

IBM

Saturn IB/V - Instrument Unit

Technical Manual

MSFC No. III-5-510-21

IBM No. 66-966-0025

Flight Control Computer

(S-IU-501)



MSFC No. III-5-510-21

IBM No. 66-966-0025

TECHNICAL MANUAL

FLIGHT CONTROL COMPUTER
(S-IU-501)

Prepared under Contract

NAS 8-14000

by

International Business Machines Corporation

Federal Systems Division

Huntsville, Alabama

15 July 1966

LIST OF EFFECTIVE PAGES

NOTE: The portion of the text affected by the changes is indicated by a vertical line in the outer margins of the page.

Page	Issue
Title	Original
A	Original
i thru iv	Original
1-1 thru 1-3	Original
1-4 Blank	Original
1-5 thru 1-19	Original
1-20 Blank	Original
2-1 thru 2-11	Original
2-12 Blank	Original
2-13	Original
2-14 Blank	Original
2-15	Original
2-16 Blank	Original
2-17 thru 2-39	Original
2-40 Blank	Original
2-41	Original
2-42 Blank	Original
2-43 thru 2-45	Original
2-46 Blank	Original
3-1	Original
3-2 Blank	Original

* The asterisk indicates pages changed, added, or deleted by the current change.

LIST OF RELATED DOCUMENTS

The following manuals have been published under NASA contracts as source and reference information on IU systems and/or components.

Title	Number		Published Date	Changed Date
	IBM	MSFC		
Structure Repair Manual	65-966-0011H	III-5-510-1	1 September 1965	
Auxiliary Power Distributors	65-966-0013H	III-5-510-2	1 October 1965	
Power Distributor	65-966-0014H	III-5-510-3	1 October 1965	
Emergency Detection System Distributor	65-966-0015H	III-5-510-4	1 October 1965	
Control Distributor	65-966-0016H	III-5-510-5	1 October 1965	
Measuring Distributors	65-966-0017H	III-5-510-6	1 October 1965	
56 Volt Power Supply (S-IU-202 thru 212/501 thru 515)	65-966-0018H	III-5-510-7	1 October 1965	1 May 1966
5 Volt Measuring Voltage Supply	65-966-0019H	III-5-510-8	1 October 1965	
Switch Selector, Model II	66-966-0001	III-5-510-9	1 February 1966	
Component Handling Equipment, Description and Operating Instructions	66-966-0005	III-5-510-10	1 February 1966	
C-Band Radar Transponder Set SST-135C	66-966-0008	III-5-510-11	31 July 1964	1 March 1966
C-Band Radar Transponder Set SST-135C (S-IU-204, 205, and 501)	66-966-0020	III-5-510-18	15 June 1966	
Control Accelerometer	66-966-0012	III-5-510-12	1 March 1966	
Control-EDS Rate Gyro Package	66-966-0013	III-5-510-13	1 March 1966	
Control Signal Processor	66-966-0014	III-5-510-14	1 March 1966	
Flight Control Computer	66-966-0015	III-5-510-15	15 March 1966	
Flight Control Computer (S-IU-204 and 205)	66-966-0021	III-5-510-19	15 June 1966	
Flight Control Computer (S-IU-501)	66-966-0025	III-5-510-21	15 July 1966	
Instrumentation System Description (S-IU-202 thru 204/501 thru 503)	65-966-0021	III-5-509-1	1 June 1966	

Flight Control Computer
List of Related Documents

List of Related Documents (Cont)

Title	Number		Published Date	Changed Date
	IBM	MSFC		
Navigation, Guidance, and Control System	66-966-0003	III-5-509-2	1 February 1966	1 May 1966
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	1 March 1966	
System Description and Component Data	66-966-0006	III-5-509-4	1 March 1966	
Environmental Control System Description	66-966-0009	III-5-509-5	15 March 1966	
Apollo-Saturn Emergency Detection System Description (SA-203 and AS-204)	66-966-0010	III-5-509-6	1 June 1966	
Apollo-Saturn Emergency Detection System Description (AS-501)	66-966-0022	III-5-509-7	1 August 1966	
Command Decoder (S-IU-204 thru 212/501 thru 515)	66-966-0019	III-5-510-17	1 June 1966	
Battery (S-IU-201 thru 212/501 thru 515)	66-966-0024	III-5-510-20	1 August 1966	

CONTENTS

Section	Page	Section	Page
ILLUSTRATIONS	iii		
TABLES	iv		
I INTRODUCTION AND DESCRIPTION	1-1		
1-1 Introduction	1-1	2-4 Functional Operation During S-IC Burn Mode	2-3
1-2 Description	1-1	2-5 Functional Operation During S-II Burn Mode	2-7
1-3 Equipment Characteristics	1-2	2-6 Functional Operation During S-IVB Burn Mode	2-7
		2-7 Functional Operation During S-IVB Coast Mode	2-9
		2-8 Subassembly Functional Descriptions	2-10
II PRINCIPLES OF OPERATION	2-1	III PREVENTIVE MAINTENANCE AND REPAIR	3-1
2-1 Introduction	2-1	3-1 Preventive Maintenance	3-1
2-2 General Operation	2-1	3-2 Repair	3-1
2-3 System Tie-In	2-1		

ILLUSTRATIONS

Number	Title	Page	Number	Title	Page
1-1	Flight Control Computer (Three- Quarter View)	1-1	2-15	ϕ Y and ϕ P S-II Filter	2-22
1-2	Flight Control Computer Construction	1-3	2-16	ϕ Y and ϕ P S-IVB Filter	2-22
2-1	Physical and Functional Relationship of the Saturn V Control System Components (2 Sheets)	2-4	2-17	ϕ R S-IC Filter	2-23
2-2	S-IC and S-II Burn Modes, Functional Block Diagram	2-11	2-18	ϕ R S-II Filter	2-23
2-3	S-IVB Burn Mode, Functional Block Diagram	2-13	2-19	Typical Bending Mode Filter, Functional Block Diagram	2-24
2-4	S-IVB Coast Mode, Functional Block Diagram	2-15	2-20	50 Ma Servo Amplifier	2-25
2-5	DC Amplifier	2-17	2-21	Servo Scaling Board (Typical)	2-25
2-6	DC Amplifier Scaling Board (Typical)	2-17	2-22	50 Ma Servo Amplifier, Functional Block Diagram	2-26
2-7	DC Amplifier, Functional Block Diagram	2-18	2-23	Servo Comparator	2-26
2-8	$\Delta\phi$ Y and $\Delta\phi$ P S-IC Filter	2-19	2-24	Servo Comparator, Functional Block Diagram	2-27
2-9	$\Delta\phi$ Y and $\Delta\phi$ P S-II Filter	2-19	2-25	Spatial Amplifier	2-28
2-10	$\Delta\phi$ Y and $\Delta\phi$ P S-IVB Filter	2-20	2-26	Spatial Amplifier, Functional Block Diagram	2-29
2-11	$\Delta\phi$ Y, $\Delta\phi$ P, ϕ Y, and ϕ P S-IVB Filter	2-20	2-27	Spatial Comparator	2-30
2-12	$\Delta\phi$ R S-IC Filter	2-21	2-28	Spatial Comparator, Functional Block Diagram	2-31
2-13	$\Delta\phi$ R S-II Filter	2-21	2-29	Spatial Synchronizer	2-32
2-14	ϕ Y and ϕ P S-IC Filter	2-22	2-30	Spatial Synchronizer, Functional Block Diagram	2-33
			2-31	TM Amplifier	2-34
			2-32	TM Amplifier, Functional Block Diagram	2-34
			2-33	Limiters	2-35

ILLUSTRATIONS (Continued)

Number	Title	Page	Number	Title	Page
2-34	Limiter, Functional Block Diagram	2-35	2-39	Switching Circuit, Functional Block Diagram (2 Sheets)	2-39
2-35	Switching Control	2-36	2-40	Redundancy Check Matrix	2-43
2-36	Switching Circuit A1	2-36	2-41	Redundancy Check Matrix, Func- tional Block Diagram	2-44
2-37	Switching Circuit A2	2-37	2-42	Mother Board (Typical)	2-45
2-38	Switching Circuit A3	2-37			

TABLES

Number	Title	Page	Number	Title	Page
1-1	Electronic Subassemblies	1-2	2-1	Bending Mode Filters Switchpoints . .	2-24
1-2	Flight Control Computer, Leading Particulars	1-7	2-2	Relay Switching Cross Reference . .	2-38
1-3	Flight Control Computer Sub- assemblies, Leading Particulars .	1-9			

SECTION I

INTRODUCTION AND DESCRIPTION

1-1 INTRODUCTION

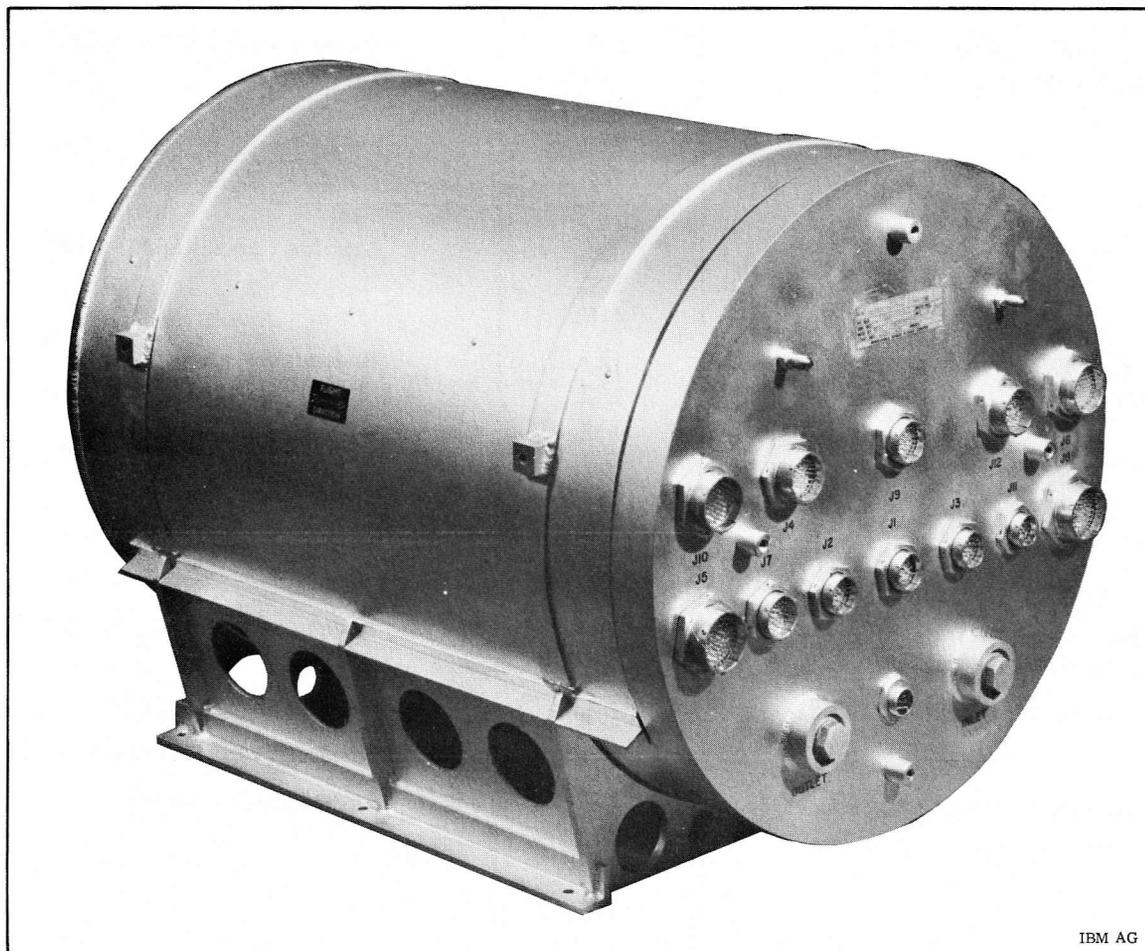
This publication contains the principles of operation and maintenance and repair instructions for the Flight Control Computer of the AS-501 launch vehicle of the Saturn V program.

NOTE

The symbol ϕ is used throughout this publication to represent attitude correction signals and is synonymous with the symbol ψ which is used in some related documents.

1-2 DESCRIPTION

The Flight Control Computer (see Figure 1-1) is located in the Instrument Unit (IU) of the Saturn V launch vehicle. The Flight Control Computer is used to control the flight path of the launch vehicle during liftoff and burn of the S-IC stage; during burn of the S-II stage; and during first burn, coast, and second burn of the S-IVB stage. During liftoff and flight, the Flight Control Computer receives signals from various sensing devices within the launch vehicle. These signals include attitude correction signals from the Launch Vehicle Digital Computer (LVDC), applied through the Launch Vehicle Data Adapter (LVDA); angular rate information from the Control-EDS Rate



IBM AG

Figure 1-1. Flight Control Computer (Three-Quarter View)

Gyro Package or the S-IVB Control Rate Gyro; and timing and programming information which is supplied by the LVDC through the LVDA and the IU Switch Selector. The LVDC, the LVDA, the Control-EDS Rate Gyro Package, and the Switch Selector are all part of the Instrument Unit.

Provisions have also been made which will allow the Apollo spacecraft to assume control of the vehicle. When this prerogative is exercised, the attitude correction signals are then supplied by the spacecraft's guidance system.

1-3 EQUIPMENT CHARACTERISTICS

Typical electronic subassemblies that make up the Flight Control Computer are illustrated in

Figure 1-2. Table 1-1 lists, in alpha-numeric-symbolic sequence, all of the subassemblies used in the Flight Control Computer. Each subassembly listing includes the official nomenclature, all reference designations, and common names of the subassemblies.

Table 1-2 lists the leading particulars of the Flight Control Computer. Table 1-3 lists, in alpha-numeric-symbolic sequence, the electrical and physical characteristics of the electronic subassemblies contained within the Flight Control Computer. The number of each type subassembly used in the Computer is also listed in this table.

Table 1-1. Electronic Subassemblies

Official Nomenclature	Ref. Desig.	Common Name
Control Switching Circuit (Figure 2-35)	J129	Switching control
DC Amplifier Scaling Board (Figure 2-6)	J35, J36, J50, J51, J92, J118	DC amplifier scaling board
Limiter (Figure 2-33)	J95, J96, J101	Limiter ($\Delta\phi Y$, $\Delta\phi P$, $\Delta\phi R$)
Matrix Switching Circuit (Figure 2-40)	1A1, 1A2, 1A3, 2A1, 2A2, 2A3, 3A3, 4A3	Redundancy check matrix
Mother Boards No. 1, 2, 3, 4, 5, 6, and 7 (Figure 2-42)	---	Mother boards 1, 2, 3, 4, 5, 6, and 7
Servo Comparator (Figure 2-23)	J86 J112	Yaw servo comparator Pitch servo comparator
Servo Scaling Board (Figure 2-21)	J80, J81, J106, J107	Servo scaling board

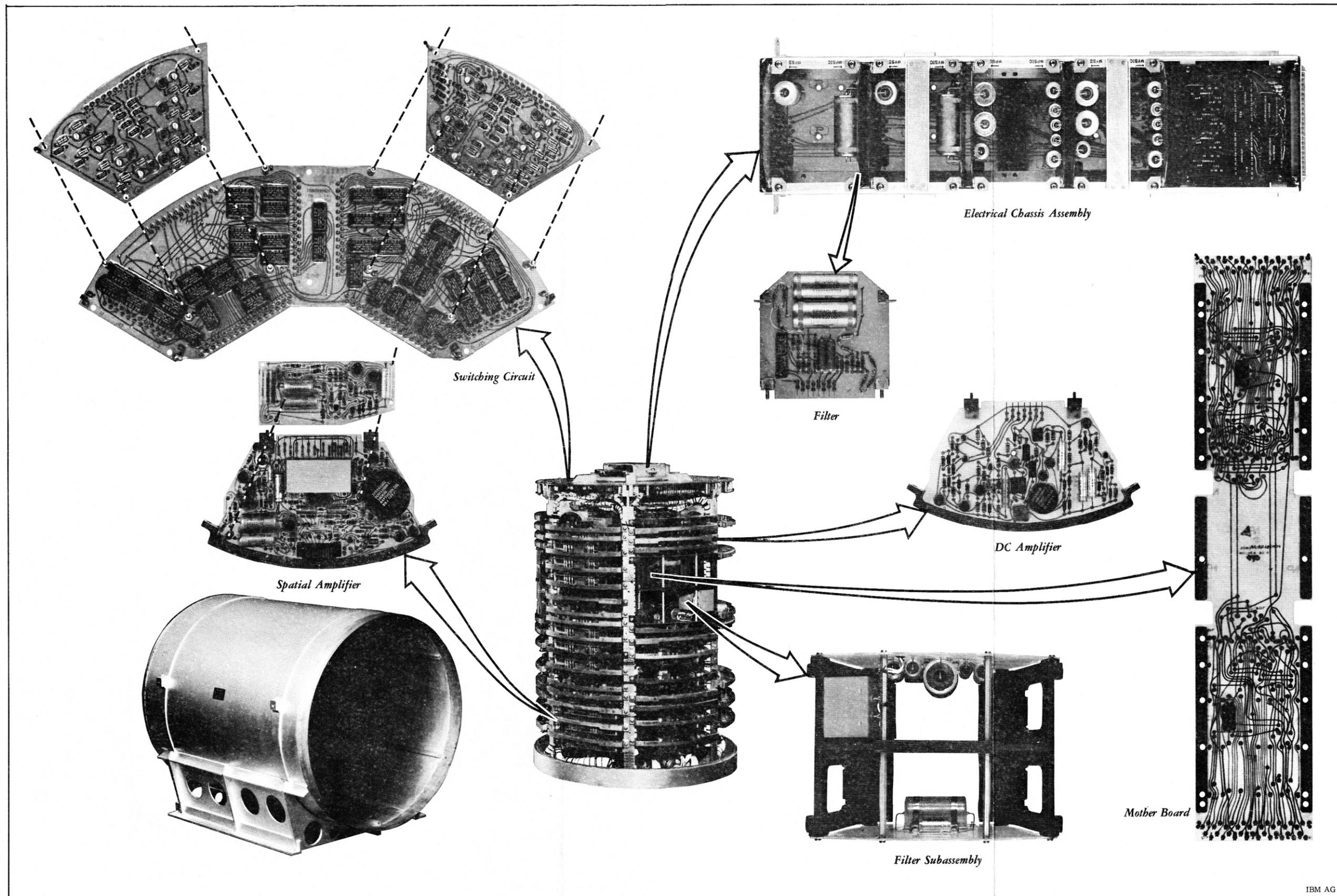


Figure 1-2. Flight Control Computer Construction

Table 1-1. Electronic Subassemblies (Cont)

Official Nomenclature	Ref. Desig.	Common Name
Spatial Amplifier (Figure 2-25)	J60, J61, J62	Pitch spatial amplifier
	J65, J66, J67	Roll-yaw I spatial amplifier
	J71, J72, J73	Roll-yaw II spatial amplifier
Spatial Comparator (Figure 2-27)	J63	Pitch spatial comparator
	J68	Roll-yaw I spatial comparator
	J74	Roll-yaw II spatial comparator
Spatial Synchronizer (Figure 2-29)	J70	Spatial synchronizer
Switching Circuit A (Figure 2-36)	A1	Switching circuit A1
Switching Circuit B (Figure 2-37)	A2	Switching circuit A2
Switching Circuit C (Figure 2-38)	A3	Switching circuit A3
Telemetry Amplifier (Figure 2-31)	J64	Pitch TM amplifier
	J69	Roll-yaw I TM amplifier
	J75	Roll-yaw II TM amplifier
45V DC Amplifier (Figure 2-5)	J34, J46, J47	DC amplifier ($\Delta\phi Y$)
	J48, J54, J55	DC amplifier ($\Delta\phi P$)
	J49, J52, J53	DC amplifier ($\Delta\phi R$)
	J31, J32, J33, J37	DC amplifier (ϕR)
	J89	DC amplifier (ϕY S-IC and S-II buffer) (ϕY S-IVB amplifier)
	J90	DC amplifier (ϕP S-IC and S-II buffer) (ϕY S-IVB amplifier)
	J91	DC amplifier (ϕR S-IC and S-II buffer) (ϕY S-IVB amplifier)
	J115	DC amplifier ($\Delta\phi R$ S-IC and S-II buffer) (ϕP S-IVB amplifier)

Table 1-1. Electronic Subassemblies (Cont)

Official Nomenclature	Ref. Desig.	Common Name
45V DC Amplifier (Cont)	J116	DC amplifier ($\Delta\phi P$ S-IC and S-II buffer) (ϕP S-IVB amplifier)
	J117	DC amplifier ($\Delta\phi Y$ S-IC and S-II buffer) (ϕP S-IVB buffer)
	J85	DC amplifier (pitch servo comparator)
	J111	DC amplifier (yaw servo comparator)
50 Ma Servo Amplifier (Figure 2-20)	J78	1 yaw 50 ma servo amplifier
	J79	2 yaw 50 ma servo amplifier
	J82	3 yaw 50 ma servo amplifier
	J84	4 yaw 50 ma servo amplifier
	J104	1 pitch 50 ma servo amplifier
	J105	2 pitch 50 ma servo amplifier
	J108	3 pitch 50 ma servo amplifier
$\phi Y P$ S-I Filter (Figure 2-8)	J125	$\Delta\phi Y$ S-IC filter
	J127	$\Delta\phi P$ S-IC filter
$\phi Y P$ S-II Filter (Figure 2-9)	J126	$\Delta\phi Y$ S-II filter
	J128	$\Delta\phi P$ S-II filter
$\phi Y P$ S-IV Filter (Figure 2-10)	J44, J45	$\Delta\phi Y$ S-IVB filter
	J56, J57	$\Delta\phi P$ S-IVB filter
	ϕ & $\phi Y P$ S-IV Filter (Figure 2-11)	J88
J114		$\Delta\phi P$ S-IVB filter
J87		ϕY S-IVB filter
J113		ϕP S-IVB filter
ϕR S-I Filter (Figure 2-12)	J97	$\Delta\phi R$ S-IC filter
ϕR S-II Filter (Figure 2-13)	J98	$\Delta\phi R$ S-II filter
$\phi Y P$ S-I Filter (Figure 2-14)	J121	ϕY S-IC filter
	J123	ϕP S-IC filter
$\phi Y P$ S-II Filter (Figure 2-15)	J122	ϕY S-II filter
	J124	ϕP S-II filter

Table 1-1. Electronic Subassemblies (Cont)

Official Nomenclature	Ref. Desig.	Common Name
ϕYP S-IV Filter (Figure 2-16)	J40, J41	ϕ Y S-IVB filter
	J29, J30	ϕ P S-IVB filter
ϕ R S-I Filter (Figure 2-17)	J99	ϕ R S-IC filter
ϕ R S-II Filter (Figure 2-18)	J100	ϕ R S-II filter

Table 1-2. Flight Control Computer, Leading Particulars

Characteristic	Specification
NOTE	
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.	
Overall Dimensions	
Height	21 inches
Width	20 inches
Depth	30 inches
Overall Weight	150 pounds maximum
Duty Cycle	100% continuous operation
Warmup Time	30 minutes
Input Power	22 to 32 Vdc, 320 watts maximum (28 volts nominal)
Operating Temperature Range	-55 to +100° C (-67 to +212° F)
Acceleration	20 G's peak
NOTE	
Reliability factor is for one Flight Control Computer and includes the probability that a failure may occur in a redundant circuit.	
Reliability Factor	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

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

Characteristic	Specification
<p>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>	
Pressurization	Gaseous nitrogen to 20 psia at +17 to +27° C (+63 to +81° F)
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)
Electrical Connectors	12
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
Insulation	Irradiated polyolefin (white)
Temperature Rating	+135° C (+275° F) maximum
Potential Rating	600 Vrms

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

Characteristic	Specification
NOTE	
Unless otherwise indicated herein, or in Table 1-2, the characteristics and specifications shown in Table 1-2 are applicable to the Flight Control Computer Subassemblies.	
Control Switching Circuit Function Construction Type Overall Dimensions Height Width Depth Overall Weight Warmup Time Input Power External Electrical Connections Electrical Characteristics Input Signal Output Signal	1 per computer Controls burn mode, test, and telemetry signals Printed circuit board 8.00 inches 5.75 inches 2.00 inches 4.8 ounces maximum None required None required Forked plug-in type connector 22 to 32 Vdc Combined burn mode, test, and telemetry signals
Limiter Function Construction Type Overall Dimensions Height Width Depth Overall Weight Warmup Time Input Power External Electrical Connections Electrical Characteristics Input Signal Linearity Limit Voltage Load Output Signal	3 per computer Limits signals to prevent overcompensation Printed circuit board 4.75 inches 8.00 inches 1.00 inch 4.3 ounces maximum None required None required Forked plug-in type connector 0 to 45 Vdc, positive or negative ±0.5% ±10% maximum change over full range of input 15 kilohms ±5% Linear function of input up to limit voltage

