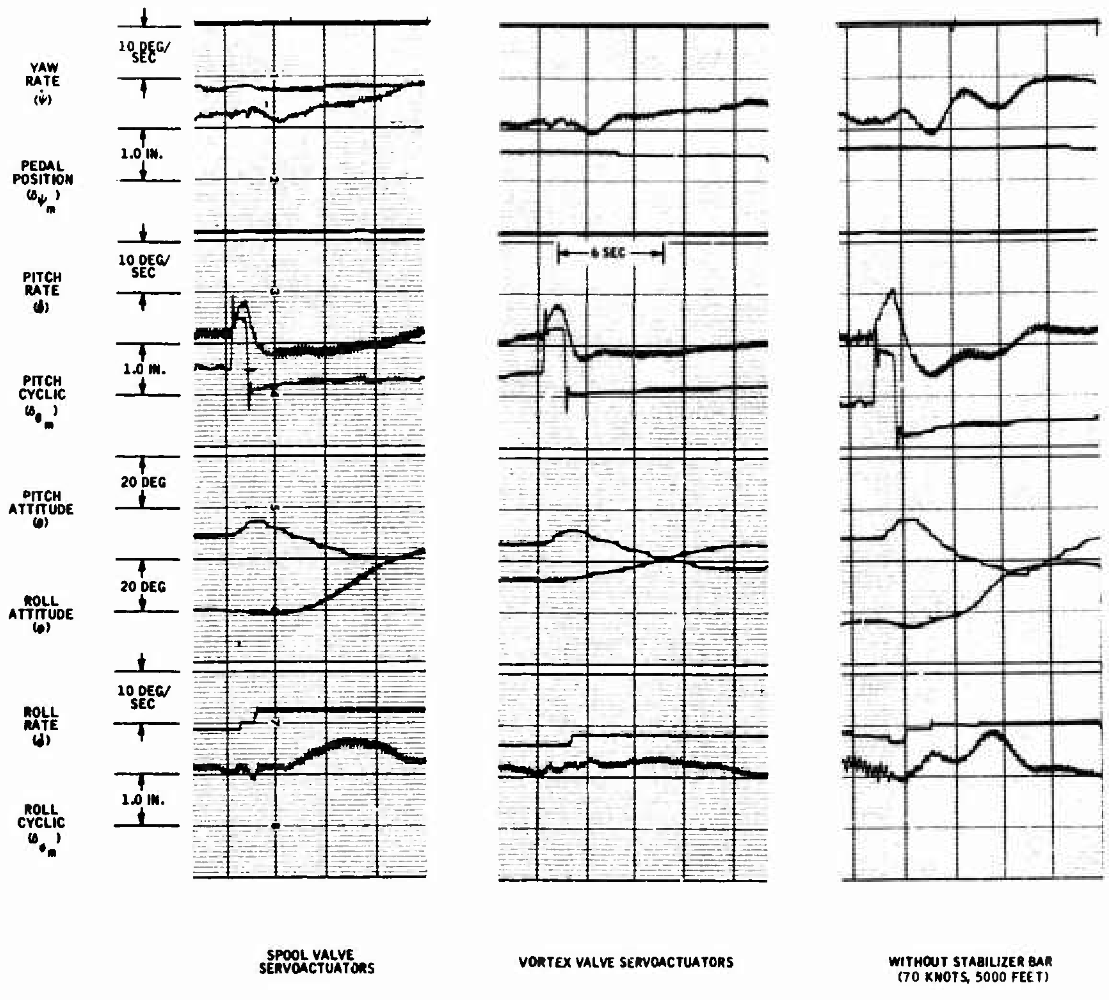


Figure 86. Pitch Pulse - Down (60 kn, 10,000 ft).

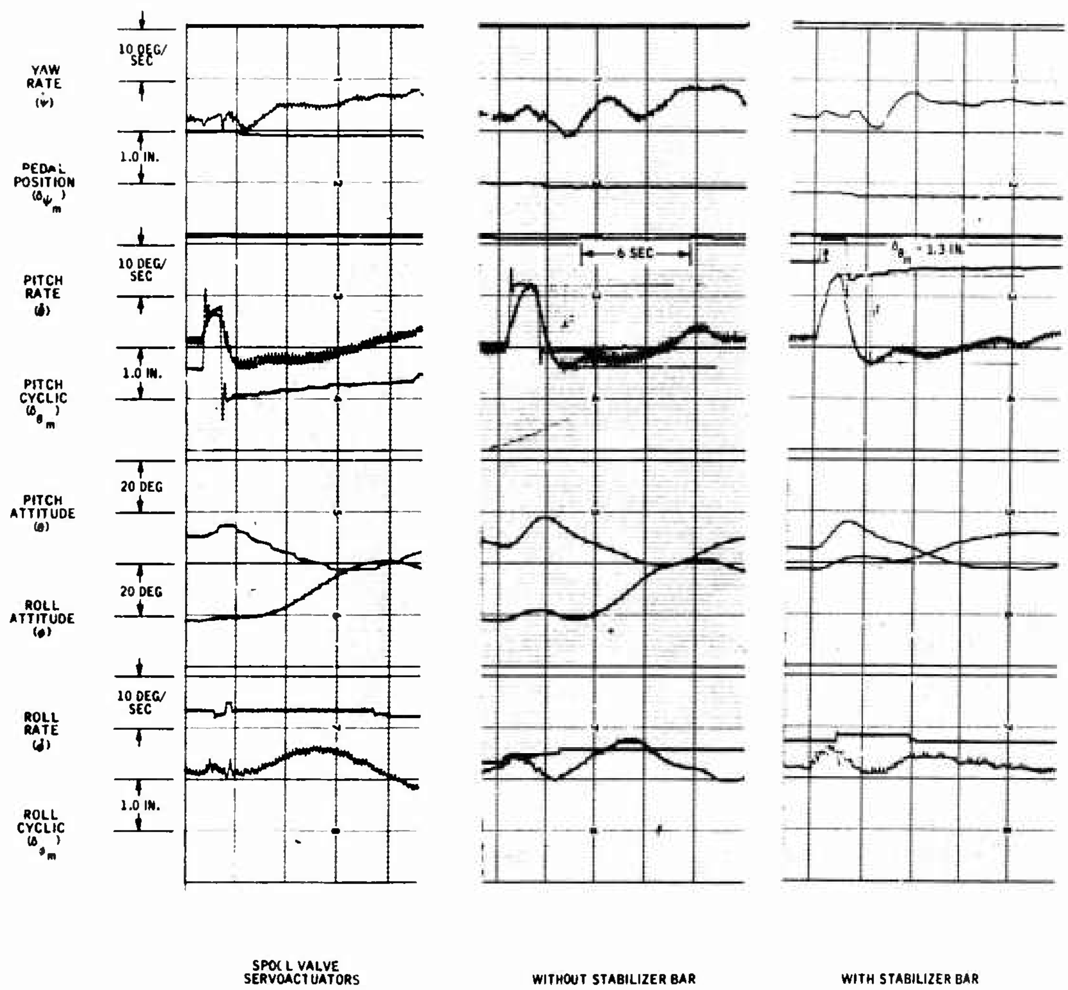


Figure 87. Pitch Pulse - Down (90 kn, 3000 ft).

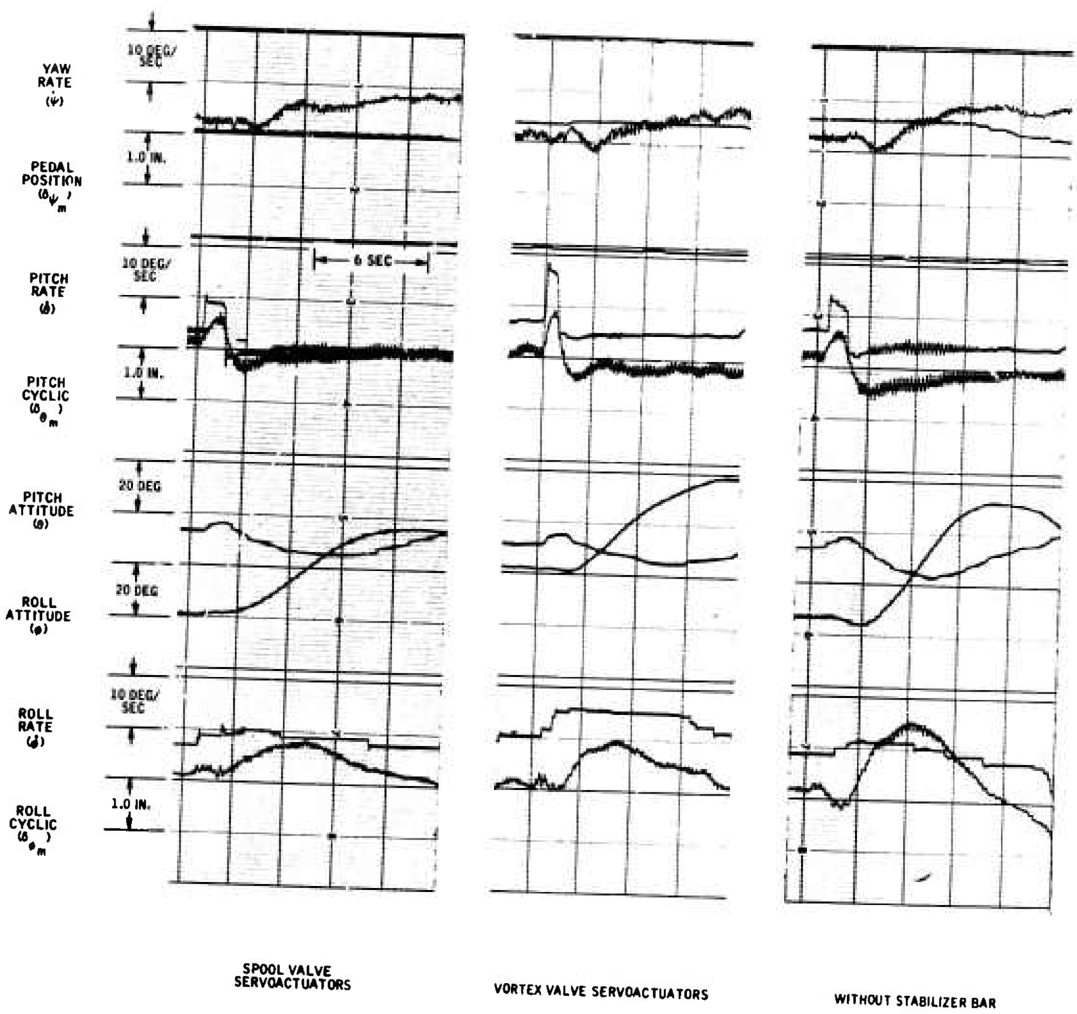


Figure 88. Pitch Pulse - Down (120 kn, 3000 ft).

