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JB-47-E FLY-BY-WIRE FLIGHT TEST PROGRAM (PHASE I)

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Hydraulic Research Manufacturing Co.

TECHNICAL REPORT AFFDL-TR-69-40

SEPTEMBER 1969

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76

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FOREWORD

The effort described in this document was performed by Hydraulic Research and Manufacturing Company under United States Air Force Contract No. AF33(615)-2526, Project 8225, "Fly-by-Wire Experimental Program". The work was administered under the direction of the Air Force Flight Dynamics Laboratory, FDCL, Research and Technology Division, Wright-Patterson Air Force Base, Ohio. Mr. Vernon R. Schmitt was the Air Force program monitor. Major J. P. Sutherland, R.C.A.C., was the Task Engineer. Flight Test Directors and Project Pilots were U.S.A.F. Major T. Scanlan and U.S.A.F. Major B. Fredricks.

The report covers work performed between August 1, 1966 and August 15, 1968, and was released by the author in October, 1968.

This technical report has been reviewed and is approved.

H.W. BASHAM
Chief, Control Elements Branch
Flight Control Division
AF Flight Dynamics Laboratory

ABSTRACT

This report is a description of the research investigation on a fly-by-wire control system applied to a B-47 test aircraft. The investigation is primarily confined to the pitch axis of the test aircraft and is divided into three phases.

Phase I system operation is based on using an electrical non-redundant primary flight control system with control inputs being generated by the normal control column motion.

Phase II system operation is based upon adding a side stick controller, pitch rate and nose acceleration feedback, and electrical roll axis control to the Phase I system.

Phase III incorporates a 4 channel redundant actuator with hydraulic logic into the Phase II system.

Flight test of the Phase I system has been completed with over 40 flight hours, including touch and go landings being accumulated on the fly-by-wire system by four different test pilots. Pilot comments on the system operation and performance were favorable. This report documents the complete Phase I system and test results and briefly describes the Phase II and III systems.

TABLE OF CONTENTS

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Continued)

B-47 TEST DESCRIPTION

The following section of this report is a documentation of the research investigations conducted on the Fly-By-Wire Control System for the pitch axis of a B-47 aircraft. This section is arranged in the following manner.

1. Introduction
2. Summary of Investigation
3. Recommendations
4. Mechanization Description
5. Test Procedure
6. Test Results and Analysis

INTRODUCTION

The Fly-By-Wire investigation undertaken for the pitch axis of the B-47 aircraft was the first phase of an extended Fly-By-Wire B-47 program. As such it is known as Phase 1. Hereafter, the investigation will be called "Phase 1 B-47 Fly-By-Wire" or "Phase 1."

The object of the Phase 1 program was to design, build, and evaluate in flight test a nonredundant, single-axis, electrical primary flight control system in a B-47 aircraft. These objectives have been accomplished.

SUMMARY OF INVESTIGATION

The first phase of the Fly-By-Wire Control System investigation as applied to the pitch axis of the B-47 aircraft has been completed and flight-tested in B-47 aircraft No. 0-53-2280. Fig. 1 is a photograph of the test aircraft taken by the chase aircraft during the second test flight.

The completed phase of the B-47 Fly-By-Wire system included the investigation, design, assembly, and lab test of an electrical, nonredundant primary flight control system for the elevator surface of the aircraft. Fig. 2 shows the system being evaluated in the B-47 tail section in the WPAFB Flight Dynamics Laboratory, prior to installation in the test aircraft and flight test evaluation.

The Phase 1 system operation was based on using an electrical position transducer attached to the control column. The position transducer created electrical signals which drive a servoactuator. This electrical control chain was installed in parallel with the mechanically controlled actuator. A switching block was constructed which incorporated redundant solenoid valves to transfer control from the mechanically operated actuator to the Fly-By-Wire actuator and back. A manually operated override spool was incorporated in the same block in order to provide emergency transfer from the Fly-By-Wire actuator to the mechanically operated actuator. When the Fly-By-Wire actuator was used, the mechanical actuator was bypassed so that its force output was zero, even though the control cables continued to operate its control valve. The electrically controlled servo actuator then positioned the elevator to provide the desired pitch control. Because of the nonredundancy of the electrical control chain, the mechanical system was used as a backup. Reversion from the electrical system to the mechanical was mechanized to occur by any of the following five different methods.

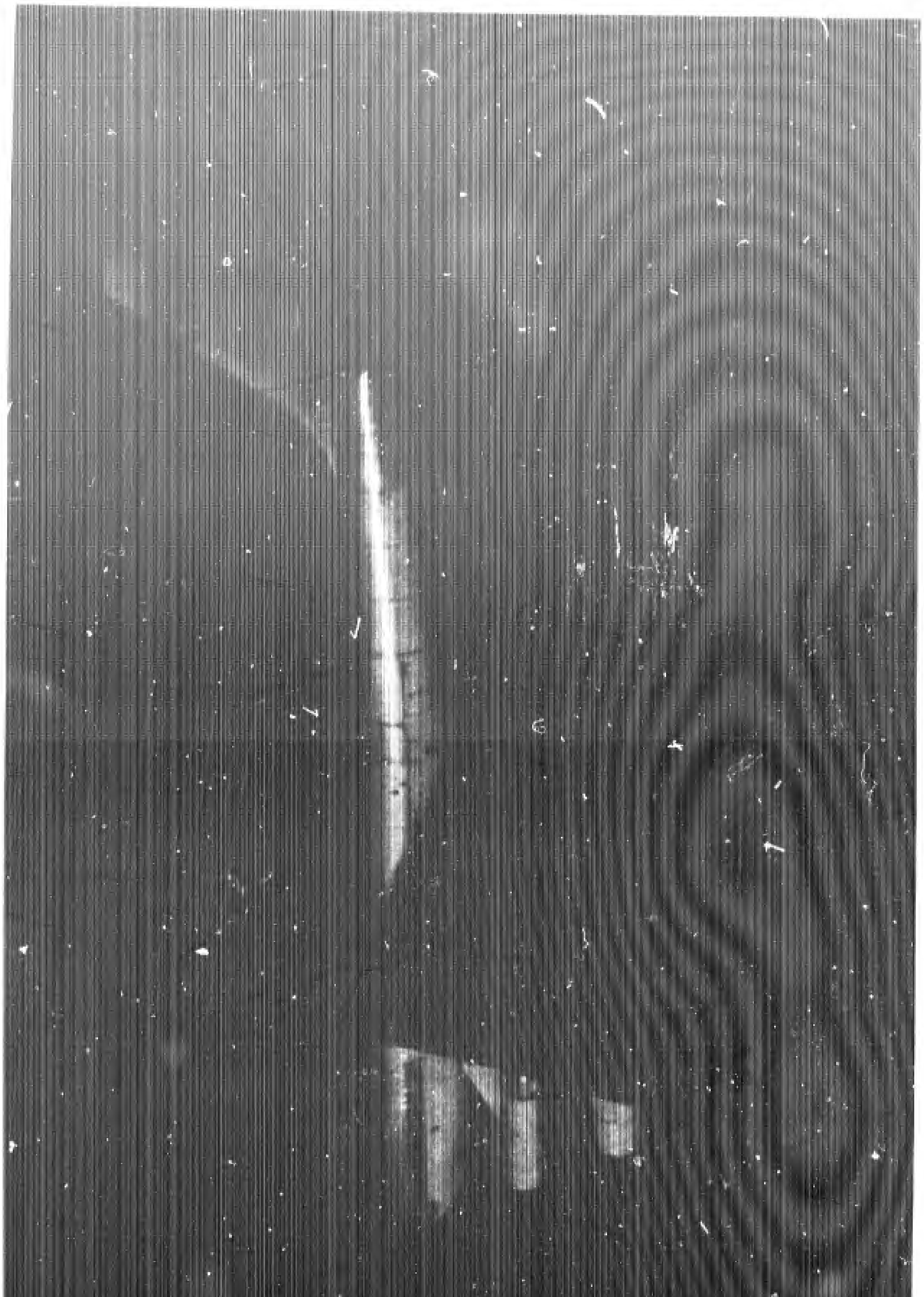


Figure 1. B-47 Test Aircraft

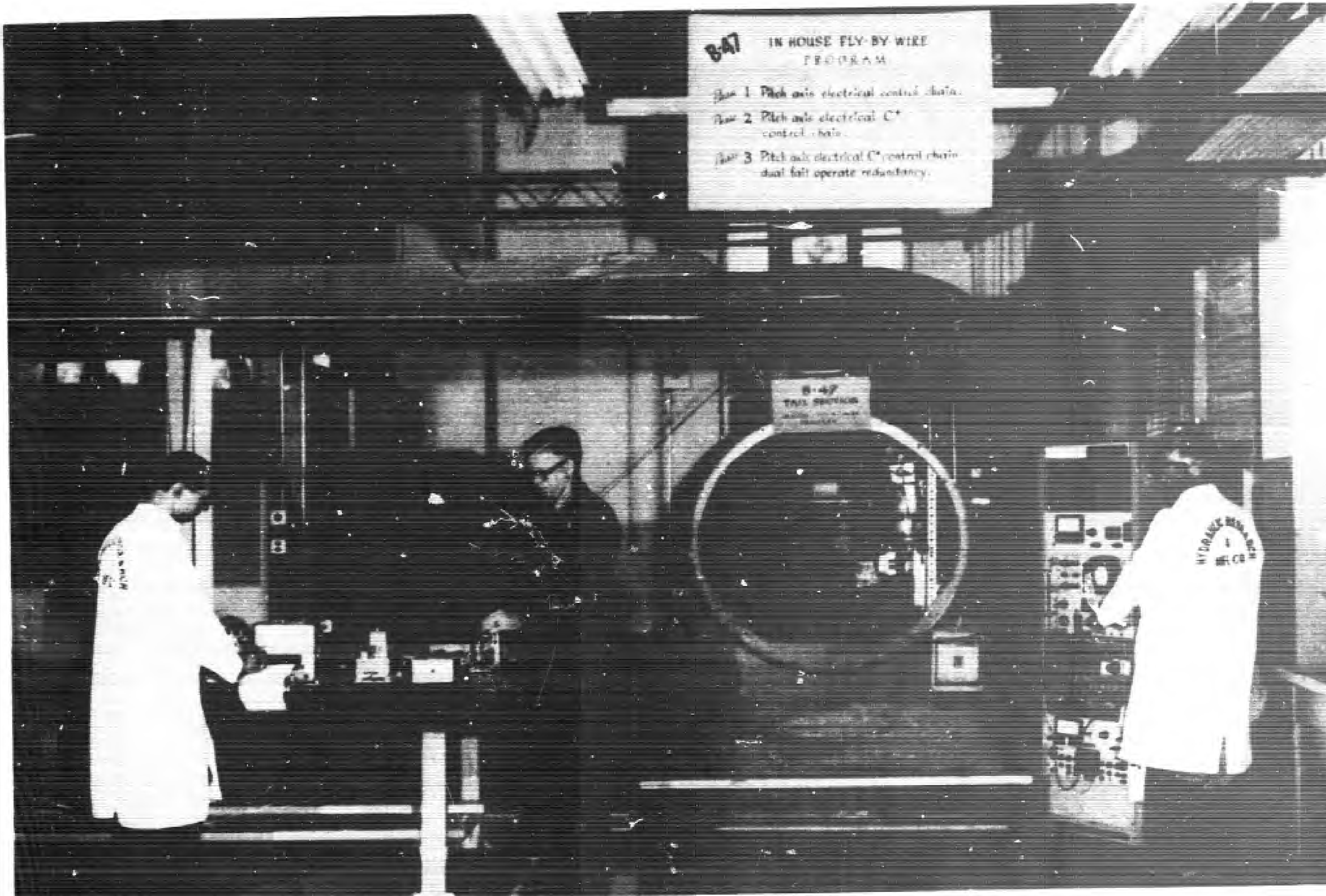


Figure 2. B-47 Tail Section Mockup

1. Automatically, with excess normal acceleration. A "g" limiter (modified F-100 MB3) caused the switching block to transfer control.
2. Manually, using the autopilot disconnect switch on the control column. Operation of the switch caused the switching block to transfer control.
3. Automatically, upon a particular level of rate disagreement between the elevator surface and control column. Microswitches mounted in the tail section detected this disagreement, causing the switching block to transfer control.
4. Manually, using the Fly-By-Wire electrical engage switch connected to the switching block.
5. Manually, using the cable-controlled reversion spool in the switching block.

Pilot notification of the system status was by pressure-switch-controlled indicating lights, green for the normal system, and amber for the Fly-By-Wire.

Fig. 3 shows the "g" limiter, servoamplifier, and junction box as mounted in the nose of the test aircraft. Fig. 4 shows the servoactuator, feedback position transducer, and switching block as mounted in the test aircraft.

The first flight test occurred on December 14, 1967. Over 45 flight hours, including touch-and-go landings, were accumulated on the Fly-By-Wire Phase 1 system by four different Air Force test pilots. Comments on the systems operations and performance were favorable. The principal improvement over the normal hydromechanical elevator control system was the aircraft response to control inputs in the high "Q" flight conditions.

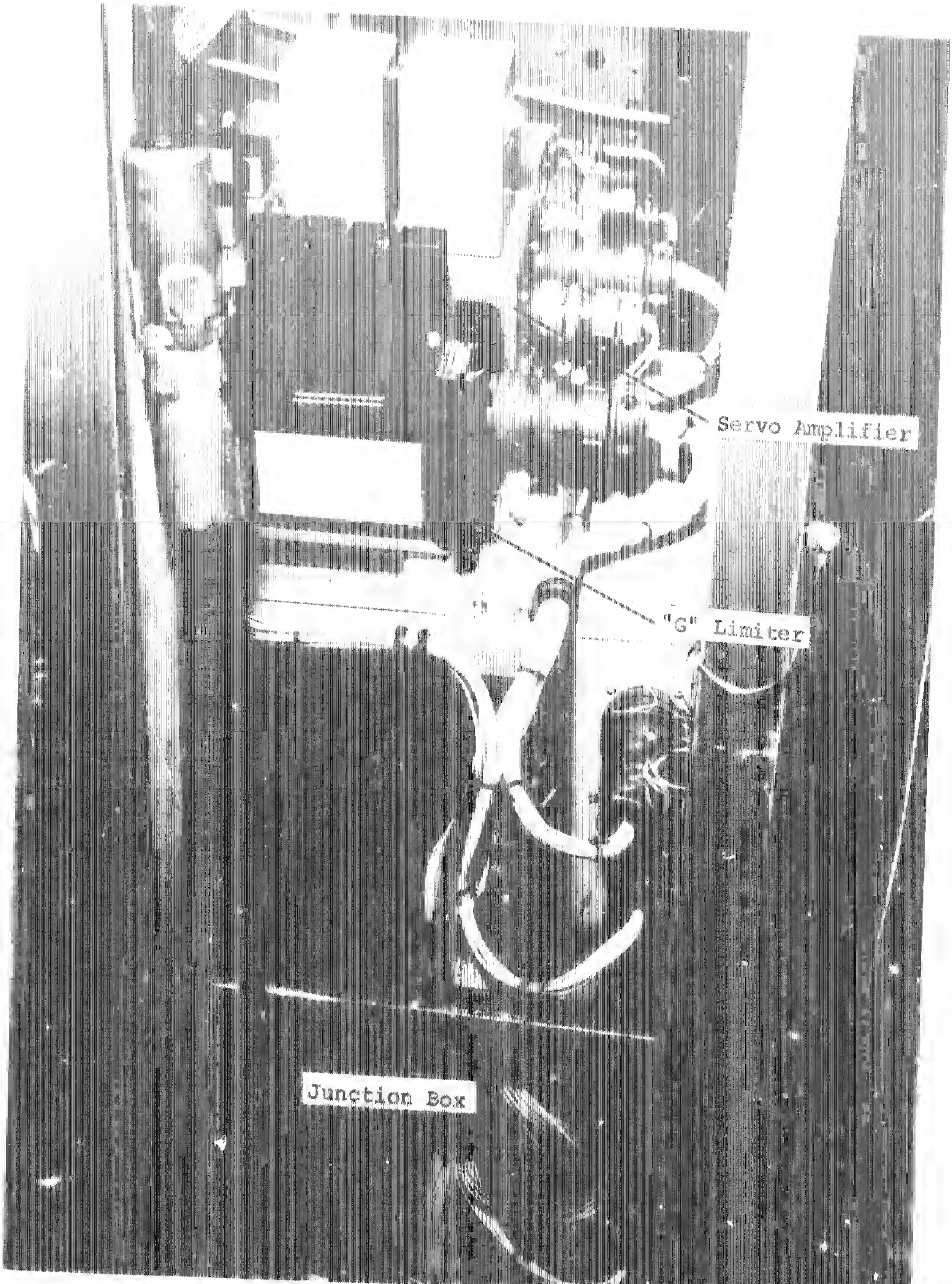


Figure 3. "G" Limiter, Servo Amplifier and Junction Box

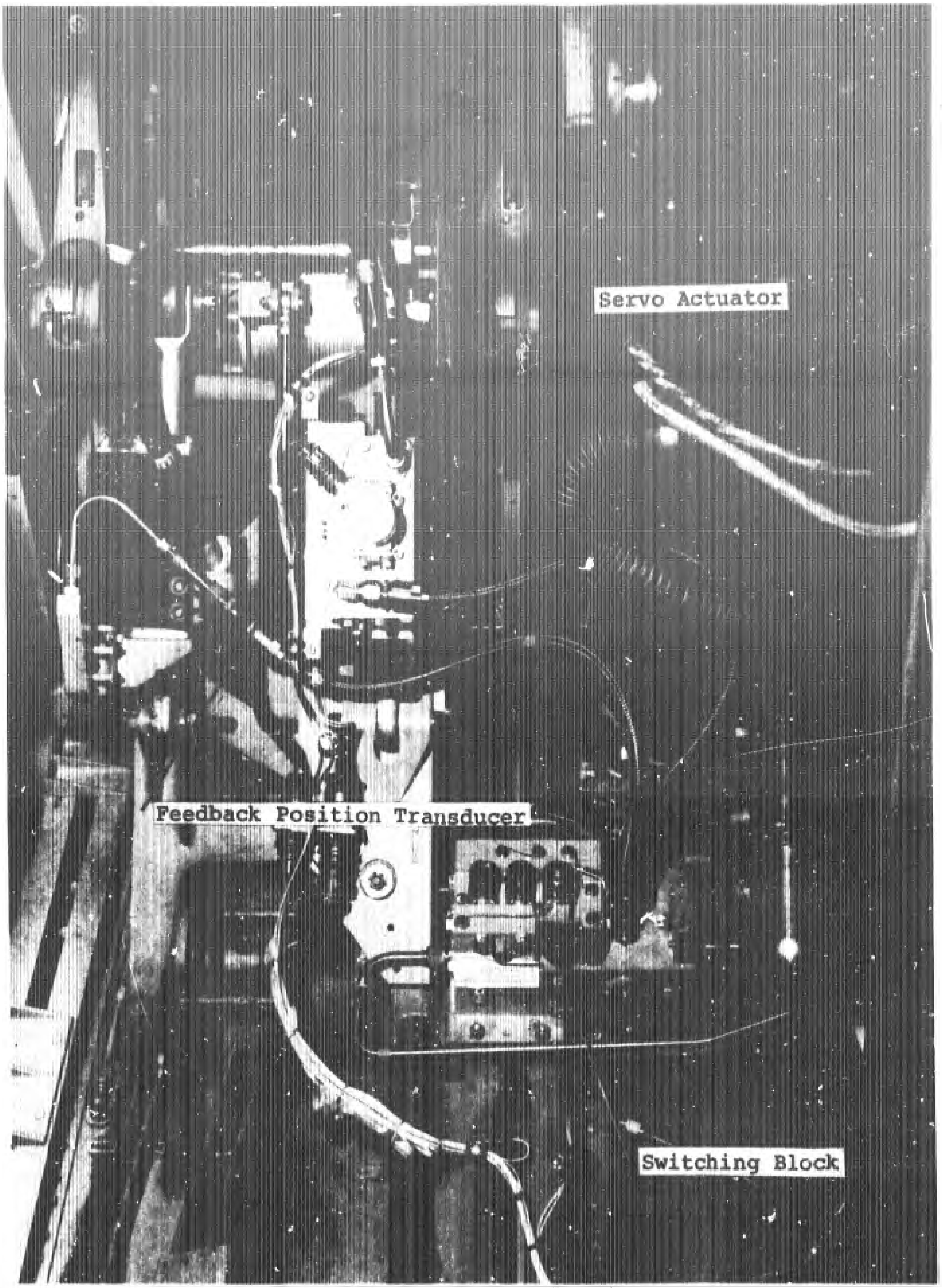


Figure 4. Servo Actuactor, Feedback Position Transducer and Switching Block

PROBLEMS ENCOUNTERED DURING THE PHASE 1 PROGRAM

There were three in-flight malfunctions with the Fly-By-Wire test system during the entire flight test program. The first malfunction was the failure of the "g" limiter to operate during the first test flight. This was traced to incorrect wiring of the accelerometers when initially installed in the test aircraft. The second malfunction occurred after about 20 hours of flight testing, and was an inability to engage the Fly-By-Wire mode of operation. This was isolated to a failure at a terminal strip in one of the permanently installed aircraft junction boxes. The third malfunction, an intermittent loss of hydraulic boost, occurred on March 25, 1968. This malfunction was traced to the lack of a cable tensioning system for the cable used in the aircraft to actuate the manual reversion spool.

