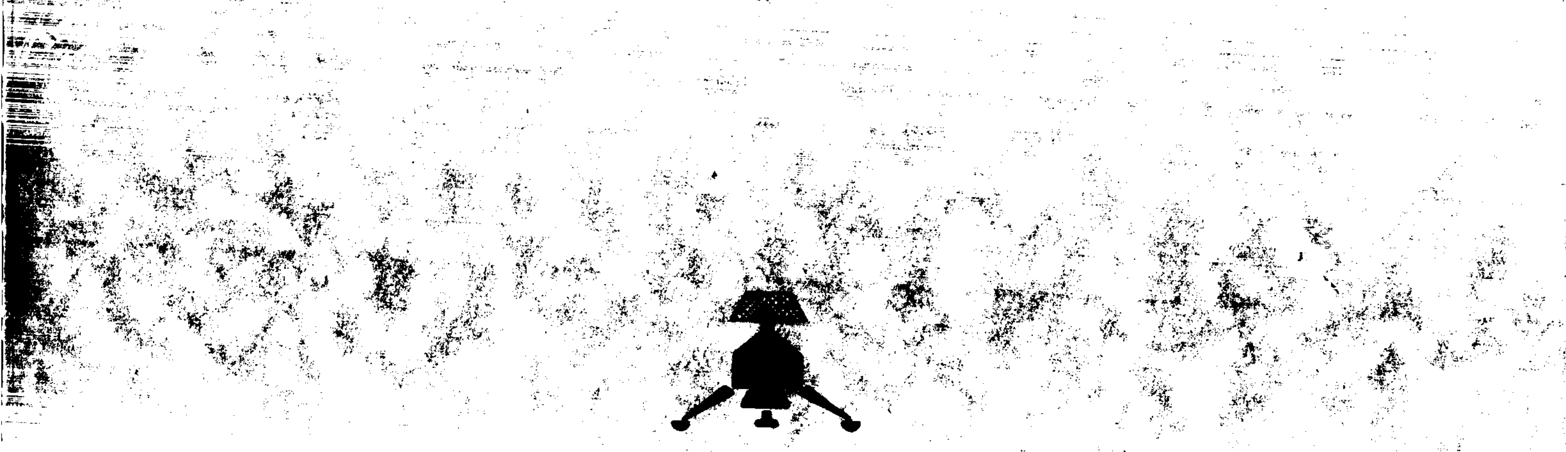

SURVEYOR SPACECRAFT
A-21A
MODEL DESCRIPTION

HUGHES

HUGHES AIRCRAFT COMPANY
SPACE SYSTEMS DIVISION

224848

SQT 48952-R

COPY #

1 SEPTEMBER 1964

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JPL 950056

This work was performed for the Jet Propulsion Laboratory, California Institute of Technology, sponsored by the National Aeronautics and Space Administration under Contract NAS7-100.

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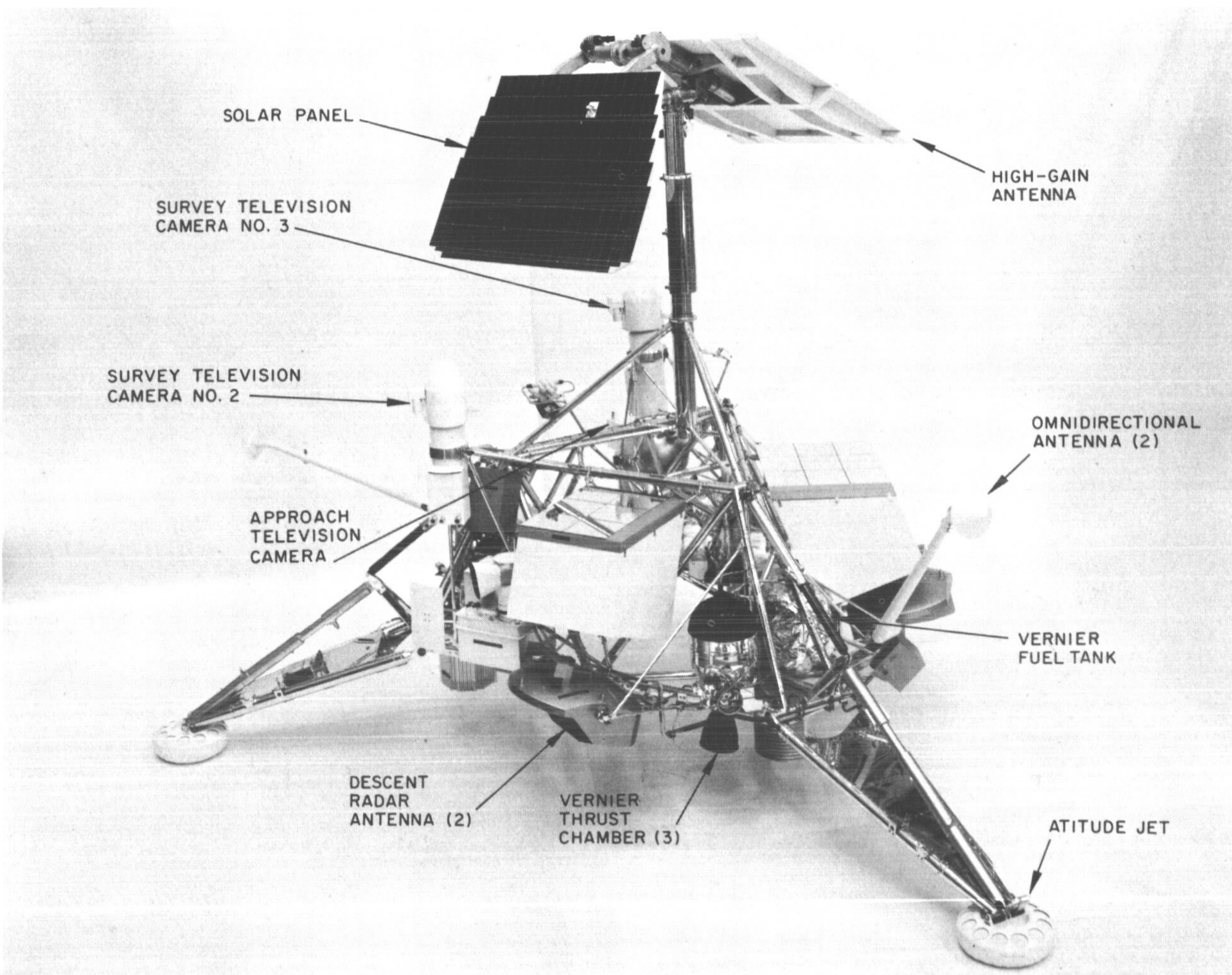


SURVEYOR SPACECRAFT A-21A

INTRODUCTION

SCOPE

This document describes the general design and overall performance of the A-21A model of Surveyor spacecraft. The material presented in the document describes the spacecraft design in its present state of development and predicts the performance expected of the spacecraft when operated under realistic conditions in a specified environment. Although the spacecraft performance is controlled, constrained, or influenced by a large number of factors other than those associated with the design and construction of the spacecraft itself, this document defines spacecraft performance in terms of the spacecraft design and mechanization only. The description of those scientific instruments provided by JPL represents the current Hughes Aircraft Company understanding of the design and operation of these instruments. A complete definition of spacecraft performance in total context with the numerous functional, operational, and administrative interfaces that exist is beyond the scope of this document.

MISSION OBJECTIVES

The Surveyor spacecraft is being designed and built by Hughes Aircraft Company under the direction of the California Institute of Technology Jet Propulsion Laboratory (JPL) for the National Aeronautics and Space Administration (NASA). The Surveyor spacecraft vehicle has been conceived and designed to effect a transit from earth to the moon, perform a soft lunar landing, and transmit back to earth basic scientific and engineering data relative to the moon's environment and characteristics.

To obtain maximum utility, the spacecraft has been designed to accommodate various alternative payloads. The basic spacecraft elements of structure, telecommunications, power generation, propulsion, and flight control provide the capability to perform the earth-moon transit and make a soft lunar landing while maintaining two-way communication. This basic grouping of spacecraft elements, designated as the "basic bus," can thus provide transportation, power, and communication services to the designated variety of payloads. The A-21 series of spacecraft, which constitutes the first four Surveyor launches, carries an engineering payload. The purpose of the A-21 series is to demonstrate successful transit and soft lunar landing and to gather basic engineering data relative to the performance of the spacecraft in the environments encountered in transit. The collection and transmission of scientific data is a secondary objective for this series of spacecraft. The A-21A series of spacecraft utilizes essentially the same basic bus but carries a different payload, consisting of various scientific instruments. The primary purpose of the A-21A series is the collection and transmission of scientific data relative to the lunar environment. The design and performance of the A-21 spacecraft is covered in HAC document 224847, "Surveyor Spacecraft A-21, Model Description."

GENERAL DESCRIPTION

Spacecraft

The general arrangement of the spacecraft and identification of its various elements are shown in the frontispiece. The spacecraft is composed of several electronic and mechanical assemblies mounted on a spaceframe constructed of thin-walled aluminum alloy tubular members. The configuration of the spaceframe is dictated by the selection of a tripod landing gear with three foldable legs for use in the soft landing. Center of gravity of the vehicle is kept low to obtain stability over a wide range of landing conditions. Thermal control of the equipment over the extreme temperature range of the lunar surface ($+260^{\circ}$ to -260° F) is accomplished by a combination of passive, semi-passive and active methods. This design represents the latest state of the art in the application of structural and thermal design principles to light-weight space structures.

Launch Vehicle

The spacecraft will be launched on its 66-hour transit to the moon by the Atlas/Centaur boost vehicle (figure 1-1). Under the direction of the NASA Lewis Research Center, the Atlas/Centaur vehicle is being specifically designed by General Dynamics/Astronautics to meet the launch requirements of the Surveyor mission. The folded spacecraft is housed within a conical breakaway shroud on

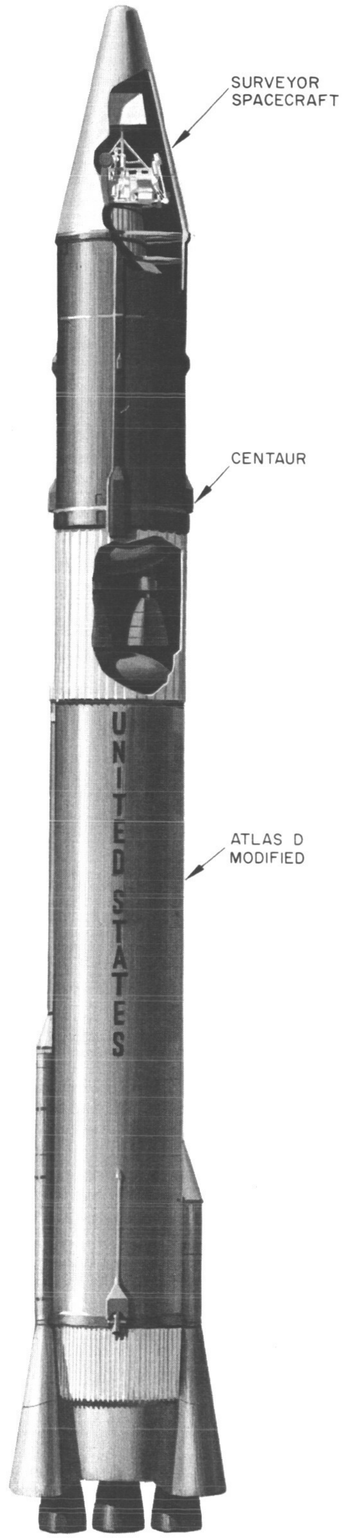


FIGURE 1-1. ATLAS/CENTAUR LAUNCH VEHICLE

top of the second stage Centaur. The Centaur, complete with its guidance system, fuel tankage, and liquid hydrogen engines, is located directly atop the first stage Atlas rocket.

MISSION

Launching and Tracking Operation

Launchings will originate from Cape Kennedy, Florida, with computation and space flight control from JPL Space Flight Operations Facilities (SFOF) in Pasadena. Upon being boosted by the Atlas to an altitude of approximately 60 miles, the breakaway nose shroud will be jettisoned. The Centaur will then inject the spacecraft into an earth/moon trajectory (figure 1-2). At the conclusion of the Centaur thrust phase, but before separation, the Centaur programmer will generate commands that extend the spacecraft landing gear and the omni antennas. The high-power transmitter mode is also commanded on at this time to aid initial spacecraft acquisition. Initial acquisition will occur at the Johannesburg, South Africa Deep Space Instrumentation Facility (DSIF) station with later tracking provided by stations at Canberra, Australia, and Goldstone, California. Both transmission and reception during this phase are from the spacecraft. The solar panel is deployed automatically at separation of the spacecraft from Centaur.

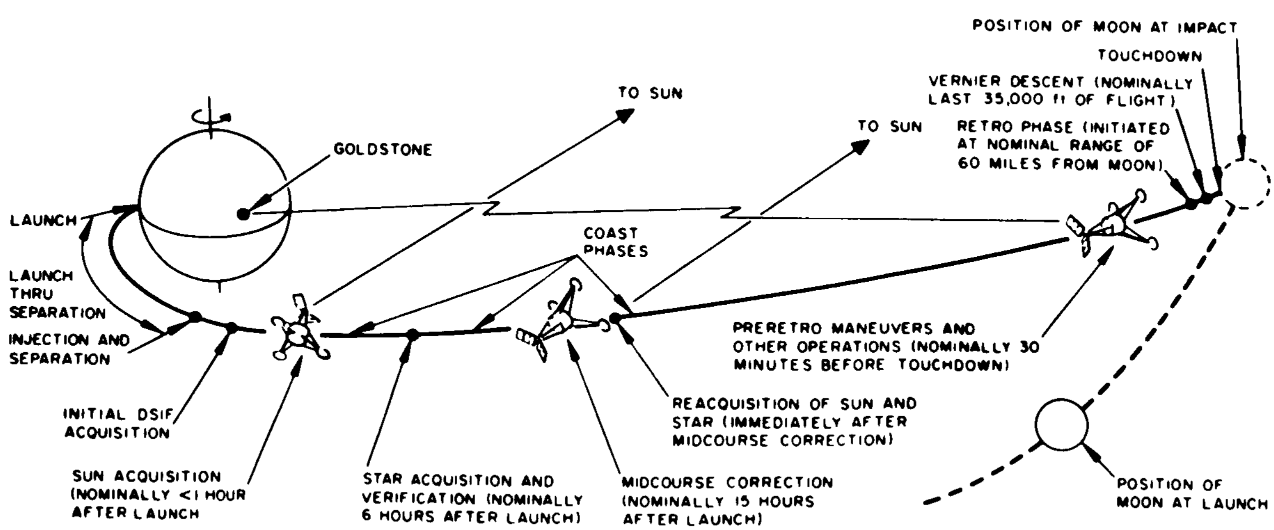


FIGURE 1-2. EARTH/MOON TRAJECTORY

Midcourse Correction

Three reaction gas jets, located on the landing gear legs, position the spacecraft to acquire and track the sun and the star Canopus. When the appropriate sensors lock on to these celestial points, a space coordinate system is established in space which will thenceforth be automatically maintained until the midcourse maneuver. The solar panel is oriented to achieve direct sun illumination and begins to generate electrical energy for spacecraft operation and battery charging. Tracking data received in sequence from the DSIF tracking stations are processed and used by SFOF to compute the midcourse correction. Upon radio command from earth, the spacecraft turns through a series of angular maneuvers to align the vernier engine thrust axis relative to the spacecraft velocity vector. The required magnitude and direction for the midcourse maneuver is transmitted from the Goldstone tracking station to the spacecraft, where it is received and stored. Then the execute command causes three liquid-fueled vernier rocket engines to operate at a specific average thrust level for a specific period of time. This action provides a midcourse alteration of the spacecraft trajectory which will ultimately bring the vehicle to the selected lunar landing area. After the midcourse correction is completed, the spacecraft reacquires the sun and the star Canopus to maintain its previous attitude.

Terminal Descent and Soft Lunar Landing

Approximately 66 hours after launch, the spacecraft approaches the moon. Upon command from the Goldstone tracking station, the spacecraft changes attitude to align the thrust of its rétro-rocket with the spacecraft velocity vector. A downward looking television camera views the moon's surface during the approach for the purpose of transmitting pictures of the landing area back to earth. As the spacecraft approaches the moon at a relative speed of about 9000 feet per second, the altitude marking radar generates a signal at a slant range of 60 miles, which, after a suitable delay, initiates ignition of the solid propellant main retro-rocket motor. This ignition and subsequent burning ejects the altitude marking radar from the retro-rocket nozzle and begins to decelerate the spacecraft. At an altitude of approximately 40,000 feet, the main retro-rocket burns out, and its empty case is separated from the spacecraft approximately 8 seconds later. At this point the spacecraft is close enough to the surface of the moon to receive reliable control signals from its altimeter and doppler velocity radar system. Signals from this system are processed by the flight control electronics to control the throttle valves on the three vernier rocket engines to maintain the proper attitude and rate of descent. The spacecraft continues to decelerate until at an altitude of 14 feet, the vernier engines are turned off. At this time both horizontal and vertical components of velocity are small. The spacecraft falls the short remaining distance to the surface of the moon with the landing shock absorbed by the landing gear and the crushable blocks. This sequence is illustrated in figure 1-3.

Lunar Operation

After landing on the surface of the moon, the spacecraft is commanded to align the solar panel to the spacecraft-sun line and the high-gain planar array antenna to the spacecraft-earth line. Subsequent commands connect one of the transmitters to the planar array antenna and switches that transmitter to the mode (high or low power) which will provide a usable bandwidth consistent with the power available from the solar panel and battery. After initial touchdown survival conformation, the various payload experiment subsystems can be turned on to provide information relative to the lunar environment via the telecommunications subsystem.

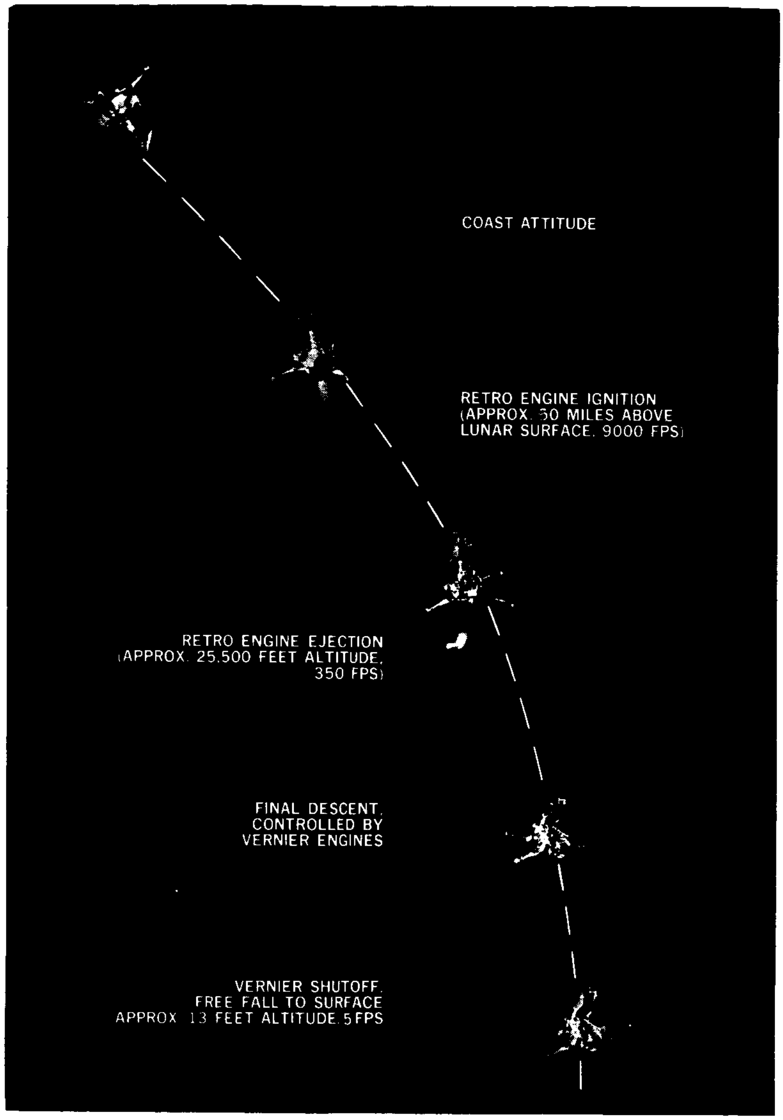


FIGURE 1-3. SPACECRAFT TERMINAL DESCENT

II. SPACECRAFT DESCRIPTIVE SUMMARY

GENERAL ARRANGEMENT

Design of the Surveyor spacecraft is dictated by the configuration of the launch vehicle, the established DSIF and Space-flight Operations Complexes, reliability objectives, and the nature of the spacecraft mission. The use of state-of-the-art design criteria, components of proven reliability, and the avoidance of unproven circuits or designs, contribute toward a high probability of success.

Figure 2-1 is a basic block diagram of the spacecraft system. The stowed, midcourse, and post-landing configurations of the A-21A are shown in figure 2-2. The principal elements comprising the system are listed below.

Surveyor Spacecraft Subsystem Elements

Structural and Vehicular Subsystem

Spaceframe — provides the basic structure for the spacecraft.

Landing Leg Mechanism — absorbs major portion of the shock of landing.

Crushable Blocks — absorb part of landing shock after relatively large landing leg mechanism deflections.

Antenna/Solar Panel Positioner (A/SPP) — orients the planar array antenna toward the earth and the solar panel toward the sun.

Pyrotechnic Devices — mechanically actuate pin pullers, separation nuts, tank valves, electrical power control switches, locking plungers and a detonator.

Electrical Cabling — interconnects spacecraft units.

Thermal Compartments — provide temperature-controlled environment for thermally sensitive units.

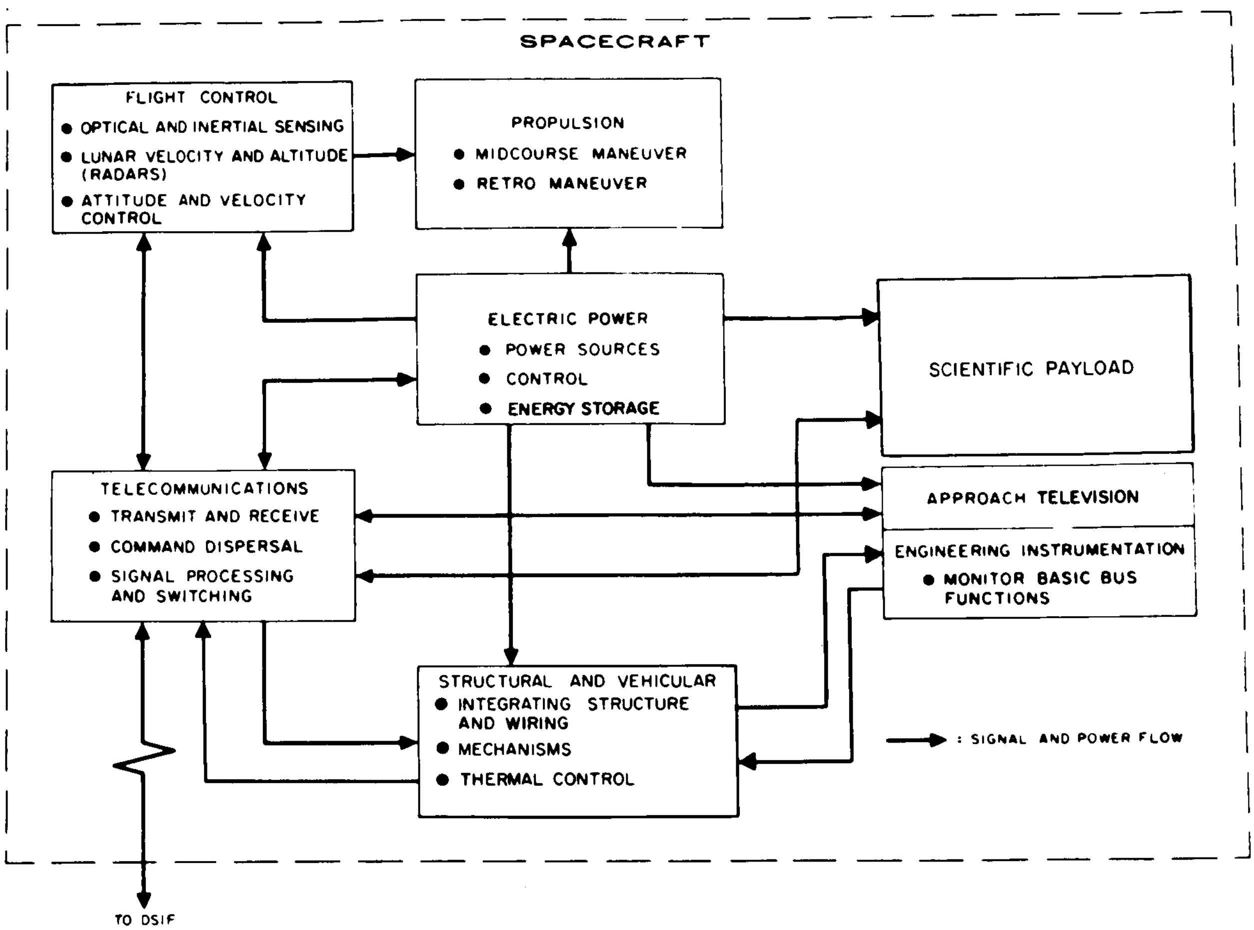


FIGURE 2-1. SPACECRAFT SYSTEM BLOCK DIAGRAM

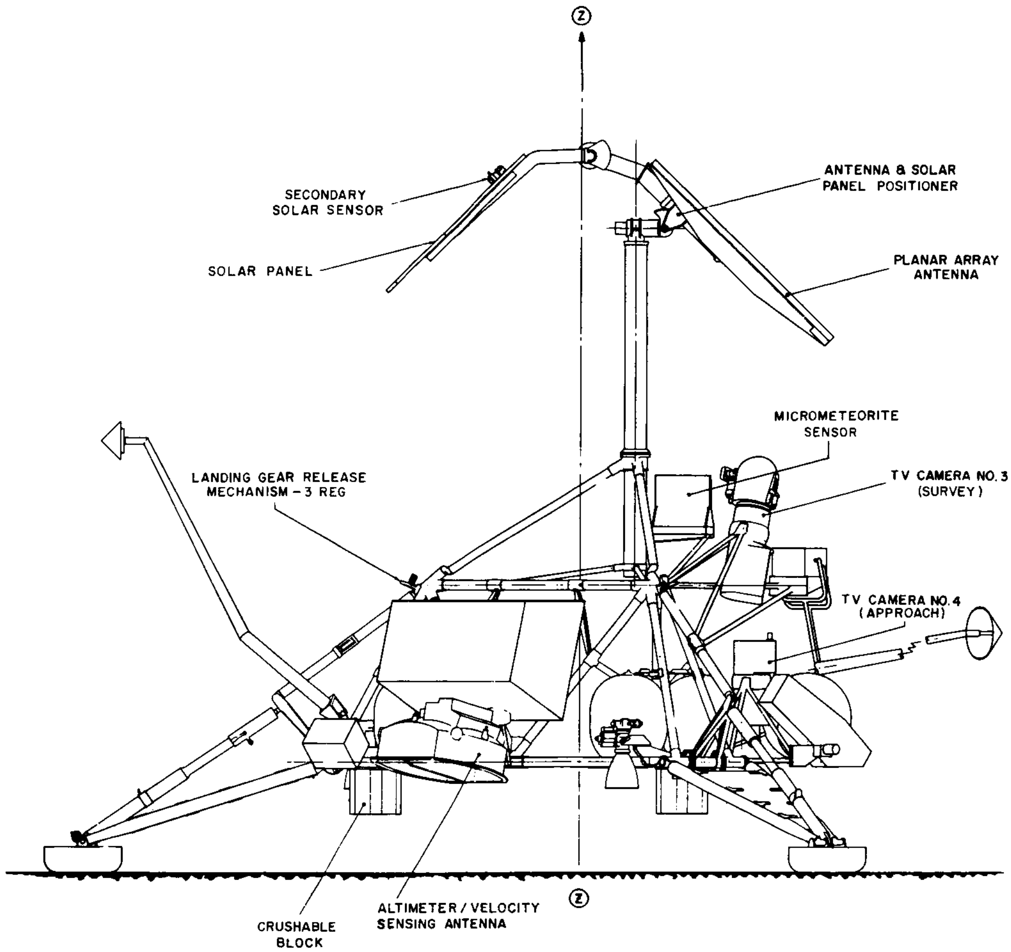
Engineering Instrumentation

Temperature and Acceleration Sensors – provide for earth monitoring of space-frame status and performance.

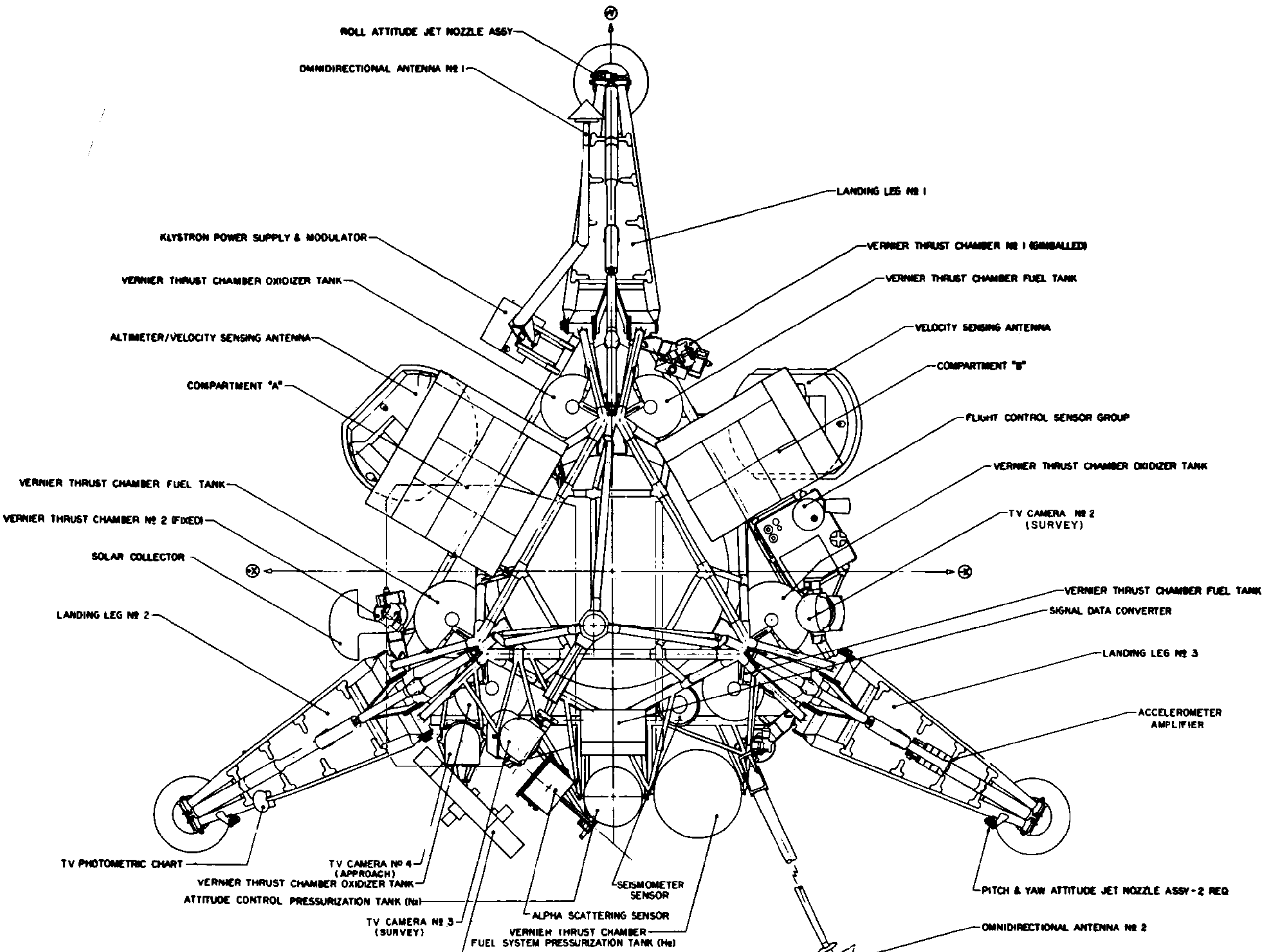
Propulsion

Vernier Engines – supply reaction forces for midcourse correction, attitude control during retro-rocket burning, and attitude and velocity control during terminal descent.

Main Retro-Rocket – decelerates spacecraft on approach to lunar surface prior to final descent.



POST LANDING CONFIGURATION



MIDCOURSE CONFIGURATION

NOTE: — ANTENNA & SOLAR PANEL POSITIONER HAS BEEN OMITTED FROM THIS VIEW FOR CLARITY

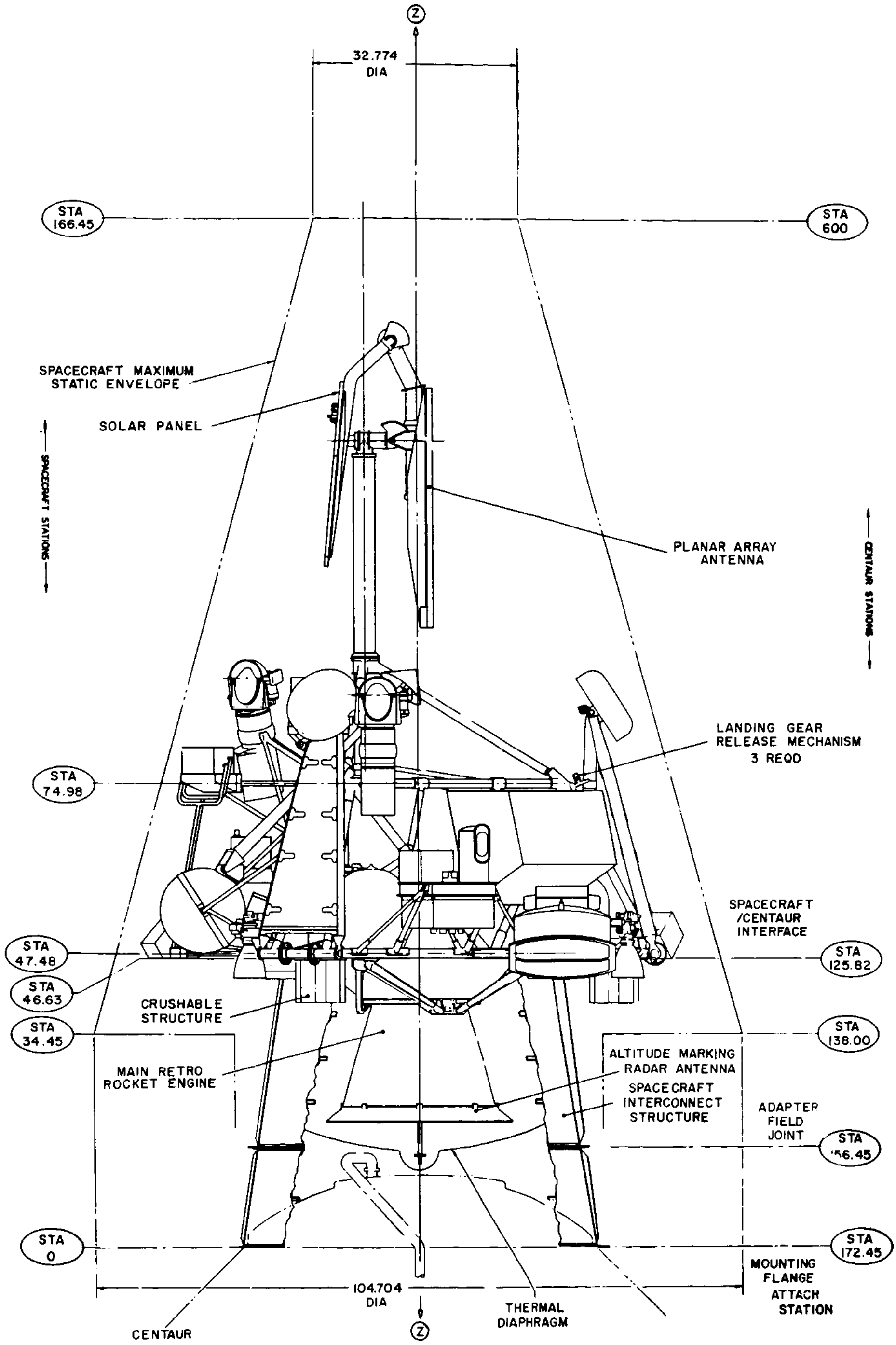


FIGURE 2-2. SPACECRAFT GENERAL ARRANGEMENT

Electrical Power

Solar Panel — charges battery and powers spacecraft during transit and lunar day.

Main Battery — provides electrical energy storage for the spacecraft.

Battery Charge Regulator (BCR) — controls and regulates battery charge from solar panel.

Boost Regulator (BR) — converts unregulated battery power to regulated power for spacecraft.

Telecommunications

Transmitters (2) — transmit engineering telemetry data in transit and engineering and scientific payload data from lunar surface.

Receiver/Transponders (2) — receive earth-transmitted commands and route these commands to the central decoder unit. Also provides two-way doppler tracking in conjunction with transmitters.

Antennas — two omnidirectional antennas for data transmission and command reception and one planar array antenna for wide band data transmission during landing and lunar phases.

Central Decoder Unit — contains a receiver-decoder selector, two central command decoders and five subsystem decoders that process earth commands and control on-off operations and time-interval operations.

Signal Processors — gather the engineering and verification signals from various subsystems and provide appropriate signal conditioning for telemetry.

Low Data Rate Auxiliary — provides for transmission at low bit rates for use with low-power transmitter and limited information bandwidth.

Flight Control

Inertial Reference Unit — provides three-axis rotational reference while spacecraft is not under control of optical or radar sensors. Also provides an acceleration reference for spacecraft flight.

Primary Sun Sensor — provides for accurate control of the spacecraft roll axis once sun acquisition is obtained.

Inertia Switch — closes at a nominal g level to predict retro rocket burnout for retro ejection timing.

Canopus Sensor — identifies and tracks the star Canopus for accurate spacecraft attitude control reference.

Flight Control Electronics — processes guidance signals from the flight control sensors (inertial, optical, and radar) for stability and maneuvering.

Secondary Sun Sensor — makes initial sun detection for gross alignment of spacecraft roll axis during transit, and for solar panel positioning toward the sun during lunar operation.

Radar — the altitude marking radar (AMR) initiates firing of the retro-rocket on approach to lunar surface. The radar altimeter and doppler velocity sensor (RADVS) measures slant range and three-axis velocity of spacecraft during descent phase, controlling the rate of descent and attitude via the vernier engines.

Attitude Jet System — provides reaction forces for spacecraft orientation maneuvers and attitude control during period from Centaur separation through pre-retro-rocket firing.

Roll Actuator — provides roll control moments during vernier engine thrust, via Vernier Engine No. 1.

Approach Television Subsystem

Approach Television Camera — provides pictures of lunar surface from a range of 1000 miles to about 80 miles above lunar surface.

Scientific Payload

Survey Television Experiment Subsystem — provides pictures of portions of lunar surface, free space, and of the spacecraft after landing.

Soil Mechanics Surface Sampler — qualitatively determines the mechanical characteristics of the lunar surface.

Alpha Scattering Experiment Subsystem — gathers information to determine lunar surface elemental composition.

Micrometeorite Detection Experiment — measures lunar ejecta resulting from micrometeorites impacting the lunar surface.

Seismometer Experiment Subsystem — measures physical disturbances on the moon.

BASIC DESIGN CONCEPTS

The primary design objective has been to maximize the probability of successful spacecraft operation within the basic limitations imposed by launch vehicle capabilities, the extent of knowledge of transit and lunar environments, and the current technological state of the art. In keeping with this primary objective, design policies have been established which (1) minimize spacecraft complexity by placing responsibility for mission control and decision making on earth-based equipment wherever possible; (2) provide the capability of transmitting a relatively large number of different data channels from the spacecraft; (3) include provisions for accommodating a relatively large number of individual commands from the earth; and (4) make all subsystems as autonomous and independent as practicable.

These basic design policies complement each other and provide a large degree of flexibility in controlling the real-time operation of the spacecraft. Complete control of spacecraft operation is achieved through a loop that is closed through earth-based equipment and decision-making processes. The only portions of spacecraft operations that are not subject to this earth/spacecraft control loop are those associated with certain portions of the attitude stabilization and terminal descent phases, and solar panel deployment where earth control is complicated by requirements for critical timing. Although this design concept places greater demands on earth-based equipment and facilities, it provides flexibility in control and data-transmission adaptability and growth potential. This concept enables the same basic spacecraft design to accommodate a wide range of possible payloads and missions.

PAYLOAD INTEGRATION

The A-21A series of Surveyor spacecraft vehicles has been designed to accommodate the following experiment subsystems:

a. Survey television experiment subsystem.

b. Soil mechanics surface sampler experiment subsystem.

c. Alpha scattering detector experiment subsystem.*

*Instruments furnished by JPL

d. Micrometeorite detector experiment subsystem.*

e. Seismometer experiment subsystem.*

Any combination of experiment subsystems may be accommodated, limited primarily by the total injection weight capability of the launch vehicle.

Flexibility is provided to accommodate the potentially wide spectrum of different electrical and functional requirements imposed on the spacecraft by the various scientific instruments in a typical payload. This flexibility is achieved through the use of electronic auxiliary units designed and built by Hughes specifically for each instrument. An experiment auxiliary unit provides the electrical/functional interface between the spacecraft basic bus and a particular instrument by conditioning and normalizing electrical signals between the two.

When the desired experiment complement for a specific spacecraft has been determined, it is only necessary to remove the appropriate auxiliaries and other elements of the experiment subsystems which are being deleted (off loaded). In some cases it may also be necessary to adjust the spacecraft lateral center of gravity by adding ballast. This approach permits the exact definition of the experiment complement to be delayed until relatively late in the spacecraft fabrication/test cycle.

*Instruments furnished by JPL

III. STRUCTURAL AND VEHICULAR SUBSYSTEM

The structural and vehicular subsystem provides support, alignment, thermal protection, electrical interconnection, mechanical actuation, and touch-down stabilization for the spacecraft and its components. This subsystem includes the basic spaceframe, landing leg mechanism, crushable blocks, omni-directional antennas mechanisms, antenna/solar panel positioner, pyrotechnic devices, electrical cabling, and thermal compartments.

SPACEFRAME

The spaceframe is the basic structure of the spacecraft. It provides mounting surfaces and attachments for the landing gear, the Centaur interconnect structure, the main retro rocket, the vernier engines and associated tanks, thermal compartments, crushable blocks, mast, flight control sensor group, descent control radars, flight control sensors, and scientific payload. Figure 3-1 illustrates the basic spaceframe. The spaceframe is made up of thin wall aluminum tubing, with the frame members interconnected to form a modified hexagon around the retro rocket. Attachment points for the retro rocket, attachment points for the Centaur interconnect structure, and landing leg hinge points are provided at the corners of the frame. The mast is attached to the top of the spaceframe and supports the planar array antenna and solar panel through a positioning mechanism.

LANDING LEG MECHANISM

The landing leg mechansim (figure 3-2) is made up of the landing leg, the intermediate A-frame, the shock absorber, the footpad, and the lock strut. The landing leg, footpad, and shock absorbers maintain attitude stability and absorb the forces of impact during touchdown. The landing legs provide long radius attachment points for the cold gas attitude control jets. The landing leg is hinged to the lower corner of the spaceframe with the aluminum honeycomb footpad attached to the outer end. The shock absorver, intermediate A-frame, and

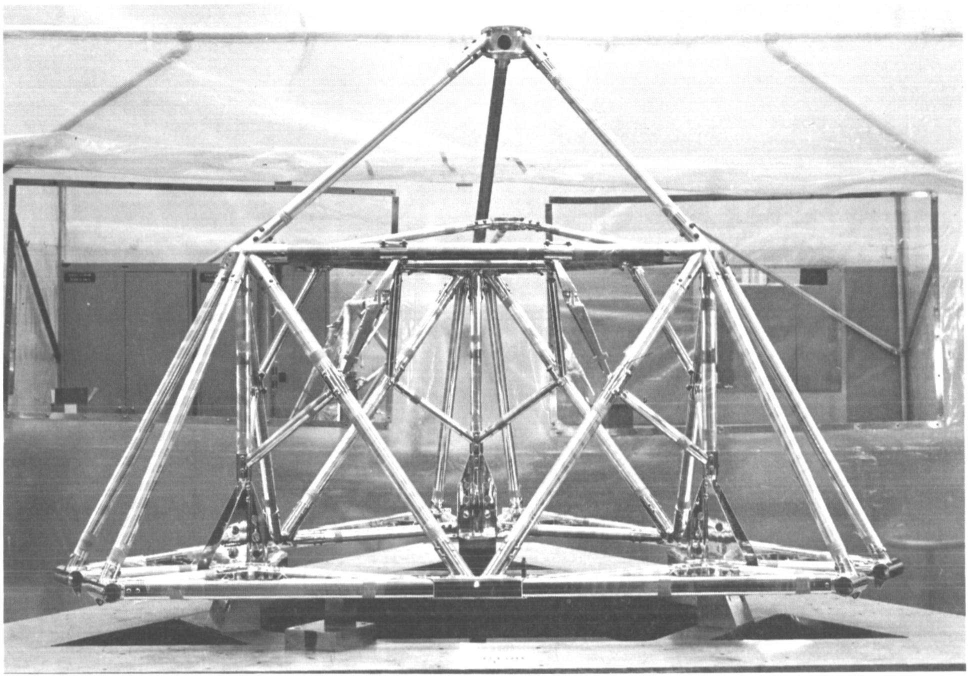


FIGURE 3-1. BASIC SPACEFRAME

FIGURE 3-2. LANDING LEG AND FOOT PAD

lock strut are interconnected to permit folded stowage of the landing gear by telescopic action of the lock strut. Torsion springs at the leg hinge extend the legs when the squib-actuated pin pullers are operated on Centaur command. The legs can also be extended by earth command. The lock strut locks in the extended position and forms, with the shock absorber, a straight line member from footpad to the spaceframe upper corner. The hydraulic shock absorber compresses with landing load and absorbs the landing shock. Crushing of the footpads absorbs some of the impact energy. After landing, the shock absorbers are locked in place by squib-actuated pin pullers.

CRUSHABLE BLOCKS

The crushable blocks of aluminum honeycomb are attached to the bottom of the spaceframe at each corner to absorb part of the shock of large landing loads. The blocks contact the lunar surface upon occurrence of any relatively large landing leg deflections and absorb energy by crushing.

OMNIDIRECTIONAL ANTENNAS MECHANISMS

The omnidirectional ("omni") antennas are mounted on the end of folding booms, hinged to the spaceframe, with omnidirectional antenna boom A near landing leg 1 and omnidirectional antenna boom B near landing leg 3. The omni antenna booms are stowed by folding against the spaceframe. Pins retain the booms in the stowed position and squib-actuated pin pullers release the booms on Centaur command. Torsion springs deploy the omni antenna after release. Omni antenna boom release is effected by a command from Centaur after the landing leg is extended and locked into position. Earth commands can also initiate omni antenna boom extension.

ANTENNA/SOLAR PANEL POSITIONER (A/SPP)

The A/SPP connects the high gain planar array and solar panel to the top of the mast by hinge connections. The planar array antenna has three axes of rotation with respect to the mast: roll, polar, and elevation. The solar panel has one axis of rotation with respect to the planar array antenna. Stepping motors rotate the axes in either direction in response to commands from earth. This freedom of movement permits orienting the planar array antenna toward the earth and the solar panel toward the sun simultaneously after landing.

PYROTECHNIC DEVICES

The pyrotechnic devices mechanically actuate the mechanisms, switches, and valves listed in table 3-1, which lists the items, their locations, functions, and quantity required. All pyrotechnic devices are vased on the "1 watt, 1 ampere, no fire for 5 minutes" range safety design requirement. A total of 36 pyrotechnic devices is used.

ELECTRICAL CABLING

The electrical cabling interconnects the spacecraft components to provide correct signal and power flow. Cable design permits installation or removal of the assemblies by disconnecting the installed wire and cable connectors. The cable connecting the two thermal compartments is routed through a thermal tunnel to minimize heat loss from the compartments. To further minimize heat losses during the lunar night, a disconnect "tear strip" is installed in the wall of compartment A. This tear strip contains leads for RADVS power, 19-ampere squib power, pre SS & A 22-volt unregulated power, and 22-volt power return. On command from the DSIF, a squib pin puller separates the tear strip, removing one part from the compartment. The aluminized Mylar super-insulation then flows in to close the hole. Figure 3-3 shows the disconnect before and after actuation. Cables exiting from the thermal compartments contain thermally insulating wire inserts (nichrome or perma-nickle wire) in all wires (except those through the tear strip) to minimize heat loss from the compartments during the lunar night. Figure 3-4 illustrates the wiring harness.

The scientific payload of the spacecraft is designed around five experiment subsystem installations, each of which may be installed or removed individually without effect on other experiment subsystems or the spacecraft basic bus, except for weight and balance. To minimize the amount of cable weight left in the vehicle when an experiment subsystem is removed, individual harnesses are employed (figures 3-4 and 3-5). A connector is installed in compartment B for each of four experiment subsystems. All wiring from the spacecraft to the experiment subsystem is contained in a separate cable assembly that can be installed or removed without compromising the reliability of the basic bus harness assemblies. The harnesses for the survey television experiment subsystem also use external basic bus connectors as TV disconnect points.

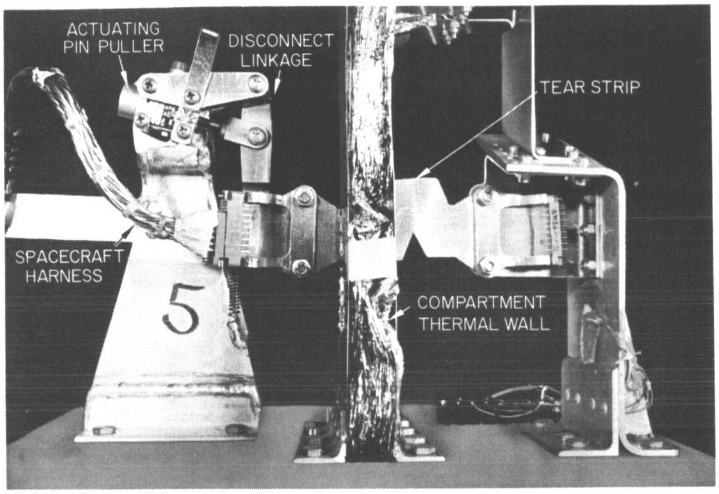
TABLE 3-1. PYROTECHNIC DEVICES

Type	Location	Quantity
Pin Puller	Omnidirectional antennas	2
	Landing leg locks	3
	Planar antenna/solar panel locks	7
	Alpha scattering deployment	3
	Soil mechanics-surface sampler	1
	Electrical harness disconnect device	1
Separation nut	Retro-rocket attachments	3
Valve operation	Helium tank valves	2
	Nitrogen tank valves	2
Locking plunger	Landing leg shock absorbers	3
	Nitrogen tank valve	1
Locking plunger	Landing leg shock absorbers	3
Pyrotechnic electrical switch	RADVS power controls	4
	(engineering mechanism auxiliary)	
Ignitor	Retro-rocket	1

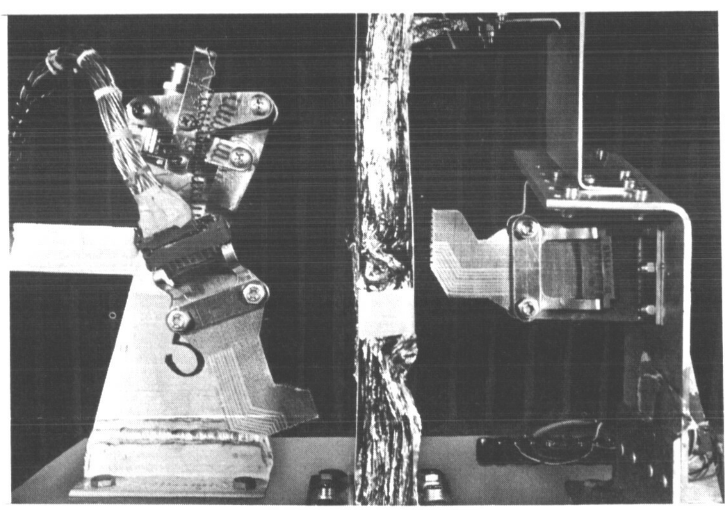
Cable assemblies are attached to the spaceframe by suitable brackets and clips. Slack cabling is provided around mechanically active points. The Centaur electrical interface is established through a 52-conductor connector assembled as part of basic bus wiring harness 1. The connector mounts on the bottom of the spaceframe between landing legs 1 and 2 and mates with a Centaur connector when the spacecraft is mounted.

THERMAL COMPARTMENTS

Two thermal compartments (A and B) are provided to house electronic items for which thermal control is needed throughout the mission. (See figure 2-2 for compartment placement.) The equipment in these compartments is mounted on a thermal tray (figure 3-6) which distributes heat throughout the compartment. A thermal shell surrounds the entire compartment to isolate it from the lunar surface environments (figure 3-7). An insulating blanket, composed of

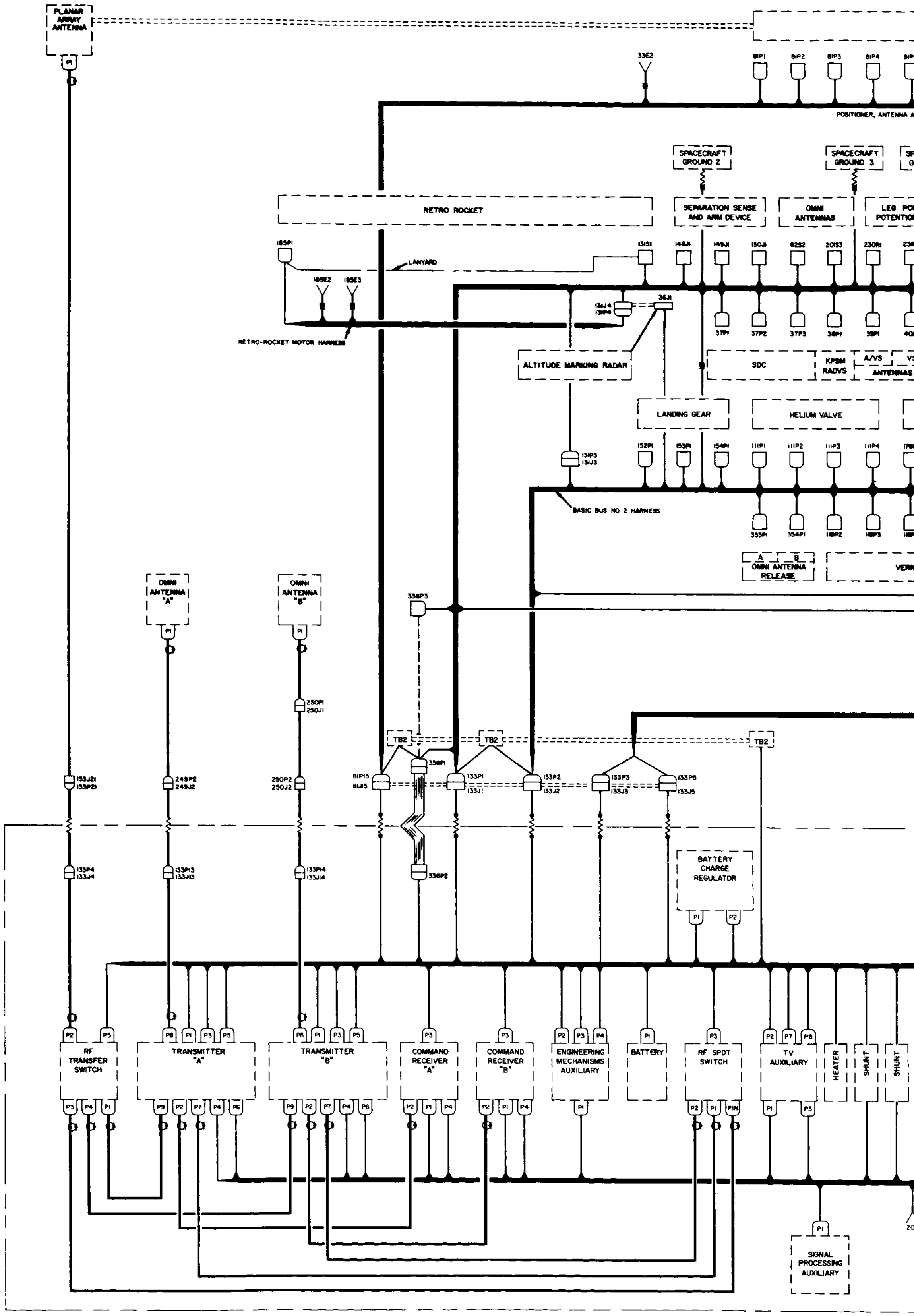


a) Before Actuation



b) After Actuation

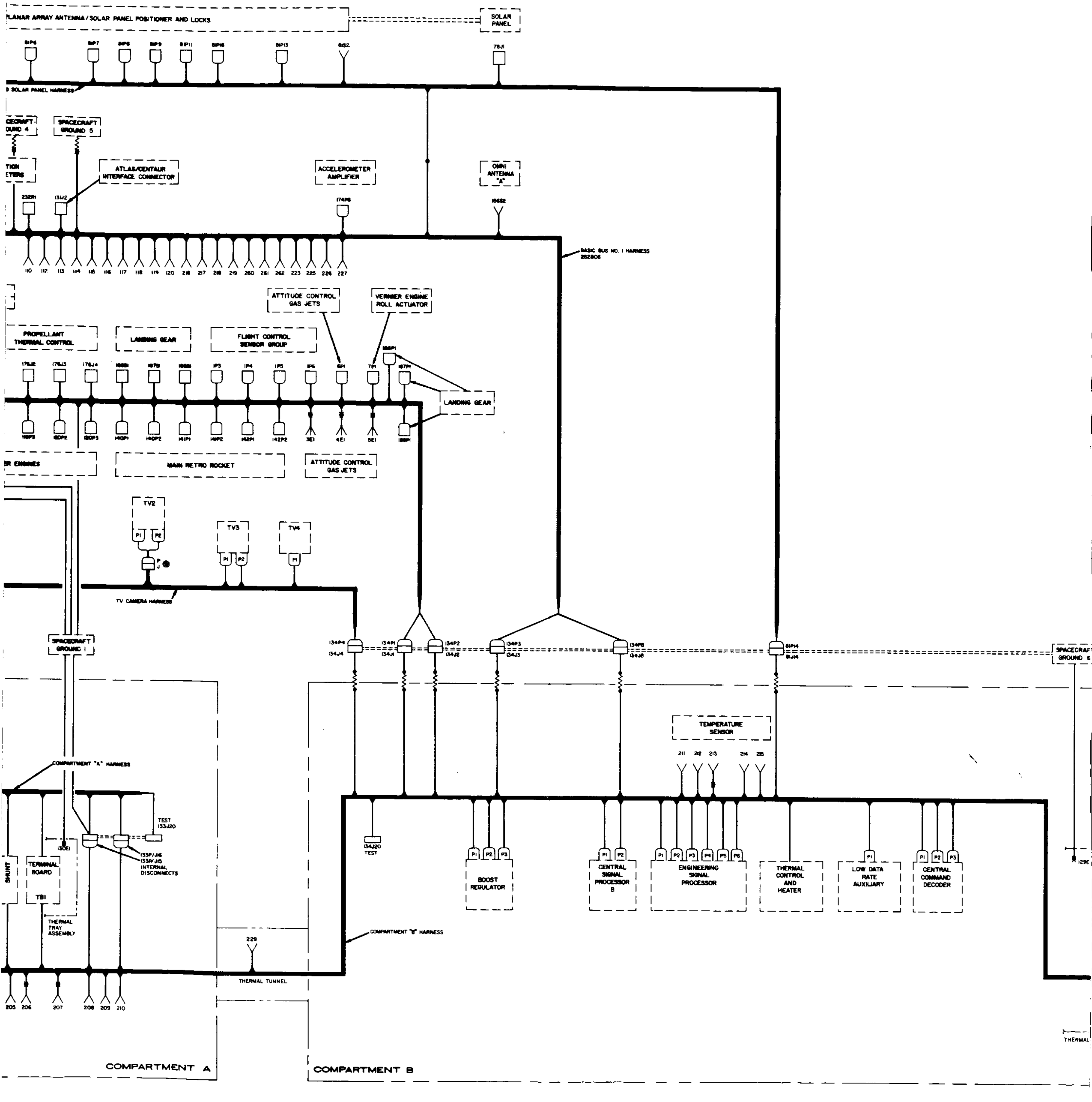
FIGURE 3-3. ELECTRICAL HARNESS DISCONNECT DEVICE



PLANAR ARRAY ANTENNA
SPACECRAFT GROUND 2
SPACECRAFT GROUND 3
RETRO ROCKET
SEPARATION SENSE AND ARM DEVICE
OMNI ANTENNAS
LANYARD
RETRO-ROCKET MOTOR HARNESS
ALTITUDE MARKING RADAR
SDC
KPSM RADVS
A/VS
ANTENNAS
LANDING GEAR
HELIUM VALVE
BASIC BUS NO 2 HARNESS
A
B
OMNI ANTENNA RELEASE
OMNI ANTENNA "A"
OMNI ANTENNA "B"
TB2
TB2
TB2
BATTERY CHARGE REGULATOR
RF TRANSFER SWITCH
TRANSMITTER "A"
TRANSMITTER "B"
COMMAND RECEIVER "A"
COMMAND RECEIVER "B"
ENGINEERING MECHANISMS AUXILIARY
BATTERY
RF SPDT SWITCH
TV AUXILIARY
HEATER
SHUNT
SHUNT
SIGNAL PROCESSING AUXILIARY

A

23-1


PLANAR ARRAY ANTENNA/SOLAR PANEL POSITIONER AND LOCKS
SOLAR PANEL
SOLAR PANEL HARNESS
SPACECRAFT GROUND 5
ATLAS/CENTAUR INTERFACE CONNECTOR
ACCELEROMETER AMPLIFIER
OMNI ANTENNA "A"
BASIC BUS NO. 1 HARNESS 282806
ATTITUDE CONTROL GAS JETS
VERNIER ENGINE ROLL ACTUATOR
PROPELLANT THERMAL CONTROL
LANDING GEAR
FLIGHT CONTROL SENSOR GROUP
MAIN RETRO ROCKET
ATTITUDE CONTROL GAS JETS
TV2
TV3
TV4
TV CAMERA HARNESS
SPACECRAFT GROUND 1
SPACECRAFT GROUND 6
TEMPERATURE SENSOR
COMPARTMENT "A" HARNESS
TEST 133J20
TERMINAL BOARD
TB1
THERMAL TRAY ASSEMBLY
133P/J16 133P/J15 INTERNAL DISCONNECTS
134J20 TEST
BOOST REGULATOR
CENTRAL SIGNAL PROCESSOR B
ENGINEERING SIGNAL PROCESSOR
THERMAL CONTROL AND HEATER
LOW DATA RATE AUXILIARY
CENTRAL COMMAND DECODER
COMPARTMENT "B" HARNESS
THERMAL TUNNEL
COMPARTMENT A
COMPARTMENT B

