d. A sequence timer signal immediately follows as backup to shut off the pitch-and-yaw pneumatic controls. Also, the timer opens the gas-generator arm-and-fire circuits and connects the inertial-reference-package accelerometer to the velocity integrator.

e. The satellite rocket engine achieves steady-state thrust.

f. Shortly before satellite rocket engine shutdown, the sequence timer arms the pitch-and-yaw pneumatic control circuit and activates the engine cut-off safety switch which has been preventing premature engine shutdown command from the velocity integrator. The integrator determines when the satellite has attained the required velocity to be gained and sends a shutdown signal to the engine relay box.

g. The engine relay box cuts off electrical power to cause the gas-generator valve to close and effect satellite rocket engine shutdown. Simultaneously, the engine relay box actuates the pneumatic control circuit to turn on the pitch-and-yaw pneumatic controls.

h. The sequence timer disconnects the accelerometer from the velocity integrator to complete propulsion system sequencing.

4-49. POWER INPUT REQUIREMENTS. The power input requirements utilized during the sequence of system operation is supplied by the 22 to 29.5-volt, unregulated, direct-current, battery power. Peak demand is 9.75 amperes and minimum power requirement for nominal 120-second operation is 500 ampere-seconds.

4-50. PROPELLANT SUBSYSTEM.

4-51. GENERAL. The major propellant subsystem items and the order in which they will be discussed are: Propellant-Tank Assembly and Mount, Propellant Feed-and-Load and Pressurization Systems, Propellant-Tank Ullage Orientation System. The propellant tank assembly, its mount, and the related propellant systems comprise the propulsion system equipment that is utilized to properly supply propellants to the turbopumps of the satellite rocket.

4-52. PROPELLANT-TANK ASSEMBLY AND MOUNT. The propellant-tank assembly is comprised of a forward, spherical tank and an aft, hemispherical section that are connected together by a truncated cone and a tank-support fitting to form a single installation unit. The forward tank has a volume of 49.12 cubic feet and is used to contain the oxidizer. The aft tank, with a volume of 37.80 cubic feet, is for the fuel. The single unit forms an egg-shaped tank assembly. The oxidizer sphere forms the compartment division between the tanks, and the truncated cone completes the aft tank enclosure. The oxidizer tank has a combined pressure inlet and vent, while the fuel tank has a separate pressure inlet and vent. Each tank has a single propellant opening with a T-fitting that serves both as propellant inlet and outlet. The line on one side of the tee connects to the propellant-load-and-dump, quick-disconnect. The line on the other side connects to the satellite rocket engine. Both tank sections have inspection and cleanout covers bolted on opposite ends of the complete assembly. Screen baffles are installed in the tanks to prevent excessive liquid motion. The tank assembly is installed within the satellite midbody structure so that the tank support fitting engages inner midbody structure. Tiedown fittings retain the assembly in this position.

4-53. PROPELLANT FEED-AND-LOAD AND PRESSURIZATION SYSTEMS.

4-54. The propellant feed-and-load system and the pressurization system are integrally related in their respective system functions. This integral relationship exists throughout the time from loading propellants into the satellite and lasts until completion of propellant-tank and helium-sphere venting, subsequent to satellite rocket engine shutdown. The feed-and-load system provides the equipment for routing propellants into and from the propellant tanks, and consists mainly of the satellite portion of the propellant-load-and-dump, quick-disconnect fittings and the lines that connect the fittings to the tanks, plus the propellant feed lines that connect the tanks to the engine pumps.

4-55. The pressurization system provides the means of maintaining the desired pressure in the tanks from the time of propellant loading until completion of the venting operation that follows satellite rocket engine shutdown. The principle components of the system include the two helium pressurization spheres on the aft equipment rack, the pressurizing lines that interconnect the spheres and the propellant tanks, the lines for tank and sphere venting, the line for pressurizing the oxidizer-pump, double-lip seal, and the various valves and controls installed in the system lines.

4-56. For propellant loading, external propellant-loading lines and external vent lines are connected to the appropriate quick-disconnect fittings of the satellite. One loading line and one vent line is for the fuel tank. This provides a closed-loop loading system between each propellant tank and its respective ground-support propellant weighing tank. In this manner, the propellant tanks can be pressurized during the pumping of propellants into the satellite. By means of ground-equipment controls, the propellants can be forced back into the weighing tanks or, alternatively, into the external dump tanks, using gas pressure to effect the reverse flow. The external lines are released from the satellites immediately prior to liftoff by remotely applying pneumatic pressure to release actuators in the quick-disconnect fittings. The release system is backed up by an independent lanyard release device.

4-57. The two helium pressurization spheres can be loaded when the external helium source is connected to the helium-fill, quick-disconnect fitting of the satellite. Each sphere has a capacity of 2200 cubic inches and is loaded with dry helium to a pressure of 3000 psi and at a maximum temperature of 120°F.

4-58. Using the closed-loop propellant loading setup, the propellant tanks can be pressurized to full operating pressure either by use of the external vent lines during propellant loading (paragraph 4-56) or through the helium system during the process of loading the helium spheres. With the latter method, the oxidizer tank is pressurized by helium flowing from the spheres through the main pressure regulator; and the fuel tank is pressurized by helium flowing through the small-capacity bypass regulator. Helium-line check valves located between the regulators and the propellant tanks permit helium flow to the tanks but prevent backflow of propellants to the regulators. A normally-closed, squib-operated valve prevents main-regulator gas flow to the fuel tank until an electrical impulse from the guidance-and-control sequence timer ignites the squib. (Refer to sequence of system operation, paragraph 4-48.) The valve then opens to connect the main-regulator gas flow to the fuel tank to supply a greater rate of gas flow to the tank. This increased flow which is needed during engine operation is greater than can be supplied through the bypass regulator.

4-59. The combination of the bypass regulator and squib-operated valve are incorporated into the system to help minimize the possibility of oxidizer and fuel backflowing and mixing in a common pressurization line prior to engine ignition. The two components perform a backup function to the pressurization-line check valves by blocking further backflow, should propellants leak through the valves. After ignition, high gas flow in the lines prevent propellant backflow and therefore any possibility of mixing in these lines.

4-60. After satellite rocket engine shutdown, the propellant tanks, the spheres, and the tank and sphere lines are vented when a signal from the guidance and control sequence timer ignites the squib-operated vent valves, causing them to open. In order to minimize disturbances to the attitude control system, which must hold a fixed orientation of the satellite after engine shutdown, all vent lines terminate in impulse nullifiers, which are simply open-end T-fittings that allow equal amounts of gas to escape in opposite directions.

4-61. A line, which taps off of the low-pressure side of the main regulator, routes helium through a low-pressure regulator to the oxidizer-pump double-lip seal, to force into the overboard drain any oxidizer that leaks past the oxidizer-pump primary seal.

