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TEST PLAN for NAVSTAR NAVIGATION SYSTEM

VOLUME II
ERS TEST SATELLITE DESIGN

Prepared under Contract NAS-12-539 for
ELECTRONICS RESEARCH CENTER
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

TRW
SYSTEMS GROUP

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1. INTRODUCTION

The satellite proposed for the NAVSTAR test program is one member of an existing family of Environmental Research Satellites (ERS). Excellent performance and high reliability have been demonstrated by these satellites, 15 of which have been placed in orbit as secondary payloads on other launches.

The test satellite is configured to provide a highly accurate check on the BINOR range measurements at a ground station by carrying a MISTRAM "B" transponder in addition to the BINOR range equipment. The satellite weighs 68 pounds and carries as the experimental payload a stable oscillator, a BINOR code generator, an L-band transmitter and a MISTRAM B transponder for obtaining highly accurate range and range-rate data at the MISTRAM ground facility. A separate VHF transmitter is included which will transmit status performance data on the satellite systems. The satellite is also command controlled by a VHF command link.

The anticipated nominal orbit is circular at an altitude of 300 nautical miles, inclined at 30 degrees to the equatorial plane. The only ground facility for the MISTRAM system is at Cape Kennedy, Florida; thus the test exercises would be conducted over this facility. The test satellite will be capable of operating during each pass over the MISTRAM site. At this altitude, the duration of each pass does not exceed 10 minutes and the maximum number of consecutive orbits that are visible to the ground station is three. The satellite antenna patterns and attitude must be such that RF downlink margins are adequate throughout each pass. The satellite configuration which fulfills these requirements is described in Section 2.

A simpler test satellite design which does not include the MISTRAM system is described in Section 3. The satellite in this case weighs about 32 pounds and carries only the L-band BINOR code system as the experimental payload. The orbit for this configuration would be the same as for the MISTRAM configuration.

2. SATELLITE DESIGN WITH MISTRAM

2.1 GENERAL DESCRIPTION

The basic satellite structure is an octagonal prism, 16 inches across a diagonal and 14 inches long. The structure is closed with solar panels on the eight side faces, and on one end. The satellite will be stabilized by a single-boom, gravity-gradient stabilization system which includes passive, magnetic-hysteresis damping. When stabilized, one end of the prism will face the earth with an accuracy of at least ±5 degrees. On the earth-pointing face are mounted the L-band antenna and a pair of X-band antennas. Both L- and X-band antennas have nominal 130-degree, broad-beam patterns giving visibility for ground station elevation angles as low as 10 degrees.

As shown in Figure 1, the satellite will have a VHF command and telemetry system to control operations and telemeter performance data. These communications subsystems are flight-proven hardware. The electrical power system is designed to meet all payload requirements on all visible passes over Cape Kennedy, even under worst-case eclipse conditions. The power system uses solar cells as the primary source and a rechargeable NiCd battery as a secondary source. The satellite payload will be exercised for a minimal 10-minute period while the satellite is within range of the MISTRAM facility in Florida.

An +8-dbw satellite EIRP at L-band (from the 4-watt transmitter and the +2 db antenna gain at the beam edges) is required in order to receive the BINOR range code signal in an aircraft with low antenna gain. The power budget (Table 1) shows that the aircraft can receive the satellite signal with a 6-db margin at a slant range of 1000 nautical miles. At a 10-degree elevation angle to the satellite, this slant range occurs with the selected 300 nautical mile circular orbit for the tests. The minimum required signal power of -132 dbm is based on the acquisition and thermal noise accuracy requirements for BINOR range code reception by the BINOR receiver.

2.2 PAYLOAD

The experimental payload consists of a 5.12-MHz stable oscillator, a BINOR code generator, and an L-band transmitter to transmit the BINOR

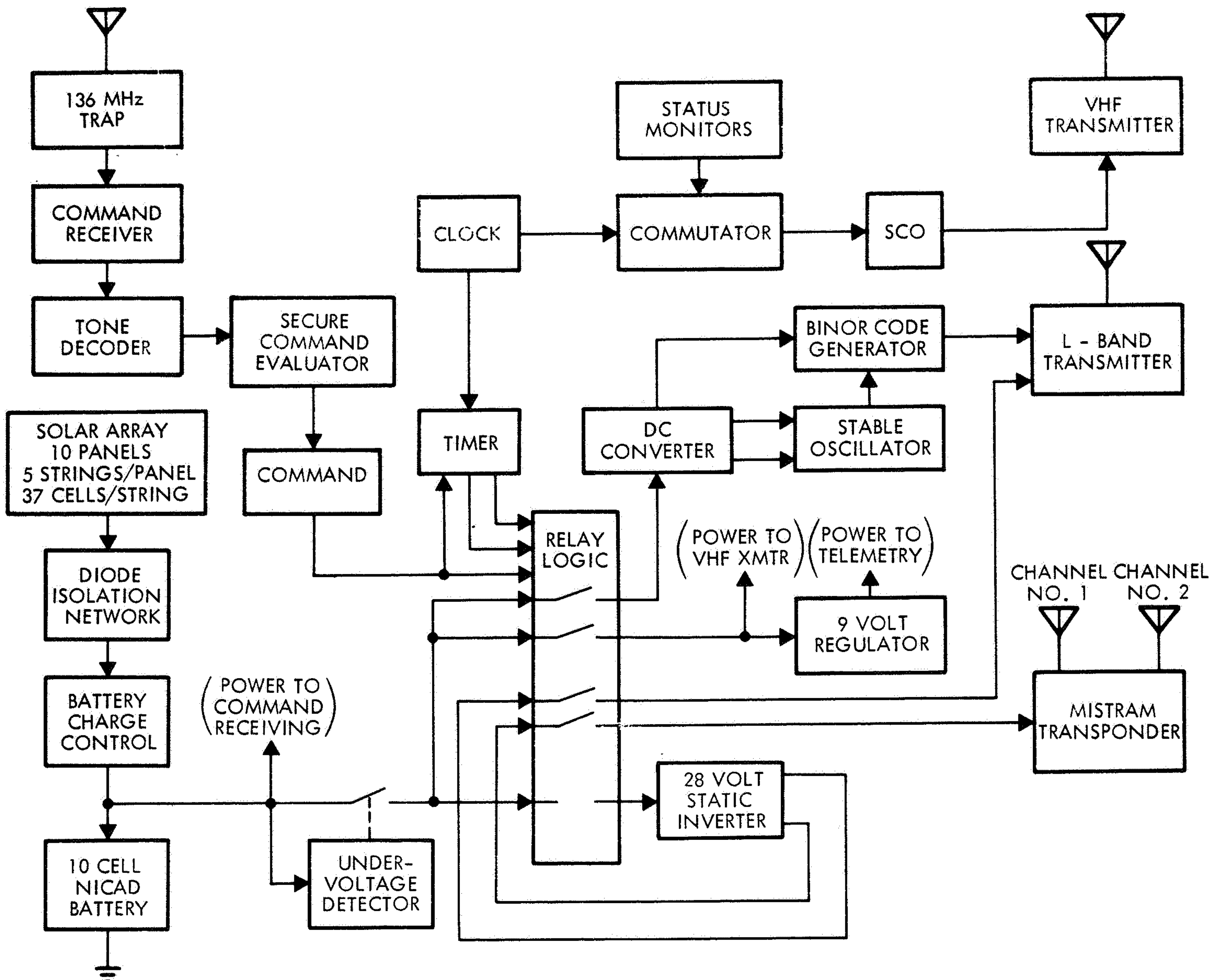


Figure 1. Test Satellite System Block Diagram

Table 1. L-Band RF Power Budget Satellite-to-Aircraft

f_o = 1600 MHz	
ERS transmitter power (4 watts)	+36 dbm
ERS antenna gain	+ 2 db
Space loss (1000 nmi slant range)	162 dbm
Aircraft antenna gain (turnstile)	0 db
Polarization loss	1 db
Circuit losses	1 db
Net transmission loss	162 db
Received signal power	-126 dbm
Minimum required signal power	-132 dbm
Margin	+ 6 db

code. In addition, the MISTRAM B transponder is included to evaluate the performance of the satellite navigational system.

2.2.1 Stable Oscillator

The reference signal from which the time clock and all the timing pulses for the range measurements are derived is provided by an ultra-stable oscillator. It consists of a precision crystal-controlled oscillator housed in a triple proportional oven. The unit operates at 5 MHz nominally; the remote frequency control allows fine frequency correction to be applied to counteract a long-term frequency drift. Two identical oscillators are used to provide a fail-safe redundancy. Both units are on all the time so that a failure of one will switch the other into operation without warm-up delay. Specifications of the unit are as follows:

Operating frequency	5 MHz
Temperature control	Triple proportional oven
Drift rate	$1\text{-}2 \times 10^{-12}$ per hour $1\text{-}2 \times 10^{-11}$ per day, maximum
Aging rate	$1\text{-}2 \times 10^{-12}$ per hour after seventh day of continuous operation
Power supply	±12 volts

Power supply sensitivity	$\pm 1 \times 10^{-11}$/5 percent change in supply
Load sensitivity	$\pm 10^{-11}$/20 percent load
Power consumption	2.5 watts, including remote frequency control
Remote frequency control, bi-directional	Fine control resolution $\pm 10^{-11}$ Coarse control resolution $\pm 2 \times 10^{-7}$
Remote frequency control signal	Digital pulses, 50 to 100 msec
Signal output level	1 volt rms into 50 ohms
Harmonics	Not less than 40 db down
Spurious outputs	Not less than 80 db down
Temperature range	-20 to +60°C
Temperature sensitivity	$\pm 1 \times 10^{-12}$/°C
Weight	6 pounds, including remote frequency control
Dimensions	3.5 × 4.5 × 9.0 in. maximum

2.2.2 BINOR Code Generator

The BINOR code generator, shown in Figure 2, will generate a binary code to be used by the aircraft or the ground station to determine distance to the satellite. The code length of the generator will be selectable by command to be either 2^{13} or 2^{15} bits. The generator will transmit the BINOR code for a number of code periods followed by several periods of the highest frequency (clock) component of the code. The sequence repeats itself after a short period to allow an unmodulated carrier to be transmitted in between code transmissions to simulate the actual NAVSTAR modulation format. Table 2 shows the transmission sequence from the code generator which continually repeats itself.

The code generator consists of a synchronously-gated binary counter which is selectable, by command, to be either 13 or 15 stages long. The square wave output of each binary stage is summed in analog fashion by a resistor summing network. The summing network output drives the

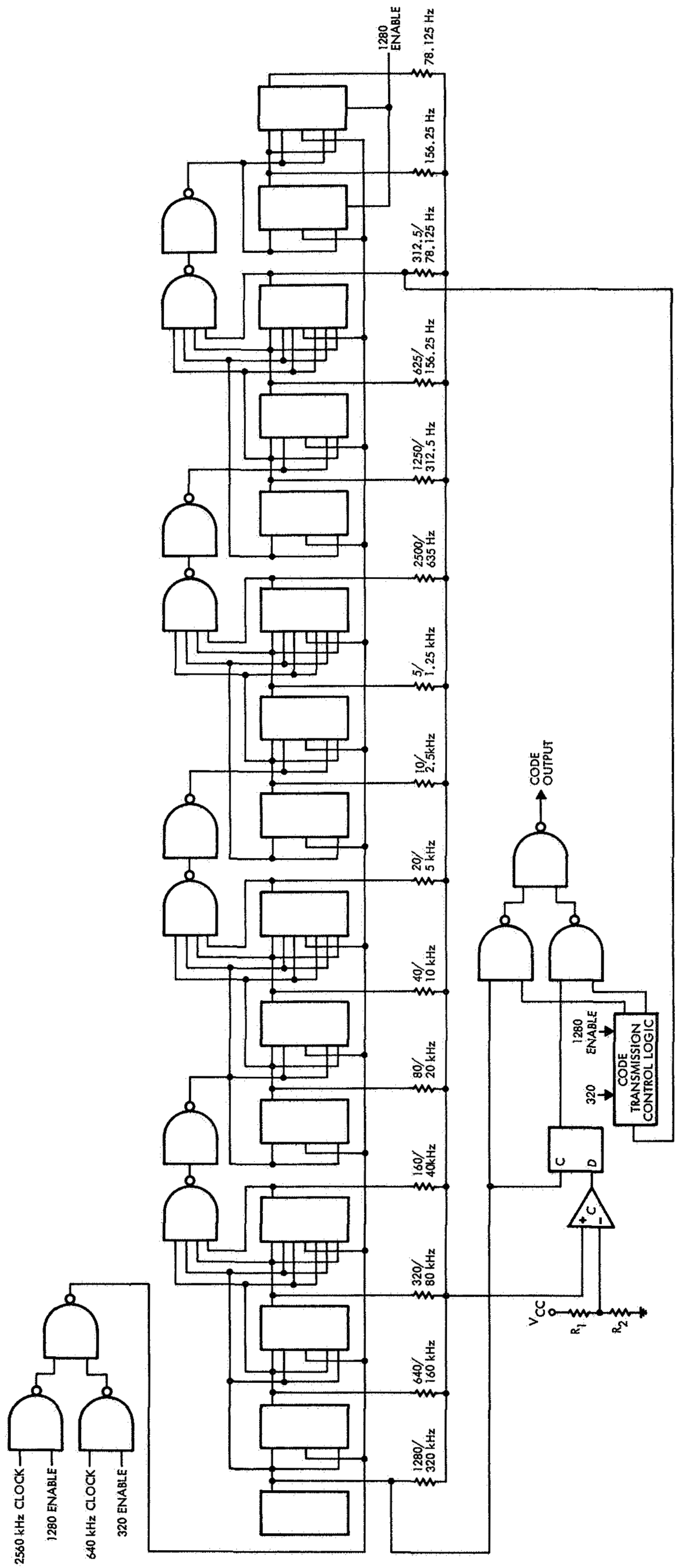


Figure 2. BINOR Code Generator

Table 2. BINOR Code Generator Transmission Sequence

Event	Duration*	
	320 kHz Clock	1280 kHz Clock
No signal	0.48 sec (38)	0.48 sec (38)
BINOR code	0.56 sec (44)	0.69 sec (54)
Clock frequency	0.18 sec (14)	0.18 sec (14)

*Numbers in parentheses represent code periods; the code period is 1/78.125 second long

comparator with its threshold set (R1, R2, and V_{cc}) such that the binary code is generated by the following rule. During each bit time of the highest frequency square wave or clock (320 or 1280 kHz) the number of square waves in binary state "0" are subtracted from the number of square waves in binary state "1." If the result of this subtraction is negative, the code is put in binary state "0"; if the result is zero (possible only when the number of binary stages is even) or positive, the code is put in the binary "1" state. Therefore, the code bit rate is equal to twice the highest square-wave frequency and the code period is equal to the period of the lowest frequency component (78.125 Hz). The counter stages are synchronously gated because the summing process requires a synchronous square-wave transition of each stage with respect to the transitions of the other stages.

