NASA

ASTRIONICS SYSTEM HANDBOOK

SATURN LAUNCH VEHICLES

GEORGE C. MARSHALL SPACE FLIGHT CENTER
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
HUNTSVILLE, ALABAMA

VII. 1
X1.7

ASTRIONICS SYSTEM HANDBOOK CHANGE

The enclosed pages change, delete, or supplement the information in the Astrionics System Handbook (1 August 1965). Insert these pages and destroy the pages they replace.

When updating your handbook, retain the existing tabs except the tab for Chapter 11 which will be replaced. Replacement covers and binding posts are provided for the soft-bound editions.

REPLACEMENT INSTRUCTIONS

1. Remove and replace title page and preface

2. Remove Table of Contents and page I; replace Table of Contents

3. Remove and replace chapters 1, 2, 3, 4, and 5

4. Remove pages 6-i/6-ii and 6.1-1/6.1-2; replace page 6.1-1/6.1-2

5. Remove page 6.4-1/6.4-2 and replace with Section 6.4

6. Remove page 7-i/7-ii

7. Remove and replace pages 7.1-1 through 7.1-4

8. Remove and replace page 10-i/10-ii

9. Remove page 11-i/11-ii, page II, and tab for Chapter 11; add new Chapter 11 tab and text

10. Remove and replace Chapters 13 and 14

11. Remove page 15-i/15-ii

12. Remove and replace pages 15.1-1 and 15.1-2

13. Remove and replace pages 15.4-17 and 15.4-18

14. Remove pages 15.5-3/15.5-4 and 15.5-5/15.5-6 and replace with pages 15.5-3 and 15.5-4

15. Remove and replace pages 15.5-9 and 15.5-10

16. Add Section 15.6 to Chapter 15

17. Remove and replace pages A-1 through A-4 and B-1 through B-6

Questions concerning this change package may be directed to:

ASTRIONICS SYSTEM HANDBOOK

SATURN LAUNCH VEHICLES

1 August 1965

Changed 15 August 1966

PREFACE

This second revised edition of the Astrionics System Handbook has been developed under the direction and overall supervision of Dr. Rudolf Decher of the Astrionics Systems Engineering Office.

This description of the Saturn Astrionics System has been generated by personnel of the Astrionics Laboratory, the staff of the Astrionics Systems Engineering Office, and by personnel of the International Business Machines Corporation working under Contract NAS8-14000.

The handbook will be updated and expanded as it becomes necessary due to changes or refinements in the system concept and hardware. Sections not contained in the first release of this document will be made available within three months.

LUDIE G. RICHARD
Chief,
Systems Engineering Office
Astrionics Laboratory

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	15 Aug 1966
a	Original
*b	15 Aug 1966
*i thru vii	15 Aug 1966
viii Blank	Original
*I Deleted	15 Aug 1966
*1-i thru 1-ii Deleted	15 Aug 1966
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CHAPTER 1
INTRODUCTION

SECTION 1.1
PURPOSE OF DOCUMENT

The intent of this document is to provide a system description of the Astrionics System for the Saturn IB and V Launch Vehicles. It is not intended to go too deeply into any given subject but rather to give an overall picture of the Astrionics System from functional and operational viewpoints. General mission requirements and system capabilities are briefly discussed in Chapter 1, Introduction, to provide a total view of the Astrionics System. The subsequent chapters of Part I present a functional description of the various subsystems and the involved hardware. The operational phases of the Astrionics System, including pre-launch checkout, are discussed in Chapter 11 and indicate how the system will be used during a typical Saturn V Apollo mission. Astrionics hardware which performs several different functions (e. g., Launch Vehicle Digital Computer and Data Adapter) is described in Part II.

The Astrionics System description includes all of the electrical and electronic equipment on board the vehicle. It also includes the launch site electronic support equipment. However, this particular description does not cover the individual stage relay circuitry which controls certain stage functions. It does describe the signal flow through the system to the point of energizing this special circuitry so that an overall understanding of system operation is presented. Likewise, stage propellant utilization systems and internal engine sequencing systems are not covered; since, for the purpose of this description, they are considered a part of the propulsion system.

Since Astrionics Systems of Saturn IB and Saturn V are very similar, this document is devoted primarily to Saturn V. The areas in which the Saturn IB Astrionics System deviates from the Saturn V Astrionics System are listed in Section 1.4. Where applicable, these deviations are specified in the text of that particular chapter.

SECTION 1.2
SATURN LAUNCH VEHICLES

Figures 1.2-1 and 1.2-2 illustrate the Saturn IB Vehicle and Saturn V Vehicle, respectively, and include some characteristic vehicle data. Both vehicles have the same upper stage (the S-IVB Stage) which is propelled by a restartable engine to provide injection into escape trajectory from a parking orbit. The Instrument Unit is mounted on top of the S-IVB Stage and is very similar for Saturn IB and Saturn V.

The primary mission of Saturn IB is to serve as a launch vehicle for the Apollo Spacecraft earth orbital flight tests. These earth orbital flights will simulate certain phases of the lunar landing mission and will provide flight tests for the spacecraft and the S-IVB/IU Stage. Saturn V is the launch vehicle for the actual Apollo lunar landing missions. The typical profile of a lunar landing mission is described in Section 1.3.

The primary mission of the Saturn Vehicles is the successful accomplishment of the Apollo mission.

In addition, the Saturn Vehicles are capable of performing other types of missions which can be generally classified as the insertion of heavy payloads into earth orbits and escape trajectories. This may include:

- Transfer between earth orbits
- Rendezvous in earth orbit
- Direct ascent and injection into escape trajectory
- Injection into escape trajectory following extended earth-orbit phases
- Extended missions beyond injection

Saturn Vehicles are numbered consecutively, beginning with 201 for the first Saturn IB Flight Vehicle and 501 for the first Saturn V Flight Vehicle. The first few vehicles of each series are considered as research and development vehicles.

NOTICE

SATURN/APOLLO COORDINATE CHANGES

The XYZ coordinates used in this book have been changed to reflect the Project Apollo Coordinate System Standards (SE 008-001-1). Under these standards, the XYZ axes are defined as: the X axis is vertical through the launch site, the Z axis is parallel to the aiming azimuth, and the Y axis completes the right-handed system. Vehicle axes for roll, pitch and yaw are also defined as X, Y, and Z, respectively.

This coordinate scheme becomes effective for vehicles 206 and 502 (and subsequent). The conversion from the old system to the new is as follows:

OLD		NEW
X	to	Z
Y	to	X
Z	to	Y

A detailed explanation of the coordinate systems can be found in Chapter 2.

IBM B128
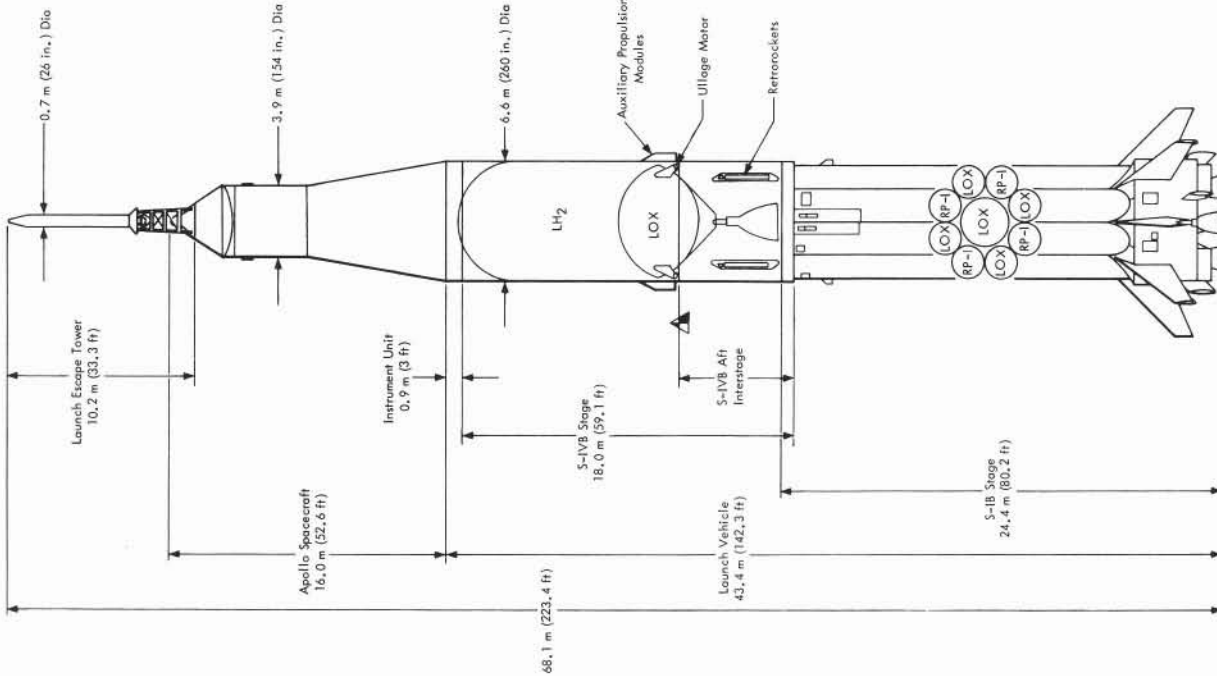
SATURN IB
- Overall Length . . . . : 68.1m (223.4 ft)

S-IB
- Dry Weight . . . . : 39,861 kg (87,800 lbs)
- Main Propulsion Engine. . : Rocketdyne H-1 (8)
- Total Nominal Thrust . . . : 7,116,000 N (1,600,000 lbs) (sea level)
- Propellants . . . . : LOX and RP-1
- Engine Cant Angle . . . . : Inboard (4) 3°
  Outboard (4) 6°
- Gimbal Pattern . . . . . . : ±8° Square Pattern (outboard engines only) (4)
- Retromotors . . . . : Thiokol TE-29-1B (4) ①
- Propellant . . . . : Solid
- Arrangement . . . . : Equally spaced circumferentially on aft interstage.

S-IVB
- Dry Weight . . . . : 10,805 kg (23,800 lbs)
- Main Propulsion Engine. . : Rocketdyne J-2 (1)
- Total Nominal Thrust . . . : 912,250 N (205,000 lbs) (vacuum)
- Propellants . . . . : LOX and LH₂
- Engine Cant Angle . . . . : None
- Gimbal Pattern . . . . . . : ±7° Square Pattern
- Attitude Control Engines : TAPCO (6)
- Nominal Thrust (Each) . . : 666 N (150 lbs) (vacuum)
- Propellants . . . . : Hypergolic (MMH & N₂O₄)
- Arrangement . . . . : Two auxiliary propulsion modules containing 3 engines each, located at positions I and III on aft skirt.
- Ullage Motors/Engines : Thiokol TX-280 (3)
- Propellant . . . . : Solid
- Arrangement . . . . : Equally spaced circumferentially on aft skirt.

INSTRUMENT UNIT
- Dry Weight . . . . : 1,884 kg (4,150 lbs)

Note:
① Retromotors are used for S-IB retrograde acceleration.
▲ Flight Separation

0.7 m (26 in.) Dia
3.9 m (154 in.) Dia
6.6 m (260 in.) Dia

Auxiliary Propulsion Modules
Ullage Motor
Retrorockets

LH₂
LOX

Launch Escape Tower
10.2 m (33.3 ft)
Instrument Unit
0.9 m (3 ft)
S-IVB Aft Interstage

Apollo Spacecraft
16.0 m (52.6 ft)
S-IVB Stage
18.0 m (59.1 ft)

Launch Vehicle
43.4 m (142.3 ft)
S-IB Stage
24.4 m (80.2 ft)

68.1 m (223.4 ft)

LOX RP-1
LOX RP-1
RP-1 LOX
LOX
LOX RP-1
RP-1 LOX

Figure 1.2-1 Saturn IB Launch Vehicle Characteristic Data

IBM B129

SATURN V

- Overall Length .... : 110.6m (362.9 ft)

S-IC

- Dry Weight .... : 136,200 Kg (300,000 lbs)
- Main Propulsion Engines : Rocketdyne F-1 (5)
- Total Nominal Thrust .. : 33,864,500 N (7,610,000 lbs)
- Propellants .... : LOX and RP-1
- Engine Cant Angle .... : None
- Gimbal Pattern .... : ±5° 9' Square Pattern (Outboard Engines) (4)
- Retromotors .... : 8
- Propellants .... : Solid
- Arrangement .... : Two in each F-1 engine fairing

S-II

- Dry Weight .... : 37,183 kg (81,900 lbs)
- Mainstage Propellant Capacity .... : 440,380 kg (970,000 lbs)
- Main Propulsion Engines : Rocketdyne J-2 (5)
- Total Nominal Thrust .. : 4,561,250 N (1,025,000 lbs)
- Propellants .... : LOX and LH$_2$
- Engine Cant Angle .... : None
- Gimbal Pattern .... : ±7° Square Pattern (Outboard Engines) (4)
- Ullage Motors/Engines. : RS-U-501 (8)
- Nominal Thrust (Each). : 101,900 N (22,900 lbs) (vacuum)
- Propellants .... : Solid
- Arrangement .... : Equally spaced around S-IC/S-II Interstage, in line with S-IC fins and positions I, II, III, & IV.

S-IVB

- Dry Weight .... : 12,440 kg (27,400 lbs)
- Mainstage Propellant Capacity .... : 104,420 kg (230,000 lbs)
- Main Propulsion Engine : Rocketdyne J-2 (1)
- Total Nominal Thrust . : 912,250 N (205,000 lbs)
- Propellants .... : LOX and LH$_2$
- Engine Cant Angle .. : None
- Gimbal Pattern ...... : ±7° Square Pattern
- Attitude Control Engines: TAPCO (6)
- Nominal Thrust (Each) : 670 N (150 lbs) (vacuum)
- Propellants .... : Hypergolic (MMH & N$_2$O$_4$)
- Arrangement .... : Two auxiliary propulsion modules containing 3 engines each, located at positions I and III on aft skirt.
- Ullage Motors/Engines : Thiokol TX-280 (2)
- Nominal Thrust (Each) : 15,560 N (3,500 lbs) (vacuum)
- Propellant .... : Solid
- Arrangement .... : Equally spaced circumferentially on aft skirt.
- Hydrogen Venting Ullage Engines......... : Gemini OAMS (2)
- Nominal Thrust (Each) : 320 N (72 lbs) (vacuum)
- Propellants .... : Hypergolic (MMH & N$_2$O$_4$)
- Arrangement .... : One in each of two auxiliary propulsion modules located at positions I and III on aft skirt. ①
- Retromotors .... : TE-29-1B (4) ①
- Propellants .... : Solid
- Arrangement .... : Positions I, II, III, and IV. S-II/S-IVB Interstage.

INSTRUMENT UNIT

- Dry Weight .... : 1,884 kg (4,150 lbs)

Note:

① Retromotors are used for S-II retrograde acceleration

▲ Flight Separation

0.7 m (26 in.) Dia

3.9 m (154 in.) Dia

6.6 m (260 in.) Dia

Auxiliary Propulsion Module
Attitude Control and Hydrogen Venting Ullage Engines

10.1 m (396 in.) Dia

10.1 m (396 in.) Dia

Launch Escape Tower 10.2 m (33.3 ft)

Apollo Spacecraft 16.0 m (52.6 ft)

Instrument Unit 0.9 m (3 ft)

S-IVB Stage 18.1 m (59.3 ft)

S-II Stage 24.8 m (81.5 ft)

Launch Vehicle 85.8 m (281.8 ft)

S-IC Stage 42.1 m (138 ft)

110.6 m (362.9 ft)

LH$_2$

LOX

LH$_2$

LOX

LOX

RP-1

Ullage Motor
Retromotor

Ullage Motor

Retromotor (Under Engine Fairing)

Figure 1.2-2 Saturn V Launch Vehicle Characteristic Data

SECTION 1.3
SATURN V/APOLLO MISSION PROFILE

The mission of the Apollo Project is to land 2 Astronauts on the moon and return the total crew of 3 Astronauts safely to earth.

The overall Apollo/Saturn Vehicle, composed of the Saturn V Launch Vehicle and the Apollo Spacecraft, is shown in Figure 1.2-2. The Saturn V Launch Vehicle consists of three propulsion stages and an Instrument Unit (IU). The IU contains the navigation, guidance, control, communication, and power supply equipment common to the main propulsion stages.

The profile of the Saturn V Launch Vehicle Mission is illustrated in Figure 1.3-1. Beginning with prelaunch checkout of the Astrionics System, the mission is divided into seven operational phases which are referenced to specific vehicle events. These phases and their approximate time durations are shown in Table 1.3-1. The following paragraphs describe these phases in detail.

Table 1. 3-1 Saturn V Operational Phases

Phase	Approximate Duration
Prelaunch	22 hours
S-IC Burn	150 seconds
S-II Burn	350 seconds
S-IVB First Burn	180 seconds
Orbital Coast	4.5 hours
S-IVB Second Burn	330 seconds
Translunar Coast	2 hours

PRELAUNCH AND LIFT-OFF

The prelaunch phase includes a complete Astrionics System checkout, prelaunch program and component activity (countdown), and the actual vehicle lift-off. This first phase begins weeks before launch and continues until the moment the vehicle leaves the launch pad.

During system checkout, all components are thoroughly tested by a complex of ground support equipment. The testing activities include stage and system checkout while the vehicle is still in the Vertical Assembly Building and final checkout on the launch pad. These test activities are sometimes referred to as "pre-rollout" (VAB) and "post-rollout" (pad) tests.

Preparation for launch is the next step in the prelaunch phase. During this operation, the Inertial Platform is aligned to the launch azimuth, the final launch and flight constants are loaded into the guidance computer, and all final preparations are completed.

The final step in the prelaunch phase is the countdown and actual lift-off. After ignition of the five F-1 engines, complete vehicle control is transferred from the ground to the IU. The vehicle is held down for approximately 6-9 seconds after ignition to allow the engines to develop sea-level thrust. The holddown arms then permit a "soft" release during the first 15 centimeters (6 inches) of vertical travel.

S-IC BURN

The second phase of the Astrionics System operation occurs during S-IC burn. During this phase, the vehicle lifts from the pad and is boosted on a precalculated path through the denser portion of the atmosphere. This burn time will normally continue for approximately 150 seconds and terminate when the fuel supply is depleted.

Boost navigation calculations are initiated in the final seconds before lift-off and continue until the vehicle reaches earth orbit. The boost navigation routine of the flight program calculates position, thrust acceleration, and velocity in a space-fixed, inertial coordinate system. These quantities provide

the necessary inputs to the guidance routines of the flight program.

Guidance of the vehicle throughout the first-stage burn is achieved through a preset time-tilt (pitch) program. Steering commands are generated as functions of time and thus guide the vehicle on a precalculated course towards the orbital objective. The vehicle is first "rolled" into correct alignment with the flight azimuth and then "tilted" into the correct flight attitude.

Attitude control during first-stage burn is accomplished by swivelling the four outer propulsion engines to change the direction of thrust and thereby change vehicle attitude. Thrust directions are oriented about the roll, pitch, and yaw axes of the vehicle. The rate at which the vehicle can change its attitude or direction is controlled by programming throughout the flight to prevent structural damage to the vehicle.

Vehicle events are initiated and monitored by the guidance computer in an operation called "sequencing". Generally, these sequence commands are used to initiate or terminate a particular vehicle operation (e.g., inboard engine cutoff or fire retrometers). A detailed flight sequence may be found in Section 4.3.

Final sequencing is often referred to as "staging" and is defined as the series of events leading to stage separation. The main staging events of the S-IC Stage are: outboard engine cutoff, inboard engine cutoff, ignition of the S-II ullage motors, firing of the separation device, and ignition of the S-IC retromotors. After separation, the S-IC Stage falls to earth in a ballistic flight path.

S-II BURN

S-II Stage ignition occurs almost simultaneously with separation of the S-IC Stage. This phase of the flight is designed to continue until fuel depletion (350 seconds after S-IC separation).

Near the beginning of this operational phase, the guidance system scheme changes from a standard time-tilt mode to a path adaptive mode. This new mode is designed to guide the vehicle towards its orbital goal with greater accuracy and fuel economy. Path adaptive guidance will be used for the remainder of the boost flight. Navigation and attitude control operations are a continuation of the scheme used in the S-IC phase.

The launch escape tower is jettisoned and second plane separation occurs early in this phase as part of the normal sequence. Fuel depletion sensors provide a start to staging operations. The five J-2 propulsion engines are shutdown simultaneously as the S-II/S-IVB Stages are prepared for staging. Basically, the staging sequence is as follows: sense low-fuel level, cutoff all 5 engines, fire the S-IVB ullage motors, fire the separation device, and fire the S-II retromotors. After separation, this stage also falls to earth in a ballistic flight path.

S-IVB FIRST BURN (ORBITAL INJECTION)

The first burn of the S-IVB Stage is designed to give the vehicle its final boost into earth orbit. The burn time of the S-IVB Stage's single J-2 engine is approximately 180 seconds. Engine cutoff will occur at the instant the vehicle reaches its predetermined velocity and position. After cutoff, the engine is prepared for a second burn period which will occur later in the flight.

Boost navigation calculations, used in the two previous stages, are continued throughout the S-IVB first burn. These calculated quantities provide the necessary inputs to the guidance computations which become more exacting as the vehicle approaches orbital injection and cutoff velocity. The path adaptive guidance scheme continues throughout this flight phase.

Attitude control for the S-IVB burn phase differs from the control of the other 2 stages because of the single-engine configuration. Since the single engine can only control pitch and yaw movements, an auxiliary steering system is utilized to control the roll functions. This system consists of ON-OFF thrust nozzles which are fix-mounted on the aft portion of the S-IVB structure. The lateral and opposing thrust of these nozzles positions the vehicle about its roll axis as commanded by the guidance and control systems.

Third-stage sequencing begins at S-II engine cutoff and continues for the remainder of the S-IVB/IU flight. Initial sequencing involves the preparation and ignition of the J-2 engine and jettisoning of the S-IVB ullage rockets. Engine cutoff at orbital insertion is commanded by the guidance system. Thereafter, the engine is purged for several minutes in preparation for a later restart when lunar transfer injection is commanded.

JETTISON S-II
AFT INTERSTAGE

S-II STAGE CUTOFF &
JETTISON — S-IV B
STAGE IGNITION

INSERTION INTO
PARKING ORBIT &
S-IVB ENGINE CUTOFF

COAST

S-IVB RE-IGNITION
④

S-II IGNITION &
LAUNCH ESCAPE
SYSTEM JETTISON

185 KILOMETERS (100 NAUT. MILES)

S-IVB ENGINE
CUTOFF
⑤

BEGINNING OF
TRANSPOSITION
MANEUVER

S-IC STAGE CUTOFF
& JETTISON
①

CM/SM
TURN AROUND

DOCKING

S-IC STAGE IGNITION
& VEHICLE LAUNCH

S-IV B SEPARATION

SATURN V/APOLLO

TO MOON

IBM B130

Figure 1. 3-1 Saturn V/Apollo Mission Profile

ORBITAL COAST

The orbital coast phase of the Saturn Mission follows cutoff of the S-IVB engine. The vehicle, which is now reduced to the S-IVB/IU Stage and Apollo Spacecraft, will orbit the earth at an approximate altitude of 185 kilometers (100 nautical miles).

The coast phase is scheduled to continue for a maximum of 3 orbits. During this flight period, the flight program/Astrionics System performs seven major functions: navigation, guidance, attitude control, event sequencing, data management (telemetry), ground command processing, and equipment evaluation and checkout. When all systems are checked and determined ready and the vehicle is positioned in the lunar insertion window, the S-IVB engine is restarted for injection into the translunar trajectory.

Orbital navigation includes computations to solve for position, and velocity based on insertion conditions. The computations are carried out by utilizing mathematical models of the earth and its atmosphere to approximate the effects of the physical world on the vehicle. The orbital guidance function is computed by using navigational inputs, time, and vehicle attitude. Generally, the vehicle attitude will be maintained at 90 degrees to the local vertical. Since the J-2 propulsion engine is cutoff, the auxiliary steering system is used for all attitude maneuvers.

Sequencing in coast flight is primarily involved in purging the J-2 engine after cutoff and then preparing it for a later restart.

S-IVB SECOND BURN (TRANSLUNAR INJECTION)

The second-burn flight of the S-IVB Stage is used to "break away" from earth orbit and thus inject the launch vehicle/spacecraft into a translunar trajectory. During this burn period, the velocity is considerably increased (approximately 40 percent) to enable the vehicle to escape the earth's gravitational pull.

The Navigation and Guidance Systems will make the final computations and course corrections needed to place the spacecraft into orbit around the moon. Shutdown of the J-2 engine will occur when the correct velocity has been achieved.

TRANSLUNAR COAST

The final operational phase for the Saturn Launch Vehicle is the translunar coast phase. During this phase, the primary function of the vehicle is to maintain a steady attitude while transposition and docking maneuvers are performed. These maneuvers consist of:

- Jettisoning the forward section of the IU/Apollo interstage adapter

- CSM separation from the LEM/IU/S-IVB Configuration and CSM turn around

- CSM docking to the LEM/IU/S-IVB configuration.

Soon after this point in time, the mission of the Saturn Launch Vehicle is considered completed, and the S-IVB/IU Stage is separated from the LEM/CSM configuration.

SECTION 1.4

ASTRIONICS SYSTEM

The overall Astrionics Systems of the Saturn IB and V Launch Vehicles are shown in the simplified block diagrams, Figures 1.4-1 and 1.4-2, respectively. The major portion of the Astrionics equipment is located in the IU, which is mounted on top of the S-IVB Stage. During flight, the Astrionics System performs, or is involved in, the following main functions:

- Navigation, guidance, and control of the vehicle

- Measurement of vehicle parameters

- Data transmission between vehicle and ground stations (up and down)

- Tracking of the launch vehicle

- Checkout and monitoring of vehicle functions in orbit

- Detection of emergency situations

- Generation of electrical power for system operation

The operational lifetime of the S-IVB/IU Astrionics System is 4-1/2 hours for Saturn IB and 7 hours for Saturn V. The operational lifetime is limited only by the capacity of the power supply (batteries) and the water supply of the environmental control system which is sufficient to complete the presently defined launch vehicle missions. With increased power and water supply capacity, the operational lifetime of the Astrionics System can be extended for longer duration missions if required.

NAVIGATION, GUIDANCE, AND CONTROL

The Saturn Astrionics System provides navigation, guidance, and control of the vehicle from launch until separation of the S-IVB/IU from the spacecraft. The equipment involved in these functions are the ST-124-M Inertial Platform Assembly, the Launch Vehicle Digital Computer and Launch Vehicle Data Adapter, the Flight Control Computer, the Rate Gyros,

and finally the propulsion engine actuators and the auxiliary propulsion system. The Saturn inertial navigation and guidance system can be updated by data transmission from ground stations through the IU command system. The Inertial Platform Assembly carries three integrating accelerometers which measure the thrust acceleration in a space-fixed reference frame. In addition, the platform gimbal angles indicate the attitude of the vehicle in the platform reference frame. The LVDA serves as the input/output device for the LVDC and also performs the necessary data processing.

The LVDC performs computations for navigation, guidance, and control functions. The position and velocity of the vehicle is obtained by combining accelerometer measurements with computed gravitational acceleration. This information is the input to the guidance computations which determine the required thrust vector orientation and engine cutoff time according to the guidance scheme stored in the memory of the LVDC.

Attitude control during powered flight is accomplished through swivelling of propulsion engines by means of hydraulic actuators to obtain the proper thrust vector orientation. The actuator commands are generated in the Flight Control Computer. The Flight Control Computer combines attitude error signals from the LVDA and angular rate signals from Rate Gyros to provide stable attitude control of the vehicle. The attitude error signal is generated in the LVDC by comparing the required thrust vector orientation (from guidance computations) with the actual vehicle attitude (obtained from platform gimbal angles).

During coast flight periods, attitude control is achieved by the auxiliary propulsion system. This system consists of 6 nozzles which are arranged in 2 modules and mounted on the aft end of the S-IVB Stage. The auxiliary propulsion system is also controlled by the Flight Control Computer located in the IU.
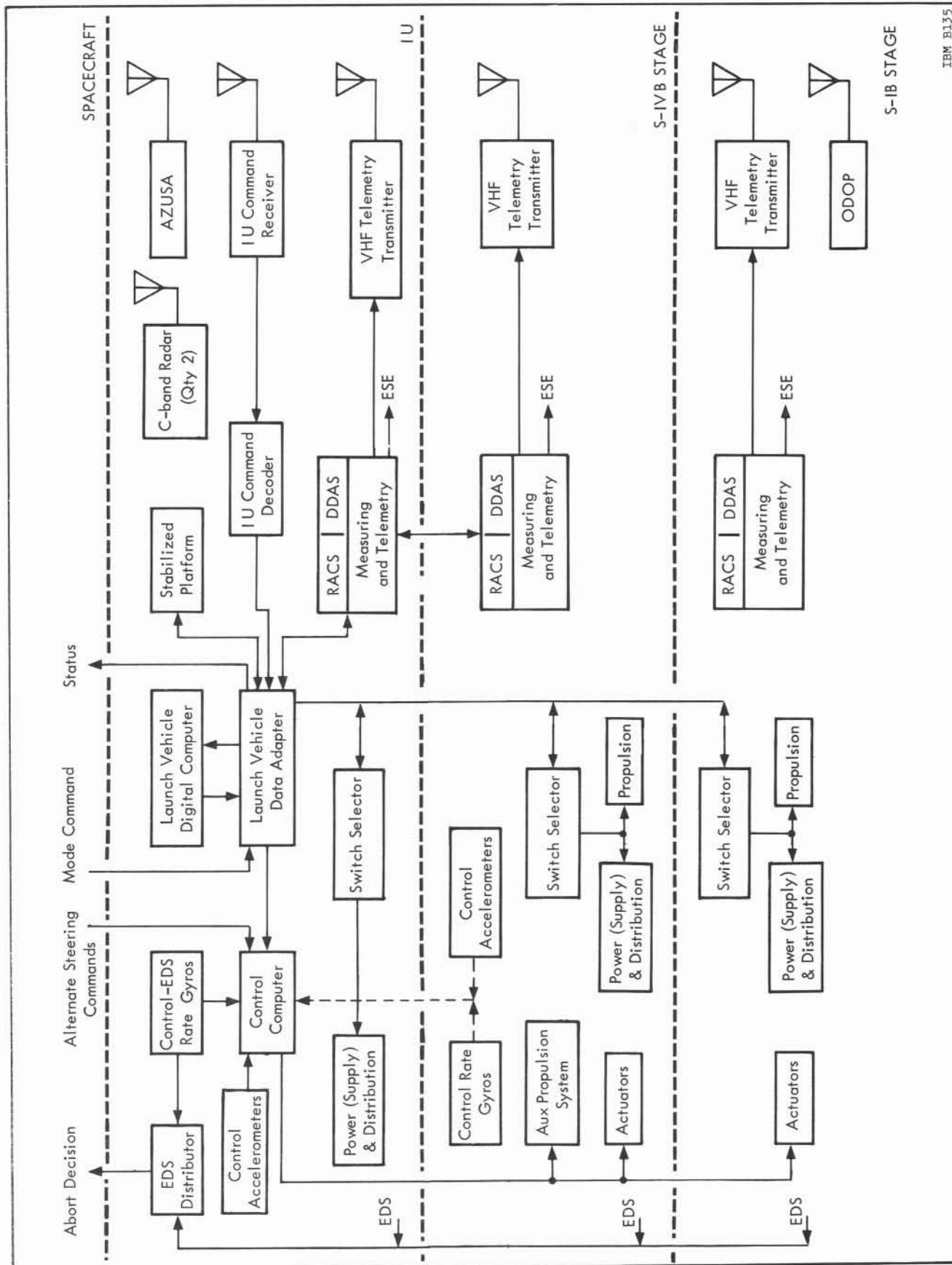
Figure 1. 4-1 Saturn IB Astrionics System (Operational Vehicle)
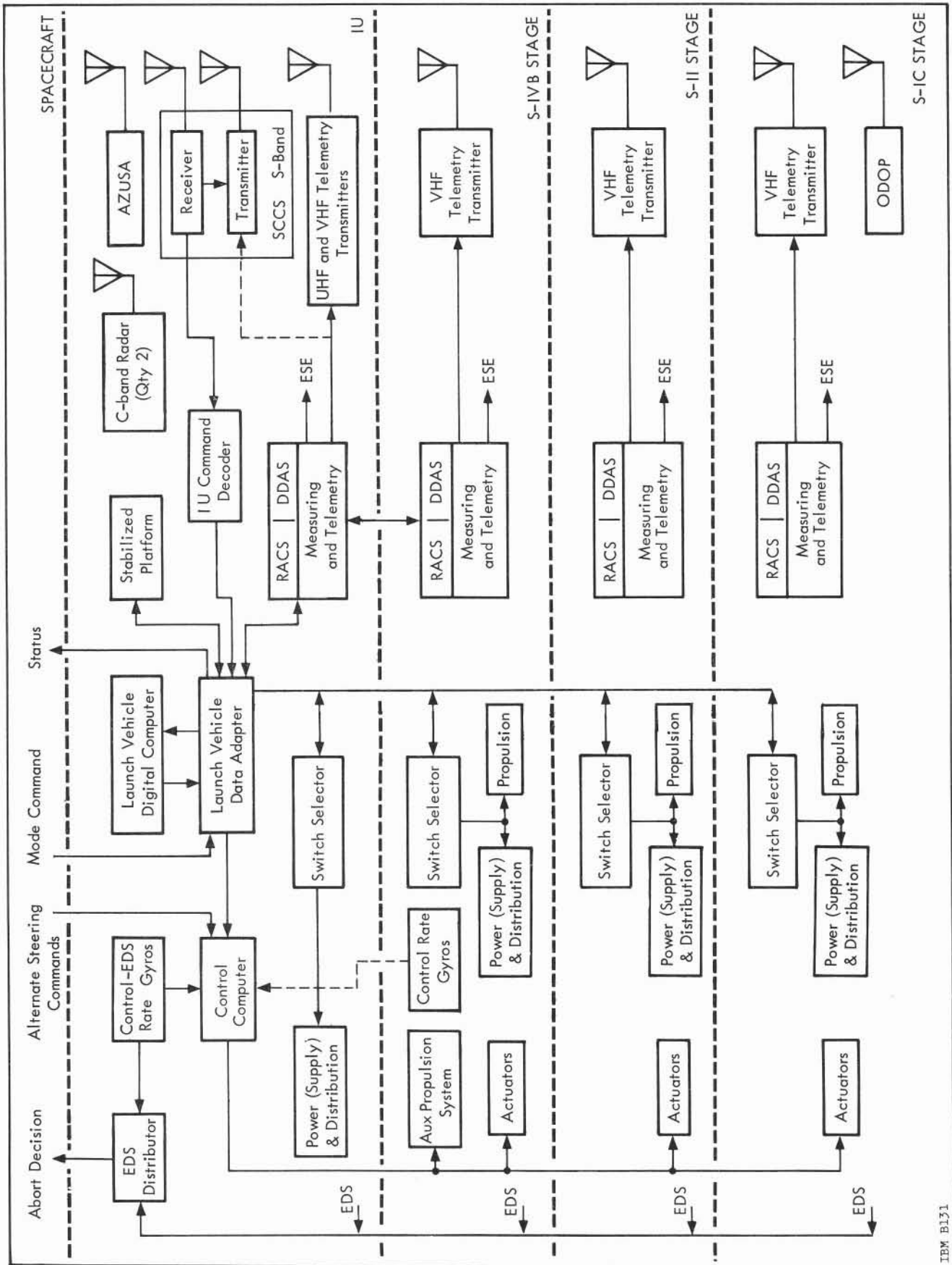
Figure 1.4-2 Saturn V Astrionics System (Operational Vehicle)

IBM B131

Flight sequence control (e. g. , vehicle staging, engine ignition and cutoff) is performed by the LVDC. The flight program, stored in the LVDC memory, generates the necessary flight sequence commands which are transmitted through the LVDA and Switch Selector to the proper circuit in the particular vehicle stage.

MEASUREMENTS AND DATA TRANSMISSION

Each vehicle stage is equipped with a measuring and telemetry system, including RF transmitter and antennas. For efficient utilization of available bandwidth and to obtain the required accuracy, three different modulation techniques are used in each stage telemetry system. These three are: frequency modulation/frequency modulation (FM/FM), pulse code modulation/frequency modulation (PCM/FM), and single sideband/frequency modulation (SS/FM) which is employed in research and development only.

In Saturn IB vehicles, telemetry data is radiated from the vehicle to ground stations in the VHF band (225-260 MHz). The PCM/FM system of the S-IVB Stage and the IU are interconnected to provide a redundant transmission path and to make S-IVB measurements available to the LVDA. All flight control data is transmitted through the PCM/FM system.

In Saturn V Vehicles, the PCM/FM telemetry data of the S-IVB and IU is transmitted in VHF band (225-260 MHz) and in the UHF band (2200-2300 MHz). The UHF-band transmission is provided primarily for transmission over the longer ranges after the vehicle has left the parking orbit. In addition, the PCM/FM data can be transmitted through the communication and command system transponder. This arrangement provides high reliability through redundancy in transmission paths.

The telemetry system of each stage has a separate output via coaxial cable to the electronic support equipment, which is used with the digital data acquisition system for vehicle checkout before launch.

The Instrument Unit command system permits data transmission from ground stations to the IU for insertion into the LVDC.

TRACKING

The Saturn Vehicles carry several tracking transponders. The ODOP Transponder is located in the first stage of Saturn IB and V Launch Vehicles. The Instrument Unit is equipped with two C-band Radar Transponders, an AZUSA Transponder, and the CCS Transponder (S-band tracking).

EMERGENCY DETECTION SYSTEM

The emergency detection system collects special measurements from each stage of the launch vehicle. Based on these measurements, critical states of the vehicle which may require mission abort are detected, and the information is sent to the Spacecraft for display and/or initiation of automatic abort.

SPACECRAFT INTERFACE

Several lines cross the IU/spacecraft interface for exchange of signals. Alternate steering commands from the spacecraft navigation and guidance system may be used to control the launch vehicle during S-II and S-IVB powered flight phases. This type of operation is considered as backup in case of a failure of the IU navigation and guidance system. During coast flight, the Astronaut may control the attitude of the vehicle through manually generated commands. In any case, a mode command must be sent first from the spacecraft to the LVDA to perform the necessary switching before the IU Flight Control Computer can accept the steering signals from the spacecraft. To indicate the state of the launch vehicle, certain measurements are sent to the spacecraft and displayed to the Astronaut.

Before launch, automatic checkout of the vehicle system is controlled by the launch computer complex and the electronic support equipment. This system also includes the digital data acquisition system.

SECTION 1.5
SYSTEM RELIABILITY CONSIDERATIONS

From the overall reliability goal of 0.85 assigned to the Saturn Launch Vehicles, a goal of 0.992 was apportioned to the Instrument Unit which contains the majority of the Astrionics System identified in Section 1.4. This goal is applied only to the components required to accomplish and support the guidance, control, and sequencing functions during the Saturn IB earth orbital mission (4.7 hours) and the flight portion of the Saturn V lunar mission (6.8 hours).

The Instrument Unit design concept requires a digital/inertial guidance scheme and an attitude error/attitude rate control scheme. A non-redundant system implementation of these design concepts would have a reliability of approximately 0.98 which is not commensurate with the 0.992 goal. In order to achieve reliability which is commensurate with the IU goal, six forms of redundancy are utilized within the IU:

- Triple modular redundancy with majority voting

- Dual redundancy

- Pair and spare redundancy

- Quad redundancy

- Backup redundancy at the functional level

- LVDC program backup redundancy

TRIPLE MODULAR REDUNDANCY

Triple Modular Redundancy (TMR) is employed extensively throughout the Launch Vehicle Digital Computer (LVDC) and Launch Vehicle Data Adapter (LVDA). The binary logic required to perform a given function is triplicated, and the outputs of the 3 channels thus formed are fed to a majority voter. The output of the voter coincides with the majority of the inputs to the voter (i.e., three logical "1's" or two "1's" and a "0" produce a "1" output; two "0's" and a "1" or three "0's" produce a "0" output). The effects of all single component part failures are negated by this scheme and significant reliability enhancement results.

DUAL REDUNDANCY

The LVDC and LVDA utilize dual redundancy in the computer memory and power supply areas. Normally, the output from one of the dual memories (memory A) is utilized. However, if an error is detected in memory A (by either parity checking or half-select current monitoring), output is switched to memory B.

The power supplies which furnish excitation for the LVDC and LVDA circuits are designed so that the predominant mode of failure is no output. Two supplies, each capable of supplying the required current, are actively "OR'ed" by diodes. Thus, failure of any one supply can be tolerated.

PAIR AND SPARE REDUNDANCY

"Pair and Spare" is the term applied to the IU control system redundancy scheme. This circuitry, which is used in the Control Signal Processor and Flight Control Computer, processes slowly varying dc voltage levels. Once again, the required circuitry is triplicated. The three channels are the "active" or "command" channel, the "standby" or "spare" channel, and the "reference" channel. Output is normally from the active channel. A differential amplifier is used to compare the active channel output with the reference channel output. If a difference in level occurs, output is switched to the standby channel. All single failures are detected and negated by this scheme.

A form of "pair and spare" redundancy is also utilized in the LVDA ladder output circuitry. As in the control system, an active channel is compared with a reference channel. A disagreement initiates a switchover which allows a standby channel to provide the output to the Flight Control Computer. Some of the hardware in the LVDA ladder circuitry is utilized for all three vehicle axes by a time sharing arrangement.

QUAD REDUNDANCY

Quad redundancy is employed in the Mod II Switch Selector. Individual transistors are replaced

by 4 transistors in a series-parallel configuration. Similarly, single diodes and resistors are replaced by groups of four.

BACKUP REDUNDANCY

The Emergency Detection System (EDS) constitutes backup redundancy at the functional level. The EDS cannot negate the mission effects of the failures which it detects. It does, however, provide for the crew's safe departure from a defunct vehicle. Thus, while the probability of the IU accomplishing the translunar injection is in the order of .99, the probability of not injuring the crew (in the case of a launch vehicle failure) is increased to 0.9999 by utilization of the EDS.

PROGRAM BACKUP REDUNDANCY

The logical decision-making capability of the LVDA/LVDC is used to achieve backup redundancy for various components.

The LVDC flight program compares the digital word present in the Switch Selector register with the word issued by the LVDA. If a disagreement is detected, the LVDA issues the binary complement of the original digital word to the Switch Selector. This complement word causes the Switch Selector to issue the same output that the original word would have generated. The effects of single failures in the Switch Selector register are negated by this scheme.

The LVDC flight program performs a reasonableness test on the inputs from the gimbal resolvers of the ST-124M Inertial Platform Assembly. An unreasonable input will cause switchover from the fine mode of resolver operation to the coarse mode (which is sufficient to allow mission success). If both fine and coarse resolver inputs are unreasonable, the LVDC/LVDA will cause the Launch Guidance Failure indication in the spacecraft to be affirmative. This failure indication is an EDS manual abort parameter. As opposed to mission success redundancy, this function of the LVDC flight program constitutes an EDS type of redundancy.

The flight program evaluates the Platform system inertial velocity outputs in a manner similar to that described for the gimbal resolvers. For each inertial axis, the velocity encoding and processing hardware has dual redundancy. Reasonableness tests are performed on the output of the initially selected channel. If an unreasonableness is detected, the program switches to the redundant channel. If the redundant channel is reasonable, its output will be utilized in the normal fashion. An unreasonable decision on the redundant channel will cause the LVDC flight program to use a modified navigation scheme and initiate the Launch Guidance Failure indication previously described. This arrangement provides mission redundancy for failures of the optisyns, Accelerometer Signal Conditioner, and associated LVDA hardware. EDS-type redundancy is provided for double failures of this hardware and single failures within the Inertial Platform accelerometers and associated servo loops.

CHAPTER 2
NAVIGATION AND GUIDANCE

SECTION 2.1

THE NAVIGATION, GUIDANCE, AND CONTROL SYSTEM

The problem of directing a ballistic missile or space vehicle, to accomplish a given mission, is customarily discussed in terms of three separate functions: navigation, guidance, and control. The boundaries between these 3 areas are to some extent arbitrary and conventional. The 3 terms, navigation, guidance, and control, will be used in this text according to the following definitions:

- Navigation is the determination of position and velocity of the vehicle from measurements made onboard the vehicle.

- Guidance is the computation of maneuvers necessary to achieve the desired end conditions of a trajectory (e.g., an insertion into orbit).

- Control is the execution of the maneuver (determined from the guidance scheme) by controlling the proper hardware.

A block diagram of the overall Saturn V navigation, guidance, and control system is shown in Figure 2.1-1. (This figure is also true for the Saturn IB Vehicle if the S-II Stage Switch Selector and engine actuator blocks are omitted.) The 3-gimbal stabilized platform (ST124-M) provides a space-fixed coordinate reference frame for attitude control and for navigation (acceleration) measurements. Three integrating accelerometers, mounted on the gyro-stabilized inner gimbal of the platform, measure the 3 components of velocity resulting from vehicle propulsion. The accelerometer measurements are sent through the LVDA to the LVDC. In the computer, the accelerometer measurements are combined with the computed gravitational acceleration to obtain velocity and position of the vehicle.

The LVDA is the input/output device for the LVDC. It performs the necessary processing of signals, from different sources, to make these signals acceptable to the computer.

According to the guidance scheme (programmed into the computer), the maneuver required to achieve the desired end conditions is determined by the LVDC. The instantaneous position and velocity of the vehicle are used as inputs. The result is the required thrust direction (guidance command) and the time of engine cutoff.

Guidance information stored in the LVDC (e.g., position, velocity) can be updated through the IU command system by data transmission from ground stations. The IU command system provides the general capability of changing or inserting information in the LVDC.

Control of the launch vehicle can be divided into attitude control and discrete control functions. For attitude control, the instantaneous attitude of the vehicle is compared with the desired vehicle attitude (computed according to the guidance scheme). This comparison is performed in the LVDC. Attitude correction signals are derived from the difference between the existing attitude angles (gimbal angles) and the desired attitude angles. In the Flight Control Computer, these attitude correction signals are combined with signals from control sensors to generate the control command for the engine actuators. The required thrust direction is obtained by swivelling the engines in the propelling stage and thus changing the thrust direction of the vehicle. Since the S-IVB Stage has only 1 engine, an auxiliary propulsion system is used for roll control during powered flight. The auxiliary propulsion system provides complete attitude control during coast flight of the S-IVB/IU Stage.

Commands for flight sequence control are generated in the LVDC according to a stored program. These commands are transferred through the LVDA to the Switch Selector of the corresponding vehicle stage. Examples of flight sequence control are engine ignition, cutoff, and stage separation. The Switch Selector in the addressed stage activates the necessary circuit to perform the commanded function.

Attitude and sequence control of the launch vehi-
cle is described in Chapters 3 and 4, respectively.
The Inertial Platform, LVDA and LVDC, which are
involved in navigation, guidance, and control opera-
tions, are described in Chapters 14 and 15.
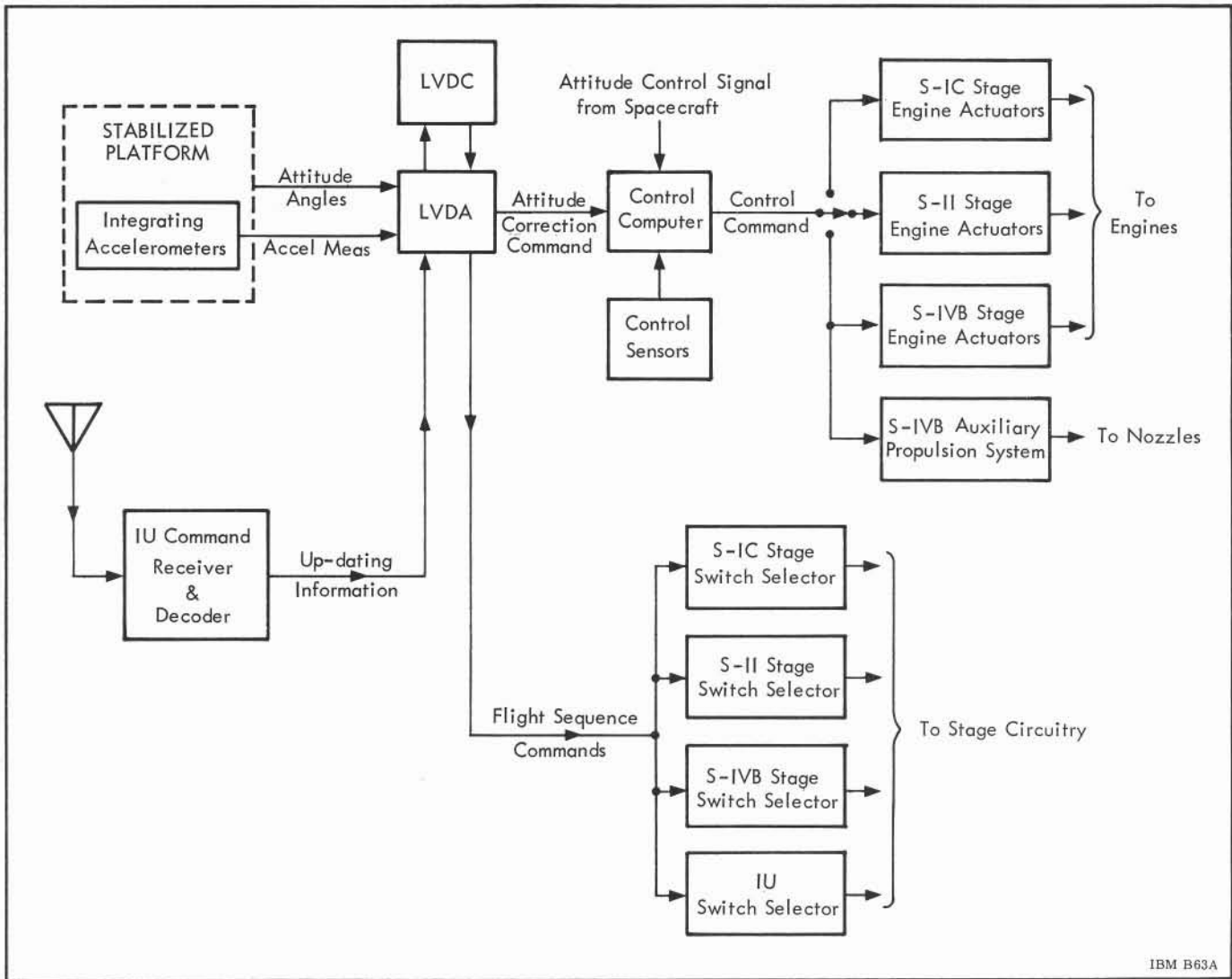
Figure 2.1-1 Block Diagram of Saturn V Navigation, Guidance, and Control System

SECTION 2.2
NAVIGATION

2.2.1 GENERAL DESCRIPTION

During powered flight, the Saturn IB/V Navigation System determines vehicle position, velocity, and acceleration from measurements made on board the vehicle. The initial measurements are supplied by integrating accelerometers which are mounted on an Inertial Platform within the ST-124M-3 Inertial Platform Assembly*.

Three coordinate reference systems (three-dimensional) are associated with the Saturn IB/V Navigation System. Using these coordinate systems and the accelerometer measurements, the LVDC is programmed to compute the instantaneous vehicle velocity and position. These values are presented to the guidance system in terms of X Y Z coordinates. **

During coast flight (orbital flight), vehicle position, velocity, and acceleration are computed from equations of motion which are based on the flight conditions at the time of orbital insertion. During this phase of the flight, data for navigation and guidance calculations can be updated from the ground stations via a radio command system.

2.2.2 NAVIGATION COORDINATE REFERENCE SYSTEMS

References for vehicle origin, position, and destination must be defined before a navigation scheme can function. There are three primary coordinate systems used for Saturn Navigation:

- The gravitational coordinate system
- The navigational or Platform coordinate system ($X_S Y_S Z_S$)
- The inertial coordinate system ($X_I Y_I Z_I$)

The references for these coordinate systems are either earth fixed or space fixed. An earth-fixed reference vector will exist in a line from the earth's

center through a point on the earth's surface (e. g., launch site). As the earth rotates, the vector will remain oriented between these 2 points and is therefore earth fixed. A space-fixed vector reference will be aimed at a point in space and is completely independent of the earth's revolution. After a quarter revolution of the earth, the earth-fixed vector will have rotated 90 degrees while the space-fixed vector has remained pointed in its original direction.

GRAVITATIONAL COORDINATE SYSTEM

The gravitational (U V W) system is a non-rotating, rectangular coordinate system whose origin is at the center of the earth. See Figure 2.2-1. It is used in computing the effect that gravity has upon the Platform-mounted accelerometers. The W axis (not shown in the figure) is aligned with the earth's spin axis and points south. The V axis is in the equatorial plane and intersects the equator at the longitude of the launch site at launch time. The U axis which is perpendicular to the W and V axes completes the right-handed orthogonal system.

NAVIGATIONAL OR PLATFORM COORDINATE SYSTEM

A navigational ($X_S Y_S Z_S$) coordinate system is used for navigation and guidance computations. Its origin is also at the center of the earth. See Figure 2.2-1. The system is earth fixed until Platform release (approximately 8 to 10 seconds before launch). At that time, the X_S axis is established, and the system becomes permanently space fixed.

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* For the purpose of brevity, the ST-124M-3 Inertial Platform Assembly will hereinafter be referred to as the Platform.

** The XYZ coordinates used in this book have been changed to reflect the Project Apollo Coordinate System Standards (SE 008-001-1). This coordinate scheme becomes effective for vehicles 206 and 502 (and subsequent). For further details refer to Section 1.2.
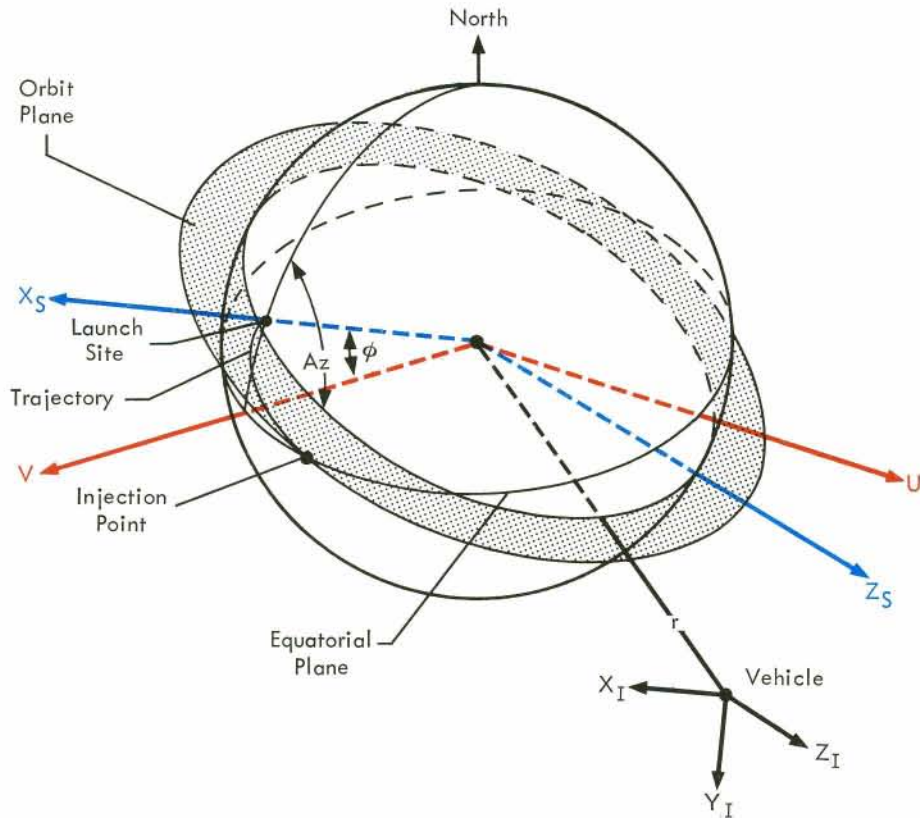
Note: Y and W coordinates have been omitted for clarity.

UVW   Gravitational Coordinate System—A non-rotating, rectangular coordinate system. The W axis is collinear with the earth's spin axis and points south. The V axis is perpendicular to the W axis and oriented so that the VW plane contains the launch site at launch time. The U axis completes a right-hand orthogonal system.

$X_S Y_S Z_S$ Navigational Coordinate System—A non-rotating, rectangular coordinate system. The X_S axis is parallel to the gravity vector at the launch site. The Z_S axis is parallel to the earth-fixed firing azimuth and is positive downrange. The Y_S axis completes a right-hand orthogonal system.

$X_I Y_I Z_I$ Inertial Coordinate System—A coordinate system which is parallel to the $X_S Y_S Z_S$ system and with its origin at the vehicle's Platform—sometimes written as $X_M Y_M Z_M$.

ϕ Geodetic latitude of the launch site.

Az   Launch azimuth, measured in clockwise direction from north.

r   Radius vector from the earth's center to the vehicle.

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Figure 2.2-1 Navigation Coordinate Systems

INERTIAL COORDINATE SYSTEM

The inertial (X_I Y_I Z_I) coordinate system is used for velocity and acceleration measurements. Unlike the first two systems, the X_I Y_I Z_I system has its point of origin at the Platform, see Figure 2.2-1. The X_I Y_I Z_I coordinates are parallel with the X_S Y_S Z_S coordinates. Thus, any increment of acceleration measured at the Platform is an indication of motion along the X_S Y_S Z_S axes. Before launch, both the navigation and accelerometer coordinate systems are earth fixed, but at the moment of Platform release (approximately 8 to 10 seconds before launch), both systems become space fixed. This system is sometimes referred to as the X_M Y_M Z_M system.

2.2.3 NAVIGATION CALCULATIONS

BOOST NAVIGATION

The acceleration values, measured by the Platform accelerometers, are processed within the LVDA and forwarded to the LVDC as increments of velocity. Velocity is integrated to determine position. The basic navigation scheme is shown in Figure 2.2-2.

Additional forces, such as gravity and initial velocity, must be considered, and the effect induced by these forces must be compensated for within the navigational computations. The gravitational acceleration, which is acting upon the accelerometers, is derived from the gravitational potential of the earth. This gravitational acceleration varies in relation to position and distance from the earth. A nominal value is 32.2 feet/second/second (32.2 ft/s^2) at the earth's surface and will decrease as vehicle altitude increases.
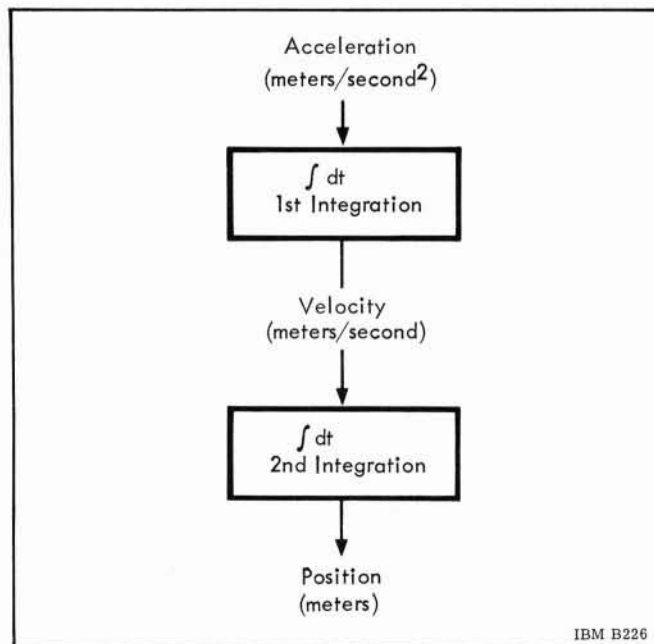
The quantities and equations required for boost navigational computations are listed in Table 2.2-1. A definition of the terms used in these and other navigation equations may be found in the List of Abbreviations, Signs, and Symbols at the rear of this book.

Figure 2.2-3 is a detailed flow diagram of the navigation scheme. Starting at the upper left of the figure, accelerometer measurements (integrated into velocity values) are obtained from the Platform via the LVDA. Within the LVDC, initial velocity (imparted by the spinning earth), gravitational velocity (caused by the effects of gravity), and the Platform velocity are algebraically summed. Their sum is inertial vehicle velocity in the space-fixed X_S Y_S Z_S coordinate system. This velocity is integrated and added to the initial position (distance from earth's center to the launch site) to determine vehicle position. The value of position is used in guidance computations and to compute the gravitational velocity which is "looped back" to the accelerometer readings. Values of Platform velocities are differentiated to obtain components of acceleration.

ORBITAL NAVIGATION

During orbital coast flight, the navigational program periodically computes the vehicle position, velocity, and acceleration from equations of motion which are based on vehicle conditions at the time of orbital insertion. In orbit, navigation and guidance information in the LVDC can be updated by digital data transmission through the radio command system.

Orbital navigation includes the computations necessary to determine position, velocity, and total acceleration of the S-IVB and IU relative to the navigation coordinate system during earth orbit. These computations are carried out in an indirect fashion, making use of mathematical models of the earth, its atmosphere, and the vehicle to approximate the effects of the physical world on the vehicle. See Table 2.2-1.

Additional navigational computations are used in maintaining vehicle attitude during orbit. These computations establish a local vertical which is used as a reference for attitude control. Normally, the attitude of the vehicle roll axis will be maintained at 90 degrees with respect to this local vertical.

Figure 2.2-2 Basic Navigation Scheme

Table 2.2-1 Navigation Computations

Boost Navigation

The boost navigation routine combines calculated gravitational accelerations with measured Platform velocities to compute position and total velocity in an earth centered inertial coordinate frame.

The gravitational acceleration components, $\ddot{X}_g$, $\ddot{Y}_g$, and $\ddot{Z}_g$ are computed during boost as:

$$\ddot{X}_g = X_S \, S - P \sin \phi$$

$$\ddot{Y}_g = Y_S \, S + P \cos \phi \sin Az$$

$$\ddot{Z}_g = Z_S \, S - P \cos \phi \cos Az,$$

where S and P are computed from equations which characterize the gravitational model and include values of GM (universal gravitational constant), a_e (equatorial radius of the earth), ϕ (geodetic launch site latitude), Az (firing azimuth), and established constants. These accelerations are then used to compute a change in position and velocity due only to the gravitational force. The gravitational position change is combined with the updated change in position due to thrust force to arrive at a new inertial position. The velocity change is added to the accumulated platform measurements to arrive at the current inertial velocity. The state quantities thus calculated provide the necessary navigational input to the guidance routines.

The computations performed are:

Change in position from gravitation

$$\Delta X_g = [\, I_{gX} + \ddot{X}_g \, \Delta t/2] \, \Delta t$$

Change in position from acceleration

$$\Delta X_T = [\, \dot{X}_M + \dot{X}_{M \, Past}] \, \Delta t/2$$

New space-fixed position

$$X_S = X_S + \Delta X_g + \Delta X_T + \dot{X}_o \, \Delta t$$

Gravitation computations

$$\ddot{X}_g = g \, (X_S, \, Y_S, \, Z_S)$$

Gravitational velocity

$$I_{gX} = I_{gX} + \Delta t/2 \, (\ddot{X}_g + \ddot{X}_{g \, Past})$$

Space-fixed velocity

$$\dot{X}_S = \dot{X}_M + I_{gX} + \dot{X}_o$$

Corresponding notation also holds for the Y and Z directions.

The computational method described is adequate for all periods of powered flight other than cutoff. Because the navigational computations are done only at intervals greater than 1 second, position and velocity must be extrapolated to provide proper values at S-IVB cutoff. A special routine is included in the terminal calculations to provide the necessary extrapolations, thus ensuring correct navigational quantities from which to initiate orbital navigation.

Table 2.2-1 Navigation Computations (Cont.)

Orbital Navigation

Orbital navigation includes the computations necessary to determine position, velocity, and total acceleration of the S-IVB and IU relative to the navigation coordinate system during earth orbit. These computations are carried out in an indirect fashion, making use of mathematical models of the earth, its atmosphere, and the vehicle to approximate the effects of the physical world on the vehicle. Certain simplifying assumptions were made with regard to the drag model and the rotation of the earth's atmosphere. The equations characterizing the mathematical model are:

$$\ddot{X} = \ddot{X}_g + \ddot{X}_D + \ddot{X}_V$$

$$\ddot{Y} = \ddot{Y}_g + \ddot{Y}_D + \ddot{Y}_V$$

$$\ddot{Z} = \ddot{Z}_g + \ddot{Z}_D + \ddot{Z}_V,$$

where for the first equation, $\ddot{X}$ is the total acceleration in the space-fixed X direction caused by gravitation ($\ddot{X}_g$), atmosphere drag ($\ddot{X}_D$), and venting accelerations ($\ddot{X}_V$). Corresponding notation also holds for the Y and Z directions. The system is then numerically integrated to yield position and velocity of the vehicle at any time.
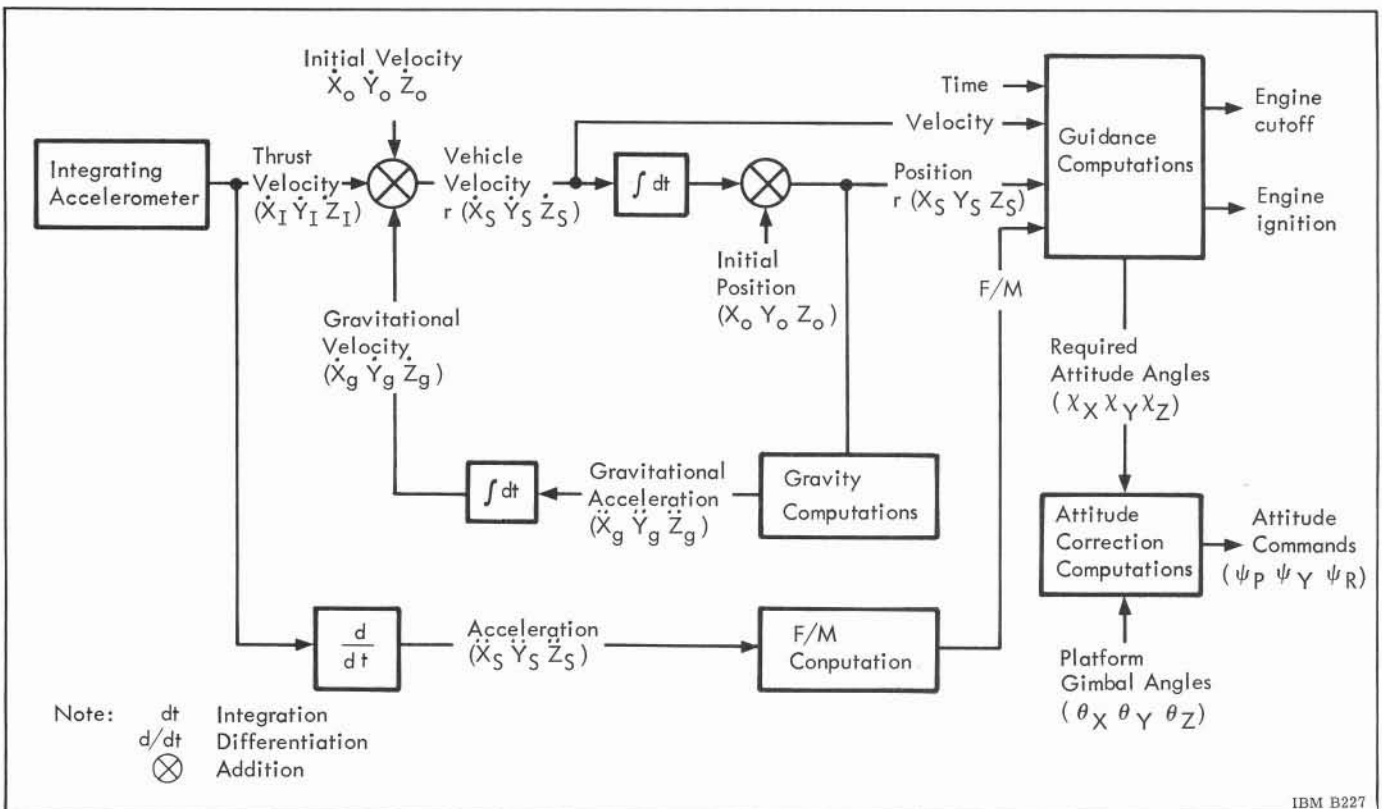
Figure 2.2-3 Navigation and Guidance Flow Diagram

SECTION 2.3
GUIDANCE

2.3.1 GENERAL CONSIDERATIONS

The guidance function of the launch vehicle is accomplished by computing the necessary flight maneuvers to meet the desired end conditions of the flight (e. g. , inserting the payload into earth orbit). Guidance computations are performed within the LVDC by programmed guidance equations which use navigation data and mission requirements as their inputs. Simply, the guidance program compares the launch vehicle's present position and heading with the predetermined "target in space", and the computed differences are issued to the attitude control equipment as maneuvering commands. The control system then alters the vehicle's thrust direction until the vehicle is on its newly computed heading.

Figure 2. 3-1 is a simplified flow diagram of the guidance computations within the LVDC. These computations are actually a logical progression of computed formulas which have a guidance command as their solution. After the desired attitude has been determined by the "best path" program, the guidance computations might be reduced into the following equation:

$$\psi = \chi - \theta$$

where: ψ is the attitude error command*
(all measured in the $X_S Y_S Z_S$ system**)

χ is the desired attitude

θ is the vehicle attitude

Calculating these attitude error commands completes the guidance scheme.

Input data to the guidance computations are the state variables of the vehicle position, velocity, and acceleration (i. e. , the navigation measurements). The result of the guidance computations is the required thrust direction (guidance command) and the time of engine cutoff and re-ignition for the S-IVB Stage. The thrust direction is expressed as three Euler angles $\chi_X \chi_Y \chi_Z$ defining the thrust vector orientation in the space-fixed coordinate system $X_S Y_S Z_S$. One final

computation, however, involves transforming the attitude commands into the vehicle reference frame from the Platform coordinate references ($\psi_X \psi_Y \psi_Z$ to $\psi_R \psi_P \psi_Y$). These values are then utilized by the control system to form the engine thrust angle commands.

PATH ADAPTIVE GUIDANCE

Overall system performance requires that the guidance scheme minimize propellant consumption for flight maneuvers and avoid excessive structural loads caused by such maneuvers. The guidance system must correct numerous flight perturbations, such as atmospheric perturbations from wind, unsymmetrical air flow because of vehicle dissymmetry, flight path deviations caused by nonstandard vehicle and engine characteristics and performance (this might even include engine-out cases), control inaccuracies, and emergency situations.

In addition, the required end conditions of the powered flight phase, at insertion into earth orbit and at injection into translunar trajectory, must be met with a high degree of accuracy to avoid additional propellant consuming maneuvers.

To meet this requirement, path adaptive guidance has been developed for Saturn Vehicles. The path adaptive guidance scheme does not constrain the vehicle to a standard trajectory, as with many ballistic missiles. Based on the instantaneous state of the vehicle, the path adaptive guidance scheme computes a new minimum propellant flight path to achieve the mission-determined end conditions. Thus, perturbations occurring during flight will be corrected in an optimum way.

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* Sometimes referred to as $\Delta\phi$ in other documentation.

** The XYZ coordinates used in this book have been changed to reflect the Project Apollo Coordinate System Standards (SE 008-001-1). This coordinate scheme becomes effective for vehicles 206 and 502 (and subsequent). For further details refer to Section 1. 2.

Several path adaptive guidance schemes have been developed for Saturn Vehicles. The polynomial guidance scheme and the iterative guidance mode have been successfully used in Saturn I flights. The iterative guidance mode will be used with Saturn IB and V and is described in Paragraph 2.3.2.

During first-stage flight (S-IB or S-IC Stage propulsion), the vehicle transverses the dense portion of the atmosphere where high aerodynamic pressure occurs. To avoid excessive structural loads caused by guidance maneuvers, no guidance constraints are applied during this flight phase. Open loop guidance, in the form of a time-tilt program, is used. Path adaptive guidance begins with the ignition of the second stage (S-II Stage for Saturn V and S-IVB Stage for Saturn IB).

Figure 2.3-1   LVDC Guidance Functions

TRAJECTORY CONSTRAINTS AND LAUNCH WINDOW

A large number of constraints apply to the trajectory of a launch vehicle. These constraints result from mission rules; environmental conditions such as the atmosphere and location of launch point; operational requirements such as safety restrictions of launch azimuth and tracking requirements; and hardware limitations such as structural load limits and available propulsion means. The dominating and most severe requirement applying to the choice of trajectories is optimal propellant utilization.

A combination of the various constraints generates a limited time period for launch; i.e., a launch window to meet mission requirements. There are 2 types of launch windows for the launch vehicle in the Saturn V/Apollo mission: the ground launch window for ascent into parking orbit and the orbital launch window for translunar injection.

The launch azimuth at the ground and the orientation (inclination and descending node) of the parking orbit are varying with time. A brief explanation of the reasons for time varient launch azimuth, inclination, and descending node seems advisable at this point. At any instant of time a plane can be defined which contains the launch site, the center of the earth, and the moon at the desired time of arrival. In order to maintain this plane, it is necessary to vary the launch azimuth on the ground continuously, since the launch site is moving with the rotating earth and the moon is moving. When range safety constraints are introduced, the launch azimuth will be limited to a certain band of values. For launches from Cape Kennedy, the band is approximately 45 to 110 degrees measured east of north and will be encountered twice each day. Within this band, a 26-degree range of azimuths will guarantee at least a 2-1/2 hour launch window. Normally, this variation will be applied to the portion of the band where launch azimuth varies linearly with the time of launch. Of the two daily launch periods, one will generally lead to a shorter coasting arc in orbit than the other. Since the geographic position of the launch site is fixed, it follows that one of the launch periods leads to time of coast in parking orbit such that ignition in orbit occurs within a range close to the launch site – sometimes referred to as the "Atlantic opportunity". The second launch period calls for coasting arcs of greater length – commonly referred to as "Pacific opportunities".

Since launch vehicle payload capability is degraded by powered plane changes, it is assumed
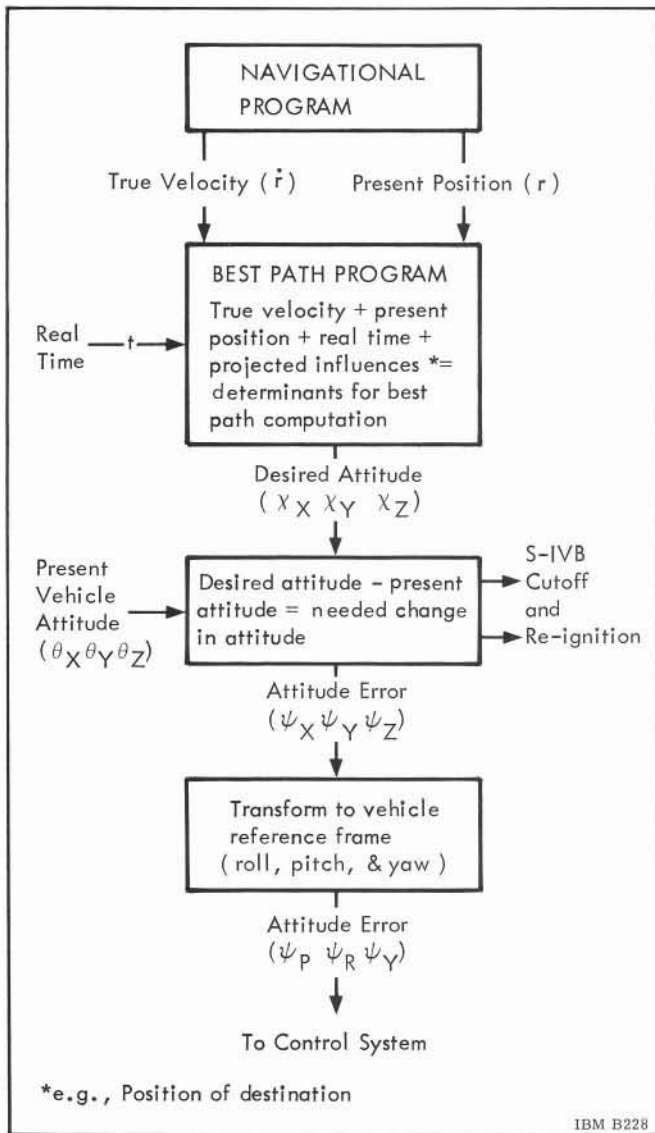
that in the nominal case there will be no such maneuvers. If it is desired to go into an orbit with an inclination to the earth's equator (greater than the latitude of the launch site), then it is possible to launch directly into the properly selected launch azimuth. There are other constraints which might be present which affect the launch window problem; e. g., launch during daylight, proper lighting of the launch site, and lighting of the return landing sites. These, however, do not affect the basic geometrical considerations.

2.3.2 ITERATIVE GUIDANCE MODE

The Iterative Guidance Mode (IGM) was developed to meet the mission flexibility requirement of large space vehicles with minimum propellant consumption. The scheme is based on optimizing techniques using calculus of variations to determine a minimum propellant flight path which satisfies the mission requirements. Experience with hundreds of minimum propellant trajectories for various orbital injection missions has demonstrated that the optimum thrust direction, relative to the local vertical, is very nearly a linear function of time during vacuum flight. Moreover, the size of the angle between the optimum thrust direction and the local horizon is never very large. These observations show a remarkable agreement with the mathematical results obtained from the calculus of variations when a flat earth model having a constant gravitational field is used, and position and velocity constraints are imposed at cutoff. A closed solution can be obtained with this mathematical model and it yields an explicit equation for the optimum thrust direction. This equation has the form:

$$\chi_P = \arctan (A + Bt) \qquad (2.3\text{-}1)$$

where χ_P is the optimum thrust direction for minimum propellant consumption and t is the time. Constants A and B are determined by the specified cutoff velocity and position, the initial values of the state variables, the vehicle thrust acceleration, and the engine specific impulse. The comparison of this equation with the results of trajectory studies suggests the use of the approximation:

$$\chi_P = A + Bt \qquad (2.3\text{-}2)$$

A rectangular injection coordinate system $(X_V Y_V Z_V)$ (Figure 2.3-2) is established with the origin at the center of the earth and with the X_V axis lying along the vertical which intersects the calculated cutoff position. Simplified equations of motions are derived to approximate the motion over an oblate earth with a realistic gravitational field. These equations of motion are solved during flight to determine the instantaneous range angle to cutoff, the time-to-go to cutoff, and the gravitational effects occurring over the remaining flight time. This information is used to compute values for A and B continuously during flight. In practice, only the value of A need be calculated when computation rates on the order of one or more per second are used (except during the last few seconds before cutoff when both A and B are computed and held constant over the remaining flight time). This is necessary because the equation gives an indeterminate command angle at the cutoff point.

The iterative guidance scheme will be activated early in the S-II Stage burn period and will continue in operation until insertion into parking orbit. Subsequently, IGM will operate until lunar transit injection with the second burn of the S-IVB Stage. The guidance scheme generates commands for the pitch-yaw angle of the thrust direction and the cutoff velocity.

The inclination of the parking orbit cutoff plane, the descending node, and the launch azimuth are functions of time of launch. During the last few minutes prior to liftoff, these quantities will be computed in the launch ground computer and the results will be put into the LVDA. At the same time, the Platform will be aligned to the direction of the desired launch azimuth. Figure 2.3-3 shows how the quantities, dependent on time of launch, vary for a typical launch day. Very simple representations of these curves, as a function of time of launch, can be obtained.

Examination of the loop shown in Figure 2.3-4 between vehicle tracking and telemetry of the LVDC and Platform, along with transmission of data in real time to the MCC, the various possibilities of updating information to the computer, of overriding commands such as the ignition equation, and of performing an alternate mission can be seen. In the case of abort (e. g., failure somewhere in the S-II to the nominal mission), no change in input is required. The solution of time-to-go and other parameters in the iterative scheme settle out quite rapidly (after 2 or 3 cycles). This statement also holds for changes in desired terminal conditions which might be built into the computer. Neither the accuracy nor the near optimality of the scheme is degraded.

In summary, the effort has been directed towards generality of guidance equations with a minimum amount of input — both for the nominal mission and for abort and alternate missions. Since a minimum amount of "tampering" with the main program

Note: Y coordinates have been omitted for clarity

$X_S Y_S Z_S$ Navigation or Platform coordinate system

$X_I Y_I Z_I$ Inertial coordinate system – sometimes written as $X_M Y_M Z_M$

$X_4 Y_4 Z_4$ Target plane coordinate system

$X_V Y_V Z_V$ Injection coordinate system

ϕ Launch site latitude

ψ_T Total range angle

i Inclination of orbit

DN   Descending Node

AN   Ascending Node

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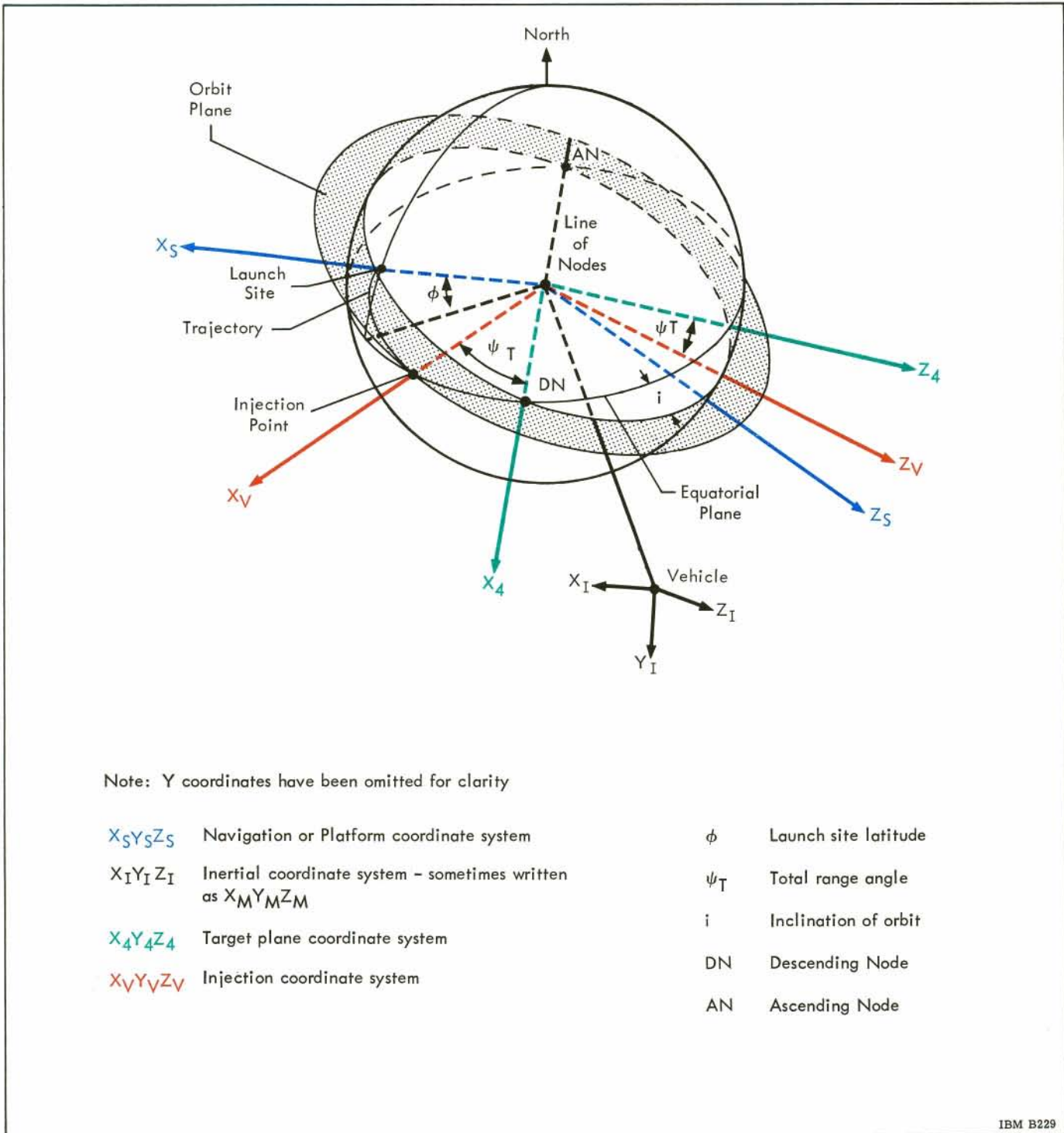
Figure 2.3-2 Guidance Coordinate Systems

in the LVDC is necessary, efficiency is at a high level, and checkout is greatly simplified. A great deal of consideration was given to the question: Is it better to have one set of guidance equations with the extraneous computation of the quantities which are nulled for a special case, or to have separate equations for the ascent-into-orbit and the flight-out-of-orbit? Close coordination with personnel who would actually be involved with the flight program on the LVDC resolved the answer: Extraneous computation, with the minimum amount of change to the flight program, is more desirable. The time required for programming a computer from one flight to the next is reduced by this method. In fact, theoretically it is possible to program the computer, change the vehicle configuration, change the mission, and never perform another checkout of the LVDC. Practically, of course, a high-confidence level is necessary for crew safety and for accomplishing the desired mission – so checkouts are performed again but of a much more limited nature than if a completely new computer program were required.

The iterative guidance equations require more LVDC capacity than most of the other guidance schemes optimized for minimum propellant consumption, but considerable flexibility is gained. The same set of guidance command equations is applicable to almost all orbital missions and can be formulated for use with any number of high thrust stages. The small number of presettings that must be calculated for a flight represents physical quantities such as vehicle exhaust velocity, nominal cutoff time, and desired cutoff position and velocity. This is an important characteristic of the scheme since these presettings may be determined without resorting to time-consuming statistical methods. The accuracy and propellant economy with the scheme are excellent. The fuel required to attain the desired cutoff conditions (at orbit insertion) is within 5 kilograms (11.1 pounds) of that required using exact minimum propellant equations obtained with the calculus of variations. This economy is obtained even under severe perturbations such as an engine failure in the first stage of a two-stage, multi-engine vehicle.

2.3.3 GUIDANCE FUNCTIONS IN FLIGHT

The following is a brief summarizing description of guidance functions during the various phases of the Saturn Launch Vehicle Mission in the Apollo Program.
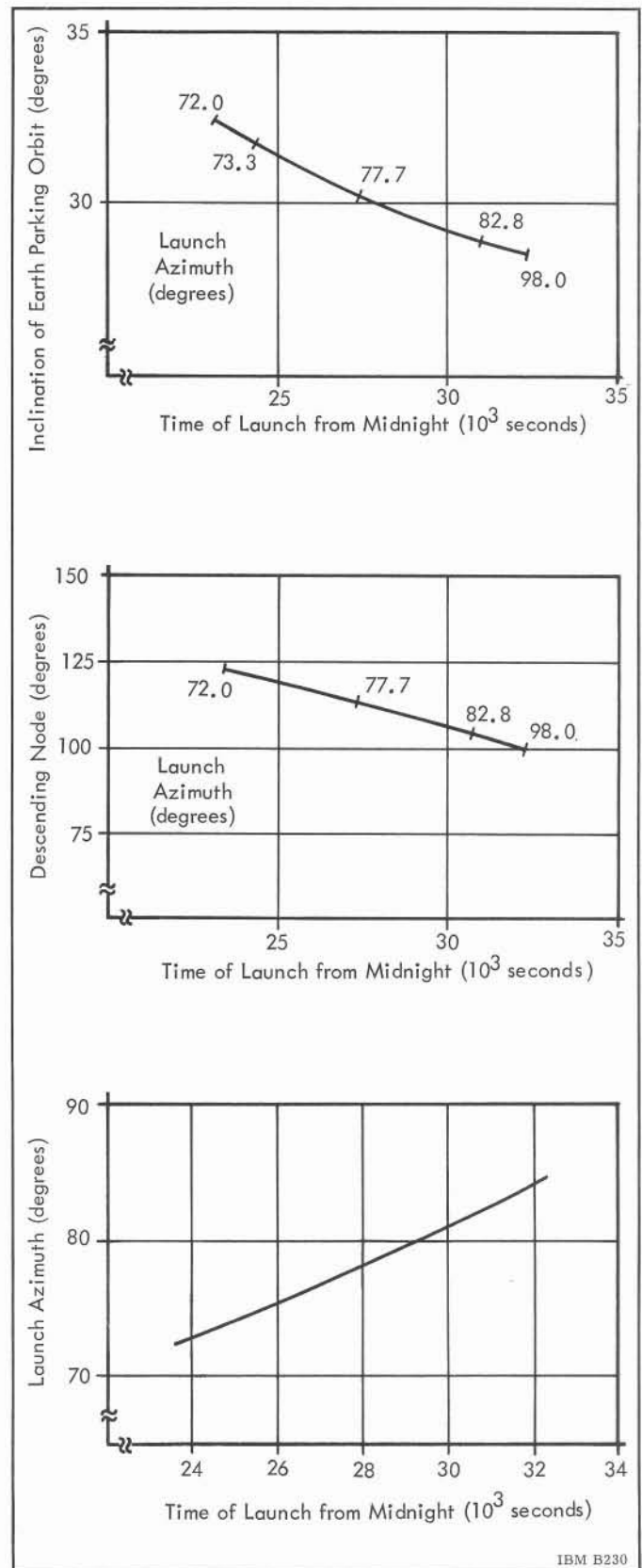
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Figure 2.3-3 Launch Window Parameters

Before launch, the Platform is erected with the X_S axis vertical and Z_S axis pointing in the direction of the launch azimuth. Since the launch azimuth is varying with time, the Platform is torqued to maintain this orientation. Just prior to liftoff, the Platform is released and becomes space-fixed oriented. The Z_S axis now determines the flight azimuth.

FIRST-STAGE FLIGHT

The vehicle lifts off vertically from the launch pad and maintains its lift-off orientation long enough to clear the ground equipment. It then performs a roll maneuver to align the vehicle with the flight azimuth direction (On the launch pad, the vehicle always has a roll orientation fixed to the launching site). This maneuver gives the vehicle control axes the correct alignment to the flight plane thus simplifying the computations in the attitude control loop.

During first-stage propulsion, a time-tilt (pitch) program, stored in the LVDC, is applied simultaneously with the described roll maneuver. The pitch angle of the vehicle is commanded according to the tilt program which is a function of time only and is independent of navigation measurements. However, navigation measurements and computations are performed through the flight, beginning at the time the Platform is released (i.e., approximately 8 to 10 seconds before liftoff). Cutoff of the first stage engines occurs when the fuel level in the tanks reaches a predetermined level. Thereafter, the first stage is separated from the launch vehicle.

SECOND AND THIRD-STAGE FLIGHT

After ignition of the S-II Stage, adaptive guidance (i.e., the iterative guidance mode) is used during all propelled flight phases of the mission. The iterative guidance mode which has been described in paragraph 2.3.2, computes the pitch and yaw angle of the required thrust direction to guide the vehicle on a minimum propellant trajectory into the predetermined parking orbit.

S-II Stage engine cutoff is initiated when the propellant in the S-II tank is consumed to a predetermined level. Following separation of the S-II Stage, the S-IVB Stage engine is ignited.

By this time the vehicle has reached approximately the orbital altitude and the S-IVB propulsion provides the necessary velocity for the circular parking orbit. When the predetermined velocity has been obtained, the guidance computations command engine cutoff.

Acceleration, velocity, and aerodynamic pressure for a typical Saturn V powered flight trajectory into earth orbit are shown in Figure 2.3-5. The step in the F/M curve around 415 seconds is caused by a change of the propellant mixture ratio to increase the specific impulse which, however, reduces the thrust and therefore cannot be applied earlier.

ORBITAL FLIGHT

During orbital coast flight, the navigation program continually computes the vehicle position and
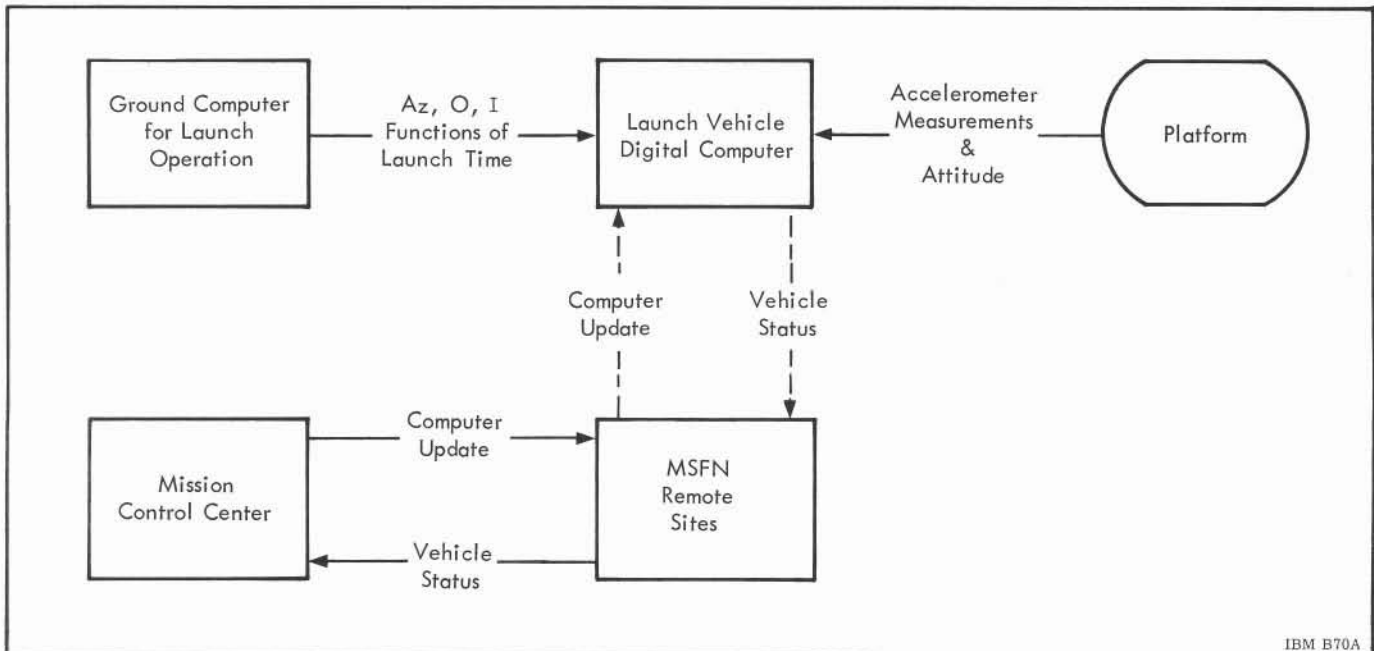
Figure 2.3-4 Linkage Between Vehicle and Ground
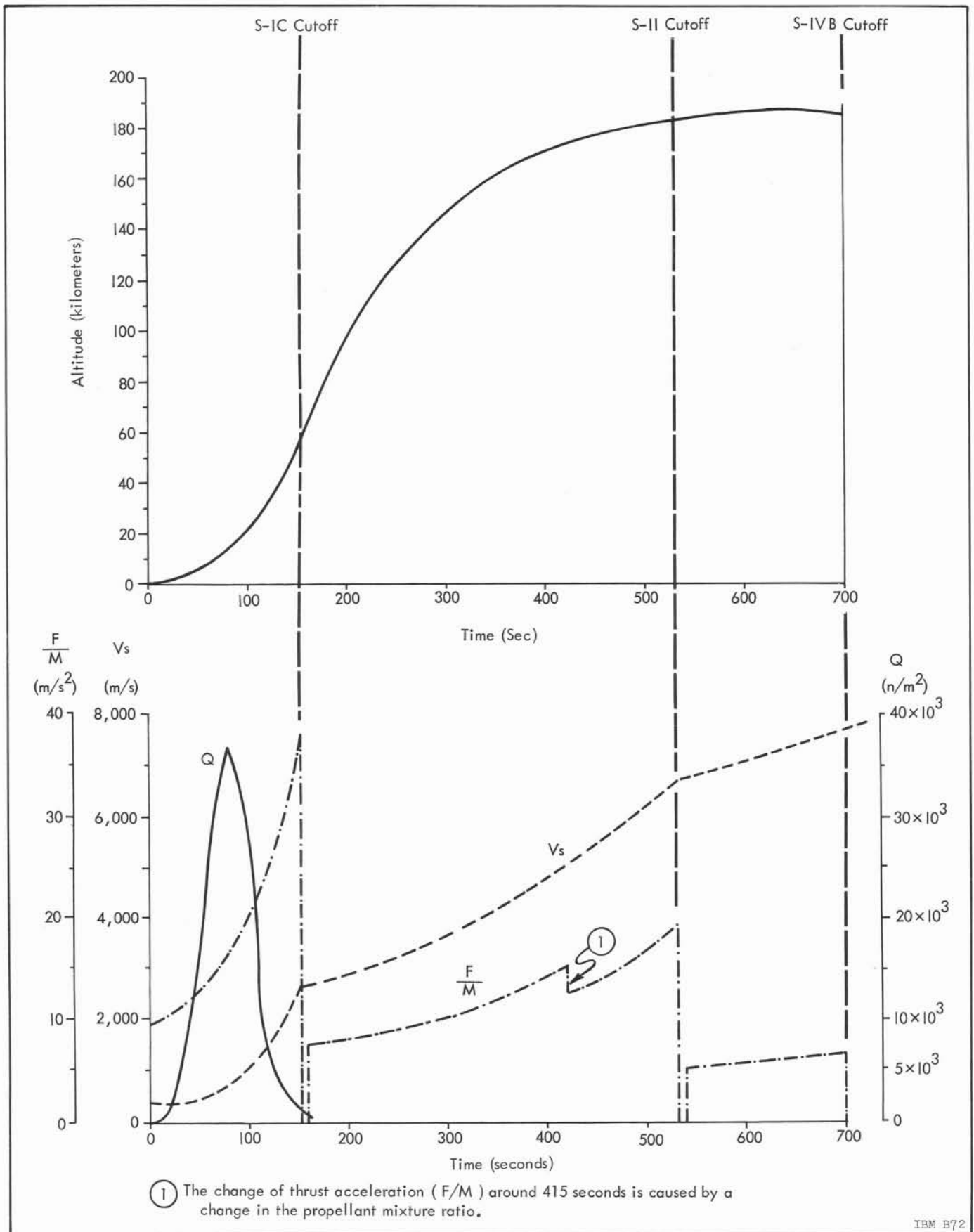
Figure 2.3-5 Altitude, Acceleration (F/M), Velocity (V_s), and Aerodynamic Pressure (Q)
for a Typical Saturn Trajectory

velocity from the equations of motion based on inser-
tion conditions. Attitude of the vehicle roll axis in
orbit is normally maintained at 90 degrees with re-
spect to the local vertical. The local vertical is de-
termined from navigational computations. The time
of re-ignition of the S-IVB engine and the required
thrust orientation for powered flight-out-of-orbit are
computed during each orbit.

In orbit, navigation and guidance information
in the LVDC can be updated by data transmission from
ground stations through the IU radio command system.

TRANSLUNAR INJECTION

When the computed time of re-ignition occurs,
the S-IVB engine is ignited. The same guidance equa-
tions are used again for the translunar injection. The
S-IVB propulsion is cutoff when the proper energy
(velocity) for translunar injection is achieved.

In the following flight phase, up to and through
the transposition maneuver, navigation and guidance
computations continue.

CHAPTER 3
ATTITUDE CONTROL

SECTION 3.1
ATTITUDE CONTROL SYSTEM

3.1.1 GENERAL CONSIDERATIONS

The attitude control requirement of the Saturn Launch Vehicle can be divided into attitude control during powered flight and attitude control during coast flight. Attitude control and stabilization during powered flight are accomplished by swivelling of the propulsion engines to control thrust vector direction. During coast flight (in orbit and after translunar injection), attitude control is provided by the auxiliary propulsion system of the S-IVB Stage. This auxiliary propulsion system is used also during powered flight of the S-IVB Stage for roll control which cannot be achieved with the single engine of the S-IVB Stage.

Control of the launch vehicle during ascent into earth orbit can be divided into 2 phases. During first-stage powered flight, the vehicle traverses the dense portion of the atmosphere. Large side forces (resulting from aerodynamic flow and wind) can act on the vehicle. Maneuvers during this flight phase are limited because of structural constraints. Structural loads from aerodynamic forces are kept within a tolerable range by controlling the vehicle to achieve a minimum angle of attack. Aerodynamic pressure reaches a maximum at approximately 12 kilometers (7.4 miles) altitude 77 seconds after lift-off. For a typical Saturn V trajectory, see Figure 2.3-5.

From the viewpoint of vehicle control, maneuvers during vacuum flight are limited only to the extent that structural bending and propellant sloshing are not dangerously excited through the nature of the control command.

The objective of the attitude control system during propulsion of the vehicle is to appropriately orient the thrust vector, relative to the vehicle, such that the required attitude commands are performed in a satisfactorily damped mode of rotation.

The vehicle cannot be considered rigid but must be treated as distributed masses connected by an elastic structure. Forces acting on these masses, resulting from atmospheric perturbations or active control of the vehicle, excite the complex spring-mass system and cause body bending. Since the structure possesses low damping, oscillatory bending modes of considerable amplitude can be produced, to which control sensors may be subjected at their particular location. Thus, incorrect information about the total vehicle behavoir may cause self-excitation of the bending modes and instability of the vehicle control system.

The vehicle is aerodynamically unstable — which means that the center of pressure is located forward of the center of gravity. As an example, Figure 3.1-1 is a plot of the center of pressure and the center of gravity for the first phase of the Saturn V flight and shows that the vehicle is unstable except for a short period of time around the 60th flight second.

Much of the vehicle characteristic data varies widely with time and the individual stages. In addition, some can be predetermined only to a certain degree, and tolerances must be imposed. Thus a wide operating range of the control system must be provided.

A typical frequency spectrum for the Saturn V Vehicle during powered flight can be seen in Figure 3.1-2. The frequency bands, rather than single frequencies, are the result of changing vehicle-state conditions because of propellant consumption as a function of flight time. The frequency band marked "engine-reaction-zero" represents the sinusoidal response where the inertial effects of the engine induce forces at the engine gimbal point equal and opposite to the side forces obtained by swivelling the thrust vector. The net effect is that the vehicle bending mode response to the engine deflection is zero at this frequency. This is sometimes known as the "tail-wags-dog" frequency.

The basic control scheme is an attitude/attitude-rate scheme. Accelerometer control may be added to produce drift minimum or load minimum.

The control law for the thrust vector deflection angle (β) is:

$$\beta = a_0 \psi + a_1 \dot\phi + g_2 \dot\gamma \quad \text{(Saturn IB)} \qquad (3.1\text{-}1)$$

$$\beta = a_0 \psi + a_1 \dot\phi \quad \text{(Saturn V)}$$

Where a_0 and a_1 are gain factors, ψ and $\dot\phi$ are attitude error angle and attitude angular rate respectively, $\dot\gamma$ is the lateral acceleration* measured by the accelerometers with its sensitive axis perpendicular to the vehicle longitudinal axis, and g_2 is the corresponding gain factor. The attitude error angle (ψ) is obtained from the LVDC/LVDA, the attitude rate ($\dot\phi$) from the rate gyros, and the lateral acceleration ($\dot\gamma$) from control accelerometers.

The lateral acceleration control is used during first-stage propulsion* to reduce structural loads from aerodynamic forces and to provide drift-minimum control.

The acceleration measured by the body-mounted Control Accelerometers differs from that measured by the guidance accelerometers. The guidance accelerometers are mounted on the stable platform and measure accelerations with respect to an inertial space coordinate system while the Control Accelerometers are used to measure side or lateral acceleration with respect to the vehicle coordinates.

The choice of the control sensor (rate gyros and accelerometers) locations is very important for attitude stabilization of the vehicle. Stabilization of large launch vehicles, with respect to bending and torsion modes, is usually obtained by shaping networks in each of the three control sensor channels.

Elastic body stability may be attained either by phase stabilization or by attenuation of the bending frequencies. Each method has its advantages. Phase stabilization, especially of the lower bending modes, makes it possible to maintain adequate bandwidth at the control frequency. Also, a phase-stabilized bending mode will damp out faster after excitation than a mode that depends upon attenuation for stabilization. In contrast, stabilization by attenuation is preferable for the higher bending modes since these mode shapes

*Used only on Saturn IB Vehicles
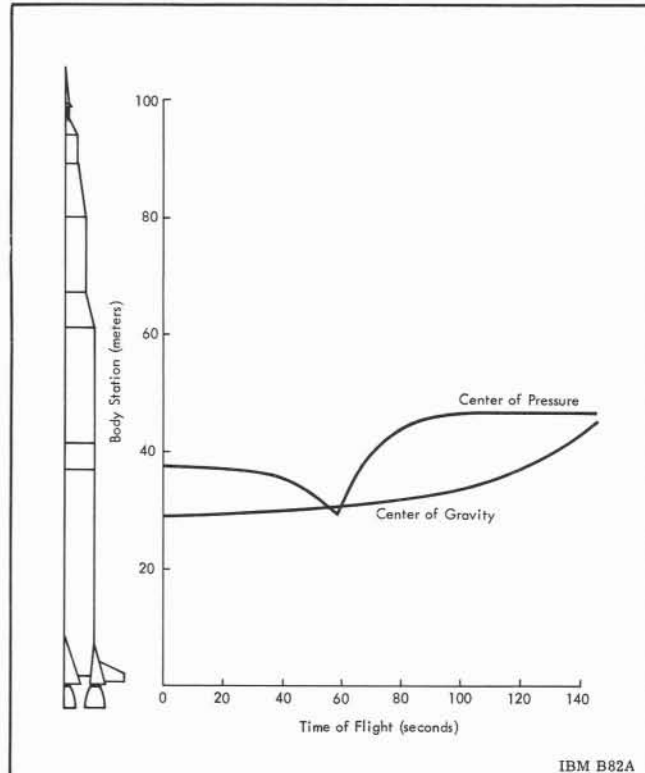
Figure 3.1-1 Variations of CP and CG During Flight

are generally more difficult to define and because it is possible to compensate bending mode shapes and frequencies satisfactorily by attenuation. Thus for practical reasons, phase stabilization is applied to the lower frequencies and gain stabilization (stabilization by attenuation) to the higher frequency spectrum. In the case of large launch vehicles like Saturn V, some bending modes are so close to the control loop frequency that they are best phase stabilized. Thus, for Saturn V during first-stage operation, the first and second bending modes will be phase stabilized and higher modes will be attenuated. During the second-stage powered flight, only the first bending modes will be phase stabilized and higher modes will be attenuated. During the third-stage powered flight, the bending mode frequencies are higher than those for the other stages; therefore, all the bending modes of this stage will be stabilized by attenuation. All torsional modes are at a relatively high frequency and will be attenuated for all stages.