4-62. **PROPELLANT-TANK ULLAGE ORIENTATION SYSTEM.** The ullage orientation system provides a means of orienting any gas in the propellant-tank ullage away from the propellant feed lines during satellite rocket engine starting. This eliminates a source that could interrupt propellant ignition and cause possible engine malfunction. The system is required for this purpose, since, during the satellite coast period to engine starting, there is no gravity field to maintain the gas-phase ullage oriented away from the feed lines. However, as soon as the engine achieves full thrust, an effective gravity field is produced that causes the gas to become buoyant and float to the top of the propellants in the tanks.

4-63. The system consists of two, solid-propellant, ullage-orientation rockets that are mounted on the after side of the aft equipment rack. The rockets are fired by the guidance-and-control sequence timer. (Refer to paragraph 4-48.) Each rocket produces a nominal forward thrust of 128 ± 10 pounds over a burning time of 20 ± 2 seconds. After the rockets have burned for 16 seconds, engine ignition occurs. This provides a 4-second burning-time overlap between rocket thrust and engine thrust and thereby maintains the forward thrust and the artificial gravity produced in the vehicle.

4-64. The rocket-mounting brackets orient the rocket thrust vectors through the Samos Satellite center-of-gravity. Each bracket incorporates a release mechanism that jettisons its rocket casing when rocket thrust drops below a predetermined value, thus lightening the vehicle by approximately 17.5 pounds total weight.

4-65. ENGINE SUBSYSTEM.

4-66. GENERAL. The engine subsystem consists of the satellite rocket engine, its related propellant flow and control systems, and engine mounting provisions. The Turbopump Propellant-Feed Assembly, Gas Generator Assembly, and the Thrust Chamber Assembly are the three major subassemblies of the engine. The Engine Propellant-Flow System, Engine Electrical Control System, and the Engine Mount and Gimbal Assembly are important associated engine subsystem items. Both the major subassemblies and the associated engine subsystem items are discussed in subsequent paragraphs. The engine propellant-flow and electrical-control systems, although separately discussed, are closely related and interconnected to perform properly integrated engine operation and control functions during engine operation.

4-67. TURBOPUMP PROPELLANT FEED ASSEMBLY. The turbopump propellant feed assembly of the engine pumps fuel and oxidizer from the propellant tanks to the thrust chamber of the engine. The assembly consists of a single-stage, impulse-type turbine, an oxidizer pump and a fuel pump which are gear-coupled to the turbine shaft, and a gear housing that serves as the assembly frame. The fuel pump shaft is sealed from the gear case with a primary seal and a double-lip seal. To provide maximum protection against oxidizer leaking into the gear case, this double-lip seal is pressurized with low-pressure helium gas which forces any oxidizer that leaks past the primary seal to flow into the overboard drain.

4-68. The turbine is designed to operate at 24,000 rpm and is driven by hot gases from the gas generator assembly (refer to paragraph 4-70). Exhaust gases are ducted overboard through the turbine exhaust duct located along the right-hand side of the engine.

4-69. The two centrifugal-type propellant pumps, for any given rotational speed, produce a flow rate that is essentially constant over large variations in outlet pressure. This results in essentially constant volumetric mixture ratios of propellants delivered to the engine thrust chamber over the engine operating range. The design speeds for the fuel and oxidizer pumps are 24,570 and 13,946 rpm, respectively, and nominal flow rates are 15.1 pounds per second for fuel and 39.0 pounds per second for oxidizer.

4-70. GAS GENERATOR ASSEMBLY. The gas generator assembly provides a means of starting and maintaining turbine operation of the satellite rocket engine. The generator assembly consists of a small combustion chamber assembly; a solid-propellant, turbine starter assembly; a gas-generator, bipropellant valve and its associated solenoid valve; and a pair of cavitating venturies.

4-71. The solid-propellant container is attached to the combustion chamber. Upon ignition of this charge by a dual-squib igniter, the resultant, hot combustion gases pass through the combustion chamber and start the single-stage turbine. The bipropellant valve subsequently opens (refer to paragraphs 4-75 and 4-76), allowing fuel and oxidizer to enter the combustion chamber where they combine and ignite hypergolically. The venturies regulate the propellant flow through the bipropellant valve to the generator. After the solid-propellant grain is expended, the gases from fuel and oxidizer ignition continue to drive the turbine until engine shutdown. During operation, generator temperature is held to safe limits by film cooling and by regenerative cooling.

4-72. THRUST CHAMBER ASSEMBLY. The thrust chamber assembly provides the medium wherein the propellants, as received from the turbopump feed assembly, unite to produce the desired forward thrust. The assembly is an integral unit consisting of a regeneratively-cooled combustion chamber, a nozzle throat section, and a divergent nozzle section, and contains cooling passages within the walls. The oxidizer, used as a thrust chamber coolant, enters the chamber cooling passages through a manifold ring located slightly aft of the nozzle throat. The flov is divided in this ring by orifices. Part of the oxidizer cools the nozzle section in a double-pass pattern, while both portions are used to cool the throat and combustion chamber sections.

4-73. The thrust-chamber propellant injector is a "one-on-one" impinging type with an additional circle of fuel orifices fed by a separate manifold for film-cooling the combustion chamcer walls. The injector design is a cavity in the fuel section into which the fuel valve fits, thus allowing the fuel valve poppet to be located close to the fuel injector orifices. Free or unoccupied volumes are held small so that, when the fuel diaphragm ruptures, fuel injection starts with minimum delay and with minimum "hammer" as the free space fills with propellant. This arrangement is not needed for the oxidizer since oxidizer flow is accelerated gradually, and oxidizer is injected for an interval before the fuel valve opens.

4-74. A nozzle closure is used to assure combustion-chamber ignition when the engine is in a vacuum environment, and consists of a cone sealed to the exit of the nozzle and of a diaphragm installed over the cone and covering the nozzle exit. On the ground, the cone extends forward within the nozzle toward the nozzle throat, but when the vehicle ascends to vacuum environment, the air trapped forward of the cone causes it to collapse against the covering diaphragm. Thus, until the diaphragm is expelled by engine ignition, it retains the sealed cone which sustains the force of air pressure within the combustion chamber at approximately 7-1/2 pounds per square inch absolute.

4-75. ENGINE PROPELLANT-FLOW SYSTEM. (See figure 4-8.) The propellant flow system provides a means of controlling propellant flow from the turbine pumps to the thrust-chamber propellant injector and the gas-generator combustion chamber. The system consists mainly of the plumbing from the pumps to the injector and the gas generator plus the following flow control valves: the oxidizer valve, the fuel valve, the gas-generator, bipropellant valve, the gas-generator solenoid valve, and the pilot-operated, fuel-control solenoid valve. Except for the oxidizer valve, control valve actuation is initiated by the engine electrical-control system (refer to paragraph 4-76) through its interconnection with the two solenoid valves. The oxidizer valve is hydraulically actuated to open and spring-loaded to close whenever the pressure of oxidizer flow respectively exceeds or is less than the valve spring load. Closing of the gas-generator and fuel-control solenoid valves (refer to paragraph 4-76) causes hydraulic pressure in the secondary propellant lines to respectively open the bipropellant valve and the fuel valve, thus admitting propellants to the gas generator and fuel to the thrust-chamber propellant injector.

