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Saturn IB/V - Instrument Unit

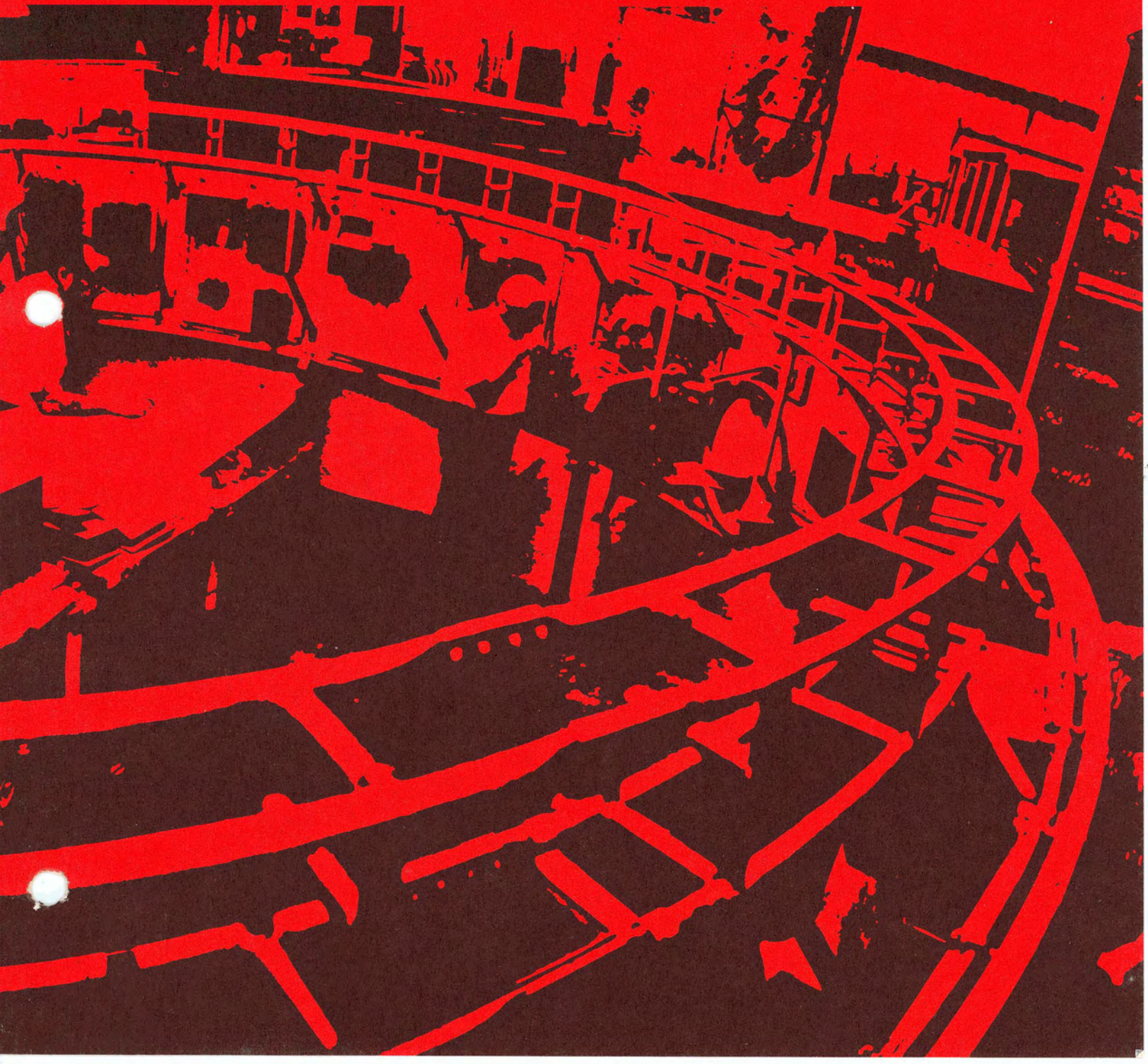
Technical Manual

MSFC No. III-5-510-19

IBM No. 66-966-0021

Flight Control Computer

(S-IU-204 and S-IU-205)



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TECHNICAL MANUAL

FLIGHT CONTROL COMPUTER

(200 Series Vehicles)

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RELATED DOCUMENTS

The following technical manuals have been published under NASA contract on the IU, its sub-systems, and components. Another listing, including technical manuals, handbooks, reports, etc., is located in Technical Manual IBM No. 66-966-0006.

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Title	Number		Title	Number	
	IBM	MSFC		IBM	MSFC
GENERAL			ENVIRONMENTAL CONTROL SUBSYSTEM		
Apollo-Saturn Emergency Detection System Description (AS-200 Series Vehicles)	66-966-0010	III-5-509-6	Environmental Control System Description	66-966-0009	III-5-509-5
Apollo-Saturn Emergency Detection System Description (AS-500 Series Vehicles)	66-966-0022	III-5-509-7	GUIDANCE AND CONTROL SUBSYSTEM		
Astrionics System Handbook	None	None	Control Accelerometer	66-966-0012	III-5-510-12
Component Handling Equipment, Description and Operating Instructions	66-966-0005	III-5-510-10	Control-EDS Rate Gyro Package	66-966-0013	III-5-510-13
EDS Q-Ball Sensor Assembly (50Z12580-1)	66-966-0027	III-5-510-22	Control Signal Processor	66-966-0014	III-5-510-14
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Structure Repair Manual	65-966-0011H	III-5-510-1	Flight Control Computer (500 Series Vehicles)	66-966-0025	III-5-510-21
System Test Reference Manual Volume I: Control System Volume II: Guidance System Volume III: ST-124M-3 Inertial Stabilizer System	66-966-0004	III-5-509-3	Navigation, Guidance, and Control System Description	66-966-0003	III-5-509-2
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C-Band Radar Transponder Set SST-135C (S-IU-203)	66-966-0008	III-5-510-11	ELECTRICAL SUBSYSTEM		
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* These manuals are at the S-IU-201 level. No plans exist to update their contents to later vehicle configurations.

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SECTION I

INTRODUCTION AND DESCRIPTION

1-1 INTRODUCTION

This manual contains the principles of operation, and maintenance and repair instructions for the Flight Control Computers which are used in the Instrument Units of the AS-200 Series Saturn Launch Vehicles. The purpose of this manual is to provide a comprehensive understanding of the physical construction and detailed operation of the Flight Control Computer.

The description in Sections I through III is basic to all AS-200 Series Flight Control Computers. Any differences which exist between Flight Control Computers of different effectivities (AS-204 through AS-212) will be covered in the Vehicle Data Sheets in Section IV. The complete description of any particular Flight Control Computer will consist of the basic description in Sections I through III, as modified by the particular Vehicle Data Sheets in Section IV.

1-2 DESCRIPTION

The Flight Control Computer (see Figure 1-1) is used to control the flight path of the vehicle during lift-off and burn of the S-IB Stage, during burn of the S-IVB Stage, and during coast of the S-IVB Stage. During lift-off and flight, the Flight Control Computer receives input signals from various sensing devices within the launch vehicle. These signals include attitude error signals from the Launch Vehicle Digital Computer (LVDC), applied through the Launch Vehicle Data Adapter (LVDA); angular rate information from the Control-EDS Rate Gyro Package; lateral acceleration information from the Control Accelerometers; and timing and programming information supplied by the LVDC through the LVDA and the Instrument Unit (IU) Switch Selector.

The symbol $\Delta \phi$ is used throughout this manual to represent attitude error signals. The symbol $\Delta \phi$ is synonymous with the symbol ψ which is used in some related documents.

1-3 EQUIPMENT CHARACTERISTICS

Table 1-1 lists the overall electrical and physical characteristics of the Flight Control Computer. Table 1-2 lists the electrical and physical characteristics of the electronic subassemblies of the Flight Control Computer. The number of each type subassembly used in the Computer is also listed in this table. Typical electronic subassemblies that make up the Flight Control Computer are illustrated in Figure 1-2.

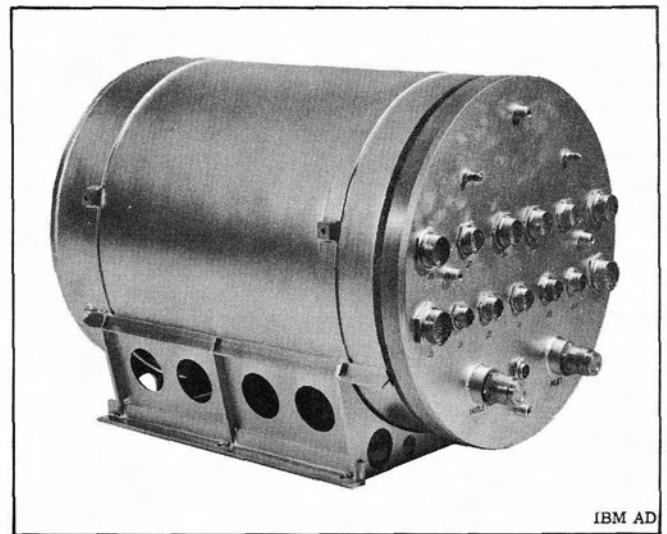


Figure 1-1. Flight Control Computer

Table 1-1. Flight Control Computer Leading Particulars

Characteristic	Specification
	<p style="text-align: center;">NOTE</p> <p>Overall dimensions are maximum and include the Flight Control Computer case and cradle. Measurements indicated are obtained when viewing the Flight Control Computer from the connector end.</p>
Overall Dimensions	
Height	21 inches
Width	20 inches
Depth	30 inches
Overall Weight	140 pounds maximum
Duty Cycle	100% continuous operation
Warmup Time	30 minutes
Input Power	22 to 32 Vdc, 320 watts maximum
Operating Temperature Range	-55 to +100 °C (-67 to +212 °F)
Acceleration	20 G's peak
Reliability Factor	0.999 for a Saturn mission consisting of (1) earth orbit insertion, (2) three earth orbits, (3) translunar injection, and (4) 2 hours of translunar coast
	<p style="text-align: center;">NOTE</p> <p>The reliability factor for the Flight Control Computer includes the possibility that a failure may occur in a redundant circuit.</p>
Pressurization	Gaseous nitrogen to 20 psia at +17 to +27 °C (+63 to +81°F)
	<p style="text-align: center;">NOTE</p> <p>The Flight Control Computer case assembly is constructed so that a pressure drop to less than 14.7 psia will not be encountered over a two-year period in a perfect vacuum environment.</p>
Cooling	Fluid coolant lines
Case	Spot welded and dip brazed aluminum alloy
Cradle	
Type	Dip brazed aluminum alloy
Lifting Lugs	4
Case Cover	
Type	Spot welded and dip brazed aluminum alloy
Lifting Lugs	4
Transducer	50 psia range, motional pick up (pressure) transducer. Mounted on the inside of the case cover
Timer	Elapsed time indicator mounted on the case cover (activated when 28 Vdc power is applied to the Flight Control Computer)

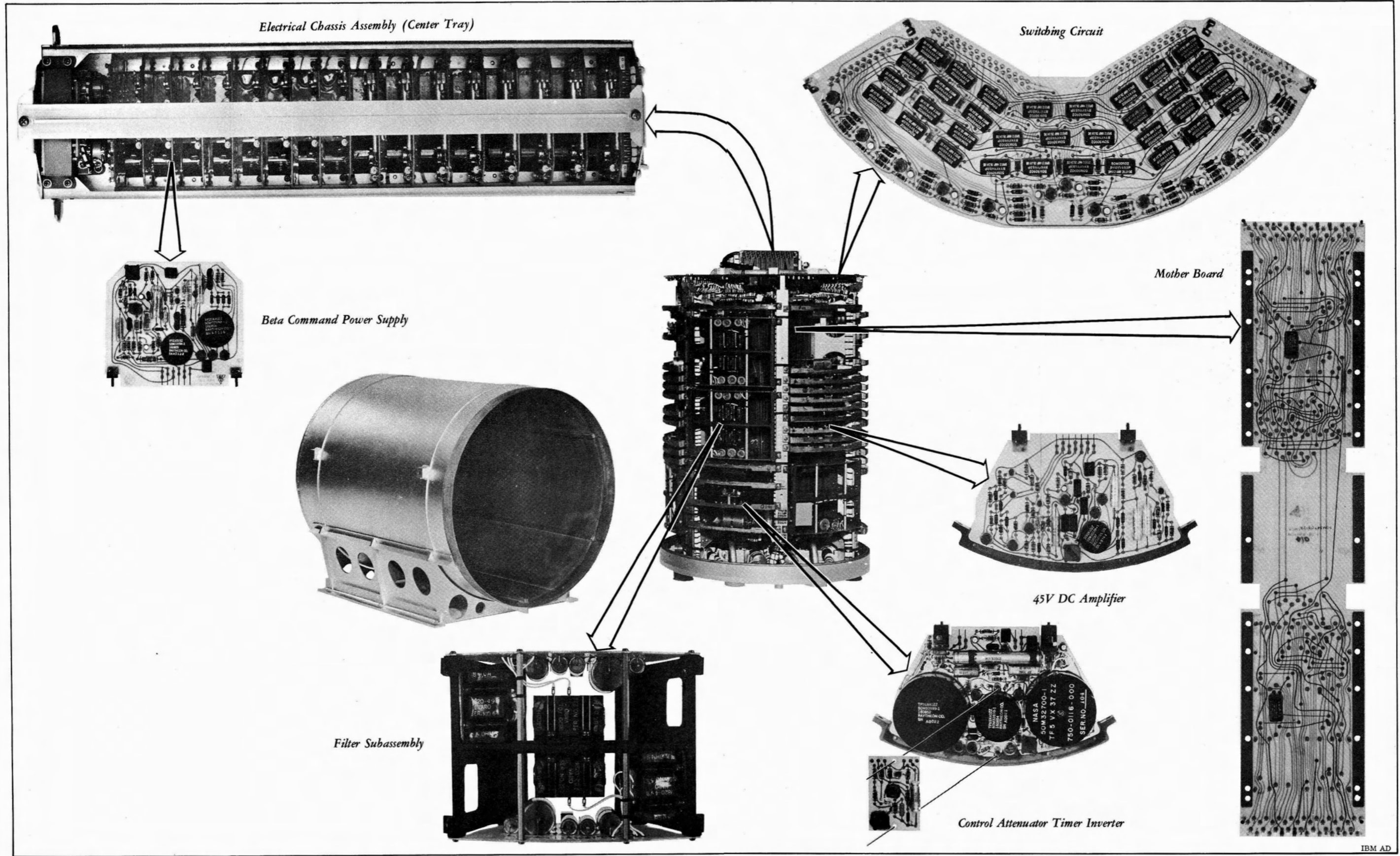


Figure 1-2. Flight Control Computer, Typical Subassemblies

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Table 1-1. Common Names of Major Subassemblies (Cont)

Official Nomenclature	Ref. Desig.	Common Name	Official Nomenclature	Ref. Desig.	Common Name
Scaling Board (Cont)	J51	ϕ Roll Scaling Board	Switching Circuit B (Figure 2-42)	A2	Switching Circuit A2
	J36	ϕ Pitch Scaling Board		A3	Switching Circuit A3
	J85	ϕ Pitch Scaling Board		J64	Pitch TM Amplifier
	J84	ϕ Yaw Scaling Board		J69	Roll-Yaw I TM Amplifier
Servo Comparator (Figure 2-29)	J107	Pitch Servo Amplifier Compartor	J75	Roll-Yaw II TM Amplifier	
	J110	Yaw Servo Amplifier Comparator	J117	1 Yaw 12 Ma Servo Amplifier	
Servo Inverter (Figure 2-21)	J56-A	Servo Inverter	J118	1 Pitch 12 Ma Servo Amplifier	
	J94-A	Servo Inverter (Standby)	J119	2 Yaw 12 Ma Servo Amplifier	
Spatial Amplifier (Figure 2-31)	J60, J61, J62	Pitch Spatial Amplifier	J120	2 Pitch 12 Ma Servo Amplifier	
	J65, J66, J67	Roll-Yaw I Spatial Amplifier	J121	3 Yaw 12 Ma Servo Amplifier	
	J71, J72, J73	Roll-Yaw II Spatial Amplifier	J122	3 Pitch 12 Ma Servo Amplifier	
Spatial Comparator (Figure 2-33)	J63	Pitch Spatial Comparator	J123	4 Yaw 12 Ma Servo Amplifier	
	J68	Roll-Yaw I Spatial Comparator	J124	4 Pitch 12 Ma Servo Amplifier	
	J74	Roll-Yaw II Spatial Comparator	45V DC Amplifier (Figure 2-5)	J32, J46, J47	DC Amplifier ($\Delta\phi Y$)
J70	Spatial Synchronizer	J33, J48, J49		DC Amplifier ($\Delta\phi P$)	
Switching Circuit A (Figure 2-41)	A1	Switching Circuit A1	J34, J52, J53	DC Amplifier ($\Delta\phi R$)	

Table 1-1. Common Names of Major Subassemblies (Cont)

Official Nomenclature	Ref. Desig.	Common Name	Official Nomenclature	Ref. Desig.	Common Name	
45V DC Amplifier (Cont)	J37, J38, J39	DC Amplifier (ϕ R)	ϕ Y, ϕ P SI Filter (Figure 2-9)	J90	$\Delta \phi$ Y S-IB Filter	
	J54	DC Amplifier (ψ Y)		J91	$\Delta \phi$ P S-IB Filter	
	J55	DC Amplifier (ψ P)		J40, J44, J45	$\Delta \phi$ Y S-IVB Filter	
	J80, J88, J89	DC Amplifier (ϕ P)			$\Delta \phi$ P S-IVB Filter	
	J81, J82, J87	DC Amplifier (ϕ Y)			J41, J94, J95	
	J106	DC Amplifier (Pitch Servo Comparator)		ϕ R, ϕ R SI Filter (Figure 2-10)	J29	ϕ R S-IB Filter
	J111	DC Amplifier (Yaw Servo Comparator)		ϕ Y, ϕ P SI Filter (Figure 2-12)	J30	$\Delta \phi$ R S-IB Filter
		J78	ϕ Y S-IB Filter			
50 Ma Servo Amplifier (Figure 2-27)	J103, J104, J105	Pitch 50 Ma Servo Amplifier	ϕ Y, ϕ P SIV Filter (Figure 2-13)	J79	ϕ P S-IB Filter	
	J112, J113, J114	Yaw 50 Ma Servo Amplifier		J96, J97, J109	ϕ Y S-IVB Filter	
				J98, J99, J108	ϕ P S-IVB Filter	
ψ Y, ψ P Filter (Figure 2-8)	J56	ψ Y Filter ψ P Filter				

Table 1-2. Flight Control Computer, Leading Particulars

Characteristic	Specification
<p style="text-align: center;">NOTE</p> <p>Overall dimensions are maximum, and include the Flight Control Computer case and cradle. Measurements indicated are obtained when viewing the Flight Control Computer from the connector end.</p>	
<p>Overall Dimensions</p> <p style="padding-left: 40px;">Height</p> <p style="padding-left: 40px;">Width</p> <p style="padding-left: 40px;">Depth</p> <p>Overall Weight</p> <p>Duty Cycle</p> <p>Warmup Time</p> <p>Input Power</p> <p>Operating Temperature Range</p> <p>Acceleration</p>	<p>21 inches</p> <p>20 inches</p> <p>30 inches</p> <p>130 pounds maximum</p> <p>100% continuous operation</p> <p>30 minutes</p> <p>22 to 32 Vdc, 320 watts maximum</p> <p>-55 to + 100 °C (-67 to 212 °F)</p> <p>20 G's peak</p>
<p style="text-align: center;">NOTE</p> <p>Reliability factor is for one Flight Control Computer and includes the probability that a failure may occur in a redundant circuit.</p>	
<p>Reliability Factor</p>	<p>0.999 for a Saturn mission consisting of (1) earth orbit insertion, (2) three earth orbits, (3) translunar injection, and (4) two hours of translunar coast</p>
<p style="text-align: center;">NOTE</p> <p>The Flight Control Computer case assembly is constructed so that a pressure drop to less than 14.7 psia will not be encountered over a two-year period in a perfect vacuum environment.</p>	
<p>Pressurization</p> <p>Cooling</p> <p>Case</p>	<p>Gaseous nitrogen to 20 psia at +17 to +27 °C (+63 to +81 °F)</p> <p>Fluid coolant lines</p> <p>Spot welded and dip brazed aluminum alloy</p>

Table 1-2. Flight Control Computer, Leading Particulars (Cont)

Characteristic	Specification
Cradle	
Type	Dip brazed aluminum alloy
Lifting Lugs	4
Case Cover	
Type	Spot welded and dip brazed aluminum alloy
Lifting Lugs	4
Transducer	50 psia range, motional pick up (pressure) transducer. Mounted on the inside of the case cover
Timer	Elapsed time indicator mounted on the case cover (activated when 28 Vdc power is applied to the Flight Control Computer)
Electrical Connectors	13
Primary Chassis	Welded and riveted aluminum alloy containing fluid coolant lines
Dual Filter Chassis	Magnesium alloy casting
Interconnections (Welded Modules)	
Point-to-Point Connections	Two nickel strips per connection
Spacing	Glass bead spacers used between nickel strips
Interconnecting Wiring	
Wire Material	Silver coated copper wire
AWG Wire Size	24 and 26
Strands	19 (minimum)
Insulation	Irradiated polyolefin (white)
Temperature Rating	+135 °C (+274 °F) maximum
Potential Rating	600 Vrms

Table 1-3. Flight Control Computer Subassemblies, Leading Particulars

Characteristic	Specification
<p>NOTE</p> <p>Unless otherwise indicated herein, or in Table 1-2, all of the characteristics and specifications shown in Table 1-2 are also applicable to the Flight Control Computer Subassemblies.</p>	
Beta Command Power Supply	8 per computer
Function	Power conversion (dc to dc) to supply 60 Vdc ($\pm 1\%$)
Construction Type	Printed circuit board
Overall Dimensions	
Height	4.75 inches
Width	5.25 inches
Depth	1.25 inches
Overall Weight	8 ounces maximum
Warmup Time	1 minute
Input Power	22 to 32 Vdc, 3 watts maximum
External Electrical Connections	Forked plug-in type connector
Electrical Characteristics	
Output Voltages	
Position Indicator Potentiometers Source Voltage (pins 1 and 5)	-60.0 (± 0.6) Vdc when output load is a 5,000-ohm ($\pm 1\%$) 1 watt resistor
Telemetry Voltage (pins 3 and 6)	-1.0 (± 0.03) Vdc
Output Ripple	Not greater than 20 mV rms
Output Spikes	Not greater than 400 mV peak-to-peak at 8 kHz
Overload Protection	(1) With output load reduced to 1.0 ($\pm 1\%$) ohms, maximum load current is not greater than 80 milliamperes, and (2) when output load returns to 5,000 ohms ($\pm 1\%$) normal output voltages are reestablished

Table 1-3. Flight Control Computer Subassemblies, Leading Particulars (Cont)

Characteristic	Specification
Beta Feedback	1 per computer
Function	Provides adjustment of feedback signal
Construction Type	Printed circuit board
Overall Dimensions	
Height	4.5 inches
Width	5.2 inches
Depth	0.8 inch
Overall Weight	4.8 ounces maximum
Warmup Time	None
Input Power	4 Vdc Maximum at 50 microamperes maximum
External Electrical Connections	Forked plug-in type connector
Control Attenuator Timer (CAT)	1 per computer
Function	Creates variations in gain as a function of time and activates microswitches at preset intervals
Construction Type	Gearbox type construction using precision aluminum alloy castings with all rotating shafts turning on ball bearings
Overall Dimensions	
Height	3.50 inches
Width	4.87 inches
Depth	2.25 inches
Overall Weight	29 ounces maximum
Warmup Time	None
Input Power	115 (± 1.15) Vac, 400 Hz, single-phase
External Electrical Connections	Forked plug-in type connector

Table 1-3. Flight Control Computer Subassemblies, Leading Particulars (Cont)

Characteristic	Specification
CAT Inverter	1 per unit
Function	Power conversion (dc to 400 Hz)
Construction Type	Printed circuit board
Overall Dimensions	
Height	4.75 inches
Width	8.50 inches
Depth	2.00 inches
Overall Weight	22 ounces maximum
Warmup Time	1 minute
Input Power	22 to 32 Vdc, 10 watts maximum
External Electrical Connections	Forked plug-in type connector
Electrical Characteristics	
Input Signal	22 to 32 Vdc
Efficiency	60% minimum
Regulation	±1% (voltage and frequency)
Power (pins 4 and 3)	6 to 10 watts
Output Voltage	115 Vrms, 400 Hz (±1%)
Servo Inverter Detector	1 per computer
Function	Monitor and switch outputs of servo inverters
Construction Type	Printed circuit board
Overall Dimensions	
Height	4.75 inches
Width	8.00 inches
Depth	0.75 inch
Overall Weight	7 ounces maximum
Warmup Time	1 minute

Table 1-3. Flight Control Computer Subassemblies, Leading Particulars (Cont)

Characteristic	Specifications
Input Power	22 to 32 Vdc, 240 milliwatts nominal, 2.24 watts maximum at trip point
External Electrical Connection	Forked plug-in type connector
Electrical Characteristics	
Input Signal	66 volts peak-to-peak (nominal) at 1200 Hz
Input Impedance	100,000 ohms minimum
Trip Point	10% input frequency change or 10% decrease of input amplitude ($\pm 3\%$)
Load	600-ohm relay
S-IB Power Distribution Assembly	1 per computer
Function	Applies +28 Vdc power to subassemblies
Construction Type	Printed circuit board
Overall Dimensions	
Height	4.75 inches
Width	8.00 inches
Depth	1.00 inch
Overall Weight	7 ounces maximum
Warmup Time	None
Input Power	None
External Electrical Characteristics	
Input Signal	+28 Vdc
Output Signal	+28 Vdc
Scaling Board	6 per computer
Function	Used with dc amplifiers to permit varied usage
Construction Type	Printed circuit board

Table 1-3. Flight Control Computer Subassemblies, Leading Particulars (Cont)

Characteristic	Specification
Overall Dimensions	
Height	4.75 inches
Width	8.00 inches
Depth	1.00 inch
Overall Weight	5 ounces maximum
Warmup Time	None
Input Power	None required
External Electrical Connections	Forked plug-in type connector
Servo Amplifier Comparator	2 per computer
Function	Monitor and switch outputs from servo amplifiers
Construction Type	Printed circuit board
Overall Dimensions	
Height	4.50 inches
Width	8.00 inches
Depth	1.00 inch
Overall Weight	7 ounces maximum
Warmup Time	1 minute
Input Power	22 to 32 Vdc, 1.4 watts maximum
External Electrical Connections	Forked plug-in type connector
Electrical Characteristics	
Input Signal	8.0 milliampere differential input current to produce trip point
Input Impedance	Not less than 20,000 ohms
Load	600-ohm relay coil

Table 1-3. Flight Control Computer Subassemblies, Leading Particulars (Cont)

Characteristic	Specification
Output Signal	(1) Yaw channel A or B or (2) pitch channel A or B
Switch Point Tolerance	±1% of the differential input signal
Telemetry Output	A discrete signal at the switch point
Servo Inverter	2 per computer
Function	Power conversion (dc to ac)
Construction Type	Printed circuit board
Overall Dimensions	
Height	4.75 inches
Width	8.00 inches
Depth	1.75 inches
Overall Weight	26 ounces maximum
Warmup Time	5 minutes
Input Power	22 to 32 Vdc, 10 watts maximum
External Electrical Connections	Forked plug-in type connector
Electrical Characteristics	
Input Signal	22 to 32 Vdc, 300 milliamperes (maximum)
Output Signal	1200 Hz, 66-volt peak-to-peak square wave
Spatial Amplifier	9 per computer
Function	To produce trip and full-ON output conditions
Construction Type	Printed circuit board
Overall Dimensions	
Height	4.75 inches
Width	8.00 inches
Depth	1.00 inch

Table 1-3. Flight Control Computer Subassemblies, Leading Particulars (Cont)

Characteristic	Specification
Overall Weight	13.6 ounces maximum
Warmup Time	1 minute
Input Power	22 to 32 Vdc, 7.28 watts maximum
External Electrical Connections	Forked plug-in type connector
Electrical Characteristics	
Input Signal	210 (± 10) microamperes into 300 turns to produce trip point; 320 (± 15) microamperes into 300 turns to produce full-ON condition
On Time	65 milliseconds (for 210 microampere input)
Load	Four 600-ohm relay coils in parallel and one 1200-ohm counter in parallel with the relays
Output	0.840 (± 0.04) Vdc (positive and negative) trip output; 1.28 (± 0.06) Vdc (positive and negative) full output
Switch Rate	5 pulses per second at 50% duty cycle
Spatial Comparator	3 per computer
Function	Monitor and switch outputs from two associated spatial amplifiers
Construction Type	Printed circuit board
Overall Dimensions	
Height	4.75 inches
Width	8.00 inches
Depth	1.00 inch
Overall Weight	7 ounces maximum
Warmup Time	1 minute
Input Power	22 to 32 Vdc, 7 watts maximum
External Electrical Connections	Forked plug-in type connector

Table 1-3. Flight Control Computer Subassemblies, Leading Particulars (Cont)

Characteristic	Specification
Electrical Characteristics	
Input Signal	The output signals supplied by two spatial amplifiers
Input Impedance	Not available
Load	600-ohm relay coil
Output Signal	Output pulse width in milliseconds/total period is (nominal) 100/200 for 50% duty cycle at 5 Hz
Switching Accuracy	10% or less
Telemetry Output	A discrete signal at the switch point
Spatial Synchronizer	1 per computer
Function	Synchronizes spatial control signals to control relay packages I and II in S-IVB Stage
Overall Dimensions	
Height	4.75 inches
Width	8.00 inches
Depth	1.00 inch
Overall Weight	7 ounces maximum
Control Signal	S-IVB burn mode signal
Switching Circuit A1	1 per computer
Function	Establishes switching combinations
Construction Type	Printed circuit board
Overall Dimensions	
Height	15.00 inches
Width	6.00 inches
Depth	1.25 inches

Table 1-3. Flight Control Computer Subassemblies, Leading Particulars (Cont)

Characteristic	Specifications
Overall Weight	18 ounces maximum
Warmup Time	1 minute
Input Power	22 to 32 Vdc, 14 watts maximum
External Electrical Connections	Forked plug-in type connector
Electrical Characteristics	
Input Signal	22 to 32 Vdc
Output	Resistance and voltage combinations
Switching Circuit A2	1 per computer
Function	Establishes switching combinations
Construction Type	Printed circuit board
Overall Dimensions	
Height	15.00 inches
Width	6.00 inches
Depth	1.25 inches
Overall Weight	20 ounces maximum
Warmup Time	1 minute
Input Power	22 to 32 Vdc, 14 watts maximum
External Electrical Connections	Forked plug-in type connector
Electrical Characteristics	
Input Signal	22 to 32 Vdc
Output	Resistance and Voltage combinations
Switching Circuit A3	1 per computer
Function	Establishes switching combinations
Construction Type	Printed circuit board

Table 1-3. Flight Control Computer Subassemblies, Leading Particulars (Cont)

Characteristic	Specification
Overall Dimensions	
Height	15.00 inches
Width	6.00 inches
Depth	1.25 inches
Overall Weight	17 ounces maximum
Warmup Time	1 minute
Input Power	22 to 32 Vdc, 14 watts maximum
External Electrical Connections	Forked plug-in type connector
Electrical Characteristics	
Input Signal	22 to 32 Vdc
Output	Resistance and voltage combinations
Telemetry Amplifier	3 per computer
Function	To produce a 1.52 ($\pm 3\%$) Vdc output in response to a spatial full-ON condition
Construction Type	Printed circuit board
Overall Dimensions	
Height	4.75 inches
Width	8.00 inches
Depth	1.25 inches
Overall Weight	7 ounces maximum
Warmup Time	1 minute
Input Power	22 to 32 Vdc, 3.5 watts maximum
External Electrical Connections	Forked plug-in type connector

Table 1-3. Flight Control Computer Subassemblies, Leading Particulars (Cont)

Characteristic	Specifications
Electrical Characteristics	
Input Signal	Zero to 215 milliamperes
Gain	11.9 volts per ampere (voltage gain)
Input Impedance	0.5 ohms maximum
Load Limit	5,000 ohms minimum
Output Signal	1.52 ($\pm 3\%$) Vdc excursion, positive or negative
Cut-Off Frequency	500 Hz minimum
Null Offset	With no input signal applied, null offset is less than 0 ± 10 mV referred to input
12 Ma Servo Amplifier	8 per computer
Function	Signal amplification
Construction Type	Printed circuit board
Overall Dimensions	
Height	4.75 inches
Width	5.00 inches
Depth	1.75 inches
Overall Weight	14 ounces maximum
Warmup Time	5 minutes
Input Power	22 to 32 Vdc, 0.56 watts maximum
External Electrical Connections	Forked plug-in type connector
Electrical Characteristics	
Input Signal	Current levels, in microamperes: 20.00; 21.20; 14.13; 84.80; 42.40
Load	Dual 1000-ohm resistive load
Output Signal	12 milliamperes

Table 1-3. Flight Control Computer Subassemblies, Leading Particulars (Cont)

Characteristic	Specification
Linearity	Within 2%
Null Offset	With no input signal applied and input terminated by 50,000 ohms, null offset is less than 0 ± 20 mV
45V DC Amplifier	22 per computer
Function	Dual input 45-volt differential amplifier
Construction Type	Printed circuit board
Overall Dimensions	
Height	4.50 inches
Width	8.00 inches
Depth	0.80 inch
Overall Weight	7 ounces maximum
Warmup Time	1 minute
Input Power	22 to 32 Vdc, 1.96 watts maximum
External Electrical Connections	Forked plug-in type connector
Electrical Characteristics	
Input Signal	A differential value between zero and 45 volts, positive or negative, at 10 Hz maximum
Gain	Between 0.5 and 25.0 as determined by the selected value of feedback selector
Input Impedance	20,000 ohms at gain of 0.5, 2 megohms at gain of 25.0
Output Impedance	100 ohms maximum
Load Limit	5,000 ohms
Output Signal	A differential value between zero and 45 volts, positive or negative, as developed across a 5,000-ohm load
Non-linearity	Not more than 2%

Table 1-3. Flight Control Computer Subassemblies, Leading Particulars (Cont)

Characteristic	Specification
Dynamic Response	Not more than 5 degrees phase shift with 10 Hz or lower frequency input signals
Null Offset	With no input signal applied, null offset is less than 0 ± 10 mV referred to input
50 Ma Servo Amplifier	6 per computer
Function	Signal amplification
Construction Type	Printed circuit board
Overall Dimensions	
Height	4.75 inches
Width	8.00 inches
Depth	2.00 inches
Overall Weight	16 ounces maximum
Warmup Time	10 minutes
Input Power	22 to 32 Vdc, 2.8 watts maximum
External Electrical Connections	Forked plug-in type
Electrical Characteristics	
Input Signal	
ϕ Y or ϕ P	33.6 or 48.4 Vdc, positive and negative
ϕ Y or ϕ P	7.465 Vdc, positive and negative
Current Gain	3.7 milliamperes per milliampere turn ($\pm 2\%$)
Input Current	Not more than 0 ± 30 microamperes into 450 turns per stack for full output signal swing
Output Impedance	Greater than 40,000 ohms from zero to 20 Hz
Output Current	A full output swing between zero and 50 milliamperes, positive and negative, as developed across a 100-ohm valve load

Table 1-3. Flight Control Computer Subassemblies, Leading Particulars (Cont)

Characteristic	Specifications
Nonlinearity	Not more than 2%
Dynamic Response	The 3 db down frequency is 55 (± 5) Hz
Null Offset	With no input signal applied, null offset is less than 0 ± 50 mV referred to input
Source Impedance	Not less than 20,000 ohms and isolated by at least 150,000 ohms from the 28 Vdc supply voltage
$\dot{\gamma}$ Y and $\dot{\gamma}$ P Filters	1 per computer
Function	Bending mode filters used to filter out angular errors resulting from frame vibrations
Overall Dimensions	
Height	4.75 inches
Width	8.00 inches
Depth	1.00 inch
Filter Type	Low pass
$\Delta\phi$ Y and $\Delta\phi$ P S-IB Filters	2 per computer
Function	Bending mode filters used to filter out angular errors resulting from frame vibrations. Also provides time sequenced attenuation control
Overall Dimensions	
Height	4.7 inches
Width	8.0 inches
Depth	1.0 inch
Attenuation Switchpoints	Switchpoint 1 (100 seconds after lift-off) Switchpoint 2 (110 seconds after lift-off) Switchpoint 3 (120 seconds after lift-off)
Filter Type	Low pass

Table 1-3. Flight Control Computer Subassemblies, Leading Particulars (Cont)

Characteristic	Specification
$\Delta \phi$ Y and $\Delta \phi$ P S-IVB Filters	6 per computer
Function	Bending mode filters used to filter out angular errors resulting from frame vibrations
Overall Dimensions	
Height	4.75 inches
Width	8.00 inches
Depth	1.00 inch
Filter Type	Low pass
ϕ R and $\Delta \phi$ R S-IB Filters	2 per computer
Function	Bending mode filters used to filter out angular errors resulting from frame vibrations. Also provides time sequenced attenuation control
Overall Dimensions	
Height	4.7 inches
Width	8.0 inches
Depth	6.5 inches
Attenuation Switchpoint	Switchpoint 3 (120 seconds after lift-off)
Filter Type	Low pass
ϕ Y and ϕ P S-IB Filters	2 per computer
Function	Bending mode filters used to filter out angular errors resulting from frame vibrations. Also provides time sequenced attenuation control
Overall Dimensions	
Height	4.75 inches
Width	8.00 inches
Depth	6.50 inches

Table 1-3. Flight Control Computer Subassemblies, Leading Particulars (Cont)

Characteristic	Specification
Attenuation Switchpoints	Switchpoint 1 (100 seconds after lift-off) Switchpoint 2 (110 seconds after lift-off) Switchpoint 3 (120 seconds after lift-off)
Filter Type	Low pass
ø Y and øP S-IVB Filters	6 per computer
Function	Bending mode filters used to filter out angular errors resulting from frame vibrations. Also provides time sequenced attenuation control
Overall Dimensions	
Height	4.7 inches
Width	8.0 inches
Depth	6.5 inches
Attenuation Switchpoints	Switchpoint 4 (S-IVB ignition + approximately 20 seconds) Switchpoint 5 (S-IVB ignition + 200 seconds)
Filter Type	Low pass

SECTION II

PRINCIPLES OF OPERATION

2-1 INTRODUCTION

This section contains the principles of operation of the Flight Control Computer and is divided into two parts; General Operation and Detailed Operation. The General Operation portion (Paragraph 2-2 through 2-7) is limited to the functional block diagram theory of the Flight Control Computer. The Detailed Operation portion contains a detailed circuit description of each component of the Flight Control Computer. The Detailed Operation portion of this manual will be supplied at a later date.

2-2 GENERAL OPERATION

The explanation presented, describes the manner in which the Flight Control Computer is connected to the launch vehicle systems, and how the systems function with the Flight Control Computer to provide control commands to the Saturn launch vehicle.

