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APPLICATIONS TECHNOLOGY SATELLITE ADVANCED MISSIONS STUDY

FINAL REPORT
VOLUME I OF II

FAIRCHILD
SPACE & ELECTRONICS DIVISION

prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

NASA Lewis Research Center
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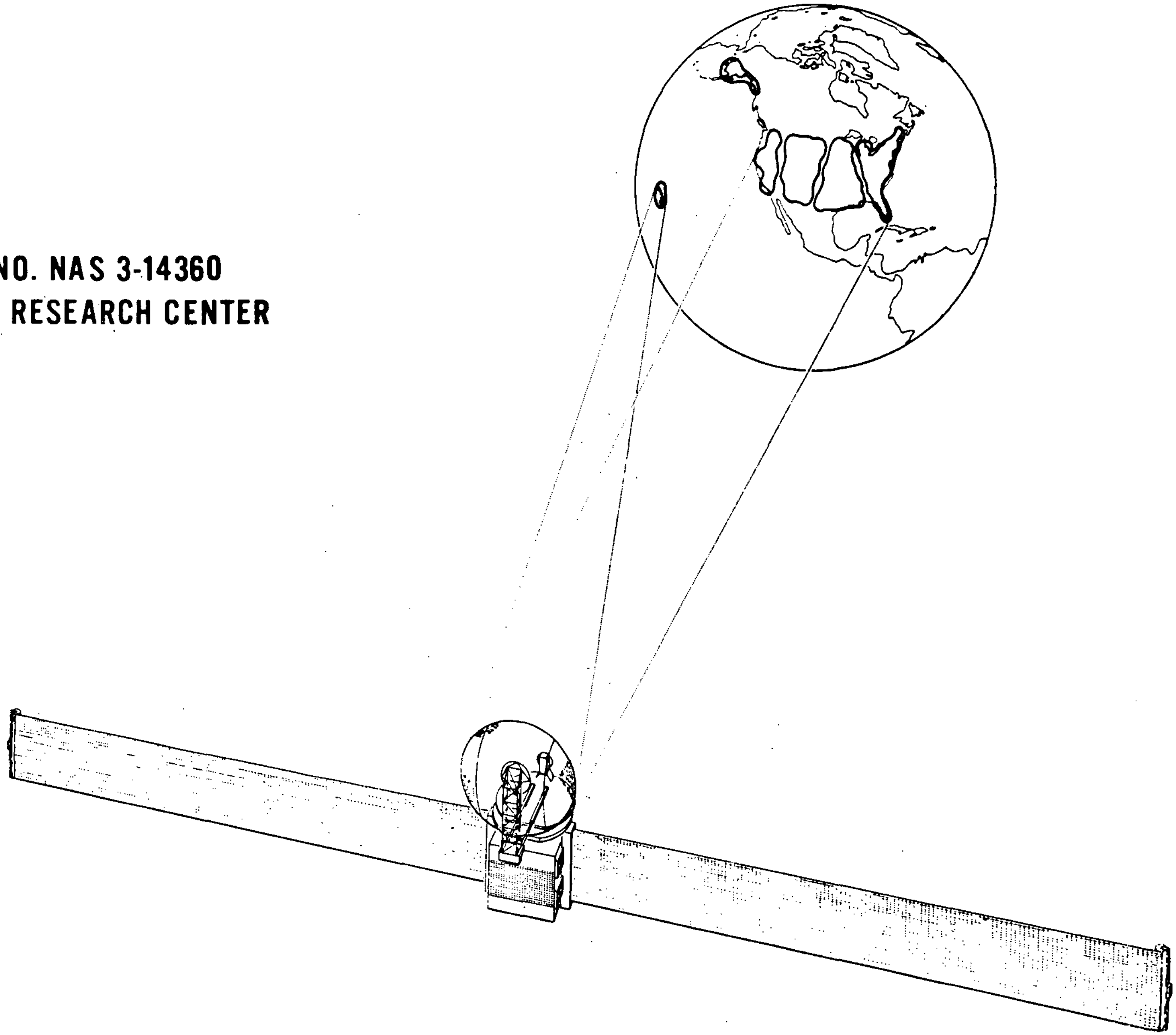

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16. Abstract

Four different spacecraft configurations were developed for geostationary service as a high power communications satellite. The first configuration is a Thor-Delta launch into a low orbit with a spiral ascent to synchronous altitude by ion engine propulsion. The spacecraft is earth oriented with rotating solar arrays. Configuration #2 is a direct injection Atlas/Centaur/Burner II vehicle which when in orbit is sun-oriented with a rotating transponder tower. Configurations #3 and #4 are Titan IIIC launches, and are therefore larger and heavier than Configuration #2. They are both sun-oriented, with rotating transponder towers and are directly injected into orbit. Technology discussed in this report includes high power (up to 2 kW) transmitters with collectors radiating heat directly into space, and contoured antenna patterns designed to illuminate particular earth regions. There is also a review of potential users of the services which can be performed by this type satellite in such areas as information networking, Public Broadcasting and Educational Television.

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SUMMARY

The objective of the Applications Technology Satellite Advanced Mission Study was to develop several approaches to the design of a spacecraft capable of demonstrating the feasibility of high power microwave communication satellites which produce shaped multi-beams to illuminate desired areas of the earth. Additionally, the satellites are suitable for use in the demonstration of information networks comprised of small user terminals.

Included in the scope of the work accomplished was the preliminary design of several possible spacecraft approaches and the associated prime experiments together with supporting analysis and tradeoff studies. Additional experiments compatible with the capabilities of the spacecraft were defined. The program implications, such as gross implementation schedules and resources; manufacturing, test and support; and critical research and development requirements were identified.

This report identifies potential users of wideband information networking systems whose operations would be enhanced by the use of a high power microwave communication satellite in geostationary orbit operating into small earth terminals. These potential users require technical data, operational experience, and hardware prove-out before committing their resources. An Applications Technology Satellite designed to meet this requirement is the ideal response to the requirement for the experimental data, experience and hardware that is essential for system specification.

The Titan III C launched spacecraft versions were found to be the most suitable for meeting the objectives.

SECTION 1

INTRODUCTION

This document is the result of a study performed at Fairchild Industries Space and Electronics Division under NASA contract NAS3-14360 and Amendment Number 1 issued by the Lewis Research Center. This section describes the scope of the study and the constraints under which it was performed as directed by the Statement of Work.

1.1 MISSION OBJECTIVES

The study was performed with the aim of providing sound technical plans for developing an Applications Technology Satellite whose main mission objectives are to demonstrate the feasibility of high power communication satellites using shaped multibeams and to demonstrate the use of such satellites in the development of information networks comprised of small user terminals.

1.2 STUDY OBJECTIVES

The objective of this study has been to develop a document presenting several approaches to the design of a spacecraft capable of fulfilling the mission objectives described above. Specifically the study included:

- Definition of three possible spacecraft approaches and associated prime experiments.
- Analysis of the approaches including tradeoffs and comparisons.
- Definition of additional experiments utilizing the capabilities of the spacecraft.

- Identification of critical research and development requirements.
- Determination of hardware and facilities requirements for manufacturing, test and support.
- Determination of gross implementation schedules and estimation of the resources required.

1.3 CONSTRAINTS

The study was performed under a certain set of constraints imposed by the Statement of Work. These constraints are outlined below.

1.3.1 GENERAL CONSTRAINTS

- The projected first launch date is 1976
- The spacecraft is placed in geostationary orbit
- The spacecraft operates for a minimum period of two years with all subsystems operating within specification

1.3.2 BOOSTERS

The following three launch vehicles were considered:

- Titan IIIC (ATS - AMS III A and B) Figure 1.3.2-1
- SLV3C CENTAUR (ATS-AMS II)* Figure 1.3.2-2
- TAT (9c)/DELTA (ATS-AMS I) Figure 1.3.2-3

Apogee kick motors, Burner II, and electric third stage were considered for final orbit injection.

1.3.3 SPACECRAFT POSITIONING

- The attitude control permits antenna beam pointing accuracy of ± 0.2° at the sub-satellite point and ± 0.2° in rotation about the boresight axis for five years

* This series vehicle will be superceeded in the 1976 time period. Launch vehicles considered in the study were SLV3D/Centaur D - 1A/Burner II (ATS-AMS II) and Delta 2910 (ATS-AMS I).

	AMS III A	AMS III B
SPACECRAFT WEIGHT	1725 kg (3800 lb)	1485 kg (3270 lb)
VIDEO/AUDIO/DATA TRANSPONDERS	10	5
AUDIO/DATA TRANSPONDERS	10	5
SOLAR ARRAY POWER (SOL)	9.6 kW	7.5 kW
EXTENDED ARRAY SPAN	42.5m (130 ft)	33.6m (102 ft)
EQUIPMENT MODULE	1.8m CUBE	
MISSION LIFE	2-5 YEARS	

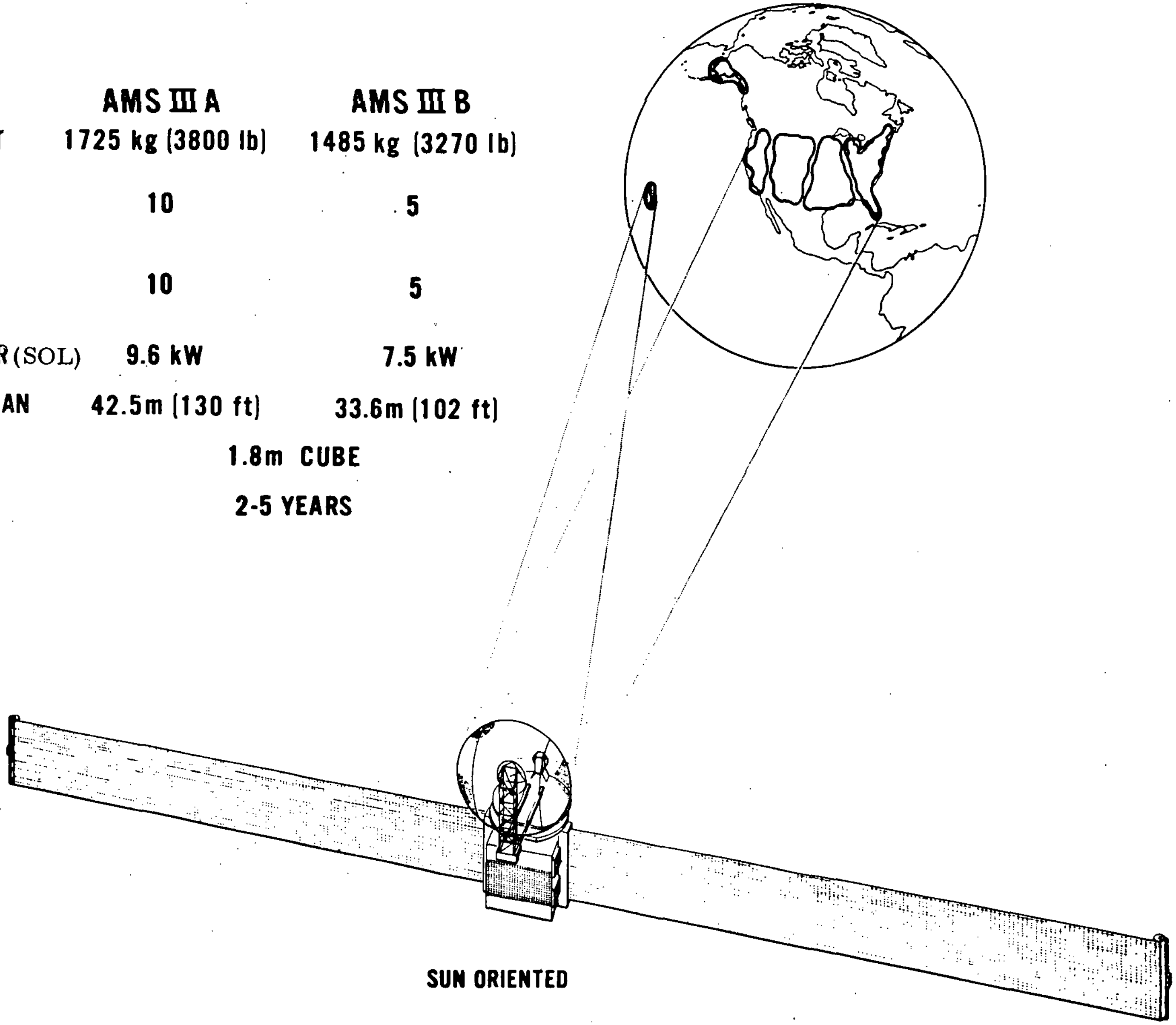


Figure 1.3.2-1. ATS-Advanced Mission Spacecraft III (Direct Ascent-Titan IIIC Launch)

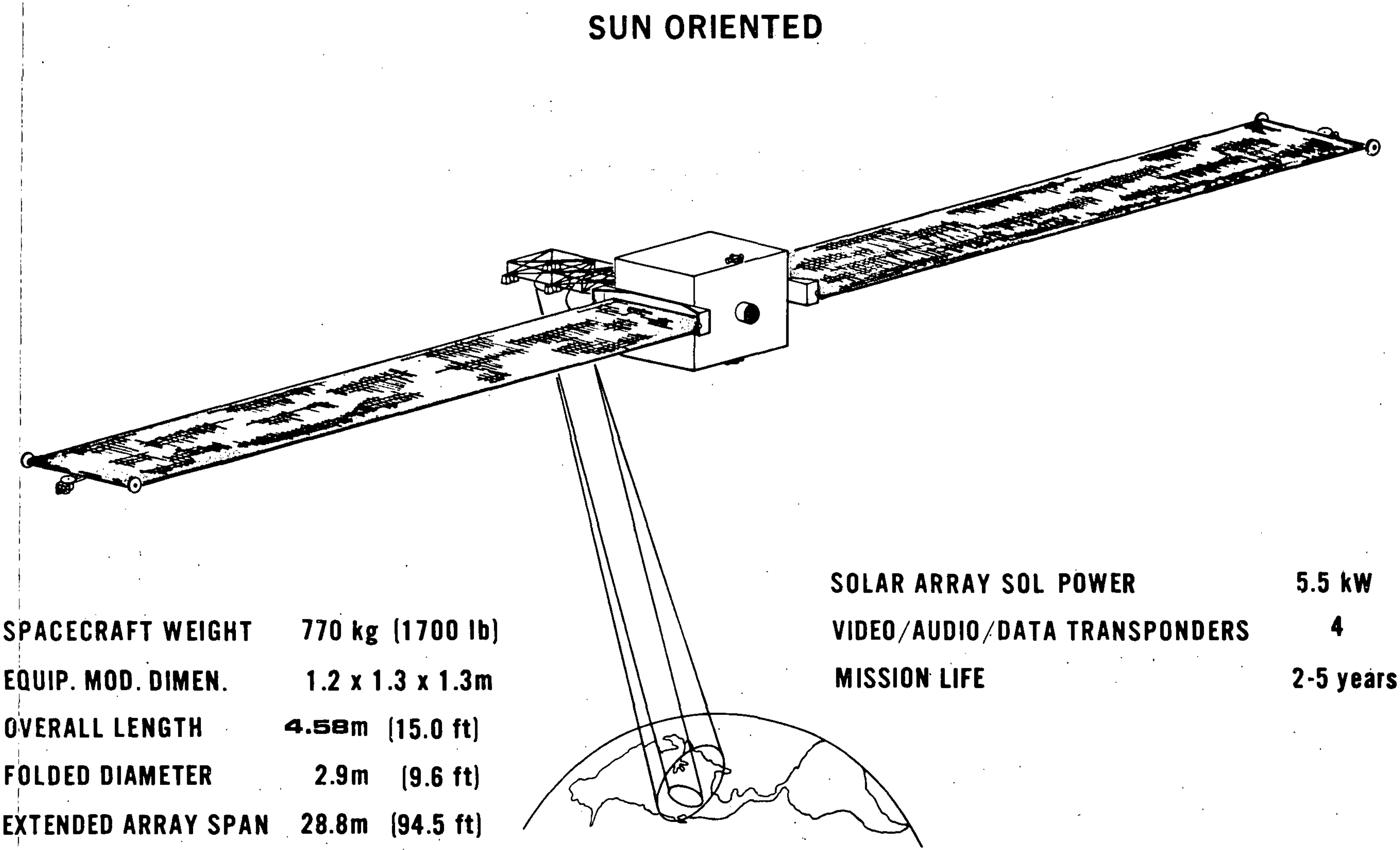


Figure 1.3.2-2. ATS-Advanced Mission Spacecraft II (Direct Ascent - SLV3D/Centaur D-1A/Burner II Launched)

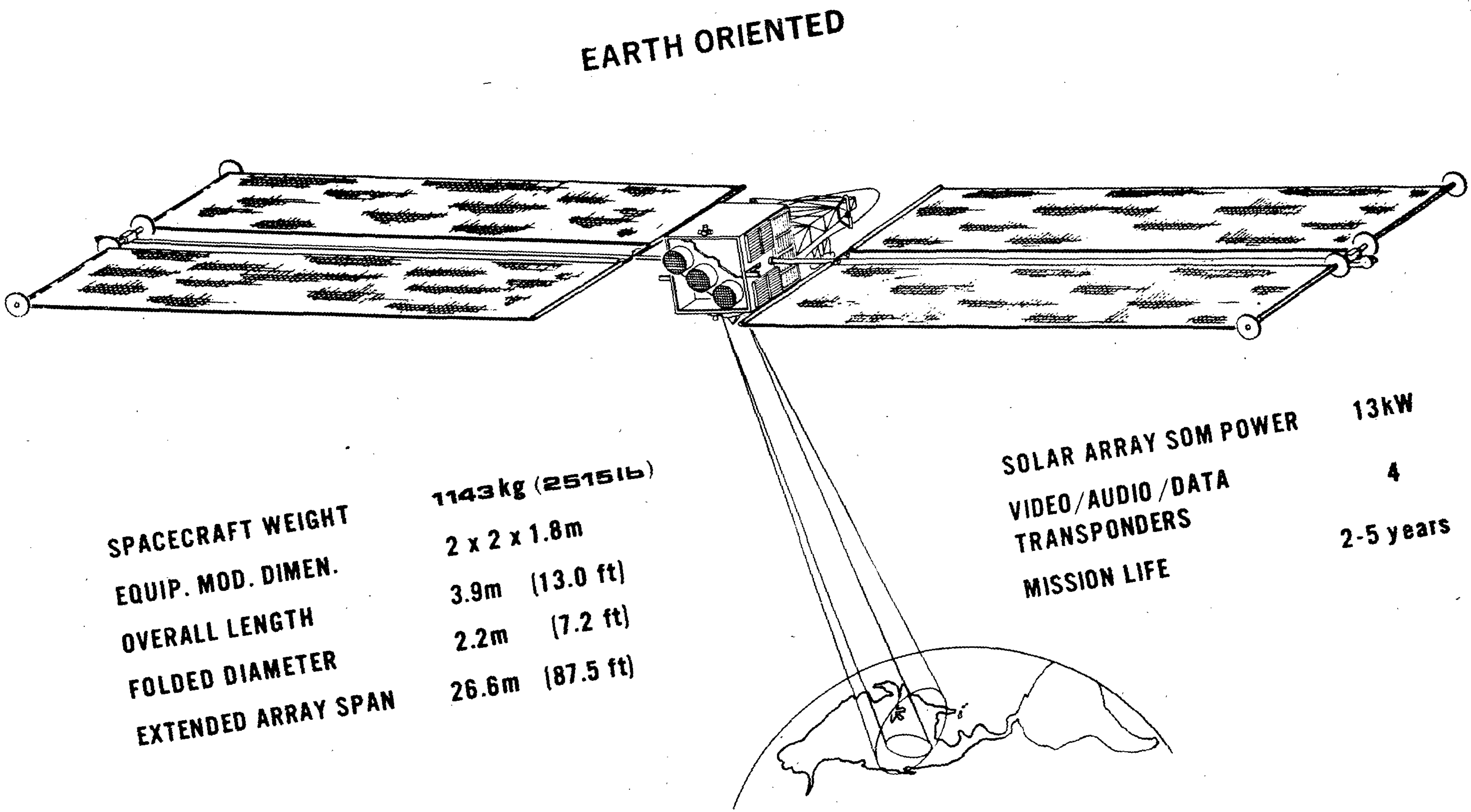


Figure 1.3.2-3. ATS-Advanced Mission Spacecraft I (Spiral Ascent - Delta 2910/30 Cm Ion Engine Launched)

- The spacecraft maintains station keeping to ± 0.2° for five years
- The spacecraft has a minimum ΔV of 30.48 m/sec (100 ft/sec) for repositioning

1.3.4 POWER CONSTRAINTS

- The prime power source is a photovoltaic array having both high and low voltage sections
- Energy storage is provided for housekeeping and operation of the power amplifier cathode heaters at 50% level during solar eclipse
- Space vacuum is used for high voltage insulation whenever possible

1.3.5 MICROWAVE TRANSMISSION

- Microwave power amplifiers are klystrons and traveling wave tubes (TWT) of 50% or greater efficiency.
- The spacecraft configuration allows for open envelope tube operation
- Angle modulation is the format
- Output signal level ranges from saturation to 6 dB below saturation
- The spacecraft is capable of repeater-type operation

1.3.6 EXPERIMENTS

The spacecraft incorporates the specified experiments:

- Generates more than 1 kW microwave power in the 11.7 to 12.2 GHz band

- Generates multibeam shaped patterns for the controlled illumination of desired areas on earth with contoured beam patterns
- Generates high power with solar arrays
- Uses gallium liquid metal slip rings for efficient power transfer from solar array to spacecraft body
- Demonstrates efficient methods for heat rejection from the transmitter power amplifier
- Demonstrates multibeam transmission to and from small ground terminals
- Directly generates high voltages from chains of solar cells

In defining other experiments, satellite to satellite communications, optical space to earth communications and station sensing by optical methods were considered.

1.4 REACTOR-THERMOELECTRIC POWER SYSTEMS

During the final weeks of the Advanced Mission Study period, two meetings were held between personnel from the U. S. Atomic Energy Commission and Fairchild Industries to discuss the possibility of utilizing a nuclear electrical power source as an alternative to the photovoltaic array. Available time and resources did not permit a full tradeoff study, but a report was provided by the Space Nuclear Systems Reactor Power Systems Branch and is included as Appendix A of this report so as to provide all of the available material in a single compilation.

SECTION 2

HIGH POWER COMMUNICATION SATELLITE MISSIONS

2.1 MISSIONS AND REQUIREMENTS

Existing point-to-point communication satellite systems utilize a satellite producing relatively modest effective isotropic radiated power (e.i.r.p.) in conjunction with a limited number of sophisticated earth terminals in order to achieve the necessary level of system performance. Technology advancements makes it economically attractive to implement networking systems with satellites of higher e.i.r.p. achieved through the use of efficient power amplifiers, contoured beam pattern antennas and related equipment, serving many simplified earth terminals.

For an application requiring a given satellite e.i.r.p. for reception by earth terminals having given antenna and receiver characteristics, consideration must be given to the relative influence of the satellite transmitter power and antenna gain in providing this e.i.r.p. The gain provided by the antenna is, of course, inversely related to its beamwidth or coverage area. Therefore, the required earth coverage area limits the gain available from the satellite antenna. In an application requiring only spot coverage, the satellite antenna can provide a greater proportion of the e.i.r.p., thus relaxing the transmitter power requirements. However, for an application requiring large area coverage to small earth terminals, such as would be the case for an information networking satellite, the satellite antenna gain is limited and this application can be satisfied only by a high-power satellite transmitter.

Another system consideration relating to antenna gain or beamwidth is the problem of antenna orientation. A highly directional satellite antenna imposes strict limitations on satellite attitude. Similarly, a highly directional earth station antenna makes initial installation alignment difficult, is subject to perturbations due to adverse weather, imposes limitations on satellite station-keeping, and requires that it have tracking capability, which is not economically feasible for a small station. Again, the use of high satellite transmitter power is the solution.

Ideally, the satellite antenna pattern footprint would exactly fit the contour of the service area with a uniform and adequate signal strength to conserve power and minimize interference with other services. Unfortunately, service areas are nearly always of an irregular contour, and simple beam antennas deliver circular or elliptical foolprints. As the following sections of this report will show, multibeam antennas producing contoured patterns are feasible, and - although there is a power loss in the satellite in deriving the contoured patterns - power conservation is achieved.

The application of these high-power technologies to communication satellites serving information networking systems comprised of many small terminal users makes the realization of the promise of innovative solutions to the problems of education, health care delivery and other areas of public concern near at hand. There are real needs by prospective users for operational wide-band information networking systems employing high power geostationary satellites. These prospective users require experimental data and actual demonstration experience to more fully understand how to best obtain the desired benefits and to appreciate the limitations of those systems.

By performing information networking experiments, they can gain the insight required to properly configure and specify their operational systems and to confidently commit public resources for the earth terminal hardware, software, personnel and training. An ATS-AMS configured to facilitate a working demonstration for several representative systems is the ideal response to these user needs.

2.2 BASELINE INFORMATION NETWORKING EXPERIMENTS

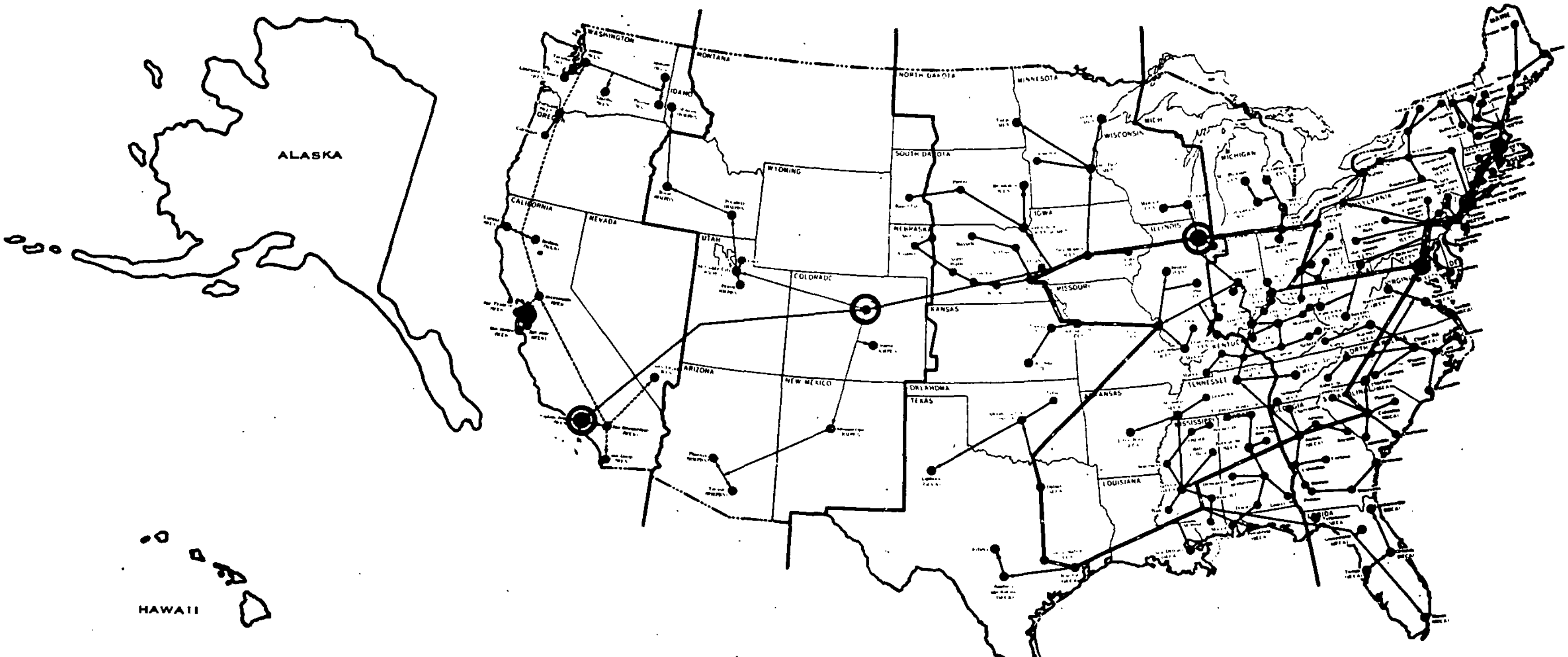
A number of information networking baseline experiments were developed for the Advanced Missions Study. Several are discussed in the following section.

2.2.1 PUBLIC BROADCASTING SYSTEM NETWORKING EXPERIMENTS

This experiment would provide experience and background in the utilization of a communication satellite for the PBS Interconnecting Service. As shown in Figure 2.2-1, program material would be provided for each of the four CONUS time zones plus the states of Alaska and Hawaii from the six originating stations. Over two hundred non-commercial television stations would receive the program material for local broadcasting. Storage centers in the Central, Mountain, and Pacific Zones would provide the appropriate time delay.

Several system configurations would be available to the service. Ten high quality relay channels would permit full time coverage of two independent TV channels to each of the five time zones. The ATS-AMS would relay the program material from the originating/storage station(s) in each time zone to the PBS stations with multibeam antenna patterns closely fitted to the contour of the zone for efficient usage of power.

- PBS STATIONS RECEIVE CCIR RELAY QUALITY (S/N = 56 dB, 3m ANTENNAS) VIDEO/DATA
- INTER-ZONE DISTRIBUTION VIA ATS-AMS III FROM ORIGINATING/STORAGE STATIONS (7.6m/60W)
- ORIGINATION TO ORIGINATION/STORAGE RELAY VIA ATS-AMS III & NEW YORK
- TRANSPORTABLE UPLINK STATIONS 3m/1kW



- ORIGINATING STATION (6)
- STORAGE STATION (3)

Figure 2.2-1. ATS-AMS III Baseline PBS Information Networking Experiment

Other configuration options include a reverse feed capability. The ATS-AMS would make available a means to route program material from properly equipped PBS stations or remote pick-up vans back to a network center for coordination and redistribution.

Through frequency reuse techniques such as polarized antenna feeds and buffer zone spacing, only six frequencies are necessary to service the ten channels. Referring to Figure 2.2-1, the Eastern Time Zone would be assigned frequency channels A and B. The adjacent Central Time Zone would be assigned Channels C and D. These same four frequencies would also be assigned as channels A' and B', C' and D' for the Mountain and Pacific Time Zones respectively, with additional isolation achieved through cross polarization of the earth terminal antennas. Alaska and Hawaii coverage would utilize the 5th and 6th frequencies.

2.2-2 SPECIAL INTEREST AREAS EXPERIMENT

Throughout the United States, there are areas where PBS television reception is difficult if not impossible due to the terrain or lack of a nearby PBS outlet. With relatively modest receiving equipment (as compared to the PBS broadcasting station); good quality television could be displayed, for instance, at schools, community centers, and other appropriate places for public gathering. Outlined in the CONUS Figure 2.2-2 is the Appalachian area and the area serviced by the Federation of Rocky Mountain States (Arizona and Nevada are reported not to be members at this writing) where this service would be particularly beneficial.

2.2.3 RAINFALL ATTENUATION EXPERIMENT

A problem most peculiar to the southeastern region of the United States is the possibility of temporary outages due to increased attenuation during heavy rainfall. The ATS-AMS multibeam antenna would permit increasing the power to the particular feed(s) associated with the outage link to correct the problem. (See Figure 2.2-3.)

- RECEIVE PBS NETWORK DISTRIBUTION DIRECT VIA ATS-AMS III
- TASO GRADE 1, (S/N = 46dB, 1m ANTENNA)

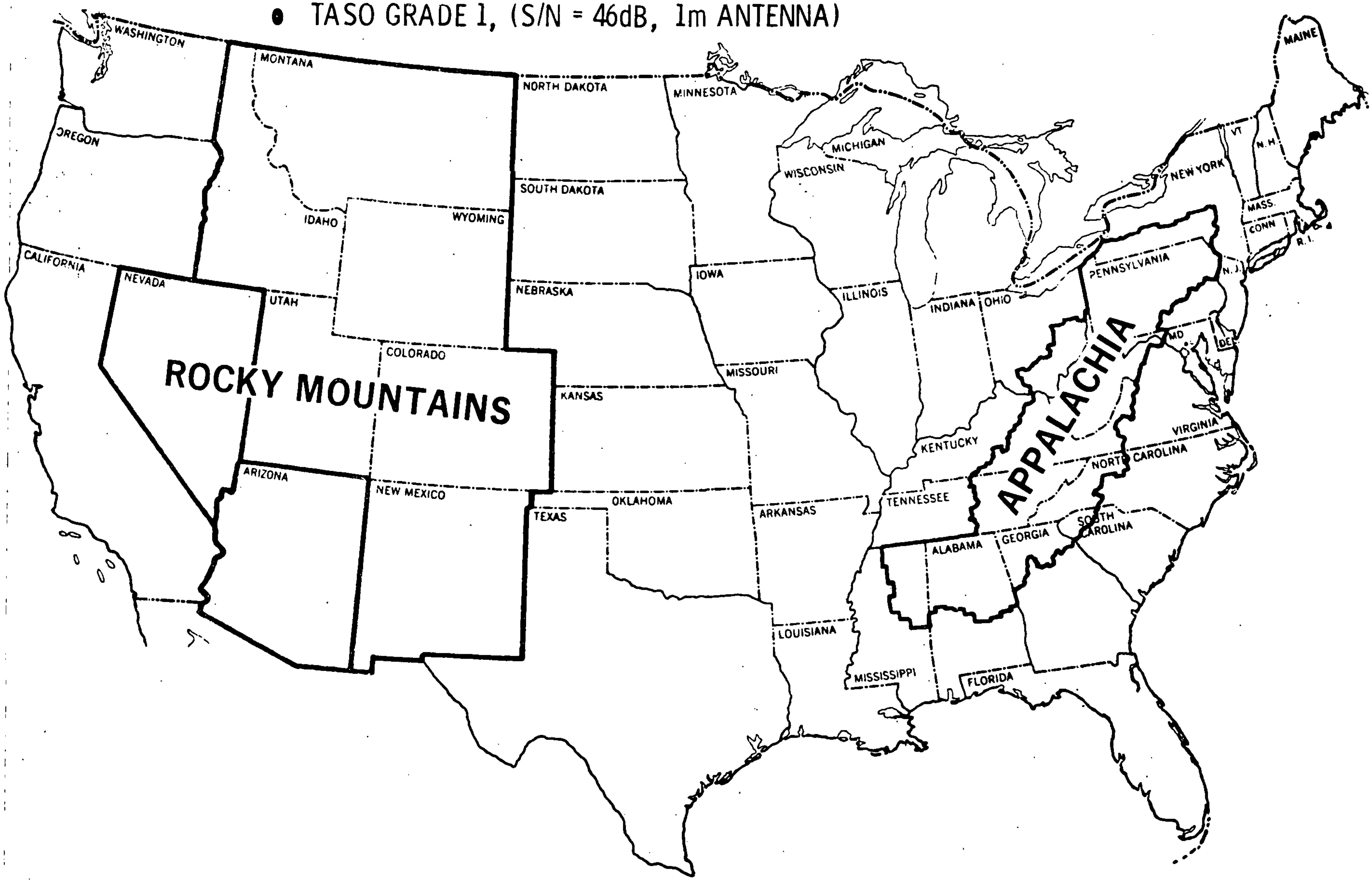


Figure 2.2-2. ATS-AMS III Baseline Special Interest Areas Information Networking Experiment

- PBS OR ITV APPLICATION
- INCREASE ATS-AMS III RF POWER OUTPUT 10dB

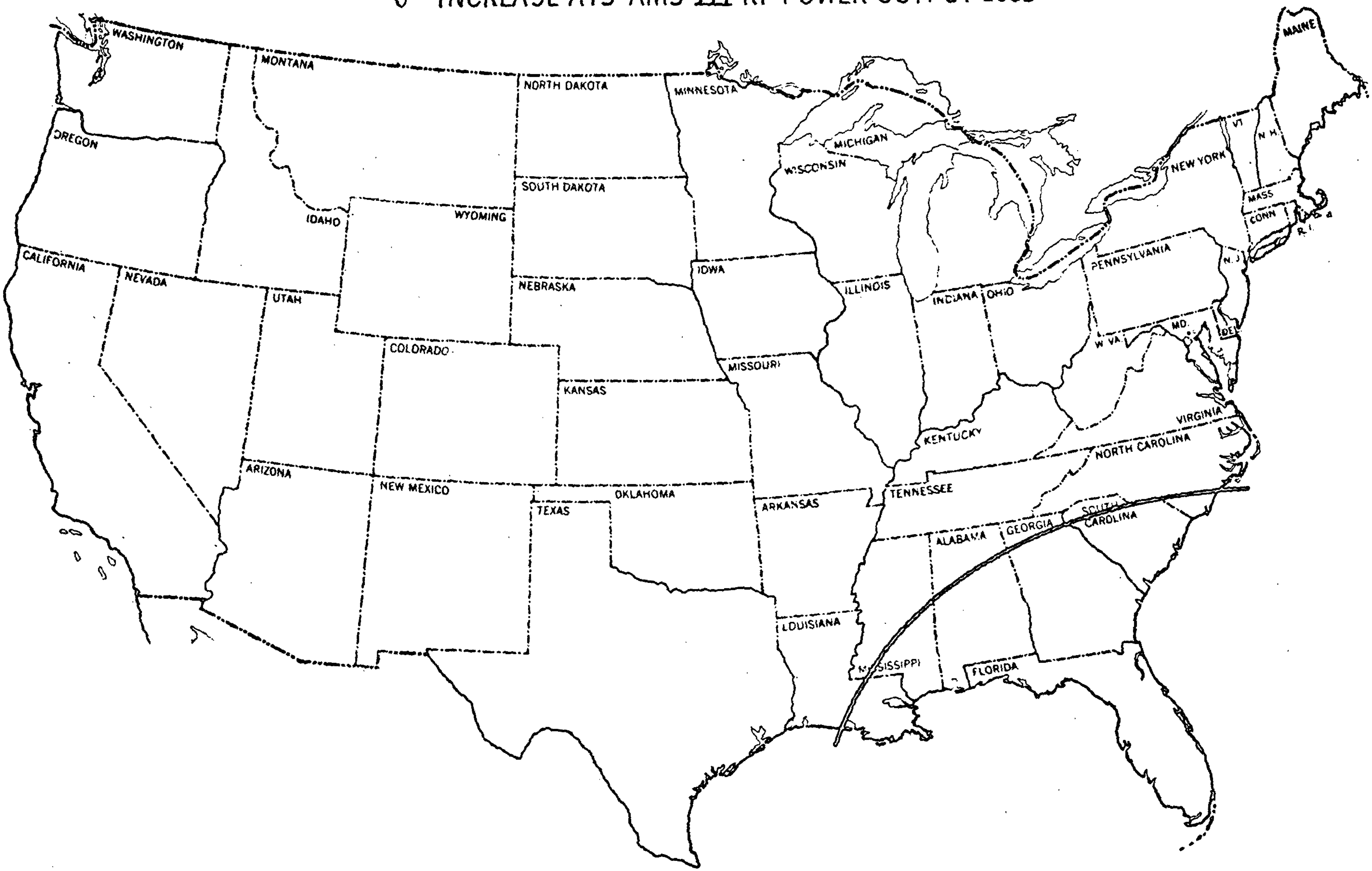


Figure 2.2-3. ATS-AMS Baseline Rainfall Attenuation Information Networking Experiment

2.2.4 CULTURAL REGION INTERACTIVE TV EXPERIMENT

As shown in Figure 2.2-4, ten centers are identified throughout six cultural regions in the CONUS. This experiment would provide additional background and experience in interactive in-school instructional television, computer-aided instructional television, medical information and remote diagnosis, etc. Experiments with a wide variety of interactive services could be facilitated. For example:

- Prominent Lecturer - A service can be envisioned to bring the knowledge and personality of prominent individuals to persons residing in a given region for a continuing education series. Students would observe and hear the lecturer on a conventional television display and would interact with the lecturer by means of voice response circuit. Total "classroom" attendance would be limited in size as it is with a conventional classroom; i.e., a maximum of about 30. An attractive option for quadrupling the number of simultaneous classes would be the use of frame rates on the order of 6 to 8 per second rather than the standard 30. This would not be a serious compromise since the actions of a lecturer and the display of material such as visual aid charts does not entail a great amount of motion.

- Popular Classroom - Increasing the size of an individual class necessitates restrictions in the degree of interaction between an instructor and the individual student. A service to classroom sizes limited by the channel capacity of the satellite could make use of a multiple choice response, say on the order of 5 choices. A large number of classrooms distributed throughout the region could observe the presentation on a conventional television display. Periodically, the instructor conducts a short multiple choice quiz, with the students entering their response on a small attachment at each desk. The results could be instantaneously

- SCHOOLS RECEIVE TASO GRADE 1 (S/N = 46dB, 1m ANTENNA) VIDEO/AUDIO/DATA
- INTER-REGIONAL DISTRIBUTION VIA ATS-AMS III FROM REGIONAL CENTERS (3m/380W)
- AUDIO/DATA TALKBACK (38W) TO REGIONAL CENTER

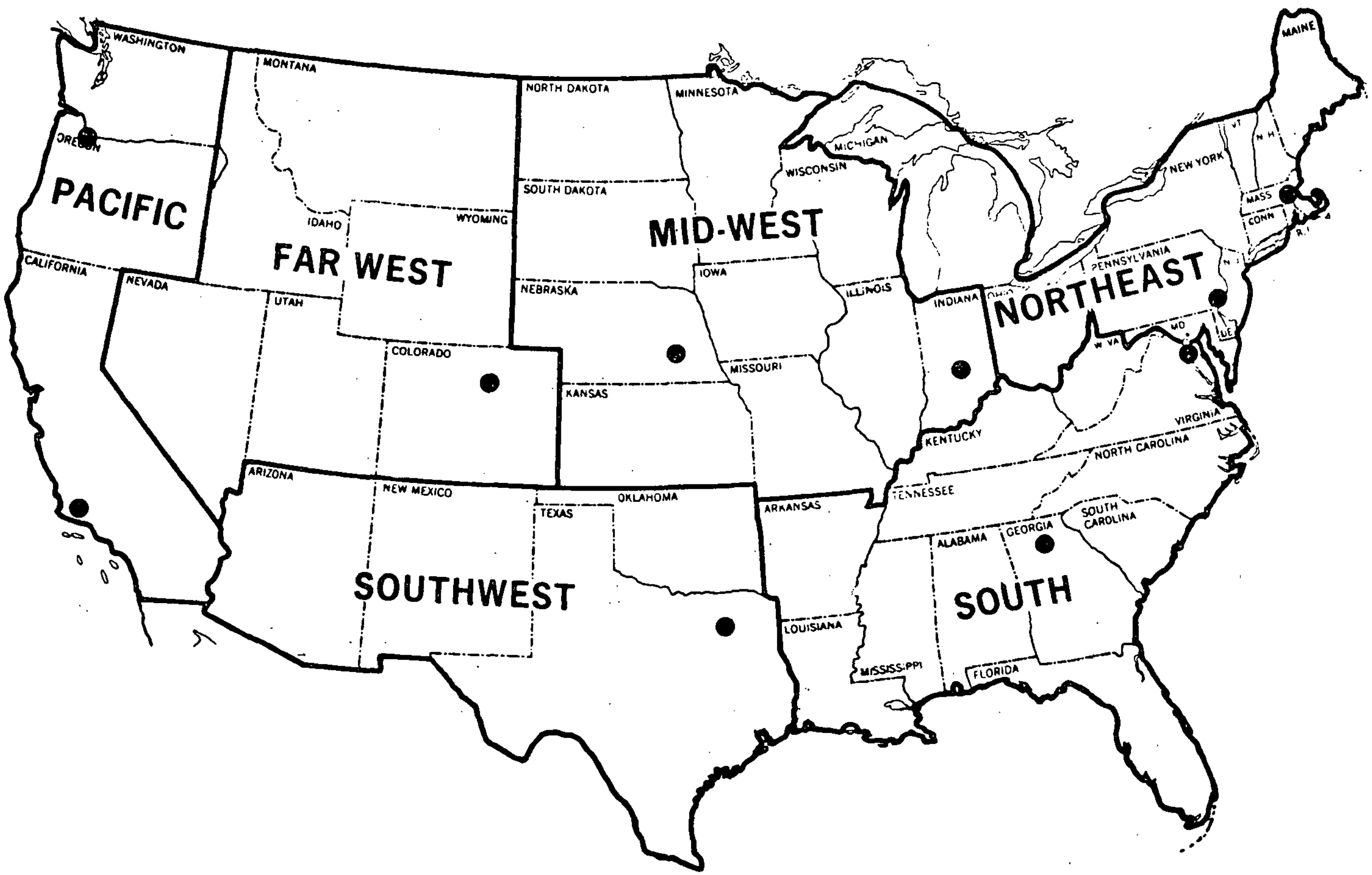


Figure 2.2-4. ATS-AMS III Baseline of Regional Interactive TV Information Networking Experiment

scored and recorded in each remote classroom, tabulated and transmitted to the origination center where the totaled response would be displayed to the instructor. He would then have an indication of the level of comprehension in his distributed classes and would proceed accordingly.

- Computer-Aided Instructional Television - Through the use of a computer directed access library at the regional origination center with keyboard terminals and television display "frame grabbers" in the classrooms, students would have the capability to pursue an array of research and programmed instruction courses.

The ATS-AMS would provide ten independent video channels for allocation among the six regions. Assuming one channel per originating center, no more than two channels would be allocated to a given region. Therefore, only six frequencies would be utilized. Again, additional isolation would be achieved through antenna cross polarization in addition to buffer zone spacing.