NOTE

Prior to the opening of these two valves, oxidizer has already started to flow into the propellant injector.

~~SECRET~~
LMSD-437237

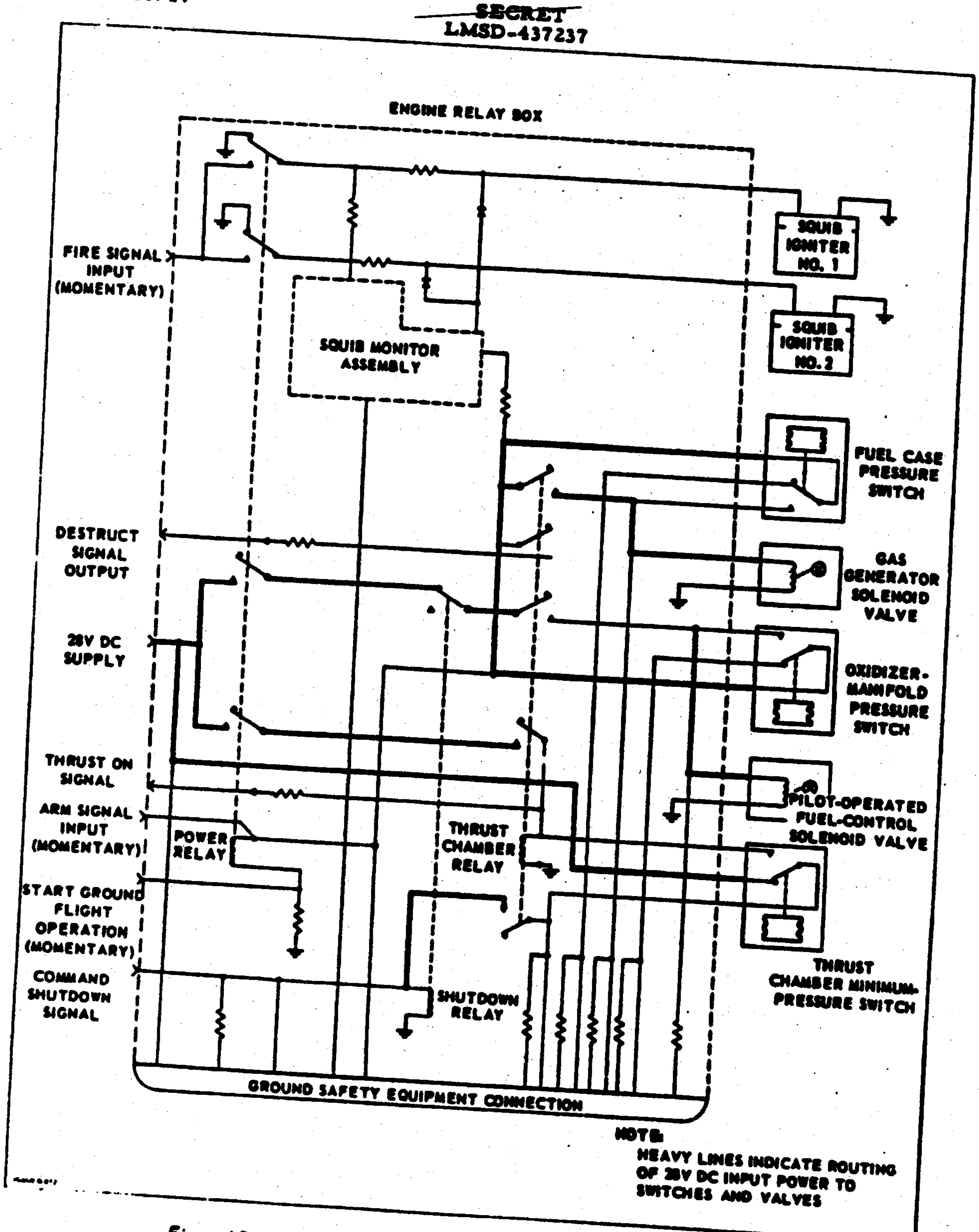


Figure 4-9. Satellite Rocket Engine, Electrical Control System Schematic

~~SECRET~~

Conversely, opening of the solenoid valves (refer to paragraph 4-76) causes hydraulic balance pressure in the secondary lines to close the respective flow-control valves. Thus, combustion stops in both the gas generator and the thrust chamber, and the turbine rapidly decelerates with a drop in pump pressure. The oxidizer valve closes last under spring pressure to provide after-flow of oxidizer during shutdown. The deriving and utilizing of hydraulic actuating power from the propellant system in this manner eliminates the added weight and complexity of an auxiliary hydraulic system.

4-76. ENGINE ELECTRICAL-CONTROL SYSTEM. (See figure 4-9.) The electrical-control system provides the electrical power for satellite rocket engine starting and shutdown, and also for initiating the actuation of the flow control valves to establish conditions essential for proper engine operation. The system consists mainly of a simple electrical circuit containing three relays (power, shutdown, and thrust chamber relays) and a squib monitor assembly. Connected to the circuit are the dual-squib igniter assembly for engine starting, three pressure switches (fuel case, oxidizer manifold, and thrust chamber minimum-pressure) for flow-control-valve operation, and the two solenoid valves described in paragraph 4-75. In addition, output circuits, for monitoring the sequence and function of the control system, and signal inputs are associated with the circuit. Provisions are made whereby a ground safety assembly can be plugged into the system for ground testing for the prevention of damage to the engine in the event of malfunction.

4-77. The power relay serves as an arming device for satellite rocket engine starting. While the power relay is de-energized, the squib firing circuits inside the engine relay box receive no external power, but when the relay is energized, the firing circuits can be fired by the sequence timer to ignite the solid-propellant turbine starter charge. Gases from this ignition start the turbine, causing pressure buildup that actuates the three pressure switches. (See paragraph 4-76.) The thrust-chamber pressure switch is the last to be actuated. It is actuated, only after it senses a sure start, to complete the circuit that energizes the thrust chamber relay. Prior to energizing this relay, power is connected through the fuel-case pressure switch and the oxidizer-manifold pressure switch to energize and close the gas-generator solenoid valve and the fuel-control solenoid valve, respectively. This, in turn, causes the bipropellant valve and the fuel valve to be opened (refer to paragraph 4-75). After the thrust chamber relay is energized, power passes directly through the relay to the solenoid valves, thereby bypassing the pressure switches. To meet the fast actuation requirements of the fuel valve, its associated solenoid valve is pilot-operated rather than direct-acting to meet substantial increased fuel flow with only a 70-percent increase in electrical current.

