POLARSAT

CONCEPT DEFINITION STUDY FOR
A POLAR ORBITING
COMMUNICATION SATELLITE SYSTEM
PROVIDING COMPLETE ARCTIC COVERAGE

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1.0 INTRODUCTION

1.1 Scope

This report describes the system design for a constellation of polar-orbiting satellites that will augment the communication service provided by a geosynchronous spacecraft operative in the UHF frequency band. The additional service is intended for users in the north of Canada above 75° N latitude, however, the system can provide service at lower latitudes as well.

This study does not attempt to fully optimize the satellite system design. The objective is to perform a conceptual analysis to establish a good baseline system configuration. Emphasis has been placed on a low-cost system which is compatible with the requirements of a UHF satellite communication system. This additional system is termed "POLARSAT".

1.2 Methodology

The study has been performed in four interlocking phases. The first three phases are described in this volume, and consists of a mission analysis study, a communications system design and a spacecraft design. The last phase is reported in a second volume, and contains the capital cost estimates for the system calculated in 1978 dollars.

1.3 Polarsat System Requirements

Table 1-1 lists the minimum system requirements of the Polarsat System. In the second column are listed the capabilities of the design presented in this report. It can be seen that a significantly greater capability than the minimum requirement can be met. This additional capability is not gained at the expense of greater complexity of the spacecraft, but merely reflects the relaxed constraints on mass that NASA Space Transportation System launches can provide.

NOTE: The Space Shuttle is the only launch vehicle that is expected to be readily available in the early 1980's for polar launches with adequate payload capability. For this reason, the shuttle has been chosen as the baseline launch vehicle in this study.

	REQUIRED	PRESENT DESIGN
o Coverage	Spherical triangle bounded by 60°W long., 142° W long., 75° N lat.	Northern hemisphere above 54° N Latitude, partial coverage to 40°N Lat.
o Communication capacity	- Minimum 2 full duplex channels with 16 Kbit/sec capacity - Compatible with each segment of an UHF geosynchronous satellite - transpond EPIRB signals	- up to 8 16 Kbit/sec full duplex channels - SHF 2 Mbit/sec fixed-to-fixed data channels - 2 frequency hopped 16 K bit full duplex channels - capability of one full duplex high power UHF channel for aircraft communication - transpond EPIRB signals
o RF Bandwidth	- 15 MHz maximum	- channelized receivers at UHF - 15 MHz minimum at SHF
o Uplink, Downlink frequencies	- compatible with UHF geosynchronous satellite	- compatible with UHF
o Availability	- 99% probability of availability	- 3 for 2 in-orbit redundancy, 24 hour continuous coverage

POLARSAT SYSTEM REQUIREMENTS

TABLE 1-1

2.0 MISSION ANALYSIS

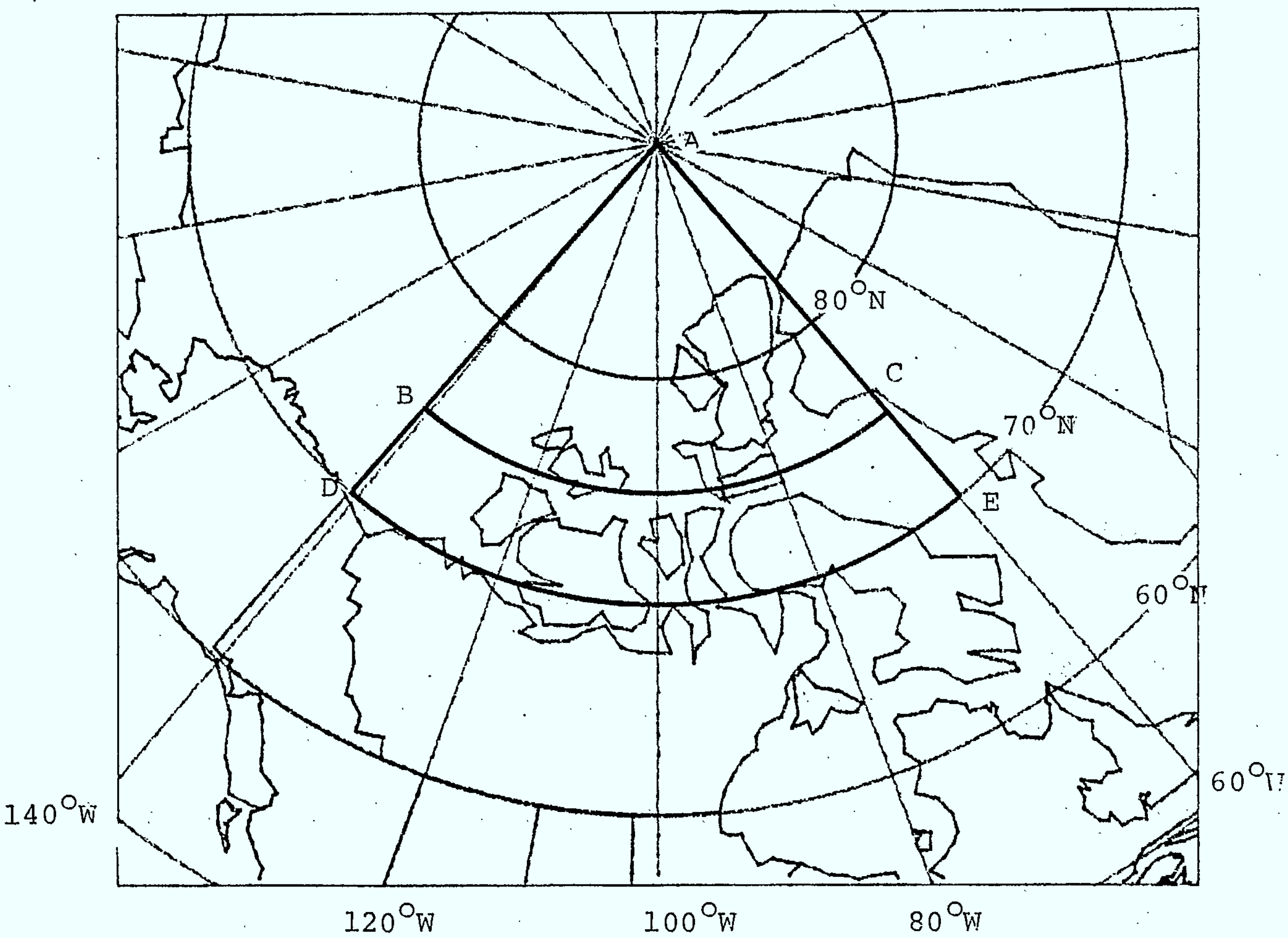
2.1 System Performance Parameters

The primary purpose of the Polarsat system is to augment the coverage area of the MUSAT communications satellite system. It is important that the system be of low cost while operating within the technical constraints imposed by the system configuration for MUSAT. This section outlines the factors affecting both the fundamental performance parameter - coverage of the Canadian North - as well as other important performance parameters which affect the Polarsat system configuration. The section also provides the basis for laying out several alternative spacecraft/orbit/attitude configurations to be considered for this mission.

2.1.1 Coverage

As a minimum, the Polarsat system must provide coverage of Canada above 75° North latitude, between longitude 60° West and 142° West (see Figure 2-1-1). Additionally, it is desirable to cover Canada above 70° North latitude and to allow a central ground station to be located in a logistically convenient location south of the seventieth parallel (preferably Cold Lake, Alberta at 54.4°N, 110.2°W). Polarsat must provide this coverage on a continous basis with full backup capability in order to provide the required availability.

The most important factor affecting coverage is the position of the spacecraft while it is being used for communications. This position is in turn governed by the satellite orbit. A spacecraft orbiting the earth travels, to a close approximation, in an elliptical path, with the centre of the earth at one of its foci. At any given time, the orbit can be simply described



ABC - REQUIRED COVERAGE ZONE
ADE - DESIRABLE COVERAGE ZONE

CANADIAN NORTHERN COVERAGE ZONE

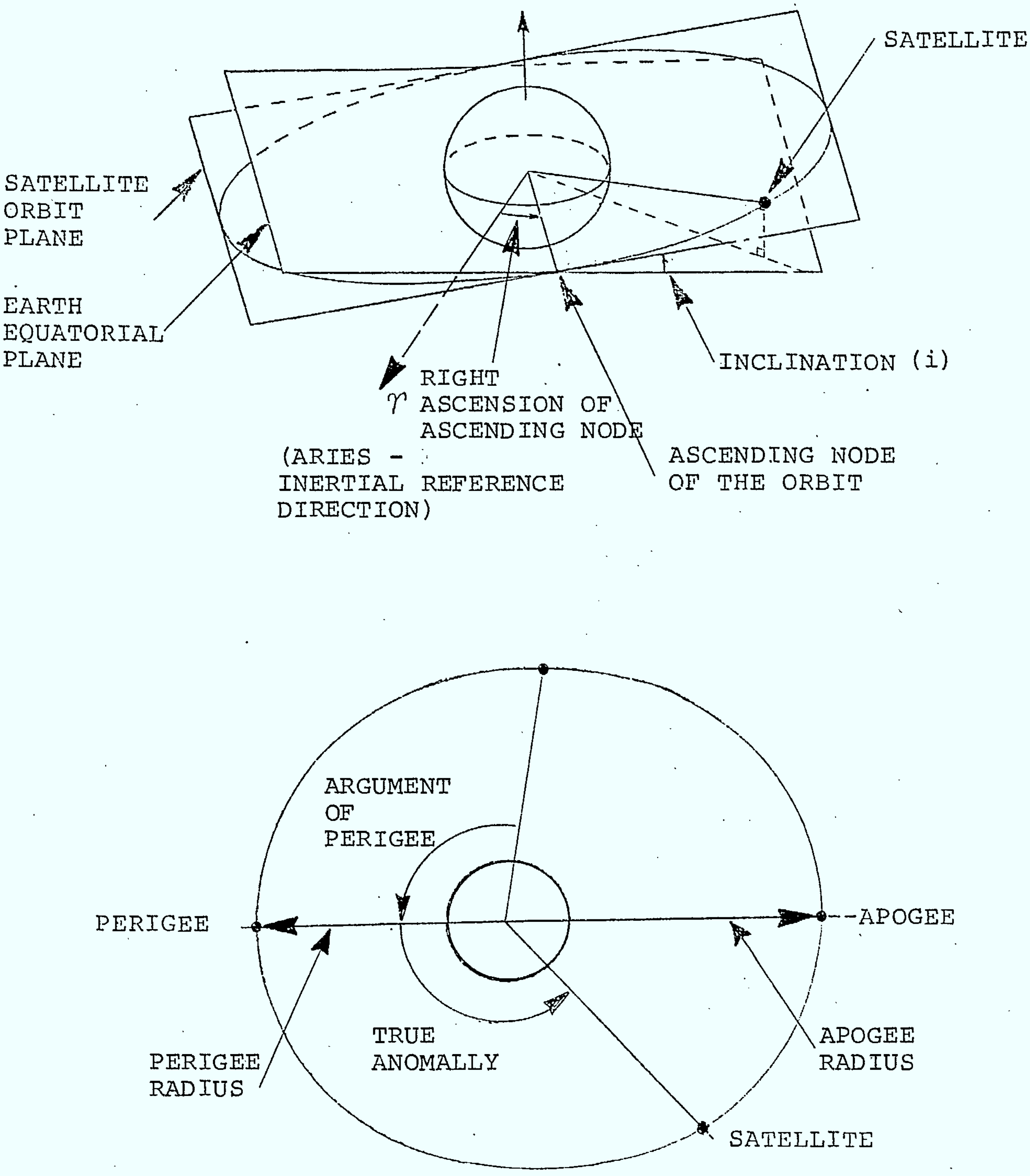
FIGURE 2-1-1

by six parameters (see Figure 2-1-2), apogee radius, perigee radius, inclination, right ascension of ascending node, argument of perigee and true anomaly.

A desirable orbit for the POLARSAT mission has a large inclination angle so that the satellite is positioned at high latitudes for at least a part of its orbit. It is also desirable that the altitude of the satellite be at least high enough to permit all the earth stations in the coverage area, plus the central ground station to view the satellite simultaneously. In addition, if the orbit is significantly elliptical, it is desirable to have an argument of perigee which locates the apogee, or high point, over Canadian latitudes. In order to keep down the number of satellites required for coverage, the satellite must spend a large portion of its orbital period within the range of true anomalies which permit the required coverage. This latter requirement can be met if, during communications operations, the satellite is high above the earth stations and moves slowly across the sky.

2.1.2 Earth Station Tracking

In order to utilize the non-tracking MUSAT earth terminals with a minimum of operational difficulties, the spacecraft should preferably remain within the field of view of the ground based antennas for several hours without requiring antenna adjustment. It is also desirable to minimize the number of antenna pointing directions required in any day, and to make them repetitive from day to day. In addition, from the point of view of ease of controlling the satellites, switch-overs from one satellite to another should be minimized.



SATELLITE ORBIT PARAMETERS

FIGURE 2-1-2

These requirements suggest that an orbit be chosen which results in a slow motion of the satellite across the sky, and that the orbital period be equal to the earth rotation period or to some multiple or sub-multiple of that period.

2.1.3 Protection Against Radiation Damage

The orbit for the POLARSAT mission must be chosen to avoid heavy radiation damage to the solar cells. The earth's magnetic field provides a mechanism which traps electrons and protons in a toroidal belt about the earth. The charged particles are held in a region about the equator extending in geomagnetic latitude over about 150 degrees and in altitude from the top of the atmosphere to the outer limits of the magnetosphere (see Figure 2-1-3).

For circular, equatorial orbits, the orbit altitude should be less than 1500 km or greater than 10,000 km in order to avoid excessive degradation of the solar array. Inclined and elliptical orbits pass through widely varying regions of radiation intensity and detailed analysis must be performed on each specific case. In general however, inclined orbits intercept less of the intense radiation field, and highly elliptical orbits can be designed to spend a comparatively short period of time in the regions of intense radiation.

2.1.4 Satellite Eclipsing

Operation of a solar-powered spacecraft while in the shadow cone of the earth requires that power be drawn from an on-board battery. To minimize battery subsystem costs, it is desirable to choose an orbit which avoids shadowing, at least during the periods of peak power requirements - i.e., while the transponder

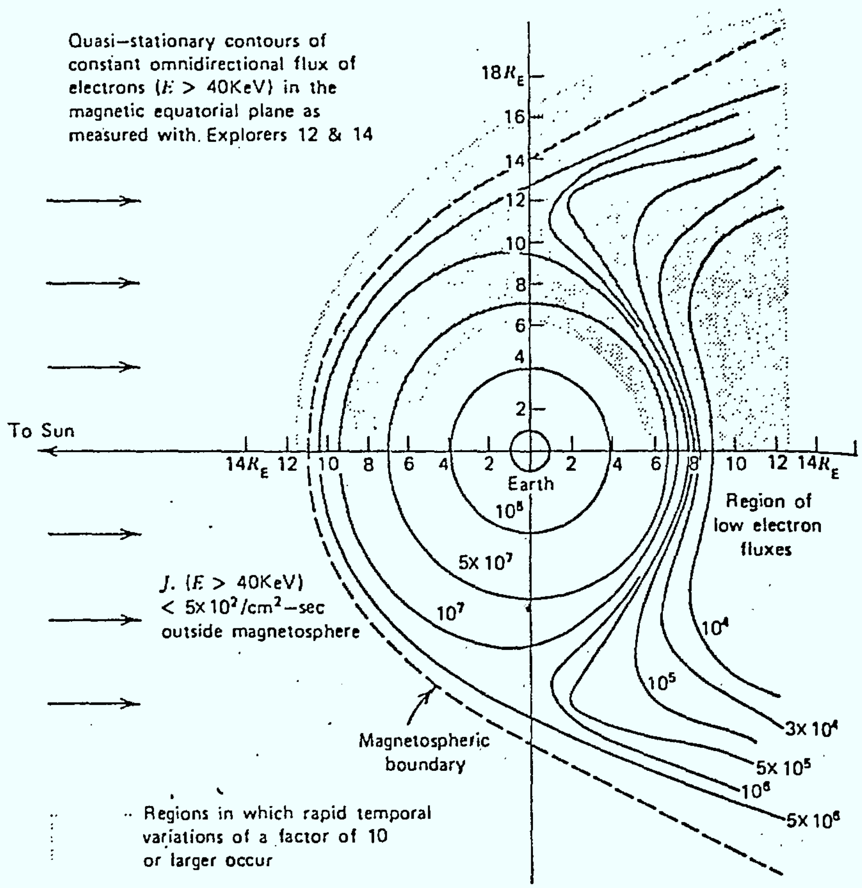


Figure 7.9 Approximate distribution of geomagnetically trapped electrons in the magnetic-equatorial plane. Fluxes shown are electrons per square centimeter per second. L. A. Frank, J. A. Van Allen, and E. Macagno, *Journal of Geophysical Research*, 1963.

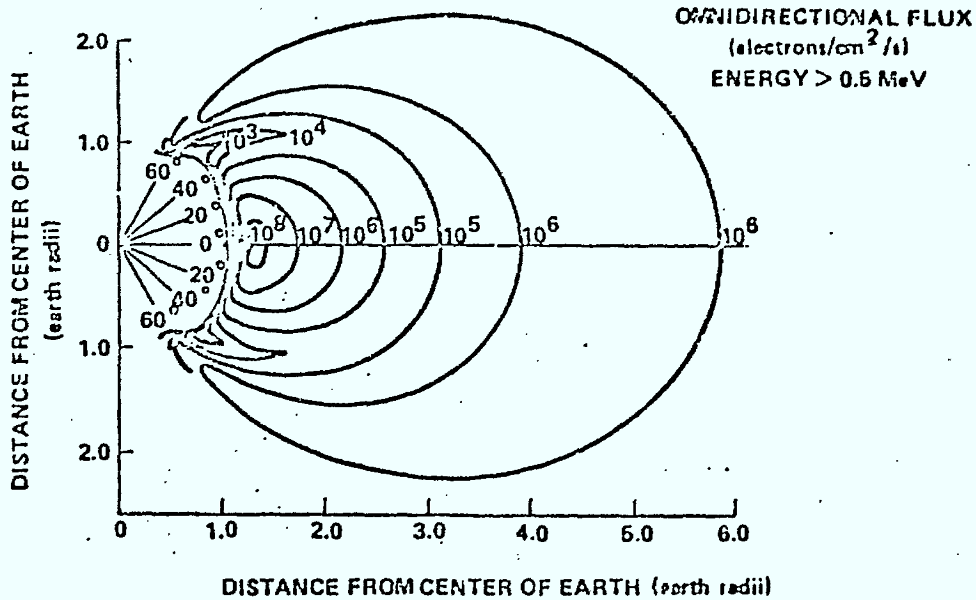


Figure II-16. Electron distribution in the earth's field (published by Vette in August 1964).

GEOMAGNETICALLY TRAPPED ELECTRONS

FIGURE 2-3-1(a)

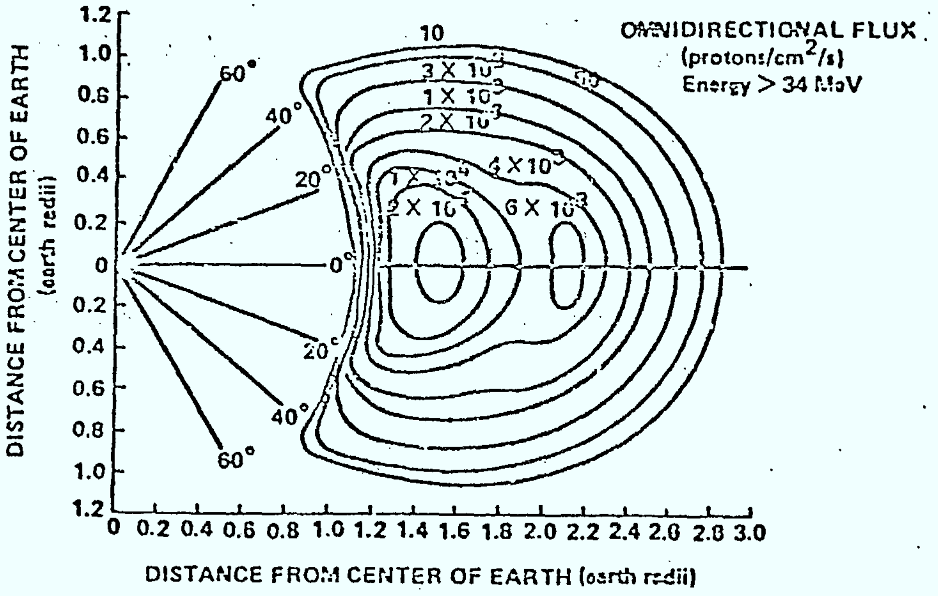


Figure II-17. Proton distribution in the earth's field (published by Vette in September 1963).

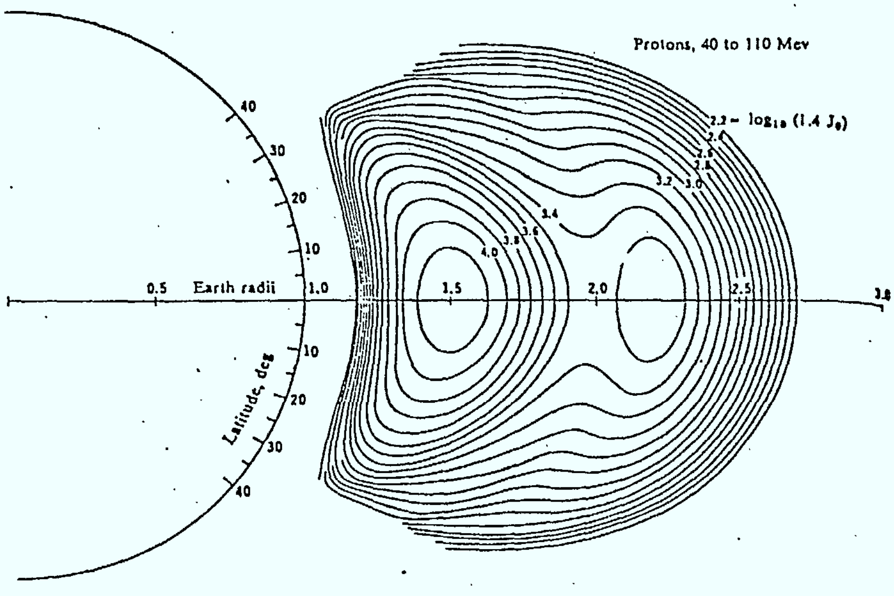


Figure 7.7 An R-λ plot of the data obtained for protons with the spacecraft Explorer 15. C. E. McIlwain, *Science*, 1966. Copyright 1966 by The American Association for the Advancement of Science.

GEOMAGNETICALLY TRAPPED PROTONS

FIGURE 2-3-1(b)

is operating. Additionally, if the satellite is well out of eclipse before the transponder is required, the problems of thermal transients caused by large variation in power dissipation can be avoided.

2.1.5 Attitude Stabilization

In order to utilize a spacecraft antenna subsystem with directive gain, either the spacecraft or the antenna must be capable of being pointed towards the Canadian coverage zone. Generally, the simplest configuration is one in which the spacecraft forms a stable platform for the antenna. The solar panel design of the spacecraft can be simplified by the choice of attitude stabilization. This is accomplished by orienting the spacecraft to limit the range of possible angles at which solar radiation can strike the surfaces on which solar cells can be mounted. This restriction in range will also prevent shadowing of the solar panels by the spacecraft antenna, and may also ease the thermal design.

The orbit/attitude configuration should also result in infrequent or automatic correction of attitude with small fuel/power requirements.

2.1.6 Orbit Stability

From the point of view of reducing both operational orbit maneuver requirements and fuel requirements, it is desirable to choose an orbit not requiring frequent or large orbit corrections. Orbits may be significantly perturbed by atmospheric drag, earth gravity field, solar and lunar gravity fields and solar radiation pressure.

The altitude of a satellite is reduced by the drag forces experienced in passing through the atmosphere. The amount of the drag force varies as the atmospheric density, the effective cross sectional area of the spacecraft and the second power of the satellite velocity. Generally speaking, drag forces begin to be appreciable for long duration missions at orbital altitudes below about 500 km.

The earth is not a perfect sphere. There is a significant equatorial bulge (oblateness) as well as variations with longitude. For inclined orbits, the bulge causes an appreciable rotation of the right ascension of ascending node. The bulge also causes a rotation in the argument of perigee of an elliptical orbit. There is, however, a critical inclination, about 63.4°, at which there is no perigee rotation due to oblateness (below an inclination of 63.4°, the perigee rotation is in the opposite direction to that when the inclination is greater than 63.4°).

Satellites in geosynchronous orbit are significantly affected by the earth's asymmetries with longitude, by solar and lunar gravitation fields and by solar radiation pressure. These orbits require periodic corrections for control of inclination, eccentricity and orbital period.

Satellites in highly elliptical orbits are appreciably affected by solar and lunar gravitation fields which can cause the perigee to drop into the high drag region of the atmosphere. This in turn causes the orbit to further decay.

2.1.7 Shuttle Launch Constraints

The Space Shuttle is normally to be launched from the John F. Kennedy Space Center in Florida, placing satellites into low altitude (200-400 km) circular orbits. The orbits can range in inclination from 28.5° to a maximum (due to range safety precautions) of 57°. Higher altitude and elliptical orbits are possible but at a cost of decreased Shuttle payload mass capability, and with the possible requirement for extra orbital maneuvering subsystem (OMS) kits. The maximum Shuttle payload weight of 65,000 lbs. (for a 28.5° inclination orbit) drops to about 56,000 lbs. for a 57° inclination orbit.

Higher inclination Shuttle orbits (56° to 104°) are possible in launches from Vandenberg Air Force Base which becomes operational in 1982. Payload capability drops to approximately 40,000 lbs. for a Shuttle orbit inclined by 90° to the equator.

Optional Shuttle equipment, pertinent to this mission, includes a spin stabilized upper stage (SSUS-D) for launching Delta class geostationary payloads and a remote manipulator system (SRMS) for payload handling. Deployment from the Shuttle may utilize the SRMS, the SSUS-D cradle, or a mechanism designed into the cradle tailored for this mission. For orbits significantly higher than 200-400 km circular, an on-board spacecraft propulsion subsystem(s) and/or the SSUS-D is required. An on-board orbit control propulsion system may also be used for this orbit boost maneuver.

The Shuttle payload bay is 15 feet in diameter, and the charges for launch are derived from a formula in which the cost is proportional to either the fraction of the 60 foot length of the payload bay or the fraction of available payload mass required. The former fraction tends to dominate and this in turn tends to make the spacecraft design somewhat pancake shaped to save on launch costs.

2.1.8 Other Factors

Several other factors will affect the selection of the satellite spacecraft/orbit/altitude configuration.

- It is desirable to minimize the relative velocity between the ground stations and the spacecraft since the resulting doppler frequency shift requires an increase in the guard bands between channels on the communications transponder.

- To minimize path loss, it is desirable to keep the orbital altitude low.

- To facilitate satellite control, it is desirable to view the satellite over wide areas for monitoring telemetry, for commanding and for tracking and orbit determination.

2.2 Spacecraft/Orbit/Attitude Configuration Selection

In order to meet the derived performance requirements of the POLARSAT mission within the constraints imposed by the MUSAT mission the following configuration must be met:

- Coverage
  - high inclination
  - high altitude
  - slow satellite angular movement
- Earth Station Tracking
  - slow satellite angular movement
  - orbital period - 24 hour on multiple or submultiple of 24 hours
- Radiation Damage
  - orbit below ∿ 1500 km or
    above ∿ 10,000 km or
    highly elliptical
- Eclipsing
  - satellite out of earth shadow during transponder operation
- Attitude Stabilization
  - Stable platform and directive antenna
  - minimize sun angle range
  - minimize frequency of attitude maneuvers
  - minimize fuel/power required for attitude maneuvers

- Orbit Stability

  - minimize frequency of orbit control maneuvers
  - minimize fuel requirement for orbit control maneuvers

- Shuttle Launch Constraints

  - Shuttle orbit altitude 200 - 400 km
  - Shuttle orbit inclination 28.5° - 104°
  - deploy using SRMS, SSUS-D or POLARSAT specific mechanism
  - SSUS-D and/or spacecraft propulsion required to boost spacecraft from shuttle orbit
  - pancake shaped spacecraft

- Doppler Velocity (Velocity of spacecraft along line of sight)

  - low earth station/satellite doppler velocity

- Path Loss

  - low altitude (n.b. this is in direct conflict with the higher priority coverage requirement)

- Satellite Control

  - satellite visibility for large percentage of the orbit period

The majority of the above configuration requirements are related to orbit, and therefore the four basic system configurations preliminarily examined in this section are oriented around orbit types.

2.2.1 Low Altitude Mission

Placing the POLARSAT spacecraft into a circular polar orbit directly from the Shuttle has the advantage of launch simplicity. However, since this orbit altitude is normally 400 km or less, the lifetime of the satellite will only be 1 to 2 years before re-entry into the atmosphere due to aerodynamic drag forces. In order to maintain the satellite in orbit, either the shuttle must go into a higher orbit or the spacecraft must have an on-board propulsion system to raise or to maintain the orbit. Since the Shuttle payload capability is drastically reduced by going to a higher orbit, and since a method of orbit correction may be desirable in any case, an on-board, propulsion system appears to be the best choice. In order to improve coverage and reduce drag, the orbit should be raised to about 1500 km, just below the intense Van Allen Radiation Belts. The minimum and maximum one way path lengths for this orbit are 1500 and 4100 km; and the required full cone angle for the spacecraft antenna is 107.5° minimum- all reasonable parameters. Due to the circular orbit, attitude stabilization for the mission could use the simple, passive gravity gradient method, although this would cause some problems if periodic orbit control is required.

This low orbit does however have some severe limitations. Most importantly, even at 1500 km the longest orbital arc over which a satellite is visible to a single ground station is only about 17.4% of an orbital period. This requires a minimum of 6 satellites even if the orbits could be phased to allow continuous coverage. For the POLARSAT mission, in order to overcome the additional requirements imposed by the requirements for mutual visibility between the ground stations in the coverage zone and a central station, to allow for backup and to eliminate phase problems, a much larger number of satellites are required.

In a low altitude orbit, satellites move rapidly across the sky causing tracking problems and doppler velocity is fairly high - 5.73 km/sec. ($\sim$0.002% of c). Further, there are frequent eclipses every day many of which will occur during transponder operation. Due to the large number of spacecraft, frequent switchovers from one satellite to another are required causing operational problems.

Based on coverage/ number of satellites plus operational difficulaties in controlling and using the satellites, the low circular orbit mission must be ruled out.

2.2.2 Medium Eccentricity, Medium Altitude Mission

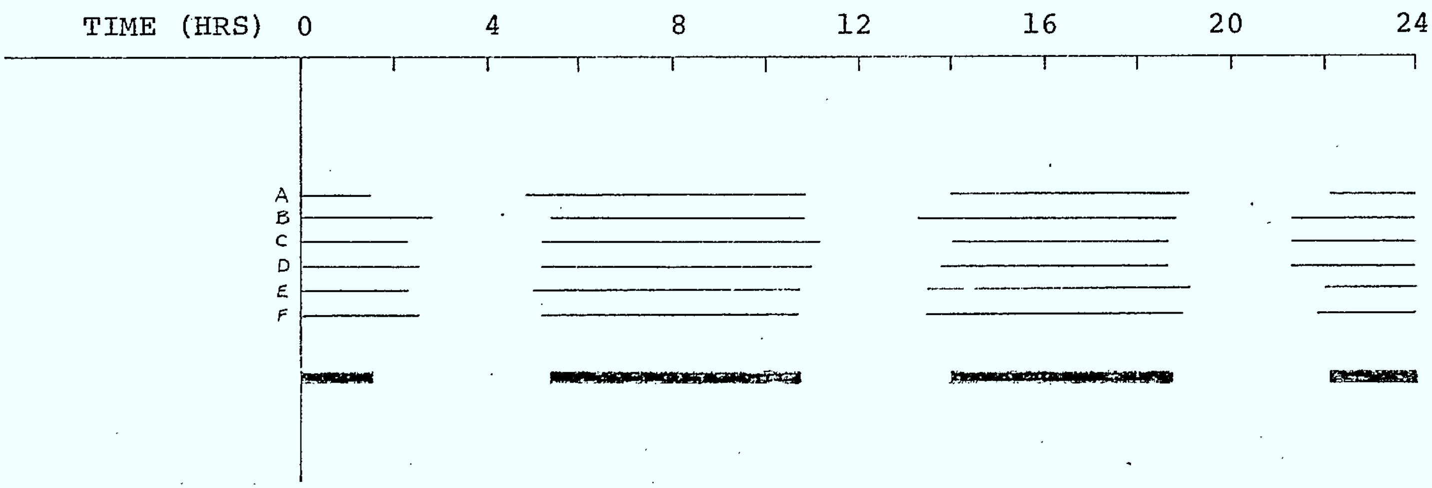
In order to reduce the number of satellites required for coverage, an orbit could be established at about 10,000 km, which is above the intense radiation belt zone. Additionally, the orbit can be made more elliptical than the low altitude orbit, which was constrained to lie within an altitude range of about 500 to 1500 km. This ellipticity results in a lower satellite velocity when the satellite is in the higher altitude portion of its orbit because the kinetic energy in the lower part of the orbit is transferred to gravitational potential energy in the higher portion. This means that not only is the satellite in view for longer periods of time during the favourable high altitude portion of the orbit, but also the satellite appears to move more slowly across the sky. In order to maintain the apogee over Canadian latitudes, without consuming excessive amounts of propellants, the orbit must be at the critical inclination of 63.4°. Choosing an orbit with a perigee altitude of 10,000 km and an apogee of 24,000 km results in an 8 hour orbit which just meets the Canadian coverage requirements (with a central station at Cold Lake) using a two satellite system (see Figure 2-2-1). The satellite still moves fairly rapidly around apogee resulting in a wide range of earth station pointing angles during a pass (up to about 70° azimuth and 60° elevation). Doppler velocity is fairly low (about 1.6 km/sec. maximum) and the minimum and maximum one way path lengths are 18,000 km and 29,000 km, respectively. The satellite experiences some eclipsing when transponder operation is required and requires a battery subsystem capable of operating the spacecraft for at least 1/2 hour per orbit. Except for battery management, there are no severe satellite control problems. The orbit eccentricity is too high to permit gravity gradient stabilization, and either spin stabilization or three-axis stabilization is required. A dual-spin configuration can handle the power requirements and is likely the cheapest alternative. Some on-board propulsion is required, not only for periodic orbit control due to perturbations, but also to raise the spacecraft from the Shuttle orbit. Two large maneuvers are required for orbit acquisition, and at least one should be by use of a solid rocket motor.

As mentioned previously, two satellites just barely provide adequate coverage. Four satellites are required to maintain 24 hours coverage, allowing for one spacecraft failure.

While the medium altitude, medium eccentricity mission results in improved performance over the low altitude mission, there are still significant difficulties in following the satellite from the ground stations, in satellite eclipsing and in orbit acquisition.

2.2.3 High Inclination Geosynchronous Mission

The normal geostationary mission has the advantage of 24 hour coverage, over large areas, with only very slight



o Thin lines represent coverage of stations:

A - Cold Lake, Alberta
B - North Pole
C - 70^{o}N, 60^{o}W
D - 75^{o}N, 60^{o}W
E - 70^{o}N, 142^{o}W
F - 75^{o}N, 142^{o}W

o Thick lines represent mutual coverage for all six stations

o One satellite
o 8 hour orbit

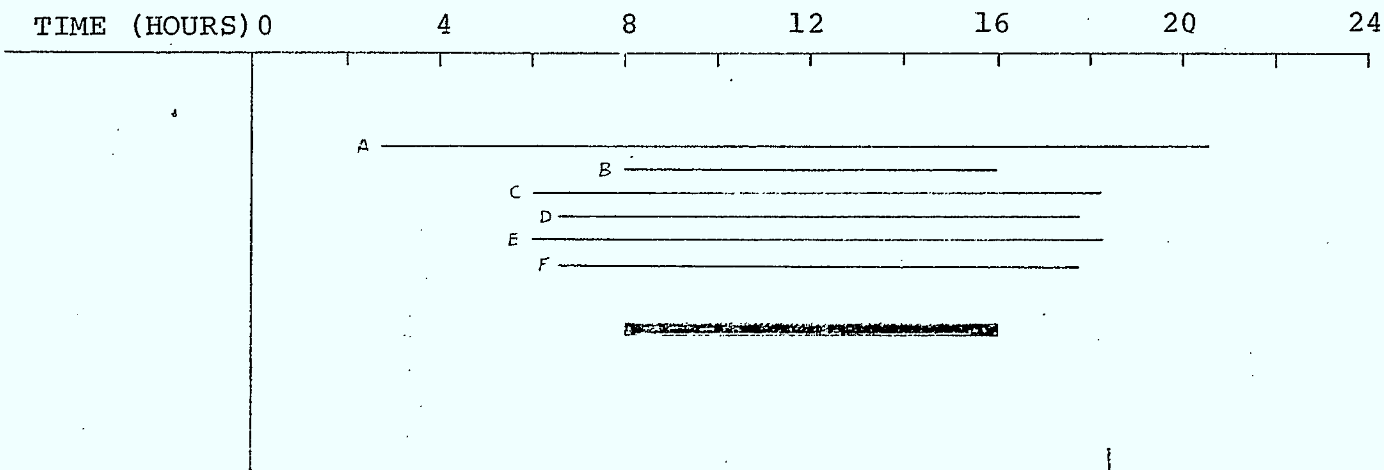
TYPICAL MEDIUM ALTITUDE ORBIT COVERAGE

FIGURE 2-2-1

apparent satellite motion. However, since the satellite is located over the equator, it cannot cover the polar regions. On the other hand, a satellite in an inclined geosynchronous orbit at high inclination can cover the polar regions for at least a part of its orbit. The spacecraft could be launched like a geostationary satellite from the normal 28.5^{o} inclination shuttle orbit into a highly elliptical transfer orbit. At apogee however, instead of firing the apogee motor to take out the orbit inclination as well as raising the perigee to synchronous altitude, the apogee motor can be oriented in such a manner that the inclination is unchanged or even increased.

For an inclined, 24 hour orbit, circular, a minimum of three satellites are required to permit operation in the Canadian Northern Coverage Zone (see Figure 2-2-2). The range of acceptable inclination angles for three satellite coverage is 28.5^{o} (the optimum shuttle orbit inclination) to 90^{o}. The three satellites just barely provide coverage at 28.5^{o} inclination and six are required for backup in the event of the failure of a single satellite. For inclinations of 62^{o} and higher, five satellites are required. A four satellite system results in gaps of about two hours when a satellite fails. One possibility is to place the satellites into elliptical 24 hour orbits at the 63.4^{o} critical inclination, - i.e., a higher altitude version of the mission discussed in Section 2.2.2.