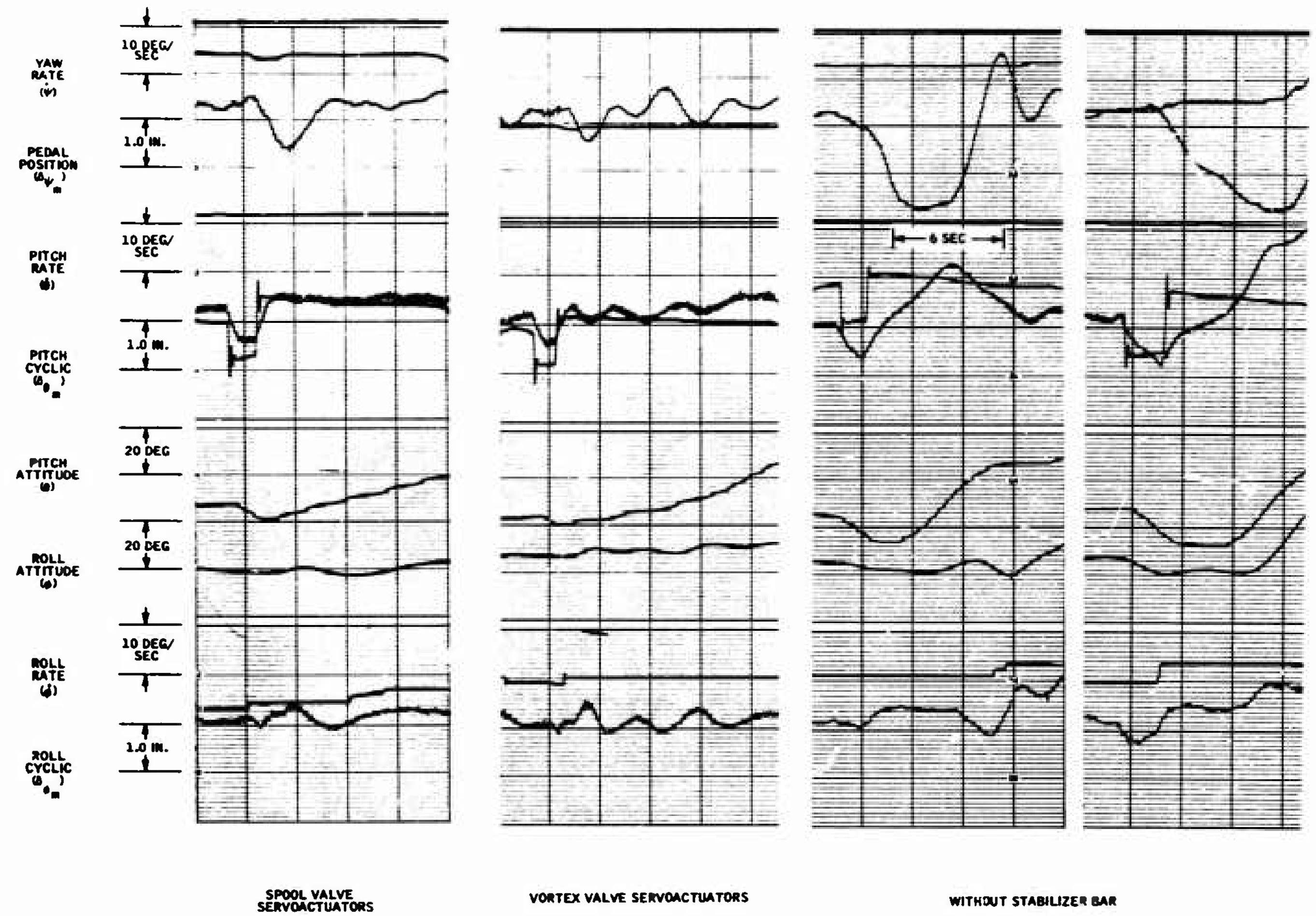


Figure 89. Pitch Pulse - Up (Hover, 3000 ft).

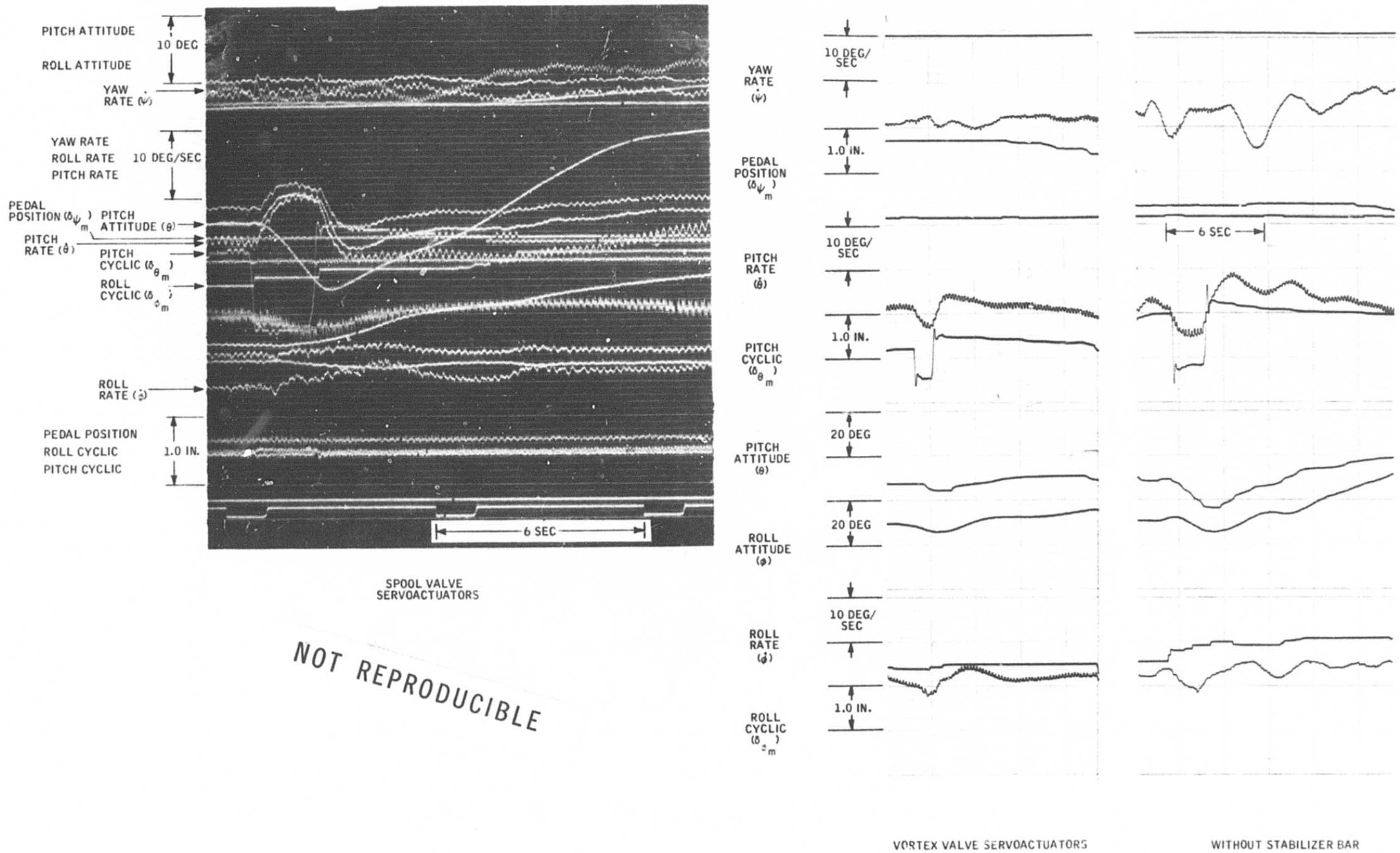


Figure 90. Pitch Pulse - Up (60 kn, 3000 ft).

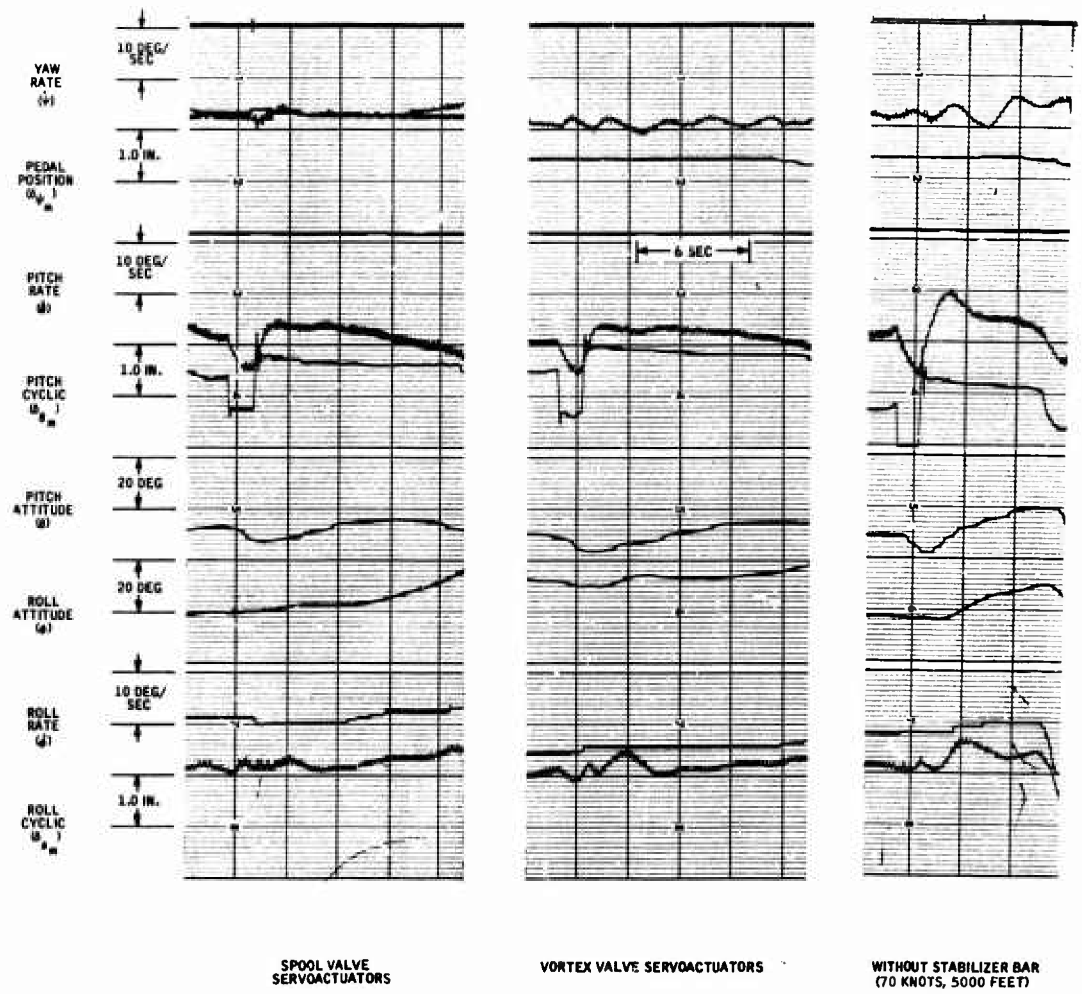


Figure 91. Pitch Pulse - Up (60 kn, 10,000 ft).

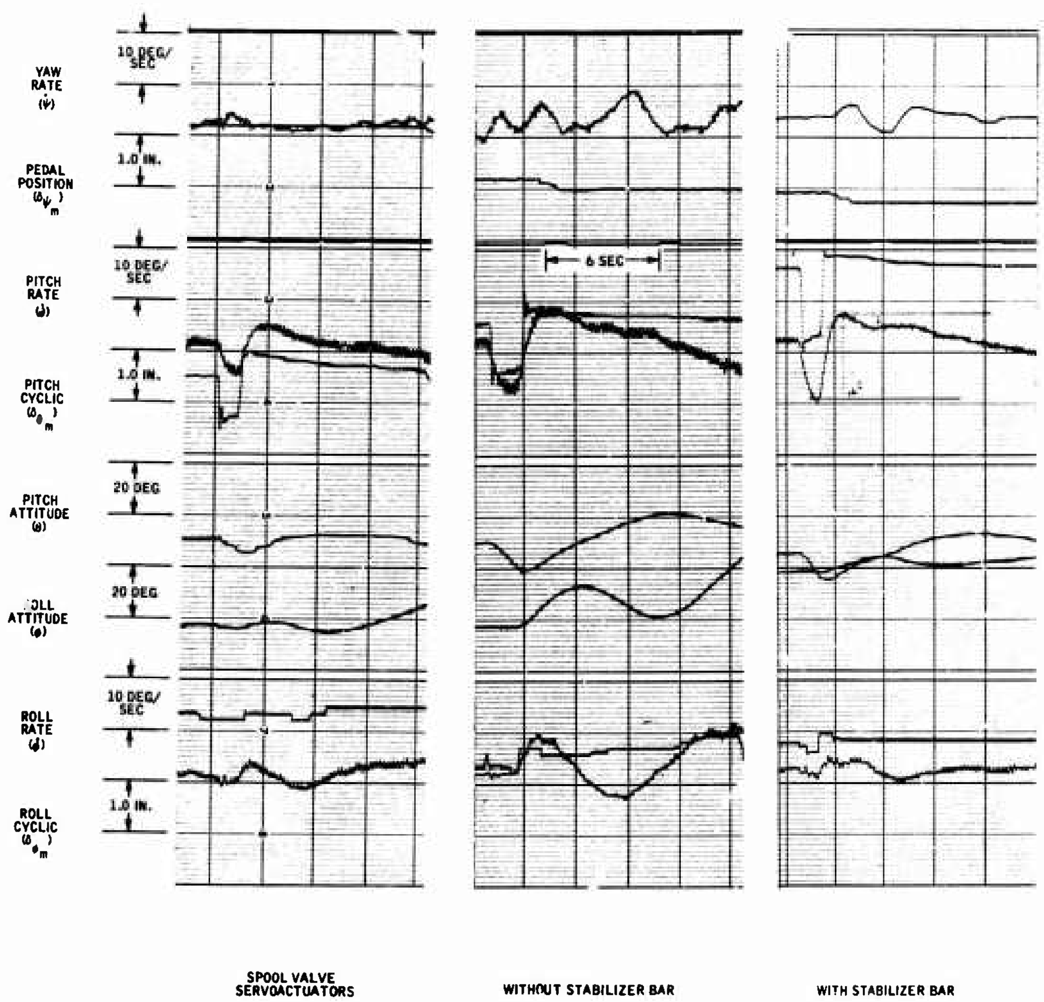


Figure 92. Pitch Pulse - Up (90 kn, 3000 ft).