4-78. The energized thrust-chamber relay, in connecting electrical power to the two solenoid valves, also establishes conditions so that now only the shutdown relay or the power relay can de-energize the solenoid valves and thus cause engine shutdown. The engine will then shut down if the power input to the control box is interrupted; the voltage drops sufficiently to allow the power relay to drop out (be de-energized); the shutdown relay is energized by an external 28-volt command signal; or the chamber pressure decreases because of pump cavitation or any other cause that returns the thrust-chamber minimum-pressure switch to its original, de-actuated position. In the de-actuated position, the switch completes a circuit that energizes the shutdown relay. Any of these four actions remove power from the solenoid valves, causing them to open. Thus, the gas-generator valve and the fuel valve will close to effect satellite rocket engine shutdown.

4-79. ENGINE MOUNT AND GIMBAL ASSEMBLY. (See figure 4-21.) The engine mount consists of a tubular frame integral with the satellite rocket engine to attach the engine to the satellite. The engine thrust chamber assembly is attached to the mount and gimbal assembly by the two hydraulic actuators of the gimbal system to thus provide for satellite pitch-and-yaw control. (Refer to the guidance and control system, paragraph 4-143, for a discussion of the actuator control.)

4-80. The gimbal system nominally provides ± 5 degrees of deflection in a square pattern. A ring, containing four equally spaced bearings, surrounds the thrust chamber forward of the injector. The bearings engage four pins, two located on the forward face of the injector, and two, on the engine mount structure. Rotation of two opposite pins provides deflection in one plane. In a similar manner, rotation of the other two pins permit deflection in the perpendicular plane. Thrust is thus transmitted through each pair of bearings to the satellite via the engine mount structure.

SAMOS SATELLITE INTERNAL ELECTRICAL POWER SYSTEM

4-81. INTERNAL ELECTRICAL POWER SYSTEM FAMILIARIZATION.

4-82. GENERAL. (See figure 4-10.) The Samos Satellite internal electrical power system includes the satellite-borne equipment that furnishes electrical power for operating the Samos Satellite and payload reconnaissance equipment from immediately preceding launch and throughout the satellite reconnaissance lifetime. Prior to the use of the satellite-borne equipment, ground power equipment is used to furnish the required power to the vehicle subsystems during the checkout and related activities at the guided missile assembly building and the launch operations building. The ground equipment also provides the means of switching from ground to the satellite-borne equipment at the appropriate time prior to launch.

4-83. DESIGN CONCEPTS OF INTERNAL ELECTRICAL POWER SYSTEM. The major objective of the internal power system is to furnish electrical power in the proper form to the associated Samos Satellite and payload subsystems for a time interval compatible with the vehicle mission. A design which incorporated and universally satisfied the requirements of all satellite subsystems insofar as practicable, while preserving a high degree of efficiency and compatibility between the associated subsystems was employed, since the various satellite systems have electrical load requirements of the same relative magnitude.

4-84. FUNCTIONAL DESCRIPTION OF INTERNAL ELECTRICAL POWER SYSTEM. (See figure 4-11.)

4-85. The internal electrical power system for flight configuration I consists of three major groups of equipment designated as prime energy, power conversion, and power control equipment. The prime energy equipment consists of 12 silver-peroxide/zinc primary batteries. The power conversion equipment includes two 2000-cycle and one 400-cycle inverter, two 2000-cycle and one 400-cycle load limiter, two +28-volt and one 400-cycle regulator,

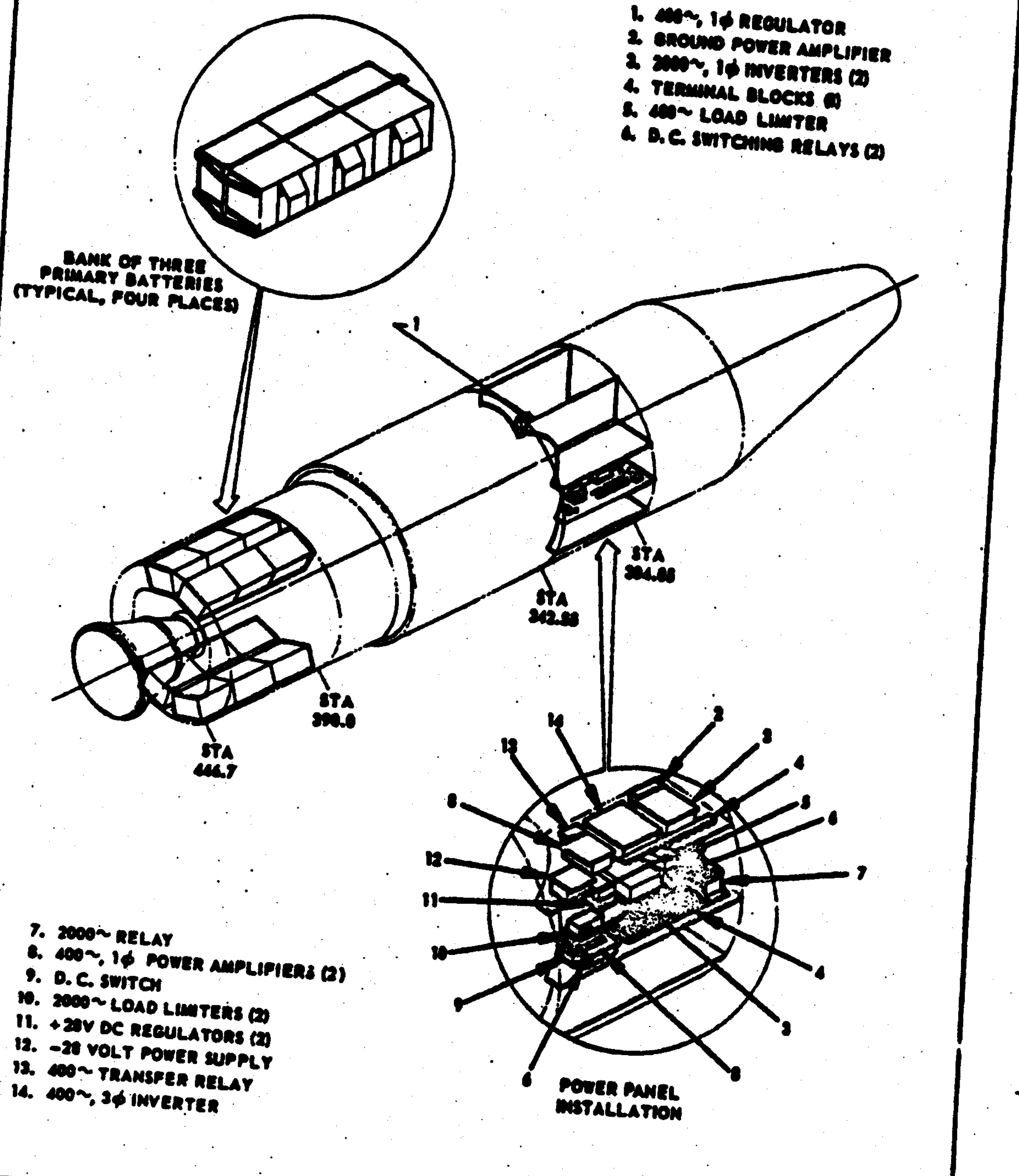


Figure 4-10. Samos Satellite Internal Electrical Power System, Location of Components