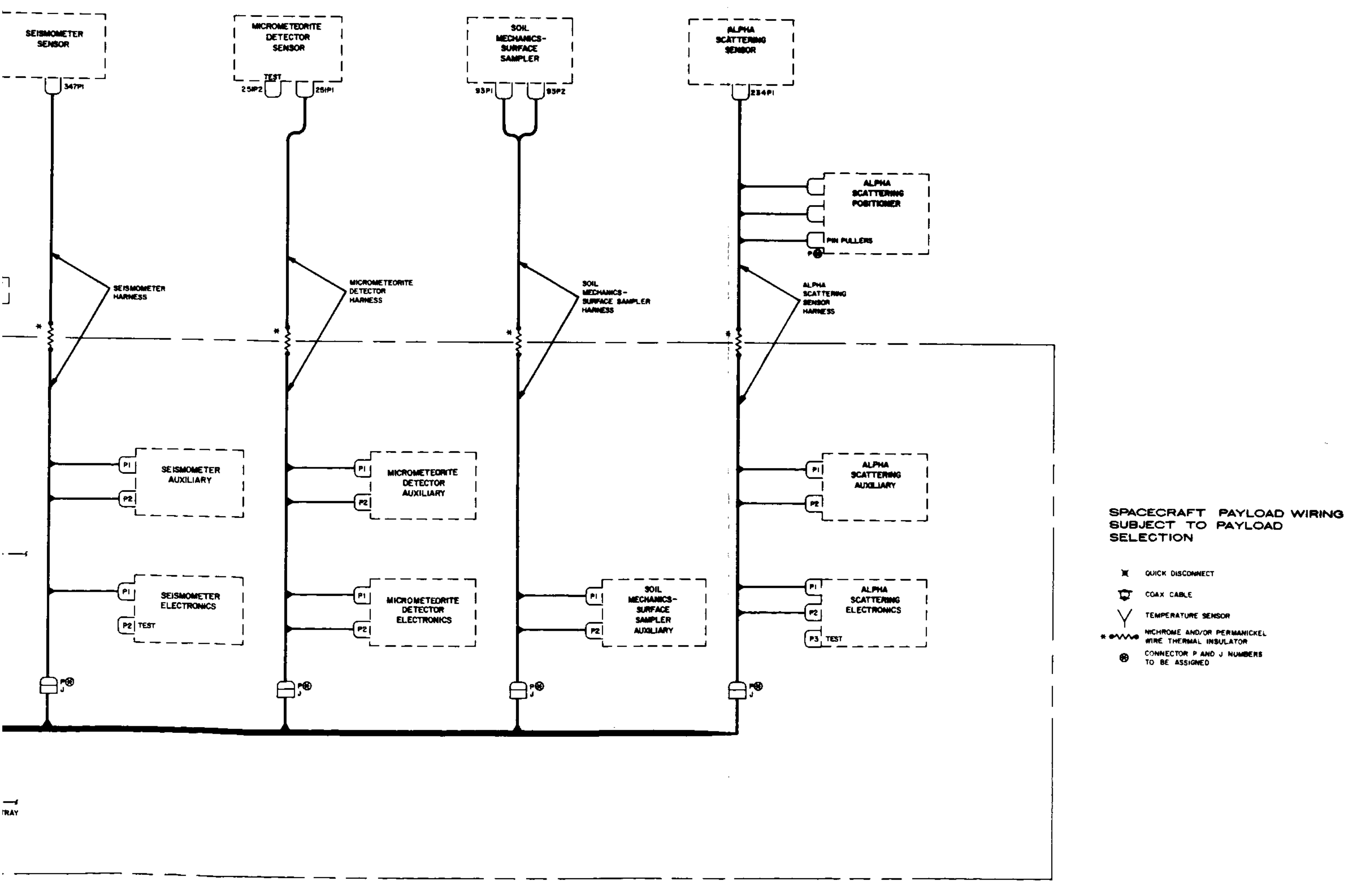


FIGURE 3-4. TYPICAL HARNESS INTERCONNECTION

B

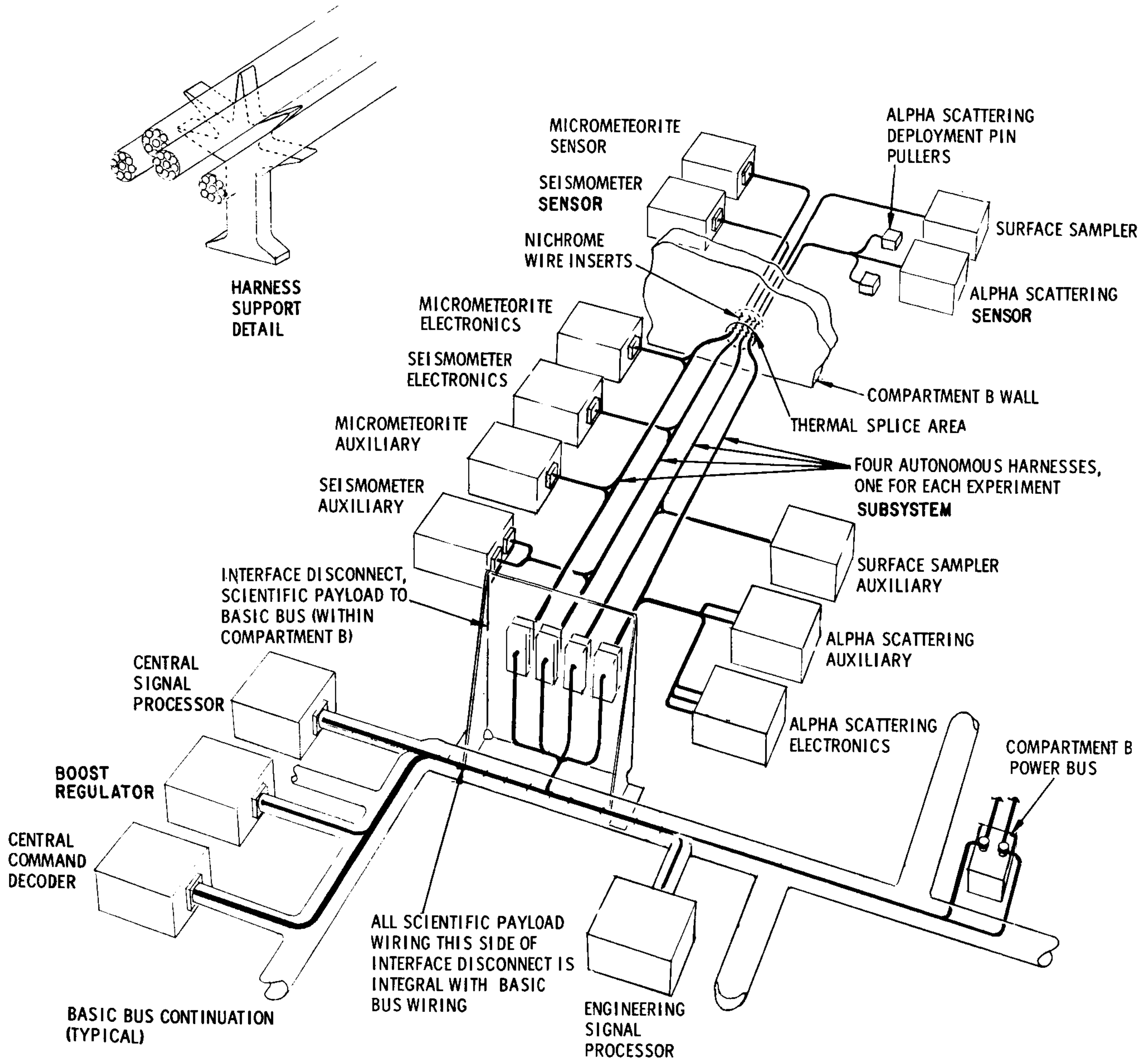


FIGURE 3-5. HARNESS REMOVAL CONCEPT

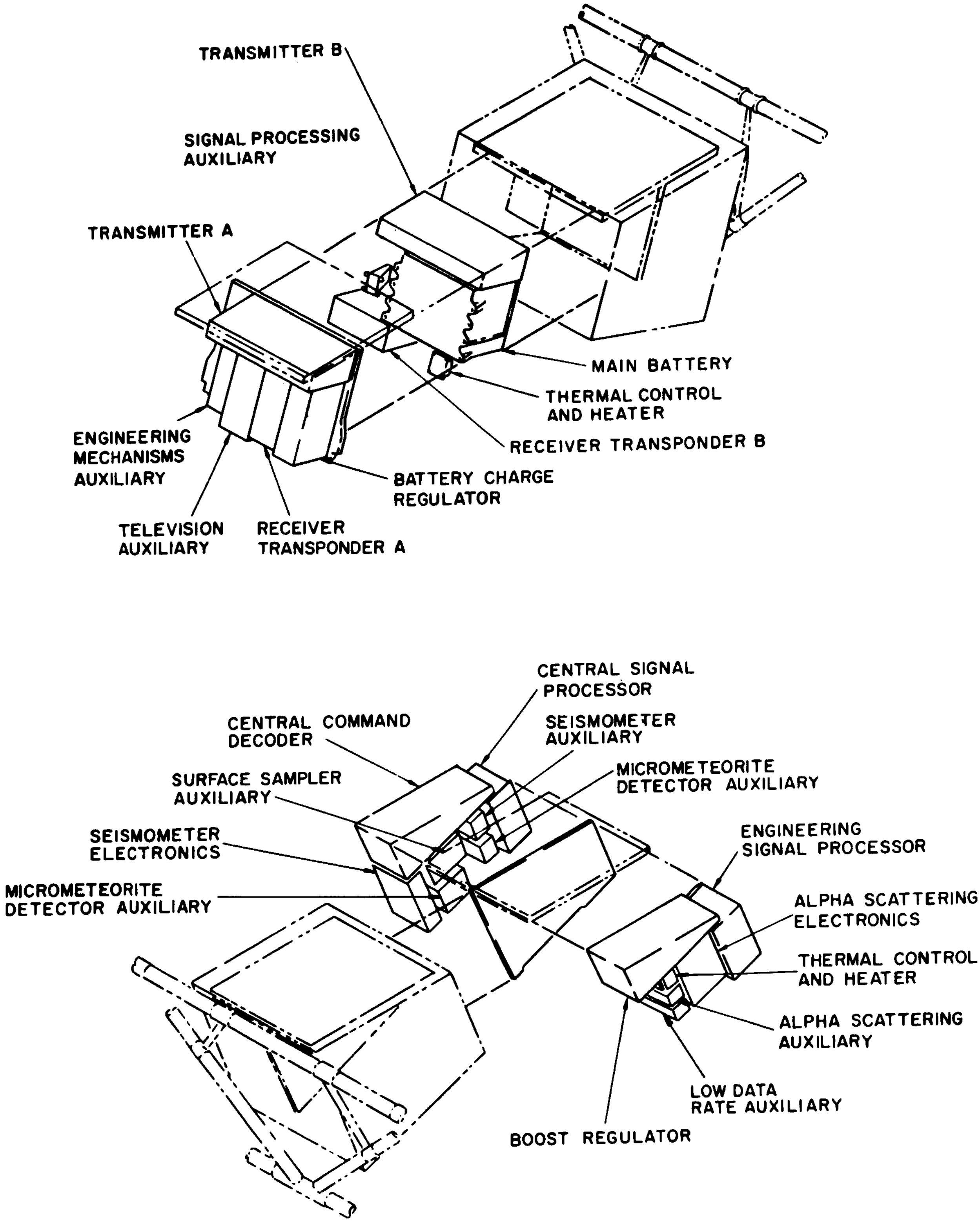
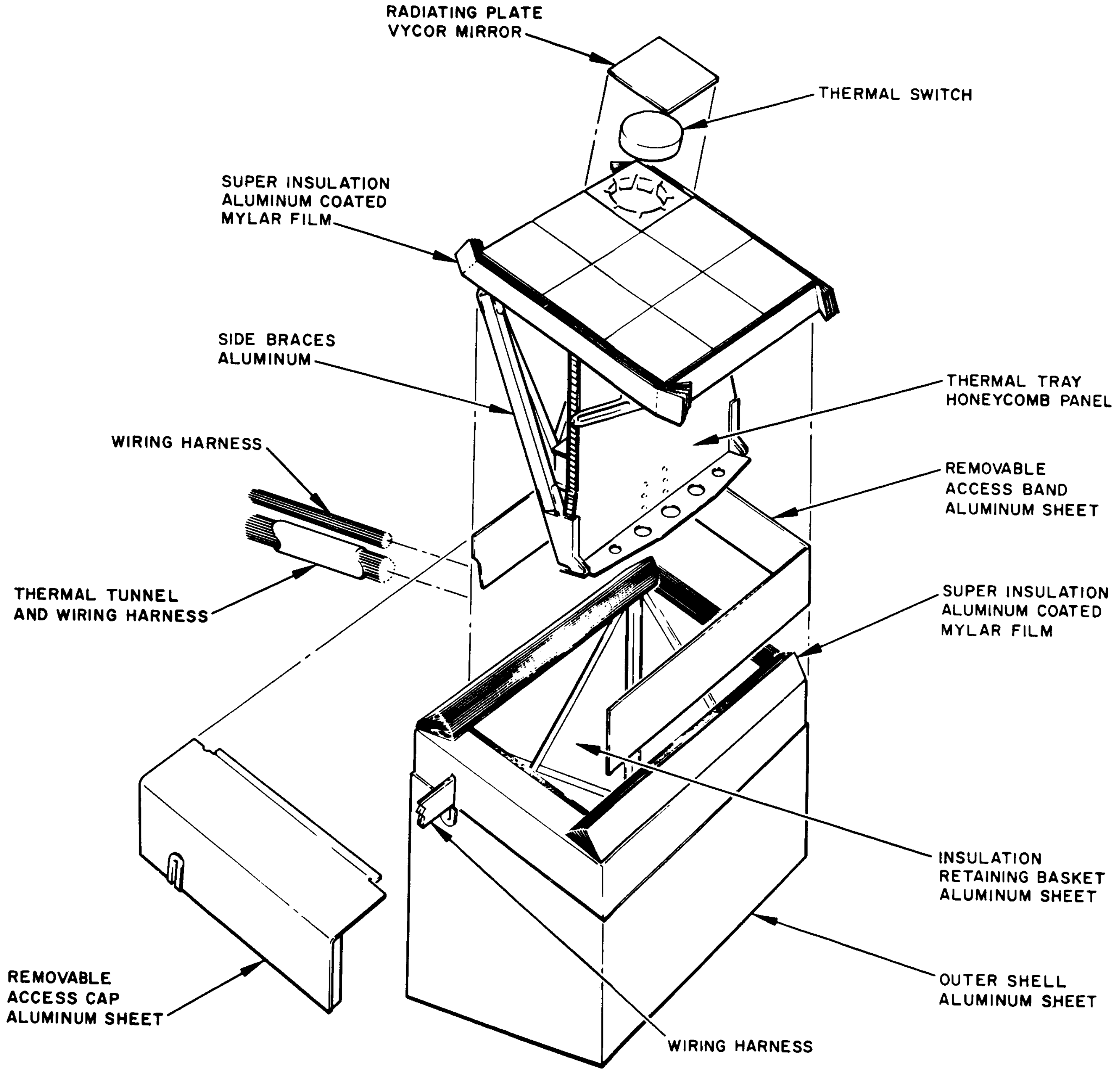


FIGURE 3-6. THERMAL TRAY ASSEMBLY



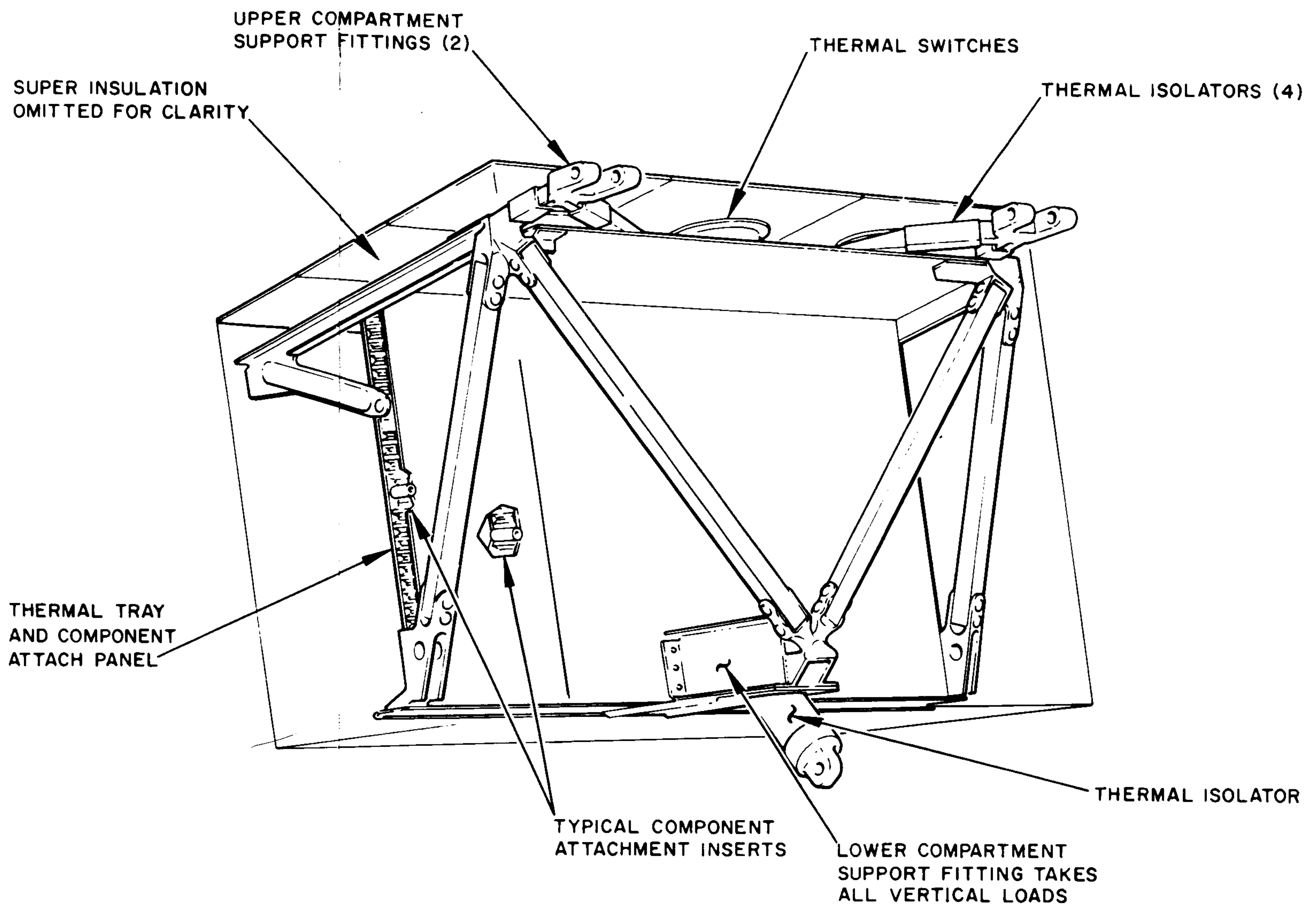


FIGURE 3-7. TYPICAL THERMAL COMPARTMENT DESIGN

27-1 27-2

approximately 75 sheets of 1/4 mil thick mylar, is installed between the inner shell and the outer protective cover of the compartment. Each of these sheets is coated on one side with an extremely thin layer of vacuum-deposited aluminum to enhance its insulating qualities. The compartment design employs thermal switches (figure 3-8) which are thermo-mechanical devices, capable of varying the thermal conductance between the inner compartment and the external radiating surface. These thermal switches automatically provide a conductive path to the radiating surface to permit dissipation of electrically generated heat from within the compartments, thereby maintaining the thermal tray temperature below +125° F. Compartment dissipating capability is illustrated in figure 3-9. The compartments each contain a thermal control and heater assembly to maintain the temperature of the thermal tray above 40° F for compartment A and above 0° F for compartment B. Components located within the compartments are identified in table 3-2 and figure 3-6. The heater control system may be bypassed by earth command if this becomes necessary.

Definitive and descriptive documents for the structural and vehicular subsystem are listed in Appendix A, items 1 thru 10.

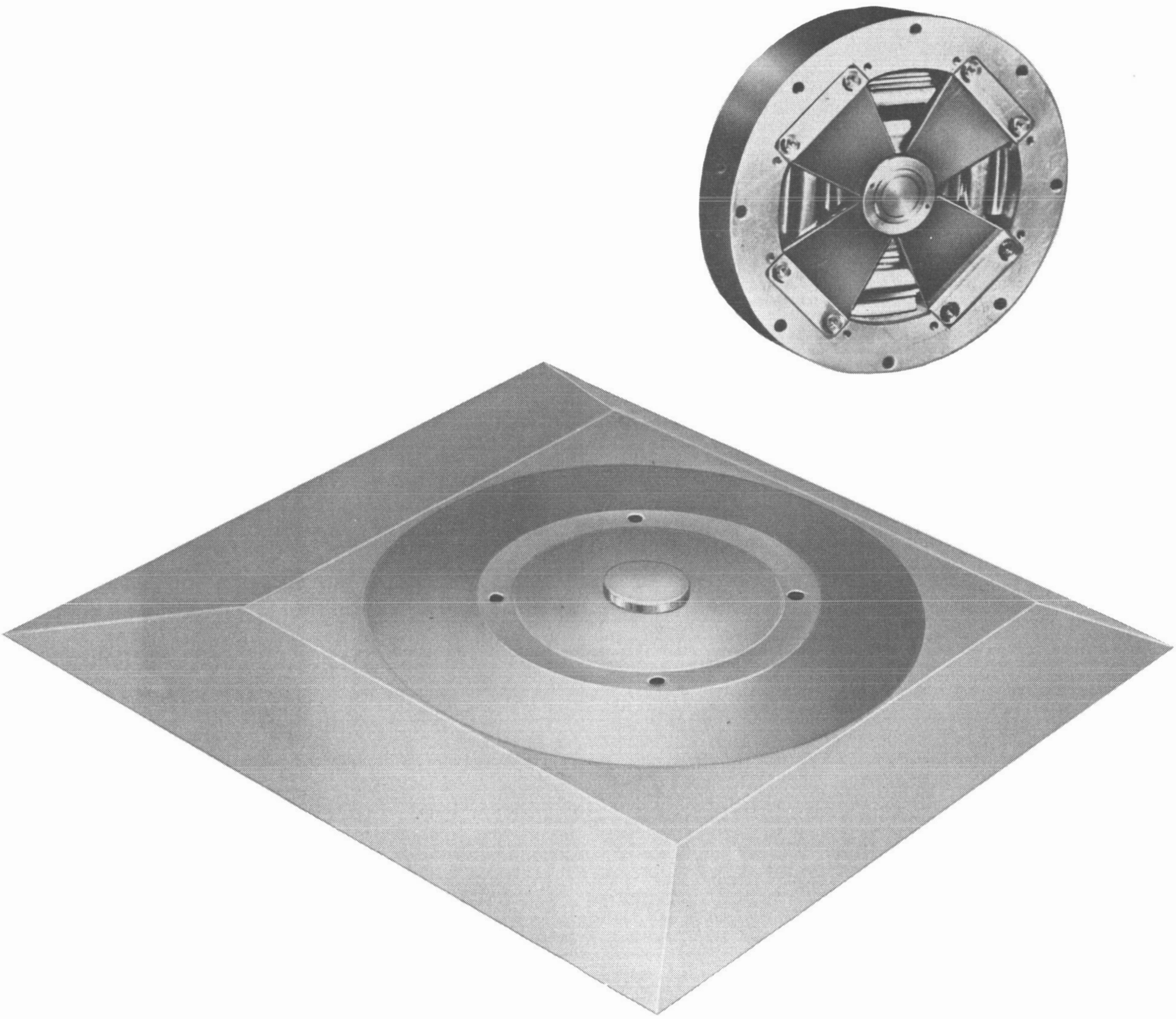


FIGURE 3-8. THERMAL SWITCH

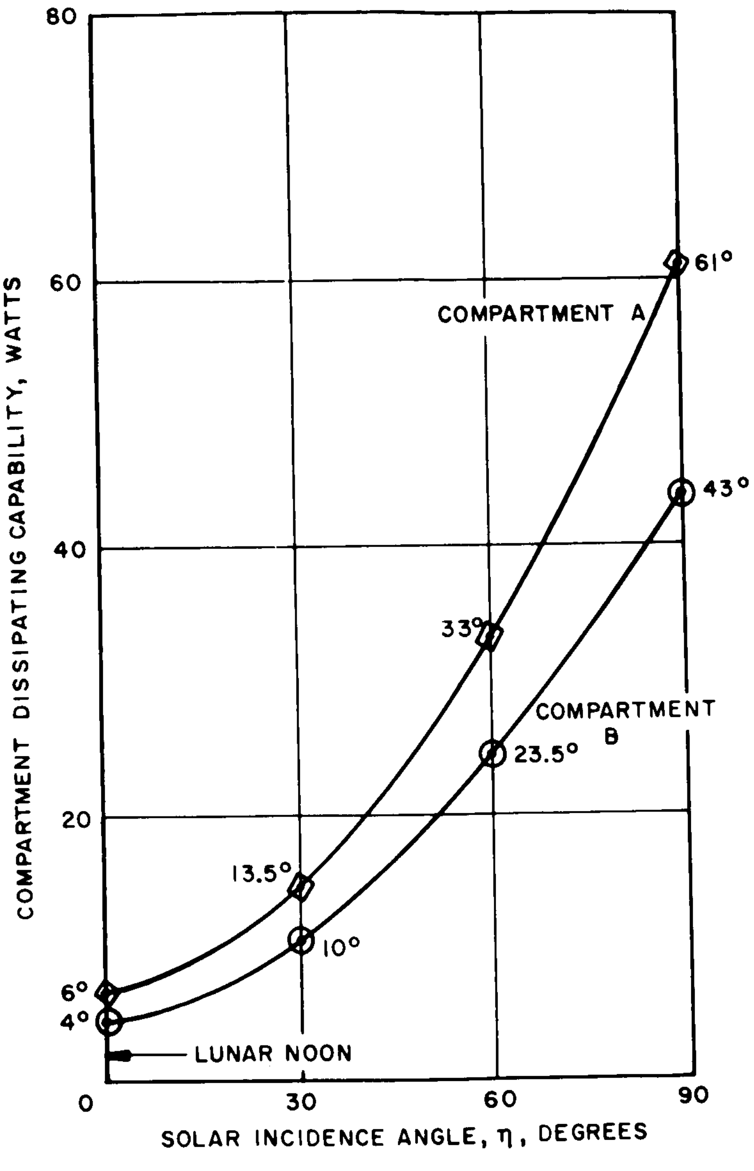


FIGURE 3-9. COMPARTMENT DISSIPATING CAPABILITY

TABLE 3-2. THERMAL COMPARTMENT COMPONENT INSTALLATION

Compartment A	Compartment B
Receivers (2)	Central command decoder
Transmitters (2)	Boost regulator
Rf SPDT switch	Central signal processor
Signal processing auxiliary	Engineering signal processor
Rf transfer switch	Low data rate auxiliary
Battery	Thermal control and heater assembly
Battery charge regulator	Resistor, thermal calibrated temperature sensor
Engineering mechanisms auxiliary	Wiring harness, compartment B
Television auxiliary*	Soil mechanics-surface sampler auxiliary*
Thermal control and heater assembly	Seismometer auxiliary*
Resistor, thermal calibrated temperature sensor	Seismometer electronics*
Meter shunt	Micrometeorite detector auxiliary*
Wiring harness, compartment A	Micrometeorite detector electronics*
	Alpha scattering auxiliary*
	Alpha scattering electronics*

*Part of scientific payload.

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IV. ENGINEERING INSTRUMENTATION

Temperature and acceleration sensors are provided for telemetric monitoring of spaceframe status and performance. There are two types of temperature sensors, a basic sensor, and a high accuracy sensor, both of which are made up of platinum resistance wire. The basic temperature sensors are provided with a 5.0 ma constant current source and the high accuracy temperature sensors are provided with a 2.5 ma constant current source. The constant current sources are contained in the ESP.

There are sixty-three temperature sensors included in the basic bus, all of which have the capability of being monitored while still on the launching pad. These sensors are distributed among the spacecraft units as follows:

Flight control units	7 sensors
Mechanisms	3 sensors
Radar units	6 sensors
Electrical power units	3 sensors
Transmitters	2 sensors
Approach TV	1 sensor
Vehicle structure	25 sensors
Propulsion units	15 sensors
Survey TV	1 sensor

Accelerometers, switches, and potentiometers are installed to measure loading and displacement during the thrust, transit, and landing phases. Three of the accelerometers are installed near the retro-rocket/Centaur attachment points and one on the FCSG. Accelerometer amplifiers are installed on landing leg 3. Full-scale range of the accelerometer system is approximately ±15 g peak. Position of the landing legs is measured by a potentiometer at the leg hinge points. Full extension of the landing lock struts and omni antennas is indicated by mechanism-actuated switches. Lock strut 1 has an "omni antenna extend" switch to inhibit omni antenna extension until landing legs are fully deployed.

Definitive and descriptive documents for the engineering instrumentation are listed in Appendix A, items 11 thru 13.

V. PROPULSION SUBSYSTEM

The propulsion subsystem components supply the reaction forces for maneuvering the spacecraft during the midcourse correction and lunar landing phases of the mission. Figure 5-1 illustrates the elements of the propulsion and flight control subsystems.

The propulsion subsystem consists of three-hypergolic-fueled variable-thrust, vernier engines for midcourse velocity vector correction and landing phase maneuvering and a solid propellant retro-rocket engine for supplying the principal deceleration force during the landing maneuver. The propulsion subsystem is controlled by flight control through preprogrammed maneuvers. commands from earth, and maneuvers initiated by flight control sensor signals.

VERNIER ENGINES

The vernier engine system (figure 5-2) supplies the reaction forces for midcourse maneuver velocity vector correction, attitude control during retro-rocket engine burning, and velocity vector and attitude control during terminal descent to the lunar surface. The vernier engine system consists of three thrust chamber assemblies and a feed system. The feed system is composed of three fuel tanks, three oxidizer tanks, a high-pressure helium tank, propellant lines, and the necessary valves for system arming, operation, and deactivation.

The thrust of each engine can be throttled over a range of approximately 30 to 104 pounds by a bipropellant throttling valve using control signals supplied by flight control (see Appendix B, item 27). Engine firing is accomplished by solenoid-controlled, helium-actuated on/off valves controlled by the flight control. Throttling valves are controlled individually while the on/off valves operate collectively from a single signal. Vent valves permit purging of the engine cooling jacket to clear out decomposed gases. The oxidizer is nitrogen tetroxide (N_2O_4) with 10 percent by volume nitric oxide (NO) to depress the freezing point. The fuel is monomethyl hydrazine monohydrate (72 MMH. 28 H_2O). Fuel and oxidizer ignite hypergolically when mixed in the thrust chamber.

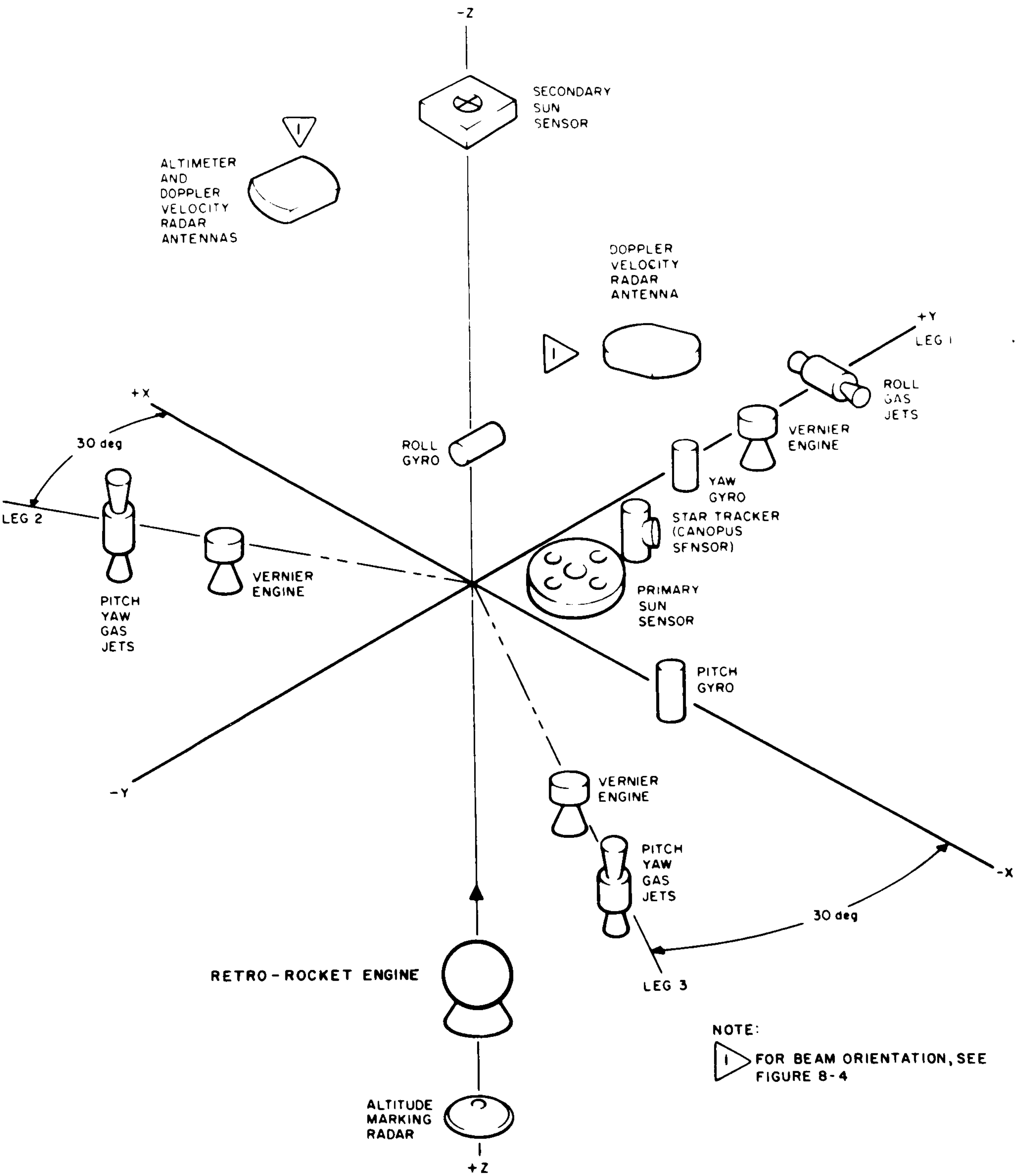


FIGURE 5-1. ELEMENTS OF PROPULSION AND FLIGHT CONTROL SUBSYSTEM

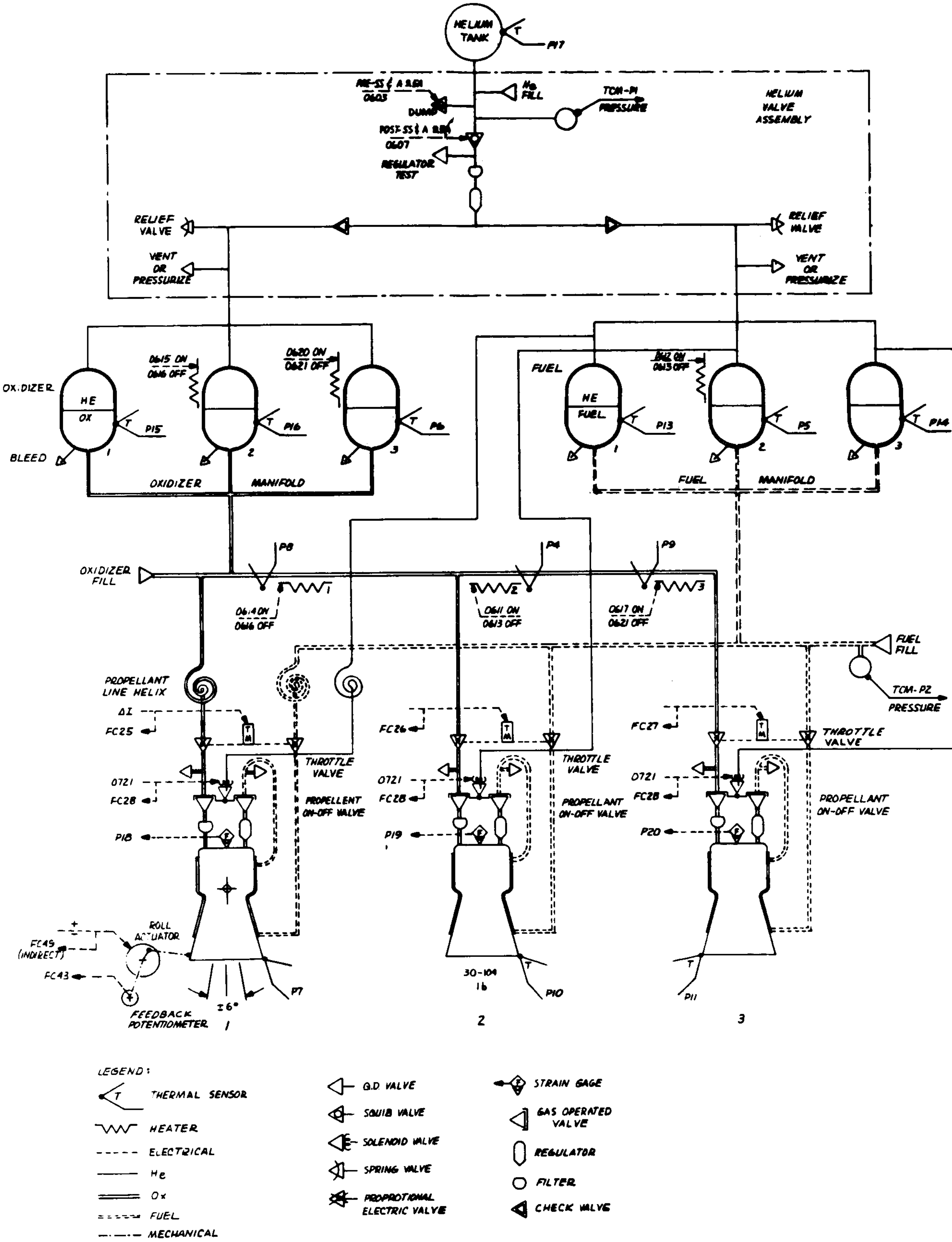


FIGURE 5-2. VERNIER PROPULSION SYSTEM, FUNCTIONAL SCHEMATIC DIAGRAM

The thrust chambers (figures 5-3 and 5-4) are located near the hinge points of the three landing legs on the bottom of the main spaceframe. Number 1 engine is hinge (swivel) mounted on an electro-mechanical motor roll actuator to rotate engine about an axis in the spacecraft x-y plane for roll control. The spacecraft control moment arm of each engine is approximately 36 inches in length. The specific impulse and total impulse vary with engine thrust. The approximate thrust of each engine is monitored by strain gages installed on each engine mounting bracket. The thermal control design of the vernier engine and feed system maintains the temperature of all portions of the system between 0° and 100° F during nonthrust periods, from launch to touchdown, preventing propellant freezing or overheating by a combination of active and passive thermal controls utilizing surface coatings and electrical and solar heating.

Other system components are thermally isolated from the spaceframe to ensure that the spacecraft structure acts as neither a heat source nor heat sink. Fuel and oxidizer are each contained in three tanks with one pair of tanks near each engine. Fuel and oxidizer tanks each have an interconnecting propellant manifold line system to all tanks and all engines. The arrangement of the tanks on the spaceframe is illustrated in figure 5-5. Tanks, and some segments of the propellant lines are electrically heated to condition the propellant temperature from 20° to 100° F. Thermal sensors on all tanks, all engines, the helium tank, and the three propellant line segments permit telemetering thermal data for DSIF monitoring. Fuel and oxidizer tanks each contain positive expulsion bladders which deflate around the central standpipe to permit complete expulsion and assure propellant cohesion under zero-g conditions. Helium release and dump valves are squib-operated units activated by 9.5 ampere pulsed constant current sources. The helium tank stores gas under pressure to force the propellants into the thrust chambers. Valves permit release of helium to the system, regulation of pressure, and dumping of residual helium. Tank capacity is given in Appendix B, item 27.

MAIN RETRO-ROCKET

The main retro-rocket (figure 5-6), which performs the major portion of the deceleration of the spacecraft during lunar landing maneuver, is a spherical, solid-propellant unit with a partially submerged nozzle.

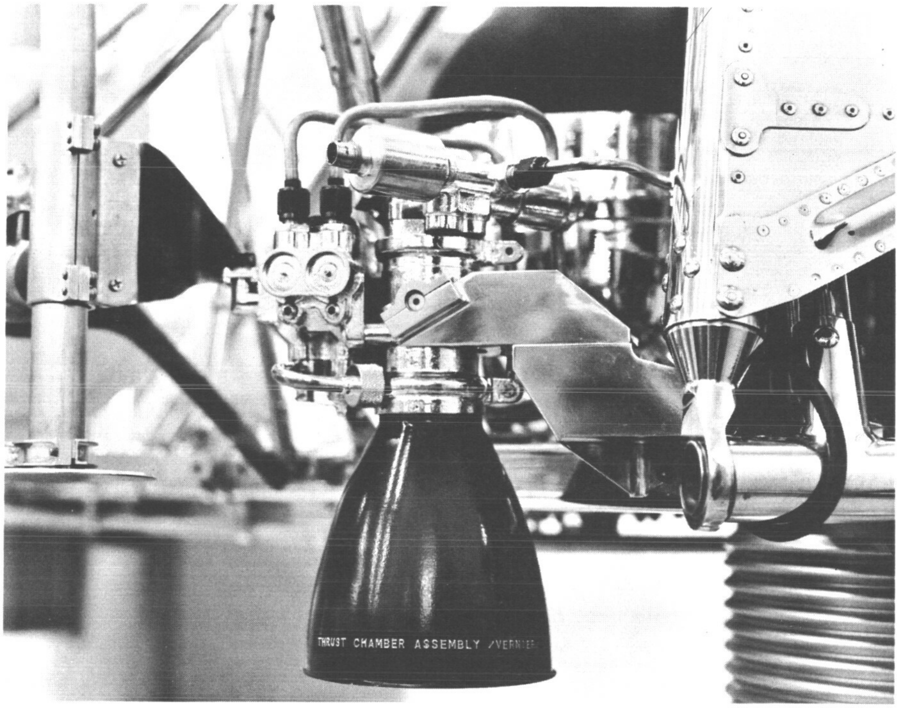


FIGURE 5-3. VERNIER ENGINE ASSEMBLY

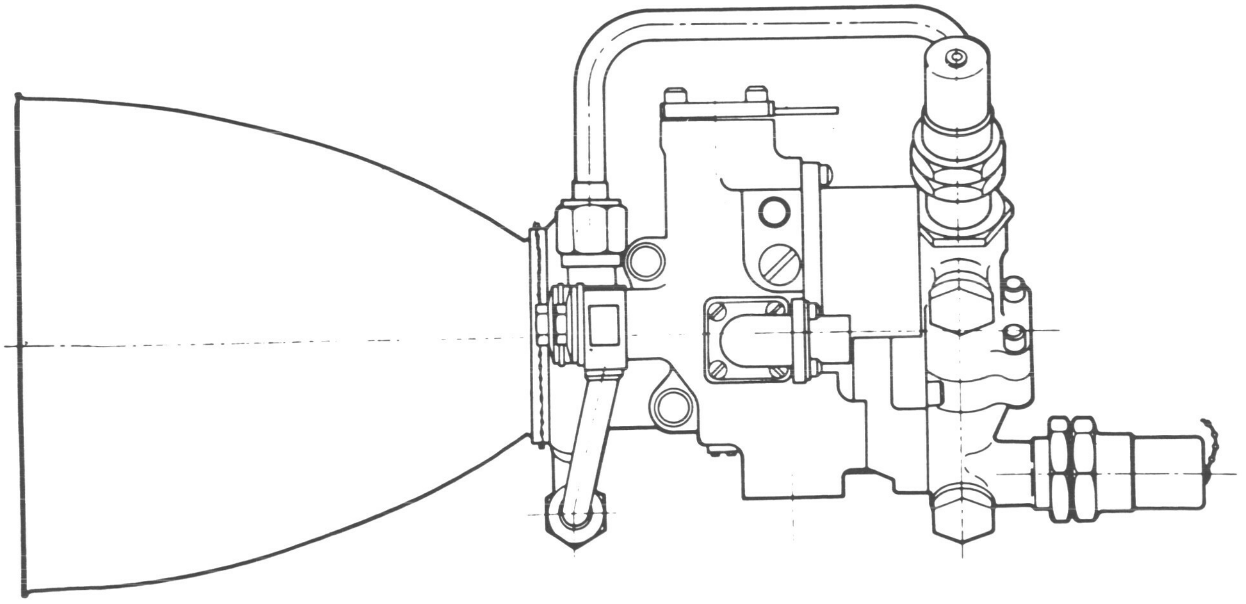


FIGURE 5-4. VERNIER THRUST CHAMBER

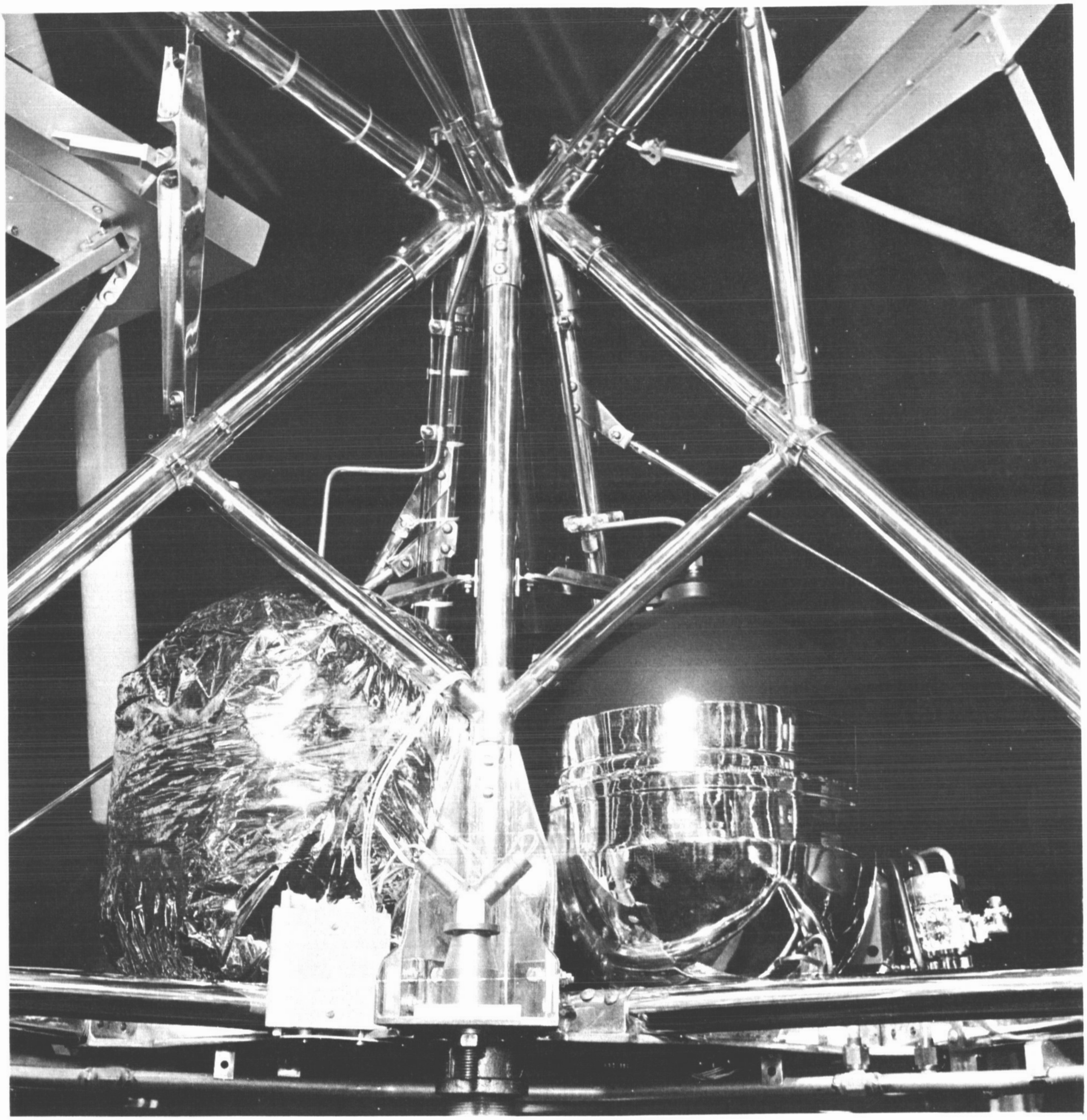
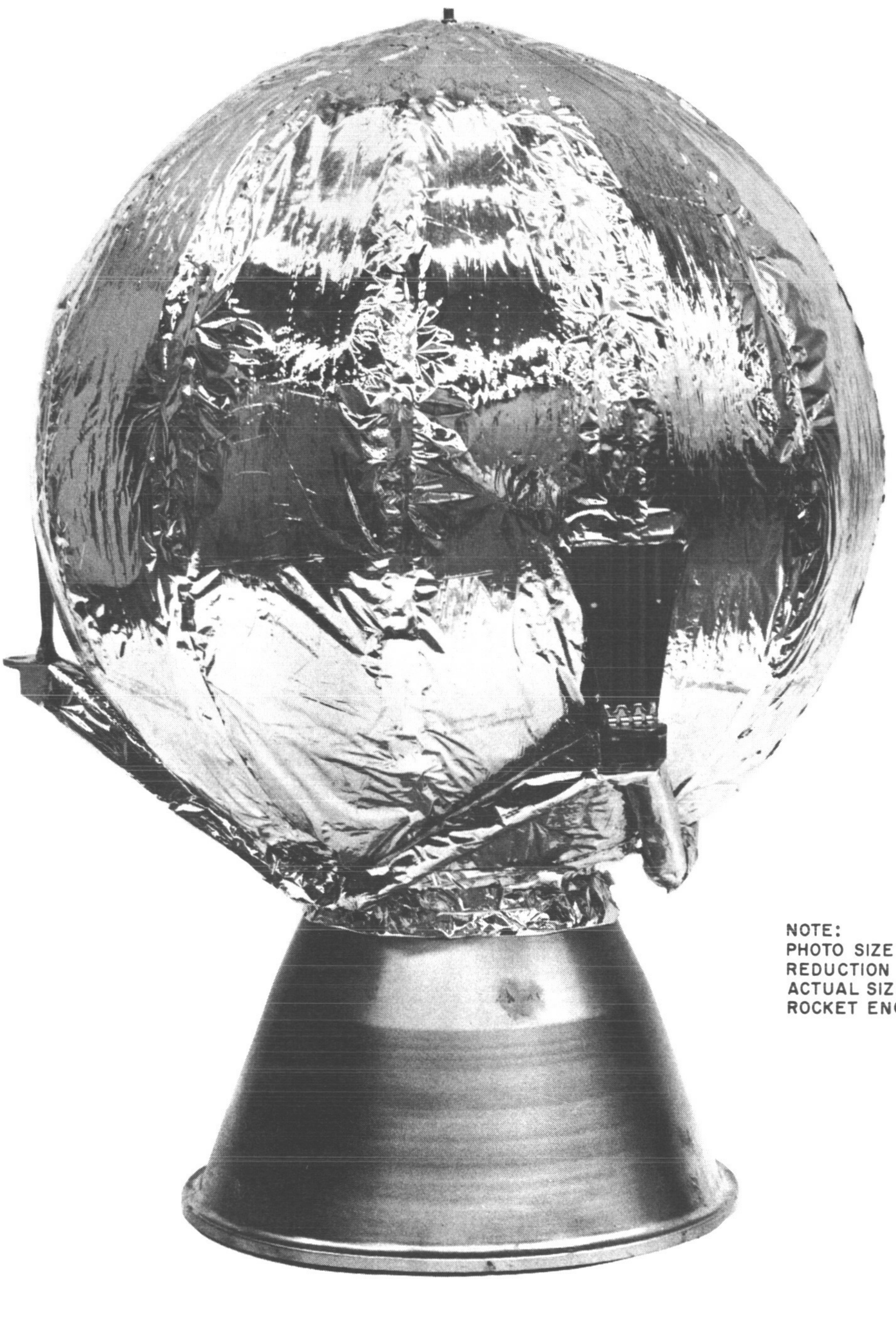


FIGURE 5-5. VERNIER PROPULSION TANKS AND SPACEFRAME



NOTE:
PHOTO SIZE IS 10:1
REDUCTION FROM
ACTUAL SIZE OF RETRO
ROCKET ENGINE.

FIGURE 5-6. MAIN RETRO-ROCKET ENGINE

The unit is attached at three points on the main spaceframe near the landing leg hinges, with explosive nut disconnects for post-firing ejection. Friction clips around the nozzle flange provide attachment points for the altitude marking radar. Retro-rocket igniter gas pressure ejects the altitude marking radar when the retro firing sequence is initiated. Retro-rocket ignition squibs and retro release explosive nuts operate from a pulsed 19-ampere constant-current source. Commands are initiated by the flight control system.

The retro-rocket safety and arming device (required by the Air Force Eastern Test Range) has dual firing single bridgewire squibs for firing the retro-rocket igniter. In addition, provisions for local and remote safe and arm actuation and remote indication of inadvertent firing of the squibs are included. Both mechanical and electrical isolation exists between squib initiator and pyrogen igniter in the safe condition.

The retro-rocket with propellant, illustrated in cross section by figure 5-7, weighs approximately 1332 pounds. The engine thrust may vary from 8000 to 10,000 pounds over the temperature range of 50° to 70° F. The required total impulse is 50,000 lb sec (see Appendix B, item 27).

A strain gage is installed on the motor case surface for telemetering case pressure information during firing. Three thermal sensors are installed for monitoring retro-rocket nozzle temperature before ignition.

The thermal control design of the retro-rocket engine is completely passive, depending on its own thermal capacity, insulating blankets, and surface coatings to maintain the "cold spot" propellant temperature above 17° F at the time of ignition. Because of the thermal gradient through the engine and the prelaunch engine temperature, the 17° F temperature will be reached at the three engine attachment points only. The bulk temperature of the propellant grain will be above +50° F.

The retro-rocket engine is of spherical design utilizing a partially-submerged nozzle to minimize overall length. The engine utilizes an existing PBAA composite-type propellant and conventional grain geometry. The nozzle has a graphite throat and a plastic exit cone. The case is of high-strength steel insulated with an asbestos-filled phenolic to maintain the case at a low temperature level during burning.

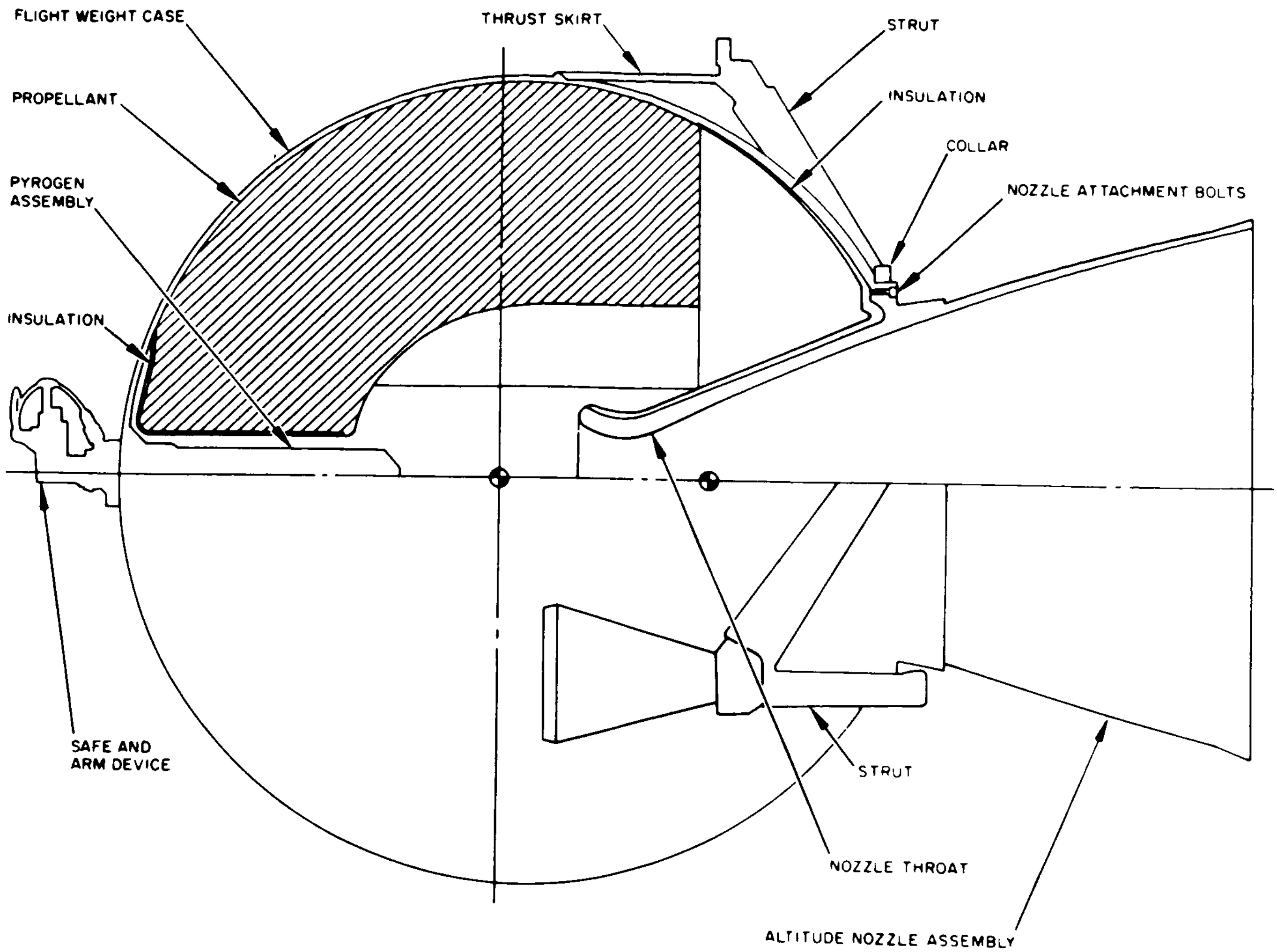


FIGURE 5-7. RETRO-ROCKET ASSEMBLY

Definitive and descriptive documents for the Propulsion Subsystem are listed in Appendix A, items 14 thru 17.

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VI. ELECTRICAL POWER SUBSYSTEM

Electrical power is supplied by a solar cell array (solar panel) whose position can be oriented with respect to the sun by command, and a sealed primary silver zinc main battery. Three 29-volt dc regulated buses and a 22-volt dc unregulated bus are provided by the electrical power subsystem to the spacecraft. Figure 6-1 is a block diagram of the power subsystem.

A 29-volt "essential" bus is provided so that power to the central command decoder will not be interrupted in the event that the overload trip circuit is actuated. The 29-volt flight control bus provides a separate bus for the flight control electronics. The 29-volt nonessential bus satisfies all other requirements for 29-volt regulated power. This bus may be disabled automatically or by command in the event of an overload.

The 22-volt bus provides unregulated power to those circuits not requiring regulated power such as switches, solenoids, and actuators, and to electronic units providing their own regulation.

SOLAR PANEL

The solar panel assembly, installed at the top of the mast, consists of a series/parallel-connected array of 792 solar cell modules arranged on a planar honeycomb substrate approximately 9 square feet in area. The panel can be periodically adjusted, via commands, to compensate for the apparent motion of the sun and will remain perpendicular to the incident solar radiation within a few degrees. The solar panel is the prime source of power during transit and the lunar day. The panel is capable of providing a minimum of 81 watts.

BATTERY

A 14-cell series-connected, silver-zinc rechargeable battery, located in compartment A, provides energy storage for the spacecraft. The minimum capacity of the battery when fully charged is 3375 watt-hours at a discharge rate of 0.5 ampere (see Appendix B, item 26). The output voltage of the 14-cell

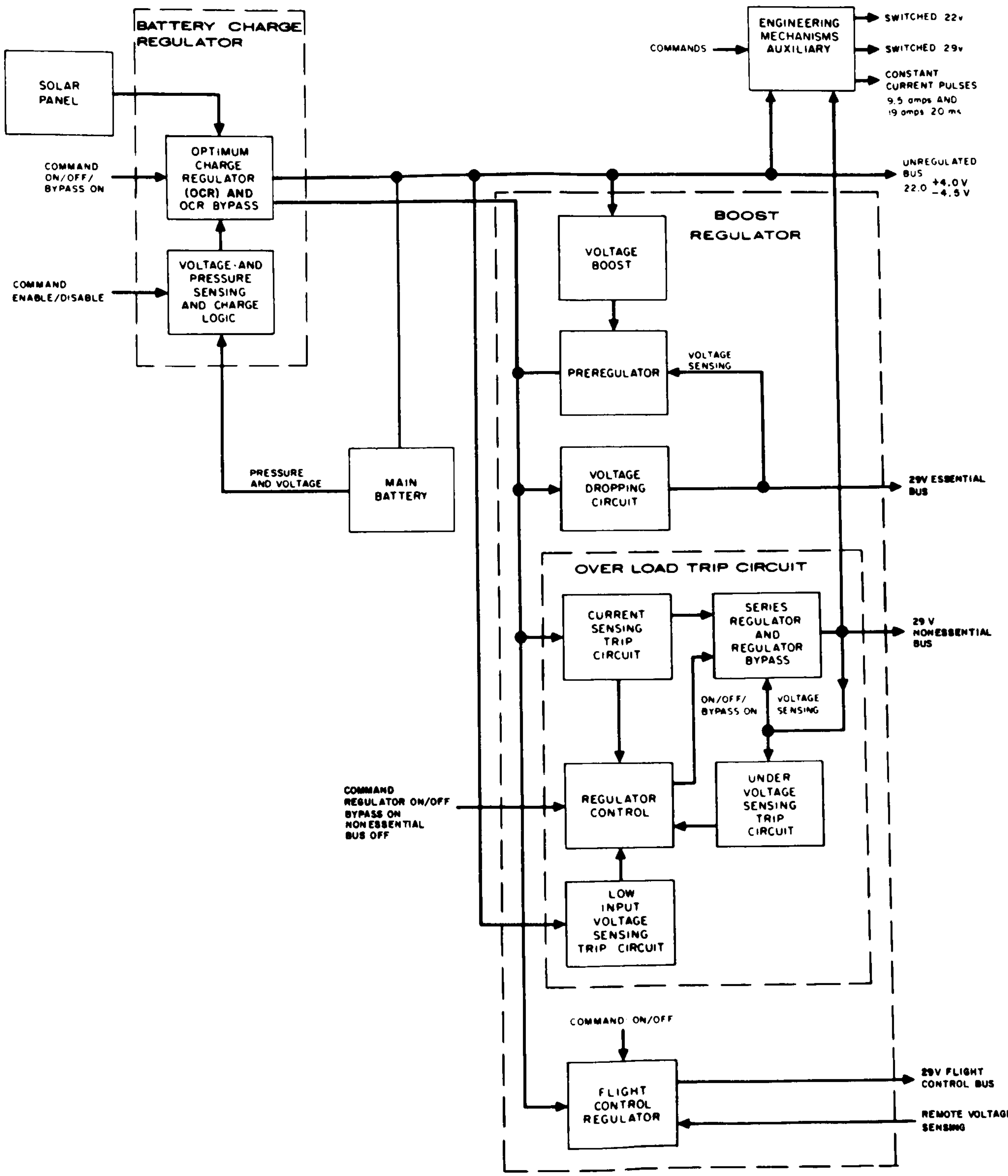


FIGURE 6-1. ELECTRICAL POWER SUBSYSTEM BLOCK DIAGRAM

battery as measured on the load side of the mating electrical connector of the battery receptacle will be 22 (+4.0, -4.5) volts for all operating and environmental conditions, including temperature from 40° to 125° F from no load to full load. The battery is intended primarily to handle peak loads and lunar night loads and is charged by the solar panel.