(A) Frequency Spectrum During First-Stage Propulsion.

(B) Frequency Spectrum During Second-Stage Propulsion.

(C) Frequency Spectrum During Third-Stage Propulsion.

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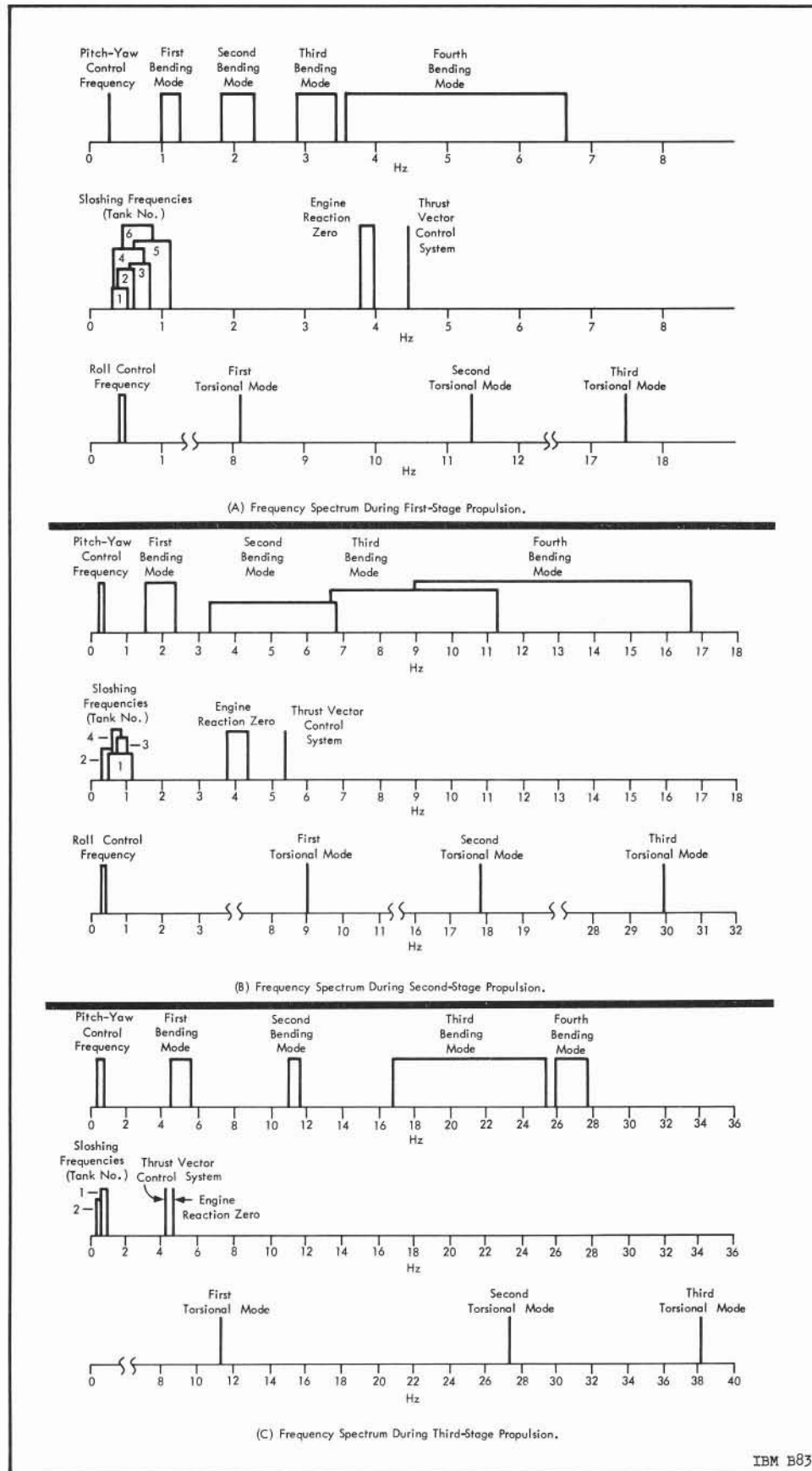
Figure 3. 1-2 Typical Saturn V Frequency Spectrum

To provide the desired phase, it is necessary that the rate gyro be properly located on the vehicle. Figure 3.1-3 exhibits the Saturn configuration and the first two bending mode shapes.

The attitude gyros and rate gyros are both located in the Instrument Unit. Since the control loop gain at the bending mode frequencies is much less through the attitude loop than through the rate loop, the location of the attitude gyro is not as sensitive and is usually determined by other considerations - primarily the need to have one centrally located unit for all stages. The location of the body-mounted accelerometers (used only in the Saturn IB first-stage propulsion phase) is also in the Instrument Unit. However, should design changes dictate, the accelerometers could be located elsewhere on the vehicle.

3.1.2 ATTITUDE CONTROL FOR POWERED FLIGHT

Attitude control of the launch vehicle during powered flight is accomplished by swivelling of the main propulsion engines and thereby changing the orientation of the thrust vector.

The attitude control system of the Saturn V Launch Vehicle provides attitude control and stabilization from launch until separation of the S-IVB/IU
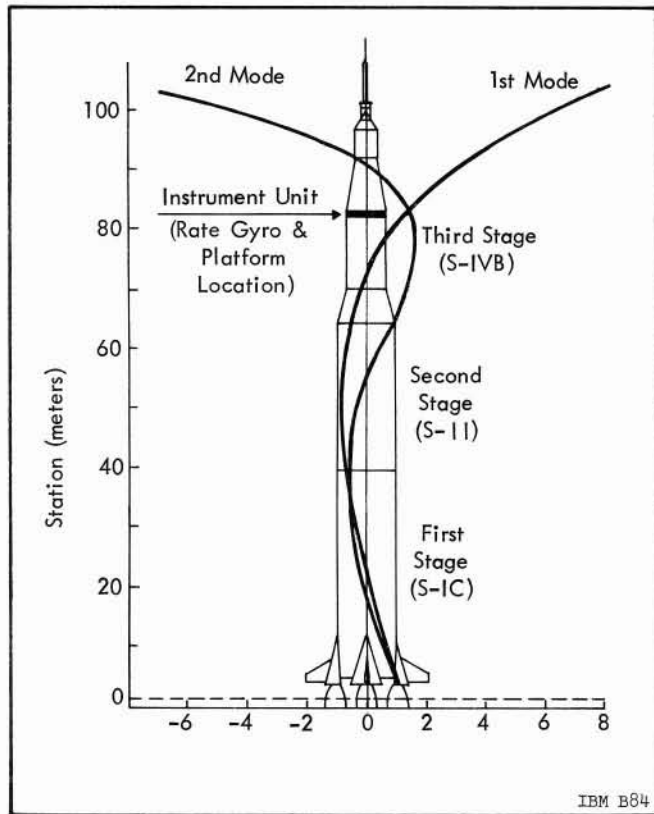
Figure 3.1-3 Shape of the First and Second Bending Modes (Saturn V)

Stage from the Apollo Spacecraft. For the second-burn phase of the S-IVB Stage, the Apollo navigation guidance and control system is available as a backup system and sends input signals to the Saturn Flight Control Computer. This backup capability for the S-II and S-IVB first-powered flight is under consideration.

The S-IB Stage has eight engines. The four outer engines are swivelled in pitch and yaw by two hydraulic actuators at each engine (see Figure 3.1-4). The S-IC and S-II Stages have five engines each. The four outer engines can be swivelled in pitch and yaw by two hydraulic actuators at each engine. The S-IVB Stage is propelled by a single engine which can be swivelled in pitch and yaw. In addition, the S-IVB Stage is equipped with an auxiliary propulsion system consisting of 6 nozzles (2 sets of 3 nozzles) mounted on the outside of the aft end of the S-IVB Stage. Four of the six nozzles are required for roll control of the S-IVB Stage which cannot be achieved with a single main propulsion engine. Figure 3.1-5 shows the layout of engine actuators and nozzles for the stages of the Saturn V Vehicle. The polarity tables given in this figure indicate which actuators or nozzles are required to perform a pitch, yaw, or roll maneuver.

The main components of the Saturn attitude control system are shown in Figure 3.1-6. The platform gimbal angle readings (θ_X θ_Y θ_Z)* indicate the orientation of the vehicle in the navigation coordinate system (X_S Y_S Z_S). The LVDC computes the required thrust vector orientation (i.e., the desired gimbal angles (χ_X χ_Y χ_Z) according to the guidance scheme and generates the difference $\Delta \theta = \theta - \chi$). From the attitude angle differences ($\Delta \theta_X$ $\Delta \theta_Y$ $\Delta \theta_Z$), attitude error signals (ψ_R ψ_P ψ_Y — with respect to the body-fixed roll, pitch, and yaw axes) are generated and sent to the Flight Control Computer.

In addition to the attitude error signals, the Flight Control Computer receives signals from control sensors — attitude rate signals (ϕ_P ϕ_Y ϕ_R — with respect to the vehicle's pitch, yaw, and roll axes) and lateral acceleration signals ($\ddot{\gamma}_P$, $\ddot{\gamma}_Y$ — along the pitch and yaw axes). All control signals fed into the Flight Control Computer are analog signals. The Flight Control Computer processes and combines these signals according to the control law (Equation 3.1-1) to generate the control signals for the engine actuators and attitude control nozzles.

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* The XYZ coordinates used in this book have been changed to reflect the Project Apollo Coordinate System Standards (SE 008-001-1). This coordinate scheme becomes effective for vehicles 206 and 502 (and subsequent). For further details refer to Section 1.2.

The number and exact location of the control sensors (rate gyros and control accelerometers) depend on the vehicle configuration and the results of further studies of vehicle dynamics.

The Saturn IB Vehicle attitude control system will use signals from two Control Accelerometers (during first-stage powered flight only). There are two different schemes under consideration for the S-IB powered flight phase. In the first scheme, lateral acceleration information is supplied by Control Accelerometers located in the IU, and angular rate signals are provided by the Control-EDS Rate Gyros also located in the IU. In the second scheme, the Control Accelerometers and an additional Control Rate Gyro package would be located on the thrust cone* at the aft end of the S-IVB Stage. After separation of the S-IB Stage, only the Control-EDS Rate Gyros in the IU will be used.

Control studies of the Saturn V Vehicle have indicated that no appreciable reduction in bending moments is obtained with body-fixed control accelerometers. Therefore, no control accelerometers will be used in Saturn V Vehicles. During the S-IC powered flight, either the Control-EDS Rate Gyros in the IU or a separate Control Rate Gyro package on the thrust cone of the S-IVB Stage will be used. After the first-stage separation, only the Control-EDS Rate Gyros in the IU will be used.

3.1.3 ATTITUDE CONTROL FOR COAST FLIGHT

During coast flight, attitude control of the vehicle is accomplished by means of the S-IVB Stage auxiliary propulsion system which contains six attitude control engines (thrust nozzles). When fired, the nozzles produce torques about the center of gravity of the vehicle. They are controlled in a pulse-type manner (full thrust or OFF) by the spatial amplifiers in the Flight Control Computer. The attitude control engines are turned ON when the input signal to the spatial amplifier exceeds certain limits and turned OFF when the input signal falls below a set threshold. The control scheme generates thrust pulses of variable duration at changing intervals which are controlled by the pseudo rate modulators in the spatial amplifier. Over a certain range of the input signal, the output of the modulator is pulse-width and frequency modulated and is a function of the input signal magnitude. Above this range, the output (thrust) is continuously ON; below this range, it is OFF.

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*The thrust cone is the cone-shaped structure on which the S-IVB J-2 engine is mounted.

A phase diagram (Figure 3.1-7) shows the angle and angular rate relationship for a limit cycle operation. When the vehicle is rotating in a positive direction with positive angle (corresponding to point A on the diagram), it will reach point B when the combination of ψ and $\dot{\phi}$ exceeds the upper side of the attitude deadband — thus firing the rocket motor. The angular rate of $\dot{\phi}$ decreases and changes to a negative quantity, placing the vehicle at point C on the diagram where the signal, still positive, falls below the level required to keep the motor ON, and the torque disappears. From this point, the vehicle drifts at a constant angular velocity toward condition D where the negative deadband limit is exceeded.

The attitude control engines of the S-IVB Stage auxiliary propulsion system are used to control the vehicle attitude in roll during powered flight phases and in pitch, yaw, and roll during coast flight phases. These engines are located in 2 modules 180 degrees apart on the aft end of the S-IVB Stage. One module is located over position I and the other over position III as shown in Figure 3.1-5. Each module has three hypergolic attitude control engines. One engine of each module is used to control the vehicle's attitude in pitch while the other two are used in yaw and roll control.

The attitude engine control signal is composed of an attitude error signal (ψ) and a vehicle turning rate signal ($\dot{\phi}$). The body mounted Control-EDS Rate Gyros, located in the Instrument Unit, supply the desired vehicle turning rate information. Attitude error information (ψ) may come from 2 sources. When the Instrument Unit is controlling the vehicle attitude, error information is supplied by the LVDC through the LVDA as shown in Figure 3.1-8. When the Spacecraft is controlling the vehicle attitude error, information is obtained from the Spacecraft over Spacecraft/IU interface lines.

Limits are imposed by the LVDC for IU-generated attitude error signals and by the IU Flight Control Computer for Spacecraft-generated attitude error signals. It should be noted in the figure that attitude error limits of ±2.5 degrees in pitch and yaw and ± 3.5 degrees in roll are imposed on the attitude error signals. These error signals are used only by the auxiliary propulsion system.

The attitude error signals from the spacecraft may originate in the Apollo navigation, guidance, and control system or may be generated by the Astronaut through manual control. In any case, the limiters in the IU Flight Control Computer will limit the angular rate. These limits prevent excessive propellant

S-IB Control Signal Polarity Table			
Actuator No.	Signal & Actuator Movement		
	$+\psi_R$	$+\psi_Y$	$+\psi_P$
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

S-IB Actuator Layout

S-IB Stage

Yaw or Z Axis

Pitch or Y Axis

CG

$+\psi_R$
$+\dot\phi_R$

$+\psi_P$
$+\dot\phi_P$
$+\ddot\gamma_Y$

$+\psi_Y$
$+\dot\phi_Y$ $+\ddot\gamma_P$

Roll or X Axis

NOTES:

1. All signal arrows indicate positive vehicle movements.
2. Vehicle pitches over position I.
3. Engine actuator & control engine layouts shown as viewed from aft end of vehicle.
4. $+\beta$ indicates engine deflection required to correct for positive vehicle movement.
5. Actuator movements noted in the Polarity Table are characteristic for all input signals of like polarity.
6. Beginning with vehicle 206, the roll, pitch, and yaw axis will also be known as the X, Y, Z axis, respectively.

IBM B86A

Figure 3.1-4 S-IB Engine and Actuator Configuration

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. Beginning with vehicle 502, the roll, pitch, and yaw axes will also be known as the X, Y, Z axes, respectively.

S-IC & S-II Actuator Layouts

S-IVB Actuator and Nozzle Layout
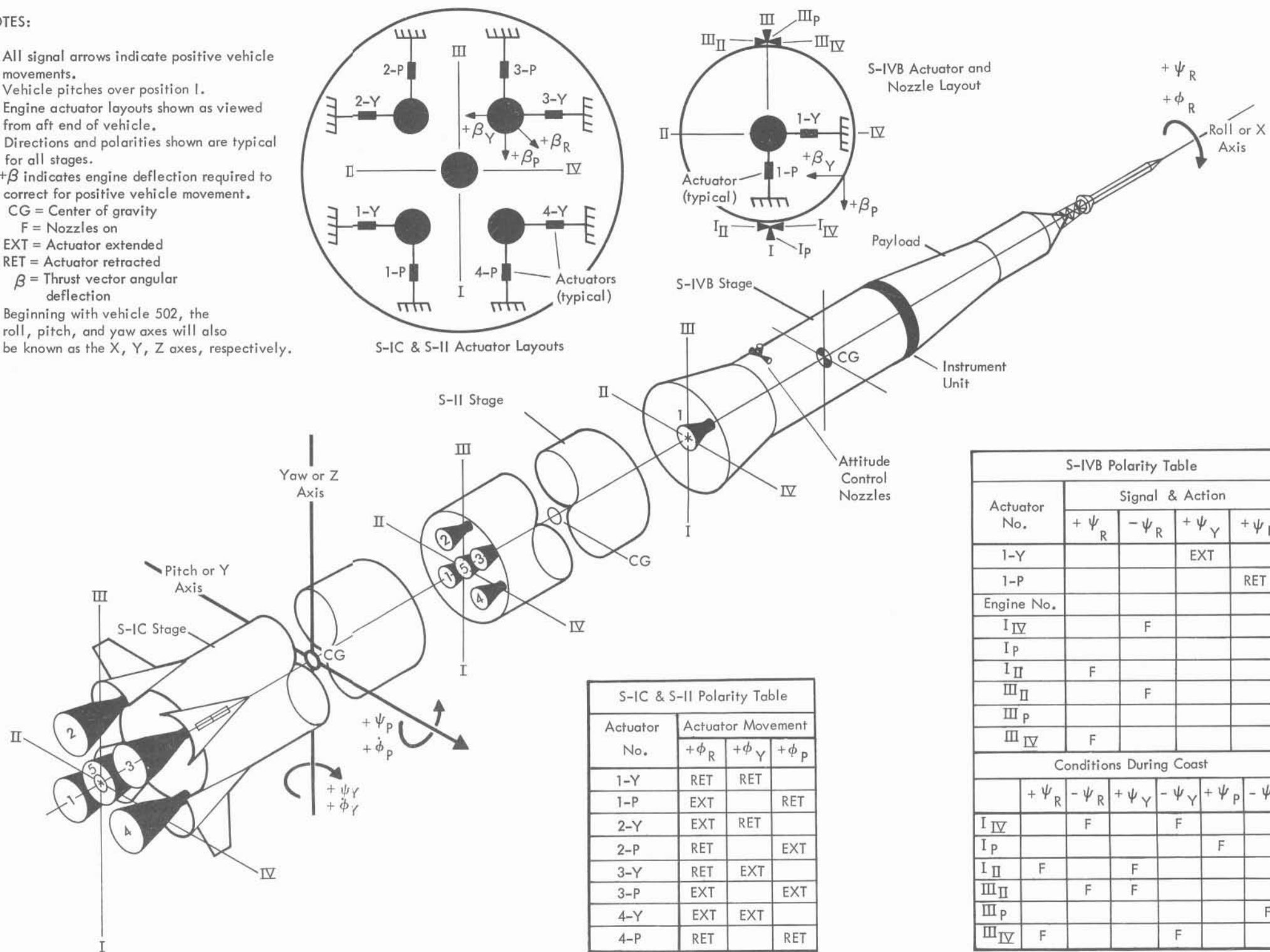
S-IC & S-II Polarity Table			
Actuator No.	Actuator Movement		
	$+\phi_R$	$+\phi_Y$	$+\phi_P$
1-Y	RET	RET	
1-P	EXT		RET
2-Y	EXT	RET	
2-P	RET		EXT
3-Y	RET	EXT	
3-P	EXT		EXT
4-Y	EXT	EXT	
4-P	RET		RET

S-IVB Polarity Table				
Actuator No.	Signal & Action			
	$+\psi_R$	$-\psi_R$	$+\psi_Y$	$+\psi_P$
1-Y			EXT	
1-P				RET
Engine No.				
I IV		F		
I P				
I II	F			
III II		F		
III P				
III IV	F			

Conditions During Coast						
	$+\psi_R$	$-\psi_R$	$+\psi_Y$	$-\psi_Y$	$+\psi_P$	$-\psi_P$
I IV		F		F		
I P					F	
I II	F		F			
III II		F	F			
III P						F
III IV	F			F		

Figure 3.1-5 Saturn V Engines, Actuators, and Nozzle Arrangement

IBM B87A
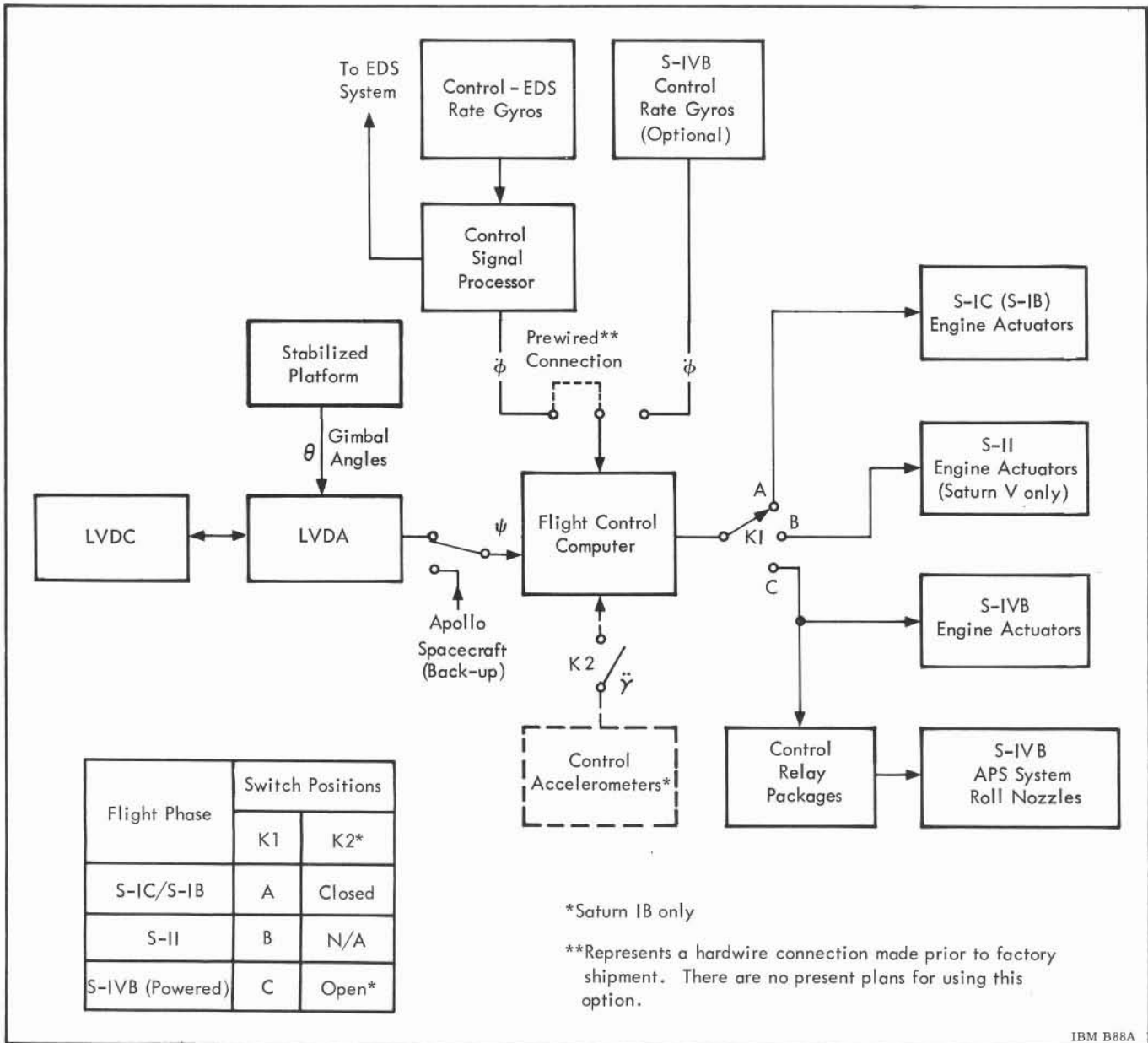
Figure 3.1-6 Saturn Control System Block Diagram (Powered Flight)

usage which would result from large angular rate commands while driving the vehicle to the desired attitude.

The Apollo Spacecraft attitude reference system can follow the instantaneous vehicle attitude. This is accomplished by driving the command display unit servo motor with an error signal which is formed by differencing the commanded and actual gimbal angles. When the Astronaut wishes to maintain a particular attitude orientation, he can use the computer to set the command display unit command resolver to the desired gimbal value. The difference between the commanded and actual gimbal angles results in an error

signal which is resolved into vehicle coordinates and is given to the IU Flight Control Computer as an attitude error signal. The S-IVB attitude control system then operates in the limit-cycle mode about this command attitude.

3.1.4 ATTITUDE SIGNAL PROCESSING

A processing and flow diagram for attitude control signals during powered flight is illustrated in Figure 3.1-9. The analog attitude angle difference signals (θ_X θ_Y θ_Z) from the gimbal angle resolvers are converted in the LVDA to digital signals with a readout rate of 25 signals per second. The guidance

command angles (χ_X χ_Y χ_Z) are computed approximately once per second. The attitude angle differences ($\Delta\theta$) are then transformed into the body-fixed axis system to generate the attitude error signals (ψ_P ψ_Y ψ_R). This transformation is performed by using simplified equations for the Euler angle rotations. Attitude control signals are generated 25 times per second to minimize vehicle stabilization problems and provide near continuous attitude control (the coefficients A_1 through A_5 are calculated approximately once per second). The gain to be used in these equations is defined in Table 3.1-1.

The attitude error signals (ψ) from the LVDC are converted to analog voltages by ladder decoders in the LVDA. The resolution of the ladders is 0.0575 degree with a maximum value of 15.3 degrees. The analog attitude error signal is then fed to the Flight
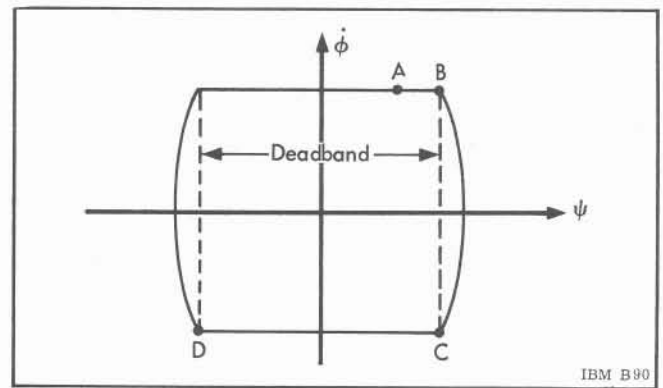
Figure 3.1-7 Limit Cycle Phase Diagram

Control Computer where it is combined with the angular rate signals and lateral acceleration signals (for Saturn IB). A functional description of the Flight Control Computer is given in Section 3.3.
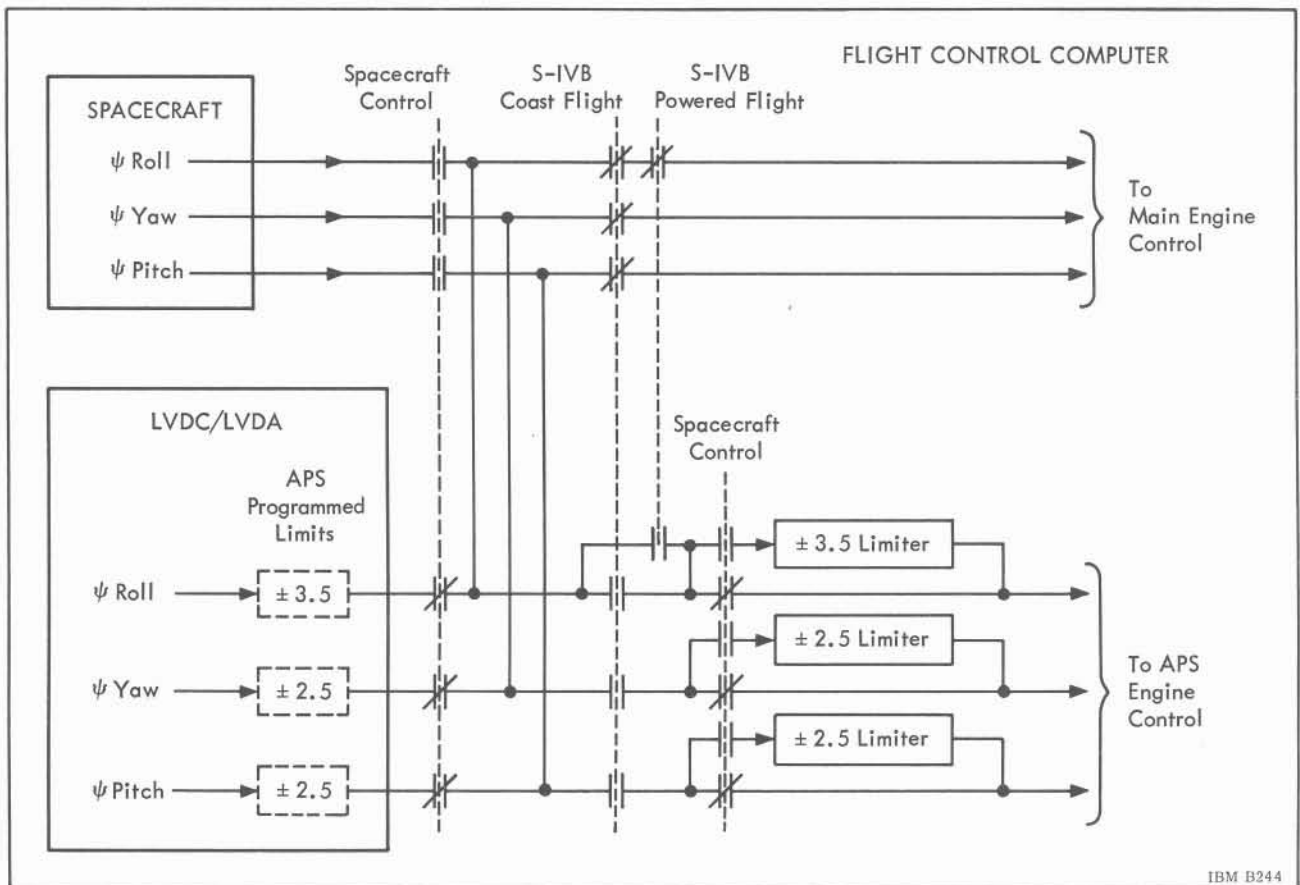
Figure 3.1-8 Attitude Error Signal Sources Functional Block Diagram
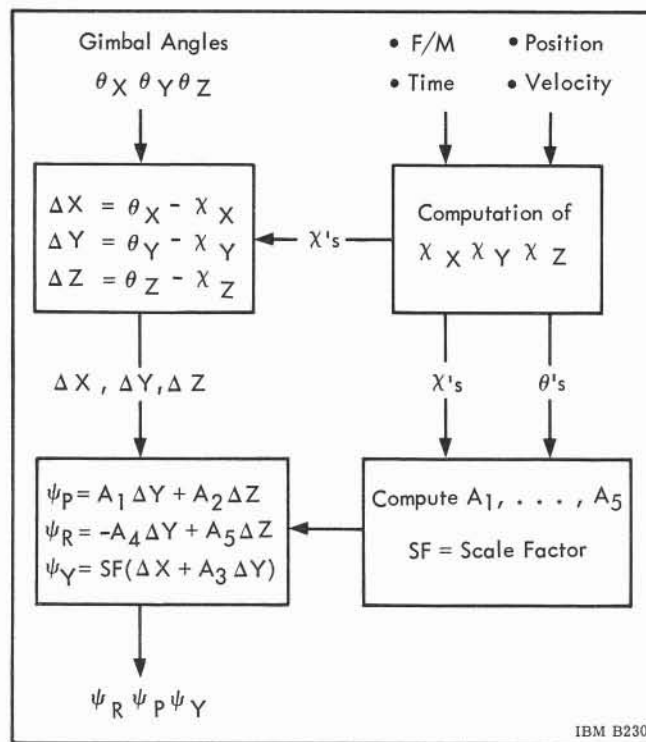
Figure 3.1-9 Attitude Signal Flow Diagram

Table 3.1-1 Typical Attitude Parameters for Saturn V

Flight Mode	Attitude Control Gain	Attitude Rate Limits			Limits on Magnitude of Attitude Commands		
		Yaw	Pitch	Roll	Yaw	Pitch	Roll
S-IC Burn	1	$\pm 12°/s$	$\pm 12°/s$	$\pm 12°/s$	$15°$	$15°$	$15°$
S-II Burn	1	$\pm 12°/s$	$\pm 12°/s$	$\pm 12°/s$	$15°$	$15°$	$15°$
S-IVB 1st Burn	1	$\pm 12°/s$	$\pm 12°/s$	$\pm 12°/s$	$15°$	$15°$	$3.5°$
Coast	1	$\pm 2.4°/s$	$\pm 2.4°/s$	$\pm 2.4°/s$	$2.5°$	$2.5°$	$3.5°$
S-IVB 2nd Burn	1	$\pm 12°/s$	$\pm 12°/s$	$\pm 12°/s$	$15°$	$15°$	$15°$
Coast	1	$\pm 2.4°/s$	$\pm 2.4°/s$	$\pm 2.4°/s$	$2.5°$	$2.5°$	$2.5°$

SECTION 3.2
CONTROL SENSORS

3.2-1 RATE GYROS

Single degree of freedom rate gyros are used on the Saturn Vehicles to sense the angular rate of movement of the vehicles about the roll, yaw, and pitch axes. The same type of rate gyros are employed in two separate packages, the Control-EDS Rate Gyro package and the Control Rate Gyro package. The Control-EDS Rate Gyro package is located in the IU on the Saturn IB and V Vehicles. It is used for attitude control and EDS operation. The location of the Control Rate Gyro package, and whether it will be used, depends on further studies of the vehicle bending characteristics.

The rate gyro (Figure 3.2-1) uses a microsyn differential transformer pickoff which is coupled to the gimbal. Angular rate about the input axis of the rate gyro will be translated to an ac voltage output signal from the microsyn. The amplitude of this signal has a definite relationship to the angular rate. This relationship, called the scale factor, is approximately 200 mV rms per degree/second. The rate gyro has the capability of sensing angular rates to 20 degrees/seconds in either counter-clockwise or clockwise direction.

Each rate gyro is equipped with self-checking capabilities. A self contained electro-magnetic coil
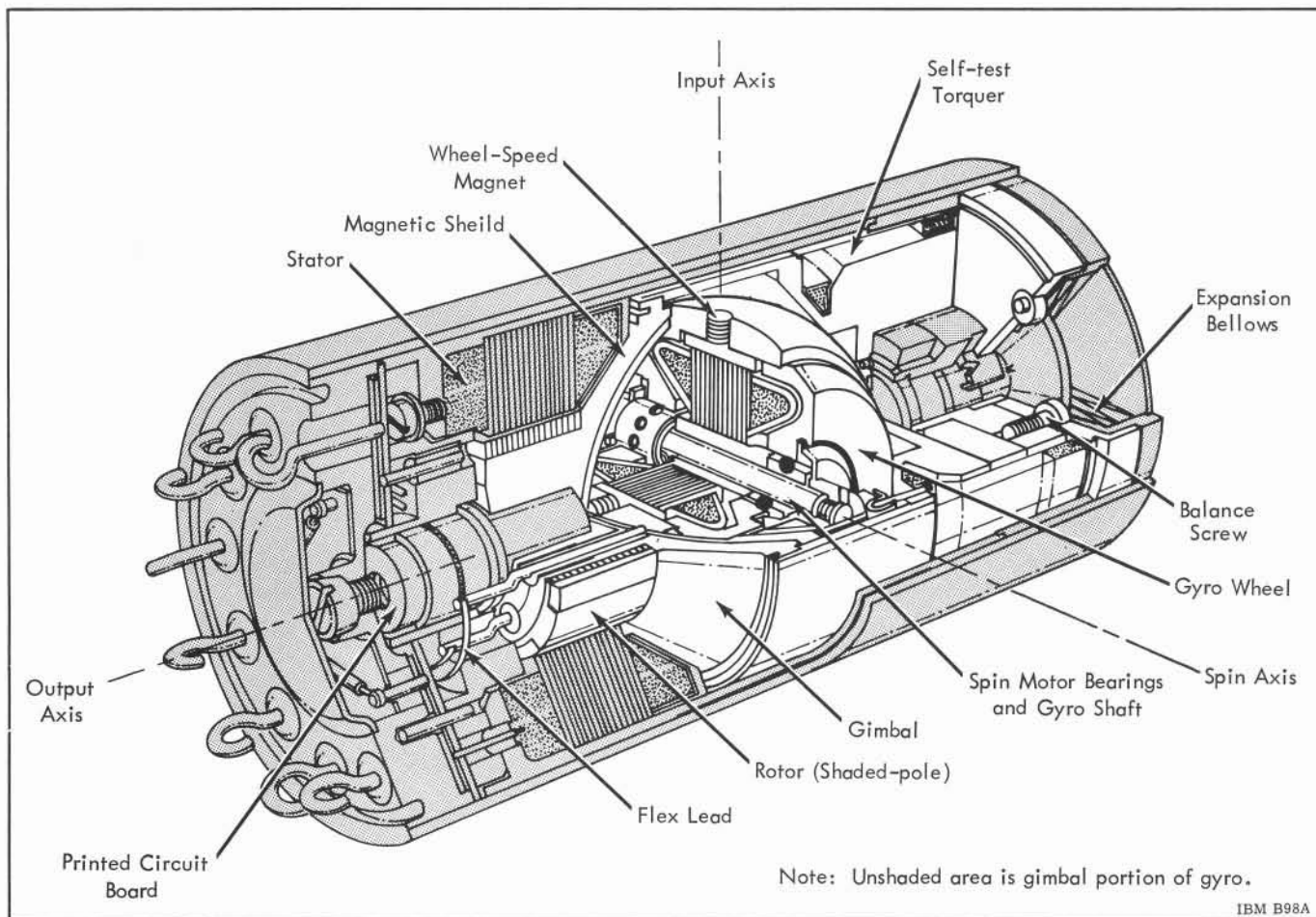
Figure 3.2-1 Cutaway View of a Self-Test Rate Gyro

can be remotely excited to cause the rate gyro gimbal to be deflected. This deflection provides an output from the microsyn without subjecting the gyro at a rate input. A built-in generator, integral with the gyro rotor, provides ac pulses (the frequency is the indication of the gyro rotor speed).

The rate gyros are fluid-damped devices which maintain a relatively constant damping factor over the range of temperatures in which the damping fluid retains its fluid characteristics. When the gyros are employed in low-temperature environments which exceed the gyro ratings (such as will be found in several locations of the Control Gyro package), the package must be supplied with heaters to ensure proper operation.

3.2.2 CONTROL-EDS RATE GYRO PACKAGE

The Control-EDS Rate Gyro (Figure 3.2-2) contains nine rate gyros arranged in a triple redundant configuration (Figure 3.2-3). Each group of three rate gyros provides output signals for angular rate about the pitch axis ($\dot{\phi}_P$), yaw axis ($\dot{\phi}_Y$), and roll axis ($\dot{\phi}_R$). Each group of gyros is completely independent of the two other groups. Thus, there are nine separate output signals – one from each gyro.

The comparison and selection of the output signals are accomplished in the Control Signal Processor.

Each of the gyro groups receives 400 Hz, single-phase power from a separate power supply. These supplies are located in the Control Signal Processor.

This triple redundant configuration increases the reliability for the Control-EDS Rate Gyro package which is used throughout the mission.

3.2.3 CONTROL SIGNAL PROCESSOR

The Control Signal Processor converts the ac rate gyro signals into dc signals required as inputs to the Flight Control Computer. It also compares and selects output signals from corresponding redundant rate gyros. A block diagram of the Control Signal Processor is given in Figure 3.2-4.

The 400 Hz output signal from each rate gyro is demodulated to generate a dc voltage which is proportional to the rate output of the gyro. A telemetry output from each demodulator is provided.

Two of the three demodulated outputs (of one channel) are compared within a comparison circuit,

Pitch and Roll Rate Gyros

Printed Circuit Board Assembly

Microsyn Excitation Chokes

Yaw Rate Gyros

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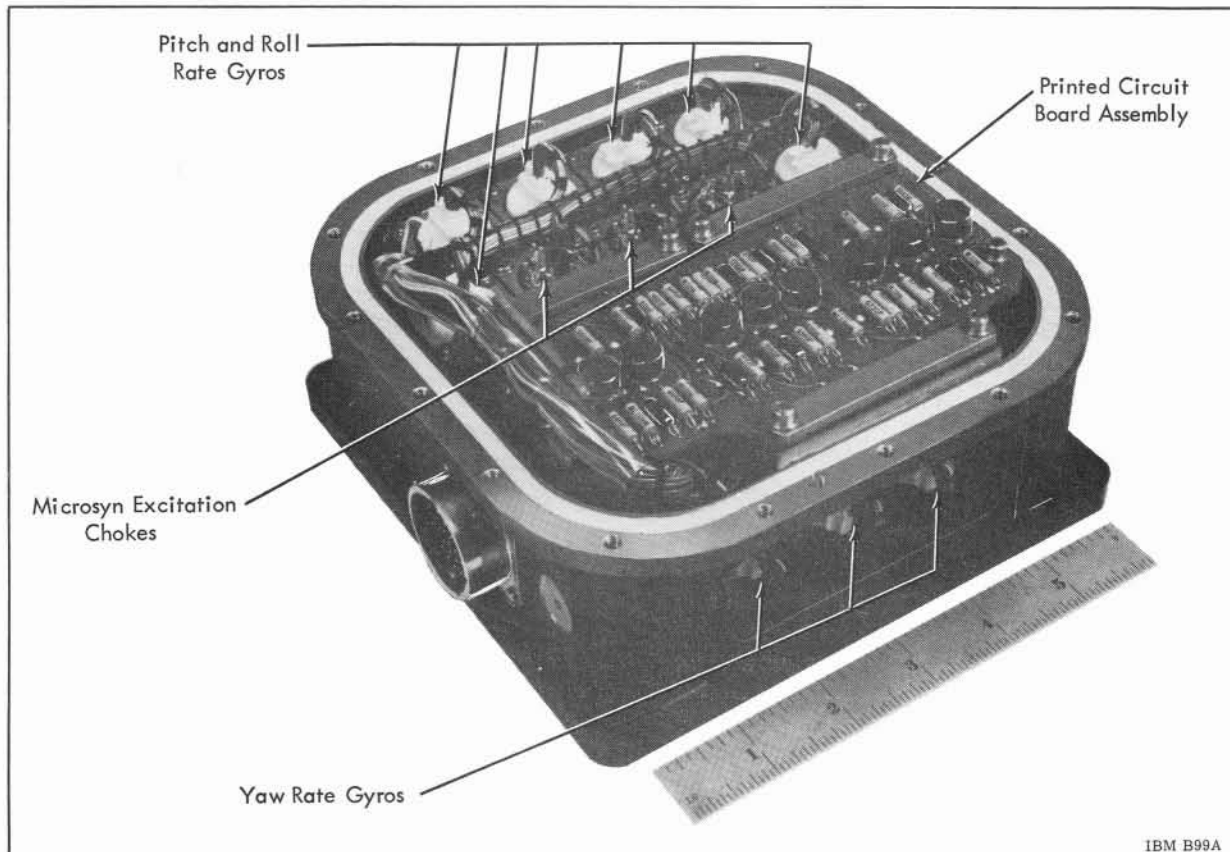
Figure 3.2-2 Control-EDS Rate Gyro Package with Covers Removed

and switching logic selects a good demodulator output. The dc outputs (0 - 45 volts) of each channel are used as inputs to the Flight Control Computer. When the dc output of the two compared demodulators differs by a preset amount, a relay selects the output of the spare demodulator as the input to the Flight Control Computer. The relay remains in this position until reset by an external command signal. Closure of the relay is indicated by a telemetry output.

Each rate switch receives the 400 Hz output signal from one gyro and energizes a relay when the amplitude of the signal exceeds the predetermined magnitude. This relay generates an EDS signal which is indicated by the telemetry.

The angular velocities of the gyro rotors are measured by wheel speed sensor circuits which provide outputs to ESE and telemetry for monitoring purposes. The input signals to the speed sensor is a frequency which is proportional to the angular velocity of the gyro

rotor. If the frequency of any one input deviates more than ± 50 Hz, the wheel speed sensor will generate a discrete signal for ESE and telemetry indicating gyro overspeed or underspeed conditions. Output signals to the EDS are provided if any of the rate gyro outputs exceed a predetermined value.

The Control Signal Processor contains three power supplies operating from three separate 28-volt batteries. Each power supply provides 60 Vdc for one roll, yaw, and pitch demodulator, and 28 Vdc for the comparison circuit and one roll, yaw, and pitch rate switch.

Three static inverters, also located in the Control Signal Processor, operate from unregulated battery power to generate closely regulated 26 Vac (400 Hz) signals. Each inverter provides motor and microsyn power to one roll, yaw, and pitch rate gyro, and a reference excitation to one roll, yaw, and pitch demodulator.
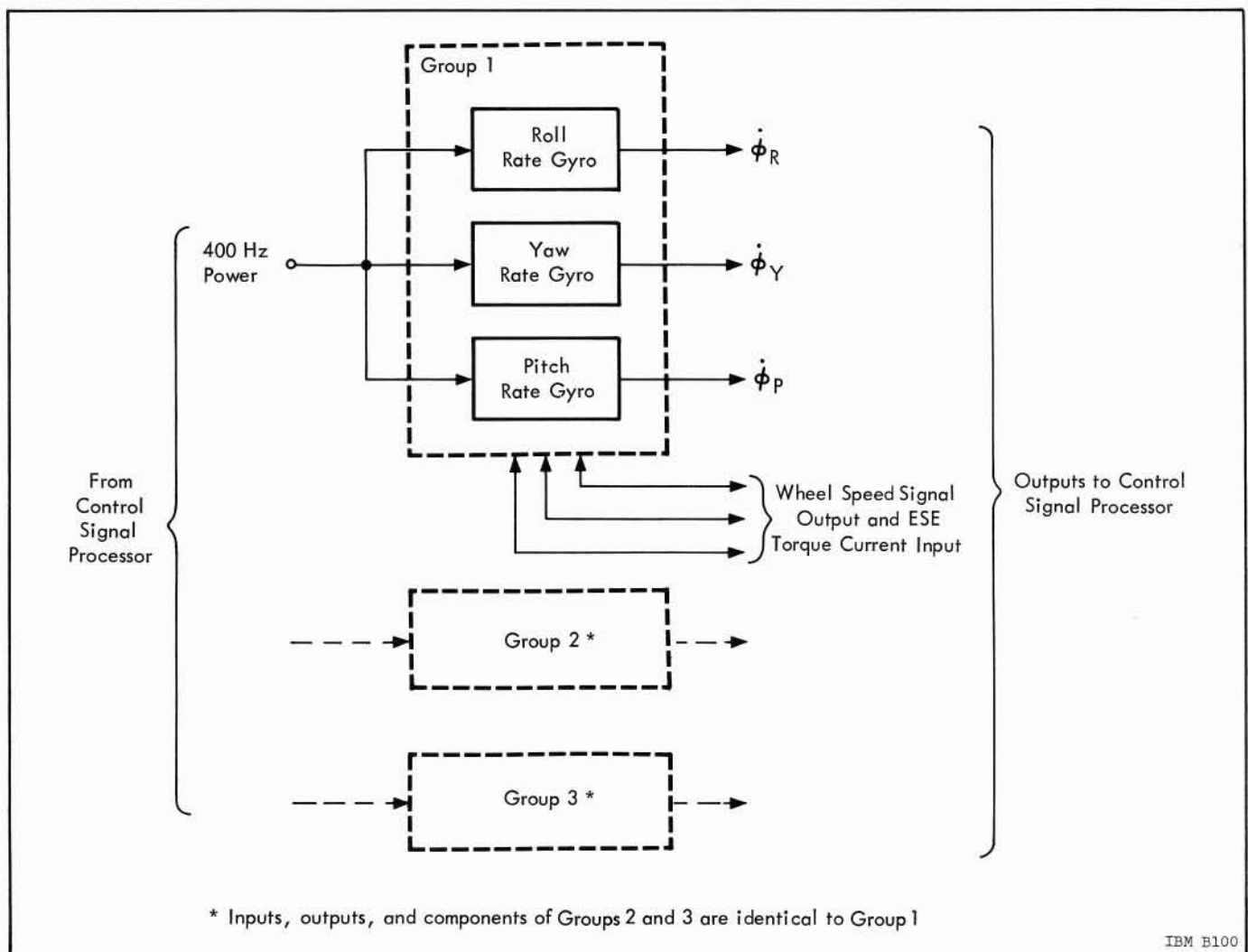
Figure 3.2-3 Control-EDS Rate Gyro Block Diagram
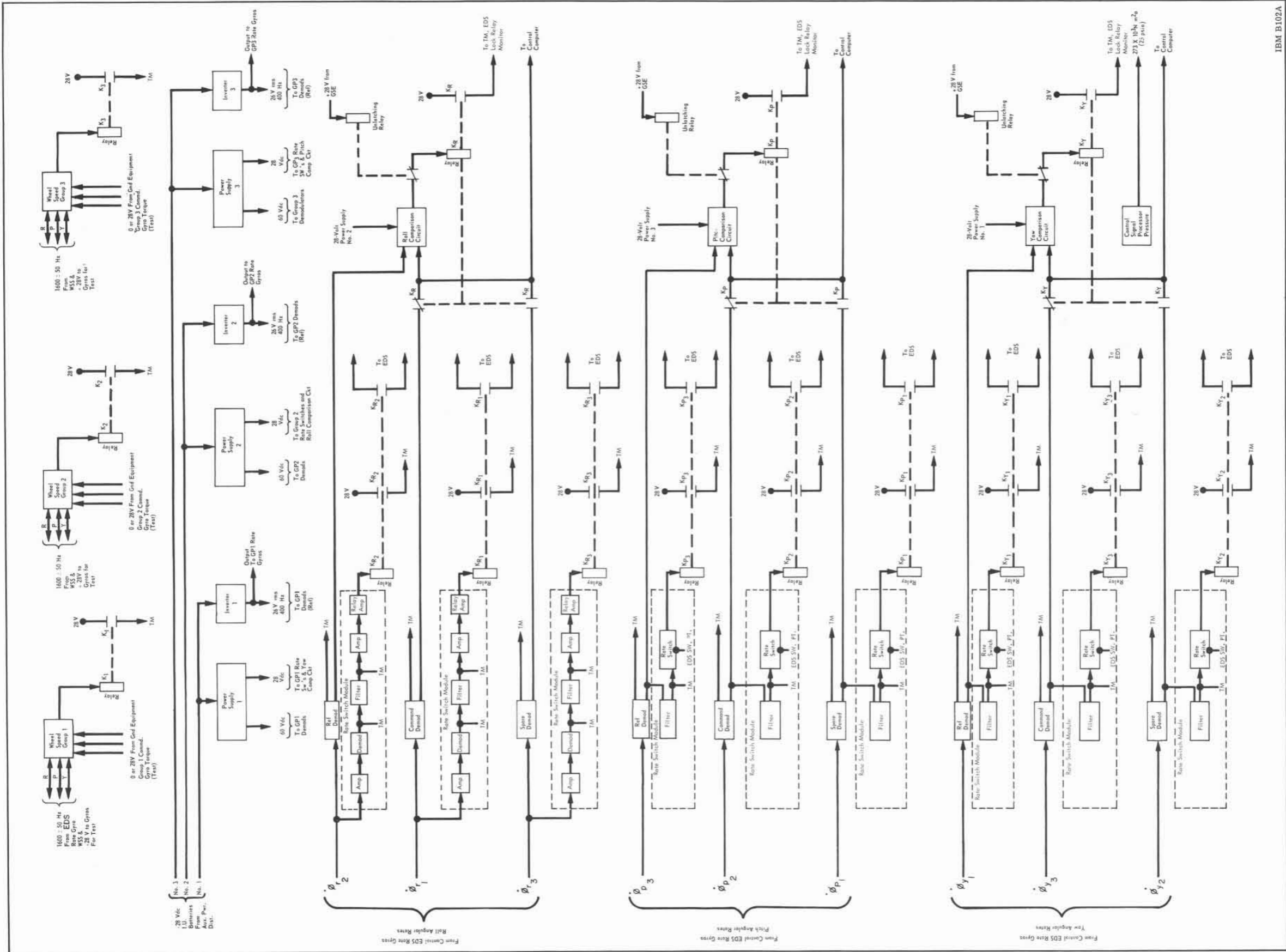
IBM B102A

Figure 3.2-4 Control Signal Processor Block Diagram

Since the Control-EDS Rate Gyro package is located on an IU cold plate, it will be maintained at a temperature which will not require the use of heaters within the package.

3.2.4 CONTROL ACCELEROMETER

Two Control Accelerometers are used during first stage burn on the Saturn IB Vehicles. They are used to sense the lateral acceleration of the vehicle (perpendicular to its longitudinal axis) in the pitch and yaw planes. The output of these instruments is used in the control system to reduce structural loading and to control vehicle deflection due to the lateral winds.

The Control Accelerometer is a linear, spring-mass, fluid damped system with an inductive pickoff. Its excitation is from a self-contained static inverter which converts the 28-volt dc input voltage to a 400-hertz ac voltage (see Figure 3.2-5). The ac output of this sensor is proportional to the applied acceleration in its sensitive axis. This signal is then amplified and demodulated within the assembly to produce a dc output voltage proportional to acceleration. This output is fed to the Flight Control Computer.

The heart of the accelerometer is its spring-mass system. The seismic mass is supported and restrained by a non-pendulous spring suspension system. The springs constrain the mass to pure translational motion with practically no friction and provide extremely low cross-axis sensitivity at the same time.

The spring-mass system is fluid dampened with a silicone fluid. As the accelerometer is subjected to acceleration, the mass moves with respect to the case and causes a positive displacement of the fluid with a resultant damping action. In addition to this effect, the damping fluid provides resistance to shock and vibration. A change in fluid volume with any temperature change is compensated for by an internal bellows.

A special feature of these accelerometers is their self-checking capability. By exciting a separate electro-magnetic coil (force coil), it is possible to deflect the seismic mass by a calibrated amount and thus check the electrical output of the instrument without subjecting it to acceleration. This provides a means of remotely checking the accelerometers after their installation on the vehicle to assure proper operation before flight. Table 3.2-1 lists the accelerometer characteristics.

Table 3.2-1 Control Accelerometer Characteristics

Measuring range	0 to ± 10 m/s^2
Natural frequency	9 Hz
Damping ratio	0.8 $\pm$ 0.2 of critical
Cross axis sensitivity	0.002 g/g
Scale factor	1.0 Vdc/m/s^2
Power	8.4 watts
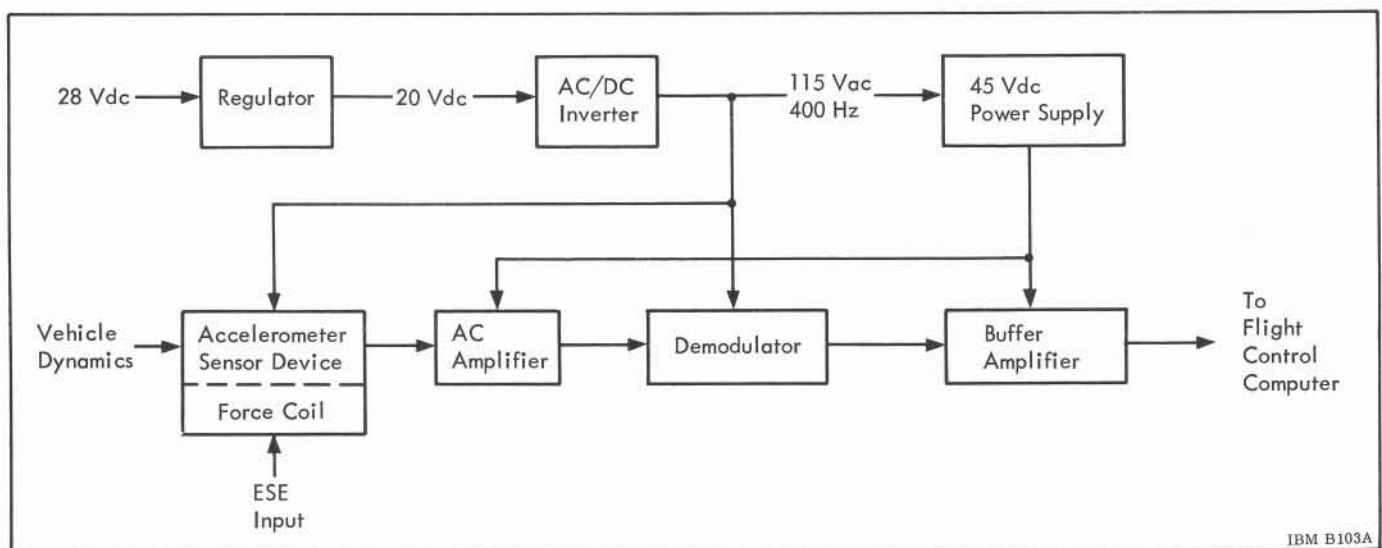
Figure 3.2-5 Control Accelerometer Block Diagram

SECTION 3.3
FLIGHT CONTROL COMPUTER

3.3.1 GENERAL

The Flight Control Computer is an analog device which generates the proper control commands for the engine actuators and the S-IVB auxiliary propulsion system. The computer accomplishes this by processing and combining the analog signals from the LVDA, control sensors, and Apollo Spacecraft.

The Flight Control Computer is equipped with eight input channels to accept eight different control signals listed in Table 3.3-1. The essential elements of the Flight Control Computer (for powered flight operation) are shown in the simplified block diagram, Figure 3.3-1. The filters and Control Attenuation Times (CAT's) provide the proper time-variable adjustment of the gain factors a_0 and a_1 (also g_2 for Saturn IB) during flight. Filter networks shape the control signals to achieve phase and gain stabilization. CAT's are used only on Saturn IB Vehicles since lateral acceleration measurements are not included in the Saturn V control scheme.

The processed signals are then summed and amplified by either servo or spatial amplifiers. These signals are the control signals for the engine servo actuators which control the thrust vector deflection angle.

The Flight Control Computer also generates the control signals for operation of the S-IVB auxiliary propulsion system which provides roll control during S-IVB powered flight and complete attitude control during coast flight. A simplified diagram of the Flight Control Computer configuration during coast flight is given in Figure 3.3-2. The attitude error signal (ψ) is provided by the LVDC/LVDA or the Apollo Spacecraft. Signals from the spacecraft are limited in the Flight Control Computer while the signals supplied by the LVDA have been limited in the LVDC/LVDA. Angular rate signals ($\dot{\phi}$) are obtained from the Control-EDS Rate Gyros in the IU. The spatial amplifiers generate the control signals (ON-OFF) for the auxiliary propulsion system attitude control engines.

The Saturn V and the Saturn IB Flight Control Computers are designed according to the same general scheme. The configurations of both computers, however, are somewhat different for the powered flight phases since the Saturn V is a 3-stage vehicle while Saturn IB is a 2-stage vehicle. See paragraph 3.3.2. Both types of computers have the same configuration for S-IVB powered flight attitude control and coast flight attitude control.
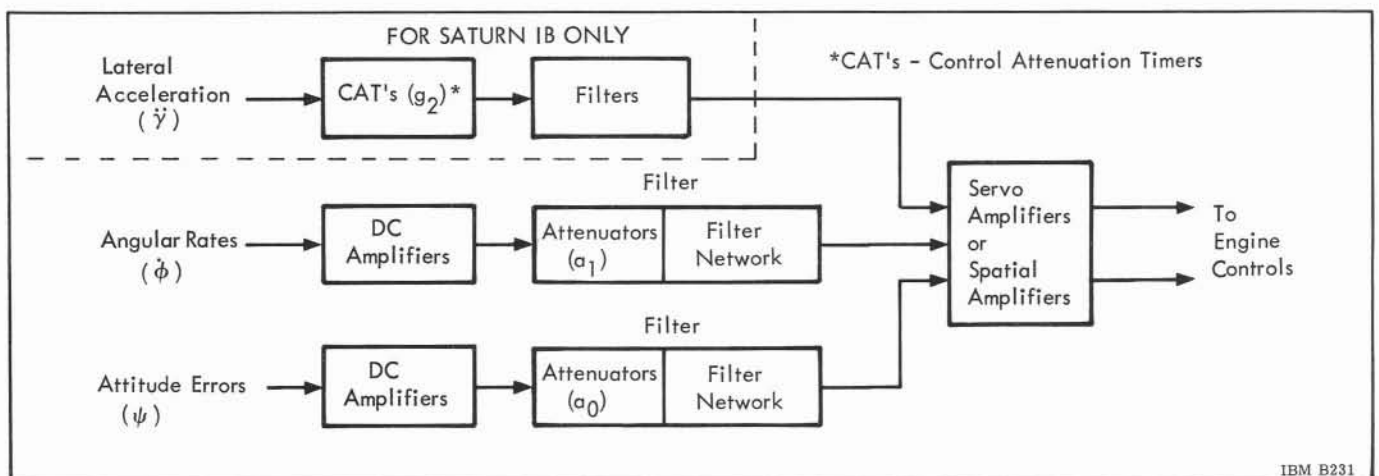
Figure 3.3-1 Simplified Diagram of the Flight Control Computer for Powered Flight

3.3.2 FLIGHT CONTROL COMPUTER OPERATION DURING POWERED FLIGHT

The Saturn IB Flight Control Computer differs from the Saturn V Flight Control Computer in several areas.

The Saturn IB Flight Control Computer contains a channel for Control Accelerometer signals (for first-stage powered flight). One type of servo amplifier is used to control the engine actuators of the S-IB Stage. Another type of servo amplifier (the same used in Saturn V) controls the S-IVB engine actuators. The engine actuator control loops for the S-IB Stage use electrical feedback. The computer contains amplifiers, filters, and the necessary switching devices for two vehicle stages.

The Saturn V Flight Control Computer has no channels for Control Accelerometer signals. The same type of servo amplifier is used to control the engine actuators in each stage (S-IC, S-II, and S-IVB). The computer contains amplifiers, filters, and the corresponding switching relays for three vehicle stages. The control loops for the engine actuators use mechanical feedback.

Table 3.3-1 Input Signals to the Flight Control Computer

Input Channel	Symbol	Gain* Factor	Signal Sources
Lateral Acceleration			
Pitch	$\ddot{\gamma}_P$	$\left.\rule{0pt}{16pt}\right\}g_2$	Saturn IB Control Accelerometers
Yaw	$\ddot{\gamma}_Y$		
Angular Rate			
Pitch	$\dot{\phi}_P$	$\left.\rule{0pt}{22pt}\right\}a_1$	Control-EDS Rate Gyros
Yaw	$\dot{\phi}_Y$		
Roll	$\dot{\phi}_R$		
Attitude Error			
Pitch	ψ_P	$\left.\rule{0pt}{22pt}\right\}a_0$	LVDA/LVDC or Spacecraft
Yaw	ψ_Y		
Roll	ψ_R		

*Time-varying functions that determine which of the three terms of the control equation will be dominant. The control equation is defined as:

$$\beta_c = a_0\,\psi + a_1\,\dot{\phi}\ (+\ g_2\,\ddot{\gamma}\ \text{for Saturn IB})$$
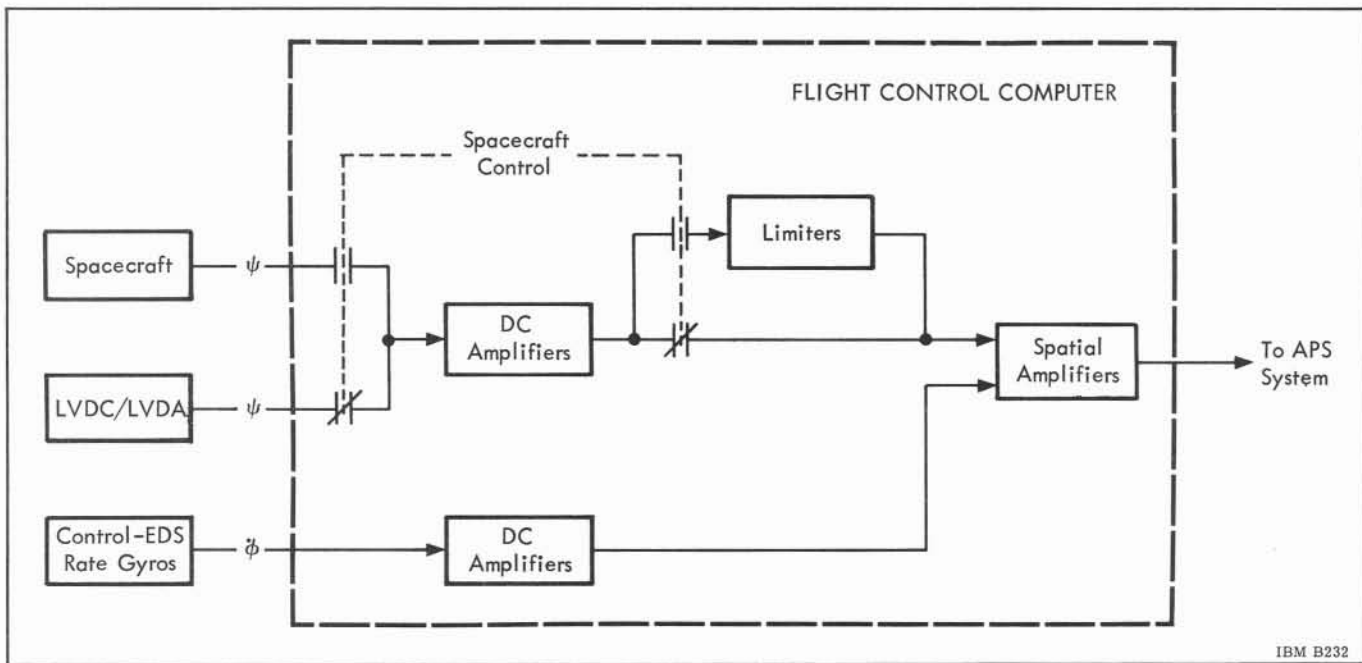
Figure 3.3-2 Simplified Diagram of the Flight Control Computer for Coast Flight

SATURN IB FLIGHT CONTROL COMPUTER

Figure 3.3-3 is a block diagram of the Saturn IB Flight Control Computer. During first-stage powered flight, each input signal to the computer passes through a dc amplifier and filter before it enters the servo amplifiers.

Attenuation (gain control) is accomplished by either of two methods: by motor-driven, cam-controlled potentiometers called Control Attenuation Timers (CAT's) or relay-switched, fixed-resistor networks. The CAT's provide attenuation that varies continuously as a function of time, while the relay-switched resistor networks change attenuation in discrete steps at predetermined time intervals. These relays are commanded by signals from the LVDC/LVDA via the IU Switch Selector and are often referred to as "switchpoints". Each filter block shown in the figure represents a relay-resistor (attenuator) network and its associated filter network. A different set of attenuators and filters is used during the S-IVB powered flight phases since the dynamic characteristic of the vehicle is changed after separation of the S-IB Stage and aerodynamic forces no longer exist. The required filters for each stage are selected by relays according to the flight program. No filters are used for operation of the S-IVB auxiliary propulsion system (during coast flight and for roll control during powered flight). The actual filter networks are passive RLC networks which provide gain and phase stabilization in the control loop to prevent undesirable signal frequencies from affecting the attitude control.

In the CAT device, a rocker arm drives two coupled potentiometers which are electrically connected to the appropriate channels. The outputs from the potentiometers are time-varying functions of the input signals; i.e., the $\ddot{\gamma}$ signal inputs to the attenuator appear at the attenuator output modified by g_2. The cam is driven by a 115V, 400-cycle, single-phase, synchronous motor. The cam speed is determined by a gear train arrangement.

A typical Saturn IB time program for a_0, a_1, and g_2 is shown in Figure 3.3-4. The gain factor (g_2) has a maximum value during the time period of high aerodynamic pressure, thus emphasizing the lateral acceleration control signals. These Control Accelerometer signals are used during the S-IB flight phase only. The gain coefficients (g_2) for Control Accelerometer signals are varied by CAT circuits.

The other gain factors (a_0 and a_1) are changed by relay switching of resistors.

The Flight Control Computer contains eight servo amplifiers for the actuators of the S-IB Stage and six servo amplifiers for the actuators of the S-IVB Stage. The servo amplifiers are magnetic amplifiers which sum and amplify the filtered and scaled control signals to generate the control signals for the engine actuators. The output stage of the amplifier can provide up to 12 milliamperes (S-IB) or 50 milliamperes (S-IVB) of current to the torque motor-operated control valve of each hydraulic servo actuator. A portion of this output current is returned through a feedback network to provide required accuracy in gain and linearity.

There is one servo amplifier for each engine actuator in the S-IB Stage (i.e., four pitch servo amplifiers and four yaw servo amplifiers). The three yaw output control signals (ψ_Y $\dot{\phi}_Y$ $\ddot{\gamma}_Y$) to the Flight Control Computer are fed into each of the four yaw amplifiers. Correspondingly, the three pitch input channels are connected to each of the four pitch amplifiers. The two roll input signals (ψ_R and ϕ_R) are sent to all eight S-IB servo amplifiers since it is necessary to deflect the engines in pitch and yaw simultaneously for execution of a roll maneuver.

The six servo amplifiers for the S-IVB powered flight are used in a triple redundant configuration to control the two servo actuators of the single J-2 engine. The output signals of two yaw servo amplifiers are compared, and if they agree, one is used to control the yaw actuator. If those two signals disagree, the output of the third yaw servo amplifier is switched to the actuator. The same scheme applies to the pitch channel.

To achieve APS thruster synchronization of roll control during S-IVB powered flight, the output signal of the yaw-roll I spatial amplifier configuration feeds both control relay packages. Although not shown in Figure 3.3-3, the yaw-roll II signals are interrupted during this period and will not be used until the coast flight control operation.

SATURN V FLIGHT CONTROL COMPUTER

Figure 3.3-5 is a block diagram of the Saturn V Flight Control Computer. The attitude error sig-
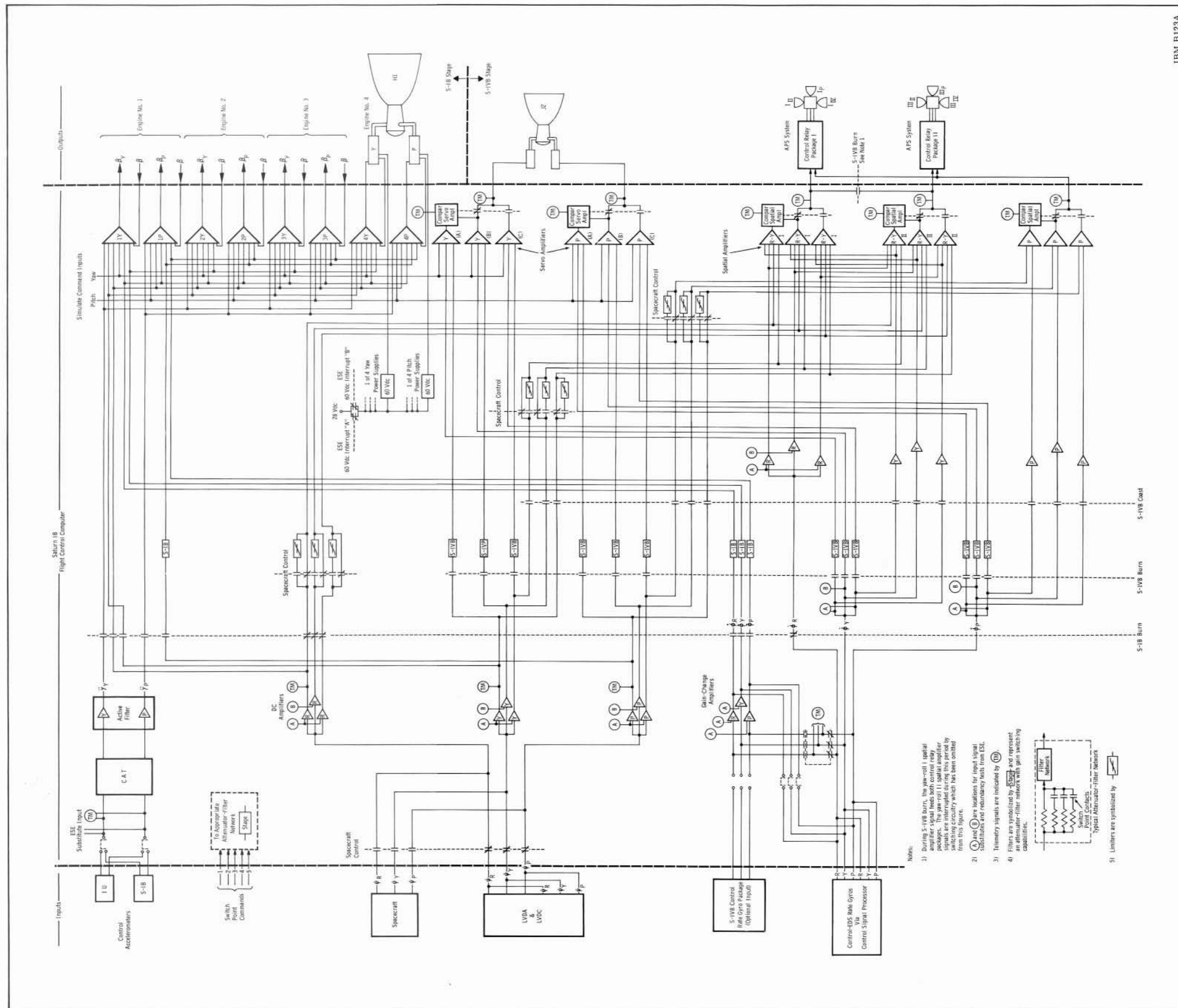
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Figure 3.3-3 Saturn IB Flight Control Computer Block Diagram

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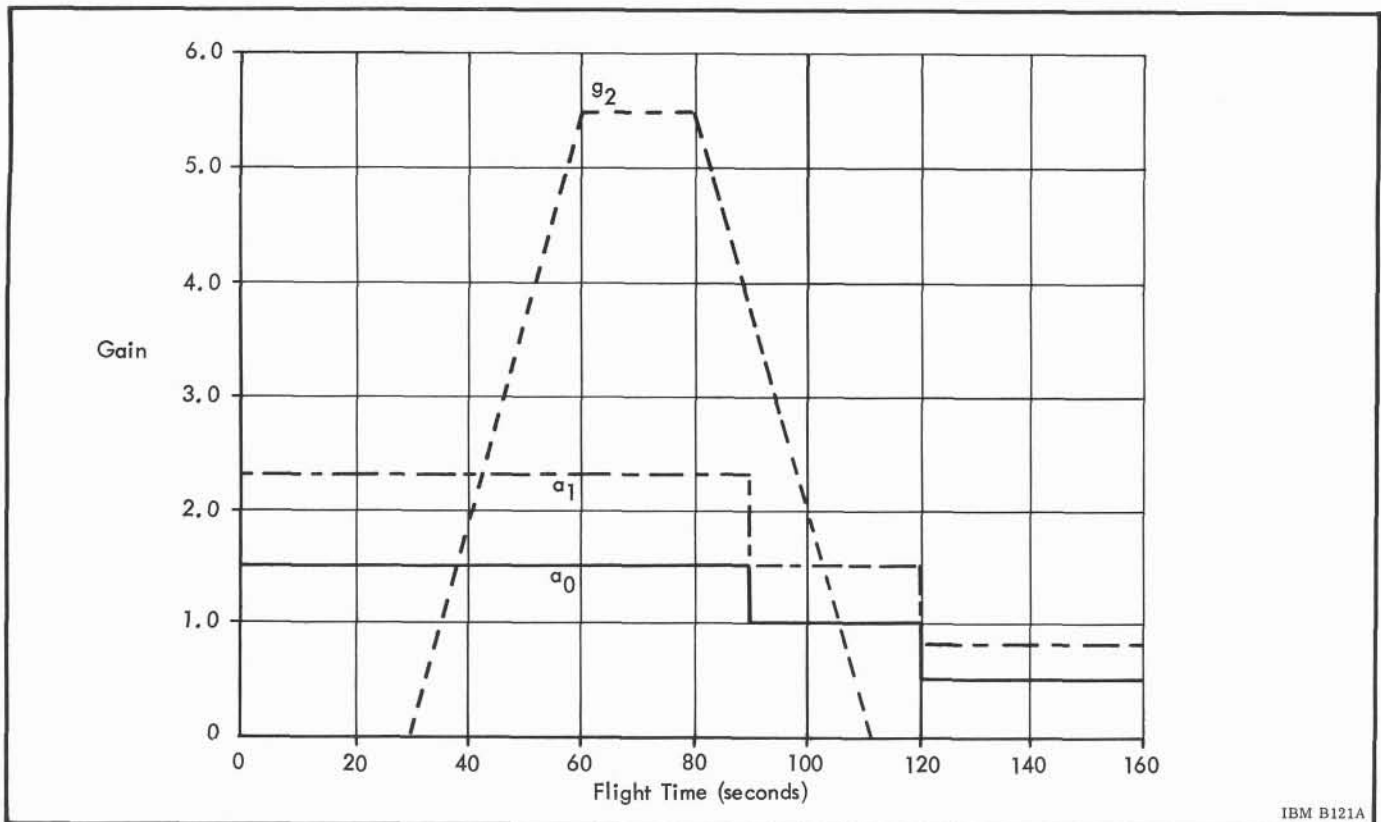
Figure 3.3-4 Typical Saturn IB Gain Program for a_0, a_1, g_2 Coefficients

nals (ψ), originating either from the LVDA or the spacecraft, are switched by a quad redundant relay configuration to three scaling amplifiers in each of the three channels (yaw, pitch, and roll). A different set of filters is switched in for each powered flight phase (S-IC, S-II, S-IVB). As in the Saturn IB Flight Control Computer, filters provide the proper adjustment of the gain coefficients a_0 and a_1.

The attitude error signals and angular rate signals from the filters are summed in the magnetic servo amplifiers and the resulting signal is used to generate the actuator control signals. The Flight Control Computer contains eight servo amplifiers of the same type. Each amplifier provides a maximum output current of 50 milliamperes to torque the motor-operated control valves of the actuators (see Figure 3.3-6). Each servo amplifier controls one of the eight engine actuators in the S-IC Stage or S-II Stage.

The roll signals (ψ_R and $\dot{\phi}_R$) are distributed to all eight servo amplifiers. The yaw signals (ψ_Y and $\dot{\phi}_Y$) are connected to all four yaw servo amplifiers and the pitch signals (ψ_P and $\dot{\phi}_P$) are connected to all four pitch servo amplifiers. The switching of input signals, filters, and servo amplifiers in the Flight Control Computer is controlled by the LVDC/LVDA through Switch Selector circuitry. See Section 4.2. Since the S-IVB Stage has only two engine actuators, it should be noted that six of the eight servo amplifiers are used in a triple redundant arrangement. The control configuration for third-stage burn is otherwise similar to that described for the S-IVB powered flight phase of Saturn IB Flight Control Computer.

3.3.3 FLIGHT CONTROL COMPUTER OPERATION DURING COAST FLIGHT

The Flight Control Computer configurations of the Saturn IB and Saturn V are identical for coast flight attitude control (Figures 3.3-3 and 3.3-5).

No filters are contained in the signal channels (ψ and $\dot{\phi}$) of the APS. Attitude error signals (ψ) from the spacecraft are limited in the Flight Control Computer between the dc amplifiers and the spatial amplifiers. The limiter circuits provide roll, pitch, and yaw signal limiting and are switched into the circuit only during the spacecraft control mode of operation.
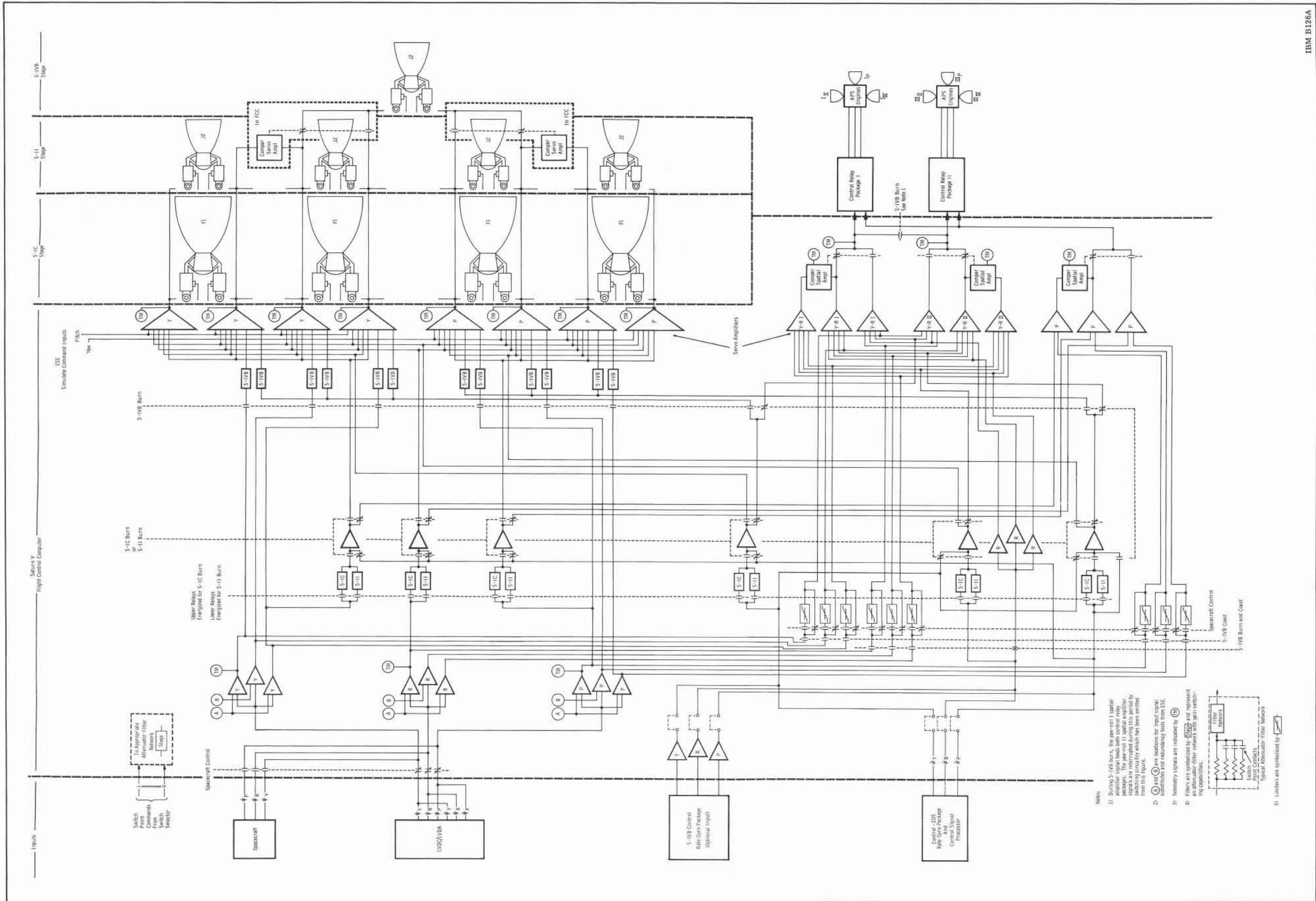
Figure 3.3-5 Saturn V Flight Control Computer Block Diagram

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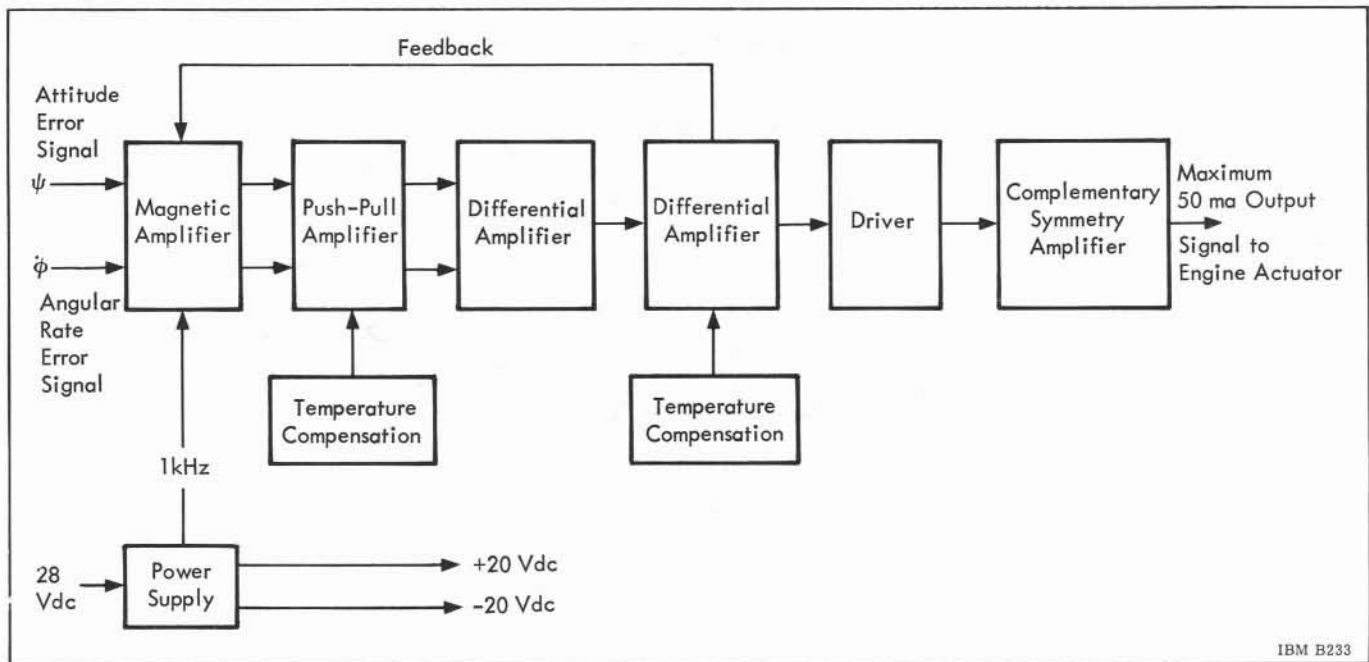
Figure 3.3-6 50-Milliampere Servo Amplifier Block Diagram

The signals from the dc amplifiers are sent directly to the spatial amplifiers. The spatial amplifiers sum the input signals (ψ and $\dot{\phi}$) and compare the resulting signal level with a predetermined threshold level. When the combined control signals exceed the limits set by the threshold, a switching amplifier energizes the relays in the control relay package, which in turn operates the values of the proper attitude control engines.

The Flight Control Computer contains nine spatial amplifiers which are used in a triple redundant configuration (three spatial amplifiers for each pitch, yaw, and roll control). A comparator circuit selects 1 of 2 amplifiers out of the 3 triple-redundant amplifiers in each channel. The spatial amplifiers for pitch control switch either the (+) pitch or the (-) pitch nozzle ON. They require only the two input signals (ψ_P and $\dot{\phi}_P$). The remaining 2 groups of spatial amplifiers both control yaw and roll maneuvers. Each of these amplifiers receives roll and yaw input signals (ψ_R, ψ_Y, $\dot{\phi}_R$, $\dot{\phi}_Y$) since the same nozzles are used for yaw and roll control. The control relay package which operates the valve coils is discussed in Section 3.4.

3.3.4 REDUNDANCY IN THE FLIGHT CONTROL COMPUTER

The attitude rate signals ($\dot{\phi}$) and the attitude error signals (ψ) enter the Flight Control Computer via separate duplex lines. The duplex lines continue until they branch into the three scaling amplifiers as shown in Figure 3.3-7. (The switchover between attitude error signals from the LVDA or the spacecraft is accomplished by relays arranged in a quad redundant configuration.)

For the S-IVB powered flight and coast flight attitude control, the Flight Control Computer has a triple redundant configuration in each channel. For each of the six input control signals (ψ_P, ψ_Y, ψ_R, $\dot{\phi}_P$, $\dot{\phi}_Y$, $\dot{\phi}_R$), there are three redundant, completely independent channels. Each channel contains individual amplifiers, filters (if used), and relays. These triple redundant signal channels are connected to a triple redundant arrangement of servo amplifiers or spatial amplifiers. The outputs of two of the three triple redundant servo (or spatial) amplifiers are compared and, if they agree, one output is applied to the engine actuator (control relay package). If the compared signals do not agree, the output of the third servo (spatial) amplifier is used.
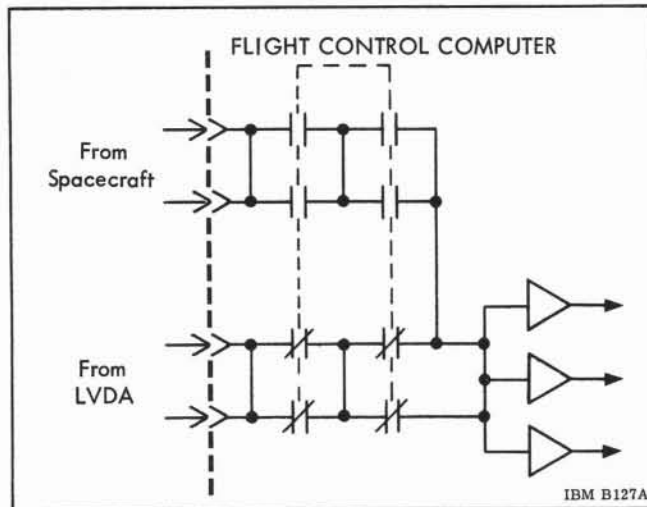
Figure 3.3-7 Redundant Cabling in the Flight
Control Computer

SECTION 3.4
ENGINE CONTROLS

3.4.1 ENGINE SERVO ACTUATORS

The angular direction of the thrust vector of Saturn IB and V engines is controlled by swivelling the gimbal-mounted propulsion engines. This control is obtained by linear hydraulic servo-actuators in the pitch and yaw planes.

The actual arrangement of the actuators for the multi-engine stages and the single-engine S-IVB Stage of the Saturn Launch Vehicle is shown in Figure 3.4-1. Independent hydraulic systems for each gimbaled engine are used to provide the following advantages:

- Flight system can be qualified by single-engine test.

- One system can be verified at a time.

- Malfunction and replacement can be isolated to 1 system.

- Post flight performance comparisons are easily accomplished and problem areas are quickly isolated.

- Redundant capability.

For the multi-engine vehicle stages, the redundant capability is further increased by sizing the swivel range so that, despite the failure of one control engine, the remaining gimbaled engines can maintain satisfactory control. The failure modes may be:

- Loss of thrust (engine out).

- Loss of hydraulic pressure (hydraulic system failure).

- Any actuation malfunction caused by servo valve failure, failure in signal lines, or Flight Control Computer output stage.

During operation of the rocket engines in the S-IC and S-II Stages, the hydraulic power supply is provided by a hydraulic pump driven by the engine turbine and an accumulator-reservoir to cover peak loads. An auxiliary pump, driven by an electric motor, provides hydraulic pressure for static checkout of the hydraulic system.

For the first-stage engines, the hydraulic pressure is obtained directly from the fuel turbopump; as the fuel itself is used as the hydraulic fluid. The available low operating pressure requires a large servo valve and actuator piston area; however, the pressurization system is very simple and even the accumulator-reservoir can be omitted as a result of the almost unlimited availability of pressurized fuel.

The transformation of the electrical signal from the Flight Control Computer, which commands the angular deflection (β) of the engine into a linear displacement, is performed by servo actuator assemblies. The servo valves are four-way, flow-control devices, with internal dynamic pressure feedback and mechanical feedback of the actuator piston position. Figure 3.4-2 is a symbolic representation of the valve. The servo valve operates in the following manner. The error signal from the Flight Control Computer is applied to the torque motor coil which displaces the flapper valve and restricts the flow of hydraulic fluid in one branch (e.g., the left one in Figure 3.4-2). The increasing pressure in that branch drives the second stage power spool to the right, which in turn allows hydraulic pressure build up on the left side of the actuator piston, causing a linear displacement of the actuator to the right. The mechanical feedback linkage displacement places a counter torque on the flapper valve. When this feedback torque equals the applied torque from the servo valve torque motor, no further motion of the actuator can occur. The potentiometer measures the actuator position for telemetry purposes.

In the S-IB Stage servo actuators only, an electrical potentiometer is used as a feedback transducer. It applies a "β" feedback signal to the Flight Control Computer servo amplifier. The mechanical feedback provides higher reliability, which results primarily from the elimination of the feedback potentiometer and its associated wiring, connections, and power supply, and reduction in corresponding failure

modes. Further, the mechanical feedback has an important advantage in that the actuator goes to a null position ($\beta = 0$) if the "β" command signal from the Flight Control Computer fails in the zero level.

The characteristics of the hydraulic servo actuator systems are given in Table 3.4-1.

3.4.2 AUXILIARY PROPULSION SYSTEM

The APS system, shown in Figure 3.4-3, consists primarily of spatial amplifiers, control relay

packages, and attitude control nozzles. The figure is not detailed since the APS system operation has been generally explained and illustrated in previous sections. The purpose of this section is to explain the design characteristics of the system and to examine component functions.

SPATIAL AMPLIFIERS

The Flight Control Computer contains 3 sets of triple-redundant spatial amplifiers. One set of amplifiers is used during S-IVB powered flight, and all three sets are used for coast flight control. For
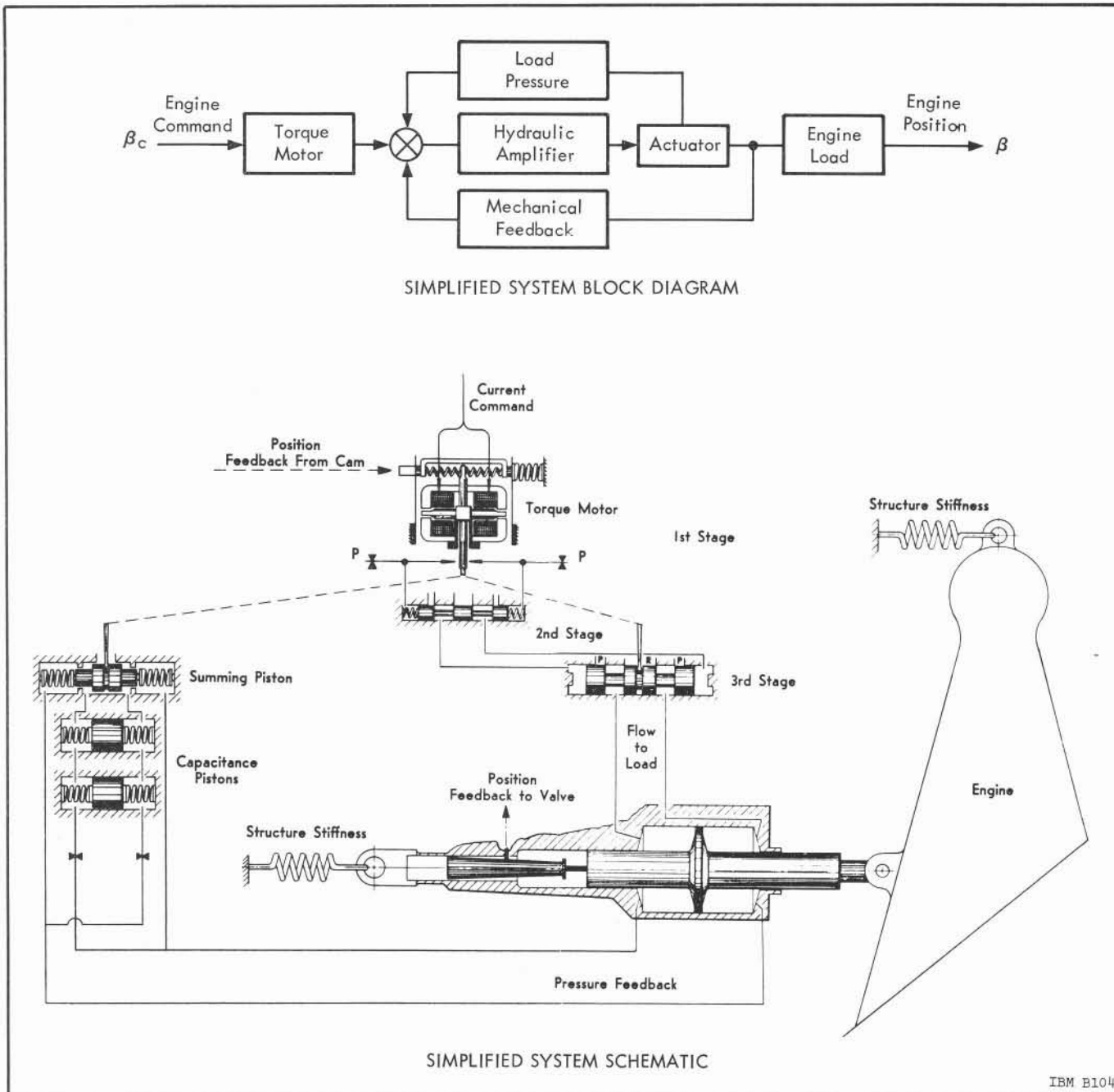
Figure 3.4-1 Hydraulic Actuator System
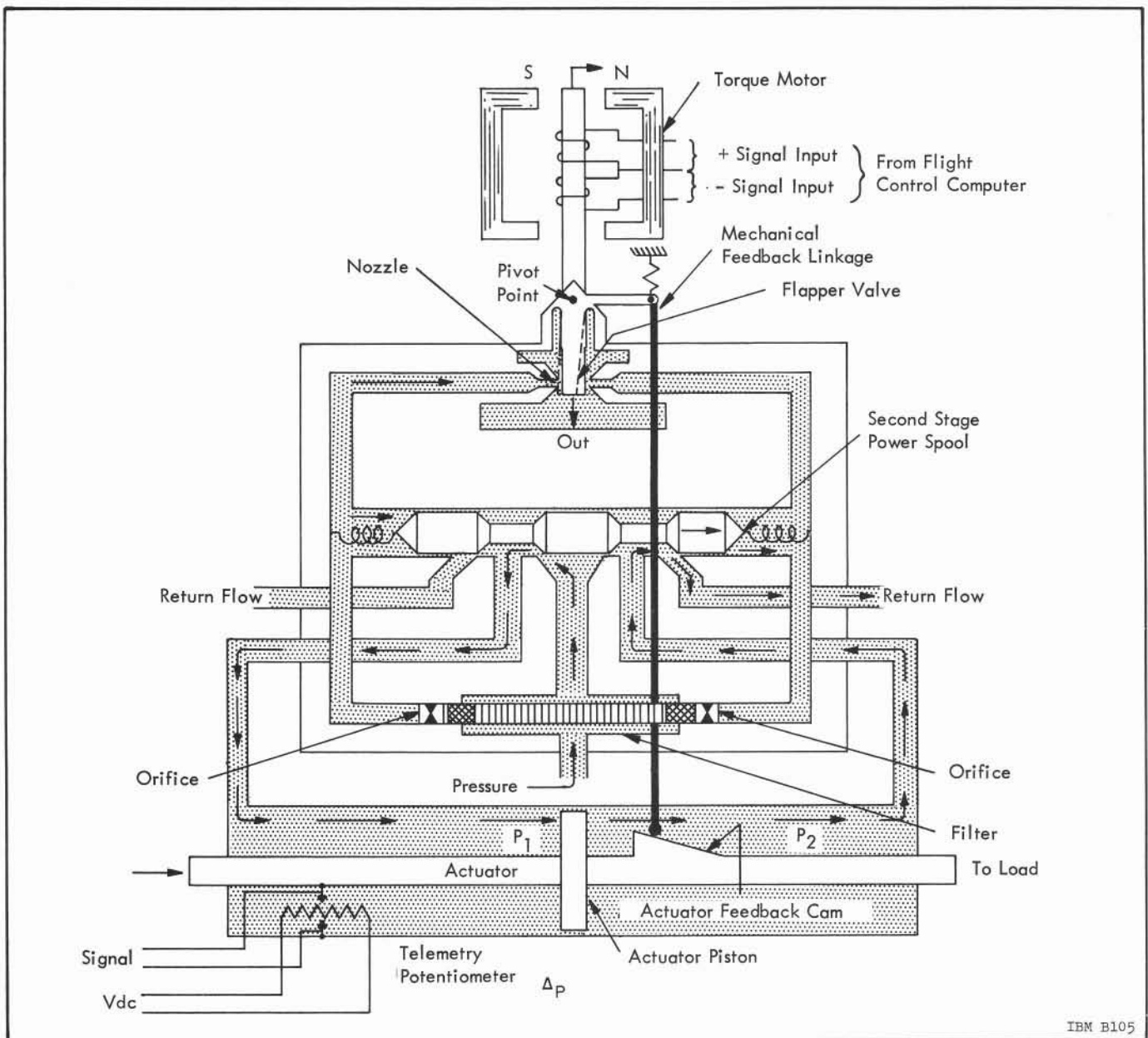
Figure 3.4-2 Flow-through Valve and Actuator with Mechanical Feedback

S-IVB powered flight, one set of spatial amplifiers (yaw-roll I) is used to feed both control relay packages. This scheme allows synchronized firing of the attitude control nozzles (thrusters) for roll attitude manuevers. The signals for the other set of spatial amplifiers used in powered flight (yaw-roll II) are interrupted. The switching circuitry used to achieve this configuration is referred to as the "spatial synchronizer" and is located within the Flight Control Computer. During coast flight, the control relay packages operate independently.

The purpose of these amplifiers is to develop ON-OFF control signals for the S-IVB thruster noz-

zles. The output signal is proportional (in duration) to the yaw, pitch, and roll error signals generated by the LVDC/LVDA or spacecraft. A block diagram of the spatial amplifier is shown in Figure 3.4-4.

Attitude and rate signals are applied to the magnetic amplifier where they are algebraically summed. The output signal of the magnetic amplifier is then amplified by 2 differential amplifier stages. Depending upon the amplitude and polarity of the signal, one of the two schmitt triggers will change state. The triggered signal is then used to drive relays within the Control Relay Unit. The relays transmit stage power to fuel and oxidizer solenoid valves of the commanded hypergolic engine(s).

Table 3.4-1 Servo Actuator Characteristics (Design Goals)

Parameter	S-IC F-1 Engine	S-II J-2 Engine	S-IVB J-2 Engine	S-IB H-1 Engine
Engine				
Moment Arm				
meters	1.620	0.301	0.295	0.699
inches	63.8	11.9	11.6	27.5
Moments of Inertia				
kg m^2	42,000	1,990	1,970	750
slug ft^2	56,760	2,690	2,660	1,015
Servo Actuator				
Piston Area cm^2	368	83.9	76	32.3
in^2	57	13	11.78	5
Stroke cm	29.2	7.7	7.4	9.7
inches	11.5	3.02	2.9	3.82
Supply Pressure				
N/cm^2	12.4x10^6 to 15.2x10^6	24.2x10^6	25.2x10^6	20.7x10^6
psig	1,800-2,200	3,500	3,650	3,000
Torque Motor Current				
ma	50	50	50	12
System				
Angular Deflection				
degrees	±5.15	±7	±7	±7.99
Angular Velocity				
deg/s	5	9.6	8.0	
Load Resonance	8.0	7.9	7.1	

The duration of the output control signal is determined by comparison of the input error signals (ψ and $\dot{\phi}$) and cutoff feedback signal. The greater the amplitude of the input signals, the longer time it will take the cutoff feedback signal to overcome the input error signals. This action results in proportionately longer duration output signals.

To this point, the discussion of the auxiliary propulsion system has been concerned with the pitch channel. The yaw and roll channels are similar in all details except that their combined error signals are mixed to ensure minimum fuel consumption. The yaw-roll mixing scheme is designed so that opposing engines will not fire simultaneously. The scheme also prevents one engine from adding to an error already existing in another channel.

Deadbands are introduced within the spatial amplifier circuit (and APS system in general) to conserve fuel and avoid "oversteering". In effect, a limiting tolerance is built into the system which will allow an attitude adjustment to be made by the APS system only if the adjustment is over a certain pre-determined level. A typical roll deadband is shown in Figure 3.4-5.

The operation of the auxiliary propulsion system during coast phase is illustrated in Figure 3.4-6. The operation starts at point A, which represents a given attitude error (ψ) and attitude rate ($\dot{\phi}$) of the vehicle. The end state is the limit cycle within the deadband and about the commanded attitude angle.