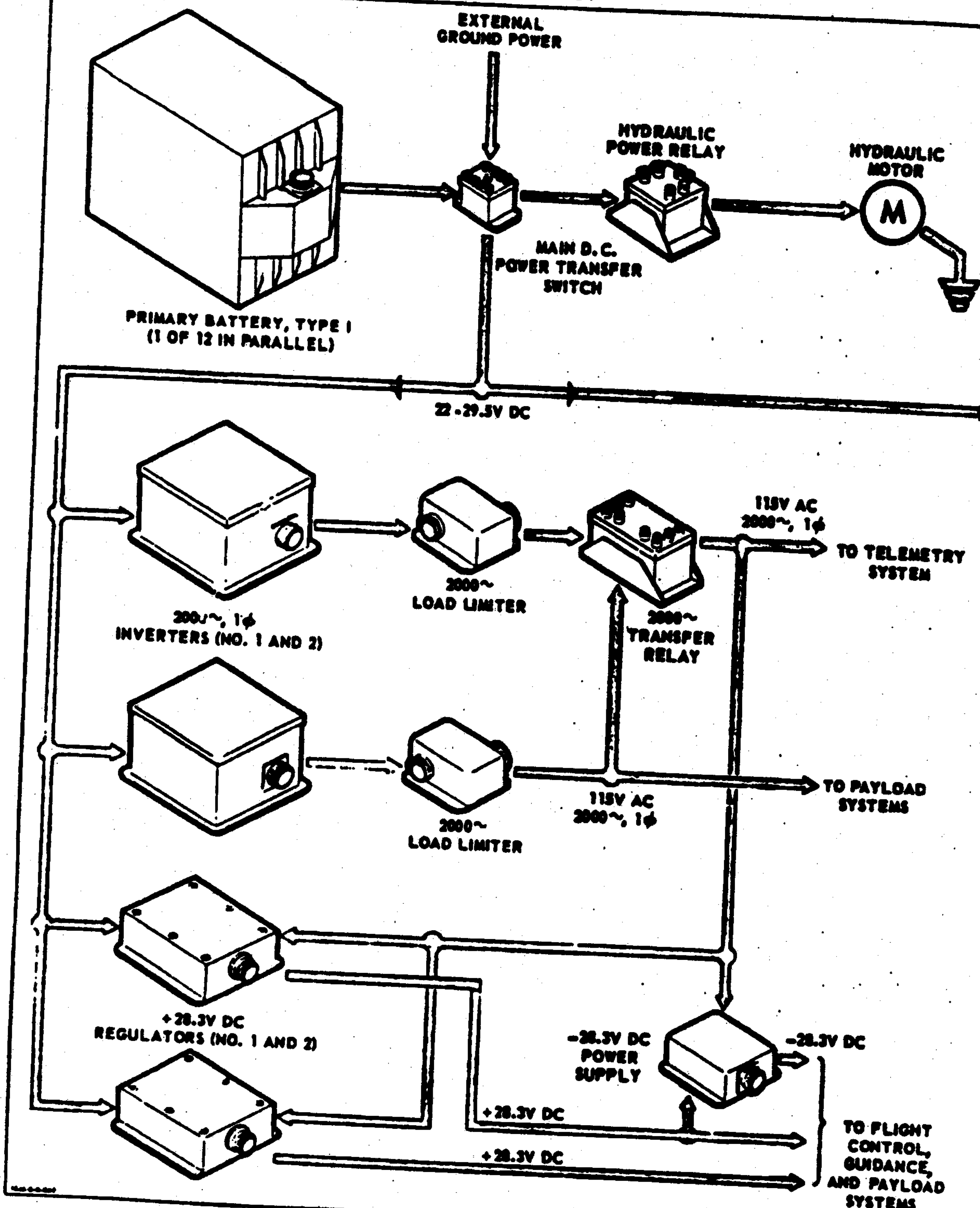


Figure 4-11. Samos Satellite Internal Electrical Power System, Functional Schematic (Sheet 1 of 2)

