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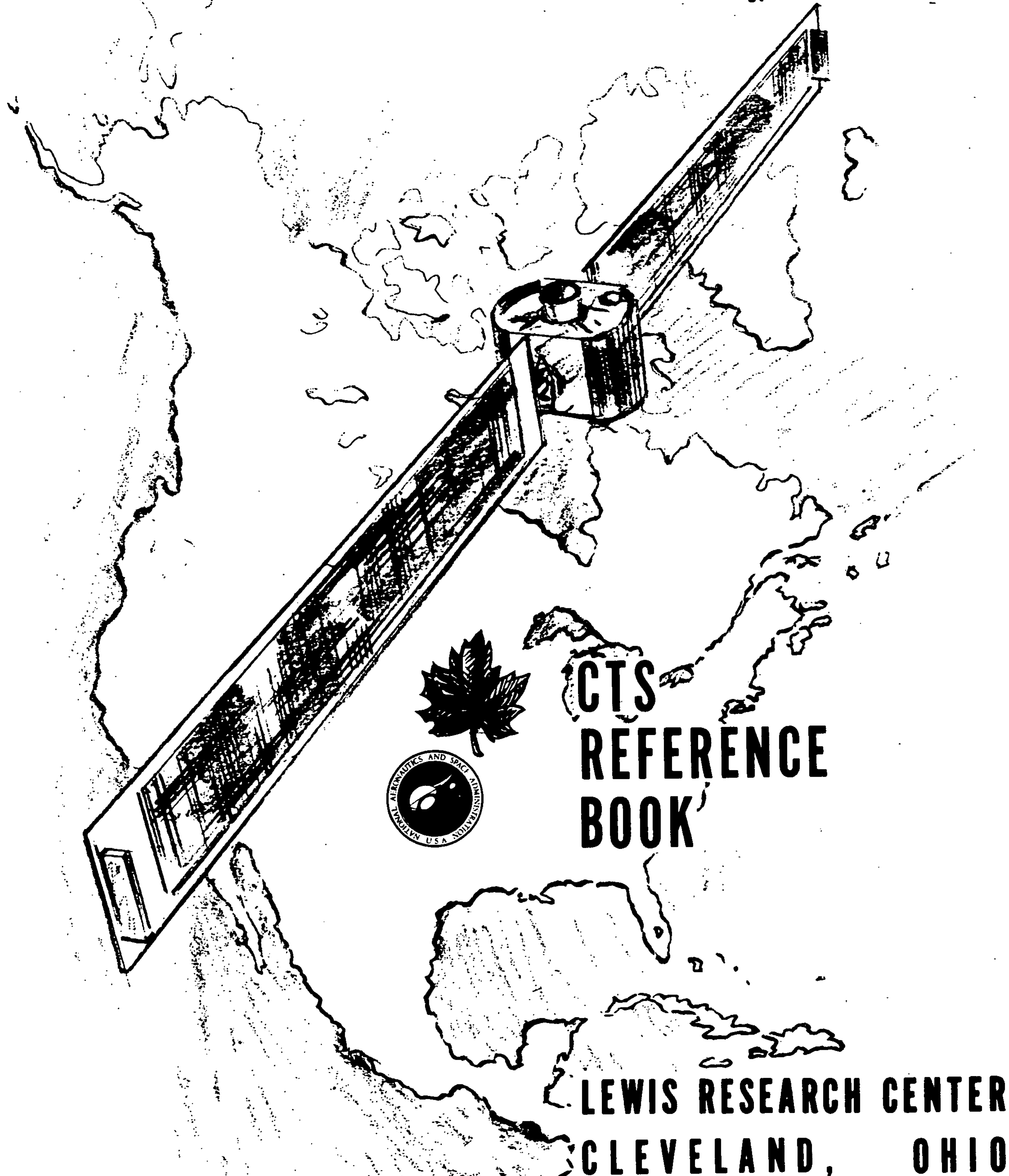

CTS REFERENCE BOOK

LEWIS RESEARCH CENTER
CLEVELAND, OHIO

OCTOBER 15, 1975

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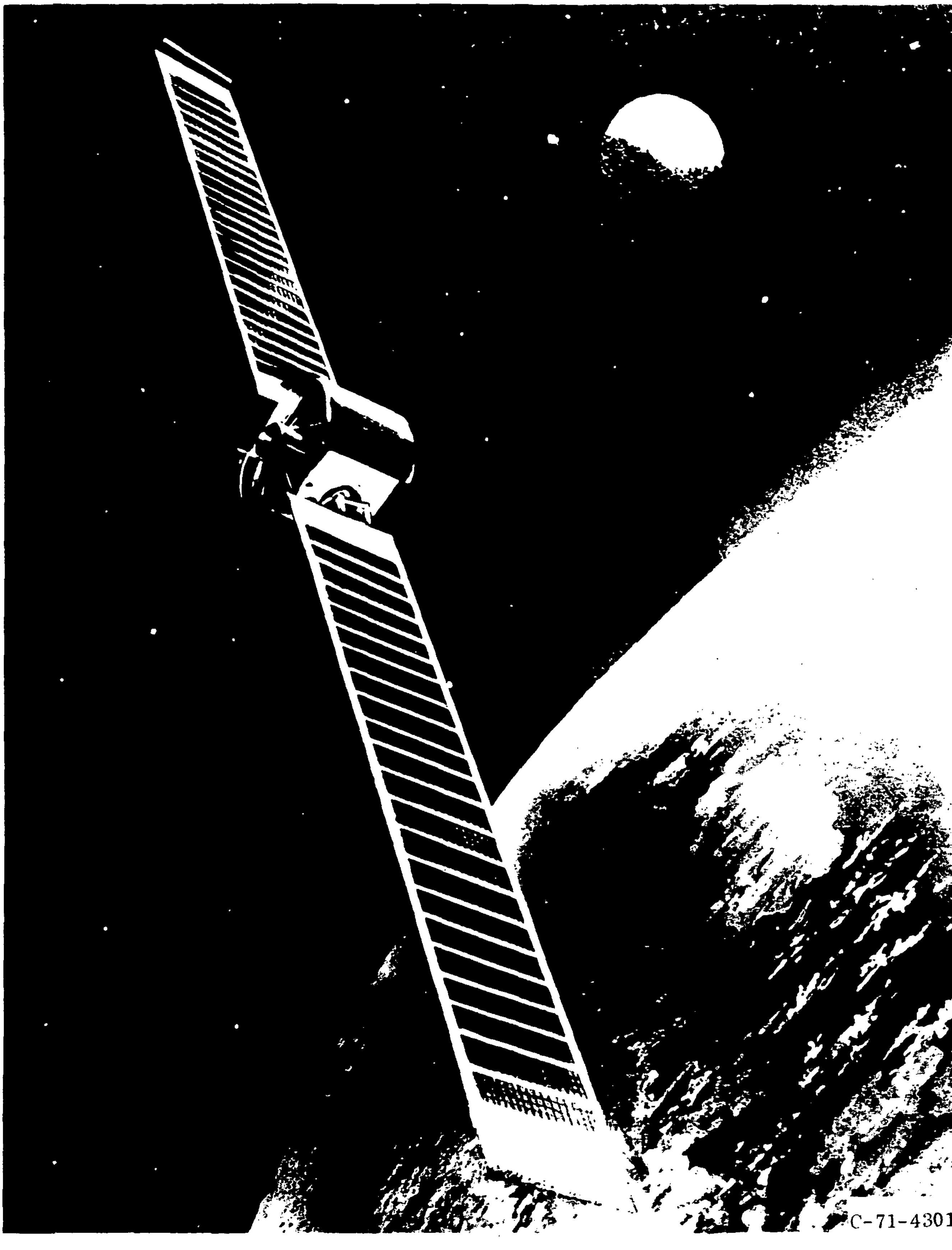

Artist's concept of CTS on station

NASA TM X-71824

CTS REFERENCE BOOK

Lewis Research Center
Cleveland, Ohio

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

FOREWORD

The Communications Technology Satellite (CTS) Reference Book has been generated as a reference document, primarily for those persons associated with the CTS program. As such, certain knowledge has been assumed on the part of those who will use it. Some of the technical data presented will change as the program reaches fruition, and errors have undoubtedly crept into the text. For these reasons, this document should not be used as the basis for design, and the correctness of data contained herein must be verified before their inclusion in other documents. A word of appreciation is due the main contributors, both Canadian and U.S., who interrupted busy schedules to supply the information provided herein.

Editors:

J. G. Kennard

A. W. Nice

CTS Project Office
NASA Lewis Research Center

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I. INTRODUCTION

The Communications Technology Satellite (CTS) is a cooperative program of the United States and Canadian governments. The program is in consonance with the National Aeronautics and Space Administration's (NASA) designated authority for international cooperation with other nations in the peaceful application and advancement of space and aeronautics technology. This program is the latest in the series of cooperative U. S. and Canadian space experiments that started in September 1962 when NASA launched the first Canadian-built satellite in the Alouette program. The Alouette satellites were used to conduct scientific observations of the ionosphere.

In January 1969, ISIS I was launched to study the upper atmosphere and to obtain a better understanding of the aurora borealis. The launch of ISIS II followed in March 1971.

On April 20, 1971, the governments of the United States and Canada signed a Memorandum of Understanding. They agreed to undertake, on a joint basis, the development and launching of an experimental satellite, designated the Communications Technology Satellite, to extend communications technology to much higher power levels of transmission than had been previously used. This would permit the use of small, low-cost, ground terminals that would make communications services practical in areas not now served. Under this agreement, Canada is designing and building the spacecraft at their Communications Research Centre (CRC). NASA is providing spacecraft test facilities at the Lewis Research Center and the Goddard Space Flight Center (GSFC) and a high-efficiency 200-watt traveling-wave-tube amplifier and power supply that will operate in the 12- to 14-GHz band. A NASA Thor-Delta model 2914 launch vehicle will place the satellite in geostationary orbit. A consortium of European nations, through the European Space Research Organization (ESRO), is also participating in association with the Canadian Government. U. S. and Canadian experimenters will share equally in the time allocation during the satellite's expected 2-year life.

PROGRAM OBJECTIVES

The objective of the CTS program is to advance the technology of both spacecraft-mounted and related ground-based components and systems applicable to high-radiated-RF-power satellites. In order to achieve this objective, the spacecraft will demonstrate new technology applications and conduct experiments on components and systems that will be applicable to future commercial communications satellites. The program will also include communications experiments with user agencies, universities, and industrial groups in the United States and Canada.

TECHNOLOGY OBJECTIVES

Specific objectives include demonstrations of

- A 12-GHz traveling-wave tube (TWT) with about 50 percent efficiency and with a nominal RF output power of 200 watts and the associated power processor required to convert the solar array power into an acceptable form to operate the TWT
- The operation of an unfurlable solar-cell array delivering over 1 kW of useful power to the spacecraft
- A three-axis stabilization system to maintain antenna boresight pointing accuracy to $\pm 0.1^{\circ}$ in pitch and roll and $\pm 1^{\circ}$ in yaw on a spacecraft with large flexible appendages
- Color television transmission at 12 GHz from a satellite to small, low-cost ground terminals
- Uplink television transmission at 14 GHz from small terminals
- Audio broadcast to very small ground terminals
- Two-way voice communication, wideband data transmission, and data relay

II. MISSION PROFILE

MISSION OVERVIEW

In order to meet the communications objectives of the CTS program, the spacecraft will be placed in a synchronous, equatorial orbit, at 116° west longitude and will be maintained within a $\pm 0.2^{\circ}$ east-west deadband about the nominal position for a 2-year period. The following distinct phases are identified:

- Launch
- Transfer orbit
- Apogee injection (motor firing)
- Spacecraft drift to station
- Three-axis attitude acquisition
- In-orbit operation

The first four phases are the responsibility of NASA, while CRC is responsible for the three-axis attitude acquisition and in-orbit operation.

The CTS will be launched by a Thor-Delta model 2914 launch vehicle. The launch phase terminates with injection of the spacecraft, spinning at 60 rpm, into an elliptical inclined transfer orbit - the apogee of which is biased above the synchronous altitude. After injection, the spacecraft Tracking, Telemetry, and Command System is employed. Ground tracking for orbital determination is initiated as soon as ground-station visibility permits. As the spacecraft approaches apogee, the attitude determination process commences. Injection into synchronous orbit could occur as early as the second apogee and as late as the tenth. Following determination of the perigee injection attitude, the spacecraft is precessed into the apogee-motor firing attitude (AMFA). The execution of the precession maneuver must comply with the constraints of spin-axis - sun-line angle, antenna patterns, and attitude determination. Following apogee-motor firing the spacecraft attitude will be maintained in an orientation which satisfies the thermal and power design limits. At the time of station acquisition at 116° west longitude, the spin axis will be normal to the orbital plane.

During the first four phases of the mission, a central Spacecraft Control Center at GSFC will be staffed by CRC and NASA personnel. They will continuously monitor the spacecraft status and provide orbital and attitude determination results. With the spacecraft at the target longitude and with the spin axis normal to the orbital plane, mission control will be transferred to the CRC ground station at Ottawa, where despin and attitude acquisition maneuvers will be accomplished by CRC. GSFC, at this point, will continue to act as ground-station backup through three-axis stabilization.

For the remainder of the mission the spacecraft is controlled in a three-axis mode with the + yaw axis pointing toward the earth, the + pitch axis normal to the orbital plane (i.e., south), and the + roll axis along the orbital velocity vector. East-west station keeping is performed to maintain the spacecraft within $\pm 0.2^{\circ}$ of the desired longitude. There is no active north-south stationkeeping control. Rather, the inclination is controlled passively by biasing the initial orbital inclination and selecting an initial ascending node close to 270° such that communications experimenters' requirements can be met throughout the 2-year lifetime.

MISSION DESCRIPTION

Launch Phase

The Thor-Delta model 2914 vehicle (with TE 364-4 third stage) will be launched from the Kennedy Space Center (KSC) Complex 17 at an azimuth of 90° down the Eastern Test Range. Major spacecraft events are shown in table II-1.

TABLE II-1. - MAJOR SPACECRAFT EVENTS

Event	Time from lift-off, sec	Altitude	
		km	n mi
1 - Lift-off	0	------	------
2 - Solid-motor burnout (6)	38.6	------	------
3 - Solid-motor burnout (3)	77.8	------	------
4 - Solid-motor jettison	85.0	------	------
5 - Main-engine cutoff	226	90	49
6 - First- and second-stage separation	234	------	------
7 - Second-stage ignition	238	100	54
8 - Fairing jettison	276	126	68
9 - Second-stage cutoff (first)	556	185	100
10 - Coast	----------	↓	↓
11 - Second-stage restart	1486	↓	↓
12 - Second-stage cutoff (second)	1494	↓	↓
13 - Third-stage spinup	1523	↓	↓
14 - Second-stage jettison	1525	↓	↓
15 - Third-stage ignition	1538	↓	↓
16 - Third-stage burnout	1582	↓	↓
17 - Spacecraft separation	1682	↓	↓
18 - Apogee firing; nominal fifth apogee crossing	17×10^{4}	36 470	19 692
19 - Station acquisition	1.7×10^{6}	36 342	19 623

The time of launch is constrained by spacecraft thermal and power considerations to two periods per day, each having a minimum duration of 1.5 hours. During the transfer orbit, a $90^\circ \pm 25^\circ$ angle between the sun line and the spin axis will be maintained, and the eclipse duration will be limited to a maximum of 30 minutes. Following lift-off, the first-stage burn and a partial burn (300 sec) of the second stage will place the spacecraft into a 185-km (100-n mi) circular parking orbit with an inclination of 28.3°. After a 15.5-minute coast in the parking orbit, upon the first southerly equator crossing (descending node), the second stage will be reignited. This second-stage burn (43 sec) will be followed by third-stage spinup, stage separation, and third-stage burn to provide an orbital inclination change of 1.5°. This will inject the CTS payload into a transfer orbit with the apogee at synchronous altitude. Transfer-orbit perigee and inclination are 185 km (100 n mi) and 26.8°, respectively. The 1.5° plane change maximizes the spacecraft weight into synchronous orbit. The nominal transfer-orbit parameters and their 99 percent dispersions are given in table II-2.

TABLE II-2. - NOMINAL TRANSFER-ORBIT PARAMETERS

Initial perigee latitude, deg	0±0.26
Initial perigee longitude, deg	4±0.48 (east)
Perigee altitude, km (n mi)	185±16.7 (100±9)
Apogee altitude,[a] km (n mi)	36 342±1038 (19 623±560)
Orbital period, hr	10.52±0.35
Orbital inclination, deg	26.8±0.55

[a]CTS is biased by +555 km (300 n mi).

Transfer-Orbit Phase

During the transfer-orbit phase the NASA Satellite Tracking and Data Network (STDN) supports spacecraft telemetry and command requirements, provides orbital predictions, and determines spacecraft attitude. After injection into the transfer orbit, the spacecraft telemetry and command functions and orbital determination are initiated as soon as ground-station visibility permits. A coherent S-band transponder is employed for ranging and range rate and is compatible with pseudorandom noise (PRN) or tone ranging. This S-band transponder, in conjunction with an encoder and decoder, serves as the command and telemetry system and is compatible with the GSFC STDN S-band system. A sufficient number of ground stations and an adequate tracking schedule are provided to determine precise transfer-orbit parameters for spin-axis attitude determination and apogee-motor aiming. Figure II-1 shows the ground-station view periods through the tenth apogee for some of the candidate ground stations. A station mask with a 10° minimum elevation is used.

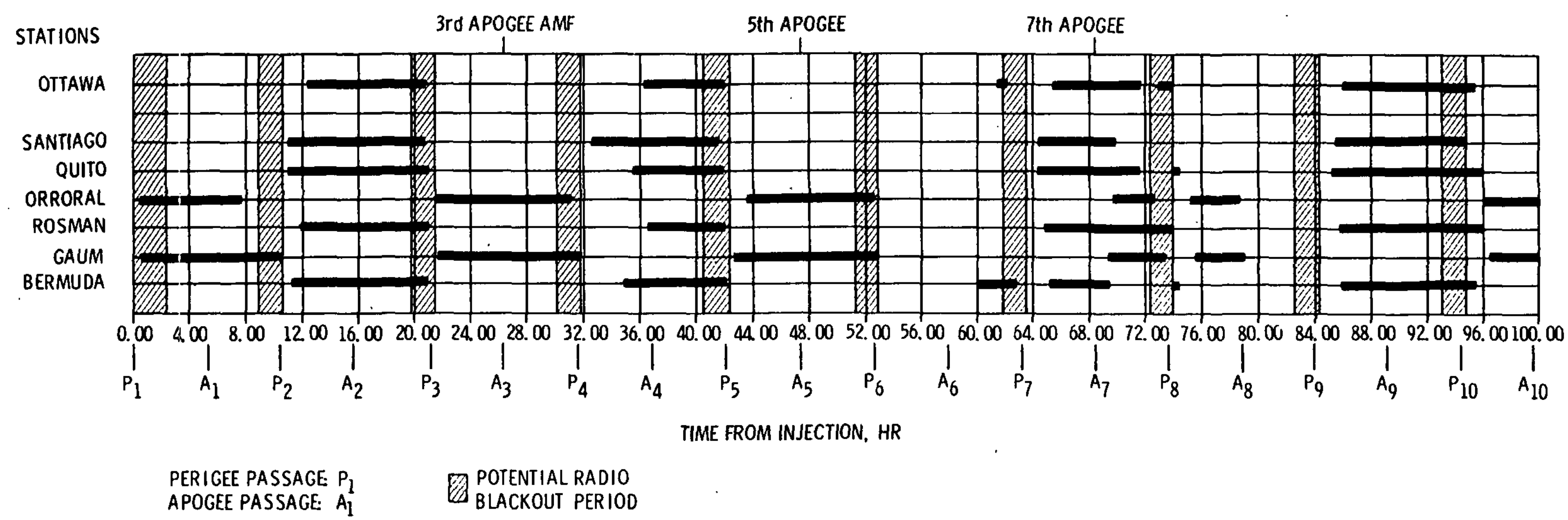


Figure II-1. - Station view periods from transfer orbit injection through tenth apogee.

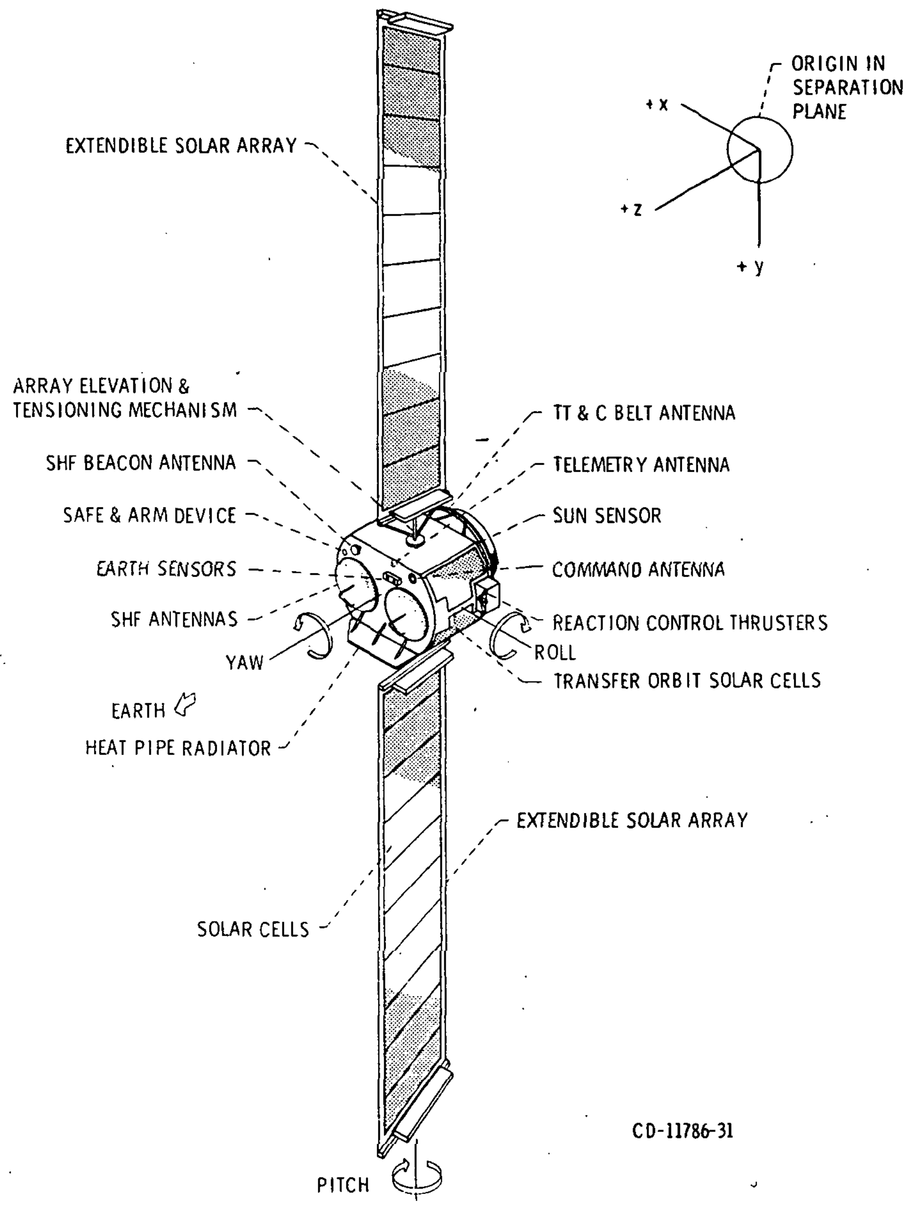


Figure II-2. - CTS configuration on station in synchronous orbit.

The spacecraft is oriented into an approximate apogee-motor firing attitude 7 hours after injection. Subsequent attitude corrections (spin-axis precessions) are performed as more precise orbital and spacecraft attitude determinations are obtained. The final apogee-motor firing attitude is selected to correct for launch-vehicle-induced orbital dispersions and to circularize and provide orbital plane change in order to minimize the final orbital inclination error. Under nominal conditions, the final attitude maneuver is completed at least 3.5 hours prior to the third apogee.

The spacecraft remains in the perigee injection attitude until sufficient attitude information is available for the spin-axis precession maneuver. Calibration of the high-thrust engine of the Reaction Control System (RCS) or execution of the precession maneuver for the apogee-motor aiming does not occur until attitude information is available.

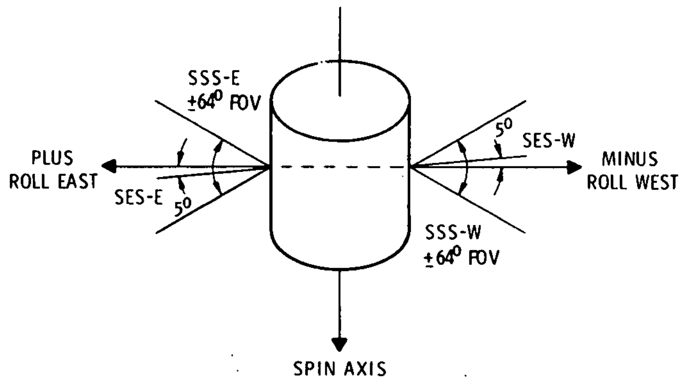
Data from two spinning digital sun sensors (SSS) and two spinning earth sensors (SES) provide attitude determination. The spinning sun sensor assemblies are located on the east and west faces of the spacecraft, referenced to the in-orbit geometry (fig. II-2). Each sun sensor has a field of view (FOV) of $\pm 64^{\circ}$, referenced to the normal to the spin axis, as shown in figure II-3(a). One sensor is considered the primary sun sensor and the other a redundant sun sensor. The selection is made by a command to the Attitude Control Electronics Assembly (ACEA). The sun sensor provides a pulse for time indexing with the earth sensor data and for executing the precession maneuver. Additionally, the sun sensors provide the sun azimuth, as shown in figure II-3(b). Pulse information is provided in every fifth telemetry main frame; azimuth information is provided in each main frame. Perigee Injection Attitude (PIA) sun angle - spin axis normal constraint of $\pm 25^{\circ}$ falls within the $\pm 64^{\circ}$ field of view of the sun sensor. Therefore, continuous data are available except during transfer-orbit shadow periods.

The Spinning Earth Sensors are located on the east and west faces of the spacecraft with a cant angle of 5° to the spin-axis normal, as shown in figure II-3(a). Each sensor provides a space-to-earth pulse and an earth-to-space pulse (referenced to the sun pulse) such that the attitude can be determined from the width of the earth crossings.

An example of earth sensor geometry is shown in figure II-3(c) for the case where the spin axis is normal to the equatorial plane. The attitude information telemetered to the ground is

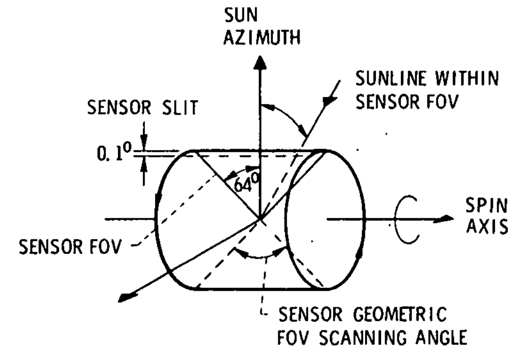
- Sun pulse
- West-earth-sensor, space-to-earth pulse
- West-earth-sensor, earth-to-space pulse
- East-earth-sensor, space-to-earth pulse
- East-earth-sensor, earth-to-space pulse
- Sun azimuth

The sun azimuth is read out in each main frame. The first five pieces of information

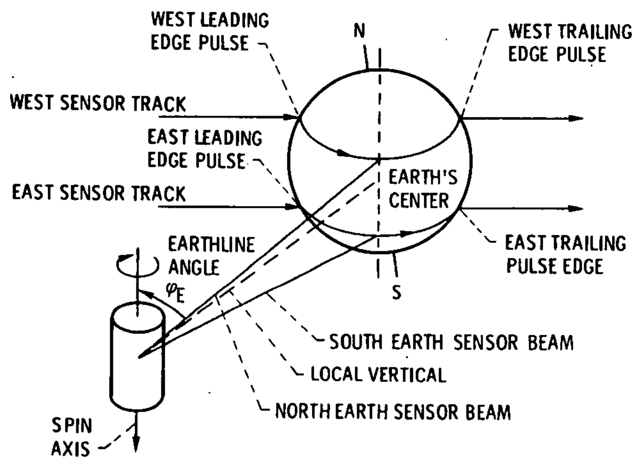


SES-E (SPINNING EARTH SENSOR - EAST FACE)
SES-W (SPINNING EARTH SENSOR - WEST FACE)
SSS-E (SPINNING SUN SENSOR - EAST FACE)
SSS-W (SPINNING SUN SENSOR - WEST FACE)

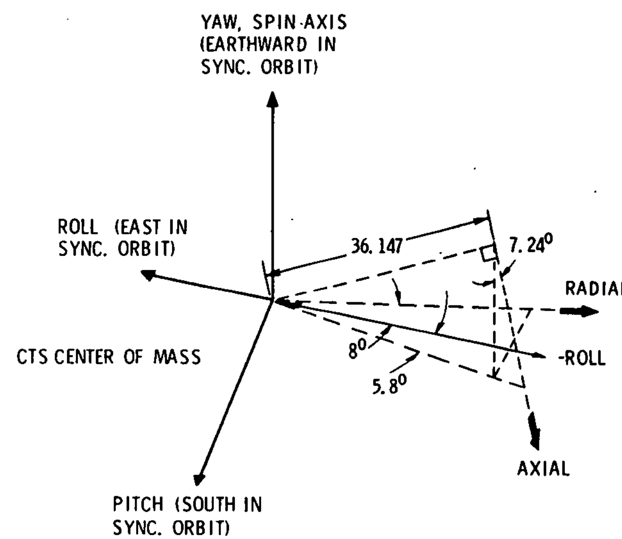
(a) Sensor configuration.



(b) Sun sensor azimuth geometry.



(c) Earth sensor geometry.



(d) Radial and axial thruster geometry.

Figure II-3. - Attitude control geometry.

are encoded and stored in one spin period. Starting with the next spin period or main frame, they are read out in successive main frames. The data from a spin period require five main frames to be read out completely.

In the perigee-injection attitude the earth sensor coverage data are available for only a few minutes after transfer-orbit injection. Shortly thereafter, the sensors lose the earth and do not reacquire until the spacecraft approaches apogee. Because the earth sensor data near perigee are outside its 18 500- to 37 000-km (9989- to 19 978-n mi) operating range, the data are of degraded quality and cannot be used for attitude determination.

The sun sensor alone does not provide a sufficiently redundant data set for attitude determination. Therefore, the spin-axis precession maneuver will not commence until the initial state of spin-axis attitude is determined, with the earth sensor data near first apogee. A coarse precession maneuver may be executed in the neighborhood of the first apogee, where the earth sensor data are available to use in monitoring the maneuver. During the maneuver, the spin axis is precessed in a manner that does not violate the thermal and power design limits of the spacecraft, that maintains adequate antenna patterns, and that provides earth sensor data to use in monitoring the maneuver. The precession maneuver is performed with the 22.5-N (5-lbf) thruster by using a series of pulses synchronized with the spin period. Each pulse is 135 msec long. The locations of the 22.5-N (5-lbf) axial and radial thrusters are shown in figure II-3(d).

A command to the Transfer Orbit Electronics (TOE) specifies the interval between the leading edge of the sun pulse and the leading edge of the thruster pulse; the number of pulses, to a maximum of 31 thruster pulses; and the thruster to be used. The TOE terminates the firing after the number of fired pulses reaches the commanded value. A planned maneuver early in the transfer orbit is scheduled for calibration of the thruster prior to the precession maneuver. Orbital and attitude information is continually provided during the transfer-orbit operations such that the random errors in the attitude determination process can be reduced and a precise apogee-motor firing attitude can be obtained.

Apogee-Injection Phase

The second and third stages of the launch vehicle will place the spacecraft into an inclined, elliptical transfer orbit having an apogee biased above the nominal synchronous altitude. The apogee motor raises the perigee to near-synchronous altitude and changes the orientation of the orbital plane in inertial space by reducing the orbital inclination and possibly performing a nodal rotation. Nominally, the orbit established following injection has a semimajor axis other than synchronous. Hence, the drift rate is established for which the trajectory was designed, and the motor impulse is sized. A small inclination bias remains such that by establishing the synchronous orbit with the proper ascending node, the orbital inclination is passively controlled. For a node at about 270°, the orbital inclination decreases for some period of time in the mission such that the communications experimenters' tracking requirements can be satisfied over much of the 2-year lifetime. The injection node of 270° may be attained by launching directly for a node of 270° or by aiming for another node and then rotating the node to $270^{\circ}\pm30^{\circ}$. The yearly launch opportunity can be increased by launching at a node other than 270° to satisfy the spin-axis - sun-angle and shadow constraints in the transfer orbit. Then the node is rotated with either the apogee motor or the hydrazine system to get a satisfactory passive orbital inclination in the final orbit.

The factors which influence the selection of the apogee for firing the apogee motor, include spacecraft health, the time to drift to station, the convergence of the attitude determination process, and network support considerations. An apogee will not be considered unless at least two stations will be viewing the spacecraft during the motor firing.

In order to limit the time for final-station acquisition to a minimum of 20 days, an apogee injection sufficiently close to the final station and compatible with drift rate and direction caused by apogee altitude dispersions will be selected. The apogee-motor firing is commanded from the ground and places the spacecraft in a drift orbit toward the final station. Orbital inclination following apogee injection is to be less than 0.9° at a node of $270^{\circ}\pm30^{\circ}$.

Preliminary results indicate an apogee bias of 740 km (400 n mi) high and a transfer-orbit inclination of 27.2° to be optimum. For this orbit, the apogee-motor

TABLE II-3. - STANDARD TRANSFER-ORBIT PARAMETERS

Parameter	Initial injection	Transfer trajectory	Mission orbit
Time from lift-off, sec	555.65	1681.6	-------------
Latitude, deg	26.237 N (geodetic)	12.297 S (geodetic)	0
Longitude, deg	58.304 W	5.362 E	116 W±0.2
Geocentric radius, km (n mi)	6478 (3495)	6507 (3511)	-------------
Inertial flightpath angle, deg	-----------------	1.01	-------------
Eccentricity	0.0022466	0.730	0
Inclination, deg	28.3	27.5	0.9 (initial)
Argument of perigee, deg	117.58	180.0	(a)
Anomalistic period, hr	-----------------	10.52±0.35	24
Perigee, km (n mi)	180.2 (97.2)	185 (100)	36 342 (19 623)
Apogee, km (n mi)	213.5 (114.9)	36 470 (19 692)	36 342 (19 623)
Maximum eclipse, min		30	0 to 72
Spin-axis - sun-line angle, deg	-----------------	90±25	-------------

[a]Nonapplicable.

fuel is 310 kg (682 lbm), and the spacecraft weight delivered to the station is 355 kg (782 lbm). Apogee-motor hardware and post-apogee-motor-firing precession fuel are included in these estimates. This assumes no nodal rotation and a 6-deg/day nominal drift rate to station. The orbital parameters listed in table II-3 are for a "standard" transfer orbit with no bias and are included only to indicate the general shape of the transfer- and drift-orbit trajectories.

Station-Acquisition Phase

Following apogee injection, the spacecraft spin axis is oriented and kept normal to the orbital plane throughout the station-acquisition phase, with the apogee-motor nozzle pointing northward. During this phase, apogee injection errors are corrected, and a drift rate is optimized until the orbit is synchronized at the operating longitude, 116° west. The detailed velocity maneuver sequence required for orbital correction and drift to station depends on the injection errors and must be defined on the basis of orbital determination results following apogee injection. No inclination corrections are programmed. Consequently, all orbital correction maneuvers will be in the orbital plane. The velocity maneuvers provide a sequence of Hohmann transfers, each of which is followed by detailed orbital and attitude determinations prior to succeeding maneuvers.

The nominal orbit subsequent to the apogee-motor firing has a semimajor axis either greater than or less than the synchronous value, depending on whether the trajectory is designed for an eastward or a westward drift. The actual orbit is affected by launch vehicle dispersions at perigee, the uncertainties in the apogee-motor firing attitude, errors in the apogee-motor impulse, and the coning error of the apogee motor in the transfer orbit.

The actual drift orbit will be calculated after injection by the use of an orbital determination program. Some adjustment to the drift rate may be necessary to assure that the spacecraft arrives on station in an acceptable period of time. Adjustments to the orbital velocity are accomplished with the 22.5-N (5-lbf) radial thruster operating with a series of 135-msec pulses per revolution. The pulse timing is referenced to the sun line. The maneuvers are commanded from a ground station.

During the drift to station, the spacecraft spin-axis orientation with respect to the sun line must satisfy the thermal and power design limits. As the spacecraft approaches the operating station, orbital velocity corrections are made such that the orbit is circular at station acquisition. If not before, upon arrival at 116° west longitude, the spin axis is oriented normal to the equatorial plane. The mission control will then be transferred to the CRC ground station at Ottawa, where a spacecraft despin and further attitude acquisition maneuvers will be commanded and analyzed.

Attitude-Acquisition Phase

The spacecraft launch and the mission up to station acquisition are a NASA responsibility. After station acquisition, with the satellite spinning about its yaw axis normal to the final orbital plane, spacecraft control will be transferred to CRC. The interface between NASA and CRC control is termed the spacecraft maneuver phase. The spacecraft is then despun, the solar arrays deployed, and the spacecraft oriented such that the yaw axis points to the center of the earth, the pitch axis is normal to the orbital plane and points south, and the roll axis points along the orbital velocity vector.

The solar panels which point north and south have sun sensors that permit them to track the sun while the spacecraft centerbody yaw axis remains coincident with the earth radius vector. However, only one degree of freedom of rotation between the solar panels and the spacecraft centerbody is provided. Therefore, the solar panel normal is displaced from the earth-sun line by the amount of the sun's declination. This displacement results in a maximum solar panel power loss of 8.25 percent. The maximum power loss occurs only twice a year, at the solstices, when the declination is $23\frac{1}{2}^{\circ}$.

In-Orbit Operation Phase

Throughout the duration of the mission, the spacecraft experiences east-west perturbations caused by earth oblateness and solar radiation pressure. It experiences north-south perturbations because of solar-lunar gravitational attractions. The north-south perturbations will not be corrected. East-west stationkeeping is performed throughout the mission with the RCS to maintain the spacecraft on station to within $\pm 0.2^{\circ}$ of the nominal 116° west longitude.

SPACECRAFT TESTS ON STATION

Prior to station acquisition, the Telemetry, Tracking, and Command System (TT&C) is powered by the Body Solar Array (BSA). This permits the spacecraft health team to assess thermal and power parameters. Once on station, the appropriate three-axis stable conditions that are necessary to support the mission will be established. With the arrays extended and providing power, each subsystem is checked and initial evaluations are made by the health team.

OPERATION ON STATION

Spacecraft operations are organized into experiment and housekeeping operations and can be active or partially passive. Experiments are as follows:

- Transmitter Experiment Package
- Attitude Control System Experiment
- Solar Array Technology Experiment
- Super-High-Frequency (SHF) Communications Experiments (include U.S. and Canadian experiments)

Housekeeping systems are as follows:

- SHF Beacon and Antenna
- Telemetry, Tracking, and Command System
- Deployable Solar Array System
- Power Systems
- Electrical Integration Subsystem
- Attitude Control System
- Structure Assembly
- Reaction Control System
- Thermal Control System

The solar arrays track the sun automatically under normal operations, and the Attitude Control System (ACS) compensates for main body movements to maintain the proper attitude. Most operations, except those occurring during eclipses and nonstandard procedures, will involve the SHF Communications Experiment modes and the SHF antenna pointing changes to support users. The ACS Experiment and the Solar Array Technology Experiment (SATE) will obtain most of their data during normal housekeeping events. Some special active events are scheduled for later in the mission.

III. SPACECRAFT DESCRIPTION

GENERAL EQUIPMENT CONFIGURATION

The basic spacecraft structure consists of a central thrust tube, forward and aft equipment platforms, and north and south equipment panels, as shown in figure III-1. The apogee motor is mounted within the central thrust tube on the same structural member as the aft platform. The launch vehicle mating surface is at the base of the central thrust tube. In the launch configuration the overall height and outside diameter are 188 and 183 cm (74 and 72 in.), respectively. The outer diameter of the aft equipment platform is supported by struts that extend down to the base of the central thrust tube. The central thrust tube is manufactured out of aluminum alloy rings joined together by magnesium sheets rolled into tubular sections. The forward edges of the north and south panels are supported laterally by struts which extend to the forward equipment platform.

The curved east and west panels of the spacecraft are used to carry the majority of body-mounted solar cells. Since the forward platform carries the SHF antennas and earth sensors, which must be very accurately alined, considerable care has had to be taken to thermally isolate it from the rest of the spacecraft. This isolation is accomplished by supporting the center of the forward platform on the cylindrical section of the central thrust tube and supporting the outer edges of the forward platform by struts which are attached to the base of the central thrust tube.

The forward and aft equipment platforms and north and south equipment panels are 2.54 cm (1 in.) thick aluminum honeycomb panels.

The extendible solar array is mounted on the north and south panels. A flat-pack design is used for the stowed configuration. Each panel contains a single blanket and a single offset boom per sail. Jettisonable stowage containers are provided for the solar blankets. The enclosure release mechanism will be actuated on command and provides for disconnection of electrical leads to the solar panels on the enclosure lids. Each sail will be extended in a concertina-like fashion (fig. III-2) by means of a single silver-plated stainless steel extendible BISTEM boom. The booms are located on the shadowed side of the blankets. The boom is 0.18 mm (0.007 in.) thick, 3.9 cm (1.5 in.) in diameter, and 7.37 m (24 ft) long when fully deployed. The array is instrumented with strain gages and accelerometers to provide information on its flexible-body dynamics and its effect on the accuracy of the three-axis attitude control system.

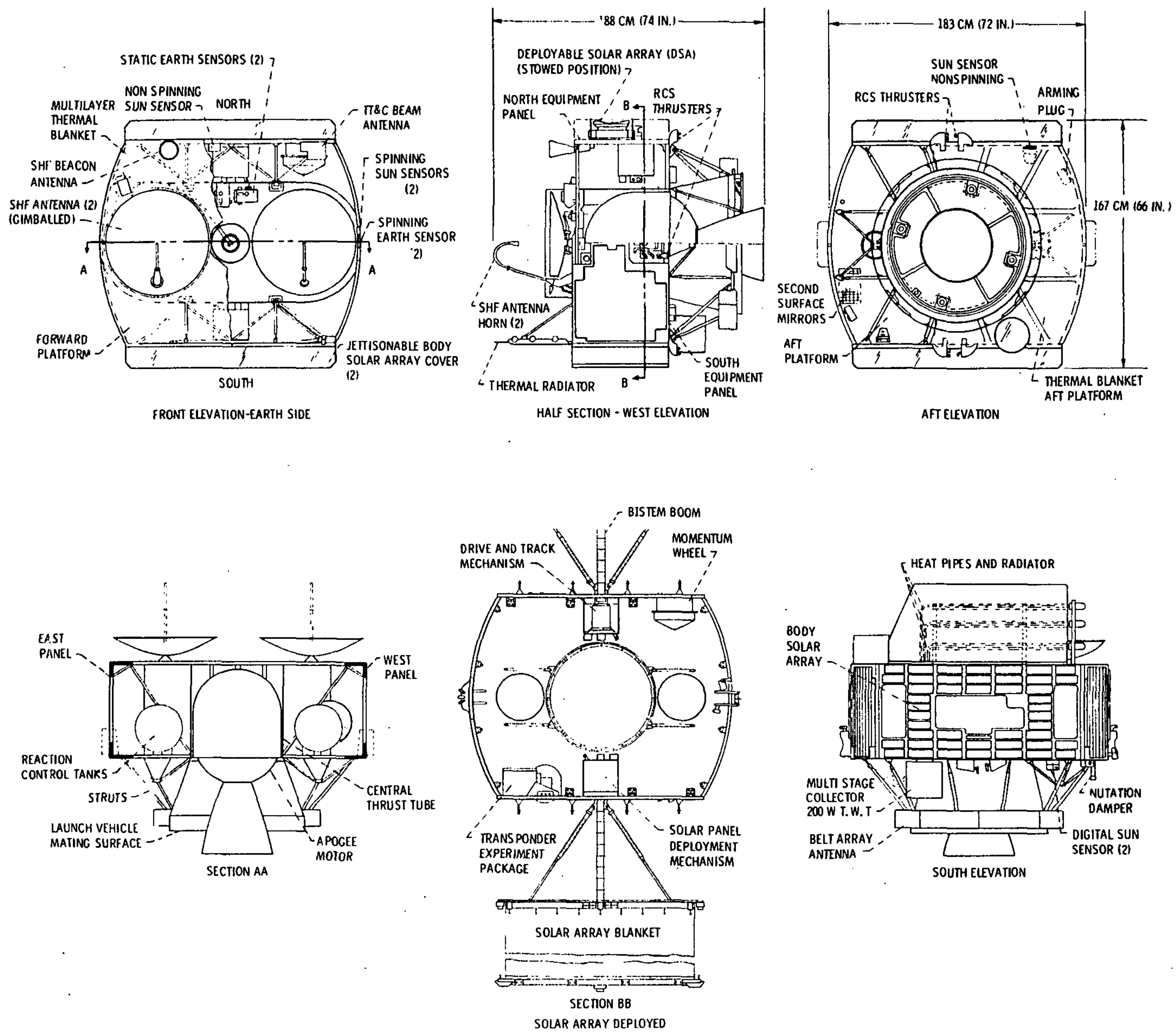
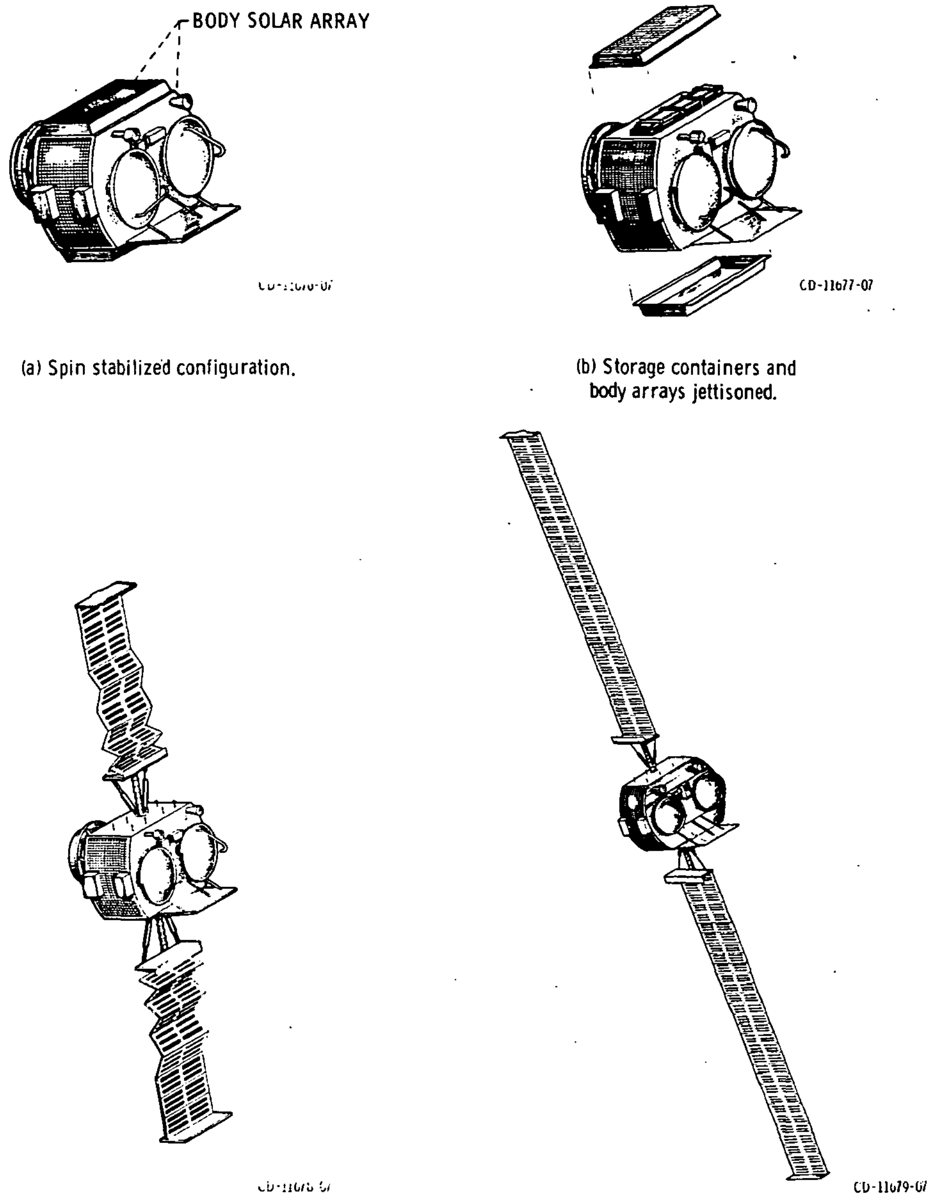


Figure III-1. - CTS spacecraft configuration.



(a) Spin stabilized configuration.

(b) Storage containers and body arrays jettisoned.

(c) Array partially deployed.

(d) Array fully deployed.

Figure III-2. - Extendible solar array actuation sequence.

Mounted on the forward platform are the two SHF narrow-beam (2.5° for 3-dB beamwidth) antennas, the TT&C conical beam and belt antennas, the static infrared earth sensors, and the SHF beacon antenna. The 12-GHz, high-efficiency TWT is mounted at the intersection of the aft platform and the south panel. The Multistage Depressed Collector (MDC) of the TWT protrudes from the aft end of the spacecraft. The Variable Conductance Heat Pipe Subsystem (VCHPS) and Radiator are mounted as an extension to the south panel and are held in place by struts to the forward platform.

Projecting through the curved east and west panels of the spacecraft are the spinning earth sensors, the spinning sun sensors, the radial reaction control jet, and the yaw and east/west low-thrust reaction control jets. The pitch, roll, and Wheel Control of Spacecraft Momentum and Attitude (WHECON) offset roll/yaw low-thrust jets and the axial reaction control jet are mounted on the aft platform. The solar array deployment, tracking and drive mechanisms, and slip ring assemblies are mounted on the north and south panels of the spacecraft.

The Reaction Control System hydrazine fuel tanks are on the aft platform. Integral with the separation ring around the aft end of the thrust tube is a 32-element, 2-GHz belt array antenna, which, in conjunction with the TT&C antennas on the forward platform, will provide near omnidirectional coverage for telemetry and command. This is particularly important during the transfer-orbit and attitude-acquisition phases of the mission.

The forward platform is thermally isolated from the rest of the spacecraft and insulated from outer space by multilayer blankets. The solar-cell panels on the east and west faces are also thermally isolated from the spacecraft.

Components mounted on the aft platform include the Reaction Control System, batteries, and the housekeeping components required to operate during transfer and drift orbits. Battery thermal control is achieved by using Second-Surface Mirrors (SSM) to increase the heat radiated from the battery mounting area via the south panel. Heaters are provided to maintain critical components within their survival temperature range when these components are not in use. Examples are the 200-watt TWT and the thrusters for the Reaction Control System.