A fourth problem occurred on the ground while the aircraft was in the hanger for a periodic maintenance inspection. The operation of the bypass valve used in the Fly-By-Wire actuator became intermittent, due to a valve sleeve distortion. This distortion was caused by an excessive amount of clamping force used to hold the sleeve in position.

The problems encountered with the test system were all of an initial system "wiring-out" nature, did not repeat themselves after correction, and were not in the electrical control channel.

RECOMMENDATIONS - B-47 FLY-BY-WIRE

The basic electrical Fly-By-Wire control channel has been incorporated in the test aircraft as part of the Phase 1 Fly-By-Wire investigation. It is therefore suggested that different general types of controllers be constructed and evaluated as input devices to the Fly-By-Wire system. This investigation could generate significant information for design of the controller elements of Fly-By-Wire control systems.

MECHANIZATION DESCRIPTION

General

The general arrangement of the Phase 1 Control System is shown in Fig. 5 . Referring to that figure, the operation of the system is as follows. The electrical control channel is located in parallel with the standard hydromechanical elevator control system. A position transducer (400 H_z excited LVDT) is attached to the bottom of the control column linkage. This position transducer creates an electrical signal which is fed to the servoamplifier. In the servoamplifier, the 400 H_z output is demodulated and filtered to create a d-c input for the servoamplifier. The servoamplifier controls the electrohydraulic servovalve. The servovalve directs flow to the drive areas of the electrohydraulic actuator, controlling the rate and direction of actuator motion. The position of the electrohydraulic actuator is measured by a second position transducer. The output of this second transducer is fed back to the servoamplifier where it is demodulated, filtered, and subtracted from the demodulated and filtered output of the control column position transducer.

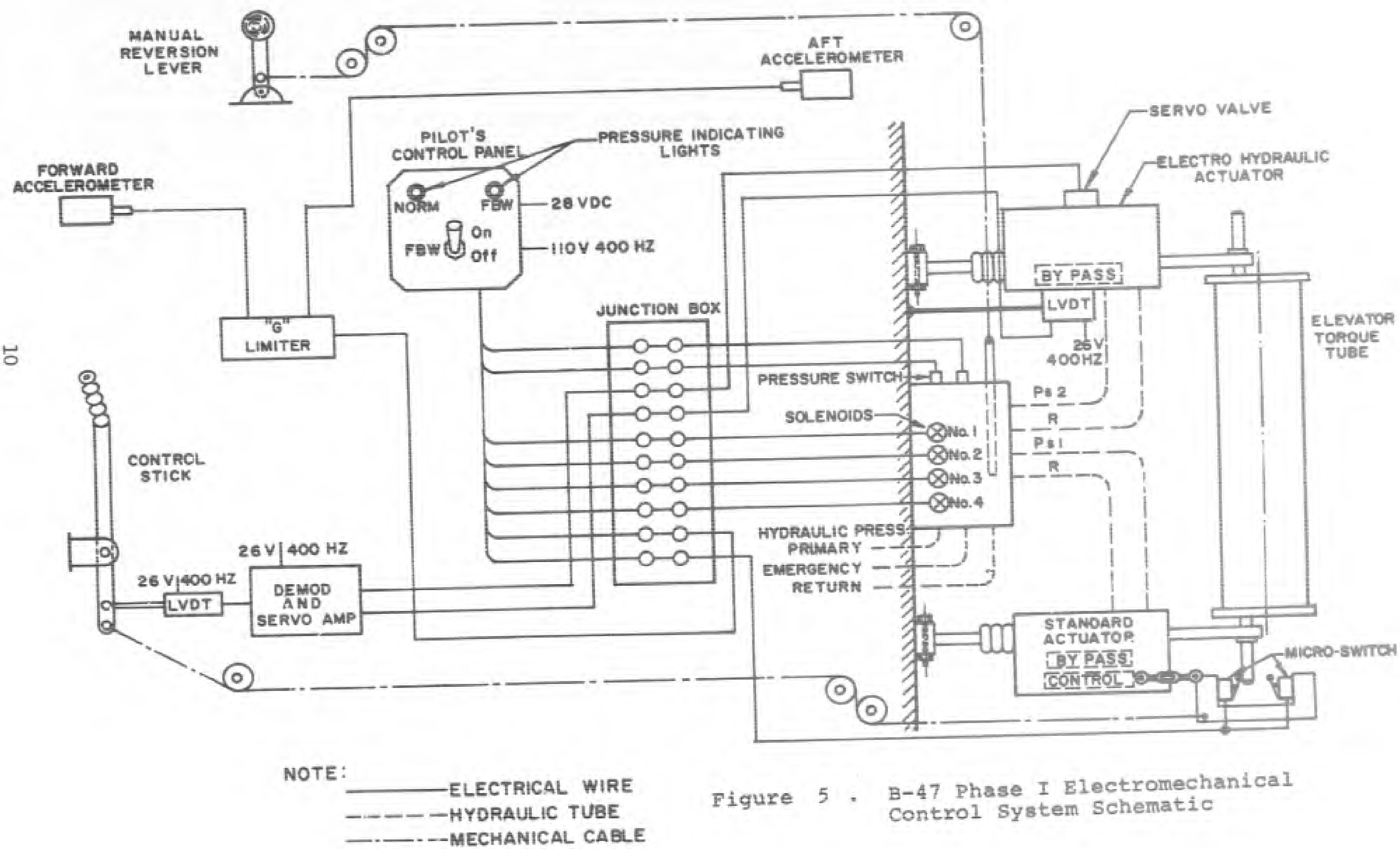


Figure 5 . B-47 Phase I Electromechanical Control System Schematic

To energize the Fly-By-Wire control system, a control panel incorporating indicator lights and a two-position toggle switch is used. When the toggle switch is thrown, solenoids in the switching block mounted in the tail of the aircraft are actuated. These solenoids transfer hydraulic supply pressure from the standard elevator actuator, and apply it to the electrohydraulic actuator. Both actuators incorporate a bypass valve which, upon loss of hydraulic supply pressure to the actuator, interconnects the drive areas of the actuator.

A manual disconnect lever cable connected to the switching block operates a manual transfer valve. When the manual disconnect lever is used, the output of the solenoid valves is overridden, and supply pressure is removed from the electrohydraulic actuator and reconnected to the standard actuator. Reversion is also provided by the "g" limiter. The "g" limiter mixes signals from a forward and aft accelerometer so that normal acceleration in excess of a predetermined level ($\pm 1/2$ "g's" for the flight test) causes the solenoid valves to transfer the elevator control back to the standard actuator.

Microswitches are incorporated on elevator torque tube linkage in order to cause the system to revert to the standard actuator upon an input-to-output-rate disagreement above a certain level.

Pressure switches connected to the output pressure lines of the switching block are connected with lights on the Fly-By-Wire control panel. A green light was used to indicate pressurization of the standard actuator. An amber light was used to indicate pressurization of the electrohydraulic actuator.

Fig. 6 shows the location of the Phase 1 System components in the test aircraft.

Fig. 7 shows the location of the rate gyro and cable tensioning system installed as Mod I and Mod II during the flight test period.

Table I lists the 21 major items, quantity and source, used to make up the Phase 1 System. Those items listed with HR&M as source were designed, assembled, and tested at WPAFB by HR&M personnel; and will be discussed in detail in the individual item mechanization description following. The other items were obtained commercially or from Air Force supply channels, and were either incorporated directly or modified to some degree before incorporation.

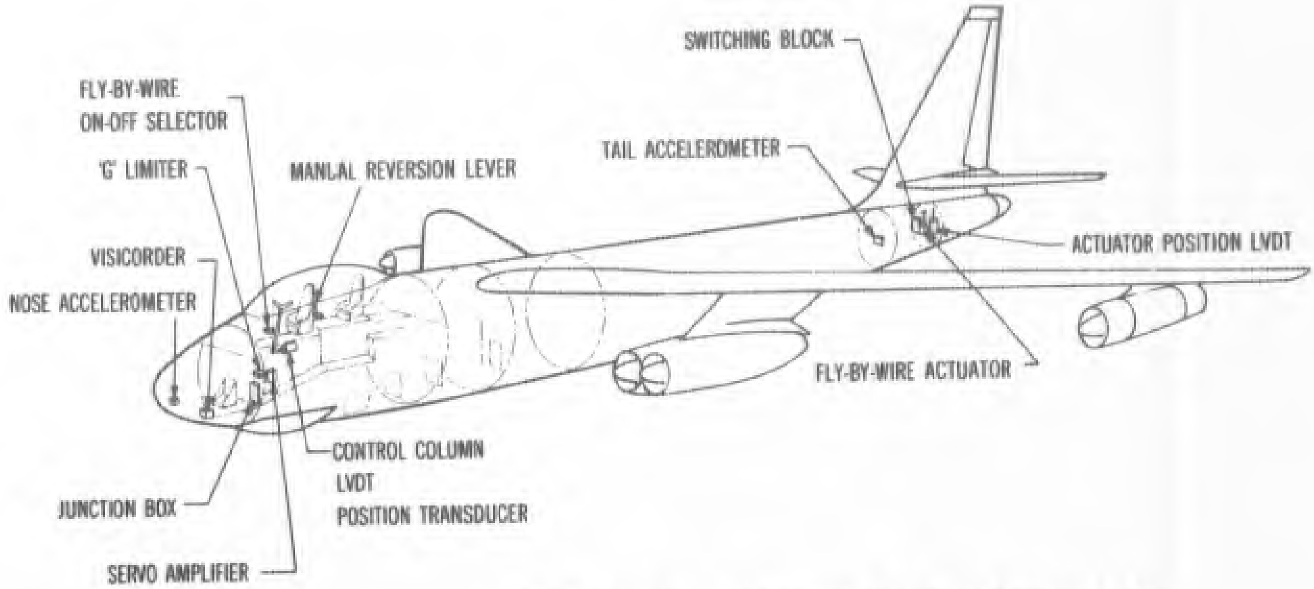
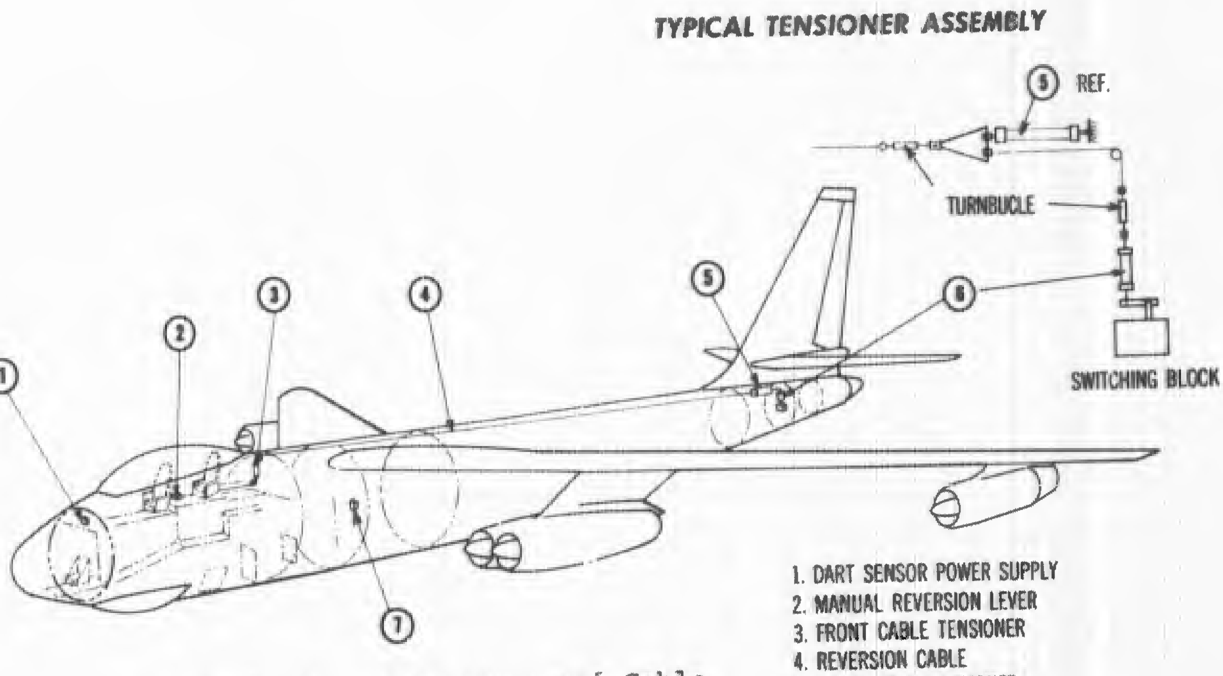


Figure 6 Phase 1 System Components



1. DART SENSOR POWER SUPPLY
2. MANUAL REVERSION LEVER
3. FRONT CABLE TENSIONER
4. REVERSION CABLE
5. REAR CABLE TENSIONER
6. FORCE RELIEF UNIT
7. DART RATE SENSOR

Figure 7 Rate Gyro and Cable Tensioning System

MODIFICATIONS:
MOD 1 CABLE TENSIONING SYSTEM
MOD 11 DART RATE SENSOR

TABLE I

ITEMS PROVIDED FOR PHASE 1 FLY-BY-WIRE AIRCRAFT INSTALLATION

Item No.	Name	Part No.	Qty.	Source
1	Servoamplifier	D-10040	1	HR
2	Control panel	C-10092	1	HR
3	Junction box	D-10083	1	HR
4	Stick LVDT	B-10093	1	HR (Collins LMT-11105)
5	Feedback LVDT	B-10093	1	HR (Collins LMT-11105)
6	"G" limiter calibrator	BG67C-1	1	USAF
7	"G" limiter power supply	BG49A-1	1	JSAF
8	Accelerometer fwd	GG47A-1	1	USAF
9	Accelerometer aft	GG47A-1	1	USAF
10	Torque tube subassembly (ref. dwg D-10001) which includes:			HR
	R.H. elevator crank	D-10002	1	
	Torque tube spacer	D-10003	1	
	Transducer standoff	B-10046	1	
11	Actuator subassembly (ref. dwg D-10001) which includes:		1	HR
	Elevator actuator body	E-10028	1	
	Servo manifold block	D-10014	1	
	Washer	A-10049-1	4	
	Washer	A-10049-2	2	
	Washer	A-10049-3	2	

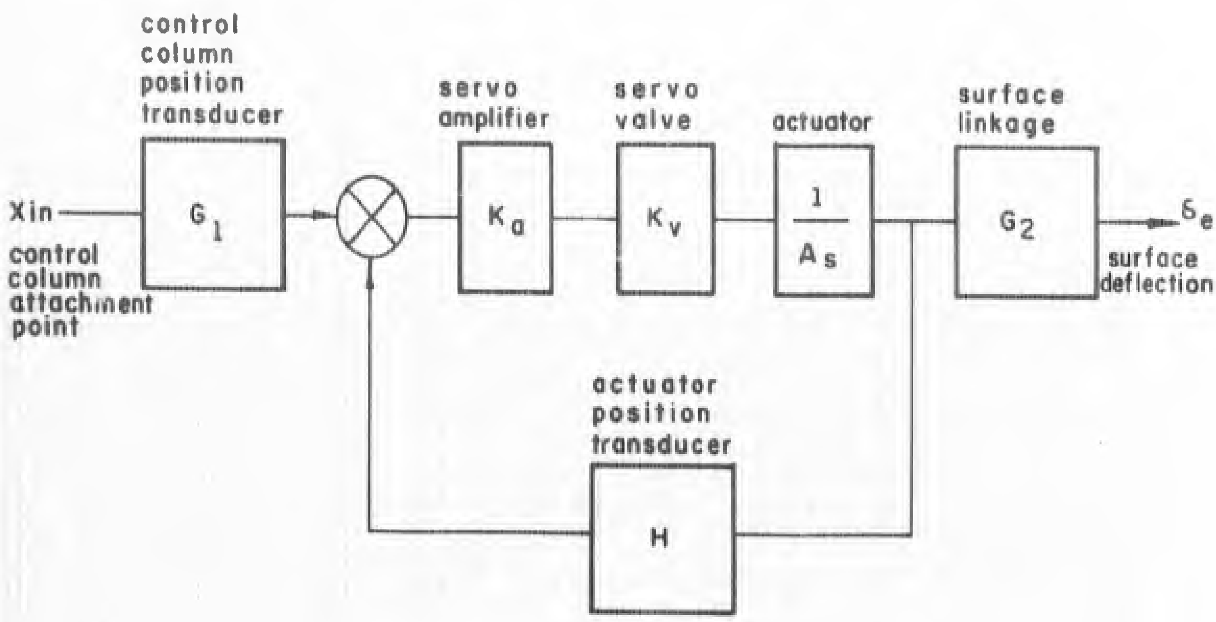
TABLE I (CONCLUDED)

Item No.	Name	Part No.	Qty.	Source
12	Actuator clevis plate subassembly (ref. assembly dwg D-10001 and D-10047) which includes:		1	HR
	Actuator clevis plate	D-10004	1	
	Transducer bracket	B-10045	1	
	Trans. connecting brkt.	B-10027	1	HR
	Bushing	A-10048	1	
	1/2 x 1-1/2 shoulder bolts	----	1	
	3/8-16 nut and washer	----	2 ea.	
	1.2 x 3-1/4 shoulder bolt	----	1	
	3/16 x 1-1/2 roll pins	----	2	
	Lower strap clamp	B-10005	1	
	Upper strap clamp	B-10006	1	
13	Pilot's reversion lever subassembly	D-10057	1	HR
14	Switching block subassembly	E-10089	1	HR
15	Bronze thrust washers for mounting LVDT FWD	----	4	HR
16	Limit switch subassembly (ref. dwg B-10119) which includes:			
	Bracket upper limit switch	A-10116	1	HR
	Bracket lower limit switch	A-10117	1	HR
17	Hydraulic filter	031616	1	HR (Bendix)
18 MOD I	Cable tensioner subassembly	D-10205	1	HR
19 MOD I	Force relief unit	D-10206	1	HR
20 MOD II	Rate gyro (Nortronics)	X64836-2	1	USAF
21 MOD II	Rate gyro power supply	8-10395	1	HR

Individual Item Mechanization Description

1. Control Circuit

The electrical control channel was sized to be identical in gain, frequency response, output displacement and slew rate with the existing B-47 elevator control channel. The following Fig. 8 .llustrates the control circuit in block diagram form.