The Flight Control Computers for vehicles AS-204 and AS-205 of the Saturn IB program are functionally identical. The only difference is in the bending mode filters. This difference is explained in Paragraph 2-7, SUBASSEMBLY FUNCTIONAL DESCRIPTION under the Bending Mode Filters heading.

Paragraph 2-3 contains a general explanation of system tie-in and describes the functional and physical relationship of the Flight Control Computer with associated systems within the Saturn launch vehicle. Paragraphs 2-4 through 2-6 describes the functional relationship between the various Flight Control Computer circuits during the S-IB burn mode, the S-IVB burn mode, and the S-IVB coast mode. Paragraph 2-7 contains a functional description of each of the Flight Control Computer circuits.

2-3 SYSTEM TIE-IN

Figure 2-1 illustrates the functional and physical relationship of the Flight Control Computer to the various position sensing and position controlling devices of the Saturn IB launch vehicle.

The Flight Control Computer is contained in the Instrument Unit of the Saturn launch vehicle. Operation of the Flight Control Computer is divided into three basic functions--control of the vehicle during S-IB Stage burn, control of the vehicle during S-IVB Stage burn, and control of the vehicle during S-IVB coast.

The Flight Control Computer, which is the controlling element of the Saturn control system, accepts signals from three sources.

- Attitude correction commands from the LVDC/LVDA guidance system.
- Attitude angular rate signals from the Rate Gyro Package.
- Lateral acceleration rate signals from the Control Accelerometers.

The ST-124M-3 Inertial Platform Assembly provides a space-fixed reference for vehicle attitude and velocity measurements. The measurements are sent through the Launch Vehicle Data Adapter (LVDA) to the Launch Vehicle Digital Computer (LVDC) where they are used to calculate the vehicle's position in space. The guidance portion of the LVDC program determines the instantaneous position of the vehicle and attitude corrections (steering commands) required for the vehicle to reach its predetermined destination in space. These attitude correction commands are issued through the LVDA to the Flight Control Computer. The Flight Control Computer commands the engine actuators to move their respective engines to the desired angle. This gimbaling action has the effect of changing the direction of the thrust vector of the vehicle, thus moving the vehicle about its roll, pitch, or yaw axis. The Control-EDS Rate Gyros and the Control Accelerometers sense the rate of change of these attitude changes and movements of the vehicle. The signals thus generated are supplied as dynamic feedback to the Flight Control Computer. The Flight Control Computer sums the angular rate information from the Rate Gyros and the lateral

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acceleration information from the Control Accelerometers with the attitude correction commands from the LVDC/LVDA to produce engine commands which conform to the prescribed vehicle limits.

Note that provisions have been made which allow the Apollo spacecraft to assume control of the vehicle. When this prerogative is exercised the attitude correction signals are supplied by the Spacecraft guidance system.

The three attitude correction signals applied to the Flight Control Computer from the LVDC/LVDA are $\Delta\phi Y$, $\Delta\phi R$, and $\Delta\phi P$. The $\Delta\phi Y$ signal is the yaw attitude error signal, the $\Delta\phi R$ signal is the roll attitude error signal, and the $\Delta\phi P$ signal is the pitch attitude error signal.

The S-IB burn mode is activated by the first motion signal from the Switch Selector. The first motion signal is applied to the S-IB power distribution assembly which then applies power to the switching circuits and S-IB burn mode circuits. The switching circuits control the routing of signals in the Flight Control Computer by opening or closing appropriate relay contacts located in the signal lines.

During the S-IB burn mode, the attitude error signals are applied through contacts which are closed during S-IB burn only to the S-IB yaw servo amplifiers and S-IB pitch servo amplifiers. The $\Delta\phi Y$ error signal is applied to the S-IB yaw servo amplifiers only, the $\Delta\phi P$ error signal is applied to the S-IB pitch servo amplifiers only, and the $\Delta\phi R$ error signal is applied to both the S-IB yaw and S-IB pitch servo amplifiers.

The S-IB yaw servo amplifiers control the four gimballed engines in the S-IB Stage in the yaw axis, and the pitch servo amplifiers control the four gimballed engines in the S-IB Stage in the pitch axis. The yaw servo amplifiers and the pitch servo amplifiers, operating simultaneously, control the four gimballed engines in the roll axis.

The Control Rate Gyros, contained in the S-IVB Stage, provides angular rate error signals in the roll, yaw, and pitch axes. These signals are designated ϕR , ϕY , and ϕP , respectively. The angular rate error signals are coupled through relay contacts which are closed during S-IB burn only, and through jumpers, to the S-IB yaw and pitch servo amplifiers. These signals are added algebraically with the attitude error signals to control the rate of response of the vehicle and to eliminate overshoot at the end of the correction. The jumpers provide for selection of angular rate

signals from the Control Signal Processor or from the Control Rate Gyros. However, this selection must be made prior to launch.

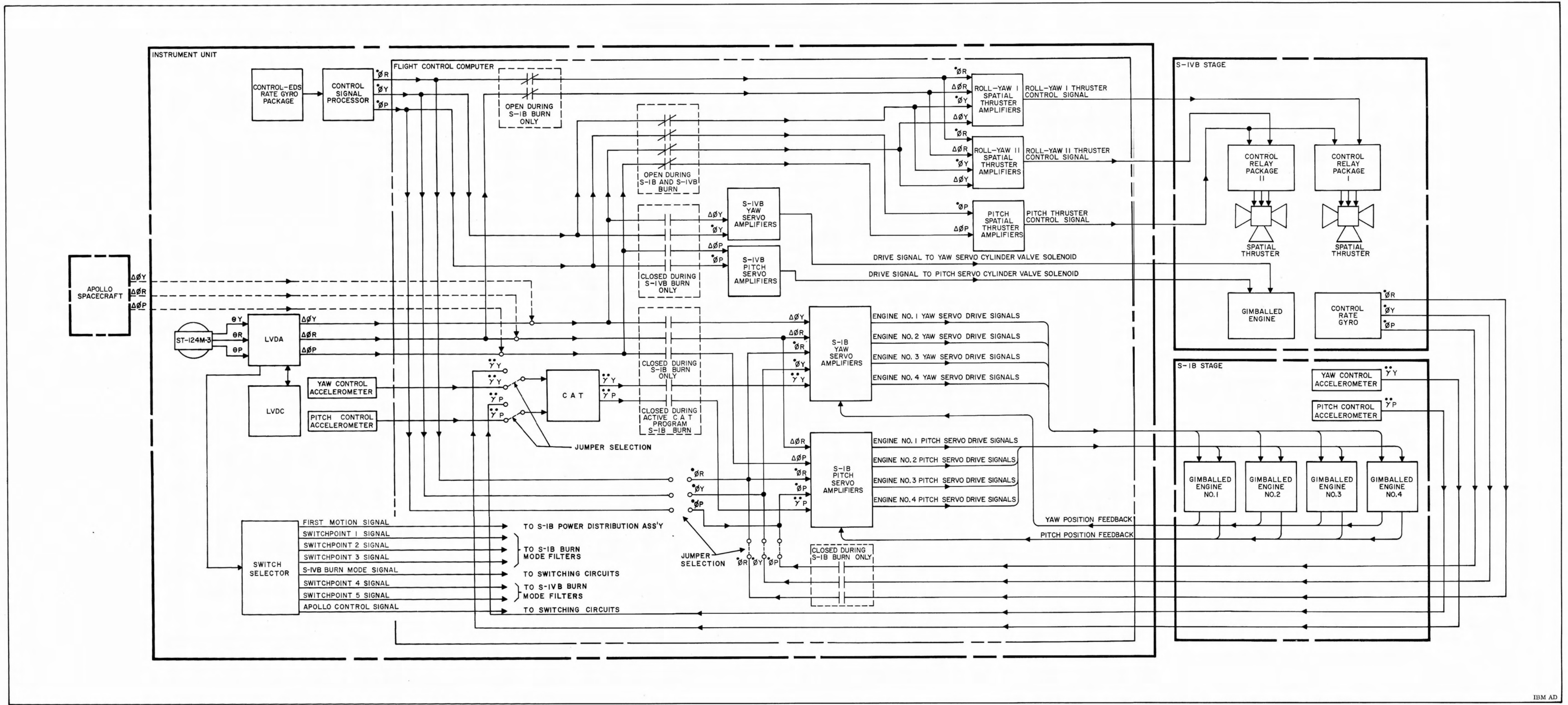
Yaw and Pitch Control Accelerometers are contained in the S-IB Stage and in the Instrument Unit. They are used to measure the lateral accelerations of the launch vehicle. Selection of the Control Accelerometers to be used during the flight is also made by jumpers and must be selected prior to launch. The output of the Yaw Control Accelerometer is called $\dot{\gamma} Y$, and the output of the Pitch Control Accelerometer is called $\dot{\gamma} P$. The acceleration error signals are applied through jumper selection to the Control Attenuator Timer (CAT) which adjusts the amplitude of the signals in relation to the phase of the flight. The lateral acceleration error signals are then applied through contacts which are closed during the active CAT program to the yaw servo amplifiers and to the pitch servo amplifiers. They are then algebraically added to the attitude error signals from the LVDC/LVDA.

During the S-IB burn mode, switchpoint signals 1 through 3 from the Switch Selector are applied to signal line filters. The filters then adjust the amplitude of the error signals in relation to the phase of the flight.

Upon completion of the S-IB burn mode, the S-IB Stage is jettisoned and an S-IVB burn mode signal from the Switch Selector is applied to the Flight Control Computer switching circuit. The switching circuit activates the signal line relays which sets the Flight Control Computer for flight control during the S-IVB burn mode.

The yaw and pitch attitude error signals ($\Delta\phi Y$ and $\Delta\phi P$) are then applied through contacts closed during S-IVB burn only to the S-IVB yaw servo amplifiers and S-IVB pitch servo amplifiers. The resultant drive signals from the S-IVB yaw and pitch servo amplifiers position the gimballed engine of the S-IVB Stage to control the yaw and pitch attitude of the vehicle during burn of the S-IVB Stage. The roll attitude error signal ($\Delta\phi R$) is applied through contacts open during S-IB burn only to the roll-yaw I and roll-yaw II spatial thruster amplifiers. The resultant control signals are applied to the control relay packages in the S-IVB Stage and operate the spatial thrusters as required to provide roll control during S-IVB burn mode.

The angular rate error signals (ϕY and ϕP) are applied from the Control Signal Processor through



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Figure 2-1. Flight Control Computer, System Tie-In Functional Block Diagram

contacts which are closed during S-IVB burn only to the S-IVB yaw servo amplifiers and S-IVB pitch servo amplifiers. These angular rate signals algebraically add with the attitude error signals from the LVDA to produce the drive signals for the gimballed engine in the S-IVB Stage. The roll angular rate error signal ($\dot{\phi}_R$) is applied through contacts which are open during the S-IB burn only to the roll-yaw I and roll-yaw II spatial thruster amplifiers. The roll angular rate error signal algebraically adds with the attitude error signals from the LVDC/LVDA to produce the roll control signals.

During the S-IVB burn mode, switchpoint signals 4 and 5 from the Switch Selector are applied to signal line filters. The filters then adjust the amplitude of the error signals in relation to the phase of the flight.

At the completion of the S-IVB burn mode, the S-IVB burn mode signal is removed. This causes the switching circuit relay contacts to transfer the yaw and pitch attitude error signals from the S-IVB yaw servo amplifiers and S-IVB pitch servo amplifiers to the spatial thruster amplifiers. The yaw attitude error signal ($\Delta\phi_Y$) is simultaneously applied to roll-yaw I and roll-yaw II spatial thruster amplifiers. The pitch attitude error signal ($\Delta\phi_P$) is applied to the pitch spatial thruster amplifiers. These attitude error signals are applied through closed contacts which are open during S-IB burn and S-IVB burn modes. The yaw and pitch angular rate error signals ($\dot{\phi}_Y$ and $\dot{\phi}_P$) from the Control Signal Processor, are also transferred to the spatial thruster amplifiers.

When the S-IVB coast mode function is completed, the Flight Control Computer is no longer needed and the S-IVB Stage, together with the Instrument Unit are jettisoned.

2-4 FUNCTIONAL OPERATION DURING S-IB BURN MODE

Figure 2-2 illustrates the functional operation of the Flight Control Computer during the S-IB burn mode. At lift-off, the LVDC/LVDA, through the Switch Selector, initiates a first motion signal which is applied to the Flight Control Computer S-IB power distribution assembly. The first motion signal causes the S-IB power distribution assembly to apply dc power to switching circuits A1 and A3 and the S-IB burn mode circuits. The application of power to the switching circuits energizes relays K13 through K24 of A1 and relays K1 through K15 of A3.

Redundant relay switching is used throughout the Flight Control Computer to achieve a high degree of reliability. To simplify the functional block diagram of Figure 2-2, redundant relay contacts are shown as a single set of contacts and are identified as to which relays they represent.

The $\Delta\phi_Y$, $\Delta\phi_R$, and $\Delta\phi_P$ attitude error signals are applied to the Flight Control Computer from the LVDC/LVDA. In the Flight Control Computer, the attitude error signals are coupled through the redundant closed relay contacts of switching circuit A1 and applied to dc amplifiers J46, J34, and J48, respectively.

The amplified $\Delta\phi_Y$ error signal from dc amplifier J46 is routed through switching circuit A1 to $\Delta\phi_Y$ S-IB filter J90. This filter is a bending mode filter which filters out angular yaw errors resulting from physical deflections of the Saturn IB structure. The deflections occur when the Saturn IB launch vehicle frame vibrates at its resonant frequency during high thrust loadings. By means of the switchpoint 1, 2, and 3 signals from the LVDA (applied through the Switch Selector) the $\Delta\phi_Y$ S-IB filter also adjusts the amplitude of the $\Delta\phi_Y$ attitude error signal in relationship to the phase of the flight.

The $\Delta\phi_Y$ attitude error signal from $\Delta\phi_Y$ S-IB filter J90 is then coupled to the four yaw 12 ma servo amplifiers J117, J119, J121, and J123. The resultant drive signals from the yaw 12 ma servo amplifiers are applied to the four yaw servo cylinders in the S-IB Stage. Each drive signal causes the associated yaw servo cylinder to mechanically position the four gimballed engines of the S-IB Stage as required to correct for errors in the yaw axis. The yaw servo cylinders are also mechanically linked to position indicator potentiometers, which supply the position feedback signal for the yaw 12 ma servo amplifiers. Each position indicator potentiometer receives its source voltage from an associated yaw beta command power supply (J125 through J128).

The amplified $\Delta\phi_P$ attitude error signal from dc amplifier J48 is routed through switching circuit A1 and $\Delta\phi_P$ S-IB filter J91 to pitch 12 ma servo amplifiers J118, J120, J122, and J124. The $\Delta\phi_P$ filter is a bending mode filter which also adjusts the amplitude of the $\Delta\phi_P$ attitude error signal in relation to the phase of the flight. The amplitude adjustment is controlled by the LVDC/LVDA through the Switch Selector. The resultant drive signals from the pitch 12 ma servo amplifiers are applied to the four pitch servo cylinders in the S-IB Stage. The drive signals

cause the associated pitch servo cylinders to mechanically position the four gimballed engines of the S-IB Stage to correct for errors in the pitch axis. The pitch servo cylinders are also mechanically linked to position indicator potentiometers, which supply the position feedback signal for the pitch 12 ma servo amplifiers. Each pitch position indicator potentiometer receives its source voltage from an associated pitch beta command power supply (J129 through J132).

The amplified $\Delta\phi R$ attitude error signal from dc amplifier J34 is coupled through switching circuit A1 and $\Delta\phi R$ S-IB filter J30 to the four yaw 12 ma servo amplifiers and the four pitch 12 ma servo amplifiers. The $\Delta\phi R$ filter is a bending mode filter which also adjusts the amplitude of the $\Delta\phi R$ error signal in relation to the phase of the flight. The amplitude adjustment is controlled by the LVDC/LVDA through the Switch Selector. The resultant drive signals from the pitch and yaw 12 ma servo amplifiers cause the four gimballed engines in the S-IB Stage to correct for errors in the roll axis.

Angular rate error signals ($\dot{\phi}Y$, $\dot{\phi}R$, and $\dot{\phi}P$) for the S-IB burn mode are supplied by either the Control-EDS Rate Gyro Package (through the Control Signal Processor), located in the Instrument Unit, or by the Control Rate Gyros located in the S-IVB Stage. Selection of the component to be used during flight is determined prior to lift-off by means of the jumper selector.

The $\dot{\phi}Y$, $\dot{\phi}R$, and $\dot{\phi}P$ angular rate error signals are coupled through switching circuit A3 to filters J78, J29, and J79, respectively. Filters J78, J29, and J79 are bending mode filters which filter out angular position errors resulting from physical deflections of the missile structure. Filters J78, J29, and J79 are also used to adjust the amplitude of the $\dot{\phi}Y$, $\dot{\phi}R$, and $\dot{\phi}P$ error signals in relation to the phase of the flight. The $\dot{\phi}Y$ signal from $\dot{\phi}Y$ filter J78 is applied to the yaw 12 ma servo amplifiers, the $\dot{\phi}P$ signal from $\dot{\phi}P$ filter J79 is applied to the pitch 12 ma servo amplifiers, and the $\dot{\phi}R$ filter J29 is applied to the yaw and pitch 12 ma servo amplifiers. The $\dot{\phi}Y$, $\dot{\phi}R$, and $\dot{\phi}P$ angular rate error signals are algebraically added with the $\Delta\phi Y$, $\Delta\phi R$, and $\Delta\phi P$ attitude error signals in the 12 ma servo amplifiers to produce the servo drive signals.

Switchpoint signals which control the amount of attenuation that filters J29, J30, J78, J79, J90, and J91 provide for their associated error signal is received from the LVDC/LVDA through the IU

Switch Selector. The Switch Selector initiates switchpoint No. 1 signal at T plus 100 seconds, switchpoint No. 2 signal at T plus 110 seconds, and switchpoint No. 3 signal at T plus 120 seconds. Switchpoint No. 1 and No. 2 signals are applied to filters J78, J79, J90, and J91 only. Switchpoint No. 3 signal is applied to filters J29, J30, J78, J79, J90, and J91.

The lateral acceleration error signals ($\ddot{\gamma}Y$ and $\ddot{\gamma}P$) for the S-IB burn mode are normally supplied by the Yaw and Pitch Control Accelerometers located in the Instrument Unit. However, the acceleration error signals can be supplied by the Yaw and Pitch Control Accelerometers located in the S-IB Stage by changing the jumper selectors prior to lift-off. The acceleration error signals are position error signals for yaw and pitch with respect to the lateral acceleration of the Saturn launch vehicle.

The $\ddot{\gamma}Y$ and $\ddot{\gamma}P$ error signals are applied to Control Attenuator Timer (CAT) J134 which controls the amplitude of the $\ddot{\gamma}Y$ and $\ddot{\gamma}P$ error signals in relationship to the altitude of the missile. The $\ddot{\gamma}Y$ error signal from the CAT is coupled through $\ddot{\gamma}Y$ filter J56 (bending mode filter), dc amplifier J54, and relay contacts which are closed during the active CAT program to the yaw 12 ma servo amplifiers. The $\ddot{\gamma}P$ error signal from the CAT is coupled through $\ddot{\gamma}P$ filter J57 (bending mode filter), dc amplifier J55, and relay contacts which are closed during the active CAT program to the pitch 12 ma servo amplifiers. The $\ddot{\gamma}Y$ and $\ddot{\gamma}P$ error signals are algebraically added with the attitude and angular rate error signals in the 12 ma servo amplifiers to produce the servo drive signals.

The ac excitation signal required for the operation of the magnetic amplifiers in the yaw and pitch 12 ma servo amplifiers is normally supplied by servo inverter J56A. Servo inverters J56A and J94A each convert the 28 Vdc from S-IB power distribution assembly J100 to 60 Vac, 1 kHz. However, only the output of servo inverter J56A is coupled through servo inverter detector J94B to the 12 ma servo amplifiers. If a malfunction occurs in servo inverter J56A, servo inverter detector J94B detects the malfunction and automatically applies the output of servo inverter J94A to the 12 ma servo amplifiers. The ac power required for the operation of CAT J134 is provided by CAT inverter J90A. The CAT inverter inverts the 28 Vdc from the S-IB power distribution assembly to 115 Vac, 400 Hz.

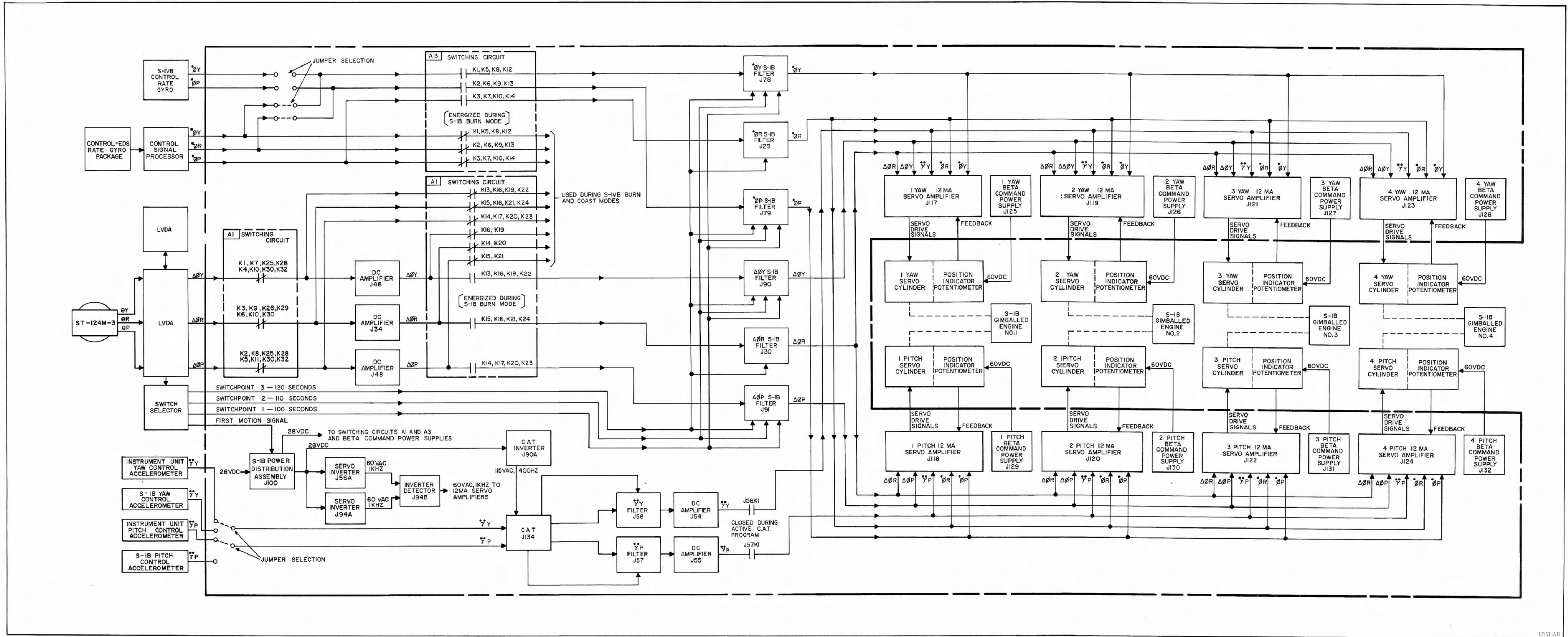


Figure 2-2. S-IB Burn Mode, Functional Block Diagram

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2-5 FUNCTIONAL OPERATION DURING S-IVB
BURN MODE

Figure 2-3 illustrates the functional operation of the Flight Control Computer during the S-IVB burn mode. At the completion of the S-IB burn mode, the S-IB Stage is jettisoned and the S-IVB Stage is activated. The LVDA, through the Switch Selector, applies an S-IVB burn mode signal to the Flight Control Computer. In the Flight Control Computer, the S-IVB burn mode signal is applied to switching circuit A2 causing relays K1 through K12 to become energized.

Redundant relay switching is used throughout the Flight Control Computer to achieve a high degree of reliability. To simplify the functional block diagram of Figure 2-3, redundant relay contacts are shown as a single set of contacts and are identified as to which relays they represent.

The $\Delta\phi Y$, $\Delta\phi R$, and $\Delta\phi P$ attitude error signals are normally applied to the Flight Control Computer from the LVDC/LVDA. In the Flight Control Computer, the attitude error signals are routed through the redundant closed relay contacts of switching circuit A1 and applied to triple redundant dc amplifiers. The $\Delta\phi Y$ attitude error signal is applied to dc amplifiers J32, J46, and J47; the $\Delta\phi R$ attitude error signal is applied to dc amplifiers J33, J48, and J49; and the $\Delta\phi P$ attitude error signal is applied to dc amplifiers J32, J52, and J53.

Guidance of the Saturn launch vehicle during the S-IVB burn mode can also be controlled by the Apollo spacecraft. Switchover is accomplished upon receipt of an Apollo control signal from the Switch Selector. The Apollo control signal is applied to switching circuit A1 causing relays K1 through K12 of A1 to become energized.

Triple redundancy circuits are used for the S-IVB Stage to achieve a high reliability. Basically, this means that the input signals have three paths to follow. Outputs from the normal path and a second path are continuously compared with each other. If a malfunction occurs in either of these two paths, the comparison circuit automatically detects the malfunction and switches to the third path. The probability of a malfunction occurring in two paths at the same time is very low.

The amplified $\Delta\phi Y$ error signals from dc amplifiers J46, J47, and J32 are coupled through switching circuit A2 to ϕY S-IVB filters J44, J45, and

J40, respectively. These are bending mode filters which filter out angular yaw errors resulting from the physical deflection of the Saturn launch vehicle. These deflections occur when the frame vibrates at its resonant frequency during high thrust loadings.

The $\Delta\phi Y$ error signal outputs from the $\Delta\phi Y$ filters are applied to their respective yaw 50 ma servo amplifier. Yaw 50 ma servo amplifiers J112 through J114 will then develop servo drive signals in proportion to the input error signals. However, only the servo drive signal from yaw 50 ma servo amplifier J113 is coupled through the normally closed contacts of the comparator relay to the yaw servo actuator in the S-IVB Stage. The yaw servo drive signal causes the yaw servo actuator to mechanically position the S-IVB gimballed engine as required to correct for errors in the yaw axis.

The outputs from yaw 50 ma servo amplifiers J112 and J113 are continuously compared by yaw servo amplifier comparator J110. If a malfunction occurs in either of the two separate and identical channels, yaw servo amplifier comparator J110 automatically detects the malfunction and causes the normally closed relay contacts in the J113 output path to open and the open relay contacts in the J114 output path to close. The servo drive signal from yaw 50 ma servo amplifier J114 is then applied to the yaw servo actuator in the S-IVB Stage.

The amplified $\Delta\phi P$ error signals from dc amplifiers J48, J49, and J33 are coupled through switching circuit A2 to $\Delta\phi P$ S-IVB filters J94, J95, and J41, respectively. These are bending mode filters which filter out angular pitch errors resulting from the physical deflections of the Saturn launch vehicle.

The $\Delta\phi P$ error signal outputs from the $\Delta\phi P$ filters are applied to their respective pitch 50 ma servo amplifier. Pitch 50 ma servo amplifier J103 through J105 then develop servo drive signals in proportion to the input error signals. However, only the servo drive signal from pitch 50 ma servo amplifier J104 is coupled through the normally closed contacts of the comparator relay to the pitch servo actuator in the S-IVB Stage. The servo drive signal causes the pitch servo actuator to mechanically reposition the S-IVB gimballed engine to correct for errors in the pitch axis.

If a malfunction occurs in either of the two separate and identical pitch channels, pitch servo amplifier comparator J107 automatically opens the

relay contacts in the J104 output and closes the relay contacts in the J105 output. The servo drive signal from pitch 50 ma servo amplifier J105 is then applied to the pitch servo actuator in the S-IVB Stage.

Roll errors are corrected by the two spatial thrusters on the S-IVB Stage. This is necessary because the S-IVB Stage obtains its power from a single gimballed engine and roll errors cannot be effectively corrected with a single engine.

The amplified $\Delta\phi_R$ error signals from dc amplifiers J34, J52, and J53 are applied to triple redundant roll-yaw I spatial amplifiers J65 through J67. The roll-yaw I spatial amplifiers will then develop spatial thruster control signals in proportion to the input error signals. However, only the spatial thruster control signal from roll-yaw I spatial amplifier J66 is coupled through the normally closed contacts of the comparator relay to spatial synchronizer J70. The spatial thruster control signal is simultaneously applied to control relay packages I and II in the S-IVB Stage by spatial synchronizer J70. The control relay packages then activate their associated spatial thruster to correct for errors in the roll axis.

Roll-yaw I spatial comparator J68 compares the outputs of roll-yaw I spatial amplifiers J65 and J66. If a malfunction occurs in either of the two separate and identical channels, roll-yaw I spatial comparator J68 will automatically open the closed relay contact in the J66 output and close the relay contact in the J67 output. The control signal for control relay packages I and II are then applied from roll-yaw I spatial amplifier J67.

The angular rate error signals (ϕ_Y , ϕ_P , and ϕ_R) are supplied by the Control-EDS Rate Gyro Package through the Control Signal Processor. The ϕ_Y and ϕ_P angular rate error signals are each coupled through switching circuit A3 and applied to triple redundant bending mode filters. ϕ_Y is applied to ϕ_Y S-IVB filters J97, J96, and J109; and ϕ_P is applied to filters J99, J98, and J108. By means of the switchpoint signals 4 and 5 from the LVDA (applied through the Switch Selector), the filters adjust the amplitude of the ϕ_Y and ϕ_P angular rate error signals in relation to the phase of the flight.

The ϕ_Y angular rate error signals from filters J97, J96, and J109 are applied to yaw 50 ma servo amplifiers J112 through J114, respectively. The ϕ_Y error signal is then algebraically added to the $\Delta\phi_P$ attitude error signal by the yaw 50 ma servo amplifiers to produce the yaw servo drive signal.

The ϕ_P angular rate error signals from filters J99, J98, and J108 are applied to pitch 50 ma servo amplifiers J103 through J105, respectively. The ϕ_P error signal is then algebraically added to the $\Delta\phi_P$ attitude error signal by the pitch 50 ma servo amplifiers to produce the pitch servo drive signal.

The ϕ_R error signal is coupled through switching circuit A3 and applied to dc amplifiers J37 through J39. The amplified ϕ_R error signals from the dc amplifiers are applied to triple redundant roll-yaw I spatial amplifiers J65 through J67. The ϕ_R angular rate error signals are then algebraically added to the $\Delta\phi_R$ attitude error signals by the roll-yaw I spatial amplifiers.

2-6 FUNCTIONAL OPERATION DURING S-IVB COAST MODE

At the completion of the S-IVB burn mode, the gimballed engine in the S-IVB Stage is shut down. Positioning of the Saturn IB launch vehicle is then controlled by the spatial thrusters located on the S-IVB Stage. This is called the S-IVB coast mode. Figure 2-4 illustrates the functional operation of the Flight Control Computer during the S-IVB coast mode. As in the S-IVB burn mode, triple redundancy circuits are used for the S-IVB coast mode to achieve a high reliability.

Redundant relay switching is also used throughout the Flight Control Computer to achieve a high degree of reliability. To simplify the functional block diagram of Figure 2-4, redundant relay contacts are shown as a single set of contacts and are identified as to which relays they represent.

Attitude error signals ($\Delta\phi_Y$, $\Delta\phi_P$, and $\Delta\phi_R$) for the S-IVB coast mode are supplied by the LVDC/LVDA or Apollo Spacecraft in the same manner as for the S-IVB burn mode. The amplified $\Delta\phi_Y$ attitude error signals from redundant dc amplifiers J46, J47, and J32 are coupled through switching circuit A2 and simultaneously applied to triple redundant roll-yaw I spatial amplifiers J65 through J67 and triple redundant roll-yaw II spatial amplifiers J71 through J73. The roll-yaw I and II spatial amplifiers then develop spatial thruster control signals in proportion to the input error signals. However, only the spatial thruster control signals from roll-yaw I spatial amplifier J66 and roll-yaw II spatial amplifier J72 are normally used. The control signal from J66 is coupled through the normally closed contacts of the comparator relay to control relay package I, and the control signal from J72 is coupled through the normally closed contacts of the comparator relay to

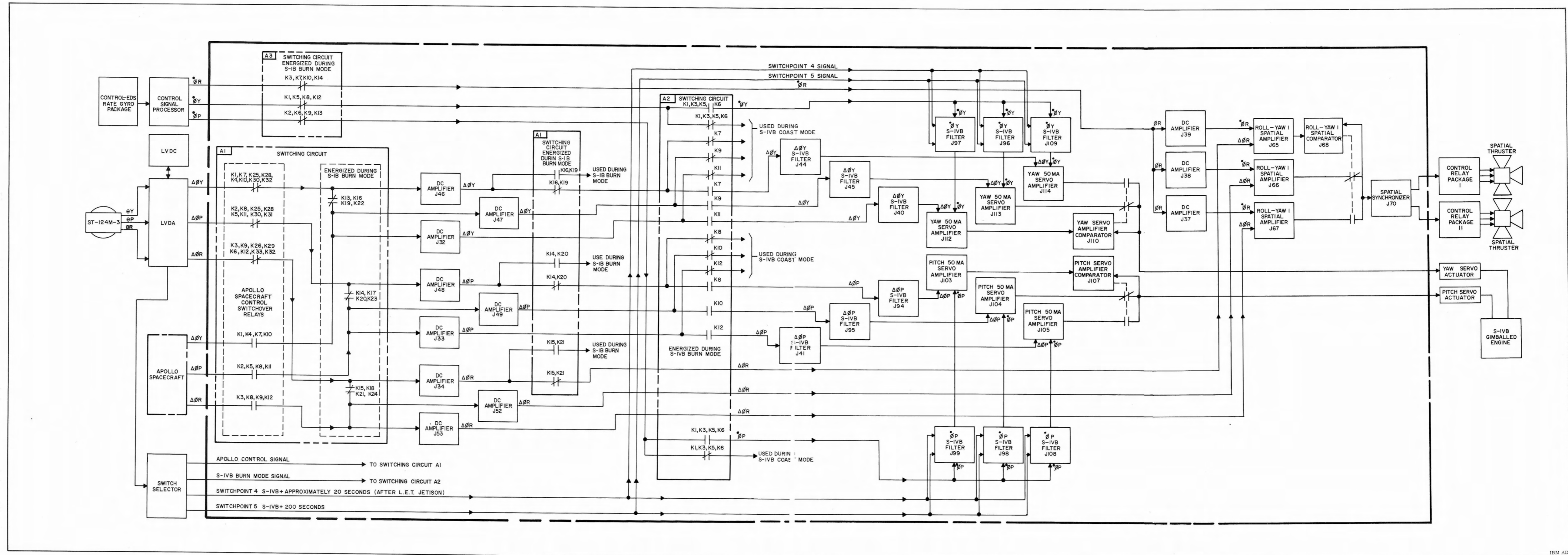


Figure 2-3. S-IVB Burn Mode, Functional Block Diagram

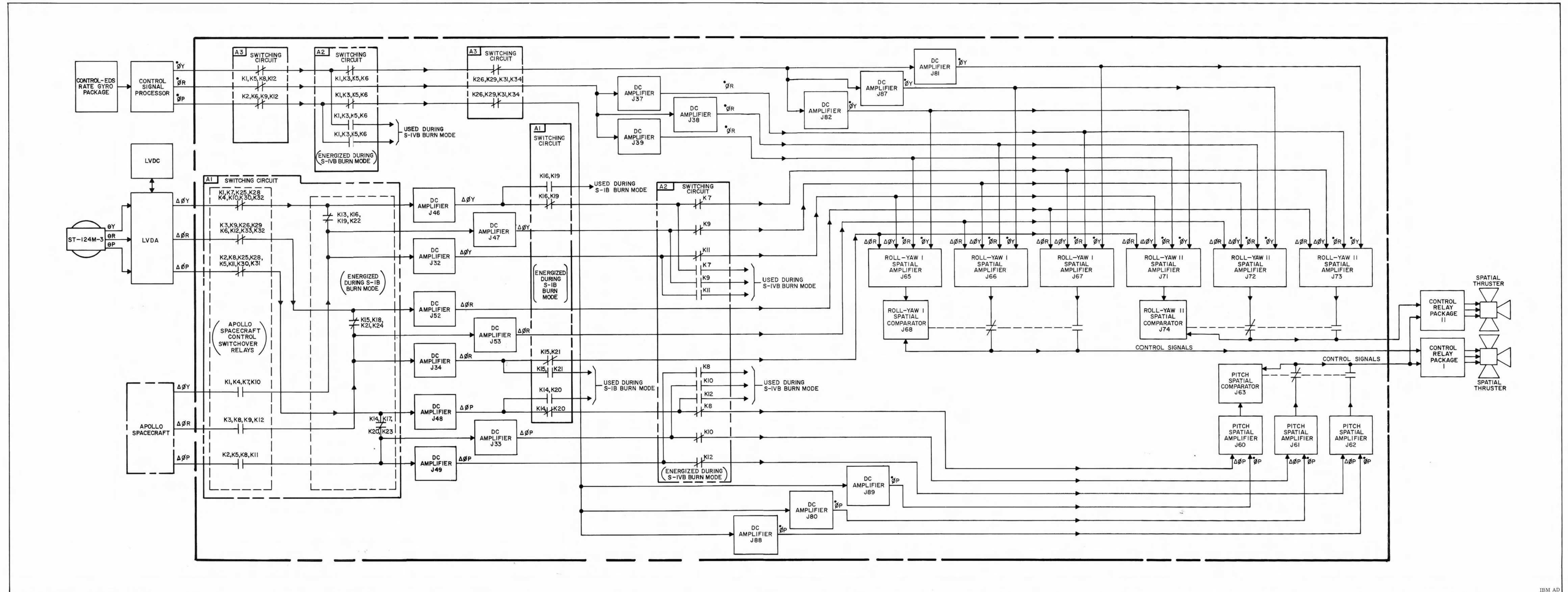


Figure 2-4. S-IVB Coast Mode, Functional Block Diagram

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control relay package II. The control relay packages then activate their associated spatial thruster to correct for errors in the yaw axis.

Roll-yaw I spatial comparator J68 compares the outputs of roll-yaw I spatial amplifiers J65 and J66. If a malfunction occurs in either of the two separate and identical channels, roll-yaw I spatial comparator J68 automatically opens the closed relay contact in the J66 output and closes the relay contact in the J67 output. The control signal for control relay package I is then applied through roll-yaw I spatial amplifier J67.

Roll-yaw II spatial comparator J74 compares the outputs of roll-yaw II spatial amplifiers J71 and J72. If a malfunction occurs in either of the two separate and identical channels, roll-yaw II spatial comparator J74 automatically opens the closed relay contact in the J72 output and closes the relay contact in the J73 output. The control signal for control relay package II is then applied through roll-yaw II spatial amplifier J73.

The amplified $\Delta\phi R$ error signals from redundant dc amplifiers J34, J52, and J53 are applied to triple redundant roll-yaw I spatial amplifiers J65 through J67 and triple redundant roll-yaw II spatial amplifiers J71 through J73. The $\Delta\phi R$ error signals are algebraically added to the $\Delta\phi Y$ error signals by the roll-yaw I and II spatial amplifiers to produce control signals for the control relay packages.