The apogee motor is insulated to ensure that propellant temperature remains above -7° C (18° F) at the end of 10 transfer orbits. There is a thin stainless-steel mesh shield attached to the aft end of the central thrust tube to protect the spacecraft from the apogee-motor plume temperatures.

When deployed, the extendible solar array must permit satisfactory heat radiation from the north and south panels. For this reason, it has been necessary to provide a mechanism which elevates the end of each sail 76 cm (30 in.) above its respective panel.

The weight breakdown of the spacecraft is summarized in table III-1.

TABLE III-1. - SPACECRAFT WEIGHT BREAKDOWN

System	Weight	
	kg	lbm
Telemetry, Tracking, and Command System	14.6	32.2
Super-High-Frequency Communications	67.2	147.9
Power Conditioning	21.5	47.3
Batteries	14.9	32.7
Body Solar Array	3.2	7.1
Flexible Solar Array Blankets	13.8	30.3
Flexible Solar Array Structure including Slip Rings and Deployment and Acceleration Mechanism	46.3	101.8
Wiring Harness and Electrical Integration	15.5	34.0
Attitude Control System	24.7	54.3
Basic Structure	55.1	121.2
Thermal Control	12.7	28.0
Reaction Control System (RCS) Hardware	18.0	39.5
RCS Fuel	25.1	55.2
Balance Weights	3.2	7.1
Apogee Motor and Fuel	341.0	750.2
Spacecraft Pad Weight	676.8	1488.9
Spacecraft Usable Weight in Synchronous Orbit	347.1	763.6
Lift-off Margin	+7.6	+16.7

THERMAL CONTROL SYSTEM

Throughout the various phases of the mission, including prelaunch operations, there will be wide variations in the amount of heat dissipated by components within and on the outer surfaces of the spacecraft. During the spin-stabilized phase, the power dissipation will be relatively low, typically 55 to 90 watts. In synchronous orbit the spacecraft will be three-axis stabilized, and the power dissipation will normally be 240 to 600 watts. Provision must be made for survival of components through eclipse periods and for operation of the spacecraft in synchronous orbit with any or all of the experiments switched off. The thermal design is also complicated by the presence of an apogee motor and by the need for low thermal distortion of the forward platform in order to provide a dimensionally stable platform for the SHF antennas and attitude sensors. An exploded view of the thermal control components and their location on the spacecraft is shown in figure III-3.

Components of the Thermal Control System (TCS) include multilayer insulation blankets, radiating areas of Second-Surface (quartz) Mirrors (SSM) on aluminum honey-

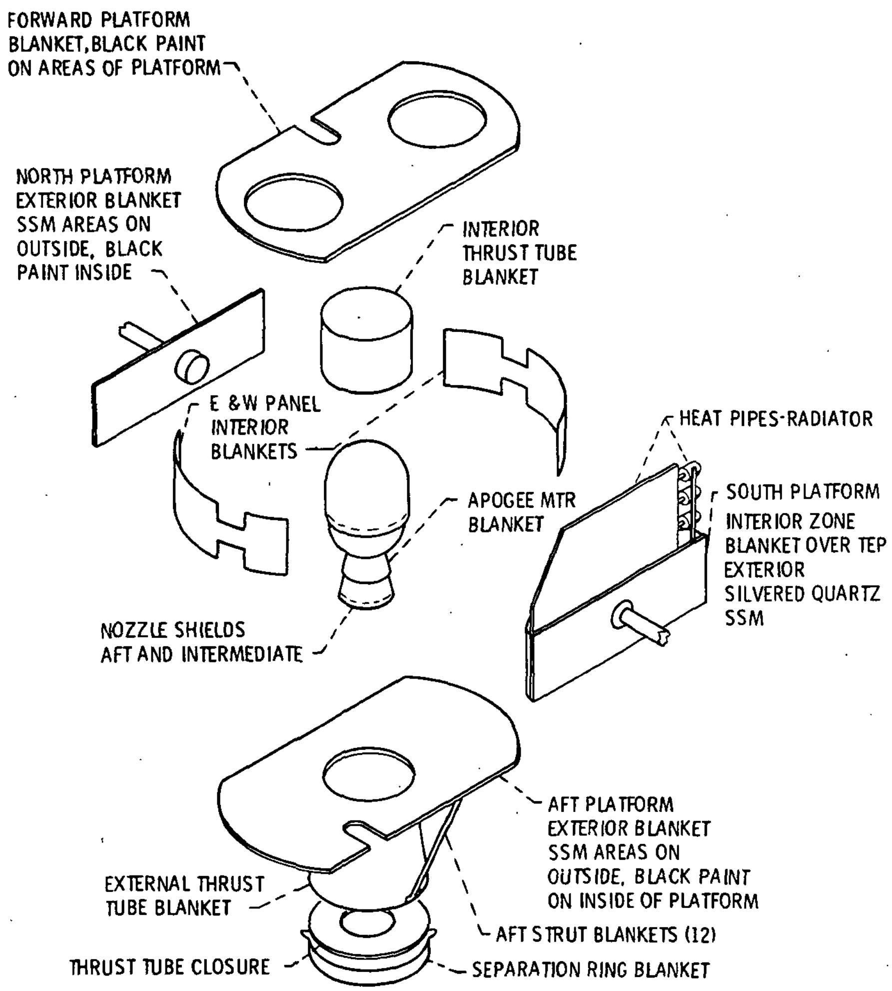


Figure III-3. - Thermal control components.

comb platforms, various other thermal coatings, commandable heaters, and aluminum doubler plates. A Variable Conductance Heat Pipe Subsystem (VCHPS) rejects the heat dissipated by the 200-watt Transmitter Experiment Package (TEP) through an externally mounted radiator on the south panel. These heat pipes use methanol as the working fluid. All major dissipating components are mounted directly on the north and south panels. Low dissipating units are mounted on the forward and aft platforms. In the synchronous mission phase, the Deployable Solar Arrays (DSA) are extended approximately 76 cm (30 in.) beyond the north and south panels to allow satisfactory heat rejection to space.

Since the lateral conductance of the honeycomb panels (north and south) is insufficient to transfer some of the highly concentrated heat, thermal doublers bonded to the honeycomb internal face sheet are used in these areas. When major dissipating components are not operating, substitute heaters for these components are energized by command. In addition, eight commandable thermal control heaters are available to compensate for the wide range of power dissipations. Two of the eight thermal control heaters are available during the spinning phase of the mission.

SPACECRAFT SPIN-STABILIZED MODE

The spacecraft configuration during the spin-stabilized mode is shown in figure III-2(a). The spacecraft will be spinning about the yaw axis at a nominal rate of 60 rpm throughout the transfer-orbit, apogee-injection, and station-acquisition phases. The deployable solar array will be in the stowed position. The east and west panels and the north and south jettisonable enclosures carry the body-mounted solar cells that, together with the battery, provide the spacecraft with power during spin-stabilized operation.

Commands can be received by either the belt array on the thrust tube or the high-gain antenna on the forward platform. Telemetry can be transmitted on either antenna, selection being made by command from the ground. The sun sensors and spinning earth sensors are used for attitude determination and provide reference timing for the axial and radial thrusters.

SPACECRAFT THREE-AXIS STABILIZED MODE

The spacecraft configuration during the three-axis stabilized mode is shown in figure III-2(d). The spacecraft is on-station with the solar array deployed. The forward platform contains the static infrared earth sensor, which provides roll and pitch information required for keeping this platform facing the earth (fig. III-4). Yaw information is provided for two $8\frac{1}{2}$-hour periods in each 24-hour orbital period by sun sensors on the

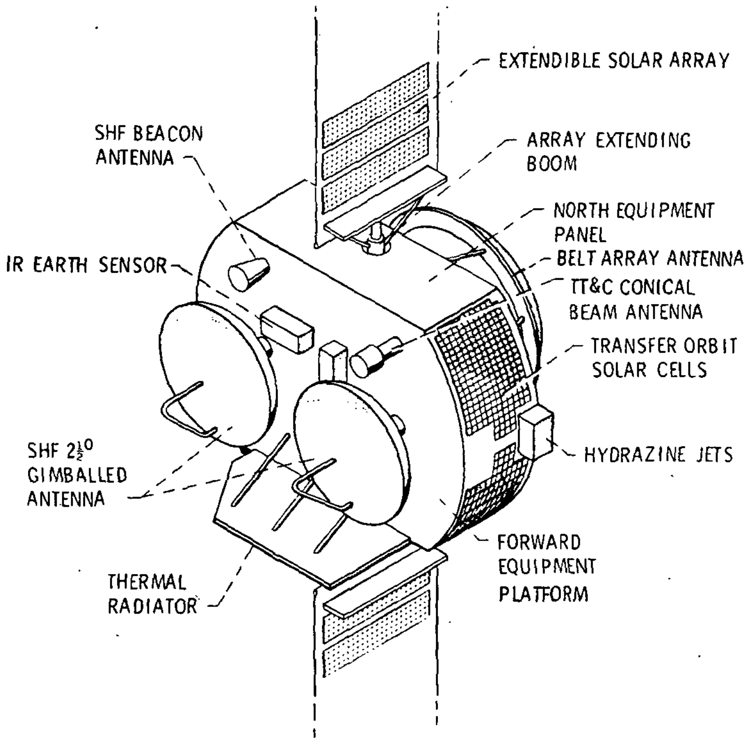


Figure III-4. - Structural view of forward equipment platform.

east and west panels. The two SHF antennas are gimbaled, and each is capable of pointing its beam anywhere on the earth's disk.

When the deployable solar array is in its fully extended operational configuration, the axes of the extendible booms coincide nominally with the pitch axis of the spacecraft. Orientation of the array panels toward the sun throughout the daily orbit is maintained by rotating the array once a day around the pitch axis. The inboard ends of the solar arrays are approximately 76 cm (30 in.) away from the north and south surfaces of the spacecraft. This is to allow a suitable view into space for the radiators on the north and south platforms. Figure III-5 shows the components with the forward platform and thermal shields removed.

SUPER-HIGH-FREQUENCY COMMUNICATIONS SYSTEM

The SHF Communications System will provide microwave signals in the 12-GHz region at power levels greater than heretofore available. It features a number of advanced technology units which will make possible low-cost ground terminals and permit communications with remote areas otherwise beyond economical limits. The communications system consists of the following elements: high-gain receiving and transmitting antennas; a receiver; an amplifier; filter circuits; a high-power, high-efficiency output stage (Transmitter Experiment Package, or TEP); and a "spotting" beacon and its antenna. A simplified block diagram of the SHF system is shown in figure III-6. The frequency plan, showing both the receive and transmit bands, is given in figure III-7.

TRANSMITTER EXPERIMENT PACKAGE

The TEP is a nominal 200-watt microwave power amplifier and consists of three major hardware assemblies, as shown in figure III-8:

- Power Processor Subsystem (PPS)
- Output Stage Tube (OST), which includes
    - Traveling Wave Tube (TWT)
    - Multistage Depressed Collector (MDC)
- Variable Conductance Heat Pipe Subsystem (VCHPS), which include
    - Radiator
    - Reservoirs
    - Heat pipes

Significant performance features of the TEP are noted in table III-2, and a block diagram of the TEP Power Processor Subsystem (PPS) is shown in figure III-9.

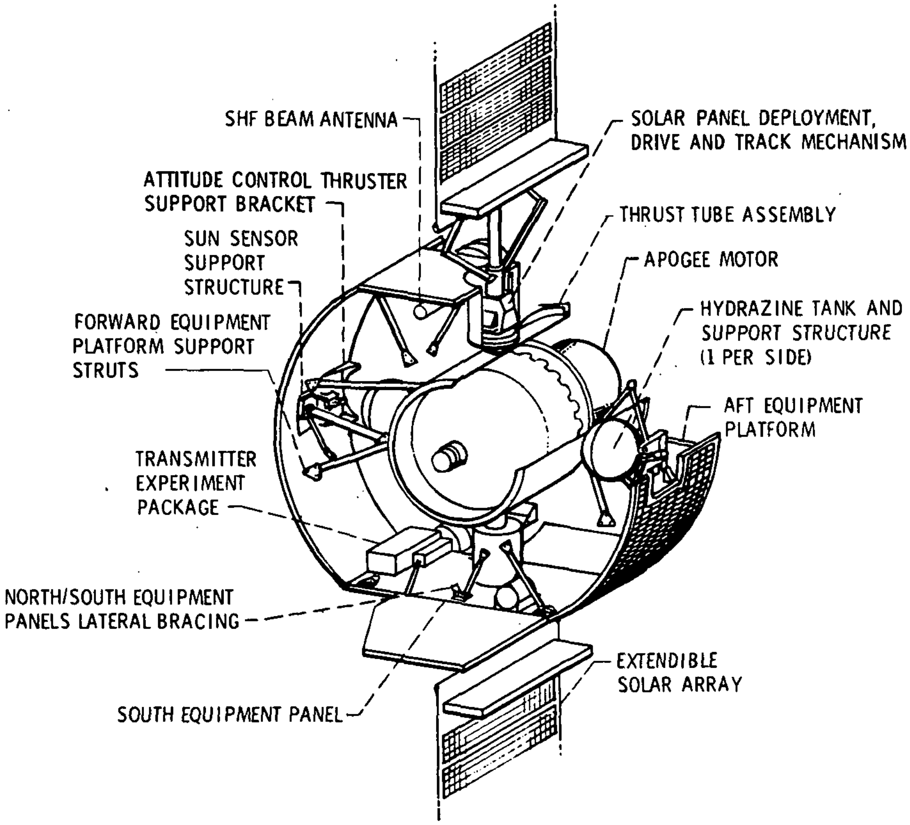


Figure III-5. - Structural view with forward platform and thermal shields removed.

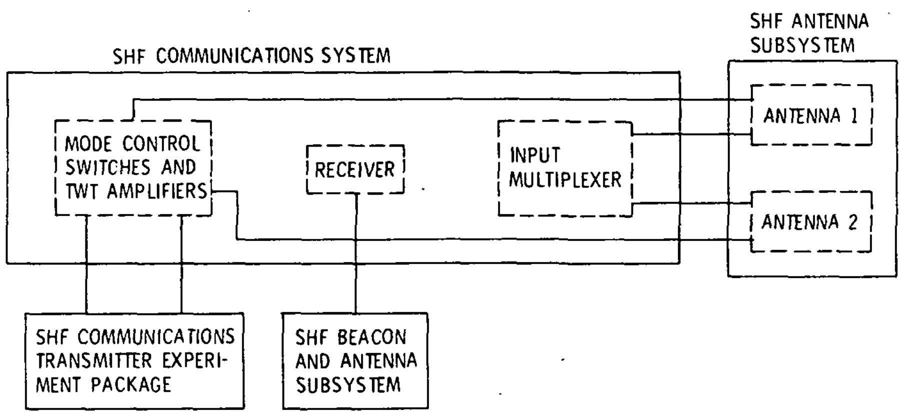


Figure III-6. - Super-High-Frequency Communications System block diagram.

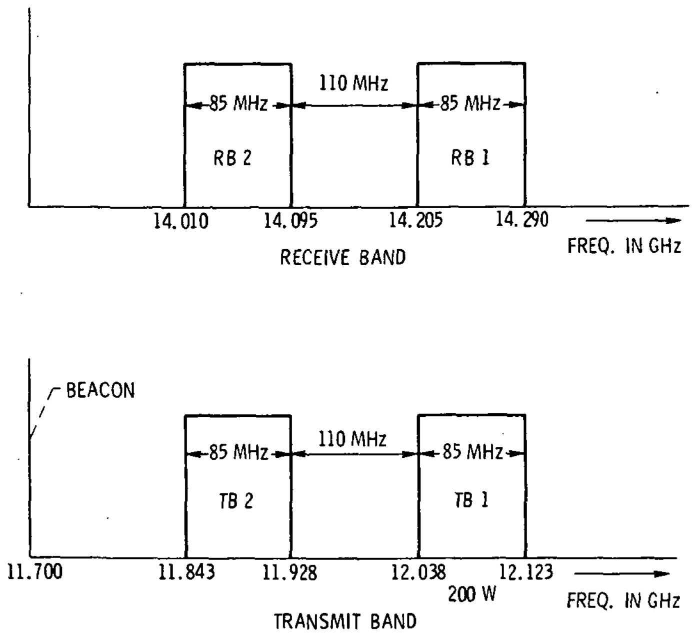


Figure III-7. - Frequency plan.

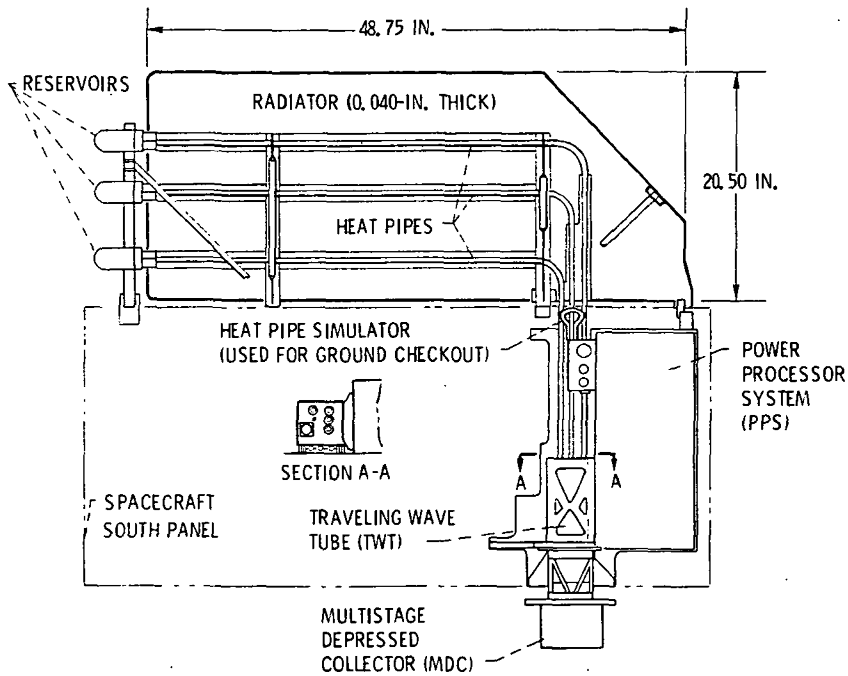


Figure III-8. - Transmitter Experiment Package and Variable Conductance Heat Pipe Subsystem.

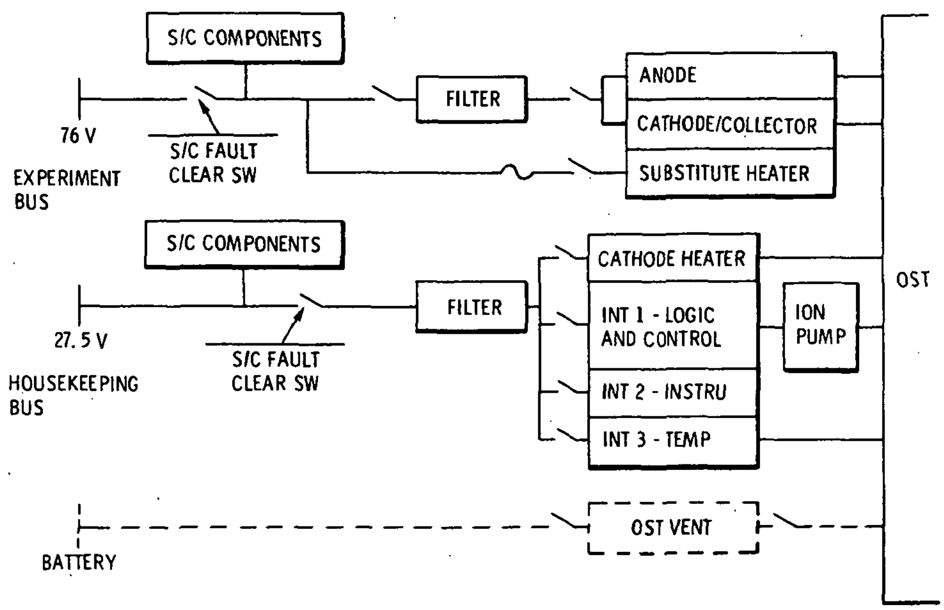


Figure III-9. - TEP Power Processor Subsystem block diagram.

TABLE III-2. - PERFORMANCE FEATURES OF

TRANSMITTER EXPERIMENT PACKAGE

Subsystem	Features
Output Stage Tube (OST)	RF amplifier at 12 GHz Nominal 200-watt output (saturated) 30-dB saturated gain 50 Percent efficiency (target)
Power Processor Subsystem (PPS)	500-Watt output Protection circuits 20 Commands and 37 telemetry channels
Variable Conductance Heat Pipe Subsystem (VCHPS)	196-Watt transport and reject capability at 50° evaporator Performance adequate with one of three pipes failed

Functionally, the PPS performs the following operations:

- Conditions solar array power for the OST
- Provides circuit protection for the PPS and OST
- Controls the TEP in response to commands
- Processes telemetry signals
- Contains TEP auxiliary heaters

The high-powered TWT provides an RF power output of nominally 200 watts with about 100 milliwatts of drive power. The drive power is supplied from one of the two 20-watt TWT's. The entire OST consists of an advanced-design coupled-cavity traveling wave tube, a novel multistage depressed collector, and a spent-beam refocusing arrangement. A velocity taper technique is employed to prevent loss of synchronism between the circuit wave and the electron beam in order to maximize efficiency. Overall efficiency is further improved by the collector design which features schemes for independently effecting the sorting of low- and high-velocity beam components to maintain a high ratio of recovered kinetic beam power to the beam input to the collector.

The VCHPS supplements the baseplate thermal capability by providing additional heat transport from the baseplate. The VCHPS also acts to block heat flow to a cold baseplate, if necessary.

The PPS supplies the OST with the necessary high dc voltages and provides the required voltage regulation. Table III-3 lists the PPS performance requirements and figure III-10 shows the overall physical dimensions of the unit. The PPS also provides the command, telemetry, protection, and logic functions necessary for proper ground control, as well as diagnostic information.

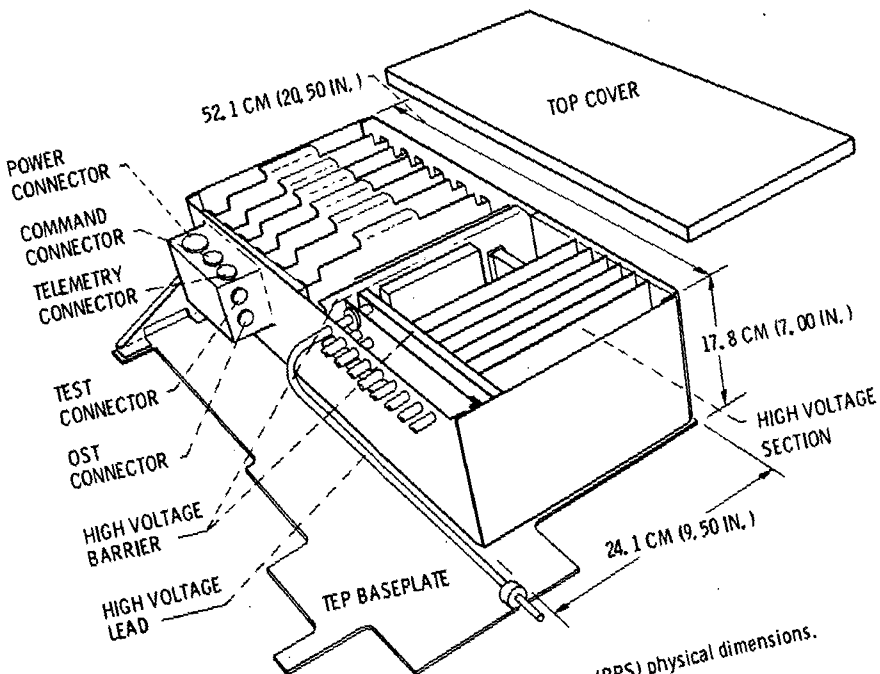


Figure III-10. - Power Processor Subsystem (PPS) physical dimensions.

TABLE III-3. - PERFORMANCE REQUIREMENTS OF

POWER PROCESSOR SUBSYSTEM

Converts 28-V and 76-V dc power to -
Nine high-voltage collector supplies (cathode voltages, -11.3 kV; current, 72 mA; regulation, ±1 percent; ripple, 0.01 percent)
Cathode heater supply voltage, 4.2 V; current, 1.5 A; (floating at -11.3 kV)
Anode supply (voltage, 350 V±200 V; current, 100 mA)
Ion pump supply (voltage, 4 kV; current, 50 μA)
Protects system against -
Excess body current in OST
Excess pressure in OST
Excessive arcing
Low voltage on input lines (tolerates 95 and 45 V)
Power conversion efficiency, 85 percent

TABLE III-4. - PERFORMANCE SPECIFICATIONS

OF OUTPUT STAGE TUBE

Nominal RF output power, W	200
Target efficiency, percent	50
Center frequency, GHz	12.0805
Bandwidth (3 dB), MHz:	
Minimum	85
Maximum	250
Saturated gain, dB	30
Small signal gain variation (peak to peak), dB	5
Second-order phase deviation, deg/MHz^2	0.2
Noise figure, dB	40

The overall physical dimensions of the OST microwave amplifier are shown in figure III-11, and some of the more important performance specifications are provided in table III-4. The input and output sections are the tube region where the RF input interacts with an electron beam and amplification takes place by conversion of beam kinetic energy. The Multistage Depressed Collector (MDC) increases the overall efficiency of the amplifier by converting the kinetic energy remaining in the spent electron beam to potential energy and thus reducing the power required to operate the tube. The MDC consists of a set of 10 electrodes (fig. III-11), which form equipotential surfaces and collect electrons at near zero velocity. This technique minimizes heat, and any remaining kinetic energy is dissipated as heat, which is radiated to space through the collector enclosure. The MDC, which protrudes out of the spacecraft, can operate to 300° C

TABLE III-5. - MAJOR POWER REQUIREMENTS OF

TRANSMITTER EXPERIMENT PACKAGE

[Total energy requirement of special instrumentation not to exceed 5 W-hr during ascent, transfer orbit, and drift to station.]

Period of operation	Nominal power requirement, W	
	From housekeeping bus	From experiments bus
Ground testing	26.2	585
On launch pad (umbilical)	26.2	80.0
During ascent and transfer orbit	1.2	0
PPS enable:		
All instrumentation, command, and protection on	16.8	.5
With cathode heater at half power	20.2	.5
With cathode heater at full power	26.2	.5
Eclipse operation:		
With cathode heater at half power	5.0	0
With cathode heater at full power and special instrumentation on	6.1	0
Substitute heaters	0	[a]71 - 82

[a]At 76 V.

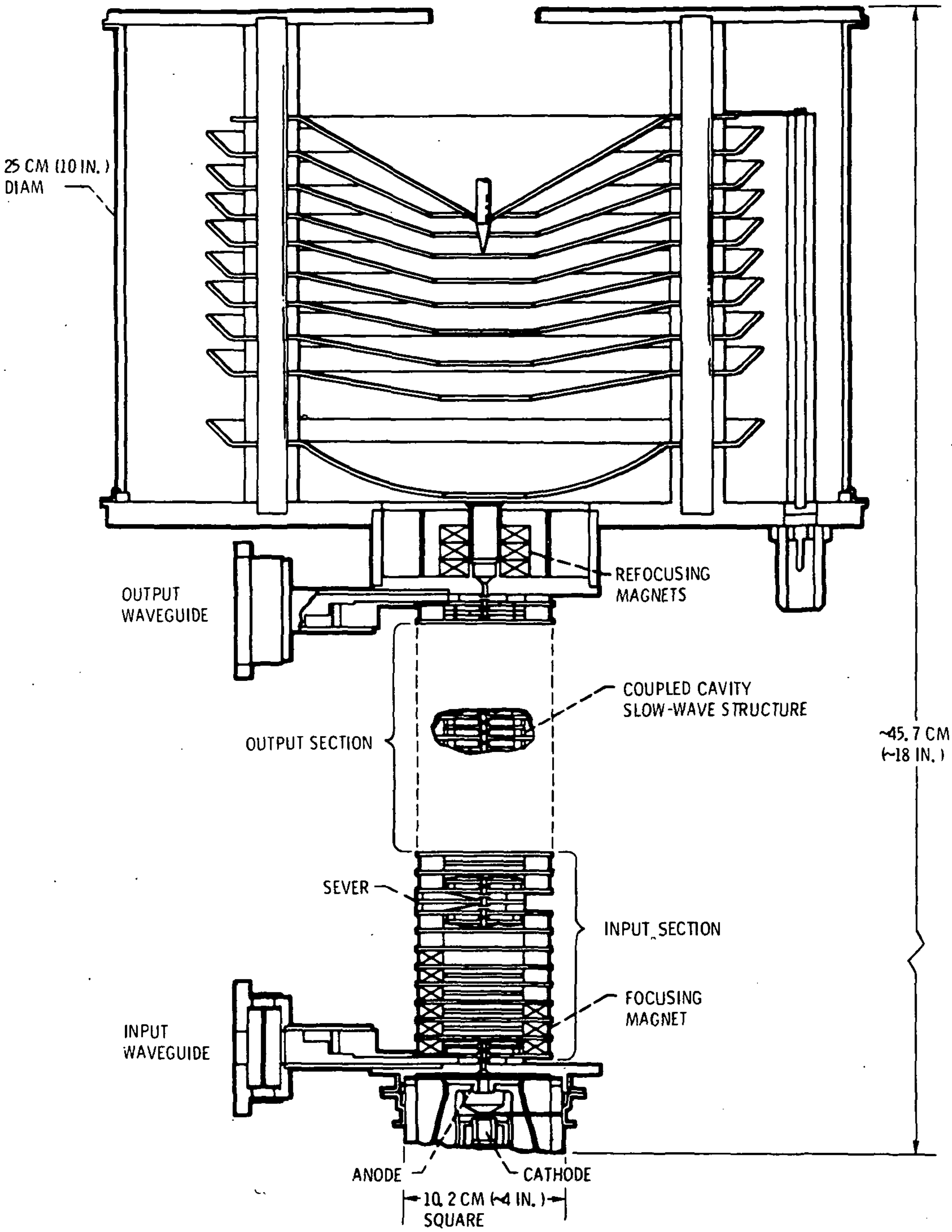


Figure III-11. - Coupled-Cavity Traveling Wave Tube with Multistage Depressed Collector.

(572° F). The TWT, which is located inside the spacecraft, operates at about 55° C (131° F).

Major Power and Signal Interfaces

The spacecraft will supply power to the TEP from two sources: an experiments bus and a housekeeping bus. These requirements are identified in table III-5 and are shown in figure III-9. The nominal voltage of the experiments bus is 76 volts. Nominal end-of-life and beginning-of-life voltages at maximum power are 65 and 91 volts, respectively. The housekeeping bus provides a nominal 27.5 volts ±3 percent, from the solar array during normal sunlight operation and from the battery system during eclipse. The TEP also includes two voltage-protection circuits in the event of housekeeping-bus malfunction or experiments-bus malfunction either through under- or overvoltage situations.

Table III-6 identifies the major telemetry interfaces for TEP measurement. A sampling rate of once per second is used. Some instrumentation is operated from the housekeeping bus during eclipse and launch operations. The telemetry data are encoded by one of two encoders and are allocated such that not all TEP data are available unless the correct encoder is operating.

TABLE III-6. - MAJOR TELEMETRY INTERFACES OF

TRANSMITTER EXPERIMENT PACKAGE

Cathode heater voltage, V dc	0 - 10
Cathode voltage, kV	0 - -15
Beam current, mA	0 - 100
Tube body current, mA	0 - 15
Anode voltage, V dc	0 - 600
Collector voltages (3), kV	0 - -10
Collector currents (10), mA	0 - 40
OST and PPS temperatures, °C (°F)	-55 - +225 (-65 - +435)
RF power (2), W	0 - 250
OST internal pressure, N/m^2 (torr)	1.33×10^{-4} to 1.33×10^{-6} (10^{-6} - 10^{-8})
Bus currents, A	0 - 10
Heat-pipe temperatures (6), °C (°F)	-73 - +93 (-100 - +200)

Commands for the operations of the TEP are routed to the PPS and cover a variety of functions whose control is based on evaluation of TEP data received on the ground.

TEP Performance

Transponder and TEP interface performance levels required to achieve satisfactory operation have been established. Some of these levels are

- The maximum Voltage Standing Wave Ratio (VSWR) of the load on the output port of the OST must be less than 1.25.
- The useful bandwidth or passband of the OST is defined as 12.038 GHz to 12.123 GHz (85 MHz).
- The TEP output power is a nominal 200 watts averaged across the 85-MHz band.
- The gain of the OST when operating under a single-carrier saturated condition is 30 dB (+2 dB, -1 dB) over the mission life and averaged across the useful bandwidth.
- The OST will have a maximum noise figure of 40 dB.

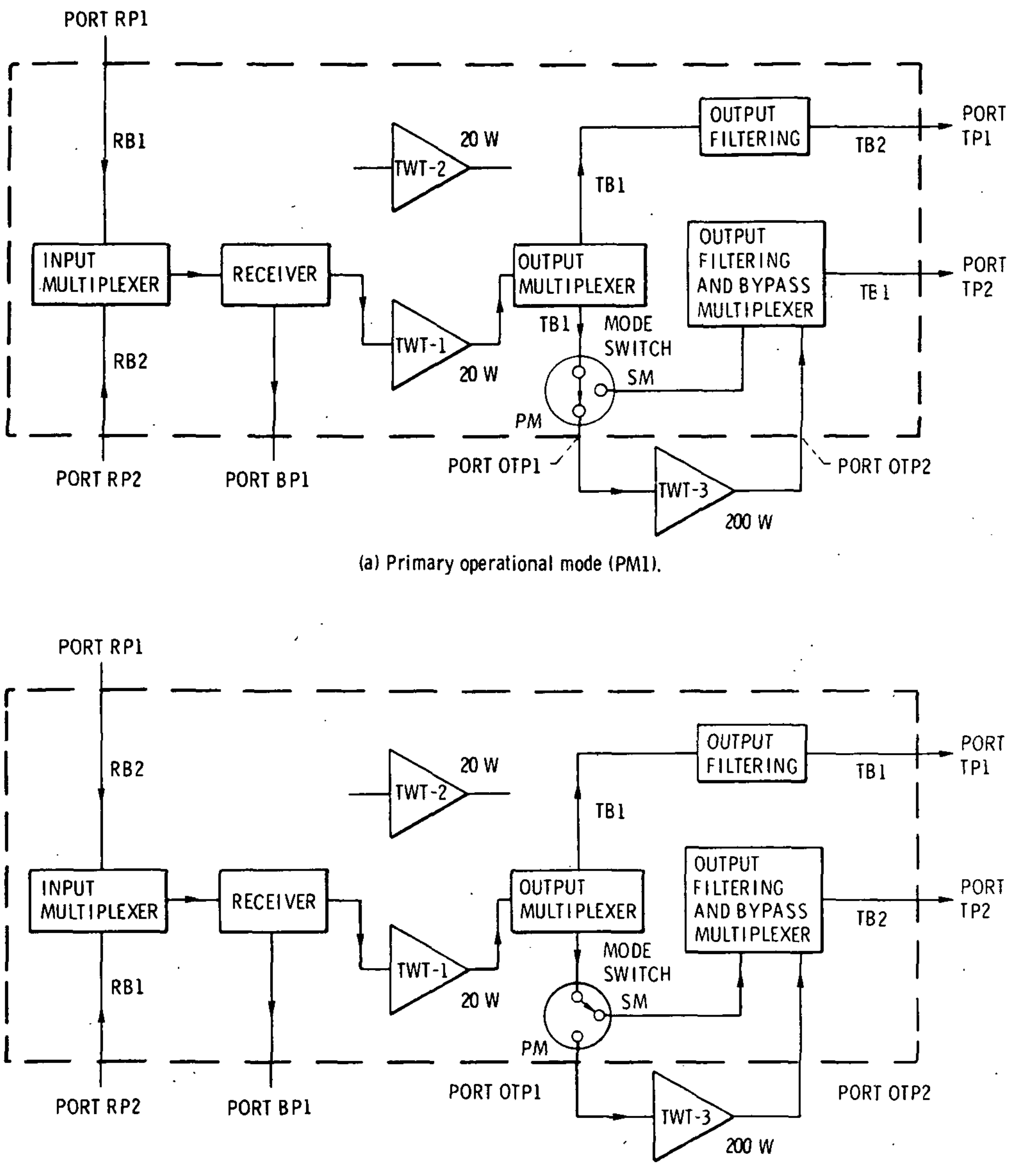
In the primary transponder configuration, only a small fraction of the output power of the 20-watt TWT drives the 200-watt TWT and its antenna. The remainder of the output power drives the second antenna.

SUPER-HIGH-FREQUENCY TRANSPONDER

The major elements of the SHF transponder are shown in figure III-12, which depicts both primary and secondary modes of operation. It is a multiport microwave system. One receive port and one transmit port are connected to a single antenna by means of an orthomode coupler to allow for simultaneous reception and transmission of orthogonal linearly polarized signals. Similarly, the second receive port and second transmit port are connected to an identical antenna. The transponder is designed so that a signal of the proper frequency and amplitude received by one antenna will be translated in frequency (without inversion), amplified, and transmitted by the other antenna. The two receiving frequency bands (RB-1, 14.010 to 14.095 GHz; and RB-2, 14.205 to 14.290 GHz) and the two transmitting frequency bands (TB-1, 11.843 to 11.928 GHz; and TB-2, 12.038 to 12.123 GHz) are capable of simultaneous operation.

The transponder has the following components:

- Input multiplexer
- Two receivers (including preamplifiers, mixers, amplifiers, translation oscillators, a parametric amplifier and filters)
- Mode control switches



(a) Primary operational mode (PM1).

(b) Secondary operational mode (SM1).

Figure III-12. - Transponder configuration in primary and secondary modes.

- Two 20-watt TWT amplifiers (TWT-1 and TWT-2)
- Output multiplexers
- Filtering circuits

For primary-mode operation, either 20-watt TWT (TWT-1 or TWT-2) is used both as an output amplifier and also as a driver amplifier for the 200-watt TWT amplifier (TWT-3). The saturated gain is nominally 122 dB, averaged across the bands. In the secondary mode, a single 20-watt TWT (TWT-1 or TWT-2) is used. Bidirectional capability is maintained by routing signals from the 20-watt TWT directly to one antenna and through a bypass multiplexer to the second antenna. The nominal saturated gain in the secondary mode is 114 dB, averaged across the bands.

The frequency response of the transponder is ±1 dB over bands RB-1 and TB-1 and within ±1.25 dB over bands RB-2 and TB-2. The noise temperature is less than 2315 K for all modes of operation when the parametric amplifier is not used and 1315 K when it is. The major technology advances in the transponder include the 14-GHz parametric amplifier, the 12-GHz field effect transistor amplifiers, graphite-fiber epoxy composite filters, and low-loss isolators.

Parametric Amplifier

One receiver of the transponder incorporates, for a first stage of low-noise amplification at 14 GHz, a single-stage solid-state parametric amplifier. The components of the amplifier are a main cavity, a pump generator consisting of a 2.5-GHz transistor oscillator followed by two frequency quadruplers, a five-port circulator, and the required stabilizing and power conditioning circuits. Gain and noise figures of the amplifier are 13 and 3.3 dB, respectively.

Field-Effect Transistor Amplifiers

A pair of 12-GHz Field-Effect Transistor Amplifiers (FETA's) is used in a redundant fashion to provide linear amplification of the communications signals prior to the 20-watt TWT's. Each amplifier provides a gain of 24 dB over the frequency band 11.83 to 12.13 GHz.

Low-Loss Isolators

Incorporated in the transponder are a number of isolators capable of high, medium, or low power operation. The units have, in addition to low loss characteristics, low masses. Typically the loss is less than 0.1 dB for a unit capable of handling 250 watts continuous wave (CW) and the weight is less than 140 g (0.3 lbm).

20-Watt Traveling Wave Tube

Two 20-watt traveling wave tubes (TWT's) are used in redundant fashion. Their principal characteristics are a small signal gain greater than 61 dB, a high dc-to-RF efficiency, and a mass of 3.07 kg (6.8 lbm) each (including a power regulator).

SUPER-HIGH-FREQUENCY ANTENNAS

The SHF Antenna Subsystem consists of two steerable parabolic reflectors with single feeds which incorporate linear, orthogonally polarized receive and transmit functions. A block diagram of one antenna is shown in figure III-13. The electrical axis of each antenna is steerable in any direction 8.5° from a line parallel to the spacecraft yaw axis and passing through the vertex of the reflector. Beam width of the antennas is 2.5° included cone angle to the -3-dB points. The transmit gain is 36.9 dB on axis, while the receive gain is 37.9 dB on axis. Isolation between transmit and receive channels is more than 25 dB.

The antennas are steered in increments of 0.12° at a slew rate of 3° per minute. Six channels of telemetered data on the antenna subsystem are transmitted to the ground, and 11 antenna subsystem commands are received by the spacecraft.

SUPER-HIGH-FREQUENCY BEACON

The SHF Beacon, operating CW at 11.7 GHz, provides a convenient signal source to locate and lock on to the spacecraft. It operates at either of two power levels on command (200 or 12 mW). A reference signal is obtained at a frequency of 108.333 MHz from the local oscillator of the transponder receiver and multiplied up to the operating frequency.

The beacon consists of four assemblies (fig. III-14):

- An Injection Locked Oscillator provides the required power conditioning, command control, and telemetry signal. Its package provides the structure for the SHF Beacon as a whole.
- A Reference Signal Generator frequency translates (with appropriate amplification) the 108.333-MHz, 50-mW signal from the SHF Transponder local oscillator.

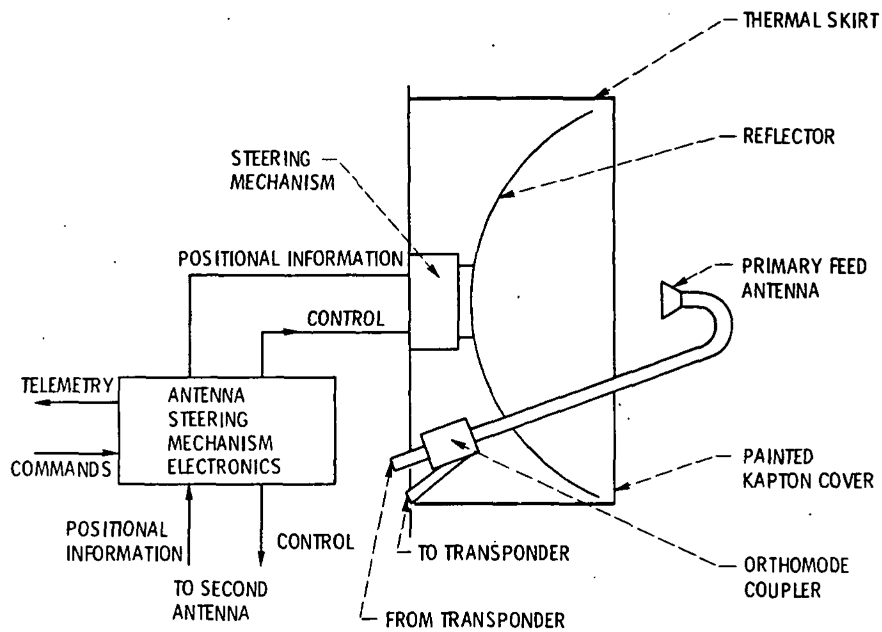


Figure III-13. - SHF Antenna block diagram.

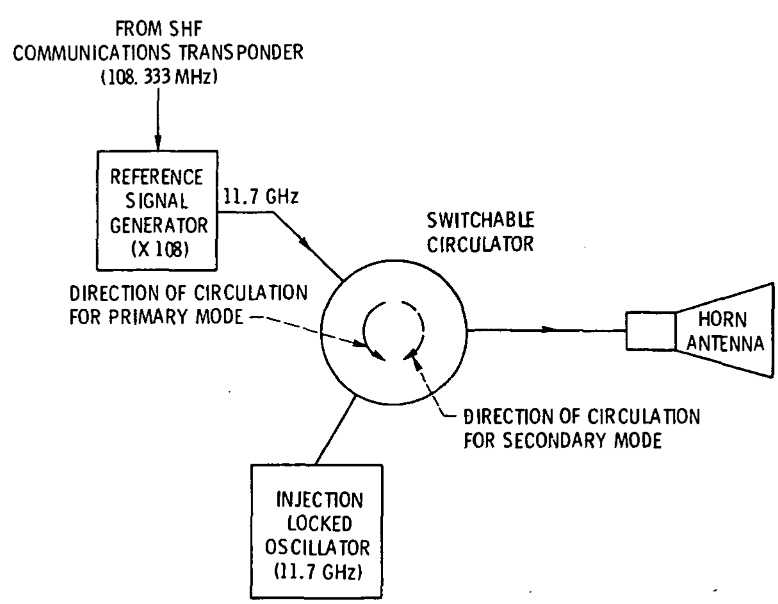


Figure III-14. - SHF Beacon.

- A Switchable Circulator Assembly routes to an antenna the RF signal from the Reference Signal Generator either directly without gain or indirectly with gain by the Injection-Locked Oscillator.
- A Horn Antenna Assembly provides circularly polarized, earth-coverage transmission of the signal.

The Reference Signal Generator contains four separate multiplier stages: two ×2 balanced transistor doublers, one ×3 single-ended transistor tripler, and one ×9 varactor multiplier - together with additional amplifying and filtering stages. A horn antenna is an integral part of the beacon package. The radiated signal is circularly polarized in the right-hand sense, with a beam width of about 16° to the -3-dB points.

TELEMETRY, TRACKING, AND COMMAND SYSTEM

A functional block diagram of the Telemetry, Tracking, and Command (TT&C) System is shown in figure III-15. The system is basically a receiver/transmitter combination. Considerable redundancy is built in to enhance the reliability of the system. The transponder is designed to be compatible with the NASA world-wide Satellite Tracking and Data Network (STDN). Redundant decoders are used to accept commands transmitted by ground control stations; and a partially redundant encoder conditions spacecraft telemetry signals for transmission to the ground by one of two transmitters.

The tracking function complies with GSFC/STDN Unified S-Band (USB) standards, which feature a digital ranging technique to determine ranges for distances greater than 15 000 kilometers (8093 n mi). After range has been established, it is possible to disable the full code modulation (which results in sideband components to 2 MHz from the carrier) and still maintain ranging by continuing to modulate the carrier with only a two-bit clock code. This limits the spectral distribution due to ranging to two single spectral lines (496 kHz above and below the carrier frequency). For distances less than 15 000 kilometers (8093 n mi), a number of different frequency tones are modulated on the uplink carrier. The spacecraft returns the signals on the downlink, and the ranging system determines range by measuring the phase shift of the tones. The higher frequency tones provide system accuracy, and the lower frequencies provide range ambiguity resolution. Frequencies are as follows: 500 kHz, 100 kHz, 20 kHz, 4 kHz, 300 Hz, 160 Hz, 40 Hz, and 10 Hz.

The command function employs a complex modulation scheme consisting of a PCM/PSK/FM/PM format in which pulse code modulation (PCM) phase shift keys (PSK) an oscillator which, in turn, frequency modulates (FM) a subcarrier oscillator and finally phase modulates (PM) a ground transmitter. The TT&C System characteristics are given in table III-7.

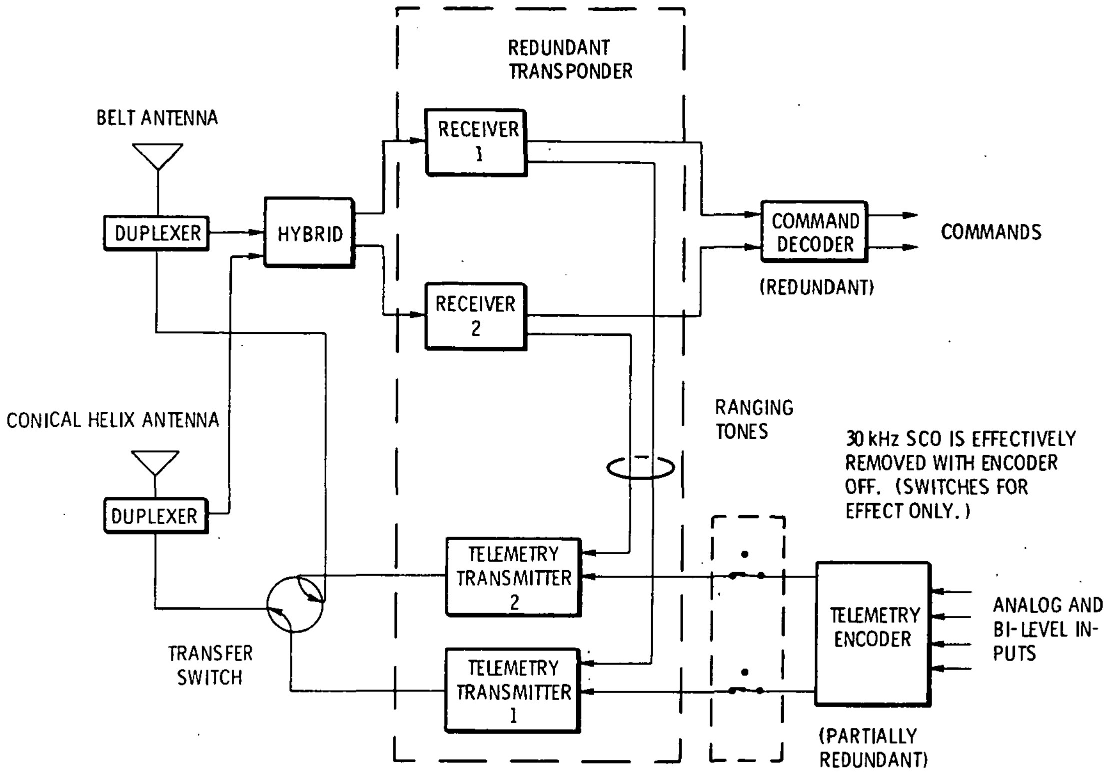


Figure III-15. - Telemetry, Tracking, and Command System (TT&C) block diagram.

Onboard the spacecraft, decoders reduce the serial data stream to discrete commands. The decoder includes a demodulator, a clock pulse generator, several shift registers, a select matrix, buffers, control logic, and supporting power supplies and regulators.

The telemetry function is provided by a PCM/FM/PM System in which pulse code modulation frequency modulates a voltage-controlled oscillator and finally phase modulates a 2-watt transmitter. The PCM/FM/PM system characteristics are given in table III-8. Two encoders are available to process the signals from various sensors into a suitable PCM format for modulation of the transmitter.

The SHF Antenna Subsystem consists of a conical beam antenna mounted on the earth-facing surface of the spacecraft and a belt array mounted around the apogee-motor thrust tube. The belt array will be used primarily during the transfer orbit, while the conical-beam antenna will be used in synchronous orbit to provide additional gain. The operating frequencies are 2277.50 MHz space to ground and 2097.198 MHz ground to space.