Where G_1 = 0.820 volts/in.

K_a = 84.4 ma/volt

K_v = 0.623 cis/ma

H = 0.820 volts/in.

A = 2.7 in.^{2}

G_2 = 12°/in.

Figure 8 B-47 Control Circuit Block Diagram

Note that the control circuit is that of a first-order lag circuit with a -3 db break frequency at 2.6 H_z.

2. Servoamplifier

The servoamplifier design was based on utilizing the following major purchased components.

a. Analog devices No. 211 chopper stabilized operational amplifier (Qty. 1)

b. Technipower 15-volt power supply model No. PM 15.8-0.100 (Qty. 2)

c. Collins LVDT demodulator (designed for operation with Collins LMT-11105 LVDT's) (Qty. 2)

d. Electronic specialty relay No. 806B5 3N-4-A-5K (3) (Qty. 1)

e. Triad 110 V to 28V 400 H_z transformer No. 10938 (Qty. 1)

All these components were MIL specification (MIL-E-5400J) items with the exception of the No. 211 operational amplifier. However, this amplifier was compatible with the MIL specifications. The demodulator was used to convert the 400 H_z output of the column and actuator LVDT's to a rectified d-c signal with 800 H_z ripple. The output of the demodulator was filtered using a two-section R.C. filter. This filter gave 46 db attenuation to the 800 H_z ripple and an input-to-output servoamplifier response 3 db down at 34 H_z (ref. Fig. 9 . and 10). The servoamplifier break frequency

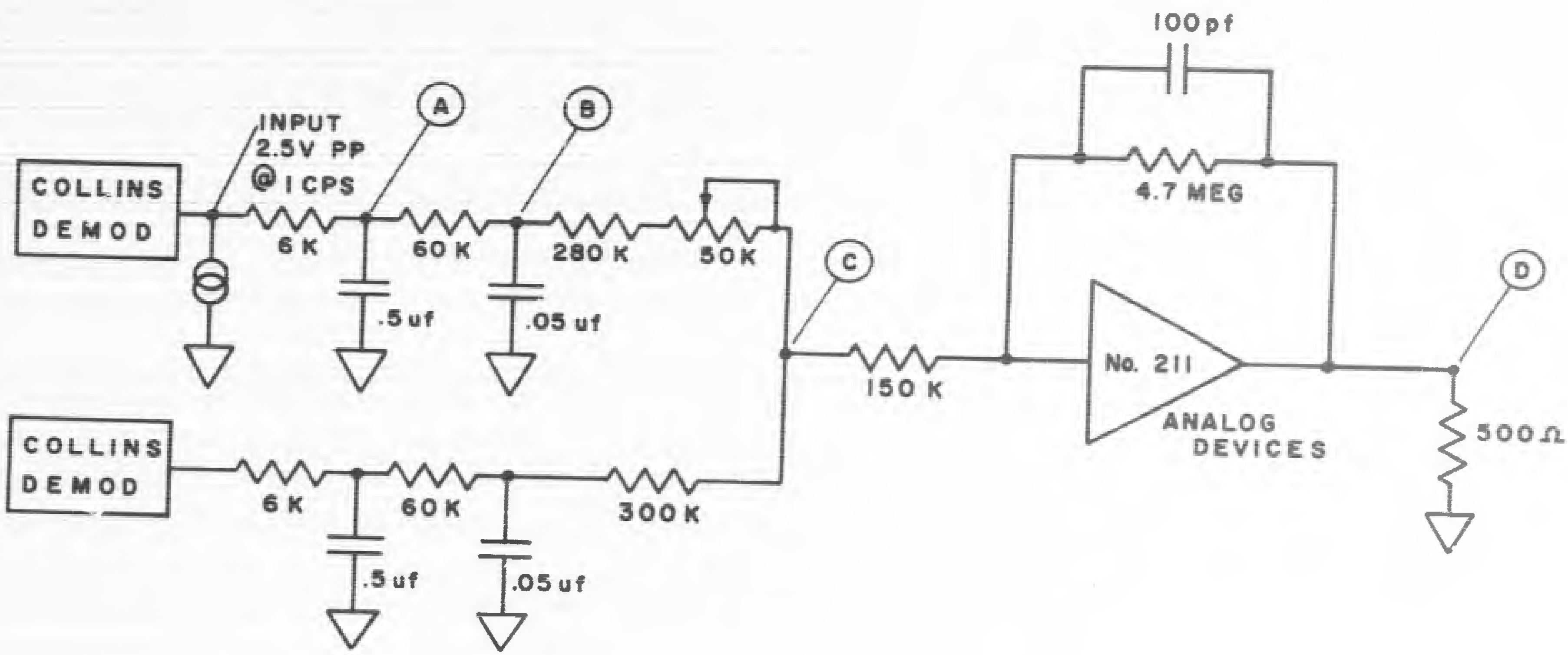


Figure 9 . Servo Amplifier Response Test
Input-Output Relationships
Signal To Noise Ratio - 46 Db

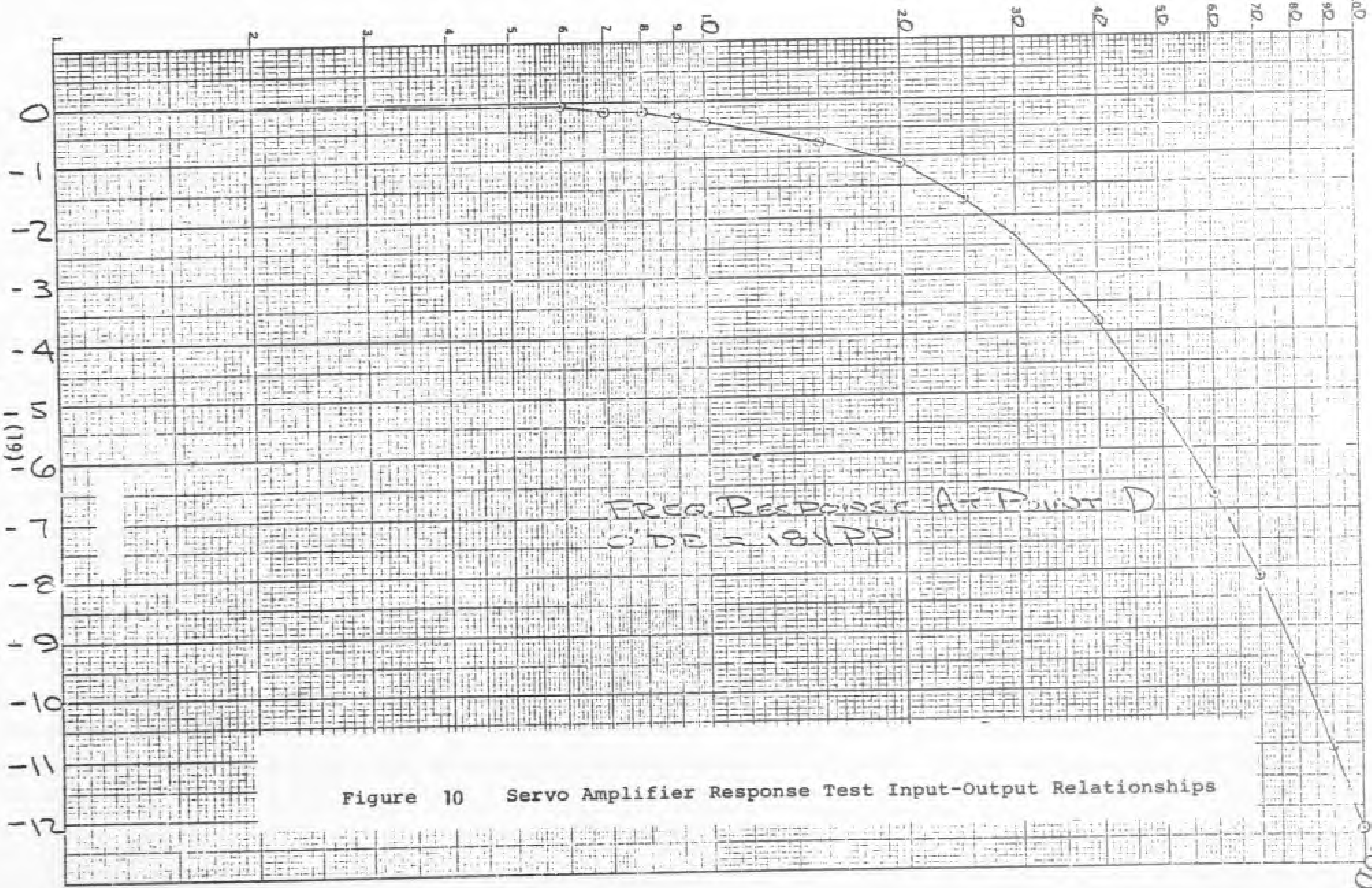


Figure 10 Servo Amplifier Response Test Input-Output Relationships

was designed to be more than a decade away from the control loop break frequency. The servoamplifier output is ± 13 volts at 20 ma. The input impedance is greater than 6,000 ohms at all frequencies. The electronic specialty relay is used to ground the input of the operational amplifier when not in the Fly-By-Wire mode. This is done in order to prevent operating the servovalve without hydraulic pressure being applied. The Triad transformer is used to reduce the 115-volt 400 H_z excitation signal to the required 26-volt LVDT excitation voltage. Fig. 11 shows the servoamplifier external and internal appearance as it was used in the test aircraft. Fig. 12 is the servoamplifier circuit schematic.

2. Servoactuator

The electrohydraulic actuator design used for the Fly-By-Wire system utilized a bypass spool, sleeve and piston rod of a standard B-47 elevator actuator. These components were housed in an actuator body designed to mount a Hydraulic Research and Manufacturing flapper-nozzle servovalve. This servovalve (HR&M P.N. 22265500) was designed to match the flow gain and pressure gain of the characteristics of the standard actuator control spool. The bypass valve operated to interconnect the two drive areas of the actuator pistons when the supply pressure level drops below 400 psi. The actuator was designed to accept 3000-psi operation as well as the 1500-psi supply pressure of the test aircraft.

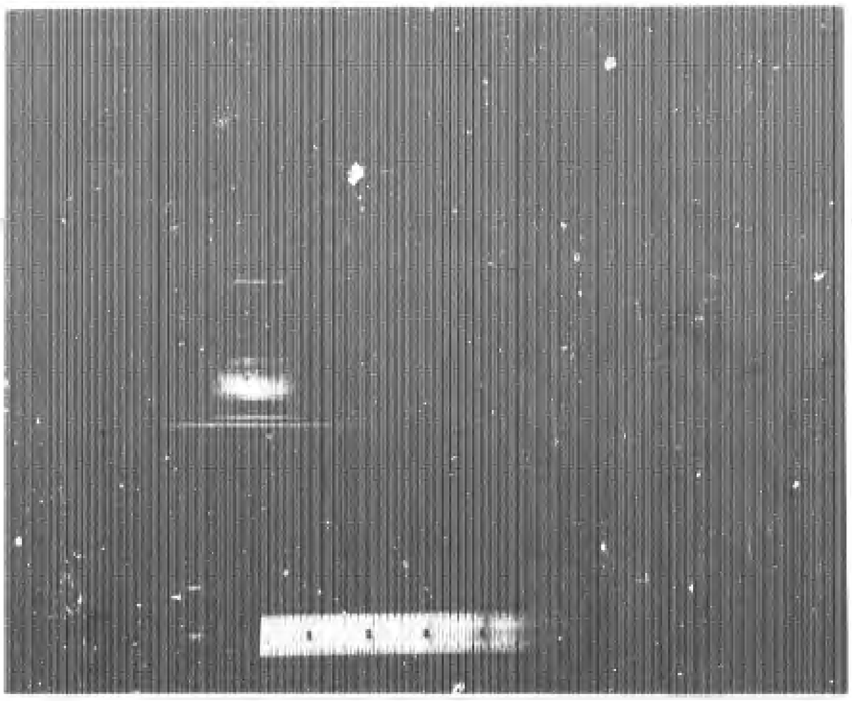
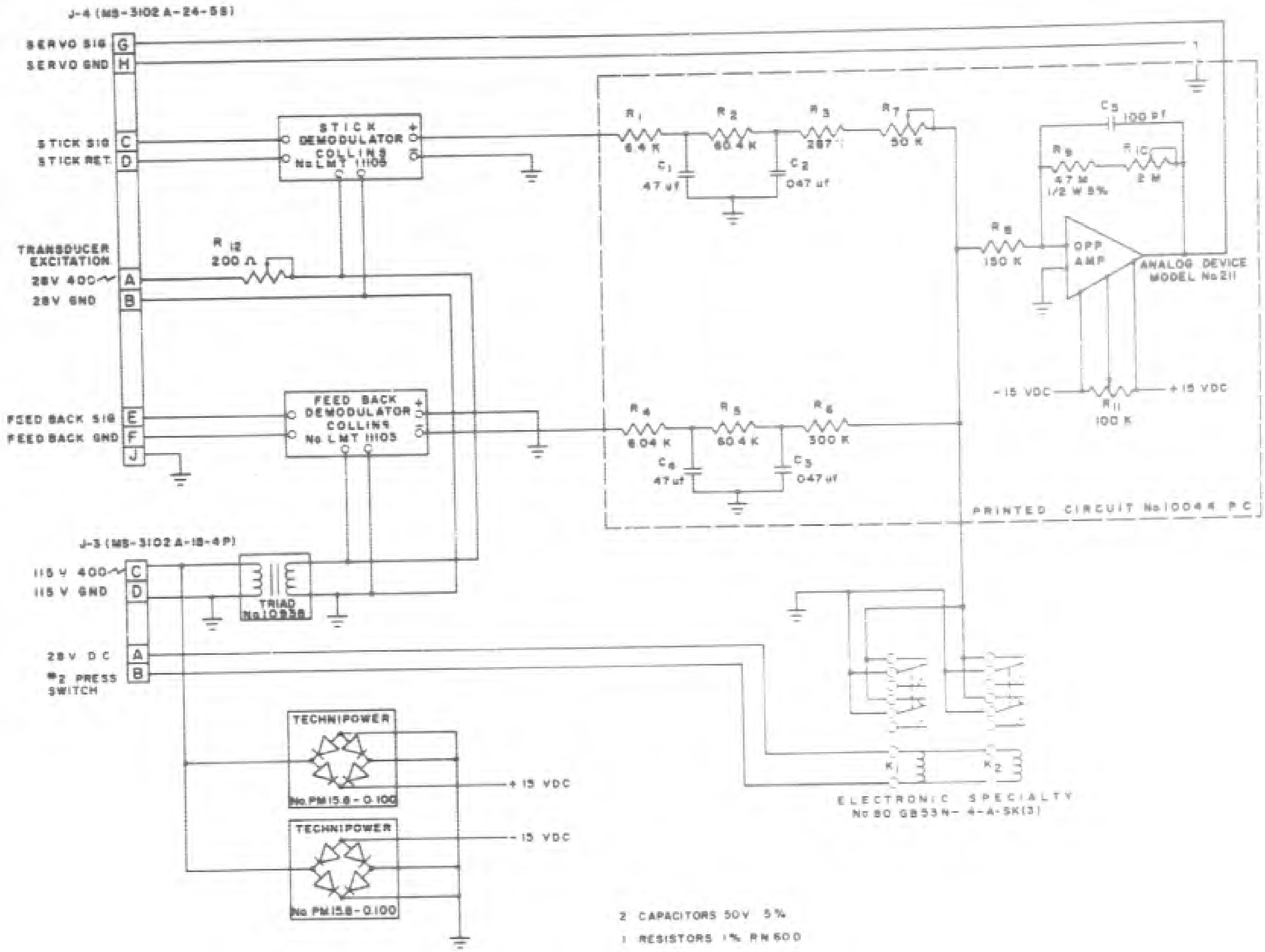


Figure 11a. External View of the B-47 Amplifier

Figure 11b. Internal View of B-47 Amplifier



NOTE

1 RESISTORS 1% RN 60D

2 CAPACITORS 50V 5%

Figure 12 B-47 Servoamplifier Schematic Diagram

3. Position Transducers

The position transducers used to create electrical signals proportional to column and actuator positions were identical. The units used were G.L. Collins Corporation LMT-11105 transducers having the following characteristics.

Input voltage - 26 $\pm$ 1.3 v @ 400 $\pm$ 20 H_z
Power requirements-less than 1 watt
Scale factor - 6 volts $\pm$ 1 percent/in.
Electrical stroke - $\pm$ 1.812 in.
Size - 0.750 in. dia x 12 in. long

For mechanical nulling purposes a $\pm$ 0.125 in. adjustable bearing end was incorporated at one end of each transducer. In order to seal the transducer from dust, a Gortite Neoprene Bellows was mounted over the body of the transducer. Fig. 13 shows the actuator position transducer before installation in the test aircraft.