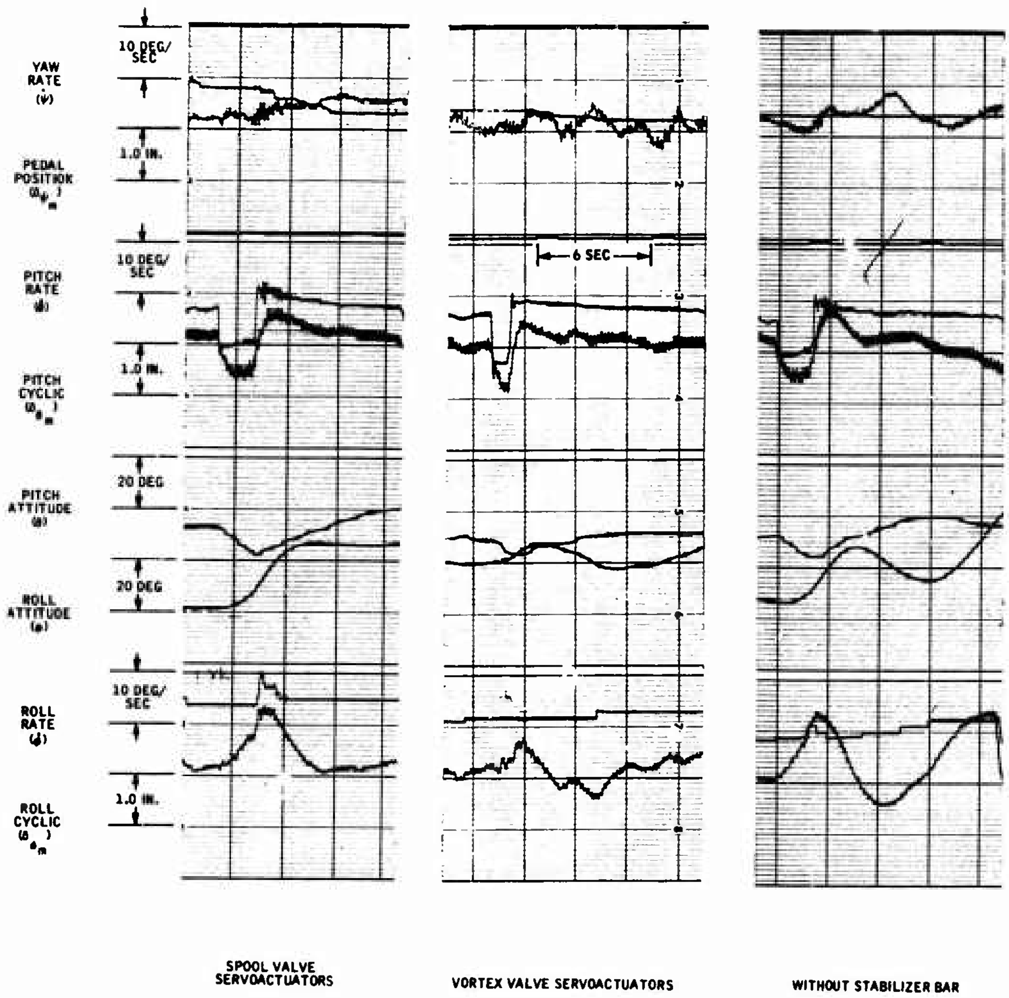


Figure 93. Pitch Pulse - Up (120 kn, 3000 ft).

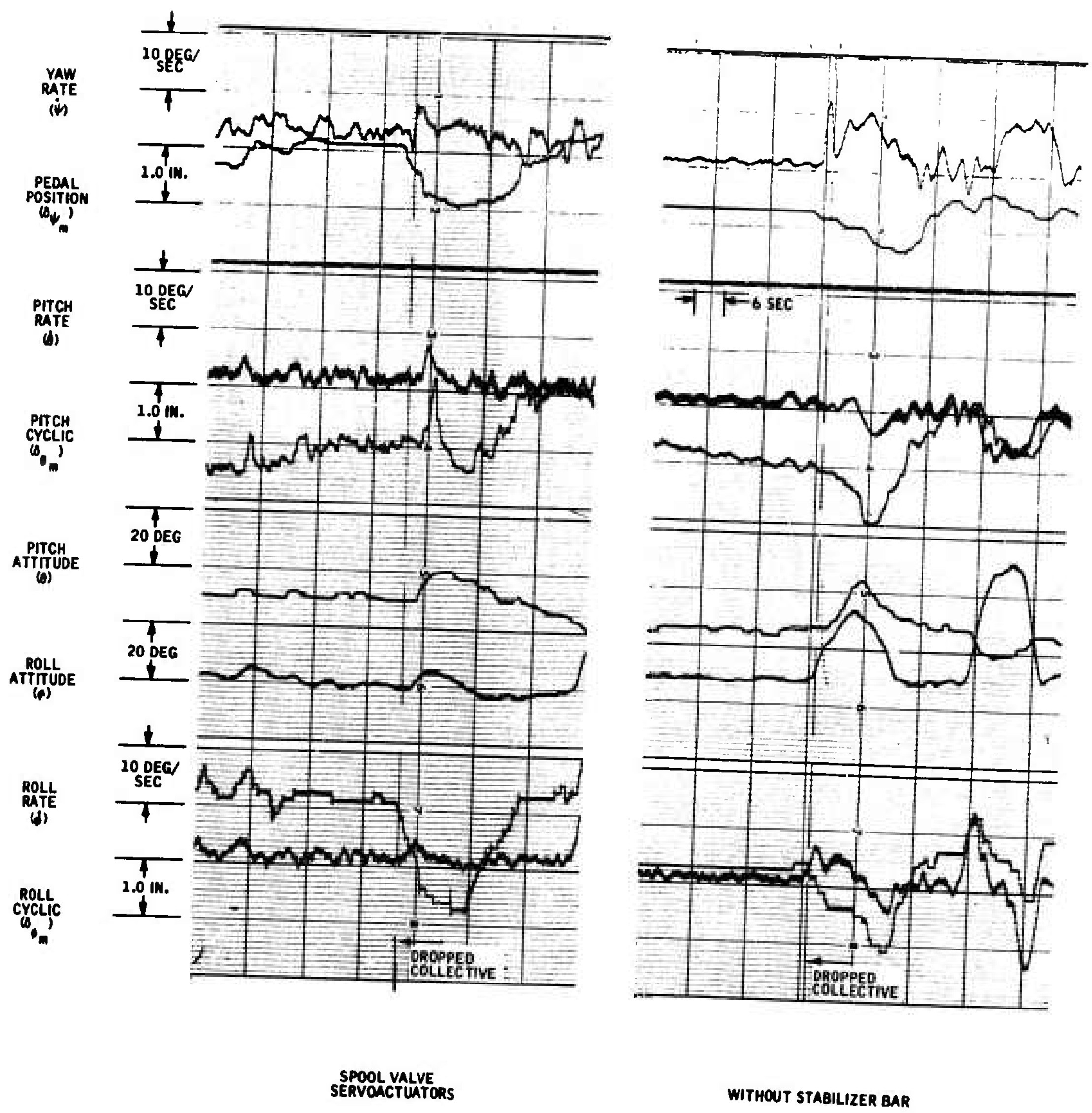


Figure 94. Autorotation - 60 kn, 3000 ft.

APPENDIX III
MECHANICAL PROBLEM AREAS ENCOUNTERED DURING INSTALLATION, GROUND TEST, AND FLIGHT TEST OF THE THREE-AXIS HYDROFLUIDIC SAS

ROLL SERVOACTUATOR TRAVEL INTERFERENCE

During preliminary mockup of the servoactuator installation, it became apparent that full forward travel of the roll servoactuator and supply hoses would be restricted by the position of the link rod which connects the pilot's and copilot's throttle control.

Several solutions to the problem were advanced and rejected, and finally the control link was simply removed. A placard was installed to note the inoperative throttle.

Two relay boxes, a wire terminal junction board, and considerable wiring also restricted full forward travel of the roll servoactuator and its supply hoses. The relays were moved to the left, and the terminal strip was moved to the right of its original location.

The cutout area in the bulkhead at station 66.00 was enlarged at the lower right-hand corner to allow clearance for the roll servoactuator supply hoses.

One solution to the three problem areas mentioned above would be to move the roll servoactuator further aft and mount the support bracket on the longitudinal bulkhead, with the support bracket arms extended laterally.

REMOVAL OF HEATING SYSTEM DUCT CONTROL BOX AND FLEXIBLE DUCT TUBING

The design layout made it necessary to modify bulkheads at stations 66.00 and 78.00 to provide a structural location for the support brackets and idler arms used with the roll and pitch servoactuators.

The lightening holes formerly used to route heater system ducting were covered with doublers and stringers. The main duct and control box were removed to allow space for the support brackets, idler arms, and flexible hydraulic supply hoses.

It might be possible to shorten the idler arm used to support the pitch servoactuator and move its support bracket inboard toward the center of the fuselage. This change would allow the duct control box to remain in its present position, with some compromise concerning the flexible tube routing.

CONTROL FORCES APPLIED TO SERVOACTUATOR LINKAGE BY MULTIPLE HOSE LOOPS

The force generated by five hose loops extending aft first became apparent in the roll axis. The combined force pushed the roll servoactuator linkage forward and displaced the cyclic control stick to the left approximately 1 in.

A similar problem existed in the yaw axis, as all flexible supply hoses extended aft. A tendency to drive right pedal forward existed.

No problems were encountered in the pitch axis, as the hose loop forces were balanced.

When loop-caused forces were measured with the aircraft's hydraulic boost on and FSAS off, a 0.5-lb force applied to the servoactuator linkage displaced the cyclic stick in roll approximately 0.5 in. Approximately 1 lb of force was applied by the combined five hose loops.

The 3000-psi hose used is excellent, and it is made up of alternating layers of neoprene, fabric, and woven stainless-steel wire. However, the hose is not as light or flexible as might be desired for this application.

Loop-caused forces were measured with and without system pressure. No difference could be detected.

The hose problem in the roll axis was corrected by locating two hose loops forward beneath the roll servoactuator. The wiring which was routed under this area was covered with a linen bakelite plate. This smooth surface allowed the hose to slide relatively free without becoming entangled.