BATTERY CHARGE REGULATOR UNIT

The battery charge regulator unit, located in compartment A, is incorporated in the basic bus to provide charging logic and voltage conversion to enable the varying output voltage of the solar panel to be used to charge the battery. The battery charge regulator unit contains the optimum charge regulator, battery charge logic circuit, and power switching circuits for all A/SPP stepping motors. The function of the optimum charge regulator circuit is to couple the solar panel to the battery with maximum power transfer. The optimum charge regulator accepts power from the solar panel at varying voltages corresponding to maximum solar panel power output. It delivers this power to the battery at the battery terminal voltage. The operation of the optimum charge regulator can be controlled by commands from earth.

The battery charge logic circuit provides sensing, logic, and control of functions necessary to:

a. Provide automatic charging of the battery until the battery manifold pressure reaches 65 ±3 psi (provided the battery voltage is below 27 volts).

b. Automatically restore battery charging when the battery manifold pressure drops below 60 ±5 psi.

c. Accept all available power for charging until the battery reaches 27.3 volts. At this point, the battery is potential-limited and a tapered charge results from charging battery impedance.

d. Respond to earth commands which override all automatic battery charging functions.

e. Prevent damage to the charging logic or to any other component in the power control unit in the event of an open or shorted battery.

BOOST REGULATOR UNIT

The boost regulator unit accepts unregulated power and delivers regulated power. The boost regulator receives an input voltage of 17.0 to 27.3 volts dc and provides 29.0 ±1 percent volts to the regulated output. This voltage specification is met only at the terminals of the regulator. The voltage available at a unit connector is less than this value by the line drop through the connecting wiring plus an additional switch drop of up to 0.5 volt, when transistor switching is required. The maximum current output of the boost regulator is 7.0 amperes at the specified regulated voltage.

The overload trip circuit, located in the boost regulator unit, provides overload protection for the boost regulator and undervoltage protection to all nonessential regulated loads. When an overload causes the output voltage to drop to 27.75 ±0.25 volts, the voltage to the nonessential regulated loads will drop to zero and remain at 0 volts for 20 to 1000 milliseconds. This period allows the individual load switches in all equipment on at the time to turn off automatically. If an overload is still present after the overload trip circuit has recovered, the voltage will again drop to zero for 20 to 1000 ms, and so on in a cyclic manner. When the battery potential drops below approximately 17.00 ±0.25 volts, the overload trip circuit will remove all nonessential loads from the regulated output in this same manner. The operation of the overload trip circuit can be controlled by earth commands.

Definitive and descriptive documents for the electrical power subsystem are listed in Appendix A, items 18 thru 23.

VII. TELECOMMUNICATIONS SUBSYSTEM

INTRODUCTION

The telecommunications subsystem consists of three interconnected groups which provide command reception and decoding, and telemetry signal processing and transmission. A data link group provides r-f transmission and reception. A command decoding group provides decoding logic circuits for all earth commands. A signal processing group provides commutation, analog to digital conversion, and sub carrier modulation circuits for processing of most analog, digital and video data channels.

DATA LINK GROUP

The data link group comprises two transmitters, two receivers (figure 7-1), two omnidirectional antennas, and a high gain planar array antenna. A transponder mode is employed during the transit phase to permit two-way doppler shift measurements. The earth-to-spacecraft link is a PCM-FM-PM system. The spacecraft-to-earth link is operated PCM-FM-PM during transit (transponder mode) and PCM-FM-FM, PCM-FM-PM, or direct FM during lunar operation. The total information bandwidth of the system is dependent on the mode of system operation. There are four modes of operation that can be selected by command from earth. These modes, and their usable information bandwidths while operating at lunar distance, are as follows:

a. Mode A — High gain antenna with transmitter in high-power mode; nominal information bandwidth is 220 kcps.

b. Mode B — High gain antenna with transmitter in low-power mode; nominal information bandwidth is 2 kcps.

c. Mode C — Omnidirectional antenna with transmitter in high-power mode; nominal information bandwidth is 1000 cps.

d. Mode D — Omnidirectional antenna with transmitter in low-power mode; nominal information bandwidth is 10 cps.

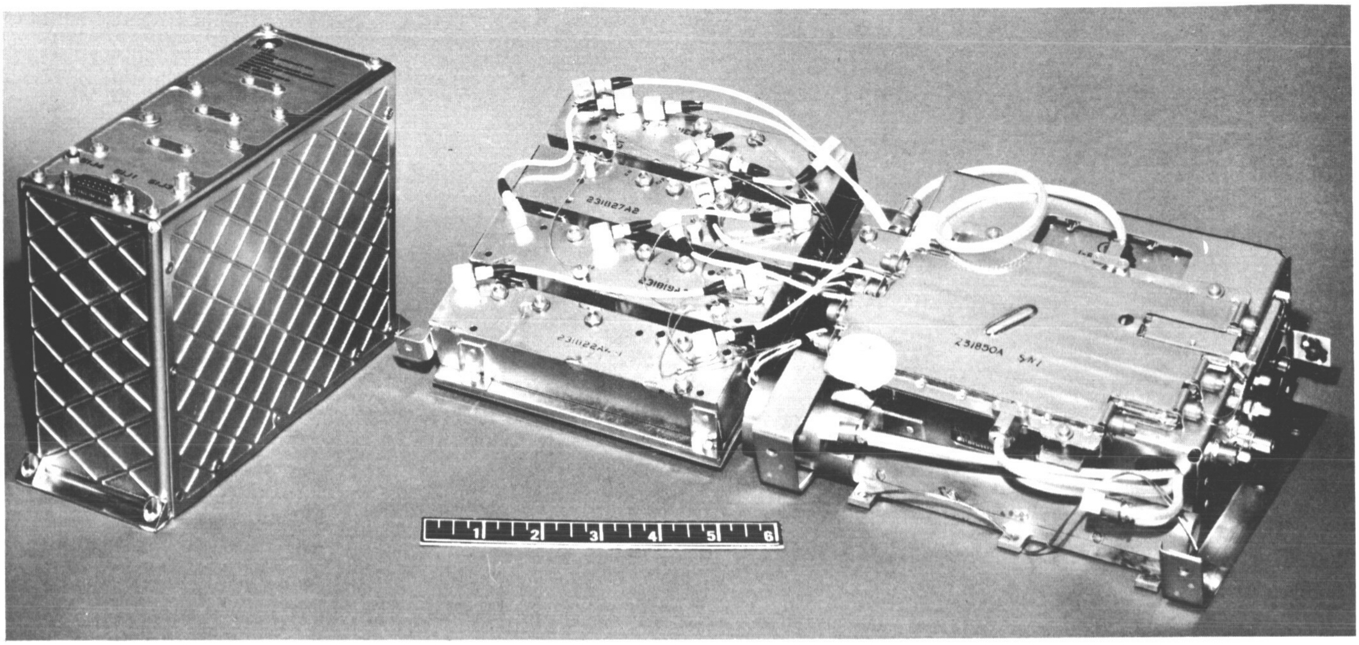


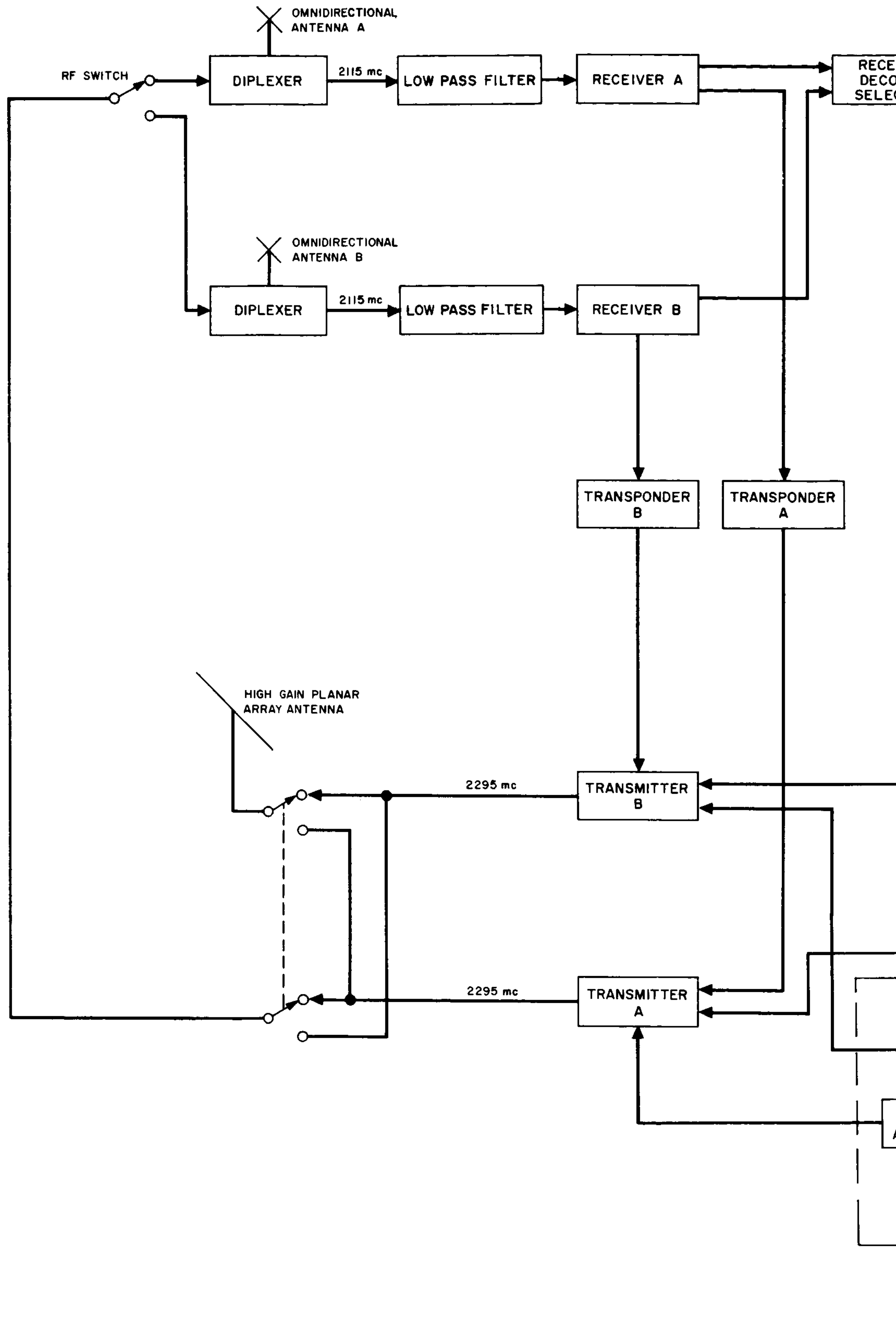
FIGURE 7-1. COMMAND RECEIVER AND TRANSMITTER

Figure 7-2 is a block diagram of the telecommunications subsystem Transmitter Receiver, and Transponder Interconnections.

Design redundancy is employed in dual transmitting and receiving systems (with the exception of omni antenna coverage redundancy) to ensure reception of commands from earth and ensure that the desired data will be transmitted back to earth.

Two identical transmitters, located in compartment A, are provided; each transmitter incorporates switching to provide either a high- or low-power output. This is accomplished by switching a traveling-wave tube (TWT) amplifier into, or out of, the circuit. Both transmitters incorporate provisions for frequency modulation and phase modulation. Each transmitter consumes about 70 watts in high-power mode and about 7 watts in the low-power mode. Transmitter performance parameters are as follows:

Nominal output frequency	=	2295 mc (See Appendix B, Item 13)
Nominal output power	=	40 dbm (10 watts) in high-power mode (See Appendix B, Item 14)



OMNIDIRECTIONAL ANTENNA A
RF SWITCH
DIPLEXER
2115 mc
LOW PASS FILTER
RECEIVER A
RECE
DECO
SELE
OMNIDIRECTIONAL ANTENNA B
DIPLEXER
2115 mc
LOW PASS FILTER
RECEIVER B
TRANSPONDER B
TRANSPONDER A
HIGH GAIN PLANAR ARRAY ANTENNA
2295 mc
TRANSMITTER B
2295 mc
TRANSMITTER A

51-7 A

70-940-188

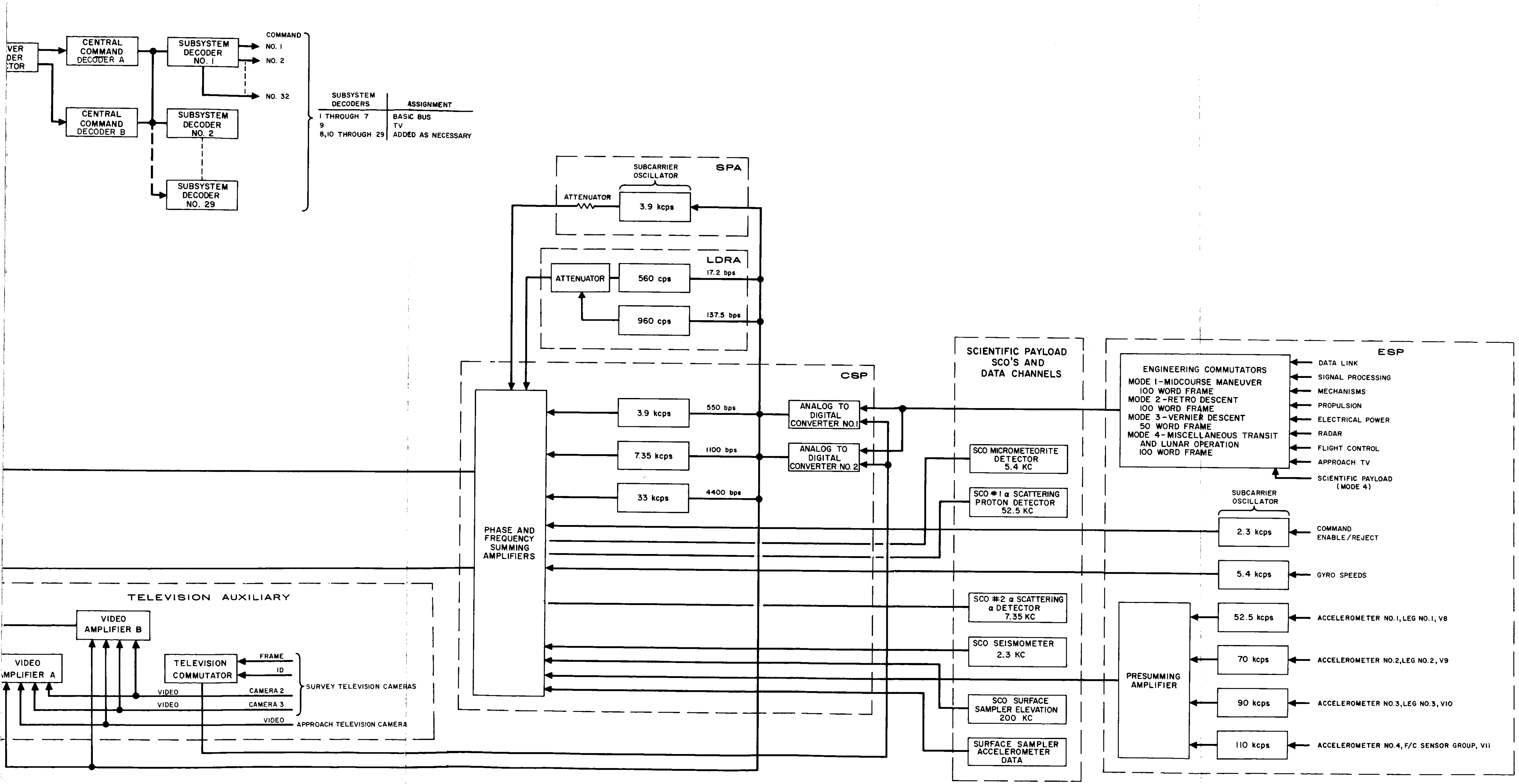


FIGURE 7-2. TELECOMMUNICATIONS SUBSYSTEM BLOCK DIAGRAM

51-2 B

Nominal output power	= 20 dbm (0. 1 watt) in low-power mode

Two identical receiver/transponders, located in compartment A, are a part of the command link. Their nominal signal input frequency is 2115 mc (see Appendix B, Item 18). These receivers are of the double-conversion type and are frequency modulated and crystal-controlled. During the transponder mode, when enabled by command, one of the receivers is phase-locked with the received signal and provides excitation for an accompanying transmitter. Either of the two receivers can perform this transponder function. Each receiver is permanently connected to one transmitter for transponder operation, thus providing two redundant transponders provided the omnis are intact. In this mode the received signal is retransmitted at a ratio of 240 to 221. Together the receivers require about 2. 82 watts continuous unregulated power for lunar operation.

Three telecommunications antennas are provided on the spacecraft. Two are omnidirectional antennas for command reception and transponder operation, and the third is a planar array antenna capable of radiating sufficient effective power for real-time television transmission. An antenna switching function is included to allow use of alternate antennas. Each of the two omnidirectional antennas is mounted on an extendable boom. One antenna is permanently connected to receiver A and the other to receiver B. The transmitters can be switched to use either the omnidirectional or the high-gain antenna. Each of the omnidirectional antennas consists of a turnstile half-wavelength dipole exciting a turnstile slotted cone, with the following nominal characteristics:

Gain	Greater than -10 db (composite pattern, both antennas, see Appendix B, Item 17)
Polarization	Right-hand circular
Impedance	50 ohms
Frequency	S-band

The high-gain antenna, installed with the solar panel on top of the mast, is a planar array with the following characteristics:

Gain	Approximately 27 db (see Appendix B, Item 16)
3-db beamwidth	8. 0 degrees E-plane, 6. 5 degrees H-plane
Polarization	Right-hand circular

Impedance	50 ohms
Frequency	S-band

Thermal control of the antennas is passive, using surface coatings to maintain temperatures within acceptable limits.

COMMAND DECODING GROUP

The command decoding group accepts earth-transmitted command messages from the spacecraft receivers, generates sync and timing signals from each command, checks each command for correct address and command complements, and provides output signals to command the addressed sybsystems. The system can process direct commands (which control on-off operations) and quantitative commands (which control time-interval operations). The spacecraft is mechanized to handle a total of 324 direct commands and quantitative commands with a resolution of one part in 1024.

The command decoding unit, located in compartment B, is the basic element of the command decoding group. The command decoding unit consists of one receiver-decoder selector, two identical central command decoders for redundancy, and five subsystem command decoders. Figure 7-3 illustrates the relationship of the receiver-decoder selector, central command decoders, and subsystem command decoders. The command link transmits information in the form of two types of standard-length serial binary digital words. The digital words are of the direct command type and the quantitative command type. One or more direct commands are always used in conjunction with a quantitative command to select the appropriate function in the flight control subsystem that requires a selectable time interval operation.

Fill-in words may be transmitted at all times when no commands are being transmitted, during periods when the command link requires frequent use. The fill-in word is a direct command which maintains word and bit sync, but does not select a subsystem decoder.

Complement checks are made on the address bits and their complements, and similarly, in the case of direct commands (and fill-in words), on command bits. Failure to "check" generates a message reject signal that is telemetered to earth. The magnitude bits of the quantitative commands are not complement checked; however, these are telemetered to earth for verification before execution.

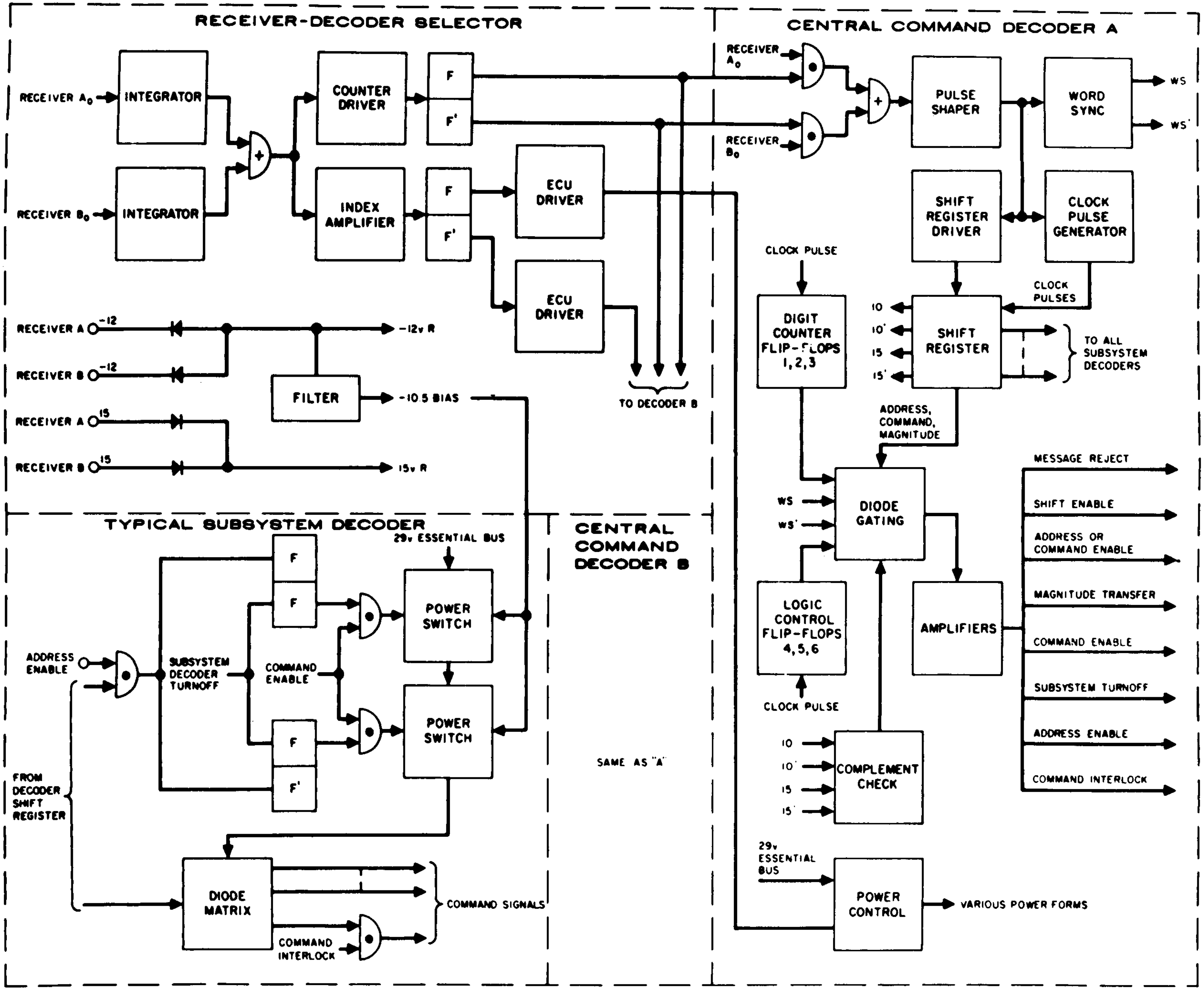


FIGURE 7-3. COMMAND DECODING BLOCK DIAGRAM

A receiver-decoder selector is provided to select one of the two spacecraft receivers and one of the two central command decoders for use in processing command messages transmitted from earth. Two flip-flops in the receiver-decoder selector form a four-state counter, each state corresponding to one of the four combinations of two receivers and two central command decoders. Both receivers will always be operating. One receiver select signal will permit the output of the corresponding receiver to be processed in the operating central command decoder. The other select signal will inhibit the corresponding receiver output at the input of the central command decoder. The nonselected central command decoder remains off. Whenever command modulation is interrupted for a period exceeding 500 milliseconds, the receiver-decoder selector automatically switches to another of the four possible combinations of receiver and decoder. Thus, if the receiver in use fails, a new receiver-decoder combination will be automatically selected. To avoid unintentional switching of the desired combination between commands, modulation must always be present; therefore, a fill-in word, which in essence is a dummy command, is required to be transmitted between normal command transmissi ons.

The central command decoder (figure 7-4), which synchronizes and controls the operation of the entire command decoding subsystem, provides the major processing of the command messages. Synchronization and timing information are derived from the incoming messages. Control signals are generated in a timed sequence and delivered to all subsystem command decoders to control their operation.

Three types of command words are provided - a direct command, a quantitative command, and an interlock command. Each direct command consists of an address, address complement, command, command complement and sync information. The quantitative command, which controls time durations of operations in flight control, also contains an address, an address complement, and sync information, but has no command complement. The interlock command, which must precede an irreversible or critical command, prevents the inadvertent execution of such commands. It also consists of an address, address complement, command, command complement, and sync information.

The central command decoder compares the address and command bits to their respective complements. Failure to check results in message rejection and initiation of a message reject signal transmitted via telemetry to earth. If

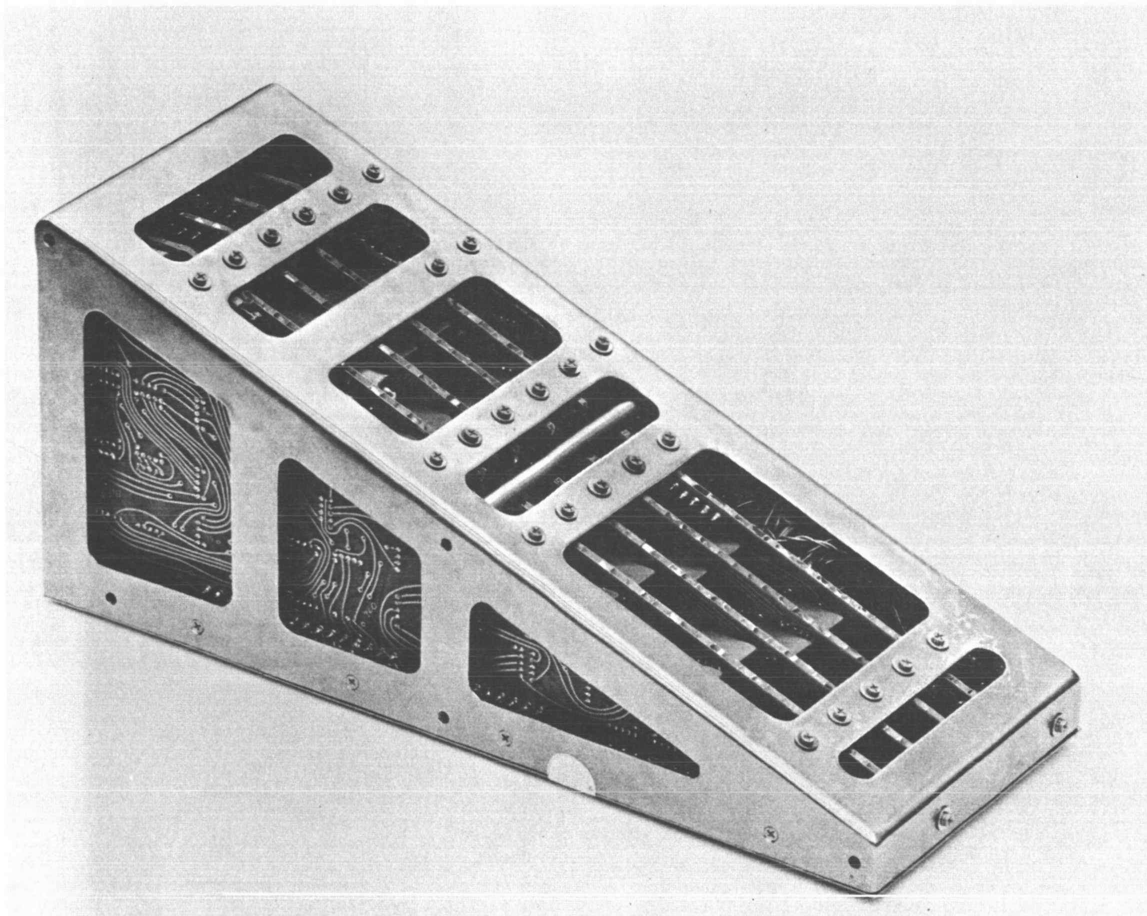


FIGURE 7-4. CENTRAL COMMAND DECODER

both address and command complement checks are successful, a command enable signal is generated and transmitted to earth.

The address and command information bits are delivered to all subsystem decoders where they are decoded under the control of the address enable signal and the command enable signal, both of which are generated in the central command decoder. The address that is assigned to all quantitative commands is decoded in the central command decoder.

The function of a subsystem command decoder is to supply actuating or command signals to chosen locations by deciphering the digital information supplied by the central command decoder under the control of the central command decoder. Subsystem decoders are available with three sizes of matrices so that 8 (3 modules), 20 (4 modules), or 32 (5 modules) possible command outputs are available. The maximum number of subsystem command decoders that may be addressed is 29.

At present, the basic bus uses 7 subsystem command decoders and the scientific payload uses 5.

The twelve are assigned as follows:

a. Data link and television approach camera (No. 4).

b. Signal processing.

c. Electrical power.

d. Vehicle and mechanisms.

e. Engineering mechanisms auxiliary.

f. Flight Control.

g. Television survey camera (No. 2).

h. Television survey camera (No. 3).

i. Soil mechanics surface sampler.

j. Alpha Particle Scattering

k. Micrometeorite detector.

l. Seismometer

Command interface signals from the central command decoder are fed simultaneously to a matrix and to an OR gate. Also, to the same OR gate of the selected subsystem decoder, the address enable signal is fed from the central command decoder. When all signals are present, a set pulse is fed out of the gate to a pair of parallel flip-flops (for redundancy). If the central command decoder address and command complement checks are both successful, a command enable signal will be generated. This signal turns on the power amplifier of the selected subsystem decoder, energizing its matrix. The states of the command interface signals at this time determine which command output of the matrix is allowed to go high.

SIGNAL PROCESSING GROUP

The signal processing group gathers the engineering and verification signals from various subsystems and provides the appropriate signal conditioning. The signal processing subsystem comprises the engineering signal processor, central signal processor, low data rate auxiliary, and signal processing auxiliary.

The engineering signal processor, located on compartment B, processes data from the Surveyor spacecraft and puts it in a suitable form preparatory to being transmitted to earth. This processor handles all data required to assess the performance of the basic bus (engineering data).

The engineering signal processor is composed of the following major components: four commutators, two current sources (thermal measurements), one command enable and reject channel; and four accelerometer channels.

The engineering signal processor contains a commutator capable of four modes of operation. Commutator modes 1, 2, and 4 consist of 98 words of data plus two sync words (a total of 100 words). Commutator 3 consists of 48 words of data plus 2 sync words (a total of 50 words). Each word consists of 11 digital bits, 10 bits of data plus one parity bit. Table 7-1 contains a sample frame format with the first four words of the frame illustrated. The first two positions on the frame (sync complement and sync) are also explained. To illustrate the entire 100 words of the message frame is beyond the scope of this document.

The sampling formats of commutator modes 1, 2. and 3 have been established primarily by telemetry requirements for the following critical periods during transit or terminal descent: midcourse maneuver and correction, commutator mode 1; terminal descent using high gain antenna, commutator mode 2; and terminal descent using omnidirectional antennas, commutator mode 3. The sampling format of commutator mode 4 has been established by all other telemetry requirements excluding those previously listed.

Each of these commutators can be operated at any time by command (but only one at a time) at any one of five bit rates: 17.2, 137.5, 550, 1100, or 4400 bits per second. The bit rate is controlled by the operating analog-to-digital converter located in the central signal processor. Selection of the bit rate is limited only by bandwidth available at a given time. The frame rate of each commutator for each bit rate is listed in table 7-2.

TABLE 7-1. PORTION OF ESP COMMUTATOR DATA FRAME

Word No.	Mode No.	Signal Commutated
00	1	Sync Complement
	2	Sync Complement
	3	Sync Complement
	4	Sync Complement
0	1	Sync
	2	Sync
	3	Sync
	4	Sync
1	1	Primary Sun Sensor Pitch Error (also occurs on mode 1, words 21, 41, 61 and 81)
	2	Doppler Velocity V_x (also occurs on mode 2 of word 51)
	3	Doppler Velocity V_x (also occurs on mode 3 of words 21, 31 and 41)
	4	Omni No. 1 Transmitted A Power
2	1	Vernier lines No. 2 Temp
	2	Doppler Velocity V_y (also occurs on mode 2 of word 52)
	3	Doppler Velocity V_y (also occurs on mode 3 of words 22, 32 and 42)
	4	Transmitter A Temperature
3	1	Primary Sun Sensor Yaw Error (also occurs on mode 1 of words 23, 43, 63 and 83)
	2	Doppler Velocity V_z (also occurs on mode 2 of word 53)

TABLE 7-1. PORTION OF ESP COMMUTATOR DATA FRAME (Cont)

<table>
<tr><th>Word No.</th><th>Mode No.</th><th>Signal Commutated</th></tr>
<tr><td>3
(Cont)</td><td>3</td><td>Doppler Velocity V_z (also occurs on mode 3 of words 23, 33 and 43)</td></tr>
<tr><td></td><td>4</td><td>Static Phase Error A (also occurs on mode 4 of words 23, 43, 63 and 83)</td></tr>
<tr><td>4</td><td>1
2
3</td><td>Rate Mode Z01, Sun Mode Z2, Star Mode Z3, Cruise Mode, Inertia Switch, Pitch Precession Enable Z19, Signal B05, Manual Lockon Z26, Nominal Thrust Bias Z18, Altimeter Reliable RORA, and Doppler Reliable RADVS</td></tr>
<tr><td></td><td>4</td><td>Main Battery Temp (also occurs on mode 4 of word 42)</td></tr>
<tr><td colspan="3">Sync complement is 0 0 0 1 1 1 0 1 1 0 1 (this is the barker code complement that is read out and sent to earth on the next commutator word preceding sync to further establish commutator frame synchronization)

Sync is 1 1 1 0 0 0 1 0 0 1 0 (this is a calculated series of digits least likely to occur at random, called the barker code, and indicates the start of each commutator frame. This defines the states of the eleven digit times, T301 thru T311, and is read out and sent to earth to indicate the start of a commutator frame.</td></tr>
</table>

There are five different types of inputs available in the engineering commutators; two analog voltage types, two temperature measurement types, and one digital type. There are 89 high-level inputs available for processing analog signals which vary from 0 to +5 volts. A number of these inputs provide sampling on more than one commutator and on more than one word in a given commutator. The basic accuracy of these high-level commutator inputs will be approximately ±0.2 percent of full scale. This tolerance, however, does not include errors that exist in the signal sources, such as, transducers, voltage

TABLE 7-2. TIME REQUIRED FOR ONE FRAME OF COMMUTATED DATA

Sampling Rate		Time Required (Seconds)			
Bits per Second	Words per Second	Commutator 1 (100 words per frame)	Commutator 2 (100 words per frame)	Commutator 3 (50 words per frame)	Commutator 4 (100 words per frame)
17.2	1.5	64	64	32	64
137.5	12.5	8	8	4	8
550	50	2	2	1	2
1100	100	1	1	0.5	1
4400	400	0.25	0.25	0.125	0.25

dividers, etc. There are 11 low-level differential inputs available for processing analog signals varying from 0 to +100 millivolts. These inputs are applied to a differential amplifier with a nominal gain of 50. The amplifier output is then processed as an ordinary 0 to +5-volt high-level signal. The absolute accuracy of the data processed on these inputs is ±2 percent of full-scale without the use of the calibration data. The absolute accuracy is ±1 percent when using a calibration curve defined by three calibration points generated within the engineering signal processor. Most of the data processed by these inputs will consist of current measurements of the power control system. The resistance value of each current shunt is such that the expected voltage drop never exceeds 100 millivolts.

There are 64 inputs available for processing basic temperature measurements. These measurements are obtained by supplying a constant current of 5 milliamperes to temperature sensors which vary in impedance as a function of temperature but do not exceed 1000 ohms. The expected accuracy of these measurements is ±4° C, depending on the calibration accuracy of the particular temperature sensors.

There are seven inputs available for processing high-accuracy temperature measurements. These measurements are obtained by supplying a constant current of 2.5 milliamperes to temperature sensors which vary in impedance as a function of temperature but do not exceed 2000 ohms. The expected accuracy of these measurements is approximately ±1.0° C.

The digital inputs handle certain quantities to be transmitted that consist only of an indication of which of two states prevails regarding a mechanical device or electrical quantity. Such a signal is defined as being either on or off. If it is on, it is defined as having a current of not greater than 3 microamperes, or being a voltage between +5 and +10 volts (or an open circuit). If a signal is off, it is between -3 and +1 volts and is capable of accepting 0.2 milliampere. There are a total of four digital words (40 inputs) available in the engineering signal processor.

Reject and enable signals from the central command decoder modulate a 2.3-kc subcarrier oscillator in the engineering signal processor. This subcarrier oscillator will be commanded on when needed and when bandwidth is available. The subcarrier oscillator will remain at its 2.3-kc frequency when no enable or reject signal is present. When the reject signal is present, it will deviate the subcarrier oscillator to a higher frequency for a period equal to the period of the enable signal (approximately 21 milliseconds). When the enable signal is present, it will deviate the subcarrier oscillator to a lower frequency for a period equal to the period of the reject signal (approximately 21 milliseconds).

The central signal processor (figure 7-5), located in compartment B, combines the outputs from the engineering signal processor and various basic bus subsystems. These signals are processed and sent to the frequency- or phase-modulation inputs of the transmitter. The subsystem has the capacity for handling analog data and, where desirable, converting it to digital data for subsequent transmission. Synchronizing patterns, parity digits, and timing signals for controlling commutators are generated in the central signal processor. The central signal processor contains two analog-to-digital converters, three subcarrier oscillators, six summing amplifiers, power switches, and an analog-to-digital isolation amplifier.

The analog-to-digital converters accept d-c analog signals and convert these signals into 10-digit binary numbers. The converter functions by making successive comparisons of the analog voltage with increments of a reference voltage furnished by a binary voltage weighter network. With each comparison, a logical

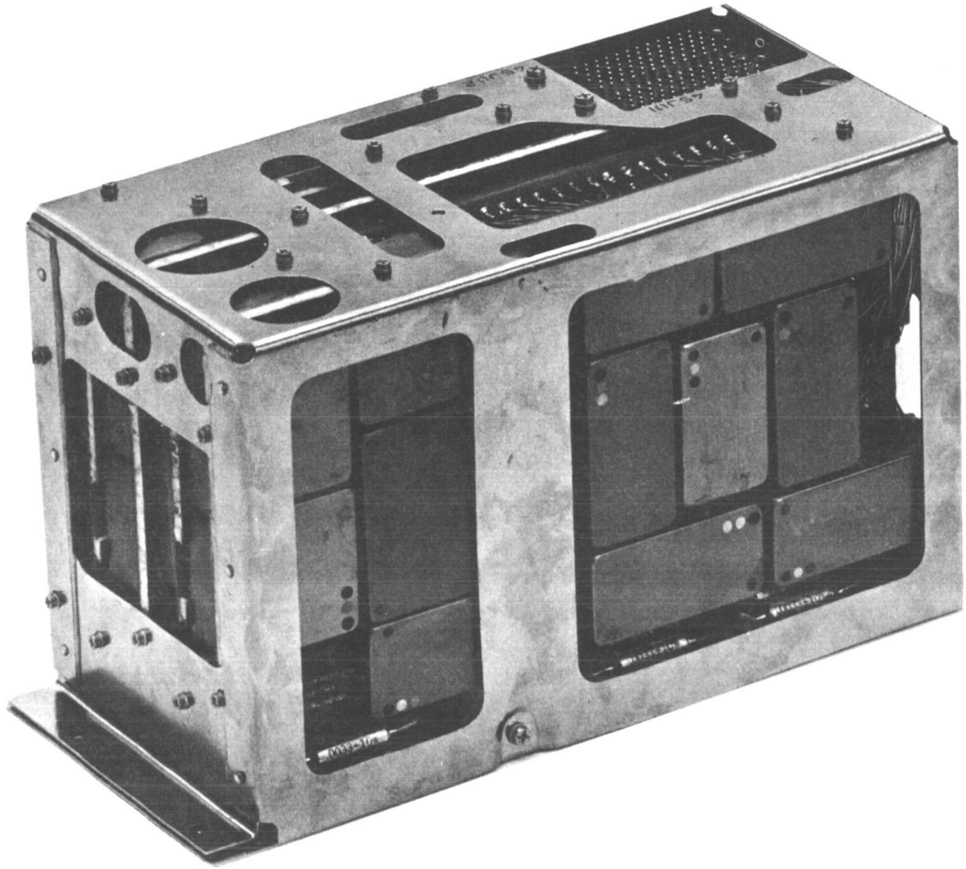


FIGURE 7-5. CENTRAL SIGNAL PROCESSOR

decision is made, and the increment of reference voltage is allowed to remain or is subtracted out. This process continues until the error in the binary representation is less than one part in 2^{10} at full scale. The output of the comparator will be the serial binary representation of the voltage being digitized. These signals are sent to the analog-to-digital timing circuitry to be combined with a parity bit, and sync words in a readout amplifier.

The signal processing subsystem contains two analog-to-digital converters for redundancy and output circuitry common to both. Either analog-to-digital converter will operate when it receives power on signals from the subsystem decoders and bias voltages from the ESP commutator ECU. The power on signals from the subsystem decoders, available upon command from earth will turn on electronic conversion units in the selected A/D converter. These supply voltages are applied to master switches which enable commutated analog signals to one converter and inhibit signal inputs to the converter that is off.

Subcarrier oscillators are modulated by the outputs of the analog-to-digital converters. Five subcarrier oscillator frequencies are provided, each associated with a different rate of data transmission. Center frequencies of these subcarrier oscillators are shown in figure 7-2.

Output signals from the analog-to-digital converter subcarrier oscillators are combined in the summing amplifiers and are sent to either of the two transmitters. The transmitters may then be either phase modulated or frequency modulated. This system uses individual "final" summing amplifiers to provide signals for the four different modes: (1) frequency modulate transmitter A, (2) frequency modulate transmitter B, (3) phase modulate transmitter A, and (4) phase modulate transmitter B. Two additional summing amplifiers are provided to "pre-sum" outputs of various other sources. This presumming reduces the number of inputs to the final summing amplifiers and thereby reduces the circuit complexity required for amplification. The gain of each individual channel through the summing circuitry is such that the desired transmitter frequency deviation is produced. A series gate which provides isolation is located at the output of the analog-to-digital converters. This gate provides a path to the Centaur data link subsystem for transmission of engineering data before separation.

The low data rate auxiliary, located in compartment B, provides for data transmission at lower bit rates than the lowest clock rates of the analog-to-digital converters. Data rates of 17 3/16 and 137 1/2 bits per second are obtained by dividing the 550-bit-per-second clock pulse from the central signal processor. These low bit rates are available to all commutators upon command. Two subcarrier oscillators, at 560 and 960 cps, are provided in the low data rate auxiliary for use with the low data rates.

The low data rates are used primarily with the low-power transmitter and onmi directional antenna, when the information bandwidth must be limited to maintain adequate operating margins.

The signal processing auxiliary located in compartment A, provides a 3.9-kc subcarrier oscillator which modulates the r-f carrier at a phase modulation index of 0.3 radian peak. By increasing the ratio of carrier to sideband power the low modulation index enhances the probability of carrier acquisition by the DSIF, while at the same time permitting the reception of engineering data. This mode of operation is commanded off when a higher modulation index is desired, at which time the central signal processor can provide a similar subcarrier oscillator at a higher modulation index. Power for the signal processing auxiliary is obtained from the 29-volt nonessential bus.

Definitive and descriptive documents for the telecommunications subsystem are listed in Appendix A items 24 thru 34.

VIII. FLIGHT CONTROL SUBSYSTEM

The flight control subsystem controls the spacecraft velocity and attitude during the transit phase of the Surveyor mission. This phase covers the period from separation of the spacecraft from the Centaur vehicle to spacecraft touchdown on the lunar surface. The basic functions performed by the flight control subsystem include: (1) attitude stabilization and orientation during the entire transit phase, (2) midcourse trajectory correction based on radio command data, and (3) terminal phase retro maneuver and vernier descent for landing of the spacecraft in an upright position on the lunar surface. Three principal forms of reference — celestial reference, inertial reference, and descent radar control — are used. Figure 8-1 is a block diagram of the flight control subsystem.

FLIGHT CONTROL SENSOR GROUP

The flight control sensor group is made up of a group of optical and inertial sensors and the flight control electronics. The assembly is mounted on the spaceframe between legs 1 and 3 near the hinge point of leg 3.

The inertial reference unit (IRU) provides a three axis rotational reference and an acceleration reference along the spacecraft roll axis. The unit consists of three orthogonally mounted, strapped down integrating rate gyros with associated temperature control circuitry and a linear force balance accelerometer. Each IRG comprises a gyro with a single degree of freedom, which integrates spacecraft rotation rate to obtain rotation angle. (See Appendix B, item 9.) Spacecraft rotation with respect to inertial reference is obtained by "torquing" a gyro at a precise rate for a precise period of time. The flight control electronics produce control pulses causing the gas jet attitude control system to rotate the spacecraft to track gyro precession. The accelerometer measures spacecraft acceleration during midcourse velocity vector correction and lunar descent phases and transmits the information to the flight control electronics. Three-phase 26-volt 400-cycle power operates the gyro motors. The gyro signal generators,

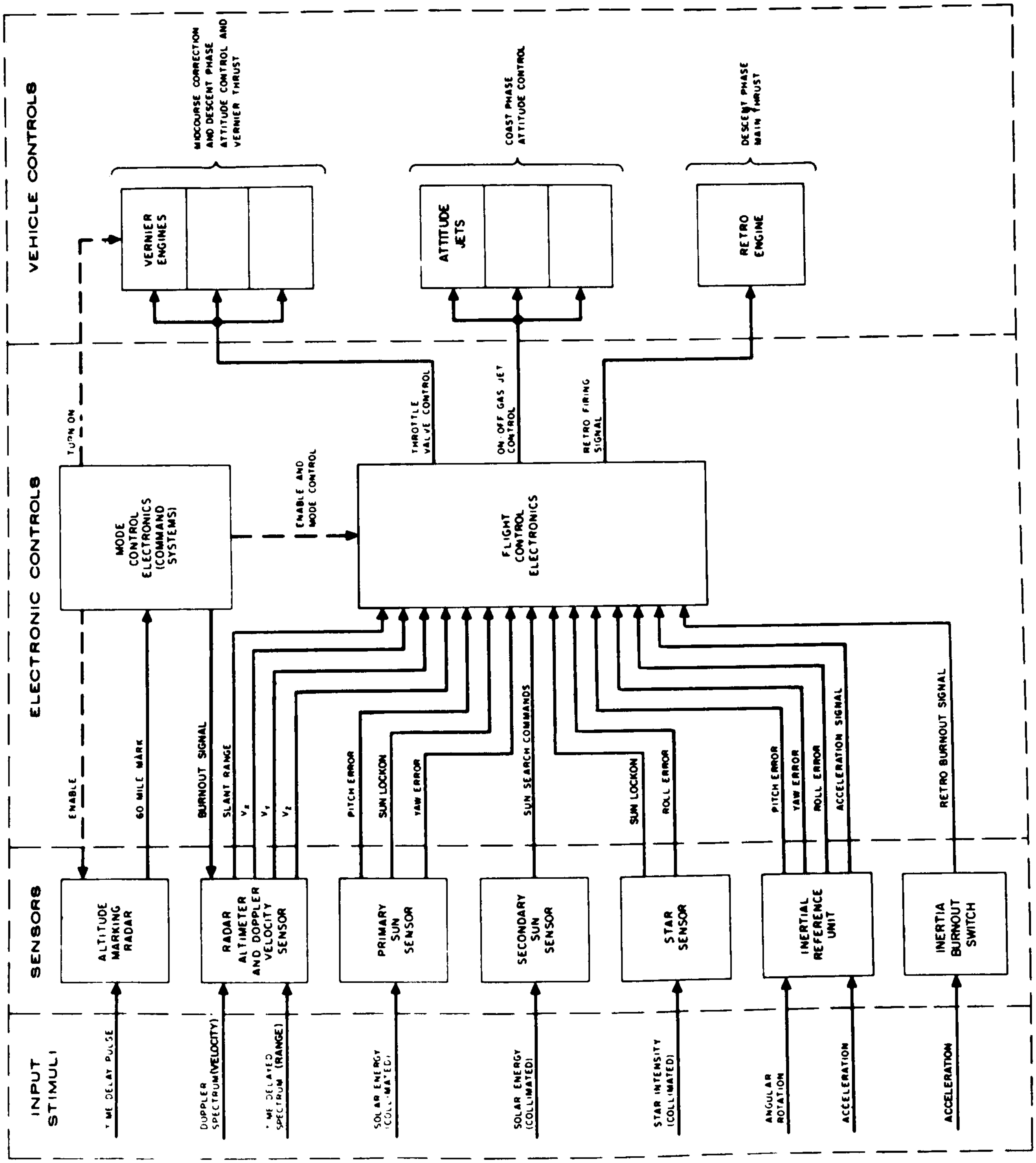


FIGURE 8-1. FLIGHT CONTROL BLOCK DIAGRAM

temperature controllers, and accelerometer utilize single-phase 10-volt 400-cycle, 22-volt dc, and 29-volt dc power, respectively. Gyro motor speed signals are telemetered to earth.

Temperature of the three gyros must be maintained within a closely restricted range in the vicinity of 180° F for all spacecraft flight attitudes. Internal gyro temperature sensing and control circuitry and heaters are employed to maintain the specified temperatures. To reduce the effect of all external perturbations, the inertial reference unit is largely thermally decoupled from the rest of the spacecraft. However, to dissipate internally generated heat, the gyros and associated circuitry are coupled to the IRU radiator. The radiator has the capability of radiating the total internal electrical heat dissipation of the IRU unit, excluding the gyro heaters, as well as direct and reflected energy from the sun and spacecraft to maintain a thermal balance within the unit.

The primary sun sensor detects deviation of the spacecraft roll axis from the sun-spacecraft line during coast phases. The sun sensor also supplies a signal to open the Canopus sensor shutter when the sun falls within a defined sun sensor field of view. The primary sun sensor consists of one lockon and four directional cadmium-sulphide photoconductive cells. The lockon cell provides a signal for transfer of control of the spacecraft attitude from the secondary sun sensor to the primary sun sensor. The directional cells supply signals enabling the flight control subsystem to control the spacecraft roll axis to within less than 0.2 ±0.3 degree limit cycle deadband, of the spacecraft-sun line.

Thermal control of this unit results from the combined effect of (a) conduction between the case and the support base and (b) the outer case surface treatment, which minimizes temperature gradients between the support base and the primary sun sensor case. The surface treatment of the inner cavities surrounding the photoconductive cells is designed to optimize heat flow between the housing and the cells.

The inertia switch, a spring-restrained mass that operates a single pole single throw switch, closes at a nominal 3.5 g as retro-rocket thrust decays. This g level corresponds to a thrust level on a constant decay curve. Therefore, by timing a command-to-initiate retro-eject signal (initiated at the instant the inertia switch opens) a safe prediction of the end of retro-burning can be made.

The telemetering accelerometer is an engineering instrumentation sensor that measures acceleration during descent. The accelerometer is a spring restrained, seismic mass that drives the pickoff arm of a linear potentiometer connected across 29 volts dc.

The Canopus sensor detects, identifies, and locks on to the star Canopus. This function establishes a fixed spacecraft roll attitude relative to the Canopus line of sight. In combination with the primary sun sensor, the Canopus sensor establishes the required celestial three-axis reference. The Canopus sensor uses a photomultiplier tube (See Appendix B, item 11) with suitable elements to (1) establish the presence, in the 8-degree nominal field of view, of a star of Canopus brightness; (2) provide error signals related to the angle of the star on the roll field of view; and (3) measure and telemeter star intensity as an aid to star map-making for positive Canopus identification. The Canopus sensor contains internal electrical and mechanical control systems operating from 3-phase, 26-volt, 400-cycle power and 29-volt dc power. The unit is designed to supply information enabling the flight control subsystem to hold Canopus alignment to 0.2 ±0.3 degree including limit cycle deadband. For thermal control, the external surfaces of the sensor case are finished so as to have the capability of continuously radiating the internal electrical dissipation as well as the direct and reflected solar energy.

The flight control electronics consists of control circuits, the programmer-decoder, and the ac/dc electrical conversion unit circuits. The location of this on the spacecraft is such that the primary radiating surfaces are not subjected to direct solar irradiation in a normal transit attitude. The external surfaces of the case are finished to obtain capability of continuously radiating any reflected solar energy, as well as the power dissipated during coast phase, without causing perturbations to the thermal control of other items within the spacecraft. With no additional thermal control, the unit is capable of radiating the thrust phase level of dissipation for a maximum period of 10 minutes.

The flight control circuits accept guidance signals and process them for control of propulsion systems to achieve the desired stability or controlled maneuver. The control circuits are installed on five circuit boards located in the flight control electronics unit. The circuits include (1) control mode switching circuits for control mode selection; (2) logic circuits for input signal processing;

and (3) analog circuits for converting sensor outputs to commands for the propulsion system.

The flight control programmer is a digital unit that provides yes/no output signals for initiating and controlling sequences within the flight control electronics and power management units of the spacecraft. Basically, the programmer generates the signals for attitude control, sun lockon, and star lockon according to certain command sequences; provides a timed sequence of output signals during the main retro-staging sequence; and generates precision radio-command variable time delays for controlling attitude maneuvers and midcourse velocity correction.

Most of the inputs to the flight control programmer are transmitted commands or magnitude information from the flight control programmer subsystem decoder or the central decoder unit, respectively. Other inputs, classified as special inputs, are from the altitude marking radar, the legs-down switches, the spacecraft separation sensors, and the inertia switch. These input signals, when they arrive in certain predetermined sequences, enable the advanced flight control programmer logic to generate the specified outputs for controlling the flight operation of retro staging sequence.

Seven output latch amplifiers serve a dual purpose of providing mode memory capability and interface buffering between the programmer and flight control electronics. Operations involving timed delays are performed by a 10-bit magnitude shift register/scalar counter which provides two types of timing: auto external time delays and auto internal time delays. Electrical power is taken from the 29-volt dc flight control bus.

The power supply electrical conversion units develop and control the d-c and a-c voltages required for operation of the flight control programmer-decoder control circuits, inertial reference unit, Canopus sensor, and primary and secondary sun sensors. The electrical conversion units are installed in the flight control electronics, and obtain input power from the 29-volt dc and 22-volt dc buses.

SECONDARY SUN SENSOR

The secondary sun sensor effects the initial sun detection and supplies signals to enable gross alignment of the spacecraft roll axis to the sunline during transit and positioning of the solar panel during lunar operation. The secondary sun sensor, an assembly of five cadmium-sulphide photo-conductive cells, is mounted on the solar panel. Four of the cells are directional cells and one is a lockon cell. The axis of the secondary sun sensor is aligned with the principal axis of the solar panel cells. The solar panel is erected to the transit position and its axis aligned to the spacecraft roll axis after separation from the Centaur vehicle. Since the four directional cells of the secondary sun sensor view a complete hemisphere, with each cell viewing one quadrant, a yaw (pitch) maneuver in any direction will bring the sun into view of one or more cells. Signals from the secondary sun sensor produce successive yaw and pitch maneuvers to orient the spacecraft roll axis to within the field of view of the primary sun sensor. Control is transferred to the primary sun sensor when the sun is within the field of view of the primary sun sensor lockon cell. The secondary sun sensor operates from approximately 5 volts dc power, which is taken from the flight control electrical conversion unit during transit and from the signal processing electronics conversion unit during lunar operation. Protective diodes are provided for operating from either source in case of "other source" failure. The output levels of all five cells are telemetered to earth.

Thermal control of this unit is obtained from the combined effects of conduction between the sensor case and the support structure; surface treatment of the case is expected to maintain the unit temperature between -100 and +160°F during transit and -125 and +235°F during the lunar day.

RADARS

The altitude-marking radar generates an "altitude mark" signal at a preset slant range from the lunar surface to initiate the terminal descent phase of the spacecraft flight. The altitude-marking radar is a single-package pulse-type fixed-range measuring radar with a single output signal that can be preset for slant ranges of 52 to 60 miles (See Appendix B, item 20). The altitude-marking radar mounts in the retro-rocket nozzle and is retained by friction clasps around the nozzle flange, with spring washers between altitude-marking radar and the flange. When retro-rocket ignition begins, the gas generated by the ignitor develops sufficient pressure to eject the altitude-marking radar from the nozzle. The altitude-marking radar is powered from the 22 volts dc bus through a break-away plug that also carries input commands, the output altitude-mark signal, and telemetry information. The altitude-marking radar operates at a frequency of 9.3 kmc. Figure 8-2 is a block diagram of the altitude-marking radar.

The thermal control designed for this unit makes it operable only to that point of the mission in which it generates the 60-mile mark trigger signal. Use of thermal isolation of the altitude-marking radar electronics, combined with active heating, ensures transit survival and attainment of the required operating temperature. During the transit phase of the mission, all altitude marking radar components are maintained at approximately 20°F by means of thermostatically controlled heaters up to the time of required operation.

The radar altimeter and doppler velocity sensor (RADVS) measures slant range and orthogonal three-axis velocity of the spacecraft with respect to the lunar surface during the descent phase from retro-rocket burnout to near touch-down. the RADVS utilizes common circuitry and components to perform two separate functions, altitude determination and three-axis velocity determination. It comprises four assemblies: (1) an r-f section (KPSM) mounted on the omni antenna 1 structure near leg 1 hinge point; (2) the altimeter/velocity antenna, beams 1 and 4, mounted under compartment A between legs 1 and 2; (3) the velocity sensing antenna, beams 2 and 3, mounted under compartment B between legs 1 and 3; and (4) the signal data converter mounted on the spaceframe, above leg 3 hinge, between legs 1 and 3. The antenna assemblies also contain the

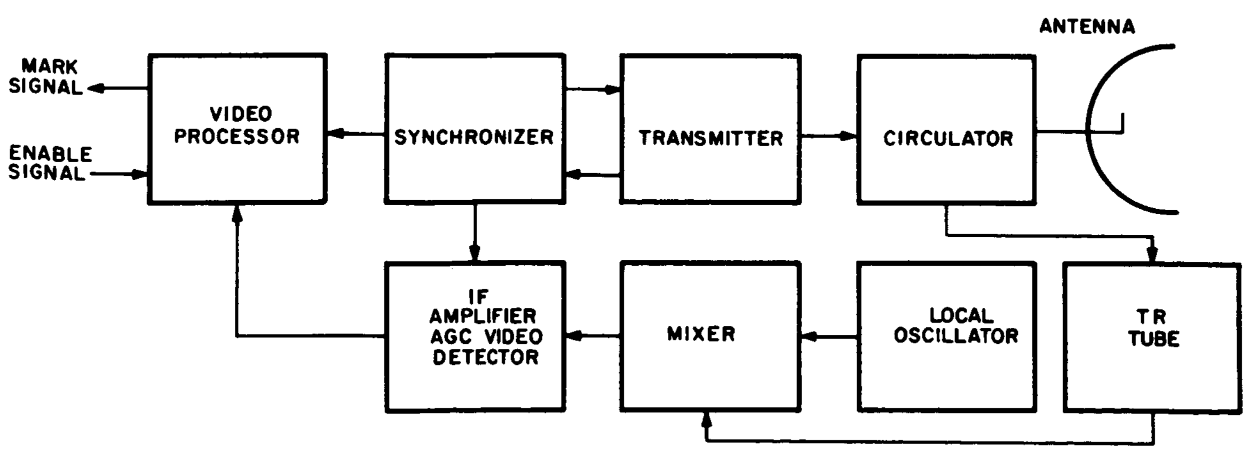


FIGURE 8-2. ALTITUDE MARKING RADAR BLOCK DIAGRAM

receiver microwave components, crystal mixer, and preamplifier for each beam, as illustrated in figure 8-3. The r-f section contains the two klystrons and the altimeter klystron modulator. The altimeter klystron operates at a mean frequency of 12.9 kmc whereas the velocity klystron operates at 13.3 kmc. A waveguide connects the antenna assemblies to the r-f section. Figure 8-4 illustrates the orientation of the four radar beams with respect to the spacecraft coordinate system. Figure 8-5 is a functional block diagram of the RADVS.

Radar beam 4 is parallel to the spacecraft Z axis and develops altitude or range information. Beams 1, 2, and 3 develop the three-axis velocity information with respect to spacecraft coordinates. Beams 1, 2, and 3 are oriented about 25 degrees divergent to the spacecraft +Z axis and pass through three corners of a square parallel to the X-Y plane. Beams 1 and 2 form a plane parallel to the X axis. Beams 2 and 3 form a plane parallel to the Y axis. Slant range along the Z axis is determined in beam 4, whose transmitted r-f is swept in frequency.