The amplified $\Delta\phi P$ error signals from redundant dc amplifiers J48, J49, and J33 are coupled through switching circuit A2 to pitch spatial amplifiers J60, J61, and J62. The pitch spatial amplifiers develop spatial thruster control signals in proportion to the input error signal. However, only the spatial thruster control signal from pitch spatial amplifier J61 is coupled through the closed contacts of the comparator relay and simultaneously applied to control relay package I and control relay package II. The control relay packages then actuate their associated spatial thruster to correct for errors in the pitch axis.

Pitch spatial comparator J63 compares the output of pitch spatial amplifiers J60 and J61. If a malfunction occurs in either of the two separate and identical channels, pitch spatial comparator J63 automatically opens the closed relay contacts in the J61 output and closes the relay contact in the J62 output. The control signal for the control relay packages is then applied through pitch spatial amplifier J62.

The angular rate error signals (ϕY , ϕP , and ϕR) are supplied by the Control-EDS Rate Gyro Package through the Control Signal Processor. The ϕY error signal is coupled through switching circuits A2 and A3 to dc amplifiers J81, J82, and J87. The amplified ϕY error signals from the dc amplifiers are simultaneously applied to their respective triple redundant roll-yaw I spatial amplifiers and the triple redundant roll-yaw II spatial amplifiers where they are algebraically added to existing yaw and roll error signals.

The ϕR error signal is coupled through switching circuit A3 to redundant dc amplifiers J37 through J39. The amplified ϕR error signals from the dc amplifiers are simultaneously applied to their respective triple redundant roll-yaw I spatial amplifiers and the roll-yaw II spatial amplifiers, where they are algebraically added to existing yaw and roll error signals.

The ϕP error signal is coupled, through switching circuits A2 and A3, to dc amplifiers J80, J88, and J89. The amplified ϕP error signals from the dc amplifiers are applied to their respective triple redundant pitch spatial amplifiers where they are algebraically added to the $\Delta\phi P$ attitude error signals.

2-7 SUBASSEMBLY FUNCTIONAL DESCRIPTIONS

The following paragraphs contain functional descriptions of the subassemblies contained within the Flight Control Computer. The subassemblies are discussed in the following order:

- DC Amplifier
- Bending Mode Filters
- 12 Ma Servo Amplifier
- Control Attenuator Timer
- Inverter
- Servo Inverter Detector
- Beta Command Power Supply
- 50 Ma Servo Amplifier
- Servo Amplifier Comparator
- Spatial Amplifier
- Spatial Comparator

- Spatial Synchronizer
- TM Amplifier
- S-IB Power Distribution Assembly
- Switching Circuits

DC Amplifier

The Flight Control Computer contains 22 identical dc amplifiers which are used throughout the Flight Control Computer to amplify yaw, pitch, and roll error signals. The yaw, pitch, and roll error signals are carried on isolated, balanced lines. To maintain this isolation and provide maximum amplification with relatively little distortion, each side of the balanced lines is amplified by the dc amplifiers. A photograph of the dc amplifier subassembly is shown in Figure 2-5. A functional block diagram of the dc amplifier is shown in Figure 2-6.

The balanced error signal input is applied to push-pull differential amplifier Q5, which amplifies the balanced error signal and provides compensation for fluctuations in current due to component, or temperature variations. The output of Q5 is further amplified by push-pull amplifier Q7 and push-pull amplifier Q9 and applied to bridge connected complementary symmetry amplifier Q10, Q11, Q12. Amplifier Q10, Q11, Q12 provides maximum undistorted amplification while maintaining ground isolation for the output signal. The amplified balanced error signal output is taken from the junction of Q10A and Q11 and the junction of Q10B and Q12.

Temperature changes which will effect the overall gain of the dc amplifier are sensed by temperature compensators Q6 and Q8. Q6 and Q8 control the gain of the dc amplifier by varying the bias level on Q5 and Q7 in proportion to the temperature change.

In order to develop a balanced error signal output of sufficient amplitude, the amplifier stages in the dc amplifier require a higher voltage source than the available 28 Vdc power source. To accommodate this power requirement, each of the dc amplifiers contain an inverter and regulating circuit which converts the 28 Vdc source to 60 Vdc.

The 28 Vdc source is applied to the inverter Q1, Q2, which converts the 28 Vdc source to a 5 kHz signal of sufficient amplitude. This 5 kHz signal is then rectified by bridge rectifier CR5, CR7, CR8, and CR9. The resulting stepped-up dc voltage is then routed through series regulator Q4 to the amplifier circuits. A sample of the output voltage is compared in comparator-amplifier Q3 with a stable reference voltage. A regulator control voltage, produced by this comparison, is applied to Q4. This control voltage controls the internal impedance, and consequently the voltage drop of series regulator Q4. If the 60 Vdc output voltage varies, the regulator control voltage varies proportionally. The variation in control voltage produces a corresponding variation in the voltage drop across Q4 and returns the output voltage to the proper level.

The circuits in which the dc amplifiers are used have various gain requirements. In order to

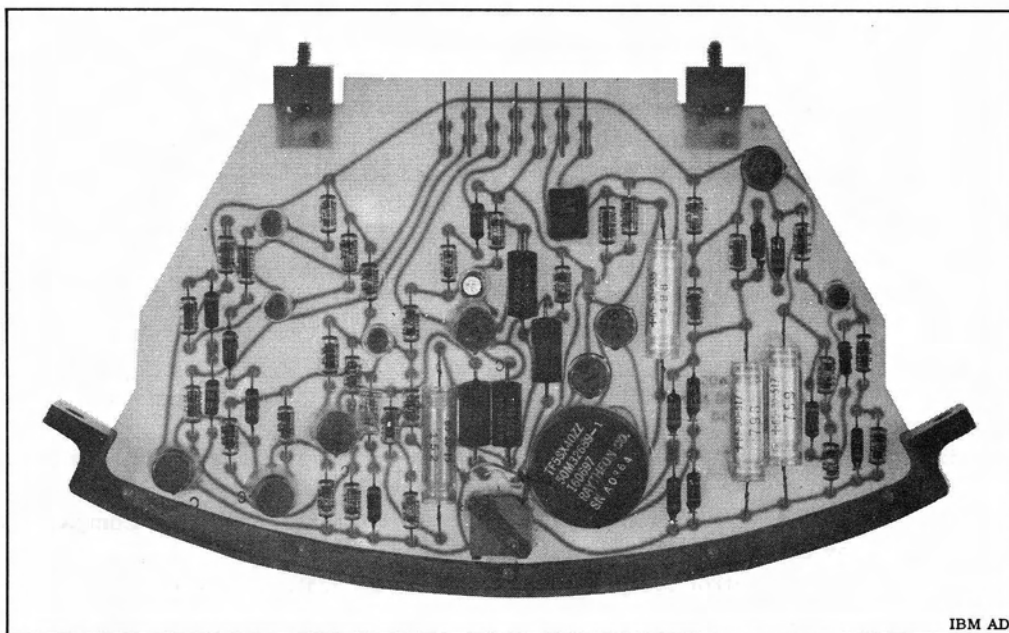


Figure 2-5. DC Amplifier Subassembly

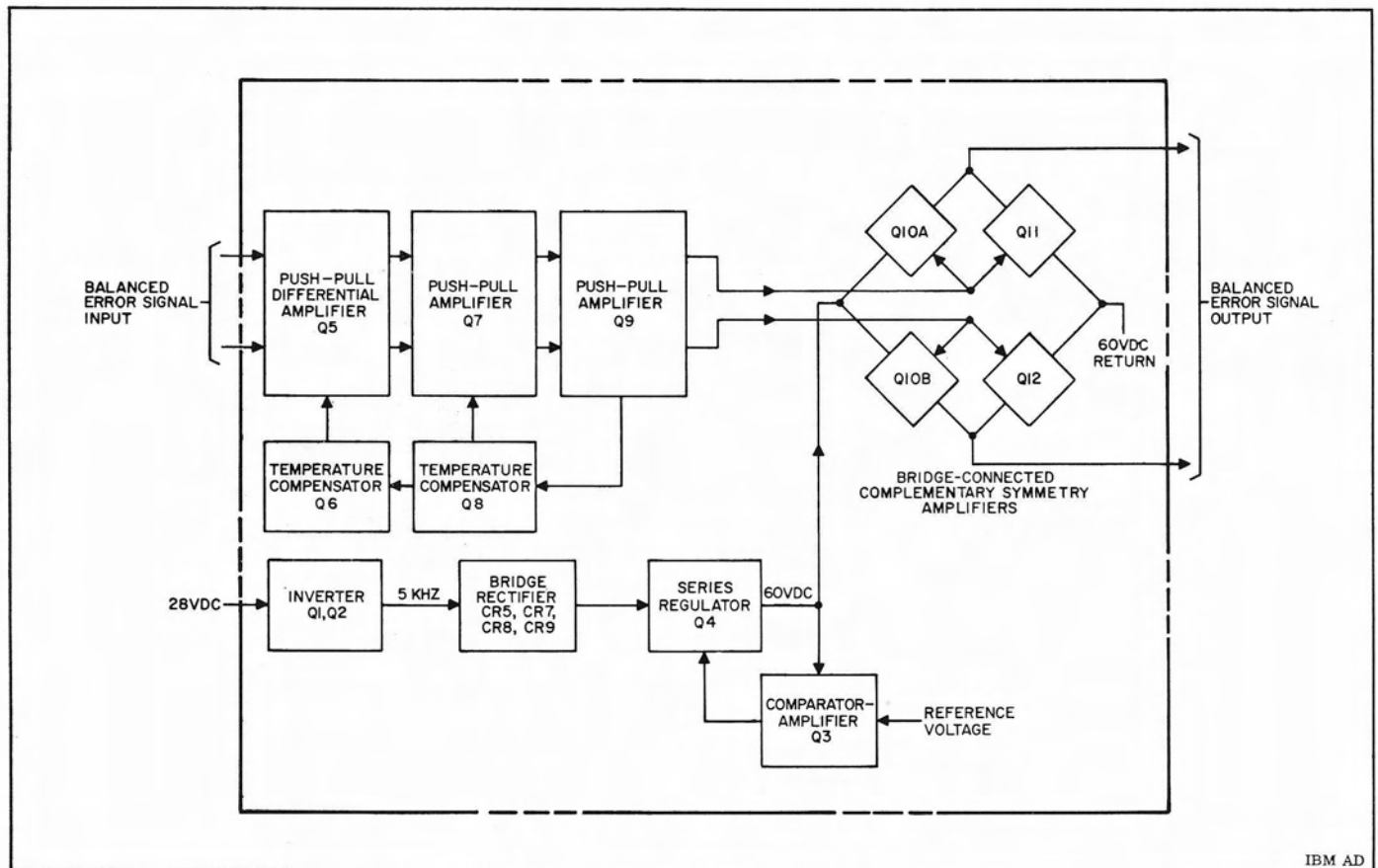


Figure 2-6. DC Amplifier, Functional Block Diagram

make a standard dc amplifier compatible with all of these circuits, a scaling board (See Figure 2-7) is associated with each dc amplifier. The scaling boards consist of selected summing and feedback resistors which determine the overall gain of the dc amplifier.

Bending Mode Filters

The Flight Control Computer contains 18 bending mode filters located in the signal lines throughout the Flight Control Computer. Photographs of typical filters are shown in Figures 2-8 through 2-13. A functional block diagram of a typical bending mode filter is shown in Figure 2-14.

The bending mode filters for vehicles AS-204 and AS-205 are functionally identical. However, the frequency range of the AS-204 and AS-205 filters are different due to the inherent differences in the structure of the vehicles.

Each bending mode filter contains a low pass filter and an attenuation network. The low pass filter filters out position error signals resulting from the physical deflections of the Saturn launch vehicle structure. These deflections occur when the Saturn

launch vehicle frame vibrates at its resonant frequency during high thrust loadings.

The low pass filter output is then coupled through an attenuation network which controls the level of the output signal in proportion to the phase of the flight. Maximum attenuation is provided by attenuating resistors A. As the flight progresses, relay A is energized by a switchpoint signal from the Switch Selector. This action places attenuating resistors B in parallel with attenuating resistors A, decreasing the amount of attenuation. As the flight progresses still further, relays B and C are sequentially energized by switchpoint signals from the Switch Selector. Each time, the amount of attenuation is decreased.

The basic difference between bending mode filters is the number of switchpoints. For the S-IB burn mode, the $\Delta\phi Y$ S-IB, $\Delta\phi P$ S-IB, ϕY S-IB, and ϕP S-IB filters each have three switchpoints, and the $\Delta\phi R$ S-IB and ϕR S-IB filters each have one switchpoint. For the S-IVB burn mode, the ϕP S-IVB and ϕY S-IVB filters each have two switchpoints. The $\Delta\phi Y$ S-IVB and $\Delta\phi P$ S-IVB filters do not require time sequenced attenuation control; however, each filter has a fixed attenuation network.

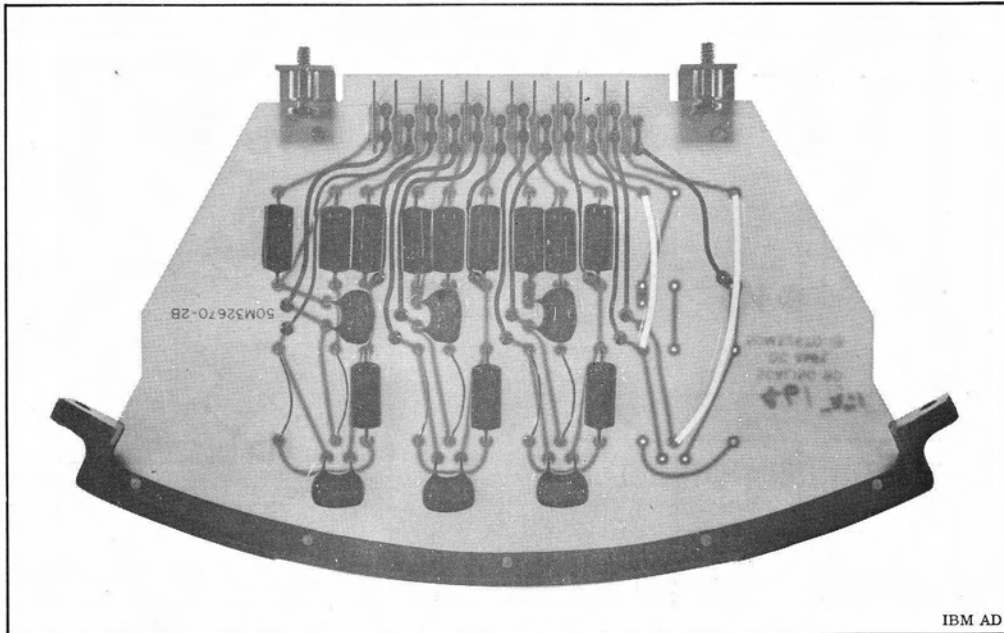


Figure 2-7. Scaling Board

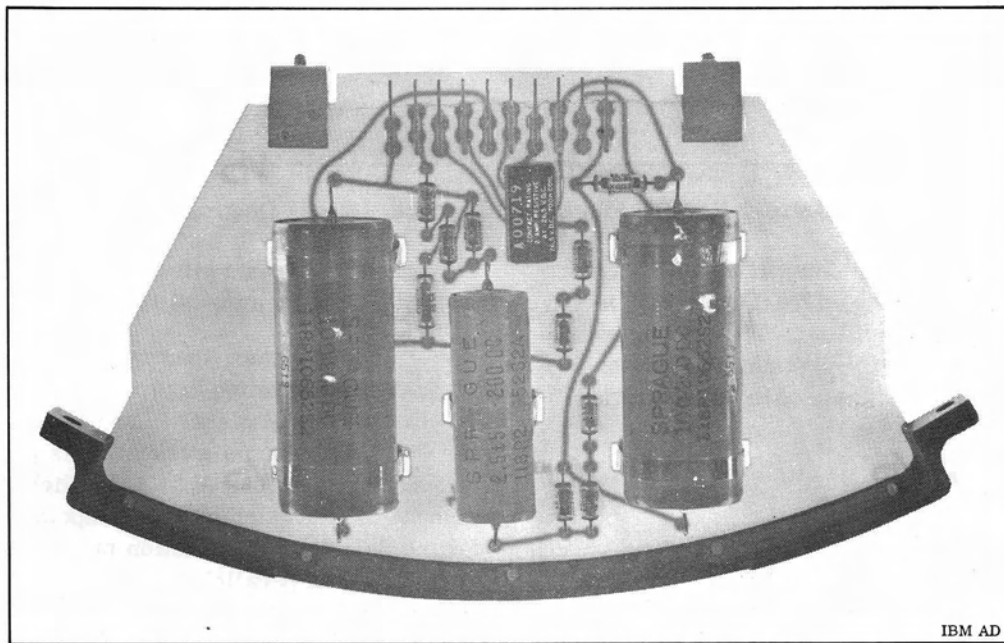


Figure 2-8. γY , γP Filter (Typical)

12 Ma Servo Amplifier

The Flight Control Computer contains four yaw 12 ma servo amplifiers and four pitch 12 ma servo amplifiers. The 12 ma servo amplifiers control the servo cylinders in the S-IB Stage by producing drive signals in proportion to the combined attitude, angular rate, and lateral acceleration error signal inputs. The servo cylinders mechanically position the four gimballed engines in the S-IB Stage in proportion to the drive signals received from the 12 ma servo

amplifier. A photograph of the 12 ma servo amplifier subassembly is shown in Figure 2-15. A functional block diagram of the 12 ma servo amplifier is shown in Figure 2-16.

Attitude, angular rate, and lateral acceleration error signals are applied to coil windings in magnetic amplifier Q1, Q2. The magnetic amplifier algebraically combines the attitude, angular rate, and lateral acceleration error signals and produces a proportional, balanced output. The balanced output is

combines the attitude error, angular rate, and lateral acceleration signals and produces a proportional, balanced output. The balanced output is then amplified by push-pull amplifier Q4, push-pull amplifier Q5, and push-pull amplifier Q6, Q7. The collector current for Q6, Q7 is routed through the feedback developing resistor network and solenoid coil windings in the S-IB Stage hydraulic actuator. Thus when an error signal is applied to push-pull amplifier Q6, Q7, the push-pull amplifier collector currents operate the S-IB Stage hydraulic actuator. A portion of this drive signal is developed across the feedback developing resistors and applied to magnetic amplifier L1 as a gain control feedback.

The 66 V_{p-p}, 1150 Hz excitation signal for magnetic amplifier L1 is received from the servo inverter detector and coupled through transformer T1 to the magnetic amplifier. The 66 V_{p-p} 1150 Hz output signal from T1 is also rectified by bridge rectifier CR1 through CR4 to provide dc power for the 12 ma servo amplifier. Operating level bias for magnetic amplifier L1 is provided by bias control amplifier Q3.

A dc position feedback signal from the position indicator potentiometer on the S-IB Stage hydraulic cylinder, is coupled through the beta feedback subassembly to magnetic amplifier L1. The beta feedback subassembly (see Figure 2-16) provides for adjustment of the dc position feedback signal. The dc position feedback signal opposes the error signal in magnetic amplifier L1 to prevent overcorrection of the gimballed engines in the S-IB Stage.

2-11 CONTROL ATTENUATOR TIMER

The Control Attenuator Timer (CAT) controls the presence and amplitude of the yaw and pitch lateral acceleration signals (\ddot{y}_Y and \ddot{y}_P) applied to the 12 ma servo amplifier during the S-IB Stage burn mode. A photograph of the CAT subassembly is shown on Figure 2-17. A block diagram of the CAT is shown on Figure 2-18.

The CAT consists of switches S1 through S3 and potentiometers R1 through R4 (R1 and R2 are not used) which are mechanically positioned by motor B1. At liftoff, a first motion signal is coupled through the closed contacts of switch S2 to the S-IB power distribution assembly. At the same time, 115 V, 400 Hz from the CAT inverter is applied to B1, causing B1 to begin rotating and switch S1 to close. The closing of S1 applies a CAT zero indication signal (28 Vdc) to the Ground Support Equipment (GSE).

Thirty seconds into the flight (T_1 plus 30 seconds), the rotation of B1 causes switch S3 to close, which applies +28 Vdc to relays K1 in the \ddot{y}_Y and \ddot{y}_P filters. Energizing relays K1 in the \ddot{y}_P and \ddot{y}_Y filters, allow the \ddot{y}_P and \ddot{y}_Y signals to be applied to the 12 ma servo amplifiers through CAT potentiometers R3 and R4, respectively. The amplitude of the \ddot{y}_P and \ddot{y}_Y signals during the S-IB burn mode are controlled by the position of the wiper arms of R3 and R4. These wipers are positioned by motor B1. At T_1 plus 30 seconds, R3 and R4 provide maximum signal attenuation; between T_1 plus 30 seconds and T_1 plus 60 seconds the attenuation is steadily decreased; from T_1

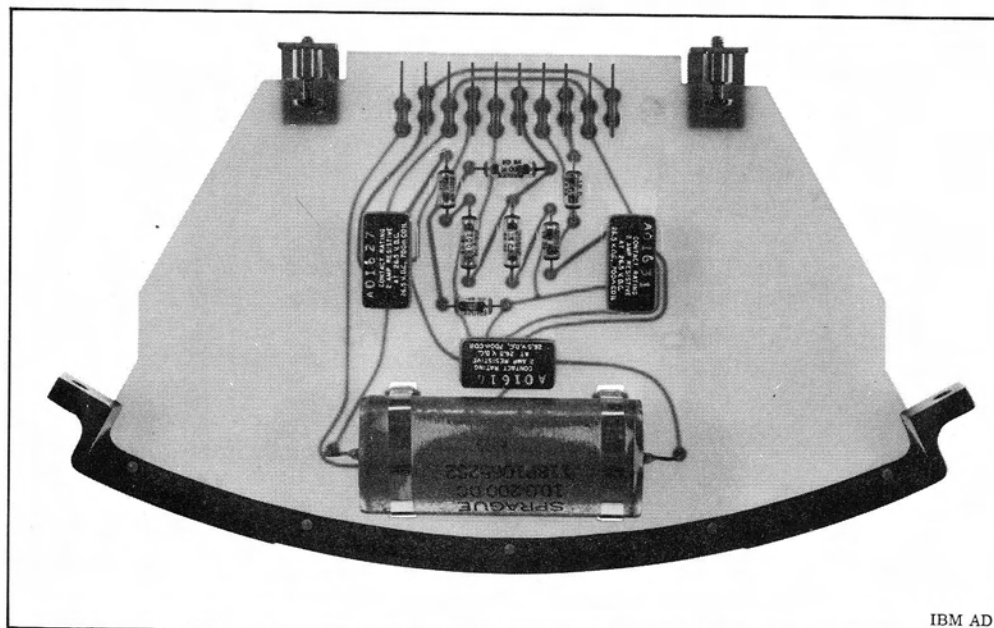
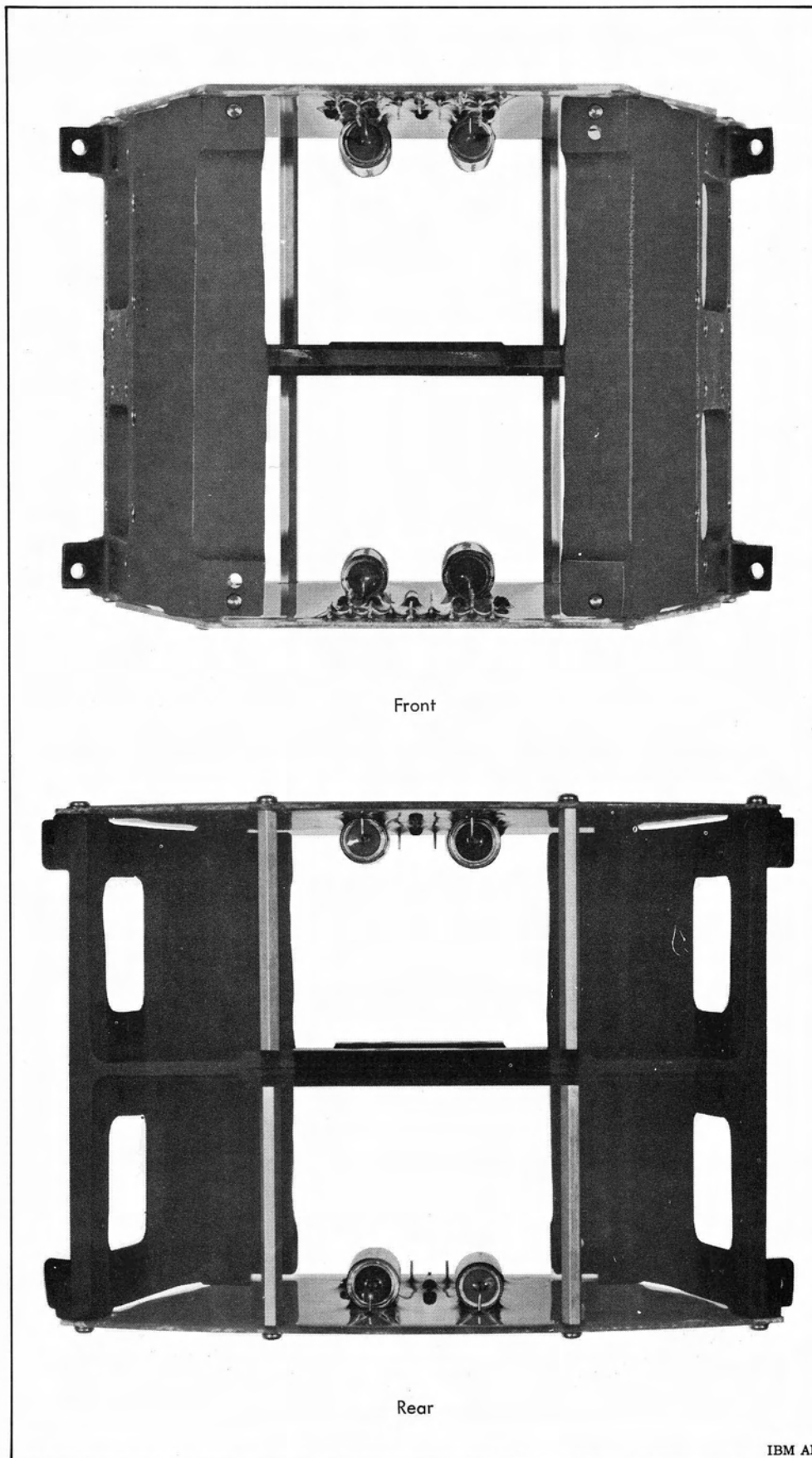
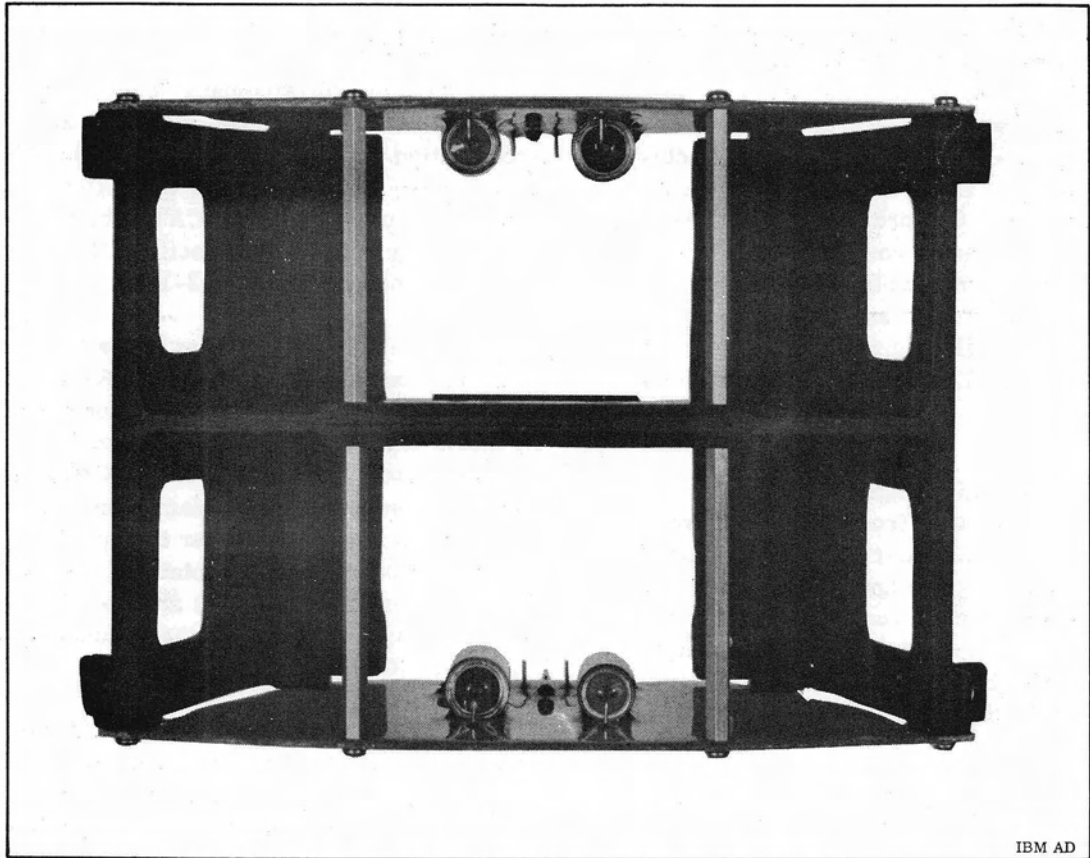


Figure 2-9. $\Delta\phi Y$, $\Delta\phi P$ S-IB Filter (Typical)



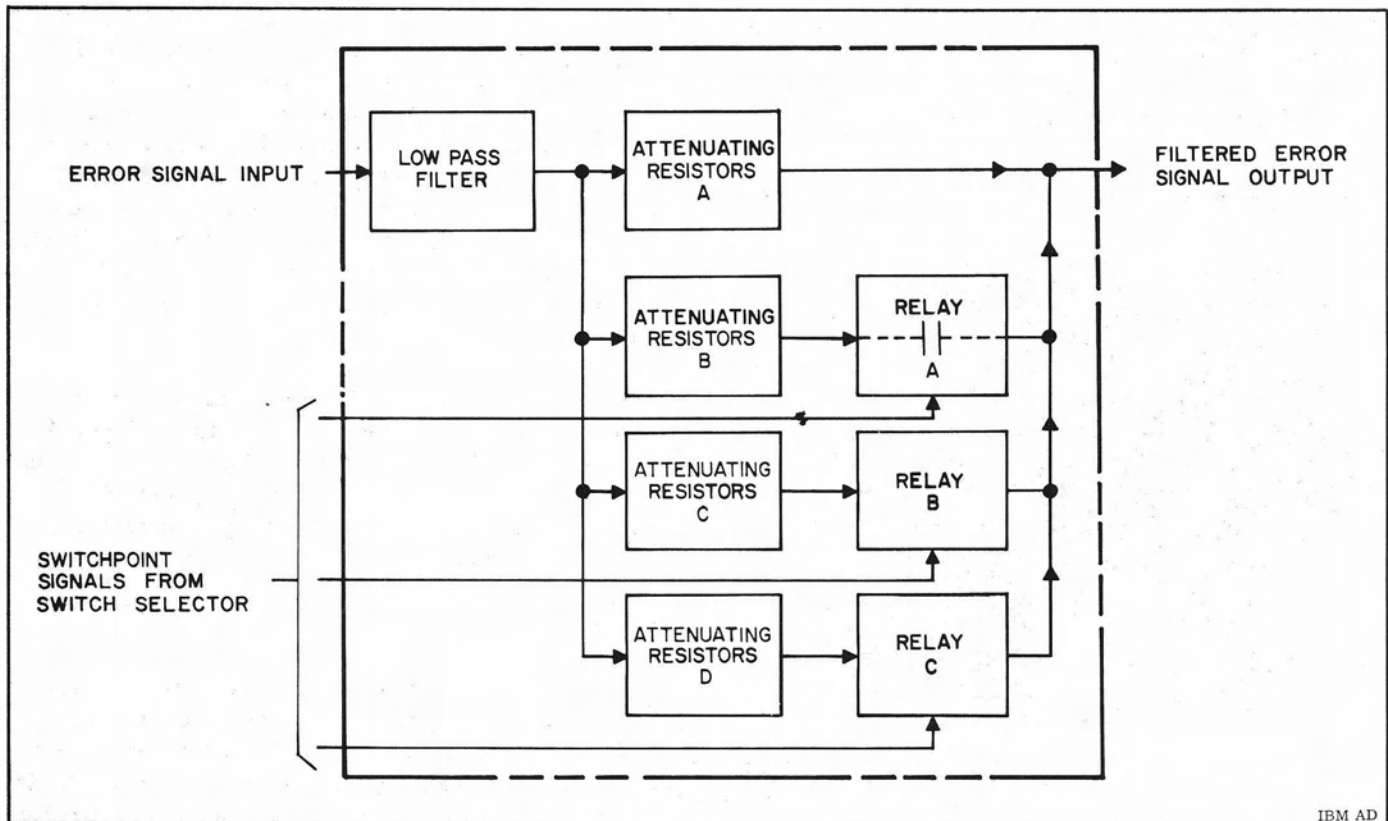
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Figure 2-10. $\Delta\phi Y$, $\Delta\phi P S$ -IVB Filter (Typical)



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Figure 2-13. $\phi Y, \phi P$ SIV Filter (Typical)



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Figure 2-14. Typical Bending Mode Filter, Functional Block Diagram

then amplified by push-pull amplifier Q4, push-pull amplifier Q5, and push-pull amplifier Q6, Q7. The collector current for Q6, Q7 is routed through the feedback developing resistor network and solenoid coil windings in the S-IB Stage servo cylinder actuator valve. Thus when an error signal is applied to push-pull amplifier Q6, Q7, proportional signals developed within the solenoid coil windings of the S-IB Stage servo cylinder actuator valve operate the valve. A portion of this drive signal is developed across the feedback developing resistors and applied to magnetic amplifier Q1, Q2 as a gain control feedback.

The 1 kHz excitation signal for magnetic amplifier Q1, Q2 is received from the servo inverter and coupled through transformer T1 to the magnetic amplifier. The 1 kHz output signal from transformer T1 is also rectified by bridge rectifier CR1 through CR4 to provide dc power for the 12 ma servo amplifier. Operating level bias for magnetic amplifier Q1, Q2 is provided by bias control amplifier Q3.

A dc position feedback signal from the position indicator potentiometer on the S-IB Stage servo cylinder, is coupled through the beta feedback subassembly to magnetic amplifier Q1, Q2. The beta feedback subassembly (see Figure 2-17) provides for adjustment of the dc position feedback signal. The dc position feedback signal opposes the error signal in magnetic amplifier Q1, Q2 to prevent over-correction of the gimballed engines in the S-IB Stage.

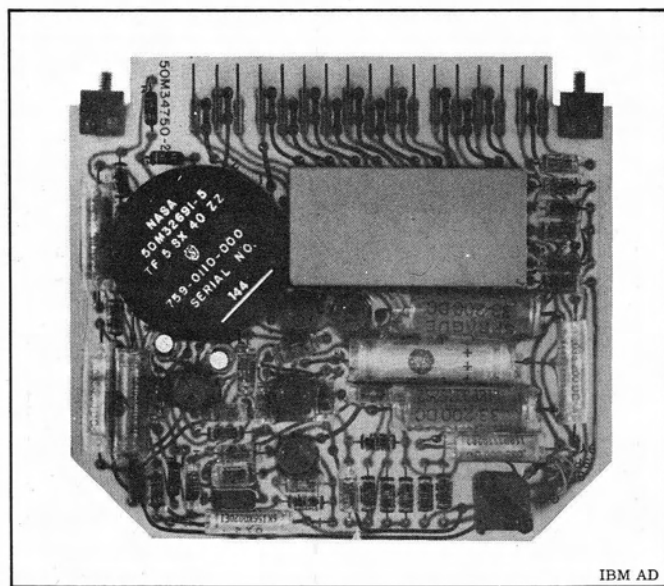


Figure 2-15. 12 Ma Servo Amplifier Subassembly

Control Attenuator Timer

The Control Attenuator Timer (CAT) controls the presence and amplitude of the yaw and pitch lateral acceleration error signals ($\ddot{\gamma}Y$ and $\ddot{\gamma}P$) applied to the 12 ma servo amplifier during the S-IB Stage burn mode. A photograph of the CAT subassembly is shown in Figure 2-18. A functional block diagram of the CAT is shown in Figure 2-19.

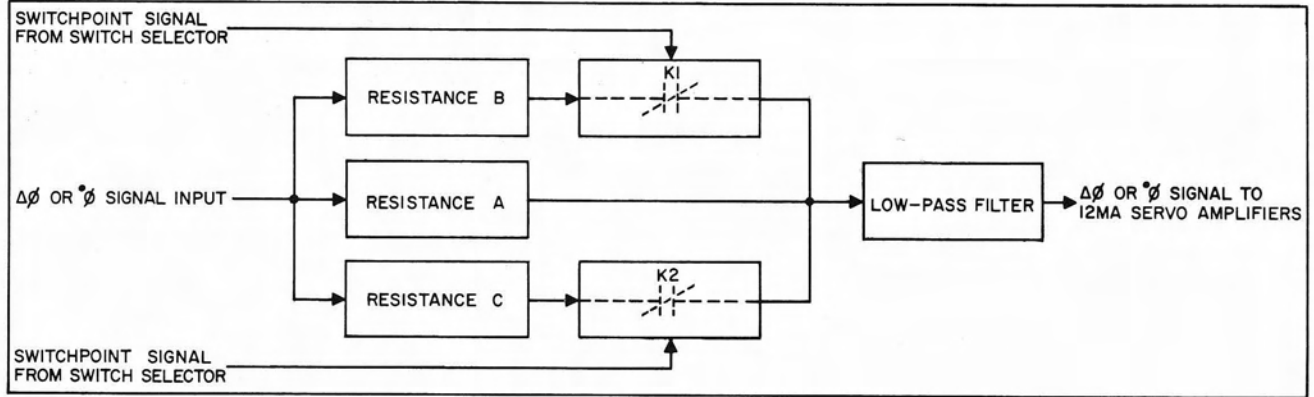
The CAT consists of switches S1 through S3 and potentiometers R1 through R4 (R1 and R2 are not used) which are mechanically positioned by motor B1. At liftoff, a first motion signal is coupled through the closed contacts of switch S2 to the S-IB power distribution assembly. At the same time, 115 Vac, 400 Hz from the CAT inverter is applied to motor B1, causing motor B1 to begin rotation and close switch S1. The closing of switch S1 applies a CAT zero indication signal (28 Vdc) to the Ground Support Equipment (GSE) via telemetry.

Thirty seconds into the flight (T plus 30 seconds), the rotation of motor B1 causes switch S3 to close, which applies +28 Vdc to relays K1 in the $\ddot{\gamma}Y$ and $\ddot{\gamma}P$ filters. The energizing of relays K1 in the $\ddot{\gamma}P$ and $\ddot{\gamma}Y$ filters allows the $\ddot{\gamma}P$ and $\ddot{\gamma}Y$ error signals to be applied to the 12 ma servo amplifiers through CAT potentiometers R3 and R4, respectively. The amplitude of the $\ddot{\gamma}P$ and $\ddot{\gamma}Y$ error signals during the S-IB burn mode are controlled by the position of the wiper arms of potentiometers R3 and R4. These wipers are positioned by motor B1. At T plus 30 seconds, potentiometers R3 and R4 provide maximum signal attenuation; between T plus 30 seconds and T plus 60 seconds the attenuation is steadily decreased; from T plus 60 seconds to T plus 80 seconds the least amount of attenuation is provided; and between T plus 80 seconds and T plus 110 seconds the attenuation again increases to maximum.