USE OF NEOPRENE WASHERS ON CLEVIS ENDS OF YAW SERVOACTUATOR LINKAGE

The yaw-axis bellcrank bearing was not modified. The spherical bearings allowed the servoactuator and rod linkage to sway enough to cause hangup on the edges of the enlarged cutout in the bulkhead at station 52.00.

This problem was corrected by installing 1/8-in.-thick x 1-1/8-in.-dia. neoprene washers on either side of the bearings at the forward and aft clevis fittings. The addition of the washers restrained torsional movement enough so the danger of hangup ceased.

SHIMMED COUPLING USED TO ATTACH YAW SERVO-ACTUATOR TO CONTROL TUBE LINKAGE

A complicated, possible source of failure -- shearing of a very small screw -- could allow the yaw servoactuator to rotate or sway enough to hang up on structure.

MODIFICATION OF FLOOR AREA ABOVE SERVOACTUATOR POSITIONS

In a future installation, a great deal of time would be saved by installing 10-32 plate nuts at all former rivet holes, and leave the structure so modified.

MODULE AND ASSOCIATED EQUIPMENT LOCATIONS

A future installation could be simplified if the modules, regulators, solenoids, and other equipment were mounted further aft. The supply, return, and control lines could be routed beneath the floor section.

This change would allow simple access to the servoactuators without the need to dismantle and disconnect multiple hydraulic lines and other equipment.

USE OF STEEL FLARED FITTINGS AND STAINLESS-STEEL LINES

This proved to be an excellent choice of material, particularly in a test program where fittings were removed and installed many times. No failures or extreme leakage was experienced throughout the program.

APPENDIX IV
CONTROL TUBES TEST REPORT*

1.0 ABSTRACT

1.1 Object

To apply a 50-percent overload in compression and tension on five linkage assemblies of the YG1053A01 Three-Axis Hydrofluidic Stability Augmentation System to demonstrate that the tubes have sufficient strength.

1.2 Conclusion

No buckling, breakage or permanent deformation was obtained by overloading the linkages. Their strengths are adequate for use in the system.

2.0 UNITS TESTED

The five linkages tested for the YG1053A01 Three-Axis Hydrofluidic Stability Augmentation System are:

- One 299-008-021-1 Actuator Assembly, 66.20 in. pin-to-pin
- One 299-008-017-33 Actuator Assembly, 24.37 in. pin-to-pin
- One 299-008-017-45 Actuator Assembly, 34.13 in. pin-to-pin
- One 299-008-017-41 Tube Assembly, 15.30 in. pin-to-pin
- One 299-008-017-53 Tube Assembly, 21.87 in. pin-to-pin.

3.0 REFERENCE DOCUMENTS

1) Honeywell Memo, B. Johnson from W. M. Posingies, "Testing of Primary Linkage Control Tubes," dated 9 July 1970.
2) Summary of Telecon between W. M. Posingies of Honeywell and Leo Norman and Frank Depaw of Bell, May 26, 27 and 28, 1970, regarding stabilizer bar disenabling blocks, deviations to materials indicated in drawings, and deviations to tapers on primary linkage tubes.
3) Bell Helicopter Drawings 299-008-017 and 299-008-021.

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*Honeywell Aero Engineering Test Report No. QFA-06190 dated July 1970.

4.0 PROCEDURES AND RESULTS

A loading fixture which could accept all of the linkages was made up as shown in Figure 95. The ram piston areas were used to make a conversion table of hydraulic pressure versus pound force load. A 10,000-psi hydraulic hand pump and pressure gauge assembly was obtained for powering the ram.

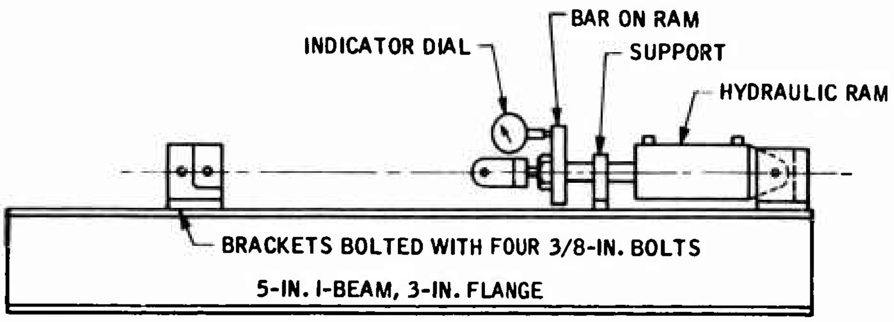


Figure 95. Linkage Load Fixture.

Each linkage assembly was inserted in the loading fixture, and the following procedure was used for loading in each direction:

1) Half of maximum load was applied and removed at least three times to take out slack. The last load was reduced to 100 lb in preparation for the next step.
2) The dial indicator was zeroed at the 100-lb load.
3) The maximum load, listed in Table VIII, was applied and then dial indicator was read to give a measure of the change in length.
4) The load was reduced to 100 lb and the indicator checked for zero.
5) Steps 3 and 4 were repeated three times.
6) The maximum load was applied, and the linkage was deflected by hand to observe for any weakness, tendency to buckle or deform, or inability to spring back.

Vortex Valve servoactuator 24SA31AA0002 was used for all three actuator linkages. The tubes were changed to obtain the different assemblies. The 299-008-017-65 clevis was left on the servoactuator for all three tests, so the 299-008-021 linkage was 65.5 in. long instead of the specified length of 66.20 in.

The values obtained for changes in length with loads are given in Table VIII. The readings indicate that no permanent set occurred for any of the linkages and that the elastic limit was never reached.

TABLE VIII. DEFORMATION OF LINKAGE ASSEMBLIES WITH LOAD

Assembly No.	Load (lb)	Dial Indicator Reading (Hundredths of an In.) Compression	Tension
299-008-021-1*	100	0	0
	1155	25.0	12.0
299-008-017-33*	100	0	0
	1000	8.0	8.5
299-008-017-45*	100	0	0
	750	10.5	11.0
299-008-017-41	100	0	0
	1000	11.0	12.0
299-008-017-53	100	0	0
	750	13.0	14.0

*Some deflection may occur in actuator center lock mechanism when under high load.

The tube assemblies were very rigid. A side load on the actuator assemblies produced some deflection - up to ±1/16 in. at the large flange with a relatively light hand load on the 299-008-021-1 assembly - but spring-back was very positive. Some of this deflection could be observed between the actuator and the piston rod. There appeared to be no difference in deflection action between a compressive and a tensile load. The actuator was not hydraulically powered.

5.0 INSTRUMENTATION

The following instruments were used for this test:

- Dial indicator, 0.001-in. divisions, 406-010, with magnetic holder
- Hydraulic pump assembly, including:
  - a) Heise gauge, 10,000 psi, 961-005, in panel box
  - b) Blackhawk hand pump, 10,000 psi
- Rulers and scales

6.0 RECOMMENDATIONS

None. Data submittal only.

APPENDIX V
CALIBRATION CURVES OF VARIOUS TRANSDUCERS

All sensors used during the flight test were calibrated by the contractor's Development and Evaluation Laboratory. This appendix presents the data that were obtained.

Test data for the three rate gyros used to measure aircraft turning rates are presented in Figures 96, 97, and 98.

Figures 99 and 100 are calibration curves for the vertical gyro. Figure 101 is the curve for the heading gyro, and Figure 102 shows the performance of the three accelerometers.

The performance of the four potentiometers used to measure pilot cyclic, collective, and rudder controls is shown in Figures 103 through 106, respectively.

RATE OUTPUTS:

INPUT (DEG/SEC CW)	OUTPUT (%)	INPUT (DEG/SEC CCW)	OUTPUT (%)	INPUT (DEG/SEC CCW)	OUTPUT (%)	INPUT (DEG/SEC CW)	OUTPUT (%)
20	00.00	20	100.00	20	100.00	20	00.00
15	10.88	15	88.77	15	89.35	15	11.43
10	23.96	10	75.80	10	76.38	10	24.58
5	36.82	5	62.70	5	63.35	5	37.40
0	49.80			0	50.18		

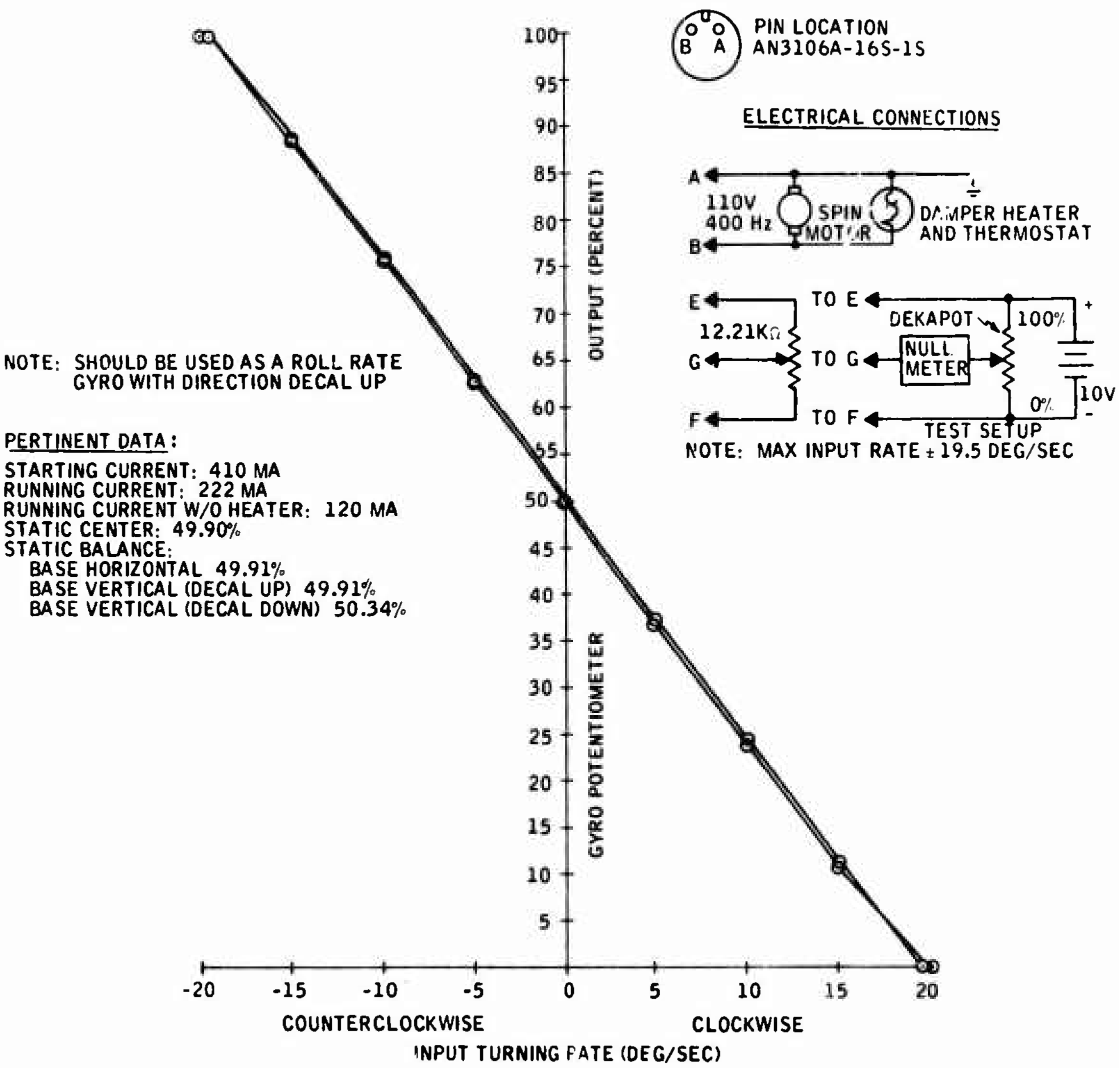
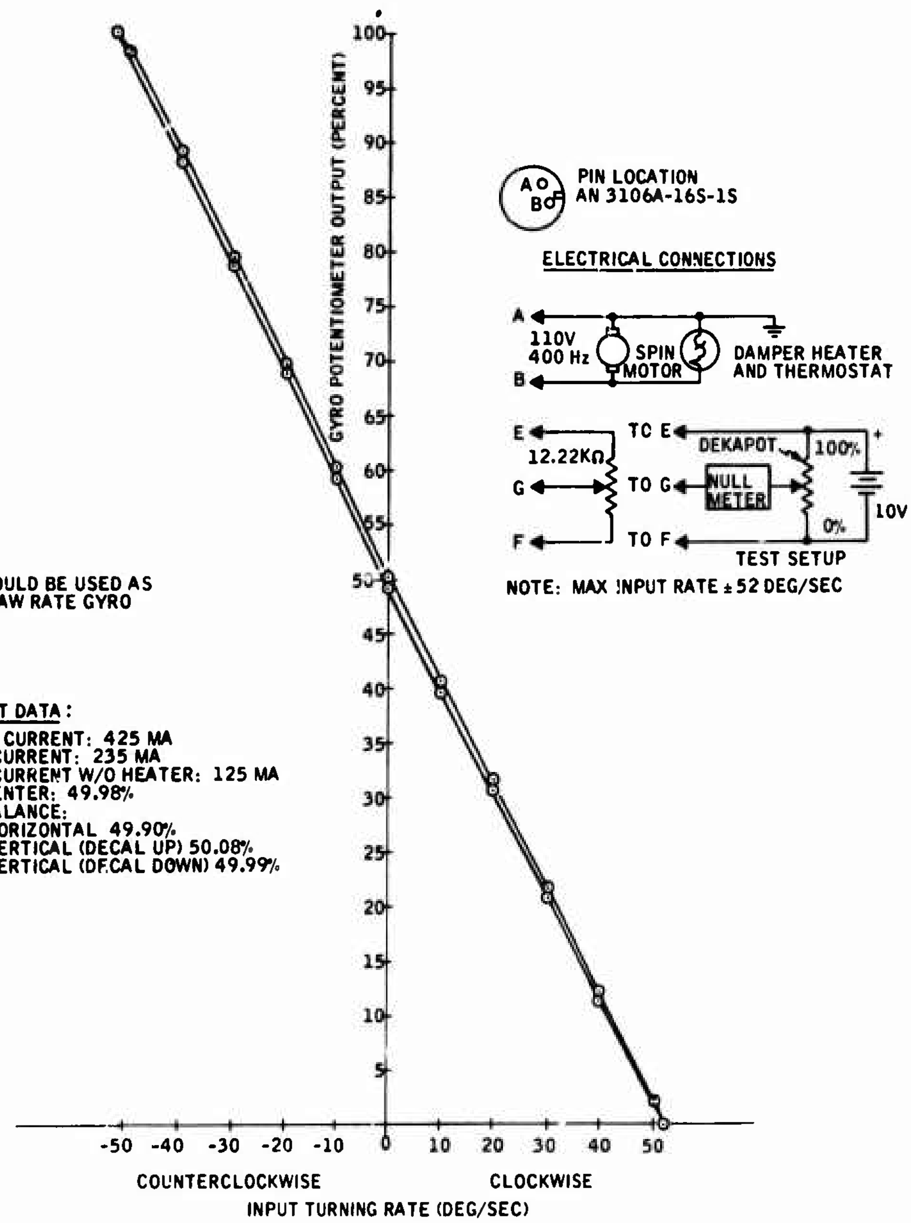