Satellites in 24 hour orbits experience eclipses during transponder operation and require a battery subsystem. Like the previously described mission, the range of ground station pointing angles during a pass (for one satellite failed operation) is fairly large. Path lengths are comparable to a geostationary mission, doppler frequency shift is very low and radiation damage is easily dealt with. Attitude stabilization could be dual spin, and orbit determination and control requirements



o Thin lines represent coverage of stations

A - Cold Lake, Alberta
B - North Pole
C - 70°N, 60°W
D - 75°N, 60°W
E - 70°N, 142°W
F - 75°N, 142°W

o Thick line represents mutual coverage for all six stations

o One satellite

o Geosynchronous inclined (28.7°) orbit

TYPICAL INCLINED GEOSYNCHRONOUS ORBIT COVERAGE

FIGURE 2-2-2

are not a problem (inclination control can likely be avoided).

The synchronous mission results in good performance, but with a requirement for at least 4 and possibly 6 orbiting satellites for high reliability.

2.2.4 Critical Inclination, High Eccentricity Mission

The performance of the missions described in sections 2.2.2 and 2.2.3 improve with increasing inclination. To simplify orbit acquisition problems and get good coverage performance, an orbit with a low perigee (near the Shuttle orbit altitude) and a high apogee (over Canadian latitudes) should be considered. For such an orbit, satellite motion is slow near apogee when required for communications coverage, and fast near perigee when passing through the radiation belts. For orbits with periods between 4 and 12 hours, 2 or 3 satellites are required for coverage without backup (fewer satellites for longer orbital periods) and 3 to 6 satellites to ensure high reliability, 24 hour coverage. As well as good coverage, low satellite angular motion and avoidance of radiation, the high eccentricity mission has potential for avoidance of operation during eclipses and also results in reasonably low doppler frequency shifts.

Because of the highly eccentric orbit, gravity gradient attitude stabilization is impractical, and a dual-spin design is probably most suitable. A solid propellant motor on this spacecraft could be used to boost the spacecraft into the highly elliptical orbit, and an on-board auxiliary propulsion system may be used to boost the perigee slightly from the Shuttle orbital altitude, and to correct any orbit errors. In normal operation, the propulsion system may be used to make periodic orbit and attitude corrections.

The critically inclined, high eccentricity mission appears to offer the most promise of meeting the mission performance requirements at low cost.

2.3 Performance Analysis

Based on the preliminary mission analysis, the results of which were outlined in Section 2.2, the critically inclined, high eccentricity mission has been chosen as the most promising candidate. Section 2.3 presents the results of some more detailed performance analysis required to define the spacecraft/orbit/altitude configuration.

2.3.1 Coverage Analysis

Several different critically inclined, high eccentricity orbits were chosen for further analysis. All were chosen with a perigee altitude of 750 km, well above the atmospheric drag region. Table 2.3.1 shows the orbit parameters varied in this study. A number of computer analysis coverage runs were performed considering the station locations shown in Table 2.2.3. Station #1 is the desirable location of the central ground station, Cold Lake Alberta. Station #2, 4 and 6 are at the corner points of the required Northern Coverage Zone and station #2, 3 and 5 are at the corner points of the desired Northern Coverage Zone. Figure 2-3-1 combines the coverage (visibility vs time) results for all six orbit cases and all six stations for a period of one day and for a typical <u>single</u> satellite. The first six lines for each case represent the visibility times for station 1 through 6 and the heavier seventh line represents the mutual visibility of all six stations required for normal operation. For all six orbit cases the mutual visibility is most severely restricted (about 2 hours per day) by the Cold Lake station. Table 2.3.3 presents the average period of mutual visibility each day for the six cases. From the table it is obvious that at least 3 satellites are required for coverage, without backup, using the 4 hour orbit, and at least 2 are required

TABLE 2.3.1

ORBIT PARAMETERS

Case	Period	Apogee Height
I	4 hours	6414.7 km
II	6 hours	10385.3 km
III	8 hours	13929.3 km
IV	9 hours	15588.2 km
V	10 hours	17186.6 km
VI	12 hours	20232.1 km

TABLE 2.3.2

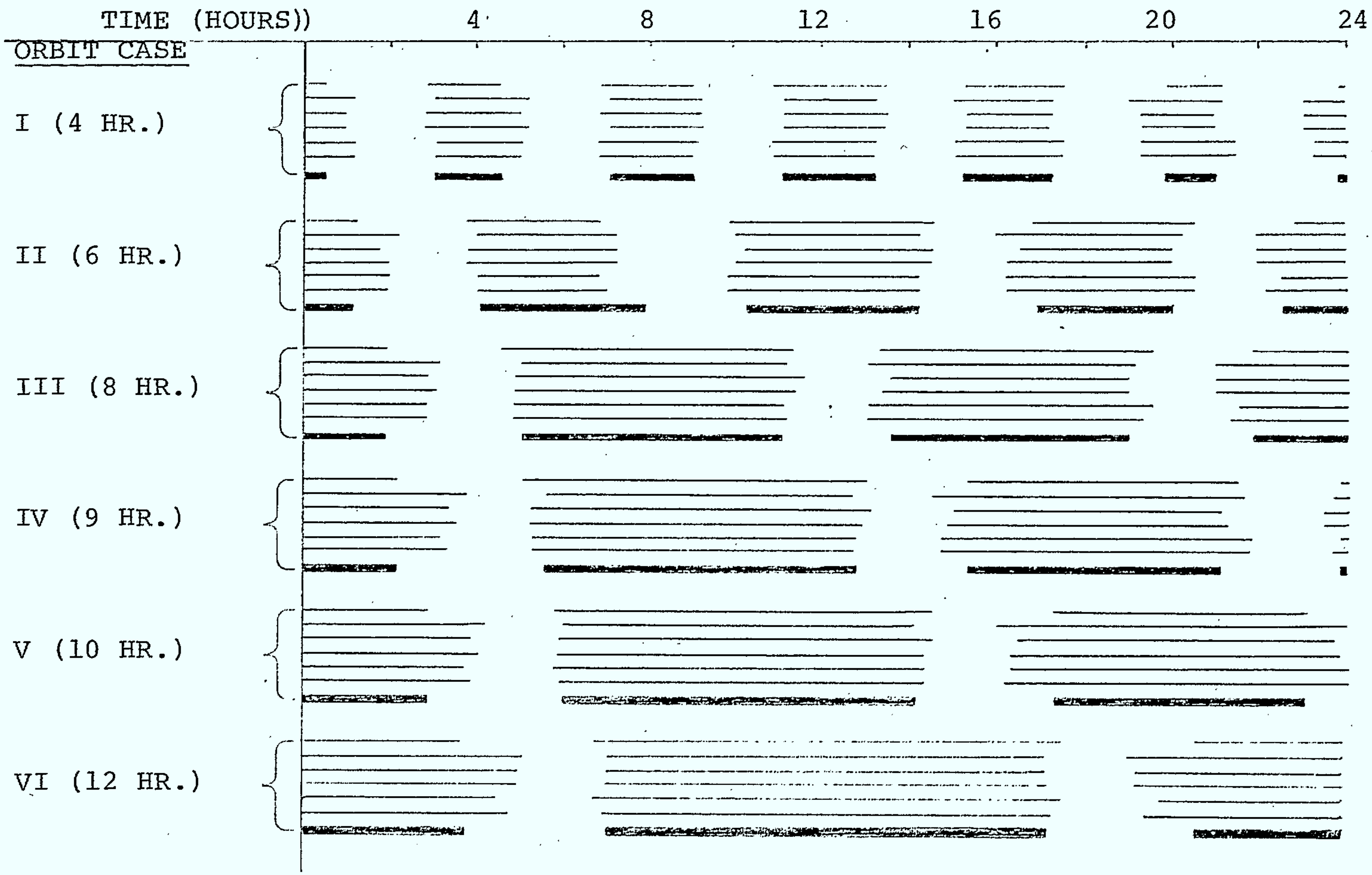
STATION LOCATIONS

Station #	Latitude	Longitude
1	54.4° N	110.2° W
2	90.0° N	arbitrary
3	70.0° N	60.0° W
4	75.0° N	60.0° W
5	70.0° N	142.0° W
6	75.0° N	142.0° W

TABLE 2.3.3

AVERAGE DAILY TOTAL TIME OF MUTUAL COVERAGE

Orbit Period	Visibility Time	% of 24 Hours
4 hours	9.5	40
6 hours	13.6	57
8 hours	15.5	65
9 hours	16.3	68
10 hours	16.8	70
12 hours	18.3	76



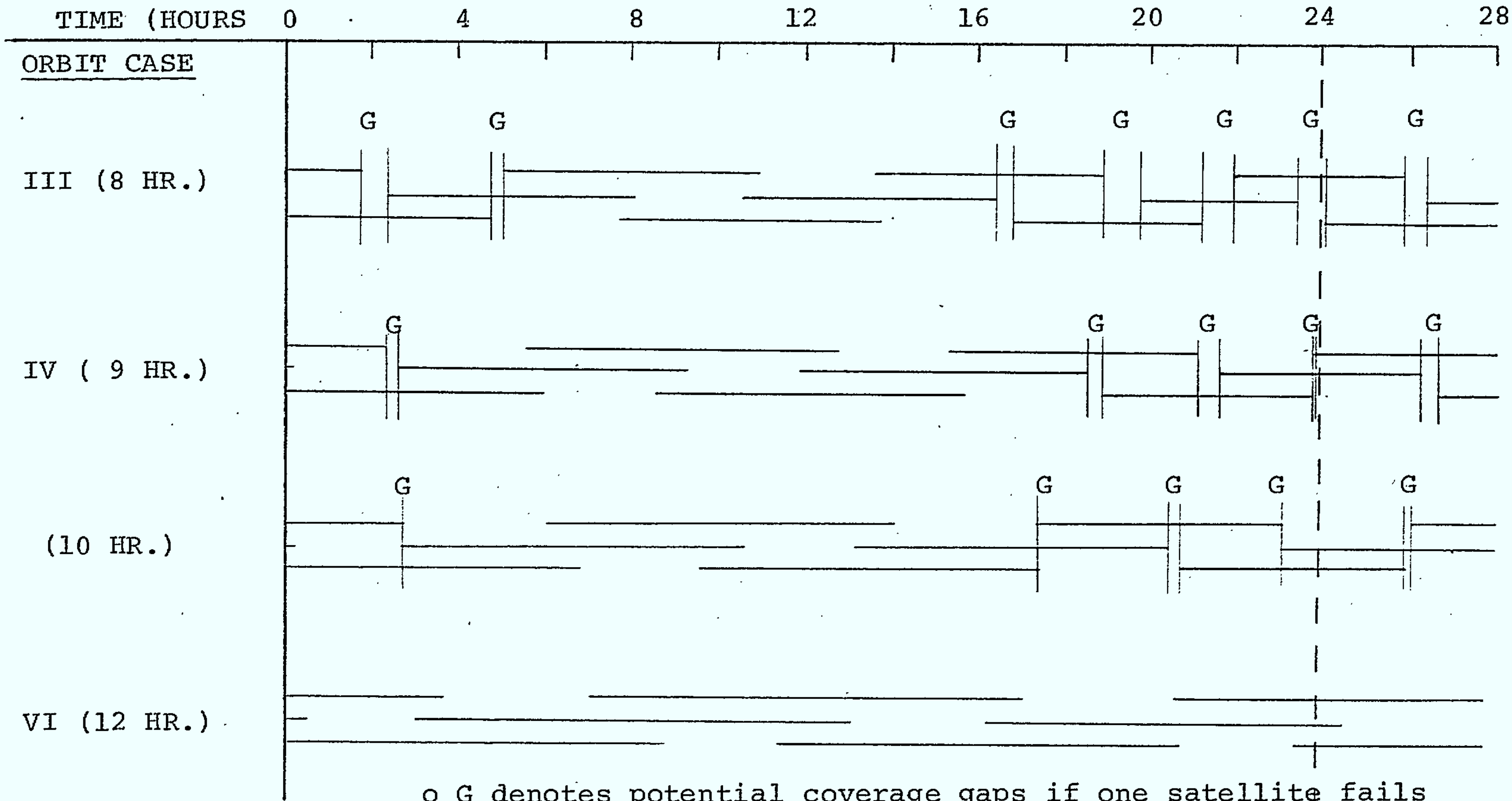
(COVERAGE TIMES BASED ON 5 DEGREE MINIMUM ELEVATION ANGLE)

TYPICAL SINGLE SATELLITE COVERAGE TIMES

FIGURE 2-3-1

for the rest. Therefore, the 4 hour orbit has been rejected. In order to provide backup, a 3 satellites for 2 redundancy scheme is proposed. This requires that each satellite be visible for at least twice as long as it is out of view - ie., it should be visible for at least 16 hours per day. On this basis, the 6 hour orbit must be rejected, and the 8 hour orbit is not acceptable if the Cold Lake station is desired. In order to further test the practicality of the 3 for 2 redundancy scheme, further coverage runs were made for these configurations - the results of which are shown in Figure 2-3-2. This figure shows that for the one satellite failed case, there are coverage gaps not only for the 8 hour orbit, but also for the 9 and 10 hour orbits. This is despite the fact that the latter two have orbits with an average duty cycle ratio of coverage to non-coverage in excess of 2:1. The reason for the gaps is that the actual individual visibility periods and gaps are not constant so that the effective instantaneous duty cycle ratio may actually be less than 2:1. This variation in visibility period lengths is due to the fact that some apogees occur at Canadian longitudes giving relatively longer coverage than when the apogees occur over the other side of the globe. The coverage restriction for the 8-10 hour orbits is due to the limitation imposed by the southern (Cold Lake) station. These latter orbits do provide adequate coverage for the stations above 70°N latitude.

The 12 hour orbit has the advantage of being a submultiple of 24 hours, and therefore the satellite tracks, in earth coordinates, will repeat from day to day. (n.b. for closest repetition of tracks, the orbit period should be about 11.967 hours - one half of a sidereal day, or the time taken for the earth to rotate 369° in inertial space). In normal operation, the 12 hour orbit is used nominally from two hours before apogee to two hours after apogee, and covers Canada down to latitudes of approximately 40°N. In one satellite failed operation, the satellite is



o G denotes potential coverage gaps if one satellite fails
o Coverage times are based on 5 degree minimum elevation angle

TYPICAL COVERAGE TIMES WITH 3 FOR 2 REDUNDANCY

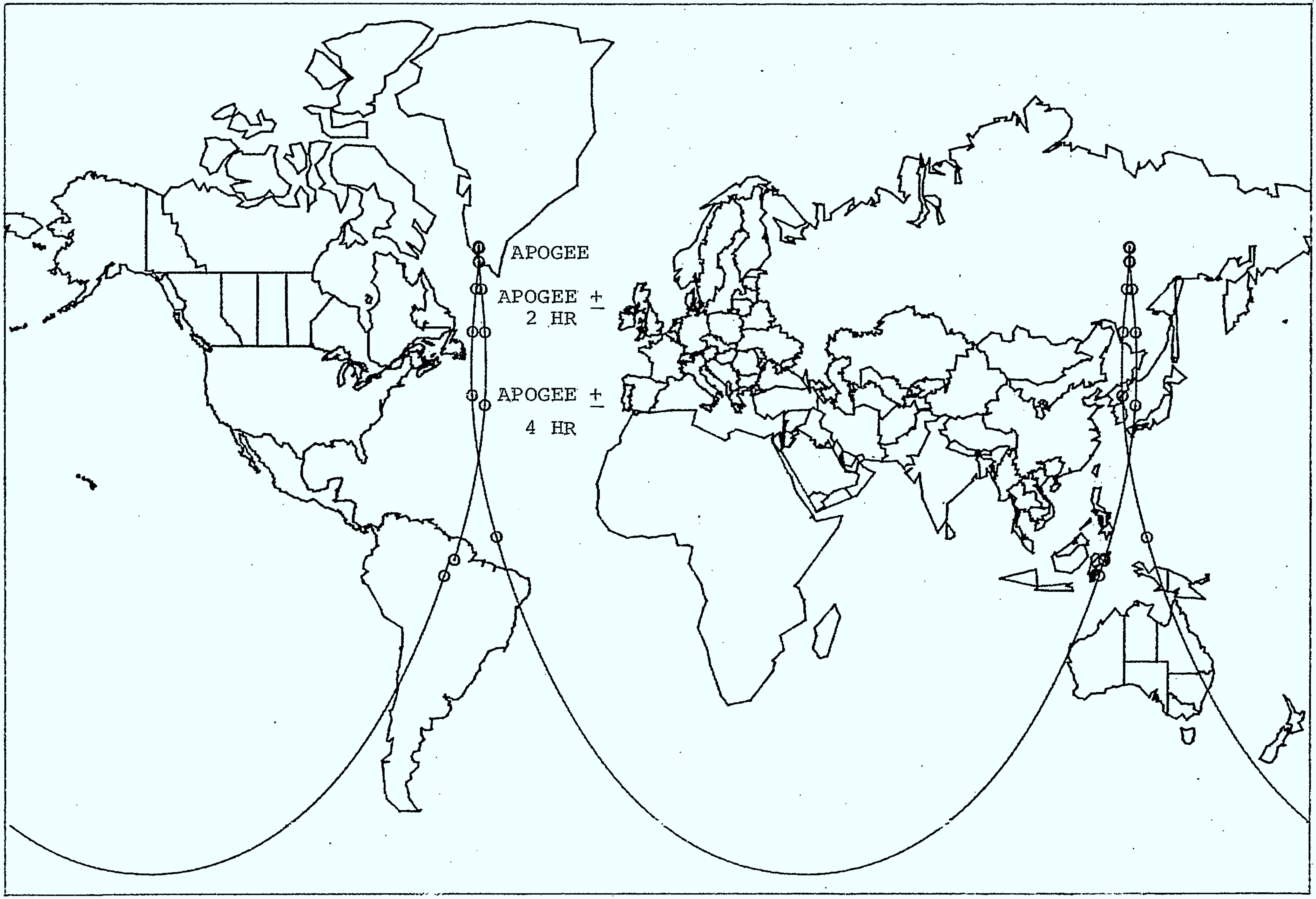
FIGURE 2-3-2

used nominally from two hours prior to apogee to four hours after or from four hours prior to apogee to two hours after, and covers Canada down to a latitude of about 50 -55°N. Figure 2-3-3 shows a typical sub-satellite track for a 12 hour orbit, and Figures 2-3-4 to 2-3-9 show typical views of the earth from the satellite. The horizons at the far side of the earth are 34.5°N at apogee, 38.6°N at apogee plus 2 hours and 55.6°N at apogee plus 4 hours (n.b. Cold Lake is at 54.8°N but can be covered since the orbit may be phased to prevent it ever being in this very worst configuration).

On the basis of coverage, the 12 hour orbit appears to be most suitable for this mission.

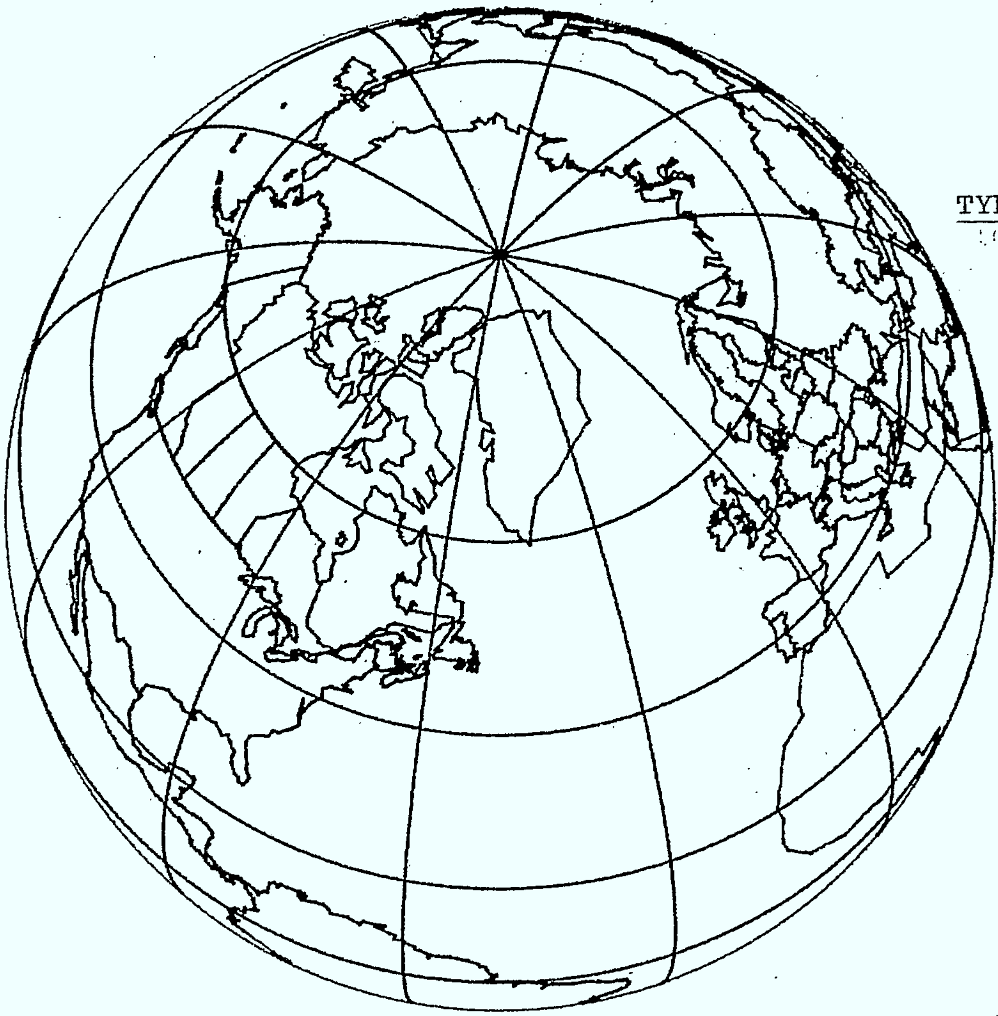
2.3.2 Earth Station Tracking

In order to minimize the motion of the satellite in the sky, its velocity at apogee should result in a rate of change of longitude of nearly zero. Figure 2-3-10 shows this rotation rate versus orbit period. The rate is near zero for a 12 hour orbit making it an ideal choice. Figures 2-3-11 and 2-3-12 show plots of AZ-EL for Cold Lake, Alverta, and the North Pole. In both cases, the satellite motion for either the Apogee ± 2 hours case, or the Apogee ± 4 hours case can be easily handled with a non-tracking earth station antenna at UHF frequencies. Figure 2-3-13 shows the larger variations experienced with an 8 hour orbit. From the point of views of minimizing earth station tracking problems, the 12 hour orbit appears to be the optimum.