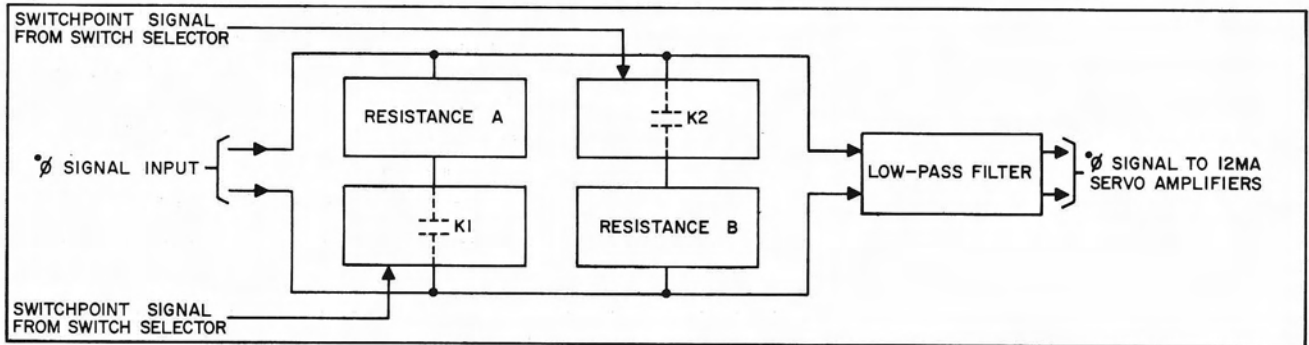
At T plus 120 seconds, the rotation of motor B1 causes switch S3 to open, deenergizing relays K1 in the $\ddot{\gamma}Y$ and $\ddot{\gamma}P$ filters. Between T plus 120 seconds and T plus 125 seconds, the rotation of motor B1 opens switch S2 and applies CAT zero command signal to the GSE via telemetry.

Inverter

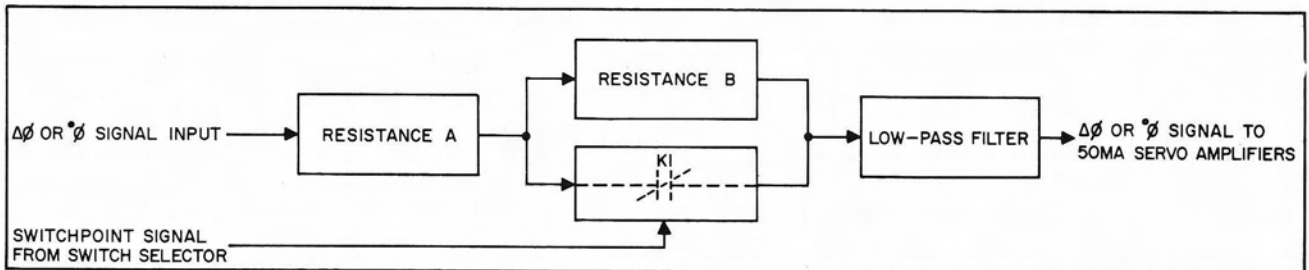
The Flight Control Computer contains a CAT inverter and a servo inverter, which differ only in output voltage and frequency. The CAT inverter converts the 28 Vdc primary power to the 115 Vac, 400 Hz required for operation of the CAT. The servo inverter



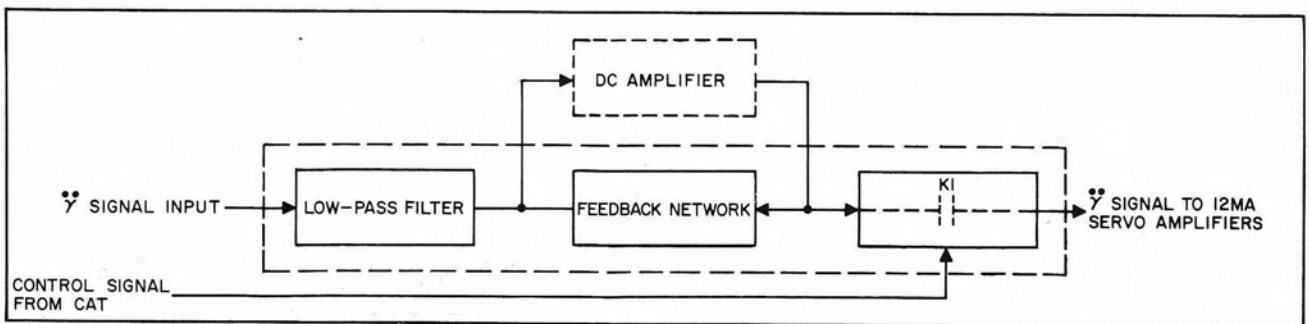
A. $\Delta\phi Y$ S-IB FILTER, $\Delta\phi P$ S-IB FILTER, $\Delta\phi R$ S-IB FILTER AND ϕR S-IB FILTER



B. ϕY S-IB FILTER AND ϕP S-IB FILTER



C. $\Delta\phi Y$ S-IVB FILTER, $\Delta\phi P$ S-IVB FILTER, ϕY S-IVB FILTER AND ϕP S-IVB FILTER



D. $\phi\phi Y$ FILTER AND $\phi\phi P$ FILTER

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Figure 2-13. Filter, Block Diagrams

plus 60 seconds to T_1 plus 80 seconds the least amount of attenuation is provided; and between T_1 plus 80 seconds and T_1 plus 110 seconds, the attenuation again increases to maximum.

At T_1 plus 120 seconds, the rotation of B1 causes switch S3 to open, deenergizing relays K1 in the $\dot{\gamma}Y$ and $\dot{\gamma}P$ filters. Between T_1 plus 120 seconds and T_1 plus 125 seconds, the rotation of B1 opens switch S2; this disconnects the +28 Vdc control signal from the relay in the S-IB power distribution assembly, which, in turn, disconnects the 115 V, 400 Hz power from B1. The CAT program is therefore terminated.

2-12 CONTROL ATTENUATOR TIMER INVERTER AND SERVO INVERTER

The Flight Control Computer contains a CAT inverter and two servo inverters which differ only in output voltage and frequency. The CAT inverter converts the 28 Vdc primary power to the 115V, 400 Hz required for operation of the CAT. The servo inverters convert the 28 Vdc primary power to the 66 V_{p-p}, 1150 Hz required for operation of the eight 12 ma servo amplifiers. Photographs of the CAT and servo inverters are shown on Figure 2-19. A block diagram of the inverters is shown on Figure 2-20.

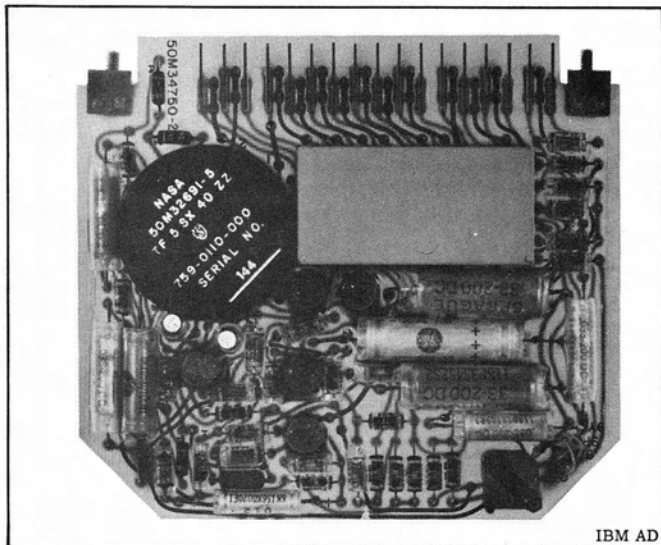


Figure 2-14. 12 Ma Servo Amplifier

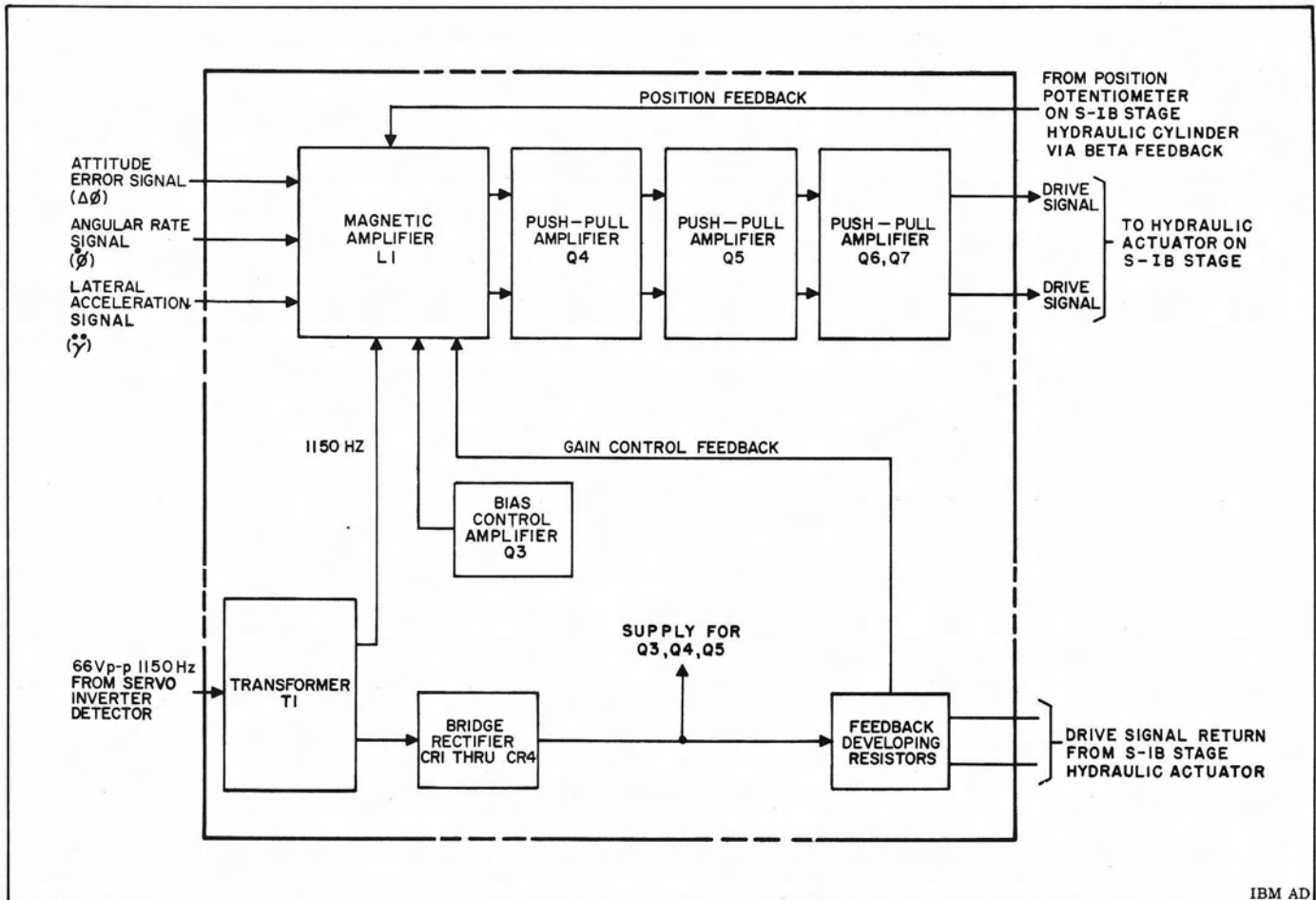


Figure 2-15. 12 Ma Servo Amplifier Block Diagram

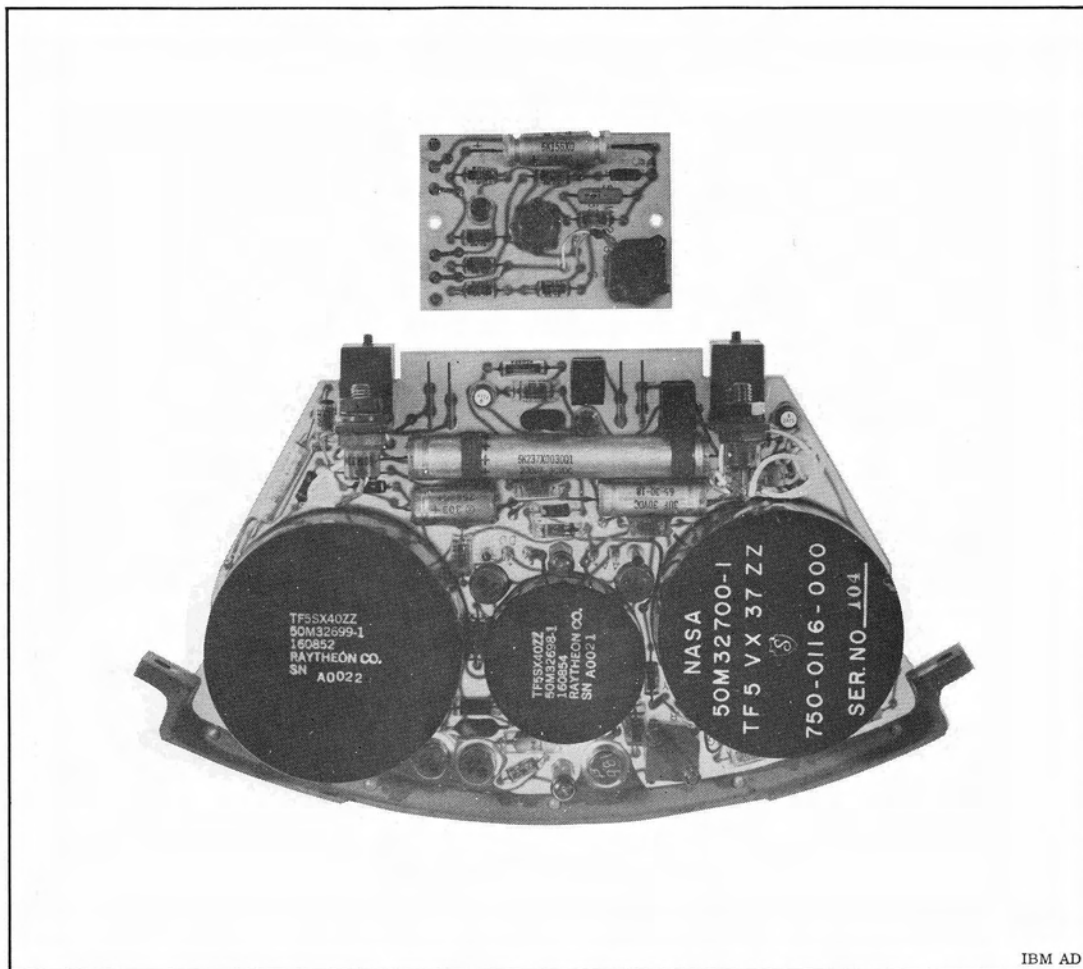


Figure 2-20. Control Attenuator Timer Inverter Subassembly

Regulation of the ac output level is provided by chopper Q8, which regulates the negative side of the 28 Vdc primary power applied to driver-amplifier Q6, Q7. Chopper Q8 intermittently applies the negative dc primary power to filter T3, C4 where it is filtered to produce the negative source for driver-amplifier Q6, Q7. The rate at which chopper Q8 switches is controlled by the output frequency of comparator-multivibrator Q10, Q11 which is coupled through driver Q9 to chopper Q8. The frequency of comparator-multivibrator Q10, Q11 is determined by the comparison of a reference voltage and the output of full wave rectifier CR7, CR8. When the ac output voltage of transformer T2 increases, the output of full wave rectifier CR7, CR8 will also increase, causing the frequency of comparator-multivibrator Q10, Q11 to decrease. The decreased frequency of comparator-multivibrator Q10, Q11 increases the OFF time of chopper Q8 and decreases the dc voltage applied to driver-amplifier Q6, Q7. Decreasing the dc voltage decreases the gain of driver-amplifier Q6, Q7. When the ac output decreases, the resulting frequency

increase of comparator-multivibrator Q10, Q11 decreases the OFF time of chopper Q8, and increases the dc voltage applied to driver-amplifier Q6, Q7. Increasing the dc voltage increases the gain of driver-amplifier Q6, Q7.

Servo Inverter Detector

The servo inverter detector is used to monitor the output of the servo inverter. If the servo inverter fails, the servo detector will automatically switch in a standby servo inverter. A photograph of the servo inverter detector subassembly is shown in Figure 2-23. A functional block diagram of the servo inverter is shown in Figure 2-24.

The 60 Vac, 1 kHz signal from servo inverter J56A is normally coupled through the closed contacts of switchover relay K1 to the 12 ma servo amplifiers. The 60 Vac, 1 kHz signal from servo inverter J94A is normally terminated at the open contacts of switchover relay K1. The ac signal from servo inverter

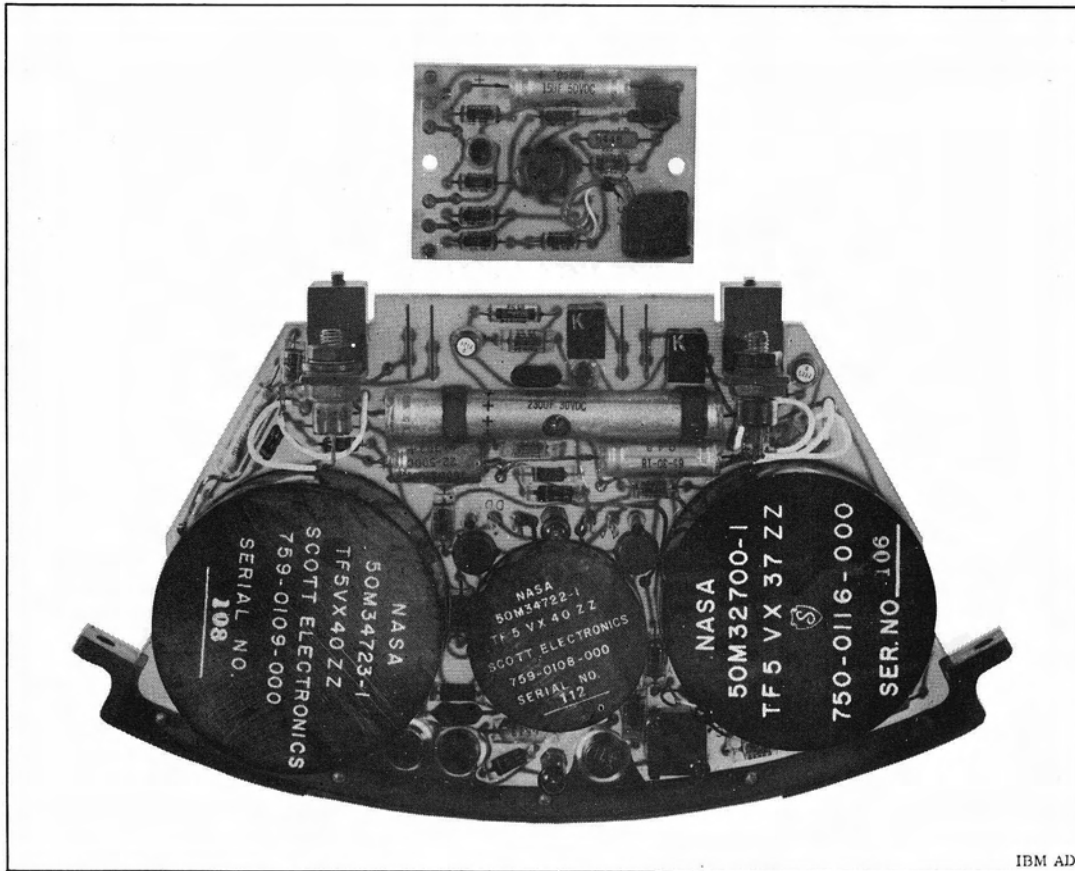


Figure 2-21. Servo Inverter Subassembly

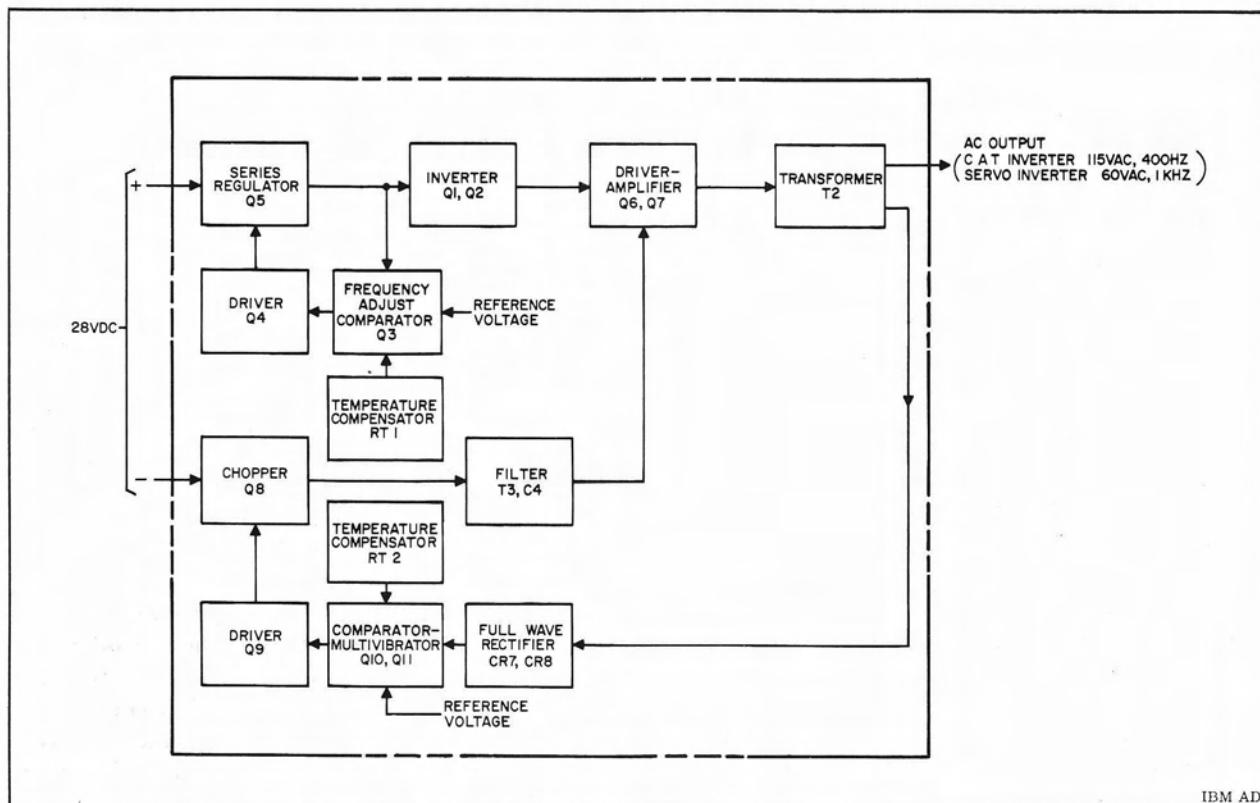


Figure 2-22. CAT and Servo Inverters, Functional Block Diagram

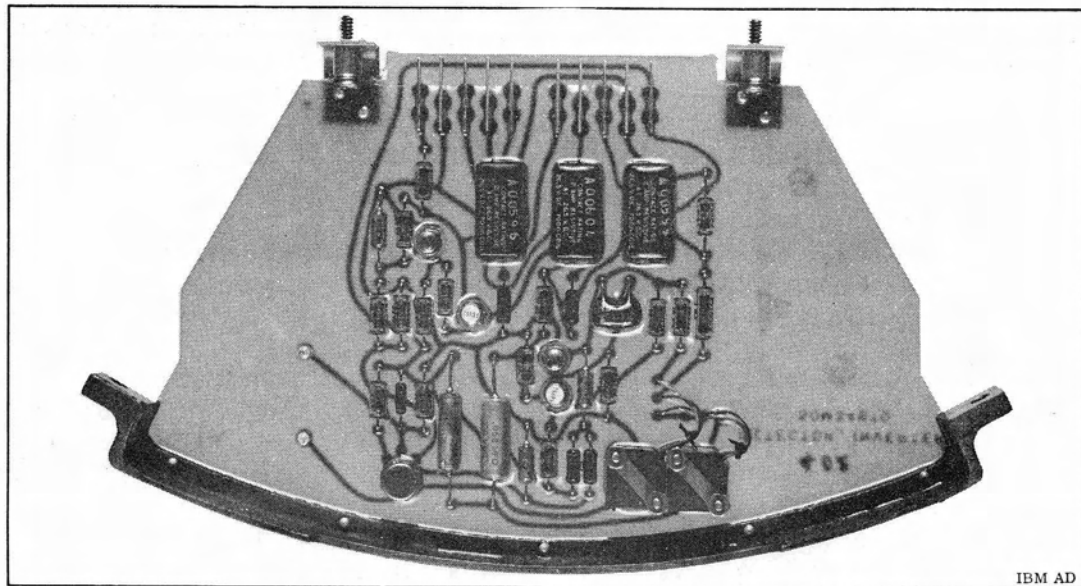


Figure 2-23. Servo Inverter Detector Subassembly

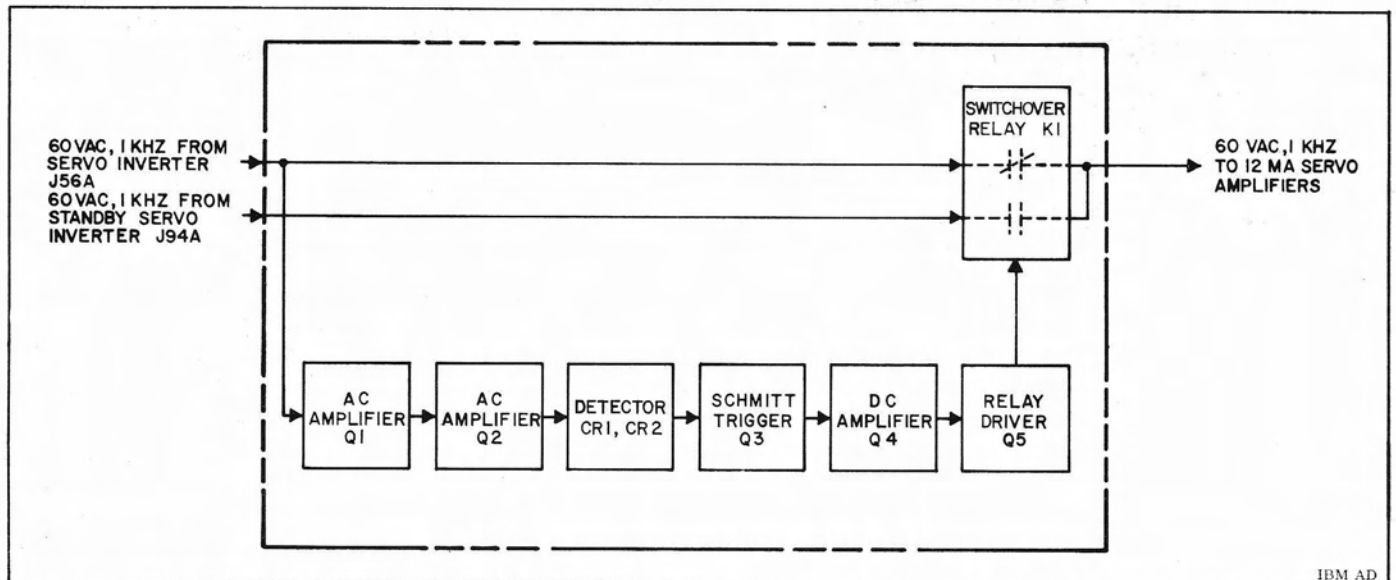


Figure 2-24. Servo Inverter Detector, Functional Block Diagram

J56A is also applied to ac amplifier Q1. The ac signal is then amplified by ac amplifiers Q1 and Q2 and applied to detector CR1 and CR2. Detector CR1 and CR2 converts the ac signal to a proportional dc voltage and applies it to schmitt trigger Q3.

Schmitt trigger Q3 is designed so that with no signal applied, it will cause dc amplifier Q4 and relay driver to conduct. Therefore, while a dc signal is applied to schmitt trigger Q3, Q4 and Q5 will be cutoff.

If servo inverter J56A fails, the dc signal will no longer be applied to schmitt trigger Q3 and it will

change state, causing Q4 and Q5 to conduct. The conduction of relay driver Q5 will in turn energize switchover relay K1. With K1 energized, the output of standby servo inverter J94A will be applied to the 12 ma servo amplifiers.

Beta Command Power Supply

The Flight Control Computer contains eight beta command power supplies. Four are used in conjunction with the position indicator potentiometers on the four yaw servo cylinders in the S-IB Stage to develop position feedback signals for the yaw 12 ma servo amplifiers; and four are used in conjunction

with position indicator potentiometers on the four pitch servo cylinders in the S-IB Stage to develop position feedback signals for the pitch 12 ma servo amplifiers. A photograph of the beta command power supply subassembly is shown in Figure 2-25. A functional block diagram of a beta command power supply is shown in Figure 2-26.

The purpose of the beta command power supply is to change the 28 Vdc source to 60 Vdc. To accomplish this purpose, the dc source voltage is first converted by inverter Q1, Q2 to a 1 kHz signal. The 1 kHz signal is then rectified by bridge rectifier

CR1 through CR4 and applied to chopper Q3. The chopper intermittently applies the rectified voltage to filter L2, C5 where it is filtered to produce the 60 Vdc output. The rate at which Q3 switches is controlled by multivibrator Q5, Q7 which is coupled through driver Q4 to Q3.

Differential amplifier Q8 compares the 60 Vdc output with a reference voltage and produces an output in proportion to the difference. The output of the differential amplifier Q8 is then used to regulate the frequency of multivibrator Q5, Q7. When the output voltage increases, the output of Q8 causes the frequency of multivibrator Q5, Q7 to decrease, which increases the ON and OFF time of chopper Q3, resulting in a decreased output voltage. When the output voltage decreases, the frequency of multivibrator Q5, Q7 is increased, which decreases the OFF time of chopper Q3, resulting in an increased output voltage.

Overload protection of the beta command power supply is provided by Q6, which senses an overload condition and applies a control signal to the multivibrator Q5, Q7, causing it to cutoff chopper Q3. Periodically, chopper Q3 is turned ON and the output load condition is sampled. If the overload is still present, the control signal again causes multivibrator Q5, Q7 to cutoff chopper Q3.

50 Ma Servo Amplifier

The Flight Control Computer contains triple redundant yaw 50 ma servo amplifiers and triple redundant pitch 50 ma servo amplifiers. The purpose

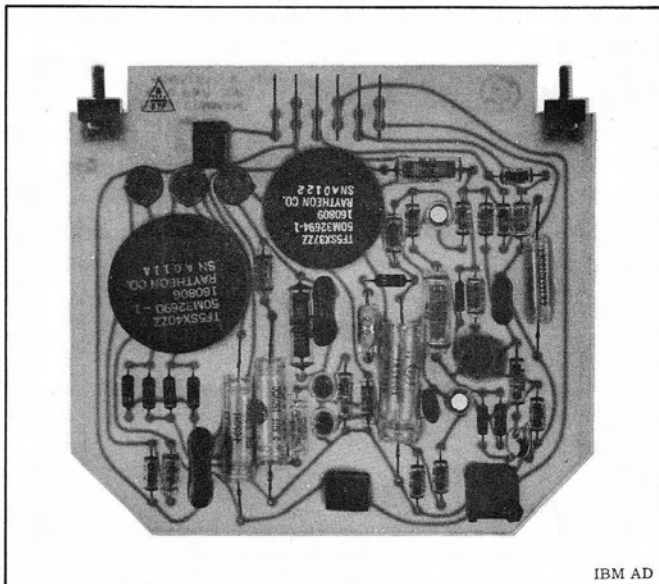


Figure 2-25. Beta Command Power Supply Assembly

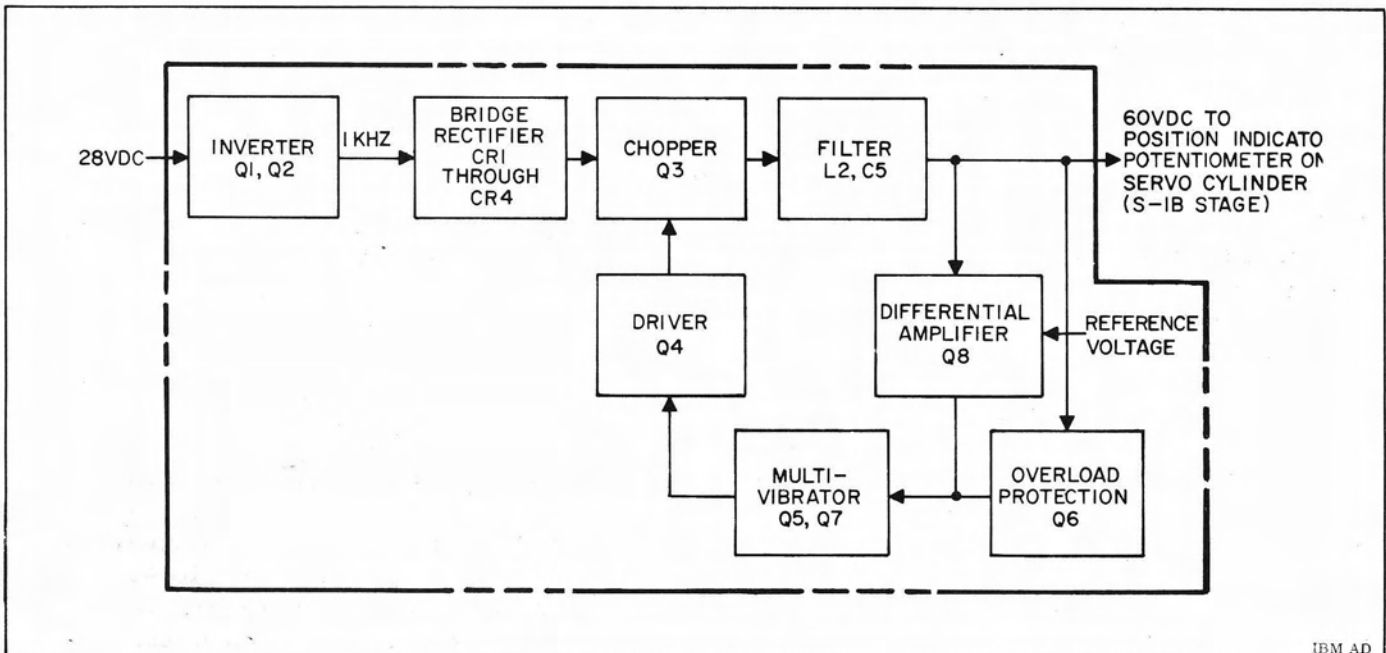


Figure 2-26. Beta Command Power Supply, Functional Block Diagram

of the 50 ma servo amplifiers is to develop drive signals for the pitch and yaw servo cylinder solenoids in the S-IVB Stage in proportion to the yaw and pitch error signals. Triple redundancy is used to ensure a high reliability factor within the Flight Control Computer. A photograph of the 50 ma servo amplifier subassembly is shown in Figure 2-27. A functional block diagram of a 50 ma servo amplifier is shown in Figure 2-28.

Attitude ($\Delta\phi$) and angular rate ($\dot{\phi}$) error signals are applied to coil windings in magnetic amplifier CR12 through CR15. The error signals are algebraically combined by magnetic amplifier CR12 through CR15 to produce a proportional balanced output signal. The balanced output signal of magnetic amplifier CR12 through CR15 is amplified by push-pull amplifier Q6 and applied to differential amplifier Q10, Q11. Differential amplifier Q10, Q11 amplifies the balanced signal and produces a single ended output signal. The output signal of Q10, Q11 is amplified by differential amplifier Q12 and applied to driver Q14, Q15. The driver further amplifies the signal and applies it to complimentary symmetry amplifier Q7, Q9. The

resulting output of Q7, Q9 is the drive signal for the servo cylinder solenoid in the S-IVB Stage.

Differential amplifier Q12 also develops a feedback signal which is applied to a coil winding in magnetic amplifier CR12 through CR15. The feedback signal opposes the input error signals of the magnetic amplifier so as to provide a damping action which will prevent the servo cylinder in the S-IVB Stage from overcorrecting.

The excitation signal for magnetic amplifier CR12 through CR15 is provided by inverter Q4, Q5 which converts the dc source to a 1 kHz signal. Regulation of Q4, Q5 is provided by chopper Q2. Changes in the output voltage of filter L3, C2 are sensed by a comparator-multivibrator Q1, Q3, which increases or decreases the output frequency of Q1, Q3 in proportion to the difference between the output voltage of filter L3, C2 and a reference voltage. The output of Q1, Q3 controls the ON and OFF time of chopper Q2.