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

Characteristic	Specification
<p>Matrix Switching Circuit</p> <p>Function</p> <p>Construction Type</p> <p>Overall Dimensions</p> <p> Height</p> <p> Width</p> <p> Depth</p> <p>Overall Weight</p> <p>Warmup Time</p> <p>Input Power</p> <p>External Electrical Characteristics</p> <p> Input Signal</p> <p> Output Signal</p>	<p>1 per computer</p> <p>Redundant relay check</p> <p>Printed circuit board</p> <p>8.00 inches</p> <p>5.75 inches</p> <p>2.00 inches</p> <p>13.5 ounces maximum</p> <p>None required</p> <p>22 to 32 Vdc (28 volts nominal)</p> <p>+28 Vdc, 0 Vdc, or open</p> <p>+28 Vdc or 0 Vdc</p>
<p>Scaling Board</p> <p>Function</p> <p>Construction Type</p> <p>Overall Dimensions</p> <p> Height</p> <p> Width</p> <p> Depth</p> <p>Overall Weight</p> <p>Warmup Time</p> <p>Input Power</p> <p>External Electrical Connections</p>	<p>10 per computer</p> <p>Used with dc amplifiers and servo amplifiers to permit varied usage</p> <p>Printed circuit board</p> <p>4.75 inches</p> <p>8.00 inches</p> <p>1.00 inch</p> <p>5.0 ounces maximum</p> <p>None required</p> <p>None required</p> <p>Forked plug-in type connector</p>
<p>Servo Comparator</p> <p>Function</p> <p>Construction Type</p> <p>Overall Dimensions</p> <p> Height</p> <p> Width</p> <p> Depth</p> <p>Overall Weight</p> <p>Warmup Time</p> <p>Input Power</p>	<p>2 per computer</p> <p>Monitor and switch outputs from servo amplifiers</p> <p>Printed circuit board</p> <p>4.50 inches</p> <p>8.00 inches</p> <p>1.00 inch</p> <p>7.0 ounces maximum</p> <p>1 minute</p> <p>22 to 32 Vdc, 1.4 watts maximum (28 volts nominal)</p>

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

Characteristic	Specification
External Electrical Connections	Forked plug-in type connector
Electrical Characteristics	
Input Signal	3.2 microamperes differential input current (to produce trip point)
Input Impedance	Not less than 20,000 ohms
Load	One 600-ohm relay coil
Output Signal	(1) Yaw channel B or C or (2) Pitch channel B or C
Switch Point Tolerance	± 5% of the differential input signal
Telemetry Output	A discrete signal at the switch point
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
Overall Weight	13.6 ounces maximum
Warmup Time	1 minute
Input Power	22 to 32 Vdc, 7.28 watts maximum (28 volts nominal)
External Electrical Connections	Forked plug-in connector
Electrical Characteristics	
Input Signal	200 (±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 200 microampere input)
Load	Four 600-ohm relay coils in parallel and one 1200-ohm counter in parallel with the relays
Output	Up to trip: 0 Vdc Trip to full-ON: Pulsating level from 0 Vdc to primary power voltage
Switch Rate	Full-ON: Primary power voltage 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

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	7.0 ounces maximum
Warmup Time	1 minute
Input Power	22 to 32 Vdc, 7 watts maximum (28 volts nominal)
External Electrical Connections	Forked plug-in type connector
Electrical Characteristics	
Input Signal	The output signals supplied by two spatial amplifiers
Input Impedance	Not available
Load	600-ohm relay coil
Output Signal	(1) Yaw channel B or C, or (2) Pitch channel B or C
Switching Point	Circuit switches when outputs of two compared spatial amplifiers differ by amount equal to 50% ($\pm 10\%$) of duty cycle.
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	7.0 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	2.00 inches

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

Characteristic	Specification
Overall Weight	22.2 ounces maximum
Warmup Time	1 minute
Input Power	22 to 32 Vdc, 14 watts maximum (28 volts nominal)
External Electrical Connections	Forked plug-in type connector
Electrical Characteristics	
Input Signal	22 to 32 Vdc
Output	None
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	2.00 inches
Overall Weight	28.5 ounces maximum
Warmup Time	1 minute
Input Power	22 to 32 Vdc, 14 watts maximum (28 volts nominal)
External Electrical Connections	Forked plug-in type connector
Electrical Characteristics	
Input Signal	22 to 32 Vdc
Output	None
Switching Circuit A3	1 per computer
Function	Establishes switching combinations
Construction Type	Printed circuit board
Overall Dimensions	
Height	15.00 inches
Width	6.00 inches
Depth	2.00 inches
Overall Weight	27.8 ounces maximum
Warmup Time	1 minute
Input Power	22 to 32 Vdc, 14 watts maximum (28 volts nominal)
External Electrical Connections	Forked plug-in type connector

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

Characteristic	Specification
Electrical Characteristics	
Input Signal	22 to 32 Vdc
Output	None
TM 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	7.0 ounces maximum
Warmup Time	1 minute
Input Power	22 to 32 Vdc, 3.5 watts maximum (28 volts nominal)
External Electrical Connections	Forked plug-in type connector
Electrical Characteristics	
Input Signal	0 to 210 milliamperes
Gain	9.52 volts per ampere (voltage gain)
Input Impedance	0.5 ohm 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
45V DC Amplifier	21 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	7.0 ounces maximum
Warmup Time	1 minute
Input Power	22 to 32 Vdc, 1.96 watts maximum (28 volts nominal)
External Electrical Connections	Forked plug-in type connector
Electrical Characteristics	
Input Signal	A differential value between 0 and 45 volts, positive or negative, at 10 Hz maximum

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

Characteristic	Specification
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 0 and 45 volts, positive or negative, as developed across a 5,000-ohm load
Non-linearity	Not more than 2%
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
50 Ma Servo Amplifier	8 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	16.0 ounces maximum
Warmup Time	10 minutes
Input Power	22 to 32 Vdc, 2.8 watts maximum (28 volts nominal)
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
$\Delta\phi$ Y or $\Delta\phi$ P	7.4 Vdc, positive and negative
Current Gain	1.1 milliamperes per turn ($\pm 2\%$)
Input Current	Not more than 0 ± 30 microamperes into 450 turns per stack for full output signal switch
Output Impedance	Greater than 40,000 ohms from 0 to 20 Hz
Output Current	A full output swing between 0 and 50 milliamperes, positive and negative, as developed across a 100-ohm valve load
Non-linearity	Not more than 2%

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

Characteristic	Specification
Dynamic Response	The 3 db down frequency is 55 (± 5) Hz
Null Offset	With no input signal applied, null offset is less than 0 ± 20 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
$\Delta\phi Y$ and $\Delta\phi P$ S-IC Filter	2 per computer
Function	Bending mode filters used to filter out angular errors resulting from frame vibrations; also provide time sequenced attenuation control
Overall Dimensions	
Height	4.75 inches
Width	8.00 inches
Depth	1.00 inch
Attenuation Switchpoints	Switchpoint 1 (at 100 seconds of S-IC burn) Switchpoint 2 (at 120 seconds of S-IC burn)
Filter Type	Low pass
$\Delta\phi Y$ and $\Delta\phi P$ S-II Filter	2 per computer
Function	Bending mode filters used to filter out angular errors resulting from frame vibrations. Also provide time sequenced attenuation control
Overall Dimensions	
Height	4.75 inches
Width	8.00 inches
Depth	1.00 inch
Attenuation Switchpoints	Switchpoint 3 (at 60 seconds of S-II burn) Switchpoint 4 (at 190 seconds of S-II burn)
Filter Type	Low pass
$\Delta\phi Y$ and $\Delta\phi P$ S-IVB Filter	2 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	6.50 inches
Filter Type	Low pass
$\Delta\phi Y$, $\Delta\phi P$, ϕY , and ϕP S-IVB Filter	2 per computer

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

Characteristic	Specification
<p>Function</p> <p>Overall Dimensions</p> <p> Height</p> <p> Width</p> <p> Depth</p> <p>Attenuation Switchpoint</p> <p>Filter Type</p> <p>$\Delta\phi R$ S-IC Filter</p>	<p>Bending mode filters used to filter out angular errors resulting from frame vibrations. Also provide time sequenced attenuation control</p> <p>4.75 inches</p> <p>8.00 inches</p> <p>6.50 inches</p> <p>Switchpoint 5 (at 270 seconds of S-IVB second burn)</p> <p>Low pass</p> <p>1 per computer</p>
<p>Function</p> <p>Overall Dimensions</p> <p> Height</p> <p> Width</p> <p> Depth</p> <p>Attenuation Switchpoints</p> <p>Filter Type</p>	<p>Bending mode filter used to filter out angular errors resulting from frame vibrations. Also provides time sequenced attenuation control</p> <p>4.75 inches</p> <p>8.00 inches</p> <p>1.00 inch</p> <p>Switchpoint 1 (at 100 seconds of S-IC burn) Switchpoint 2 (at 120 seconds of S-IC burn)</p> <p>Low pass</p>
<p>$\Delta\phi R$ S-II Filter</p> <p>Function</p> <p>Overall Dimensions</p> <p> Height</p> <p> Width</p> <p> Depth</p> <p>Filter Type</p>	<p>1 per computer</p> <p>Bending mode filter used to filter out angular errors resulting from frame vibrations</p> <p>4.75 inches</p> <p>8.00 inches</p> <p>1.00 inch</p> <p>Low pass</p>
<p>ϕY and ϕP S-IC Filter</p> <p>Function</p> <p>Overall Dimensions</p> <p> Height</p> <p> Width</p> <p> Depth</p>	<p>2 per computer</p> <p>Bending mode filters used to filter out angular errors resulting from frame vibrations. Also provide time sequenced attenuation control</p> <p>4.75 inches</p> <p>8.00 inches</p> <p>1.00 inch</p>

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

Characteristic	Specification
Attenuation Switchpoints	Switchpoint 1 (at 100 seconds of S-IC burn) Switchpoint 2 (at 120 seconds of S-IC burn)
Filter Type	Low pass
ϕ Y and ϕ P S-II Filter	2 per computer
Function	Bending mode filters used to filter out angular errors resulting from frame vibrations. Also provide time sequenced attenuation control
Overall Dimensions	
Height	4.75 inches
Width	8.00 inches
Depth	1.00 inch
Attenuation Switchpoints	Switchpoint 3 (at 60 seconds of S-II burn) Switchpoint 4 (at 190 seconds of S-II burn)
Filter Type	Low pass
ϕ Y and ϕ P S-IVB Filter	2 per computer
Function	Bending mode filters used to filter out angular errors resulting from frame vibrations. Also provide time sequenced attenuation control
Overall Dimensions	
Height	4.75 inches
Width	8.00 inches
Depth	6.50 inches
Attenuation Switchpoint	Switchpoint 5 (at 270 seconds of S-IVB second burn)
Filter Type	Low pass
ϕ R S-IC Filter	1 per computer
Function	Bending mode filter 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 S-II Filter	1 per computer
Function	Bending mode filter used to filter out angular errors resulting from frame vibrations

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
Filter Type	Low pass

SECTION II

PRINCIPLES OF OPERATION

2-1 INTRODUCTION

This section contains the principles of operation of the Flight Control Computer. The explanation presented describes the manner in which the Flight Control Computer is connected to the launch vehicle systems, and how the overall system functions to provide control commands to the launch vehicle.

2-2 GENERAL OPERATION

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 launch vehicle. Paragraphs 2-4 through 2-7 describes the functional relationship between the Flight Control Computer circuits during the various flight modes. Paragraph 2-8 contains a functional description of each of the Flight Control Computer circuits.

2-3 SYSTEM TIE-IN

Figure 2-1 illustrates the physical and functional relationship of the Flight Control Computer to the various position sensing and position controlling devices of the launch vehicle. Sheet 1 of Figure 2-1 illustrates the physical relationship between the engines, actuators, and spatial thrusters of the Saturn V launch vehicle. Sheet 2 of Figure 2-1 is a function block diagram of the Saturn V control system.

Operation of the Flight Control Computer is divided into four basic functions - control of the vehicle during S-IC burn mode, control of the vehicle during S-II burn mode, control of the vehicle during S-IVB burn mode, and control of the vehicle during S-IVB coast mode.

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

- Attitude error signals ($\Delta\phi$) from the LVDC/LVDA guidance system

- Attitude angular rate error signals ($\dot{\phi}$) from the Control-EDS Rate Gyro Package

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 computes the attitude corrections (steering commands) required for the vehicle to reach its pre-determined destination in space.

The attitude correction commands are issued through the LVDA to the Flight Control Computer. The Flight Control Computer then 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 Gyro Package (of if selected, the S-IVB Control Rate Gyro) senses 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 with the attitude error signals from the LVDC/LVDA to produce engine commands which conform to the prescribed vehicle limits.

Provisions have been made which will allow the Apollo spacecraft to assume control of the vehicle. When this prerogative is exercised, the attitude error signals are then supplied by the spacecraft guidance system.

The three attitude control 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-IC burn mode is activated prior to lift-off by the S-IC burn mode substitute signal from the GSE. The S-IC burn mode substitute signal is applied to the Flight Control Computer switching 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. The relay code symbols on Figure 2-1 indicate which relays are energized.

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

The resultant drive signals from the four yaw servo amplifiers are coupled through relay contacts which are closed during the S-IC burn mode only and applied to the four gimballed engines in the S-IC stage. Drive signals from the four pitch servo amplifiers are coupled through relay contacts which are closed during the S-IC burn mode only and also applied to the actuators of the four gimballed engines in the S-IC stage. The yaw servo amplifiers control the four gimballed engines in the S-IC stage in the yaw axis, and the pitch servo amplifiers control the four gimballed engines in the S-IC stage in the pitch axis. The yaw servo amplifiers and the pitch servo amplifiers, operating simultaneously, result in control of the vehicle about the roll axis.

Angular rate error signals ($\dot{\phi} Y$, $\dot{\phi} R$ and $\dot{\phi} P$) are developed by the Control-EDS Rate Gyro Package in the Instrument Unit. These signals are applied to the Flight Control Computer via the Control Signal Processor. In the Flight Control Computer, the angular rate error signals are coupled through jumpers and closed relay contacts to the yaw and pitch servo amplifiers. The $\dot{\phi} Y$ angular rate error signal is applied to the four yaw servo amplifiers; the $\dot{\phi} P$ angular rate error signal is applied to the four pitch servo amplifiers; and the $\dot{\phi} R$ angular rate error signals are applied to both the 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 error signals from the Control-EDS Rate Gyro Package in the IU, or from the Control Rate Gyros in the S-IVB stage. However, this selection must be made prior to launch.

During the S-IC burn mode, switchpoint 1 and 2 signals from the Switch Selector are applied to signal line filters (not shown on Figure 2-1). The filters then adjust the amplitude of the error signals in relation to the phase of the flight.

Upon completion of the S-IC burn mode, the S-IC stage is jettisoned and an S-II burn mode signal from the Switch Selector is applied to the Flight Control Computer switching circuit. The indicated relays are energized. This connects the Flight Control Computer for S-II burn mode operation.

Operation of the Flight Control Computer during S-II burn mode is functionally identical to the operation during the S-IC burn mode. Attitude and angular rate error signals are received and processed in the same manner as for the S-IC burn mode. However, drive signals from the four yaw servo amplifiers and the four pitch servo amplifiers are now coupled to the four gimballed engines in the S-II stage through relay contacts which are closed during S-II burn mode only. The yaw servo amplifiers control the four gimballed engines of the S-II stage in the yaw axis, and the pitch servo amplifiers control the four gimballed engines in the pitch axis. The yaw servo amplifiers and the pitch servo amplifiers, operating simultaneously, result in control of the vehicle about the roll axis.

During the S-II burn mode, switchpoint 3 and 4 signals 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-II burn mode, the S-II stage is jettisoned and an S-IVB burn mode signal from the Switch Selector is applied to the Flight Control Computer switching circuit. The indicated switching circuit relays are energized, setting the Flight Control Computer for flight control during the S-IVB burn mode.

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 $\Delta\phi Y$ attitude error signal from the LVDC/LVDA or the spacecraft is coupled through relay contacts which are closed during S-IVB burn only and applied to three of the yaw servo amplifiers. The 4 yaw servo amplifier is not used during the S-IVB burn mode. The identical drive signal outputs from the three servo amplifiers are applied through relay contacts which are closed during S-IVB burn only to the yaw servo comparator. The drive signal output from the yaw servo comparator positions the gimbaled engine in the S-IVB stage to control the yaw attitude.

Similarly, the $\Delta\phi P$ attitude error signal is coupled through relay contacts which are closed during S-IVB burn and is also applied to only three of the pitch servo amplifiers, the 4 pitch servo amplifier being unused. Again, identical drive signal outputs from the three servo amplifiers are applied through relay contacts which are closed during S-IVB burn to the pitch servo comparator. The drive signal output from the pitch servo comparator positions the gimbaled engine in the pitch attitude.

Roll attitude errors can not be corrected with a single gimbaled engine. Therefore, the roll attitude corrections during S-IVB burn mode are made by the S-IVB stage spatial thrusters. The $\Delta\phi R$ attitude error signal is coupled through closed relay contacts and applied to the triple redundant roll-yaw I and roll-yaw II spatial amplifiers.

The $\dot{\phi}Y$ and $\dot{\phi}P$ angular rate error signals from the rate gyros are coupled through relay contacts which are closed during S-IVB burn only, and like $\Delta\phi$, are applied to the three yaw servo amplifiers and three pitch servo amplifiers. These angular rate signals algebraically add with the attitude error signals ($\Delta\phi$) from the LVDC/LVDA to produce the drive signals for the gimbaled engine in the S-IVB stage. The $\dot{\phi}R$ angular rate error signal is coupled through closed relay contacts and applied to the triple redundant roll-yaw I and roll-yaw II spatial amplifiers. The roll angular rate error signal algebraically adds with the $\Delta\phi R$ attitude error signal from the LVDC/LVDA to produce the roll control signals. These control signals are coupled through the spatial synchronizer to the spatial thrusters in the S-IVB stage.

Upon completion of the S-IVB burn mode, the S-IVB burn mode signal is removed and the S-IVB stage enters the coast mode. Removal of the S-IVB burn mode signal causes the switching circuit relay contacts to transfer the yaw and pitch attitude error signals from the yaw and pitch servo amplifiers to

the spatial amplifiers. The yaw attitude error signal ($\Delta\phi Y$) is simultaneously applied to the roll-yaw I and the roll-yaw II spatial amplifiers. The pitch attitude error signal ($\Delta\phi P$) is applied to the pitch spatial amplifiers. The $\Delta\phi Y$ and $\Delta\phi P$ signals are applied to the spatial amplifiers through relay contacts which are only closed during the S-IVB coast mode.

The roll attitude error signal ($\Delta\phi R$) and the roll angular rate error signal ($\dot{\phi}R$) are applied to the roll-yaw I and the roll-yaw II spatial amplifiers in exactly the same manner employed during the S-IVB burn mode. The spatial amplifiers algebraically add the angular rate error signals ($\dot{\phi}Y$, $\dot{\phi}P$, $\dot{\phi}R$) to the attitude error signals ($\Delta\phi Y$, $\Delta\phi P$, $\Delta\phi R$) to produce vehicle control signals. The control signals are applied through the spatial synchronizer to the control relay packages in the S-IVB stage. The control relay packages actuate their associated spatial thrusters to correct for errors in vehicle position during the coast mode.

At a designated time, the S-IVB stage gimbaled engine is restarted to reach orbital escape velocity. When this action occurs the S-IVB burn mode signal is re-applied to the Flight Control Computer switching circuits. This is called the S-IVB second burn mode. The operation of the S-IVB second burn is identical to the operation of the previously described S-IVB burn mode. However, during the S-IVB second burn mode, the switchpoint 5 signal from the Switch Selector is applied to signal line filters. The filters then adjust the amplitude of the error signal in relation to the phase of the flight.

2-4 FUNCTIONAL OPERATION DURING S-IC BURN MODE

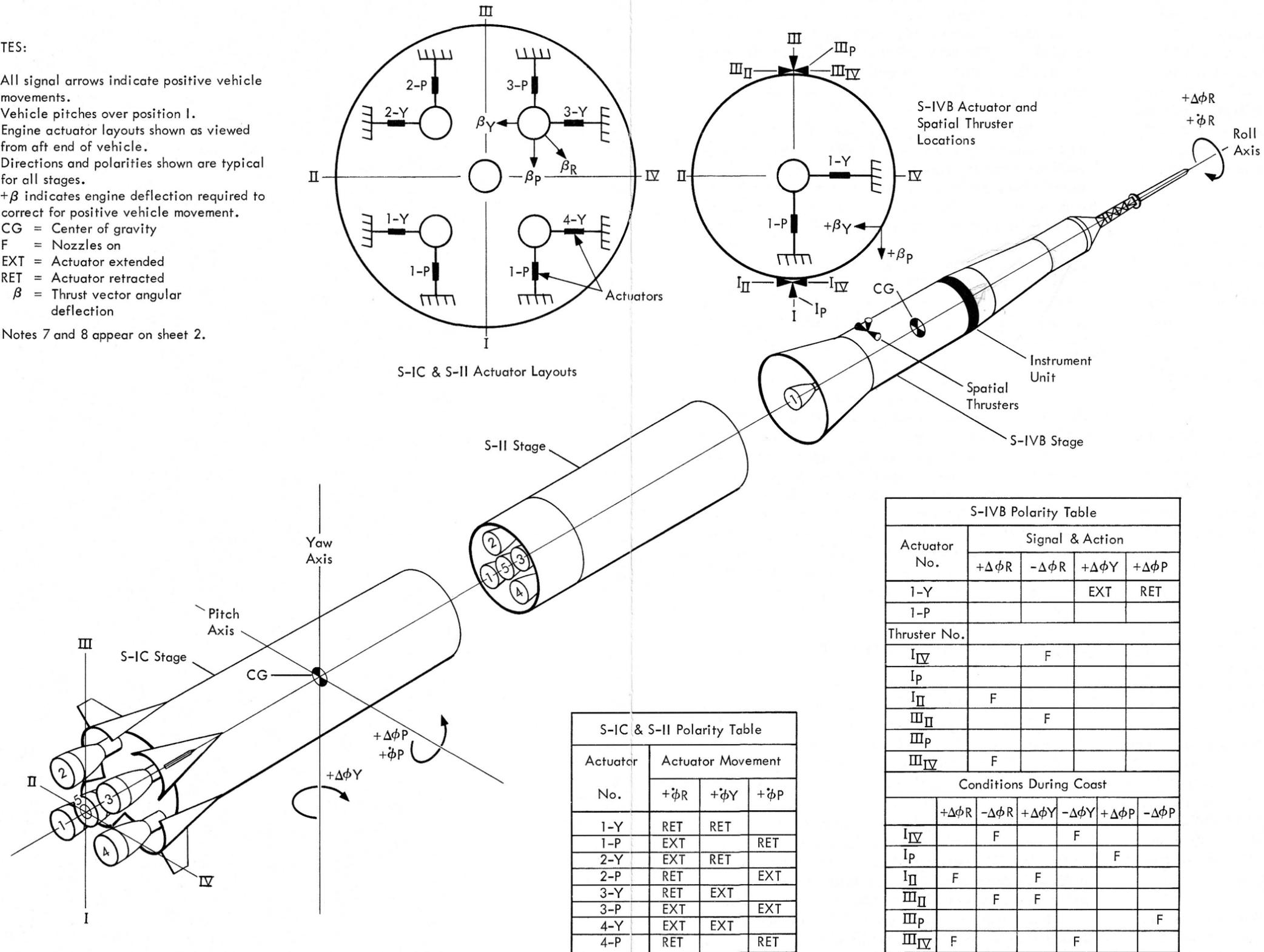
Figure 2-2 illustrates the functional operation of the Flight Control Computer during the S-IC and S-II burn modes. The S-IC burn mode will be discussed in Paragraph 2-4 while the S-II burn mode will be discussed in Paragraph 2-5.

The S-IC burn mode is initiated prior to lift-off by the S-IC burn mode substitute signal from the GSE. The S-IC burn mode substitute signal is applied to the switching circuits of the Flight Control Computer (see Figure 2-39 for a functional diagram of the switching circuits). Operation of the appropriate relays within the switching circuits conditions the Flight Control Computer for attitude control during the S-IC burn mode. The relays which are activated during the S-IC burn mode are identified by the relay code symbols called out on Figures 2-2 and 2-39.