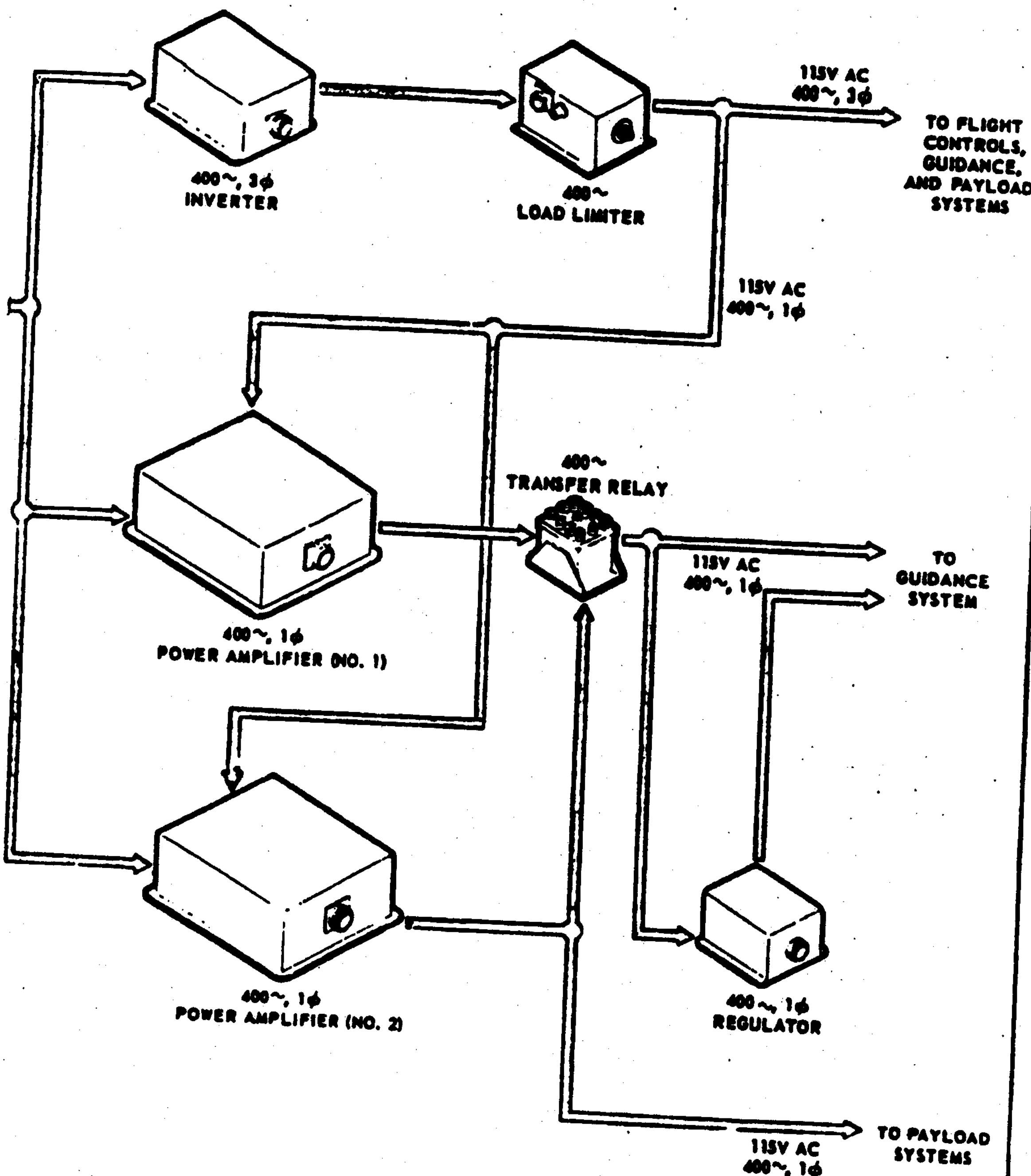


Figure 4-11. Samos Satellite Internal Electrical Power System, Functional Schematic (Sheet 2 of 2)

a -28-volt power supply, and two 400-cycle power amplifiers. The power control equipment includes such items as relays, switches, umbilical connector, wiring harness, and protection devices. (Refer to paragraph 4-89 for major equipment item descriptions.)

4-86. The batteries furnish sufficient primary electrical power to meet the electrical load requirements of the vehicle in figure 4-12. The system supplies both + and -28-volt direct current power, 2000-cycle, single-phase, 115-volt power; 400-cycle, 3-phase, 115-volt power; 400-cycle, single-phase, 115-volt, precision regulated power; and unregulated battery power. The unregulated battery power is directly furnished for satellite equipment, such as the hydraulic pump drive motor, that can utilize the 22 to 29.5 battery voltage range. For equipment that requires more closely controlled voltage, the internal electrical power system utilizes the two +28-volt regulators which operate in series with the unregulated battery source and which also draw power from the 2000-cycle inverters to provide regulated +28-volt direct current. In addition, the -28-volt power supply unit operates from the 2000-cycle system and is arranged to track, in voltage regulation, the +28-volt regulator. Positive and negative direct current from the power supply unit and the regulator are required for the flight control and guidance systems and the payload. The 400-cycle, 3-phase inverter furnishes the basic a-c power to the satellite guidance and control system, the recorders, and payload; the 400-cycle power amplifiers, synchronized with the reference phase from the inverter, furnish the required guidance-and-control system, single-phase, 400-cycle, reference power. During countdown, the outputs of the + and -28-volt regulators and the a-c power inverter are monitored from the launch operations building.

4-87. A power junction box serves to interconnect the system components. The box includes the necessary bus bars and terminal strips for all interconnected components. System terminal grounding is accomplished at the junction box with a ground bus connection to the vehicle frame. Power distribution leads are carried from the junction box to the power distribution box. From the power distribution box, power is distributed to the various subsystems.

4-88. Ground-checkout and performance-monitor leads are taken from the power junction box to the electrical umbilical connector. The internal power system components can be monitored during launch countdown. An external-internal power transfer switch is mechanically designed as an integral part of the umbilical connector. It incorporates an external solenoid which actuates the switch to permit switching from external to internal power during countdown prior to launch. Provisions are also made in the power transfer switch for lanyard operation in the event of failure of the solenoid mechanism or of the pneumatic, umbilical power disconnect. The switch also incorporates contractors to furnish a power-transfer signal to the system-checkout, and to the monitor-console panel in the launch operations building.

4-89. MAJOR EQUIPMENT ITEMS DESCRIPTION.

4-90. GENERAL. (See figure 4-10.) The major equipment items of the internal electrical power system that are discussed in the following paragraphs include the batteries, inverters, regulators, power amplifiers, and negative power supply. With the exception of the batteries which are installed on the aft equipment rack, the major items are located on the power panel of the forward equipment rack.

4-91. PRIMARY BATTERIES, TYPE I. (See figure 4-10.) Twelve primary, silver-peroxide/zinc batteries serve as the prime electrical energy source for flight configuration I Samos Satellites. The batteries are installed on the aft equipment rack (figure 4-4), grouped in four banks of three batteries each. Each battery weighs 106 pounds and consists of 16 series-connected cells enclosed within a magnesium case. Battery capacity rating is 360 ampere-hours at a nominal potential direct current rating of 25.0 volts and with an energy-to-weight factor of 75 watt-hours per pound. Each battery incorporates parallel-connected diodes in series with the discharge lead connection to provide circulating current protection. The diodes are for isolation of a battery unit in the event of malfunction, to thus prevent the discharge of the complete battery bank. Each battery is equipped with a pressure relief valve designed to maintain internal pressures within the range of 5 to 15 pounds per square inch.

4-92. 2000-CYCLE, STATIC INVERTERS. (See figure 4-10.) Two 2000-cycle, single-phase, static inverters, controlled by an inductance-capacitance circuit are used in the internal power system to convert d-c battery power to alternating current power. The two inverters are transistor-oscillator power amplifiers that furnish the main 115 volt, 2000 cycle, single-phase a-c power for the vehicle telemetering equipment and for the payload. Dual inverters are used to meet the heavy electrical loads of such equipment.

4-93. PLUS 28-VOLT REGULATORS. (See figure 4-10.) The internal electrical power system utilizes two 28-volt d-c regulators that operate in series with the unregulated batteries and which draw power from the 2000-cycle inverters to provide regulated 28-volt direct current. The 28.3 d-c voltage regulation is met in the internal power system. A voltage-booster concept is used to hold the efficiency of regulation to the highest degree possible. Regulation is accomplished by sensing the output potential of the batteries plus the booster-series combination. The booster operates from the 2000-cycle inverter power source, thus eliminating the requirement for an associated inverter. The regulator is constructed with efficient heat transfer between diodes and the metallic case structure. All components are mounted mechanically rigid, and the entire unit is potted to provide a sealed construction.

4-94. NEGATIVE 28-VOLT POWER SUPPLY. (See figure 4-10.) The -28-volt power supply is a conventional full-wave transformer-rectifier arranged for operation from the 2000-cycle inverters. Regulation is accomplished through the magnetic-amplifier circuit of the No. 1 +28-volt regulator.

4-95. 400-CYCLE, 3-PHASE INVERTER. (See figure 4-10.) The 400-cycle, 3-phase inverter is a transistorized power inverter. Its frequency is controlled by a 19.2-kilocycle, quartz-crystal-controlled oscillator. The unit incorp ates power input filters to restrict noise reflection into the battery circuit. T e filters eliminate noise feedback that would pass from the inverter transis r switching circuits through the battery to the low-signal-level electronic eq ment.