CONTROL RELAY PACKAGE

Each control relay package consists of 12 miniature, double-pole, double-throw relays. Two such packages are used to control the S-IVB attitude control engine valve coils. These valve coils are an integral part of the engine and control the fuel and oxidizer poppet valves. The relays are driven in groups of eight from the three S-IVB attitude control
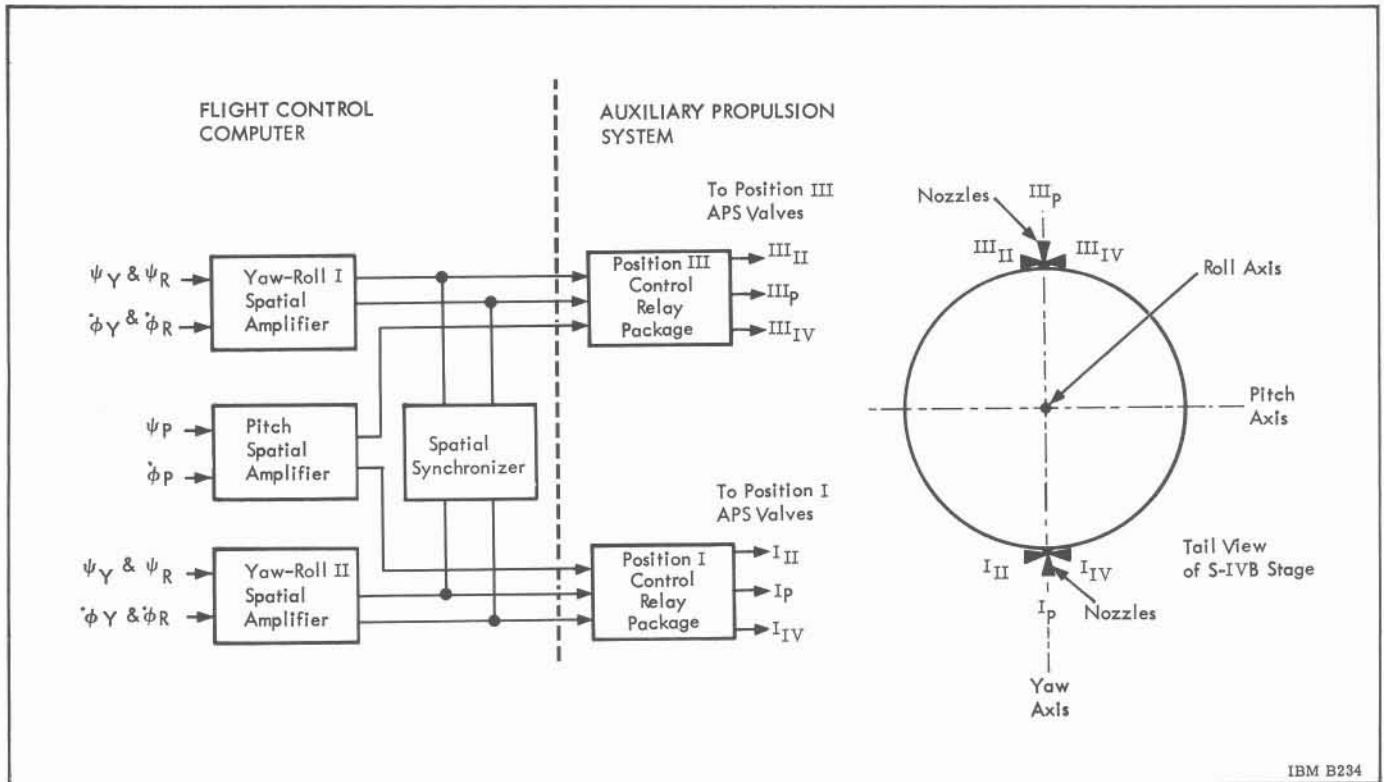
Figure 3.4-3 Auxiliary Propulsion System Block Diagram
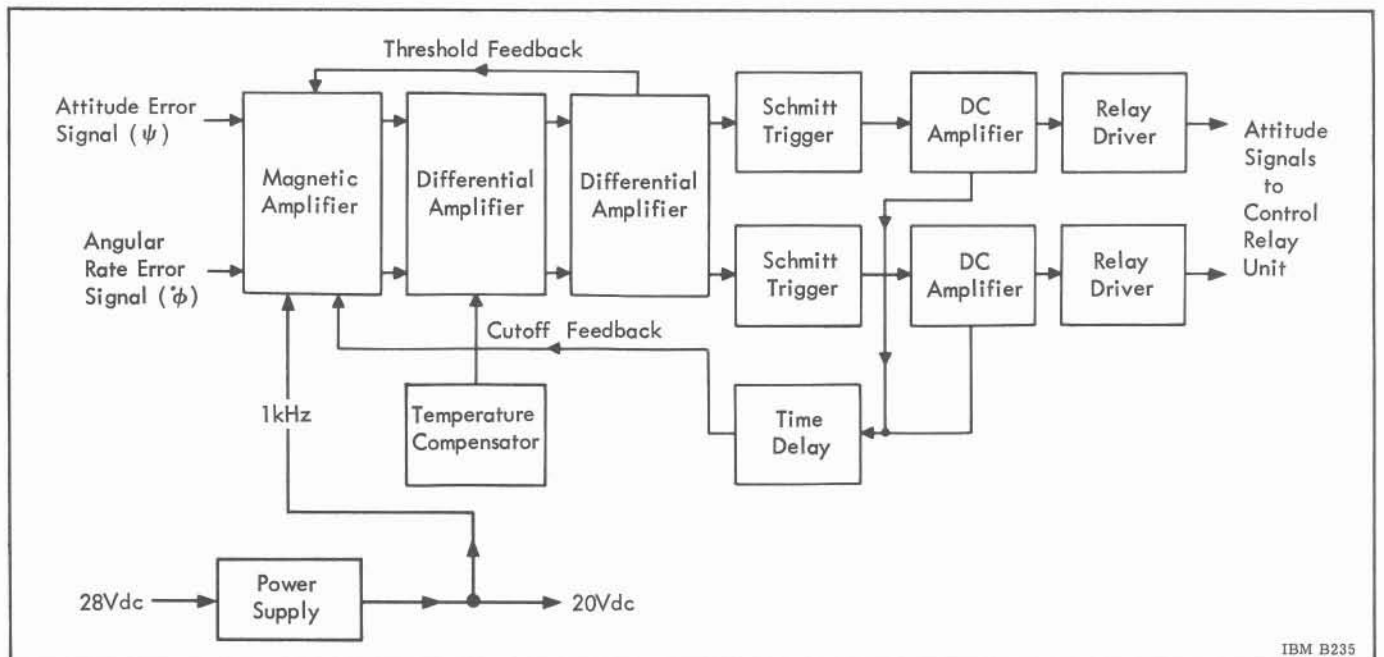
Figure 3.4-4 Spatial Amplifier Block Diagram

spatial amplifiers in the Flight Control Computer.
Figure 3.4-7 shows the arrangement of the relays,
valve coils, and quad-redundant fuel and oxidizer
poppet valves associated with 1 amplifier in the pitch
or yaw axis.

As an example, if the attitude control signal
requires that nozzle I_{II} be turned ON, the spatial
amplifier energizes coils K1 through K4, which closes
the associated contacts (1, 1', etc). Valve coils 1,
1', etc., are energized and open fuel valves 1, 2, 3,
and 4 and oxidizer valves 1' and 2', 3' and 4' in the
quad valve network. This turns on nozzle I_{II}.

The control relay package was designed to take
advantage of the redundant valve configuration by pro-
viding failure isolation from the predominant modes
of relay failure (open-circuited coils, failure-to-make
contacts, and welded contacts). This relay redundancy
can be shown by considering relay 1 as an example.
When relay coil K1 is energized, it closes contacts
1 and 1' which causes fuel valve 1 and oxidizer valve
1' to be open. An open in coil K1 or a failure of con-
tacts 1 and 1' to close results in no operation of fuel
valve 1 and oxidizer valve 1'. However, this does
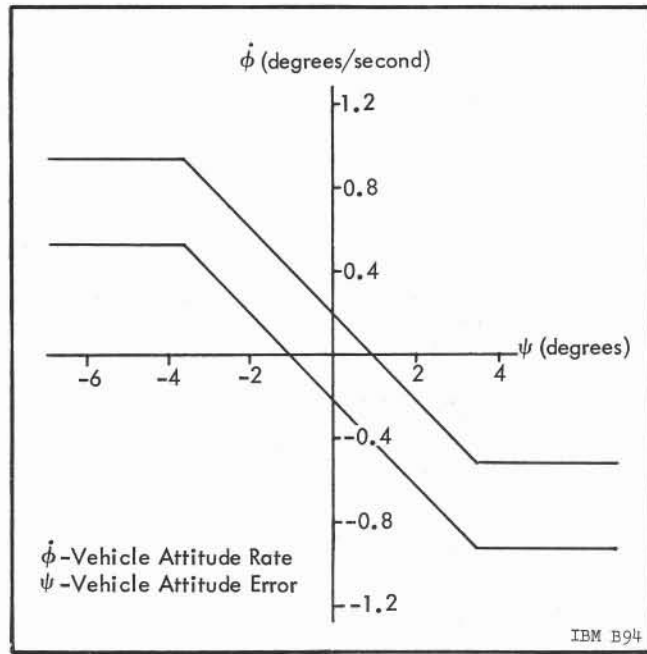
Figure 3.4-5 Typical Roll Deadband for S-IVB
Auxiliary Propulsion System
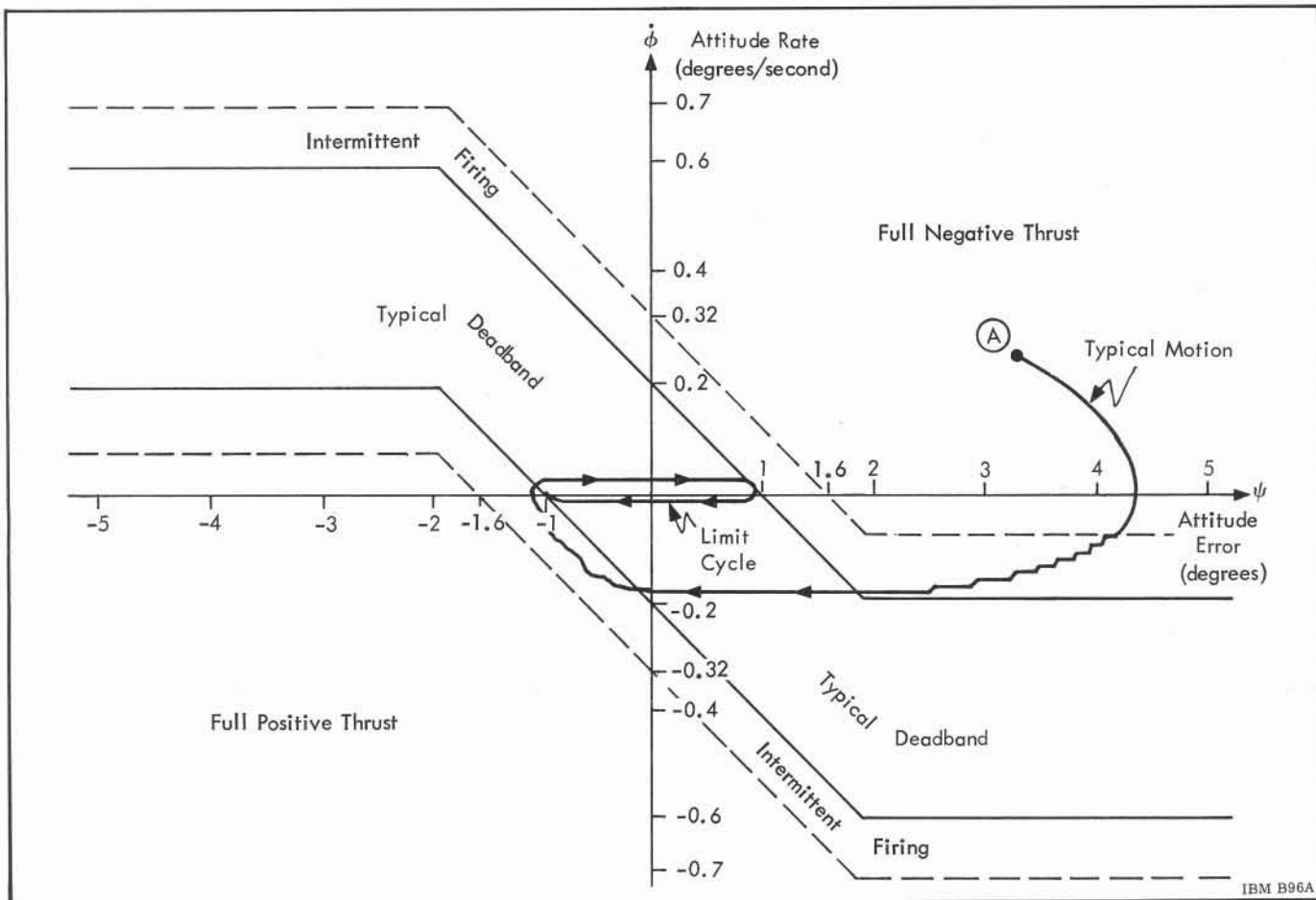
Figure 3.4-6 Typical Coast Phase APS Operation

not cause a failure of the attitude control system, since the quad-redundant valves provide an operative flow path through valves 2, 4 and 2', and 4'. If, on the other hand, either or both sets of contacts of relay 1 should weld, or otherwise become permanently closed, fuel valve 1 and oxidizer valve 1' would be open. However, control of fuel and oxidizer would be maintained by valves 3 and 3' respectively.

Reliability predictions have shown that 85 percent of relay coil failures are due to an open coil, and 93 percent of contact failures are due to failure to close. The design of the control relay package iso-lates the control system from these predominate failures, as well as welded contacts.

The control relay package was introduced into the design of the S-IVB attitude control system to im-prove the interstage cabling design between the IU Flight Control Computer and the S-IVB valve coils by reducing the current requirements from 12 amperes to 200 milliamperes. Therefore, the control relay package saves about 100 pounds of cabling and re-duces the Flight Control Computer spatial amplifier's power requirements by switching the power to the valve coils in the S-IVB Stage.
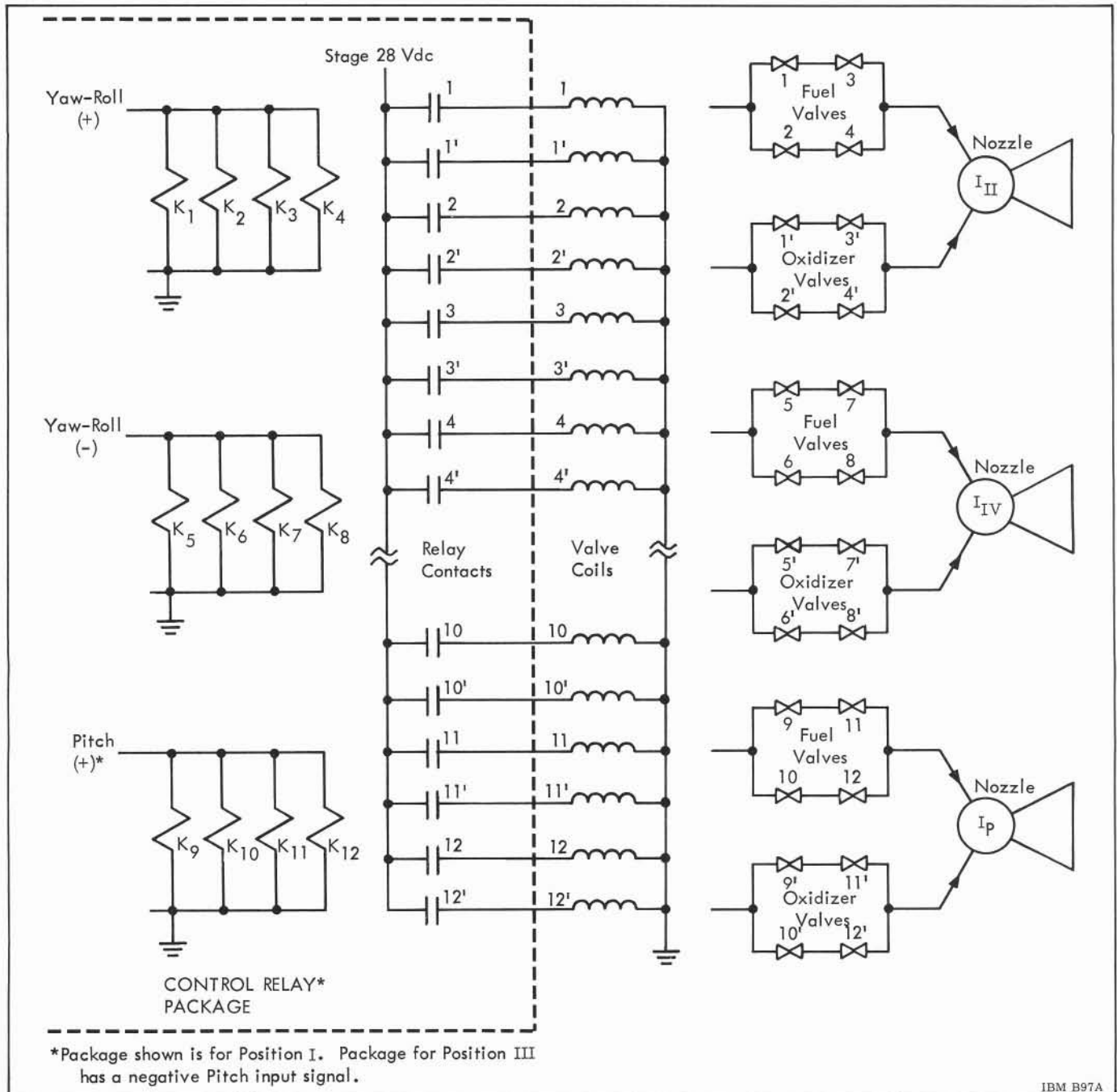
Figure 3.4-7 Control Relay Unit and Quad-Redundant Valves

CHAPTER 4
MODE AND SEQUENCE CONTROL
SECTION 4.1
INTRODUCTION

The terms mode and sequence control will be defined to explain their relationship with the Astrionics System. The LVDC memory contains a predetermined number of sets of instructions which, when initiated, induce the entire Astrionics System, or portions of the system, to operate in a particular mode. An example of a mode operation is the solving of guidance equations with interlaced attitude control and sequencing commands during the first stage flight. Each mode consists of a predetermined sequence of events. The computer also generates the appropriate control signals (e.g., engine cutoff and ignition, stage separation). Such a predetermined sequence of events is initiated by commanding the computer to operate in a particular mode.

Mode selection and initiation is accomplished through one of 3 actions: the successful completion of a previous mode, computer switching to a new mode based on real time, or an event occurrence. In addition to these internal mode selections which the system makes, mode switching is also accomplished by commands from the ground checkout equipment (before launch), the IU command system, or the Apollo Spacecraft. A built-in safeguard feature gives the system the capability of refusing conflicting commands or commands that would be detrimental to vehicle safety.

Examples of modes of operation that occur during flight in the Astrionics System are:

- First-stage guidance, control, and sequencing.

- Second-stage guidance, control, and sequencing.

- Third-stage guidance, control, and sequencing.

- Orbital operations —

    Vehicle attitude stabilization commanded by the IU.

    Spacecraft command of the vehicle attitude.

- Guidance, control, and sequencing of the third stage (second burn) into lunar trajectory —

    Using the IU navigation, guidance, and control system.

    Using the spacecraft navigation, guidance, and control system.

- Control of stage sequence functions from ground stations through the IU command system (emergency mode).

- Insertion of data into the LVDC through the IU command system.

The mode and sequence control scheme is very flexible since changes of modes and flight sequences are made by changing the instructions and programs in the LVDC memory, rather than by hardware modifications. The equipment involved in mode and sequence control of the launch vehicle is indicated in Figure 4.2-1. The LVDC issues flight sequence commands through the LVDA to the Switch Selector of the particular stage where the command must be executed. The Switch Selector of the addressed stage activates the appropriate circuit to perform the commanded function (e.g., engine cutoff, power ON or OFF). The Switch Selector operation is discussed in Section 4.2.

Mode commands from the spacecraft to the LVDA change the operational mode of the IU guidance and control system. When the Astronaut wants to control the attitude of the vehicle during coast flight, in

Astrionics System
Section 4.1

orbit, and after translunar injection, he initiates the proper mode command to the LVDA. The mode command is processed by the LVDA and LVDC, and the necessary sequence signals are generated to allow attitude control signals from the spacecraft to feed into the IU Flight Control Computer. Control is returned to the IU attitude control system when the Astronaut issues another mode command. In a similar manner, the back-up guidance mode may be selected. In this mode, the navigation and guidance system of the spacecraft guides the vehicle during powered flight.

During coast flight, the IU command system may be used to command mode changes from the ground.

Before launch, the ground checkout equipment exercises all operational modes of the Astrionics System. This includes simulated launch and orbit programs. The launch computer commands a particular mode of system operation by sending a coded command to the LVDA which "reads" it into the

LVDC. The mode command is decoded, and the set of instructions or program, which is defined by the decoded mode command, is selected from the LVDC memory. The LVDC then begins to execute (within itself or elsewhere in the system) the instructions in that mode program.

Some events of the flight sequence are controlled by "discrete" signals generated by the LVDC according to a time program. Other events of the flight sequence are initiated when a particular event has occurred. For example, a signal is sent to the IU indicating S-IC engine cutoff — the LVDC then sends a signal to shutdown the S-II recirculation system.

A new time base for the time-controlled program is initiated during operation as a result of certain flight events (e.g., at lift-off, by the actuation of the first stage propellant level sensors, and S-IVB engine cutoff). A discrete signal is given to the LVDC as a reference for each time base.

SECTION 4.2
SWITCH SELECTOR

4.2.1 SYSTEM DESCRIPTION

Each stage, and the Instrument Unit, of the Saturn Launch Vehicle are equipped with a Switch Selector. The Switch Selector consists of electronic and electromechanical components which decode digital flight sequence commands from the LVDA/ LVDC and activate the proper stage circuits to execute the commands.

Each Switch Selector can activate, one at a time, 112 different circuits in its stage. The selection of a particular stage Switch Selector is accomplished through the command code. Coding of flight sequence commands and decoding of the stage Switch Selectors reduces the number of interface lines between stages and increases the flexibility of the system with respect to timing and sequence. In the Saturn V Launch Vehicle, which contains 4 Switch Selectors, 448 different functions can be controlled using only 28 lines from the LVDA. (Two IU + 28-volt and two signal return lines from the Control Distributor are also used.) Flight sequence commands may be issued at time intervals of 100 milliseconds.

Figure 4.2-1 illustrates the Saturn V Switch Selector configuration. As shown, all Switch Selector control lines are connected through the Control Distributor in the IU to the LVDA and the electrical support equipment.

To maintain power isolation between vehicle stages, the Switch Selector is divided into two sections: the input section (relay circuits) of each Switch Selector receives its power from the IU; the output section (decoding circuitry and drivers) of each Switch Selector receives its power from the stage in which the Switch Selector is located. The input and output are coupled together through a diode matrix. This matrix decodes the 8-bit input code and activates a PNP output driver, thus producing a Switch Selector output.

Each Switch Selector is connected to the LVDA through 22 lines:

> Stage select lines - 2
> Read command lines - 2
> Reset (forced) lines - 2
> Bit register output lines - 8
> Verification lines - 8

In addition, there are 2 lines for IU + 28 Vdc and 2 lines for signal return between the Control Distributor and the Switch Selectors.

The wire pairs for stage select, read command, forced reset, IU + 28 Vdc, and signal return are redundant. Only one of each pair is required for normal operation.

All connections between the LVDA and the Switch Selectors, with the exception of the stage select inputs, are connected in parallel as shown in Figure 4.2-2.

The output signals of the LVDA switch selector register, with the exception of the 8-bit command, are sampled at the Control Distributor in the IU and sent to IU PCM telemetry. Each Switch Selector also provides 3 outputs to the telemetry system within its stage.

The Switch Selector is designed to execute flight sequence commands given by the 8-bit code or by its complement. This feature increases reliability and permits operation of the system despite certain failures in the LVDA switch selector register, line drivers, interface cabling, or Switch Selector relays.

The flight sequence commands are stored in the LVDC memory and are issued according to the flight program. When a Programmed Input/Output (PIO) instruction is given, the LVDC loads the 15-bit switch selector register with the computer data. The LVDA switch selector register word format is shown in Figure 4.2-3.

Notes: • Signal return lines from the Switch Selectors, through the Control Distributor, to the LVDA are not shown in this figure.
• The letters used to label interstage connections between units are not actual pin or cable connectors. The letter code is denoted below:

a = 8-digit command (8 lines)
b = Force reset (register) (1 line + 1 redundant line)
c = Register verification (8 lines)
d = Read command (1 line + 1 redundant line)
e =
f =   Stage select lines
g =   (1 line + 1 redundant line)
h =
j = b, c, d, e, f, g, and h to IU telemetry
k = Register test
    Zero indicate   To stage telemetry
    Sw Sel output   (1 line each)
m = +28 Vdc from the Instrument Unit

Figure 4.2-1 Saturn V Switch Selector Configuration

IBM B113A

Switch Selector register bits 1 through 8 represent the flight sequence command. Bits 9 through 13 select the Switch Selector to be activated. Bit 14 resets all the relays in the Switch Selectors in the event data transfer is incorrect as indicated by faulty verification information received by the LVDA. Bit 15 activates the addressed Switch Selector for execution of the command. The LVDC loads the switch selector register in two passes: bits 1

through 13 are loaded during the first pass and, depending on the feedback code, either bit 14 or bit 15 is loaded during the second pass.

After the Switch Selector input relays have been "picked" by the 8-bit command, the complement of the received code is sent back to the LVDA/LVDC over eight parallel lines. The feedback (verification information) is returned to the digital

Notes:

1. 22 lines between each Switch Selector and LVDA.

2. 26 lines between all Switch Selectors and LVDA (Saturn IB) (3 Switch Selectors).

3. 28 lines between all Switch Selectors and LVDA (Saturn V) (4 Switch Selectors).

4. 30 lines (maximum number) available to Switch Selectors from LVDA (5 Switch Selectors).

5. 4 lines (IU +28 Vdc and signal return) to all Switch Selectors from Control Distributor (Saturn IB and V).

Figure 4.2-2 LVDC - Switch Selector Interconnection Diagram
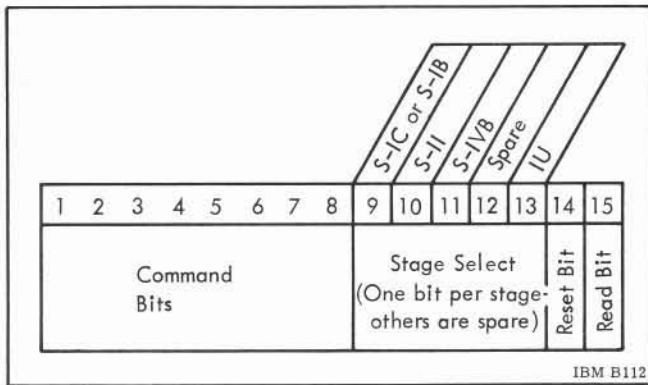
Figure 4.2-3 Switch Selector Register Word Format

input multiplexer of the LVDA and is subsequently compared with the original code in the LVDC. If the feedback agrees with the original code, a read command is given. If the feedback does not agree with the original code, a reset command is given (forced reset), and the LVDC/LVDA reissues the 8-bit command in complement form.

A typical operation cycle, to initiate a given function in a particular stage, is accomplished as follows:

- The 8 verify lines are sampled to ensure that all stage select relays have been reset, thereby ensuring an unwanted stage is not selected. Zero voltage on all lines indicates that this condition exists. The presence of IU + 28 Vdc on the verify lines indicates that the verify lines are enabled because a stage Switch Selector is not reset. Having detected this situation, the LVDC commands a "force reset" and then rechecks the verify lines.

- The LVDC inserts the 8-bit flight command into the LVDA switch selector register. At the same time, a stage select command is sent to the appropriate Switch Selector. Application of the stage select command completes the signal return path for the input relays and allows the 8-bit command to be stored in the Switch Selector input relays.

- The verification lines are sampled to determine if the command transfer was correct.

- If the verification is correct, the LVDC sends a read command to the Switch Selector; activating it, and causing the proper output to occur.

- If the verification is not correct, the LVDC initiates a reset command to prepare the Switch Selector to receive the complement of the original command.

- After the forced reset is completed, the LVDC loads the stage select command, and the complement of the previously sent command, into the LVDA switch selector register.

- Neglecting the verification information that occurs after the complement command is inserted, the LVDC initiates a read command which produces an output from the selected switch selector channel. The read command also energizes the automatic reset circuitry which returns the Switch Selector to the reset condition to prepare it for a new cycle.

4.2.2 COMPONENT DESCRIPTION

A description of Switch Selector operation is given in the following paragraphs (refer to the simplified schematic diagram and timing diagram shown in Figures 4.2-4 and 4.2-5, respectively) The circuit functions of the Switch Selector are as follows:

Stage select

Input relays

Verification

Reset (forced)

Read command

Reset (automatic)

Decoding matrix

Output matrix and telemetry

Test outputs

The Switch Selector operates on positive logic (i.e., +28 Vdc for a binary "1" and 0 Vdc for a binary "0").

STAGE SELECT

The purpose of the stage select command is to enable the desired Switch Selector to receive the coded flight sequence command from the LVDA. The stage select command (1 bit) is transmitted individually on a separate line to each Switch Selector. Because of the danger of overloading the LVDA, no more than 2 Switch Selectors can be addressed simultaneously.
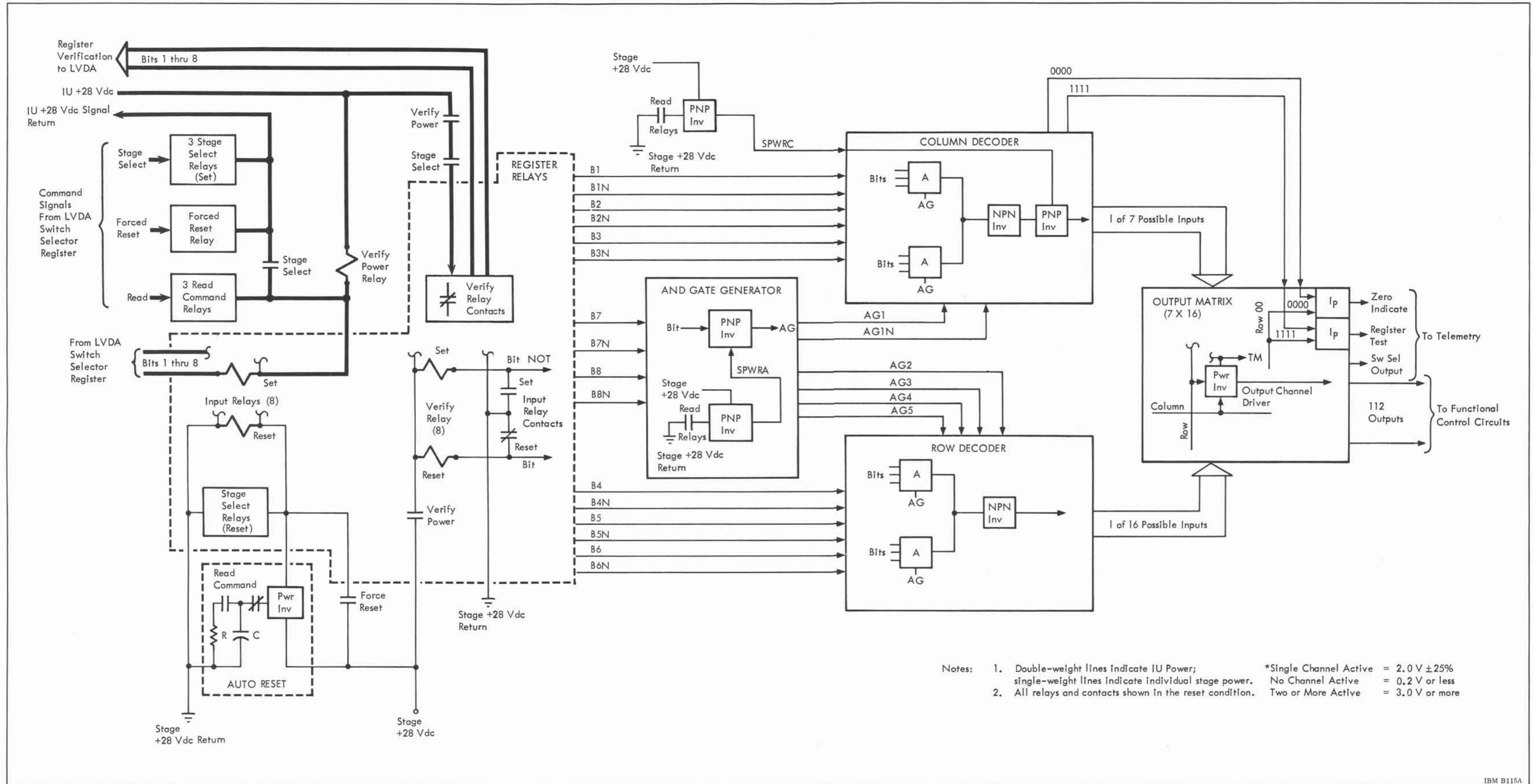
Figure 4.2-4 Switch Selector (Model II) Simplified Diagram

The stage select command sets three magnetic latch relays in the addressed Switch Selector. The setting of these relays completes a ground path for the read command relay, the set side of the input relays, and the verify power relay (see Figure 4.2-4). This action conditions the read and input relays to receive commands from the LVDA and energizes the verify power relay. When the verify power relay contacts close, IU + 28 Vdc power is applied through a set of stage select contacts to the verify relay contacts; and stage + 28 Vdc is applied to the verify relays. The Switch Selector can now accept the 8-bit command from the LVDA, store it in the input relays, and make verification information available to the LVDA
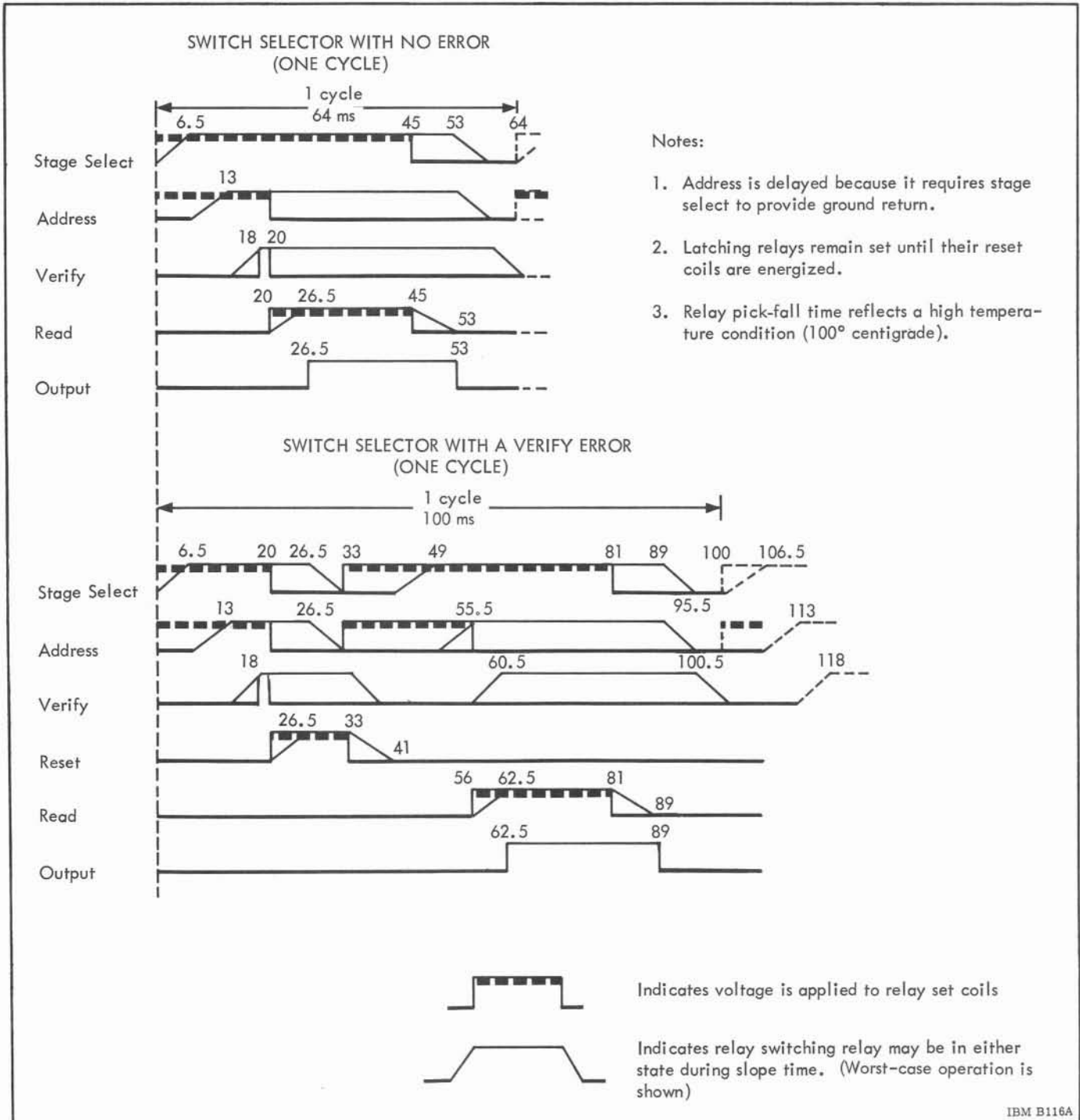
Figure 4.2-5 Typical Switch Selector Timing Diagram

INPUT RELAYS

The input register of the Switch Selector is made up of eight magnetic latch relays, controlled by the 8-bit coded command from the LVDA. The coded command (consisting of 8 bits in parallel) is available to the Switch Selector during the same period of time the stage select signal is present. As soon as the stage select relays are set, a signal return path is provided for the coded command through the input relay coils (See Figure 4.2-4). This allows the coded command to be stored in the input register.

Once the coded command is stored in the input register, binary information is available to the verification register, column decoder, row decoder, and the AND gate generator.

VERIFICATION

A feedback circuit is used to provide verification information back to the LVDC. This circuit consists of 8 relays, and is controlled by the input relay contacts. When a "1" is applied to the input relay set coil, stage ground is switched to the verify relay set coil causing the normally closed verify relay contact to open; indicating a "0" to the LVDC. When a "0" is applied to the input relay set coil, the input relay contacts remain in the reset condition. Stage ground remains applied to one side of the verify relay reset coil, causing the verify relay contact to remain in its normally closed position. Therefore, the verification output of a Switch Selector is the complement of the input command.

READ COMMAND

Immediately after verification of the original command, the read command is initiated. If verification of the original command proved to be false, the read command will not be initiated by the LVDC until the forced reset, stage select, and complemented command have been transmitted to the Switch Selector.

Three conventional (non-latching) relays are picked when the read command is received from the LVDA. One side of the coils of these relays is connected to the read command signal, the other side is connected to the IU signal return through the stage select relay contacts. Thus, a stage select signal must have been received before a read command relay can be energized. When the read relay contacts close, stage ground is applied to the switch

power inverters (SPWRA and SPWRC) in the decoder matrix. These inverters generate voltages (SPWRA and SPWRC) which activate the AND gate generators and the column decoder circuitry allowing the coded command to be decoded and an output to occur.

RESET (FORCED)

A forced reset is initiated by the LVDC when the LVDC determines that the verification information is not the complement of the given command. The reset signal is applied to the forced reset relay coil causing the forced reset contacts to close. This action applies stage + 28 Vdc to the stage select and input relay reset coils forcing the Switch Selector into the reset condition shown in Figure 4.2-4. After the forced reset has been completed, the stage select command and the complement of the original flight sequence command are transmitted to the Switch Selector. The complement command is not verified by the LVDC before the read command is issued.

RESET (AUTOMATIC)

An automatic reset pulse is generated when stage power is initially applied to the Switch Selector and when the read command is issued. The automatic reset circuitry consists of a power inverter, an RC network, and a set of read command contacts.

Activation of the read command relay contacts is not required to produce the "Power ON" reset pulse. Prior to the application of stage power to the Switch Selector, the read command contacts are in the reset condition (as shown in Figure 4.2-6) and capacitor C is in a discharged state. When power is applied to the Switch Selector, stage 28 Vdc is applied to the emitter of the power inverter and capacitor C begins to charge through base resistor (R3) and thermistor (RT1). Under these conditions the power inverter turns on, producing an output of approximately 28 Vdc at the collector. This output is applied across the input relay reset coils and the stage select reset coils forcing the Switch Selector into a reset condition.

The duration of the reset pulse is determined by the charge time of capacitor C through base resistor R3 and thermistor RT1. When C charges sufficiently, the power inverter is turned off and the reset pulse is removed from the reset coils.
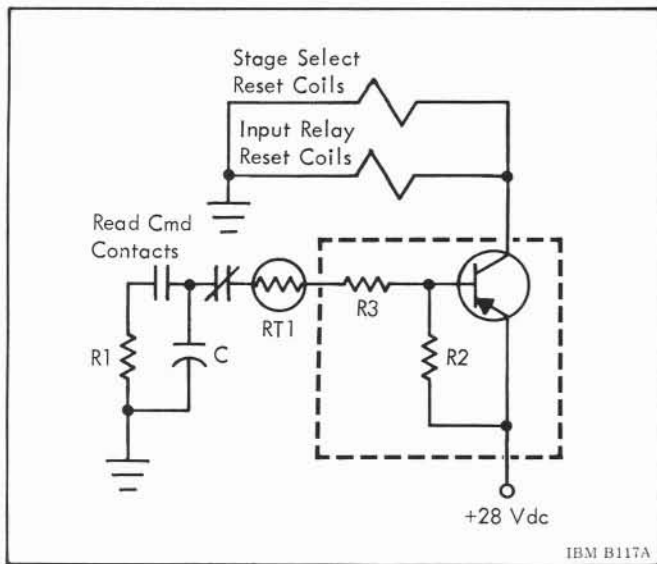
Figure 4.2-6 Automatic Reset Circuitry
Simplified Diagram

The automatic reset pulse initiated by the read command is developed exactly the same as the "Power ON" reset pulse. When the read command is issued, the normally open read command relay contact closes, and the normally closed read command relay contact opens. This allows capacitor C to discharge through resistor R1. The power inverter remains turned off (due to lack of base drive) until the read command is removed. When the read command is removed, the read command relay contacts return to the reset condition. At this time, capacitor C begins to charge providing base drive for the power inverter. The power inverter turns on, producing an output of approximately 28 Vdc at the collector. The output is applied across the input relay and stage select relay reset coils forcing the Switch Selector into the reset condition as shown in Figure 4.2-4.

As previously stated, the duration of the reset pulse is determined by the charge time of capacitor C through base resistor R3 and thermistor RT1. When C is sufficiently charged. the power inverter is turned off and the reset pulse is removed from the reset coils.

DECODING MATRIX

The decoding matrix is comprised of PNP inverters, NPN inverters, and AND circuits arranged so that selection of a particular output channel can be made by either the true or complement input code.

This matrix can be divided into three sections as follows (see Figure 4.2-4):

- AND gate circuitry
- Row decoder
- Column decoder

The object of these circuits is to generate a particular column and row signal for each input code and to send these signals to the output matrix upon receipt of the read command signal.

The 8-bit coded command is transferred to the decoding matrix by operation of the input relay contacts. When a binary "1" is applied to an input relay set coil, the normally open input relay contact closes and the normally closed input relay contact opens.

This causes stage ground to be switched from the verify relay reset coil to the verify relay set coil. The bit NOT input to the decoding matrix is now tied to stage ground, while the bit input line to the decoding matrix is at stage + 28 Vdc. When a binary "0" is applied to an input relay set coil, the input relay contacts remain in the reset condition. With the input relay contacts in the reset condition, the bit NOT input to the decoding matrix is at stage + 28 Vdc and the bit input to the decoding matrix is a 0 Vdc.

The 8-bit command is sub-divided into groups for decoding and output driver selection. The least three significant bits (1, 2, and 3) are decoded to enable the column selection circuitry. The next three bits (4, 5, and 6) are decoded to enable the row selection circuitry. Bits 7 and 8 and SPWRA (controlled by read command relays) are applied to the AND gate generator where they are decoded to supply the appropriate AND gates to the column and row decoders. Bit 8 is applied to the AND gate generator where it is decoded to determine whether the coded command is a true or complement word. The listing below illustrates the Boolean expressions for decoding bits 7 and 8.

AG1	=	B8	· Read Command
AG1N	=	B8N	· Read Command
AG2	=	AG1	· B7
AG3	=	AG1	· B7N
AG4	=	AG1N · B7N	
AG5	=	AG1N · B7	

where: B8 = Bit 8 = Bit 8 is a binary "1".

B8N= Bit 8 NOT = Bit 8 is a binary "0".

When decoding a typical word (01010101) where bits 8 through 1 are arranged from left to right, the column is determined by bits 8, 3, 2 and 1, or 0101, or more completely by $AG1N \cdot B3 \cdot B2N \cdot B1$. Similarly, the row is determined by bits 8, 7, 6, 5, and 4, or 01010, or more completely $AG5 \cdot B6N \cdot B5 \cdot B4N$. However, if the input register transfer circuitry shows a failure, the complement of the original word must select the same output channel in the output matrix. Therefore, the location defined by 01010101 must also be defined by 10101010. This complement code is decoded in the same manner as the original code. The column is determined by bits 8, 3, 2, and 1, or 1010, or more completely by $AG1 \cdot B3N \cdot B2 \cdot B1N$. The row is determined by 8, 7, 6, 5, and 4, or 10101, or more completely by $AG3 \cdot B6 \cdot B5N \cdot B4$. Therefore, the total expression to define this particular output channel is:

$$\begin{aligned}
\text{Channel XY} = {} & AG1N \cdot AG5 \cdot B6N \cdot B5 \cdot \\
& B4N \cdot B3 \cdot B2N \cdot B1 \\
+ {} & AG1 \cdot AG3 \cdot B6 \cdot B5N \cdot \\
& B4 \cdot B3N \cdot B2 \cdot B1N
\end{aligned}$$

OUTPUT MATRIX AND TELEMETRY

The 112 output channels of the Switch Selector are arranged in a 7 by 16 matrix configuration (7 columns and 16 rows). The coordinate selection is similar to the X-Y coincidence method common to core memory technology. An output channel driver (PNP inverter) with the location X-Y in the matrix is activated when the row decoder circuitry (X) and the column decoder circuitry (Y) are turned ON by application of the read command. When activated, the output channel driver produces a + 28 Vdc output to the control circuitry of the selected function.

In addition to producing an output voltage, each output channel driver also produces a separate telemetry output. The 112 telemetry outputs are tied together to produce one switch selector telemetry output signal. This signal is transmitted via PCM and DDAS telemetry. The telemetry signal indicates whether none, one, or more than one output channel is ON at any given time. If one output channel is activated, the telemetry signal will have a magnitude of 2.0 Vdc ± 20 percent; if no output channel is active, the telemetry signal will have a magnitude of 0.2 Vdc or less; and if more than one output channel is activated, the telemetry signal will have a magnitude of 3.0 Vdc or more. When the Switch Selector is operating properly, only one output channel will be active at any given time.

TEST OUTPUTS

In addition to the telemetry output described, each Switch Selector has two telemetry outputs which are used to verify that the input selection relays can be set and reset properly. These signals (register test and zero indicate) are special outputs of the output matrix. The register test output is generated by an output driver when the input address selection is all ones and the read command has been issued. The zero indicate output is generated when the input address selection is all zeros (read command is not required). The zero indicate and register test signals are transmitted via PCM and DDAS telemetry.

SECTION 4.3
SATURN V OPERATION SEQUENCE

The Saturn V Operation sequence starts during the prelaunch phase at approximately 22 hours when the electrical power from the ground support equipment is applied to all stages of the launch vehicle. Figure 4.3-1 provides an overall view of the prelaunch sequence. During this time, the sequencing is controlled from the launch control center-launcher-umbilical tower complex, utilizing both manual and automatic control to checkout the functions of the entire Astrionics System. After the umbilical is disconnected, the sequencing is primarily controlled by the flight program within the LVDC.

Since flight sequencing is time related, the sequencing operation is divided into seven time periods. These time periods are referred to as time bases and progress from time base No. 1 (T_1) through time base No. 7 (T_7). Each time base is related to a particular flight event. These time bases and their initiating events are as follows:

T_1	Lift-off
T_2	Inboard engine cutoff (S-IC)
T_3	Outboard engine cutoff (S-IC)
T_4	Engine cutoff (S-II)
T_5	Engine cutoff (S-IVB)
T_6	Begin restart preparations (S-IVB)
T_7	Engine cutoff (S-IVB)

The flight program (described in Chapter 11) uses these time bases in issuing sequential commands to the appropriate stages. The essential sequenced events, as they occur within each time base, are listed in Table 4.3-1.

Note: Time periods are expressed in seconds except where designations H (hours) or M (minutes) are used. The abbreviation "TBD" indicates "To Be Determined".

Table 4.3-1 Typical Saturn V Flight Sequence, Time Base 1 through 7

Nominal Flight Time Hrs:Min:Sec	Command	Stage	Remarks	Time from Base (Seconds)
00:00.0	Liftoff — Start of Time Base No. 1 (T_1)		Liftoff sets the computer time base. Fuel pressuring valve No. 1 is sequenced open at this time.	$T_1 + \ \ 0.0$
00:05.0	Auto-Abort Enable Relays Reset	IU		$T_1 + \ \ 5.0$
00:14.0	Multiple Engine Cutoff Enable	S-IC	This signal enables cutoff of any engine due to rough combustion or low thrust. Detects combustion instability by monitoring accelerometers located on each engine.	$T_1 + \ 14.0$
00:25.0	S-IC Telemeter Calibrate ON	S-IC		$T_1 + \ 25.0$
00:27.0	Telemeter Calibrator In-Flight Calibrate	IU		$T_1 + \ 27.0$
00:30.0	S-IC Telemeter Calibrate OFF	S-IC		$T_1 + \ 30.0$
00:32.0	Telemeter Calibrator Stop In-Flight Calibrate	IU		$T_1 + \ 32.0$
00:49.5	Fuel Pressurizing Valve No. 2 Open & Tape Recorder Record	S-IC	Fuel Pressuring Valve sequence times are preliminary. Fuel pressuring valve No. 1 (normally open) is opened at liftoff.	$T_1 + \ 49.5$
00:60.0	Enable Launch Vehicle Engine EDS Cutoff			$T_1 + \ 60.0$
01:30.0	Telemeter Calibrator In-Flight Calibrate	IU		$T_1 + \ 90.0$
01:34.8	Telemeter Calibrator Stop In-Flight Calibrate	IU		$T_1 + \ 94.8$
01:35.3	Fuel Pressurizing Valve No. 3 Open	S-IC	Refer to time base $T_1 + 49.5$	$T_1 + \ 95.3$
01:40.0	Flight Control Computer Switch Point No. 1	IU		$T_1 + 100.0$
01:55.0	S-IC Telemeter Calibrate ON	S-IC		$T_1 + 115.0$
01:59.2	Regular Calibrate Relays ON	S-IVB		$T_1 + 119.2$
02:00.0	Flight Control Computer Switch Point No. 2	IU		$T_1 + 120.0$

Figure 4.3-1 Typical Saturn V Astrionics System Prelaunch Sequence

Table 4.3-1 Typical Saturn V Flight Sequence, Time Base 1 through 7 (Cont.)

Nominal Flight Time Hrs:Min:Sec	Command	Stage	Remarks	Time from Base (Seconds)
02:00.2	S-IC Telemeter Calibrate OFF	S-IC		$T_1 + 120.2$
02:04.2	Regular Calibrate Relays OFF	S-IVB		$T_1 + 124.2$
02:13.5	Fuel Pressurizing Valve No. 4 Open	S-IC	Refer to time base $T_1 + 49.5$	$T_1 + 133.5$
02:13.7	Fast Record ON	S-IVB		$T_1 + 133.7$
02:13.9	Start Recorders	S-II		$T_1 + 133.9$
02:14.1	IU Tape Recorder Record ON	IU		$T_1 + 134.1$
02:14.3	LOX Tank Strobe Lights OFF	S-IC		$T_1 + 134.3$
02:14.5	S-IC Two Engine Out Auto-Abort Inhibit Enable	IU		$T_1 + 134.5$
02:14.7	S-IC Two Engine Out Auto-Abort Inhibit	IU		$T_1 + 134.7$
02:14.9	Excessive Rate, (P, Y, and R) Auto-Abort Inhibit Enable	IU		$T_1 + 134.9$
02:15.1	Excessive Rate (P, Y, and R) Auto-Abort Inhibit	IU		$T_1 + 135.1$
02:15.3	Two Adjacent Outboard Engines Out Enable	S-IC		$T_1 + 135.3$
02:15.5	Inboard Engine Cutoff—Start of Time Base No. 2 (T_2)	S-IC	The S-IC inboard engine will be cutoff by the LVDC through the S-I Switch Selector at a predetermined time. At this time, the LVDC will monitor the downrange accelerometer. If sufficient downrange velocity exists, the LVDC will initiate Time Base No. 2 (T_2).	$T_2 + 0.0$
02:15.9	Start First PAM-FM/FM Calibration	S-II		$T_2 + 0.4$
02:20.9	Stop First PAM-FM/FM Calibration	S-II		$T_2 + 5.4$
02:23.8	Separation Camera ON	S-IC		$T_2 + 8.3$
02:24.3	LOX Tank Strobe Lights OFF	S-IC		$T_2 + 8.8$

Table 4.3-1 Typical Saturn V Flight Sequence, Time Base 1 through 7 (Cont.)

Nominal Flight Time Hrs:Min:Sec	Command	Stage	Remarks	Time from Base (Seconds)
02:25.0	S-II Ordnance Arm	S-II		T_2 + 9.5
02:25.2	Arm EBW Retro and Separation	S-IC		T_2 + 9.7
02:25.5	S-II LH$_2$ Recirculation Pumps OFF	S-II		T_2 + 10.00
02:25.7	Q-Ball Power OFF	IU		T_2 + 10.2
02:25.9	Camera Lights ON	S-II		T_2 + 10.4
02:26.1	S-IC Telemetry Measurement Switch Over	S-IC		T_2 + 10.6
02:26.3	Enable Outboard Engine Cutoff	S-IC		T_2 + 10.8
02:27.3	Outboard Engines Cutoff — Start of Time Base No. 3 (T_3)		S-IC outboard engine cutoff initiates time base T_3. There are two different methods by which the S-IC outboard engines can be cutoff normally: (1) Actuation of any two LOX level sensors will initiate a 1-second timer which will cutoff S-IC outboard engines upon expiration. (2) Actuation of either of two fuel lower level sensors will initiate a 1-second timer which will cutoff the S-IC outboard engines upon completion.	T_3 + 0.0
02:27.5	Camera Motor ON	S-II		T_3 + 0.2
02:27.7	S-II Ullage Trigger	S-II	The S-II ullage rockets reach full thrust approximately 0.170 seconds after the signal to trigger the ullage rockets is initiated by the computer. Ullage rockets are required to be at full thrust when the separation structure is severed. The S-II ullage rockets must burn until the S-II Stage engine reaches approximately 30 percent thrust.	T_3 + 0.4
02:27.9	S-IC/S-II Separation	S-IC	Separation is initiated when the F-1 outboard engines thrust has decayed to the 10 percent level. Circuit delays of approximately 10-15 ms are built into the separation circuitry so that retrorocket ignition occurs after separation device ignition to prevent possible unseating of the S-II Stage propellants.	T_3 + 0.6

Table 4.3-1 Typical Saturn V Flight Sequence, Time Base 1 through 7 (Cont.)

Nominal Flight Time Hrs:Min:Sec	Command	Stage	Remarks	Time from Base (Seconds)
02:28.0	Camera Event Mark	S-II	Provide Visual Record	T_3 + 0.7
02:28.1	Switch Engine Control to S-II, Enable S-II Engine Out and Second Separation Indication "A"	IU	Engine gimballing control is switched from the S-IC Stage to the S-II Stage. This is accomplished in the IU.	T_3 + 0.8
02:28.2	S-II Engines Cutoff Reset	S-II		T_3 + 0.9
02:28.3	Engines Ready Bypass	S-II		T_3 + 1.0
02:28.4	Prevalves Lockout Reset	S-II		T_3 + 1.1
02:28.6	S-II Engine Start	S-II	The S-IC/S-II Stages are separated by a minimum of 31 cm (1 ft) at this time. The J-2 engine chilldown begins at this time. When the 0.5 second chilldown timers expire, the start tank discharge valves open regardless of whether the thermocouples indicate proper chilldown temperature. Physical separation signal is interlocked in the S-II engines start command. The thermocouple override signals to the S-II engines are supplied when power is transferred to the S-II Stage.	T_3 + 1.3
02:28.7	Camera Event Mark	S-II		T_3 + 1.4
02:28.8	Enable S-II Engine Out and S-II Second Separation Indication "B"	IU		T_3 + 1.5
02:29.0	Engines Ready Bypass Reset	S-II		T_3 + 1.7
02:29.9	S-II Hydraulic Accumulators Unlock	S-II	Unlocking the hydraulic accumulators enables gimballing of the S-II Stage engines.	T_3 + 2.6
02:33.9	S-II Start Phase Limiter Cutoff Arm	S-II	A start sequence halt, due to timer malfunction, etc., will prevent the malfunction cutoff circuitry being armed by the engine sequencer. This signal is a backup for arming the mainstage OK switch to give cutoff if a malfunction in the engine sequencer has occurred. This signal will enable the individual cutoff circuitry of each J-2 engine to shutdown the engine when mainstage OK is not energized.	T_3 + 6.6

Table 4.3-1 Typical Saturn V Flight Sequence, Time Base 1 through 7 (Cont.)

Nominal Flight Time Hrs:Min:Sec	Command	Stage	Remarks	Time from Base (Seconds)
02:34.1	Activate PU System	S-II		T_3 + 6.8
02:34.9	S-II Start Phase Limiter Cutoff Arm Reset	S-II	This signal prevents inadvertent engine cutoff before depletion.	T_3 + 7.6
02:38.4	IU Tape Recorder Record Off	IU		T_3 + 11.1
02:38.6	Stop Data Recorders	S-II		T_3 + 11.3
02:38.8	Fast Record OFF	S-IVB		T_3 + 11.5
02:57.8	S-II Second Plane Separation	S-II		T_3 + 30.5
02:57.9	Camera Event Mark	S-II	Visual recording of separation event.	T_3 + 30.6
02:58.9	Camera Event Mark	S-II	Visual recording of separation event.	T_3 + 31.6
03:00.0	Water Coolant Valve Open	IU	No cooling is available for the IU from the end of prelaunch phase until this time. This time depicted is typical.	T_3 + 32.7
03:03.6	L.E.T. Jettison "A"	IU	The launch escape system is jettisoned approximately 6 seconds after inter-stage (second plane) separation.	T_3 + 36.3
03:03.8	L.E.T. Jettison "B"	IU		T_3 + 36.5
03:05.3	Camera Eject No. 1	S-II	Separation of visual recording devices.	T_3 + 38.0
03:05.9	Camera Eject No. 2	S-II	Separation of visual recording devices.	T_3 + 38.6
03:06.4	Camera Eject No. 3	S-II	Separation of visual recording devices.	T_3 + 39.1
03:28.6	Flight Control Computer Switch Point No. 3	IU		T_3 + 61.3
05:30.0	Start Second PAM-FM/FM Calibration	S-II		T_3 + 182.7
05:35.0	Stop Second PAM-FM/FM Calibration	S-II		T_3 + 187.7
05:38.6	Flight Control Computer Switch Point No. 4	IU		T_3 + 191.3
05:50.0	Telemeter Calibrator In-Flight Calibrate	IU		T_3 + 202.7
05:55.0	Telemeter Calibrator Stop In-Flight Calibrate	IU		T_3 + 207.7
06:00.0	Measurement Control Switch No. 2 Activate	S-II		T_3 + 212.7

Table 4.3-1 Typical Saturn V Flight Sequence, Time Base 1 through 7 (Cont.)

Nominal Flight Time Hrs:Min:Sec	Command	Stage	Remarks	Time from Base (Seconds)
06:38.2	S-II LH$_2$ Step Pressurization	S-II		$T_3 + 250.9$
07:00.7	Start Third PAM-FM/FM Calibration	S-II		$T_3 + 273.4$
07:05.7	Stop Third PAM-FM/FM Calibration	S-II		$T_3 + 278.4$
08:08.1	Regular Calibrate Relays ON	S-IVB		$T_3 + 340.8$
08:08.3	Telemeter Calibrator In-Flight Calibrate	IU		$T_3 + 341.0$
08:12.9	Regular Calibrate Relays OFF	S-IVB		$T_3 + 345.6$
08:13.3	Telemeter Calibrator Stop In-Flight Calibrate	IU		$T_3 + 346.0$
08:13.9	Rate Gyro OFF	S-IVB		$T_3 + 346.6$
08:14.1	Charge Ullage Ignition ON	S-IVB		$T_3 + 346.8$
08:14.3	S-II/S-IVB Ordnance Arm	S-II		$T_3 + 347.0$
08:14.5	IU Tape Recorder Record ON	IU		$T_3 + 347.2$
08:14.7	Fast Record ON	S-IVB		$T_3 + 347.4$
08:14.9	Start Recorders	S-II		$T_3 + 347.6$
08:15.1	Start Recorder Timers	S-II		$T_3 + 347.8$
08:15.3	S-IVB Engine Cutoff OFF	S-IVB		$T_3 + 348.0$
08:15.5	S-II LOX Depletion Sensor Cutoff Arm	S-II		$T_3 + 348.2$
08:15.7	S-II LH$_2$ Depletion Sensor Cutoff Arm	S-II		$T_3 + 348.4$
08:30.7	Cutoff S-II J-2 Engines — Start of Time Base No. 4 (T_4)	S-II	The computer time base is reset at this time. Simultaneous shutdown of all five J-2 engines is initiated by any two of five propellant depletion sensors (5 located in LOX tank and 5 in LH$_2$ tank) sensing the shutoff level of either the LOX or LH$_2$ propellant tanks.	$T_4 + 0.0$

Table 4.3-1 Typical Saturn V Flight Sequence, Time Base 1 through 7 (Cont.)

Nominal Flight Time Hrs:Min:Sec	Command	Stage	Remarks	Time from Base (Seconds)
08:30.8	Prevalves Close OFF	S-IVB		T_4 + 0.1
08:30.9	Fire Ullage Ignition ON	S-IVB	The effective burn time of the S-IVB ullage rockets is approximately 4 seconds. The ullage rockets must burn until enough thrust is present in the J-2 engine to maintain propellant seating.	T_4 + 0.2
08:31.0	S-II/S-IVB Separation	S-II	The time shown represents the time when the J-2 engines should be at 10 percent thrust. Circuit delays are built into the separation circuitry so that retrorocket ignition occurs after separation device ignition to prevent possible unseating of the S-IVB Stage propellants.	T_4 + 0.3
08:31.1	Engine Ready Bypass	S-IVB	This signal is effective only on unmanned flights and prevents the engine from being cutoff by any of the malfunction cutoff modes, which are: (1) Start tank depressurized (2) Mainstage not OK.	T_4 + 0.4
08:31.2	S-IVB Engine Start ON	S-IVB	This is the time when a minimum of 3.1 meters (10 feet) clearance between S-IVB engine and separated aft interstage is reached with one retrorocket out. J-2 engine chilldown begins at this time. The J-2 engine contains a sequencer which sequences the necessary commands to the engine for starting.	T_4 + 0.5
08:31.3	S-IVB Engine Start Interlock Bypass ON	S-IVB	This interlock will be used for unmanned flights only.	T_4 + 0.6
08:31.4	Flight Control Computer S-IVB Burn Mode ON	IU	This signal enables the auxiliary propulsion system to control roll and enables J-2 engine gimballing.	T_4 + 0.7
08:31.6	LOX Chilldown Pump OFF	S-IVB		T_4 + 0.9
08:32.4	Chilldown Shutoff Pilot Valve Close ON	S-IVB		T_4 + 1.7
08:32.6	Fuel Chilldown Pump OFF	S-IVB		T_4 + 1.9
08:33.2	S-IVB Engine Out Indication "A" Enable ON	IU		T_4 + 2.5

Table 4.3-1 Typical Saturn V Flight Sequence, Time Base 1 through 7 (Cont.)

Nominal Flight Time Hrs:Min:Sec	Command	Stage	Remarks	Time from Base (Seconds)
08:33.4	S-IVB Engine Out Indication "B" Enable ON	IU		$T_4 + 2.7$
08:34.0	LOX Tank Flight Pressure System ON	S-IVB		$T_4 + 3.3$
08:34.2	Fuel Injection Temperature OK Bypass	S-IVB	The start tank discharge valve will open at this time regardless of whether the thermocouple signal is present.	$T_4 + 3.5$
08:34.4	Engine Start OFF	S-IVB		$T_4 + 3.7$
08:36.0	First Burn Relay ON	S-IVB		$T_4 + 5.3$
08:38.0	Emergency Playback Enable ON	S-IVB		$T_4 + 7.3$
08:39.2	PU Activate ON	S-IVB		$T_4 + 8.5$
08:40.5	Charge Ullage Jettison ON	S-IVB		$T_4 + 9.8$
08:43.0	Fire Ullage Jettison ON	S-IVB		$T_4 + 12.3$
08:46.3	Ullage Charging Reset	S-IVB		$T_4 + 15.6$
08:46.5	Ullage Firing Reset	S-IVB		$T_4 + 15.8$
08:49.0	Fast Record OFF	S-IVB		$T_4 + 18.3$
08:49.6	IU Tape Recorder Record OFF	IU		$T_4 + 18.9$
08:52.0	Emergency Playback Enable OFF	S-IVB		$T_4 + 21.3$
08:53.1	Telemeter Calibrator In-Flight Calibrate	IU		$T_4 + 22.4$
08:58.1	Telemeter Calibrator Stop In-Flight Calibrate	IU		$T_4 + 27.4$
09:02.5	Regular Calibrate Relays ON	S-IVB		$T_4 + 31.8$
09:07.5	Regular Calibrate Relays OFF	S-IVB		$T_4 + 36.8$
11:31.6	Cutoff S-IVB Engine — Start of Time Base No. 5 (T_5)	S-IVB	Initiated by deactuation of S-IVB thrust OK pressure switches at S-IVB cutoff. J-2 engine cutoff will be at a predetermined velocity. Time shown is typical sequence time.	$T_5 + 0.0$

Table 4.3-1 Typical Saturn V Flight Sequence, Time Base 1 through 7 (Cont.)

Nominal Flight Time Hrs:Min:Sec	Command	Stage	Remarks	Time from Base (Seconds)
11:31.9	70-pound Ullage Engines ON	S-IVB	The 312 Newton (70-pound) ullage engines are actuated to dampen out the cutoff transients.	T_5 + 0.3
11:32.0	Ullage Thrust Present ON	IU		T_5 + 0.4
11:32.1	Prevalves Close ON	S-IVB	The main fuel and LOX valves must have sufficient time to close before the prevalves are closed. The minimum time to close the LOX and LH_2 prevalves is 0.425 seconds.	T_5 + 0.5
11:32.3	First Burn Relay OFF	S-IVB		T_5 + 0.7
11:32.5	PU Activate OFF	S-IVB		T_5 + 0.9
11:32.7	LOX Tank Flight Pressure System OFF	S-IVB	This signal sets the first burn relay which closes the main LOX pressurization shutoff valve.	T_5 + 1.1
11:32.9	Coast Period ON	S-IVB		T_5 + 1.3
11:34.2	Engine Pump Purge Control Valve Enable ON	S-IVB	The J-2 engine must be purged for 10 minutes after engine shutdown.	T_5 + 2.6
11:34.4	PU Inverter and DC Power OFF	S-IVB		T_5 + 2.8
11:36.6	Flight Control Computer S-IVB Burn Mode OFF	IU		T_5 + 5.0
TBD	S/S FM Transmitter OFF	S-IVB		TBD
TBD	S/S FM Transmitter Group OFF	S-IVB		TBD
11:36.8	Aux Hydraulic Pump Coast Mode ON	S-IVB		T_5 + 5.2
11:37.0	Aux Hydraulic Pump OFF	S-IVB		T_5 + 5.4
11:38.6	Continuous Vent Valve Open ON	S-IVB	Continuous vent valve open command active for only 2 seconds. The valve will stay open during the restart sequence.	T_5 + 7.0
11:40.6	Continuous Vent Valve Open OFF	S-IVB		T_5 + 9.0
11:41.6	S-IVB Engine Out Indication "A" Enable OFF	IU		T_5 + 10.0

Table 4.3-1 Typical Saturn V Flight Sequence, Time Base 1 through 7 (Cont.)

Nominal Flight Time Hrs:Min:Sec	Command	Stage	Remarks	Time from Base (Seconds)
11:41.8	S-IVB Engine Out Indication "B" Enable OFF	IU		T_5 + 10.2
12:21.7	312 Newton (70-Pound) Ullage Engines OFF	S-IVB		T_5 + 50.1
12:21.9	S-IVB Ullage Thrust Present OFF	IU		T_5 + 50.3
12:32.1	Prevalves Close OFF	S-IVB		T_5 + 60.5
12:32.3	Chilldown Shutoff Pilot Valve Close OFF	S-IVB		T_5 + 60.7
12:32.5	Azusa Transponder Power OFF	IU		T_5 + 60.9
TBD	IU Tape Recorder Playback Reverse ON	IU		TBD
TBD	Emergency Playback Enable ON	S-IVB		TBD
TBD	Emergency Playback Enable OFF	S-IVB		TBD
TBD	Tape Recorder Playback Reverse OFF	IU		TBD
TBD	Slow Record ON	S-IVB		TBD
TBD	Slow Record ON	S-IVB		TBD
TBD	Flight Transmitter Group OFF	S-IVB		TBD
TBD	Flight SCO Group OFF	S-IVB		TBD
TBD	Orbit Transmitter Group OFF	S-IVB		TBD
TBD	Engine Pump Purge Control Valve Enable OFF	S-IVB		TBD
TBD	PU Inverter and DC Power ON	S-IVB		TBD
TBD	Begin Restart Preparations — Start of Time Base No. 6 (T_6)		T_6 initiated by the LVDC or by ground command; based upon mission and launch window.	T_6 + 0.0
TBD	312 Newton (70-pound) Ullage Engines ON	S-IVB	The ullage engines are turned on to seat propellants for S-IVB restart.	T_6 + 0.2
TBD	S-IVB Ullage Thrust Present ON	IU		T_6 + 0.4

Table 4.3-1 Typical Saturn V Flight Sequence, Time Base 1 through 7 (Cont.)

Nominal Flight Time Hrs:Min:Sec	Command	Stage	Remarks	Time from Base (Seconds)
TBD	Continuous Vent Valve Close ON	S-IVB	Active for approximately 2 seconds.	T_6 + 1.0
TBD	LH$_2$ Repress. Control Valve Open ON	S-IVB		T_6 + 1.8
TBD	LOX Chilldown Pump ON	S-IVB	The chilldown pump is started to condition LOX system for restart.	T_6 + 2.2
TBD	Continuous Vent Valve Close OFF	S-IVB	Continuous vent valves should remain closed for remainder of mission.	T_6 + 3.0
TBD	Fuel Chilldown Pump ON	S-IVB		T_6 + 5.0
TBD	Prevalves Close ON	S-IVB		T_6 + 15.0
TBD	Prevalves Close OFF	S-IVB	Prevalves opened to prevent bubbles in suction line at engine start.	T_6 + 305.0
TBD	LOX Repress. Control Valve Open ON	S-IVB		T_6 + 305.2
TBD	S-IVB Restart Alert	IU		T_6 + 317.0
TBD	Aux. Hydraulic Pump Flight Mode ON	S-IVB		T_6 + 321.6
TBD	Aux. Hydraulic Pump Coast Mode OFF	S-IVB		T_6 + 321.8
TBD	PU Fuel Boil Off Bias Cutoff ON	S-IVB	The PU boiloff bias is switched off before second burn so propellants will be burned at proper mixture to obtain minimum residual.	T_6 + 325.0
TBD	S-IVB Engine Cutoff OFF	S-IVB		T_6 + 325.6
TBD	Engine Ready Bypass	S-IVB		T_6 + 325.8
TBD	LH$_2$ Repress. Control Valve Open OFF	S-IVB		T_6 + 326.0
TBD	Fuel Chilldown Pump OFF	S-IVB		T_6 + 326.2
TBD	LOX Chilldown Pump OFF	S-IVB		T_6 + 326.4
TBD	Chilldown Shutoff Pilot Valve Close ON	S-IVB		T_6 + 326.6
TBD	LOX Tank Repress. Control Valve Open OFF	S-IVB		T_6 + 326.8
TBD	Engine Start ON	S-IVB		T_6 + 327.0

Table 4.3-1 Typical Saturn V Flight Sequence, Time Base 1 through 7 (Cont.)

Nominal Flight Time Hrs:Min:Sec	Command	Stage	Remarks	Time from Base (Seconds)
TBD	LOX Tank Flight Press. System ON	S-IVB		$T_6 + 327.4$
TBD	Coast Period OFF	S-IVB		$T_6 + 327.6$
TBD	S-IVB Engine Out Indication "A" Enable ON	IU		$T_6 + 327.8$
TBD	S-IVB Engine Out Indication "B" Enable ON	IU		$T_6 + 328.0$
TBD	Flight Control Computer S-IVB Burn Mode ON	IU		$T_6 + 334.8$
TBD	Fuel Injection Temperature OK Bypass	S-IVB		$T_6 + 335.0$
TBD	Engine Start OFF	S-IVB		$T_6 + 335.2$
TBD	312 Newton (70-pound) Ullage Engines OFF	S-IVB		$T_6 + 337.2$
TBD	S-IVB Ullage Thrust Present OFF	IU		$T_6 + 337.4$
TBD	Second Burn Relay ON	S-IVB		$T_6 + 337.6$
TBD	PU Activate ON	S-IVB		$T_6 + 340.0$
TBD	Flight Control Computer Switch Point No. 5	IU		$T_6 + 597.0$
TBD	Point Level Sensor Arming	S-IVB		$T_6 + 655.7$
TBD	Cutoff S-IVB Engine — Start of Time Base No. 7 (T_7)	S-IVB	Time base T_7 is initiated by deactuation of the S-IVB thrust OK pressure switches at S-IVB cutoff.	$T_7 + 0.0$
TBD	Point Level Sensors Disarming	S-IVB		$T_7 + 0.3$
TBD	Prevalves Close ON	S-IVB	The main fuel and LOX valves must have sufficient time to close before the prevalves are closed. The minimum closing time of the LOX and LH_2 prevalves after engine cutoff is 0.425 second.	$T_7 + 0.5$
TBD	Second Burn Relay OFF	S-IVB		$T_7 + 0.7$
TBD	PU Activate OFF	S-IVB		$T_7 + 0.9$
TBD	PU Inverter and D.C. Power OFF	S-IVB		$T_7 + 1.3$

Table 4.3-1 Typical Saturn V Flight Sequence, Time Base 1 through 7 (Cont.)

Nominal Flight Time Hrs:Min:Sec	Command	Stage	Remarks	Time from Base (Seconds)
TBD	LOX Chilldown Pump Purge Control Valve Open OFF	S-IVB		$T_7 +$ 1.6
TBD	LOX Tank Flight Press. System OFF	S-IVB		$T_7 +$ 1.8
TBD	Coast Period ON	S-IVB		$T_7 +$ 2.0
TBD	Flight Control Computer S-IVB Burn Mode OFF	IU		$T_7 +$ 5.0
TBD	Aux Hydraulic Pump Flight Mode OFF	S-IVB		$T_7 +$ 5.2
TBD	LV/SC Separation Sequence Start	IU		$T_7 +$ 8.3
TBD	Prevalves Close OFF	S-IVB		$T_7 +$ 60.5
TBD	Chilldown Shutoff Pilot Valve Close OFF	S-IVB		$T_7 +$ 60.7
TBD	LH$_2$ Tank Vent Valve Open ON	S-IVB		TBD
TBD	LOX Tank Vent Valve Open ON	S-IVB		TBD
TBD	LH$_2$ Tank Vent Valve Open OFF	S-IVB		TBD
TBD	LH$_2$ Tank Vent Valve Close ON	S-IVB		TBD
TBD	LH$_2$ Tank Vent Valve Close OFF	S-IVB		TBD
TBD	LOX Tank Vent Valve Open OFF	S-IVB		TBD
TBD	LOX Tank Vent Valve Close ON	S-IVB		TBD
TBD	LOX Tank Vent Valve Close OFF	S-IVB		TBD

CHAPTER 5
MEASURING AND TELEMETRY

SECTION 5.1
INTRODUCTION

The combined measuring and telemetry system is essentially an independent operating subsystem within the overall Astrionics System.

The combined measuring and telemetry systems of the Saturn Launch Vehicles measure physical quantities and signals onboard the vehicle and transmit the data to ground stations. The complexity of the launch vehicle and its missions dictate a large number of measurements. The data transmitted by the measuring and telemetry systems supply information for the following operations:

- Automatic preflight checkout of the vehicle.

- Monitoring of vehicle performance during powered flight.

- Postflight evaluation of vehicle performance.

- Monitoring and checkout of the vehicle during orbital flight.

- Verification of commands received in the vehicle from ground stations.

Figure 5.1-1 illustrates the signal flow through the system. The transducers convert the physical quantities to be measured (e.g., pressure, temperature, etc.) into electrical signals. These signals are modified, when necessary, by signal conditioning devices into voltages suitable as inputs to the telemetry system. Some quantities are already in electrical form and need no transducers or conditioning. The measuring distributor feeds the conditioned signals
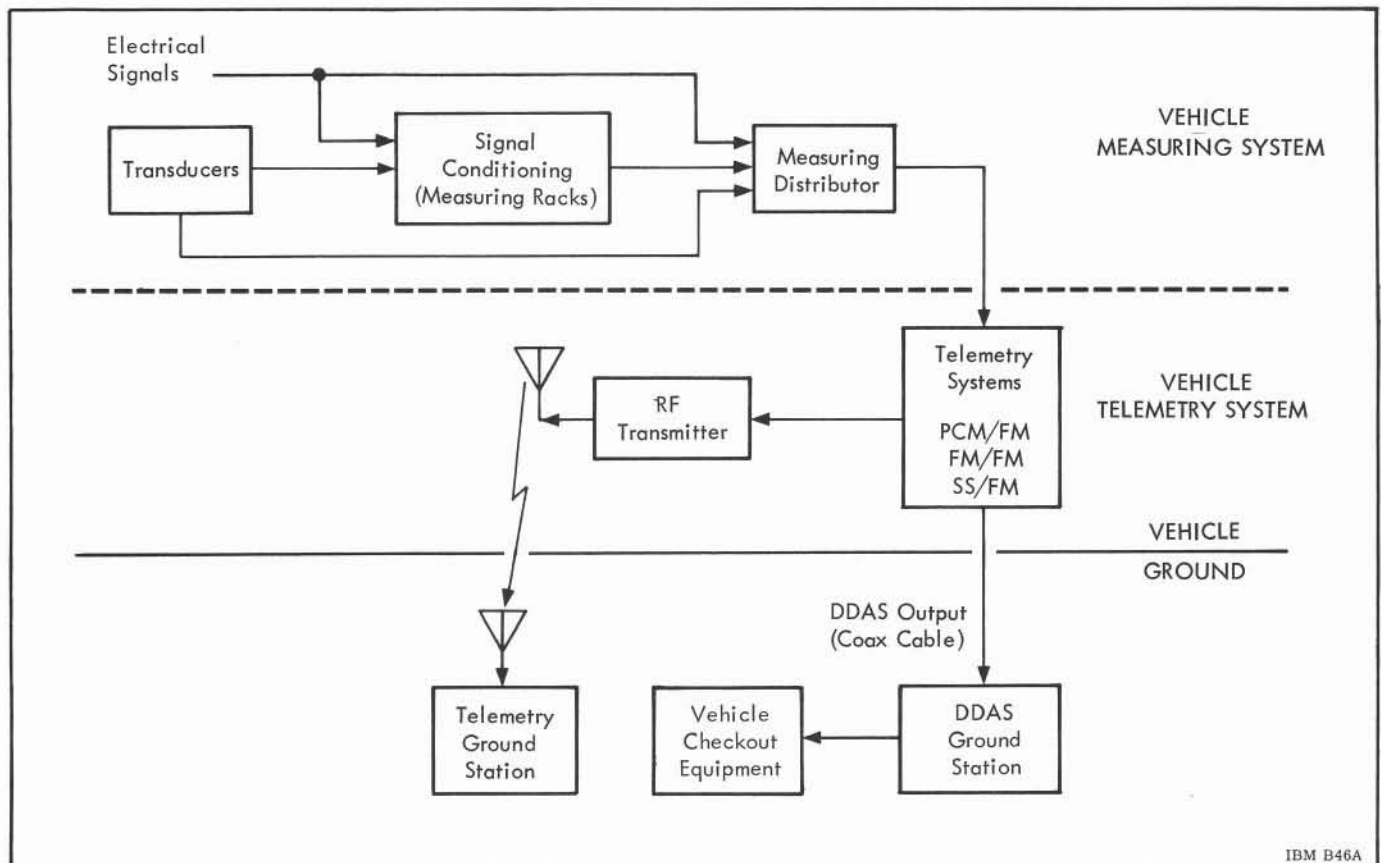
Figure 5.1-1 Measuring and Telemetry System

to the telemetry system. In the telemetry system, the signals are inserted on RF carriers and transmitted during flight to the telemetry ground stations. Before launch, the measuring and telemetry systems send digital data by coaxial cable from each stage of the vehicle to the checkout facility. The digital information is used for automatic checkout of the vehicle on the launch pad. In this mode, the telemetry system operates with the launch site Digital Data Acquisition System (DDAS).

Each stage of the launch vehicle has an independent measuring and telemetry system, DDAS output, and RF transmission equipment. The telemetry system of the S-IVB Stage is also connected to the telemetry system in the IU for the purpose of digital data acquisition and alternate mode transmission.

For monitoring of digital data, the IU telemetry system has an interface with the LVDC through the LVDA. As the measuring and telemetry system in each stage is essentially the same, the IU measuring and telemetry system will be used as an example throughout this chapter.

To simplify vehicle checkout, data handling, etc., standardization is practiced wherever feasible. A remote automatic checkout system is standardized on each stage to provide automatic checkout of measurements from ground equipment.

An airborne television system is also used to transfer data from the vehicle to the ground. This system is used on Saturn V Vehicles to observe the rocket engines during flight.

SECTION 5.2
MEASURING SYSTEM

5.2.1 GENERAL

The measuring system includes transducers, measuring racks (which contain the signal conditioning modules), measuring distributors, measuring rack selectors, and measuring voltage supplies. The measuring system converts the signal or quantity to be measured into an electrical signal that is acceptable to the telemetry system. According to this definition, the measuring system ends at the input to the telemetry system (at the subcarrier oscillator, multiplexer, etc.). Figure 5.2-1 illustrates typical components of the measuring system. The measurements in the launch vehicle cover the areas of:

Propulsion   Guidance and
Structure   control
Flight mechanics   Environment

Physical quantities to be measured, such as, pressure, temperature, and vibrations must be transformed by transducers into electrical signals for transmission by telemetry. Measurements of electrical signals (voltages, currents, and frequencies) originating in onboard equipment are used for monitoring the performance of this equipment and the sequence of flight events (e.g., stage separation, engine cutoff, and others). The signals to be measured exist in analog and digital form. Measurements of all electrical signals do not require transducers.

The measurements made in the various stages of Saturn IB and V Launch Vehicles are listed in Table 5.2-1. The number and type of measurements given in the table are best estimates and are subject to changes as the development of vehicles proceeds. This applies particularly to the figures for operational vehicles since a larger number of measurements is necessary for performance evaluation and testing of research and development vehicles. Because of the large number of different measurements, no detailed list of measurements is given. Table 5.2-2 indicates some of the quantities measured in a typical Saturn V Vehicle.

Certain measurements will be used as inputs to the emergency detection system (see Chapter 4). Operational measurements are those measurements used for flight (mission) control. Operational measurements indicate the state of the vehicle and its systems and also provide trend information (beginning with the orbital flight phase). All operational measurements are transmitted over the PCM telemetry links of the S-IVB Stage and the IU. Typical operational measurements for the Saturn V Vehicles are given in Table 5.2-3.

5.2.2 TRANSDUCERS

Generally, transducers change energy from one form to another. In measuring techniques, they are used as electromechanical measuring instruments which contain sensing devices for converting mechanical and environmental quantities into electrical signals. Evaluation of vehicle performance and inflight monitoring requires the measurement of a large variety of physical quantities onboard the vehicle. Therefore, many different types of transducers are used. Some typical transducers and the corresponding measurements are listed in Table 5.2-2. Because of the many types of transducers and measurements used in launch vehicles, no attempt is made to give a complete description of all transducers on each stage. The following measurement descriptions generally apply to the S-IB and S-IC Stages. The number of transducers used in each stage may be found in Table 5.2-1. Figure 5.2-2 illustrates several typical transducers.

BENDING MODE VIBRATION MEASUREMENTS (ALL STAGES)

Bending measurements are made using force-balance accelerometers. The principle of operation of the force-balance accelerometer is shown in Figure 5.2-3. These instruments operate as a subminiature servosystem, which is responsive to linear acceleration along its sensitive axis. Due to inertia, the acceleration sensitive mass, commonly called the paddle wheel, moves relative to the position de-
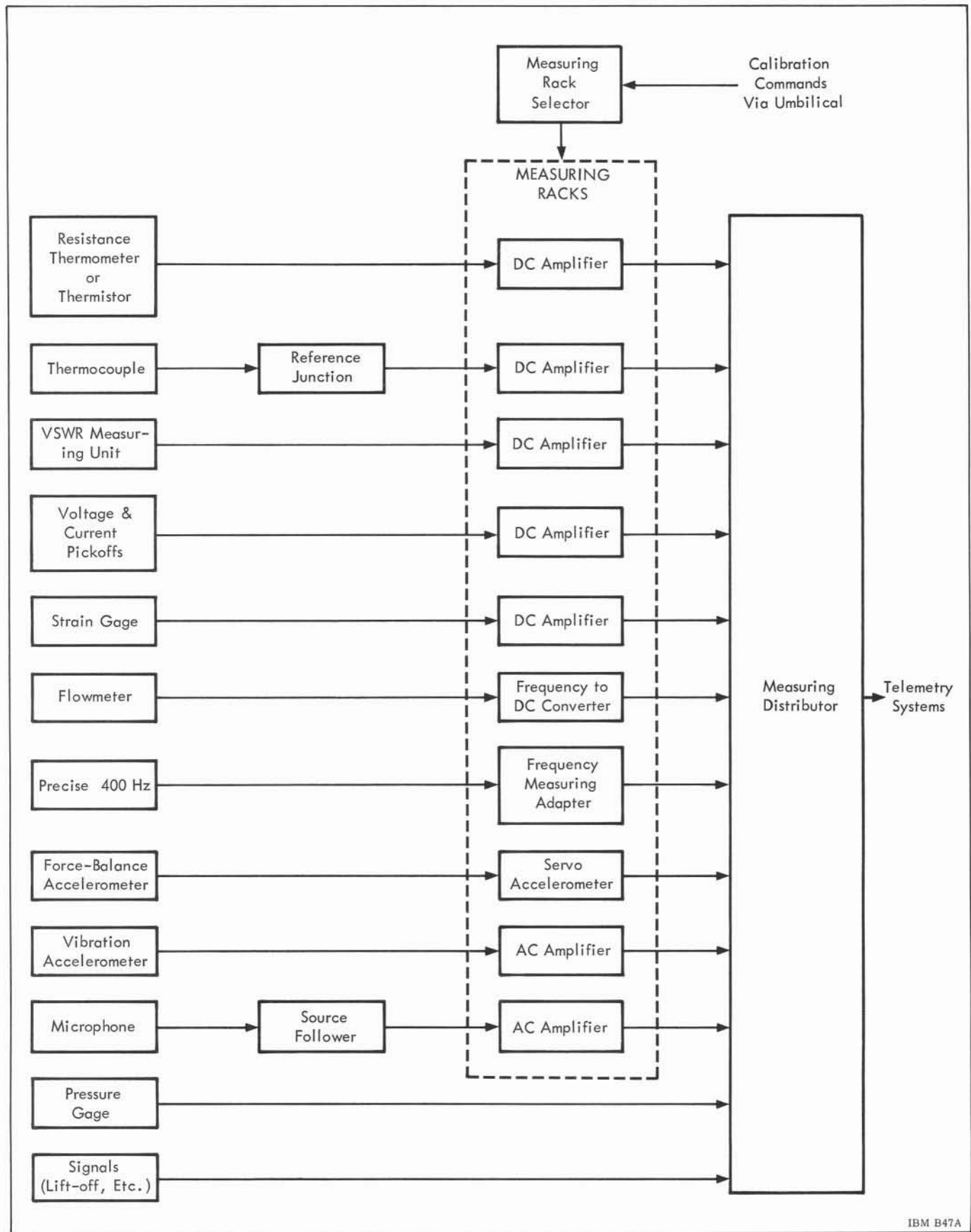
Figure 5. 2-1 Typical Saturn Measuring System

Table 5.2-1 Number of Measurements, Transducers, and Measuring Racks (Typical)

SATURN IB						
Stage	Measurements		Transducers		Measuring Racks *	
	R & D	Oper	R & D	Oper	R & D	Oper
S-I B	542	263	305	106	19	5
S-IV B	432	218	298	115	17	7
IU	278	134	92	45	9	4
Total Vehicle	1252	615	695	366	45	16
SATURN V						
Stage	Measurements		Transducers		Measuring Racks *	
	R & D	Oper	R & D	Oper	R & D	Oper
S-I C	901	297	723	157	27	5
S-II	913	446	589	180	35	8
S-IV B	432	218	298	115	17	7
IU	278	134	92	45	10	4
Total Vehicle	2524	1077	1702	497	89	24

* Signal conditioning panels

tector when acceleration is applied. The position-error detector and servoamplifier generate a feedback signal to the restoring mechanism. The electromechanical servoaction results in a balance between the input force proportional to the acceleration and the feedback force proportional to the current in the restoring coil. The restoring current, or the voltage it develops across a series resistor, provides the output of the accelerometer, and is a precise measurement of acceleration.

The advantage of the force-balance acceler-ometer over earlier potentiometer type accelero-meters is that friction is almost negligible. The force-balance accelerometer will sense both extremely slow vibrations and continuous acceleration in a single direction. The same instruments are employed to sense lateral acceleration along the pitch and yaw axes in the IU. Similar measurements are also made

to determine the bending or flapping actions of the vehicle fins.

FLIGHT MECHANICS MEASUREMENTS

Longitudinal acceleration measurements are made to measure the thrust decay of engines. These measurements are made with the same type of acceler-ometers that are used for strain and vibration meas-urements. Longitudinal acceleration measurements to determine thrust and thrust decay of the S-IVB Stage are made in the IU.

Angular velocity measurements are made using Rate Gyros (see Chapter 3). The instrument is accurate within a range of ±10 degrees per second. In case of a catastrophic situation, the vehicle also carries Rate Gyros to measure angular movement in excess of 100 degrees per second along the pitch and yaw axes.

Propulsion system measurements are made by pressure transducers, flowmeters, tachometers, and liquid level sensors. A brief functional description for each is given in the following paragraphs.

PRESSURE MEASUREMENTS

Pressure measurements are made by two different methods: potentiometric and strain gage.

The potentiometric transducer is basically a wirewound potentiometer with the wiper mechanically connected to a pressure controlled diaphragm as illustrated in Figure 5.2-4. The output voltage is obtained by applying 5 Vdc to the potentiometer. The magnitude of the output will vary from 0 to 5 volts, depending upon the position of the wiper. Since the output voltage is within the limit required by the telemetry subsystem, no signal conditioning is required.

Table 5.2-2 Typical Saturn V Measurements

Quantity Measured	Typical Transducer
Acoustic energy	Microphone
Temperature	Resistance thermometer, thermocouple, or thermistor
Pressure	Bourdin tube or bellows pressure transducers
Vibration	Piezoelectric accelerometer
Liquid flow rate	Turbine-type flowmeter
Liquid level	Level probe
Strain	Strain gauge
RPM (turbopump)	Tachometer
Acceleration	Force balance accelerometer
Position	Potentiometer
Angular velocity	Rate gyro
Voltage, current, and frequency	——
Signals	——
Guidance and Control Signals	——
RF & telemetry signals	——

Table 5.2-3
Typical Saturn V Operational Measurements

S-IVB Auxiliary Propulsion System

Propellant level
Helium supply pressure
Propellant temperature

S-IVB Main Propulsion System

Gas supply pressure
Propellant valve positions
Propellant masses
Engine sequencing signals
Tank pressure
Propellant flow
Thrust chamber pressure
Hydraulic system pressure and fluid temperature

S-IVB/IU Electrical System

Battery voltages and currents
Bus voltages and currents
Inverter voltages

S-IVB Attitude Control and Stabilization System

Attitude control signals (pitch, yaw, roll)
Angular velocity (pitch, yaw, roll)
Engine actuators position
Control computer temperature
Servo amplifier signals
Spacial amplifier signals
Valve positions

Navigation, Guidance, and Digital Signal

Steering commands (pitch, yaw, roll)
Space fixed velocity (3 components)
Space fixed displacement (3 components)
Computer time
Time to S-IVB second ignition and cutoff
Accelerometer output (X, Y, Z)
Gyro servo output (X, Y, Z)
Gimbal angles (pitch, yaw, roll)
Launch Vehicle Digital Computer signals
Temperatures of computer and platform components
Air bearing supply pressure
Switch Selector output

Environmental Control System

Pressure and temperature of coolant at several places
Cold plate temperature
Gas supply pressure

Instrumentation and Communication System

Power output of the PCM transmitter (S-IVB & IU)

Approximately 70 per cent of the total pressure measurements are made with the potentiometric type transducer.

On the S-IB Stage most of the potentiometric type pressure transducers may be calibrated while on board the vehicle by using the calibration valve that is associated with each transducer. The calibration valve, connected between the transducer and the source of pressure being measured, may be coupled to a calibration line that is connected through a quick disconnect coupling to a controlled pressure source. When the calibration line is connected to the valve, the transducer sensor is mechanically positioned to sense the controlled pressure. The transducer automatically returns to sense the system pressure when the calibration line is removed. All other type pressure transducers are calibrated electrically.

In addition to the potentiometric pressure transducer, pressure measurements are made with unbonded strain gages (Figure 5.2-5). Each strain gage transducer consists of four individual strain gages mechanically connected to a pressure sensitive diaphragm and arranged in such a manner that compression on two of the gages results in tension on the
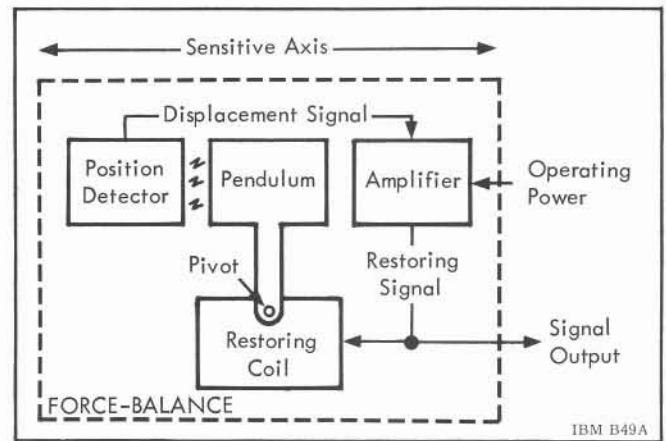
Figure 5.2-3 Force-balance Accelerometer Block Diagram

other two. This action, in effect, results in an output when the strain gage transducer is excited by an external voltage. The output voltage of the strain gage transducer is approximately 40 millivolts at full pressure; therefore, the signal must be fed into a signal conditioner (dc amplifier) before it can be applied to the subcarrier oscillator. In some cases, the signal conditioning is incorporated in the transducer package.
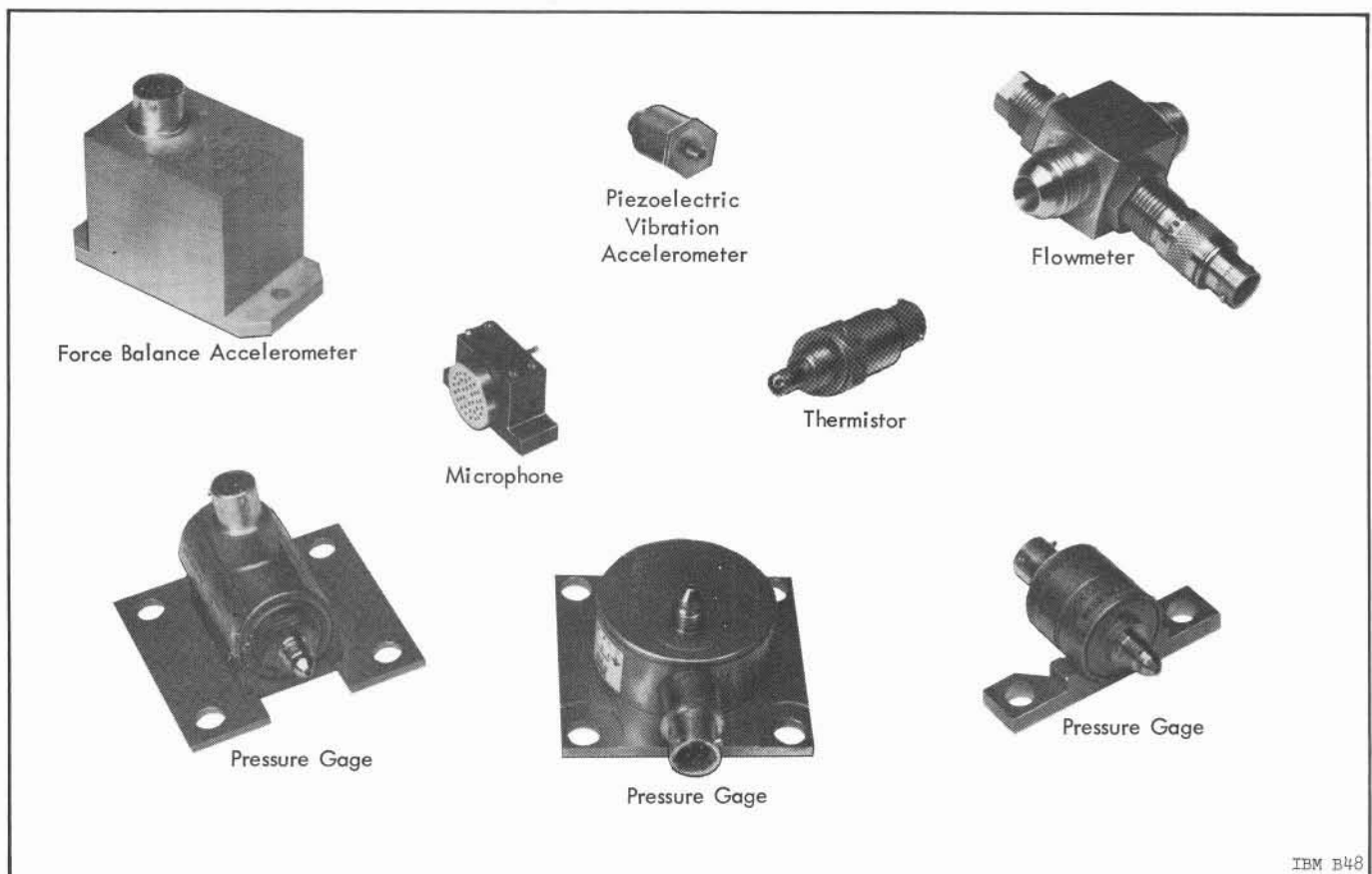
Figure 5.2-2 Typical Transducers
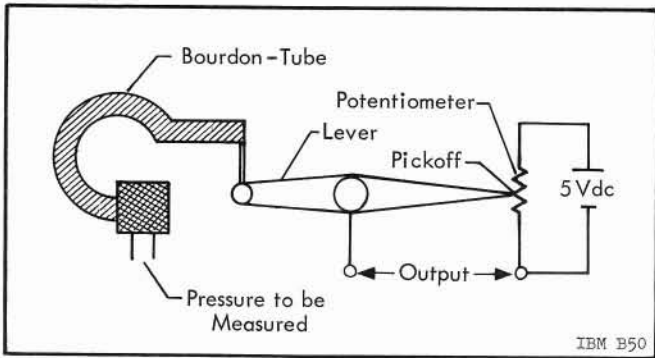
Figure 5.2-4 Bourdon-tube Potentiometer Type
Pressure Transducer

RPM MEASUREMENTS (S-IB STAGE ONLY)

This measurement utilizes a variable reluc-
tance type of tachometer mounted on the turbine. As
the turbine turns, the varying reluctance of a magne-
tic path generates an ac voltage. The frequency of
the output voltage is proportional to the rpm of the tur-
bine. This frequency is too high for the bandwidth of
the available telemetry channels; therefore, signal
conditioning is required to divide the frequency by 32
to make the output signal compatible with available
telemetry channels.

The signal conditioner is a magnetic frequency
divider which uses the rectangular hysteresis charac-
teristics of a saturable core.

LIQUID LEVEL MEASUREMENTS (S-IB STAGE ONLY)

The liquid level of the propellant is measured
by two different methods: the discrete and the con-
tinuous method. The discrete method (Figure 5.2-6)
utilizes a photo-electric cell to obtain signal output.
The sensor consists of a light source, photo-electric
cell, and a prism. Because the liquid diffuses the
light rays, no output is obtained until the level of the
liquid drops below the prisms. As the level of the
liquid drops below the prism, light rays from the
light source are reflected back to the photo-electric
cell by the prism faces, which are at 45-degree angles
to the light rays. Fifteen sensors are located along
the length of the tank. When the output from one sen-
sor is detected, it is possible to determine the amount
of propellant (fuel or lox) that was consumed in the
time interval that lapsed between outputs from this
sensor and one that was previously initiated.