The number of binary stages and the input clock drive frequency is selected by means of the 1280-320 enable signals. For a 2^{13} bit code, the last two binary stages are forced into opposite output states by the "1280 enable" signal, in order to cancel their effect in the summing process, and the 640 kHz input square wave is gated into the first binary stage (by the 320 enable signal). This results in a 320-kHz high frequency component (or clock) in the code; the 13th binary stage will have a 78.125 Hz square wave output. For a 2^{15} bit code, the last two stages are enabled and a 2560-kHz input square wave is gated into the first binary stage, resulting in a 1280-kHz frequency component in the code. The "code transmission control logic" allows the complete code or the highest frequency component (320 to 1280 kHz) to be alternately gated out of the generator in a periodic fashion.

The code output goes to a phase modulator for eventual transmission on the L-band frequency carrier.

2.2.3 L-Band Transmitter

The L-band transmitter tentatively selected for this mission is a flight-qualified unit being built for the U.S. Navy for telemetry of missiles and spacecraft. In order to simulate the time division multiplexing of the NAVSTAR satellites, the transmitter is turned off for 0.1 second at the start of every BINOR code transmission sequence by a signal from the code transmission logic in the code generator. The transmitter delivers a nominal 4 watts of RF power at a frequency of nominally 1560 MHz. It requires a power input of nominally 40 watts at 27 volts ±10 percent. The unit has a volume of 32 cubic inches and weighs 2 pounds. It meets all environmental requirements including a temperature range of -30 to +85°C.

2.2.4 MISTRAM Transponder

The MISTRAM B transponder, when installed in a flight article and operated in conjunction with the MISTRAM ground stations, provides a precision vehicle trajectory measurement system which provides position and velocity data. Its characteristics are listed in Table 3.

The MISTRAM B transponder, Figure 3, is all solid state except for a microwave amplifier tube. The transponder simultaneously receives two microwave signals and transmits two microwave signals, each of which is phase coherent with one of the received signals. One of the two transmit-receive channels, the range channel, maintains an exact ratio of 2054:2037 between the frequencies of the transmitted and received signals (nominally 68 MHz). The calibration channel, offset 256 MHz from the range channel, always maintains the same frequency difference between transmitted and received signals as that in the range channel (nominally 68 MHz).

The transponder, which is packaged in a machined magnesium housing, weighs approximately 17 pounds and has a volume of 400 cubic inches. Upper and lower cover plates are removable for access to the component parts, tuning adjustments, and test points. Thirteen mounting bolts and a number of screws hold the three-part housing together. The interfaces between the upper cover, body, and lower cover provide

Table 3. Electrical Characteristics of MISTRAM B Transponder

Item	Maximum	Nominal Value	Minimum	Remarks
Supply voltage (vdc)	32	28 50 W	25	Transients not to exceed 25% for 300 msec or 50 V for 10 sec
Input signal levels (dbm)	-35		-110	
Input frequencies (MHz)	7892.5	7892	7883.5	Calibration channel
	8148.5	8148	8147.5	Range channel
Output frequencies (MHz)	7960.5	7960	7951.5	Calibration channel
	8216.5	8216	8215.5	Range channel
Power output (watts)			1.0	Per channel
Phase delay variation at 256 MHz difference frequency (deg)	90			
Phase delay variation at 8 MHz difference frequency (deg)	4.0			
Phase jitter (deg)	18			At -76 dbm input level
Phase locked loop noise bandwidth (kHz)		40		Equivalent IF noise bandwidth

metal-to-metal contact bonding and are treated for corrosion resistance. Packings are employed at each interface to seal the transponder against pressure leakage and moisture intrusion. The magnesium housing of the transponder and the adjacent vehicle frame act as a heat sink during operation so that accessory cooling equipment is not normally required.

2.2.5 Antennas

The test satellite antenna system will require one L-band antenna and two X-band antennas. Both antenna systems will provide coverage

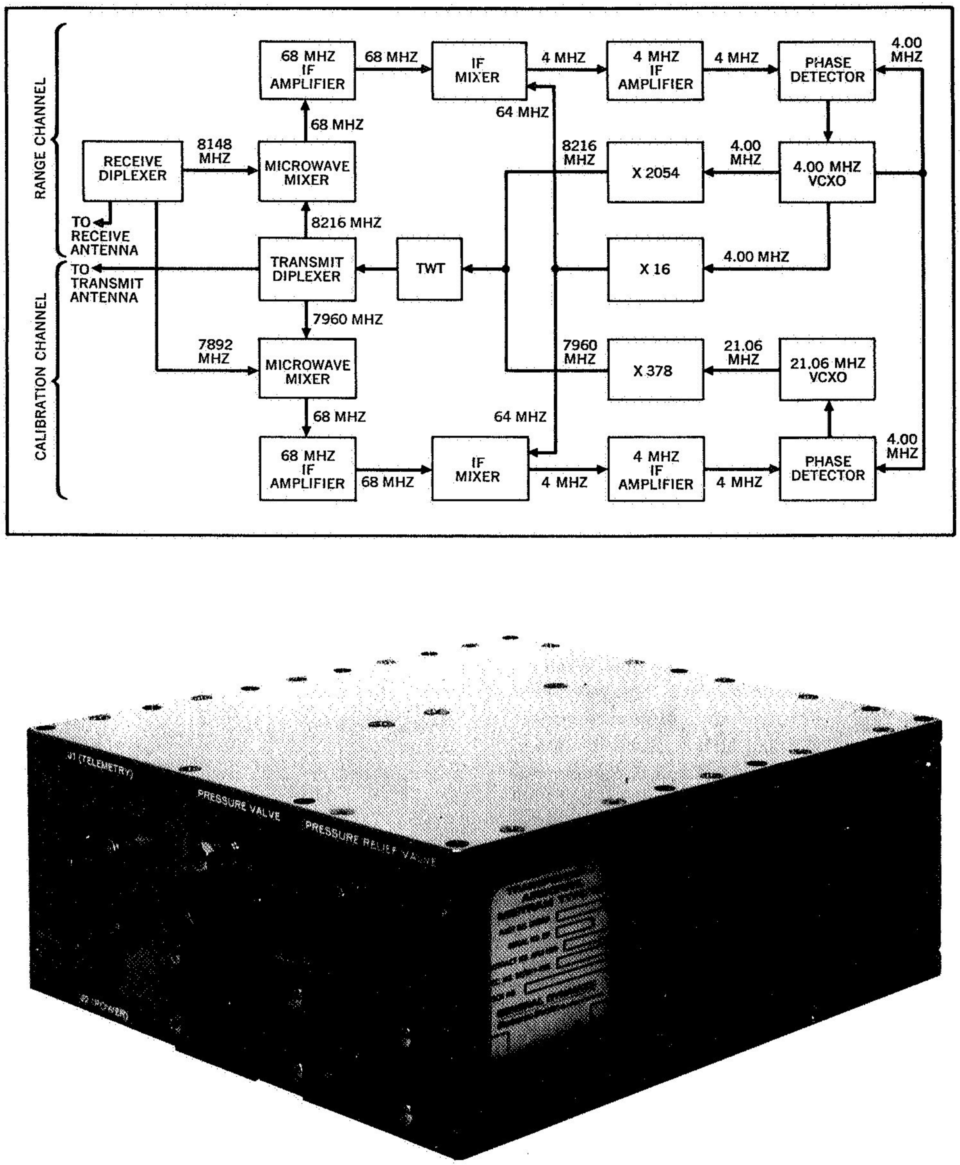
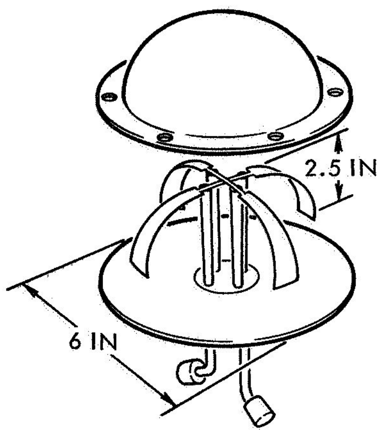


Figure 3. MISTRAM Type B Transponder

over a 130-degree conical beam oriented normal to one end of the satellite. The X-band system requires linear polarization with 60 db or greater isolation between the two antennas. Right hand circular polarization and a minimum gain of +2 db at the outer edges of the conical beam is required for the L-band antenna.

The proposed L-band antenna configuration is a curved arm turnstile as shown in Figure 4. Figure 5 shows the antenna in its center position on one end of the satellite with the X-band antenna mounted symmetrically



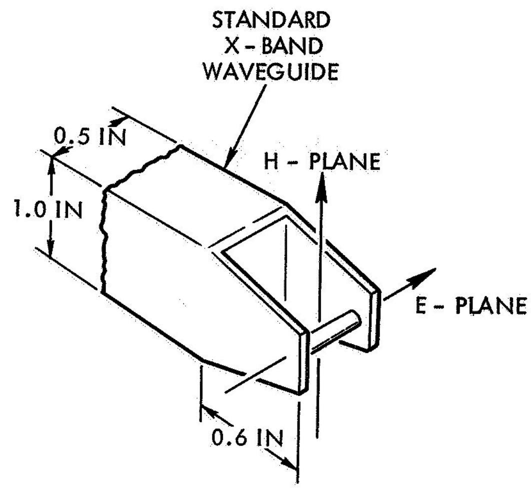


Figure 4. Satellite Antennas

Figure 5. Mounting of Satellite Antennas

nearby. The dipole elements will be fed through a hybrid coupler to maintain phase quadrature and insure equal power division. Each dipole element will be impedance matched to provide a VSWR of 1.5 to 1 or less with reference to an impedance of 50 ohms. Figure 6 is a typical radiation pattern of the curved arm turnstile antenna.

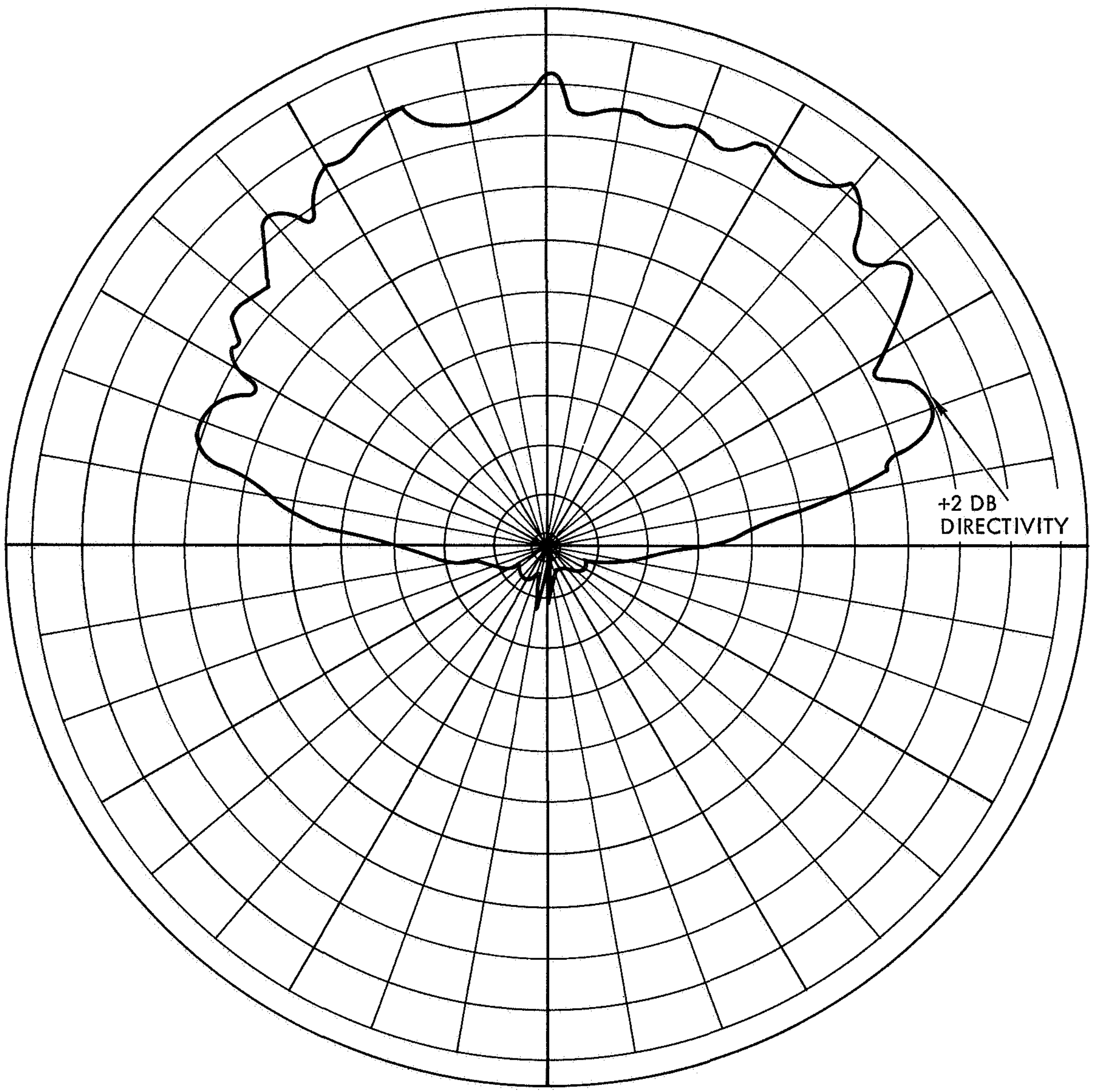


Figure 6. L-Band Antenna, Curved Arm Turnstile, RHC Polarization

The proposed X-band antenna system will utilize two waveguide antennas of the type shown in Figure 4. The tapered H-plane and post configuration is to provide beam broadening in the H-plane. The length of taper and the size of the E-plane post will be optimized to produce the required 130-degree beamwidth. E-plane beamwidth for this type of antenna is adequate without further modification. The antenna will be mounted with the H-plane in line and spaced eight wavelengths apart to minimize coupling. Each antenna will be impedance matched to provide a VSWR of 1.5 to 1 or less. Radiation patterns for H-plane and E-plane linear polarizations are shown in Figures 7 and 8.