NOTES:

1. All signal arrows indicate positive vehicle movements.
2. Vehicle pitches over position I.
3. Engine actuator layouts shown as viewed from aft end of vehicle.
4. Directions and polarities shown are typical for all stages.
5. $+\beta$ indicates engine deflection required to correct for positive vehicle movement.
6. CG = Center of gravity
F = Nozzles on
EXT = Actuator extended
RET = Actuator retracted
 β = Thrust vector angular deflection

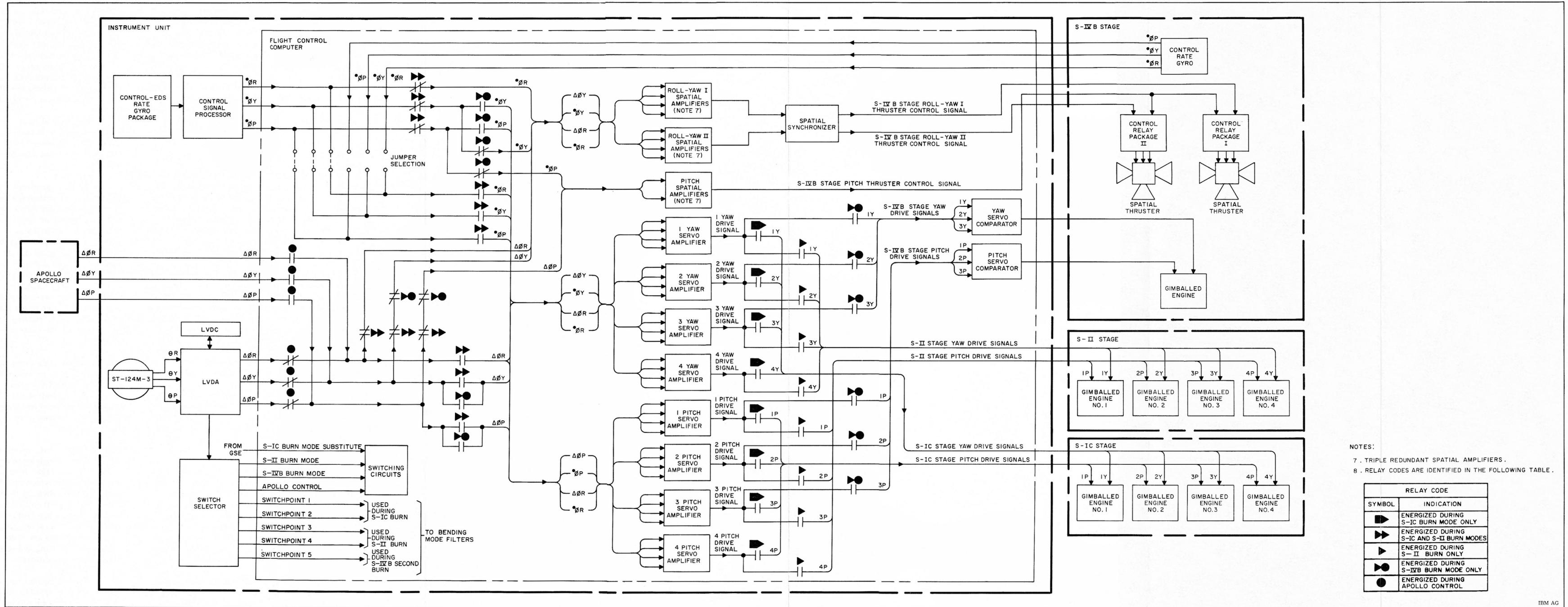
7. } Notes 7 and 8 appear on sheet 2.
8. }



Actuator No.	Actuator Movement		
	$+\phi_R$	$+\phi_Y$	$+\phi_P$
1-Y	RET	RET	
1-P	EXT		RET
2-Y	EXT	RET	
2-P	RET		EXT
3-Y	RET	EXT	
3-P	EXT		EXT
4-Y	EXT	EXT	
4-P	RET		RET

Actuator No.	Signal & Action					
	$+\Delta\phi_R$	$-\Delta\phi_R$	$+\Delta\phi_Y$	$+\Delta\phi_P$		
1-Y			EXT	RET		
1-P						
Thruster No.						
I _{IV}		F				
I _p						
I _{II}	F					
III _{II}		F				
III _p						
III _{IV}	F					
Conditions During Coast						
	$+\Delta\phi_R$	$-\Delta\phi_R$	$+\Delta\phi_Y$	$-\Delta\phi_Y$	$+\Delta\phi_P$	$-\Delta\phi_P$
I _{IV}		F		F		
I _p					F	
I _{II}	F		F			
III _{II}		F	F			
III _p						F
III _{IV}	F			F		

Figure 2-1. Physical and Functional Relationship of the Saturn V Control System Components
(Sheet 1 of 2)



- NOTES:
 7. TRIPLE REDUNDANT SPATIAL AMPLIFIERS.
 8. RELAY CODES ARE IDENTIFIED IN THE FOLLOWING TABLE.

RELAY CODE	
SYMBOL	INDICATION
◼	ENERGIZED DURING S-IC BURN MODE ONLY
◻	ENERGIZED DURING S-IC AND S-II BURN MODES
◻	ENERGIZED DURING S-II BURN ONLY
◻	ENERGIZED DURING S-IVB BURN MODE ONLY
●	ENERGIZED DURING APOLLO CONTROL

Figure 2-1. Physical and Functional Relationship of the Saturn V Control System Components
(Sheet 2 of 2)

Redundant relay switching is used throughout the Flight Control Computer to achieve a high degree of reliability. To simplify the functional block diagram (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 via the LVDA. In the Flight Control Computer, the attitude error signals are coupled through the redundant closed relay contacts of switching circuit A2 and applied to dc amplifiers J34, J49, and J48, respectively.

The amplified $\Delta\phi Y$ attitude error signal from dc amplifier J34 is routed through switching circuit A3 to $\Delta\phi Y$ S-IC filter J125. J125 is a bending mode filter which filters out angular yaw errors resulting from physical deflections of the vehicle structure. The deflections occur when the launch vehicle frame vibrates at its resonant frequency during high thrust loadings.

The $\Delta\phi Y$ attitude error signal from filter J125 is coupled through switching circuit A3 to dc amplifier J117. Dc amplifier J117 is used as a buffer to provide impedance matching and sufficient power to drive the 50 ma servo amplifiers. The amplified $\Delta\phi Y$ attitude error signal from dc amplifier J117 is coupled through switching circuit A1 and applied to the four yaw 50 ma servo amplifiers J78, J79, J82, and J84. Each yaw 50 ma servo amplifier develops drive signals which are proportional to the combined input levels and polarities. The drive signals from the four yaw 50 ma servo amplifiers are coupled through switching circuit A3 and applied to respective yaw servo actuators in the S-IC stage. The four yaw servo actuators mechanically position the four gimballed engines of the S-IC stage as required to correct for errors in the yaw axis.

In a manner identical to the processing of the $\Delta\phi Y$ error signal, the amplified $\Delta\phi P$ attitude error signal from dc amplifier J48 is coupled through switching circuit A3 to the $\Delta\phi P$ S-IC bending mode filter J127. The $\Delta\phi P$ attitude error signal from J127 is coupled through switching circuit A3 to dc amplifier J116. Dc amplifier J116 is also used as a buffer to provide impedance matching and sufficient power to drive the 50 ma servo amplifiers.

The $\Delta\phi P$ attitude error signal from dc amplifier J116 is coupled through switching circuit A1 and applied to the four pitch 50 ma servo amplifiers J104,

J105, J108, and J110. Each pitch 50 ma servo amplifier develops a drive signal which is proportional to its combined input levels and polarities. The drive signals from the four pitch 50 ma servo amplifiers are coupled through switching circuit A3 and applied to respective pitch servo actuators in the S-IC stage. The four pitch servo actuators mechanically position the four gimballed engines of the S-IC stage as required to correct for errors in the pitch axis.

The $\Delta\phi R$ attitude error signal is processed in a manner similar to both the $\Delta\phi Y$ and $\Delta\phi P$ error signals. The difference is that the $\Delta\phi R$ signal is applied to both the yaw and pitch 50 ma servo amplifiers as follows:

From dc amplifier J49, the $\Delta\phi R$ signal is routed through switching circuit A3 and $\Delta\phi R$ S-IC bending mode filter J97 to dc amplifier J115. Operating as a buffer, dc amplifier J115 provides impedance matching and sufficient power to drive the 50 ma servo amplifiers. From dc amplifier J115, $\Delta\phi R$ is coupled through switching circuit A1 and applied to both the yaw and pitch 50 ma servo amplifiers. The $\Delta\phi R$ signal is algebraically added to the $\Delta\phi Y$ signal by the yaw 50 ma servo amplifiers, and to the $\Delta\phi P$ signal by the pitch 50 ma servo amplifier. The resultant drive signals are applied to the yaw and pitch servo actuators in the S-IC stage, causing the four gimballed engines to be positioned to correct for errors about the roll axis.

Angular rate error signals ($\dot{\phi}Y$, $\dot{\phi}R$, and $\dot{\phi}P$) for the S-IC burn mode are supplied by either the Control-EDS Rate Gyro Package located in the Instrument Unit or by the Control Rate Gyro located in the S-IVB stage. Selection of the component to be used during flight is determined prior to liftoff 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 J121, J99, and J123, respectively. These are bending mode filters which filter out angular rate errors resulting from physical flexings of the vehicle structure. The $\dot{\phi}Y$, $\dot{\phi}R$ and $\dot{\phi}P$ angular rate error signals from the filters are coupled through switching circuit A3 to dc amplifiers J89, J91, and J90, respectively. The dc amplifiers are used as buffers to provide impedance matching and sufficient power to drive the 50 ma servo amplifiers.

The $\dot{\phi}Y$ angular rate error signal from dc amplifier J89 is coupled through switching circuit A1 to yaw 50 ma servo amplifiers J78, J79, J82, and J84;

the $\dot{\phi}P$ angular rate error signal from dc amplifier J90 is coupled through switching circuit A1 to pitch 50 ma servo amplifiers J104, J105, J108, and J110; and the $\dot{\phi}R$ angular rate error signal is coupled through switching circuit A1 to both the yaw and pitch 50 ma servo amplifiers. The $\dot{\phi}Y$, $\dot{\phi}P$, and $\dot{\phi}R$ angular rate error signals are algebraically added to the $\Delta\phi Y$, $\Delta\phi P$, and $\Delta\phi R$ attitude error signals by the 50 ma servo amplifiers. The purpose of the angular rate error signals is to control the rate of response of the vehicle and to eliminate overshoot at the end of the correction.

Adjustment of the signal amplitudes is accomplished by scaling resistor networks included with the dc amplifiers, filters, and 50 ma servo amplifiers. These scaling resistor networks are set at a predetermined value to control the overall gain of the Flight Control Computer. In addition $\Delta\phi Y$ S-IC filter (J125), $\Delta\phi P$ S-IC filter (J127), $\dot{\phi}Y$ S-IC filter (J121), and $\dot{\phi}P$ S-IC filter (J123) contain relay-switched resistor networks which change attenuation in discrete steps at predetermined time intervals. These relays are energized during S-IC burn mode by switchpoint 1 and 2 signals from the LVDC/LVDA, through the Switch Selector.

2-5 FUNCTIONAL OPERATION DURING S-II BURN MODE

Upon completion of the S-IC burn mode, the S-IC stage is jettisoned and an S-II burn mode signal from the Switch Selector is applied to the Flight Control Computer switching circuits. This energizes the switching circuit relays that set the Flight Control Computer for guidance control during the S-II burn mode. Figure 2-2 illustrates the functional operation of the Flight Control Computer during the S-II burn mode. Relays are identified by the appropriate relay code on Figure 2-2.

The operation of the Flight Control Computer during the S-II burn mode is functionally identical to the operation during the S-IC burn mode. Refer to Paragraph 2-4 for a functional description of the S-IC burn mode. The S-IC bending mode filters are switched out of the circuit and the S-II bending mode filters are switched in by switching circuit A3. The yaw and pitch 50 ma servo amplifiers are connected through switching circuit A3 to the yaw and pitch servo actuators in the S-II stage.

Drive signals from the four yaw 50 ma servo amplifiers are applied to respective yaw servo actuators in the S-II stage. The four yaw servo actuators

mechanically position the four gimballed engines of the S-II stage as required to correct for errors in the yaw axis. Drive signals from the four pitch 50 ma servo amplifiers are applied to respective pitch servo actuators in the S-II stage. The four pitch servo actuators mechanically position the four gimballed engines of the S-II stage as required to correct for errors in the pitch axis. Combined yaw and pitch drive signals cause the servo actuators to position the four gimballed engines of the S-II stage as required to correct for errors in the roll axis.

Switchpoint 3 and 4 signals from the Switch Selector are applied to $\Delta\phi Y$ S-II filter (J126), $\Delta\phi P$ S-II filter (J128), $\dot{\phi}Y$ S-II filter (J122), and $\dot{\phi}P$ S-II filter (J124) during the S-II burn mode. These signals energize relays within the filters that switch in resistor networks that change attenuation of their respective signals at predetermined time intervals.

2-6 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-II burn mode, the S-II stage is jettisoned and the S-IVB stage is activated. The LVDC, through the LVDA and the Switch Selector, applies the S-IVB burn mode signal to the Flight Control Computer. The functional operation of the Flight Control Computer is the same during first burn and second burn of the S-IVB stage.

Normally, guidance of the launch vehicle is controlled by the LVDC/LVDA during the S-IVB burn mode; however, control may be assumed 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 A2 which energizes the indicated switchover relays in switching circuit A2. This allows the $\Delta\phi Y$, $\Delta\phi R$, and $\Delta\phi P$ attitude error signals from the spacecraft guidance system to be applied to the same dc amplifiers as the $\Delta\phi Y$, $\Delta\phi R$, and $\Delta\phi P$ signals from the LVDC/LVDA. Also, the Apollo control signal energizes other relays in switching circuit A2 which open the paths for the $\Delta\phi Y$, $\Delta\phi R$, and $\Delta\phi P$ signals from the LVDC/LVDA, so that only one set of attitude error signals are used.

During the following discussion, it will be assumed that the $\Delta\phi Y$, $\Delta\phi R$, and $\Delta\phi P$ attitude error signals are applied to the Flight Control Computer from the LVDC through the LVDA. In the Flight Control Computer, the attitude error signals are routed

through the redundant closed relay contacts of switching circuit A2 and applied to triple redundant dc amplifiers. The $\Delta\phi Y$ attitude error signal is applied to dc amplifiers J34, J46, and J47; the $\Delta\phi R$ attitude error signal is applied to dc amplifiers J49, J52, and J53; and the $\Delta\phi P$ attitude error signal is applied to dc amplifiers J48, J54, and J55.

The amplified $\Delta\phi Y$ attitude error signals from dc amplifiers J34, J46, and J47 are coupled through switching circuits A1 and A3 to $\Delta\phi Y$ S-IVB filters J44, J45, and J88, respectively. These are bending mode filters which filter out attitude yaw errors resulting from the physical flexing of the launch vehicle. These flexings 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 amplifiers. The yaw 50 ma servo amplifiers (J78, J79, and J82) then develop servo drive signals in proportion to the input error signals. However, only the servo drive signal from 50 ma servo amplifier J79 is coupled through the normally closed contacts of the comparator relay of yaw servo comparator J86 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 50 ma servo amplifiers J78 and J79 are continuously compared by yaw servo comparator J86. If a malfunction occurs in either of the two separate and identical channels, yaw servo comparator J86 automatically detects the malfunction and causes the normally closed relay contacts in the J79 output path to open and the open relay contacts in the J82 output path to close. The servo drive signal from 50 ma servo amplifier J82 is then applied to the yaw servo actuator in the S-IVB stage.

Similarly, the amplified $\Delta\phi P$ attitude error signals from dc amplifiers J54, J55, and J48 are coupled through switching circuits A1 and A3 to $\Delta\phi P$ S-IVB filters J114, J57, and J56, respectively. These filters are also bending mode filters and are used to filter out attitude pitch errors resulting from the physical flexing of the launch vehicle.

From the $\Delta\phi P$ filters, the $\Delta\phi P$ attitude error signals are applied to their respective pitch 50 ma servo amplifiers. The pitch 50 ma servo amplifiers (J104, J105, and J108) also develop servo drive signals in proportion to the input error signals. However, only the servo drive signal from 50 ma servo

amplifier J105 is coupled through the normally closed contacts of the comparator relay of pitch servo comparator J112 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 comparator J112 automatically opens the relay contacts in the J105 output and closes the relay contacts in the J108 output. The servo drive signal from 50 ma servo amplifier J108 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 corrected with a single engine.

When under LVDC/LVDA control, the three $\Delta\phi R$ attitude error signals are coupled directly through switching circuits A1 and A3 to roll-yaw I spatial amplifiers J65, J66, and J67. When the $\Delta\phi R$ attitude error signal originates from the Apollo spacecraft, relays in switching circuit A1 are opened and the direct path of the $\Delta\phi R$ signals from the dc amplifiers to the roll-yaw I spatial amplifiers is open. At the same time, other relays in switching circuit A1 are closed, and the three $\Delta\phi R$ signals are routed through limiters J95, J96, and J101 to the roll-yaw I spatial amplifiers. The limiters limit the amplitude of the $\Delta\phi R$ error signal within a preset range such that feedback from the ϕR channel to the spatial amplifiers holds the rate of displacement change within safe limits.

Roll-yaw I spatial amplifiers J65, J66, and J67 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 is coupled through the normally closed contacts of the comparator relay of roll-yaw I spatial comparator J68 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 thrusters 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 to the spatial synchronizer is then applied from roll-yaw I spatial amplifier J67.

The angular rate error signals ($\dot{\phi}Y$, $\dot{\phi}P$, and $\dot{\phi}R$) are supplied by the Control-EDS Rate Gyro Package through the Control Signal Processor. The $\dot{\phi}Y$ and $\dot{\phi}P$ angular rate error signals are each coupled through switching circuit A2 and applied to triple redundant bending mode filters. The $\dot{\phi}Y$ signal is applied to $\dot{\phi}Y$ S-IVB filters J40, J41, and J87; and the $\dot{\phi}P$ signal is applied to filters J29, J30, and J113. By means of the switchpoint 5 signal from the LVDC/LVDA (applied through the Switch Selector), the filters adjust the amplitude of the $\dot{\phi}Y$ and $\dot{\phi}P$ angular rate error signals in relation to the phase of the flight. The switchpoint 5 signal is applied during the S-IVB second burn mode only.

The $\dot{\phi}Y$ angular rate error signals from filters J41, J40, and J87 are applied to yaw 50 ma servo amplifiers J78, J79, and J82, respectively. The $\dot{\phi}Y$ error signal is then algebraically added to the $\Delta\phi Y$ attitude error signal by the 50 ma servo amplifiers to produce a yaw servo drive signal which conforms to the prescribed vehicle limits.

Similarly, the $\dot{\phi}P$ angular rate error signals from filters J29, J30, and J113 are applied to pitch 50 ma servo amplifiers J104, J105, and J108, respectively. The $\dot{\phi}P$ error signal is then algebraically added to the $\Delta\phi P$ attitude error signal by the 50 ma servo amplifiers to produce a pitch servo drive signal which conforms to the prescribed vehicle limits.

The $\dot{\phi}R$ error signal is coupled through switching circuit A2 and applied directly to dc amplifiers J31, J32, and J33. The amplified $\dot{\phi}R$ error signals from the dc amplifiers are applied to triple redundant roll-yaw I spatial amplifiers J65 through J67. The $\dot{\phi}R$ angular rate error signals are algebraically added to the $\Delta\phi R$ attitude error signals to produce spatial thruster control signals which conform to the prescribed vehicle limits. These signals are coupled through the spatial comparator and spatial synchronizer to the control relay packages in the S-IVB stage. The control relay packages then activate their associated spatial thruster to correct for errors in the roll axis.

2-7 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 and

the launch vehicle enters the S-IVB coast mode. Positioning of the launch vehicle is then controlled by the spatial thrusters located on the S-IVB stage. 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 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. The relay code indicates when the relays will be energized. However, during the S-IVB coast mode, these relays will be in their deenergized positions to conserve power.

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 in the same manner as for the S-IVB burn mode. The amplified $\Delta\phi Y$ attitude error signals from redundant dc amplifiers J34, J46, and J47 are coupled through switching circuits A3 and A1 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 closed contacts of the comparator relay to spatial synchronizer J70. During the S-IVB coast mode, the spatial synchronizer is not in use and the control signal is routed through closed relay contacts in the spatial synchronizer to control relay package I in the S-IVB stage. The control signal from J72 is coupled through the closed contacts of the comparator relay and spatial synchronizer J70 to control relay package II in the S-IVB stage. The control relay packages then activate their associated spatial thrusters to correct for errors in the yaw axis.

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, 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 from roll-yaw I spatial amplifier J67.

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, 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 from roll-yaw II spatial amplifier J73.

The amplified $\Delta\phi R$ attitude error signals from redundant dc amplifiers J49, 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 any existing $\Delta\phi Y$ error signals by the roll-yaw I and II spatial amplifiers to produce control signals for the control relay packages.

The amplified $\Delta\phi P$ attitude error signals from redundant dc amplifiers J48, J55, and J54 are coupled through switching circuits A3 and A1 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.

When the vehicle is under spacecraft control, an Apollo control signal is applied to the Flight Control Computer switching circuits. The indicated relays are energized and the $\Delta\phi R$, $\Delta\phi Y$, and $\Delta\phi P$ attitude error signals are now supplied by the Apollo spacecraft. Limiters J95, J96, and J101 are also switched into the $\Delta\phi R$ attitude error signal lines. The limiters limit the attitude error signal amplitudes within a preset range such that feedback from the ϕR , ϕY , and ϕP channels to the spatial amplifiers holds the rate of displacement change within safe limits.

The angular rate error signals ($\dot{\phi Y}$, $\dot{\phi P}$, and $\dot{\phi R}$) are supplied by the Control-EDS Rate Gyro Package through the Control Signal Processor. The $\dot{\phi Y}$

error signal is coupled through switching circuits A2 to dc amplifiers J89, J90, and J91. The amplified $\dot{\phi Y}$ error signals from the dc amplifiers are simultaneously applied to their respective triple redundant roll-yaw I and roll-yaw II spatial amplifiers where they are algebraically added to existing yaw and roll error signals to produce spatial thruster control signals which conform to the prescribed vehicle limits.

The $\dot{\phi R}$ error signal is coupled through switching circuit A2 to redundant dc amplifiers J31, J32, and J33. The amplified $\dot{\phi R}$ 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 also algebraically added to yaw and roll error signals to produce spatial thruster control signals which conform to prescribed vehicle limits.

The $\dot{\phi P}$ error signal is coupled, through switching circuits A2 and A3, to dc amplifiers J80, J88, and J89. The amplified $\dot{\phi P}$ 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 to produce spatial thruster control signals which conform to prescribed vehicle limits.

2-8 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 and DC Amplifier Scaling Board
- Bending Mode Filters
- 50 Ma Servo Amplifier and Servo Scaling Board
- Servo Comparator
- Spatial Amplifier
- Spatial Comparator
- Spatial Synchronizer
- TM Amplifier
- Limiter
- Switching Circuit
- Redundancy Check Matrix
- Mother Board

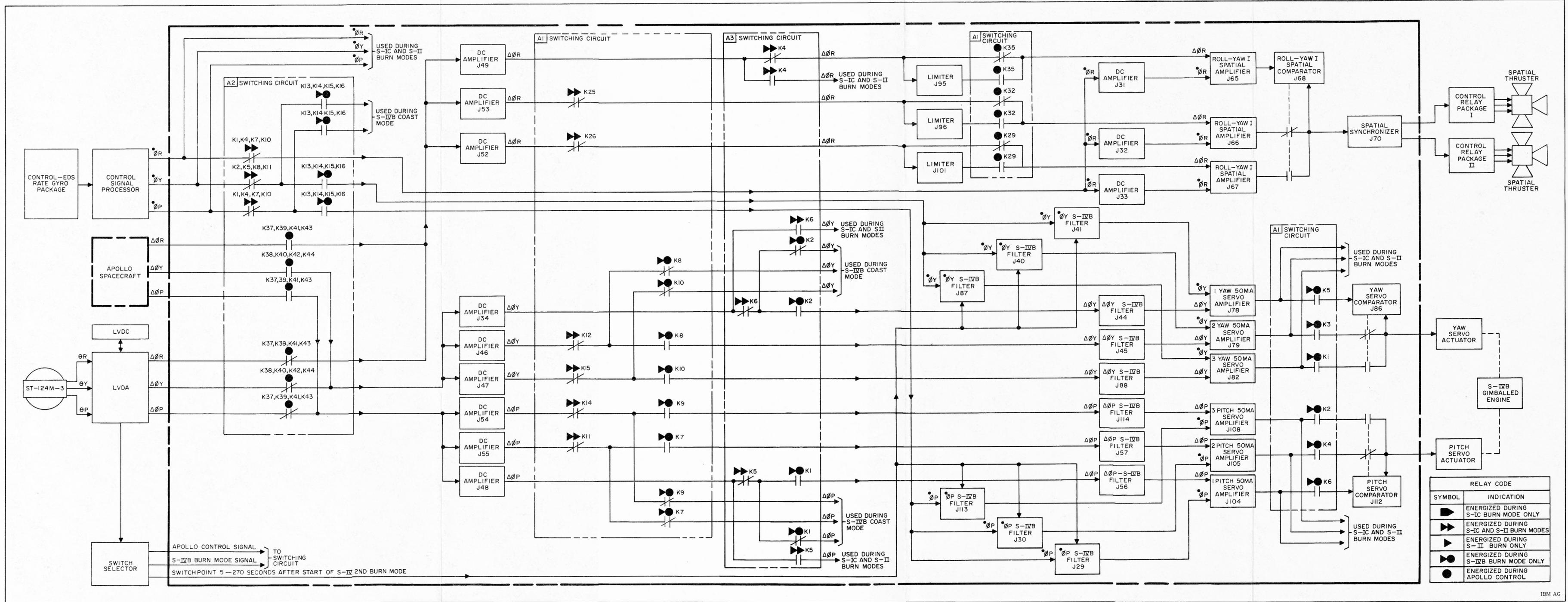


Figure 2-3. S-IVB Burn Mode, Functional Block Diagram

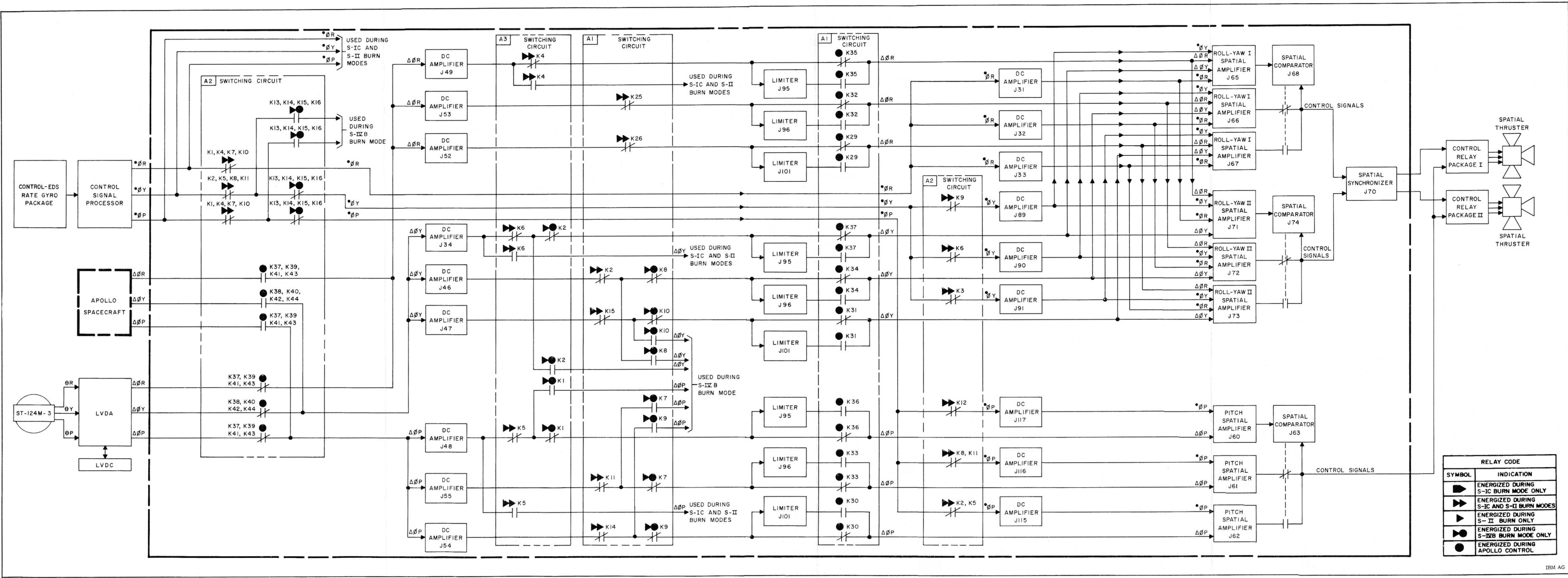


Figure 2-4. S-IVB Coast Mode, Functional Block Diagram

DC Amplifier and DC Amplifier Scaling Board

Dc amplifiers 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 line is amplified by the dc amplifiers. The dc amplifier scaling boards contain feedback and resistive attenuating networks which are used to set the gain of the dc amplifiers to a predetermined level.