Transmitters

There are two transmitters. Either one, or neither, may be on for long periods. Both may be on for periods not to exceed 20 minutes. On command, either transmitter is capable of operating with either receiver. The carrier frequency is derived from a selected receiver when the uplink is on and from a separate oscillator when no uplink signal is present. The modulation consists of the detected uplink signal from the selected receiver and/or telemetry from whichever of the two encoders is on.

The transmitter will provide a minimum of 2.0 watts output over a bandwidth of 20 MHz to the 1-dB points. Figure III-16 is a functional block diagram of the transmitter. Frequency multiplication by a factor of 60 occurs within the transmitter.

Telemetry Encoder

The PCM Encoder Unit is responsible for gathering analog and digital information from various subsystems in the spacecraft by means of time-division multiplexing. It performs analog-to-digital conversion and organizes all the information into a serial bit stream which frequency modulates a Sub-Carrier Oscillator (SCO) which is part of the encoder. The PCM Encoder Unit also contains the Transfer Orbit Electronics (TOE).

TABLE III-7. - CHARACTERISTICS OF TELEMETRY, TRACKING, AND COMMAND SYSTEM

Total number of commands	255
Command bit rate, bits/sec	1000
Command format, bits:	
Synchronization word	14
Address word	7 + 1 decoder
Command word	8 + 1 parity
Value command word	16 + 1 parity
Execute word	8 + 1 parity
Command PSK frequency, kHz	2.0
Subcarrier oscillator frequency, kHz	70

TABLE III-8. - CHARACTERISTICS OF PCM/FM/PM SYSTEM

Type	PCM
Word length, bits	8
Bit rate, bits/sec	1536
Words per frame	192
Overall accuracy, percent	2
Frame time, sec	1
Sampling rate, samples/sec	1
Bilevel measurements (maximum)	240
Analog measurements (maximum)	312
Eight-bit digital measurements	19

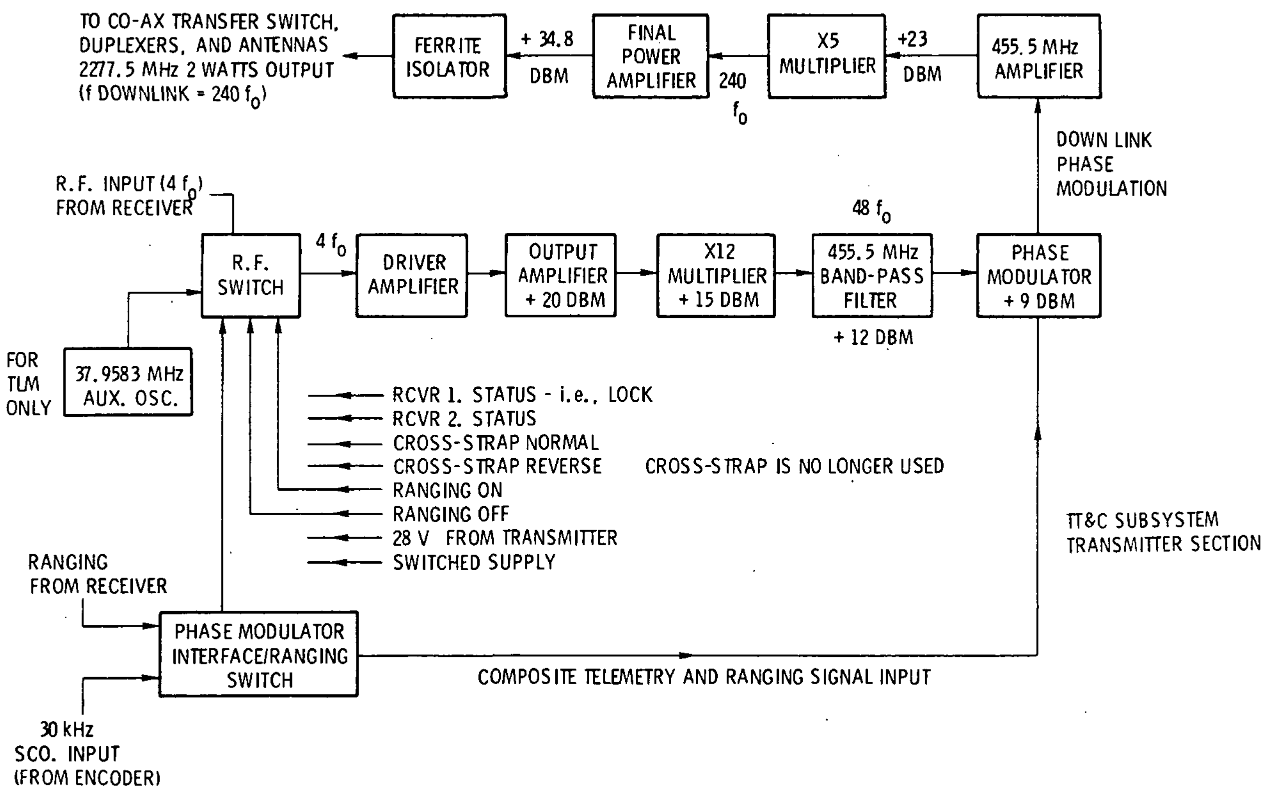


Figure III-16. - Transmitter functional block diagram.

The encoder assembly contains two independent, identical encoders (each including a TOE) used as a redundant pair. Only one encoder will be used at a time; the other will be unpowered (except for a "hold on" voltage on certain sections).

The PCM data stream may further be described as a serial wave train of the two-level Non-Return to Zero (NRZ-S) type. The digital data to be displayed include

- Transfer-Orbit Electronics
- Nonspinning Earth Sensor Assemblies A and B
- Attitude Control Electronics
- Pitch Bias
- Roll Bias
- Momentum Wheel Speed Counter
- Wheel Control Pulse Width
- Noise Filter Flag
- Wheel Torque Bias
- Offset Thruster Pulse Count
- ACEA Subcommutation Address
- Offset Thruster Pulse Width
- Thruster Control Unit
- TCU Subcommutation Address
- Nonspinning Sun Sensor
- Spinning Sun Sensor

In further detail, the encoder functions as follows:

- Analog channels: The telemetry encoder is capable of accepting continuously variable input signals and converting them to binary form.
- Analog-to-digital converter: This subunit converts the analog signals into an 8-bit binary code. The scale of the converter is adjusted to give quantization steps of 20 mV each (i. e., full scale will be 5. 10 V±20 mV). The error in the input level which gives a particular binary code will not deviate by more than 20 mV from the theoretical level.
- Flag inputs (bilevel): The telemetry encoder accepts, from other subsystems, binary signals "or flags" and incorporates them into the PCM format.
- Digital data: The telemetry encoder also accepts, from other subsystems, digital data in the form of trains of binary pulses three words (24 bits) long and incorporates them into the telemetry format.

The telemetry encoder provides these other subsystems with a continuous train of shift clock pulses and a telemetry enabling pulse gate which indicates when the digital data are to be received. The interface requirements are the same as that of the flag inputs.

Transfer-Orbit Electronics

Contained within the PCM Encoder Unit is the Transfer Orbit Electronics (fig. III-17). The TOE digitally encodes the time between occurrence of the pulses which are generated by the sun sensor (spinning mode) and the east and west earth sensors (spinning mode). The TOE also controls the firing of the radial and axial thrusters when operated in the pulsed mode during the spinning phase of the mission.

Measurement of the relative timings of the following pulse data is accomplished by counting by means of a 1536-bit/sec clock during the following intervals:

- The period of rotation of the satellite (i. e., the timing between successive sun pulses)
- The intervals between the leading edge of the sun pulse and the leading and trailing edges of the east earth sensor
- The intervals between the leading edge of the sun pulse and the trailing edges of the west earth sensor (These intervals are encoded during one revolution of the spacecraft and transmitted to the ground sequentially. These data are provided to the attitude determination software.)

The thruster firing output section of the TOE is used to generate pulse outputs for driving the radial and axial thrusters. The leading edge of these pulses is referenced to the leading edge of the sun sensor pulse. The time delay is transmitted from the ground as a 9-bit word which is part of a 16-bit value word. A further 5 bits of this word indicate the number of pulses required. Duration of the pulses is nominally 135 msec. An additional bit of the value word indicates whether the pulse will go to the radial thruster or to the axial thruster.

The TOE telemetry and command functions are shown in figure III-18.

Receivers

Two solid-state receivers (a redundant pair) function as a 2-GHz Phaselock Loop Command Receiver Subsystem (PLCRS). The command receiver tracks the incoming signal and demodulates the PSK/PCM data stream contained on a subcarrier that phase modulates the uplink carrier. Together with the telemetry transmitter, the command receiver forms a ranging transponder. The ranging signal, which phase modulates the uplink carrier, is demodulated in a phase detector to baseband and then fed to the modulator input of the S-band telemetry transmitter.

Each receiver is a superheterodyne and employs a phaselock carrier tracking loop, a separate wideband (modulation) phase detector, and a coherent AGC detector. In operation, the receiver is phase locked by sweeping the unmodulated ground transmitter frequency across a portion of the uplink frequency assignment. When phaselock is ac-

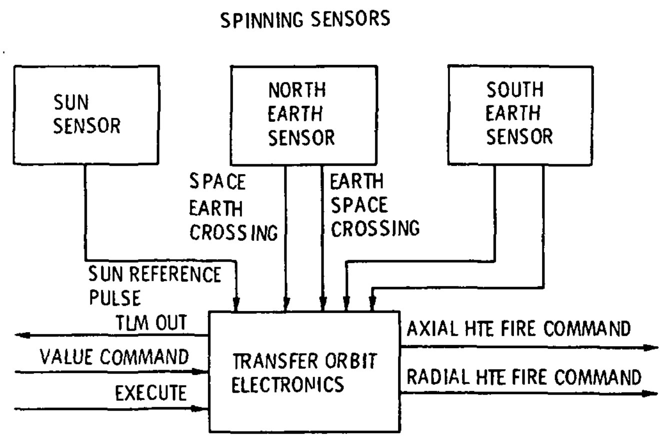
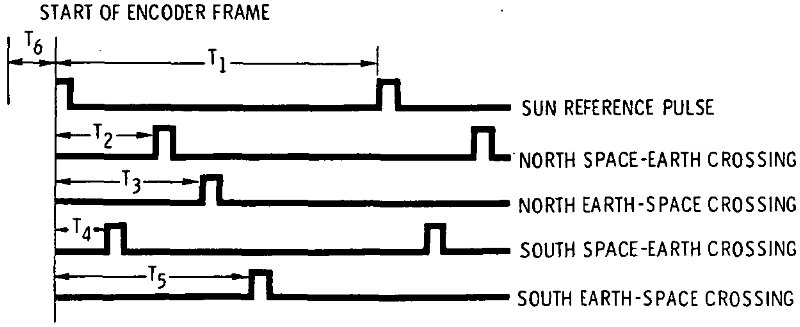


Figure III-17. - Transfer Orbit Electronics (TOE) functional block diagram.

TELEMETRY FUNCTION

PROVIDES ON-BOARD PROCESSING OF ANALOG EARTH AND SUN SENSOR PULSES; TIMES, ENCODES, AND TELEMETERS THE FOLLOWING TIME INTERVALS



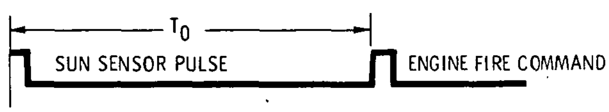
INFORMATION IS ENCODED AND STORED DURING ONE REVOLUTION AND SEQUENTIALLY READ OUT DURING NEXT 5 TELEMETRY FRAMES EACH FRAME IS FORMATED AS SHOWN BELOW.

YES/NO	1 MSB	2	3	4	5	6	7
8	9	10	11	12	13	14	15
1	2	3	1	2	3	4	5
IDENTIFIER			T_6				

COMMAND FUNCTION

ON-BOARD PROCESSING IS REQUIRED TO FIRE THE HIGH THRUST ENGINES IN A PULSE MODE TO PRECESS THE SPACECRAFT FOR ATTITUDE REORIENTATION.

ENGINE FIRING IS REFERENCED TO THE OCCURRENCE OF A SUN SENSOR PULSE.



T_0, WHICH IS DERIVED FROM A KNOWLEDGE OF SPACECRAFT SPIN RATE AND THE DESIRED PRECESSION AXIS, IS LOADED INTO THE TRANSFER ORBIT ELECTRONICS BY A VALUE COMMAND.

THE VALUE COMMAND HAS THE FOLLOWING FORMAT:

MSB	"	"	"	"	"	"	"	LSB	MSB	LSB		
TIME INTERVAL									NO. OF PULSES REQUIRED		AXIAL OR RADIAL SELECT	NOT USED

Figure III-18. - Transfer Orbit Electronics (TOE) telemetry and command functions.

quired, the ground transmitter frequency will be set to the nominal center frequency of the transponder and the transponder will not lose phaselock when subjected to the normal Doppler frequency offsets. When locked to the incoming signal, the receiver will furnish

- The demodulated command signal to the command decoder
- A reference frequency, coherent with the receiver carrier, to drive either transmitter
- The demodulated uplink (ranging and command) signal to the transmitted modulators
- "In-lock" indications to the transmitters
- Housekeeping telemetry signals to the telemetry subsystem
- Signal strength (AGC voltage)
- Phase error in the phaselock loop

The squelch setting of the receiver is -109 dBm or lower. The receiver will perform normally with signal levels to -73 dBm. The initial noise figure is less than 8 dB and may degrade 2 dB throughout its life. Front-end selectivity for 0.25-dB bandwidth is 25 MHz minimum and 100 MHz maximum for 60 dB down. Operating frequency is 2097.198 MHz. The receivers will drop lock at -132 dBw. A receiver functional block diagram is given in figure III-19.

The design Bit Error Rate (BER) of 10^{-4} will give, on the average, six error-free 1-second telemetry frames received from seven transmitted. Thus, an observer would have to wait only 2 or 3 seconds to ensure that any one parameter of interest had not been equivocated.

The expected telemetry signal margin at Ottawa, Canada, is 12 dB. For STDN stations with 30-foot aperture antennas, the margin is 16 dB, assuming cryogenic parametric amplifiers.

Decoders

The Decoders are part of a Pulse Code Modulation (PCM) Command System compatible with the "Aerospace Data Systems Standards of NASA" (GSFC Document X-500-63-2, Part 2, Section 3). They accept the phase-shift-keyed (PSK) subcarrier from the command receivers, demodulate the subcarrier, decode the binary information obtained, and transform the pulses into unique command signals for the spacecraft.

The main components of the Decoders are a demodulator, a reset pulse generator, a command shift register, a main shift register, a value shift register, control logic, a matrix circuit, value and verify buffers, various power supplies, regulators, timers, power switches, and Electro Magnetic Interference (EMI) filters. A functional block diagram of the decoders is shown in figure III-20.

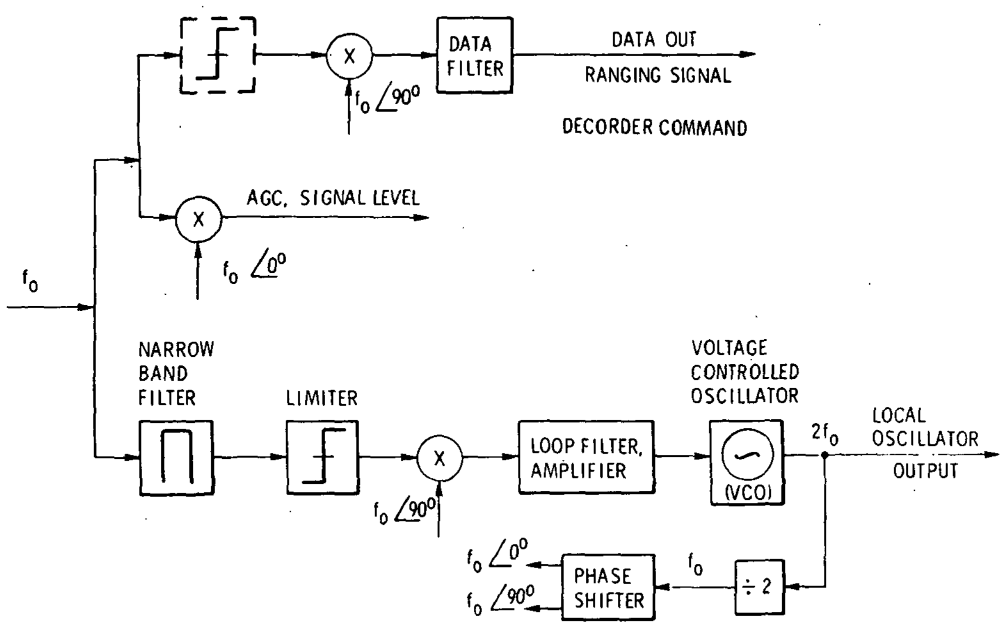


Figure III-19. - Receiver functional block diagram.

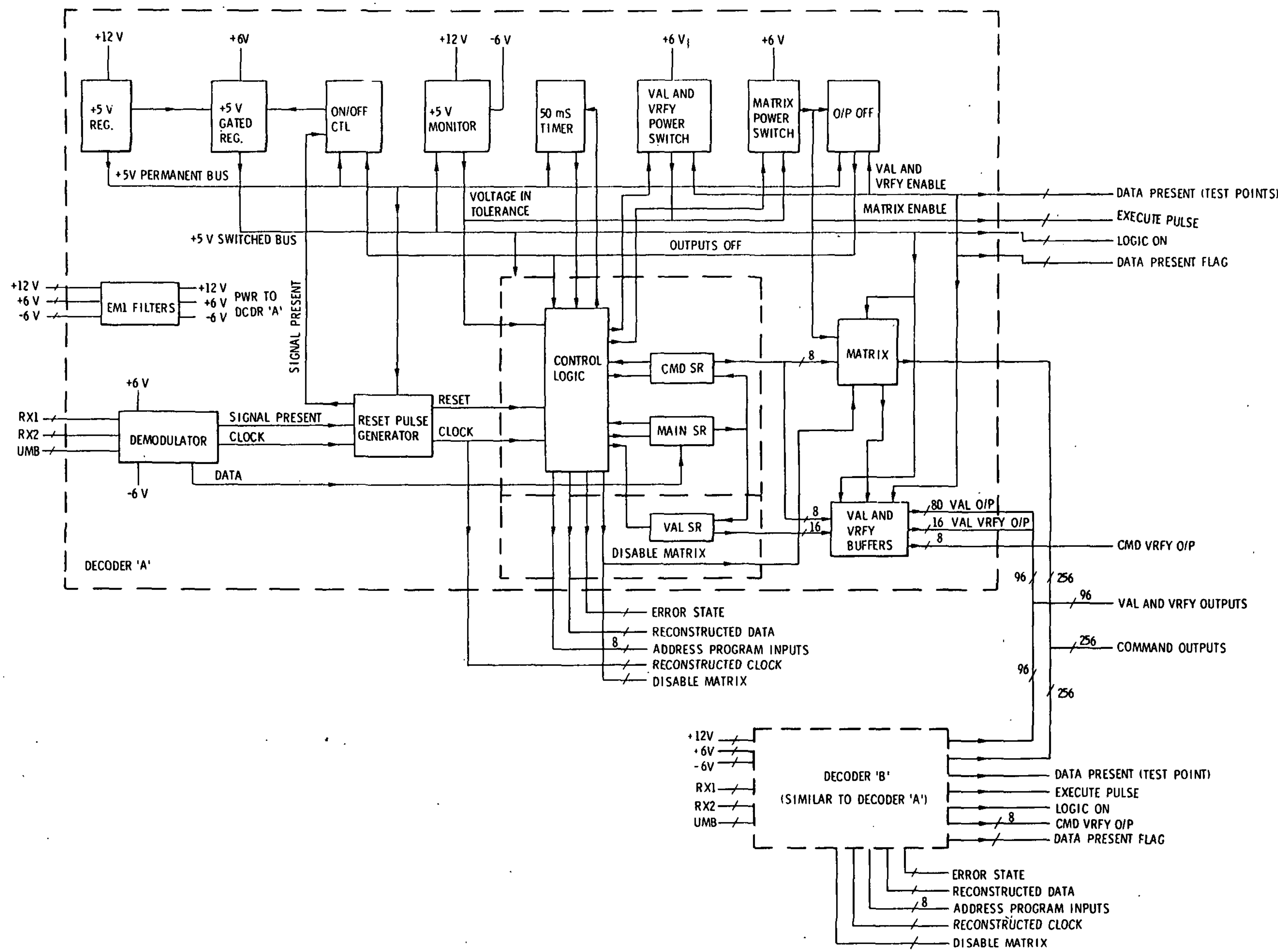


Figure III-20. - Decoder functional block diagram.

The basic command consists of a binary pulse train with a Pulse Code Modulation Non-Return to Zero (PCM-NRZ) format. The pulse code sequence is phase-shift-keyed onto a 2-kHz subcarrier oscillator (fig. III-21), the phases of the "zero" and "one" states being 180° apart. A sinusoidal signal of the same frequency as the bit rate is also transmitted to assist in identifying the bit transition in the PCM train. Since the bit rate is 1 kHz, exactly half the subcarrier frequency, the phase of the bit-rate sinusoid may be used to identify the reference or "zero" phase of the 2-kHz subcarrier.

The phase of the sine wave will remain constant with respect to the bit train but may be adjusted in 1° increments over a range of $\pm 25^{\circ}$. Zero phase occurs when a positive-slope zero crossing of the sinusoid coincides with a bit transition. The composite PSK/PCM signal will be transmitted as frequency modulation on a 70-kHz subcarrier, which subcarrier will be demodulated in the command receiver. The command decoder will receive only the 2-kHz phase-shift-keyed subcarrier and the 1-kHz bit-rate sinusoid.

The two command decoders are identical and are capable of accepting an 8-bit code and decoding it to activate one of 255 outputs. Corresponding commands from each decoder are combined in pairs before the output from the combined decoders is formed. Each decoder is capable of storing a value in the form of a 16-bit binary number. The corresponding bits from each decoder are split into five groups of 16 parallel lines and are combined in pairs for setting logic stores within the subsystems. The decoder is so designed that a single failure will not cause a continuous command or inhibit any command output from the other decoder.

Of the 255 commands available, eight are "value" commands and the remaining 247 are "simple" commands. Simple commands are those for which the selected command decoder will issue a single 50-msec execute pulse from the specified output channel immediately following receipt of the execute frame of the command message. Value commands contain a 16-bit word in the command message, which is transferred by the command decoder to the specified register of a subsystem upon receipt of the execute frame. Value commands are identified as such by a 5-bit code in the command word. The remaining 3 bits specify the register to which the 16 value bits are to be routed. The 3-bit address limits the total number of value commands to 8, or 2^3.

By the inclusion of a succession of execute frames in the command message, the duration of the execute pulse of a command may be extended in increments of approximately 17 msec. Repetitive execute pulses may be obtained from a single command (simple or value) by the introduction of additional execute frames separated by "post-message sync" intervals.

Any command may be verified at the option of the satellite controller. The decoders provide outputs to the telemetry encoder to enable the individual bits of the command word and of the command value frame of a command message to be transmitted to ground for verification before the execute frame of the command message is transmitted to the

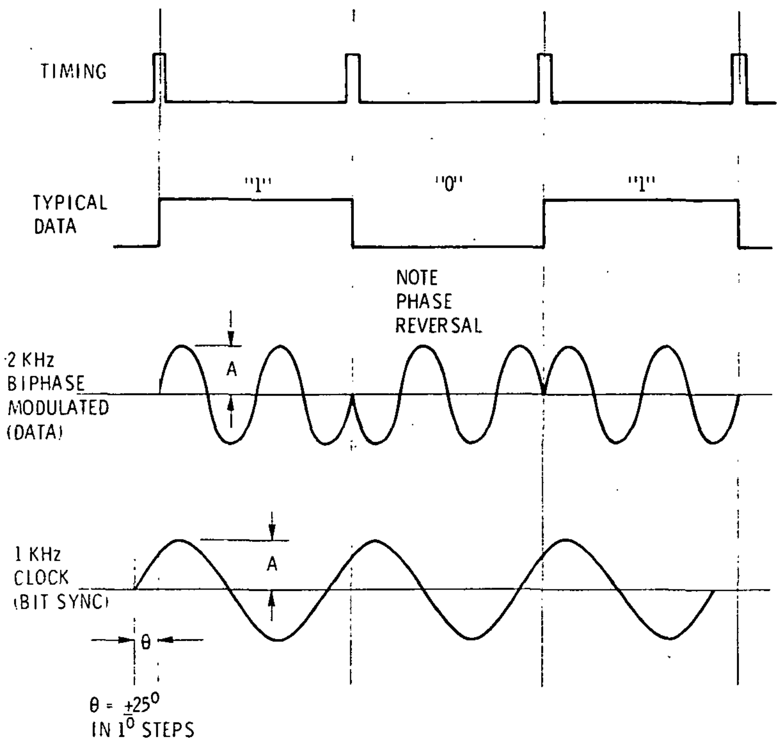


Figure III-21. - Pulse code sequence.

spacecraft. Postmessage sync intervals, in multiples of 17 bits, must be inserted into the command message to keep the command decoder active while the verification is taking place.

Command Format

The format is shown diagramatically in figure III-22. The frequency of the subcarrier oscillator is 70 kHz, the frequency of the PSK oscillator is 2 kHz, and the bit rate is 1000 bits per second. A description of the format follows:

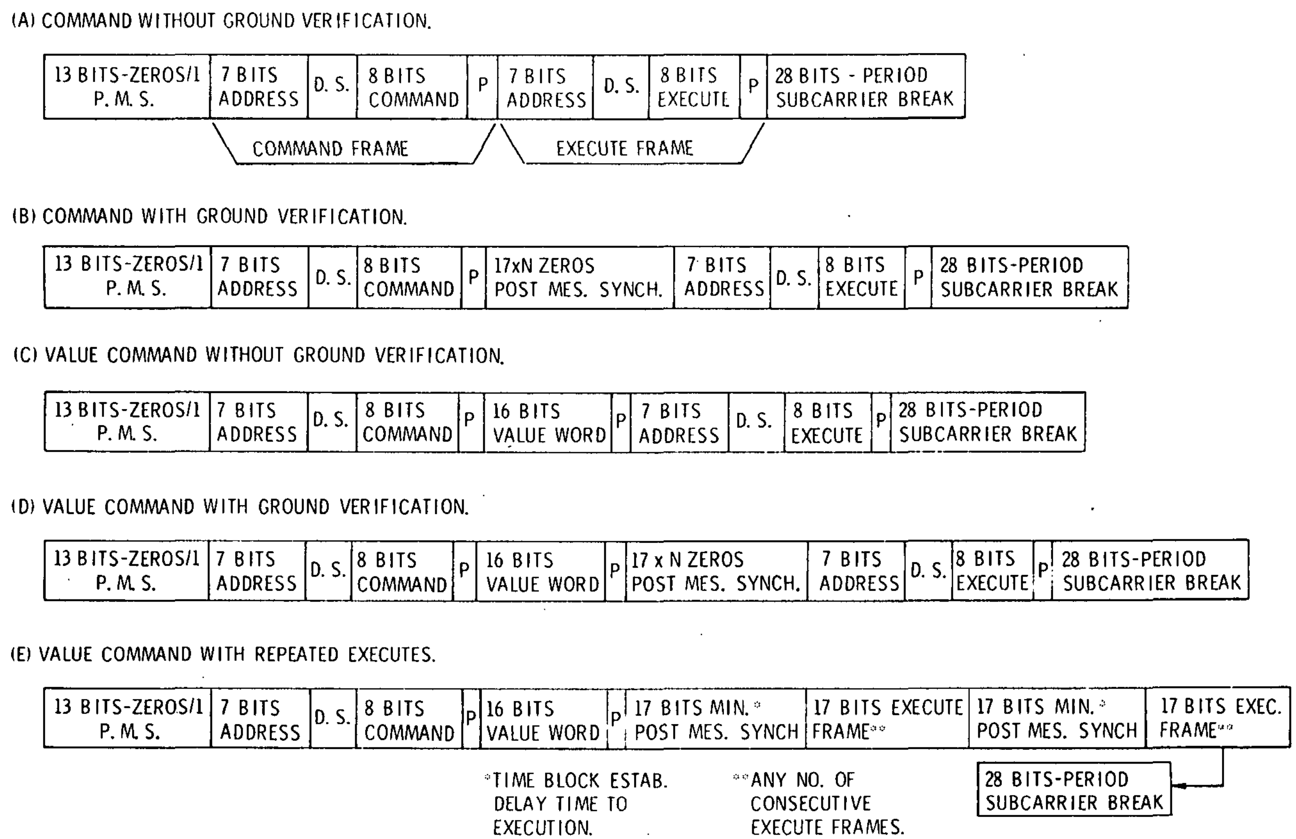
- Bit synchronization signal (bit clock tone): The 70-kHz subcarrier oscillator is frequency modulated with a sinusoid of 1 kHz combined with a 2-kHz biphase modulated wave. The depth modulation is 50 percent±5 percent.

The relative phase of this sinusoid is fixed with respect to the bit transition time. Reference phase is when a positive-going zero crossing of the sinusoid coincides with a bit transition. This phase will be variable on the ground.

- Premessage synchronization: Every command message will be preceded by a premessage synchronization code consisting of 13 "zeros" followed by one "one." The purpose of this code is to allow transients to settle in the bit-rate filter and to indicate uniquely the beginning of a message.

Until the beginning of the premessage synchronization code, the decoder is in a quiescent state, drawing minimum power.

- Command frame: The command frame immediately follows the premessage synchronization code. This frame consists of 17 bits and is divided into four parts:
    - Address: The address is a 7-bit word which is specific to the spacecraft. The address must be recognized by the decoder before the decoding process is allowed to continue.
    - Decoder selector bit: The bit immediately following the address is used for selecting which of the two decoders is to provide the command output from its decoding matrix. The same command can be sent to either decoder by altering this one bit. Thus, the same command cannot be transmitted simultaneously to both decoders from a single ground source.
    - Command word: The command word is an 8-bit binary number representing the channel to be activated. The decoder stores the command word, which is decoded by using a logic matrix which activates one of the 255 output channels.
    - Parity check: The command word will be immediately followed by a single bit giving the "one parity" of the previous 8 bits. A parity check is taken by the decoder and must prove correct before the decoding process is allowed to proceed.



NOTES: P. M. S. = PREMESSAGE SYNCHRONIZATION (14 BITS); N = INTEGER; P = PARITY (1 BIT); D. S. = DECODER SELECT (1 BIT)

Figure III-22. - Command format.

- Value word: A standard frame of 17 bits, called the value word, immediately follows certain types of command frames and represents a 16-bit number which is transferred into a register or stored in another subsystem. The 17th bit is a "one parity" digit for the other 16 bits. The decoder is capable of storing this number for eventual transfer in parallel form to a register designated by the previous command frame.

  The command decoders provide outputs to the telemetry encoder to enable the 8 bits of the command word and the 16 bits of the value frame to be transmitted to ground by the telemetry system for verification of the spacecraft command encoder status before an execute frame is sent. A single output carrying logic-level signals is provided to indicate "data present" whenever a successful parity check has been carried out on a command word, the command is in the decoder store, and the ground verification outputs are energized.
- Postmessage synchronization: After a command frame (and value if required) have been transmitted and before the execute frame is transmitted, the decoder is kept active by a continuous train of "zeros" modulated with the bit-rate sinusoid. In the event of execute without ground verification, there will be no postmessage synchronization. The execute frame follows immediately on the command or value frame without a break.
- Execute frame: The execute frame is the final frame of the message and consists of the spacecraft address followed by an execute command word which is uniquely identified by the decoder. On recognition of the execute command word, the decode matrix is energized and the command channel already specified by the command word is activated for 50 msec. After transmission of the execute frame, there will normally be a 28-bit (minimum) message interval which enables the decoder to return to the quiescent state. If required, however, a succession of execute frames may be sent. Since the interval between execute frames is less than 50-msec duration of the channel activation pulse, a continuous execute signal is available.
- Message interval: The minimum interval between successive messages is 28 bits, during which there is no subcarrier.

When there is no subcarrier present, the decoder is in a quiescent state with power applied only to the filter circuits. The logic circuits are switched on by detection of the subcarrier. Further power is conserved by only switching power to the decode matrix during the execute pulse. A message interval will cause the decoder to return to the quiescent state.

Beam Antenna

The beam antenna assembly (fig. III-23) is made up of two separate antennas: a dish for the command function, and a microstrip patch for the telemetry function. Radiation is right-hand circular. The dish for receiving commands is 22.9 cm (9 in.) in diameter and 1.9 cm (0.75 in.) deep. The antenna patch is mounted on a 7.0-cm by 10.3-cm (2.75-in. by 4.06-in.) plate. The combined patterns of these antennas, together with that of the belt array, whose description follows, are shown in figure III-24.

Belt Antenna

The belt antenna used for command and telemetry consists of interlaced circular arrays of microstrip radiators (fig. III-25). There are a total of 128 elements: 64 elements (or facets) for command, and 64 for telemetry. The power for both systems is distributed to the elements equally by a system of "T" branches in the coaxial lines and in the strip lines to the radiating elements. Power is halved in each branch of the T's.

The belt antenna surrounds the thrust tube around the apogee motor. It is 117 cm (46 in.) in diameter. Each ring is 12.7 cm (5 in.) wide and is attached to the spacecraft by clamps around the spacecraft struts. A schematic of the TT&C antenna system is given in figure III-26.

DEPLOYABLE SOLAR ARRAY

The Deployable Solar Array (DSA) provides power for all spacecraft loads after the spacecraft is despun and the solar array is deployed in synchronous orbit. The solar array subsystem consists of the following major assemblies (fig. III-27):

- Flexible Solar Array
- Stowage and Support Structure
- Solar Array Mechanical Assembly (SAMA)
- Deployment and Tensioning Mechanism
- Drive and Tracking Mechanism and Array Orientation Electronics
- Array Instrumentation

The DSA is stowed during launch, transfer orbit, and synchronous drift orbit (spin-stabilized mode) in a flat, folded configuration between two honeycomb panels. Interleaf material is inserted between folds of the solar array blanket to protect the cells and the interconnects while the DSA is in the stowed position. Prior to deployment of the flexible array, the north and south Jettisonable Body Solar Arrays (JBSA) are jettisoned

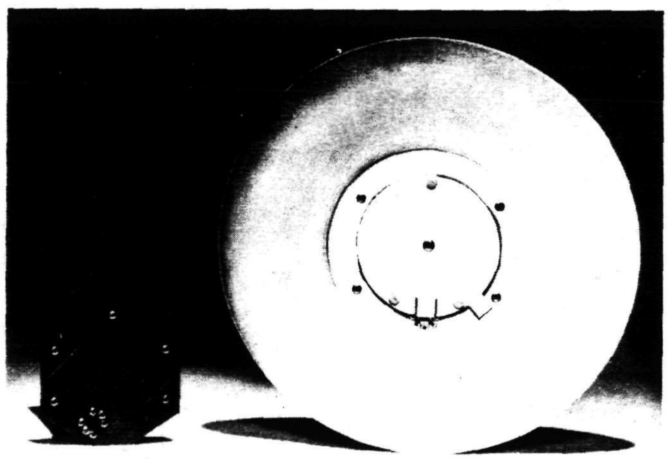


Figure III-23. - Telemetry, Tracking, and Command System (TT&C) beam antenna.

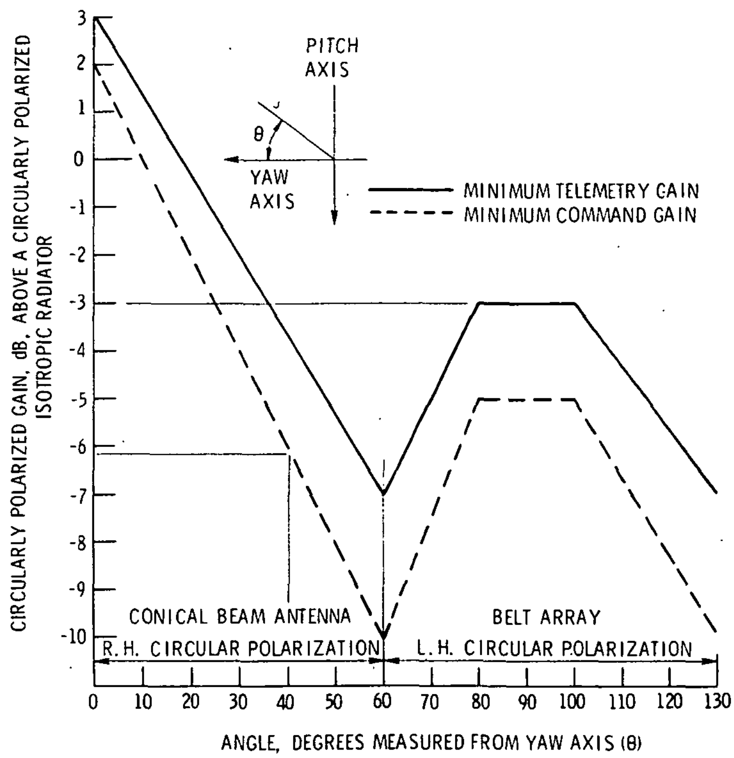


Figure III-24. - Telemetry, Tracking, and Command System combined antenna patterns.

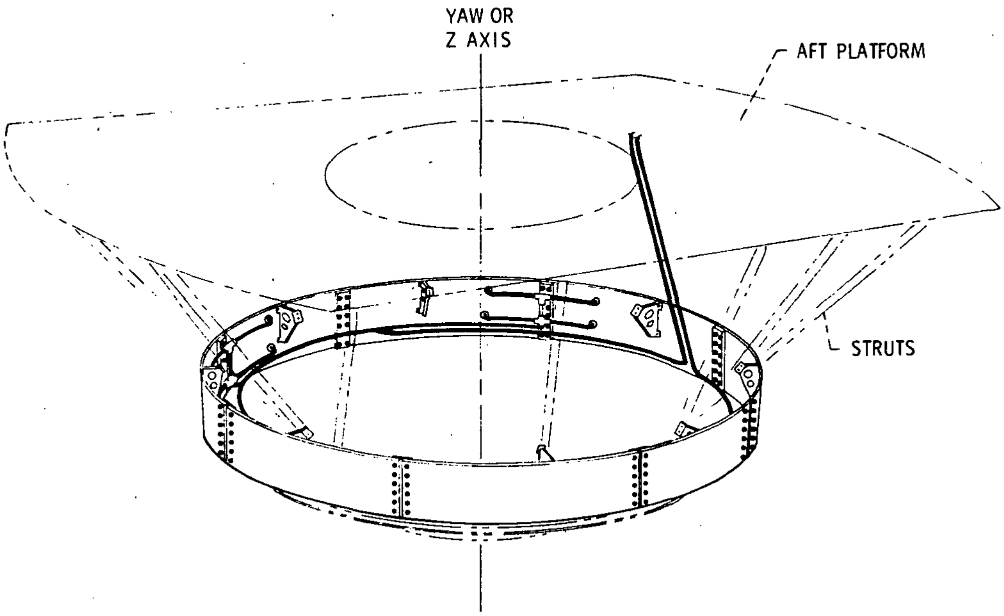


Figure III-25. - Belt antenna array configuration.

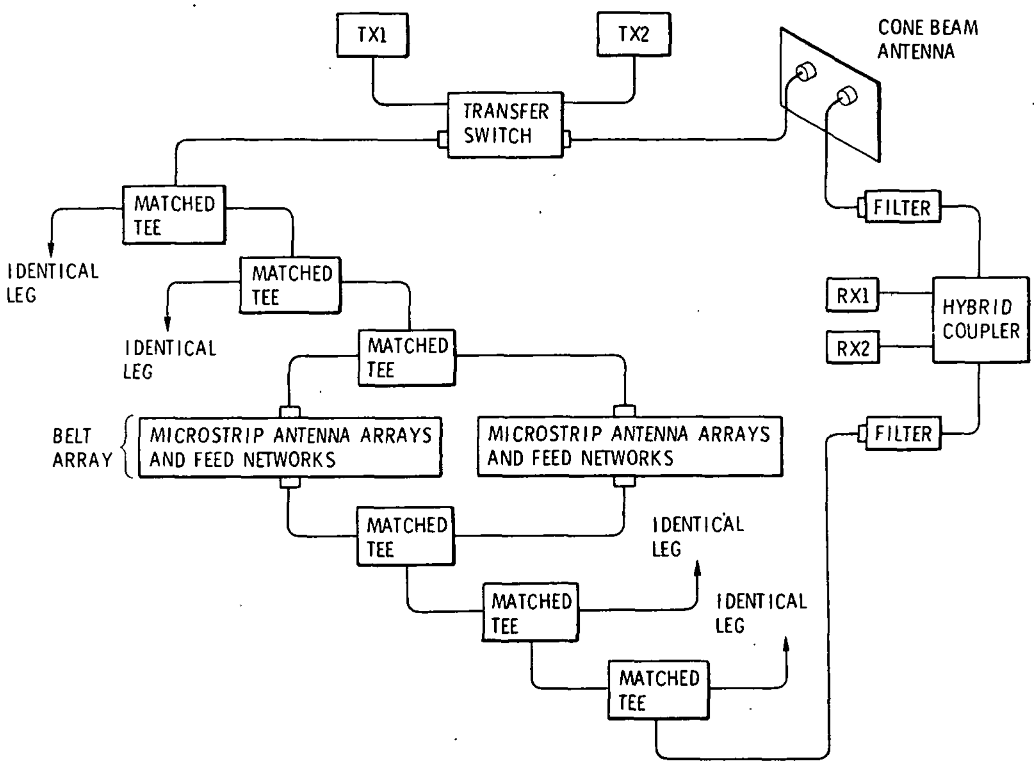


Figure III-26. - Telemetry, Tracking, and Command System antenna schematic.

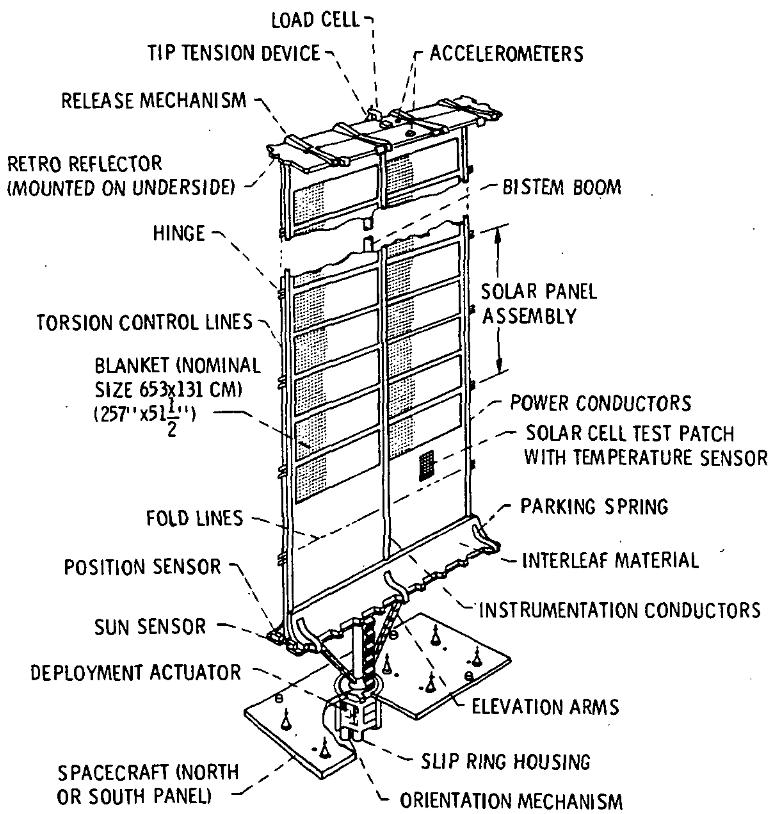


Figure III-27. - Deployable solar array.

by pyrotechnic actuators and a mechanical linkage which releases the flexible arrays for deployment. The deployment actuator is energized and deployment begins after the stowed array is elevated a distance of 76.2 cm (30 in.) by means of the BISTEM boom. Array foldout continues at a nominal rate of 2.5 cm (1 in.) per second. At full extension of the array, the tensioning mechanism at the tip of the boom is activated to provide a nominal tension of 3.63 to 4.54 kg (8 to 10 lbm) and automatically switches off the deployment sequence. Overall characteristics of the DSA are provided in table III-9.

TABLE III-9. - CHARACTERISTICS OF DEPLOYABLE SOLAR ARRAY

Configuration	2 flat blankets each 130 cm by 653 cm (51 in. by 257 in.)
Total number of cells	25 272
Number of cells in series:	
Housekeeping	81
Experiments	162
Number of cells in parallel:	
Housekeeping	60
Experiments	126
Operating temperature range, $^{\circ}C$ ($^{\circ}F$)	45 - 55 (113 - 131)
Number of panels:	
Active	52
Blank	8
Boom length, m (ft)	7.37 (24)
Boom diameter, cm (in.)	3.4 (1.34)
Solar cells:	
Thickness, cm (mil)	0.020 (8)
Area, cm (in.)	2 by 2 (0.79 by 0.79)
Coverglass thickness, cm (mil)	0.010 (4)
Power output, W:	
Beginning of life	1257
End of life	919

Flexible Solar Array Blanket

The solar array is composed of two identical blankets. Each blanket has the following components:

- Flexible substrate
- Solar-cell modules
- Flat cable conductors

The blankets are subdivided into eight Solar Panel Assemblies (SPA), which are hinged to permit stowing for launch. The substrate is a laminate of 25-μm (0.001-in.) thick Kapton and 35-μm (0.0014-in.) thick glass fiber bonded together by means of a polyester adhesive. The total thickness of the substrate is 65 μm (0.0026 in.).

The basic power generating elements of the solar array are the connector integrated solar cells, which consist of the following:

- A 1-ohm cell, 200 μm (0.0079 in.) thick and 2 cm by 2 cm (0.79 in. by 0.79 in.)
- Cerium-doped coverglass, 100 μm (0.0039 in.) thick
- Silver mesh interconnector welded to the cell n-contact
- Coverglass adhesive to bond the coverglass to the cell

The 3-by-9 assembly of connector integrated cells forms a solar-cell module. The strings of the experiments section are made by a series connection of 18 modules, and the strings of the housekeeping section are made by a series connection of nine modules.

The electrical block diagram of one solar array blanket is shown in figure III-28 and is composed of eight solar panel assemblies. Each blanket is divided into two sections: one for experiment power, and the other for housekeeping power. The experiments section is a parallel connection of a group of strings arranged physically in a 2-by-21 pattern, each string consisting of cells in a 3-by-162 pattern. The housekeeping section is a parallel connection of strings in a 2-by-10 pattern, each string consisting of cells in a 3-by-81 pattern.

Wiring

The wiring system consists of three different types of flat cable conductors: the bus and common ground wiring for transfer of the array power to the spacecraft, and the instrumentation wiring for different sensors on the blanket and the deployment system. The integration of the wiring system by means of the hinge joints of adjacent solar panel assemblies is made by special types of bridge pieces soldered to the wiring. The electrical connection of the solar array to the Solar Array Mechanical Assembly (SAMA) is made through specially designed terminal boards connected to solar panel assemblies.

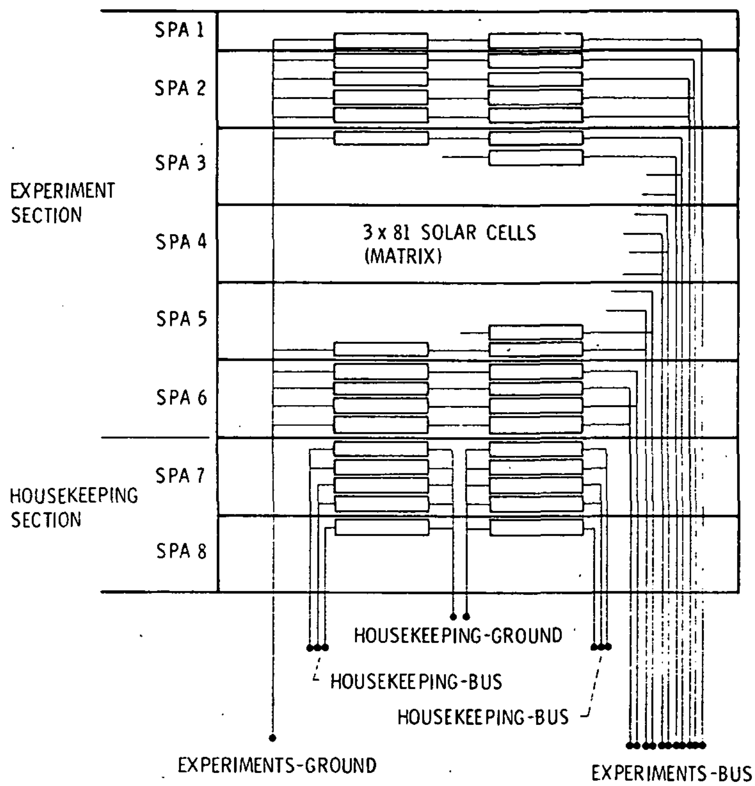


Figure III-28. - Electrical block diagram of one solar array blanket.

Power

The total output power of the two blankets required for the experiments and housekeeping sections over the mission life of 2 years are

- Experiments, 1016 watts beginning of life (BOL) and 743 watts end of life (EOL)
- Housekeeping, 241 watts beginning of life (BOL) and 176 watts end of life (EOL)

The power/weight ratio of the flexible array blanket is specified to be not less than 77.2 watts/kg (35 watts/lbm) at 60° C after 2 years in synchronous orbit.

Solar Array Mechanical Assembly

Deployment and orientation of the solar array are performed by the Solar Array Mechanical Assembly (SAMA). Each sail is independently driven by a single axis drive and tracking mechanism consisting of a stepping motor and gear train. Rotation is bidirectional and about the pitch axis of the spacecraft. A sun sensor on each sail can be used to control the drive mechanism. For normal tracking, the rotation rate is 1 revolution per day. During initial solar acquisition and also during reacquisition, fast slew rates of up to 30° per minute are provided. The SAMA consists of the following major units:

- Deployment Actuator
- Drive and Tracking Mechanism
- Array Instrumentation

A functional block diagram of the array control is shown in figure III-29. The deployment actuator uses a BISTEM boom which is 3.4 cm (1.34 in.) in diameter, is made of 0.17-mm (0.007-in.) thick stainless steel, and is silver plated on the outside. The actuator mechanism consists of twin spools around which the BISTEM element is stored. The spools are supported on shafts located in a lightweight rigid housing. The spools are geared together and driven by a dc motor while rollers around the spools contain and guide the deploying element. A rotary potentiometer telemeters rotation of the spools.

The drive and tracking mechanism performs the two basic tasks of acquiring and tracking the sun. Subsequent to initial deployment and following an eclipse of the sun or loss of track, a nominal fast slew mode is required for sun acquisition. After acquiring the sun, the control system will track the sun at 1 revolution per day, maintaining the solar array normal to the sun to within $\pm 1.0^{\circ}$.