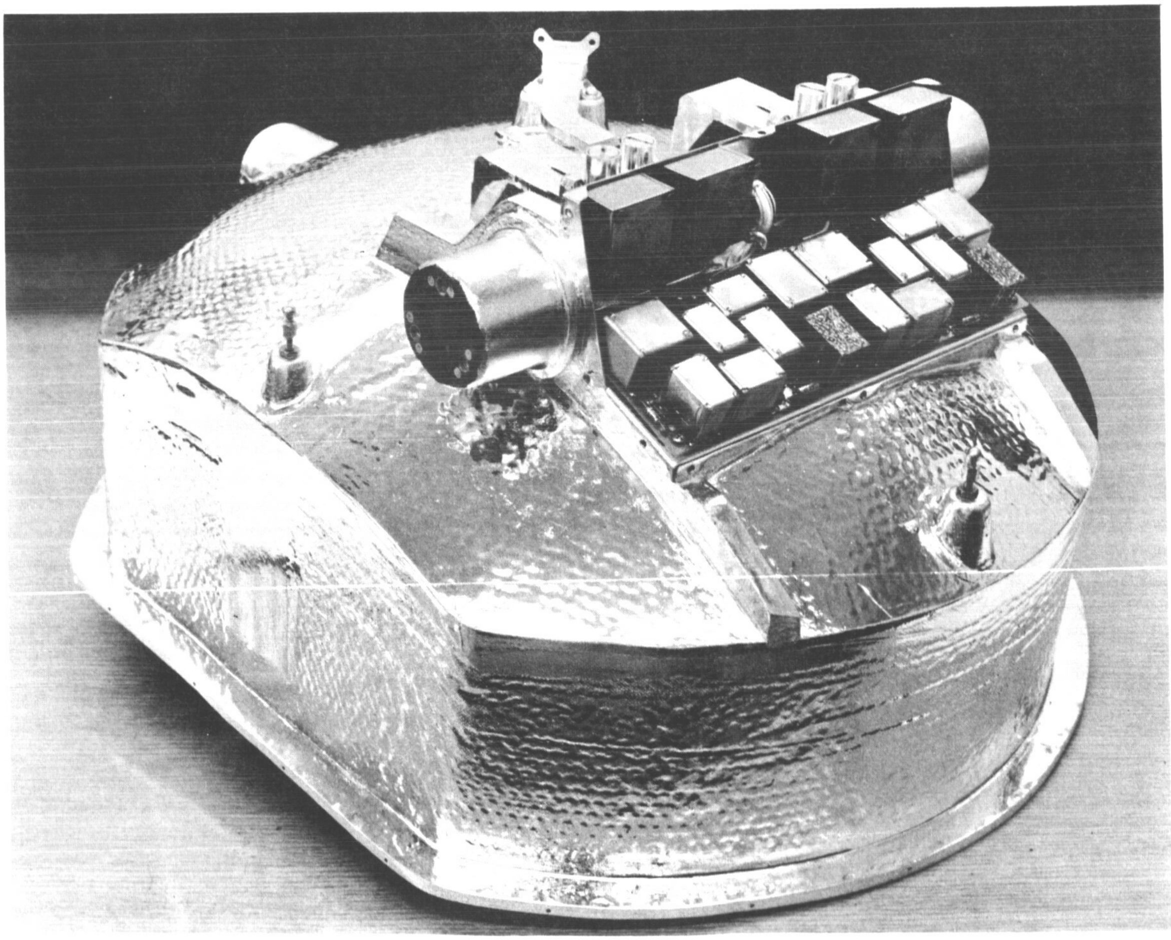


FIGURE 8-3. RADVS ASSEMBLY WITH PREAMPLIFIER COVER REMOVED

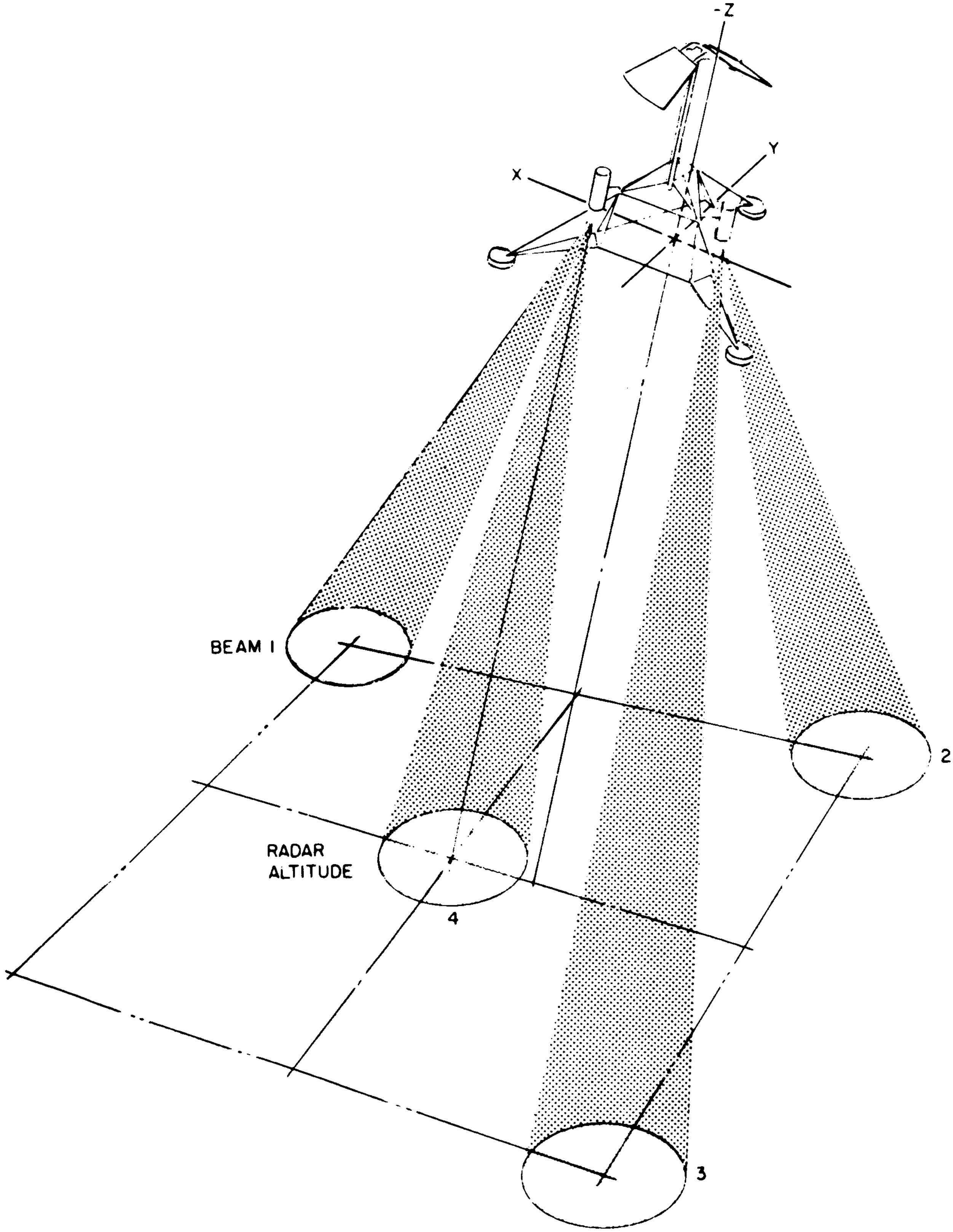


FIGURE 8-4. RADVS BEAM ORIENTATION

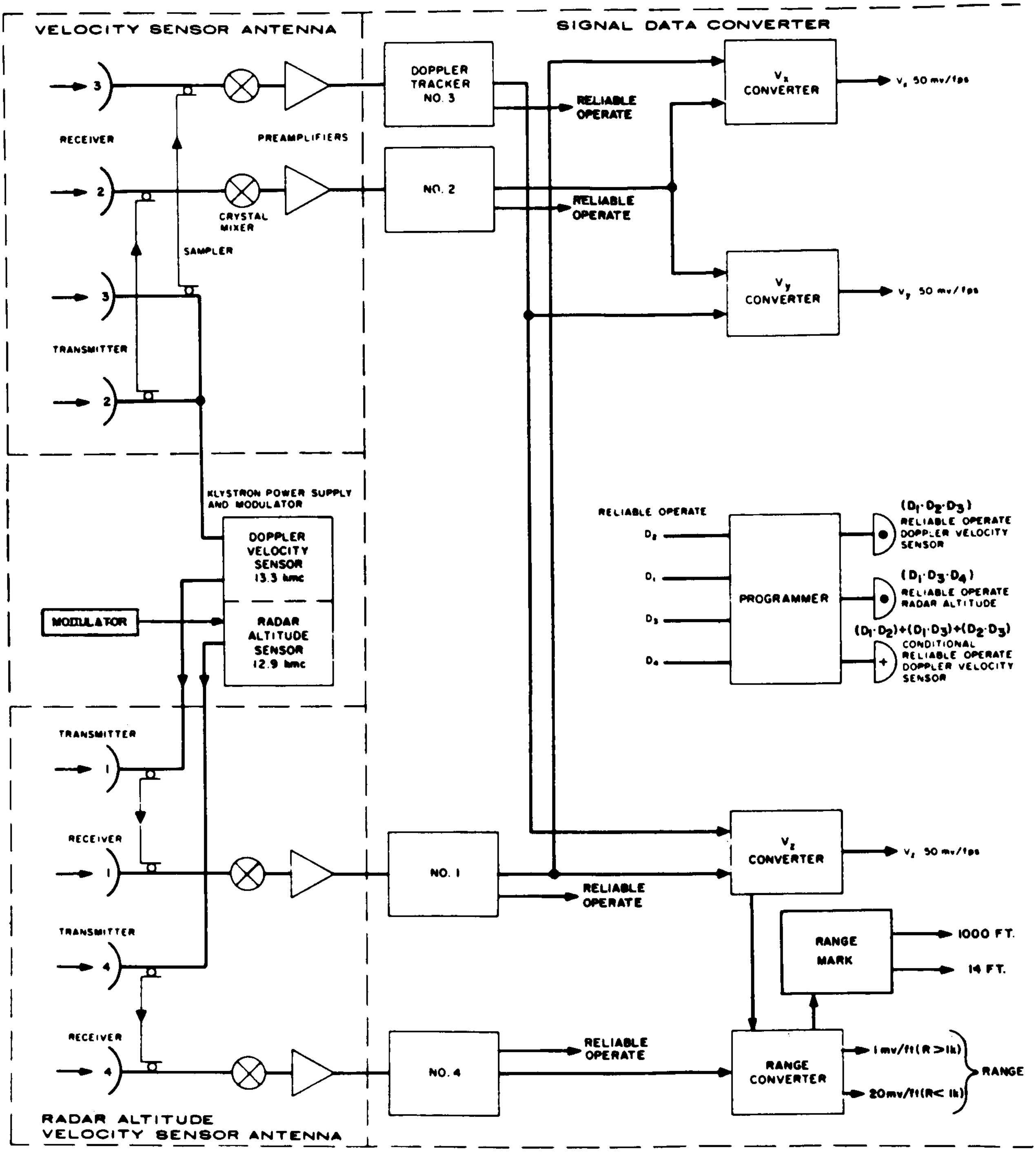


FIGURE 8-5. RADVS BLOCK DIAGRAM

A sample of the transmitted frequency in each beam is mixed with the received energy frequency, and their difference extracted by a crystal detector. The beam 4 difference frequency is proportionate to the r-f "round trip" time and therefore to range along the beam 4 path. The difference frequencies of beams 1, 2, and 3 are proportional to the velocity of the spacecraft along their respective beam paths.

Range is determined by compensating the beam 4 difference frequency for the doppler frequency shift errors to produce an "audio" signal corresponding to range along the spacecraft Z axis. Velocity is determined by summing the sensed doppler frequency shift of the reflections of each pair of constant-frequency beams with divergent paths. Three simultaneous equations are solved in the signal data converter to produce a doppler "audio" corresponding to the velocity along each spacecraft axis. The method of summing determines whether axial or transverse velocity is computed. By using each pair of velocity beams, i.e. 1-2, 1-3, 2-3, the velocity along the X, Z, and Y axes, respectively, is determined. Velocity and altitude capabilities and accuracies for RADVS are given in Appendix B, item 21.

Thermal control of the RADVS is passive, relying on surface coating and thermal capacity to ensure acceptable temperatures from the beginning to completion of its required operation. Power is supplied to the RADVS system from the unregulated 22-volt dc bus.

ROLL ACTUATOR

The roll actuator consists of a two-phase induction motor, a gear train, and an angular position transducer. This assembly provides roll-control moments during vernier engine thrust phase operation by swiveling vernier engine 1 over the range of ±5.5 degrees in a plane parallel to both the Z-axis and to a line through vernier engines 2 and 3 in response to electrical signals from the flight control subsystem. The actuator motor operates from 26-vrms 400-cps power with one phase fixed and the control phase variable. Position feedback is provided by an induction potentiometer operating from 10-vrms 400-cps power.

ATTITUDE JETS

The attitude jets supply the reaction forces for spacecraft orientation maneuvers during the period from Centaur-Surveyor separation through preretro-rocket firing.

The attitude jet system consists of a spherical tank containing approximately 4.5 pounds of nitrogen under high pressure; regulating, and dumping valves for gas supply control; and three pairs of opposed gas jets with solenoid operated valves for each jet. (See Appendix B, item 12.) The gas jet pairs are installed at the ends of the three landing legs shown in figure 3-2. Number one jet pair lies in the X-Y or horizontal plane for roll maneuvers. Jet pairs 2 and 3 are approximately parallel to the vertical or Z axis. Cumulative thrust produces pitch rotation, and differential thrust produces yaw rotation. Execution of specific maneuver commands is accomplished by actuation of the required solenoid valves to release nitrogen gas to the designated nozzles. The six solenoids are connected to solid-state switches in the flight control electronics. Each gas jet supplies a thrust of approximately 0.057 pound at a radius of 70 inches from the center of gravity for angular acceleration. The moment capability of the attitude jet system about each axis is:

Roll axis	±4.0 in-lb
Pitch axis	±4.25 in-lb
Yaw axis	±7.0 in-lb

The temperature of the nitrogen tank and the number 2 attitude jet are measured for telemetry to earth.

The thermal control of this system is passive, utilizing surface coatings and its own thermal capacity to maintain the temperatures of the gas supply and the jet valves within acceptable limits throughout the mission.

Definitive and descriptive documents for the flight control subsystem are listed in Appendix A, items 35 thru 44.

IX. APPROACH TELEVISION SUBSYSTEM

The approach television subsystem provides pictures of the lunar surface over the range from 1000 to 80 ±20 miles (slant range) above the lunar surface. During the approach interval covering this altitude range, up to 100 individual television frames will be taken in response to commands from earth. The approach television subsystem is composed of the downward-looking approach television camera and the television auxiliary unit, together with the appropriate cables and connectors. The approach television camera is illustrated in figure 9-1. The physical location of the approach television camera and its relation to other spacecraft items and structures is shown in figure 2-2.

The approach television camera field of view is fixed relative to the spacecraft coordinate system and is approximately 6.4 degrees by 6.4 degrees. A fixed-focus camera lens is employed which will provide in-focus pictures at all altitudes greater than 300 feet at the maximum aperture of f:4. The iris may be preset before launch over the range of f:4 to f:22. The center of the field of view of the approach television subsystem may be manually adjusted at the time of spacecraft assembly to be parallel to the spacecraft Z axis within ±0.5 degree.

An overall block diagram of the approach television subsystem is shown in figure 9-2. Only the approach camera itself (designated TV camera 4) and the switch for its electrical heater blanket are unique to the approach television subsystem. All other elements illustrated are shared with the survey television subsystem and/or other spacecraft subsystems. The approach television camera itself is completely self-contained, requiring only primary electrical power and command signals for operation. Upon command from earth at the appropriate point in the lunar approach phase, the camera will convert images of the lunar surface into complete composite-video signals that include all necessary synchronization and blanking signals. Such earth commands, received from the central command decoder turn the camera on and off, initiate the picture taking logic, and control the camera thermal power.

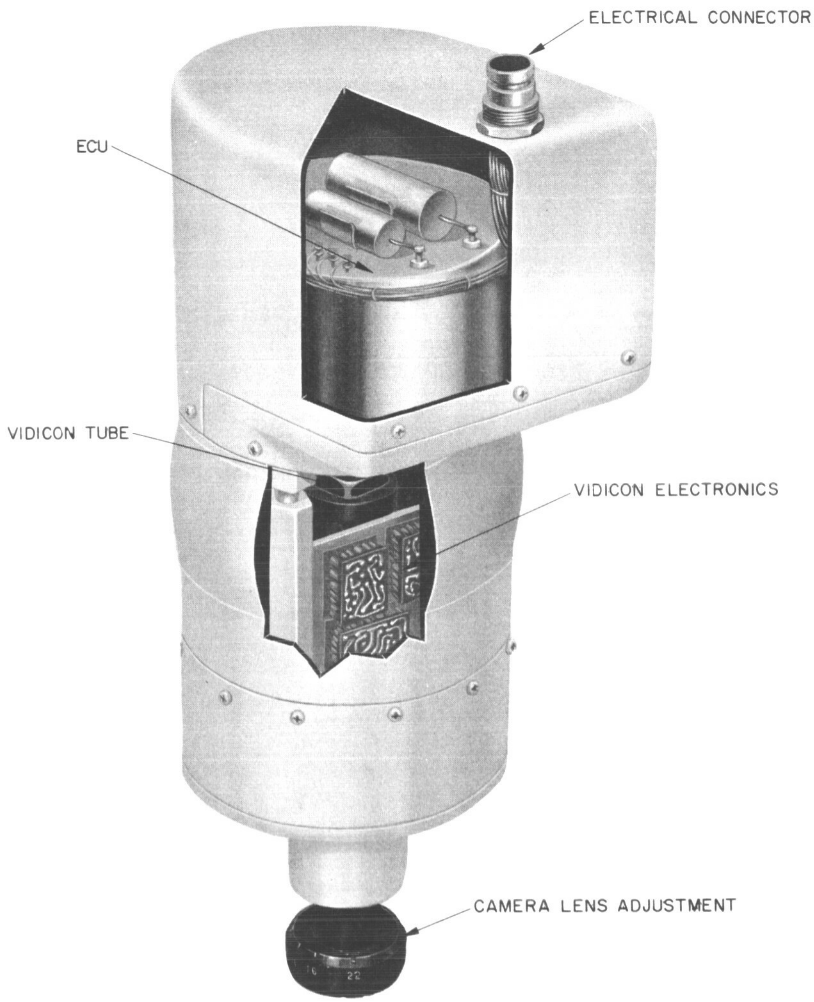


FIGURE 9-1. SURVEYOR TELEVISION APPROACH CAMERA

Electrical power for camera operation and for the active thermal control is delivered to the camera via the spacecraft harness from the basic bus central power control and distribution system. This electrical power consists of +29 vdc regulated voltage and +22 vdc unregulated voltage.

The composite single-frame video output from the camera is sent to the television auxiliary unit, passed through a summing amplifier, and then sent to the spacecraft transmitter. Two identical video outputs from the camera are conducted through separate cables to individual summing amplifiers in the TV auxiliary from which the signals are fed through individual cables to each spacecraft transmitter channel. Camera condition is observed by analog signals which are sent from the camera to the engineering signal processor. There they are

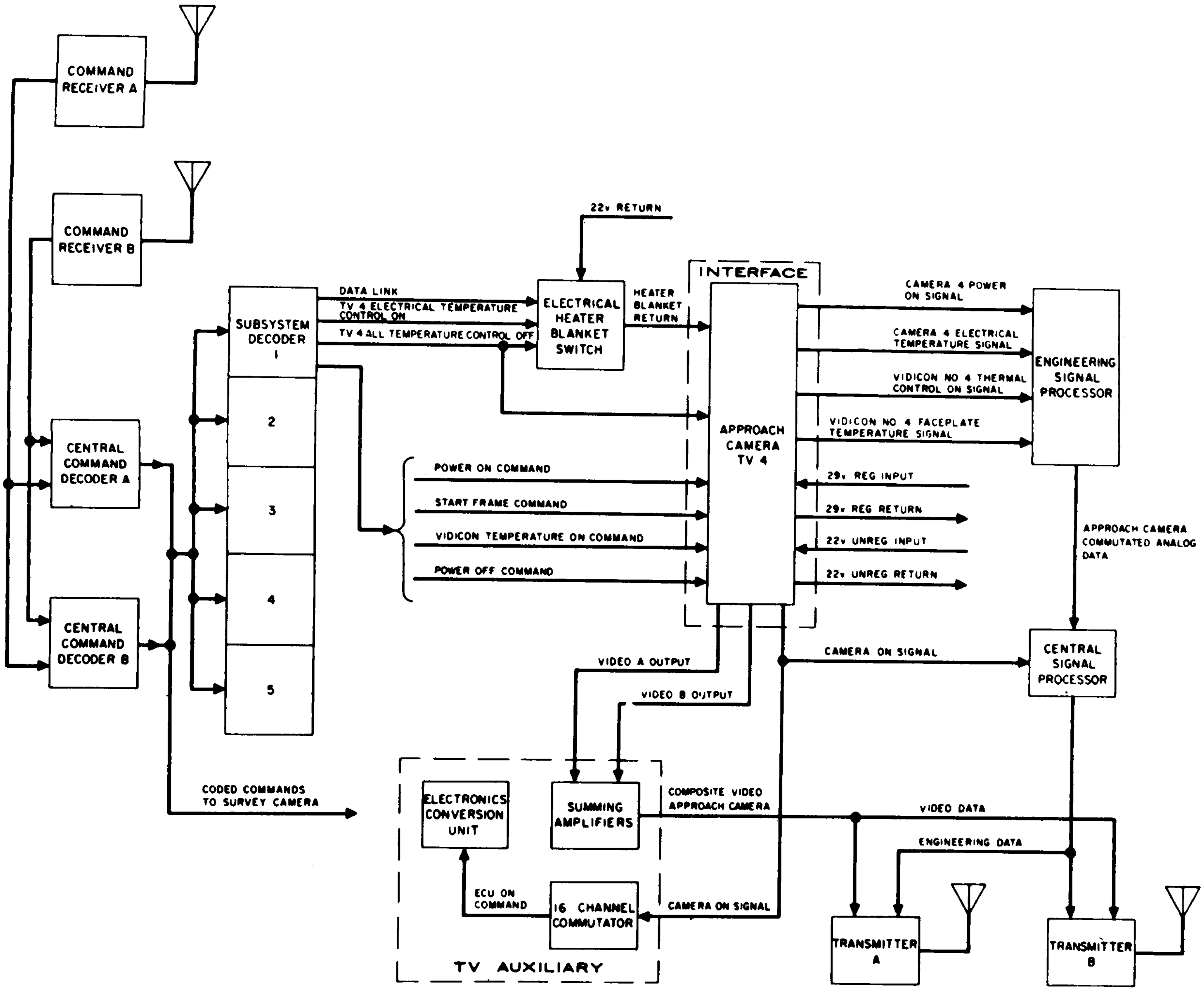


FIGURE 9-2. APPROACH TELEVISION SUBSYSTEM BLOCK DIAGRAM

commutated and sent to the central signal processor to be digitized and combined in sequence with other engineering data for transmission via the spacecraft transmitter between picture sequences. Analog data outputs from the camera, sent in this manner, typically consist of temperature and camera operational status indications.

Closed-loop thermal control of the camera vidicon faceplate is provided to maintain the temperature of the vidicon within appropriate operating limits during camera operation. The camera and associated circuitry have been designed to permit initial warmup of the camera electronics and the camera vidicon faceplate before actual camera operation.

The approach television camera provides 600 line-per-frame slow scan operation. Sensitivity of the camera is adjustable before launch to cover the range from 800 to 3000 foot-lamberts, with acceptable results at levels as low as 100 foot-lamberts.

The approach camera is located and mounted on the spacecraft in a position that will allow limited observation of an illuminated calibration chart and collimation lens that may be provided within the spacecraft shroud of the Centaur launch vehicle at the option of JPL. This arrangement allows prelaunch testing of the approach camera, with a resulting assessment limited by lighting and focus constraints. Observation of either the in-shroud calibration chart or the lunar surface is controlled by a shutter within the camera that prevents any light from reaching the vidicon faceplate until it is opened at the appropriate point in the sequence. The shutter also protects the vidicon from direct exposure to sunlight.

Figure 9-3 illustrates the picture-taking sequence controlled by the logic and timing circuitry of the camera that may be initiated upon command from earth. Horizontal and vertical generators are both free running and unsynchronized to reduce camera circuit complexity. These generators, which produce the camera synchronizing and blanking pulses, start automatically upon application of camera power and run continuously. The nominal 200-millisecond vertical blanking pulses represent the principal timing waveform for the picture-taking sequence. When a start frame command is received from the central command decoder, the camera video logic circuitry is enabled. The first vertical blanking pulse to appear thereafter initiates a picture-taking sequence and triggers the camera shutter to provide a nominal 20-millisecond exposure of the vidicon.

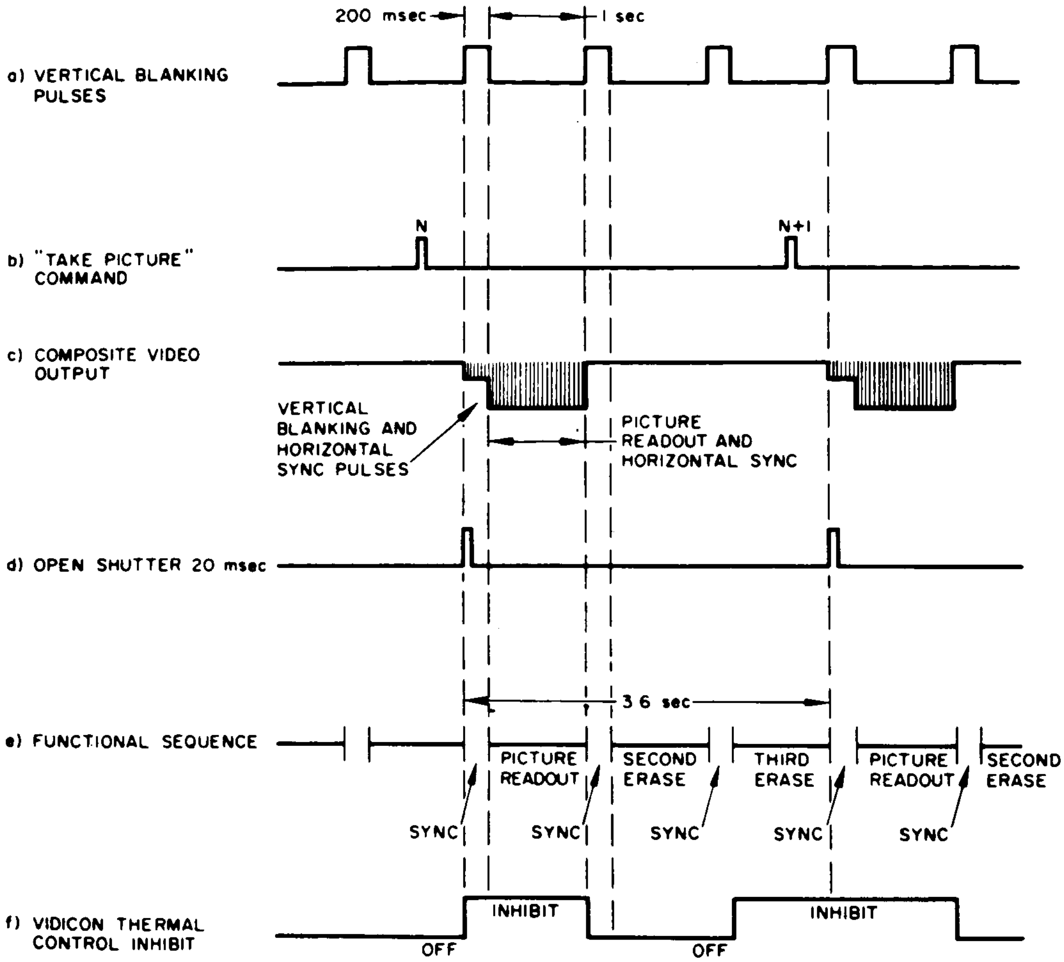


FIGURE 9-3. APPROACH TELEVISION PICTURE-TAKING SEQUENCE

The first data transmitted during a picture-taking sequence consists of the vertical blanking pulse with superimposed horizontal synchronizing pulses. This permits receiving and display monitors on earth to be synchronized with a vertical blanking pulse and with the horizontal synchronizing pulses before receipt of picture video. Single frame picture readout takes place in the interval following the vertical blanking pulse. Transmission of the composite video information requires approximately 1.2 seconds and is terminated at the start of the next (second) vertical blanking pulse. The second blanking pulse closes the video gate which feeds video to the television auxiliary and resets the shutter.

Three frame periods are required for proper vidicon erasure so that the subsequent start frame command must not be received at the spacecraft until after the third blanking pulse of the start frame sequence. This permits three erasure frames to occur (picture readout being the first one) in approximately 3.6 seconds. An alternate sequence of operation is possible in that the camera can receive a start frame command during the second erase scan (picture readout being the first), and the TV camera will take a picture and respond normally. In this case the previous image will not be completely erased. This alternate method can provide capability for transmission of a degraded picture every 2.4 seconds.

The defining document for the television subsystem is listed in Appendix A as item 45.

X. SCIENTIFIC PAYLOAD

INTRODUCTION

The basic design policies involved in providing the capability to accommodate various payload experiment subsystem combinations and the detailed implications on spacecraft weight, center of gravity, and electrical harness design, are described in Sections 2, 3 and 11. This section defines the other spacecraft provisions for scientific experiment subsystems in detail and describes the experiments themselves, with emphasis on electrical interfaces and operation.

An experiment subsystem is defined as all spacecraft items, other than those provided by the basic bus, which are uniquely necessary for the installation, mounting, deployment, thermal control, and functional performance of a scientific experiment on the lunar surface. Typically, an experiment subsystem is composed of the following items:

a. An instrument sensor assembly or mechanism or a combination of the two.

b. Mounting provisions and/or deployment or manipulative mechanisms to perform a secondary mechanical function by initiating or terminating mechanical action.

c. An instrument electronics unit directly associated with the calibration and operation of the instrument sensor.

d. An instrument auxiliary unit to provide the electrical interface and act as an "adapter" to match the electrical requirements of the instrument sensor and its electronics unit (if any) to the power, command and data transmission facilities provided by the basic bus.

e. Electrical cables necessary to provide interconnections between the other subsystem elements.

An "instrument" is defined as a sensor and its directly associated electronics. The electronics conversion unit and the instrument auxiliary are not considered a part of the instrument. The scientific instruments and accompanying electronics units and/or mechanisms required are either built by Hughes or furnished by JPL. The instrument auxiliaries are provided by Hughes.

The design approach outlined in Section 2 is implemented to provide basic bus subsystems that are, to the greatest extent possible, independent of instrument complement. The spacecraft basic bus provides only two power forms, unregulated 22 vdc battery power, and regulated 29 vdc power. The telemetry system is designed to accept only input signals in the range from 0 to 5 volts. The central command decoder is part of the basic bus, with provision for connection to subsystem command decoders for each instrument.

The auxiliary unit for each experiment subsystem then provides the interface circuitry between the standardized basic bus subsystems and each individual instrument as defined above. Each auxiliary unit contains a standard subsystem command decoder capable of supplying 8, 20, or 32 commands, as required. Power switches for turning the instrument on and off, for controlling heater power, and mode changes, etc., are located in these units. The instrument signal outputs are conditioned by the auxiliary unit as necessary to bring them into the proper voltage and impedance ranges. Subcarrier oscillators are also included in the auxiliary units. Special power supplies (ECU's) can be included where power form requirements are different from the basic bus supplies (none are necessary for the present scientific payload). Thus the basic bus subsystems do not have to be changed appreciably when the payload instrument complement is changed, and each experiment is largely independent of other experiments.

The A-21A Surveyor Spacecraft carries the following five experiment payload subsystems:

1. Survey Television Experiment Subsystem - transmits pictures of selected portions of the lunar surface and other scientific instruments on command from earth and is composed of two survey cameras and the television auxiliary.
2. Soil Mechanics - Surface Sampling Experiment Subsystem - investigates lunar surface properties and is composed of an instrument, and an instrument auxiliary unit.
3. Alpha Scattering Experiment Subsystem - performs elemental analysis of lunar surface material and is composed of a sensor, an instrument

digital electronics unit, instrument auxiliary unit, and a deployment mechanism.

4. Micrometeorite Detector Experiment Subsystem - provides data of individual lunar ejecta and is composed of a sensor, an instrument electronics unit, and an instrument auxiliary unit.

5. Seismometer Experiment Subsystem - measures physical disturbances of the moon and is composed of a sensor, an instrument electronics unit, and an instrument auxiliary.

The following descriptions of the individual experiment subsystems include detailed discussions of the functional elements incorporated in each auxiliary.

SURVEY TELEVISION EXPERIMENT SUBSYSTEM

The survey television experiment subsystem provides the capability of observing the lunar surface, portions of the spacecraft, and large sections of free space, on command from earth. This capability is provided through an experiment subsystem consisting of two cameras and a television auxiliary. Each of the cameras can be commanded to alter its angular field of view and to change the angular orientation of the center of the field of view with respect to the basic spacecraft coordinate system. Provisions are also made for inserting colored or polarizing filters into the camera optical system on command from earth. This arrangement permits colorimetric and polarimetric evaluation of individual television pictures. These features are augmented and made practical by the provisions for commanding the camera optical systems to alter focus distance and lens aperture to adjust for variation in object distance and light intensity. Provision is made to alter the lens opening (iris) either on direct command from earth or automatically, as desired. With both cameras viewing the same area, stereoscopic coverage may be obtained. The Survey Television Subsystem is also used to observe the operation of selected instruments carried by the spacecraft.

A functional block diagram of the survey television subsystem is illustrated in figure 10-1. All of the items appearing on this diagram are shared with the approach television subsystem. The telecommunications subsystem is shared with other spacecraft subsystems. The survey television camera is illustrated in figure 10-2. The location of the survey television camera on the spaceframe and its relation to other units is illustrated in the general arrangement drawing (figure 2-2).

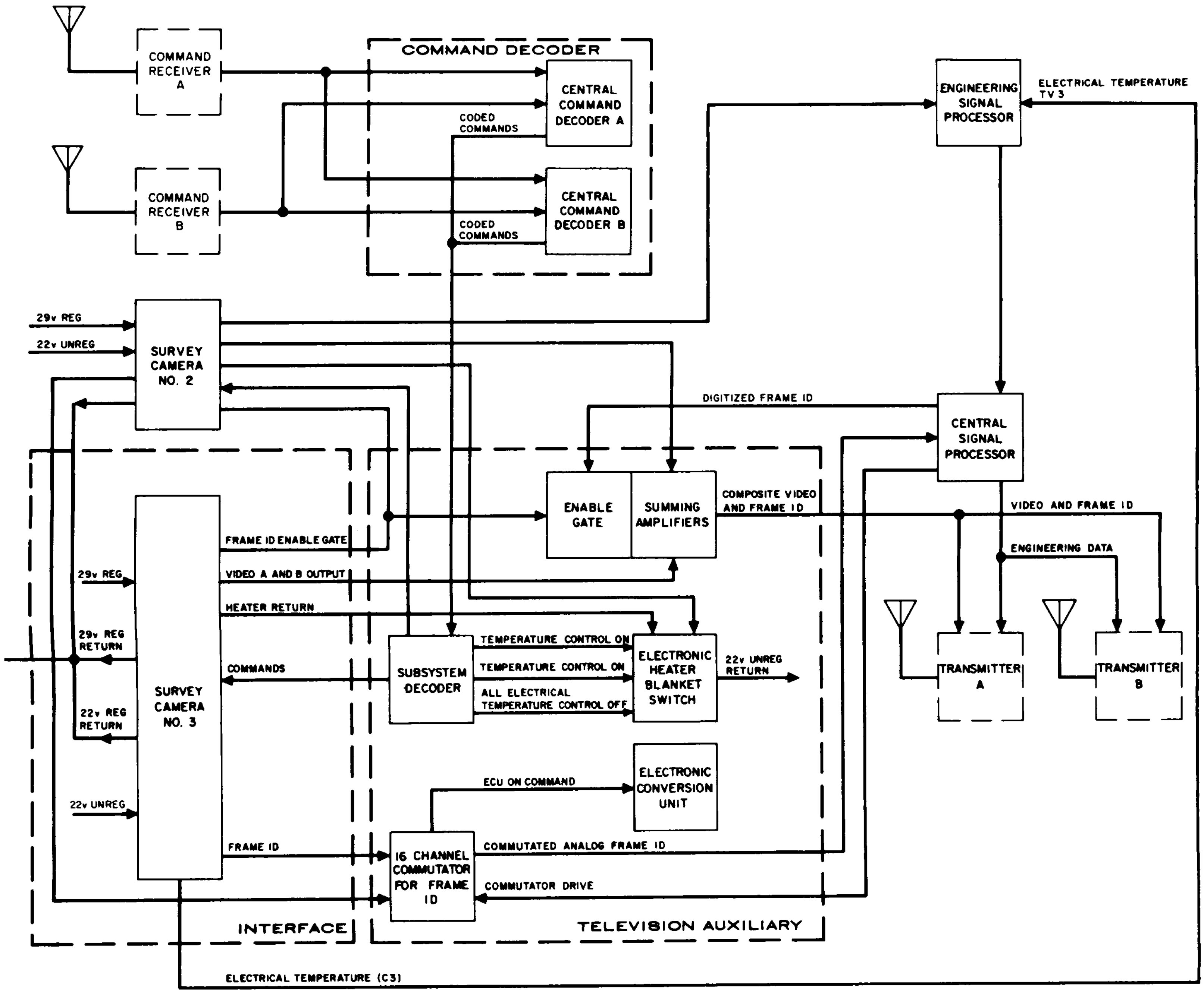


FIGURE 10-1. SURVEY TELEVISION SUBSYSTEM, FUNCTIONAL BLOCK DIAGRAM

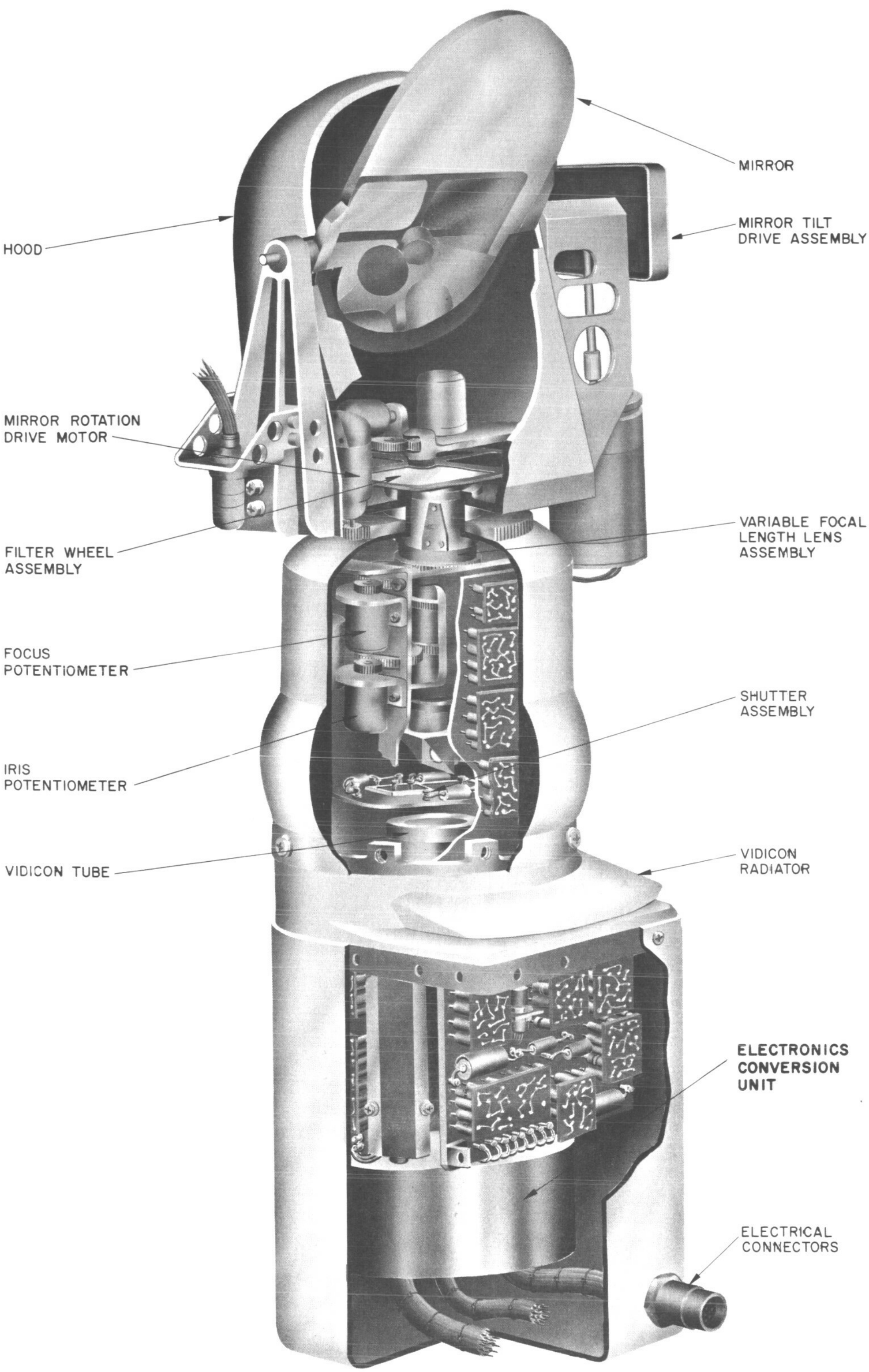


FIGURE 10-2. SURVEY TELEVISION CAMERA

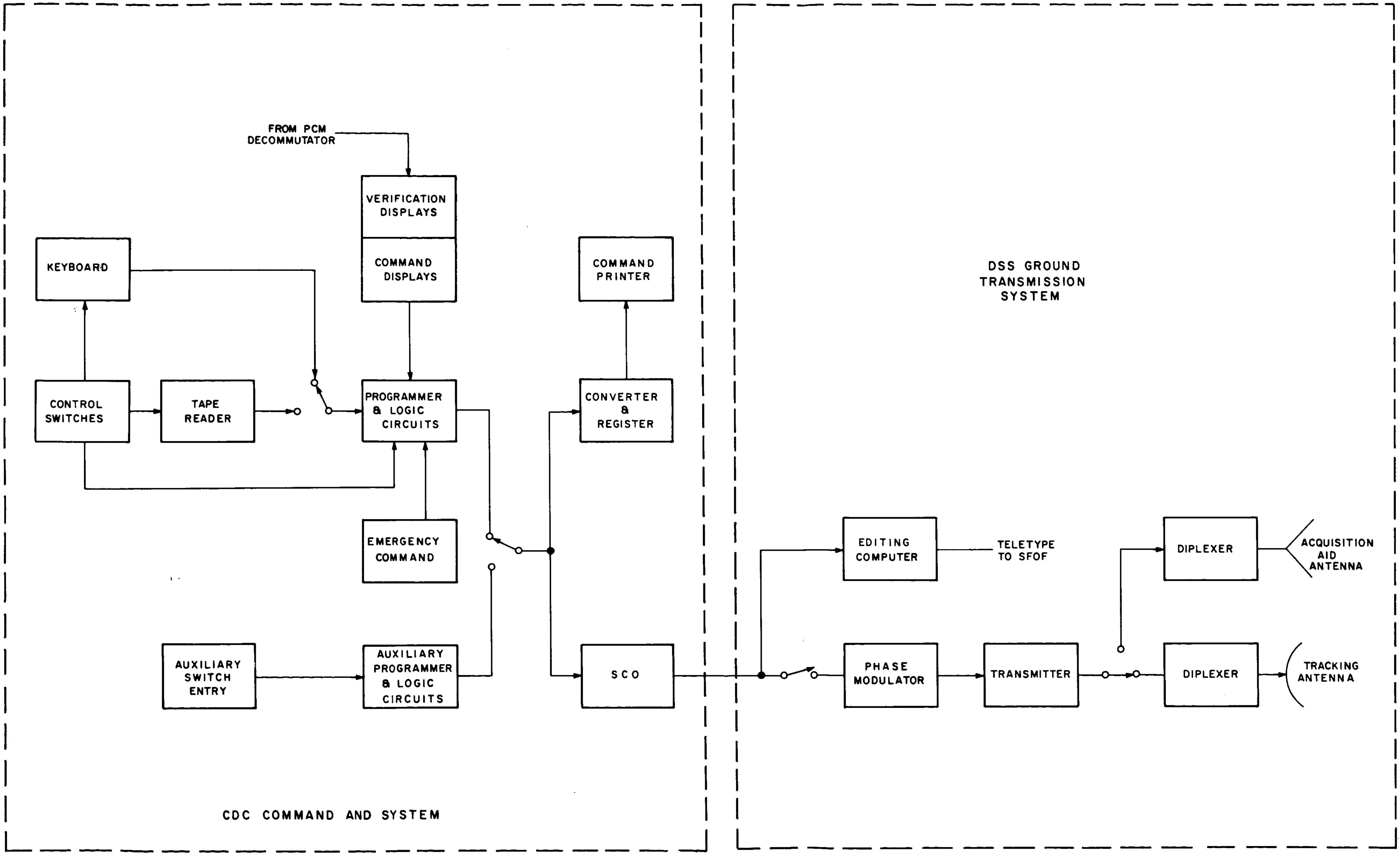
The relationship of the spacecraft survey television experiment subsystem to the CDC and the DSS ground transmission system is illustrated in figure 10-3. The CDC, in its command operation, provides commands for the spacecraft. The DSS transmitter processes and modulates the CDC commands for transmission to the spacecraft.

The survey television experiment subsystem receives the commands from the DSS, performs the desired operations, and transmits video, camera, and command confirmation information back to the DSS. (The DSS is located at the DSIF). Upon reception, the DSS amplifies the signal from the spacecraft and prepares it for demodulation and decommutation in the CDC. After demodulation by the CDC, the signal is sent to discriminators and decommutators in the CDC Telemetry System and to video processing in the CDC Television System.

TV frame identification information is sent to the television system for video display and recording on 35 mm film. Selected telemetry signals may be monitored on binary, decimal, and analog meter displays. In the television system, a permanent record of video displays may be obtained shortly after readout by using a Land camera to photograph the output of the television monitor.

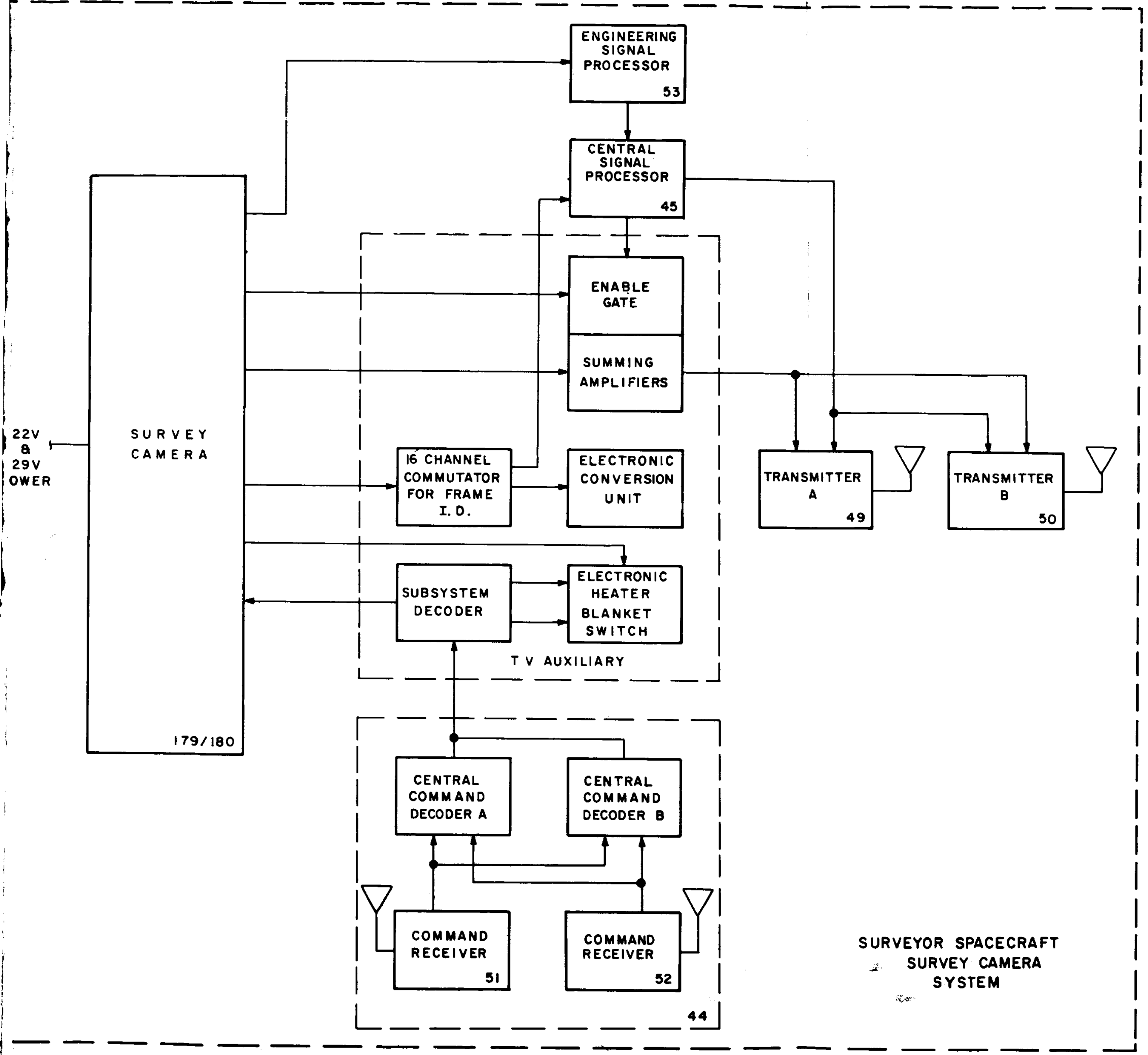
The Survey television camera converts optical images within its field of view into complete composite video signals which include horizontal synchronization and vertical blanking pulses. In performing this function, the camera is completely self-contained, requiring only electrical power inputs and commands received from the central power distribution system and the television auxiliary. The commands received by the survey camera are decoded by a subsystem decoder in the television auxiliary. The subsystem decoder receives coded earth commands via the command receiver and central command decoder. All power for operation and active thermal control of the camera is supplied from the central power distribution system as +29 vdc regulated voltage and +22 vdc unregulated voltage.

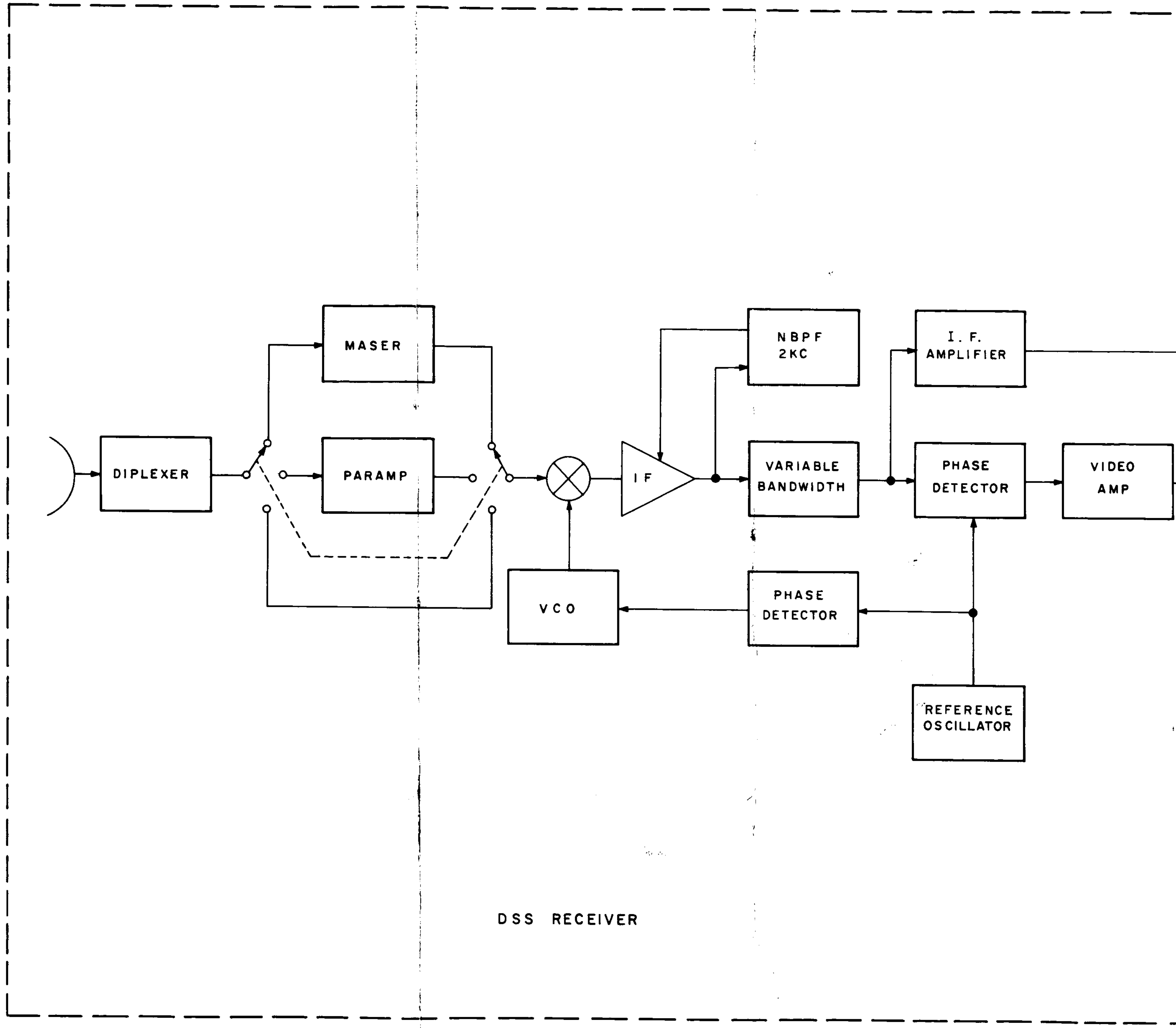
The principal signal output from the camera is the composite signal-frame video. Two identical signal-frame video outputs are supplied from the camera to the summing amplifier of the TV auxiliary unit through individual cables. Frame identification information from the camera is received by the summing amplifiers through an enabling gate. Frame identification data from the camera contains all optical, electronic, thermal, and angular position information necessary to describe the relative location and engineering characteristics pertinent to each


FROM PCM DECOMMUTATOR
VERIFICATION DISPLAYS
COMMAND DISPLAYS
KEYBOARD
COMMAND PRINTER
DSS GROUND TRANSMISSION SYSTEM
CONTROL SWITCHES
TAPE READER
PROGRAMMER & LOGIC CIRCUITS
CONVERTER & REGISTER
EMERGENCY COMMAND
EDITING COMPUTER
TELETYPE TO SFOF
DIPLEXER
ACQUISITION AID ANTENNA
AUXILIARY SWITCH ENTRY
AUXILIARY PROGRAMMER & LOGIC CIRCUITS
SCO
PHASE MODULATOR
TRANSMITTER
DIPLEXER
TRACKING ANTENNA
CDC COMMAND AND SYSTEM

91-1 A

91-2





B

91-3

91-4

C

91-5

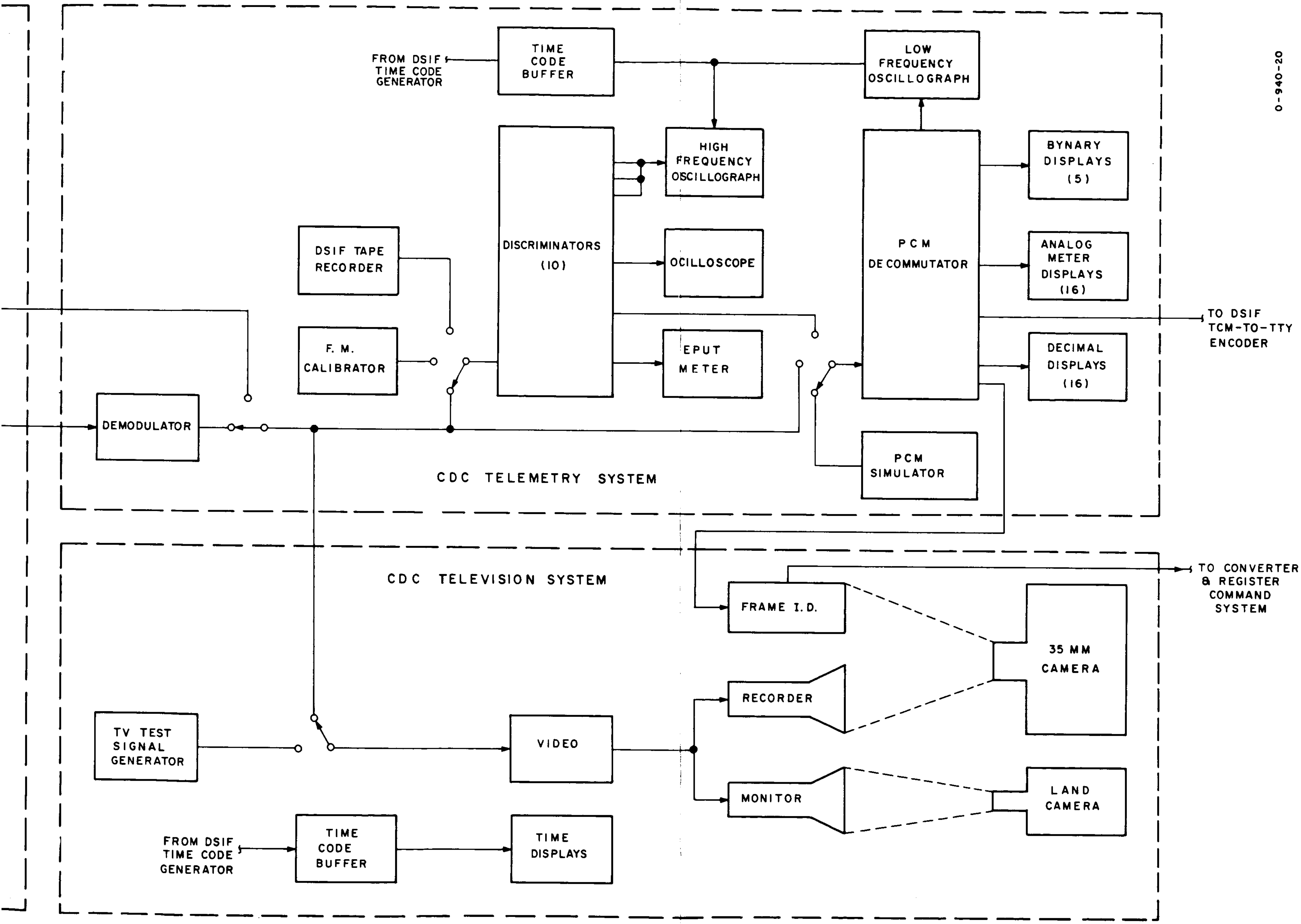


FIGURE 10-3. SURVEY TV CAMERA/GROUND EQUIPMENT INTERFACE, FUNCTIONAL BLOCK DIAGRAM

91-6

televised scene. The frame identification signal consists of analog data that has been serially commutated in the TV auxiliary and routed to the central signal processor. The central signal processor converts the analog data to digital form. This signal is then returned to the TV auxiliary and summed with composite video as pulse coded data in the summing amplifiers. The total signal from the summing amplifiers is sent to the transmitters in a sequence consisting of composite video information with frame identification data following immediately thereafter (figure 10-4).

The survey television camera is equipped with a variable focal length lens that can be commanded to set at either its maximum focal length position to provide a narrow-angle (6.4 x 6.4 degrees nominal) viewing capability or at its minimum focal length position to provide wide-angle (25.4 x 25.4 degrees nominal) viewing capability. The focus distance of the lens is also commandable to cover the range from 4 feet to infinity. The camera indirectly views the particular scene to be televised by means of a gimbaled mirror assembly. The elevation gimbal axis is mounted inside of the azimuth gimbal axis. Stepping motors are provided in the mirror assembly to position the mirror surface angularly in either azimuth or elevation in accordance with earth commands. The optical centerline of the camera field of view may be stepped in increments of approximately 3 degrees in azimuth and 5 degrees in elevation. The mechanical motion of the mirror itself in elevation is in increments of approximately 2-1/2 degrees per step, or exactly half the angular increment of the camera elevation optical path. A one-to-one relationship exists between the mechanical motion of the mirror in azimuth and the corresponding rotation of the camera field of view in azimuth. This occurs because azimuth motion corresponds to angular rotation around an axis very nearly parallel to the optical centerline of the camera lens.

Temperature control of the survey camera is provided through a passive thermal radiator for the vidicon faceplate and small electrical heaters located on the electronic chassis within the camera. In addition, closed-loop thermal control is also provided for the vidicon faceplate to achieve proper camera operation.

The survey camera is capable of operating in either a 600-line TV normal mode or in a 200-line TV emergency mode. The camera is turned on in the normal mode and then may be switched to the emergency mode if required. The intent of the latter mode is to permit the transmission of degraded quality pictures in a limited signal bandwidth not as wide as that normally required for television

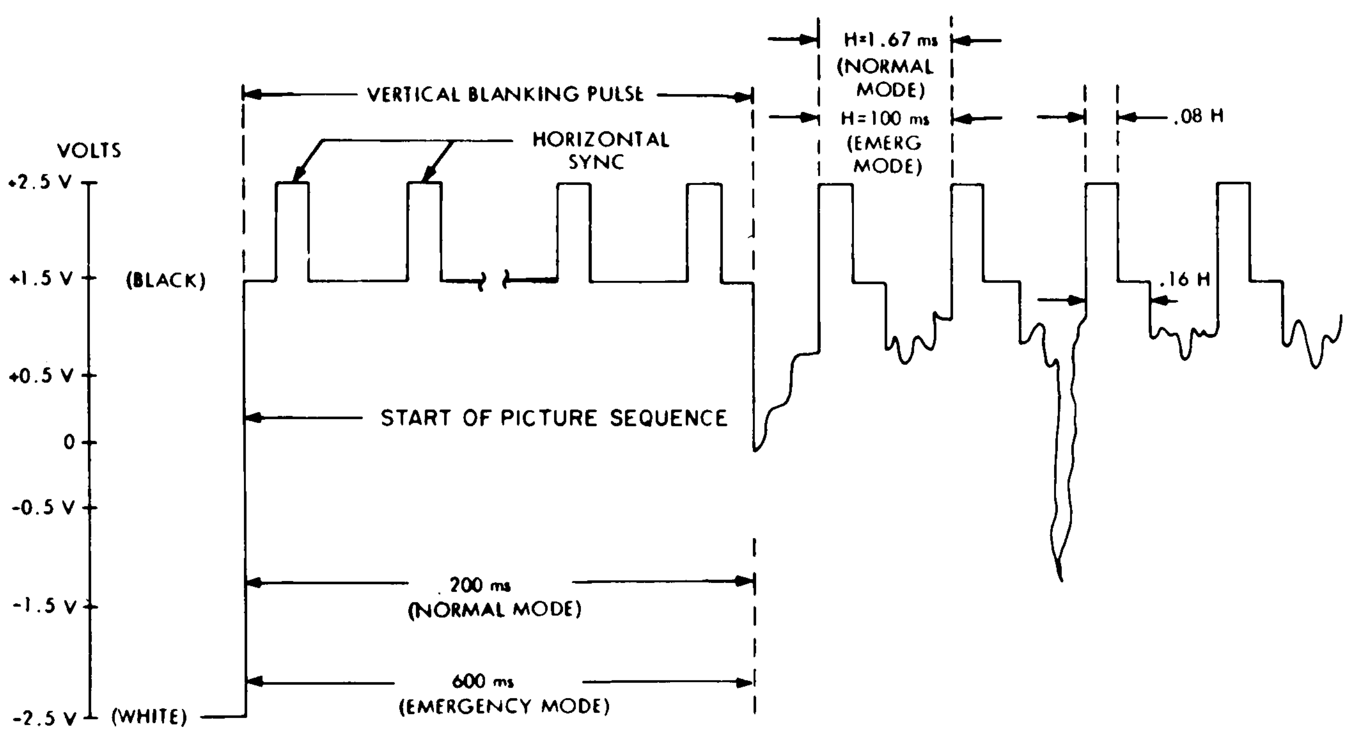


FIGURE 10-4. SURVEY TELEVISION COMPOSITE VIDEO OUTPUT

transmission. The basic objective in providing this mode of transmission is to increase the probability of successful television transmission to earth when the high-gain antenna or the high powered transmitter is not usable. (Successful receipt and display of emergency mode television pictures requires use of appropriate ground equipment compatible with the narrow-band transmission of the emergency mode.)