A photograph of the dc amplifier is shown in Figure 2-5 and a photograph of the dc amplifier scaling board is shown in Figure 2-6. A functional block diagram of the dc amplifier is shown in Figure 2-7.

Error signals are coupled through an attenuating network on the dc amplifier scaling board before they are applied to the dc amplifiers. This reduces the signal to a predetermined level. A sample of the dc amplifier output is also applied to the scaling board to provide a degenerative feedback.

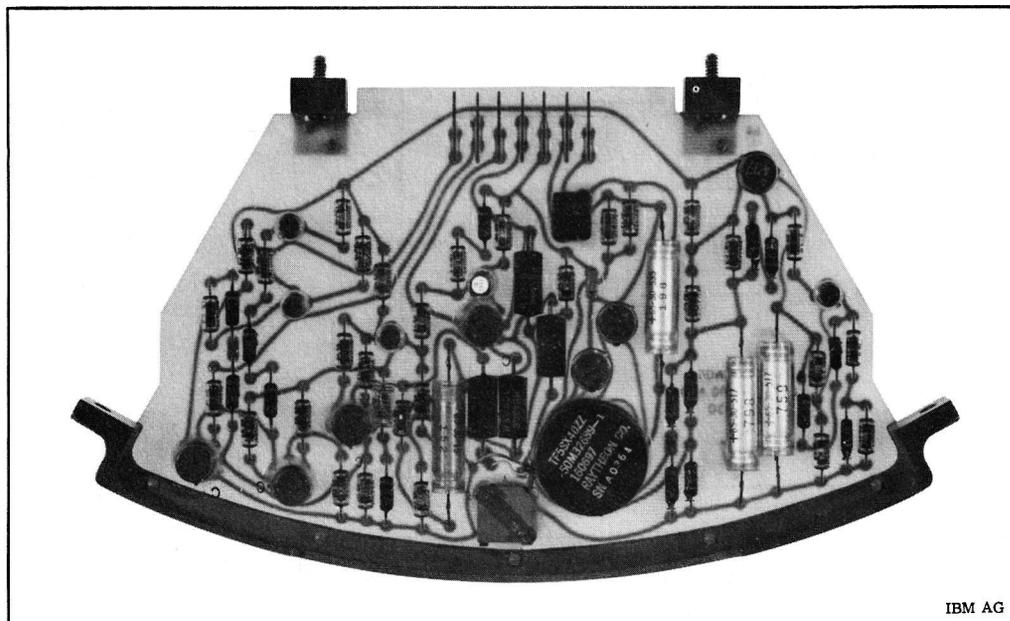


Figure 2-5. DC Amplifier

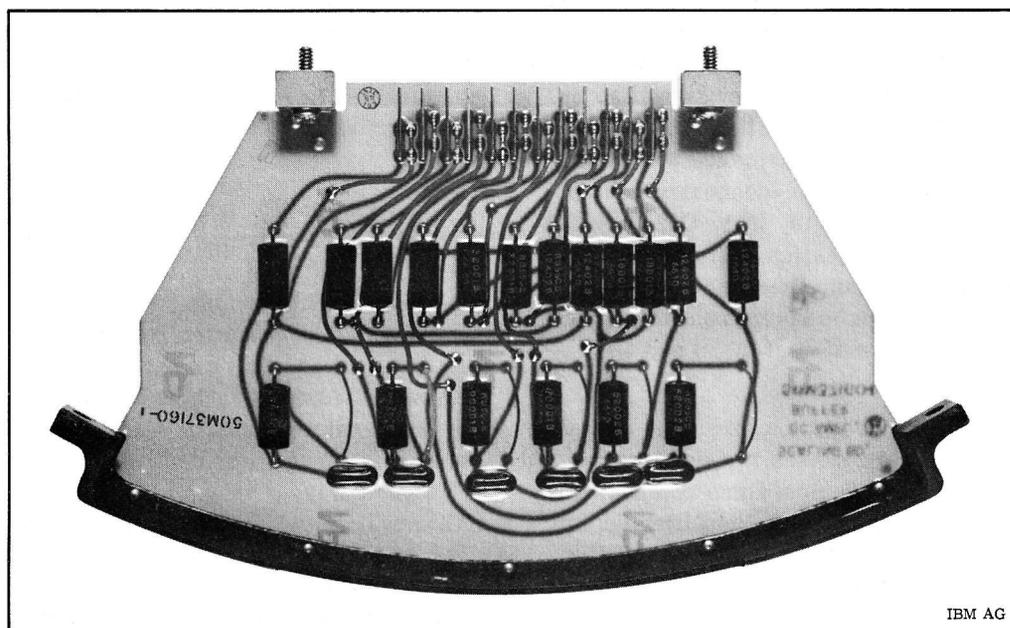


Figure 2-6. DC Amplifier Scaling Board (Typical)

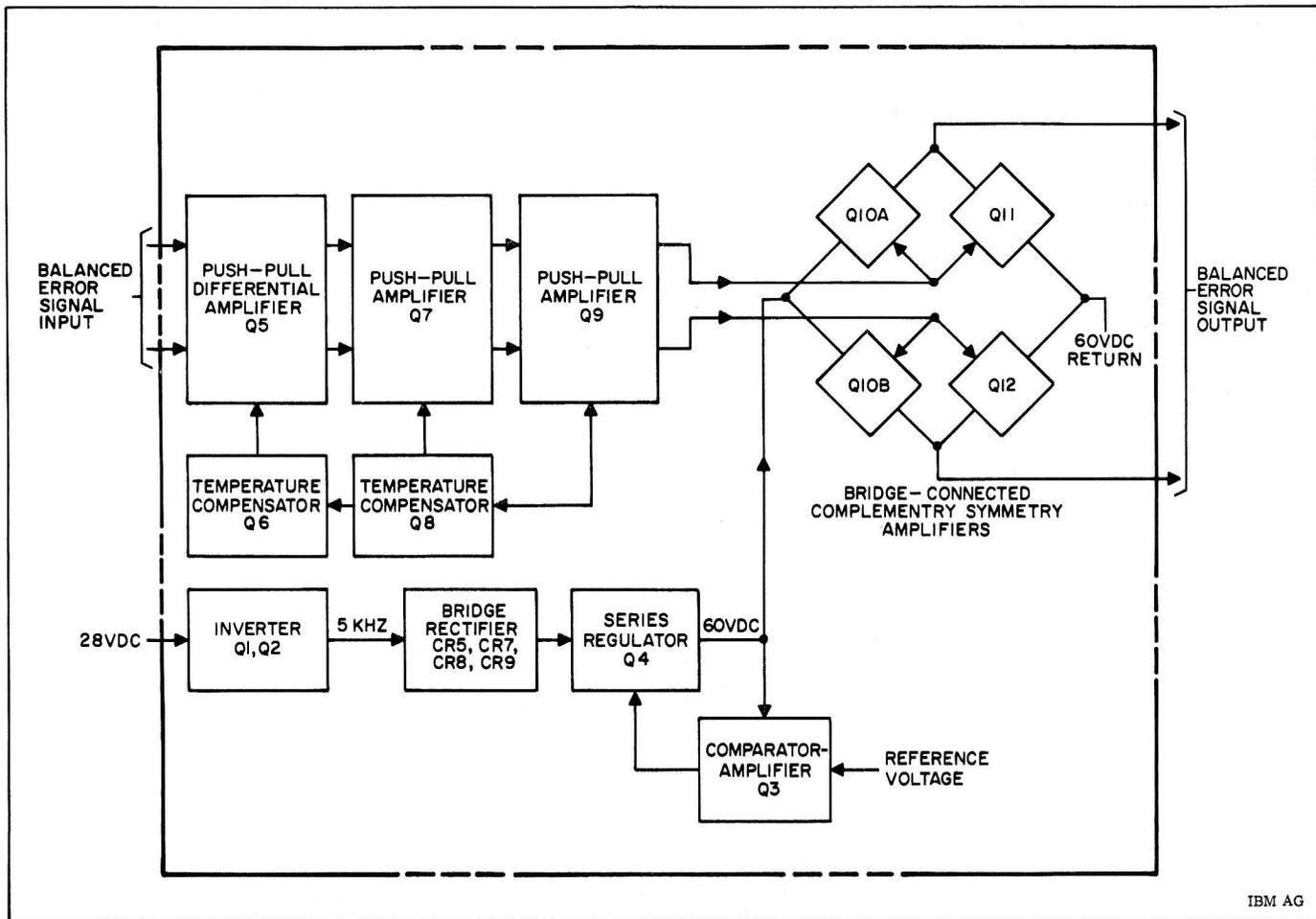


Figure 2-7. DC Amplifier, Functional Block Diagram

The balanced error signal input to the dc amplifier 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 push-pull differential amplifier 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. Bridge-connected complementary symmetry 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 effect the overall gain of the dc amplifier are sensed by temperature compensators Q6 and Q8. Temperature compensators Q6 and Q8 control the gain of the dc amplifier by varying the bias level on push-pull differential amplifier Q5 and push-pull amplifier 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 contains 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 rectifiers 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 series regulator 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 series regulator Q4 and returns the output voltage to the proper level.

Bending Mode Filters

The Flight Control Computer contains bending mode filters located in the signal lines throughout the Flight Control Computer. Photographs of the various bending mode filters are shown in Figures 2-8 through 2-18. A functional block diagram of a typical bending mode filter is shown in Figure 2-19.

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 flexings of the vehicle structure. These flexings occur when the launch vehicle frame vibrates at its resonant frequency during high thrust loadings.

The low pass filter output is coupled through an attenuation network which controls the level of the output signal in relation to the phase of the flight. Minimum attenuation is provided by parallel attenuating resistors A, B, and C. As the flight progresses, relay A is energized by a switchpoint signal from the Switch Selector. This action removes attenuating resistors B from attenuating network, increasing the amount of attenuation. As the flight progresses still further, relay B is energized by a second switchpoint signal from the Switch Selector. Each time, the amount of attenuation is increased.

Each of the bending mode filters is specially designed to filter out attitude errors resulting from

the physical flexing of the vehicle structure for its associated signal line during a specific mode. The amount of attenuation and time sequenced attenuation change is determined by the individual mission. Table 2-1 lists the bending mode filters and the switchpoint signals applied. As indicated in the table, some of the filters do not require time sequenced attenuation control; however, each of these filters have a fixed attenuation network.

50 Ma Servo Amplifier and Servo Scaling Boards

The Flight Control Computer contains four yaw 50 ma servo amplifiers and four pitch 50 ma servo amplifiers. The purpose of the 50 ma servo amplifiers is to develop drive signals for the pitch and yaw servo actuators in proportion to the yaw and pitch error signals. The servo scaling boards contain resistor attenuating networks which are used to attenuate the input signals to the 50 ma servo amplifiers and to provide impedance matching and isolation. A photograph of the 50 ma servo amplifier is shown in Figure 2-20 and a photograph of the servo scaling board is shown in Figure 2-21. A functional block diagram of a 50 ma servo amplifier is shown in Figure 2-22.

Attitude ($\Delta\phi$) and angular rate ($\dot{\phi}$) error signals are applied to coil windings in magnetic amplifiers CR12 through CR15. The error signals are algebraically combined by magnetic amplifiers CR12 through CR15 to produce a proportional balanced output signal. The balanced output signal of magnetic amplifiers CR12 through CR15 is amplified by push-pull amplifier Q6 and applied to differential amplifier Q10, Q11. Differential amplifier Q10, Q11 amplifies the balanced

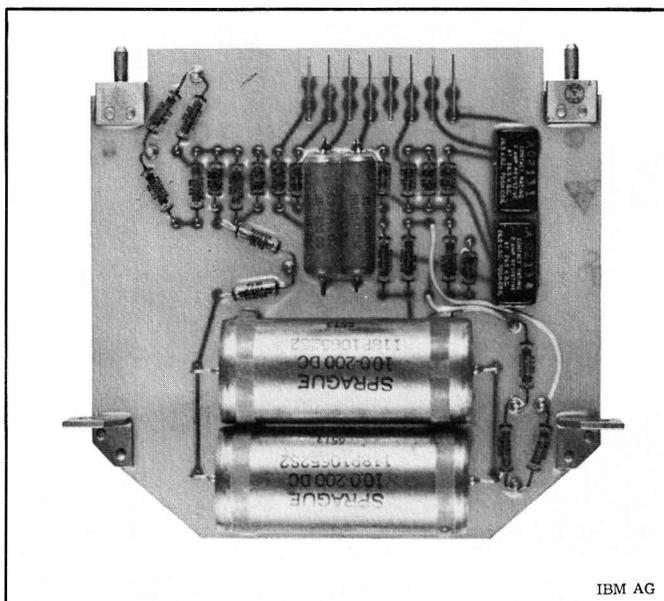


Figure 2-8. $\Delta\phi$ Y and $\Delta\phi$ P S-IC Filter

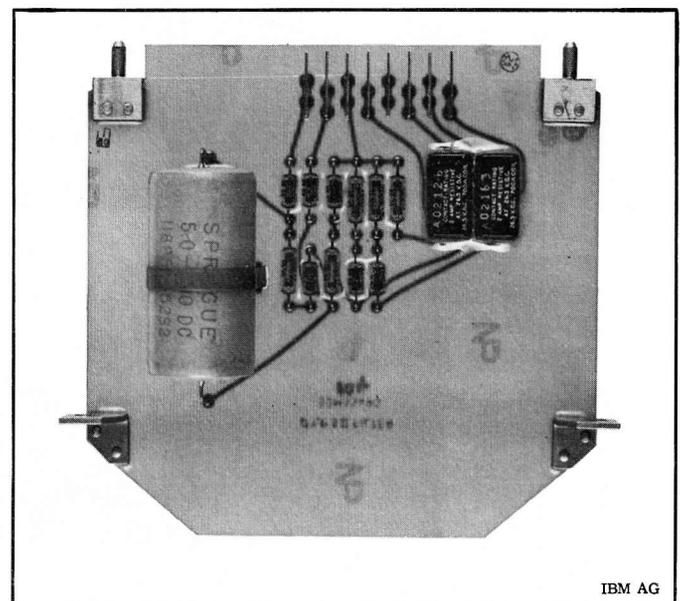
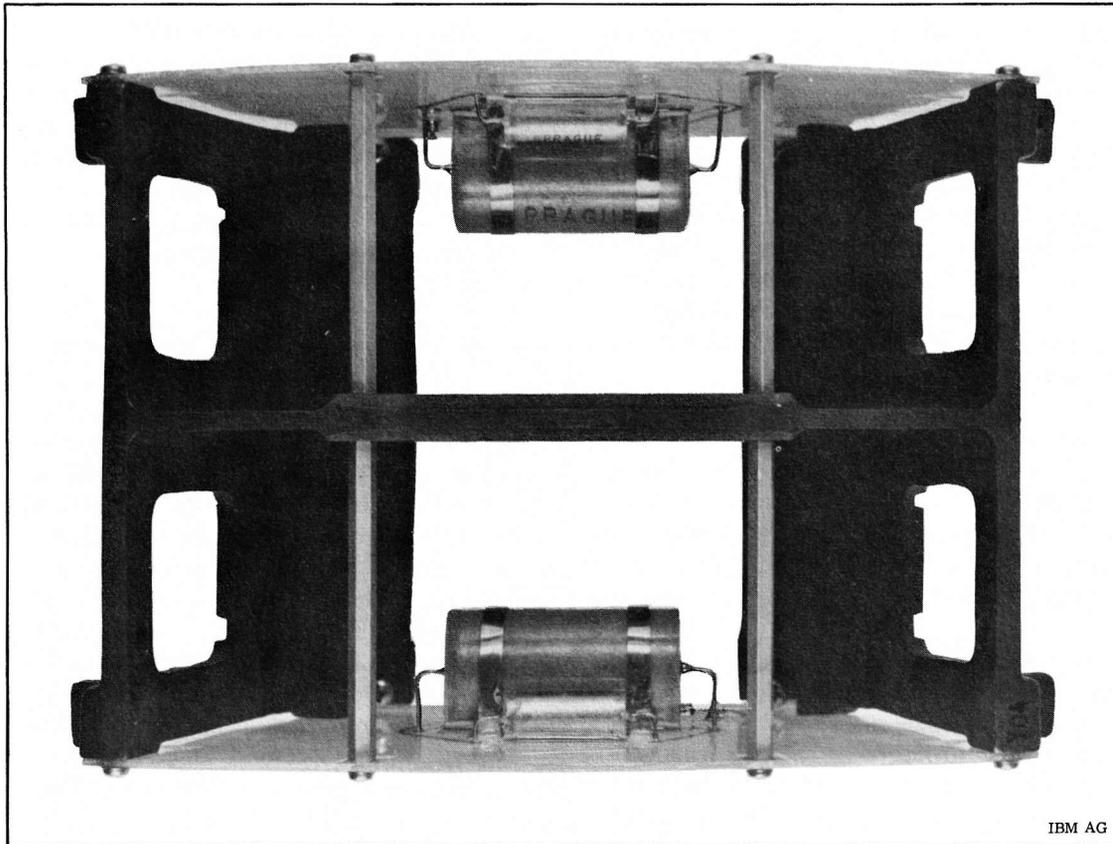
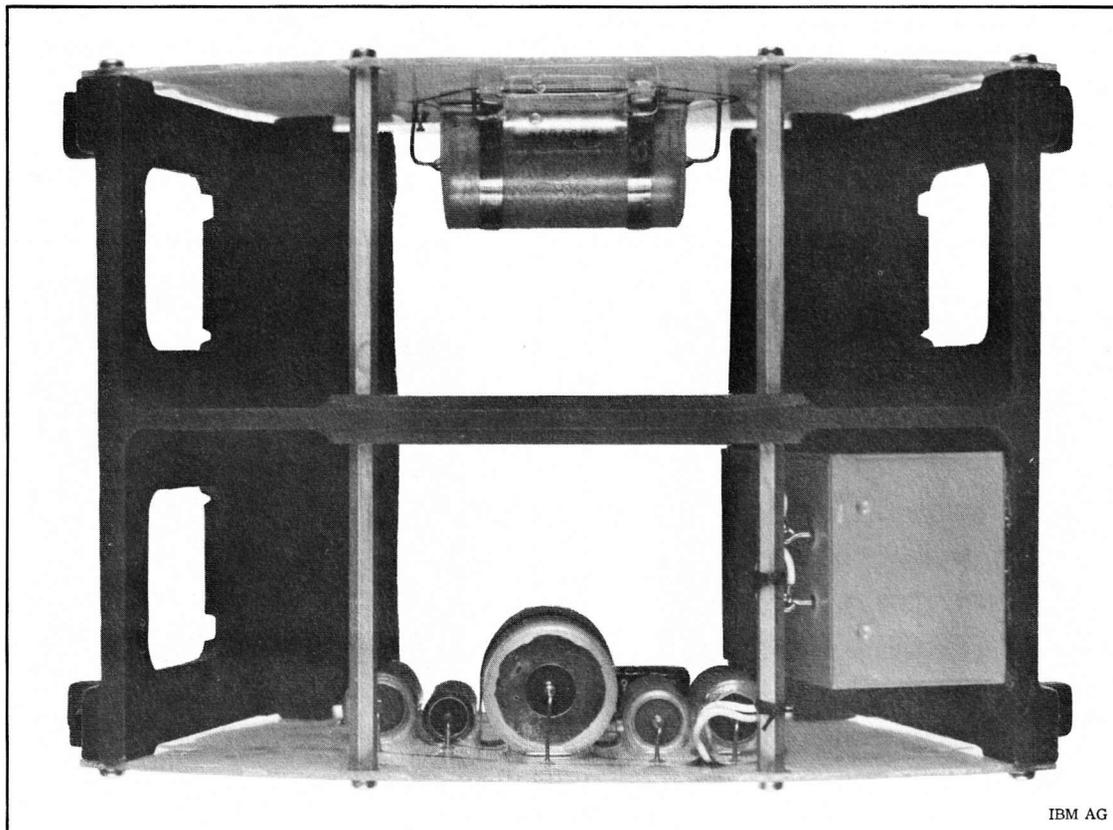


Figure 2-9. $\Delta\phi$ Y and $\Delta\phi$ P S-II Filter



IBM AG

Figure 2-10. $\Delta\phi Y$ and $\Delta\phi P$ S-IVB Filter



IBM AG

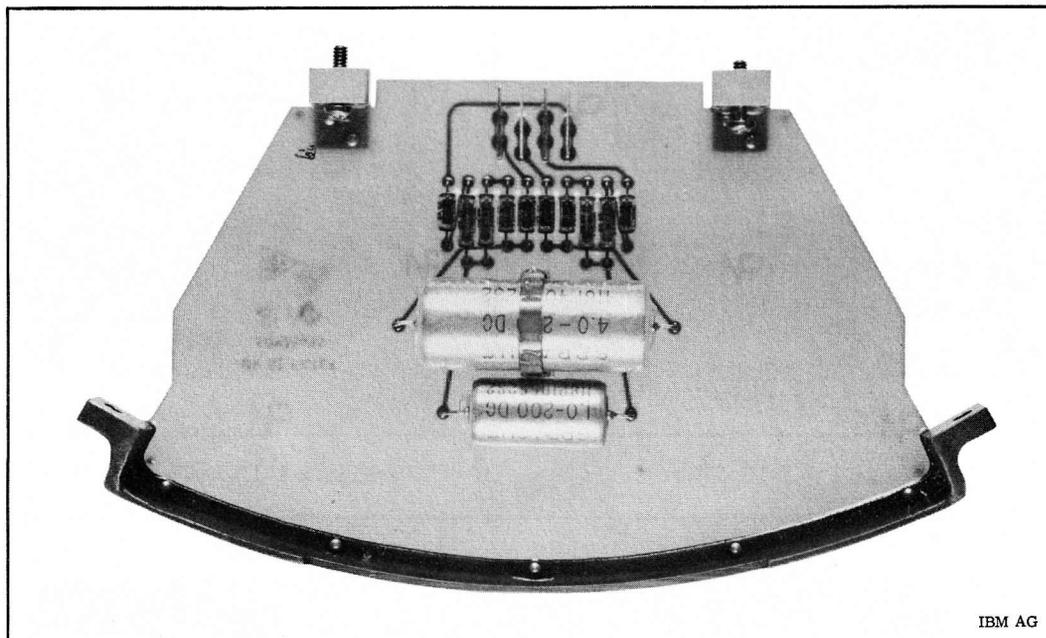
Figure 2-11. $\Delta\phi Y$, $\Delta\phi P$, ϕY , and ϕP S-IVB Filter

signal and produces a single-ended output signal. The single-ended output signal of differential amplifier Q10, Q11 is amplified by differential amplifier Q12 and applied to driver Q14, Q15. Driver Q14, Q15 further amplifies the signal and applies it to complementary symmetry amplifier Q7, Q9. The resulting output of complementary symmetry amplifier 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 mag-

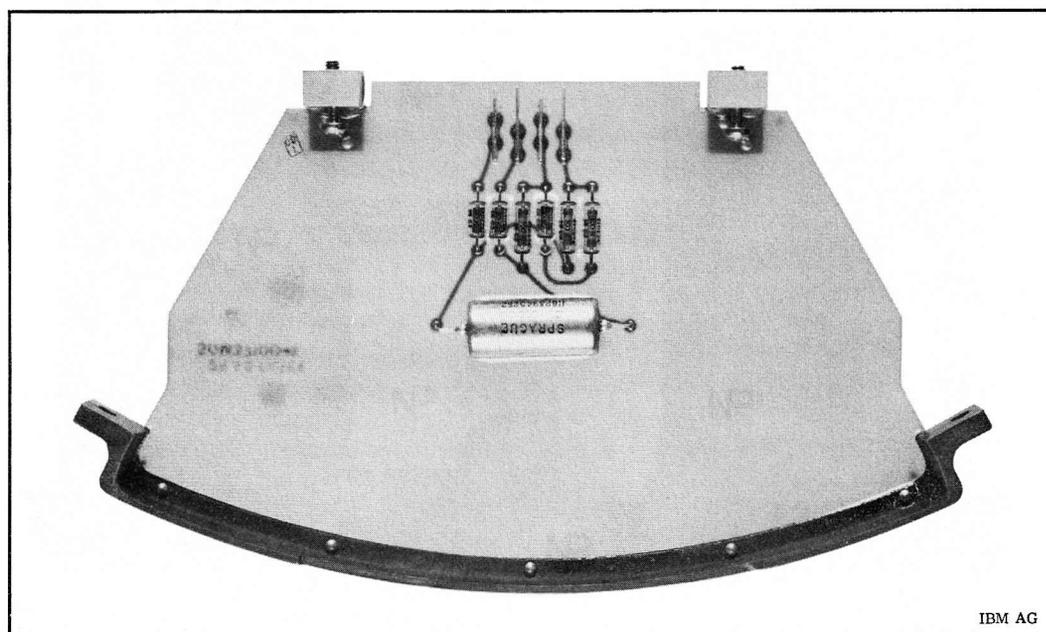
netic amplifier CR12 through CR15. The feedback signal opposes the input error signals of magnetic amplifier CR12 through CR15 so as to provide a damping action which prevents the servo cylinder in the S-IVB stage from overcorrecting.