Figure 96. Gyro Output Versus Input Turning Rate for GG79-Type Rate Gyro Used as a Roll Rate Gyro.

RATE OUTPUTS:

INPUT (DEG/SEC)	OUTPUT (%)	INPUT (DEG/SEC)	OUTPUT (%)	INPUT (DEG/SEC)	OUTPUT (%)	INPUT (DEG/SEC)	OUTPUT (%)
50	01.63	50	98.01	50	98.85	50	02.49
40	11.27	40	88.36	40	89.22	40	12.13
30	20.86	30	78.89	30	79.53	30	21.76
20	30.53	20	69.07	20	69.79	20	31.58
10	39.68	10	53.22	10	60.26	10	40.54
0	49.46			0	50.30		



NOTE: MAX INPUT RATE ±52 DEG/SEC

NOTE: SHOULD BE USED AS A YAW RATE GYRO

PERTINENT DATA:

STARTING CURRENT: 425 MA
RUNNING CURRENT: 235 MA
RUNNING CURRENT W/O HEATER: 125 MA
STATIC CENTER: 49.98%
STATIC BALANCE:
BASE HORIZONTAL 49.90%
BASE VERTICAL (DECAL UP) 50.08%
BASE VERTICAL (DECAL DOWN) 49.99%

Figure 97. Gyro Output Versus Input Turning Rate for GG79-Type Rate Gyro Used as a Yaw Rate Gyro

RATE OUTPUTS:

INPUT (DEG/SEC CW)	OUTPUT (%)	INPUT (DEG/SEC CCW)	OUTPUT (%)	INPUT (DEG/SEC CCW)	OUTPUT (%)	INPUT (DEG/SEC CW)	OUTPUT (%)
15	01.00	15	97.76	15	98.17	15	01.79
10	17.33	10	82.03	10	82.48	10	17.73
5	33.23	5	65.79	5	66.41	5	34.15
2	43.14	2	56.35	2	56.69	2	43.87
0	49.78			0	50.20		

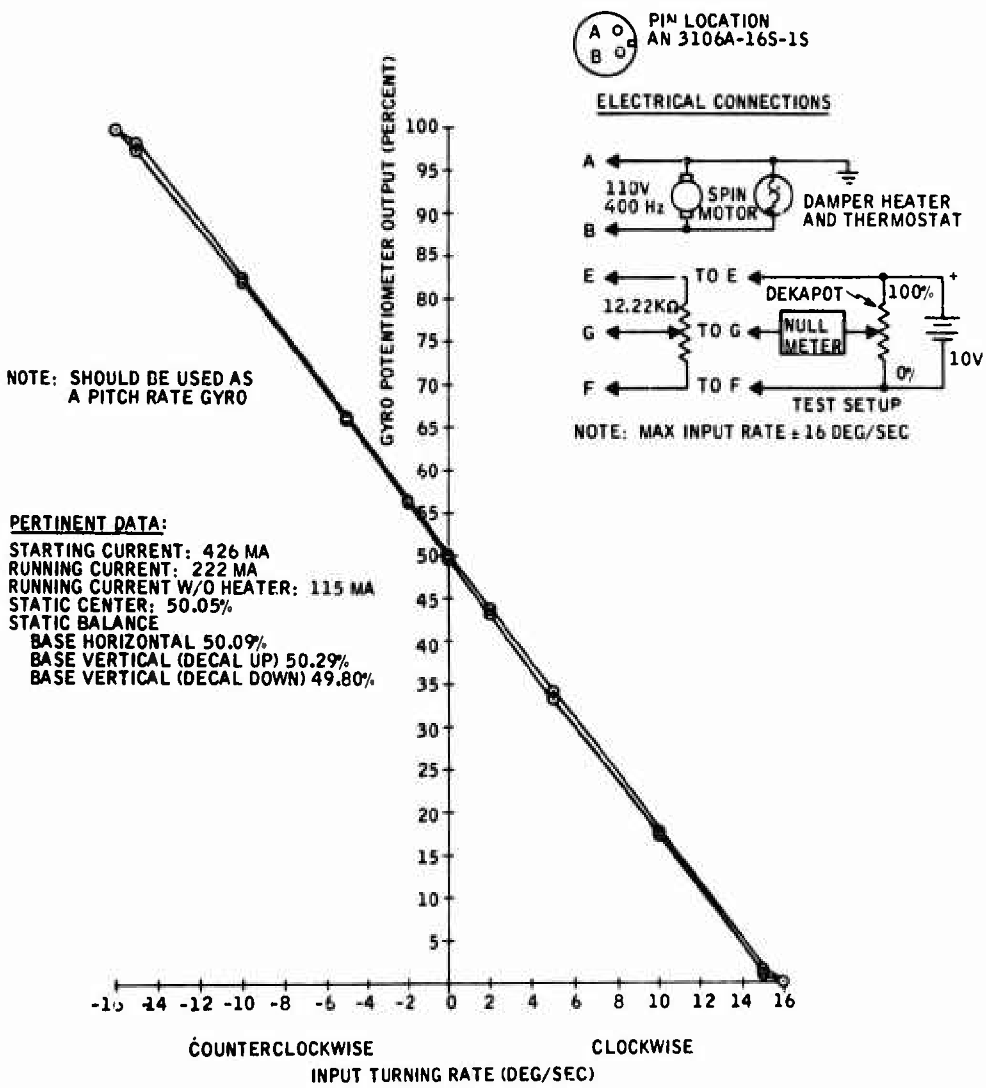


Figure 98. Gyro Output Versus Input Turning Rate for GG79-Type Rate Gyro Used as a Pitch Rate Gyro.

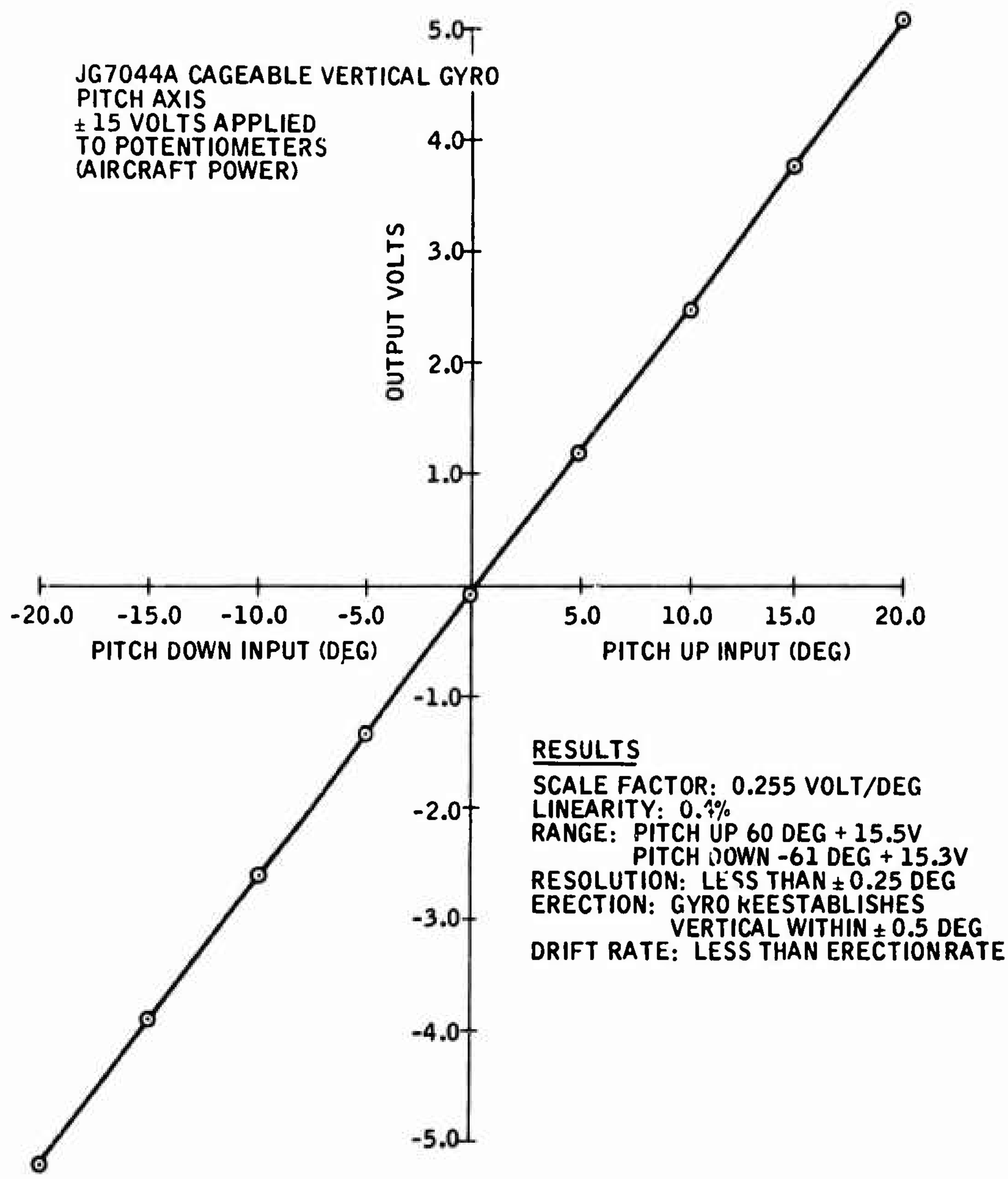


Figure 99. Pitch Output Voltage Versus Pitch Input Angle for JG7044A Cageable Vertical Gyro Used on Three-Axis FSAS Program.