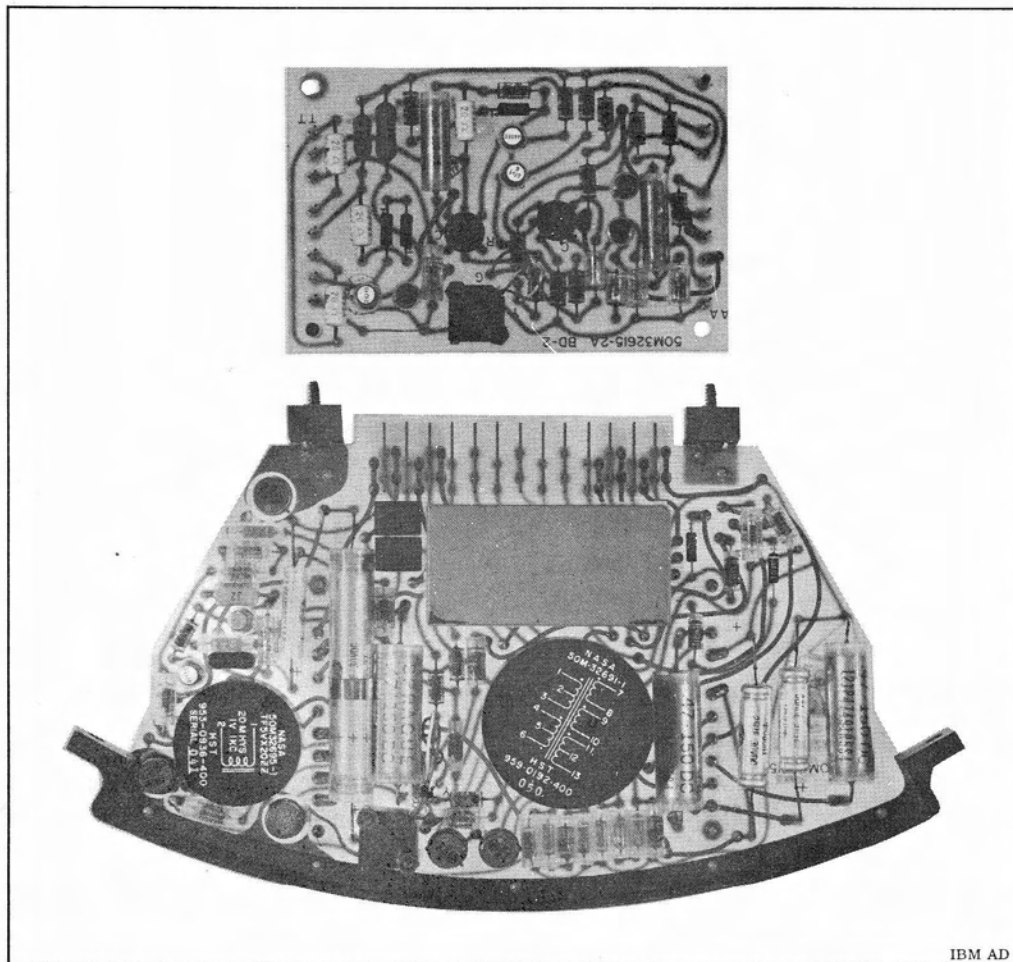


Figure 2-27. 50 Ma Servo Amplifier Subassembly

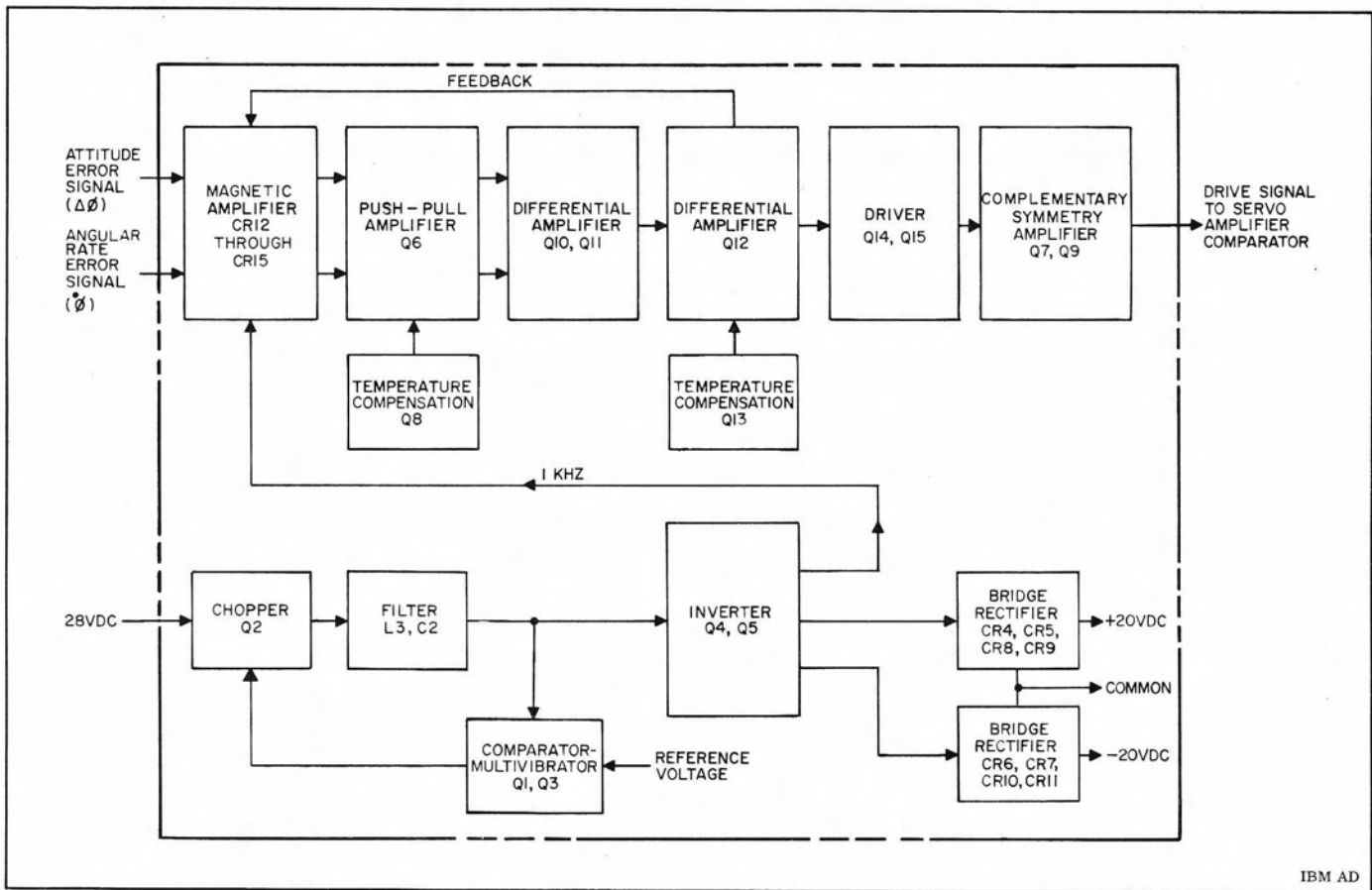


Figure 2-28. 50 Ma Servo Amplifier, Functional Block Diagram

Inverter Q4, Q5 also applies a 1 kHz signal to bridge rectifiers CR4, CR5, CR8, CR9, and CR6, CR7, CR10, CR11. The 1 kHz signal is rectified by the bridge rectifiers to produce the ± 20 Vdc necessary for the operation of the amplifier stages in the 50 ma servo amplifier.

Temperature changes which will affect the overall performance of the 50 ma servo amplifier are sensed by temperature compensators Q8 and Q13. Temperature compensators Q8 and Q13 control the gain of the 50 ma servo amplifier by varying the bias level on push-pull amplifier Q6 and differential amplifier Q12 in proportion to the temperature change.

Servo Amplifier Comparator

The Flight Control Computer contains a yaw servo amplifier comparator and a pitch servo amplifier comparator. The yaw servo amplifier comparator is used to monitor the outputs of the triple redundant yaw 50 ma servo amplifiers. The pitch servo amplifier comparator is used to monitor

the outputs of the triple redundant pitch 50 ma servo amplifiers.

The purpose of the servo amplifier comparator is to compare the identical drive signal outputs of two of the triple redundant 50 ma servo amplifiers (one of which is applied to a servo cylinder actuator in the S-IVB Stage). When a malfunction occurs in either of the two 50 ma servo amplifiers being compared, the servo amplifier comparator automatically terminates the signal being applied to the servo cylinder actuator and applies the drive signal from the third (standby) 50 ma servo amplifier to the servo cylinder actuator. A photograph of the servo amplifier subassembly is shown in Figure 2-29. A functional block diagram of the servo amplifier comparator is shown in Figure 2-30.

Identical drive signals from the triple redundant 50 ma servo amplifiers (A, B, and C) are applied to the servo amplifier comparator. The drive signal from 50 ma servo amplifier B is normally routed through closed relay contacts in switchover relays K1, K3, K4 to a servo cylinder

signals and produces a proportional balanced output signal. The balanced output signal is then amplified by differential amplifier Q8 and differential amplifier Q10, Q11. The resulting balanced signal output causes Schmitt trigger Q12 or Schmitt trigger Q13 to change state. The polarity of the input error signals determine which Schmitt trigger will change state.

When the output signal of Q10, Q11 causes Schmitt trigger Q12 to change state, the resulting output signal is amplified by dc amplifier Q14 and relay driver Q16. When the output signal of Q10, Q11 causes Schmitt trigger Q13 to change state, the resulting output signal is amplified by dc amplifier Q15 and relay driver Q17. The output signal from relay driver Q16 or Q17 is the control signal for the control relay package in the S-IVB Stage.

The level of input signal required to cause the Schmitt trigger to change state is controlled by the threshold feedback from Q10, Q11. A preset level of the threshold feedback is developed by Q10, Q11 and

applied to a coil winding in magnetic amplifier L2 when error signals are applied to magnetic amplifier L2. The polarity of the threshold feedback opposes the polarity of the input error signals; therefore, for one of the Schmitt triggers to change state, the level of the input error signals must be of sufficient amplitude to overcome the threshold feedback.

The duration of the spatial thruster control signal developed by the spatial amplifier assembly is determined by the comparison of the input error signals and the time delayed cutoff feedback. A preset level of cutoff feedback is developed by Q14 or Q15 and applied through a time delay R15, R20, C3, C4 to a coil winding in magnetic amplifier L2. The cutoff feedback signal opposes the input error signal; however, due to the time delay of the cutoff feedback, the input error signals hold Q12 or Q13 in a changed state until the cutoff feedback has reached sufficient amplitude to overcome the input error signals. When this happens, Q12 or Q13 returns to its normal state. The greater the amplitude of the input error signals, the longer it will take for the cutoff feedback signal

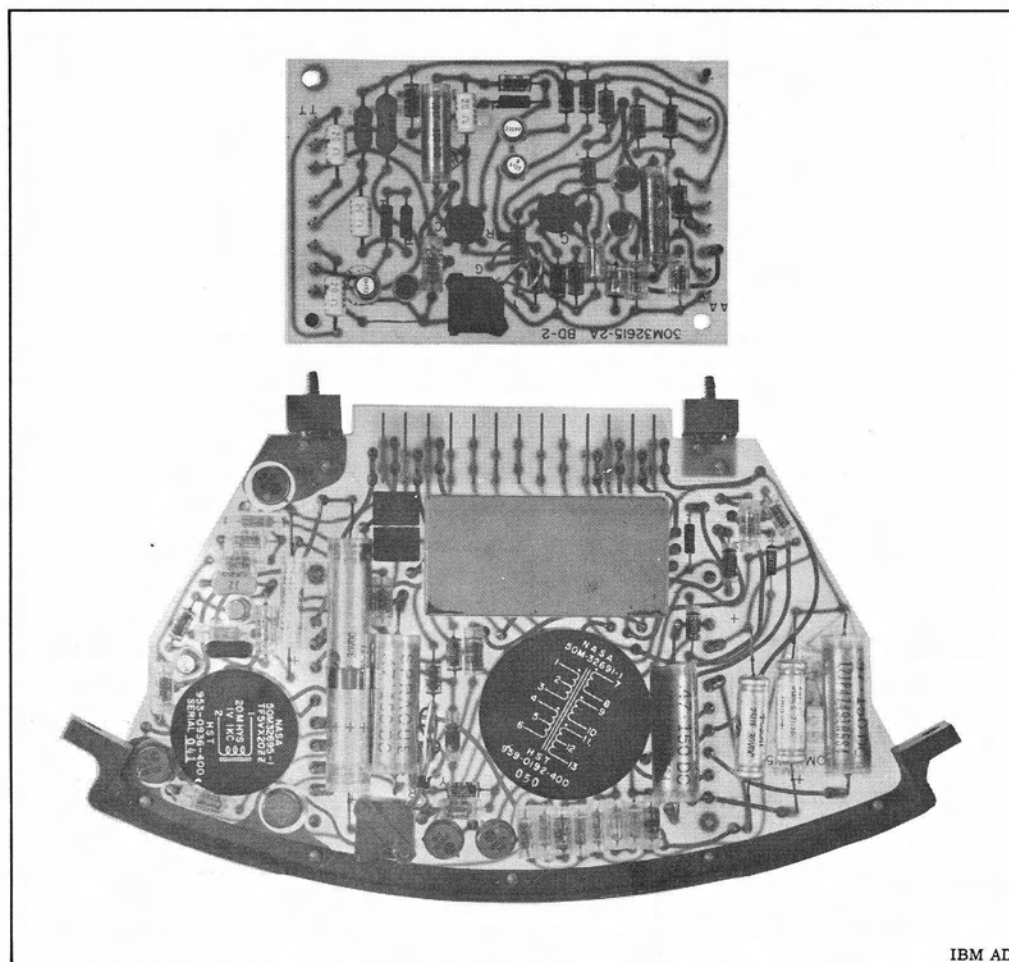


Figure 2-25. 50 Ma Servo Amplifier

to overcome the input error signals, resulting in proportionally longer duration output signals.

The 28 Vdc source for the spatial amplifier is regulated by series regulator Q1. A sample of the output voltage is compared in comparator-amplifier Q3 with a stable reference voltage. A regulator control voltage, produced by this comparison, is amplified by driver Q2 and applied to series regulator Q1. This control voltage controls the internal impedance, and consequently the voltage drop, of Q1. If the 20 Vdc output voltage varies, the regulator control voltage varies proportionally. The variation in control voltage produces a corresponding variation in the voltage drop across Q1 and returns the output voltage to the proper level.

The excitation signal required for the operation of magnetic amplifier L2 is provided by inverter Q4, Q5, which converts the +20 Vdc output of series regulator Q1 to a 1 kHz signal.

Temperature changes which would affect the performance of the spatial amplifier are sensed by temperature compensator Q9. Temperature compensator Q9 controls the gain of the spatial amplifier by

varying the bias level for differential amplifier Q8 in proportion to the temperature change.

2-18 SPATIAL SYNCHRONIZER

The Flight Control Computer contains one spatial synchronizer which is used in conjunction with the roll-yaw I spatial amplifiers and the roll-yaw II spatial amplifiers. The purpose of the spatial synchronizer is to insure spatial thruster (S-IVB Stage) synchronization during the S-IVB burn mode. A photograph of a spatial synchronizer is shown on Figure 2-31. A block diagram of the spatial synchronizer is shown on Figure 2-32.

The S-IVB burn signal from the Switch Selector is applied to relay driver Q1, Q3 and relay driver Q2, Q4 during the S-IVB burn mode, causing them to conduct. Conduction of Q1, Q3 energizes switchover relays K1, K2, and conduction of Q2, Q4 energizes switchover relays K3, K4.

With K1, K2 and K3, K4 energized, the signal lines from roll-yaw II spatial amplifiers B and C are terminated at open relay contacts. The two signal lines from roll-yaw I spatial amplifier B are coupled

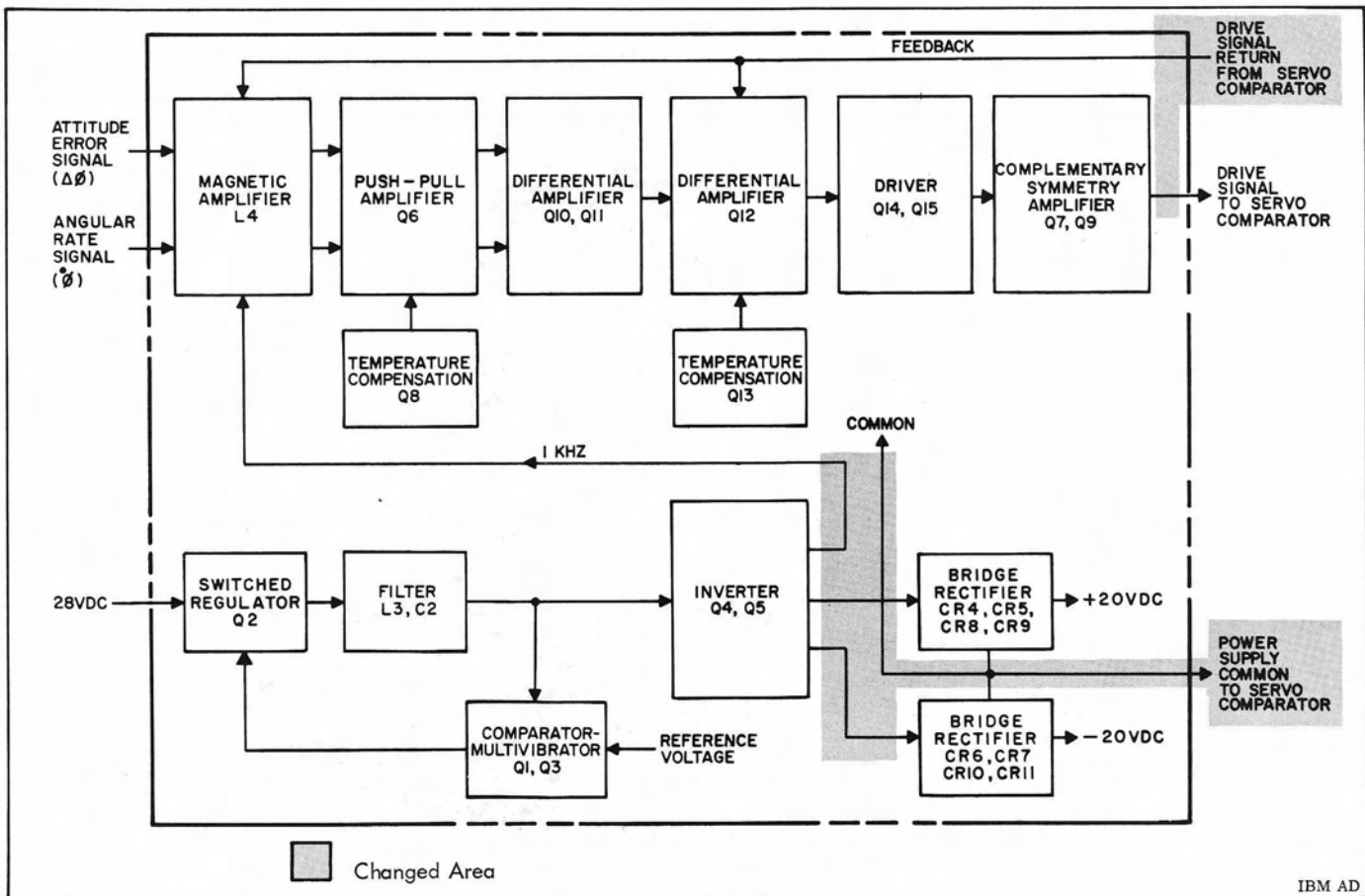


Figure 2-26. 50 Ma Servo Amplifier Block Diagram

The balanced output signal is then amplified by differential amplifier Q8 and differential amplifier Q10, Q11. The resulting balanced signal output causes schmitt trigger Q12 or schmitt trigger Q13 to change state. The polarity of the input error signals determine which schmitt trigger will change state.

When the output signal Q10, Q11 causes schmitt trigger Q12 to change state, the resulting output signal is amplified by dc amplifier Q14 and relay driver Q16. When the output signal of Q10, Q11 causes schmitt trigger Q13 to change state, the resulting output signal is amplified by dc amplifier Q15 and relay driver Q17. The output signal from relay driver Q16 or Q17 is the control signal for the control relay package in the S-IVB Stage.

The level of input signal required to cause the schmitt trigger to change state is controlled by the threshold feedback from differential amplifier Q10, Q11. A preset level of threshold feedback is developed by differential amplifier Q10, Q11 and applied to a coil winding in magnetic amplifier Q6, Q7 when error signals are applied to magnetic amplifier Q6, Q7. The polarity of the threshold feedback opposes the

polarity of the input error signals; therefore, for one of the schmitt triggers to change state, the level of the input error signals must be in sufficient amplitude to overcome the threshold feedback.

The duration of the spatial thruster control signal developed by the spatial amplifier assembly is determined by the comparison of the input error signals and the time delayed cutoff feedback. A preset level of cutoff feedback is developed by relay driver Q16 or Q17 and applied through a time delay R15, R20, C3, C4 to a coil winding in magnetic amplifier Q6, Q7 when the input error signals cause schmitt trigger Q12 or Q13 to change state. The cutoff feedback signal opposes the input error signals; however, due to the time delay of the cutoff feedback, the input error signals hold schmitt trigger Q12 or Q13 in a changed state until the cutoff feedback has reached sufficient amplitude to overcome the input error signals. When this happens, schmitt trigger Q12 or Q13 returns to its normal state. The greater the amplitude of the input error signals, the longer time it will take the cutoff feedback signal to overcome the input error signals, resulting in proportionally longer duration output signals.

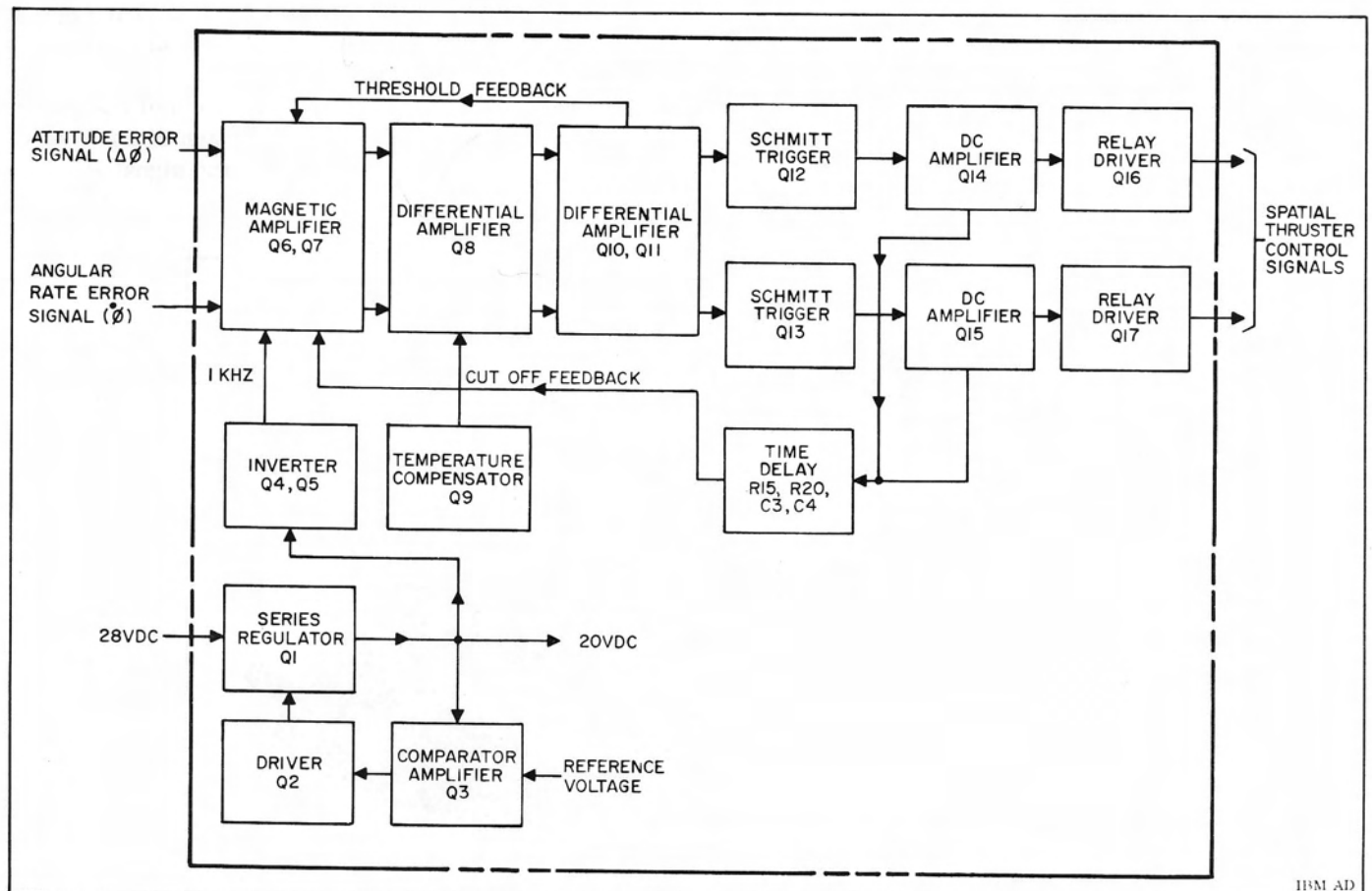


Figure 2-32. Spatial Amplifier, Functional Block Diagram

The 28 Vdc source for the spatial amplifier is regulated by series regulator Q1. A sample of the output voltage is compared in comparator-amplifier Q3 with a stable reference voltage. A regulator control voltage, produced by this comparison, is amplified by driver Q2 and applied to series regulator Q1. This control voltage controls the internal impedance, and consequently the voltage drop, of series regulator Q1. If the 20 Vdc output voltage varies, the regulator control voltage varies proportionally. The variation in control voltage produces a corresponding variation in the voltage drop across series regulator Q1 and returns the output voltage to the proper level.

The excitation signal required for the operation of magnetic amplifier Q6, Q7 is provided by inverter Q4, Q5, which converts the +20 Vdc output of series regulator Q1 to a 1 kHz signal.

Spatial Comparator

The Flight Control Computer contains a roll-yaw I spatial comparator, which is used in conjunction with the triple redundant roll-yaw I spatial amplifiers, a roll-yaw II spatial comparator, which is used in conjunction with the triple redundant roll-yaw II spatial amplifiers, and a pitch spatial comparator, which is used in conjunction with the triple redundant pitch spatial amplifiers. The purpose of the spatial comparator is to compare the spatial thruster control signal outputs of two of its associated triple redundant spatial amplifiers, one of which is applied to the control relay control package in the S-IVB Stage. When a

malfunction occurs in either of the two spatial amplifiers being compared, the spatial comparator automatically terminates the control signal being applied to the control relay package and applies the control signal from the third (standby) spatial amplifier to the control relay package. A photograph of a spatial comparator is shown in Figure 2-33. A functional block diagram of the spatial comparator is shown in Figure 2-34.

Identical spatial thruster control signals from the triple redundant spatial amplifiers (A, B, and C) are applied to the spatial comparator. The signal lines from spatial amplifier B are normally coupled through closed relay contacts in switchover relays K1, K3 to the spatial TM amplifier and the control relay package in the S-IVB Stage. The signal lines from spatial amplifier A are applied to dummy load R7, R8 which represents the load presented by the TM amplifier and the control relay package. The signal lines from spatial amplifier C are terminated at open relay contacts in switchover relays K1, K3. The signal lines from spatial amplifiers A and B are also applied to isolation network CR2, CR3, CR5, CR6.

Each of the spatial amplifiers has two output signal lines; however, a spatial control signal will appear on only one signal line at a time. The signal line that a signal will appear on is determined by the polarity of the input error signals applied to the spatial amplifiers. To simplify the description of the spatial comparator, the signal lines from each of the spatial amplifiers are called signal line 1 and signal line 2 in Figure 2-34.

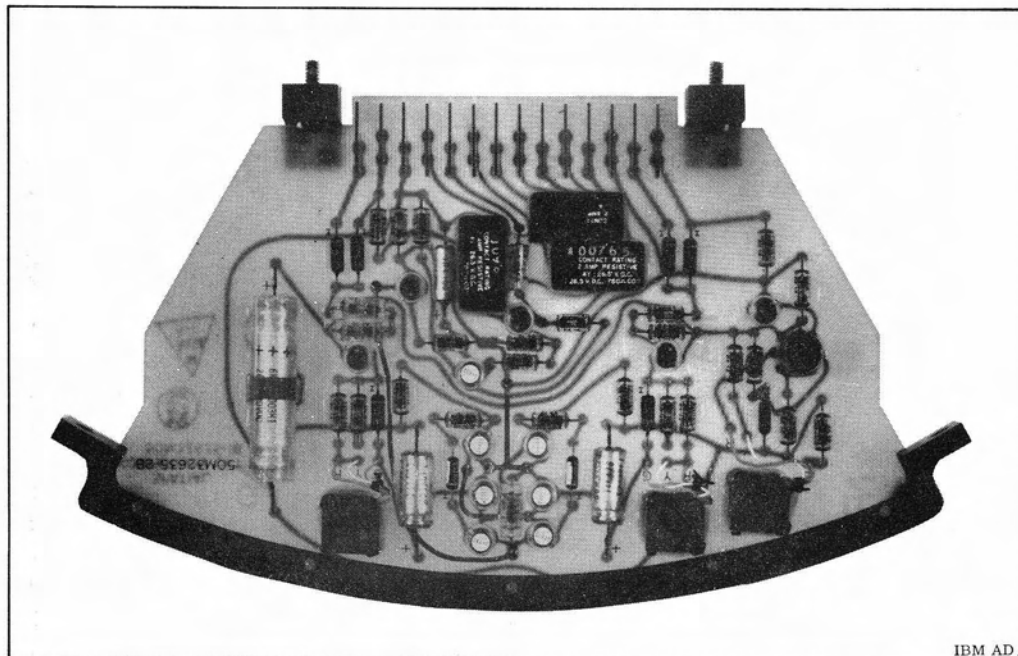


Figure 2-33. Spatial Comparator Subassembly

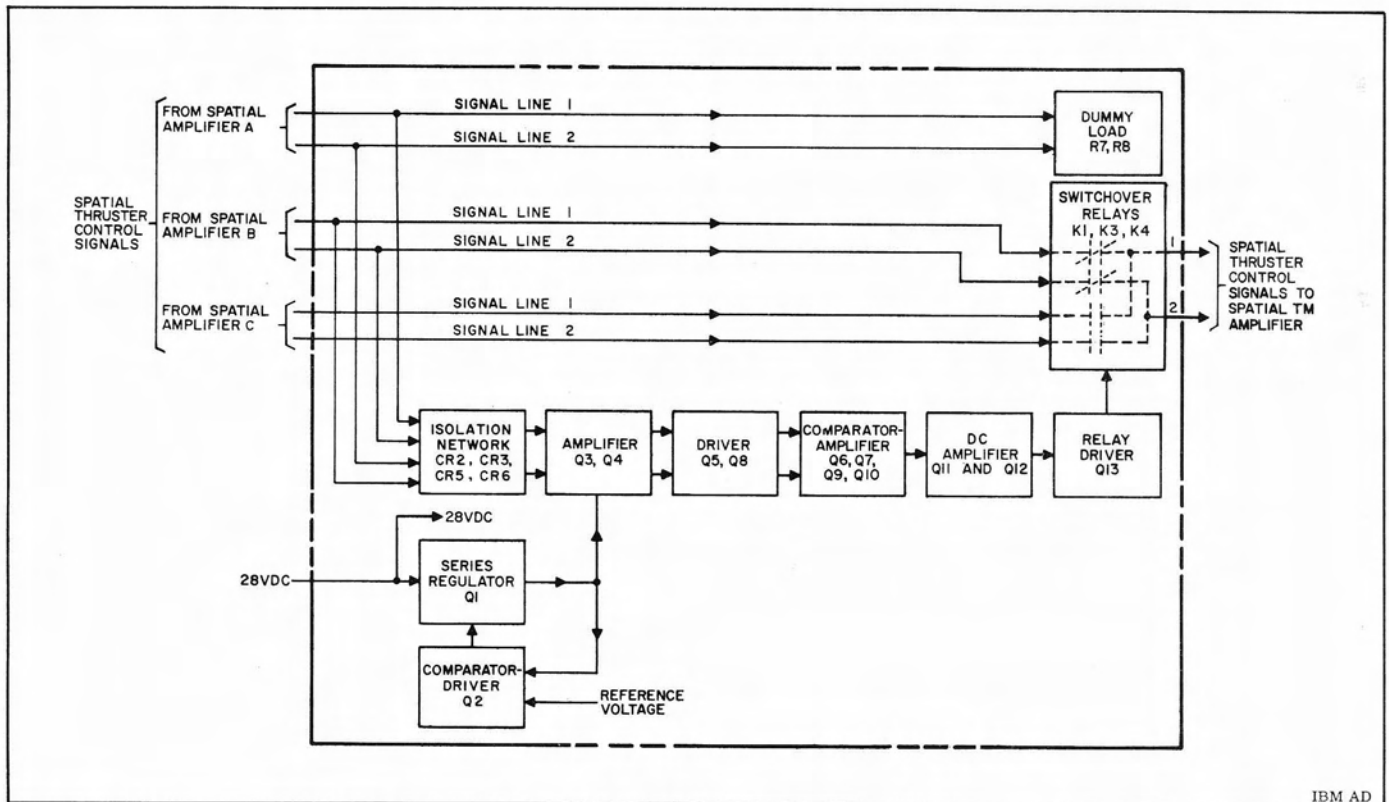


Figure 2-34. Spatial Comparator, Functional Block Diagram

When the spatial amplifiers produce a spatial thruster control signal on signal line 1, the signals on signal line 1 of spatial amplifiers A and B are coupled through isolation network CR2, CR3, CR5, CR6 to amplifier Q3, Q4. The signals are then amplified by amplifier Q3, Q4 and driver Q5, Q8, and applied to comparator-amplifier Q6, Q7, Q9, Q10. Under a normal balanced condition (both inputs to comparator-amplifier are equal), no signal appears at the output of the comparator-amplifier. When a malfunction occurs in either of the spatial amplifiers (A or B), an unbalanced condition exists within the comparator-amplifier which results in a signal being applied to dc amplifiers Q11 and Q12. The signal from comparator-amplifier Q6, Q7 is then amplified by dc amplifiers Q11 and Q12 and applied to relay driver Q13, causing relay driver Q13 to conduct. The conduction of relay driver Q13 energizes switchover relays K1, K3. With switchover relays K1, K3 energized, the two signal lines from spatial amplifier B are terminated at open relay contacts and the two signal lines from spatial amplifier C are coupled through closed relay contacts to the TM amplifier and the control relay package in the S-IVB Stage.

When the spatial amplifiers produce a spatial thruster control signal on signal line 2, the signals

on signal line 2 from spatial amplifiers A and B are coupled through isolation network CR2, CR3, CR5, CR6 to amplifier Q3, Q4. The signals on signal line 2 are then processed in the same manner as the signals on signal line 1.

The dc source voltage for amplifier Q3, Q4 is coupled through series regulator Q1. Control of series regulator Q1 is provided by comparator-driver Q2, which applies control signals to Q1 in proportion to the difference between the output voltage and a reference voltage.

Spatial Synchronizer

The Flight Control Computer contains one spatial synchronizer which is used in conjunction with the roll-yaw I spatial amplifiers and the roll-yaw II spatial amplifiers. The purpose of the spatial synchronizer is to provide spatial thruster (S-IVB Stage) synchronization during the S-IVB burn mode. A photograph of a spatial synchronizer is shown in Figure 2-35. A functional block diagram of the spatial synchronizer is shown in Figure 2-36.

An S-IVB burn signal from the Switch Selector is applied to relay driver Q1, Q4 and relay driver Q2,

Q3 during the S-IVB burn mode, causing them to conduct. Conduction of relay driver Q1, Q4 energizes switchover relays K1, K2 and conduction of relay driver Q2, Q3 energizes switchover relays K3, K4.

With switchover relays K1, K2 and K3, K4 energized, the signal lines from roll-yaw II spatial amplifiers B and C are terminated at open relay contacts. The two signal lines from roll-yaw I spatial amplifier B are coupled through closed relay contacts

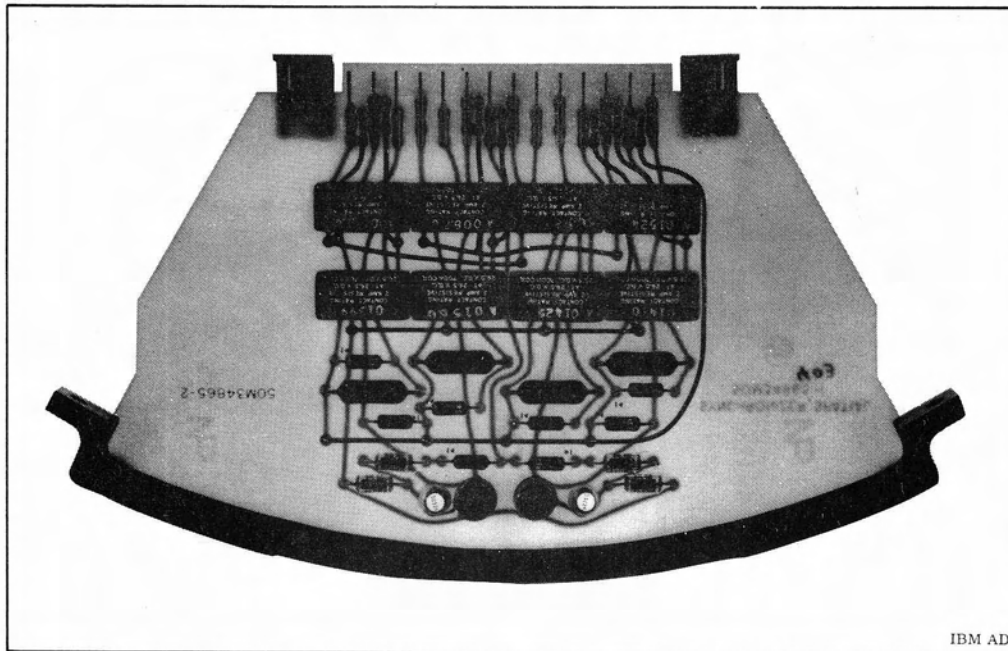


Figure 2-35. Spatial Synchronizer Subassembly

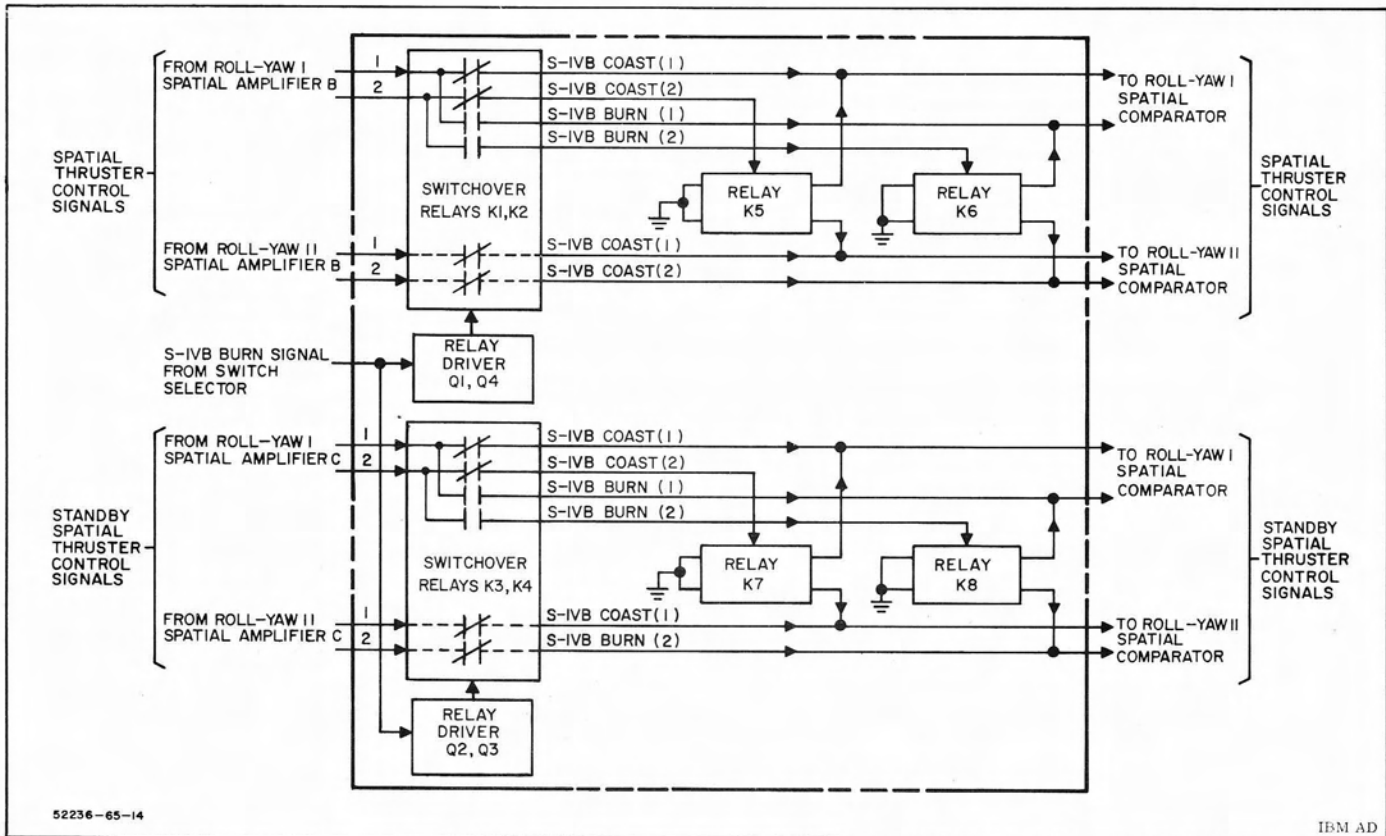


Figure 2-36. Spatial Synchronizer, Functional Block Diagram

to relays K5 and K6, and the two signal lines from roll-yaw I spatial amplifier C are coupled through closed relay contacts to relays K7 and K8.