TYPICAL SATELLITE TRACK, 12 HOUR ORBIT

FIGURE 2-3-3

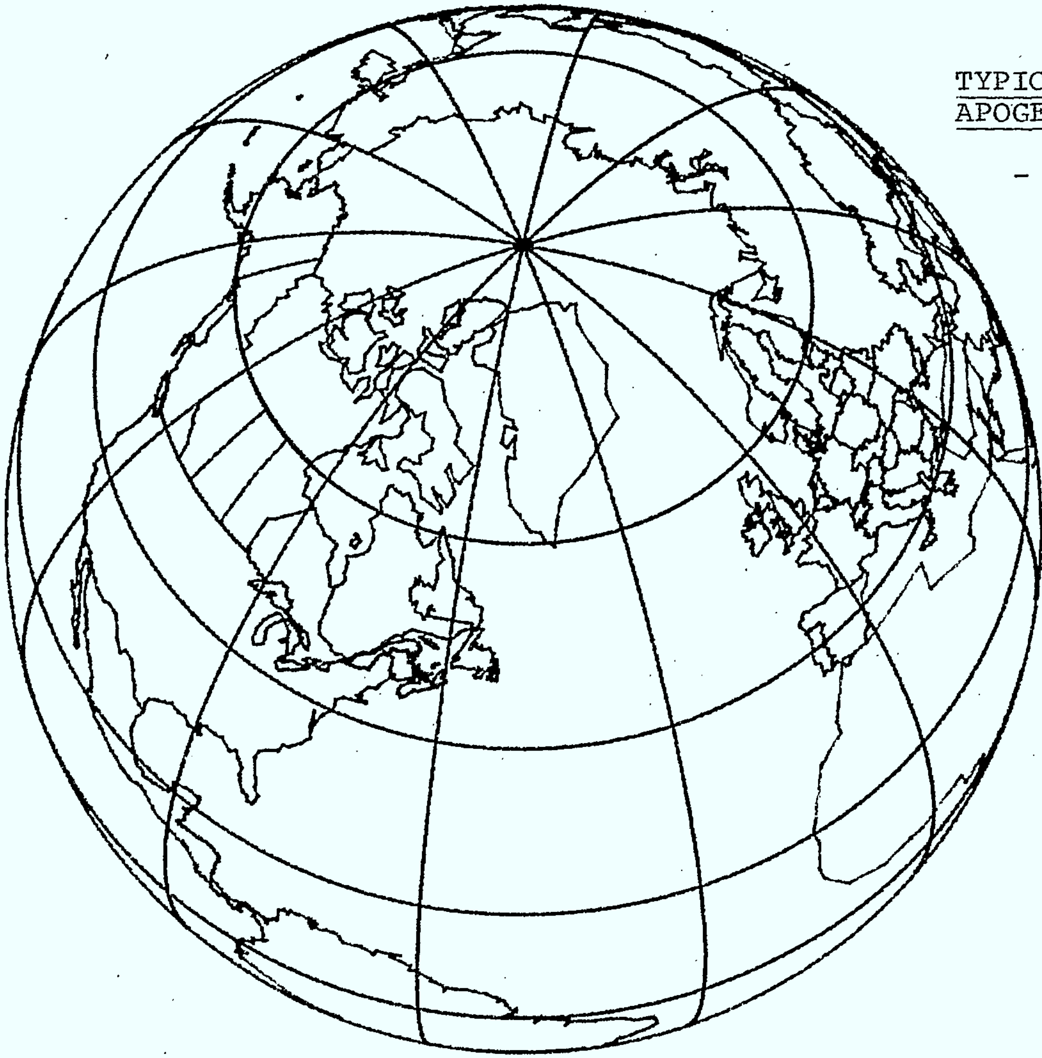


TYPICAL NEAR SIDE VIEW FROM APOGEE

\- 12 HOUR ORBIT -

FIGURE 2-3-4

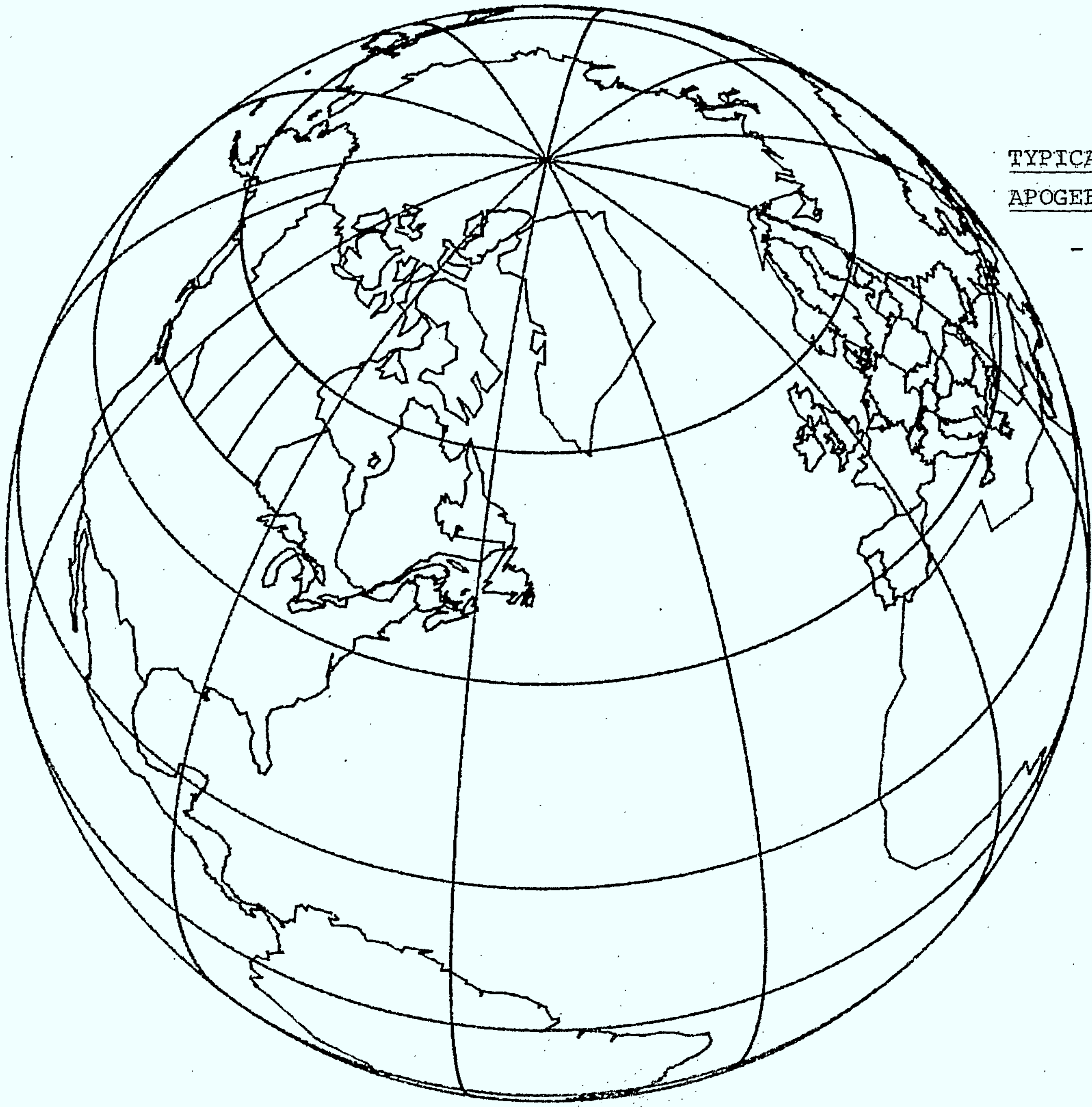
CANADIAN ASTRONAUTICS LIMITED



TYPICAL NEAR SIDE VIEW FROM APOGEE + 2 HOURS

- 12 HOUR ORBIT -

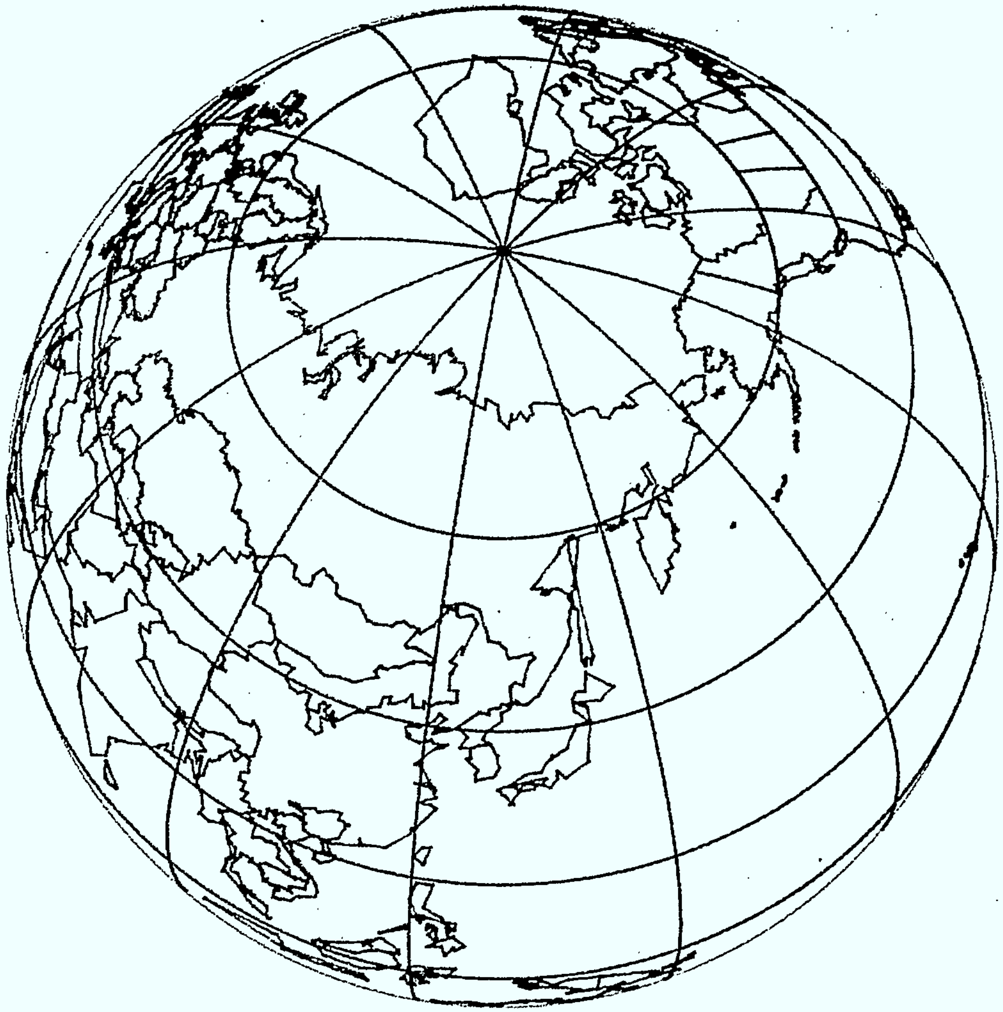
FIGURE 2-3-5



TYPICAL NEAR SIZE VIEW FROM
APOGEE + 4 HOURS

- 12 HOURS -

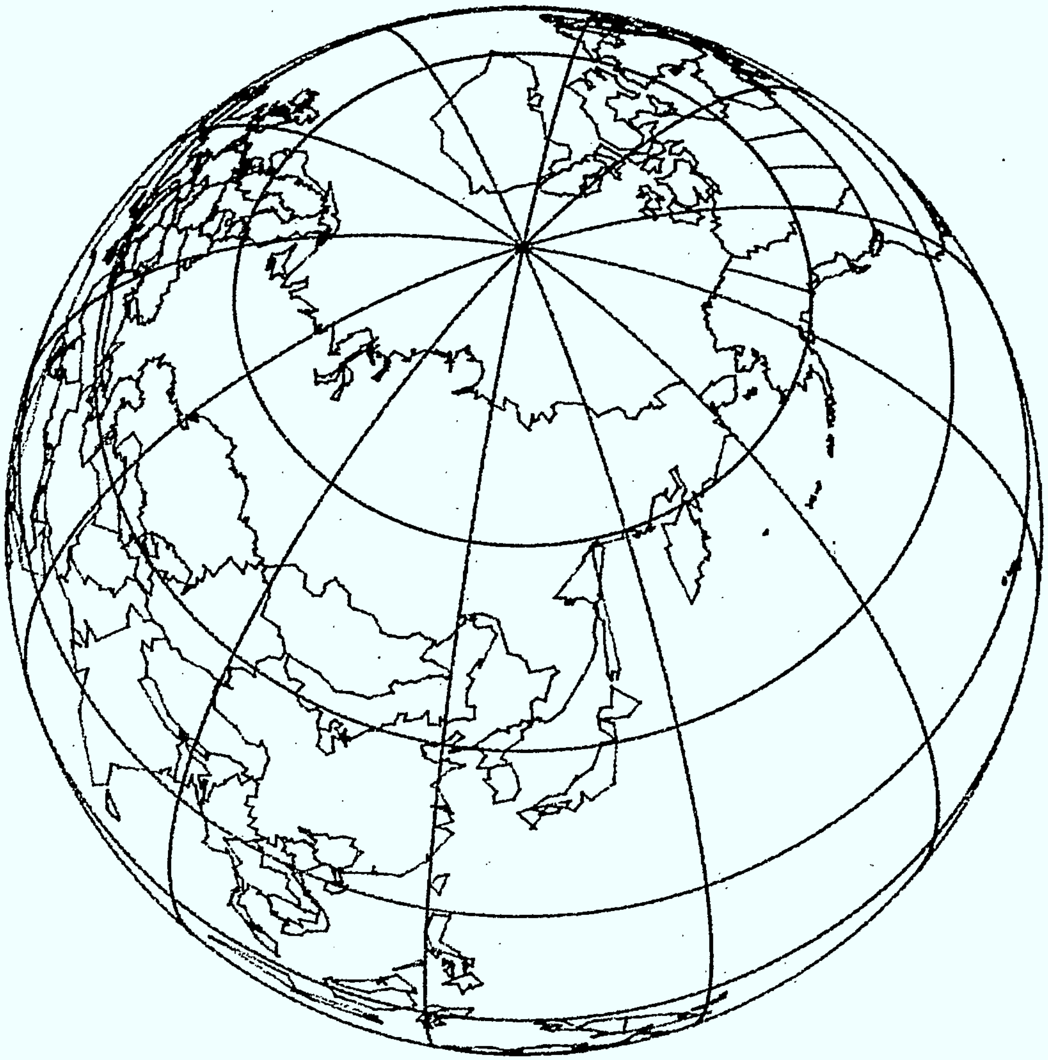
FIGURE 2-3-6



TYPICAL FAR SIDE VIEW
FROM APOGEE

- 12 HOUR ORBIT -

FIGURE 2-3-7



TYPICAL FAR SIDE VIEW
FROM APOGEE + 2 HOURS

- 12 HOUR ORBIT -

FIGURE 2-3-8

FAR SIDE
TYPICAL VIEW

TYPICAL FAR SIDE VIEW FROM
APOGEE + 4 HOURS

FIGURE 2-3-9

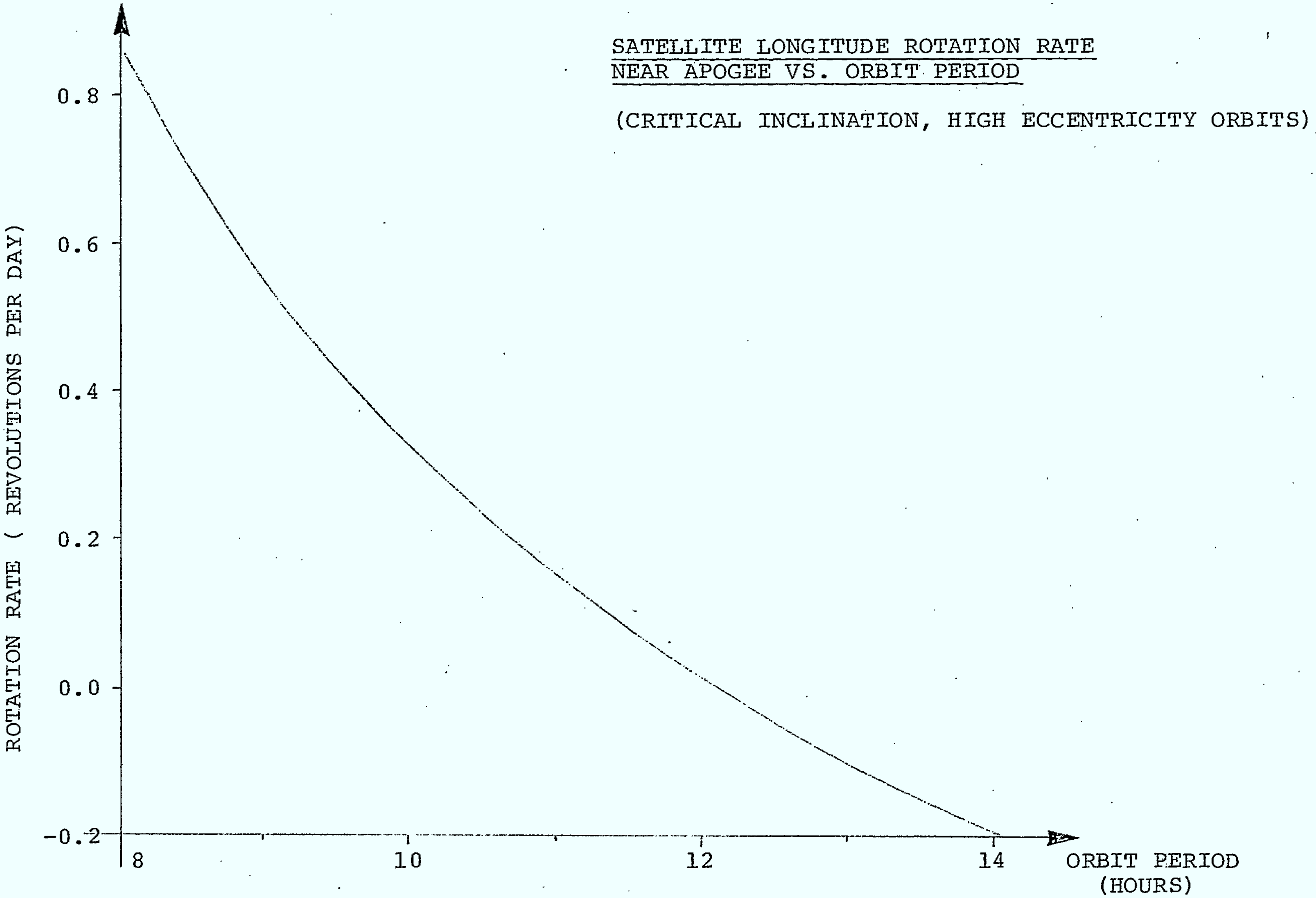
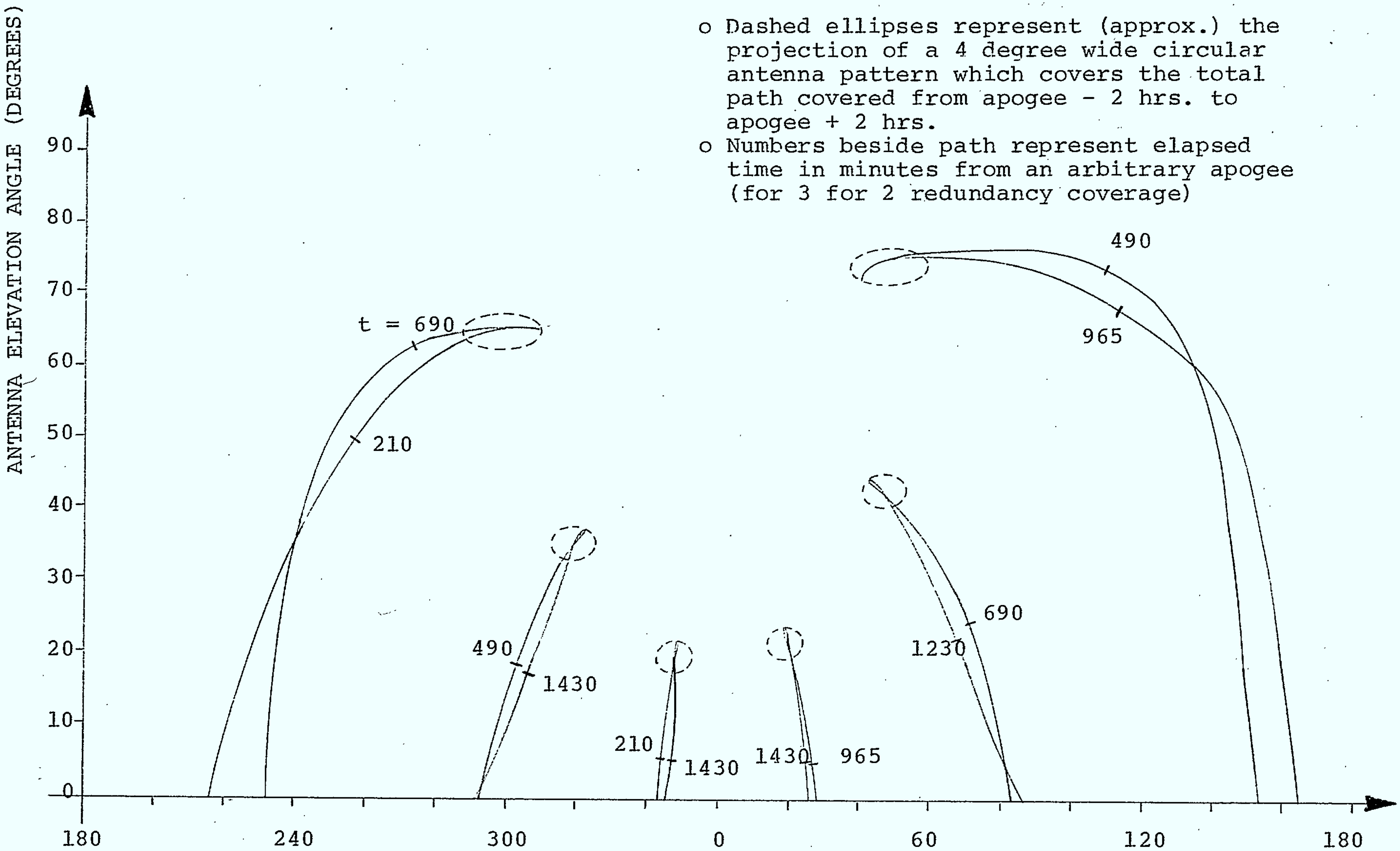

SATELLITE LONGITUDE ROTATION RATE
NEAR APOGEE VS. ORBIT PERIOD
(CRITICAL INCLINATION, HIGH ECCENTRICITY ORBITS)
ROTATION RATE (REVOLUTIONS PER DAY)
0.8
0.6
0.4
0.2
0.0
-0.2
8
10
12
14
ORBIT PERIOD
(HOURS)

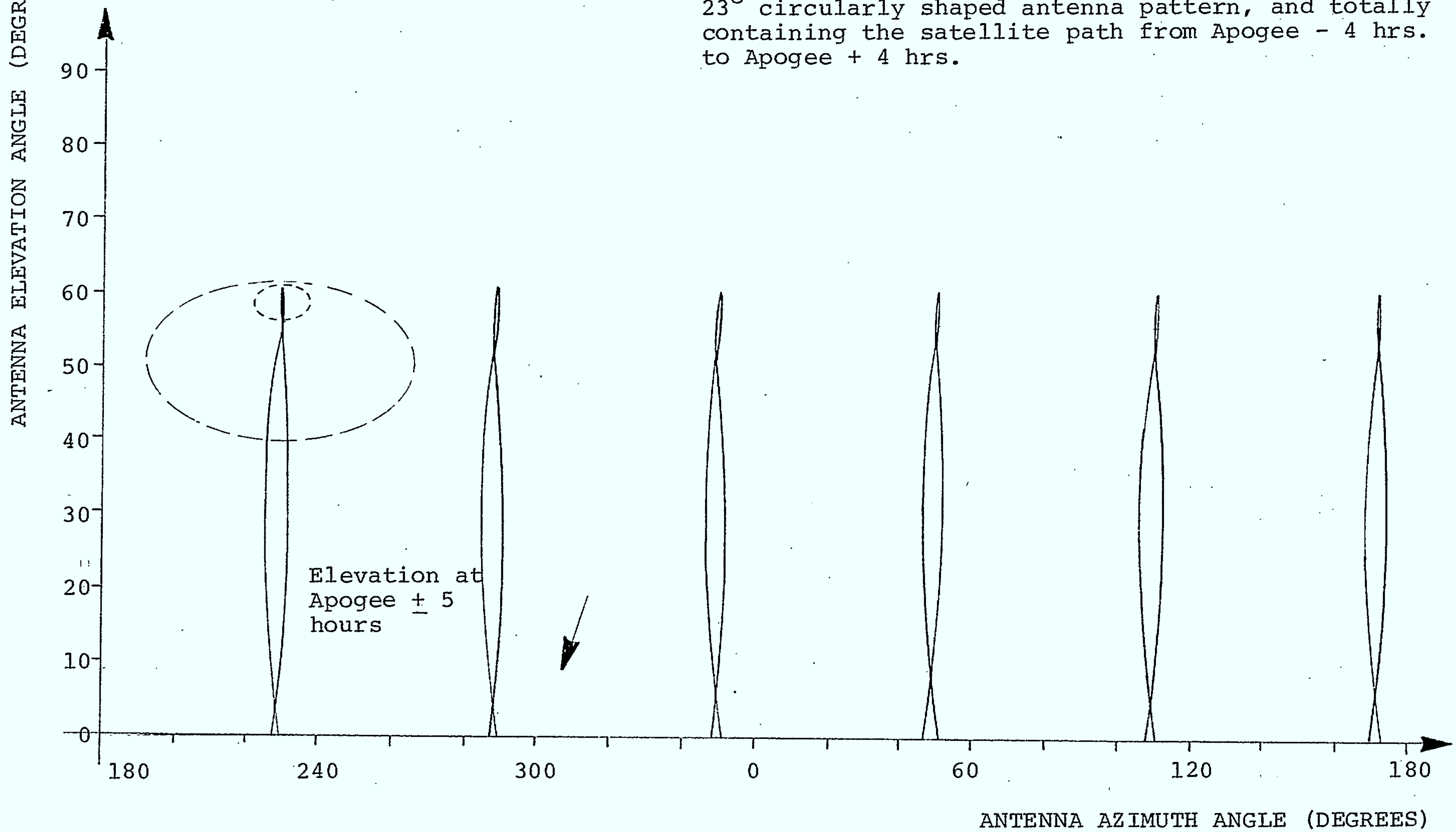
FIGURE 2-3-10



TYPCIAL SATELLITE AZ-EL PATH (12 HR ELLIPTICAL ORBIT, VIEWED FROM COLD LAKE)

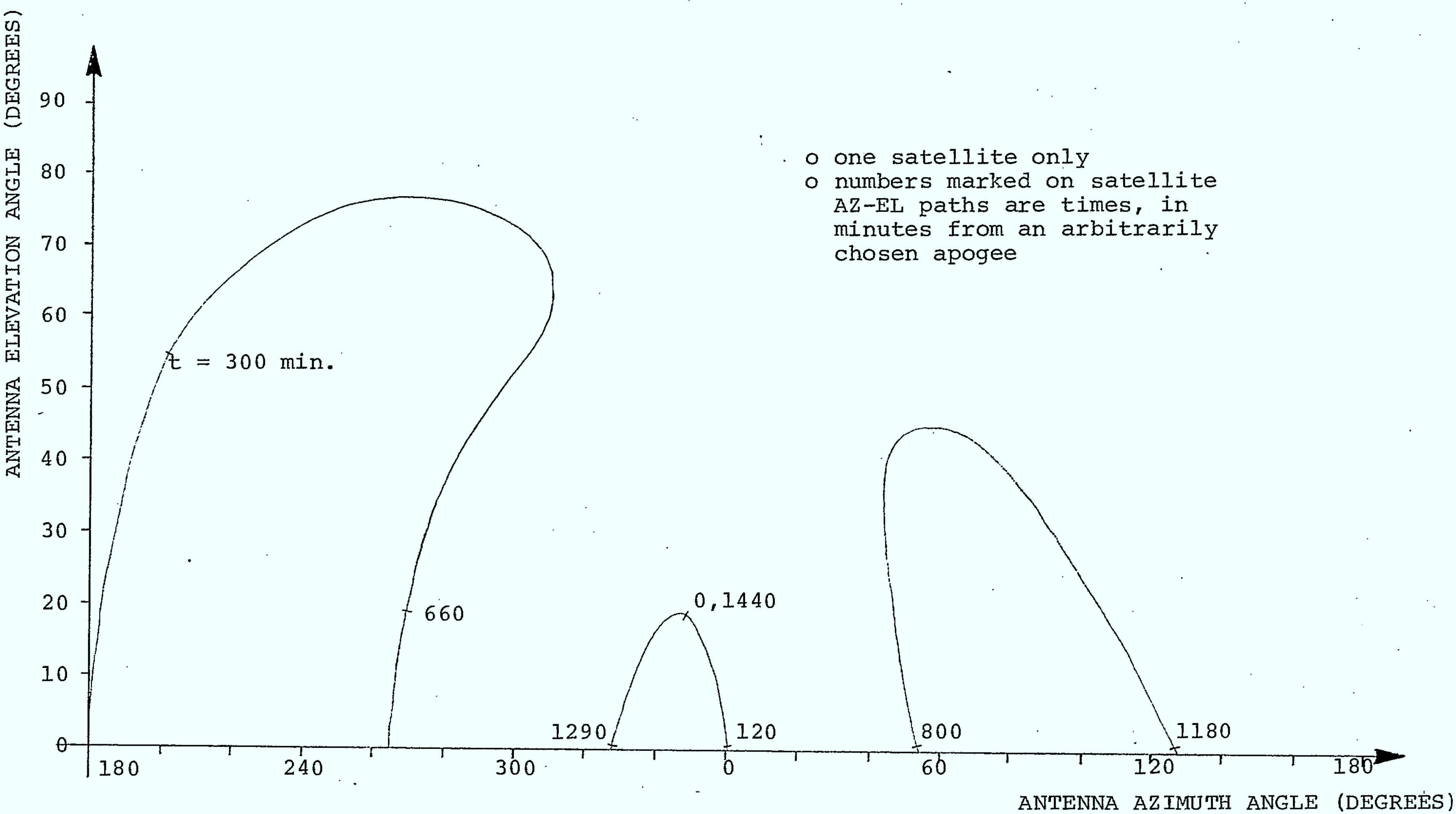
FIGURE 2-3-11

- o Small, dashed ellipse - sky area viewed by a $4\frac{1}{2}^{o}$ circularly shaped antenna pattern, and totally containing the satellite path from Apogee - 2 hrs. to Apogee + 2 hrs.
- o Large, broken line ellipse - sky area viewed by a 23^{o} circularly shaped antenna pattern, and totally containing the satellite path from Apogee - 4 hrs. to Apogee + 4 hrs.



TYPICAL SATELLITE AZ-EL PATH (12 HR. ELLIPTICAL ORBIT, VIEWED FROM NORTH POLE)

FIGURE 2-3-12



TYPICAL SATELLITE AZ-EL PATH (8 HR ELLIPTICAL ORBIT, VIEWED FROM COLD LAKE)

FIGURE 2-3-13

2.3.3 Radiation Degradation Calculation

The Polarsat orbit passes through the region of trapped charged particles known as the Van Allen radiation belts, (Figures 2-1-3 a and b). It is necessary to determine the effect of this flux of energetic particles on the output of the solar array. This has been done by means of a program which calculates the equivalent 1 MEV (million electron volt) fluence of particles incident on the array over a given period of time.

The 1 MEV equivalent electron fluence is a standard bench-mark against which various cells have been tested. The radiation damage caused by electrons and protons over a range of energies has been measured in the laboratory on typical cells, and referenced to the flux of 1 MEV electrons required to produce equivalent damage. Measured values of the particulate radiation at various altitudes and inclinations have been reduced to values of MEV equivalent electron flux. These values are tabulated in Ref. (3) and were used as a data base for this program.

The program sums the radiation contribution from each region through which the satellite will pass in its orbit and outputs the equivalent 1 MEV electron fluence per year for a given density of shielding.

A knowledge of the equivalent radiation fluence, the solar cell properties and the spacecraft design life permits one to calculate the array power output degradation as a function of time. This has been done for a range of perigee altitudes and orbit periods for an inclination of 63.4° The results of these calculations are shown in Figures 2-3-14 and 2-3-15.

The degradation of power output for various cell thickness is then determined from the curves of Figure 2-3-16.

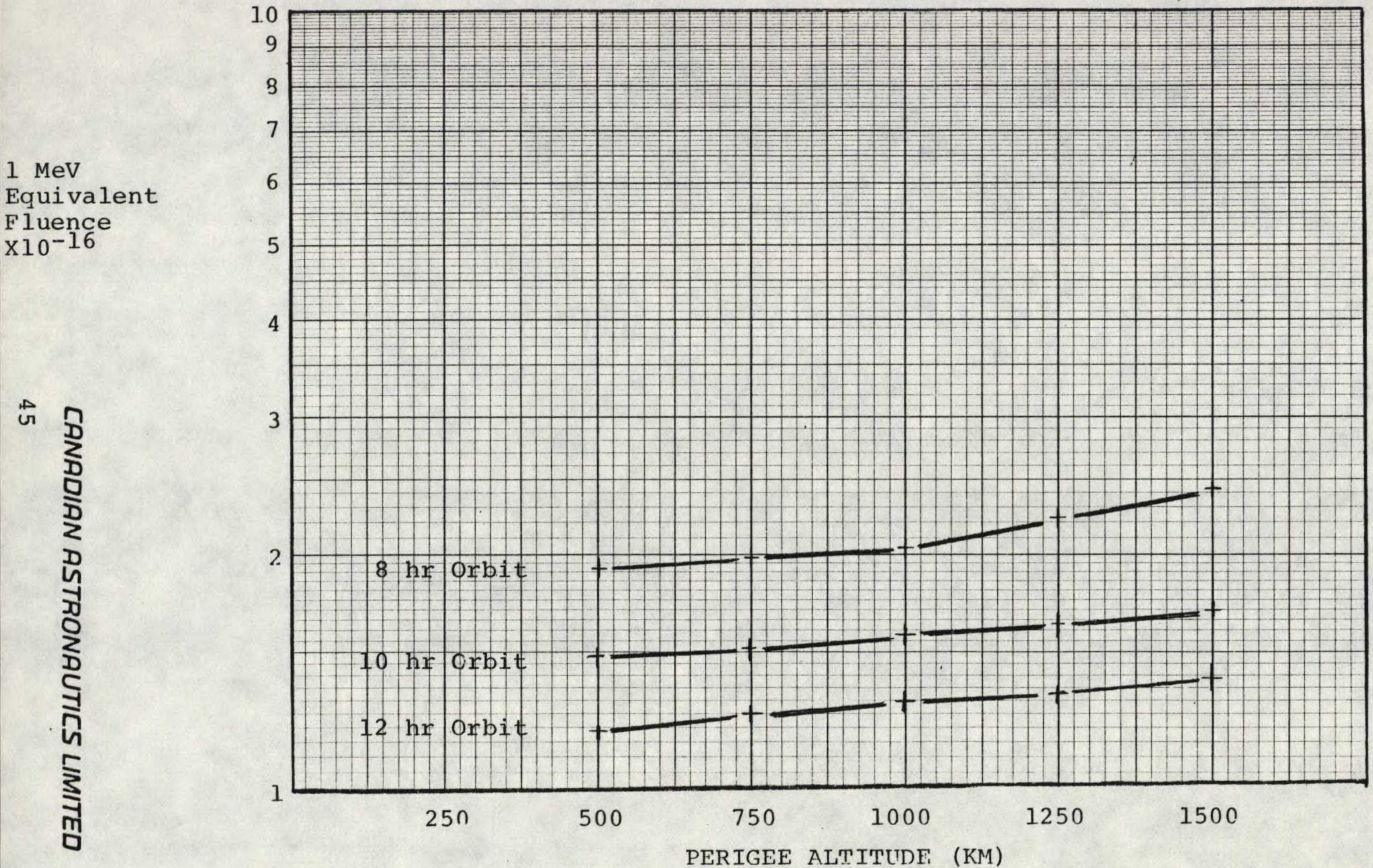


FIGURE 2-3-14

1 MeV Equivalent Fluence vs Perigee Altitude for a 6 mil coverglass, 5 year life

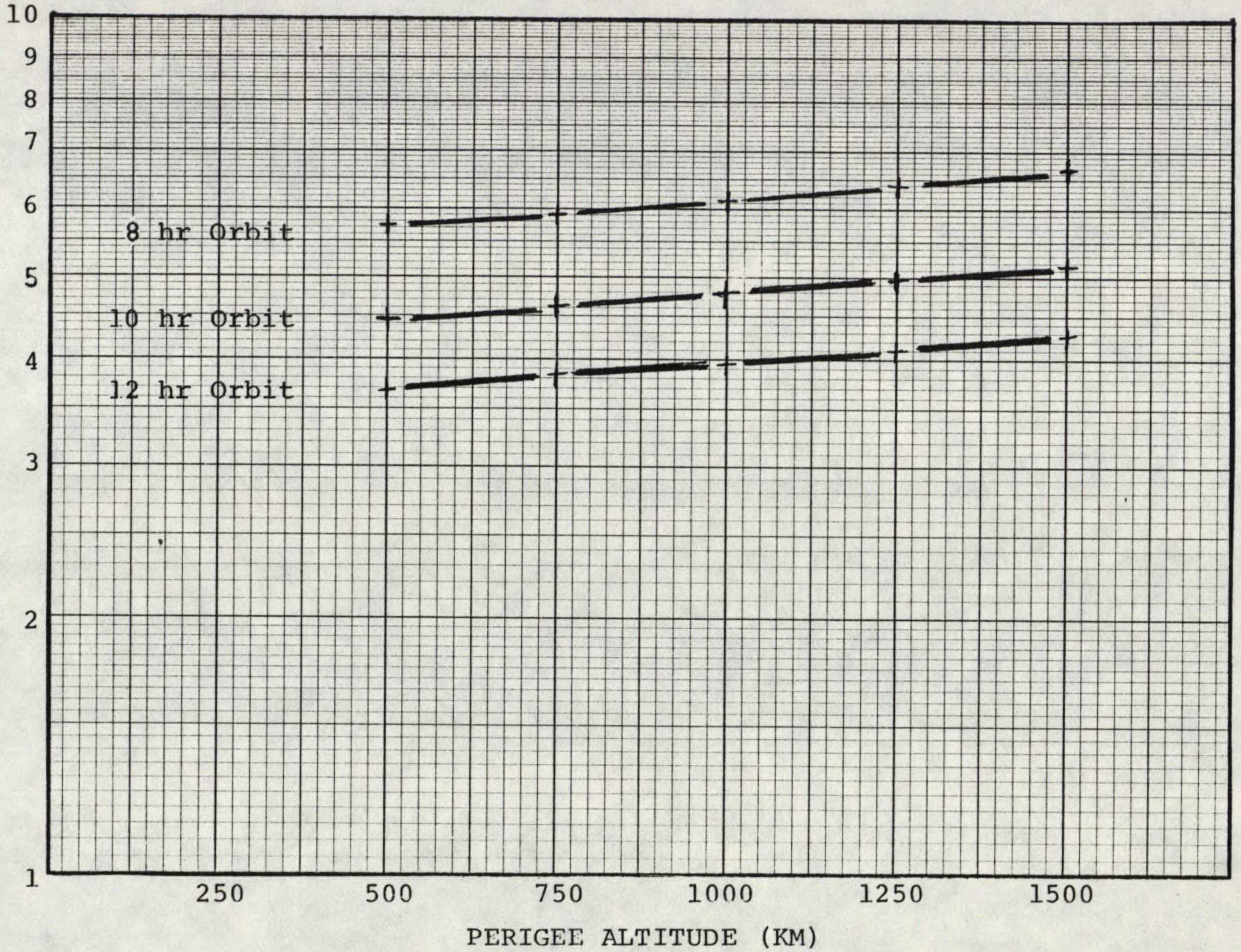
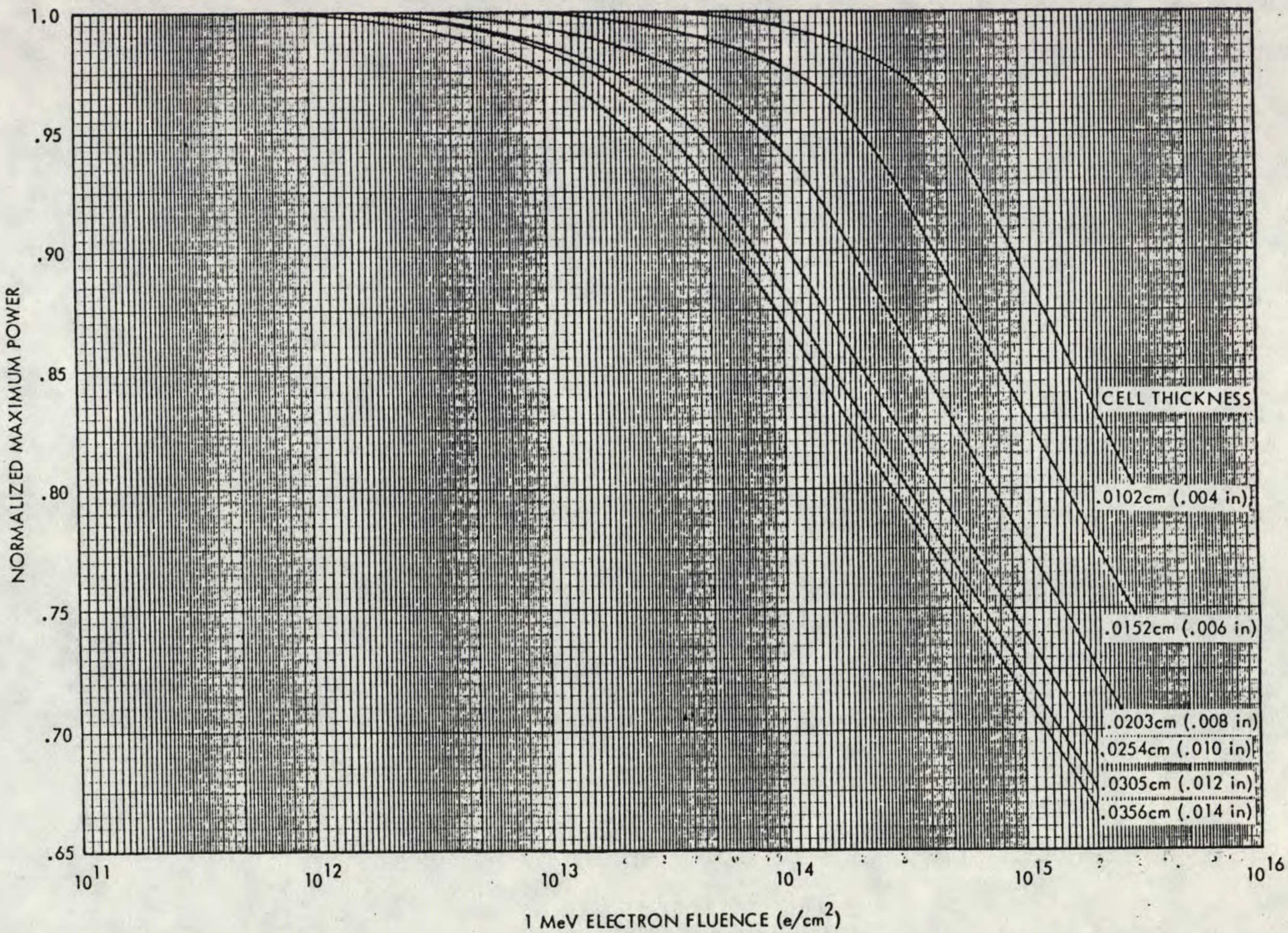


FIGURE 2-3-15

1 MeV Equivalent Fluence vs Perigee Altitude for a 12 mil coverglass, 5 year life



Normalized Maximum Power vs. 1-MeV Electron Fluence for 7-13 ohm-cm N/P Silicon Cells. At 135 mW/cm^2 AM0 Illumination Intensity, 28°C. (Ref. 3)

FIGURE 2-3-16

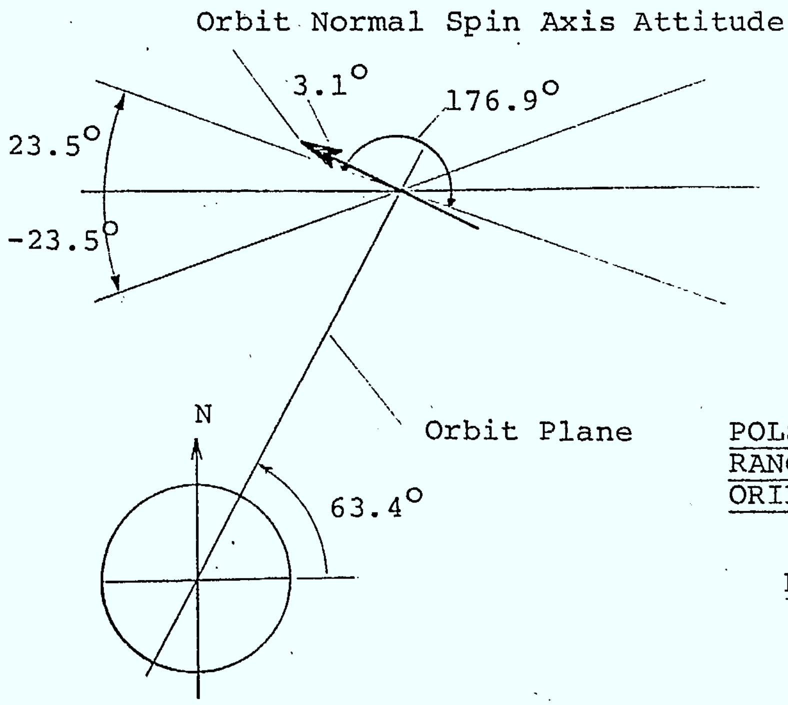
2.3.4 Satellite Eclipsing

In order to avoid the requirement for a large battery system to operate the payload, the 12 hour orbit should be free of eclipsing for at least 4 hours before and after apogee (for the one satellite failed condition). The worst case eclipse condition will occur near winter solstice when the sun is in the southern hemisphere and casting an earth shadow north of the equator. For the 12 hour orbit, under the worst case conditions, the satellite will be free of eclipsing for 4 hours 32 minutes after apogee which allows adequate protection even for the longer coverage passes. There is also, as with geo-stationary orbits, the possibility of long duration lunar eclipses. These events are rare however, and with three satellites operating, this lunar eclipses can be avoided.

The 8 hour orbit produces potential eclipsing problems when one satellite has failed. In this case, eclipse-free satellite operation is required for at least 160 minutes before or after apogee, but eclipses can actually occur within 150 minutes of apogee.

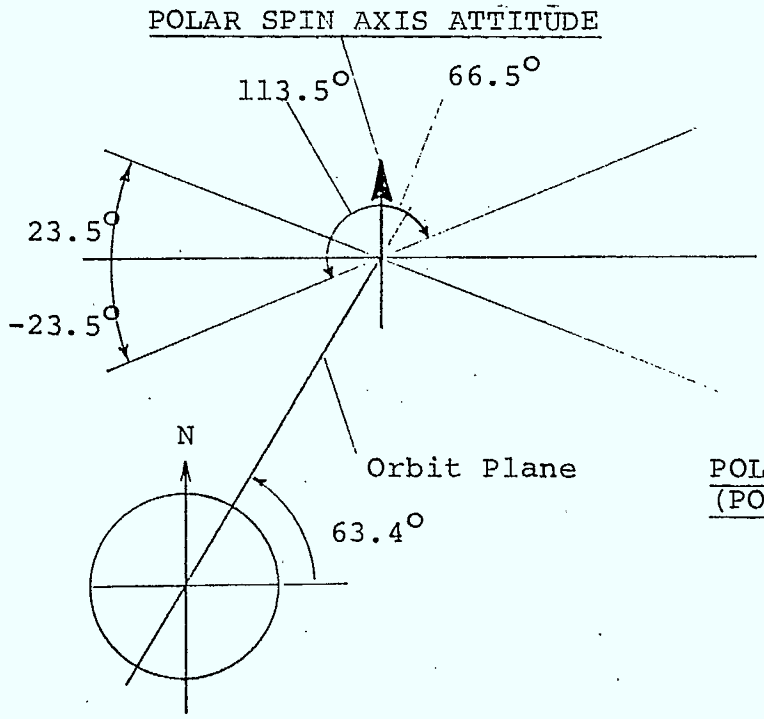
2.3.5 Attitude Control

The baseline stabilization method chosen for this mission is dual-spin. From the point of view of space-craft antenna pointing, the ideal solution is to orient the spin axis perpendicular to the orbit plane. The antenna despin control mechanism controls the azimuthal pointing variation experienced as the satellite moves around the earth, and the variation in elevation of the Canadian coverage zone is small. However, due to the large orbit inclination and the seasonal variation in solar declination angle, the spacecraft solar panel design must allow for sun angles of 3.1 to 176.9° on the space-craft (see Figure 2-3-17).



POLSARSAT SUN ANGLE RANGE (ORBIT NORMAL ORIENTATION)

FIGURE 2-3-17



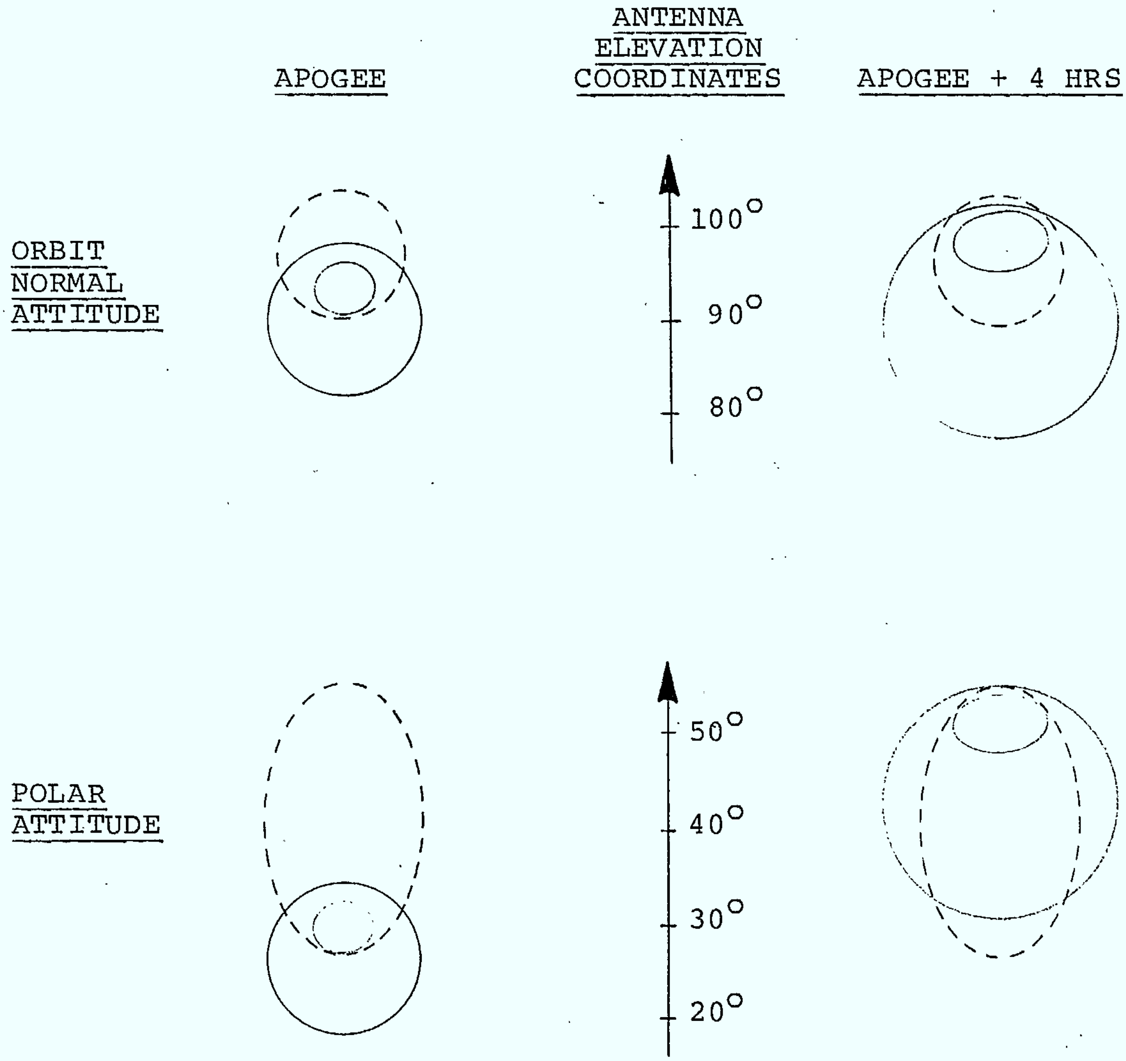
POLARSAT SUN ANGLE RANGE (POLAR ORIENTATION)

FIGURE 2-3-18

To eliminate this problem, the spin axis may be oriented perpendicular to the ecliptic plane which results in sun angles of 90° at all times. This, however, causes large variations in the spacecraft antenna elevation angle to the coverage zone. A compromise design is to orient the spin axis parallel to the earth-polar axis which results in sun angles between 66.5° and 113.5° - the same as for a geostationary satellite (see Figure 2-3-18). This altitude results in a larger range of beam elevation angle requirements than for the orbit normal case, but the situation can be accommodated by the spacecraft antenna. Figure 2-3-19 shows for both cases the projected sphere of the earth, the coverage area (above 70°N latitude) and a typical elliptical antenna pattern required for coverage (the 12 hour orbit is assumed for this analysis). The mean antenna elevation angle (angle between spin axis and antenna bore sight) is 41.5° and the total north-south range required is 27.6°. The total east-west range is 9.5°.

As well as easing the sun angle constraints for geometrical placement of solar cells, the polar spin axis orientation may also ease the thermal design of the spacecraft by allowing dissipation of heat out the ends of the cylinder. Attitude control maneuvers are also less frequent and of smaller magnitude than for orbit normal attitude. This is because there is now no requirement to follow the orbit plane which is precessing about the earth polar axis at about 51° per year.

The antenna despin system may be controlled by inputs from earth sensors, the boresight being controlled to point in azimuth towards the centre of an earth chord. Since the earth appears to move in elevation angle, the sensors must be aligned in elevation so as to cover the earth over the entire range. As the satellite approaches perigee, after terminating communications operations, the sensor will leave the earth and the despin system goes into a constant speed mode. The attitude determination



- Antenna Elevation is an angle in the plane of the spin axis measured from the spin axis-antenna end
- 12 hr. critically inclined, high eccentricity orbit
- The view drawn represent projections on a visual sphere centered on the spacecraft
- Solid circles - Earth View
- Shaded ellipses - Coverage zone above 70°N latitude
- Dashed ellipses - Projections of a typical antenna beam required for coverage and circles

ANTENNA BEAM POINTING

FIGURE 2-3-19

system can make use of the same earth sensors used for the despin system.