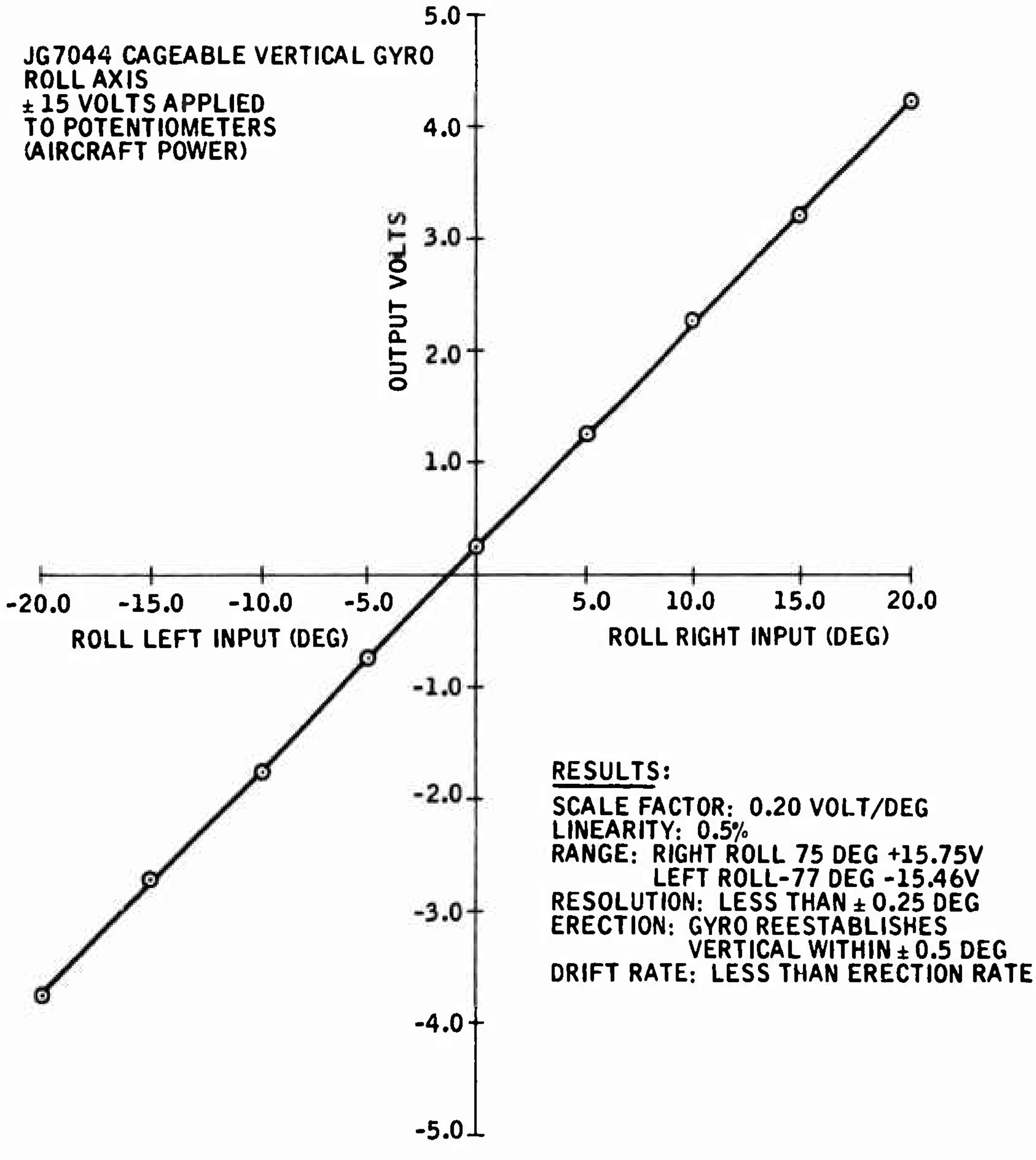


Figure 100. Roll Output Voltage Versus Roll Input Angle for JG7044A Cageable Vertical Gyro Used on Three-Axis FSAS Program.

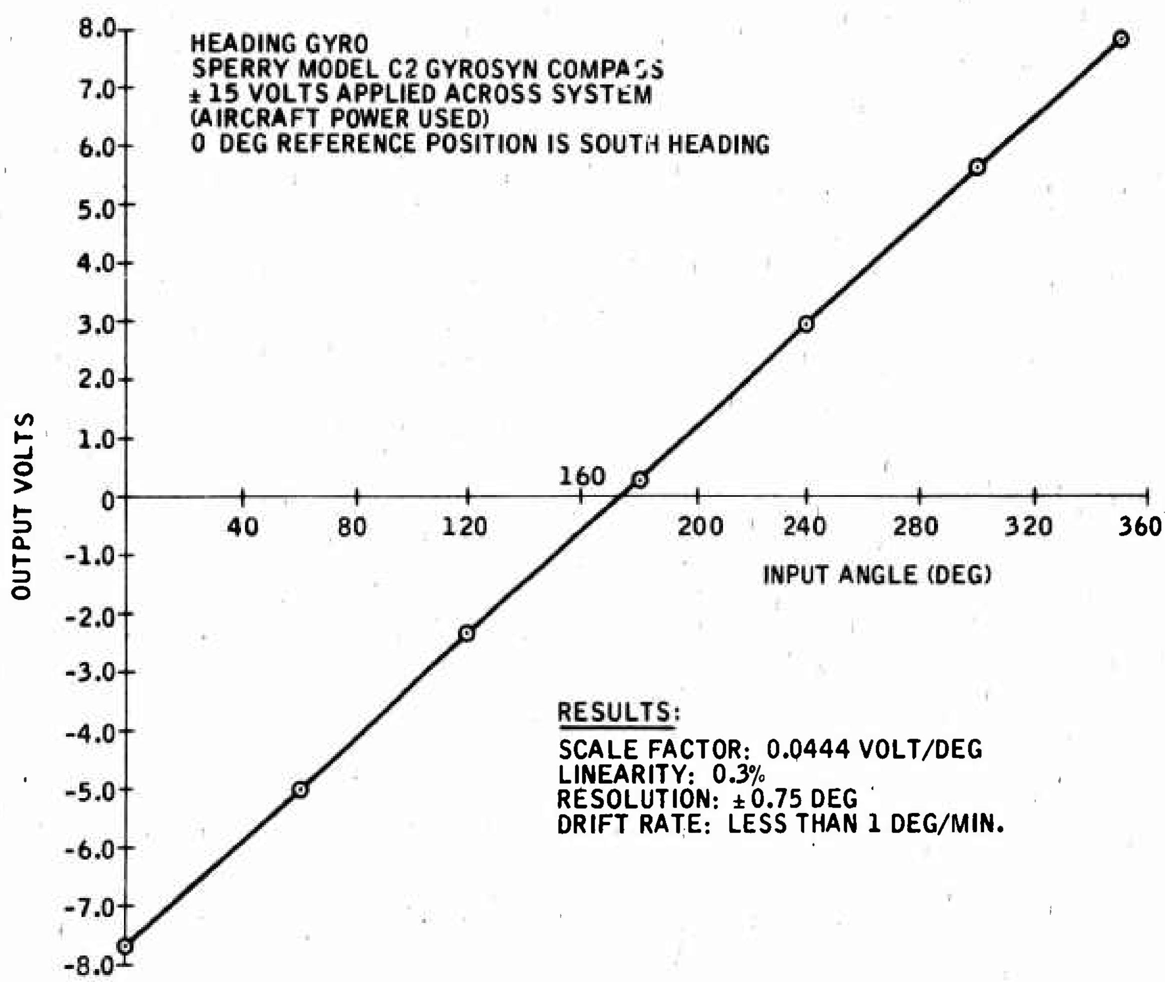


Figure 101. Yaw Output Voltage Versus Input Angle for Heading Gyro Used on Three-Axis FSAS Program.

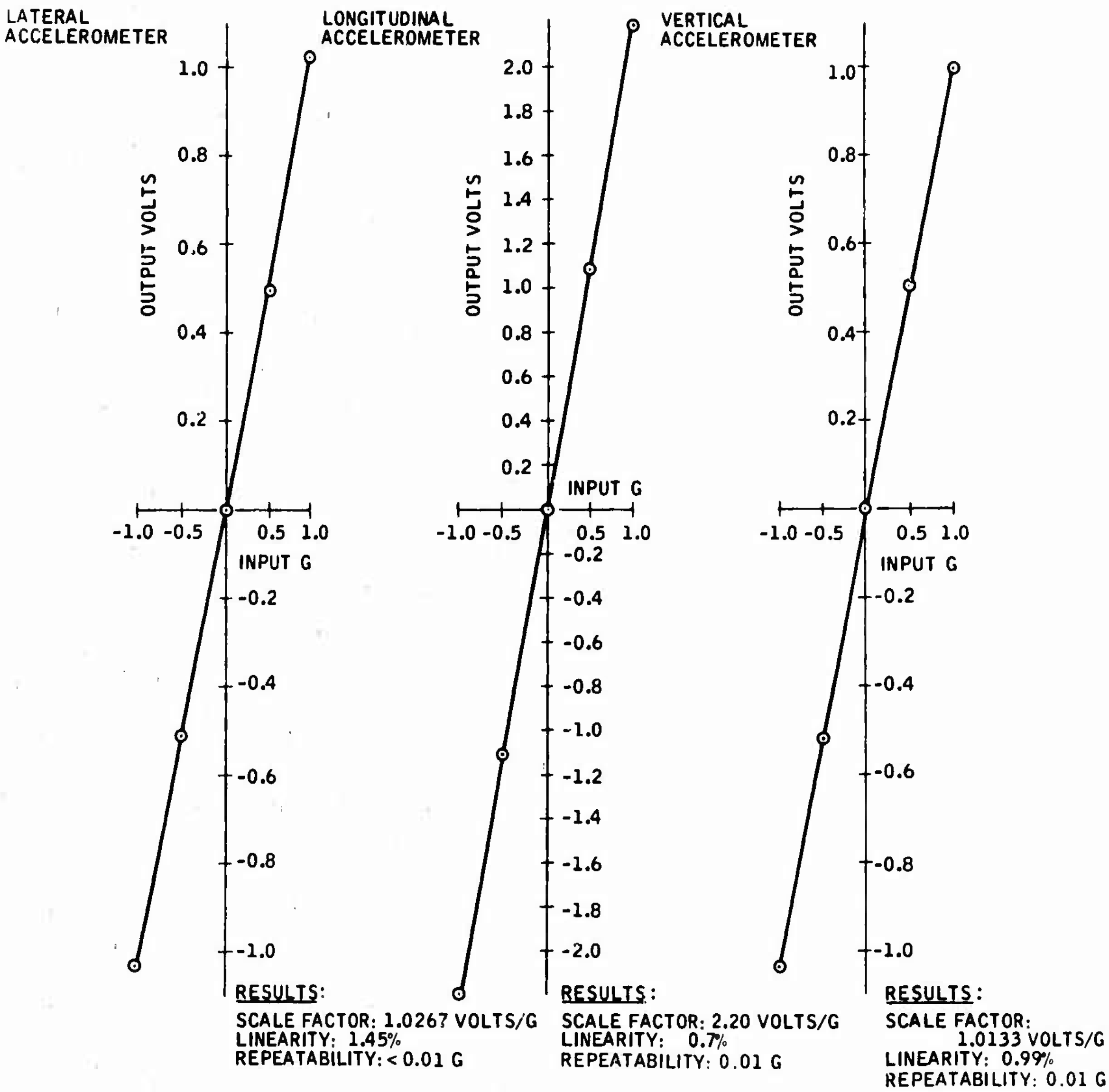


Figure 102. Output Voltage Versus Input G-Level for Downer Model 4310 Linear Accelerometer Used on Three-Axis FSAS Program.