4-9 . 400-CYCLE, SINGLE-PHASE, A-C REGULATOR. (See figure 4-10.) The 400-cycle, single-phase, a-c regulator accepts and maintains a root mean square output of 115 volts. The voltage output is controlled by means of an input series magnetic amplifier. Integral circuits prevent phase shift between the input and output voltage from exceeding 3.5 pe cent. All power-handling elements in the unit are located on the base of the assembly for maximum heat transfer. All components are rigidly mounted, and the entire assembly is potted to insure vibration-proof operation.

4-97. 400-CYCLE, SINGLE-PHASE, SYNCHRONIZED, POWER AMPLIFIER. (See figure 4-10.) The power amplifier is a transistorized, static-electronic unit which receives a precision 400-cycle frequency signal from one phase of the three-phase inverter in order to synchronize and maintain its frequency to that of the three-phase power distribution system. One watt of power is drawn from the three-phase supply. The main power output is supplied by the unregulated d-c voltage through a push-pull, transistor switching-circuit. The output voltage is maintained at 115 (±1.15) volts root mean square, and the output current is transformer-sensed so that if the current exceeds the rating of the power amplifier, a current-limiting action occurs to prevent unit damage. All components are mechanically secured, and the entire unit is potted to provide maximum protection against vibration. All heat-producing components are in good thermal contact with the baseplate assemb'; for efficient heat transfer by conduction.

4-98. POWER AND SERVICE REQUIREMENTS.

4-99. ELECTRICAL POWER REQUIREMENTS. During countdown and until just prior to launch, satellite electrical power requirements are met by external auxiliary power supplied by the pad power supply and signal processing set. From just prior to launch, when the external-internal power transfer switch is actuated, until the end of Samos Satellite reconnaissance lifetime, the power requirements of the satellite subsystems and payload subsystems are met by the satellite internal electrical power system. Figure 4-12 lists the estimated electrical loads and energy requirements during the time the internal electrical power system supplies power. Three general groupings are shown: the satellite subsystems, the payload subsystems, and a summation of values indicated in the first two groups.

4-100. SYSTEM SERVICE REQUIREMENTS. Upon verification of internal system operability, no service requirements are needed inasmuch as the satellites are used for single mission purposes.

4-101. AIR CONDITIONING OF SAMOS SATELLITE AND GROUND SUPPORT EQUIPMENT. Air conditioning provisions are utilized when testing and checking out the Samos Satellite electronic equipment at the guided missile assembly building and when the satellite is located at the launch pad complex. During these times, properly conditioned air is delivered to the satellite electronic equipment from a ground air-conditioning unit whose duct is coupled to the satellite connector located on the right-hand access panel of the forward equipment rack. At the launch site, when the ground air-conditioning unit is operating and prior to the electronic equipment being turned on, the conditioned air delivered to the satellite mainly performs a drying function. When the equipment is turned on, in addition to drying, cooling of the equipment begins and continues until separation of the coupling at launch.

4-102. The ground air conditioning unit at the launch pad complex is located in the launch pad and service building. It provides conditioned air for the internal electrical power system of the satellite and for the checkout and service equipment located within the launch pad complex.

ESTIMATED VALUES FOR SATELLITE SUBSYSTEMS:

Component	Ascent Phase, Satellite Electrical Load		
	Duty Cycle (minutes)	Maximum Demand (kilowatts)	Energy (kw-hr)
Ascent Guidance	17.0	0.020	0.006
Engine Ignition	1.7	0.526	0.015
Flight Control	20.0	0.013	0.004
Hydraulic System	2.0	1.960	0.065
Horizon Scanner	10.0	0.012	0.002
Subtotal		2.531	0.092

Component	Orbit Phase, Satellite Electrical Load		
	Duty Cycle (minutes per day)	Maximum Demand (kilowatts)	Energy (kw-hr/day)
Attitude Control System	Continuous	0.440	1.058
Data-Link Transmitter	100	0.100	0.167
Command Transmitter	260	0.100	0.433
Programmer and Timer	Continuous	0.009	0.216
Decoder	262	0.015	0.066
Telemeter	130	0.230	0.500
Tape Recorder	300	0.078	0.390
Subtotal		0.972	2.830

ESTIMATED VALUES FOR PAYLOADS SUBSYSTEMS

Payload Subsystem	Maximum Power Demand (kilowatt)		Energy (kw-hr/day)
	Ascent	Orbit	
E-1 Payload Readout	0	0.076	0.320
F-1 Payload Readout	0	0.601	0.832

SUMMATION OF POWER REQUIRED FOR SATELLITE AND PAYLOADS SUBSYSTEMS

Satellite Subsystems Plus Payloads	Ascent Energy (kw-hr)	Maximum Orbit Power (kilowatt)	Total Energy (kw-hr/day)
E-1	0.092	1.048	3.150
F-1	0.092	1.573	3.662
E-1 and F-1	0.092	1.649	3.982

Figure 4-12. Samos Satellite Internal Electrical Power Requirements

4-103. MAJOR ACCESS AND INSPECTION FEATURES. Four access panels enclose the forward equipment area and form the outer surface of the satellite between stations 304 and 326. The right-hand panel contains the access covers for the air-conditioning coupling and the electrical umbilical connection. Each of the four primary-battery installation areas of the aft equipment rack are enclosed by an access panel.

4-104. INTERNAL ELECTRICAL POWER SYSTEM OPERATIONAL READINESS CHECKOUT CONCEPT.

4-105. GENERAL. A Samos Satellite, upon arrival at the launch base area, is received at the guided missile assembly building in the squadron maintenance area, where it is inspected and tested to verify the readiness of satellite equipment for satisfactory operation prior to transporting the satellite to the launch pad. At the guided missile assembly building, the internal electrical power system undergoes visual inspection followed first by preliminary checkout at the subsystem level and then by integrated systems level testing wherein the overall subsystems are sequentially checked.

4-106. SUBSYSTEM CHECKOUT CONCEPT. The internal electrical power system initially undergoes subsystem checks independent of other satellite subsystem checks. Herein, such equipment as the auxiliary power checkout console, the vehicle power monitor console, and the universal power supply, are used to check the output of system voltages and current. The voltages observed are the plus and minus 28-volt dc; 400-cycle, single- and three-phase ac; 2000-cycle, single-phase ac; and the prime-energy-source voltage of 22 to 29.5 volts. In addition to voltage and current measurements, 400-cycle and 2000-cycle, a-c power frequency plus phase rotation of the 400-cycle, three-phase inverter are measured. A programmed series of load conditions are undergone in order to determine the performance of system inverter and regulator components. The load checkout sequence permits a complete, accurate, and relatively quick checkout of the internal electrical power system under actual Samos Satellite electrical load conditions.

4-107. ELECTRICAL CHECKS DURING SATELLITE SYSTEMS CHECKOUT. The overall Samos Satellite systems checkout is an essentially complete, timed sequence of events check of satellite operations wherein dynamic system testing is performed by automatic sequencing of satellite functions throughout a programmed flight. The extent of the checkout of the internal electrical power system is one of performance monitoring in which the overall output of the system is monitored to assure performance within the required limits.