4. Switching Block

The switching block was designed to transfer supply pressure from the elevator actuator to the electrohydraulic actuator and back again. This transfer was accomplished either electrically by using solenoid valves or mechanically using a manual transfer valve. The switching block was designed (Ref. Fig. 14) so that the manual transfer valve overrode the solenoid valve control. Four pilot solenoids (2 HR&M P.N 55367, 2 HR&M P.N. 84000018) were used to provide redundancy of these elements. These pilot solenoids operate two switching spools by applying or removing supply pressure from a

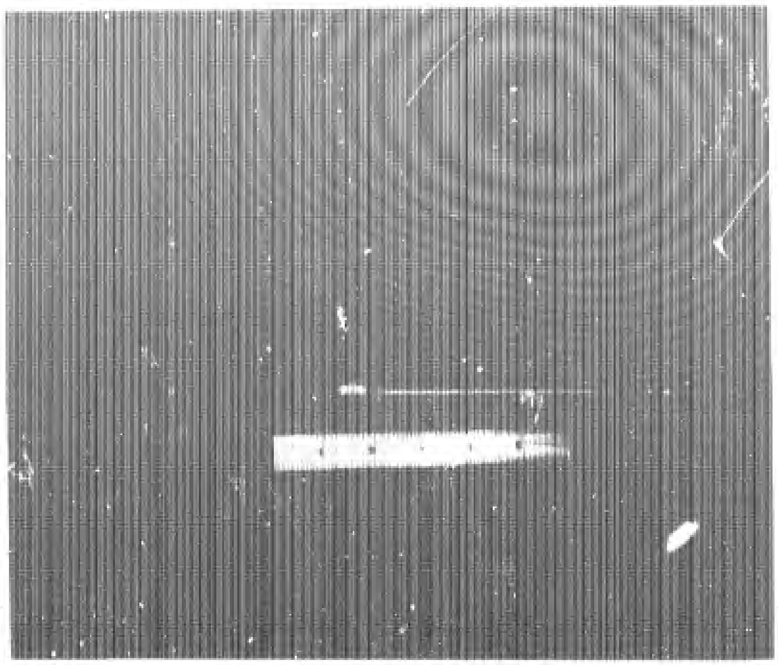


Figure 13 B-47 Actuator Position Transducer

drive piston on one end of each switching spool. The arrangement of the solenoid valves was such that loss of the 28-volt aircraft electrical supply would transfer elevator control back to the standard elevator actuator. In addition, the arrangement of the solenoid valve output interconnection was made so that any single solenoid valve failure (while in the Fly-By-Wire mode of operation) would not prevent turning off the Fly-By-Wire system. The switching block output was indicated by HR&M pressure-operated switches (HR&M Part No. 84000-17), which were connected to indicator lights on the cockpit-mounted Fly-By-Wire control panel. A green light was used

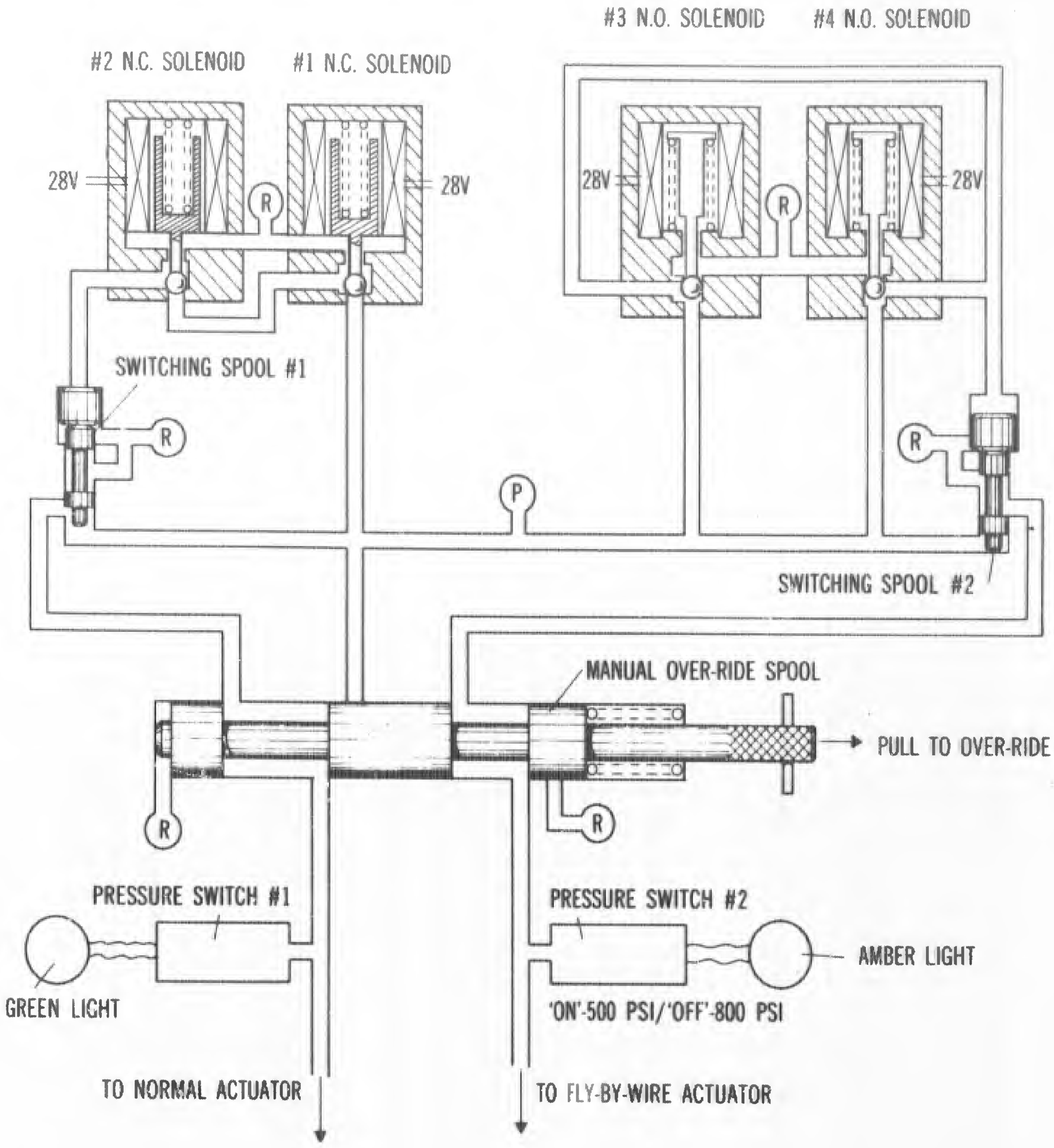


Figure 14 B-47 Switching Block Schematic
(Shown in F-B-W Off Mode)

to indicate pressurization of the standard actuator. An amber light was used to indicate pressurization of the Fly-By-Wire actuator. The wrong light or both lights being on or off would directly indicate solenoid malfunction. The failure mode table, Table II lists the effect of each solenoid failure for both manual and Fly-By-Wire mode of operation. Fig. 15 shows the switching block before assembly into the test aircraft.

5. Fly-By-Wire Junction Box

To provide instrumentation connections and convenient test points for systems checkout, all the Fly-By-Wire wiring was run through a junction box. Fig. 16 shows the internal construction of the junction box. Relays for the Fly-By-Wire disconnect circuit were mounted in the junction box.

6. Fly-By-Wire Control Panel

Fig. 17 shows the front of the Fly-By-Wire control panel. Two (2) press-to-test indicator lights were used to indicate mode of operation. A single-pole, double-throw toggle switch was utilized to engage or disengage the Fly-By-Wire system. Fig. 18 shows the aircraft cockpit installation of the Fly-By-Wire control panel.

TABLE II

FLY-BY-WIRE FAILURE MODE

Solenoid No.	Solenoid d-c Power Loss	Failure Jam	Switching Block Mode of Operation	Immediate Effect	Subsequent Effect
1	x		Normal	None	Green light stays on in switching to FBW
1		x	Normal	None	Green light stays on in switching to FBW
1	x		FBW	Green light comes back on	———
1		x	FBW	None	None
2	x		Normal	None	Green light stays on in switching to FBW
2		x	Normal	None	Green light stays on in switching to FBW
2	x		FBW	Green light comes back on	———
2		x	FBW	None	None
3	x		Normal	None	Amber light does not come on in switching to FBW
3		x	Normal	None	Amber light does not come on in switching to FBW
3	x		FBW	Amber light goes out	———
3		x	FBW	None	In switching to FBW amber light does not come on

TABLE II (CONCLUDED)

FLY-BY-WIRE FAILURE MODE

Solenoid No.	Solenoid d-c Power Loss	Failure Jam	Switching Block Mode of Operation	Immediate Effect	Subsequent Effect
4	x		Normal	None	Amber light does not come on in switching to FBW
4		x	Normal	None	Amber light does not come on in switching to FBW
4	x		FBW	Amber light goes out	—
4		x	FBW	None	Amber light does not come on in switching to FBW

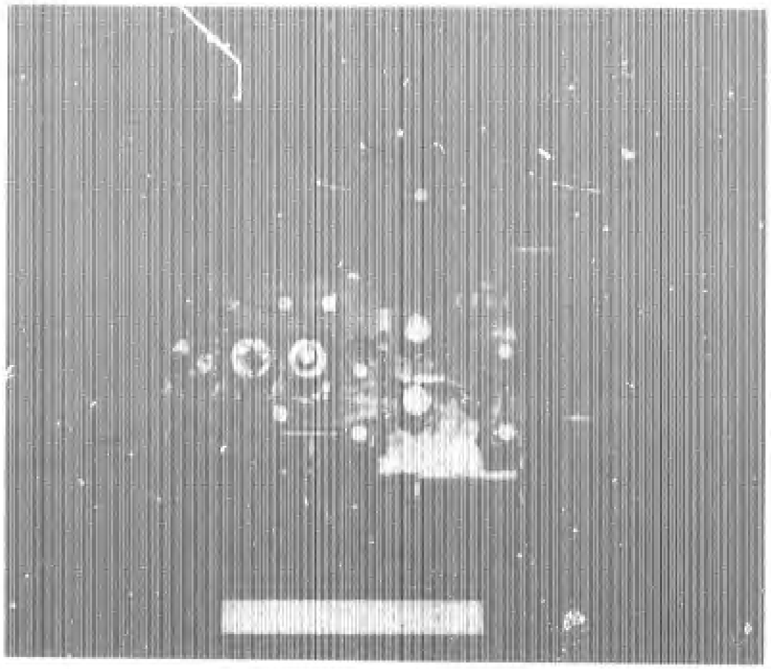


Figure 15 B-47 Switching Block

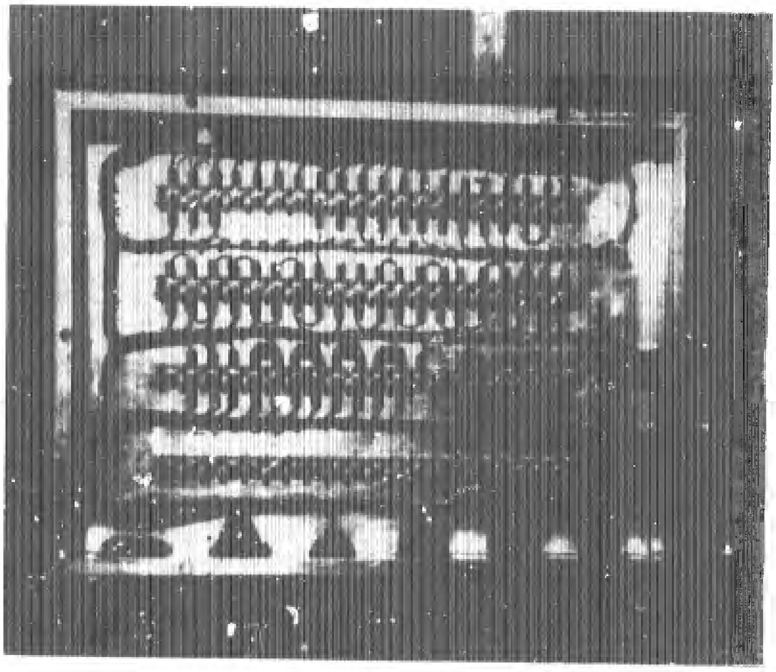


Figure 16 Internal View of B-47 Junction Box

Figure 17 Fly-By-Wire Control Panel

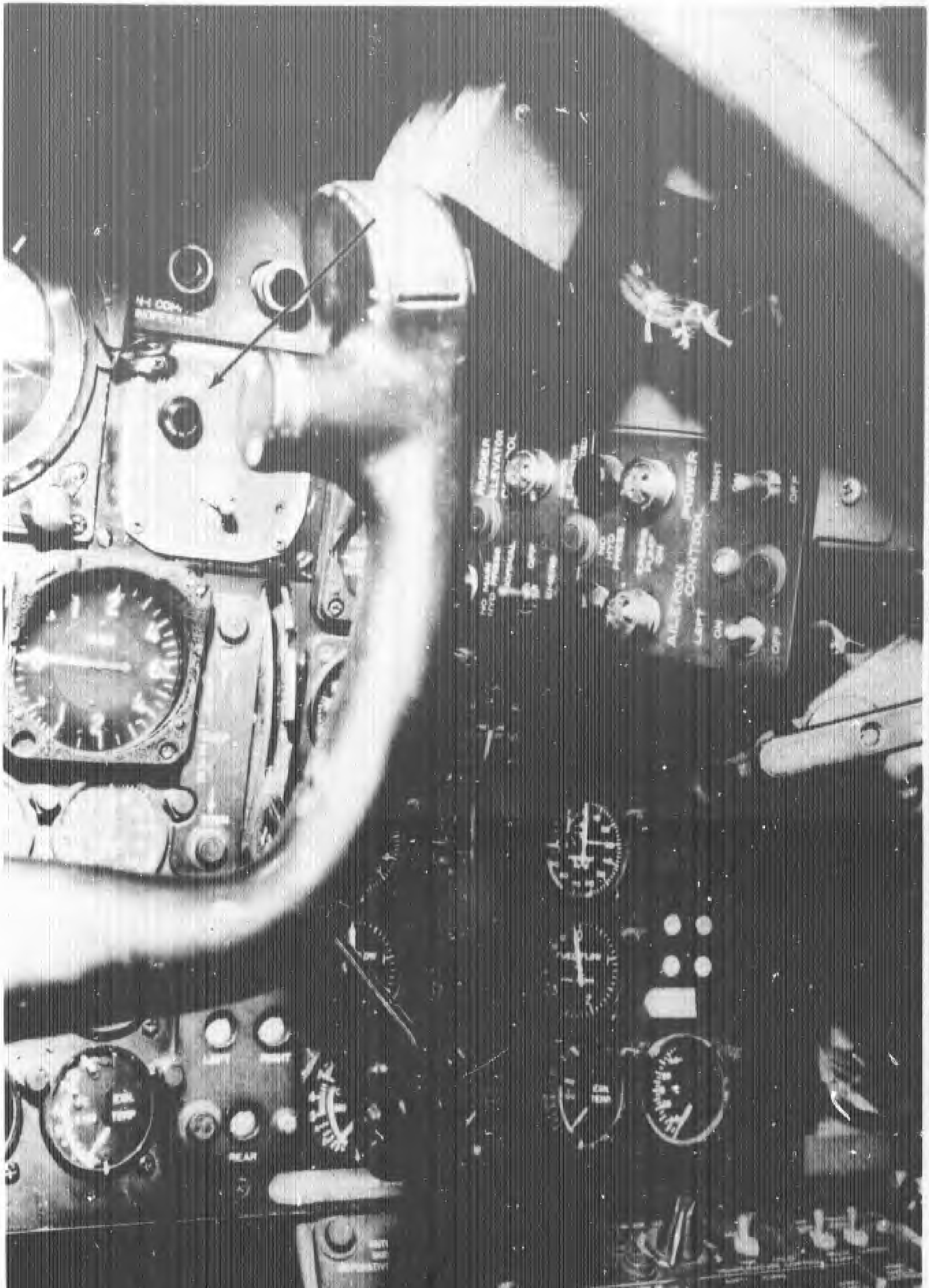


Figure 18. Fly-By-Wire Control Panel Installed in B-47 Test Aircraft

7. "g" Limiter

The "g" limiter used for normal acceleration reversion from the Fly-By-Wire system utilized an F-100 "g" limiter. The switchout point was reset to a level more acceptable for B-47 operation. The "g" limiter utilized a d-c type accelerometer to saturate the magnetic core of a-c transformers. These transformers were used to connect an a-c 400 H_z voltage to a-c operated relays. A buildup of acceleration causes the accelerometers to saturate the transformers, causing the relays to lose their a-c pull-in voltage and drop out. Two accelerometers are used and their outputs summed in the "g" limiter control box. One accelerometer is located in the nose of the aircraft. A second unit is located in the tail section. Rotational acceleration was therefore cancelled out and only normal acceleration caused the a-c relays to drop out.

8. Fly-By-Wire Disconnect System

Fig. 19 shows the Fly-By-Wire disconnect circuit used to cause reversion from the Fly-By-Wire system. Interlock relays were used to prevent reengagement of the Fly-By-Wire until the Fly-By-Wire switch was first turned off.

9. Hydraulic Connections

The Fly-By-Wire actuator used the normal aircraft elevator-rudder 1500 psi, 3.4 gpm hydraulic supply. Since the Fly-By-Wire actuator flow and pressure requirements were essentially the same as the standard elevator actuator's,

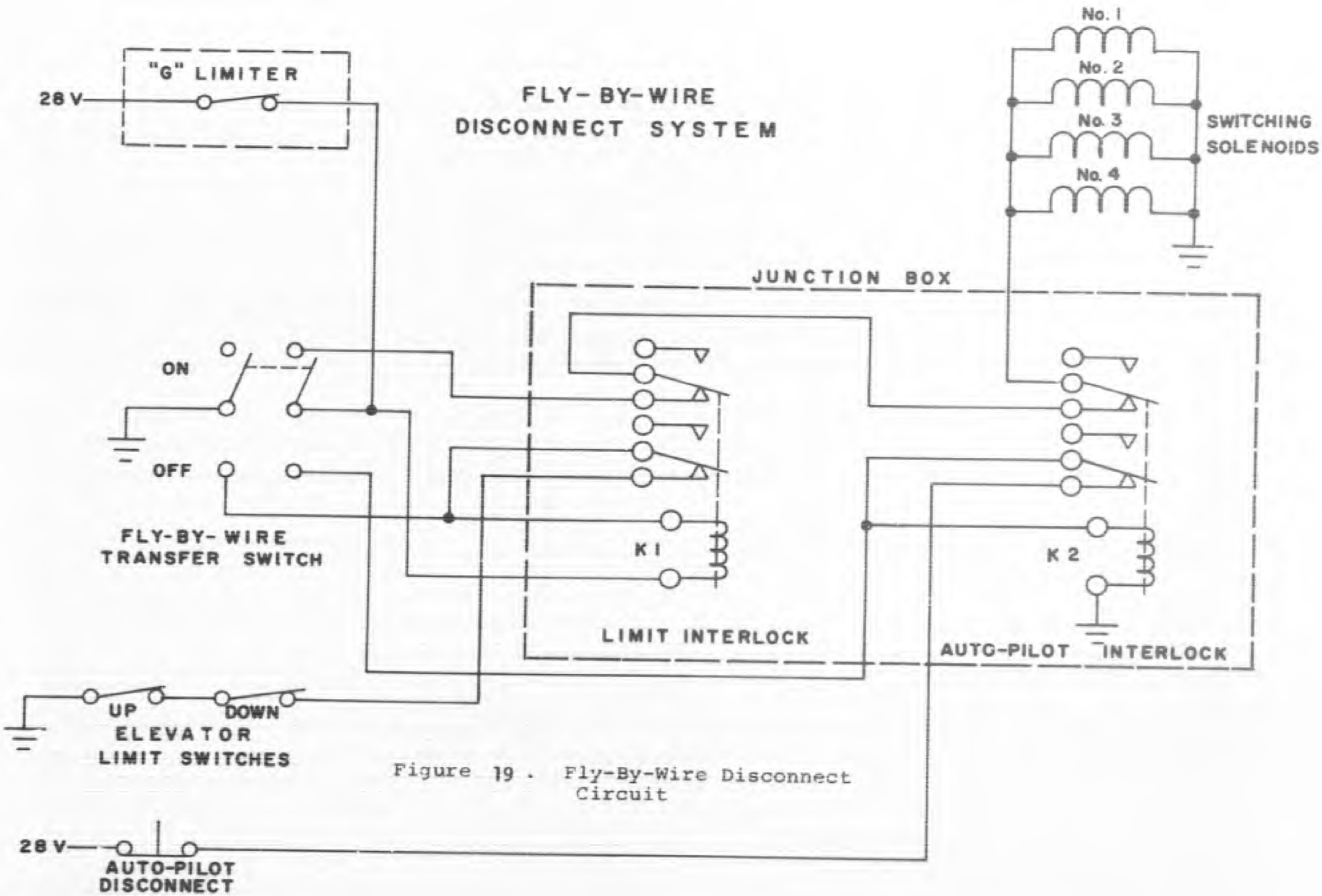


Figure 19. Fly-By-Wire Disconnect Circuit

no additional hydraulic power demand was placed on the aircraft supply. To provide additional hydraulic fluid filtering, a Bendix 031618 filter was incorporated between the aircraft supply and the switching block. This filter has a 2-micron nominal, 5-micron absolute filtration rating, and a flow rating of 10 gpm for MIL-O-5606 fluid at 150°F. A pressure transducer (Stratham Laboratory Model 2890P10-3 MG-350) was connected downstream of the filter in order to monitor hydraulic pressure. Fig. 20 is a schematic of the hydraulic connections.

10. Manual Reversion Lever

The manual reversion lever assembly used to actuate the manual reversion spool was mounted on the right side of the pilot's seat in the test aircraft (Ref. Fig. 21). The lever was designed to lock in the down position by moving the lever arm sideways 3/8 in. Cable tension held the lever in the up position, and returned the lever to the up position when unlocked. Connection of the reversion assembly to the switching block was so arranged that with the reversion lever up, the pilot solenoid valves in the switching block controlled the normal and Fly-By-Wire actuator pressurization. With the reversion lever locked in the down position, the manual reversion spool in the switching block was positioned so that the Fly-By-Wire actuator was connected to hydraulic return, and the standard actuator was pressurized.