2.3.6 Other Factors

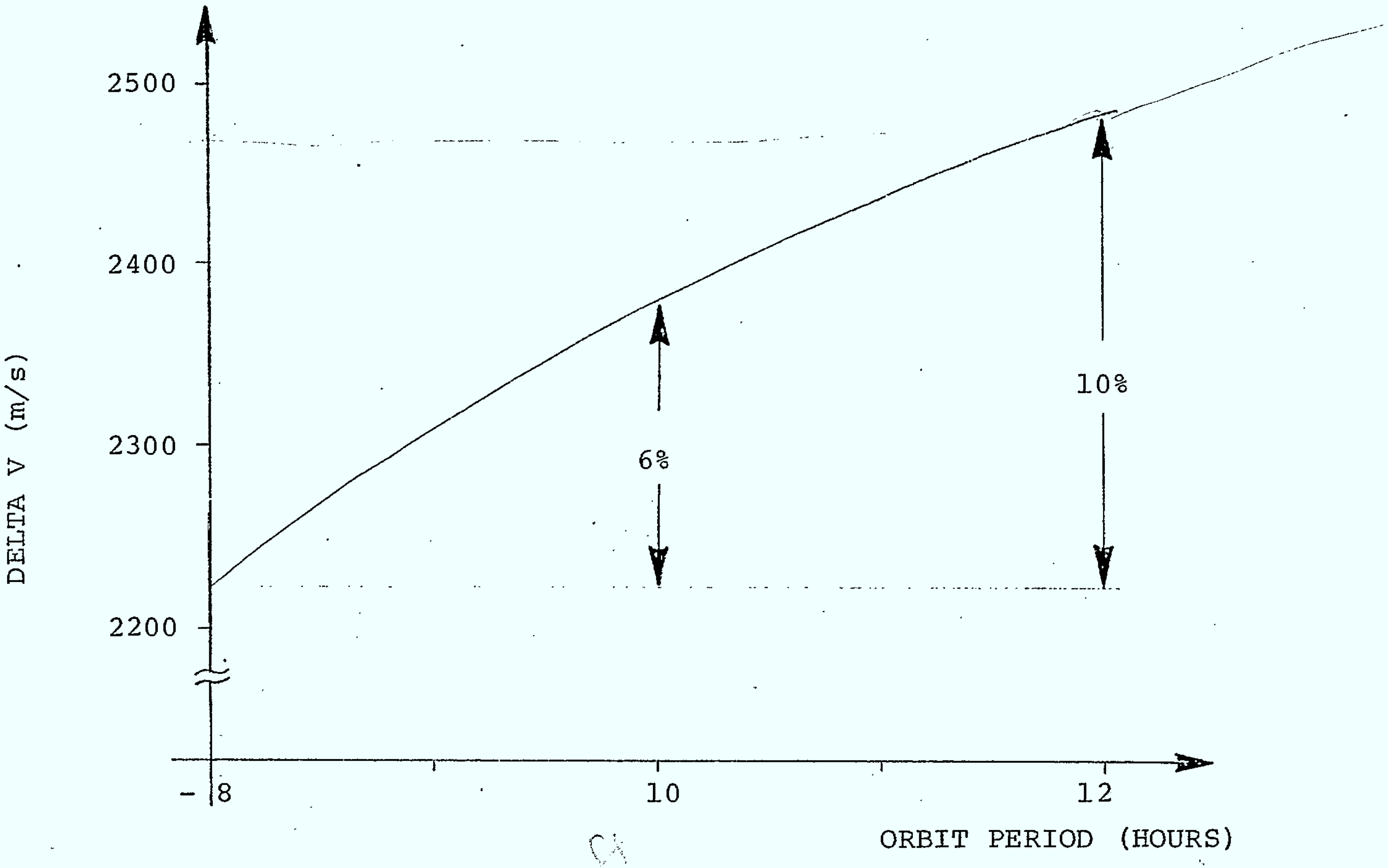
For the preferred 12 hour orbit, the maximum doppler velocity experienced is about 2.3 km/sec (for the 8 hour orbit, the maximum is about 1.9 km/sec). These maximums occur at the beginning or end of the part of the orbit used during the one satellite failed condition.

For the 12 hour orbit, the maximum and minimum, satellite to earth station, one-way ranges are 42,000 km and 24,000 km for stations above 70^{o}N latitude with one satellite failed. For the 8 hour orbit, the maximum range is about 32,300 km and the minimum is about 14,000 km.

Utilizing the polar spin axis orientation discussed in section 2.3.5, solar cells mounted on a coaxially cylindrical spacecraft body will suffer a maximum power loss of 8% due to the 23.5^{o} maximum solar incidence angle.

Launch from the Shuttle orbit into an elliptical orbit can be by solid or liquid motor built into the spacecraft. The velocity change (ΔV) requirements for the motor are shown in Figure 2-3-20. The extra ΔV for boosting the satellite into a 12 hour orbit is only 10% greater than for boosting it into an 8 hour orbit, and the motor required then is approximately 16% heavier.

An auxilliary propulsion system is required to correct for errors in the boost and to raise the perigee or to maintain it periodically in order to counteract the reduction of altitude due to solar and lunar gravitational forces. The propulsion system is also used for periodic control of spacecraft attitude and for orbit phasing. A total ΔV budget of 100 m/s should be adequate for a five year life.



DELTA V REQUIREMENTS FOR PERIGEE KICK MOTOR

FIGURE 2-3-20

2.4 Mission Sequence

The Polarsat mission sequence begins with a launch by the Space Shuttle from Vandenberg Air Force Base, California, into a 300 km altitude circular parking orbit inclined by 63.4 degrees. From the Kennedy Space Center, Florida, the maximum inclination of a parking orbit is only 57 degrees due to range safety constraints.

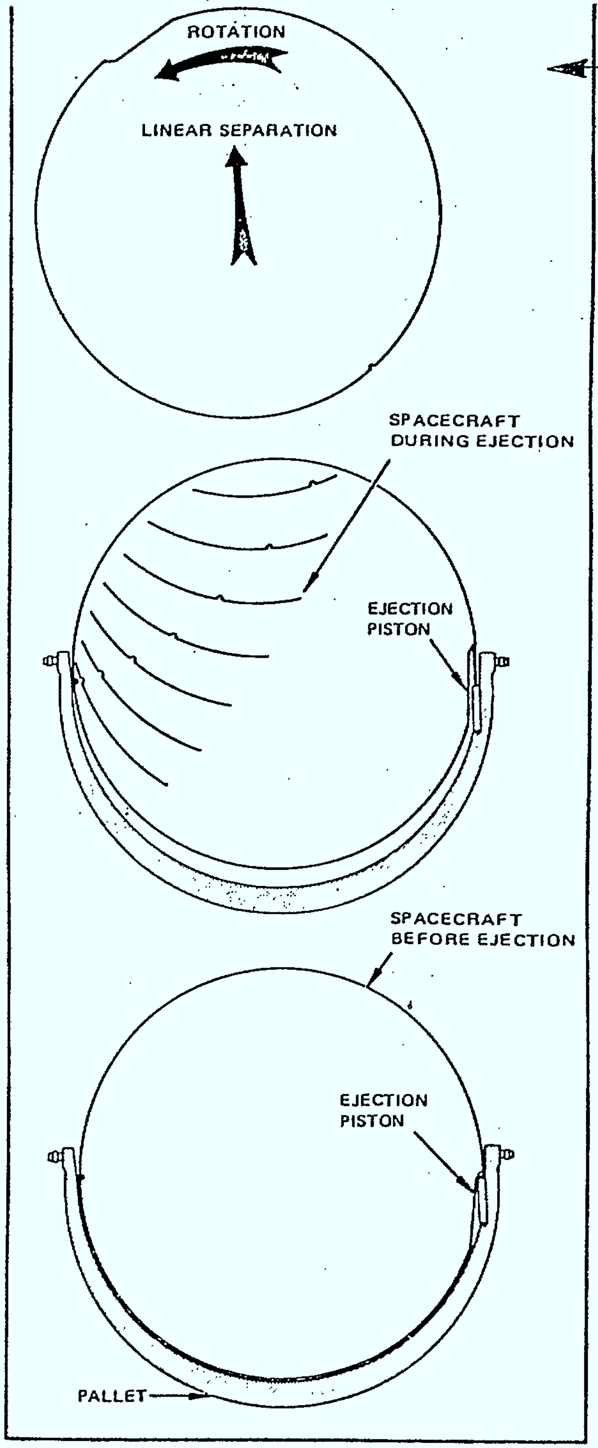
In the parking orbit, Polarsat can be deployed from the Shuttle with the Shuttle Remote Manipulator system (SRMS). The SRMS which is being developed in Canada, provides the capability for remotely controlled removal of payloads from the cargo bay and for positioning them in a particular orientation before release.

In a typical deployment sequence, the SRMS attaches itself to a point on the despun section of the spacecraft, the attach/release devices are actuated and the spacecraft is then positioned about 10 meters from the Shuttle. After the SRMS joints are locked in a stable position, the spacecraft despin motor is commanded to provide at least 3 RPM of rotational motion to the spinning section of the spacecraft. This takes about 35 seconds and one revolution of the spacecraft spinning section. The SRMS then releases the spacecraft, retracts itself and the Shuttle executes a small maneuver to initiate positive separation from the spacecraft. After several minutes, the spacecraft fires a pair of thrusters to increase its spin rate to approximately 20 - 30 RPM.

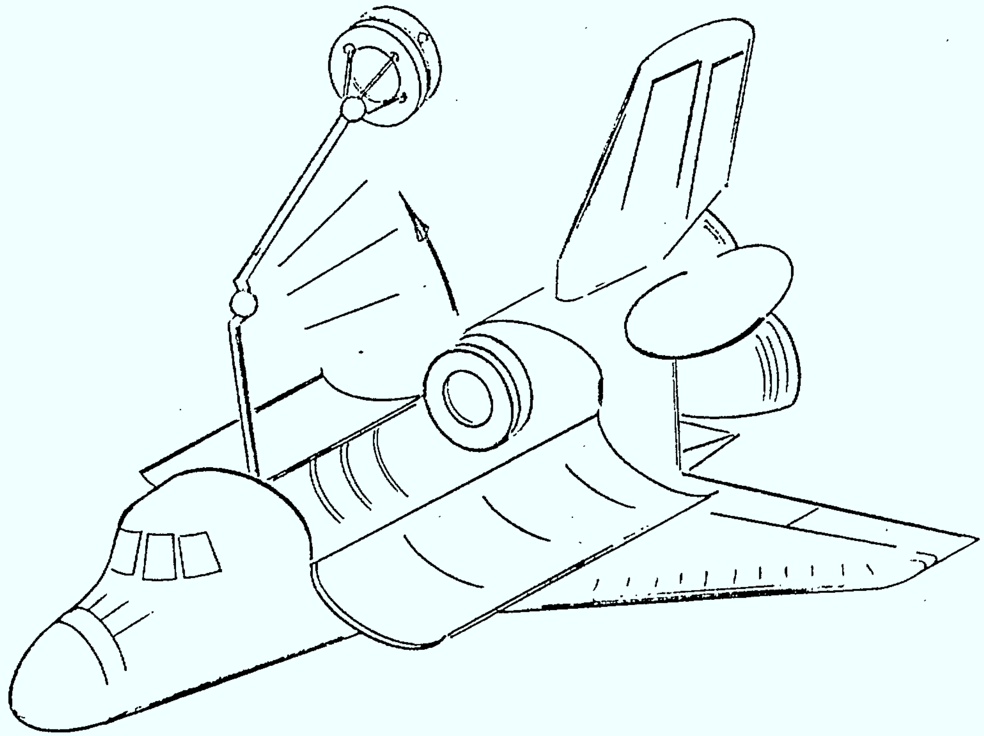
An alternate spacecraft deployment option that may be considered in future studies if vehicle dynamics make the baseline technique unfeasible is the "FRISBEE" technique proposed by Hughes Aircraft. The spacecraft is constrained in the Shuttle cargo bay by latches on each side. When these are released a spring or other such actuator pushes one side of the spacecraft up and out of the bay. The other side is temporarily constrained by a pivot joint so that the spacecraft receives both a translational and rotational impulse in a single, simple motion. Figure 2-4-1 illustrates the baseline deployment maneuver and the optional 'FRISBEE' approach.

The spacecraft attitude at deployment is colinear with the satellite velocity vector at the southernmost point in the orbit (one quarter orbit after North to South equator crossing). The perigee kick motor is fired at this southernmost point, boosting the apogee to an altitude of approximately 39,700 km.

On a subsequent apogee, the perigee may be boosted from 300 km to operating altitude (say 500 to 750 km). If the axial engines of the auxiliary propulsion system are oriented in the opposite direction to the perigee kick motor (PKM) they may be used without requiring any spacecraft attitude reorientation. Alternatively if the axial engines must be oriented in the same direction as the PKM, spacecraft may be reoriented by 90° and the radial engines used, or a 180° reorientation can be performed followed by an axial engine boost maneuver. Following the rough orbit acquisition maneuvers, several corrective maneuvers will likely be required to adjust the orbit period and other orbital parameters. These maneuvers may be performed either before or after reorientation to the final polar attitude. Final orbit and attitude touch up maneuvers will have to be made within a few days after attitude acquisition and periodically thereafter. A block diagram of the mission sequence is shown in Figure 2-4-2.



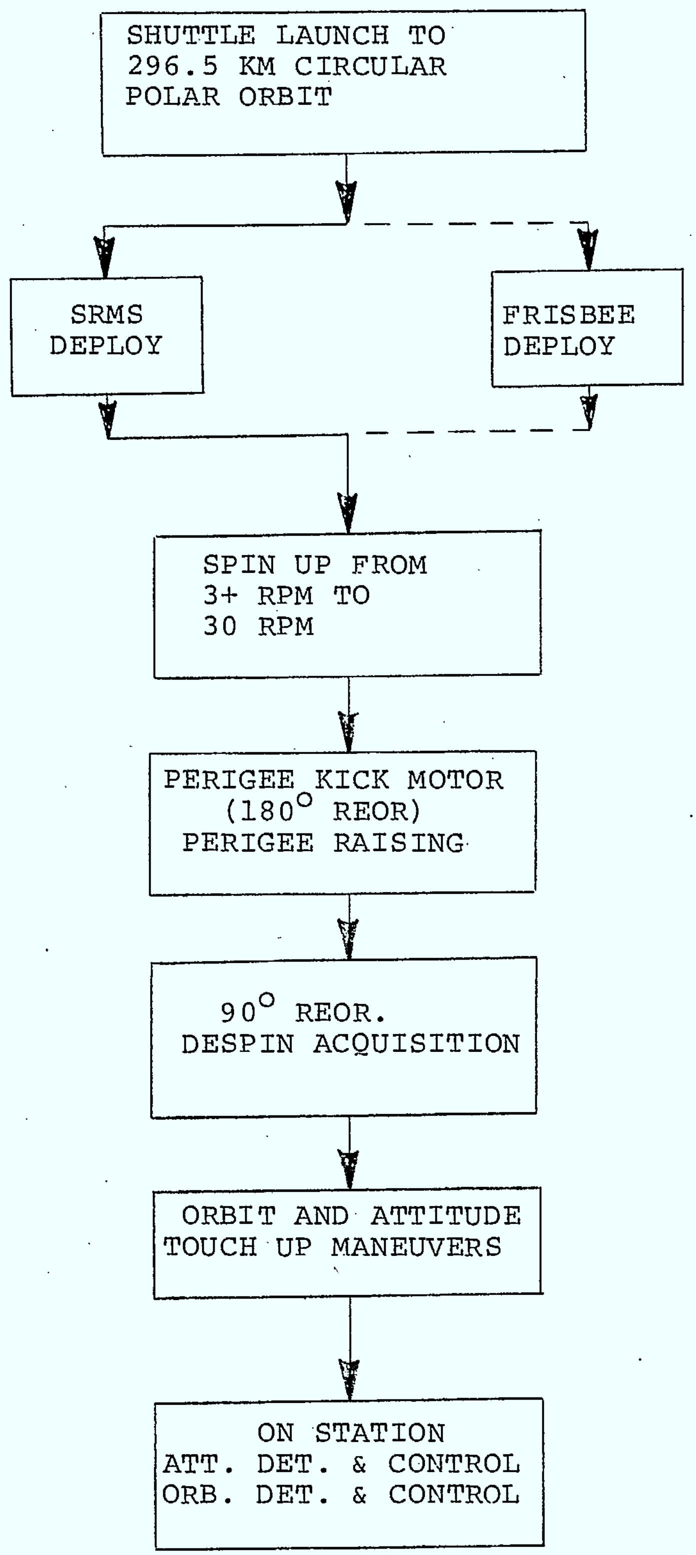
a) HUGHES "FRISBEE" DEPLOYMENT



b) SRMS DEPLOYMENT

SPACECRAFT DEPLOYMENT OPTIONS

FIGURE 2-4-1



MISSION SEQUENCE BLOCK DIAGRAM

FIGURE 2-4-2

2.5 Summary of Mission Analysis

The analysis performed for the POLARSAT mission indicates that the baseline mission should utilize a dual-spin satellite oriented parallel to the earth polar axis, and in a critically inclined, highly elliptical, 12 hour orbit. No attempt has been made to fully optimize the mission, or to examine in detail the performance. Rather, the objective has been to perform a conceptional analysis to establish a good baseline spacecraft/orbit/ attitude configuration for the POLARSAT mission.

3.0 COMMUNICATIONS SYSTEM DESIGN

3.1 General

The Polarsat system provides the communications capabilities for duplex transmission of voice and data between UHF, mobile stations and fixed SHF base stations. In addition, an SHF communications capability for transmission between fixed stations is also provided. Mobile to mobile transmission is achieved using a double satellite hop via a fixed station. Further links to the Polarsat System include a wide-band channel for duplex voice and data transmission using spread spectrum frequency hopping and a channel for the reception of EPIRB distress signals. Spacecraft control, monitoring and orbit determination are accomplished using a telemetry, tracking and command (TT&C) link from a central control station.

The frequency plan for the Polarsat System is presented in section 3.2 and the up and down link budgets for the communications channels are derived in section 3.3. The spacecraft communications system and ground communication system are discussed in sections 3.4 and 3.5 respectively.

Shown in Table 3-2-1 is a summary of the link design for the communications channels of the Polarsat System and the channel capacity. The up and downlink budgets for the channels are given in Appendix A.

TYPE OF CHANNEL	DATA RATE (Kbps)	EIRP/Ch (dBW)	Min. Uplink C/N_o (dB-Hz)	Min.Dnlk C/N_o (dB-Hz)	Min.Overall C/N_o (dB-Hz)	No. of Ch.	CHANNEL FUNCTION
UHF-SHF	16	-1.5	55.9	56.7	53.3	7	Team-Pack UHF TX
	16	-1.5	58.3	56.7	54.4	8	Shipborne UHF TX
	16	-1.5	57.6	56.7	54.1		Airborne UHF TX
	16	-1.5	55.9	56.7	53.3		Small Airborne UHF TX
	-	-9.1	32.1	49.1	32.0	10	EPIRB Distress TX
SHF-UHF	16	23.3	56.8	55.2	52.9	8	Team-Pack UHF RX
	16	23.3	56.8	49.8	49.0		Shipborne UHF RX
	16	23.3	56.8	53.4	51.8		Airborne UHF RX
	16	29.2	56.8	49.8	49.0		Small Airborne UHF RX
SHF-SHF	2000	15.0	79.0	73.2	72.2	1	2 Mbps Data Tx-Rx
COMMAND SHF	.256		36.1	-	-	1	Command Tx
TELEMETRY SHF	.256	-22.1		36.1	-	1	Telemetry Rx

RF LINK DESIGN SUMMARY

TABLE 3-2-1

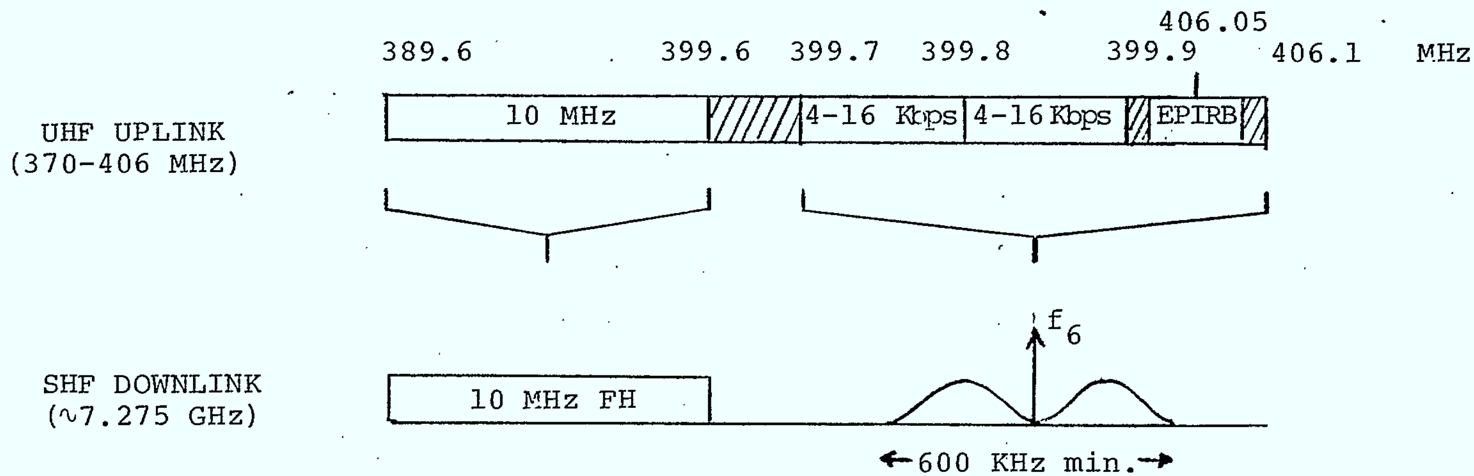
3.2 Frequency Plan

3.2.1 General

The communications links consists of UHF and SHF up and downlinks. The UHF links connect the spacecraft to the remote mobile stations which include aircraft, ships and transportable team packs. The SHF links connect the spacecraft to fixed SHF stations and the central control station. These links are consistent with the MUSAT earth segment as given in Reference 5 and 6. End-to-end transmissions are carried on UHF-SHF, SHF-UHF and SHF-SHF communications paths.

3.2.2 UHF-SHF Path

The frequency plan for the UHF-SHF path has the following signals: EPIRB distress signal, 8 frequency multiplexed 16 Kbps channels, and a 10 MHz wideband frequency hopped channel as shown in Figure 3-2-1. Bandwidth for eight 16 Kbps channels has been provided extending below 399.9 MHz in accordance with the ITU Allocation Tables, for this service. The 10 MHz frequency hopped channel is retransmitted by a separate transponder as considered in Section 3.4. The 16 Kbps channels carry voice and data on UHF frequencies spaced 25 KHz apart which are distinct from the UHF frequencies assigned to the MUSAT System. The UHF uplink spectrum is received by the spacecraft as individual uplink carriers in the band from 389.6 to 406.1 MHz. This is necessitated by the mode of operation of the MUSAT UHF ground segment. The 16 Kbps channels and EPIRB channel are FM modulated on a single 7.275 GHz carrier to enable removal of the doppler shift on the received SHF downlink transmission.



FREQUENCY PLAN FOR UHF-SHF PATH

FIGURE 3-2-1

The Polarsat orbit selected from the Mission Analysis Studies in section 2.0 results in a maximum satellite radial velocity of 2.3 Km/sec at the orbit positions corresponding to apogee plus or minus four hours. At the downlink frequency of 7.275 GHz, the doppler shift is $\pm$55.8 KHz. The doppler shift on the maximum UHF uplink frequency of 406 MHz is $\pm$3.1 KHz.

3.2.3 SHF-UHF Path

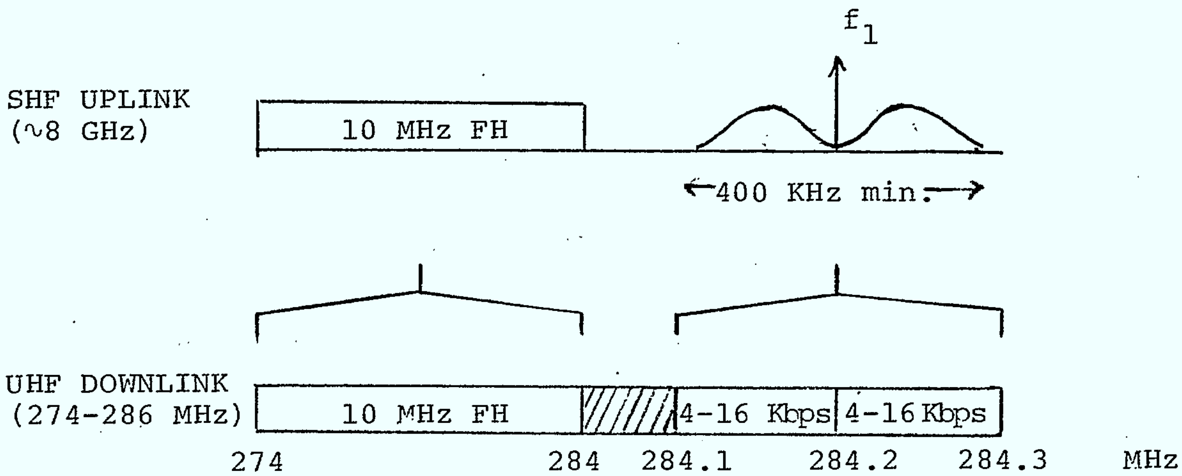
The frequency plan for the SHF-UHF path is shown in Figure 3-2-2. The SHF-UHF path serves as the return path for the UHF-SHF path except for the simplex EPIRB distress signal component of the UHF-SHF spectrum. Due to a doppler shift of $\pm$61 KHz on the 8 GHz SHF uplink, the eight 16 Kbps channels and command channel are FM modulated on the 8 GHz SHF uplink carrier. In the spacecraft, this SHF uplink signal is received and FM modulated using a tracking demodulator which removes the uplink doppler shift. The 10 MHz frequency hopped channel is retransmitted by a separate transponder. On the UHF downlink portion of the SHF-UHF path, the signals are transmitted on individual UHF carriers in the band from 274 to 286 MHz.

3.2.4 SHF-SHF Path

The SHF-SHF path carries duplex transmission of a 3 MHz wideband channel for high speed 2 Mbps data. The SHF-SHF frequency plan is shown in Figure 3-2-3.

3.2.5 TT&C Paths

The TT&C transmissions consist of the telemetry, command and ranging signals. The telemetry signal is transmitted on the SHF downlink of the UHF-SHF path

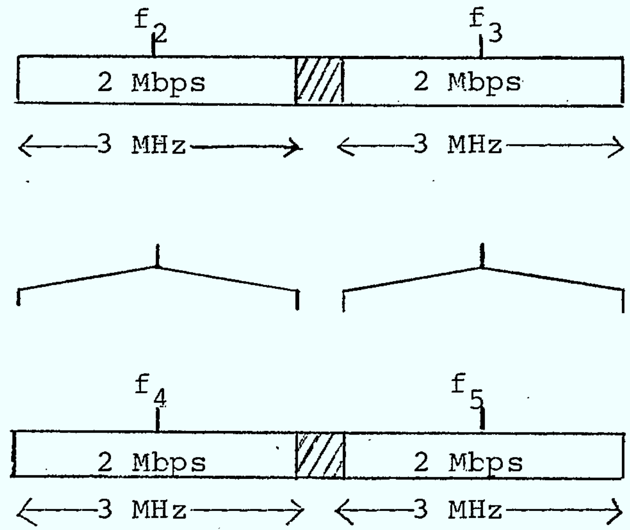


FREQUENCY PLAN FOR THE SHF-UHF PATH

FIGURE 3-2-2

SHF UPLINK
(∿8 GHz)

SHF DOWNLINK
(∿7.275



FREQUENCY PLAN FOR THE SHF-SHF PATH

FIGURE 3-2-3

and the command signal is transmitted on the SHF uplink of the SHF-UHF path. Both the telemetry and command signals are frequency multiplexed with the other signal components of the SHF-UHF and UHF-SHF paths. The composite frequency multiplexed signal frequency modulates the up and downlink SHF frequencies as further discussed in section 3.4.7. For a 256 bps telemetry and command system the bandwidth required is about 500 Hz for a non-coherent FSK subcarrier modulation scheme.

The ranging signal consisting of a narrowband tone is provided for the purposes of satellite tracking and orbit determination. It is transmitted together with the telemetry and command subcarriers as described above.

3.3 Link Analysis

3.3.1 General

The RF link budgets for the Polarsat system are given in Tables A-1 to A-19 of Appendix A. The link design involves the selection of the S/C UHF and SHF effective isotropic radiated power (eirp), S/C UHF and SHF receiver figure of merit (G/T), uplink and downlink loss factors, and the SHF ground station eirp and G/T. The Polarsat links are designed to permit operation with MUSAT earth stations as defined in Reference 5 and 6. As input to the link design, the following factors are required: spacecraft orbit, coverage area, frequency plan, and the overall per channel carrier power to noise density ratio. Polarsat, with the orbit selected from the mission analysis studies in Section 2.0, provides coverage to the required regions with the use of fixed UHF and SHF antennas having a half-power beamwidth of approximately 20 degrees. This corresponds to a boresight antenna gain of 20 dB. At the perimeter of the coverage area, however, the gain will be reduced by about 3 dB or half-power. Thus an edge gain of 17 dB is used for the SHF and UHF antennas in the link design. The selection of spacecraft eirp and G/T will specify the required transmitter RF power and the receiver noise temperature. Due to the change in spacecraft distance over the operational portion of the orbit from apogee to apogee plus or minus 4 hours, a free-space path loss variation of 3.1 dB must be included in the link design. The frequency plan specifies the links that are required, the number of channels and the frequencies for which the path loss must be determined. The Polarsat frequency plan is given in Section 3.2. The final factor determining the link design, the overall per channel carrier to noise density ratio, is arrived at from a consideration of the channel demodulator requirements for above threshold operation or operation not exceeding a maximum bit error rate in the case of digital transmission. The individual channel carrier-to-noise density ratios required are considered in the sections that follow. Subject to these inputs, the link design involves a trade-off of ground segment versus space segment link gains. The overall gain of a link is determined by the loss factors included. The loss factors consist of the free space path loss and other propagation losses arising from polarization mismatch, ionospheric attenuation and multipath. The propagation loss factors used in the link budgets of Tables A-1 to A-19 of Appendix A were taken from Reference 5. For the SHF part of the link design, the link gain is concentrated in the ground segment so as to minimize spacecraft SHF weight and power requirements. A typical SHF ground station receiving system with a 30 ft. antenna and a G/T of 32.3 dB/^{O}K is selected for the baseline design. The UHF link design requires the allocation of sufficient downlink RF power to enable satisfactory reception at a UHF mobile station. Since the UHF mobile antenna gains are limited by portability constraints and number of units required, the UHF RF downlink power dominates the spacecraft transmitter power requirements. As a result it is not considered necessary to use a larger ground station SHF antenna. The specific link design for each of the channels specified in the frequency plan is considered in the following sections.

3.3.2 SHF Uplink and Downlink Modulation

The SHF downlink and uplink transmissions for the SHF-UHF and UHF-SHF paths are angle modulated by the fixed 16 Kbps, telemetry, EPIRB and command channels to enable removal of the doppler shift produced by spacecraft radial motion. The link and modulation design of the SHF up and downlinks is governed by the requirements for adequate sideband and SHF carrier signal strengths. For angle modulation, the signal strengths of the sideband and carrier components of the total modulated SHF signal, are approximately related by the following expression:

$$J_o^2(\beta) = \frac{P_{carrier}}{P_{sidebands} + P_{carrier}}$$

where:

$P_{carrier}$ is power in residual SHF carrier

$P_{sidebands}$ is power in modulation sidebands

β is Overall modulation index

The signal strength of the remaining SHF carrier after modulation must be sufficient to permit angle demodulation using a tracking phase lock loop. The phase lock loop demodulator is constrained to operate over a loop bandwidth equal to plus or minus the doppler shift of the SHF carrier to enable lock-on. Typically a phase-lock loop demodulator requires about a 7 dB carrier to noise ratio in the loop bandwidth. For a maximum doppler shift to the SHF downlink of approximately $\pm$56 KHz, the carrier to noise density ratio of the SHF downlink carrier must be a minimum of 54.5 dB-Hz. For a maximum doppler shift to the SHF uplink of $\pm$61.6 KHz the carrier to noise density ratio of the SHF uplink carrier must be a minimum of 57.9 dB-Hz. The minimum up and downlink SHF carrier signal strengths for these carrier to noise density ratios are determined in the link budgets given in Tables 19 and 20 of Appendix A.

The minimum RF power in the modulation sidebands is the total of the minimum required RF powers for successful demodulation of each sideband component. The sideband components or modulating channels include the eight duplex 16 Kbps channels, ten EPIRB distress signals, and the command and telemetry channels. The minimum RF power for each component is determined in the sections that follow. The carrier and sideband RF powers for the SHF up and downlinks are given in Tables 3-3-1 and 3-3-2. Also indicated in these tables is the overall modulation index determined from the expression above for angle modulation. Using the Carson Rule for bandwidth calculation, the SHF downlink bandwidth, for 95% power transmission, is approximately 1.1 MHz and similarily the SHF uplink bandwidth is approximately 1.1 MHz. These bandwidths assume a baseband modulating signal for the SHF Uplink and downlink as shown in Figure 3-3-1.

SHF UPLINK

ANGLE MODULATION

UPLINK SIGNAL COMPONENT		MINIMUM POWER (dBw)
1) Carrier		45.7
2) Sidebands		
Command	50.9	
8 16 Kbps channels	53.6	
Total Sideband		55.5
3) Total Power		55.9
4) Overall Modulation Index		1.8

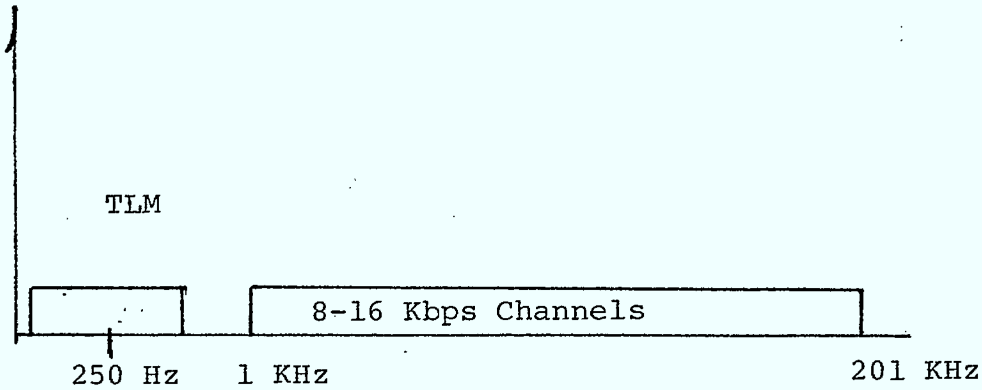
TABLE 3-3-1

SHF DOWNLINK

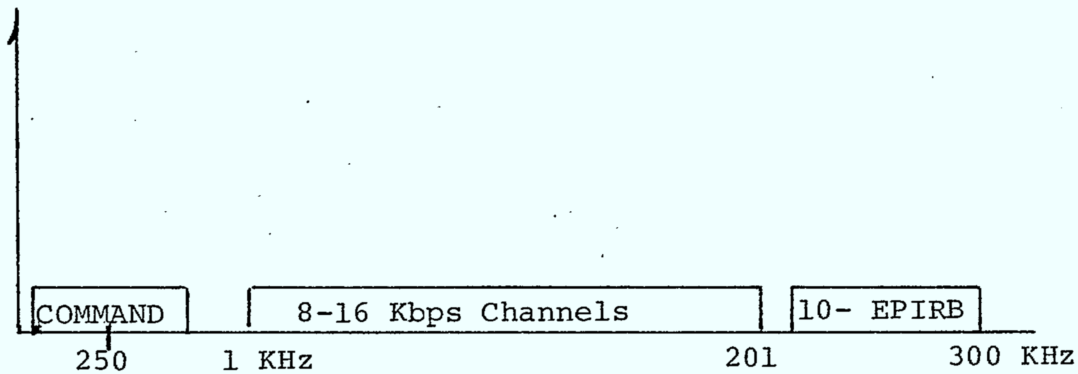
ANGLE MODULATION

DOWNLINK SIGNAL COMPONENT		MINIMUM POWER (dBw)
1) Carrier		-3.7
2) Sidebands		
Telemetry	-12.1	
EPIRP for 10 channels	-16.1	
8 16 Kbps channels	- 9.5	
Total Sideband		-7.0
3) Total Power		-2.0
4) Overall Modulation Index		0.8

TABLE 3-3-2



BASEBAND FOR SHF UPLINK



BASEBAND FOR SHF DOWNLINK

TABLE 3-3-3

3.3.3 16 Kbps Channels

Duplex transmission of voice and data is handled by a UHF-SHF forward path and an SHF-UHF return path from the fixed base stations. Due to the relatively weak UHF transmit signal levels, the UHF-SHF path is uplink limited. For the SHF-UHF patch, adoption of a downlink-limited design is desirable so as to limit the spacecraft UHF downlink transmit eirp by allocating more power to the SHF ground segment.

The objective of the link design for the transmission of 16 kbps voice and data channels is a minimum overall carrier to noise density ratio of 49 dB-Hz. This is dictaged by demodulator requirements for satisfactory operation. For the UHF-SHF path, the minimum overall carrier-to-noise density ratio is 53.3 dB-Hz and for the SHF-UHF path it is 49 dB-Hz. The link budgets are found in Tables A-2 to A-5 and A-14 to A-17. One of the 16 kbps channels is required for communications to/from a small jet aircraft. Since the jet uses an omni-directional antenna, a higher downlink eirp is required. The link budget is given in Table A-9 of Appendix A.

3.3.4 2 Mbps Wideband Channel

Duplex transmission of the 2 Mbps wideband channel is handled by the SHF-SHF forward and return paths between fixed stations. A downlink limited design is selected with a minimum overall carrier-to-noise density ratio objective of 73.0 dB-Hz to insure a bit error rate not exceeding 10^{-5}. The link budgets are found in Tables A-10 and A-11 of Appendix A.