The ATS-AMS would incorporate a capability for one talk-back transponder per regional center. Each transponder would accommodate up to 25 audio channels per transponder or 6,250 low speed data circuits. Through operational queing, the number of participants would be primarily limited by the user peripheral equipment.

2.2.5 ALASKA MEDICAL INFORMATION NETWORKING EXPERIMENT

It has been dramatically demonstrated through the early ATS Spacecraft and elsewhere that medical information transmitted via communications networks can save lives in areas such as Alaska. An experiment to demonstrate the benefits of a two-way video/voice/data transmission capability between isolated regions and supporting services, such as The National Library of Medicine or cooperating medical centers in the Continental United States can be viewed as as a step in the direction of establishing a medical services program covering the entire State of Alaska. The services provided can be in the form of diagnostic assistance, or can be in the form of medical information for upgrading paramedical personnel capabilities.

2.2.6 EXPERIMENT PLANNING AND COORDINATION

It is important to recognize that attention must be addressed to the entire area of experiment planning and coordination with the users. Studies, possibly including physical simulations, are necessary to establish methods and techniques for the information networking experiments, such as multipoint multiple access cueing and control techniques.

Having the capability and the flexibility to provide service to unsophisticated earth terminals, the ATS-AMS would be valuable to many other information networking experiments. Section 7.2 discusses these in detail.

SECTION 3

ASCENT TRAJECTORIES AND LAUNCH SEQUENCES

Three ascent trajectories and four launch vehicles were considered during the study with the following combinations being recommended for continued consideration.

- Direct ascent to geosynchronous equatorial orbit using a current Titan III C and a growth version
- Direct ascent to geosynchronous equatorial orbit using an Atlas, Centaur/Burner II launch vehicle combination
- Injection into a 28.5° inclined parking orbit below the Van Allen belt using a Delta 2910 launch vehicle combination and orbit raising to geosynchronous equatorial by means of 30 cm Mercury ion thrusters

Injection into a 28.5° inclined parking orbit above the Van Allen belt using a Titan III B/Tandem BII launch vehicle combination and orbit raising to geosynchronous equatorial orbit with ion engines was also considered but the payload capability does not appear adequate for more than a minimal mission.

Highlights of the trajectory and launch vehicle analyses are given in the following subsections.

3.1 TITAN III C TRAJECTORY AND LAUNCH SEQUENCE

3.1.1 LAUNCH VEHICLE CHARACTERISTICS AND CAPABILITIES

The Titan III C launch vehicle is a four stage vehicle consisting of two 5-segment solid fuel rocket motors, a standard core and a Transtage.

The payload capability into a synchronous equatorial orbit (characteristic velocity 11.9 meters/second (39 ft/sec) is currently quoted at 1,550 kg (3,423 lbs). An increased payload capability of 1724 kg (3900 lbs) can be obtained by adding a Burner II above the Transtage. However, a recent communication from the Martin Marietta Denver Division indicates that further hardware improvements and trajectory optimization to the Titan III C could realize payload capabilities of up to 1724 kg (3900 lbs). The payload capability vs launch mission characteristic velocity for the current Titan III C is shown by Figure 3.1-1. Payload adapters trusses and separation devices are considered to be a part of the payload.

The payload fairing for the Titan III C is 3.05 m diameter and 10.7 m long (35 ft). A sketch of this fairing is given in Figure 3.1-2. The Titan III C profile is shown by Figure 3.1-3. The payload fairing is fully developed and is currently in use for all Titan III C vehicles.

3.1.2 ASCENT TRAJECTORY

The Titan III C is launched from the Eastern Test Range at 93° azimuth and ascends to a parking orbit with an altitude of 167 km (104 mi). At the first equatorial crossing (descending node) the transtage places the spacecraft into an elliptical transfer orbit with apogee at synchronous altitude of 35,786 km (22,000 mi). At the apogee equatorial crossing (ascending node) the Transtage performs a second burn and places the spacecraft into synchronous equatorial orbit. A ground track of this standard ascent trajectory is given in Figure 3.1-4. As shown by this figure, the spacecraft achieves synchronous orbit south of Malaysia. To place the spacecraft south of the United States, either an eastward drift to station is required or an ascent trajectory which requires multiple equatorial crossings in either the parking orbit or the transfer orbit. Minimum weight and time penalty occurs if injection into transfer orbit occurs at the second parking orbit equatorial crossing (ascending node) and injection into

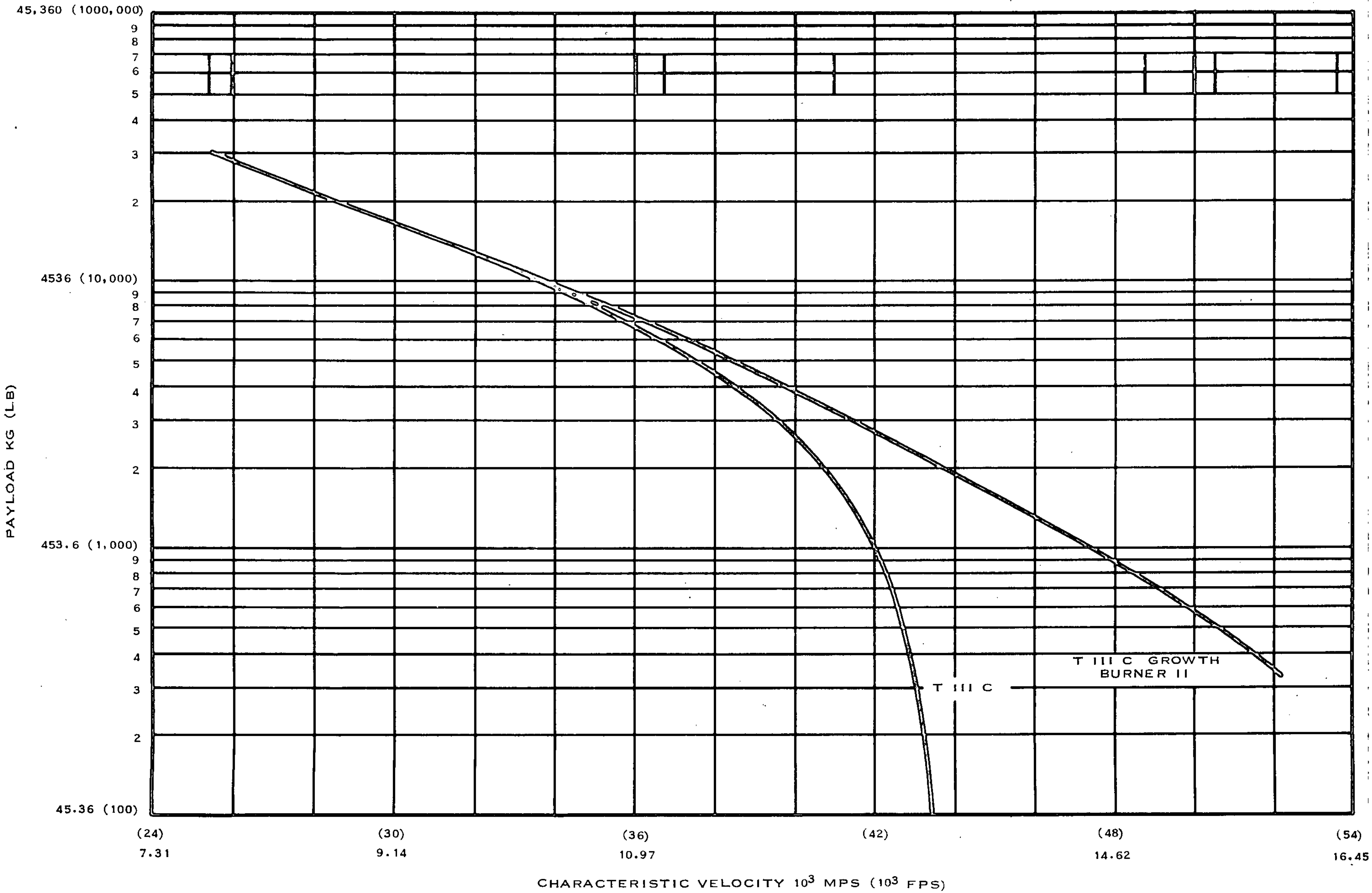


Figure 3. 1-1. Titan IIIC (T IIIC-26), Payload Weight vs. Characteristic Velocity, ETR

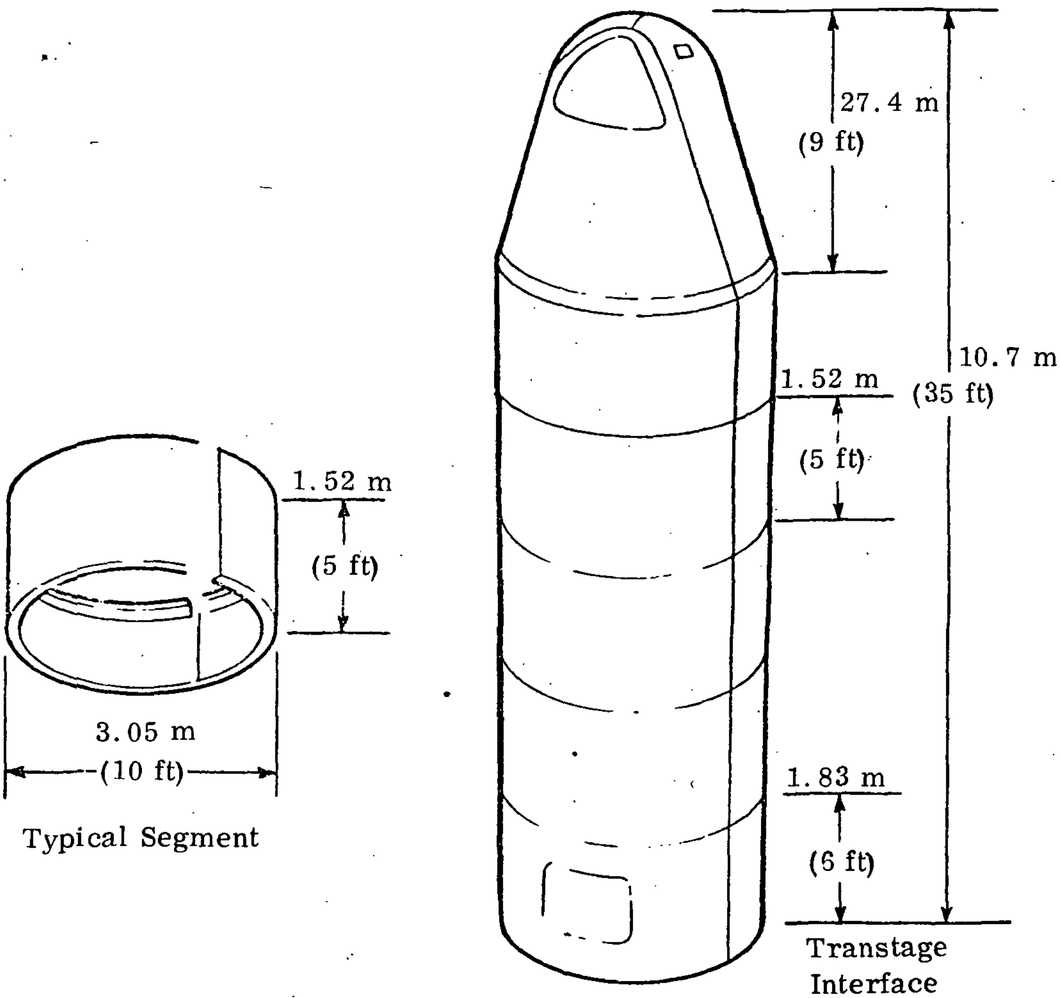


Figure 3.1-2. Titan IIIC Payload Fairing

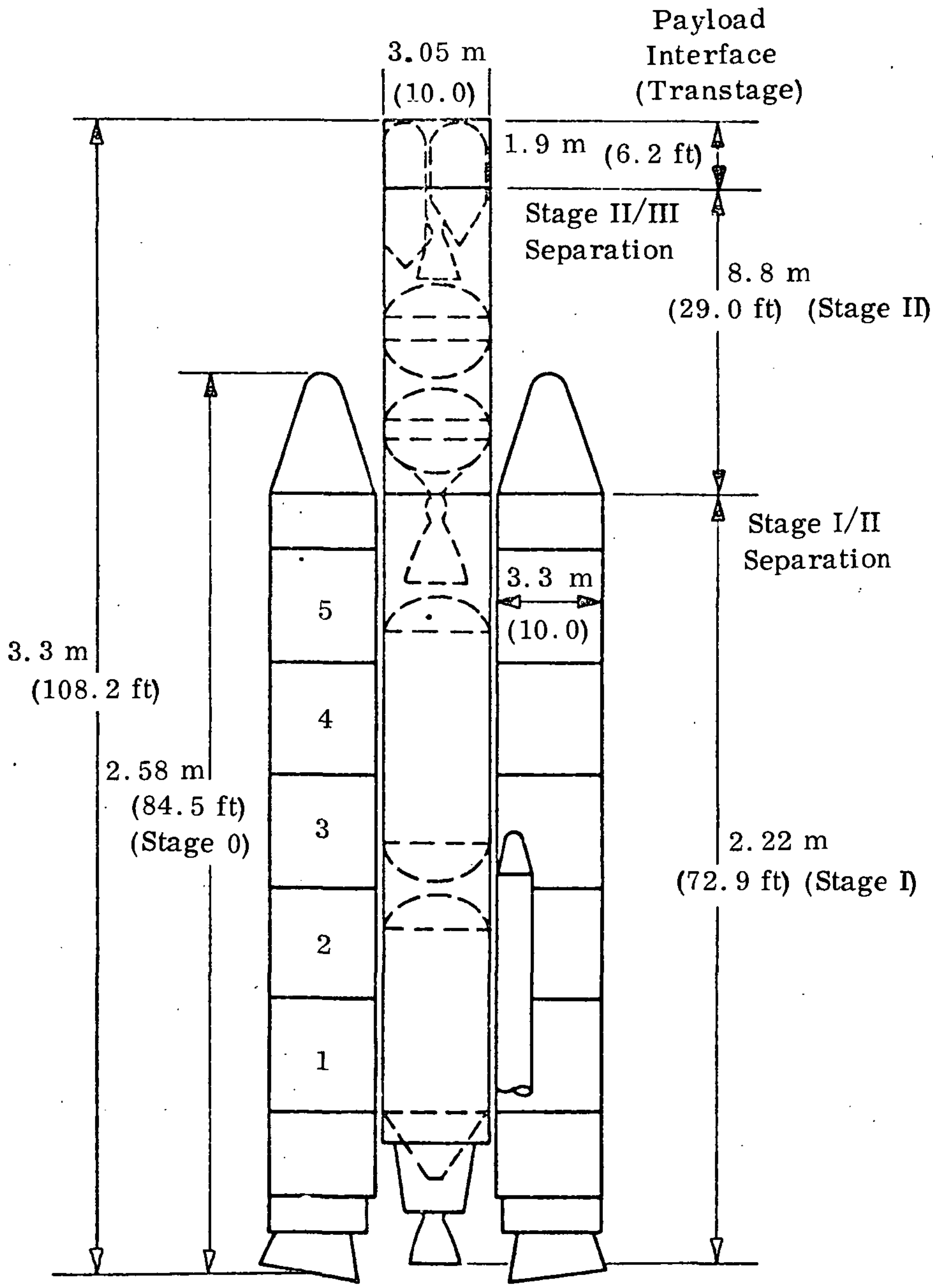


Figure 3.1-3. Titan IIIC Profile

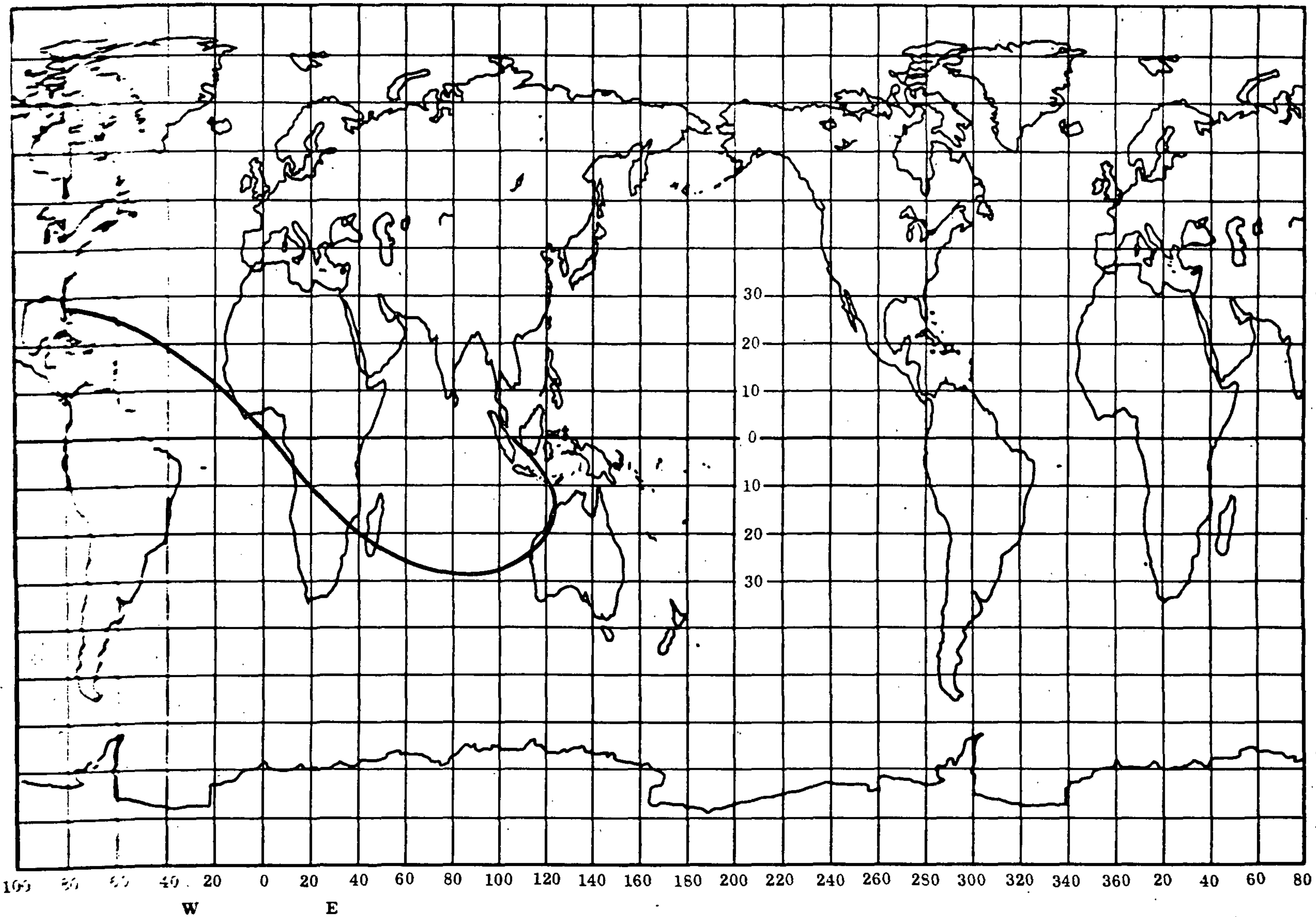


Figure 3.1-4. Titan IIIC Ascent Trajectory Ground Trace

synchronous equatorial orbit occurs at the first apogee equatorial crossing (ascending node). The orbit errors that can be expected with a Titan III C launch are as follows:

apogee altitude	352 km (218 mi)
perigee altitude	326 km (202 mi)
period error	± 11 min
orbit eccentricity	.0066
inclination error	± .165 deg
geocentric longitude	± .2 deg.

The period error is equivalent to a station change of ±2.75 degrees per day.

3.1.3 LAUNCH SEQUENCE

A typical launch sequence for the Titan III C is tabulated in Table 3.1-1 and illustrated in Figure 3.1-5.

Table 3.1-1. Titan III Typical Flight Sequence

Time (Sec)	Description
0	Liftoff from ETR LC 40, begin vertical rise.
10	Start pitchover with inertial pitch rate.
20	Start angle-of-attack attitude control.
30	Begin zero-lift flight.
80	Terminate zero-lift flight; initiate inertial pitch rate.
108.77	Acceleration of 2.18 g; start staging sequence.
111.70	Stage I ignition; simulated thrust buildup.
122.85	Solid rocket motor jettison.
126.00	Initiate inertial pitch rate.
131.00	Initiate inertial pitch rate.
258.24	Initiate Stage I tailoff Stage II ignition.
258.94	Jettison Stage I.
289.00	Jettison payload fairing.
450.00	Initiate inertial pitch rate.
462.56	Initiate Stage II tailoff.
476.06	Jettison Stage II, park orbit inject.
1350.19	Start first Stage III burn.
1656.59	First Stage III shutdown; inject into final transfer orbit.
20422.00	Start second Stage III burn.
20529.99	Second Stage III shutdown; inject into final orbit; begin vehicle reorientation.
20647.99	End vehicle reorientation; jettison payload forward of Station 77 into a 35,786 km circular orbit.

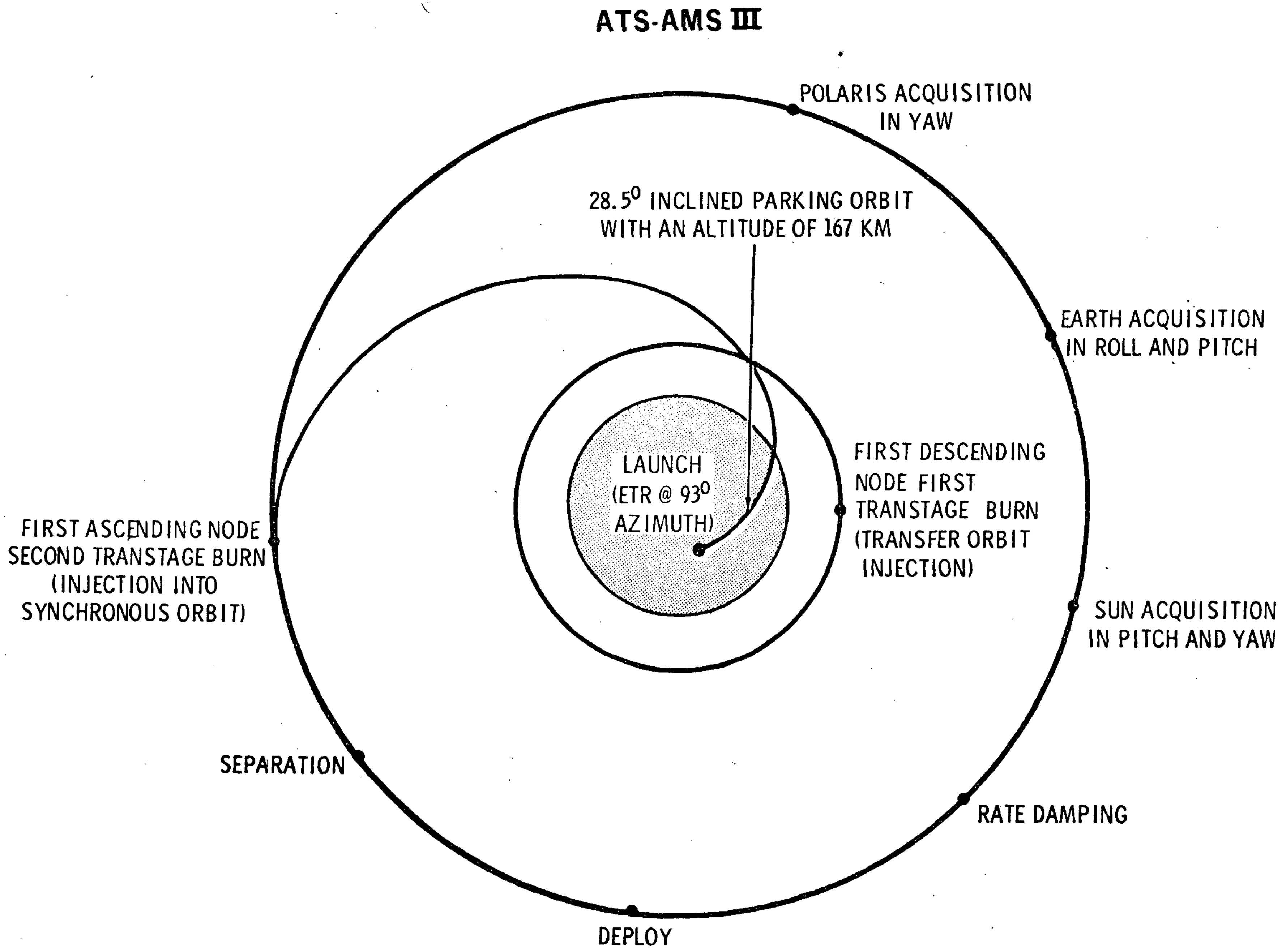


Figure 3.1-5. Positioning of S/C Over USA Obtained by Eastward Drift After Injection into Orbit

3.2 ATLAS/CENTAUR/BURNER II TRAJECTORY AND LAUNCH SEQUENCES

3.2.1 LAUNCH VEHICLE CHARACTERISTICS AND CAPABILITIES

The Atlas SLV-3D/Centaur D-1A is a two stage liquid propellant launch vehicle that is capable of injecting a payload of 1860 kg (4100 lbs) into a 185 x 35,786 km (100 x 19,323 nautical miles) elliptical transfer orbit. The overall Atlas/Centaur Profile is shown by Figure 3.2-1 and the allowable Spacecraft envelope is shown by Figure 3.2-2. The Burner II structure and the Burner/Centaur interfaces are shown on Figure 3.2-3. The Burner can accommodate either the TE-364-3 or TE-364-4 apogee kick motors which under standard conditions of propellant loading have the following total impulse capabilities.

TE-364-3	190,058 kg-sec (419,000 lb-sec) ± 6% (I_{SP} = 287 sec)
TE-364-4	358,344 kg-sec (790,000 lb-sec) ± .75% (I_{SP} = 284 sec)

The Burner II is an inertially guided stage and takes its attitude reference from the Centaur at separation. A recent telephone conversation with Convair San Diego indicated that the three Sigma accuracy of vehicle attitude at transfer was ± 1.2°.

The AKM can be offloaded or extended to accommodate the exact payload requirements with the propellant with the propellant weight determined by the relationship

$$W_f = W_P \left(1 - e^{\frac{-\Delta V}{g I_{SP}}}\right)$$

The Atlas Centaur normally utilizes a 185 km (100 mi) altitude parking orbit with a transfer orbit 185 x 35,786 km (19,323 n mi) inclined at 28.5° to the equator; the required ΔV at apogee is 183 m/sec (6000 ft/sec). It is possible to make a plane change at perigee to match a payload with the

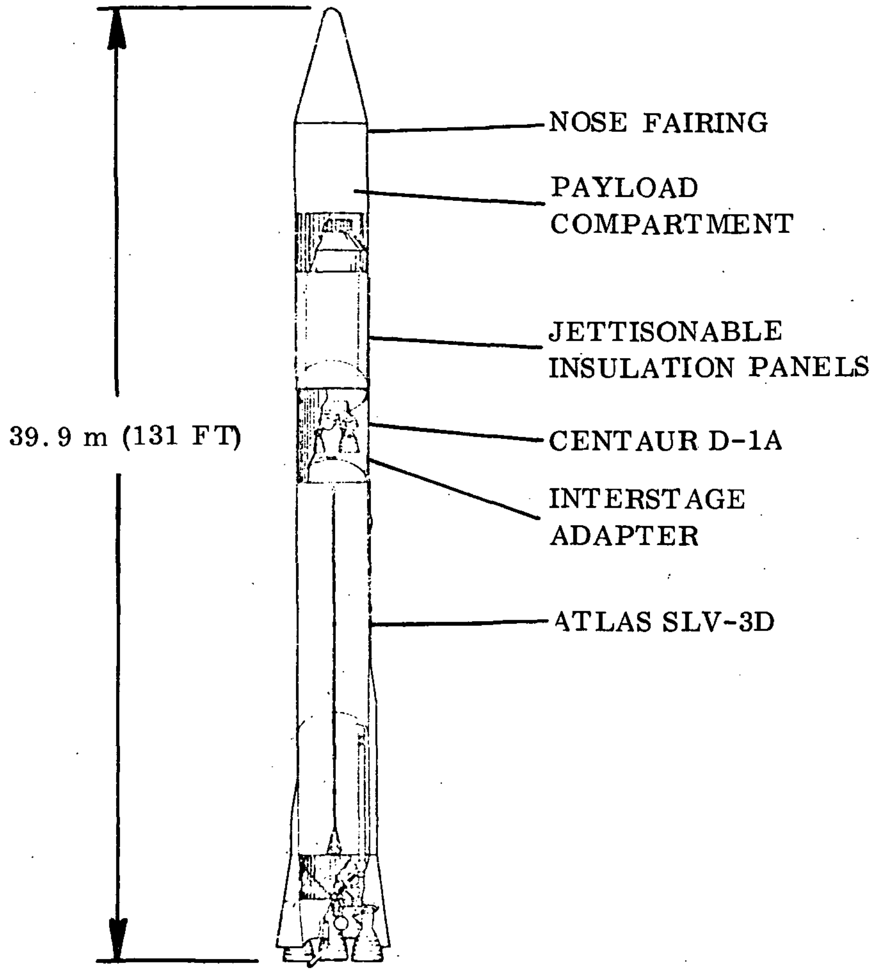
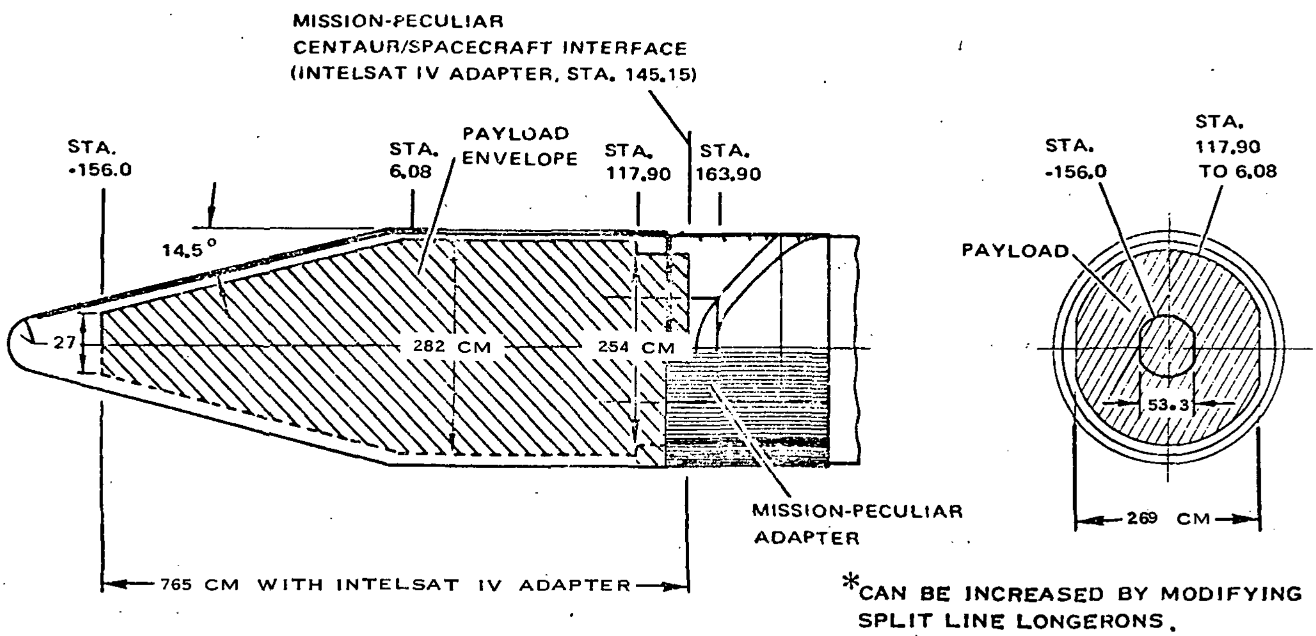


Figure 3.2-1. Atlas/Centaur Configuration



FIBERGLASS FAIRING ADVANTAGES:

- RF TRANSPARENT IN ALL DIRECTIONS
- INSIDE SKIN REMAINS COOL

Figure 3.2-2. Spacecraft Envelope for Centaur D-1A

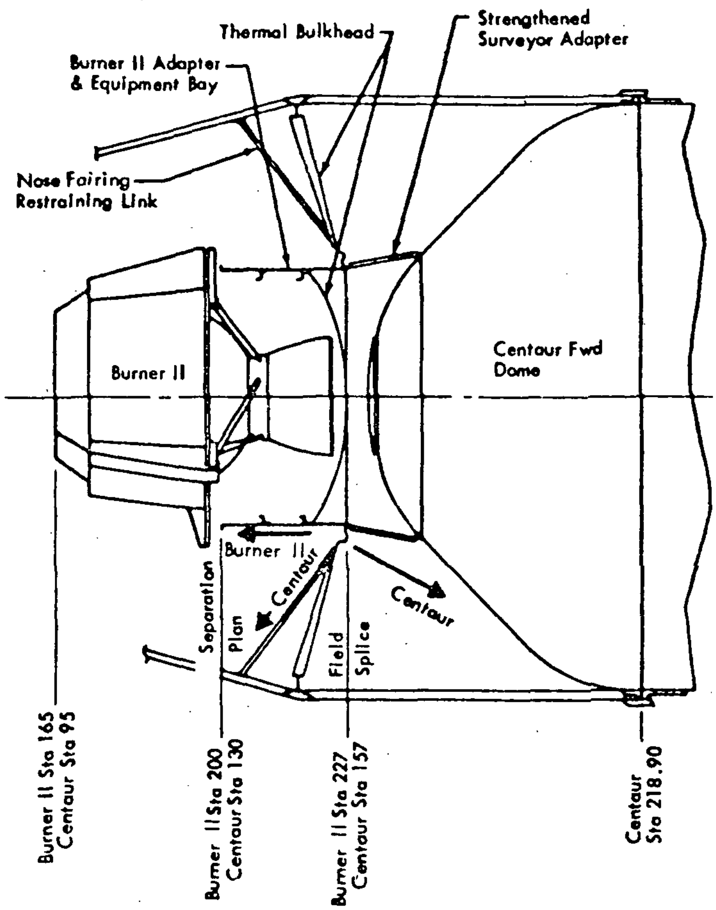


Figure 3.2-3. Burner II/Centaur Physical Interface

standard propellant loading of an apogee kick motor. Figure 3.2-4 shows the synchronous transfer orbit payload capability as a function of perigee plane change and the AKM propellant weight requirement as a function of final orbit inclination. As an example, the burnout weight of an apogee impulse Burner II with full time telemetry and coast control is 155 kg (341 lbs) and the propellant weight with a TEM-364-3 AKM is 660 kg (1453 lbs). Using this propellant loading and a final orbit inclination of 0^{o} and a perigee plane change of 10^{o} the synchronous transfer orbit payload including the spacecraft adapter would be 1450 kg (3200 lbs). It is also possible to use the Convair OV1-B as an apogee kick stage. The OV1-B also uses the Thiokol TEM-364-3 and-4 motors and has compatible performance to the Burner II.

3.2.2 ASCENT TRAJECTORY

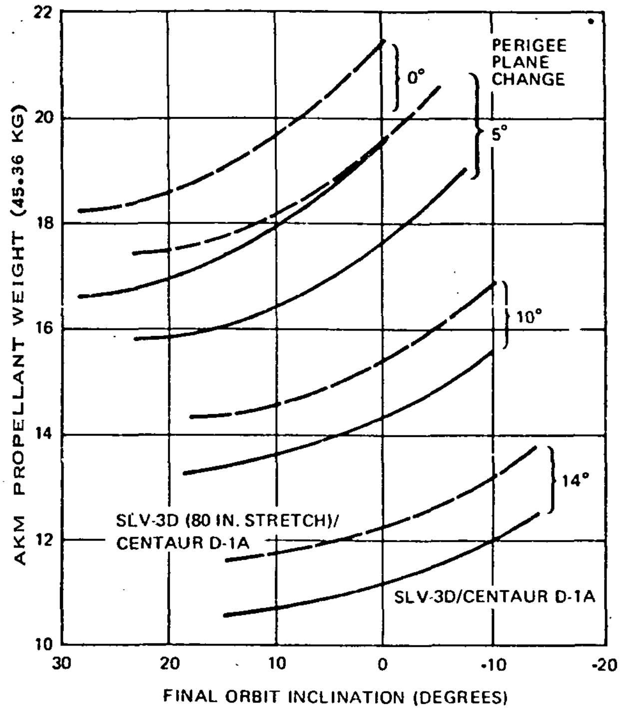
The ascent trajectory for the Atlas/Centaur/Burner II will be essentially the same as that described for the Titan III C except that with the Centaur launch vehicle, the weight penalty for transfer orbit injection at the second equatorial crossing is approximately 68 kg (150 lbs) and a weight optimum ascent trajectory would involve a first descending node perigee burn, a first ascending node apogee burn and a synchronous attitude coast to station. The orbit errors that can be expected with an Atlas/Centaur/Burner II launch are as follows:

period error	19.4 min
orbit eccentricity	.006
inclination error	± .37 deg
geocentric longitude	± .2 deg

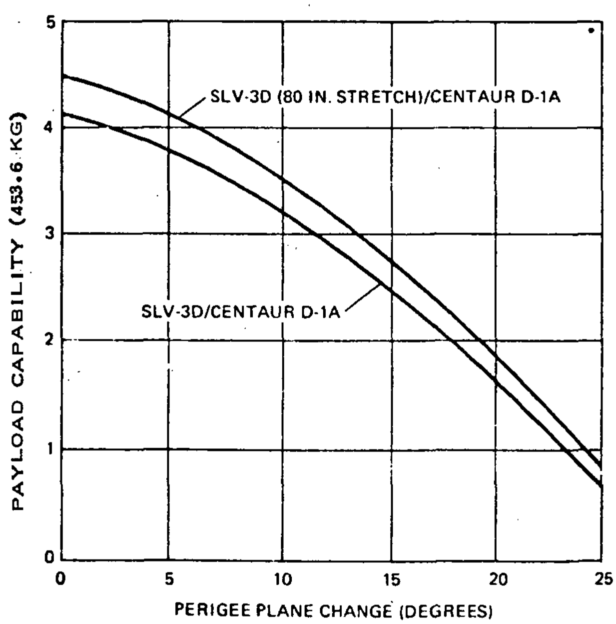
The period error is equivalent to a station change of 4.85 degrees per day.

3.2.3 LAUNCH SEQUENCE

A typical launch sequence for the Atlas/Centaur is tabulated in Table 3.2-1 and illustrated in Figure 3.2-5. Following separation



AKM Propellant Weights for Synchronous Altitude



Synchronous Transfer Orbit Payload Capability

Figure 3.2-4. Synchronous Transfer Orbit Payload Capability

Table 3.2-1. Typical synchronous equatorial mission, Atlas/Centaur sequence of events.

Events	Time (Approximate) (sec)	Time (Approximate) (sec)
2-Inch Motion (Liftoff)	Liftoff	T
Start Roll Program (Launch Azimuth 101 deg)		T + 2
End Roll Program		T + 15
Start Pitch and Yaw Program		T + 15
Booster Engine Cutoff (Guidance Discrete, Staging Acceleration 5.7g)	BECO	T + 151.1
Jettison Booster Package	BECO + 3.1	T + 154.2
Admit Guidance Steering	BECO + 8	T + 159.1
Jettison Insulation Panels	BECO + 45	T + 196.1
Sustainer Engine Cutoff (Propellant Depletion)	SECO	T + 241.0
Atlas/Centaur Separation	SECO + 1.9	T + 242.9
Fire Atlas Retrorockets	SECO + 2	T + 243.0
Centaur First Main Engines Start (SECO + 11.5 sec)	MES 1	T + 252.5
A/P Admit Guidance, Constant Attitude Steering	MES 1 + 4	T + 256.5
Jettison Nose Fairing	MES 1 + 12	T + 264.5
Begin Guid Clsd Loop Steering	MES 1 + 17	T + 269.5
Centaur First Main Engine Cutoff	MECO 1	T + 627.0
Centaur Second Main Engine Start (First Equatorial Crossing)	MES 2	T + 1520.0
Centaur Second Main Engine Cutoff	MECO 2	T + 1594.2
Spacecraft Separation	MECO 2 + 135	T + 1729.2
Start Turnaround	MECO 2 + 140	T + 1734.2
Tank Blowdown	MECO 2 + 305	T + 1899.2
V 1/2 On (H_2O_2 Depletion)	MECO 2 + 555	T + 2149.2
Power Changeover	MECO 2 + 605	T + 2199.2

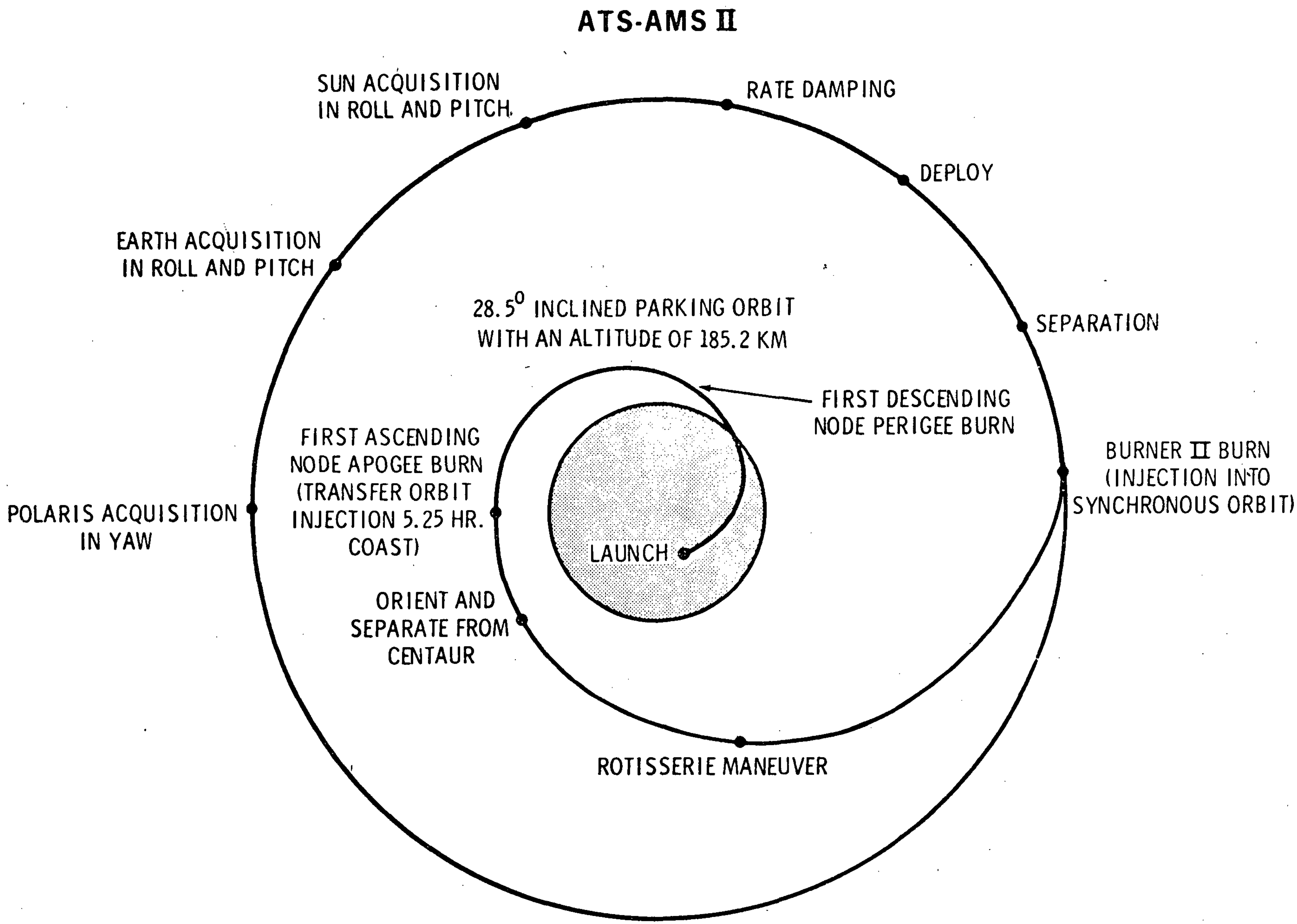


Figure 3.2-5. Spacecraft Positioning Over Final Station Will be Obtained by a Combination of Multiple Equatorial Crossings in Either the Parking Orbit or the Transfer Orbit and Eastward Drift.

from the Centaur, the Burner II and the attached spacecraft payload would coast to apogee with a coast time of 5.25 hours. During the coast period, the Burner II guidance equipment would perform a slow roll of approximately one-half revolution per minute to average the gyro drift errors and provide a better thermal balance for the spacecraft.

3.3 DELTA/ION ENGINE LAUNCH AND INJECTION SEQUENCES

3.3.1 LAUNCH VEHICLE CHARACTERISTICS AND CAPABILITIES

The vehicle configurations currently available for new mission planning are described by a four digit number, e.g. Delta 2914. The interpretation of the individual digits is defined as follows:

```
     Number of Solids—┐ ┌—Second Stage
        First Stage—┐ │ │ ┌—Third Stage
                    │ │ │ │
                    2 9 1 4
```

First Stage

0 Long Tank Thor with Rocketdyne Engine - Vehicles of this configuration are shown in the text as containing the last 3 digits only.