The excitation signal for magnetic amplifier CR12 through CR15 is provided by inverter Q4, Q5 which converts the dc source of a 1 kHz signal. Regulation of inverter Q4, Q5 is provided by chopper Q2. Changes in the output voltage of filter L3, C2 are sensed by comparator-multivibrator Q1, Q3 which



IBM AG

Figure 2-12. $\Delta\phi R$ S-IC Filter

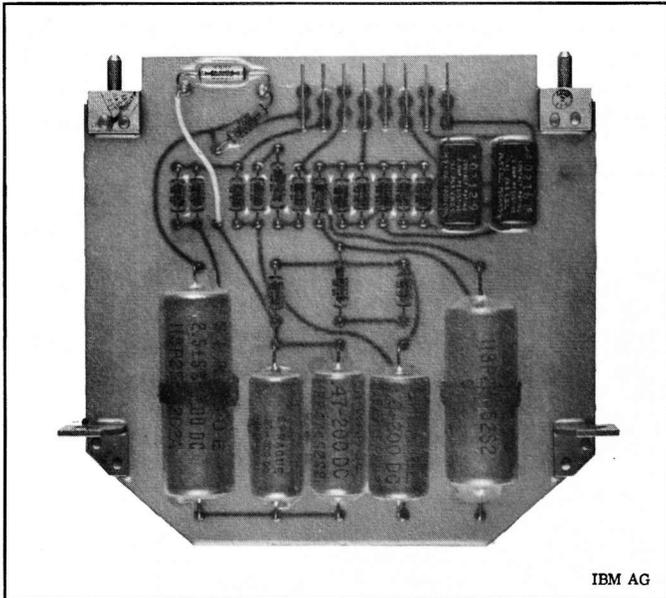


IBM AG

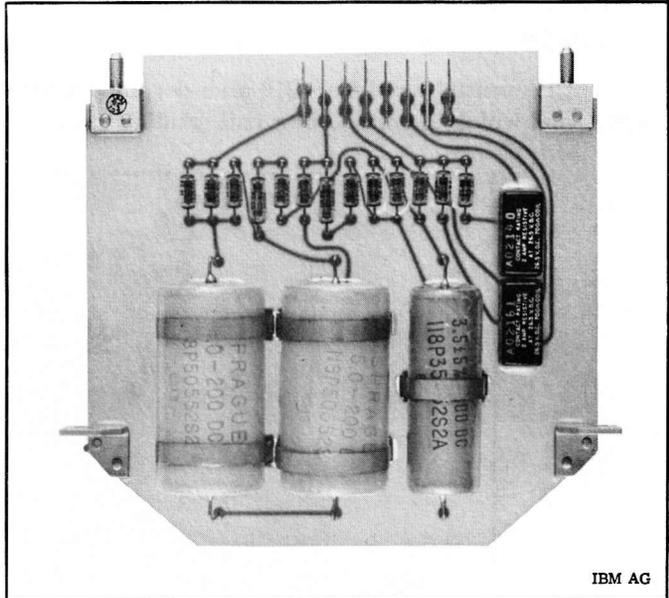
Figure 2-13. $\Delta\phi R$ S-II Filter

increases or decreases the output frequency of comparator-multivibrator Q1, Q3 in proportion to the difference between the output voltage of filter L3, C2 and a reference voltage. The output of comparator-multivibrator Q1, Q3 controls the ON and OFF time of chopper Q2.

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 rectifiers to produce the + and - 20 Vdc necessary for the operation of the amplifier stages in the 50 ma servo amplifier.



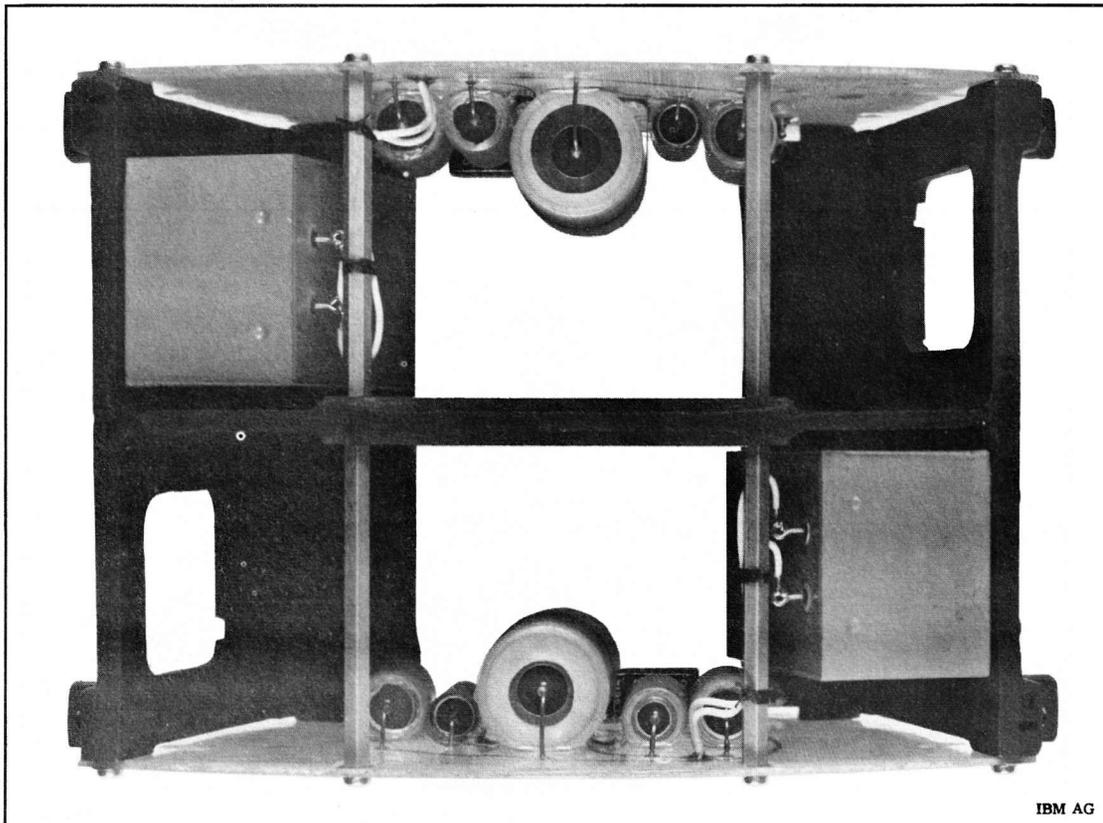
IBM AG



IBM AG

Figure 2-14. ϕ Y and ϕ P S-IC Filter

Figure 2-15. ϕ Y and ϕ P S-II Filter



IBM AG

Figure 2-16. ϕ Y and ϕ P S-IVB Filter

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

Servo Comparator

The Flight Control Computer contains a yaw servo comparator and a pitch servo comparator. The

yaw servo comparator is used to monitor the outputs of the triple redundant yaw 50 ma servo amplifiers during the S-IVB burn mode. The pitch servo comparator is used to monitor the outputs of the triple redundant pitch 50 ma servo amplifiers during the S-IVB burn mode. A photograph of the servo comparator is shown in Figure 2-23. A functional block diagram of the servo comparator is shown in Figure 2-24.

The purpose of the servo comparator is to compare the identical drive signal outputs of two of its

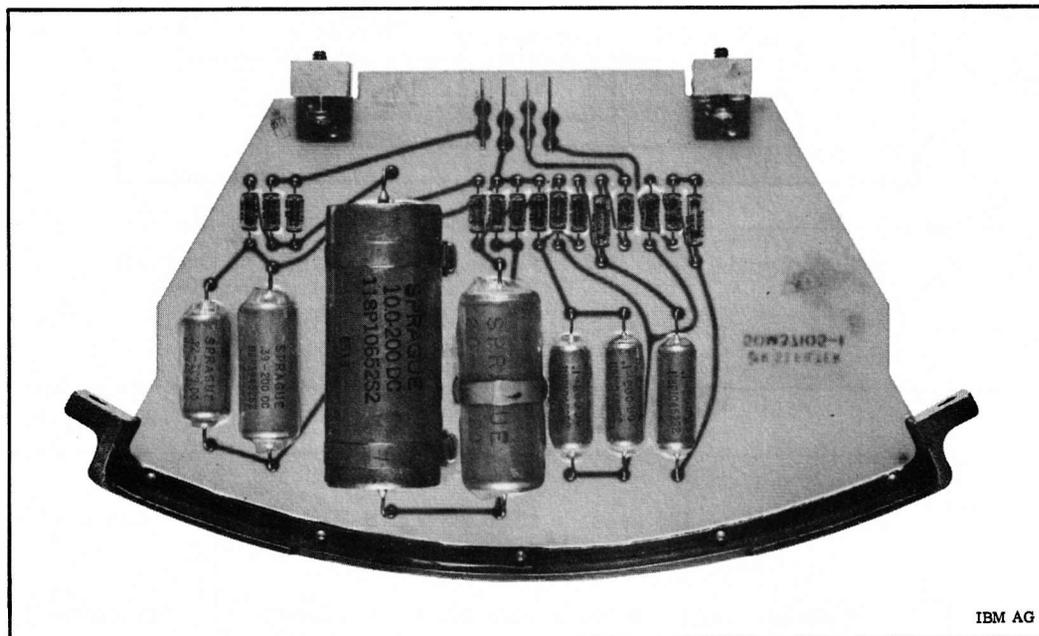


Figure 2-17. ϕ R S-IC Filter

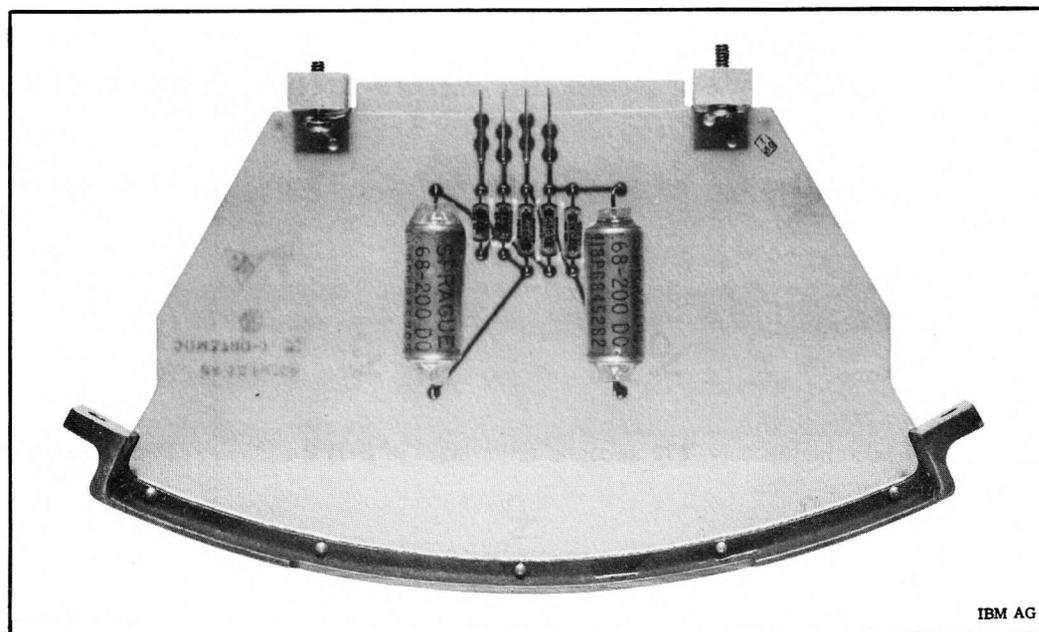


Figure 2-18. ϕ R S-II Filter

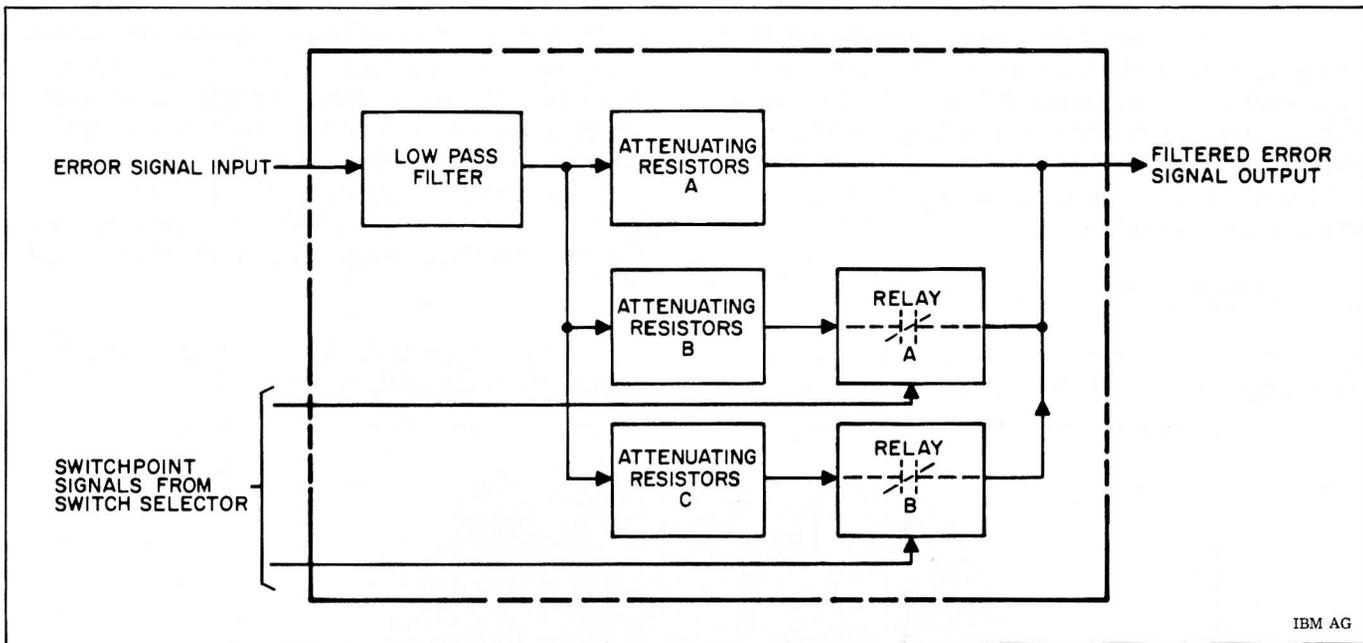


Figure 2-19. Typical Bending Mode Filter, Functional Block Diagram

Table 2-1. Bending Mode Filters Switchpoints

Filter	Switchpoint	Switchpoint Function
$\Delta\phi Y$ S-IC } $\Delta\phi P$ S-IC }	Switchpoint 1 - 100 seconds after start of S-IC burn.	Decreases signal level.
$\dot{\phi} Y$ S-IC } $\dot{\phi} P$ S-IC }	Switchpoint 2 - 120 seconds after start of S-IC burn.	Decreases signal level.
$\Delta\phi R$ S-IC } $\dot{\phi} R$ S-IC }	None	
$\Delta\phi Y$ S-II } $\Delta\phi P$ S-II }	Switchpoint 3 - 60 seconds after start of S-II burn.	Decreases signal level.
$\dot{\phi} Y$ S-II } $\dot{\phi} P$ S-II }	Switchpoint 4 - 190 seconds after start of S-II burn.	Decreases signal level.
$\Delta\phi R$ S-II } $\dot{\phi} R$ S-II }	None	
$\Delta\phi Y$ S-IVB } $\Delta\phi P$ S-IVB }	None	
$\dot{\phi} Y$ S-IVB } $\dot{\phi} P$ S-IVB }	Switchpoint 5 - 270 seconds after start of S-IVB second burn.	Decreases signal level.