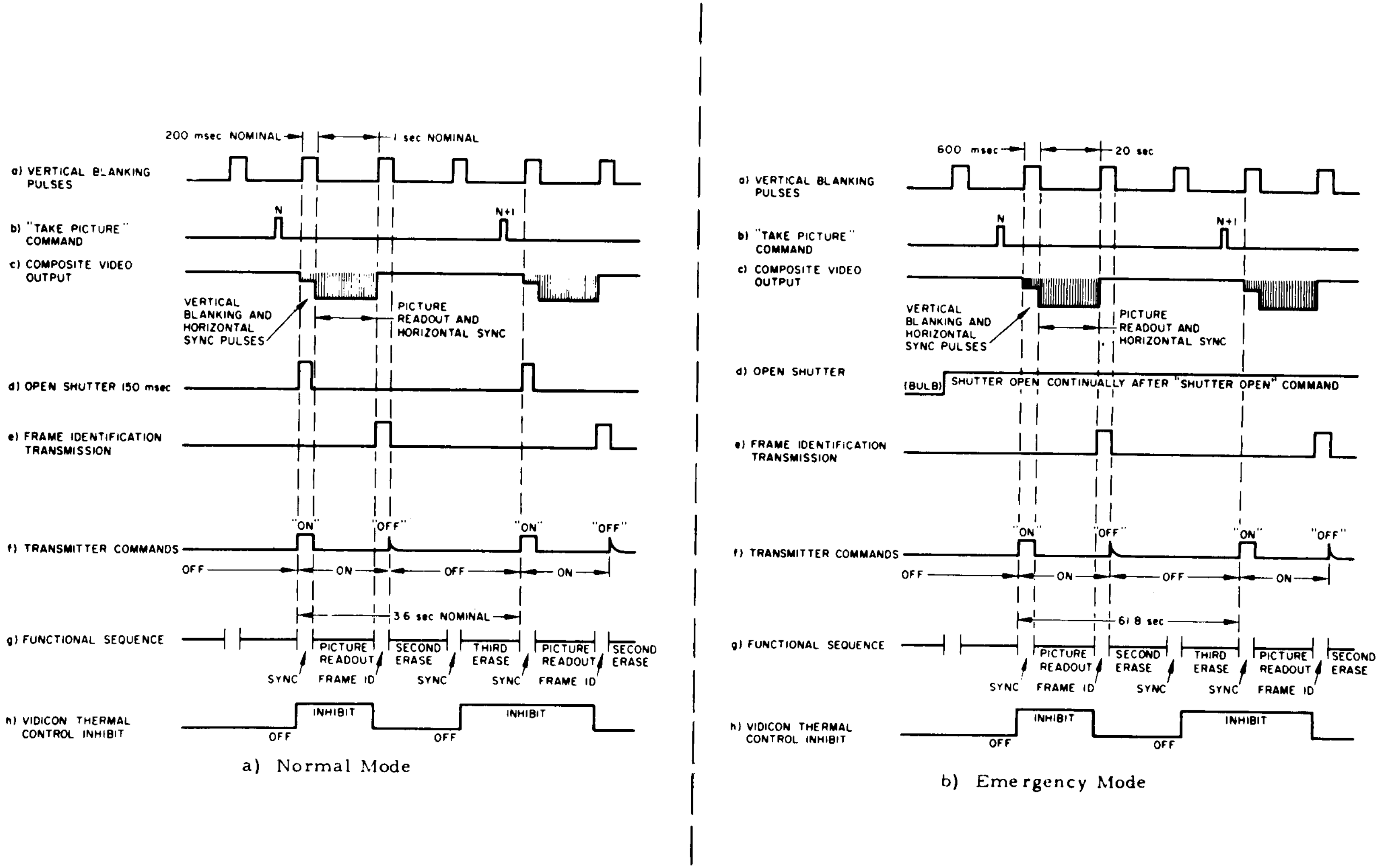
The survey camera is equipped with a focal plane shutter that provides a nominal exposure of 150 milliseconds. The shutter is also capable of being opened for an indefinite period of time on receipt of proper earth command. This operating mode is not directly equivalent to a "bulb" exposure for picture-taking purposes. The maximum equivalent exposure time is about 1 second, regardless of continued exposure beyond this interval. A reflex beam splitter is provided in front of the shutter in the camera to sample the total light input passed by the lens. If the sampled light is sufficiently intense and the iris is not a f:22, the circuitry driving shutter is inhibited and prevents the shutter from opening until the excessive light intensity is removed. At f:22 in the manual mode the shutter will open regardless of light intensity. Light-intensity changes may occur as a result of changes: (1) in the area of coverage viewed by the camera, (2) in the angular orientation of the camera field of view, or (3) in the direct or reflected illumination of the scene viewed by the camera because of a change in sun angle. The same light-sensing arrangement is also employed in the automatic iris control circuitry. Provision is made to override the shutter inhibit function on receipt of ground commands.

The camera horizontal synchronizing pulses and the vertical blanking pulses are independently generated when the camera is turned on (power applied) and remain until it is turned off. The television system design does not require a fixed timing relationship between the horizontal and vertical oscillators. Therefore, they are both free-running and independent of each other. This design approach permits a significant saving in circuit complexity and power required. In the normal mode the vertical blanking pulse represents the principal timing waveform of the picture sequence. When a start frame command is received (via the command receivers and central command decoder) from the TV auxiliary subsystem decoder, the camera video logic circuitry is enabled. The first vertical blanking pulse that appears after an earth-initiated start frame command, in turn, initiates a complete picture sequence. This blanking pulse triggers the camera shutter to provide vidicon exposure and, simultaneously, causes a signal to be generated to turn the transmitter on.

The first signal transmitted during a typical picture-taking sequence consists of the vertical blanking pulse with superimposed horizontal synchronizing pulses. This waveform is intended to permit ground monitors to be synchronized before receipt of the actual picture video. Single-frame picture readout takes place in the interval following vertical blanking (figure 10-5a). Transmission of the composite video signal requires about 1.2 seconds in the normal mode and is terminated at the start of the next (second) vertical blanking pulse. The second blanking pulse closes the video gate which provides a video signal to the transmitter, resets the camera shutter, and acts as an enable pulse to close the frame identification input gate to the video summer in the TV auxiliary. Several complete sequences of frame identification information are transmitted during the enable pulse. The trailing edge of this pulse initiates a transmitter-off command that terminates transmission for that particular picture-taking sequence.

Three frame periods are necessary for proper vidicon erasure. Therefore, the subsequent start frame command must not be received at the spacecraft until after the third vertical blanking pulse of the start frame sequence. This will permit three complete erasure frames to occur (picture readout erasure occurs during the first frame). The three erasure frames require nominally 3.6 seconds in the normal mode and 61.8 seconds in the emergency mode (figure 10-5b). It is possible for the spacecraft to receive a start frame command during the second erase scan and the TV camera to take a picture and respond normally. In this case, since the third erasure has not yet occurred, the previous TV image will not be completely erased from the vidicon. Through this specialized operating sequence, the capability is provided to transmit a degraded (imperfectly erased) picture approximately every 2.4 seconds.

The survey television camera is equipped with a filter wheel that may be rotated on receipt of proper earth commands to occupy one of four different positions, each providing use of a different optical filter. It is therefore possible, before time of camera assembly, to select combinations of clear, colored, or polarizing filters, as desired, to enable observation of lunar scenes through optical filters with known characteristics.



a) Normal Mode

b) Emergency Mode

FIGURE 10-5. SURVEY TELEVISION PICTURE-TAKING SEQUENCE

SOIL MECHANICS-SURFACE SAMPLING EXPERIMENT SUBSYSTEM

The purpose of the soil mechanics-surface sampling experiment subsystem is: (1) to qualitatively determine the mechanical characteristics of the lunar surface; (2) to provide means for lunar soil manipulation; and (3) to map surface elevations within its area of operation. The surface properties to be investigated are: nature of the material, shear strength, modulus of elasticity and surface contour.

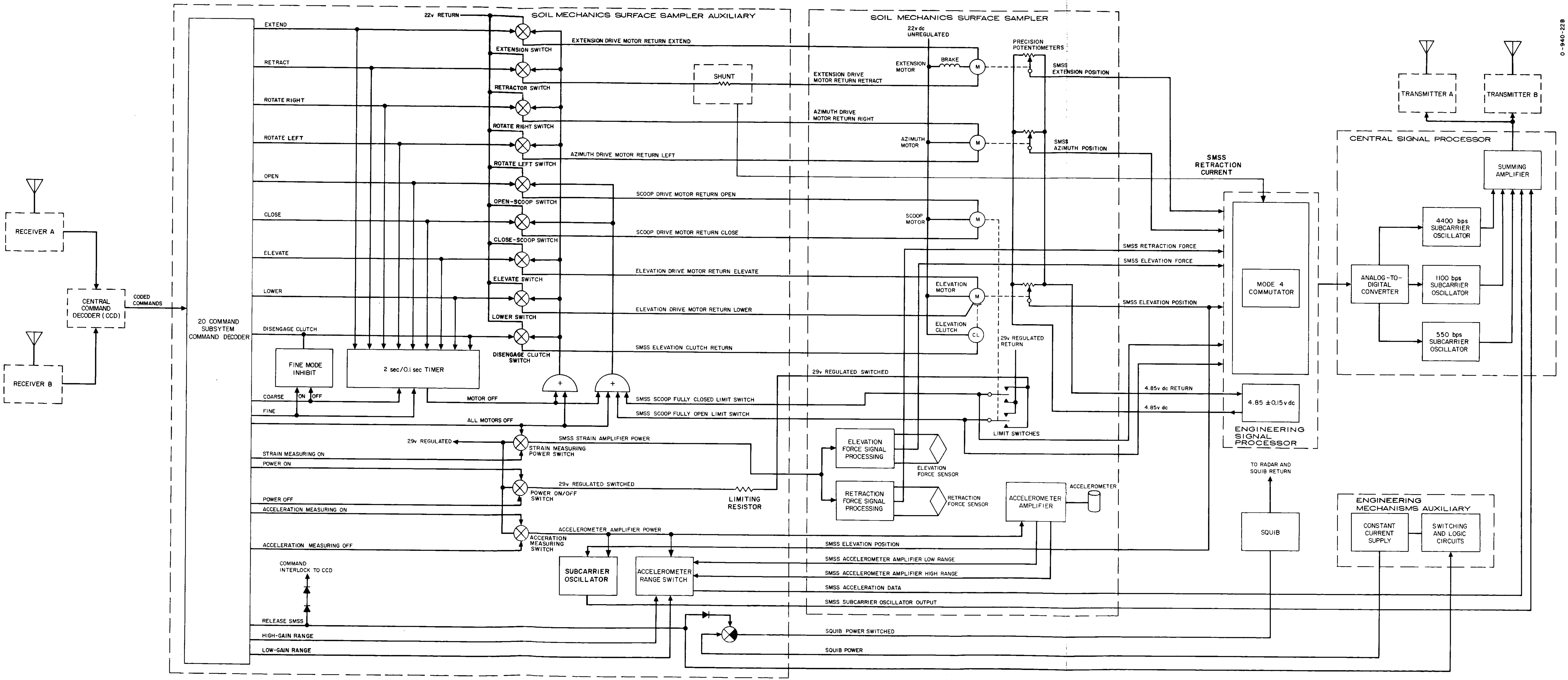
The soil mechanics-surface sampling experiment subsystem will test the lunar surface by digging, scraping, picking, and contour mapping (in conjunction with television viewing) in a limited area near the Surveyor spacecraft. The experiment subsystem will measure those parameters necessary to determine the lunar soil properties listed above.

Experiment Subsystem Characteristics

The soil mechanics-surface sampling experiment subsystem is composed of an instrument, an instrument auxiliary unit, electrical cables necessary to provide interconnections between the instrument and the instrument auxiliary unit, and mounting bracketry necessary to mount the instrument on the spaceframe. Figure 10-6 is a functional block diagram of the experiment subsystem.

Instrument Description

The basic soil mechanics-surface sampler consists of a clamshell-shaped device (scoop) attached to the end of an extension arm (see figure 10-7). The scoop and arm are mounted on a horizontally and vertically hinged base which is attached to the spaceframe. The extension arm assembly is manipulated by three electric motors that control the azimuth, elevation and extension motions. A fourth motor operates the scoop door. Potentiometers are used to measure extension, elevation, and azimuth positions of the instrument. Limit switches are provided to indicate the open and closed positions of the scoop door. An acceleration measuring system, providing a dual-range output (0 to 50 g and 0 to 2000 g), is mounted on the instrument to measure vertical deceleration of the scoop during the picking action. Two force transducers are provided; one measures vertical forces over the range from 0.1 to 3.0 pounds and the other measures retraction forces over the range of 0.1 to 20.0 pounds.



A

99-1

B

99-2

FIGURE 10-6. SOIL MECHANICS SURFACE SAMPLER EXPERIMENT SUBSYSTEM, FUNCTIONAL BLOCK DIAGRAM

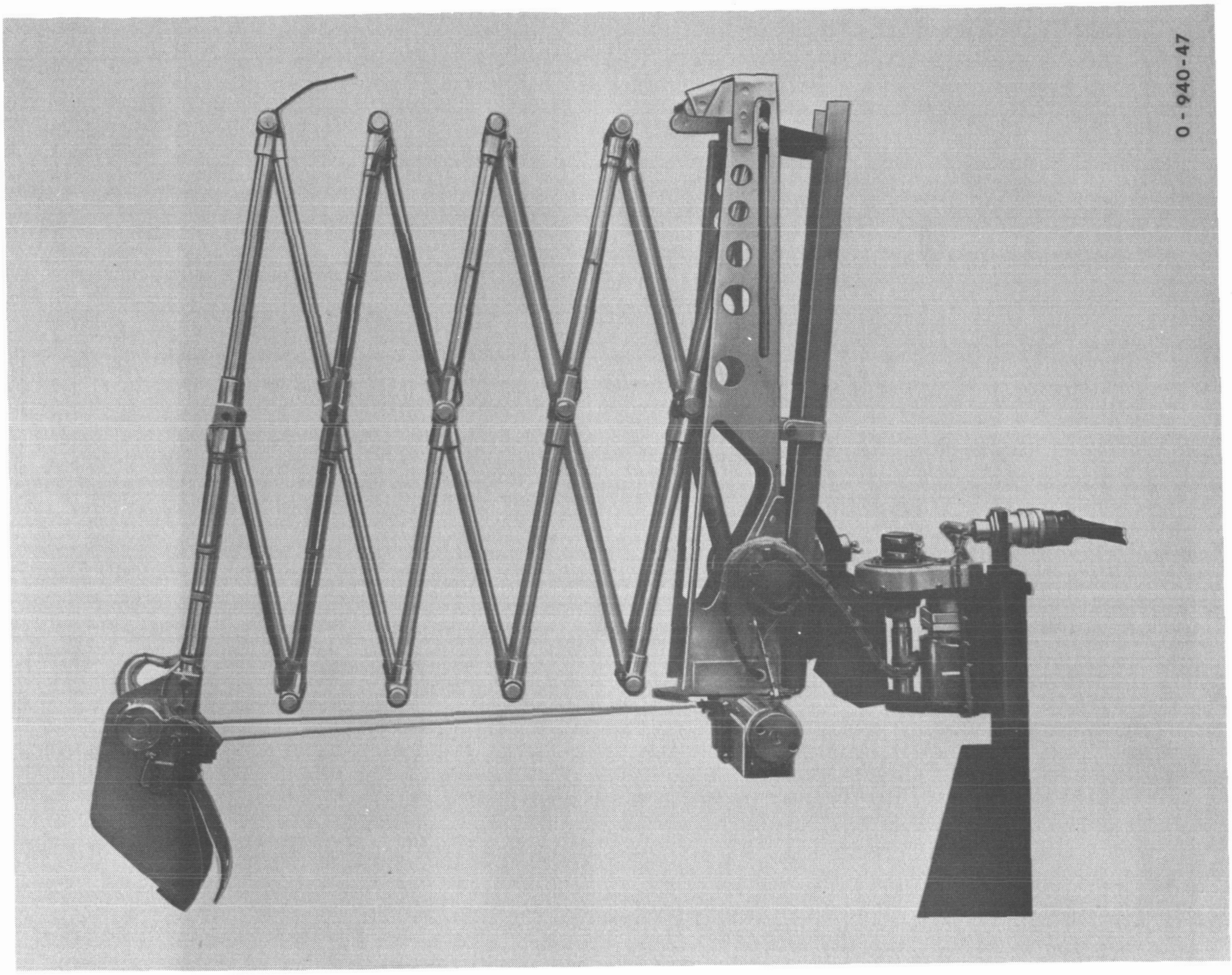


FIGURE 10-7. SOIL MECHANICS SURFACE SAMPLER INSTRUMENT PARTIALLY EXTENDED

Alternate light and dark striped markings on the inside and outside of the scoop may be viewed by the survey television to determine penetration depth of the scoop, grain size, and the level of lunar material contained within the scoop.

Operation of the instrument consists of manipulation in extension, elevation and azimuth; disengagement of the elevation drive by a clutch device for picking action; retraction of the extension mechanism along the surface for digging; and lowering the scoop to the surface with the elevation drive disengaged for point-to-point mapping of the lunar surface.

The instrument can perform lunar-surface testing within a sector defined by its operating limits in azimuth, extension, and retraction. Either coarse or fine incremental movements position the scoop within its sector of operation. The fine movements occur in maximum increments of 0.6 inch in extension, 2.5 degrees in elevation, and 3.0 degrees in azimuth. Motor control and coarse or fine increment selection by earth command is accomplished by appropriate logic and timing action in the instrument auxiliary unit.

Three series-wound, split-field, direct-current motors move the instrument in azimuth, elevation, and extension-retraction. Another motor opens and closes the scoop door. An electromechanical clutch disengages the elevation drive train so that the scoop can fall freely for picking action.

The motors and the clutch receive power directly from the spacecraft 22 vdc unregulated supply and are controlled by solid-state switches located in the instrument auxiliary unit. Only one motor or clutch will be operated at any one time. The instrument mechanism is capable of developing the following forces at the scoop:

a. Extension — 1-lb radial push.

b. Retraction — 20-lb radial pull.

c. Azimuth — 1-lb either way at maximum extension.

d. Elevation — 30-lb up or down at maximum extension.

e. Scoop jaw — 6-lbs measured at edge.

The instrument is equipped with precision potentiometers to indicate static instrument position within the envelope of operation. It also has two force-transducer systems, an accelerometer transducer system, and two limit switches to indicate fully open or fully closed scoop positions. The potentiometers receive excitation from the 4.85 vdc potentiometer supply in the engineering signal processor (ESP). Their outputs are processed through commutator mode 4 in the ESP. In addition, the output from the elevation position potentiometer is routed via a selector switch and SCO in the instrument auxiliary unit for processing during picking action only. This is necessary to measure the radial velocity of the scoop at impact.

Each of the two force-transducer systems consist of a strain gage bridge, a signal conditioning amplifier, and an interconnecting cable provided to measure vertical and retraction forces. The signal conditioning amplifiers contain a regulated bridge power supply and a d-c amplifier. Spacecraft 29 vdc regulated power is supplied to the signal conditioning amplifiers which generate a precise bridge excitation voltage. The bridge outputs are routed to a d-c amplifier the output of which is processed via the Mode 4 commutator. One strain gage bridge is located so that elevation forces (from 0 to 3 pounds) can be measured. Another strain gage bridge is mounted so that radial forces from 0 to 20 pounds applied to the scoop along the tape can be measured.

An acceleration transducer system, consisting of a sensor, an amplifier, and an interconnecting cable, measures scoop deceleration at impact during picking action. The sensor is mounted on the scoop near the blade. The output signal from the sensor is routed to the amplifier where it is simultaneously amplified through dual channels. The ratio of the two channel gains is approximately 40 to 1 with the amplitude of both channel outputs ranging from 0 to 5 volts dc. When the sensor output is low, the high-gain channel can be monitored. Conversely, when the sensor output is high, the low-gain channel can be monitored. Both channel outputs are routed to the soil mechanics surface sampler auxiliary where, upon earth command, a switch will select the desired gain. The selected output is sent from the instrument auxiliary unit to the telecommunications subsystem, where it is modulated directly upon the main carrier. Two limit switches, mounted on the scoop mechanism, indicate whether the scoop door is fully open or fully closed. The switches are excited by 29 vdc regulated voltage.

The switch outputs are routed to two destinations, (1) to digital channels on the Mode 4 commutator to be transmitted back to earth, and (2) to the instrument auxiliary unit to turn off the scoop motor when the scoop is either fully open or fully closed.

The soil mechanics surface sampler instrument is designed for the following thermal conditions:

Operating temperature range	-65 to 257°F
Survival temperature	-300 to 257°F

Prior to touchdown, the instrument mechanism is secured to the spacecraft by a clamp which may be released by a pyrotechnic device. The pyrotechnic device is energized from a 9.5 ampere squib power supply in the engineering mechanism auxiliary and controlled by a switch in the instrument auxiliary unit. A command received from earth by the telecommunications subsystem is routed via the instrument auxiliary unit subsystem decoder to the engineering mechanism auxiliary and simultaneously to a switch in the instrument auxiliary unit. The command simultaneously turns on the squib power supply in the engineering mechanism auxiliary and the switch in the instrument auxiliary unit. The squib power supply turns off automatically after approximately 20 milliseconds. The actuating command is interlocked to prevent inadvertent firing of the squib.

Instrument Auxiliary Unit

The instrument auxiliary unit provides command decoding, signal processing, and power management for the soil mechanics-surface sampling experiment subsystem. These functions are performed by a 20-command subsystem command decoder, power switches, a current measuring shunt, a timer, a range selector switch, an SCO, and appropriate logic circuitry located within the instrument auxiliary unit.

The subsystem command decoder processes the inputs received from the CCD and provides pulse output commands required in the instrument auxiliary unit and in the instrument. Commands from the subsystem decoder perform the functions listed in table 10-1.

The 2 sec/0.1 sec timer limits the operating time of the instrument motors or clutch to either 2 ± 0.4 seconds (coarse) or 0.1 ± 0.02 second (fine). Normally, the instrument is turned on in the coarse timing mode with the fine mode selectable upon earth command. During fine mode operating, switching logic prevents release of the elevation clutch to prevent damage to the instrument. The fine and coarse time intervals for motor and clutch operating directly govern the fine and coarse incremental movements, respectively, of the instrument. The instrument auxiliary unit contains a current shunt that measures extension-retraction motor current during scoop retraction only. The shunt output is sent to a current measuring channel in the Mode 4 commutator.

ALPHA SCATTERING EXPERIMENT SUBSYSTEM

The alpha scattering experiment subsystem performs compositional analysis of lunar surface materials. A portion of the lunar surface adjacent to the spacecraft is bombarded with 6 mev alpha particles. Backward scattered alpha

TABLE 10-1. SOIL MECHANICS-SURFACE SAMPLER COMMANDS

No.	Command Title	Sequence Initiated
0.	SMSS power on	Turns on 29 v power used in the auxiliary and in the instrument. Enables all experiment functions.
1.	SMSS power off	Turns off 28 v power used in the auxiliary and in the instrument. Disables all experiment functions.
2.	Extend SMSS	Power applied to extension motor to move sampler out.
3.	Retract SMSS	Power applied to extension motor to move sampler in.
4.	Rotate SMSS left	Power applied to azimuth motor to move sampler left.
5.	Rotate SMSS right	Power applied to azimuth motor to move sampler right.
6.	Elevate SMSS	Power applied to elevation motor to raise sampler.
7.	Lower SMSS	Power applied to elevation motor to lower sampler.
8.	Open SMSS scoop	Power applied to scoop motor to open scoop.
9.	Close SMSS scoop	Power applied to scoop motor to close scoop.
10.	Disengage SMSS clutch	Power applied to clutch to permit pick action operation.
11.	All motors off	Removes all power from motor, clutch and strain measuring function.
12.	SMSS coarse mode	Switches timer from fine mode to coarse mode.
13.	SMSS fine mode	Switches timer from coarse to fine mode. Inhibits disengage clutch command.
14.	SMSS strain measuring on	Applies power to strain measuring function.
15.	SMSS acceleration measuring on	Applies power to accelerometer measuring function and SCO.

TABLE 10-1. SOIL MECHANICS-SURFACE SAMPLER COMMANDS (Cont)

No.	Command Title	Sequence Initiated
16.	SMSS acceleration measuring off	Removes power to accelerometer measuring function and SCO.
17.	SMSS high-g range	Sets accelerometer selector switch to low gain setting.
18.	SMSS low-g range	Sets accelerometer selector switch to high gain setting.
19.	Release SMSS	Energizes squib actuated pin-puller to release SMSS from stowed position.

particles, as well as protons generated within the sample by the incident alpha particles, are detected by solid-state surface barrier detectors.

Alpha Scattering Experiment Characteristics

The alpha scattering experiment subsystem is composed of a sensor, an instrument digital electronics unit, an instrument auxiliary unit, a deployment mechanism, and electrical cables.

The alpha scattering experiment subsystem detects scattered alpha particles from all elements except hydrogen and helium. The proton system detects protons from lithium, boron, nitrogen, fluorine, sodium, magnesium, aluminum, silicon, phosphorus, and sulfur. Although the sensitivity of the measurement varies from one element to another, the detection threshold is expected to be about 1 percent by weight. Operation of the sensor will be impaired by surface contaminants thicker than 0.1 micron.

The important scientific measurements are (1) the energies of the protons and scattered alpha particles, and (2) the intensity of protons and scattered alpha particles as a function of energy. The energy of each detected particle is determined by pulse height analysis. The pulse height, which is proportional to the energy of the detected particles, is converted to a time signal which is then coded into a 9-bit word. These 9-bit words, which comprise the instrument output, identify the pulse height channels or energies of the detected particles.

The instrument is packaged in two parts – the digital electronics and the sensor. The digital electronics contains the command memory circuits, power converter, and digital logic to convert the variable width pulses from the sensor into synchronous binary form for transmission. The command memory uses spacecraft commands to set flip-flops controlling the outputs of the detectors in the sensor. This enables proper operation if a detector becomes noisy or is contaminated with radioactive material.

Sensor Description

The sensor, which will be deployed directly to the lunar surface, contains a radioactive source of alpha particles with two alpha detectors and four proton detectors. The detectors are of the surface barrier type. The alpha detector outputs are amplified and applied to a pulse height-to-time converter. The alpha data pulse output of the height-to-time converter is applied to a gated clock in the alpha scattering instrument digital electronics unit. The proton detectors are similar to the alpha detectors except that the proton detectors have a thin metal foil in front of them to stop alpha particles. To prevent registering background radiation as a proton event, guard detectors are provided. The outputs of the guard detectors are amplified and applied to a discriminator that inhibits the proton channel pulse height-to-time converter. Signals from the command memory control the operation of each alpha detector and each proton detector with the associated guard detector.

When measuring composition, the sensor must operate in a vacuum and requires an alpha source of 25 to 100 millicuries of Curium-242. However, in testing the instrument, it is not necessary to use the same alpha source required for lunar operation in vacuum. Weak alpha sources can be mounted close to the detector so that reproducible spectra are obtained at atmospheric pressure. For vacuum tests, a different set of sources may be employed. Tests must be of sufficient length to allow accumulation of enough counts to avoid large statistical uncertainties. The instrument requires special treatment since it has an operating temperature range of -40°C to +50°C and a survival temperature range of -166°C to +75°C. At low temperatures, temperature control is provided by a 5-watt electrical heater. At high temperatures, the thermal design of the sensor case together with appropriate surface finish prevent the temperature of the instrument from rising above the survival temperature.

High accuracy temperature sensors, in the sensor and in the instrument electronics unit, will be monitored at regular intervals upon command from earth to maintain the temperature requirements of this instrument. The weight of the instrument electronics unit is 3.2 ± 0.2 pounds. A typical operating sequence for this instrument is as follows:

a. Standard sample — 3 hours (sensor located in stowed position).

b. Background count — 3 hours (sensor deployed to background count position).

c. Initial count on the lunar surface — 6 hours (sensor deployed to lunar surface).

d. Data accumulation — minimum of 24 hours of data (sensor deployed to lunar surface).

In order to obtain reliable data, the total interrupted time in steps a and b must not exceed 30 minutes, and the interrupted time in step c 1 hour. The time between the initiation of step a and of step c must not exceed 7-1/2 hours for the same reason.

Instrument Digital Electronics Unit

The instrument digital electronics unit, located within compartment B (see figure 3-6), weighs 5.0 ± 0.2 pounds.

The control circuits for the two instrument outputs, alpha and proton pulses, are identical except for the readout rates. The alpha readout bit rate is 2200 cps, while the proton readout frequency is 550 cps. The logic circuit uses a gated 900-kc clock and a 7-bit counter to convert the pulse from the sensor into a 7-bit binary word. The 7-bit word is shifted into a storage register to minimize the effective dead time, and then into an output register where a sync bit and parity check are generated. This results in a 9-bit word consisting of a sync bit followed by 7 data bits and a parity bit. If no data have been received since the previous word, a sync bit followed by 8 zeroes is generated.

The power converter within the digital electronics unit generates instrument operating voltages from the spacecraft 29 vdc supply.

Instrument Auxiliary Unit

The instrument auxiliary unit provides the electrical interface between the instrument and the spacecraft. The instrument auxiliary unit (see figure 10-8) contains a command decoder capable of providing 20 commands, two subcarrier oscillators, two power switches, and three silicon-controlled rectifiers (SCRs). The instrument auxiliary unit is similar in appearance to the seismometer instrument auxiliary unit shown in figure 10-15.

The "Alpha Data Output" and the "Proton Data Output" signals from the instrument digital electronics unit modulates individual SCO's in the instrument auxiliary unit. These outputs are present simultaneously and are digital, NRZ, with a format consisting of a sync bit followed by 7 data bits, with the most significant bit first, and a parity bit following the least significant bit. A digital "one" is represented by 0 ± 0.25 volts and a digital "zero" by 5 ± 0.25 volts. Source impedance is approximately 5000 ohms.

The outputs of the SCOs are applied to a presumming amplifier and then to two phase-summing amplifiers in the central signal processor. The output of the two phase-summing amplifiers is phase modulated onto the main carrier of the spacecraft transmitter.

The two analog outputs provided by the high accuracy temperature sensors in the experiment are both in the range of 0 to 5 volts and are processed by the ESP.

Electrical power for the instrument operation and active thermal control is delivered to the instrument via the payload harness from the electrical power subsystem at 29 vdc regulated and 22 vdc unregulated. Power for the experiment is controlled by power switches within the instrument auxiliary unit. The instrument power requirements from the 29 v and 22 v busses are 1.4 and 5 watts, respectively.

Deployment Mechanism

The sensor deployment mechanism provides the means for stowing the instrument sensor prior to spacecraft touchdown, and deploying it, in an irreversible sequence, from the stowed; to the background count; to the lunar surface position.

Support for the sensor in the stowed position (figure 10-9a) is provided by a mounting platform. The mounting platform also serves as a dust cover for the sensor, holds the standard sample in the proper position for calibration, and positions the sensor in proper relationship to the spacecraft to provide the required thermal radiation view factor for standard sample count. In the stowed position, the sensor is held in place by a steel band connected to the mounting platform.

Release of the sensor from the stowed to the background count position (figure 10-9b) is accomplished by activating an explosive squib actuated pin-puller which disconnects the steel band. Proper orientation of the sensor to the lunar surface is provided by a small orientation band which rotates the sensor as it is being deployed from the stowed to the background count position. As the deployment mechanism swings out, after release, the orientation band unreels until the sensor is 15 degrees past the vertical axis. At this point the band begins to exert a force on the sensor drum, causing the sensor to rotate as it and the deployment arm continue out toward the background count position.

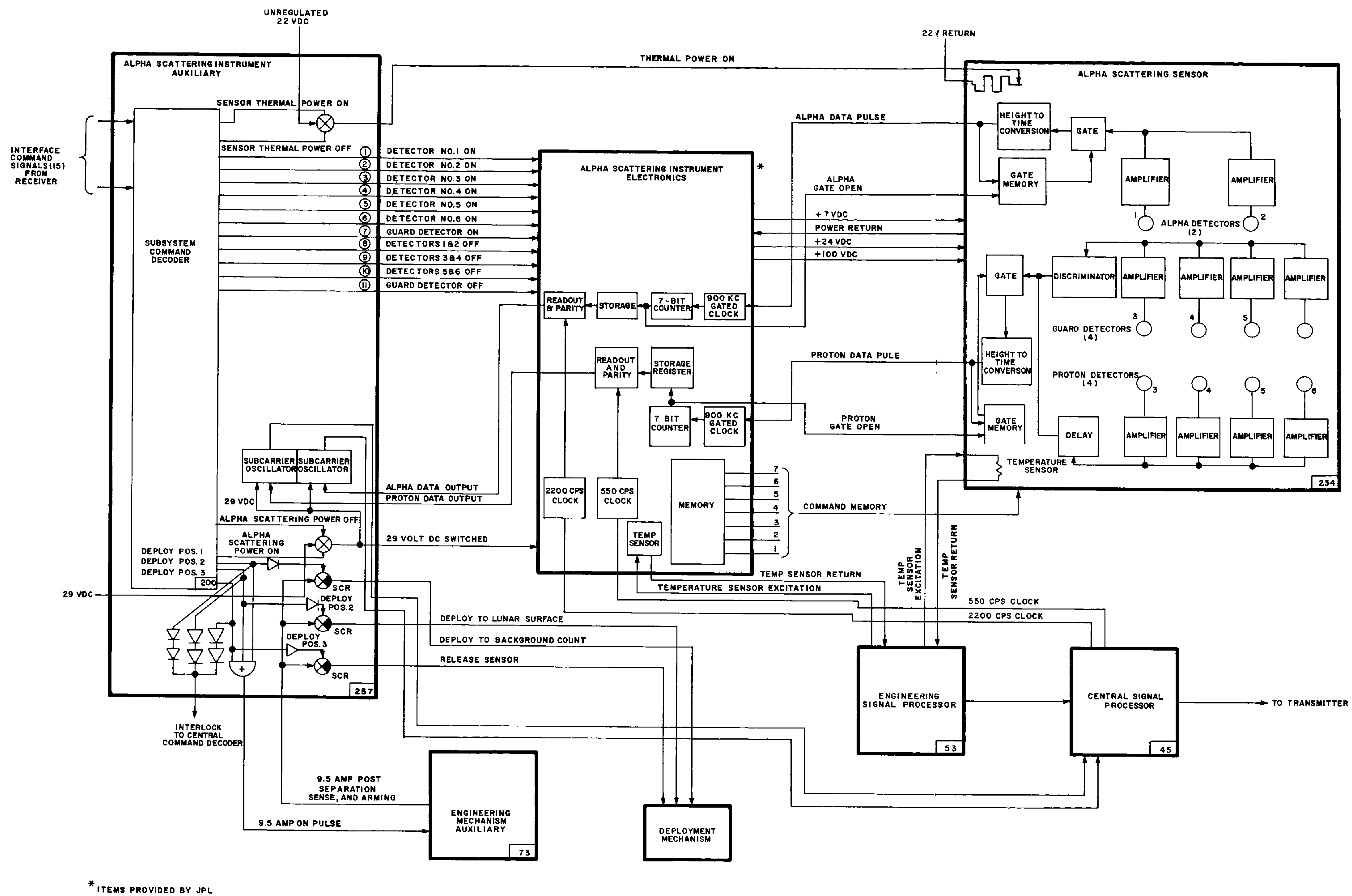


FIGURE 10-8. ALPHA SCATTERING EXPERIMENT SUBSYSTEM, FUNCTIONAL BLOCK DIAGRAM

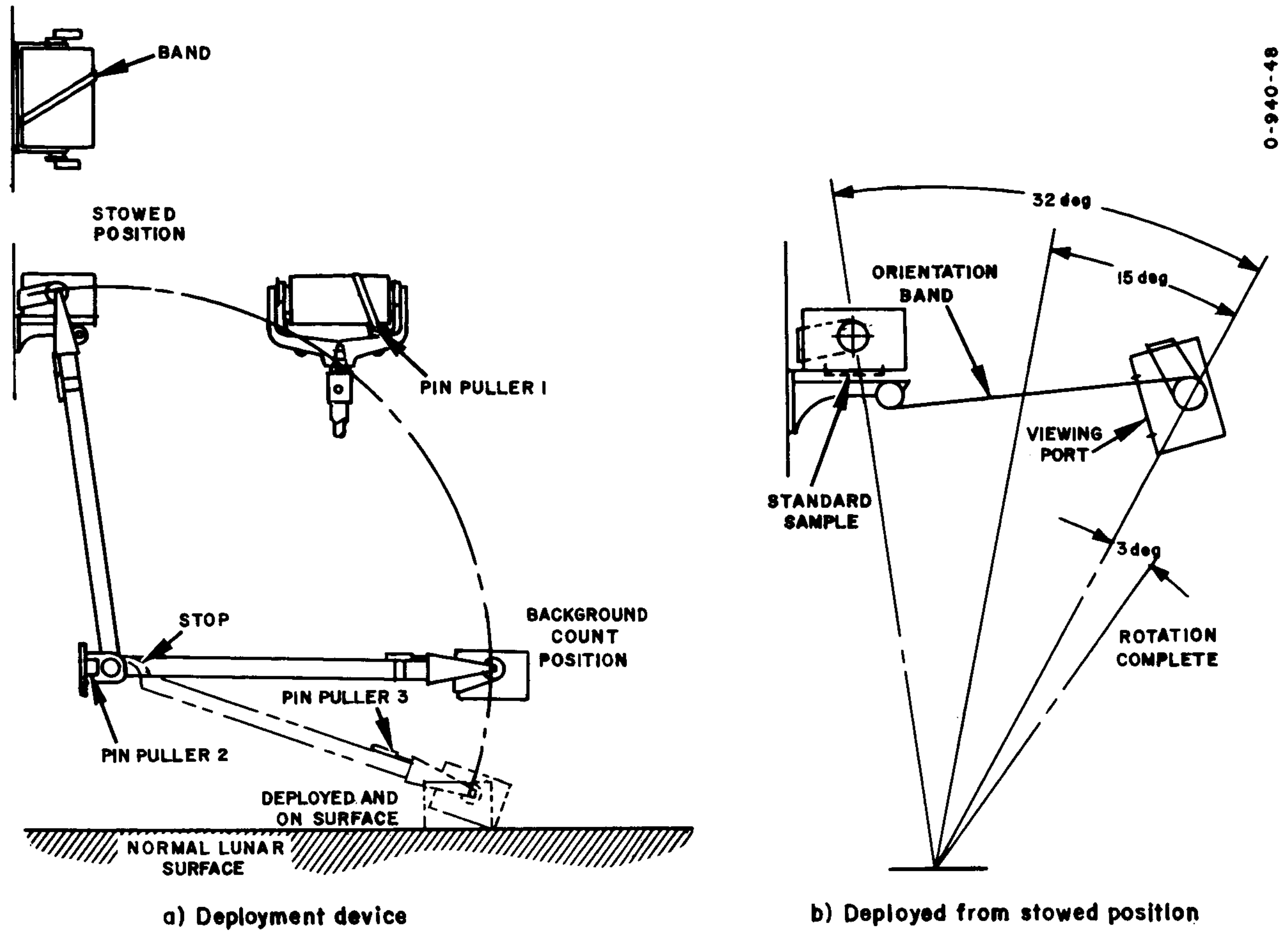


FIGURE 10-9. ALPHA SCATTERING SENSOR DEPLOYMENT MECHANISM

When the sensor reaches the correct position, a detent retains the deployment arm and instrument sensor in position, and the band slips out of its disconnect slot.

Final deployment of the sensor from the background count to the lunar surface position (figure 10-9a) is controlled by two explosive squib actuated pin-pullers. The first pin puller releases the deployment arm background count position detent, allowing arm and sensor to settle to the lunar surface. The second a pin-puller releases the spring loaded yolk which connects the arm to the sensor, allowing the sensor viewing port face to conform to the lunar surface. The only connection between the sensor and the spacecraft is the electrical cable from the sensor.

Each of the three pin pullers utilized in the deployment sequence are energized by switch power controlled by command from earth. The weight of the deployment mechanism and support, less wiring harness, is approximately 5.5 pounds.

The deployment mechanism and sensor are positioned on the spacecraft so that the maximum practicable view for the sensor is achieved with as little interference as possible with the operating envelope of the soil mechanics-surface sampler. The relationship between the soil mechanics-surface sampler and the alpha scattering sensor on the spacecraft can be seen in figure 2-2. The common areas of potential interference between the sensor and the soil mechanics-surface sampler are illustrated in figure 10-10.

MICROMETEORITE DETECTOR EXPERIMENT SUBSYSTEM

The micrometeorite detector experiment provides data on the number, momentum, and kinetic energy of individual lunar ejecta resulting from micro-meteoroids impacting the lunar surface in the vicinity of the Surveyor spacecraft.

Micrometeorite Detector Characteristics

The micrometeorite detector experiment subsystem consists of the sensor and an instrument electronics unit both contained in one package, an instrument auxiliary unit, and associated interconnecting cables. The experiment subsystem block diagram (figure 10-11) shows functionally how the sensor and the instrument electronics unit are tied together.

Sensor Description

The sensor contains three detectors bonded to an impact plate with detectors consisting of a microphone and two thin film capacitors bonded to each side of the common impact plate. The signal from the microphone, a 100 KC crystal, will be related to the momentum of a particle striking the plate, while the capacitor signals indicate gross trajectory and kinetic energy of the particle. Signals from the three detectors are amplified in the sensor unit and then are applied to the instrument electronics unit for digital processing.

A temperature sensor, located in the sensor unit, monitors unit temperature. It receives excitation voltage from a source within the ESP and its output of 0 to 5 volts is processed through the Mode 4 commutator.

A proportionally controlled heater, located in the sensor unit, provides the heating required if the experiment is operated near the day-night terminator. The

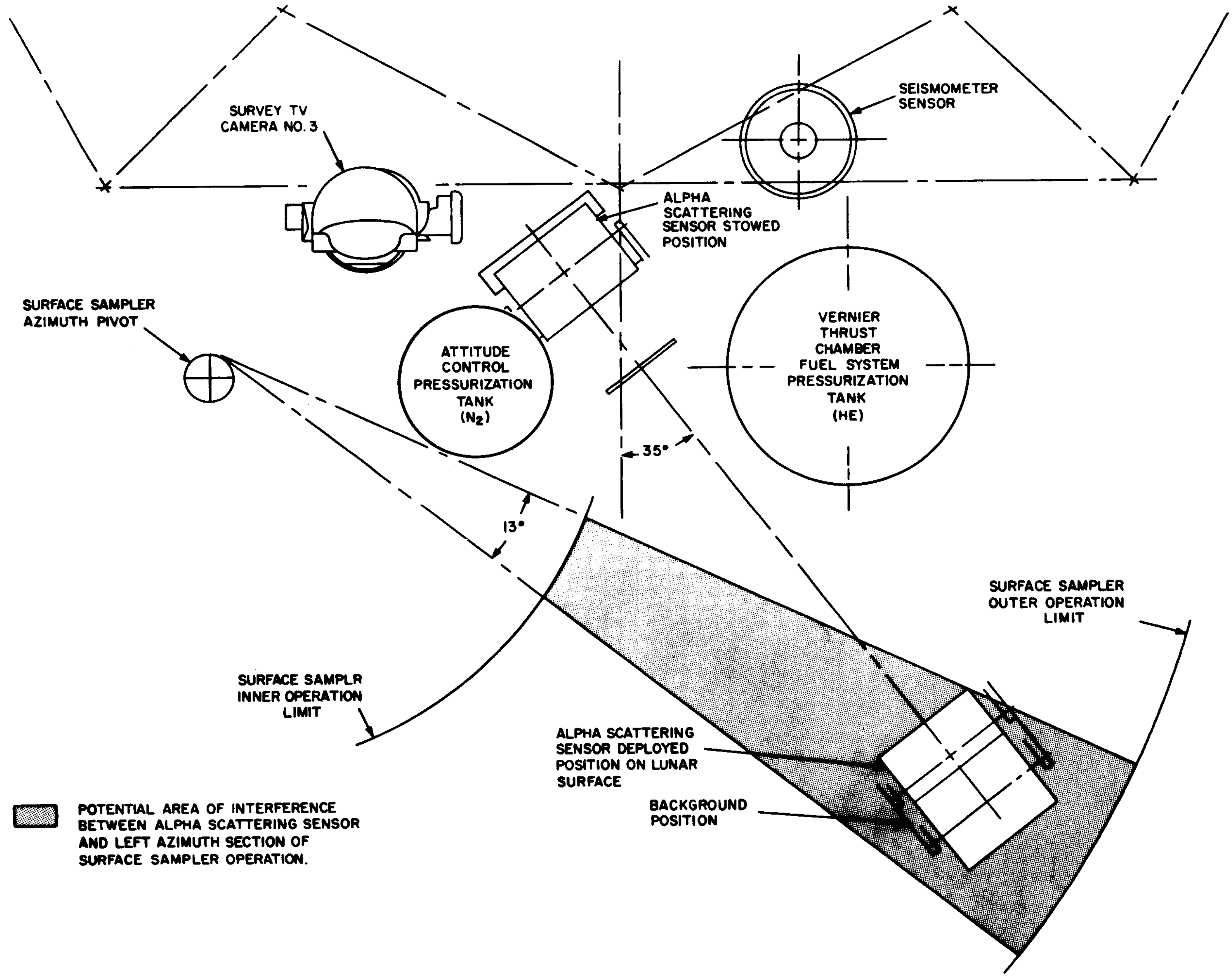


FIGURE 10-10. POTENTIAL AZIMUTH MECHANICAL INTERFERENCE BETWEEN ALPHA SCATTERING AND SURFACE SAMPLING EXPERIMENTS

sensor is designed to survive the lunar night without heating. The heater receives power directly from the spacecraft 22 vdc unregulated supply and is turned on or off by a switch located in the instrument auxiliary unit. The sensor includes a buffer amplifier and calibration transducer as part of a calibration circuit. Pulses from the instrument electronics unit are routed through the buffer amplifier on the transducer to physically shock the plate. The impact is picked up by the microphone for calibration of the capacitor sensors and of the acoustical transducer. The sensor assembly can be calibrated at any time during operation upon command.

Figure 10-12 shows the sensor and its basic field of view. As mounted on the spacecraft, the field of view of the sensor is reduced by approximately 10 percent on one side by compartment A and the mast support structure. The field of view from the other side is reduced by approximately 6 percent because of the shadowing from TV camera 2 and omnidirectional antenna B.

Instrument Electronics Unit

The instrument electronics unit contains microphone and capacitor film pulse height analysis (PHA) logic circuitry, microphone and capacitor film clearing circuits, detector calibration circuitry, and commutation circuitry as shown in figure 10-11. Energy and momentum signals from the sensor unit are sent to the electronics unit for pulse height analysis. Every impact registered by a detector is counted and stored in accumulation registers. During intervals when no ejecta impacts occur, the commutator does not cycle and a series of ones are fed continuously to the telecommunications subsystem.

The output signal in this case will resemble a 100-cps square wave. When a particle hit occurs, the commutator will cycle one time and read out the data from the pulse height analyses (PHA) circuits and from the registers. As the commutator cycles, it will generate a 001 to identify the start of a data word. Three bits are used for readout of the pulse height analyzed momentum signal from the acoustical transducer. Four bits are used for readout of the pulse height analyzed kinetic energy signal from the capacitors. Two bits are used to identify the side on which the impact occurred. Three bits are used to read out the momentum accumulation and four bits are used to read out the energy accumulation. Although the commutator will cycle once when an impact occurs, it may also be cycled by a command from the earth. The commutator output is routed

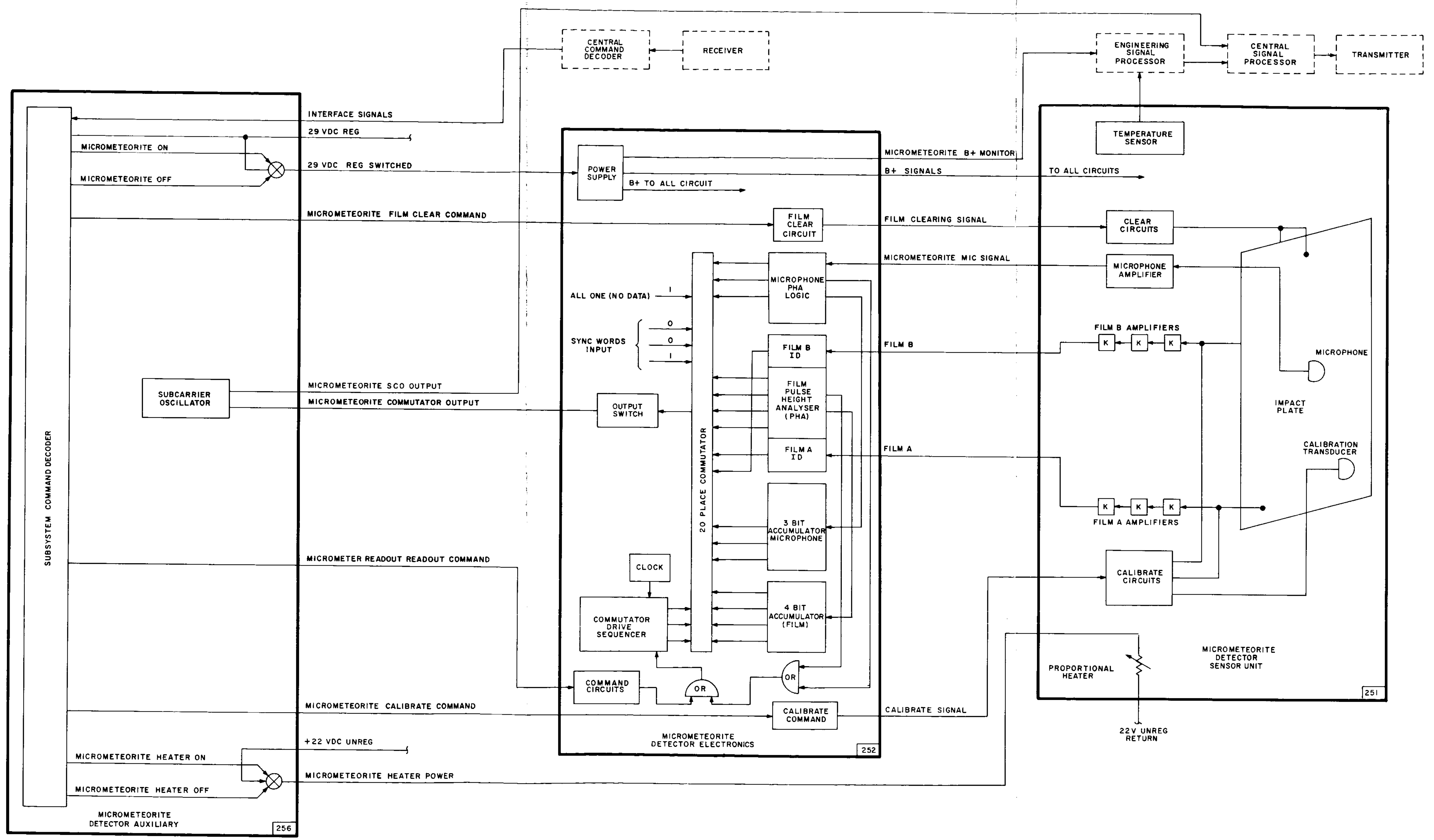


FIGURE 10-11. MICROMETEORITE DETECTOR EXPERIMENT SUBSYSTEM, FUNCTIONAL BLOCK DIAGRAM

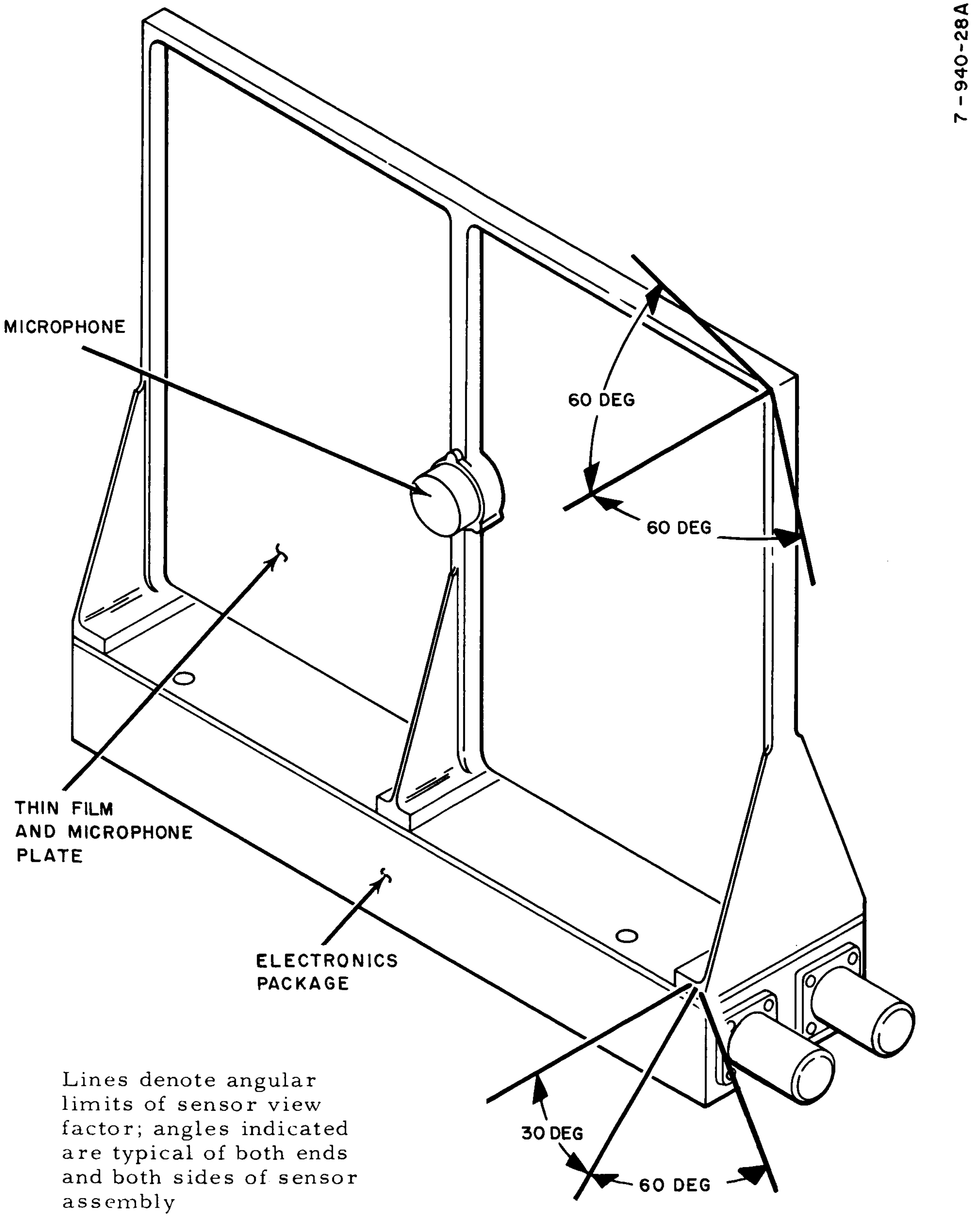


FIGURE 10-12. VIEW FACTOR OF MICROMETEORITE DETECTOR INSTRUMENT ASSEMBLY

to the instrument auxiliary for processing. The output of the conversion unit is monitored through the Mode 4 commutator.

Upon earth command the capacitor clear circuits in the instrument electronics unit will generate sufficient power to electrically burn out any shorts in the thin film capacitors.

An electronic conversion unit (ECU) in the instrument electronics unit generates B+ power for circuit elements in both the sensor and in the instrument electronics unit. This conversion unit receives its power from the spacecraft 29 vdc regulated supply through a switch in the instrument auxiliary.

Instrument Auxiliary Unit

The instrument auxiliary unit for the micrometeorite detection experiment, located in compartment B, provides the electrical interface between the instrument and the Surveyor spacecraft. The instrument auxiliary unit has the same physical appearance as the seismometer instrument auxiliary unit shown in figure 10-15. This auxiliary unit performs command decoding, signal processing, and power management functions for the experiment. The electronic components that accomplish these functions include a subsystem command decoder, two power switches, and an SCO.

The subsystem command decoder receives signals from the central command decoder, decodes all commands addressed to the micrometeorite detector experiment subsystem and furnishes pulse output commands to the instrument electronics unit and the sensor.

Signal processing is accomplished in the instrument auxiliary unit by modulating a subcarrier oscillator with the micrometeorite commutator output from the instrument electronics unit. The subcarrier oscillator output is applied to summing amplifiers in the central signal processor for transmission via the telecommunications subsystem.

Two solid-state switches control the 29 vdc regulated and 22 vdc unregulated power, respectively. One switch controls the power from the spacecraft 29 vdc regulated supply used in the instrument auxiliary unit and in the instrument. The other switch controls the power from the spacecraft 22 vdc unregulated supply used in the sensor heating element.

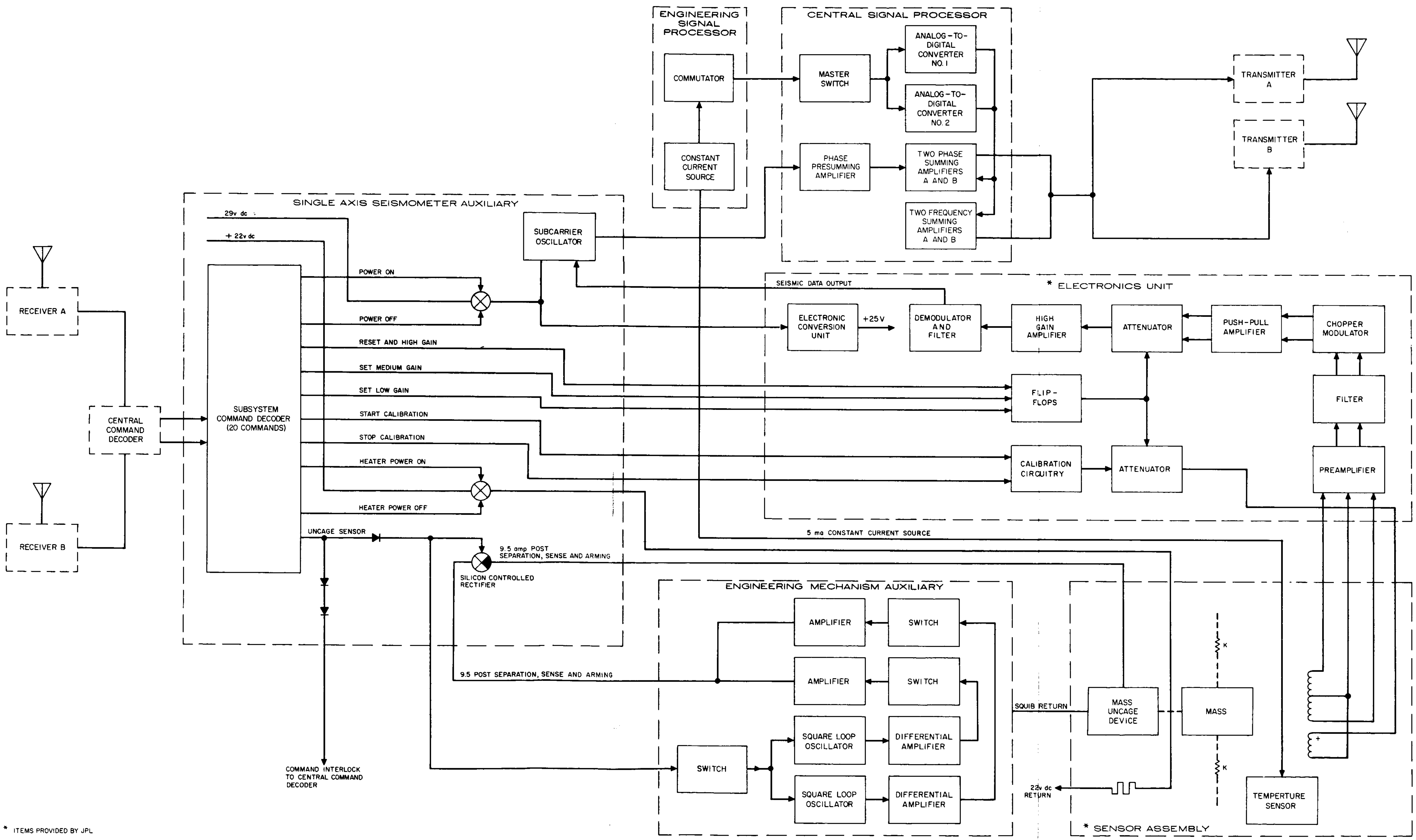


FIGURE 10-14. SEISMOMETER EXPERIMENT SUBSYSTEM, FUNCTIONAL BLOCK DIAGRAM

a single, low frequency coupling between the two. The d-c drift from the preamplifier is not amplified by the high-gain, low-noise amplifier. To utilize the advantages of a balanced arrangement in the preamplifier, the seismometer main coil is center-tapped to provide a push-pull output. The output of the preamplifier is filtered and applied to the modulator which is followed by another stage of push-pull amplification. The modulated and amplified signal is then applied through an attenuator to the following stages of a-c amplification. It is then demodulated and filtered.

The mechanical structure of the sensor is composed of four basic elements. (1) a cylindrical magnet assembly forming an annular gap, (2) a fixed center-tapped cylindrical coil centered within the magnet gap, with several turns of wire wound on the same form but independent of the main coil, for instrument calibration, (3) the elastic suspension consisting of two preformed springs, and (4) a cylindrical housing approximately 4.5 inches in diameter and 4.0 inches high. Thermal control is achieved using super insulation augmented by a special heater. A temperature sensor is included.

To calibrate the instrument, a start calibrate command pulse is routed via calibrate circuitry in the instrument electronics unit to produce a current step in the auxiliary coil of the sensor. The step is subsequently turned off by the stop calibration command pulse after a satisfactory transient at the output is obtained. Since it is desirable to maintain the calibration output pulse at a constant percentage of full-scale reading independent of the gain setting, a step attenuator is used to vary the amplitude of the calibration pulse inversely with the gain setting of the amplifier. Mechanical caging is performed by clamping the seismometer mass against the support frame. Uncaging is performed by actuating a squib device to cut the holding cable.

The sensor is rigidly attached to the spacecraft at a location where maximum mechanical coupling is afforded and where the amplification factor due to spacecraft structure is at a minimum. The location is shown in figure 2-2. The seismometer may operate as much as 15 degrees off vertical depending on the landing attitude of the spacecraft.

The instrument electronics unit, which is located in compartment B, weighs 1.2 ± 0.05 pounds.

Instrument Auxiliary Unit

The instrument auxiliary unit (figure 10-15) provides the electrical interface between the instrument and the Surveyor spacecraft. This unit performs command decoding, signal processing, and power management functions required for the subsystem. The auxiliary consists of a subsystem command decoder capable of providing 20 commands, two electronic power switches, one silicon-controlled rectifier (SCR), and one subcarrier oscillator (SCO).

The principal output of the instrument is the seismic data, in the form of an analog signal of 0 to 5 volts, containing information ranging from 1/20 to 20 cps with a source impedance of about 5000/ohms. This signal directly modulates a subcarrier oscillator within the instrument auxiliary unit. The modulated signal in turn is routed to the central signal processor where further signal conditioning is provided. A temperature sensor whose output is an analog signal of 0 to 5 volts is excited by a 5-milliampere constant current source from the engineering signal processor (ESP). This signal is conditioned within the ESP. Electrical power for the operation and thermal control of the instrument is delivered via the payload harness from the basic bus electrical power subsystem. This electrical power consists of 29 volts dc regulated voltage and 22 volts dc unregulated voltage. Operation of the instrument requires approximately 3 watts of regulated power and 0.5 watt of unregulated power.

The defining documents for the scientific payload are listed in Appendix A, items 46 through 61.

SEISMOMETER EXPERIMENT SUBSYSTEM

The seismometer experiment subsystem measures (1) number, magnitude, and spatial distribution of natural moonquakes, (2) background noise level and spectrum (e.q. seismic background noise correlated, if possible, with thermal or other sources such as, effect of temperature on lunar surface materials) (3) elastic properties and structure both near the lunar surface and at depth; (4) internal constitution - internal damping (Q), density versus depth, temperature versus depth, and type and state of lunar materials versus depth, and (5) distribution of meteorite impacts (number and energy released depends upon the ability to differentiate between impacts and moon quakes).

Seismometer Experiment Subsystem Characteristics

The seismometer experiment subsystem consists of the sensor, instrument auxiliary unit, and instrument electronics unit. Figure 10-13 shows the seismometer instrument, and figure 10-14 is a functional block diagram of the seismometer experiment subsystem.

The short period seismometer consists of a coil mounted within the flux gap of a suspended permanent magnet, which serves as the inertial mass. An auxiliary coil is wound on top of the main coil, for calibration purposes. Motion of the suspended magnet relative to the stationary main coil induces a voltage proportional to the differential velocity. The subsystem has flat response to ground displacement (during quakes) for frequencies above its natural frequency and flat response to ground acceleration (rate of movement) below its natural frequency. Since the velocity transducer output differentiates ground motion, this output is flat to ground velocity above the natural frequency. With an instrument amplifier that is flat over the specified range, the overall experiment response is that of the sensor itself, in the above frequency range.