1 Extended Long Tank Thor with Rocketdyne MB-3 Engine.

2 Extended Long Tank Thor with Rocketdyne H-1 Engine.

Number of Solids

3, 6, 9 - Number of First Stage strap-on solid motors.

Second Stage

0 1.65 m (5.5 ft) diameter second stage and fairing with Aerojet General AJ10-118F propulsion system

1 2.44 m (8.0 ft) diameter second stage and fairing with the Aerojet General AJ10-118F propulsion system.

Third Stage

0 No third stage.

2 United Technology Center FW-4D.

3 Thiokol TE-364-3.

4 Thiokol TE-364-4.

For the ATS-AMS I configuration 2910 is recommended. The outboard profiles of the Delta Launch vehicles are shown in Figure 3.3-1. The allowable payload envelope is shown by Figure 3.3-2. A special attach fitting would have to be designed since the 5414 attach fitting shown in the payload envelope drawing is capable of supporting 499 kg (1100 lbs) only. Overall dimensions would however be comparable and a sketch of the 5414 fairing is shown for reference in Figure 3.3-3. The Delta 2910 launch vehicle capabilities are given by Figure 3.3-4 which shows useful payload as a function of circular orbit altitude. As indicated previously in this report, the design weight for the spacecraft is 1135 kg (2500 lbs) which will permit an initial circular orbit altitude (2040 km (1100 n.m.) using current Delta capabilities. By 1976 the Delta capabilities can be expected to grow from the current synchronous equatorial orbit payload capability of 703 kg (1550 lbs) to 786 kg (1730 lbs). This would imply an increased second stage capability as indicated by Figure 3.3-5 and make possible a circular orbit altitude of approximately 2593 km (1400 n.m.) with the 1135 kg (2500 lbs) payload. The ion engines used for raising the orbit from the inclined medium altitude parking orbit to synchronous altitude and removing the 28.5° orbit inclination are described in later sections of this report.

3.3.2 ORBIT RAISING TRAJECTORY COMPUTATIONS

The orbit raising trajectories (without plane change) were calculated by means of a computer program based on the "variation of parameters" method of analysis. Initial conditions of "starting power","spacecraft weight", and "parking orbit altitude" were used

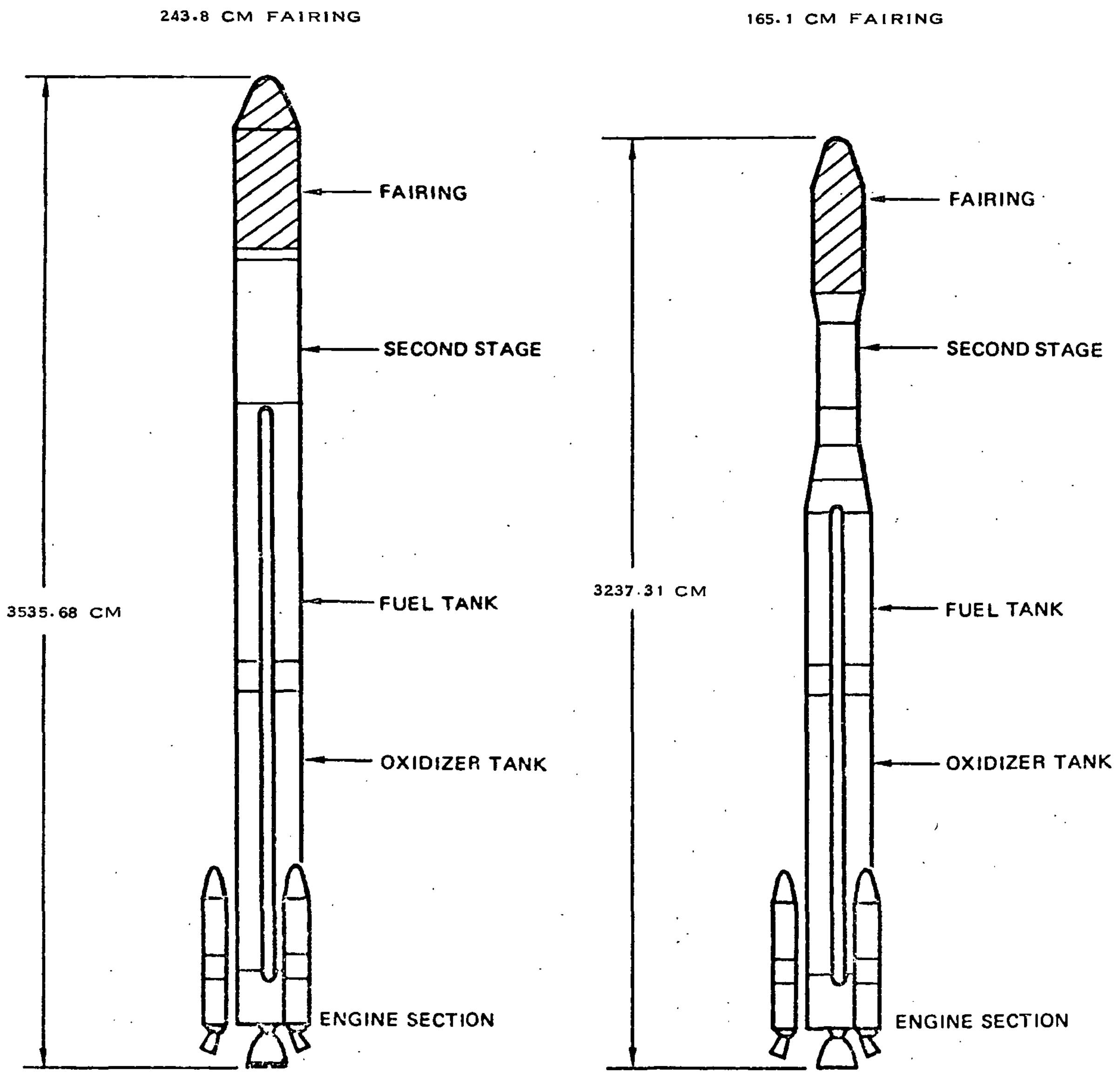
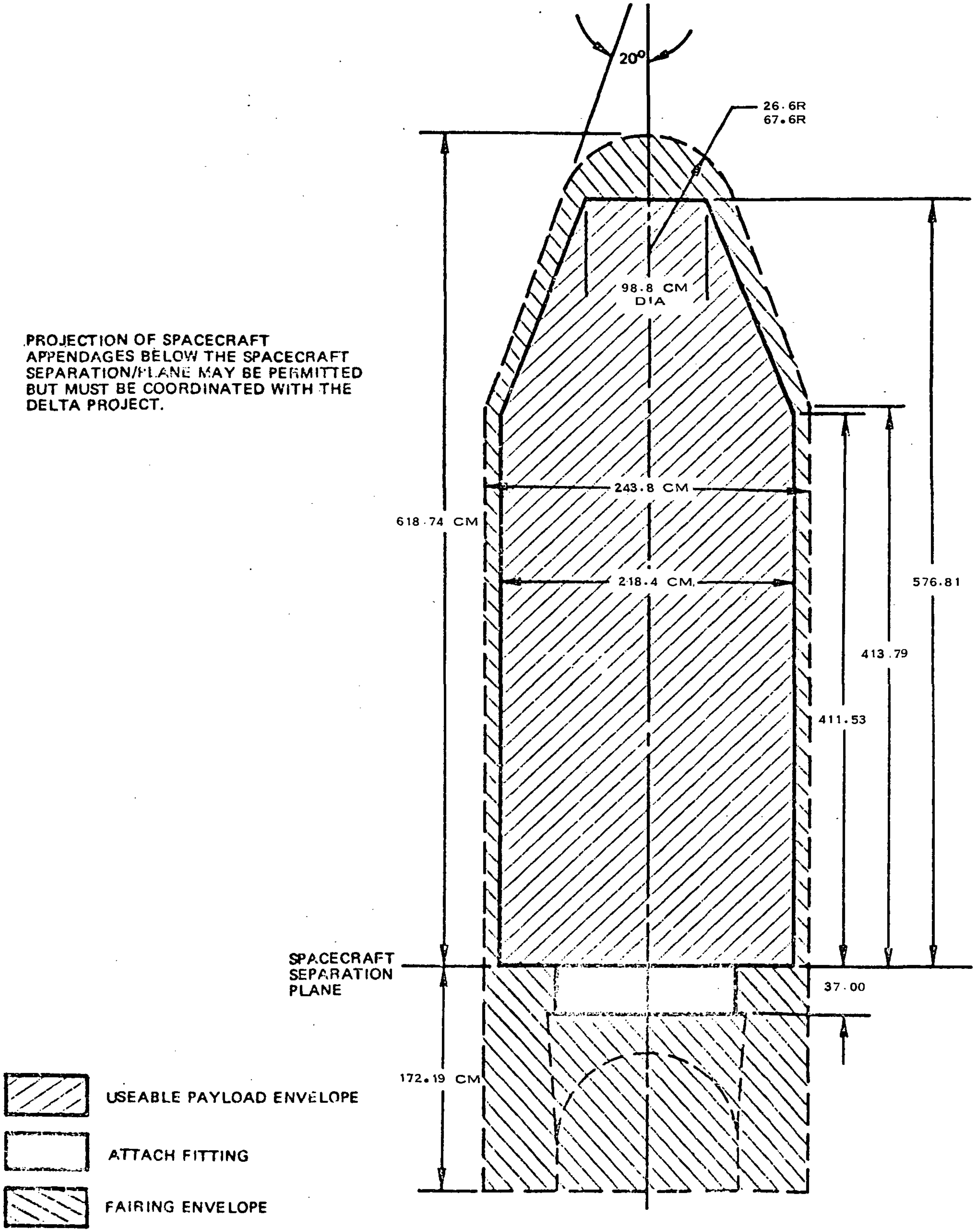
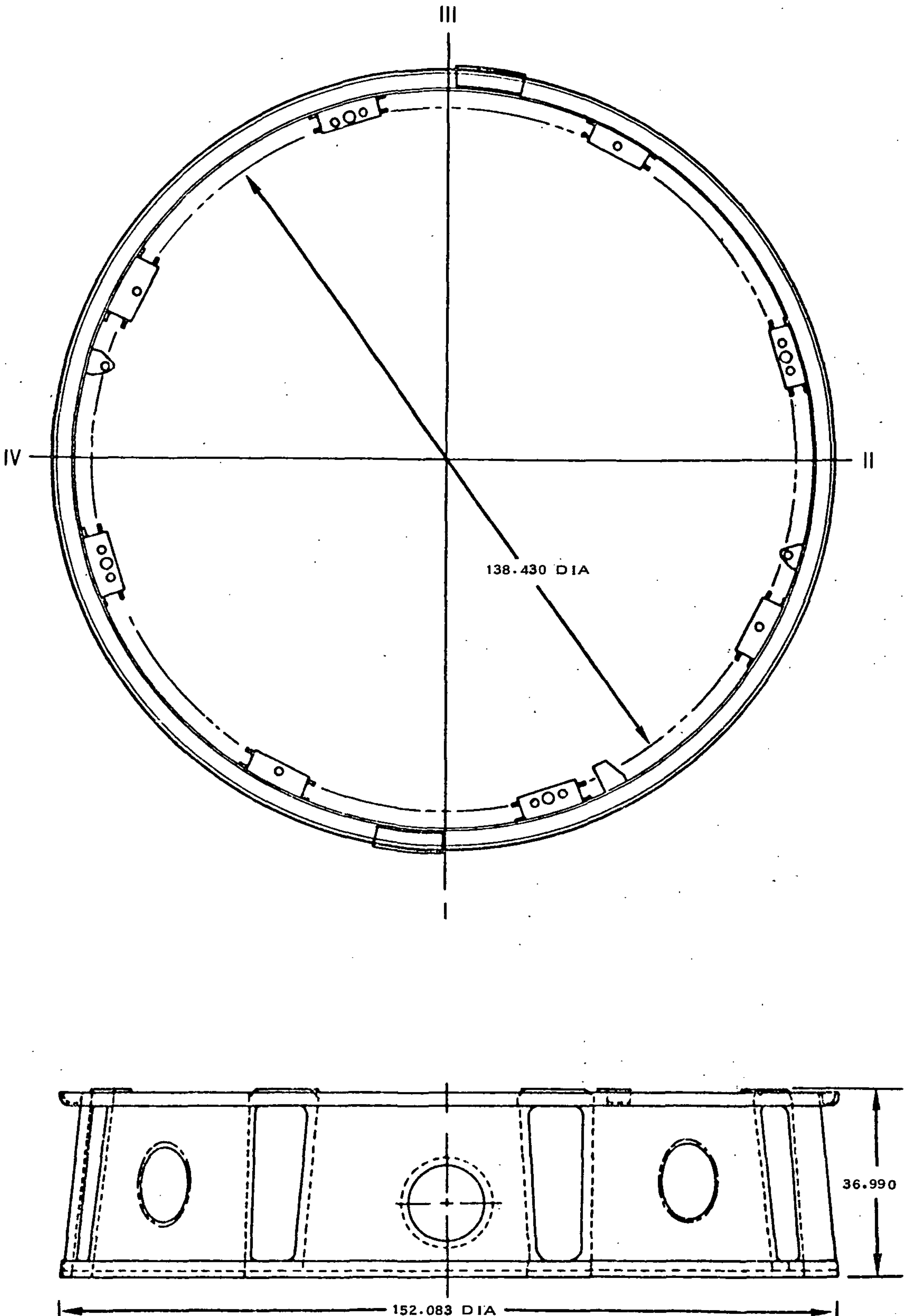


Figure 3.3-1. Delta Outboard Profile



Note: All dimensions in Centimeters

Figure 3.3-2. Payload Envelope, Two Stage, 5414 Attach Fitting



Note: All dimensions in Centimeters

Figure 3.3-3. 5414 Conical Attach Fitting Detailed Dimensions

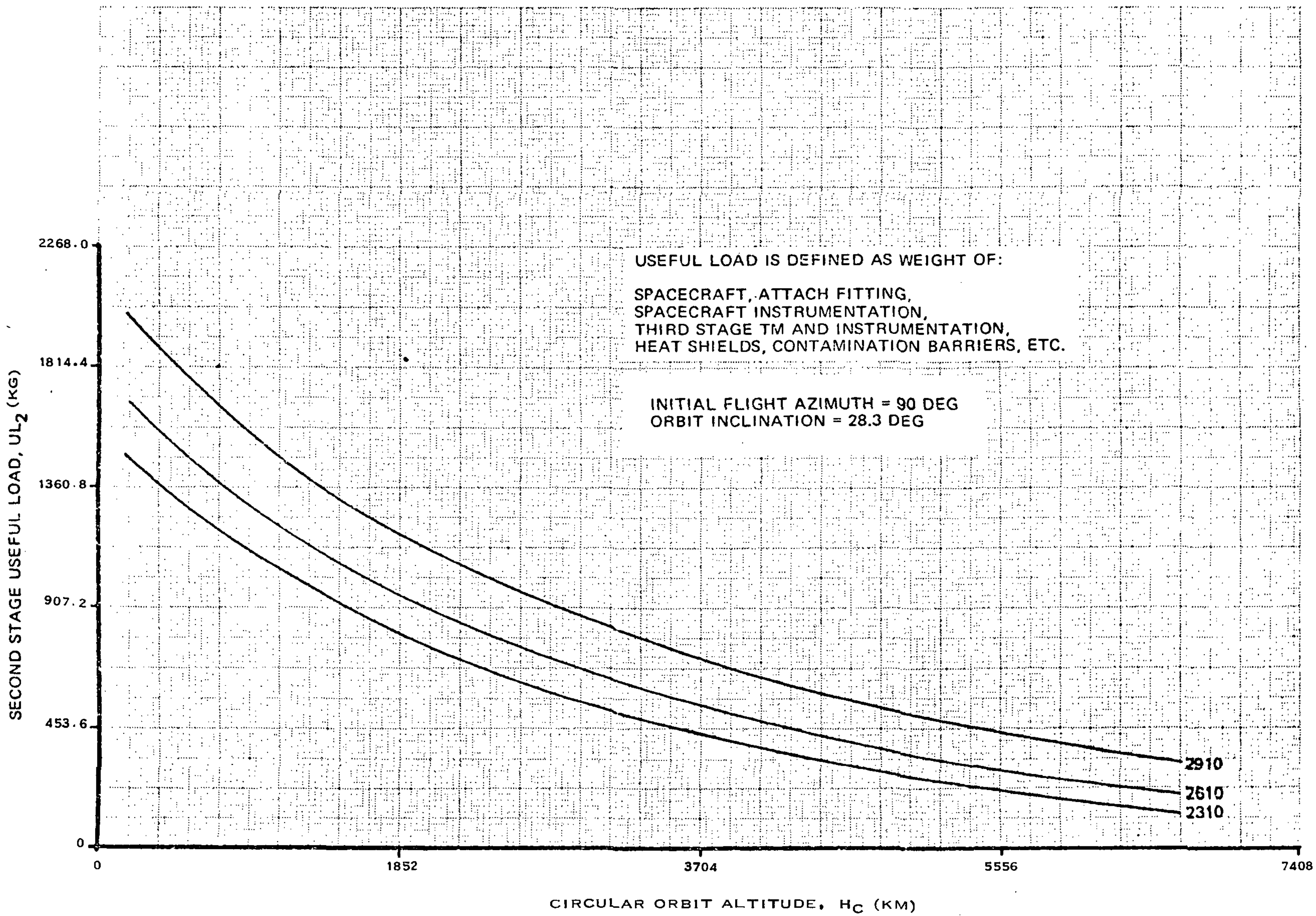


Figure 3.3-4. Delta 2910 Capabilities

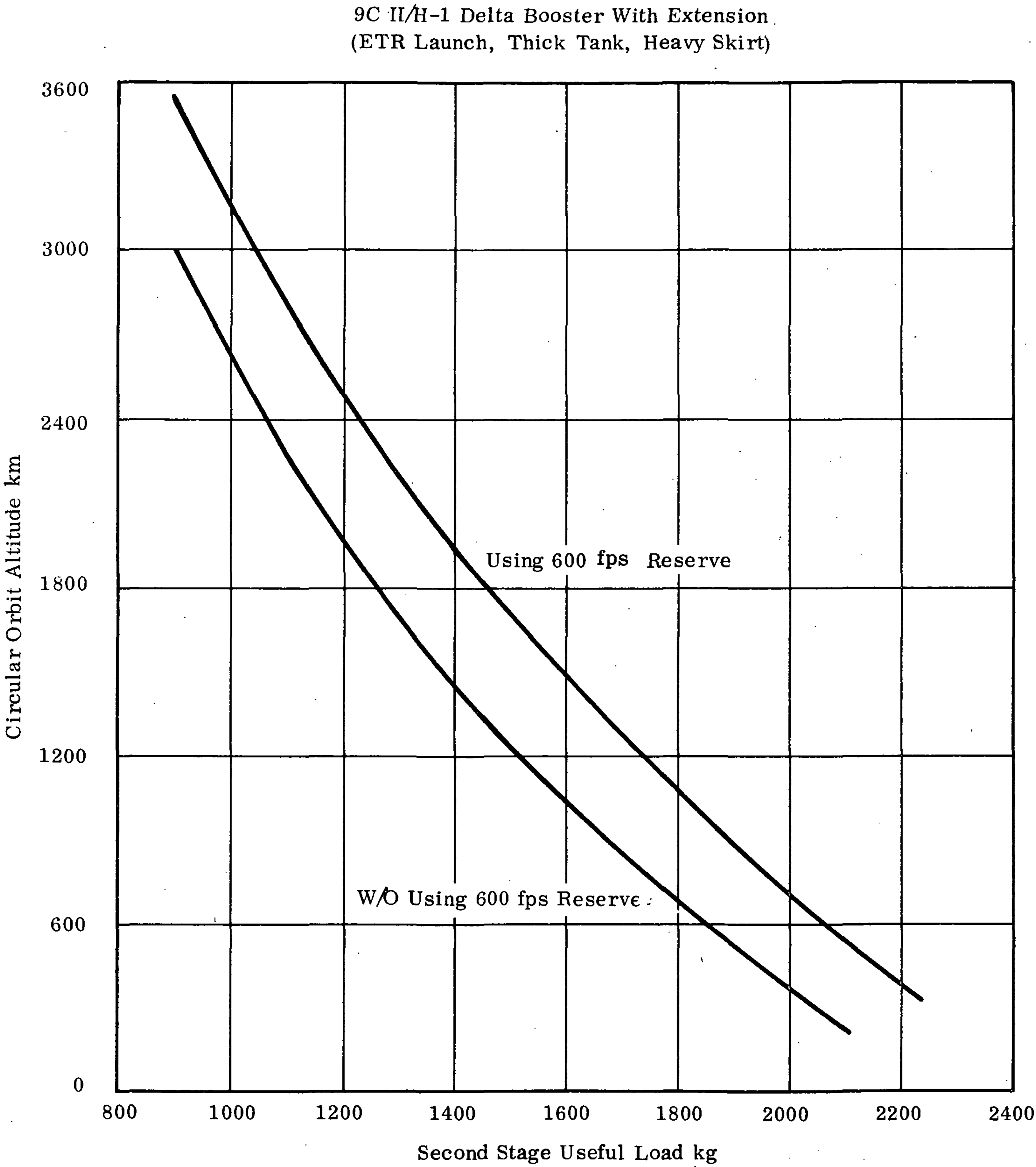


Figure 3.3-5. Delta 2910 Load vs. Altitude

to compute "time of ascent", "fuel consumed", and "fraction of solar array power remaining". A design nominal trajectory with a starting power of 13 kW, a spacecraft weight of 1130 kg (2500 lbs) and a parking orbit altitude which lies on the "burn to depletion" delta capability curve at 2700 km (1460 nm) has been selected and ascent trajectories for the starting powers, spacecraft weights and parking orbit altitudes surrounding this nominal initial state have been computed.

The matrix summarizing these computations is given in Table 3.3-1.

In selecting the actual initial conditions, it is useful to consider the sensitivities of the computed outputs (fuel consumption, array degradation and time of ascent) to changes in initial conditions. These sensitivities as determined from the summary matrix are as follows:

	Initial Power	Initial Altitude	Spacecraft Weight
Fuel Consumption	-0.4536 kg/kW	-0.0115 kg/km	+0.105 kg/kg
Array Degradation	-1.15% /kW	-0.0027 %/km	0.0032 %/kg
Ascent Time	-34.2 days/kW	-0.045 days/km	+ 0.335 days/kg

The minus sign indicates a decreasing output parameter with increasing initial condition. Array degradation is relatively insensitive to changes in initial conditions being most sensitive to changes in initial power. Fuel consumption is only moderately sensitive to change in initial condition being most sensitive to change in spacecraft weight. Ascent time is quite sensitive to initial condition being most sensitive to changes in initial power.

Examination of the Delta payload vs. altitude curve indicates that the sensitivity is -0.643 kg/km. The weight coefficient of the deployed, rotatable solar array is 19.1 kg per kW. These values in-

Table 3.3-1. Low Altitude Ascent Trajectory Summary

Weight of Fuel Consumed (Kilograms)

Initial Power (kW)	Initial Altitude (km)	Initial Weight (kg) 680	907	1134	1361
11 kW	4630	67.7	90.1	112.8	135.6
	3704	73.5	97.9	123.0	147.6
	2778	80.2	107.4	134.2	161.7
	1852	88.2	117.5	147.4	177.8
	1019	96.1	127.8	160.5	193.1
13 kW	4630	69.7	91.4	111.2	133.9
	3704	74.8	97.0	121.4	145.4
	2778	80.1	106.3	$\boxed{132.2}$	158.6
	1852	87.2	116.2	144.8	173.4
	1019	95.6	126.7	158.1	189.9
15 kW	4630	68.0	90.3	113.2	136.7
	3704	74.7	98.6	124.1	149.0
	2778	81.2	108.5	135.0	162.8
	1052	89.0	118.3	148.7	176.0
	1019	97.2	129.4	162.1	191.4

Ratio Of: Power Lost/Initial Power

Initial Power (kW)	Initial Altitude (km)	Initial Weight (kg) 680	907	1134	1361
11 kW	4630	.490	.515	.534	.548
	3704	.513	.537	.555	.569
	2778	.527	.550	.568	.582
	1852	.541	.564	.581	.595
	1019	.542	.565	.582	.595
13 kW	4630	.478	.503	.520	.535
	3704	.501	.524	.541	.555
	2778	.514	.537	$\boxed{.553}$	.567
	1852	.527	.550	.556	.580
	1019	.529	.551	.568	.581
15 kW	4630	.462	.488	.508	.524
	3704	.486	.511	.531	.546
	2778	.501	.526	.545	.559
	1852	.516	.541	.559	.571
	1019	.517	.542	.560	.572

Ascent Time (Days)

Initial Power (kW)	Initial Altitude (km)	Initial Weight (kg) 680	907	1134	1361
11 kW	4630	98.2	138.5	181.6	226.1
	3704	111.5	157.6	207.5	258.9
	2778	124.5	177.0	231.9	290.8
	1852	140.9	199.3	262.5	329.4
	1019	142.3	209.2	275.2	344.2
13 kW	4630	86.4	117.5	144.2	179.4
	3704	95.2	125.6	163.7	202.9
	2778	101.4	140.5	$\boxed{152.1}$	226.5
	1852	112.3	157.8	205.0	254.9
	1019	119.3	166.2	215.9	268.1
15 kW	4630	68.5	96.8	127.4	160.1
	3704	78.9	110.8	146.6	183.2
	2778	87.9	125.1	163.5	205.3
	1852	99.3	140.6	185.8	221.1
	1019	105.1	148.6	195.4	231.4

dicate that little is to be gained by increasing the parking orbit altitude and that increase in initial power will have a significant effect on only the ascent time and the actual available "start of mission" power. The net effect however of lowering parking orbit altitude to increase spacecraft payload capabilities (and hence spacecraft weight) will be 0.52 kg/km after the effects on solar array degradation and fuel consumption are considered. Lowering the parking orbit altitude and increasing the weight as discussed would increase the ascent time by almost a month.

The previous discussion refers to the problem of raising the orbital altitude from the nominal design parking orbit to synchronous altitude without an orbit plane change. For most launch vehicles and launch sequences, a plane change of 28.5° would also be required to produce the desired synchronous equatorial orbit. This could be accomplished after ascent to synchronous altitude but would extend the time required to attain the mission orbit and would result in a significant additional expenditure of fuel. A savings in both time and fuel consumption can be achieved by maneuvering the spacecraft during ascent to provide a component of thrust normal to the trajectory plane.

3.3.3 LAUNCH SEQUENCES

Typical launch ascent sequences for the two-stage Delta launch vehicle are tabulated in Table 3.3-2 and illustrated in Figure 3.3-6. The computer program used to determine the orbit raising ascent trajectories was given in Appendix C of the interim report together with an outl ine of the assumptions and relationships used in the computation. Typical curves of spacecraft altitude vs. normalized ascent time and ion engine thrust vs. normalized ascent time was given in Figure 3.3-7 and 3.3-8. A tabulation of synchronous equatorial orbit injection errors is not appropriate since both the

Table 3.3-2. Typical Sequences of Events for Two Stage Missions

Events	Time (Seconds) 2 Stage Delta
Solid Motor Ignition	T + 0
Liftoff Signal to Guidance System	T + 0
Solid Motor Burnout	T + 38
Solid Motor Separation	T + 95
Main Engine Cutoff - MECO (M)	T + 219
Blow Stage I/II Separation Bolts	M + 8
Start Stage II Ignition	M + 12
Fairing Separation	M + 48
Second Stage Engine Cutoff, Command No. 1 - SECO 1 (S1)	M + 328
Stage 2 Engine Re-Start	S1 + 3044
Stage 2 Engine Cutoff, Command No. 2 - SECO 2 (S2)	S1 + 3057

ATS-AMS I

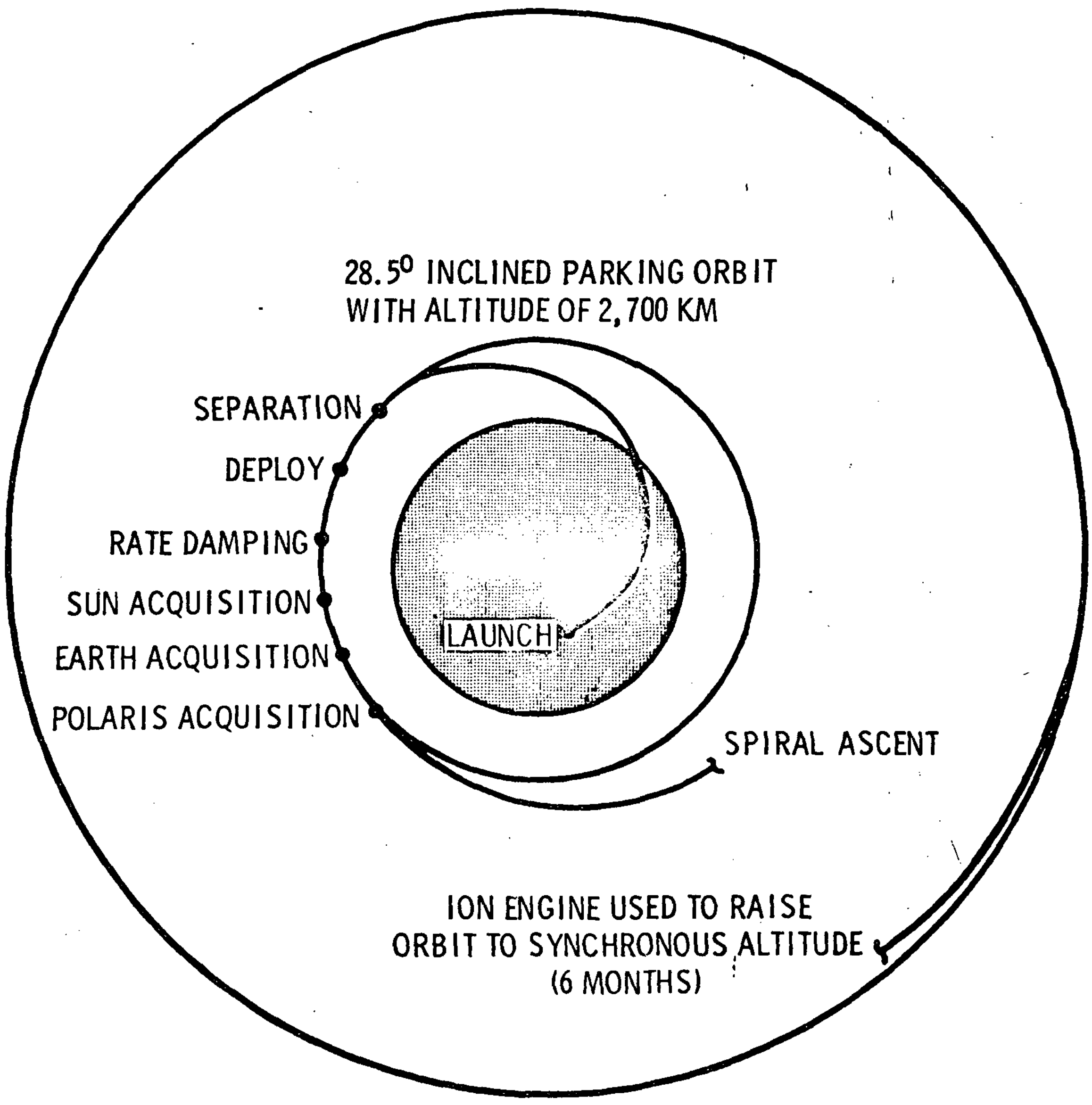


Figure 3.3-6. During the Final Stage of the Spiral Ascent, the Necessary Adjustment to Obtain Final Orbit Position Will Be Made

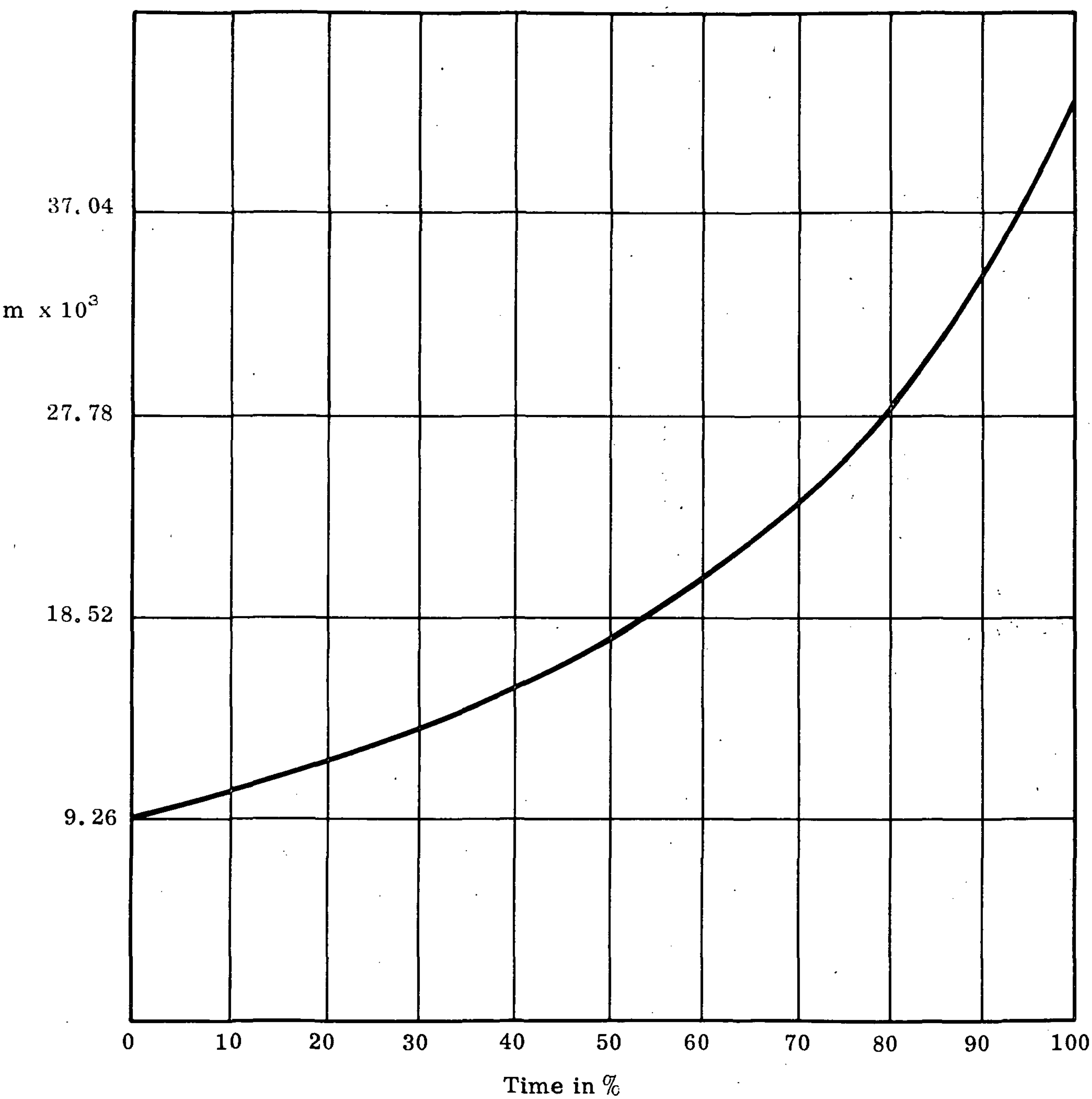


Figure 3.3-7. Typical Orbit Radius vs. Time

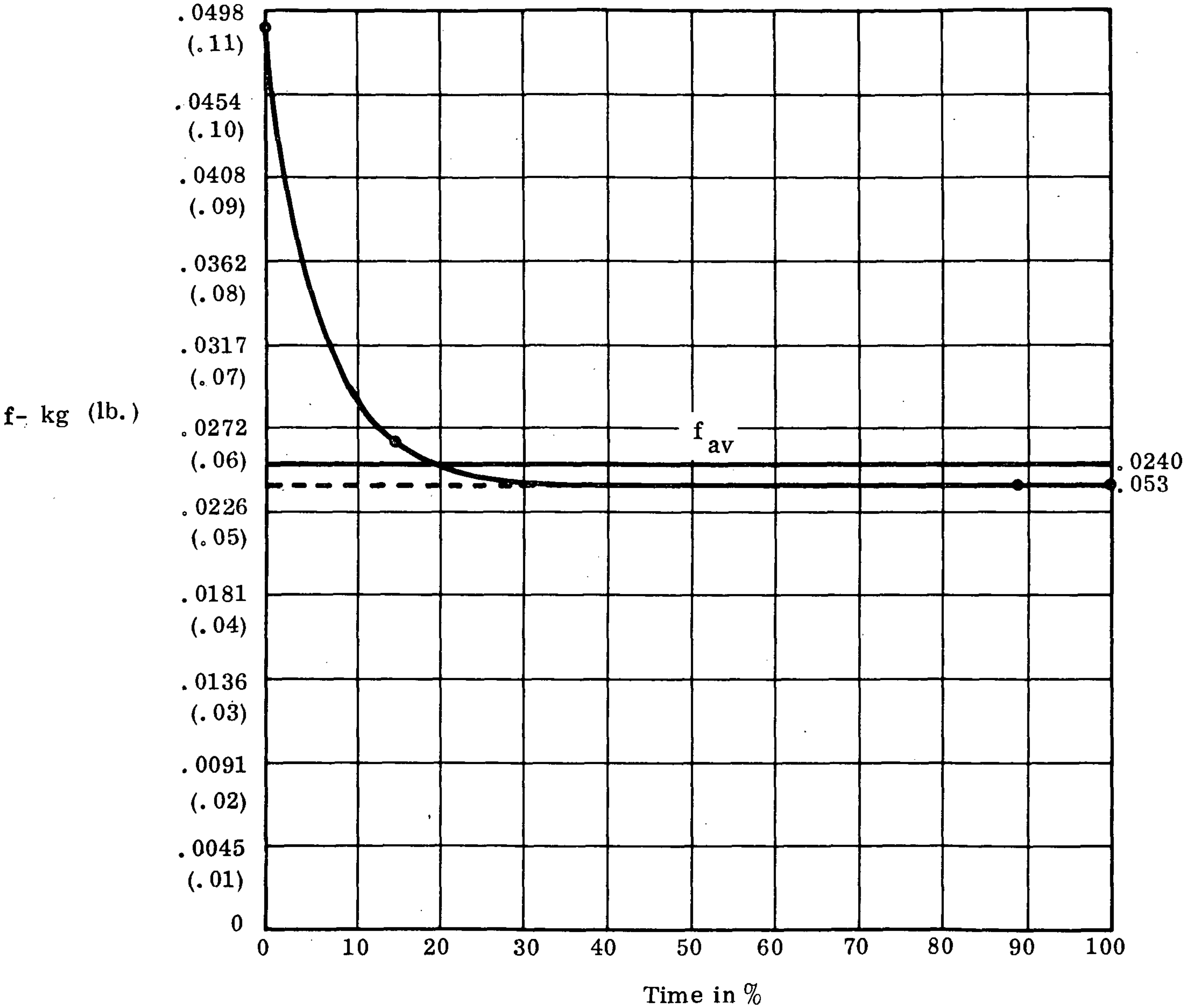


Figure 3.3-8. Typical Thrust Profile

altitude and thrust level of the ion engines can be commanded during the orbit raising phase of the ascent mission to bring the spacecraft to its exact station. The ascent phase of the mission would for this launch vehicle combination require approximately 6 months.

SECTION 4

SPACECRAFT DESCRIPTION

4.1 SUMMARY OF SPACECRAFT CHARACTERISTICS

The preliminary spacecraft and subsystem designs based upon the approaches for accomplishing the mission objectives are described in this Section. For convenience in referencing throughout the report, a designator has been assigned to each of the models as follows:

- ATS-AMS III A - A 1,725 kg, (3800 lb) Titan III C/Growth Burner II - launched, sun oriented spacecraft having fixed solar arrays which ascends directly to synchronous orbit. It incorporates all of the required operational experiments and technological experiments - including antennas producing contoured beam patterns. The antennas, power amplifiers and transponders are mounted on the earth pointing rotating antenna tower. Also included is sufficient capacity for about 500 kg (1100 lb) of undefined experiments.

- ATS-AMS III B- A 1,485 kg (3270 lb) version of the ATS-AMS III A having a lesser communication capability. It is launched on a Titan III C such as those currently being produced. The ATS-AMS IIIA and B are basically the same satellite. For a two-launch program, it might be desirable to launch the smaller III B first, followed by the larger III A.

- ATS-AMS II - A 783 kg (1,723 lb) Atlas/Centaur/Burner II - launched, sun oriented spacecraft generically similar to ATS-AMS III. Although the updating would not be overly difficult, this spacecraft was designed prior to both the 1971 World Administrative Conference (WARC) and Contract Amendment Number 1 and therefore incorporates 12.88 - 13.25 GHz uplink and antennas producing CONUS, time zone, half time zone and spot pattern beams.
- ATS-AMS I - A 1,167 kg (2,567 lb) Delta 2910 - launched, earth oriented spacecraft having rotating solar arrays which achieves synchronous orbit by utilizing three ion engines to raise the spacecraft from a lower parking orbit. This spacecraft design also preceded the WARC and Contract Amendment Number 1.

The significant characteristics for these four spacecraft are summarized by the Tables 4.1-1 (ATS-III B), 4.1-2 (AMS-IIIA), and 4.1-3 (AMS-II)and 4.1-4 (AMS-I).

The weight budget summary is stated in Table 4.1-5. It is important to note the portion of the budget allocated to the structure. For spacecraft of this type, normally 14 to 20% of the total is required for the structure. ATS-AMS III A and III B have been sized at 16.5% and 18% respectively, being basically the same design. However, ATS-AMS I and II could only be allocated about 7.4% and 12.7% of the total respectively. This would required an extremely difficult and possibly marginal structural design unless the allocations for the subsystems were reduced.