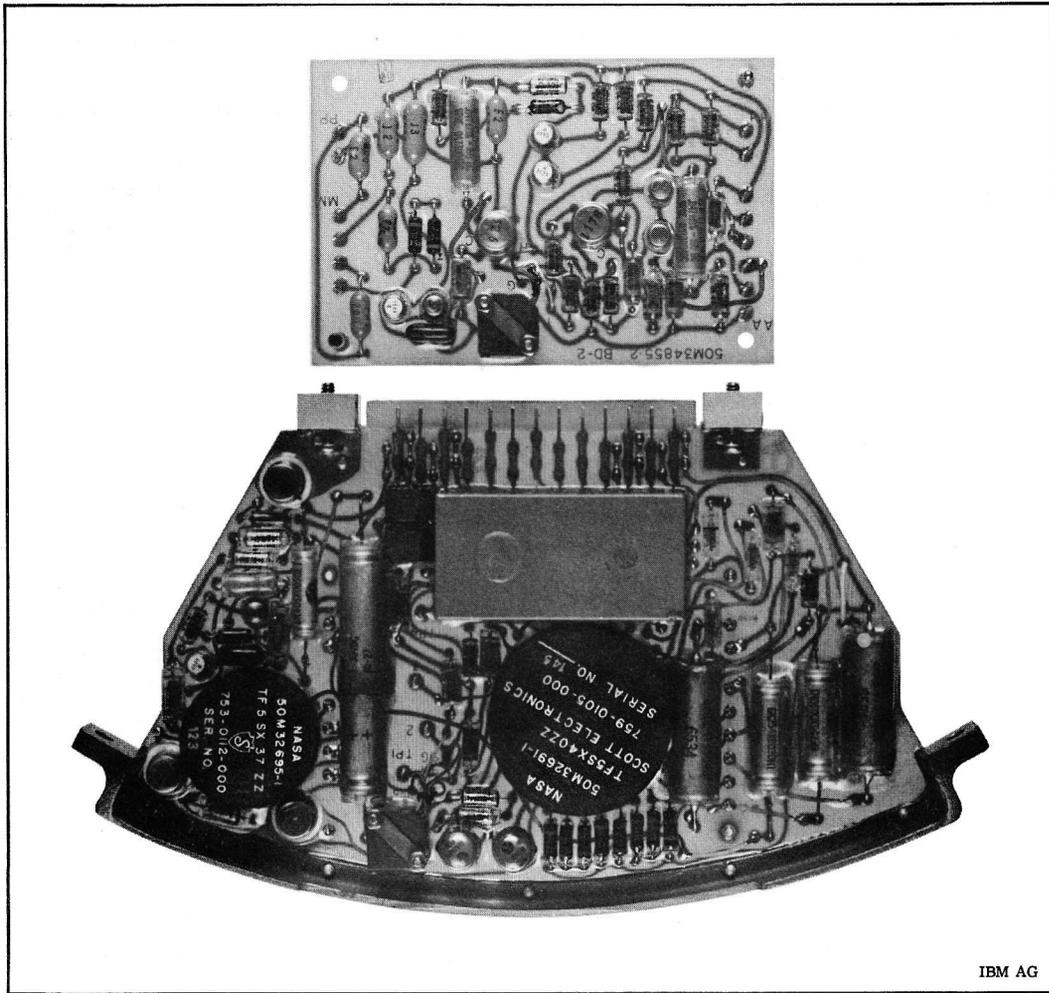


Figure 2-20. 50 Ma Servo Amplifier

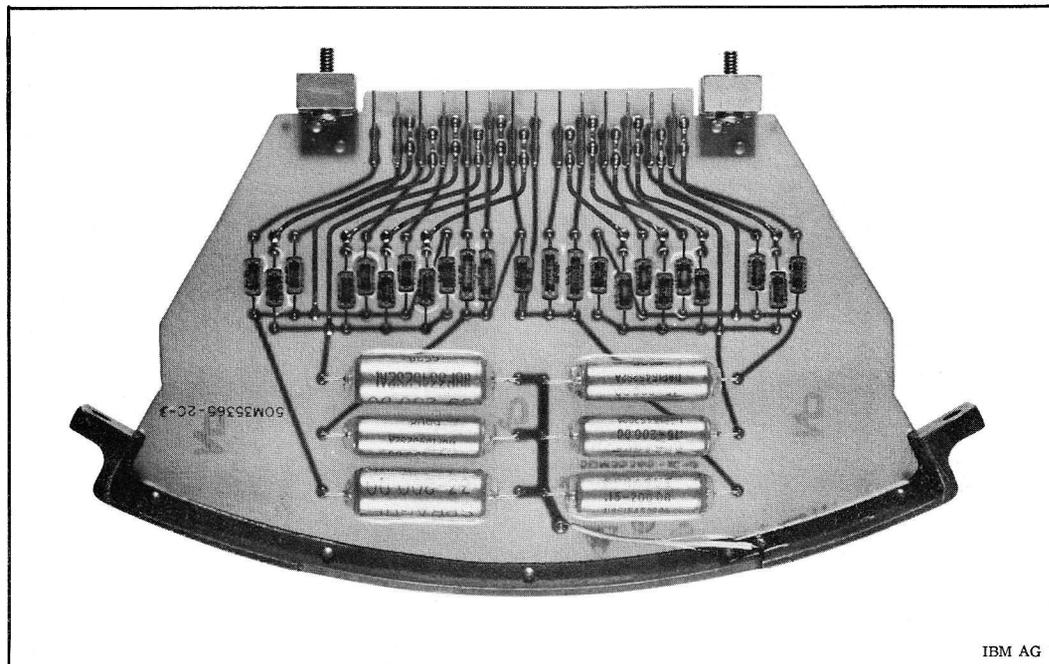
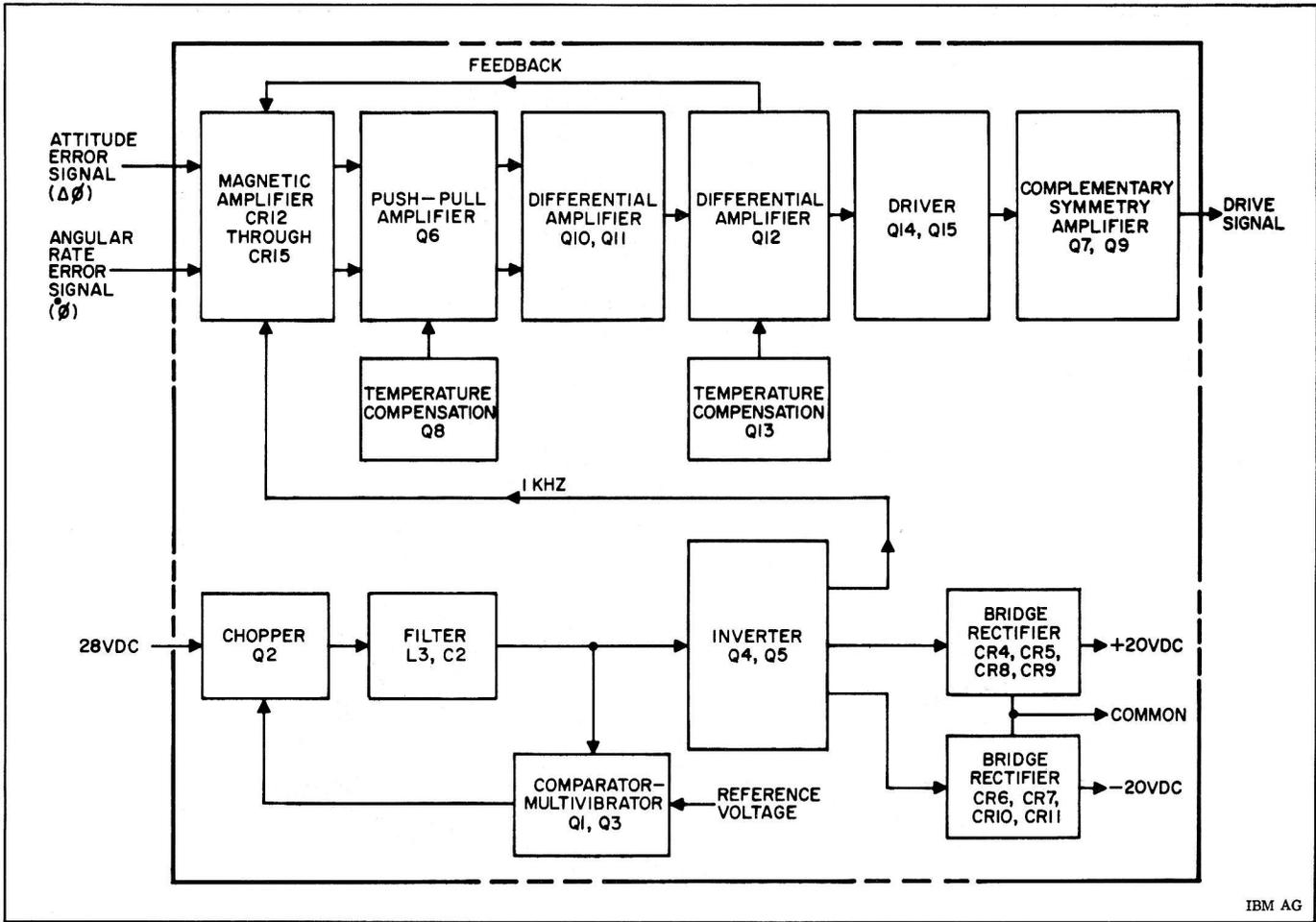
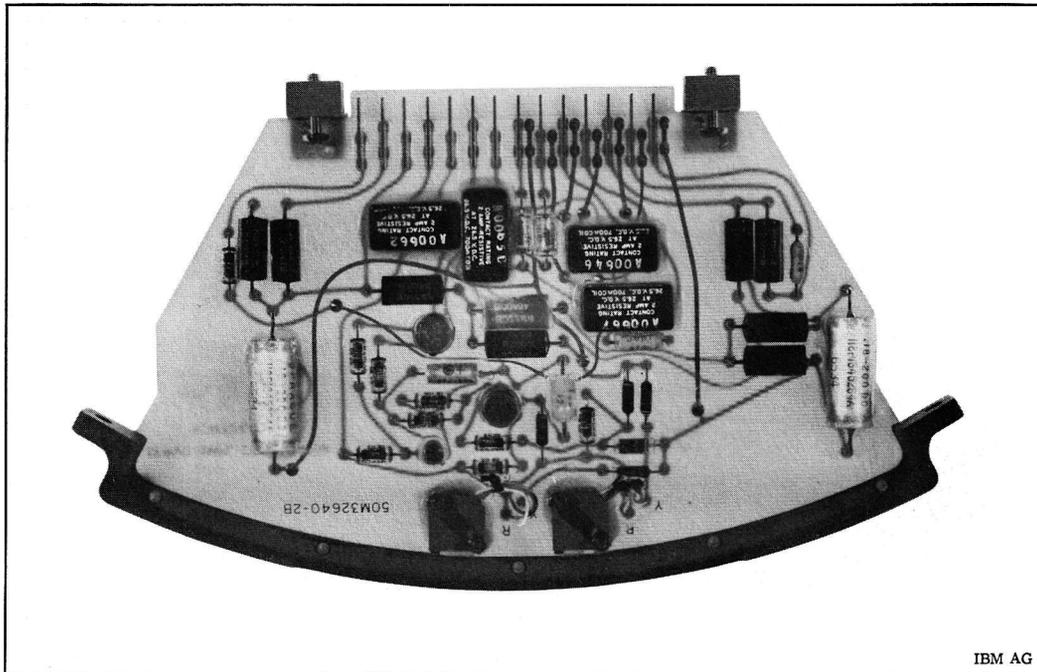


Figure 2-21. Servo Scaling Board (Typical)



IBM AG

Figure 2-22. 50 Ma Servo Amplifier, Functional Block Diagram



IBM AG

Figure 2-23. Servo Comparator

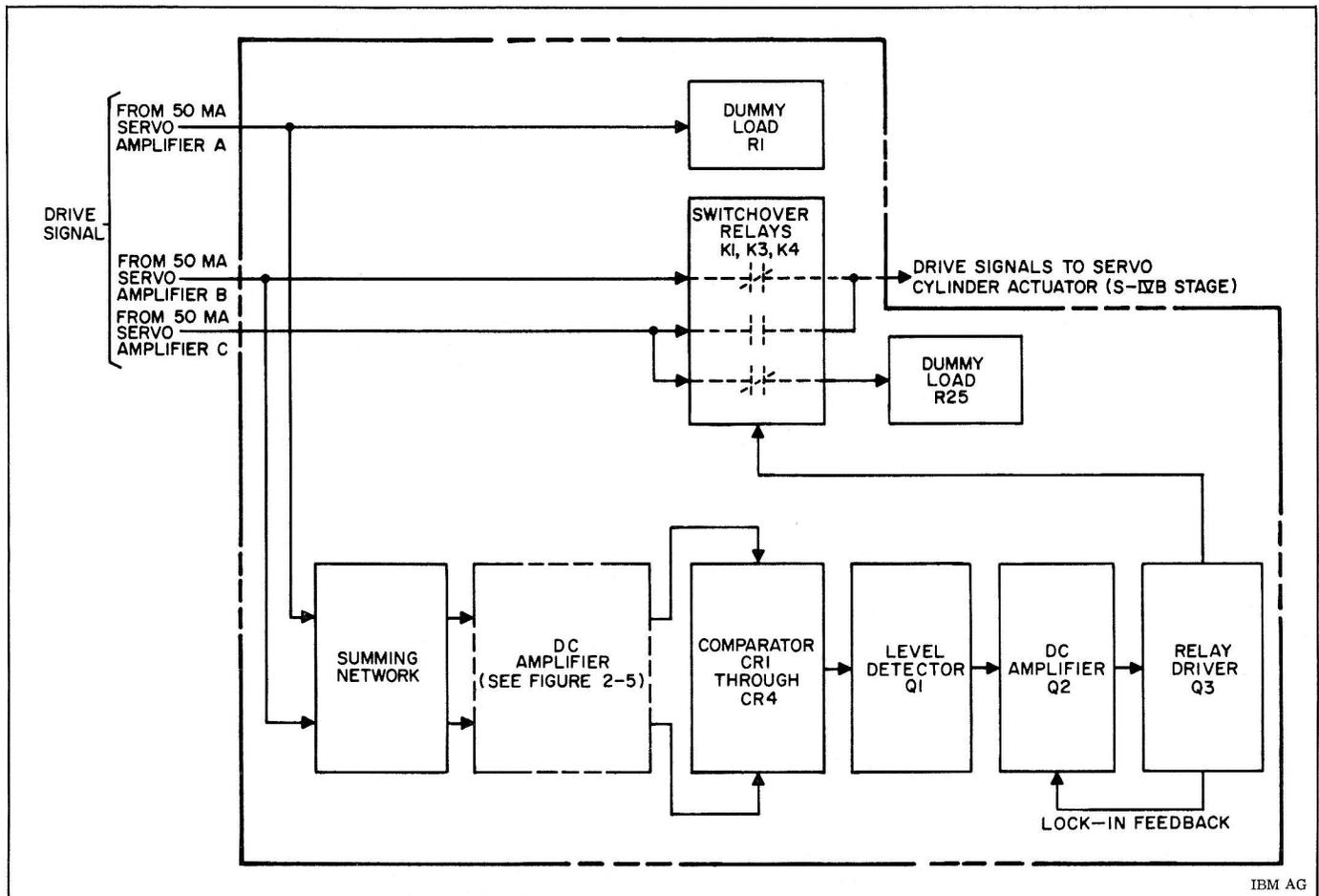


Figure 2-24. Servo Comparator, Functional Block Diagram

associated triple redundant 50 ma servo amplifiers, one of which is applied to a servo actuator in the S-IVB stage. If a malfunction occurs in either of the two 50 ma servo amplifiers being compared, the servo comparator automatically terminates the signal being applied to the servo actuator and applies the drive signal from the third (standby) 50 ma servo amplifier to the servo actuator.

Identical drive signals from the triple redundant 50 ma servo amplifiers (A, B, and C) are applied to the servo 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 actuator in the S-IVB stage. The drive signal from 50 ma servo amplifier A is applied to dummy load R1, which represents the load presented by the servo actuator. The drive signal from 50 ma servo amplifier C is normally routed through closed relay contacts in switchover relays K1, K3, K4 to dummy load R25, which also represents the load presented by the servo actuator.

The drive signals from 50 ma servo amplifiers A and B are also coupled through a summing network to

a dc amplifier. The drive signals are amplified by the dc amplifier and applied to comparator CR1 through CR4.

Under a normal balanced condition (both inputs to comparator CR1 through CR4 are equal), there is no signal applied to level detector Q1. If a malfunction occurs in either of the 50 ma servo amplifiers (A or B), an unbalanced condition occurs within comparator CR1 through CR4 causing level detector Q1 to change states. The resulting output of level detector Q1 is amplified by dc amplifier Q2 and applied to relay driver Q3, causing relay driver Q3 to conduct. The conduction of relay driver Q3 energizes switchover relays K1, K3, K4, and causes a lock-in feedback to be applied to dc amplifier Q1. The lock-in feedback forward biases dc amplifier Q1, which will in turn keep the relay driver Q3 locked in a conducting state.

With switchover relays K1, K3, and K4 energized, the drive signal from 50 ma servo amplifier B is terminated at open relay contacts, and the drive signal from 50 ma servo amplifier C is removed from dummy load R25 and routed through closed relay contacts to a servo actuator in the S-IVB stage.

Spatial Amplifier

The Flight Control Computer contains triple redundant roll-yaw I spatial amplifiers, triple redundant roll-yaw II spatial amplifiers and triple redundant pitch spatial amplifiers. The purpose of the spatial amplifiers is to develop spatial thruster control signals for the control relay packages in the S-IVB stage in proportion to the yaw, pitch, and roll error signals. Triple redundancy is used to ensure a high reliability factor within the Flight Control Computer. A photograph of the spatial amplifier is shown in Figure 2-25. A functional block diagram of the spatial amplifier is shown in Figure 2-26.

Attitude ($\Delta\phi$), and angular rate ($\dot{\phi}$) error signals are applied to the coil windings in magnetic amplifier Q6, Q7. The magnetic amplifier algebraically combines the attitude and angular rate error 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 balance output signal of differential amplifier Q10, Q11 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 differential amplifier 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 differential amplifier 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 of sufficient amplitude to overcome the threshold feedback.

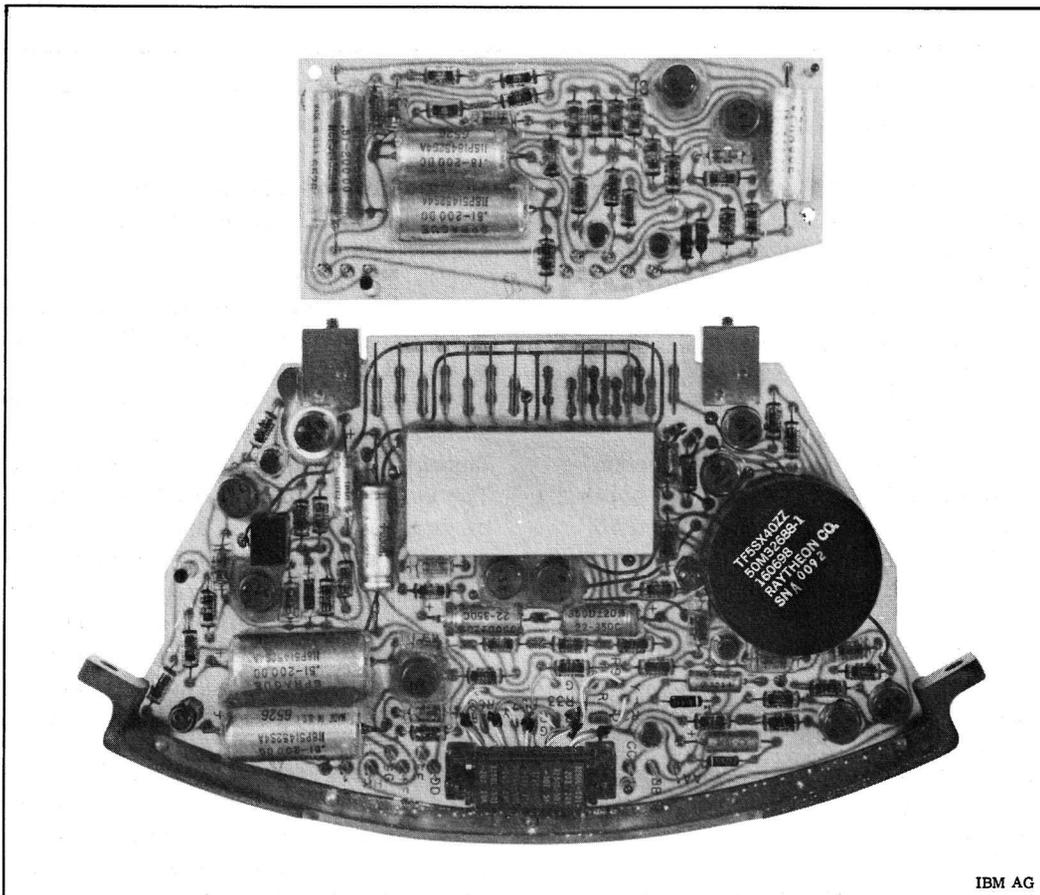


Figure 2-25. Spatial Amplifier

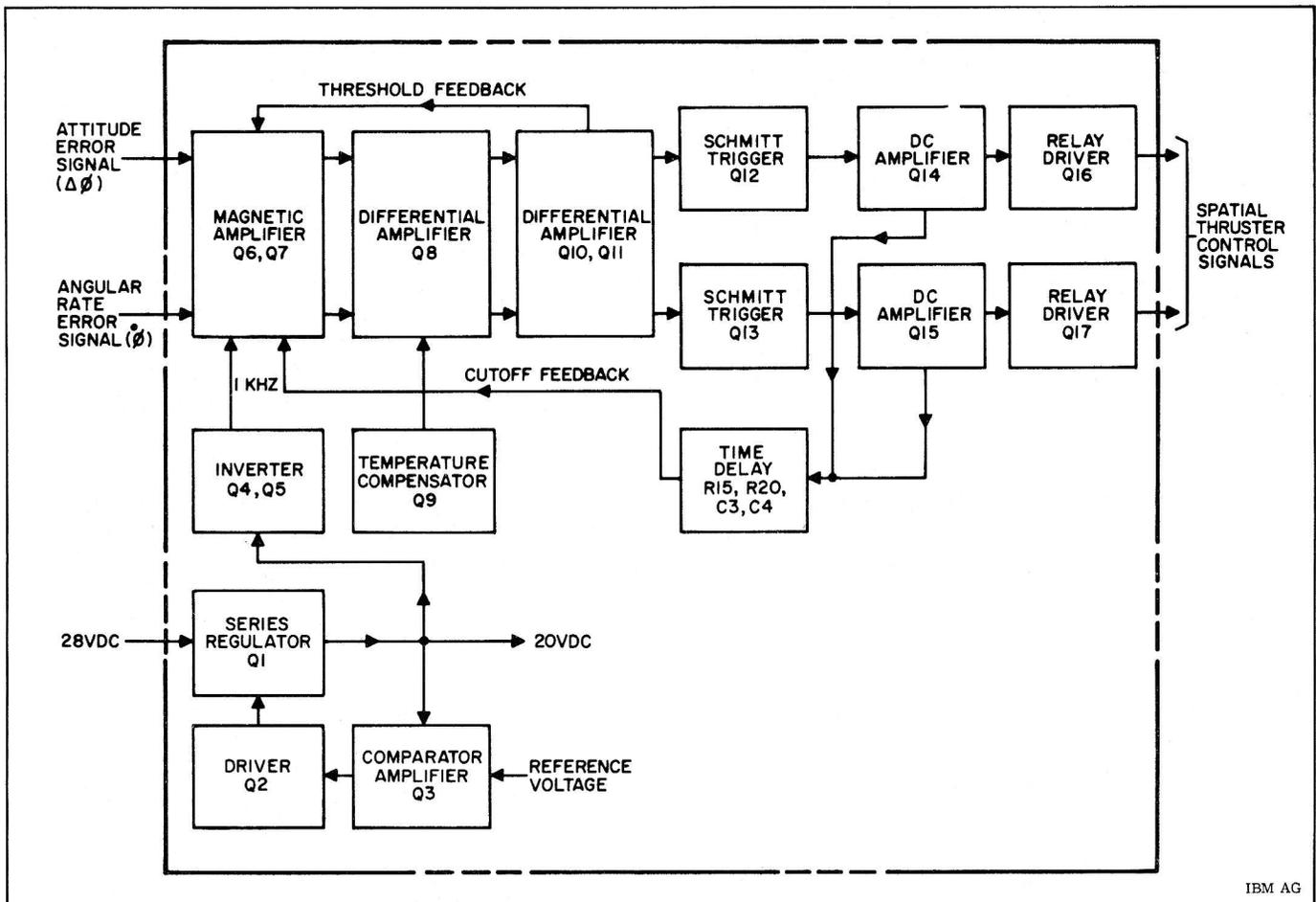


Figure 2-26. Spatial Amplifier, Functional Block Diagram

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 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 takes the cutoff feedback signal 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 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 three spatial comparators which are used in conjunction with the triple redundant roll-yaw I spatial amplifiers, the triple redundant roll-yaw II spatial amplifiers, and 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 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 the spatial comparator is shown in Figure 2-27. A functional block diagram of the spatial comparator is shown in Figure 2-28.

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 2-28.

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 amplifiers 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 Q6, Q7, Q9, Q10 are equal), no signal appears at the output of comparator-amplifier Q6, Q7, Q9, Q10. When a malfunction occurs in either of the spatial amplifiers (A or B), an unbalanced condition exists within comparator-amplifier Q6, Q7, Q9, Q10, 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

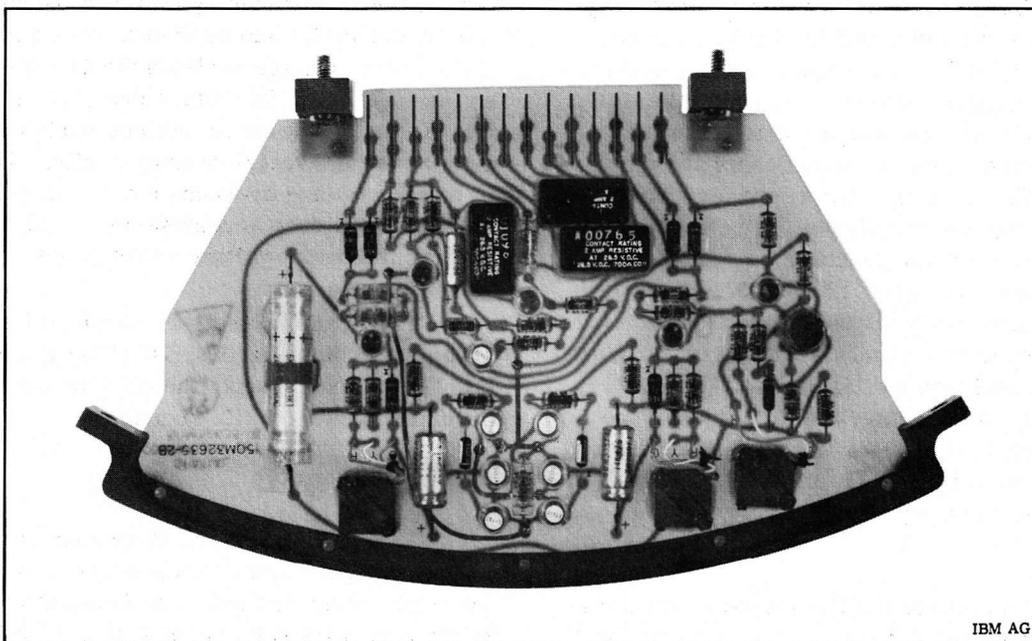


Figure 2-27. Spatial Comparator

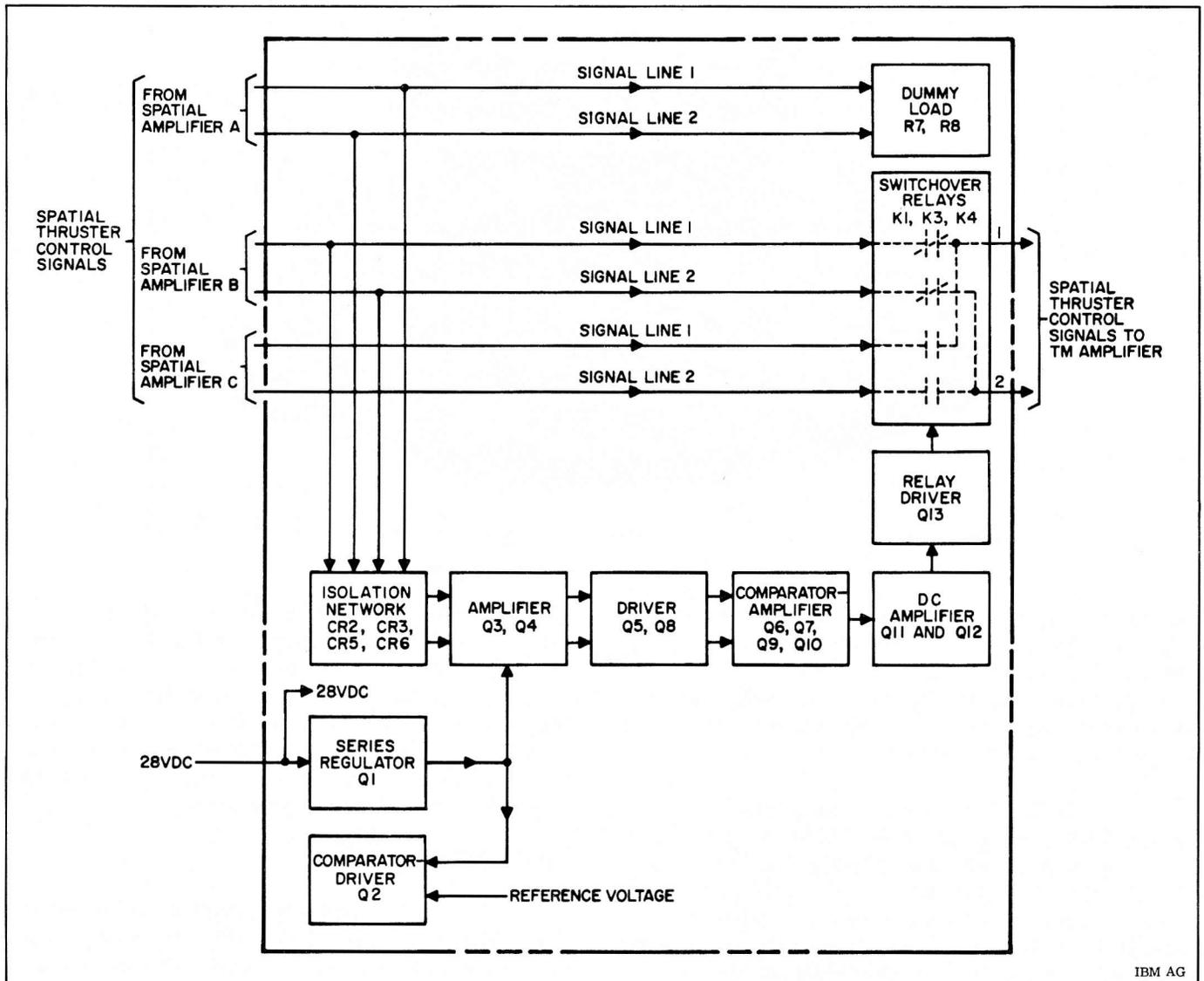


Figure 2-28. Spatial Comparator, Functional Block Diagram

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 series regulator 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 the spatial synchronizer is shown in Figure 2-29. A functional block diagram of the spatial synchronizer is shown in Figure 2-30.