Relays K5 and K7 are energized when a spatial thruster control signal (ground) is present on signal line 1 of roll-yaw I spatial amplifiers B and C. The resulting closed contacts of relay K5 apply a ground signal to the roll-yaw I spatial comparator and the roll-yaw II spatial comparator. Likewise, the resulting closed contacts of relay K7 apply a ground signal to the roll-yaw I spatial comparator and the roll-yaw II spatial comparator.

Relays K6 and K8 are energized when a spatial thruster control signal (ground) is present on signal line 2 of roll-yaw I spatial amplifiers B and C. The resulting closed contacts of relay K6 apply a ground signal to the roll-yaw I spatial comparator and the roll-yaw II spatial comparator. Likewise, the resulting closed contacts of relay K8 apply a ground signal to the roll-yaw I spatial comparator and the roll-yaw II spatial comparator.

The S-IVB burn signal from the Switch Selector is removed from relay drivers Q1, Q4 and Q2, Q3 at the completion of the S-IVB burn mode, (beginning of S-IVB coast mode) causing relay drivers Q1, Q4 and Q2, Q3 to become cut OFF. Switchover relays K1, K2 and K3, K4 are deenergized when relay drivers Q1, Q4 and Q2, Q3 are cut OFF. The signal lines from roll-yaw I spatial amplifiers B and C are then coupled through closed relay contacts in switchover relays K1, K2 and K3, K4 to the roll-yaw I spatial

comparator. The signal lines from roll-yaw II spatial amplifiers B and C are also coupled through closed relay contacts in switchover relays K1, K2 and K3, K4 to the roll-yaw II spatial comparator.

TM Amplifier

The Flight Control Computer contains three TM amplifiers. The TM amplifiers are used to monitor the output signal lines of the spatial comparators and develop a telemetry signal when spatial thruster control signals are present. A photograph of a TM amplifier is shown in Figure 2-37. A functional block diagram of the TM amplifier is shown in 2-38.

Signal lines from the spatial comparator are coupled through coil windings in magnetic amplifier CR1 through CR4 to the control relay package in the S-IVB Stage. The magnetic amplifier produces an output signal when a spatial thruster control signal is present on one of the signal lines from the spatial comparator. The polarity of the magnetic amplifier output is determined by which signal line the spatial thruster control signal is carried on. The output of the magnetic amplifier is coupled through temperature compensator RT1 to produce the telemetry output signal.

The excitation signal for the magnetic amplifier is developed by inverter Q1, Q2 which converts the 28 Vdc source to a 1 kHz signal. The amplitude of the magnetic amplifier output is preset by the 28Vdc bias control.

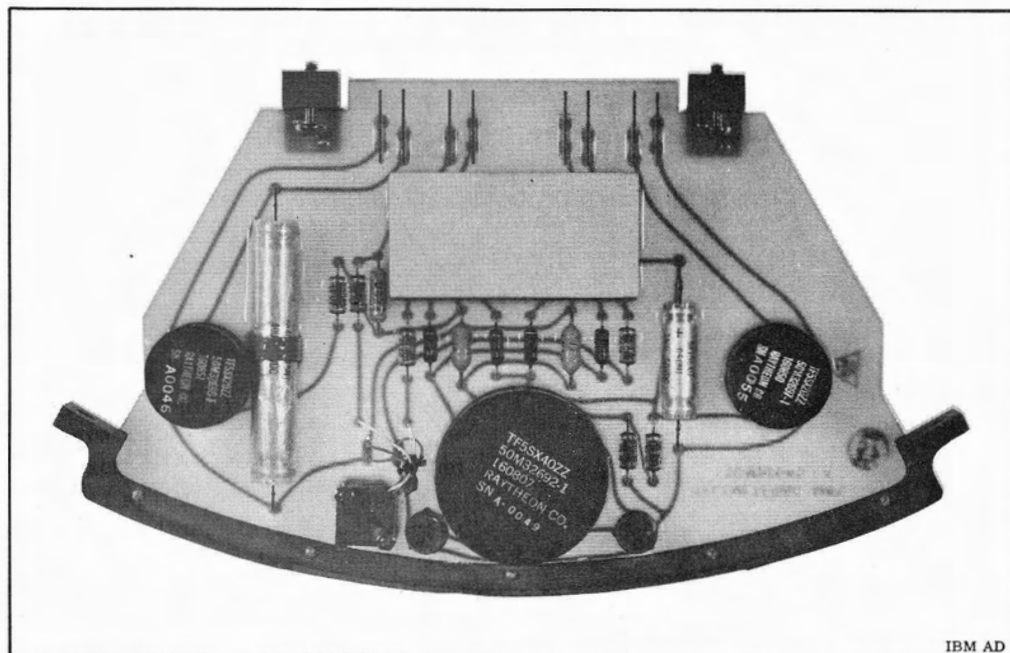


Figure 2-37. TM Amplifier Subassembly

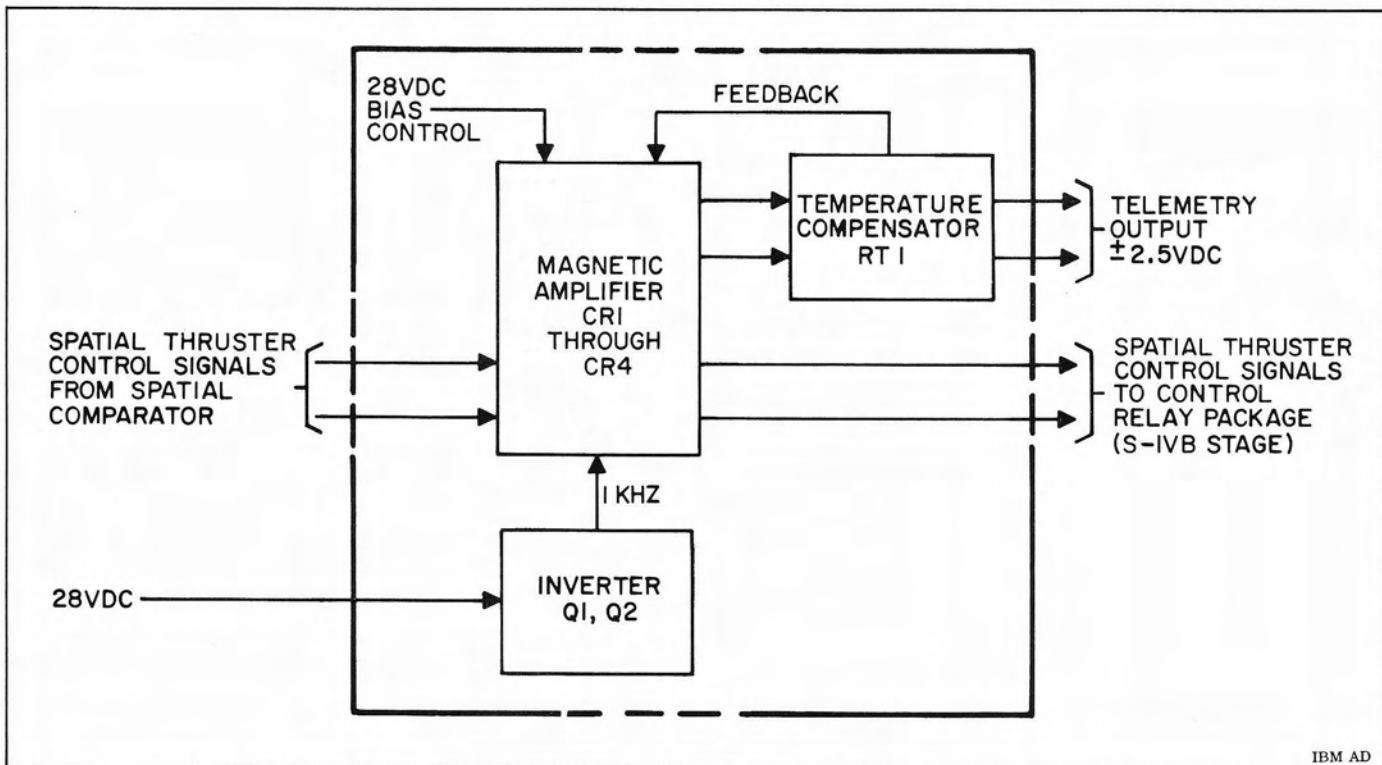


Figure 2-38. TM Amplifier, Functional Block Diagram

Temperature changes which affect the overall performance of the TM amplifier are sensed by temperature compensator RT1. RT1 controls the gain of the magnetic amplifier by applying a feedback in proportion to the temperature change.

S-IB Power Distribution Assembly

The S-IB power distribution assembly is used during the S-IB burn mode to control the application of power to the S-IB burn mode circuitry. A photograph of the S-IB power distribution assembly is shown in Figure 2-39. A functional block diagram of the S-IB power distribution assembly is shown in Figure 2-40.

The S-IB power distribution assembly is activated at lift-off when a first motion signal from the Switch Selector energizes relays K3, K4, K8, and K11. The contacts of these relays are connected in a series-parallel arrangement to provide a highly reliable switching action. This series-parallel arrangement is shown as a single set of contacts in Figure 2-40.

The operation of relays K3, K4, K8, and K11 results in the 28 Vdc source being applied directly to the CAT and servo inverters. The 28 Vdc source is also coupled through the closed contacts of relays K1

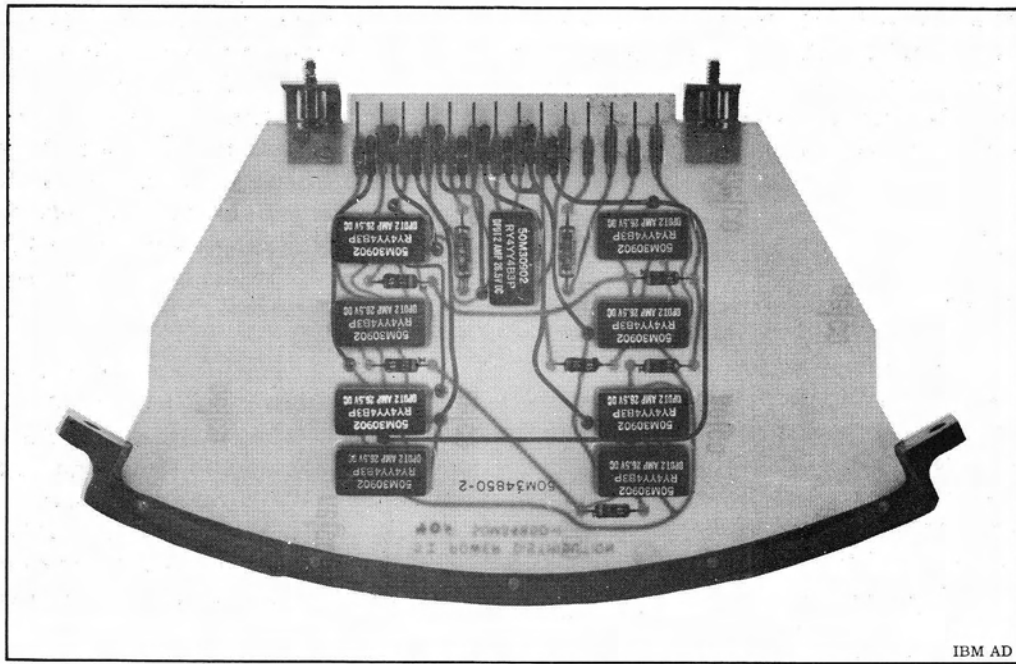
and K2 to the beta command power supplies; and through the closed contacts of relays K5 and K6 to switching circuits A1 and A3.

Relay K7 is energized by the 28 Vdc signal from the CAT. The resultant closed contacts of relay K7 couples the 115 Vac, 400 Hz power from the CAT inverter to the CAT.

These relays remain energized throughout the S-IB burn mode. At the completion of the S-IB burn mode, the first motion signal and the 28 Vdc signal from the CAT are removed, causing the relays to return to their original deenergized condition.

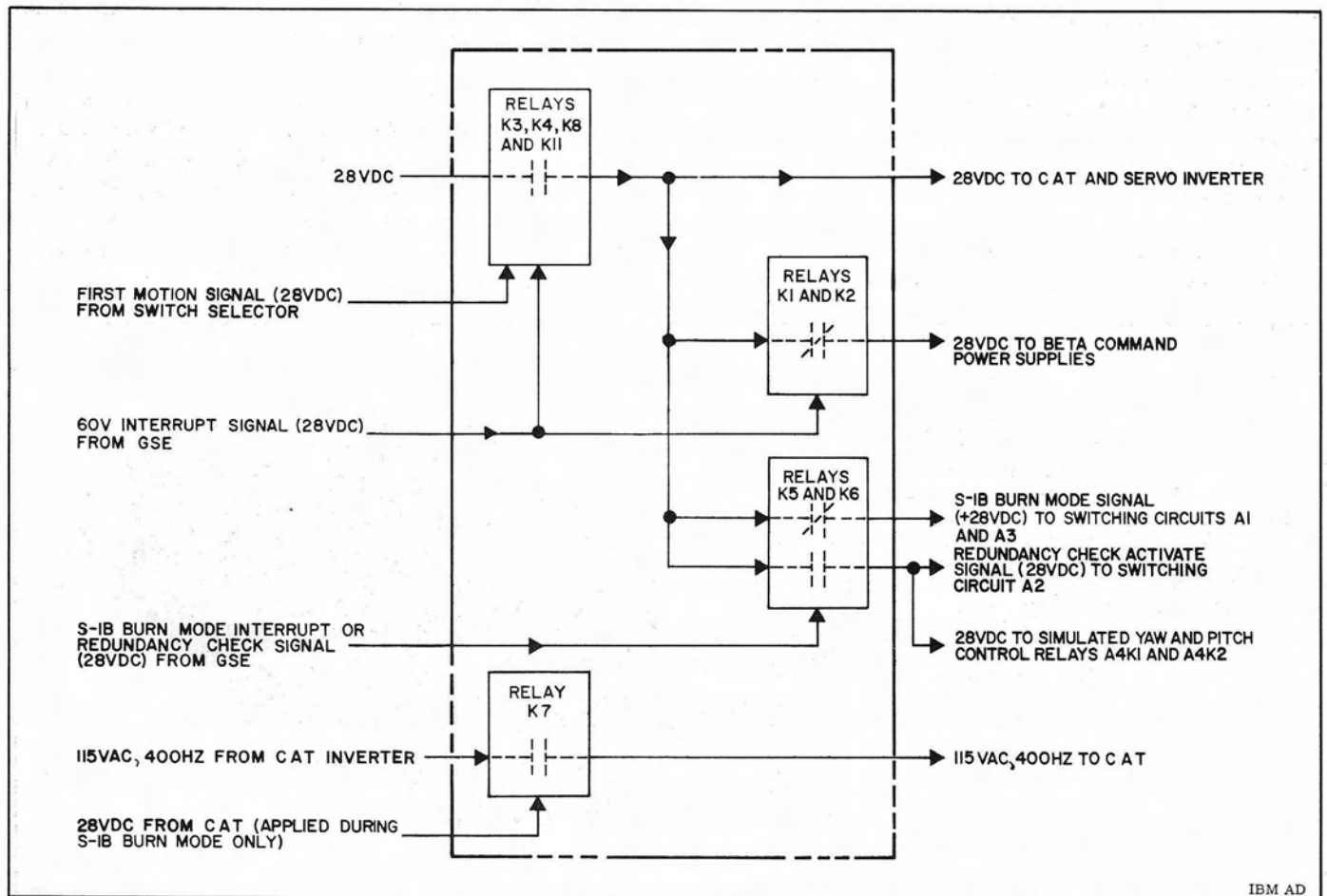
Provisions for control of the S-IB power distribution assembly during GSE testing is also provided. During simulated S-IB burn, the 60 Vdc output of the beta command power supplies can be interrupted by a 28 Vdc signal from the GSE. This signal, referred to as the 60 V interrupt signal, energizes relays K1, K2 and K3, K4, K8 resulting in +28 Vdc being removed from the input of the beta command power supplies. When the +28 Vdc input is removed the 60 Vdc output is interrupted.

The S-IB burn mode interrupt or redundancy check signal from the GSE is applied to relays K5 and K6. During the S-IB burn test, it is used to interrupt



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Figure 2-39. S-IB Power Distribution Assembly



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Figure 2-40. S-IB Power Distribution Assembly, Functional Block Diagram

the application of power to switching circuits A1 and A3, and switches in simulated yaw and pitch signals by applying 28 Vdc to relays A4K1 and A4K2. During the switching relays redundancy check, it is used to apply power to the redundancy check matrix in switching circuit A2.

Switching Circuits

The Flight Control Computer contains three switching circuit assemblies, A1, A2, and A3. These switching circuits operate together to control the routing of error signals during the S-IB burn mode, S-IVB burn mode, and S-IVB coast mode. They are also used during systems test to control the routing of test signals from the GSE. Each switching circuit assembly contains a group of relays and relay drivers with the exception of switching circuit A2 which also contains a redundancy check matrix. The redundancy check matrix is controlled by the GSE and is used to check the switching action of the various relays on the switching circuits.

The following functional description of the switching circuits describes only the activating sequence and basic functions of the relay groups. The actual switching functions and location of the relay contacts in relation to the other Flight Control Computer circuitry is illustrated in Figure 2-2 through 2-4. Photographs of the switching circuits are shown in Figures 2-41, 2-42, and 2-43. A functional block diagram of the switching circuits is shown in Figure 2-44.

At lift-off, the S-IB burn mode signal from the S-IB power distribution assembly is applied to

relay drivers Q9 through Q16 of switching circuit A1; and to relay drivers Q1 through Q8 of switching circuit A3. This causes each of the relay drivers to conduct, which in turn, energizes relays K13 through K24 of A1 and relays K1 through K15 of A3. The contacts of these relays then connect the input error signals to the S-IB burn mode guidance control circuits. At the completion of the S-IB burn mode, the S-IB burn mode signal is removed and these relays become deenergized.

S-IVB burn mode switching is activated by the S-IVB burn mode signal from the Switch Selector. The S-IVB burn mode signal is applied to relay drivers Q1 through Q14 of switching circuit A2. This causes the relay drivers to conduct, which in turn, energize relays K1 through K12 of A2. The contacts of these relays then connect the input error signals to the S-IVB burn mode guidance control circuits.

At the completion of the S-IVB burn mode, the S-IVB burn mode signal is removed and the A2 switching circuit relays are deenergized. The contacts of the switching circuit relays are connected so that when they are in the deenergized condition, the input error signals are connected to the S-IVB coast mode guidance control circuits.

Guidance of the Saturn launch vehicle is normally controlled by the signals developed in the LVDA. However, guidance during the S-IVB burn and coast modes can also be controlled by the Apollo Spacecraft. Switchover is accomplished upon receipt of an Apollo control signal from the Switch Selector. The Apollo control signal is applied to relay drivers

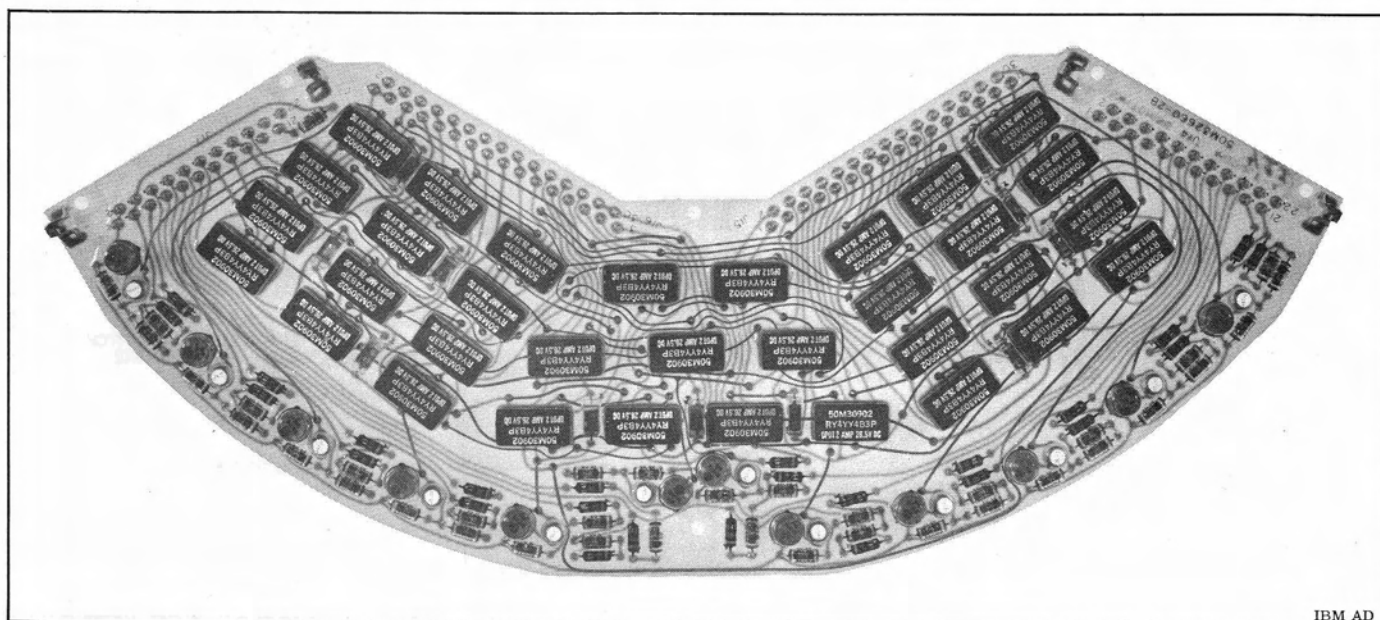


Figure 2-41. Switching Circuit A1

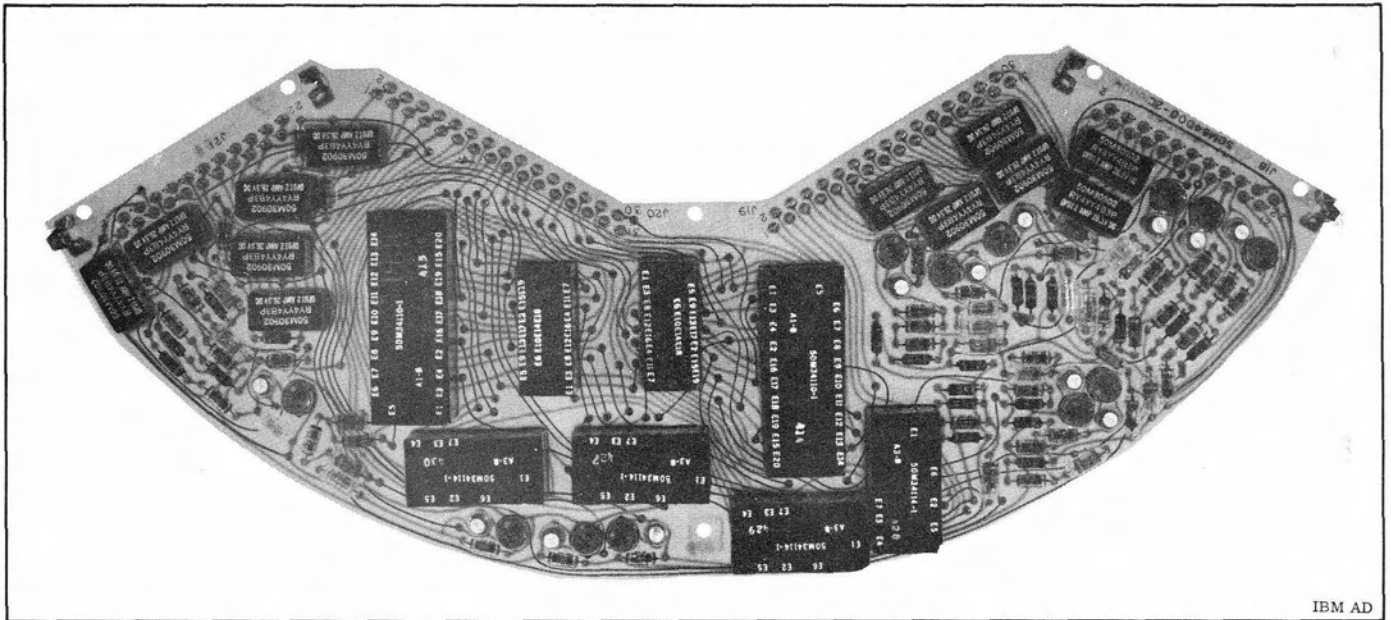


Figure 2-42. Switching Circuit A2

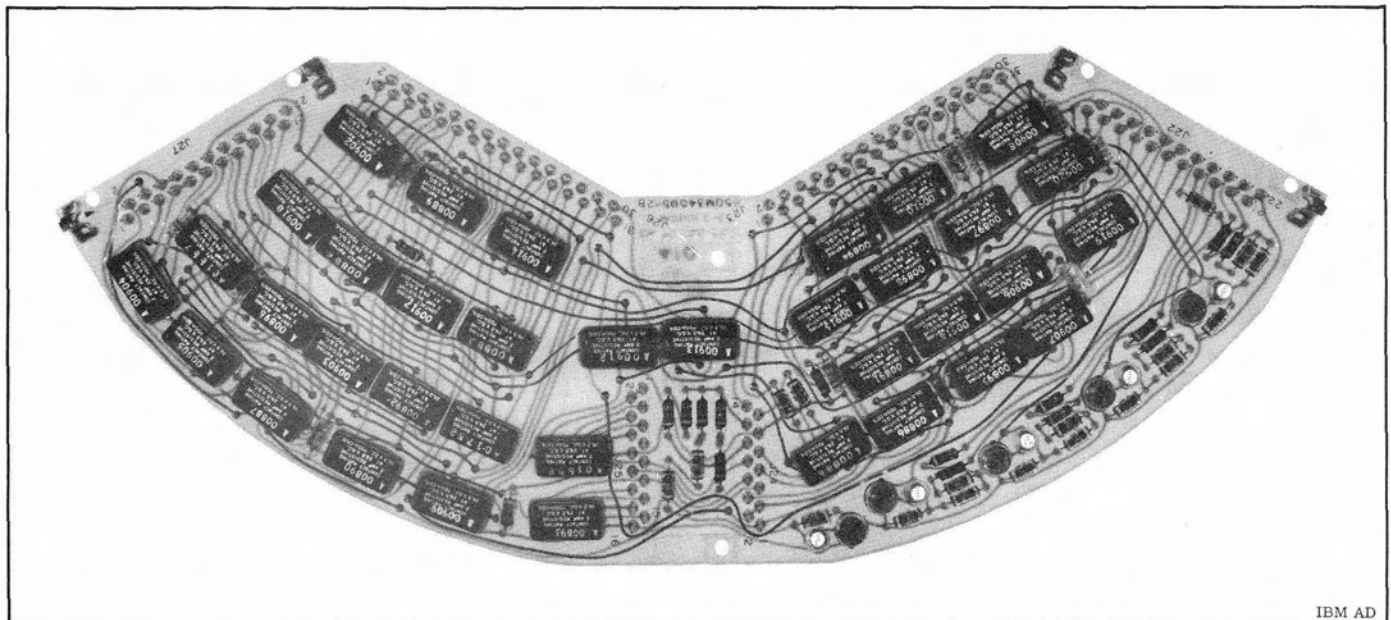


Figure 2-43. Switching Circuit A3

Q1 through Q8 of switching circuit A1, causing relays K1 through K12 to become energized.

The switching circuits are also used during systems test to control the routing and application of test signals from the GSE. Control of the testing sequence and setup is controlled by the application of test signals from the GSE to the switching circuits. These test signals set the Flight Control Computer for simulated S-IB burn mode operation, S-IVB burn mode operation, and S-IVB coast mode operation by energizing appropriate switching circuit relays.

Redundant relay switching is used throughout the Flight Control Computer to achieve a high degree of reliability and positive switching action. The switching action of these redundant relay contacts is tested by the redundancy check circuit in switching circuit A2 (Figure 2-44).

The redundancy check circuit consists of a diode matrix and four identical matrix drivers. The polarity (+ or -) or absence (0) of a matrix driver input signal determines the polarity of its four outputs. Note 3 lists the three conditions that can exist.

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The resultant matrix driver outputs are then applied to the diode matrix, setting-up bias conditions for the diodes.

The redundancy check circuit is activated upon receipt of 28 Vdc from the S-IB power distribution assembly. The input signals to the four matrix drivers is supplied by the GSE and is called the matrix code. The matrix code determines the matrix driver outputs, which in turn, controls the biasing of the diode matrix. This action results in 28 Vdc appearing at one of the diode matrix outputs. The diode matrix outputs are connected to the relay drivers shown on Figure 2-44. Therefore, the

relay or groups of relays that will be energized is determined by the matrix code.

Note 4 lists the switching circuit relays that are energized upon receipt of a specific matrix code (matrix driver inputs) from the GSE. The matrix codes are normally applied in the sequence shown; however, individual relays or groups of relays may be energized by applying the appropriate matrix code.

2-8 DETAILED OPERATION

(To Be Supplied)

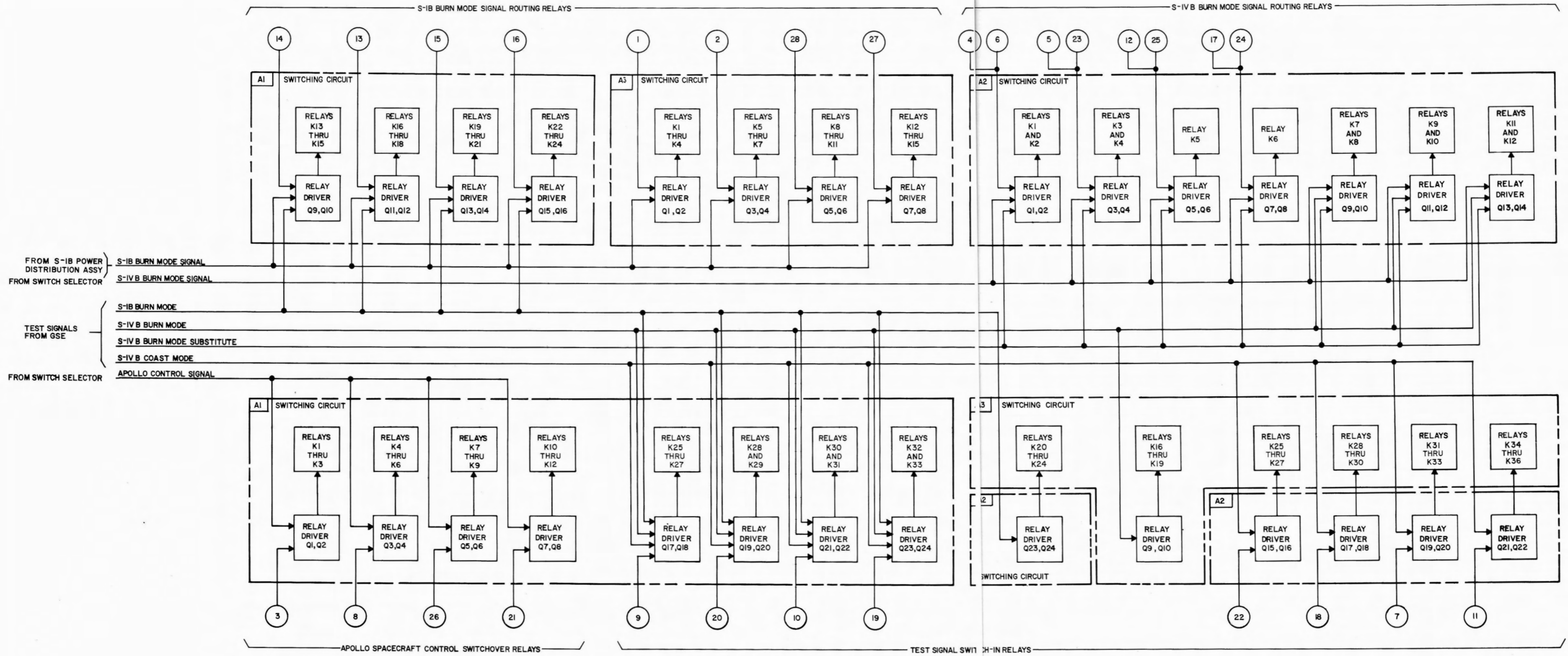


Figure 2-44. Switching Circuits, Functional Block Diagram (Sheet 1 of 2)

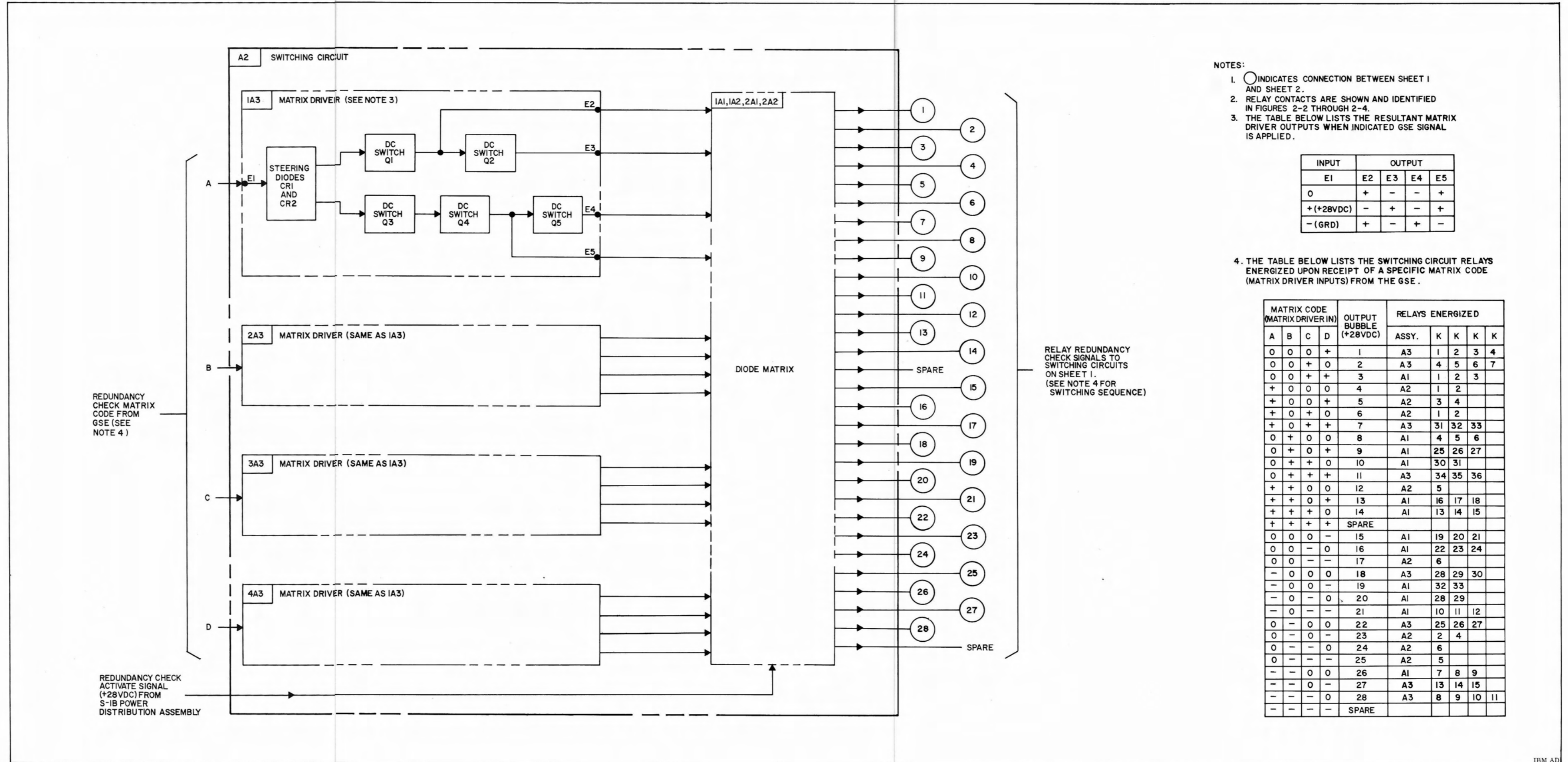
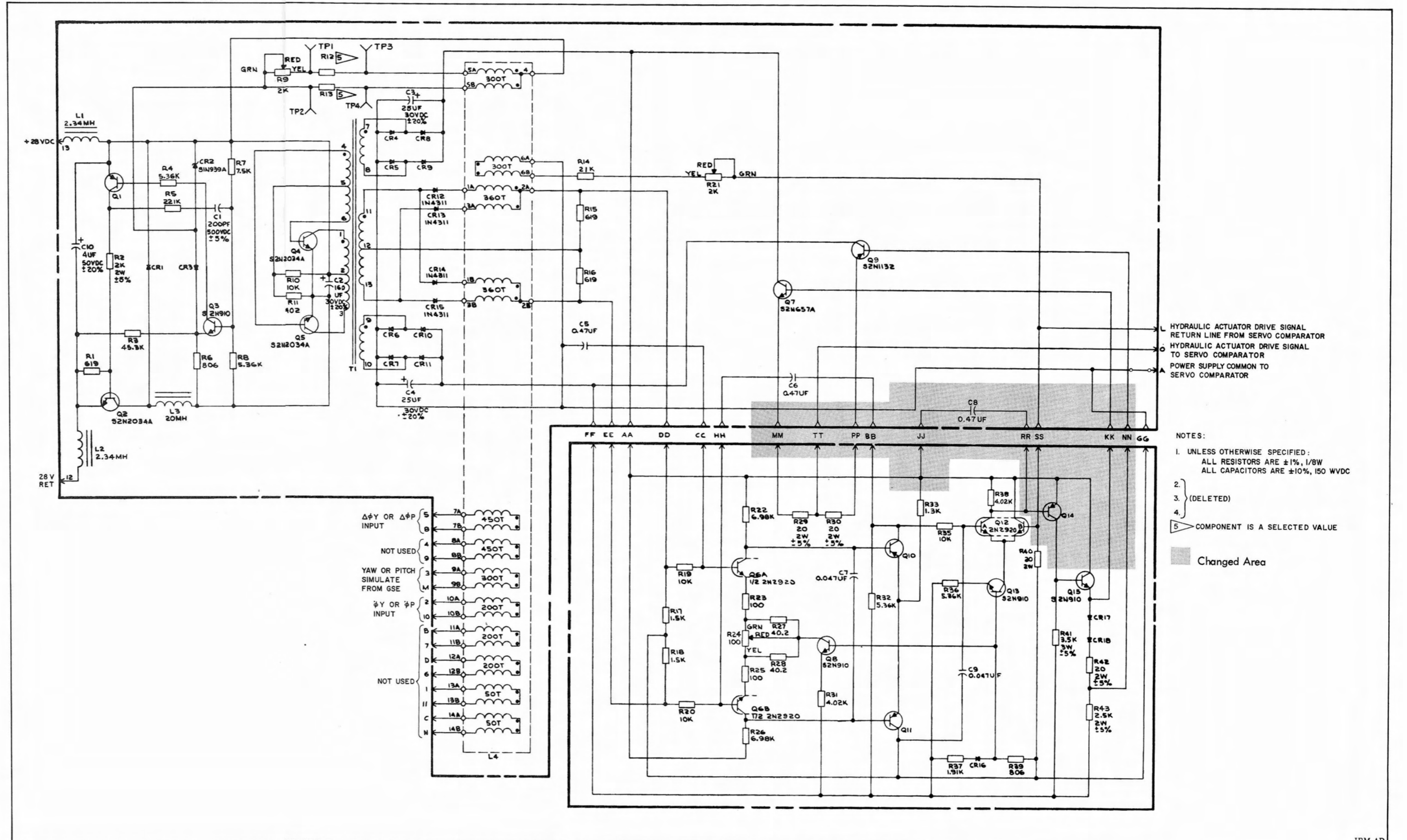