Table 4.1-1. ATS-AMS III B Spacecraft Characteristics (Sheet 1 of 3)

General

o Baseline Information Networking Experiments	1. PBS Networking
	2. Special Interest Areas
	3. Rainfall Attenuation
	4. Cultural Regions ITV
	5. Alaska Medical
o Technology Experiments	1. Shaped multibeam antenna contoured beam patterns
	2. Generate 2 kW Microwave Power
	3. High voltage power generation and conditioning
	4. Liquid metal slip rings
	5. Vectorable ion attitude engines
	6. High power temperature control
o Orbit	1. Geostationary, 110° West Longitude
	2. Sun oriented
	3. Three-Axis stabilized
	4. Rotating antenna, transponder and PA tower

o Launch	1. Titan III	
	2. 1976 ETR	
	3. Direct Ascent	
	4. Transtage injection	
o Life	1. Two years in orbit (Five-year design life)	
o Weight - kilograms - (lbs)	1. Structure	268 (590)
	2. Subsystems	876 (1,927)
	3. Undefined Experiments and Contingency	341 (706)
	Total	1,485 (3,270)
o Power Demand-Watts	1. Peak	5,400 kW
o Dimensions - centimeters (ft)	1. Folded Diameter	274 (9)
	2. Length	609 (20)
	3. Span	3,640 (111)
	4. Equipment	183 x 183 x 183 (6 ft cube)

Table 4.1-1. ATS-AMS III B Spacecraft Characteristics (Sheet 2 of 3)

Communications		Power	
• Antenna - centimeters (ft)	1. One 344 cm (11.3 ft) parabolic reflector, switchable multiple offset feeds, up to six shaped beams 2. One 22.85 x 51.8 (9 x 20.4 inch) parabolic reflector	• Photovoltaic Array	1. 2 extended articulated panels 2. Panel on portion of sun side of equipment module
• Frequency Band	1. 11.7 - 12.2 GHz down link 2. 14.0 - 14.5 GHz up link	• Power Available	1. 7.5 kW solar array SOL 2. 6.1 kW solar array EOL 3. 36 Ah Battery
		• Total Array Area - square meters (ft^2)	1. 68 (730)
• Power Amplifiers 50% efficiency or greater	1. One 1 kW Klystron (experiment) 2. One 2 kW TWT-MDC (experiment) 3. Five 200 W TWT - MDC 4. Five 5 W TWT	• Array Voltages	1. 67 V 2. Biased 5, 10, 15 kV (experiment)
		• Conditioned Voltages	1. 1 to 15 kV 2. 30.5 V
		• Energy Storage for Periods of Solar Eclipse	1. Housekeeping functions 2. PA cathode heaters at 50% nominal power
• Transponders - Angle modulation format	1. Simultaneous linear translation of up to 5 independent 23 MHz video/ voice/ data channels; 4 D/L frequencies, 5 V/L frequencies 2. Simultaneous linear translation of up to 190 independent 44 KHz voice/ data channels or 30,000 data channels	• Insulation	1. Space vacuum 2. Kapton - external wiring 3. Teflon-internal wiring

Table 4.1-1. ATS-AMS III B Spacecraft Characteristics (Sheet 3 of 3)

Control		Thermal	
• Antenna Beam Pointing Sub-Satellite Point	1. ± 0.2° for latitude and longitude 2. ±0.2° in rotation about the boresight axis 3. Five years	• Waste Heat Rejection	1. Over 5.0 kW
		• High Power Amplifiers	1. Open envelope operation 2. Direct radiation to space
• Attitude Accuracy	1. Better than 0.03 degrees in 3 axis up to 10 degrees off local vertical	• Control Techniques	1. Heat Pipes 2. Louvers 3. Thermal Coatings 4. Superinsulation
• Yaw (N-S) Reference	1. Interferometer primary 2. Polaris sensor backup	• Equipment Module Environment	1. Average $293^{\circ}K \pm 10^{\circ}$ 2. Mounting surface $293^{\circ}K \pm 15^{\circ}$
• Pitch and Roll Reference	1. Interferometer primary 2. Horizon sensor backup	**TT&C**	
• Attitude Control Actuators	1. Momentum wheels 2. (2) 5 cm ion engines 3. Hydrazine thruster backup	• Telemetry	1. Stadan compatible
		• Command	1. 1536 discrete commands
• Repositioning Capability - meters/second	1. ΔV 30.48 (100 ft/sec) minimum		
• Stationkeeping	1. 0.2 degrees N-S, E-W for five years		
• Orbit Control Actuator	1. 30 cm ion engine N-S 2. Hydrazine thruster E-W		

Table 4.1-2. ATS-AMS III A Spacecraft Characteristics (Sheet 1 of 3)

General

• Baseline Information Networking Experiments	1.	PBS Networking
	2.	Special Interest Areas
	3.	Rainfall Attenuation
	4.	Cultural Regions ITV
	5.	Information Networking
• Technology Experiments	1.	Shaped multibeam antenna contoured beam patterns
	2.	Generate 2 kW Microwave Power
	3.	High voltage power generation and conditioning
	4.	Liquid metal slip rings
	5.	Vectorable ion attitude engines
	6.	High power temperature control
• Orbit	1.	Geostationary, 110° West Longitude
	2.	Sun oriented
	3.	Three-Axis stabilized
	4.	Rotating antenna, transponder and PA tower

• Launch	1.	Titan III C/Growth Burner II	
	2.	1976 ETR	
	3.	Direct Ascent	
	4.	Transtage/Growth Burner II injection	
• Life	1.	Two years in orbit (Five-year design life)	
• Weight-kilograms - (lbs)	1.	Structure	284 (627)
	2.	Subsystems	1,089 (2,396)
	3.	Undefined Experiments	352 (774)
	4.	Total	1,725 (3,800)
• Power Demand-Watts	1.	6,930 peak (EOL)	
• Dimensions - centimeters (ft)	1.	Folded Diameter	274 (9)
	2.	Length	609 (20)
	3.	Span	4,250 (130)
	4.	Equipment	183 x 183 x 183 (6 ft cube)

Table 4.1-2. ATS-AMS III A Spacecraft Characteristics (Sheet 2 of 3)

Communications		Power	
• Antenna - centimeters	1. One 344 cm(11.3 ft) parabolic reflector, switchable multiple offset feeds, up to six shaped beams	• Photovoltaic Array	1. 2 extended articulated panels 2. Panel on portion of sun side of equipment module
	2. One 22.85 x 51.8 (9 x 20.4 inch) parabolic reflector	• Power Available	1. 9.6 kW solar array SOL 2. 7.8 kW solar array EOL 3. 48 Ah battery
• Frequency Band	1. 11.7 - 12.2 GHz down link	• Total Array Area - square meters (ft^2)	1. 87 (810)
• Power Amplifiers	1. One 2 kW Klystron (experiment) 2. One 2kW TWT-MDC (experiment) 3. Ten 100 to 200 W variable TWT-MDC 4. Ten 5 W TWT	• Primary Voltages	1. 67 V 2. Biased 5, 10, 15 kV (experiment)
		• Conditioned High Voltages	1. 1 to 15 kV 2. 30.5 V
		• Energy Storage for Periods of Solar Eclipse	1. Housekeeping functions 2. PA cathode heaters at 50% nominal power
• Transponders	1. Simultaneous linear translation of up to 10 independent 23 MHz video/voice/data channels; 6 frequencies 2. Simultaneous linear translation of up to 150 independent 44 KHz voice/data channels or 50,000 data channels	• Insulation	1. Space vacuum 2. Kapton-external wiring 3. Teflon-internal wiring

Table 4.1-2. ATS-AMS III A Spacecraft Characteristics (Sheet 3 of 3)

Control		Thermal	
• Antenna Beam Pointing Sub-Satellite Point	1. $\pm 0.2^{\circ}$ for latitude and longitude 2. $\pm 0.2^{\circ}$ in rotation about the boresight axis 3. Five years	• Waste Heat Rejection	1. Over 5.0 kW
• Attitude Accuracy	1. Better than 0.03 degrees in 3 axis up to 10 degrees off local vertical	• High Power Amplifiers	1. Open envelope operation 2. Direct radiation to space
• Yaw (N-S) Reference	1. Interferometer primary	• Control Techniques	1. Heat Pipes 2. Louvers 3. Thermal Coatings 4. Superinsulation
• Pitch and Roll Reference	1. Interferometer primary 2. Horizon sensor backup	• Equipment Module	1. Average $290^{\circ}K \pm 10^{\circ}$ 2. Mounting surface $293^{\circ} \pm 15^{\circ}$
• Attitude Control Actuators	1. Momentum wheels 2. (2) 5 cm ion engines 3. Hydrazine thruster backup	**TT&C**	
• Repositioning Capability-meters/second	1. ΔV 30.48 (100 ft/sec) minimum	• Telemetry	1. Stadan compatible
• Stationkeeping	1. 0.2 degrees N-S, E-W for five years	• Command	1. 1536 discrete commands
• Orbit Control Actuator	1. 1 or 3 30 cm ion engines N-S 2. Hydrzaine thruster E-W		

Table 4.1-3. ATS-AMS II Spacecraft Characteristics (Sheet 1 of 3)

General

- **Baseline Information Networking Experiments**
  1. ETV to small terminals
  2. Bush communication
  3. Information distribution
  4. Video exchange through small terminals
  5. Information Networking
- **Technology Experiments**
  1. 2 kW microwave power
  2. High voltage solar array
  3. Liquid metal slip rings
  4. Vectorable ion attitude engines
  5. High power temperature control
- **Orbit**
  1. Geostationary
  2. Sun oriented
  3. Three-Axis stabilized
  4. Rotating antenna, transponder and PA tower
- **Launch**
  1. SLV 3D/Centaur D-1A/BII
  2. 1976 ETR
  3. Direct Ascent
  4. Burner II injection
- **Life**
  1. Two years in orbit (Five-year design life)
- **Weight - kilograms - (lbs)**
  1. Structure 98 (216)
  2. Subsystems 685 (1507)
  3. Total 783 (1723)
- **Power Demand - Watts**
  1. Peak 3,900
- **Dimensions-centimeters (ft)**
  1. Folded Diameter 293 (9.6)
  2. Length 458 (15)
  3. Span 2,880 (94.5)
  4. Experiment Module 112 x 132 x 127 (44 x 52 x 50 in)

Table 4.1-3. ATS-AMS II Spacecraft Characteristics (Sheet 2 of 3)

Communications		Power	
• Antenna - centimeters	1. One 122 x 244 (4 x 8 feet) elliptical parabolic cassegrain, 15 feeds, 4 beams 2. One 76.2 (2.5 feet) diameter circular parabolic cassegrain, 6 feeds, 1 beam	• Photovoltaic Array	1. 2 extended fixed panels
		• Power Available	1. 5.5 kW solar array SOL 2. 4.45 kW solar array EOL 3. 42 Ah battery
• Frequency Band	1. 11.7 - 12.2 GHz downlinks 2. 12.88 - 13.25 GHz uplinks	• Total Array Area - meters	1. 69.8 (649 sq. ft.)
• Power Amplifiers 50% efficiency or greater	1. One 1kW Klystron (experiment) 2. One 2 kW TWT-MDC (experiment) 3. Four 200 W TWT-MDC	• Primary Voltages	1. 30.5 V
		• Conditioned High Voltages	1. 1 to 16 kV 2. Biased 5, 10, 15 kV (experiment)
• Transponders - Angle modulation format	1. Simultaneous linear translation of up to 4 independent video/voice/data channels	• Energy Storage for Periods of Solar Eclipse	1. Housekeeping functions 2. PA cathode heaters at 50% nominal power
		• Insulation	1. Space vacuum 2. Kapton - external wiring 3. Teflon - internal wiring

Table 4.1-3. ATS-AMS II Spacecraft Characteristics (Sheet 3 of 3)

Characteristic		Value
Control		
• Antenna Beam Pointing Sub-Satellite Point	1.	± 0.2° for latitude and longitude
	2.	± 0.2° in rotation about the boresight axis
	3.	Five years
• Attitude Accuracy	1.	Better than 0.03 degrees in 3 axis up to 10 degrees off local vertical
• Yaw (N-S) Reference	1.	Interferometer primary
	2.	Polaris sensor backup
• Pitch and Roll Reference	1.	Interferometer primary
	2.	Horizon sensor backup
• Attitude Control Actuators	1.	Momentum wheel
	2.	(2) 5 cm ion engines
	3.	Hydrazine thruster backup
• Repositioning Capability - meters/second	1.	ΔV 30.48 (100 ft/sec) minimum
• Stationkeeping	1.	0.2 degrees N-S, E-W for five years
• Orbit Control Actuator	1.	(3) 30 cm ion engines
Thermal		
• Waste Heat Rejection	1.	Over 3.8 kW
• High Power Amplifiers	1.	Open envelope operation
• Control Techniques	1.	Heat Pipes
	2.	Louvers
	3.	Thermal Coatings
	4.	Superinsulation
• Equipment Module Environment	1.	Average $293^{o}K \pm 10^{o}$
TT&C		
• Telemetry	1.	Stadan compatible
• Command	1.	1536 discrete commands

Table 4.1-4. ATS-AMS I Spacecraft Characteristics (Sheet 1 of 3)

General

• Baseline	1. ETV to *small terminals* 2. Bush communication 3. Information distribution 4. Video and high speed data exchange through small terminals 5. Tracking &	• *Launch*	1. *Delta 2910/Ion engines* 2. 1976 ETR 3. Spiral ascent from 2780 km (1500 n. mi) parking orbit
		• Life	1. Two years in orbit (Five-year design life)
• Technology Experiments	1. 2 kW microwave power 2. *High voltage solar array* 3. Liquid metal slip rings 4. Vectorable ion attitude engines 5. High power temperature control	• Weight - kilograms - (lbs)	1. Structure 84 (185) 2. *Subsystems* *1083* (2383) 3. Total 1167 (2567)
		• *Power Demand-* Watts	1. *Peak* 7267
• Orbit	1. Synchronous equatorial 2. Earth oriented 3. Three-axis stabilized 4. Rotating solar array	• Dimensions - centimeters	1. Folded 218 (7 ft 2 in) 2. Length 396 (13 ft) 3. Span 2660 (87.5 ft.) 4. Equipment Module 201 x 201 x 178 (79 x 79 x 70in)

Table 4.1-4. ATS-AMS I Spacecraft Characteristics (Sheet 2 of 3)

Communications	
o Antenna - centimeters	1. One 122 x 244 (4 x 8 feet) elliptical parabolic cassegrain, 15 feeds, 4 beams
	2. One 76.2 (2.5 feet) diameter circular parabolic cassegrain, 6 feeds, 1 beam
o Frequency	1. 11.7 - 12.2 GHz down link
	2. 12.88 - 13.25 GHz up link
o Power Amplifiers 50% efficiency or greater	1. One 1 kW Klystron (experiment)
	2. One 2 kW TWT - MDC(experiment)
	3. Four 200 W TWT-MDC
o Transponders - Angle modulation format	1. Simultaneous linear translation of up to 4 independent video/voice/data channels

Power	
o Photovoltaic Array	1. 2 rotating double panels
o Power Available	1. 13 kW solar array SOL
	2. 5.5 kW solar array EOA
	3. 4.45 kW solar array EOL
	4. 87 Ah battery
o Total Array Area - square meters	1. 144 (1340 square feet)
o Primary Voltages	1. 30.5 V to 16 kV
o Energy Storage for Periods of Solar Eclipse	1. Housekeeping functions
	2. PA cathode heaters at 50% nominal power
o Insulation	1. Space vacuum
	2. Kapton - external wiring
	3. Teflon - internal wiring

Table 4.1-4. ATS-AMS I Spacecraft Characteristics (Sheet 3 of 3)

Control		
• Antenna Beam Pointing Sub-Satellite Point	1.	± 0.2° for latitude and longitude
	2.	± 0.2° in rotation about the boresight axis
	3.	Five years
• Attitude Accuracy	1.	Better than 0.03 degrees in 3 axis up to 10 degrees off local vertical
• Yaw (N-S) Reference	1.	Interferometer primary
	2.	Polaris sensor backup
• Pitch and Roll Reference	1.	Interferometer primary
	2.	Horizon sensor backup
• Attitude Control Actuators	1.	Momentum wheel
	2.	(2) 5 cm ion engines
	3.	Hydrazine thruster backup
• Repositioning Capability - meters/second	1.	ΔV 30.48 (100 ft/sec) minimum
• Stationkeeping	1.	0.2 degrees N-S, E-W for five years
• Orbit Control Actuator	1.	30 cm ion engine

Thermal		
• Waste Heat Rejection	1.	Over 2.7 kW
• High Power Amplifiers	1.	Open envelope operation
	2.	Direct radiation to space
• Control Techniques	1.	Heat Pipes
	2.	Louvers
	3.	Thermal Coatings
	4.	Superinsulation
• Equipment Module Environment	1.	Average $293^{o}K \pm 10^{o}$
	2.	Mounting surface $293^{o}K \pm 15^{o}$
TT&C		
• Telemetry	1.	Stadan compatible
• Command	1.	1536 discrete commands

Table 4.1-5. Weight Budget Summary
kg (lbs)

	AMS-IIIB		AMS-IIIA		AMS-II		AMS-I	
ANTENNA & FEEDS	106	(234)	107	(235)	541	(125)	54	( )
COMMUNICATION S/S	168	(370)	288	(635)	118	(256)	118	(256)
POWER & DISTRIBUTION S/S	305	(670)	392	(860)	233	(512)	448	(986)
ATTITUDE CONTROL S/S	183	(403)	183	(403)	*170	(370)	**109	(240)
ORBIT CONTROL S/S	34	(75)	34	(75)	34	(75)	227	(500)
THERMAL CONTROL S/S	45	(100)	50	(110)	28	(62)	62	(136)
T&C	35	(76)	35	(76)	35	(76)	39	(85)
STRUCTURES	268	(590)	284	(627)	98	(216)	84	(185)
UNDEFINED EXPERIMENTS AND CONTIGENCIES	341	(706)	352	(774)	-		-	
TOTALS	1485	(3270)	1725	(3800)	770	(1700)	1143	(2515)

*Includes 28 kgs of backup hydrazine propellent

**Does not include backup hydrazine propellent

The spacecraft subsystem descriptions are organized into the following subsections:

4.2 SPACECRAFT CONFIGURATION

4.2.1 GENERAL DESCRIPTION - ATS-AMS III

The ATS-AMS III - A consists of a cube shaped equipment module having a rotatable platform on its forward face, a cluster of three 30 cm ion engines on its aft face and extendable body fixed solar panels on the two faces which are normally parallel to the orbit plane. The spacecraft is attached to the Titan IIIC by an adapter section of four sets of tubular trusses. These trusses will contain provisions for guiding the spacecraft during separation from the booster. The concept shown is similar to the system used on ATS-F. The spacecraft is 660.2 cm (256.0 inch) high and will fit into a 274.32 cm (108 inch) diameter payload envelope. A sketch of the spacecraft is shown in Figure 1.3.2-1 and an overall layout is presented in Figure 4.2-1.

4.2.1.1 Equipment Module

The equipment module cube measures 177.8 cm (70.0 inches) along all edges. The internal structure is a tubular center section which contains four fittings for attaching the spacecraft to the adapter sections. The upper and lower sections consists of extruded angle framework with machined fittings at the corners. The top section of the module contains a cylindrical section which houses the bearings and the cylindrical portion of the rotating platform support structure. This support structure consists of an outer channel ring section which is attached to the cylindrical portion by a series of tapered bulkheads. This structure will also contain provisions for securing the rotating platform during launch. Attached to the center section structure and extending into the lower module section is a tubular framework which is provided for attaching one or three 30 cm ion engines. The outside of the module will be covered with honeycomb sandwich panels. Several of these panels will be removable for equipment access. Although it is anticipated that for thermal reasons, the majority of the equipment

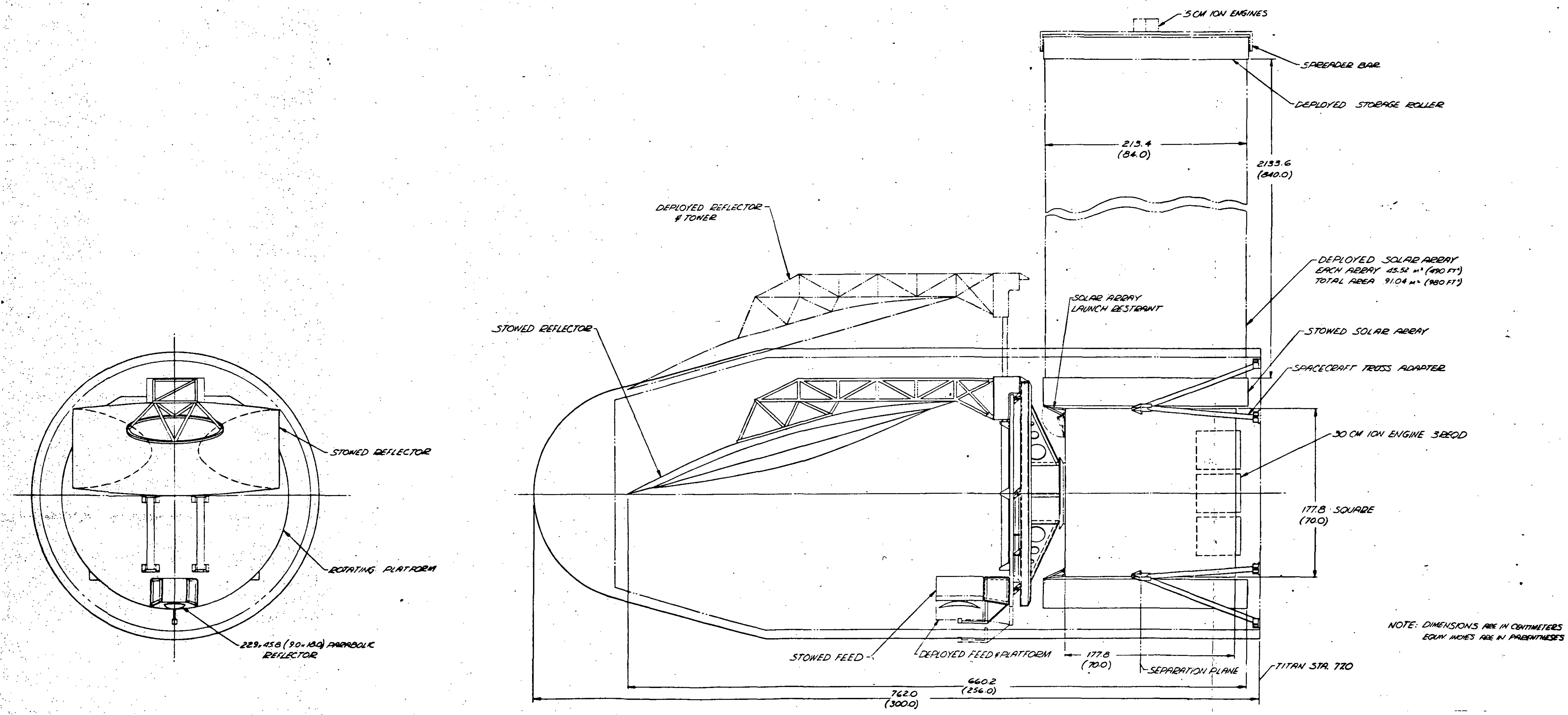


Figure 4.2-1. ATS-AMS IIIA Sun Oriented Spacecraft 25 Ft. Fairing Titan Booster

will be mounted to the module side panels additional shelves could be added to the upper and lower side of the center section structure. See Figure 4.2-2.

4.2.1.2 Rotating Platform

The rotating platform consists of a reinforced honeycomb panel which contains all of the communication equipment. The platform is secured to the support structure during launch by a series of shear fittings and explosive devices. Auxiliary framework structure to support super insulation blankets can be added to the platform to thermally control the electronic equipment. Also attached to this platform are the main antenna feeds. The reflector and its supporting tower are attached to the support structure through shear fittings and explosive release devices. The 344.42 cm (135.6 inch) diameter parabolic reflector is constructed of honeycomb panels supported on radial ribs. The tower is a welded tubular truss with outrigger structure near the top to provide a mounting surface for the antenna. Once the spacecraft is in orbit, the reflector and tower are deployed outward 110.49 cm (43.5 inches) through a telescopic tubing arrangement. The platform with the feed package and the communication equipment is then deployed in the opposite direction approximately 20.32 cm (8.0 inches). The reason for the deployment of both sections is twofold: first to achieve the proper focus between the feed and the reflector and second to maintain proper balance of the spacecraft. It is anticipated that a motor driven screw jack will be used to drive both the platform and the reflector and tower to their deployed positions. A similar mechanism will be used to tilt the platform thereby accurately aligning the antenna if required.

4.2.1.3 Solar Arrays

The details of the solar array design are presented in Section 4.5. A basic design goal was not to exceed a 10 to 1 aspect ratio of length to width of the array. Each of the two arrays are 213.4 cm (84.0 inches wide by 2133.6 cm (840.0 inches long) and provide for 45.52 m^2

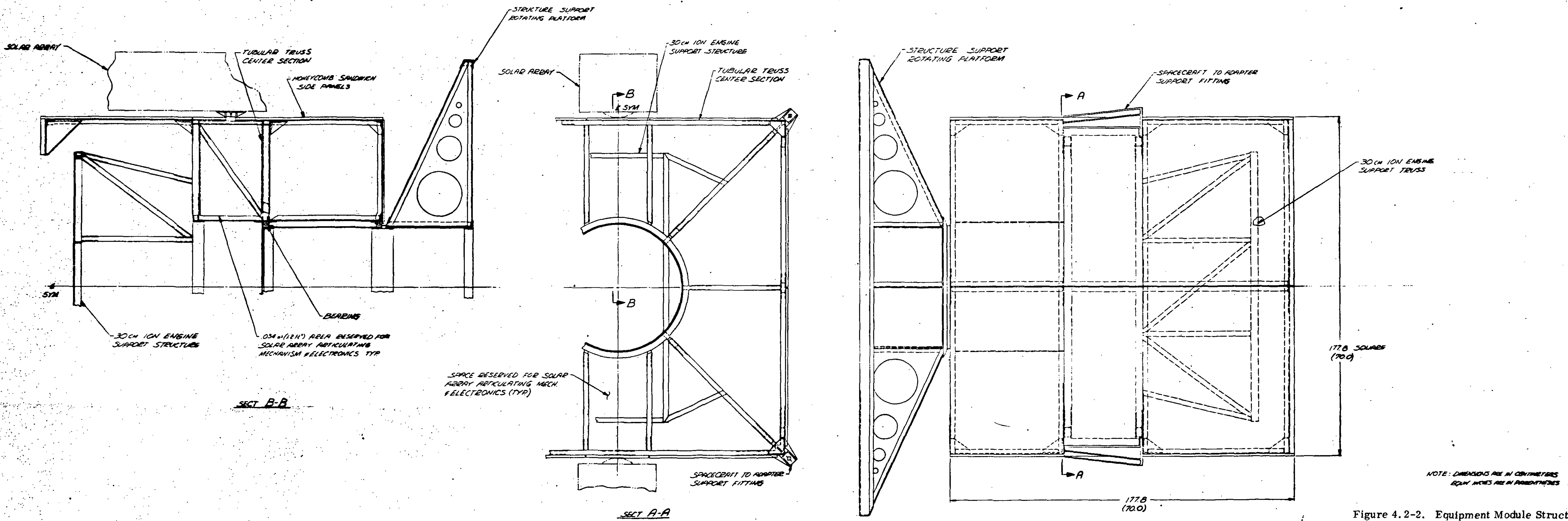
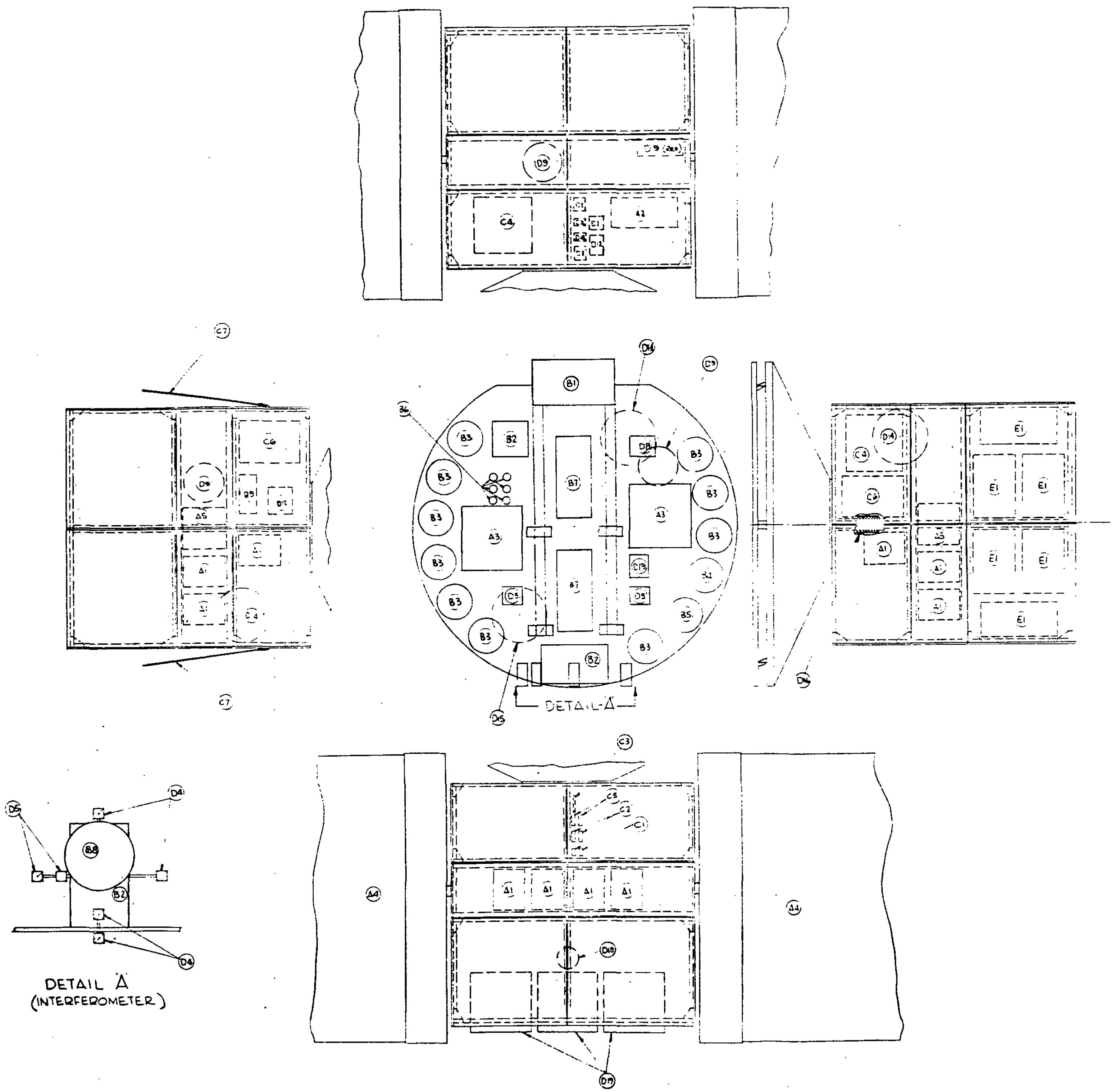


Figure 4.2-2. Equipment Module Structure ATS-AMS III

$(490\ ft^2)$ of solar cell area. The total available area for solar cells is $91.04\ m^2$ $(980\ ft^2)$. The deployment sequence is such that the arm links are deployed first, then the array is unrolled as the roller is deployed to the end of the arm links. With this deployment scheme there is no need for slip rings between the array and the equipment module. The arrays are articulated to track the sun from solstice to solstice.

4.2.1.4 Equipment Installation

The location of the equipment in the Equipment Module as shown in Figure 4.2-3 was based primarily on thermal and CG considerations. No attempt was made to locate any of the equipment into sub-system groupings. The small antenna, the feed package and all the communication equipment is located on the upper rotating platform. This will eliminate the need for a rotating RF joint and will minimize the wave guide lengths between the feed package and associated equipment. Since the platform is earth oriented, the earth sensors and electronics are also located here. The majority of the equipment will be centrally located on the platform to facilitate the addition of any required thermal insulation. The TWT's and Klystron are located near the outer perimeter of the platform and are thermally isolated from any structure. This location was selected to provide ample distance between these packages and other electronic equipment and to allow direct radiation to space. The remainder of the equipment is distributed throughout the equipment module. The lower section of the module with approximately $1.303 m^3$ $(47\ ft^3)$ of available volume has been reserved for experiment packages. The hydrazine and ion engine fuel tanks, the momentum wheel and the batteries will be located in the tubular truss section of the module. Since the anticipated CG of the spacecraft should fall in this area, this location of the fuel tanks will minimize any CG changes during fuel consumption. The remainder of the equipment will be located in the upper section of the module and the center shelf.



ITEM	NOMENCLATURE	SIZE IN CM³ (INCHES)	QTY
A 1	BATTERY	22.56 × 30.48 × 19.05 (8.9 × 12.0 × 7.5)	2
A 2	BATT. CHG. REG.	22.86 × 48.26 × 25.4 (9.0 × [illegible])	1
A 3	CONVERT. NET.	44.7 × 95.5 × 5.24 (17.6 × 37.6 × 2.0)	1
A 4	SOLAR ARRAY	213.4 × 2133.6 (84.0 × 840.0)	2
A 5	ARRAY DRIVE ELEC	.034 m³ (1.2 ft³)	2
B 1	REFLECTOR	344.42 DIA (135.6 DIA)	1
B 2	FEED PKG.	50.8 × 25.4 × 50.8 (20.0 × 10.0 × 20.0)	1
B 3	200w TWTA	25.4 DIA × 55.8 (10.0 DIA × 22.0)	10
B 4	2000w TWTA	25.4 DIA × 55.8 (10.0 DIA × 22.0)	1
B 5	1000w KLYSTRON	25.4 DIA × 45.72 (10.0 DIA × 18.0)	1
B 6	5w PA TWT	5.08 DIA × 35.56 (2.0 DIA × 14.0)	6
B 7	TRANSPONDER	15.24 × 25.40 × 60.96 (6.0 × 10.0 × 24.0)	2
B 8	ANTENNA	22.86 × 51.80 (9.0 × 20.4)	1
C 1	DIPLEXER	2.54 × 5.08 × 7.62 (1.0 × 2.0 × 3.0)	2
C 2	TRANSMITTER	2.54 × 7.62 × 10.16 (1.0 × 3.0 × 4.0)	2
C 3	LOW PASS FILTER	1.27 DIA. × 12.7 (0.5 DIA × 5.0)	1
C 4	DACU	41.91 × 41.91 × 19.05 (16.5 × 16.5 × 7.5)	2
C 5	RECEIVER	2.54 × 7.62 × 10.16 (1.0 × 3.0 × 4.0)	2
C 6	CDD	44.19 × 32.26 × 12.95 (17.4 × 12.7 × 5.1)	2
C 7	ANTENNA	0.64 DIA × 91.44 (.25 DIA. × 36.0)	2
			5
D 1	FINE SUN SENS.		2
D 2	SUN SENS ELEC		1
D 3	RATE GYRO		1
D 4	INTERFER. HORN	7.8 × 7.8 × 15.73 (3.07 × 3.07 × 6.199)	6
D 5	EARTH SENS. HD	13.21 × 14.49 × 11.18 (5.2 × 5.7 × 4.4)	2
D 6	EARTH SENS ELEC	18.54 × 15.49 × 7.28 (7.3 × 6.1 × 7.0)	1
D 7	POLARIS SENSOR	29.21 × 12.95 × 11.18 (11.5 × 5.1 × 4.4)	1
D 8	CONTROL COMP	10.16 × 10.16 × 6.35 (4.0 × 4.0 × 2.5)	1
D 9	MOMENTUM WHEEL	30.75 DIA × 12.06 (12.125 DIA × 4.75)	3
D 10	MOMENT. WHL. ELEC		1
D 11	HYDRAZINE JET		
D 12	JET VALVE DR.	10.16 × 15.24 × 20.32 (4.0 × 6.0 × 8.0)	1
D 13	HYDRAZINE [illegible]		
D 14	HYDRAZINE TANK	15.24 DIA (6.0 DIA)	2
D 15	SLIP RINGS		1
D 16	5 cm. ION ENG.	13.0 DIA × 4.60 (5.1 DIA × 1.81)	6
D 17	30 cm. ION ENG.		3
D 18	ION ENG. FUEL	15.24 DIA (6.0 DIA)	1
D 19	INTERFER. ELEC	12.7 × 20.32 × 21.34 (5.0 × 8.0 × 8.4)	1
E 1	EXPERIMENTS	0.431 m³ (15.2 ft³) VOLUME	

Figure 4.2-3. ATS-AMS III A Sun Oriented Spacecraft Equipment Layout

4.2.2 GENERAL DESCRIPTION (ATS-AMS II)

The ATS-AMS II consists of a rectangular equipment module which has a rotatable antenna tower on its forward face, a 30 cm ion engine on its aft face and extendable, body-fixed solar panels on the two faces which are normally parallel to the orbit plane. The 5 cm vectorable ion engine attitude control thrusters are located at the tips of the solar array panels. Two 3-jet manifolds of hydrazine thrusters are mounted at the aft sides of the equipment module for emergency and backup use during periods of high attitude control disturbance torque or in the event of ion engine failure. A sketch of the spacecraft is shown in Fig. 1.3.2-2 and a list of the major spacecraft characteristics is given in Table 4.1-3.

The equipment module is a rectangular box 200.66 cm (79 inches) square and 177.80 cm (70 inches) high. The internal structure consists of a tapered cylinder running vertically through the center of the module and supported at the top and bottom. The cylinder is 40.64 cm (16 inches)diameter at the top and 60.76 cm (24 inches) diameter at the bottom. The top 30.48 cm (12 inches) houses the bearing for the rotatable tower. A vertical shear web, the full depth of the module, extends from the cylinder to the vertical centerline of each side, dividing the module into four (4) compartments.

A horizontal equipment mounting shelf is located 30.48 cm (12 inches) from the bottom of the module above the 30 cm ion engine and extends the full area of the module except inside the cylinder. The shelf is manufactured in two (2) parts to allow continuity of heat pipes which are integral with the shelf. The vertical shear webs in this area are not continuous. A horizontal heat shield is positioned inside the cylinder in the same plane as the shelf.

The tower is a welded tubular structure 50.80 cm (20 inches) square with a height of 261.62 cm (103 inches) that rotates on two (2) bearings housed in the main equipment module with a 30.48 cm (12 inch) vertical dimension between bearings. The base of the tower extends into the main equipment module to attach to both bearings. The first segment of the tower above the main equipment module contains the communications equipment package 50.8 cm x 50.8 cm x 50.8 cm (20 inches x 20 inches x 20 inches). The positioning of the communications equipment in this area makes the use of rotating RF joints unnecessary. On top of the tower are located two (2) earth-oriented parabolic antennas. The upper antenna being 76.20 cm (30 inches) diameter and the lower 243.84 cm x 121.92 cm (96 inches x 48 inches).

The experimental ion engine projects 21.59 cm (8.5 inches) below the main equipment module and is mounted on gimbals to allow a 15° movement in all directions. The gimbals are controlled by two (2) electric motors which can be operated independently or simultaneously. Both the main reflector and the sub reflector of the 243.8 cm x 121.92 cm (98 inch x 48 inch) cassegrain antenna and the 76.2 cm (30 inch) reflector will be honeycomb sandwich construction. The subreflector of the large antenna will be attached to the main reflector by tubular structure. The overall layout of the spacecraft is shown by Figure 4.2-4.

The Burner II three point attachment is adapted to the spacecraft eight point attachment by means of three Main I beam vertical longerons. These longerons are attached to a horizontal ring frame which is supported laterally by a triangulated structure of angle sections. The mating structure with the spacecraft is attached to the ring frame by struts and shear plates at eight (8) places.

4.2.3 GENERAL DESCRIPTION - ATS - AMS I

The ATS - AMS I consists of a rectangular equipment module supporting a large elliptical aperture parabolical antenna from a tower structure on its forward face, a small circular aperture parabolic antenna on its earth viewing face, having a cluster of three 30 cm ion engines on its aft face and supporting two, large area, extendible and rotatable solar

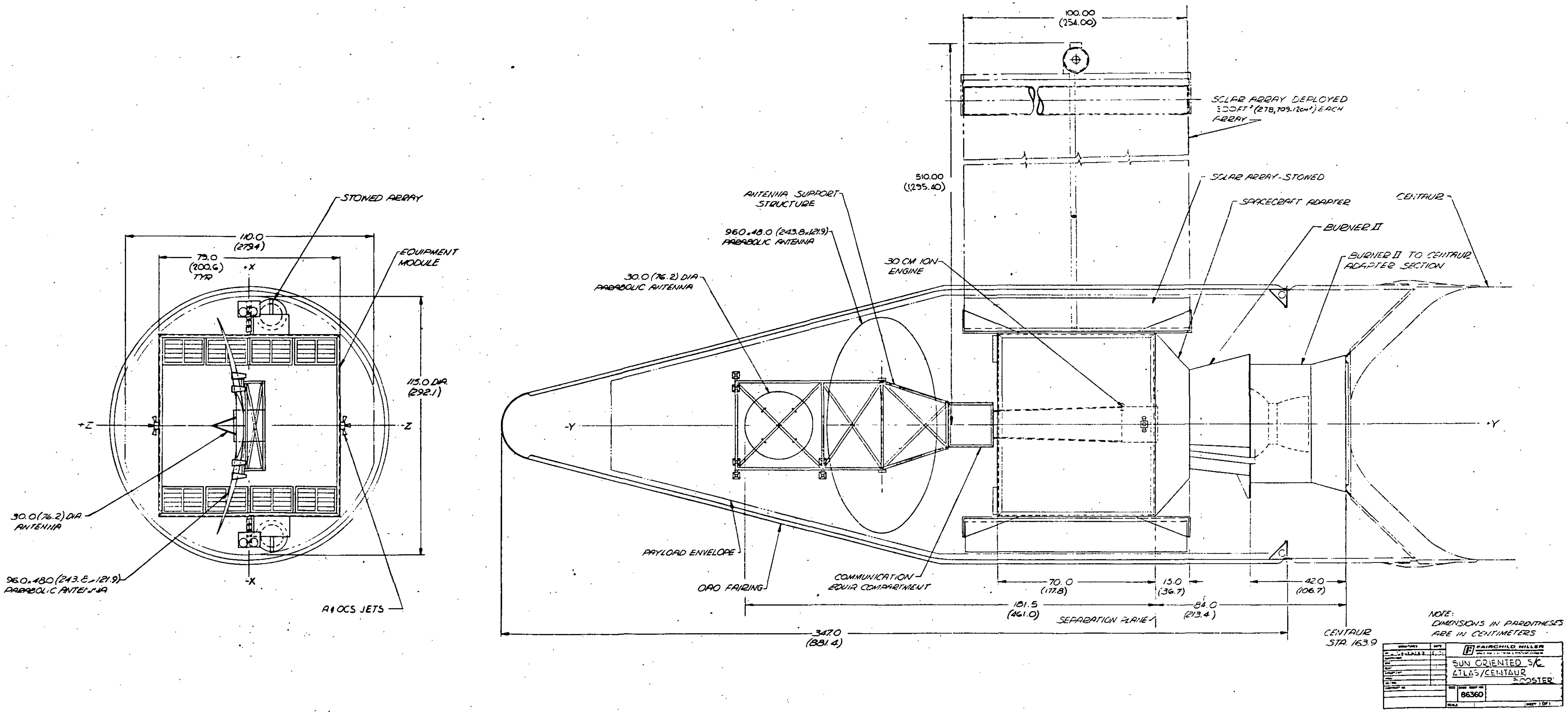


Figure 4.2-4. Layout of ATS-AMS II

panels from the two faces which are normally parallel to its orbit plane. A sketch of the spacecraft is shown in Figure 1.3.2-3.

The equipment module is a rectangular box measuring 111.8 cm x 132.1 cm x 128.3 cm high (44 inches x 52 inches x 50.5 inches). This represents a change in the basic shape of the equipment module from 116.8 cm (45 inches) square by 144.8 cm (57 inches) high. The change was made to provide additional volume for solar array stowage and to allow radiant cooling of the 30 cm ion engine by allowing them to extend 21.6 cm (8.5 inches) below the bottom of the spacecraft. The module structure consists of aluminum angle framework and honeycomb sandwich side panels and top cover. The bottom of the module is open to permit the 30 cm ion engines to gimbal. An equipment mounting shelf is located 33.3 cm (13.1 inches) from the bottom of the module. This shelf is also made from honeycomb sandwich construction. Selection of honeycomb for the side panels and equipment shelf was based on two requirements. One requirement was to provide sufficient stiffness for equipment mounting. The other consideration was the location of heat pipes. By installing the heat pipes inside the honeycomb panels, smooth mounting surfaces for equipment could be maintained. There are two vertical full depth sheet metal beams to provide for support of the solar array drive and slip ring assemblies. A third beam, 90° to the two beams is also provided. Along the bottom edge of the module are eight fittings used for attaching the spacecraft to the adapter section. A structural platform attached to two corner fittings is provided for the installation of the three 30 cm ion engines.

The spacecraft is attached to the DELTA 2910 by an adapter section which consists of welded tubes which join a rectangular framework to a circular frame. The welded tubular structure was selected for the adapter section and a one piece construction primarily for cost consideration. A one piece constructed adapter would require extensive and costly tooling to manufacture because of the transition from a rectangular shape at one end to a circular section at the other end. The bottom circular frame attaches to a flange on the third stage engine of the Delta similar to the existing 37 x 31 inch circular adapter. The upper rectangular framework contains eight fittings that match the fittings on the equipment module, and are used for attaching the spacecraft to the adapter section with explosive bolts.

The tower section is constructed from welded rectangular tubing. Provided in the tower section are mounting provisions for attaching the 2.438 m x 1.219 m (96.0 inch x 48.0 inch) parabolic antenna. Outrigger structure has also been provided to support the stowed solar array during the launch phase. Both the main reflector and the sub reflector of the 2.438 m x 1.219 m (98.0 inch x 48.0 inch) cassegrain antenna and the .762 m (30 inch) reflector will be of honeycomb sandwich construction. The main consideration in the selection of honeycomb sandwich construction for the antennas was surface accuracy of the reflectors. Although formed sheet metal frames with a mesh for the reflector would be less expensive the surface accuracy of this type of construction would be more difficult to maintain. The sub reflector of the large antenna will be attached to the main reflector by tubular structure. An overall layout of the spacecraft is given by Figure 4.2-5.

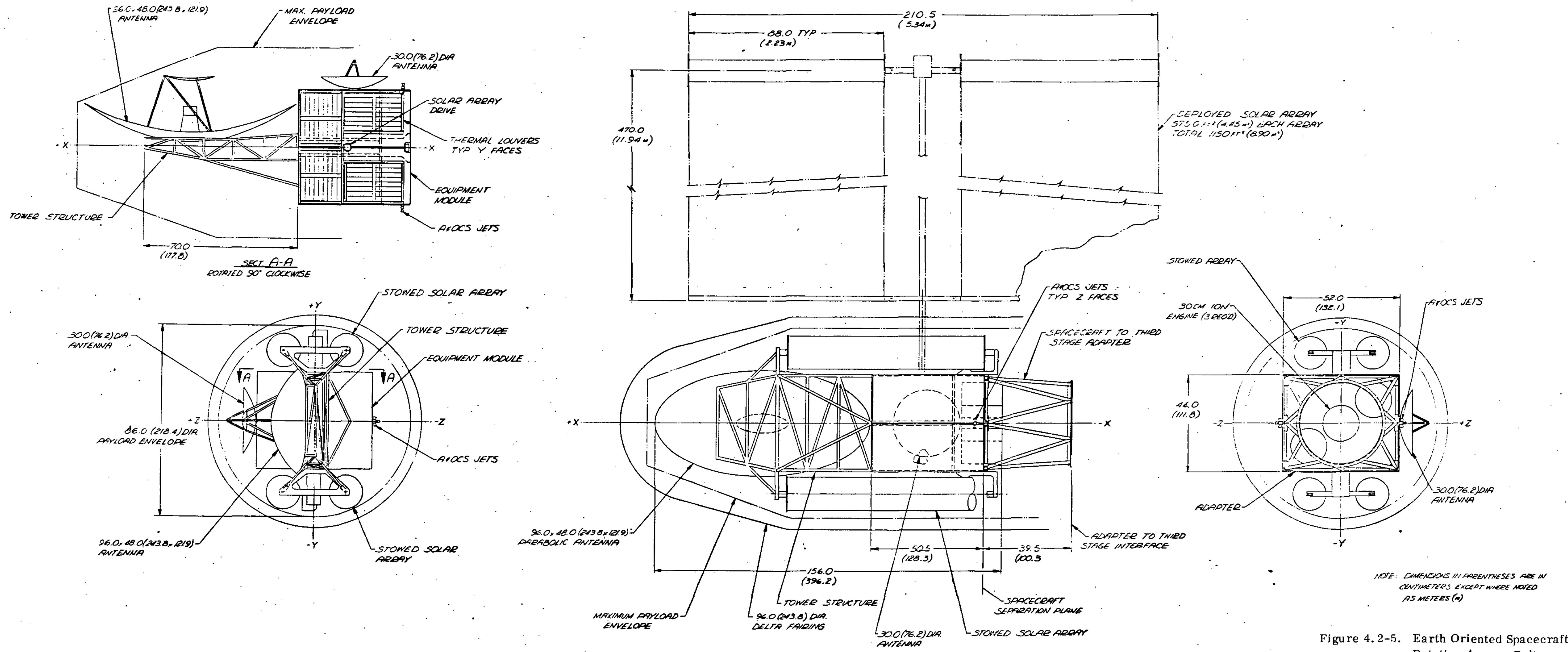


Figure 4.2-5. Earth Oriented Spacecraft Rotating Arrays Delta Booster 2910

4.3 ANTENNAS AND FEEDS

4.3.1 ATS-AMS III ANTENNAS

4.3.1.1 Coverage

The Public Broadcasting System Networking experiments require individual coverage of the four CONUS time zones in addition to Alaska and Hawaii, in order to distribute television programs throughout each of these zones. Signal separation between zones is an essential part of the experiment, so as to permit frequency re-use without interference from one zone to another. Part of this separation is achieved by contouring with multiple narrow-beamwidth beams. Figure 4.3.1-1 shows in heavy lines the 3 dB contour for the PBS experiment for ATS-AMS IIIA. The dashed lines show the reduced coverage of ATS-AMS IIIB. The various channels distributed throughout each zone of ATS-AMS IIIB are symbolized on Figure 4.3.1-1 as "f_A, f_B," etc. The same channels that are received in Alaska are also distributed to Hawaii by means of a 1/2^o beam.