3.3.5 Command/Telemetry Channel

The command and telemetry transmissions are carried on the SHF path from the central control station to the spacecraft. These transmissions can be implemented by a straight forward non-coherent FSK system at the low data rate of 256 bps considered. The up and down links are designed for a minimum carrier to noise density ratio which results in a bit error rate not exceeding 10^{-6}. These transmissions are carried as separate channels with a bandwidth of about 500 Hz. The link budgets are found in Tables A-12 and A-13 of Appendix A.

3.3.6 EPIRB Channels

The reception of EPIRB distress signals is considered in the link budgets given in Tables A-6 and A-7 of Appendix A. The link design is uplink limited due to the weak EPIRB transmit level of a maximum 5 watts and a nominal pattern loss resulting from poor antenna orientation. A downlink which allows no more than 0.1 dB degradation to the uplink is selected. The minimum overall carrier-to-noise density ratio is 32 dB-Hz. This is about 5 dB lower than the worst-case SARSAT value for the 406 MHz EPIRB signals.

3.3.7 Frequency Hopped 16 Kbps Channels

In addition to the eight 16 Kbps voice and data channels on fixed frequencies, there is a capability provided for two full duplex 16 kbps channels carried on frequencies which are selected according to a code. This transmission method is termed spread spectrum frequency hopping. For the Polarsat system, an RF bandwidth of 10 MHz has been selected for frequency hopping of two 16 kbps channels. As a minimum, the requirements for successful demodulation of the individual 16 Kbps channels apply, since the demodulation takes place after a wideband correlator.

This requirement is the same for the fixed 16 Kbps voice and data channels. Detailed spread spectrum receiver considerations, however, may result in a requirement for higher RF signal levels. For a 10 MHz RF bandwidth and an overall carrier-to-noise density ratio of 50 dB-Hz, the RF signal-to-noise ratio is -20 dB.

It is estimated that a margin of 2 dB is required to take account of possible spread spectrum receiver requirements. This means that the minimum overall carrier-to-noise density ratio of the individual frequency hopped channels is 51 dB-Hz. This is satisfied in the link budgets for the fixed UHF-SHF path. For the SHF-UHF path, the UHF downlink for each of the Frequency hopped 16 Kbps channels requires 2 more dB of downlink eirp as shown in the link budget of Table 4-8 of Appendix A.

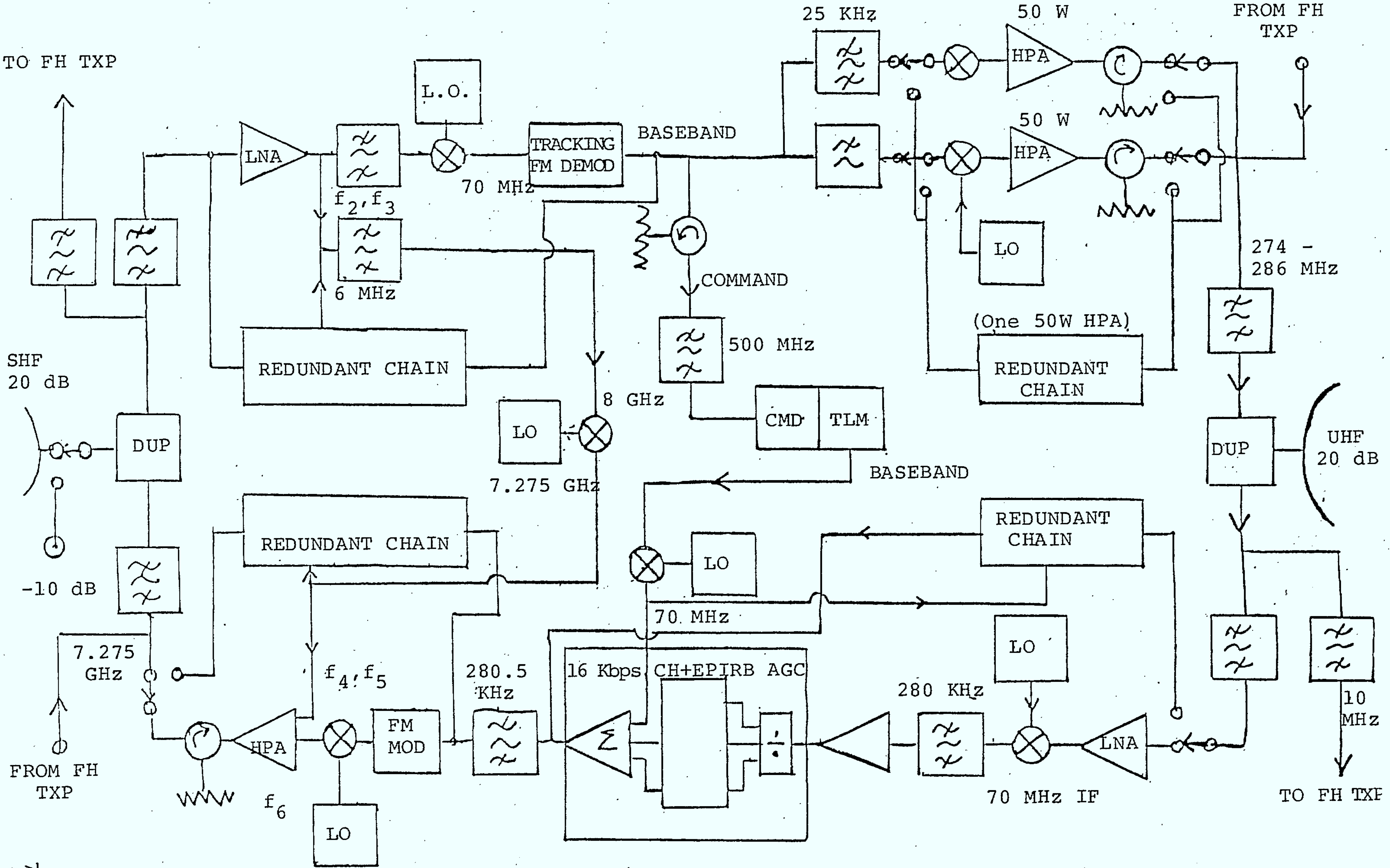
3.4 Spacecraft Communications System

3.4.1 General

The spacecraft communications system consists of two main subsystems: a transponder for the fixed 16 Kbps channels as shown in Figure 3-4-1 and a transponder for the frequency-hopped 16 Kbps channels as shown in Figure 3-4-2. The provision for a frequency hopped subsystem allows for operation in a substantial electromagnetic interference environment. For fixed frequency operation, the maximum tolerable per-channel interference level is determined by the margin above demodulator threshold. Since the fixed frequency system is designed to a nominal threshold to minimize spacecraft power and weight, little interference immunity is expected. For this reason channelized automatic gain control (AGC) has been incorporated in the design of the UHF-SHF chain of the fixed frequency transponder. This insures that a single high level interferer within one of the fixed 16 Kbps channels does not result in the suppression of all channels as would occur in a wideband AGC responding to total received power.

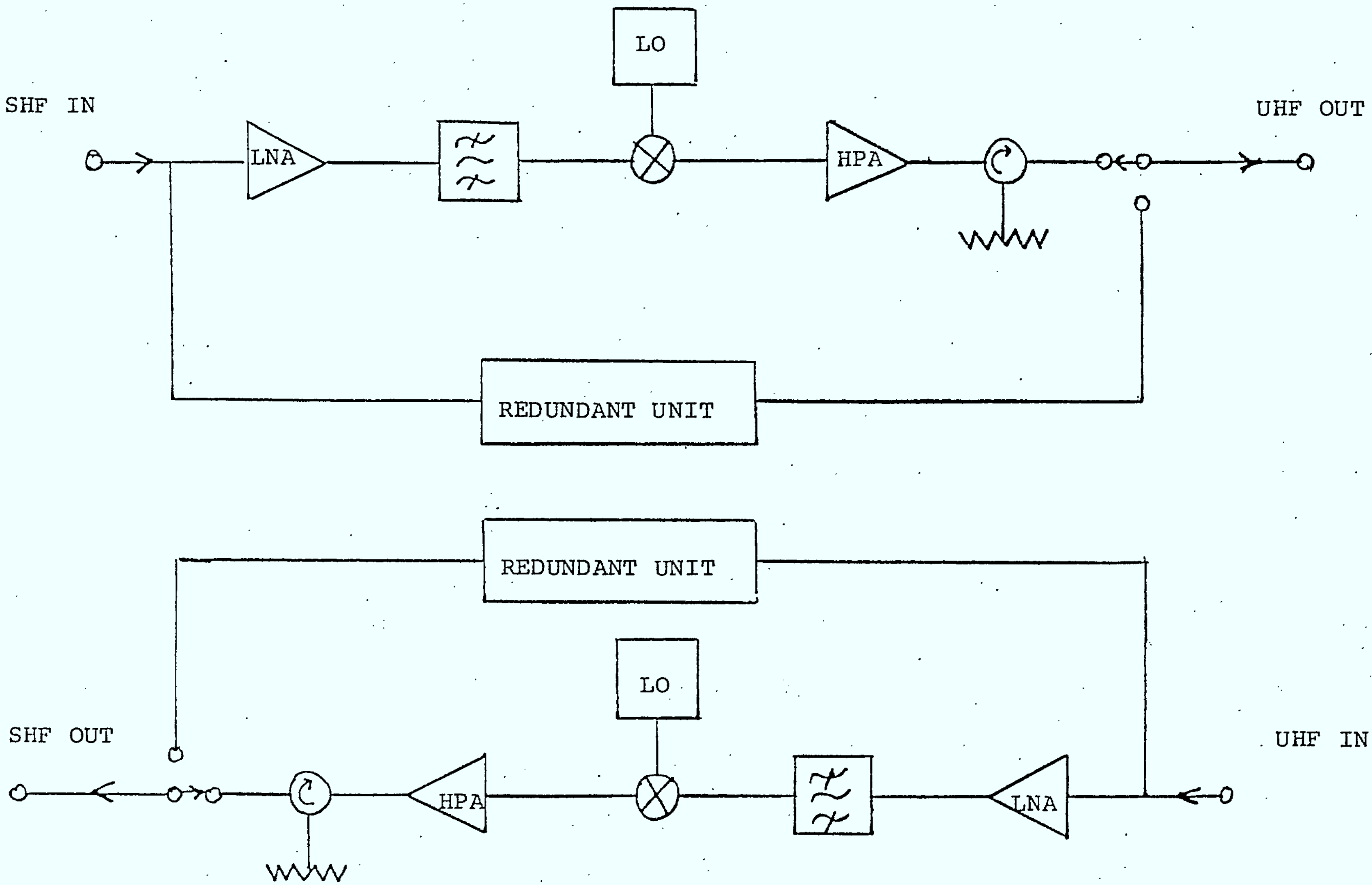
3.4.2 UHF-SHF Chain

The channelized AGC is part of the UHF-SHF chain consisting of a wideband UHF receiver followed by down-conversion to a 70 MHz IF. The channelized AGC controls the level of each downconverted uplink UHF channel to a preset value. This value is adjusted according to the manitude of per channel FM deviation desired in the FM modulator which FM modulates the fixed 16 Kbps channels, telemetry and EPIRB channels onto an intermediate frequency carrier. This modulated signal is then upconverted and amplified to the required SHF downlink power level. It is expected that the total required SHF output power level (including the SHF-SHF wideband data



POLARSAT TRANSPONDER

FIGURE 3-4-1



FREQUENCY HOPPED (FH) TRANSPONDER

FIGURE 3-4-2

link) of 33 dBm can be achieved with a solid state GaAs Fet class A power amplifier (e.g., MSC 88010).

3.4.3 Transponder Power Budget

The Polarsat transponder power budget shown in Table 3-4-3 indicates the breakdown of total transponder power among the communications SHF and UHF subsystems. The housekeeping power budget is given in Section 4.5. The transmitter power budget in Table 3-4-4 is based on the per channel eirp determined for each downlink from the link analysis in the previous section, and for the nominal number of channels to be carried. The transmitted power required for transmission to a small jet aircraft is obtained from a battery source since the service is expected to be intermittent. For an additional four duplex 16 Kbps channels, 50 watts more transmitter power is required giving a total of 181.5 watts.

3.4.4 SHF-UHF Chain

The SHF-UHF chain of the fixed frequency transponder consists of a wideband receiver followed by downconversion to a 70 MHz IF. A frequency tracking phase-lock-loop FM demodulator extracts the baseband spectrum from the SHF uplink signal. An AGC on reception is not required since the uplink signal is hard-limited by the FM demodulator prior to demodulation. The baseband spectrum is filtered so that one 25 KHz channel for communications to a jet aircraft, is applied to a 50 watt high power amplifier (HPA) while the remaining 16 Kbps channels are upconverted and amplified to a total power output level of 50 watts for 3 channels (total of 4 channels with jet channel) or approximately 100 watts for 7 channels at the UHF downlink frequency band. This arrangement allows simultaneous transmission to a jet aircraft and to other mobile UHF stations.

	NO.	POWER (WATTS)	TOTAL POWER (WATTS)
Total Transmitter Power	1	111	111
SHF LNA	4	0.5	2
SHF/70 MHZ LO	2	0.5	1
25 KHz/UHF LO	3	0.5	1.5
IF/SHF LO	2	0.5	1
SHF/SHF LO	1	0.5	0.5
UHF/70 MHZ LO	2	0.5	1
BSBD/70 MHZ LO	1	0.5	0.5
SHF/UHF LO	2	0.5	1
UHF/SHF LO	2	0.5	1
TRACKING FM DEMODULATOR	2	1.0	2
FM MODULATOR	2	1.0	2
70 MHZ AMPLIFIERS	14	0.5	7
TOTAL POWER			131.5

POLARSAT TRANSPONDER

POWER BUDGET

TABLE 3-4-3

Downlink	Per Channel EIRP (dBW)	Antenna Gain (dB)	Per Channel PWR (dBw)	No. of Channels	Total RF PWR (dBw)	RF PWR PER TMTR (dBw)	DC Power (Watts)
UHF-SHF for fixed 16 Kbps	-1.5	17	-18.5	8	-9.5		
UHF-SHF for FH 16 Kbps	-1.5	17	-18.5	2	-15.5		
UHF-SHF for EPIRB	-9.1	17	-26.1	10	-16.1	2.8	11
UHF-SHF for Telemetry	-22.1	-10	-12.1	1	-12.1		
SHF-SHF for 2 Mbps	15.0	17	-2.0	2	1.0		
SHF-UHF for fixed 16 Kbps	23.3	17	6.3	7	14.8	14.8	100
SHF-UHF to small jet	29.2	17	12.2	1	12.2	12.2	50 (on battery)
SHF-UHF for FH 16 Kbps	25.3	17	8.3	2	11.3	11.3	50
TOTAL							161

POLARSAT TRANSMITTER POWER BUDGET

FIGURE 3-4-4

3.4.5 Transponder Redundancy

As shown in the transponder block diagram, switches are provided to enable the connection of one redundant chain to replace either UHF 50 watt HPA. Redundant chains are also provided for the other receiver and transmitter subsystems in the manner shown in the transponder block diagrams of Figures 3-4-1 and 3-4-2. The redundant chain for the SHF receiver and FM demodulator of the SHF-UHF chair is permanently powered, to insure reception of command information despite the failure of one receiver/ demodulator chain.

3.4.6 SHF-SHF Chain

A duplex SHF-SHF chain is provided for the transmission of 2 Mbps data. This SHF uplink transmission is received through the SHF-UHF wideband LNA and is then filtered from the rest of the SHF uplink followed by downconversion to the desired SHF downlink centre frequency.

3.4.7 TT&C

The tracking telemetry and command signals of the Polarsat system are FM modulated onto the SHF up and downlink carriers of the SHF-UHF and UHF-SHF path as discussed in Section 3.2.5. After the SHF uplink signal is received and FM demodulated in the spacecraft the command subcarrier is filtered out from the rest of the baseband modulating signal consisting of eight 16 Kbps channels. The command subcarrier to then undergoes demodulation by a non-coherent FSK demodulator to extract the command information.

The telemetry information from various spacecraft sensors and subsystems, is FSK modulated at 256 bps into an intermediate frequency subcarrier and then frequency multiplexed with the UHF uplink signals at 70 MHz IF. This composite frequency multiplexed signal consisting of telemetry, eight 16 Kbps and the EPIRB distress channels FM modulate the downlink SHF carrier.

3.5 Ground Communications System

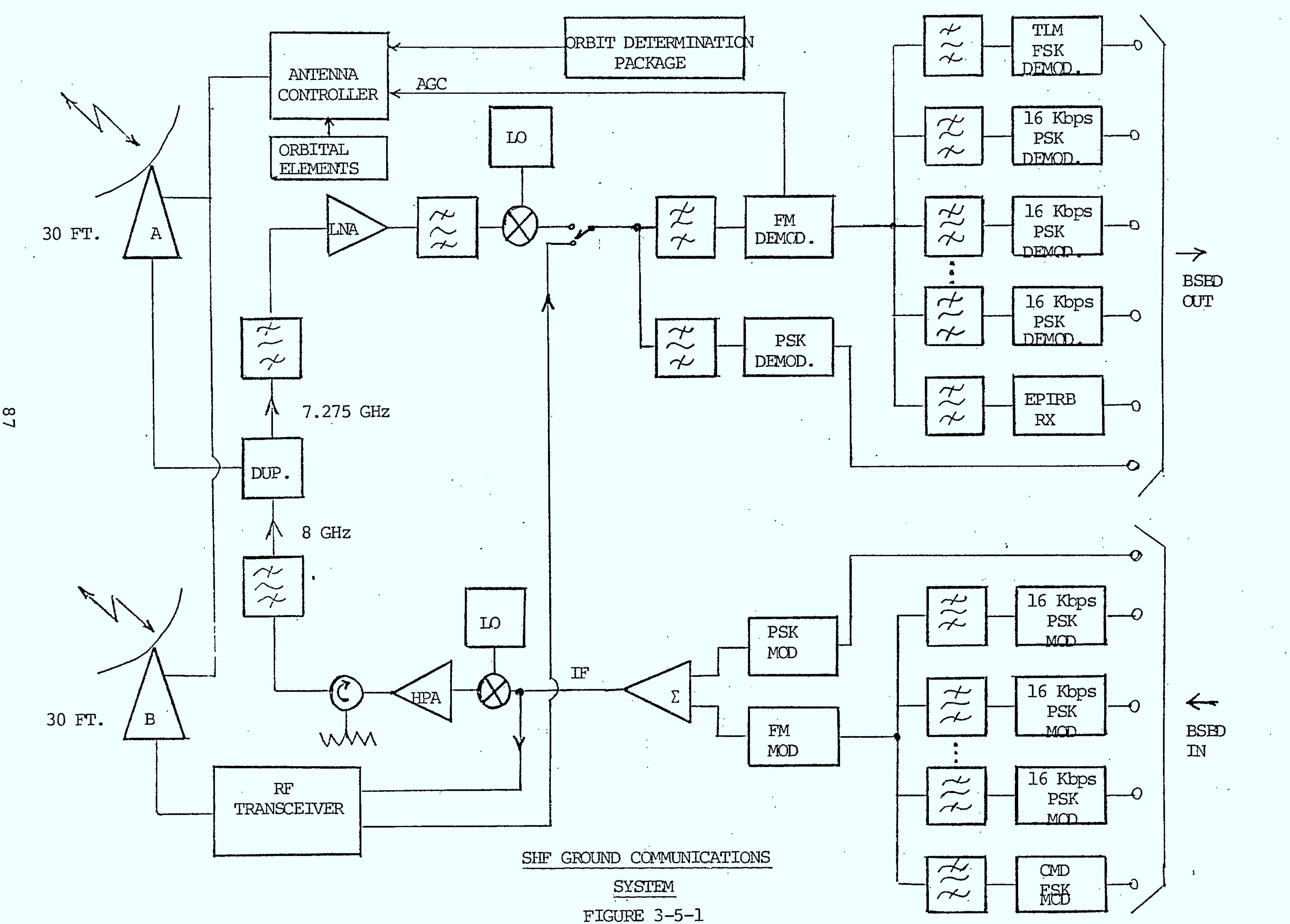
The Polarsat ground communications system is incorporated into the UHF and SHF ground segment. Since the UHF ground segment is defined according to the MUSAT System requirements, this section will consider only the SHF ground communications system. The parameters used for the MUSAT UHF Mobile Stations, taken from Reference 5 and 6, are given in Table 3-5-1. As indicated in Section 3.3, a standard 30 ft. dish with a G/T of 32.2 dB/^{O}K has been selected for the SHF ground station, from RF link considerations. The spacecraft is acquired at a minimum elevation angle of 5^{O} and tracked at an angular speed of 0-15 degrees/hour. The pattern of spacecraft visibility for constellations consisting of two or three spacecraft is considered in Section 2.0. To ensure continuity of SHF downlink reception, two tracking antennas per ground station site are required. Further, the same SHF downlink transmissions from each satellite are offset in frequency so as to avoid multipath problems that might arise if the same transmission frequencies are received from two satellites simultaneously. In this way, uninterrupted downlink reception is realized. This arrangement is shown in the SHF Ground communications system block diagram in Figure 3-5-1. The downlink signals from each antenna are received and down-converted to an intermediate frequency. The stronger of these two sets of downlink signals is selected. This selected signal is then filtered to separate the UHF-SHF downlink signal from the SHF-SHF downlink signal. The UHF-SHF downlink signal is FM demodulated to remove the downlink doppler shift and produce the baseband spectrum. This spectrum is filtered and each signal component undergoes further processing to extract the desired information. The telemetry subcarrier is demodulated by a non-coherent FSK demodulator, the 16 Kbps voice and data channels are PSK demodulated and the 406 MHz EPIRB uplink

	Team Pack	Shipborne	Airborne	Small Airborne
Antenna Gain (RX)	13 dB	10 dB	9.5 dB	0 dB
Antenna Gain (TX)	16.9 dB	13 dB	9.5 dB	0 dB
Receive G/T	-15.1 dB/^{O}K	-17.4 dB/^{O}K	-17.4 dB/^{O}K	26.9 dB/^{O}K
Transmit EIRP	26 dBW	31.1 dBW	26.6 dBW	26 dBW
Max. Data Rate	16 Kbps	16 Kbps	16 Kbps	16 Kbps
Min. C/N_O	49 dB-Hz	49 dB-Hz	49 dB-Hz	49 dB-Hz

MUSAT UHF MOBILE STATION PARAMETERS

TABLE 3-5-1

carrier frequency is measured to obtain a doppler history and the EPIRB identification code (if one is provided) is extracted by demodulation. The 2 Mbps data signal is PSK demodulated. Of all these baseband signals, the 16 Kbps voice and data channels and the 2 Mbps wideband channel are routed to an exchange facility which performs appropriate signalling functions and interfaces with the telephone switching networks (at a Regional Control Station). Demand assigned multiple access (DAMA) operation may also be performed by the exchange if required. The output from the exchange is routed to the individual transmitter channel modules consisting of a PSK modulator and 25 MHz filter. The 16 Kbps voice and data subcarriers are combined along with a 256 bps FSK modulated command subcarrier. This composite baseband signal FM modulates an intermediate frequency carrier which is then combined with the 2 Mbps data subcarrier. This combined IF signal enters a splitter which distributes the signal to the RF transceiver systems of the two antennas. Each transmitter subsystem of the RF transceiver system upconverts the IF signal to produce two SHF carriers. The two SHF carrier frequencies in one antenna RF transmitter are different from these in the other antenna transmitter systems. Those signals are amplified by their respective high power amplifiers and output to the antenna. The transmitter output power requirement per antenna RF transceiver system is indicated in the table shown in Figure 3-5-2. These power levels are based on the eirp values required from the link calculations. The antennas have a half-power beamwidth of 0.3 degrees. The antennas require pointing instructions based on the results of an orbit determination package. The pointing instructions may be generated by a programmed tracking antenna controller. The controller orbit parameters are checked periodically against the orbit determination package results and updated with new orbit elements if required. The orbit determination



SHF GROUND COMMUNICATIONS SYSTEM

FIGURE 3-5-1

Uplink	Per Channel EIRP (dBW)	Antenna Gain (dB)	Per Channel PWR(dBW)	No. of Channels	Total RF Power(dBW)
SHF-SHF for 2 Mbps data	66.8	55	11.8	2	14.8
SHF-SHF for Command	50.9	55	-4.1	1	-4.1
SHF-UHF for Fixed 16 Kbps Channel	44.6	55	-10.4	8	-1.4
SHF-UHF for FH 16 Kbps Channels	44.6	55	-10.4	2	-7.4

SHF TRANSMITTER RF POWER BUDGET

FIGURE 3-5-2

package uses the range-rate measurements of a ranging tone passed through the spacecraft transponder on the SHF-SHF path.

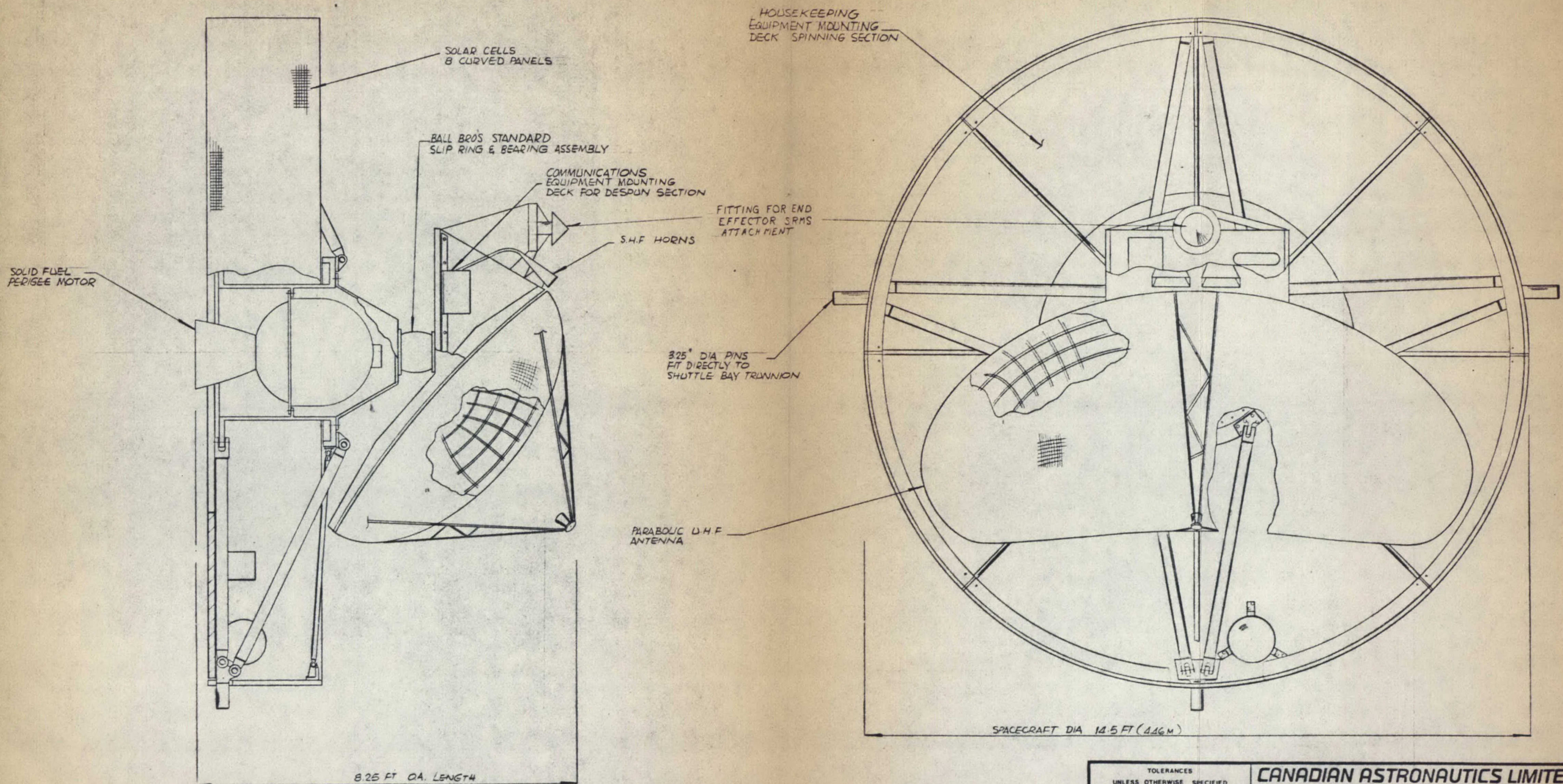
4.0 SPACECRAFT DESIGN

4.1 Spacecraft Configuration

The spacecraft designed to carry the Polarsat communications payload is shown in Figure 4-1-1 The design is similar to that of a conventional dual-spin satellite such as ANIK A, with the exception that the rotating cylindrical portion has a much larger diameter-to-length ratio. This feature contributes to the large moment-of-inertia ratio required for attitude stability. As with a large number of geosynchronous communications satellites, the solar array is mounted on the cylindrical side surfaces of the drum. Because the drum rotates, the thermal design of the array is simplified. All of the housekeeping subsystems except for the TT&C are housed in the spinning cylinder.

The despun platform carries the communication payload, which consists of the transponder deck and the antennas. Only DC power and baseband/command and telemetry signals are transferred across the rotating joint. This is accomplished by means of a standard 40 element slip-ring unit which is an integral part of the despin bearing assembly.

There are four antennas mounted on the despun platform. The largest of these is a parabolic reflector with a focal-point feed which provides a coverage at UHF tailored for the calculated change in view angle from the spacecraft to the earth. On the same boresight, two horn antennas are provided to give similar coverage pattern at SHF (7-8 GHz). The boresight of all these antennas are fixed at an angle of 41.5° from the spacecraft spin axis, so that the antennas will point to the required coverage area when the spacecraft is in the active portion of its orbit, with its spin axis aligned parallel the earth's spin axis. The fourth antenna consists of an array of four patch antennas



POLARSAT — SPACE CRAFT CONFIGURATION

FIGURE 4-1-1

CANADIAN ASTRONAUTICS LIMITED
Suite 201, 1024 Morrison Drive, Ottawa, Ontario K2H 8K7
(613) 820-8280

TOLERANCES UNLESS OTHERWISE SPECIFIED

☐ METRIC — DIMENSIONS IN ☐ mm ☐ cm UNLESS OTHERWISE SPECIFIED — NO DECIMAL ±.5 mm, .X ±.1 mm

☑ BRITISH — DIMENSIONS IN INCHES UNLESS OTHERWISE SPECIFIED — FRACTIONS UNDER 6" ± 1/64, 6" - 24" ± 1/32, OVER 24" ± 1/16; DECIMALS .XXX ± .005, .XX ± .010, .X ± .015

ANGLES ± 1/2°

	NAME	DATE
DRAWN BY	JSG	3 APRIL
CHECKED BY	RB	3 APR.
MECHANICAL		
ELECTRICAL		
APPROVED BY	[illegible]	[illegible]

PROJECT: POLARSAT

TITLE: CONFIGURATION

DRAWING NO:

REVISION

SCALE 1:12 SHEET OF

designed to give omni-directional coverage for command and telemetry purposes. A diplexer combines the high and low gain signals to the receivers. The link calculations for command and telemetry assume 10 dB nulls in the omni-directional pattern while maintaining a bit-error rate of less than one part in 10^6. This spacecraft design is convenient from the point of view of assembly and test. The housekeeping units are all selected from standard flight qualified designs. These are integrated within the spinning unit and can be assembled and tested separately from the payload section. In the same manner, all the antennas can be assembled and tested together as a unit prior to integration with the transponder payload. The transponder itself is built up to on an equipment deck which then becomes the third major spacecraft subassembly. The antennas and transponders are then integrated, checked out and assembled to the spinning section.

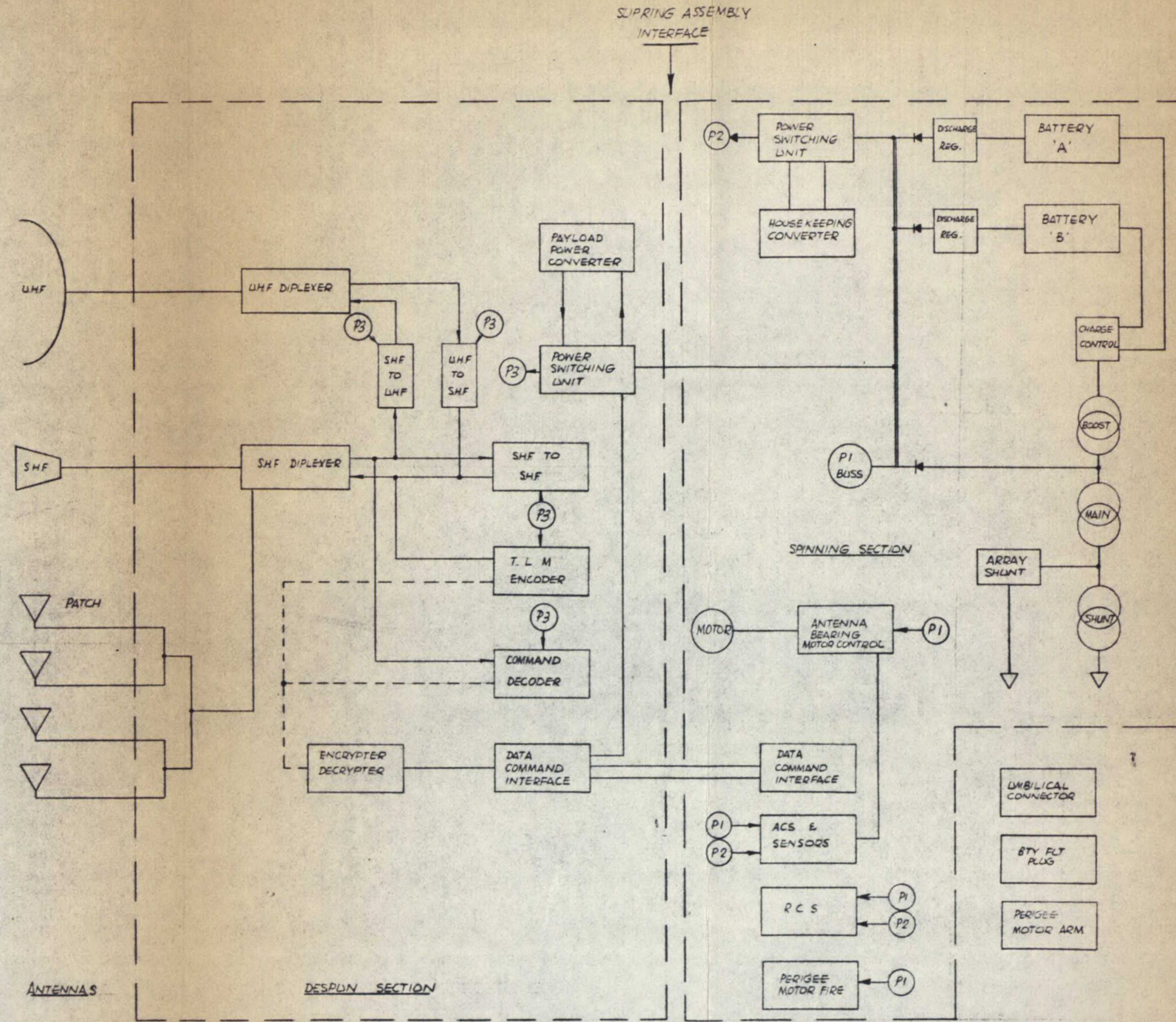
Note that any of the three major subassemblies as well as the perigee motor are interchangeable between the three or four flight model spacecraft. This will permit greater flexibility during the assembly and test phases of the program.

In general, the spacecraft will be relatively inexpensive to manufacture and test because it is simple, and because of the conventional design approach.

4.2 Spacecraft Block Diagram

An overall block diagram of Polarsat is shown in Figure 4-2-1. The most significant feature of this design is its simplicity.

9/a



POLARSAT ~ BLOCK DIAGRAM

FIGURE 4-2-1

TOLERANCES UNLESS OTHERWISE SPECIFIED		CANADIAN ASTRONAUTICS LIMITED Suite 201, 1024 Morrison Drive, Ottawa, Ontario K2H 8K7 (613) 820-8280			
☐ METRIC	☐ BRITISH		NAME	DATE	PROJECT POLARSAT
DIMENSIONS IN ☐ mm ☐ cm UNLESS OTHERWISE SPECIFIED	DIMENSIONS IN INCHES UNLESS OTHERWISE SPECIFIED	DRAWN BY	JSG	3 APR 16	
	FRACTIONS UNDER 6" ± 1/64	CHECKED BY			TITLE
	6" - 24" ± 1/32	MECHANICAL			
	OVER 24" ± 1/16	ELECTRICAL			
NO DECIMAL ± .5mm	DECIMALS XXX ± .005				DRAWING NO:
.X ± .1 mm	XX ± .010				REVISION
	X ± .015				
ANGLES ± 1/2°		APPROVED BY			SCALE: — SHEET OF

7 6 5 4 3 2 1

4.3 Spacecraft Subsystems

4.3.1 Transponder

The transponder operation is functionally described in Section 3.4.1. All of the transponder units are mounted on an equipment platform which doubles as a despun section structural member. This configuration has several advantages.

Firstly, all the signal interconnects to the antennas are kept short, and do not require articulation. This minimizes the feed losses and simplifies assembly.

Secondly, the transponder integration and testing is facilitated because of the ease of access to all components. Test jigs are also simple and external power interfaces are the primary power bus voltage.

The integration and test of the transponder and antennas is straight-forward and can be carried out independently of the housekeeping (spinning) section of the spacecraft.

Thirdly, the modularity of the spacecraft design should result in cost savings because any of the flight transponder payloads can be interchanged during the integration phase. This should provide some additional flexibility in scheduling to meet program milestones.

4.3.2 Antennas

There are four antennas on Polarsat, a UHF dish, two SHF horns, and a set of SHF omni-patches. Each of these antennas must operate with a diplexer for transmit and receive.

The UHF antenna is a focal-point-feed parabola with a shaped reflector. The reflector dimensions are approximately 3.8 m x 2.2 m with the feed 1.3 m in front of the vertex. The boresight of this antenna is oriented at an angle of 41.5^{o} from the spacecraft spin axis. This orientation, in conjunction with the resulting 16^{o} East-West by 27^{o} North-South beam provides coverage of the northern hemisphere from four hours before apogee to four hours after apogee, for a spacecraft in a 63.4^{o}, 12 hour inclined orbit.

The large parabolic reflector is mounted above the transponder deck on the despun platform. There are two options for the feed, a Cassegrain or a focal-point design. For the conceptual design, a focal-point feed has been chosen. In either case, the UHF diplexer is mounted beneath the antenna, directly on the transponder equipment deck.

The antenna reflector is made of wire mesh bonded onto a ribbed substructure. Mesh is chosen to minimize the effects of asymmetric solar pressure torques which can affect the spacecraft attitude.