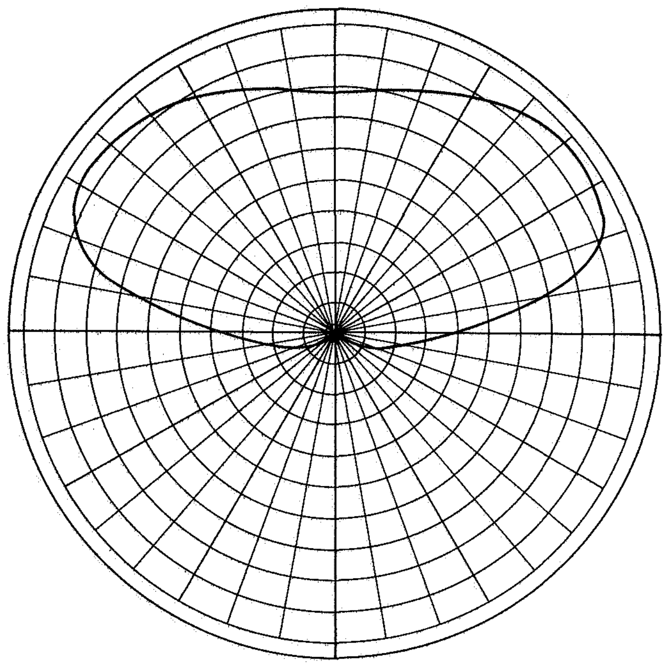


Figure 7

X-Band Antenna, H-Plane Pattern

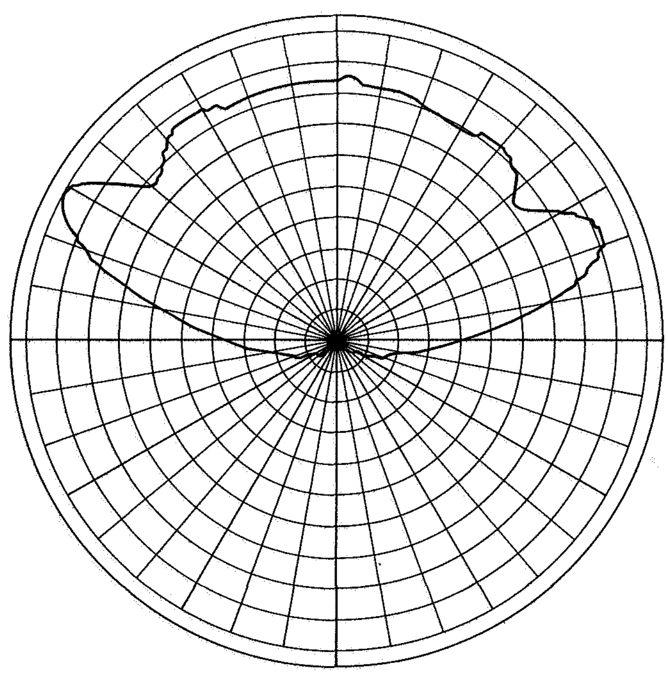


Figure 8

X-Band Antenna, E-Plane Pattern

2.3 SUPPORTING SUBSYSTEMS

The supporting subsystems for the satellite are the VHF telemetering system and instrumentation, the VHF command and control system, the electrical power system, the structure, the attitude stabilization system, and the thermal control.

2.3.1 Attitude Stabilization

The satellite will be stabilized to point toward the earth at all times to within ±5 degrees by a simple gravity-gradient stabilization system (Figure 9). The gravity-gradient stabilization and damping subsystem selected is extremely simple, with no moving parts, and is ideally suited for low-altitude, earth-oriented missions. It consists of a deployable boom, about 40 feet in length, and a series of fixed permeable hysteresis rods in strategic areas of the satellite. With the boom deployed, gravity-gradient torques will orient the satellite with the boom along the earth's radius vector. The librational motion of the satellite in the earth's magnetic field, due to initial tumbling conditions and external disturbances, will produce an alternating magnetization of the rods. The rods have lossy hysteresis characteristics and will dissipate energy as this occurs, thus damping the satellite motion.

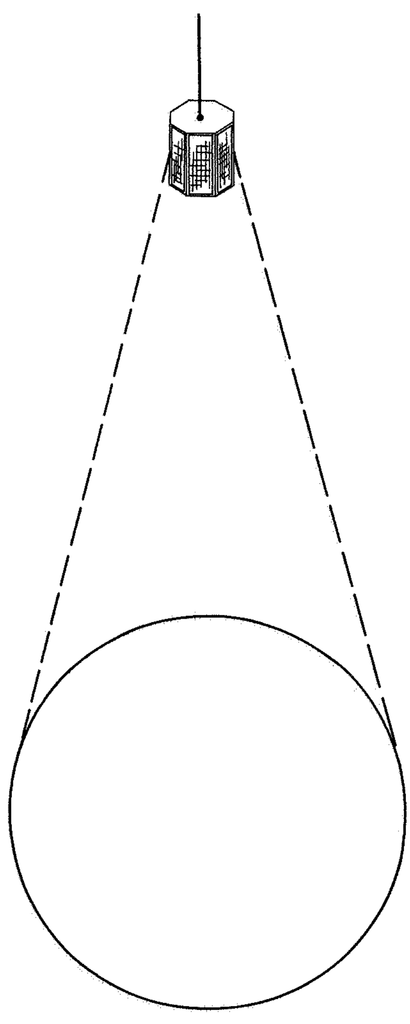


Figure 9. Gravity-Gradient Stabilized ERS

The proposed gravity-gradient stabilization and damping system is virtually identical to that first used on Satellite 1964-26A, launched into a 550-nautical-mile orbit on June 3, 1964, under the sponsorship of the U.S. Navy. This satellite, designed by the Applied Physics Laboratory of Johns Hopkins University, achieved gravity-gradient stabilization from a peak libration angle of 40 degrees to 7 degrees in a period of approximately four days. At least 10 satellites of this type have flown successfully.

Since the gravity-gradient torques align a specific axis of the satellite with the local vertical, with no preferred orientation, it may be possible for the satellite to be upside-down after the original capture if tipoff rates are greater than expected. If this occurs, the satellite can be inverted by using a simple technique developed by the Naval Research Laboratory. The boom is partially retracted so that the effective satellite moment of inertia is reduced to one-half the nominal value. The satellite will then start to tumble at twice the orbital rate. One-quarter orbit later, the satellite will be turned over, the boom is re-extended, and the damping rods cause it to be captured in the proper orientation. This type of maneuver has been performed successfully many times on various NRL satellites.

The stabilization performance requirements that have been established are that the satellite axis be aligned to the earth's radius vector to within 5 degrees after six weeks in orbit for nominal worst-case booster separation conditions. System analysis by TRW Systems on similar configurations indicate that these requirements can be achieved for an installed weight of approximately 2.5 pounds for the entire stabilization subsystem.

The stabilization subsystem is made up of a single, deployable boom and a set of fixed hysteresis damping rods. The boom would be selected from among the standard SPAR Aerospace Products (formerly deHavilland of Canada) STEM's (Storable Tubular Extendible Member) which have flown on scores of satellites, one of SPAR's smaller MINISTEMS, or similar booms produced by Fairchild-Hiller or Westinghouse. These booms are made of a flat strip of beryllium copper which assumes a tubular shape of high strength when deployed and is coiled on a small drum for storage. The selected boom would be about 40 feet long, carry a tip weight of 0.8 pound, and have an extension and retraction such as the Clifton Precision Products D.C. gear motor type 7801, requiring 12 volts and using up to 2 watts of power. The final selection of the type of boom will be made during the preliminary design study, based on performance requirements, availability, and power requirements.

The boom mechanism is released when a pyrotechnic guillotine cuts a tie-bolt holding the unit to the satellite apex. The guillotine will be a type similar to Holex model 2802 (0.5 ampere no-fire, 1.5 ampere all fire, 5 millisecond functioning time for the recommended 5-ampere firing current). The motor limits the boom extension rate to 1.5 ft/sec, thus preventing buckling of the boom from tumbling motion before deployment. Tumbling after deployment is negligible.

The damping mechanism consists of permeable rods attached in a grid at the spacecraft's equatorial plane, exactly like the damping mechanism for the magnetically-stabilized Test and Training Satellite (one of the ERS series) built by TRW for NASA Goddard Space Flight Center and launched successfully on December 13, 1967. Additional grids could be placed on the backs of the solar panels if necessary. Typical rods will be arranged as shown in Figure 10. Eddy current damping can be added for even faster acquisition times by placing the permeable rods within aluminum tubes. This concept was used by TRW in the design of a magnetic stabilization system for the German 625A satellite. Foam rubber corks would be used to block the ends of the tubes and support the rod.

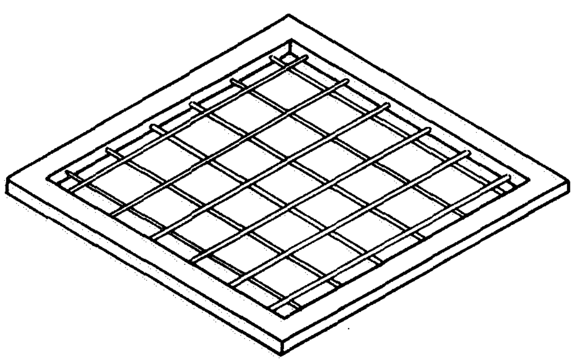


Figure 10
Typical Hysteresis Damping Rod Configuration for ERS

This would allow the rods to float within the tubes and maintain sliding contact only with the internal surface of the tube. This way the tube not only provides eddy current damping but also protects the permeable rods from the effects of launch vibration and thermal cycling.

Tests will be conducted to determine the optimum rod pattern and spacing for maximum damping. The L/D ratio of the rods should be as large as possible for best damping. Since manufacturing tolerances limit the minimum rod diameter to about 0.045 inch, the rods will be located in a spot to allow them to be as long as possible. At the central plane they can be up to 15 inches in length.

Acquisition time and the steady-state performance of the stabilization subsystem will be analyzed with a special computer program developed by TRW Systems for gravity gradient and magnetically stabilized satellites of this type. The differential equations describing the six-degree-of-freedom motion of an orbiting body under the influence of external perturbing forces and torques are strongly nonlinear and must be integrated numerically. All of the torques acting on the satellite are small so "rough" solutions often mask the true response.

The significant mission requirements which affect the design include the orbit altitude, eccentricity, and inclinations, the required acquisition time and maximum long-term pointing error, and such satellite characteristics as mass properties, magnetic moment, and exposed area. The gravity-gradient subsystem will be designed to meet the required performance, in terms of acquisition time and long-term pointing error, for a worst-case condition and the performance for a nominal case will also be determined on the in-line hybrid computer designed to dispense completely with the complex nonlinear mathematical simulation of the hysteresis phenomenon. It consists of three major components: digital and analog computers and a hardware component which utilizes permeable rods, identical to those used on the flight spacecraft, to generate the hysteresis torque required for the simulation. This computer combines the best features of analog and digital simulation with the experimental measurement of hysteresis damping of permeable rods.

The in-line hybrid computer was first applied to the design of the attitude control system of the Test and Training Satellite (TTS). For this reason, the dynamic equations presently programmed are oriented specifically to the simulation of this satellite. The results demonstrated that this simulation concept provides the most realistic description yet attempted of the hysteresis damping phenomenon of permeable rods, and therefore increases significantly the accuracy of the simulation. For satellite stabilization systems which use a large number of damping rods (like the TTS), the unavoidable variation in the hysteresis damping capability of these rods can be included in the simulation, thus eliminating the need to rely on approximations based on the average properties of the rods.

2.3.2 Mechanical

Preliminary design of an octagonal prism model of the ERS has been completed. This model is ideally suited to the NAVSTAR test program since:

- It can accommodate the payload and supporting equipment with excellent packaging efficiency.
- The cross sectional area is exactly right for the requirement of the solar cell power supply.
- The flat end surface provides a good ground plane and mounting surface for the L-band and X-band antennas.
- The prism configuration is very adaptable to the single-boom gravity-gradient stabilization system.
- This configuration allows the VHF command and telemetry dipole antenna elements to be mounted without interfering with the microwave antenna patterns.
- The ejection system is designed for secondary payload applications without modifications to the booster vehicle.

The prism satellite measures 16 inches across the diagonal of the octagon and 14 inches high along the side face. The top is flat and will accommodate the microwave antennas. The bottom is slightly conical for structural strength and will accommodate additional solar cells as well as the ejection tube and gravity-gradient boom exit port.

The satellite has an all-aluminum structure consisting of the two octagonal end panels made of a shallow sheet aluminum drum. At each vertex is a vertical aluminum structural member which joins the upper and lower end panels and provides attachment points for the two mounting shelves. The two mounting shelves are located near the bottom panel and near the midplane of the prism. The shelves are supported at the center by a machined aluminum column which also serves as a guide during separation from the booster. Most of the structural loads during the launch are taken through the center column or ejection tube. The remainder of the loads are taken through four attachment points into the ejection canister which surrounds the spacecraft until ejection. The central mounting shelf serves as a heat sink for the transmitters and accommodates the hysteresis damping rod matrix. The top panel bears only the weight of the microwave antennas. Excessive resonant vibration of this panel is prevented by structural cross members. The four VHF antenna elements are mounted at opposing vertices on the vertical members near the midplane. The eight rectangular solar panels close out the structure and form the prism. These panels also serve as an integral part of the structure and bear part of the structural loads imposed by the launch. Figure 11 shows a model of an 11-inch-high version of the octagonal ERS mounted in the ejection canister. The proposed NAVSTAR test configuration would be identical except for an increase in height of 3 inches.

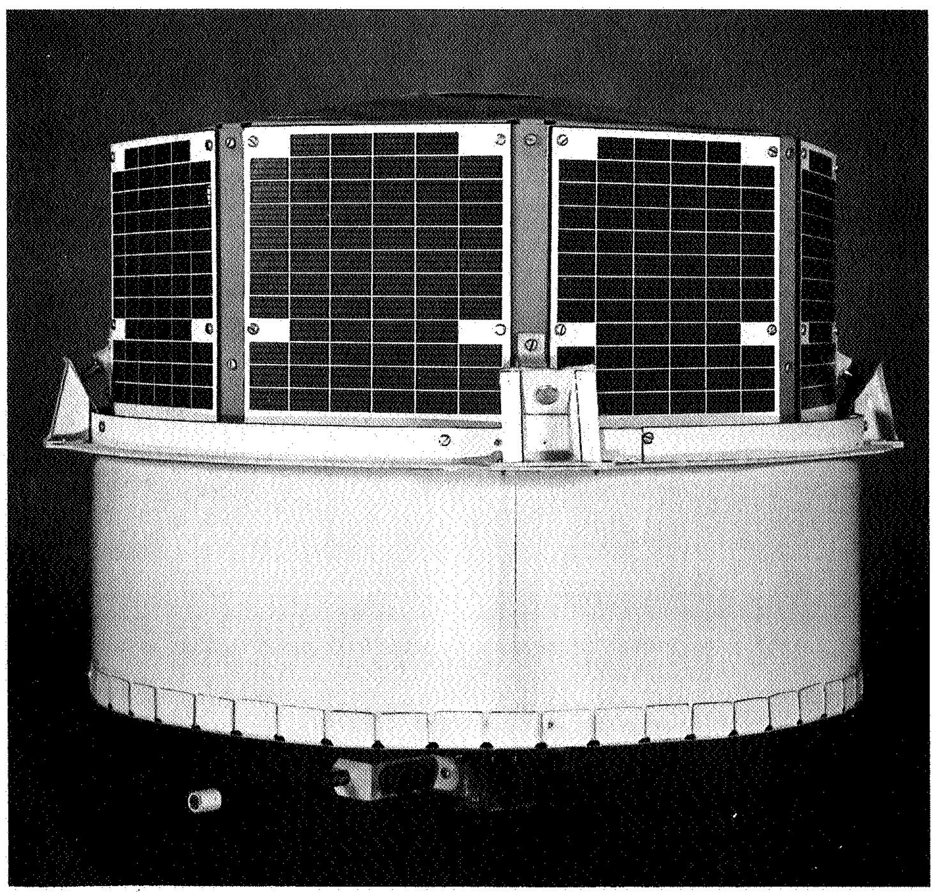


Figure 11. ERS Prism Satellite Mockup

A preliminary layout of the equipment within the structure is shown in Figure 12. The MISTRAM transponder occupies the upper shelf with the remainder of the experiment payload mounted on the underside of the middle shelf. The payload support hardware is mounted on a shelf at the bottom of the prism, but not directly to exterior-facing surface for greater thermal isolation. This layout indicates that the size of the satellite is adequate but not excessive to accommodate the required hardware. Table 4 summarizes the sizes and weights of the various components. The estimated total weight of the installed hardware is 58 pounds. The estimated weights for the satellite structure is 6 pounds and for the ejection canister 4 pounds. The total weight of the booster-mounted package is nominally 68 pounds.