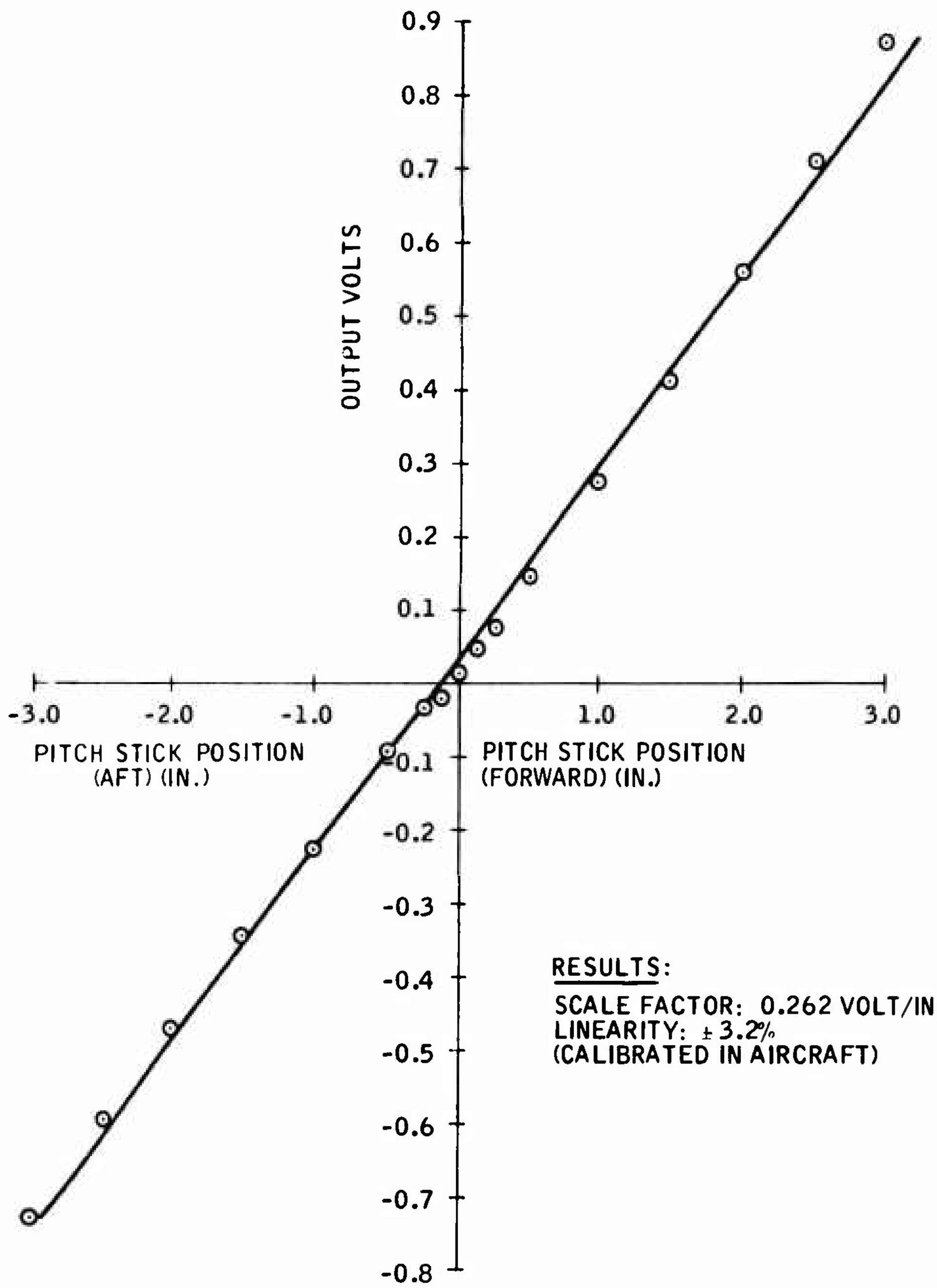


Figure 103. Cyclic Stick (Pitch) Position Calibration -- Output Voltage Versus Pitch Stick Position.

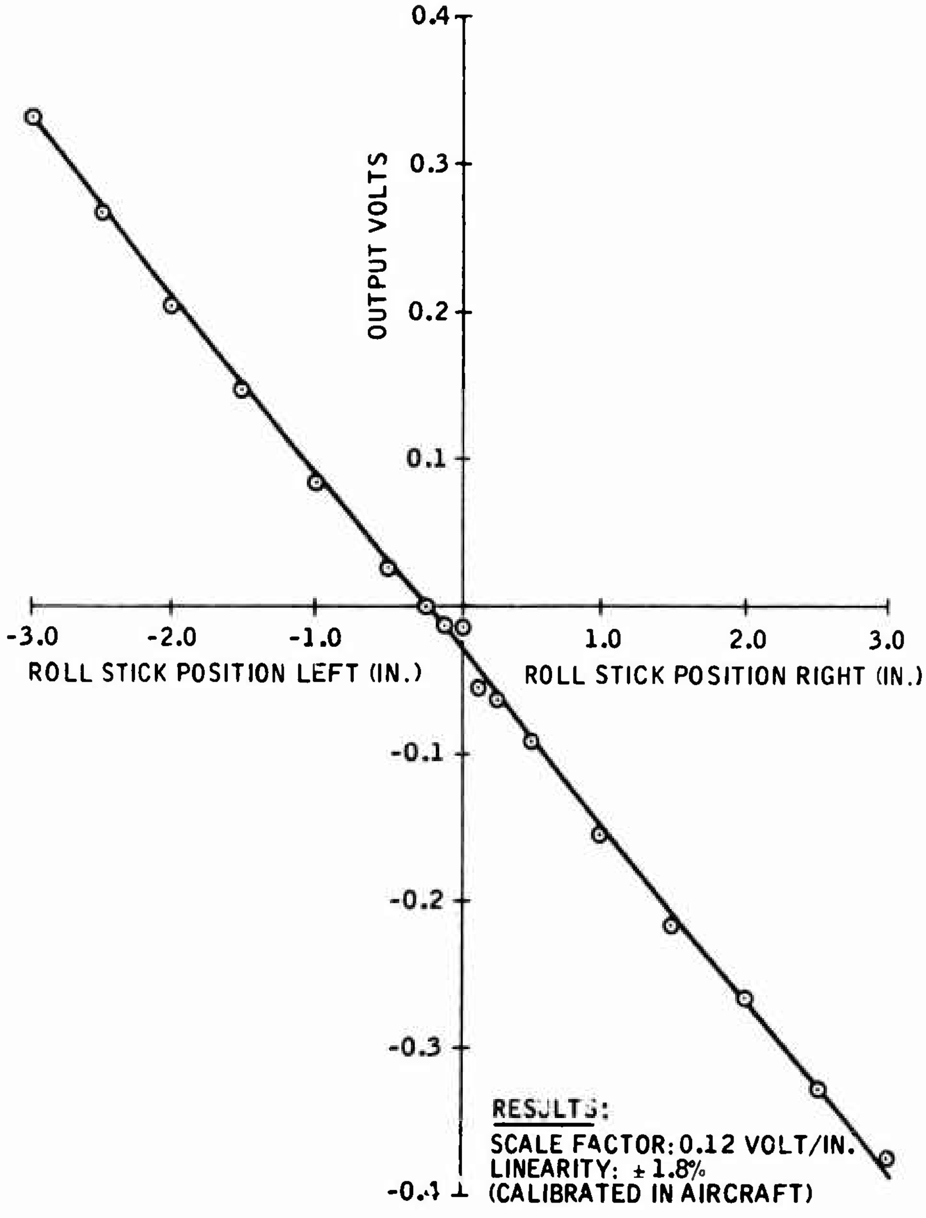


Figure 104. Cyclic Stick (Roll) Position Calibration -- Output Voltage Versus Roll Stick Position.

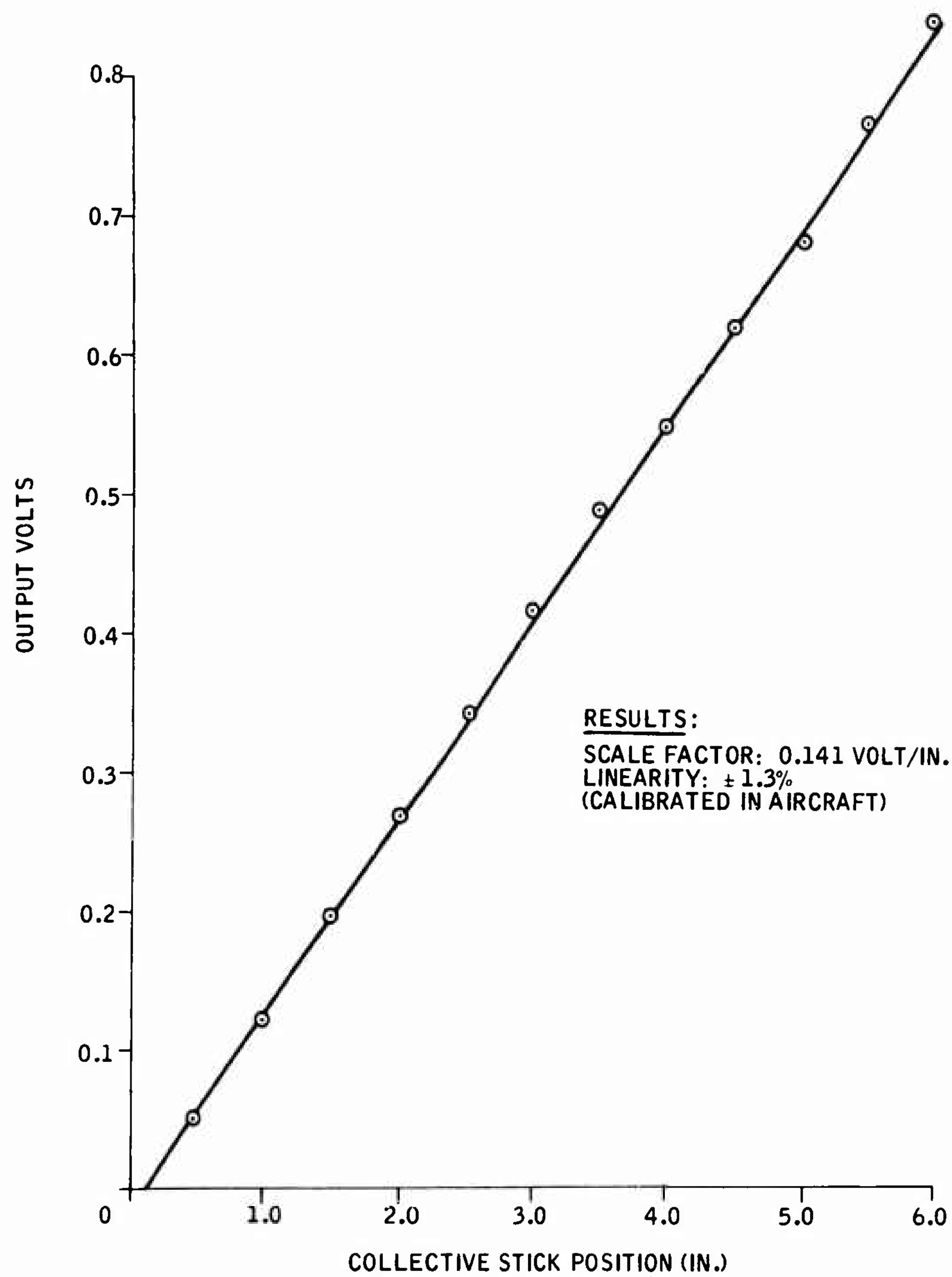


Figure 105. Collective Stick Position -- Output Voltage Versus Collective Stick Position.