The SHF horn antennas are designed to provide the same coverage pattern as the UHF antenna. The boresight of these antennas is thus parallel to that of the parabolic dish. The pyramidal horns are approximately .3m long with an aperture .26 m x 0.09 m. They are designed to mate with WR112 waveguide. These horns mount directly to the transponder deck, where it is easy to provide solid mechanical mounts. Each horn is fed by waveguide from the SHF diplexer.

	GAIN	COVERAGE
UHF up downlink	20 dB	$16^{\circ} \times 27^{\circ}$
SHF up downlink	20 dB	$16^{\circ} \times 27^{\circ}$
SHF cmd tlm	-10 dB	omni-directional

POLARSAT ANTENNAS

TABLE 4.3.1

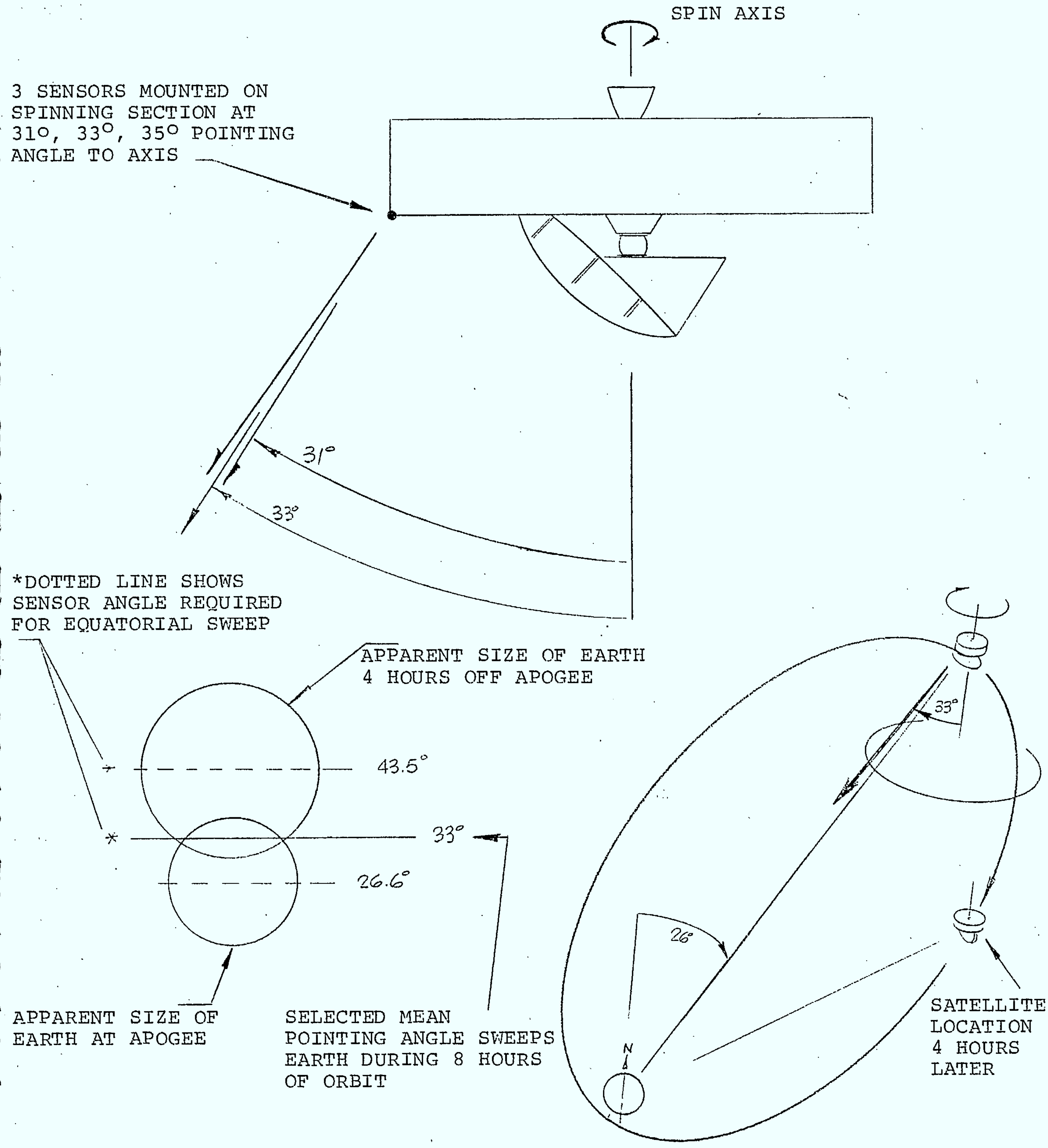
The SHF patch omni-directional antenna is also fed from the SHF diplexer. The four SHF patch antennas are mounted around the periphery of the despun platform at 90° intervals. This antenna provides an omni-directional coverage for command and telemetry during the station and attitude acquisition maneuvers.

The link calculations presented in Section 3 assumed the gains listed in Table 4.3.1 for the three antennas. It is noted that 10 dB nulls in the omni-directional pattern have been assumed, however, there may be a deeper null in the direction of the perigee motor throat. This is not considered to be a problem as the spacecraft attitude throughout the mission is planned to avoid this orientation with respect to the ground control station(s).

4.3.3 Attitude Control System

The spacecraft is stabilized by causing one of the sections of the spacecraft body to spin at about 30 rpm. The spin axis is aligned parallel to the earth's spin axis by firing a series of pulses using the reaction control subsystem thrusters (Section 4.3.4). Once aligned, it remains inertially stable except for small perturbing torques such as solar radiation pressure. These small torques together with the slow rotation of the orbit plane necessitate periodic corrections to be made.

The determination of the spacecraft attitude is done using earth horizon sensors mounted on the spinning body. Figure 4.3.1 illustrates the geometry involved. The sensors are oriented in such a manner that the sensors periodically intersect the earths' horizon. The sensors operate in the IR at about 15 μm wavelength. By measuring the difference between the earth dwell times of the sensors using a digital clock pulse counting technique, the roll angle can be determined. Because



SENSOR GEOMETRY

FIGURE 4-3-1

of roll-yaw coupling due to orbit motion, measurements made over an extended orbital arc all the inertial direction of the spin axis to be computed. An accuracy of ± 0.1 degrees represents the state-of-the-art limit with this technique and is sufficient to enable the spin axis attitude to be controlled to within one degree tolerance. Provision must be made in the computations for the earth's shape and the satellite's orbit.

The measurement of pitch angle of the spinning section is easily derived from the time of intersection of the sensor with the earth's horizon signals. Clock pulses are counted up from the onset of earth's horizon once roll and yaw have been determined. The pitch angle of the despun section is measured digitally using a pipper clocking signal from a small magnetic or optical pickup (mounted between the spinning and despun sections) and either of the earth horizon signals. Clock pulses are counted up from the onset of earth horizon to the pipper signal and then counted down until the earth horizon signal disappears. The remainder gives a measurement of despun section pitch angle with respect to local vertical. This signal output drives a servo loop controlling the despin electric motor and hence the despun section is locked onto the earth. A commandable digital offset register is used to steer the despin section to any arbitrary pitch angle or to correct for biases.

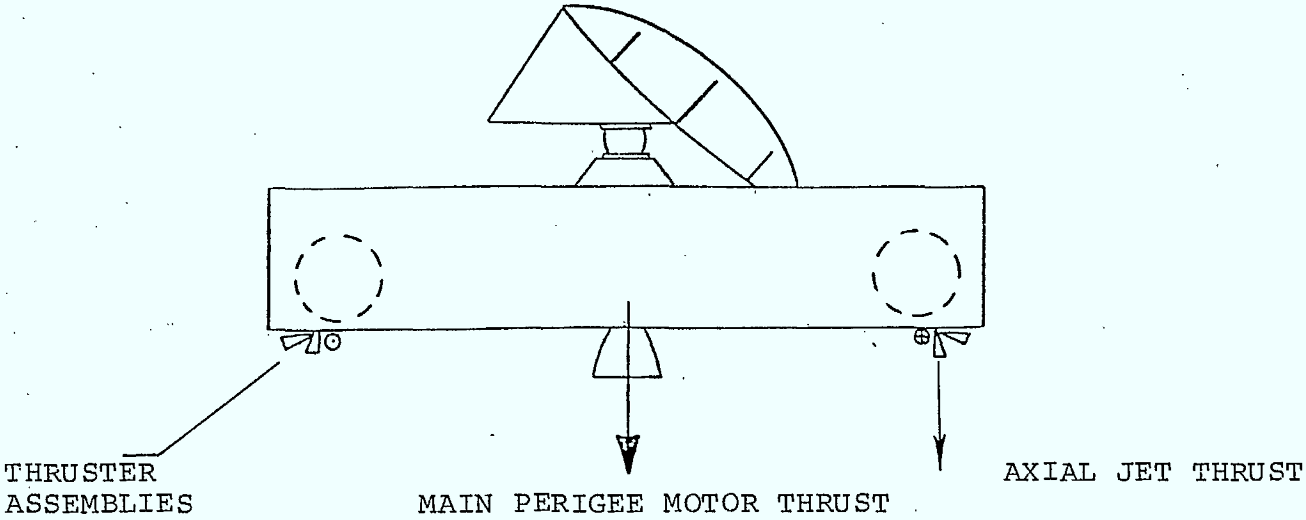
During the perigee of the orbit, the despin motor is switched to a constant-speed mode as the earth will not be in the field-of-view of the horizon sensors at this time. When the earth horizon is required, the despun platform will reaquire the earth.

During the orbit and acquisition phases at the start of the mission, a dual-slit sun sensor is used to aid in attitude determination at times when the horizon sensors may not be effective. The sun sensor slit has a 90 degree field of view and the two slits are aligned at an angle with respect to each other to eliminate geometric ambiguities.

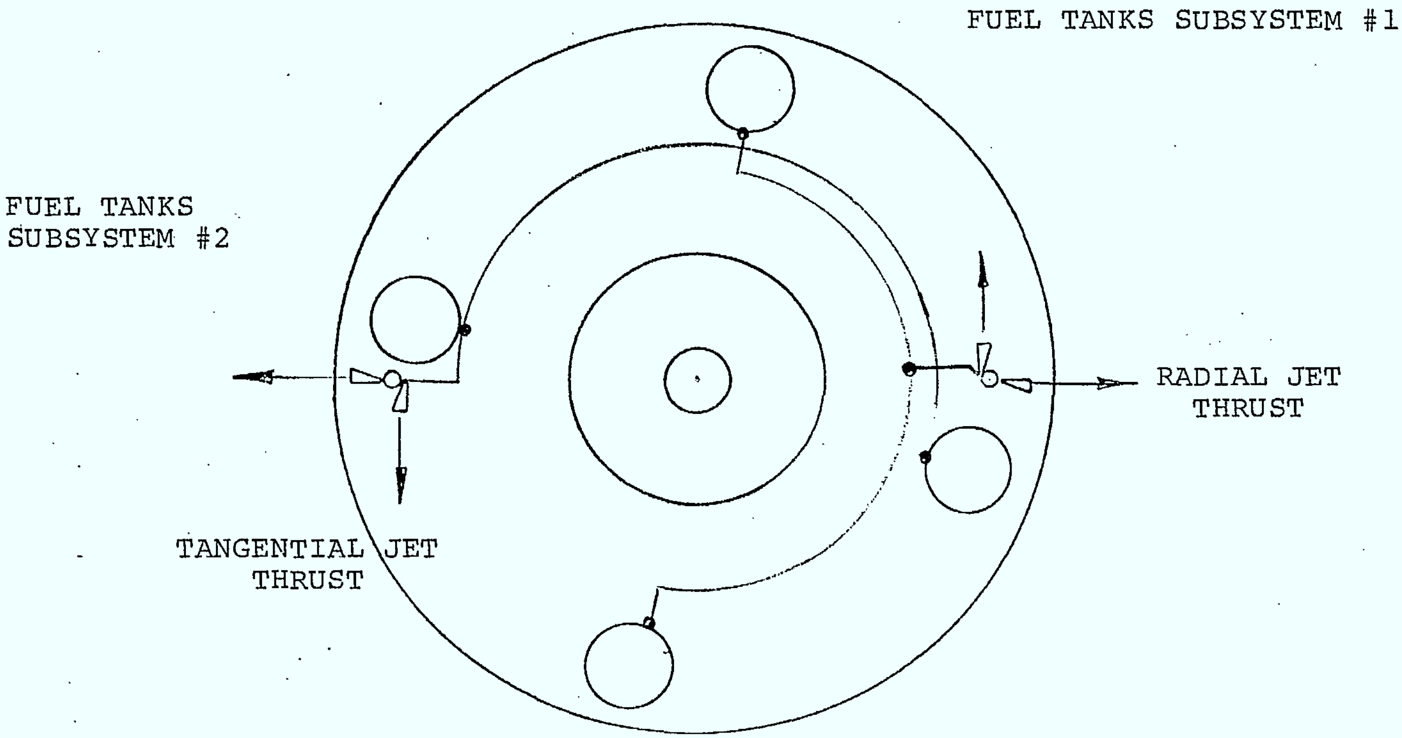
4.3.4 Reaction Control System

The orbit trim maneuvers, orbit and attitude correction maneuvers and spin speed changes are performed using a monopropellant hydrazine reaction control system. The propulsion system is redundant consisting of two sets of thruster assemblies, two sets of opposed fuel tanks, plumbing and thruster control electronics. Figure 4.3.2 illustrates the RCS configuration. Each thruster assembly is composed of one axial, one radial and one tangential thruster. The specific impulse typically achieved with a catalytic decomposition type thruster is about 220 sec. for continuous thrusting and 175 sec. for pulsed operation.

Orbit correction maneuvers are performed using the radial jets, whose thrust vector passes through the center of gravity of the spacecraft. By firing a radial jet in a pulsed mode, synchronized with the spin rate of the spacecraft, a velocity change can be effected which can be in any selectable direction normal to the spin axis. There is also in this type of maneuver, a parasitic axial velocity change component which is largely nullified in any orbit correction maneuver sequence. Attitude correction maneuvers are effected by firing an axial jet in spin synchronized, pulsed mode, allowing an inertially fixed torque to be applied to the spacecraft. Spin-up of the spacecraft is effected by firing tangential thrusters in continuous mode providing a torque about the spin axis.



SIDEVIEW OF SPACECRAFT



VIEW FROM BOTTOM OF SPACECRAFT

AUXILIARY PROPULSION SUBSYSTEM CONFIGURATION

FIGURE 4.3.2

4.3.5 Power Subsystem

The Polarsat power system is required to convert the incident solar flux into electrical power, regulate and store this power, convert the voltage level at which this power is available to appropriate levels for spacecraft subsystems, and switch the power to the proper subsystems.

The end-of-life power required for the Polarsat housekeeping and payload is estimated at 191.5 watts which corresponds to 320 watts beginning-of-life (40% degradation over five years). The power system design for this spacecraft is a conventional shunt-tapped array with a boost section for battery charging. The system is equipped with two batteries, giving full redundancy. The power converters are standard high-reliability units which have been used recently on the Viking lander program. Power switching is accomplished with relays. The power subsystem block diagram is shown in Figure 4.3.3.

4.3.5.1 Solar Array

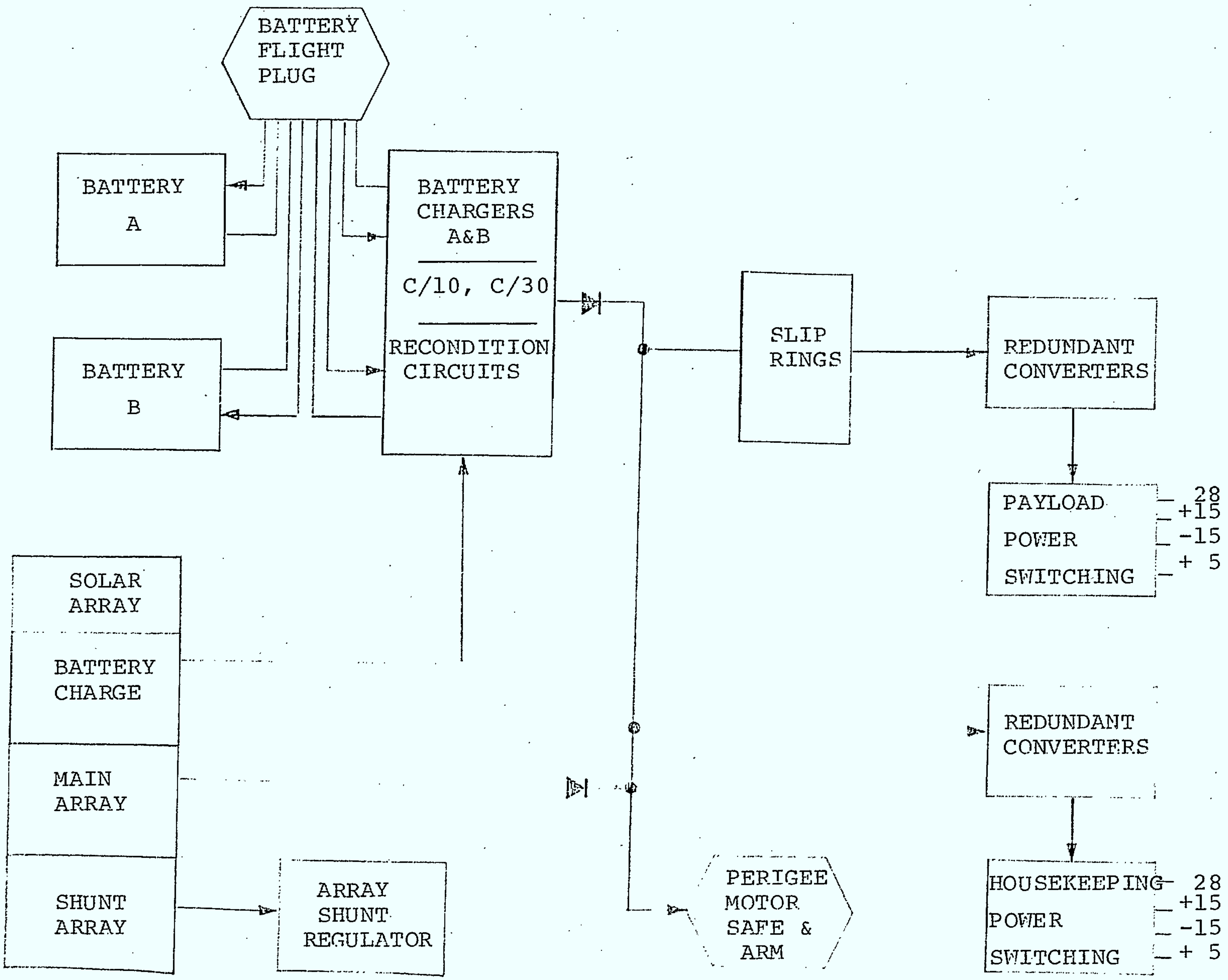
The solar array forms the curved sides of the cylindrical spacecraft body. A preliminary design, for an array with 400 watts beginning-of-life power, is presented in Table 4.3.2. This design is based on typical parameters for 10Ω-cm n on p silicon cells. A conservative degradation factor of 10% per year will result in an array output power of 240 watts at the end of five years in orbit. Thus the array is 48.5watts oversized for the payload.

The array is divided into 3 parts, the main array, a shunt-tapped array, and a battery charge array. The main array is sized to provide the required operating power at beginning of life while running at minimum

	CELLS IN SERIES STRING	CELLS IN PARALLEL (NORMAL SUN)	TOTAL CELLS FOR CYLINDRICAL ARRAY	AREA m^2
MAIN	63	90 + 6	19,000	7.6
SHUNT	20	90 + 6	6,000	2.4
BATTERY CHARGE	20	6	380	.15
			25,380	10.15

SOLAR ARRAY DESIGN (400 watt B.O.L.)

TABLE 4.3.2



POLARSAT POWER SUBSYSTEM

FIGURE 4-3-3

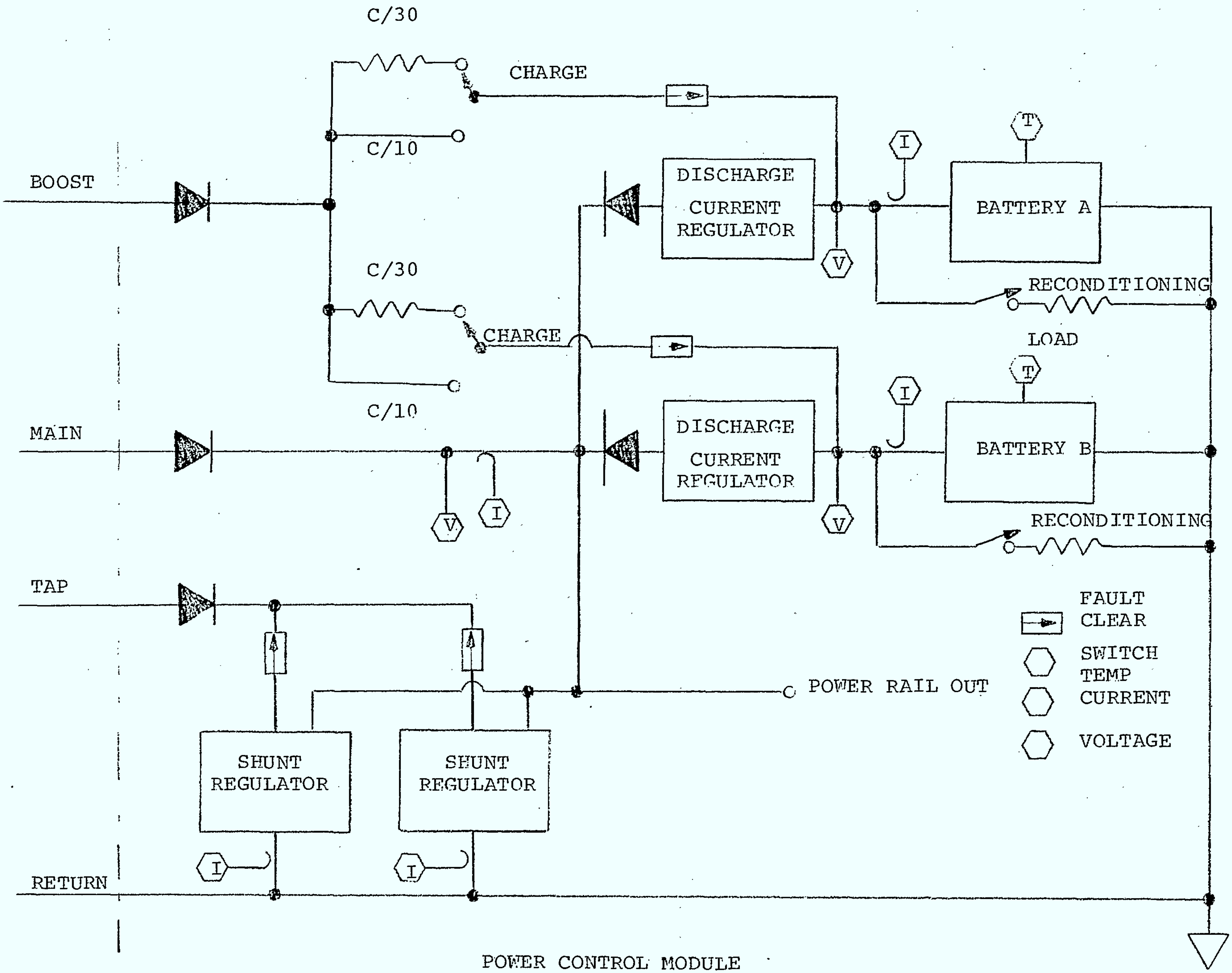
temperature. At this time, the shunt is fully on. The shunt-tapped array is then sized to provide the additional power that will be required at end-of-life with the array at maximum temperature. In this condition, the shunt is fully off. At conditions intermediate between these two extremes, the shunt is proportionally turned on or off. A residual load is always left on the array to limit the voltage peak after an eclipse.

The batteries are charged from an additional array which is used to provide sufficient short-circuit current to charge each battery at C/10. This section of the array produces a voltage which is added in series with the main array voltage in order to provide the additional potential required to fully charge the batteries. As the battery reaches full charge, the charging current is automatically tapered by this type of charge circuit.

4.3.5 Battery

The Polarsat battery is planned to be redundant and of sufficient capacity to support a load of 60 watts for one hour at 50% depth of discharge with one battery failed. These requirements result in a 4 amp-hr capacity battery. The resulting battery size and mass is summarized in Table 4.3.3.

Each battery is charged by means of a boost section of the solar array which is sized such that the short-circuit current of the cells will provide a C/10 charge rate. Both batteries can be trickle-charged on command. The charge circuitry is shown in Figure 4.3.4.



POWER CONTROL MODULE

FIGURE 4.3.4

Requirement: 60 watt-hr, 28 volt bus, 50% D.O.D.
results in 4 amp-hr cell

Cell Size: 8.1 cm x 2.11 cm x 5.41 cm
190 gm each G.E.

Battery: 28 cells/battery - 5.32 kg of cells/battery
and total assembly of 7 kg/battery
total estimated battery size (l,w,j) = 59 cm x 5.41 cm x 8.2 cm

Spacecraft battery compliment:

- 14 kg of battery
- 4 cell pack units each 30 cm x 5.4 cm x 8.2 cm
- battery discharge regulator
- battery charge controller

POLARSAT BATTERY DESIGN

TABLE 4.3.3

In normal operation, both batteries are on line at the same time. Discharge current regulators ensure that the batteries share the load equally.

4.3.5.3 DC-DC Converters

The requirements for regulated voltages for logic circuits and analogue signal conditioning is provided by DC to DC converters. These are modular devices which are assembled into the required converter configuration by the spacecraft integrator. The devices chosen to establish the baseline are manufactured by Power-Cube Corporation. Two of these (redundant) units are mounted on the spacecraft, one for housekeeping units, and one for the payload.

4.3.5.4 Power Switching Unit

Power is controlled by means of two relay boxes called power switching units (PSU's), one on the spinning and one on the despun portion of the spacecraft.

4.3.5.5 Battery Flight Plug, Arming Connector, Umbilical Connector

Each battery is connected to its charge rail and its loads via the pins of a connector receptacle located on the spacecraft exterior. For ground operations, the pins of this receptacle are used for charging the batteries and/or for operating the loads from external power supplies. A mating connector (the Battery Flight Plug) is provided which completes the battery connections for flight use. Connections are made between the charge rail inputs of this receptacle and the charging lines of the spacecraft umbilical connector to permit battery charging during countdown until liftoff.

A separate connector is installed on the exterior of the space craft which, when installed with a mating connector, enables the perigee motor firing circuits. An umbilical connector is provided for ground test of the integrated spacecraft and for check-out prior to launch.

4.3.6 Structure

The Polarsat structure conceptual design meets the constraints of the payload and mission which include the baseline three point interface to the STS cargo bay, the mechanical environments discussed in Section 4.4, perigee motor firing loads and structural dynamics critiera. Figures 4-3-5 and 4-3-6 show the housekeeping section structure and Figure 4-3-7 shows the despun communications section structure. The main features are:

- o The communications platform, bearing assembly, and perigee motor, which are roughly located at the cargo bay centerline, are supported in the X direction by a compound truss which spans between the longeron attach points. Z axis loads are also reacted at the two longeron points and Y direction loads are reacted at the keel point. The keel fitting is also required to support X axis loads to prevent spacecraft rotation about its (YY) axis.

- o A star-shaped truss is used to carry the primary loads. For this report, steel was consdered but the members may be aluminum, steel or perhaps composite material. A trade-off is required to determine whether it if cost effective to design the structure for minimum weight, thereby decreasing the required size of the perigee motor.

- o The perigee motor is mounted in an aluminum thrust tube which has heavy flanges to provide moment continuity for the truss members spanning the cargo bay. It can be either a machined casting or an aircraft-type rivetted ring/sheet/stringer assembly. the motor is mounted on an integral high strength ring.

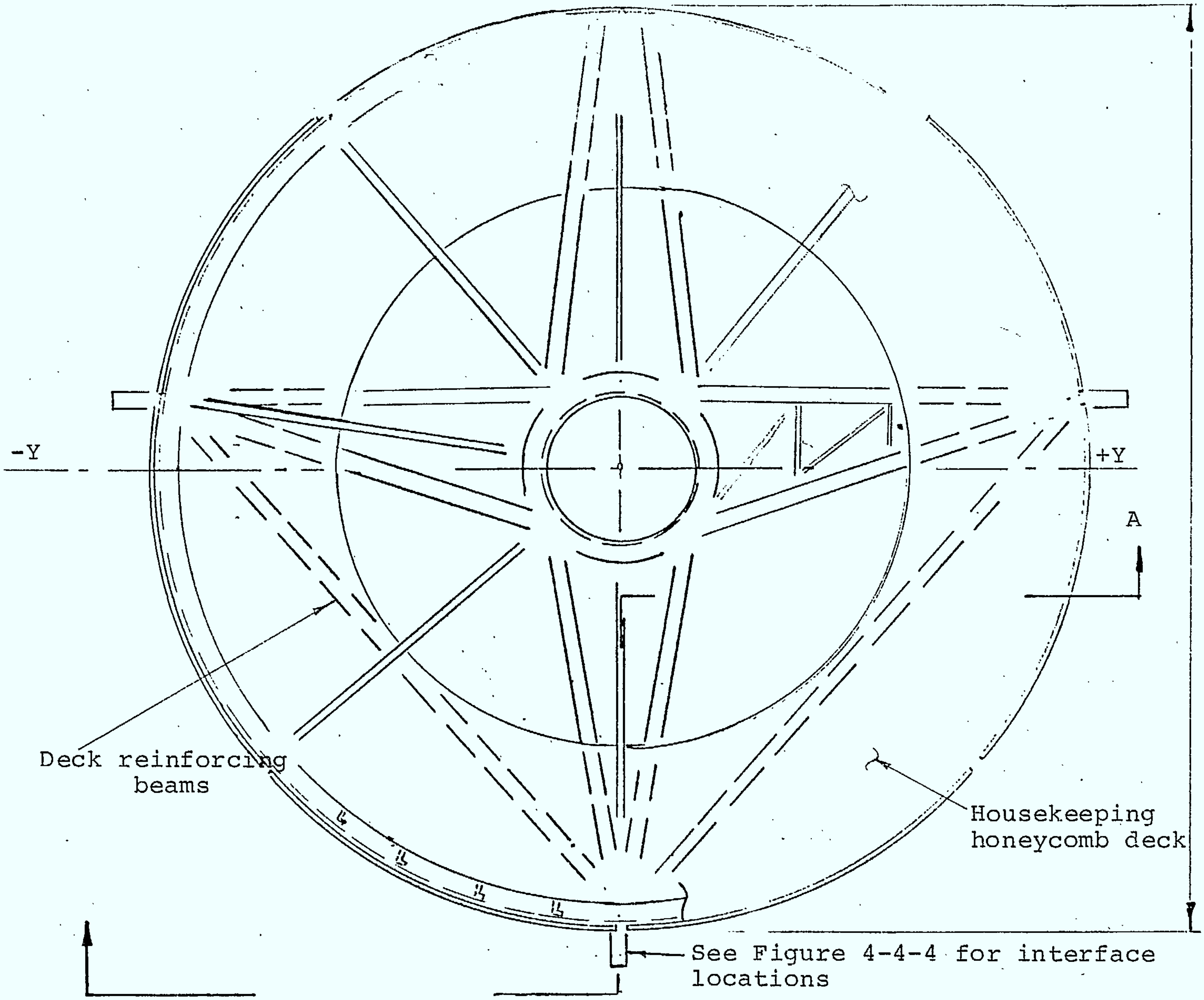
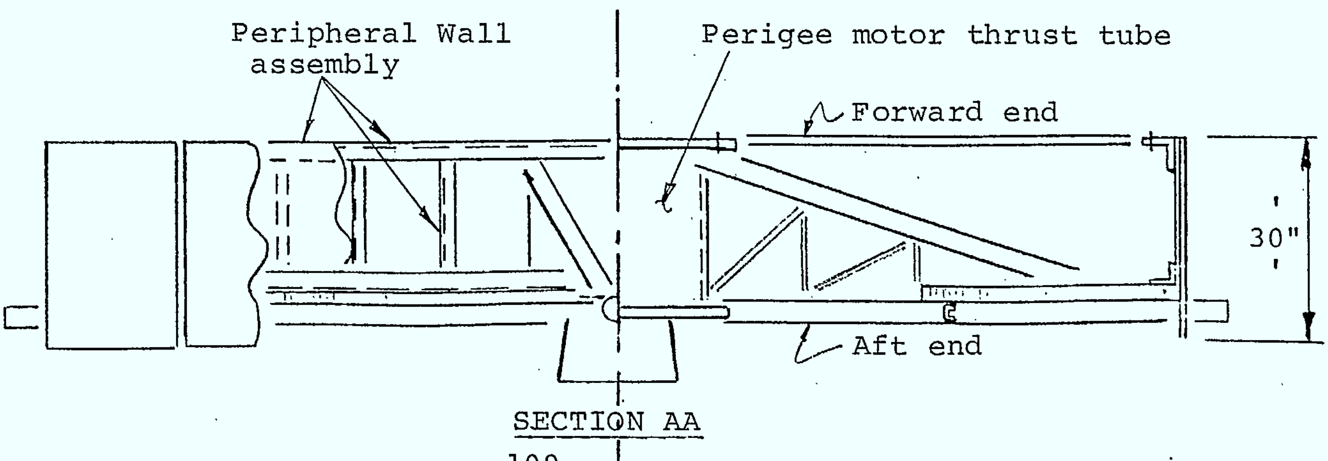


FIGURE 4-3-5

STRUCTURE, SPINNING SECTION



SECTION AA

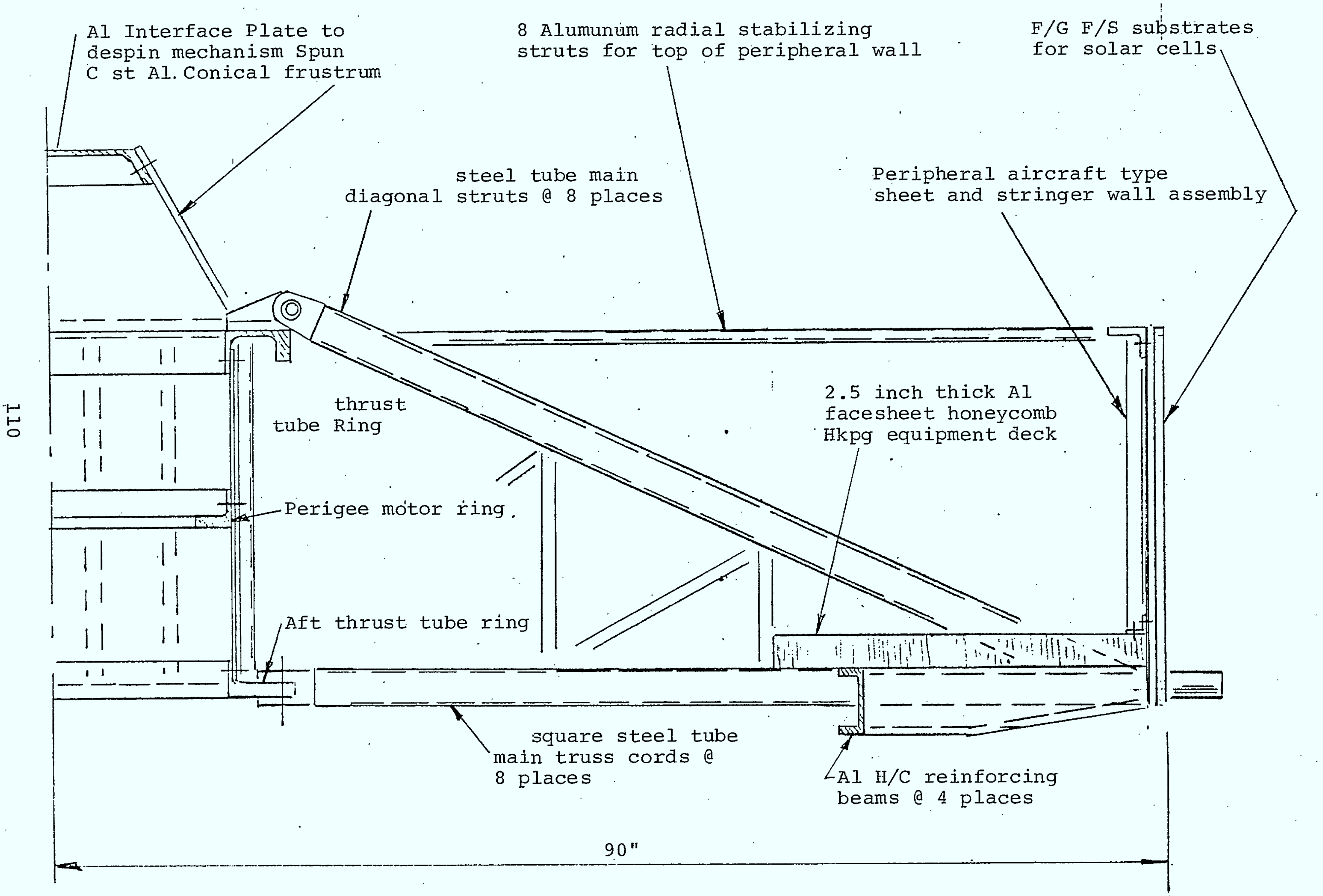


FIGURE 4-3-6 STRUCTURAL ELEMENTS, SPINNING SECTIONS

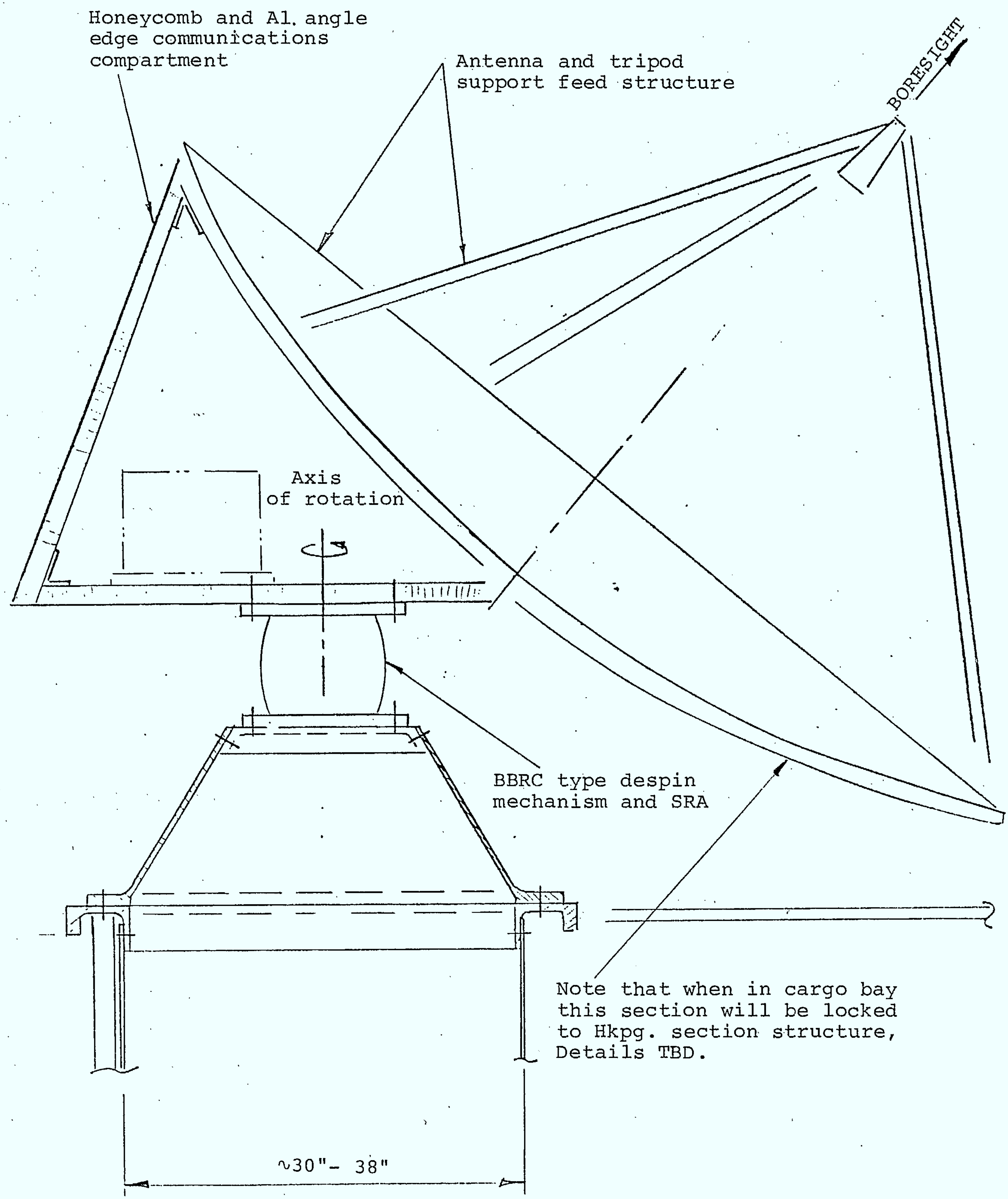


FIGURE 4-3-7 STRUCTURE, DESPUN SECTION

- The housekeeping equipment is mounted on a 2.5 inch thick aluminum honeycomb platform which is supported in the X direction by the lower surfaces of the main truss and four in-plane reinforcing beams. The outer periphery is stiffened. The deck is a thermal control surface as discussed in Section 4.3.7.