Figure 2-44. Switching Circuits, Functional Block Diagram (Sheet 2 of 2)

Semiconductor Complement

Reference Designation	Function
CR1	CEMF Suppressor
CR2	Voltage Reference
CR3	Temperature Compensator
CR4 thru CR11	Bridge Rectifier
CR12 thru CR15	Magnetic Amplifier Steering Diode
CR16 thru CR18	Temperature Compensator
Q1	1/2 Comparator-Multivibrator
Q2	Switched Regulator
Q3	1/2 Comparator-Multivibrator
Q4, Q5	Inverter
Q6	Push-Pull Amplifier
Q7	1/2 Complementary Symmetry Amplifier
Q8	Temperature Compensator
Q9	1/2 Complementary Symmetry Amplifier
Q10, Q11	Differential Amplifier
Q12	Differential Amplifier
Q13	Temperature Compensator
Q14	1/2 Driver
Q15	1/2 Driver



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Figure 2-56
50 MA Servo Amplifier Schematic Diagram

Changed 1 February 1968

THE CONTENTS OF THE PRECEDING SECTIONS OF THIS MANUAL APPLY TO THE VEHICLE SPECIFIED ON THESE DATA SHEETS EXCEPT FOR THE FOLLOWING DIFFERENCES:

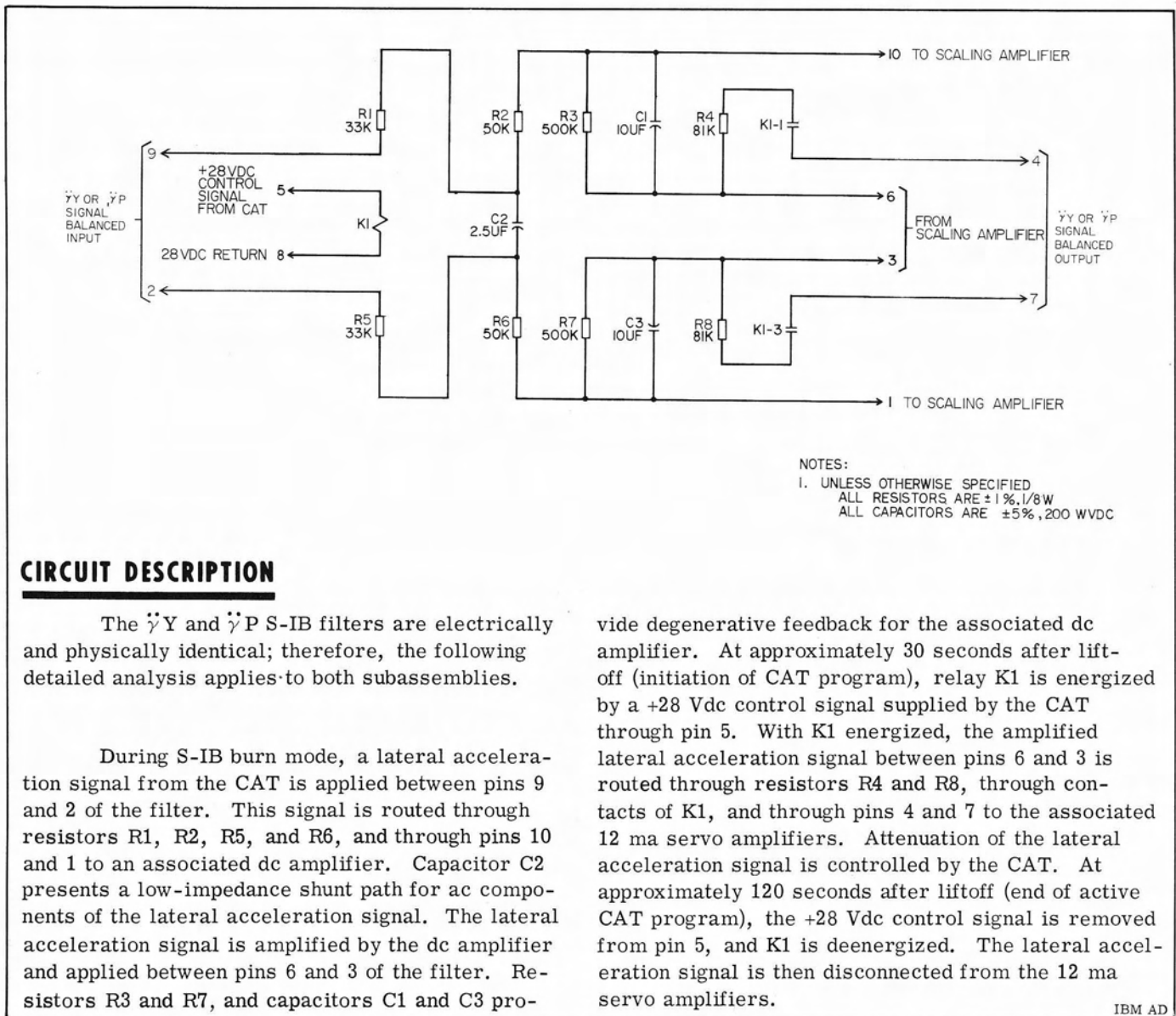
NOTICE

The Flight Control Computer, which is installed in Vehicle 204, was originally designed for the S-IU-208 effectivity. Therefore, the Flight Control Computers for both effectivities are identical.

lines are shown in Figures 204-1 through 204-7 in the following order:

- ϑ̄Y, ϑ̄P S-IB Filter Figure 204-1
- Δϕ Y, Δϕ P S-IB Filter Figure 204-2
- Δϕ R S-IB Filter Figure 204-3
- Δϕ Y, Δϕ P S-IVB Filter Figure 204-4
- ϕ̇ Y, ϕ̇ P S-IB Filter Figure 204-5
- ϕ̇ Y, ϕ̇ P S-IVB Filter Figure 204-6
- ϕ̇ R S-IB Filter Figure 204-7

1 The filter circuits which are used to remove bending mode frequencies from the control signal



CIRCUIT DESCRIPTION

The ϑ̄Y and ϑ̄P S-IB filters are electrically and physically identical; therefore, the following detailed analysis applies to both subassemblies.

During S-IB burn mode, a lateral acceleration signal from the CAT is applied between pins 9 and 2 of the filter. This signal is routed through resistors R1, R2, R5, and R6, and through pins 10 and 1 to an associated dc amplifier. Capacitor C2 presents a low-impedance shunt path for ac components of the lateral acceleration signal. The lateral acceleration signal is amplified by the dc amplifier and applied between pins 6 and 3 of the filter. Resistors R3 and R7, and capacitors C1 and C3 pro-

vide degenerative feedback for the associated dc amplifier. At approximately 30 seconds after lift-off (initiation of CAT program), relay K1 is energized by a +28 Vdc control signal supplied by the CAT through pin 5. With K1 energized, the amplified lateral acceleration signal between pins 6 and 3 is routed through resistors R4 and R8, through contacts of K1, and through pins 4 and 7 to the associated 12 ma servo amplifiers. Attenuation of the lateral acceleration signal is controlled by the CAT. At approximately 120 seconds after liftoff (end of active CAT program), the +28 Vdc control signal is removed from pin 5, and K1 is deenergized. The lateral acceleration signal is then disconnected from the 12 ma servo amplifiers.

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Figure 204-1. ϑ̄Y, ϑ̄P S-IB Filter Schematic Diagram

S-IU-204 Data Sheet

Flt Cont Cmpt (200), Sect. IV

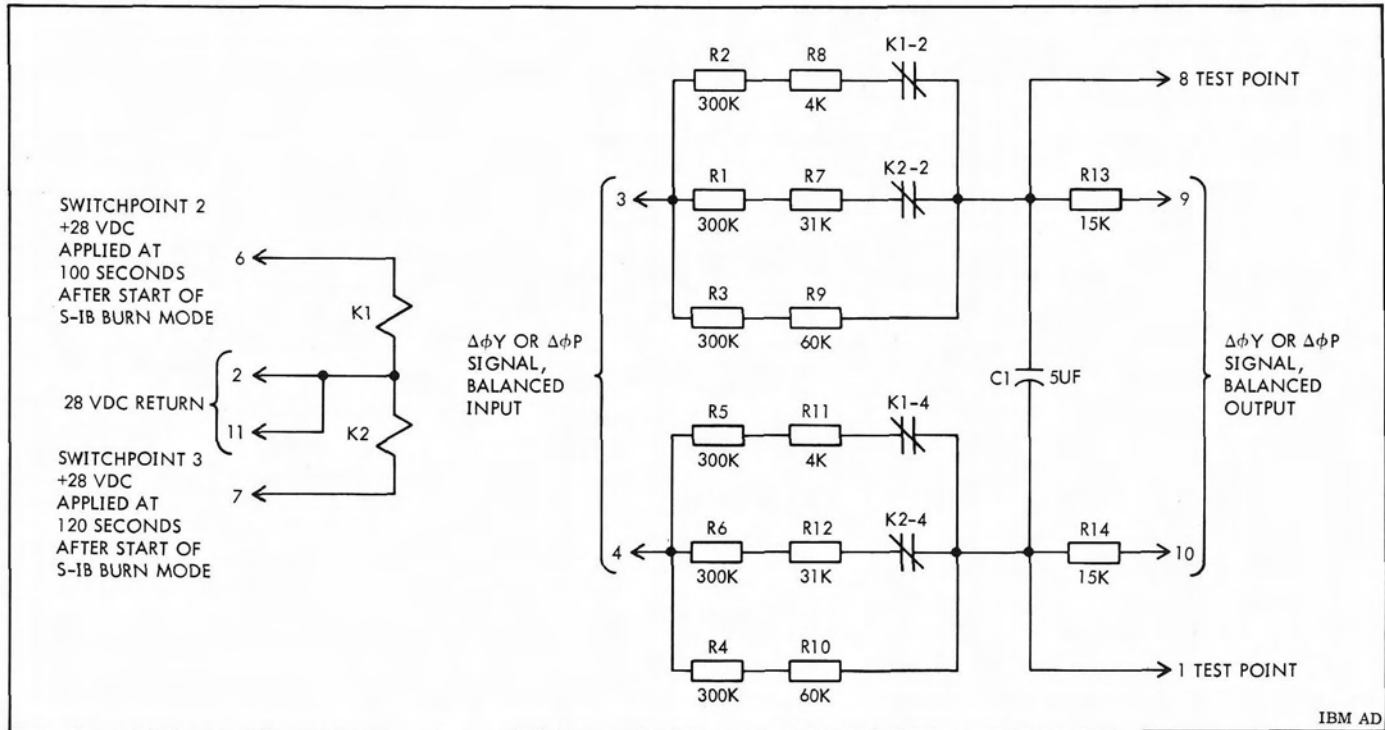
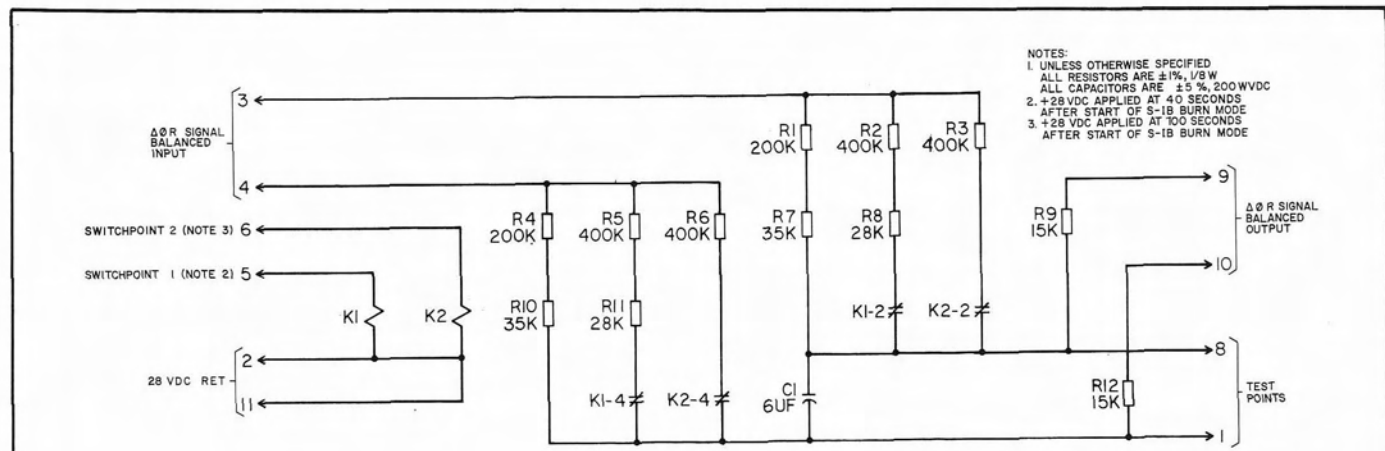


Figure 204-2. $\Delta\phi Y$, $\Delta\phi P$ S-IB Filter Schematic Diagram



- NOTES:
 1. UNLESS OTHERWISE SPECIFIED, ALL RESISTORS ARE $\pm 1\%$, $1/8$ W. ALL CAPACITORS ARE $\pm 5\%$, 200 WVDC.
 2. +28 VDC APPLIED AT 40 SECONDS AFTER START OF S-IB BURN MODE.
 3. +28 VDC APPLIED AT 100 SECONDS AFTER START OF S-IB BURN MODE.

CIRCUIT DESCRIPTION

During the S-IB burn mode, the roll attitude error signal is applied between pins 3 and 4 of the $\Delta\phi R$ S-IB filter. This signal is routed through the series-parallel attenuation networks comprising resistors R1 through R12, and through pins 9 and 10 to associated 12 ma servo amplifiers. During the first 40 seconds of flight, relays K1 and K2 are de-energized, and series resistance in each side of the signal line is approximately 126 kilohms. At 40 seconds after liftoff, the switchpoint 1 signal (+28

Vdc) from the Switch Selector is applied through pin 5 to relay K1. With K1 energized, the series impedance in each side of the signal line is increased to approximately 163 kilohms. At 100 seconds after liftoff, the switchpoint 2 signal from the Switch Selector is applied through pin 6 to relay K2. With K2 energized, the series impedance in each side of the signal line is increased to approximately 250 kilohms. Capacitor C1 presents a low-impedance shunt across the signal line for ac components of the $\Delta\phi R$ signal.

Figure 204-3. $\Delta\phi R$ S-IB Filter Schematic Diagram

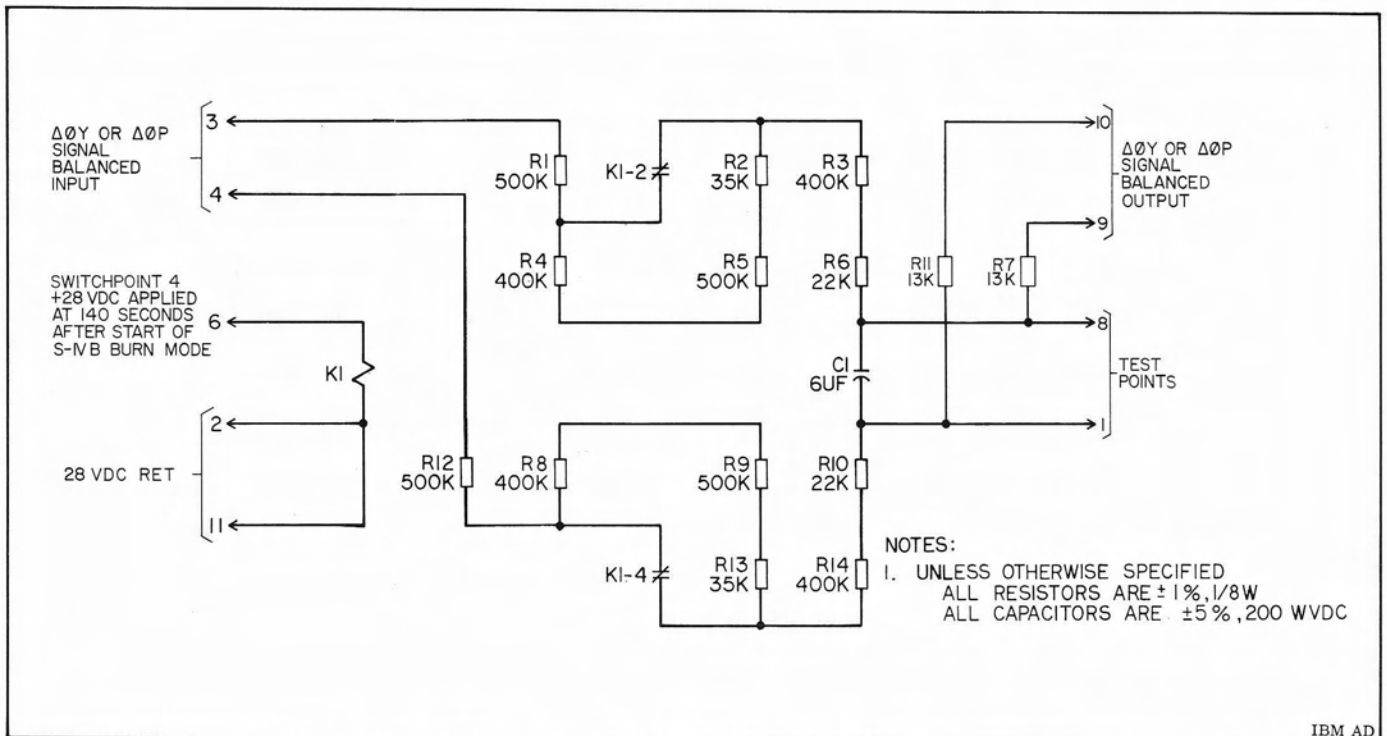


Figure 204-4. Δφ Y, Δφ P S-IVB Filter Schematic Diagram

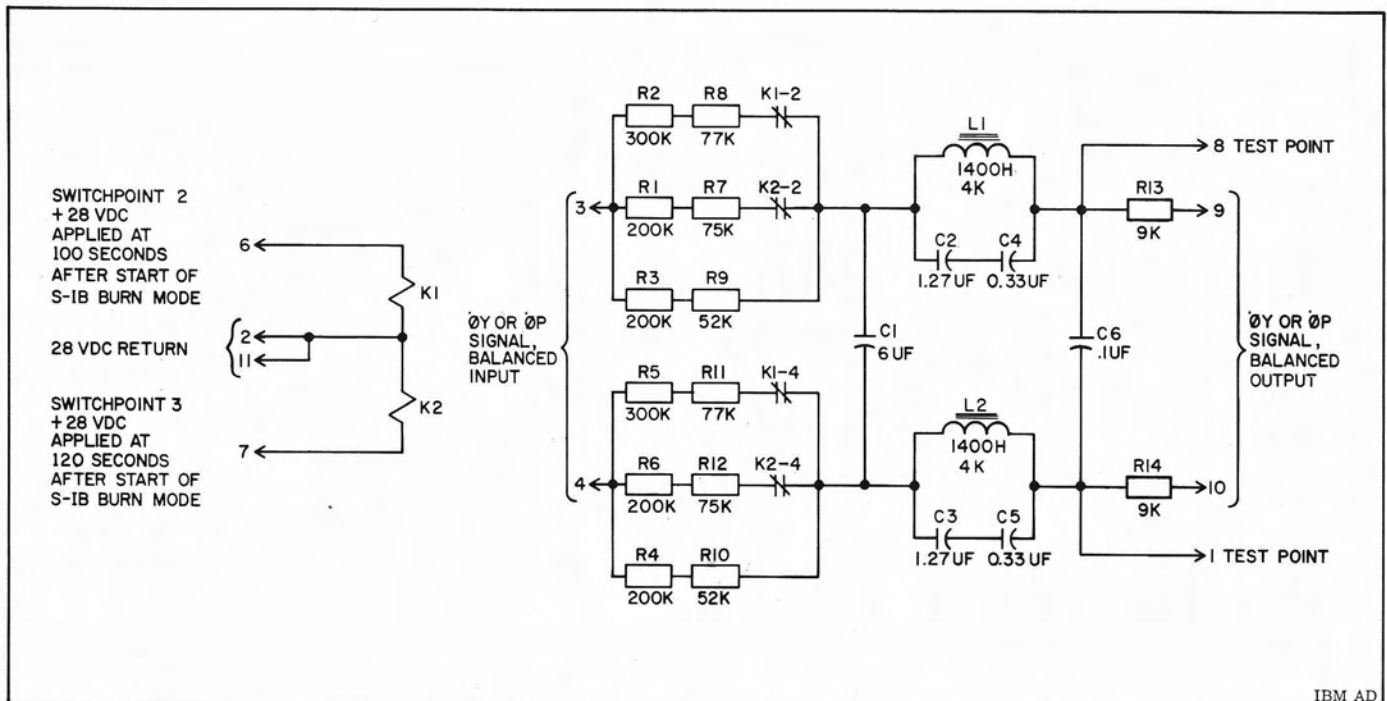


Figure 204-5. φ Y, φ P S-IB Filter Schematic Diagram

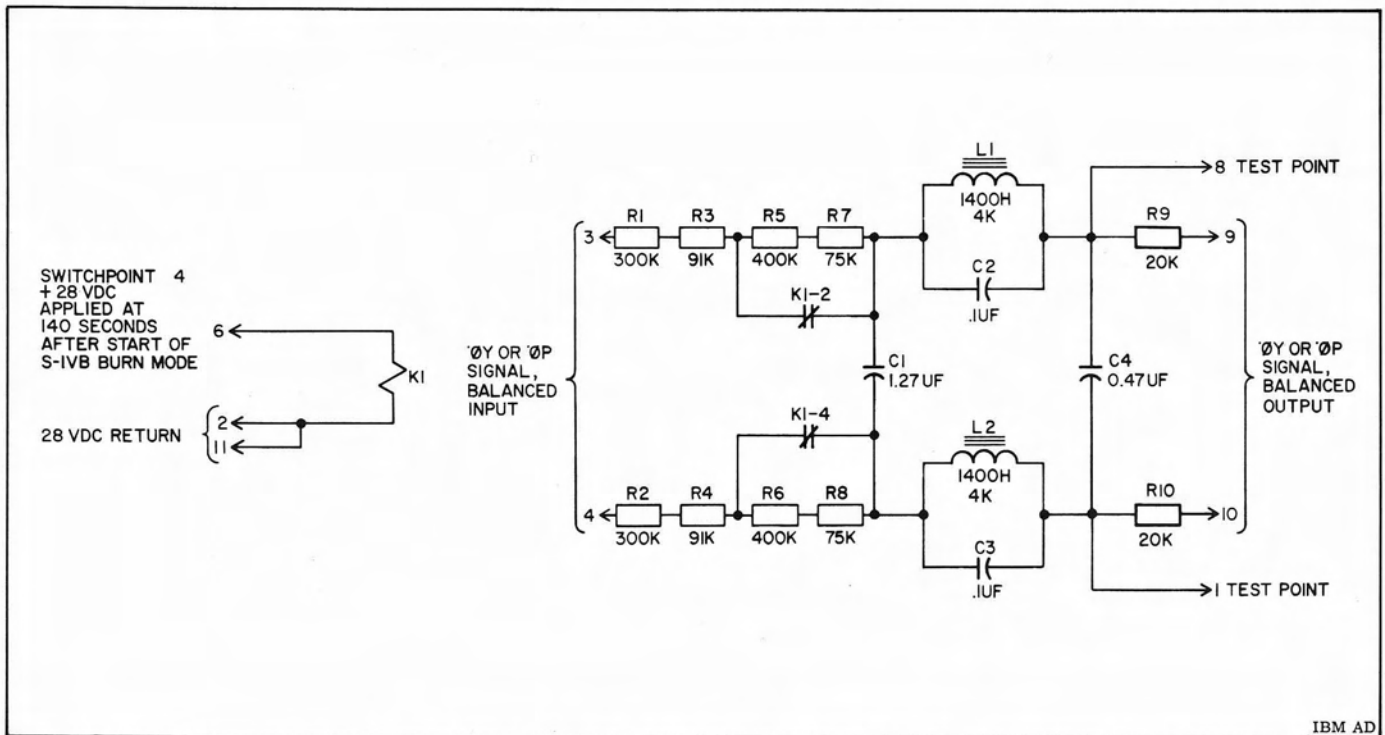
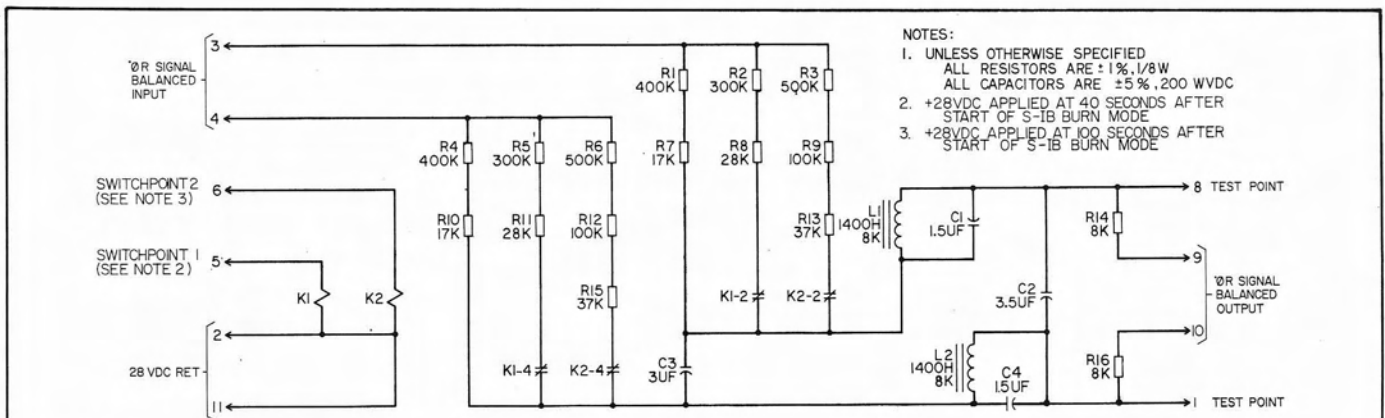


Figure 204-6. ϕ Y, ϕ P S-IVB Filter Schematic Diagram



- NOTES:
1. UNLESS OTHERWISE SPECIFIED ALL RESISTORS ARE $\pm 1\%$, 1/8W ALL CAPACITORS ARE $\pm 5\%$, 200 WVDC
 2. +28VDC APPLIED AT 40 SECONDS AFTER START OF S-IB BURN MODE
 3. +28VDC APPLIED AT 100 SECONDS AFTER START OF S-IB BURN MODE

CIRCUIT DESCRIPTION

During the S-IB burn mode, the roll angular rate (ϕ R) signal is applied between pins 3 and 4 of the filter. During the first 40 seconds of flight, this signal is routed through the series parallel network comprised of resistors R1 through R13, and R4 through R15; then through inductors L1 and L2, through resistors R14 and R16, and through pins 9 and 10 to the associated 12 ma servo amplifiers. Series resistance in each side of the signal line is approximately 159 kilohms. At 40 seconds after liftoff, the switchpoint 1 signal (+28 Vdc) from the Switch Selector is applied through pin 5 to relay K1. With K1 energized, the series resistance in each side of the signal line is increased to approximately 268 kilohms.

At 100 seconds after liftoff, the switchpoint 2 signal from the Switch Selector is applied through pin 6 to relay K2. With K2 energized, the series resistance in each side of the signal line is increased to approximately 433 kilohms. Inductors L1 and L2 and capacitors C1 and C4 are connected to form low-Q parallel L/C (tank) circuits in each side of the signal line. These parallel L/C circuits provide maximum signal rejection at approximately 3.3 Hz. However, since circuit Q is quite low (less than 5), effective signal rejection over a wide frequency range is obtained. Capacitors C3 and C2 present low-impedance shunts across the signal line for ac components of the ϕ R signal.

Figure 204-7. ϕ R S-IB Filter Schematic Diagram

2

The yaw and pitch simulate circuitry has been eliminated in the S-IU-204 Flight Control Computer. This circuitry is shown in a simplified block diagram in Figure 204-8. (It also appears on sheets 1, 2, and 7 of the basic Flight Control Computer Schematic Diagram, Figure 2-46, which is located in the back pocket of this manual.) The wiring and components which have been removed are shown in shaded areas in Figure 204-8. All other wiring and components are unaffected by this change.

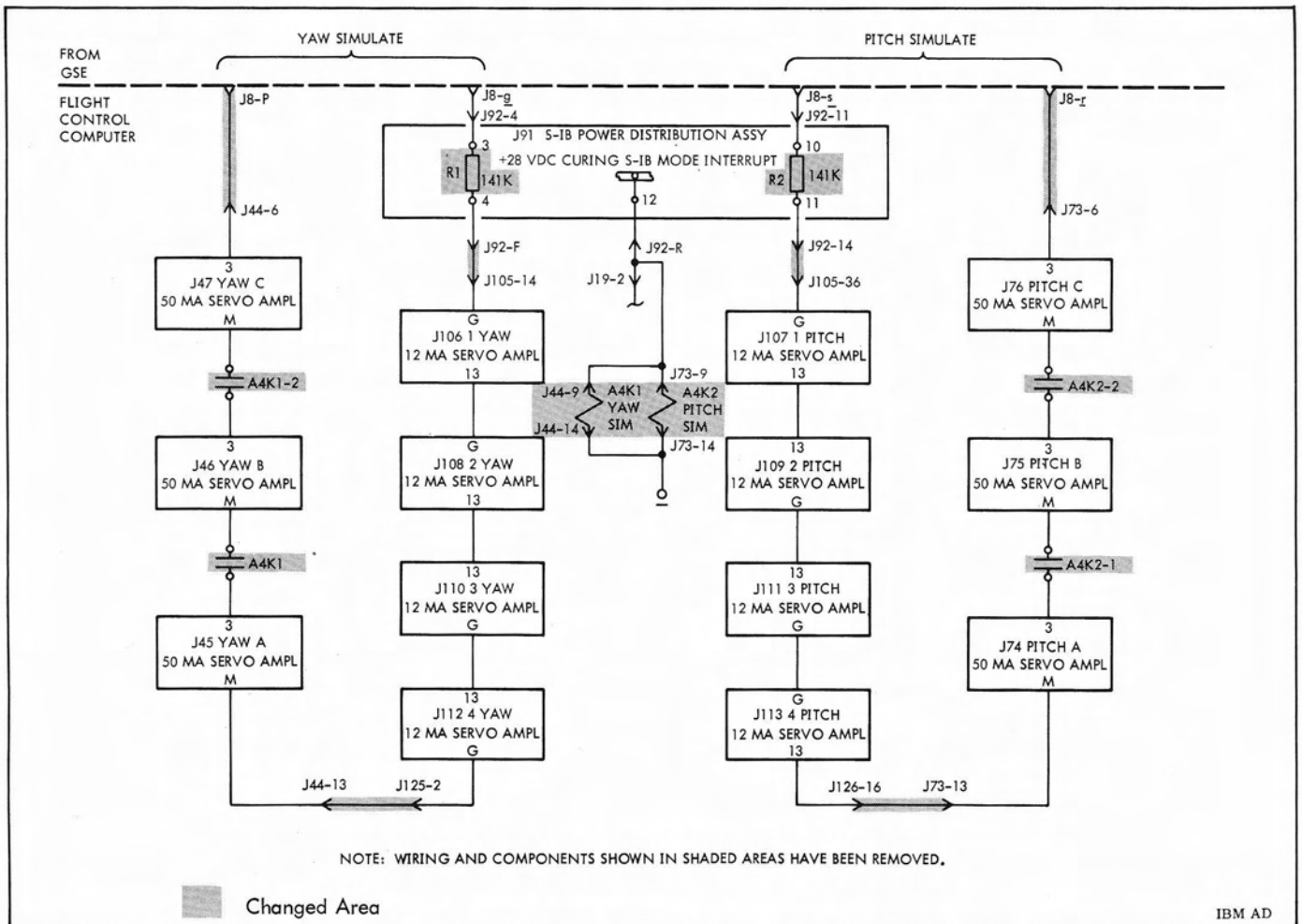


Figure 204-8. Yaw-Pitch Simulate Circuit, Simplified Block Diagram

THE CONTENTS OF THE PRECEDING SECTIONS OF THIS MANUAL APPLY TO THE VEHICLE SPECIFIED ON THESE DATA SHEETS EXCEPT FOR THE FOLLOWING DIFFERENCES:

NOTICE

The Flight Control Computer, which is installed in Vehicle 205 was originally designed for the S-IU-209 effectivity. Therefore, the Flight Control Computers for both effectivities are identical.

1 In the S-IU-205 Flight Control Computer, the yaw and pitch simulate circuitry has been eliminated in the same manner as in the S-IU-204 Flight Control Computer. See change number 2 in the S-IU-204 Data Sheets for detailed information about this change.

2 The filter circuits which are used to remove bending mode frequencies from the control signal lines are shown in Figures 205-1 through 205-5 in the following order:

- \ddot{Y} , \ddot{P} S-IB Filter Figure 205-1
- $\Delta\phi Y$, $\Delta\phi P$ S-IB Filter . . . Figure 205-2
- $\Delta\phi Y$, $\Delta\phi P$ S-IVB Filter . . . Figure 205-3
- ϕY , ϕP S-IB Filter Figure 205-4
- ϕY , ϕP S-IVB Filter Figure 205-5

The $\Delta\phi R$ S-IB Filter and the ϕR S-IB Filter are identical to those used in the S-IU-204 Flight Control Computer. For detailed information about these filters, see the S-IU-204 Data Sheets.