At launch and during boost the satellite is mounted in a container which is the only interfacing item with the injection bus vehicle. The only satellite interface connections are four machine fasteners and a 22-volt electrical coupling for the pyrotechnic pin puller. This canister is a cylindrically-shaped sheet metal structure with a stiffening flange at its open end. This and a machined fitting at its truncated apex form the basic support for the satellite and incorporate the ejection mechanism. Four equidistant sheet steel fittings, which nest with the underside of the open-end flange and with the outer surface of the cylinder, provide the four hard-points for attachment of the satellite/canister assembly to the booster support structure.

The apex fitting is designed to support a hard-anodized tubular ejection post coincidental with the axis of the cylinder. Additionally, it is flanged to receive the pin puller sheet metal support channel. The pin from the puller passes through the wall of this fitting and enters a hole in the satellite fitting, effectively locking it to the canister. All loads up the ejection post are carried in shear and bending by this pin puller.

At the open-end hard-points, four adjustable sway-braces bear against the satellite frame through load-distributing pads near the equatorial plane. With the satellite balanced such that its center of gravity is at the centroid of the prism, loads parallel to equatorial plane will pass, in bearing, through these sway braces directly into the hard-point

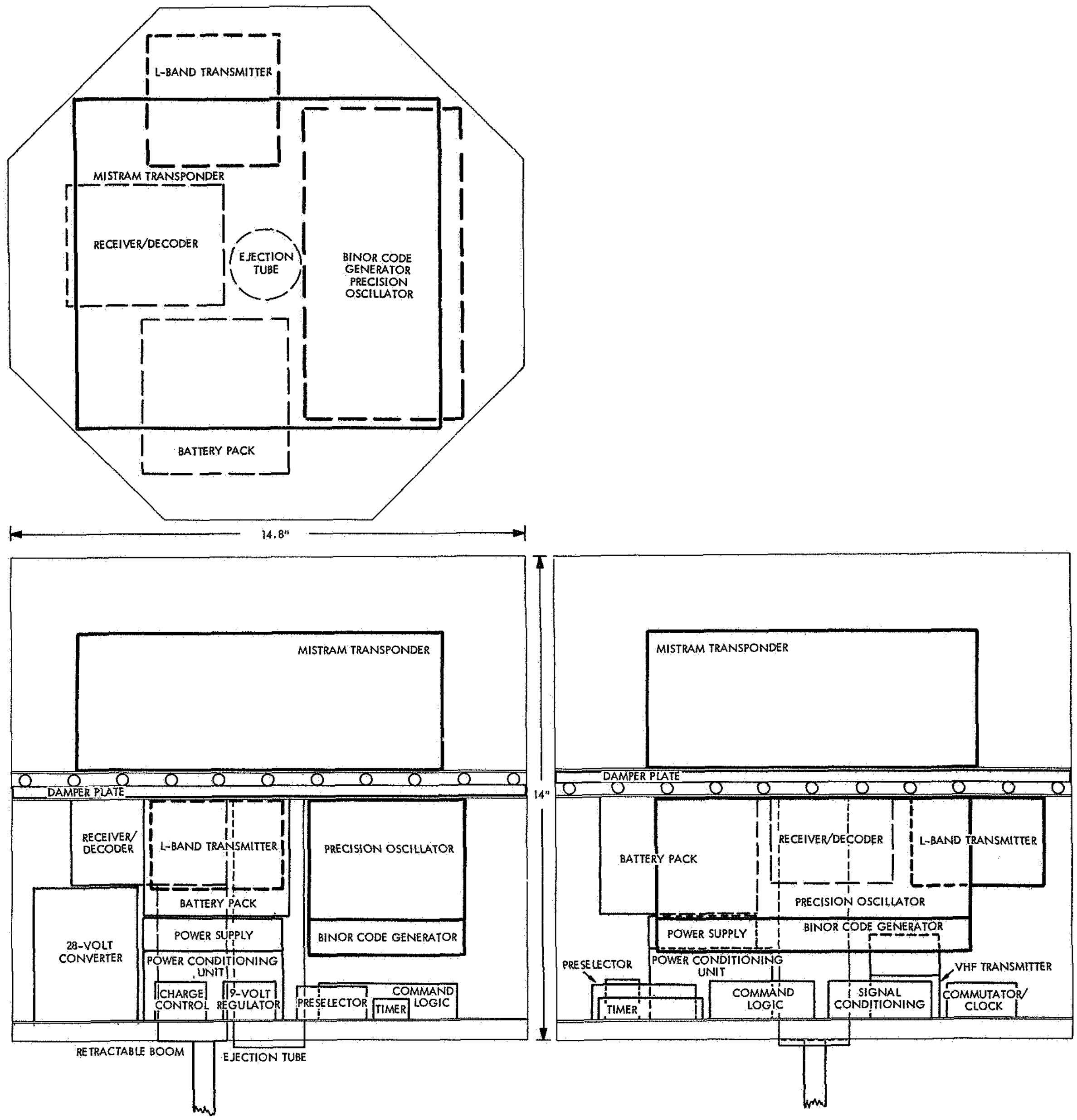


Figure 12. Preliminary Layout of Equipment in Satellite

fittings and into the booster support structure. The small imbalance from the eccentricity of the center of gravity to the sway brace plane produces relatively small loads at the lower fitting which are carried in shear through the cone to the hard-point fitting and pass into the booster support structure.

Down loads (those tending to move the satellite deeper into the canister) are reacted by the sway braces directly.

When the satellite is installed in the canister, the VHF antennas are located at the vertices near the equatorial plane. Each antenna element

Table 4. Satellite Weight and Volume Summary

	Size (in.)	Volume (in.^{3})	Weight (lb)
Payload			
L-band transmitter	3-3/4 × 3-3/4 × 2-1/2	35.0	2.0
BINOR code generator	9 × 4-1/2 × 1	40.5	2.0
Precision oscillator	9 × 4-1/2 × 3-1/2	141.5	4.0
MISTRAM transponder	9-1/2 × 10-1/2 × 4	400.0	17.5
Waveguide feeds	1 × 1/2 × 24 (est.)	9.0	0.2
VHF Telemetry			
VHF transmitter	4-3/4 × 2 × 1-1/8	10.7	0.3
Subcarrier oscillator	3-1/8 × 2 × 1-1/8	7.0	0.2
Commutator	3 × 1 × 1	3.0	0.2
Clock	1-1/2 × 1 × 1	1.5	0.1
Signal conditioning	3 × 4 × 1	12.0	0.3
VHF Command			
Preselector	3 × 2 × 1-1/8	7.0	0.2
Receiver/decoder	4-1/2 × 3-1/2 × 2-1/4	35.0	1.7
Command logic	4 × 3 × 1	12.0	0.3
Timer	3 × 1 × 1	3.0	0.2
Power			
Battery (10-cell)	4-1/4 × 4-1/8 × 3-3/4	66	6.4
28-volt static inverter	3 × 4 × 6	72	2.5
Power conditioning unit	4 × 3 × 2-1/2	28	0.8
Battery charge control	1 × 1 × 1.5	1.5	0.1
Low voltage sensor	1 × 1 × 1.5	1.5	5.0
Solar panels (1850 cells)	(external)	—	5.0
Stabilization			
Retractable boom	2 × 2 × 4-1/2	18.0	0.4
Damper plate and heat sink	14.5 octagon × 5/8	108.0	10.0
Structure, etc.			
Satellite shell (without panels)			6.0
Canister			4.0
Antennas			0.5
Wire and cable			3.0
		1012.2	68.0
Available volume		2430.0	
Packing density		42%	

is coiled into an antenna cup. The antenna elements are made of preformed steel tape and are grooved in such a way that they are retained in the antenna cups when held against a knife edge on the canister. During ejection, as the satellite rises on the ejection tube, the antennas are released and move toward their deployed positions.

As on all other ERS, ejection from the injection bus vehicle is by means of a compressed helical spring of stainless steel released by a squib-actuated pin puller. During boost and injection the satellite is contained within its canister, its antennas suitably coiled and supported. Ejection is by signal from the bus which actuates the pin puller and unloads the ejection springs. Spring actuation results in a separation velocity of approximately 3 ft/sec from the launch vehicle.

2.3.3 Thermal Control

Thermal control requirements of the ERS series are satisfied by passive techniques, including the selection of exterior surface materials, careful arrangement and mounting of components to take advantage of conductive and radiative heat exchange, and appropriate use of low emissivity coatings or insulation to increase the effective thermal time constant of critical components.

In general, satellite temperature is determined by a balance of heat generated within the satellite, heat absorbed from external sources (i.e., sun and earth) and heat emitted by the satellite. Mean temperatures can be controlled by changing the effective absorptivity to emissivity ratio of the outside coating of the satellite. Most of the outside surfaces of the ERS are covered by solar cells which are protected with cover glass.

For a gravity-gradient stabilized satellite, the sun-satellite line sweeps out, once per orbit, a cone whose half-angle is the angle between the solar vector and the orbit plane normal. The cone half angle increases a maximum of one degree per day until the cone degenerates into a plane. This is the condition in which the satellite rotates about the axis normal to the orbit plane once per orbit with the solar vector normal to that axis. The cone then moves to the other side of the satellite and the half-angle decreases until a minimum is reached three months later. If the orbit plane is normal to the ecliptic plane, the cone degenerates into a line.

In any case, the cone reaches a minimum half-angle equal to the angle between the orbit plane and the normal to the ecliptic.

It is clear therefore that surfaces of the satellite are exposed to the sun or to deep space for relatively long periods. The external thermal radiation properties of the solar arrays are such that the equilibrium temperature of arrays pointing at the sun can rise to approximately $120^{o}C$. Similarly, shaded panels can reach dangerously low temperatures. Steps must be taken to limit temperature fluctuations of solar panels and further limit temperature gradients and fluctuations in sensitive components. Batteries and critical electronic subsystems, for example, must be insulated from the solar panels and, if possible, mounted in good thermal contact with each other.

Previous studies of thermal control associated with gravity-gradient stabilized satellites have indicated that the internal satellite structure should be relatively open and internal solar array surfaces coated with high emissivity black Cat-a-lac epoxy resin base paint (ϵ = 0.86). This allows shelf-mounted equipment to exchange heat with all solar array panels, thereby effectively averaging the sink temperature to which the power dissipated in the equipment is radiated. Also, with a relatively open internal structure it is possible that those solar panels which are illuminated can effectively reradiate a portion of the absorbed solar energy to shadowed panels. This provides an effective way of decreasing the peak temperature of illuminated panels for greater electrical efficiency. At the same time, the temperature of shaded panels is increased, thereby avoiding destructively low temperatures. For example, a solar array panel which would otherwise rise to approximately $120^{o}C$, can be made to operate at approximately $64^{o}C$ if the back of each panel is treated to maximize heat exchange between hot and cold panels. Equipment shelves and components can be coated with paint in certain local areas, leaving other areas bare to achieve the proper local emissivity. In high power dissipation areas, black paint can be used while in areas of low dissipation a combination of black paint or Rinshed-Mason silicone base aluminum paint and bare metal can be used. A striped pattern can be used in some areas to achieve the desired emissivity without requiring the development of a coating having a particular emissivity. Low emissivity regions on equipment shelves can sometimes be used effectively to enhance radiation exchange between solar array panels by the reflection of radiation from a hot panel to colder panels.

In reaching a specific design for the thermal control of the satellite, a TRW thermal analyzer computer program will be used. The TRW TAP is an n-dimensional, asymmetric, finite difference, IBM 7094 digital routine in which the heat transfer parameters are entered as their electrical analogs in an R-C network. The specified boundary conditions are applied to the network and the program obtains a solution through an iterative process. Radiation configuration factors required to establish the network are obtained with the TRW Shape Factor 1 program. Areas of interest are divided into a specified number of flat plate elements. The incremental configuration factor between an element of the emitting area and an element of the receiving area is determined and summed with all increments to produce the configuration factor for the total area. To eliminate shadowing errors, the program verifies that the line of sight between each receiving and emitting element is uninterrupted before including the corresponding incremental configuration factor in the total.

The total radiation heat transfer coefficient is calculated with the TRW Script F program. Script F represents the net radiation exchange between two areas. Direct radiation, and radiation from one area to another via reflections from all the other areas of the enclosure, are considered. The computer program is a matrix inversion routine; the matrix being an array of reflectivities and configuration factors between all surfaces. To apply these computer programs, two models of the satellite are required. One model is used to determine the overall temperature distribution, the other to obtain more detailed temperature distributions and component temperatures.

2.3.4 VHF Telemetry

The VHF telemetry system included for this mission (Figure 13) will provide housekeeping information (temperatures, voltages, etc.), monitor the functional parameters of the payload, and serve as a beacon

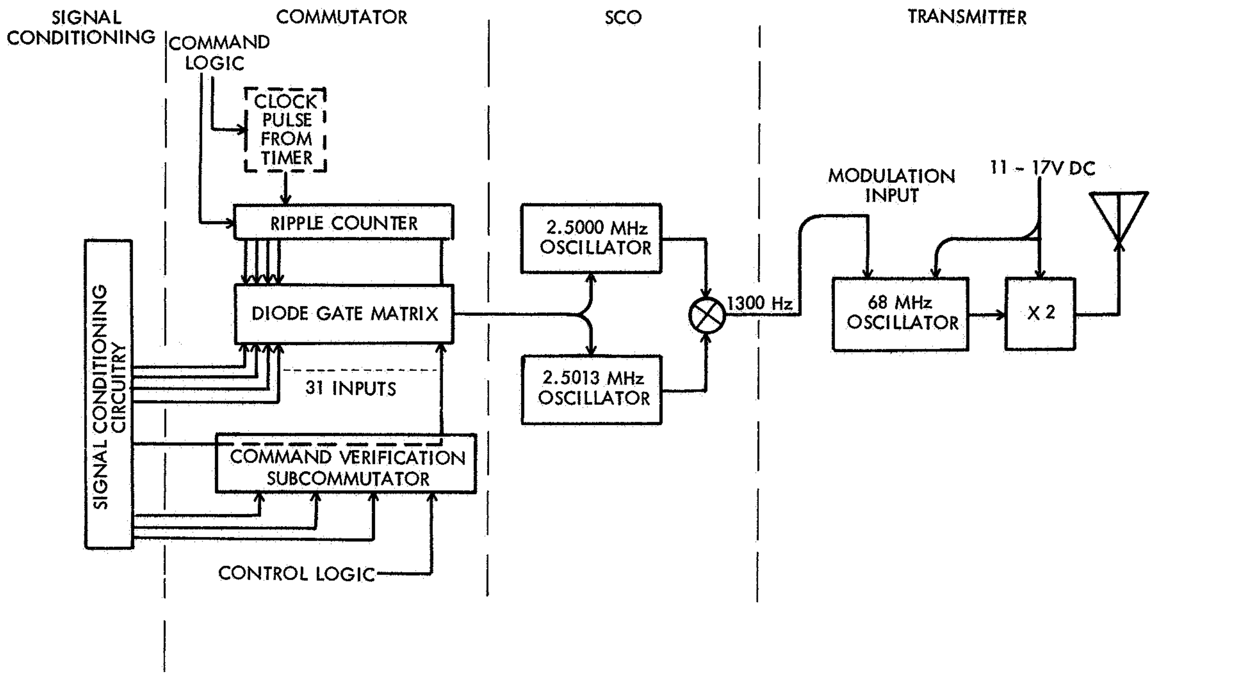


Figure 13. Telemetry System Functional Block Diagram

for NASA/STADAN interferometer Minitrac stations. The entire telemetry and signal processing system is nearly identical to that of earlier ERS satellites.