The continuous method of measuring liquid
level is used in the lower portion of the propellant
tank to determine the amount of propellants remain-
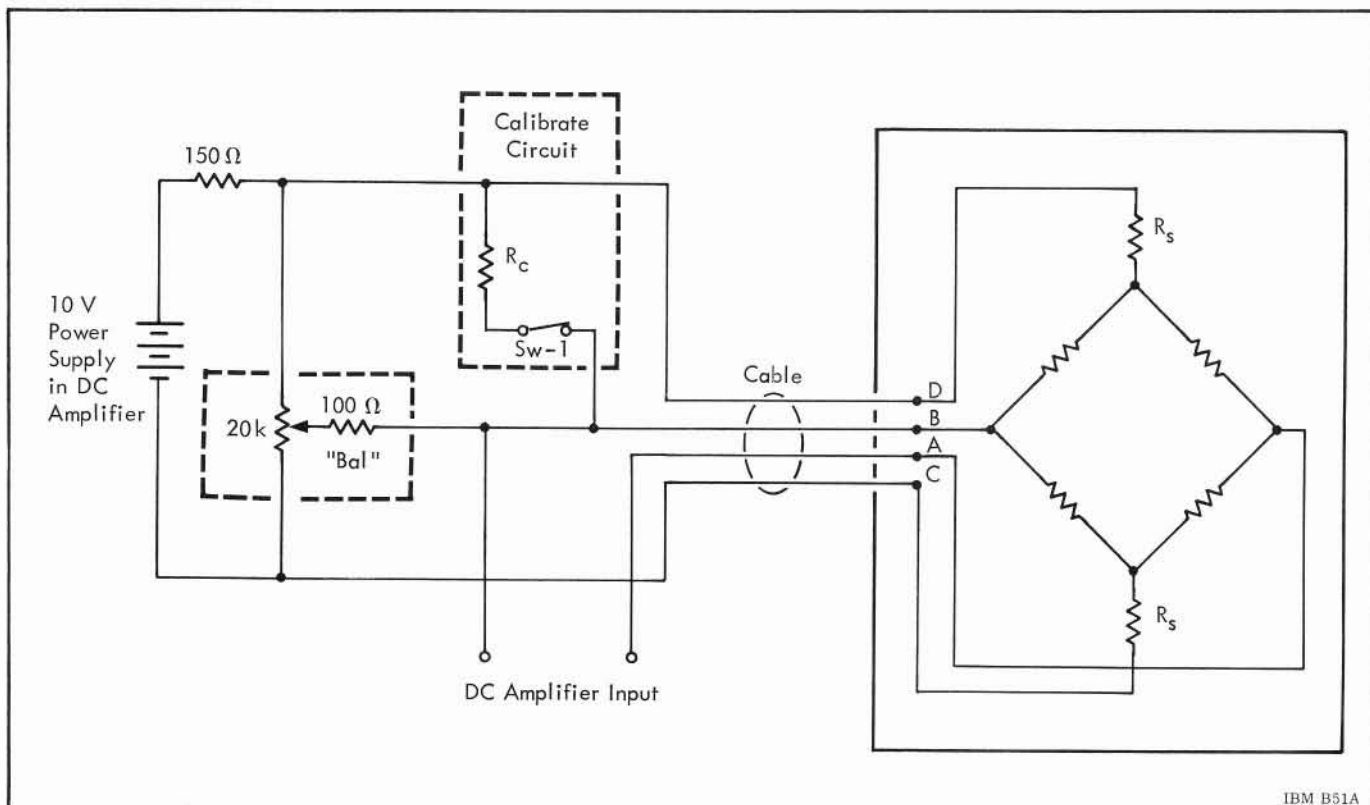
Figure 5.2-5 Strain-gage Type Pressure Transducer
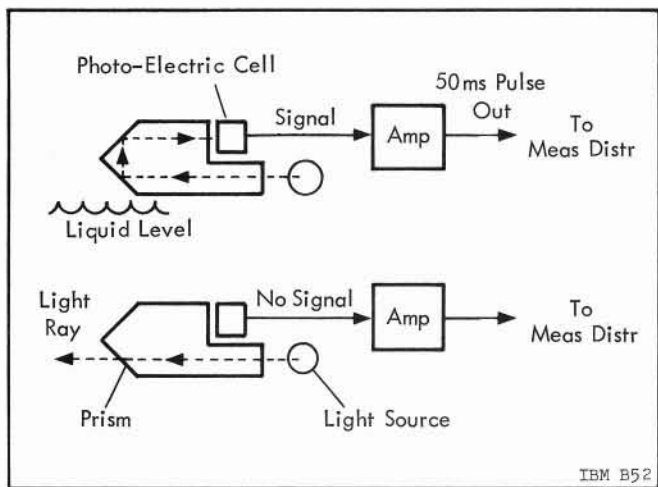
Figure 5.2-6 Liquid Level, Discrete, Functional Diagram

ing after engine cutoff. Principles of operation of the continuous method are illustrated in Figure 5.2-7. The sensor consists of two tandem capacitors with the dielectric being the propellant. The top capacitor (C_s) is approximately 40 inches in length and exhibits a capacitance proportional to the height of the liquid
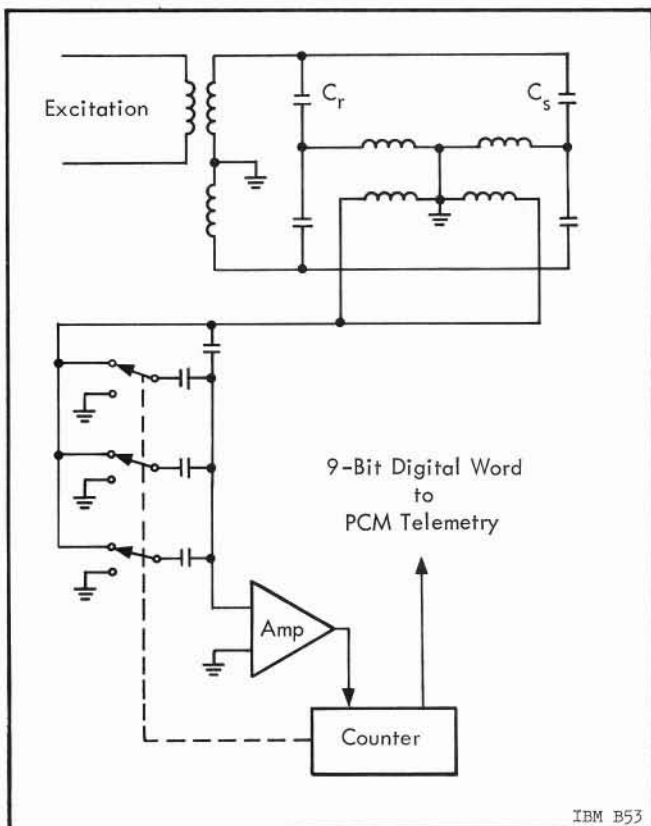
Figure 5.2-7 Liquid Level Sensor Electrical Schematic

and the dielectric constant of the liquid. The bottom capacitor (C_r) is approximately 3 inches in length and is submerged in propellant when a measurement is made. The capacitance of C_r is proportional to the dielectric constant of the liquid.

Each capacitor is connected in a bridge circuit. The outputs of the two bridge circuits are connected as the excitation to a third bridge circuit which is always kept in balance by an electronic servo system. The feedback elements of the servosystem are nine binary weighted capacitors controlled by a counter. The counter provides a parallel binary output for telemetry.

FLOW RATE MEASUREMENTS

Flowmeters convert the rate of flow of a fluid (amount per unit time) within a pipe into an electrical signal. The rate of flow is represented by the frequency of the signal. The output signal must be signal conditioned to form a 0 Vdc to 5 Vdc signal. Most flowmeters are equipped with a mechanical locking device to prevent damage to the turbine during purge operations.

The flowmeter casing (see Figure 5.2-8) consists of a short piece of pipe the same size as the piping being measured. Within the casing is a free-turning turbine. External to this casing is a coil surrounded by a permanent magnetic field. Fluid pumped through the piping and flowmeter causes the turbine to turn at a rate directly proportional to the velocity of the fluid. The metal turbine blades disturb the magnetic field, thus inducing a current into the coil. This current is changed each time a turbine blade disturbs the magnetic field. The frequency of
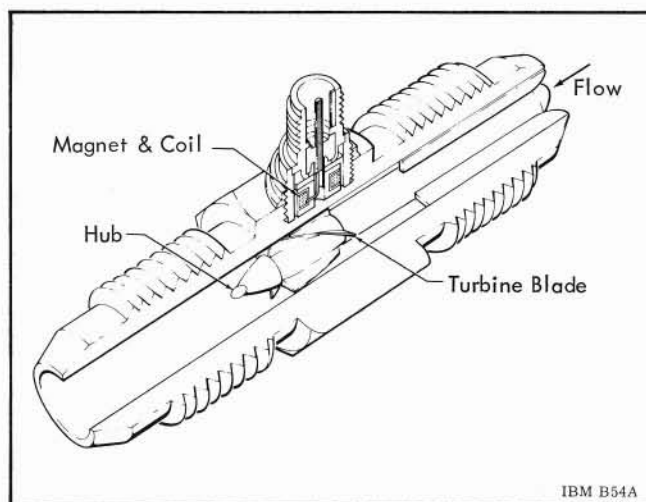
Figure 5.2-8 Basic Principles of a Flowmeter

the current change is proportional to the turbine speed which is proportional to the fluid velocity. The ac signal output is sent to a dc amplifier signal conditioning module or to a frequency to dc converter signal conditioning module.

TEMPERATURE AND RADIATION MEASUREMENTS

The principal temperature and radiation measurements are in the following areas:

- Cryogenic temperature measurements

- Heat flux (radiation and convective)

- Temperature measurements

- Fire detection temperature measurements

- General environmental temperature measurements

Cryogenic Measurements. An important factor in engine operating efficiency is the density of the oxidizer and the fuel. As density is inversely proportional to temperature, temperature measurements are performed at critical points in these systems during flight. The operating efficiency of the engines can be determined from these and other measurements.

Cryogenic measurements are performed by means of resistance thermometers located in the fuel and lox tanks and in the plumbing for both of these systems. These positive temperature-coefficient resistors are fabricated of high purity platinum and require precise calibration techniques.

The maximum temperature range of these devices is from -200°C to +300°C (-328°F to +572°F). Data reduction is accomplished through a straightforward temperature versus resistance method. As in the other temperature measurements, dc amplifiers are employed to increase the output signal level to the required 0 to 5-volt telemetry input level.

Heat Flux (Thermal Radiation) Measurements. Calorimeters are located at strategic points on the engine heat shield to measure thermal radiation and thus permit a precise calculation of the minimum insulation weight allowable at these points.

One type of calorimeter measures both thermal radiation and convective heat transfer. The heating rate is determined from the temperature versus time characteristics of the copper slug.

Another type of calorimeter utilized in some measurements on the Saturn Vehicle is the "thin film" or "membrane" calorimeter. This calorimeter employs a thin disc of constantan welded to a copper heat sink. An insulated copper lead is attached to the center of the disc and another insulated copper lead is attached to the copper heat sink. The temperature differential between the center of the disc and the circumference of the disc produces the emf output which is telemetered to the ground. The emf output of the membrane calorimeter is a linear function of the incident heating rate with the slope of the output curve being determined by the physical size of the constantan disc.

Fire Detection Measurements (S-IB and S-IC Stages Only). The prelaunch fire detection system is operative during the period prior to vehicle lift-off. When the air temperature, measured at critical points, rises at a rate beyond a specified value, an alarm signal is initiated in the blockhouse and an automatic engine cutoff occurs.

A series of thermocouples, arranged in loops with several thermocouples in each loop, are used. The thermocouples are located in back of the heat shield and in the engine compartment. The rate of rise of the temperature of the air is measured in the immediate vicinity of the thermocouple, rather than the radiated heat from some point remote from the measuring device. The system does not define the exact location because of the series loop configuration.

General Temperature Measurements. Surface temperature measurements are made with thermocouples, thermistors, and resistance thermometers. Thermocouples are normally used when the measuring range is 150°C (302°F) or greater, and resistance thermometers and thermistors are used for ranges of less than 150°C. The resistance thermometers are extremely accurate positive temperature coefficient resistors. Thermistors are semiconducting devices which exhibit a high-negative-temperature coefficient of resistance. Ambient air measurements down to -20°C (-4°F) are also performed with thermistors.

Thermocouples and Zone Boxes. Three types of thermocouples are used on the vehicles. In the circuit consisting of 2 metals, an emf will be produced at the measuring junction if a temperature difference exists between the measuring and the reference junctions. The 3 types in use are: chromel/alumel, iron/constantan, and platinum/platinum 10 percent rhodium.

Thermocouples provide measurements of surface heat, fuel temperature, ambient air temperatures, and are used as fire detection gages.

In a laboratory, the reference junction is usually maintained at 0°C (32°F). For vehicular purposes, such a temperature limitation is impractical; therefore, an alternate method has been selected. Changes in reference junction temperature during flight are compensated for electrically by means of a bridge circuit with a resistance thermometer in one leg. A typical bridge circuit with zone box is shown in Figure 5.2-9. The reference junction is contained within the zone box. The zone box is the junction between the vehicle network wiring and the thermocouple leads. A resistance thermometer is in thermal contact with the reference junction. The resistance thermometer has a known positive temperature coefficient. Since the resistance thermometer controls the resistance of one leg of the bridge, the output emf of the bridge circuit will follow the temperature induced emf output of the thermocouple (reference junction). The bridge circuit is so calibrated that it will cancel any emf produced by the reference junction as a result of an increase or decrease in temperature. The dc amplifier increases the output signal level to the required telemetry input level of 0 to 5 volts.

VIBRATION MEASUREMENTS

Vibration measurements are made to determine the structural strength and stability of the Saturn Vehicle. Vibration and stress sensors are mounted on the propellant tanks, turbopumps, engine mountings and engine combustion chamber, and at structural junction points of the vehicle. Two types of accelerometers are used as vibration sensors on board the vehicles; the piezoelectric and the strain gage types.
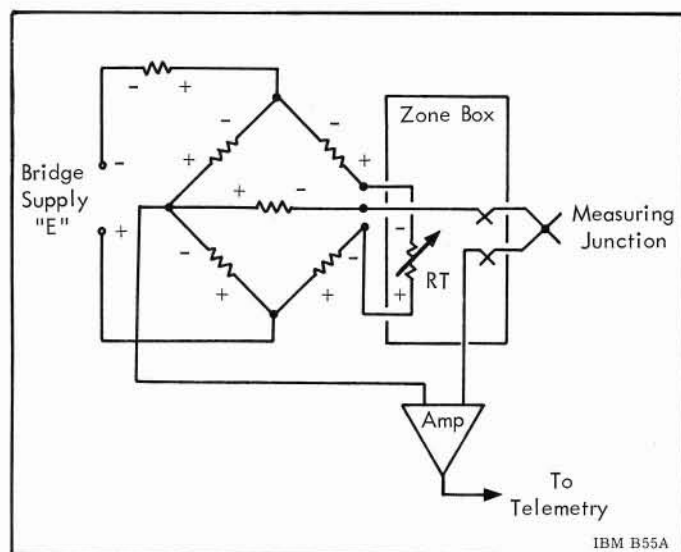
Figure 5.2-9 Typical Bridge Circuit

Piezoelectric Accelerometer. Because of its small size, light weight, and frequency response, the piezoelectric accelerometer is extensively used for flight vibration instrumentation. The frequency response of a typical piezoelectric accelerometer ranges from about 5 hertz to several kilohertz. For flight measurements, the useful frequency response is limited to the bandwidth of the telemetering channel. This type of accelerometer will not respond to near constant acceleration. The effects of vehicular acceleration are thus eliminated from the measurements, and only local vibration is detected by the sensor.

The piezoelectric accelerometer (Figure 5.2-10) consists of a seismic mass which applies a force to a piezoelectric crystal, causing it to generate an electrical signal. The electrical output is proportional to the force applied to the crystal and is an indication of acceleration or vibration. The high-impedance output of the device is coupled to a high-input-impedance emitter-follower stage to maintain adequate low-frequency response.

The emitter-follower can be integral to the accelerometer case, or it can be physically removed but joined electrically through a coaxial cable. Typical sensitivity is in the order of 32 mv/g. Temperature environment is the determining factor in the choice of crystal material to be used. A sufficiently broad range of sensor types is available to permit vibration measurements to be made in almost any temperature environment. The accelerometers are capable of operating up to an acceleration level of several hundred g's and to a lower limit determined by the associated electronic circuits. In flight applications, the range is normally from ±3g to ±70g. Tests have indicated that the acoustic environment encountered on the vehicle will have negligible effect on accelerometer output. This is the sensor most frequently used for vibration measurements.

Strain Gage Accelerometer. The accelerometer (Figure 5.2-11) consists of a mass suspended from strain-sensitive wires connected in a Wheatstone bridge. When a force is applied to the mass, the resistances of the supporting wires change, thus unbalancing the bridge and causing an output voltage. The heavy arrows show the lower resistance circuit of an unbalanced bridge. The extent of the unbalance is indicated by the increase in voltage at the meter. In practice, a bias voltage is used to produce a 2.5-volt potential at the output of the amplifier when the mass is at rest. The bias prevents the output from going negative as the movements of the mass
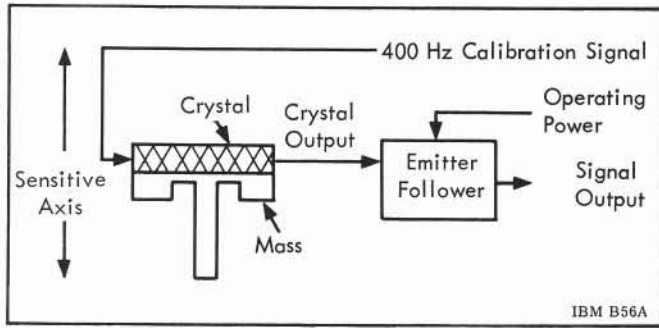
Figure 5.2-10 Piezoelectric Accelerometer
and Emitter Follower

is in both directions. Furthermore, the bias confines
the output voltage to the 0 to 5-volt range.

ACOUSTIC MEASUREMENTS

An acoustic transducer converts a sound into
a proportional electrical signal. The transducer con-
sists of two pieces — a microphone and a source
follower (see Figure 5.2-12). The output signal must
be signal conditioned to form a 0 Vdc to 5 Vdc signal.
As sound is a movement of air molecules, an acoustic
transducer can operate only in the atmosphere.

Microphone - The microphone uses the piezoelectric
effect of a crystalline structure to produce an output.

The unit is sensitive to sound pressure levels of 110 db
to 190 db in the frequency ranges of 50 Hz to 3150 Hz.
The microphone consists of a crystal solidly bonded
to a mass with a sound-sensitive diaphragm attached
to the crystal. The microphone is mounted directly
to the structure with the diaphragm facing the area
to be measured for sound. Sound pressure applied
to the diaphragm causes it to vibrate and this vibra-
tion causes compression-tension forces to be felt by
the crystal. The crystal generates a small electrical
signal proportional to the applied sound pressure.
The amplitude of the signal represents the db level
and the frequency represents the frequency of the
sound. The electrical output of another crystal/mass
assembly (which is mounted opposite the first) is
coupled directly to the output of the first. Unwanted
crystal outputs induced by structural vibrations are
thus cancelled out and are not felt on the microphone
output. The output of the microphone is coupled via
coaxial cable to the source follower.

Source Follower - The source follower electrically
isolates and increases the current of the microphone
output. The source follower is a transistorized unit
less than 3 inches in its longest dimension. The
source follower receives operating power from the
Measuring Rack. The output of the source follower
is routed to an ac amplifier signal conditioning module.
Prior to launch, the source follower and subsequent

Figure 5.2-11 Strain-gage Accelerometer Block Diagram

Figure 5.2-12   Acoustic Transducer Functional Diagram

components can be calibrated by application of a 400 Hz signal.

RF POWER MEASUREMENTS

The VSWR measuring unit is the only rf power transducer used. It contains eight coupler assemblies, each of which measures the transmitted or reflected power of a telemetry transmitter. Each of the eight output signals from the unit is low-level dc and is fed to a dc amplifier signal conditioner module.
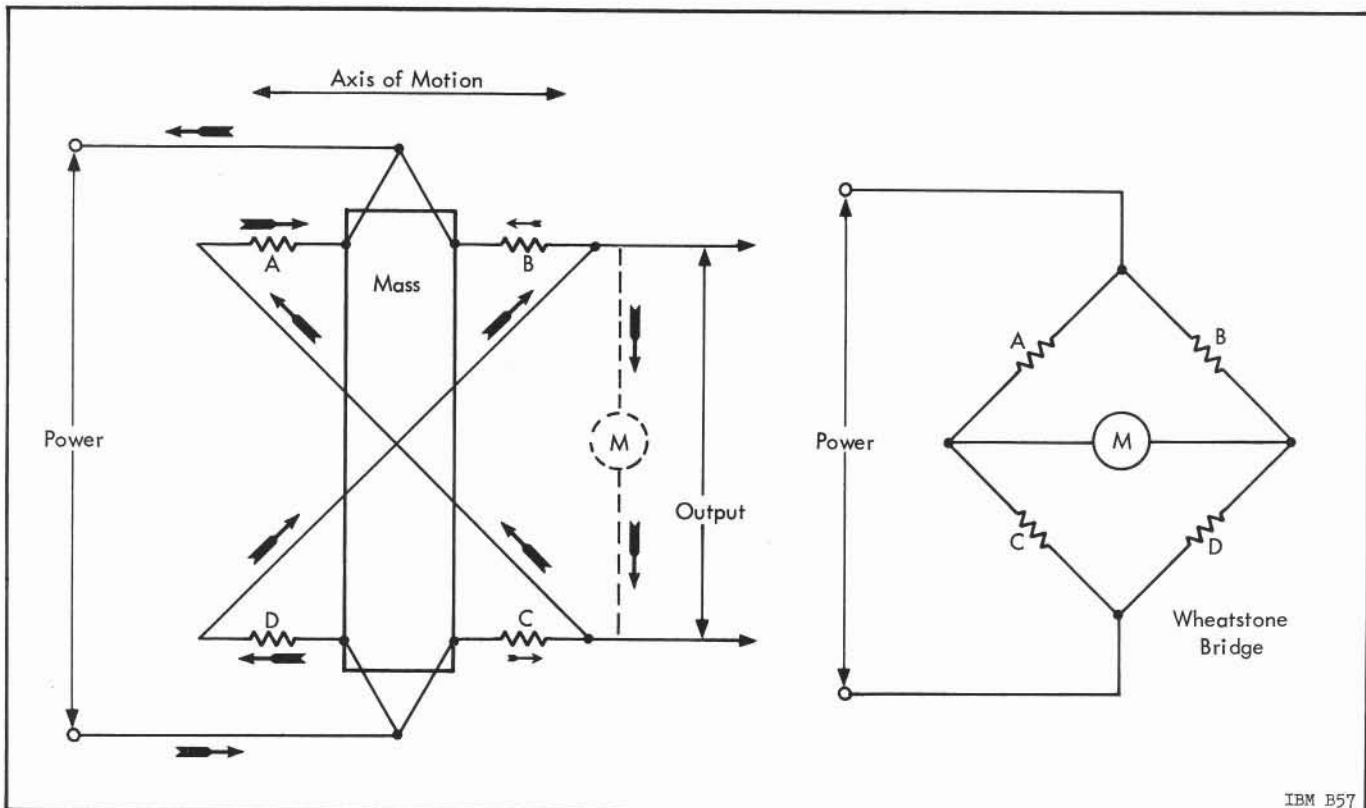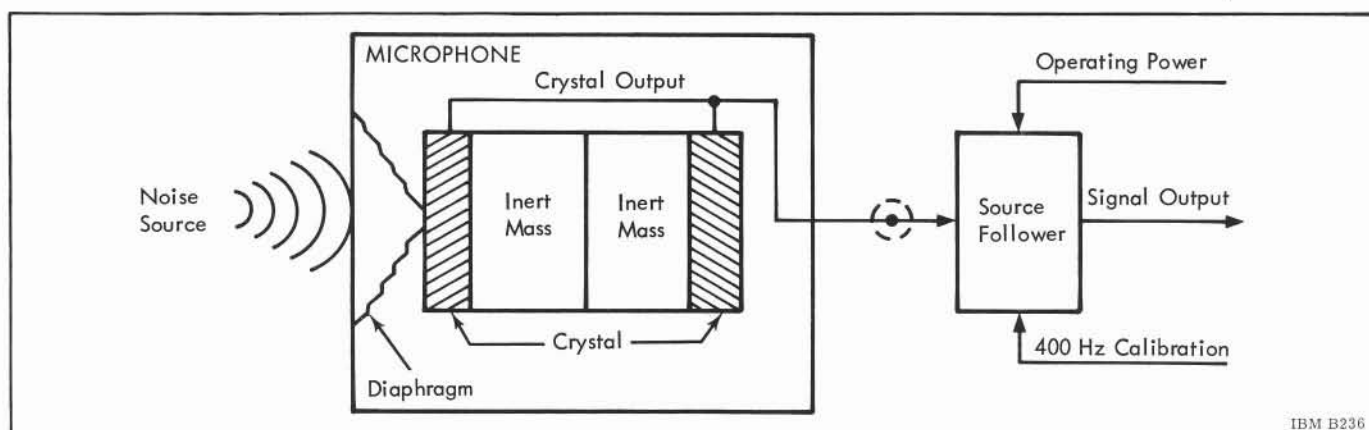
A typical rf input signal (see Figure 5.2-13) is in the frequency range of 230 MHz to 260 MHz, with 22 watts of power. Each measurement is inductively picked off by a coupling loop inserted into the electromagnetic field about the input signal conductor. The loop is inserted so as to be sensitive to energy travelling in one direction only; therefore, one loop is necessary for transmitted power monitoring and another loop for reflected power monitoring. A crystal detector rectifies the voltage in each pickoff line.

5.2.3 SIGNAL CONDITIONING

The following discussion on signal conditioning applies primarily to the Instrument Unit.

The signal conditioning system takes the signal to be measured and converts it to a signal that is acceptable to telemetry. The signal may be an electrical signal from the vehicle or it may be the output from a transducer. Certain transducers have output signals which do not require signal conditioning. These signals are fed directly to the Measuring Distributor.

A modular concept is used in the IU signal conditioning system. This consists of a number of measuring racks, each having a maximum of 20 measuring modules (channels) plus 2 selector channels.

These modules which plug into the Measuring Rack consist of dc amplifiers, ac amplifiers, channel selectors (for calibration) and other modules. The amplifiers have the provisions for plug-in signal conditioning cards to obtain a very flexible measuring system with a minimum number of components.

The rack is a fabricated sheet metal structure with a tight fitting cover. It is not pressurized or hermetically sealed but has a gasket seal on the cover to protect the modules from foreign material. The internal electrical connections are made through a multilayer printed circuit board with flexible integral cable to the external connectors. The multilayer board is also the mechanical support for the measuring module connectors. A typical measuring rack of the type used in the IU is shown in Figure 5.2-14.

Signal conditioning in other vehicle stages is accomplished in a similar way. Signal conditioning modules are mounted in measuring racks containing 20 modules each. In the S-IVB Stage, signal conditioning modules are mounted on panels distributed throughout the stage. The number of measuring racks in each stage is given in Table 5.2-1.

DC AMPLIFIERS

There is only one type of dc amplifier module used on the IU. By making use of individually tailored range cards, this one amplifier is used for measuring temperatures, pressures, currents, voltages, error signals, and other measurements.

The dc amplifier module converts dc measurement signals or low-frequency ac measurement signals into a 0 Vdc to 5 Vdc range. The dc amplifier module physically consists of two parts: a range card and a fixed gain amplifier. The range card plugs into the
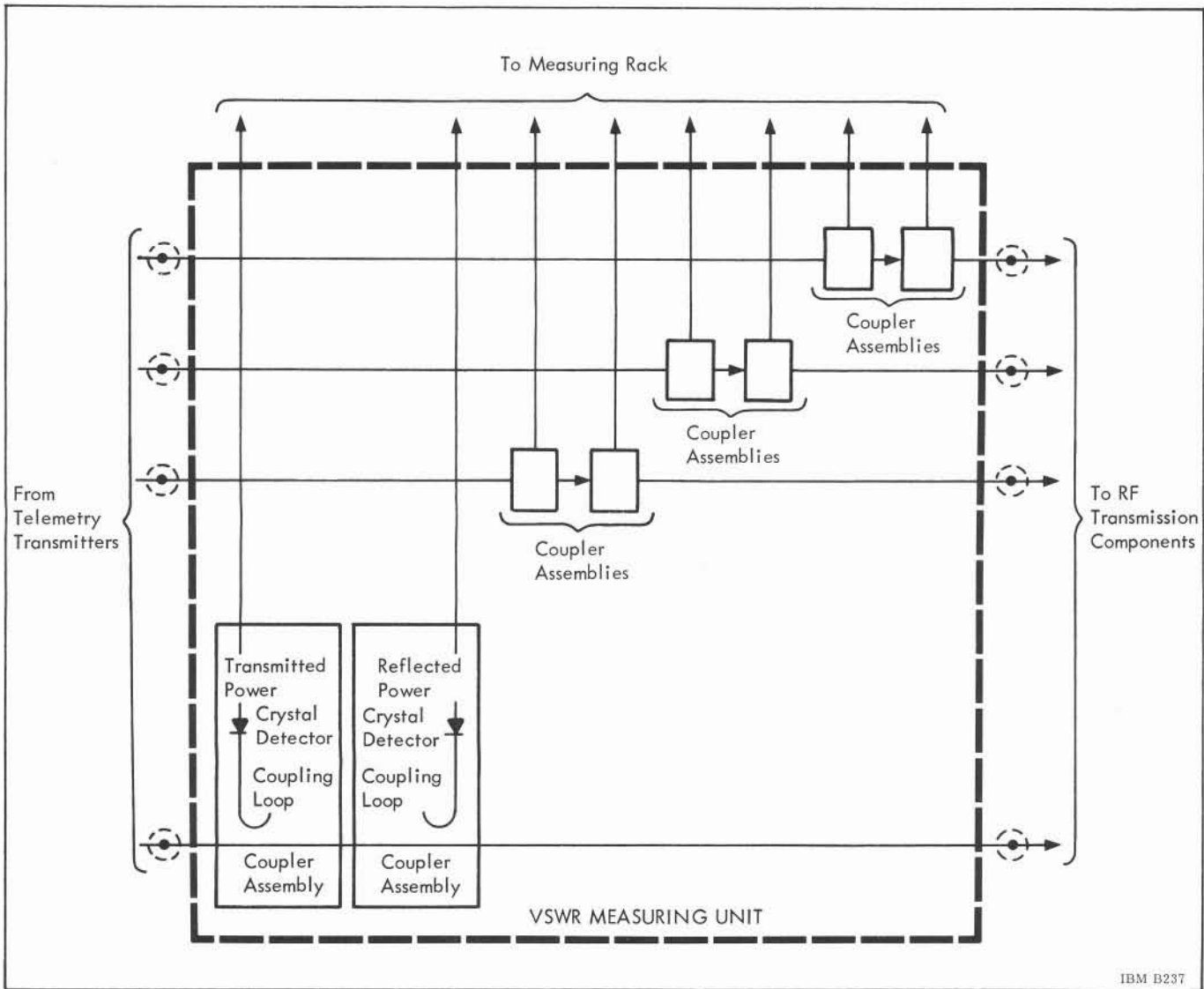
Figure 5.2-13 RF Power Transducer Functional Diagram

amplifier portion. Different types of range cards are selected to adapt the measurement signal to the fixed amplifier. Regulated power used by the range card and the amplifier is generated on the amplifier portion (see Figure 5.2-15). The output regulator provides a precise 10 Vdc to the range card for use by the bridge network and the calibration network.

The input signal to a dc amplifier module is first routed straight through the amplifier portion to the range card. The range card conditions the signal to a 50 mV range and sends it back to the amplifier portion for amplification to a 5 V range. The amplified signal is then routed back to the range card for addition of bias voltage if necessary. From the range card the signal is returned to the amplifier portion and on out to the measurement distribution equipment.

Two relays are provided for checkout and calibration of the amplifier and associated measuring system. These relays are operated remotely through the Remote Automatic Calibration System (RACS).

AC AMPLIFIERS

Only one type of ac amplifier is used in the IU. The ac amplifier has the same type range card as the dc amplifier. It is used primarily for vibration and acoustic measurements.

The amplifier has a gain of 240 and a frequency response of 50 to 3000 hertz. Since practically all vibration and acoustic measurements are on SS/FM telemetry, the lower frequency response limit was designed to be 50 hertz. The output is transformer isolated from the input and ground.

Signal Conditioning Modules
(1 channel per module)

Channel Selector Modules
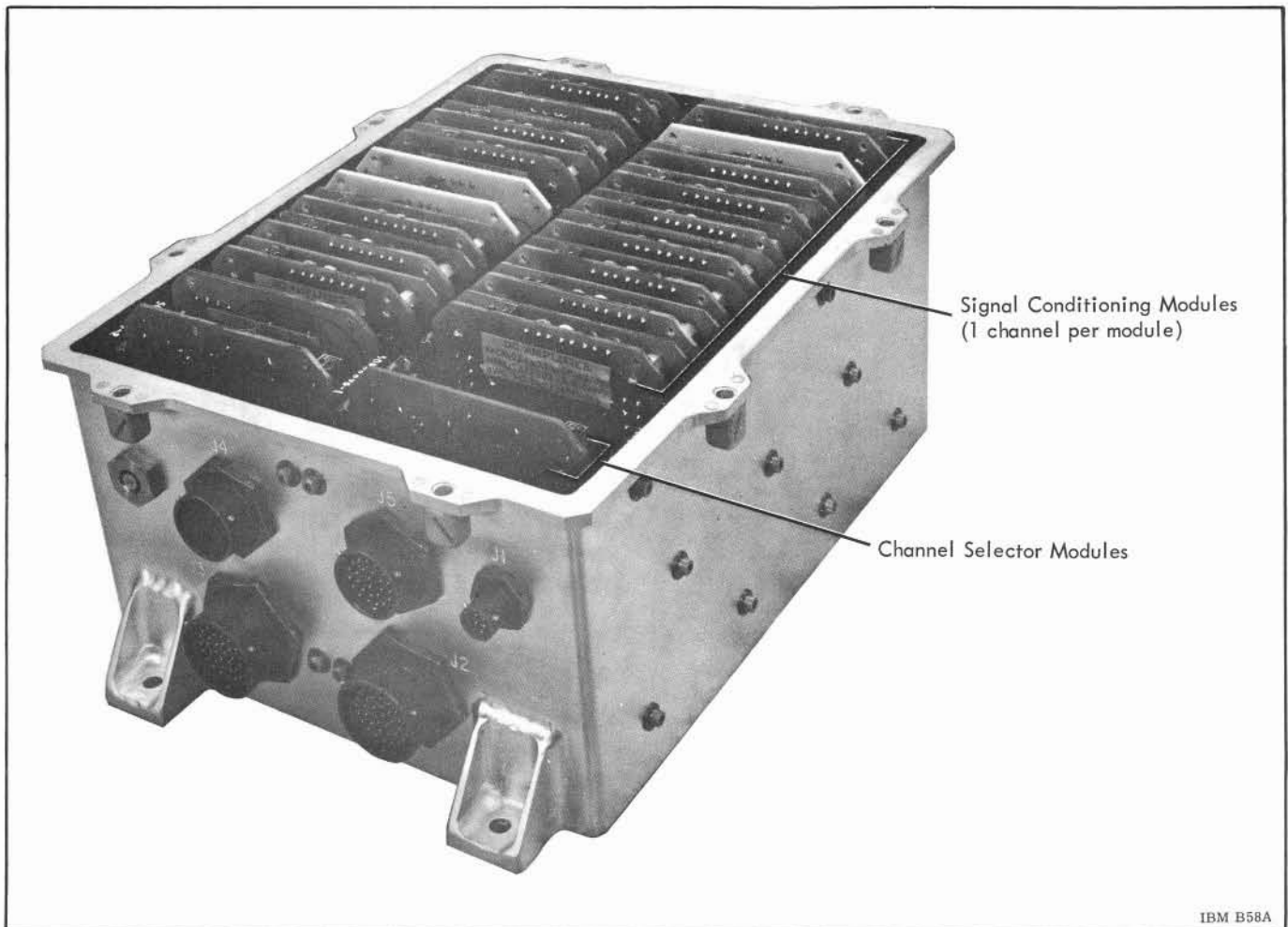
IBM B58A

Figure 5.2-14 Typical Measuring Rack

The amplifier contains an internal dc-to-dc converter to supply an isolated power source for the amplifier and to provide an isolated power source for use with the vibration transducers and emitter followers. The ac amplifier has the same type limiting circuits and calibration relays as the dc amplifier.

RANGE CARD

The range card is a printed circuit card used with the dc and ac amplifiers. The card provides the necessary flexibility to obtain the measurements required on the Saturn Vehicle with a small number of different type amplifiers. Thus, only one type of dc amplifier and one type of ac amplifier are used on practically all the measurements requiring signal conditioning. The range card contains bridge completion resistors, calibration resistors, bias networks gain adjustment, or other conditioning circuits.

Range cards are individually tailored to particular measurements but all range cards fall into two general categories: a voltage divider range card which transforms an input voltage into a proportional 50 mV range and a bridge range card which monitors varying resistive networks and transforms the varying resistance into a proportional 0 to 50 mV range. Bridge range cards are used with resistive type temperature transducers (the transducer is the variable leg of the bridge).

A circuit diagram of a typical bridge card for temperature measurements is shown in Figure 5.2-16. In this case, the transducer is a thermistor. Small range changes can be made by the range adjustment potentiometer. By changing resistors, the range can be changed by a large amount and several ranges of temperature can be measured by using one basic range card. The system can be calibrated by using the Remote Automatic Calibration System (RACS) to shunt fixed resistors across the bridge.
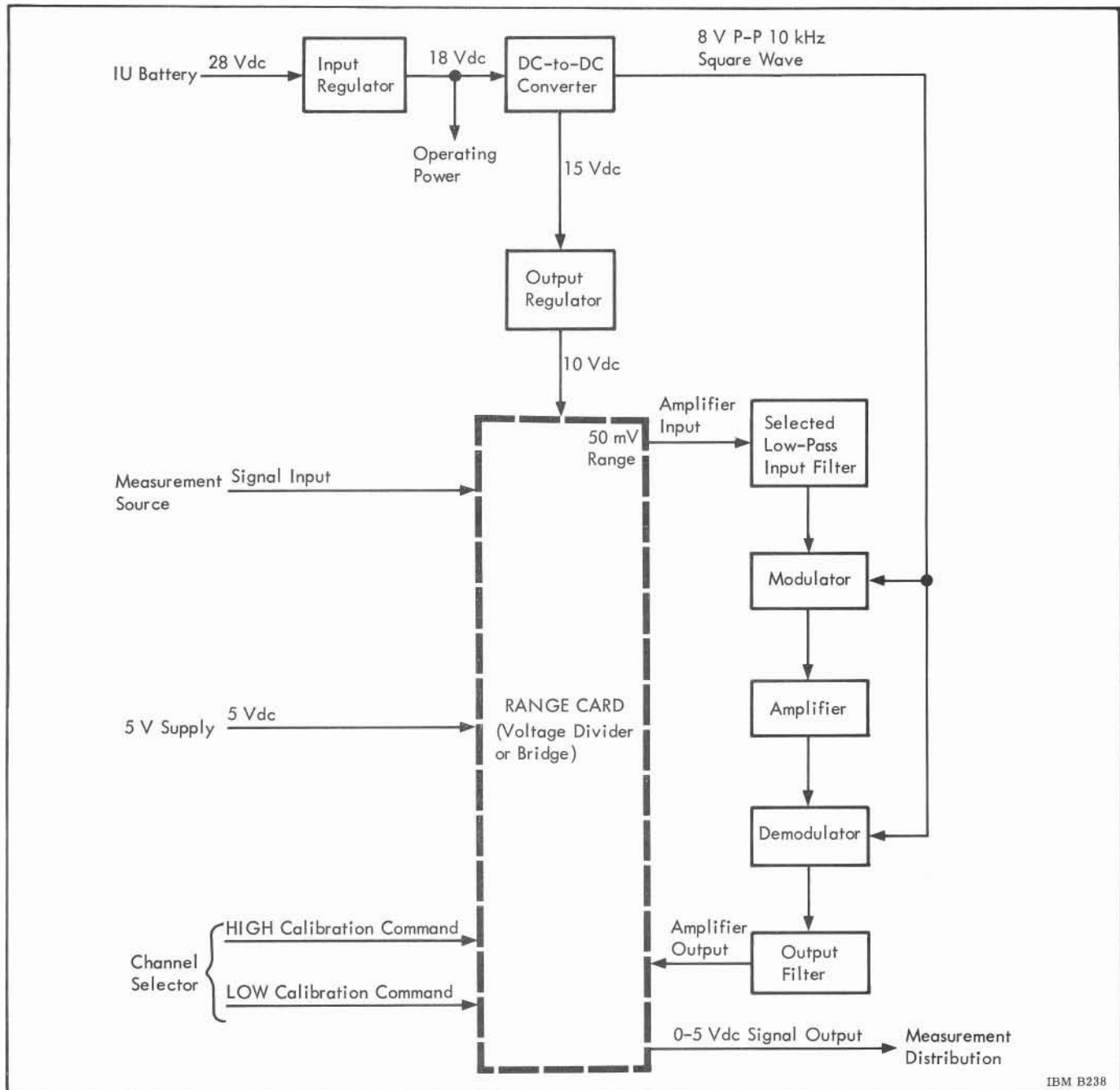
Figure 5.2-15 DC Amplifier Block Diagram

SPECIAL MODULES

There are several measurements that require signal conditioning that are not compatible with the ac or dc amplifier. Special signal conditioning modules are used for these measurements. Examples of these modules are: the frequency-to-dc converter used with flowmeters, the servo accelerometer unit used with the force balance accelerometers, and the frequency measuring unit used to accurately monitor the 400-hertz vehicle supply. Most of the special modules have relays for calibration and checkout.

5.2.4 MEASURING DISTRIBUTION

The measuring distributors accept the 0 to 5-volt output of the signal conditioning modules and routes them to the proper telemetry channels. Most measurements in the measuring system are connected to the distributor to be directed to their preassigned channel. The distributors provide versatility in changing channel assignments; changes are made by physically rearranging jumper wires within the measuring distributors. The versatility of the distributors eliminates extensive cable changes and allows channel
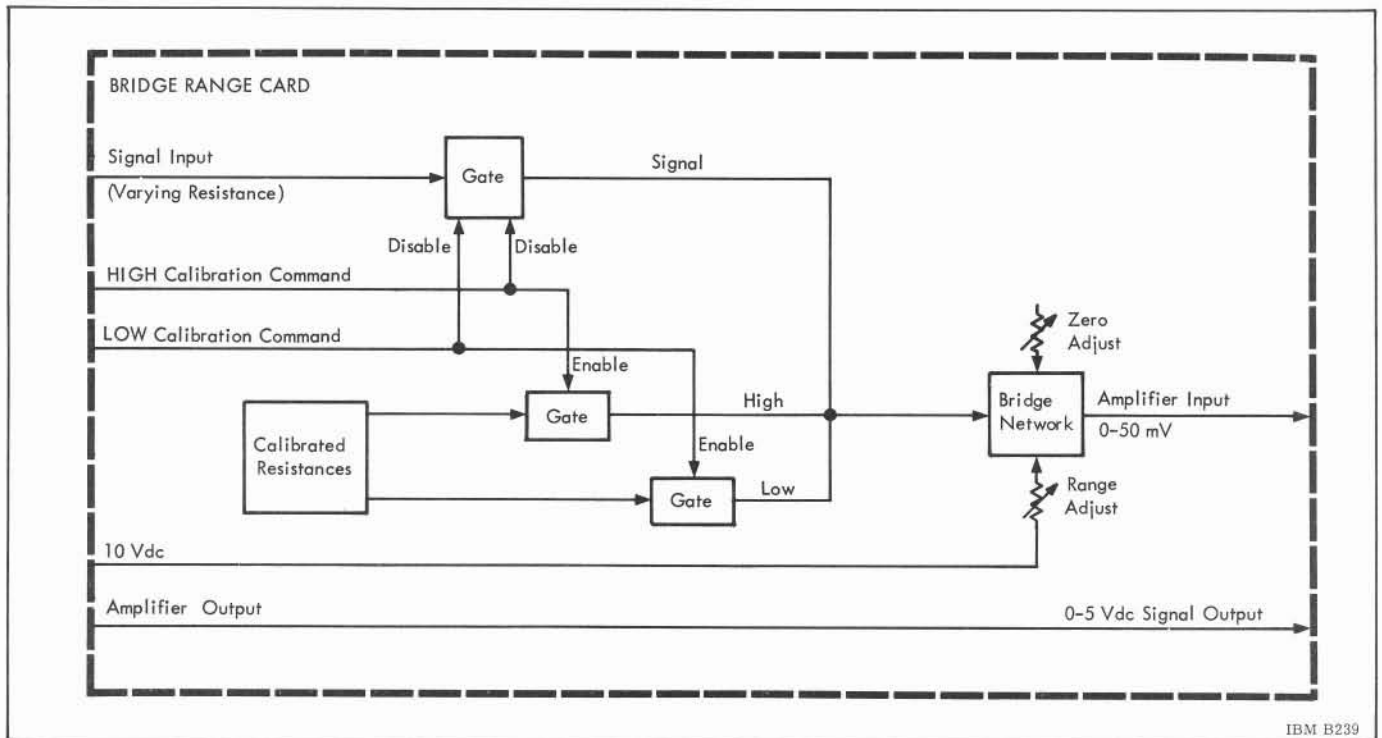
Figure 5.2-16 Typical Range Card for Temperature Measurements

changes to be made just prior to launch. More than one measuring distributor may be used depending on the number of measurements to be made and the physical location within the IU of the measurements to be made.

Switching functions connect different sets of measurements to the same telemetry channels during different flight periods. These switching functions, controlled by the Control Distributor, are performed in the Measuring Distributor. Switching functions, controlled from the ground via the umbilical cable, connect measurements not required during flight to digital data acquisition system channels for ground checkout and return the channels to flight measurements after checkout.

5.2.5 MEASUREMENT CALIBRATION

The Remote Automatic Calibration System (RACS) permits a remote calibration of the measuring system prior to launch. During vehicle checkout, calibration of measurements is accomplished through the RACS and various corrections can be made by adjusting potentiometers on the modules. Prior to launch time, the RACS may be operated if the system drifts or deviates from the final adjustments. The data obtained is used to correct the flight data for more accurate measurements.

Each signal conditioning module contains 2 relays and the necessary circuit to simulate the trans-

ducer as well as the upper (HI) end and the lower (LO) end of the signal range for that particular measurement. The transducer is connected to the module in the RUN mode.

A binary-coded signal is sent from the GSE through the umbilical cable to the Measuring Rack Selector in the vehicle to select a particular measurement for calibration. Figure 5.2-17 shows the RACS for the IU. Each stage has a separate RACS. The signals are generated in the GSE either by a manual keyboard or from a computer program. These signals are decoded on the vehicle and distributed to the various Measuring Racks to operate the checkout relays.

The calibration command consists of 13 bits in parallel. There are 6 bits for rack selection, 5 bits for channel selection, and 2 bits for mode selection. (One checkout relay is for the HI mode, or checkpoint, and the other is for the LO mode, or checkpoint.) The same code is used on all stages to provide a 31-rack capacity. The IU RACS is designed for 14 racks.

On the vehicle, the code is received in the Measuring Rack Selector. The rack select code is decoded and the signal is addressed to the proper Measuring Rack. Buffer circuits for mode codes and channel codes are provided in the Measuring Rack Selector which acts as a distribution point to the various Measuring Racks. The signals go to the channel selector module in the selected Measuring

Rack. Operation of the channel selector requires the following conditions:

> Rack select signal present
> Channel select signal present
> Mode signal present (HI or LO)

There are two channel selectors in each Measuring Rack. One decodes channels 1 through 10 and the other decodes channels 11 through 20. The channel selector contains the necessary diode matrices and latch circuits to complete the operation of the RACS.

The channel select modules contain 20 AND gates. When the 3 conditions mentioned above are fulfilled, the addressed AND gate "latches in" and energizes the corresponding calibration relay in the signal conditioning module. This relay will remain energized until another signal is sent to change the mode.

In addition to HI and LO mode, there is a third mode of operation, called RUN mode, which is selected when both calibration relays are in an unenergized state. The RUN mode is the normal mode of operation with the transducer or input signal connected to the signal conditioning module. During calibration (following HI or LO mode), the RUN mode is achieved by sending the proper code to the vehicle. Since the RACS is used only before launch, it is wired in such a way that the power is removed at launch. Removal of the power causes all relays to switch to RUN mode. This eliminates the possibility of leaving any module in the checkout state (HI or LO).

Any number of channels can be selected individually, simultaneously, or in any sequence or combination (with any combination of a HI, LO, or RUN selection at the GSE keyboard). The GSE display panels are used to monitor the signal (code) sent to the vehicle (there is no code feedback from the vehicle). The calibration results are observed through the telemetry system and connected ground checkout system.
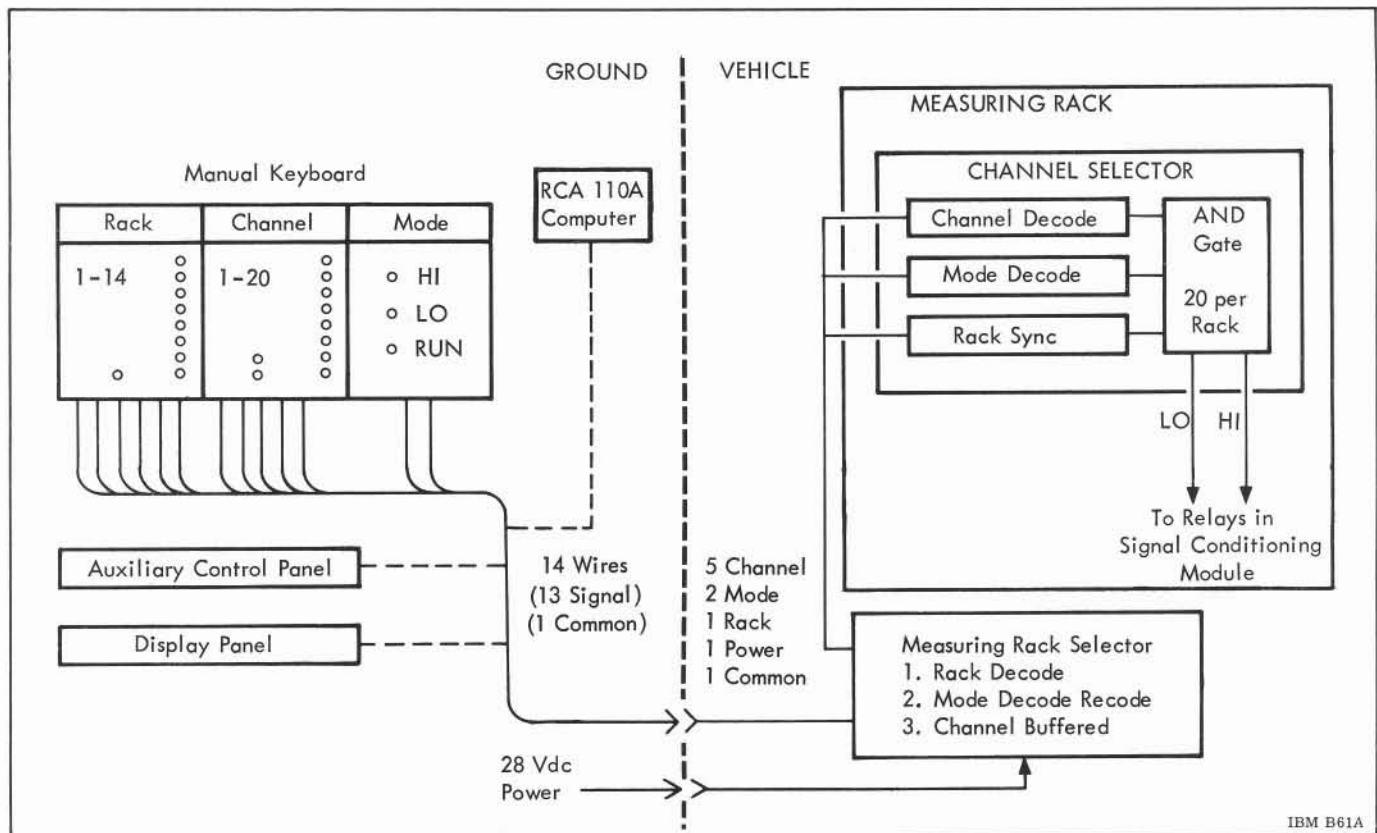

Figure 5.2-17 Block Diagram of RACS for the IU

SECTION 5.3
TELEMETRY

5.3.1 GENERAL

Each stage of the Saturn Vehicle carries an independent telemetry system. These systems modulate the signals from the measuring system onto RF carriers for transmission to ground stations. Three different modulation techniques are applied in the telemetry system of each stage:

- FM/FM telemetry applies frequency modulation/frequency modulation with Pulse Amplitude Modulation (PAM) and triple Frequency Modulation (FM^3) as auxiliary techniques

- SS/FM telemetry uses single sideband modulation/frequency modulation

- PCM/FM telemetry uses pulse code modulation

These different modulation techniques provide efficient transmission of the large number and variety of measuring data which have different requirements of bandwidth and accuracy (Table 5.3-1).

DATA CATEGORIES

Low-response/low and medium-accuracy data may be sufficiently defined by sampling at a rate of 10 hertz, or less. In terms of frequency response this would be 2 or 3-hertz maximum variation. This category includes temperature, pressure, and other measurements where rapid variations are not very likely or are not of interest.

Low-response/high-accuracy data requires accuracies of 1 percent or better. Examples are longitudinal acceleration and combustion chamber pressure measurements.

Medium-response/medium-accuracy data requires a 5 to 40-hertz response. Data of this category can be handled by sampling at 100 to 125 hertz per channel or less, or by FM/FM channels below the 3-kilohertz subcarrier using the standard deviation ratio of 5.

High-response/medium-accuracy data requires a 50 to 1000-hertz channel response. This type of data is readily handled by FM/FM subcarriers.

Wide-band data has a bandwidth of roughly 50 to 3000 hertz and normally no dc component of interest. Vibration data and sound intensity fall within this classification.

Event measurements constitute a special category. Interest in this data is limited to whether or not

Table 5.3-1 Data Categories

Type	Frequency Response (hertz)	Accuracy Required (percent)
1. Low-response/low and medium-accuracy	2-3	2-5
2. Low-response/high-accuracy	2-3	1
3. Medium-response/medium-accuracy	5-40	2
4. High-response/medium-accuracy	50-1000	2
5. Wideband low-accuracy	50-3000	5
6. Event measurements	---	---
7. Digital measurements	---	---

an event has occured and, if so, the time of occur-
rence with respect to a specified time resolution.

In addition, much data originates in digital
form, e.g., signals from the LVDC.

MODULATION METHODS

The wide-band data-carrying capability of
standard FM/FM telemetry is very poor. The FM/FM
system has one channel with a frequency response of
0 to 1050 hertz at a subcarrier deviation ratio of 5;
the other channels have decreasing frequency responses
down to 6 hertz. The sum of all the channel band-
widths is only 4700 hertz. The bandwidth efficiency
of a standard FM/FM system with subcarriers operat-
ing at a deviation ratio of 5 is about 1.6 percent. A
corresponding figure for a PAM/FM system is about
3.5 percent; for a PCM/FM system, it is about 3.0
percent.

The necessity to transmit wide-band data
(acoustic and vibration measurements) led to the use
of SS/FM telemetry. An RF carrier is frequency
modulated with single-sideband, amplitude-modulated
carriers. The bandwidth efficiency of such a system
is roughly 10 times greater than that of FM/FM. It
is capable of transmitting 45 kilohertz of data over the
same bandwidth used by the FM/FM link to transmit
4700 hertz of data.

A theoretical comparison of the signal/noise
performance of the SS/FM with a standard FM/FM
system is useful. Since single-sideband subcarriers
do not possess wide-band gain, one would logically
expect that SS/FM would perform less favorably in
this respect than FM/FM, probably by a factor equal
to $\sqrt{3}$ times the deviation ratio, which is the wide-
band gain of the FM subcarrier. However, if the
modulating signal of such a system is to be vibration
data, other factors can more than compensate for
the lack of wide-band gain in the subcarrier.

These compensating factors result from a
prior knowledge of the nature and characteristics of
vibration data. A Gaussian or normal curve is a
good approximation of amplitude distribution char-
acteristics of such data. Since the summation of
Gaussian functions is a Gaussian function itself, the
composite signal modulating the transmitter (in an
SS/FM system carrying vibration data) could be ex-
pected to resemble a Gaussian function. The peak-
to-peak amplitude of the data applied to an FM sub-
carrier must be limited to the band edges of the
subcarrier channels, the deviation being ± 7.5 percent

of center frequency, to prevent adjacent channel
interference. Thus, when data possesses a high
peak-to-rms ratio, the signal capacity of the channel
is reduced below its signal capacity for a sine wave
modulating signal (1.41 peak-to-rms ratio). Corre-
spondingly, Gaussian data with a peak-to-rms ratio
of 4.0 reduces rms data capability to 2.83.

No such inherent peak data restriction exists
in the SS/FM system. The data peaks of the individual
channels add in a random manner, resulting in an
amplitude distribution of the composite signal similar
to that of the data at the channel inputs and having a
peak factor of approximately the same magnitude.
Thus, with Gaussian-type characteristics and an
identical number of channels, the SS/FM system will
accommodate two or more times as much peak carrier
deviation per channel as an FM/FM system.

Data which originates in digital form and data
requiring high accuracy are transmitted through the
PCM/FM system. The digital data transmission link
is relatively insensitive to imperfections in the trans-
mission channel. Below a threshold noise level, the
accuracy is not appreciably affected by noise in the
channel. Nonlinearity in the transmission channel is
of little consequence. By a relatively simple and
reliable operation, the serial digital data which con-
tains noise and deteriorated rise time can be regener-
ated into its original noise-free form.

Also, digital data transmission is compatible
with real-time data processing by digital computers
at ground stations. A separate digital data output of
the PCM/FM system is used for automatic preflight
checkout of the launch vehicle. This is called the
digital data acquisition system. The Mod 301 PCM/
DDAS Assembly is also connected with the LVDC for
checkout of the S-IVB/IU Stage in orbit. During flight,
all operational data is transmitted over the PCM/FM
link.

MULTIPLEXING METHODS

To increase the data handling capacity of te-
lemetry systems, several data channels are multi-
plexed. Two types of multiplexing are utilized: fre-
quency-division multiplexing and time-division
multiplexing.

Frequency-division multiplexing is the process
of simultaneous frequency sharing of one RF carrier
transmission link by dividing the available bandwidth
into a number of channels, each with a separate center

frequency. Spacing is provided between these channels to allow each channel to be frequency modulated or deviated about the center frequency. After modulation occurs, the subcarrier signals are combined linearly, and the resultant composite signal is used to frequency modulate an RF carrier. FM/FM and SS/FM are examples of this frequency division multiplexing.

Time-division multiplexing is the process of sequential sampling of two or more data sources and applying the data samples to a common output in a fixed sequence. The samples may or may not be separated by a "dead time" space. Usually some marker is made a part of the output signal to designate the beginning or end of a sampling cycle. This is for identification and synchronization purposes at the data reduction point.

The Saturn telemetry systems make use of several types of time-division multiplexers. The Mod 270 Multiplexer Assembly is a time-division multiplexer for analog data. It is used in conjunction with the FM/FM telemetry system and the PCM/FM system. The Mod 245 Multiplexer Assembly provides time-division multiplexing for wide-band data and is used for SS/FM telemetry and FM/FM telemetry. The remote digital submultiplexers and digital multiplexers are used in connection with the Mod 301 PCM/DDAS Assembly. Remote analog submultiplexers are used for analog data connected to the Mod 270 Multiplexer Assembly.

TRANSMISSION CARRIERS

The number of RF carriers allotted to each modulation technique is chosen to provide an appropriate balance of data transmission capability to handle the quantities and varieties of data originating on the stage. The telemetry equipment associated with each stage consists of a "building-block" arrangement, which may be connected in numerous combinations to satisfy specific measuring requirements. The number of carriers allocated to FM/FM, SS/FM, or PCM/FM and also the specific combination of building blocks vary from stage to stage and from vehicle to vehicle.

TYPICAL S-IVB/IU TELEMETRY SYSTEM

Figure 5.3-1 is a functional flow diagram of a typical instrumentation system for Saturn V. Flight (mission) control data is sent in parallel to the IU PCM/FM and S-IVB PCM/FM systems as indicated in the figure. Therefore, in the event of a failure in one of the PCM systems (IU or S-IVB), the other PCM system can be used to transmit the flight control data.

In operational vehicles, the number of multiplexers and FM/FM systems will be reduced, and the SS/FM telemetry will be omitted.

Table 5.3-2 lists the type and number of telemetry systems and multiplexers used in the stages of Saturn Vehicles. Because of the smaller quantity of measurements required in operational vehicles, the number of telemetry systems is also reduced. (Figures given for operational vehicles are "best guess" at the present time.)

From one to six time-division multiplexers are synchronized from a central timing source located in the PCM/DDAS Assembly. Each time-division multiplexer provides an output to the PCM/DDAS Assembly which combines the outputs into a single serial wavetrain. The individual analog samples are digitized and combined into a digital format which is transmitted via coaxial cable to the ground checkout equipment. This data is also transmitted via a PCM/FM carrier for in-flight monitoring.

Each of the time-division multiplexers has a second data output which is identical to the output provided to the PCM/DDAS Assembly except that it is conditioned for PAM transmission. These outputs may modulate a 70-kilohertz, voltage-controlled oscillator in FM/FM telemeter assemblies. This arrangement provides redundant transmission of some multiplexer outputs using both PAM and PCM techniques.

In the following paragraphs, the telemetry systems are described in the order of data flow. Starting from the telemetry equipment closest to the data source, the descriptions continue toward the transmitting equipment. Figure 5.3-1 should be used as a reference to these system descriptions.

5.3.2 SLOW SPEED MULTIPLEXING

MOD 245 MULTIPLEXER

The Mod 245 Multiplexer Assembly is a time-division multiplexer which is used in conjunction with an SS/FM Assembly multiplexer and FM/FM multiplexer link. This unit can sample a maximum of 80 data channels and provide 16 channels of output data. It is primarily used for multiplexing wideband vibration data.

The Mod 245 Multiplexer Assembly, Figure 5.3-2, consists of 20 plug-in subassemblies. Of these 20 subassemblies, 4 are common subassemblies
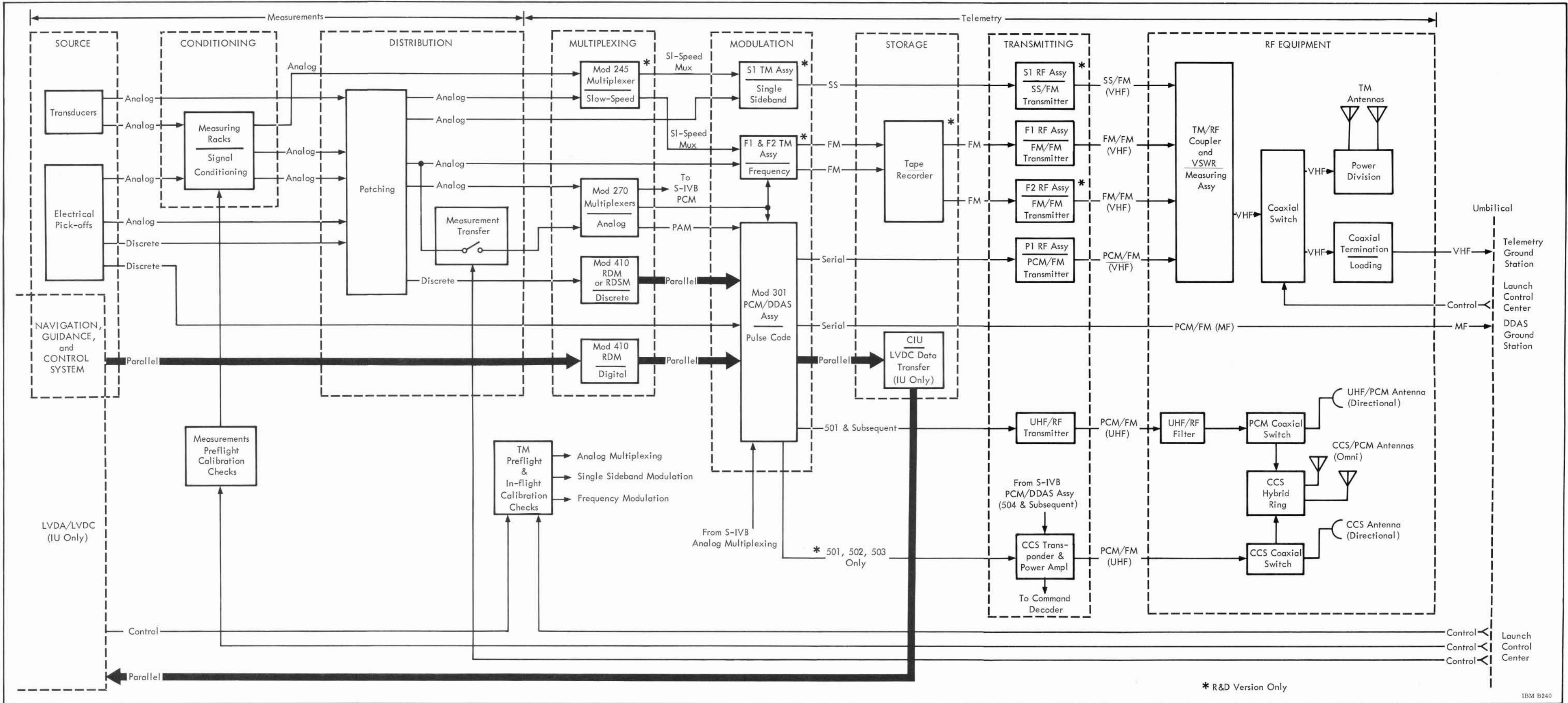
Figure 5.3-1 Saturn V Instrumentation System Functional Data Flow

Changed 15 August 1966

Table 5.3-2 Telemetry Systems in the Various Saturn Vehicle Stages

Stage	Telemetry System	Number of Systems		Number of Multiplexers		Transmitter	
		R&D	Oper	R & D	Oper	Frequency	Power
IU	FM/FM SS/FM PCM/FM	2 1 1	1 - 1	1 CIU 2 Mod 270 1 Mod 245 1 RDM Mod 410	1 CIU 2 Mod 270 2 RDM Mod 410	Saturn IB: 225-260 MHz Saturn V: 225-260 MHz and 2200-2300 MHz	20 W
S-IVB	FM/FM SS/FM PCM/FM	3 1 1	1 - 1	4 Mod 270 1 Mod 245	2 Mod 270	225-260 MHz	20 W
S-II	FM/FM SS/FM PCM/FM	3 2 1	2 - 1	5 Mod 270 1 Mod 245 7 RASM 2 RDSM	2 Mod 270 2 RDSM 2 RASM	225-260 MHz	20 W
S-IC	FM/FM SS/FM PCM/FM	3 2 1	1 - 1	4 Mod 270 2 Mod 245 1 RDSM	1 RDSM 2 Mod 270	225-260 MHz	20 W
S-IB	FM/FM SS/FM PCM/FM	2 1 1	1 - 1	3 Mod 270 1 Mod 245 2 RDSM	2 Mod 270 1 RDSM	225-260 MHz	20 W

Note: CIU = Computer Interface Unit
RDSM = Remote Digital Submultiplexer
RASM = Remote Analog Submultiplexer
RDM = Remote Digital Multiplexer (Mod 410)

(1 logic and timing unit, 1 regulator unit, 1 inverter unit, and 1 driver and voltage-controlled oscillator unit). The remaining 16 subassemblies are multiplexer units.

The data units are made up of 4 types. They are as follows:

● Two multiplexed channels with a sample period of 5.7 seconds each.

● Four multiplexed channels with a sample period of 2.9 seconds each.

● Five multiplexed channels with a sample period of 2.3 seconds each.

● One continuous channel.

The 3 types of multiplexer units all use 12-second sampling cycles. The different sample times are separated by "dead time". The sampling cycles are separated by longer periods of "dead time". Frequency response of the input is 0 to 5000 hertz. Input impedance is a constant 100 kilohms. Input signal level can be 0 to +5 volts peak to peak, or 0 to +5 volts dc. Output impedance is 1 kilohm at 0 to 5000 hertz. A dummy unit is used for continuous data.

Additional timing information is also transmitted using a 960 ± 70-hertz voltage controlled oscillator for data reduction purposes. This is fed into the special service channel in the SS/FM Assembly.

5.3.3 SS/FM TELEMETRY

The Saturn SS/FM telemetry system is designed specifically for transmission of the large volume of vibration data from the Saturn Vehicle. This system can transmit 15 channels, each having a response
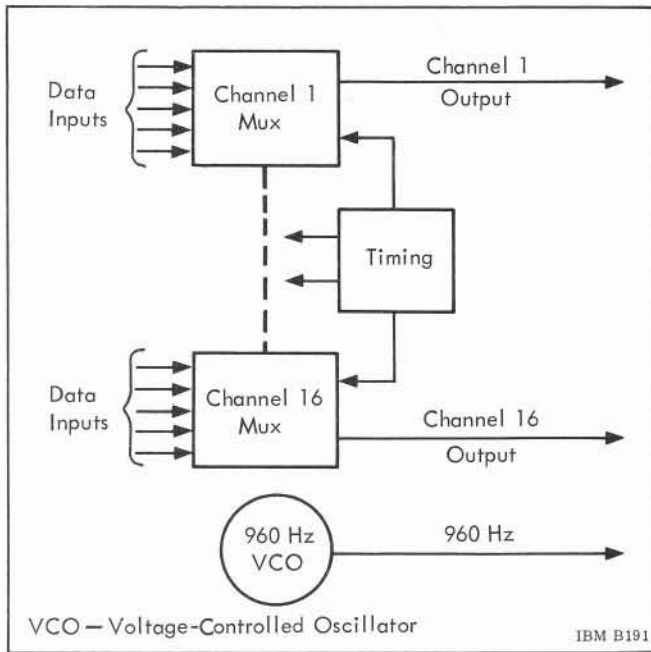
Figure 5.3-2 Mod 245 Multiplexer Block Diagram

of 30 to 3000 hertz, for a total data bandwidth of approximately 45 kilohertz within the standard telemetry carrier bandwidth.

A basic SS/FM telemetry system is comprised of a slow speed multiplexer, an SS Telemetry Assembly, and an RF assembly similar to the FM/FM RF Assembly. The SS/FM Assembly is used in conjunction with a slow speed multiplexer (Mod 245) to expand its data-handling capability by time-sharing specific data channels. SS/FM telemetry is not used in operational vehicles.

SS TELEMETRY ASSEMBLY

Each of the 15 data inputs is fed to a balanced modulator and heterodyned with a 455-kilohertz carrier (Figure 5.3-3). The output of the modulator is fed to a mechanical bandpass filter (455 to 458 kilohertz) which passes only the upper sideband. The output of the filter is fed to a second balanced modulator where it is translated to the proper baseband frequency. The baseband position is determined by the carrier supplied from the frequency synthesizer. The two balanced modulators and the mechanical bandpass filter for each data channel make up the channel units. The channel units are identical for all channels. bandpass filter for each data channel make up the channel units. The channel units are identical for all channels.

The second modulator carriers are supplied to the channel unit from the frequency synthesizer

and comb filters. These 15 subcarriers transpose the data to an assigned frequency between 4.74 and 72 kilohertz. These are the lower sideband outputs of the second modulators.

Both sidebands of the second modulators go to the summing amplifier No. 1 and the 200-kilohertz low-pass filter where all upper sidebands of the second modulator are filtered out. Here a 75.835-kilohertz pilot tone is summed into the channel unit outputs. The pilot tone provides a signal at the receiving station for demodulation of the 15 channels.

After leaving the 200-kilohertz low-pass filter, the composite signal is passed through a signal regulator. From the signal regulator, the composite signal is summed with the signal from the non-translated special service channel. The frequency response of this channel is 0 to 1200 herts. Its normal use is to carry an 890 to 1030-hertz synchronizing signal provided by the Mod 245 Multiplexer Assembly for demultiplexing the signals at the receiver. The output of the summing amplifier No. 2 is used to frequency modulate an RF transmitter.

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All the carrier signals and the pilot tone are generated internally in the airborne SS/FM assembly by a single 910.025-kilohertz crystal oscillator. This signal is used to drive a flip flop which provides the 455.012-kilohertz carrier signal and a frequency synthesizer provides all sub-carriers and the pilot tone from the comb filters.

To provide a 3-kilohertz information bandwidth and allow sufficient guardband, a channel spacing of 4.74 kilohertz is used. This spacing is convenient to generate in the synthesizer and allows an adequate guardband of 1.74 kilohertz. The 75.83-kilohertz pilot tone falls just above the highest baseband frequency. It is used as a reference in the ground demodulation equipment to regenerate the basic 455 and 4.74-kilohertz frequencies. Since the amplitude of the transmitted 75.83-kilohertz pilot tone is regulated, it is also used as an automatic gain control.
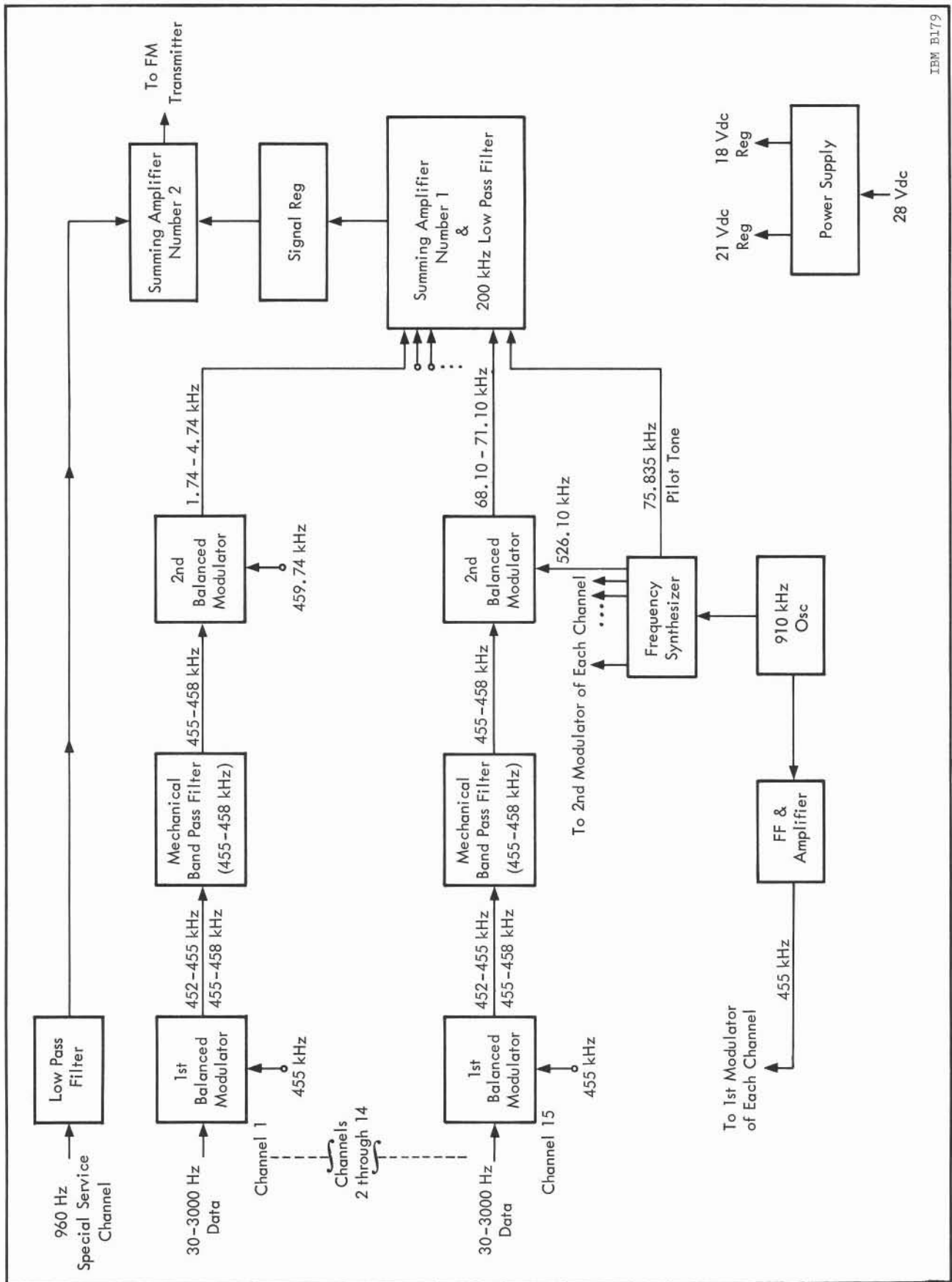
IBM B179

Figure 5.3-3 Airborne SS Telemetry Assembly Block Diagram

5.3.4 FM/FM TELEMETRY

The FM/FM telemetry system frequency multiplexes the input data. The conditioned signals derived from pressure, temperature, etc., modulate the frequency of the subcarrier oscillators. The combined output of several subcarrier oscillators is again frequency modulated on the RF carrier. Standard IRIG frequency channels are assigned. The data bandwidth capabilities of these channels are shown in Table 5.3-3. Slow-varying data is applied to the low-frequency channels; data with higher frequency variations must be applied to the higher frequency channels. FM/FM systems are susceptible to various sources of noise and distortion (fluctuation noise, crosstalk, harmonic distortion of the individual subcarrier signals, subcarrier data feed-through, and distortion of individual subcarrier signals due to band-limiting filters), which affect the accuracy of the telemetered data. The basic components of an FM/FM telemetry system are: a Mod 270 Multiplexer Assembly, an FM/FM Assembly and an RF Assembly.

Each of the subcarrier oscillators of an FM/FM system may be preceded by another subcarrier group. Each of these "sub-subcarriers" would, in this case, be modulated by the actual measurement values. This technique is called FM/FM/FM, or FM^3. The technique trades bandwidth for additional channels; therefore, the frequency response of the channels is much lower. Since another modulation stage is used, the technique is less accurate than FM/FM.

A time-multiplexed signal may be fed into a subcarrier oscillator of an FM/FM system to increase the channel capacity. Time multiplexing consists of sequentially sampling a number of data signals with an electronic or mechanical commutator. The commutator output appears as a train of pulses of varying amplitude. This technique is known as Pulse-Amplitude Modulation (PAM/FM/FM).

The basic modulation scheme and principal components used (subcarrier oscillators, mixer, power amplifier, and transmitter) are essentially the same for the FM/FM system in each stage. Figure 5.3-4 shows a typical Saturn Stage FM/FM system. Each channel receives a signal from the Measuring Distributor. This input signal modulates a voltage-controlled subcarrier oscillator. The frequency-modulated signals from the various subcarrier oscillators are combined in the mixer amplifier. The composite signal is frequency modulated on a radio-frequency carrier in the VHF band (225 to 260 mega-hertz) for transmission to the ground stations. The transmitter provides a signal power level of approximately 20 watts.

The available standard IRIG channels may be utilized in different ways. The system shown in the figure uses 13 continuous data channels (channels 2 through 14). The input signals from the Measuring Distributor are in the range of 0 to +5 volts dc (2 to 7 volts peak-to-peak ac when a bandpass filter is used in place of a channel or subchannel oscillator).

When used, FM^3 is applied to the channels above channel 13. A subcarrier channel of 70 kilohertz (±30 percent) is used for time-multiplexed signals from the Mod 270 Multiplexer Assembly. When this PAM technique is applied, all IRIG channels above 30 kilohertz will not be used. The Mod 270 solid-state, time-division multiplexer operates at a rate of 3600 samples per second. It consists of 30 primary channels which are sampled 120 times per second. In addition, 10 submultiplexers may be used on selected main channels; each sampled at a rate of 12 per second. The multiplexer accepts input signals in the range of 0 to 5 volts dc and provides two PAM wave train outputs. One wave train modulates the 70-kilohertz, voltage-controlled subcarrier oscillator with a frequency deviation of ±30 percent (i.e., 49 to 91 kilohertz for an input signal from 0 to 5 volts dc). The second wave train is used for the PCM system.

RF ASSEMBLY

The RF Assembly accepts single sideband FM and FM analog signals from the telemetry assemblies and provides the highly-stable, close-tolerance carrier frequency (±0.01 percent). It further provides the capability of varying the carrier frequency in a manner (±1 percent of the best straight line approximation for 125-kilohertz deviation at modulation frequencies from 300 hertz to 100 kilohertz) proportional to the input signal amplitude, thereby providing an RF link from the measurement systems in the flight vehicle to the ground receiving station.

5.3.5 PAM MULTIPLEXING

MOD 270 MULTIPLEXER

The Mod 270 Multiplexer Assembly (time division) is basically a 30 by 120 multiplexer (30 channels, each sampled 120 times per second). It accepts voltage inputs in the range of 0 to 5 volts (high level data) and provides two PAM wavetrain outputs, one (with pedestal) for input to a 70-kilohertz ±30 percent

Table 5.3-3 IRIC Subcarrier Channel (±7.5% Channels)

| Channel	Center Frequencies (kHz)	Lower Limit (kHz)	Upper Limit (kHz)	Nominal Frequency Response (Hz)	Nominal Rise Time (ms)	Maximum Frequency Response (Hz)	Minimum Rise Time (ms)
2	0.56	0.518	0.602	8	42	42	8.33
3	0.73	0.675	0.785	11	32	55	6.40
4	0.96	0.888	1.032	14	24	72	4.86
5	1.30	1.202	1.398	20	18	98	3.60
6	1.70	1.572	1.828	25	14	128	2.74
7	2.30	2.127	2.473	35	10	173	2.03
8	3.00	2.775	3.225	45	7.8	225	1.56
9	3.90	3.607	4.193	59	6.0	293	1.20
10	5.40	4.995	5.805	81	4.3	405	0.864
11	7.35	6.799	7.901	110	3.2	551	0.635
12	10.50	9.712	11.288	160	2.2	788	0.444
13	14.50	13.412	15.588	220	1.6	1,088	0.322
14	22.00	20.350	23.650	330	1.1	1,650	0.212
15	30.00	27.750	32.250	450	0.78	2,250	0.156
16	40.00	37.000	43.000	600	0.58	3,000	0.117
17	52.50	48.562	56.438	790	0.44	3,938	0.089
18	70.00	64.750	75.250	1,050	0.33	5,250	0.067
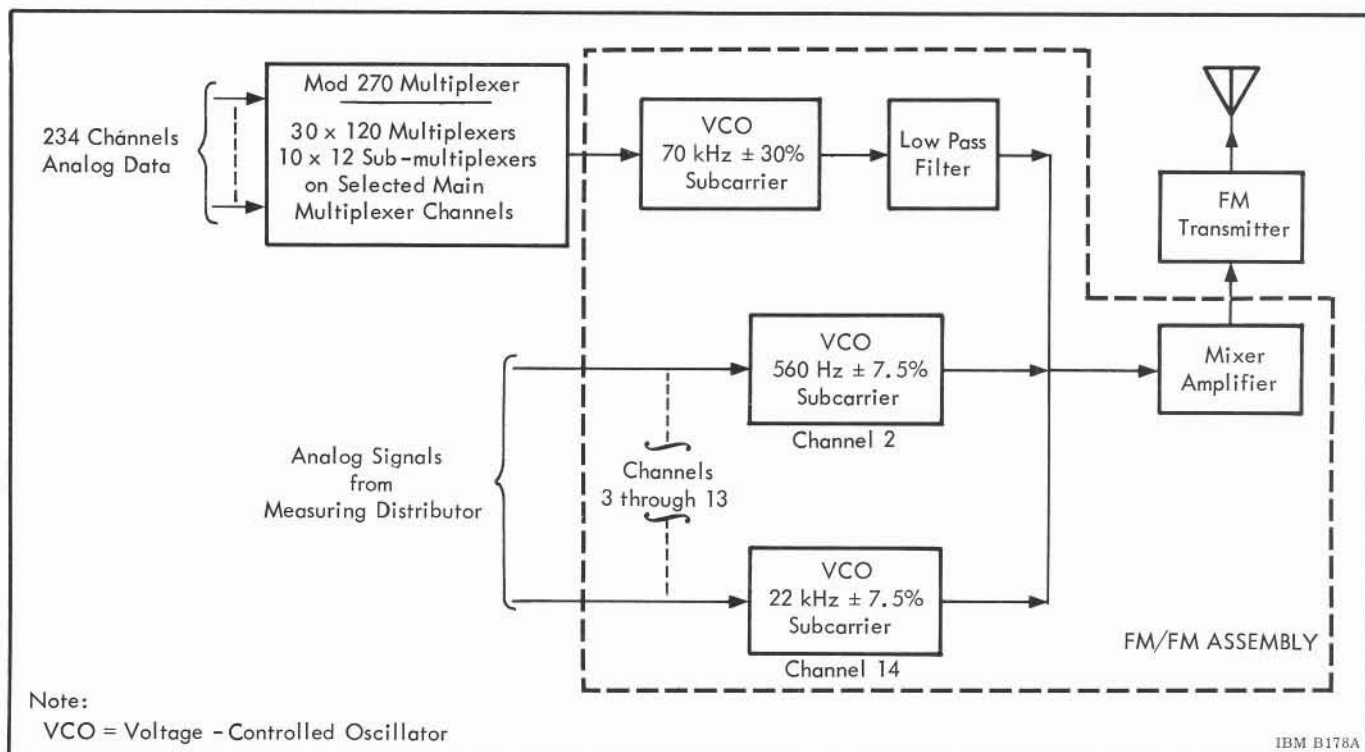
Figure 5.3-4 Typical R&D FM/FM Telemetry System Block Diagram

FM/FM subcarrier and a second (without pedestal) for the PCM/DDAS Assembly. A block diagram of the Mod 270 Multiplexer Assembly is shown in Figure 5.3-5.

A time-division multiplexer consists of three major parts: a set of switches or gates for each output channel, a timing device (logic generator), and a power supply.

The gates, acting as electronic switches, sequentially connect the data inputs to the multiplexer output on commands from the timing circuits.

The timing, or logic, circuits provide the signals which open and close the data gates at the desired times and in the desired sequence.

Multiplexers are normally located in proximity to data sources in order to minimize cabling weight. Each multiplexer houses sampling gates for up to 234 data channels. By means of flexible plug-in modules, variations in numbers of channels and sample rates may be accomplished within the multiplexers. These multiplexers also accept pre-sampling filters where required for specific measurements. Remotely located submultiplexer assemblies may also be used to increase the capacity of each individual multiplexer to 270 channels.

Twenty-seven of the thirty primary channels are data channels while the remaining three are utilized for amplitude references and PAM frame identification. Ten-channel gate modules, which plug into the assembly, may be used to submultiplex primary channels 1 through 23. Each of these 10 sub-channels then provides a sampling rate of 12 times per second. Any number of the 23 channels may be submultiplexed or utilized as 120-sample-per-second channels. When used at the primary sample rate, a submultiplexer "dummy" card (or a pre-sampling filter if required) is inserted in place of the submultiplexer module. The four remaining primary data channels (24, 25, 26, 27) are always utilized as 120-sample-per-second channels. Each of the 27 primary data channels may be submultiplexed external to the basic assembly.

A zero amplitude reference is inserted on channel 28 of the output wavetrains except in the frame corresponding to sub-channel 10, where a 5-volt amplitude reference is inserted to provide master frame identification. Channels 29 and 30 carry a 5-volt reference level and are bridged together to form PAM frame identification. Except for the channel 29-30 bridge, all channels are inserted as 50 percent duty cycle pulses on both output wavetrains.

A zero pedestal of 1.2 volts is added to the wavetrain output applied to the 70-kilohertz VCO of the PAM/FM/FM telemetry system. The output wavetrains have a pulse rate of 3600 pps.

A calibration generator, located in the assembly, provides in-flight calibration capability. Calibration is initiated by an external command which in the Saturn telemetry system is provided by the TM Calibrator (Mod II). When a calibrate command is received, the calibration generator delays its sequence until the beginning of the following master frame. It then applies a sequence of five calibration voltages to all data channels (channels 28, 29, and 30 are unaffected by the calibration). The calibration voltage levels are in order: 0, 25, 50, 75, and 100 percent of full-scale data voltage. Each voltage level is sustained for one master frame (83.4 milliseconds), requiring approximately 400 milliseconds for the complete sequence. Paragraph 5.3.7 describes the calibration function in greater detail.

Internal logic functions of the Mod 270 Multiplexer Assembly operate from its internal 3600-hertz multivibrator clock. However, the sample and frame rates can be synchronized to external pulses of the correct frequency and waveform. In the Saturn telemetry system, these synchronizing signals are provided by the Mod 301 PCM/DDAS Assembly. The synchronizing waveforms consist of a 3600-hertz square wave and a 4 pps pulse approximately 278 microseconds in duration occurring during channel one of each 30th frame.

The Mod 270 Multiplexer Assembly also provides synchronizing signals as outputs. These are identical to the waveforms previously described, except the frame sync pulse has a repetition rate of 12 pps instead of 4 pps. See Figure 5.3-6. These synchronization outputs are normally utilized to control remote submultiplexer packages.

The submultiplexers are composed of selection gates that subdivide the main multiplexer channels. Ten submultiplexer channels are connected to each main multiplexer channel, thus reducing the sampling rate from 120 per second to 12 per second.

Each gate functions in the following manner: The input side of a gate is tied to the data to be sampled, and the output side is tied to the main multiplexer channel. The normally open gate closes upon application of a current pulse to the transformer to allow information to pass through to the main multiplexer. This current pulse is obtained by grounding
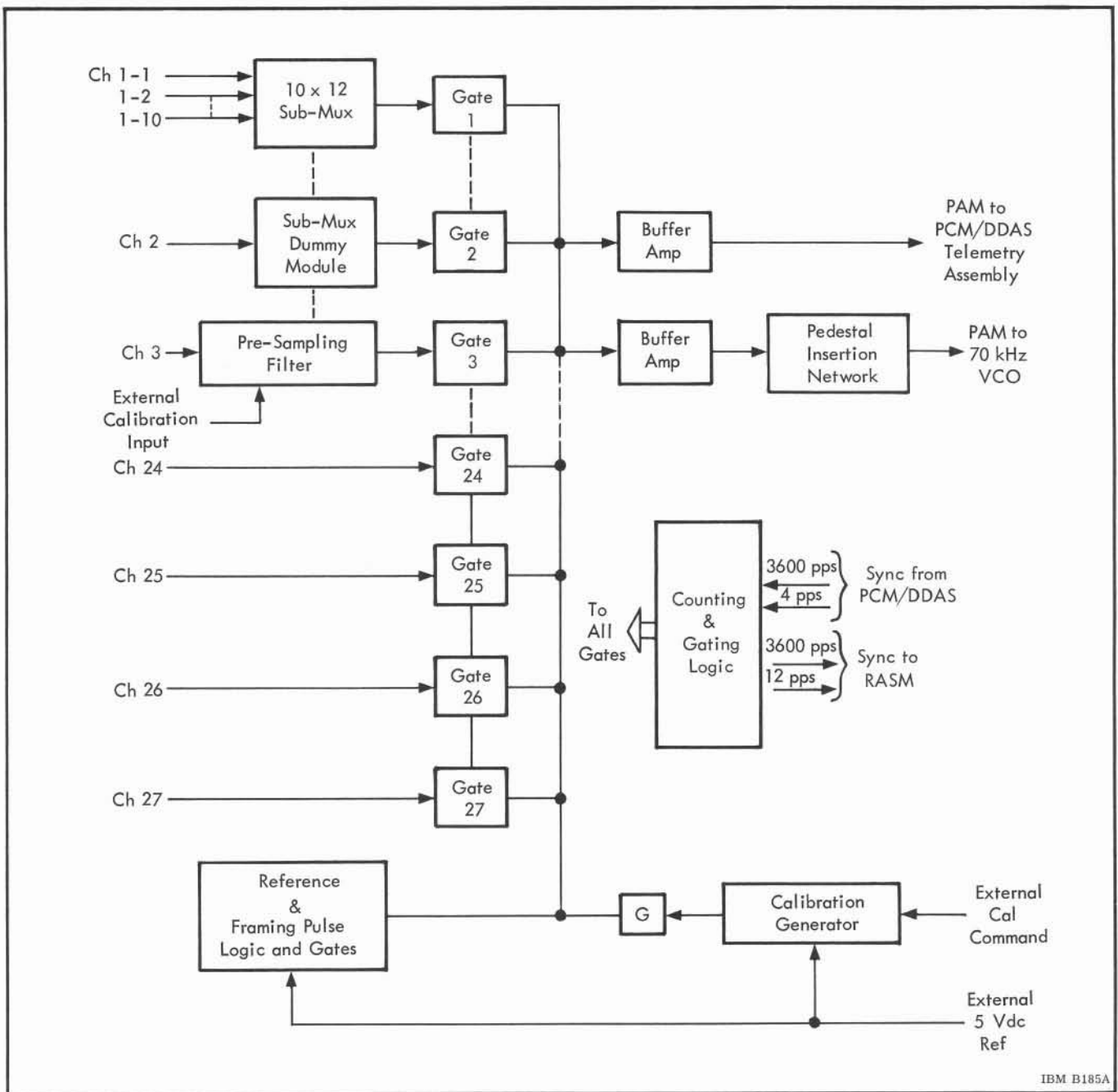
Figure 5.3-5 Mod 270 Multiplexer Block Diagram

one side of the transformer and connecting the other side to the decoded output of the "T" counter.

The gates are connected in columns of ten, and all outputs of a particular column are connected to one main multiplexer channel input.

The submultiplexer "dummy" card is used when a sampling rate of 120 times per second is required. This card, used in place of the submultiplexer card, provides a direct path from the data being sampled to the main multiplexer board. An RC load mounted on this board (the same as on all submultiplexer cards) prevents data spikes from being passed through.

The characteristics of the Mod 270 Multiplexer Assembly are given in Table 5.3-4.

5.3.6 PCM/DDAS TELEMETRY SYSTEM

The PCM/DDAS telemetry system serves a dual purpose in the Saturn Launch Vehicle. This

Table 5.3-4 Mod 270 Multiplexer Assembly
Performance Characteristics

Characteristic	Specification
Time Stability	
Clock rate	3600 Hz + 2%
Duty cycle	139 us ±5%
Linearity (best straight line)	± 0.1%
Accuracy	
Output of amplifier (without pedestal)	±0.1%
Output of amplifier (with pedestal)	±0.5%
Pedestal Stability	1.2 V ±50 mV
Reverse Current (all channels at 5 V except channel under test which is short circuited)	2 uA (max)
Input Impedance of Channel (excluding 100k input terminating resistor)	5 mego (min)
Output No. 1 (without pedestal)	100 Ω (max)
Output No. 2 (with pedestal)	2k (max)

system functions as a telemetry link for digital data (PCM/FM) and as a part of the DDAS. PCM/FM is required on the launch vehicle in order to obtain sufficient accuracy, with acceptable bandwidth efficiency, for digital data transmission from data sources such as the LVDC. This accuracy is possible because of two main reasons: first, the scheme requires no digital to analog conversion for digital and discrete inputs, and second, transmission of a serial bit configuration allows a good signal to noise ratio for a weak and deteriorated signal received at a distant ground station.

Before launch, PCM/DDAS also provides measurements to the launch computer in the ground checkout stations (via coaxial cable).

SYSTEM COMPONENTS

To provide flexibility, the system is "built up" from several assemblies which may be combined as required in a particular stage application. The following assemblies may be used to "build up" a PCM/DDAS telemetry system:

- Mod 301 PCM/DDAS Assembly

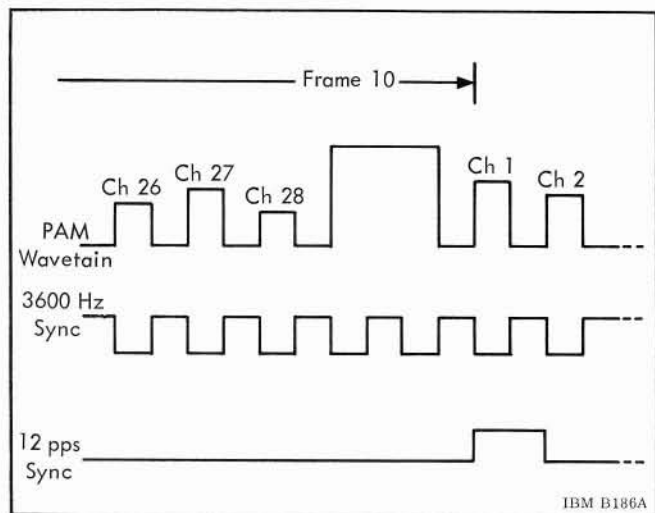- Mod RDSM-1D Remote Digital Submultiplexer

Figure 5.3-6 Mod 270 Multiplexer Assembly
Waveforms

- Mod 410 Remote Digital Multiplexer

- Mod 270 Multiplexer Assembly

- A remote analog submultiplexer which may be used with a Mod 270 Multiplexer Assembly

- Computer Interface Unit

- PCM RF Assembly

The Mod 301 PCM/DDAS Assembly is the central piece of equipment in both the telemetry link and the DDAS link. It provides the output signals for telemetry and automatic checkout.

The Mod RDSM-1D Remote Digital Submultiplexer is used to connect digital data sources to the Mod 301 PCM/DDAS Assembly.

The Mod 410 Remote Digital Multiplexer is used in the IU to connect the LVDC digital data, and discrete sources to the Mod 301 PCM/DDAS Assembly.

The Mod 270 Multiplexer Assembly is used to connect analog data into the Mod 301 PCM/DDAS Assembly.

The remote analog submultiplexer may be used to submultiplex data being fed through the Mod 270 Multiplexer Assembly (on the S-II Stage of the Saturn V Vehicle only).

The Computer Interface Unit (CIU) allows the LVDC, operating through the LVDA, to read selected measurement data which is being transmitted through the Mod 301 PCM/DDAS Assembly.

The PCM/RF Assembly contains the signal conditioner, power amplifier, and RF transmitter for the telemetry link.

SYSTEM USES

During preflight checkout, the telemetry system presents digital data over coaxial cables to one or more locations remote from the vehicle. These measurements are available to digital computers in real time through a special data-receiving facility interfaced with the computers. The data-receiving facility also provides outputs for display of selected channels in either digital or analog form for visually determining the status and readiness of vehicle subsystems and tape records the DDAS inputs for analysis at a later time.

During the launch, earth orbiting, and lunar-injection phases, there are times when information processed by the LVDC is desired at the ground station. Since the information to be telemetered is dependent on particular missions and has a random characteristic, provision will be made in the telemetry to accomodate these outputs. Specific PCM telemetry system channels are assigned to accommodate the LVDA 40-bit outputs. The assigned channels are sampled at a rate of 240 times per second.

The LVDA identifies valid data by the presence of a validity bit which has no significance to the telemetry but is transmitted as part of the data telemetered to the ground. The ground computer automatically determines the existence of valid data by recognizing the validity bit in a data word. The validity bit is present with the valid data for at least 4.5 milliseconds to ensure at least one transmission of the valid data.

During flight, the DDAS function is performed between the telemetry system, LVDA, and LVDC. Upon request, data in digital form is made available to the LVDC during flight and is used by the LVDC to perform vehicle checkout.

The telemetry system in the S-IVB/IU functions during launch, earth orbit, and lunar-injection phase of the mission. During these phases, periodic checks are required of the vehicle's performance or operating status. This is accomplished by inserting specific segments of the telemetered information into the computer.

During orbital checkout, which is initiated by a command signal to the LVDC via the IU command, the LVDC requires a real-time value of measurements, which are part of the total measurements being telemetered by the S-IVB/IU Stage telemetry system.

Selective transfer of measurement values from the S-IVB/IU Stage telemetry systems to the LVDA is accomplished through the DDAS/Computer Interface Assembly. This assembly contains the timing and comparison logic necessary to separate the selected channel from the IU PCM/DDAS format. A time division multiplexer, located in the S-IVB Stage, transfers S-IVB measurements needed for checkout and mission control to the IU PCM/DDAS Assembly.

The LVDA signifies the specific data channel to be transferred by means of a 12-bit channel address. Upon receipt of a "data-request" signal from the LVDA, the DDAS/Computer Interface Assembly initiates a transfer sequence which consists of:

- Awaiting the next appearance of the signified channel in the PCM/DDAS format.

- Writing the data sample into a 10-bit holding register within the assembly.

- Providing a "data-ready" signal to the LVDA indicating that the selected data is available.

So long as the "data-request" signal remains at the request level, subsequent samples of the selected channels are transferred into the holding register as they appear in the format of the PCM/DDAS Assembly. When the LVDA returns the "data-request" signal to the standby level, the last value transferred remains in the holding register until another transfer sequence is initiated.

When the LVDA receives the "data-ready" signal, it branches to a sub-routine which operates to transfer the data from the telemetry output register to the LVDA. Synchronization between the telemetry system and the LVDA is accomplished in the following manner: Each time the telemetry receives an address from the LVDA, followed by a valid "data-request" signal, it recognizes this input as the initiation of a new data-seeking cycle as well as a signal to read in the data. Upon this recognition by telemetry, it first resets its output data register and then begins seeking the data requested by the LVDA. The LVDA and LVDC ensures that a new address with a valid read bit is not generated until data from the telemetry output register has been received in response to the previous address.

MOD 301 PCM/DDAS ASSEMBLY

The Mod 301 PCM/DDAS Assembly performs six major functions as follows:

- Scans the PAM wavetrains of several (1 to 6) Mod 270 PAM Multiplexer Assembles in a programmed sequence and combines these wavetrains into a single PAM wavetrain.

- Encodes the PAM samples in this wavetrain into 10-bit digital form.

- Accepts data in digital form and programs it into selected time slots in the output serial format.

- Generates the required frame and master frame identification codes, combines these codes with the digital and encoded analog data, and arranges the desired serial format for output.

- Provides a 600-kilohertz FM modulated carrier as the DDAS output, and an NRZ modulating output for the PCM/RF Assembly.

- Provides the synchronization outputs necessary to synchronize the Mod 270 Multiplexer Assemblies and remote digital multiplexers.

- Provides a parallel digital output to the CIU.

Figure 5.3-7 is a functional block diagram of the Mod 301 PCM/DDAS Assembly. It is composed of the six functional subsystems listed below:

- PAM scanner (and associated program patch).
- Analog-to-digital converter.
- Digital multiplexing and formating logic.
- Clock programming and timing logic.
- DDAS voltage controlled oscillator.
- Power supplies.

PAM Scanner. The PAM scanner connects one or more Mod 270 Multiplexer Assembly outputs in a programmed sequence to the ADC input for digitizing. By means of the scanner program patch, any multiplexer arrangement compatible with the system application can be accomodated. Up to three multiplexer arrangements (modes) may be programmed at the scanner patch, and the automatic switching between modes is accomplished by applying external 28-volt dc commands.

The scanner gates are 4-transistor, back-to-back, balanced configurations which switch both legs of the PAM circuit of each PAM multiplexer. The transformer coupling of the switching logic input of the gates provides a favorable impedance condition for the PAM signals.

The PAM scanner and program patch are packaged on printed circuit cards which plug into the Mod 301 PCM/DDAS Assembly.

Analog-to-Digital Converter. The analog-to-digital converter encodes the PAM signals received through the PAM scanner by the successive approximation method. The digital output is fed in parallel form to a parallel storage register according to commands received from the programming and timing logic. Figure 5.3-8 shows a functional block diagram of the ADC. Its operation is non-synchronous (not synchronized with other parts of the system) and at a clock rate of approximately 250 kilohertz which is provided by a blocking oscillator.

Encoder command pulses at the system word rate are provided by the programming logic. When an encode command is received by the ADC, it begins an encode cycle at its own clock rate; this requires approximately 56 microseconds. At the end of the cycle, the ADC register switches have been set to the digital equivalent of the analog quantity present at the buffer input during the encode cycle. The logic levels from the register switches provide a parallel digital output until the next encode command arrives. The register switches are then reset and a new encode cycle follows.

Digital Multiplexing and Formatting Logic (See Figure 5.3-9). The function of the digital multiplexing and formatting logic is to combine the encoded data from the ADC, externally generated digital data, and frame (and master frame) identification words into the required output sequence. The specific time slot into which data is inserted is controlled by command pulses from the programming and timing logic. The 10-bit output of the ADC is transferred into the parallel storage register by each word rate clock pulse (so long as the encoder inhibit bus is unenergized). When the encoder inhibit bus is energized, the ten AND gates at the ADC output are disabled, and the ADC output is not transferred into the parallel storage register.