Deviations between the shown coverages and the actual time zones are due to physical constraints in positioning adjacent multiple feeds. To obtain overlap of adjacent patterns would require an additional large antenna, which was not felt to be justified since the 3 dB coverage can be supplemented by stations outside the contours shown with a higher gain facility. These patterns are obtained from a satellite position of 110^o west longitude, which provides a particularly excellent coverage of the more heavily populated Southeastern area of Alaska, as shown in heavy lines in Figure 4.3.1-2, without excessive spillover into Canada. Positioning the satellite at a point further west increases spillover into Canada and does not

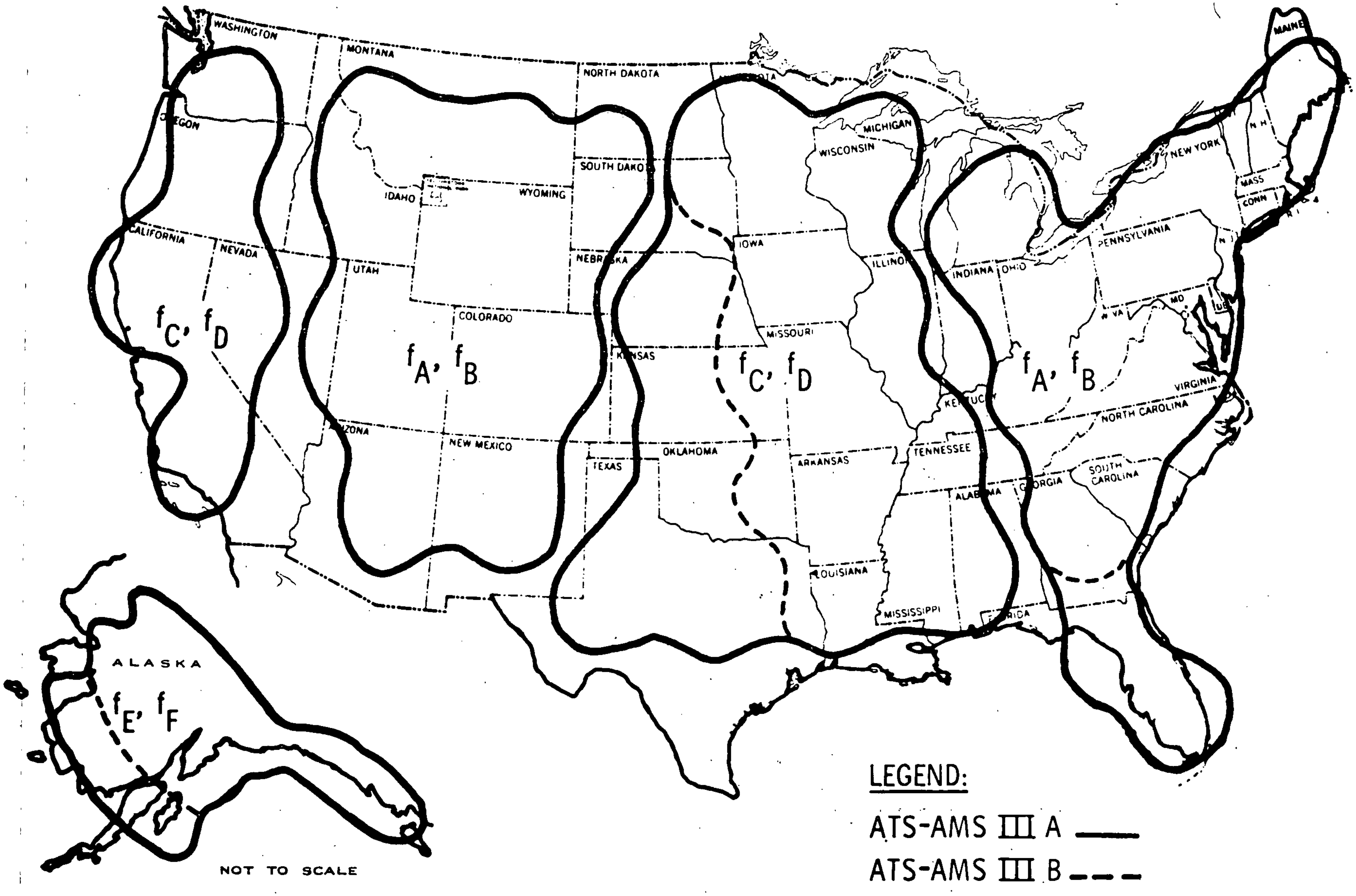
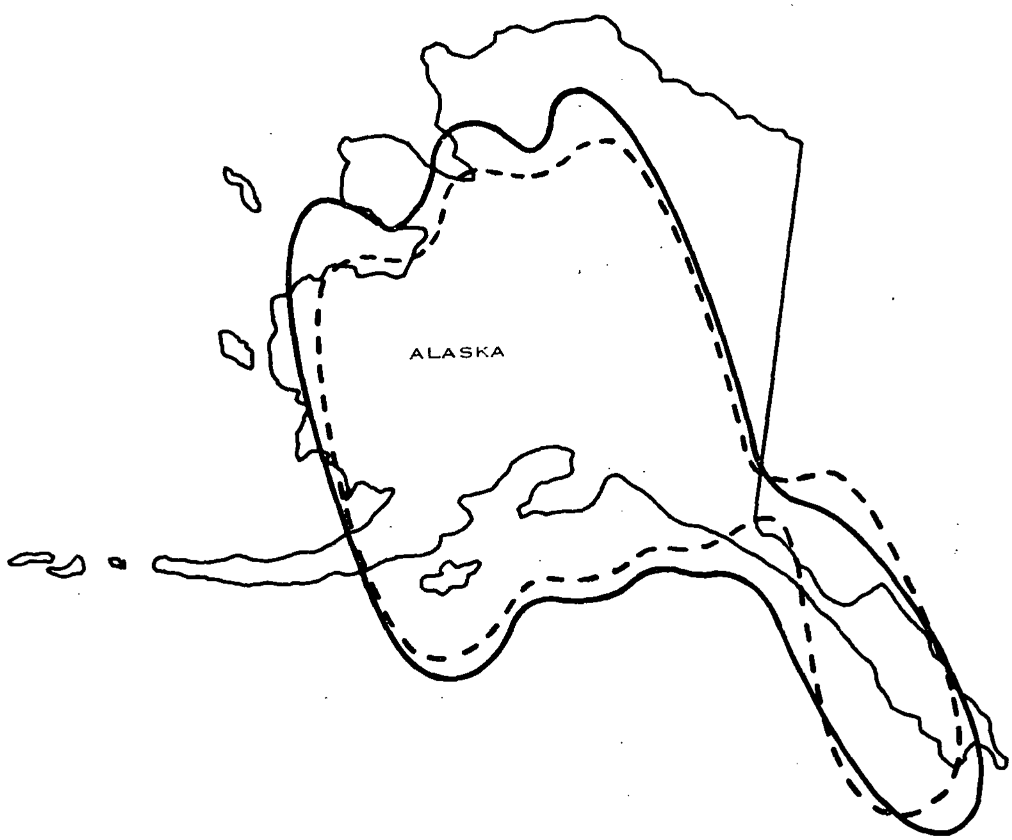


Figure 4.3.1-1. ATS-AMS IIIA Multibeam Contoured Coverage for PBS Experiment



Legend:

Sat. 110° W. Long. ————

Sat. 122° W. Long. – – – –

Figure 4.3.1-2. ATS-AMS III Alaska Coverage from Two Satellite Locations improve coverage of Alaska, in addition to a poorer CONUS coverage. Figure 4.3.1-2 shows in dotted lines the Alaska coverage from a satellite location of 122° west longitude.

The Cultural Region Interactive Television (ITV) experiments will be carried out with the same antenna used for the PBS experiment, and the desired coverage is achieved by re-assigning the necessary feeds so as to follow as close as possible the ITV regional boundaries. As shown in Figure 4.3.1-3, coverage of the six cultural regions is achieved with very little overlap into neighboring regions. The main restriction is the position that the feeds have been assigned for the PBS experiment. However, the amount of switching required to accomplish this transfer from time-zone coverage to cultural-region coverage is reasonable in weight and complexity. The re-assignment of feeds is the basic method to be employed in obtaining the desired coverages. Additional adjustments in the position of the feeds may improve the final coverages for the various experiments. In addition, coverage of Alaska is proposed for ATS-AMS IIIB, for medical communications.

Reception of video will be carried out by only those feeds that are directed to the areas containing originating stations. Therefore, satellite reception will be through 1/2° beams, relaxing the power requirements of the ground transmitters and, indirectly, the coordination constraints on earth station sites. For ATS-AMS IIIB, Alaska reception can be switched to any of the feeds that cover that region, for medical communications.

4.3.1.2 Large Reflector

Figure 4.3.1-4 shows the 3.44 meter (11.3 feet) parabolic offset reflector with the feed matrix at a focal length of 3.1 meters. This corresponds to the relatively large f/D ratio of 0.9, which minimizes

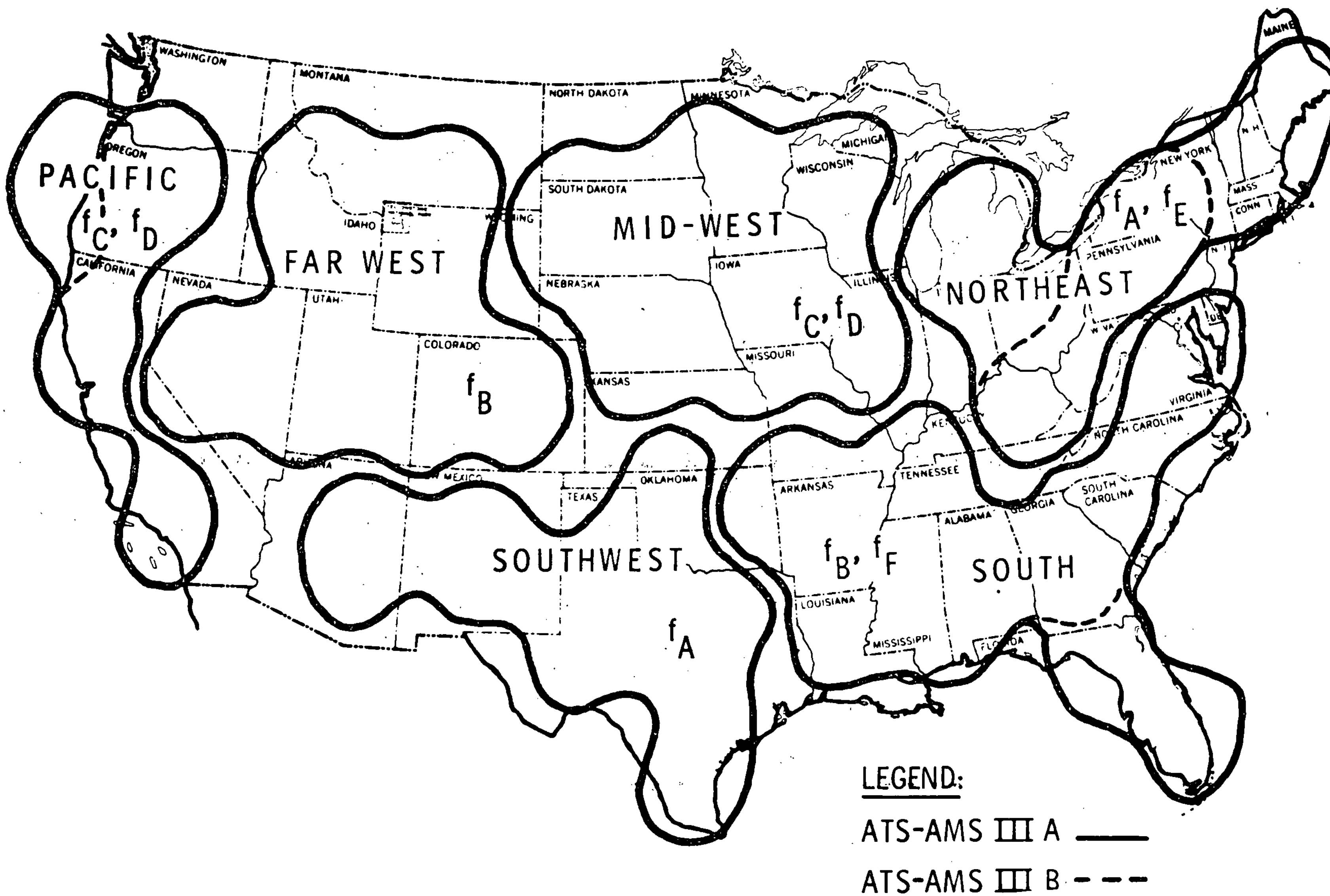


Figure 4.3.1-3. ATS-AMS III Multibeam Contoured Coverage for ITV Experiment

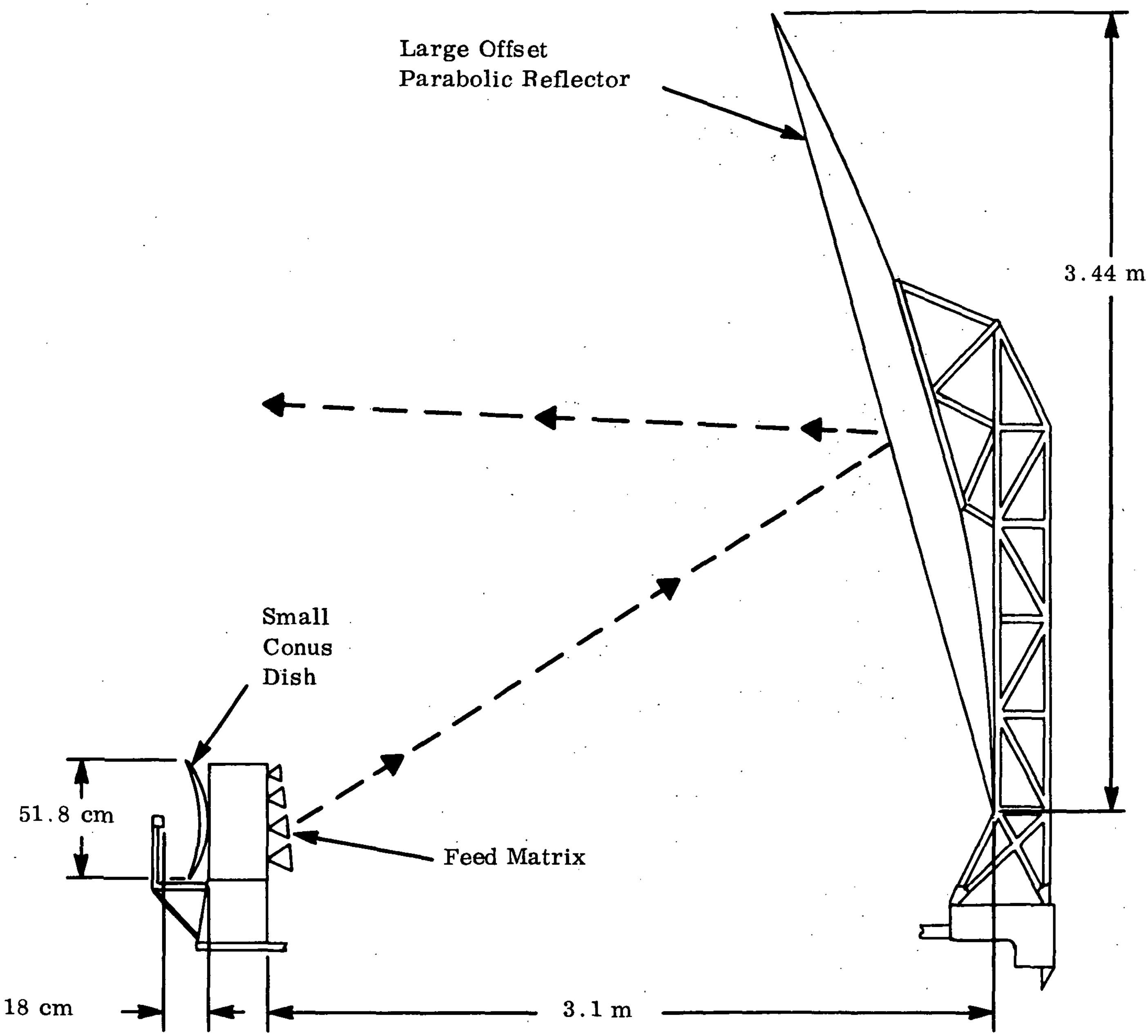


Figure 4.3.1-4. ATS AMS III Antennas

the comatic aberrations inherent in scanning. As shown in Table 4.3.1-1 the reflector and its supporting structure are estimated to weigh 45 kg (100 lbs), and are deployed approximately 1.3 m (4.16 ft.) back from their position at launch.

4.3.1.3 Feed Matrix

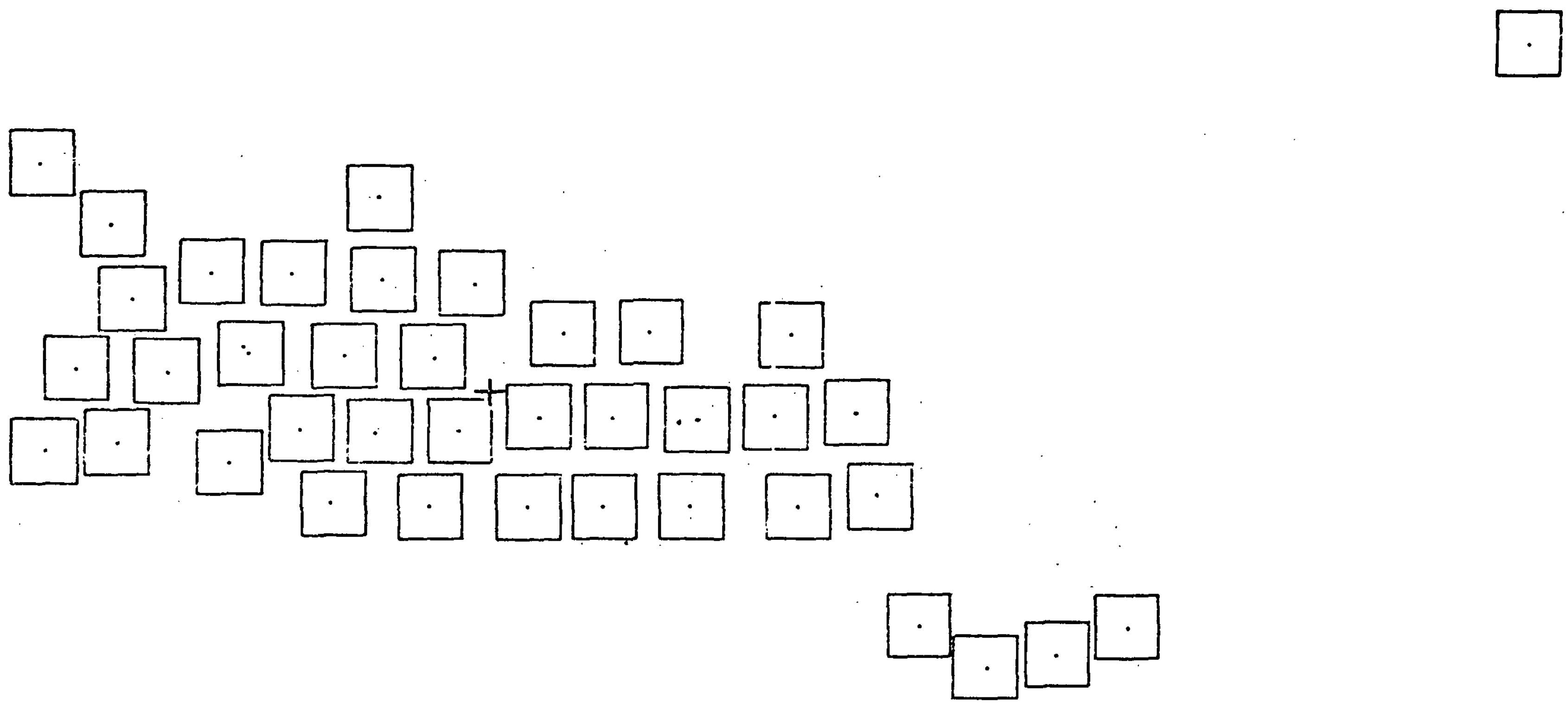
A total 37 of the 39 feeds are required for PBS coverage of CONUS, Alaska and Hawaii. The number of feeds assigned to each region is shown in Table 4.3.1-1. The position of the feeds is shown in Figure 4.3.1-5. However, adjustments are expected to be required in compromising the final coverage desired. The feeds are waveguide-fed horns flared in both planes to achieve tapering of the reflector illumination. The dimensions of each feed are 3.64 cm x 3.64 cm (1.43 x 1.43 in) and the total feed matrix is expected to exceed 20 x 50 cm (7.9 x 19.7 in) in size.

Some switching is required in order to re-assign transmitter and feeds from one experiment to the next. However, not all the feeds and transmitters have to be re-assigned between experiments, since many are common to several experiments. The complete switching system requires circulators, switches, and hybrids; Table 4.3.1-2 shows the approximate weight of all these components, which total 106 kg (235 lbs), including feeds and waveguides.

Selected feeds have orthogonal ports to which the receivers are connected, permitting full utilization of polarization for transmitter-to-receiver isolation in the satellite transponder. Additional isolation of transmitter power at the receiver is achieved by inserting a short strip of waveguide at the front end of the receiver, which filters the transmitter power by acting beyond cut-off at the 12 GHz transmitter frequencies.

Table 4.3.1-1. ATS-AMS III Feed Assignment

PBS Experiment			ITV Experiment		
Time Zone	No. of Feeds AMS III-A	No. of Feeds AMS III-B	Cultural Region	No. of Feeds AMS III-A	No. of Feeds AMS III-B
EASTERN	8	6	NORTHEAST	4	(2)
CENTRAL	12	8	SOUTH	8	6
MOUNTAIN	8	8	MID-WEST	6	6
PACIFIC	4		SOUTHWEST	6	6
ALASKA	4	8	FAR-WEST	6	6
HAWAII	1		PACIFIC	4	(4)



SCALE: 8 cm/in

Figure 4.3.1-5. ATS-AMS III Conceptual Feed Layout

Table 4.3.1-2. ATS-AMS III Antenna and Feed Subsystem Equipment List

Item	Quantity		Total Weight Kilograms (lbs)		Power Required Watts
	III-A	III-B	III-A	III-B	
Large Reflector (including support)	1	1	45.0(100)	45.0(100)	at deployment only
Small Reflector	1	1	2.3(5)	2.3(5)	None
Feeds (including waveguides)	39	36	35.0(77)	32.0(71)	None
RF Switches	22	32	10.0(22)	14.5(32)	None
RF Couplers and Hybrids	38	39*	9.5(21)	9.1(20)	None
RF Circulator	10	6	4.5(10)	2.7(6)	None
Reflectometers**	10	6	18.2(40)	10.9(24)	None
TOTALS			106.3(235)	106.0(234)	None

* All couplers.
** If required. Not in totals.

4.3.1.4 Small Reflector

A small parabolic dish of dimensions 22.8 cm x 51.8 cm (9 x 20.4 inches) is used for CONUS coverage. This small dish is located on the back of the main feed matrix, so as to produce no additional blockage or scattering to the large reflector (see Figure 4.3.1-4). The large dimension of this small dish is vertical, to correspond with the narrower width of the CONUS beam. A single feed is located at a focal length of 18 cm (7.1 in).

The weight of the small reflector, including the feed, is approximately 2.3 kg (5 lbs), as indicated in Table 4.3.1-2.

4.3.2 ATS-AMS I AND II ANTENNAS

The initial Statement of Work set forth as a mission objective for the ATS-AMS I and II is the following:

> "Paragraph 2.1.2 to develop the technology for the controlled illumination of desired areas of the earth, with shaped mutli-beam transmission, using antennas with major beam dimensions as small as 0.5^{o}, and having axial ratios less than 3."

For the AMS I and II requirement, two reflectors and various feed combinations are used to achieve beamwidths ranging from 0.65^{o} to 7.5^{o}. The 0.65^{o} width was chosen on the basis of fitting a reflector inside the available shroud volume without folding. The 7.5^{o} width was chosen to provide better E-W coverage. Beam variations and selection is by means of switches controlled by ground command through the T & C link. A small reflector is used for transmission and/or reception within regions, broadcast zones or CONUS, while a large reflector is used for transmission and/or reception on spot beams. Both reflectors use a Cassegrain feed system.

4.3.2.1 Coverage

The CONUS beam is $7.5^{\circ} \times 4.2^{\circ}$ at the half-power points and provides coverage of the entire CONUS. The broadcast zone beam is $2.25^{\circ} \times 4.2^{\circ}$ at the half-power points. Any one of three of these beams may be selected to provide coverage for the eastern, central or western portion of the CONUS. The regional beam is $2.25^{\circ} \times 2.1^{\circ}$ at the half-power points. Anyone of six of these beams may be selected to provide coverage for the upper or lower sections of the eastern, central, or western portions of the CONUS. The spot beam formed by the large reflector is $0.65^{\circ} \times 1.3^{\circ}$ at the half-power points. Of the fifteen beams available, various combinations may be selected to form high gain communication links to desired areas.

4.3.2.2 Small Reflector

The principal dimensions of the small reflector are shown in Figure 4.3.2-1 as is the location of one pair of feeds; the other two pairs are behind the one shown. The feeds, arranged in a 2 x 3 matrix, are waveguide fed horns with a small flare to a dimension of 3 cm for good reflector illumination. The waveguide is fed through the rear of the reflector from the component package which houses the beam forming and beam selection matrix. The estimated weight for the matrix is 9.4 kg (20.75 lbs). An additional 2.3 kg (5 lbs) is estimated for the reflector and support structure for a total of 11.7 kg (25.8 lbs).

The beam forming and beam selection matrix which consists of hybrid junctions and switches is utilized to energize the six feeds individually, in pairs, or all six at once. A $2.25^{\circ} \times 2.1^{\circ}$ regional beam is formed by selecting one feed, while the $2.25^{\circ} \times 4.2^{\circ}$ broadcast zone beam is formed by selecting a combined pair.

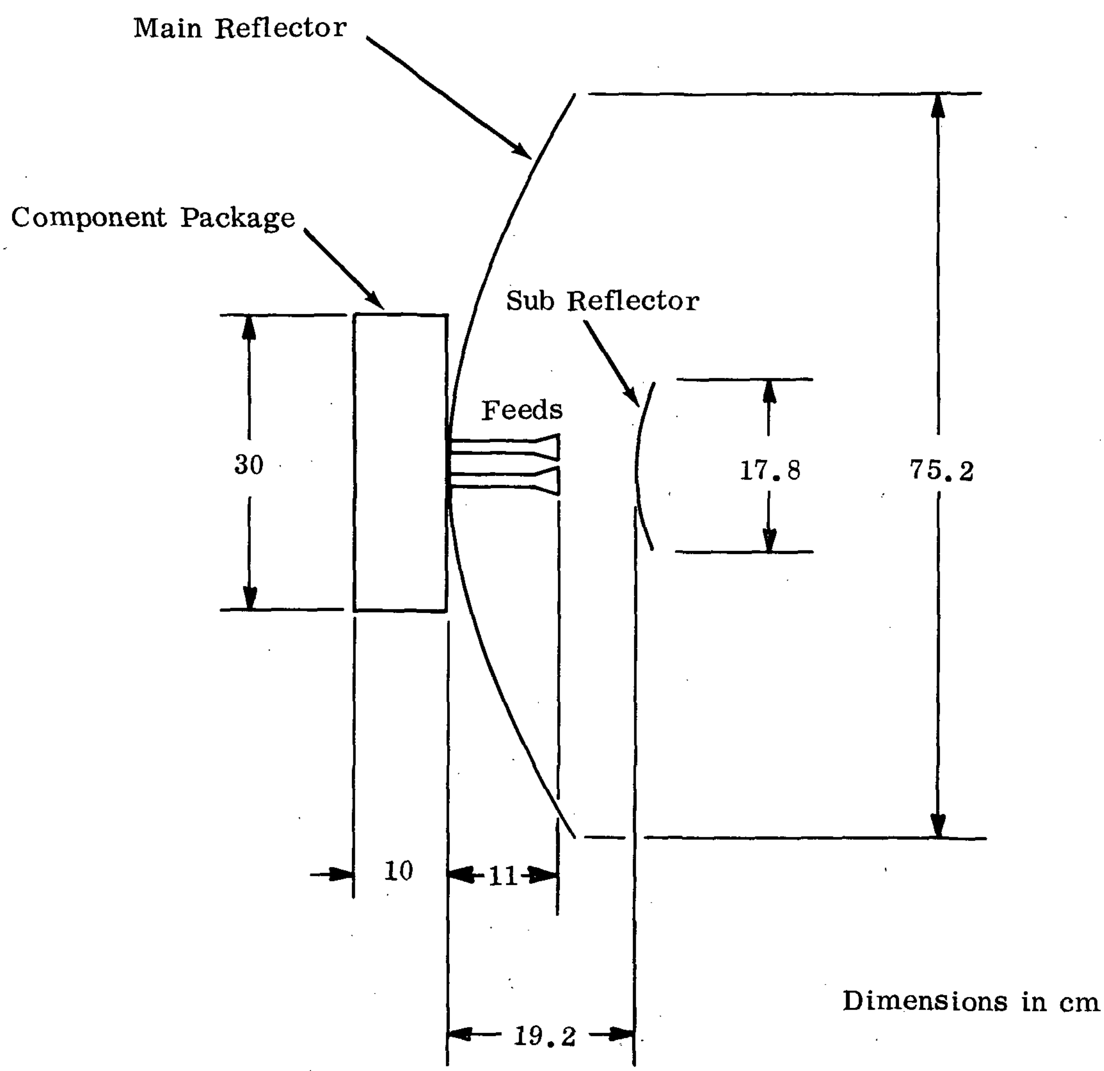


Figure 4.3.2-1. Small Antenna Dimensions

The 7.5^{o} x 4.2^{o} CONUS beam is formed by combining all six feeds. Beam formation is summarized in Table 4.3.2-1.

4.3.2.3 Large Reflector

The pertinent dimensions of the large reflector are shown in Figure 4.3.2-2. Although the feeds are not shown, the location of the phase center is. A total of fifteen feeds are used, arranged in a 3 x 5 matrix with the 5 feed row aligned with the smaller reflector dimension and the E-W direction. The feeds are waveguide fed horns, flared in both planes to achieve tapering of the reflector illumination. The waveguide is fed through the rear of the reflector to the component package which houses the beam selection matrix. The estimated weight for the matrix and feeds is 28.8 kg (64 lbs). An additional 13 kg (29 lbs) is estimated for the reflectors and support structure for a total of 41.8 kg (93 lbs).

In the E-plane, the horn aperture dimension is 4.5 cm and in the H-plane the aperture is 3 cm. The horns are aligned such that their E-planes are parallel to the plane containing the smaller reflector dimension. This arrangement produces about a 7 dB edge taper in both planes for the reflector. The estimated pattern shape for this reflector illumination is a beam with half-power width of 0.65^{o} x 1.3^{o}. With respect to the 3 x 5 beam matrix, beam pointing in the N-S direction is fixed at 0^{o} and $\pm 1^{o}$, while beam pointing in the E-W direction is fixed at 0^{o}, $\pm 1.5^{o}$, and $\pm 3^{o}$. Thus the total overall pattern coverage as measured from the outer 3 dB point of the first beam to the outer 3 dB point of the last beam is 2.65^{o} in the N-S direction and 7.3^{o} in the E-W direction. The beam cross-level in the N-S plane is estimated at -6.4 dB, while the cross over in the E-W plane is estimated at -4 dB.

Table 4.3.2-1. ATS-AMS I & II Beam Formation Summary

Beam Location	Beam Size (In Deg.)	Feed(s) Excited
Center (CONUS)	7.5 x 4.2	all six
left (western time zone)	2.25 x 4.2	1 & 4
center (center time zone)	2.25 x 4.2	2 & 5
right (eastern time zone)	2.25 x 4.2	3 & 6
upper left	2.25 x 2.1	1
lower left	2.25 x 2.1	4
upper center	2.25 x 2.1	2
lower center	2.25 x 2.1	5
upper right	2.25 x 2.1	3
lower right	2.25 x 2.1	6

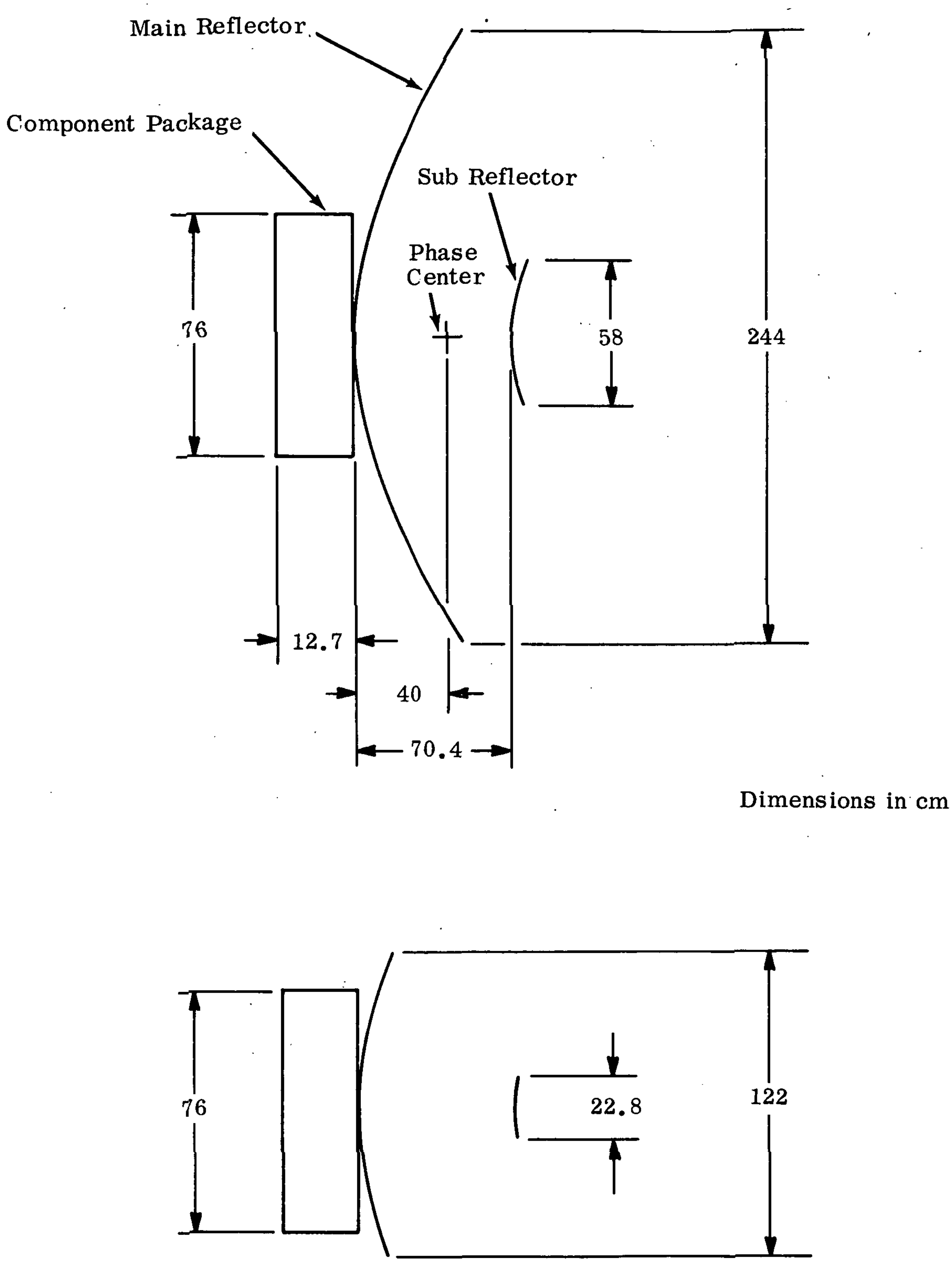


Figure 4. 3. 2-2. Large Antenna Dimensions

The receive beam selection matrix will permit the connection of any one of the fifteen feeds (beams) to any one channel. The transmit beam selection matrix permits the connection of any of 4 feeds to one transmit channel on an area basis. The receive matrix with separate TDA frontends permits complete versatility with only 0.1 dB insertion loss. While the transmit matrix does not have complete versatility, considerable weight reduction has been achieved, and the estimated insertion loss is only 0.3 dB.

4.4 COMMUNICATIONS TRANSPONDER SUBSYSTEM

4.4.1 COMMUNICATIONS FREQUENCY PLAN-ATS-AMS-III

4.4.1.1 Choice Of Up-Link Frequencies

A Federal Communication Commission Public Notice dated July 29, 1971, gives information on the results of the recent World Administrative Conference (WARC) which revised the Table of Allocations and Articles governing conditions of use for Space Services. The changes are scheduled to enter into force on January 1, 1973; and require a significantly different up-link frequency band since the U.S. proposed 12.75 - 13.25 GHz was not allocated to the FIXED-SATELLITE service. Instead the WARC adopted the 250 MHz upband 12.5 - 12.75 GHz region 2, shared with FIXED and MOBILE, and the 500 MHz band 14-14.5 GHz shared with existing services:

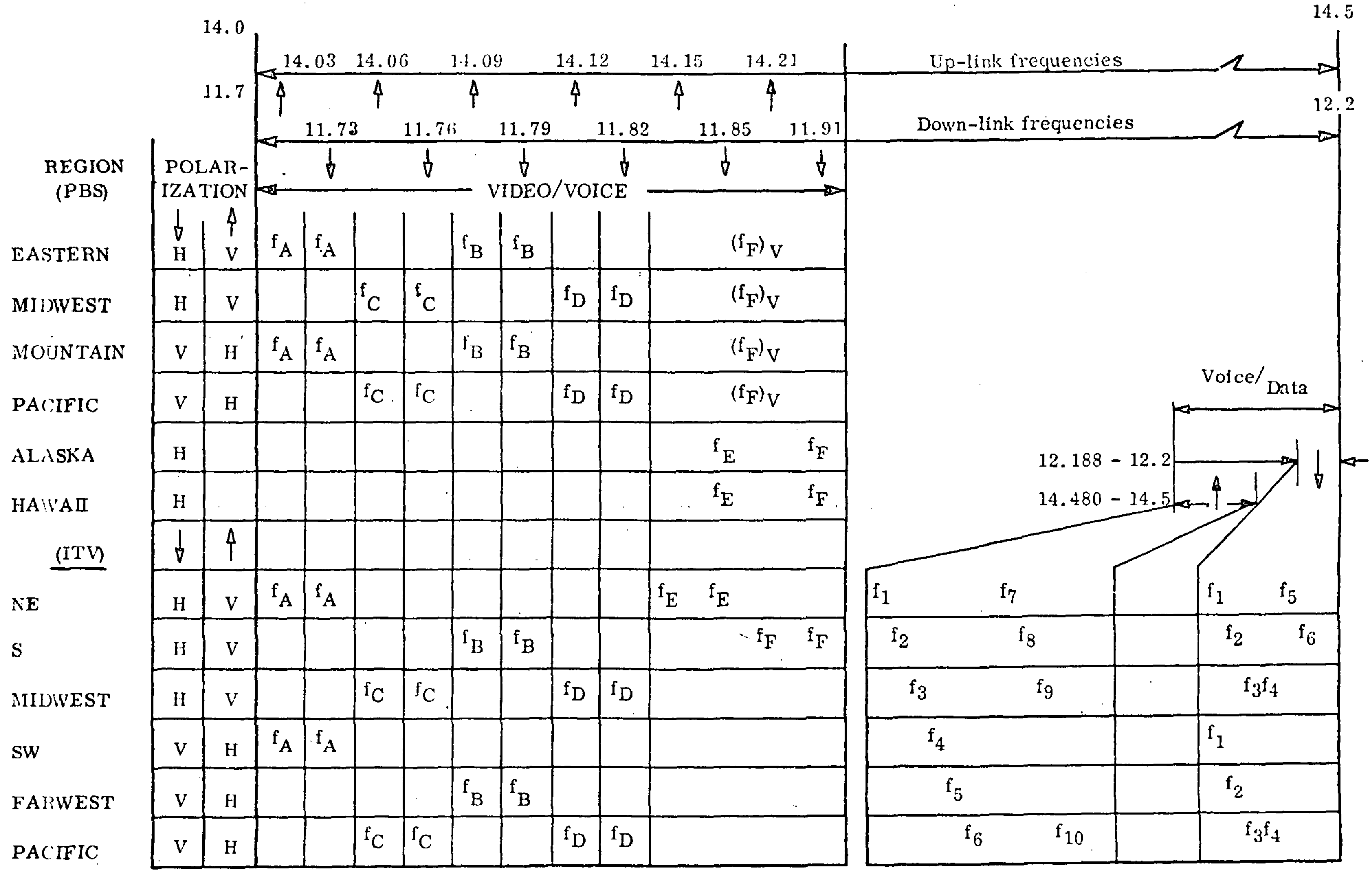
14.0	-	14.3 GHz	FIXED-SATELLITE (UP) RADIONAVIGATION
14.3	-	14.4 GHz	FIXED - SATELLITE (UP) RADIONAVIGATION SATELLITE
14.4	-	14.5 GHz	FIXED FIXED - SATELLITE (UP) MOBILE

4.4.1.2 Down Link Frequencies

As specified in the Statement of Work, the down-link band is 11.7 - 12.2 GHz.

4.4.1.3 Frequency Plan (Reference Figure 4.4.1.3-1)

Television signals carrying video/voice from ground originating stations to the ATS-AMS III are received in the 14.0 - 14.5 GHz band and transmitted to provide regional coverage at 11.7 - 12.2 GHz. With 24.5 MHz useful bandwidth for each voice video/channel, carriers are conservatively spaced on 30 MHz centers.



$(f_F)_V$ Uplink video/voice frequency for portable earth stations received on CONUS coverage antenna. All other video/voice channels are received and transmitted via the large 3.44 meter antenna

Note: Each frequency number designates a band of frequencies 1.0 MHz wide

Figure 4.4.1.3-1. Typical Frequency Plan (ATS-AMS III A)

For the baseline experiments, alternate beam cross polarization and identical frequencies are used in alternate regions so that six frequencies suffice to provide the 10 receive (uplink) and 10 transmit (downlink)channels for ATS-AMS III A, and three frequencies are needed for the 5 channels for ATS-AMS III B. Figure 4.4.1.3-1 shows a video/voice frequency assignment plan for the PBS network. Interactive TV experiments utilizing the full 10 channels of ATS-AMS III A. The frequencies for each region are designated by the same letters indicated in the coverage patterns of Section 4.3, Antennas and Feeds Subsystem.

Voice/data channels for interactive educational programs are in the same frequency bands as for video/voice. Allowance has been made for 10 uplink channels requiring 10 frequency bands for ATS-AMS III A, and 5 channels for ATS-AMS III B, each 1.0 MHz wide, received at the CONUS coverage antenna. The down-link voice/data channels to the origination stations is sent via spot beams from the same feeds as for the video receive channels. Again, identical frequencies are shared in alternate zones; thus six frequency bands suffice for ATS-AMS III A, and three frequency bands for ATS-AMS III B. The response to the student is on the video/voice channels with data assigned to the baseband slot.

Sufficient frequency spectrum is available for doubling the video/voice channels and increasing the number of audio/data channels by at least five times to user and programmer capacity requirements, and additional channels could be provided by ATS-AMS III at the expense of the undefined experiments weight reserve.