A high gain, low noise amplifier, located in the instrument electronics unit, increases the very small sensor output power up to the levels required by the telecommunications subsystem. In designing the amplifier it has been assumed that the moon is seismologically much quieter than the earth, and maximum possible gain is required. However, if this is not true, a high gain amplifier could saturate on background noise. To avoid this possibility, two gain changes are available on command, one-tenth and one-hundredth maximum gain. A chopper-amplifier is preceded by a direct-coupled, low noise preamplifier with

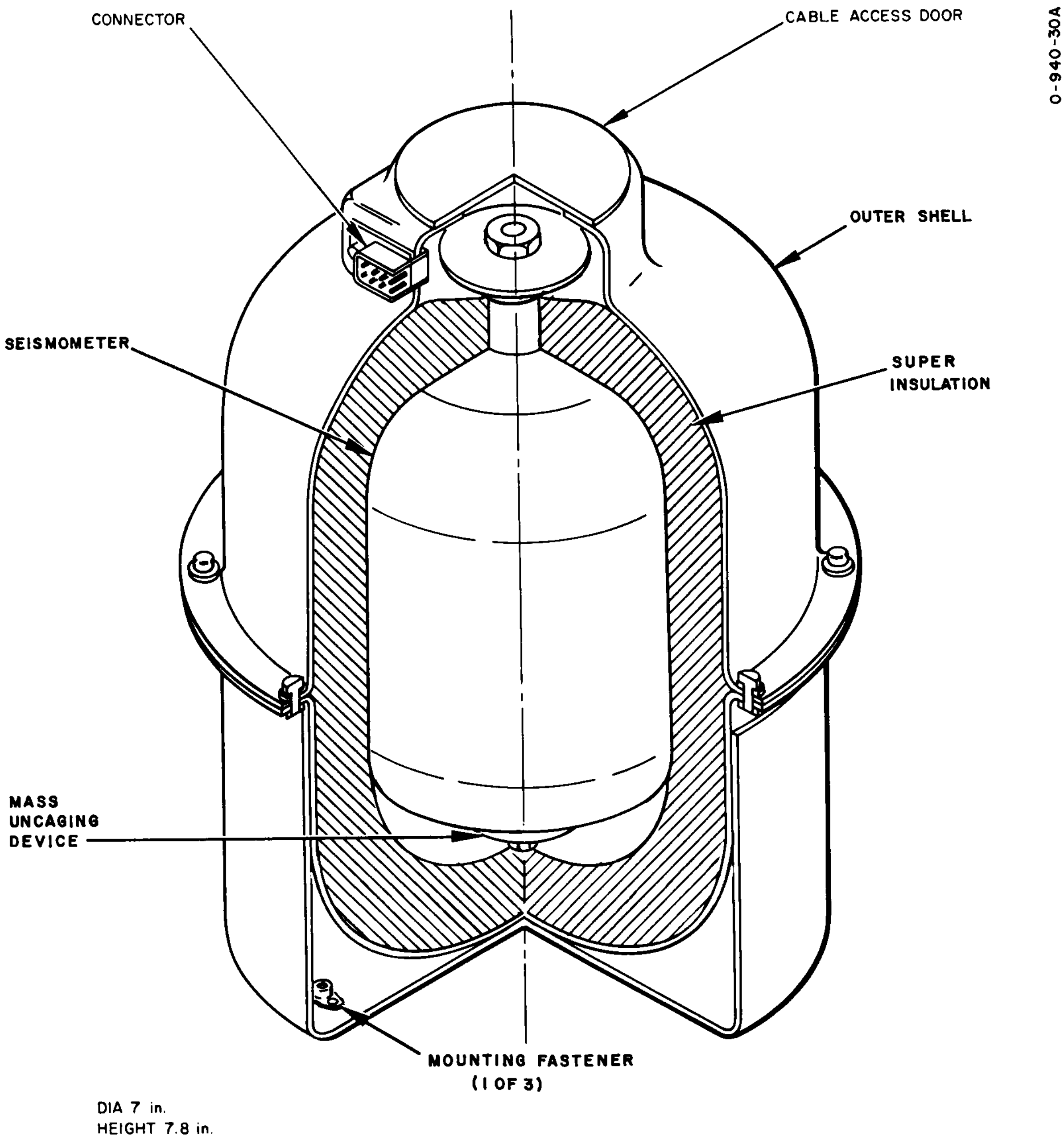


FIGURE 10-13. SEISMOMETER INSTRUMENT

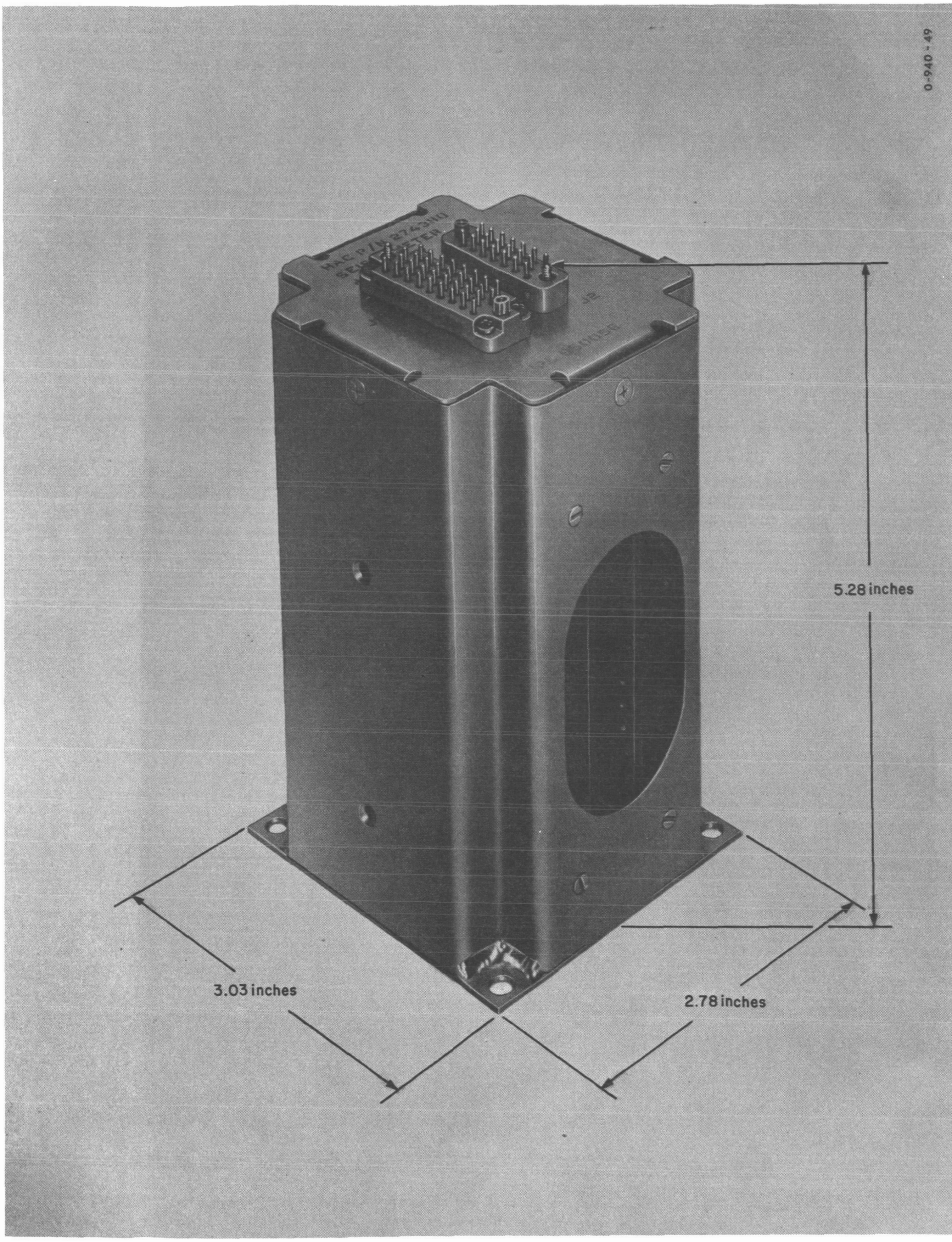


FIGURE 10-15. SEISMOMETER INSTRUMENT AUXILIARY UNIT

XI. SPACECRAFT MASS PROPERTIES

WEIGHT

Total spacecraft weight is limited by the capability of the Atlas/Centaur launch vehicle to a nominal 2150 pounds. A detailed weight breakdown (as of 19 June 1964) is given in tables 11-1 and 11-2. The specific payload for each mission will be selected several months in advance of the flight to provide an optimum payload combination which meets weight and cg constraints. This selection is necessary since the total weight of the spacecraft, including all five scientific experiments shown, may exceed the boost capabilities of the Atlas/Centaur. Those instruments not required for a particular mission will be deleted. Fluctuations in exact details of design result in periodic revision of the figures given in this table; however, the current spacecraft weight status may be obtained by referring to the current Surveyor Weight Status Report.

CENTER OF GRAVITY AND INERTIAS

Spacecraft center-of-gravity, radius-of-gyration, and moment-of-inertia conditions are specified for three major configurations: stowed for launch at liftoff weight; deployed for midcourse and retromaneuvers at separated weight; and deployed for touchdown at landed weight. The limits of the center-of-gravity travel in the X-Y plane are defined by the first two conditions, and the limits of the center-of-gravity travel along the Z-axis are defined by the first and last conditions.

Spacecraft-center-of-gravity limits in the stowed configuration at launch weight are shown in figure 11-1 as established by Atlas/Centaur control and stability capabilities.

Center-of-gravity limits after Surveyor/Centaur separation for midcourse and retro maneuvers are limited by the attitude correction capabilities of the flight control and vernier engine subsystems during retro-rocket burning. The spacecraft center of gravity, before retro-rocket installation, coincides with

TABLE 11-1. SURVEYOR A-21A WEIGHT SUMMARY

Element	Weight* (pounds)
Basic bus	698.81
Usable propellant	1432.08
Scientific payload	90.01
Separated weight	2221.90
Landed weight	624.72
*Based on Payload Combination 1	

TABLE 11-2. SURVEYOR A-21A DETAILED WEIGHT STATUS

Item Description	Current Design Weight (pounds)
Basic Bus	698.81
Flight control system	49.02
Sensor group flight control	35.39
Inertial reference unit	8.13
Canopus sensor	4.92
Wiring harness	0.95
Electronics	17.89
Support	3.50
Sensor secondary solar	0.35
Attitude control system	13.28
Attitude jets	1.62
N_2 tank and control	9.08
Actuator, roll	1.08
Nitrogen	1.50

TABLE 11-2. SURVEYOR A-21A DETAILED WEIGHT STATUS (Cont)

Item Description	Current Design Weight (pounds)
Electronics	103.86
Data link	32.29
Antenna, planar array	8.90
Antenna, omnidirectional A	0.32
Antenna, omnidirectional B	0.32
Rf transfer switch	0.83
Rf spdt switch	0.48
Transmitter A	6.84
Transmitter B	6.84
Command receiver and transponder A	3.88
Command receiver and transponder B	3.88
Central command decoder	5.44
Central signal processor	4.85
Doppler altitude/velocity system	34.18
Signal data converter	9.64
Klystron power supply	10.62
Antenna, altitude/velocity sensor	6.60
Antenna, velocity sensor	6.02
Waveguide assembly	1.30
Boost regulator	6.78
Approach TV camera 4	6.80
Engineering signal processor	5.73
Engineering mechanism auxiliary	3.35

TABLE 11-2. SURVEYOR A-21A DETAILED WEIGHT STATUS (Cont)

Item Description	Current Design Weight (pounds)
Thermal control assembly	0.24
Altitude marking radar	8.43
Insulation, altitude marking radar	0.92
Signal processing auxiliary	0.34
Battery charge regulator	3.30
Low data rate auxiliary	0.60
Electrical Power	54.90
Solar panel	8.50
Battery	46.40
Mechanisms	28.55
Positioner, antenna/solar panel	24.38
Boom etc, omnidirectional antenna A	2.20
Boom etc, omnidirectional antenna B	1.16
Separation sensing and arming	0.81
Spacecraft vehicle	218.46
Spaceframe, basic structure	59.87
Installation hardware	22.54
Thermal paint	1.00
Landing gear installation	38.28
Landing gear 1	11.63
Landing gear 2	11.63
Landing gear 3	11.63
Pin puller, gr rel (3)	0.33
Auxiliary crushable blocks	3.06
Thermal compartment A	25.16

TABLE 11-2. SURVEYOR A-21A DETAILED WEIGHT STATUS (Cont)

Item Description	Current Design Weight (pounds)
Thermal compartment B	18.06
Wire harness, basic bus	44.20
Pneumatic lines	0.74
Release mechanism, main retro	1.50
Engineering measurement sensors	4.45
Latch, spacecraft to Centaur	0.33
Heat collectors	0.78
Spacecraft propulsion	223.16
Propulsion system, vernier	73.42
Valve assembly, helium	3.04
Tank, helium	20.31
Tank, fuel (3)	9.96
Tank, oxidizer (3)	9.96
Thrust chamber assembly (3)	18.44
Thermal control	5.62
Lines and miscellaneous fittings	6.09
Helium	2.50
Propellant, unusable	4.20
Rocket engine, main retro	143.04
Rocket engine, main retro	139.39
Insulation, main retro	3.65
Contingency	8.00
Propellant, usable	1433.08
Vernier propellant	156.50
Retro-Rocket propellant	1236.20

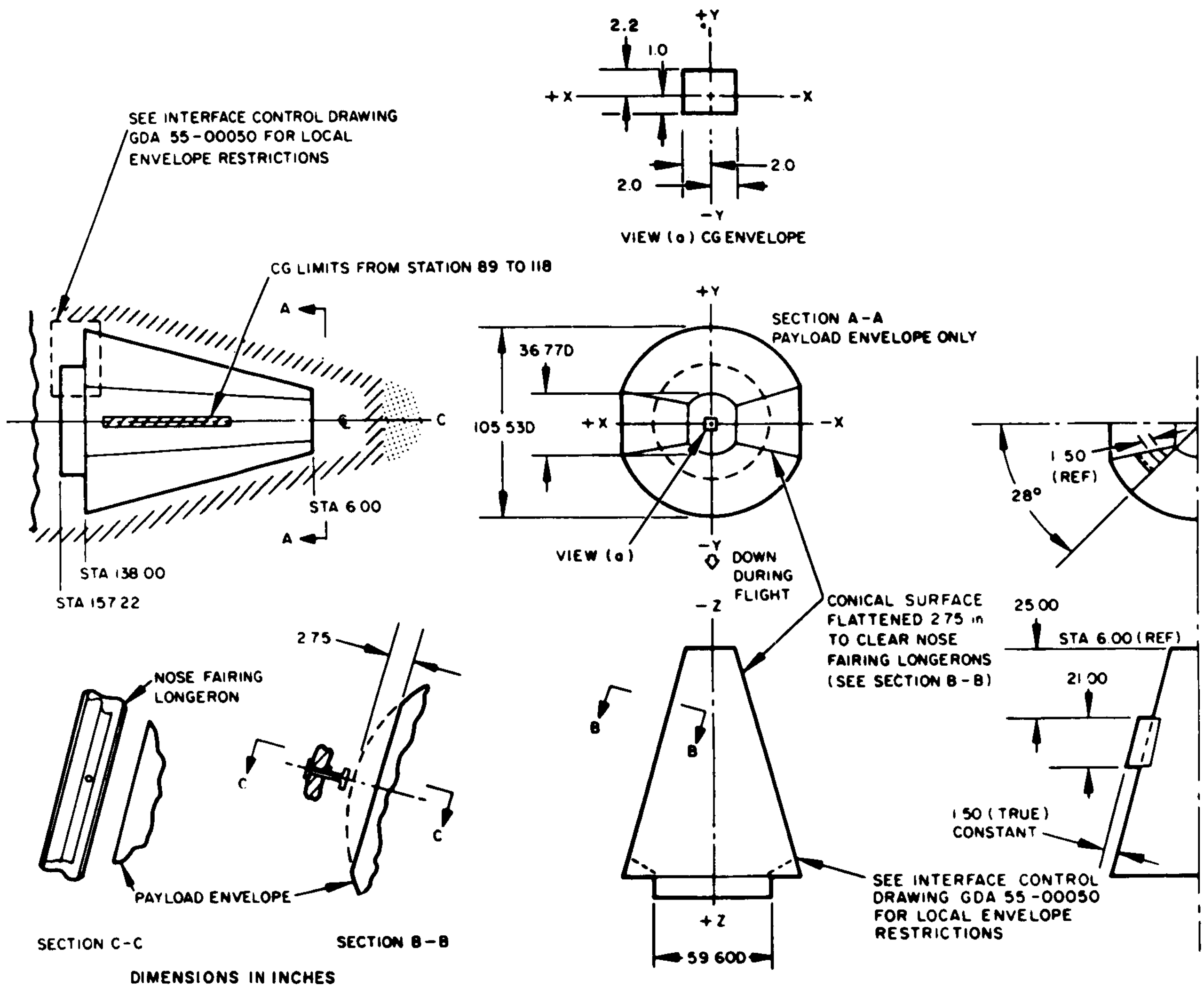


FIGURE 11-1. ATLAS/CENTAUR PAYLOAD ENVELOPE

the Z-axis within 1.0 inch. The spacecraft center of gravity for the separated weight conditions must coincide with the retro-rocket thrust axis within 0.18 inch radial. The position of the thrust axis is adjusted to coincide with the center of gravity at the time of installation to coincide with the spacecraft center of gravity to meet this requirement.

The limits of travel of the vertical center-of-gravity location in the touch-down configuration are designed to the landing site assumptions, described in Section XIII, so the spacecraft will not topple when landing.

EFFECTS OF PAYLOAD COMBINATIONS

The concept of designing the spacecraft for autonomous operation and removal of one or more of the approved scientific instruments has been outlined in the previous sections. The approved list of scientific experiment subsystems for the A-21A series of spacecraft consists of the following:

Survey television experiment subsystem (which can include either one or both of two television cameras)

Soil mechanics-surface sampler experiment subsystem

Alpha scattering experiment subsystem

Micrometeorite detector experiment subsystem

Seismometer experiment subsystem

The television experiment can be installed in any of three different combinations (i.e., TV camera 3 only, TV camera 2 only, or both TV cameras). When it is added to the other four experiments the total could be considered potentially as seven different possible experiment combinations. However, because of the relative locations of the television cameras on the spaceframe, the use of TV camera 2 by itself is not considered practical. Its capability to view the other scientific instruments is inferior to that of TV camera 3. Furthermore, its location is not as advantageous for center-of-gravity adjustment purposes as that of TV camera 3. Accordingly, the possible television combinations are restricted to 1) TV camera 3 only, or (2) both TV cameras 3 and 2.

The scheme employed in analyzing weight and balance effects consists of starting with a full complement of scientific experiments and then removing them in various combinations, to achieve particular configurations. In some cases removal of one or more experiment subsystems requires that ballast be added to either leg No. 2 or No. 3, or both, to keep the spacecraft lateral center of gravity within acceptable limits. Ballast is added at the footpad in increments of one pound up to a maximum weight of five pounds per leg. Use of ballast in excess of this figure potentially requires redesign of the leg and extension mechanism.

In some instances proper cg control cannot be achieved by adding 5 pounds at one or more footpads. In these cases it would be necessary to add supplemental ballast on or near the main spaceframe structure to achieve proper cg position.

An itemized weight summary for each scientific payload experiment is given in Table 11-3. Table 11-4 illustrates 13 different payload combinations considered. (Those combinations previously studied but which could not be brought into proper balance by adding footpad ballast have not been shown.)

When the total dry landed weight of the spacecraft drops below 586 pounds, it will be necessary to add supplemental ballast (which can be evenly distributed around the geometric center of the spaceframe) to enable final descent to be made consistent with the minimum vernier engine thrust level of 30 pounds per chamber.

As various payload combinations are selected for flight it will be necessary to provide the vernier fuel and oxidizer necessary to accommodate the total spacecraft/payload weight including ballast (if any). The total amount of main retro propellant loaded must also be adjusted for total spacecraft weight.

SPACECRAFT COORDINATE SYSTEM

Spacecraft components are located by a right hand Cartesian coordinate system whose axes are denoted X, Y, and Z. The Z coordinate axis (figure 2-2) is the height or vertical length of the spacecraft. A negative movement along the Z axis is toward the top of the spacecraft, the X-Y plane is perpendicular to the Z axis, the positive Y coordinate axis is along the center line of spacecraft leg No. 1. The X axis is perpendicular to the Y axis, with the positive direction taken so as to make the coordinate system right handed.

The origin lies in the X-Y plane at the geometric center of a triangle formed by three tooling ball holes. The X-Y plane is approximately the attachment level between the Surveyor vehicle and the Surveyor-Centaur adapter.

Since the coordinate system origin is no physical point on the spacecraft, tooling holes on the legs are used to locate the origin. The origin in the X-Y plane is 23.75 inches from the tooling holes.

Pitch, yaw and roll axes are defined as the spacecraft X, Y and Z axes respectively, the origin being as defined above. The location of the center of gravity is determined, but may change with changes in component location and weight changes. Tolerances on the center of gravity are within a one inch radius from the origin in the X-Y plane, and Z = -16 to -19.5 inches (above the X-Y plane).

TABLE 11-3. ITEMIZED WEIGHT SUMMARY FOR EACH SCIENTIFIC PAYLOAD EXPERIMENT

(All Weights Shown in Pounds)

Experiment Nomenclature	Primary Instrument or Sensor	Instrument (Sensor) Mounting Provisions	Instrument Electronics (if any)	Spacecraft Electronic Auxiliary	Removable Wiring Harness	Miscellaneous	Experiment Subsystem Weight Totals	Remarks
Television camera 2	16.13 ± 0.20	1.70 ± 0.05	Included in camera	1.74 + 0.22[a] - 0.11	1.90 ± 0.10	Charts[a] 0.16 ± 0.01	21.61 + 0.32 -0.25	a) When TV cameras 2 and 3 are installed, only one TV auxiliary and one set of test charts is required.
Television camera 3	16.13 ± 0.20	1.50 ± 0.05	Included in camera	1.72 + 0.22[a] - 0.11	1.90 ± 0.10	0.16 ± 0.01[a]	21.43 + 0.32 - 0.25	
Surface sampler	8.80 ± 0.10	0.83 ± 0.05	b	2.80 ± 0.10	1.80 ± 0.45		14.23 ± 0.47	b) Weight of accelerometer and strain gauges included in sampler.
Micrometeorite detector	2.00 ± 0.10	0.50 ± 0.02	3.00 ± 0.15	0.90 ± 0.05	1.50 ± 0.37		7.90 ± 0.42	c) Includes weight of circuitry in sensor assembly.
Alpha scattering	3.20 ± 0.20	Deployment mechanism and support 4.80 ± 0.10	5.00 ± 0.20	1.15 ± 0.05	2.25 ± 0.66		16.40 ± 0.73	
Seismometer	5.80 ± 0.20	0.10 ± 0.02	1.20 ± 0.05	1.00 ± 0.05	0.85 ± 0.22		8.95 ± 0.37	
Nonremovable harness and provisions						1.37 ± 0.34	1.37 ± 0.34	
Category totals	52.06	19.45	9.20	7.57[d]	10.20	1.53[d]	90.01 + 1.16[d] - 1.14	d) These totals include weight of one set of charts and one TV auxiliary only.

TABLE 11-4. PAYLOAD COMBINATIONS SUMMARY

Payload Combination Number	Experiment Combinations: TV Camera 3	Both TV Cameras	Surface Sampler	Micro-meteorite Detector	Alpha Scattering	Seismom-eter	Spacecraft Design Results: Payload Weight, pounds	Resulting Center of Gravity, inches: X	Y	Z	Ballast Weight (if any), pounds: Leg 2	Leg 3	Total Usable Propellant Weight, pounds	Spacecraft Gross Weight, pounds	Dry Landed Weight, pounds	Propellant Weight - lb: Vernier	Retro
1		X	X	X	X	X	90.01	0.10	0.23	0.25			1433.08	2221.90	624.72	151.38	1281.7
2		X	X		X	X	82.11	0.20	0.19	0.28			1416.84	2197.76	616.82	149.74	1267.1
3		X	X	X	X		81.06	0.26	0.35	0.44			1414.94	2194.81	615.77	149.54	1265.4
4		X	X	X		X	73.61	0.22	0.49	0.54			1399.50	2171.92	608.32	148.00	1251.5
5		X	X		X		73.16	0.37	0.30	0.48			1398.40	2170.37	607.87	147.90	1250.5
6		X	X	X			64.66	0.39	0.62	0.73			1381.26	2144.73	599.37	146.26	1235.0
7	X		X	X	X		61.33	0.73	0.36	0.81		2.50	1379.48	2142.12	598.54	147.47	1246.7
8	X		X	X	X	X	70.28	0.72	0.34	0.80		0.70	1394.17	2163.96	605.69	145.98	1233.5
9	X			X	X	X	56.05	0.47	0.67	0.81		2.60	1368.92	2126.38	593.36	144.92	1224.0
10*		X	X				55.39	0.54	0.55	0.77			1362.26	2116.46	590.10	144.26	1218.0
11	X		X	X		X	53.88	0.69	0.42	0.81		4.20	1367.80	2124.69	592.79	144.80	1223.0
12	X		X		X		53.43	0.78	0.23	0.81		4.80	1368.15	2125.19	592.94	144.85	1223.3
13**	X				X	X	48.15	0.54	0.64	0.84		3.00	1353.71	2103.67	585.86	143.41	1210.3

*Includes 1.64 pounds ballast, evenly distributed about the geometric center, to obtain minimum dry-landed weight of 586 pounds.

**Includes 6.00 pounds ballast, evenly distributed about the geometric center, to obtain minimum dry-landed weight of 586 pounds.

Note: Weight and balance assumptions are:

Nitrogen tank located at x = 0.00
y = -35.50

Signal data converter located at x = 1.00
y = -23.00

Weights are as of 19 June 1964 weight report

Locations and weights of scientific payload primary instrument or sensor are:

	X	Y	Z	Weight (pounds)
Survey TV camera 2	-33.20	-6.50	82.90	15.96
Survey TV camera 3	15.00	-25.96	79.85	15.96
Surface Sampler	23.00	-36.20	51.00	8.80
Micrometeorite detector	7.50	-16.00	86.34	2.00
Seismometer	-10.56	-21.00	52.16	5.80
Alpha scattering	10.50	-34.00	70.87	3.20

XII. SPACECRAFT THERMAL CONTROL

To obtain maximum reliability with minimum complexity, passive and semipassive thermal control techniques have ben used wherever practicable. This design was dictated by the requirement to achieve, with a minimum weight penalty, adequate thermal control over the spacecraft for relatively long periods of time while operating on a lunar surface having an exceptionally wide range of high and low temperatures. Although passive thermal control provides a relatively simple and lightweight solution to real-time operational thermal problems, it required the development and verification of special insulation techniques and surface treatments having unique thermal properties. Semipassive thermal control is achieved through the use of self-actuating mechanical thermal switches which control temperature by varying thermal conductivity. Through utilization of these control techniques, the active electrical heating requirements and thermal control power consumption are held to a minimum.

METHODS OF THERMAL CONTROL

The Surveyor thermal control system consists of the individual subsystem or unit thermal control systems, integrated to provide acceptable thermal environments for all components during all phases of spacecraft operation. The passive methods employed include (1) special preparation of spacecraft external surfaces to achieve optimum thermal absorption and emissive characteristics, and (2) superinsulated compartments to house critical equipment. The active methods employed are (1) electrical heaters, (2) thermal conduction paths controlled by bimetallically operated thermal switches to maintain the compartment temperatures within acceptable limits.

The spacecraft thermal control design is based on the following assumptions:

a. The spacecraft will remain with the positive-thrust (Z) axis pointed toward the sun except during the following periods:

(1) Before initial sun acquisition.

(2) An eclipse period of 1 hour maximum (full shadow).

(3) Midcourse correction maneuver.

(4) Terminal descent maneuver.

b. For the normal transit attitude, the positive thrust axis of the spacecraft will be pointed at the sun.

c. Solar intensity is 442 Btu/hr-ft^2 nominal (same as during transit).

d. Maximum temperature of the illuminated lunar surface is +260°F at a solar angle of 0 degrees from zenith. The surface temperature is assumed to vary as a function of solar angle in accordance with figure 12-1.

e. Radiative properties of lunar surface:

Infrared emissivity	ϵ =	0.875
Solar absorptivity	α =	0.93

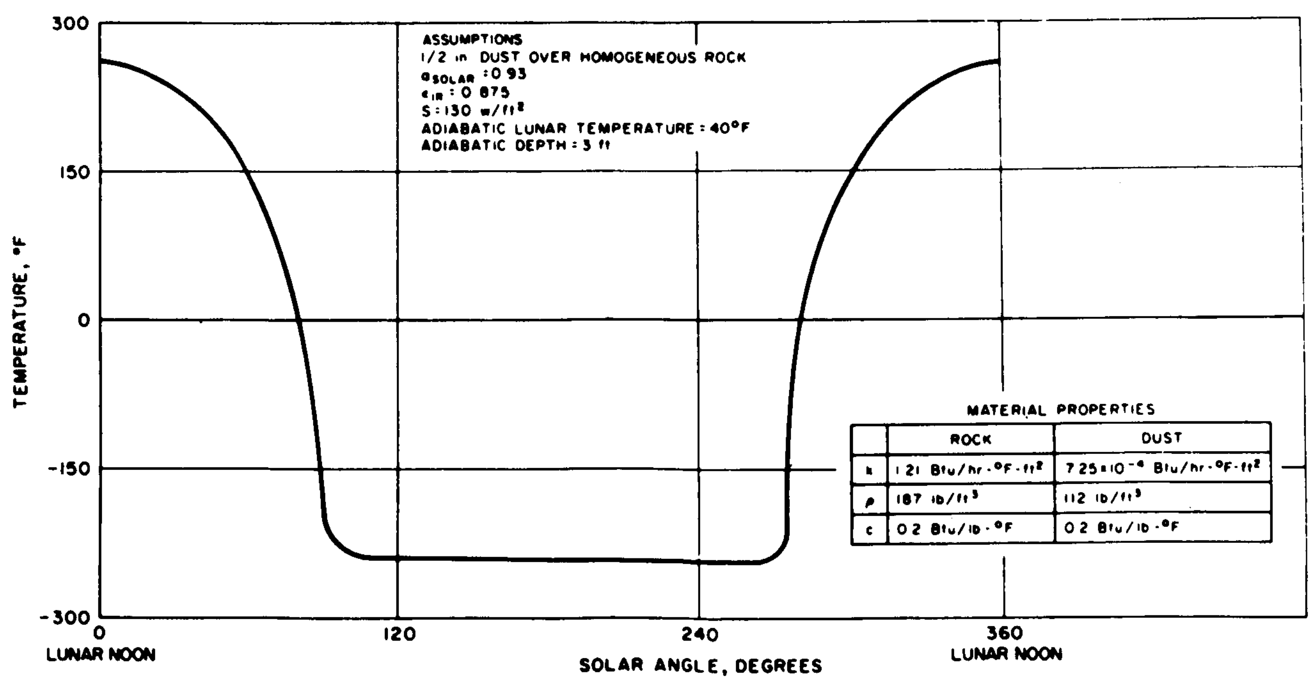


FIGURE 12-1. LUNAR SURFACE TEMPERATURE

f. Minimum temperature of lunar surface at night is -245°F at the equator and -260°F at 65 degrees latitude.

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XIII. OPERATIONAL SEQUENCE AND FUNCTIONAL PERFORMANCE

The expected performance of the spacecraft during a standard operational mission is described in the following paragraphs. Emphasis is placed on spacecraft performance based on design capabilities and their relation to the mission operational phases. The three major phases of a typical mission are prelaunch, transit, and lunar.

PRELAUNCH PHASE

Air Force Eastern Test Range (AFETR)

The AFETR imposes various constraints on the spacecraft, primarily to ensure that adequate safety measures are provided to protect personnel. The "General Range Safety Plan," AFMTCP 80-2, provides the authority for such constraints as are reflected in the spacecraft design. Typical constraints affecting spacecraft design are summarized as follows:

a. Squibs must be capable of withstanding an energy of 1 watt at 1 ampere for 5 minutes without firing.

b. An explosive destruct system, capable of actuation by radio command, must be provided for destruction of solid propellant (provided by GD/A).

c. Pressure vessels (i.e., nitrogen and helium tanks) must meet minimum proof test requirements.

d. A safe and arming mechanism to ensure against inadvertent retro-rocket firing must be provided.

Other AFETR restrictions such as launch azimuth conditions do not influence the spacecraft design as directly as those listed above.

Prelaunch Operations at AFETR

The final prelaunch countdown check on the Surveyor spacecraft itself (other than a few critical power and safety circuits that are hard-line wired to meet

AFETR requirements) is accomplished by means of an r-f link. Inherent in this checkout is the operational verification of the spacecraft transmitter, receiver, command decoding, and signal processing. During this countdown, the spacecraft pulse code modulation (PCM) data channels and Centaur telemetry can be monitored to provide the initial-condition values for these channels. Checks of the television and spacecraft transponder operation, as well as the ability of the spacecraft power system to properly supply the electrical load (i.e., with external power removed) can also be made.

The success of the mission is dependent on the establishment of definite initial conditions for some of the spacecraft subsystems. At the time of launch, the spacecraft battery must be fully charged, the marking range of the altitude marking radar selected, the Canopus sensor field of view adjusted properly for the particular launch data, the gyro temperatures within their operational range, and the temperatures of some spacecraft items such as the retro-rocket, vernier propellant, shock absorbers, compartments A and B, flight control electronics, and Canopus sensor at their prescribed launch values.

Once the spacecraft is launched, external control of the spacecraft is not available until the initial DSIF acquisition is accomplished. Therefore, spacecraft operational conditions required during this interval must be established during the final countdown. These conditions are as follows:

a. Flight control coast phase electronics must be on, and the spacecraft flight control system must be in the rate-stabilized mode so that the angular rates imparted to the spacecraft as a result of the separation from Centaur can be nulled.

b. The nitrogen must be inhibited from flowing to the jets so that it will be conserved during countdown and boost, until Centaur separation.

c. The accelerometer amplifiers must be on to permit accelerometer data to be transmitted via the telemetry channels provided by the Centaur.

d. One transmitter and its associated receiver/transponder interconnection must be on at the time of launch to permit and to facilitate the initial DSIF acquisition.

e. The transmitter traveling-wave tube (TWT) filaments must be on so that the transmitter will be ready for high power operation when the preseparation command for spacecraft transmitter high power turnon is received from the Centaur.

f. The spacecraft signal processor must be operating in the proper mode to provide the one channel of PCM data via the Centaur and the spacecraft telemetry links during the period between boost and the completion of the initial DSIF acquisition.

TRANSIT PHASE

The transit phase of the Surveyor mission includes the following events:

a. Launch through separation.

b. DSIF acquisition.

c. Sun acquisition.

d. Canopus acquisition and verification.

e. Coast phase 1.

f. Midcourse correction maneuver.

g. Coast phase 2.

h. Pre-terminal descent maneuver.

i. Terminal descent.

   1. Main retro descent.

   2. Vernier descent.

   3. Touchdown.

Launch thru Separation

Provisions are included on the spacecraft to permit the telemetering of PCM and accelerometer data via the Centaur telemetry system during this phase of the mission. This same PCM data will be simultaneously telemetered via the spacecraft telemetry system so that it will also be available following the separation of the spacecraft from the Centaur. The accelerometer data will indicate the boost vibration experienced by the spacecraft.

For the spacecraft to perform properly after it separates from the Centaur, certain operations must be accomplished before separation. Signals that will cause these operations to occur are to be provided by the Centaur. Required operations are described in the following paragraphs.

The landing gear and omnidirectional antennas are launched in their stowed (i.e., nonextended) positions so that the spacecraft can fit within the envelope of the shroud. The landing gear must be extended if the cold gas attitude control system is to operate properly since the attitude jets are installed near the ends of the legs and their moment control capability depends on the legs being extended. Consequently, gas jet actuation is inhibited until the legs are extended. The omnidirectional antennas must also be extended to provide the desired radiation pattern coverage for accomplishing the initial DSIF acquisition; the spacecraft high-power transmitter must be turned on to ensure that a signal of sufficient amplitude will be available for accomplishing the initial DSIF acquisition; and the solar panel must be aligned so that it can convert solar energy into electrical power.

Although it is possible to command these functions from the ground in the event that the Centaur fails to deliver the signals required by the spacecraft, the ground-to-spacecraft communication link must be established before this can be done.

The reduction of possible separation-induced angular rotation is accomplished by the coast phase attitude control system operating in a rate-stabilized mode. The coast phase attitude control system controls the spacecraft attitude by operating the three pairs of nitrogen gas jets located on the ends of the landing legs, and as indicated in table 13-1, it is the system used for attitude control throughout transit except during the midcourse and terminal descent thrusting phases. In the rate-stabilized mode, the system closes an electrical feedback loop around each of the gyros so that the spacecraft rotational rates about each of the spacecraft axes are sensed and reduced to approximately zero. To keep the gyro gimbals from hitting their stops and to limit the amount of nitrogen used to stabilize the vehicle (approximately 1 percent of the total amount carried), the rotational rates induced by separation are required to be less than 3 deg/sec.

The spacecraft is mechanized so that two signals automatically generated during the preseparation and separation sequences enable the gas jet system. These signals are (1) a signal produced by all three landing legs extending in response to the Centaur command described previously, and (2) a signal generated by the separation of the spacecraft from the Centaur. As a backup,

TABLE 13-1. SUMMARY OF OPERATIONAL MODES PROVIDED BY COAST PHASE ATTITUDE CONTROL SYSTEM

Operational Mode	Mechanization Description	Phase of Mission When Utilized
Rate-stabilized mode	Closed electrical loop provided around gyros so that spacecraft angular rates are sensed and reduced to approximately zero (within the system dead-band).	Immediately after separation of spacecraft from Centaur
Inertial mode	System operates so that changes in angular position of spacecraft in space are sensed. The attitude of the spacecraft is controlled so that it remains fixed in space.	During midcourse correction, pre-retro phase, retro burning phase, during entire vernier descent for roll, and during the constant velocity portion for pitch and yaw
Maneuvers	Mechanization is same as inertial mode, except that a fixed current is applied to the gyro torquer for the time commanded from the ground. This current results in a nominal spacecraft rate of 0.5 deg/sec for the commanded time, resulting in a given angular movement.	Before midcourse and terminal descent
Automatic sun acquisition	Spacecraft maneuvered at 0.5 deg/sec in yaw and pitch in response to secondary sun sensor logic until sun appears in primary sensor field of view.	Sun acquisition

TABLE 13-1. SUMMARY OF OPERATIONAL MODES PROVIDED BY COAST PHASE ATTITUDE CONTROL SYSTEM (Cont)

Operational Mode	Mechanization Description	Phase of Mission When Utilized
Automatic star acquisition	Spacecraft maneuvered at 0.5 deg/sec in roll until Canopus appears within field of view of Canopus sensor.	Canopus acquisition
Optical (or celestial) reference	Spacecraft pitch and yaw attitude controlled by error signals provided by primary sun sensor, and spacecraft roll attitude controlled by error signal provided by the Canopus sensor.	Coast phases

commands which either permit the gas jet system to be enabled or permit the system to be shut off at any time during transit are provided to accommodate nonstandard situations.

Automatic solar panel deployment is initiated by two squib-firing pulses. These pulses, generated when the spacecraft senses separation from the Centaur, initiate the solar panel launch-lock pin pullers and the roll axis launch-lock pin puller. The unlocking of the solar panel axis is sensed by a limit switch that initiates a pulse generator, which in turn supplies a continuous train of pulses to the solar panel stepping motor. When the solar panel relock position is reached, the pulse generator output is transferred to the roll axis stepping motor through the use of a solar panel relock limit switch. When the roll axis relock position is reached, it is sensed by a third limit switch and the pulse generator is disabled. After DSIF acquisition, power to the automatic deployment logic is commanded off.

DSIF Acquisition

The communication link between the spacecraft and the ground must be established and maintained by the DSIF stations to permit the spacecraft to be controlled and tracked during the mission. As the first step in this sequence, the DSIF must locate the spacecraft transmitter signal and tune the ground receiver to lock on to this signal. If the boost and separation phases have been standard, the spacecraft will come within view of the tracking station with one of its transmitters operating in the high-power mode, frequency controlled by its narrow band voltage-controlled oscillator, and transmitting one channel of engineering data over one of the two spacecraft omni antennas. The spacecraft receiver can phase-lock to the ground transmitter signal during the second part of the acquisition procedure in which the ground transmitter is tuned until its signal appears within the pull-in range of the spacecraft receiver. This will result in a shift in the spacecraft transmitter frequency, since with the achievement of transponder operation, the spacecraft transmitter frequency will be controlled by the spacecraft receiver frequency, which in turn will be controlled by the ground transmitter.

With the spacecraft transmitter operating in the high-power mode, the r-f power appearing at the antenna feed will typically be a minimum of 2.7 watts. The transmitter frequency is expected to change from the time of launch to the time of acquisition, primarily because of the temperature change which the transmitter experiences during this period, so that the spacecraft transmitter frequency will be uncertain at the time of acquisition. The frequency stability of the transmitter is expected to be within ±20 parts per million (ppm) with a temperature coefficient of less than 0.5 ppm/°F.

The two circularly polarized omnidirectional antennas mounted on the spacecraft are used simultaneously for reception but only one-at-a-time for transmission. These antennas do not have the same level of gain throughout the entire 4π steradians, primarily because of spacecraft shadowing. They are installed on the spacecraft so that the nulls in the pattern of one antenna tend to cover the peaks of the other. At the time of acquisition, since the transmitter will be transmitting over only one antenna and cannot be commanded to transmit over the other antenna until acquisition is achieved, the coverage provided by one antenna alone is pertinent to spacecraft transmission. Also, since both receivers will be operating continuously throughout the mission, the coverage provided by both antennas is of concern for spacecraft reception.

The composite coverage of the two antennas used for receiving will typically provide a gain of at least -10 db for single polarization over 99 percent of the possible aspect angles, a gain of at least -6 db for dual polarization over the upper hemisphere of the spacecraft, and a gain of at least -7 db for dual polarization over the lower hemisphere. Transmission, however, must consider an appreciably lower antenna gain whenever the spacecraft attitude is unknown. Thus, the typical gain of the antenna at this time is expected to be at least -30 db for 99 percent of the possible aspect angles, with no nulls deeper than -30 db over 10 degrees wide.

There is a constraint on the amount of time that the spacecraft transmitter can be operated continuously on high power in the transit environment expected in the launch-to-acquisition period. Continuous transmitter high-power operation will result in the transmitter power amplifier tube exceeding its operating temperature. Thus, since the high-power operation is initiated just before separation, the acquisition must be accomplished within 1 hour after launch or the transmitter will overheat. Under standard conditions, the spacecraft is expected to be acquired within 20 minutes maximum after the transmitter is switched to high power, and the transmitter can be commanded to low power if telemetered temperature data indicates that the transmitter is overheated.

Each of the spacecraft receivers is a conventional crystal mixer superheterodyne-type FM receiver. A typical receiver is expected to have the following characteristics: (1) a noise figure no greater than 14 db, (2) a bandwidth of no more than 13.5 kc for getting a ground command into the receiver, (3) a threshold signal-to-noise ratio in this bandwidth (assuming the signal is centered in the passband) of greater than 12 db, and (4) a dynamic range of at least 40 db. The stability of the spacecraft receivers is the same as that of the spacecraft transmitters, so that the uncertainty in the knowledge of the spacecraft receiver frequency (to which the ground transmitter must be tuned during the acquisition) is the same as that indicated in the previous discussion of transmitter stability. As in the case of the transmitters, the uncertainty at the time of acquisition can be reduced by measuring the receiver frequency just before launch.

The one channel of PCM data which is being transmitted during this phase of the mission includes signals which can aid in the acquisition (such as the transponder static phase error, the receiver agc, the transponder phase-lock signal, etc.). This data modulates a subcarrier signal which, in turn, phase-modulates the transmitter carrier signal. The subcarrier oscillator utilized to modulate the carrier signal provides a low modulation index so that a very high percentage of the transmitted power is in the carrier center frequency thereby enhancing the probability of the ground receiver acquiring the carrier signal.

Sun Acquisition

The spacecraft next must acquire and lock on to the sun to (1) establish the vehicle attitude relative to the sun on which the passive thermal control of most spacecraft components during transit depends, (2) align the solar panel so that it faces the sun and can begin to convert solar energy into required electrical power, and (3) aid in establishing a known and accurate vehicle reference before the midcourse and terminal descent maneuvers. The acquisition and lockon are accomplished automatically in response to a single ground command by a sequence of spacecraft maneuvers produced by the gas jet system. The gas jet system is controlled by the flight control subsystem in accordance with signals provided by the secondary and primary sun sensors.

Because the secondary sun sensor is mounted on the solar panel (figure 13-1), the solar panel must be stepped to its transit position from the stowed position in which it is launched before sun acquisition can be achieved. Under normal conditions, automatic deployment of the solar panel is initiated by spacecraft Centaur separation. If the solar panel is not deployed automatically, it can be accomplished by means of ground commands. To accomplish this, the solar panel/planar array combination must be rotated approximately 60 degrees about the spacecraft roll axis, and the solar panel then rotated up from the roll axis by 85 degrees. These rotations, when accomplished by ground command, will result in a stepping of the panel in 1/8-degree increments per command. When the solar panel is stepped to its transit position, the active face of the panel is perpendicular to the spacecraft roll axis and points in the direction of the top of the spacecraft. The spacecraft can then be commanded to acquire the sun.

At the time the spacecraft receives the command to acquire the sun, its attitude is expected to be random. The acquisition of the sun from this random orientation involves a sequence of rotations (controlled automatically by the flight control programmer) at a nominal rate of 0.5 deg/sec. In general, the vehicle is first rotated about the spacecraft yaw axis until the sun lies in the plane determined by the roll and yaw axes, and is then rotated about the spacecraft pitch axis until the roll axis is within the primary sun sensor field of view.

The field of view of the secondary sun sensor is one hemisphere. Each of the four cells of this sensor has a view of one quadrant of the upper hemisphere of the spacecraft and provides an output signal when the sun is within that quadrant. These output signals are processed to produce 0.5 deg/sec rate commands for the

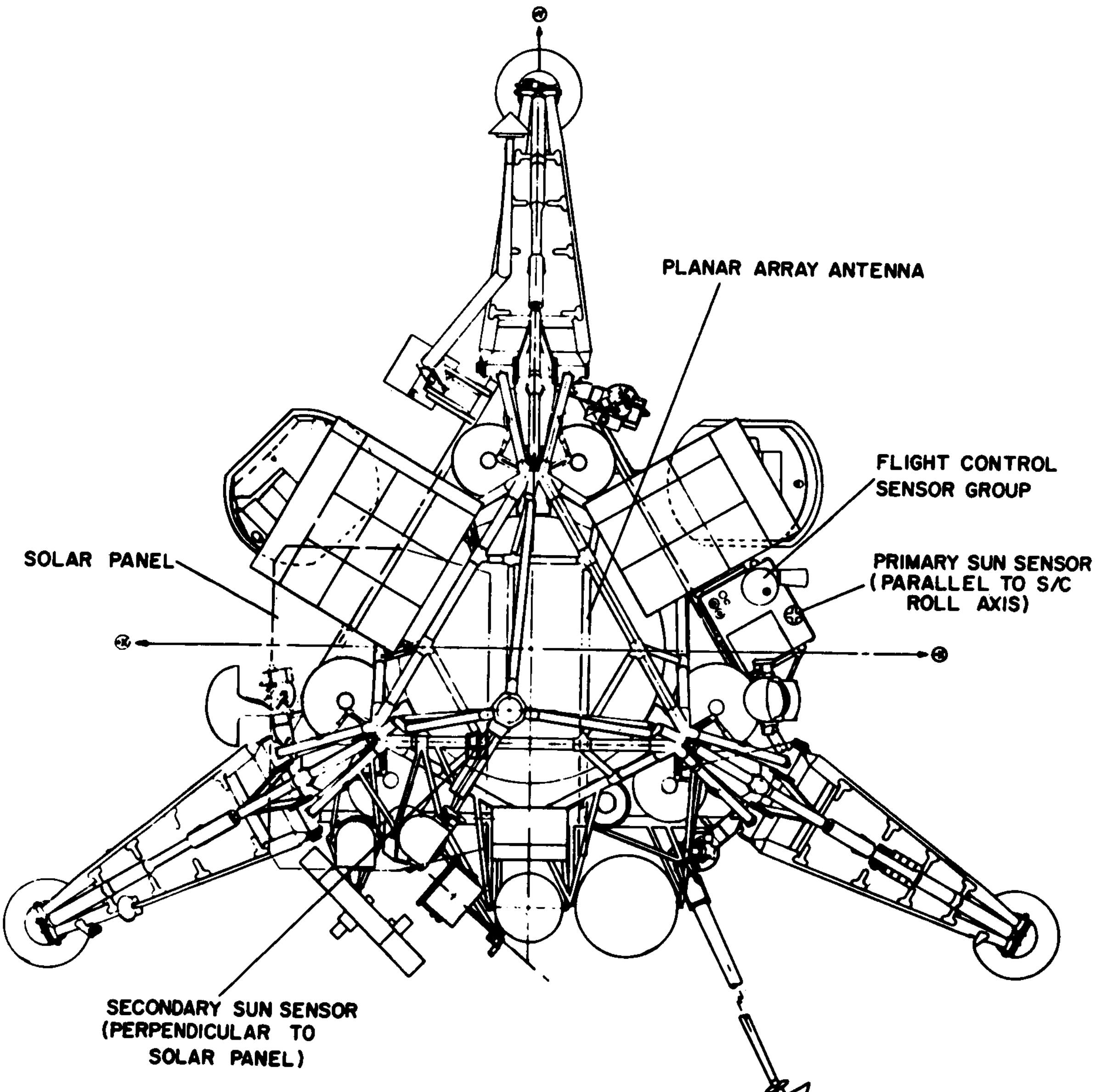


FIGURE 13-1. SUN SENSORS, LOCATIONS AND ORIENTATIONS

yaw and pitch attitude control loops as indicated in figure 13-2. In the event that no secondary sun sensor cell is illuminated (i.e., sun is not in the upper hemisphere), the spacecraft will still be commanded to execute a yaw maneuver which will eventually result in the sun's appearing in the sensor field of view.

When the sun moves within the field of view of the primary sun sensor, a sun lockon indicate signal is generated and control is switched to the optical mode in both the pitch and yaw channels. The primary sun sensor is mounted on top of the flight control sensor group and has a field of view that is aligned parallel to the spacecraft roll axis (figure 13-1). Simultaneous nulling of the pitch and yaw error angles is accomplished by the attitude control system in response to the signals provided by the primary sun sensor.

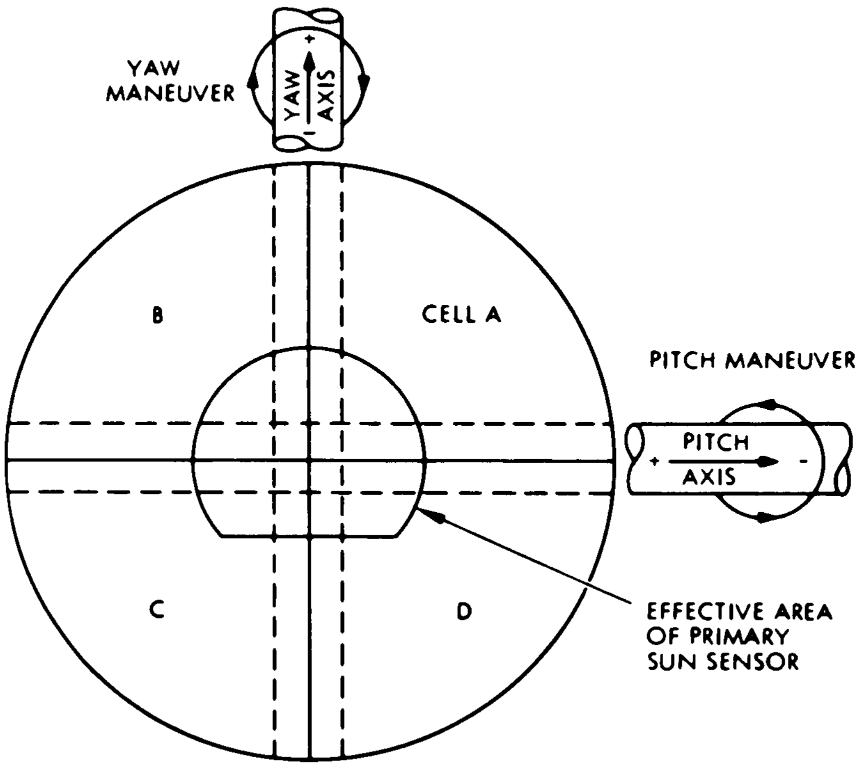
Sun lockon must be achieved within an estimated time of 1 hour after launch to ensure that certain parts of the spacecraft do not suffer permanent damage due to extremely high or low temperature.

Canopus Acquisition and Verification

With the spacecraft locked on to the sun, only the pitch-yaw attitude is controlled. Before the execution of the midcourse and terminal descent, where it is desired to thrust in a given direction in inertial space, the roll attitude must be established.

The spacecraft is mechanized to accomplish this by means of a star sensor which is adjusted before launch so that with the spacecraft locked on to the sun, the star canopus will appear within its field of view when the spacecraft is rolled to the proper attitude. The ground command to initiate the automatic star acquisition results in a spacecraft roll via the gas jet system at the nominal rate of 0.5 deg/sec while the spacecraft remains locked on to the sun. The spacecraft continues to roll until a star of the proper brightness (i.e., the expected brightness of Canopus) appears in the sensor field of view. When this occurs, a lockon signal is generated by the sensor, and the roll attitude is controlled by the error signal developed by the star sensor.

Although the Canopus sensor is designed to discriminate against stars that may appear in the sensor field of view, it also provides an output signal which makes it possible to verify that the object on which the sensor is actually locked is indeed Canopus. Verification can be accomplished by commanding the spacecraft to perform a complete 360-degree roll while the telemetered star sensor output



SECONDARY SUN SENSOR AS VIEWED ALONG ROLL AXIS (CENTER OF CIRCLE SHOWN IS ROLL AXIS WITH TOP OF SPACECRAFT POINTING OUT OF PAPER)

LOGIC USED TO COMMAND SPACECRAFT MANEUVERS TO ACQUIRE SUN FROM SECONDARY SUN SENSOR CELL SIGNALS

CELLS ILLUMINATED	COMMANDED MANEUVER
A	+ YAW
D	+ YAW
NONE	+ YAW
A AND D	+ YAW
B	- YAW
C	- YAW
B AND C	- YAW
A AND B	+ PITCH
C AND D	- PITCH

FIGURE 13-2. SECONDARY SUN SENSOR ORIENTATION LOGIC

(star intensity) is monitored. Thus a map of all stars having intensities in the sensitivity range of the Canopus sensor and falling within the 360-degree band swept out by the sensor field of view as the vehicle rotates is generated. A comparison between the positions of stars on this map and those on a map prepared before launch for the particular launch date permits Canopus to be verified.

The Canopus sensor is not required to perform in a radiation environment such as might be expected in the Van Allen belt. The sensor is expected to be inoperative while it is in the radiation environment, and to become operational as soon as the radiation level is reduced. Since the first midcourse correction is not planned to occur before 8 hours after launch, it will be possible to acquire and verify lockon to Canopus in time to establish the spacecraft roll reference required for the maneuvers before this correction.

Coast Phase 1

The coast phases of the mission are characterized by the spacecraft coasting with its attitude servoed to the sun and to the star Canopus with the low-power transmitter on in the transponder mode to permit continuous two-way doppler tracking. The transmitted power will be radiated via an omnidirectional antenna, with signal processing equipment on to provide PCM engineering data at a rate consistent with the signal-to-noise ratio available at the ground receiver as a result of these operating conditions. Coast phase 1 occurs before, and coast phase 2 occurs after, the midcourse correction.

Since the spacecraft is operating under coast conditions for most of the mission, the amount of nitrogen gas required for maintaining the attitude of the spacecraft during this phase represents the greatest portion (typically 51 percent for the 3 σ case) of the total amount of nitrogen expected to be used for standard operations accomplished by the coast phase attitude control system. The amount of nitrogen gas carried on the spacecraft is nominally 4.5 pounds. For a standard mission, this quantity of gas is expected to provide the nominal capability indicated in table 13-2 with a 3 σ probability plus a reserve quantity for nonstandard situations.

With the spacecraft locked on to the sun, the solar panel will be facing the sun and will convert the solar energy impinging on it into necessary electrical power. For the typical mission sequence for the transit phase, the total spacecraft electrical load is in excess of the power which will be provided to the loads

TABLE 13-2. COAST PHASE ATTITUDE CONTROL CAPABILITY

Mode/Maneuver	Time/Number of Maneuvers
Rate-stabilized mode, including dissipation of separation rates	30 minutes
Inertial hold	1 hour, 50 minutes
Optical mode (celestial hold)	65 hours
Sun acquisition	1
Star acquisition	1
Star verification	4
Roll maneuvers	6
Yaw maneuvers	5*

*Maneuvers required under standard conditions for two midcourse corrections.

by the solar panel via the optimum change regulator. This situation results in power being supplied continuously by the spacecraft battery to make up the difference between the power demand of the loads and the power being supplied by the solar panel (figure 13-3). Thus, the battery will not be fully charged at the time of landing.

Of the total electrical load indicated in figure 13-3, the power required for the operation of spacecraft components within compartment A and compartment B must be within the dissipation capability of these compartments. The typical standard transit phase results in the compartment dissipations indicated in figure 13-4. During the coast phases, compartment B has a steady dissipation of 17 watts with transient peaks of up to 37 watts, while compartment A has a steady dissipation of 25 watts with transient peaks of up to 75 watts. At present, compartment A has more radiating area than compartment B and is partially shaded during the coast phase, while compartment B is fully illuminated by the sun. Thus, it is able to dissipate more energy than compartment B during this period.

The power dissipation in the compartments results in the temperatures of the compartment trays as indicated in figure 13-5. During the coast phase, these

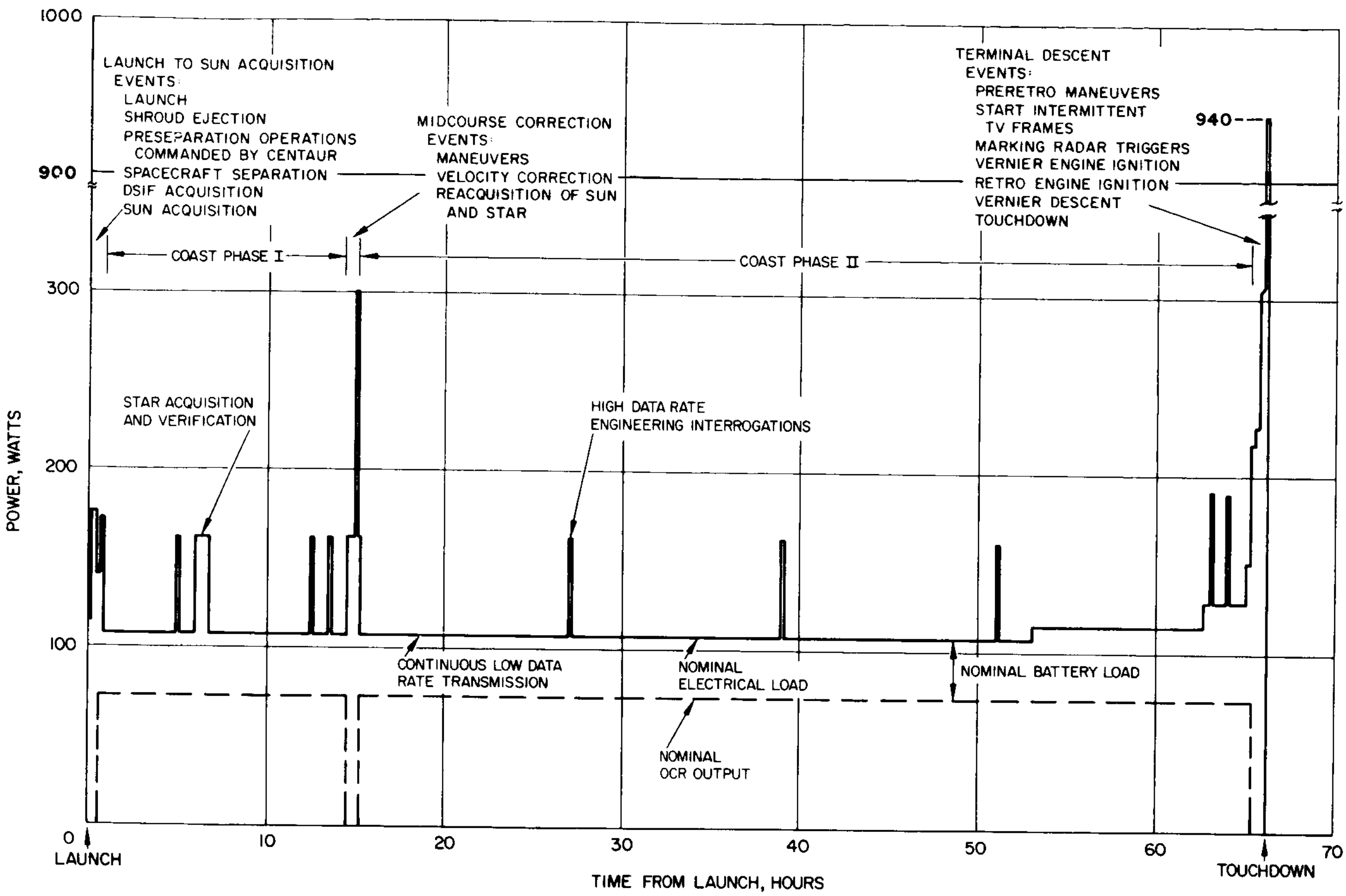


FIGURE 13-3. TYPICAL STANDARD TRANSIT OPERATIONS AND POWER PROFILE

temperatures are well within the estimated allowable upper limit of 125°F. The expected typical performance of other spacecraft equipment during the coast phases are shown in table 13-3.

During the coast phases, the spacecraft can be commanded to transmit PCM data continuously in any of the four data arrangements (i.e., commutator modes) of the engineering signal processor. Briefly, the types of data found on these modes are described below.