Up to ten 10-bit groups of digital data from external sources can be programmed into selected time slots at any of the four system sampling rates. The data is accepted in parallel. A zero-volt level
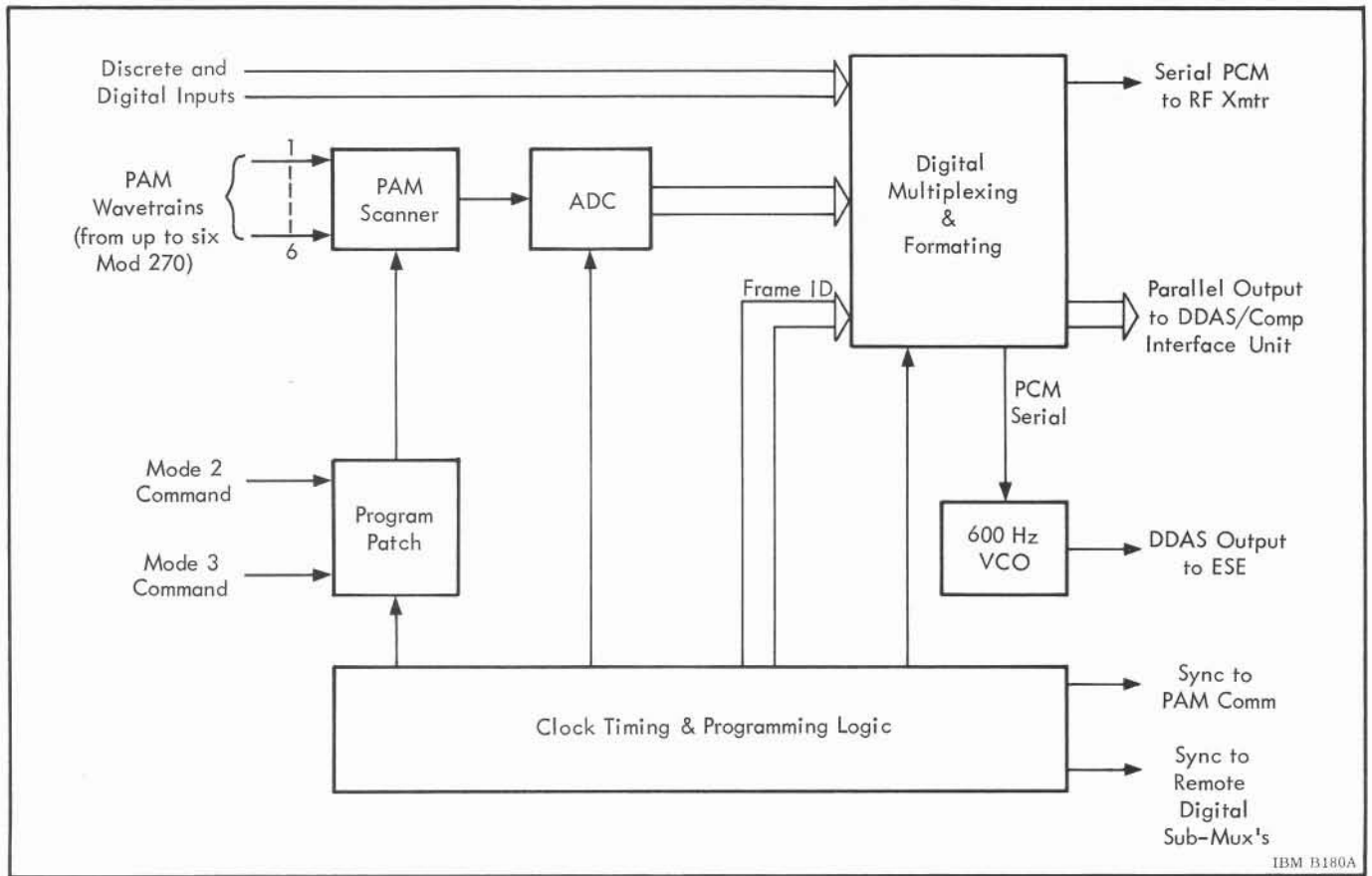
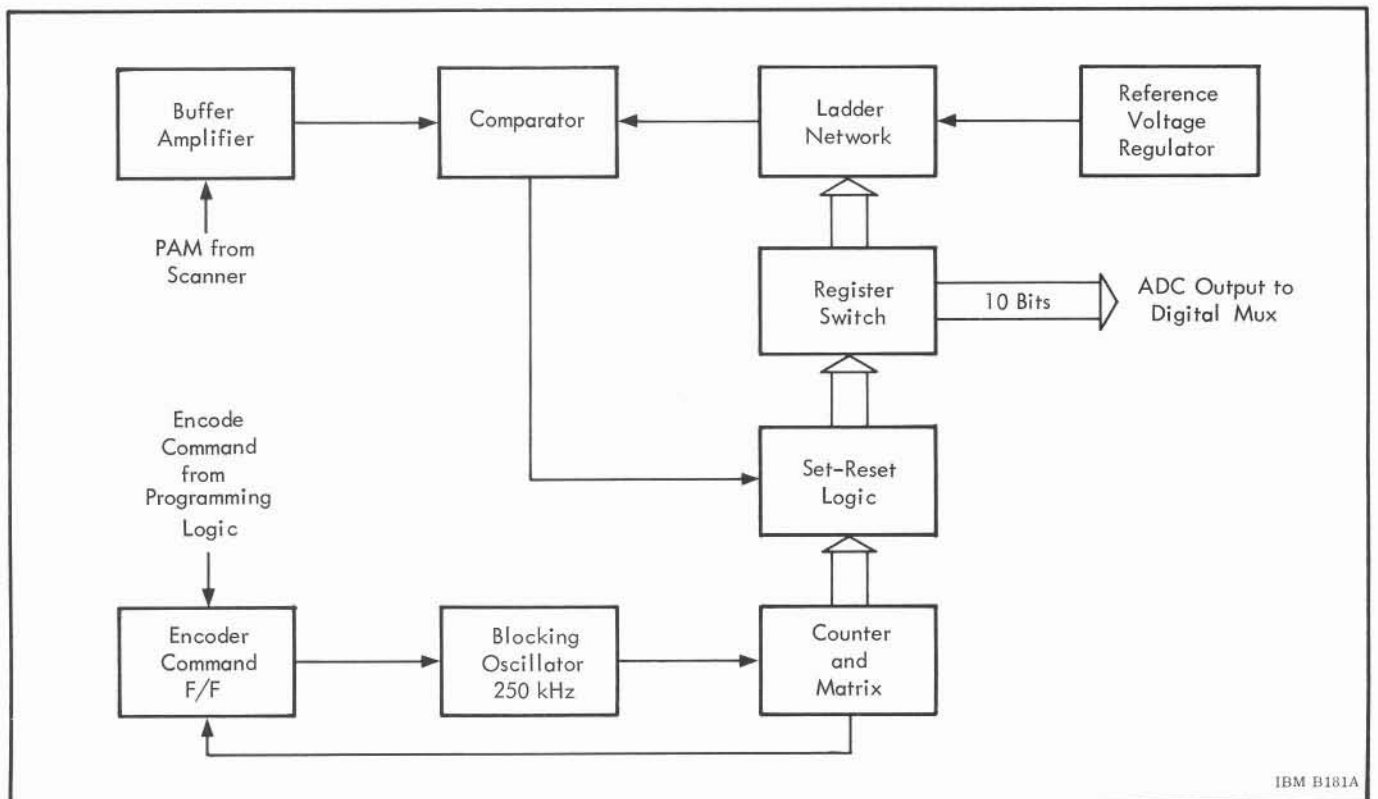Figure 5.3-7 PCM/DDAS Assembly Block Diagram

Figure 5.3-8 Analog-to-Digital Converter Block Diagram

represents a logical "0" and a positive level represents a logical "1". Inputs are buffered and then shifted into a magnetic core register before being transferred into the parallel storage register. The MCR provides temporary storage and dc isolation of the data source. Each MCR, along with its ten associated buffers and other circuits, is powered by an individual supply which is dc isolated. This permits monitoring of several digital data sources without interconnecting their dc commons.

The digital input section has 10 channels. Each channel will accept a 10-bit input in parallel form.

To increase the data handling capacity, the input section may be fed from 10-channel remote digital multiplexers.

A WRITE command to an MCR causes the 10 bits of data to be stored in the magnetic cores. The WRITE command is programmed (at the command program patch) to occur before the time for transfer of the data into the PSR. Typically, the command occurs during the previous word time, but it can be programmed to occur at any word time after the previous sample of the specific channel is read into the PSR. For example, several 10-bit data sets can
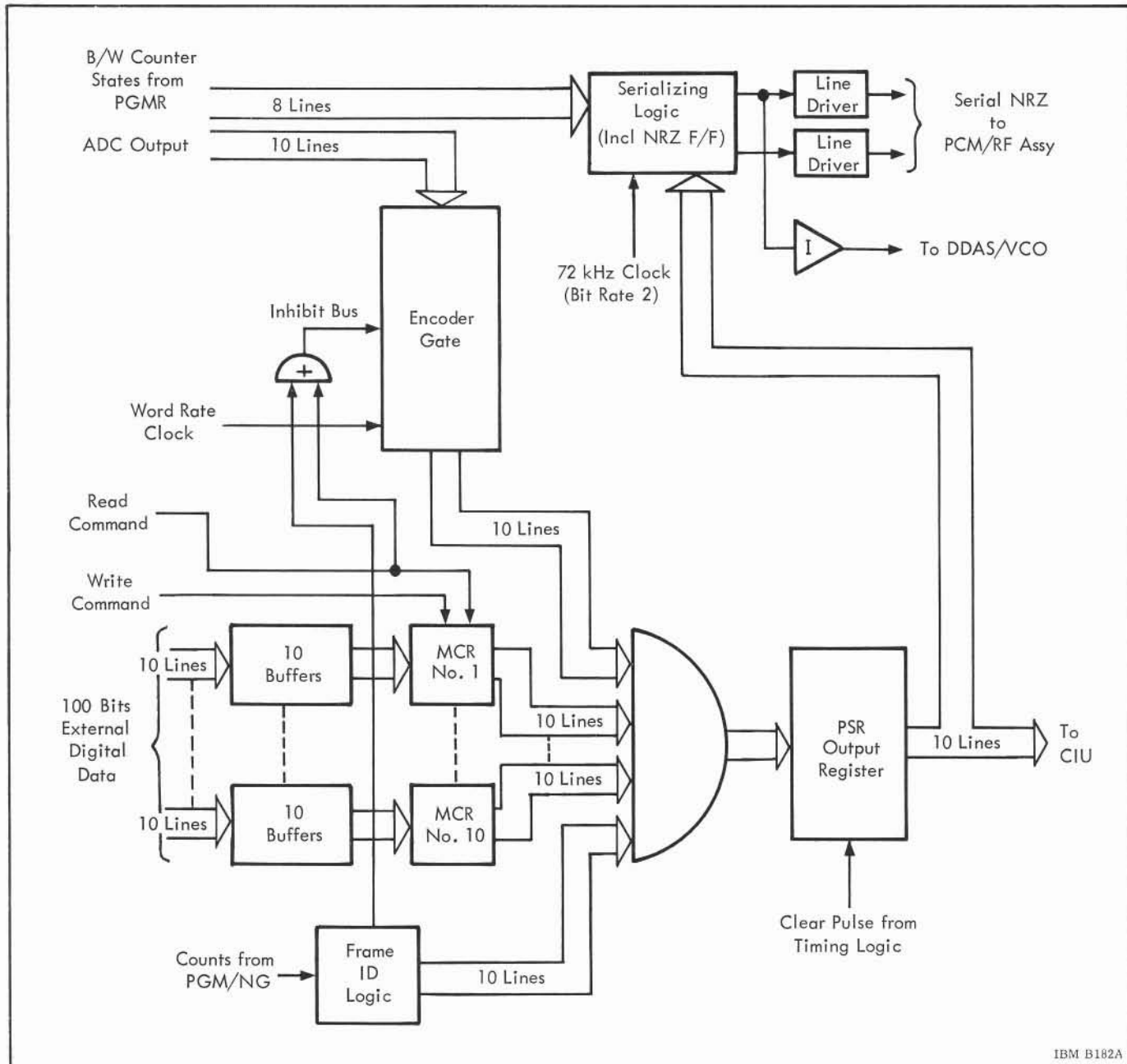
Figure 5.3-9 Digital Multiplexing and Format Logic

be written into their MCR's simultaneously and then placed into specific time slots in the output format. This mode of operation is utilized in monitoring a 40-bit set of data from the LVDC.

The READ command to the MCR is timed to transfer the data into the output register during the least significant bit time of the previous word time. READ commands are also OR'd together to provide a function for inhibiting the transfer to ADC data through the encoder gate.

The frame identification logic generates three unique code groups and inserts them consecutively during the last three word times of the PCM/DDAS frame. Once each 30th frame, the frame ID logic receives a signal from the frame ID reversal flip flop. This signal causes the frame ID logic to complement (reverse) all bits in the frame ID code of the 30th frame. This forms the master frame identification code. The code group is as follows:

Word 29B	1 0 1 1 0 1 1 1 1 0
Word 30A	1 0 1 0 0 0 1 0 0 1
Word 30B	1 1 0 0 0 0 0 1 1 0

The serializing logic shifts the contents of the PSR bit-by-bit into an NRZ flip flop, thus forming the serial NRZ output. The states of the bits per word counter are decoded, combined with the PSR outputs, and clocked to provide set-reset pulses to the NRZ flip flop. The two complementary outputs of the NRZ flip flop are buffered and provided at an output connector for use as a modulating input to the PCM/RF Assembly. One side of the NRZ flip flop provides a modulating input to the DDAS/VCO.

Clock, Programming, and Timing Logic. (See Figure 5.3-10) The clock, programming, and timing logic provides the timing signals necessary for the Mod 301 PCM/DDAS Assembly as well as signals required to synchronize logic in other telemetry assemblies.

One phase of the 72-kilohertz clock provides bit rate pulses to the serializing logic while the other phase steps the B/W counter. The outputs from each trigger of the B/W counter are decoded in the serializing logic and used to select stages of the PSR to form the serial bit train which controls the NRZ flip flop.

Count "2" of the B/W counter is decoded and initiates a series of three successive timing pulses which are 1.75 microseconds apart. The first (clear) pulse occurs approximately 2 microseconds after the

least significant bit is transferred to the NRZ flip flop and resets all 10 stages of the PSR. The second pulse (READ clock) provides the correct timing phase for the transfer of data into the PSR from the ADC, the frame ID logic, or an MCR. The third pulse (WRITE clock) provides the proper phasing for writing external digital words into the MCR's. This pulse also signals the ADC to proceed with digitizing the succeeding analog signal.

The reset pulse of the B/W counter steps the group counter. This pulse occurs during the 2^2 bit time because of offset in the serializing logic. This allows settling time for the analog data inputs, (which are gated by the scanner in synchronism with the group counter) before the ADC begins digitizing. The group counter is a divide-by-two counter (one flip flop).

The reset pulse of the group counter steps the channel counter at a rate of 3600 pps. This consists of a divide-by-five counter and a divide-by-six counter which together form a divide-by-thirty counter.

The reset of the channel counter steps the divide-by-ten frame counter and the divide-by-three multiplexer counter. Both of these counters are stepped at the PCM/DDAS frame repetition rate of 120 times per second.

The 4 counters define each time slot in the PCM/DDAS format. The divide-by-two group counter provides the timing for the interlacing of two multiplexer groups. Each group is comprised of up to three Mod 270 Multiplexers which are controlled by the divide-by-three multiplexer counter. The divide-by-three multiplexer counter provides timing for the sharing of specific mainframe time slots. The divide-by-thirty channel counter steps in sychronism with the sampling action of the 30 by 120-channel analog gates (each sampled 120 times per second) in the Mod 270 Multiplexer Assemblies. The divide-by-ten frame counter steps in synchronism with the 10 by 12 submultiplexers (10 gates sampled 12 times per second).

The counts of each of the 4 counters are decoded and routed to the command program patch. A programming arrangement provides selection of READ and WRITE commands corresponding to specific time slots. These commands provide the timing signals to the MCR's necessary to place external digital data inputs into a specific format. Decoded outputs of the group counter and multiplexer counter are also routed to the scanner program patch, which provides selection of the multiplexer scanning sequence. Up to three multiplexer-scanning sequences may be programmed
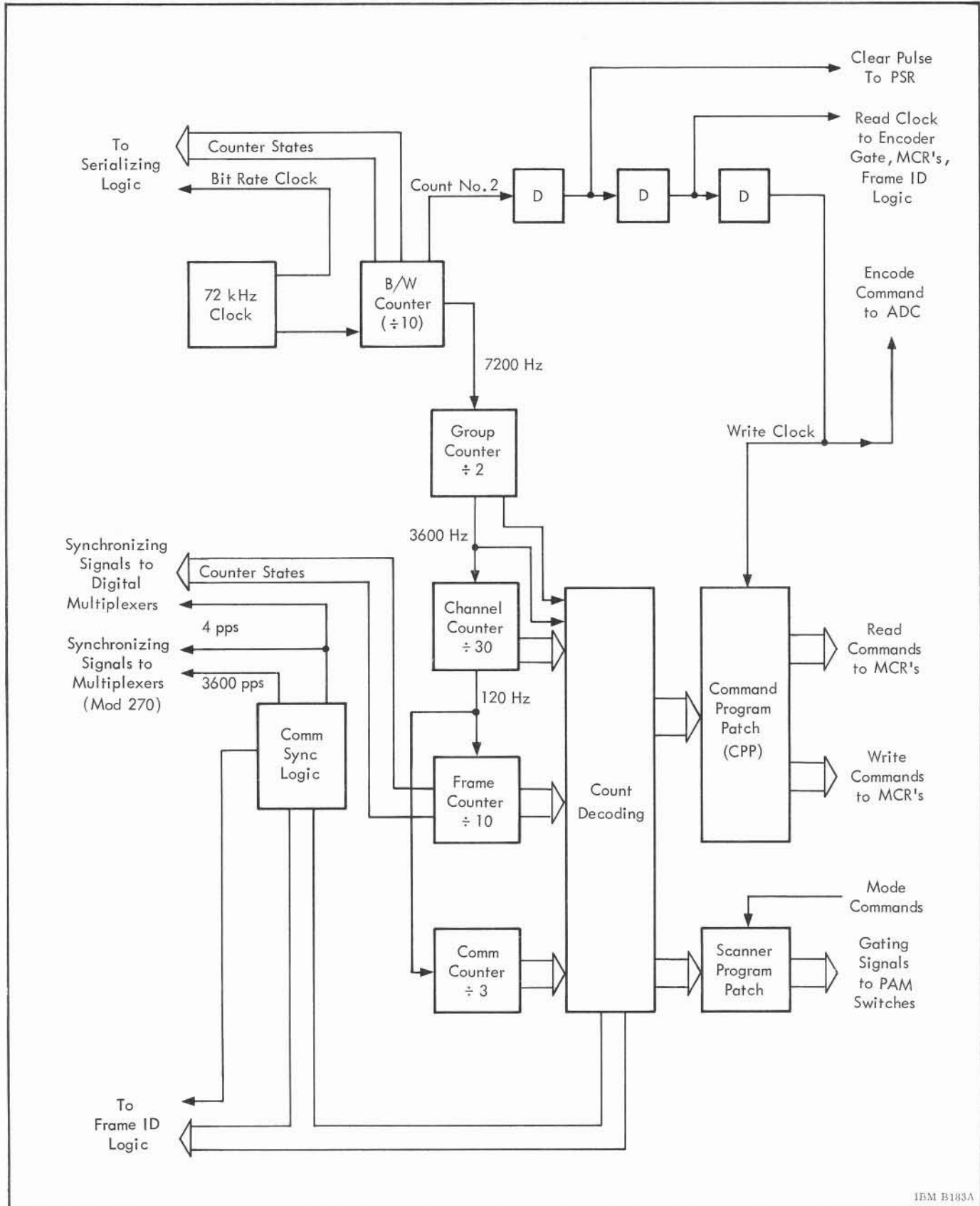
Figure 5.3-10 Clock Programming and Timing Logic Block Diagram

on the scanner program patch. A specific sequence is selectable by an externally generated mode command. The mode commands are 28-volt dc signals from GSE, the vehicle command, or other sources appropriate to the application. (Note: Mode 1 does not require application of an external mode command voltage.)

The control logic also generates 3 sets of waveforms used for synchronizing logic in other assemblies to the clock and frame rates of the Mod 301 PCM/DDAS Assembly. Two sets of waveforms provide the correct waveform shape, frequency, and phase for synchronizing Mod 270 Time Division Multiplexers. The two waveform sets are identical except for the offset in phase necessary for interlacing group A and group B multiplexers. Each set consists of a 3600-hertz square wave and a 278-microsecond pulse with a repetition rate of 4 times per second.

The third waveform set consists of an output from each stage of the divide-by-ten frame counter and a 3600-hertz square wave. This set of waveforms is utilized to synchronize remote digital multiplexer assemblies (Model 410).

DDAS Voltage Controlled Oscillator. The DDAS/VCO provides an FM modulated carrier (600 kilohertz) for transmission of the PCM/DDAS signal (via coaxial cable) to DDAS receiving equipment.

A 600-hertz transistor multivibrator is bias modulated by the serial NRZ data. An amplifier preceding the multivibrator presents an impedance of approximately 50 kilohms to the NRZ data input and also provides a non-linear modulating characteristic which makes the frequency deviation relatively insensitive to variations in the input data logic levels. A frequency deviation of approximately ± 35 kilohertz is used.

A common-collector buffer stage drives a filter which is designed to remove harmonics from the multivibrator output.

Power Supplies. There are six separate power supplies used in the Mod 301 PCM/DDAS Assembly.

A 28 Vdc-to-28 Vdc converter regulates the primary 28 Vdc power to the assembly and isolates the digital return in the Mod 301 PCM/DDAS Assembly from the vehicle 28 Vdc return line. It supplies 28 Vdc ± 1 percent at a nominal 25 watts output to the remaining power supplies in the system.

The 6 Vdc card accepts + 28 volts from the converter and provides + 6 Vdc at 250 mA, and - 6 Vdc

at 50 mA. The + 6 Vdc output provides collector voltage to the programming, timing, and scanning portions of the Mod 301 PCM/DDAS Assembly. The - 6 Vdc output provides base bias voltage to the same circuits.

The +18 Vdc and +5 Vdc card accepts, regulates, and converts 28 Vdc to +18 Vdc and +5 Vdc. All circuit returns on this card are common. These voltages are used in the non-critical and digital circuits in the ADC (encoder) section of the assembly.

The -9 Vdc supply card provides bias voltage for the encoder section of the Mod 301 PCM/DDAS Assembly.

The +15 Vdc regulated supply obtains its input voltage from the +18 Vdc supply. The +15 Vdc is used as the reference voltage from which the comparison voltage is derived for the ladder network in the encoder.

The + 12 Vdc supply is a simple Zener-resistor divider. It is used to provide bias voltage for the circuits on command program patch B.

REMOTE DIGITAL SUBMULTIPLEXER
(MOD RDSM-1D)

The Remote Digital Submultiplexer provides additional digital data handling capability to the PCM telemetry system. A maximum of 100 inputs are provided (see Figure 5.3-11). These inputs are sampled sequentially in groups of 10 and then applied to an output register with 10 outputs. The inputs may be utilized individually or in groups to form digital words. The RDSM handles digital or discrete information only; i.e., its inputs and outputs are set voltage levels which represent either ON-OFF conditions or binary numbers.

The PCM programmer located in the Mod 301 PCM/DDAS Assembly controls the sequential sampling action of the RDSM. The control signal consists of a 4-bit binary word which arrives in parallel format and represents timing information.

The RDSM provides a high degree of flexibility of application by use of isolated input gates and isolated power supplies. The unit has two isolated power supplies; regulated 20 Vdc and + 5 Vdc gate-collector supply. The regulated 20 Vdc is derived from the 28 Vdc vehicle supply by a dc-to-dc converter. The +5 Vdc gate supply is developed on each gate card by rectifying and filtering the 3600 pps clock signal from the PCM programmer.

The binary inputs to word gates are 0 ± 0.5 volt for a logical "0" input and $7 + 3$ volts for a logical "1" input (In a Type IV RDSM, a logical "1" input is approximately 28 Vdc). The binary outputs from the RDSM and the PCM programmer are the same, 0 ± 0.5 volt and 5 ± 1 volts.

An auxiliary card is available with the RDSM. It may be added to the unit to adapt it to a specific input. This auxiliary card is a divide-by-six ($\div 6$) card which has an output signal that is used to cycle the liquid level measuring system.

REMOTE DIGITAL MULTIPLEXER (MOD 410)

One of these assemblies will be utilized in the Instrument Unit to take the 40-bit (parallel) words from the LVDC/LVDA and provide 10-bit words at the output to feed the LVDC words into the Mod 301 PCM/DDAS Assembly for transmission to the ground. Another Mod 410 Assembly will be utilized for other digital and discrete inputs.

A detailed diagram is given in Figure 5.3-12. There are 40 data input bits from the LVDA which are fed into the MCR cards for storage and subsequent insertion into the proper time slots of the system format. These time slots will be determined by the command program patch. From the Mod 301 PCM/DDAS Assembly multiplexer, there are two sync pulse inputs to the 410 timing logic; a 3.6-kilohertz group sync pulse, and a 4-hertz master frame sync pulse. Delay circuits in the timing logic provide the correct sequencing of the write, clear, and read commands. The timing logic also steps the 3, 5, 6, and 10 counters. The outputs of the counters are decoded by the command program patches A and B to provide assembly timing which is sufficient to accurately define the occurence of time slots in the system format.

DDAS/COMPUTER INTERFACE UNIT

A system interconnection diagram for the DDAS/Computer Interface Unit is illustrated in Figure 5.3-13. Figure 5.3-14 is a block diagram showing logic and circuit arrangements for the DDAS/Computer Interface Unit. Positive true logic levels and positive circuit triggering is assumed.

The buffer and dc isolation circuits provide the required input impedance to data and synchronizing signals from the PCM/DDAS. The circuits also contain magnetic coupling elements which provide isolation between the circuits of the PCM/DDAS and other cir-

cuits within the assembly. It should be noted that dc isolation is not required between the LVDA and the assembly logic circuits.

The timing and resetting logic circuits derive the common reset for the counter array and a clock of the correct phase for transfer of data into the holding register from the synchronization signals.

The counter array consists of three binary counters (A, B, and C) each having an internal reset which provides a recycle each 30, 10, and 3 counts, respectively. Each counter is also reset to a count of 1 by the common reset from the timing and resetting logic.

Counter A consists of 5 flip flops, arranged to provide a divide-by-thirty binary counter. Its input consists of the restored 3600 pps synchronization input to the assembly. At the end of count period 30, counter B is internally reset to a count of 1. The internal reset of counter A is also provided as an input to counters B and C.
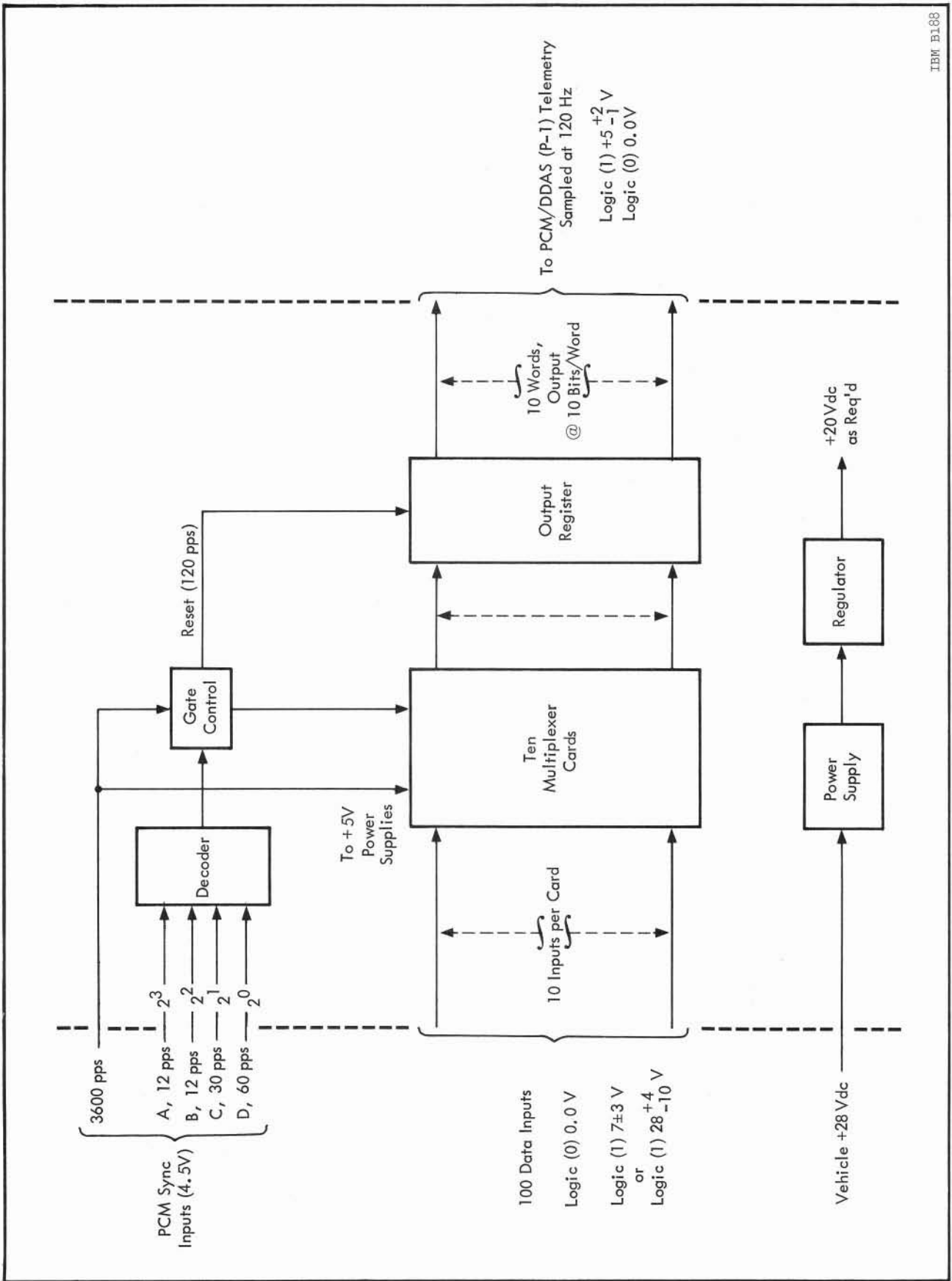
Counter B consists of 4 flip flops configured as a divide-by-ten binary counter. Its input is triggered 120 times per second by the internal reset of counter A. At the end of count period 10, counter B is internally reset to a count of 1.

Counter C consists of 2 flip flops configured as a divide-by-three binary counter. Its input is triggered at 120 times per second by the internal reset of counter A. At the end of count period three, counter C is internally reset to a count of 1.

The twelve address lines from the LVDA are first buffered, and logic levels are transposed to the internal logic levels of the assembly. They are then compared to the states of the counter array by three comparators and two zero detectors.

Comparator A compares address bit 1 with waveform b, and address bits 2 through 6 with the states of flip flops one through five, respectively, of counter A.
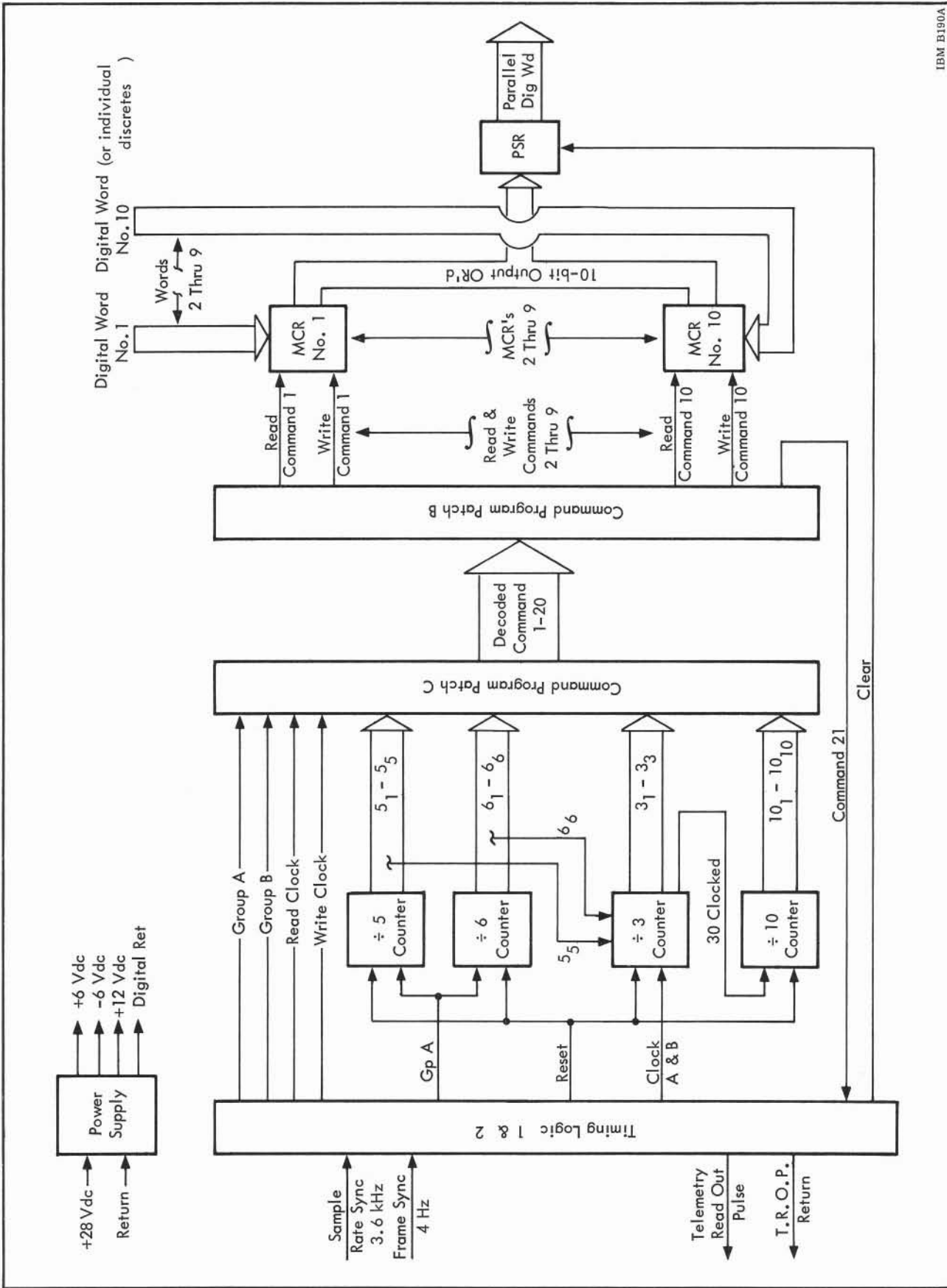
If address bits 7 through 10 are all in the zero state, zero detector No. 1 provides an output which is combined in the OR gate OR_1 with the output of comparator B. Hence OR_1 provides a true output for either an in-register, or all zeros condition of the four address bits.

IBM B188

Figure 5.3-11 Remote Digital Submultiplexer

To PCM/DDAS (P-1) Telemetry
Sampled at 120 Hz

Logic (1) +5 $^{+2}_{-1}$ V
Logic (0) 0.0V

10 Words, Output
@ 10 Bits/Word

Reset (120 pps)

Output Register

+20 Vdc
as Req'd

Gate Control

Regulator

Ten Multiplexer Cards

Decoder

To + 5V Power Supplies

Power Supply

10 Inputs per Card

2^3
2^2
2^1
2^0

3600 pps

A, 12 pps
B, 12 pps
C, 30 pps
D, 60 pps

PCM Sync
Inputs (4.5V)

100 Data Inputs

Logic (0) 0.0 V

Logic (1) 7±3 V
or
Logic (1) 28 $^{+4}_{-10}$ V

Vehicle +28 Vdc

Figure 5.3-12 Mod 410 Multiplexer Block Diagram

Comparator C compares address bits 11 and 12 with flip flops one and two, respectively, of counter C.

If address bits 11 and 12 are both in the zero state, zero detector No. 2 provides an output which is combined in OR gate OR$_2$ with the output of comparator C. Hence OR$_2$ provides a true output for either an in-register, or all-zeros condition of address bits 11 and 12.

The zero detector provides a means by which the address bits may instruct that the status of either (or both) counter B and C be disregarded. This will provide access to the various sampling rates of the PCM/DDAS. The outputs of comparator A, OR$_1$, and OR$_2$ are applied to the data entry AND gate (AND$_1$). The in-sync detector provides a check of the correct synchronization state of the counters and prevents a transfer of data from occurring when the assembly timing is not synchronized with the PCM/DDAS. The output of the in-sync detector is then applied to AND$_1$ as a synchronization pulse.

The data request signal (after buffering and logic level transposition) is also applied as a true logic level to AND$_1$. If the in-sync detector output to AND$_1$ is true, the next "comparison true" signal then activates AND$_1$ and a clock pulse passes through to clear the holding register. After a delay of a few microseconds, the data inputs are gated into the holding register. The gating pulse also triggers the monostable multivibrator, which provides a "data ready" signal to the LVDA.

PCM/RF ASSEMBLY

The Model II PCM/RF Assembly uses the PCM pulse train from the PCM/DDAS Telemeter as modulation for a vhf carrier. The unit is all solid-state circuitry except for the regulated tube-type power amplifier.

The double-ended pulse train from the PCM/DDAS Telemeter is first fed to the PCM signal conditioner (see Figure 5.3-15). This module translates the positive-going and negative-going input excursions into corresponding positive and negative output voltages. This flip-flop circuit maintains the positive or negative voltage output until it is changed by the next input excursion of the opposite polarity. These positive or negative voltages are fed through a jumper plug before being used as the modulation to the transmitter module. The jumper plug is located on an external connector that could furnish the signal to a tape recorder.

The positive level of the two-level input causes the crystal-controlled transmitter module to transmit at its high-band edge, and the negative level at its low-band edge. The power of this carrier is then amplified and filtered to remove the upper harmonics and spurious frequencies. This results in a carrier with a power of 20 watts.

UHF/RF TRANSMITTER ASSEMBLY

The UHF/RF Transmitter Assembly uses the PCM pulse train from the PCM/DDAS Telemeter
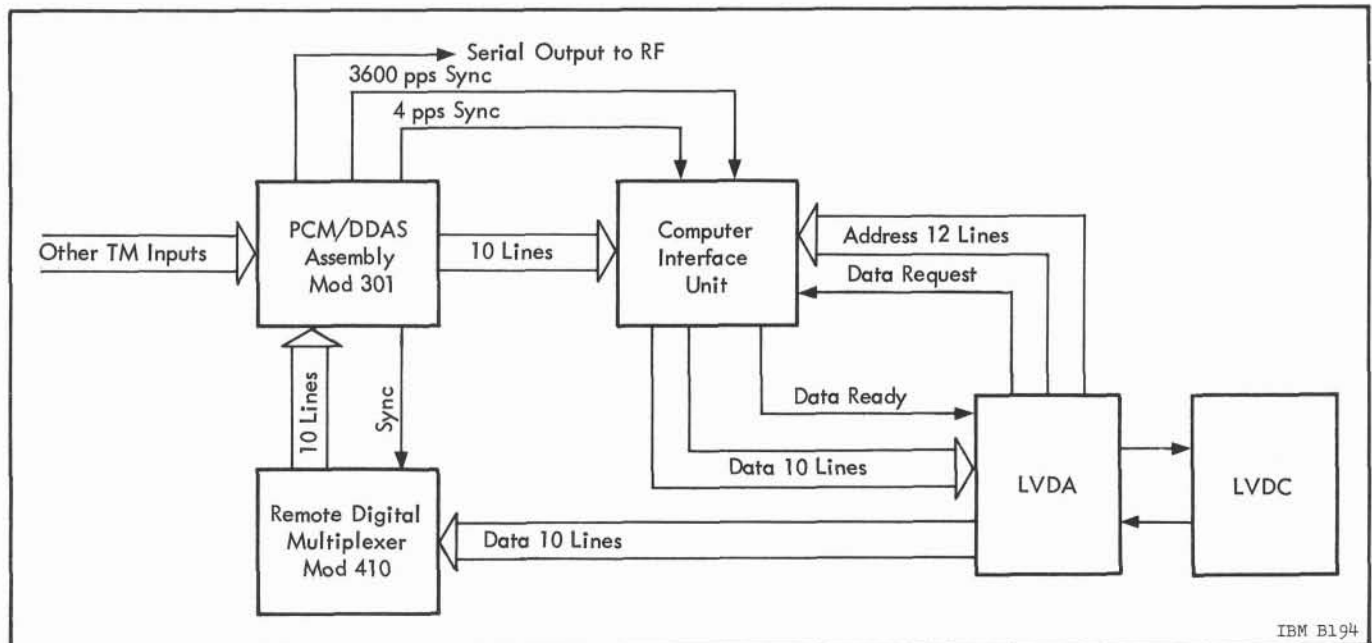
Figure 5.3-13 Computer Interface Unit System Interconnection Diagram

IBM B195
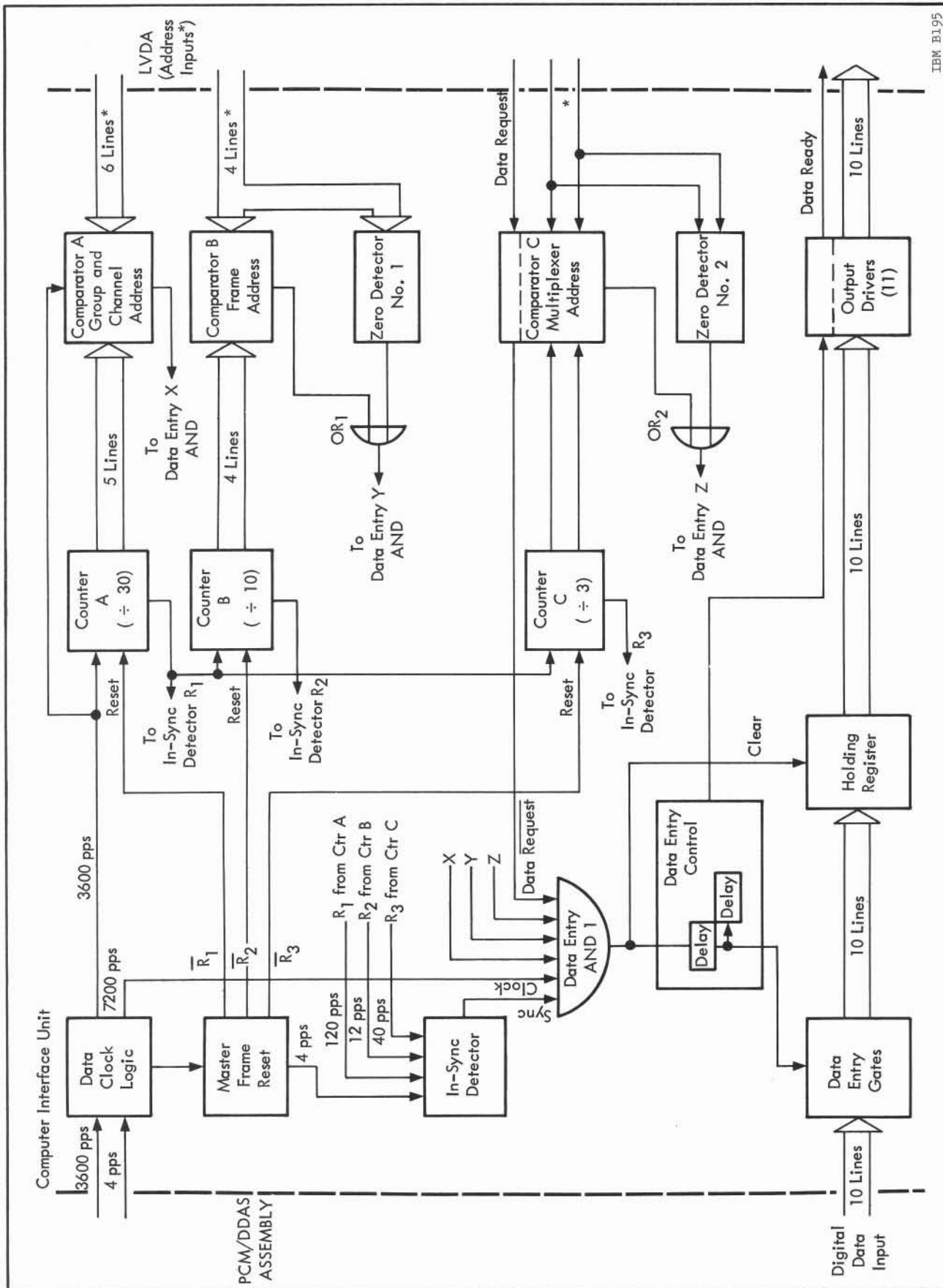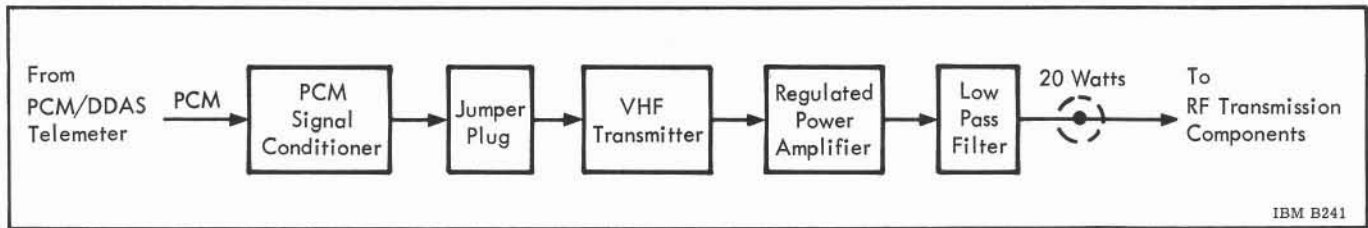
Figure 5.3-14 Computer Interface Unit Block Diagram

Figure 5.3-15 PCM/RF Assembly Block Diagram

Assembly as modulation for a UHF carrier. The unit operates in parallel with the PCM/RF Transmitter Assembly. See Figure 5.3-1 for the location and function of the UHF/RF Transmitter Assembly within the telemetry system. The unit uses a combination of solid-state devices and vacuum tubes to transmit through the RF transmission components.

A UHF/RF Filter serves as a bandpass filter for the output of the UHF/RF Transmitter. The filter ensures that noise generated by the transmitter is not radiated by the UHF antenna (which, if unfiltered, could overlap the signal received by the CCS receiving antennas).

CCS TELEMETRY

Information on the Command and Communication System (CCS) – as related to the Saturn Telemetry System – may be found in Section 6.4.

5.3.7 TELEMETRY CALIBRATION

The purpose of TM calibration (see Figure 5.3-16) is to assist in determination of telemetry equipment accuracy. This is done by furnishing selected telemetry equipment with known, precise inputs in place of the normally broadcast measurements signals. The normally broadcast signals are compared to the calibrated broadcast signals to obtain the exact degree of telemetry equipment accuracy.

TM calibration is used during flight and prelaunch checkout. During flight, TM calibration is controlled by the program of the LVDC. For prelaunch checkout, TM calibration is controlled by ESE in the control center.

Two components and special circuitry in other components provide TM calibration. The two components are the TM Calibrator and the TM Calibrator Controller. Special circuits to accept calibration inputs are located in F2 and P1 Multiplexers, F1 and F2 Telemeters, and S1 Telemeter.

The TM Calibrator and TM Calibrator Controller function together to furnish and control the calibration inputs to the telemetry components. These two units together are referred to as the Calibrator-Controller Assembly. The Calibrator selects the calibration signal level and selects the component to be calibrated. The Controller furnishes regulated power to the Calibrator and also serves as an interface unit, routing and isolating all signals to and from the Calibrator.

The Calibrator-Controller Assembly furnishes six outputs for calibration of model 270 multiplexers or FM telemeters, and three outputs to calibrate single sideband telemeters. Each of these units that are to be calibrated contain circuitry to accept the calibration command and apply the calibrated signals to their individual data channels. While up to nine components can be calibrated by the Calibrator-Controller Assembly, the full capability is not used. For clarity of explanation, this description of operation assumes that full capability is used.

To increase system flexibility, all inputs and outputs of the Calibrator-Controller Assembly are routed through a Measuring Distributor. This allows the inputs and outputs to be patched to selected telemetry equipment without modification to launch vehicle wiring.

During prelaunch checkout, the calibration capabilities are greatly expanded. Any of several preflight calibrations may be selected in addition to the normal inflight calibration.

The equipment is operated in either the inflight mode or the prelaunch mode. During inflight mode, the telemetry equipment is calibrated by preset signals in a preset sequence when commanded to do so by the LVDC program. During prelaunch mode, the LVDC command may be simulated or variable calibration signals may be furnished to the telemetry equipment.
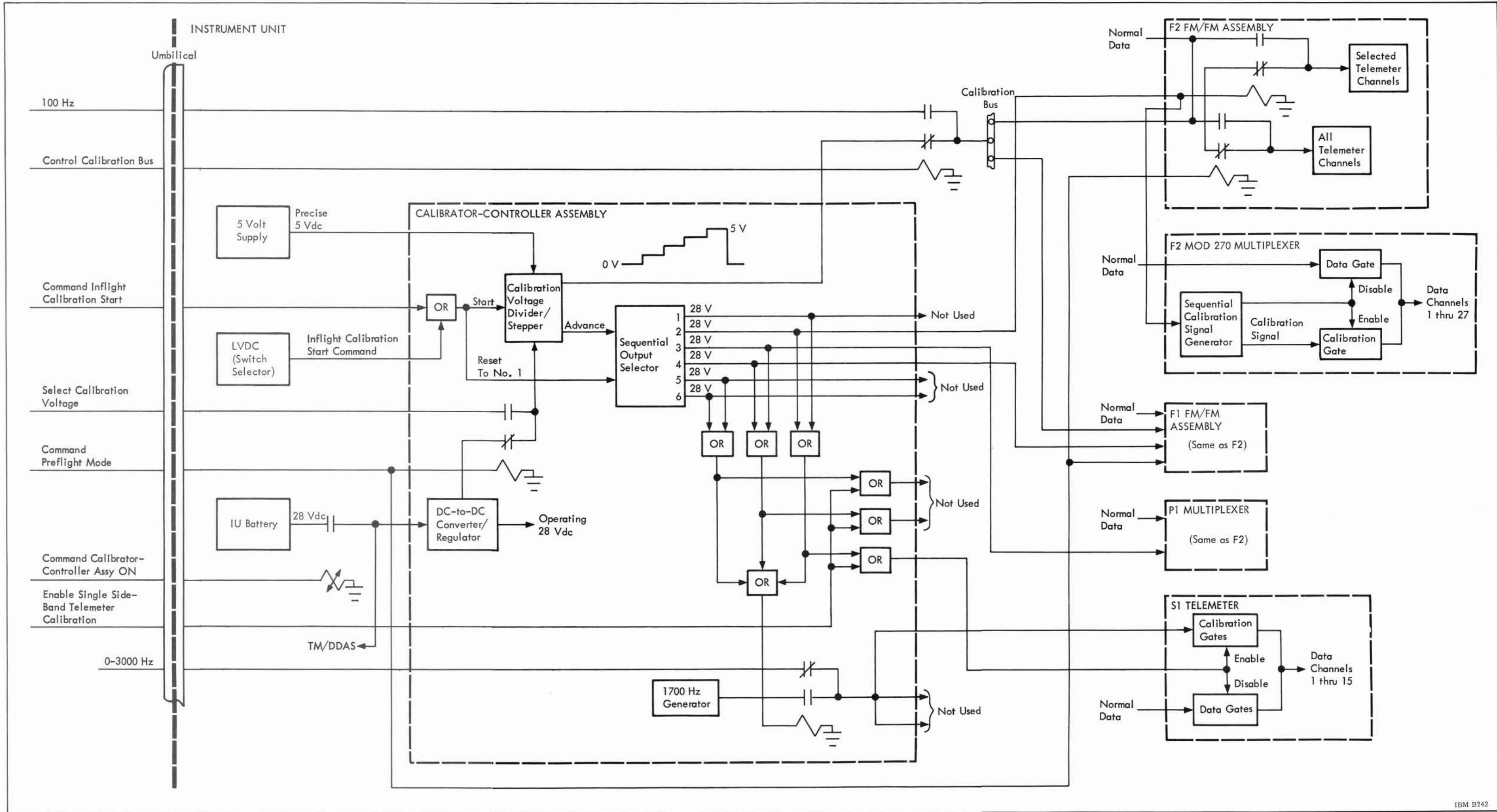
Figure 5.3-16 Telemetry Calibration Functional Diagram

Changed 15 August 1966

INFLIGHT MODE

During flight, upon receiving a 4.2-second, 28 Vdc command from the LVDC, the Calibrator-Controller Assembly will:

- Furnish one 0 to 5 Vdc calibration signal in 1.25 Vdc steps, to be sent to the FM telemeters being calibrated.

- Furnish 28 Vdc commands that are used to enable FM telemeters to receive calibration signals or initiate a calibration sequence in the model 270 multiplexers.

- Furnish a 28 Vdc command that enables the single sideband telemeters to receive calibration signals.

- Furnish a 1700 Hz calibration signal to the single sideband telemeters being calibrated.

The equipment uses 28 Vdc from the vehicle power supply and 5 Vdc from the 5 V Supply. The 5 Vdc is used for the precise 0 to 5 Vdc calibration signal sent to the FM telemeters. The 28 Vdc is regulated by the Controller and used for operating power for the Calibrator and the 28 Vdc enable signals sent to calibrate the telemetry equipment.

The inflight calibration start command originates in the LVDC and is routed through the Switch Selector to a clock circuit in the Calibrator-Controller Assembly.

The clock starts a voltage divider and stepper circuit which produces a precise 0 to 5 Vdc, stepped voltage output. This sequentially stepped output goes from 0.00 to 1.25 to 2.50 to 3.75 to 5.00 and back to 0.00 in a period of 700 milliseconds. The stepped voltage runs through six sequences and is fed to the calibration bus for distribution to the FM telemeters. Each time the stepped voltage finishes a sequence, the output selector circuit advances to select the next telemeter or mux to be calibrated.

When an FM telemeter is selected for calibration, the 28 Vdc signal from the Calibrator-Controller Assembly energizes a relay circuit in the telemeter assembly. This relay inhibits the normal measurement data for 700 milliseconds and substitutes the stepped calibration voltage present at the calibration bus. This calibration is performed only on those FM telemeter channels patched to the YES bus. When another unit is selected for calibration, the relay re-

turns to its original position and normal measurement data is once more allowed to flow to the oscillator channels.

When a Model 270 Mux is selected for calibration, the 28 Vdc signal from the output selector starts an automatic circuit in the mux. This automatic circuit inhibits the normal measurement data input and substitutes an internally generated, stepped 0 to 5 volt signal like the signal generated by the Calibrator-Controller Assembly. When a mux receives the calibrate command, the calibration sequence is delayed until the start of the next mux master frame. Each voltage step is held for one master fram, thus requiring five master frames or approximately 400 milliseconds for a complete calibration sequence. The mux then returns the normal measurement data to its data channels.

While the multiplexers and FM telemeters are being calibrated, the Calibrator-Controller Assembly also selects and calibrates the single sideband telemeters. The Calibrator-Controller Assembly produces a 1700 Hz, 1 volt peak-to-peak calibration signal. Single sideband telemeters are selected and calibrated with the 1700 Hz signal similar to the method used for the selection and calibration of the FM telemeters. A calibration sequence for a single sideband telemeter lasts the time necessary for two multiplexers or FM telemeters to be calibrated, that is, 1400 milliseconds. Normal measurement data is then allowed to return to the data channels of the single sideband telemeter.

PRELAUNCH MODE

During prelaunch mode and while being controlled from the LCC, the Calibration equipment can:

- Simulate the inflight calibration start command from the LVDC.

- Transfer the calibration bus to ESE control and supply an ESE generated calibration signal.

- Select single sideband telemeters for calibration and supply an ESE generated calibration signal.

- Control the output of the Calibrator-Controller Assembly to allow any step of the 0 to 5 Vdc calibration signal to be applied to the calibration bus.

The airborne equipment uses the same 28 Vdc and 5 Vdc power used in the inflight mode. During prelaunch mode, however, the 28 Vdc may be switched OFF, thus turning the Calibrator-Controller Assembly OFF.

An inflight calibration start command simulation signal may be originated by the ESE. This signal is fed to an OR circuit in the Calibrator-Controller Assembly and starts the same calibration sequence that is started by the inflight calibration start command from the LVDC. This signal must be present for at least 4.2 seconds.

The Calibrator-Controller Assembly's signal to the calibration bus may be switched out and substituted by an input from the ESE. This input is a 100 Hz signal with the primary purpose of calibrating those FM telemeter channels that carry frequency-intelligent measurement data.

A command from the ESE can cause the single sideband telemeters to be switched to the calibration bus in the Calibrator-Controller Assembly. A sweeping 0 to 3000 Hz calibration signal is then applied to the bus from the ESE. This 1 V peak-to-peak signal sweeps from 3000 Hz to 0 Hz to 3000 Hz in a 10 second period.

Application of a preflight calibration mode command causes those FM telemeter channels connected to the YES bus and the NO bus to be switched to the calibration bus. This command also allows the 0 to 5 Vdc calibration voltage from the Calibrator-Controller Assembly to be completely controlled from the ESE.

After the preflight calibration mode command has been sent, the 0 to 5 Vdc output is under the complete control of the ESE. The ESE may select any of six possible signals to be supplied by the Calibrator-Controller Assembly to the calibration bus. Any of the five voltage steps or a continually stepping voltage may be selected. Each of these six signals may be applied to the calibration bus for as long as desired.

SECTION 5.4
TELEVISION SYSTEM (SATURN V)

The Saturn V Launch Vehicle television system will be used to provide both real-time and permanent visual data on the performance of certain vehicle functions. No television is planned for use on Saturn IB operational vehicles.

A block diagram of the vehicle and ground equipment is illustrated in Figures 5.4-1 and 5.4-2. Table 5.4-1 lists the television system characteristics.

Figure 5.4-1 illustrates the path of the video signal from the camera, through the video register, to the transmitter. A description of the circuitry follows:

The television cameras are small, having an outside diameter of 7 centimeters (2.8 inches) and a length of 35 centimeters (13.8 inches) The Saturn V television system will include two such cameras.

The video register unit is a sync generator and video multiplexing unit. This unit is capable of driving up to 4 cameras, accepting the video outputs of each, and multiplexing the output at either a field, frame, or 2-frame rate. The Saturn V system will be multiplexed at a 2-frame rate.

Since there are actually 4 areas to be viewed in the engine compartment and only 2 cameras, a split fiber optic bundle is used on each camera. Each bundle is split at the lens' end and uses two objective lenses. The view covered is shown in Figure 5.4-3.

The output of the video register unit is fed to the transmitter, which is an FM system.

The ground system shown in Figure 5.4-2 provides 2 units for recording the demodulated signal as it is received and a system for real-time viewing.
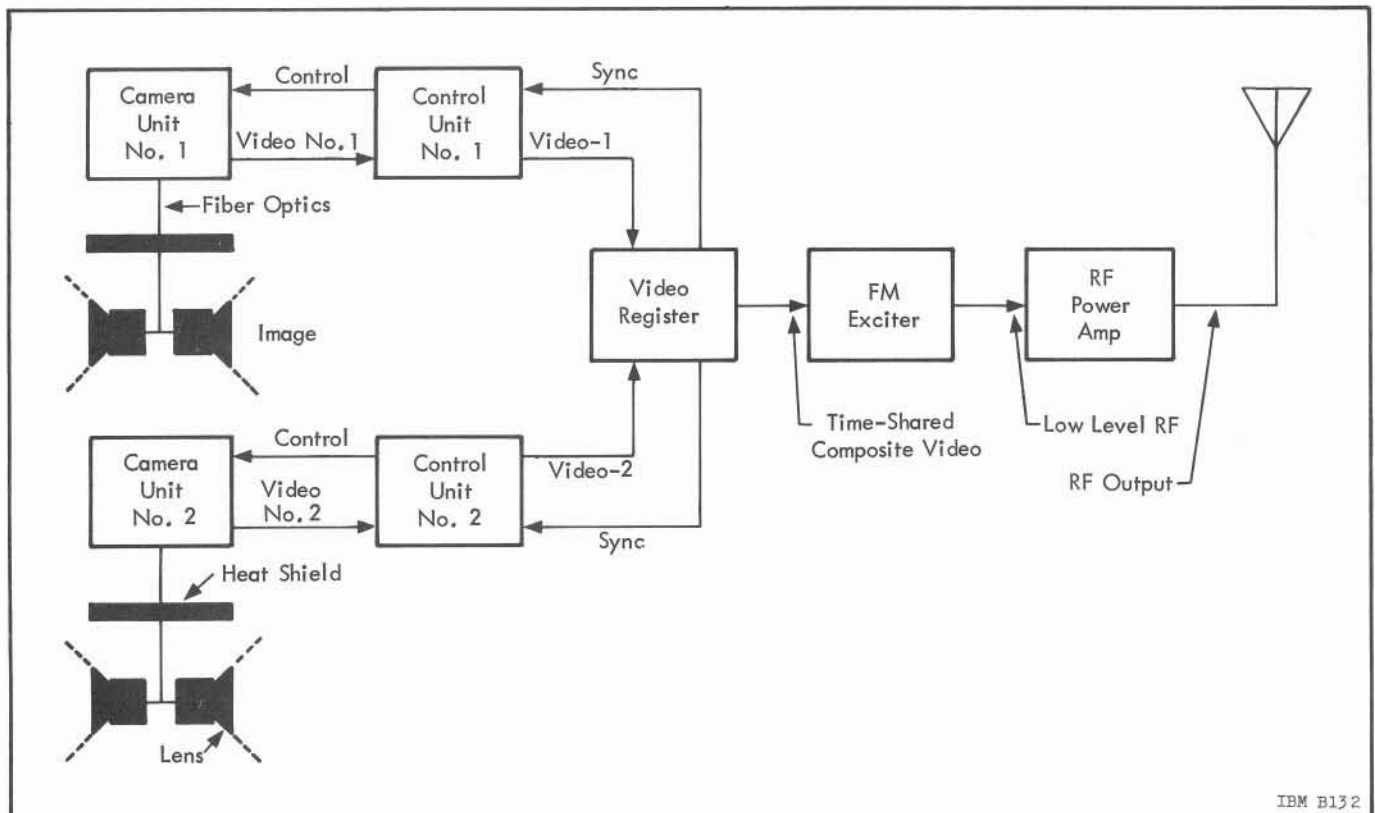
Figure 5.4-1 S-IC Television Function Block Diagram

Table 5.4-1 Saturn V Launch Vehicle
Television Characteristics

Transmitter

Video bandwidth	8 MHz
Modulation	FM
Deviation	10 MHz (for complete video
Output power	2. 5 watts (minimum)
Unmodulated frequency	1705 MHz ± 0. 20%
Video resolution (horizonal) of received picture	500 lines

Closed Circuit Camera System

Camera light sensitivity	10. 76 lumens/meter^2 (1. 0 foot candle)
Video bandwidth	8 MHz
Frame rate	39/second
Scanning	2:1 Interlace

Specifications of Television
Ground Station for Support
of Saturn Television Systems

Parametric amplifier

Gain	20 db (minimum)
Noise figure	1. 35 db
Frequency range	1700 to 1720 MHz

Receiver

Frequency range	1700 to 1720 MHz
Gain	90 db (minimum)
Noise figures	12 db (maximum)

Signal Processing and Distributing
Amplifier

Video bandwidth	8 MHz
Number of outputs	4

Sequence Decoder

Video bandwidth each output	8 MHz
Number of outputs selectable	1 to 16
Switching time	0. 1 us

Video Tape Recorder

Video bandwidth	5. 5 MHz
Tape speed	38 cm/s (15 in. /s)
Recorder time	96 minutes

Kinescope Recorder

Camera frame rate	30/second
Film capacity	365. 8 meters/minute (1200 feet/minute)

Viewing Monitor

Video bandwidth	8 MHz
Video resolution (horizontal)	600 lines
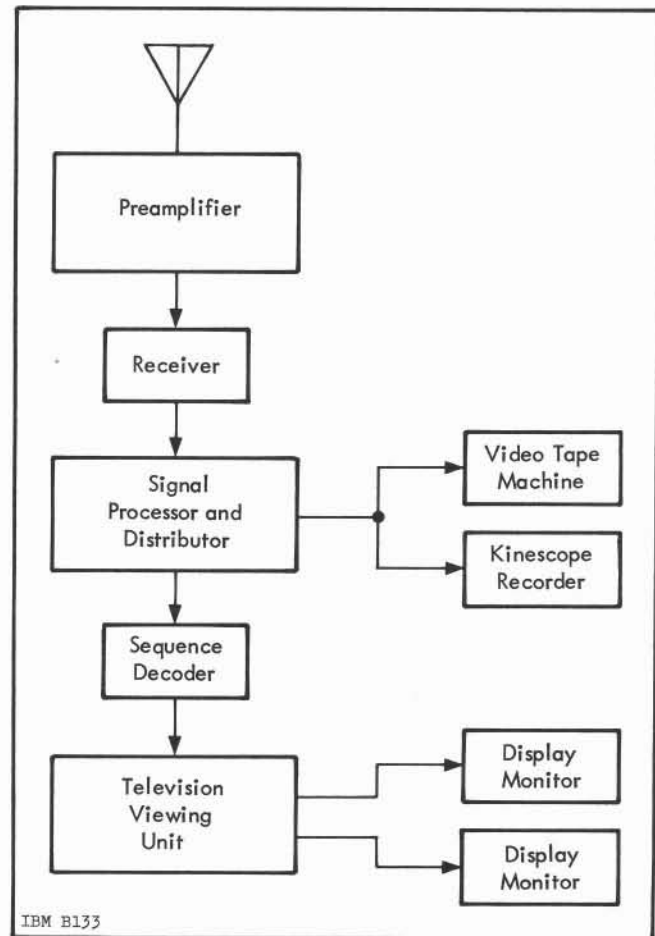
Figure 5.4-2 Television Ground Receiving Station

The systems used for recording the video are one broadcast-type video recorder and one kinescope recorder. Video recorded on both of these systems is still in a multiplexed sequence. For real-time viewing, special equipment is required (i.e., sequence decoder and television viewing unit).

The function of the sequence decoder is to demultiplex the video signal and feed the output of each camera to the same monitor. If the sequence decoder were used alone, the video being fed to each monitor would be composed of 2 frames of video followed by 2 blank frames.

To eliminate the "blinking" picture caused by the above signal, the television viewing unit is provided. This unit uses a storage system and command signals from the sequence decoder to fill the blank 2-frame time with video information from the previous frame. This gives a steady picture. The picture information rate, however, would be at 7-1/2 frames per second. Although the picture appears steady, motion would be jerky.
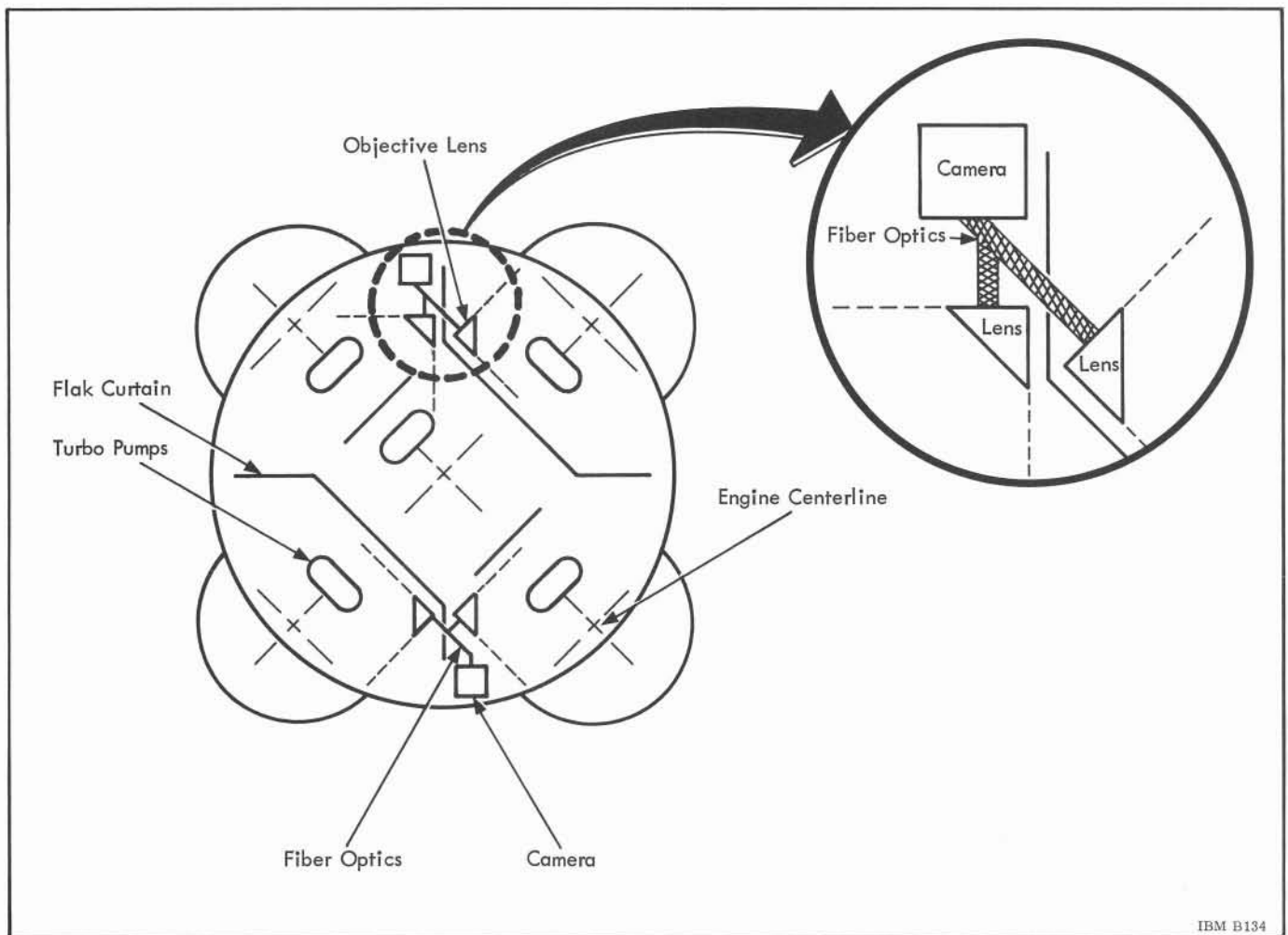
Figure 5.4-3 S-IC Tentative Television Optic Layout

SECTION 5.5
OPTICAL INSTRUMENTATION

Recoverable, inflight 16mm motion picture cameras are used by Saturn IB/V Launch Vehicles to record specific events (e.g., stage separation). The cameras are mounted in camera capsules within ejection tubes. The complete units are located in close proximity to the subjects the cameras are to record.

When the cameras have completed their missions, the capsules (with cameras intact) are pneumatically ejected from their mounting tubes. Following this ejection, camera capsule stabilization flaps are deployed. At an altitude of 4300 meters (14,100 feet), a paraballoon is inflated and causes the stabilization flaps to fall away. A recovery radio transmitter and flashing light beacon, located on the paraballoon, are then turned ON. (Saturn IB capsules do not use the beacon recovery aid.)

After touchdown, the capsule effuses a dye marker to aid visual sighting of the capsule and a shark repellent to protect the camera capsule, parabaloon, and recovery team.

CAMERA CAPSULES

The camera capsule, shown in Figure 5.5-1 contains 3 compartments—the camera compartment, recovery systems compartment, and lens compartment. The camera compartment provides:

- Rigid camera mounting
- Controlled atmosphere
- Camera protection
- Control system (recovery aids deployment)

The recovery systems compartment provides the following aids:

- Stabilization flaps
- Paraballoon
- Radio transmitter
- Light beacon (Saturn V only)
- Dye marker
- Shark repellent

The lens compartment provides:

- Secure lens mounting
- Controlled atmosphere
- Lens protection
- Double bulkhead protection (against water leaks)

The Saturn IB radio transmitter is pulse modulated and operates on a frequency 242 megahertz at 15 watts for a lifetime of 36 hours (20 hours on Saturn V). The light beacon, used on Saturn V, operates at a rate of 40 flashes per minute.

CAMERA MISSIONS

On AS-501 and 502, the S-II Stage carries two camera capsules mounted on the structure at Positions I and III. From this position, the capsules view first plane separation, J-2 engine start, engine gimballing, and second plane separation. In addition, the S-II Stage on AS-502 carries 2 rings of separation tracking aid lights on the aft interstage. The rings each contain fourteen 62-watt lamps. The upper lamps provide S-IC camera reference, and the lower lamps are for the S-II cameras.

On AS-502 and 503, the S-IC stage is equipped with 4 recoverable film camera capsules. Two of these cameras, located on the S-IC forward interstage, view forward during the first plane separation and J-2 engine start. The other 2 capsules, mounted on top of the LOX tank at Positions II & IV, contain pulse cameras which view aft into the LOX tank through fiber optics bundles. The LOX tank is lit by strobe lights which operate at a rate of 5 flashes per second. One strobe light provides spotlight illumination of the tank bottom; and the other provides a wide beam for general tank illumination. The strobe lights and pulse cameras are operated and synchronized by a pulse from the camera film marking timer at a rate of 5 per second. The timer also provides for film marking in binary code. A similar timer is used on the S-II stage.

Table 5.5-1 summarizes the coverage provided by the optical instrumentation system.
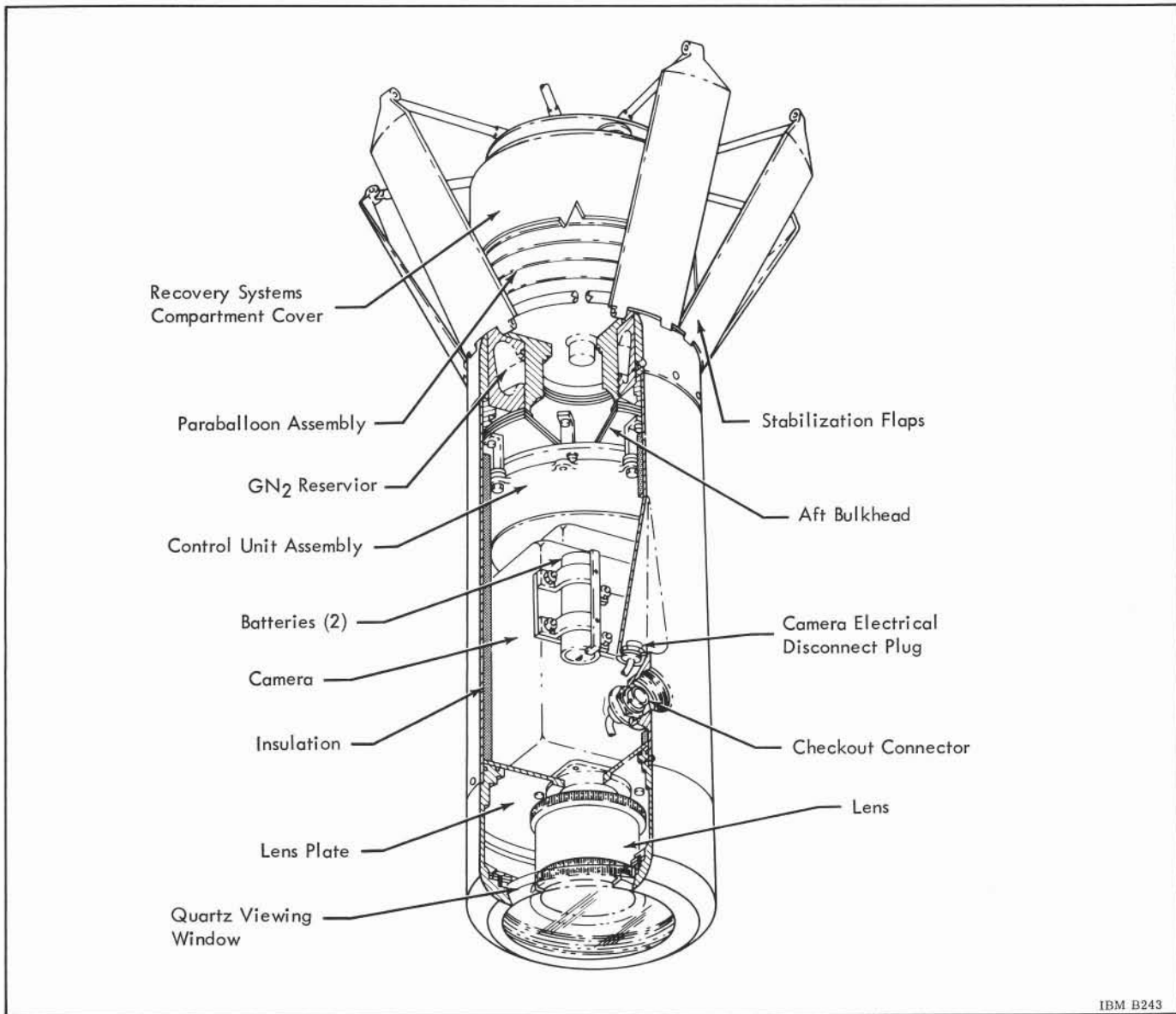
Figure 5.5-1 Typical Direct-View Camera Capsule

Table 5.5-1 Saturn Onboard Motion Picture Coverage

Location of Film Cameras	Coverage Provided	No. of Cameras	Frame Rate	Running Time (seconds)	Vehicle Effectivity	Illumination
Top of S-IB Stage between Positions I & II and III & IV	S-IB/S-IVB separation and J-2 engine start	2	128	29	201, 202, 203	Natural
Top of S-IC LOX tank at Positions II & IV	Interior of LOX tank	2	5	226	502, 503	Strobe lights
S-IC forward inter-stage at Positions I & III	S-IC/S-II separation and J-2 engine start	2	100	27	502, 503	Natural with separation tracking aid lights
S-II thrust cone at Positions I & III	S-IC/S-II separation, S-II second plane separation and J-2 engine start	2	100	38	501, 502	Natural with separation tracking aid lights

CHAPTER 6
RADIO COMMAND SYSTEMS
SECTION 6.1
INTRODUCTION

The Saturn Vehicles carry two different types of radio command systems; for range safety and for data transmission to the Instrument Unit.

The Instrument Unit command system provides digital data transmission from ground stations to the Astrionics System in the S-IVB/IU Stage. This system will be used to up-date guidance information or command certain functions in the S-IVB/IU Stage. The Instrument Unit command system will not be used during powered flight phases.

The range safety command system provides a means to terminate the flight of the vehicle by radio command from the ground in case of emergency situations in accordance with range safety requirements.

Each powered stage of the vehicle is equipped with two command receivers/decoders and the necessary antennas to provide omni-directional receiving characteristics (range safety requirements). The command destruct system in each stage must be completely separate and independent of those in other stages. In case of vehicle malfunctions which cause trajectory deviations larger than specified limits, the vehicle will be destroyed by the range safety officer by means of the range safety command system. The range safety system is active until the vehicle has achieved orbit. After successful insertion into earth orbit, the destruct system is deactivated (safed) by command from the ground. The early Saturn IB and V Vehicles will be equipped with a tone command system (AN/DRW-13) which will be replaced in later vehicles by a secure range safety command system.

SECTION 6.4
SATURN COMMAND AND COMMUNICATION SYSTEM

The Command and Communication System (CCS) is a phase-coherent receiver-transmitter capable of establishing a communication link between the Unified S-Band (USB) ground stations and IU of the Saturn V Launch Vehicle. Specifically, the CCS will: receive and demodulate command up-data for the LVDC/LVDA in the IU; transmit Pulse Code Modulated (PCM) mission control measurements originating in the S-IVB and IU to the USB ground stations for processing; and retransmit the Pseudo-Random Noise (PRN) range code that is received from the USB ground stations. The CCS is located in the IU and consists of a transponder, power amplifier, and antenna system.

The operational requirements of the CCS include command up-data, down-link telemetry in which ranging is desirable but not mandatory. Command up-data is not required on a continuous basis throughout the entire moon mission. The requirement for command varies according to the particular phase of the flight profile. During the launch phase, there is no requirement for up-data since the launch is of such short duration that there is not sufficient time to analyze and make an action decision on unpredictable situations. The LVDC is triple-redundant and is programmed for all predictable situations.

After insertion into an earth parking orbit, command up-data will be required on a non-continuous basis. If all systems are functioning properly, the computer will be corrected (if necessary) and up-dated with the data required to accomplish injection. At a distance of approximately 20,000 kilometers (10,800 nautical miles), the S-IVB/IU separates from the spacecraft and is no longer needed for mission success. The CCS requirement is then complete.

During the launch and earth orbit phases, VHF transmitters are the prime telemetry source and make CCS telemetry unnecessary. At distances of approximately 10,000 kilometers (5400 nautical miles) and greater, VHF transmission is inadequate, primarily due to vehicle and ground antenna gains. After injection, the CCS transponder and a UHF transmitter become the prime telemetry links. The mission

control data from the S-IVB Stage and the IU are interconnected so that both sources of data are available at either source. This makes the mission control data double-redundant during the launch and earth orbit phase. See Figure 6.4-1.

The telemetry data from the S-IVB and IU have different PCM formats. However, the mission control data occupies the same time slots in both formats making the recovery of the mission control data simple, regardless of which receiver is demodulating the PCM stream.

Under normal circumstances, there is no mandatory tracking requirement of the CCS. During launch, tracking is provided by C-band beacons and surface radar with the spacecraft transponders providing tracking during other phases. The CCS transponder will have PRN turnaround capability and may be used as a backup to the CSM transponder in case of failure or desire for a cross-check.

CCS TRANSPONDER OPERATION

The CCS transponder consists of a double-conversion, phase-coherent receiver and a continuous wave transmitter in which the received and
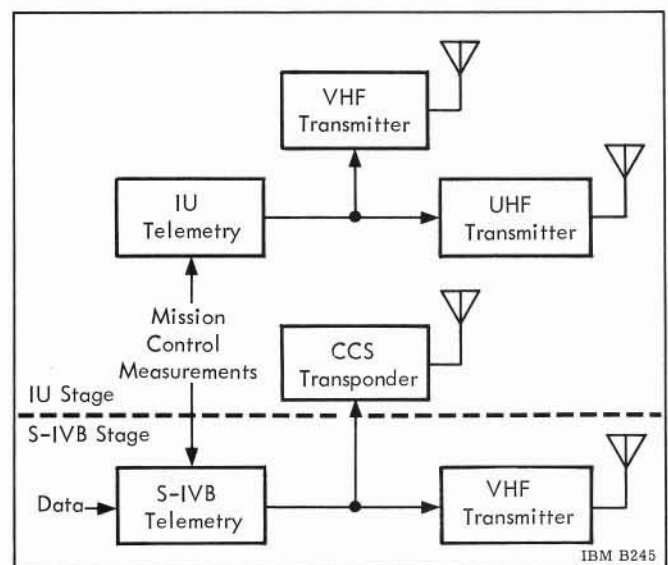
Figure 6.4-1 Telemetry System

transmitted signal are integrally related in both frequency and phase. Figure 6.4-2 illustrates the operation of the transponder.

The received frequency is designated 221f and is nominally 2101.8 megahertz where f is nominally 9.51 megahertz. The first mixer converts the 221f input to the first IF of 5f using a local oscillator of 216f. The first IF amplifier has a bandwidth of 4 megahertz and a gain of approximately 100 db with an AGC range of 120 db. After amplification, the 5f signal is converted to f in the second mixer using 6f as a local oscillator. The second IF amplifier is at a frequency of f and has a fixed gain of +54 db and a bandwidth of 16 kilohertz, which is determined by a crystal bandpass filter. The second IF output is then limited to a constant signal plus noise level of +15 dbm before being presented as one input to the loop phase detector.

The other input to the loop phase detector is from a crystal Voltage-Controlled Oscillator (VCO) which has been divided by two. Both local oscillators (216f and 6f) are the 2f VCO multiplied by 108 and 3. This makes the 216f and 6f local oscillators frequency and phase coherent with the 2f VCO. Since the loop phase detector is phase comparing the converted input and the divided VCO, any phase difference between the two will cause an output from the phase detector which is related to the phase difference.

The output of the phase detector is then processed by the loop filter and applied as a control voltage to the VCO. This control voltage is such that the phase difference between the converted signal and the divided VCO will be reduced. The amount and rate of reduction are dependent upon the loop filter, the limiter, phase detector, and the VCO. The receiver is now considered in a "locked" condition and the VCO will follow the input signal in phase and frequency and will contain the up-link doppler shift.

The command up-data is frequency modulated on a 70-kilohertz subcarrier and then phase modulated onto the RF carrier. The PRN range code is directly phase modulated onto the RF carrier. The second mixer has a second output of f, or approximately 9.51 megahertz. The f output is then amplified and limited before wide-band demodulation using the divided VCO. The output of the demodulator will be the 70-kilohertz subcarrier and the PRN range code at baseband. The demodulated signal is then passed through a 70-kilohertz bandpass filter with a bandwidth of 20 kilohertz to remove the command subcarrier. The subcarrier is limited and FM-detected by using a pulse-averaging detector. The FM-detected output is video amplified and presented as an output to be further processed by the command decoder. The baseband PRN range code is one input of the transmitter modulator to be described later.
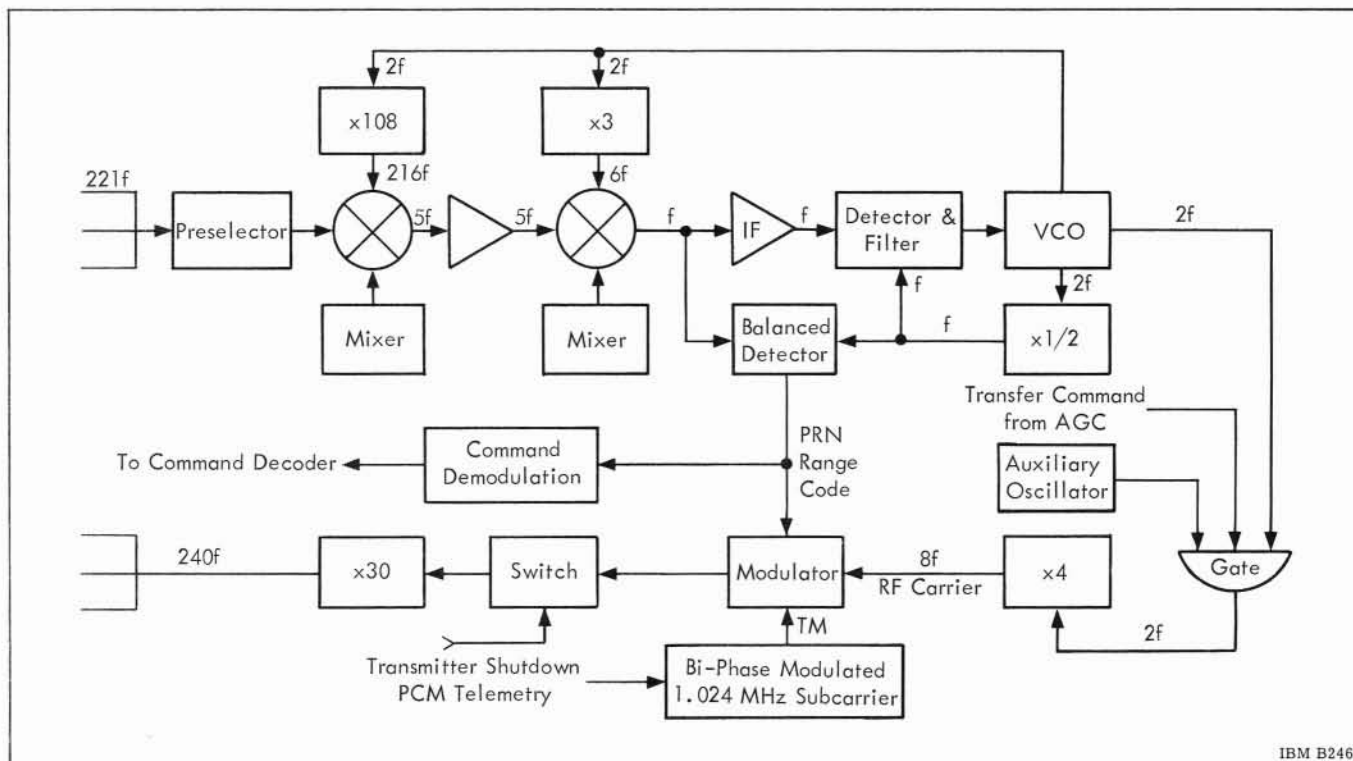
Figure 6.4-2 CCS Transponder Block Diagram

The transmitter section of the transponder receives 2f from the VCO and multiplies it by 4 to 8f. The 8f is then phase modulated and multiplied by 30 to 240f. Thus, the received and transmitted frequencies are integrally related by a ratio of 240 to 221. The phase modulator of the transmitter has two modulation inputs, the PRN code and the down-link telemetry subcarrier. Either modulation input may be separately set to modulate the down-link RF carrier to 2 radians for a peak modulation capability of 4 radians.

The down-link telemetry subcarrier consists of a 1.024-megahertz crystal oscillator that is bi-phase modulated by 72-kilohertz, non-return-to-zero PCM data which contains mission control measurements. This subcarrier is compatible with the USB ground stations although its bit rate is different from the Lunar Excursion Module and Command and Service Module PCM bit rates.

An auxiliary oscillator is included so that when the receiver is unlocked, the auxiliary oscillator will provide a noise-free carrier for the transmitter. When locked, the auxiliary oscillator is gated out by the receiver AGC.

AGC is provided by a second phase detector which phase detects the unlimited second IF and the divided VCO (phase shifted 90 degrees). This detector will therefore have an output directly proportional to the amplitude of the input signal. The AGC system is designed to maintain the signal input to the limiter at a constant power (within 3 db) over the operating signal dynamics.

Since the CCS and LEM transponders have identical frequencies, a possibility of interference exists if the two are operated simultaneously at close range. The CCS transponder has the capacity to shut down the transmitter by an external command. This shut down command will be initiated before the spacecraft transponder is used. The CCS receiver will remain alive so that commands may be received at all times. It is desirable that the CCS transponder transmitter be turned back ON after sufficient separation from the LEM transponder. This will allow turnaround ranging to better define the trajectory of the S-IVB/IU after separation from the spacecraft.

Table 6.4-1 shows the main specifications of the CCS transponder.

ANTENNA SYSTEM

The antenna system for CCS is designed to meet 3 requirements:

- To provide adequate coverage for prelaunch and launch conditions.

- To provide the wide angular coverage required at low altitude during parking orbit (where roll maneuvers will be performed).

- To provide adequate coverage of all stations which are optically visible during the period from injection to spacecraft separation.

To meet these requirements, a system consisting of an omnidirectional antenna pair for the receiver and an omnidirectional pair and a directional antenna for the transmitter is required. The omnidirectional pair for the receiver will be permanently connected and will essentially maintain omnidirectional angular coverage for the receiver from launch throughout the mission. The transmitter will operate into an omnidirectional pair from launch through injection to a point approximately 10,000 kilometers (5400 nautical miles) from the earth. At this distance, the path loss becomes great enough to require additional gain from the vehicle antenna, and the transmitter is then switched to a directional antenna having an initial

Table 6.4-1 CCS Transponder Characteristics

Frequency Received	2101.8 Hz
Frequency Transmitted	2282.5 Hz
Offset Ratio	240/221
Noise Figure (including preselector)	13 db
Threshold Noise Bandwidth	400 Hz
Strong Signal Noise Bandwidth	2330 Hz
Carrier Threshold (S/N = +6 db)	-126 db
Dynamic Range	100 db (120 db capability)
Transmitter Power	500 MW
VCO Gain	400 Hz/volt
Phase Detector Gain	0.35 volt/degree
Phase Multiplication	110.5
Tracking Range	±185 kHz
Input Power	28 ±4 Vdc @ 32.5 W
Weight	9.3 kg (20.5 lbs)
Size	13.7 x 24.9 x 37.9 cm (5.4 x 9.8 x 14.9 in.)

gain of about 6 db. The beamwidth of this antenna
is adjustable, so that it may be reduced. (As the
range becomes greater, the angle subtended by the
earth becomes smaller, and the required gain in-
creases.) The orientation of the directive antenna
pattern remains fixed with respect to the vehicle,
and directing of the pattern toward the ground stations
is accomplished by attitude control of the vehicle.

The omnidirectional antenna elements are
rectangular half-loops, located diametrically opposite

on the vehicle body and driven with equal power. The
directional antenna is a 3 x 3 array of Archimedian
spirals, connected through a switchable system of
phasing and power-dividing transmission lines. The
antenna system is shown in Figure 6.4-3. The CCS
transponder and the UHF telemeter transmitter may
be switched to share an omniantenna or to their own
separate directional antennas as just described.
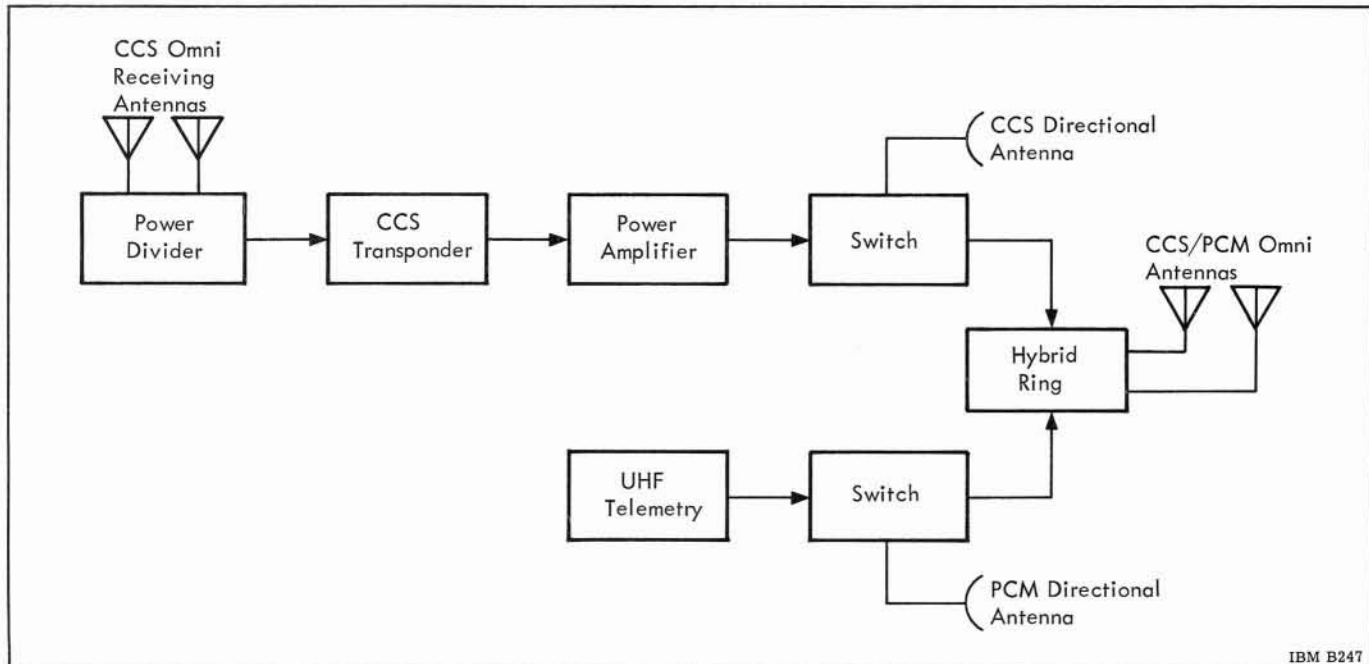
Figure 6.4-3 CCS Antenna System

SECTION 7.1
SATURN TRACKING INSTRUMENTATION

The purpose of radio tracking is the determination of the vehicle's trajectory. Tracking data is used for mission control, range safety, and post-flight evaluation of vehicle performance.

The Saturn vehicles carry several tracking transponders as listed in Table 7.1-1. A combination of tracking data from different tracking systems provides the best possible trajectory information and increased reliability through redundant data. The tracking of the Saturn Launch Vehicle may be divided into 4 phases: (1) powered flight into earth orbit, (2) orbital flight, (3) injection into translunar trajectory, and (4) coast flight after injection.

Continuous tracking is required during powered flight into earth orbit. Because of the long burning time (700 seconds) of the 3-stage Saturn V Launch Vehicle, the end of the powered flight phase cannot be covered sufficiently from land-based tracking stations. Therefore, a tracking ship will be located in the Atlantic to obtain the very important tracking data during insertion which is required for orbit determination. Figure 7.1-1 shows tracking stations used during the powered flight into earth orbit. The number of stations which can "see" the vehicle depends on the launch azimuth. Station visibilities are indicated in Figure 7.1-2 for launch azimuths of 72.5 degrees and 105 degrees. The figure is based on a typical Saturn V powered flight trajectory.

In addition, the Saturn Launch Vehicle will be tracked from S-band stations at Cape Kennedy and on the Atlantic tracking ship. These stations have dual tracking capability; i.e., they can simultaneously track the two S-band transponders on the vehicle (one in the IU and the other in the Apollo Spacecraft). The S-band station on Bermuda has only a single capability and will track the Apollo Spacecraft transponder.

During orbital flight, tracking is accomplished by S-band stations of the Manned Space Flight Network and by C-band radar stations listed in Table 7.1-2. The S-band stations, including the Deep Space Instrumentation Facility, can track the Apollo Spacecraft to the moon and will also be involved in tracking after injection. Tracking information collected during orbital flight may be used to update the Saturn guidance system before injection.

In addition to land-based stations, five tracking ships, equipped with C-band radar, and S-band stations, will be available. One of these ships will be used for insertion tracking in the Atlantic.

Tracking requirements for the launch vehicle during second burn of the S-IVB Stage and during the flight period following injection have not been completely defined. Stations listed in Table 7.1-2 will participate in this operation. The DSIF and MSFN stations have essentially the same capabilities. All S-band tracking stations will be equipped for reception of PCM telemetry at VHF and UHF.

Table 7.1-1 Saturn Tracking Instrumentation

| Tracking System | Transponder Location | |
	Saturn I B	Saturn V
S-band	-	IU
C-band radar (2)	IU	IU
AZUSA/GLOTRAC	IU	IU
ODOP	S-IB Stage	S-IC Stage
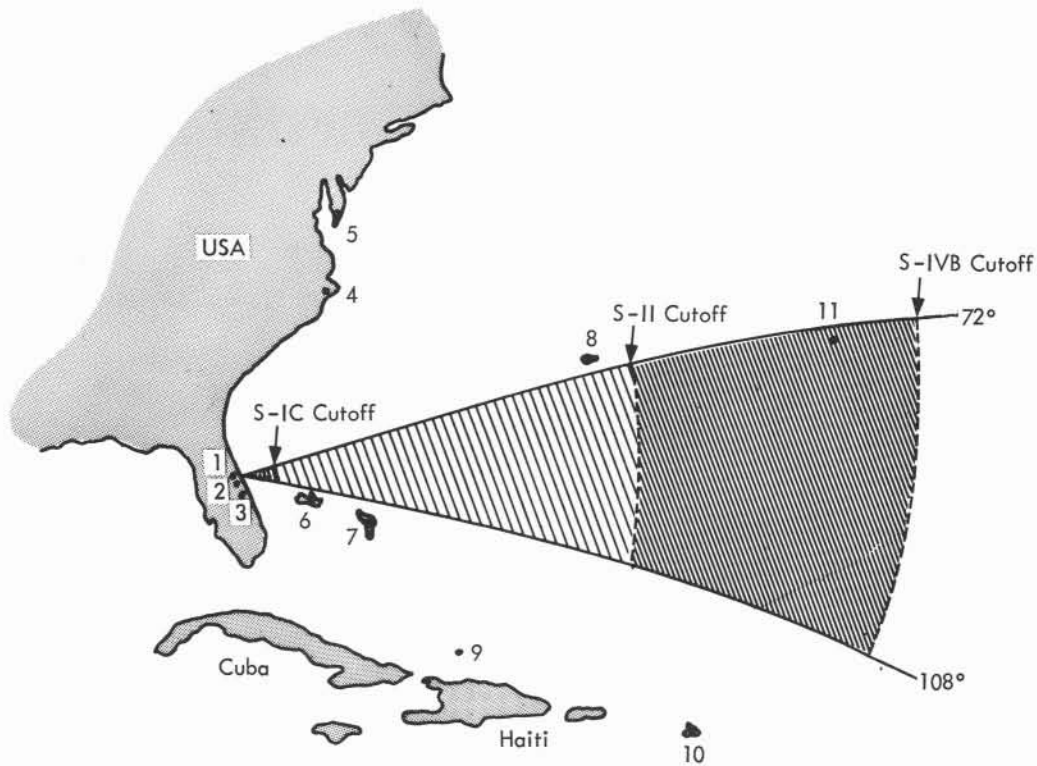
Number	Location	Tracking System	Number	Location	Tracking System
1.	Cape Kennedy, Florida	C–Band Radar ODOP AZUSA/GLOTRAC	7.	Eleuthera Island	GLOTRACK Receiver
2.	Patrick, Florida	C–Band Radar	8.	Bermuda Island	C–Band Radar GLOTRAC Transmitter & Receiver
3.	Valkaria, Florida	(Not used presently for Saturn tracking)	9.	Grand Turk Island	C–Band Radar GLOTRAC Receiver
4.	Cherry Point, North Carolina	GLOTRAC Receiver	10.	Antigua	C–Band Radar GLOTRAC Transmitter & Receiver
5.	Wallops Island, Virginia	C–Band Radar	11.	Atlantic Ship (Position depends on launch azimuth)	C–Band Radar
6.	Grand Bahama Island	C–Band Radar AZUSA			

IBM B73

Figure 7. 1-1 Launch Phase Tracking Stations
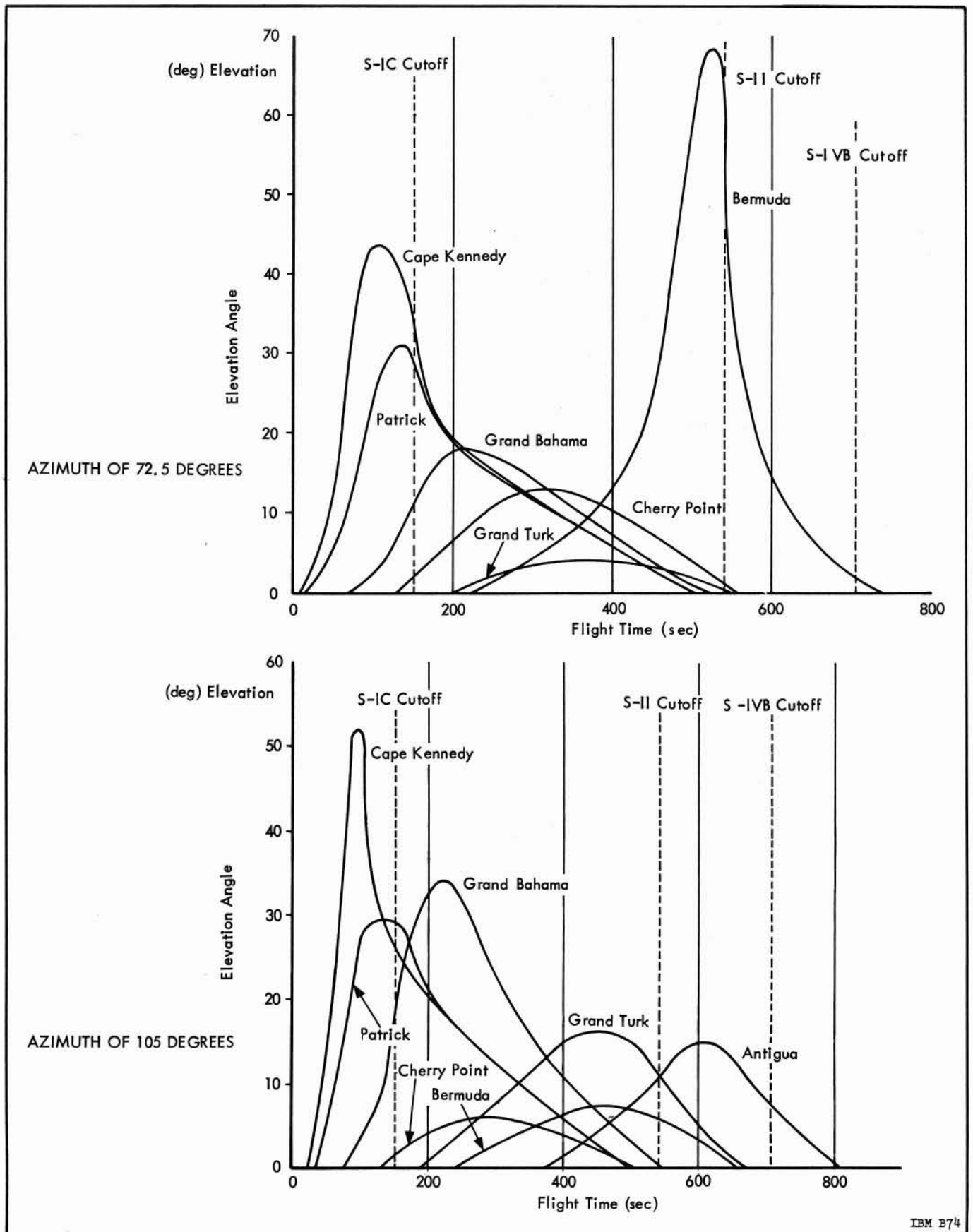
Figure 7. 1-2 Station Visibility for Saturn V Powered Flight

Table 7. 1-2 Orbital Tracking Stations

Name and Location	S-Band Tracking			C-Band Radar
	Single Dual	Ant. Dia		
Cape Kennedy, Florida	D	9.2 m	MSFN	X
Bermuda	S	''	''	X
Antigua Island	S	''	''	X
Canary Island	S	''	''	X
Ascension Island	D	''	''	X
Canarvon, Australia	D	''	''	X
Hawaii, USA	D	''	''	X
Guaymas, Mexico	S	''	''	
Wallops Island, USA	D	''	''	
Guam	D	''	''	
Texas Station, USA	S	''	''	
Canberra, Australia	D	26 m	DSIF/ JPL	
Goldstone, USA	D	''	''	
Madrid, Spain	D	''	''	
Goldstone, USA	D	''	MSFN	
Madrid, Spain	D	''	''	
Canberra, Australia	D	''	''	
Point Aguello, USA				X
White Sands, USA				X
1 Ship (Atlantic)	D			X
4 Ships	S			X

9. 2 m = 30 ft   26 m = 85 ft

MSFN - Manned Space Flight Network
DSIF   - Deep Space Instrumentation Facility
JPL   - Jet Propulsion Laboratory

CHAPTER 10
LAUNCH SITE SUPPORT SYSTEMS

(Deleted)

CHAPTER 11
FLIGHT PROGRAM

SECTION 11.1
INTRODUCTION

A flight program is defined as a set of instructions which controls the LVDC operation from seconds before liftoff until the end of the launch vehicle mission. These instructions are stored in core memory within the LVDC.

The flight program performs many primary functions during the launch vehicle mission. These functions include: navigation, guidance, attitude control, event sequencing, data management, ground command processing, and hardware evaluation. The definition for these functions will change with the mission objectives as R and D or operational; lunar or orbital; or manned or unmanned.

The LVDC flight program may be divided into five parts for the purpose of discussion: the powered-flight major loop, the orbital-flight program, the minor loop, interrupts, and telemetry. The powered flight major loop contains guidance and navigation calculations, timekeeping, and all other repetitive functions which do not occur on an interrupt basis. The orbital flight program consists of an executive routine concerned with IU equipment evaluation during orbit and a telemetry time-sharing routine to be employed while the vehicle is over receiving stations. In addition, in the orbital flight program, all navigation, guidance, and timekeeping computations are carried out on an interrupt basis keyed to the minor loop. The minor loop contains the Platform gimbal angle and accelerometer sampling routines and control system computations. Since the minor loop is involved with vehicle control, minor loop computations are executed at the rate of 25 times per second during the powered phase of flight. However, in earth orbit, only a rate of 10 executions per second is required for satisfactory vehicle control.

Several subroutines or secondary functions are available to the main program. They are utilized at specific times within the main loop or a certain phases of the flight. (e.g., boost navigation computations) In addition, an interrupt routine permits interruption of the normal program operation to free the LVDC for priority work. These interrupt operations may occur at any time within the program. Actually, the minor

loop is an interrupt operation which will occur at intervals of 25 times per second. (i.e., Every 40 milliseconds, the main program is interrupted while gimbal angle readings, accelerometer samplings, and control system routines are processed.)