4.4.2 COMMUNICATIONS TRANSPONDER - ATS-AMS-III

4.4.2.1 General

The basic configuration for the ATS-AMS III communications transponder and matrix assembly is shown in Figure 4.4.2-1. The video/

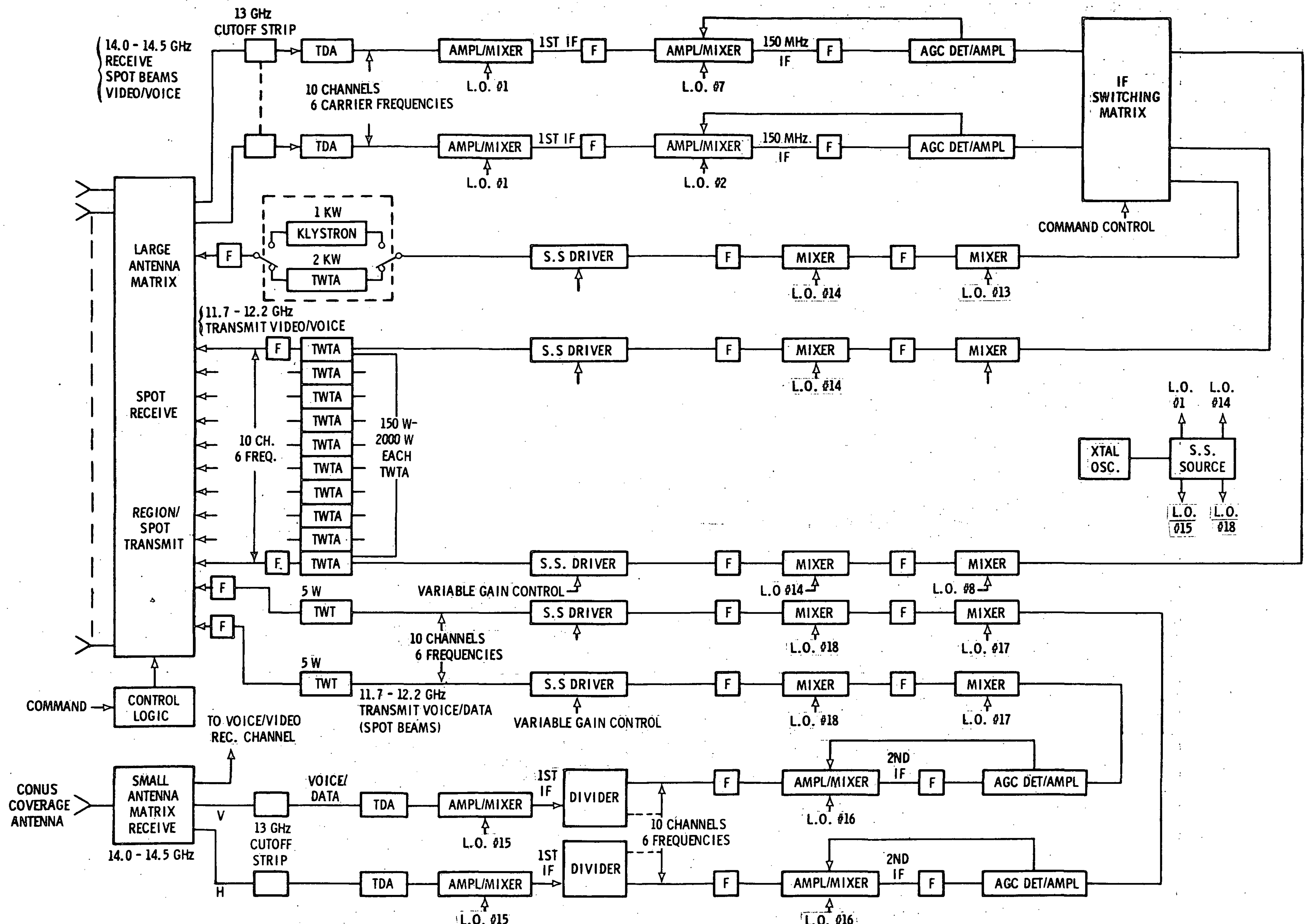


Figure 4.4.2-1. ATS-AMS III Communication Transponder

voice repeater is capable of:

- Receiving video/voice channels 30 MHz wide (24.5 MHz video/voice FM bandwidth) on spot beams on the large antenna and on the small CONUS coverage antenna.
- Transmitting video/voice channels from the large antenna via contoured pattern beams after translation to the 11.7 - 12 GHz frequency band.
- Providing IF and RF switching of individual channel frequencies and feeds upon ground command as desired to change experiments and provide a remote feed capability.

The voice/data repeater is capable of receiving channels 1.0 MHz wide at 14.0 - 14.5 GHz on the CONUS antenna, translating to the 11.7 - 12.2 GHz frequency band and transmitting on spot beams from the large antenna.

4.4.2.2 Video/Voice Transponder

The input signal is amplified in a tunnel diode amplifier (TDA) with a gain of 15 - 20 dB and a noise figure of 6 dB nominal.

The output of the TDA is down-converted in two stages providing optimum rejection of image frequencies. AGC controls the second amplifier mixer output. Local oscillator (LO) signals for each mixer stage are derived from a solid state microwave source using a Gunn type oscillator phase locked to a crystal reference oscillator. Intermediate frequencies at the second IF are the same (150 MHz) for all transponders to permit flexibility of video channel switching. IF switching is provided on individual channels (and frequencies) to direct programs to different locations as required by each networking experiment.

IF signals are amplified and up-converted in two steps to 11.7 - 12.2 GHz with LO signals derived from the same reference oscillator as for the first LO. After filtering, each output is fed into a high powered TWTA for transmission to the desired areas on selected beams via the contoured pattern antenna. Nominally, 200 watts output power will supply the required power levels at most earth station locations for the desired quality of video.

The 1000 watt Klystron or the 2000 watt TWTA is placed in operation to achieve acceptable quality video under rainfall attenuation conditions or with 2 to 4 carriers in a backed-off mode. This latter capability must first be developed under laboratory test conditions.

Output power control (e.g. back-off of TWT input power) is varied via the power amplifier driver upon ground command. Solid state drivers are IMPATT diode negative reactance amplifiers (watts of output) and fundamental Gunn diode amplifiers (hundreds of milliwatts). IMPATT diode amplifiers have gains of 30 dB, Gunn diode amplifiers have 10 dB gain. Efficiencies are 6 - 10%.

Control logic is used for all switching functions and for controlling gain (e.g. for increasing or decreasing power of TWT's). Upon switching to the high power TWT or Klystron the high voltage is first switched off and then reapplied when the appropriate device is connected.

4.4.2.3 Voice/Data Transponder

Voice/data signals are received from multiple, widely dispersed locations on the small CONUS coverage antenna. After pre-amplification (TDA), the carrier frequencies are down-converted into stages. Following an initial down-conversion to a 1 GHz center frequency, the composite signal is divided and filtered into individual 1.0 MHz frequency bands and down-converted to second IF's below 500 MHz. Up-conversion is in two stages, with six frequency bands shared among the ten down link channels for ATS-AMS III A.

Five watt TWT's are used as power amplifiers. Because of the multiple access carrier voice/data operation, the TWT's are backed off 10.8 dB to the linear operating region of the tube to maintain carrier-to-intermodulation product levels of 24 dB. Allowing for antenna feed losses, the power delivered at the multibeam antenna is 25 milliwatts per 45 kHz channel. An alternate approach is to precede a lower power TWTA (1.5 W) with predistortion circuitry, resulting in the same performance with 6 dB output backoff. Multi-carrier operation in spaceborne power amplifiers is recommended for future experimentation and development. Solid state drivers for the 5 watt TWTA's are linear Gunn effect devices to insure that additional intermodulation products are not introduced at the amplifier input.

4.4.2.4 Transmitter/Receiver Isolation

Isolation filters and antenna polarization are combined to reduce transmitter out-of-band noise and transmitted output power below the noise level of the receiver. Three and four section microwave filters with -40 and -50 dB isolation respectively are used at the output of each TWTA to reduce transmitter - generated noise. Cutoff waveguide strip is inserted before the TDA in the 14 GHz receive channel to provide -100 dB out-of-band filtering. Filters are lightweight (ounces) at these frequencies.

4.4.2.5 Transponder Characteristics

Transponder input and output levels are summarized in Table 4.4.2.5-1. Link analyses to support these levels are included in Section 5.1. Overall weight and prime power for the transponder is estimated in Table 4.4.2.5-2.

An outline drawing of the high powered TWTA is included in Section 5.8, Thermal Control. The Klystron or 2000 W TWTA is not operated simultaneously with the other TWT's except for a standby power of about 100 watts, due to limitation of power at EOL.

Table 4.4.2.5-1. Transponder Input/Output Levels

Type Signal	Signal Level	Bandwidth	Output Power
Video/Voice	-57 to -84 dBM	24.5 MHz	200 W
Voice/Data	-81 dBM	1 MHz	5 W
Receiver Noise Powers	-124.0 dBW at 24.5 MHz bandwidth		
	-137.8 dBW at 1.0 MHz bandwidth		

Table 4.4.2.5-2. ATS-AMS II Communication Subsystem Equipment List

Item	Quantity		Total Weight		Power Required Watts	
	ATS-AMS III A	AMS III B	III A	III B	III A	III B
1000 W	1	1	17 (37)	17 (37)	2000 33 W St By	2000 33 W St By
2000 W	1	1	13 (28)	13 (28)	4000 65 W St By	4000 65 W St By
TWTA (150-200 W)	10	5	127 (280)	63 (140)	4000 65 W St By	2000 33 W St By
5W TWT	10	5	22 (50)	11 (25)	150	75
TDA	12	9				
MIXERS	74	43				
L.O. SOURCE (S)	18	14				
I F AMPLIFIERS	21	12	109 (240)	63 (140)	400	275
SOLID ST. DRIVERS	21	11				
SWITCHING MATRIX	1	1				
FILTERS - BP	84	46				
- Isolation	20	11				
WAVEGUIDES						
			288 (635)	168(370)	4648 W Max.	2448 W (4416 W Max)

4.4.3 COMMUNICATIONS TRANSPONDER - ATS-AMS-I AND II

4.4.3.1 General

The initial Statement of Work identifies as part of mission objective (2.1.1) the capability "To demonstrate the high power communications technology using transmitter output of one kilowatt or more in space in the frequency band of 11.7 - 12.2 GHz." Mission constraint 3.5 adds the requirement 3.5 for considering "Klystron and traveling wave tubes which have efficiencies of 50% or greater" and constraint 3.14 states "The spacecraft shall be capable of repeater-type operation."

The communication transponder concept herein described is directed toward achieving the objectives stated in particular relating to experimentation on high power TWT's and Klystrons. The basic configuration for the spaceborne communications transponder is shown in Figure 4.4.3.1-1.

The spaceborne communications transponder for ATS-AMS-I and II will provide a capability for receiving up to four video/voice channels each 40 MHz wide (34 MHz usable bandwidth) in the 12.75 - 13.25 GHz band, on 15 separate spot beam feeds on the large antenna and on a small CONUS coverage antenna, and for transmitting up to four video/voice channels in the 11.7 - 12.2 GHz band on either the large or small antenna for spot beam, region at CONUS coverage.

4.4.3.2 Transponder Description

The received signal from each beam is fed into a tunnel diode amplifier (TDA) with a noise figure of about 5 dB ensuring a receiver noise figure of approximately 6 dB. For each of the 15 spot beams into the large antenna the output of the TDA is down-converted to intermediate frequencies in the 280 - 440 MHz band by mixing with a local oscillator signal programmed by ground command. A low pass

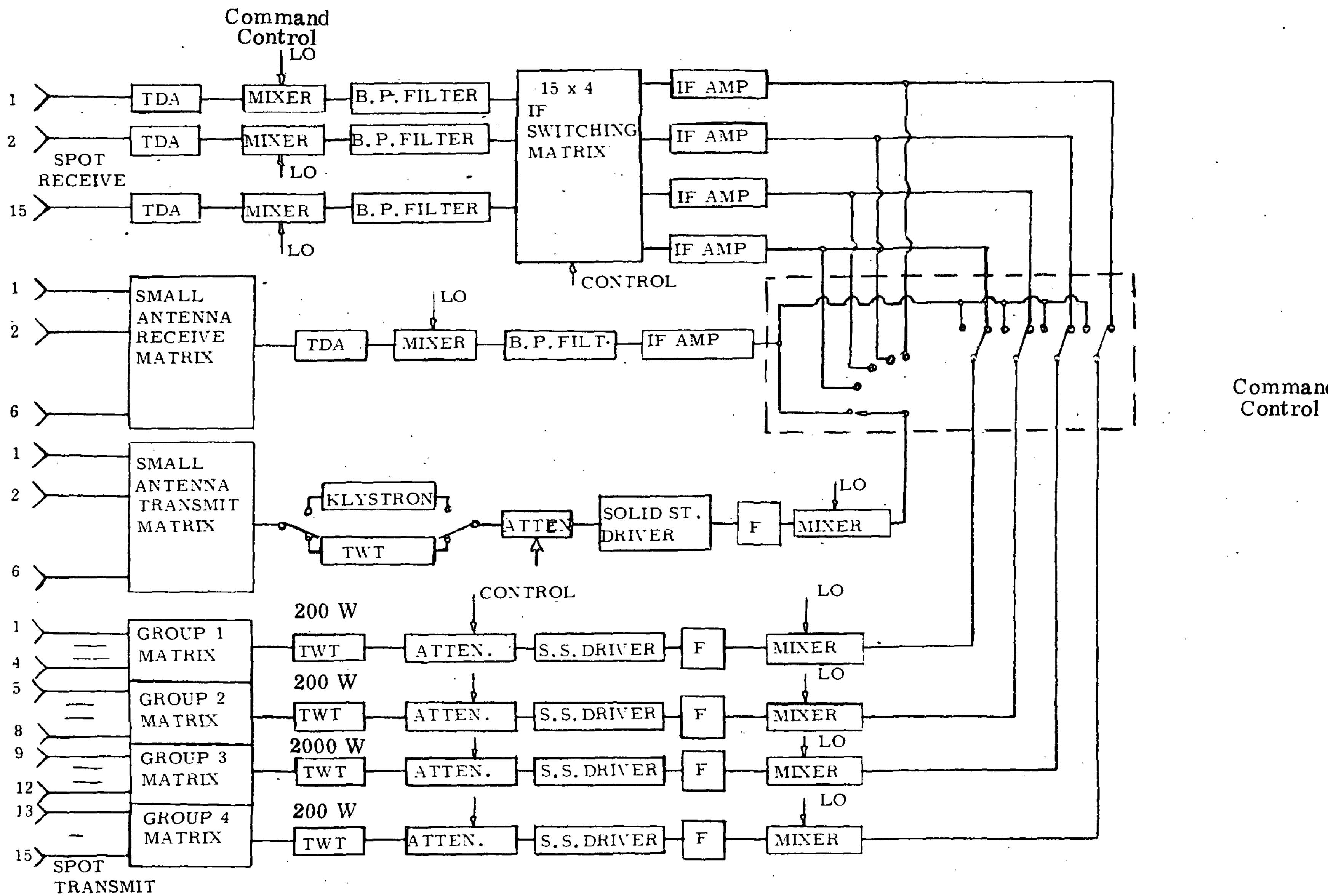


Figure 4.4.3.1-1. Satellite Repeater Block Diagram ATS-AMS -I/II

filter selects the lower sideband from the mixer output. The IF switching matrix transfers each receive channel to the appropriate transmit channel (s) on the four 200 watt transmitters (single carrier on a 34 MHz useful bandwidth on each transponder) or to the 2000 W TWTA (or 1000 W Klystron) for transmission on a single channel on the small CONUS antenna. Up to four simultaneous channels received on the small antenna are down converted with a single LO for up conversion and transmission via the 1000 watt Klystron or 2000 Watt TWT over the small antenna or for up-conversion and transmission over one of the selected 200 Watt TWT's. A 40 MHz band pass filter is assumed at the front end of each of the four up-converters (mixers) to select the appropriate IF band from the 160 MHz TWT band (100 MHz for the Klystron).

The switch matrix following the IF amplifiers performs the channel transfer described also upon ground command. A variable gain control provides, upon ground command, the proper drive level for either the TWT's or Klystron amplifier for various experiments and for necessary back-off when using more than one channel.

Up to four simultaneous video channels will be transmitted by the 2 KW TWT and no more than two RF channels by the 1000 Watt Klystron.

Redundant modules are used for each of the active solid state subassemblies, except for the 15 TDA's. These are all on stand-by and can be switched in, in event of failure or intolerable degraded performance, by means of the control logic.

Control logic is used for all switching functions and for controlling gain. Before switching the high power TWT or Klystron, the high voltage is first switched off and then reapplied when the appropriate device is connected, preventing the generation of high power transients with all of the attendant wide band spectral components.

The overall prime power required for the transponder is governed principally by the efficiency of the 2000 Watt TWT and 1000 Watt Klystron and will be approximately 4 KW. The transponder, independent of the feed matrix assemblies, should weigh under 64 kg (140 pounds) and can be contained in two packaged 10 x 20 x 56 cm (4" x 8" x 22") exclusive of the power amplifier tubes. Weight and power estimates for the ATS-AMS I and II transponder components are summarized in Table 4.4.3.2-1.

Table 4.4.3.2-1. Communication Subsystem Equipment List ATS AMS I & II

Item	Quantity	Total Weight		Power Required Watts
		Kilograms	(Lbs)	
1000 W Klystron	1	17	(37)	2000 33 W Standby
2000 W TWTA	1	13	(28)	4000 66 W Standby
200 W TWTA	4	51	(112)	1600 26 W Standby
TDA	16			
Mixers	21			
IF Amplifiers	5			
Solid State Drivers	5	37	(81)	200 W
Cont. Atten.	5			
15 x 4 Switch Matrix	1			
Band Pass Filters	21			
Wave Guides				
		118	(256)	4259 W MAX

4.5 POWER SUPPLY SUBSYSTEM

Two power supply subsystem configurations were used in the preliminary designs: Shunt Regulation and Direct Energy Transfer. Of the two, Shunt Regulation represents the more conventional class of space-proven power subsystems. It, too, is a form of a direct energy transfer system in that there is no element in series between the solar array and the low voltage loads. The selection of this configuration for a synchronous altitude mission is supported by many tradeoff studies*. In the AMS case it is used for housekeeping power for the ATS-AMS I and serves as the main power supply for the ATS-AMS-II and III. The second configuration, Direct Energy Transfer, achieves power conditioning and regulation by a switching technique that permits the transfer of power generated by the array at high voltage directly to those spacecraft loads that require high voltage. This configuration has been studied for its applicability as the main power supply for the ATS-AMS-I mission and as the configuration for the high voltage array experiments of ATS-AMS-II and III.

The following paragraphs summarize the characteristics and capabilities of the power subsystems selected for each mission.

4.5.1 ATS-AMS-III Power Subsystem Configuration

The power subsystem for ATS-III is required to provide power over a 2-5 year period primarily from solar arrays with batteries supplying housekeeping and communication keep alive during occult (darkness).. With solar array degradation of 18% over the first two years of operation available power for high voltage communications experiments will decrease with the requirements for housekeeping,

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* See, for example, Barna, G. and Newell, R., "Design of a Multi-Kilowatt Photovoltaic Power System for Manned Space Stations", IECEC 1967.

low voltage communications and orbit control (ion engine thrust power) expected to remain essentially constant. The PBS and ITV communications experiments postulated have been based on available power at the end of a two-year period. Additional increased power experiments may be performed during the initial two years (from start of life) as described in Section 5.5, Power Supply Analysis. The power supply subsystem has been sized on the maximum power requirements anticipated.

During each day in orbit the demands placed on the power subsystem will vary according to the following schedule during equinox:

- Orbit control, once a day for 0.5 hour will require high voltage power for firing the ion engines, and 30.5 VDC communications keep-alive and housekeeping (antenna tower drive, telemetry and command and attitude control).

- Occult during equinox with periods of darkness up to 1.2 hours with battery power for housekeeping and communications keep-alive.

- Primary communications for the rest of the day (22.3 hours at equinox to 23.5 hours at solstice). The power requirements are for 3.05 VDC housekeeping and low voltage communications; in addition to high voltage communication and margin for undefined experiments and battery recharge at 67 Volts.

The power requirements for ATS-AMS-III at end of life are summarized in Table 4.5-1, for a typical day in orbit. The maximum period for occult power, power for housekeeping, keep-alive for communications and low voltage communications for ATS-AMS-III A and B are as outlined in Table 4.5-2.

Table 4.5-1. ATS-AMS-III Power Subsystem Load Requirements (EOL)

Operational Mode	ATS-AMS-III A	ATS-AMS-III B
Primary Communications ***		
• 30.5 V DC Power	725 W (830 W)	575 W (660 W)
• High Voltage Bus Power	4150 W (4840 W)	3400 W (3970 W) *
• Margin for Battery ** Charge and Undefined Experiments (67 V)	(1330 W)	(840 W)
Total @ 67 V	(7000 W)	(5470 W)
Orbit Control (0.5 Hr/Day)		
• 30.5 V DC Power	538 W (601 W)	238 W (437 W)
• High Voltage Bus Power	2000 W (2330 W)	2000 W (2330 W)
Total @ 67 V	(2931 W)	(2767 W)
Occult (Max. 1.2 Hr/Day)		
• 30.5 V DC Power	538 W (601 W)	383 W (437 W)

Notes: 1. Parenthesis represents 67 V primary bus power

* By absorbing the bulk of the margin (less 160 W) for additional HV communications the 2000 W output TWT may be operated at full saturated power with a tube efficiency of 50%.

** Battery discharge, recharge and capacity requirements are analyzed in Section 5.5.

*** Solstice condition (worst case)

Table 4.5-2. AMS-III 30.5 VDC Power Requirements

30.5 V DC Bus Power (During Orbit Control and Occult)	ATS-AMS-III A	ATS-AMS-III B
Antenna Tower Drive	10 W	10 W
Telemetry and Command	45 W	45 W
Attitude Control	170 W	170 W
1. Total, housekeeping	225 W	225 W
2. Communications keep-alive	213 W	158 W
3. Margin	100 W	-
4. Total (1, 2, 3)	538 W	383 W
30.5 V DC (Bus) Power (Primary Communications)		
1. Total, housekeeping	225 W	225 W
2. Low power communications	400 W	275 W
3. Margin	100 W	75 W
4. Total (1, 2, 3)	725 W	575 W

Analysis supporting the estimates shown are given in Section 5.5, Power Supply Analysis.

The ATS-AMS III spacecraft power subsystem block diagram is shown in Figure 4.5-1. From the figure, it is seen that power is delivered from the low voltage solar array (at 69 V) to a 67 V bus (with a 2 V drop through an isolation diode and harness). From the solar array the 67 V bus is maintained at voltage by the voltage sensor calling for the partial shunt regulator (dissipator) to request power from the array or to dissipate power as required to maintain the bus at 67 Volts. The partial shunt dissipator parallels 67 percent of the array.

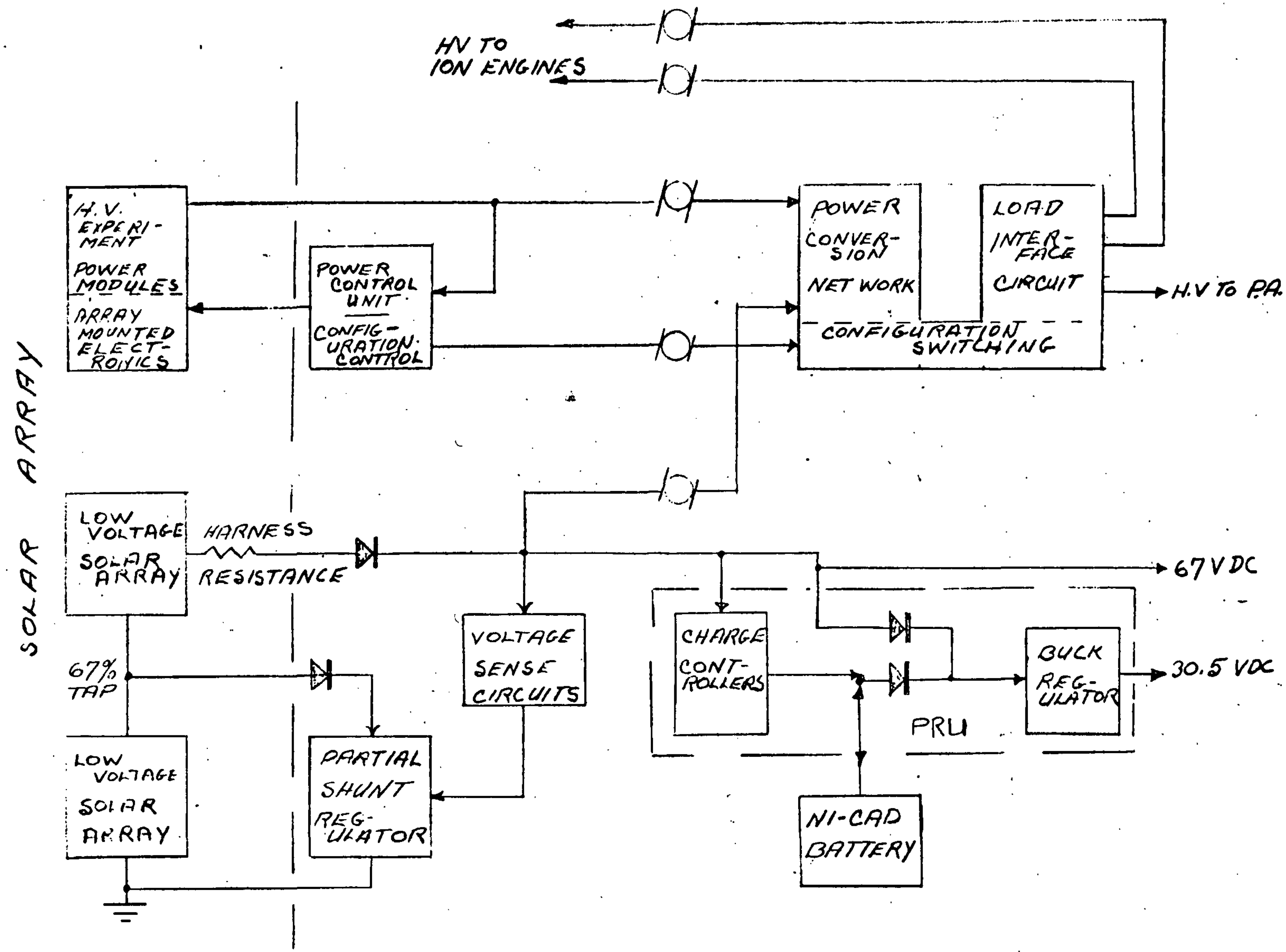


Figure 4.5-1. ATS-AMS-III Power Subsystem Block Diagram

The power regulation unit consists of the charge controller and discharge (buck) regulator of the PWM (pulse width modulation) type. The discharge regulator provides 30.5 V power for low voltage communication and housekeeping power from the 67 V bus during daytime and housekeeping power from the battery during occult. The regulator functions as a nondissipative, tapered current, temperature compensated bucking charger. It's conversion efficiency is 89%. Isolation diodes from both sources each take a one volt drop. The charge controllers operating off the 67 V bus maintain and recharge the battery during each day to replenish the occult discharge (maximum of 1.2 hours/day during equinox). Each string of batteries requires its own dissipative-type charge controller containing current, voltage and temperature sensing and limiting functions. Charge current is controlled so as not to damage the battery with high end-of-charge voltages or overcharge rates.

The Power Converter Network conditions high voltage power for operation of the ion engines (during orbit control) and of the power amplifiers with power normally supplies from the 67 V bus. The conversion efficiency is about 86%. As it is contemplated supplying high voltage to several power amplifiers in parallel there will be a load interface circuit providing load isolation, protection and load switching.

Also shown in the diagram are high voltage solar array (HVSA) modules with array-mounted electronics for conducting high voltage experiments. The modules of the high voltage array are connected in series-parallel combination to provide power directly to the high voltage power loads.

The power control unit (PCU) reconfigures the power converter network and provides voltage regulation for the HVSA. Upon ground command, the high voltage modules can be substituted for converter modules to supply biased high voltage directly to the load.

As shown in Figure 4.5-1, an interface exists between the main spacecraft body and the antenna support platform. Only DC power is transferred across the rotating joints. Power transfer is accomplished by means of liquid-metal (gallium) sliprings located on the main drive shaft. The power converter network is located on the rotating platform thereby minimizing the number of sliprings that must carry high voltage power. However, power at a 1.2 kV level will be returned from the platform to the ion engines for orbit control. A detailed study will determine the best location for the Power Control Unit. At present, in deference to the location of the high voltage solar array experiment, it is located within the main spacecraft body.

Table 4.5-3 summarizes the characteristics of the ATS-AMS-III power subsystem.

4.5.2 ATS-AMS II POWER SUBSYSTEM CONFIGURATION

The power subsystem for this direct-ascent spacecraft is required to provide primary power at low voltage with high voltages obtained directly from the solar array as an experiment. The Shunt Regulator Subsystem, shown in functional block diagram form in Figure 4.5-2, has been selected for this application. The bulk of the array power is generated at 30.5 VDC. A converter network, powered from the regulated bus, provides high voltage to the ion engines or transmitter tubes. An on-board control unit, containing a stored-program computer, reconfigures the network by ground commands according to a spacecraft operational requirements. Otherwise, the shunt regulator subsystem operates as described in the following paragraph.

The state of the subsystem can be in only one of three operational modes depending upon available solar array power, battery charge

Table 4.5-3. Summary of Power Subsystem Characteristics for ATS-AMS III Spacecraft

Requirements	ATS-AMS III A		ATS-AMS III B	
Primary Array Voltage	67 V		67 V	
High Voltage Requirement	Experiment		Experiment	
Housekeeping Power	538 W		383 W	
CAPABILITIES				
Start of Ascent Array Power	9.6 kW at equinox		7.5 kW at equinox	
End of Ascent Array Power	9.6 kW at equinox		7.5 kW at equinox	
End of Mission Array Pwr	7.8 kW at solstice		6.1 kW at solstice	
COMPONENT AREA/WT.	AREA	WEIGHT	AREA	WEIGHT
Solar Array*	87 m^2	177 kg	68 m^2	134 kg
o 4 cm^2, N-P, 0.20 mm thick				
o 0.10 mm fused silica C. G.				
o 0.05 mm Kapton Substrate				
* Includes deployment and/or rotation or articulation mech.				
Battery				
o 128 cells, 12 Ahr NiCd		94 kg		
o 128 cells, 9 Ahr NiCd				71 kg
Electronics				
o Power Regulation Unit		7 kg		5.5 kg
o Shunt Dissipator		54 kg		42 kg
o Power Control Unit		3 kg		3 kg
o Power Converter Network		43 kg		35 kg
Harness		14 kg		14 kg
TOTAL		392 kg (860 lb.)		305 kg (670 lb.)

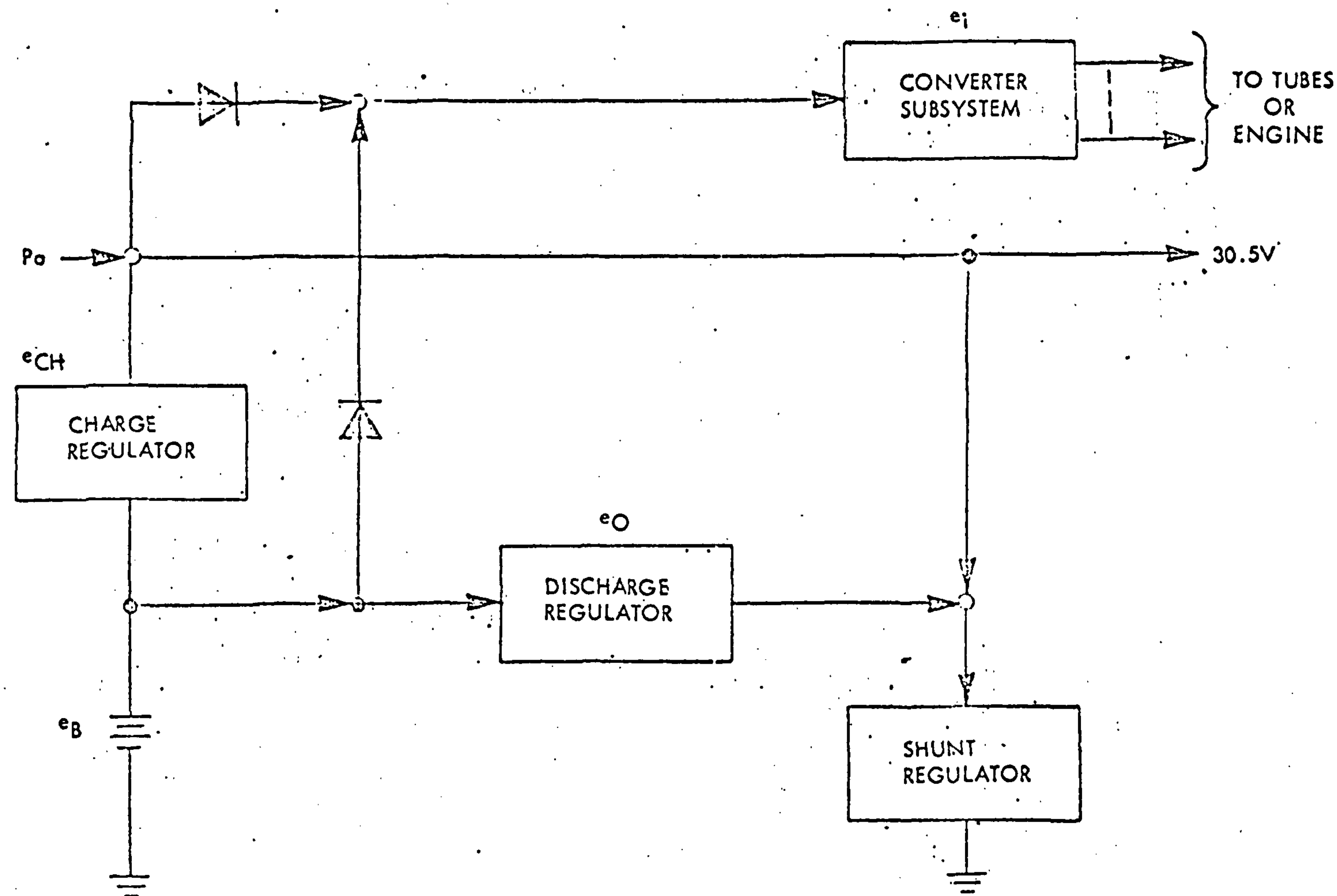


Figure 4.5-2. Low Voltage Power Subsystem Block Diagram - AMS-II

requirements and instantaneous spacecraft electrical loads. The modes are: battery discharge, battery charge and shunt regulation. In the first mode, which occurs during occult and periods of peak demand, the battery discharges through a PWM-type voltage-boost regulator. In the second mode the array is capable of charging the battery and supplying spacecraft loads. In this case the dissipative-type charge controller maintains the regulated solar array bus with both the boost and shunt regulators turned OFF. The third mode occurs when array power is in excess of load and battery charge requirements. The shunt regulator dissipates this excess and acts to maintain voltage regulation. The three regulators are turned ON and OFF in proper sequence and proportioned as a result of the measurement of very small changes in regulated bus voltage which correspond to source - load power conditions.

The solar array design features will be similar to the blanket-type array later described in detail for the ATS-AMS I mission. Its area, however, will be 51 m^2 (550 ft^2). Other subsystem characteristics are summarized in Table 4.5-4.

The high voltage array experiment requires 0.9 m^2 (9.7 ft^2) allocated to two 1000 VDC blocks of solar cells. The blocks each contain nearly 4000 1 x 1 cm solar cells, blocking and coupling diodes, and semiconductor load, stacking, shorting and regulating switches. Configuration experiments and voltage regulation are controlled from the on-board computer which is otherwise used to reconfigure the modular converter network. Upon ground command, the blocks can be substituted for converter modules to provide biased high voltage to the transmitters. The experiment can be located either on array or, as described earlier, on the sun-facing side of the spacecraft.

4.5.3 ATS-AMS I POWER SUBSYSTEM CONFIGURATION

The ATS-AMS I mission requires that the solar cell array be capable of providing power to a cluster of ion engines during spiral-out ascent from low to synchronous orbit and then be capable of reconfiguration to provide high power at high voltages to TWT and Klystron transmitters. Sections of the array will be dedicated to categories of voltage requirements summarized as follows:

- Attitude control, telemetry, battery charging and other housekeeping requirements at 30.5 V nominal,
- Ion engine requirements with accelerator and screen voltages up to 1.2 kV,
- RF power amplifier requirements with cathode and collector voltages up to 16 kV,
- Special experiment voltages as required.

A functional block diagram of the ATS-AMS I power subsystem configuration is shown in Figure 4.5-3. The solar array is divided into low and high voltage sections. The low voltage section delivers power at 30.5 V through a shunt regulator power subsystem to the housekeeping loads. The high voltage section contains blocks or modules, capable of delivering nominally 1000 V. The blocks are connected in the proper series - parallel combination to provide power directly to the high voltage high power loads. Precise voltage regulation across a series chain of blocks is achieved by shorting out the proper number of series solar cells in each block. The reconfiguration and regulation functions are controlled by an on-board computer on a continuous basis. Although the low and high voltage arrays are shown independently, it is possible that, upon detailed trade-off and optimization analysis, the sections could be combined with a net savings of array area and weight.

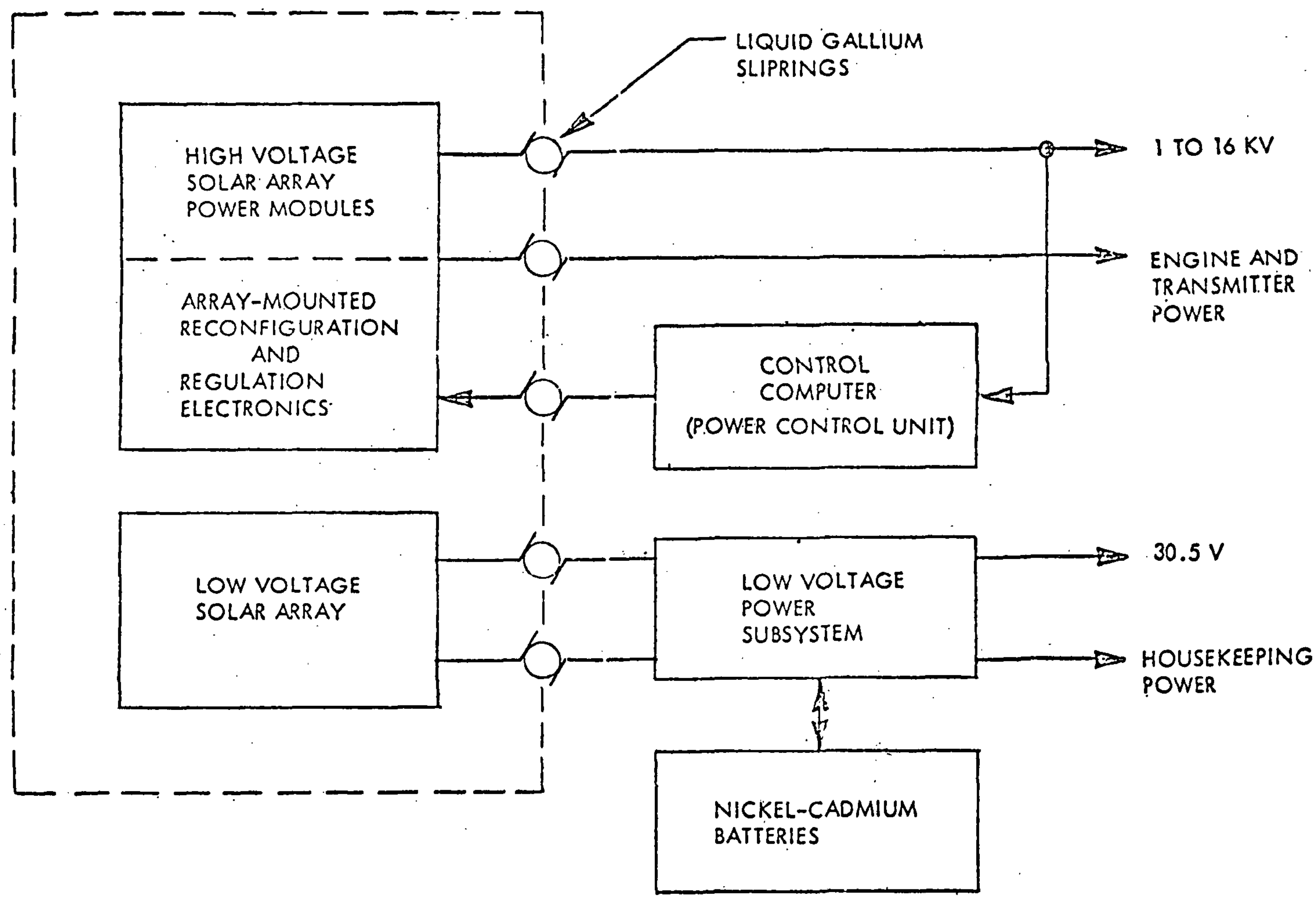


Figure 4.5-3. High Voltage Solar Array - Power Supply Subsystem Functional Block Diagram ATS-AMS-I

The low voltage section of the array will require approximately 15 m^2 (165 ft^2) of area covered by 2 x 2 (8 x .8 m) cells. Consequently, the array has been sized to provide 144 m^2 (1550 ft^2) of total area.

The array is capable of providing 13 kW of power to the ion engine cluster in a 2778 km (1733 mi) initial parking orbit.

The ascent program will place the spacecraft in orbit at synchronous altitude after approximately 182 days with 44.7 percent of the initial high voltage power (5.8 kW) remaining. However, the program assumes that the initial spacecraft weight is 1134 Kg (2500 lb). With a 144 m^2 (1550 ft^2) solar array the total spacecraft weight is estimated to be 1210 kg -- (2662 lb) a negative margin of 76 kg (167 lb). This implies an increase in ascent time of about 25 days and an increase in array degradation of about one percent. Therefore, the absence of reasonable power and weight margins and the imprecision of estimates of certain of the spacecraft component characteristics suggests that other configurations, of both power subsystem and spacecraft, be considered.

Table 4.5-4 summarizes the salient characteristics of the ATS-AMS I power subsystem.

4.5.4 Design of the Solar Array

The solar array has been sized to produce sufficient power after two years at solstice without array articulation. However, the capability to articulate the array through a range of at least ±23.5 degrees is included in the spacecraft design as an experiment. The obvious advantages are a savings of about 8 percent in initial design power and the potential reduction in the variation of shunt dissipator power in the spacecraft by purposely "spoiling" the acceptance angle during periods of light load.

Table 4.5-4. Summary of Power Subsystem Characteristics for ATS-AMS I and II

Requirements	ATS-AMS I		ATS-AMS II	
Primary Array Voltage	30.5		30.5	
High Voltage Requirement	1.2 - 16 KV		Experiment	
Housekeeping Power				
Capabilities				
Start of Ascent Array Power	13 KW		5.5 KW	
End of Ascent Array Power	5.8 KW		5.5 KW	
End of Mission Array Pwr	5.5		4.6 KW	
Component Area/Wt.	Area	Wt.	Area	Wt.
Solar Array*	144 m^2	306	51 m^2	102 Kg
o 4 cm^2, N-P, 0.20 mm thick				
o 0.10 mm fused silica C.G.				
o 0.05 mm Kapton Substrate				
* Includes deployment and/or rotation or articulation mech.				
Battery				
o 60 cells, 15 Ahr NiCd				48
o 120 cells, 15 Ahr NiCd				(105)
Electronics		(100)		
o Power Regulation Unit		7		5
o Shunt Dissipator		8		33 Kg
o Power Control Unit		3		3 Kg
o Power Converter Network		10		28 Kg
Harness		14		14 Kg
Total		448 (986)		233 (513)

The solar array system for ATS-AMS-III is similar to that for ATS-AMS I and II.

One of the panels for a solar array system, similar to that shown for ATS-AMS-III, is shown for ATS-AMS I and II in Figure 4.5-4.

It consists of identical arrays deployed on each side of the spacecraft. The arrays are non-rotating and non-retracting. Each array has a drum for storing the array during launch, a folding arm linkage, launch restraint structure and an array mechanism. No slip rings are required for powering the attitude control ion engines since these engines and their fuel are mounted on the end of the deployed boom. The engines are mounted to fire parallel to the axis of the boom and to provide attitude control of the spacecraft by electronically deflecting the thrust vector.

In the ATS-AMS III A case, the array aspect ratio is 9.55 to 1 (corresponding to an 87 square meter area). The array width of 244 cm (96 inches) is limited above by the antenna view angle of 1.5° and below by the ion engine contamination boundary. This boundary is the smallest view angle from the array to the engine nozzle with a projected nozzle shielding to exit material with a greater than 15° included plume angle. It is assumed that above this boundary line the array performance will not be affected by contaminants coming from the engines.

The array is deployed in a two step operation with the linkage extending first and the array extension following. Linkage deployment consists of releasing 5 hinged arms which are extendable from the spacecraft. The hinges are spring loaded to deploy the arm when a restraining cable is cut. Initial deployment is insured with a kick spring. The kick spring is necessary to overcome the small mechanical advantage of the torsion springs at full stowage.