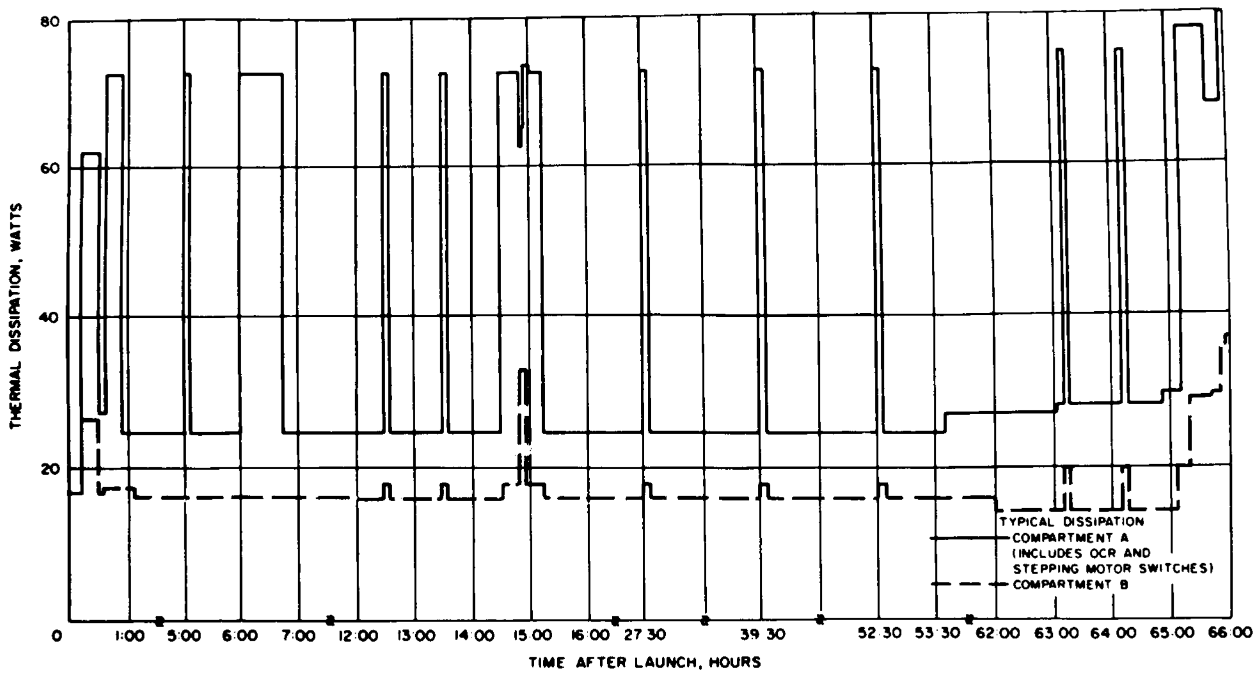


FIGURE 13-4. TYPICAL COMPARTMENT THERMAL DISSIPATION DURING STANDARD TRANSIT PHASE

Mode 1

This mode contains primarily coast and thrust phase attitude control data. It also contains most of the telemetered electrical current data and a few propulsion system temperatures. This mode is expected to be used for monitoring attitude maneuvers, sun and star acquisition, and midcourse velocity correction. It will also be of primary interest during the high-data-rate interrogation just before the midcourse and terminal maneuvers since this data can verify the proper state of the attitude control system.

Mode 2

This mode contains the same coast phase and thrust phase attitude control data which appear on mode 1; however, the number of signal samples is less since it is normally used during terminal descent when the data transmission via the high-gain planar array permits higher transmission bit rates to be used. This mode also includes signals covering the retro burning phase, vernier system temperatures, and electrical power.

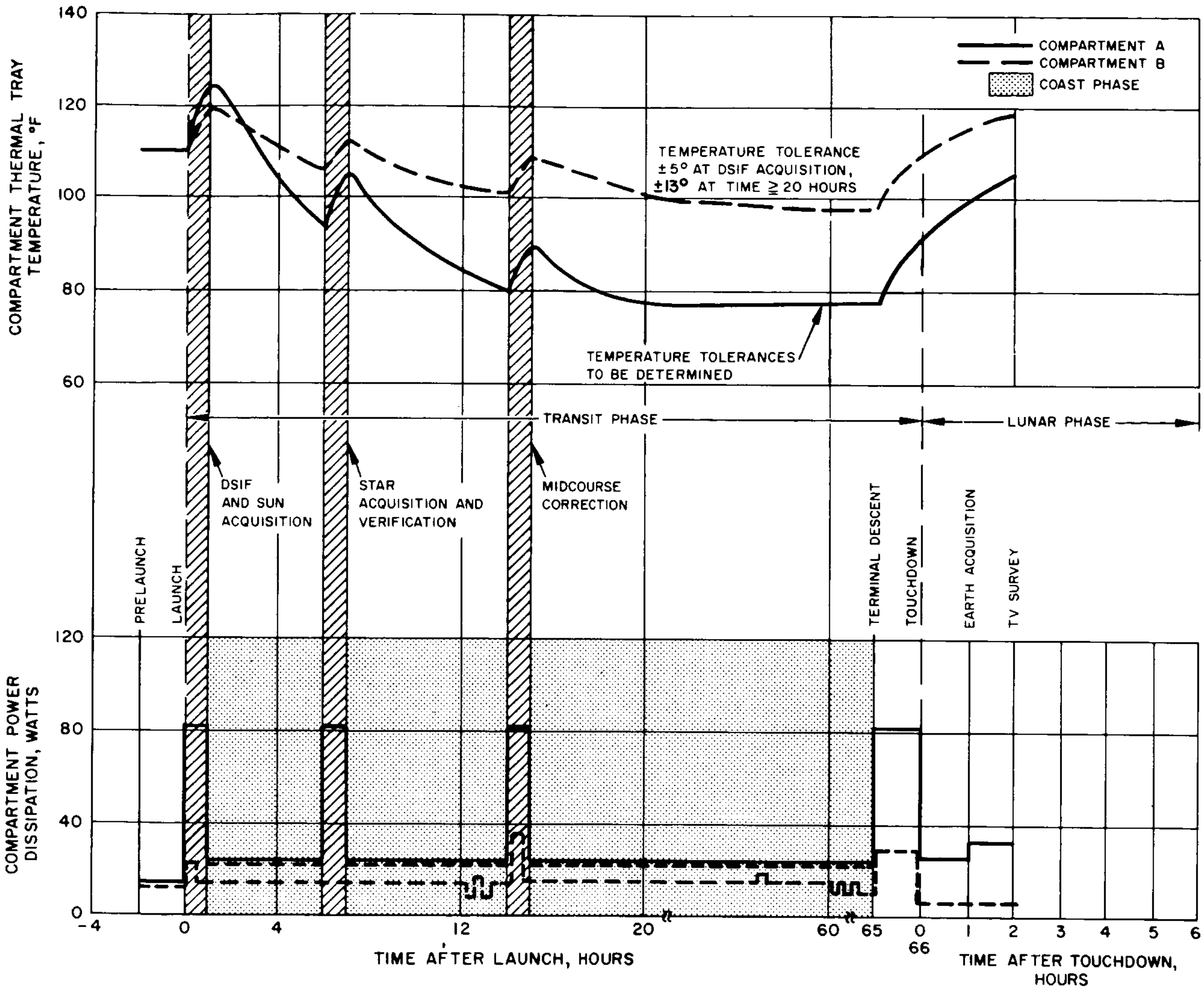


FIGURE 13-5. COMPARTMENTS A AND B TEMPERATURE PROFILES

Mode 3

This mode contains the signals used for lunar reflectivity measurements and for monitoring the vernier descent sequence. This data is provided by the doppler radar (RADVS) data and altitude control signals.

Mode 4

This mode contains primarily spacecraft electrical power, thermal mechanism positioning, and data link signals. It is intended for use in the DSIF acquisitions to position the solar panel and monitor the spacecraft power and thermal status during the mission.

TABLE 13-3. TYPICAL EXPECTED THERMAL PERFORMANCE DURING TRANSIT

Subsystem	Transit Temperatures (°F)					
	Survival Temperature Limits	Operating Temperature Limits	Prelaunch Temperature	Acquisition (Maximum Eclipse 1 hour)	Coast Phase	Touchdown (Maximum Eclipse 1/2 hour)
Vernier system*						
Painted propellant tanks	0 to 100		80 to 90	50 to 85	60 to 100	34 to 100
Blanketed propellant tanks	0 to 100		80 to 90			20 to 60
Thrust chambers						
Plane at valving	0 to 100		80 to 90	55 to 90	60 to 100	35 to 100
Barrel/ extension cone	0 to 100		80 to 90	45 to 90	20 to 60	0 to 60
Propellant transport lines						
Heated length	0 to 100		80 to 90	10 to 50	10 to 50	10 to 50
Aluminized length	0 to 100		80 to 90	30 to 90	50 to 100	0 to 100
Helium tank	0 to 100		80 to 90	20 to 90	55 to 100	25 to 100

*Temperatures listed under touchdown refer to those temperatures at time of system activation which correspond to a time of approximately 6 minutes before touchdown.

TABLE 13-3. TYPICAL EXPECTED THERMAL PERFORMANCE DURING TRANSIT (Cont)

Subsystem	Transit Temperatures (°F)					
	Survival Temperature Limits	Operating Temperature Limits	Prelaunch Temperature	Acquisition (Maximum Eclipse 1 hour)	Coast Phase	Touchdown (Maximum Eclipse 1/2 hour)
RADVS system*						
Signal data converter	-40 to 257	41 to 77	100 (maximum)		38 to 80	33 to 85
Klystron power supply	-40 to 257	-22 to 14	100 (maximum)		-24 to 16	-26 to 18
Velocity sensing antenna	-90 to 220	-90 to 220	100 (maximum)		-123 to 253	-131 to 261
Altimeter velocity sensing antenna	-90 to 220	-90 to 220	100 (maximum)		-123 to 253	-135 to 265
Shock absorber	-50 to 300	20 to 125	85 (minimum)	0 to 15	80 to 120	15 to 125
Main retro propellant	20 to 105	20 to 75	80 to 90	70 to 90	32 to 85 (30 hours after launch)	17 to 70
Solar panel	-320 to 235	-200 to 145	80 to 90	-200 to 140	117 to 140	-140 to 140

*Temperatures listed under touchdown refer to those temperatures at time of system activation which correspond to a time of approximately 6 minutes before touchdown.

TABLE 13-3. TYPICAL EXPECTED THERMAL PERFORMANCE DURING TRANSIT (Cont)

Subsystem	Transit Temperatures (°F)					
	Survival Temperature Limits	Operating Temperature Limits	Prelaunch Temperature	Acquisition (Maximum Eclipse 1 hour)	Coast Phase	Touchdown (Maximum Eclipse 1/2 hour)
Planar array antenna	-320 to 300	-255 to 275	80 to 90	-253 to 150	-166 to -100	-234 to 258
Nitrogen system						
Jets	-30 to 160	-30 to 160	80 to 90		100 to 160	
Tank	-70 to 125	-70 to 125	80 to 90		30 to 110	
Spaceframe lines	-160 to 90	-160 to 90	80 to 90			
Leg lines	-50 to 250	-50 to 250	80 to 90		150 to 250	

*Temperatures listed under touchdown refer to those temperatures at time of system activation which correspond to a time of approximately 6 minutes before touchdown.

Any of these modes can be transmitted at any of the available spacecraft bit rates: 4400, 1100, 550, 137.5, or 17.2 bits per second. The choice of the rate must be determined on the basis of the strength of the received signal at the DSIF station.

Because of losses in cabling and tolerances on the transmitter power output, the transmitter power available at the antenna varies from the nominal 100 milliwatts (low power) and 10 watts (high power) at the transmitter output. Table 13-4 indicates the expected minimum power available at the antenna. As shown, at least 24 milliwatts of power are expected to be available during the coast phases.

TABLE 13-4. POWER INTO ANTENNAS

Operational Mode	Typical Range of Values	
	Minimum	Nominal
Low-power transmitter, omnidirectional antenna	24 milliwatts	45 milliwatts
Low-power transmitter, planar array	36.2 milliwatts	58.8 milliwatts
High-power transmitter, omnidirectional antenna	2.69 watts	4.5 watts
High-power transmitter, planar array	4.06 watts	5.88 watts

Based upon the DSIF system parameters documented in Surveyor Spacecraft/ DSIF System Interface Requirements dated 29 July 1963, the expected quality of telemetry data obtained is indicated in table 13-5.

Midcourse Correction Maneuver

The soft landing of the Surveyor spacecraft at a desired lunar location with sufficient time to obtain lunar data via the Goldstone DSIF tracking station is the primary goal of a Surveyor mission.

As a result of expected dispersions in the launch vehicle injection parameters, a midcourse correction will normally be required to correct the spacecraft trajectory. The purpose of the correction is to minimize the expected miss from the desired landing spot on the moon, to optimize the probability of soft landing, and to adjust the spacecraft time of flight so that the desired visibility from the Goldstone tracking station is achieved during and subsequent to landing.

In general, the velocity increment applied to the spacecraft to effect the correction will consist of two components. The first component will be in the so-called critical plane and will correct primarily for the miss. The second

TABLE 13-5. EXPECTED QUALITY OF TELEMETRY

Data	Signal-to-Noise Ratio	Resulting Quality
PCM (at lunar distance)	8 ±1 db (rms signal to rms noise) at input to subcarrier discriminator	Bit error rate of approx. 3×10^{-3}
Accelerometer channels (Transmitted during retro burning)	>46 db (Peak signal to rms noise) at output of subcarrier discriminator for signal producing peak deviation of the subcarrier	<0.5 percent distortion due to noise
Touchdown Strain Gage Channels	20 db (rms signal to rms noise at output of subcarrier discriminator for signal producing peak deviation of the subcarrier)	10 percent distortion due to noise
TV	24 (Peak-to-peak signal to rms noise at luminance level of 100 foot Lamberts)	

component will be normal to the critical plane and will be sized to optimize the following soft landing and time-of-arrival constraints:

a. The main retro burnout velocity must be between limits imposed by the operational ranges of the doppler velocity and altimeter radars and the mechanized range/range-rate descent curve.

b. A sufficient amount of vernier fuel must be reserved to ensure that the required moment control can be exerted during the main retro phase and that the desired thrust can be maintained until touchdown during the vernier descent. The quantity of reserved fuel in excess of the amount to be used for the midcourse correction and the nominal terminal descent (defined as the fuel margin) must be large enough to ensure that a soft landing can be accomplished for the 3σ variations in specific impulse, burnout velocity, and altitude, etc.

c. The incidence angle of the unbraked velocity must be less than some maximum value (nominally 45 degrees) to ensure that the marking radar will operate properly. This constraint will automatically be satisfied by selecting the landing site properly (figure 13-6).

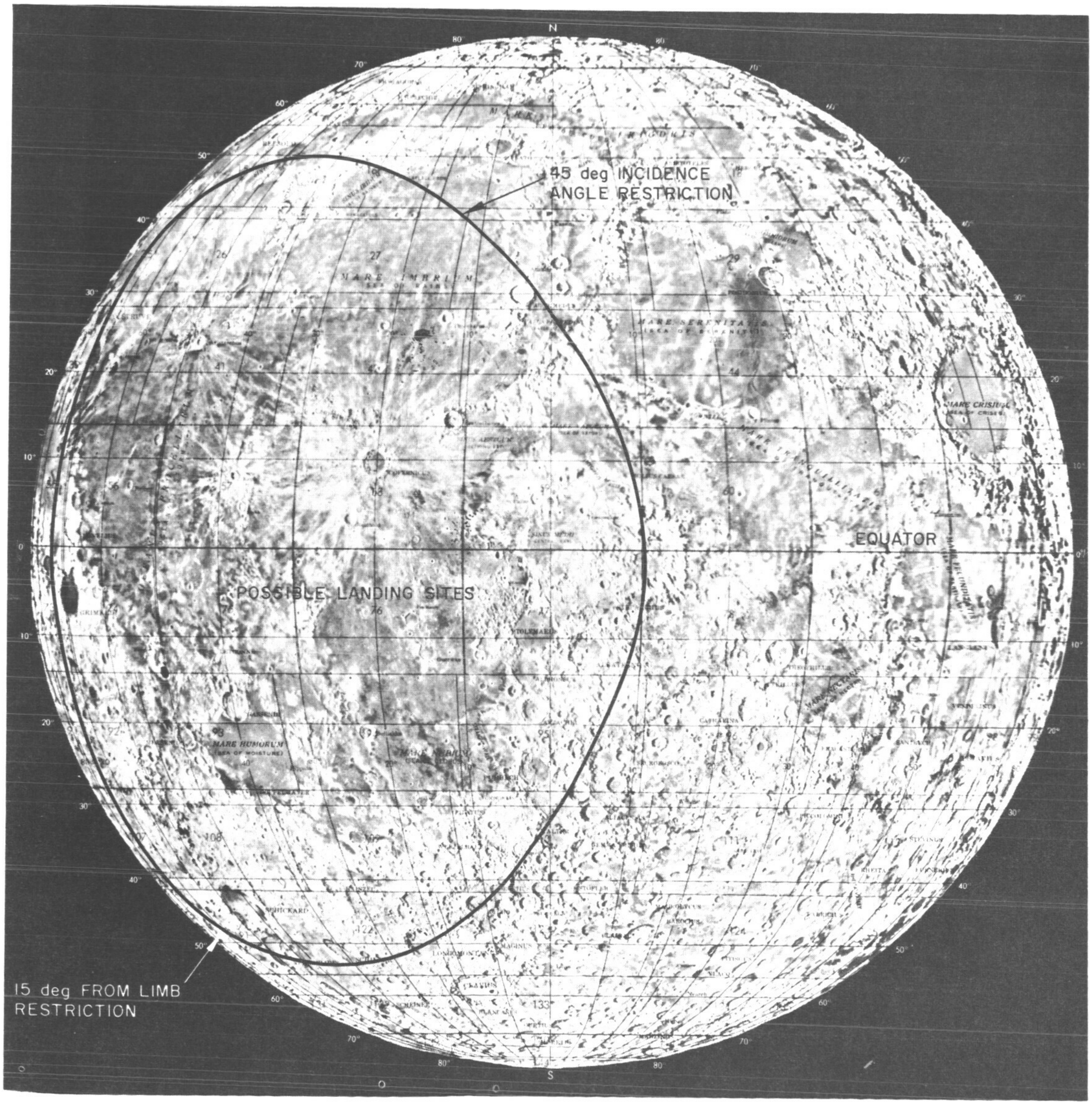


FIGURE 13-6. POSSIBLE LANDING SITES

d. The time at which the spacecraft arrives at the moon must be earlier than a minimum allowable time (nominally 3 hours) before the last lunar visibility at Goldstone to ensure that there is sufficient time to conduct TV surveys after touchdown. This time must also be later than nominally 2 hours after the first visibility at Goldstone so that the command and execution of the terminal descent can be accomplished from Goldstone.

The actual midcourse velocity correction may differ from that desired because of uncertainties in the spacecraft attitude and velocity control systems. The uncertainty in the desired landing location has been specified in JPL Specification 30240D, "Surveyor Spacecraft Design Specification," and this, in turn, sets the limits on the midcourse mechanization errors.

Figure 13-7 shows the estimated miss at the moon (99 percent) as a function of the midcourse correction magnitude. A 30-meter-per-second maneuver executed 15 hours after injection results typically in a miss of 47 kilometers if the correction is applied within 15 minutes after the start of the initial pre-midcourse maneuver. Therefore, for these assumed conditions (which are typical), the miss is within 60 kilometers. This figure illustrates that for a given correction,

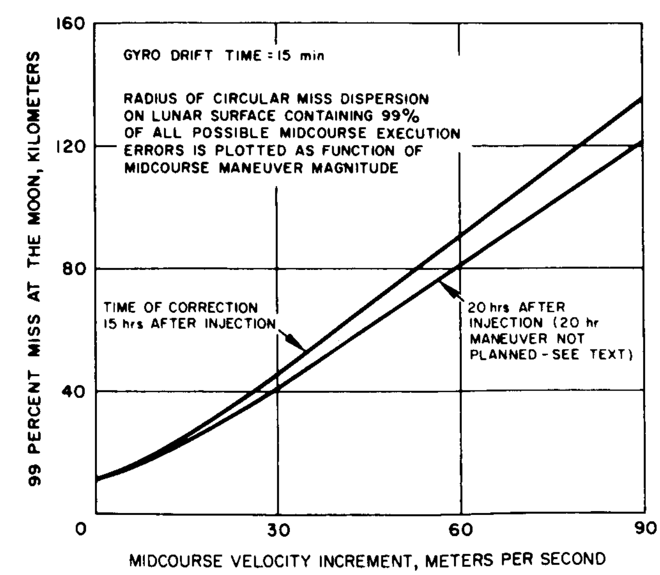


FIGURE 13-7. TYPICAL MISS AS FUNCTION OF MIDCOURSE MANEUVER MAGNITUDE FOR NORMAL IMPACT

the expected error will decrease if the correction is applied at a later time (e.g., 20 hours versus 15 hours) since the sensitivity of the trajectory to a given correction decreases with time. However, for the same reason, the amount of correction required for a given miss will be larger for a correction applied at 20 hours than one applied at 15 hours. Further, for direct ascent trajectories in 1965 and 1966, a midcourse maneuver cannot be commanded from the Goldstone tracking station at 20 hours after injection, since the spacecraft will no longer be visible from Goldstone.

The maximum midcourse correction that can be executed depends on the amount of fuel that can be allotted for the midcourse correction without jeopardizing the ability of the spacecraft to achieve a soft landing. Typically, the spacecraft will be loaded with sufficient vernier fuel to permit up to a 30 meter-per-second correction to be accomplished and to ensure a 99-percent probability that sufficient fuel is available during terminal descent for a nominal unbraked impact velocity of 2690 meters per second, incidence angles up to 45 degrees (See Appendix B, item 5) and a nominal time of flight of 66 hours. The actual maximum midcourse correction capability could exceed the nominal 30 meter-per-second (See Appendix B, item 3) value on any mission, depending on the actual time of flight, the unbraked incidence angle, and the unbraked impact velocity. For example, figure 13-8 illustrates the typical relationship between the amount of propellant available for midcourse correction and the unbraked impact velocity for various fuel margins and burnout velocities for a specific time of flight and unbraked incidence angle. Also ,indicated on this figure is a curve which depicts, as a function of impact velocity, the amount of propellant the spacecraft can use for midcourse correction and still satisfy the constraint of reserving sufficient terminal descent fuel for 99 percent of all cases. For example, for impact velocities of 2686 and 2670 meters per second, midcourse propellant can be used in each case until the fuel margin burnout velocity is reached (follow straight lines A and B). Clearly, the amount of allowable midcourse correction is a function of the actual unbraked impact velocity. Since the midcourse capability is also a function of the time of flight and incidence angle, there is no specific maximum capability for all missions, but a particular maximum capability will exist for each mission, depending on mission parameters.

To perform a midcourse correction maneuver, the spacecraft must be commanded to point in the proper inertial direction and to operate its vernier engine

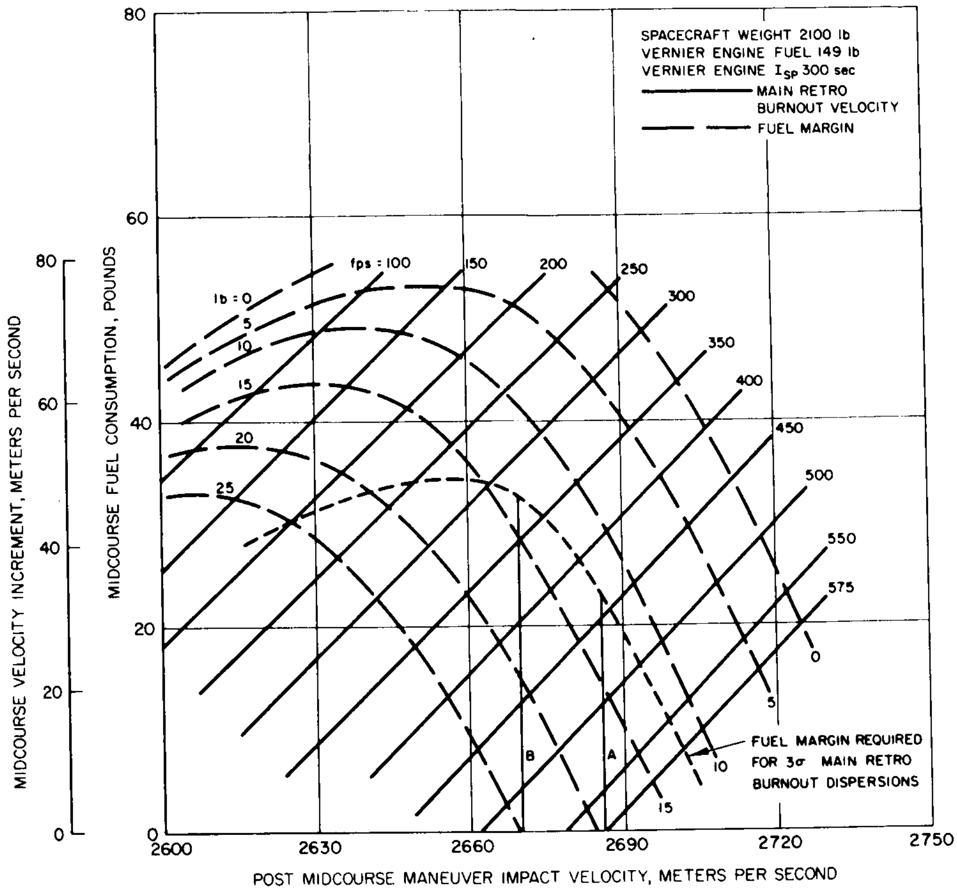


FIGURE 13-8. TYPICAL MIDCOURSE MANEUVER CAPABILITY

system to achieve the required velocity correction. The first step in this maneuver is to orient the vehicle angularly so that the thrust axis of the vernier engine system (i.e., the spacecraft Z-axis) is aligned with the desired direction for the velocity correction. The coast phase attitude control system provides the capability for maneuvering in either direction in yaw, pitch, and roll. All of these maneuvers are performed at a nominal rate of 0.5 deg/sec, and multiple angular maneuvers are performed serially, one at a time. The method used to command the spacecraft to make an angular change of a desired magnitude about any one of the three axes is as follows: the spacecraft is commanded to maneuver in the desired direction about that axis for the specific time interval required for it to maneuver through the desired angle at the fixed rate of 0.5 deg/sec.

A maneuver sequence is initiated by sending a quantitative command containing the 10-bit binary equivalent of the desired maneuver duration. This magnitude is stored in a register by the spacecraft flight control programmer, and is also transmitted back to the ground receiver where it may be compared with the magnitude commanded from the ground. When it is verified that the spacecraft has received the proper magnitude, it can then be commanded to execute the desired maneuver (roll, pitch, or yaw) in the proper direction.

The spacecraft programmer contains two clock rates which permit it to count the number of seconds stored in the programmer. For attitude maneuvers, the clock rate provided is such that the nominal storage capability of the register is 409.6 seconds. Thus, the maximum single angular change that can be commanded and controlled automatically by the spacecraft programmer is nominally 204.8 degrees. Larger single angular maneuvers can be executed only by manually controlling the maneuver time from the earth.

Normally, two maneuvers (roll-pitch or roll-yaw) are required to orient the spacecraft before applying midcourse thrust. There is no multiple command storage capability aboard the spacecraft, and the programmer can only store one magnitude at a time. Consequently, these maneuvers must be accomplished serially.

The velocity correction applied during the midcourse sequence is accomplished in a manner similar to that used for the maneuvers. The spacecraft is commanded to thrust for a desired period of time. As indicated in table 13-6, the thrust is provided by the three vernier engines to cause the spacecraft to experience a constant acceleration of nominally 0.11 earth g for the commanded time. The magnitude of the desired thrust time is sent to the spacecraft and verified in a manner similar to that used to verify the maneuver time magnitudes.

In the case of the velocity correction, the second available flight control programmer clock rate is employed. This rate is such that the maximum storage capacity of the register nominally is 51.2 seconds. This time is sufficient to provide a nominal velocity of up to 55 meters per second.

At the end of the commanded time for the midcourse thrusting, the flight control programmer provides the cutoff signal for shutting off the vernier engines. The cutoff signal can also be commanded from earth if, in an abnormal situtation, the programmer should fail to provide this signal.

TABLE 13-6. VERNIER ENGINE THRUST CONTROL MODES

Operational Mode	Description of Mechanization	Phase of Mission When Utilized
Midcourse velocity correction	Vernier engine thrust servoed to provide a fixed vehicle acceleration for the commanded time interval.	Midcourse maneuver
Main retro	The vernier engines are throttled differentially to correct for main retro thrust misalignment. Two commandable thrust values (200 and 150 lbs) are available. However, the flight control electronics will override these values when the main retro misalignment requires compensation thrust from the vernier engines different from these values.	Retro burn period
Main retro separation	Constant total vernier thrust of near maximum value (typically >280 pounds) is provided for an interval controlled automatically by the flight control programmer.	Retro separation
Acceleration control	Thrust to mass ratio maintained constant at nominally 0.9 lunar g.	Vernier descent
Velocity control by range reference	Thrust is controlled by the doppler radar signals to approximate an optimum (minimum fuel consumption) descent trajectory.	Vernier descent

TABLE 13-6. VERNIER ENGINE THRUST CONTROL MODES (Cont)

Operational Mode	Description of Mechanization	Phase of Mission When Utilized
Constant velocity control	Thrust is controlled so that a constant spacecraft velocity of nominally 5 fps is obtained.	Final vernier descent

Coast Phase 2

Following execution of the midcourse velocity correction, the spacecraft is commanded to reacquire the sun and star, thereby returning the spacecraft to the optical control mode. It will be possible for the reacquisition of the sun and star to be accomplished either by (1) performing maneuvers of the same magnitude as those performed before the midcourse thrusting, but in the opposite direction and in the reverse order, or (2) by commanding the spacecraft to acquire the sun and star as it did in the original sun and star acquisition.

Pre-Terminal Descent

Planar Array Positioning

The high-gain planar array antenna is stepped by earth commands to the position that will result in its pointing to the earth after the pre-retro maneuvers are performed. The stepping of the antenna is accomplished by ground commands, with each command resulting in a 1/8-degree step.

It is desirable to accomplish the planar array positioning prior to the execution of the terminal descent maneuvers so that the amount of time during which the system is placed in the inertial mode is kept to a minimum without reducing the number of TV pictures that can be taken after the maneuvers are completed. The angle to which the planar array must be stepped changes primarily as a function of the unbraked incidence angle, and is expected to vary between approximately 38 and 68 degrees for incidence angles of ±45 degrees off vertical. (For vertical impact, the array would have to be stepped approximately 52 degrees, involving approximately 416 commands.)

At 48.5 degrees, the planar array will begin to obscure the field of view (nominally a 5-degree half-angle cone) of the Canopus sensor sun channel. At an angle of 68 degrees, the planar array will permit only a cone of approximately

0.75 degree half-angle clear field of view. With a restricted field of view, the Canopus sensor may loose the sun reference, resulting in the roll attitude of the spacecraft reverting to the inertial mode. The approximate time required to step from 48 to 68 degrees is 80 seconds, and the amount of time required to send and verify the commands required to initiate the first roll maneuver is estimated to be approximately 155 seconds.

Thus the system could be on inertial for almost 4 minutes in addition to the presently planned time if the complete stepping sequence were accomplished for a mission where 68 degrees of stepping were required. It may be necessary to step the array out in two sequences. The array could first be stepped out to an angle which would ensure no shadowing of the Canopus sun channel (e.g., 45 degrees) before the maneuvers are performed. Then, any additional stepping that is required in excess of the initial stepping could be accomplished subsequent to the maneuvers. The time required for this additional stepping (a maximum of 92 seconds) could reduce the time available for obtaining approach TV pictures prior to retro-rocket ignition.

Maneuvers

Before the retro-thrusting period, three spacecraft maneuvers normally are performed. The descent sequence is shown in figure 1-3. The first two maneuvers (nominally a roll and a pitch or yaw maneuver) align the thrust axis of the main retro-engine approximately with the velocity vector. The last maneuver (a roll) causes the planar array to point to the earth so that the necessary telecommunication information bandwidth required for the transmission of approach TV pictures (for those missions where lunar lighting conditions permit) and high-rate (4400 bits per second) PCM and accelerometer data during the pre-retro and retro burning phases are obtained.

Each of these three maneuvers can be accomplished in the same manner as the midcourse correction maneuvers. A quantitative command representing the desired maneuver duration is sent to the spacecraft, and proper receipt of this command by the spacecraft is verified on the ground from spacecraft telemetry. The maneuver is then executed by the thrust provided by the gas jets at a nominal angular rate of 0.5 deg/sec.

Approach Television

The spacecraft can be commanded to take up to 100 TV pictures with the approach TV camera in the interval between the time that the spacecraft executes the last roll maneuver to point the high-gain planar array toward the earth and the time that the altitude marking radar signal initiates the retro ignition sequence. The first pictures will be taken when the spacecraft is at a range no less than 1000 miles from the lunar surface.

Since the present design of the spacecraft does not provide capability of transmitting TV video signals simultaneously with PCM data of one of the commutator modes, the TV picture sequencing will be in blocks normally consisting of 10 pictures. These blocks of pictures will be separated by periods in which engineering data is telemetered so that the status of the spacecraft can be monitored, particularly when necessary commands are being sent. The spacecraft will be commanded to transmit the last TV picture at a time as close to the predicted time at which the altitude-marking-radar trigger is expected to occur, but not so close as to jeopardize the transmission of PCM data which can verify that this trigger signal is generated. It is estimated that the last picture taken before retro ignition will be at a nominal range of 80 miles from the lunar surface.

The approach TV geometry is shown in figure 13-9. The field of view of the approach TV camera is nominally 6.4 by 6.4 degrees so that a picture taken at 1000 miles will be greater than 180 kilometers on each side. The coverage provided by the field of view is a function of the unbraked impact angle. The pictures taken in the pre-retro period are expected to have a negligible smear because of the angular rates of the vehicle while the cold gas attitude control system is still controlling the vehicle attitude.

In addition to the pre-retro pictures, it may be possible to obtain a few additional pictures when the landing sequence permits. Any pictures taken during the retro burning phase will be subject to degradation because of the effects of the main retro-engine and vernier engine exhaust gases and are also expected to have a minimum of five TV lines of smear because of effects of retro thrusting. Also, any picture taken between retro burnout and separation will be subject to degradation caused by vernier engine exhaust gases and is expected to have one to four TV lines of smear.

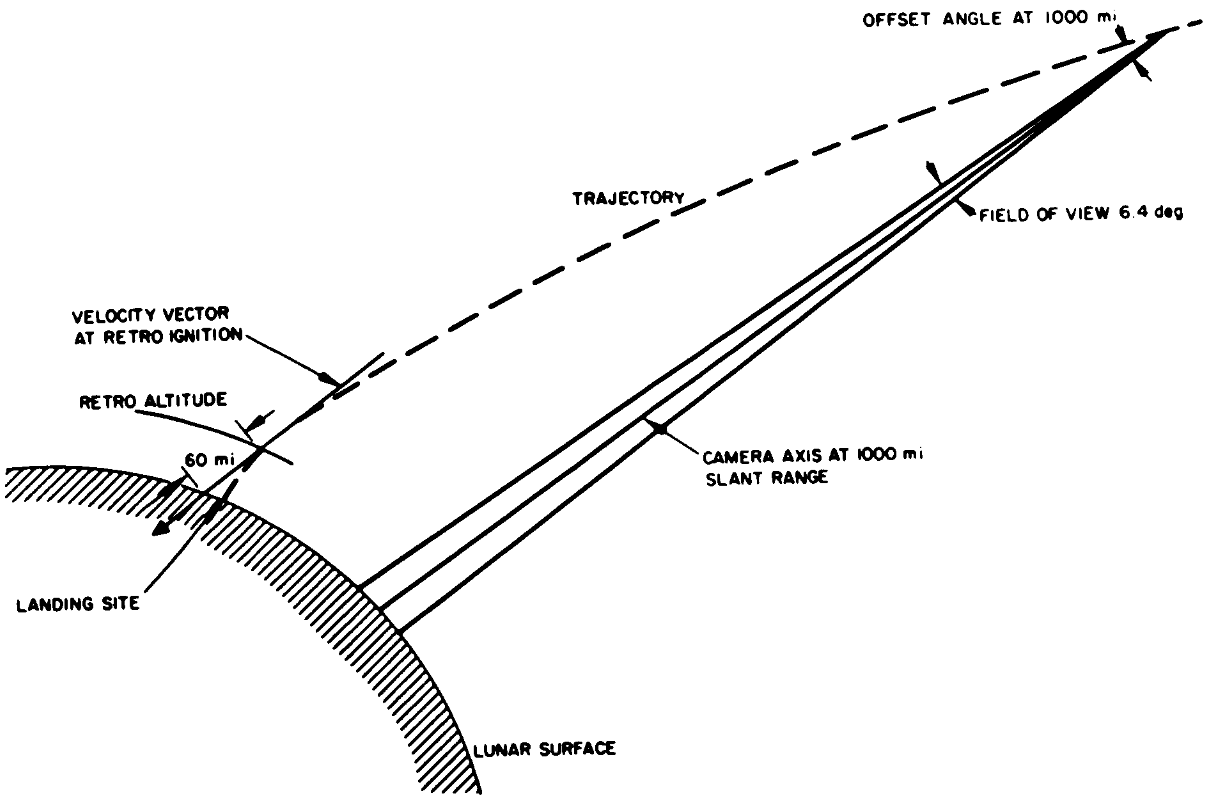


FIGURE 13-9. APPROACH TELEVISION GEOMETRY

It will probably not be possible to obtain TV pictures after retro separation when the spacecraft attitude is switched to doppler control since the planar array will not be pointing to the earth.

For those launches resulting in night landings as well as some day landings, the spacecraft will be shaded by the moon during part or all of the terminal descent phase. The maximum amount of time during which the spacecraft can be shaded during this phase is set by two different considerations: (1) the amount of time that the spacecraft can be placed on inertial hold (i.e., relinquish sun lockon) is limited primarily by the expected amount of gyro drift, and (2) spacecraft equipment not installed in thermally controlled compartments which depend on the spacecraft coast phase orientation may be damaged and become inoperative. The limitation attributable to allowable gyro drift is nominally 30 minutes. Coincidentally, the constraint due to thermal limitations of spacecraft equipment is also approximately 30 minutes. Table 13-7 shows the spacecraft equipment whose performance would be affected by being shaded during the terminal descent phase. This includes the vernier system, flight control system, landing leg shock absorbers, and the solar panel. The most critical items are the doppler radar signal data converter and the shock absorbers.

Lunar Approach Altitude Measurements

To reduce the probability of false alarm, the altitude marking radar will not be enabled until the spacecraft is at a nominal slant range of 120 ±45 miles.

For the design range of impact velocities (2615 to 2692 meters/second), the number of measurements provided in the interval before the radar marks will nominally vary from 36 (for highest velocity and latest enabling time) to 257 (for lowest velocity and earliest enabling time), assuming that the received signal exceeds the agc threshold at the time that the altitude marking radar is enabled.

Measurements of lunar reflectivity can also be obtained during the vernier descent phase by measuring the received signal strength of the radar return in the four doppler radar beams. There should nominally be a total of 520 measurements during vernier descent, assuming that the radars are locked on at the time of burnout.

Terminal Descent

The terminal descent phase comprises main retro descent, vernier descent, and touchdown.

TABLE 13-7. MAXIMUM SHADE TIME FOR SPACECRAFT EQUIPMENT

Spacecraft Equipment	Maximum Allowable Time in Shade (approximate hours)
Vernier engines (3)	0.8
RADVS signal data converter	0.5
Helium gas supply	0.8
Shock absorbers (3)	0.5
Solar panel	0.8

Main Retro Descent

The events that constitute the automatic retro sequence are illustrated in figure 13-10. The sequence is initiated by the trigger signal provided by the altitude-marking radar when the spacecraft is nominally at a slant range of 60 miles from the lunar surface, and is controlled automatically by the flight control programmer. After a prescribed delay (commanded and verified well in advance of the expected marking time and stored in the flight control programmer), the vernier engines are ignited. The main retro engine is ignited after a 1 second delay to permit the vernier engine thrust to stabilize. After a further delay of 0.55 second, the RADVS is turned on.

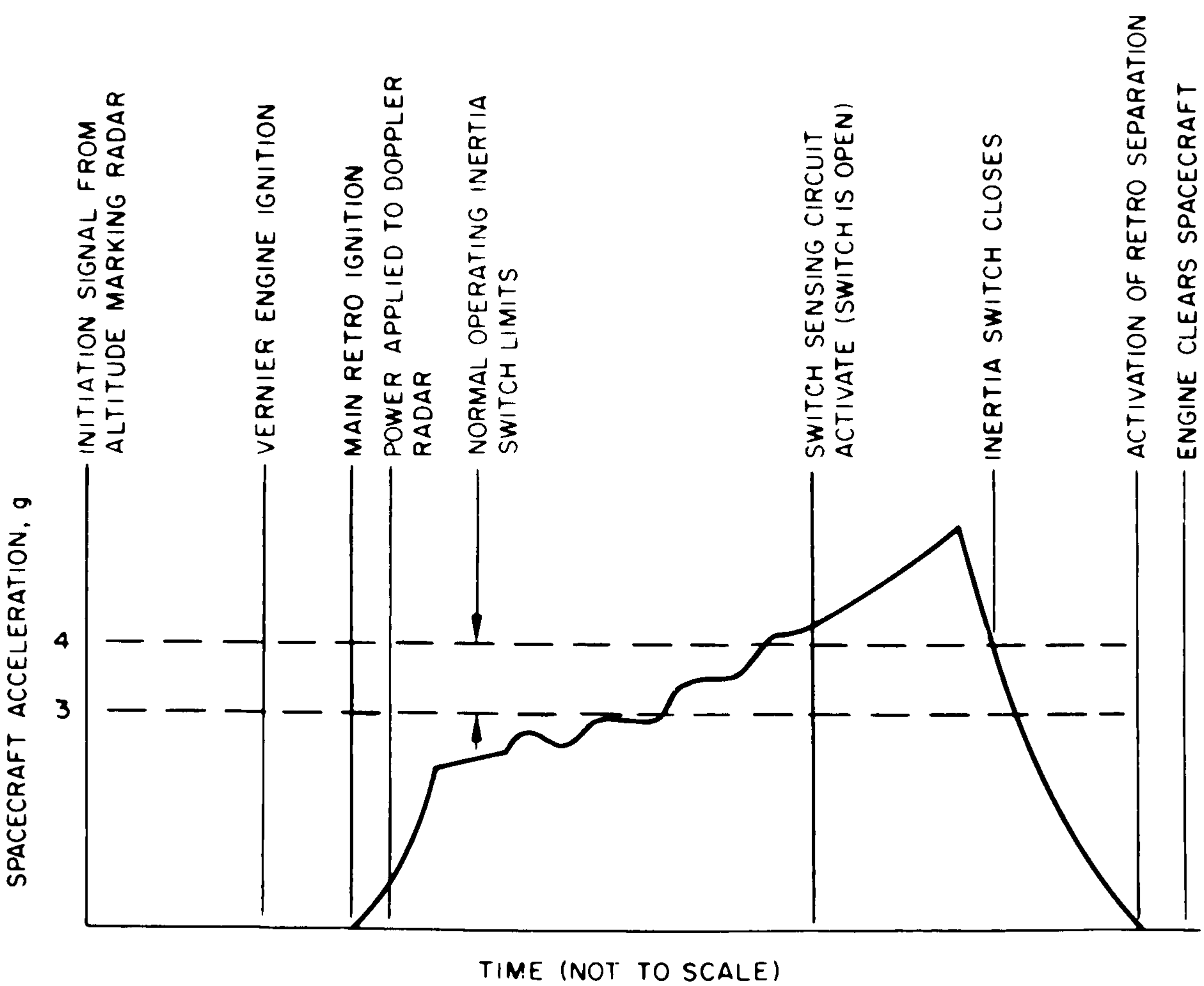


FIGURE 13-10. TYPICAL RETRO SEQUENCE OF EVENTS

The total thrust provided by the vernier engines during the retro burning phase can be commanded in advance of the retro period to one of two values, typically 150 or 200 pounds. The lower level is provided for use on those trajectories where the impact (lunar approach) velocity is low and hence the burnout velocity will be low. By reducing the vernier thrust, the burnout velocity can be increased to a more acceptable value. The descent trajectory design is based on the assumption that the doppler velocity and altitude radars cannot be used reliably while the main retro-engine is burning. Thus, the spacecraft attitude will be controlled so that the attitude will remain fixed inertially throughout the retro phase.

The main function of the vernier engines during this period is to provide the necessary moment to accomplish this control. Where possible they will also be controlled to maintain the total vernier thrust at a constant (i.e., 150 or 200 pounds) value. The thrust phase attitude control system is mechanized so that the moment demand overrides the thrust level command when the two are not compatible.

Moment control about the roll axis is provided by swiveling one of the vernier engines about a radial line perpendicular to the vehicle roll axis. The engine can be swiveled approximately ±5.5 degrees.

The average thrust provided by the main retro-engine is nominally 9000 pounds and results in the removal of the bulk of the incoming spacecraft velocity. As the grain burns out, the decrease in thrust is sensed by an inertia switch. The switch senses when the spacecraft acceleration has decreased to nominally 3.5 g and provides a signal to the flight control programmer to initiate the retro separation sequence. This signal results in the vernier engine thrust level being increased to the maximum programmed level so that the spacecraft will be experiencing the maximum deceleration at the time of separation, thereby aiding the separation. In addition, the flight control timer begins to count down. After a fixed time delay of sufficient duration to allow the main retro thrust to reduce to a negligible value, the programmer delivers a signal causing the retro separation nuts to be blown apart.

The flight control programmer continues to count down and, after a delay sufficient to permit the retro-engine to clear the spacecraft, provides an arming signal which permits the spacecraft pitch and yaw attitude control to be switched from inertial to doppler velocity reference when the doppler velocity reliable signal is generated. If this signal is already present at the time that the arming signal is generated, control will be switched immediately to doppler control.

Burnout Conditions

The spacecraft velocity and altitude at burnout are determined primarily by the main retro ignition altitude, the spacecraft velocity at main retro ignition, the thrust level provided by the main retro-engine and the vernier engines, the duration for which this thrust is provided, and the spacecraft weight at retro ignition. As shown in figure 13-11, the burnout velocity is the vector sum of the initial spacecraft velocity, the velocity increment resulting from the main retro phase, and the lunar gravity term.

Since the main retro phase velocity increment is supplied to an approximately constant total impulse (provided by the main retro and vernier engines) applied in a direction nearly opposite to the spacecraft velocity at the time of main retro ignition, the absolute value of the main retro phase velocity increment is a function primarily only of the weight of the spacecraft at the start of terminal

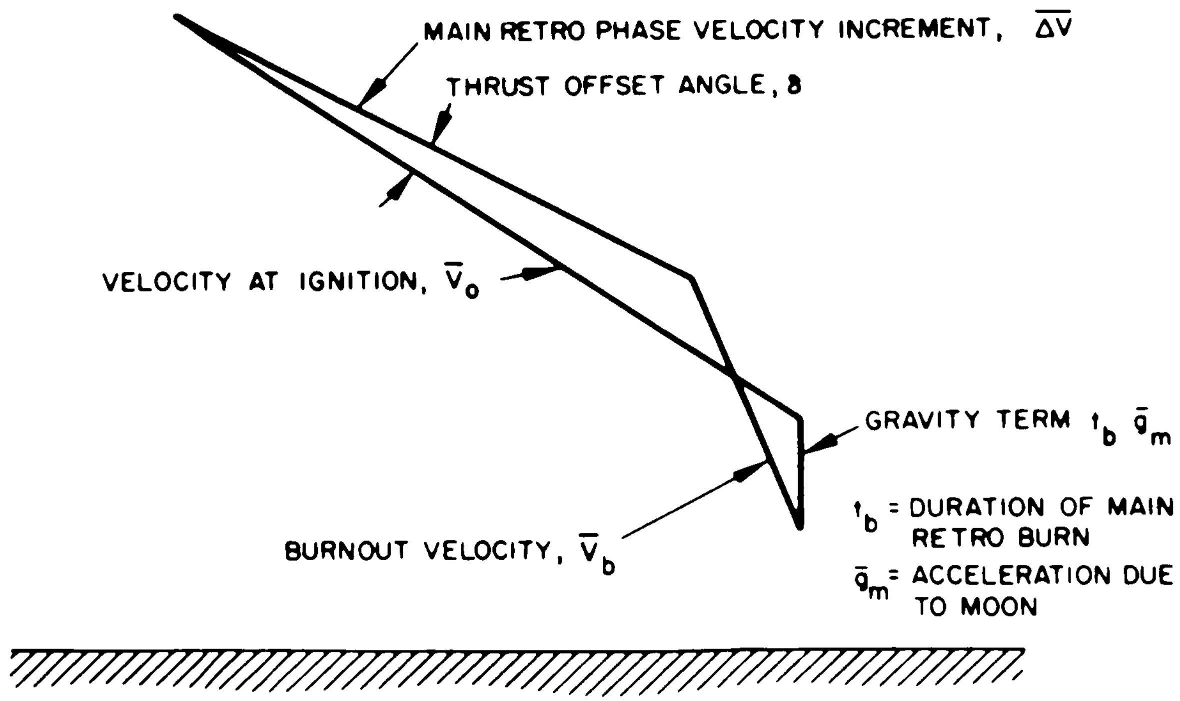


FIGURE 13-11. DETERMINATION OF MAIN RETRO BURNOUT VELOCITY

descent. This weight is determined by the amount of fuel used in making the midcourse correction. Since the spacecraft velocity at retro ignition is essentially determined by the unbraked impact velocity vector, the burnout velocity is thus primarily a function of the unbraked impact velocity and the fuel consumed at midcourse. Main retro burnout nominally occurs on the burnout locus shown in figure 13-12.

The main retro burnout constraints are determined by the requirements of the vernier descent phase which follow. During this phase, it is desired to control the spacecraft to follow the optimum slant-range-velocity contour indicated in figure 13-12. This curve corresponds to a gravity turn with a constant deceleration. If the main retro burns out at a point below the curve, maximum thrust of the vernier engines will not be sufficient to return the spacecraft to the curve, and a hard landing will result. On the other hand, if burnout occurs too far above the curve, there will not be an adequate supply of vernier fuel to effect a soft landing. Along with the nominal burnout locus, typical 3 σ dispersion ellipses for different values of impact velocity and amount of midcourse fuel used are shown.

Control of the spacecraft during the vernier phases is accomplished by the doppler velocity and altitude radars. Therefore the burnout conditions must be within the operational ranges of these sensors. These constraints are also shown in figure 13-12. The doppler radar will not operate within the desired accuracy at velocities greater than nominally 700 fps. In addition, the doppler altimeter will not function at velocities above 850 fps (nominal). Since these radars must function before the vernier descent curve is reached, the burnout velocity should be below these values. The minimum allowable burnout velocity is set by the requirement that the angles which the spacecraft roll axis makes with the vertical and the velocity vector must not exceed 45 and 75 degrees, respectively, together with the assumption that the radars cannot be used reliably while the main retro-engine is burning. The former restriction ensures that the three doppler beams intercept the lunar surface and guarantees that there is a sufficiently high signal-to-noise ratio in the doppler channels. The latter restriction arises from the requirement that the velocity component along at least one of the doppler beams be greater than a certain minimum value so that the spacecraft roll axis can be aligned with the velocity vector following retro burnout.

Since the roll axis is aligned with the velocity vector at burnout, the burnout velocity must also be within 45 degrees of the vertical and 75 degrees of the roll

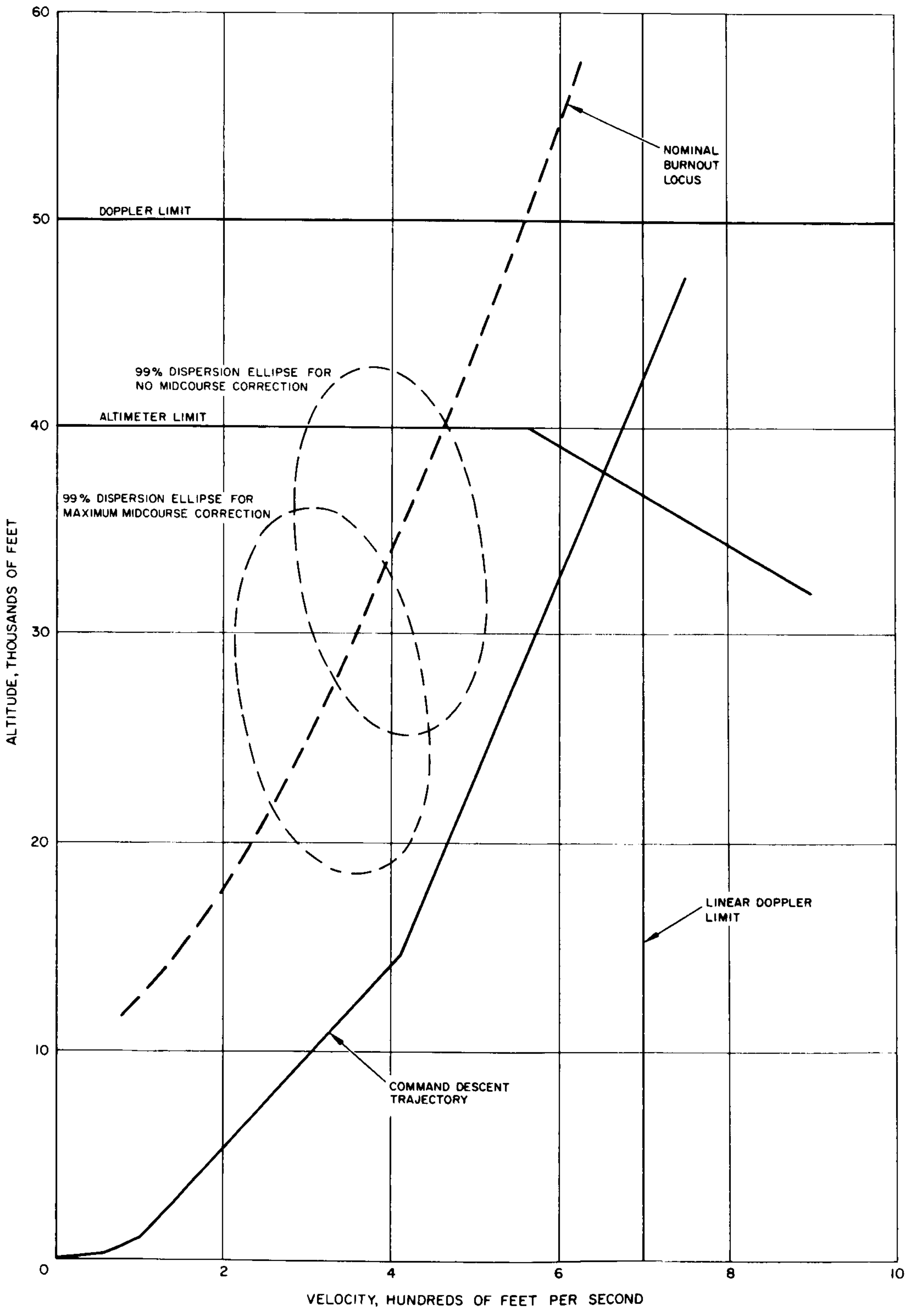


FIGURE 13-12. VERNIER DESCENT PHASE

axis at burnout. The minimum nominal burnout velocity consistent with the above constraint can be minimized by proper choice of the thrust offset angle for the main retro thrust (for vertical descent $\delta = 0$). This yields a nominal burnout velocity of 270 fps for the lowest unbraked impact speed and smallest spacecraft weight at retro ignition. The maximum nominal burnout velocity for the combination of highest unbraked impact speed and greatest spacecraft weight must be chosen so that the 700-fps constraint is not violated. This nominal velocity is typically 525 fps.

Vernier Descent

Following the separation of the main retro-engine from the spacecraft and before obtaining radar reliable (range and/or range-rate) signals and reaching the optimum descent curve (fuel-wise), the vernier engine thrust is servoed to maintain a constant thrust-to-mass ratio equivalent to nominally 0.9 lunar g. If the doppler velocity reliable signal is present, the pitch and yaw attitude control is switched to the doppler velocity reference, and the vehicle attitude is controlled by servoing the components of velocity normal to the vehicle roll axis (V_x and V_y) toward zero.

When the optimum descent trajectory curve is reached, the thrust is controlled to bring the vehicle down the desired preprogrammed range-velocity curve. At an altitude of nominally 1000 feet, the radar altimeter will provide a signal to change the scale factor of the doppler system. At a velocity of nominally 10 fps, the thrust control is switched to the doppler velocity reference. A constant velocity of nominally 5 fps is then commanded, and the pitch and yaw attitude control is switched back to the inertial hold mode.

During the vernier descent phase the spacecraft will transmit PCM data, permitting spacecraft operations to be monitored and providing data for determining the lunar reflectivity.

When the spacecraft reaches an altitude of nominally 14 feet, the vernier engines are shut off automatically by a signal from the radar altimeter to minimize the possibility of the spacecraft stirring up a dust cloud at landing.

Touchdown

With the vernier engine system cut off, the spacecraft will free-fall the remaining 13 feet to the lunar surface. As the spacecraft touches the surface, the tripod landing gear system (with the spring-damper legs) provides stability and, in conjunction with the crushable blocks on the spacecraft body, helps to absorb the kinetic energy of the landing. The spacecraft is designed so that it will not topple on landing (i.e., will settle in an upright position) on the assumed lunar terrain when (1) the vertical velocity is 20 fps or less (See Appendix B, item 4) (2) the limits of lateral velocity are as shown in figure 13-13, and (3) the center-of-gravity location (shown in figure 13-14) is not exceeded. These limits are based on a vehicle radius of gyration of 32 inches or less. Lunar operations will not be impaired by the structural loads imposed under the conditions indicated in figure 13-13 within the landed weight range of figure 13-14, provided (1) the vertical center of gravity is not below spacecraft station 63.48 and (2) the vehicle radius of gyration is 28 inches or more, under the same vertical velocity, slope, and friction coefficient noted below.

The nature of the lunar terrain at the landing site is unknown. The assumed terrain at the landing site is defined as: (1) a slope not exceeding 15 degrees, (2) protuberances 10 cm or less, (3) terrain compressive strength from 50 to 25,000 psi (for landing dynamics and design purposes), and (4) coefficient of friction with the landing gear foot pads of from 0 to 1.0. Surface dust conditions are uncertain, but reasonable means are provided to ensure proper functioning of the spacecraft and components in the presence of dust. Surface compressive strength may be different from that noted but is not included in the design requirements.

LUNAR PHASE

Postlanding Engineering Sequence

The initial period after landing will be utilized to turn off all terminal descent phase subsystems and functions not required during lunar operations. The high electrical power dissipation within the thermally controlled compartments during a normal landing will require immediate turn-off, particularly when the landing occurs under near lunar noon conditions. Engineering data transmissions to verify the spacecraft response to turn-off commands will be accomplished by means of the omnidirectional antennas with the spacecraft transmitter in the high-power mode. The functions to be turned off include: RADVS power; flight control power; approach television camera 4 power and temperature control; and vernier line, fuel, and oxidizer tank thermal control. Commands will then be

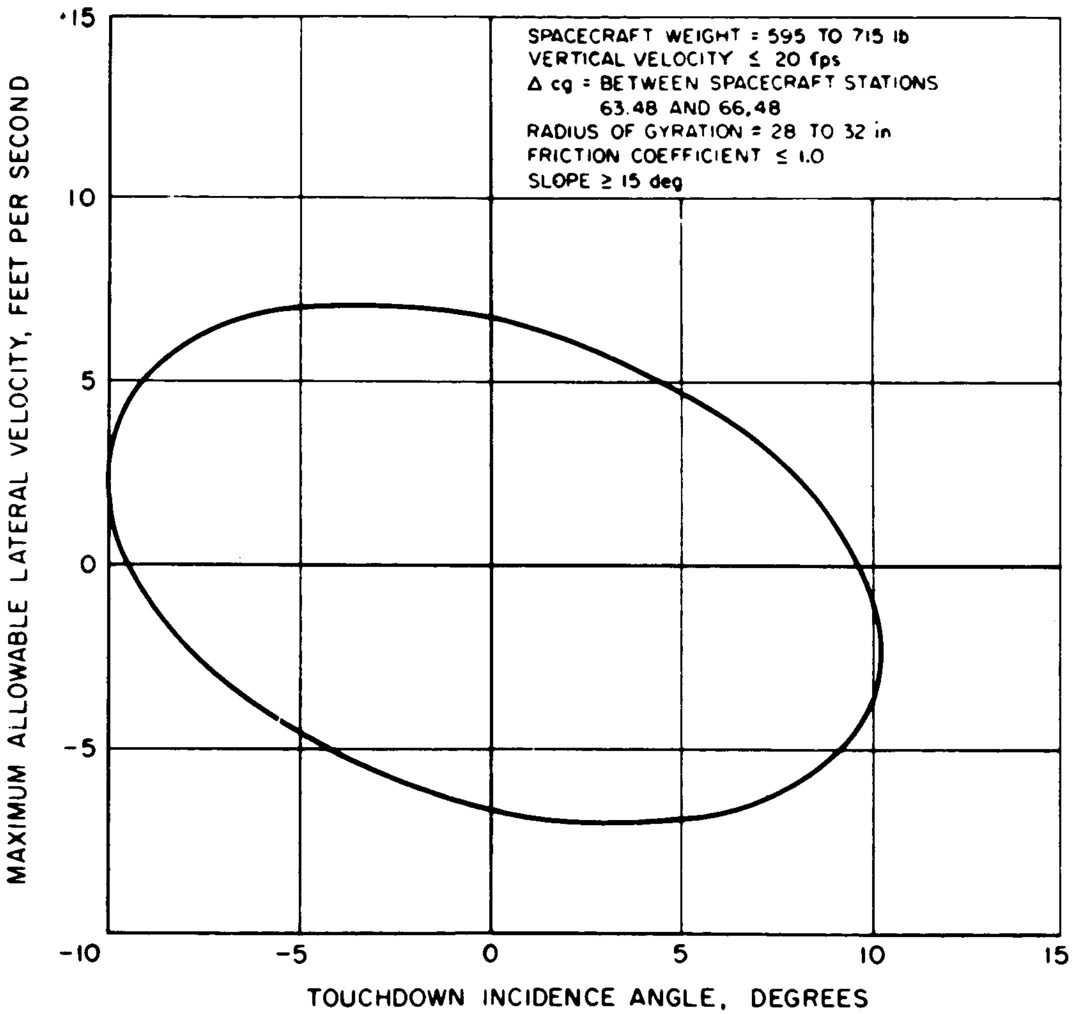


FIGURE 13-13. MAXIMUM ALLOWABLE LATERAL VELOCITY AS FUNCTION OF TOUCHDOWN INCIDENCE ANGLE

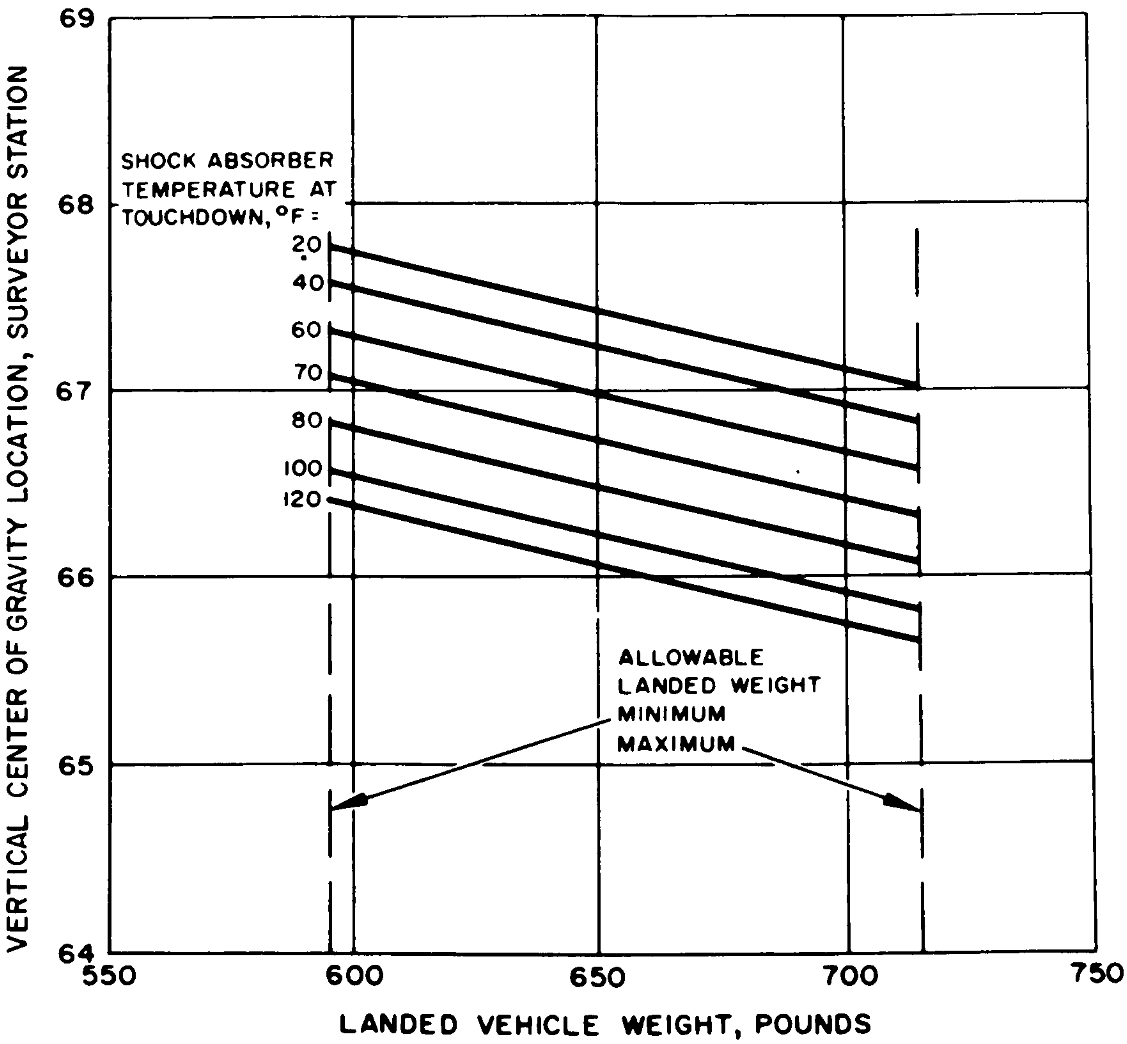


FIGURE 13-14. MAXIMUM ALLOWABLE VERTICAL CENTER OF GRAVITY AS FUNCTION OF LANDED WEIGHT

transmitted to lock the landing gears and dump helium. In a typical case, these operations are expected to be completed in approximately 4 minutes.