Each array has its own identical but electrically separate control system so that each array can be controlled individually or as a unit (one slaved to the other). There are three operational modes of control available:

- A closed-loop control using sun sensor pointing as a detector

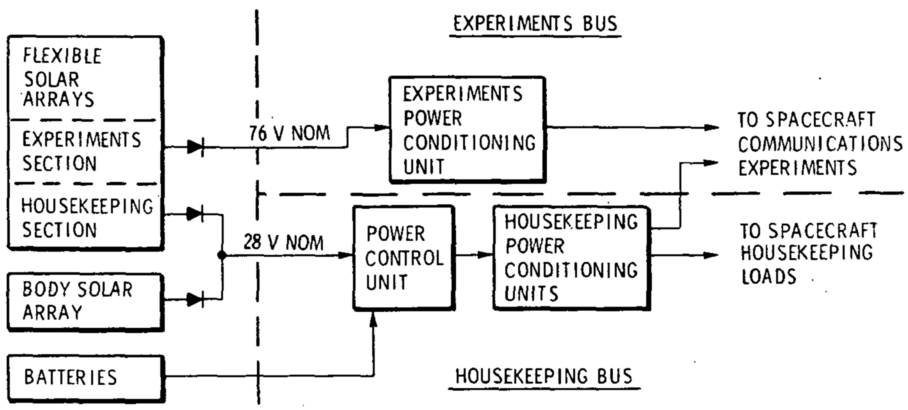


Figure III-29. - Array control functional diagram.

- An open-loop ground command control
- An open-loop spacecraft-synchronous control

In addition, a shaft encoder is provided for each array in order to monitor array orientation and facilitate ground control operation. The shaft encoders give a reading of array orientation relative to the spacecraft with a resolution better than 0.125° and read out directly to telemetry.

A slip ring assembly is used for power transfer. The unit is a gold-on-gold wet lubricated unit with 46 ring and brush circuits. Each power ring and brush circuit is designed to carry 2.0 A, and telemetry circuit rings are designed for 100 mA.

ELECTRICAL POWER SYSTEM

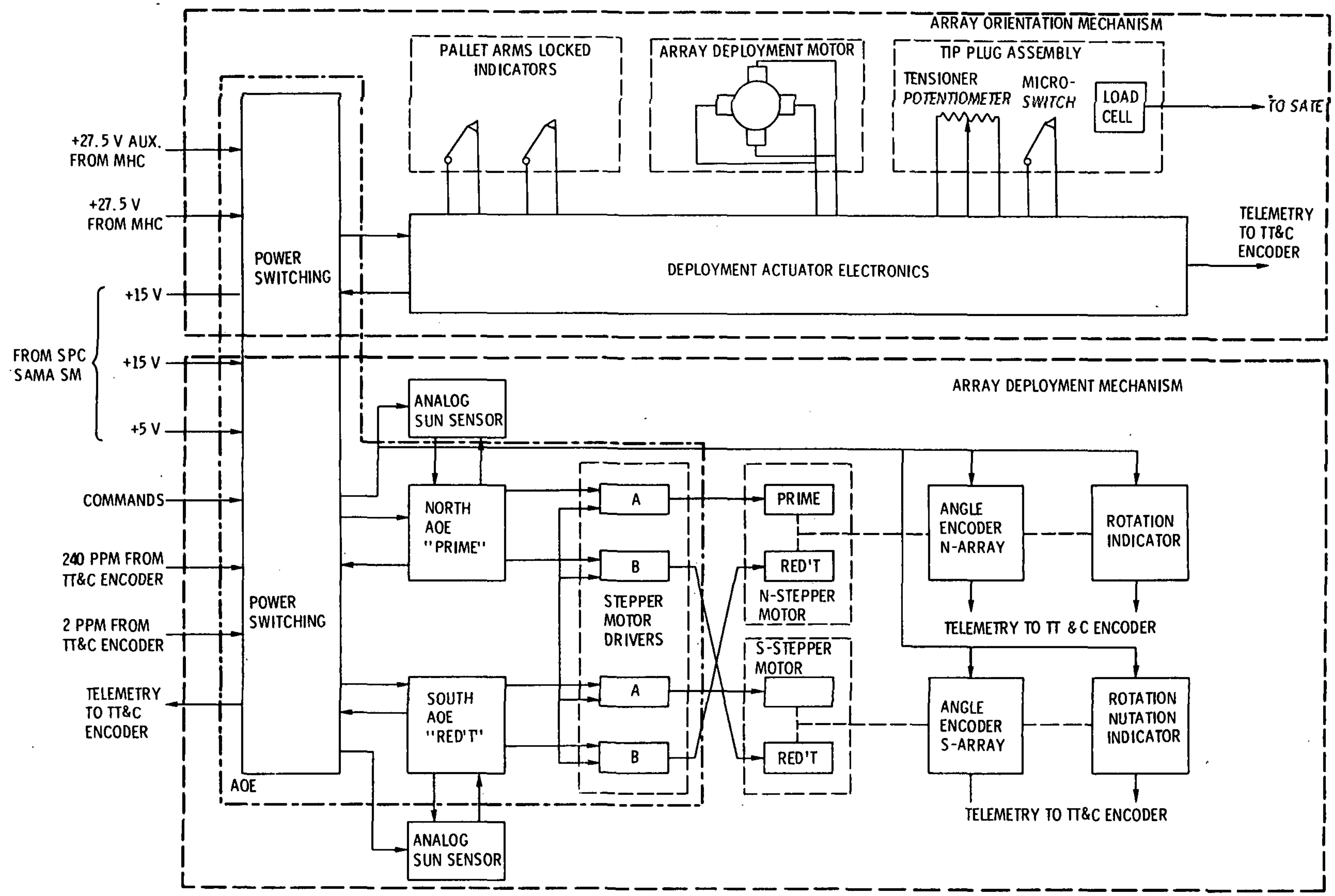
The Body Solar Array (BSA) is the primary source of electrical power from the time of spinup of the Delta third stage until the spacecraft is despun to zero rpm during acquisition. Once the spacecraft is on station in synchronous orbit, however, the primary power source is the Deployable Solar Array (DSA). Electrical power from these arrays is supplied to the spacecraft via two separate busses:

- The low-voltage housekeeping bus receives its power from either the Body Solar Array or the housekeeping section of the DSA and also from batteries. It provides power to the housekeeping subsystem loads and to spacecraft communications experiment loads.
- The high-voltage experiments bus receives its power from the experiment section of the DSA only. It furnishes power only to the communications experiments during sunlit synchronous orbit operation of the spacecraft.

These busses are shown in the simplified block diagram of the Electrical Power System (fig. III-30).

Body Solar Array

The Body Solar Array includes six panels of solar cells mounted circumferentially about the periphery of the spacecraft (fig. III-31). The four east-west panels are hard mounted to the cylindrical portion of the vehicle structure and remain there throughout the mission although they are used only until the much larger DSA is deployed. The Jettisonable Body Solar Array (JBSA) panels, located on the flattened north and south portions of the cylindrical structure, form the enclosure for the extendible DSA. The JBSA panels are, at the appropriate time, jettisoned when they are no longer needed during the three-axis acquisition sequence. This clears the way for extending the booms for DSA deployment.



NOTE: ARRAY DEPLOYMENT MECHANISM SHOWN FOR ONE ARRAY ONLY.

Figure III-30. - Electrical Power System block diagram.

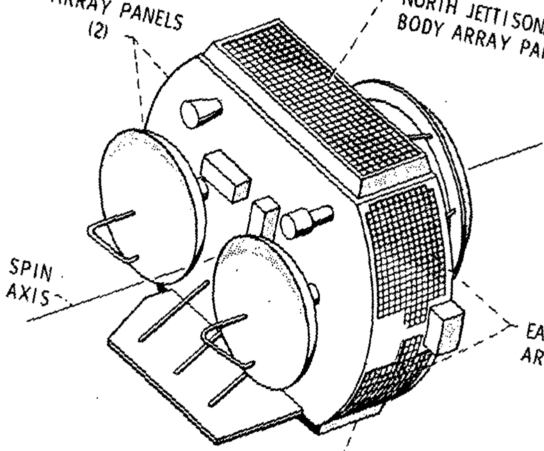


Figure III-31. - Location of Body Solar Array (BSA) panels (six panels total).

The 2-cm by 2-cm (0.79-in. by 0.79-in.) solar cells on the Body Solar Array panels are high-efficiency (12.5 percent), shallow junction cells and are 200 μm (8 mils) thick. Approximately 5460 cells in the six array panels provide an average power of 90 watts at 30 volts at equinox with the spin vector perpendicular to the sun line and 86 watts at summer solstice with the spin vector 65° to the sun line at end of life (a period of approximately 1 month).

Batteries

Batteries are needed to provide secondary power when the solar arrays are inoperative because of shadowing and also to supplement array power during peak power demands. In a nominal mission, the spacecraft draws battery power

- From 5 minutes before launch to 26 minutes after lift-off
- From the time of despin to zero rpm until the flexible solar arrays are deployed and sun oriented
- During all eclipse periods
- In a sharing mode with either array during peak power demands
- During a nonstandard mission condition when the arrays are unable to furnish power

Two batteries are provided, each consisting of twenty-four 5-A-hr nickel-cadmium cells. During synchronous orbit eclipse periods, the maximum depth of discharge is 60 percent. Critical loads are kept to within the capability of a single 5-A-hr battery in event of failure of the other. The batteries are charged at a C/10 rate until a specified voltage is achieved, after which the rate is reduced to C/20 until full recharge is achieved. Means are provided to periodically recondition the batteries by discharging to a specified level at a predetermined rate. The start and finish of the recondition discharge and the battery to be discharged are selected by ground command.

Power Control Unit

The Power Control Unit (PCU) controls the power flow from the power sources to the Housekeeping Power Conditioning Units. The block diagram of figure III-32 shows an array overvoltage controller which automatically switches off the PCU input if the array voltage rises above 47 volts. It includes diodes for each battery to enable the batteries to discharge into the unregulated bus when the bus voltage falls below the battery voltage (less the diode voltage drop).

Two series, dissipative-type, constant-current battery charge controllers - one dedicated to each battery - are provided. They are capable of supplying a continuous charge current of either C/20 (280 mA) or C/10 (500 mA). The charge controllers are

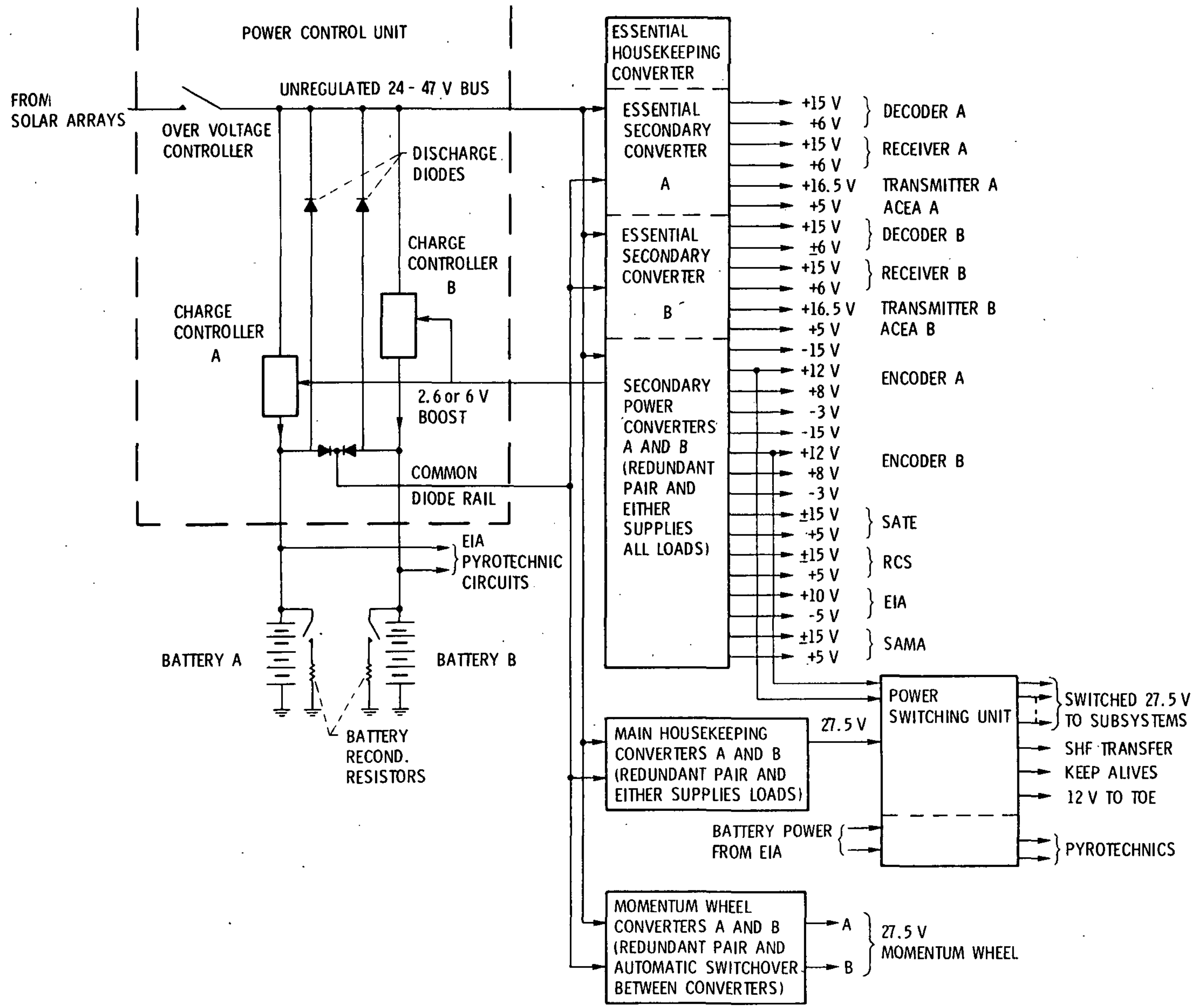


Figure III-32. - Housekeeping bus functional block diagram.

supplied from the bus with an additional 2.6- or 6.0-volt boost above the unregulated bus from the Housekeeping Power Conditioning Secondary Power Converter (SPC) unit (described later). The battery charge rate and boost voltage are selected by ground command. The batteries may be reconditioned, one at a time, on command by switching to the external discharge resistors, which control the discharge current to 2.0 A maximum.

A common diode rail, formed by combining the direct outputs of the two batteries into an OR gate configuration, maintains low-level sensing and control circuits constantly active. The 24- to 36-volt common diode rail voltage is capable of supplying 75-mA average current with peaks of 7.1 A for 20 msec. Hard connections between the battery outputs and the pyrotechnic firing circuit inputs (located in the PSU) are made within the PCU.

Essential Housekeeping Converter

The Housekeeping Power Conditioning Units include several converters, one of which is the Essential Housekeeping Converter (EHC). The EHC is composed of two parts: The Essential Secondary Converter (ESC), which is fully redundant, and the Secondary Power Converter (SPC), which is partially redundant. The two redundant ESC units are hard wired to the supply bus and must operate 100 percent of the time. One of these supplies the five controlled and regulated voltages for the TT&C Receiver A, Decoder A, and Transmitter A, and also for the ACEA A. Similarly, the other redundant converter supplies the inputs to TT&C Receiver B, Decoder B, Transmitter B, and the ACEA B. The total output of the ESC is approximately 23 watts and it operates at 74.5 percent efficiency.

The second portion of the EHC consists of the partially redundant SPC. Unlike the ESC, the SPC units are capable of being independently ground controlled. The SPC's supply the power and voltages required for a variety of loads as shown in figure III-32. The output of the SPC is approximately 19.5 watts. It operates at 73.5 percent efficiency at full load.

Main Housekeeping Converter

In terms of output power, the Main Housekeeping Converter (MHC) is the largest converter on the housekeeping bus. As is shown in figure III-32, it supplies regulated 27.5-volt power for all subsystems controlled by the Power Switching Unit (PSU), except the 12-volt power to the Transfer-Orbit Electronics (TOE) and the unregulated battery power to the pyrotechnics. The MHC is comprised of dual redundant converters, only one of which is in operation at any one time, as determined by command. Outputs of the converters are commoned to a single output line. The MHC is designed for 130-watt continuous output and operates at 87 percent efficiency.

Momentum Wheel Converter

The Momentum Wheel Converter (MWC) is also dual redundant. The MWC provides 27.5 volts for the primary and redundant momentum wheel assembly drive circuits. Only one converter supplies power at any one time. An Automatic Failure Protection Circuit is incorporated into the MWC to automatically switch to the other converter in event of failure of one of the converters. The converters can alternatively be controlled by ground command. The MWC provides a maximum output of 65 watts at 85 percent efficiency.

Power Switching Unit

A Power Switching Unit (PSU) provides means for ground-initiated on/off control of power supplied by the MHC, SPC, and batteries to spacecraft subsystems. Except for pyrotechnic circuits and the TT&C antenna transfer switches, these switching functions are accomplished with magnetically latching relays. "Keep alive" circuits supply power to various subsystems in order to maintain them actively responsive to telemetry and command signals. The keep-alive circuits are each current limited with a 1000-ohm series resistance to prevent overcurrents in event of downstream faults.

The PSU includes the pyrotechnic firing relays for the apogee-motor detonators and the JBSA panel actuators. Separate primary and redundant circuits are provided. The primary circuits are powered by one battery; the redundant circuits are powered by the other. The primary apogee-motor firing circuit in the PSU fires one detonator; the redundant circuit fires the other.

Experiments Power Conditioner

The Experiments Power Conditioner (EPC) is divided into two sections: a voltage controller and a converter section, as shown in figure III-33. The EPC provides voltage control of the 76-V experiments bus and secondary power for spacecraft experiments.

The dual-redundant voltage controller section senses bus voltage and accordingly provides on/off switching of the 76-V experiments bus by controlling the redundant relays K1, K2, K3, and K4. The voltage controller also controls on/off switching of the Experiments Power Converters A and B by means of relays K5 and K6, respectively. Bus

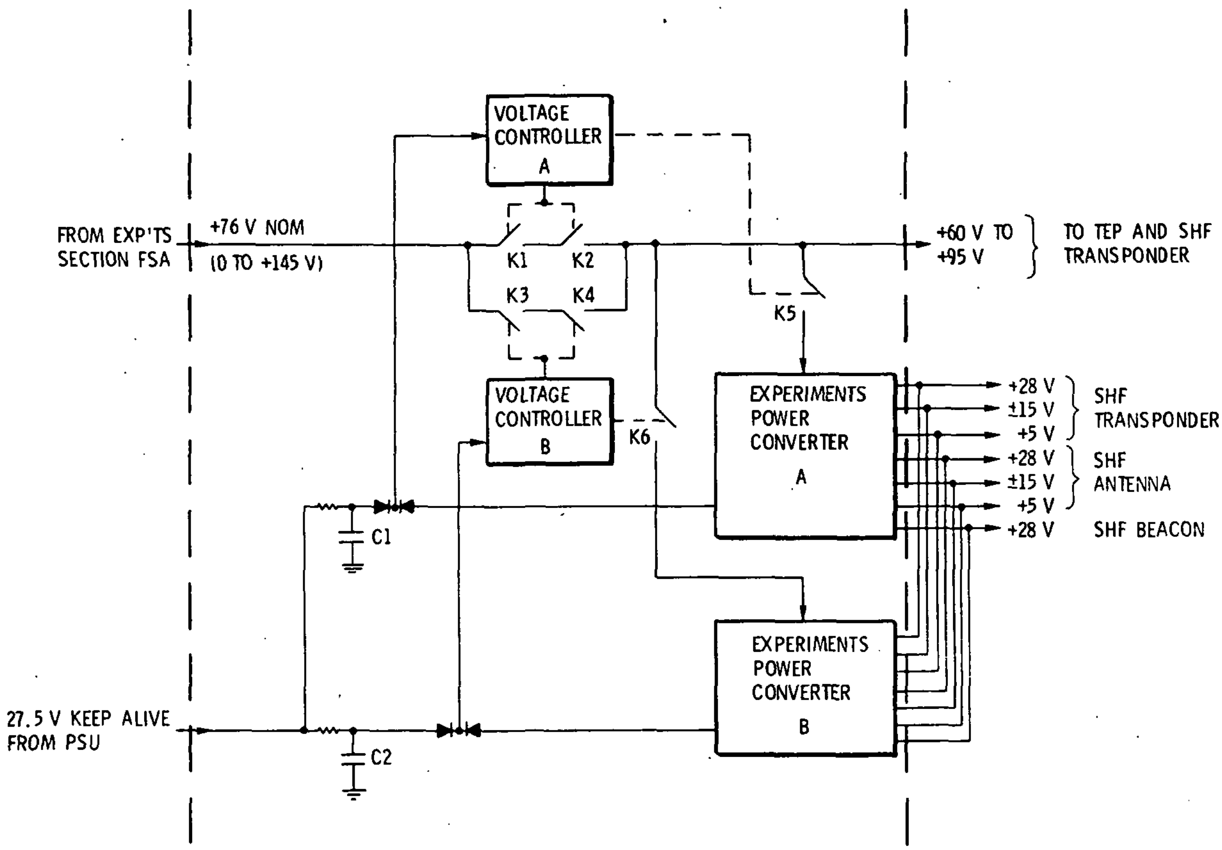


Figure III-33. - Experiments Power Conditioner functional block diagram.

relays K1 to K4 are non-latching-type relays, and only two of these relays are closed at any one time to limit thermal dissipation. Capacitors C1 and C2, which are maintained precharged by the keep-alive circuit, supply kick energy to initially close the bus relays. They are held closed by power from the corresponding Experiments Power Converter, which automatically switches on (through relays K5 or K6) as soon as the bus relays close. When an undervoltage condition occurs, as when the array enters eclipse, the voltage controller opens the bus relays K1, K2, etc. It will switch them on again when the bus voltage returns to normal, but only after the normal condition has been preceded by an overvoltage condition. This prevents relay undervoltage cycling, which could occur when the array open-circuit voltage recovers, on coming out of eclipse, only to be forced off again due to bus loading. An override command disables the overvoltage/undervoltage sense circuits and allows ground on/off commands to operate the bus relays, irrespective of the bus input (array) voltage.

The converter section of the EPC contains two regulating converters, one of which is in standby redundancy. Both converters cannot be on at the same time. The converters receive unregulated bus power and supply the secondary load busses, which are line and load regulated. The converters have the capability of supplying a continuous output load of 40 watts and operate at an efficiency of 83 percent.

Power Budget

Power requirements for the various spacecraft electrical loads are listed in table III-10. The values shown include the three basic mission operational modes: spinning phase, synchronous sunlight, and synchronous eclipse.

ELECTRICAL INTEGRATION SUBSYSTEMS

Electrical Integration Assembly

The purposes of the Electrical Integration Assembly (EIA) are as follows:

- To condition temperature sensors throughout the spacecraft for telemetry to ground
- To incorporate switching functions required to successfully integrate the spacecraft
- To accommodate other circuitry requirements necessary for electrical integration of the spacecraft

Temperature sensor signal conditioning circuits are provided in the EIA for 39 thermistors located at various thermally critical places about the spacecraft. These circuits generate analog voltage outputs covering a temperature range of -30° to $+90^{\circ}$ C (-22° to 194° F). Thermal control heaters for the batteries, the RCS, and other miscellaneous

TABLE III-10. - PREDICTED ELECTRICAL SYSTEM POWER BUDGET

[As of June 1975.]

Component	Operational mode		
	Spinning phase	Synchronous sunlight	Synchronous eclipse
	Power, W		
Transmitter Experiment Package	1.2	585.2	6.7
SHF Antennas, peak	-----	12.9	-----
Transponder	-----	98.0	-----
SHF Beacon	-----	18.0	-----
Telemetry, Tracking, and Command transmitter	12.7	12.7	12.7
Encoder and Transfer-Orbit Electronics	1.7	1.3	1.3
Receivers	5.0	5.0	5.0
Decoder	.7	.7	.7
Solar Array Mechanical Assembly	-----	8.8	3.4
Solar Array Technology Experiment	-----	5.0	5.0
Power Control Unit[a]	3.1	7.7	5.0
Essential Housekeeping Converter[a]	26.1	28.2	27.6
Main Housekeeping Converter[a]	4.6	8.5	5.6
Momentum Wheel Converter[a]	-----	6.3	4.8
Experiments Power Converter[a]	-----	18.1	-----
Power Switching Unit	.2	.3	.3
Batteries at C/20[b]	[c]10.4	[d]21.9	-----
Electrical Integration Assembly	4.9	4.9	4.9
Spacecraft Wiring Harness[a]	.7	1.5	1.2
Nonspinning Earth Sensor Assembly	-----	2.5	2.5
Spinning Earth Sensor Assembly	2.2	-----	-----
Sun Sensor Assembly	.5	.5	.5
Three-Axis Rate Gyro	-----	13.4	-----
Attitude Control Electronics Assembly	1.1	5.5	5.5
Momentum Wheel Assembly	-----	5.4	5.4
Heaters[e]	19.4	105.5	-----
Reaction Control System	13.0	18.5	11.90

[a]Variable - depends on other spacecraft loads.

[b]Variable - depends on unregulated bus voltage.

[c]1 Watt from batteries.

[d]2 Watts from batteries.

[e]Heaters are used as required.

needs are controlled from the EIA. Telemetry flag signals responsive to the various EIA functions are generated. Additional operations performed within the EIA include control of axial engine selection and firing circuits, control of the JBSA backup pyrotechnic devices, provision of a precharging circuit for the TEP power processing input filter (to avoid large switching transients on the MHC output), noise filters for the SATE accelerometers, and RCS pulse counting circuitry.

Spacecraft Harness

The Spacecraft Harness provides electrical connections for both signal and power equipment located on the forward and aft platforms and north and south panels. It also connects the Body Array and the DSA (through slip rings on the north and south panels) to the various spacecraft subsystems. The Spacecraft Harness includes the main wiring harness, umbilical connectors, separation switch, terminal blocks, grounding connectors, structural ground bonding, and the battery conditioning resistors (spaced circumferentially about the thrust tube). The main body of wiring is mounted onto the circular harness trough surrounding the apogee-motor thrust tube. Branches from the main wiring body extend from here to various structural panels (fig. III-34).

Continuous harness lacing is avoided, and instead spot ties are made at points not exceeding 15 cm (6 in.) apart. The harness insulation is designed to withstand a 500-V breakdown. Cable routing isolation methods and shielding are employed to minimize electromagnetic interference (EMI). Wire separation between sensitive and EMI generating devices is carefully maintained. Pyrotechnic cables are twisted and shielded per AFETRM-127-1, and cables are routed separately from other parts of the harness.

ATTITUDE CONTROL SYSTEM

The spacecraft Attitude Control System (ACS) (fig. III-35) has been designed to support two distinctly different mission phases. During the spinning phase, attitude control of the cylindrical rigid body is provided by means of ground-computed commands sent to the spacecraft; gyroscopic stability is assured by the mass property distribution. During the in-orbit phase, when the satellite consists of a central body from which two large sun-tracking solar arrays are deployed, stabilization and control are accomplished by means of a three-axis onboard autonomous system that uses the momentum wheel - offset thruster principle.

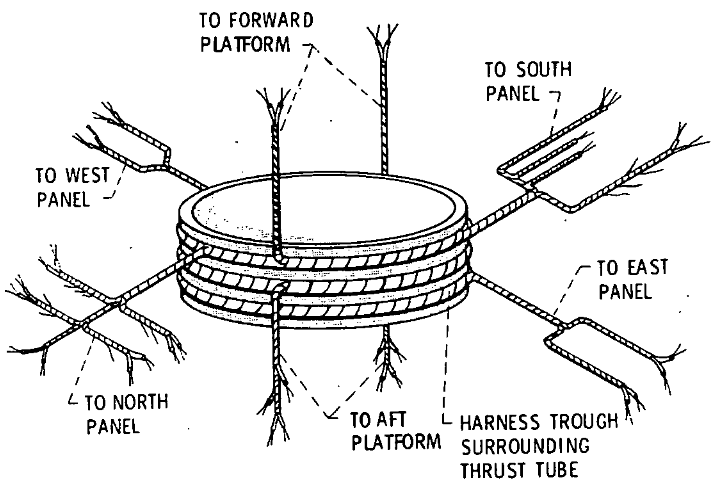


Figure III-34. - Spacecraft wiring harness configuration.

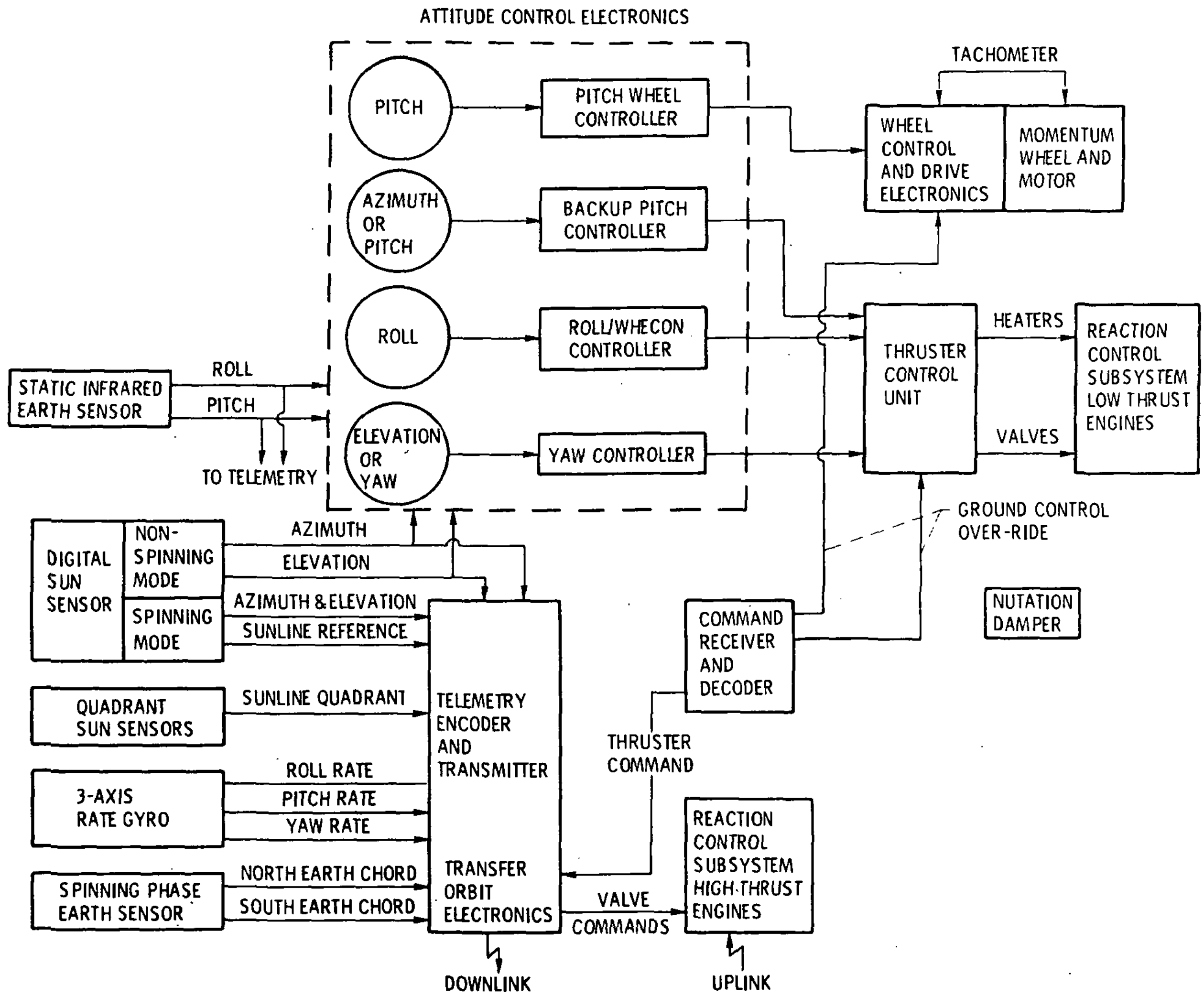


Figure III-35. - Attitude Control System block diagram.

Spinning Phase

During the transfer-orbit and station-acquisition phases, the spacecraft will be spinning at 60 rpm about its maximum moment-of-inertia axis, which is the yaw axis (I_{zz}). Attitude will be determined from measurements made with a spinning-phase earth sensor and a digital sun sensor together with orbital position data. Axial and radial hydrazine thrusters with a nominal thrust level of 22.5 N (5 lbf) will provide necessary attitude and orbit adjustments. Due to the favorable inertia properties (pitch axis/roll axis inertia ratio I_{zz}/I_{xx} of 1.3), stabilization and control during the spinning phase are generally regarded as a well-established and demonstrated technology.

Attitude-Acquisition Phase

During the attitude-acquisition phase, the spacecraft is to be transferred from a state where it is spin-stabilized at 60 rpm in synchronous orbit with the positive yaw (spin) axis pointing south, to the on-station three-axis stabilized state. An ordered sequential execution of a number of elemental spacecraft maneuvers begins with despin (together with passive or active nutation damping), deployment of solar arrays and initiation of sun autotrack, acquisition of the sun line onto the spacecraft negative roll axis, spinup of the momentum wheel, acquisition of the earth by a static infrared earth sensor (NESA) by appropriate attitude maneuvers induced by thrusters, and corresponding enabling of the autonomous on-station stabilization primary mode.

The following complement of spacecraft hardware will be used for control during the acquisition phase: digital sun sensor, three-axis rate gyro (active nutation damping and control of spin rate at low rates), telemetry and command, earth sensors (NESA), and ACEA controllers.

Three-Axis Attitude-Stabilized Phase

In the in-orbit operation phase for the remainder of the 2-year mission, the spacecraft will be attitude stabilized in a three-axis mode by autonomous onboard means with the yaw axis pointing toward the earth, the pitch axis normal to the geostationary orbital plane, and the roll axis along the orbital velocity vector (fig. III-36). The primary means of autonomous stabilization is the momentum wheel with offset thrusters. In the primary mode, the basic elements are a roll/pitch error detector earth sensor NESA, a momentum wheel, pitch thrusters, and appropriate electronics. Pitch error information detected by the earth sensor is used to control spacecraft pitch attitude through momentum wheel torque control. Pitch thrusters are used by ground command

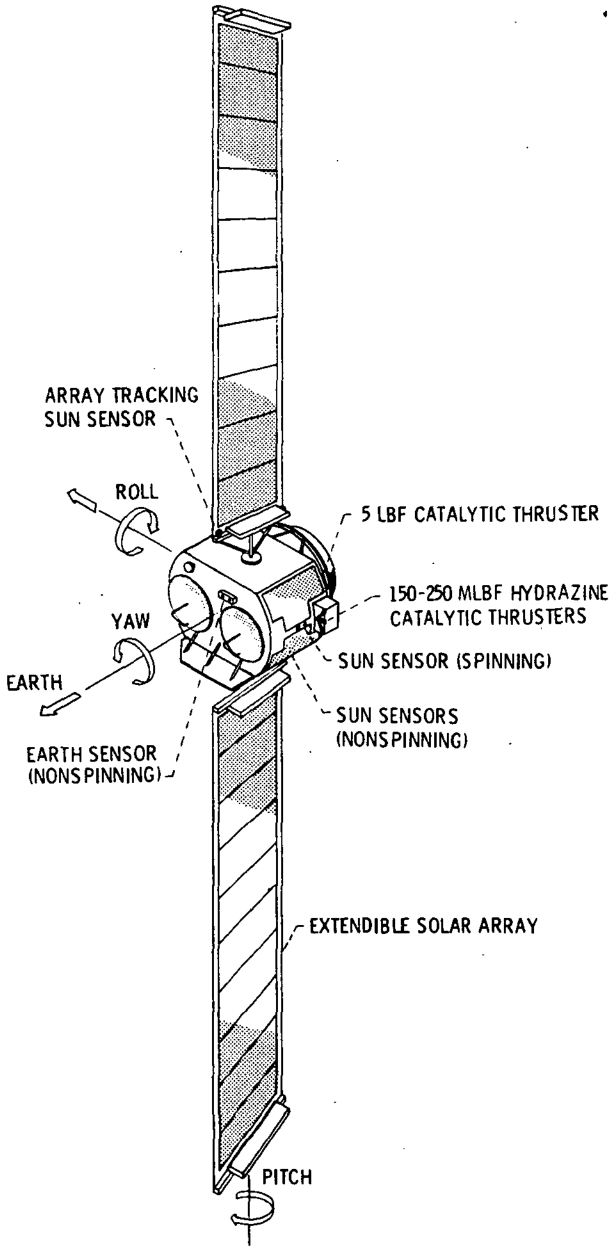


Figure III-36. - Spacecraft stabilization - ACS.

o maintain the wheel momentum within desired storage limits. Roll error signals detected by the earth sensor, appropriately processed by a roll/offset controller, are used o actuate offset hydrazine thrusters which are so arranged to produce a torque with both roll and yaw components. This torque, combined with gyrocompassing over about a quarter of an orbit, is designed to remove both roll and yaw attitude errors induced by the solar torque.

A backup mode is available for use in the possible event of momentum wheel malfunction and shutdown. This mode requires ground control of the appropriate thrusters.

Changes in controller gains, deadband, and filter configuration are designed in parameter sets to allow for capture of the spacecraft from large-angle, high-rate initial conditions and subsequent control to the in-orbit levels specified, as follows:

Roll range, deg	±0.1
Pitch range, deg	±0.1
Yaw range, deg	±1.1

Nonspinning Earth Sensor Assembly

The infrared Nonspinning Earth Sensor Assembly (NESA) will provide both pitch and roll error data and consists of a 14-μm scanning head and separate electronics. The heads are mounted on the earth-pointing face (forward platform) with the optical axis facing along the spacecraft positive yaw axis, and the electronics are mounted in the spacecraft interior. Pitch and roll information is obtained from a single head. The "A" head scans parallel to the spacecraft roll/yaw plane (offset 3.5° about the roll axis), taking in the southern hemisphere. Pitch (east-west) errors are detected by comparing the measured distances between the leading edge of the earth crossing and the center of the scan and between the center of scan and the trailing edge of the earth crossing. Roll (north-south) errors are determined by comparing the total scan length between horizon crossings to a reference chord length established for synchronous altitude. The "B" head scans parallel to the spacecraft yaw/pitch plane (offset 3.5° about the pitch axis), taking in the western hemisphere. Consequently, the pitch and roll error determination is interchanged with respect to the "A" head.

Spinning Earth Sensor Assembly

Two Spinning Earth Sensor (SES) assemblies are used. One looks east and the other looks west. Both are offset 5° from the pitch/roll plane, as shown in figure II-3(a). Each channel is an independent infrared instrument designed specifically for use in a spin-stabilized vehicle. The field of view extends radially from the spin axis and produces a circular scan as the vehicle rotates. The thermal discontinuity between earth and outer space generates a signal as the field of view is scanned across the horizon. Horizon-crossing signal data from the sensor are telemetered to the ground, where control of the spin axis is maintained by a command link.

Sun Sensor Assembly

The Sun Sensor Assembly (SSA) is used in the nonspinning and spinning modes during in-orbit operation and transfer-orbit phases, respectively. The assembly consists of five digital Nonspinning Sun Sensors (NSS), two digital Spinning Sun Sensors (SSS), a Sun Sensor Electronics (SSE) unit containing redundant signal processing channels and redundant power supplies, and interconnect cables connecting the sensor heads with the SSE.

The NSS's provide a two-axis measurement of sun-line azimuth and elevation from spacecraft reference axes. The sensors provide a parallel gray code output to the SSE, which is telemetered to the ground processing system. The digital spinning sun sensors provide a sun reference pulse when the sun crosses the roll/yaw plane (for encoding reference) and sun elevation data in gray code format.

Three-Axis Rate Gyro Package

The rate gyro package consists of three mutually perpendicular rate gyros with power supply and signal conditioning circuitry. Dynamic range is $\pm 10^{\circ}$ per second. Both rate information obtained by these gyros is telemetered for ground processing.

Attitude Control Electronics Assembly

The Attitude Control Electronics Assembly (ACEA) consists of three functional units: the Attitude Control Electronics (ACE), the Thruster Control Unit (TCU), and the Power Switching Unit (PSU). The PSU switches power to the subassemblies of the Attitude Control System upon ground command. The TCU interfaces between the Reaction Control System, the Attitude Control Electronics, and ground control. The TCU contains the necessary circuitry to arm or inhibit individual thrusters and to process thruster firing signals from the ACE or ground, plus instrumentation to measure thruster firing characteristics and telemetry and command signal conditioning.

The ACEA is the heart of the Attitude Control System. Essentially it is a special-purpose digital computer which provides the interface between the sensors, telemetry, and command, the TCU, and the Momentum Wheel Assembly. It contains the roll/offset and pitch wheel controllers. The ACEA uses digital signal processing; all filters are implemented with Digital Differential Analyzers (DDA).

Momentum Wheel Assembly

The Momentum Wheel Assembly (MWA) consists of a housing containing a flywheel, bearing assembly, and electric drive motor, together with electronics for driving the wheel and controlling and measuring the wheel angular rate. The motor produces a net acceleration torque of about 42.6×10^{-3} N-m (6 in.-oz) about the flywheel axis. At its nominal speed of 3750 rpm, the wheel has an angular momentum of 19.05 N-m/sec (14.0 ft-lb/sec). The wheel speed controller operates in one of two modes: constant wheel speed or torque control. In the constant-wheel-speed control mode, the wheel angular rate is automatically maintained at a desired value (set by ground command). In the torque control mode, wheel speed is modulated through closed-loop control in response to pitch error information in order to maintain earth pointing within the specified limits. The pitch loop controller provides continuous control of wheel speed over a range of ±10 percent off nominal.

Nutation Damper Assembly

The Nutation Damper Assembly (NDA) is a passive, viscous damper. The straight-tube damper assembly is partially filled with mercury and is mounted through the spacecraft aft platform. The NDA is designed to dissipate nutations about the spin axis caused by torques imparted at third-stage separation, precession of spin axis by RCS high-thrust engine firing, apogee-motor firing, and station acquisition by RCS high-thrust engine firing.

STRUCTURE

The spacecraft structure consists of the following items, which are shown in figure III-1:

- Basic structure, including supporting, reinforcing, or bracing struts and attachments
- East panel

- West panel
- North equipment panel
- South equipment panel
- Aft equipment platform
- Central thrust tube
- Forward equipment platform
- Struts

The primary structure on which the entire spacecraft load is carried consists of the central thrust tube, the aft platform, and the 12 auxiliary support struts. Generally, all major components of the structure fall into one of three main manufacturing categories:

- Precision machine fittings from solid aluminum alloy
- Thin-gage magnesium or aluminum sheeting
- Thin aluminum face sheet combined with aluminum honeycomb core panels

Primary Structure

The spacecraft is primarily a four-sided structure. The north and south panels are flat; the east and west panels are curved. There are two additional surfaces called the aft and forward platforms. The primary structural elements are the central thrust tube, the aft platform, and the support struts. The remaining elements form the peripheral structure, which is supported by the primary structure. The peripheral structure consists of the forward platform, the north and south panels, the east and west panels, and the trusses and bracketry connecting these platforms and panels to the primary structure.

All spacecraft equipment mounting platforms are 2.5 cm (1 in.) honeycomb with a core material made from 5052 perforated ribbon 16.8×10^{-3} mm (0.0007 in.) thick, bonded to form an egg-crate pattern. The face sheets are chemically milled 0.25-mm (0.01-in.) 2043 T3 aluminum alloy sheets bonded to the cores with adhesive. Heavily loaded areas such as strut attachment points are locally filled with a foam densification compound.

A main feature of the panels and platforms is the inserts used for structural attachment and component mounting. Basically, the structural type of insert is a solid aluminum bar bonded into the honeycombed panel and drilled on final assembly. The component type of insert is one which has been adopted from the Telsat spacecraft and is a proprietary item. The panels are predrilled and inserts placed in the holes which have been partially filled with foam.

Central Thrust Structure

The thrust tube acts as the main central column and consists of three machined aluminum rings. They are attached by thin rolled magnesium sheets which are in a cylindrical configuration for the upper portion of the tube and in a conical configuration for the lower portion. The rings and sheets are then riveted together. Two of the rings (separation and apogee) are made from 7075-T73 aluminum alloy forgings; the third (forward) is made from 2219 plate.

The struts used throughout the spacecraft are of a standard construction consisting of thin-walled tubing and act as support and stiffening columns for the honeycomb platforms.

Structural Design Requirements

The primary requirements for the design, performance qualification, and acceptance of the structure are embodied in a variety of specifications that are unique to CTS:

- The flight structure must survive a 2-year lifetime in orbit after qualification and acceptance tests and launching.
- The structure is designed to a stiffness level such that resonant frequencies do not occur below 35 Hz in the Z-axis nor below 15 Hz in the lateral axes, with uncoupling of the resonant modes.
- The structure must minimize distortions which can affect pointing of critical components.
- The structure must be electrically conductive so that it acts as a common ground plane for CTS equipment.
- The structure must support a spacecraft and adaptor weight of 726 kg (1600 lbm) with margin for the required CTS safety factors.

The structure is also designed to be compatible with the requirement that the spacecraft overall moment-of-inertia ratio be equal to or greater than 1.20 with the Deployable Solar Arrays in the stowed position. Provisions are made on the extremities of all equipment panels to accommodate as much as 6.8 kg (15 lbm) of balance weights.

REACTION CONTROL SYSTEM

The Reaction Control System (RCS) is a dual-thrust-range, blowdown-mass-expulsion, catalytic-decomposition hydrazine propulsion system. Its primary functions are to provide controlled impulse for orienting the spacecraft spin axis during spinning phases; to adjust the orbit; and, in conjunction with the Attitude Control System, to provide earth-oriented, three-axis attitude control for the 2-year in-orbit period of the mission.

The system is comprised of two propellant tanks and four thruster groups suitably interconnected and isolated by latching valves to provide redundancy for all in-orbit control functions. A schematic diagram is shown in figure III-37. The system is installed on the spacecraft aft platform in one operation as a complete unit with the exception of the Electrical Junction Box (EJB), which contains Bendix-type connectors.

Figure III-38(a) illustrates the system mounted on the spacecraft aft platform. The thrust vector geometry is shown in figure III-38(b). The RCS hardware is designed to be as light as possible while retaining structural integrity and to provide maximum utilization of fuel during all modes of spacecraft operations. The RCS is designed to have minimum probabilities for successful operation of 0.95 for the nominal 2-year mission and 0.975 for the backup possibility of 1-year of maximum operation with the pitch momentum wheel failed.

Component Selection

All components used in the RCS are selected from those which have been qualified on previous hydrazine systems or from those which are considered to be partially qualified by similarity, with further qualification performed for the parameters unique to the CTS program. The Electrical Junction Box (EJB) and Low Thrust Engines (LTE) required a rigorous qualification program. The performance requirements for the LTE are unique because of the short pulse width of 0.007 second. This required a separate qualification program for the LTE. The performance requirements for the RCS maneuvers are specified in table III-11.

High- and Low-Thrust Engines

The modules in the RCS which contain the engines are designated east, west, north, and south Rocket Engine Modules (REM). The east and west REM's contain five low-thrust engines each, which perform east-west stationkeeping and provide yaw and pitch control torques. The west REM also contains the high-thrust radial and axial engines used during the spacecraft spinning phase. North and south REM's each have three low-thrust engines: two for primary and backup offset operation, and one for roll control.

The RCS requires two engine designs: the high-thrust engine (HTE) and the low-thrust engine (LTE). The HTE is needed for the spinning mode. The HTE used on the RCS is a modified version of the 22.5-N (5-lbf) monopropellant catalytic hydrazine engine presently flying on the IDSCP/A and NATO III satellites. The engine weight has

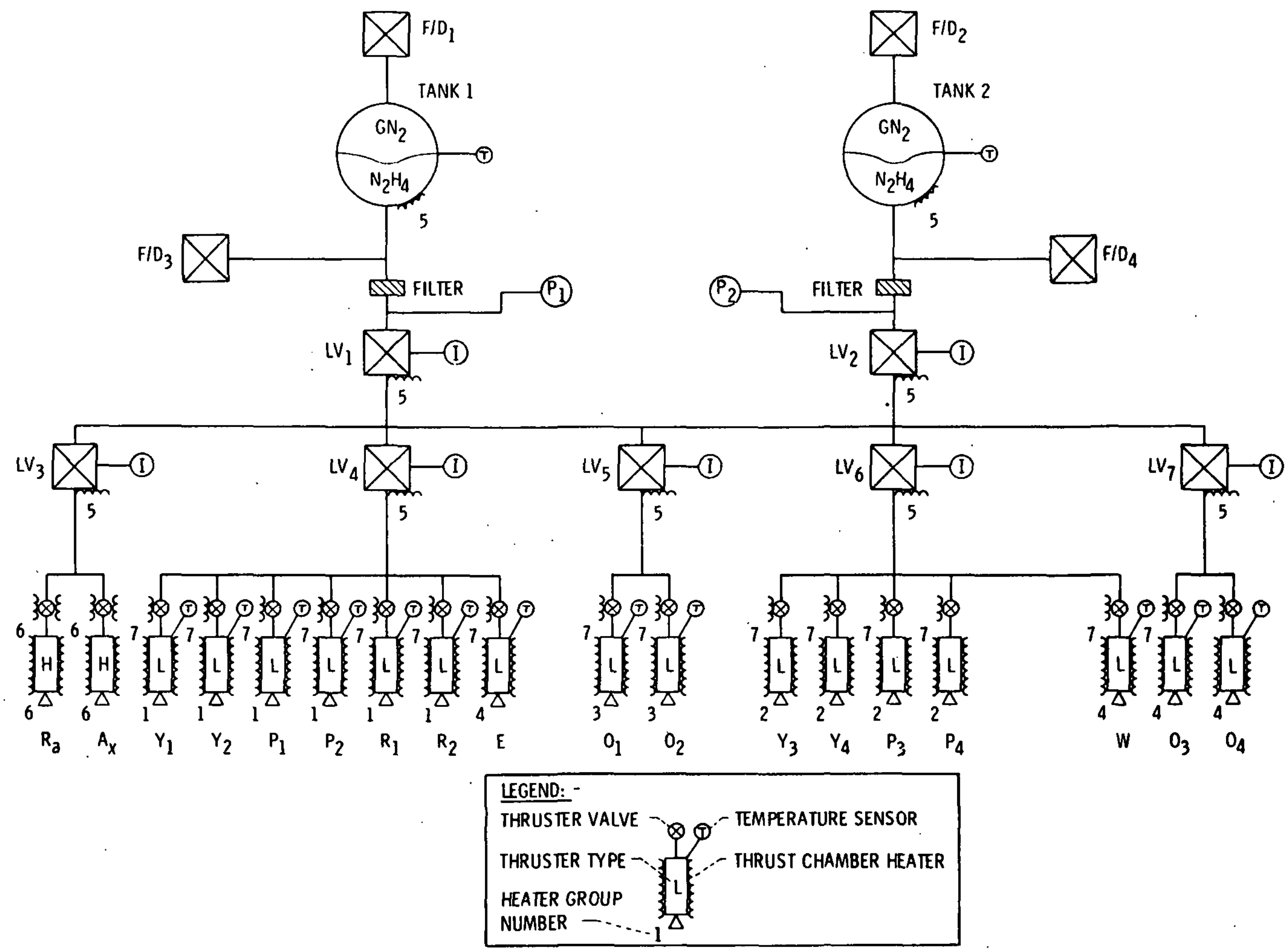
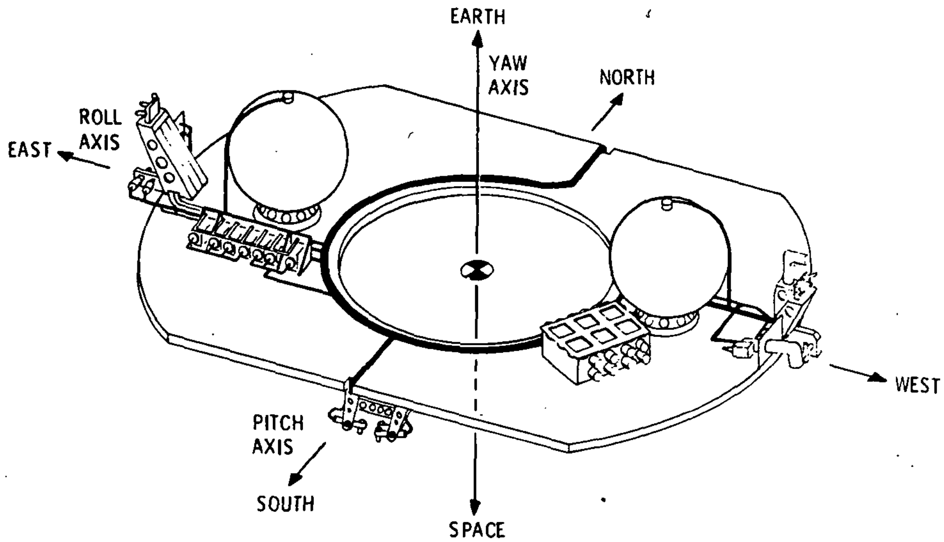
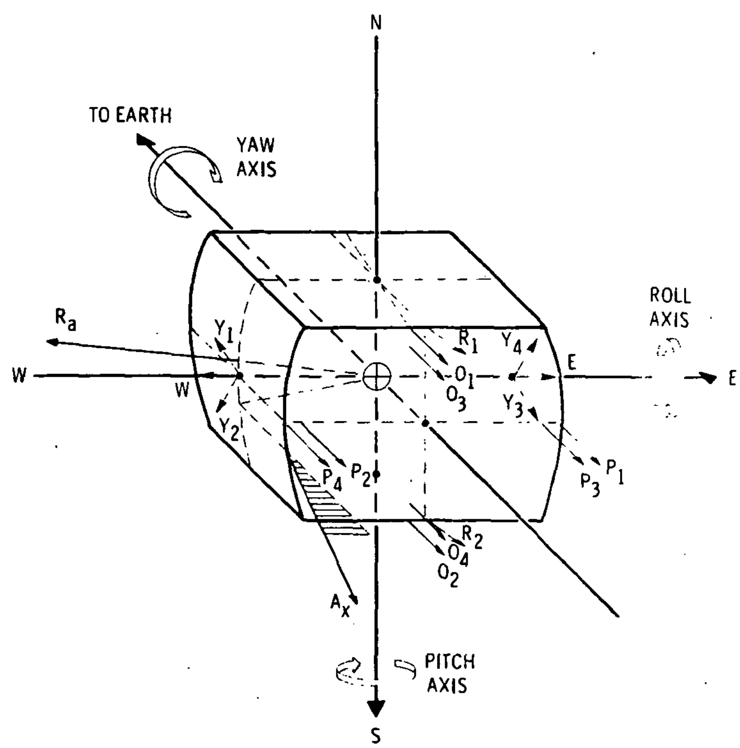


Figure III-37. - Reaction Control System propulsion schematic diagram.