4-108. In this checkout concept, the Samos Satellite is installed in the vehicle tilt and roll stand and checkout is performed of all the subsystem components while they are operated together as a system during a simulated flight. During this checkout, the output voltages and currents of the internal electrical power system are monitored to check their response under actual load conditions. Provisions are incorporated into the systems checkout equipment to permit the utilization of the equipment used in subsystem checkout should the system monitor program indicate a failure in the internal electrical power system equipment. This provision facilitates the location of a fault in the internal electrical power system at the satellite systems checkout level.

4-109. Upon successful completion of systems checkout, the satellite is then taken from the guided missile assembly building to the launch pad complex.

4-110. LAUNCH PAD AND COUNTDOWN PHASE CONCEPT. The principal activities relating to the internal electrical power system while the Samos Satellite is at the launch pad include primary battery installation prior to hoisting and mating of the Samos Satellite to the Samos Booster. Also, final checkout of Samos Satellite functions is made through auto-sequenced programmer and monitor consoles and continued as required until all equipment checks out properly. During final countdown, internal and external electrical power is monitored during the final testing, servicing, and monitoring of the Samos Satellite and Samos Booster equipment and supporting facilities.

4-111. SUPPORT PERSONNEL. The Samos Satellite during its time at the launch base is handled by a satellite crew, a satellite bench maintenance crew, and a launch pad complex maintenance crew. The satellite crew has complete responsibility over the Samos Satellite from time of arrival at the base until launch. This crew is supported in the bench areas of the guided missile assembly building by the technicians of the bench maintenance crew that are assigned to the internal electrical power system and the various satellite subsystems. At the launch pad, the satellite crew, in preparing the satellite for launch, is supported by the permanent launch pad complex maintenance crew. The Samos Satellite and its adapter are hoisted and mated to the Samos Booster by the satellite crew, which then completes all connections and installations. The launch pad complex crew provides satellite testing during preparation for launch and, therein, is augmented as needed by the satellite crew in preparing the Samos Satellite for launch.

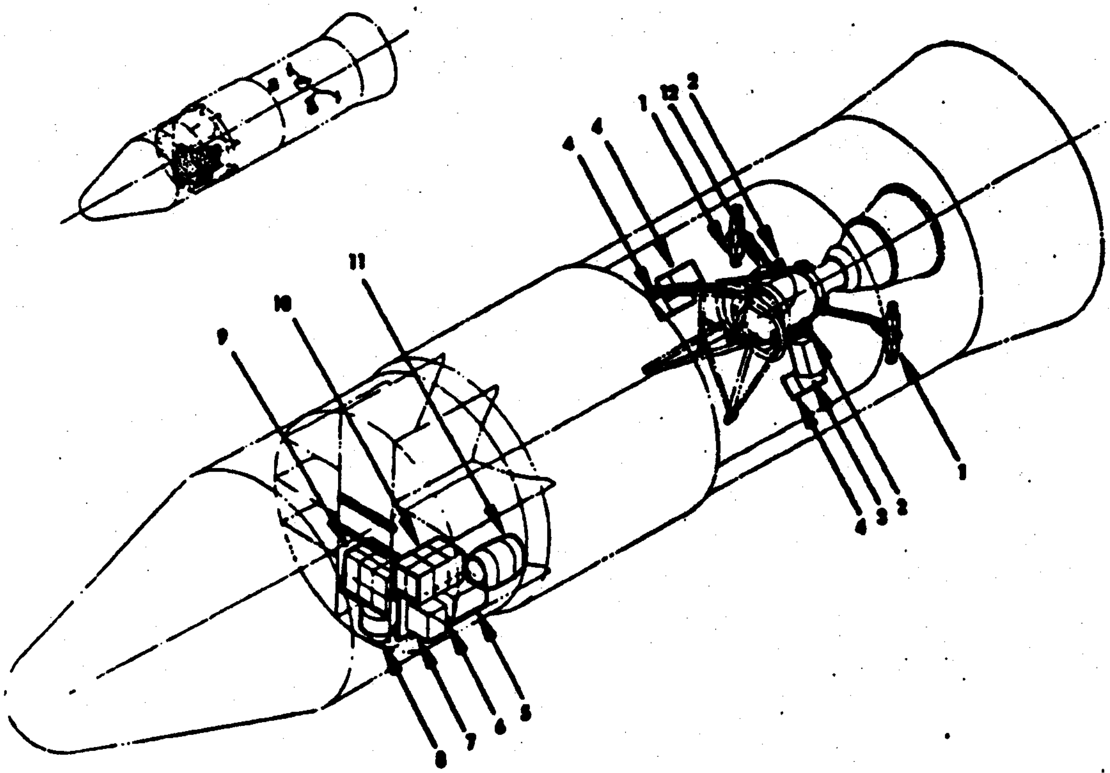
4-112. SUPPORT EQUIPMENT. The various support equipment that is used to check out and prepare the internal electrical power system for launch includes the vehicle power monitor console, the auxiliary power checkout unit, the universal power support, the discharge load tester for primary batteries, the dual battery charger, battery activation and handling equipment, and the overall systems checkout complex.

SAMOS SATELLITE-BORNE GUIDANCE AND CONTROL SYSTEMS

4-113. PURPOSE.

4-114. GENERAL. The Samos Satellite-borne guidance and control equipment positions the Samos Satellite to the correct attitude for all phases of flight after separation of the Samos Satellite from the Samos Booster, and maintains the prescribed attitude by stabilizing the Samos Satellite in pitch, roll, and yaw. A guidance system and a control system which constitute a unique combination of guidance devices and control mechanisms is provided in the Samos Satellite to perform the vital function of satellite attitude control. The guidance system provides the necessary guidance signals for the control system, and the control sytem changes the attitude of the satellite in response to the guidance signals. The location of the major equipment items that form the two systems is shown in figure 4-13.

4-115. Since the operations performed by the guidance and control systems during Samos Satellite ascent are different from those performed while the satellite is in orbit, and since different guidance signals and control mechanisms are used at these different times, the ascent and orbit phases are discussed separately. The ascent phase includes the separation of the Samos Satellite from the Samos Booster, the operation of the satellite rocket engine, and the orbit reorientation of the satellite from its horizontal attitude to a vertical,



1 GAS REACTION JETS
2 HYDRAULIC ACTUATOR
3 HYDRAULIC POWER UNIT AND INTEGRATED HYDRAULIC PACKAGE
4 HORIZON SENSOR
5 CONSTANT-RATE MOMENTUM WHEEL
6 INERTIAL REFERENCE PACKAGE (BEHIND ITEM 5)
7 ATTITUDE DAMPING PACKAGE
8 HORIZON SCANNER
9 COMPUTER
10 FLIGHT CONTROL ELECTRONICS UNIT
11 SEQUENCE TIMER
12 GAS SUPPLY SPHERE

Figure 4-13. Samos-Satellite-Borne Guidance and Control Systems, Location of Components