- Eight solar array substrates are supported at the perimeter of the spacecraft. At the aft end, they are supported by the main honeycomb platform at their forward end they are supported by the ring which forms the top edge of peripheral wall.

The solar array panels are not part of the main structure load path.

- In order to stiffen the perimeter of the spinning section deck and also provide backside radiation shielding for the solar cells, a peripheral wall is used with thin aluminum sheet and angle section stiffeners. It is supported radially by eight struts.

- The communication equipment is mounted on a separate honeycomb deck, beneath the parabolic antenna.

- The communications platform is prevented from rotation during launch by two locks which are subsequently opened by either pyrotechnics or solenoid actuation.

- o The BBRC type despin and slipring assembly is supported from the main thrust tube by a conical support structure.

- o Conventional materials fabrication and assembly techniques are used to minimize cost.

Payload structural dynamic design criteria have yet to evolve for STS, but the imposed environments are not in general severe. For Polarsat, the following guidelines were used:

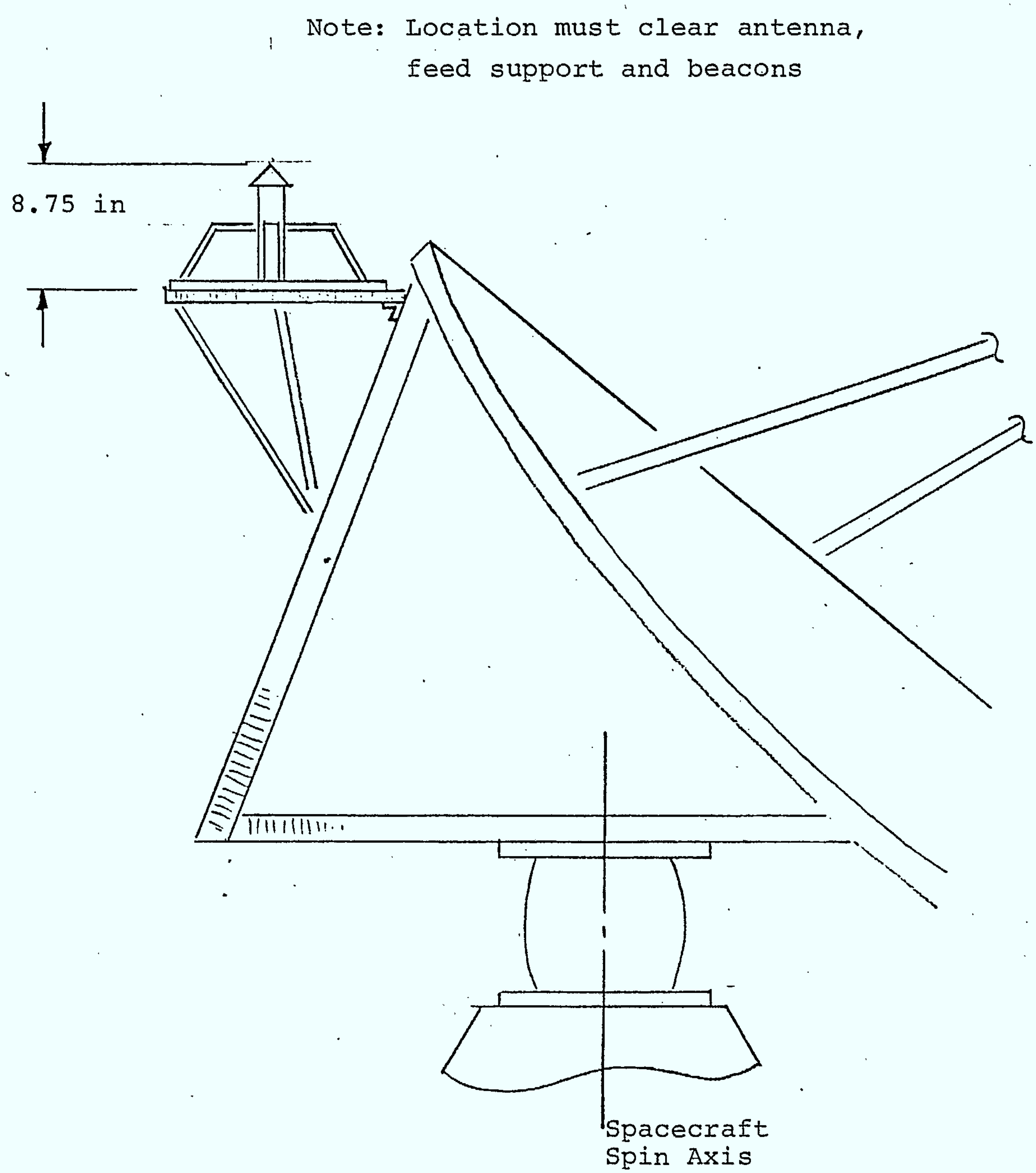
- o All primary structural modes, when the unit is stowed in the cargo bay, are to be above 15 Hz.

- o For the spacecraft configuration at perigee-motor firing, all primary structure modes are to be above 15 Hz.

- o Primary structure main modes are to be separated i.e., modes of the perigee motor/communications platform and housekeeping deck are to be separated .

- o Peripheral solar panel substrates have panel modes significantly higher than the primary structural modes. The panels are not rigidly coupled to the main body of the spacecraft.

- o Due to its size the antenna modes will be low and likely will be excited by the main structural modes. The antenna will be designed with as low a mass as possible and will have the strength to survive the resulting high acceleration responses. The antenna dish will likely be composite material for minimized weight and thermal distortion.



LOCATION FOR SRMS END EFFECTOR PICKUP POINT

FIGURE 4-3-8

The orbit is described in Section 2. Figure 4-3-9 illustrates some of the implications of the orbit on the thermal design. Since the spin axis is aligned with the earth N-S axis the sun vector will generally be directed normal to the spinning axis with a seasonal ± 23.5 degree variation. This means that both the forward and aft decks will be exposed to some shallow angle solar flux.

Figure 4-3-10 and 4-3-11 show the thermal subsystem conceptual design. For on-orbit operations the principals of the design are as follows:

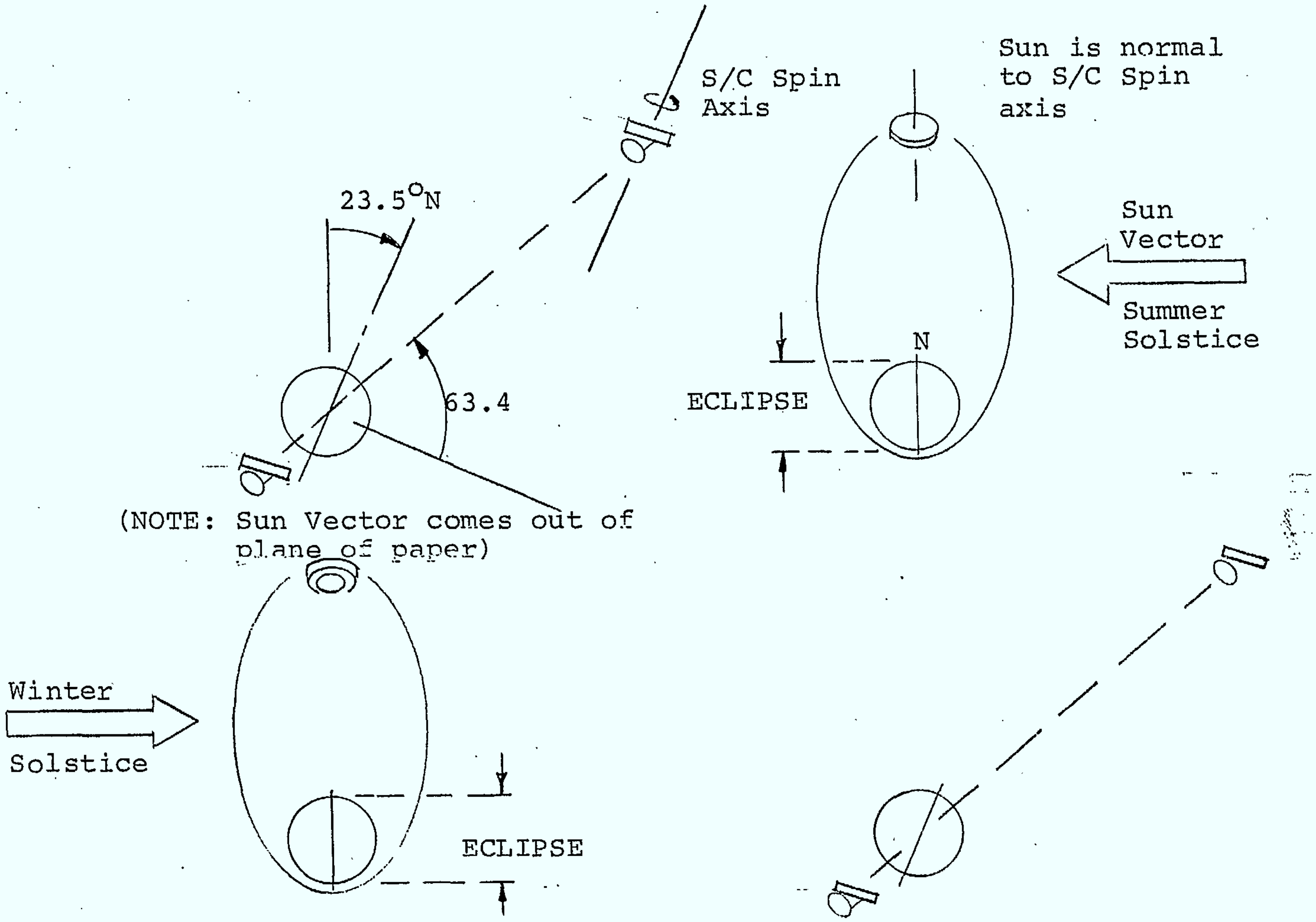
a) Thermal control is essentially passive, i.e., paints, materials, finishes and correct conductive radiative and conductive coupling.

b) Heaters will be available to prevent hydrazine freezing and to protect equipment which may be susceptible to low temperature failure. The hydrazine components shall never go below $5^{\circ}C$.

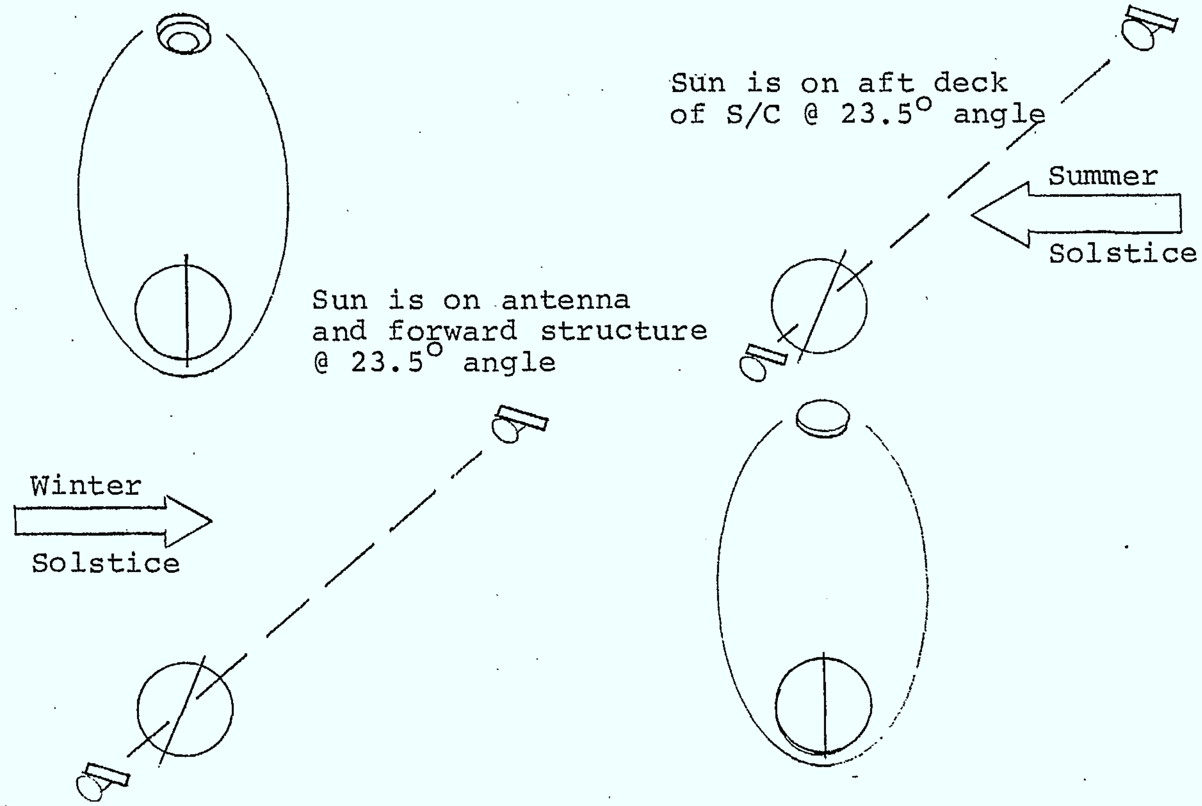
c) Temperature sensors will be on board at critical locations.

d) Heaters will be switched on/off thermostatically. Override commands will be available from the ground station based on sensor data.

e) The solar array substrate will nominally have no backside multi-layer insulation but will rely on low ε finish in the interior surface to prevent excess heat loss during eclipse. Insulation may be required when the equipment layout and thermal performance characteristics are developed during the project PDP Phase. Substrate structural connections will be high thermal resistance paths.



CASE I: SUN ALWAYS NORMAL TO S/C SPIN AXIS



CASE II: SUN @ 23.5° ANGLE TO S/C SPIN AXIS

TWO LIMITING CASES OF ON ORBIT THERMAL CONDITIONS FIGURE 4-3-9

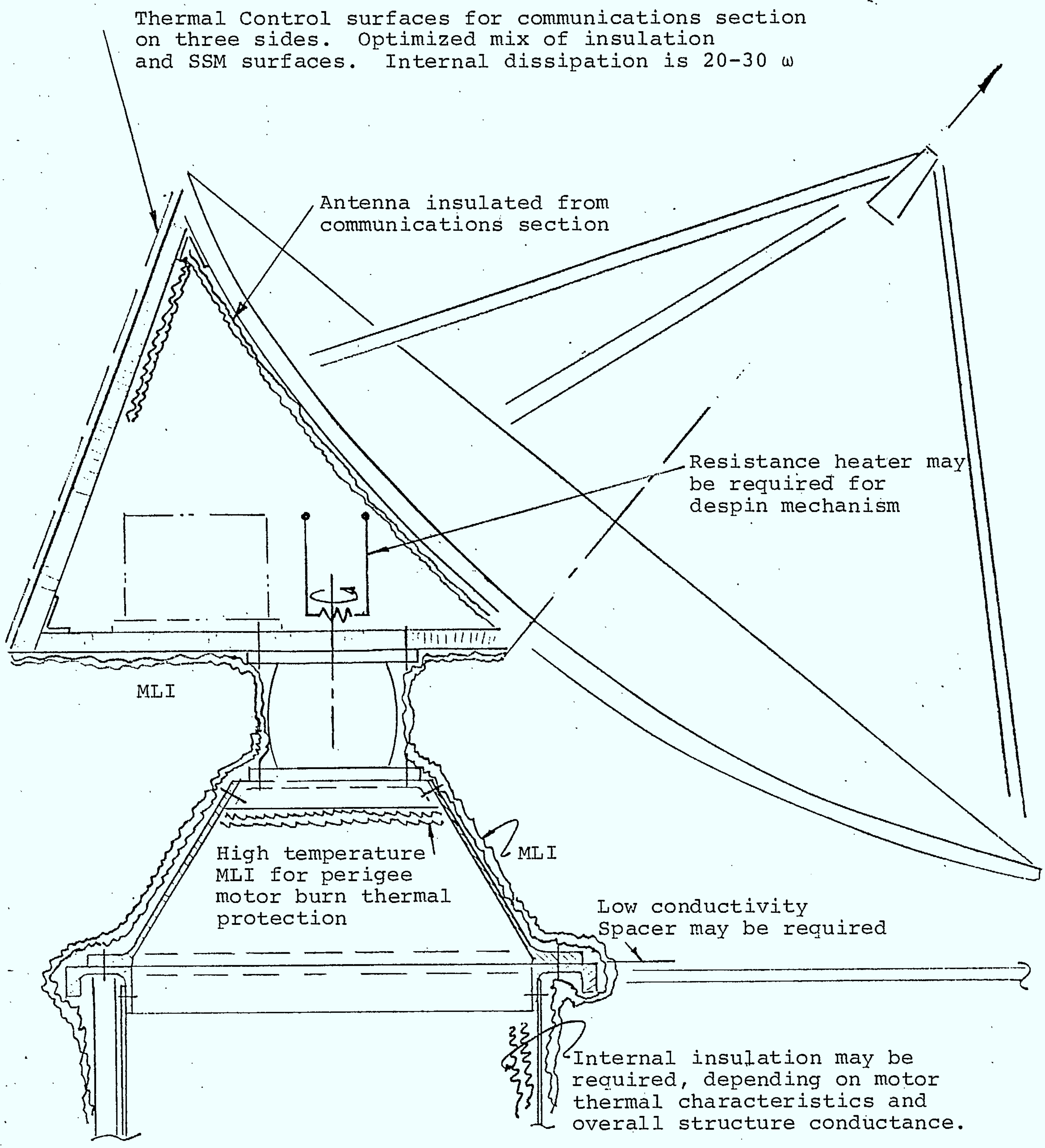
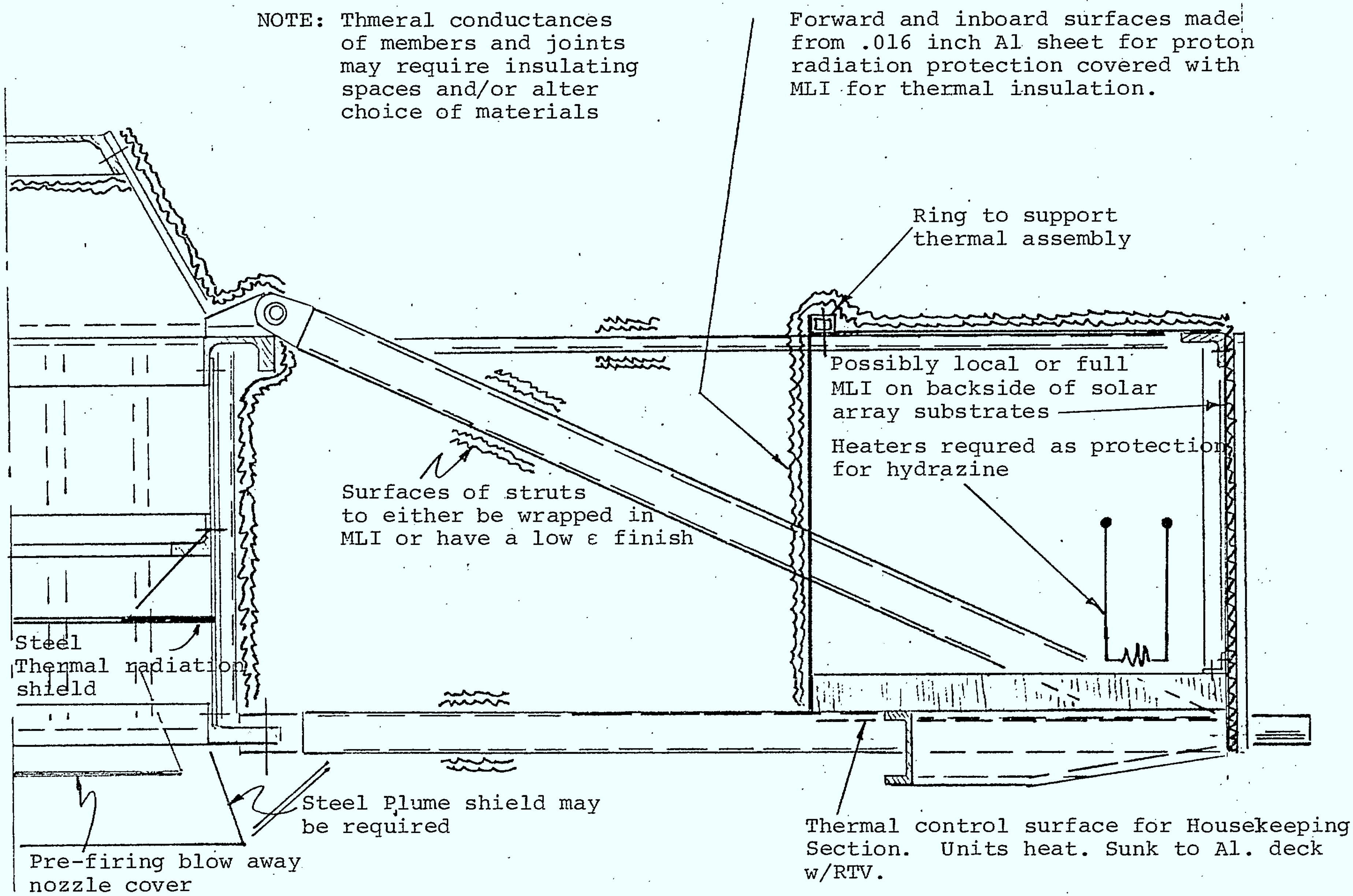


FIGURE 4-3-10

DESPUN SECTION THERMAL DESIGN



SPINNING SECTION THERMAL DESIGN - FIGURE 4-3-11

f) Because the spacecraft is spinning, each cell receives approximately $\frac{432}{\pi}$ BTU/hr.ft^2 average steady state solar radiation when the sun vector is normal to the spin axis. In addition, the surface receives heat from albedo. Earth thermal emittance also provides significant radiation to the spacecraft. During perigee and near earth altitudes this is pedominantly an irradiation to the aft honeycomb deck and during high altitudes and apogee it is an irradiation to the antenna, and forward structure. Earth irradiation is taken at 70 BTU/hr.ft^2 and albedo is considered 30%.

g) Since the spacecraft spends the majority of its time per orbit at the higher altitudes, the aft honeycomb deck has been chosen as the thermal control surface for the housekeeping units. All units are conductively mounted to it. The dissipation is estimated to vary between 50 and 200 watts. Its exterior finish is a combination of MLI and high and low emissivity finishes to give an optimum average α/ε and effective area for the total orbit and for the year. All other parts of the house-keeping enclosure are adiabatic boundaries. The units and the interior surfaces of the eeck have a high emissivity (black) finish to reduce thermal gradients.

h) The antenna is subject to large temperature gradients because of the angular variation of the solar vector over the year. It must be thermally and structurally designed to minimize the resultant distortions.

i) The despun communications deck and antenna are thermally uncoupled from the spinning housekeeping structure. The equipment dissipation is estimated to vary between 10 and 90 watts. The equipment is mounted on the flat honeycomb panel which is normal to the spin axis. The side panels act as thermal control surfaces and their exterior surface is an optimized combination of insulation and second surface mirrors to accommodate the seasonal variation of the sun vector.

j) The BBRC despin mechanism will be maintained above $0^{o}C$ and below $25^{o}C$. Heaters may be required to provide this uniform temperature.

h) In general, the spacecraft units shall be maintained between $-5^{o}C$ and $+35^{o}C$.

A critical mission phase for the thermal design occurs when the cargo bay doors are open and when the SRMS is articulating the spacecraft, at these times the thermal control will depend on:

a) The use of solar cells and fabrication techniques which can withstand high temperatures, on the order of $100^{o}C$ for limited periods of time without significant permanent damage.

b) The ability of the equipment which is mostly non-operating at this point, to withstand temperature excursions in excess of their operating range values.

This is a significant area of concern, not just for Polarsat but for all STS spacecraft. NASA is responsible for maneuvering the orbiter to achieve attitudes such that all the on-board spacecraft payloads remain within their thermal design constraints. Obviously, this will be an important interagency interface task. One possible design solution is to provide retractable sun shield to protect the spacecraft when its in the cargo bay.

The perigee motor affects the thermal design in several ways:

a) Prior to firing, it must be maintained within its design temperatures.

b) During firing, the intense heat from the motor can not be allowed to affect the other spacecraft components.

c) After firing, the exposed casing must not act as a heat loss path from the spacecraft interior.

The design soluations are:

a) The nozzle is provided with a blow-away thermal cover.

b) Insulation is provided around the thrust tube to prevent the burning motor from overheating the rest of the spacecraft.

c) The conduction path between the motor and the aft deck is carefully controlled by the proper choice of materials.

4.3.8 Perigee Motor

The Polarsat mission requires that a significant velocity addition be imparted to the spacecraft after release from the shuttle. A solid perigee kick motor has been selected for this task. This motor is similar in every way to the apogee kick motors used to circularize geosynchronous communications satellites in their orbit. The Polarsat perigee kick motor is mounted axially in the spinning section of the spacecraft with the nominal thrust vector aligned along the spin axis.

4.3.9 TT&C Subsystem

The telemetry tracking and command subsystem is part of the communications subsystem. This is a conventional approach used on several operational satellite systems. In this scheme, the command signals are a low-data-rate signal transmitted at the edge of the normal communications band. On the spacecraft, the command signal is detected and decoded. If the signal is encrypted then it must be decrypted by further signal processing prior to implementation.

Telemetry data is handled by introducing a low data rate signal on the downlink carrier.

Tables 4.3.4 and 4.3.5 list the command and telemetry functions that must be provided by the TT&C for Polarsat.

Range and range rate data is available by transmitting ranging tones through the SHF-SHF transponder, which is part of the communications payload of Polarsat.

- Perigee Motor Arm
- Perigee Motor Fire
- RCS Thruster Firing Commands
- Redundant Payload Switching
- Battery charge controls
- Despin System Bias

POLARSAT COMMAND FUNCTION LIST

TABLE 4.3.4

- Array Voltage
- Array Current
- Battery A - Voltage and current
- Battery B - Voltage and current
- Battery temperature
- Array temperature
- Rotation Rate
- Attitude sensor data
- Reaction control system tank pressure
- Command decoder contents
- Switch status flags

POLARSAT TLM FUNCTION LIST

TABLE 4.3.5

4.4 Shuttle Interface

Figure 4-4-1 shows the overall STS orbiter interfaces. Some of the general interface conditions are:

- o For moderate sized spacecraft, minimizing the overall length is more cost effective than minimizing the weight because of the NASA cost formula.

- o Structural attachment points between the spacecraft and the orbiter are located at the periphery of the 180 inch payload envelope, so it makes sense to locate as much mass as possible on the spacecraft perimeter to improve the structural dynamic behaviour.

- o A multitude of keel and longeron attach points may be used, each having the load reacting capabilities shown in Figure 4-4-2. The support system can be made either statically determinant or indeterminant.

- o A spacecraft can either interface directly to the orbiter or use an intermediate cradle.

- o The SRMS can assist in some payload preparation tasks. See Figure 4-4-3.

- o 28V D.C power is available prior to spacecraft ejection.

- o Multiple spacecraft on board an orbiter provide mission unique thermal and structural dynamic constraints.

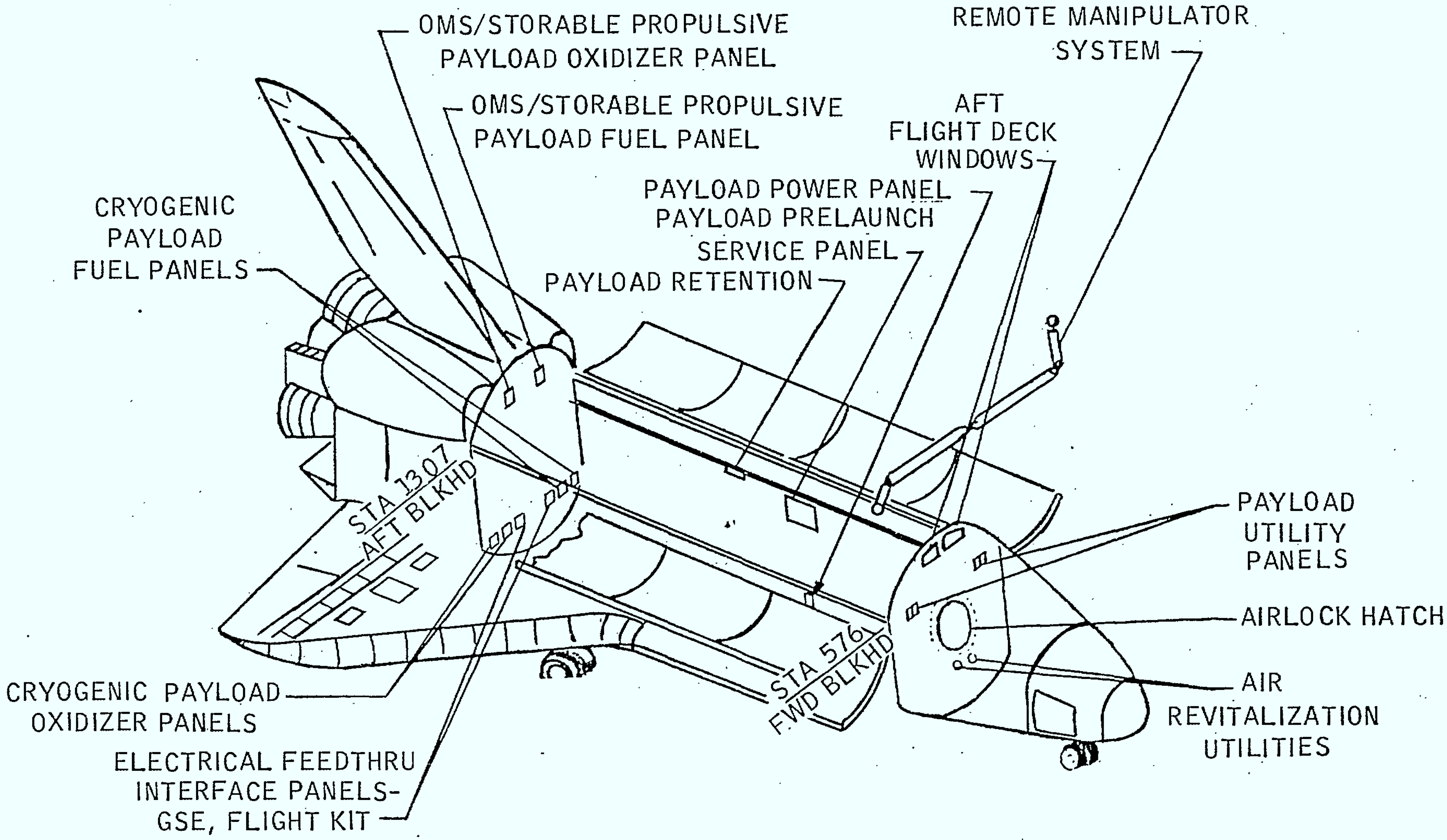
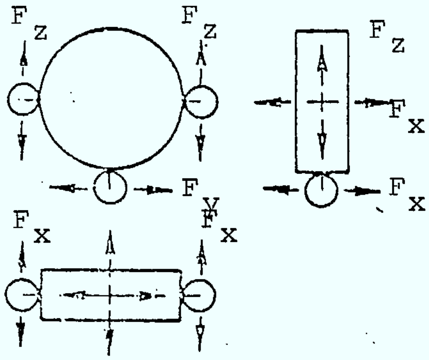
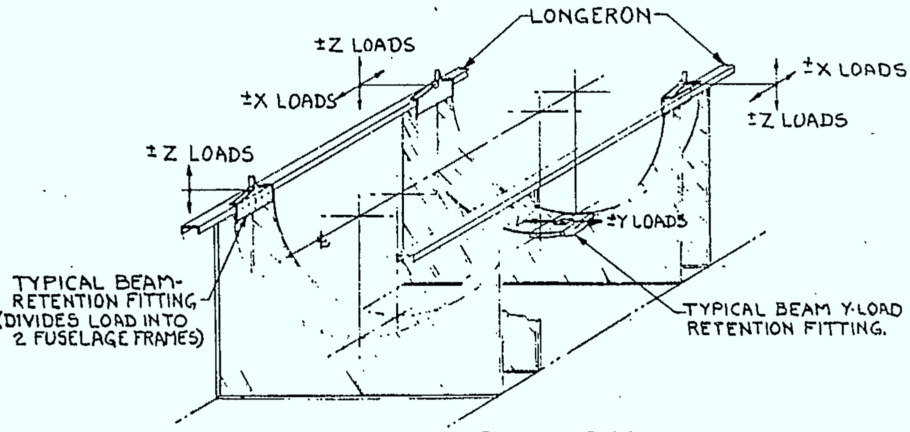


Figure 4-4-1 Payload/Orbiter interfaces

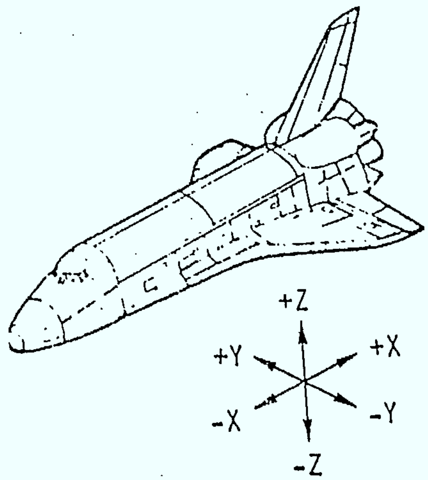


3-POINT (IN-PLANE) DETERMINATE

(2 LONGERON-SILL FITTINGS FOR $\pm X$ & $\pm Z$ LOADS)
(1 KEEL FITTING FOR $\pm Y$ & $\pm X$ LOADS)



BASELINE PAYLOAD RETENTION SYSTEM.



Coordinate System

PAYLOAD KEEL FITTING

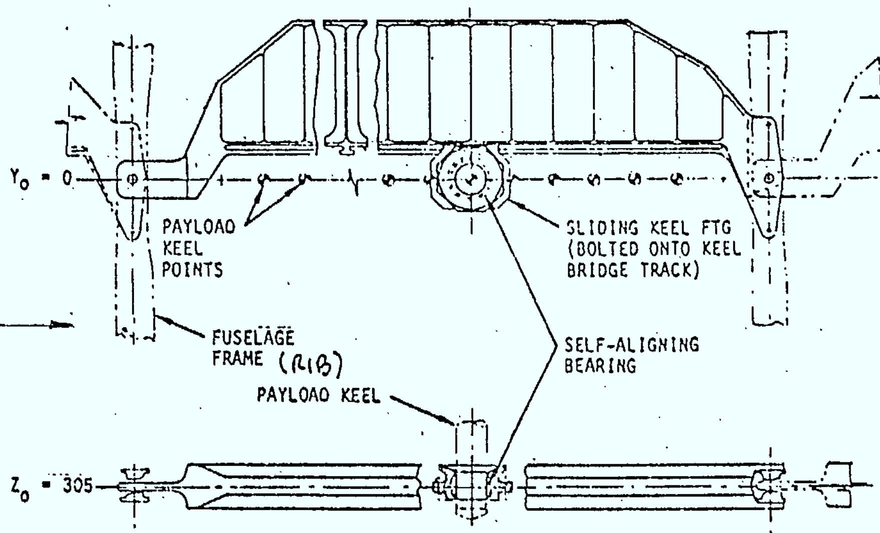
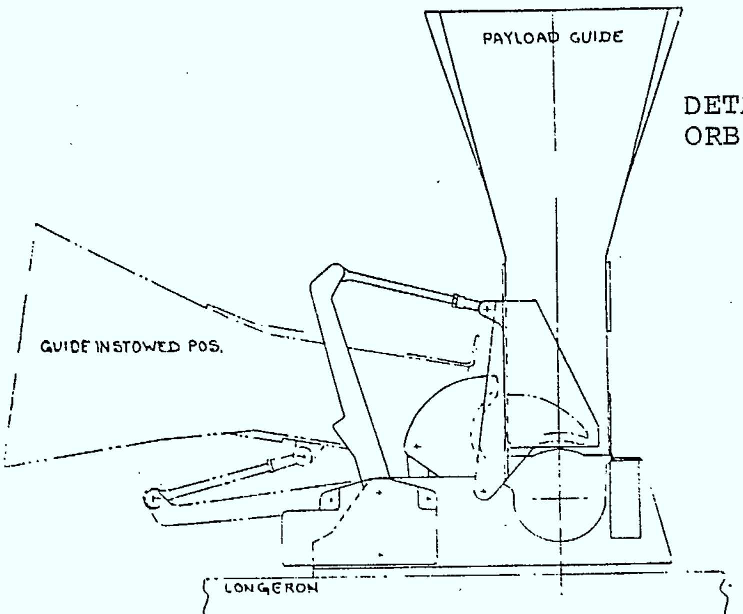
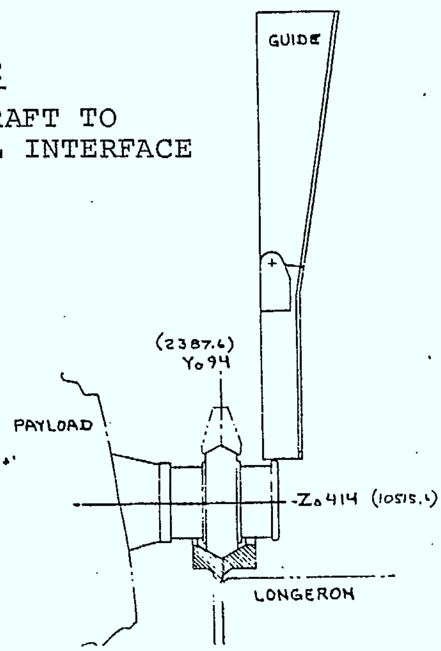


FIGURE 4-4-2

DETAILS OF SPACECRAFT TO ORBITOR MECHANICAL INTERFACE



PAYLOAD LONGERON FITTING



The selection baseline orbiter interface for Polarsat is:

- o The overall stowed length of Polarsat is 8.5 feet and the full 15 foot payload envelope diameter is utilized. Due to its relatively small size it can be located practically anywhere within the cargo bay which is accessible to the SRMS.