The VHF transmitter (Figure 14) radiates 100 milliwatts at 136.8 MHz and is normally used with the NASA/STADAN network, which has given essentially worldwide coverage for all previous ERS programs. It is also compatible with other ground stations equipped for 136 MHz. The telemetry and signal processing systems include the VHF transmitter, subcarrier oscillator, signal processing, electronic commutator, and antenna.

Data will normally be obtained at a NASA/STADAN station on magnetic tape and on oscillograph paper. Data will be received on IRIG subcarrier channel No. 5. It will be in the form of sequential pulse amplitude signals. The duration of each segment will be 0.50 second.

The telemetry system is a PAM/FM/PM system with a crystal controlled carrier frequency of 136.8 MHz. Analog data format is employed and is transmitted sequentially. The PAM/FM input to the transmitter is fed to the 68.4 MHz oscillator, which is thus phase modulated. A push-pull doubler in the next stage provides a PAM/FM/PM telemetry signal at 136.8 MHz at a level of 100 milliwatts. The modulation index is 1.0 radian peak. A careful test of the telemetry system

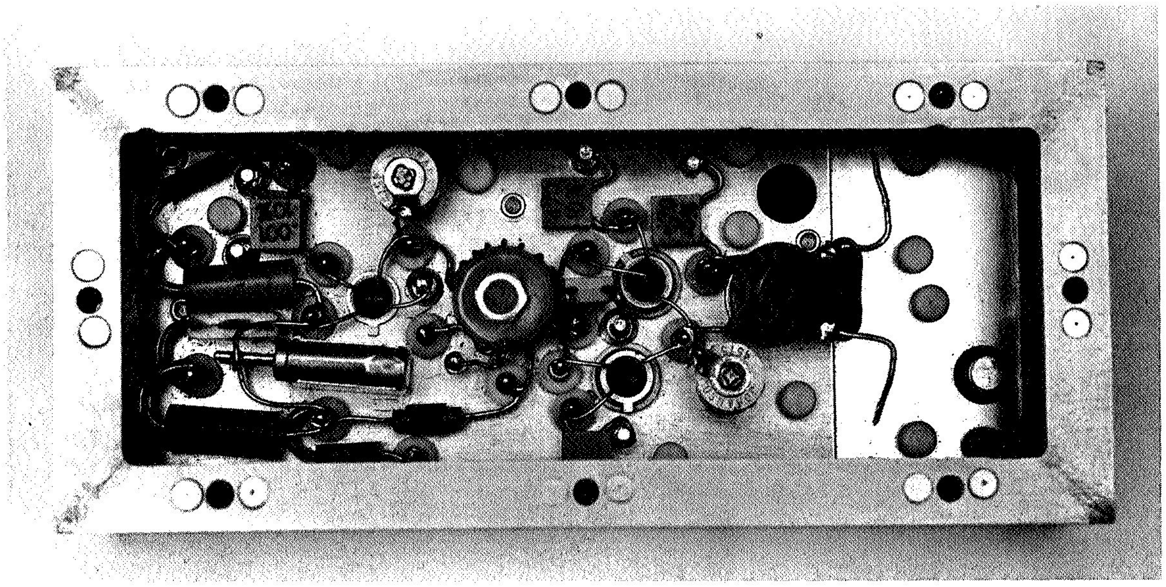


Figure 14. Telemetry Transmitter

of an earlier ERS over radio link proved its accuracy to be better than one percent under all conditions. Telemetry data reception with the 100 milliwatt transmitter has been successfully accomplished in actual flight (ERS 13, 1964 40C) with a STADAN 85-foot dish antenna throughout a 108,000 kilometer apogee. The RF power budget calculations based on a nominal 300 nautical mile altitude orbit are shown in Table 5, and indicate ample margins for quality data at the projected maximum range. Margins would be adequate for much higher orbits. The minimal antenna for the STADAN network is assumed for these calculations.

The subcarrier oscillator converts serially commutated data to an FM modulating signal. It is compatible with NASA/STADAN network equipment or other appropriate facilities and operates on IRIG 5 subcarrier band (1300 Hz $\pm$ 7.5 percent). Use of these parameters permits transmission of DC and low frequency AC data of good quality with a minimum of satellite RF power. It also results in easy-to-obtain data records. Channel 5 will normally permit rise times of less than 20 milliseconds with the selection of appropriate ground station equipment (a linear-phased, or Bessel discriminator output filter, for which $\tau\beta = 0.35$ is appropriate). By appropriate action at ground stations in selection of the discriminator output filter, rise times up to five times faster can be obtained. This SCO employs two crystal controlled oscillators, one at 2.5000 MHz and

Table 5. Downlink RF Power Budget

Transmitter power (100 mw)	+ 20 dbm	
Antenna probable gain (worst angle)	- 3 db	
Space loss at 1,200 n mi (300 n mi altitude at 5 deg)	-143 db	
Ground antenna minimum gain	+ 19 db	
Net transmission loss	-127 db	
Total received power	-107 dbm	
Receiver noise spectral density (1880°K inc 4 db N.F.)	-166 dbm/cps	
<u>Carrier Performance</u>		
Carrier modulation loss (1.00 rad)	- 2 db	
Received carrier power	-109 dbm	
Carrier loop input noise BW (B_N = 100 cps)	20 db	
Threshold S/N in B_N	6 db	
Threshold carrier power	-140 dbm	
Margin		+31 db
<u>IRIG No. 5 Subcarrier Performance</u>		
Subcarrier modulation loss (1.00 rad)	- 4 db	
Received subcarrier power	-111 dbm	
Subcarrier noise BW (2 B_L = 400 cps)	26 db	
Threshold subcarrier S/N	6 db	
Threshold subcarrier power	-134 dbm	
Margin for threshold		+23 db
Discriminator output filter (2.6 B_0 = 14), 6 cps filter	11 db	
FM improvement factor	+ 8 db	
Data quality equivalent S/N at R = 1,200 n mi		+46 db

the other at 2.5013 MHz. Each has a voltage variable capacitor pulling its crystal, but in opposite directions. The beat frequency with 3.0 volts at the input is 1300 Hz, the center frequency of IRIG Channel 5. This is changed up or down 7.5 percent by 0.6 volts either side of 3.0 volts. These levels are convenient for operation with the standard ERS power system. An IRIG Channel 5 SCO used on previous ERS missions is shown in Figure 15.

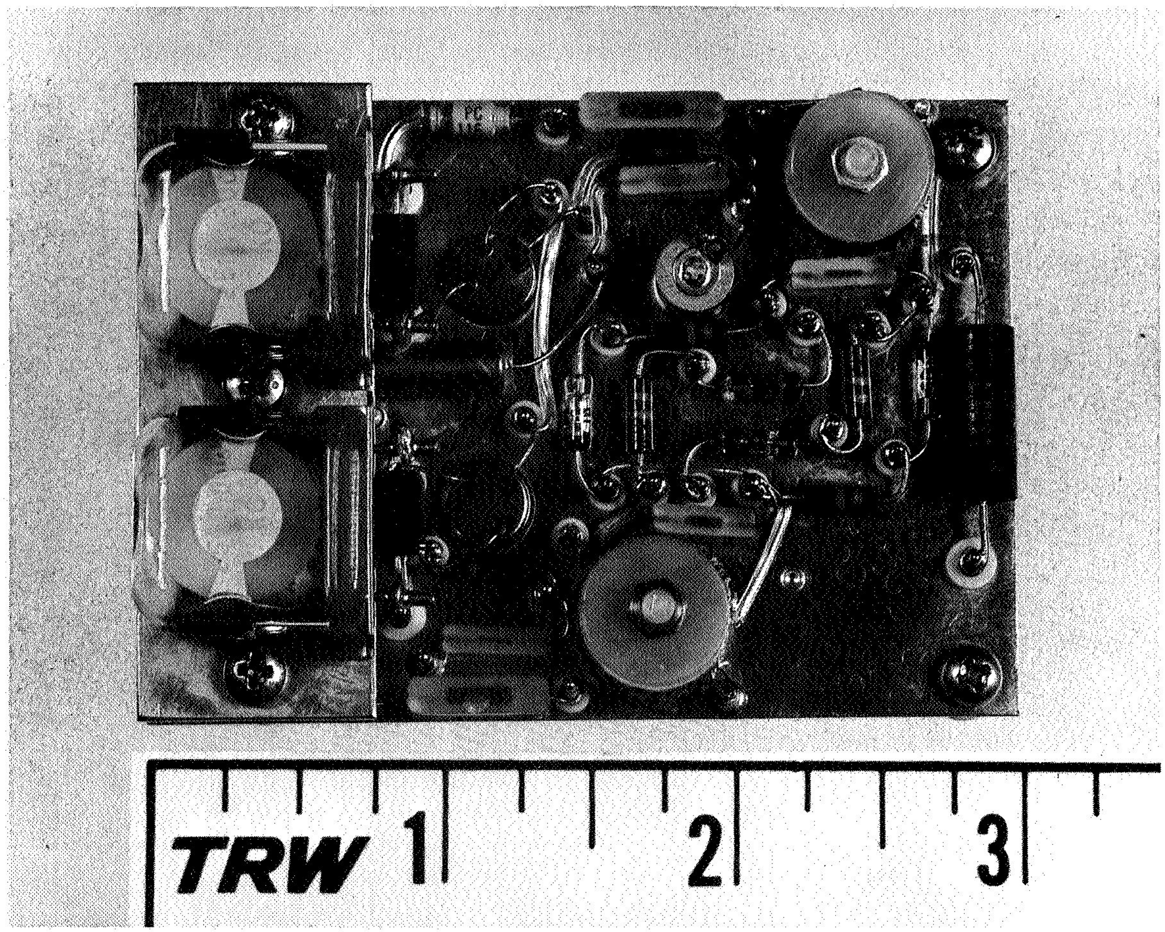


Figure 15. IRIG Channel SCO

The solid-state commutator is identical to those built for earlier ERS missions. The commutator converts analog data that are applied in parallel fashion into a series signal for modulation of the IRIG-5 SCO. In its normal mode of operation, the commutator continuously processes 31 parameters represented by 31 parallel channels of information. It sequences between channels each 500 milliseconds.

The commutator is a totally integrated design that uses the characteristics of the integrated circuits themselves to accomplish matching in the diodes and thereby achieve greater accuracy. Since integrated circuits are used, a 5 volt supply must be employed. Thus the input level to the commutator must be less than 5 volts. The signal into the commutator is 3 volts ± 0.6 bolt band edge to band edge.

The telemetered parameters can be categorized as system calibration, housekeeping, command verification, and battery state of charge. The telemetered parameters are listed in Table 6.

Table 6. Telemetry Parameter List

1.	Sync channel (Hi/Lo calibrate)	16.	28-V current
2.	Midband calibrate	17.	Solar cell array voltage
3.	Boom status	18.	Transmitter (L-band) temperature
4.	MISTRAM Channel 1 AGC	19.	Commutator temperature
5.	MISTRAM Channel 2 AGC	20.	Oscillator temperature
6.	MISTRAM Channel 1 power out	21.	Battery temperature
7.	MISTRAM Channel 2 power out	22.	Oscillator monitor
8.	MISTRAM Channel 1 phase detector	23.	Oscillator monitor
9.	MISTRAM Channel 2 phase detector	24.	Structure temperature
10.	MISTRAM temperature	25.	Spare
11.	28-V monitor	26.	Spare
12.	+12-V monitor	27.	Spare
13.	-12-V monitor	28.	Verification Command 2 and 3
14.	+5-V monitor	29.	Verification Command 5 and 6
15.	Battery bus monitor	30.	Verification Command 7 and 8
		31.	Spare

The beacon and telemetry antenna system consists of two dipole antenna elements mounted to provide a maximum of isolation of the transmitter and command and control antenna systems. The antenna elements are made from preformed steel tape.

2.3.5 VHF Command and Control

The primary function of the VHF command receiver and control system (Figure 16) is to turn on the transmitters for each test and to select the desired BINOR code clock rate. It also will be used for turning off the VHF transmitter for radio silence or maximum battery charge rate, or when the experiment is complete, and to control secondary functions such as turning off individual components.

The command receiver, tone decoder, command decoder, secure command evaluator are identical to units being used on current flights of the ERS vehicles. Different distribution and logic units are required for each ERS mission and one will be tailored to meet the specific needs of this program.

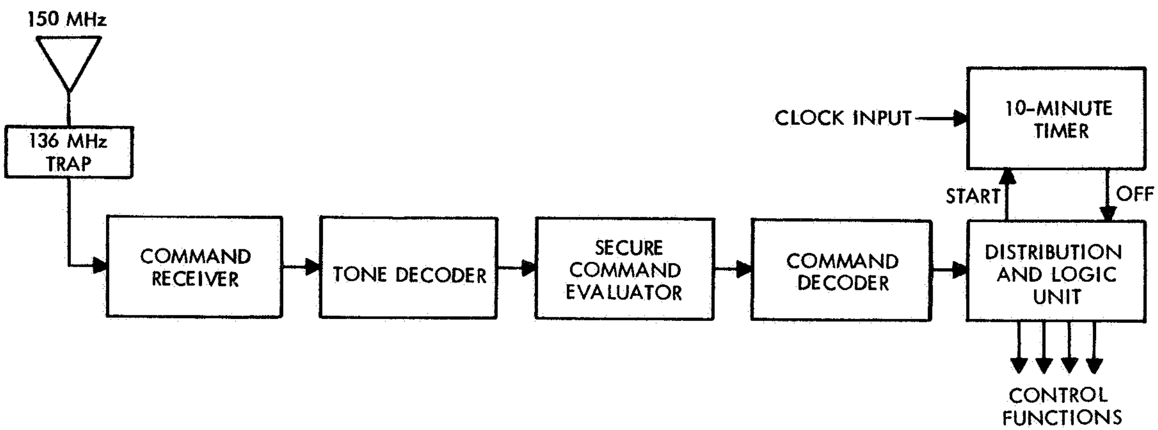


Figure 16. Command and Control Systems Functional Block Diagram

Commands will normally be initiated from the Fort Myers ground station, although commands can also be accomplished from other sites having appropriate equipment. The STADAN/Minitrack stations that are within range of the planned orbit which have command, tracking, and data acquisition capability are listed below together with their code name and locations.