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 relay drivers Q1, Q4 and Q2, Q3 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

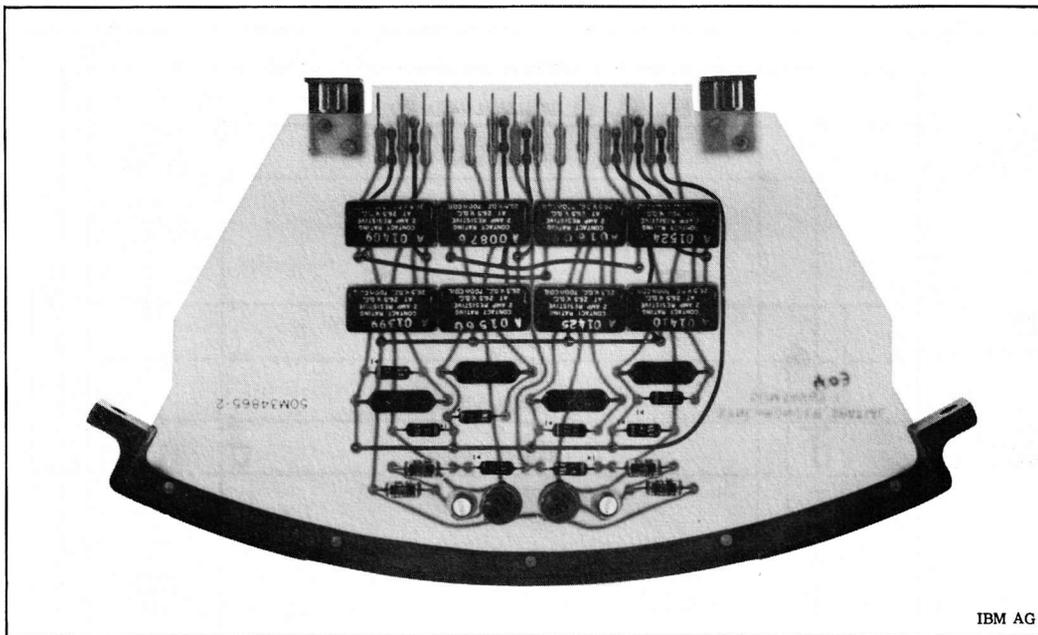


Figure 2-29. Spatial Synchronizer

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 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 de-energized when relay drivers

Q1, Q4 and Q2, Q3 are cutoff. 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 the TM amplifier is shown in Figure 2-31. A functional block diagram of the TM amplifier is shown in Figure 2-32.

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. Magnetic amplifier CR1 through CR4 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 magnetic amplifier CR1 through CR4 is coupled through temperature compensator RT1 to produce the telemetry output signal.

The excitation signal for magnetic amplifier CR1 through CR4 is developed by inverter Q1, Q2

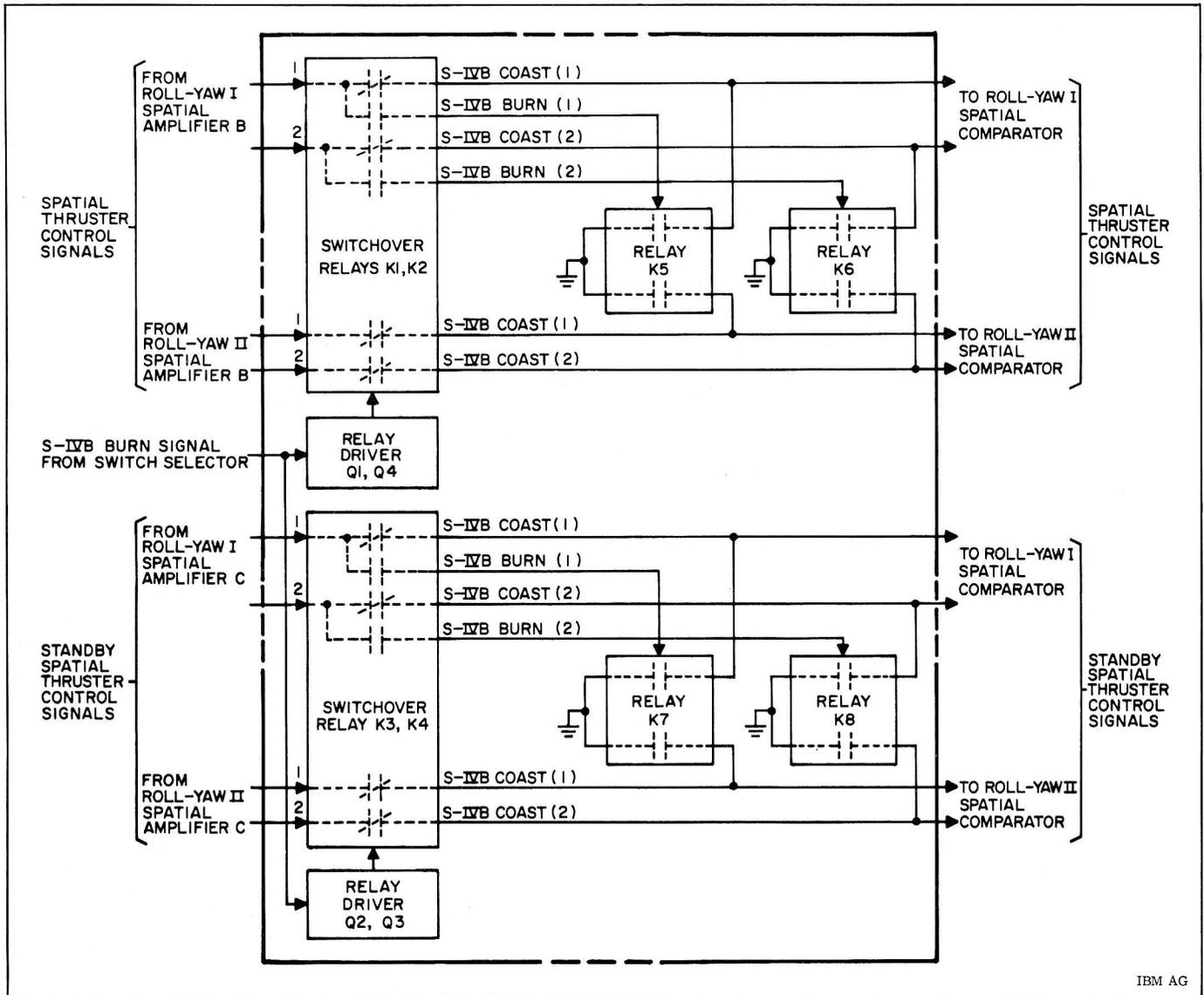


Figure 2-30. Spatial Synchronizer, Functional Block Diagram

which converts the 28 Vdc source to a 1 kHz signal. The amplitude of the magnetic amplifier output is pre-set by the 28 Vdc bias control.

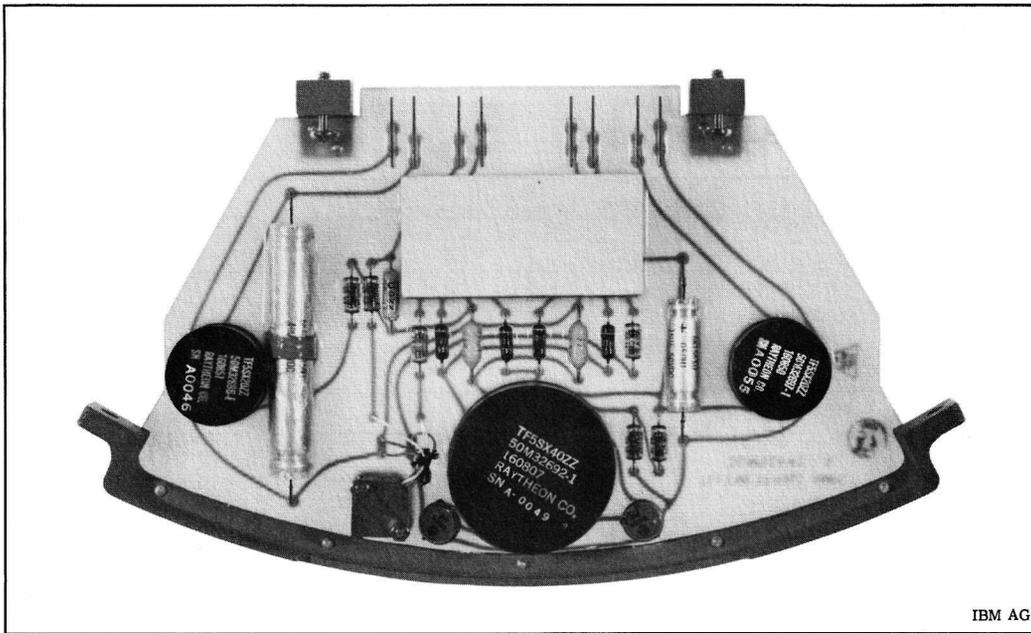
Temperature changes which affect the overall performance of the TM amplifier are sensed by temperature compensator RT1. Temperature compensator RT1 then controls the gain of magnetic amplifier CR1 through CR4 by applying a feedback in proportion to the temperature change.

Limiters

The Flight Control Computer contains triple redundant yaw limiters, triple redundant pitch limiters, and triple redundant roll limiters. The limiters are used during Apollo spacecraft control of the S-IVB

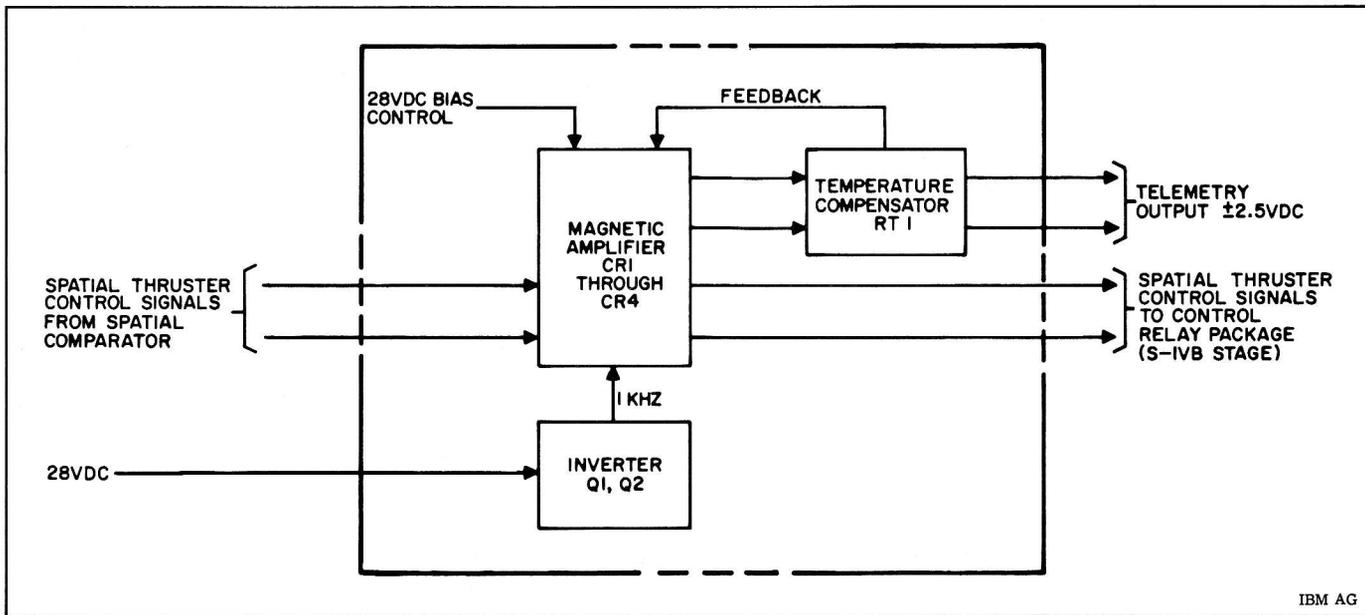
stage during the coast mode only. The purpose of the limiters is to prevent overcorrection of the S-IVB stage attitude as a result of large attitude error signals. A photograph of the limiter is shown in Figure 2-33. A functional block diagram of the limiter is shown in Figure 2-34.

Each limiter assembly actually consists of three limiters, one limiter for each of the attitude error signals ($\Delta\phi Y$, $\Delta\phi P$, and $\Delta\phi R$). The yaw attitude error signal ($\Delta\phi Y$) from the yaw dc amplifier is applied to limiter Q1. Limiter Q1 provides a dc output which is a linear function of the dc input, but which is always less than the dc input. Any error signals above 12 volts dc at the input to Q1 are limited to not more than approximately 11 volts at the output. The balanced output from Q1 is sent to the roll-yaw I spatial amplifier and the roll-yaw II spatial amplifier.



IBM AG

Figure 2-31. TM Amplifier



IBM AG

Figure 2-32. TM Amplifier, Functional Block Diagram

Limiter Q2 limits the pitch attitude error signal ($\Delta\phi P$) from the pitch dc amplifier, and operates in the same manner as limiter Q1. The balanced output of limiter Q2 is sent to the pitch spatial amplifier. Limiter Q3 limits the roll attitude error signal ($\Delta\phi R$) from the roll dc amplifier. Limiter Q3 operates in the same manner as does Q1, and its output is also sent to the roll-yaw I spatial amplifier and the roll-yaw II spatial amplifier.

Switching Circuit

The switching circuit consists of a switching control, switching circuit A1, switching circuit A2, and switching circuit A3. These subassemblies operate together to control the routing of error signals during the S-IC burn mode, S-II burn mode, S-IVB burn mode, and the S-IVB coast mode. They are also used during systems test to control the routing of test

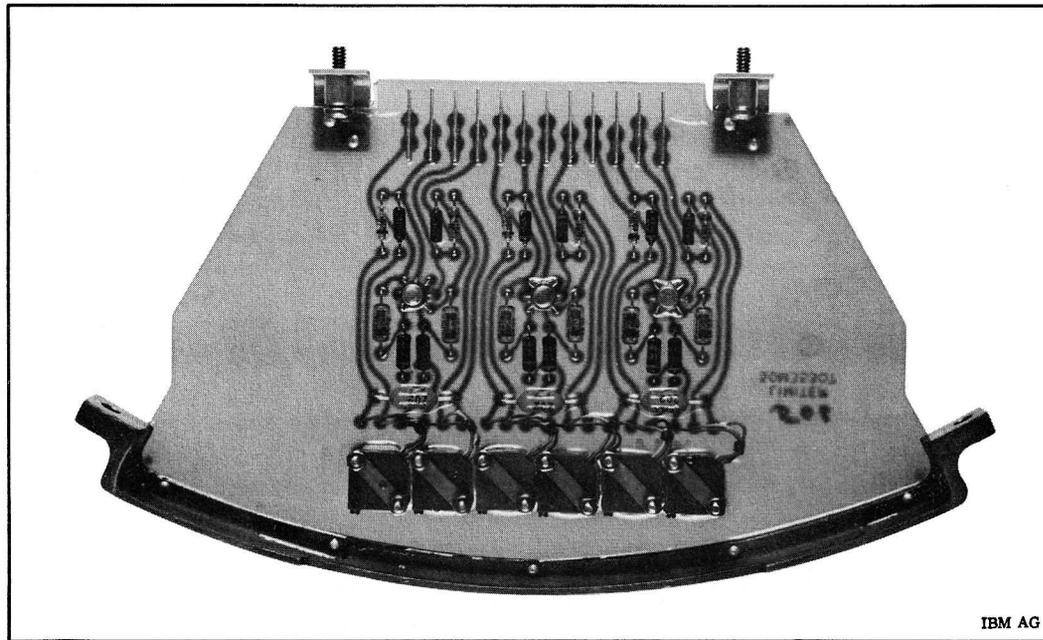


Figure 2-33. Limiter

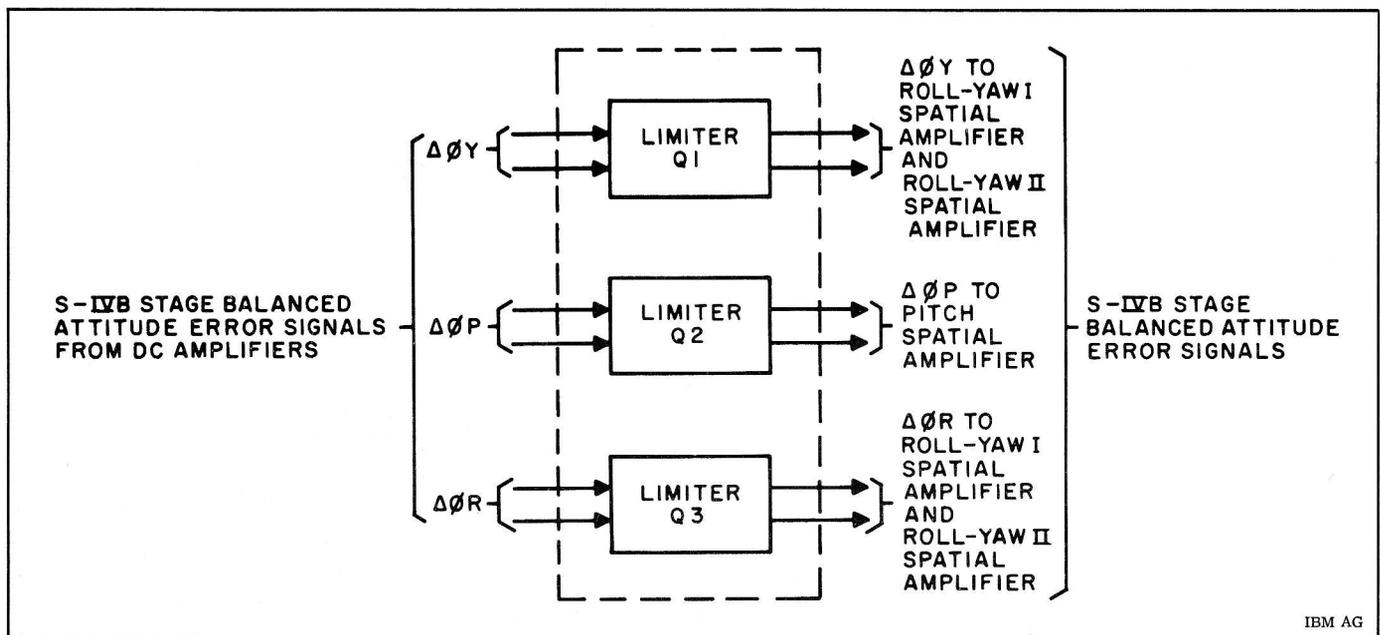


Figure 2-34. Limiter, Functional Block Diagram

signals from the GSE. Each switching circuit assembly contains a group of relays and relay drivers. The switching control circuit isolates the various switching circuit inputs. A photograph of the switching control is shown in Figure 2-35, and photographs of switching circuits A1, A2, and A3 are shown in Figures 2-36 through 2-38.

The following functional description of the switching circuit describes only the activating sequence and basic functions of the relay groups. The

actual switching function and location of the relay contacts in relation to the other Flight Control Computer circuitry is illustrated in Figures 2-2 through 2-4. A functional block diagram of the switching circuits is shown in Figure 2-39.

Prior to liftoff, the S-IC burn mode substitute signal (+28 Vdc) from the GSE is routed through the switching control isolation diodes to relay drivers on switching circuits A1, A2 and A3. This causes the relay drivers to conduct, which in turn, energize

their associated relays. The relay symbol code on Figure 2-39 indicates which relays are energized. The relays energized are also listed in Table 2-2. The contacts of these relays connect the input error signals to the S-IC burn mode guidance control circuits. The relays remain energized until the S-II burn mode signal is applied to the switching circuit.

The contacts of relays A3K21 and A3K23 connect the 28 Vdc to the S-IC burn line to provide a lock-in.

At the start of the S-II burn mode, an S-II burn mode signal (28 Vdc) from the Switch Selector is applied through the switching control isolation diodes to relay drivers on switching circuits A1, A2, and A3.