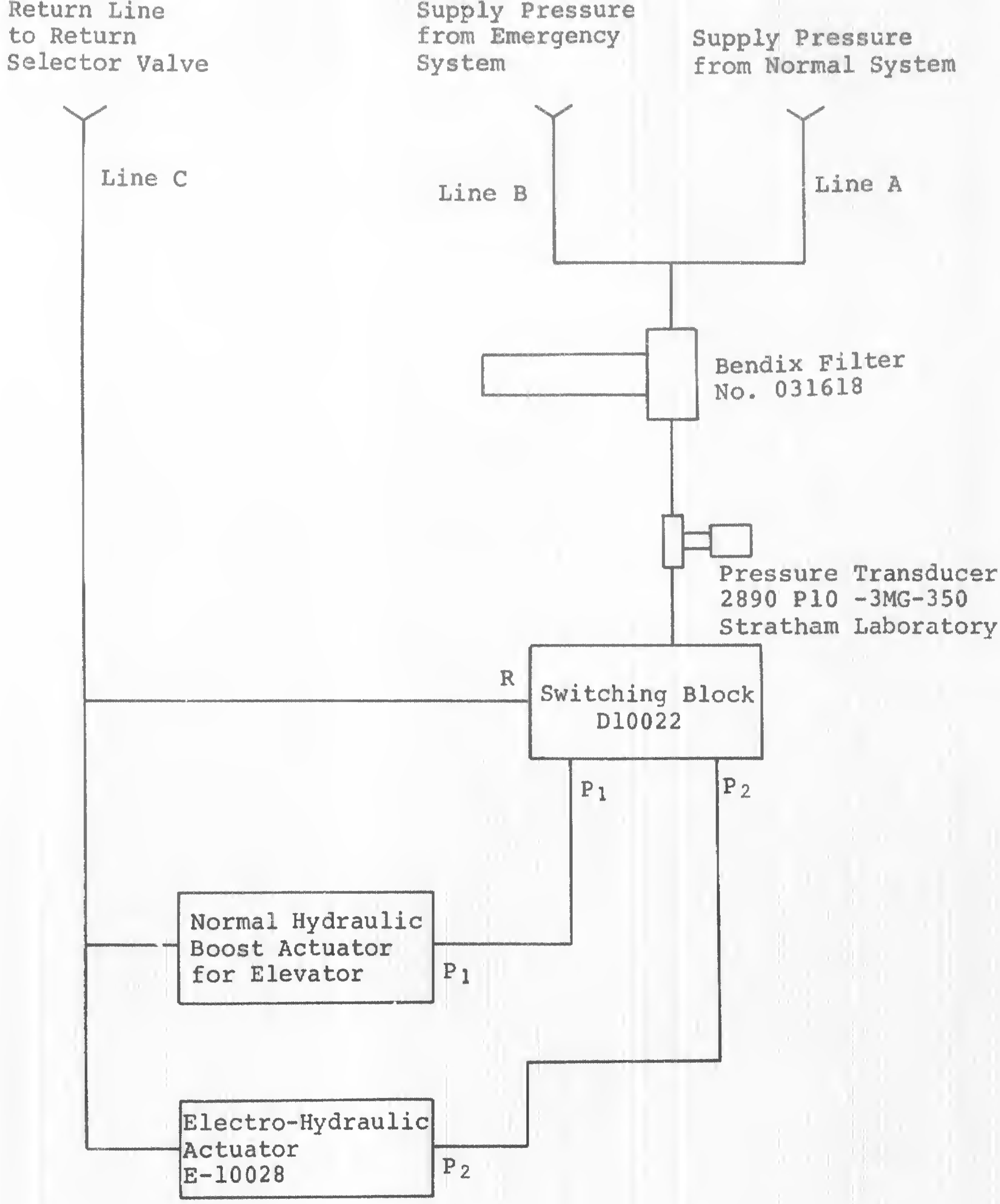


Figure 20 Hydraulic Connection Schematic
Fly-By-Wire B-47 Phase I

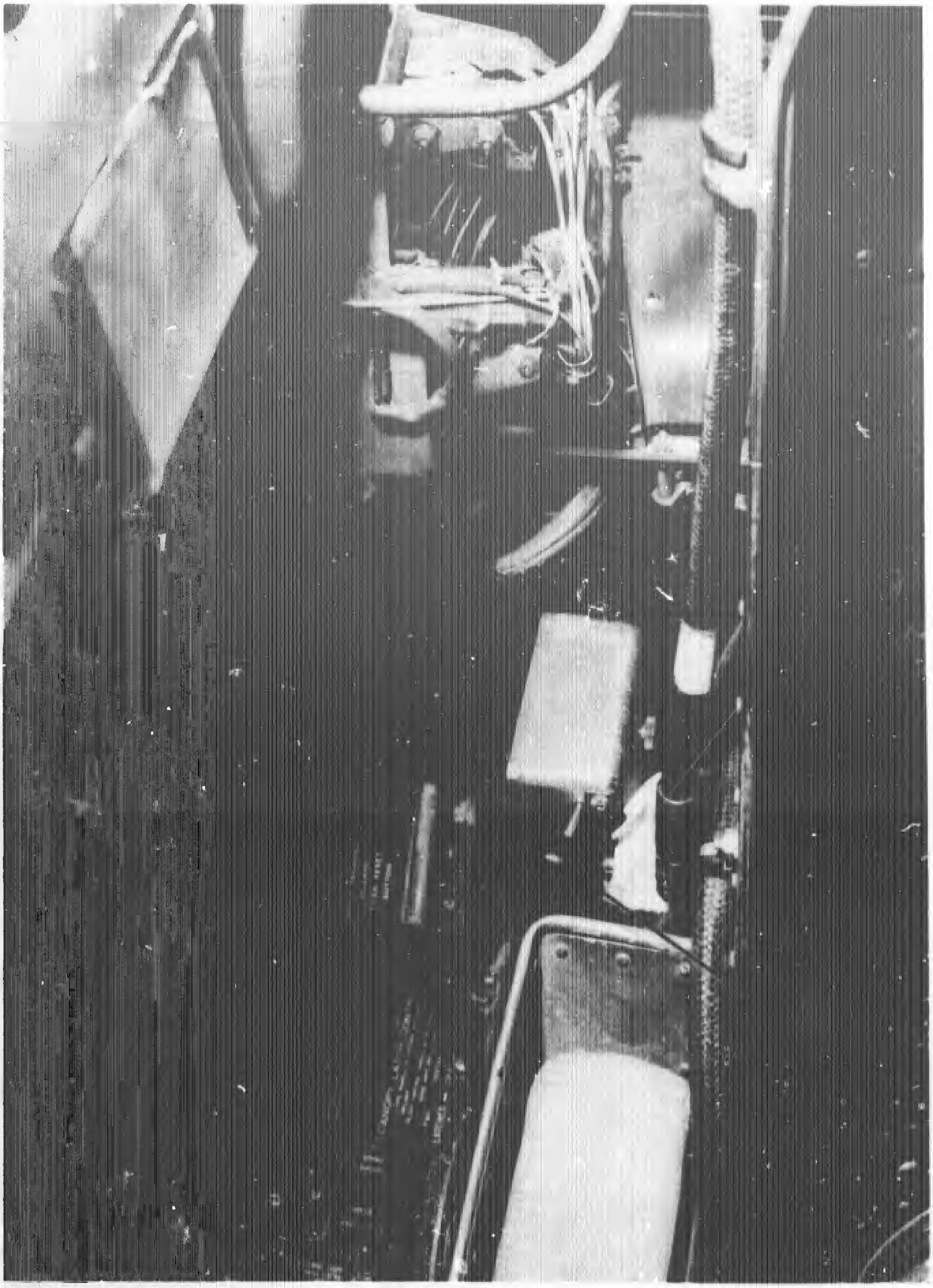


Figure 21. Manual Reversion Lever

11. Cable Tensioning System

The tensioning system for the manual reversion cable was installed partway through the flight test program. As originally installed in the test aircraft, no method of tensioning the cable was provided. To provide a more positive cable connection, a spring-loaded tensioner and a force-relief cartridge were incorporated. (Ref. back to Fig. 7 . The spring tensioner was designed to pull all the cable slack to the tail end of the test aircraft by applying an initial cable tension of 20 lb. This left a slack section of cable between where the cable tensioner attached and the manual reversion input lever on the switching block. This slack section was to allow aircraft flexing without causing the reversion spool to move. The force relief cartridge was preloaded to take 80 lb before it would extend. This cartridge was incorporated to prevent aircraft flexing from causing damage to the manual reversion spool input lever assembly with the manual reversion locked in the down position.

12. Rate Gyro

A Nortronics X-64836-2 rate gyro was installed partway through the flight test program to allow obtaining pitch rate data. A power supply, providing $\pm$ 15 volts d-c and 26 volts 400 H_z a-c output, was constructed by HR&M personnel in order to power the gyro and its pickoff. For noise elimination an RC filter having a -3 db break frequency of 28 H_z was applied to the output of the rate gyro. (Refer back to Fig. 6 for aircraft location of the components).

13. Actuator Mounting

In order to mount the Fly-By-Wire actuator into the test aircraft, it was necessary to design, have fabricated, and assemble a special torque tube and clevis plate. The torque tube replaced the existing torque tube and crank arm in the test aircraft and the clevis plate clamped to the torque tube support framework. The torque tube was designed to take over twice the load that the original torque tube could withstand, and was machined from solid 2024-T4 aluminum. The clevis plate was designed to clamp to the support framework and used a 1-in. steel pin passing through the framework and clevis plate to resist the actuator output force. To the torque tube and clevis plate were mounted brackets which supported the actuator position transducer. Fig. 22 shows the clevis plate and torque tube assembly.

TEST PROCEDURES

The object of the tests performed on the Fly-By-Wire Phase I system was evaluation of the system under various flight conditions. For the first check flight of the test system, the B-47 test aircraft was operated with the following conditions.

(a) Gross weight between 110,000 and 125,000 lb.

(b) C.G. between 20 and 25 percent MAC

(c) Test altitude 15,000 to 23,000 ft.

(d) Load limit reversion factor $\pm$ 0.5 incremental g's

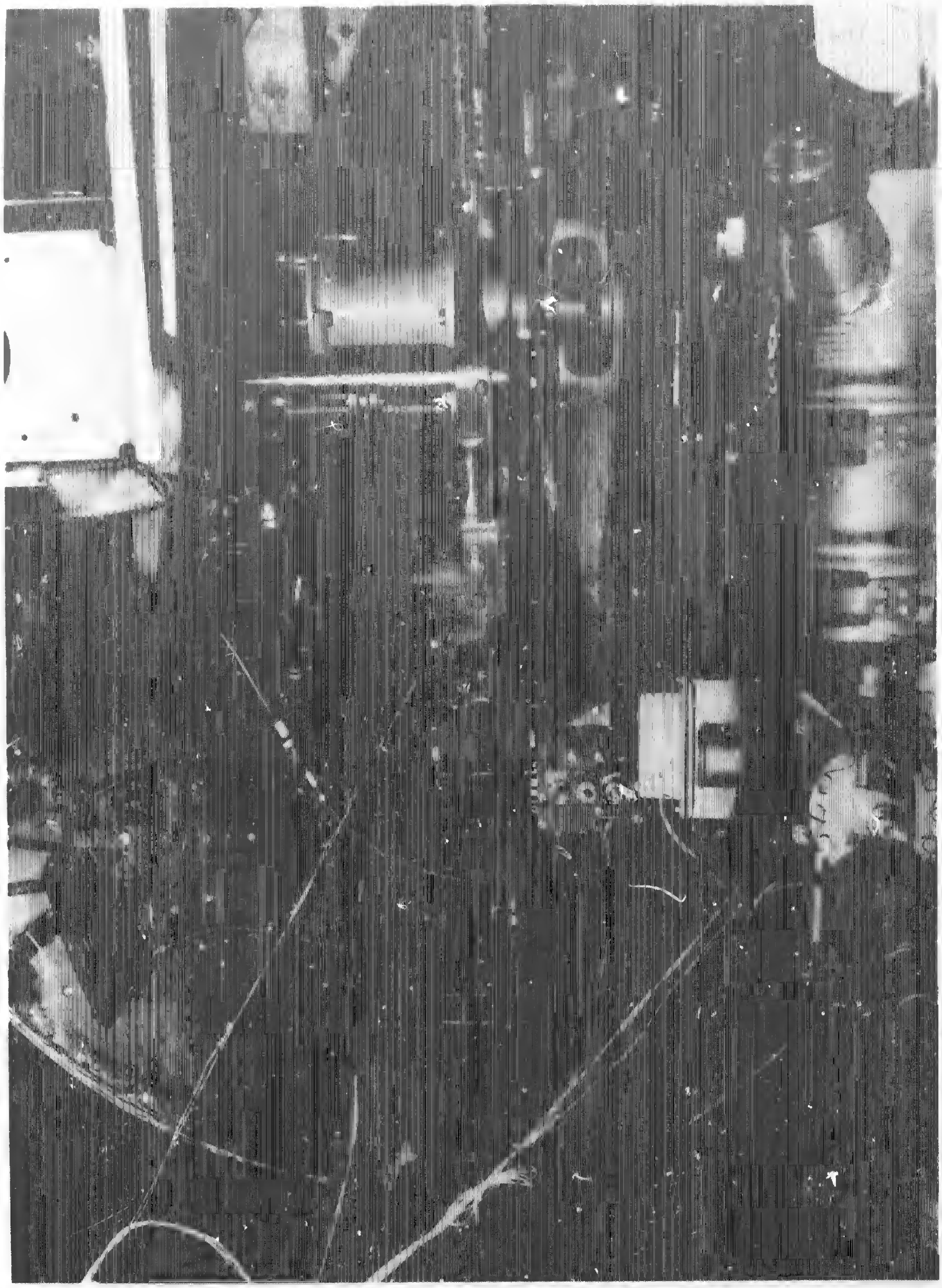


Figure 22. Clevis Plate and Torque Tube Assembly

The following test procedure was used during the first test flight.

(a) The aircraft was stabilized at 250 knots indicated air speed.

(b) The Fly-By-Wire system was engaged and qualitatively evaluated in turns, climbs, and descents with the load factors less than the "g" limiter kickout.

(c) The aircraft was accelerated to $\pm$ 50 knots from trim speed with Fly-By-Wire engaged, and the elevator effectiveness was evaluated. Conversion to and from the test system was evaluated at $\pm$ 25 knots and $\pm$ 50 knots from trim speed.

(d) The "g" limiter reversion point was evaluated by flying a level turn with decreasing radius until the + "g" reversion limit or 60 percent of maximum bank was obtained. For the "g" reversion limit, a pushover from a climbing flight path was used.

(e) The response of the Fly-By-Wire system with the autopilot engaged was evaluated.

(f) The Fly-By-Wire system was evaluated in an approach to a 1 "g" stall.

(g) The test system was evaluated at Mach 0.83.

The preceding test procedure in some or all parts was repeated during subsequent test flights. During the last several test flights, two flat and level runs at 5000 ft. and 400 knots of indicated air speed were run. During these runs, elevator pulses were applied to the aircraft, and the acceleration and pitch rate transient response were recorded.

Instrumentation used during the test flights was selective recording of the following information on a Honeywell

Visicorder mounted in the nose of the aircraft.

1. Control column position
2. Elevator surface position
3. Solenoid valve No. 1 current
4. Solenoid valve No. 2 current
5. Solenoid valve No. 3 current
6. Solenoid valve No. 4 current
7. Pitch rate
8. Forward acceleromter output
9. Aft accelerometer output
10. Servovalve current
11. Hydraulic supply pressure

TEST RESULTS AND ANALYSIS

General

The results of the B-47 Phase 1 flight test investigation were primarily qualitative. However, data were recorded both for verification of proper functional operation of the test system and to obtain pitch axis pulse response data to be used for analog simulation for further phases of the B-47 Fly-By-Wire program.

The flight test phase of the evaluation was successful. Over 40 hr of flight time by four different test pilots were accumulated on the Fly-By-Wire system during the period from 15 December 67 to 15 August 68. Although the majority of the evaluation was done at altitudes above 5000 ft. "touch and go" landings were carried out by the test pilots during the test period. The principal difference between the Fly-By-Wire control system and the normal elevator control was in the area of dynamic response, especially at higher airspeeds. The cable compliance effect was eliminated with the Fly-By-Wire system. This was observed in the data recorded during high speed, low altitude pulse test runs.

During the entire period of the test flight program, there were three problems which occurred in flight. The first problem occurred on the first test flight. The wiring polarity of the tail accelerometer had been reversed when it was installed in the test aircraft. This prevented the "g" limiter from functioning properly. The second in-flight problem occurred when the aircraft wiring connecting the Fly-By-Wire control panel to the switching block became an open circuit. The cause of this failure was aircraft structural flexing which opened the terminal strip connections in one of the permanently installed aircraft junction boxes. The third in-flight problem occurred during one takeoff when flexing of the aircraft structure created a tension load in the cable connecting the manual reversion lever to the switching block. This loading shifted the manual reversion spool to an intermediate position which removed hydraulic supply pressure from the normal actuator.

Problem 1 was solved by correcting the wiring on the "g" limiter accelerometer. Problem 2 was corrected by making new terminal connections for the open wiring. Problem 3 was corrected by installing a cable tensioning system which pulled the cable taut to the tail end of the aircraft. Some slack was provided in the tail section to prevent actuation of the manual reversion spool due to structural flexing. Furthermore, the manual reversion spool was reground to a lap condition which allowed the reversion spool to interconnect the Fly-By-Wire and normal actuator pressure lines in its intermediate position. Although the effect of this changed lap condition with Fly-By-Wire off was to allow connection of the elevator hydraulic supply pressure to its return (if the manual reversion lever was held between its two operating positions), this condition was felt to be more desirable than the original lap condition, where it was possible to turn off the supply pressure to both actuators at the same time.

One additional problem was encountered on the ground during the flight test program. While the test aircraft was in the hanger for aircraft maintenance, it was determined that the bypass valve (obtained from a normal B-47 elevator actuator) in the Fly-By-Wire actuator was operating intermittently. The cause of the intermittent operation was isolated to the amount of clamping force used to hold the sleeve of the bypass valve in position in the Fly-By-Wire actuator body. Although bypass valve housing dimensions identical to the normal elevator actuator had been used in fabricating Fly-By-Wire actuator, the end-plate used in the Fly-By-Wire actuator was less flexible than that of the normal elevator actuator. This caused the clamping action of the Fly-By-Wire end-plate to distort the bypass valve sleeve slightly, causing the bypass spool to stick. This problem was corrected by decreasing the clamping interference fit to the minimum which would prevent the bypass valve from shifting over the operating temperature range of the Fly-By-Wire actuator.

It is important to note that none of the problems that did occur during the evaluation test was with the primary Fly-By-Wire control circuit itself.