(a) Reaction Control System on aft platform.



(b) Reaction Control System thrust vector geometry.

Figure III-38. - Reaction Control System.

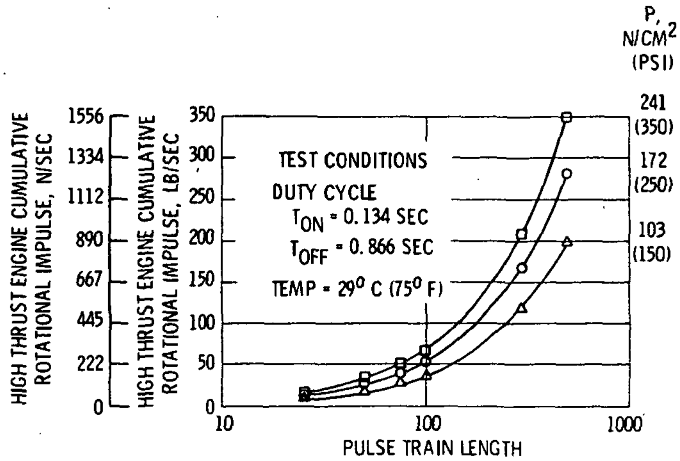
TABLE III-11. - PERFORMANCE REQUIREMENTS OF REACTION CONTROL SYSTEM

Maneuver	Primary engine	Nominal electrical time, sec		Nominal pulse train length	Nominal total number of pulses	Nominal total impulse	
		On	Off			N-sec	lbf-sec
Preapogee precession	Axial	0.135	0.865	~550	~1 100	[a]2 785	[a]626
Postapogee precession	Axial	.135	.865	~275	~550	[a]1 299	[a]292
Station acquisition	Radial	.135	.865	~550 - 755	~10 000	[a]18 949	[a]4260
Despin	Negative yaw couple	~1015	-------	1 per engine	1	1 263	284
Attitude acquisition:							
Three-axis control	Pitch, roll, and yaw	Very low duty cycle		Variable	Variable	≤245	≤55
Pitch wheel spinup	Positive pitch	0.050	15.0	600	600	~22	~5
In-orbit attitude							
Offset engines	Offset pitch	.007	2 - 2000	------	-----	823	185
Pitch momentum		.050	15.0				
Secondary mode	Yaw and offset	.007	------	------	-----	---	---
East-west stationkeeping	West	.050	.950	75 - 150	15 000	605	136
Wheel-failed backup attitude control	Pitch, roll, yaw	.007	2 - 2000	-------	-----	---	---

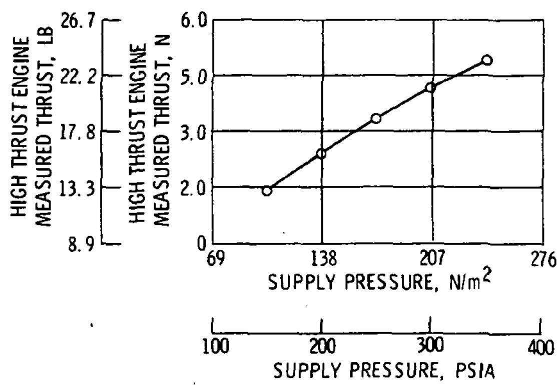
[a]At 100 percent rotational efficiency.

been reduced and its thermal standoff resistance increased for the CTS program. This resistance lowers the heat load to the spacecraft and improves engine performance. In addition, a right-angle, 60/1-area-ratio bell nozzle has been added to facilitate packaging and to further improve engine performance. The pulsed mode of operation is used for most spinning maneuvers. Electrical on-time is 0.135 second. Off-times vary between 0.776 and 0.976 second, depending on spin rate.

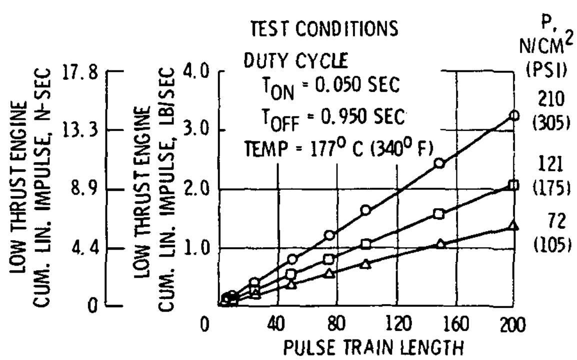
The two HTE's are ground commanded by using flight prediction data generated during engine qualification and acceptance testing. Data for the cumulative rotational impulse as a function of pulse train length and propellant supply pressure are given in figure III-39(a) for the nominal spinning duty cycle of 0.135 second on and 0.865 second off. For small-delta velocity maneuvers the spacecraft velocity change caused by a given pulse-train-length firing is simply the cumulative rotational impulse for that firing divided by the average mass of the spacecraft during the firing. The high-thrust engines are also qualified for 5 minutes of steady-state operation once each hour should initial adjustment of the orbital inclination be required. The HTE thrust as a function of supply pressure is shown in figure III-39(b). A trim orifice is used to obtain a thrust level of 22.5 N (5 lbf) ±5 percent at a supply pressure of 241 N/cm^2 (350 psia).



(a) Reaction control rotational impulse vs. pulse train length supply pressure.



(b) Reaction control high thrust engine thrust vs. supply pressure.



(c) Reaction control linear impulse vs. pulse train length supply pressure.

Figure III-39. - Reaction Control System performance.

The low-thrust engine (LTE) is needed for both attitude acquisition (despin) and in-orbit three-axis stabilized modes. The LTE used on the RCS is a modified version of the 0.45-N (0.1-lbf) monopropellant catalytic hydrazine engine presently flying on the Solrad X satellite. The engine valve interface is modified to minimize propellant hold-up volume to meet the low-minimum-impulse bit requirements of the CTS program.

The LTE is used to perform a variety of spacecraft maneuvers, some ground commanded and others controlled by onboard computers. East-west stationkeeping and pitch momentum dumping are ground commanded by using duty cycles of 0.050 second on, 0.950 second (nominal) off and 0.050 second on, 15 second (nominal) off, respectively. Data for the cumulative linear impulse delivered for these duty cycles as a function of pulse train length and propellant supply pressure is shown in figure III-39(c).

The LTE is required to produce repeatable small-impulse bits at low duty cycle during the attitude acquisition and in-orbit phases of the mission. In order to maximize specific impulse and minimize impulse bit size for this function, the LTE catalyst bed is heated to $171^{\circ} \pm 42^{\circ}$ C ($340^{\circ} \pm 75^{\circ}$ F), and the firing pulse width is reduced to a range from 0.007 to 0.020 second.

Despin from 60 to 2 rpm is performed by using the two yaw LTE's positioned to provide a pure couple and maximum moment arms. During this maneuver, these engines fire continuously.

Propellant Tanks

The propellant tanks contain a positive expulsion elastomeric (EPT-10) diaphragm separating the pressurant (nitrogen) from the propellant. Each side is loaded through a separate fill and drain valve of different size to prevent misconnection. Each tank has a maximum capacity of 12.3 kg (27.1 lbm) of hydrazine and operates in the blowdown pressure range from 241 to 72 N/cm^2 (350 to 105 psia). The constraint is a function of blowdown curve and not tank capacity. A curve of pressure against propellant remaining in each tank is shown in figure III-40. An etched-disk 10-μm absolute filter is located upstream of all control components in the feedlines from each tank.

The seven latching valves, two filters, and two pressure transducers are mounted on a common bracket to form a module. Two latching valves are located so that either propellant tank can be isolated. The remaining five latching valves provide isolation for the five groups of engines: the primary roll, pitch, and yaw engines and the east engine; the radial and axial engines; the offset engines; the backup offset engines; and the backup pitch and yaw engines and west engine. The consequences, if any thruster valve fails to close or to seal tightly, can be minimized by closing the group latching valve, thereby isolating that group of thrusters and enabling continuation of mission operation through the use of the backup thrusters.

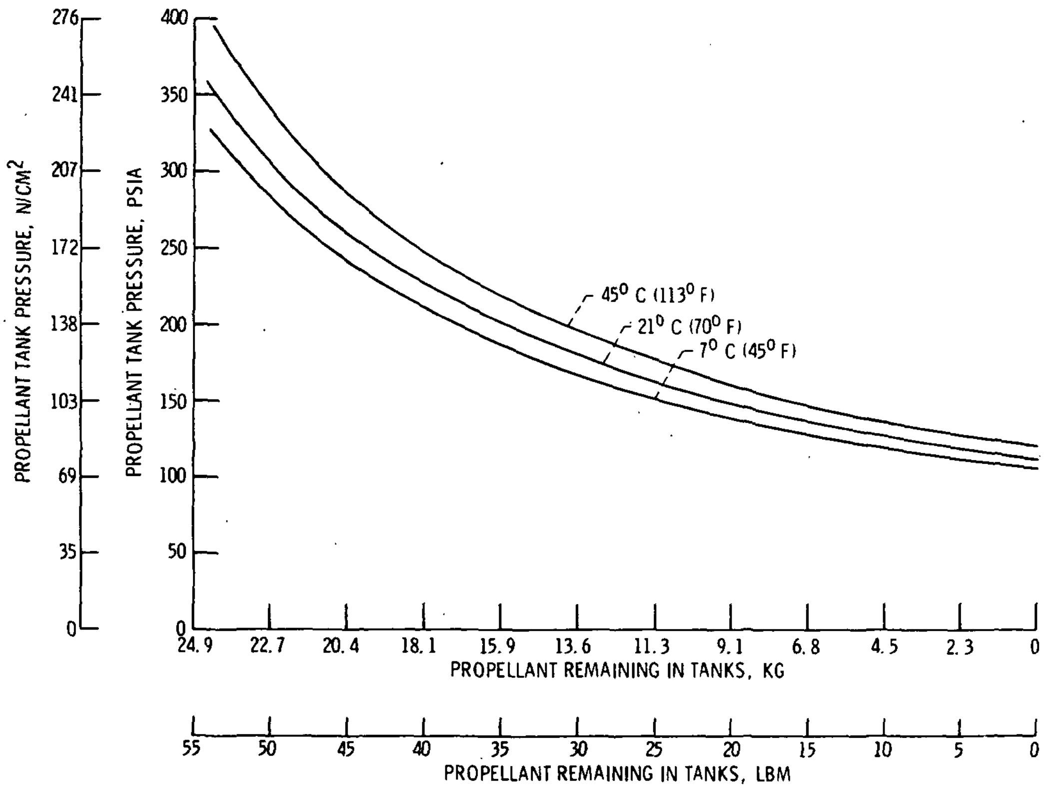


Figure III-40. - Reaction Control System propellant tanks blowdown curves.

All components (tank, fill and drain valves, filters, and tank-to-filter interconnecting tubes) are made of titanium to save weight. A transition to stainless-steel tubing is made downstream of the filters to improve thermal conductivity.

The module bracketry, which positions the rocket engines with respect to the spacecraft axes and provides thrust vector directional control, is designed to maximize the engine moment arms, to avoid engine plume impingement on spacecraft surfaces, and to permit attachment of RCS thermal blankets. These blankets are provided to protect RCS engine valves and lines from the extremes of the external thermal environment and to close the thermal gap between the complex REM's and the spacecraft panels.

Heaters

Since the CTS must operate in both spinning mode (with low internal heat dissipation) and nonspinning mode (with the high internal heat dissipation of approximately 600 watts), its thermal design permits aft platform and average internal spacecraft temperatures to vary between 5^{o} and 50^{o} C (41^{o} and 122^{o} F). Therefore, groups of thermal environment control heaters are provided to minimize command and power requirements while maintaining adequate RCS operating temperatures with design margin above the hydrazine freezing point $+2^{o}$ C (35.5^{o} F), throughout the mission. The RCS thermal and component selections are compatible with the stated maximum spacecraft temperature while these heaters are operating. Catalyst bed heaters are provided on the low-thrust engines to achieve the performance and pulse life requirements. These heaters are parallel redundant and grouped to minimize command and power requirements.

Within the electrical junction box (EJB) the RCS provides driver circuits for engine valves, latching valves, and heater groups; solid-state latches for thermal environment control heaters; and high-accuracy signal conditioning for all instrumentation (both telemetry and ground testing). Sufficient passive redundancy is provided to gain the high subsystem reliability required for the 2-year mission. Drivers are powered from the CTS-regulated 28-V dc main housekeeping bus. Logic circuitry and signal conditioning obtain power from secondary module-regulated ±15-V dc and ±5-V dc busses. Because of the combination of high power transmission and sensitive logic circuitry onboard the CTS, the subsystem is designed to comply with the stringent EMC requirements for emission and susceptibility of MIL-STD 461A Notice 2 and MIL-STD 462. The electrical interface and functions performed within the EJB are shown in figure III-41. RCS pulse counting is also provided to estimate the remaining fuel.

Propellant tank temperature (±2 percent accuracy) and pressure (±1 percent accuracy) are monitored for each tank in order to predict engine performance and estimate fuel usage during the mission. Primary LTE catalyst bed temperatures are measured over the full operating temperature range of the engines to an accuracy which allows

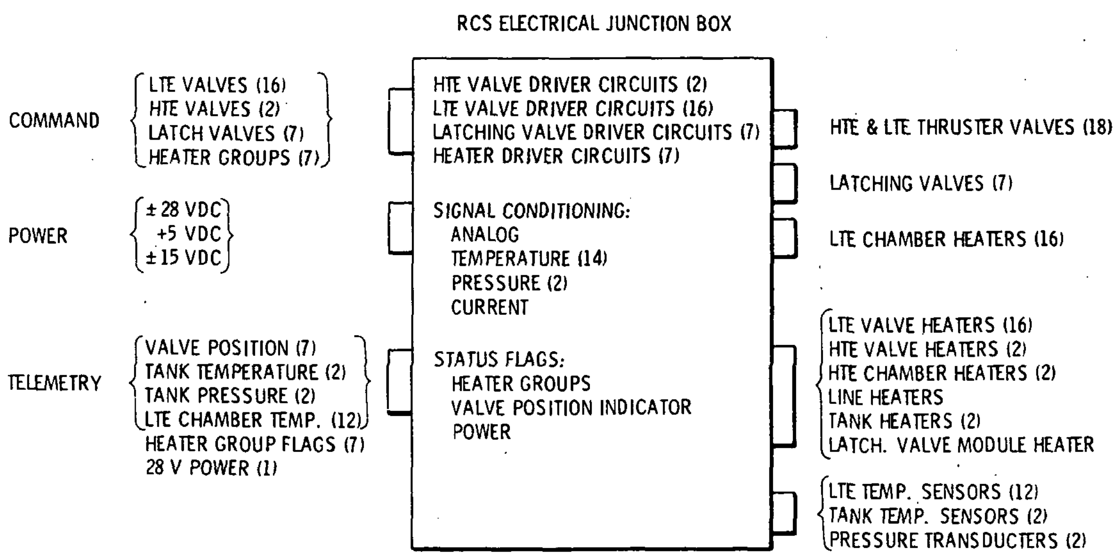


Figure III-41. - Reaction Control System electrical diagram.

the spacecraft operator to distinguish between heater-off and heater-on conditions. He can then predict engine performance by using these temperature data and engine flight performance prediction data. The total RCS 28-V dc current is monitored. The status of the heater group circuits, the position of latching valves, and the presence of RCS power are indicated by flags.

THERMAL CONTROL SYSTEM

Throughout the various phases of the mission including prelaunch operations, there will be wide variation in the amount of heat dissipated by components within and on the outer surfaces of the spacecraft. During the spin-stabilized phase the power dissipation will be relatively low, typically 55 to 90 watts. In synchronous orbit the spacecraft will be three-axis stabilized, and the power dissipation range will be 240 to 600 watts. Provision must be made for survival of components through eclipse periods and for operation of the spacecraft in synchronous orbit with any or all of the experiments switched off. The thermal design is also complicated by the presence of an apogee motor and by the need for low thermal distortion of the forward platform in order to provide a dimensionally stable platform for the SHF antennas and attitude sensors.

TABLE III-12. - SYSTEM FUNCTIONAL CONSTRAINTS

Transfer orbit	Drift orbit	Attitude acquisition orbit	Synchronous orbit
Spinning with sun line of sight $90^\circ \pm 25^\circ$ to spin axis (yaw)	Sun line of sight and power dissipations same as for transfer orbit	Three-axis stabilized with sun normal to west panel and then normal to forward deck at JBSA jettison and array deployment	Three-axis stabilized with sun
Power dissipations of 55 to 90 watts	Time duration, 24 days maximum	Nonoperating components 15° C above minimum turnon	Power dissipations of 240 to 600 watts
Time duration of 10 orbits with 30-minute maximum eclipses			Temperature limits of generally 0° to 40° C; batteries temperature limits of 0° to 25° C
Spacecraft temperature generally 0° to 40° C		Time duration of 40 minutes from array deployment to full spacecraft operation	Time duration of 2 years
Apogee-motor temperatures of -7° to 38° C			Power density maximum of 3 watts/in.^{2}

Table III-12 is a list of functional constraints for the various CTS mission phases. Table III-13 is a summary of the design dissipations used for the specific mission phases, and table III-14 is a list of the major-component design temperature requirements.

Insulation Materials

Multilayer insulation blankets are located on the spacecraft. As a fire safety precaution, Kapton is used for the two outermost layers of every blanket. Interior blankets, which are not exposed to solar input, are aluminized on both sides. Exterior blankets, which may be exposed to aerodynamic heating during launch and to solar input, are aluminized only on the side not exposed. All blankets provide for venting during launch. Ground connections are made to each metallized surface by strips of aluminum foil, 1.9 cm (3/4 in.) wide and 0.025 cm (0.001 in.) thick, to eliminate static charge buildup.

The principal means of heat removal from the spacecraft is through Second-Surface Mirror radiators located on the north and south panels. Small Second-Surface Mirrors are located on the aft platform. These radiators are made up of 2.54-cm (1-in.) square silver quartz mirrors bonded to the aluminum honeycomb faces. They have a solar absorptivity less than 0.06 and an infrared emissivity greater than 0.80. Thermal finishes are specified as either low emissivity (<0.06) or high emissivity (>0.85), and appropriate paints or finish materials are required. In some areas, thermal doubler plates are required to distribute heat dissipated by some units over a larger area than the base of the unit itself.

An exploded view of the thermal control components and their locations on the spacecraft is shown in figure III-3.

Thermal Design

The spinning phase thermal design is basically a passive design with the JBSA's and their enclosure skirts serving as the primary control of spacecraft thermal balance. The apogee motor is isolated from the spacecraft and from space while it is relying on its strong thermal coupling to the north and south Second-Surface Mirror radiating areas. In addition, a combination of coatings is used on the JBSA skirts. The skirts are covered frame structures fastened to the JBSA and extending to the edges of the north and south honeycomb panels. The combination of coatings includes oxidized stainless foil on the east-west areas and gold-coated Kapton and silver-coated Teflon on the forward and aft areas. To compensate for the internal spacecraft power dissipation variations, two commandable thermal control heaters are used. In addition, seven

TABLE III-13. - THERMAL DISSIPATIONS FOR MAXIMUM- AND

MINIMUM-TEMPERATURE CASES

Subsystem	Transfer and drift orbits		Synchronous orbit		Substitute heaters	Eclipse (72 min plus 6 min of transition)
	Maximum temperature case	Minimum temperature case	Maximum operating case	Minimum operating case		
	Thermal dissipation, watts					
SHF Communications	0	0	141.1	77.7	86.7	0
SHF Beacon			19.7	13.9	15.8	0
SHF Antenna			6.0	0	0	0
Transmitter Experiment Package			290	87.8	101	4.7
Telemetry, Tracking, and Command	17.6	15.2	16.8	14.6	-----	4.7
Solar Array Mechanical Assembly	0	0	17.7	13.1	-----	0
Power	29.2	13.6	86.7	37.1	-----	31.0
Miscellaneous Electrical	3.6	1.7	6.1	2.3	-----	2.3
Attitude Control System	4.1	2.7	35.5	16.2	-----	15.6
Reaction Control System	7.2	10.1	3.6	15.9	-----	9.7

TABLE III-14. - SUBSYSTEM DESIGN

TEMPERATURE LEVELS

[The design temperatures shown represent the range within the subsystem. Individual components may have other design temperatures within the design range shown. Acceptance temperatures are defined as design $\pm 5^{\circ}$ C. Qualification temperatures are defined as design $\pm 15^{\circ}$ C.]

Subsystem	Operation design temperature, $^{\circ}$C	
	Maximum	Minimum
SHF Communications	70	-5
SHF Beacon	55	0
SHF Antenna	70	-5
Transmitter Experiment Package	60	10
Telemetry, Tracking, and Command	50	-5
Solar Array Mechanical Assembly	95	-100
Jettison Panels	112	-80
Power	60	0
Attitude Control System	65	-50
Reaction Control System	50	5

commandable heaters are available to control temperatures within the propulsion system.

During the three-axis stabilization mission phase, Second-Surface Mirrors on the exterior of the north and south panels are used for rejecting the internal spacecraft component dissipations. All major dissipating components are mounted directly on the north and south panels. Low dissipating units are mounted on the forward and aft platforms. In the synchronous mission phase the Deployable Solar Arrays are extended approximately 76 cm (30 in.) beyond the north and south panels to allow satisfactory heat rejection to space. Since the lateral conductance of the north and south panels is insufficient to transfer the highly concentrated heat dissipations, thermal doublers are bonded to the honeycomb internal facesheets in these areas. When major dissipating components are not operating, substitute heaters for these components are energized by command. In addition, eight commandable thermal control heaters are available to compensate for the wide range of power dissipations.

The thermal subsystems are categorized in groups to facilitate design and analysis.

Central-thrust-tube thermal group. - The components of the central-thrust-tube thermal group are

- Blankets
  - Apogee-motor nozzle blanket with stainless-steel foil, which is extended to interface with aft-thrust-tube closure blanket
  - Apogee-motor belly blanket
  - Central-thrust-tube interior blanket
  - Aft-thrust-tube exterior blanket, which covers the belt antenna coaxial cable runs
  - Separation ring blanket
  - Miscellaneous collar blankets at the junctions of thrust tube and aft struts and thrust tube and aft platform, around belt antenna supports, and at junction of aft struts and aft platform
- Thermal barriers
  - Apogee-motor aft thermal barrier, which includes thermal finishes on inside aft end of nozzle
  - Apogee-motor intermediate thermal barrier
- Paint and miscellaneous
  - Includes low-emissivity finishes inside thrust tubes
  - Black paint on exterior of internal thrust tube
  - Low-emissivity finishes on belt antenna support brackets

Aft-platform thermal group. - The components of the aft-platform thermal group are

- Blankets
  - Aft-platform blanket, which includes TEP-OST sealing

- RCS tank support trusses
- Umbilical connection
- Sun sensors
- Nutation damper
- North and south REM collar blankets
- Battery zone (2)
- Aft strut (12)

- Second-Surface Mirrors on aft platform
- Black paint and edge-sealing tape

Forward-platform thermal group. - The components of the forward-platform thermal group are

- Blankets
  - Forward-platform blanket, half of which is covered with 0.1-mm (5-mil) aluminized Teflon
  - Thrust-tube-closure blanket, which is configured with clearance for apogee-motor ignition lines
  - SHF beacon antenna
  - Separate NESA sensor head blankets
  - TTC cone beam antenna
  - Safe and arm
  - Heat-pipe fitting blankets
- Black paint and edge-sealing tape

North-panel thermal group. - The components of the north-panel thermal group are

- External blankets
- Experimental converter doubler plate
- Silvered-quartz Second-Surface Mirrors
- Interior black paint and Momentum Wheel isolators (titanium)

South-panel thermal group. - The components of the south-panel thermal group are

- External blankets
- 20-Watt TWT and EPC doubler plates
- Silvered-quartz Second Surface Mirrors
- Interior black paint and edge-sealing tape

East- and west-panel thermal group. - The components of the east- and west-panel thermal group are

- Blankets for REM, spinning earth sensor, and sun sensor
- Sealing and finishing of panel edges
- Isolation from aft platform of <0.378 Btu/(hr)(^{o}C) (<0.21 Btu/(hr)(^{o}F))

TEP/heat-pipe thermal subsystem. - The 12-GHz, 200-watt TWT is mounted at the intersection of the aft platform and the south panel. Supplemental cooling for the body of the TWT and that part of the associated power conditioning that is not directly cooled by radiation to space is provided by a Variable-Conductance Heat-Pipe Subsystem, which transfers heat from these two sources to a radiator mounted to the forward platform (fig. III-8). Requirements for the heat-pipe/radiator system include the following:

- Capability to reject 196 watts at a temperature of 50° C (148° F) on the heat-pipe evaporator
- Capability to shut off at a temperature of 27° C (106° F) on the evaporator

The variable-conductance feature of this device automatically shuts off the external radiator when the TEP baseplate falls below a preset temperature. It thus augments the thermal dissipation of the south panel when the heat-transfer capability of the panel is exceeded.

The heat-pipe system includes three heat pipes, evaporator and condenser saddles, external radiating fin, and support structure. The principal control function is supplied by the heat pipe, which operates as a thermal switch to provide heat rejection when needed and minimizes supplemental heater power. The design uses methanol as the working fluid, with each heat pipe capable of transferring 150 watts to a 45.7- by 20.3-cm (18- by 8-in.) radiator.

SHF antenna thermal subsystem. - The steerable SHF antenna dishes represent a significant amount of area on the forward platform. To prevent the dishes from having a detrimental effect on the spacecraft temperatures, the antennas are enclosed with cylindrical, multilayer blanket shrouds. The top cover is a radome consisting of a single sheet of painted Kapton with a solar transmittance of ≤ 0.01. Both sides of the dishes are coated with vapor-deposited aluminum.

Battery thermal subsystem. - The temperature range for the batteries will be 0° to 35° C (32° to 121° F) and will be achieved by using Second-Surface Mirrors to increase the heat radiated from the battery mounting area. Thermal design requirements are

- To maintain uniform temperatures within each cell
- To maintain a temperature difference between cells of less than 2° C (3.6° F)
- To provide a conductance to the north panel of less than 5 Btu/(hr)($^{\circ}$C) (2.8 Btu/(hr)($^{\circ}$F))

The spacecraft thermal design will maintain the battery within the design operating temperature limits under these conditions.

The battery cells have a stainless-steel case of relatively poor thermal conductivity; therefore, aluminum heat sinks are used to increase heat transfer through the largest face. Thermal conductance to the heat sinks is maximized by clamping the heat sinks between the cells and bolting to the base plate, which has a fin connected to the north panel. Isolation from the aft platform is provided by standoffs. Dissipation requirements are specified as 0 to 5 watts for a charging rate of C/10 (A-hr capacity divided by charging time in hr) and 5.0 to 7.7 watts for a charging rate of C/20. Figure III-42 shows the thermal arrangement for the battery.

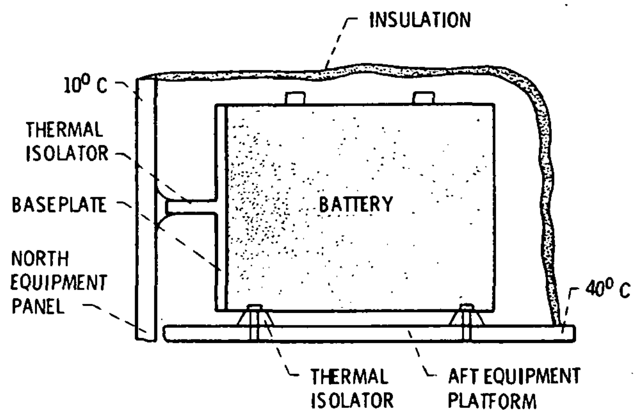


Figure III-42. - Battery thermal arrangement.

APOGEE MOTOR

The apogee motor mounted in the central thrust tube of the spacecraft is a solid-propellant motor with a high-expansion-ratio nozzle, pyrogen igniter, and electromechanical safe and arm device. (See fig. III-43 for a block diagram of the subsystem and fig. III-44 for a drawing of the entire assembly.)

Total impulse is 917 879 N-sec (206 358 lb-sec), which provides a velocity increment of 1746.2 m/sec (57 289 ft/sec) to the spacecraft. This velocity increment removes the orbital inclination and circularizes the orbit at synchronous altitude. The motor weighs 361 kg (796 lbm) before propellant is off loaded to optimize payload weight.

Propellant

The fuel is TP-H-3135 aluminized polybutadiene ammonium perchlorate. Maximum fuel weight (before off-loading) is 332.3 kg (732.7 lbm), providing a thrust of 26 901 N (6048 lbf).

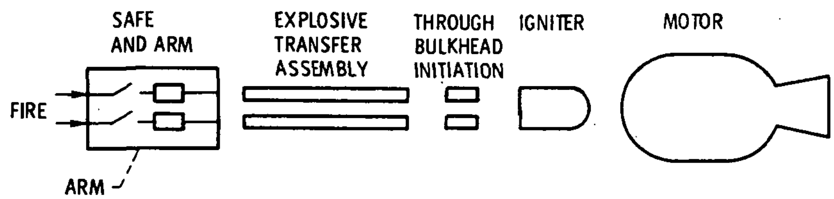


Figure III-43. - Apogee Motor Assembly block diagram.

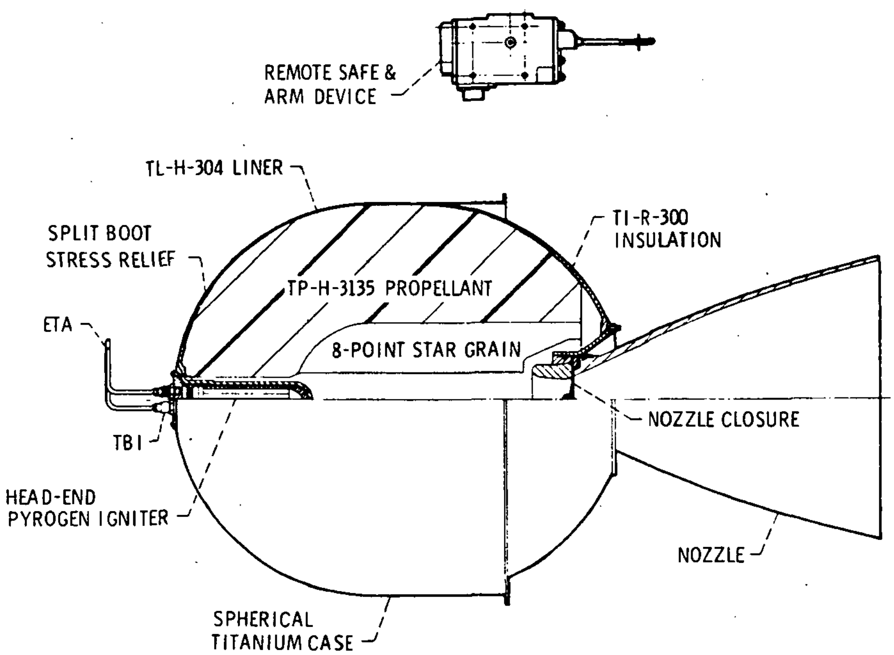


Figure III-44. - Apogee Motor Assembly.

Motor Case

The spherical case is fabricated from the titanium alloy Ti-6Al-4V, 68.5-cm (27.0-in.) outside diameter and contoured to a minimum wall thickness of 0.083 mm (0.327 in.). The exit cone throat diameter is 7.3 cm (2.87 in.), and the bell opening is 49-cm (19.29-in.) inner diameter. End-to-end length is 1.23 m (48.43 in.).

Safe and Arm Device

The remotely located safe and arm device (S&A) is an electromechanical ordnance device which, in the safe mode, mechanically and electrically prevents inadvertent or premature initiation of an igniter. Upon arming and application of the proper firing current, it initiates the explosive trains internal and external to the S&A and culminates in the initiation of a conventional igniter. The S&A can be electrically and mechanically safed. Arming can only be accomplished electrically. A safety pin is provided to prevent inadvertent arming of the S&A. With the safety pin in place and the S&A electric motor energized, the pin is locked in place to prevent arming and to prevent pin removal.

Explosive Transfer Assembly

These two devices consist essentially of silver-sheathed tubes approximately 125 cm (56.7 in.) long with 10 layers of fiberglass overwrap containing 0.82 grain per meter $\left(2\frac{1}{2}\text{ grains/ft}\right)$ of HNS II Grade B fuse-type pyrotechnic core. The assemblies serve to propagate ignition from the detonators to the pyrogen igniters within the propellant charge of the apogee motor itself. The system is redundant.

Off-Loading Capability

It is desirable to carry no more fuel than necessary to complete the mission. Although the motor is sized very nearly optimally for this mission, as much as 10 percent of the solid fuel may be removed - depending upon spacecraft actual weight and the predicted performance of the booster.

TRANSIENT EVENT COUNTER

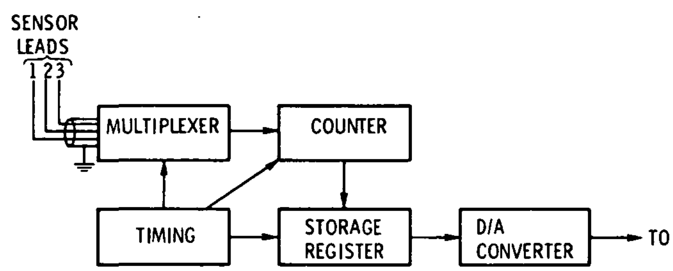
The Transient Event Counter (TEC) is a small electronic device to aid in the investigation of spacecraft charging phenomena. Many synchronous satellites have encountered anomalies which may have been caused by spacecraft charging. However, no previous satellites have carried instrumentation to specifically investigate this.

The TEC (fig. III-45(a)) is designed to sense and count transients which may be induced into spacecraft subsystems by electrical discharges due to spacecraft charging. Three sensors are located in close proximity, for electrical coupling, to the following subsystem harnesses:

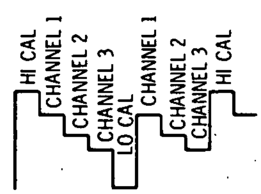
- Between NESA and ACE
- South Array Power Harness
- South Radiator Panel Instrumentation Harness

Each sensor provides an input to one of the three TEC channels. The channels are sampled sequentially, allowing about 1 second for each channel. If any transients are coupled into a sensor while that particular channel is being sampled, they will be counted (to a maximum of 63). The resulting count is read out through telemetry during the next sampling period. One frame of TEC data consists of the three sampling periods, followed by an internal calibration level. On every other frame the calibration alternates from high to low (fig. III-45(b)). This allows determination of which channel is being sampled without requiring telemetry synchronization.

Observation of the occurrence rate and general subsystem location of transients, in conjunction with other spacecraft data, should provide a better insight into the causes and effects of spacecraft charging.



(a) Block diagram.



(b) Typical telemetry output.

Figure III-45. - Transient Event Counter.

IV. LAUNCH VEHICLE DESCRIPTION

LAUNCH VEHICLE - GENERAL

The launch vehicle (fig. IV-1) is a three-stage Thor-Delta model 2914. The first-stage, liquid-propellant booster is powered by a gimbaled main engine and is initially augmented by nine externally mounted solid-propellant motors. The solid-propellant motors are jettisoned after burnout.

The second stage is a pressure-fed propulsion system. The thrust chamber assembly is mounted on a gimbal system for attitude control (pitch and yaw) during powered flight. Roll control during powered and coast flight and pitch and yaw control during coast are achieved by the second-stage cold-gas system. The second-stage guidance compartment structure houses the Flight Control System, the Inertial Guidance System, Instrumentation, the Range Safety System, and the Tracking and Power Systems.

The interstage connecting the second and third stages is a spin table, which supports the third-stage assembly. The third-stage propulsion system is a Thiokol TE-364-4 solid-propellant motor. An attachment fitting secured to the third-stage motor is used to support the spacecraft and includes the controls for event sequencing and for separating the spacecraft from the expended third-stage motor.

The spacecraft is mounted on top of the third-stage motor and is enclosed by a 2.4-meter (8-ft) diameter, two-piece aluminum fairing. The fairing structure, when assembled, forms a thin-walled housing of circular cross section and is jettisoned during second-stage powered flight at an altitude of about 120 km (65 n mi).

The launch site is Complex 17 at the Kennedy Space Center. Details of the launch site, the Eastern Test Range, and Complex 17 are shown in figures IV-2 and IV-3.

First Stage

The first stage consists of the Extended Long-Tank Thor. The engine is the Rocketdyne RS-27 configuration, which is a modification to the Rocketdyne H-1 engine and is rated at 911 887-N (205 000-lbf) thrust. The propellant burns to depletion in approximately 224 seconds.

Thrust augmentation consists of nine Thiokol Chemical Corporation solid motors designated "TX-354-5." Each set of three strap-on solids develops an additional

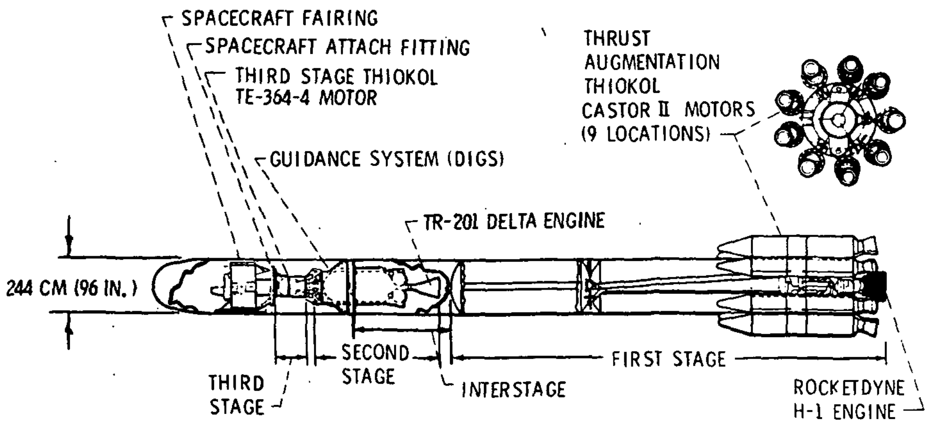


Figure IV-1. - Thor-Delta 2914 Launch Vehicle.

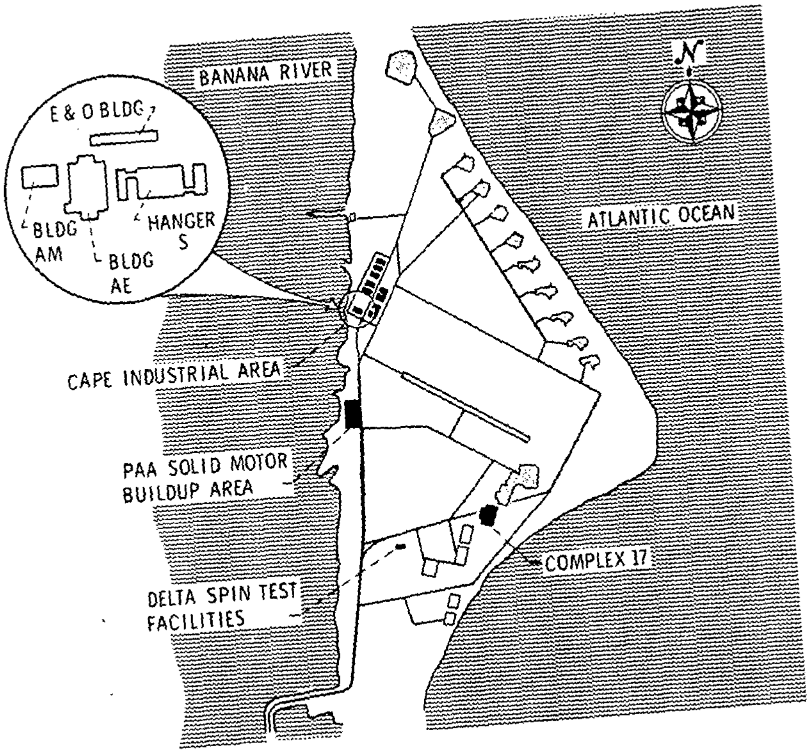


Figure IV-2. - Eastern Test Range map.

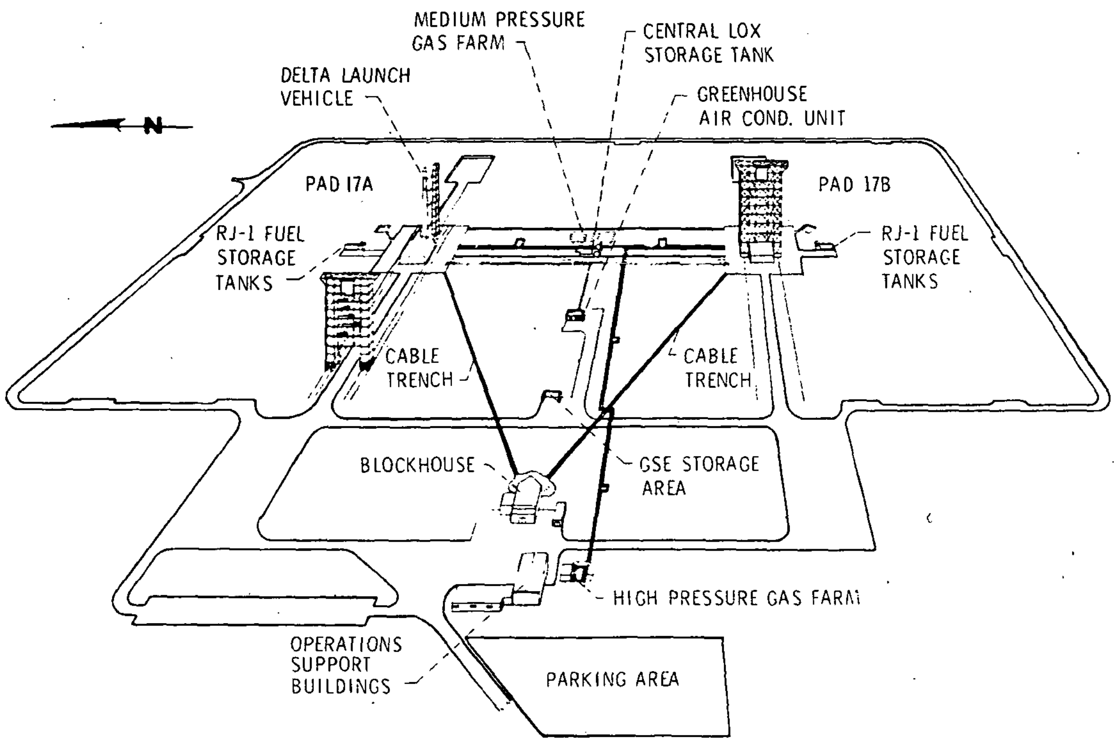


Figure IV-3. - Launch Complex 17.

467 064-N (105 000-lbf) thrust. Six thrust-augmentation solid motors are ignited at lift-off, and the remaining three are ignited at altitude approximately 40 seconds after lift-off. The expended motors are then jettisoned simultaneously about 90 seconds after lift-off.

Second Stage

The second-stage configuration uses the TRW Systems TR-201 propulsion system, which is restartable and operates on storable propellants. The engine delivers approximately 44 482-N (10 000-lbf) thrust and operates for about 350 seconds. This configuration comprises a 243.8-cm (96-in.) outside diameter skirt in which the 165.1-cm (65-in.) diameter stage is suspended. The outside skirt is attached to the first stage by a cylindrical structure. On top of the skirt a 243.8-cm (96-in.) diameter aluminum fairing protects the spacecraft and third-stage assembly. The first and second stages are separated by an explosive bolt and spring combination.

The second-stage thrust chamber assembly is mounted on a gimbal system for attitude control (pitch and yaw) during powered flight. A spin table forms the interface between the second and third stages, and the third stage is spun up to 60 rpm about 30 seconds after second-stage final cutoff. Eight spin rockets provide the necessary spinup torque.

Third Stage

The third-stage propulsion system is the Thiokol Chemical Corporation TE-364-4 solid-propellant motor and provides an average thrust of 66 590 N (14 970 lbf); burn time is about 44 seconds. After third-stage burnout, spacecraft separation is initiated. The 3731A attachment fitting connects the spacecraft with the third stage through a two-piece clamp band that is secured by two bolts. Spacecraft separation is effected by actuation of ordnance cutters to sever the two bolts. However, clamp design is such that cutting of either bolt will permit spacecraft separation. Springs are also provided to assist in retracting the clamp band and to retain the open clamp after release.

FAIRING

The Delta fairing is 792.2 cm (311.9 in.) long and 243.8 cm (96 in.) in diameter and weighs about 544 kg (1200 lbm) exclusive of thermal or acoustic insulation. It is an aluminum structure fabricated in two half-shells consisting of a hemispherical nose cap with a conical section tapering from the nose cap to the cylindrical section. The nose section is a ring-stiffened monocoque structure, half of which is fiberglass to permit RF transmission from the launch stand. The cylindrical section is an internally stiffened isogrid structure. The half-shells are joined by a contamination-free linear piston-cylinder thrusting separation system that runs longitudinally the full length of the fairing. The halves are jettisoned laterally.

V. SPACECRAFT TESTING

SCOPE

The test program uses three spacecraft: a Dynamic/Thermal Model, an Engineering Model, and a Protoflight Model. The Protoflight Model will be qualification tested prior to launch. The Protoflight Model will be stressed to levels that exceed flight values wherever possible. After the qualification program the Protoflight Model will go through a series of flight readiness tests to verify that the spacecraft has not degraded and is, indeed, ready for launch.

The testing of the Communications Technology Satellite follows the principles which have been used successfully in prior programs by NASA and the Canadian Department of Communications (DOC). The major feature of this philosophy is a heavy emphasis on complete spacecraft systems testing. The integrated spacecraft will be environmentally tested at CRC, Lewis, and GSFC. These tests will be carried out in conformance with the test plans approved by the Lewis and CRC CTS Project Managers. Test specifications and test procedures are prepared by CRC, with the Transmitter Experiment Package and facility test sections furnished by Lewis, GSFC, and KSC. At NASA facilities, test procedures are executed jointly by Lewis and CRC. CRC performs spacecraft testing at all other locations. The CTS schedule and test locations are summarized briefly in table V-1.

SPACECRAFT TESTING FACILITIES

In addition to the testing performed on the various spacecraft models at CRC, a number of NASA facilities are employed in the development and qualification of the CTS. These include the Vibration and Shock Facility at Lewis (fig. V-1) for dynamics testing of the Dynamic/Thermal Model. The Thermal Vacuum Facility (4.6 m by 18.3 m; 15 ft by 60 ft) at Lewis (fig. V-2) is used for thermal testing of the Dynamic/Thermal, Engineering, and Protoflight Models, as well as for subsystem testing of the south panel and the Deployable Solar Array.

At GSFC, the Acceleration Facility is used for dynamics tests of the Dynamic/Thermal Model. The Solar Simulation Chamber (fig. V-3) is used for thermal tests of the Dynamic/Thermal, Engineering, and Protoflight Models. STDN compatibility is

TABLE V-1. - SUMMARY OF CTS TESTS

Model	Facility			
	Communications Research Centre (Canada)	Lewis Research Center	Goddard Space Flight Center	Kennedy Space Center
Dynamic/Thermal	Integration testing, August 1972	Vibration and shock testing, October 1972 Thermal vacuum chamber compatibility testing, March 1973	Acceleration testing, November 1972 Solar simulation testing, January-February 1973	
Engineering	Integration checkout, March-September 1974 Electrical compatibility testing, September-November 1974 Vibration and acoustic testing, April-May 1975	Thermal vacuum testing, November-December 1974	Solar simulation and STDN compatibility testing, December 1974-February 1975	
Protoflight	Integration and electrical compatibility testing, October 1974-July 1975 Magnetic moment, vibration, and shock testing, July-August 1975	Thermal vacuum testing, September 1975	Solar simulation STDN compatibility, electromagnetic compatibility, static and dynamic balance, and mass properties measurements; October-November 1975	Range operations, November-December 1975 Launch, Jan. 1976

also tested at GSFC for the Engineering and Protoflight Models. Electromagnetic compatibility, static and dynamic balance, and mass properties measurements are made at GSFC.