Station Location	Station Code Name
Barstow, California	(MOJAVE)
Blossom Point, Maryland	(BPOINT)
Fort Myers, Florida	(FTMYRS)
Johannesburg, South Africa*	(JOBURG)
Lima, Peru	(LIMAPU)

*Denotes higher power command capability

Station Location	Station Code Name
Orroral, Australia*	(ORORAL)
Quito, Ecuador*	(QUITOE)
Rosman, North Carolina*	(ROSMAN)
Santiago, Chile*	(SNTAGO)
Vandenberg, California*	(AF-WTR)

The standard ERS command receiver (Figure 17) is a commercially available fixedtuned VHF AM receiver designed for satellite use and flight qualified. It measures 3.5 xy 4.5 by 1 inch and weighs less than 1 pound. It is entirely solid-state (silicon only) and is a single-conversion receiver utilizing silicon transistors. One 18.1 MHz crystal filter is used in conjunction with low-level demodulation and high-gain amplification. Dynamic range is 80 db; sensitivity is -110 dbm; bandwidth is 10 kHz, which is adequate for worst-case doppler. Only 137 mw of power are required for continuous standby operation from the unregulated solar panel output. An RF power budget is shown in Table 7, which assumes the use of equipment presently operational at NASA/STADAN stations. The power budget indicates that commands can easily be received over the maximum slant range for this mission (1000 n mi).

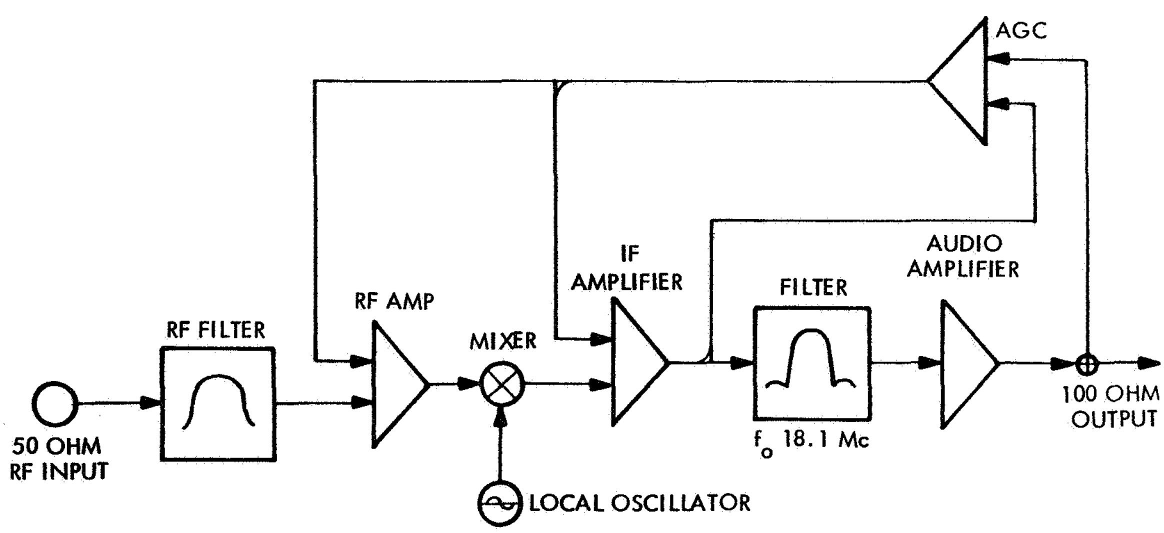


Figure 17. Command Receiver Functional Block Diagram

*Denotes higher power command capability

Table 7. RF Power Budget, Uplink

Ground transmitter power (170 W)	+ 52 dbm
Ground antenna gain	+ 13 db
Cable and miscellaneous losses	0
Polarization loss	- 3 db
Receiver antenna gain (worst angle)	- 6 db
Spacecraft diplexer insertion loss	- 3 db
Space loss at 1000 n mi (300 n mi altitude at 10 deg)	-142 db
Total received power	- 89 dbm
Receiver sensitivity	-110 dbm
Margin for weakest station	+ 21 db
Margin for most stations (8)	+ 45 db

The tone decoder is a companion piece to the command receiver, designed to operate with it. It consists of eight bandpass filters, one for the address tone and seven for execute tones. The tone frequencies are those specified to NASA Aerospace Data Systems Standards X-560-63-2. The tone decoder is housed in an enclosure which matches the command receiver so that they may be stacked. The decoder measures 3.5 by 4.5 by 1.25 inches and weighs 10 ounces.

The distribution and logic unit establishes the desired circuit configurations for effecting the selected mode of satellite operation, for causing the selected sequences, for distribution actuation signals, and for providing necessary interlocks. In contrast to such units as the command receiver, new or modified distribution and logic unit designs must be developed for each new ERS satellite mission such as in the NAVSTAR test program.

The command decoder converts received sequences of tones into as many as 21 different discrete commands, although only the following nine are anticipated:

1) Address

2) All transmitters on – BINOR clock rate 1

3) All transmitters on - BINOR clock rate 2

4) VHF telemetry off

5) MISTRAM off

6) L-band off

7) BINOR modulation off

8) Extend boom

9) Retract boom

This method is compatible with equipment existing at all STADAN stations. The command logic is arranged for commands consisting of three sequential tones. A secure command evaluator circuit evaluates each sequence of tones and prevents unwanted function execution. An address tone must occur first, followed by two information tones. Each must be of 0.5 second duration with an 0.5 second silent period between. If these requirements are not met, the system resets and awaits the next command.

Commands 2 and 3 are used to initiate the test. Command 2 turns on the L-band transmitter, the MISTRAM transponder, and the VHF telemetry transmitter. It applies the 1280 enable signal to the BINOR generator such that the 2^{15} bit code length is selected. Similarly, Command 3 turns on the three transmitters but differs in that it applies the 320 enable signal to the BINOR generator such that the 2^{13} bit code length is selected. When either Command 2 or 3 is executed, a ten-minute timer is initiated. After the elapsed time, it commands the L-band transmitter and MISTRAM transponder OFF. The VHF transmitter remains ON so that the various functional and environmental parameters of the satellite can be monitored to evaluate the performance of the system.

When sufficient data has been received following a test, the VHF transmitter is commanded OFF by Command 4. Because of the low power of the VHF telemetering system, it can remain ON almost indefinitely. However, for maximum charging of the battery after a test, it can be turned OFF by any of the STADAN stations. It need only be ON as long as telemetered data is required, or for tracking purposes.

Commands 5 and 6 are included as a back-up to the ten-minute timer or to turn off the L-band transmitter of MISTRAM transponder before the end of the ten-minute test period. Neither command would normally be used during a routine test.

Command 7 provides an unmodulated L-band signal from the ERS to aid in measuring multipath reception conditions in an aircraft.

The single boom gravity gradient attitude stabilization system has an equal probability of stabilizing either in the boom-up or boom-down position. After orbit injection the BHF transmitter (only) will be ON to allow the satellite to be tracked. The initial command will be Command 8 which will extend the G. G. boom. After stabilization has occurred, a test will be performed (Commands 2 or 3) to determine if the satellite attitude is correct. An inverted position for the satellite will be indicated by the absence of signal strength from the L-band transponder and MISTRAM transponder since their antenna patterns are nominally hemispheric. If an inverted position is indicated the boom will be retracted by Command 9 and will be reextended about 1/4 orbit later by Command 8.

Timing is involved in the control of the commutator and experiment duration. Figure 18 illustrates the timer configuration. A crystal oscillator provides the basic timing reference. Ripple counters perform the countdown function. The ripple counters are all integrated circuits mounted in flat wafer modules.

The antenna system for the command and control function consists of two dipole antenna elements protruding from opposing vertices on the equator of the satellite. The receiving dipole is located in such a way

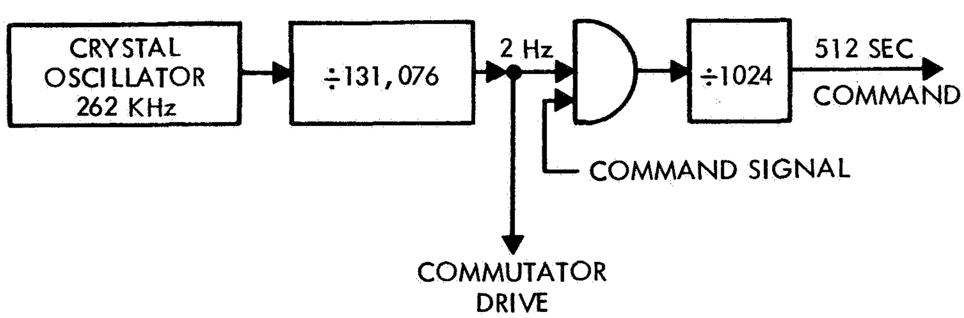


Figure 18. Timer Configuration

as to provide a maximum of isolation from the radiated VHF transmitting signals, i.e., perpendicular to that antenna. A radiation pattern for the dipole on an ERS of the 1967 40C configuration is shown in Figure 19. The antenna is made from steel tape. Methods of stowage and deployment are discussed in Section 2.3.2.

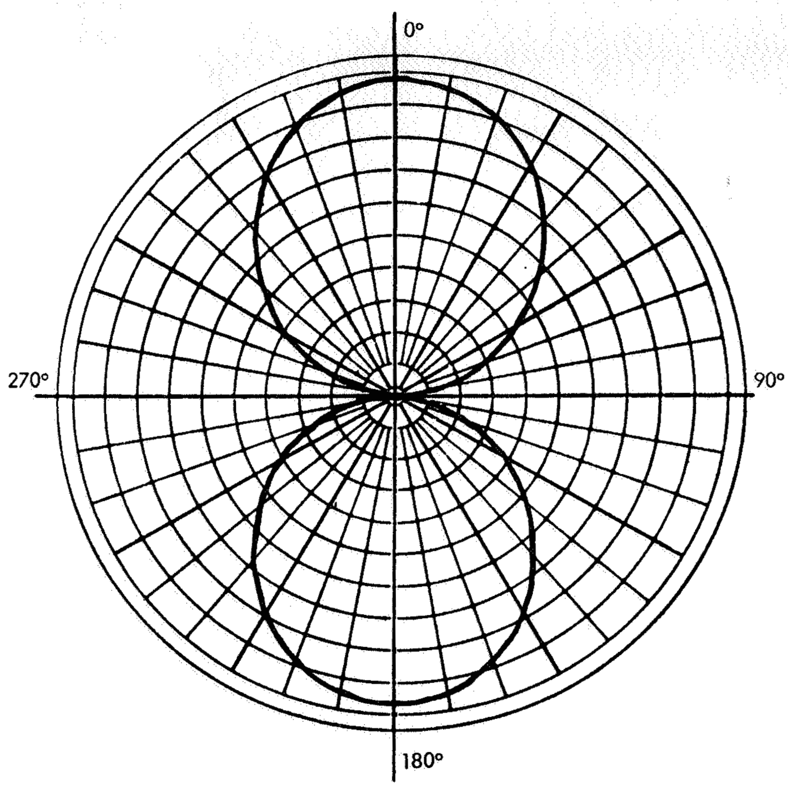


Figure 19. VHF Dipole Pattern. The antenna is parallel to the 90-270 degree line.

The command system and the transmitter operate simultaneously, and therefore steps must be taken to isolate the two. First, the receiving antennas are located at right angles to the transmitting antennas to maximize isolation between the two systems. Second, a 136 MHz trap is included at the receiver input for further isolation. This trap provides approximately 50 db isolation with a forward loss of 3 db. The receiver provides 70 db rejection to spurious signals and 60 db rejection to image signals.

2.3.6 Electrical Power Supply

The mission requirements for electrical power are set by the requirement for the payload to be exercised a maximum of three consecutive orbits under worst-case shadow conditions. This is the maximum number of consecutive orbits that the satellite is visible from the MISTRAM facility. In addition, the power system must provide the capability of payload operation each time the satellite passes over the Cape Kennedy range. i.e., nominally every 24 hours. The requirements of the payload and supporting subsystems are summarized in Table 8. The payload requires 95 watts of power for ten minutes per exercise and 4.5 watts of standby power.

Table 8. Satellite Payload Electrical Power Requirements Summary

Item	Power Supply	Normal Standby (VHF Off) (watts)	Transmitters "On" (watts)	VHF Transmitter "On" Only (watts)
MISTRAM transponder*	28V	0	45.0	0
L-band transmitter*	28V	0	45.0	0
Stable oscillator*	± 12V	3.0	3.0	3.0
BINOR generator*	+ 5V	1.2	1.2	1.2
Command system		0.3	0.3	0.3
VHF telemetry system		0	0.5	0.5
Total		4.5	95.0	5.0

*Includes power converter losses

The battery-supplemented solar cell supply is being used on many other ERS missions and is nearly identical to that for four other ERS's that have been built. Power for all satellite electrical functions is taken directly from the battery or the solar panels. Charge regulating circuitry is used to control the rate of charge without damage to the battery.

The electrical power system includes the solar cell array; a 10 cell, 12-volt, 72 amp-hour NiCd battery; a charge control circuit; two power converters; and a 9-volt regulator. The system features separate power supplies for each of the satellite functions. A separate converter supplies +12, -12, and +5 volts for the stable oscillator and BINOR generator. The VHF telemetering system operates from a 9.0 volt precision ($\pm$ 0.1 percent) regulator with the exception of the VHF transmitter, which operates from the unregulated solar array bus. The VHF command and control system operates from the unregulated solar array bus for maximum reliability.

Sufficiently high solar panel and battery bus voltages have been selected to maximize the efficiencies of voltage level conversion and regulation, while still maintaining practical considerations of reliability and mechanical mounting constraints. A computer program is used to determine and profile temperature for each solar panel throughout a simulated orbit. Another computer program is used to optimize the ratio of the number of solar cells per string to the number of battery cells for the given temperature profile.

The normal operating mode is such that about half of the average available current from the solar arrays is required to operate the continuous duty equipment. The remainder of the current is available to charge the battery. After each payload exercise, about six more orbits are required to return the battery to its fully charged state. However, the battery capacity is such that the payload can be exercised as many as three consecutive orbits. This must then be followed by approximately 15 orbits, or 24 hours at maximum charging rate before the battery is again fully charged. Under anticipated operational conditions, the electrical power system design provides an ample margin for the most severe payload exercises.