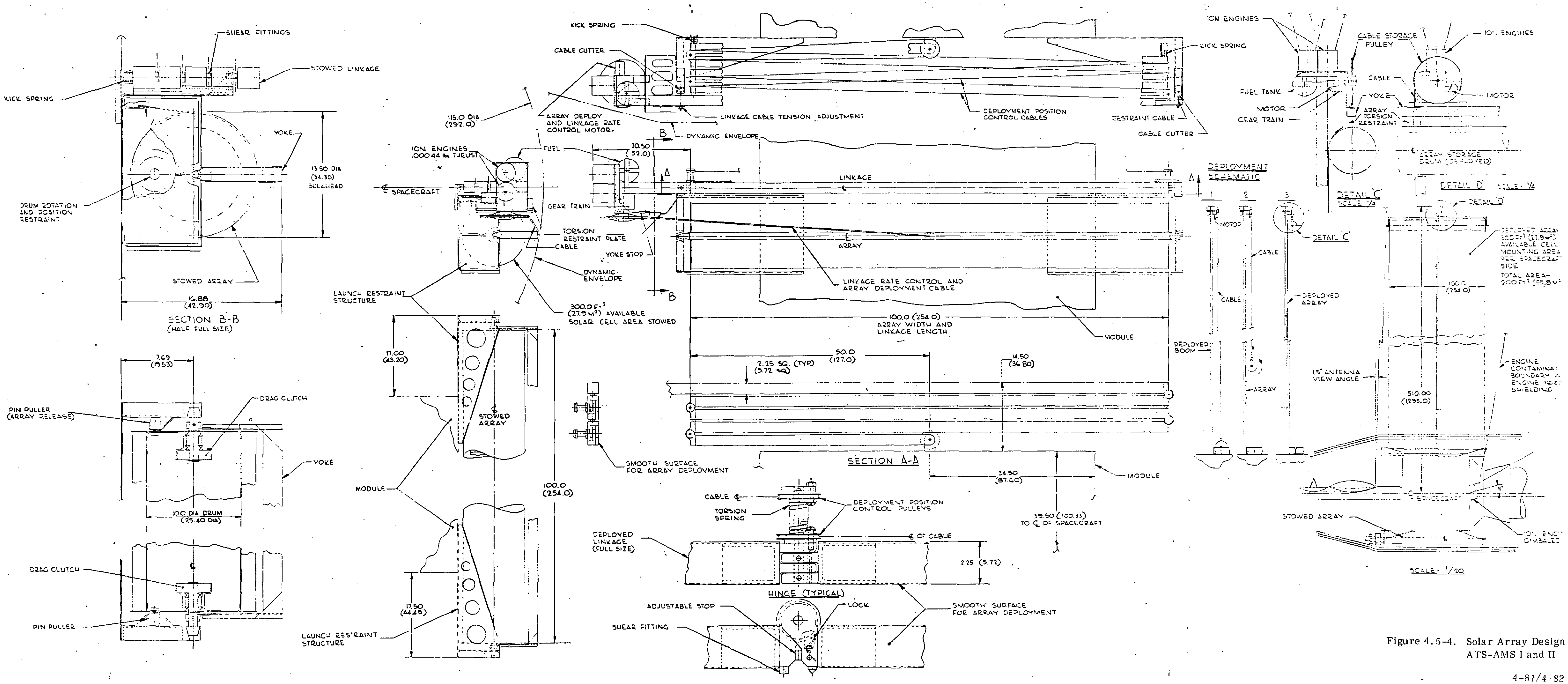


Figure 4.5-4. Solar Array Design
ATS-AMS I and II

The arms deploy in a programmed manner by using a parallel arm linkage system. The arms have pulleys fastened to them at the hinges. Cables wrapped around pulleys attached to alternate arms require that alternate arms rotate relative to one another. Using equal pulley diameters, alternate arms deploy parallel in a controlled manner. Different relative arm rotations are achieved by using different pulley diameters.

The rate that the linkage deploys is controlled by a motor, worm, gear train, drum, and cable system mounted on the end of the last arm. This same system is later used to deploy the array. A cable stored on the drum is attached to the fixed, stowed array. The rate that this cable unwinds from its storage drum on the last arm limits the arm deployment rate. Storage drum rotation is controlled by a DC motor, worm, and gear train.

Upon deployment the arms lock together forming a boom. The locks are positive, antibacklash, spring loaded locks. An adjustable stop insures boom alignment after deployment.

Launch loads from the folded arms are transmitted to the spacecraft by shear fittings on the hinges. These fittings are self-releasing when a launch restraint cable located beside them is cut by either of two pyrotechnics.

After deployment a structural boom is formed. One surface of the boom is designed to be smooth for the subsequent array deployment.

Array deployment is initiated by severing a cable which restrains the array in the launch restraint structure. The motor on the boom end winds up the cable attached to the array yoke and pulls the yoke and drum toward the boom end. Since one edge of the array is attached to the spacecraft, the bearing mounted array drum rotates relative to the translating yoke and deploys the array from the drum. A drag clutch on the yoke-drum shaft maintains array tension during deployment. The array unwinds from the drum such that there is no relative motion between the array and the boom. When the yoke and array drum reach the end of the boom they snap under an array torsion restraint plate. This plate is oversized to ensure yoke engagement and provide adequate array torsion control for the deployed system. No array torsion controls are used during deployment of the array. Array tension is determined by the drum-yoke drag clutch.

The array launch restraint structure transfers all array launch loads to the equipment module structure. The array is stored on a 25.4 cm (10 in) diameter drum. This diameter was determined from experimentation to be adequate for storage of the array. A small shift of the spacecraft occurs during array deployment. This shift is caused by the array changing its effective mass center from on the storage drum to beside the boom.

The high voltage solar array experiment will be located on the sun-facing side of the Equipment Module. It will consist of two blocks of 1 x 1 cm (.4 x .4 in) solar cells, each block capable of providing 1000 VDC. Of the 3.4 m^2 (36.6 ft^2) area available for solar cells, only 0.9 m^2 (9.7 ft^2) will be utilized to permit Optical Solar Reflector (OSR) temperature control. Accordingly, with 73 per cent OSR coverage, the solar cell temperature will be roughly 308^{o}K, 280^{o}K from standard temperature measurement conditions.

The solar array blanket structure proposed for all of the ATS-AMS spacecraft consists of solar cells of 2 x 2 cm (.8 x .8 in) size or less, 0.20 mm (0.008 inch) thick with 0.10 mm (0.004 inch) fused silica coverslides bonded to a 0.05 mm (0.002 inch) thick Kapton substrate. The average bare solar cell efficiency under 1 AU, 301°K of C start-or-mission condition is 11.4 percent. The estimated area density of the array blanket including interconnections and power switching electronics, is 4.83 kg/m^2 (0.204 lb/ft^2).

4.5.5 Power Converter Network (PCN)

For ATS-AMS III power conversion from the 67 V primary bus is required to provide high voltage for the ion engine(s) and for multiple power amplifiers (power amplifiers are on standby during orbit control when powering the ion engines). High voltage requirements for the TWT's and Klystron differ from that of the ion engine requiring reconfiguration of the PCN.

Figure 4.5-5 shows a simplified schematic diagram of a single TWT assembly with the high voltage supplies connected. Eight collector supply voltages are provided by two multiple secondary converters, each sensing a single output.

The 10 kV cathode supply is provided by a single converter using two, paralleled primary transformers with secondaries in series. Preliminary trade-offs have indicated that due to high voltage insulation and turns ratio requirements, the two-transformers, multiple secondary scheme appears to be optimum from size and

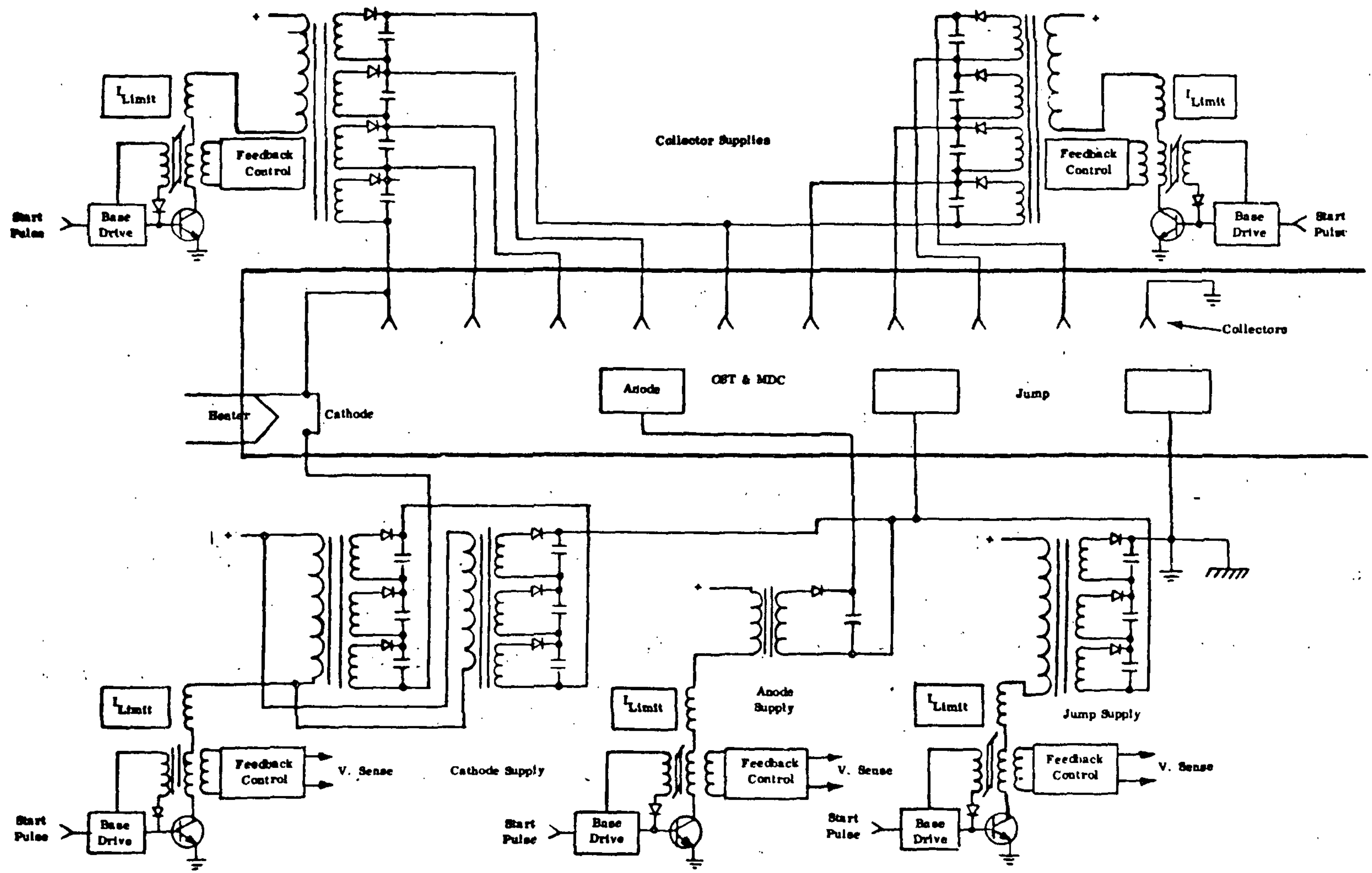


Figure 4.5-5. Simplified Schematic High Voltage Power Supplies

efficiency standpoints. Voltage sensing of the total secondary voltage is planned in order to achieve the specified regulation. Voltage adjustment can be easily achieved through the feedback control circuits.

The jump supply will employ multiple secondaries as shown in order to reduce the turns ratio between primary and secondary. Only one transformer is indicated. The anode supply is the smallest of the high voltage supplies in both voltage and power. Regulation requirements, however, dictate use of the full compliment of control circuitry. For both jump and anode supplies, full output sensing will be required. In the configuration shown conversion is based on the half-wave single step technique for converting from source (input) to the load (output).

The power converter network (PCN) for ATS-AMS-III will supply high voltage loads for up to ten 200 W TWT's (III A), a 2000 W TWT, a 1000 W Klystron, 5 W TWT's (Audio) and the ion engine.

The different power requirements for ion engines and the need for reconfiguration in shifting load between TWT's (e.g., replacing 200 W TWT's with a 2000 W TWT or Klystron) favors consideration of separate power supplies for each unit. Further study is required to evaluate the separate VS combined supply approach with potential weight saving at the expense of complexity favoring the single supply with a combined PCN. The latter approach has been postulated with weight estimates made for a PCN for ATS-AMS-III A and B as described in Section 5.5. Several techniques have been considered for the DC-DC (67 V to HV) conversion to provide both level-setting and regulation. Both single step and two-step conversion techniques have been studied. The two-step approach is more complex but generally has superior dynamic regulation and output level characteristics. Based on efficiency, relatively low parts counts and output bus characteristics a single step half-wave conversion technique representing the best compromise and is suggested for ATS-AMS-III.

4.5.6 Load Interface Circuit (LIC) and Reconfiguration Circuit

Interconnection of the Power Control Network and the various High Voltage Loads will be through the Load Interface Circuit. Each individual load will be capable of being energized or de-energized by ground command. Switching will be via high voltage reed relay assemblies configured specifically for this purpose. In addition, each high voltage lead will be fused for protective reasons and will have a fuse-defeat circuit, which upon ground command, may permit reactuation or "Try-again" techniques in the event of tube failure due to a partial short circuit.

In summary, the LIC acts as a switching and distribution system for the High Voltage Loads and also provide protection to the PCN.

Figure 4.5-6 represents a typical Load Interface Circuit showing the energize-de-energize and the fuse defeat relays. There will be a similar LIC for each high voltage load.

The Reconfiguration Circuit upon receiving appropriate ground commands modifies the PCN circuits to provide the output voltage levels required for the particular operational mode desired. This is done by changing the individual building-block voltages and by combining them via suitable relay arrangements. The individual output voltages may be varied by changing reference voltages.

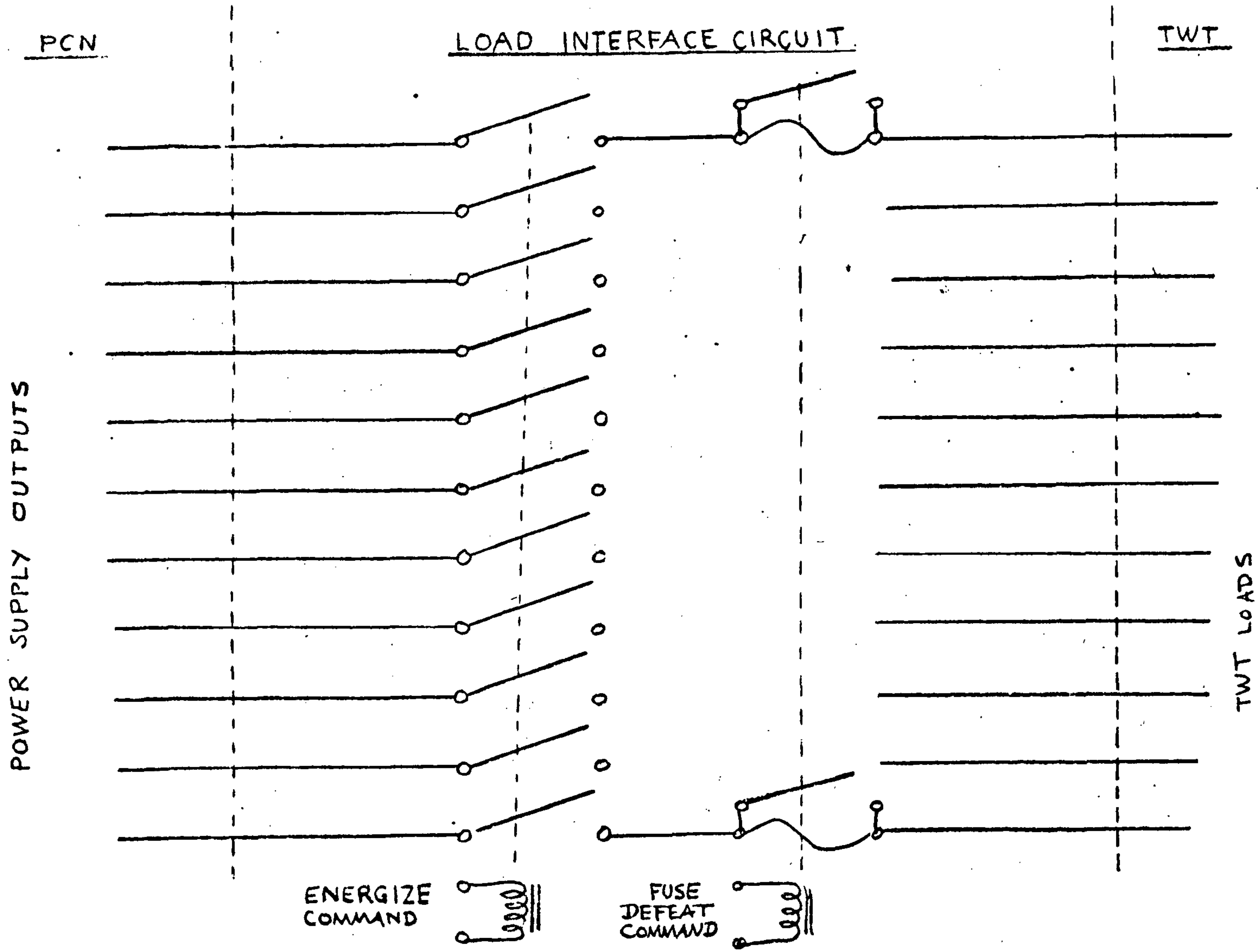


Figure 4.5-6. Typical TWT Load Interface Circuit

4.6 ATTITUDE CONTROL SUBSYTEM

4.6.1 SUBSYSTEM OPERATION AND REQUIREMENTS

The attitude control subsystem (ACS) is required to establish and maintain the attitude of the spacecraft and its payload such that in the primary control mode the boresight axis of the antenna beams point to the desired location on Earth with a tolerance of $\pm 0.2^{\circ}$ for longitude and latitude, which when converted to the spacecraft angular rotations about the orbit normal and the orbit tangent axes of the body-fixed reference frame amount to $\pm 0.03^{\circ}$ tolerance at the geosynchronous equatorial orbit altitude. A similar tolerance of $\pm 0.03^{\circ}$ is also required for the spacecraft attitude about the third axis of the mentioned reference frame (local vertical) since angular motions about this axis affect the longitude or the latitude of antenna beams not pointing along the local vertical, but offset respectively along the local meridian or the equator. The ACS is also required to orient the flats of the solar arrays so that the perpendicular to their surface tracks the projection of the spacecraft - sun line in the orbit plane with a tolerance of $\pm 0.5^{\circ}$.

As it is described in Section 4.1, the spacecraft configurations ATS-AMS III and ATS-AMS II are Sun oriented with an Earth oriented rotating antenna tower while ATS-AMS I is Earth oriented with (Sun oriented) rotating solar arrays. For attitude control purposes, it is convenient to define reference frames. The first one, also referred to as the orbital reference frame is defined as follows:

OXYZ	Orthogonal (Cartesian) axes with the origin at the spacecraft center of mass
OX	Roll axis, tangent to the orbit and oriented in the direction of motion
OY	Pitch axis, perpendicular to the orbit plane and oriented in the direction N-S (for an equatorial orbit)

OZ Yaw axis, completing a right handed reference frame, therefore pointing toward the Earth center.

The second reference frame, referred to as pseudo-inertial, is defined as follows:

$OX_IY_IZ_I$ Orthogonal reference frame with the origin coincident with that of the other frame

OX_I Axis aligned with the projection of the spacecraft to sun line in the orbit plane and oriented toward the sun.

OY_I Axis perpendicular to the orbit plane and oriented in the N-S direction for a spacecraft in equatorial orbit.

OZ_I Axis completing a right handed cartesian frame.

In the discussions that follow, motion about the orbit normal vector will be referred to as "pitch", motion about the orbit tangent vector will be referred to as "roll" and motion about the earth center vector will be referred to as "yaw". East/West stationing will require forces parallel to the roll vector and North/South stationing will require forces parallel to the pitch vector. A sketch of the spacecraft and its reference frames is shown in Figures 4.6.1-1 and 4.6.1-2 respectively for the ATS-AMS III and II and for the ATS-AMS I configurations.

The ATS-AMS III consists of a cube shaped equipment module approximately 1.8 m (6 ft.) on a side, symmetrical roll out solar arrays with a "wing-spread" of 42.5 m (140 ft.) and a rotating antenna and transponder equipment tower. The 30 cm North/South station keeping ion thruster is mounted on the face of the equipment module opposite to the face that mounts the rotating antenna tower and the axis of thrust lies along the axis of rotation. The solar arrays are firmly attached to the spacecraft body and the center line of the array is orthogonal to the spacecraft tower-thruster axis. The flats of the arrays parallel the tower thruster axis. In orbit, the spacecraft will be oriented so that the antenna tower axis of rotation is normal to the orbit plane and the center line of the solar array is parallel to the orbit plane. The spacecraft will

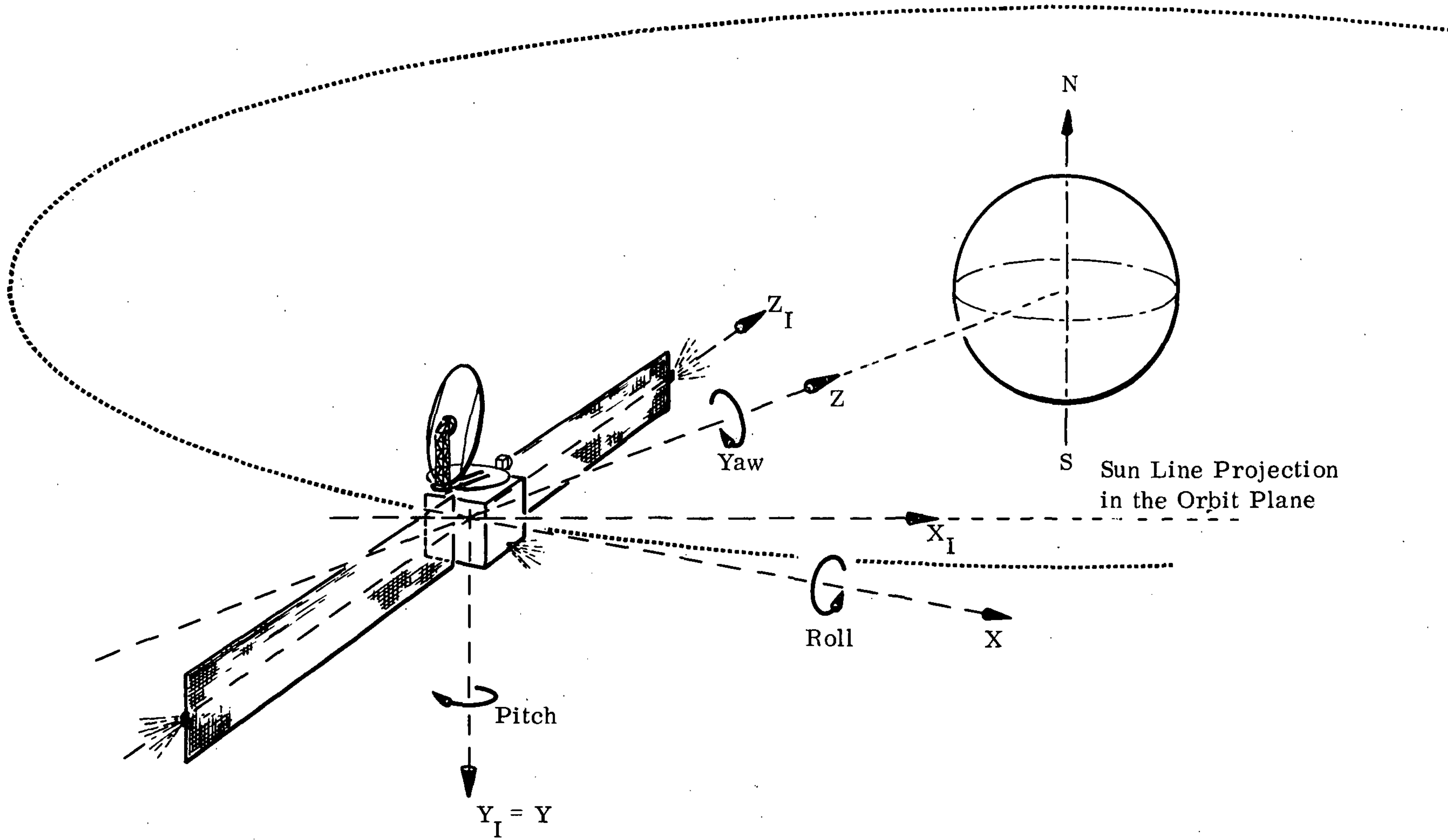


Figure 4.6.1-1. Reference Frames for the ATS-AMS III and II Configurations

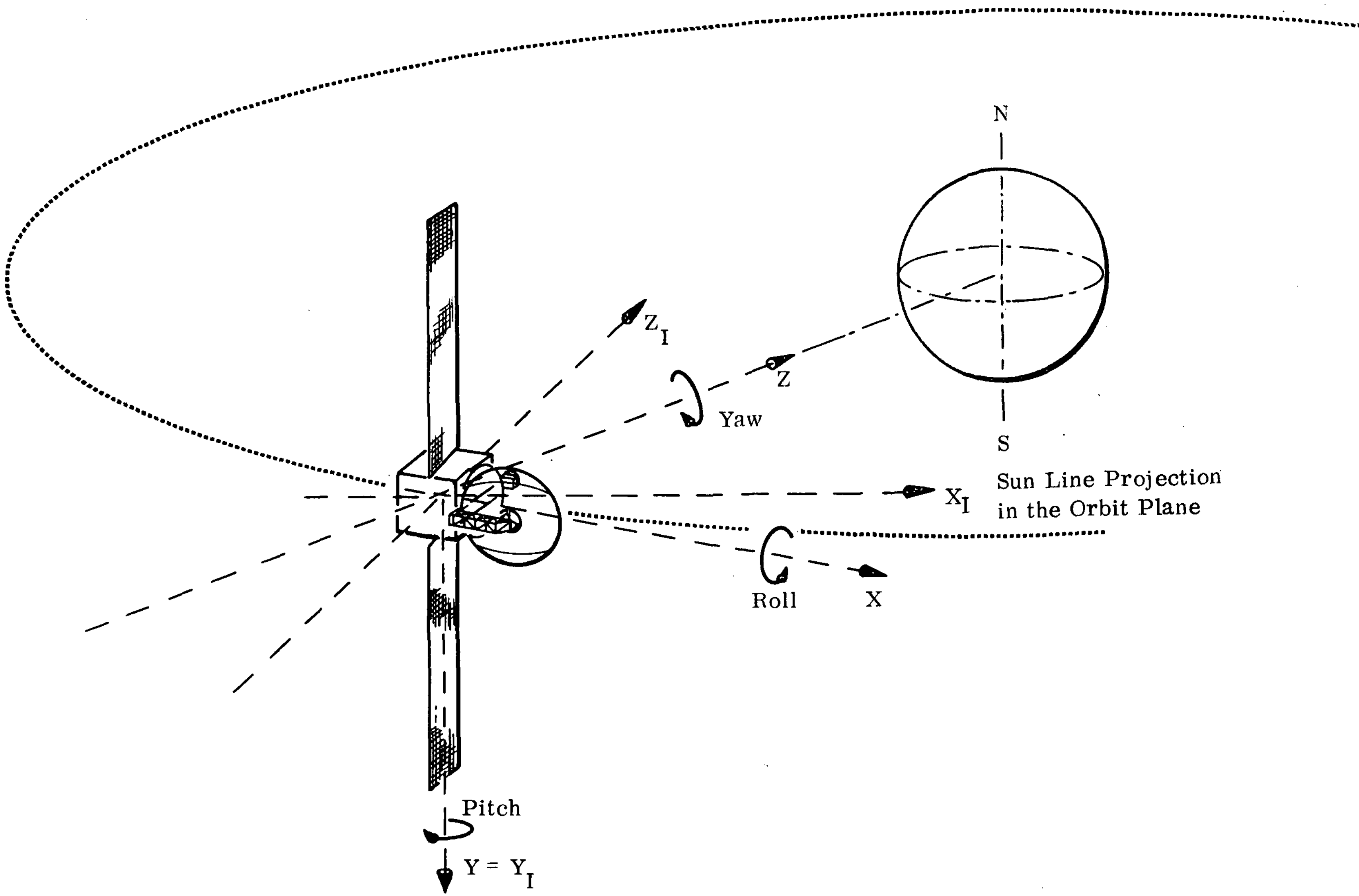


Figure 4.6.1-2. Reference Frames for the ATS-AMS I Configuration

be controlled in pitch to maintain the flats of the array normal to the projection of the sun line in the orbit plane except when the array is purposely misaligned for power regulating purposes. The spacecraft will be controlled in roll and yaw to maintain the tower-thruster axis normal to the orbit plane. An antenna drive servo loop utilizing either an earth sensor or an interferometer will be used to control the east-west orientation of the antenna beam. It is currently planned to fix the north-south axis of the beam by locating the antenna and feed structure on the spacecraft such that with the pitch axis normal to the orbit, the antenna beam center will be at the desired latitude. A capability for north-south articulation of the antenna and feed could be incorporated into the spacecraft design without significant weight, cost or complexity penalities. It should be noted that with this spacecraft configuration and with the reference frame selected, the solar array centerline alternately parallels the roll and yaw axes.

The ATS-AMS II is basically similar to the one previously described.

The ATS-AMS I configuration is also similar except for the following elements:

1. The equipment module is Earth oriented, thus the antenna and RF equipment are rigidly attached to the module.
2. A solar array drive is used to decouple the array motion from the spacecraft orbital motion, thus maintaining the array oriented to the Sun.
3. The spin axis of the array, also their centerline, is the pitch axis. Thus the antenna tower axis is oriented along the roll axis.
4. The spacecraft will be controlled in roll and pitch in order to control the N-S and E-W orientation of the antenna beam and in yaw in order to control both orientations for antenna beam pointing offset from the local vertical.

4.6.2 ACS CONFIGURATION FOR ATS-AMS III & II

The attitude control concept selected for the ATS-AMS III and II configurations is three-axis stabilization using momentum storage devices (momentum wheels) complemented by 5 cm ion thrusters. The storage devices will control or store momentum due to cyclical torques on the spacecraft and the ion thrusters will overcome long term constant disturbance torques by periodically desaturating the storage devices by expelling mass from the spacecraft. Three momentum wheels (one for each pseudo-inertial axis) will be mounted on the equipment module to provide control torques about the X_I, Y_I, and Z_I axes. A vectorable set of thrusters mounted at the tips of the solar array erection masts are used for obtaining complementary control torques about the X_I and Y_I axes.

An additional set of 5 cm ion thrusters will be located on the equipment module as indicated in Figure 4.6.1-1 to provide torquing capability about the Z_I axis.

Disturbance torques acting on the spacecraft arise from solar radiation pressure, gravity gradient, spacecraft magnetic moment and moving masses within the spacecraft. These torques have both secular and cyclic components. Solar disturbance torques will be predominantly secular and gravity gradient torques will be predominantly cyclic. The mass properties, the secular and the cyclic disturbance torque magnitude for the ATS-AMS III spacecraft are approximately:

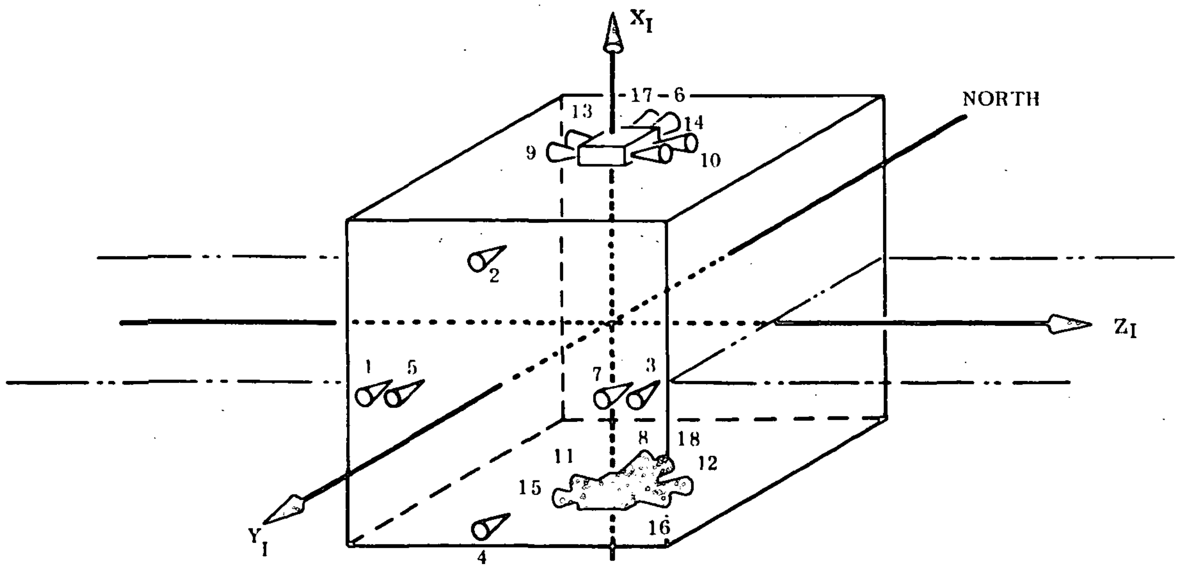
Pseudo Inertial Axes	Moment of Inertia		Secular Disturbance Torque		Cyclic Disturbance Torque	
	$kg\text{-}m^2$	$(slug\text{-}ft)^2$	Newton-meters	(ft-lb)	Newton-meter	(ft-lb)
X_I	25,100	(18,600)	——	——	$\pm 18\times10^{-8}$	$(\pm 13.3\times10^{-8})$
Y_I	23,900	(17,700)	4.06×10^{-5}	(3.01×10^{-5})	$\pm 19.4\times10^{-5}$	$(\pm 14.4\times10^{-5})$
Z_I	2,300	(1,700)	12.3×10^{-5}	(9.15×10^{-5})	$\pm 5.07\times10^{-5}$	$(\pm 3.76\times10^{-5})$

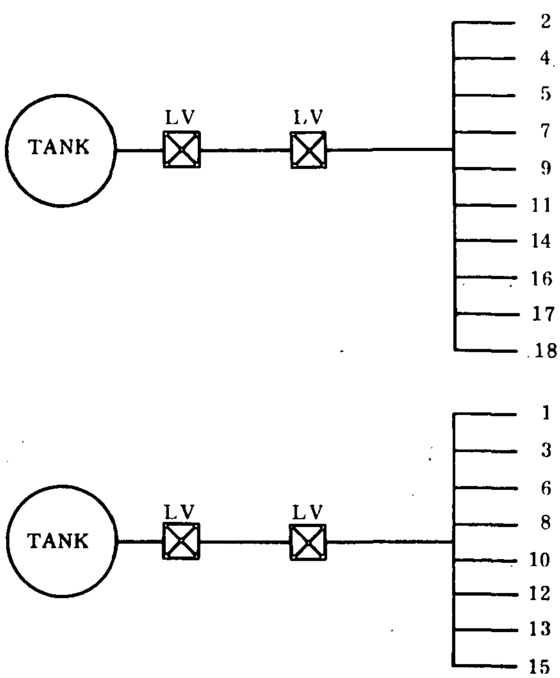
$Slug\text{-}ft^2 = 1.35\ kg\text{-}m^2$; ft-lb = 1.35 Newton-meter

The control torques obtainable from the 5 cm ion thrusters mounted on the equipment module (Z_I axis torques) are approximately one order of magnitude larger than the Z_I axis solar disturbance torques and are therefore sufficient for attitude control. The momentum storage capability of the momentum wheels was selected to provide in excess of an integrated half cycle of cyclic disturbance.

A backup hydrazine reaction control subsystem, similar to that used for ATS-F/G was also selected. This subsystem would feature 0.222 N (0.05 lbf) thrusters mounted on the spacecraft equipment module. The location, orientation and number of these thrusters is given by Figure 4.6.2-1. The primary use of this subsystem will be to provide for East/West station keeping. Tankage and plumbing is such that a backup dual redundancy for both directions of station keeping, unloading of momentum wheels, and all three axes of attitude control can be achieved, for a 1725 kg (3800 lb) spacecraft, at a weight penalty of approximately 40.7 kg (89.5 lb) per year.

The three axis stabilization concept described will provide spacecraft stabilization relative to the pseudo-inertial reference frame. The antenna drive servo loop, utilizing either an earth sensor or an interferometer will be used to provide an antenna tower spin rate of 1 r/day for antenna beam pointing on earth and pitch axis attitude control. The functional block diagram describing the overall ACS configuration is given in Figure 4.6.2-2.





Latch valves ground commandable to isolate and to jet manifold on any tank

Jet valves have double solenoids and seats

Torquing Mode	$-Y_I$	9 & 16 or 13 & 12
	$+Y_I$	10 & 15 or 14 & 11
	$-X_I$	1 or 5
	$+X_I$	3 or 7
	$+Z_I$	4 or 6 or 17
	$-Z_I$	2 or 8 or 18
	E Z_I	9 & 11 or 13 & 15
	W $-Z_I$	10 & 12 or 14 & 16
	N $-Y_I$	1 & 3 or 2 & 4 or 5 & 7
	S Y_I	17 & 18 or 6 & 8

E/W only collinear with Z_I axis when spacecraft is at point in orbit closest to sun.

Figure 4.6.2-1. Back-Up Hydrazine Thruster Matrix

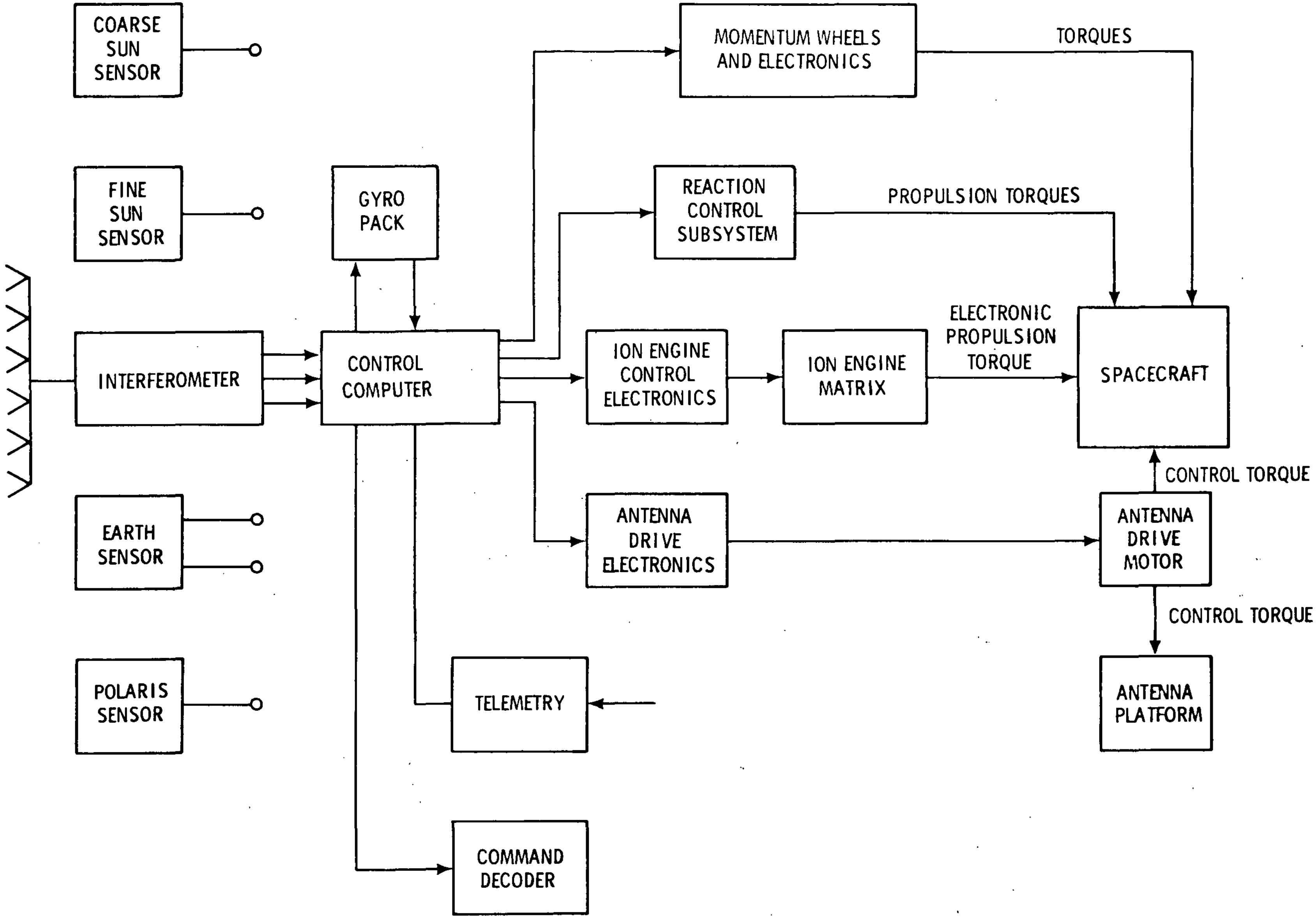


Figure 4.6.2-2. ACS Configuration, ATS-AMS II and AMS III

The sensor, actuators and associated electronics composing the ACS for the configurations ATS-AMS II and III are listed in Table 4.6.2-1, which include component weight and power. The ATS-AMS III and II, being injected directly into synchronous equatorial orbit, would not require the IRU (Inertial Reference Unit) which is required for the I configuration. A rate gyro package is, however, required for acquisition purposes. A number of control modes are possible and the use of the control computer is used to resolve the spacecraft referenced sensor outputs into the orbit centered attitude reference frame and to reconfigure the control laws to permit alternate modes of control.

Table 4.6.2-1. ACS Components for ATS-AMS II & III

EQUIPMENT	NUMBER OF UNIT	WEIGHT kg (lb)	POWER (Watts)
Hydrazine Propellant (e-w station keeping for 5 years)	1	9.1 (20)	——
Coarse Sun Sensor	6	.226 (.5) ea.	——
Fine Sun Sensor	2	.338 (.75) ea.	1.0 each
Sun Sensor Electronics	1	1.000 (2.2)	2.0
Rate Gyro Package	1	2.256 (5.0)	12.0
Interferometer	1	6.680 (15.0)	16.0
Earth Sensor Heat	2	1.760 (3.9) ea.	15
Earth Sensor Electronics	1	2.710 (6.0)	
Polaris Sensor	1	4.880 (10.8)	15
Hydrazine Subsystem (including fuel tank)	1	25.5 (56.1)	——
Jet Valve Drivers	1	2.620 (5.8)	10
5 cm Ion Engines (incl. fuel tank and fuel for 5 years)	6	6.650 (14.6) ea.	60 each
Antenna Drive & Slip Rings	1	12.700 (28)	15
Control Computer	1	1.355 (3.0)	10.0
Momentum Wheel	3	8.85 (19.5) ea.	53 each
Momentum Wheel Electronics	1	1.8 (4)	5
Hydrazine Propellant (backup for 1 year)	1	40.7 (90)	——

The $\pm .03^{o}$ attitude accuracy requirement requires use of the Interferometer as the primary attitude sensor. Solar panel orientation is sensed by a digital Sun Sensor mounted on the face of the equipment module that is on the sun side of the spacecraft and parallel to the plane of the solar array. The output of this digital Sun Sensor can, in conjunction with the Earth sensor, the known sun-hour angle and an accurate antenna tower position encoder, be used as an alternate method of attitude sensing and East-West antenna beam control. An Earth sensor and a Polaris sensor mounted on the antenna tower, are used to provide degraded mode ($.1^{o}$ instead of $.03^{o}$ accuracy) attitude sensing. The hydrazine backup subsystem in connection with the control computer can be used instead of the ion engines in a self-adaptive precision pointing jets only control mode in a manner similar to the SAPPSAC* equipment that is being flight tested as a part of the ATS-F/G program. The various sensing and control modes are summarized in the following matrix.

*Self Adaptive Precision Pointing Satellite Attitude Control

	Sensing Mode		Control Mode
1.	Interferometer (P.R.Y)	A.	Three axis - Momentum wheels complemented by the ion thrusters
2.	Sun Sensor and Clock Earth Sensor (P.R.)	B.	Three axis - Momentum wheels complemented by the hydrazine thrusters
3.	Earth Sensor (P.R.) Polaris Sensor (Y)	C.	SAPPSAC hydrazine thrusters

This 3 x 3 matrix will permit 9 modes of control.

4.6.3 COMPONENT DESCRIPTION

o Sun Sensors

Two types of Sun Sensors will be required, one for the coarse mode and one for stabilized fine pointing mode to determine where the sun is for initial stabilization. The selected sensors are both manufactured by the Adcole Corporation, Waltham, Massachusetts. The The characteristics of each sensor are listed in Table 4.6.3-1.