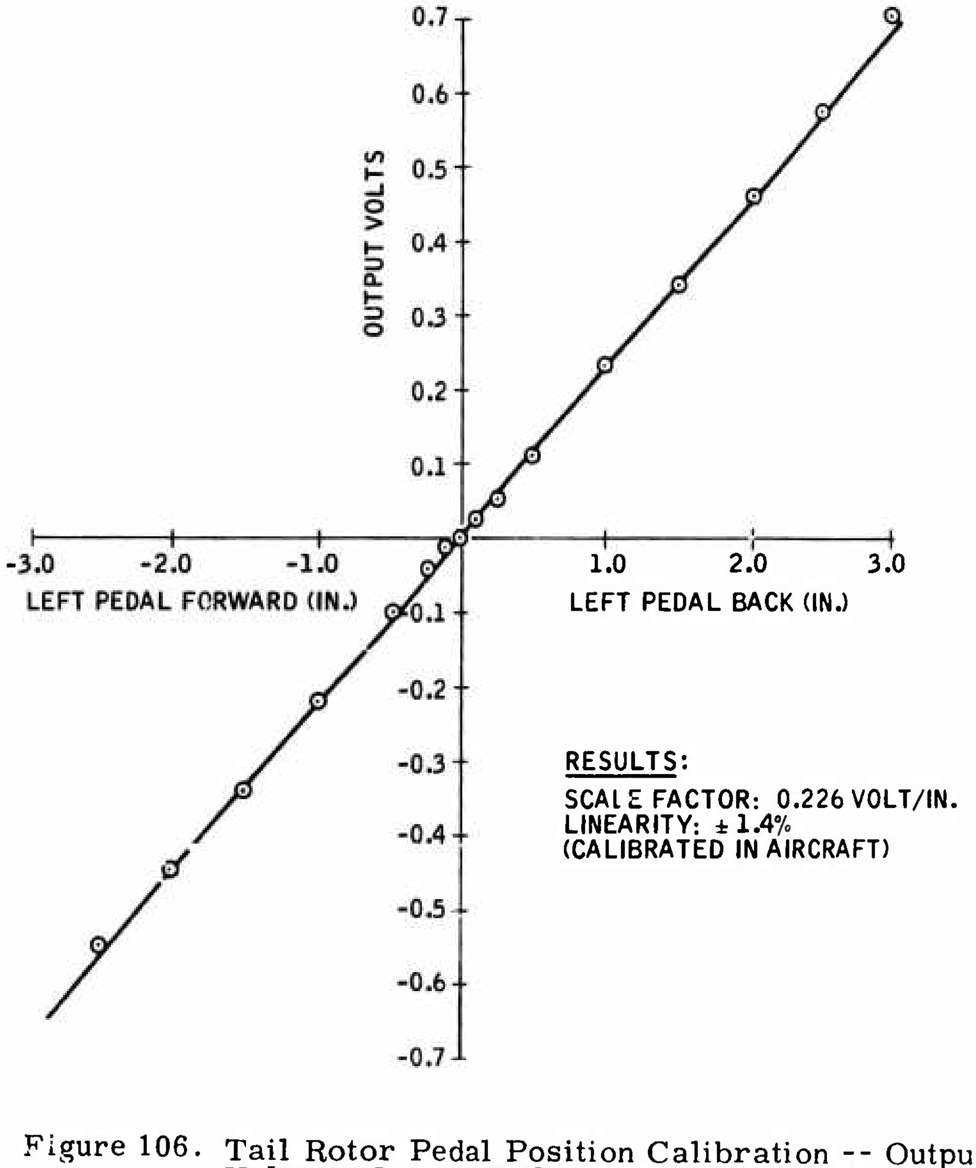


Figure 106. Tail Rotor Pedal Position Calibration -- Output Voltage Versus Tail Rotor Pedal Position.

APPENDIX VI
FLIGHT TEST PLAN FOR A THREE-AXIS HYDROFLUIDIC STABILITY AUGMENTATION SYSTEM*

1.0 SCOPE

This specification defines the scope of flight testing required to evaluate the three-axis hydrofluidic stability augmentation system (FSAS).

2.0 APPLICABLE DOCUMENTS

- Honeywell Flightworthy Test Report No. AEX-53737
- Honeywell Design Specification DS-21565-01

3.0 REQUIREMENTS

3.1 General

The objectives of the flight test evaluation of the FSAS-equipped UH-1C are:

- Quantitatively assess the FSAS's performance and capability of improving the UH-1C flying qualities
- Quantitatively compare flight test results with analytical predictions
- Recommend improvements to the FSAS
- Determine feasibility of the FSAS for possible replacement of the stabilizer bar in the UH-1 series helicopter control system and incorporation in and improvement of the current inventory aircraft.

The above objectives shall be accomplished within approximately 20 flights and shall not exceed a total of 13 hours of actual flight time.

*Honeywell document 21476-FR1 (Revised), dated July 1970.

3.2 Installation and Checkout

1) Interface and installation problems shall be recorded and sorted into categories depending on whether they are inherent problems or problems peculiar to the test hardware.

2) A ground check (helicopter not to leave ground) with rotor running shall be conducted at several power levels; the duration of the ground check is to be at the discretion of the project pilot. The project pilot shall engage and disengage the sytem, noting any problems. All safety disengage switches shall be tested for disengaging the FSAS. All instrumentation shall function properly. The FSAS null, both servoactuator output and hydrofluidic output, shall be evaluated during and after the ground check.

3.3 Flight Test Procedures

1) All flights shall be conducted at weights under 8000 lb and at mid-center-of-gravity loading conditions.

2) Continuous recordings shall be made of all takeoffs, landings, and letdowns, even though specific response tests are not made.

3) Prior to each flight, all instrumentation shall be adequately checked to assure proper operation.

4) The built-in test (BIT) button on each axis controller shall be pressed and released before and after each flight to determine satisfactory hydraulic system, controllers, and servoactuator operation. The BIT button shall be guarded so as to prevent inadvertent operation of this button by any means during flight which could jeopardize flight safety.

5) A preflight plan shall be prepared jointly by the project pilot, flight test engineer, and analyst prior to each flight. A separate plan will be required for each flight.

6) A pilot report shall be completed by the pilot after each flight.

7) Flight evaluation reports will be completed by the project analyst after each flight on which data are taken. This report will approve or disapprove recordings taken on the flight. These reports will be a basis for preparing future preflight plans.

3.4 Quantitative Flight Tests

1) A 5-minute FSAS-stabilized flight shall be conducted at (1) hover in ground effect, and (2) at 2500 ±1000 feet density altitude, 60 kn calibrated airspeed (KCAS). For these tests the project pilot shall minimize all induced inputs. These tests shall also be conducted for the basic aircraft (FSAS disengaged) for one flight.

2) The FSAS shall be quantitatively evaluated per the flight conditions listed in Table IX. Changes in the FSAS may be made between flights in an effort to optimize performance. The changes shall be based on the project pilot's qualitative evaluation and the flight recordings; this will require approximately nine flights.

3) The response of the FSAS during autorotational entries shall be measured and recorded. Entries shall be made from trim speeds of 60, 90, and velocity-never-to-exceed (V_{NE}) KCAS for one flight. Autorotational condition shall be maintained for at least 3 seconds unless terminated sooner at the discretion of the pilot.

4) All transient aircraft movements occurring in the helicopter from engaging and disengaging the FSAS, during level flight at speeds of 60, 90, and V_{NE} KCAS, as well as climbing turns to the left and to the right, shall be measured and recorded.

5) The project pilot shall provide a qualitative opinion report on each quantitative test flight. These reports will be combined by the project pilot into a summary report at the end of the flight test program.

6) The analyst will attempt to correlate project pilot qualitative comments with project pilot quantitative data, correlate Government pilot qualitative comments with Government pilot quantitative data, and compare project pilot and Government pilot qualitative comments.

3.5 Qualitative Flight Tests

The FSAS shall be qualitatively flight evaluated by two Government research test pilots designated by the Contracting Officer. The flight evaluation shall be conducted within the established flight envelope. The Contractor shall support and provide guidance as requested to the Government test pilots. The Government pilots shall fly the test helicopter from the right seat during the qualitative flight evaluation of the FSAS. If it is required for the contractor's project pilot to fly

TABLE IX. QUANTITATIVE EVALUATION OF FSAS FOR UH-1C HELICOPTER

Altitude	Hover	60 KCAS	90 KCAS	V_{NE} KCAS
Out of Ground Effect	$P_F R_F T_F Y_F$	-	-	-
	$P_A R_A T_A Y_A$	-	-	-
3000 ft ± 1000 ft	-	$P_F R_F T_F Y_F$	$P_F R_F T_F Y_F$	$P_F R_F T_F Y_F$
	-	$P_A R_A T_A Y_A$	$P_A R_A T_A Y_A$	$P_A R_A T_A Y_A$
10,000 ft	-	$P_F R_F T_F Y_F$	-	-
	-	$P_A R_A T_A Y_A$	-	-

Nomenclature:

P = Pitch pulse and step each way: one nose up, one nose down.

R = Roll pulse and step each way: one roll right, one roll left.

T = Turn each way: one turn right, one turn left, each turn at least 15 sec duration. At hover, turn should be made with pedals only; at 60 kn and above, turn should be made with lateral stick only.

Y = Pedal pulse and step each way: one yaw right, one yaw left.

Subscripts:

F = Three-axis hydrofluidic SAS

A = Free aircraft

Notes:

1. Pulses and steps shall be made at different size inputs from approximately 0.25 in. up to largest attainable at discretion of pilot. Pulse amplitude and duration must be consistent during tests at each flight condition.
2. Responses should be duplicated at least once at each flight condition to minimize need for repeating tests.

during the Government qualitative evaluation, he shall do so only as a safety/instructor pilot. Flight data resulting from these flights shall be recorded, reduced, and analyzed by the contractor as requested by the Government pilot or pilots and authorized by the Contracting Officer. The Government research test pilot's or pilots' evaluation will approximate but shall not exceed 3 hours of flight.

4.0 INSTRUMENTATION

Instrumentation for the following parameters shall be installed on the Government-furnished UH-1C test helicopter. All parameters shall be recorded on every flight on the 24-channel recorder:

1) Aircraft roll rate
2) Aircraft pitch rate
3) Aircraft yaw rate
4) Aircraft yaw attitude
5) Aircraft roll attitude
6) Aircraft pitch attitude
7) Pedal position
8) Collective position
9) Lateral cyclic position
10) Longitudinal cyclic position
11) Roll FSAS servoactuator output
12) Pitch FSAS servoactuator output
13) Yaw FSAS servoactuator output
14) Roll FSAS fluidic output
15) Pitch FSAS fluidic output
16) Yaw FSAS fluidic output
17) FSAS supply pressure
18) FSAS supply flow
19) Event marker
20) Time correlation
21) Linear accelerations -- longitudinal
22) Linear accelerations -- lateral
23) Linear accelerations -- vertical

The 8-channel recorder shall record parameters selected in the pre-flight plan to more clearly exemplify the quantitative portion of the flight test.

5.0 RECORDED DATA

Data as listed shall be recorded, reduced, and submitted in report form. Correlation between the following shall be stressed:

- Augmented versus free vehicle
- Actual flight data versus computer simulation data in USAAMRDL Technical Report 71-30, Appendix I
- Pilot qualitative evaluation versus quantitative data.