DYNAMIC/THERMAL MODEL SPACECRAFT TESTING

Structural tests of the Dynamic/Thermal Model were performed in a full-scale structure in which mass dummy components were mounted. Their primary purpose was

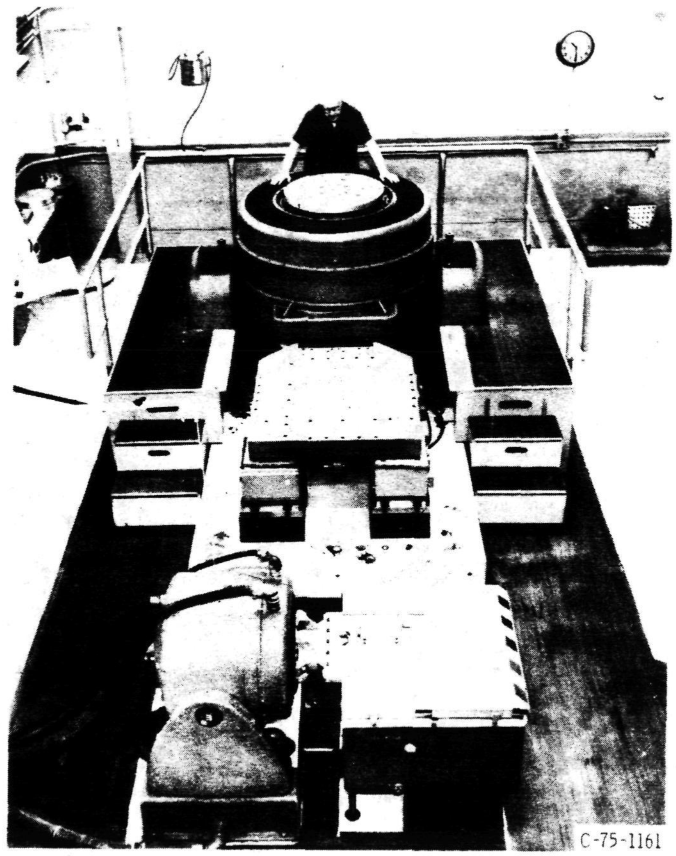


Figure V-1. - Lewis Vibration and Shock Facility.

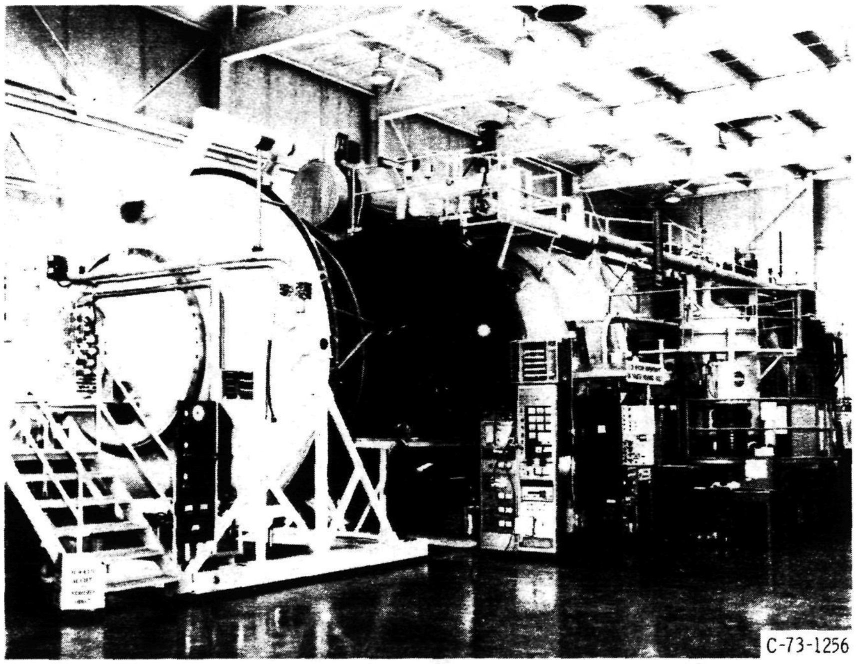


Figure V-2. - Lewis Thermal Vacuum Facility.

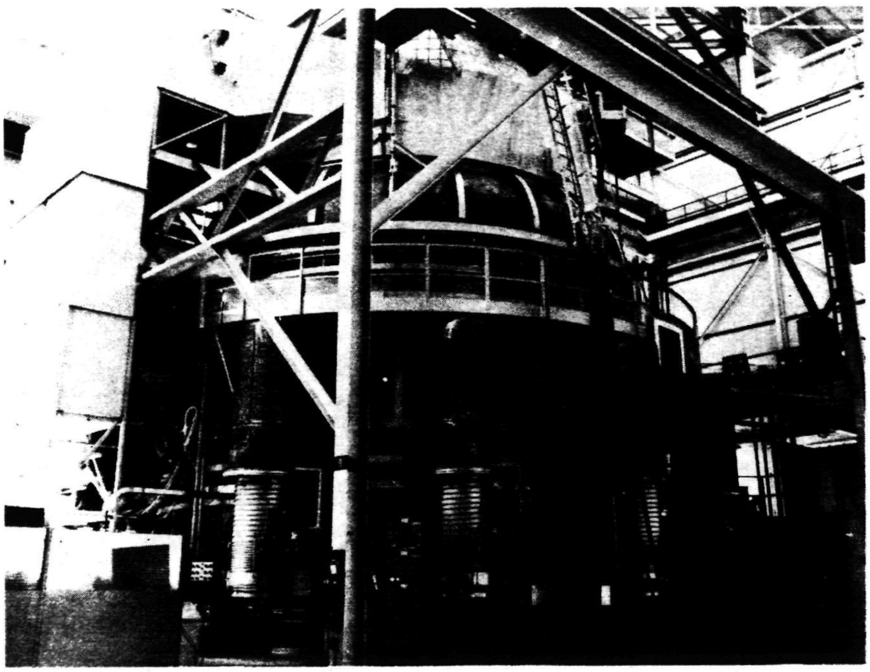


Figure V-3. - Goddard Solar Simulation Chamber.

to demonstrate the structural integrity and thermal balance of the spacecraft design and to provide test data for further development of the spacecraft structural and thermal subsystems. After completion of the structural tests, the spacecraft was converted to a thermal model in which thermal paths, component heat capacity, and heat dissipation were simulated. The following tests were performed:

- Alinement verification - Mechanical alinement of critical spacecraft surfaces was determined to assure pointing accuracy of various sensors and nozzles.
- Vibration test - The capability of the spacecraft structure to survive the qualification testing environment was evaluated and data on structural resonant frequencies and dynamic behavior were provided.
- Shock test - Spacecraft structural elements were not damaged by planned pyrotechnic events such as spacecraft separation and solar array protective cover separation.
- Acceleration test - The spacecraft structure was stressed to demonstrate adequate structural design under the most severe quasi-static load expected during the launch phase.
- Thermal vacuum test - The capability of the Lewis facility for conducting spacecraft thermal vacuum testing was demonstrated. Fits, clearances, and mounting were checked in preparation for Engineering Model Testing.
- Solar simulation test - The capability of the spacecraft thermal control system to maintain component temperatures within the required envelope during synchronous and transfer-orbit simulation phases of the mission was demonstrated.

ENGINEERING MODEL SPACECRAFT TESTING

This portion of the test program developed and demonstrated complete operational capability and intersystem compatibility of the spacecraft design to perform to the mission requirements. The tests were conducted on an electrically functional spacecraft capable of operating in all modes predicted for mission operation. The structure, component placement, and thermal control system are identical to the flight configuration. Components are functionally identical to the flight components; however, some are not potted and internal mechanical construction is not identical to the flight configuration in all components. The following tests were performed:

- Integration checkout and electrical compatibility tests - The spacecraft subsystems, as they are installed on the spacecraft, operated properly both individually and in combination with other systems.
- Thermal vacuum test - Proper spacecraft systems performance was demonstrated and evaluated in vacuum under all probable operating configurations over the entire specified temperature range.

- Solar simulation test - Thermal data were provided to verify spacecraft thermal design, and the effects of solar input to spacecraft equipment panels and structure conductivity during synchronous and drift phases were confirmed.
- Spaceflight Tracking and Data Network (STDN) compatibility test - Compatibility of hardware and software signals with STDN equipment was demonstrated.
- Performance verification tests - Functional tests performed at the beginning, during, and at the end of each test demonstrated spacecraft capability to perform within acceptable limits under test conditions.
- Vibration, shock, and acoustic tests - Capability to perform within acceptable limits under conditions of mechanical stress was verified.

PROTOFLIGHT MODEL SPACECRAFT QUALIFICATION TESTING

These Protoflight Model qualification tests are performed on the spacecraft intended for launch and will provide environments more severe than those expected in flight. This phase of the test program demonstrates the adequacy of the spacecraft to operate under the expected environmental conditions.

- Integration checkout and electrical compatibility tests - The spacecraft subsystems as they are installed on the spacecraft must operate properly both individually and in combination with other systems.
- Alinement verification tests - The mechanical alinement of the critical surfaces of the spacecraft is determined; proper alinement of such items as the earth sensor, the momentum wheel, the reaction control nozzles, and the gyros is assured.
- Vibration test - The spacecraft and components must successfully endure the qualification level sine and random vibration tests.
- Shock test - The spacecraft and components must survive the shock of detonating the spacecraft pyrotechnic devices.
- Leak tests - Various spacecraft subsystems are tested for leakage subsequent to each spacecraft exposure to vibration or vacuum conditions.
- Thermal vacuum test - Capability of spacecraft operation in vacuum over a temperature range greater than that expected in flight is demonstrated. SHF transponder communications tests are performed as part of this test.
- Corona checks - Outgassing tests are conducted to demonstrate the minimum time to turn on of all high-voltage and high-frequency components when they are exposed to vacuum.
- Performance verification tests - Spacecraft readiness and capability to perform within acceptable limits under extreme test conditions with satisfactory recovery upon return to nominal test conditions are demonstrated.

PROTOFLIGHT MODEL SPACECRAFT FLIGHT READINESS TESTING

The Protoflight Model flight readiness tests are conducted after the qualification tests to prepare the spacecraft for flight and to demonstrate that the spacecraft is ready for launching.

- Combined solar simulation and STDN compatibility test - The capability of the spacecraft to perform as required within allowable temperature limits in orbit and its compatibility with STDN equipment are verified.
- Final alinement verification test - A detailed set of measurements are taken prior to launch to determine mechanical alinement of critical surfaces for sensor and control accuracy.
- STDN compatibility tests - The adequacy of STDN tracking, telemetry, and command parameters and the adequacy of operational procedures for the intended mission requirements are demonstrated.
- Magnetic moment measurement test - The residual magnetic moment of the spacecraft is measured to permit calculating the magnetic moment disturbance torque prevailing in orbit.
- Electromagnetic compatibility test - The spacecraft must have no spurious RF emissions which are likely to compromise the performance of the launch vehicle or support equipment.
- Static and dynamic balance test - Data are provided to permit minimizing the nutation and coning motions induced in the spacecraft during the spin-stabilized phase of flight.
- Mass properties measurements test - The weight, center of gravity, and moments of inertia of the spacecraft are determined.
- Range operations tests (at KSC) - The spacecraft must have successfully survived shipment to the range and be ready for mating to the vehicle and for subsequent launch.

VI. MISSION OPERATIONS

MISSION OPERATIONS OVERVIEW

General functions of the mission operations activity include planning and coordination among the several agencies involved in placing the CTS into a geostationary synchronous orbit. Documents are prepared, specific tasks related to operations are coordinated, and such areas as launch vehicle preparations and interface documents, mission analysis, ground station interfacing, and tape simulations are surveyed.

The mission has been separated into the following six operational phases, the first four of which are the responsibility of NASA:

- Phase 1 - Prelaunch operations: all operations from flight readiness of the spacecraft to the start of countdown - including software implementation, mission analysis, trajectory optimization, personnel launch readiness, and launch site operations
- Phase 2 - Launch operations: all operations performed throughout the countdown and through separation of the spacecraft from the vehicle
- Phase 3 - Transfer-orbit operations: operations from separation through apogee-motor firing - including precession (pointing) maneuvers, selection of apogee for firing, evaluations of high-thrust engines, spin control, nutation damper evaluation, spin sensor evaluation, and apogee-motor firing
- Phase 4 - Drift operations: covers the period from apogee-motor firing to spacecraft handover to CRC mission control - including operations which trim the orbit and place the spacecraft at the appropriate longitude, spinning at 60 rpm with the orbit parameters at acceptable values
- Phase 5 - Attitude acquisition: a CRC responsibility - including despinning the spacecraft, deploying the solar arrays, and acquiring the earth and the sun for three-axis stable operations
- Phase 6 - In-orbit operations: a CRC responsibility, which includes data handling, housekeeping, and experiment operations as well as performance evaluation of communications for the 2-year mission

The existing NASA Spaceflight Tracking and Data Network (STDN) is used through the fourth phase of the mission for all operational support. This network also serves as a backup facility when requested and during the critical fifth phase of operations.

MISSION DESIGN

Goddard Space Flight Center (GSFC) has the responsibility for providing the mission design required to perform the NASA phases of the mission. GSFC's Mission and Data Operations Directorate is performing the various analyses to provide an optimized mission design of CTS. This includes maximizing the handover weight and minimizing the use of hydrazine fuel, as well as recommending the apogee for apogee-boost-motor firing. During the transfer-orbit and station-acquisition phases, GSFC will actively perform attitude and orbit determination and recommend procedures and maneuver time lines to achieve station acquisition and spacecraft handover to CRC.

Major elements of the CTS operations ground systems are

- GSFC NASA Communications Network (NASCOM) and STDN Stations
- CRC Ground Control Station
- Lewis Ground Station
- Lewis Experiment Evaluation Center
- Mission Control Center at GSFC (for CTS)
- CTS Launch Control Room (NASA Phases at GSFC)

OPERATIONS CONTROL CENTER

Functional Requirements

The GSFC Multi-Satellite Operations Control Center (MSOCC) is required to control the spacecraft in real time from shortly after launch through the operations required to place the spinning spacecraft in a geostationary orbit at the required longitude. These operations include orientation to the apogee-motor firing attitude, apogee-motor firing, attitude reorientation of the spin axis perpendicular to the orbit plane, and drift to station. Control will be transferred to the CRC control center in Ottawa, Canada, after the spacecraft is placed on station. The MSOCC will serve as a backup to Ottawa to respond to special requests including real-time support through spacecraft three-axis stabilization. During the 2-year mission, STDN telemetry data acquisition and distribution will be required as backup on a requested basis.

During the launch phases described, selected spacecraft status information will be displayed in the MSOCC and updated as available. Data will be processed and distributed as quick-look information - 4 to 12 hours of real-time data per day - until the spacecraft drifts to station. Unusual events will not be restricted to a time frame but will be reported as dictated by the urgency of the situation and the need for a quick response. Voice and data circuitry between MSOCC and participating stations will be used.

During the 2-year mission, the MSOCC will be prepared to command the CTS into a safe operational mode if the CRC station were to fail at a critical time.

Telemetry Data Acquisition

The Telemetry, Tracking, and Command System is activated on the launch pad and is supplied by the battery until the Body Solar Array can supplement the power later in the mission. The first planned acquisition of data for purposes other than monitoring temperatures and voltages is a "Spacecraft Separation from Third Stage" monitor, which introduces a jump discontinuity in the output of the Separation Ring temperature. These data are monitored by a mobile tracking station. The first STDN station to acquire data will be Orroral, Australia. From this data acquisition through the fifth apogee (nominally), the STDN station visibility chart (fig. II-1) will apply.

Telemetry Data Processing and Display

During the NASA phases, the data flow is as follows:

- Data are received by a STDN ground station and blocked to the 7.2-kilobit GSFC format.
- Blocked data are received by GSFC NASA Communications Network (NASCOM) through data lines which process it into appropriate formats for the Launch Control Room.
- The blocked data are rerouted, by NASCOM, to the Lewis Ground Station and the CRC Ground Station for remote processing into individualized formats for specific uses.

Special processing at GSFC for the NASA phases is accomplished by specialized software:

- Launch-phase software
- Orbit-determination software
- Tailoring of attitude-determination and precession requirements to CTS spinning spacecraft
- Apogee-Motor Firing Attitude (AMFA) computational software
- Spin-axis precession monitoring
- Station-acquisition maneuver computations subsequent to apogee-motor firing
- Data display formats at the NASA Mission Control Room (MCR)
- Other interfaces and associated software packages, which provide data outputs in convenient formats for project support

- Software required to process spacecraft data into formats for spacecraft health status

Launch Control Room

The data processed at GSFC are formatted and displayed on monitors in the Launch Control Room (LCR). The LCR serves as a typical project operations control center and is capable of the following functions with the aid of MSOCC:

- Accepts and processes the 7.3-kilobit/second blocked data into the 1000-bit/second digital stream and then into the 1-bit status "flags," 8-bit analog channels, and the more complex digital words of 24 bits each
- Monitors either encoder (120 flags, 156 analog channels) for acceptable operating limits and displays selected parameters and out-of-tolerance parameters
- Continuously displays spacecraft configuration and identifies changes of state
- Provides for real-time determination and display of information concerning spacecraft attitude and related constraints (i.e., power, thermal, and fuel)
- Provides time-history plots of selected parameters
- Records spacecraft data in real time for immediate diagnostic playback
- Verifies spacecraft commands, if required, prior to execution and verifies proper functional response after execution
- Provides spacecraft command capability, including command sequence generation and transmission to spacecraft or the appropriate ground station and requests spacecraft commands
- Provides communications tie-in to participating ground stations, including Lewis and CRC, by NASCOM
- Records, processes, and supplies mission data for dissemination and to be used in generation of flight reports and subsystem performance reports

A CTS data flow chart is shown in figure VI-1.

Command Operations

GSFC will verify the ability to generate and send commands by assigned STDN ground stations directly or by remote control generated at MSOCC. Commands will be by voice or directly from the keyboard at LCR. The assigned CTS ground station will be required to transmit commands to the spacecraft in real time or at prescheduled times via teletype and/or voice instructions from GSFC or CRC depending on the phase of the mission.

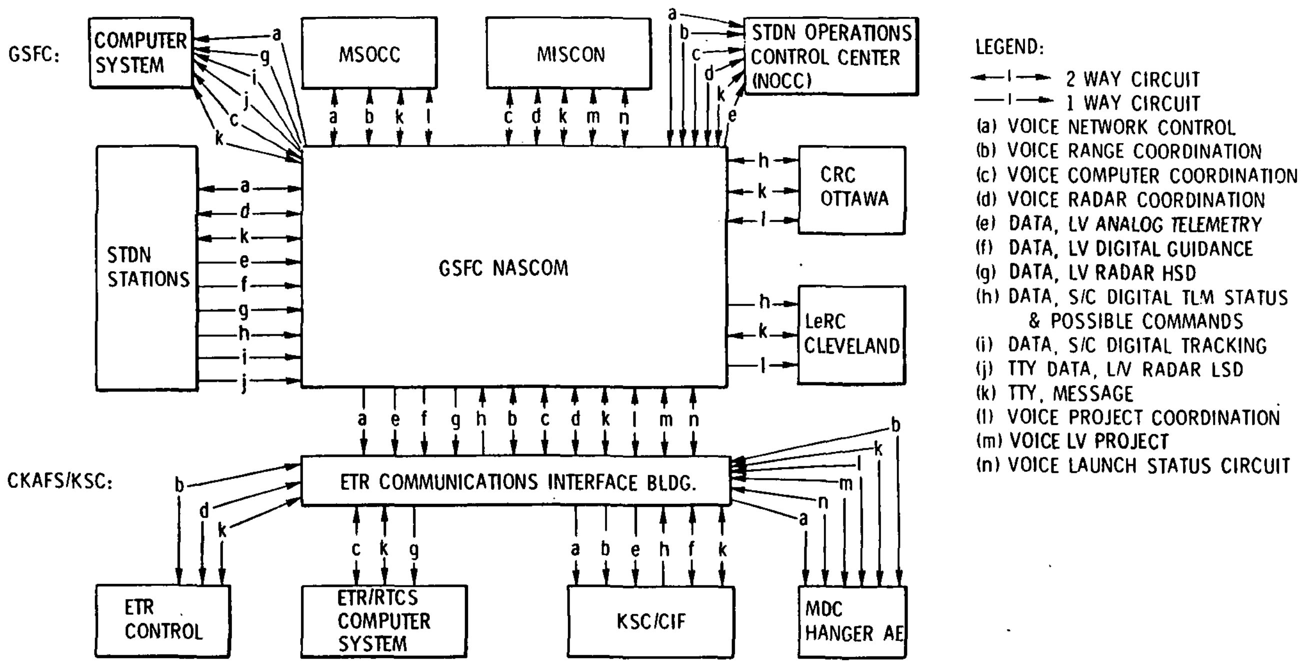


Figure VI-1. - CTS data flow chart.

Ranging

In order that GSFC support the ranging requirement, the minimum flux density for the command ranging carrier at a frequency of 2097.198 MHz will be 93.2 dBm per square meter (8.66 dBm/ft^2) at the spacecraft.

On-Station Backup

The function of NASA mission control and data operations is to place the spacecraft on station. Thereafter, NASA mission control will serve for planned backup support purposes only. This backup support involves monitoring critical parameters (voltages, currents, temperatures, flags, etc.) by using programs developed in support of the transfer-orbit phase of the mission and command capability. The backup support is available for 24 months, if required. The STDN active requirement will be no longer than 1 month from launch to on-station (nominally).

LIMIT CHECKING AND CALIBRATION

The first step in most spacecraft and experiment status monitoring will be limit checking of the spacecraft parameters in the normal telemetry stream and conversion of the raw telemetry data to calibrated engineering units for further processing or direct display. The limit checking function will be dynamic in using limit sets for the analog values. These values are selected according to spacecraft status information. In addition the limit checking will include special computations on specific functions of more than a single frame of telemetry.

CONFIGURATION STATUS MONITORING AND DISPLAY

The commanded status of the spacecraft will be monitored during each telemetry frame at LCR and MCR. This monitoring will be accomplished by a computer display of the changes in the current spacecraft configuration. In addition, any of the raw telemetry values can be displayed for operator monitoring.

The attitude of the CTS spacecraft for the spinning phase will be computed in real time at GSFC. The output of the attitude determination program will be used in conjunction with several other computer programs to display actual attitude in the Flight Dynamics Operation Control Center, to support various operations to check out-of-limits conditions, and to provide data for command control. Sensor data will be available in the LCR and MCR.

TIME LINE

A GSFC-developed mission design nominal time line for the CTS mission is given in table VI-1. At the appropriate time, with acceptable orbital elements, the control of the spacecraft is turned over to CRC for three-axis acquisition. At that time the NASA phase of the mission, excluding experiments and users, is complete. The acceptable orbital elements are inclination drift and spacecraft east-west and north-south drift, of which only east-west drift is to be controlled or limited.

TABLE VI-1. - NOMINAL TIME LINE

(a) Time from transfer-orbit injection

Time	Orbit	Event	New attitude		Duration of maneuver	Fuel usage		Approximate period of radio silence per orbit
			Right ascension, deg	Declination, deg		kg	lbm	
0^h0^m	1	Injection	150.0	-22.06		----	-----	1^h
2^h30^m		Sun sensor switching						
15^h12^m	1	Maneuver to verification attitude	-----	------	1^m59^s	0.12	0.26	
15^h45^m	2	Maneuver to apogee-motor firing attitude	333.9	-22.5	35^m12^s	2.09	4.60	1^h
	3	Collection and analysis of data						
36^h to 40^h	4	Trim maneuver	333.9	-22.5		[a].01	[a].02	1^h
48^h	5	Apogee-motor firing (5th apogee)	333.9	-22.5	$0^m35.8^s$			

(b) Time from apogee-motor firing

Time	Orbit	Event	Right ascension, deg	Declination, deg	Duration of maneuver	kg	lbm	Approximate period of radio silence per orbit
0^h0^m	1	Apogee-motor firing	333.9	-22.5	$0^m35.8^s$			[b]4.5^h
2^h0^m	1	Maneuver to 15° off orbit	347.1	-74.1	7^m54^s	0.47	1.03	0^h
26^h0^m	2	Maneuver to orbit normal	0	-89.1	2^m16^s	.14	.30	0^h
35^h to 304^h	[c]2 to 13	Four station-acquisition maneuvers	0	-89.1	103^m	5.99	13.2	0^h

[a]Per degree.
[b]If no maneuver.
[c]3-Sigma value, 25 to 30 days.

CHECKOUT OF SPACECRAFT

When the spacecraft is properly oriented with respect to sun and earth and the subsystems are responding properly, the spacecraft will undergo a limited series of checkout procedures. The Transmitter Experiment Package (TEP) special instrumentation is energized to assess temperatures. The first momentum dumping procedure is exercised and evaluated. Orbital elements and the orbit-dependent parameters are determined through tracking and analysis on an off-line computer basis during these activities. Electrical, thermal, optical, and mechanical parameters are periodically evaluated to determine the environmental adequacy and safety of the operating spacecraft subsystems. An east-west stationkeeping maneuver is conducted to evaluate the Attitude Control System (ACS), Reaction Control System (RCS), and value command functions, as well as to evaluate Attitude Control Electronics (ACE) parameter set changes. Earth sensors are switched to determine biases and the primary system is selected to minimize sun and moon interference effects.

The solar array blanket tension is determined, and the negator spring is adjusted accordingly. The array tracking steppers are optimized to minimize blanket resonance by alternate steps. When the power profile is determined to be optimal, the batteries undergo a discharge-recharge cycle. As part of spacecraft checkout, the SHF transponder is turned on and configured for TEP operation. The initial SHF Communications Technology Experiment is conducted to verify proper operation prior to configuring the spacecraft for first eclipse season.

USER SCHEDULING

Once it is through the eclipse period, the spacecraft is reconfigured to support U.S. and Canadian experimenters. The sharing of the spacecraft is accomplished by a predetermined time-sharing schedule. The sharing arrangement is given in table VI-2.

TABLE VI-2. - TIME-SHARING ALLOCATION

Time period	Days of week[a]	
	U.S.	Canada
Initial experiments period	(b)	(b)
May 9, 1976 - July 3, 1976[c]	Mon., Wed., Fri.	Tues., Thurs., Sat.
July 4, 1976 - October 2, 1976	Tues., Thurs., Sat.	Mon., Wed., Fri.
October 3, 1976 - January 1, 1977	Mon., Wed., Fri.	Tues., Thurs., Sat.

[a]Start of day, 04:00 est; Sundays alternate.

[b]Technical experiments and communication experiment (CE): Start, Feb. 22, 1976 (launch, Jan. 13, 1976).

[c]Communication experiment full time, May 9, 1976.

VII. GROUND TERMINALS

GENERAL

Prior to 1971 the only frequency bands allocated for satellite communications were below 9 GHz. Bands at 4 and 6 GHz are being used extensively in international systems such as INTELSAT and are also used by Telesat Canada for the Canadian domestic satellite system. These bands are subject to sharing constraints and power flux density limits by international agreement. In July 1971 the International Telecommunications Union World Administrative Radio Conference on Space Telecommunications allocated a number of new frequency bands to space services. The CTS band allocation (12 and 14 GHz) is one of these.

The 12- and 14-GHz bands are not allocated exclusively to space services; terrestrial services, by international agreement, may be introduced after the approval of plans for space services to ensure compatibility of the use of the band within each country. In fact, the ability to locate ground stations close to the user and thus avoid coordination problems often encountered in the lower frequency shared bands may be one of the more important features of the use of the band by domestic systems with large numbers of terminals. Communications experiments using CTS will be directed at investigating those applications that take particular advantage of the use of the 12- and 14-GHz bands and the high satellite Effective Isotropic Radiated Power (EIRP). Ground station development within the CTS program will tend to concentrate on terminals that have small-diameter antennas, are modest in cost, and can be located close to the user.

A unique feature of the spacecraft transponder is that one of the two communications channels is equipped with a high-power amplifier of 200 watts, compared with a 20-watt amplifier for the second channel. This permits the satellite to operate in a broadcast mode where one relatively large station can broadcast through this channel to any number of smaller stations within the coverage area of the high-power beam. The technical requirements of the ground terminals and consequently the cost are reduced considerably from the case of receiving the same quality signal from the low-power channel. In Canada the ground terminals are provided by the government for user experiments and, in general, are installed and maintained by the government. In the U.S. the ground terminals are provided by the user and operated and maintained by the user.

The following sections describe both Canadian and U.S. ground terminals associated with the multitude of experiments planned. Terminal characteristics vary widely:

antenna diameters range from 0.8 to 9 meters (32 in. to 30 ft), transmitter powers vary from 20 to 1200 watts, and both tracking and nontracking types are used.

CANADIAN GROUND TERMINALS

The Communications Research Centre will have a complement of ground terminals which will be made available to experimenters to conduct the CTS communications experimental program. This section describes these terminals in terms of capabilities, configuration, and performance. The terminals which will be available to experimenters include

- One 9-meter (30-ft) Ottawa ground terminal
- Two 3-meter (10-ft) transportable ground terminals
- Eight 2-meter (7-ft) ground terminals
- Eight 1-meter (32-in.) ground terminals

In general, most experimenters will interface with the terminals at baseband with such signals as television video and audio, telephony, and sound program. An Intermediate Frequency (IF) interface is available to permit interfacing of special digital modulation and demodulation equipment.

Television or sound program origination, monitoring, or distribution equipment (e.g., television cameras, tape recorders, and television receivers) are not supplied as part of the ground terminals and are the responsibility of the experimenters. The interfacing of the telephony and sound program inputs and outputs of the terminals to telephone or other systems, if required, is also the responsibility of the experimenters. Similarly, interfacing experimenter's modulators and demodulators at the IF interface will be the responsibility of the experimenter. In all cases, the experimenter must ensure that any external equipment connected into the interfaces of the terminals is compatible with the terminal equipment and meets all interface requirements.

Nine-Meter Ottawa Ground Terminal

The 9-meter (30-ft) ground terminal is the largest of the terminals to be provided for CTS experiments. It is permanently installed at the Communications Research Centre, Shirley Bay, Ottawa. The terminal is equipped for all types of signals envisaged for the CTS experiments, including television transmission and reception and sound program transmission and reception. In addition, it has the necessary equipment to act as a network control terminal for the telephone system.

As illustrated in figure VII-1, the terminal is composed of two major parts: an antenna and a control building. The control building contains the operating console for

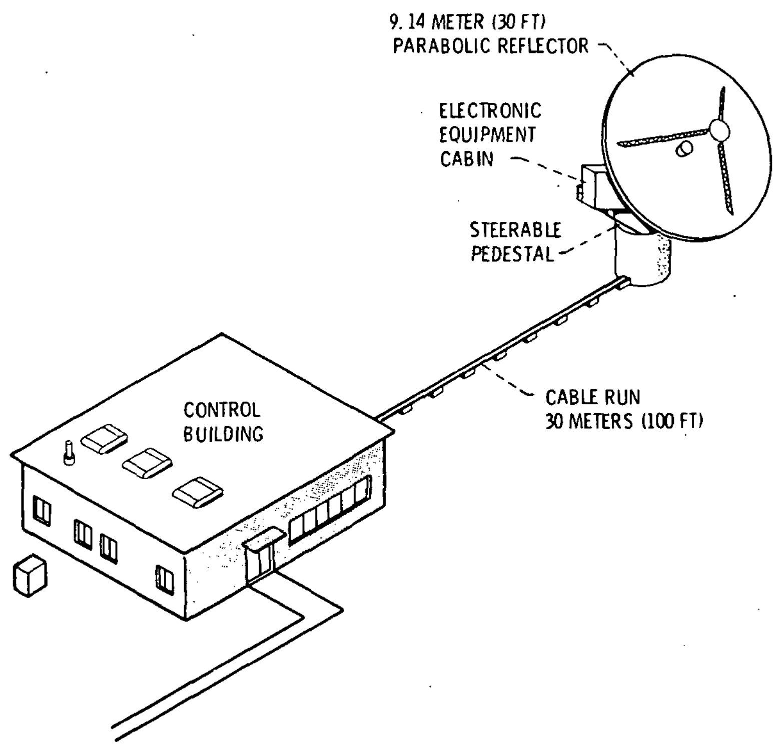


Figure VII-1. - 9-Meter Ottawa ground terminal.

the antenna and some of the ground communications equipment. The complete terminal is operated from the control building. All interfacing of the experimenter's equipment is to be done at the communications equipment racks within the control building.

Signal types and interfaces. - This section describes the available signal types and some of the interface characteristics. The experimenter must deliver his signals to and from these interfaces and must, therefore, provide all necessary cabling, terrestrial links, etc.

- One television signal including as many as three television audio channels can be transmitted to 2- or 3-meter (7- or 10-ft) terminals.
- One sound program signal can be transmitted to 1- or 2-meter (32-in. or 7-ft) terminals equipped with the optional sound program receiving unit.
- Up to a maximum of nine 1- or 2-meter (32-in. or 7-ft) terminals may simultaneously intercommunicate within the telephone system under the network control provided by the 9-meter (30-ft) terminal.
- An IF interface with a bandwidth of 85 MHz will be available for special signal applications.
- One television signal including as many as three television audio channels can be received from a 3-meter (10-ft) terminal.
- One sound program signal can be received from a small terminal equipped with the optional sound program transmitting unit.
- Up to a maximum of nine simultaneous telephone channels can be received from each of any small terminals.

Terminal characteristics. - The major terminal characteristics are defined by the following two parameters:

- System gain/temperature (G/T), 32.9 dB
- Effective Isotropic Radiated Power (EIRP), 81.7 dBW

Figure VII-2 is a block diagram of the 9-meter (30-ft) terminal's communication equipment. The basic equipment is essentially identical to that provided for the 3-meter (10-ft) terminals. An experimenter can interface with the terminal at three locations, depending on the specific requirements of the experiment. These include a baseband interface patch panel for television video and audio, a baseband patch panel for the telephone system, and an IF interface connection panel.

Three-Meter Transportable Ground Terminal

The 3-meter (10-ft) transportable ground terminal is a self-contained terminal. It will provide most of the services provided by the 9-meter (30-ft) Ottawa ground terminal for applications where it is not feasible or desirable to use the Ottawa terminal. It will be usable for television broadcast transmission or reception and will be equipped to

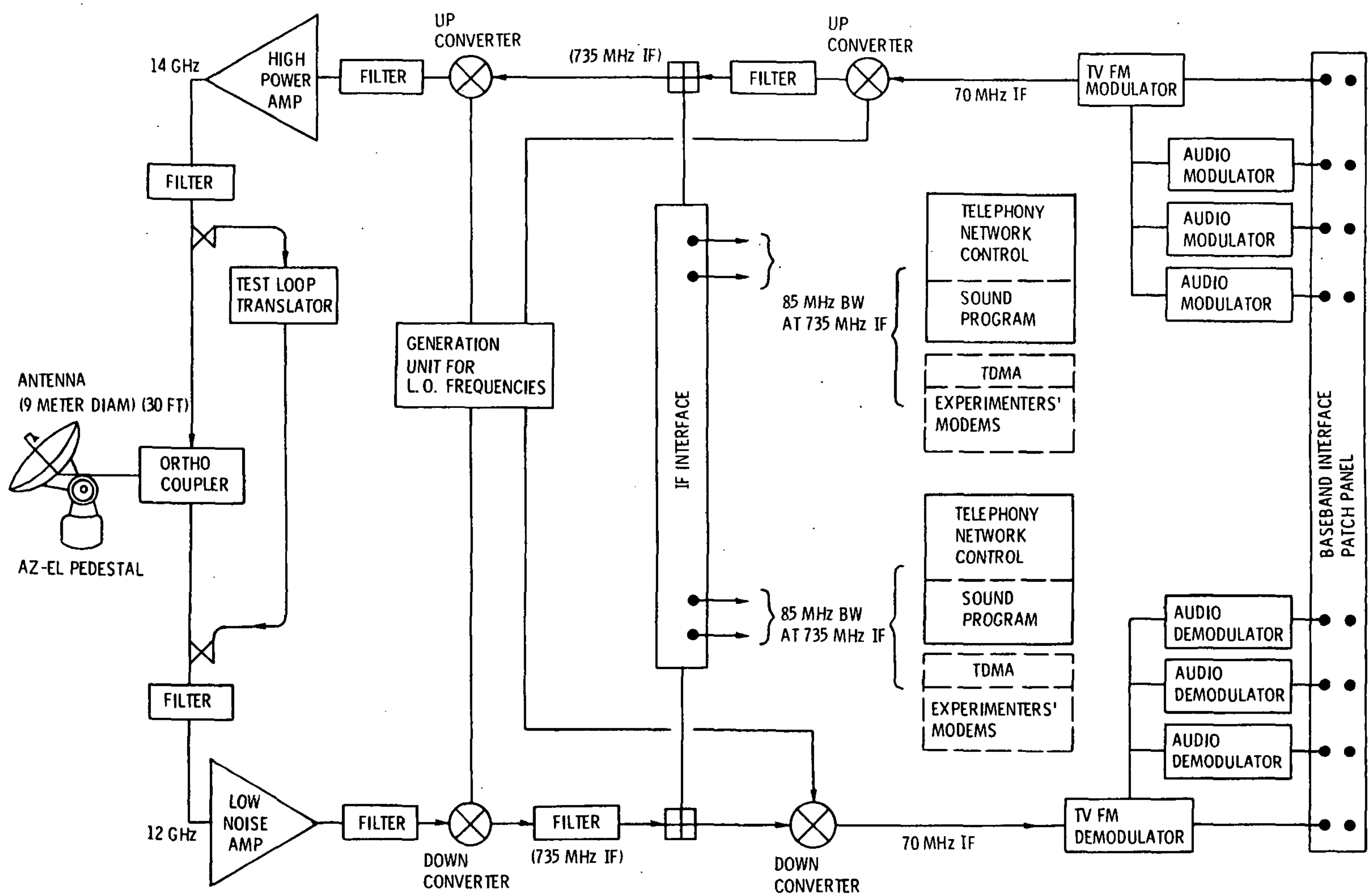


Figure VII-2. - 9-Meter terminal block diagram.

provide network control for telephone operation with smaller terminals.

Two terminals are being purchased. Both are contained in trailers (fig. VII-3). The trailers are towable for highway transport and can also be transported by rail, semi-trailer truck, and Hercules aircraft. They can be installed and operated by as few as two men. One terminal has been specially configured such that the communications equipment racks inside the trailer and the antenna with its associated mount can be packaged to permit transport by small aircraft (Twin Otter) to allow operation in remote and inaccessible regions. In this "small aircraft" configuration, the terminal equipment is estimated to weigh about 2270 kg (5000 lbm) and is composed of about 20 individual packages ranging in weight from 45 to 136 kg (100 to 300 lbm).

Signal types and interfaces. - A weatherproof interface junction box will be provided on the exterior of the trailer for use by the experimenter. In the small-aircraft configuration, interfacing is provided by patch panels housed in the experimenter's facilities.

In the standard mode of operation for television broadcast, sound program, and telephone network control, the spacecraft low-power beam must be centered on the terminal. The following signal types will be accommodated:

- One television signal including as many as three television audio channels can be transmitted to 2-meter (7-ft) terminals, the 9-meter (30-ft) terminal, or the other 3-meter (10-ft) terminal.
- One sound program signal can be transmitted to small terminals equipped with an optional sound program receiving unit.
- A maximum of nine small terminals may simultaneously communicate within the telephone system under the network control provided by one 3-meter (10-ft) terminal.
- An IF interface with a bandwidth of 85 MHz will be available for special signal applications.
- One television signal including as many as three television audio channels can be received from a 9-meter (30-ft) terminal or the second 3-meter (10-ft) terminal.
- One sound program signal can be received from a 3-meter (10-ft) terminal equipped with the optional sound program transmitting unit.
- A maximum of nine simultaneous telephone channels can be received, one from each small terminal.

Site requirements and installation. - The experimenter assumes responsibility for selection and preparation of the sites for the 3-meter (10-ft) terminals. Figure VII-4 illustrates the terminal orientation on a typical site. The terminal will be installed on the site by a CRC operating crew. Setting up and dismantling the terminal in the trailer configuration can be accomplished by a two-man crew. Dismantling in preparation for transport requires an estimated 4 hours. Setup requires an estimated 8 hours for antenna assembly, locating and acquiring the satellite, and system checkout.

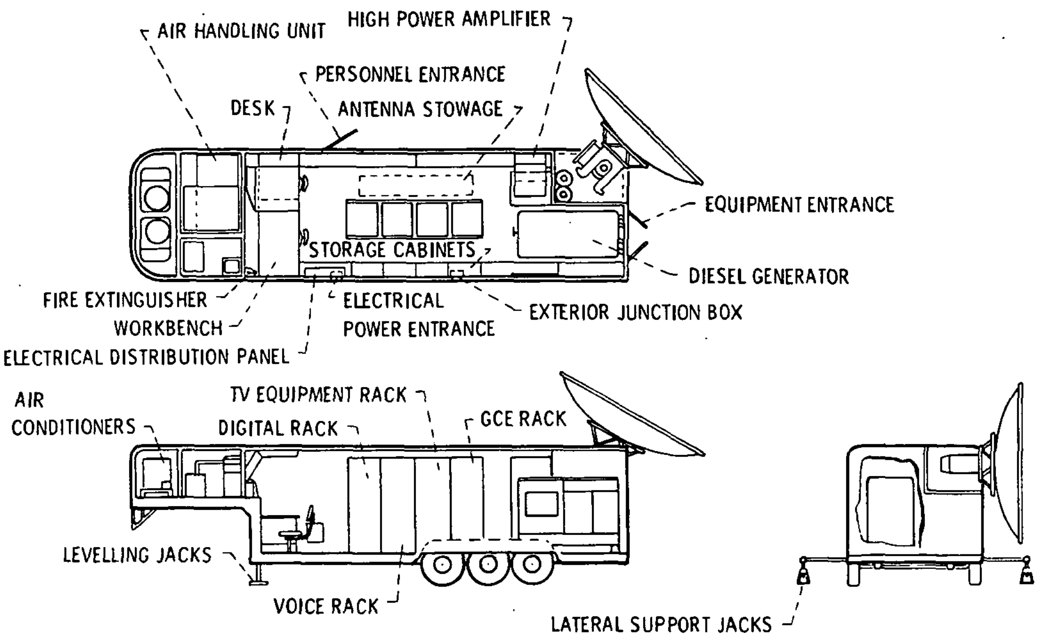


Figure VII-3. - 3-Meter ground terminal.

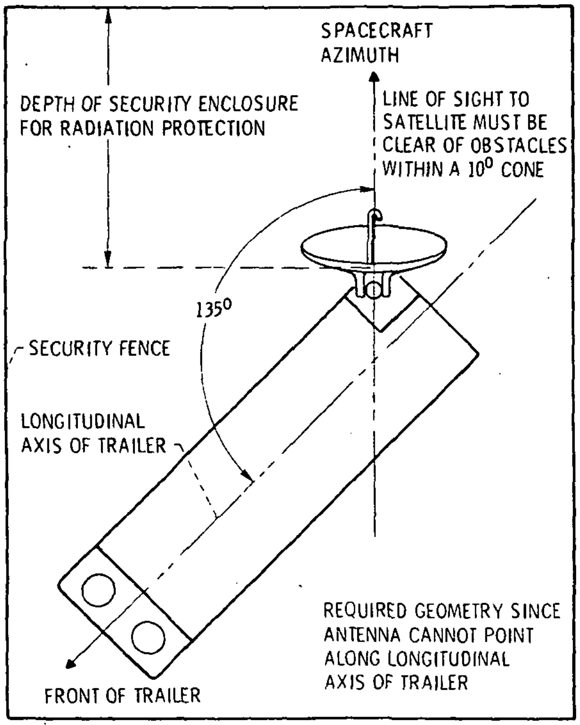


Figure VII-4. - 3-Meter-terminal orientation on site.

In the small-aircraft configuration, facilities must be provided by the experimenter. For proper operation, the terminal equipment facility temperature must be maintained between 18.3° C (65° F) and 26.7° C (80° F). The power levels capable of being transmitted by the 3-meter (10-ft) terminal are sufficiently high to present an RF radiation hazard to personnel in the beam of the antenna. Consequently, site protection is required. The terminal is equipped to operate from a 240/120-volt, single-phase, 60-Hz, three-wire service at a maximum rate of 32 kV-A and 0.8 pF. The experimenter must arrange to provide service of this capacity on site.

Terminal characteristics. - The major terminal characteristics are defined by the parameters

- System G/T, 22 dB
- EIRP, 79.5 dBW

These are minimum specified values. Figure VII-5 is a block diagram of the terminal's communication equipment. An experimenter can interface with the terminal at four locations, depending upon the specific requirements. These include the exterior junction box mounted on the outside of the trailer, an internal baseband interface patch panel for television video and audio, a baseband patch panel for the telephone and sound program systems, and an IF interface connection panel.

Two-Meter Ground Terminal

The 2-meter (7-ft) ground terminal is intended to provide television reception plus a single telephone channel. In addition, by connection of the optional sound program transmitting unit and/or the sound program receiving unit, the terminal can transmit or receive the sound program channel. Eight of these terminals are being purchased. The terminals are configured as shown in figure VII-6. They are designed to be used near the building provided by the experimenter. The antenna and antenna mount can be installed either on the roof of the building or on the ground near the building. On the antenna are two weatherproof electronics equipment packages containing the high-frequency components of the terminal. These packages are interconnected by means of cables to an indoor unit which is composed of packages containing some terminal electronics, the interface panels, and operating controls.

Signal types and interfaces. - Patch panels and connection panels are provided on the electronic packages located in the experimenter's building to allow the experimenter's equipment to interface with the terminal. Available signal types are as follows:

- One telephone channel which is capable of transmitting and receiving voice: Two interfaces are available; a standard telephone set and direct input into the telephone channel unit are provided on the baseband interface patch panel.

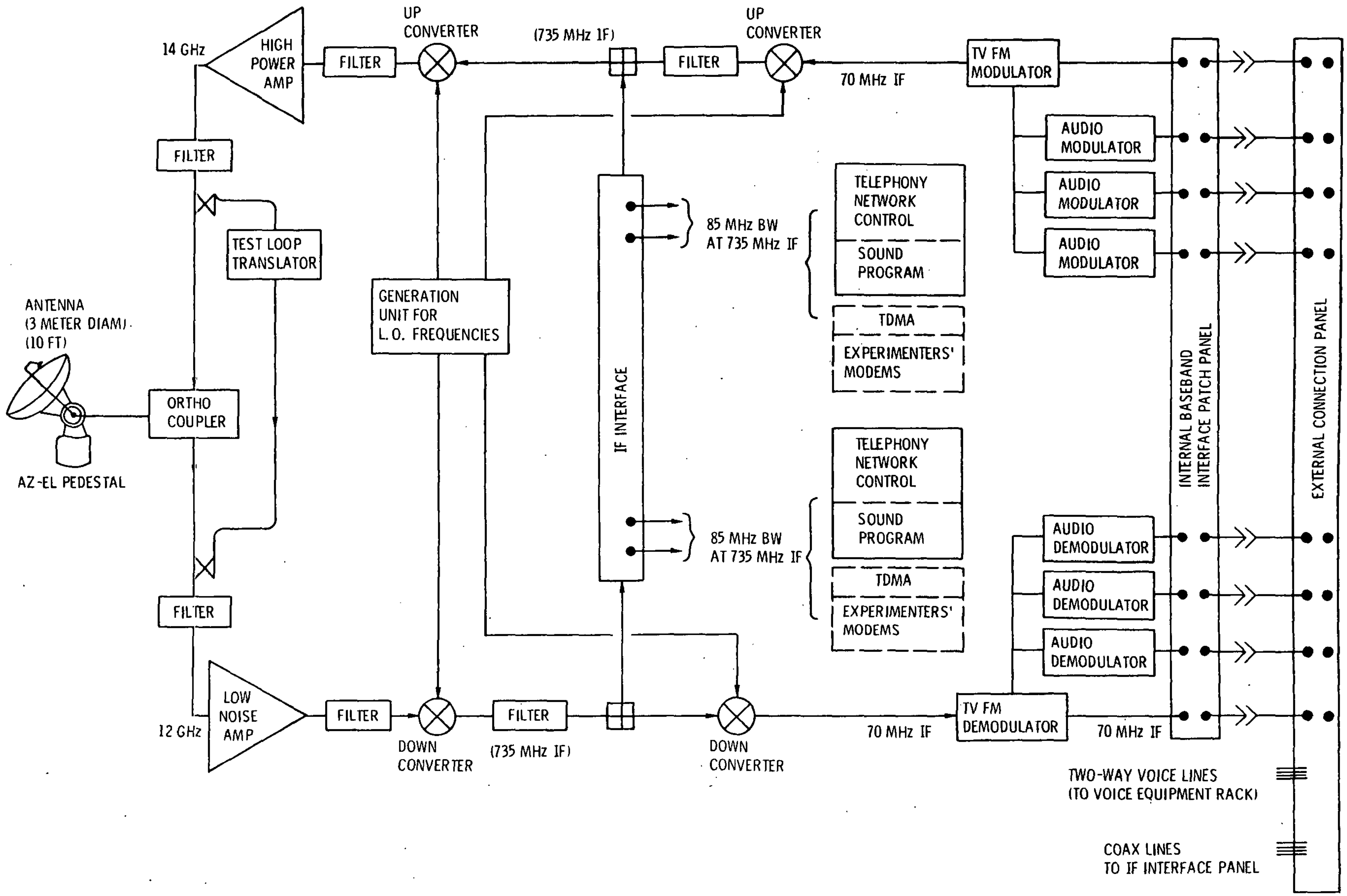


Figure VII-5. - 3-Meter-terminal block diagram.

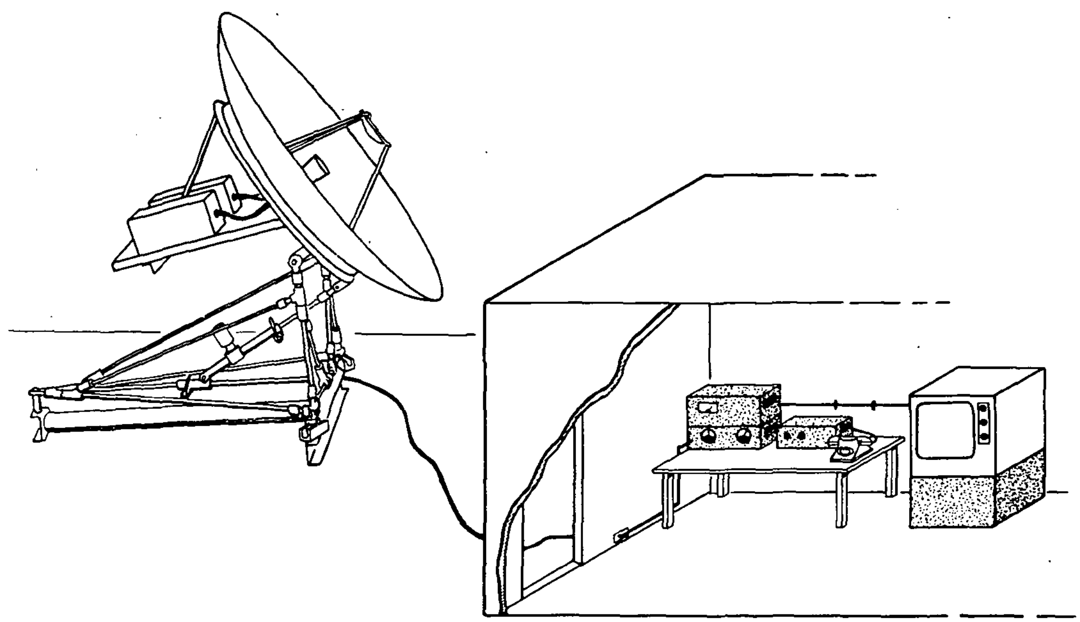


Figure VII-6. - 2-Meter ground terminal.