Specific Test Results and Analysis

1. Handling Qualities

The handling qualities of the Fly-By-Wire system in the area of precision and dynamic response were better than the normal elevator control system. The improvement in dynamic response can be readily seen by comparing Fig. 23 and Fig. 24 These two figures are tracings of flight test data recorded on the Visicorder during a 400-knot, 5000-ft altitude test flight run towards the end of the Phase 1 flight test program. Fig. 23 shows the effect of a control column motion pulse on elevator motion, nose acceleration and pitch rate with the normal elevator control system operating. Fig. 24 shows the effect of a similar pulse with the Fly-By-Wire system operating. Note that the input pulse for both figures is almost identical. However, the amplitude and rise time of the elevator motion were quite different. With the Fly-By-Wire actuator, the elevator motion had a greater amplitude and a more rapid rise time. The effect of this on the aircraft motion was to give a pitch rate pulse of less rise time and greater amplitude. The higher frequency content of the Fly-By-Wire pulse excited the structural modes of the aircraft more than the normal elevator pulse. This effect is directly observed in the pulse shape of the nose acceleration. The reason for the pulse response difference was that with the normal elevator system there is motion loss introduced by the control cables. The control cables act as a spring in connecting the control column to the elevator actuator. The loss is directly influenced by the "Q" spring system loading the control cables. At

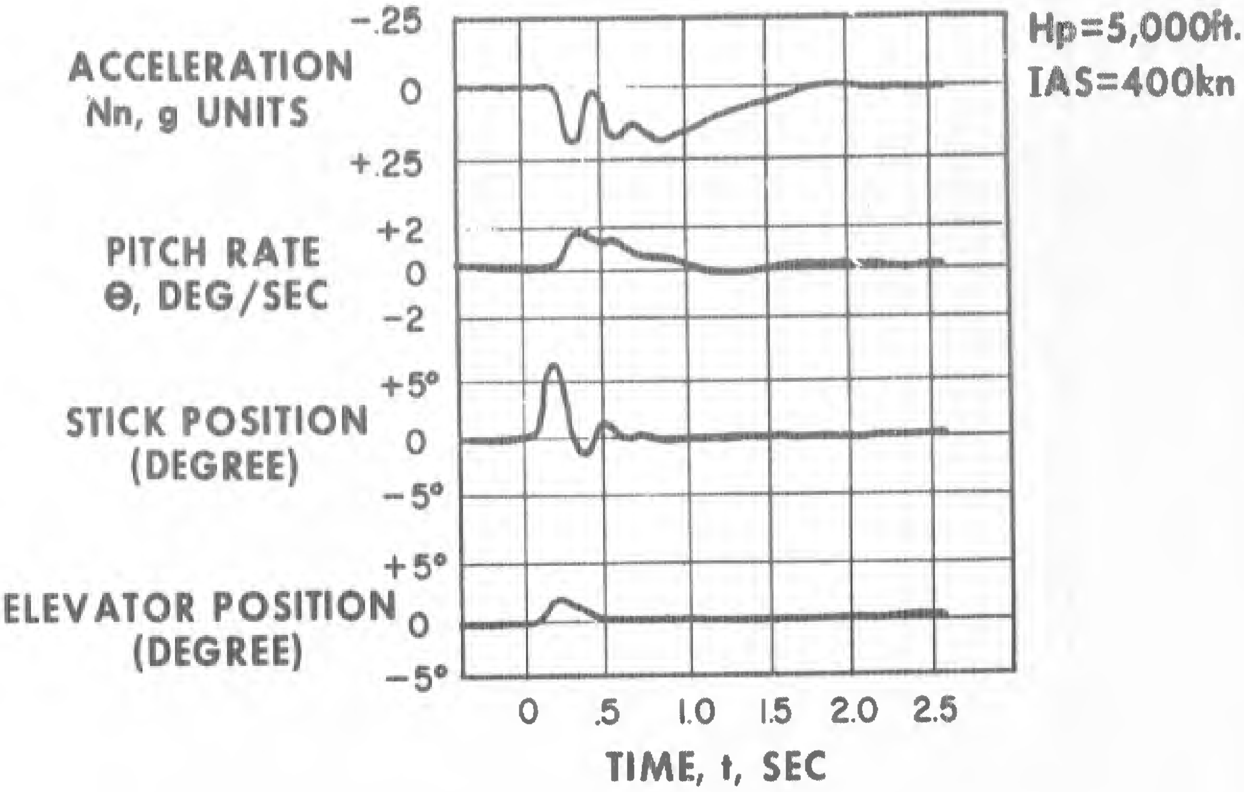


Figure 23 Pulse Response in Normal Mode

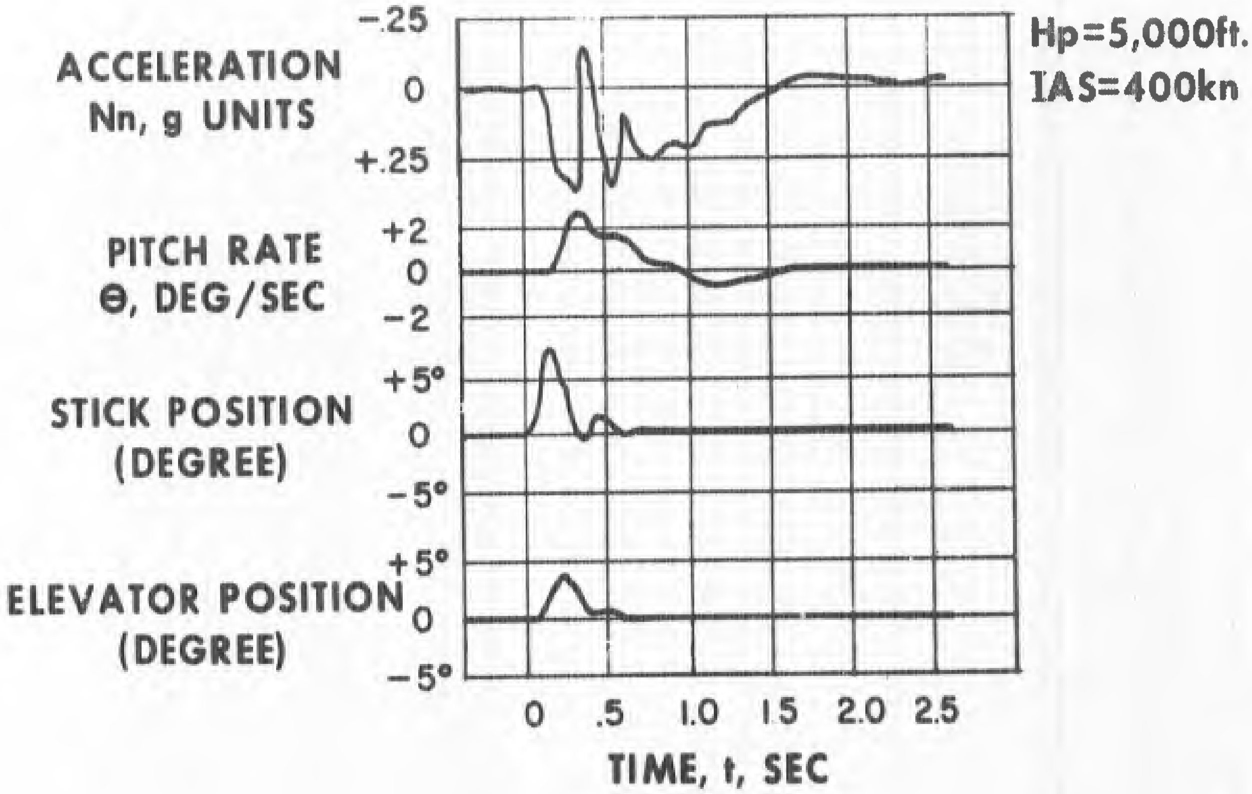


Figure 24 Pulse Reponse in Fly-By-Wire Mode

low airspeed conditions, the cable compliance acts as a slight time delay only, with the steady-state amplitude of motion at the actuator end of the cable being equal to the motion at the column end. At high air speed conditions, the "Q" spring system loads the actuator end of the cable, requiring a force from the cable proportional to cable movement. This acts to attenuate the control column motion effect, since part of the control column motion is used to stretch the control cable and part to move the control valve. In addition, the cable compliance filters the control column motion on a dynamic basis.

The engage bump obtained during switching to the Fly-By-Wire system when flying in an "out-of-trim" condition was noticeable. The engage-disengage bump on the ground and when in a trim condition was barely detectable. The reason for the out-of-trim engage bump was the "Q" spring system stretching the control cables. Thus the control column had to be moved to compensate for the cable stretch. This was effectively a change of the control column null position for the normal elevator control system. This movement of the control column for out-of-trim conditions meant that when switching into the Fly-By-Wire system, an input command voltage proportional to the "null" position change of the normal elevator control system was present, causing the engage bump.

2. G Limiter Reversion

The "g" limiter reversion had been set at $\pm$ 0.5 g's in the laboratory. This was verified in flight with the positive incremental "g's" occurring at $\pm$ 0.52 observed g's and the negative reversion unit occurring at -0.5 observed g's.

3. Autopilot Operation

The response of the Fly-By-Wire system with the autopilot on was evaluated during only the first several test flights. With the Fly-By-Wire system and autopilot engaged, a limit cycle condition occurred. This was due to the additional phase lag introduced by the autopilot system when it was driving the entire elevator cable system in order to create input signals to the Fly-By-Wire actuator. This differed from the normal autopilot operation where the autopilot sends control inputs to the normal actuator at the tail end of the aircraft. The limit cycle that occurred was of a limited amplitude, and the frequency was approximately 1.5 H_z. The limit cycle was easily damped out by the pilot restraining the control column. The limit cycle occurred intermittently with the onset usually occurring when operating in a slight out-of-trim condition. It was decided not to operate the pitch axis autopilot and Fly-By-Wire control system at the same time.

3. Mach 0.83 and 1.0 "g" Stall Evaluation

In evaluating the Fly-By-Wire at an approach to 1.0 "g" stall and at Mach 0.83, there was little difference between the normal elevator control system and the Fly-By-Wire. The only noticeable change was the difference in pulse response at the Mach 0.83 condition.

4. Pilot Comments

The following two pages comprise a letter commentary on the Fly-By-Wire control system Phase 1 operation by Major Fredericks, Test Director and Test Pilot for the Phase 1 flight tests.

DEPARTMENT OF THE AIR FORCE
HEADQUARTERS AERONAUTICAL SYSTEMS DIVISION (AFSC)
WRIGHT-PATTERSON AIR FORCE BASE, OHIO 45433

REPLY TO ATTN OF: ASTFB (Maj Fredricks/73112) 30 Oct 1968

SUBJECT: Pilot Comments on Phase I B-47 Fly-by-Wire System Tests

TO: FDCL (Maj Sutherland)

1. The following pilot comments are offered for inclusion in the "Fly-by-Wire Flight Control System, Phase I" Test Report:

a. Handling qualities. The only noticeable improvement in handling qualities offered by the test system was a decrease in backlash due to cable stretch. The fact that the test system did in fact eliminate a measurable amount of cable stretch was demonstrated by trimming the aircraft up with the test system engaged, changing the airspeed without retrimming, and observing the trim change caused by reverting to the normal aircraft elevator system. When the aircraft was slowed from the trim condition and reversion accomplished, a noticeable pitch down occurred. This can be explained by the fact that the test system by-passed the control cables connecting the control column to the q-spring and elevator actuator; it thus eliminated the lost motion caused by the slight amount of stretch experienced by the control cables in overcoming the resistive force of the q-spring. When reversion to the normal system was made at a speed below trim speed, the lost motion due to cable stretch was reintroduced into the control circuit, the result being a slight decrease in "up" elevator and a consequent pitch down. Elimination of backlash due to cable stretch made for much more positive, less sloppy, and apparently more responsive elevator control. This was most apparent in high speed, low altitude flight.

b. System reliability. The system to be tested consisted of the simple open-loop electrical connection of the control column and the elevator actuator by means of linear transducers and electrical wiring. This system operated in parallel with the normal aircraft elevator control system, and various methods, both mechanical and electrical, manual and automatic, were provided for switching between the normal and test systems. Although some difficulties occurred due to malfunctioning of these switching mechanisms, no malfunctions in the basic "fly-by-wire" system occurred during the entire period of testing. Based on his experience with this program, the undersigned has a high degree of confidence in electrical flight control systems, but feels that some improvements could be made in the design, fabrication and installation of the switching mechanisms that are provided for redundancy.

2. Please contact the undersigned if any additional comments are needed.

Barron Fredricks

BARRON FREDRICKS, III, Major, USAF
Test Director/Project Pilot
Bomber Operations Branch

B-47 FLY-BY-WIRE PHASE II

The primary objective of Phase II (currently in flight test evaluation), of the B-47 Fly-By-Wire investigation is to demonstrate the advantages fo closed-loop Fly-By-Wire flight control systems on a large flexible aircraft. These advantages are in terms of improved control performance and handling qualities. Flight test data and pilot comments obtained on the Phase II system indicate that the program will contribute significantly to the general acceptance and use of Fly-By-Wire in future military aircraft.

Phase II of the Fly-By-Wire investigation is based on using a side stick controller, pitch rate and nose acceleration feedback, and electrical roll axis control in conjuction with the Phase I system. The side stick mounts on the left side of the pilot's or copilot's seat and is of the stiff displacement category. The pitch axis command inputs are fed into essentially the Phase I pitch axis system. Aircraft pitch rate and nose acceleration electrical feedback is used in the pitch axis. The roll axis electrical inputs are fed into the roll axis autopilot roll attitude reference gyro disabled. Roll rate is used for the roll axis electrical feedback.

The particular blend of pitch rate and nose acceleration feedback is called C* feedback and is described in the material following.

Figure 25 shows the principal components used in the Phase II pitch axis

Figure 26 describes the basis for the C* control criteria.

Figure 27 describes the design procedure for the C* feedback blend.

Figure 28 is the block diagram schematic of the pitch axis system.

Figure 29 is a picture of the actual hardware components added to the Phase I mechanization in order to accomplish the Phase II system.

Figure 30 is a block diagram schematic of the roll axis system.

As of 18 April 1969, 15 different evaluation pilots, including a general officer had flown the Phase II system. Thirty four flight hours had been accumulated on the Phase II system. Evaluation pilot comments on the system were uniformly very favorable with particular comment on the ease and preciseness of control.

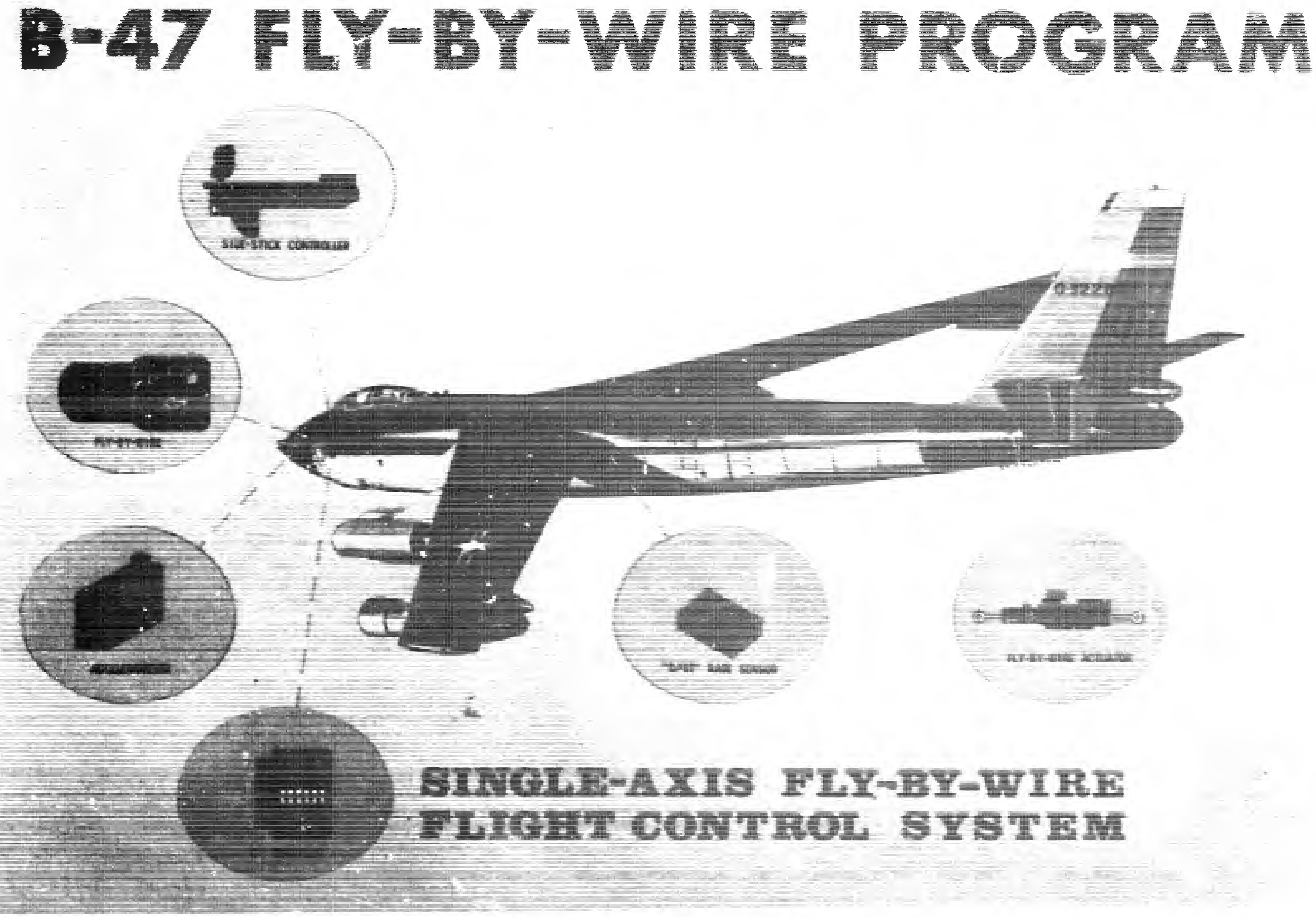


Figure 25-Phase II Pitch Axis Components

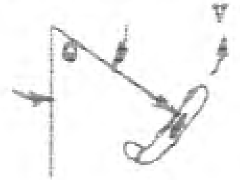
C* BASIS

PILOT RESPONDS TO A BLEND OF PITCH RATE AND NORMAL ACCELERATION, WITH THE BLEND RATIO VARYING ACCORDING TO NORMAL VARIATIONS IN AIRCRAFT RESPONSE.