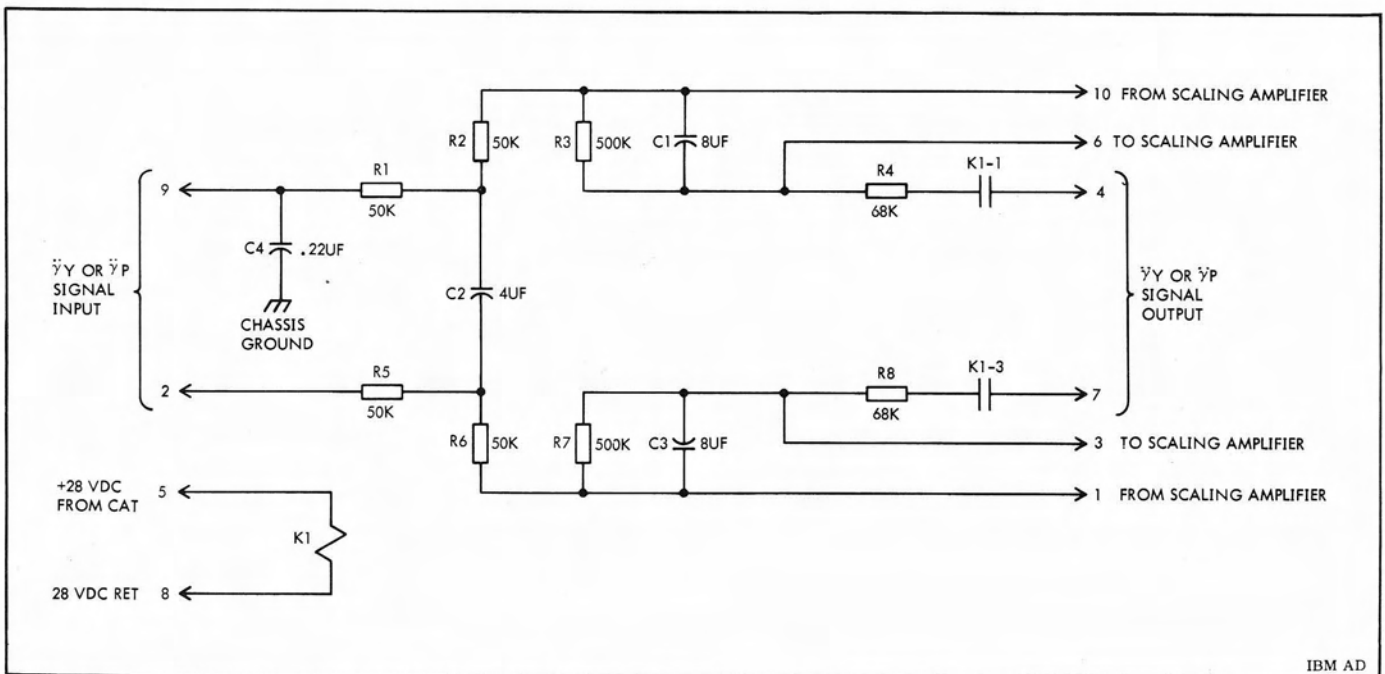


Figure 205-1. \ddot{Y} , \ddot{P} S-IB Filter Schematic Diagram

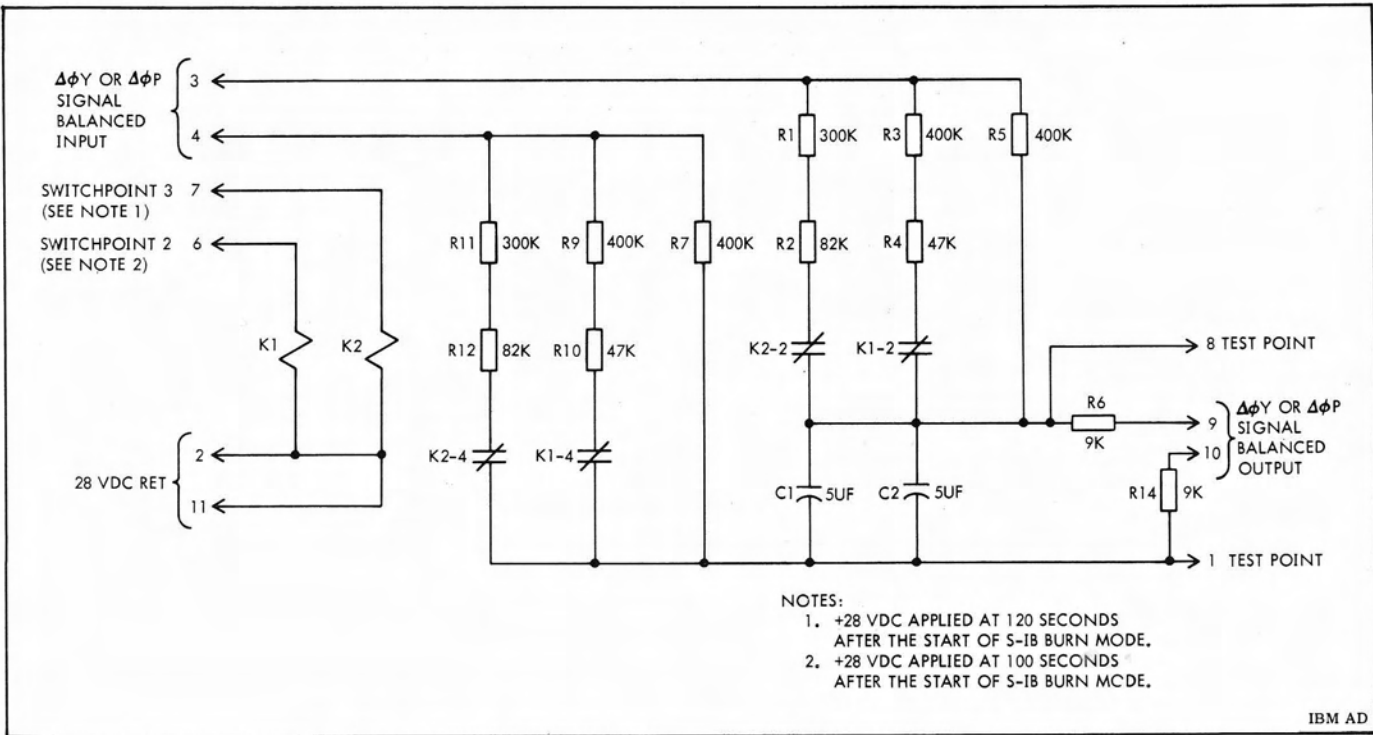


Figure 205-2. $\Delta\phi Y$, $\Delta\phi P$ S-IB Filter Schematic Diagram

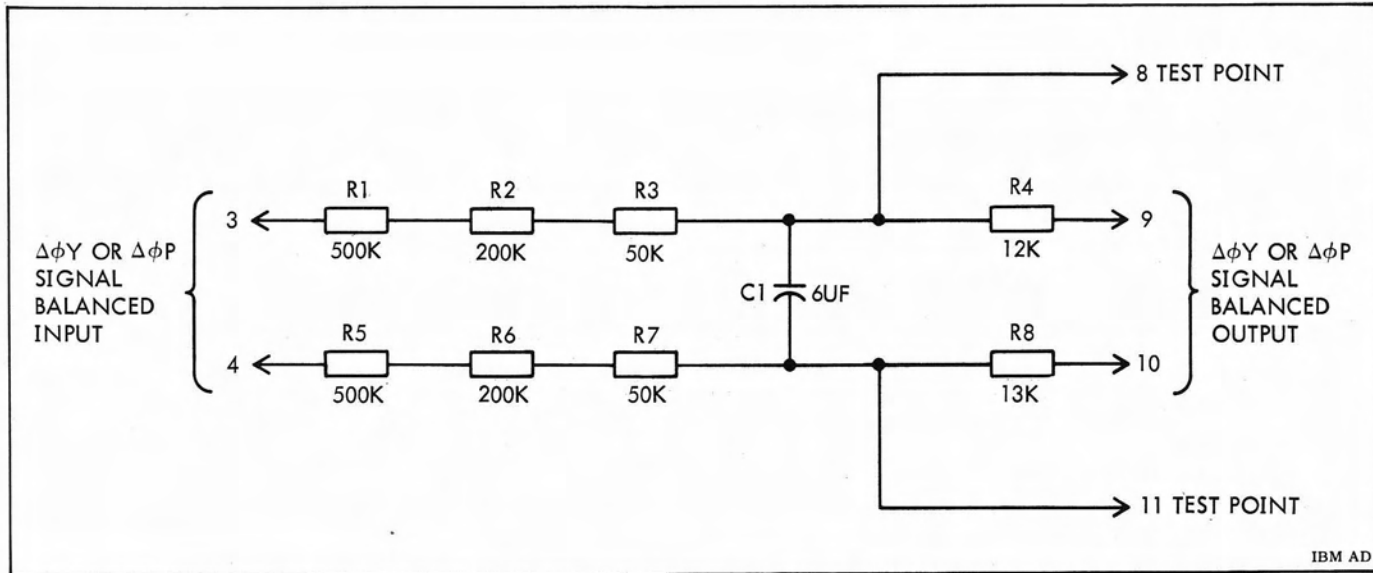
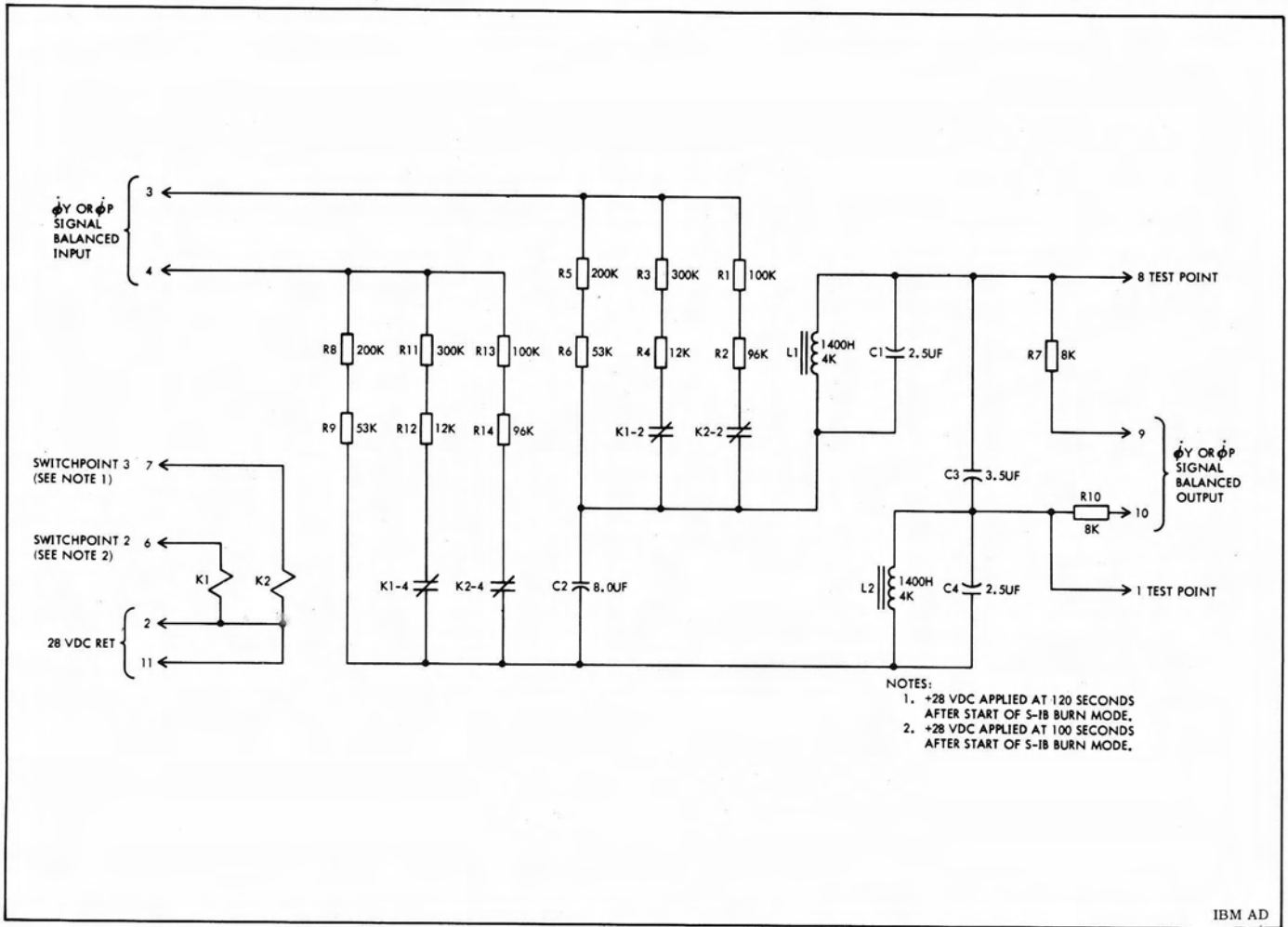


Figure 205-3. $\Delta\phi Y$, $\Delta\phi P$ S-IVB Filter Schematic Diagram



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Figure 205-4. ϕY , ϕP S-IB Filter Schematic Diagram

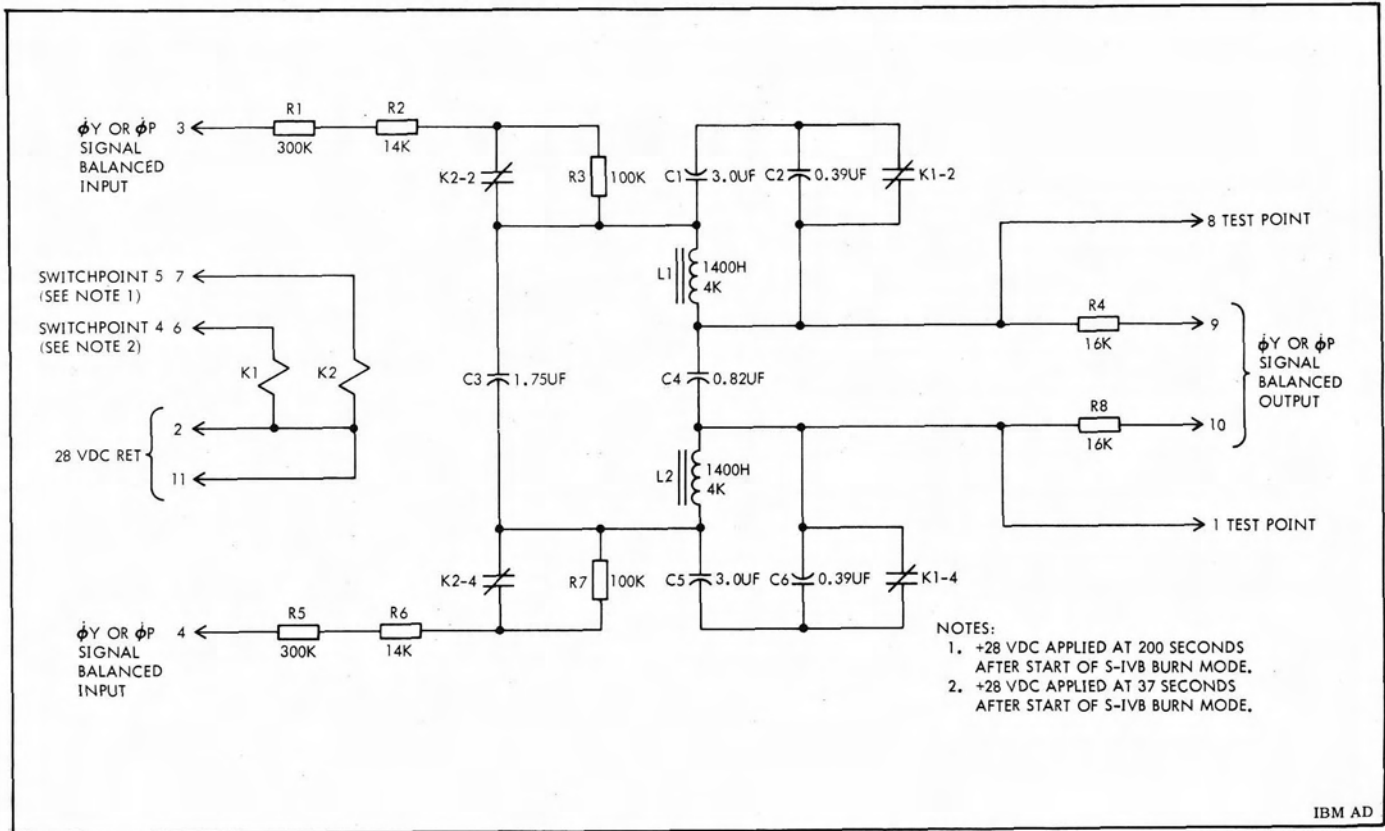


Figure 205-5. ϕY , ϕP S-IVB Filter Schematic Diagram

THE CONTENTS OF THE PRECEDING SECTIONS OF THIS MANUAL APPLY TO THE VEHICLE SPECIFIED ON THESE DATA SHEETS EXCEPT FOR THE FOLLOWING DIFFERENCES

1 The S-IU-206 filter circuits are described in the following figures:

- \ddot{Y} , \ddot{P} S-IB Filter Figure 206-1
- $\Delta\phi$ Y, $\Delta\phi$ P S-IB Filter Figure 206-2
- $\Delta\phi$ Y, $\Delta\phi$ P S-IVB Filter Figure 206-3
- $\dot{\phi}$ Y, $\dot{\phi}$ P S-IB Filter Figure 206-4
- $\dot{\phi}$ Y, $\dot{\phi}$ P S-IVB Filter Figure 206-5

The $\Delta\phi$ R S-IB Filter and the $\dot{\phi}$ R S-IB Filter are identical to those used in the S-IU-204 Flight Control Computer. For detailed information about these filters, see the S-IU-204 Data Sheets.

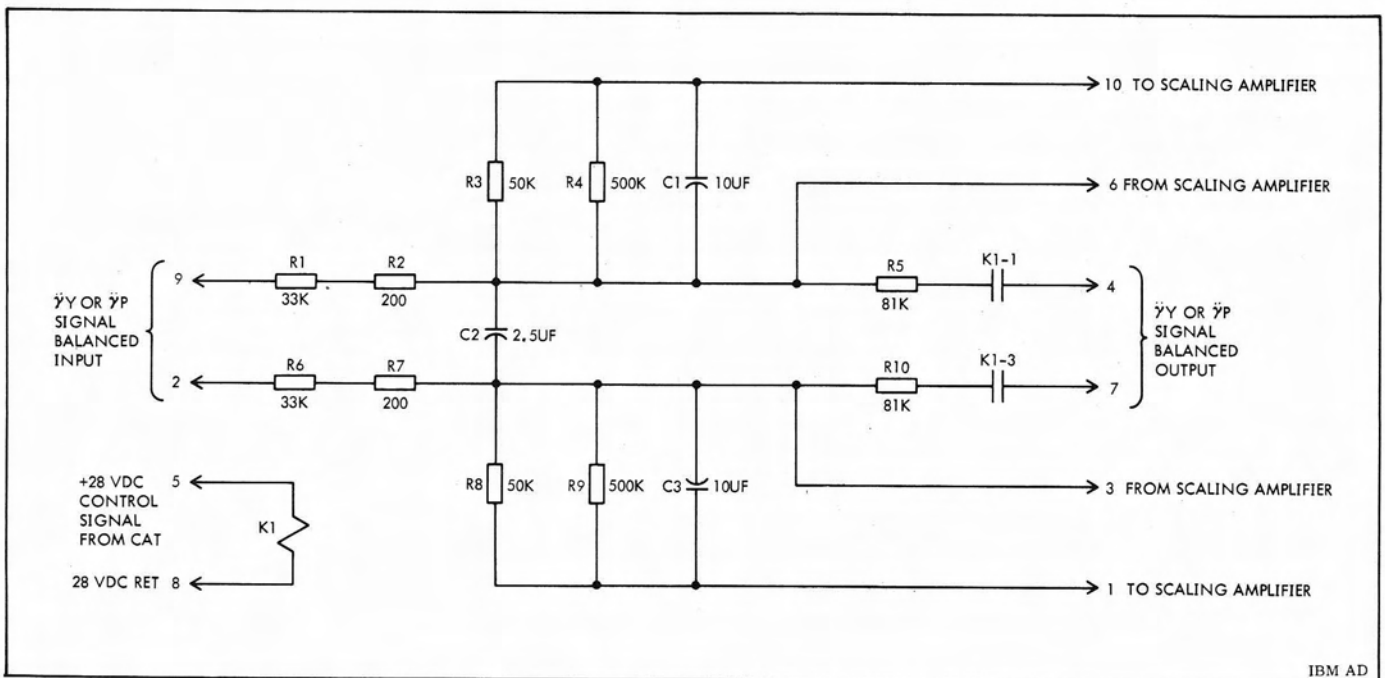


Figure 206-1. \ddot{Y} , \ddot{P} S-IB Filter Schematic Diagram

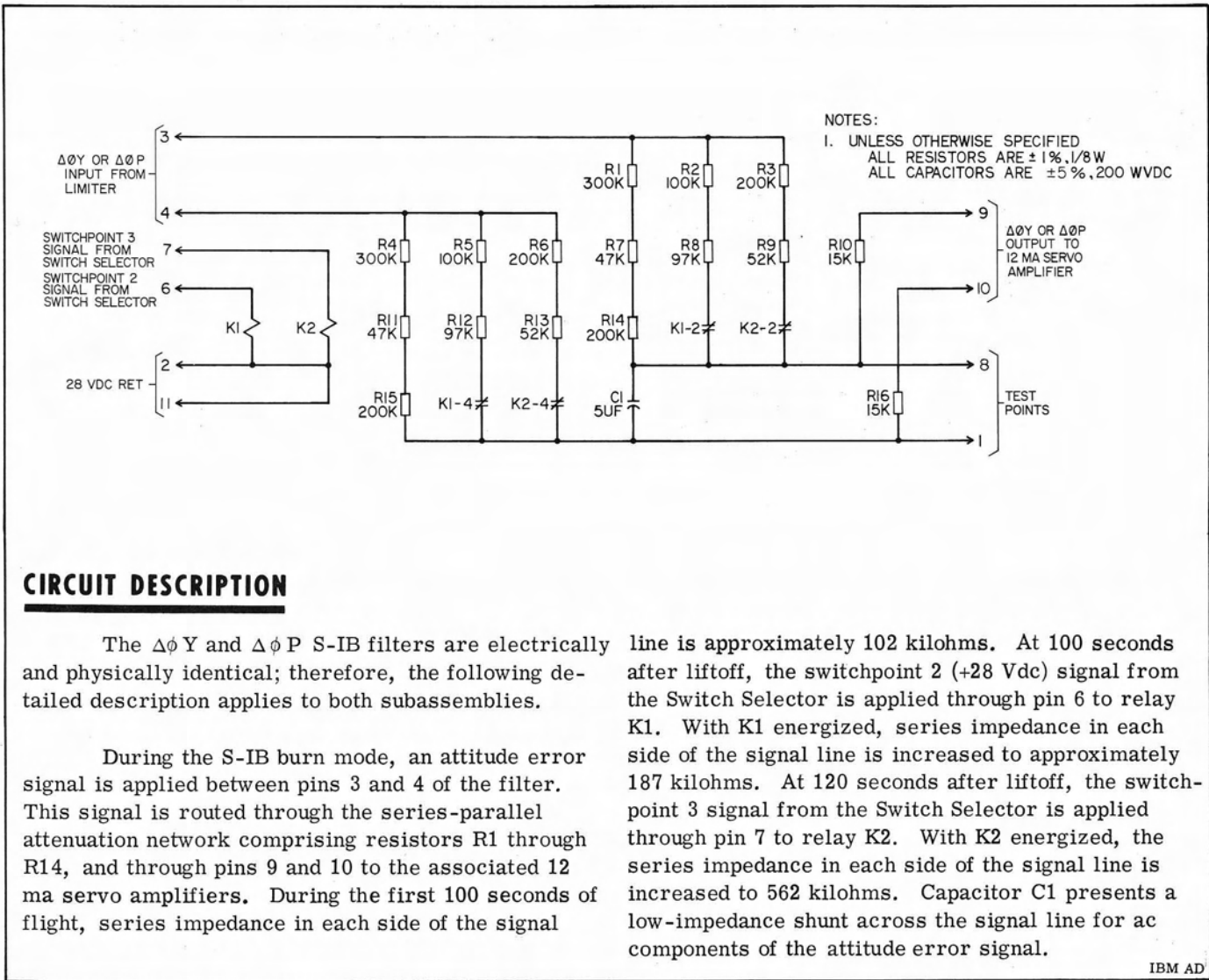
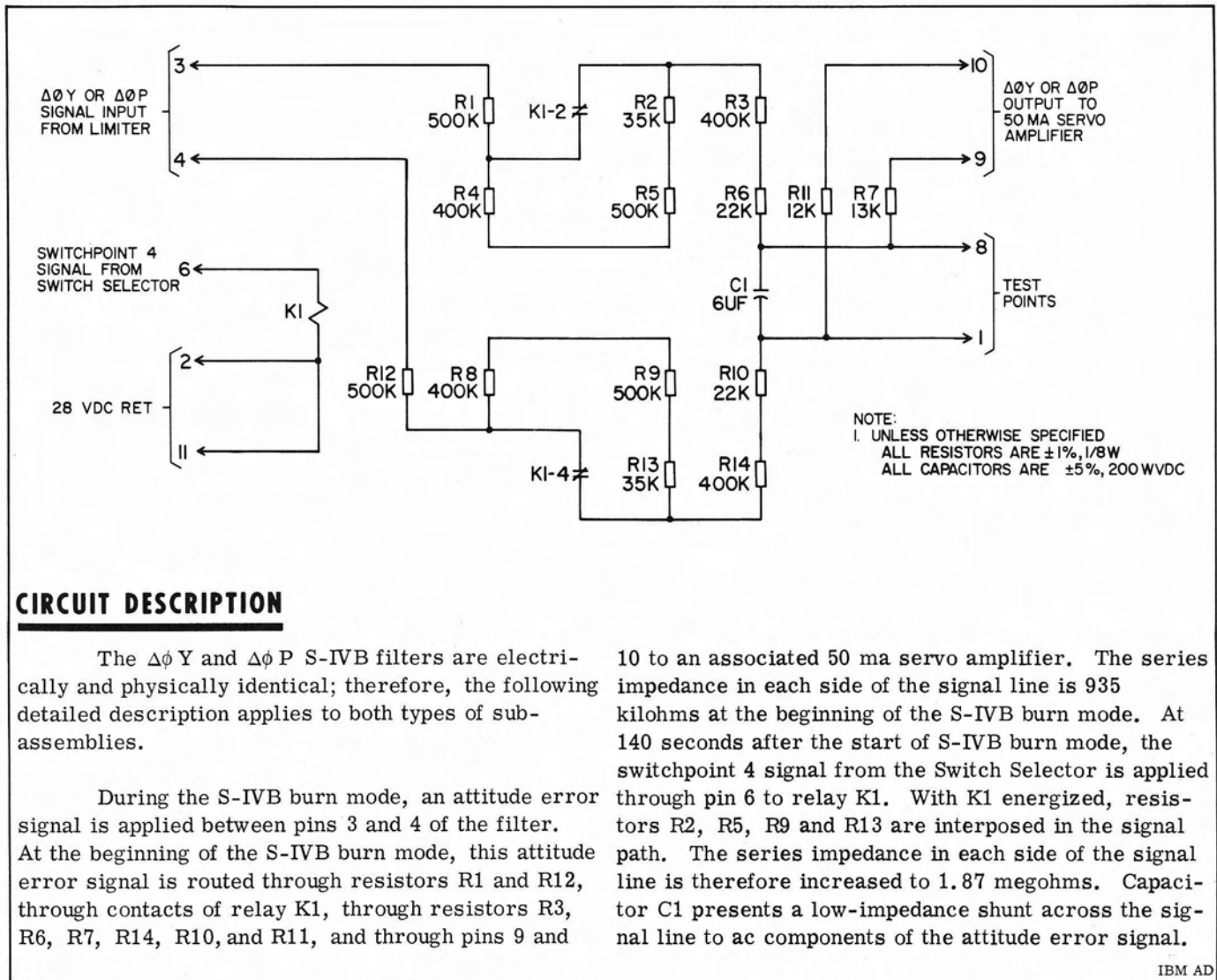


Figure 206-2. $\Delta\phi Y$, $\Delta\phi P$ S-IB Filter Schematic Diagram



CIRCUIT DESCRIPTION

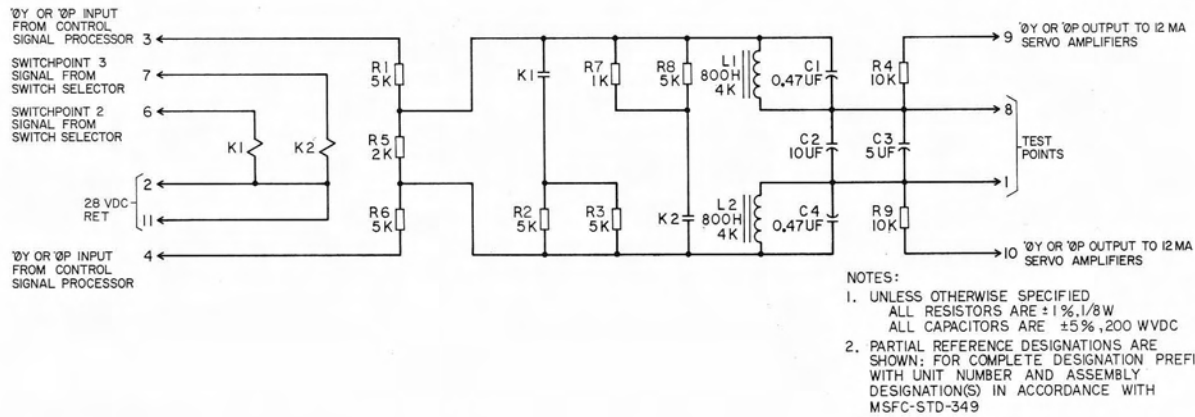
The $\Delta\phi Y$ and $\Delta\phi P$ S-IVB filters are electrically and physically identical; therefore, the following detailed description applies to both types of sub-assemblies.

During the S-IVB burn mode, an attitude error signal is applied between pins 3 and 4 of the filter. At the beginning of the S-IVB burn mode, this attitude error signal is routed through resistors R1 and R12, through contacts of relay K1, through resistors R3, R6, R7, R14, R10, and R11, and through pins 9 and

10 to an associated 50 ma servo amplifier. The series impedance in each side of the signal line is 935 kilohms at the beginning of the S-IVB burn mode. At 140 seconds after the start of S-IVB burn mode, the switchpoint 4 signal from the Switch Selector is applied through pin 6 to relay K1. With K1 energized, resistors R2, R5, R9 and R13 are interposed in the signal path. The series impedance in each side of the signal line is therefore increased to 1.87 megohms. Capacitor C1 presents a low-impedance shunt across the signal line to ac components of the attitude error signal.

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Figure 206-3. $\Delta\phi Y$, $\Delta\phi P$ S-IVB Filter Schematic Diagram



CIRCUIT DESCRIPTION

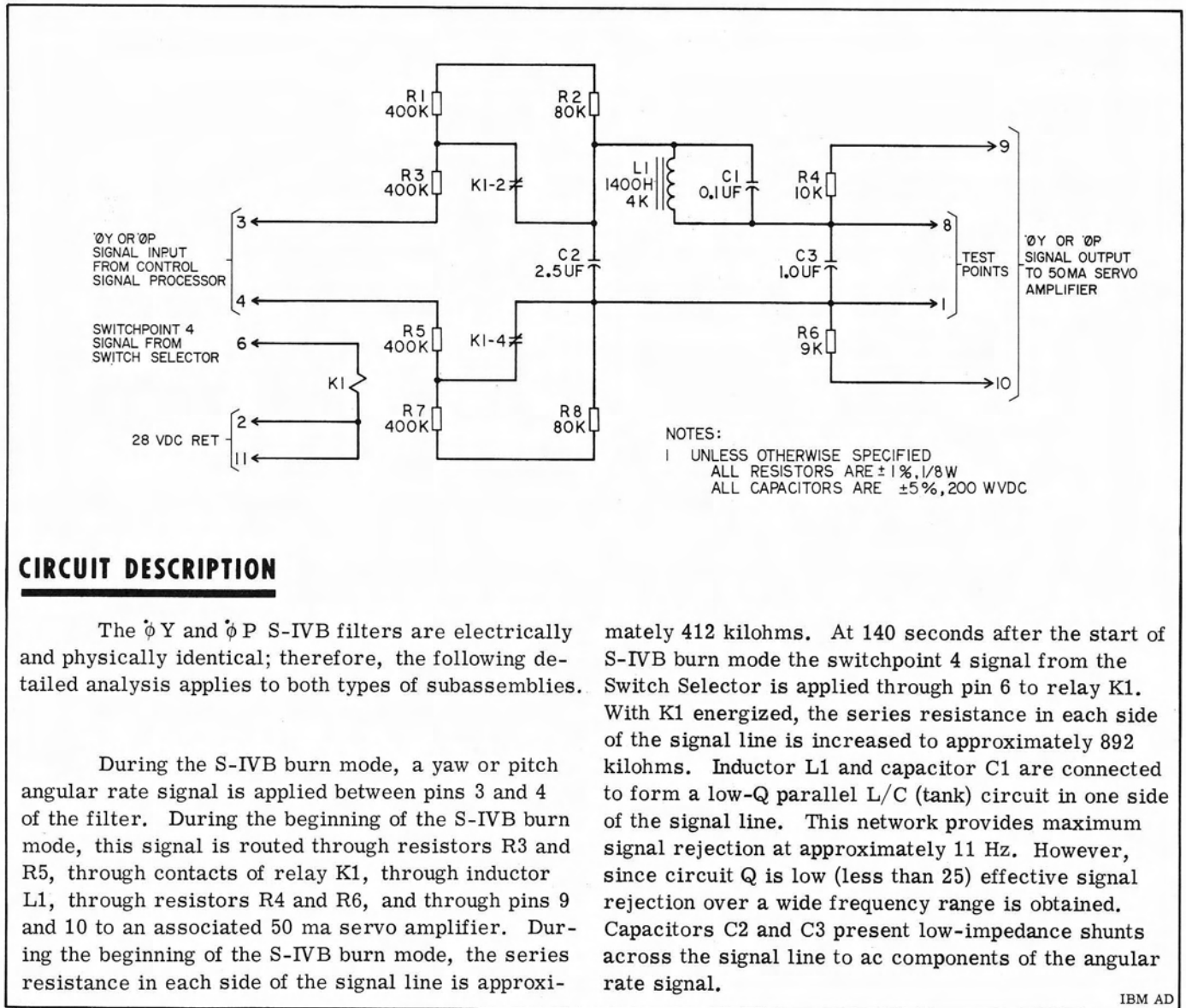
The ϕY and ϕP S-IB filters are electrically and physically identical; therefore, the following detailed description applies to both subassemblies.

During the S-IB burn mode, an angular rate signal is applied between pin 3 and 4 of the filter. This signal is routed through resistors R1 and R6, through inductors L1 and L2, through resistors R4 and R9, and through pins 9 and 10 to the associated 12 ma servo amplifiers. During the first 100 seconds of flight, resistor R5 presents a 2000-ohm resistive shunt across the signal line. At 100 seconds after liftoff, the switchpoint 2 signal (+28 Vdc) from the Switch Selector is applied through pin 6 to relay K1. With K1 energized, the shunt resistance across the

signal line is reduced to approximately 1111 ohms. At 120 seconds after liftoff, the switchpoint 3 signal from the Switch Selector is applied through pin 7 to relay K2. With K2 energized, the shunt resistance across the signal line is reduced to approximately 476 ohms. Inductors L1 and L2 and capacitors C1 and C4 are connected to form low-Q parallel L/C (tank) circuits in each side of the signal line. These parallel L/C circuits provide maximum signal rejection at approximately 8.5 Hz. However, since circuit Q is quite low (less than 9), effective signal rejection over a wide frequency range is obtained. Capacitors C2 and C3 present a low-impedance shunt across the line to ac components of the angular rate signal.

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Figure 206-4. ϕY , ϕP S-IB Filter Schematic Diagram

Figure 206-5. ϕY , ϕP S-IVB Filter Schematic Diagram

2 In the S-IU-206 Flight Control Computer, the yaw and pitch simulate circuitry has been disabled as shown in Figure 206-6. (It also appears on sheets 1, 2, and 7 of the basic Flight Control Computer Schematic Diagram, Figure 2-46, which is located in the back pocket of this manual.) In this figure, only the affected components and circuitry are shown. All other components and circuitry are not affected by this change.

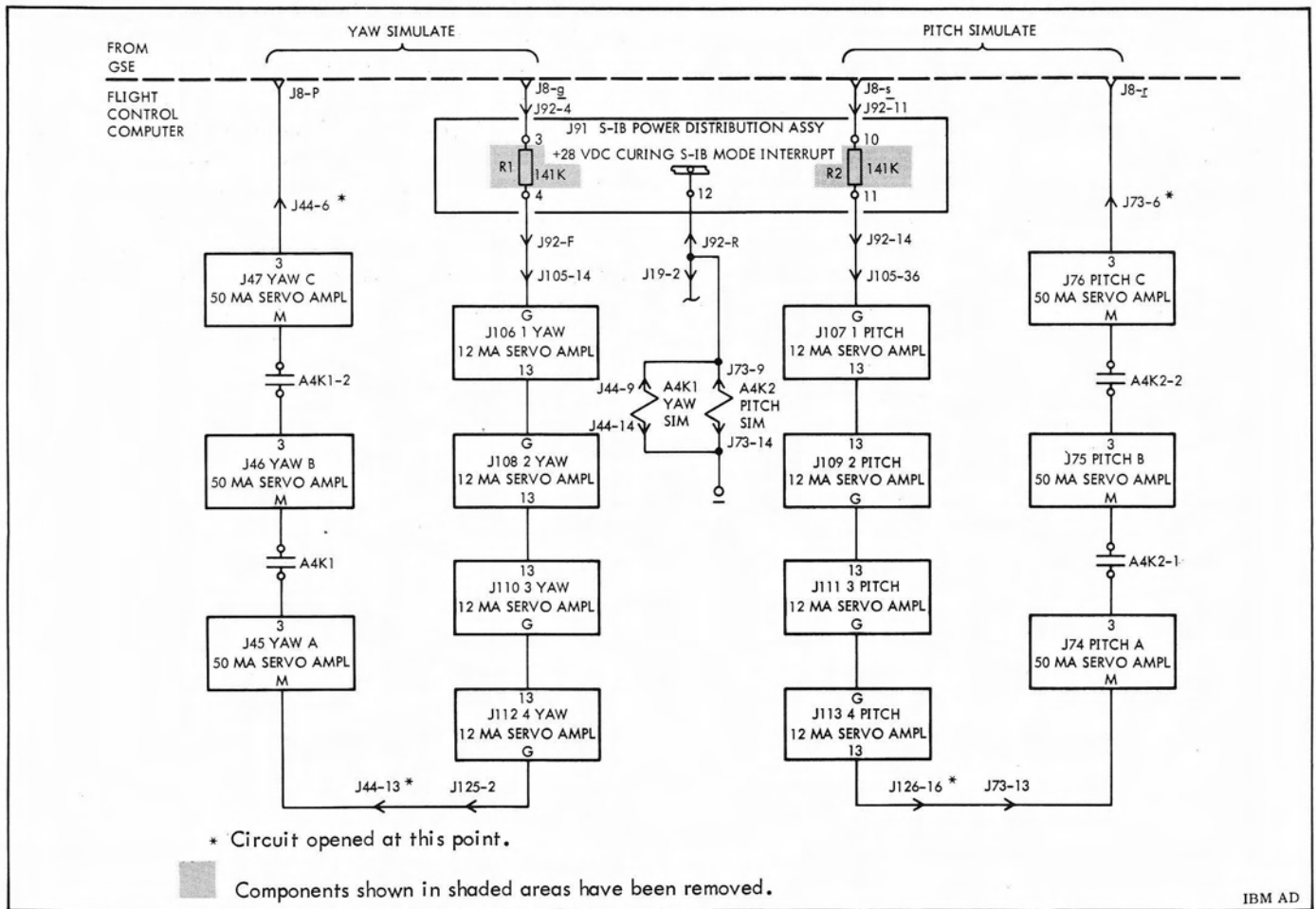


Figure 206-6. Yaw-Pitch Simulate Circuit, Simplified Block Diagram

THE CONTENTS OF THE PRECEDING SECTIONS OF THIS MANUAL APPLY TO THE VEHICLE SPECIFIED ON THESE DATA SHEETS EXCEPT FOR THE FOLLOWING DIFFERENCES:

1 The Flight Control Computer which is installed in the AS-207 Saturn Launch Vehicle is identical to the S-IU-205 Flight Control Computer. For detailed information about this Computer, refer to the S-IU-205 Data Sheets.

THE CONTENTS OF THE PRECEDING SECTIONS OF THIS MANUAL APPLY TO THE VEHICLE SPECIFIED ON THESE DATA SHEETS EXCEPT FOR THE FOLLOWING DIFFERENCES:

1 The Flight Control Computer which is installed in the AS-208 Saturn Launch Vehicle is identical to the S-IU-204 Flight Control Computer. For detailed information about this Computer, refer to the S-IU-204 Data Sheets.

THE CONTENTS OF THE PRECEDING SECTIONS OF THIS MANUAL APPLY TO THE VEHICLE SPECIFIED ON THESE DATA SHEETS EXCEPT FOR THE FOLLOWING DIFFERENCES:

1 The Flight Control Computer which is installed in the AS-209 Saturn Launch Vehicle is identical to the S-IU-205 Flight Control Computer. For detailed information about this Computer, refer to the S-IU-205 Data Sheets.

THE CONTENTS OF THE PRECEDING SECTIONS OF THIS MANUAL APPLY TO THE
VEHICLE SPECIFIED ON THESE DATA SHEETS EXCEPT FOR THE FOLLOWING DIFFERENCES

1 In the S-IU-210 Flight Control Computer, the yaw and pitch simulate circuitry has been eliminated in the same manner as in the S-IU-204 Flight Control Computer. See change number 2 in the S-IU-204 Data Sheets for detailed information about this change.

THE CONTENTS OF THE PRECEDING SECTIONS OF THIS MANUAL APPLY TO THE VEHICLE SPECIFIED ON THESE DATA SHEETS EXCEPT FOR THE FOLLOWING DIFFERENCES:

1 In the S-IU-211 Flight Control Computer, the yaw and pitch simulate circuitry has been eliminated in the same manner as in the S-IU-204 Flight Control Computer. See change number 2 in the S-IU-204 Data Sheets for detailed information about this change.

THE CONTENTS OF THE PRECEDING SECTIONS OF THIS MANUAL APPLY TO THE
VEHICLE SPECIFIED ON THESE DATA SHEETS EXCEPT FOR THE FOLLOWING DIFFERENCES

1 In the S-IU-212 Flight Control Computer, the yaw and pitch simulate circuitry has been eliminated in the same manner as in the S-IU-204 Flight Control Computer. See change number 2 in the S-IU-204 Data Sheets for detailed information about this change.

SECTION III

PREVENTIVE MAINTENANCE AND REPAIR

3-1 PREVENTIVE MAINTENANCE (Not Applicable)

3-2 REPAIR

All repairs to the Flight Control Computer will be accomplished by component replacement. If failure analysis determines repair is possible, the entire unit should be returned to the proper agency for repair.

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