The solar array output can be described by the characteristic solar cell I-V curve shown in Figure 20. The curve shifts along the current scale according to the projected area of the array normal to the sun. Other variations are caused by temperature and radiation damage. The input impedance of the complete electrical system can be described by a load line function which would appear as a discontinuous line on a current-voltage plot. The load line function changes with the operating mode of the system and with the state of charge of the battery. The important point is that the operating voltage of the solar array bus is always at the intercept of the load line function and the characteristic curve of the solar array. The 10-cell battery represents a very low input impedance to the solar array device with a bias of from 14 to 14.5 volts while charging, and from 11 to 14 volts while discharging. Thus, the impedance lines shown on Figure 20 are nearly vertical. Of the total current available from the solar array, about 240-300 milliamps is required to operate the standby equipment. The residual current,

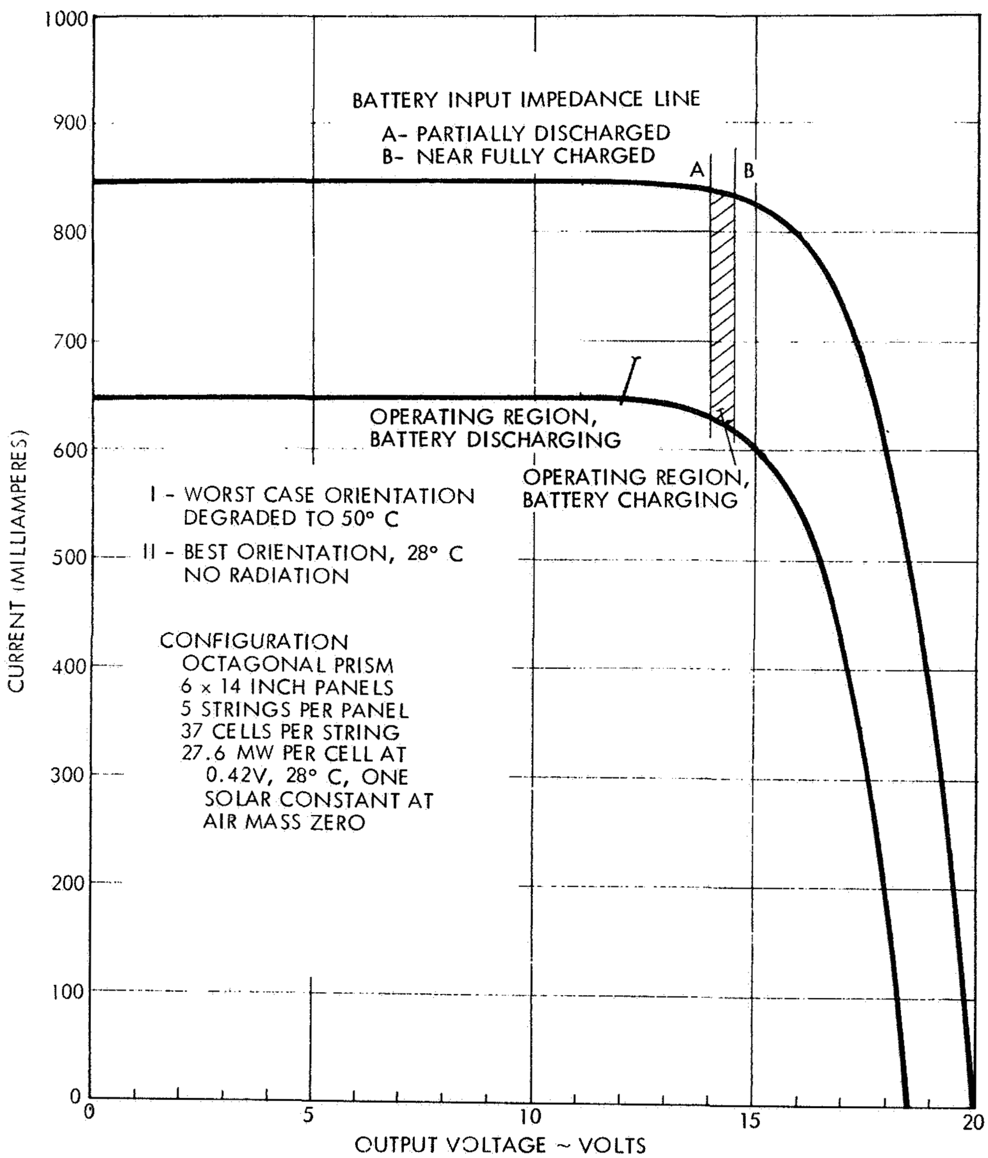


Figure 20. Solar Cell I-V Curve

from 350 to 550 milliamperes, is available to charge the battery. If the battery is fully charged, then the residual current is shunted by the charge control circuit and the excess power dissipated as heat. When the payload is exercised, the current demand is about 8.0 amperes at nominally 11.5 volts. If the satellite is not eclipsed, about 0.75 amperes is supplied by the solar array and the remainder is drawn from the battery.

As the battery charges, the cell voltage begins to exceed 1.4 volts per cell and the current to the battery is reduced by shunting the residual current to ground. At 1.43 volts per cell, the charging current is further reduced, until after 1.45 volts per cell only the current necessary to maintain the batteries in a full state of charge is supplied. This operational mode provides a high duty cycle with high reliability.

The other important function of the charge system, the low voltage control, also utilizes the discharge characteristics of the nickel-cadmium cell to perform its function. When discharging, the cell voltage starts at about 1.4 volts and reduces to approximately 1.25 volts when one-quarter of its capacity is exhausted. It remains close to this potential until nearly discharged. When 90 percent of its capacity is gone, the voltage starts to drop rapidly. A voltage of 1.1 volts is used to indicate this point. When the low voltage control senses a battery voltage of 1.1 volts per cell, it generates an internal off command. The satellite then reverts to the normal standby (minimum power) mode and begins the charge sequence.

On each face of the satellite is a panel of 185 solar cells consisting of 5 strings of 37 cells each. Each cell is 1 x 2 cm, 10 ohm-cm, n-on-p, end contact, with a conversion efficiency of 11 percent. In addition, the up-facing end of the prism (boom side) will accommodate an array of cells equivalent to 2 side panels, or 270 cells. At an operating voltage of 0.38 volt per cell, the power output of the total array ranges from 9.1 watts (end view) to 11.8 watts (side view) with the nominal average of 11.0 watts, or a current of 785 milliamperes. The design of the power system allows the operating point of the arrays to go as low as 0.33 volt per cell before the regulated lines are affected.

The usable power has been conservatively reduced to 9.0 watts to account for the specific mission use. Reduction is caused by losses in the cover glass (5%), isolation diodes (2%), losses in the battery charger gate transistor (4%), practical limitations of operating at the solar panel maximum power point (10%), and higher than ambient solar panel temperature due to nonrotation for a nonspinning satellite (6%). Six-mil thick cover glass will be used to prevent damage by radiation. Individual cells are soldered to a printed circuit on a glass epoxy substrate bonded to an aluminum plate. Similar (triangular) panels have been tested for thermal cycling, vibration, and shock far in excess of any flight requirements. The covers are bonded to each solar cell individually.

A battery of ten hermetically-sealed rectangular nickel-cadmium cells rated at 6 ampere-hours has been selected. The nickel-cadmium cell has certain advantages over other types. It has the highest energy storage capacity per pound of any rechargeable type and has a considerably longer life in terms of number of charge-discharge cycles than any other. Hermetically-sealed nickel-cadmium cells are available in either cylindrical or rectangular shapes. For this mission the rectangular variety (Gulton No. V0-6HS-4) was selected for convenience in packaging. Battery cells from this series have been orbited on previous ERS satellites.

Battery charge control circuitry (Figure 21) will perform the following functions:

a) Provide maximum rate of battery charge for maximum overall duty cycle of the satellite system

b) Prevent battery overcharge

c) Prevent excessive battery discharge

Two basic circuits are used. The charge current control circuitry provides a maximum of charge current from the solar panels to the battery except when the battery approaches a condition of full charge. At this time charge current is gradually reduced until only a trickle charge level is supplied which is sufficient to overcome internal battery losses. The low voltage control circuitry prevents excessive discharge of the battery by removing power loads from the battery upon sensing a low voltage condition.

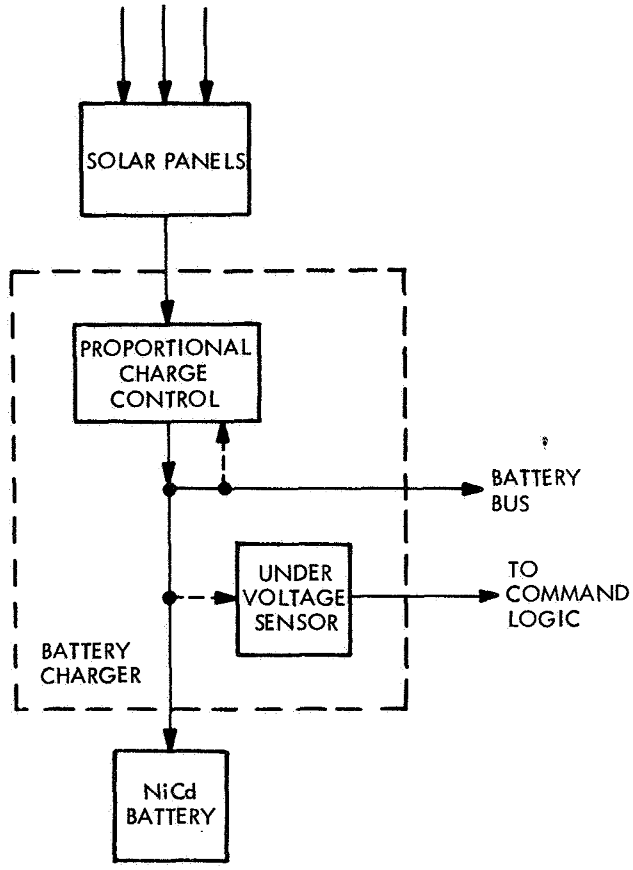


Figure 21. Battery Charge Control Block Diagram

Two separate DC converters are necessary. A 28-volt static inverter is required to operate the L-band transmitter and MISTRAM transponder for a 10-minute period each test exercise. This converter must supply nominally 3. 0 amperes for the duration of the test, which, in turn, requires 7 to 8 amperes from the secondary power source. Since only $\pm$ 10 percent regulation is required, output/input voltage ratio os 2. 25 will fulfill these requirements. Since tight regulation is not required, a high conversion efficiency can be achieved, and 90 percent efficiency is anticipated for this unit.

A second power conditioning unit is necessary to supply voltages for the stable oscillator and BINOR generator, both of which operate continuously. A separate supply is preferred for these devices to increase the overall reliability of the payload, to avoid the problem of switching transients from the 28-volt supply, and to achieve the higher efficiency necessary because of continuous operation. This power conditioning unit must supply $\pm$ 12 volts regulated to 2 percent for the stable oscillator and +5 volts regulated to 10 percent for the BINOR generator. Since these devices must operate continuously, the efficiency of conversion is particularly important to the power budget. In addition, the continuous operation means that the input voltage range must be greater than for the 28-volt converter, namely from 11 to 15 volts. However, the tighter regulation requirement ($\pm$ 2 percent on the 12-volt lines) limits the efficiency that can be achieved to nominally 80 percent. This results in a continuous raw power drain of 4.4 watts at 14 volts nominal. These components will be carefully reviewed to achieve greater power utilization efficiency.

The VHF telemetry system requires precision regulated 9.0 volts for the commutator and SCO while the transmitter uses unregulated power from the battery bus. The 9-volt regulator (Figure 22) will be a conventional series type that has been used on every ERS mission to date. It has a regulation of $\pm$ 0.1 percent over the expected range of temperatures and loads. This regulator is assembled as a cordwood module 1 x 1 x 1.5 inches.

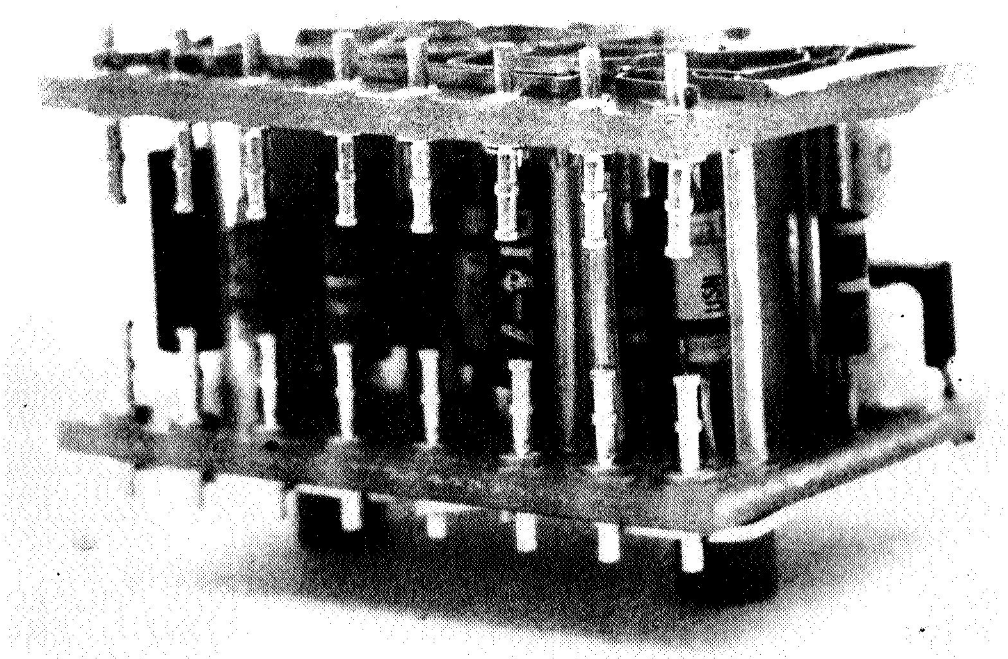


Figure 22. Nine-Volt Regulator

3. SATELLITE DESIGN WITHOUT MISTRAM

A relatively simple satellite can be used if a high accuracy tracking system, such as MISTRAM, is not included. A 12-inch octahedron (ORS-III) of the Environmental Research Satellite family would be selected for this configuration. This TRW satellite model has been piggyback launched nine times to date; three more are currently in preparation. Use of this model satellite would result in lower costs and greater selection of piggyback rides than for the larger prism satellite required for the MISTRAM configuration.