C* Equation

$$\boxed{\begin{array}{ll} C^* = K_1 N_z + K_2 \dot{\theta} & \text{(Pilot at C. G.)} \\ C^* = K_1 N_z + K_3 \ddot{\theta} + K_2 \ddot{\theta} & \text{(Pilot not at C. G.)} \end{array}}$$

NORMAL AIRCRAFT RESPONSE



$$\dot{\theta} = \frac{d\theta}{dt} = \omega$$

$$V = R\,\dot{\theta}$$

$$N_z = R\omega^2$$

Therefore;

$$N_z = R(\dot{\theta})(\dot{\theta}) = R\left(\frac{V}{R}\right)\dot{\theta}$$

$$\boxed{N_z = V\,\dot{\theta}}$$

Figure 26 - C* Control Criteria

C* DESIGN PROCEDURE

1. SELECT CROSSOVER VELOCITY

(VELOCITY AT WHICH $\dot{\theta}$ AND N_z CUES HAVE THE SAME MAGNITUDE)

2. AT CROSSOVER

$$K_1 N_z = K_2 \dot{\theta}$$

$$K_1 N_z = \frac{N_z \,(\,32.2\,)}{V_{co}}$$

LETTING $K_1 = 1$

$$K_2 = \frac{V_{co}}{32.2}$$

WHERE $\dot{\theta} = \dfrac{N_z\,(\,g\text{'s}\,)(\,32.2\,)}{V_{co}\,(\,FT\,/\,SEC\,)}$

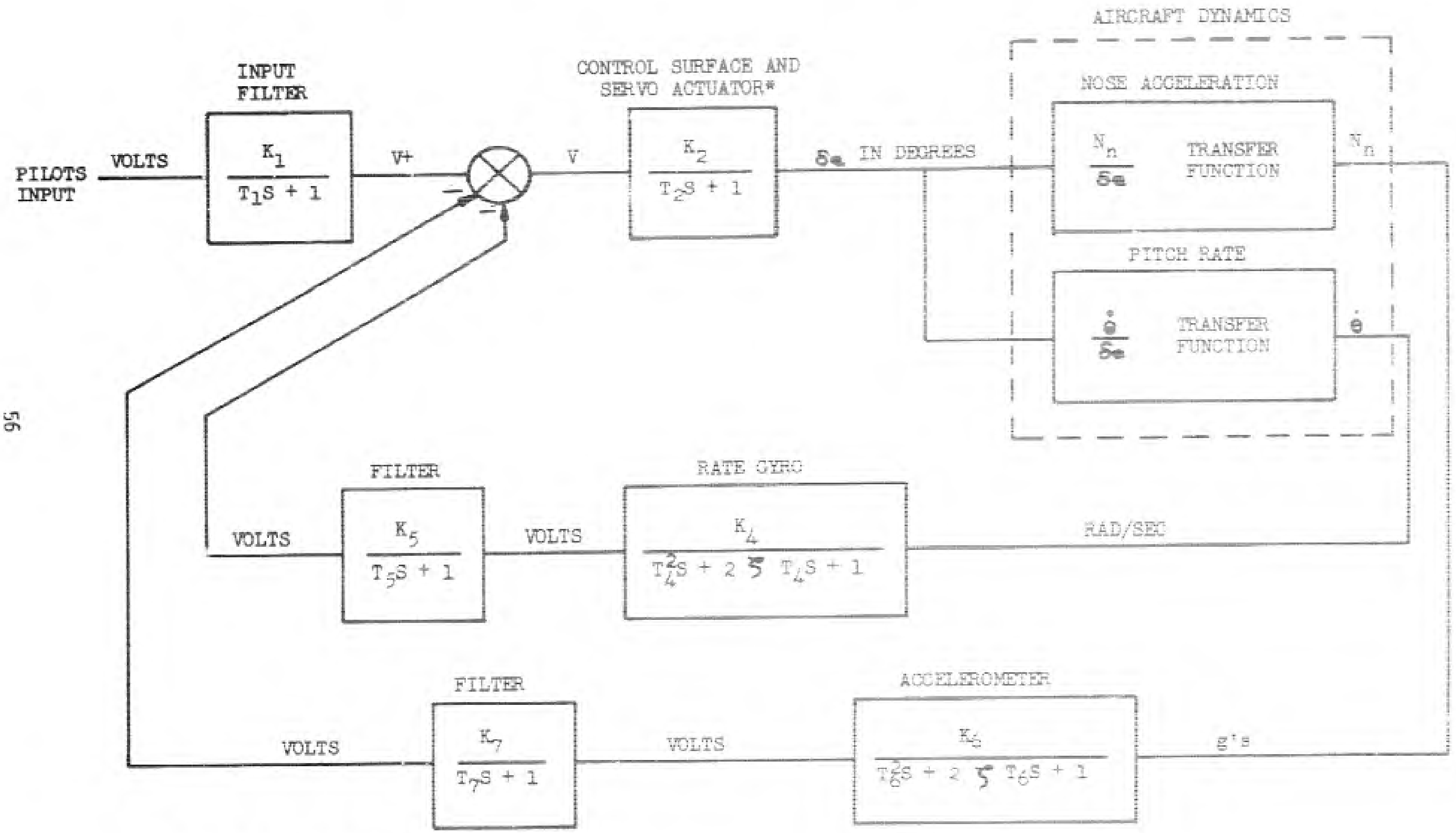
3. EXAMPLE

FOR $V_{co} = 400$ FT/ SEC

$$K_2 = \frac{400}{32.2} = 12.4$$

Figure 27 - Design Procedure for C* Blend

B-47 PHASE II C* CONTROL SYSTEM PITCH AXIS BLOCK DIAGRAM



* APPROXIMATION FOR SMALL ANGLES

Figure 28 - Pitch Axis Block Diagram (II)

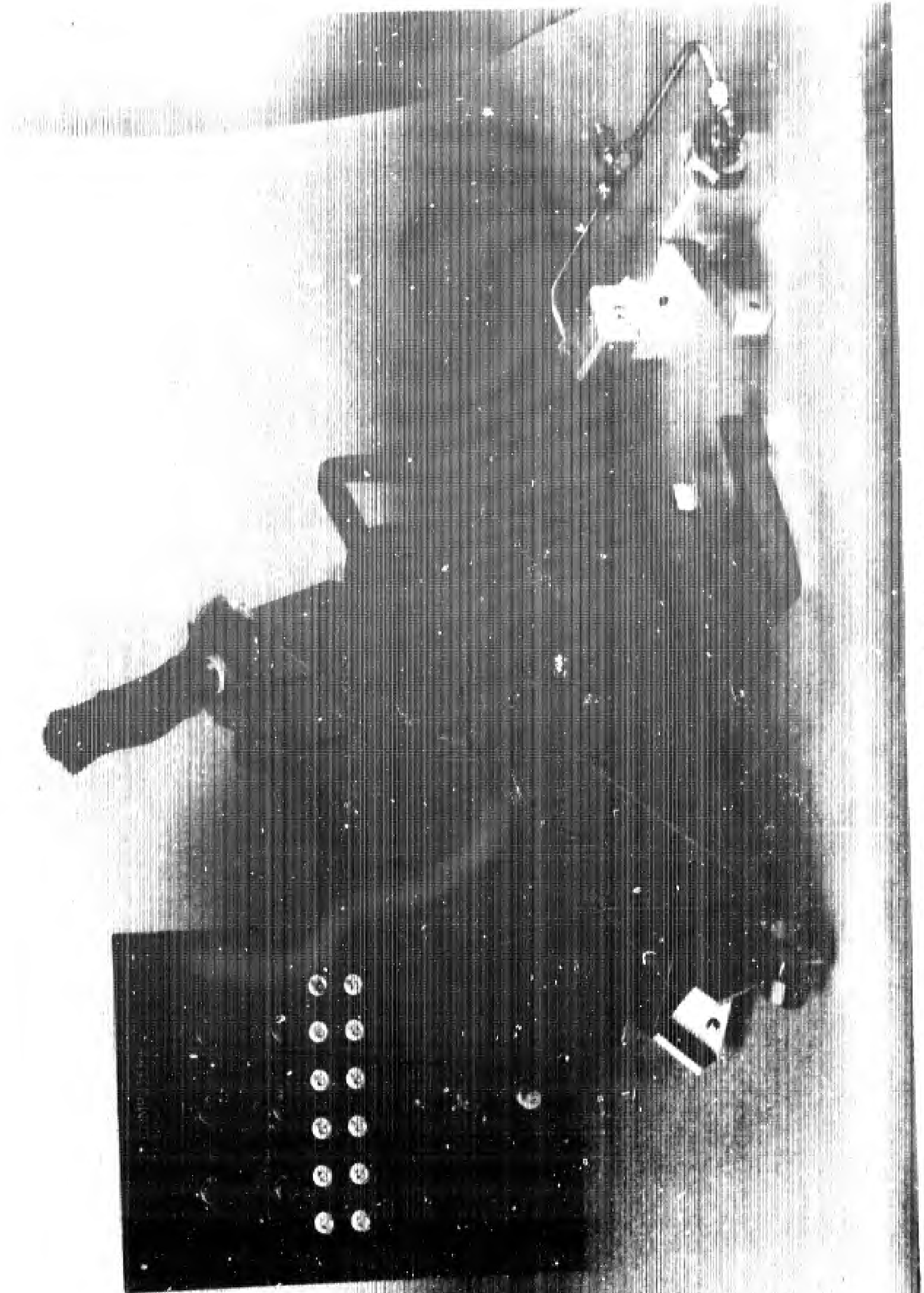
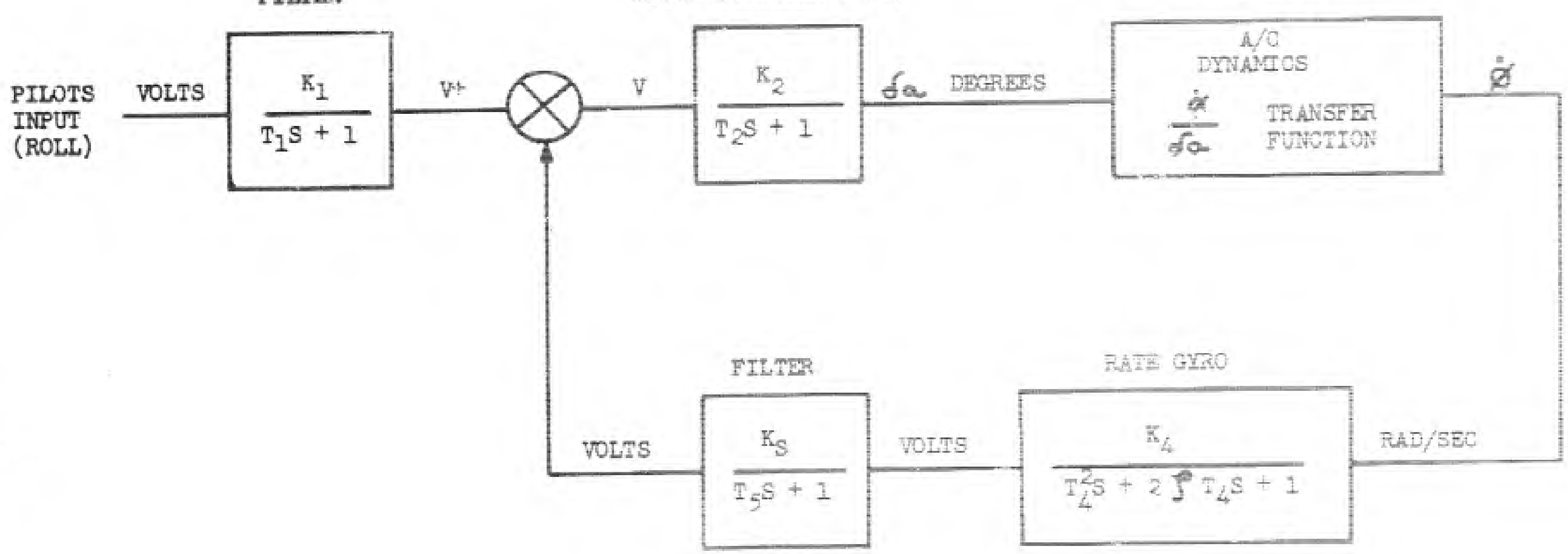


Figure 29 - Actual Flight Hardware

B-47 PHASE II CONTROL SYSTEM ROLL AXIS BLOCK DIAGRAM



* APPROXIMATION FOR SMALL ANGLES

(1) CONTROLLED BY AUTOPILOT SERVO SYSTEM

Figure 30 - Roll Axis Block Diagram (II)

B-47 FLY-BY-WIRE PHASE III

The Phase III portion of the B-47 Fly-By-Wire investigation is to evaluate a quadruply redundant electrohydraulic actuator in the test aircraft. The actuator is being provided by Hydraulic Research and Manufacturing Company and has been constructed and laboratory tested in accordance with a specification provided by AFFDL. The redundant actuator replaces the non-redundant pitch axis actuator used in the Phase II mechanization.

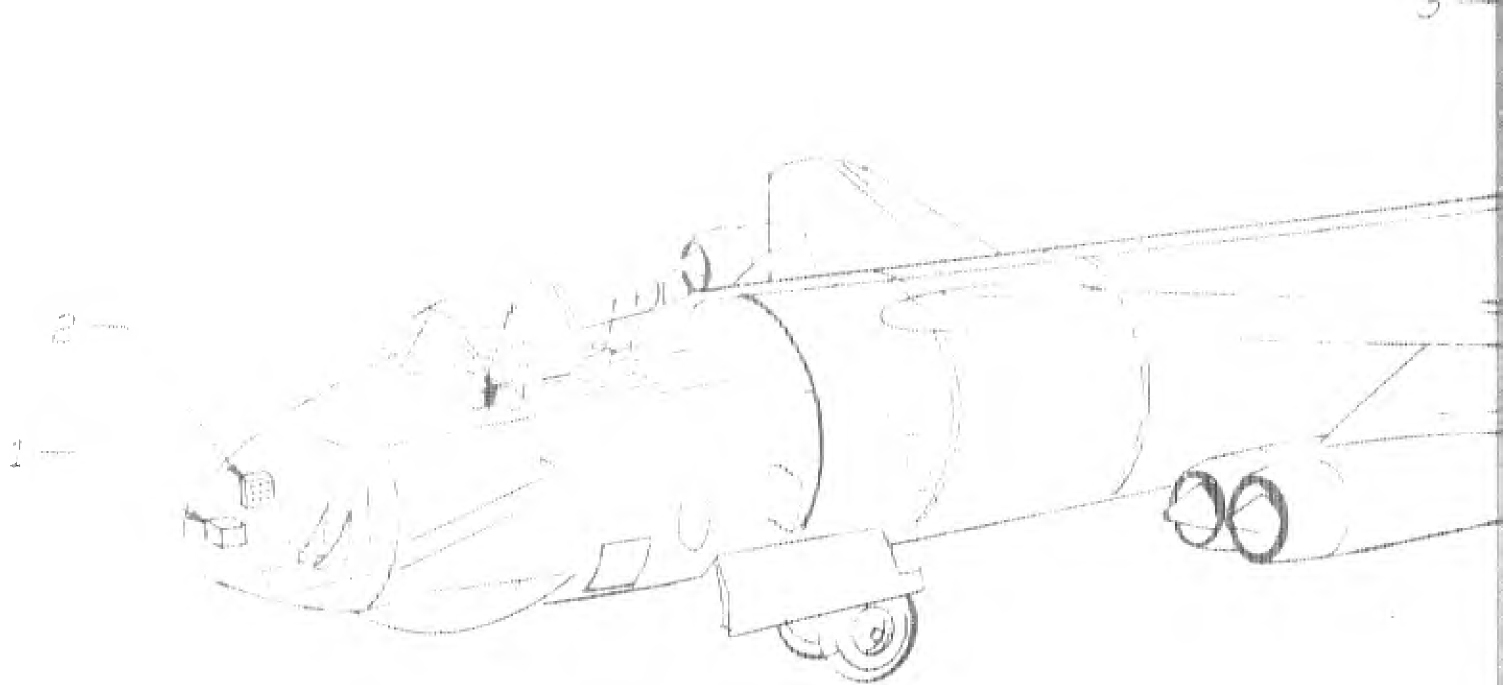
The Phase III actuator is powered by four independent, electrically driven, hydraulic power supplies. The hydraulic supply modules will be mounted in the tail section of the test aircraft. The redundant actuator uses hydraulic logic and monitoring techniques which have been designed to give the unit a dual fail-operate, fail-neutral capability. The actuator has the ability of withstanding 2 similar or dissimilar channel failures (hydraulic supply, internal or electrical input) with no performance change. When a third channel failure occurs, the actuator reverts to a damped "free" position.

Four servo amplifiers and a failure mode panel in the navigator's and pilot's station are added to the Phase II components to allow driving and monitoring the redundant actuator. The failure mode panels have provision for injecting failures into the redundant actuator during flight in order to measure the failure mode characteristics.

Figure 31 shows the physical location of the additional components for Phase III.

Figure 32 is the pitch axis block diagram with the quadruply redundant Hydrologic actuator incorporated.

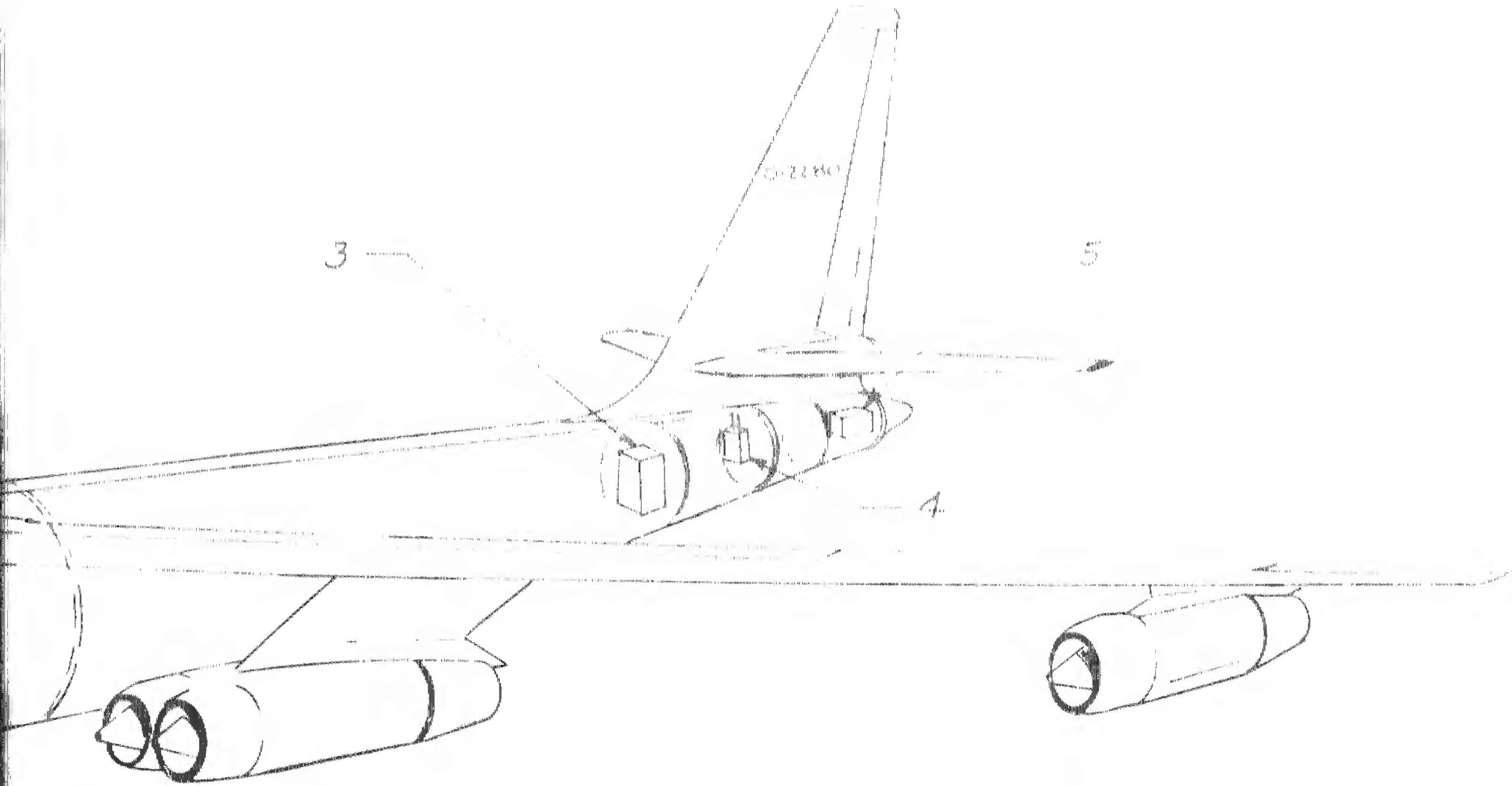
Figures 33,34&35 are pictures of the pumping modules, Hydrologic actuator and switching block, and servo amplifiers, respectively.



1. SERVO AMPLIFIERS.
2. FAILURE MODE CONTROL PANEL.
3. 3000 PSI PUMP MODULE (4 REQD).
4. ACTUATOR.
5. HYDROLOGIC SWITCHING BLOCK.

NOTE: PHASE III EQUIPMENT IS ADDED TO PHASE II MODS.

A



EQ'D).