A programmed telemetry feature is also provided as a method of monitoring LVDC and LVDA operations. The telemetry routine transmits specified information and data to the ground via IU telemetry equipment. In orbit, telemetry data must be stored at times when the vehicle is not within electromagnetic range of a ground receiving station. This operation is referred to as data compression. The stored data will be transmitted later on a time-shared basis with real-time telemetry when range conditions are favorable.

The execution time for any given major loop, complete with minor loop computations and interrupts, is not fixed. The average execution time for any given major loop in powered flight, complete with minor loop computation and interrupt processing, is called the nominal computation cycle for that mode. The computation cycle is not fixed for two reasons. First, the various flight modes of the program have different computation cycle lengths. Second, even in a given flight mode, the uncertainties of discrete and interrupt processing and the variety of possible paths in the loop preclude a fixed computation cycle length. Even though the exact length of any given computation cycle is not predictable, the longest nominal computation cycle in boost should be less than two seconds. In orbit, the computation cycle will be approximately one second for guidance and eight seconds for navigation. For the second S-IVB burn (out-of-orbit flight), the computation cycle will be approximately two seconds in length.

NOTE

This chapter is primarily concerned with the major and minor loops. However, since portions of the program are dependent upon subroutines and various interrupts, brief summaries

of these will be included. Simplified
"word" flow charts of the program are
included to give the reader a better
understanding of how the flight program
routines fit together. These routines
will vary as refinements and changes
become necessary. Therefore, all
descriptions and diagrams should be
considered as typical and for general
information only.

SECTION 11.2
FLIGHT PROGRAM DESCRIPTION

11.2.1 PRELAUNCH AND INITIALIZATION

The sequence of events which take place prior to launch are referenced to the estimated lift-off time. The sequence begins approximately 20 minutes before estimated lift-off (T-20 minutes). Up to this point in time, the LVDC has been under control of the Ground Control Computer (GCC). Ground routines that perform certain prelaunch functions and prepare the LVDC for the flight program are established. At T-20 minutes, the GCC issues a "Prepare-to-Launch" (PTL) command to the LVDC. The PTL routine performs the following functions:

- Executes an LVDC/LVDA self-test program and telemeters the results.

- Monitors accelerometer inputs and calculates the Platform off-level indications. Telemeters accelerometer outputs and time.

- Performs reasonable checks on particular discrete inputs and alerts.

- Interrogates the error monitor register.

- Keeps all ladder outputs zeroed by issuing ladder outputs.

- Processes the Guidance Reference Release (GRR) interrupt and transfers LVDC control to the flight program.

- Samples Platform gimbal angles.

At T-163 seconds, an automatic launch sequence is started. (This action is not communicated to the LVDC.) At T-15 seconds, the launch sequencer issues a GRR alert signal to the LVDC and GCC. In effect, this signal alerts the program. At T-10 seconds, a GRR interrupt signal is sent to the LVDC and GCC. With the receipt of this signal, the PTL routine transfers control of the LVDC to the flight program.

When the Guidance Reference Release (GRR) interrupt is received by the LVDC, the following events take place:

- The computer sets time base zero (T_0).

- Gimbal angles and accelerometers are sampled and stored for use by program routines.

- Time and accelerometer readings are telemetered.

- All flight variables are initialized.

- The ground computer is signaled that the LVDC is under control of the flight program.

During the time period (T_0) between GRR and Lift-off (LO), the LVDC begins to perform navigational calculations and processes the minor loops. At T-8.8 seconds, the engines are ignited. At T=0, lift-off occurs and a new time base (T_1) is initiated. T_1 is used for first-stage flight and for discrete outputs.

11.2.2 MAJOR LOOP

The major loop, shown in Figure 11.2-1*, contains the navigation and guidance calculations, timekeeping, and other repetitive operations of the flight program. Its various routines are divided by

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* The figures in this section contain word flow diagrams of a typical Saturn flight program. The major loop is illustrated first and is followed by several figures which give a functional breakdown of the various routines. A further breakdown of the routines, i.e., subroutines, is beyond the scope of this book. The following symbols are used to represent program functions:

Decision Block   Processing Block

Subroutine   Connectors

(On-page)   (Off-page)
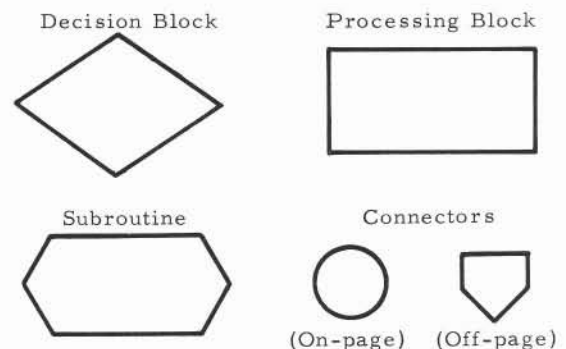
function and are illustrated by individual blocks in the figure. Depending upon mode of operation and time of flight, the program will follow the appropriate "path".

Assuming that the vehicle has just lifted off and the program has been "running" normally, a boost major loop cycle will operate as follows. Entering at block 1 of Figure 11.2-1, it is determined that it is not time for the orbital scheme. Therefore block 2, which is the accelerometer processing routine, will be the first routine to operate. Following the completion of that routine, the boost major loop will continue operating through blocks 3, 4, 5, 8, 9, 1, 2, etc., in that order. This is a repetitive cycle and will continue until it is time for the IGM mode of operation. The path during IGM operation will then include blocks 1, 2, 3, 4, 6, 7, 8, and 9.

After S-IVB cutoff, the vehicle is assumed to be in orbit so the program will necessarily follow blocks 1, 10, and 11 or 12, etc., respectively. This portion of the program is referred to as the orbital major loop.

It should be noted that the minor loop support routine (block 8) is not the minor loop as previously described. Minor loop support is a processing routine within the major loop which supports the minor loop (interrupt) function.

Block 13 includes the minor loop and other interrupt routines. Figure 11.2-2 is a general flow diagram of the interrupt and minor loop processing routines which represent the remainder of the flight program. The interrupt function is shown separately because it operates independently of the major loop. An interrupt can occur at any place and time within the operation of the major loop program.

11.2.3 INTERRUPT AND MINOR LOOP OPERATION

The LVDC/LVDA has a built-in feature which allows priority processing by interrupting the normal flight program. See Figure 11.2-2. An "interrupt" is generated when such a priority task arises. The interrupt transfers LVDC control to a special subroutine which identifies the interrupt source, performs the necessary subroutines, and then returns to the point in the program where the interruption occurred. For example, during the 50th instruction of the boost navigation program, an interrupt might occur. At the completion of that instruction, the program control branches to the interrupt routine, performs the indicated operation, and then returns to the 51st instruction step of the boost navigation program.

The boost navigation routine itself will not be affected in any way except for the time of delay caused by the interrupt operation.

The LVDC/LVDA has provision for 12 interrupts which are listed in Table 11.2-1.

The minor loop, which is initiated by an interrupt, is shown on the right of the figure. A built-in timing feature of the LVDC (under program control) causes a minor loop interrupt approximately every 40 milliseconds (25 times per second) during boost flight. Present plans indicate that the minor loop will operate approximately 21 milliseconds. This implies that the major loop will operate for only 19 milliseconds between each minor loop interrupt. During orbit, the frequency of the minor loop interrupt is reduced to 100 millisecond intervals (10 times per second) and can be considered as divided into 2 modes. The first mode is the attitude update mode (1 per second) in which the gimbal angles are sampled and the ladder (attitude) commands are computed. In the second mode, which is the attitude hold mode, the ladder commands are recomputed (9 per second) based on the previous gimbal samples. In either case, ladder commands are issued at the minimum rate of 10 per second. Under this scheme, the time spent in the minor loop is reduced from over 50 percent in powered flight to approximately 10 percent during orbit.

Table 11.2-1 Interrupts

Decreasing Priority	Function
1	Minor Loop Interrupt
2	Switch Selector Interrupt
3	Computer Interface Unit Interrupt
4	Temporary Loss of Control (TLC)
5	Command Receiver Interrupt
6	Guidance Reference Release
7	S-II Propellant Depletion/Engine Cutoff
8	S-IC Propellant Depletion/Engine Cutoff "A"
9	S-IVB Engine Out "B"
10	Program Re-Cycle (RCA-110A Interrupt)
11	S-IC Inboard Engine Out "A"
12	Command LVDA/RCA-110A Interrupt

11.2.4 FLIGHT PROGRAM ROUTINES

ACCELERATION PROCESSING

The accelerometer processing routine, shown in Figure 11.2-3, accomplishes two main objectives: it accumulates velocities as measured by the Platform, and tries to detect velocity measurement errors through "reasonableness" tests.

The quantity F/M (acceleration) is computed from Platform velocities except when such computations are expected to be "noisy". During noisy portions of the program, F/M is set to a predetermined constant. M/F is also computed in this routine for use within the IGM routine.

The reasonableness test is designed to detect large errors in the formation of velocity words. The test serves as a program switch which selects velocity word channel "B" in the event of a failure in velocity word channel "A". A velocity change is considered reasonable if it falls within ± 50 percent of the expected change, enlarged by a Reasonable Test Constant (RTC). The RTC serves two purposes. First, it provides leeway in the test when the expected change is near zero (e.g., RTC = 5). Secondly, the RTC provides a safety margin during portions of the flight when F/M computations may be questionable (e.g., RTC = 30).

Should both readings fail the reasonableness test, an appropriate velocity change is computed from current gimbal angles, time, and pre-stored values of thrust, mass, and mass flow rate. The reasonableness tests will not correct the cause of the error but will attempt to prevent such errors from causing corresponding errors in guidance and navigation quantities.

BOOST NAVIGATION

The boost navigation routine, shown in Figure 11.2-4, combines gravitational acceleration with measured Platform data to compute position and velocity in the X_S, Y_S, Z_S coordinate system. The computations used in this routine may be found in Chapter 2 of this book. Each block in the figure represents the computation of the values given in Table 11.2-2.

This boost navigation routine is used from GRR until approximately 100 seconds after the first S-IVB cutoff and again from the beginning of restart preparations until 10 seconds after the second S-IVB cutoff.

Table 11.2-2 Boost Navigation Calculations

Block	Computation Results
1	ΔX_g ΔY_g ΔZ_g
2	ΔX_T ΔY_T ΔZ_T
3	X_S Y_S Z_S
4	$\ddot{X}_g$ $\ddot{Y}_g$ $\ddot{Z}_g$
5	I_{gX} I_{gY} I_{gZ}
6	$\dot{X}_S$ $\dot{Y}_S$ $\dot{Z}_S$

PRE-IGM GUIDANCE

Pre-IGM guidance deals with the method of generating attitude commands (χ angles) during the time from Guidance Reference Release (GRR) to initiation of IGM. The word flow diagram for this period is depicted in Figure 11.2-5. This portion of the flight program is sometimes referred to as the "time-tilt" program since these early guidance commands are functions of time. It is also referred to as "open loop" guidance since vehicle dynamics do not affect or influence the guidance commands.

Consider a typical Saturn V flight, where Position I alignment is offset by -18 degrees. This offset occurs because Position I has an azimuth heading of 90 degrees at lift-off whereas the desired heading for Position I is in the direction of firing azimuth (72 degrees). After lift-off, the vehicle is maintained vertically as the roll angle is commanded to -18 degrees. When the tower has been cleared, the roll error is commanded to zero. After this command, roll and yaw commands remain at zero and the vehicle is gradually pitched (tilted) about the vehicle's Y axis to its predetermined boost heading. Rate limiting of the output commands prevents the χ angles from exceeding 1 degree per second.

Just prior to S-IC engine cutoff, the attitude commands are "frozen" until the initiation of IGM. This freeze continues into the initial portion of the S-II Stage flight to eliminate excessive flight perturbations caused by S-IC burnout and S-II ignition.

IGM ROUTINE

The IGM routine (shown in Figure 11.2-6) is primarily used to compute the steering angle commands and the "time-to-go" for the remaining burn time of the
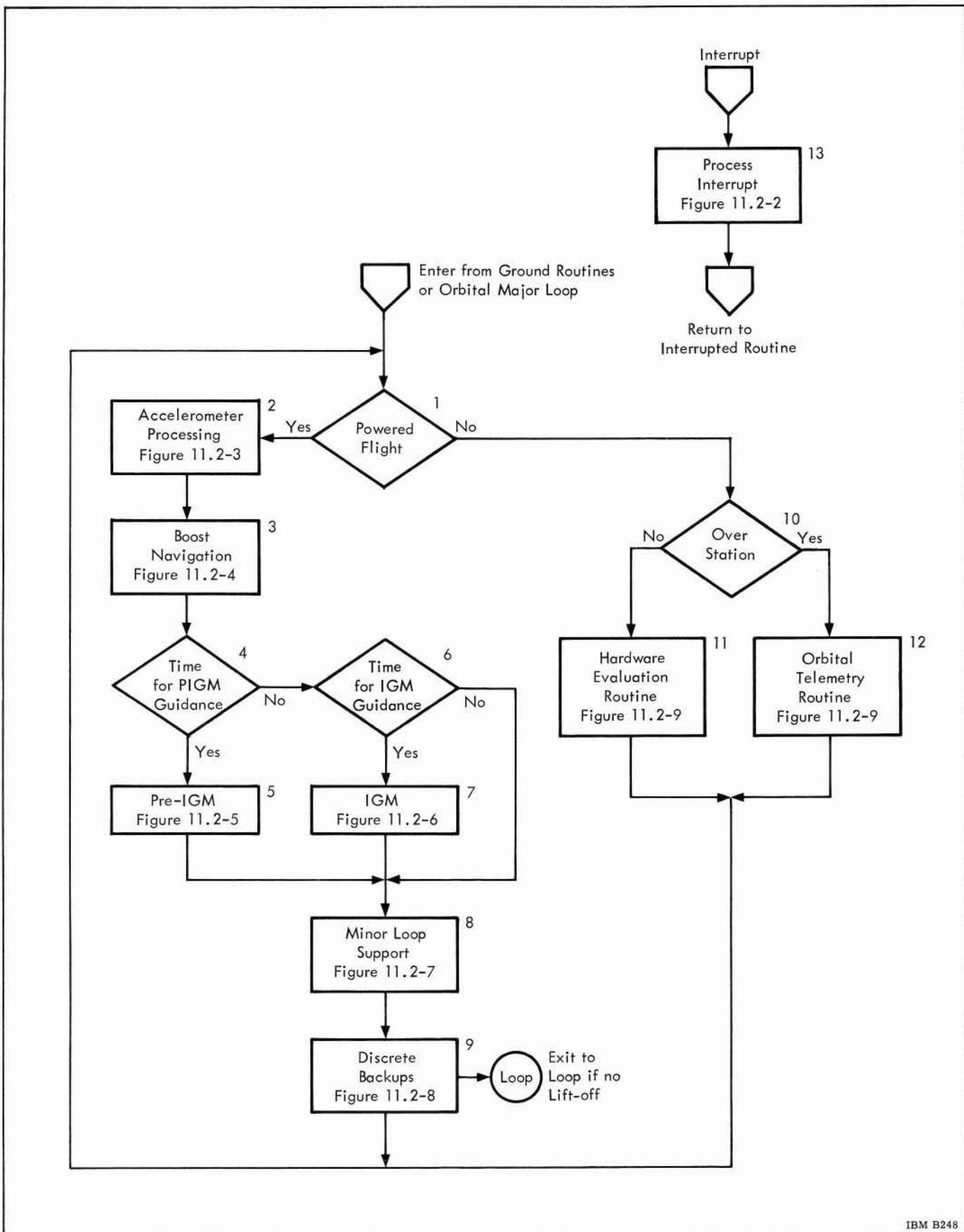
Figure 11.2-1 Major Loop
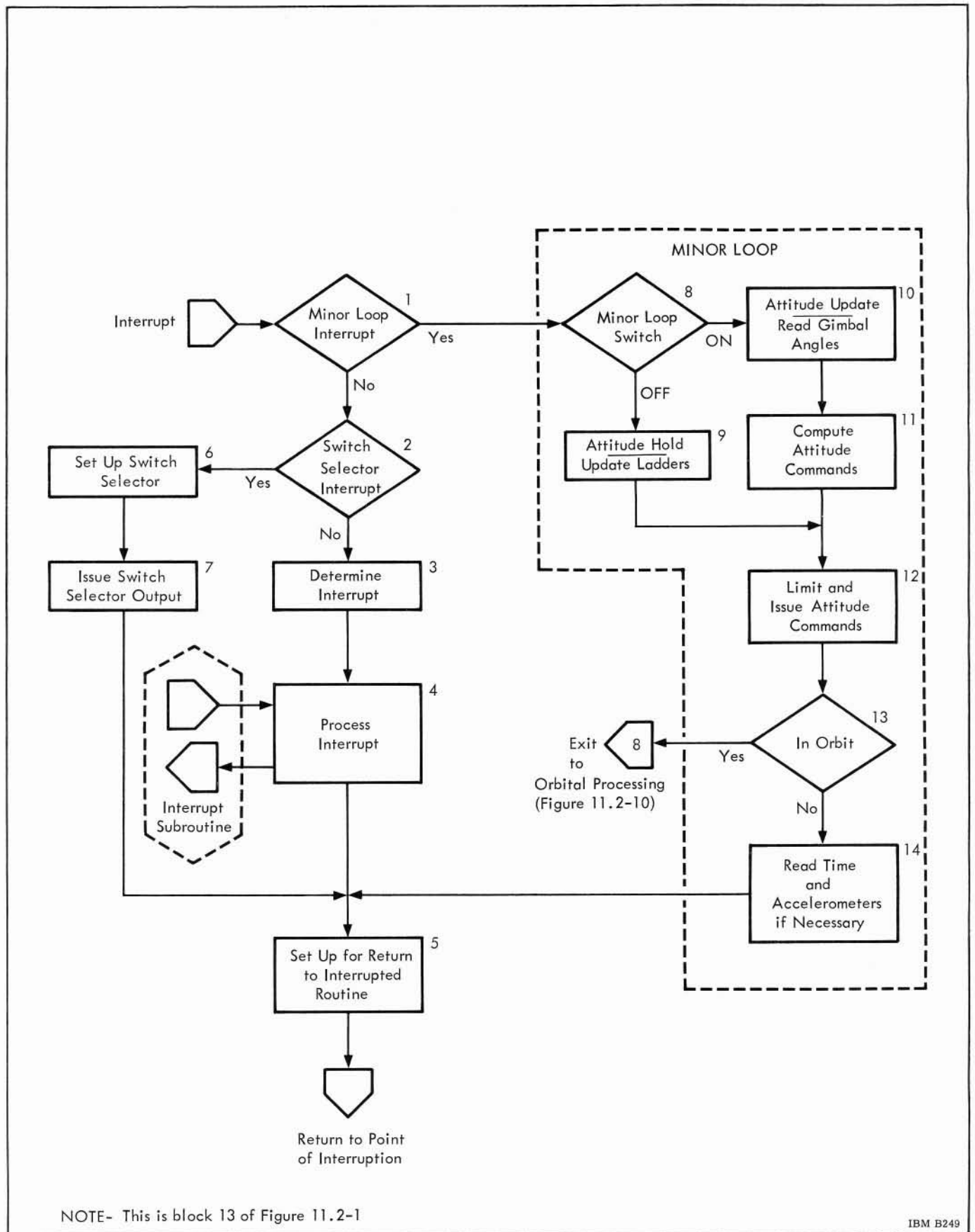
Figure 11.2-2 Interrupt and Minor Loop Processing

vehicle. Most of the values are computed in the injection coordinate system and are later transformed into the space-fixed coordinate system.

IGM commences after second-stage ignition and continues until the S-IVB engine is cutoff. Cutoff occurs when the desired velocity has been reached.

IGM is started approximately 30 seconds after S-II ignition. During S-II burn, one or more burn modes (engine mixture ratios) may be programmed. At 350 seconds after S-II ignition, S-II thrust will be terminated and stage separation occurs. The S-IVB Stage is then ignited and burns for approximately 180 seconds before injecting the vehicle into a parking orbit.

MINOR LOOP SUPPORT

Certain processing is required to support the minor loop. The first task performed within this support routine (Figure 11.2-7) is to check the X Y Z counters which record Platform resolver failures (64:1 or 2:1). The output of these resolvers are used as inputs to the minor loop. The 64:1 resolvers are tested to determine whether the backup resolver should be used instead of the normal resolver. The 2:1 counters are tested to determine if a guidance failure has occurred. If excessive Disagreement Bit (DB) failures are detected, certain counters in the disagreement bit processing routine are not reset.

Values of $D\chi$ are computed in block 6 of this routine and will be used in limiting certain attitude rates (guidance commands) during powered flight. The value of $D\chi$ is a computed increment of the latest guidance command (χ) which is gradually added to the last guidance command. This scheme provides a smooth transition from the old command to the new one. During coast flight, the smooth transition is not required, and the entire guidance command is issued when it is first computed.

The (χ) rate of change is limited to values appropriate to the particular flight mode. When this limiting occurs, the Steering Misalignment Correction (SMC) computations are inhibited.

The remainder of the routine is used to compute sine-cosine coefficients needed in attitude control equations. A check of the channel A ladder output register failures is also made, and if it fails more than three times in one major loop cycle, it is abandoned in favor of channel B.

DISCRETE BACKUPS

Certain discrete, onboard events are particularly important to the flight program since they reset the computer time base which is the reference for all sequential events. For Saturn V Vehicles, these significant events (time bases) are:

T_1	Lift-off (LO)	
T_2	S-IC Inboard Engine Cutoff (IECO)	
T_3	S-IC Outboard Engine Cutoff (OECO)	
T_4	S-II Cutoff	
T_5	S-IVB Cutoff (Boost Phase)	
T_6	S-IVB Restart	
T_7	S-IVB Cutoff (Orbital Phase)	

Since Switch Selector outputs are a function of time (relative to one of the time bases), no Switch Selector output could be generated if one of the discrete signals were missed. A backup routine is provided to circumvent such a catastrophic failure. Simply, the discrete backup routine (Figure 11.2-8) will simulate these critical signals if they do not occur when they are expected. No onboard backup for the initiation of time base six (T_6) is included in the flight program at this time.

In the cases of the backup routines for LO and IECO, special routines are established as a double safety check. In both cases, motion as well as time are confirmed before a backup discrete is issued. For LO, the backup routine is entered 10.5 seconds after GRR. If the X_S acceleration exceeds 2 m/s^2 for 6 computation cycles, the vehicle is assumed to be airborne, and the lift-off discrete is issued. For IECO, an additional safety feature is built into the routine to ensure that an on-the-pad firing of the S-II Stage cannot occur if T_1 is accidentally set. Before T_2 can be initiated, velocity along the Z_S coordinate is tested for a minimum of 500 m/s.

ORBITAL FLIGHT PROGRAM

The orbital flight program (shown in Figure 11.2-9) consists of an executive routine concerned with IU equipment evaluation during orbit and a telemetry routine which is employed while the vehicle is over or within range of a receiving station. The orbital navigation routine computes and pinpoints the access areas.

Both real-time and stored (past-history) data are transmitted on a time-shared basis. The stored data is "compressed" before being stored in the LVDC because of limited core capacity. Four types of data will be compressed during a typical flight: (1) LVDC inputs, (2) LVDC internal data, (3) LVDC outputs, and (4) PCM data read into the LVDC through the Computer Interface Unit.

ORBITAL PROCESSING

The Orbital Processing Routine (OPR) is shown in Figure 11.2-10. The routine is actually an extension of the minor loop interrupt but is only used during the orbital phase of the mission.

Navigation, guidance, and attitude control functions are carried out on a regular basis under control of the minor loop interrupt. The OPR thus serves as an executive program which cycles various routines on a timed basis. Among these routines are:

● Orbital Guidance (1 per second)

● Orbital Navigation (1 per 4 seconds)

● Read gimbal angles (1 per 8 seconds)

Entry from:
• Ground Routines
• Discrete Backups (Figure 11.2-8)
• Orbital Flight Program (Figure 11.2-9)

1

Powered Flight — No → 10 — Exit to Orbital Flight Program (Figure 11.2-9)

Yes

2 — Compute Time and Reasonableness Test Values

3 — Obtain Platform Velocity Word A

4 — Is Change A Reasonable — No → 5 — Obtain Platform Velocity Word B

Yes

6 — Is Change B Reasonable

Yes

No

7 — Form Reasonable Velocity Change

8 — Construct X Velocity Word

9 — Repeat Blocks 3 to 8 for Y and Z Velocity Words

10 — Compute F/M — No → 11 — Compute F/M from Stored Values

Yes

13 — Compute F/M from Platform Values

14 — Compute Smoothed M/F

15 — Set RTC = 5.0

12 — Set RTC = 30

2

Exit to Boost Navigation Routine (Figure 11.2-4)

NOTE - This is block 2 of Figure 11.2-1

IBM B250

Figure 11.2-3 Accelerometer Processing
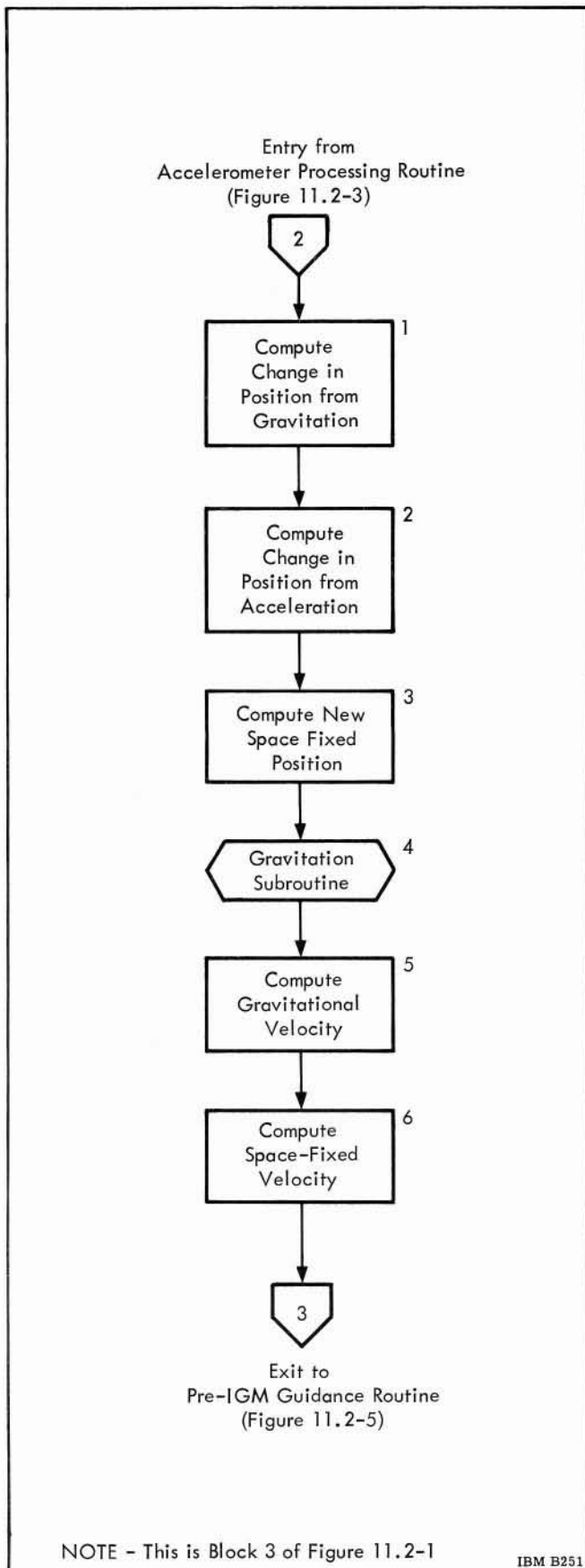
Figure 11.2-4 Boost Navigation

Entry from
Accelerometer Processing Routine
(Figure 11.2-3)

2

Compute
Change in
Position from
Gravitation — 1

Compute
Change in
Position from
Acceleration — 2

Compute New
Space Fixed
Position — 3

Gravitation
Subroutine — 4

Compute
Gravitational
Velocity — 5

Compute
Space-Fixed
Velocity — 6

3

Exit to
Pre-IGM Guidance Routine
(Figure 11.2-5)

NOTE - This is Block 3 of Figure 11.2-1

IBM B251

Figure 11.2-5 Pre-IGM Guidance

Entry from
Boost Navigation Routine

Exit to IGM
Routine
(Figure 11.2-6)

3

4

Time
for IGM — 1

Yes

No

Freeze
Guidance
Angles — 8

Yes

Tilt
Freeze — 2

No

Adjust
Time
Value — 3

Begin
Pitch — 4

Yes

Command
Roll & Yaw
to Zero — 6

No

Command
Initial
Guidance
Angles — 5

Calculate
Pitch from
Time Polynomials — 7

5

Exit to
Minor Loop Support Routine
(Figure 11.2-7)

NOTE - This is Block 5 of Figure 11.2-1

IBM B252

Figure 11.2-6 IGM Flow Diagram

Entry from:
• Pre-IGM Guidance Routine (Figure 11.2-5)
• IGM Routine (Figure 11.2-6)

```
                              ┌───┐
                              │ 5 │
                              └─┬─┘
                                │
                                ▼
┌──────────────────┐  Yes   ◇─────────◇   No   ┌──────────────────┐
│  Set Guidance   2│◄───────│  Excess  1│──────►│ Reset Resolver  3│
│Failure Disconnect│        │Reasonableness│    │   Counters       │
│  (Backup Mode    │        │  Failures  │      │                  │
│    Only)         │        ◇─────────◇         │                  │
└────────┬─────────┘                            └─────────┬────────┘
         │                                                │
         └──────────────────────┬─────────────────────────┘
                                 ▼
                           ◇─────────◇   No   ┌──────────────────┐
                           │ Excess DB 4│─────►│  Reset DB Failure 5│
                           │ Failure  │       │    Counter       │
                           ◇─────────◇        └────────┬─────────┘
                                │ Yes                  │
                                ◄─────────────────────┘
                                ▼
                           ┌──────────┐
                           │ Compute  6│
                           │ Dχ Values│
                           └────┬─────┘
                                ▼
                        ◇──────────◇
                   No   │ Limit Dχ  7│
              ◄─────────│           │
              │         ◇──────────◇
              │              │ Yes
              │              ▼
              │         ┌──────────┐
              │         │Limit Dχ and 8│
              │         │Inhibit SMC │
              │         │Computation│
              │         └────┬─────┘
              └──────────────┤
                             ▼
                        ┌──────────┐        ┌──────────────────┐
                        │Compute Trig 9│────►│ Switch to Ladder 10│
                        │Functions for│     │  B if Necessary  │
                        │ Minor Loop │      └────────┬─────────┘
                        └──────────┘                │
                                                    ▼
                                                 ┌───┐
                                                 │ 6 │
                                                 └───┘
                                            Exit to
                                      Discrete Backups Routine
                                         (Figure 11.2-8)
```

NOTE - This is block 8 of Figure 11.2-1

IBM B254

Figure 11.2-7 Minor Loop Support

Figure 11.2-8 Discrete Backups

Figure 11. 2-9 Orbital Flight Program

NOTE - This is blocks 10, 11, & 12 of Figure 11.2-1

IBM B256

Entry from
Minor Loop
(Figure 11.2-2)

8

Read and
Process
Real Time ‖1

Time
for Gimbal
Read ‖2 → Yes → Read Gimbal
Angles ‖3

No

Time
for Orbital
Navigation ‖4 ← No

Yes → Restore Normal
Inhibits ‖6

Time
for Orbital
Guidance ‖5 → No → Return to Point
of
Interruption

Yes

Orbital Navigation
Routines** ‖7

Orbital Guidance
and Restart
Routines* ‖8 → 9 Exit to
IGM Routine
(Figure 11.2-6)

* Routines include:
  • Predict positions and velocities
  • Compute gravitation
  • Compute drag
  • Compute vent thrusts
  • Accumulate position, velocity, & acceleration

** Routines include:
  • Interpolate $X_S Y_S Z_S$
  • Calculate $X_4 Z_4$
  • Compute target & perigee vectors
  • Issue χ commands
  • Prepare for restart of S-IVB
  • Calculate restart conditions

IBM B257

Figure 11.2-10 Orbital Processing

SECTION 13.1
THERMAL CONDITIONING SYSTEM

A thermal conditioning system is provided in the S-IVB Stage and Instrument Unit of the Saturn IB and V Launch Vehicles. This system is required since cooling of the Astrionics equipment by radiation is inadequate. The thermal conditioning system furnishes cooling during prelaunch and flight.

The thermal conditioning system contains 16 thermal conditioning panels (cold plates) in the S-IVB Stage and 16 in the Instrument Unit. Most of the equipment which requires cooling is mounted on the cold plates; however, some equipment such as the Launch Vehicle Digital Computer and Data Adapter have integral cooling passages and are not mounted on cold plates. Each cold plate measures 76.2 centimeters by 76.2 centimeters (30 inches by 30 inches) and contains tapped bolt holes positioned on a 5.1-centimeter (2-inch) square grid. (A cold plate is illustrated in Figure 13.1-1.) This hole pattern permits maximum flexibility of component mounting on the cold plates.

A coolant fluid of 60 percent methanol and 40 percent water by weight is circulated through the cold plates and equipment having integral cooling passages. Each cold plate is capable of dissipating approximately 420 watts.

A diagram of the environmental control system is shown in Figure 13.1-2. The manifold and methanol-water accumulator are filled with coolant through either of the quick disconnects at the S-IVB/IU interface. Also, any air which may be trapped during filling is bled out through these disconnects. The methanol-water accumulator provides damping of pressure fluctuations, a constant pump inlet pressure, compensation for thermal expansion of the coolant, and will supply additional coolant should any be lost.
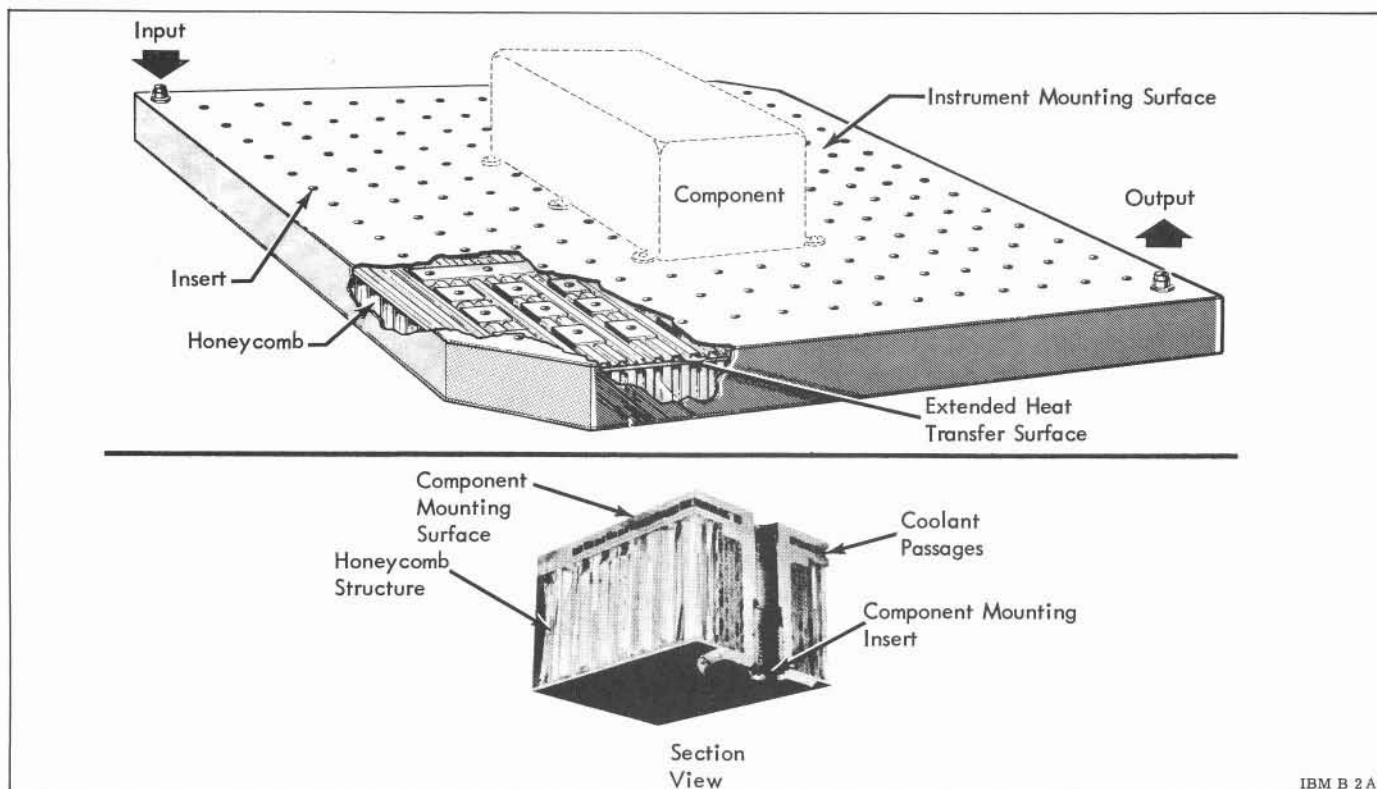
Figure 13.1-1 Thermal Conditioning Panel Details

The methanol-water accumulator and water accumulator are pressurized with gaseous nitrogen (GN_2) that is stored at 20.7×10^6 N/m^2g (3000 psig) in the 2690-cubic centimeter (165-cubic inch) storage sphere. The gaseous nitrogen is filtered by filter A, reduced to approximately 103.5×10^3 N/m^2a (15 psia) by the pressure regulator, and then applied to the methanol-water accumulator. The orifice regulator reduces the 103.5×10^3 N/m^2a (15 psia) gaseous nitrogen to 34.5×10^3 N/m^2a (5 psia) and this pressure is used to pressurize the water accumulator. The orifice regulator also vents excess pressure into the Instrument Unit compartment to prevent over-pressurizing the water accumulator. A flexible membrane is contained within both accumulators and provides a barrier that prevents mixing of the gaseous nitrogen and liquid.

The bleeder assembly is opened during filling of the accumulators. This provides an escape port for air which could otherwise be trapped and allows the flexible membranes in the accumulators to retract.

Filling of the water accumulator is accomplished manually by pouring water into the accumulator through the water fill port. The flow of water from the water accumulator to the Flight Heat Exchanger (sublimator) is controlled by water solenoid valve C. This valve is energized (closed) during prelaunch operations since cooling of the methanol-water coolant is accomplished by the preflight heat exchanger.

Filling of the 165-cubic inch storage sphere with gaseous nitrogen is controlled by GN_2 solenoid (solenoid valve A). A high pressure source of gaseous nitrogen is applied to the GN_2 vent quick disconnect. A pressure transducer indicates when the sphere is pressurized to the proper value. At this time, solenoid valve A is deenergized (closed). The solenoid valve also provides for the venting of the system during pre-flight operations, if required.

During prelaunch operations, the methanol-water coolant is circulated through a closed-loop system by electrically driven, redundant pumps. The coolant flows through the coolant temperature sensor, orifice assembly, and then through the cold plates, gas bearing heat exchanger, and internally cooled equipment within the Instrument Unit. The orifice assembly diverts a portion of the coolant to the cold plates in the S-IVB Stage. The coolant in the return line either passes through the sublimator and preflight heat exchanger to the pumps or flows through the by-pass line to the pumps, thus completing the closed loop. The orifices shown in the LVDA, LVDC, and Flight Control Computer lines maintain the proper coolant flowrate through their respective components.

The coolant temperature sensor, located at the pump outlet, senses the coolant temperature and supplies an electrical signal to the electronic control assembly which controls the modulating flow control valve. The modulating flow control valve diverts varying amounts of coolant through the sublimator and preflight heat exchanger if the coolant requires cooling. If the coolant requires no cooling, an appropriate quantity of coolant is bypassed around the sublimator and heat exchanger.

As previously mentioned, during prelaunch operations the sublimator is inactive and the preflight heat exchanger cools the closed-loop coolant. A temperature controlled methanol-water mixture, supplied by the ground support equipment, is circulated through one section of the preflight heat exchanger, thus cooling the closed-loop coolant that flows through the other section.

During flight, the operation is the same except that the sublimator is active and the preflight heat exchanger is inactive. However, the sublimator does not operate until approximately 150 seconds after lift-off, at which time the ambient pressure is low enough to sustain sublimation. Calculations indicate that during the transient phase, the thermal capacity of the electronic equipment and thermal conditioning system is sufficient to prevent over-heating of the equipment.

When the ambient pressure is low enough for sublimation, water solenoid valve C is de-energized (opens) allowing water to flow from the water accumulator to the sublimator. Water under pressure enters the sublimator and flows into the pores of the porous plate. As the water in the pores encounters the space vacuum at a temperature and pressure below the triple point of water, an ice layer forms at the outer water surface and continues to build-up back towards the water passage. As heat is transferred through the water passage and ice layer, the outer surface sublimes overboard. As the heat level is increased, the ice sublimes at a faster rate, increasing the water flow rate. The converse is also true.

In addition to the previous description of sublimator operation, a theory exists that two modes of heat transfer occur in the sublimator. This theory – sublimation and evaporation – is as follows: Water under pressure enters the sublimator (refer to Figure 13.1-3) and flows into the pores of the porous plates. As the water in the large pores encounters the space vacuum, freezing occurs and an ice plug is formed. In the small pores which are non-wetted and can retain water by surface tension, the water fills the pores to a distance where the restriction is small enough to
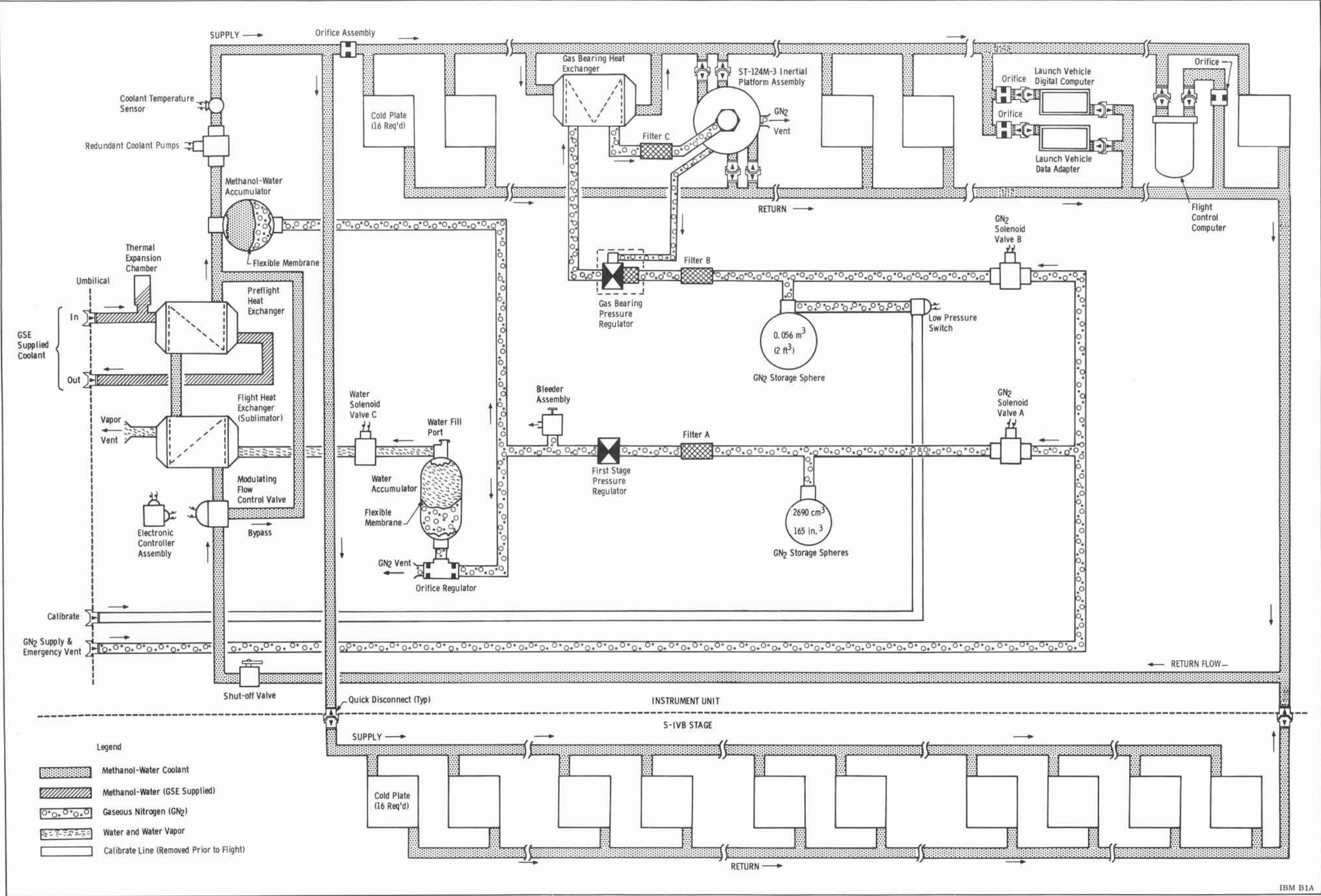
Figure 13. 1-2 Environmental Control System Mechanical Diagram

retain water at the inlet pressure. As the heat flux increases, the ice in the large pores sublimes at the solid-vapor interface and is formed at the liquid-solid interface. The vapor or steam is vented overboard. When the heat flux becomes great enough to prevent all ice from existing, any further increase in the heat load will raise the evaporation temperature. The heat load reaches a maximum when no ice exists and the

heat sink temperature is zero degrees Centigrade (32 degrees Fahrenheit).

As can be seen, the sublimator has an advantage over a heat exchanger in that the flow of water is self-regulating — as the heat flux decreases, ice plugs are formed in the pores and the flow of water also decreases.
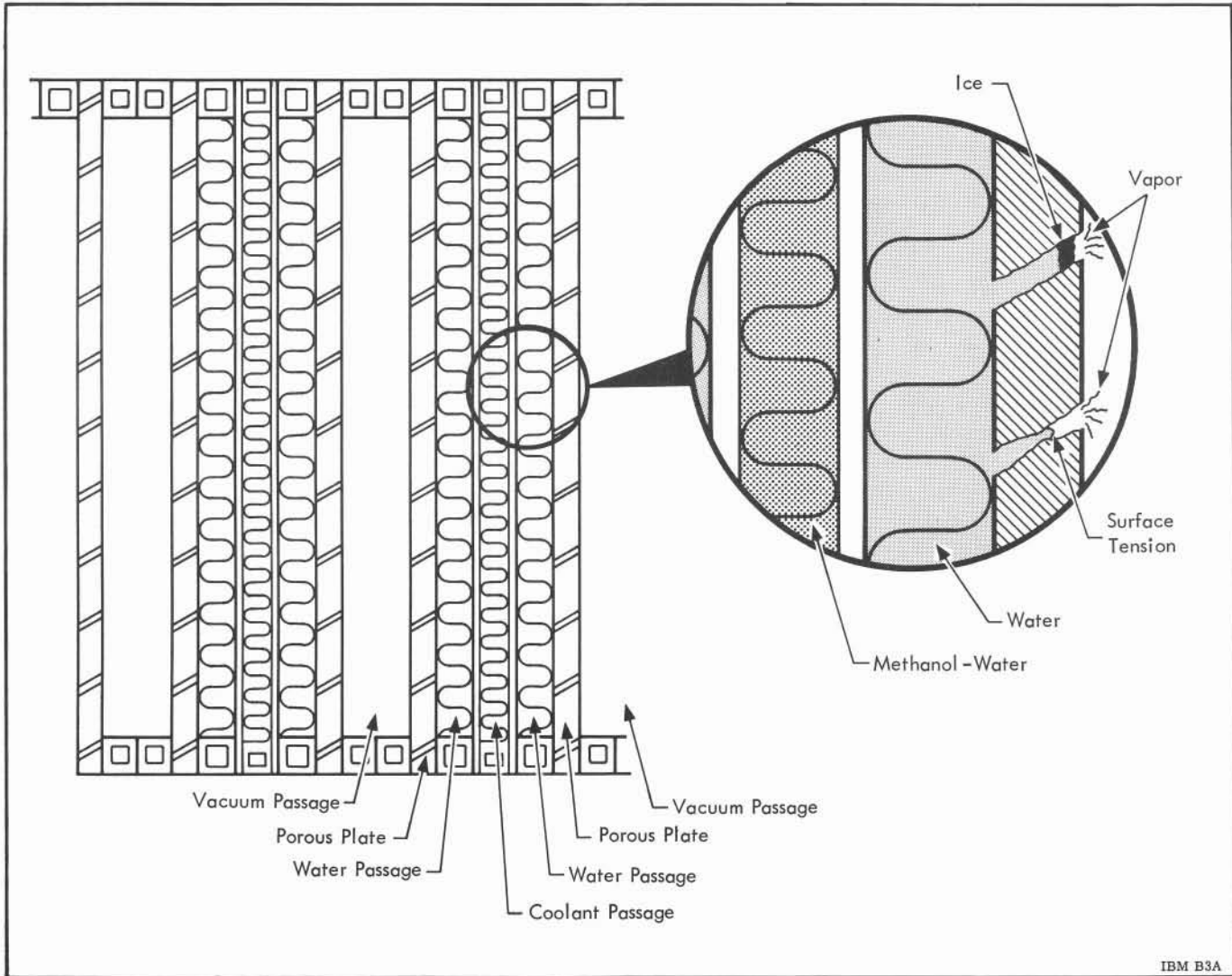
Figure 13.1-3 Sublimator Details

SECTION 13.2
GAS BEARING SUPPLY SYSTEM

The gas bearing supply system furnishes gaseous nitrogen at a regulated temperature and pressure for lubrication of the gas bearings in the ST-124-M Inertial Platform Assembly.

The system is filled through the GN_2 supply and emergency vent quick disconnect (refer to Figure 13.1-2) from a ground supplied source. A pressure transducer indicates when the pressure in the 0.056-cubic meter (2-cubic foot) storage sphere is approximately 20.7×10^6 N/m^2g (3000 psig). At this time, normally closed solenoid valve B is de-energized (closed) and the GN_2 source can be removed.

The gaseous nitrogen flows from the storage sphere, is filtered by filter B, and reduced to a pressure suitable for lubrication of the gas bearings by the pressure regulator and filter. The gaseous nitrogen is cooled or heated as required as it passes through the gas bearing heat exchanger, is filtered by filter C, and then applied to the gas bearings in the Platform. After lubricating the gas bearings, the gaseous nitrogen is vented into the Instrument Unit compartment.

A reference pressure is fed-back from the Platform to the pressure regulator and filter. This feed-back reference pressure maintains the pressure across the gas bearings constant by controlling the pressure regulator and filter output pressure; i.e., a decrease in pressure across the gas bearings causes the pressure regulator and filter output to increase.

The low pressure switch, which is calibrated through the calibrate quick disconnect, indicates when the pressure in the 0.056 cubic meter (2-cubic foot) sphere decreases below approximately 6.9×10^6 N/m^2g (1000 psig). Should an indication of a pressure decrease occur during prelaunch operations, the Platform would be shutdown until the 0.056-cubic meter sphere could be repressurized. Under normal in-flight conditions, this should not happen unless a leak or malfunction is present in the system.

PREFLIGHT AIR/GN₂ PURGE SYSTEM

The Preflight Air/GN_2 Purge System (Figure 13.2-1) consists primarily of flexible ducting located above the Instrument Unit payload interface. The system distributes temperature and pressure regulated, ground supplied filtered air or GN_2 through openings in the ducting. During preflight phases, ventilating air is furnished. During fueling, inert GN_2 is furnished to prevent accumulation of a hazardous and corrosive atmosphere.
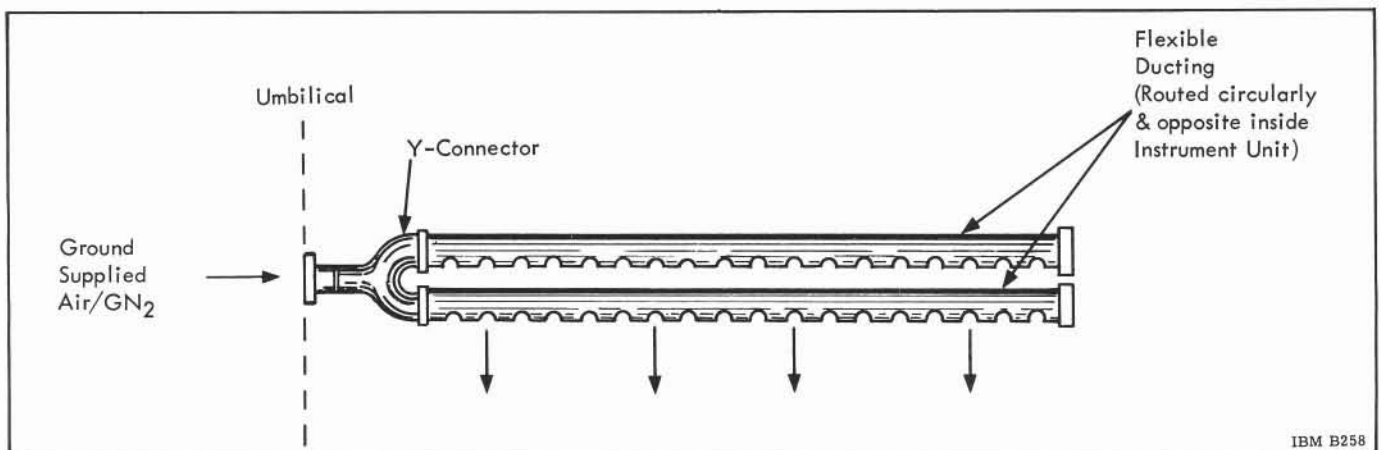
Figure 13.2-1 Preflight Air/GN₂ Purge System

SECTION 14.1
INTRODUCTION

The ST-124-M Inertial Platform system [1] (see Figures 14.1-1 and 14.1-2) provides the inertial reference coordinates, integrated acceleration data, and vehicle attitude measurements with respect to these coordinates for navigation, guidance, and control of the Saturn Launch Vehicles.

The Platform system consists of 6 assemblies:

- ST-124-M Inertial Platform Assembly [2]

- ST-124-M Platform Electronic Assembly

- Accelerometer Signal Conditioner

- ST-124-M Platform AC Power Supply

- 56 Volt Power Supply

- Gas Bearing Supply

The Platform system is self-contained and requires only inputs from the environmental conditioning system and the 28-volt dc power source.

The interface between the Platform system and other components of the Saturn navigation, guidance, and control system is shown in Figure 14.1-3. The accelerometer output signals are shaped by the

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[1] For the purpose of brevity, the ST-124-M Inertial Platform system will hereinafter (within Chapter 14) be referred to as the Platform system.

[2] For the purpose of brevity, the ST-124-M Inertial Platform Assembly will hereinafter (within Chapter 14) be referred to as the Inertial Platform

ST-124-M PLATFORM
AC POWER SUPPLY

ST-124-M PLATFORM
ELECTRONIC ASSEMBLY

ST-124-M INERTIAL
PLATFORM ASSEMBLY

ACCELEROMETER
SIGNAL CONDITIONER

SYSTEM FUNCTIONS

- Accelerometer sensing and reference.

- Vehicle attitude and programming.

- Guidance reference coordinates.

IBM B136A

Figure 14.1-1 ST-124-M Inertial Platform System (Saturn IB and V)

IBM B137
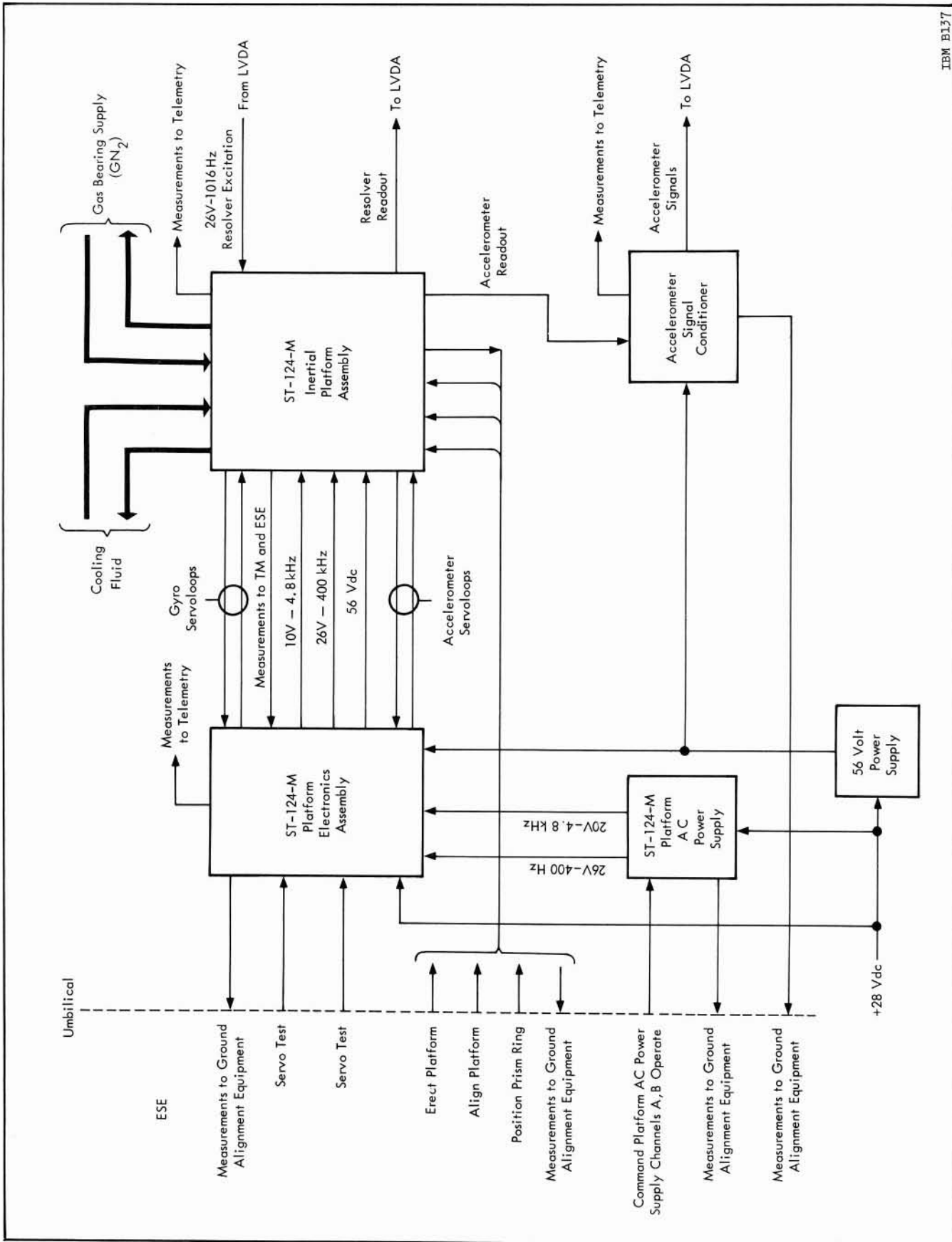
Figure 14.1-2 Inertial Platform Subsystem Block Diagram

Accelerometer Signal Conditioner and then converted into a digital code by the accelerometer signal processor in the LVDA. The signals from the gimbal resolvers are converted to binary numbers in the crossover detector in the LVDA. The resolver chain output is used for prelaunch checkout only.

The Inertial Platform servo amplifier contains the solid state electronics to close the platform gimbal servoloops, the accelerometer servoloops, and an impulse function generator for automatic checkout of each servoloop.

ESE associated with the Platform system is located at the launch site and the launch control center. At the launch site, the azimuth-laying theodolite is located approximately 700 feet from the base of the vehicle. The theodolite installation transmits signals

via underground cables to the platform ESE located in the launch umbilical tower. Platform system ESE in the launch umbilical tower consists of an inertial data box, an alignment amplifier panel, an azimuth control panel, a theodolite amplifier panel, and accelerometer monitor panels. The equipment contained in the launch umbilical tower interfaces with the onboard Platform system, the RCA-110A launch umbilical tower computer, and the data link. In the launch control center, additional Inertial Platform ESE interfaces with the launch control center RCA-110A Computer and data link. This equipment consists of a platform checkout and monitor panel, a platform control panel, an azumuth laying panel, and an azimuth-laying, video-monitor panel. Total remote control and automation of the Platform system is accomplished from the launch control center by use of the Platform system ESE, the launch control computer, and the data link.
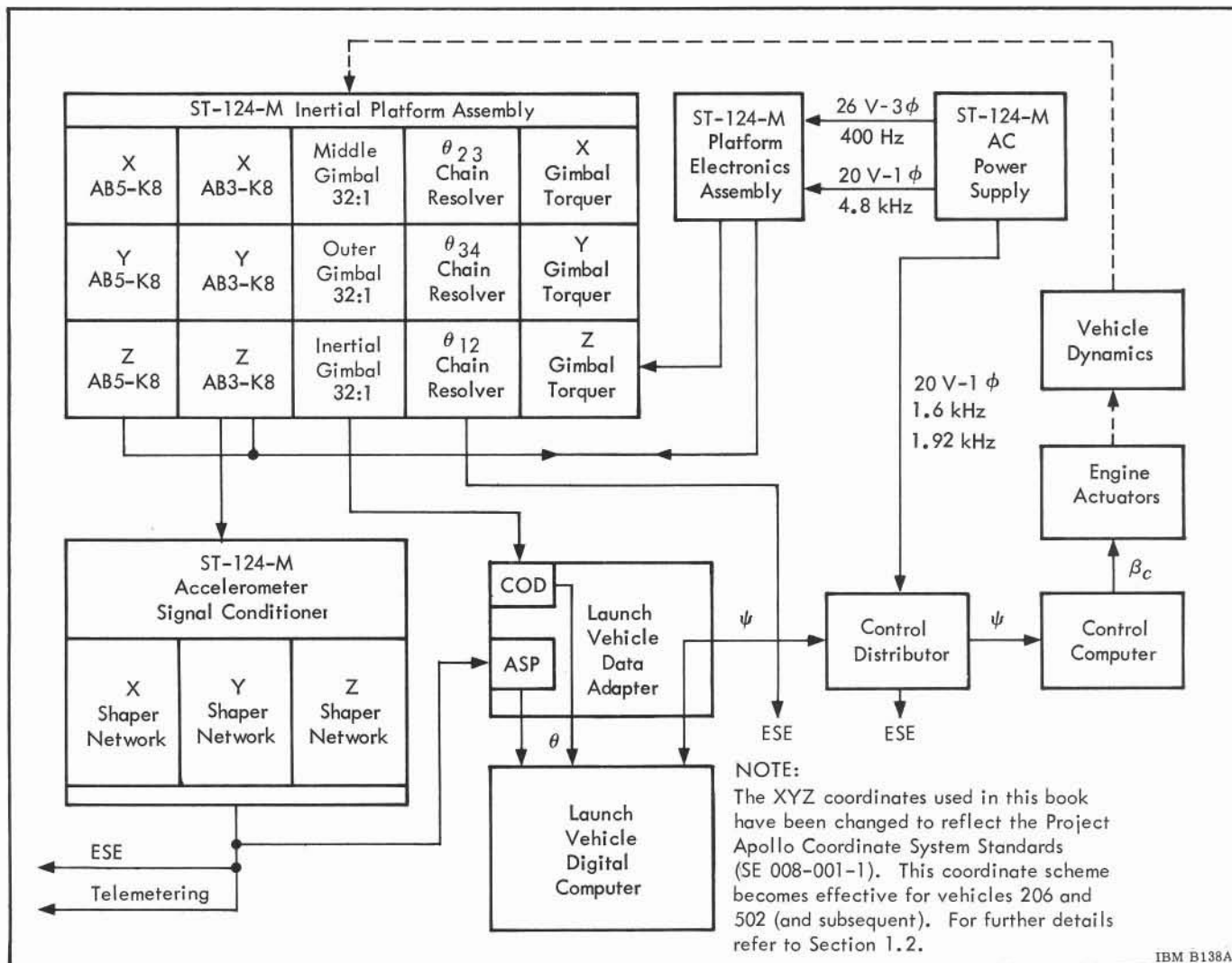
Figure 14.1-3 Inertial Platform System Signal Interface

SECTION 14.2

ST-124-M INERTIAL PLATFORM ASSEMBLY

The Inertial Platform Assembly, used in Saturn IB and V Vehicles, can take either of two configurations. These two configurations are designated ST-124-M3 and ST-124-M4 and are three- and four-gimbal systems, respectively. In the basic design of the Inertial Platform, the outer, middle, and inertial gimbals are identical in both models. The ST-124-M4 has a fourth, or redundant, gimbal and a larger base structure; it also requires a different mounting frame. The IU intercabling and wiring for these two systems are identical.

The gimbal configuration of the ST-124-M4 offers unlimited freedom about all three inertial reference axes, while the ST-124-M3 is limited to ± 45 degrees about its Z axis* (vehicle yaw at launch). For the Apollo mission, the ST-124-M3 configuration is used (Figure 14.2-1).

The inertial, or inner, gimbal provides a rotationally-stabilized table upon which three integrating accelerometers are mounted. The gimbal system allows the inertial gimbal rotational freedom. Three single-degree-of-freedom gyroscopes provide the reference for the stabilized table. The gyros have their input axes aligned along an orthogonal inertial coordinate system X_I, Y_I, and Z_I of the inertial gimbal as shown in Figure 14.2-2. The signal generator, which is fixed to the output axis of the gyro, generates electrical signals proportional to torque disturbances. These signals are transmitted to the servoelectronics which terminate in the gimbal pivot servotorque motors. The servoloops maintain the inner gimbal rotationally-fixed in inertial space.

The inner gimbal has three pendulous integrating gyro accelerometers oriented along the inertial coordinates X_I, Y_I, and Z_I (Figure 14.2-3). Each accelerometer measuring head contains a pendulous single-degree-of-freedom gyro; the rotation of the

measuring head is a measure of integrated acceleration along the input axis of the accelerometer. The Y accelerometer measures acceleration perpendicular to the thrust vector (crossrange acceleration). The outputs of the X and Z accelerometers are used to compute the pitch angle of the vehicle acceleration vector and the required cutoff velocity.

The coordinate erection and orientation system of the Inertial Platform aligns the X_I coordinate along the launch local vertical parallel to the gravity vector and aligns the Y_I coordinate perpendicular to the desired flight plane. The coordinate orientation is accomplished by use of an automatic alignment system which is a part of the ESE on the ground. The leveling alignment servoloops (Y and Z) generate torquing voltages which are a function of the earth's angular velocity, the latitude of the launch site, and the azimuth of the Z_I coordinate (launch azimuth). These voltages are applied to the Y and Z gyro torquers and cause the gyros to torque its respective gimbal until the servos are nulled. Inertial Platform coordinates are established in azimuth with the roll alignment servoloop. The output of the roll alignment servoloop is a function of the earth's angular velocity and the latitude of the launch site. Azimuth alignment being a commanded function, is accomplished by injecting azimuth alignment signals directly into the roll alignment loop and torquing the roll gimbal with the X gyro. The azimuth alignment position can be updated at any time by the launch control computer.

The reference coordinate frame of the Inertial Platform maintains a fixed relationship to the inner gimbal. This coordinate frame is accurately positioned prior to vehicle lift-off and remains space-direction fixed after vehicle lift-off. The gimbals of the Inertial Platform allow the inner gimbal rotational freedom with respect to the vehicle. The gimbal pivots are the axes about which the gimbals turn with respect to each other. The $\pm$ Y pivots couple the inner (stabilized) gimbal to the middle gimbal; the Y pivot axis is normally along the vehicle pitch axis. The $\pm$ Z pivots couple the middle gimbal to the outer gimbal; the Z pivot axis is normally along the vehicle yaw axis. The $\pm$ X pivots couple the outer gimbal to the platform

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* The XYZ coordinates used in this book have been changed to reflect the Project Apollo Coordinate System Standards (SE 008-001-1). This coordinate scheme becomes effective for vehicles 206 and 502 (and subsequent). For further details refer to Section 1.2.

mounting base; the X pivot axis is along the vehicle roll axis. The Inertial Platform pivot scheme is shown in Figure 14.2-4. The X, Y, and Z pivots are controlled by dc torque motors; the X pivot torquer has approximately twice the torque capacity of the other pivot torquers. This accommodates the reflected torque when the middle gimbal is not orthogonal with the outer gimbal. For transfer of electrical signals across the ± X and ± Y pivots, slip-ring capsules are used. A flex cable is used on the ± Z pivot because of its limited freedom.

Dual-speed resolvers (32:1) are used as shaft angle encoders on the +X, +Y, and +Z pivots. The outputs of these resolvers are converted into a digital code in the LVDA. The system also has an analog resolver chain which utilizes single-speed resolvers on each pivot and three program resolvers in the
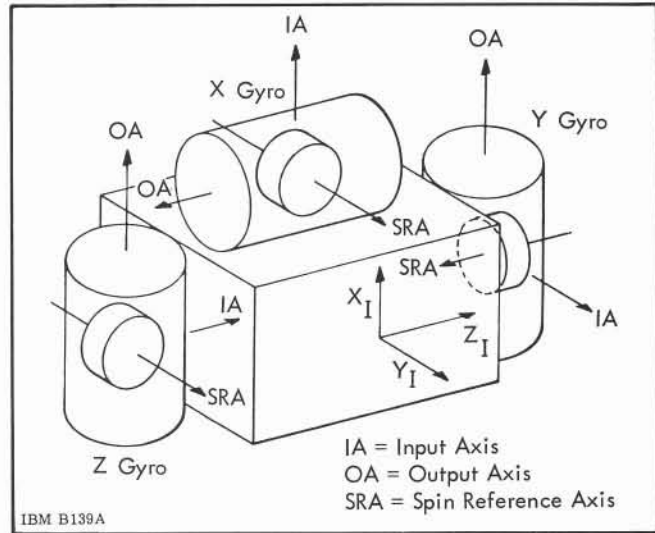
Figure 14.2-2 Orientation of Gyro Axes
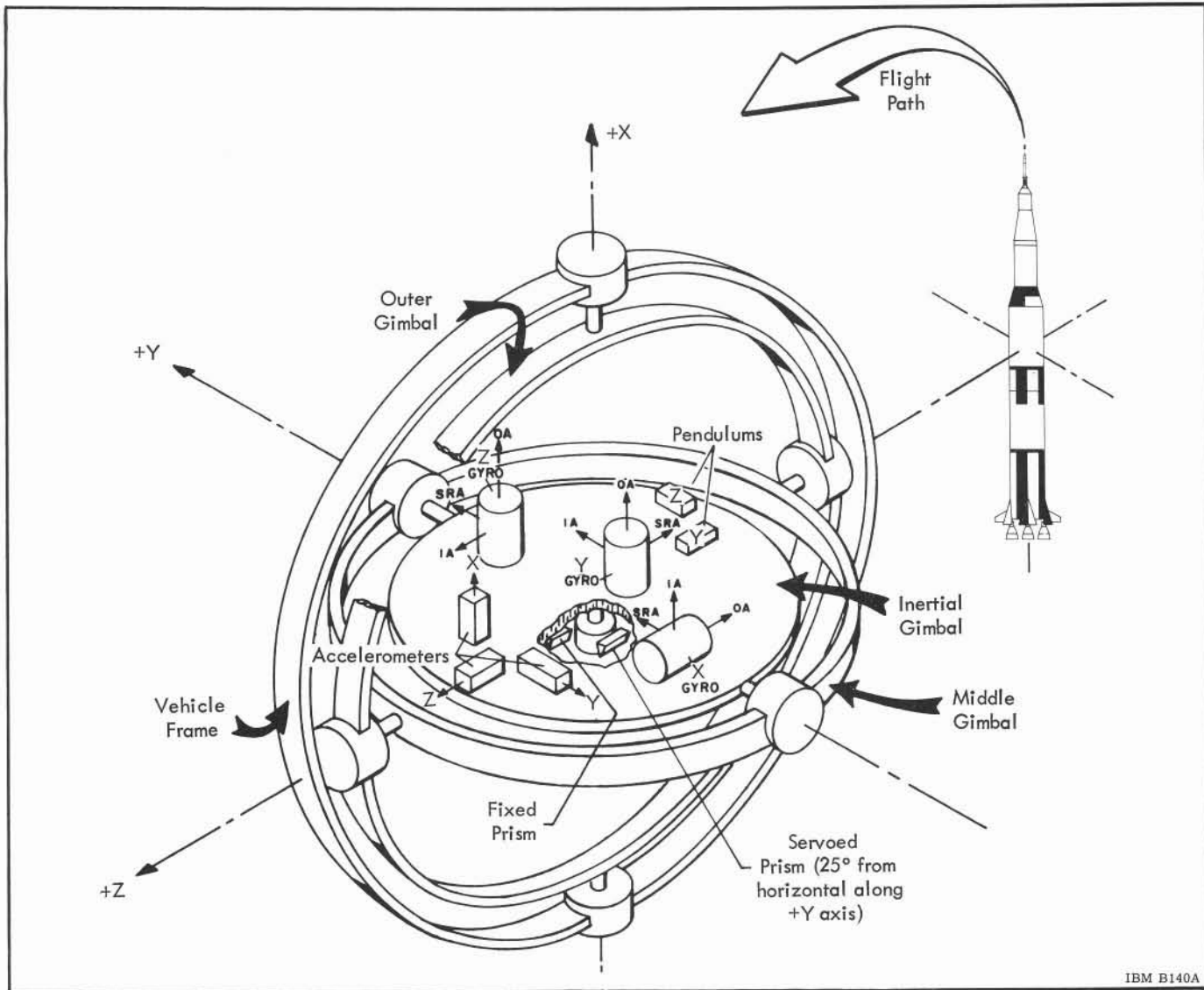
Figure 14.2-1 ST-124-M Gimbal Configuration
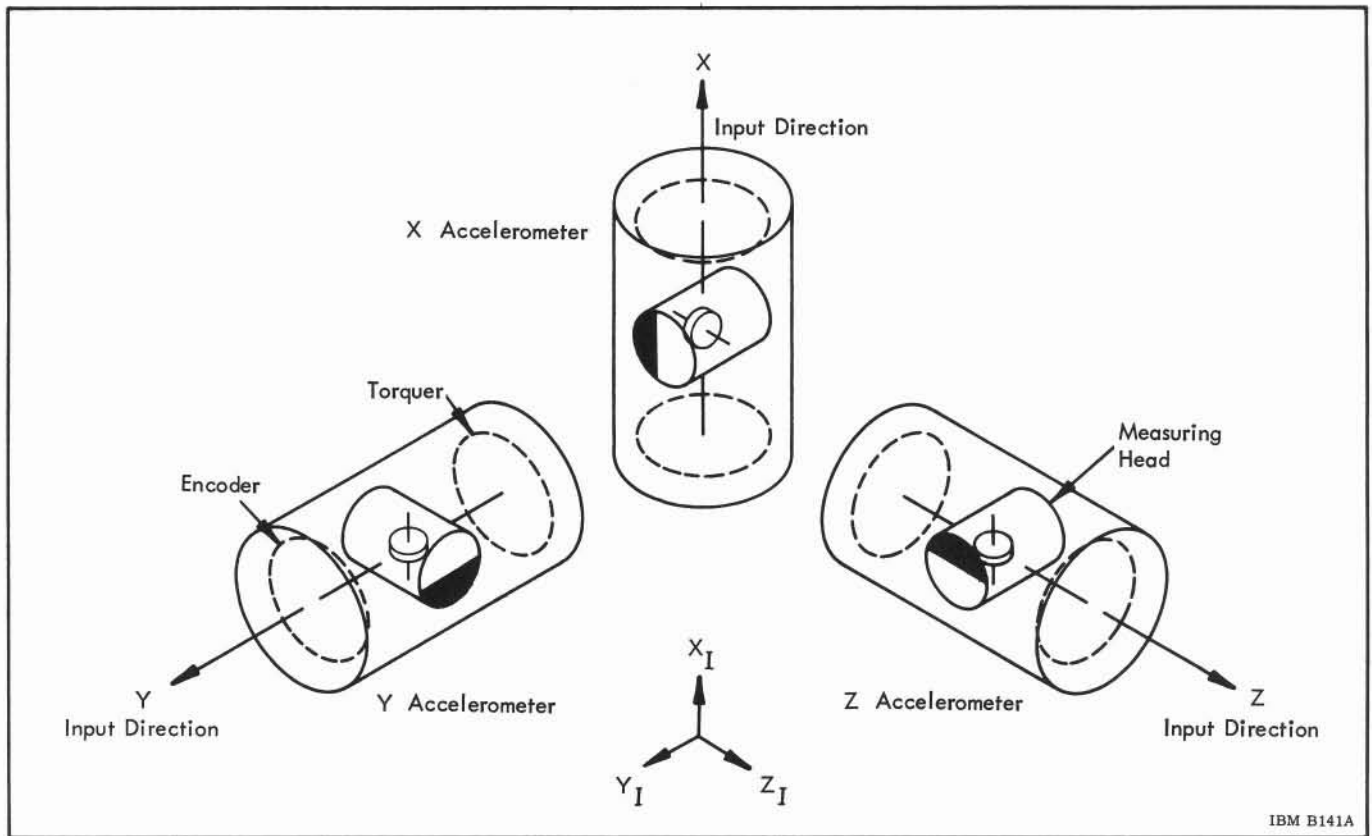
Figure 14. 2-3 Orientation of Accelerometer Axes
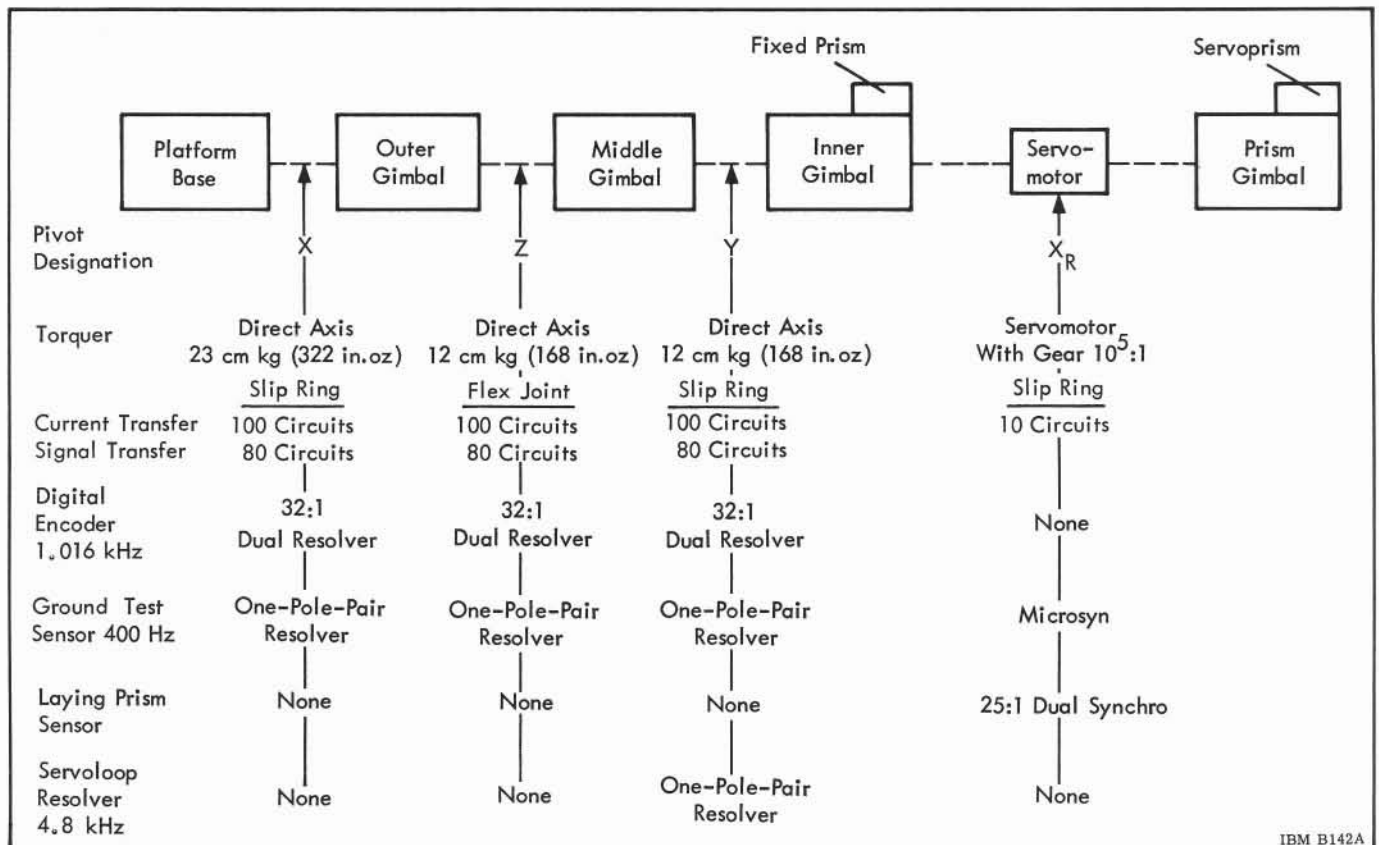
Figure 14. 2-4 Inertial Platform Gimbal Pivot Scheme

inertial data box. This is used with the ground electrical support equipment only to facilitate checkout and test. If required, the resolver chain system can be used as flight equipment to measure the gimbal angles and to provide the vehicle attitude signals.

The + X and +Y pivots each contain a 100-circuit slip-ring cartridge for passing electrical signals across the pivots. The -X and -Y pivots each contain an 80-circuit slip-ring cartridge. All critical signals are routed through the plus pivots while all power transmission is over the minus pivots. The slip rings are built up by electrodeposition of 0.015 inch of copper base with a flashing of nickel to prevent gold migration and electrodeposition of 0.015 inch of gold on the outside. The rings are dielectrically isolated and supported with a filled epoxy. A one-piece stainless steel cross-spline along the center serves as the main structural member of the ring subassembly (Figure 14.2-5). The leadwire is fastened to the ring by electrodeposition and is routed along the spline to the mounting flange.

A 90-degree V-groove is machined into the rings to serve as a guide for the brushes. Dimensional tolerance on the ring assembly is controlled by a grinding process where the V-groove in all rings are ground in a single operation. A set of miniature precision bearings and the brush blocks' mounting frame complete the ring assembly.
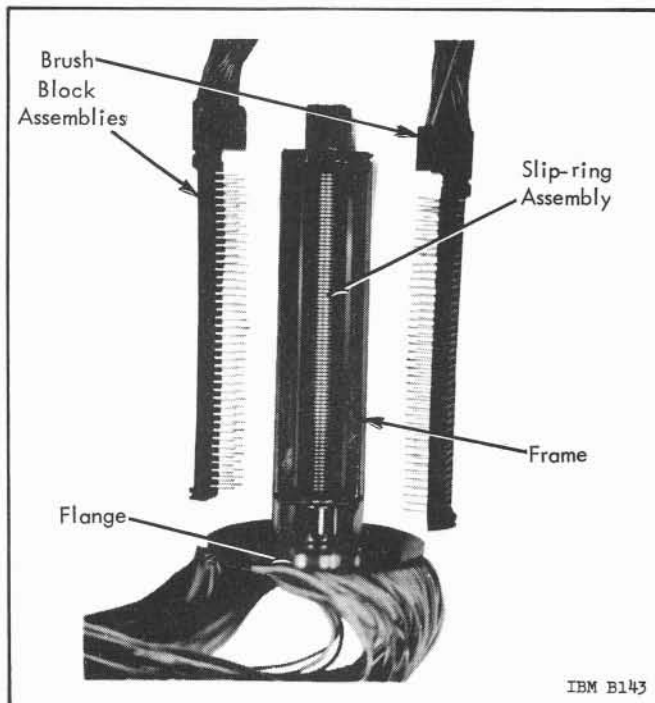
Figure 14.2-5 Slip-ring Capsule

The two brush block subassemblies allow complete redundancy for each brush. The brushes are preformed and prestressed Ney-ORO-28A (75% gold, 22% silver, and 3% nickel) material. The brush blocks are a filled epoxy structure aligned with dowel pins and secured with screws to the frame. The leadwire is an unpigmented teflon-insulated nickel-plated copper, 19/42, size 30 AWG, stranded wire.

Acceptance testing of each unit includes 100-hour run-in time, temperature cycling, noise tests, and an X-ray of the completed assembly.

Table 14.2-1 lists the slip-ring cartridge characteristics.

The X_R prism gimbal pivot is controlled through a gear reduction of 10^5:1 by a servomotor mounted to the inertial gimbal. The angle between the prism gimbal and the inertial gimbal is accurately sensed by a dual synchro (25:1) and controlled with a followup servo. A microsyn is also mounted on the X_R pivot for initial alignment of the prism gimbal to the inertial gimbal and to facilitate laboratory checkout and test.

Table 14.2-1 Slip-ring Cartridge Characteristics

Current Rating	1 A at 125 V, 400 Hz continuous per circuit
Noise Limit	10 uV/mA
Contact Resistance	0.125 ohm per circuit
Breakaway Friction	2 g cm/circuit
Environmental	
Temperature	-55°C to +100°C (67°F to +212°F) storage 0°C to +60°C (32°F to +140°F) operation
Vibration	15 g, 20-2000 Hz 3-hour test time per axis
Shock	30 g, 3 milliseconds 20 g, 11 milliseconds
Acceleration	20 g
Life	
Useful Life	8760 hours (storage and operating)
Operating Life	1000 hours will meet noise specification 5000 hours with maximum of 100% increase in noise specification
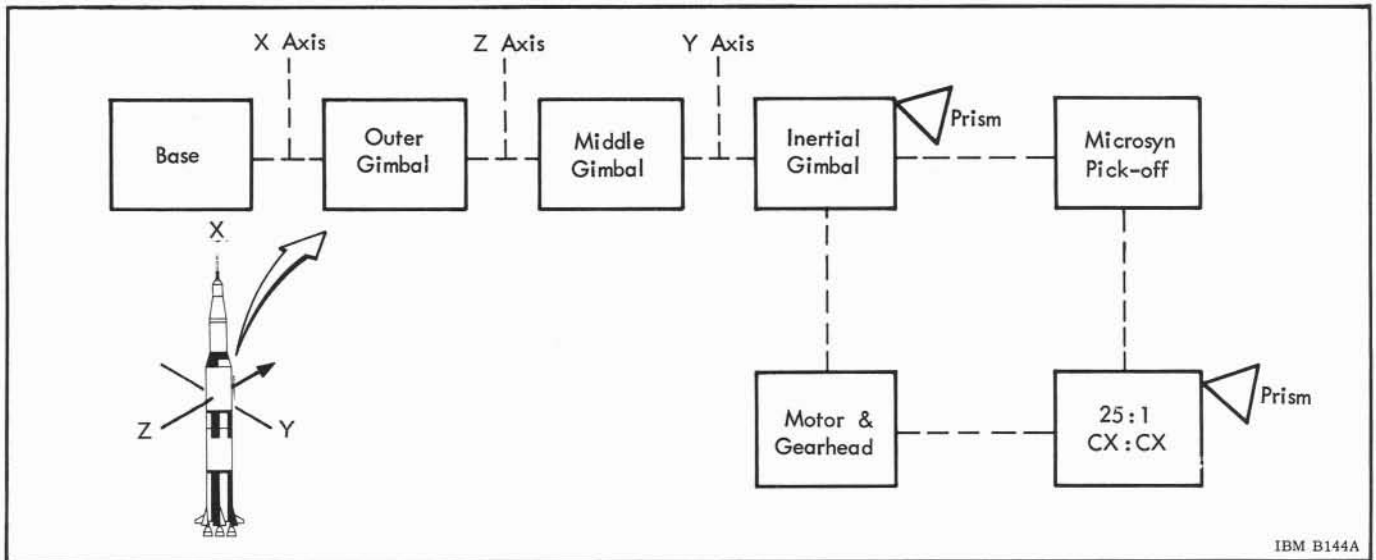
Figure 14.2-6 Platform Gimbal Arrangement

The three-gimbal ST-124-M3 Inertial Platform Assembly, as described, is the normal launch pad configuration and is illustrated in Figure 14.2-6.

The weight of the ST-124-M3 Inertial Platform Assembly is 48 kilograms (107 pounds). The Inertial Platform is mounted in the Saturn IB and V IU at control position IV on accurately machined surfaces which

are qualified to the IU coordinates to ±3 arc minutes. A window in the IU at control position IV provides a line-of-sight to the Inertial Platform azimuth laying prisms for penetration of the azumuth laying theodolite light source (Figure 14.2-7).

The electrical connectors and hemispherical covers of the Inertial Platform are sealed so that ex-
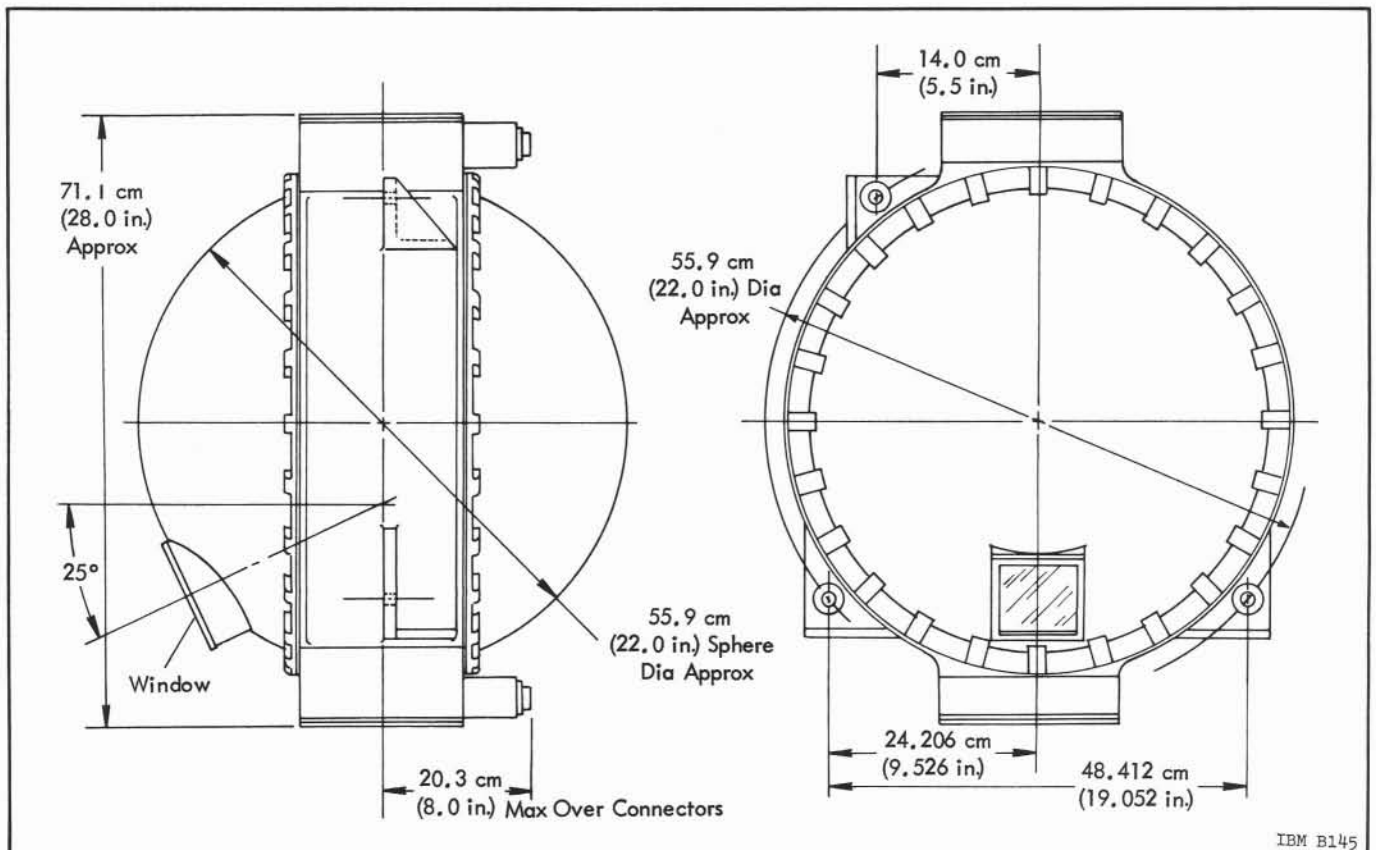
Figure 14.2-7 ST-124-M Inertial Platform Assembly

haust gas from the gas bearing components must pass through a special orifice in its base. This orifice is a pressure-regulating device which stabilizes the internal ambient pressure at 8.3 N/cm^2a (12 psia). Because of the characteristics of this orifice, the internal pressure will rise to approximately 11.75 N/cm^2a (17 psia) in a one atmosphere ambient and will drop to the controlled pressure of 8.3 N/cm^2a (12 psia) approximately one minute after lift-off.

The covers also serve as heat exchangers for removing excess heat from the Inertial Platform. The water-methanol coolant from the environmental conditioning system flows through passages in the covers removing the excess heat. The water-methanol solution is maintained at $15 \pm 1°C$ (59 $\pm 1.8°F$) by the IU environmental conditioning system. The mass temperature of the platform stabilizes at approximately $42°C$ ($107.6°F$). The hemispherical covers are fabricated of aluminum and are secured to the platform abse with cap screws and sealed with full volume "0" rings.

The Inertial Platform gimbals, pivot housings, and the base are machined from beryllium which provieds the greatest stiffness-to-weight ratio, stability after machining, and excellent heat transfer characteristics.

The inner gimbal is geometrically the most complicated gimbal and the most difficult to fabricate (Figure 14.2-8). This gimbal supports the three stabilizing gyros, three accelerometers, two pendulums, the inertial prism, the synchro prism assembly, and electronic modules. The mounting surfaces for the gyros, accelerometers, pendulums, and prisms are precision-machined surfaces. Orthogonality of the accelerometers' mounting surfaces is ± 3 arc seconds; the gyros' is ± 2 arc minutes. The pendulums are matched to the accelerometers to ± 3 arc seconds, and the prisms are matched to the inertial coordinates to ± 3 arc seconds. Gas supply passages are machined inside the inertial gimbal, and special fittings on the gas bearing components allow them to be plugged directly into the gas supply passages.

The gimbal rings are designed as spherical sections. This geometry was chosen to give maximum mechanical rigidity and stability and to provide symmetrical moments of inertia for servoloop design.

Gimbal load bearings are designed so that minimum sheer is placed on the gimbal. This is accomplished by using a pair of bearings on one pivot of each gimbal which is preloaded against each other with a 36.3 kilograms (80 pounds) load. The

other pivot of each gimbal has a single gothic arch type bearing loaded with a 2.27 kilograms (5 pounds) preload.

Gimbal pivot resolvers are assembled in separate beryllium housings and attached to the pivots with stiff diaphragms. This allows a minimum eccentricity between resolver rotor and stator, bench testing of the assembled resolver, and easy replacement without disassembly of the gimbal structure at the load bearings.

Nitrogen gas is transported across the pivots through an annulus with "0" ring seals. The total leak rate of the pivot annulus will be less than 1400 cm^3/min STP (0.05 cfm).

Structural resonance of the outer and middle gimbals is 265 hertz and 330 hertz, respectively. The structural resonant frequency of 120 hertz is obtained on the inner gimbal because of the stiffness of the beryllium gimbal rings. There is a secondary resonant frequency of the inertial gimbal at 280 hertz which is obtained from the bearings and pivot trunnions. Above 280 hertz, the gimbals attenuate base vibration inputs to provide a smooth support for the components.
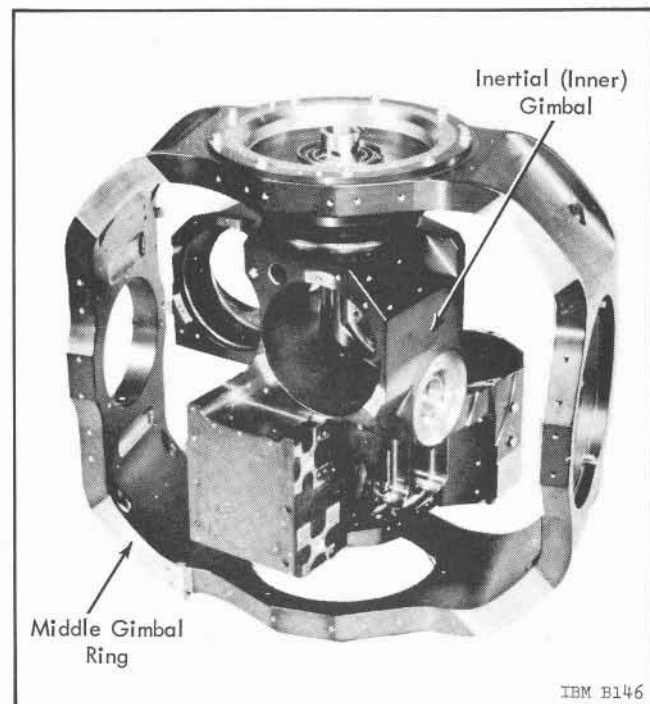
Figure 14.2-8 Gimbal Design

SECTION 14.3
GYRO AND ACCELEROMETER SERVOSYSTEM

The block diagrams of the gyro and accelerometer servoloops are shown in Figures 14.3-1 and 14.3-2, respectively. The servoloops use a 4.8-kilohertz suppressed carrier modulation system with the signal generator outputs being amplified and demodulated on the gimbals of the Inertial Platform. The dc signal from the detector output is transferred from the Inertial Platform to the Platform Electronic Assembly. The dc signal is shaped by a lag-lead stabilization network, remodulated at 4.8 kilohertz, amplified, and then demodulated prior to entering the dc power bridge. This dc power bridge provides a current source drive for the direct axis dc gimbal torquer. The 4.8-kilohertz carrier provides sufficient bandwidth for the servoloop while the current driver for the torquer maintains the gain in the servoloop independent of torquer heating and commutator brush resistance.

The Y servoloop has the Y gyro output signal phase-shaped and amplified and sent to the Y pivot torquer.[*] The X and Z gyro output signals are resolved along the X and Z coordinates of the middle gimbal by a resolver mounted along the -Y pivot. The outputs of the resolver are amplified and demodulated. The resolved signals are then shaped and amplified and fed to their corresponding gimbal pivot dc torquer. No gain compensation such as secant θ_Z is used in the outer gimbal servoloop for middle gimbal angle deviation. Figure 14.3-3 shows the hardware which comprises a gimbal servoloop.

A block diagram of the gyro servoloop and accelerometer servoloop is shown in Figures 14.3-4 and 14.3-5, respectively.

The principle function of the servosystem is to maintain α near 0 in the presence of various disturbances (see Figure 14.3-6). A logical measure of system performance is the ratio $\dfrac{T_\alpha}{\alpha}$ which describes the maximum angular deviation for specified inputs (T_α = torque about IA). For instrumentation reasons, $\dfrac{T_\alpha}{\beta}$ is used because α is very small for frequencies below ten times the resonant frequency. The angular motion β resulting from α is normally a much larger signal by the multiplication factor $\dfrac{H}{J_\beta s}$ (the gyro transfer function) and is easily measured at the input to the servoamplifier, F(s). In system design, the ratio $\dfrac{T_\alpha}{\beta}$ is made as large as possible, consistent with good transient response.

The single-axis approach to the design of the servosystem is expedient, but it ignores the effect of mechanical and electromechanical interaxis coupling. Mechanical interaxis coupling will occur if the principal axes of the inertial gimbal do not coincide with the input axes of the gyroscopes. Also, products of angular motion will occur if the moments of inertial are unequal. Fortunately, in the design and fabrication of the inertial gimbal, close tolerance controls make these effects negligible relative to other interaxis coupling.

The output signal (β) of the single-axis gyro is a measure of the angle between the gyro float and the gyro case. This is the signal used in the Inertial Platform servoloop. Because the gyro case is mounted to the inertial gimbal, disturbances about its output axis are reflected into other servoloops. Therefore, motions of the gyro case about its output axis are coupled into the servoloop for that gyro.

The effect on servostability of this pickoff coupling depends on the orientation of the gyros on the Inertial Platform. A multiplicity of possible gyro orientations exists for an Inertial Platform but it is not possible to orient three single-axis gyros with mutually orthogonal input axes in such a way that there is not at least one closed loop of gyro pickoff

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[*] The XYZ coordinates used in this book have been changed to reflect the Project Apollo Coordinate System Standards (SE 008-001-1). This coordinate scheme becomes effective for vehicles 206 and 502 (and subsequent). For further details refer to Section 1.2.

interaxis coupling. Additional considerations for orientation of the gyros on the inertial gimbal must be made. These include the coning or rectification drift and the unequal elasticity (anisoelasticity) in the gyro spin axis and input axis which give rise to a steady state drift rate. The effect of gyro orientation is a function of the direction of applied linear accelera-

tion or vibration and varies sinusoidally as the acceleration vector is rotated about the gyro's output axis.

Figure 14.3-6 is a sketch of the Inertial Platform gyro orientation. The inertial gimbal is gimbal-mounted for 3 degrees of freedom relative to the vehicle about the three mutually-perpendicular axes
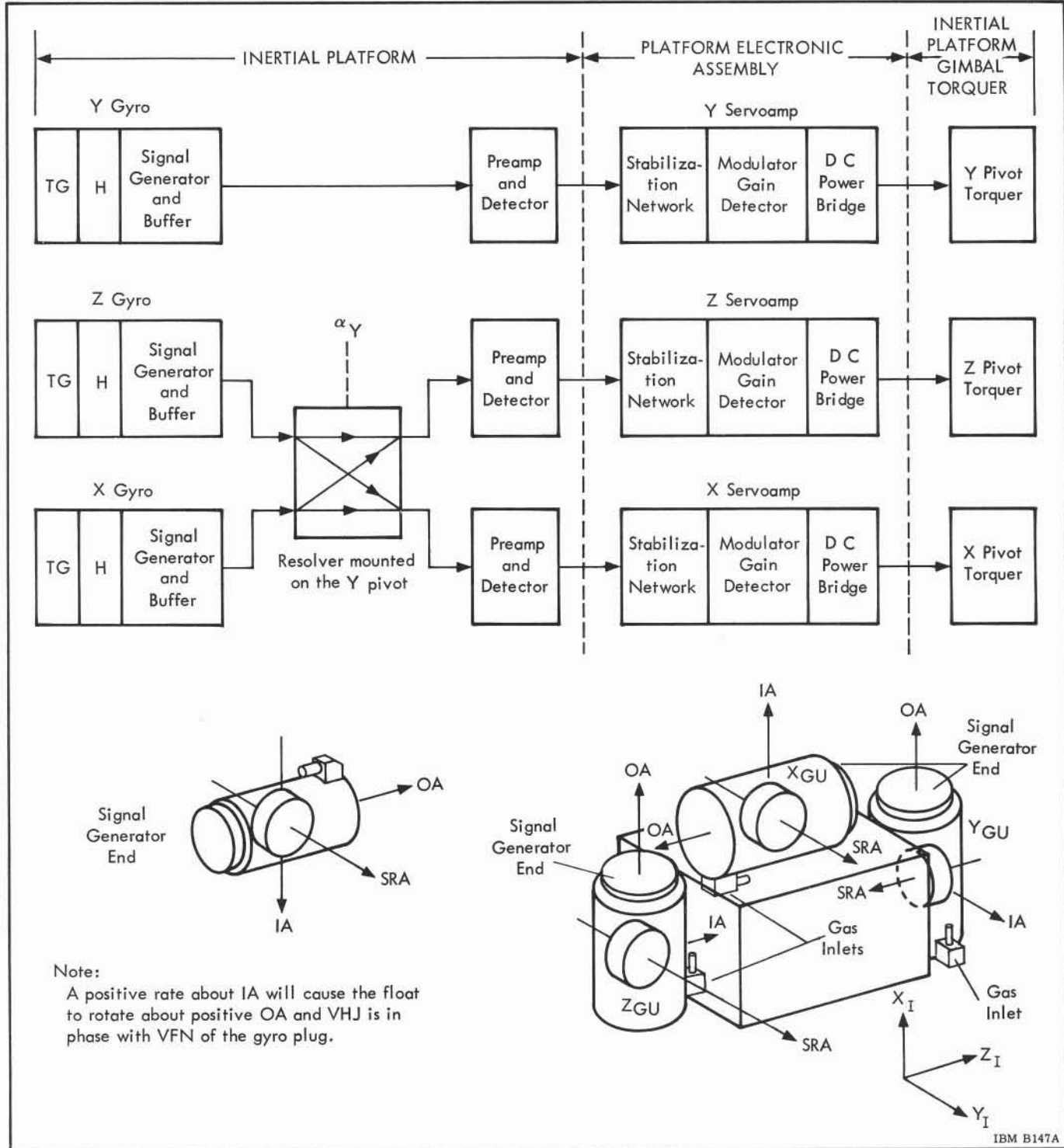
Figure 14.3-1 Gyro Servoloops Block Diagram

identified as α_X, α_y, and α_z. Each gyroscope is represented by mutually-perpendicular vectors which define the angular momentum (H), the input angular motion (α), and the output motion (β). The gyroscopes are identified with their particular platform axes. Inertial Platform angular motion about the input axis of the X gyro is coupled into the pickoff of the Y and Z gyros, and angular motion about the input axis of the Z gyro is coupled into the pickoff of the X gyro.