- One sound program signal which can be transmitted to a 3- or 9-meter (10- or 30-ft) terminal but requires the use of the optional sound program transmitting unit
- One television video channel with as many as three television audio channels which can be received from a 3- or 9-meter (10- or 30-ft) terminal
- One sound program signal which can be received from the 3- or 9-meter (10- or 30-ft) terminal when it is equipped with the optional sound program receiving unit: Two interfaces are available which can be used simultaneously, a standard output interface and a speaker interface.

Site requirements and installation. - Sites for the antenna can be selected either on the roof of a building or on the ground. Equipment is estimated to weigh about 280 kg (617 lbm). The site must have a clear line of sight to the spacecraft and be free of obstacles within a 10° cone centered on the line of sight. Radiation densities in the beam on the antenna are not expected to exceed the safe levels for human exposure. The experimenter must provide power. The terminal requires an estimated 700 watts, 120 volts, 60 Hz, single-phase, three-wire service.

Terminal characteristics. - The major terminal characteristics are defined by the minimum values:

- System G/T, 16.6 dB
- EIRP, 60.5 dBW

Figure VII-7 is a block diagram illustrating the terminal's communication equipment. Experimenters can interface with the terminal at several points. These include the high-IF-interface patch panel, the television patch panel, the baseband interface patch panel for the telephone system, the baseband patch panel of the sound program transmitting unit (if available), the baseband patch panel of the sound program receiving unit (if available), and a low-IF-interface patch panel.

One-Meter Ground Terminal

This terminal is intended to provide a single telephone channel. Also, by connection of the optional sound program transmitting unit and/or sound program receiving unit, the terminal can transmit or receive the sound program channel. Eight of these terminals will be available for experimenter's use. The terminals are designed to be used near a building provided by the experimenter. The antenna and antenna mount can be installed either on the roof or on the ground near the building. A weatherproof electronics equipment package is interconnected with cables to an indoor unit containing the interface panels and operating controls. The interconnecting cables allow separations between the indoor and outdoor units of as much as 30 meters (100 ft). The terminal can be broken into several small pieces for transport and is easily handled by two men.

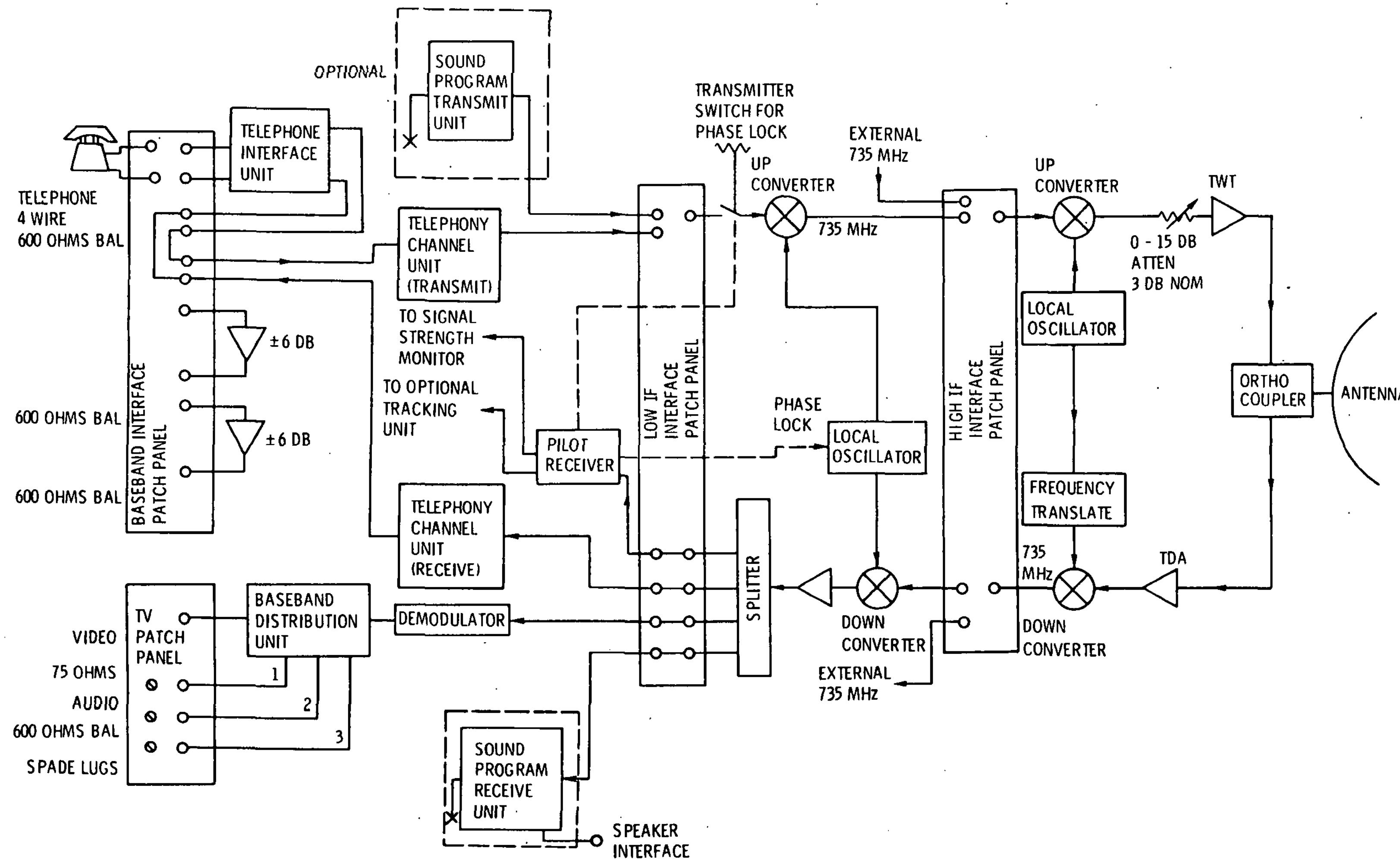


Figure VII-7. - 2-Meter-terminal block diagram.

Signal types and interfaces. - Patch panels and connection panels are provided on the electronic packages located in the experimenter's building to allow the experimenter's equipment to interface with the terminals. Available signal types and interfaces are as follows:

- One telephone channel capable of transmitting or receiving voice: Two interfaces are available - a standard telephone set and a direct input on the baseband interface patch panel.
- One sound program signal which can be transmitted to a 9-meter (30-ft) terminal with the use of the optional sound program transmitting unit
- An IF interface patch panel for patching the sound program transmitting unit in place of the telephone channel unit
- One sound program channel which is received from the 9- or 3-meter (30- or 10-ft) terminal with use of the optional sound program receiving unit: Two interfaces are available which can be used simultaneously - a standard output interface and a speaker interface.
- An IF interface patch into the receiving portion of the telephone channel unit, the pilot receiver unit, and the sound program receiving unit

Site requirements and installation. - Sites for the antenna can be selected either on the roof of a building or on the ground near the experimenter's building. The equipment is estimated to weigh about 95 kg (210 lbm), and the site must have a clear line of sight to the spacecraft. It must be clear of obstacles within a 10° cone centered on the line of sight. Power requirements are estimated at 460 watts, 120 volts, 60 Hz, single-phase, three-wire service.

Terminal characteristics. - The major terminal characteristics are defined by the following minimum specified values at the beam edge:

- System G/T, 5.2 dB
- EIRP, 48.0 dBW

Figure VII-8 is a block diagram of the terminal communications equipment. Experimenters can interface with the terminal at several points. These include the baseband patch panel for the telephone system, the baseband patch panels of the sound program transmitting and/or receiving units (if available), and a low-IF-interface patch panel. With the exception of the antenna, the equipment is essentially identical to that of the 2-meter (7-ft) terminal.

UNITED STATES GROUND TERMINALS

In the United States, each user is responsible for providing all the terminal equipment associated with his experiment. NASA is, however, planning a number of ground terminals for the purpose of demonstrating experimentally the effectiveness of various

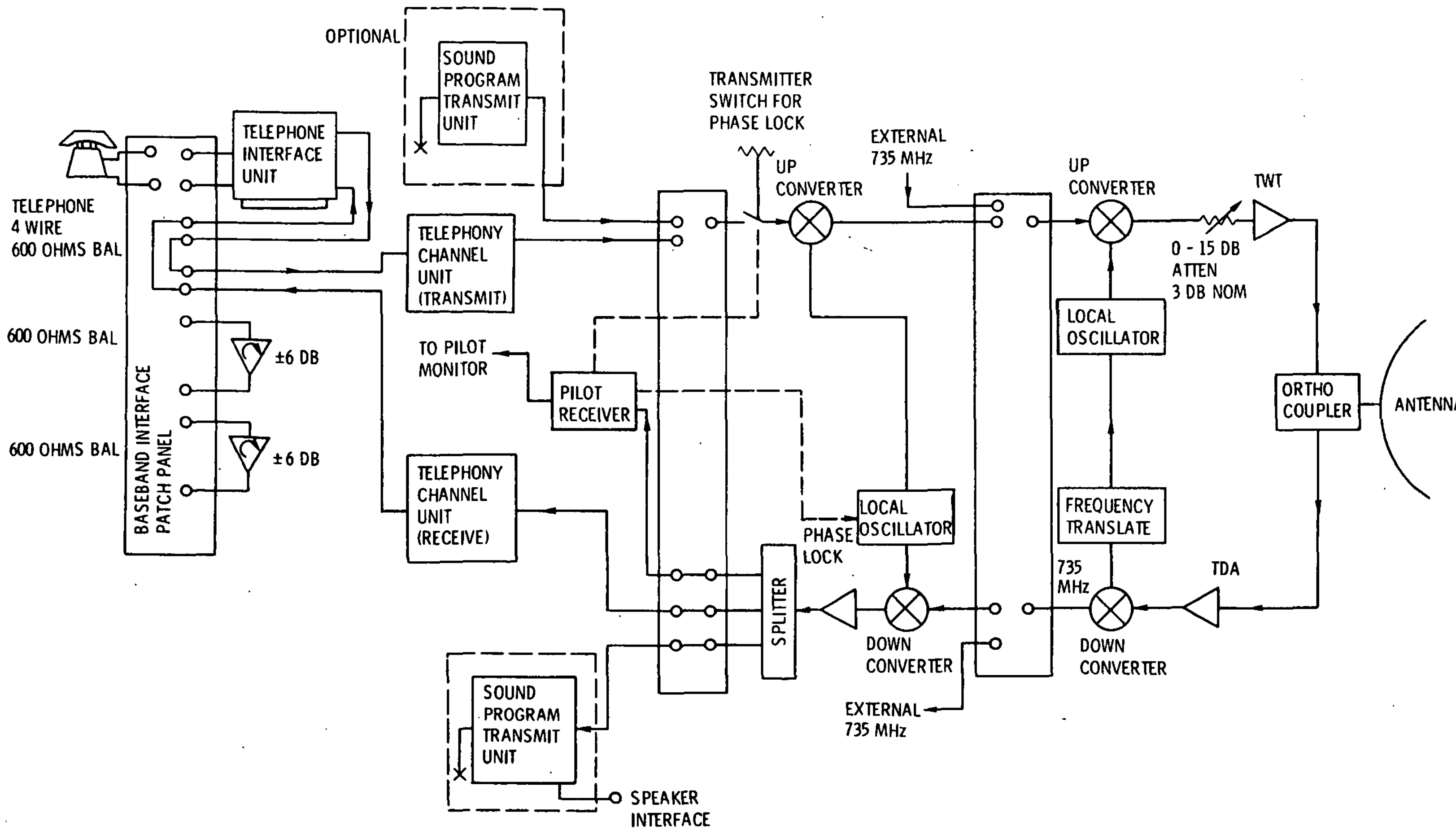


Figure VII-8. - 1-Meter-terminal block diagram.

low-cost ground station concepts and to perform experiments which require a transportable terminal at a remote location for short periods. A ground station facility is also available at Lewis for use in conjunction with appropriate experiments on a noninterference basis with its primary objective, which is to support the TEP SHF technology experiment. Several existing and planned U.S. SHF terminal characteristics are summarized in table VII-1.

TABLE VII-1. - CHARACTERISTICS OF SEVERAL EXISTING AND PLANNED U.S. SHF TERMINALS (12 TO 14 GHz)

Terminal	Antenna				Receiver		System gain/ temperature ratio, dB/K	Transmitter power, W	Antenna control
	Diameter		Peak gain (12 GHz), dB	Three-decibel beam width, deg	Preamplifier	Total system noise temperature, K			
	m	ft							
Cleveland (NASA)	5	16	52	0.4	TDA[a]	800	24	1250	Step track
Rosman (NASA)	5	16	53	.4	(b)	450	26	1250	Autotrack
TV receive only; two-way voice	3	10	48	.6	TDA[a]	900	18	500	Remote motor drive
Two-way voice	1.2	4	40	1.5	TDA[a]	900	10	20	Manual adjustment
Two-way voice	.6	2	34	3.0	TDA[a]	900	4	20	Fixed

[a]Tunnel diode amplifier.
[b]Uncooled paramplifier.

Lewis Super-High-Frequency Ground Station

The Lewis ground station facility provides the main support to the TEP SHF technology experiment and the 200-watt TWT. The facility is also available for use in conjunction with appropriate experiments upon approval. The support for the TEP includes primary activities during the following launch phases:

Prelaunch through attitude acquisition. - The primary activity during this phase to perform an assessment within the limitation of existing constraints of TEP health prior to and after each major event, where deviation from expected performance could adversely affect TEP operational capability.

In-orbit activities. - The activities to be performed in orbit include

- To perform checkout, initial startup, and initial evaluations of TEP performance as part of the CTS spacecraft communications system
- To evaluate in space the performance of the SHF high-efficiency, 200-watt Transmitter Experiment Package
- To demonstrate and evaluate the performance of a satellite communications system with low-cost ground terminals

- To obtain fundamental data to guide advancement in the state of the art of high-power microwave amplifier operation in space

The system is shown schematically in figures VII-9 and VII-10. The main operating requirements of the facility are to provide

- A high-power (85-dBW EIRP), wideband, frequency-modulated, uplink signal to the CTS at 14 GHz (video and audio)
- A low-noise, downlink receiving system at 12 GHz (video and audio)
- Capability of displaying video, audio, and telemetry data for real-time evaluation

System description. - The Lewis ground station facility consists of three major systems (table VII-2).

High-power, uplink transmitting and receiving system: The high-power, uplink transmitter consists of two identical low-power modulators, a two-channel combiner, a low-power driver, and the high-power transmitter. Baseband video and audio signals applied to low-power modulators produce a 14-GHz frequency-modulated (FM) signal at their output. Either one or two FM channels may be processed through the channel combiner. The low-power, FM, video-audio channel is amplified in the low-power driver, the output of which is used to excite the high-power klystron. The high-power transmitter system employs two high-power klystrons (type VKU 7791) with a common power supply and common input-output circuitry. Only one klystron may be operated at a time. The characteristics of the klystrons are identical except for frequency. One is fixed tuned for uplink band 1, while the other is tuned to uplink band 2. Both klystrons are air cooled, and the entire transmitter system can be operated remotely.

The receiving system consists of a 5-meter (16-ft) diameter Cassegrain parabolic reflector used to transmit signals to and receive signals from the spacecraft. A tunnel diode amplifier (TDA) is employed as a low-noise preamplifier for the two identical demodulators operating at the downlink frequency of 12 GHz. The uplink-downlink system has the capability to generate and demodulate video carriers with frequency deviation of 15 to 30 MHz peak to peak with or without pre- or deemphasis. Four audio subcarriers can be generated and demodulated, any three of which can be operated simultaneously.

Five-meter (16-ft) SHF Cassegrain antenna: The ground terminal antenna facility used for communication with the CTS spacecraft consists of a 5-meter (16-ft) diameter parabolic reflector with a Cassegrain feed (fig. VII-11). The reflector, feed assembly, and pedestal (elevation over azimuth) mounting structure are mounted on the roof of building 55 of Lewis. The associated tracking, control, and monitoring equipment is located in the Experiment Evaluation Center (EEC), which is located in building 54 at Lewis.

The antenna, with the rest of the ground station facility, is used to communicate with the CTS in the SHF transmitting frequency band of 14 GHz (gain, 54.0 dB) and in the receiving frequency band of 12 GHz (gain, 52.5 dB). A step-track system, locked to a 11.7-GHz spacecraft beacon, can continously update the antenna pointing toward the

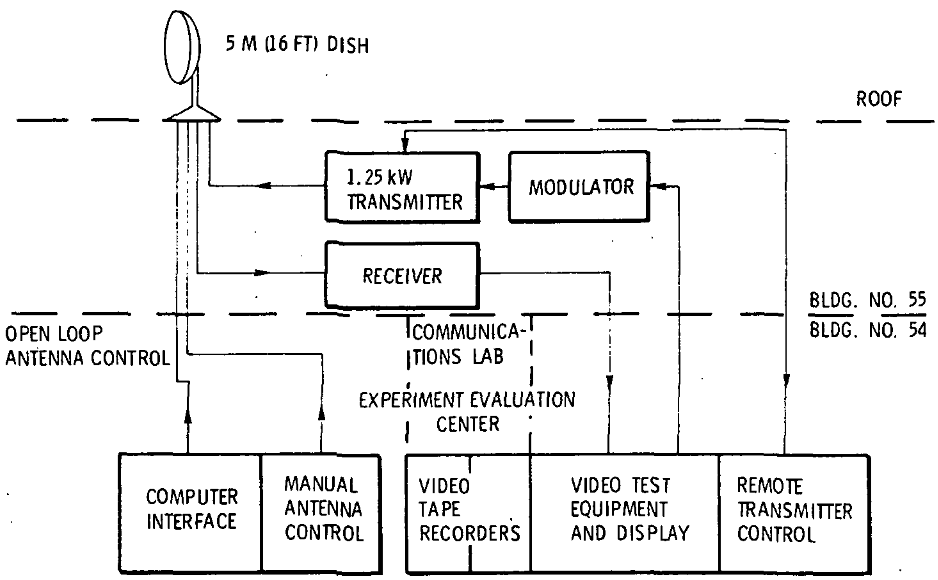


Figure VII-9. - Lewis ground station arrangement.

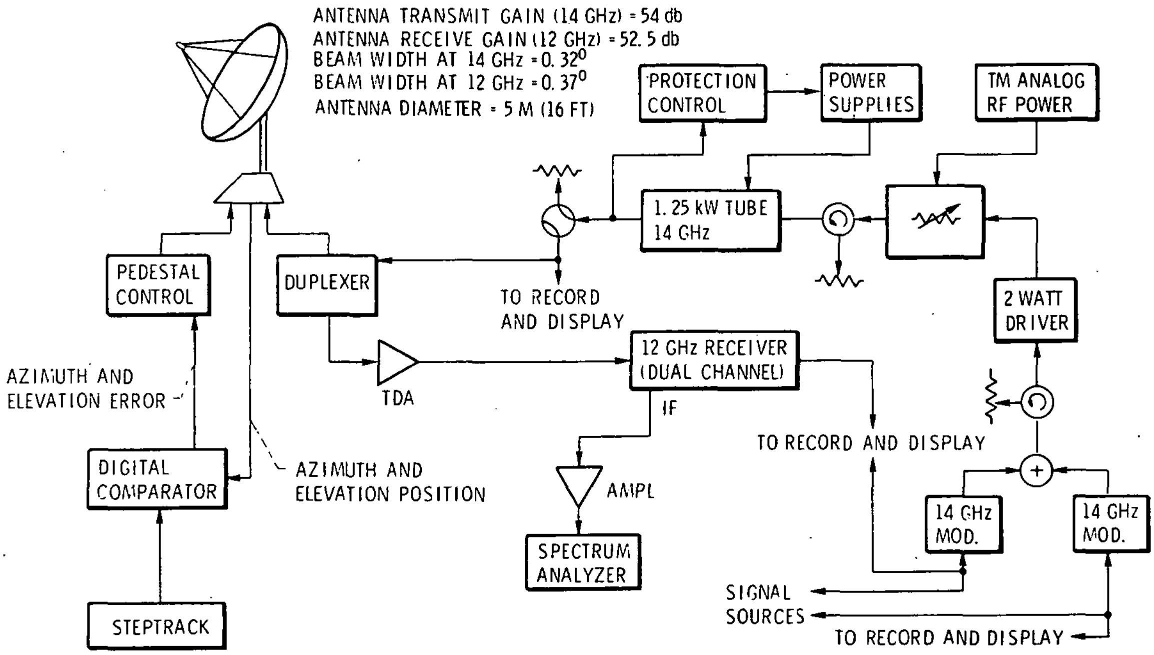


Figure VII-10. - Lewis ground station block diagram.

TABLE VII-2. - CHARACTERISTICS OF LEWIS GROUND STATION

Location:	
Area	Lewis Research Center, Cleveland, Ohio
Latitude	41° 24'45'' N
Longitude	81° 51'55'' W
Elevation, m (ft)	231 (758)
Antenna:	
Type	Parabolic reflector with Cassegrain feed
Diameter, m (ft)	5 (16)
Polarization	Linear orthogonal (transmit/receive)
Mount	Elevation over azimuth
Receiving frequency	11.983 GHz±150 MHz (11.7-GHz beacon)
Transmitting frequency	14.150 GHz±150 MHz
Gain/temperature ratio, dB	23.5
Gain-receive, dB	52.5
At 54 percent	53.06
At 50 percent	52.73
Gain-transmit, dB	54.0
At 54 percent	54.51
At 50 percent	54.17
Receiving system noise temperature, K	800
Effective Isotropic Radiated Power (EIRP), dBW	85
Beam width (3 dB), deg	
Receiving	0.37
Transmitting	0.309
Tracking accuracy (step track), deg	±0.05
Transmitter:	
Frequency, GHz	14
Modulation	FM
Maximum power output, kW	1.25
Radiofrequency bandwidth, MHz	85
Modulators (2 each):	
Frequency, GHz	14
Modulation	FM
Power output, dBm	22.6
Subcarriers (audio), MHz	7.5, 5.14, 5.41, 5.79
Preemphasis	
Video	CCIR
Audio, μsec	75
Deviations (peak to peak):	
Video, MHz	15-30
Carrier by subcarrier, MHz	1.98
Subcarrier by audio, kHz	120-200
Demodulators (2 each)[a]:	
Frequency, GHz	12
Modulation	FM

[a]Compatible with modulators, except for frequency.

Figure VII-11. - 5-Meter SHF Cassegrain Antenna.

spacecraft with an accuracy of ±0.05°. The antenna half-power beam widths are 0.30° and 0.37° on transmitting and receiving, respectively.

TEP Experiment Evaluation Center: The SHF experiment will be conducted and evaluated from the Experiment Evaluation Center (EEC). The EEC will serve as the main facility for display, switching, control, and evaluation of video, audio, and telemetry data. Both the uplink transmitter and the 5-meter (16-ft) dish antenna can be controlled from this location. All audio and video signals (generated in the communications laboratory) are routed through EEC to modulators. Likewise, all signals received from the spacecraft are similarly routed through EEC to various display and recording equipment. Some video tape recorders (VTR's) are located in the communications laboratory.

Sample link calculations. - Tables VII-3 and VII-4 show typical uplink and downlink

TABLE VII-3. - SAMPLE UPLINK CALCULATION FOR

LEWIS GROUND STATION

[Uplink frequency, 14.2 GHz.]

Characteristic	Spacecraft receiver noise temperature, K	
	1315	2315
Terminal:		
Transmitter power (1250.0 W), dBW	30.97	30.97
Feed loss, dB	-2.00	-2.00
Antenna gain (4.88 m (16.0 ft)), 0.31° half-power beam width (HPBW)	54.53	54.53
Effective Isotropic Radiated Power (EIRP), dBW	83.50	83.50
Antenna pointing error (0.05°), dB	-0.26	-0.26
Margin, dB	-3.00	-3.00
Propagation loss (23 074 statute miles; latitude, 41.4°; relative longitude, 35.1°), dB	-207.22	-207.22
Atmospheric loss (0.100% outage; CCIR Rainfall Region 2), dB	-2.23	-2.23
Polarization loss, dB	-0.25	-0.25
Spacecraft:		
Feed loss, dB	-0.00	-0.00
Antenna gain (0.70 m by 0.70 m (2.3 ft by 2.3 ft); 2.15 by 2.15 HPBW)	37.68	37.68
Antenna pointing error (0.38°), dB	-0.31	-0.31
Received carrier power, dBW	-92.03	-92.03
Noise power density, dBW/Hz	-197.41	-194.96
Bandwidth, dB (Hz) (27.0 MHz)	-123.10	-120.04
Carrier-power/receiver-noise ratio, dB	31.02	28.56

TABLE VII-4. - SAMPLE DOWNLINK CALCULATION FOR

LEWIS GROUND STATION

[Downlink frequency, 12.1 GHz.]

Characteristic	Spacecraft receiver noise temperature, K	
	1315	2315
Spacecraft:		
Output tube power (200 W), dBW	23.01	23.01
Feed loss, dB	-0.00	-0.00
Antenna gain (0.70 m by 0.70 m (2.3 ft by 2.3 ft); 2.52 by 2.521 + PBW), dB	36.28	36.28
Effective Isotropic Radiated Power (EIRP), dBW	59.29	59.29
Antenna pointing error (0.38°), dB	-0.22	-0.22
Margin, dB	-3.00	-3.00
Propagation loss (23 074 statute miles; latitude, 41.4°; relative longitude, 35.1°), dB	-205.81	-205.81
Atmospheric loss (0.100% outage; CCIR Rainfall Region 2), dB	-1.52	-1.52
Polarization loss, dB	-0.25	-0.25
Terminal:		
Feed loss, dB	-1.00	-1.00
Antenna gain (4.88 m (16.0 ft); 0.30° HPBW)	53.12	53.12
Antenna pointing error (0.05°), dB	-0.18	-0.18
Received carrier power, dBW	-99.58	-99.58
Noise power density (T=800 K), dBW/Hz	-199.57	-199.57
Bandwidth, dB (Hz) (27.0 MHz)	74.31	74.31
Terminal receiver noise power, dBW	-125.26	-125.26
Uplink noise contribution (C/N, 31.02; 28.6 dB), dB	0.95	1.80
Terminal net noise power, dBW	124.14	123.45
Terminal carrier-power/receiver-noise ratio, dB	24.56	23.87
FM improvement (M=2.00), dB	21.58	21.58
Noise weighting factor (CCIR), dB	10.20	10.20
Preemphasis improvement, dB	2.40	2.40
Signal/noise ratio, dB	58.91	58.05

power budgets for the wideband video signal, which will be processed through the CTS spacecraft by the Lewis ground station. As an example, table VII-3 shows an uplink power budget for transmission of a 1.25-kW video signal from a ground station to the spacecraft. Combining all the losses and gains, the spacecraft received carrier power is calculated to be -92.03 dBW. Likewise, the total noise contribution at the spacecraft is -123.10 dBW, which yields the difference of 31.02 dB as the carrier-power/receiver-noise ratio. More simply stated, the picture received by the spacecraft receiver is of an excellent quality.

Table VII-4 shows the downlink power budget for retransmission by the spacecraft of the video signal, using the 200-watt, spacecraft transponder. Combining losses and gains, the ground-terminal carrier-power/receiver-noise ratio is calculated as 24.56 dB. This quantity can be converted into a signal/noise ratio of 58.91 dB at the television monitor, which again is a television picture of excellent quality.

The spacecraft has the capability of utilizing either of the two onboard receivers. Tables VII-3 and VII-4 show video signal data when a low-noise ($T = 1315^{\circ}$ K) receiver and a higher noise ($T = 2315^{\circ}$ K) receiver are used.

U.S. USERS GROUND TERMINALS

Typical U.S. users ground terminals are summarized in table VII-1 and include a variety of terminal types reflecting a range of experimental uses. One important kind of user experiment involves the reception of single-channel video by several ground stations and may include the transmission of return audio. A typical FM video receiver which would be used in such an experiment consists of an outdoor mixer-down converter and an indoor intermediate-frequency receiver. This receiver would provide either baseband or VHF-AM output, or both, and could accommodate one or more audio subchannels. A tunnel diode amplifier would be included with the outdoor unit to improve receiver sensitivity. An image-enhanced mixer is also available and would not require a tunnel diode amplifier.

A 3-meter (10-ft) parabolic antenna would be used with the video experiment. The antenna system would have two-axis-limited motion pointing capability and could accommodate the anticipated north-south and east-west movement of the spacecraft. The antenna pointing system would be motor driven and remotely controlled from indoors. The antenna and outdoor receiver unit could be located as much as 30 meters (100 ft) from the indoor equipment. A block diagram of a typical FM video receiver ground terminal is shown in figure VII-12.

A video uplink or transmitting station is much more expensive than a receive-only station because of the dominating cost of the high-power amplifier. A block diagram of a typical video transmitting station is shown in figure VII-13. The video baseband signal is modulated, upconverted, amplified to a high power level, and radiated from the antenna. If the uplink station is located in the downlink coverage region, the uplink station can receive its own signal from the spacecraft with the transmitting antenna, orthomode coupler, TDA, and receiver (as shown). For some experiments the uplink is not located in or near the downlink coverage area, and thus the uplink station cannot monitor its own signals as transmitted by the spacecraft. For example, this would occur with an uplink station in the eastern U.S. transmitting to a downlink coverage area in the western U.S.

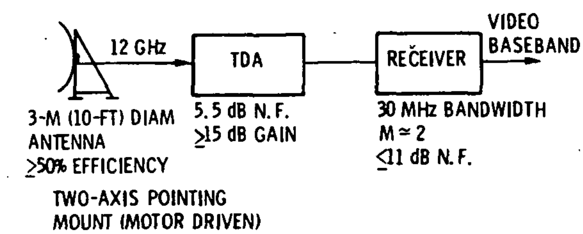


Figure VII-12. - Typical U.S. user video receive ground terminal.

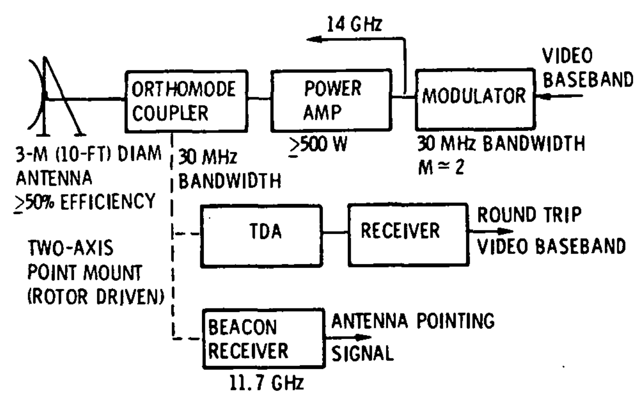


Figure VII-13. - Typical U.S. user video transmit ground terminal.

VIII. TECHNOLOGY EXPERIMENTS

UNITED STATES AND CANADIAN RESPONSIBILITIES

The authority for NASA's international activities resides in Public Law 85-568, which provides the basis for cooperation by the United States with other nations in demonstrating the peaceful purposes of space research and exploration. The CTS is such a cooperative effort and the related national responsibilities are defined in a "Memorandum of Understanding Between the United States National Aeronautics and Space Administration and the Canadian Department of Communications" signed April 20, 1971. The principal technological objectives are defined in the Memorandum of Understanding and include the requirement to conduct satellite communication systems experiments with 12- and 14-GHz terminals. Also included are requirements to develop and flight test a superefficient traveling wave tube, unfurlable solar arrays, and an accurate stabilization system for a spacecraft with large, flexible appendages.

NASA and DOC have agreed to freely share and exchange all scientific and technical results of these experiments and to make these results available through publication in appropriate journals or through established channels. Specific DOC responsibilities in technology experiments include the provision for tracking, data acquisition, command and control after the spacecraft is placed into geostationary orbit, and the provision of ground facilities in Canada for experimental programs. Similarly, specific NASA responsibilities, in the area of technology experiments, include the provision of ground facilities in the United States for the experimental program and a superefficient power tube to be used on the spacecraft along with the associated power conditioning and thermal interface equipment. Both DOC and NASA are charged with the responsibility to act as coinvestigators in meeting the principal technological objective of conducting satellite experiments with the unfurlable solar array.

TRANSMITTER EXPERIMENT PACKAGE/SUPER-HIGH-FREQUENCY EXPERIMENT

A major thrust of the overall CTS program concept is the joint objective of Canada and the United States to carry out an experimental program in communications. The CTS is expected to produce advances in the field of spacecraft communications systems by operating at the newly allocated frequencies of 12 and 14 GHz and by significantly increasing radiated power levels over those provided by existing spacecraft. In support of this objective, a number of advanced technology developments will be demonstrated. A primary component of the overall program is a technology experiment with the 200-watt, high-efficiency TWT. In combination with a transponder package and antenna system, the TWT will be capable of relaying communications signals between suitably located ground terminals. This experimental device is designated as the Transmitter Experiment Package (TEP). The more significant TEP characteristics to be evaluated or demonstrated include

- Efficiencies of the collector, overall TWT, and power processor at different RF power levels
- TWT gain, bandwidth, and frequency stability under space environment conditions
- System noise loading
- Effects of distortion, noise, and intermodulation on television picture quality
- Fundamental data on the operation of high-power, high-voltage systems

TEP Experiment Approach

The in-orbit phase of the TEP technology experiment consists of a series of checks and tests that are necessary for the initial evaluation of the TEP performance as part of the CTS communications system. The tests are designed to assess TEP status from launch through the in-orbit phase of the spacecraft operation and are as follows:

- Prelaunch tests commence about 30 days prior to launch and after the spacecraft arrives at ETR. These tests include visual inspections and functional tests.
- Drift-orbit tests consist of periodic temperature checks throughout the drift period.
- After a minimum 3-day system checkout and outgassing period, initial in-orbit TEP tests begin. The initial tests last about 6 days.
- The next phase of in-orbit TEP testing consumes about 2 days per week for the next 3 months, including tests before and after an eclipse.
- The last phase of in-orbit TEP testing is scheduled for approximately 1 day per month for the remainder of the 2-year mission.

TEP Experiment Descriptions

In-orbit tests begin after the checkout and outgassing period and are structured around five groups of tests which are performed periodically during the mission. These tests are

- Tests at maximum gain over a range of frequencies to obtain thermal and power processor readings and data related to thermal stabilization

- TEP/transponder communications tests with one and two carriers on band TB1 and observations on band TB2; video tests with temperatures stabilized and during transient temperatures
- Power processing system noise tests at different temperature levels and during thermal stabilization
- Combined video and audio tests
- Cathode emission tests

These tests are performed at various times during the mission, including abbreviated tests during eclipse periods. In addition, the transponder may be exercised through the low-gain modes.

SOLAR ARRAY TECHNOLOGY EXPERIMENT

As a result of the unique design of the Deployable Solar Array, a series of experiments will be performed on the array to support the development of analytical models describing the various solar array configurations. The experimental objectives include

- Demonstration that measurement and theory are in accord for the flexible solar array mechanics with particular attention to
  - Solar array deployment dynamics
  - Deployed static shape in sunlight and eclipse
  - Deformation history while traversing the earth's terminator
  - Natural vibrational frequencies, modes, and nodal damping
  - Magnitude of dynamic deformation in response to a known perturbation
- Establishment by flight measurement and correlation with theory of the solar and magnetic torque environment for the CTS configuration

Typical array dynamics to be studied are shown graphically in figure VIII-1. A functional block diagram to implement the experiment is provided in figure VIII-2.

Array instrumentation consists of a deflection-measuring system, which measures deflections of the tip of the array in two directions by means of strain gages attached to the boom. With the aid of ground computations, motions of the tip of the array both in plane and out of plane are obtained. Data on the rotation of the tip relative to the base are also available. Accelerometers are provided to measure the dynamic motions.

ATTITUDE CONTROL SYSTEM EXPERIMENT

The Attitude Control System (ACS) experiment will provide in-flight information with regard to spacecraft stabilization and control during the attitude-acquisition and on-station phases of the mission. In addition, the experiment will provide measure-

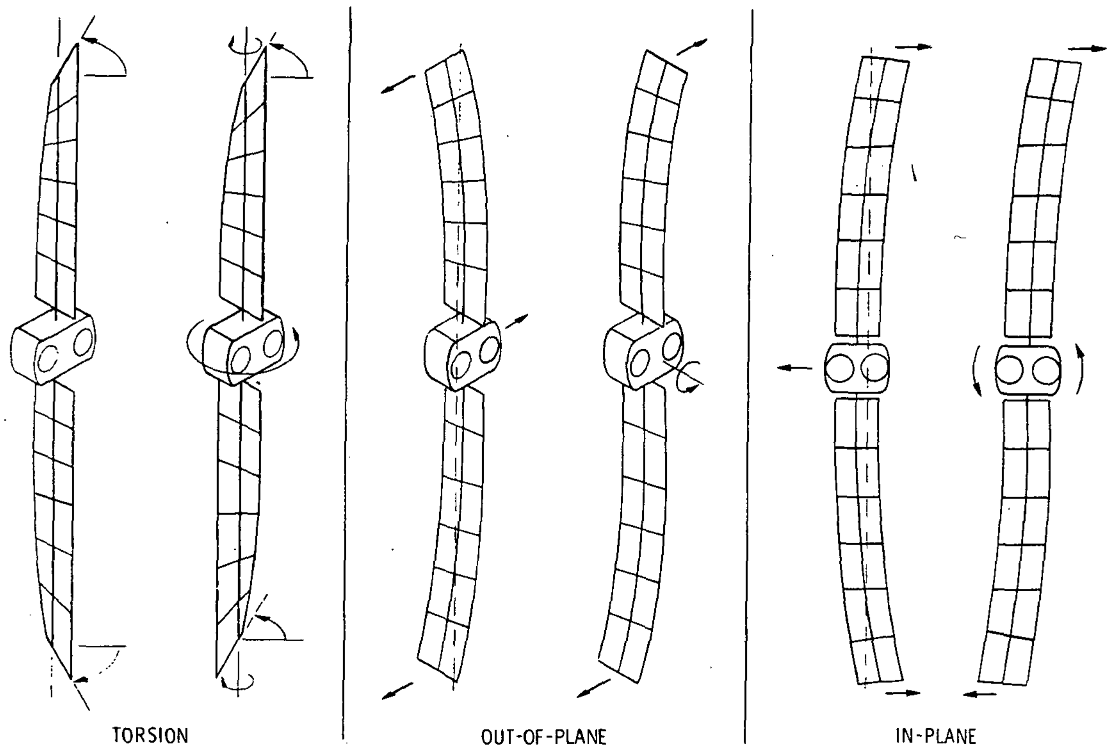


Figure VIII-1. - Solar Array Technology Experiment array dynamics.

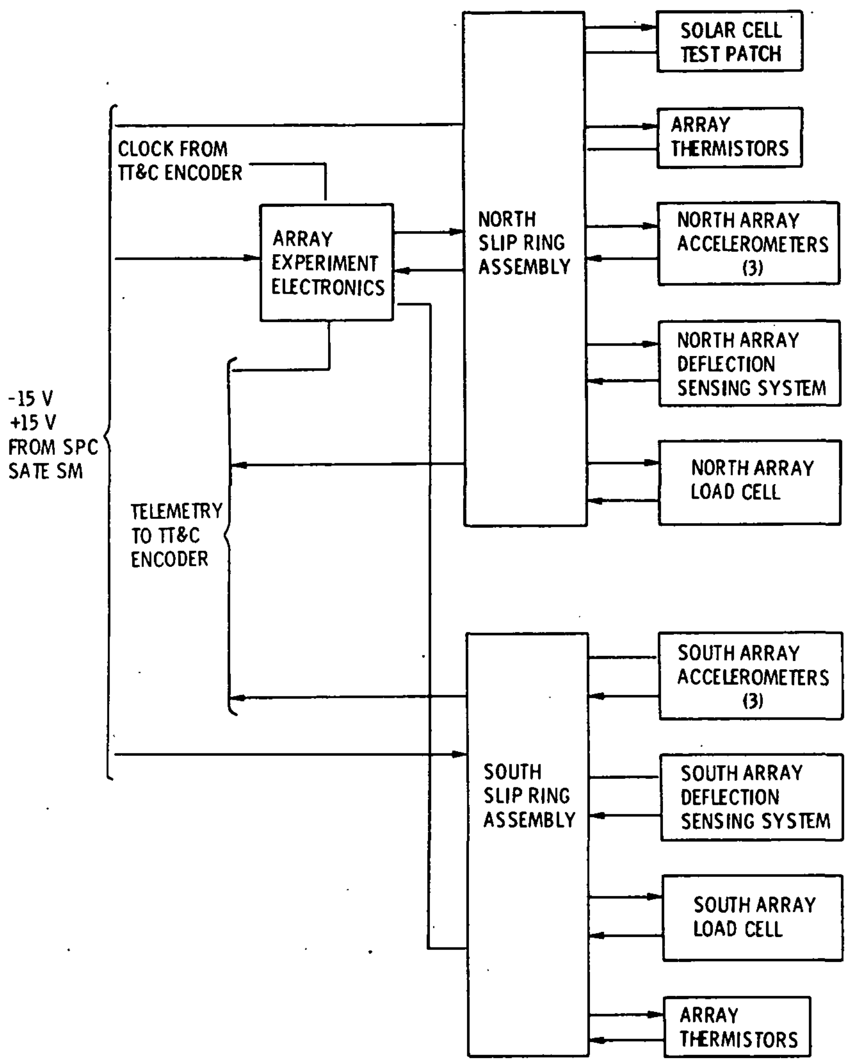


Figure VIII-2. - Solar Array Technology Experiment functional block diagram.

ments of the solar torque magnitude and the effects of spacecraft flexibility on the ACS operation. In general, the experiment will attempt to provide answers to the following questions:

- What is the appropriate solar and magnetic torque environment for the CTS class of spacecraft?
- Which method of three-axis stabilization is optimal for the CTS class of spacecraft?
- What boresight stabilization (roll, pitch, or yaw) can be achieved with the offset-thruster principle of operation?
- Is existing theory regarding flexibility effects and stabilization valid?
- Which method of array deployment is best suited for the CTS class of spacecraft (spinning or nonspinning)?
- To what extent are "ground-based control" and "man in the loop" modes of acquisition operations feasible and/or desirable?

Specific experiment objectives are cited below and require no additional instrumentation beyond that currently installed (section III). It is planned to demonstrate that flight performance is in accord with theory for stabilization and control during attitude acquisition with specific emphasis on

- Active nutation damping
- Two-axis stabilization about the sun line
- Stabilization during solar array deployment
- Intentional precession maneuvers

Additional experimental activities include a demonstration that flight measurements and theory are in accord for on-station, three-axis stabilization as implemented by

- The "momentum wheel with offset thrusters" principle
- The "zero momentum" system consisting of earth sensors, sun sensors, and hydrazine thrusters
- Generic derivatives as are feasible within the scope of the hardware and ground-based control capability

The experiment will also establish by flight measurement and correlate with theory the solar and magnetic torque environment for the CTS configuration and the effect of spacecraft flexibility on stabilization system operation.

IX. COMMUNICATIONS EXPERIMENTS

UNITED STATES AND CANADIAN USER POLICY

The combined Canadian and United States technology and communications experiment activities are planned and coordinated through a Canadian Depart of Communications (DOC) and NASA Joint Working Group (JWG) organization. The working group is divided into communication and technology experiments subcommittees whose function is

- To define experiment objectives
- To evaluate user plans or proposals
- To determine spacecraft configurations for the conduct of experiments
- To examine and correlate experimental use of the spacecraft
- To determine instrumentation requirements for monitoring, retrieval, reduction, and evaluation of experimental data

In Canada, DOC will provide ground terminals to experimenters; these are described in section VII. In the U.S., each experimenter is responsible for providing the necessary ground terminals. Typical U.S. user ground terminals are also described in section VII. An overall communication experiments schedule reflecting a 50-50 time sharing between NASA and DOC on an alternate-day basis has been jointly developed by the two agencies based on the provisions of the Memorandum of Understanding. For the Canadian experiments, DOC will maintain, and in some cases operate, the remote terminals for the experimenters. In the U.S., except for preliminary checkout of the TEP, experimenters will operate and maintain their ground terminal equipment.

UNITED STATES USERS EXPERIMENTS

In order to explore the future uses of high-powered communications satellites, U.S. users are planning a variety of experiments. These include experiments in technology, health care, community services, education, and special services. The ground station locations range from the Atlantic to the Pacific seaboards. Government, education, and industry are all represented. The categories and typical experiments are described in the following sections. The present United States users are listed in table IX-1.

TABLE IX-1. - COMMUNICATIONS EXPERIMENTS - UNITED STATES

Experiment	Description	Organization
1	Communication Link Characterization	Goddard Space Flight Center
4	College Curriculum Sharing	Ames Research Center; Carleton University, Ottawa, Canada; Stanford University, Stanford, CA
6	Transportable Earth Terminal	Communications Satellite Corporation (COMSAT)
7	Biomedical Communications	Lister Hill National Center and Department of Health, Education, and Welfare
9	Satellite Library Information Network (SALINET)	SALINET
11	Health/Communications	Veterans Administration
13	Communication Support for Decentralized Education	The Washington, Alaska, Montana, Idaho Program (WAMI)
14	Armed Forces Radio and Television Wideband Direct-User Technology	Office of Information, U.S. Armed Forces
15	Communications in Lieu of Transportation	Westinghouse Electric Corporation
16	Project Interchange	Archdiocese of San Francisco
17	Health Education Television	Association of Western Hospitals

Education Experiments

Five experiments are currently planned to investigate and determine the feasibility of a range of educational requirements from counseling to curriculum sharing. They include

- College curriculum sharing among universities with a demonstration of digital video compression techniques for both bandwidth and power reduction
- Teacher upgrading by improving teaching skills and development of instructional units and making graduate education available to teachers
- Health education programs using live and videotaped techniques for use by hospitals and health-care facilities
- Exchange of materials and teaching techniques related to computer-aided instruction between diverse areas of the country
- Investigation of telecommunication systems requiring only limited human support and providing data on career development, employment, job preparation, and counseling

Health-Care Experiments

Three experiments are planned by state and federal agencies which involve Veterans' Administration Hospitals and various levels of health-care professionals. Typical experiments include

- Conducting biomedical clinical and continuing medical experiments among thirty VA hospitals
- Demonstration of the feasibility of information exchange between research institutions and the medical community; evaluation of the broadband teleconference as a means of continuing education among health-care professionals
- Investigation of techniques for improving administration and teaching procedures for decentralized medical education

Experiments in Community and Special Services

Industry, academia, and the federal government have each planned experiments which are as diverse as the organizations themselves and are structured to demonstrate or investigate the following:

- Feasibility of a satellite library information network to improve individual and organizational capabilities for assessing and disseminating information
- Development of techniques for transmission of special services programs worldwide; conversion of analog data to digital for wideband transmission of time-compressed audio at video format speeds
- Determination if a large and geographically dispersed industrial organization can substitute video and audio communication for travel

Technology Experiments

Two experiments are planned to investigate characterization of high-frequency communication links and transportable earth terminals as extensions of existing technology. Objectives of these experiments include

- Evaluation of attenuation and signal degradation due to absorption and scattering induced by atmospheric precipitation; measurement and characterization of earth-based, man-made signals which could interfere with the uplink frequency band
- Demonstration of the suitability of transportable earth terminals to relay communications to and from a disaster area

United States Experiments Description

Each of the U. S. experiments is described here; the experiment locations and antenna footprints are shown in figure IX-1.

EXPERIMENT 1: Communications Link Characterization
Goddard Space Flight Center

Objective
- Characterize RF links at 12 and 14 GHz
- Measure natural and man-made electromagnetic environment
- Determine attenuation and signal degradation induced by precipitation
- Evaluate and optimize communications systems operation

Technique
- Phase-modulated carrier; full bandwidth
- Digital modulated signal to 85 MHz
- One-way FM video

Ground System
- Rosman, NC (transmitter)
- Greenbelt, MD (receive only)
- Blacksburg, VA (receive only)
- Columbus, OH (receive only)
- Greenbelt to Rosman (land link)

EXPERIMENT 4: College Curriculum Sharing
Ames Research Center, Carleton University, Stanford University

Objectives
- Expand scope of instructions
- Develop student/teacher interaction for remote sharing
- Develop and demonstrate digital video impression system
- Evaluate cost effectiveness of system
- Demonstrate effectiveness in reducing satellite bandwidth and power requirements
- Interconnect university systems with community colleges

Technique
- Video, point-to-point, interactive

Ground System
- Ames Research Center, CA - 2.44-m (8-ft) terminal

TABLE IX-2. - COMMUNICATIONS EXPERIMENTS - CANADA

Experiment	Description	Organization
F-1-1	Propagation Measurement	Communications Research Centre
F-1-2	Time-Division Multiple-Access (TDMA) Synchronization	
F-1-3	Frequency-Division Multiple-Access (FDMA) Demand Assignment	
F-1-4	High-Data-Rate Transmission	
F-1-5	Small-Terminal Evaluation	
F-2-1	Broadcast Signal Reception in a Metropolitan Environment	Canadian Broadcasting Corporation
F-2-2	Applications of Radio Broadcasting	
F-2-3	Special Television Demonstration	
F-3	Continuing Education Learning System	Public Service Commission
P-1-1	Community Interaction	Newfoundland Communications Technology Committee
P-1-2	Telemedicine	
P-2-1	Community Communications in New Quebec	Ministry of Communications, Quebec
P-2-2	Telemedicine in New Quebec	Directorate of New Quebec
P-2-3	Omnibus Network	University of Quebec
P-2-4	Radiotelephony with Satellite Camps	Societe de Developpement de la Baie James/SOTEL
P-2-5	Telephony and Data Transmission Studies	Research Institute of Hydro-Quebec
P-3	Multiministry Administrative and Operational Experiment	Government of Ontario
P-5	Government Teleprocessing Network	Management Committee of Cabinet, Province of Manitoba
U-1	Digital Video University-Curriculum-Sharing Experiment	Ames Research Center; Carleton University, Ottawa, Canada; and Stanford University, CA
U-2	A Satellite Communications Link for Train Location	Queen's University
U-3	Satellite-Link Radio Interferometry	University of Toronto
U-4	Performance Evaluation of a Digital Modulation-Demodulation for High-Rate Data Transmission	McMaster University
U-5	Computer Network to Supply Information to Native People in Northern Canada	University of Western Ontario
U-6	Telemedicine Experiment	University of Western Ontario
U-7	Signal Processing Techniques for Data Communication	University of Waterloo
U-8	Upgrading Mathematical Competence of Elementary School Teachers	Lakehead University
U-9	Saskquebec Education - Culture	University of Regina; Ministry of Education, Province of Quebec
I-1	Transportable Telecommunications System	Bell Canada and Telesat
E-1	Health-Care Delivery to Remote Areas	Queen Charlotte Islands General Hospital
E-2	Project Ironstar	Alberta Native Communications Society

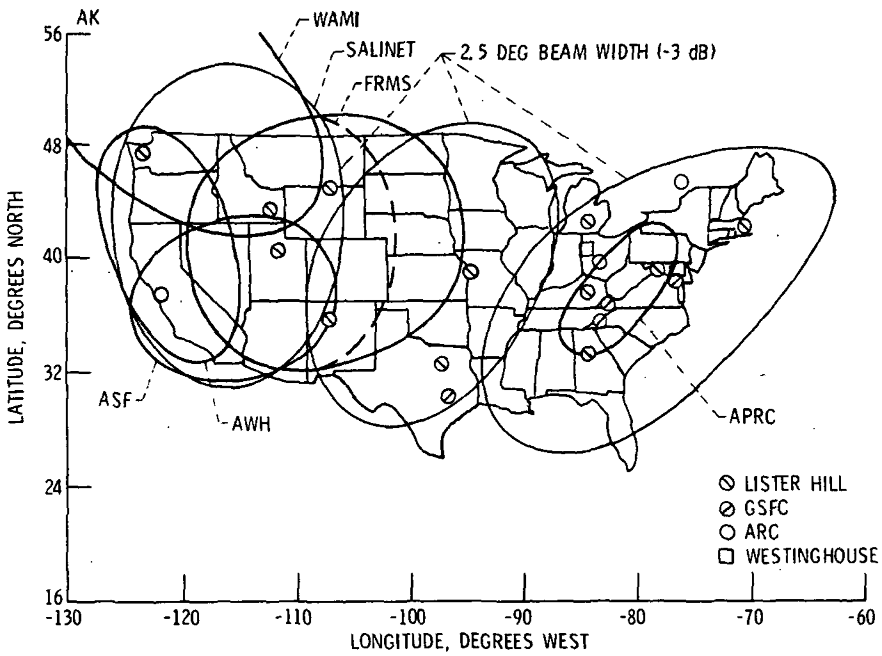


Figure IX-1. - U.S. experiment locations.