The assessment of the spacecraft survival of the landing shock will be conducted to acquire reliability, design, and subsystem operating data for postlanding operational planning. The assessment of touchdown survival may include verification of such functions as operation of the second transmitter, signal processing, planar array and solar panel positioning, solar panel output power, spacecraft temperatures and the condition and response of the thermal control system and electrical power system.

The extent to which the postlanding assessment of the spacecraft is possible or desirable is dependent on solar lighting conditions, Goldstone visibility time remaining after touchdown, and payload operational requirements. A night landing will preclude the solar panel assessment (as well as that of the survey television, which must be delayed until after earth acquisition with the high-gain antenna). A noon landing may reduce the extent of assessment possible by restricting the time available for continuous operation because of the high spacecraft temperatures encountered at that time. If touchdown occurs with the minimum of 3 hours of Goldstone visibility remaining, the evaluation of the relative importance of spacecraft assessment per se, as opposed to the acquisition of initial scientific data by the Goldstone station, may be desirable for the optimum allocation of available time.

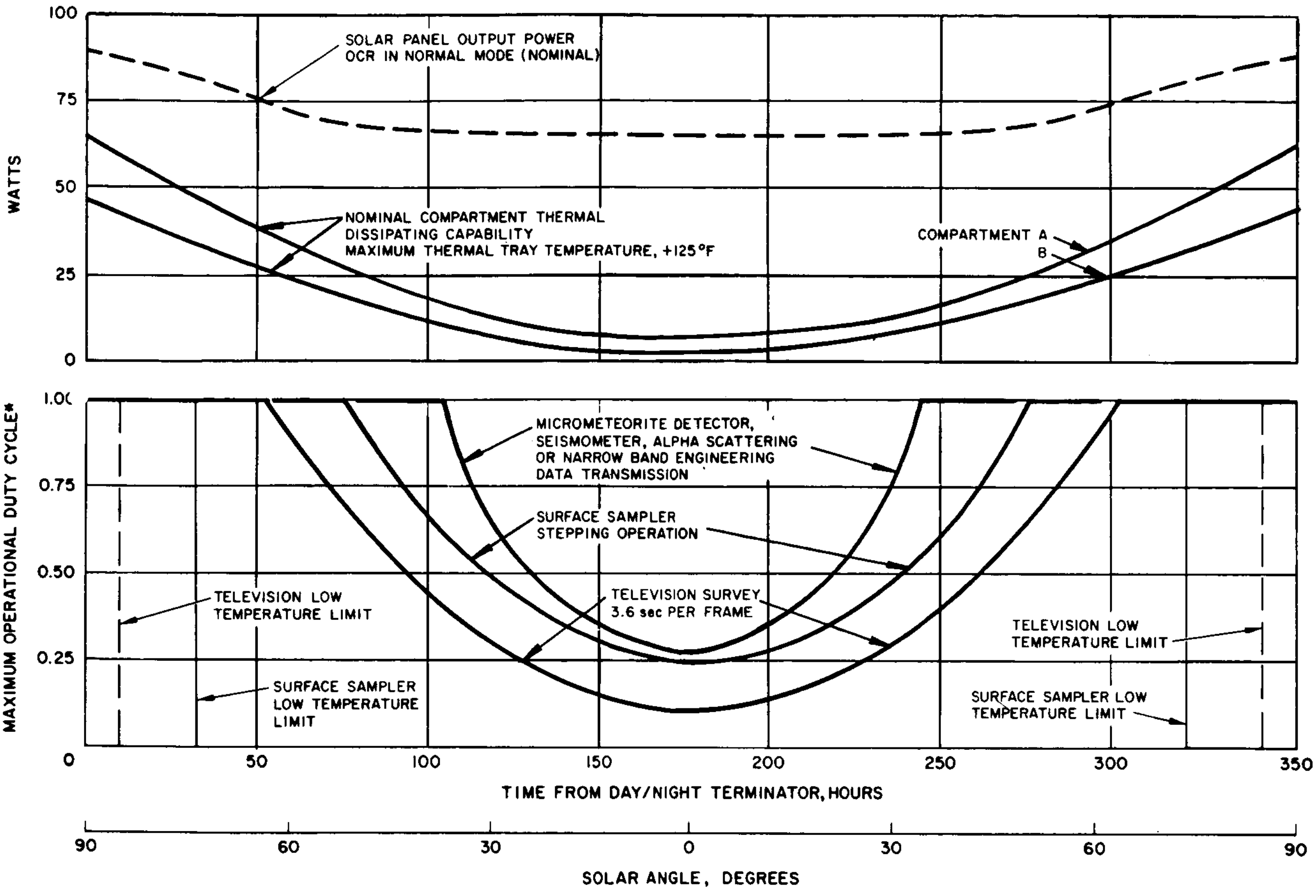
The postlanding positioning of the solar panel and planar array antenna will be accomplished for day landings and is also desirable for night landings. The positioning is required for the case of day landings to assure the necessary telecommunication bandwidth for television transmissions and for the generation of power for lunar day operations and battery charging. In the case of a night landing, it is desirable to position the solar panel to face the direction of the morning sun and the planar array to acquire the earth to provide wide-band telemetery capability. The procedures utilized and time required for the initial positioning depend on the lunar environmental conditions and, if the landing occurs near lunar noon, on the spacecraft thermally controlled compartment temperatures existing at touchdown. The angular acquisition of the earth will be accomplished by conducting a search pattern with the planar array. The search pattern required will be restricted for near day terminator landings by determining spacecraft roll attitude from the solar panel data and for night or near noon landings by the assumption of a most probable roll attitude.

After initial positioning, repositioning of the solar panel to follow the changing solar incidence angle and the planar array to correct pointing errors resulting from lunar libration must be accomplished at periodic intervals. Repositioning approximately twice per earth day during the lunar day is anticipated, although it may be desirable to position the antenna at specific times, such as prior to television operation, to obtain optimum video signal-to-noise ratio.

When the landing site is known, spacecraft attitude may be determined from two celestial bodies known in spacecraft coordinates. For day landings, the spacecraft will provide sun and earth positions in spacecraft coordinates from the solar panel and planar array positions and star positions from survey television frames with their corresponding mirror positions. For the case of a night landing, the survey cameras may be inoperable because of their low ambient temperatures and earth position may be available only if earth acquisition by the planar array is successfully accomplished.

Spacecraft Lunar Day Operational Capability

The lunar day operational capability of the Surveyor A-21A configuration is shown in Figure 13-15. The nominal solar panel output as a function of time or solar angle is shown for the case of the optimum charge regulator in the normal mode of operation and represents the maximum power level which at various times may be available to the spacecraft. The amount of power actually generated is limited by the capability of the thermally controlled compartment to dissipate the heat generated by the optimum charge regulator compartment. The degree to which the available solar panel power is utilized depends on the mission sequence of operation and is limited by the capability of the thermally controlled compartments to dissipate the heat generated by the optimum charge regulator in compartment A and the spacecraft subsystems located in both compartments A and B. In general, with a lunar day landing, sufficient solar energy is available to conduct all desired operations within the constraints imposed by the compartments. The single exception would be a landing near the day/night terminator, in which case sufficient time may not be available to fully charge the battery before the start of the lunar night. The capability of the thermally controlled compartments is shown in terms of watts of heat that may be dissipated under thermal tray



*SPACECRAFT THERMALLY CONTROLLED COMPARTMENT CAPABILITY ONLY;
EXTERNAL SENSORS NOT INCLUDED EXCEPT AS SPECIFED

FIGURE 13-15. LUNAR DAY OPERATIONAL CAPABILITY FOR EQUATORIAL LANDING

temperature conditions of +125° F and as such represents the maximum capability under the compartment standard environmental conditions. The environment assumed includes the lunar environment as well as the influence on the compartment radiator surface by the solar panel and planar array positions and temperatures.

The complement of scientific instruments consists of the following:

(a) Survey television (2)

(b) Soil mechanics surface sampler

(c) Alpha scattering instrument

(d) Micrometeorite detector

(e) Seismometer

The operational capability of the above instruments is shown in terms of the maximum steady-state cycling which is permitted within the constraints imposed by the heat dissipating capability of the thermally controlled compartments. The maximum allowable thermal tray temperature is 125° F; therefore, maximum heat dissipation by the compartments as well as maximum operation of components internal to the compartments occurs at a thermal tray temperature of 125° F.

During the On period of the instrument cycle the spacecraft is assumed in the normal mode of operation as follows:

(a) Optimum charge regulator in normal mode

(b) Transmitter on (either wide band or narrow band as required)

(c) Transmitter connected to planar array

(d) Command receivers on (2)

(e) Engineering signal processor and central signal processor on as required

(f) Boost regulator on

(g) Instrument electronic and auxiliary units on as required

During this period the compartment thermal tray temperatures are slightly below 125° F and rising. During the Off period of the instrument cycle only the command receivers are assumed to be operating with the resulting decay in the thermal tray temperatures to slightly less than 125° F.

At lunar noon, the operation of the survey television in the mapping mode (3.6 seconds per frame) typically results in thermal dissipations of approximately 36 watts and 11 watts in compartments A and B, respectively. During the Off or compartment cooling period approximately 3 watts are dissipated in compartment A by the command receivers and no dissipations occur in compartment B. Therefore, in order to limit the average heat dissipation in compartment A to 6 watts and in compartment B to 4 watts the operation of the mapping mode is limited to a duty cycle of about 10 percent or 6 minutes per hour.

The operation of the micrometeorite detector, alpha scattering instrument, and seismometer is shown in figure 13-15. The above instruments may be operated either individually or simultaneously at essentially the same duty cycle. The principal operating mode of the surface sampler, the continuous stepping of one of the four drive axes, is shown in figure 13-15. The operation in the accelerometer mode although requiring wide-band telemetry is only of short duration and is therefore not shown.

The spacecraft capability shown in figure 13-15 is subject to a number of variables. The first of these is the operating mode of the optimum charge regulator. It was previously stated that the regulator was operated in the normal mode during the On period and turned off during the instrument Off period. A tradeoff may be made, however, in battery charging versus data transmission. If the charge regulator is turned off or operated in the bypass mode during transmission, the reduction in the heat generated in compartment A would, if compartment A were the limiting compartment, allow increased transmission time. If battery charging is important during the period when the indicated duty cycle is less than 100 percent, the battery may be charged during portions of the off cycle (thereby causing added thermal dissipation within the compartment) with a resulting decrease in the allowable transmission time. Other variables affecting the illustrated capability include optimum charge regulator variations due to the dispersions associated with the solar panel output power, variations in the dissipations by spacecraft electronics, the optimum charge regulator and the boost regulator due to variations in the battery terminal voltage, and variations in the actual compartment dissipating capability due to lunar environment and compartment thermal parameters. The above variables are potentially capable of significantly affecting the illustrated capability during the noon interval.

Additional capability is available to optimize the lunar sequence of operation. At touchdown, compartment temperatures in the typical case will be less than the allowable maximum. The compartment thermal capacity due to its mass will thus provide a limited transient capability for continuous spacecraft operation until the maximum tray temperature of 125° F is reached. In the normal case, with a noon landing, the postlanding engineering assessment and the initial television survey may be completed before thermally restricted operation as indicated in figure 13-15 is required. During the region of restricted operation, the nonoperation of equipment would allow the accumulation of additional transient capability which may be used for extended transmissions at some desired time.

A tradeoff may also be made in the utilization of the planar array or solar panel. Since the above represent heat sources during the noon interval, the reduction of the panel temperatures could result in an increase in compartment heat radiating capability. Although it may be undesirable to alter the planar array position, the positioning of the solar panel off-normal to the sun to lower its temperature and yet retain a minimum output could provide additional compartment thermal dissipation capability if desired.

Spacecraft Lunar Night Operational Capability

The transit energy deficit, nominally 1000 watt-hours will result in an expected battery state of charge of 2800 watt-hours at a temperature of 70° F. During lunar night operation compartments A and B are controlled at 50° F and 10° F, respectively, to minimize heat losses. Table 13-8 presents a summary of compartment lunar night heat losses. In order to maintain the compartments at the above temperatures, a nominal 6.8 watts and 3.9 watts of heat must be dissipated in compartments A and B. The battery discharge at a temperature of 50° F and an average load of less than one-half ampere will reduce the available energy remaining for lunar night operation to about 2375 watt-hours. If the landing occurs during the day such that the battery is fully charged before the day/night terminator a nominal 3375 watt-hours of energy will be available for the night operations.

The heat dissipation required for the maintenance of compartment temperatures may be provided by heaters or, more efficiently from an operational standpoint, by the operation of the telecommunication system to acquire scientific or engineering data. The lunar night capability under various operating conditions is presented in figure 13-16. If the maintenance of communication or the survival heating of the survey television is not required, additional spacecraft night survival capability may be obtained by taking advantage of the compartment thermal capacities. The complete discharge of the battery at the above control temperatures is followed by the slow decay of compartment temperatures. The maximum extended survival time should be limited to the interval required for compartment A to decay to 0° F, the minimum allowable battery temperature.

For the case of a night landing it is desirable to allocate a small portion of the available energy for the assessment of the spacecraft touchdown survival. A nearly complete assessment of the spacecraft touchdown survival is expected.

TABLE 13-8. A21-A COMPARTMENT LUNAR NIGHT HEAT LOSS SUMMARY

Heat Path	Model A21-A Heat Loss ~ Watts	
	Compartment A* at 50°F	Compartment B** at 10°F
Basic compartment less wiring harness		
Supports	1.52	0.75
Thermal switches	1.13	0.67
Mylar super insulation	2.32	1.38
Thermal tunnel	0.76	0
Wiring harness		
Basic bus		
Low conductance inserts	0.34	0.68
3 coax cables low conductance	0.26	0
Tear strip (34 strips)	0.16	0
Scientific payload ***		
TV cameras 2 and 3	0.16	0.06
Surface sampler	0	0.16
Seismometer	0	0.06
Alpha scattering	negligible	0.09
Micrometeorite detector	0.16	0
Total compartment losses	6.8 ± 0.5	3.9 ± 0.25

*Based on design shown on drawing 261214 revision E.

**Based on design shown on drawing 261240 revision E.

***Assumes low conductance wiring harness.

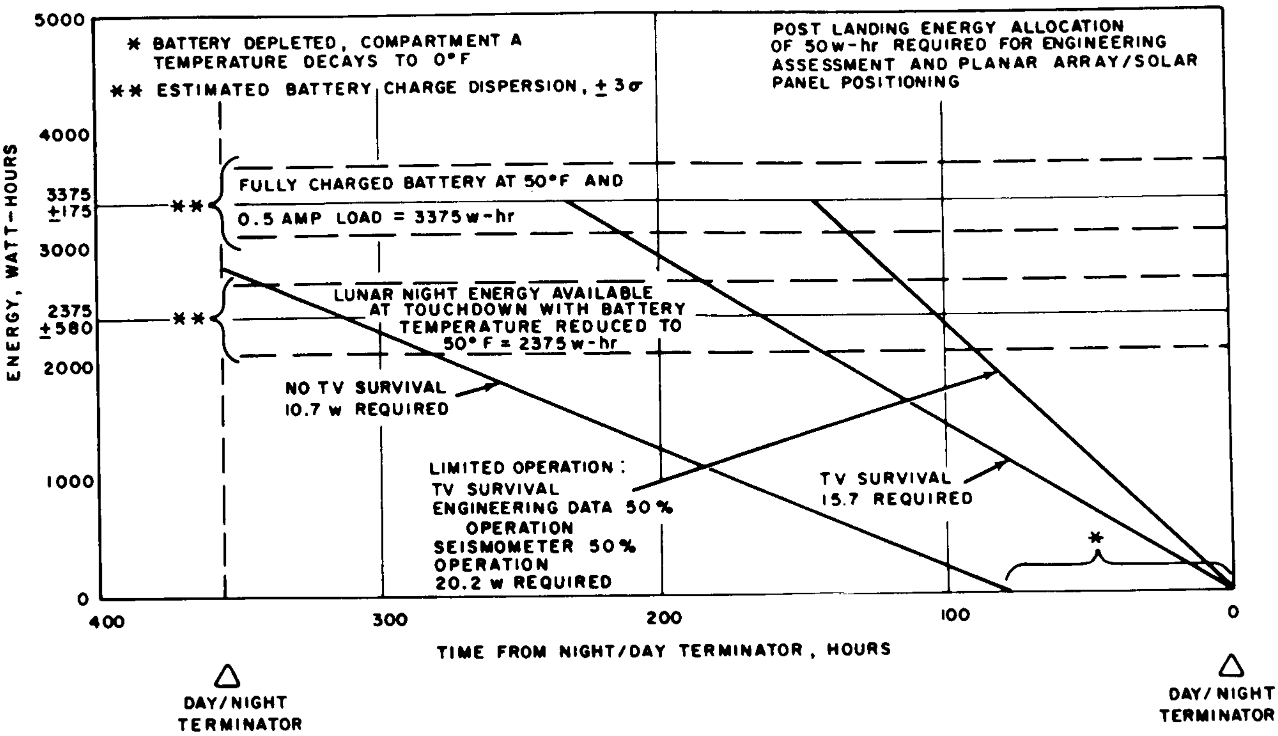


FIGURE 13-16. LUNAR NIGHT SURVIVAL CAPABILITY

It is desirable to acquire the earth with the high gain antenna for the assessment of the wide-band transmission capability. The survey television may be inoperative after touchdown, however, because of the low environmental temperatures of the lunar night.

Hypothetical Sequence of Operation

A hypothetical sequence of operation for the case of an equatorial landing is shown in figure 13-17. The touchdown survival evaluation and the sun and earth acquisition by the solar panel and planar array are completed during the first hour with the initial television survey completed in the second hour. Although the heat dissipated in the thermally controlled compartments exceeds the maximum steady-state compartment dissipating capability, the transient thermal capabilities of the compartments in the nominal case are expected to permit completion of the initial 2 hours of operations within the thermal tray temperature constraint of 125° F.

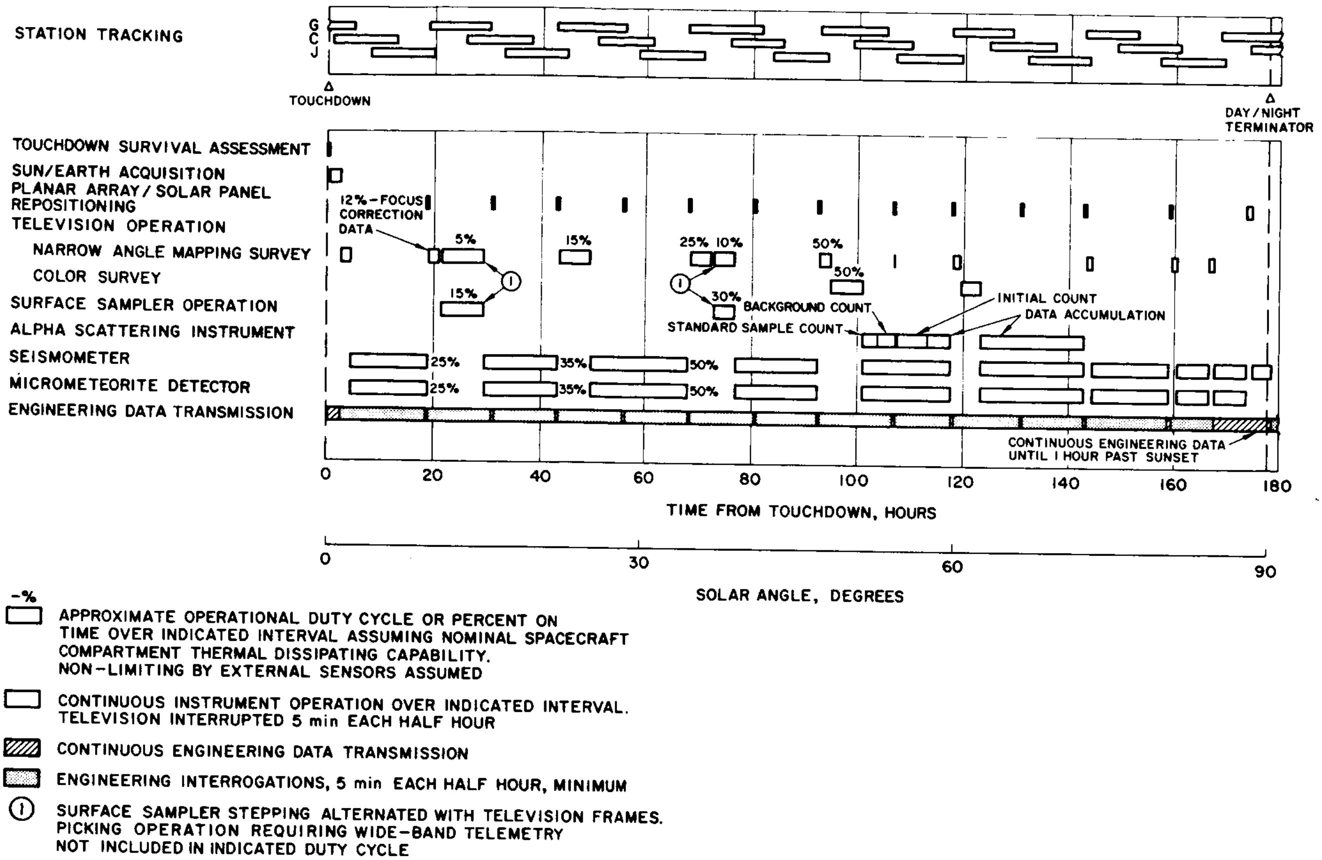


FIGURE 13-17. HYPOTHETICAL LUNAR PHASE SEQUENCE—EQUATORIAL LANDING ASSUMED

Operations during the remainder of the lunar day are shown as constrained by the compartment dissipating capability indicated in figure 13-15.

Assuming that the television camera operating capability, established by its thermal characteristics, exceeds the thermal constraints imposed by the thermally controlled compartments during the near noon interval, black and white mapping surveys of approximately 1000 frames each, at 3.6 seconds per frame, may be conducted by predetermined command tape. Two complete mapping surveys with each of three color filters are included. The television mapping operation is interrupted to monitor the spacecraft status and to allow thermal cooldown of the compartments.

If sufficient telecommunication bandwidth is available the seismometer and micrometeorite detector will be operated continuously, interrupted only for compartment thermal cooling as required and during the wide-band transmissions required by other scientific instruments. It is assumed that the sensors do not constrain operation during the noon interval.

The operation of the soil mechanics surface sampler is conducted in two modes. Scraping, digging, and mapping of the lunar surface is conducted in a narrow-band telecommunications mode with the stepping alternated with television frames in the wide-band mode as required. The picking operation is conducted in the wide-band telecommunications mode with television viewing used for remote control.

The 167° F maximum nonoperating temperature limit of the alpha scattering instrument sensor, together with its limited available view factor to space, could result under worst case conditions in potential permanent damage to the sensor if allowed to remain in the stowed position after a touchdown at lunar noon. Since the 3-hour standard sample count is conducted in the stowed position and is limited to a maximum interruption of 30 minutes the deployment of the instrument will preclude the acquisition of standard sample data. It is assumed that the above worst case situation must be alleviated by thermal design if additional thermal studies of the sensor head and its environment confirm the existence of an actual problem. Therefore this worst case condition is not considered in the illustrated typical sequence of operation. In order to conform to the instrument maximum interruption constraint, the alpha scattering experiment is not initiated until continuous spacecraft operation is permitted by the thermally controlled compartments.

During the lunar day and night additional operations are required to support the requirements of the scientific payload. The assessment of touchdown survival will provide data necessary for the operational planning associated with the scientific payload. The positions of the sun, earth, and stars in spacecraft coordinates obtained during the initial acquisition period and television survey will provide data necessary to compute spacecraft attitude. Engineering interrogations are provided each half hour to monitor spacecraft status. Repositioning of the solar panel and planar array should be performed at intervals of about 12 hours, although if the rate of lunar libration is less than maximum, repositioning of the planar array may be conducted at less frequent intervals. The solar incidence angle on the solar panel is thereby maintained at less than 3.5 degrees and the planar array positioning error may be maintained at less than 1 degree. In an optimum sequence the final positioning of the solar panel prior to the start of lunar night causes it to face in the direction of the norning sun, with a full battery charge achieved at the time of final positioning.

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APPENDICES

Appendix A lists specifications that establish design requirements, configuration, performance, and function of the Surveyor Spacecraft, Model A-21A. The list is divided into two major categories, namely, the basic bus and the scientific payload. These are further divided into the subsystems under which the specifications are listed.

Appendix B lists characteristics data for Surveyor Spacecraft, Model A-21A. These are listed under subsystems to which they apply. All items are for informational reference only, except for those that appear in the preceding sections.

Appendix C gives the physical and performance differences which exist between the A-21 and A-21A spacecraft.

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APPENDIX A. DOCUMENT AND SPECIFICATION LIST

BASIC BUS:

Structural and Vehicular Subsystem

Item	Document Number	Document Name
1	DS 230029	Spaceframe Vehicle Design Configuration
2	DS 230003	Spaceframe and Unit Integrating Structure
3	DS 238900	Landing Gear Requirements
4	DS 238901	Landing Gear Shock Absorber Column
5	DS 238904	Landing Gear Crushable Body Blocks
6	DS 236130	Antenna/Solar Panel Positioner
7	GS 226100	Electromagnetic Interface Specification, Surveyor Spacecraft and Associated Equipment
8	CS 239506	Spacecraft Design Parameters, Grounding, Bonding, and Shielding
9	DS 238704	Spacecraft Thermal Control, Compartment A
10	DS 238706	Spacecraft Thermal Control, Compartment B

Engineering Instrumentation Subsystem

Item	Document Number	Document Name
11	CS 988653-2 CS 988654-10	Temperature Sensors
12	DS 239004	Accelerometers
13	DS 239004	Engineering Measurement Sensors, Accelerometer - Amplifier System

Propulsion Subsystem

Item	Document Number	Document Name
14	PS 238611 and E155-62	Main Retro Rocket Engine Thiokol Spec, Main Retro-Rocket Engine
15	DS 238666 and PS 238610	Surveyor Vernier Propulsion System
16	PS 262526	Safety and Arming Assembly, Main Retro Rocket Engine
17	DS 234675	Roll Actuator

Electrical Power Subsystem

Item	Document Number	Document Name
18	239513	Power and Thermal Dissipation by Functional Event
19	DS 237601 and PS 237787	Solar Panel Assembly
20	PS 237790	Solar Panel Solar Cell Module
21	PS 237901	Battery
22	DS 231631	Battery Charge Regulator
23	DS 231632	Boost Regulator

Telecommunications Subsystem

Item	Document Number	Document Name
24	239511	Spacecraft Command Assignments for Scientific Payload
25	231681	Decibel Allocation and Margin Summary
26	DS 231706	Spacecraft Telecommunications Transmitters
27	DS 231707	Receiver Transponders
28	DS 231718	Telecommunication Antennas and R-F Switching
29	DS 231641	Central Decoding Unit (Central Command Decoder)
30	DS 231711	Engineering Signal Processor
31	DS 231710	Central Signal Processor
32	DS 231616	Low Data Rate Auxiliary
33	DS 231613	Signal Processing Auxiliary
34	239512	Spacecraft Data Channel Control for Scientific Payload

Flight Control Subsystem

35	DS 234600	Flight Control Sensor Group
36	DS 234630	Inertial Reference Unit
37	DS 234622	Primary Sun Sensor
38	DS 234636 and PS 234661	Inertia - Burnout Sensor Switch
39	PS 234621	Canopus Sensor
40	DS 234610	Flight Control Electronics Unit

Flight Control Subsystem (Cont)

Item	Document Number	Document Name
41	DS 234623	Secondary Sun Sensor
42	DS 232601	Altitude Marking Radar
43	PS 232902	Radar Altimeter and Doppler Velocity Sensor (RADVS)
44	PS 234641 PS 234640	Attitude Jet System - Gas Tank Gas Jet Attitude Controls
45	DS 231621	Approach Television Camera

SCIENTIFIC PAYLOAD:

Item	Document Number	Document Name
46	DS 231622	Survey Television Camera-Spacecraft Telecommunication Television
47	FR 231611	Alpha Scattering Instrument Auxiliary
48	FR 231612	Micrometeorite Ejecta Detector Instrument Auxiliary
49	FR 231715	Sample Processor Auxiliary
50	FR 231716	Soil Mechanics Instrument Auxiliary
51	FR 239319	Alpha Scattering Experiment
52	FR 239320	Micrometeorite Ejecta Detection Experiment
53	FR 239323	Soil Mechanics Experiment
54	FR 239334	Seismological Experiment
55	FR 239335	Seismometer Auxiliary
56	FR 239389	Soil Mechanics-Surface Sampler Auxiliary

Television Subsystem (Cont)

Item	Document Number	Document Name
57	FR 239390	Soil Mechanics-Surface Sampling Experiment
58	IS 239302	Micrometeorite Ejecta Detector
59	IS 239303	Alpha Scattering Experiment
60	IS 239330	Single Axis Seismometer
61	IS 239381	Soil Mechanics-Surface Sampler

APPENDIX B. CHARACTERISTICS DATA

System

Item Number	Item	Data
1	Spacecraft Injected Weight	2221.27
2	Transit Time (Nominal)	66 hr
3	Midcourse ΔV at 20 hours	30 meters/sec
4	Soft Landing Capability (60 lb payload)	15 fps (vertical); 5 fps (lateral)
5	Lunar Oblique Approach Capability	25 deg
6	Energy Removal	
	Main Retro Engine in Inertial Hold	95 percent
	Vernier System in Gravity Turn Trajectory	5 percent
7	Midcourse and Terminal Control	Ground tracked radio command
8	Attitude Control	Inertially stabilized with sun-star reference and strapped down gyro system

Flight Control

Item Number	Item	Data
9	Gyro	Kearfott 2514 α series, 10 deg input axis freedom, single degree of freedom, floated

Item Number	Item	Data	
9 (Cont)	Angular Momentum	100,000 gm/cm^2/sec	
	Motor	3-phase, synchronous, 2.6 $\pm$0.3 watts	
	Signal Generator	Electromagnetic, ac, $\pm$10 percent linearity over 10 deg	
	Signal Generator Scale Factor at 180^{o}F	0.044 $\pm$5 percent v/deg	
	Torque Generator	Permanent magnet, four pole, dc	
	Maximum Torque Rate at 100 ma	80,000 deg/hr	
	Torque Scale Factor (Calibrated at 1800 deg/hr = 0.5 deg/sec	Input Angle OFF Null (Deg)	Max. Deviation (Percent)
		0 to 2	0.05
		2 to 4	0.2
		4 to 10	1.0
	Torque Linearity	Torque Rate (deg/hr)	Percent of 800 deg/hr/ma
		0 to $\pm$1,800	0.5
		1,800 to 10,000	2.0
		10,000 to 80,000	5.0
	Drift		
	Non-Acceleration Sensitive	$\pm$1 deg/hr	
	Acceleration Sensitive	$\pm$2 deg/hr	
	Anisoelastic (from 500 to 700 cps)	0.02 deg/hr/g^2 (rms)	
	Elastic Restraint	$\pm$0.6 deg/hr/deg off null	
	Heater Power		
	Warmup	50 $\pm$5 watts	

Flight Control (Cont)

Item Number	Item	Data
9 (Cont)	Operate	10 ±0.5 watts
	Maximum Weight per Gyro	1.3 lbs
10	Accelerometer	Conner 431 OF linear force balance
	Weight	8 oz
	Range	0.01 to 0.5 g
	Scale Factor	30 v/g
	Accuracy from 0.01 to 0.16 g	±0.001 g (30) (about 1 percent full scale)
11	Canopus Sensor	1P21 tube, S4 response
	Field of View	5 x 7.2 deg at Canopus magnitude
	Adjustment of Center 5 deg Field	±15 degrees
	Null Offset	±0.1 deg
	Lockon Threshold	±0.45 magnitude from Canopus
	Power	5 watts
	Linear Region	±2 deg
	Weight Max.	4 lb
	Scale Factor	2 v/deg
12	Attitude Jet System	
	Nitrogen Weight	4.5 lb ±0.1 lb
	Thrust Level	0.057 lb +30 percent, -10 percent
	Specific Impulse	63 sec

Telecommunications

Item Number	Item	Data
13	Operating Frequencies	2295 mc (transmitter), 2213 mc (receiver)
14	Transmitter Power	0.1 and 10 watts
15	Transmitter Long-Term Stability	
	Narrow Band PM or FM	±20 ppm
	Wide Band FM	±60 ppm
16	Planar Antenna	27 ±1/2 db, 6.2 x 7.5 deg to 3 db point
	First Sidelobe	About 12 deg, down -12 to -14 db from maximum gain
	Theoretical First Sidelobe	13.2 db
	Polarization	Right-hand circular
17	Omnidirectional Antenna	
	Gain for Both	≥ -10 db, all angles, ≥ -7.5 db for 40% of angles per omni
	Polarization	Right-hand circular
18	Receiver	
	Noise Figure	13.0 ±1.0 db
	Threshold Signal-to-Noise Ratio	10.5 ±1.5 db
	Sensitivity at Threshold	
	Commands	-110 dbm
	Carrier	-128 dbm
	Noise Bandwidth	
	Commands	13 kc
	Carrier	470 kc

Telecommunications (Cont)

Item Number	Item	Data
18 (Cont)	Power (Per Receiver)	1.41 watts
	Required Dynamic Range	75 db
19	Transmitter	Traveling wave tube
	Modulation Frequencies	1 cps to 220 kc
	Maximum Frequency Deviation	±1.5 mc
20	Altitude Marker Radar	Pulsed magnetron 1.5 kw peak
	Frequency	9300 mc
	Pulse Duration (50 percent amplitude)	3.15 usec
	Pulse Repetition Rate	350 pps
	Sidelobes	Down 10 db or greater
	Antenna	Parabolic, 30 in.
	Detection system	Double gate
	Total Power (gross)	75 watts
	Accuracy at 60 Miles	
	Vertical Approach	±0.3 miles
	45 deg Approach	±1.0 miles
21	RADVS	Klystron, continuous wave, 2 watts
	Doppler Velocity Sensor Frequency	13,300 mc
	Altimeter Frequency	12,900 mc
	Antenna	One-half of 18 inch parabolic dish per beam

Item Number	Item	Data
21 (Cont)	✓ Velocity Capability	
	Velocity	0 to 3,000 fps
	Slant Range	0 to 50,000 ft
	Linearity Range	0 to 700 fps ±300 fps
	/ Altimeter Capability	14 to 40,000 ft
	Total Power (gross, including conversion efficiency)	590 watts
	Velocity Accuracy	$\pm\sqrt{1 + (2\%\ v_{total})^2}$ fps
	Altitude Accuracy	$\pm\sqrt{4^2 + (5\%\ \text{true range})^2}$ feet

Spaceframe and Landing Gear

Item Number	Item	Data
22	Spaceframe	
	Members	Thin-wall tubular, 7075 aluminum
	Joints	Machined, 7075 aluminum
	Assembly	Rivets
	Weight	
	Basic Structure	59.87 lb
	Integrative Structure	22.54 lb
23	Landing Gear	Tripod arrangement with shock absorber and damper
	Length	
	z-axis to Center of Footpad	77 in.
	Hinge to Center of Footpad	40.5 in

Spaceframe and Landing Gear (Cont)

Item Number	Item	Data
23 (Cont)	Weight	37.7 lb
	Crushing Force for Three Crushable Blocks	6000 lb

Thermal Control

Item Number	Item	Data
24	Equipment Compartment Thermal Switches	
	Conductance	
	Open	0.001 Btu/$^{\circ}$F/hr
	Closed (over switch closure temp.)	0.5 Btu/$^{\circ}$F/hr

Electrical Power Sources

Item Number	Item	Data
25	Solar Panel	
	Cell Type	p-n
	Panel Size	30 x 47 in.
	Cell Arrangement	Series-parallel - nine major groups each with 4 strings of 22 series-connected cells with equipotential points parellel-connected
	Nominal Transit Power Output at 50°C	89 watts
	Maximum Weight	8.5 lb
26	Battery	Silver oxide/zinc with potassium hydroxide

Electrical Power Sources (Cont)

Item Number	Item	Data
26 (Cont)	Nominal Terminal Voltage	17.5 to 27.3 volts
	Storage Capacity	
	1/2 Ampere Load at 50°F	3375 w-hr
	1 Ampere Load at 70°F	3800 w-hr
	Circuit	14 sealed, manifolded cells in series
	Weight	46.4 lb

Propulsion

Item Number	Item	Data
27	Vernier Propulsion System	
	Oxidizer	Hypergolic, with nitrogen tetroxide (N_2O_4) with 10 percent nitric oxide (NO)
	Fuel	Monomethylmonohydrate (•72 MMH •28 H_2O)
	Throttle Range	30 to 104 lbs
	Tank Capacity	
	Recommended with 1.5 oxidizer-to-fuel ratio	170.2 lb
	Maximum	179.6 lb
	Required Total Impulse	50,000 lb-sec
	Nozzle	Molybdenum, radiation-cooled, 86/1 exp
28	Main Retro-Rocket Engine	36-in. diameter, spherical
	Propellant	Solid, with 86 percent TF-N-3062 polymer

<u>Propulsion (Cont)</u>

Item Number	Item	Data
28 (Cont)	Average Chamber Pressure	511 psi
	Average Thrust	8800 lbs at 50°F
	Dry Weight, Including Insulation	154 lb
	Expansion Ratio	53 to 1
	Maximum Temp. Sensitivity	0.12%/°F
✓	Nominal Burning Time, 50°F	40.5 sec

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APPENDIX C. DIFFERENCES BETWEEN A-21 AND A-21A SPACECRAFT

PHYSICAL DIFFERENCES

The physical differences consist of minor variations in the basic bus configuration of each vehicle and significant differences in the payload configuration. These differences are reflected in a table of component differences (Table C-1) and a table of weight differences (Table C-2). Figure C-1 shows the placement of the difference components on the two spaceframes.

The scientific payload items of A-21A Surveyor spacecraft as listed in Table C-1 are designed to be individually removable from the A-21A spacecraft. To mechanize this concept, each payload item has an individual wire harness. An individual connector is installed on the compartment B wire harness for payload item interconnection. The A-21 spacecraft does not provide this removal concept and the associated mechanization for engineering payload items.

PERFORMANCE DIFFERENCES

The operation and performance differences between Surveyor spacecraft A-21 and A-21A include lunar night survival capability as a function of battery power, payload removal concept, data signal commutator modes available, thermal dissipation characteristics, and transit and landing correction capability.

Electrical power available to the spacecraft after touchdown is the determining factor in length of time the spacecraft will survive after lunar day-night terminator.

Electrical power available immediately after touchdown, at a battery termperature of 70°F:

A-21	1900 watt-hours*
A-21A	2800 watt-hours

*Assumes 1000 watt-hours available from the auxiliary battery.

TABLE C-1. COMPONENT DIFFERENCE, A-21 TO A-21A

Deleted from A-21*	Added to A-21A
Engineering payload Seven strain gages and amplifiers Eleven thermal sensors Four accelerometers and amplifiers Two current shunts Auxiliary battery and control Auxiliary engineering signal processor	Scientific payload Alpha scattering instrument subsystem Survey television camera 2 instrument subsystem Soil mechanics surface sampler instrument subsystem Micrometeorite detector instrument subsystem Seismometer instrument subsystem
Thermal Sensors Compartment B thermal switch (inside) PA/SP mast base	Thermal Sensors Three in compartment B ~~Three additional thermal switches on compartment B~~ Electrical harnesses associated with scientific payload experiments Lunar night electrical harness disconnect Thermal inserts and increased wire size for harness wiring in series with nichrome thermal inserts
*Components that will not appear on A-21A	

TABLE C-2. WEIGHT DIFFERENCES OF A-21 AND A-21A*

Item	A-21 Weight	A-21A Weight
Basic bus	697.18	698.81
Spacecraft vehicle	213.14	216.65
Thermal compartment B	17.59	18.06
Wire harness, basic bus	41.20	42.78
Engineering measurement sensors	4.46	4.44
Propellant, usable	1383.50	*
Vernier propellant	162.30	*
Retro-rocket propellant	1211.00	*
Engineering payload	61.95	. . .
Scientific payload	. . .	*

*See Table 11-4 in Section 11.

Electrical power available after touchdown, at a battery temperature of 40°F:

A-21 1300 watt-hours* (Main battery only)

A-21A 2375 watt-hours

Lunar night power consumption rate:

A-21 59 watts

A-21A 10.7 watts**

Hours of operation into lunar night:

A-21 22 hours

A-21A 222 hours

*Energy available from main battery assuming 750 watt-hours supplied by auxiliary battery during transit.

**Difference due to thermal insulators in electrical cables from thermal compartments and effect of lunar night disconnect.

Data signal modes are different for the A-21 and A-21A spacecraft:

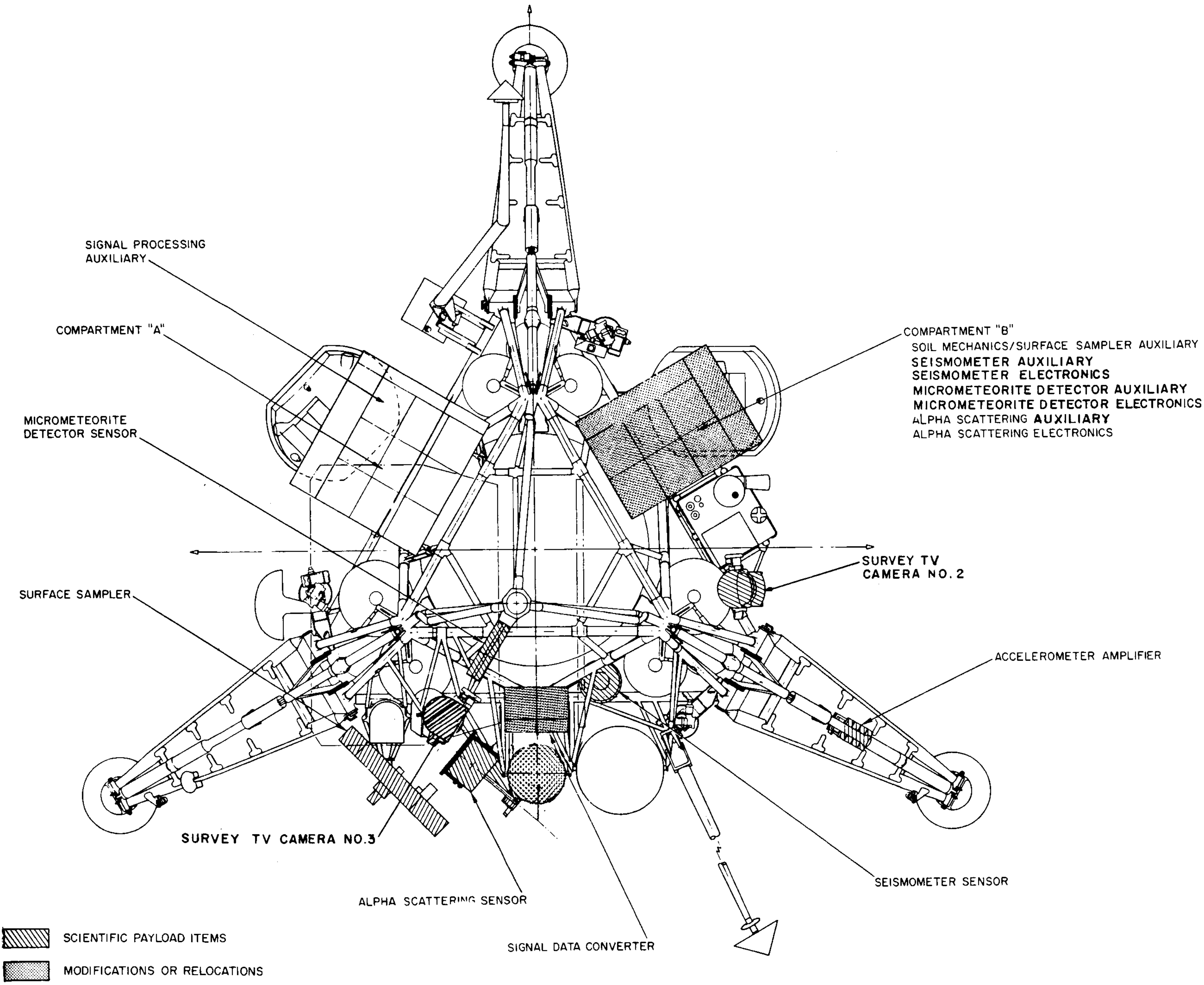
A-21 and A-21A	A-21 only
Engineering Signal Processor	Auxiliary Engineering Signal Processor
Mode 1	Coast mode
Mode 2	Transit mode
Mode 3	
Mode 4	

Thermal compartment B temperature and dissipation characteristics also differ for spacecraft A-21 and A-21A. Figure C-2 shows the difference in compartment B temperature profiles. Compartment A has the same temperature profile for both spacecraft. Figure C-3 shows the compartment dissipating capability for the two spacecraft as a function of solar incidence angle.

Midcourse correction capability for transit and landing is different for the A-21 and A-21A spacecraft. The values listed below assume the midcourse correction maneuver executed 15 hours after injection.

Spacecraft	Typical velocity increment available	Typical miss distance
A-21	44 meters per second	69 kilometers
A-21A	30 meters per second	47 kilometers

Figure C-4 shows the difference in dispersion ellipse for the two spacecraft.


SIGNAL PROCESSING AUXILIARY
COMPARTMENT "A"
MICROMETEORITE DETECTOR SENSOR
SURFACE SAMPLER
SURVEY TV CAMERA NO.3
ALPHA SCATTERING SENSOR
SIGNAL DATA CONVERTER
SEISMOMETER SENSOR
ACCELEROMETER AMPLIFIER
SURVEY TV CAMERA NO. 2
COMPARTMENT "B"
SOIL MECHANICS/SURFACE SAMPLER AUXILIARY
SEISMOMETER AUXILIARY
SEISMOMETER ELECTRONICS
MICROMETEORITE DETECTOR AUXILIARY
MICROMETEORITE DETECTOR ELECTRONICS
ALPHA SCATTERING AUXILIARY
ALPHA SCATTERING ELECTRONICS
SCIENTIFIC PAYLOAD ITEMS
MODIFICATIONS OR RELOCATIONS

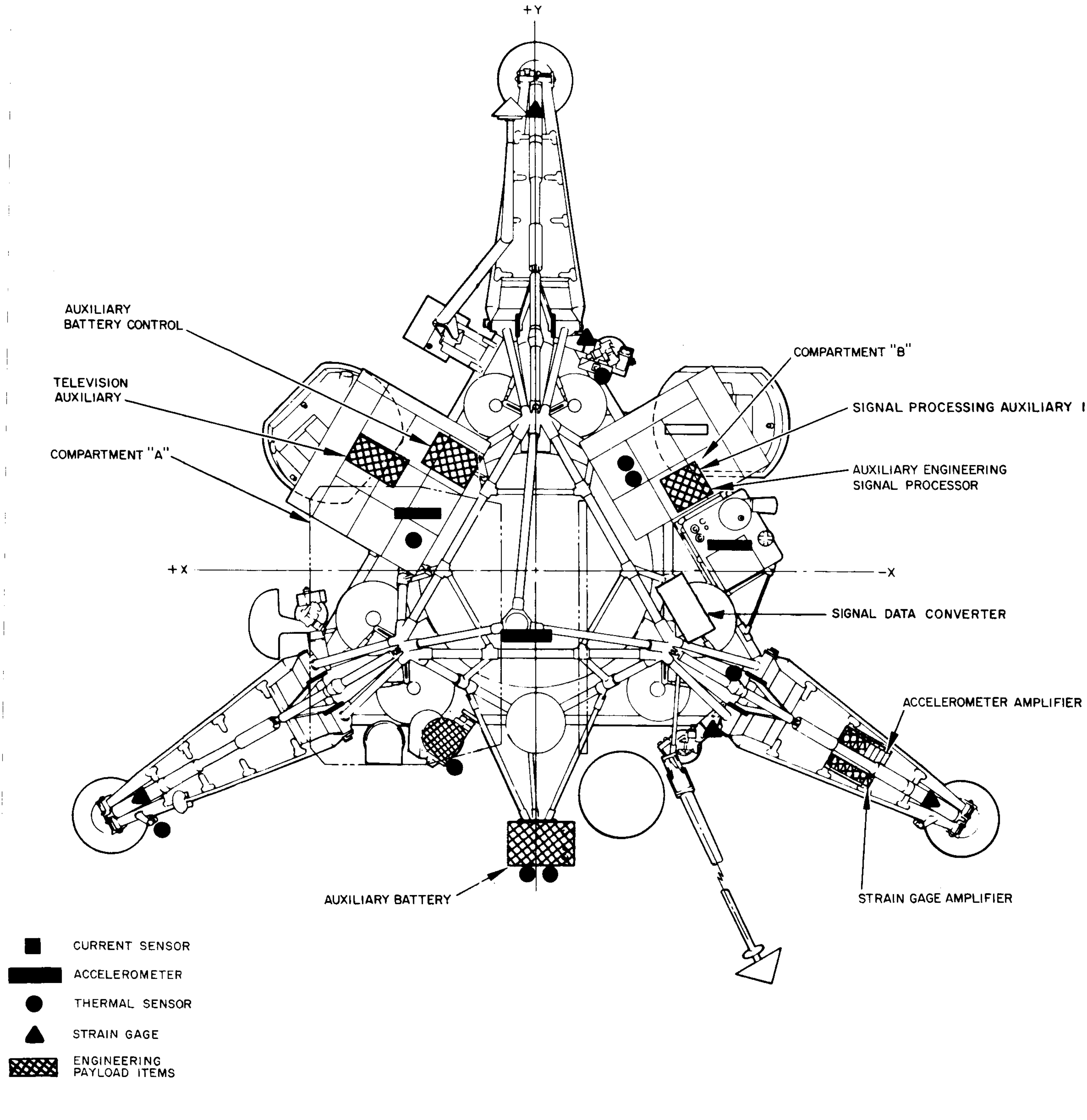


FIGURE C-1. SURVEYOR PHYSICAL DIFFERENCES

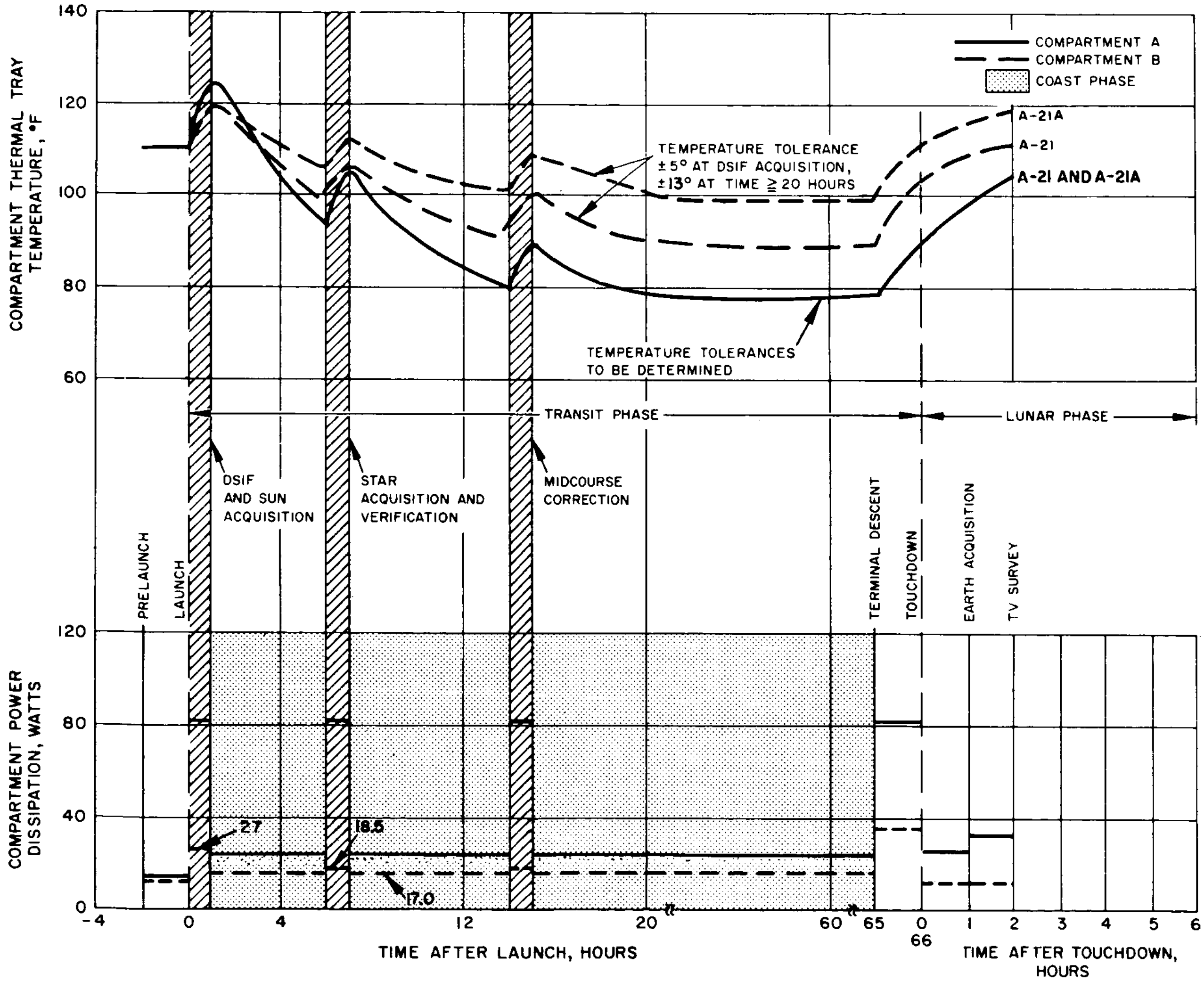


FIGURE C-2. COMPARTMENTS A AND B THERMAL TRAY TEMPERATURE PROFILE DIFFERENCES

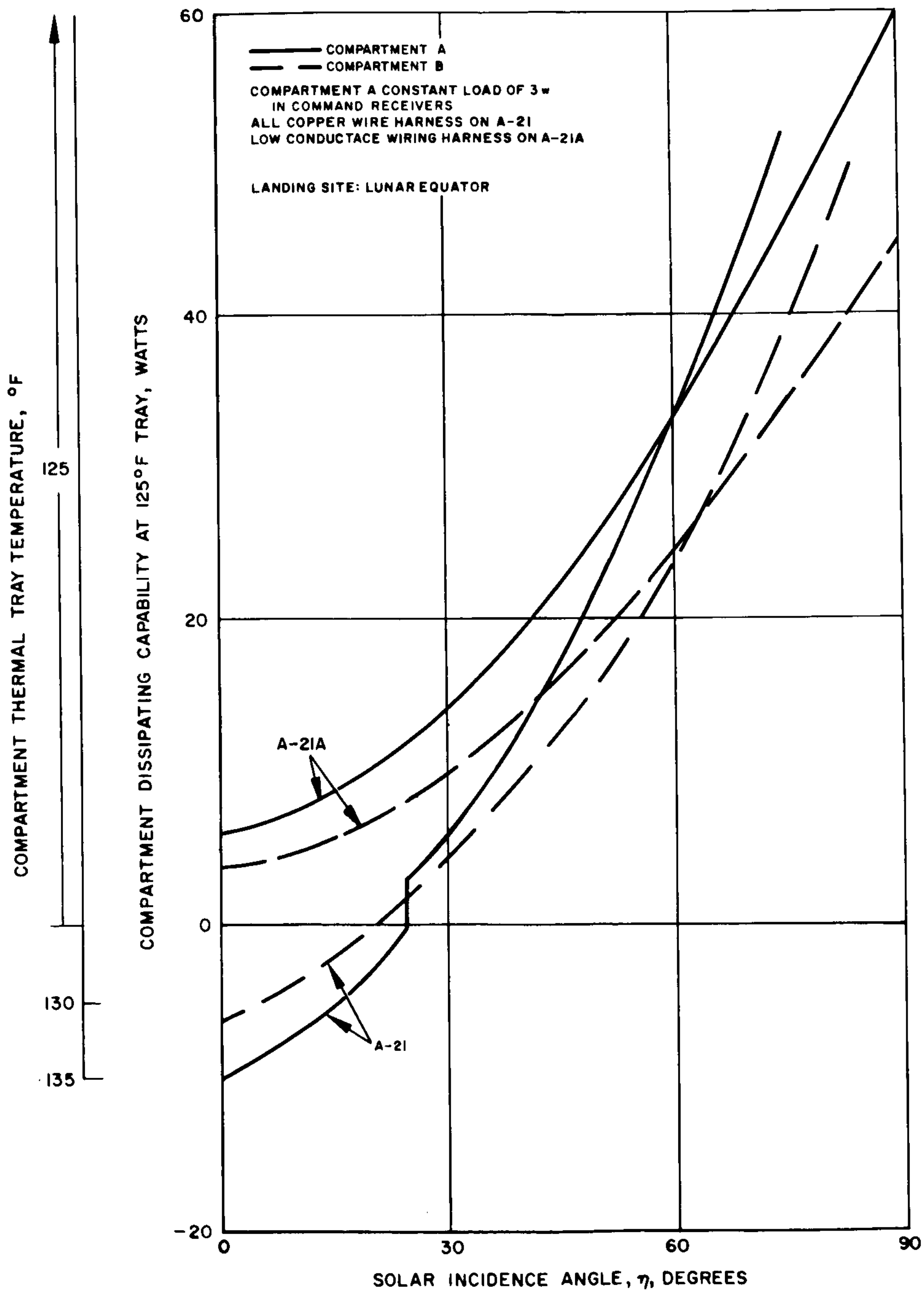


FIGURE C-3. COMPARTMENT DISSIPATING CAPABILITY DIFFERENCES, LUNAR DAY

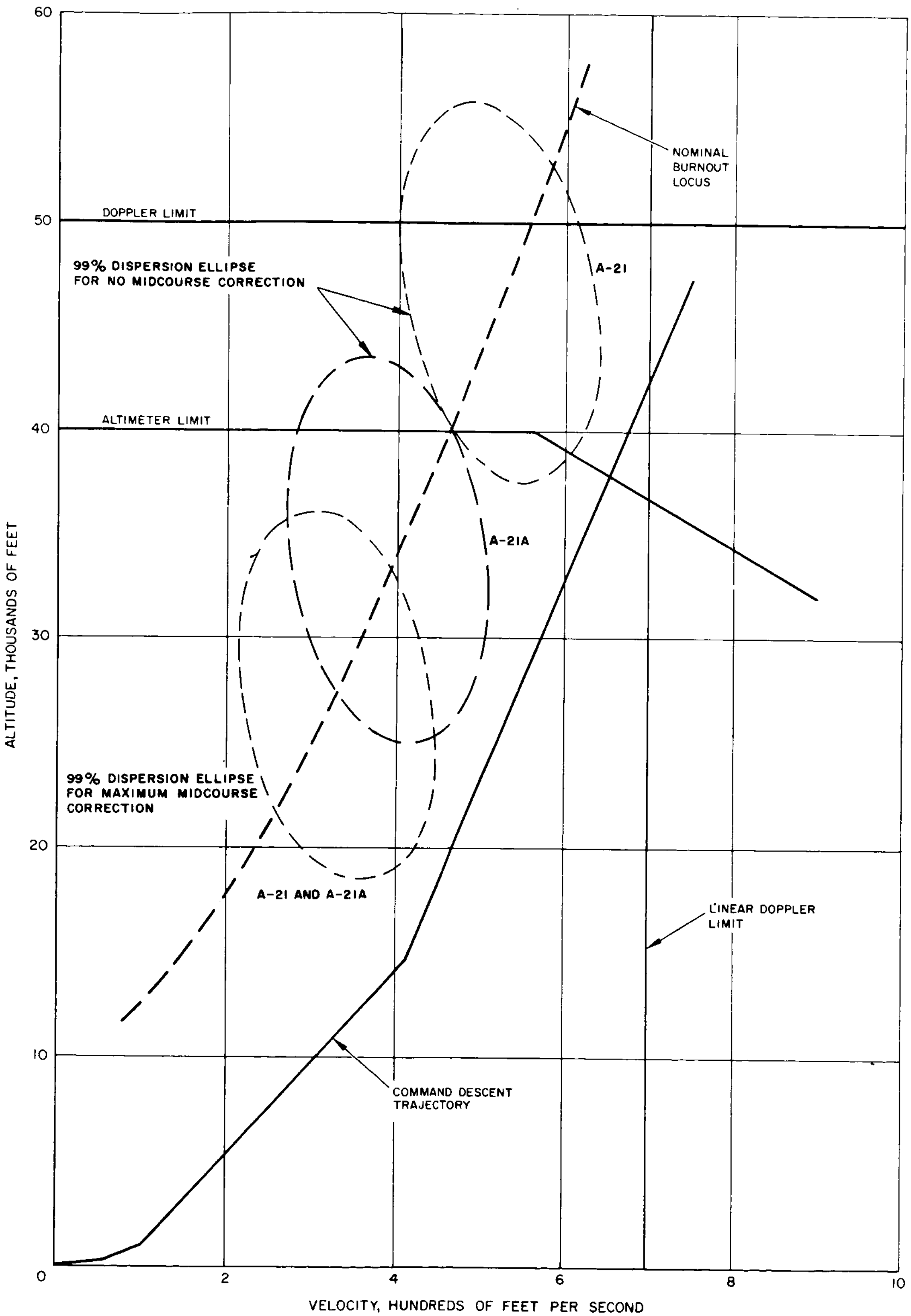


FIGURE C-4. VERNIER DESCENT PHASE DIFFERENCES