The actual pickoff signals which are fed to the amplifier can be expressed as (Figure 14. 3-6):

$$\sigma_X = \beta_X + \alpha_Z$$

$$\sigma_Y = \beta_Y - \alpha_X$$

$$\sigma_Z = \beta_Z + \alpha_X$$

The 3-axis servosystem block diagram shown in Figure 14. 3-7 illustrates this coupling through the gyro output axes. The motion about the Y axis is coupled into the Y axis servo but does not feed back into either of the other two gyro servoloops. Therefore, the Y axis servo will be disturbed by motion of the X axis but no stability problem results. There is coupling of the input angles α_X and α_Z to the output motions β_X and β_Z, respectively. Therefore, a closed loop is formed which, in addition to causing cross axis disturbances, must also be considered as to its effect on servo performance. A resolver (α_Y) provides continuous coupling between the inertial components and the gimbal about which compensating servotorques are applied. This effect is shown by the sinusoidal coupling between the X and Z axes.
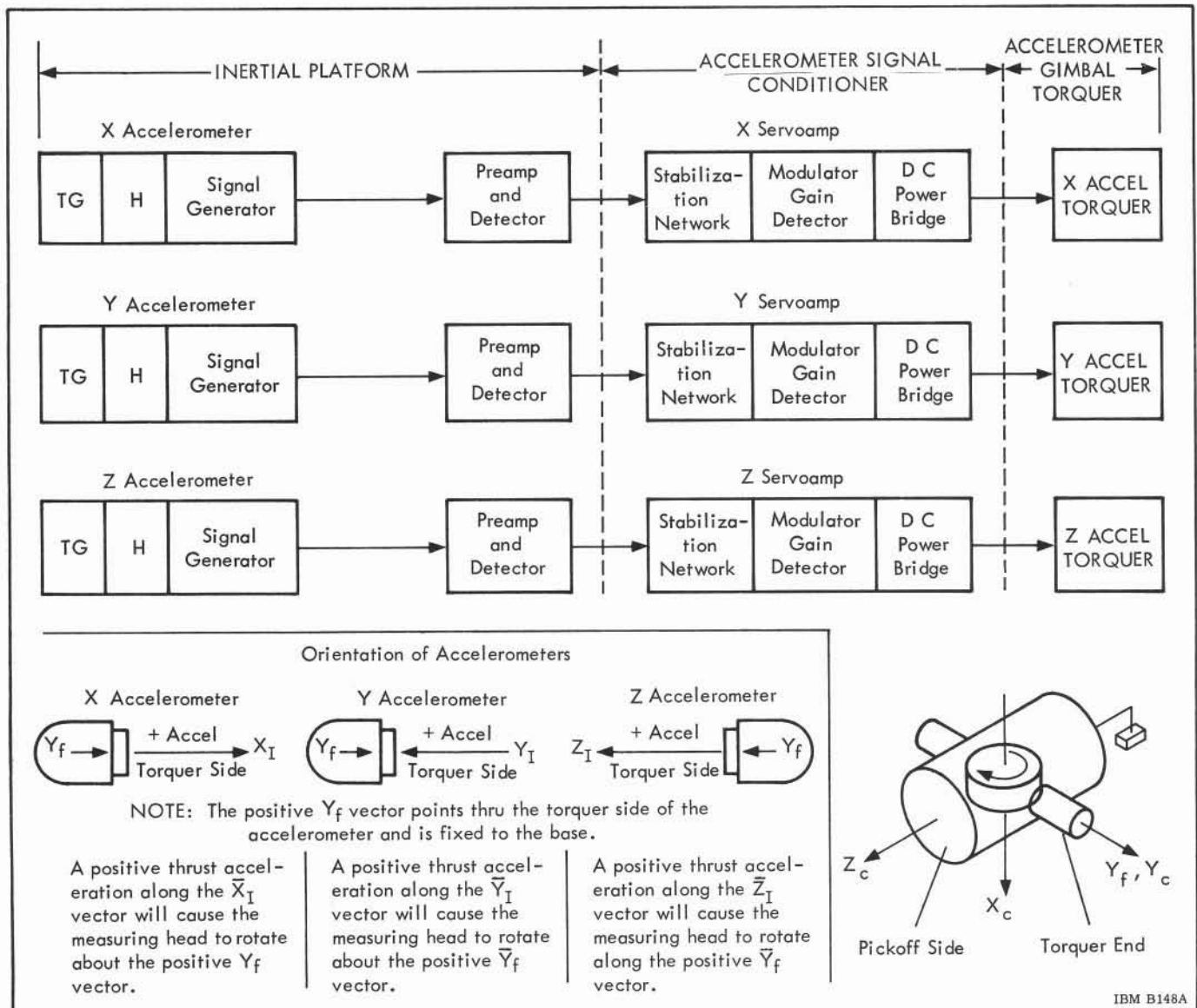
Figure 14. 3-2 Accelerometer Servoloops
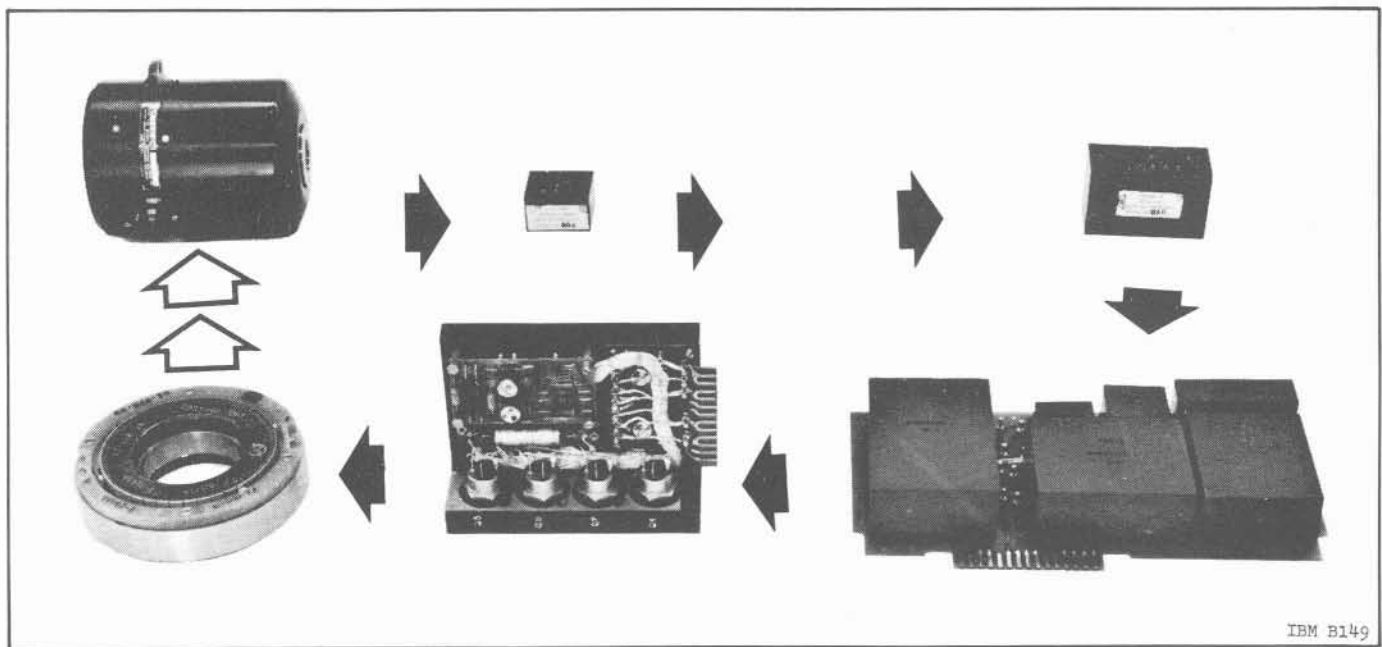
Figure 14. 3-3 Gimbal Servoloop Hardware

IBM B150

PLATFORM

SERVO BOX

K_{TG} — Torquer

3k

K_{DC} — DC Drive

$0.280 \dfrac{\text{amps}}{\text{volt}}$

$\dfrac{T_M}{V_{IN}} = \dfrac{2.4 \times 10^3 \left(\dfrac{S}{5000} + 1\right)}{\dfrac{S^2}{(1.67 \times 10^3)^2} + \dfrac{2 \times .355}{(1.67 \times 10^3)^2} + 1} \quad \dfrac{\text{g-cm}}{\text{volt}}$

K_{D2} — Detector & Filter

5k

10V -4.8 kHz

K_A — Amplifier

$K_M K_A K_{D2} = 35.7 \dfrac{\text{Vdc}}{\text{Vdc}}$

K_M — Modulator

10V - 4.8 kHz

38k

$G_N(S)$ — Network

$G_N(0) = 0.102 \dfrac{\text{Vdc}}{\text{Vdc}}$

$G_N(S) = \dfrac{(S+39)(S+161)(S+34 \pm J46.3)}{(S+2.74)(S+172)(S+216 \pm J617)}$

GAIN SETTING = 90,000 $\dfrac{\text{g-cm}}{\text{deg}}$

PLATFORM

K_{D1} — Detector & Filter

10V - 4.8 kHz

$18.2 \dfrac{\text{Vdc}}{\text{Vac}}$

K_P — Preamp

$(1.5 + J11.4)k$

K_B — Buffer

30k

$10 \dfrac{\text{Vdc}}{\text{deg}}$

GYRO

K_{SG} — Signal Generator

10V - 4.8 kHz

$Z_{IN} - (1.4 + J3.5)k$

$K_{SG} = 0.550 \dfrac{\text{Vac}}{\text{deg}}$

Max Gyro Precession $+3°^{+0°}_{-+5°}$

Figure 14. 3-4 Gimbal Electronics Block Diagram
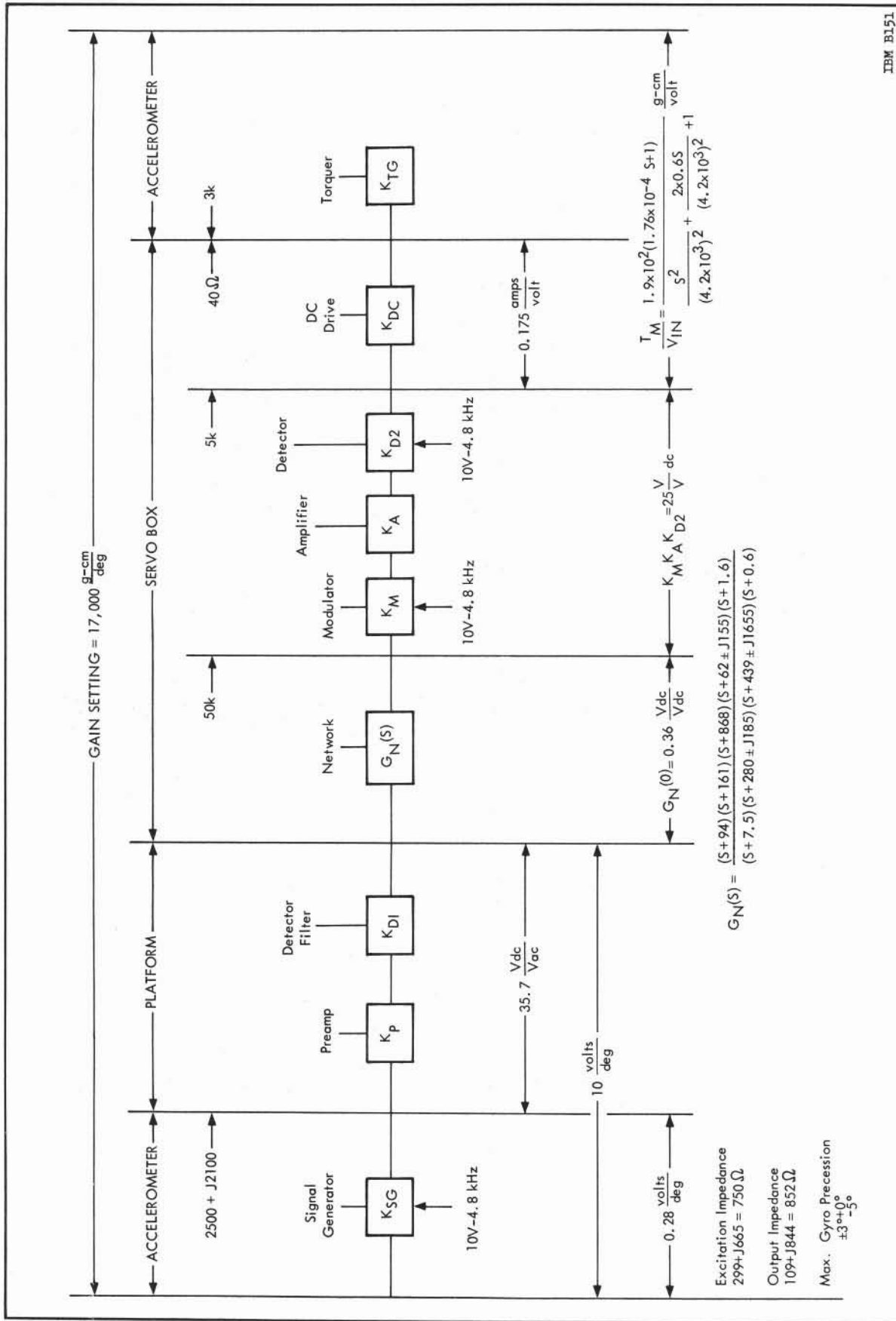
Figure 14.3-5 Accelerometer Electronics Block Diagram

IBM B151

$$G_N(S) = \frac{(S+94)(S+161)(S+868)(S+62\pm J155)(S+1.6)}{(S+7.5)(S+280\pm J185)(S+439\pm J1655)(S+0.6)}$$

$$G_N(0) = 0.36 \frac{Vdc}{Vdc}$$

$$K_M K_A K_{D2} = 25 \frac{V}{V} dc$$

$$\frac{T_M}{V_{IN}} = \frac{1.9\times10^2(1.76\times10^{-4}\ S+1)}{\frac{S^2}{(4.2\times10^3)^2} + \frac{2\times0.65}{(4.2\times10^3)^2} +1} \frac{g\text{-cm}}{volt}$$

Excitation Impedance
299+J665 = 750 Ω

Output Impedance
109+J844 = 852 Ω

Max. Gyro Precession
±3°+0°
     -5°
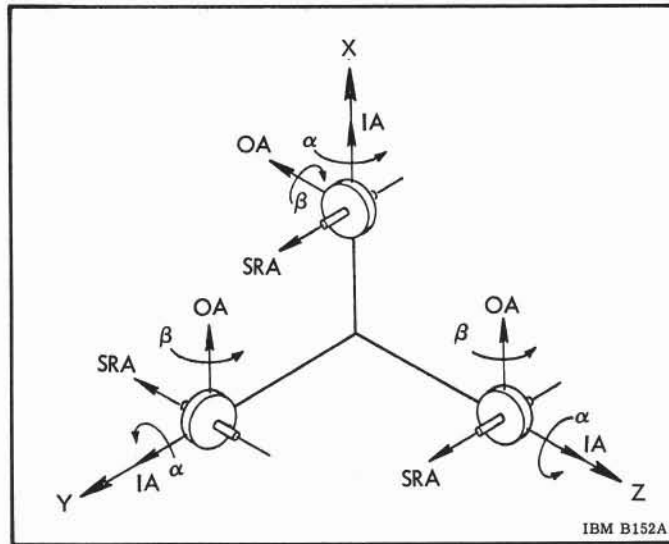
Figure 14. 3-6 ST-124-M Gyro Orientation
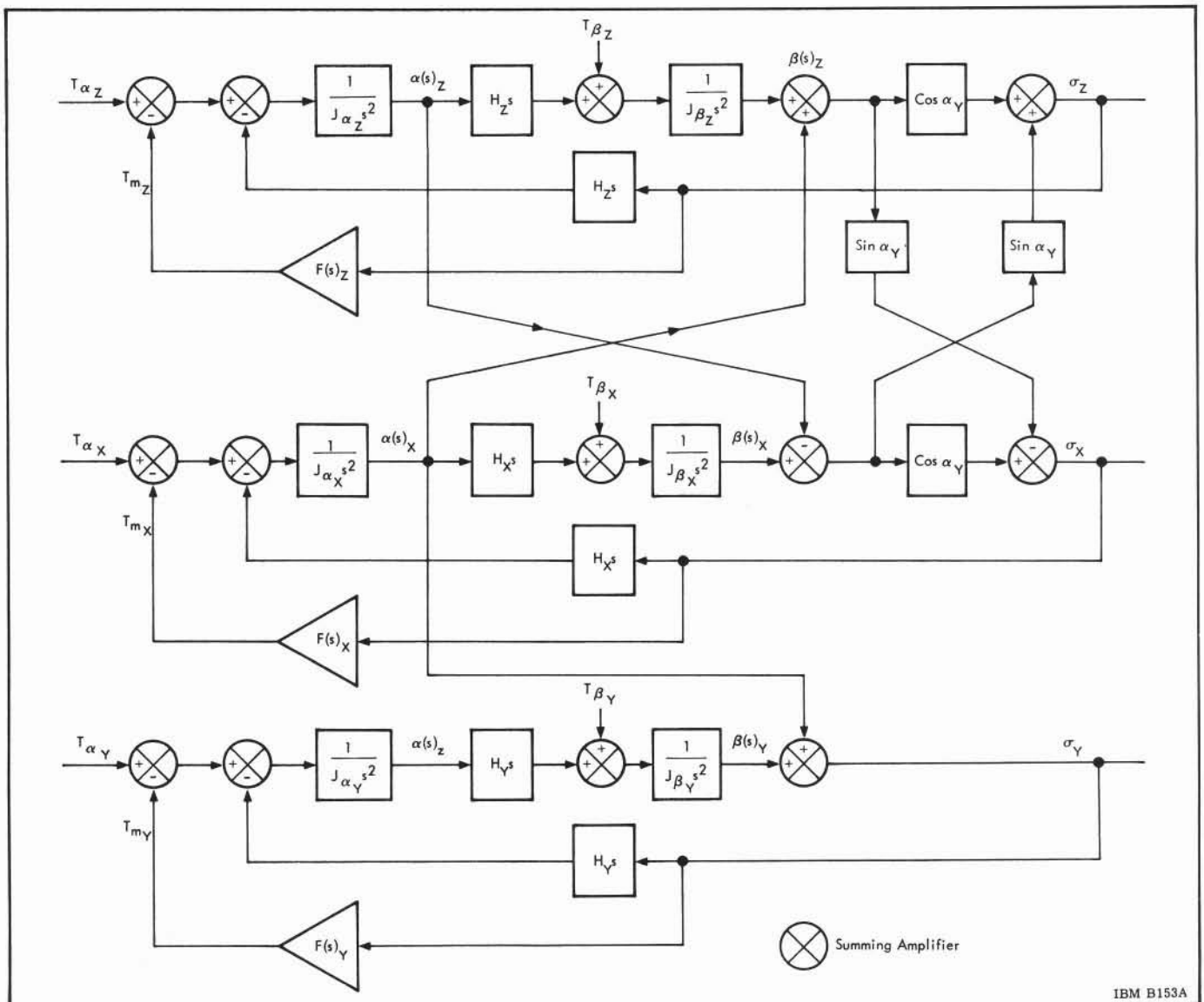
Figure 14. 3-7 Three-axis Inertial Platform Assembly Block Diagram

SECTION 14.4
GIMBAL ANGLE MULTISPEED RESOLVERS

As shown in Figure 14.4-1, the +X, +Y, and +Z gimbal pivots have multispeed analogue resolvers for angular readouts.* The phase shift of their output voltage is measured by digital techniques. A schematic block diagram is shown in Figure 14.4-2.

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* The XYZ coordinates used in this book have been changed to reflect the Project Apollo Coordinate System Standards (SE 008-001-1). This coordinate scheme becomes effective for vehicles 206 and 502 (and subsequent). For further details refer to Section 1.2.

The dual resolver provides the demanded high-readout accuracy. The digital computer system processes the measured gimbal attitude signals and computes the pitch, roll, and yaw vehicle body rates for the attitude control system.

The dual resolver has both a 32-speed and a single-speed winding on the same magnetic structure. The 32-speed winding has 32 electrical rotations for one mechanical shaft rotation. The reference or the resolver excitation (V_1) is 26 volts, 1016 hertz. The output winding drives an RC bridge network as shown in Figure 14.4-2.
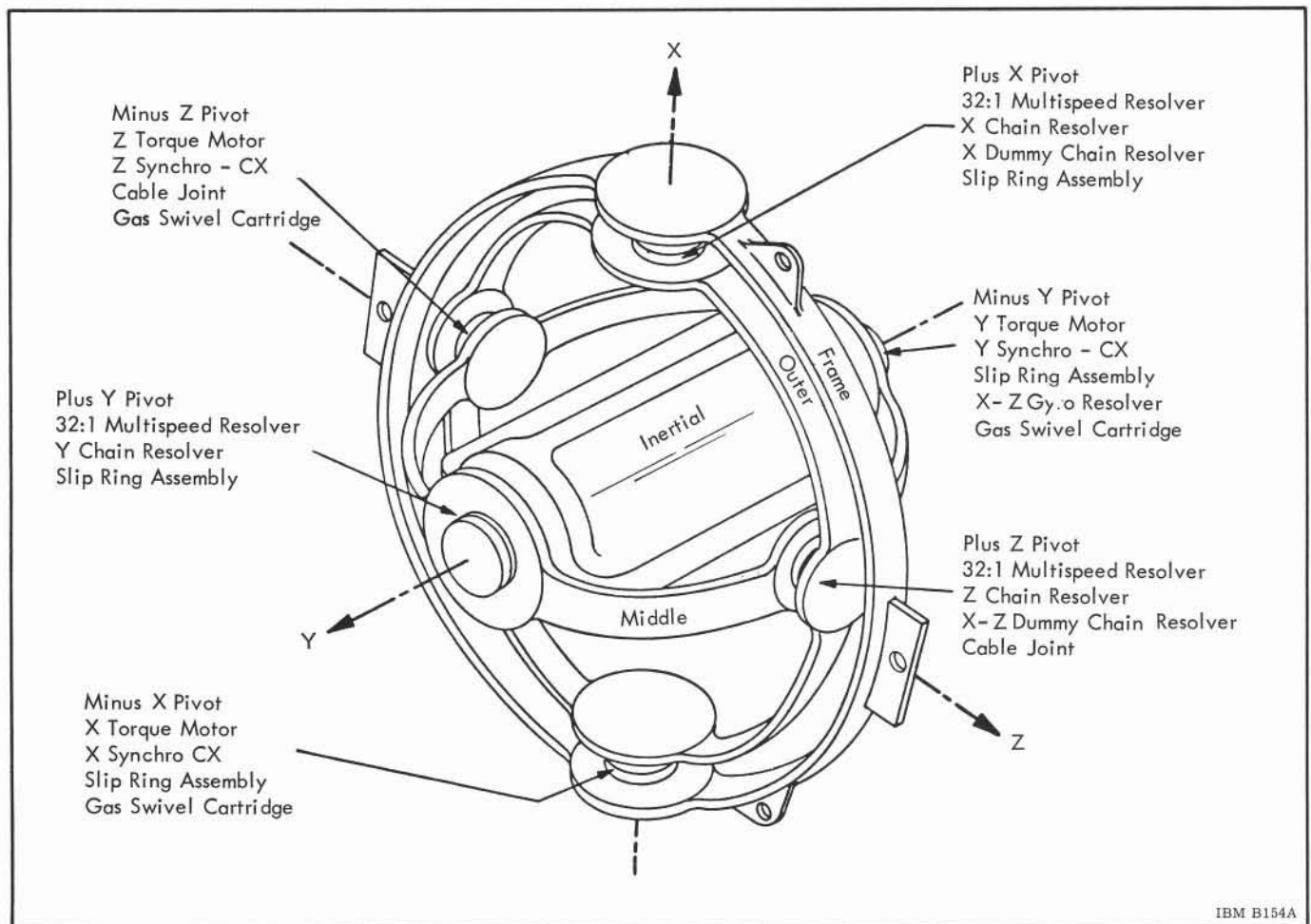
Figure 14.4-1 Three-gimbal Configuration

The voltage V_2 at the input to the start pulse generator can be expressed as

$$V_2 = \frac{E}{\sqrt{2}} \, e^{j(32\theta - \pi/4)}$$

and the voltage V_3 at the input to the stop pulse generator can be expressed as

$$V_3 = \frac{E}{\sqrt{2}} \, e^{-j(32\theta - \pi/4)}$$

where E is the open circuit resolver voltage and θ is the rotation or mechanical angle in radians.

From the ratio of $\dfrac{V_2}{V_3} = e^{j(64\theta - \pi/2)}$, it is seen that the phase shift of V_2, with respect to V_3, is 64 times the angle θ or a multiplication by a factor of 2 occurs in the bridge network.

As the instantaneous voltage V_2 passes through zero with a positive slope, a start pulse is generated

which opens a gate and a counter counts a 2. 048 megahertz clock frequency. As V_3 passes through zero with a positive slope, a stop pulse is generated which closes the gate and stops the counter. The number of cycles counted is a measure of the gimbal angle (Refer to the vector diagram in Figure 14. 4-3).
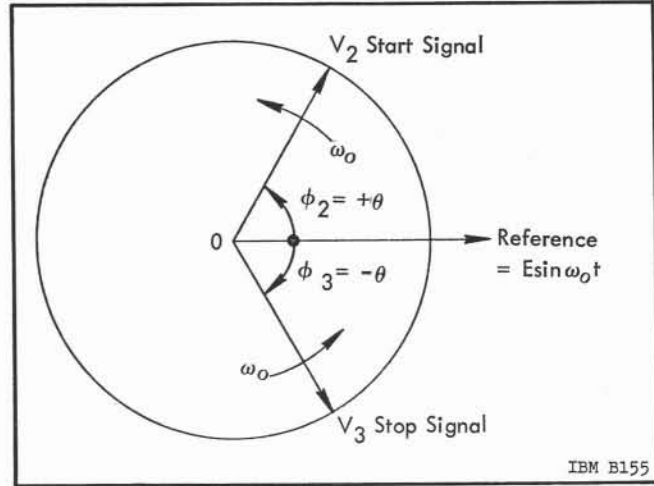
Figure 14. 4-3 Gimbal Angle Vector Diagrams
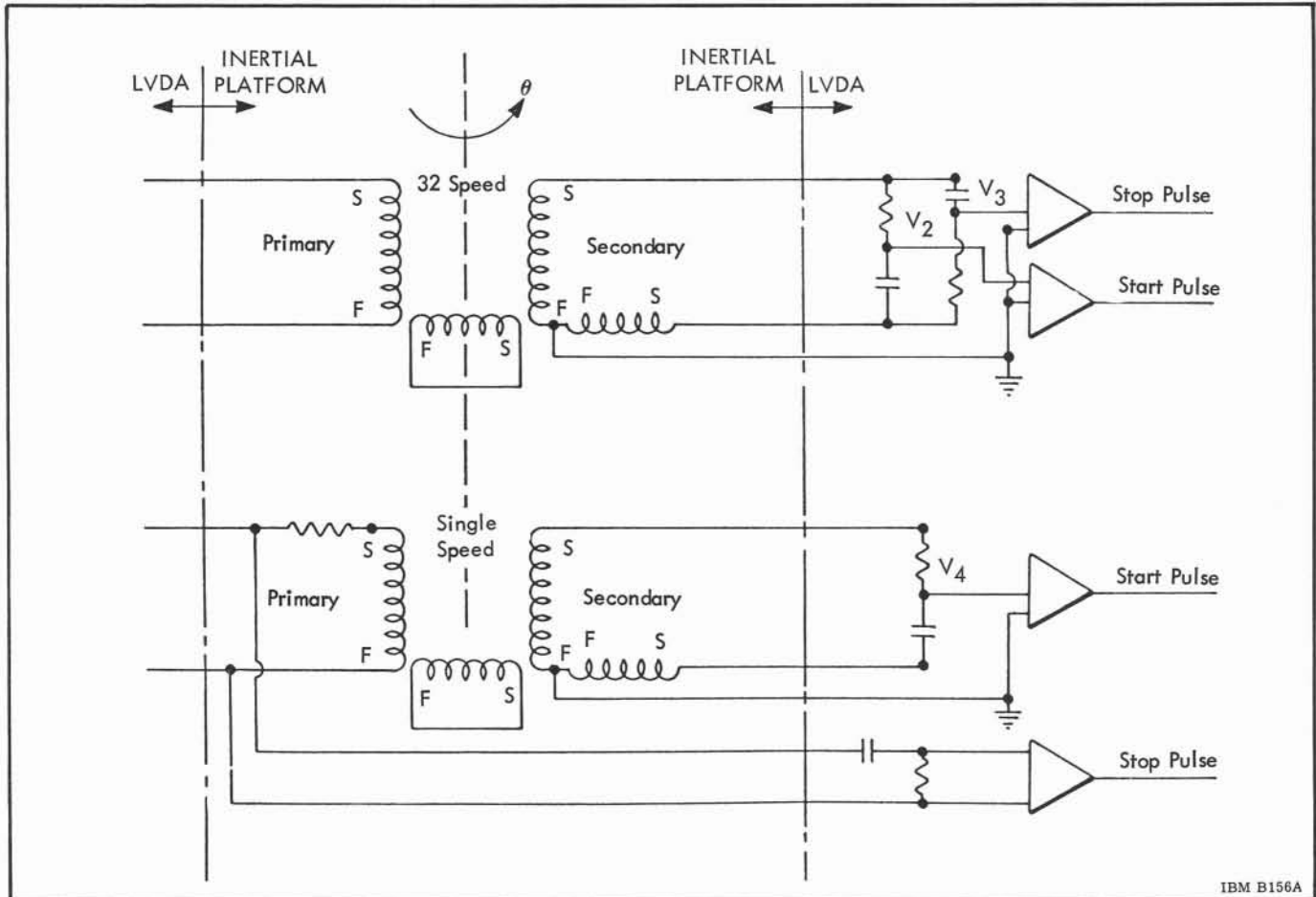
Figure 14. 4-2 Two-speed Resolver Schematic

The single-speed winding uses a single RC-passive network which generates a start pulse; the stop pulse is obtained from the reference voltage to the resolver primary as shown in Figure 14. 4-2. The method of counting is the same for the high-speed windings. This information is utilized only if the multispeed winding is lost. Thus, degraded accuracy occurs rather than complete failure.

The computer system provides two frequency reference sources, one for the multispeed winding and one for the single-speed winding. These references are controlled by the same frequency standard to insure that the gimbal angle measurement will not be lost if one reference source fails.

The multispeed system accuracy is basically insensitive to temperature variation of the resolver as well as impedance unbalance in the output windings. The open circuit voltage of the resolver is the basic reference voltage as shown in the equations.

The resolver and performance characteristics of the angular readouts are listed in Table 14. 4-1.

The Platform system also contains an analogue resolver chain system which can provide vehicle pitch, roll, and yaw attitude steering signals directly to the control system. This is a backup system and is not

planned as flight equipment. The capabilities will be utilized in the ground checkout system to program the platform gimbals for vehicle control system calibration.

The resolver chain system includes three servo-driven resolvers whose shafts are time programmed from the ground computer with a maximum rate of 2 degrees per second; operational rate is 1 degree per second. These units are located in the inertial data box.

The resolver chain schematic is shown in Figure 14. 4-4. The resolvers X_X, X_Y, and X_Z are the time-programmed units. The characteristics of the resolver chain system are listed in Table 14. 4-2.

The resolver chain performs coordinate transformation computations. From Figure 14. 4-4, it can be seen that the resolver chain provides six transformation matrices. With proper detection, two chain references (f_1 and f_2) allow vehicle roll, yaw, and pitch signals to be detected from the transformation matrices. The 1. 6-kilohertz signal on the $\overline{Y}_B$ winding and the 1. 6-kilohertz signal on the $\overline{X}_B$ at the output of the resolver chain provide the roll and pitch rates, respectively. The 1. 92-kilohertz signal on the $\overline{X}_B$ winding provides the yaw rate. If redundant steer-

Table 14. 4-1 Angular Readout Characteristics

Resolver Characteristics	32-Speed	Single-Speed
Excitation voltage	26 V ± 5%	26 V ± 5%
Harmonic content of excitation	0. 1%	0. 1%
Excitation frequency	1016 Hz ± 0. 01%	1016 Hz ± 0. 01%
Excitation power	1. 8 W	0. 08 W
Mechanical accuracy	± 10 arc sec	± 30 arc min
Secondary voltage maximum (open circuit)	5. 0 V	5. 0 V
System Characteristics		
System high speed	64:1	
System low speed	1:1	
Static accuracy	± 30 arc seconds	
Dynamic accuracy (error is proportional to input rate)	20 arc seconds at 0. 2 rad/second	
Computer clock frequency	2. 048 MHz ± 0. 01%	
Temperature range for optimum accuracy	± 30° C (54°F)	

Table 14. 4-2 Resolver Chain System Characteristics

Excitation		Demodulator Output	
f_1	26 V, 1. 6 kHz	To control computer	3 Vdc/degree of arc
f_2	26 V, 1. 92 kHz	To telemetry (fine)	$\pm$ 2. 5 Vdc/$\pm$ 3°
		(coarse)	$\pm$ 2. 5 Vdc/$\pm$ 15°
		Linear range	$\pm$ 15 degrees

ing signals were required, a third reference (f_3) could be inserted at $\overline{X}'$ and the detection system would provide an additional roll, yaw, and pitch signal at $\overline{Z}_B$ and $\overline{Y}_B$.

The detection scheme for the resolver chain is shown in Figure 14. 4-5. When the vehicle coordinate system is aligned to the navigational coordinate system, the $\overline{Y}_B$ winding will have a standing voltage of 15 volts, 1. 92 kilohertz which is in phase with vector $\overline{Y}_0$. This bias voltage is removed by a bucking voltage obtained from $\overline{Y}_0$ (f_2). The signals are filtered and, by means of signal bucking, detection filtering, modulation amplification and redetection, are separated as shown in Figure 14. 4-5. The output signals are dc with a scale factor of 3 volts per degree. The "3-sigma" accuracy of the resolver chain is 6 arc minutes over a complete cycle.
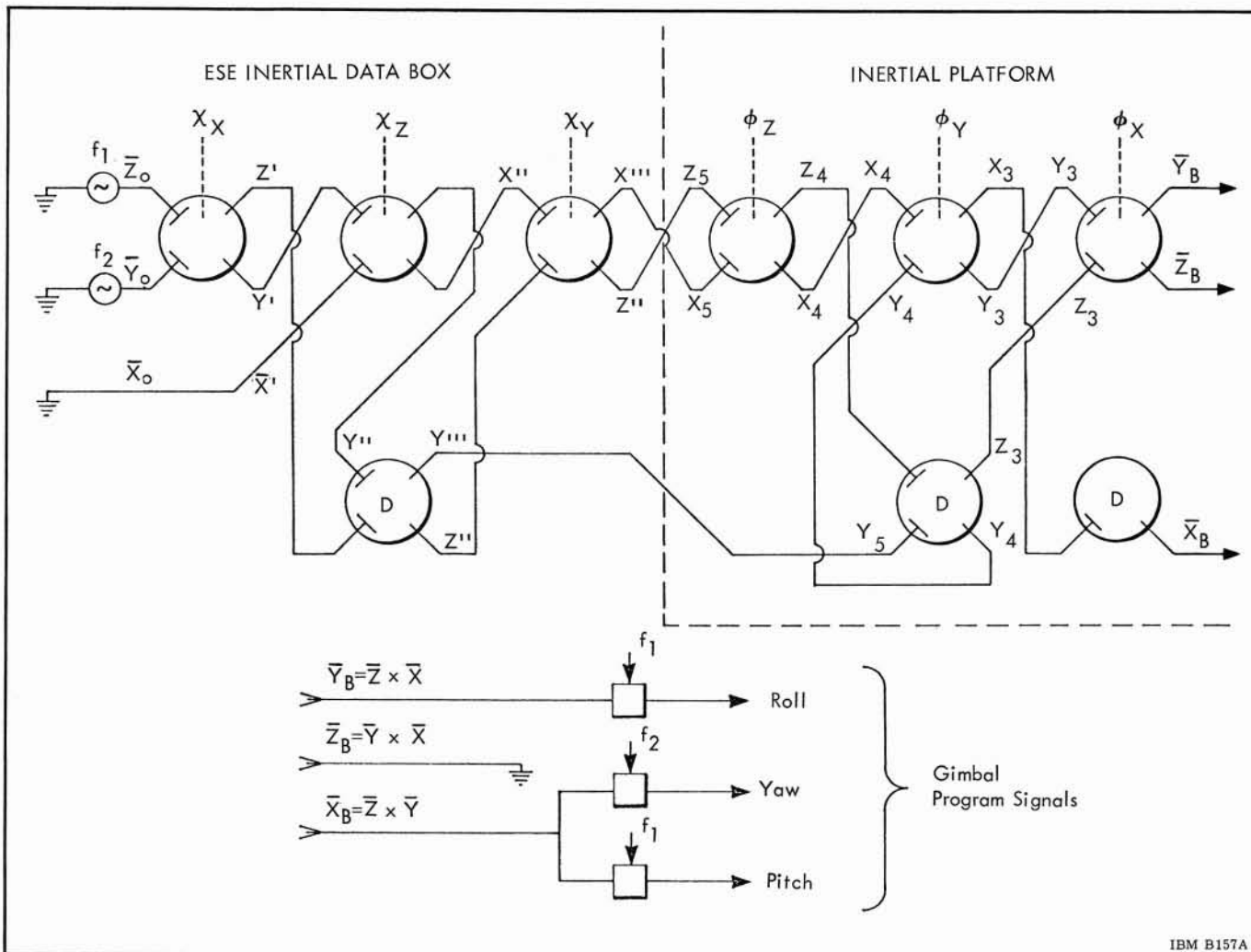
Figure 14. 4-4 Resolver Chain Schematic
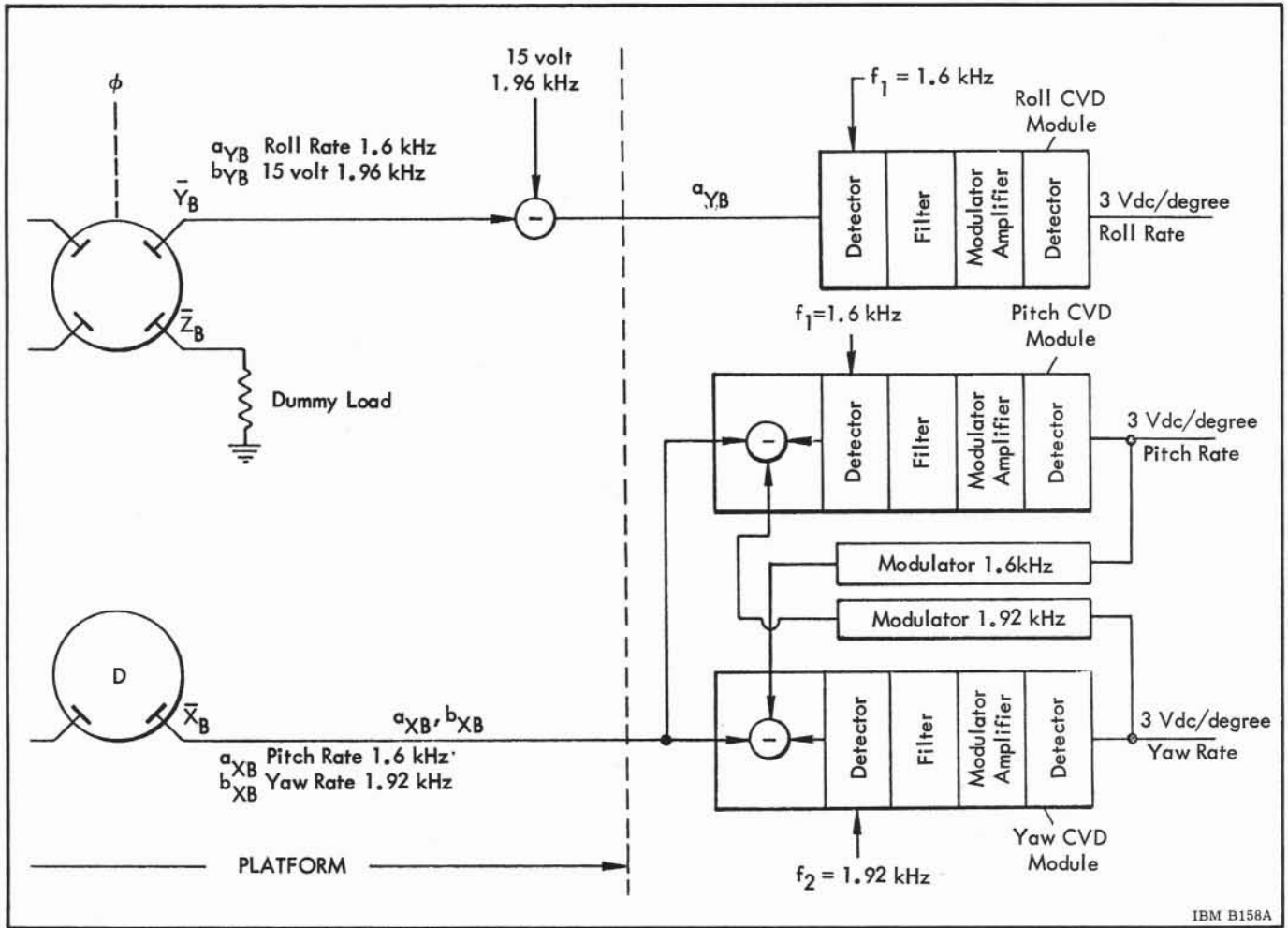
Figure 14. 4-5 Resolver Chain Signal Detection Scheme

SECTION 14.5
OTHER PLATFORM SYSTEM UNITS

PLATFORM ELECTRONIC ASSEMBLY

The ST-124-M Platform Electronic Assembly contains the electronics, other than those located in the Inertial Platform, required for the platform axis and the accelerometer stabilization. Switching electronics for controlling Platform system power and checkout functions are also located in the Platform Electronic Assembly. The electronics are printed circuit, modular constructed, and are fitted into the box with an electrical connector (for ease of assembly and maintenance).

Components or modules requiring pressurization are protected by epoxy encapsulation. Internal heat sources are heat-sinked to the main casting, heat removed by conduction into the temperature controlled mounting panels of the IU. For system evaluation, critical control signals are conditioned in the Platform Electronic Assembly and supplied to telemetry.

The Platform Electronic Assembly is a cast magnesium structure and is mounted to the vehicle frame with pads which extend out from the box structure. The box has a light-gauge sheetmetal cover, gasket-sealed, and pressurized to 1. 7 N/cm^{2}d (2 psid). Internally, the box contains a cast magnesium deck for mounting electronic components and a grooved rack for mounting printed circuit modules. The assembly weighs 19 kilograms (42 pounds).

PLATFORM AC POWER SUPPLY

The ST-124-M Platform AC Power Supply (Figure 14.5-1) furnishes the power required to run the gyro wheels, excitation for the platform gimbal synchros, frequency sources for the resolver chain references, and for gyro and accelerometer servosystems carrier. It is a regulated, solid-state, three-phase, ac power supply capable of supplying up to 250 VA continuously. With an input voltage from 25 to 30 Vdc, it produces a three-phase sine wave output, which is fixed at 26 volts (rms) line-to-line at a fixed frequency of 400 ± 0. 01 hertz. Three

single-phase, 20-volt reference-square-wave outputs of 4. 8 kilohertz, 1. 92 kilohertz, and 1. 6 kilohertz are also provided.

The oscillator generates a temperature-stable square wave of 19. 2 kilohertz derived by frequency division from a quartz crystal oscillator. This 19. 2-kilohertz square wave is the frequency source from which all the frequency outputs are derived.

The frequency divider network, referenced with the 19. 2-kilohertz square wave, produces two buffered 4. 8-kilohertz square waves. One 4. 8-kilohertz wave becomes the "clock pulse" for the cycle register; the other feeds the regulator-driven amplifier. In addition, the network provides dividers and buffers for the auxiliary outputs. All frequency-divider circuits use Johnson-type counter logic to provide symmetrical square wave outputs to eliminate complicated decoding gates. The network section utilizes integrated circuits construction.

The cyclic register produces six push-pull, square-wave, 400-hertz outputs spaced in 30-degree increments. The 4. 8-kilohertz signal from the divider network is used as the clock pulse to trigger the six register flip flops. The flip flops are bistable circuits of the same type (integrated-circuit) used in the frequency divider network.

Each cycle register flip flop drives one channel of the power amplifier. One channel consists of two cascaded push-pull stages which are transformer coupled; all transistors operate in the switching mode, which contributes to the high efficiency of the inverter. Each channel excites a push-pull primary winding of a toroidal output transformer. The secondary windings consist of three per transformer with a 4:3:1 winding ratio (A of Figure 14. 5-2). Each of the outputs has 1 winding from each of the 6 transformers (B of Figure 14. 5-2) in a combination of 3 pairs with a 4:3:1 wattage ratio. The manner in which these ratios are used in the generation of the sine wave output is illustrated in C of Figure 14. 5-2.

The output voltages contain a 15-percent harmonic distortion. A low-pass output filter reduces this distortion by approximately 10:1. The voltage regulator compares a sample of the three-phase output voltage with a stable Zener reference voltage, then amplifies the error and biases the power amplifier to close the loop.

The 19.2-kilohertz oscillator, the frequency dividers, the network dc supply, and the cycle register operate in dual redundancy. If the 400-hertz voltage is not detected in channel A, a bistable switching circuit will instantly switch power demand to channel B.

The Platform AC Power Supply box is cast magnesium and has a light-gage sheetmetal cover which is gasket-sealed and maintains a pressure of 1.7 N/cm^{2}d (2 psid). Modular-potted construction is used throughout. Motherboard printed circuit boards with integrated circuitry minimize internal wiring. The weight of the assembly is 14.5 kilograms (32 pounds).

ACCELEROMETER SIGNAL CONDITIONER

The Accelerometer Signal Conditioner accepts the velocity signals from the accelerometer optical encoders and shapes them before they are passed on to the guidance computer. Each accelerometer requires 4 shapers — a sine shaper and a cosine shaper for the active channel and a sine shaper and cosine shaper for the redundant channel. Also included are four buffer amplifiers for each accelerometer — one
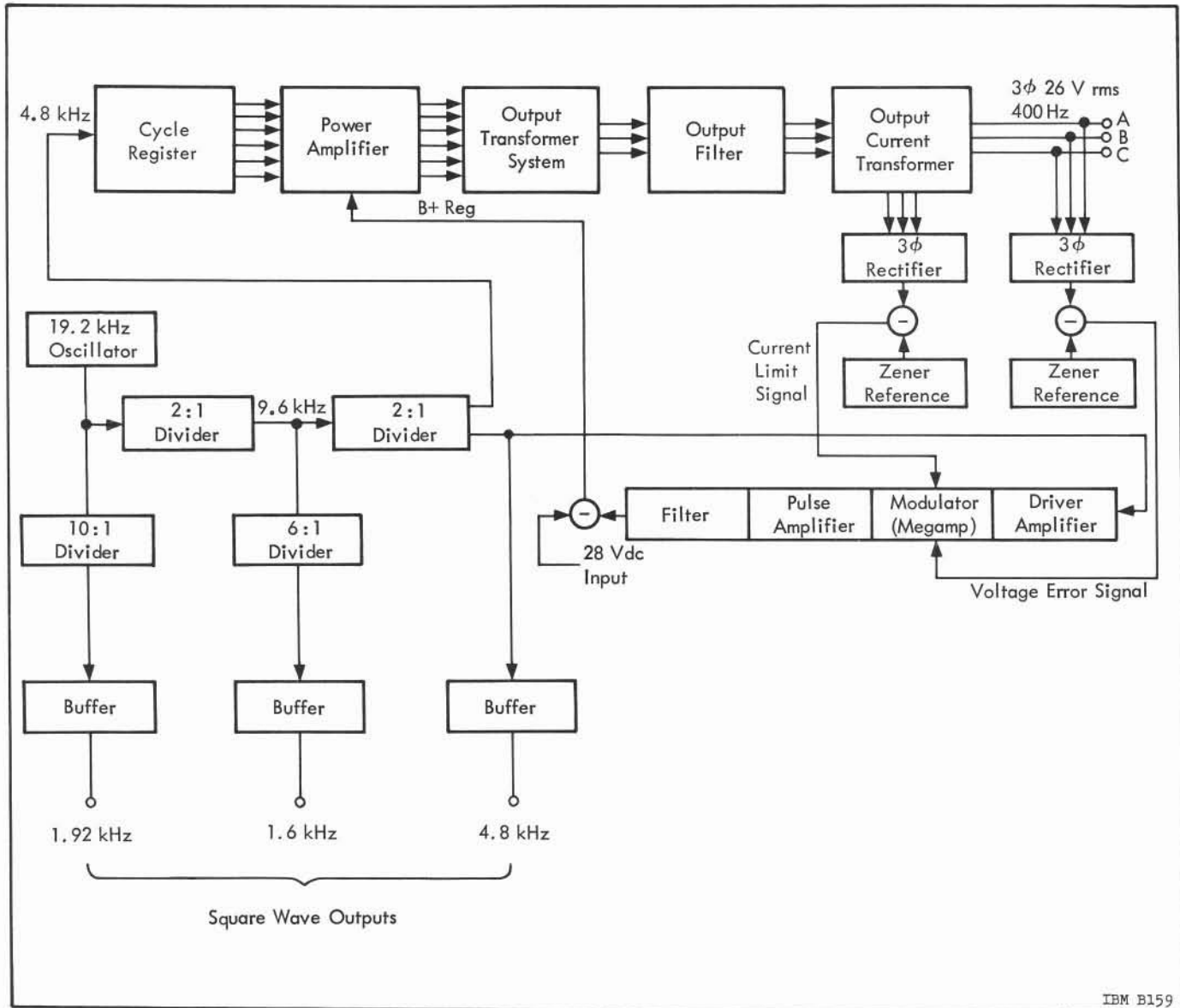
Figure 14. 5-1 Platform AC Power Supply Assembly
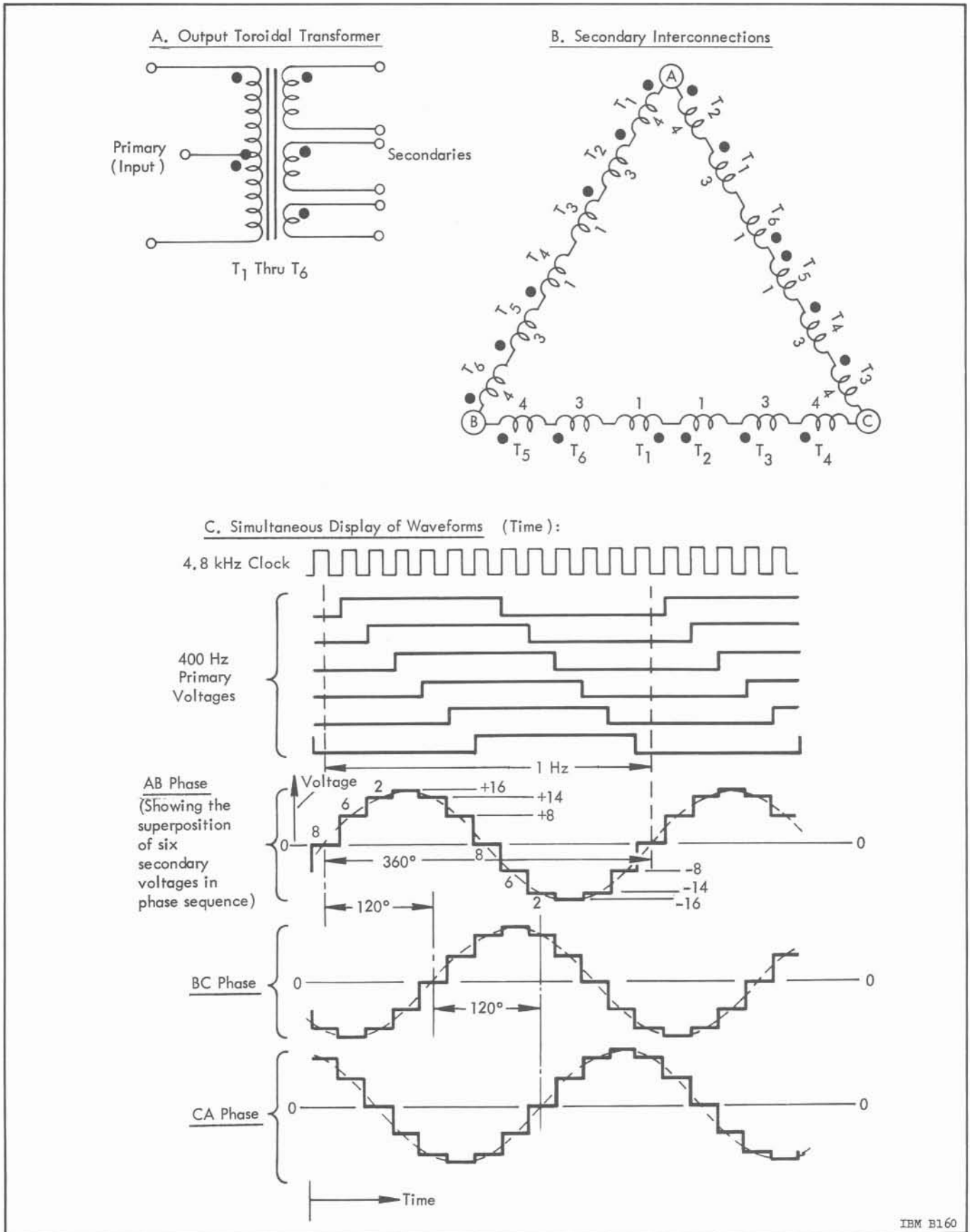
Figure 14. 5-2 Generation of 3-Phase, 400-Hertz Voltage

for each sine and cosine output. The output of these buffers is furnished to the blockhouse for a prelaunch accelerometer calibration check. The accelerometer telemetry velocity signals are also conditioned in this assembly. Characteristics of the velocity signals are listed in Table 14. 5-1.

The total assembly weighs approximately 4. 5 kilograms (10 pounds) and is fabricated from a magnesium casting. Light-gauge sheetmetal covers with gasket seals maintain an internal pressure of 1. 7 N/cm^2d (2 psid).

Table 14. 5-1 Characteristics of the Accelerometer TM Velocity Signals

Output voltage levels	0-6 volts minimum (3 volt bias)
	0-8 volts maximum (4 volt bias)
Signal rise time	50 us on square wave
Output frequency of acceleration signal	+ 60 g (50 Hz/g)
	- 30 g (50 Hz/g)

SECTION 14.6
PLATFORM ERECTION SYSTEM

The erection of the Inertial Platform to the local vertical is accomplished by two gas bearing pendulums which are fitted to the inner gimbal. The input axes of the pendulums are parallel to the Y and Z accelerometer measuring axes*.

The gas bearing pendulum is a single-axis gravity-sensing device. Its sensing element is a gas-floated slug which supports a soft iron core as shown in the cutaway view of Figure 14. 6-1. The iron slug moves inside the coils of a linear differential transformer which provides the electrical output signal. Damping of the slug motion is provided by a chamber and an exhaust orifice while the spring restraint is obtained magnetically.

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* The XYZ coordinates used in this book have been changed to reflect the Project Apollo Coordinate System Standards (SE 008-001-1). This coordinate scheme becomes effective for vehicles 206 and 502 (and subsequent). For further details refer to Section 1.2.

The characteristics of the pendulums are listed in Table 14. 6-1.

The block diagram of the erection system is shown in Figure 14. 6-2. The input axes of the Y and Z pendulums are along the inertial gimbal Y and Z axes, respectively. The pendulum output is amplified by a preamplifier on the Inertial Platform and then transmitted to the ground equipment alignment amplifier.

The alignment amplifier provides a proportional plus integral path to the torque driver amplifier, which returns the signal from the ground to the variable coil of the gyro torque generator. The erection system is a second order system with a natural frequency of 0. 05 radian per second and a damping ratio of 0. 5. The leveling accuracy of the erection system is ± 2. 5 arc seconds.
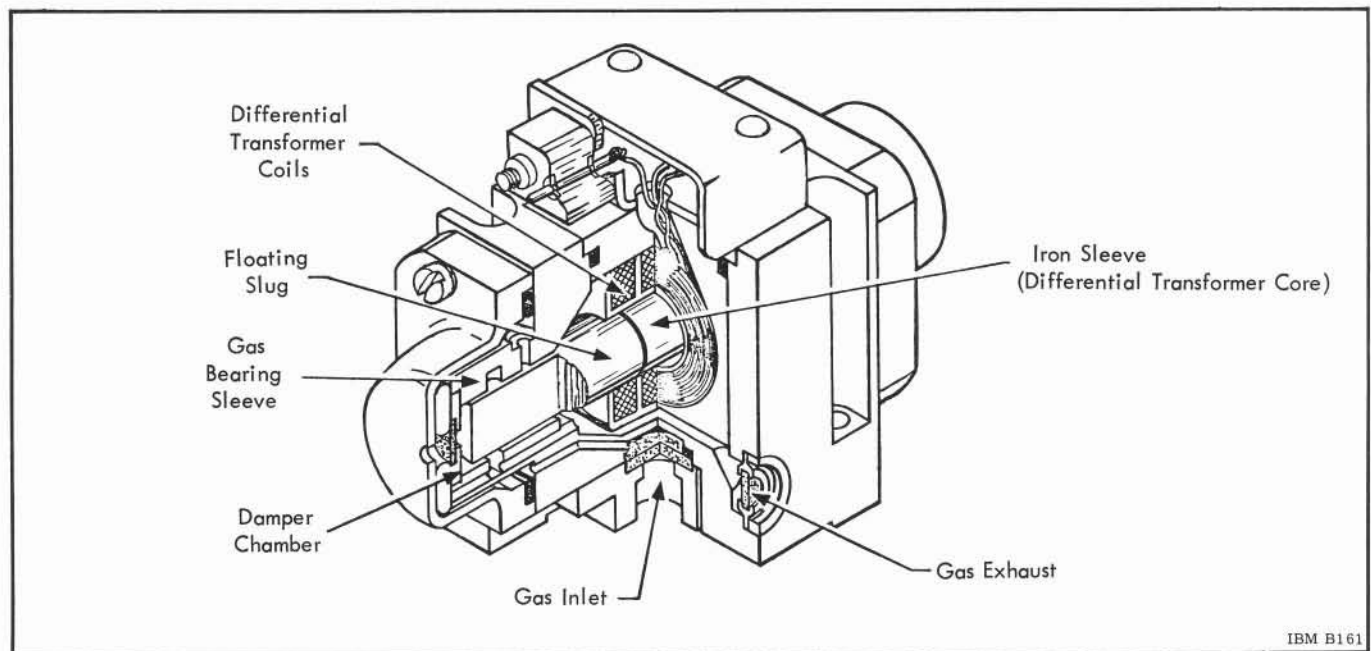
Figure 14. 6-1 Gas Bearing Pendulum

Table 14.6-1 Gas Bearing Pendulum Characteristics

Physical Characteristics	
Size	5.7 cm x 3.8 cm x 3.2 cm (2.25 in. x 1.5 in. x 1.25 in.)
Weight	92 grams (0.2 pound)
Gas Bearing	
Gas pressure	10.3 N/cm^{2}d (15 psid)
Gas flow	100 cc/min STP (6.1 in.^{3}/min)
Gas gap	0.015 to 0.02 cm (0.006 to 0.008 in.)
Signal Generator	
Type	Inductive
Excitation	4 V, 400 hertz
Sensitivity	300 mV/degree
Performance	
Leveling accuracy	± 2.5 arc seconds
Input range	± 0.5° (signal saturation)
Time constant	10 seconds
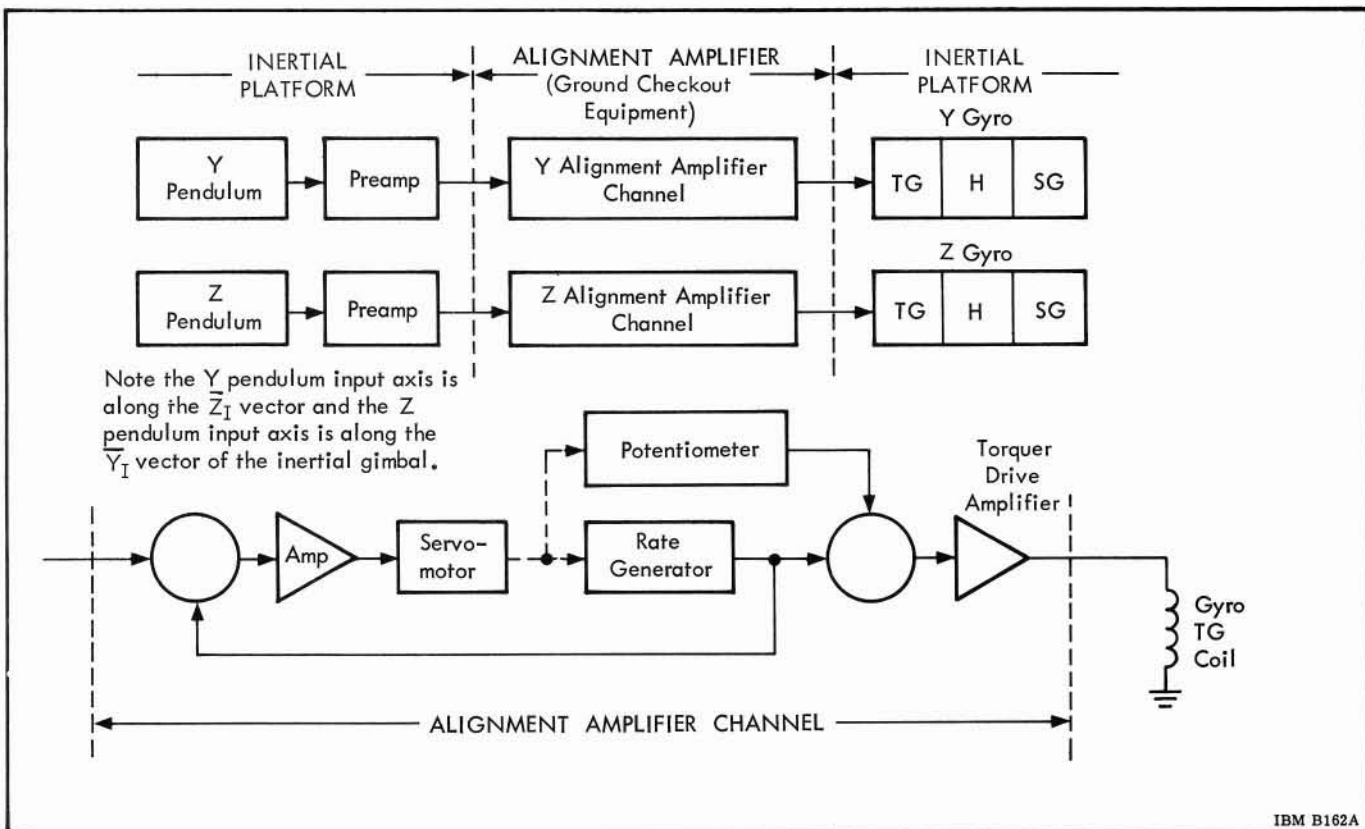
Figure 14.6-2 Inertial Platform Erection System Block Diagram

SECTION 14.7
AZIMUTH ALIGNMENT SYSTEM

The azimuth alignment system orients the Inertial Platform +Z coordinate axis* to the desired flight azimuth. To obtain the high accuracy requirement, the azimuth alignment system utilizes electro-optical techniques. The basic elements of the azimuth alignment system are an autocollimating theodolite with detection system, synchro encoder servosubsystem, launch control computer for azimuth programming and reference, and an alignment torquing servosubsystem. Fitted to the inertial gimbal is a porro prism (intermediate infrared) with its dihedral edge parallel to the Z axis of the inertial gimbal. A second prism (near infrared) is fitted to the stator of the two-speed synchro which has its dihedral edge free to rotate in the Y-Z plane (Figure 14. 7-1).

The basic element of the optical system is the SV-M2 Theodolite (Figure 14. 7-2). The system consists of an autocollimating theodolite with necessary optics and detectors, a penta mirror set, an automatic sway control system, reference prism with stand, closed circuit TV monitoring system, and support electronics and control.

The theodolite has three control channels — the synchro prism, the inertial prism, and a trihedral prism (Figure 14. 7-3). An infrared energy spectrum is generated by a tungsten filament lamp. This spectrum has a bandwidth of two microns which lie between 0. 7 and 2. 7 microns. Atmospheric absorption of infrared in this bandwidth occurs at 1. 35 and 1. 8 microns. Therefore, the control channels are divided at 1. 35 and 1. 8 microns with the 0. 7 to 1. 35 microns near infrared-band assigned to the synchro prism, the 1. 25 to 1. 8 microns intermediate infrared-band assigned to the inertial prism, and 1. 8 to 2. 6 micron far infrared-band assigned to the trihedral prism (Figure 14. 7-4). The prisms have a dichroic multi-layer coating that allows only the desired bandwidth of energy to pass.

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* The XYZ coordinates used in this book have been changed to reflect the Project Apollo Coordinate System Standards (SE 008-001-1). This coordinate scheme becomes effective for vehicles 206 and 502 (and subsequent). For further details refer to Section 1.2.

The angular position of the synchro prism is controlled directly from the theodolite. A theodolite error signal drives a servoamplifier which is located in the launch umbilical tower. The amplifier output is transmitted to the Inertial Platform where it excites a servomotor located on the inertial gimbal. The servomotor positions the synchro prism throuh a gear train of 10^5:1, reducing any angular error to zero. This loop is active throughout countdown until vehicle lift-off. The angle between the inertial gimbal (navigation coordinates) and movable prism is measured by a precision 25:1 dual-speed control transmitter synchro (Figure 14. 7-5). The inertial prism is controlled by the theodolite output signal through the roll alignment servoelectronics by torquing the roll gimbal until the prism is in the desired azimuth plane.

The trihedral prism, which is fixed to the vehicle, is used as reference for a servosystem to position the penta mirror which provides the theodolite with the capability of translatory tracking. At the vehicle elevation of the Platform system, the SV-M2 Theodolite (Saturn V, Mod 2) will compensate for a ± 35. 5 centimeters (± 14 inches) translation movement of the vehicle. The trihedral prism servo-loop increases the aperture of the autocollimator from 20 centimeters (8 inches) to 85 centimeters (32 inches) at a rate of 76 centimeters/second (32 inches/second).

Prism acquisition signals from the three channels are generated with the same infrared-coded bandwidths that produce the error signals. These are used to automatically block and/or initiate other events in the vehicle launch procedure and azimuth alignment system. Indicator lights on the theodolite control panel in the launch control center confirm that the 3 prisms are in the acquisition range of the theodolite. Two closed-circuit TV monitor systems provide the launch control center with a view of the 3 prisms and a monitor of the theodolite control and display panel in the theodolite hut.

The elevation angle (nominally 25 degrees from horizontal) of the theodolite is set by a dc motor-driven actuator which is manually controlled

from the launch control center. At the Saturn V launch complex, the theodolite can operate with a ± 35. 5 centimeter (± 14 inches) variation in elevation without requiring any adjustment.

The encoder-synchro package consists of a 25:1 dual-synchro, an 18-bit optical shaft angle encoder, and a motor-tachometer-gear assembly. The encoder is provided with an enclosure containing strip heaters to maintain the temperature within ± 5°C (9°F), maintaining the designed accuracy of ± 10 arc seconds. The encoder system has a self-

generated encoder sampling time of 20 milliseconds. The output is a cyclic gray code with logic levels of 0 or 28 volts. The 25:1 dual synchro with the platform dual synchro has a back-to-back control transmitter: control transformer error of ± 10 arc seconds.

The reflected infrared energy from the prisms is acquired by the theodolite's optics. From this coded energy, the theodolite detects deviations from the desired position of the prisms. These coded signals are separated into their respective channels by slit prism angle filters and dichroic beam splitters.
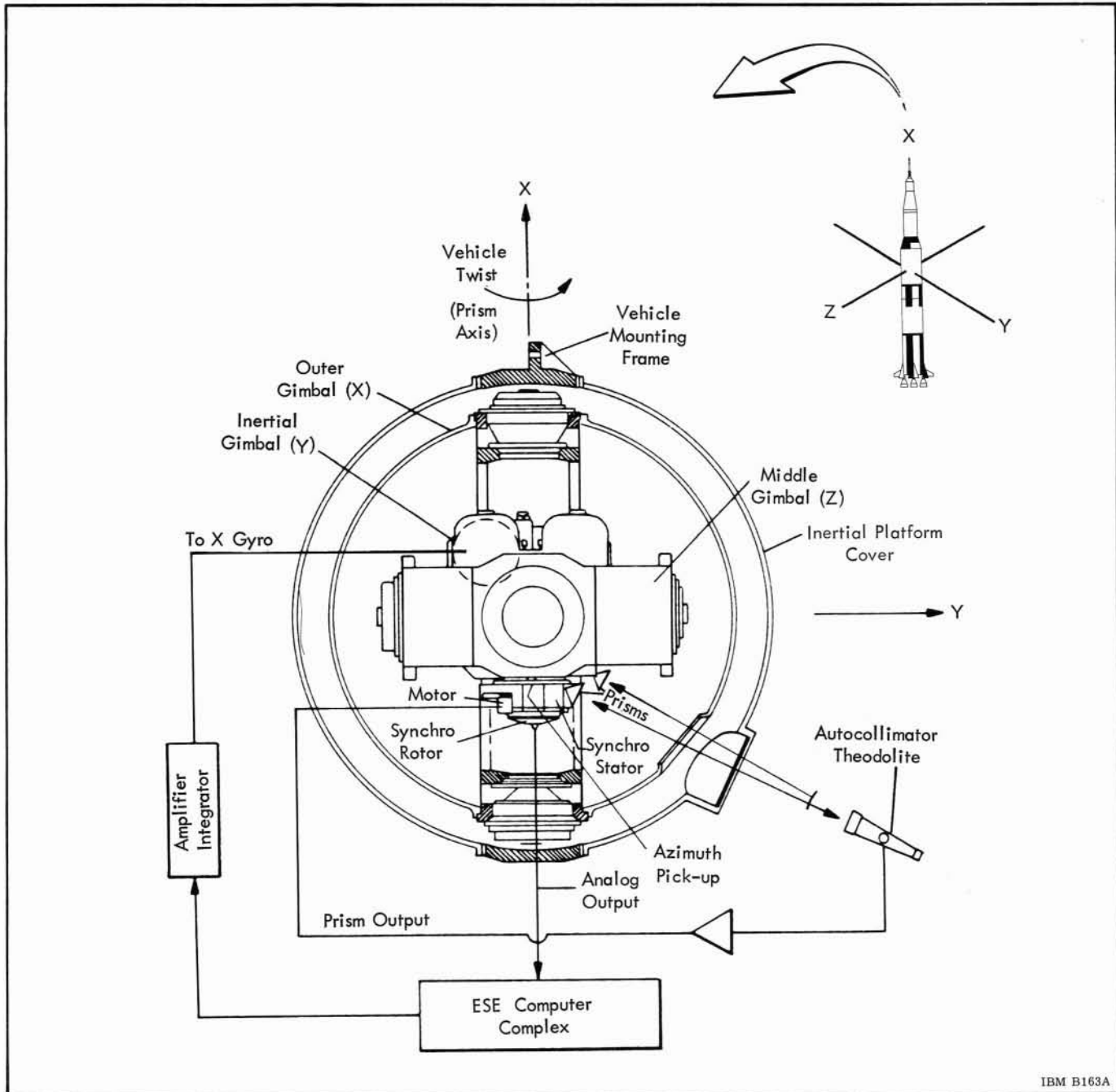
Figure 14. 7-1 Automatic Azimuth Alignment

The dichroic beam splitters have a dichroic multilayer coating similar to the prisms. Each of the three channels has acquisition detectors which generate dc signals when the prisms are within the acquisition range of the theodolite. These signals are used to indicate acquisition and to control relay switching in various other vehicle-associated control functions. When an error exists in any channel, a representative portion of the infrared energy passes through the slit prism filter to the dichroic beam splitter and on to a lead sulfide detector. A phase-oriented ac voltage is generated which is directionally proportional to the angular error. This voltage is amplified and demodulated to achieve the desired signal gradient of 100 millivolts per arc second.

The simplified azimuth block diagram in Figure 14. 7-6 shows the operation of the azimuth alignment system. After all systems have been energized, the erection system positions the inertial gimbal to the local vertical. The azimuth alignment equipment is switched into the acquisition mode. This closes

contacts A, B, C, and D and opens contacts E, F, G, and H. A bias signal is injected into the synchro prism servoloop to drive this prism into acquisition; the acquisition signal removes the bias and closes the synchro prism servoloop on the theodolite and drives the synchro prism to null.

The output of the dual-speed control transformer drives the encoder-synchro servoloop until the synchro output is zero through contact E. The RCA-110A Launch Computer reads and stores the position of the 18-bit encoder. Thus, the baseline azimuth is stored in the computer. This is the azimuth of the navigation coordinate system and is the reference from which launch azimuth is established. Any deviation from this position will generate an error signal in the CX:CT measurement causing the X gyros to torque the gimbal and reduce the error to zero.

The mission azimuth is established with contacts A, B, C, and D open and contacts E, F, G, and H closed. The launch control computer computes the
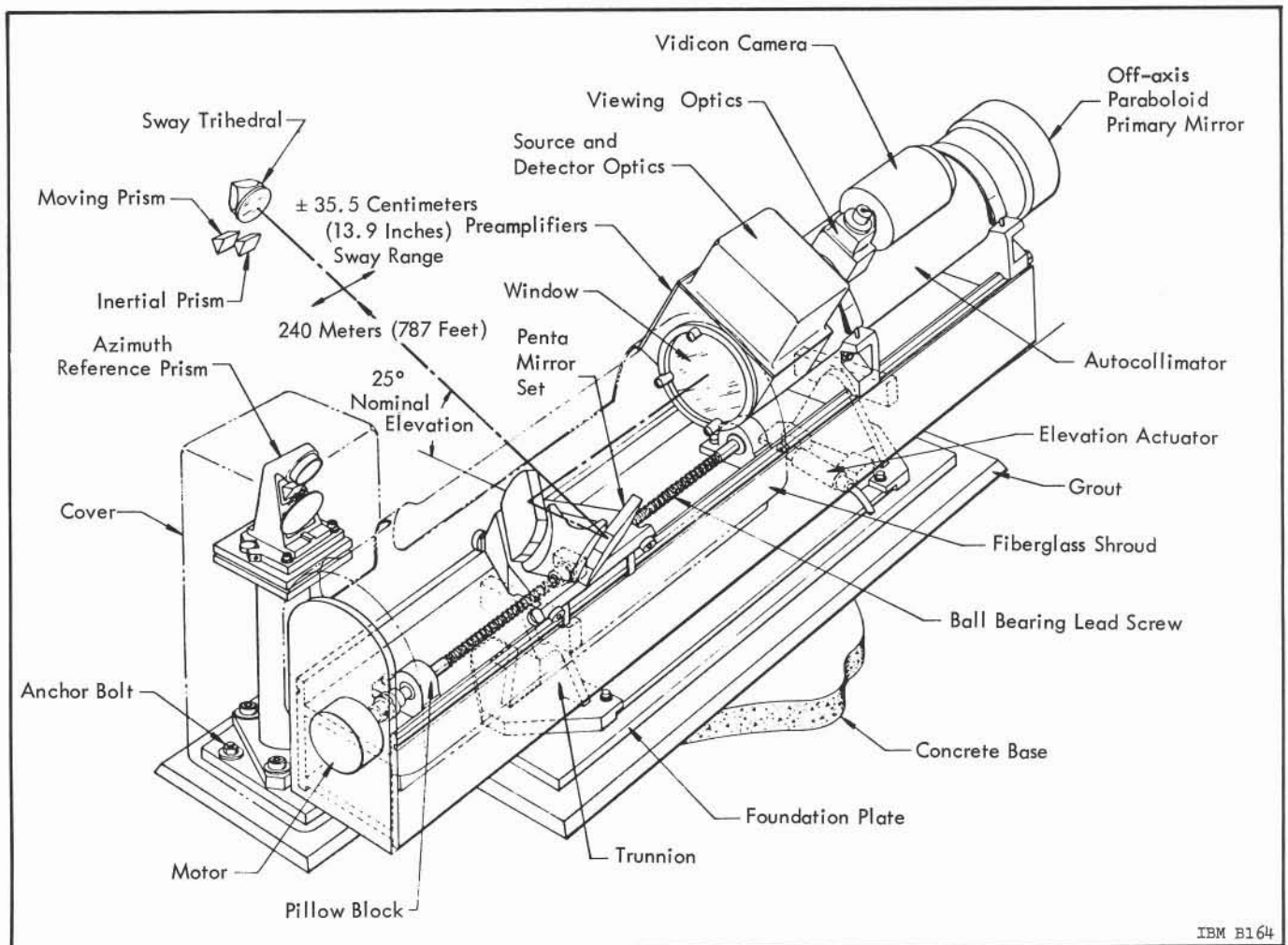
Figure 14. 7-2 Theodolite

azimuth program angle by comparing the baseline azimuth to the stored mission azimuth and torquing the synchro-encoder repeater until the encoder output agrees with the computer. The error signal from the dual-speed synchro system if fed to the X gyro alignment loop and drives the inertial gimbal to the mission azimuth which also nulls the CX:CT unbalance. The synchro prism is held fixed with respect to the optical beam from the theodolite on the baseline azimuth. The inertial gimbal (navigation coordinates) is held on the mission azimuth with the CT acting as an azimuth pickoff. The azimuth angle computation is a function of

the predicted launch time. The computer program translates any deviations from the predicted launch time into a respective azimuth angle change and repositions the inertial gimbal to the changing azimuth through the encoder synchro. The mission azimuth is displayed in the launch control center by a digital monitor.

The analysis of the azimuth alignment error shows the root sum squared system error to be less than ± 20 arc seconds.
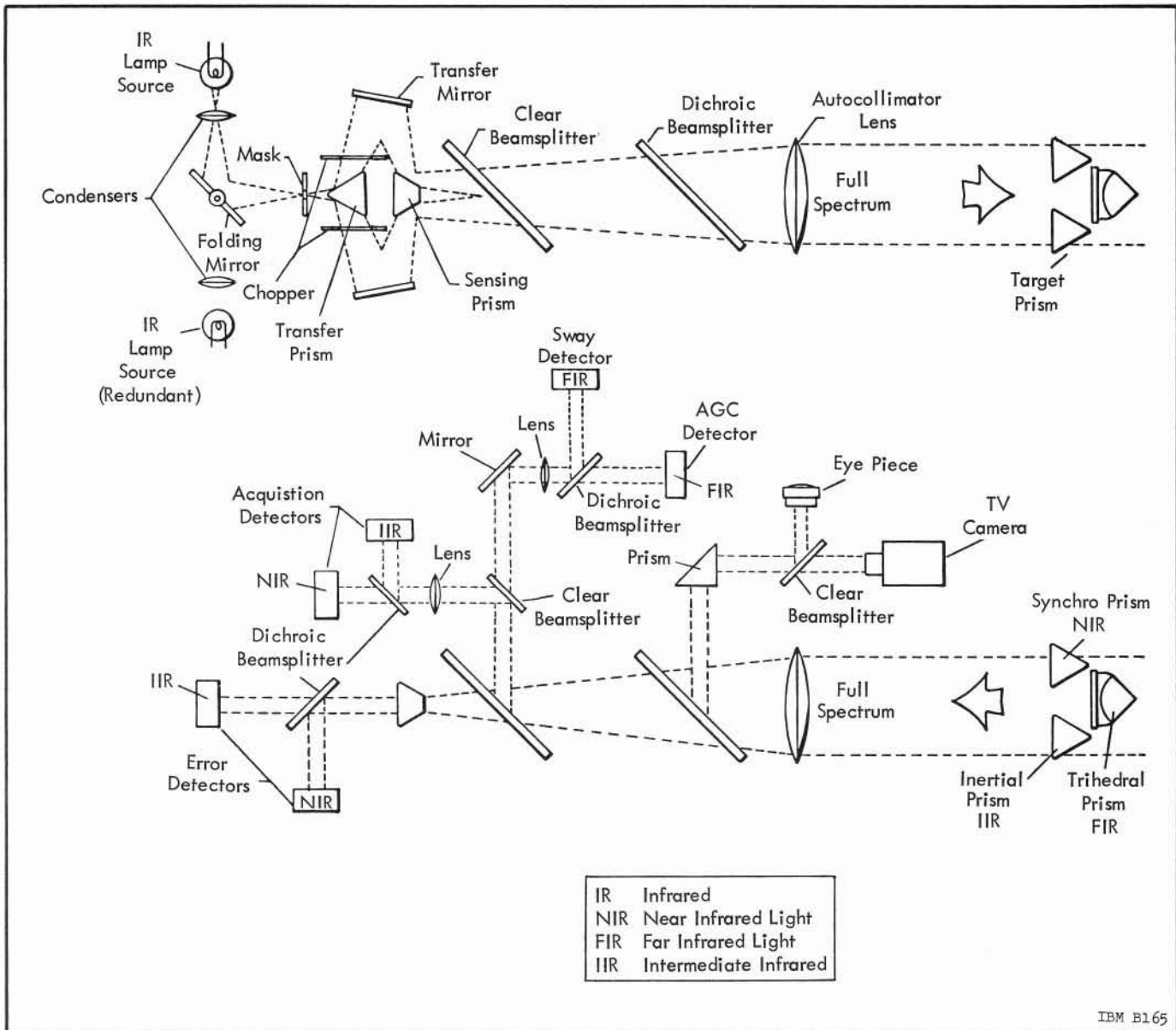
Figure 14. 7-3 Optical Schematic Diagram of SV-M2 Theodolite
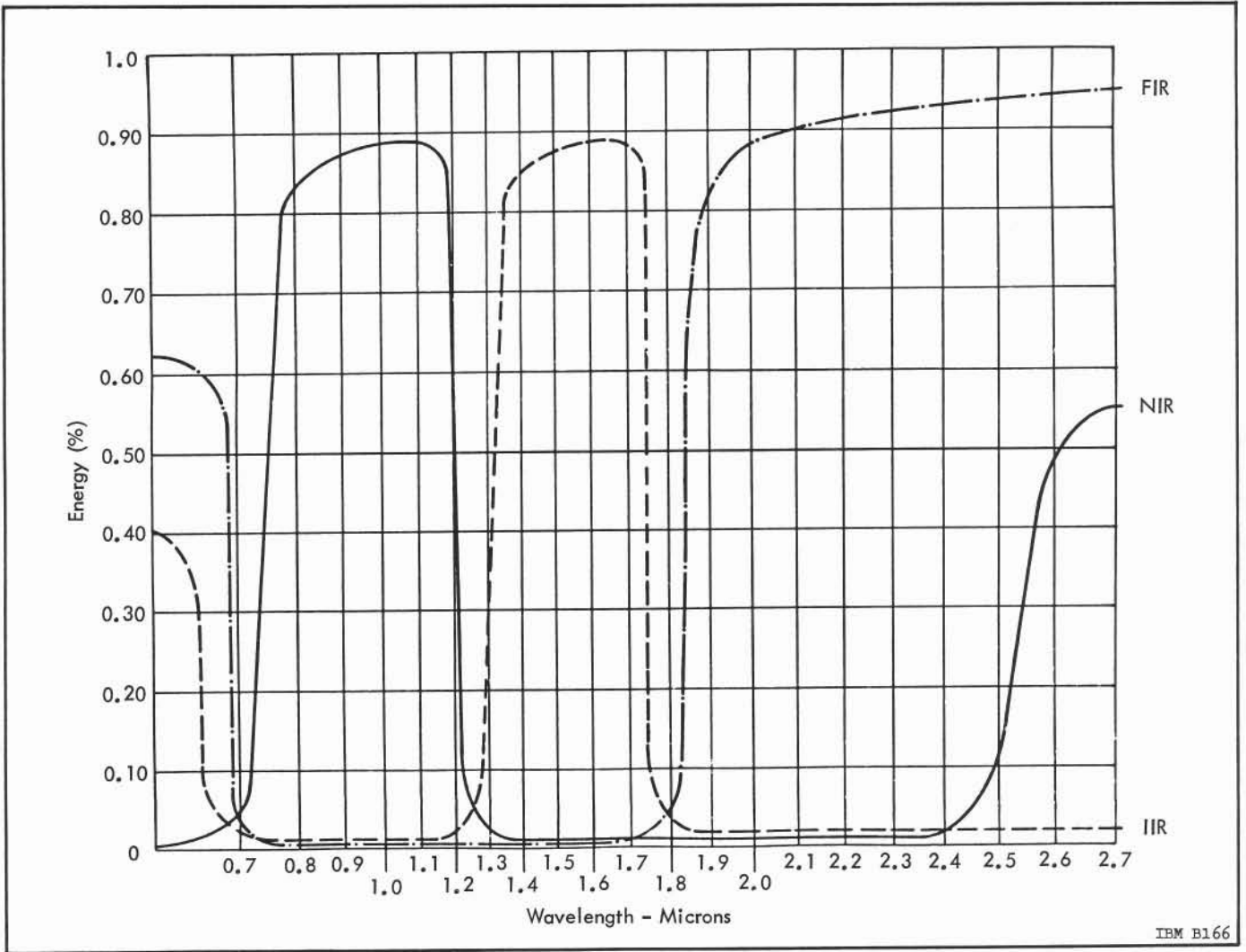
Figure 14. 7-4 Optical Spectrum
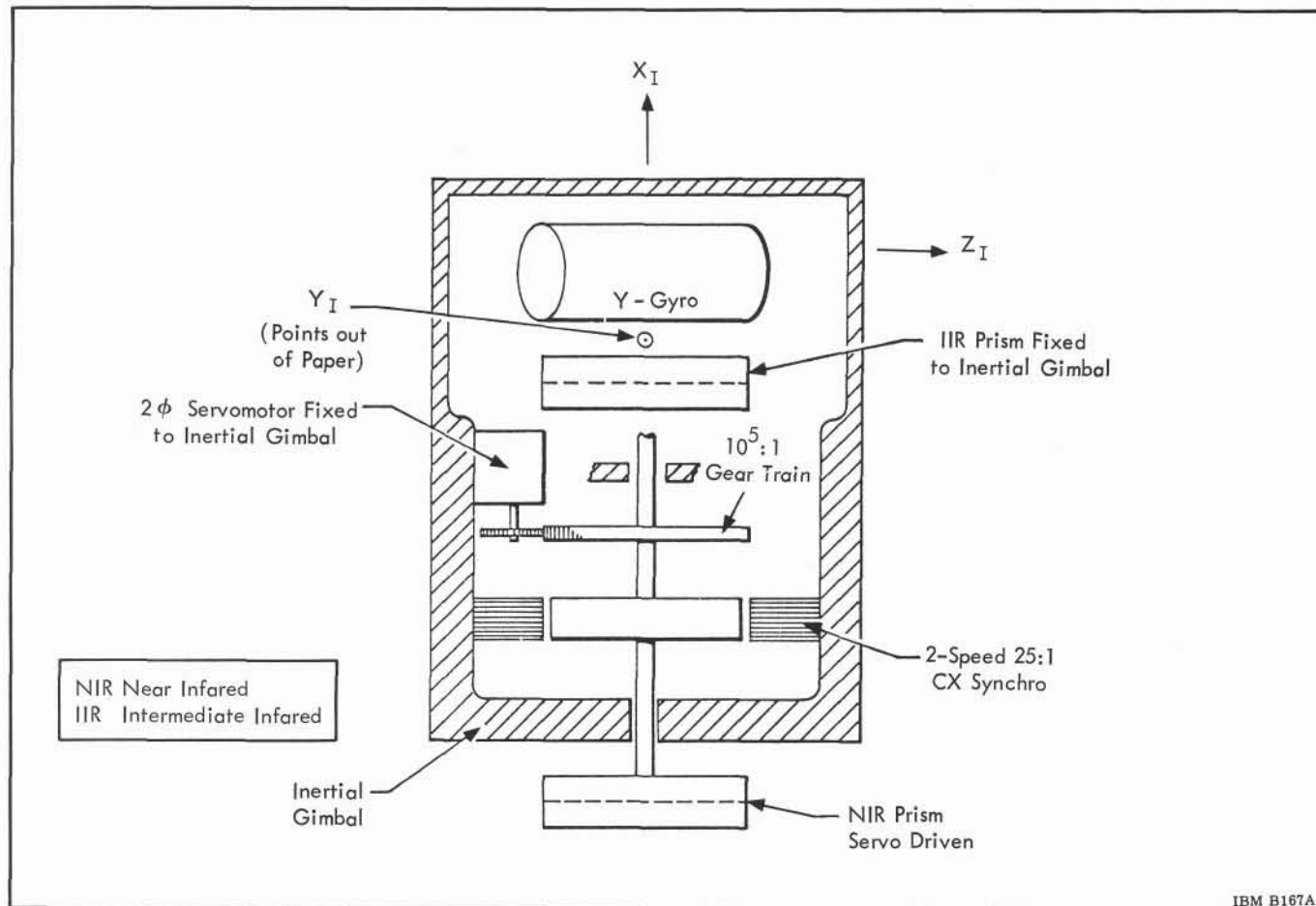
Figure 14. 7-5 Optical Gimbal Laying System

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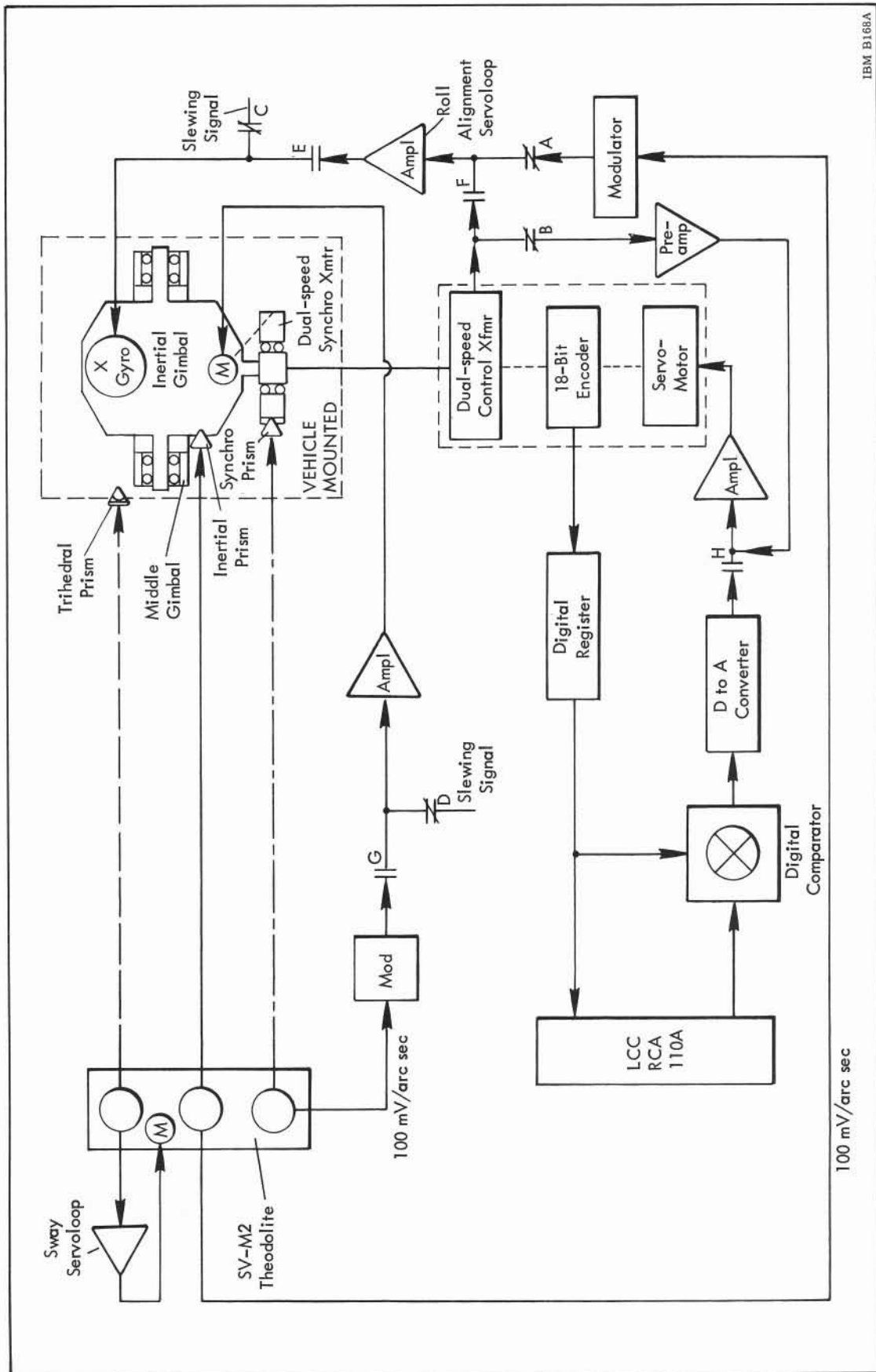
Figure 14. 7-6 Azimuth Alignment Scheme

SECTION 14.8
GAS BEARING GYRO

The ST-124-M Inertial Platform Assembly contains three AB5-K8 gas bearing single-degree-of-freedom gyros. These components measure the vehicle motion; their performance capabilities define the hardware accuracies of the guidance system. No active compensation is used in the LVDC for instrument error terms and absolute tolerances are established for the life of the instruments.

The gas bearing gyro is shown in a cutaway view of Figure 14. 8-1. The cylindrical, externally-pressurized gas bearing suspends the cylinder between the sleeve and endplates, as shown in Figure 14. 8-2, and provides both axial and radial centering. The endplates are bolted to the sleeve and their assembly forms the case of the gyro.

Dry gaseous nitrogen is passed through two rows of 24 holes with millipore discs in the sleeve to act as flow diffusers and provide the bearing stiffness. The gas in the cylinder chamber generates the hydrostatic bearing and flows symmetrically to both endplates, escaping around the hub at each end of the cylinder. The sleeve, endplates, and cylinder are constructed of beryllium with machined tolerances of 5×10^{-4} millimeters (20 microinches) in roundness squareness.

The gyro wheel shown in Figure 14. 8-2 mounts in a yoke of the cylinder endcap. The neck section of the yoke is controlled to minimize the anisoelastic drift of the wheel assembly. The endcap is mounted in the cylinder forming the gyro cylinder assembly. The cylinder is helium filled to reduce windage losses. The wheel is a two-pole, synchronous hysteresis motor. Precision bearings are fabricated to the beryl-

lium shaft with the motor laminations and windings to form the stator of the wheel. The rotor is made up of an elkonite ring with P-6 hysteresis laminations shrink-fitted into the elkonite ring and beryllium endbells which are bolted to each side.

The signal generator and torque generator, shown in Figure 14. 8-2, are coupled to the cylinder by means of a copper-shorted loop which is mounted on the cylinder. The signal generator is an ac-type, shorted-loop, single-winding sensor for the angular displacement of the gyro about its output axis. The torque generator operates like an eddy current motor. It provides only alignment torques for initial erection. The 3-phase, 400-hertz wheel power is transmitted by ribbon flex leads to standoffs on the gyro cylinder. A magnetic shield is placed between the gyro case and the signal generator. Dust covers, which are also magnetic shields, complete the assembly.

The electrical schematic for the AB5-K8 stabilizing gyro is shown in Figure 14. 8-3. The coordinate definition for a single gyro is also shown. The wheel phase rotation A-B-C, American Standard, with the gyro connected to the 26-volt, 400-hertz supply, causes the gyro wheel to rotate in a negative direction about the gyro spin reference (X_{GU}) vector. *

Table 14. 8-1 lists the gyro characteristics.

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* The XYZ coordinates used in this book have been changed to reflect the Project Apollo Coordinate System Standards (SE 008-001-1). This coordinate scheme becomes effective for vehicles 206 and 502 (and subsequent). For further details refer to Section 1. 2.

Table 14.8-1 Gyro Characteristics

Gyro Wheel

Type . . . . Synchronous hysteresis

Angular momentum . . . . 2.6×10^6 g cm^2/s

Wheel speed. . . . . 24,000 rpm

Wheel excitation . . . . 26 V, 3-phase, 400 Hz

Wheel bearing preload . . . . 3.4 kg, operating (7.5 pounds)

Wheel power at sync . . . . 8 W

Wheel life . . . . 3000 hours, minimum

Wheel mount . . . . Symmetrical

Wheel sync time . . . . 90 seconds

Gas Bearing

Gas pressure . . . . 10.3 N/cm^{2}d (15 psid)

Gas flow rate . . . . 2000 cc/min STP
                                                           (122 in.^{3}/min)

Gas gap (one side) . . . . 0.015 to 0.02 cm
                                                           (0.006 to 0.008 in.)

Orifice restrictors . . . . Millipore discs

Sleeve material. . . . . Anodized beryllium

Endplate material . . . . Anodized beryllium

Cylinder material . . . . Anodized beryllium

Signal Generator

Type. . . . . Short turn reluctance

Excitation . . . . 10 V, 4.8 kHz

Sensitivity . . . . 550 mV/° with 10 k load

Float freedom . . . . $\pm 3°{}^{+0°}_{-0.5°}$

Torquer (for platform erection and earth rate bias only)

Type. . . . . Shorted turn reluctance

Normal erection rate. 6°/min

Fixed coil excitation . . . . 26 V, 400 Hz, 45 mA

Maximum variable
coil excitation. . . . . 30 V, 400 Hz, 50 mA

Impedance
     Fixed coil resistance. . . . . 184 Ω

     Fixed coil impedance. . . . . 555 Ω ∠+31°   (400 Hz)

     Variable coil resistance. . . . . 190 Ω

     Variable coil impedance. . . . . 330 Ω ∠+53°   (400 Hz)

Table 14. 8-1 Gyro Characteristics (Cont'd)

Physical Characteristics

    Size

        Diameter........................... 7. 7 cm (3 inches)

        Length............................. 10. 3 cm (4 inches)

    Weight .... 900 grams (2 pounds)

    Mounting .... 3-point flange mounting

Temperature Characteristics

    Calibration temperature.................... 40°C (104°F) (gyro housing)

    Drift versus temperature gradient .... 0. 009°/h/°C

1. Torque Generator	9. Feeder Ports
2. Gas Passage	10. Spin Reference Axis
3. Gas Filter	11. Gimbal Angle
4. Gas Inlet	12. Spin Axis (SA)
5. Gyro Rotor	13. Gimbal
6. Output Axis (OA)	14. Input Axis (IA)
7. Gas Exhaust Port	15. Signal Generator
8. Supporting Gas Film	

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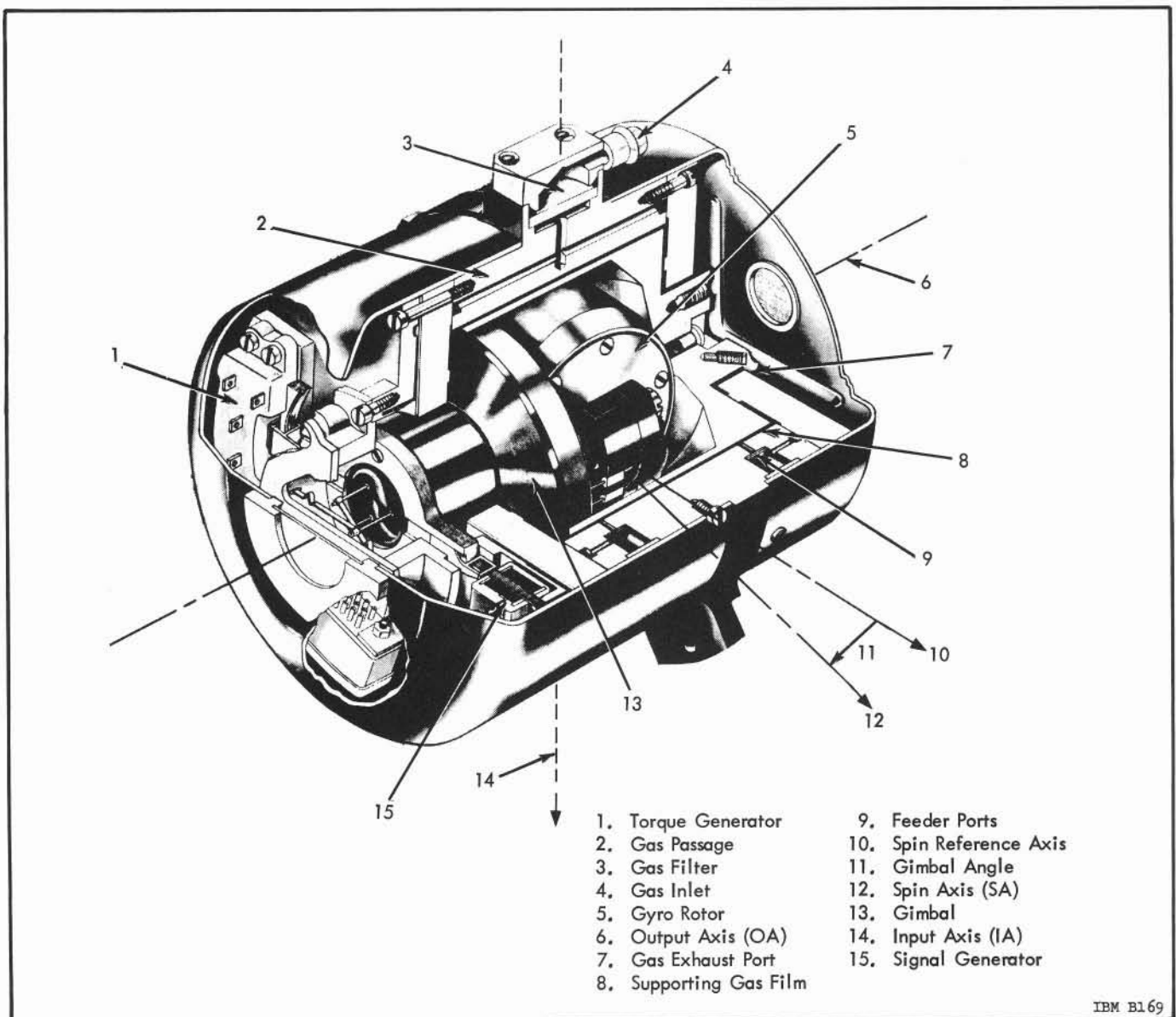
Figure 14. 8-1 Cutaway View of a Single-axis Integrating Gyro
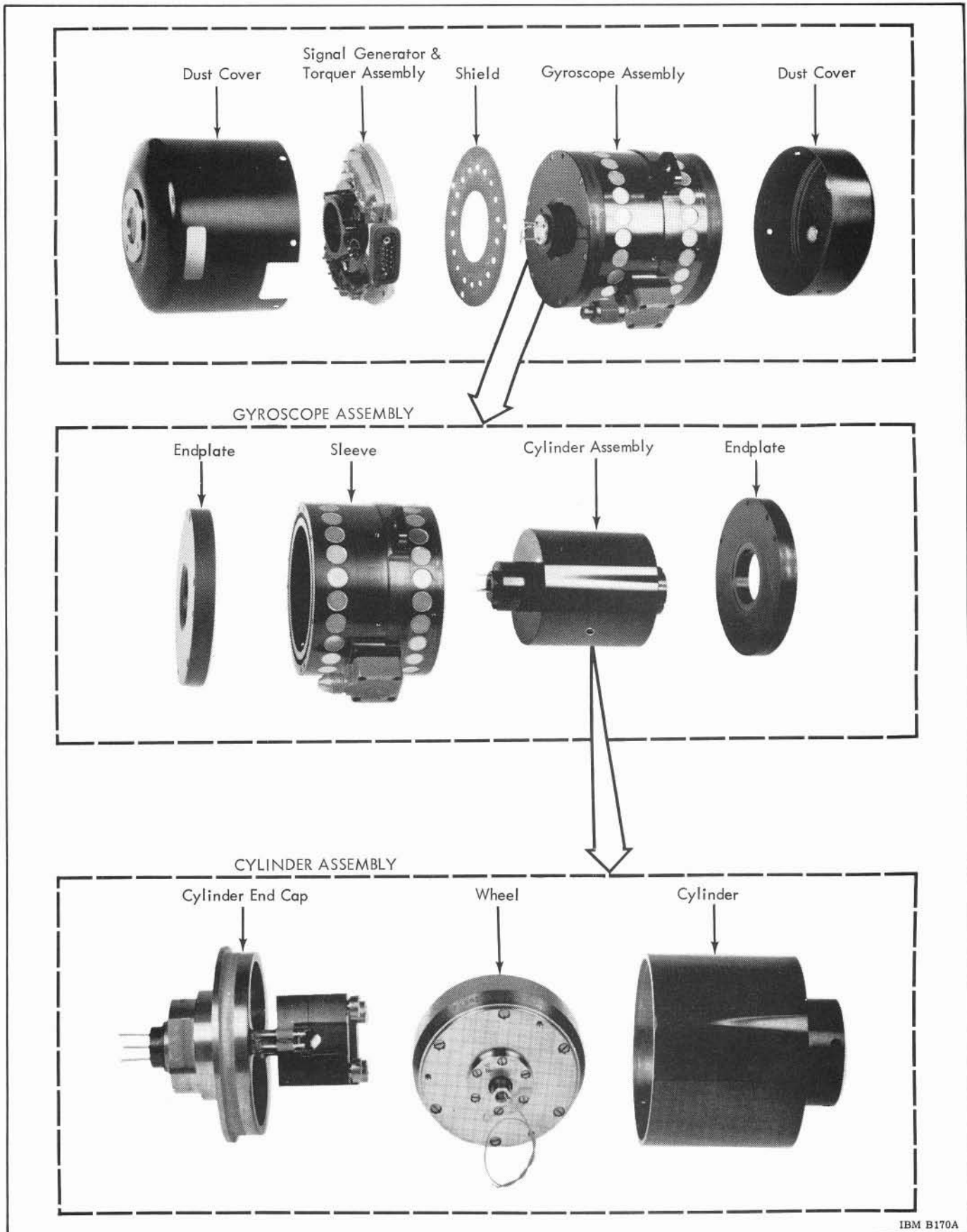
Figure 14. 8-2 Exploded View of a Gas Bearing Gyro
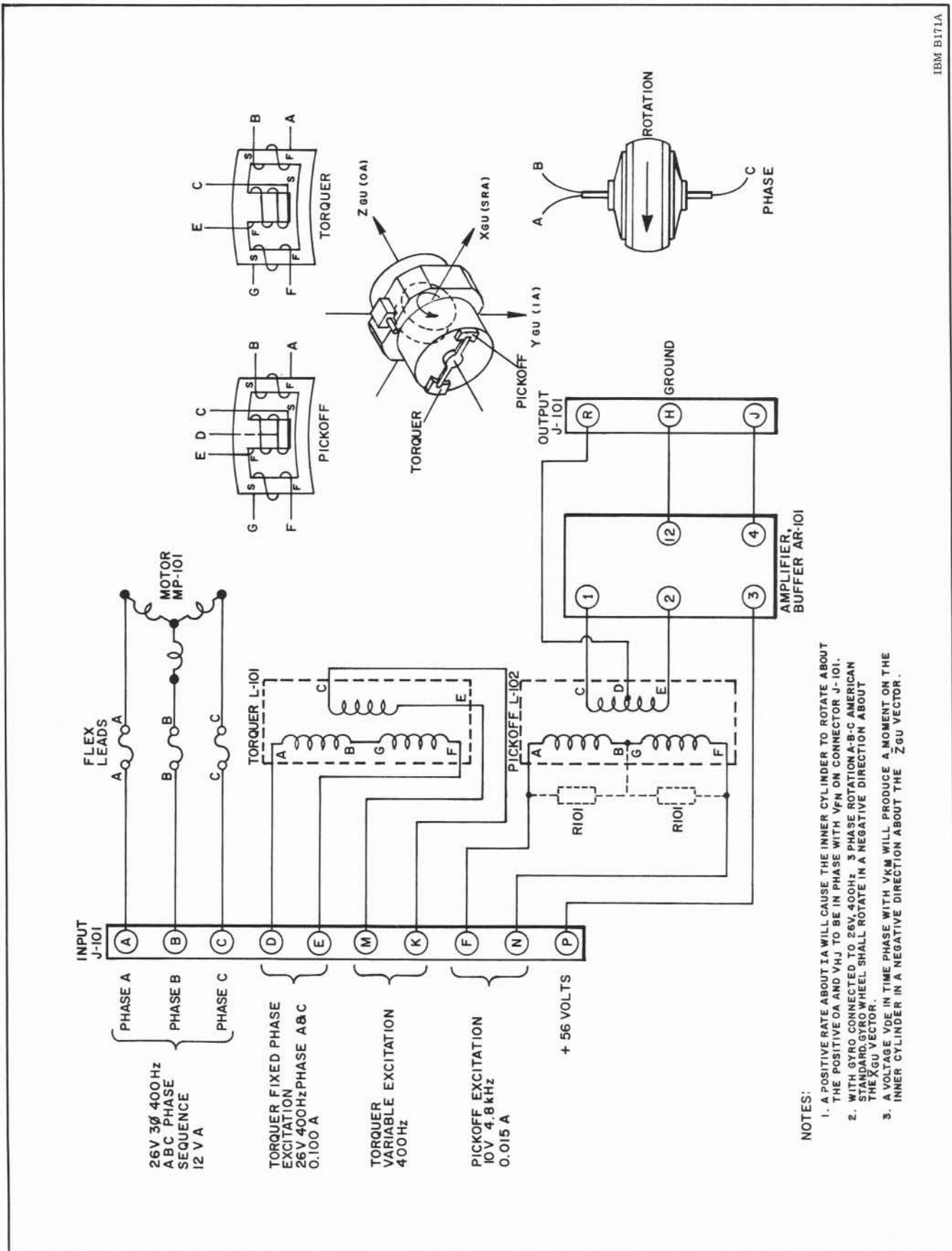
Figure 14.8-3 Electrical Schematic of an AB5-K8 Stabilizing Gyroscope

NOTES:

1. A POSITIVE RATE ABOUT IA WILL CAUSE THE INNER CYLINDER TO ROTATE ABOUT THE POSITIVE OA AND VHJ TO BE IN PHASE WITH VFN ON CONNECTOR J-I0I.