EXPERIMENT 6: Transportable Earth Terminal
COMSAT

Objectives
- Establish emergency communications with disaster areas
- Use small, highly mobile earth terminal with CTS
- Deploy system rapidly
- Establish operational satellite emergency communications system

Technique
- Duplex voice

Ground System
- COMSAT Laboratories, Clarksburg, MD
- Several remote areas in United States

EXPERIMENT 7: Biomedical Communications Experiment
Lister Hill National Center (HEW)

Objectives
- Disseminate medical information between the National Institute of Health (NIH) and the medical community
- Evaluate satellite communications for supporting continuing education among health professionals
- Develop techniques to support national biomedical network
- Evaluate distribution of audiovisual material
- Develop audiovisual distribution by electronic transmission
- Identify new techniques of audiovisual transmission

Technique
- Full duplex audio-video interactive
- One-way video with audio interactive

Ground System
- Resource centers at National Library of Medicine, MD; National Medical Center, GA; two West Coast medical libraries

EXPERIMENT 9: Satellite Library Information Network
SALINET

Objectives
- Provide information access by individuals and organizations
- Demonstrate cost effectiveness
- Develop user markets for satellite distribution

Technique
- One-way video broadcast
- Two-way audio (simplex, ATS-3)
- Two-way data (simplex, ATS-3)

Ground System - 56 sites in Rocky Mountain States
- 20 sites in North Dakota, South Dakota, Nebraska, and Kansas

Technique - Television broadcast
- Two-way voice (simplex) interactive (ATS-3)
- Two-way digital (simplex) interactive (ATS-3)

Ground System - 56 Sites in Rocky Mountain States
- 12 Public broadcasting stations
- Denver Network Control Center (DNCC) 11-m (36-ft) transmitter

EXPERIMENT 11: Health/Communications
Veterans Administration

Objectives - Establish communications for biomedical and clinical experiments
- Improve patient care
- Provide access to resources for remote hospitals
- Determine cost effectiveness of domestic satellite system

Technique - Television broadcast
- Television slow scan
- Two-way audio (simplex) interactive
- Two-way digital (simplex) interactive

Ground System - Thirty VA hospitals in west and midwest
- Resource center and headquarters at San Francisco, CA

EXPERIMENT 13: Communications Support for Decentralized Education
The Washington, Alaska, Montana, Idaho Program (WAMI)

Objectives - Increase medical school enrollment
- Provide meaningful education experience in nonmetropolitan communities
- Increase flow of knowledge between community practitioners and the university
- Broaden education opportunities in communities by providing studies under physician teachers
- Contain the cost of medical education

Technique - Full duplex audio-video interactive
- Video receive-only; audio duplex

Ground System - Four university and 14 community sites throughout the WAMI states

EXPERIMENT 14: Armed Forces Radio and Television Wideband Direct-User Technology

Office of Information, U.S. Armed Forces

Objectives
- Develop new or improved techniques for transmission of Armed Forces Network (AFN) audio and video program material worldwide
- Experiment in converting analog information to digital for wideband transmission of time-compressed audio at video format speeds
- Develop technology and application of speech plus multiplexing of program material for maximum utilization of satellite transmission
- Design suitable encoding and decoding devices for the dissemination of video transmission

Technique
- One-way video and audio
- Wideband digital

Ground System
- Washington, DC, broadcasting facility
- Los Angeles, CA, program studios
- Low-cost terminals on land and at sea

EXPERIMENT 15: Communications in Lieu of Transportation

Westinghouse Electric Corporation, Electronic Systems Support Division

Objectives
- To test the hypothesis that a large geographically dispersed industrial organization in the private sector can economically use, in a significant percentage of cases, a satellite to effectively exchange the information necessary to conduct business in terms of video, audio, and hard-copy media in lieu of transportation of the individuals desirous of the information exchange

Technique
- Full duplex video/audio/hard-copy interactive

Ground System
- Lima, OH - 4.6-m (15-ft) terminal
- Baltimore, MD - 3-m (10-ft) terminal

EXPERIMENT 16: Project Interchange

Archdiocese of San Francisco (ASF), Department of Education

Objectives

- Accelerate cost-effective application of available electronic technologies to existing systems of individualized instruction
- Utilize satellite interconnection for effective interaction of experts working toward the same system goal
- Develop favorable staff attitude and skills toward utilization of support technologies (e.g., instructional television and computers)
- Generate cost data for the introduction of advanced technological support systems in participating school systems

Technique

- One-way video (simplex) with as many as three audio channels
- Point-to-point communication via satellite with program distribution via Instructional Television Fixed System (ITFS), very-high-frequency public video and cable

Ground System

- Program originates at ASF television studios, Menlo Park, CA
- S-band to ARC satellite uplink
- San Francisco (ASF), Salt Lake City, and Pasadena reception and distribution terminals

EXPERIMENT 17: Health Education Television

Association of Western Hospitals

Objectives

- Develop cost-effective delivery of high-quality continuing education to health-care professionals in remote areas
- Develop an interconnect of existing metropolitan ground distribution systems to provide education simultaneously in multiple urban areas
- Develop live interaction of widely separated education centers
- Explore satellite utilization for health-related services (e.g., resource sharing, data sharing, and clinical consultation)

Technique

- One-way video (simplex) with as many as three audio channels
- Received satellite communication distributed via ITFS and cable in metropolitan areas

Ground System

- Program originates at ASF television studio, Menlo Park, CA
- S-band to ARC satellite uplink

- Menlo Park, Los Angeles, and Seattle reception and distribution terminals
- Additional sites to be determined

CANADIAN EXPERIMENTS

The overall objective of the SHF Communications Experiments is to explore the use of future high-power communications satellites. Experiments will be performed to demonstrate the ability to provide television and voice communications and wideband data transmission to remotely located, low-cost ground terminals. The experiments are expected to demonstrate a viable method for implementing a wide variety of sociological, economic, and scientific programs. Proposals for communications experiments generally fall into five broad categories:

- Education
- Medicine
- Community Development and Interaction
- Data Communications
- Technology

Over the operational life of 2 years, communications experiments will occupy the bulk of in-orbit time. The schedule will accommodate the requirements of more than 20 experimenters. Table IX-2 is a list of currently approved Canadian experiments and includes participation by federal and provincial government agencies as well as universities and industrial organizations.

Education Experiments

A variety of educational experiments are planned which cover the spectrum from elementary to college levels. Typical experiments are

- Testing the effectiveness of satellite communications system to increase access to university education
- Exploration of continuing education of public service personnel
- Exchange by Carleton University and Stanford University of undergraduate and graduate courses by using digital video compression techniques
- Evaluation of the feasibility of using a satellite to upgrade the mathematical competence of elementary school teachers located in remote areas

Medical Experiments

These experiments range from upgrading the capabilities of remotely located practitioners to performing real-time diagnoses through the satellite communications systems, as follows:

- Demonstration of the feasibility of providing continuing medical education and health education programs to physicians and communities located in remote areas
- Provision of consultation and diagnostician services by transmission of medical data such as electrocardiograms, X-rays, and vital signs
- Provision of audiovisual supervision of professional services to a remotely located hospital, including psychiatric consultations, data storage, and nursing and staff education
- Determination of the optimum use of two-way television in providing for the operation and management of a primary health delivery system in a remote area

Experiments in Community Development and Interaction

Linking together communities which are remote from each other and of diverse cultural backgrounds forms the basis for a group of experiments described briefly as follows:

- Demonstration of a frequency division multiple access system that will improve the two-way voice service to remote areas
- Demonstration of television coverage of special sporting events in real time
- Facilitation of the development of communications among Inuit settlements and provision of the means to demonstrate their needs and concerns
- Measurement of the increase in productivity effected by better communications between temporary camps and the head office
- Documentation of community-designed programs and comparison of these to existing communications services
- Demonstration of the benefits of an interchange of cultural and educational programs by linking two communities in the north of Saskatchewan and Quebec
- Evaluation of the benefits to native peoples in the Yellowknife Area in the Northwest Territories of having available to them, time-sharing computer facilities in London, Ontario
- Demonstration of the feasibility of interactive communications between Edmonton and native communities in northern Alberta, related to health, education, and community development

Data Communications Experiments

Experiments in this category range from signal-processing techniques to teleprocessing networks to evaluate the feasibility of future decentralization of government data processing. Typical experiments are

- Demonstration of the capability to transmit digital television and to implement simple modulation-demodulation (modem) equipment, which operates at very high rates of data transmission
- Testing the flexibility and reliability of telephony and data transmission
- Evaluation of communications links between remote areas of the Province of Ontario over which government personnel will transmit operational or administrative information related to government services
- Investigation of the feasibility of terminal-to-computer and computer-to-computer communications with a view to the future decentralization of government data processing in the Province of Manitoba
- Study of signal coding and signal processing techniques to evaluate their effectiveness in data communications
- Evaluation of the performance of a digital modem for high-rate data transmission

Technology Experiments

A variety of technological experiments will provide demonstrations of applications as diverse as transmitting the position of a train to providing information on the position of astronomical radio sources. These experiments are

- Measurement of propagation factors such as polarization isolation, low-angle effects, and precipitation attenuation
- Evaluation of the performance of a synchronization technique as a basis for making very accurate determinations of satellite position and orbit
- Evaluation of small ground terminals in a user environment to help plan future systems
- Technical measurement of signal reception in metropolitan areas
- Demonstration of the feasibility of radio conferences
- Investigation of the feasibility of using the CTS to transmit the position of a train
- Demonstration of the feasibility of using the CTS to provide a link between two radio telescopes to form a radio interferometer which will provide information on the structure and position of astronomical radio sources
- Investigation of the transmission capabilities and the station portability, reliability, and maintainability of the two-way voice terminals by using a terminal at Frobisher Bay on Baffin Island

Canadian Experiments Description

Each of the Canadian experiments is briefly described as follows:

EXPERIMENT F-1-1: Propagation Measurements
Communications Research Centre

Objectives
- Measure attenuation by precipitation using a signal transmitted by the satellite
- Measure attenuation indirectly from measurement of the sky-noise temperature
- Investigate signal fading at low elevation angles

Summary
- An experiment has been designed to investigate these three aspects of SHF propagation. A low-power, continuous-wave (CW) signal will be transmitted to the satellite from one or more of the 2-meter (6-ft) terminals located anywhere in Canada. The resulting satellite transmission will be received by a special 2.5-meter (7.5-ft) terminal equipped to measure two orthogonal polarizations. The data will be analyzed to determine the extent of polarization discrimination during precipitation and to correlate this discrimination with meteorological conditions. In addition, both direct and indirect measurements of attenuation will be used to develop and improve methods of predicting the probabilities of attenuation on earth-space paths. Variations in the delay of signals passing through the precipitation and scatter from hydrometers along the path will be measured. It is also hoped to conduct studies of low-angle fading and its correlation with meteorological conditions. Bell Northern Research is cooperating in the polarization discrimination aspects of this experiment.

EXPERIMENT F-1-2: Time-Division Multiple-Access Synchronization
Communications Research Centre

Objectives
- Demonstrate feasibility of novel TDMA synchronization scheme
- Measure timing accuracies of synchronization system
- Determine guard times required for data bursts
- Investigate natural phenomena and problems that perturb the system

Summary
- In the TDMA synchronization system being investigated, the position of the satellite is accurately measured by a control terminal cooperating

with three other terminals called ranging terminals. The measured position is then transmitted by the control terminal through the satellite to all terminals of the system. Any other terminal can then access the satellite accurately without resorting to the trial-and-error procedure normally required. This technique can, when applied to operational systems, make them more efficient and more reliable than the existing systems. The system shows particular promise for use with beamed satellite systems, where the problem of synchronization has not yet been satisfactorily solved.

In addition, this experiment serves as a basis for two other experiments: Demand Assignment and Orbit Perturbation Measurement. Once this synchronizing system is in operation, the Demand Assignment experiment will be carried out. The satellite positions calculated and measured will be recorded for off-line processing to study orbital details.

EXPERIMENT F-1-3: Frequency-Division Multiple-Access Demand Assignment - Two-Way Voice

Communications Research Centre

Objective - Demonstrate a frequency-division multiple-access demand system to improve two-way voice service to remote locations

Summary - This experiment will involve the testing and demonstration of a demand assignment algorithm and equipment by the placement of a arranged set of calls among terminals and between the terminals and the switched telephone network. Features to be tested and demonstrated will be priority, preemption, broadcast, and conference calls. Also to be tested is the sensitivity of the system to link and control station failures. These tests will be carried out over a period of 8 weeks and should provide enough information to specify both the demand assignment operating system and the necessary control equipment required for domestic multipurpose satellite systems.

EXPERIMENT F-1-4: High-Rate Data Experiment

Communications Research Centre

Objectives - Implement simple modulation-demodulation (modem) equipment at high data transmission rate
- Analyze the capability to transmit digital television

Summary - This experiment, F-1-4, has the primary purposes of constructing and testing a modem for high-rate data. This modem has been specified to be two-phase coherent PSK operating at 62.5 megabits/sec. This rate was selected to make it compatible with a rate required for digital television. The modem will be evaluated in the laboratory and by using CTS and 3- and 9-meter (10- and 30-ft) terminals. Bit error rate and burst error patterns will be obtained.

A digital television capability is available and such transmissions will be evaluated subjectively. A "dry run" of subjective tests was made in the spring of 1974. Similar tests will be conducted with the CTS link.

EXPERIMENT F-1-5: Small Terminal Evaluation
Communications Research Centre

Objective - Evaluate small ground terminals in user environment

Summary - The Small Terminal Evaluation Experiment will be undertaken by CRC with the cooperation of CTS experimenters and the installation and maintenance crew. The evaluation will cover installation, operation, and maintenance factors; transportability; survivability in the Canadian environment; radio interference; modulation techniques for voice; and feasibility of television reception with a 1-meter (30-in.) dish. The technical information gathered will contribute to a data base for the planning and design of future systems for television broadcast, voice, and other services.

EXPERIMENT F-2-1: Broadcasting Signal Reception in a Metropolitan Environment
Canadian Broadcasting Corporation

Objectives - Evaluate 12-GHz television reception with a 2-meter (7-ft) antenna in a metropolitan environment
- Demonstrate direct-to-home television reception with a 1-meter (30-in.) antenna

Summary - Television signals consisting of running programs, still slides, and test signals and originating from the CBC studios in Ottawa will be fed to CTS via the CRC main earth station at Shirley Bay. They will be received at the CBC studios in Montreal, where a television receive-only earth station with a 2-meter (7-ft) antenna and several individual-reception installations will be available at the Maison de Radio-Canada.

Measurements and demonstrations will take place during two 4-week periods in the summer of 1976 for 3 hours each day that the satellite is available to Canada. This experiment is expected to gather data regarding day-to-day variations of picture and sound quality and service reliability in a metropolitan environment when super-high-frequency (SHF), frequency-modulated (FM) television signals are beamed from a high-power satellite. Also, we expect to gain first-hand experience with home-appliance type of satellite reception apparatus, some of which will be made available by laboratories in Japan and, perhaps, Europe. The impact of these results is likely to affect technological decisions about future conventional and cable television systems.

EXPERIMENT F-2-2: Applications of Radio Broadcasting
Canadian Broadcasting Corporation

Objectives
- Demonstrate special-interest programming direct to individual broadcast stations
- Prove the relevance of extension of radio to, from, and between small communities

Summary
- In cooperation with other experimenters whose interests cover common ground, this experiment will demonstrate to a world radio conference the relevance of and the technical feasibility of CTS for radio broadcasting.

EXPERIMENT F-2-3: Special Television Demonstration
Canadian Broadcasting Corporation

Objectives
- Determine suitability of CTS for remote television broadcasting and control
- Assess transmitting and receiving from large populated areas
- Assess flexibility and handling efficiency of portable ground stations under the pressure of operations

Summary
- Pictures of the Olympic equestrian competitions will be transmitted from Bromont to Montreal, Quebec, at specific times during July 17 to August 1, 1976.

EXPERIMENT F-3: Continuing Education Learning System

Public Service Commission

Objectives
- Explore cost-effective, high-speed training techniques
- Develop a two-way educational television system
- Validate methods of professional development by using a learner-centered model

Summary
- The Staff Development Branch proposes to explore the use of two-way video as a means of linking professional training groups in shared expertise on an interprovincial or remote basis. We propose to do this by placing video cameras and receivers in the home offices of selected civil servants in a relatively remote test location (St. John's, NF) so that two or more such home office groups will be able to see each other and/or a similar professional training group in our Ottawa training laboratories. This exploration is not one of video linkage of a teacher to a receptive audience of learners. Instead the proposal is to explore the possibilities and difficulties inherent in the conduct of high-speed learning by means of audio-visual teleconference-type teacherless groups.

EXPERIMENT P-1-1: Community Interaction

Newfoundland Communications Technology Committee

Objective
- Document the use of CTS technology by community groups

Summary
- The Community Interaction broadcasts will be designed by the communities themselves, and so their content cannot be specified at this time. A number of results are expected: communities will develop comprehensive overviews of their communications needs for the first time, and pressure may be brought to bear on existing communication agencies to upgrade their services; communities will concentrate on existing education and government programs and their relevancy and may well agitate for changes in these programs; leadership and organizational skills will be developed; sophisticated communications technology will be "demystified"; and greater mutual awareness and cooperation may be promoted among communication and education agencies.

EXPERIMENT P-1-2: Telemedicine

Newfoundland Communications Technology Committee

Objectives
- Offer continuing medical education to remote areas
- Offer health education programs to remote areas
- Test the feasibility of transmitting medical data from remote areas

Summary
- The Telemedicine Experiments will be of three main types:

(1) Continuing medical education programs will be provided to practicing physicians in remote areas by one-way video but with an important component being the doctor's participation by two-way audio. By developing teaching sessions in a half dozen areas (therapeutics, anesthesia, cardiology, communication disorders, etc.) the experiment will determine whether it is possible to deliver the learning experience to physicians practicing in a remote area without their having to leave their place of work.

(2) Health education programs will be directed to several communities in the experimental area.

(3) By transmission of medical data such as electrocardiograms (ECG), X-rays (by slow scan), and vital signs, it will be determined whether first-line medical care performed by the recipients can be significantly improved at savings in time and money.

EXPERIMENT P-2-1: Community Communications in New Quebec

Ministry of Communications, Quebec

Objective
- Demonstrate real-time teleconferences with Inuit settlements

Summary
- Citizens' committees will be established in eight settlements to activate community interest and participation in the project. The communities will define the types of programs to be prepared by members of the communities themselves. A communications infrastructure will be developed in the eight communities to include the installation of radio and television equipment and the training of local residents in the technical and social animation aspects of community communications. Taped programs and live interaction will be interspersed to make full use of the interactive CTS communications system.

EXPERIMENT P-2-2: Telemedicine in New Quebec
Directorate of New Quebec

Objective - Provide support and advice to health-care personnel in remote areas

Summary - Bidirectional audio links will be established between the medical center of the University Laval in Quebec and three small Eskimo localities in northern Quebec. These links will make possible the transmission of audio, ECG, and X-rays. Highly specialized doctors will receive the information and will advise on the treatment to be given or on the need to transfer the patient to a hospital in southern Quebec.

EXPERIMENT P-2-3: Omnibus Network
University of Quebec

Objective - Test interactive system to increase access to university education

Summary - In order to ensure the widest accessibility to university education throughout the province, the University of Quebec is decentralized and operates out of various regional centers. A telephone network connects the regional centers and is used for teaching, management, and document exchange. The CTS experiment will test the utility of multidirectional communications, including broadband, for the development of the educational network. It will test short-term applications of such a network and will simulate educational applications expected to be operational in the 1980's.

EXPERIMENT P-2-4: Radiotelephony with Satellite Camps
Societe de Developpement de la Baie James/SOTEL

Objective - Test increase in productivity due to better communications between temporary camps and the head office

Summary - The Societe de Developpement de la Baie James (SDBJ) is doing surveys of different areas in the James Bay territory. This territory does not have communications facilities other than high-frequency (HF) radios. The survey group will stay in one place for periods of 2 weeks to 1 month, and it is not possible to install reliable telecommunications facilities. The SDBJ intends to use the 1-meter (30-in.) terminal to establish a telephone link between these temporary camps and its main site in the James Bay area. From there the telephone link would be connected to the national network.

EXPERIMENT P-2-5: Telephony and Data Transmission Studies

Research Institute of Hydro-Quebec

Objective - Test ability of 1-meter (30-in.) terminals to control remote generators

Summary - In many remote areas in northern Quebec Province, electricity is produced by diesel generators. At the present time these remote areas are linked by HF radio to the regional dispatching center in the city of Quebec. Hydro-Quebec proposes to establish between a few of these generators and the dispatching center in the city of Quebec a telephone link to transmit audio, facsimile, and slow-rate data.

EXPERIMENT P-3: Multiministry Administrative and Operational Experiment

Government of Ontario

Objectives - Determine capability of meeting communication needs in remote areas
- Improve government service to remote areas

Summary - Several experimental communications links will be established over which personnel from a number of Ontario Government ministries will transmit information of an operational or administrative nature. The communications links will be either exclusively satellite or in combination with the regular switched-point or "broadcast basis," using where appropriate the following communications modes:

(1) Point-to-point transmission modes: two-way audio, facsimile, teletypewriter, slow-speed data (including telemetering and physiological data), and video

(2) "Broadcast" transmission modes: audio, facsimile, and teletypewriter

"Test" transmissions will be conducted from the following city areas: Toronto and Thunder Bay; and from the following remote areas: Red Lake, Pickle Lake, Sioux Lookout, and Big Trout Lake and several base camps within the perimeter formed by Red Lake, Pikangikum, Round Lake, Fort Hope, and Thunder Bay. Should the quality and reliability of the CTS system prove satisfactory, appropriate observations will be made which would facilitate an assessment of the immediate and long-term impact which a system such as CTS could make in the delivery of government services in remote areas.

EXPERIMENT P-5: Government Teleprocessing Network
Province of Manitoba, Management Committee of Cabinet

Objectives
- Determine capability for terminal to computer communications
- Investigate high-speed digital data transmission

Summary
- For the satellite experiment the Manitoba Government Computer Centre will investigate, with the help of IBM Canada, Ltd., alternative methods of interfacing the CRC ground stations with the data processing systems at either end. An interactive data terminal equipped with a printer and a keyboard will be located in a government facility in Thompson, Manitoba. This terminal will communicate with the DOC-supplied 1-meter (30-in.) ground terminal on an audio-grade channel at 2400 or 4800 baud.

A 3-meter (10-ft) terminal will be located at the Manitoba Government Computer Centre in the Norquay Building in downtown Winnipeg. This ground station will be attached to the data processing system and the data files. The experiment will consist of a series of command transmissions and data retrievals from the data files located in Winnipeg. The successful demonstration of this capability will have a significant impact on the government's desire to decentralize its operations geographically. High-speed digital transmissions will be investigated between the 3-meter (10-ft) terminal in Winnipeg and the 9-meter (30-ft) terminal in Ottawa. In particular, the Manitoba Government Computer Centre will participate as a user in the TDMA experiment (F-1-2).

EXPERIMENT U-1: CTS Digital Video University Curriculum Sharing Experiment
Ames Research Center, Carleton University, and Stanford University

Objectives
- Demonstrate the ability to expand scope of instruction by sharing classes between universities
- Demonstrate curriculum sharing with student interaction
- Develop video compression with efficient coding and modulation
- Determine effects of rain attenuation on high-data-rate digital link

Summary - Carleton University and Stanford University will exchange courses, chosen from their respective course lists, by means of digital video compression techniques developed at the NASA Ames Research Center in California. The time allotted for the exchange is Monday to Friday from 1830 to 2000 hours est. Four of these days will be spent on course material and the fifth day will be used for housekeeping in order to deal with technical problems that may arise and also to experiment with teaching/learning innovations. There will be an emphasis on graduate engineering courses during the exchange, but individuals in other disciplines will be encouraged to become involved in the experiment.

EXPERIMENT U-2: A Satellite Communications Link for Train Location

Queen's University, Kingston, Ontario

Objectives
- Communicate control data with a moving train
- Evaluate this type of communications system in the railroad environment

Summary - There will be two experimental periods. During the first period, an antenna system, mounted on a platform designed to simulate train movements, will be used to communicate with the 9-meter (30-ft) facility at Ottawa. After the test period, modifications on the transmitting and receiving equipment will be made if necessary before the next observation period. The second phase, scheduled to be performed approximately 6 months later, will be conducted from a moving train. The equipment used in the first test will be mounted on a flat car; data will be exchanged with the Ottawa facility while the train is in motion. The success of the experiment will be measured by monitoring error rate, signal strength, and antenna beam position. The results will determine the feasibility of such a technique for continuous railroad operation.

EXPERIMENT U-3: Satellite-Link Radio Interferometry

University of Toronto, Toronto, Ontario

Objectives
- Investigate real-time correlation of data outputs from two widely separated radio telescopes
- Investigate the structure and variability of extragalactic radio sources at at resolutions from 10^{-2} to 10^{-4} second of arc

Summary - Several observing sessions will be conducted, depending on the availability of the satellite. During each session, two telescopes will observe the same cosmic source, and the broadband output of one telescope will be transmitted (by auxiliary antennas at the observatories) to the other observatory via the satellite. Here they will be combined (correlated) in real time to obtain the output "fringes" of the interferometer. The experiment will be begun with a relatively short baseline, from Algonquin Park, Ontario, to Green Bank, West Virginia. After the techniques have been demonstrated, the Owens Valley, California, telescope will be substituted for the West Virginia telescope for additional observations. The fringes will then be studied by the astronomers to infer information about the structure of the cosmic sources under investigation. Concurrently, at least in initial phases of the program, conventional tape-recorded observations will be carried out simultaneously over the same baselines in order to compare the results obtained by the two different techniques. It would also permit improvement of the precision of current measurements of cosmic source positions and of baseline lengths between radio telescopes. The solution of this problem will make possible major advances in man's understanding of the universe and has implications for future research in geophysics, particularly in plate tectonics, seismology, and geodesy.

EXPERIMENT U-4: Performance Evaluation of a Digital Modem for High-Rate Data Transmission

McMaster University

Objective - Evaluate a fast frequency-shift keying (FSK) modem for transmission of high-rate digital data

Summary - The experiment will involve the use of a fast FSK modem that is being designed by the Communications Research Laboratory of McMaster University. The modem is designed to operate at a nominal data rate of 60 megabits/second. It is expected that a variation of ±5 megabits/second will be possible with some degradation of performance. It will be connected to the terminals through an intermediate frequency (IF) interface operating at 735 MHz. By varying the rate of data transmission through the satellite, it would become possible to evaluate the effects of intersymbol interference caused by the filtering action of the channel. By varying the power level of the transmitted signal, it would become possible to evaluate the effects of nonlinearity in the channel. It is also proposed to test the suitability of the modem for the transmission of digitized color television data via the Communications Technology Satellite.

The results obtained from the experiment should demonstrate the achievable degree of closeness to the corresponding results obtained from theory and/or computer simulation. The aim in the modem design is to be within 1 decibel of the theoretically optimum performance attainable when the modem is operating in a back-to-back mode.

EXPERIMENT U-5: Exploration of a Satellite Computer Network to Supply Computerized Information to Native Peoples in Northern Canada

University of Western Ontario

Objectives
- Provide technical data to evaluate the feasibility of providing computer facilities for native people in the Far North
- Assist in establishment of native centers making use of a computer communications network

Summary
- An Information Resource Center at the Tree of Peace in Yellowknife, Northwest Territories, will be linked via CTS to a time-share computer in London, Ontario. Information will be exchanged in two directions over this network in such a way as to determine the optimum utilization of this resource. A problem of major importance is to determine to what extent information is to be distributed at the extremes of the network.

EXPERIMENT U-6: Telemedicine Experiment - Between a Remote Village (Kasechewan), a Northern Base Hospital (Moose Factory General), and a Health Science Center (University of Western Ontario)

University of Western Ontario

Objective
- Use a satellite to transmit and receive medical information with various-size terminals

Summary
- With the use of a 1-meter (30-in.) terminal at a remote station (Kasechewan) on James Bay, assistance will be given to a local nurse. The anticipated benefits include assistance with clinical examinations, help in making triage decisions, and making possible a level of medical service heretofore not considered possible or appropriate. With the use of a 3-meter (10-ft) terminal at the base hospital in Moosonee on James Bay,

the following services will be provided: supervision of a resident providing professional service such as anesthesia, transmission of image data to a Health Science Center for interpretation and consultation, and clinical consultation. The 2-meter (6-ft) terminal at the Health Science Center in London, Ontario, will be used to complete the link and to provide the necessary resources.

EXPERIMENT U-7: Studies and Evaluations of Signal Processing Techniques for Data Communication Via Satellite

University of Waterloo

Objective - Study signal coding and processing techniques

Summary - For a period of 1 month, coded and uncoded signals will be transmitted and received by using a Codex 9600 data modem at transmission rates of multiples of 2400 Hz, to 9600 Hz. Signal reception will be monitored on line. The received signal will either be processed on line or recorded in magnetic tape for off-line processing. The effect of signal coding (source and channel) will be evaluated.

EXPERIMENT U-8: Upgrading Mathematical Competence of Elementary School Teachers

Lakehead University

Objective - Study the feasibility of upgrading both the mathematics and the mathematics education of elementary school teachers in remote areas

Summary - A 1-hour interactive mathematics and mathematics education program will be broadcast. Each hour of satellite presentation will consist of a 20-minute video recorded lesson followed by a 10-minute two-way exchange by means of a two-way audio channel and a one-way telewriter signal. This will be followed by another 20-minute video recorded lesson and a second live two-way telephone and one-way telewriter link. There are to be regular assignments as well as a text to complement the satellite presentation.

EXPERIMENT U-9: Saskquebec Education - Culture

University of Regina/Quebec Ministry of Education

Objective - Interchange cultural and educational programs between two communities in diametrically different parts of Canada

Summary - The use of two 3-meter (10-ft) terminals in Zenon Park and Baie St. Paul to transmit two-way video and two-way audio programs will be investigated. Each community should organize itself prior to the experiment with the help of a social director. It is essential that this cooperative community development come from the members of the community themselves and not be something imposed from the outside. The communities will then be prepared for the dialogue implicit in the experiment itself. Thirty-two cultural and 16 educational programs between Zenon Park and Baie St. Paul over a 3-month period will be exchanged.

EXPERIMENT I-1: Transportable Telecommunications Systems

Bell Canada/Telesat

Objectives - Provide data on the transportation capability, portability, reliability, and maintainability of ground stations

- Demonstrate rapid turnaround for temporary service

Summary - The experimental program will basically consist of conducting a series of transmission tests to evaluate the overall satellite channel performance and transmission of telecommunication traffic between Ottawa and Frobisher Bay. Various types of telecommunication traffic will be transmitted over the satellite, and the user results will be evaluated. Tests carried out will be both technically and user oriented, providing a good cross section of the system capability.

EXPERIMENT E-1: Health Care Delivery to Remote Areas

Queen Charlotte Islands General Hospital, Queen Charlotte City, British Columbia

Objective - Determine optimum use of two-way television in providing a primary health system for remote areas

Summary - It is proposed to use the CTS to create communications linkages between the Queen Charlotte Islands General Hospital (QCIGH) and urban-based patient care, educational, and research resources. The project hospital is the only hospital serving the general population of the Queen Charlotte Islands. The critical factor restricting the performance of the project hospital is the lack of qualified personnel. As the current health system makes the distribution of medical specialists throughout rural areas untenable, the demonstration project proposes to explore the extent to which the effective scope of a basic health team can be increased through audio-visual and data communications to specialists in urban-based facilities.

EXPERIMENT E-2: Project Ironstar

Alberta Native Communications Society, Edmonton, Alberta

Objective - Determine and serve overall needs of native communities in Alberta's north

Summary - Project Ironstar provides for interactive communications between the studios of the Alberta Native Communication Society, in Edmonton, and some 30 native communities in Alberta's north. In the first phase, voice communications will be provided between two CTS terminals in remote locations and a similar terminal in Edmonton. A number of additional communities will be connected to the system via terrestrial facilities. In second phase, a video signal will be transmitted from Edmonton, to be received at three CTS terminals. From two of these terminals, in the High Level and Slave Lake areas, the signal will be retransmitted to a number of communities in the area. At the third terminal at Fort Chipewyan, the signal will be received directly. Provision will be made for voice return to Edmonton from any of the locations receiving the video signal.

Programming will be designed to reflect the freely expressed concerns and needs of all communities served. It is anticipated that these concerns will include health care, education, the legal system, community development and interaction, etc. The effectiveness of this program, including the introduction of new interactive techniques, will be evaluated on a continuous basis. In fact, the evaluation process is considered an integral part of the project and will later serve as a guide to determining optimum forms of communication systems for Alberta's north.

X. PROGRAM MANAGEMENT AND SUPPORT INTERFACES

AGENCY RESPONSIBILITIES

The Office of Applications (OA), NASA Headquarters, has been assigned overall responsibility for the United States portion of the CTS Project. The Associate Administrator for Applications, NASA Headquarters, is responsible for overall direction and evaluation. He has delegated Headquarters responsibility of this Program to the OA Director of Communications Programs. The Director of Communications Programs has designated A. J. Cervenka as the United States Program Manager. He is the primary point of contact in Headquarters for all matters relating to the CTS Program.

The Office of Tracking and Data Acquisition has responsibility for providing tracking and data acquisition support. The Office of Space Science will provide the launch vehicle. The Office of Manned Space Flight is responsible for providing the launch support operation.

The Canadian Department of Communications has been assigned overall responsibility for the Canadian portion of the CTS Program. The Assistant Deputy Minister for Space Programs is responsible for overall direction and evaluation. The Canadian Department of Communications has assigned program responsibility to the Communications Research Centre. Dr. I. Paghis has been designated as the Canadian Program Manager.

UNITED STATES - CANADIAN JOINT MANAGEMENT

The CTS joint management structure is shown in figure X-1. This Project is implemented under the general direction of the Assistant Deputy Minister for Space Programs of DOC and the Director of Communications Programs of NASA. The CTS Steering Committee consists of two representatives of DOC and two representatives of NASA. This group serves to review progress, to resolve disagreements, to make policy decisions, to make agency commitments, and to report to the Assistant Deputy Minister for Space Programs for DOC and the Director of Communications Programs of NASA. Reporting to the Steering Committee are the Project Managers from DOC and NASA. The Project Managers are the primary points of contact between the principal agencies and represent the level where efforts and commitments are centered.

A Joint Working Group (JWG) has been designated by NASA and DOC to be the principal mechanism for coordinating the execution of the project and for keeping all partici-

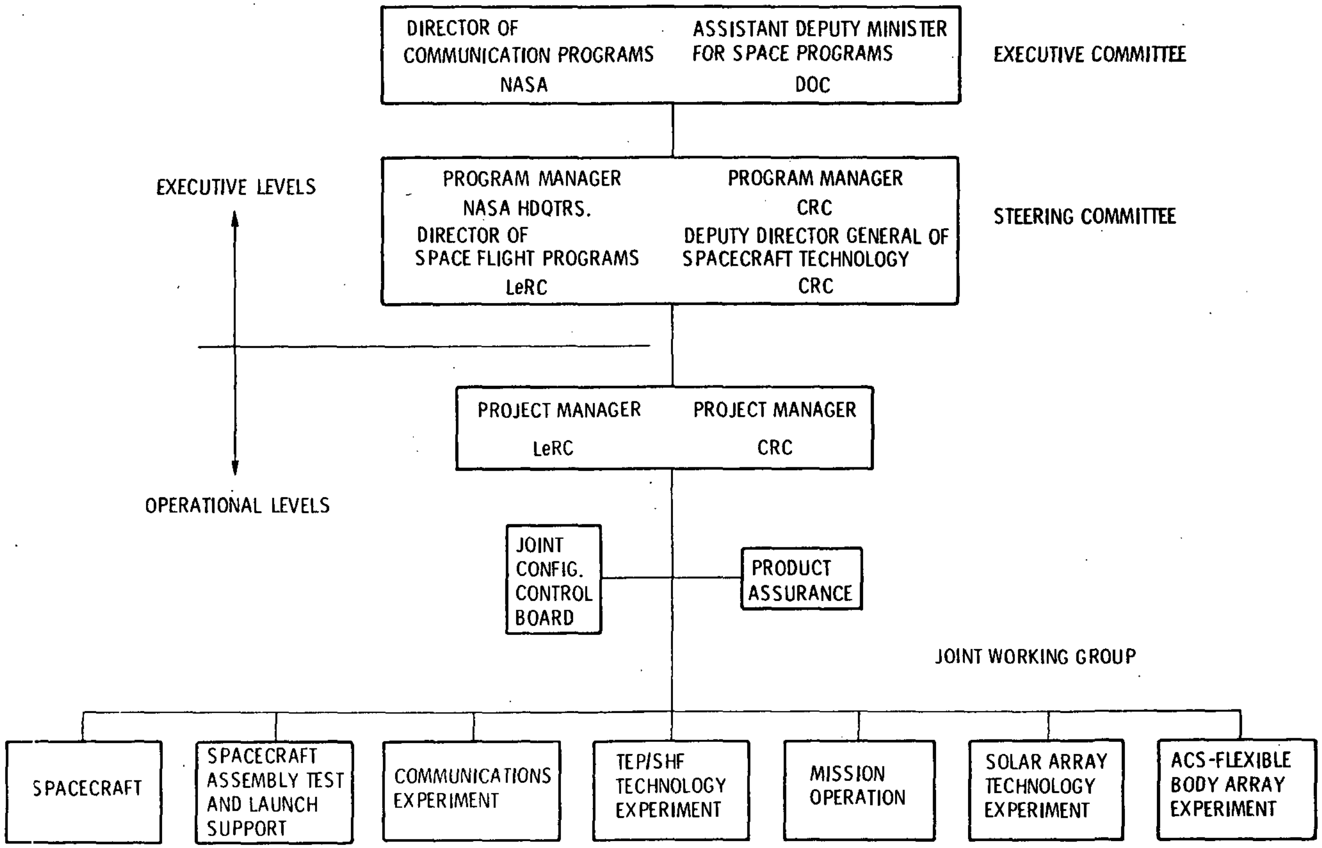


Figure X-1. - Communications Technology Satellite joint management structure.

pants informed of status. The JWG is a coordinating and reporting group and is not responsible for the project management. NASA and CRC each have their individual organizations for meeting the management responsibilities. The JWG chairmen, who are the CRC Project Manager and the Lewis Project Manager, are responsible for setting up such committees and subcommittees as may be required to carry out the coordination functions of the JWG. They must also provide the necessary support so that committees can effectively accomplish the functions assigned to them.

NASA PROJECT MANAGEMENT

Lewis Research Center has been assigned overall project management responsibility for the NASA portion of the CTS Project, reporting to the Office of Applications, NASA Headquarters, through the Communications Programs Directorate. The CTS Project Office of the Space Flight Programs Directorate has been assigned management responsibility. The Director of the Lewis Research Center has designated William H. Robbins as the CTS Project Manager. Support for the various specialized spacecraft disciplines required to effectively support the project are drawn from the Spacecraft Technology Division (STD) and the Office of Reliability and Quality Assurance.

The GSFC Mission and Data Operations Directorate has been requested to accept the mission operations responsibility for the CTS project. The Test and Evaluation Division of the Systems Reliability Directorate represents the GSFC in all matters concerning spacecraft environmental testing performed at GSFC. The Project Directorate, GSFC, represents GSFC in all matters pertaining to the Thor-Delta launch vehicle. The Director, GSFC, has arranged for Kennedy Space Center to provide launch operations support for the Thor-Delta launch vehicles.

CANADIAN DEPARTMENT OF COMMUNICATIONS PROJECT MANAGEMENT

The Canadian Communications Research Centre (CRC) has been assigned overall project management responsibility by the Canadian Department of Communications (DOC). Dr. John N. Barry is the CRC CTS Project Manager and has responsibilities similar to those of his NASA CTS counterpart, William H. Robbins.

MAJOR NASA SUPPORT INTERFACES

A Memorandum of Understanding between the Canadian Department of Communications and the United States National Aeronautics and Space Administration defines the responsibilities of each agency. Included in the NASA responsibilities are

- Providing a Thor Delta-class launch vehicle, conducting the launching into the agreed geostationary orbit, and providing the services required to achieve the launching
- Providing the heat shield and the spacecraft tiedown and separation mechanism
- Providing for the inclusion in the spacecraft of the superefficient TWT power tube and associated power conditioning and thermal interface equipment
- Providing ground facilities in the United States for the experimental program
- Establishing specifications and providing facilities for final spacecraft environmental and flight acceptance tests
- Acting as coinvestigator to the DOC in carrying out the Solar Array Experiment

Table X-I lists the major NASA support interfaces necessary to fulfill these responsibilities.

TABLE X-1. - MAJOR NASA SUPPORT INTERFACES

Responsibility	Responsible NASA Center	Office
Project management	Lewis	CTS Project Office
Launch vehicle	GSFC	Delta Project Office
Launch support	KSC	Unmanned Launch Operations
Launch services	KSC	Unmanned Launch Operations
Mission analysis	GSFC	Mission and Data Operations Directorate
Mission and data operation	GSFC	Mission and Data Operations Directorate
Network support	GSFC	Networks Directorate
Heat shield and separation mechanism	GSFC	Delta Project Office
Superefficient TWT and power conditioner	Lewis	Space Flight Programs Directorate
Ground facilities for experimental program	Lewis	Space Flight Programs Directorate
Environmental test facilities	Lewis	Space Flight Programs Directorate
	GSFC	Systems Reliability Directorate
Coinvestigator for Solar Array Experiment	Lewis	Space Flight Programs Directorate

MAJOR DEPARTMENT OF COMMUNICATIONS SUPPORT INTERFACES

Included in the DOC responsibilities identified in the Memorandum of Understanding are

- Design, fabrication, integration, and testing of the spacecraft and its subsystem except for the superefficient TWT power tube and its associated power conditioning and thermal interface equipment
- Providing, integrating, and testing the apogee-motor subsystem

- Providing flight-qualified spares of critical spacecraft subsystems and spacecraft ground checkout equipment
- Providing tracking, data acquisition, command, and control after the spacecraft is placed in geostationary orbit
- Providing ground facilities in Canada for experimental programs

APPENDIX - ABBREVIATIONS AND ACRONYMS

ACE	Attitude Control Electronics
ACEA	Attitude Control Electronics Assembly
ACS	Attitude Control System
AGC	Automatic Gain Control
AM	Amplitude Modulation; Apogee Motor
AMF	Apogee-Motor Firing
AMFA	Apogee-Motor Firing Attitude
APRC	Appalachian Regional Commission
ARC	Ames Research Center (NASA)
ASF	Archdiocese of San Francisco
ATS	Applications Technology Satellite
AWH	Association of Western Hospitals
AZ-EL	Azimuth and Elevation Adjustable Pedestal
BAL	balanced line
baud	signal speed - binary, bit/sec
BC	British Columbia
BER	Bit Error Rate
BOL	Beginning of Life
BSA	Body Solar Array
BW	bandwidth
CBC	Canadian Broadcasting Corporation
CCIR	Consultive Committee International Radio
CIF	Central Instrumentation Facilitiy
CKAFS	Cape Kennedy Air Force Station
C/N	Battery charging rate, capacity/(time in hours)
COMSAT	Communications Satellite Corporation
CRC	Communications Research Centre (Canada)
CTS	Communications Technology Satellite

CW	Continuous Wave
D/A	Digital to Analog
dBm	decibel, referred to milliwatts
dbW	decibel, referred to watts
DDA	Digital Differential Analyzers
DIGS	Delta Inertial Guidance System
DNCC	Denver Network Control Center
DOC	Department of Communications (Canada)
DS	Decoder Select
DSA	Deployable Solar Array
E	Emissivity
ECG	Electrocardiogram
EEC	Experiment Evaluation Center
EHC	Essential Housekeeping Converter
EIA	Electrical Integration Assembly
EIRP	Effective Isotropic Radiated Power
EJB	Electrical Junction Box
EMI	Electromagnetic Interface
EOL	End of Life
EPC	Experiments Power Conditioner
ESC	Essential Secondary Converter
ETA	Explosive Transfer Assembly
ETR	Eastern Test Range
ESRO	European Space Research Organization
FDMA	Frequency-Division Multiple Access
FETA	Field-Effect Transistor Amplifier
FM	Frequency Modulation
FRMS	Federation of Rocky Mountain States
FOV	Field of View
FSA	Flexible Solar Array

FSK	Frequency-Shift Keying
GCE	Ground Checkout Equipment
GSFC	Goddard Space Flight Center (NASA)
G/T	Gain/Temperature
HEW	Health, Education, and Welfare Administration
HF	High Frequency
HPBW	Half-power beam width
HSD	High-Speed Data
HTE	High-Thrust Engine
IBM	International Business Machine Corporation
IDSCP	Interim Defense Satellite Communication Program
IF	Interface; Intermediate Frequency
ISIS	International Satellite for Ionospheric Studies
ITFS	Instructional Television Fixed System
JBSA	Jettisonable Body Solar Array
JWG	Joint Working Group
KSC	Kennedy Space Center (NASA)
LCR	Launch Control Room
LSB	Least Significant Bit
LSD	Low-Speed Data
LTE	Low-Thrust Engines
LV	Launch Vehicle
MCR	Mission Control Room
MDC	Multistage Depressed Collector; Mission Director's Center
MHC	Main Housekeeping Converter
MISCON	Mission Control Center
MODEM	Modulation-Demodulation
MSB	Most Significant Bit
MSOCC	Multisatellite Operations Control Center
MWA	Momentum Wheel Assembly

MWC	Momentum Wheel Converter
N	integer
N/A	Not Applicable
NASA	National Aeronautics and Space Administration
NASCOM	NASA Communications Network
NATO	North American Treaty Organization
NDA	Nutation Damper Assembly
NESA	Nonspinning Earth Sensor Assembly
NF	Newfoundland
NIH	National Institute of Health
NOCC	Network Operations Control Center
NRZ	No Return to Zero
NSS	Nonspinning Sun Sensors
OA	Office of Applications (NASA Headquarters)
OST	Output Stage Tube (TWT + MDC = OST)
P	Parity
PAA	Pan American Airlines
PCM	Pulse Code Modulation
PCM/FM/PM	Pulse-Code Modulation/Frequency Modulation/Phase Modulation
PCM/PSK/AM	Pulse-Code Modulation/Phase-Shift Keying/Amplitude Modulation
PCM/PSK/FM	Pulse-Code Modulation/Phase-Shift Keying/Frequency Modulation
PCM	Power Control Unit
PF	Protoflight; Power Factor
PIA	Perigee Injection Attitude
PM	Phase Modulation
PMS	Premessage Synchronization
P-P	Peak to Peak
PPS	Power Processing Subsystem
PSK	Phase-Shift Keying
PSK/PCM	Phase-Shift Keying/Pulse-Code Modulation

PRN	Pseudorandom Noise
PSU	Power Switching Unit
RCS	Reaction Control System
REM	Reaction Engine Module
RF	Radiofrequency
RTCS	Real-Time Computer Systems
RX	Receiver
S&A	Safe and Arm
SALINET	Satellite Library Information Network
SATE	Solar Array Technology Experiment
S/C	Spacecraft
SCO	Subcarrier Oscillator
SDBJ	Societe de Developpement de la Baie James
SES	Spinning Earth Sensors
SHF	Super High Frequency
SPA	Solar Panel Assemblies
SPC	Secondary Power Converter
SM	Secondary Module
SR	Shift Register
SSA	Sun Sensor Assembly
SSE	Sun Sensor Electronics
SSM	Second-Surface Mirrors
SSS	Spinning Sun Sensor
STDN	Spaceflight Tracking and Data Network
SUN	Satellite User Network
TB	Transmit Band
TBD	To Be Determined
TBI	Through-bulkhead Initiation
TCU	Thruster Control Unit
TCS	Thermal Control System

TDA	Tunnel Diode Amplifier
TDMA	Time-Division Multiple Access
TEC	Transient Event Counter
TEP	Transmitter Experiment Package (OST + PPS = TEP)
TLM	Telemetry
TOE	Transfer Orbit Electronics
TT&C	Telemetry, Tracking, and Command
TTY	Teletype
TWT	Traveling Wave Tube
TX	Transmitter
USB	Unified S Band
V	Voice
VA	Veterans Administration
VAL	validate
VHF	Very High Frequency
VSWR	Voltage Standing Wave Ratio
VTR	Video Tape Recorders
VCHPS	Variable-Conductance Heat-Pipe Subsystem
WAMI	Washington, Alaska, Montana, Idaho
WHECON	Wheel Control (of S/C momentum and attitude)