- o Polarsat interfaces directly to the cargo bay using STS deployable spacecraft type longeron clamps. The spacecraft is equipped with three steel trunnion pins, approximately 3 in. dia. and 6 in.in length to form the attachment. A special bridge piece is required in the cargo bay so that X direction loads can be reacted at the keel trunnion as well as the longeron trunnions.

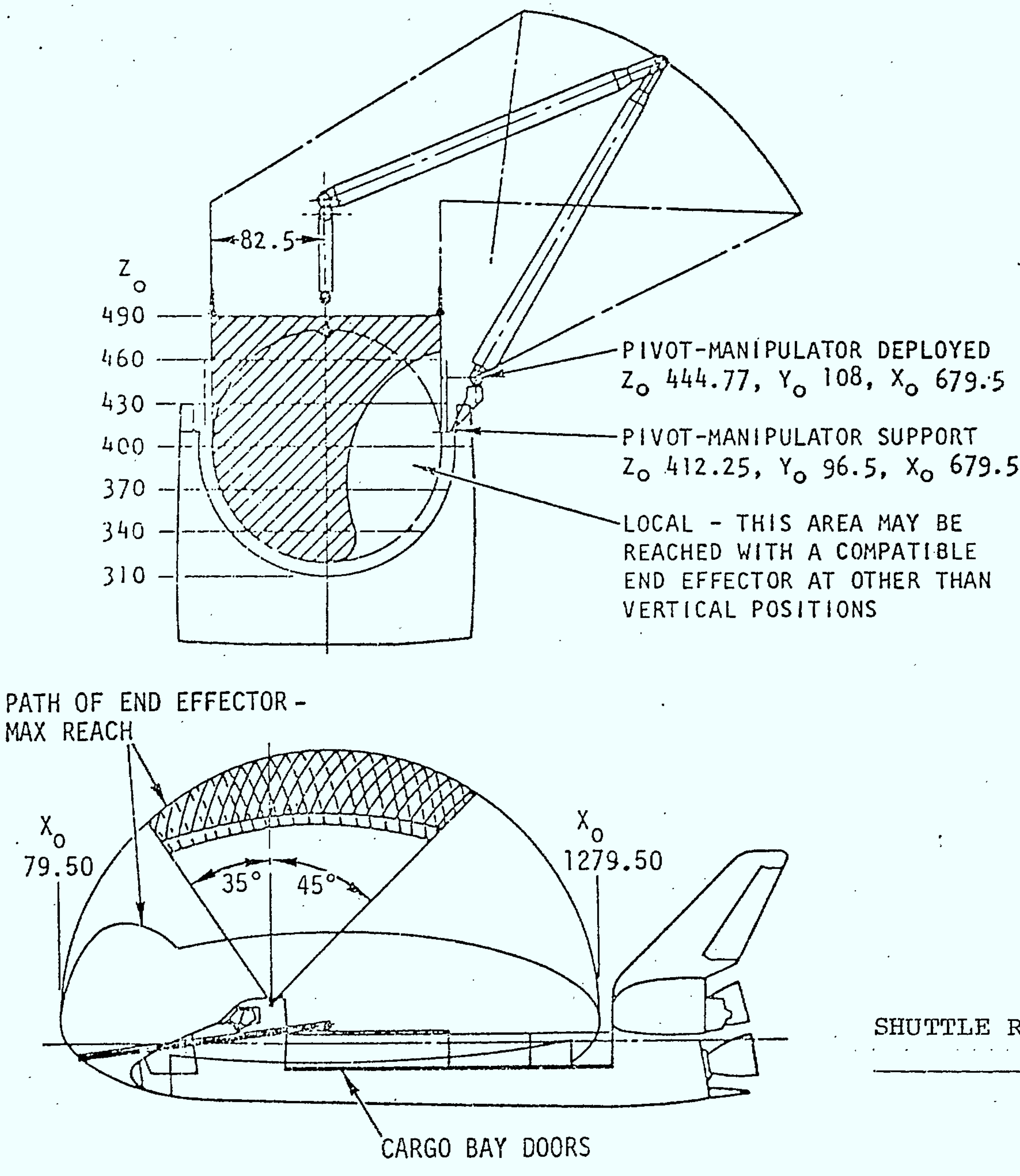
- o The SRMS is used to deploy the spacecraft. An end effector pick-up fitting is attached to the despun section of the stowed spacecraft. Initial spin up of the spacecraft is performed using the despin motor while still attached to the SRMS.

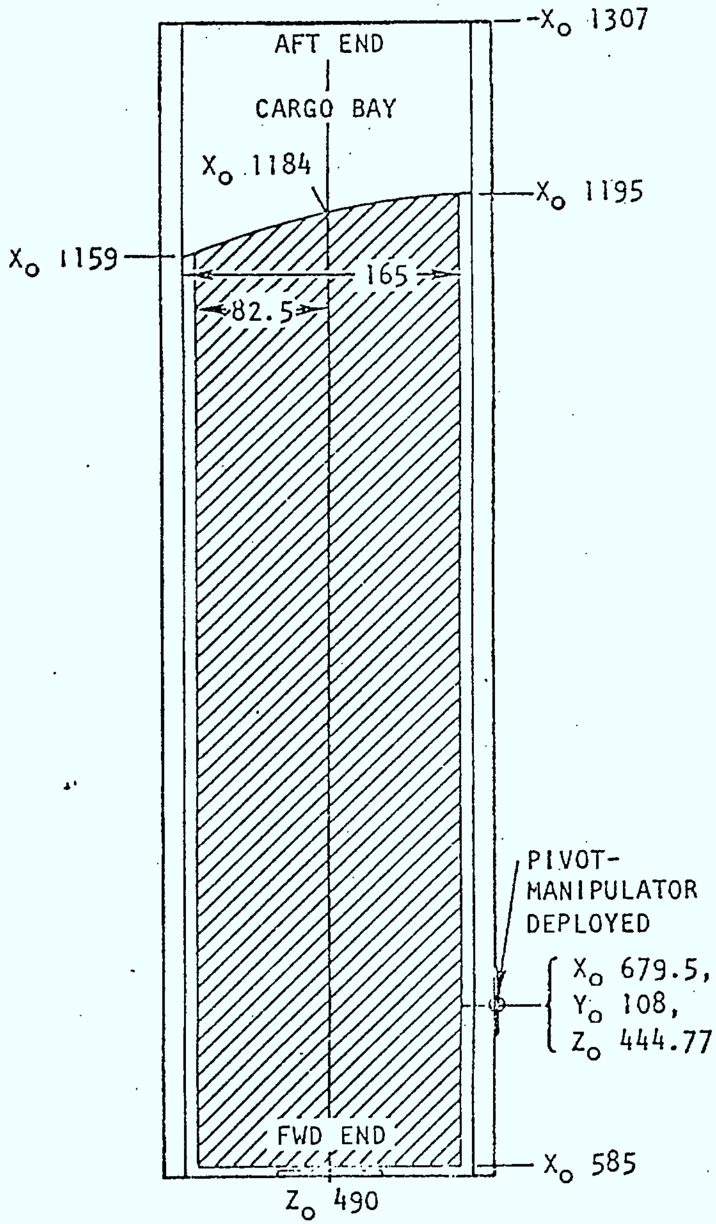
- o Spin-up to 30 rpm is done by the RCS tangential thrusters, after the SRMS has released the spacecraft and the orbiter has moved away to a safe distance.

- o During launch, the communications deck is locked to the spacecraft main body. These locks are released prior to the release of the spacecraft from the end effector.

- o There is a pull-away umbilical which is used to transmit spacecraft state of health information to the crew mission specialist, prior to spacecraft release. This umbilical circuit is broken when the spacecraft leaves the cargo bay. Spacecraft telemetry is used for data from this point in the mission.

- Polarsat will share a shuttle launch with other spacecraft. When the cargo bay doors are open it is an STS responsibility to roll the orbiter or perform other maneuvers so that the exposed spacecraft does not undergo unacceptable thermal conditions. This is a particularly important procedure since the exposed perimeter of Polarsat is a solar array substrate which is designed for a solar flux determined for in-orbit thermal conditions. Moreover, the array has an adiabatically designed substrate, so heat cannot be rejected from this backside of the array. The thermal design must accommodate worst-case estimates for the cargo bay wall temperature of $-45^{o}C$ to $+93^{o}C$, which are based on conditions with the shuttle doors open, and an assumed adiabatic space structure.





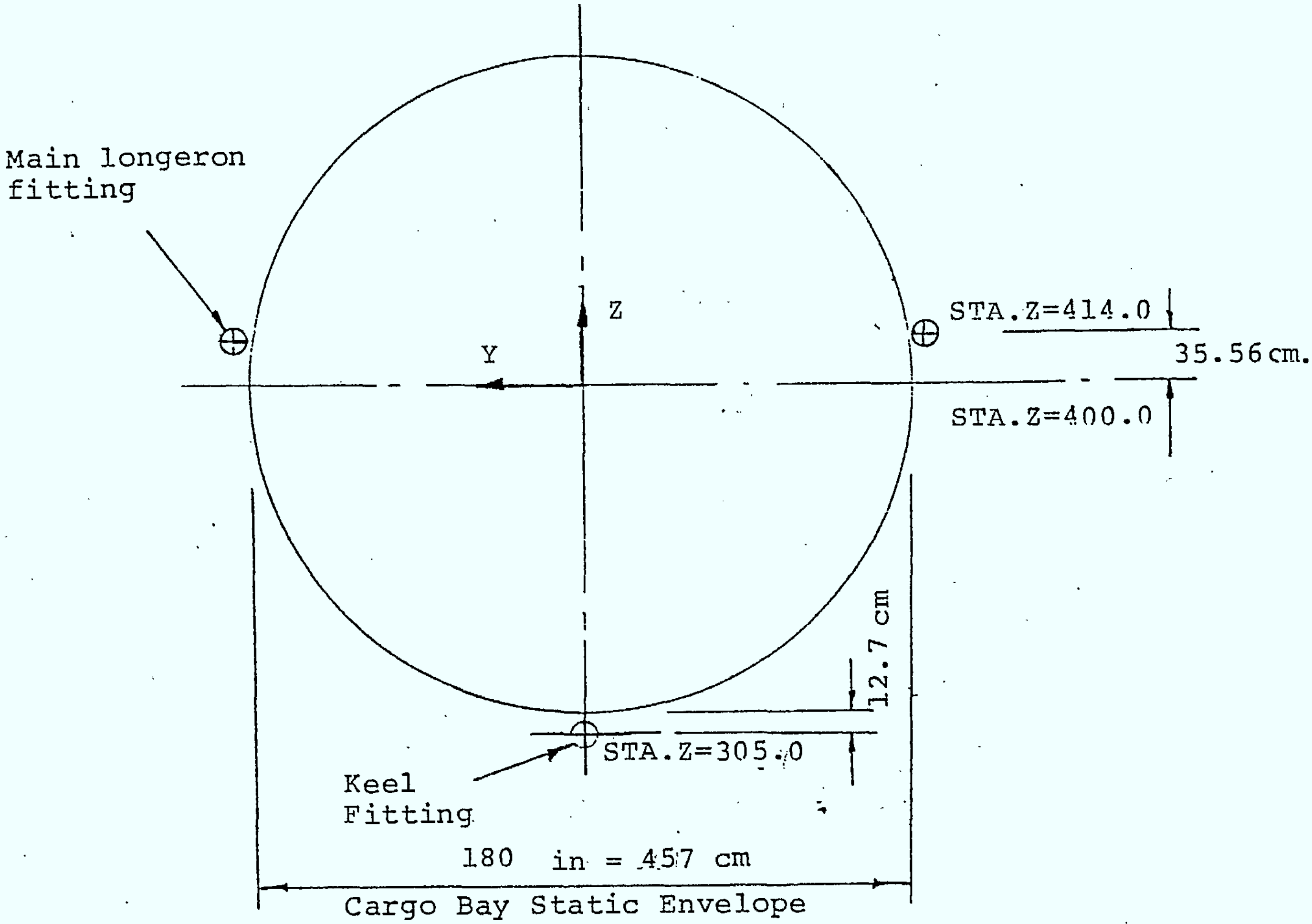
SHUTTLE REMOTE MANIPULATOR SYSTEM (SRMS)
REACH CAPABILITIES

FIGURE 4-4-3

Figures 4-4-4 and 4-4-5 show the cargo bay dimensions and interface environments. For vibration, shock and acoustic, the dynamic response to the environment must be considered in design. NASA, as part of its launch support services, will perform overall multi-spacecraft cargo bay dynamics analysis but this usually occurs after the spacecraft has been designed. Two additional design environments of note are:

- o 1/4 g sine sweep from 5 to 35 Hz to simulate transient events

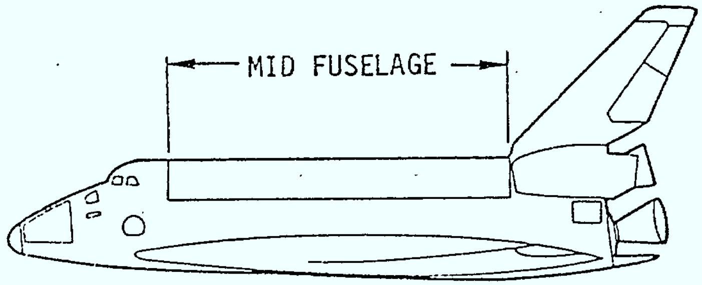
- o Basic component design shock of 20 g, 11 ms terminal peak sawtooth.



NOMINAL FLIGHT LOADS	CRASH SAFETY LOADS
$F_X = 3.3$ g's	$F_X = \pm 9.0$ g's
$F_Y = 1.0$ g	$F_Y = \pm 1.5$ g's
$F_Z = 1.5$ g's	$F_Z = {+4.5 \atop -2.0}$ g's

CARGO BAY ENVELOPE AND STATIC LOAD REQUIREMENTS

FIGURE 4-4-4

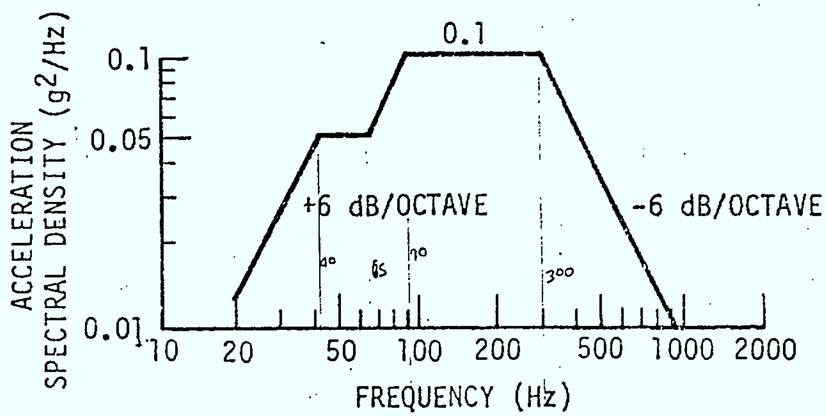


ORBITER PAYLOAD BAY INTERNAL ACOUSTIC SPECTRUM DESIGN SPEC

OCTAVE BAND CENTER FREQ (Hz)	SOUND PRESSURE LEVEL (dB)
31.5	124.5
63	130.5
125	137.0
250	140.0
500	139.5
1000	134.0
2000	128.0
4000	122.0

MAX SOUND PRESSURE WITHOUT PAYLOAD ≤145 dB OVERALL

PREDICTED RANDOM VIBRATIONS AT PAYLOAD/MID FUSELAGE INTERFACE



THIS SPECTRUM REPRESENTS MAXIMUM OF LIFTOFF, MACH 1, & MAX Q FLIGHT & DOES NOT REFLECT VARIATIONS IN PAYLOAD WEIGHT, SUPPORT STIFFNESS, & C.G. THE ACTUAL VIBRATION INPUT TO PAYLOADS WILL DEPEND ON TRANSMISSION CHARACTERISTICS OF MID FUSELAGE - PAYLOAD SUPPORT STRUCTURE & INTERACTIONS WITH EACH PAYLOAD'S WEIGHT, STIFFNESS, & C.G.

ACOUSTIC AND RANDOM VIBRATION LOADS

FIGURE 4-4-5

4.5 Equipment List and Mass Properties

4.5.1 Equipment List

The equipment compliment for a Polarsat spacecraft is given in Table 4.5.1. This list is broken down by subsystem, and contains a mass and volume estimate for each of the units identified. The total spacecraft mass is estimated to be 1058 kg of which 646 kg is the perigee motor. Approximately 25% of the on-station mass is payload.

4.6 Power Budget

The DC power required by the various elements of the payload is summarized in Table 4-6-1. It can be seen that considerable growth capability is included in this baseline design.

TABLE 4.5.1

POLARSAT EQUIPMENT LIST

SUBSYSTEM	ITEM	L cm	W cm	H cm	UNIT MASS kg	POWER WATTS	# ON S/C	TOTAL MASS PER S/C kg	
ATTITUDE CONTROL	Sun Pippers	4	4	4	1.7	2.5	1	1.7	
	Rotation Sensor	10	4	4	0.5	1	1	0.5	
	Horizon Sensors	15	6	6	.15	5	4	.6	
						8.5		2.8	PAGE TOTAL

TABLE 4.5.1

POLARSAT EQUIPMENT LIST

SUBSYSTEM	ITEM	L cm	W cm	H cm	UNIT MASS kg	POWER WATTS	# ON S/C	TOTAL MASS PER S/C kg
Antennas	UHF Reflector & Feed	380	220	-	23		1	23
	SHF Horn	30	9	25	2		2	2
	Omni Patches	15	15	2	0.5	-	4	2
	SHF Antenna Switch	10	5	2	0.5	-	1	0.5
								27.5 PAGE TOTAL

TABLE 4.5.1

POLARSAT EQUIPMENT LIST

SUBSYSTEM	ITEM	L cm	W cm	H cm	UNIT MASS kg	POWER WATTS	# ON S/C	TOTAL MASS PER S/C kg
Auxillary Propulsion	Hydrazine Thrusters	–	–	–	3	23.5	2	6
	Hydrazine Tanks & Fuel				5.7		4	23
	RCS Controller	15	15	15	4	2	1	4
	Perigee Motor				646		1	646
						25.5		679 PAGE TOTAL

TABLE 4.5.1

POLARSAT EQUIPMENT LIST

SUBSYSTEM	ITEM	L cm	W cm	H cm	UNIT MASS kg	POWER WATTS	# ON S/C	TOTAL MASS PER S/C kg
POWER	Power Switching Unit	20	15	5	1.5	2	2	3
	Battery	30	6	8.2	3.5		4	14
	Solar Cells &Coverglass	-	-	-	-	-	-	20
	Charge Control	15	5	3	.2	-	2	.4
	Shunt Regulator	25	15	5	1	1	2	2
	Converters	20	7	3	1	2	4	4
	Bty. Dis. Reg.	15	5	3	.1	-	8	.8
PAGE TOTAL						5		44.2

TABLE 4.5.1

POLARSAT EQUIPMENT LIST

SUBSYSTEM	ITEM	L cm	W cm	H cm	UNIT MASS kg	POWER WATTS	# ON S/C	TOTAL MASS PER S/C kg	
TT&C	TLM Encoder	15	10	8	2	3	1	2	
	CMD Decoder	15	10	8	2	3	1	2	
	Data/CMD Interface	15	10	8	2	3	2	4	
	Encrypter Decrypter	20	10	8	2	5	1	2	
						14		10	PAGE TOTAL

TABLE 4-5-1

POLARSAT EQUIPMENT LIST

SUBSYSTEM	ITEM	L cm	W cm	H cm	UNIT MASS kg	POWER WATTS	# ON S/C	TOTAL MASS PER S/C kg
STRUCTURE	SPINNING SECTION							
	Main Truss Assy.							45
	Shuttle Fittings						3	9
	H/C Platform & Beam Stiffeners							54
	Peripheral Ring Assy							20
	Thrust tube & rings							18
	Conical Adaptor							9
	BBRC Despin mechanism, SRA							18
	Brg. Motor Ctl.							2
	Misc. Structure							5
	Solar Panel substrates							22
	DESPUN SECTION							
	H/C communications area							15
	Thermal							9
								226 PAGE TOTAL

TABLE 4.5.1

POLARSAT EQUIPMENT LIST

SUBSYSTEM	ITEM	L cm	W cm	H cm	UNIT MASS kg	POWER WATTS	# ON S/C	TOTAL MASS PER S/C kg
PAYLOAD	UHF Diplexer	30	20	10	10	–	1	10
	SHF Diplexer	10	8	5	1	–	1	1
	UHF Filters	25	15	4	0.5	–	5	2.5
	SHF Filters	20	7	5	0.2	–	8	1.6
	IF Filters	6.5	2.5	2.5	0.1	–	6	0.6
	UHF LNA	5	5	2	0.4	0.5	4	1.6
	SHF LNA	5	5	2	0.5	0.5	4	2.0
	Mixer/L.O.	10	8	5	0.7	0.5	15	10.3
	FM Demod.	13	8	5	1.0	0.5	2	2.0
	FM Mod.	13	8	5	1.0	0.5	2	2.0
	UHF AGC/Channel	6.5	6	2.5	0.1	1	5	0.5
PAYLOAD (CONT)	OUTPUT AMP							
	UHF	20	20	8	5	50	3	15
	UHF hopped	20	20	8	5	50	2	10
	SHF	30	7	3	0.63	11	2	1.3
	SHF (FH)	10	5	2	0.25	4	2	0.5
	IF AMP	6	3	2	0.1	1	4	0.4
	SHF Waveguide	–	–	–	–	–	1	2
	UHF Coax	–	–	–	–	–	1	1.5
	Switching Control Box	15	10	8	2	1	1	2
	UHF Isolators	5	5	5	0.2	–	5	1.0
	SHF Isolators	2	2	2	0.1	–	4	0.4
							PAGE TOTAL	68.2

Transponder	131.5
Housekeeping	
ACS	8.5
RCS	25.5
Power	5
TT&C	14
Despin Brg.	7
TOTAL EOL	191.5
ARRAY DEGRADES TO 60% in 5 years	
TOTAL BOL	319.2
Array Area Capability	400
Margin	80.8

POLARSAT POWER BUDGET

TABLE 4.6.1

5.0 CONCLUSION AND RECOMMENDATION

5.1 Conclusion

This report presents a conceptual design for a polar-orbing communications satellite system which exceeds the baseline requirement of augmenting a geosynchronous UHF communications satellite system UHF for coverage of the Canadian Arctic. Indeed, the Polarsat system can operate independently of the full system, and can easily be expanded to provide at least 30% of the full geosynchronous system communications capacity. The system is capable of 24 hour all-Canada coverage by employing additional satellites.

The conceptual design presented consists of three spacecraft in elliptical orbits of a 12 hour period with a perigee height of 750 km and an apogee height of 20232 km. A single communications and satellite control station is part of the system. This station may be located as far south as Cold Lake Alberta.

The spacecraft conceptual design is optimized for a shuttle launch vehicle, and is thus designed to minimize hardware and assembly and test costs. The large antenna needed for the UHF links is not required to be a deployable structure because of the larger payload diameter permitted by the shuttle bay dimensions. This feature of shuttle design will reduce the system cost.

5.2 Recommendation

The capability of the conceptual design is sufficiently promising to warrant a detailed engineering design phase for Polarsat. This phase would identify specific hardware and result in detailed engineering drawings from which more accurate cost projections can be made.

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6. Overview of the Multipurpose UHF Satellite System.

7. NASA JSC 07700, Vol. XIV; Space Shuttle System Payload Accommodations.

8. Rockwell International; SD78-SR0002; Physical, Dynamic, Thermal, Electrical and Data Processing Characteristics of the Space Shuttle Cargo Bay.

APPENDIX A

LINK BUDGETS

LINK	TABLE
1) Uplink from Team Pack Station for one 16 Kbps Channel	A-2
2) Uplink from Shipborne Station for one 16 Kbps Channel	A-3
3) Uplink from Airborne Station for one 16 Kbps Channel	A-4
4) Uplink from EIRB for one EPIRB Distress Signal	A-5
5) SHF Downlink for one 16 Kbps Channel	A-6
6) SHF Downlink for one ERIRB Distress Signal	A-7
7) UHF Downlink to Shipborne Station for one Frequency-hopped 16 Kbps Channel	A-8
8) UHF Downlink to Small Airborne Station for one 16 Kbps Channel	A-9
9) SHF Uplink for one 2 Mbps Channel	A-10
10) SHF Downlink for one 2 Mbps Channel	A-11
11) SHF Uplink for Command Channel	A-12
12) SHF Downlink for Telemetry Channel	A-13
13) UHF Downlink to Team Pack Station for one 16 Kbps Channel	A-14
14) UHF Downlink to Shipborne Station for one 16 Kbps Channel	A-15
15) UHF Downlink to Airborne Station for one 16 Kbps Channel	A-16
16) SHF Uplink for one 16 Kbps Channel	A-17
17) SHF Uplink Carrier for Modulated Uplink	A-18
18) SHF Downlink Carrier for Modulated Downlink	A-19

TABLE A-1

UHF UPLINK FROM TEAM PACK STATION
FOR ONE 16 Kbps CHANNEL

1)	Frequency	370 - 406 MHz
2)	TX EIRP	26 dBW
3)	Path Loss	173.4 - 176.5 dB
4)	Propagation Losses	
	a) Polarization Loss	0.2 dB
	b) Ionospheric Loss	5.0 dB
	c) Multipath Loss	3.0 dB
5)	Pointing Inaccuracy Loss	1.0 dB
6)	Antenna Gain	17 dB edge
7)	Received Signal	-139.6 → -142.7 dBw/Hz
8)	Noise Density (1000^{o}K)	-198.6 dBW/Hz
9)	C/N_o	59 - 55.9 dB-Hz

TABLE A-2

UHF UPLINK FROM SHIPBORNE STATION
FOR ONE 16 Kbps CHANNEL

1) Frequency	370-406 MHz	
2) TX EIRP	31.1 dBW	
3) Path Loss	173.4 - 176.5 dB	
4) Propagation Losses		
a) Polarization	0.2	
b) Ionospheric	5.0	
c) Multipath	10.0	11.9 dB
5) Alignment Loss	1.5	
6) Antenna Gain	17 dB edge	
7) Received Signal	-137.2 → -140.3 dBW/Hz	
8) Noise Density (1000°K)	-198.6 dBW/Hz	
9) C/N_o	61.4 - 58.3 dB-Hz	

TABLE A-3

UHF UPLINK FROM AIRBORNE STATION
FOR ONE 16 Kbps CHANNEL

1) Frequency	370 - 406 MHz	
2) TX EIRP	26.6 dBW	
3) Path Loss	173.4 - 176.5 dB	
4) Propagation Losses		
a) Polarization	-	
b) Ionospheric	5.0 dB	
c) Multipath	6.0 dB	8.1 dB
5) Alignment Loss	1.0 dB	
6) Antenna Gain	17 dB edge	
7) Received Signal	-137.9 → -141.0 dBW/Hz	
8) Noise Density (1000°K)	-198.6 dBW/Hz	
9) C/N_o	60.7 - 57.6 dB/Hz	

TABLE A-4

SHF DOWNLINK FOR ONE 16 Kbps CHANNEL

1)	Frequency	7.275 GHz
2)	EIRP	-1.5 dBW
3)	Path Loss	198.6 - 201.7 dB
4)	Propagation Loss	0.8 dB
5)	Polarization Loss	0.1 dB
6)	Antenna Gain	55 dB (30 ft. dish)
7)	Received Signal Strength	-146.0 → -149.1 dBW/Hz
8)	Noise Density (190 $^{\circ}$K) (G/T = 32.2 dB/ $^{\circ}$K)	-205.8 dBW/Hz
9)	C/N_o	59.8 - 56.7 dB-Hz

TABLE A-5

UHF UPLINK FROM EPIRB FOR
ONE EPIRB DISTRESS SIGNAL

1)	Frequency	406.05 MHz
2)	EPIRB	7 dBW (5 watts max.)
3)	Pattern Loss (Nominal)	3.0 dB
4)	Path Loss	173.4 - 176.5 dB
5)	Propagation Loss	
	a) Polarization	3.0
	b) Ionsopheric	5.0
	c) Multipath	3.0
6)	Alignment Loss	0.0 dB
7)	Antenna Gain	17.0 dB edge
8)	Received Signal	-163.4 → -166.5 dBW/Hz
9)	Noise Density ($1000^{o}K$)	-198.6 dBW/Hz
10)	C/N_o	35.2 - 32.1 dB-Hz

<u>TABLE A-6</u>

SHF DOWNLINK FOR ONE
EPIRB DISTRESS SIGNAL

1)	Frequency	7.275 GHz
2)	EIPR	-9.1 dBW
3)	Path Loss	198.6 - 201.7 dB
4)	Propagation Loss	0.8 dB
5)	Polarization Loss	0.1 dB
6)	Antenna Gain	55 dB (30 ft. dish)
7)	Received Signal Strength	-153.6 → -156.7 dBW/Hz
8)	Noise Density ($190^{o}K$) ($G/T = 32.2$ $dB/^{o}K$)	-205.8 dBW/Hz
9)	C/N_o (0.1 dB degradation to uplink)	52.2 - 49.1 dB-Hz

TABLE A-7

UHF DOWNLINK TO SHIPBORNE STATION FOR ONE FREQUENCY-HOPPED 16 Kbps CHANNEL

1)	Frequency		274-286 MHz
2)	EIRP		25.3 dBW
3)	Path Loss		170.3 - 173.4 dB
4)	Propagation Losses		
	a) Polarization	0.2	
	b) Ionospheric	4.0	11.3 dB
	c) Multipath	10.0	
5)	Alignment	1.0	
6)	Antenna Gain		10 dB
7)	Received Signal Strength		-146.3 - -115.9 dBW-Hz
8)	Noise Density (550 ^{o}K) (G/T = -17.4 dB/ ^{o}K)		-201.2 dBW/Hz
9)	C/N_o		54.9 - 51.8 dB-Hz

TABLE A-8

UHF DOWNLINK TO SMALL AIRBORNE STATION
FOR ONE 16 Kbps CHANNEL

1)	Frequency		274-286 MHz
2)	EIRP		29.2 dBW
3)	Path Loss		170.3 - 173.4 dB
4)	Propagation Loss		
	a) Polarization	-	
	b) Ionospheric	4.0	
	c) Multipath	6.0	7.7 dB
5)	Alignment	1.0	
6)	Antenna Gain		0.0 dBi
7)	Received Signal Strength		-148.8 → -151.9 dBW/Hz
8)	Noise Density (490^{o}K) ($G/T = 26.9$ dB/^{o}K)		-201.7 dBW/Hz
9)	C/N_o		52.9 - 49.8 dB-Hz

TABLE A-9

SHF UPLINK FOR ONE 2 Mbps CHANNEL

1) Frequency	8 GHz
2) EIRP	66.8 dBW
3) Path Loss	199.4 - 202.5 dB
4) Propagation Loss	0.8 dB
5) Polarization Loss	0.1 dB
6) Antenna Gain	17.0 dB edge
7) Received Signal Strength	-116.5 → -119.6 dBW/Hz
8) Noise Density ($1000^{o}K$)	-198.6 dBW/Hz
9) C/N_o	82.1 - 79.0 dB-Hz

TABLE A-10

SHF DOWNLINK FOR ONE 2 Mbps CHANNEL

1)	Frequency	7.275 GHz
2)	EIRP	15.0 dBW
3)	Path Loss	198.6 - 201.7 dB
4)	Propagation Loss	0.8 dB
5)	Polarization Loss	0.1 dB
6)	Antenna Gain	55 dB (30 ft. dish)
7)	Received Signal Strength	-129.5 → -132.6 dBW/Hz
8)	Noise Density (190°K) (G/T = 32.3 dB/°K)	-205.8 dBW/Hz
9)	C/N_o	76.3 - 73.2 dB-Hz

TABLE A-11

SHF UPLINK FOR COMMAND CHANNEL

1) Frequency	8 GHz
2) EIRP	50.9 dBW
3) Path Loss	199.4 - 202.5 dB
4) Propagation Loss	0.8 dB
5) Polarization Loss	0.1 dB
6) Antenna Gain	-10 dB
7) Received Signal Strength	-159.4 → -162.5 dBW/Hz
8) Noise Density (1000°K)	-198.6 dBW/Hz
9) C/N_o ($P_e = 10^{-6}$, fsk 256 bps)	39.2 - 36.1 dB-Hz

TABLE A-12

SHF DOWNLINK FOR TELEMETRY CHANNEL

1)	Frequency	7.275 GHz
2)	EIRP	-22.1 dBW
3)	Path Loss	198.6 - 201.7 dB
4)	Propagation Loss	0.8 dB
5)	Polarization Loss	0.1 dB
6)	Antenna Gain	55 dB (30 ft. dish)
7)	Received Signal Strength	-166.6 → -169.7 dBW/Hz
8)	Noise Density (190°K) (G/T = 32.2 dB/°K)	-205.8 dBW/Hz
9)	C/N_o $P_e = 10^{-6}$, fsk 256 bps)	39.2 - 36.1 dB-Hz

TABLE A-13

UHF DOWNLINK TO TEAM PACK STATION
FOR ONE 16 Kbps CHANNEL

1) Frequency	274-286 MHz
2) EIRP	23.3 dBW
3) Path Loss	170.3 - 173.4 dB
4) Propagation Losses	
a) Polarization	0.2
b) Ionospheric	4.0
c) Multipath	3.0
5) Alighment Loss	1.0
6) Antenna Gain	13 dB
7) Received Signal Strength	-142.2 → -145.3 dBW/Hz
8) Noise Density (646^{o}K) (G/T = -15.1 dB/^{o}K)	-200.5 dBW/Hz
9) C/N_o	58.3 - 55.2 dB-Hz

TABLE A-14

UHF DOWNLINK TO SHIPBORNE STATION
FOR ONE 16 Kbps CHANNEL

1)	Frequency		274-286 MHz
2)	EIRP		23.3 dBW
3)	Path Loss		170.3 - 173.4 dB
4)	Propagation Losses		
	a) Polarization	0.2	
	b) Ionospheric	4.0	11.3 dB
	c) Multipath	10.0	
5)	Alignment		1.0
6)	Antenna Gain		10 dBi
7)	Received Signal Strength		-148.3 → -151.4 dBW/Hz
8)	Noise Density (550°K) (G/T = -17.4 dB/°K)		-201.2 dBW/Hz
9)	$(C/N_o)_D$		52.9 - 49.8 dB-Hz

TABLE A-15

UHF DOWNLINK TO AIRBORNE STATION
FOR ONE 16 Kbps CHANNEL

1)	Frequency		274 - 286 MHz
2)	EIRP		23.3 dBW
3)	Path Loss		170.3 - 173.4 dB
4)	Propagation Losses		
	a) Polarization	-	7.7 dB
	b) Ionospheric	4.0	
	c) Multipath	6.0	
5)	Alignment		1.0
6)	Antenna Gain		9.5 dBi
7)	Received Signal Strength		-145.2 → -148.3 dBW/Hz
8)	Noise Density (490°K) (G/T = -17.4 dB/°K)		-201.7 dBW/Hz
9)	$(C/N_o)_D$	-	56.5 - 53.4 dB-Hz

TABLE A-16

SHF UPLINK FOR ONE 16 Kbps CHANNEL

1)	Frequency	8 GHz
2)	EIRP	44.6 dBW
3)	Path Loss	199.4 - 202.5 dB
4)	Propagation Loss	0.8 dB
5)	Polarization Loss	0.1 dB
6)	Antenna Gain	17 dB edge
7)	Received Signal Strength	-138.7 → -141.8 dBW/Hz
8)	Noise Density (1000°K)	-198.6 dBW/Hz
9)	C/N_o	59.9 - 56.8 dB-Hz

TABLE A-17

SHF UPLINK CARRIER
FOR MODULATED UPLINK

1)	Frequency	8 GHz
2)		
2)	EIRP	45.7 dBW
3)	Path Loss	199.4 - 202.5
4)	Propagation Loss	0.8 dB
5)	Polarization Loss	0.1 dB
6)	Antenna Gain	17 dB edge
7)	Received Signal Strength	-137.6 - -140.7 dBW/Hz
8)	Noise Density ($1000^{o}K$)	-198.6 dBW/Hz
9)	C/N_o	61.0 - 57.9 dB-Hz

TABLE A-18

SHF DOWNLINK CARRIER
FOR MODULATED DOWNLINK

1)	Frequency	7.275
2)	EIRP	-3.7 dBW
3)	Path Loss	198.6 - 201.7
4)	Propagation Loss	0.8 dB
5)	Polarization Loss	0.1 dB
6)	Antenna Gain	55 dB (30 ft. dish)
7)	Received Signal Strength	148.2 - -151.3
8)	Density (190^{o}K) (G/T=32.3 dB/^{o}K)	-205.8 dBW/Hz
9)	C/N_o	57.6 - 54.5 dB-Hz

TABLE A-19

POLARSAT: CONCEPT DEFINITION STUDY FOR A POLAR ORBITING COMMUNICATION SATELLITE SYSTEM PROVIDING COMPLETE ARCTIC COVERAGE.