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B-47 FLY-BY-WIRE - PHASE III

REDUNDANT HYDROLOGIC

4 CHANNEL ACTUATOR

Figure 31 - Phase III Component Location

1 B

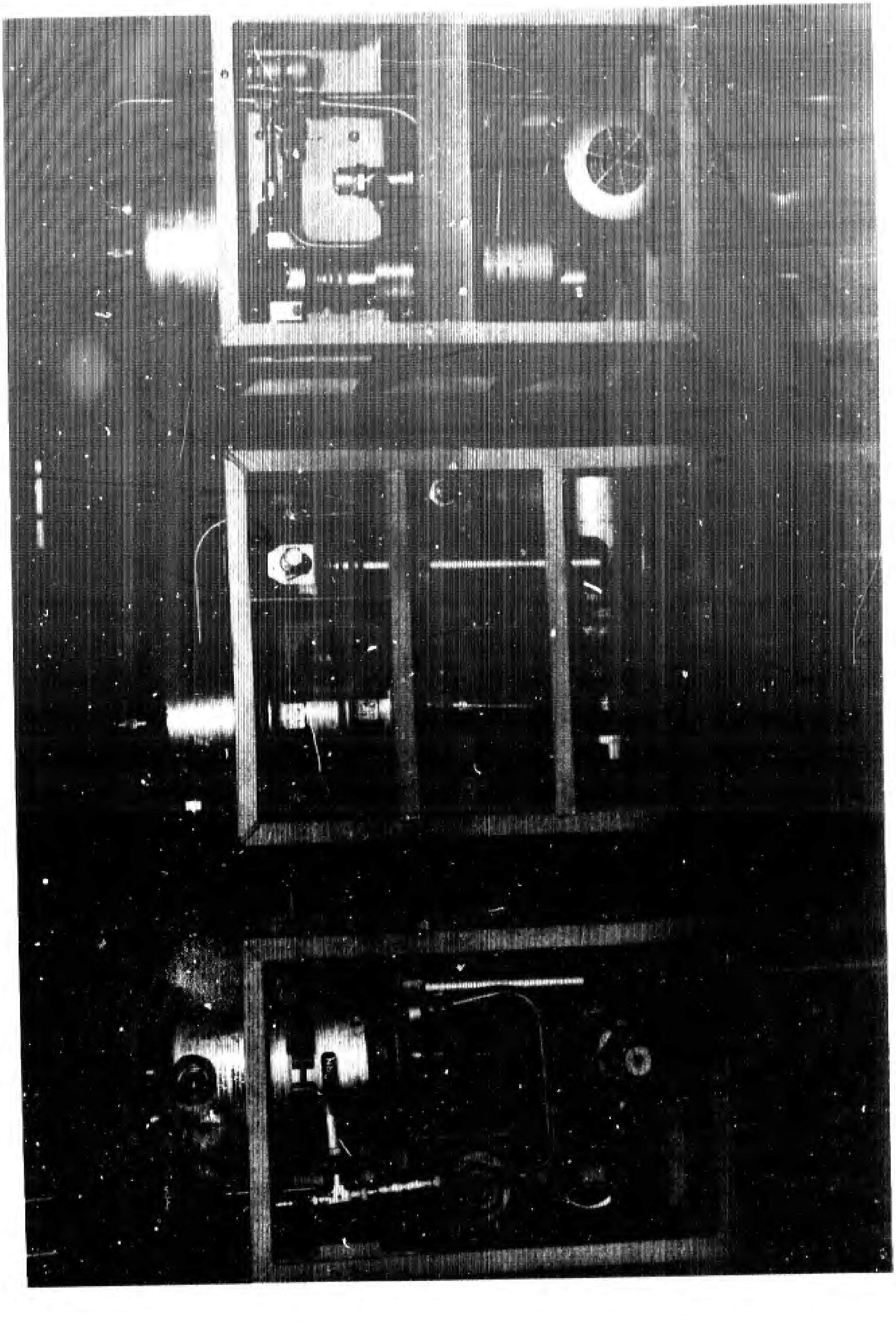
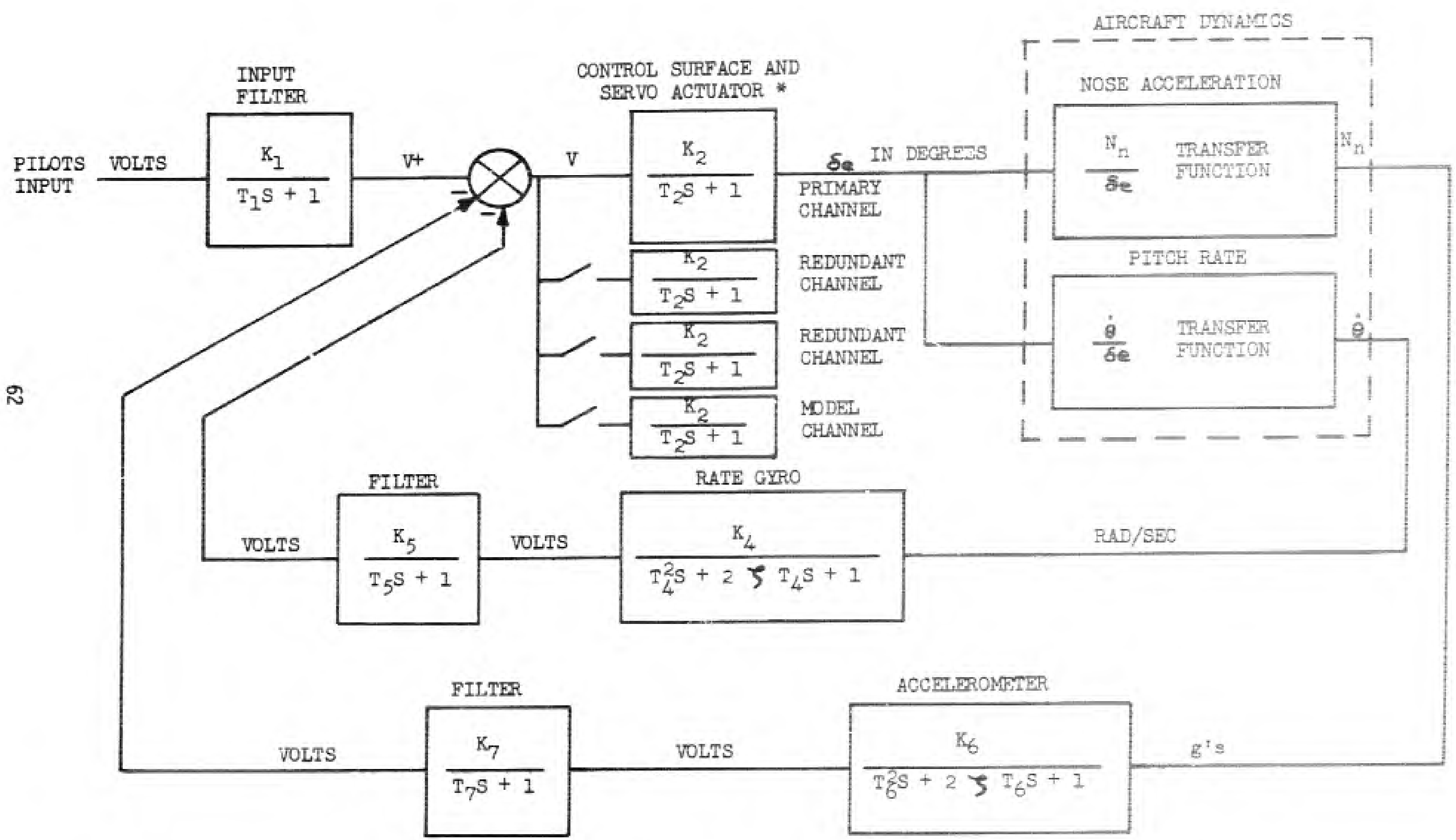


Figure 33 - Pumping Module

B-47 PHASE III C* CONTROL SYSTEM PITCH AXIS BLOCK DIAGRAM
WITH REDUNDANT HYDROLOGIC 4 CHANNEL ACTUATOR



* APPROXIMATION FOR SMALL ANGLES

Figure 32 - Pitch Axis Block Diagram (III)

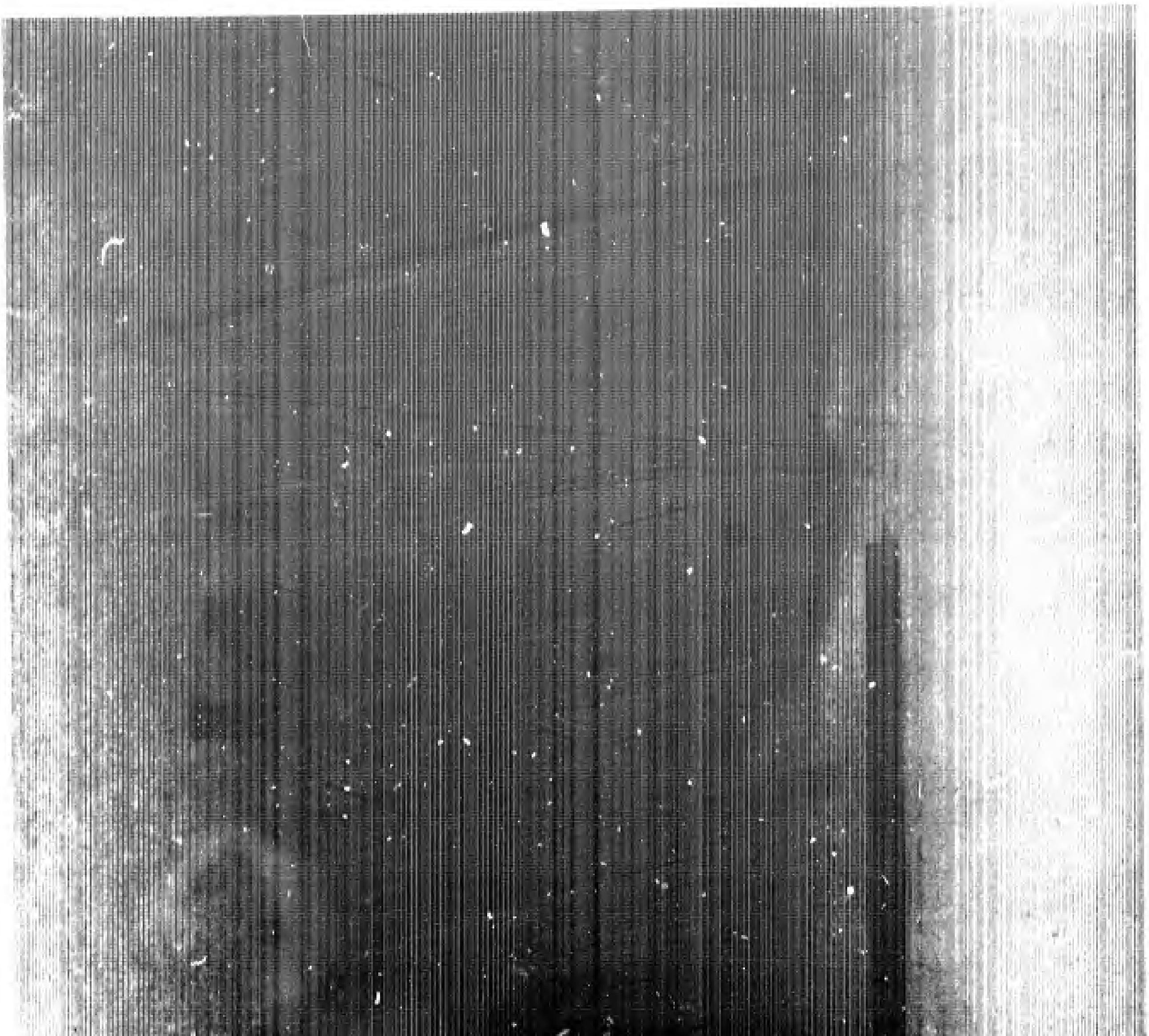


Figure 34 - Hydrologic Actuator

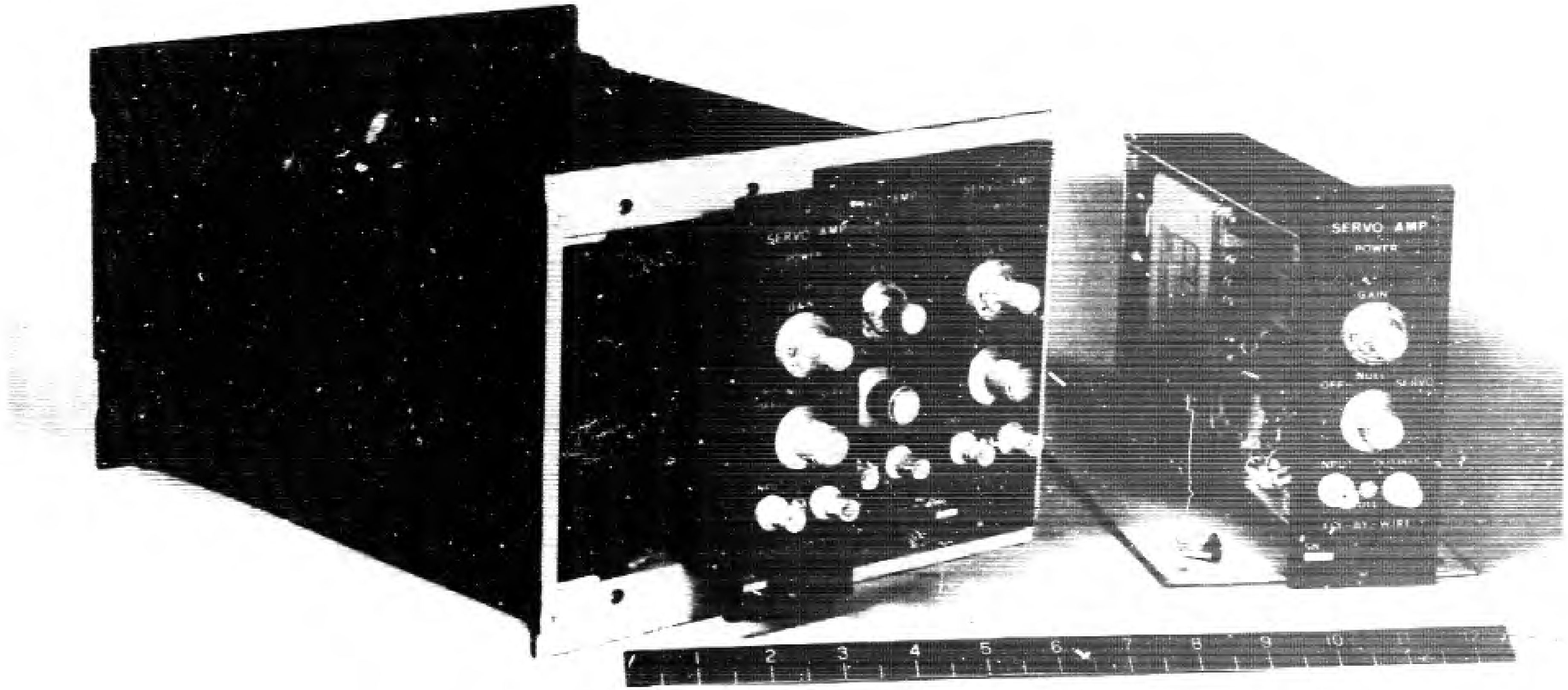


Figure 35 - Servo Amplifiers

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PHASE I of a 3 PHASE FLIGHT TEST PROGRAM

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FLIGHT CONTROL DIVISION (FDCL)
WRIGHT - PATTERSON AIR FORCE BASE, OHIO

13. ABSTRACT
This report is a description of the research investigation on a fly-by-wire control system applied to a B-47 test aircraft. The investigation is primarily confined to the pitch axis of the test aircraft and is divided into three phases.

Phase I system operation is based on using an electrical non-redundant primary flight control system with control inputs being generated by the normal control column motion.

Phase II system operation is based upon adding a side stick controller, pitch rate and nose acceleration feedback, and electrical roll axis control to the Phase I system.

Phase III incorporates a 4 channel redundant actuator with hydraulic logic into the Phase II system.

Flight test of the Phase I system has been completed with over 40 flight hours, including touch and go landings being accumulated on the fly-by-wire system by four different test pilots. Pilot comments on the system operation and performance were favorable. This report documents the complete Phase I system and test results and briefly describes the Phase II and III systems.

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Security Classification

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
FLIGHT CONTROL SYSTEMS						
ELECTRICAL FLIGHT CONTROL						
FLY-BY-WIRE						
REDUNDANT SYSTEMS						
ACTUATORS						
CONTROLLERS						
AIRCRAFT EQUIPMENT						
C* CRITERIA						
STABILITY & CONTROL AUGMENTATION						
FAILURE DETECTION SYSTEMS						
B-47 FLIGHT TESTS						
HYDROLOGIC						

INSTRUCTIONS

1. ORIGINATING ACTIVITY: Enter the name and address of the contractor, subcontractor, grantee, Department of Defense activity or other organization *(corporate author)* issuing the report.

2a. REPORT SECURITY CLASSIFICATION: Enter the overall security classification of the report. Indicate whether "Restricted Data" is included. Marking is to be in accordance with appropriate security regulations.

2b. GROUP: Automatic downgrading is specified in DoD Directive 5200.10 and Armed Forces Industrial Manual. Enter the group number. Also, when applicable, show that optional markings have been used for Group 3 and Group 4 as authorized.

3. REPORT TITLE: Enter the complete report title in all capital letters. Titles in all cases should be unclassified. If a meaningful title cannot be selected without classification, show title classification in all capitals in parenthesis immediately following the title.

4. DESCRIPTIVE NOTES: If appropriate, enter the type of report, e.g., interim, progress, summary, annual, or final. Give the inclusive dates when a specific reporting period is covered.

5. AUTHOR(S): Enter the name(s) of author(s) as shown on or in the report. Enter last name, first name, middle initial. If military, show rank and branch of service. The name of the principal author is an absolute minimum requirement.

6. REPORT DATE: Enter the date of the report as day, month, year; or month, year. If more than one date appears on the report, use date of publication.

7a. TOTAL NUMBER OF PAGES: The total page count should follow normal pagination procedures, i.e., enter the number of pages containing information.

7b. NUMBER OF REFERENCES: Enter the total number of references cited in the report.

8a. CONTRACT OR GRANT NUMBER: If appropriate, enter the applicable number of the contract or grant under which the report was written.

8b, 8c, & 8d. PROJECT NUMBER: Enter the appropriate military department identification, such as project number, subproject number, system numbers, task number, etc.

9a. ORIGINATOR'S REPORT NUMBER(S): Enter the official report number by which the document will be identified and controlled by the originating activity. This number must be unique to this report.

9b. OTHER REPORT NUMBER(S): If the report has been assigned any other report numbers *(either by the originator or by the sponsor)*, also enter this number(s).

10. AVAILABILITY/LIMITATION NOTICES: Enter any limitations on further dissemination of the report, other than those imposed by security classification, using standard statements such as:

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12. SPONSORING MILITARY ACTIVITY: Enter the name of the departmental project office or laboratory sponsoring *(paying for)* the research and development. Include address.

13. ABSTRACT: Enter an abstract giving a brief and factual summary of the document indicative of the report, even though it may also appear elsewhere in the body of the technical report. If additional space is required, a continuation sheet shall be attached.

It is highly desirable that the abstract of classified reports be unclassified. Each paragraph of the abstract shall end with an indication of the military security classification of the information in the paragraph, represented as *(TS)*, *(S)*, *(C)*, or *(U)*.

There is no limitation on the length of the abstract. However, the suggested length is from 150 to 225 words.

14. KEY WORDS: Key words are technically meaningful terms or short phrases that characterize a report and may be used as index entries for cataloging the report. Key words must be selected so that no security classification is required. Identifiers, such as equipment model designation, trade name, military project code name, geographic location, may be used as key words but will be followed by an indication of technical context. The assignment of links, rules, and weights is optional.