2. WITH GYRO CONNECTED TO 26V, 400Hz, 3 PHASE ROTATION A-B-C AMERICAN STANDARD, GYRO WHEEL SHALL ROTATE IN A NEGATIVE DIRECTION ABOUT THE $\bar{X}$GU VECTOR.

3. A VOLTAGE VDE IN TIME PHASE WITH VKM WILL PRODUCE A MOMENT ON THE INNER CYLINDER IN A NEGATIVE DIRECTION ABOUT THE $\bar{Z}$GU VECTOR.

SECTION 14.9

PENDULOUS GYRO ACCELEROMETER

The AB3-K8 instrument is a pendulous gyro accelerometer; a cutaway view and schematic is shown in Figure 14.9-1 and Figure 14.9-2, respectively.

The accelerometer is a single degree-of-freedom gyro unbalanced about its output axis. The gyro motor and flywheel of the gyro accelerometer are shifted along the spin reference axis to obtain desired pendulosity about the gyro output axis. The pendulous cylinder is machined with a pair of pivots mounted into a frame with a set of Class-7 bearings. Thus, the cylinder is free to rotate about the gyro input axis, which is aligned in the acceleration-measuring direction. A torque is produced by the unbalance or pendulosity which is proportional to the acceleration to which the pendulous mass is subjected. The precession angle of the gyro is proportional to the integral of the acceleration. A signal generator measures the precession angle and a servo closes the loop to a direct-axis torquer which is mounted on the gyro input axis. Thus, the gyro measuring head is stabilized and the pendulous weight is held perpendicular to the input axis. The speed of the measuring head or gyro, with respect to inertial space, is proportional to thrust acceleration along the input axis, and the position of the measuring head is a measure of thrust velocity. An optical incremental encoder is mounted on the input axis and provides a measure of the thrust velocity. The power source for the synchronous spin motor is referenced from a crystal-controlled frequency, guaranteeing a constant accelerometer scale factor, or velocity increment/revolution, about the input axis.

The symmetrical gyro wheel is mounted in a cylinder that is suspended on a hydrostatic gas bearing. The gyro is constructed the same as the AB5-K8 gyro except there are two rows of 18 holes for feeding gas into the bearing. All nonmagnetic parts are machined from beryllium material except the endplates. which are fabricated of Monel to reduce the servoloop nutation frequency.

The electrical schematic for the AB3-K8 accelerometer is shown in Figure 14.9-3. The wheel phase rotation A-B-C, American Standard, with the gyro connected to a 26-volt, 400-hertz, 3-phase power supply, causes the wheel to rotate in a positive direction about the spin reference axis vector.

Table 14.9-1 lists the accelerometer characteristics.

Table 14.9-1 Accelerometer Characteristics

Gyro Wheel	
Type	Synchronous hysteresis
Angular momentum	94,000 g cm^2/second (207 pounds)
Wheel speed	12,000 rpm
Wheel excitation	26 V, 3-phase, 400 Hz
Wheel sync time	90 seconds
Wheel power at sync	4.5 W
Wheel life	3000 hours, minimum
Wheel mount	Symmetrical
Wheel bearing preload	907.2 grams (1.99 pounds) operating

Table 14.9-1 Accelerometer Characteristics (Cont'd)

Gas Bearing

    Gas pressure . . . . 10.3 N/cm^2d (15 psid)

    Gas flow rate . . . . 2400 cc/min STP
                                                 (146 $in.^3$/min)

    Gas gap . . . . 0.015 to 0.02 cm

                                                 (0.006 to 0.008 in.)

    Orifice restrictors . . . . Millipore discs

    Sleeve material . . . . Anodized beryllium

    Endplate material . . . . Monel

    Cylinder material . . . . Anodized beryllium

Signal Generator

    Type . . . . 4-pole shorted turn reluctance

    Excitation . . . . 10 V, 4.8 kHz

    Sensitivity . . . . 285 mV/° with 700 Ω load

    Float freedom . . . . $\pm 3°{}^{+0°}_{-0.5°}$

Torque Motor

    Type . . . . Direct-axis dc torquer

    Maximum torque . . . . 1.44 kg cm at 1.1A and 44V

    DC resistance . . . . 32.6 ohms

    Inductance . . . . 12.5 millihenries

Velocity Pickoff

    Type . . . . Digital encoder (optical grid)

        Count . . . . 6000 count/revolution

        Resolution . . . . 5 cm/s/bit (1.96 in./s/bit)

        Output . . . . Incremental with redundant
                                                  channels

Physical Characteristics

   Size

        Diameter . . . . 8.3 cm (3.25 inches)

        Length . . . . 12.8 cm (5 inches)

    Weight . . . . 1200 g (2.64 pounds)

    Mounting . . . . 3-point flange mounting

Table 14. 9-1 Accelerometer Characteristics (Cont'd)

Performance

    Accelerometer scale factor. . . . . 300 m/s/revolution of output axis
    (985 feet/s/revolution)

    Pendulosity . . . . 20 g cm

Temperature Characteristics

    Calibration temperature . . . . 40°C ambient (104°F)
    (on housing gyro)

    Ambient temperature range for
    required accuracies . . . . 40°C ± 3°C (104°F ± 5. 4°F)

1. Gas Inlet
2. Encoder Discs
3. Encoder Electronics
4. Supporting Gas Film
5. Servo Pickoff
6. Gyro Output Axis
7. Torquer
8. Gyro Input Axis
9. Accelerometer Measuring Direction
10. Slip Ring Capsule
11. Feeder Ports
12. Spin Reference Axis
13. Gyro Rotor
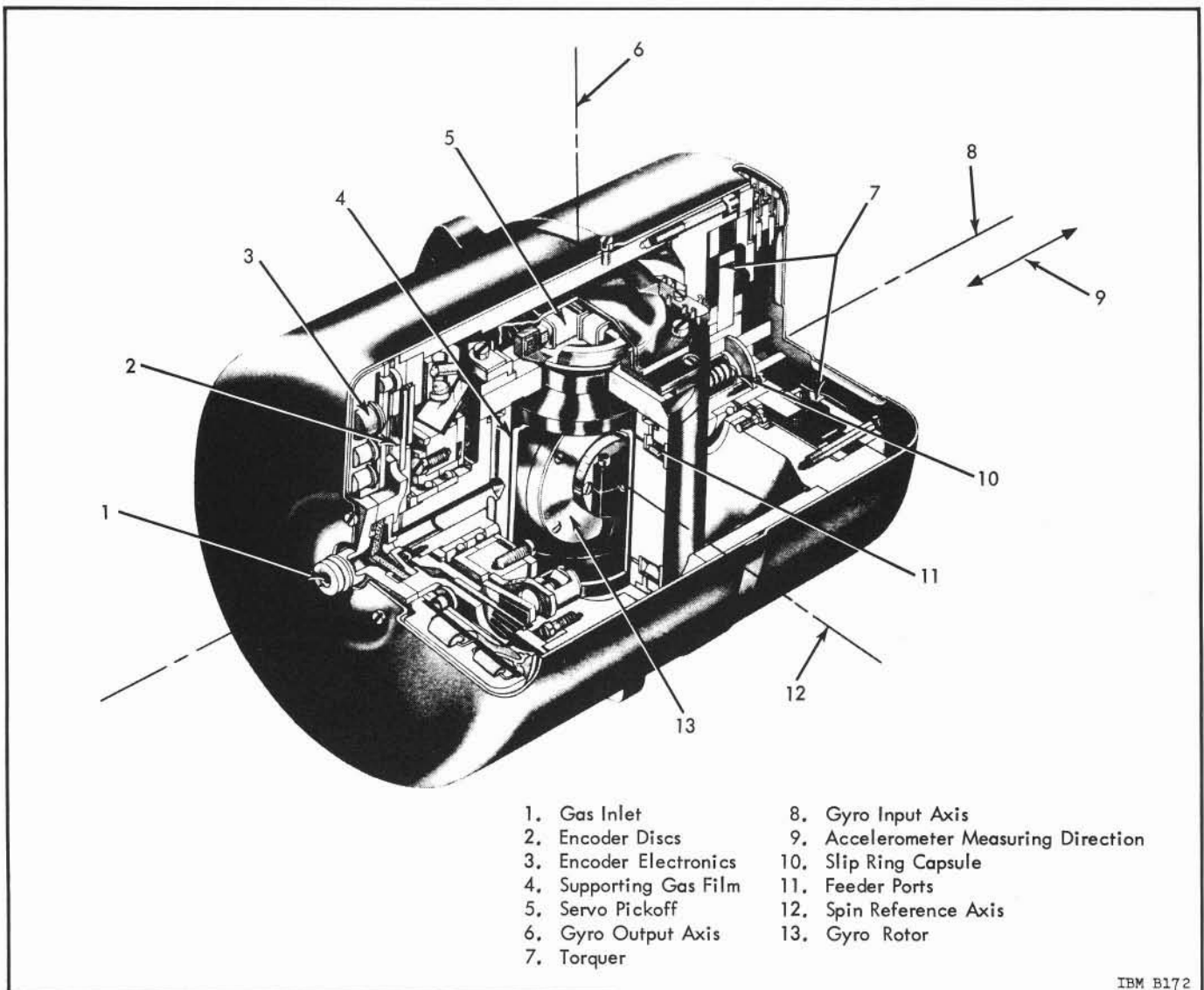
IBM B172

Figure 14. 9-1 Cutaway View of a Pendulous Integrating Gyro Accelerometer
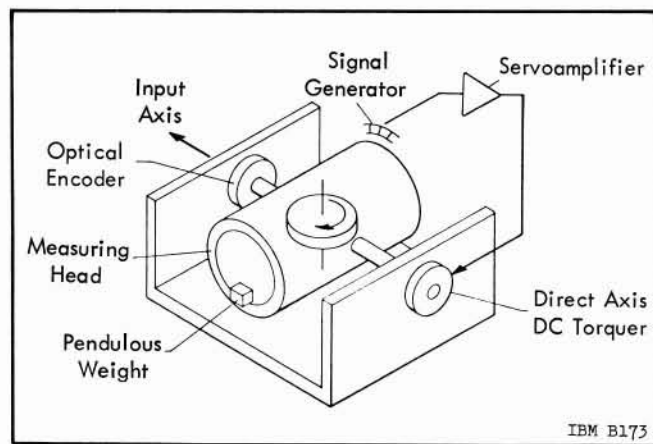
Figure 14.9-2 Pendulous Integrating Gyro
Accelerometer Schematic

IBM B174A

NOTE:
The XYZ coordinates used in this book have been changed to reflect the Project Apollo Coordinate System Standards (SE 008-001-1). This coordinate scheme becomes effective for vehicles 206 and 502 (and subsequent). For further details refer to Section 1.2.
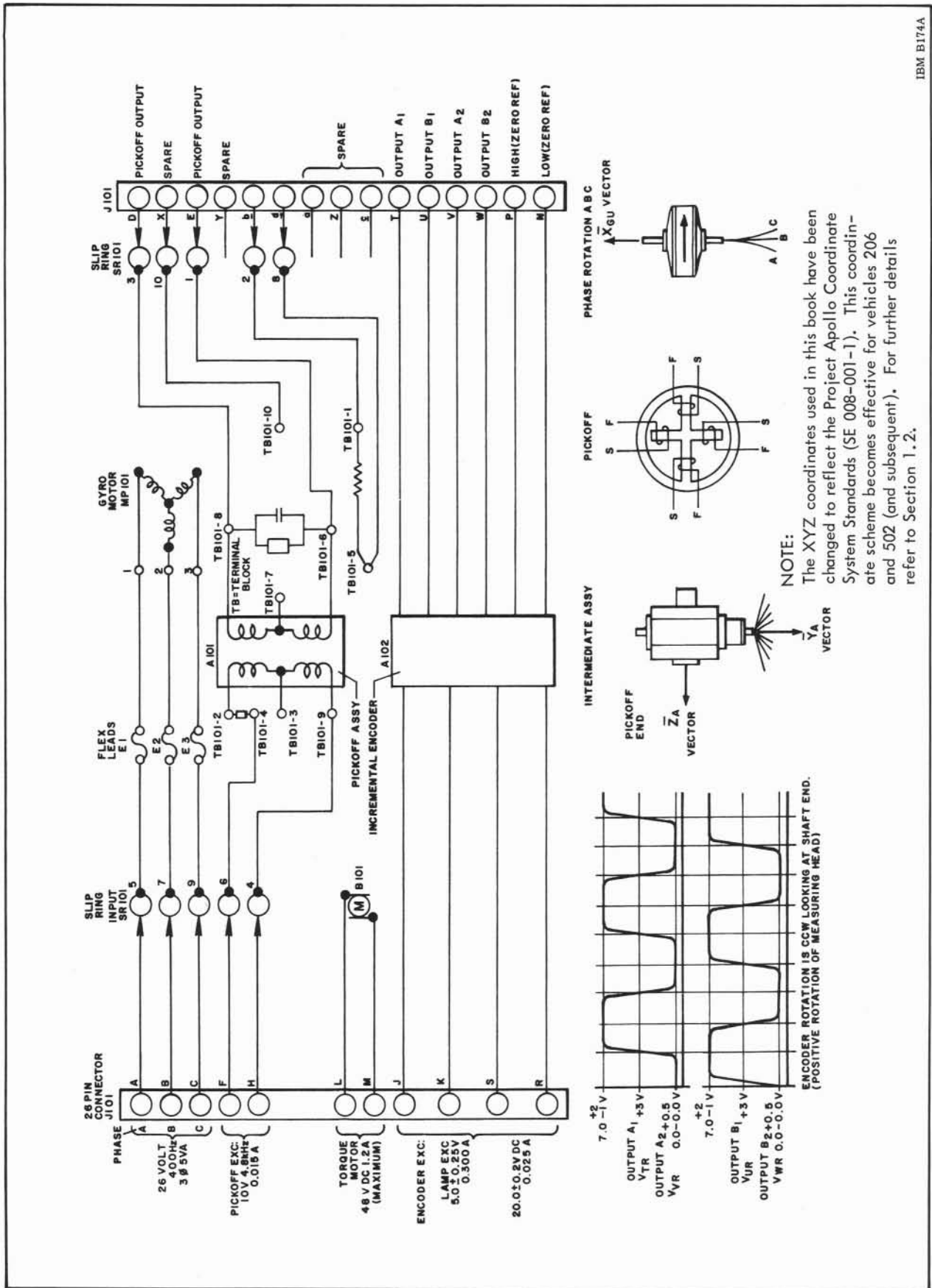
Figure 14. 9-3 Electrical Schematic of an AB3-K8 Acceleometer

SECTION 14.10
POWER AND GAS REQUIREMENTS

The input power requirements and the expected heat dissipation of the system assemblies are shown in Figure 14.10-1. The input power figures show the normal (quiescent) requirements and the peak (maximum disturbing forces and g loading) requirements.

INERTIAL PLATFORM HEATERS

The Inertial Platform has 150 watts of resistive heating with the application of 115 volts, 60 hertz, or 400 hertz. The heaters are used to assist the Inertial Platform to reach its optimum operating temperature. Thermostats, located inside the platform, control the heaters and turn them off when the operating temperature is reached. The platform component heat dissipation maintains this temperature. The heaters will also be available if inflight environment dictates the necessity for heating.

Inflight platform temperature is controlled by circulating a constant-temperature fluid through ducts in the platform covers. The exterior of the Inertial Platform is painted with aluminum to provide an emissivity of approximately 0.4, thus providing a near-constant temperature radiating surface. Blowers will be utilized to circulate the internal gaseous nitrogen and maintain normal temperature gradients across the gimbals.

Gaseous nitrogen is supplied from a 0.056 cubic meter (two cubic feet) storage reservoir, pressurized to 20.7×10^6 N/m^2 (3000 psi), and regulated to 10.3×10^4 N/m^{2}d (15 psid) supply to the Inertial Platform. Temperature conditioning of the gas will be from the water-methanol IU coolant system. The Inertial Platform requires 1.4×10^{-3} cubic meters per minute (0.5 cubic feet per minute) of gaseous nitrogen.

Table 14.10-1 lists the power supply specifications.

Table 14.10-1 Power Specifications

28 Vdc	
Voltage regulation	± 2.0 Vdc
Ripple content	0.5 V
Normal current	9 A
Peak current	11 A
56 Vdc	
Voltage regulation	± 3 V
Ripple content	0.25 V
Normal current	1.0 A
Peak current	6.0 A
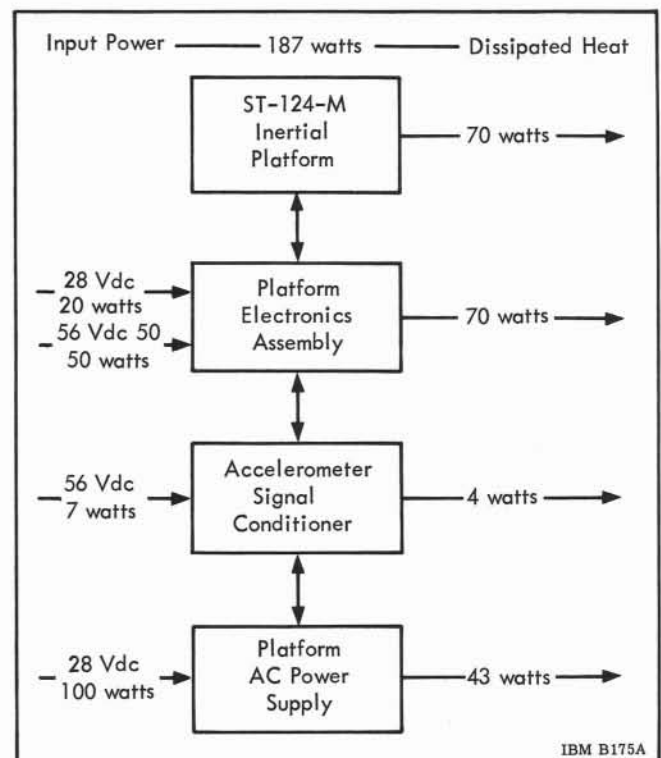
Figure 14.10-1 System Power Requirements and Heat Dissipation

SECTION 15.1

INTRODUCTION

The LVDA and the LVDC together contain the equipment and perform the functions which usually represent an electronic computer system. The LVDA is the input/output device of the LVDC. Any signal to or from the computer is routed through the LVDA. In addition, the LVDA serves as central equipment for interconnection and signal flow between the various systems in the Instrument Unit. The LVDA converts signals from digital to analog form and vice versa. The separation into LVDA and LVDC offers several advantages, the 2 boxes, each having approximately half the size and weight of a combined unit, can be mounted much easier in the vehicle. Also, flexibility is gained since functional or operational changes in the Saturn Astrionics System will essentially affect the LVDA only and can be implemented without changing the LVDC.

The LVDA and the LVDC are involved in the following main operations:

Prelaunch checkout
Navigation and guidance computations
Vehicle sequencing
Orbital checkout

The interconnections between LVDA, LVDC, and other Astrionics equipment is indicated in Figure 15.1-1.

The LVDC contains the logic circuits, the memory and the timing system required to perform arithmetic operations. The LVDA houses circuitry for temporary storage and conversion of data for communication with the LVDC and the power supplies for the LVDA and the LVDC.
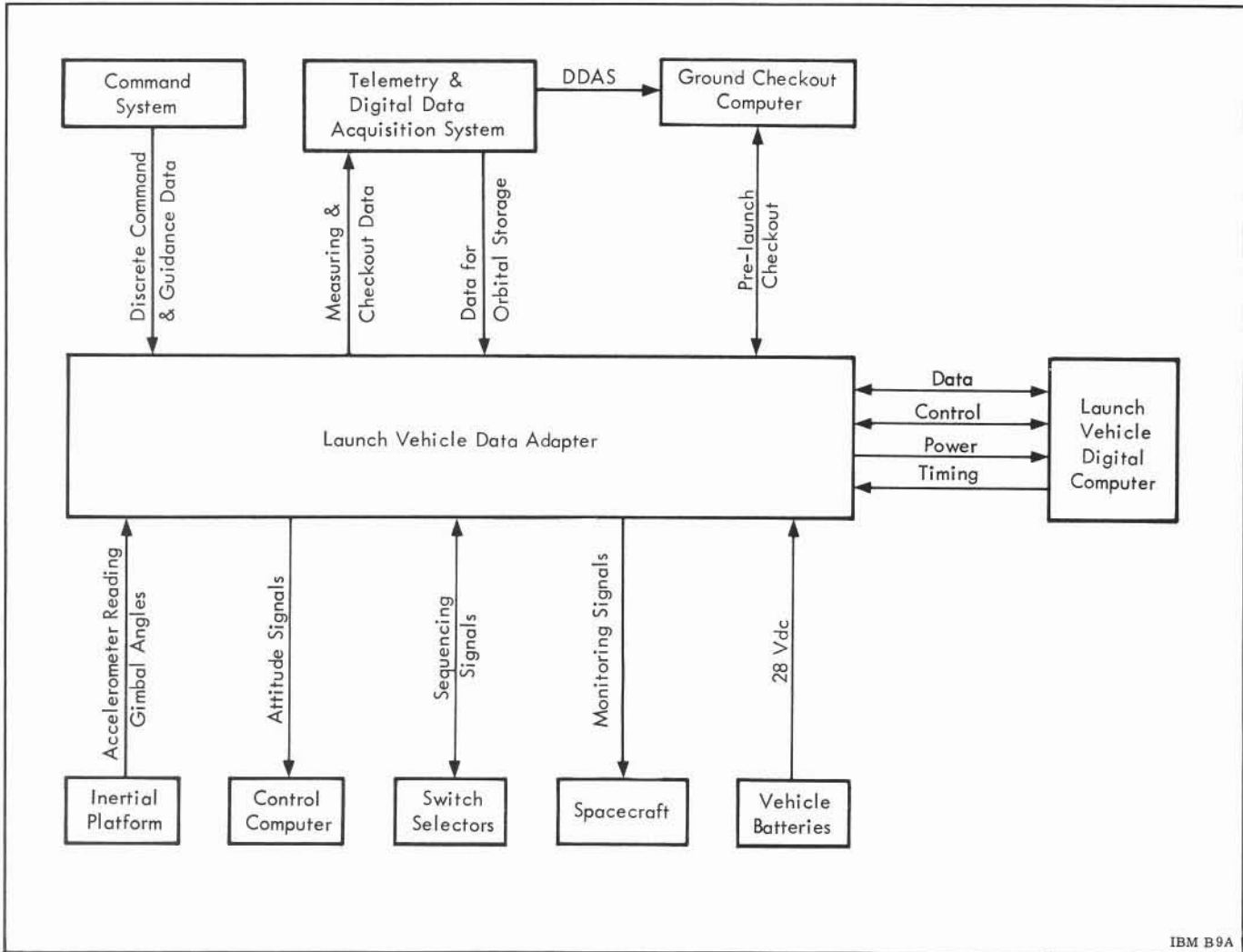
Figure 15.1-1 Connections Between LVDC, LVDA, and the Astrionics System

non-cycle times. These failures are combined into 2 types, "on current failures" and "no current failures" by the error-detector sensing circuits. Timing signals which enable the error monitors to sense the different types of errors at the appropriate times are common to both error monitors.

Information flows from the memory buffer registers in 2 directions, through the inhibit drivers into the memory and into the computer to be used. The memory select latches provide separate control over data flowing in each direction. Separate control is maintained so that errors occurring in the non-selected memory can be corrected from the selected memory without changing the selection of the memory buffer which is providing information to the computer.

15.4.5 COMPUTER INSTRUCTIONS

The computer uses a complement of 18 single-address instructions which are composed of a 4-bit operation code and a 9-bit operand address. The 4-bit operation code can select 1 of 16 different instructions to be executed; this range is extended to 18 by grouping 3 instructions under one operation code, then using bits A8 and A9 of the address to discriminate between them. Bits A8 and A9 serve no other function for the instructions which are grouped. A list of the 18 instructions available is given in Table 15.4-2 with the operation code and a brief description of each.

The 9-bit operand address permits selection of 512 memory addresses for use as operands or data storage locations. The memory is divided into a number of sectors, each containing 256 addresses. Address bits A1 through A8 select one of the addresses within the sector. Bit A9 determines whether the address will be in a sector previously selected by the program or in a special sector called "residual memory". Consequently, bit A9 is called the residual bit; residual memory is selected when A9 is a "1".

Instructions which do not require that data be read from the memory frequently use the operand address for special purposes. These special purposes are pointed out in the List of Instructions, Table 15.4-2, where they occur. For example, the shift instructions (SHF) always manipulate the contents of the accumulator and thus requires no operand address; this frees the operand address for use as shift control. One instruction which makes extensive use of its operand address is PIO. A complete breakdown of the PIO addresses is given in Table 15.4-3.

INTERRUPT

A limited program interrupt feature is provided to aid the input-output processing. An external signal can interrupt the computer program and cause a transfer to a subprogram. A computer interrupt occurs when the instruction in progress is completed. All interrupts automatically generate a HOP 400 (octal) instruction. The HOP constant designates the start of the subprogram. Storage of the contents of the instruction counter, sector and module registers, syllable latch, accumulator, and product-quotient registers must be accomplished by the subroutine itself. These values are stored for an eventual return to the interrupted program. Protection against multiple interrupts and interrupts during MPY and DIV operations is also provided.

The interrupt signal may be generated by a timed source. The rate at which it is generated is controlled by changing the magnitude of a number which is being continually summed. When the summed number reaches a predetermined value, the interrupt signal is generated. This is accomplished in the LVDA.

The main program can be resumed by addressing the contents of residual memory word 777 with a HOP instruction.

Certain discrete input signals are allowed to cause an interrupt. These are useful in causing the input-output subprogram to give immediate attention to an input or output operation.

The interrupt feature facilitates the timing of input-output operations by causing a transfer to an input-output subprogram. The interrupt signal is generated in the LVDA and may be set to interrupt at the highest rate at which any input-output quantity must be handled. This method avoids the necessity of keeping track of time expired since last entering the input-output subprogram.

The automatic interrupt also makes it possible to permit certain discrete inputs to cause an interrupt. Allowing discrete inputs to interrupt makes it possible to demand that the program give attention to an important discrete input. Communications between the LVDC and the vehicle telemetry monitoring system are thus facilitated.

Table 15.4-2 List of Instructions

Instruction	Operation Code 4 3 2 1	Description
HOP	0 0 0 0	Transfers program to memory location specified by HOP constant and controls simplex/duplex operation of memory. Operand address specifies memory location of HOP constant used to load the registers shown in HOP constant format (below). Full HOP constant MUST be specified each time. First instruction following HOP comes from new location.

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26
I M 1	D U P I N	D S 4	D S 3	D S 2	D S 1	D M 3	D M 2	D M 1	D U P D N	0	A 8	A 7	A 6	A 5	A 4	A 3	A 2	A 1	S Y L	I S 4	I S 3	I S 2	I S 1	I M 3	I M 2

Instruction	Operation Code 4 3 2 1	Description
MPY	0 0 0 1	Multiplies contents of memory location specified in operand address by contents of accumulator. Uses 24 high-order bits of each operand to form 26-bit product. Product of multiplicand and 12 low-order bits of accumulator is available by addressing 775 with second instruction following MPY; final product is available to fourth instruction following MPY. Program continues while MPY is in progress; concurrent use of accumulator is permitted.
SUB	0 0 1 0	Subtracts contents of memory location specified in operand address from contents of accumulator. Places remainder in accumulator.
DIV	0 0 1 1	Divides contents of memory location specified by operand address into contents of accumulator. The 24-bit quotient is available to the eighth instruction following divide by addressing the P-Q register (775). Program continues while DIV is in progress; concurrent use of accumulator is permitted.
TNZ	0 1 0 0	Conditional transfer. Transfers operand address (A1-A8) to instruction counter and A9 to syllable select if accumulator contents are not zero. Next instruction comes from new syllable and address. If accumulator is zero, perform next instruction in sequence.
MPH	0 1 0 1	Multiplies contents of memory location specified in operand address by contents of accumulator. Uses 24 high-order bits of each operand to form 26-bit product. Holds up program until multiplication is complete. Product is available from accumulator or P-Q register with instruction following MPH.
AND	0 1 1 0	AND's contents of memory location specified in operand address with contents of accumulator. Result is placed in accumulator.
ADD	0 1 1 1	Adds contents of memory location specified in operand address to contents of accumulator. Sum is placed in accumulator.

Figure 15.5-1 LVDA-LVDC Interconnection Block Diagram
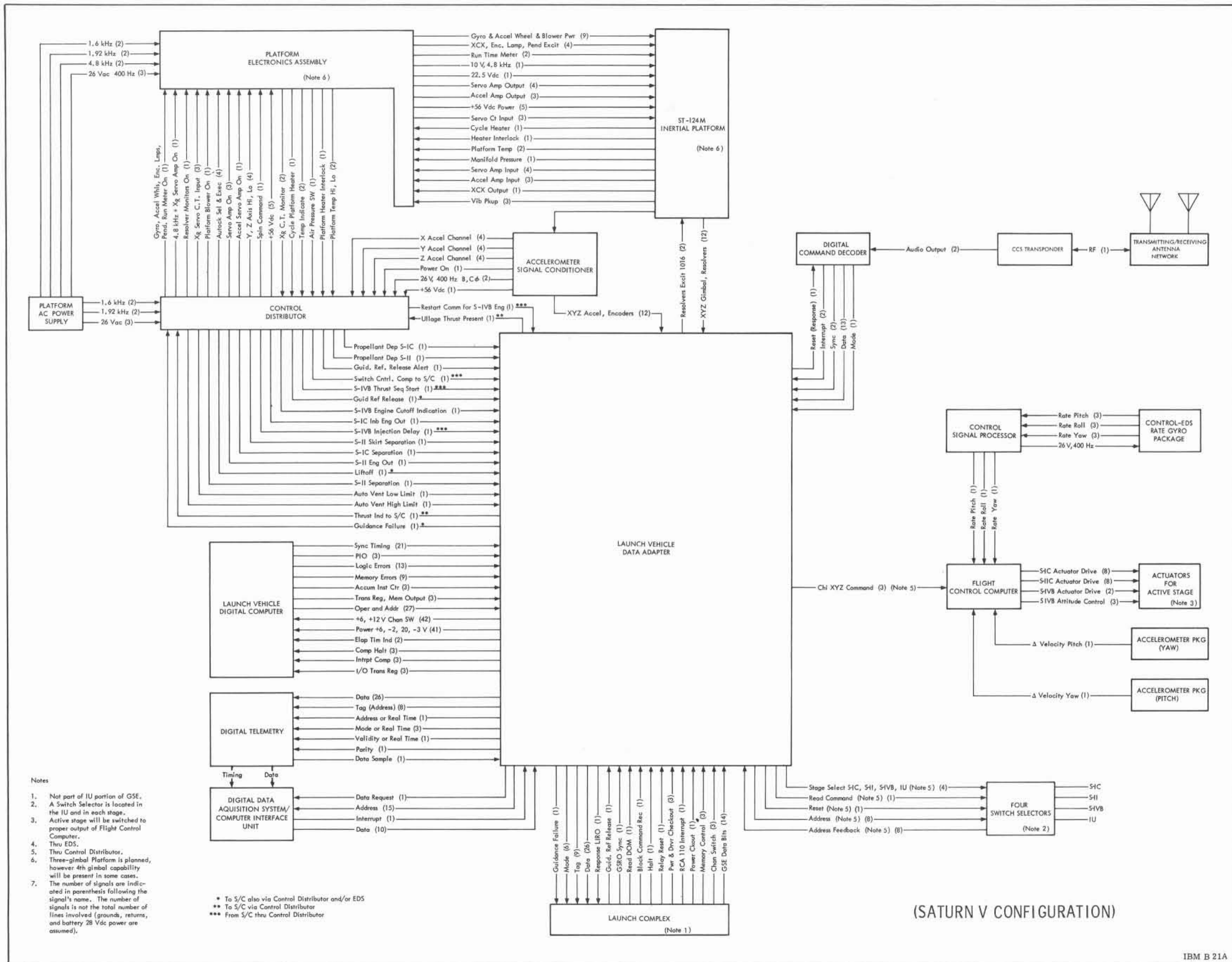
Figure 15.5-2 LVDA-IU Equipment Interconnection Block Diagram

CIRCUIT DESCRIPTIONS

For the following description of various circuits in the LVDA, refer to Figure 15. 5-3.

<u>Delay Lines</u>. The LVDA contains three glass delay lines which serve as temporary storage. The delay lines will be connected together as follows:

- Three delay lines are used in a redundant manner for normal LVDA operation.

- One delay line is used for telemetry operations.

The triple modular redundant delay line is organized around the LVDC timing such that the information it contains remains synchronized with the LVDC operation cycle. The total circulation time of the delay line and its associated electronics is equal to the basic LVDC operation cycle time of 81. 9 microseconds (42-bit times). The delay line is divided into three 14-bit word times corresponding to the three LVDC phase times. Furthermore, the four clock times into which each LVDC bit time is divided are used to time-share the delay line among 4 "channels" of 512-kilohertz serial information. Hence, twelve 14-bit words can be stored in a single delay line by operating the line at a 2. 048-megahertz rate. Table 15. 5-3 illustrates how these word locations are used.

In performing a PIO operation, the LVDC sends out or looks for information only during phase B and C times. Real time is assigned to a phase A word time to facilitate the use of real-time information in the data output multiplexer. However, real time is made available to the LVDC during phase B via the digital input multiplexer register and the serial latch.

The velocity accumulations, which are the processed outputs of the accelerometer optisyns, are arranged to provide duplex redundancy, matching the duplexed optisyns. One line contains outputs Z_1 and X_2, another X_1 and Y_2, and a third Y_1 and Z_2. When the LVDC calls for a given velocity accumulation, it receives the processed output of one of the optisyns on the selected accelerometer during phase B and the outputs of the other optisyns on the same accelerometer during phase C. These 2 values are processed separately in the LVDC such that any one of the delay lines or any optisyn could fail without failing the system.

The real-time accumulation is voted upon in TMR voters during every circulation; thus the values in all 3 lines will always agree. The duplex operation of the accelerometer processors does not allow voting, so there is no guarantee that the absolute value of the 2 readings will agree. Real time is accumulated in 246. 1-microsecond increments, while the least significant bit in the velocity measurement has a weight of 0. 05 meter per second.

The delay line channels with data written at X (channel 3) and W (channel 2) clock times are used to time two of the functions in the computer program. Both of these functions occur during phase B. Time-to-go until the next LVDC interrupt for Switch Selector operations is counted down in channel 2. Time remaining before the start of the next minor loop is counted down in channel 3. These 2 countdowns occur at a rate of one count every 492. 2 microseconds, and will generate an interrupt when they pass through zero. The length of the count will be determined by the LVDC,

Table 15. 5-3   Use of Word Locations in the Delay Line

Channel Number	Write Time	Phase Time			Digital Computer Use
		Phase A	Phase B	Phase C	
1	Z clock	Spare	Spare	Interrupt storage	Read channel
2	W clock	Spare	Switch select or interrupt countdown	Interrupt limiting	Write channel
3	X clock	One-millisecond countdown	Minor loop interrupt countdown	Interrupt inhibit	Write channel
4	Y clock	Real-time accumulation	Velocity accumulation Z_1, X_1, or Y_1	Velocity accumulation X_2, Y_2, or Z_2	Read channel

which loads a value of time-to-go to initiate each count. Also, during phase A of channel 3, a 0. 8-millisecond delay for use in the digital-to-analog converter is generated by counting 10 circulations of the delay line.

LVDC interrupts are stored during phase C of channel 1. Once the LVDC recognizes an interrupt, it will set the corresponding bit in phase C of channel 2, and will reset this bit in channel 1. The associated circuit will prevent a new interrupt from being recognized in this bit position until the previous interrupt has disappeared. The only constraint, therefore, on the length of the interrupt signal will be that it must last for at least 81. 9 microseconds, the circulation time through the delay line and associated circuits. Interrupts may be stored without being recognized by setting the corresponding bits in phase C of channel 3.

Channel 1 may be conveniently read by the computer, while channel 2 may be conveniently written into from the computer. As many as 3 of the normal 14 bits may be sacrificed if it is desired to use either of these 2 channels in the opposite manner.

Interrupt Register. As a means of notifying the LVDC that immediate attention be given to an external operation, an interrupt line is wired from the LVDA to the LVDC. The interrupt register (delay line) is capable of accepting 12 different signals and storing them until the LVDC has acted upon them. The signals are OR'ed together so that only one interrupt line to the LVDC is required. After an interrupt, the LVDC branches to a subroutine to read the interrupt register by means of a PIO operation. A computer analysis is then made, testing the highest priority bit positions first in case more than one interrupt signal is stored in the register. During this testing, the LVDC stores the contents of the memory address register and the instruction counter and branches to an interrupt subroutine. While in this subroutine, the LVDC does not recognize further interrupts. The next to last instruction of the testing the highest priority bit positions first in case more than one interrupt signal is stored in the register. During this testing, the LVDC does not recognize further interrupts. The next to last instruction of the interrupt subroutine is a PIO instruction addressed to the interrupt register to reset the particular bit causing the interrupt. The hardware provides a time delay to prevent further immediate interrupts from the same source. The source must disappear and return before another interrupt is honored from that source. This prevents slow-acting devices such as relays from regenerating interrupts while they are being activated by discrete outputs which occur during the interrupt

subroutine. Each interrupt signal must be at least 84 microseconds in duration to assure storage in the delay line.

The LVDC is also capable of inhibiting the interrupt, as commanded by the program, with PIO instructions whenever the function of the subroutine warrants this precaution. Examples of interrupts are:

- An interrupt which is timed to ensure regular processing of guidance data;

- An interrupt from the CIU indicating that requested data are available.

Nine of the interrupt signals will be generated external to the LVDA and LVDC and three will be generated internal to the LVDA and LVDC.

Interrupt Circuits. The operation of the interrupt circuits is as follows. Upon receipt of an interrupt signal, the LVDA will insert a bit in a time slot reserved for this particular interrupt in a TMR delay line register called the interrupt storage register. When the bit is placed in the register, the LVDA signals the LVDC that an interrupt has been received. Upon receipt of the LVDA interrupt signal, the LVDC will read the interrupt register and determine which interrupt has been received. The LVDC will then insert a bit in the corresponding time slot in another delay line register, called the interrupt limiting register, to indicate that the LVDC has acknowledged that particular interrupt. The logic of the interrupt storage and inhibit registers will be so arranged that once a particular interrupt appears, at the LVDA interface and interrupts the LVDC, it will not be capable of causing another interrupt unless it disappears and reappears again. Another delay line storage register, called the interrupt inhibit register, is provided. When the LVDC places a bit in a particular time slot, this bit will prevent the corresponding bit in the interrupt storage register from interrupting the LVDC. Any combination of bits may be placed in the interrupt inhibit register.

Switch Selector Register. This register is loaded by the LVDC whenever the computer wishes to give commands to specific vehicle devices such as fuel valve controls. The register has a 15-bit storage capacity and is loaded by a PIO instruction. The register thus controls the outputs of 5 Switch Selectors located within the stages of the vehicle. The 15 bits are used as follows:

SECTION 15.6
INPUT-OUTPUT DATA PROCESSING

The solution to the navigation, guidance, and control problems of the Saturn Launch Vehicles requires processing of most input and output data by the LVDA/LVDC system. Generally, this processing is required for equipment compatibility, temporary data storage, and data transfer. There are six processing areas defined in the basic LVDA/LVDC scheme. These areas are:

- Gimbal angle processing
- Accelerometer processing
- Switch Selector processing
- Interrupt processing
- Telemetry processing
- Attitude command processing

Gimbal angle processing is an example of the complex processing required for data compatibility. The LVDA receives this data from the Inertial Platform in sinusoidal form and must then convert this intelligence to an acceptable digital form before the LVDC flight program can utilize it. The data is then stored within the LVDA until called for by the flight program. A PIO instruction then transfers the data to the LVDC.

The processing, which is detailed in the following paragraphs, is a combination of flight programming, timing, and LVDA/LVDC circuitry and logic. The preceding sections in this chapter, as well as Chapter 11, will be helpful in understanding how this processing is accomplished and how the resultant input-output data is actually used by the flight program.

GIMBAL ANGLE PROCESSING

The pitch, yaw, and roll gimbal angles are measured by resolvers on the Inertial Platform. Two separate resolvers, coarse and fine, are used for each gimbal angle. The coarse resolver has a ratio of 1:1, while the fine resolver has a ratio of 32:1. Through electrical phase shifting, the two ratios are converted into three ratios — 1:1, 2:1, and 64:1. In the case of the 64:1 ratio, each mechanical degree of

rotation will cause 64° of electrical rotation to occur. These three ratios are designated as coarse, backup, and fine, respectively. Since the fine gimbal reading is the most accurate, it is normally used in the processing scheme. The backup ratio, as its name implies, serves as a backup reading in case of malfunctions or errors. The coarse ratio is not used.

Figure 15. 6-1 is a block diagram of a typical resolver system. The diagram incorporates the resolver located on the Platform and part of the LVDA circuitry. Two sinusoidal output voltages of the resolver secondaries are transmitted to an RC phase and shaping network in the LVDA. The waveforms are then sensed for their zero "crossover" points by 2 Crossover Detectors (COD's). The start COD output signal starts counters A and B at the zero crossover point of the positive going wave. The counters are stopped in the same manner by the output from the stop COD.

The overall flow diagram of the resolver signal processing is shown in Figure 15. 6-2. Two identical 11-bit counters operate at the 2. 048 MHz oscillator frequency. The two outputs are sent to the counter selector circuit, where one of the counters is selected, and its contents serialized and sent to the LVDC. The counter outputs are also sent to a subtract circuit where a difference is taken (A-B). If both counters are working properly, the difference (A-B) will be zero and either counter can be used. The limit checking circuitry is such that if the difference is within the limits of -4 and +3, either counter can be used. If the difference is greater than these limits, a disagreement bit will be set. The disagreement bit is part of the gimbal angle word placed in the accumulator when a gimbal angle PIO is issued. A decision must then be made as to which counter reading is correct. A comparison between individual counter readings and past readings is made by the flight program, and the best counter will be used. The checking process is repeated in hopes that both counters will work properly at the next reading. If a disagreement bit is set by three incorrect readings of the same angle during one computation cycle the
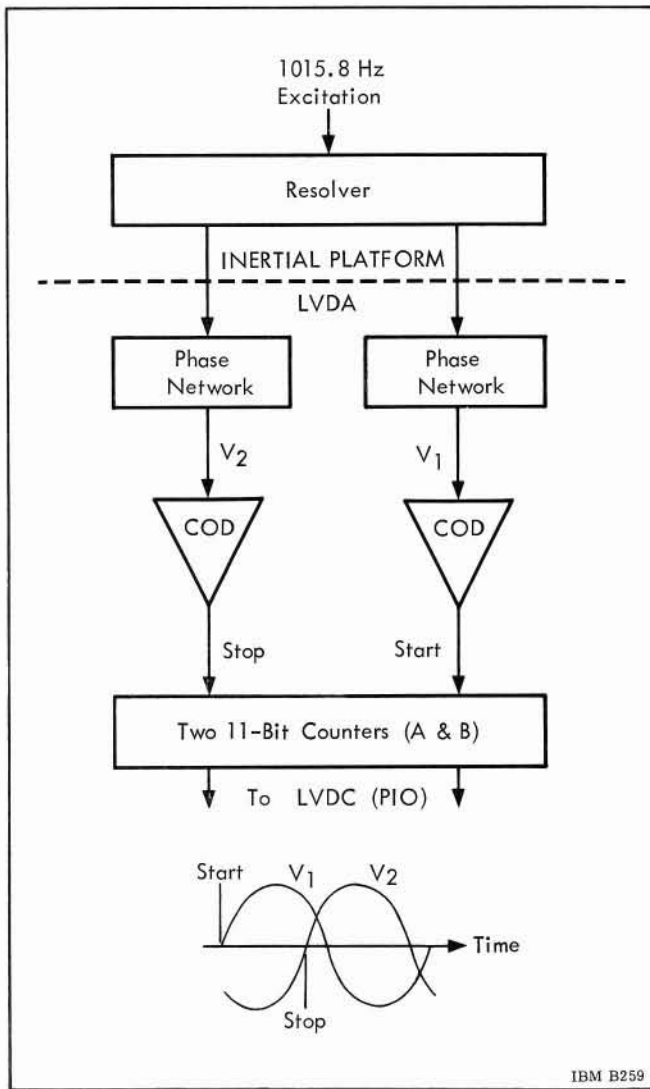
Figure 15. 6-1 Resolver System Block Diagram

disagreement bit will be inhibited, and the good counter will be used throughout the remainder of the flight.

ACCELEROMETER PROCESSING

The ST-124-M Inertial Platform System contains three accelerometers, which sense acceleration in each of the three (X, Y, and Z) orthogonal directions. Acceleration is indicated as a shaft rotation; the speed of rotation being proportional to acceleration. Since acceleration causes the accelerometer shaft to be displaced as a function of time, the shaft position (with respect to a zero reference) is proportional to velocity, and the accelerometer is referred to as an integrating accelerometer.

Each of the accelerometer shafts is linked to an optical encoding device called an optisyn. The optisyn converts the accelerometer shaft rotation to

two pairs of sinusoidal signals, one pair is referred to as the direct channel and the other pair as the redundant channel. Each pair of signals consists of two sinusoidal waves approximately 90 degrees out of time phase. These signals are passed through the Accelerometer Signal Conditioner for wave shaping purposes before being received by the LVDA. Chapter 14 contains detailed information on the accelerometer assembly and the Accelerometer Signal Conditioner.

The LVDA, by means of accelerometer processor logic, receives the velocity information from the optisyns and converts it into a form that can be utilized by the LVDC. The LVDA accumulates velocity as measured by the Inertial Platform, and the LVDC attempts to detect velocity errors through reasonableness tests.

Each pair of signals generated by the optisyns appears at the LVDA as two voltages displaced in time as shown in Figure 15. 6-3. The frequency of these square waves is directly proportional to the change in velocity sensed by the accelerometer. The LVDA tests the voltage level of each signal every 82. 03 microseconds. As illustrated in the figure, there are four unique combinations of voltage levels definable at the times t_1, t_2, t_3, and t_4. The resulting code progresses in a positive direction as follows: 01, 00, 10, 11, 01, etc. It should be noted that this code does not progress in the same manner as a true binary code: 00, 01, 10, 11, 00, etc. Instead, it changes only one bit at a time, and is called a reflected binary code. (The term "gray code" is also used.)

The LVDA recognizes changes in the optisyn output signals as increments in velocity. The frequency of the square waves is 1500 cycles per accelerometer head revolution. Therefore, one revolution of the accelerometer head presents 6000 increments of velocity. Since the accelerometer integrates acceleration at the rate of 300 meters per second in one revolution, the resolution of the system is 0. 05 meters per second for each increment.

An optisyn cycle is a progression through all the bit configurations that the optisyn can generate before it begins to repeat itself. Since the range of optisyn cycles (velocities) attained will be large, it is necessary to provide a means of counting and storing the number of optisyn cycles for processing by the LVDC. The count and store functions are provided in the LVDA by three duplexed accumulators called accelerometer processors. These processors consist of channel 4 of each of the three TMR delay lines, and the associated logic circuitry. The storage
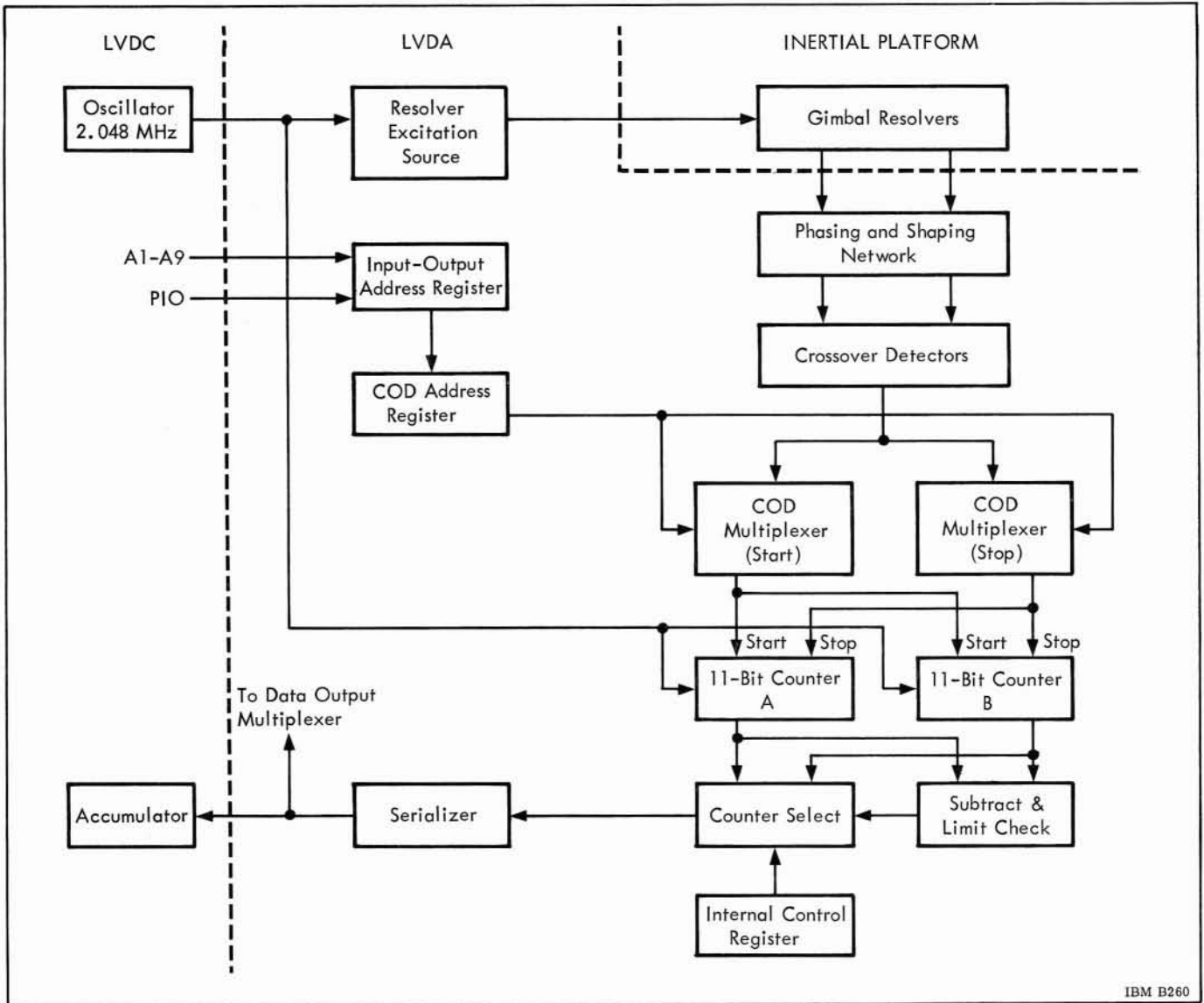
Figure 15. 6-2 Resolver Signal Processing
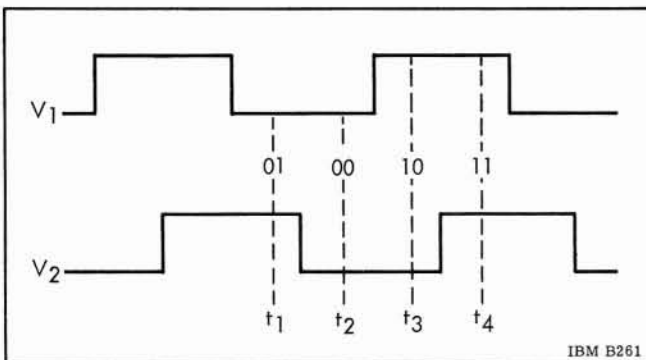
Figure 15. 6-3 Optisyn Signals at the LVDA

function of the accelerometer processors is performed by channel four (phase times B and C) of the delay lines. Figure 15.6-4 illustrates this velocity information storage.

The location of velocity information from the X, Y, and Z accelerometers in the three TMR delay lines is shown in Figure 15. 6-5. The X_1, Y_1, and Z_1 velocity information is derived from direct channel optisyn signals, while X_2, Y_2, and Z_2 velocity information is derived from redundant channel optisyn signals. As the figure indicates, the arrangement is such that no velocity information would be lost should one of the TMR delay lines fail.

At each optisyn sampling time, the accelerometer processor compares the new values of the R and Q bits (low order bit and high order bit, respectively) received from the optisyn with the old values to determine if an optisyn cycle has been completed. The R and Q bits are then converted from reflected to true binary and stored as the two low order bits in the delay line. If the completion of a positive cycle was

CHAN	WRITE TIME	PHASE A (1-14)	PHASE B (1-14)	PHASE C (1-14)
1	Z			INTERRUPT STORE — 1 2 3 4 5 6 7 8 9 10 11 12
2	W	12	SWITCH SELECTOR INTERRUPT COUNTER — 1 2 3 4 5 6 7 8 9 10 11 12 13	INTERRUPT LIMITING — 1 2 3 4 5 6 7 8 9 10 11
3	X	LADDER PULSE COUNTER — 12 1 2 3 4 5	TIMED INTERRUPT COUNTER — 1 2 3 4 5 6 7 8 9 10 11 12 13	INHIBIT INTERRUPT STORE — 1 2 3 4 5 6 7 8 9 10 11
4	Y	REAL TIME ACCUMULATION — 1 2 3 4 5 6 7 8 9 10 11 12 13	VELOCITY ACCUMULATION — R Q 3 4 5 6 7 8 9 10 11 12	VELOCITY ACCUMULATION — R Q 3 4 5 6 7 8 9 10 11 12

IBM B262

Figure 15.6-4 TMR Delay Line Configuration

sensed, a "1" is added to the 10 high order bits. Conversely, if completion of a negative cycle was sensed, a "1" is subtracted. The 10 high order bits are left unchanged if an optisyn cycle has not been completed.

SWITCH SELECTOR PROCESSING

The Saturn V system utilizes the computing system for control of mode and sequence of functions in all stages. The Switch Selector provides the communications link between the LVDC/LVDA (through the control distributor), the IU, and each stage. The Switch Selector is the device that has direct control of functional operation of the equipment on each stage. The sequencing scheme has the capability of selecting five Switch Selectors in the Saturn Vehicle – one per stage, one for the IU, and one spare (see Figure 4.2-1). Control of the individual units is commanded by the LVDC.

Each operation of the switch selectors and the processing components requires a PIO instruction word from memory in accordance with the stored program. At the proper point in the program, the computer will issue commands to initiate and operate the Switch Selector sequencing. These instructions will utilize increments of 15-bit Switch Selector computer words by transferring these bits to the LVDA from either the accumulator or a location in memory (through the transfer register). The 15-bit word is divided as follows: 5 bits for stage addressing, 8 bits for commanding an operation within that stage, 1 read bit, and 1 reset bit. The 15-bit Switch Selector register output is applied to each of the Switch Selectors. To increase the reliability of the operation, a two-way communication between the LVDC/LVDA and the Switch Selectors has been established. The LVDC/LVDA transmits command data to the Switch Selector and the Switch Selector transmits the complement of the command data back to the LVDC/LVDA for verification.

Switch Selector processing is accomplished in the following manner: (see Figure 15.6-6). The system is first tested for a hung stage by the Switch Selector routine. The program then issues a PIO instruction which loads the Switch Selector registers. The 15-bit computer word includes one of the five address bits being set to select the stage for Switch Selector operation along with a coded command word indicated by the eight command bits. The other two bits, the reset bit and the read bit, are zeros. At this time, a time delay occurs which is determined by the Switch Selector interrupt counter (18 milliseconds minimum). This delay is required to allow time for relays in the addressed Switch Selector to function. After the delay, the program issues another PIO instruction that loads the original eight command bits into the accumulator

Phase B	Phase C		
Z₁	X₂	Delay Line No. 1	(Channel 4)
X₁	Y₂	Delay Line No. 2	(Channel 4)
Y₁	Z₂	Delay Line No. 3	(Channel 4)
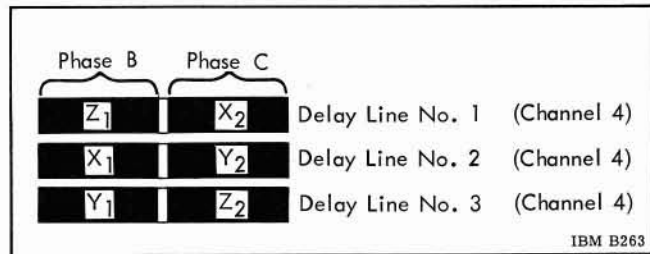
IBM B263

Figure 15.6-5 TMR Delay Line Storage of Velocity Information
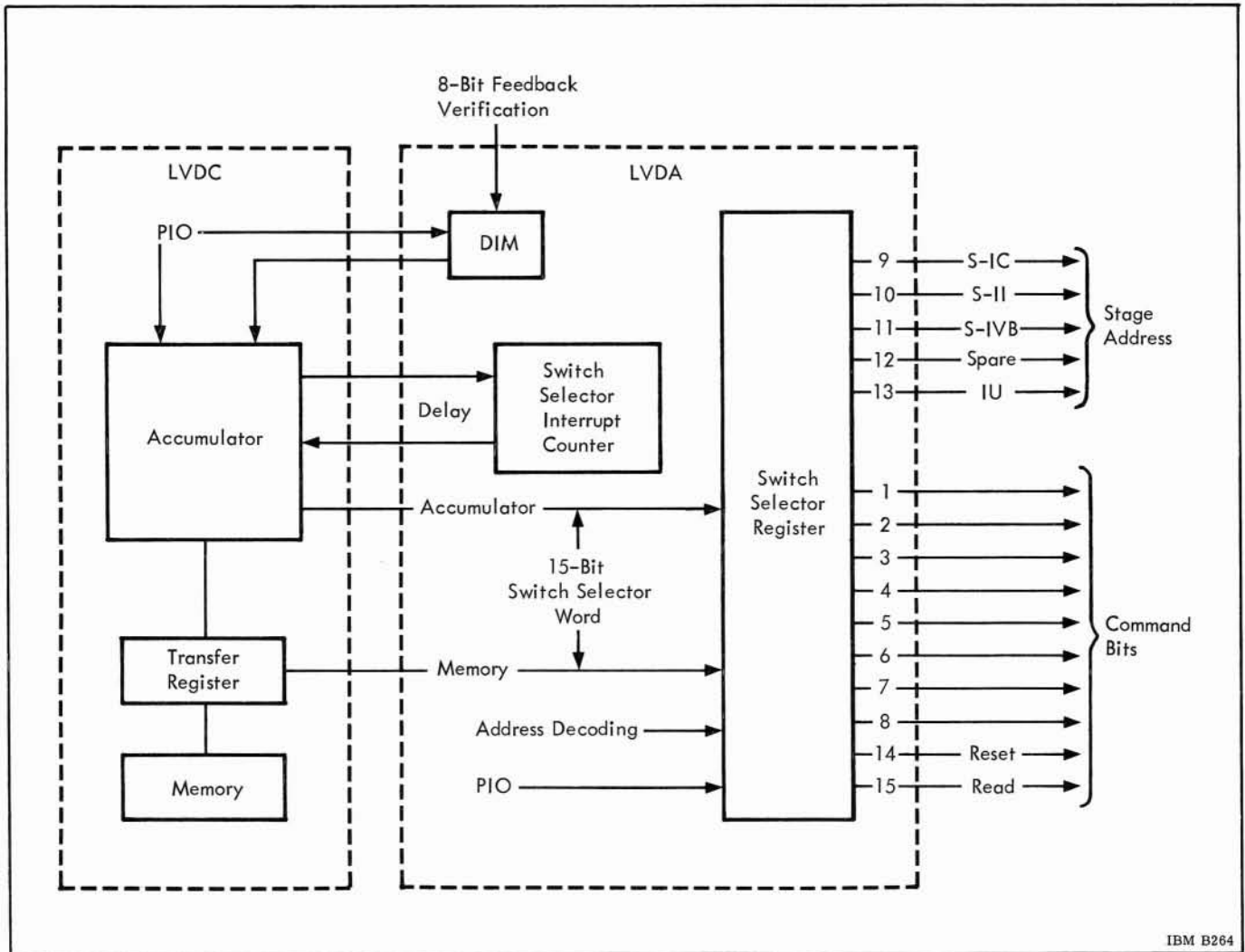
Figure 15. 6-6 Switch Selector Processing

in complement form. The computer program then checks to verify that the feedback bits are the complement of the original command to the Switch Selector. There are two alternatives available at this point: If the feedback bits are correct, the program issues a word that sets the register bits to zero (except the address bit – for telemetry – and read bit). This turns ON the function that was selected by the original computer word. The read command must remain set for at least 25 milliseconds. If the feedback bits are incorrect, the program issues a PIO switch selector code with all 15 bits being zeros except bit 14, the reset bit. The operation then must assume a delay to allow all relays in the selected stage to reset. After the delay, the program issues a PIO instruction that sets the complement of the original command word along with the stage select bit or address. Due to the dual command decoding ability of the stage Switch Selector, this operation can select the same stage operation. The program must then go through another delay

as determined by the Switch Selector interrupt counter. At the termination of the delay the program issues a PIO Switch Selector 15-bit word which is all zeros (except the address bit and read bit). This allows the commanded stage operation to be actuated.

Further information on the Switch Selector and its operation may be found in Chapter 4.

INTERRUPT PROCESSING

If some event should occur that requires immediate attention by the LVDC flight program, an interrupt signal is generated which sends the program to a special subroutine. Once the cause of the interrupt signal has been dealt with, the computer returns to the main program and continues on from the point of interruption. At present, three interrupt signals are developed internal to the LVDC/LVDA: minor loop interrupt, Switch Selector interrupt, and simul-
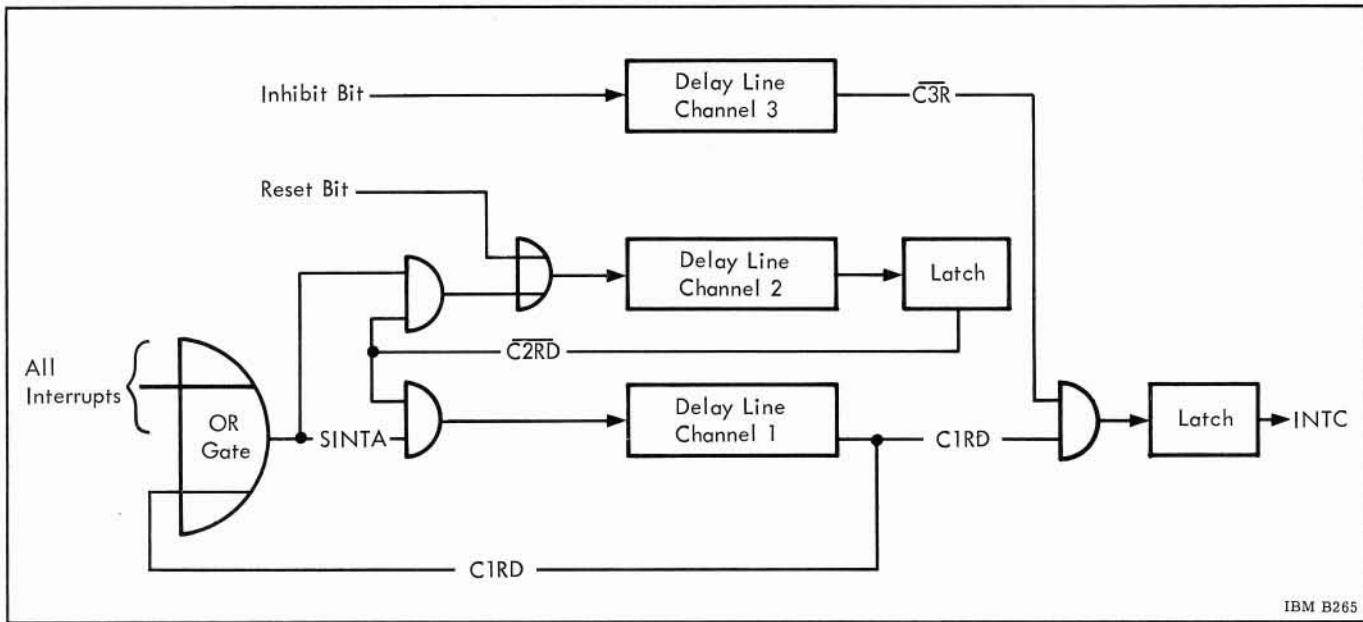
Figure 15. 6-7 Interrupt Processing Logic

taneous duplex memory error interrupt (TLC). One interrupt comes from the Saturn Ground Control Computer, and the remaining eight interrupts are developed by various control elements in the spacecraft.

Interrupt processing requires the use of the Phase C time of three channels of the LVDA delay line. See Figure 15. 6-4.

All of the interrupts to the LVDA are clocked individually and are OR'ed together so that when an interrupt does occur, it is sent over one line (SINTA) to Channel 1 of the Delay Line. See Figure 15. 6-7. Each particular interrupt bit can be identified by the bit time that is sent to the delay line. Since only one interrupt is processed at a time, a certain priority is set up. This priority depends on the order that the various interrupt input lines are sampled. Upon leaving the delay line, the interrupt bit sets an interrupt computer latch (INTC). The output is sent to the LVDC to initiate an interrupt subroutine. If any other interrupts are encountered during the process, these bits are also put into the delay line, and upon completion of the present interrupt subroutine, the new interrupt is then dealt with. Once the LVDC interrupt subroutine is completed, an interrupt reset bit is sent to the second channel of the delay line in the LVDA. This reset bit corresponds in relative time to the interrupt bit in Channel 1 that it is meant to reset. The INTC signal being sent to the computer is reset. After passing through the delay line, the reset bit sets a latch ($\overline{C2RD}$ = 0) for one bit time, and effectively causes a delay of one machine cycle. This disables the AND gate to Channel 1 of the delay line. This is

necessary in the event that the mechanism feeding the interrupt signal to the delay line (such as a slow-acting relay) had not yet reset its interrupt signal, another interrupt process would not be started. If the original interrupt had disappeared (SINTA = 0), the process stops and the next interrupt, if any, is then processed. If the original interrupt has not yet reset (SINTA = 1), the reset bit is recirculated in Channel 2 of the delay line until the interrupt does disappear.

If for some reason, the programmer does not desire a certain interrupt signal to be processed, a PIO command will be issued to inhibit any or all interrupts that might be sent to the LVDC over the INTC line. These inhibit bits are fed into Channel 3 of the delay line and correspond in relative time to the interrupt bits of Channel 1 that they are to inhibit. When an interrupt signal is produced (but an interrupt inhibit signal is also produced), the interrupt signal, C1RD, is activated as usual; but the interrupt computer signal, INTC, is held off since $\overline{C3R}$ equals 0 for that bit time.

TELEMETRY PROCESSING

In order to monitor conditions existing in the LVDC and LVDA during flight, the Mod 410 Digital Multiplexer (Chapter 5) samples information contained in telemetry registers within the LVDA at a rate of 240 times per second; i. e. , every 4. 17 milliseconds. These registers are recipients of two specific types of telemetry, LVDA telemetry and LVDC telemetry. The differences that distinguish the two types arise not only from where the telemetry information originates but also the method of processing this information.

LVDA Telemetry

Information handled by the LVDA is designated to be telemetered if:

- The information (other than LVDC telemetry information) is passed between the LVDA and LVDC.

- The LVDC issues a PIO instruction requesting information to be telemetered by the LVDA.

- Equipment error is indicated by an error monitor register.

The existence of any one of the above conditions causes the construction of an LVDA telemetry word.

All LVDA telemetry words have identical formats. The data is contained in 26 bits (which is formatted in several ways for its particular application). These bits are combined with 8 tag bits to define the type of data being telemetered and 5 bits of real time that identify the time at which the data was designated to be telemetered. Referring to Figure 15.6-8, the 39 bits of data, tag, and real time are channeled into the Data Output Multiplexer (DOM), which essentially consists of a 4-channel delay line and a distributor. Each channel is capable of storing a 39-bit word referred to as a "DOM word".

A word is stored in the DOM until the data portion of the DOM word is gated into a 26-bit buffer register. At the same time, the real time portion is transferred into a 5-bit real time/mode-validity register and the tag portion is transferred into an 8-bit tag register. The commutator action of the DOM selects a delay line channel which is currently storing a DOM word and directs this word into the registers. However, the word also remains in the DOM until it enters the Mod 410 Multiplexer. Every 4.17 milliseconds, the data stored in these registers is read into the Mod 410 Multiplexer along with the proper parity bit for a total of 40 telemetry bits. (Odd parity is used for telemetry transmission.) The multiplexer then issues a sync pulse and another word is gated from the DOM into the buffer, real time, and tag registers. If the DOM is empty, all zeros are continually read into the multiplexer.

Tag bits 1 through 7 identify the source of the 26 data bits contained in the DOM. Bit 8 is used to distinguish between LVDC and LVDA telemetry. This bit is "0" for all LVDA telemetry words.

An error indication in the error monitor register is the only condition under which an LVDA telemetry word is created without the issuance of a PIO instruction by the LVDC. The error monitor register stores disagreements between the inputs to certain voters and duplex circuits in the LVDA and LVDC. When an error occurs, an error time word is forced into the DOM. This word contains real time, phase time, clock time and the bit gate at which the error occurred, and the instruction counter setting at the time of the error. With this information the circuit causing the error can be determined.

LVDC Telemetry

Any information available to the LVDC can be made part of LVDC telemetry. By issuing a telemetry PIO instruction, the LVDC circumvents the DOM and loads the tag, real time/mode validity, and buffer registers with an LVDC telemetry word. This operation destroys the current contents of these registers thus giving the LVDC telemetry precedence over LVDA telemetry. However, a DOM word destroyed in these registers is still stored in the DOM and will be read again into the register at the occurrence of the next sync pulse.

It should be noted that an LVDC telemetry word may destroy another LVDC word if they both occur between adjacent sync pulses. But since telemetry PIO instructions are issued from the flight program, this hazard can be avoided by programming telemetry instructions at least 4.17 milliseconds apart.

An LVDC telemetry word is constructed from 26 bits of data, 9 tag bits, 3 computer mode bits and 1 validity bit. See Figure 15.6-8. The three mode bits are obtained from the LVDA mode registers. The validity bit is set to a "1" only while the LVDC is loading the telemetry registers. An indication of the validity of a telemetered LVDC word is necessary because the telemetry registers may be in the process of being
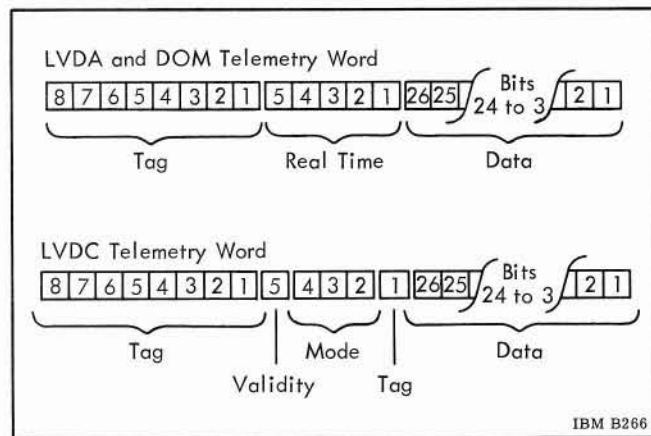
Figure 15.6-8 Telemetry Word Formats

loaded when the multiplexer is reading the telemetry registers. The validity bit being a "1" under such conditions indicates that the word in the telemetry registers is only partially loaded and, therefore, invalid.

Eight LVDC address bits are directed into the tag register. Bit 8 of this register is set to "1" to indicate LVDC telemetry and CIU operation. One address bit, three mode bits, and the validity bit are channeled into the real time/mode-validity register. The 26 bits of data are placed in the buffer register. After these registers are read into the Mod 410 Digital Multiplexer, the registers are again loaded from the DOM.

ATTITUDE COMMAND PROCESSING

The LVDC generates roll, pitch, and yaw attitude commands in a digital format. Under program control, these commands are transferred to the LVDA which processes and converts the digital data to analog voltages.

The LVDA processing operation includes digital to analog conversion, error detection, amplifying and scaling, telemetry buffering, and storage.

The LVDA continuously presents these analog commands to the Flight Control Computer where they are further processed before becoming the steering commands for the vehicle's attitude maneuvers.

Figure 15. 6-9 is a block diagram of attitude command processing within the LVDA. Three identical 9-bit ladder registers temporarily store the digital commands (roll, pitch, or yaw) as they are alternately received from the LVDC. Two of the registers provide inputs for redundant channels within the D/A converter area. The third channel provides an input for a reference channel which is used in error detection. When a ladder group address PIO instruction is issued by the flight program, the ladder registers receive 9 bits of the computer data word (bits 17 through 25). The ladder decoders then convert the digital data stored in the ladder registers into an analog voltage. Each ladder decoder converts the 9 bits of incoming digital data. The first 8 bits (LR1 to LR8) determine the magnitude of the analog signal, and the ninth bit (LR9) determines the polarity.

The decoder produces a negative output when LR9 is a "1" and a positive voltage when LR9 is a "0"

The outputs from ladder decoders A and B are applied to three output multiplexer-drivers. Each driver, when enabled by the proper control signal, will sample and store (hold) its respective pitch, roll, or yaw signal. The drivers contain "hold capacitors" which store these analog voltages for a limited time. See Figure 15. 6-10. Since the stored voltage will tend to decay and cause erroneous output signals, the hold capacitors must be recharged at regular intervals. During boost flight, these capacitors are allowed to store the signal voltage for a maximum of 40 milliseconds before being recharged by an updated attitude command. During coast phase, the quality of the attitude commands are not as critical and only need to be updated at 100 milliseconds. * When enabled by control signals, bipolar electronic switches couple the decoder outputs to the appropriate hold capacitors and ladder amplifier. The control signals cause these switches to act as either open or closed circuits. The telemetry buffers convert the signal voltages to levels which are compatible with the telemetry equipment.

An error detector compares the reference ladder decoder output with both the ladder decoder A output and the output of the selected output multiplexer-driver. If the comparisons are within designated tolerances, the error detector allows the normally selected channel A to remain selected. If either or both comparisons are out of tolerance, the error detector will select channel B. Once the error detector has selected channel B, the channel will remain selected until the flight program commands a change.

Control circuits, under address decoder control, select the appropriate output multiplexer-driver and enable the error detector to sample the ladder decoder outputs. During the time when the ladder registers are being loaded, the control circuits also inhibit the drivers and error detector from sampling the decoder outputs.

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* The flight program routine for updating these commands is only cycled once per second, but the same routine reloads the ladder registers 10 times per second (100 milliseconds). This method of programming saves considerable computer time for other operations.
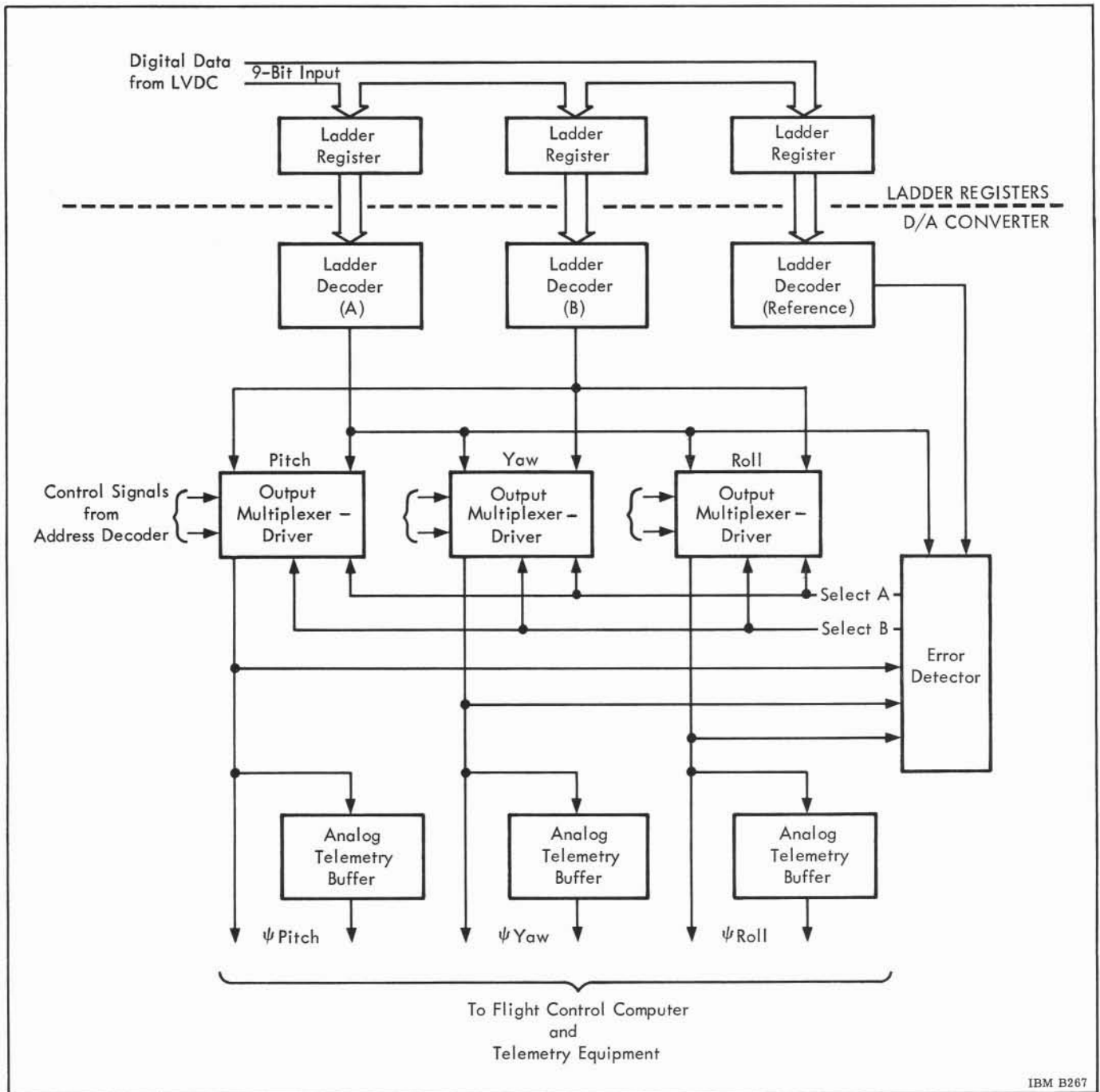
Figure 15.6-9 Attitude Command Processing Block Diagram
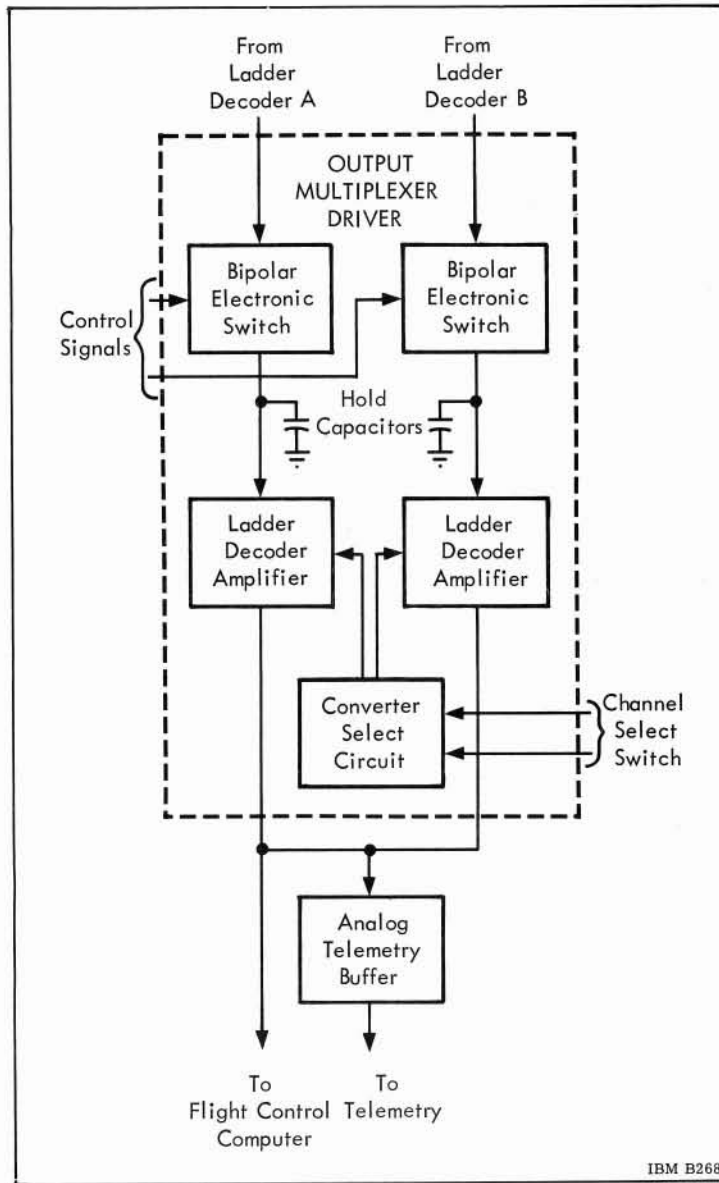
Figure 15.6-10 Output Multiplexer-Driver Block Diagram

ABBREVIATIONS

A

A	(A$_z$) Flight Azimuth
A	Ampere
ADC	Analog-to-Digital Converter
AFC	Automatic Frequency Control
AG	AND Gate
AGC	Automatic Gain Control
AM	Amplitude Modulation
AN	Ascending Node
APS	Auxiliary Propulsion System
ASP	Accelerometer Signal Processor
AVP	Address Verification Pulse
A$_z$	(A) Flight Azimuth
Acc/Reg	Accumulator Register
A/S	Add/Subtract

B

B/W	Bits/Word

C

CAT	Control Attenuation Timer
cc	Cubic Centimeter
CCS	Command and Communication System
CCSL	COD Counter Sequence Latch
CDSV	Change Data Sector-Voted
CG	Center of Gravity
CIU	Computer Interface Unit
CL	Center Line
COD	Crossover Detector
CM	Command Module
cm	Centimeter
Const	Constant
CP	Center of Pressure
CPP	Command Program Patch
CRP	Command Received Pulse
CSM	Command and Service Module
CT	Control Transformer
CSTN	Single Step Operation - Not
CSTV	Single Step Operation - Voted
CVD	Command Voltage Demodulator
CX	Control Transmitter
CX:CT	Control Transmitter:Control Transformer

D

DATAV	Normal Data Input from External Equipment
DB	Disagreement Bit

db	Decibel
DCS	Digital Command System
DCW	Data Command Word
DDAS	Digital Data Acquisition System
deg	Degree
DIN	Special Data Input
DN	Descending Node
DOMS	Data Output Multiplexer Serializer Latch
DSIF	Deep Space Instrumentation Facility
D/R	Downrange

E

EAMV	Error in Even Memories - Voted
EBMV	Error in Odd Memories - Voted
EDS	Emergency Detection System
EF	Emitter Follower
e.g.	Exempli Gratia (for example)
ESE	Electrical Support Equipment
Etc.	Et Cetera (and so forth)

F

FCM	Flight Combustion Monitor
FIR	Far Infrared
FM	Frequency Modulation
F/M	Thrust Acceleration (Force/Mass)

G

GSE	Ground Support Equipment
g	Gravity
GM	Universal Gravitational Constant
GRR	Guidance Reference Release
GRRA	Guidance Reference Release Alert
GU	Spin Reference

H

H	Spinning Mass (Angular Momentum)
HALTV	Halt - Voted
HOPC1V	Generate HOP Constant - Voted
Hz	Hertz (1 cycle per second)

I

i	Inclination of Orbit (angle)
IA	Input Axis
ICD	Internal Control Discrete
i.e.	Id Est (that is)
IECO	Inboard Engine Cutoff

IGM	Iterative Guidance Mode		Oper	Operational
IIR	Intermediate Infrared		OPR	Orbital Processing Routine
INTC	Interface Output Latch (Interrupts LVDC)			
INTV	Interrupt - Voted			
IR	Infrared			
IRIG	Interrange Instrumentation Group			

P

PAM	Pulse Amplitude Modulation
PCM	Pulse Code Modulation

J

JPL	Jet Propulsion Laboratory

PIO	Process Input Output
PM	Phase Modulation
pps	Pulses per Second
PRF	Pulse Repetition Frequency
psi	Pounds per Square Inch
psia	Pounds per Square Inch Absolute
psid	Pounds per Square Inch Differential
psig	Pounds per Square Inch Gage
PRN	Pseudo-Random Noise
PSK	Phase Shift Keyed
PSR	Parallel Storage Register
PTL	Prepare to Launch
PU	Propellant Utilization (System)

K

K	Kelvin
k	Kilohm
kg	Kilogram
kHz	Kilohertz (Kilocycle)
kW	Kilowatt

L

LEM	Lunar Excursion Module
LES	Launch Escape System
LET	Launch Escape Tower
LO	Lift-Off
LOX	Liquid Oxygen
LSD	Least Significant Digit
LTE	Laboratory Test Equipment
LVDA	Launch Vehicle Data Adapter
LVDC	Launch Vehicle Digital Computer

Q

Q	Aerodynamic Pressure

R

R	Reset
r (or) R	Vehicle Position
$\dot{r}_o$	Initial Velocity
RACS	Remote Auto. Calib. System
RASM	Remote Analog Submultiplexer
R and D	(R&D) Research and Development
RDSM	Remote Digital Submultiplexer
rms	Root Mean Square
RSO	Range Safety Officer
RTC	Reasonable Test Constant
RUNV	Start Signal - Voted

M

m	Meter
MAP	Message Acceptance Pulse
MCC	Mission Control Center
MCR	Magnetic Core Register
MHz	Megahertz (Megacycle)
Mono	Monostable (Multivibrator)
ms	Millisecond
MSD	Most Significant Digit
MSFN	Manned Space Flight Network
mV	Millivolt
MW	Megawatt

S

S	Set
s	Second
S&A's	Safe and Arm Devices
SA	Spin Axis
SC	(S/C) Spacecraft
SCCS	Saturn Command and Control System
SCO	Subcarrier Oscillator
Servo	Servomechanism
SDH	Simplex Driver - High
SDI	Simplex Driver - Intermediate
SDL	Simplex Driver - Low
SG	Signal Generator
SM	Service Module
SR	Shift Register
SRA	Spin Reference Axis
SS	Single Sideband

N

N	Newton
N	"Not" Condition, a Binary "O"
NIR	Near Infrared
NRZ	Non-Return-to-Zero
N/A	Not Applicable

O

OA	Output Amplifier
OECO	Outboard Engine Cutoff

T

TBD	To be Determined
TD	Time Delay
TDH	TMR Driver - High Current
TDL	TMR Driver - Low Current
TDM	TMR Driver - Medium Current
TG	Torque Generator
TLC	Simultaneous Memory Errors
TLCV	Two Simultaneous Memory Errors - Voted
TMR	Triple Modular Redundancy
TRS	Transfer Register Serial
TRSV	Transfer Register Serial - Voted

U

UHF	Ultra-High Frequency
us	Microsecond (s)
USB	Unified S-Band

V

VCO	Voltage-Controlled Oscillator
VHF	Very-High Frequency
VSWR	Voltage Standing Wave Ratio

W

W	Watt

X

X	Roll Axis

Y

Y	Pitch Axis

Z

Z	Yaw Axis

SIGNS AND SYMBOLS

$\int$	Integral
∂	Partial Differential
$\sqrt{}$	Square Root
$\sim$	Difference
$\propto$	Varies as
$>$	Greater than
$<$	Less than
$\geq$	Greater than or equal to
$\leq$	Less than or equal to
∞	Infinity
$\cong$	Congruent to
$\approx$	Nearly Equal to
$\angle$	Angle
Δ	Change

	Greek Alphabet		
α	alpha	ν	nu
β	beta	ξ	xi
γ	gamma	o	omicron
δ	delta	π	pi
ϵ	epsilon	ρ	rho
ζ	zeta	σ	sigma
η	eta	τ	tau
θ	theta	υ	upsilon
ι	iota	ϕ	phi
κ	kappa	χ	chi
λ	lambda	ψ	psi
μ	mu	ω	omega

β	Feedback signal from S-IB actuators to Flight Control Computer.
β_c	Command to actuator from Flight Control Computer. $(\beta_R \ \beta_P \ \beta_Y)$
$\ddot{\gamma}_P \ \ddot{\gamma}_R$	Lateral acceleration (S-IB only).
$\Delta X g$	Change to X_S caused by gravitation.

ΔX_T	Change to X_S caused by Platform-measured acceleration.
θ	Platform gimbal angle. Indicates the difference between vehicle attitude and Platform attitude.
$\theta_X \theta_Y \theta_Z$	Platform gimbal angles (vehicle attitude).

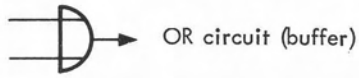|			
ϕ	(ϕ_L) Geodetic latitude of the launch site.	r	(R) Vehicle position measured as a radius vector from the center of the earth to the vehicle
$\dot{\phi}$	Rate of angular attitude change.		
$\dot{\phi}_R \dot{\phi}_P \dot{\phi}_Y$	Rate of attitude change in vehicle coordinates.	$\dot{r}$	Vehicle velocity after gravitational and initial velocity have been acounted for
χ	Desired attitude change as determined by guidance computations.	S	Same as P
		UVW	Gravitational coordinate system
$\chi_R \chi_P \chi_Y$	Guidance command angles (desired vehicle attitude).	V_S	Velocity
$\chi_X \chi_Y \chi_Z$	Guidance command angles (Euler angles)	W	Same as P
ψ	Attitude error signal	$X_4 Y_4 Z_4$	Target plane (flight plane) coordinate system
$\psi_R \psi_P \psi_Y$	Attitude error signals (steering commands)	$\dot{X}_g \dot{Y}_g \dot{Z}_g$	Components of gravitational velocity
ψ_T	Total range angle		
a_0	Gain factor for attitude error (ψ) channels	$\ddot{X}_g \ddot{Y}_g \ddot{Z}_g$	Components of gravitational acceleration
a_1	Gain factor for attitude rate $(\dot{\phi})$ channels	$X_I Y_I Z_I$	Inertial coordinate system (also expressed as $X_M Y_M Z_M$)
a_e	Equatorial radius of the earth		
g_2	Gain factor for lateral acceleration $(\ddot{\gamma})$ channels (Saturn IB only)	$\dot{X}_I \dot{Y}_I \dot{Z}_I$	Components of thrust velocity (measured by the integrating accelerometers of the Platform)
I_{gX}	Computed gravitational velocity in the X_S direction	$X_0 Y_0 Z_0$	Initial position - launch site to earth's center
LH_2	Liquid Hydrogen	$\dot{X}_0 \dot{Y}_0 \dot{Z}_0$	Initial velocity
N_2O_4	Nitrogen tetroxide	$X_S Y_S Z_S$	Space-fixed navigational coordinate system (used for navigation and guidance computations)
P	Constant used in computing gravitational acceleration and using:		
		$\dot{X}_S \dot{Y}_S \dot{Z}_S$	Inertial velocity
	a_e = equatorial radius of the earth	$X_V Y_V Z_V$	Injection coordinate system
	GM = universal gravitational constant		
	J = constant		
	ϕ = geodetic latitude of launch site		
	A_Z = firing azimuth		

AND gate

AND gate with inhibiting input

OR circuit (buffer)
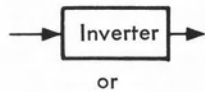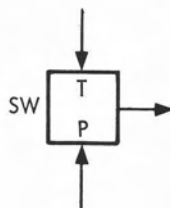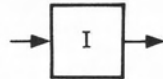
Bistable multivibrator (flip flop). Reset output is significant immediately after Reset input has occurred. The same is true for the Set side.

S

R

Differentiating circuit which produces pulses at the leading and trailing edges of gate waveforms. A minus sign placed inside symbol indicates that only the trailing-edge pulse is significant. A plus sign indicates that the leading edge is significant. The plus and minus signs have nothing to do with polarity, but merely denote leading or trailing edge.

$\dfrac{d}{dt}$

MVB — Monostable multivibrator. Time period of MVB is shown adjacent to symbol.

— ms

Passive time delay element. (Tau)

D — Time delay

Summing amplifier

Inverter

or

I

Logical inversion. Inverts from significant to inhibiting and vice-versa. The term "significant", as used herein, implies nothing about the polarity of a given waveform, as it is not necessary to know the polarities to understand the general operation of the decoder logic. It is only necessary to remember that when the output of a given element is "significant", the element receiving such an input will be activated to its significant state, or will perform its prescribed function.

SW
T
P

Bistable electronic switch (such as a silicon controlled switch). Input P is the "plate" or anode, and is connected to some power source. When P is present and trigger T occurs, the switch is said to be "on" and the output is then significant. Switch can be turned off only by an interruption of the power.

FF	— Flip-Flop (Bistable Multivibrator)
ms	— Millisecond
MVB	— Multivibrator
R	— Reset
S	— Set
SR	— Shift Register
SW	— Switch
T	— Trigger
TD	— Threshold Detector

ALPHABETICAL INDEX

ABBREVIATIONS

A

A	Ampere
ADC	Analog-to-Digital Converter
AFC	Automatic Frequency Control
AG	AND Gate
AGC	Automatic Gain Control
AM	Amplitude Modulation
APS	Auxiliary Propulsion System
ASP	Accelerometer Signal Processor
AVP	Address Verification Pulse
AZ	Flight Azimuth
Acc/Reg	Accumulator Register
A/S	Add/Subtract

B

B/W	Bits/Word

C

C. A. T.	Control Attenuation Timer
cc	Cubic Centimeter
CCS	Command Communication System
CCSL	COD Counter Sequence Latch
CDSV	Change Data Sector - Voted
CG	Center of Gravity
CIU	Computer Interface Unit
COD	Crossover Detector
CM	Command Module
cm	Centimeter
Const	Constant
CP	Center of Pressure
CRP	Command Received Pulse
CSM	Command and Service Module
CT	Control Transformer
CSTN	Single Step Operation - Not
CSTV	Single Step Operation - Voted
CVD	Command Voltage Demodulator
CX	Control Transmitter
CX:CT	Control Transmitter:Control Transformer

D

DATAV	Normal Data Input from External Equipment
db	Decibel
DCS	Digital Command System

DCW	Data Command Word
DDAS	Digital Data Acquisition System
deg	Degree
DIN	Special Data Input
DOMS	Data Output Multiplexer Serializer Latch
DSIF	Deep Space Instrumentation Facility
D/R	Downrange

E

EAMV	Error in Even Memories - Voted
EBMV	Error in Odd Memories - Voted
EDS	Emergency Detection System
EF	Emitter Follower
e. g.	Exempli Gratia (for example)
ESE	Electrical Support Equipment
Etc.	Et Cetera (and so forth)

F

FCM	Flight Combustion Monitor
FIR	Far Infrared
FM	Frequency Modulation
F/M	Thrust Acceleration

G

GSE	Ground Support Equipment
g	Gravity
g (r)	Gravitational Acceleration
GU	Spin Reference

H

H	Spinning Mass (Angular Momentum)
HALTV	Halt - Voted
HOPC1V	Generate HOP Constant - Voted
Hz	Hertz (Cycle)

I

IA	Input Axis
ICD	Internal Control Discrete
i. e.	Id Est (that is)
IIR	Intermediate Infrared
INTC	Interface Output Latch (Interrupts LVDC)
INTV	Interrupt - Voted
IR	Infrared
IRIG	Interrange Instrumentation Group

J

JPL	Jet Propulsion Laboratory

K

K	Kelvin
k	Kilohm
kg	Kilogram
kHz	Kilohertz (Kilocycle)
kW	Kilowatt

L

LH_2	Liquid Hydrogen
LO	Lift-Off
LOX	Liquid Oxygen
LSD	Least Significant Digit
LTE	Laboratory Test Equipment
LVDA	Launch Vehicle Data Adapter
LVDC	Launch Vehicle Digital Computer

M

m	Meter
MAP	Message Acceptance Pulse
MCC	Mission Control Center
MCR	Magnetic Core Register
mHz	Megahertz (Megacycle)
Mono	Monostable (Multivibrator)
ms	Millisecond
MSD	Most Significant Digit
mV	Millivolt
MW	Megawatt

N

N	Newton
N	"Not" Condition, a Binary "O"
NIR	Near Infrared
NRZ	Non-Return-to-Zero

O

OA	Output Amplifier
Oper	Operational

P

PAM	Pulse Amplitude Modulation
PCM	Pulse Code Modulation
PIO	Process Input Output
PM	Phase Modulation
pps	Pulses per Second
PRF	Pulse Repetition Frequency
psi	Pounds per Square Inch
psia	Pounds per Square Inch Absolute
psid	Pounds per Square Inch Differential
psig	Pounds per Square Inch Gage
PSK	Phase Shift Keyed
PSR	Parallel Storage Register
PU	Propellant Utilization (System)

R

R	Reset
$\dot{r}$	Vehicle Position
$\dot{r}_0$	Initial Velocity
RACS	Remote Auto. Calib. System
RASM	Remote Analog Submultiplexer
R and D	(R&D) Research and Development
RDSM	Remote Digital Submultiplexer
RSO	Range Safety Officer
RUNV	Start Signal - Voted

S

S	Set
s	Second
SA	Spin Axis
SC	(S/C) Spacecraft
SCCS	Saturn Command and Control System
SCO	Subcarrier Oscillator
Servo	Servomechanism
SDH	Simplex Driver - High
SDI	Simplex Driver - Intermediate
SDL	Simplex Driver - Low
SG	Signal Generator
SM	Service Module
SR	Shift Register
SRA	Spin Reference Axis
SS	Single Sideband

T

TBD	To be Determined
TD	Time Delay
TDH	TMR Driver - High Current
TDL	TMR Driver - Low Current
TDM	TMR Driver - Medium Current
TG	Torque Generator
TLCV	Two Simultaneous Memory Errors - Voted
TMS	Triple Modular Redundancy
TRS	Transfer Register Serial
TRSV	Transfer Register Serial - Voted

U

UHF	Ultra-High Frequency
us	Microsecond (s)

V

VCO	Voltage-Controlled Oscillator
VHF	Very-High Frequency

W

W	Watt

X

X_{GU}	X Gyro Spin Reference

Y

Y_{GU}	Y Gyro Spin Reference

Z

Z_{GU}	Z Gyro Spin Reference

SIGNS

$\int$	Integral
∂	Partial Differential
$\sqrt{}$	Square Root
$\sim$	Difference
α	Varies as
$>$	Greater than
$<$	Less than
$\geq$	Greater than or equal to
$\geq$	Less than or equal to
∞	Infinity
$\cong$	Congruent to
$\approx$	Nearly Equal to
$\angle$	Angle

	Greek Alphabet		
α	alpha	ν	nu
β	beta	ξ	xi
γ	gamma	o	omicron
δ	delta	π	pi
ϵ	epsilon	ρ	rho
ζ	zeta	σ	sigma
η	eta	τ	tau
θ	theta	υ	upsilon
ι	iota	ϕ	phi
κ	kappa	χ	chi
λ	lambda	ψ	psi
μ	mu	ω	omega

SYMBOLS

AND gate

AND gate with inhibiting input

OR circuit (buffer)
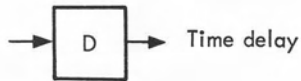
Bistable multivibrator (flip flop). Reset output is significant immediately after Reset input has occurred. The same is true for the Set side.

Differentiating circuit which produces pulses at the leading and trailing edges of gate waveforms. A minus sign placed inside symbol indicates that only the trailing-edge pulse is significant. A plus sign indicates that the leading edge is significant. The plus and minus signs have nothing to do with polarity, but merely denote leading or trailing edge.

Monostable multivibrator. Time period of MVB is shown adjacent to symbol.

MVB
— ms

Passive time delay element. (Tau)

Time delay

Summing amplifier

Inverter

or

I

Logical inversion. Inverts from significant to inhibiting and vice-versa. The term "significant", as used herein, implies nothing about the polarity of a given waveform, as it is not necessary to know the polarities to understand the general operation of the decoder logic. It is only necessary to remember that when the output of a given element is "significant", the element receiving such an input will be activated to its significant state, or will perform its prescribed function.

SW

T
P

Bistable electronic switch (such as a silicon controlled switch). Input P is the "plate" or anode, and is connected to some power source. When P is present and trigger T occurs, the switch is said to be "on" and the output is then significant. Switch can be turned off only by an interruption of the power.

FF — Flip-Flop (Bistable Multivibrator)
ms — Millisecond
MVB — Multivibrator
R — Reset
S — Set
SR — Shift Register
SW — Switch
T — Trigger
TD — Threshold Detector

ALPHABETICAL INDEX

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