Figure 23 shows the octahedral satellite in orbit with its broad-beam directional L-band antenna pointed at the earth by a single boom, gravity gradient orientation system. This orientation also causes the VHF command and telemetry dipole antennas to maintain a favroable r-f linkage

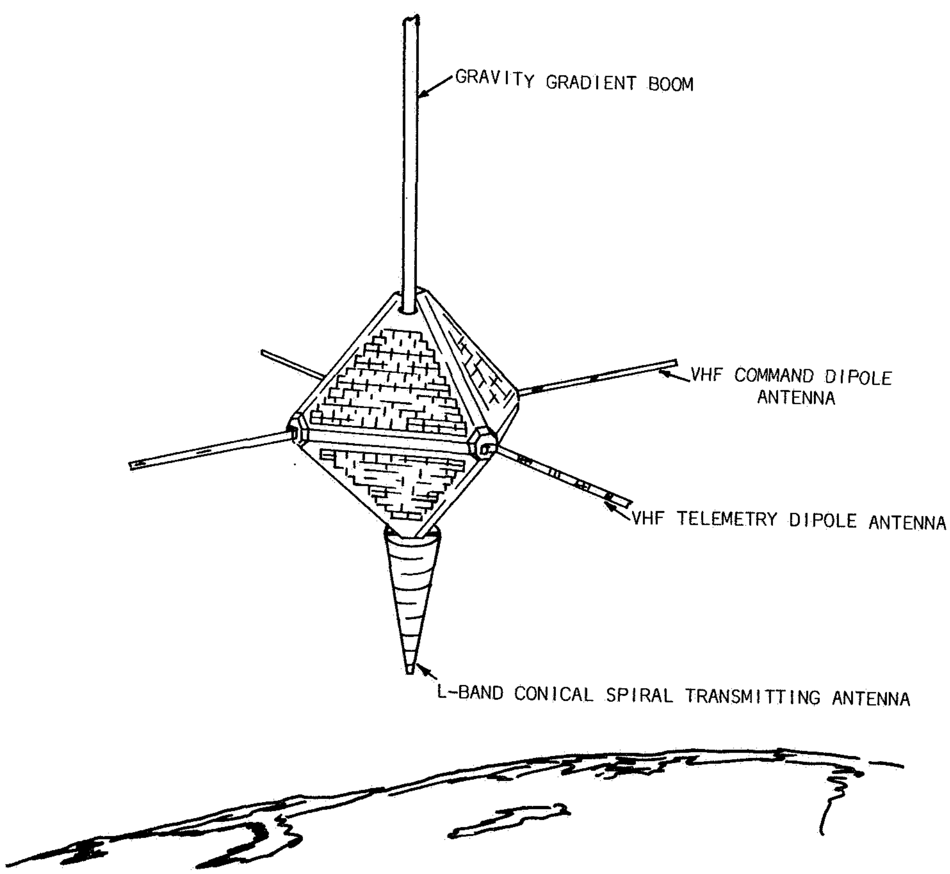


Figure 23. System Concept, Navstar Test Satellite, Octahedral Configuration

with ground stations. The gravity gradient system is basically the same as that used in the MISTRAM design (Paragraph 2. 3. 1); in summary, it employs a passive, magnetically lossy permeable rod motion damper. The gravity gradient boom is initially extended via command. The boom is retractable in case polarity reversal is required. This type of gravity gradient system can be used for low orbits and elliptic orbits up to approximately 1000 nautical miles.

Figure 24 illustrates a radiation pattern of the broad-beam, L-band antenna. It is a conical log spiral antenna and was developed at TRW for an earlier mission. Its gain and bandwidth characteristics are similar to those of the turnstile antenna described in Paragraph 2. 2. 5.

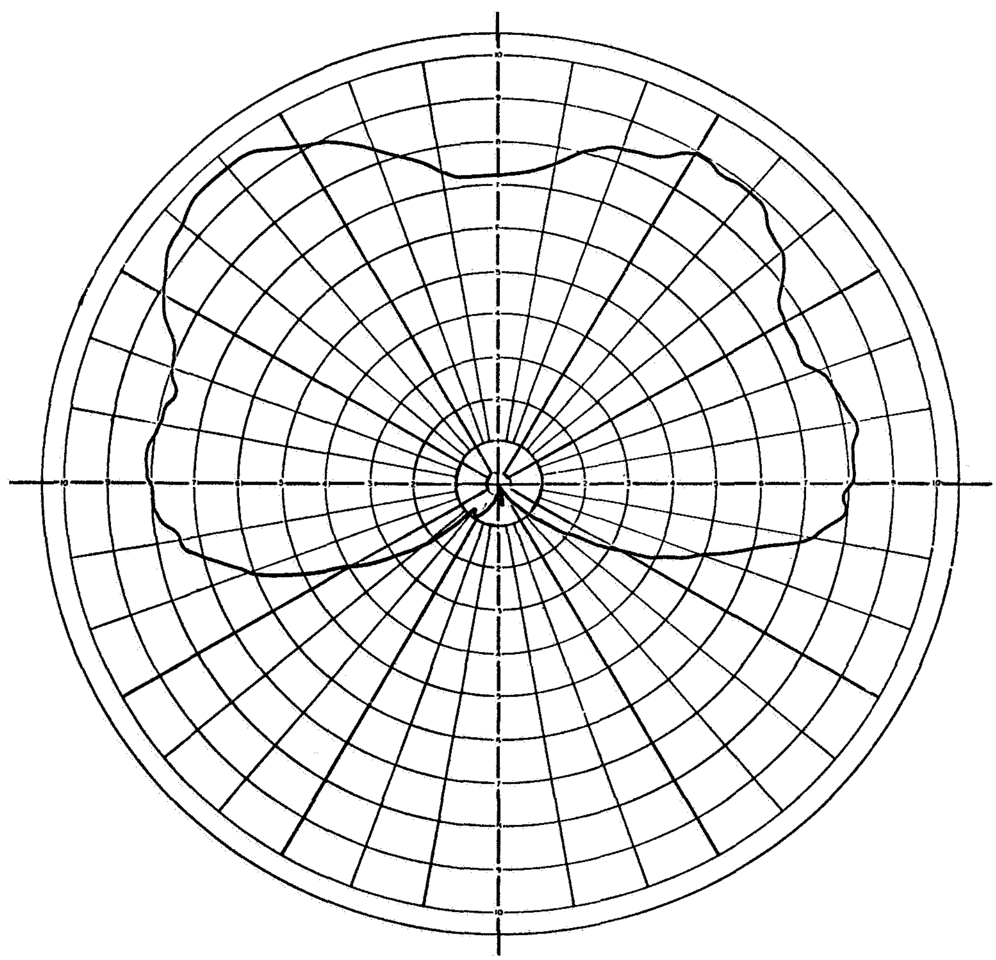


Figure 23. Conical Log Spiral Antenna Pattern

Figure 25 is a functional block diagram of the system. The VHF beacon and telemetry, VHF command and control, stable oscillator*, BINOR code generator, and L-band transmitter are identical to those described in Section 2 for the MISTRAM test satellite design. Using the VHF beacon and tracking by the STADAN stations, orbit prediction of between 50 to 100 meters accuracy can be provided. The VHF subsystem and most of the electrical power subsystem are nearly identical to those of earlier, successful ERS satellite missions.

Table 9 presents electrical power considerations, including power requirements and particularly important performance capabilities. Based on nine minutes operation of the high power L-band transmitter per pass, the battery supplemented power system will permit full mission

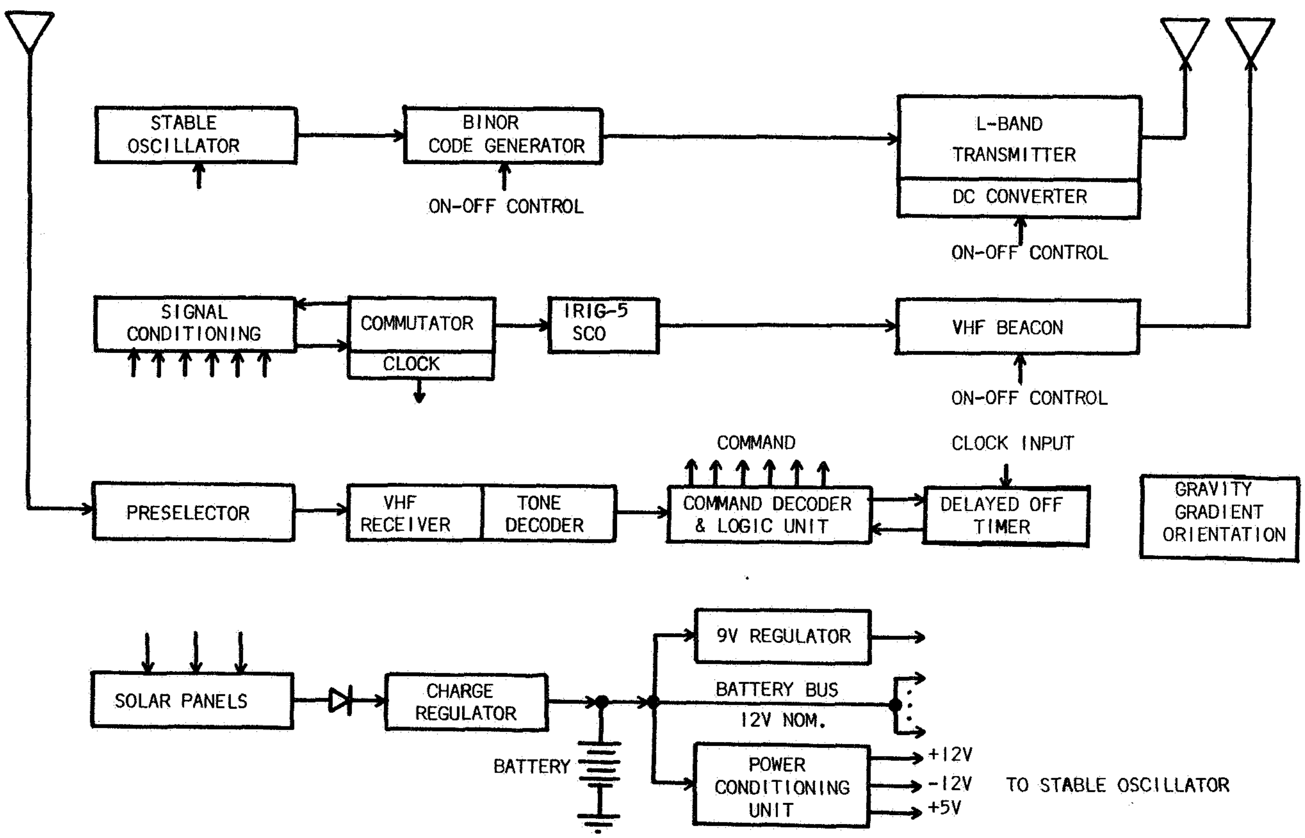


Figure 25. Test Satellite Functional Block Diagram

*Except that no remote frequency control is provided due to power limitations in the standby mode of satellite operation.

Table 9. Electrical Power Requirements and Power System Performance Characteristics

ELECTRICAL POWER REQUIREMENTS			
	Power (watts)		
	Standby or Minimum Power Mode	VHF Beacon On	*L-Band Transmitting
L-Band Transmitter	0	0	47
**Stable Oscillator	1.4	1.4	1.4
BINOR Generator	0	0	1.0
VHF Beacon	0	0	0.25
VHF Receiver	0.15	0.15	0.15
Other Electronics	0.10	0.10	0.60
Total	1.65	1.90	50.4

*Normal 9 minutes duration per exercise.
**Power level after initial warmup.

POWER SYSTEM PERFORMANCE CHARACTERISTICS (Orbit Sun Time: 65 Minutes; Orbit Eclipse Time: 31 Minutes)	
Current required at 12.5 volts during test exercise	4000 ma
Battery drain during 9 minute exercise (pass duration)	0.60 ah
Current available from sunlit solar panels	400 ma
Current for standby operations	207 ma
Current available for charging	193 ma
Usable charging current (Eff = 0.85)	164 ma
Recharge per orbit	0.18 ah
Orbits to recharge per exercise	3
Maximum long-term operation	5 per day
Maximum consecutive operating orbits (for 70 percent discharge)	10 operations
Charge time after 70 percent discharge	36 hours

tests for one pass per orbit for ten successive orbits before the battery is discharged to 70 percent capacity. Recharge time after a one-pass operation is approximately three orbits, or less than five hours. Recharge time after a ten successive pass operation is 36 hours. On a long termbasis, approximately 4 to 5 tests can be performed each day.

Table 10 presents packaging and weight parameters. Approximately 50 percent is a reasonable packaging density for this class of satellite and, for example, is lower than that of two of the most recently launched octahedral satellites. Satellite weight is approximately 29 pounds. Total piggyback weight on the launch vehicle is 32 pounds and includes the three-pound launch canister.

Table 10. Packaging and Weight Parameters

	Size (in.)	Volume (cu in.)	Weight (lbs)
Payload			
L-Band Transmitter	3-3/4 x 3-3/4 x 2-1/2	25.0	2.0
BINOR Code Generator	9 x 4-1/2 x 1	40.5	2.0
Precision Oscillator	9 x 4-1/2 x 3-1/2	141.5	4.0
VHF Telemetry			
VHF Transmitter	4-3/4 x 2 x 1-1/8	10.7	0.3
Subcarrier Oscillator	3-1/8 x 2 x 1-1/8	7.0	0.2
Commutator	3 x 1 x 1	3.0	0.2
Clock	1-1/2 x 2 x 1	1.5	0.1
Signal Conditioning Unit	3 x 2 x 1	6.0	0.2
Command			
Preselector	3 x 2 x 1-1/8	7.0	0.2
Receiver/Decoder	4-1/2 x 3-1/2 x 2-1/4	35.0	1.7
Command Logic	4 x 3 x 1	12.0	0.3
Timer	3 x 1 x 1	3.0	0.2
Power			
Battery (10-cell)	4-1/4 x 4-1/8 x 3-3/4	66.0	6.4
Power Conditioning Unit	4 x 3 x 2-1/2	28.0	0.8
Battery Charge Control	1 x 1 x 1.5	1.5	0.1
Low Voltage Sensor	1 x 1 x 1.5	1.5	0.1
Solar Panels (External)	--	--	2.5
Stabilization			
Retractable Boom	2 x 2 x 4-1/2	18.0	0.4
Damper Plate and Heat Sink	8 x 8 x 5/8	40.0	2.0
Structure, Etc.			
Satellite Shell (without panels)	--	--	3.0
Canister	--	--	3.0
Antennas	--	--	1.0
Wire and Cable	--	--	1.5
Total		457.0	32.2
Available Volume		926.0	
Packaging Density		49.4%	

Table 11 summarizes the most important features of the satellite by subsystem. The four-watt L-band radiated power level was selected because this unit was an available off-the-shelf vendor item. A higher power level can be provided. If this is accomplished with solid state devices, there would be a corresponding reduction in the maximum transmitter duty cycle and an increase in cost. If it is accomplished with a traveling wave tube (TWT), packaging and satellite size must be reassessed; again there would be some cost increase.

Table 11. Satellite Parameter Summary

General	
Spacecraft model	12-inch octrahedron
Spacecraft weight	29.2 pounds
Total weight on launch vehicle (including 3-pound launch canister)	32.2 pounds
Payload	
Basic payload	L-band transmitting system (1600 MHz)
Radiated power	4 watts
Transmitted data	BINOR code
Data format	PCM/PSK
"ON" time per exercise	9 minutes
Maximum operating time (at 70% battery depletion)	9 minutes per orbit for 10 orbits
Long term duty cycle:	
Full sun	8 exercises per day
Maximum eclipse	5 exercises per day
Electrical Power	
Solar panel (n or p) initial raw power	6 watts
Battery (NiCd)	75 watt hours
Beacon Telemetry	
Frequency (STADAN net)	136.8 MHz
Radiated power	0.1 watt
Transmitted data	PAM housekeeping data
Data format	PAM/FM/FM
Subcarrier channel	IRIG-5
Number of telemetry parameters	24
Command	
Frequency	150 MHz
Commands	8
Off timer	Delayed L-band off (9 minutes)
Special Features	
Satellite operation technique	Gravity gradient
Orientation system motion damper	Passive permeable rod matrix