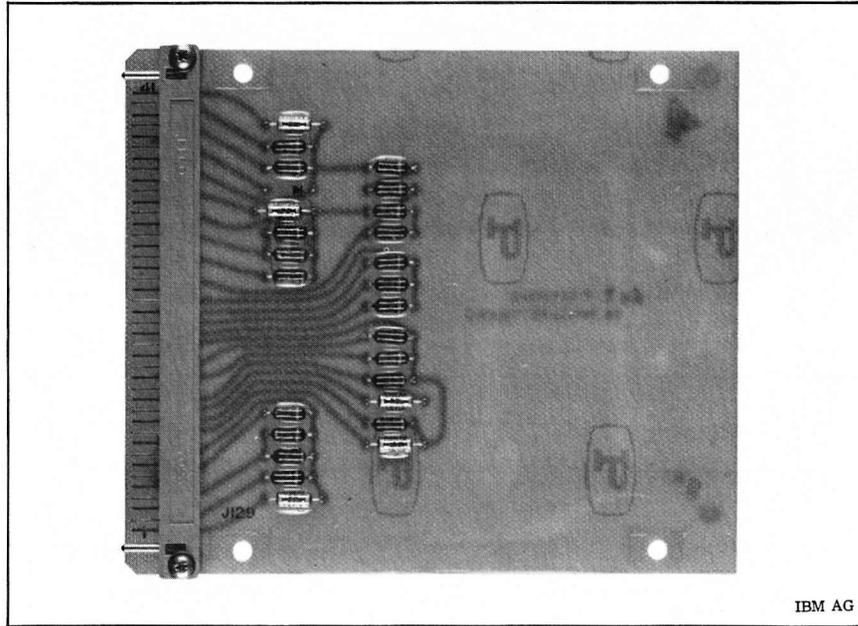


Figure 2-35. Switching Control

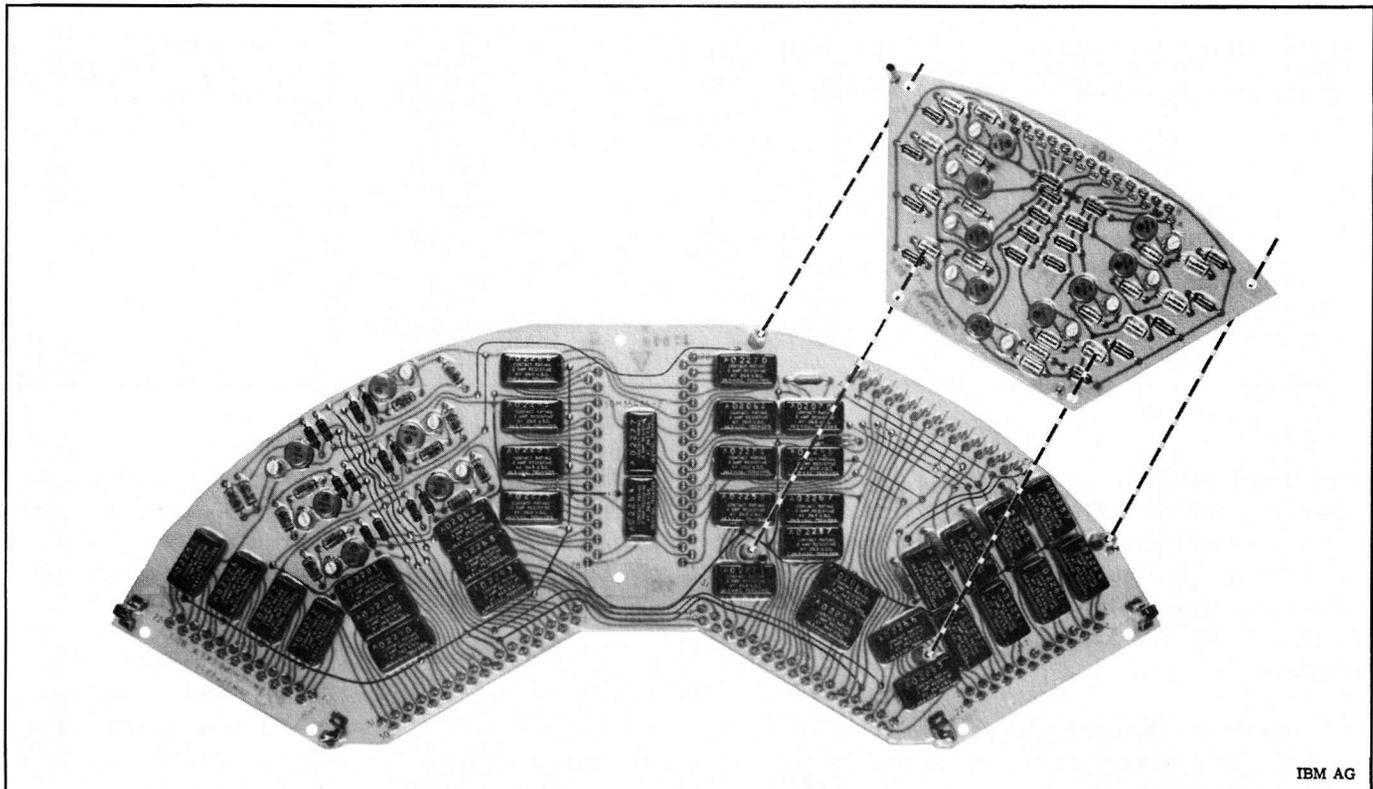


Figure 2-36. Switching Circuit A1

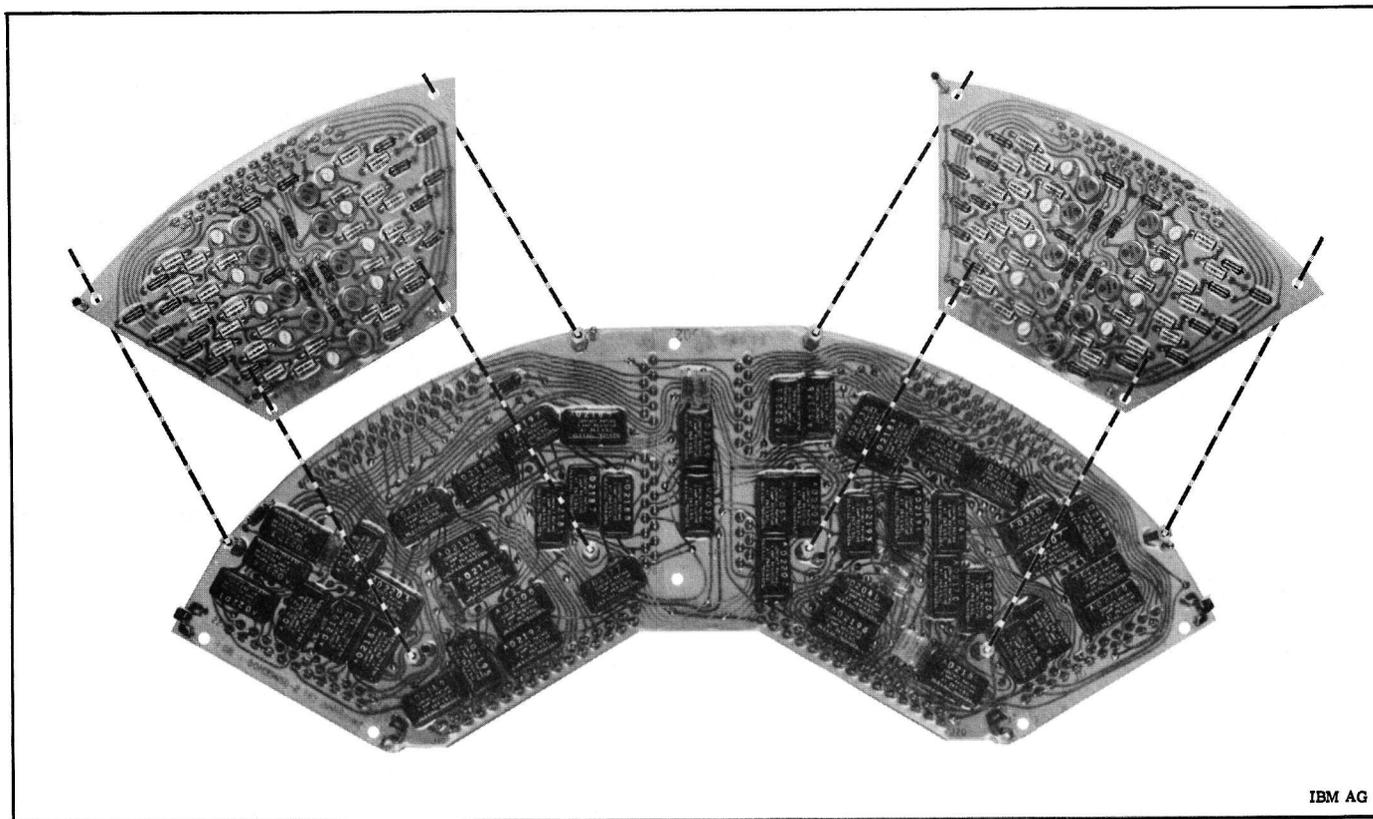


Figure 2-37. Switching Circuit A2

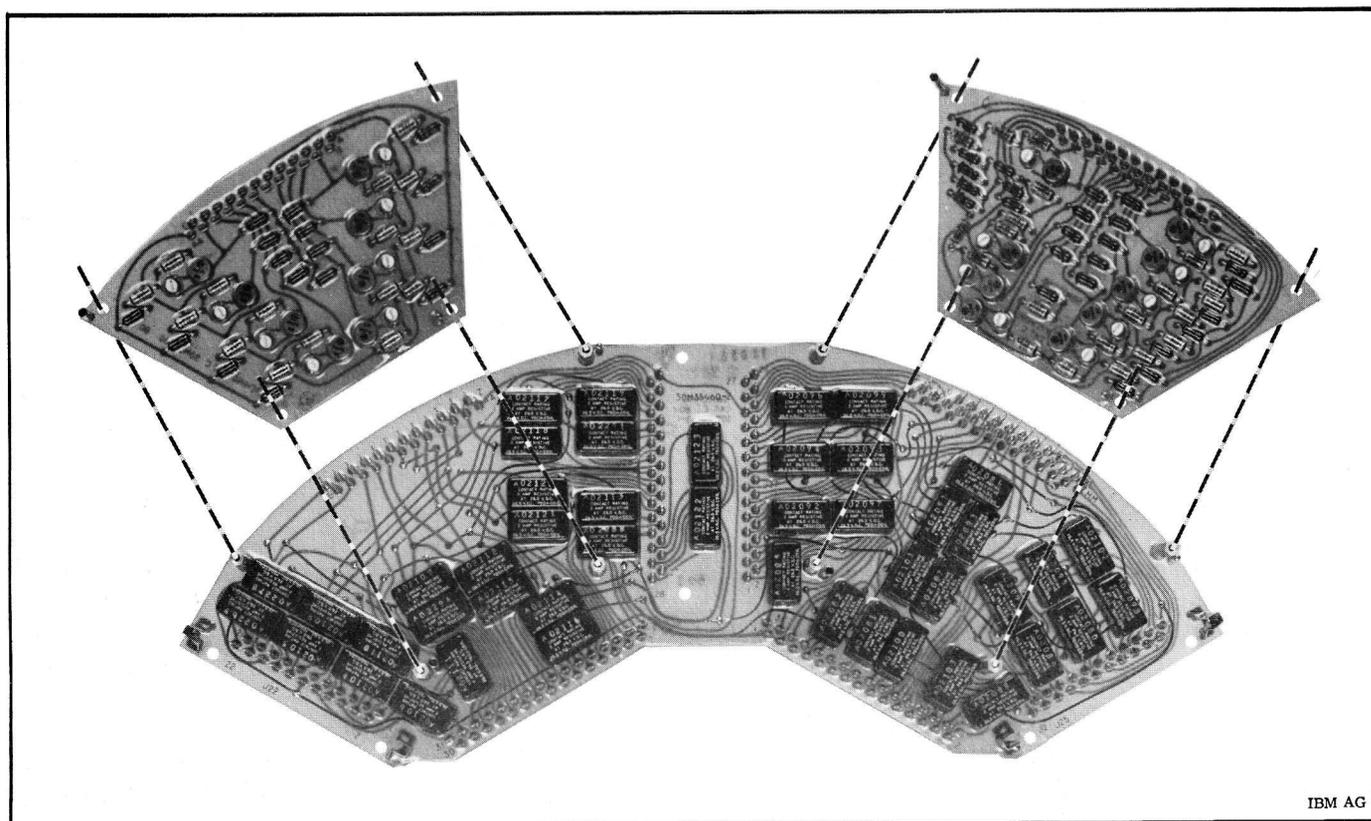


Figure 2-38. Switching Circuit A3

Table 2-2. Relay Switching Cross Reference

Signal In	Relays Energized	
	Switching Circuits	Relays
First Motion (S-IC Burn Mode)	A1	K11 thru K28
or	A2	K1 thru K12
S-IC Burn Mode Substitute	A3	K17 thru K23 and K36 thru K47
S-II Burn Mode	A1	K11 thru K28
or	A2	K1 thru K12
S-II Burn Mode Substitute	A3	K7 thru K16 and K24 thru K35
S-IVB Burn Mode	A1	K1 thru K10
or	A2	K13 thru K16
S-IVB Burn Mode Pressure	A3	K1 and K2
or		
S-IVB Burn Mode Substitute		
Apollo Control	A1	K29 thru K37
or	A2	K37 thru K44
IU Guidance Failure		
or		
Apollo Control Substitute		
or		
IU Guidance Failure Substitute		
S-IC Test	A2	K20 thru K36
	A3	K19 thru K23
S-II Test	A2	K20 thru K36
	A3	K3
S-IVB Test	A2	K20 thru K36
Coast Test	A2	K20 thru K36 and K45 thru K48

This causes the relays drivers to conduct, which in turn, energizes the indicated relays. The relays energized are listed in Table 2-2. The contacts of relays A3K13 and A3K16 are opened. This removes the 28 Vdc lock-in from the S-IC burn line. The Flight Control Computer is now set for guidance control during the S-II burn mode.

Upon completion of the S-II burn mode, the S-II burn mode signal is removed and the relays become de-energized. At the start of the S-IVB burn mode, the S-IVB burn mode signal (28 Vdc) from the

Switch Selector is applied through the switching control isolation diodes to relay drivers on switching circuits A1, A2, and A3. This causes the relay drivers to conduct, which in turn, energize the indicated relays. The relays energized are listed in Table 2-2. 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 switching circuit relays are de-energized. The contacts of the

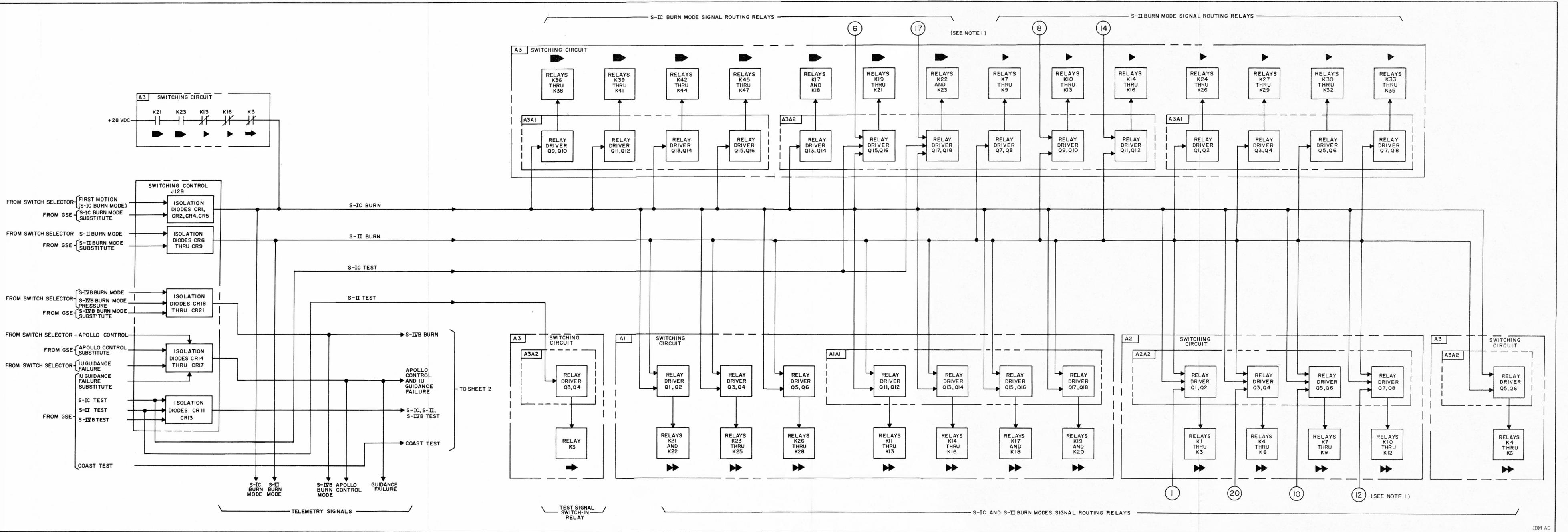


Figure 2-39. Switching Circuit, Functional Block Diagram (Sheet 1 of 2)

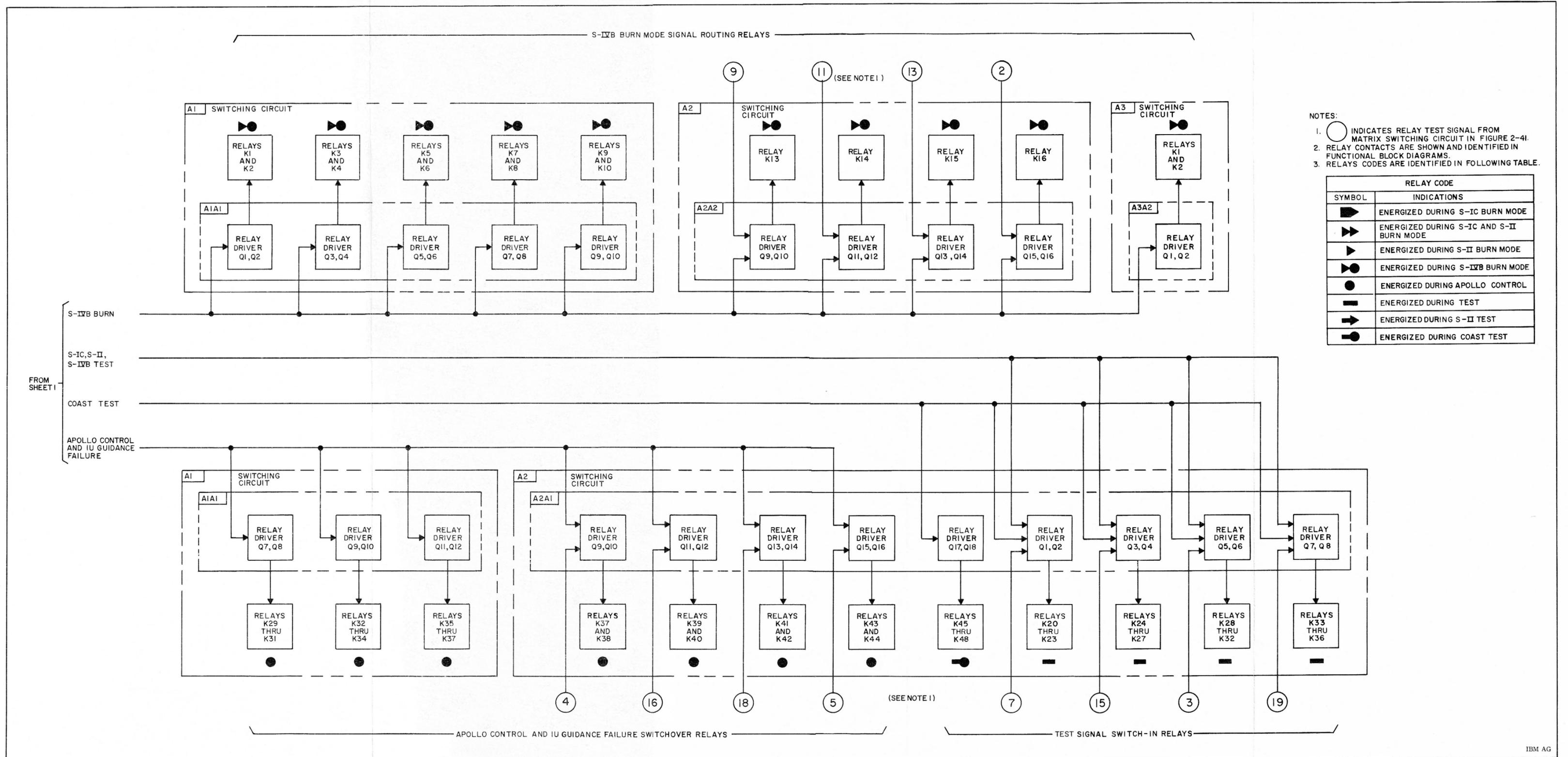


Figure 2-39. Switching Circuit, Functional Block Diagram (Sheet 2 of 2)

switching circuit relays are connected so that when they are in the de-energized condition, the input error signals are connected to the S-IVB coast mode guidance control circuits.

Guidance of the launch vehicle is normally controlled by the signals developed in the LVDC/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 or IU guidance failure signal from the Switch Selector. The Apollo control signal is coupled through the switching control isolation diodes to relay drivers on switching circuits A1 and A2. This causes the relay drivers to conduct, which in turn, energize the indicated relays. The relays energized are listed in Table 2-2.

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.

The outputs of the switching control isolation diodes are also sent to external telemetry equipment. This provides a means of monitoring the switching circuit during flight.

Redundancy Check Matrix

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 matrix. A photograph of the redundancy check matrix is shown in Figure 2-40. A functional block diagram of the redundancy check matrix is shown in Figure 2-41.

The redundancy check matrix consists of a diode matrix and four identical matrix drivers. The polarity (\pm) or absence (0) of a matrix driver input signal determines the polarity of its four outputs. Note 1 lists the three conditions that can exist. 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 the matrix enable signal from the GSE. This signal energizes the matrix enable relay causing 28 Vdc to be supplied to the matrix. 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, control 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 2-39. Therefore,

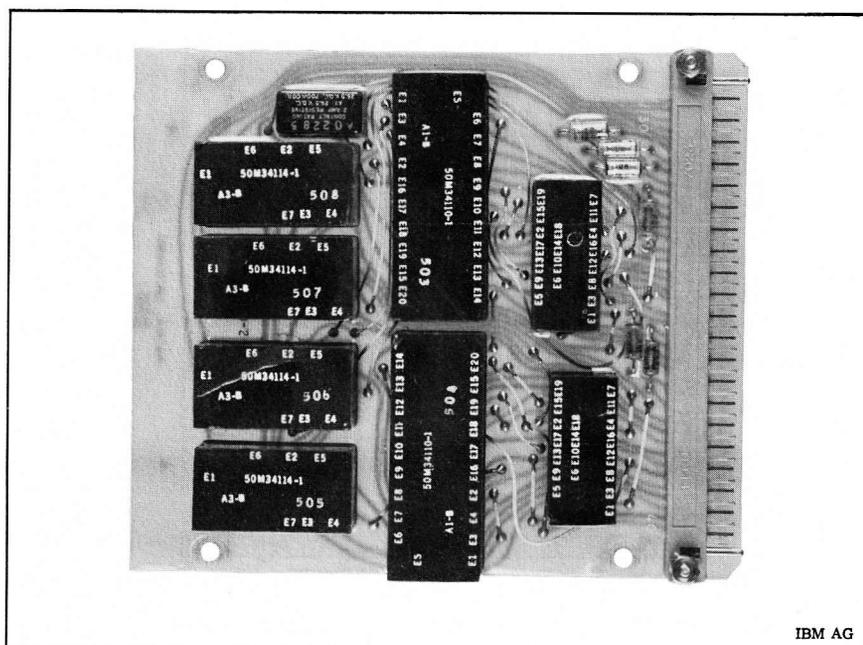
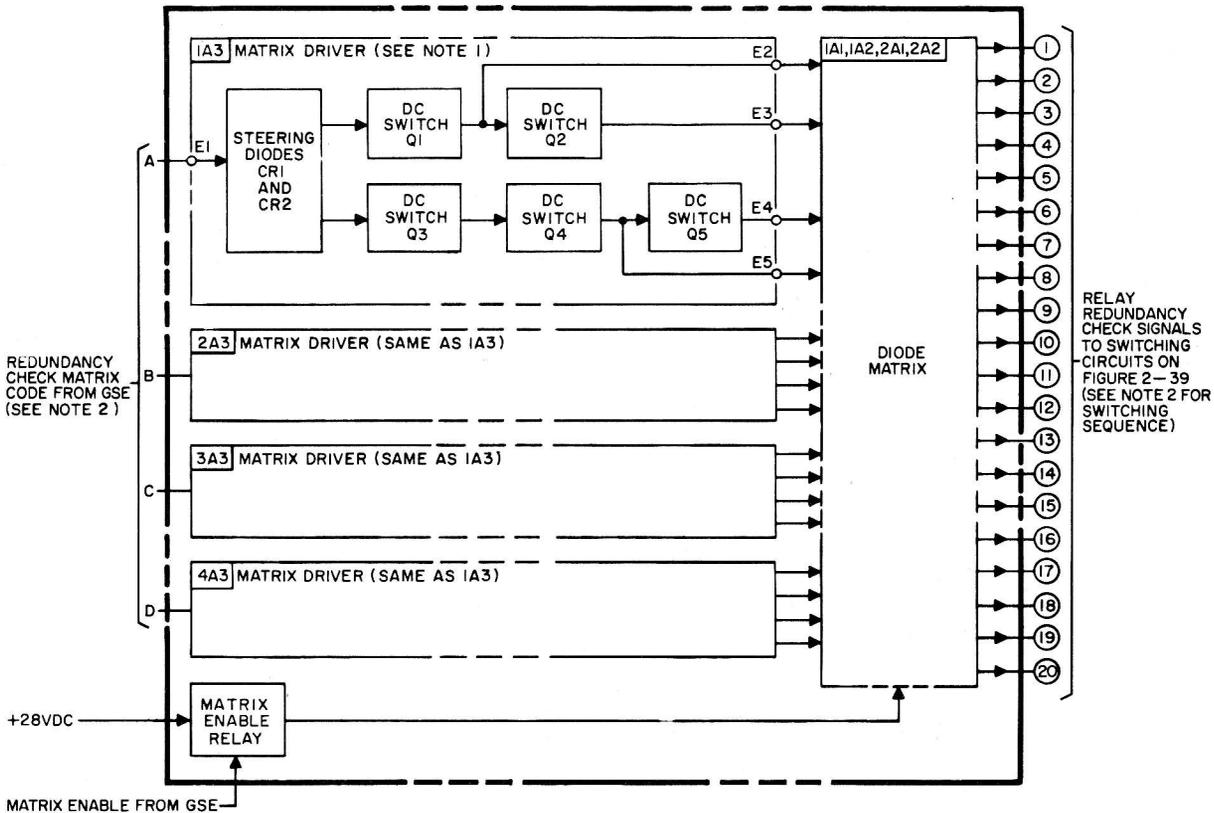


Figure 2-40. Redundancy Check Matrix



NOTES:
1. THE TABLE BELOW LISTS THE RESULTANT MATRIX DRIVER OUTPUTS WHEN INDICATED GSE SIGNAL IS APPLIED.

INPUT	OUTPUT				
E1	E2	E3	E4	E5	
0	+	-	-	+	
+(+28VDC)	-	+	-	+	
-(GRD)	+	-	+	-	

2. THE TABLE BELOW LISTS THE SWITCHING CIRCUIT RELAYS ENERGIZED UPON RECEIPT OF A SPECIFIC MATRIX CODE (MATRIX DRIVER INPUTS) FROM THE GSE.

MATRIX CODE (MATRIX DRIVER IN)				OUTPUT BUBBLE +28VDC	RELAYS ENERGIZED					
A	B	C	D		ASSY	K	K	K	K	K
0	0	0	+	1	A2	1	2	3		
0	0	+	0	2	A2	16				
0	0	+	+	3	A2	28	29	30	31	32
+	0	0	0	4	A2	37	38			
+	0	0	+	5	A2	43	44			
+	0	+	0	6	A3	19	20	21		
0	+	0	0	7	A2	20	21	22	23	
0	+	0	+	8	A3	10	11	12	13	
0	+	+	0	9	A2	13				
+	+	0	0	10	A2	7	8	9		
0	0	0	-	11	A2	14				
0	0	-	-	12	A2	10	11	12		
-	0	0	0	13	A2	15				
-	0	-	0	14	A3	14	15	16		
-	0	-	-	15	A2	24	25	26	27	
0	-	0	0	16	A2	39	40			
0	-	0	-	17	A3	22	23			
0	-	-	0	18	A2	41	42			
-	-	0	0	19	A2	33	34	35	36	
-	-	-	0	20	A2	4	5	6		

Figure 2-41. Redundancy Check Matrix, Functional Block Diagram

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

The table in Note 2 of Figure 2-41 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.

Mother Boards

The Flight Control Computer contains seven mother boards (mother boards 1 through 7). A photograph of a typical mother board is shown in Figure 2-42.

The mother boards consist of a printed circuit wiring board and a series of sequentially numbered connector jacks. The various Flight Control Computer subassemblies are plugged into the mother board jacks. The mother board then provides interconnection between these subassemblies.

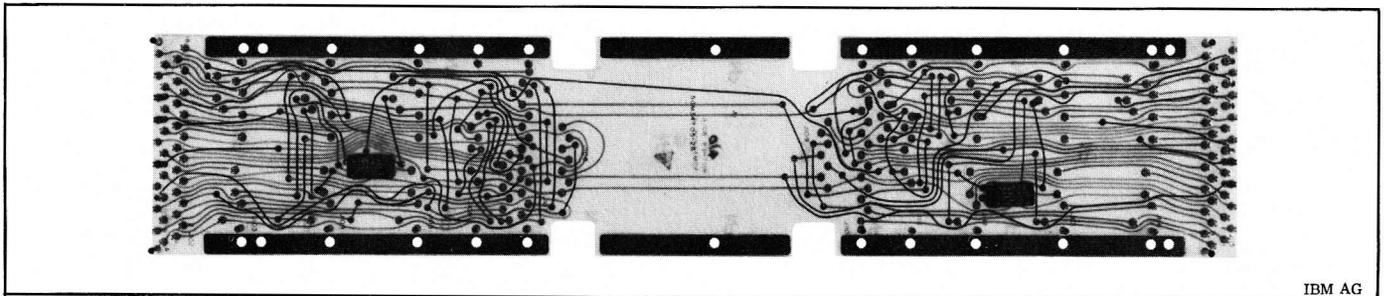


Figure 2-42. Mother Board (Typical)

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 manufacturer for repair.

IBM®

Space Systems Center
Huntsville, Alabama

Printed in USA