Table 4.6.3-1. Digital Solar Aspect Sun Sensor Characteristics

MODE	COARSE	FINE
Information Axes	Two	Two
Size (cm)	8.1 x 8.1 x 2	10 x 9.7 x 3.3
Model No. (Adcole)	14135	15380
Field of View (deg)	128 x 128	64 x 64
Resolution (deg)	.5	.0039
Accuracy (min)	15	———
Weight (kg)	.23	.34
Power (W)	None	None
AMPLIFIER & STORAGE REGISTER CHARACTERISTICS		
Size (cm)		19.3 x 11.4 x 3.8
Weight (kg)		1.02
Power (W)		2
Subsystem Output	1 or 2, 8 bit Binary Numbers	1 or 2, 14 bit Binary Numbers

Six coarse mode sensors, providing 4 π steradians of viewing (with sufficient overlap) and two fine pointing sensors will be used. A coarse sensor will be mounted at the tip of each solar array erection mast where the center line of the field of view of each sensor is parallel to the centerline of the masts. Additionally, a coarse sensor will be mounted (1) on the top of the reflector at its center of rotation (when not deployed) and directed in the Northern direction, (2) on the south face and directed in the Southern direction, (3) such that its FOV is in the X_I direction, and (4) with its FOV in the $-X_I$ direction. Redundancy is not considered necessary for the coarse mode sensors because of the relatively short life requirement and their inherent high reliability.

A common electronic box will be used as an amplifier and storage register since signals from only one sensor at a time will be processed. Selected circuit redundancy will be employed to meet the long life requirements. The characteristics of the electronic box are also tabulated.

- <u>Interferometer</u>

The spacecraft interferometer consists of the following components

a. Antenna Array Assembly

b. Receiver/Converter Assembly

The interferometer is a precision, wide field-of-view attitude sensor for the spacecraft attitude control system.

It measures the phase difference of signals received at paired antenna elements and converts this information into digital data, which is then related to spacecraft attitude.

The interferometer can be used as a three axis sensor. The interferometer accuracy requirements are specified in the following:

a. Attitude Determination - The space angle between the line-of-sight from the spacecraft (SC) to the ground emitter and the SC apparent yaw (Z)-axis shall be measured with an accuracy of ± 0.02 degrees (3σ) in pitch and roll axes over a field-of-view of ± 12.5 degrees about the yaw (Z)-axis.

b. Relative Angle Measurement - The space angle between the ground emitters shall be measured with an accuracy of ± 0.015 degrees (3σ) in pitch and roll axes within a field-of-view ± 12.5 degrees about the SC yaw (Z)-axis.

o Earth Sensor

A space proven, non-moving part radiation balance earth sensor, manufactured by Quantic Industries was selected. The characteristics of the sensor are given in Table 4.6.3-2.

Table 4.6.3-2. Earth Sensor Characteristics

Size	20.3 cm diameter by 17.8 cm (8 x 7 in)
Weight	3.2 kg (7 lbs.)
Power	2.2 W
Volts	22-31 DC
Type	Radiation Balance, No Moving Parts, 24 Thermocouple Detectors
Accuracy	0.1° (Non-Linear Off Axis)
Field-of-View	$\pm 5^{\circ}$ Linear
	$\pm 12^{\circ}$ Saturated
Sensitive Axes/Head	2
Altitude Range	35,000 - 36,134 km (18,900 - 19,500 nmi)
Output	Analog .5V/deg.
Sun Rejection	Avoid by static switching

The sensor contains 24 thermocouple detectors separated into four quadrants. Error signals are generated by adding and subtracting the signals from the thermocouples in each quadrant. These calculations are performed and the resultant error signals amplified by a electronic circuits mounted in the detector head. The output is and analog signal of .5VDC/degree of error for each axis which is linear with $\pm 5^{\circ}$ of offset and non-linear from $\pm 5^{\circ}$ to $\pm 12^{\circ}$. The sensor and amplifier are saturated at $\pm 12^{\circ}$.

If the sun is in the field of view, a large error signal will be generated. This signal will exceed the threshold level and the sensor output will be commanded off by logic in the electronics. The design of the detectors is such that the sensor will not be damaged by direct sunlight.

- Polaris Sensor

The Polaris sensor selected is a non-moving part sensor manufactured by Honeywell and similar to the Canopus sensor used on Mars Mariner flights 4 through 9. Such a sensor is currently specified for ATS-F/G. The diurnal motion of Polaris ($\pm$55 arc-min) appears as an oscillation in yaw once per orbit. To accommodate this oscillation, the sensor is gimballed and scans electronically. The instantaneous, scan and total field of view are given in Table 4.6.3-3 along with size, weight and power data.

- Attitude Control Computer

A small, lightweight, solid state computer made by CDC (model 469) was selected for use in the ACS. A description is given in Table 4.6.3-4.

A redundant back-up unit could be incorporated although the MTBF is computed to be in excess of 250,000 hours. Alternatively, the memory could be doubled to 32 k providing internal redundancy with only a slight increase in weight and size over the present 16 k memory.

Table 4.6.3-3. Polaris Sensor Characteristics

Physical Characteristics		
Weight	Size	Power
Optics/Electronics 17.6 kg Baffle 4.4 22.0 kg	Optics/Elec. 29.2cm x 11.1cm x 13cm Baffle 18.2cm x 13.5cm x 31.6cm	10.0 watts normal 17.0 watts sun-in-field (Includes LIC Power)

Performance

- Accuracy 3 sigma: 0.024° / 0.050° specified
- Data Resolution .0137° (analog)
- Noise Angle 3 sigma: 0.012° test data MM '69 / 0.020° computed noise
- Output Data 9 bits (8) (bits + sign)

SCAN CHARACTERISTICS

- Inst. F.O.V. 1° yaw x 11° roll
- Scan. F.O.V. 3° yaw x 11° roll @1200 Hz
- Total F.O.V. 7° yaw x ±14° roll (linear range) / 9° yaw x ± 14° roll (Saturated range)
- Roll Offset: 5 steps, commanded

Operating Characteristics

- Spectral Bandpass 0.3 to 0.6 microns
- Optics: Aperture 1.7 cm
  Focal Length 2.13 cm (f/1.25)
  Resolution 197 lines/cm
- Detector: Image Dissector Tube, Electrostatic Deflection
- Photocathode Type: S-11
  Uniformity 20%
- Time constant < 1 s
- Operating Temperature (baseplate) +10°C to +30°C
- Vibration Environment: Mariner Mars '69 / Titan IIIC

Table 4.6.3-4. Attitude Control Computer

GENERAL DATA
Weight: 1.36 kg. (3 lbs)
Size: 10.2 cm x 10.2 cm x 6.4 cm (4" x 4" x $2\frac{1}{2}$") with 16-bit 4 k memory, expandable to 65 k. 16 k memory used.
Power: 10 W
Input Voltages: ±15 VDC, ±5 VDC
Environment: MIL-E-5400K, Class 2
MTBF > 250,000 h Cooling not required
Circuits: 245 CMOS and PMOS Devices
INPUT/OUTPUT
16-Bit Parallel, Party Line, Buss. 1/0
Serial Channels: 1 Input, 1 Output
4-Bit Address Control; External Clock Input
400 kHz Burst Rate; 1000 kHz Continuous Rate
CENTRAL PROCESSOR
Type: Binary, Parallel, General Purpose, Fractional, Fixed Point, Two's Complement
Instructions: 44; 16-Bit Inst/Data Words
Addressable Hardware Register File: 16 Reg
(8 Accumulators, 4 Index, 4 P-Registers)
Interrupts: 3 Ext. Levels, Direct Execute
Execution Times: Add 2.4μ sec, Divide 30.4μ sec,
Multiply 10.4μ sec.
Double Precision Add 3.6μ sec.
MEMORY
Type: Random Access, Work-Organized, Non-Destructive Readout, Electrically Alterable, Plated Wire Memory
16-Bit Words, 4K DRO/NDRO, Expandable to 65K
Cycle time: 1.6u sec; Access Time; 1.0μ sec.

o <u>Momentum Wheel</u>

The Bendix type 1880026 momentum wheel as used on the ATS/F Program was selected. The reliability of the wheels and their driving electronics is high. The entire unit is hermetically sealed in helium at ½ atmosphere pressure. The physical and electrical characteristics of the wheel are listed below:

Diameter		30.7 cm (12 in)
Height		12.1 cm (4.76 in)
Weight		8.85 kg (19.5 lb)
Momentum	(N-m-s)	11.4 @ r/min
Stall Torque		.015 kg-m (20 in-oz)
Stall Power	(W)	53
Synchronous Speed	(r/min)	1500
Supply Frequency	(Hz)	400

A tachometer is included in the housing and provides a non-symmetrical saw tooth wave from which both rotor speed and direction of rotation can be obtained.

o <u>5 cm Ion Engines</u>

Data on the 5-cm mercury bombardment thruster for use as the ACS thrusters was obtained from Le RC and is included in this section for reference.

Thrust vectoring in the range of $\pm 24^{\circ}$ has been assumed as attainable by the time flight date occurs. Also, projected performance data has been considered as obtainable by then (.5 mlb thrust, 2180 sec Isp and a total power of 50 W).

The characteristics of the ion engine are as follows:

13 cm dia, 30 cm long

Dry weight, 2.1 kg - Propellant 6.2 kg (20 k hrs)

Thermal design: synchronous orbit

Structural design: THORAD qualified

The development goals for the engine are summarized in the following:

Thruster input power, 60 W

Isp, 1830 sec

Overall efficiency, 26.6%

Power/Thrust, 68 w/mkg

Thrust, 0.88 mkg

9000 hours durability design minimum

4.6.4 ACS CONFIGURATION FOR ATS-AMS I

The attitude control subsystem selected for ATS-AMS I is three axis stabilization, using 5 cm vectorable ion thrusters mounted at the tips of the solar array erection masts for roll-yaw axis attitude control and a momentum wheel for pitch axis control. The use of the wheel is recommended because of the cyclic character of the pitch solar disturbance torques in the I configuration (versus the secular pitch disturbance torques in the II & III configurations).

The equipment selections for the attitude control subsystem are determined by the previously stated pointing accuracy requirements, the characteristics of the cyclic and secular distrubance torques and the spacecraft mass properties. These requirements are summarized in Table 4.6.4-1.

Because of the required maneuvering during the spiral-out or orbit raising portion of the mission, the tight pointing accuracy requirements during the synchronous altitude porition of the mission, and the need for sensor backup over a five year life, the selection of attitude control sensors is a particularly difficult problem. A multiple mode approach using different types of sensors was selected and the recommended combinations are tabulated below.

Mode	Spiral Out			Direct Ascent		
	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Initial Acquisition Mode	Sun Sensor Updated Inertial Reference Unit			Earth Sensor	Earth Sensor	Sun Sensor
Spiral Out Mode	Sun Sensor Updated Inertial Reference Unit			——	——	——
Primary Mission Mode	Two-Station Interferometer			Two-Station Interferometer		
Alternate/Backup Mode	Earth Sensor	Earth Sensor	Polaris Sensor	Earth Sensor	Earth Sensor	Polaris Sensor

Table 4.6.4-1. Attitude Control Requirements - Configuration 1

A.) H_O = 2778 km (1500 nmi) - initial altitude for spiral ascent

B.) H_F = 35,786 km (19,323 nmi) - final (synchronous) altitude

Spiral Out Mode, Earth Pointing, Rotating Solar Paddles

	Axis	Pointing Accuracy (deg.)	Disturbance Torque Secular (N-m) x 10^{-5}	Secular (lb-ft) x 10^{-5}	Cyclic (N-m) x 10^{-5}	Cyclic (lb-ft) x 10^{-5}	Moment of Inertia (kg.-m^2) #
A.)	Roll	$\pm .5$	50.0	37.0	8.4	6.2	$18,800 + 410 \sin^2 \theta p$**
	Pitch	$\pm .5$	2.2	1.6	20.0	15.0	950
	Yaw	$\pm .5$	≈ 0	≈ 0	16.0	12.0	$19,200 + 410 \cos^2 \theta p$*
			x10^{-6}	x10^{-6}	x10^{-6}	x10^{-6}	
B.)	Roll*	$\pm .03$	5.2	3.8	27	20	
	Pitch*	$\pm .03$	.2	.2	200	150	
	Yaw*	$\pm .03$	≈ 0	≈ 0	27	20	

* $\pm .1^o$ when in backup mode

\# Slug ft^2 = 1.35 kg. m^2

** θp is the solar paddle cycle referred to a zero condition when the array flats are parallel to the spacecraft roll (thrust) axis

The equipment selection for ATS-AMS I is the same as for II and III except for the following differences and additions.

1. An inertial reference unit is needed instead of a rate gyro package, for attitude determination during the spiral-out mode
2. A momentum wheel is required for pitch control.
3. A Magnetic Control Assembly is also used at low altitudes as an alternate means of surmounting the larger magnetic and gravity gradient disturbance torques.

Using the sensor equipments described in the previous sections, the initial acquisition pointing error will be less than 0.1°, the spiral out mode pointing error will be less than 0.5°, the primary mission mode accuracy will be less than 0.03° and the alternate mission mode accuracy will be less than $.1^{\circ}$.

The momentum storage capabilities of the flywheel was selected to provide in excess of an integrated half cycle of cyclic disturbance. A block diagram of the attitude control subsystem is given by Figure 4.6.4-1.

As indicated by the block diagram, the roll and yaw loops are stabilized by 5 cm mercury bombardment ion engines mounted at the tips of the solar panels and the pitch loop is stabilized by a momentum wheel. The momentum wheel is periodically unloaded by the auxiliary hydrazine system which also serves as a backup for all three axes of attitude control actuation, North-South station keeping, and East-West station keeping.

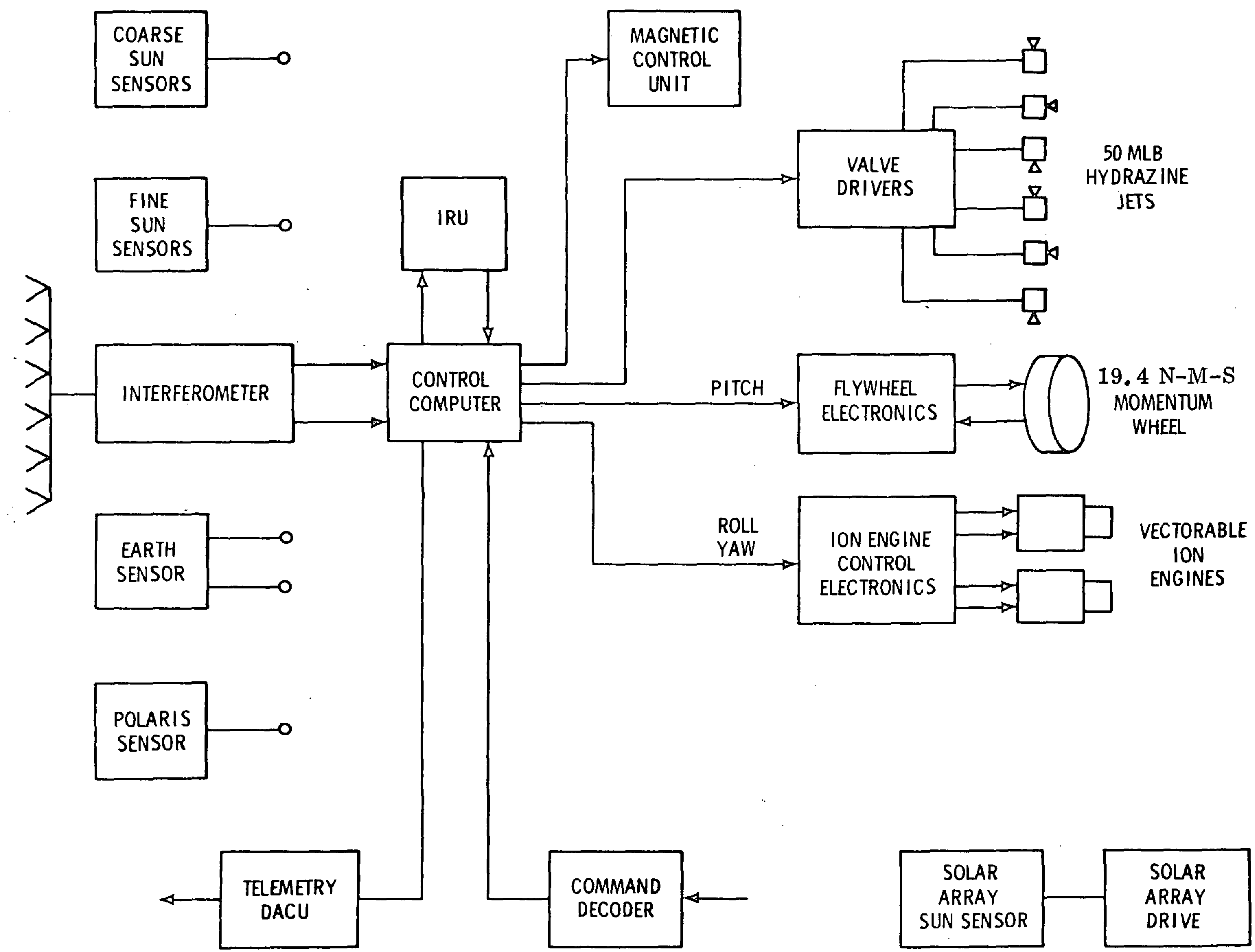


Figure 4.6.4-1. ACS Configuration, ATS-AMS I

A description of the additional equipment follows:

Inertial Reference Unit

A lightweight, strap down IRU being developed by Autonetics Division of North America Rockwell was selected. The proposed IRU will employ a pair of tiny electrostatically suspended, two degree-of-freedom gyros which have wide angle pickoffs with a 4-axis precision electrostatic levitation servo, acceleration outputs also are available. The two gyros effectively would serve as multisensors, each performing functions of both a gyro and an accelerometer.

The gyros use the unusual concepts of a mass unbalance modulation pickoff and a rotor pumping scheme in which levitation servos keep the ball running at constant speed. The small size of the gyros - 3.3 cm in diameter by 3.3 cm long - is attributable to the fact that they have only a single moving part and very few parts in total. Weight of the two-axis, free rotor gyro is 0.18 kg (0.4 lb.). The rotor itself is 1 cm in diameter and weighs 1 gram. The gyro has an uncompensated random drift rate of 0.01 degrees/hour and a steady state maximum acceleration capability of 30 g.

The full system would consist of the two inertial multisensors, an electronics section, computer power supply, and battery, packaged in a central support frame attachable to the spacecraft.

Initially, the attitude of each rotor is found by open-loop gyro-compassing. Attitude is computed 128 times per second. Direction cosines are generated for body-to-inertial and body-to-navigation frame transformations which are used for resolving rotor attitude into vehicle attitude and acceleration signals into a locally level set.

Whenever the sinusoidal mass unbalance signal passes through zero amplitude, a zero-crossing detector generates a pulse from which a high-speed digital clock can measure rotor period and time between zero crossing on successive axes. Angular readout is found from phase angles derived from zero crossing times.

The computer accepts spin axes phase angle information and acceleration from the inertial unit and computes velocity, position and attitude. These data are outputted in digital form.

Momentum Wheel

The Bendix type 1823408 momentum wheel as used in the OGO Program was selected. The wheel will be operated at a nominal speed of 1500 r/min (at a control frequency of 800 Hz) producing 19.4 N-m-s (14.4 ft-lb-s) of angular momentum.

OGO wheels of this type have been operated continuously for over 8 years by TRW on earth and for over 6 years in space. The reliability of the wheels and their driving electronics is high. The entire unit is hermetically sealed in helium at $\frac{1}{2}$ atmospheric pressure.

The physical and electrical characteristics of the wheel are listed below:

Diameter	(cm)	34.7 cm
Height	(cm)	11.8 cm
Weight	kg (lbs)	8.2 (18)
Momentum	N-m-s (lb-ft-s)	19.4 (14.4)
Stall torque	N-m (lb-ft)	.0189 (.014)
Stall Power	W	53
Running Power	W	10
Saturated Speed	r/min	2500
Supply frequency	Hz	800

A tachometer is included in the housing and provides a non-symmetrical saw tooth wave from which both rotor speed and direction of rotation can be obtained.

An associated amplifier converts the DC error signal to 800 Hz AC at 26 V for use as the control and reference phase voltages in the fly wheel motor.

Magnetic Control Assembly

The Ithaco unit designed for Nimbus D was selected.

The assembly consists of 3 permanent magnets mounted orthogonally whose magnetic state can be altered by external command. The circuits for controlling the magnets, the sensors for measuring the dipole moment of each magnet and a magnetometer for Earth field sensing are all included in the housing.

To fully magnetize a 10,000 pole cm (10 amp-meter^2) magnet requires only 1 W-s of energy. Once magnetized, no additional power is required until the magnetic state is to be changed.

Each magnet can be magnetized to any desired state and will remain there indefinitely. The state of each magnet is sensed and telemetered to the ground. The maximum field strength is less than 100 mG at a distance of 56 cm. The unit weighs approximately 6 lbs (2.7 kg) and uses approximately 1 W.

This unit will not compensate for the large magnetic moment associated with the 30 cm engines. The vector direction of the ion engine magnetic moment is colinear with the thrust force but can be set for either direction (either in the same sense as the thrust force or opposite). Thus, for an added number of engines, a magnetic moment of approximately 35,000 pole-cm (35 amp-m^2) will have to be compensated.

An additional single Ithaco magnet unit of the propermaximum magnetic moment capacity will be required. This unit will weigh approximately .91 kg (2 lbs.) and will be 7.62 cm (3 in) in diameter and 30.48 cm (12 in) in length. The magnetic control assembly magnetizing circuitry can be utilized to accomplish the necessary alterations in magnetic state of the magnet.

A summary Table 4.6.4-2 of the equipment required for configuration I, including weight and power, follows:

Table 4.6.4-2. ACS Components for ATS-AMS I

EQUIPMENT	NO's OF UNITS	WEIGHT kg.	(lb)	POWER W
Coarse Sun Sensor	6	.226	(.5) each	———
Fine Sun Sensor	2	.338	(.75) each	1.0 each
Sun Sensor Electronics	1	1.000	(2.2)	2.0
Inertial Reference Unit	1	1.355	(3.0)	10
Interferometer	1	6.680	(15.1)	16
Earth Sensor Head	2	1.760	(3.9) each	15
Earth Sensor Electronics	1	2.710	(6.0)	
Polaris Sensor	1	4.880	(10.8)	15
Hydrazine Subsystem	1	25.300	(56)	———
Jet Valve Drives	1	2.620	(5.8)	10
5 cm ion Engines (incl. fuel tanks & fuel for 5 years)	4	8.300	(18.4) each	60 each
Solar Array Drive & Slip Rings	1	12.700	(28)	15
Control Computer	1	1.355	(3.0)	10
Momentum Wheel	1	8.150	(18)	10
Wheel Electronics	1	1.355	(3.0)	5
Magnetic Control Unit	1	2.710	(6)	1
Hydrazine Propellant (backup for 1 year)	1	25.5	(56)	

TOTAL WEIGHT: 135 kg (298 lbs.)

4.7 ORBIT CONTROL SUBSYSTEM

4.7.1 SUBSYSTEM OPERATION AND REQUIREMENTS

The orbit control subsystem capabilities will include station acquisition and station keeping for the ATS-AMS I, II, and III configurations and will include the ability to generate sufficient total impulse to raise the orbit from approximately 1500 N. M. altitude to 19,323 N. M. altitude and change the plane of the orbit inclination 28.5^{o} for the ATS-AMS I configuration.

The orbit will be controlled both East/West and North/South to within an equivalent earth longitude and latitude tolerance of $\pm$ .2^{o}. The spacecraft will have a minimum V of 30.48m/sec (100 ft/sec) for re-positioning, The axis of thrust will be maintained through the spacecraft center of mass within acceptable limits.

4.7.2 ORBIT CONTROL CONFIGURATION FOR ATS-AMS III and II

The orbit control subsystem equipment consists of a 30 cm mercury bombardment ion engine and the backup hydrazine reaction control subsystem. Both the ion engine and hydrazine thrusters are started and stopped by ground command. The estimated weight and power requirements of the 30 cm ion engine for the ATS-AMS II and III configurations are listed in Table 4.7.2-1. The weight requirements of the backup hydrazine reaction control system were previously listed in Section 4.6.

4.7.3 COMPONENT DESCRIPTION

4.7.3.1 Ion Engine

For the North/South stationing, a 30 cm ion engine will be mounted on the spacecraft $+Y_I$ Face (bottom end). The nominal thrust vector is parallel to the Y_I axis to produce motion in the $-Y_I$ direction.

Table 4.7.2-1. Weight and Power Summary for ATS-AMS II and III

	Weight kg (lb.)	Power Max.	Standby
30 cm ion engines (1)	11.3 (25.0)	2000	160 W
Propellant (Isp > 2000) (for 5 years)	13.4 (29.5)	---	---
Tankage and plumbing	9.1 (20.0).	---	---
Totals	34 (74.5 lbs.)	2000 W	160 W

4.7.3.2 Hydrazine Reaction Control Subsystem

The primary use of the hydrazine reaction control subsystem will be to provide for East/West station keeping. In addition it can be used as a backup for unloading the momentum wheels, attitude control, and North/South station keeping.

Hydrazine (N_2H_4) will be utilized as the propellant. Tankage and plumbing is such that a backup dual redundancy for both directions of stationkeeping can be achieved. Orientation of the thrusters is given by Figure 4.6.2-1.

4.7.4 ORBIT CONTROL CONFIGURATION FOR ATS-AMS I

The equipment selection for the ATS-AMS I is approximately the same as for the ATS-AMS II and III. The only differences between the two configurations are:

a) Propellent and power requirements

b) Ion engine gimbal control orientation and usage

c) Hydrazine reaction control subsystem usage

The estimated weight and power requirements of the 30 cm ion engine cluster for the ATS-AMS I configuration are listed in Table 4.7.4-1

The weight requirements of the hydrazine reaction control system were previously listed in Table 4.6.4-2.

Table 4.7.4-1 Orbit Control Weight and Power Summary for ATS-AMS I.

	Weight kg (lbs)	Nom.	Power Max.	Standby
30 cm ion engines (3)	34 (75)	1500 W	6000 W	160 W
Propellant (Isp 2000)	159 (349)	---	---	---
Tankage and plumbing	24 (52)	---	---	---
Gimbal and drive motors	11 (24)	---	40 W	---
Totals	228 (500)	1500 W	6000 W	160 W

4.7.4.1 Ion Engine Gimbal Control

For the spiral out ascent mode, a cluster of three 30 cm Hg bombardment ion engines will be grouped on the spacecraft -X face (aft end). The nominal thrust vectors are parallel to the X axis to produce motion in the +X direction.

A two-axis gimbal system will be required permitting $\pm 15^{\circ}$ of rotation about axes parallel to the pitch (y) and yaw (z) axes. The motor driven gimbals will operate at a relatively slow rate (approximately 10 rpm) throughout their regular travel range slewing at approximately 15°/minute.

4.7.4.2 Hydrazine Reaction Control Subsystem

The primary use of the hydrazine reaction control subsystem will be to unload the pitch momentum wheel when it is saturated. In addition, it can be used as backup attitude control, North South station keeping, and East-West station keeping.

Hydrazine (N_2H_4) will be utilized as the propellant.

4.8 THERMAL CONTROL SUBSYSTEM

Spacecraft thermal control design is dictated primarily by the electronic and optical components' functional reliability for a five year mission life. Due to power fluctuation and changing orbital environmental conditions, the spacecraft heat rejection requirements may vary from 300 to 5000 watts. The ATS-AMS thermal design provides the flexibility to accommodate the variations in heat rejection rates while maintaining the temperatures within required limits. Variable heat rejection in a narrow radiator temperature band is achieved by louver assemblies which vary the effective emittance of the spacecraft radiator surfaces. A typical louver assembly is capable of an effective emittance variation from 0.14 to 0.7. Therefore, a 100% louvered radiator generates a heat control ratio of 5. Lower turndown ratios are achieved by proportioning both louvered radiators and passive radiators.

The designs of ATS-AMS III and II are similar because they are all direct ascent, solar oriented spacecraft. Table 4.8-1 gives the weight details.

4.8.1 ATS AMS III

Preliminary analysis of the ATS AMS III A sun oriented spacecraft has led to a design concept which maintains individual components within their allowable temperature range with a minimum expenditure of cost and weight. The spacecraft body proper contains the batteries, TT&C, attitude control, low voltage power conditioning, experiment packages and antenna drive. With the exception of the antenna drive this equipment is mounted directly to the three sides of the spacecraft which do not receive direct sunlight (Figure 4.8.1-1). The temperature of these three surfaces is maintained between 273°K and 308°K (32-95°F) with the aid of thermal control louvers.

The subsolar side of the spacecraft is a mounting platform for the high voltage solar array experiment. The sunfacing side contains

Table 4.8-1. Thermal Control Weight Estimates

Item	Quantity	Total Weight Kilograms	
ATS-AMS III A			
Thermal Blankets			
1	34.03 sq. ft.	1.54	(3.40)
2	63.20 sq. ft.	8.17	(18.01)
Heat Pipes	116.00 ft	7.89	(17.40)
Louvers	54.00 sq. ft.	24.49	(54.00)
OSR's	27.22 sq. ft	1.23	(2.72)
Paint			
Exterior	40.00 sq. ft.	1.09	(2.40)
Interior	204.17 sq. ft.	5.56	(12.25)
Total		49.98	(110.18)
ATS-AMS II			
Thermal Blankets			
1	115.20 sq. ft	5.23	(11.52)
2	8.34 sq. ft	1.07	(2.37)
Heat Pipes	107.00 ft.	7.28	(16.05)
Louvers	13.72 sq. ft	6.22	(13.72)
Paint			
Exterior	75.84 sq. ft	1.61	(4.56)
Interior	227.50 sq. ft	4.85	(13.65)
Total		28.06	(61.87)
ATS-AMS I			
Thermal Blankets			
1	35.00 sq. ft	1.59	(3.50)
2	19.44 sq. ft	4.78	(10.53)
Heat Pipes	163.92 ft.	11.15	(24.59)
Louvers	17.855sq. ft	8.10	(17.86)
OSR's	13.67 sq. ft	0.62	(1.37)
Paint			
Exterior	1150 sq. ft	31.30	(69.00)
Interior	160 sq. ft	4.35	(9.60)
Total		61.89	(136.45)

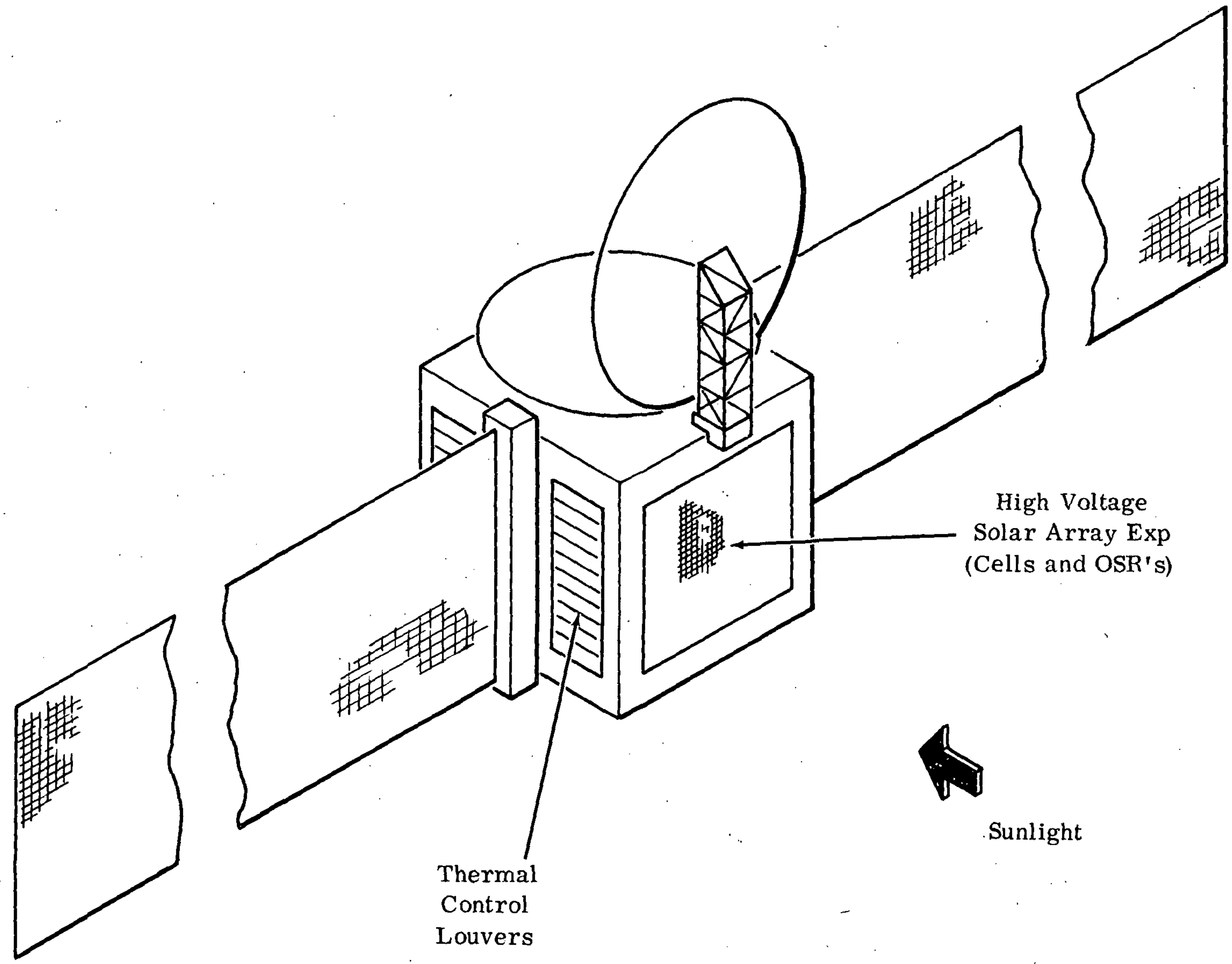


Figure 4.8.1-1. ATS-AMS III Thermal Configuration

solar cells and optical solar reflectors. The backside which sees the inside of the spacecraft serves as a mounting platform for low power components.

The north side of the spacecraft serves as the mount for the rotating feed and antenna assembly. Multilayer insulation is used to prevent heat leaks from the spacecraft in this area.

The south side of the spacecraft is used to mount the 30 cm ion engines and the solar array shunts. The shunt power transistor junctions are allowed to range between $233^{o}K$ and $413^{o}K$. Internal insulation prevents the spacecraft interior from reacting to this temperature excursion. The ion engines are high temperature heat dissipators which radiate to the shunts' mounting area. However, the engine operation does not interfere with the shunts because shunts are off when the ion engines fire.

The high power transponders and the high voltage power supply are mounted on the antenna platform. The power amplifier TWT's dissipate 2000 watts of heat from radiator/collectors which are integral with the TWT design. These radiators attain temperatures close to the TWT body when operating. Ceramic insulation is used to thermally isolate the hot TWT from the platform (Figure 4.8.1-2). The transponders and converter require $273^{o}K$ to $308^{o}K$ temperatures control for a heat dissipation range of from 1400 W to 1750 W. This is a relatively steady heat load which can be controlled by the use of passive radiators provided that the individual transponder and converter packages can share their heat dissipations. Heat sharing is made possible by using heat pipes in the platform. The detail thermal design of the platform is a major problem which requires extensive thermal anslysis. A major concern is the low temperature problem which occurs when the high power transponders are turned off. The radiator might require variable conductance heat pipes which will conserve heat at low temperatures, so that the transponders and converters do not cool below $273^{o}K$.

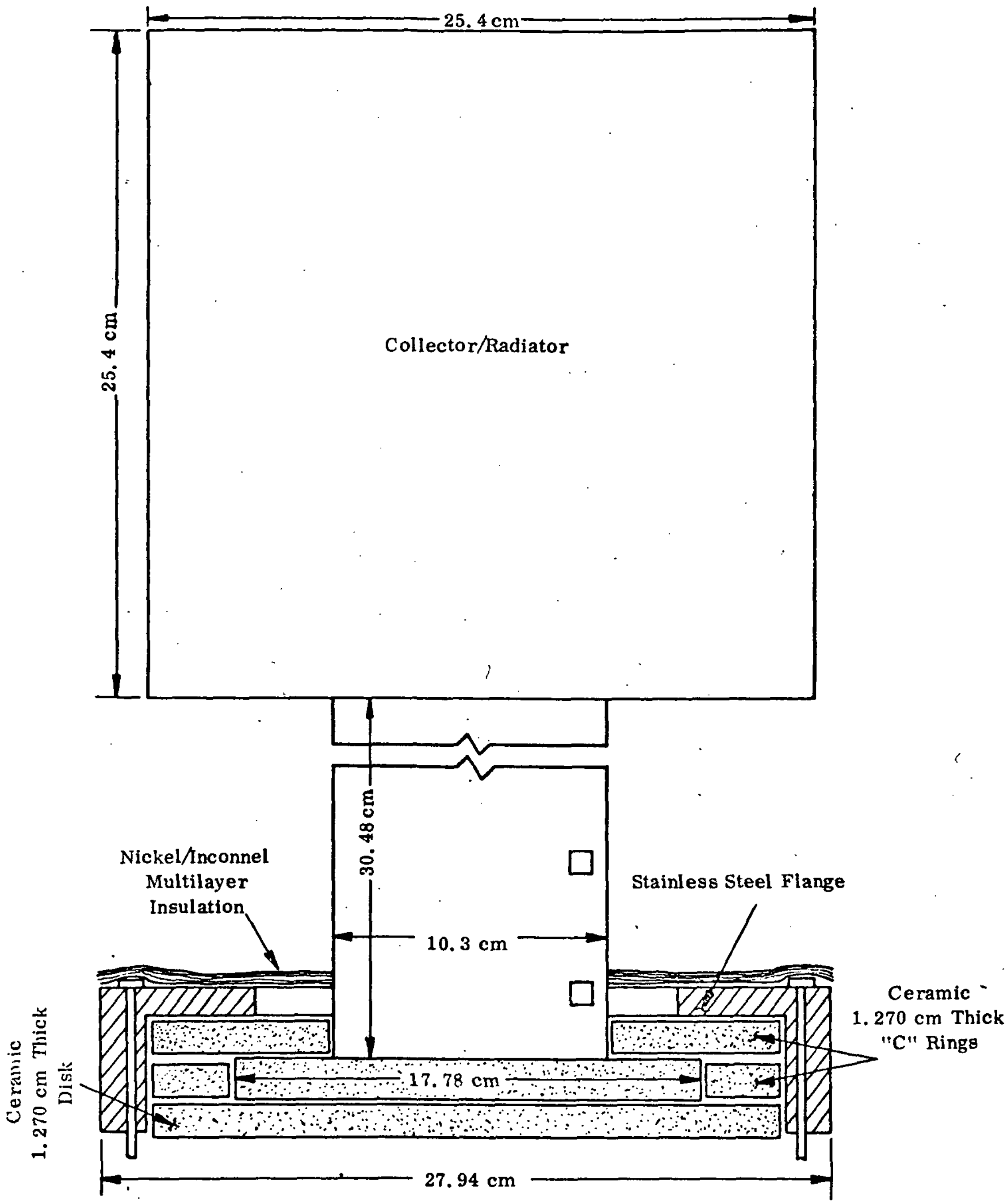


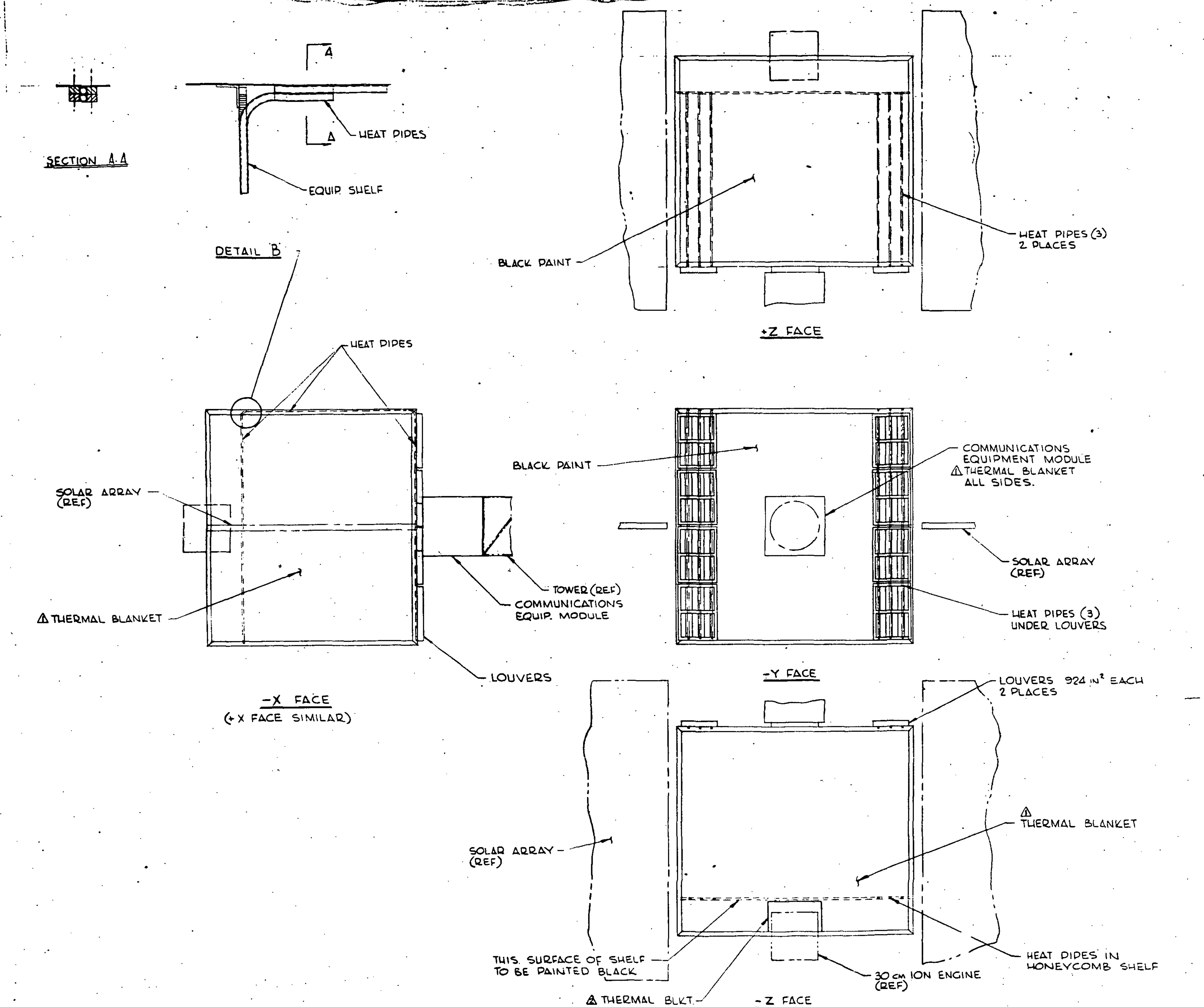
Figure 4.8.1-2. Power Amplifier Tube Insulation Support Design

4.8.2 ATS AMS II

The ATS AMS II has a low temperature radiator on one side only. The north side of the spacecraft has a 1.0m^2 (10.76 ft^2) louvered area. The side facing away from the sun has 2.0 m^2 (21.52 ft^2) of passive radiator area. The two radiators and an internal equipment shelf are conductively coupled by "U" shaped heat pipes (Figure 4.8.2-1). Heat pipes are required in this version because the internal shelf must be used to mount equipment having significant heat dissipations. In the larger AMS III there is available area for mounting direct to the space viewing sides.

4.8.3 ATS AMS I

The preliminary thermal design ATS AMS I earth oriented spacecraft uses a louvered 4.5 m^2 (48.4 ft^2) low temperature radiator. The two sides of the spacecraft where solar arrays are mounted will be louvered to yield 2.25m^2 (24.2 ft^2) each. The side facing away from the earth will have a 2.25 m^2 (24.2 ft^2) passive radiator. The louvered radiator sides are conductively coupled with heat pipes to achieve the required load distribution to maintain 278^{o}K-308^{o}K radiator temperature. The passive radiator surface may require heaters to maintain the equipment temperature above 5°C during quiescent mode. The heat pipes are of 1.27cm (0.5 in) outside diameter grooved/ammonia configuration similar to those used on ATS-F & G spacecraft and have a spacing of 8.9 cm 3 1/2 in) based on Fairchild in-house evaluations. The remaining three sides are superinsulated with 30 layers of 1/8 mil perforated double aluminized mylar separated by 15 denier nylon netting with random perforations not exceeding 5% area. The effective emittance of this insulation is better than 0.01. Figure 4.8.2-2 illustrates this configuration.



△1 THERMAL BLANKET –
OUTSIDE COVER - 1 SHT. 5 MIL KAPTON
FILLER - 30 SHTS. .125 MIL MYLAR, ALUMINIZED BOTH SIDES
SEPARATORS - 31 SHTS. NYLON NETTING
INSIDE COVER - 1 SHT. 5 MIL MYLAR, ALUMINIZED ONE SIDE

△2 THERMAL BLANKET -
OUTSIDE COVER - 10 SHTS. .5 MIL NICKEL FOIL
SEPARATORS - 9 SHTS. INCONEL MESH
FILLER - 20 SHTS. .5 MIL CRINKLED KAPTON, ALUMINIZED ONE SIDE
20 SHTS. .125 MIL MYLAR, ALUMINIZED BOTH SIDES
SEPARATORS - 21 SHTS. NYLON NETTING
INSIDE COVER - 1 SHT. 5 MIL MYLAR, ALUMINIZED ONE SIDE

Figure 4.8.2-1. Thermal Control Sun Oriented S/C

FAIRCHILD HILLER
THERMAL CONTROL
SUN ORIENTED S/C
[redacted]
CODE IDENT NO 86360
SHEET 1 OF 1

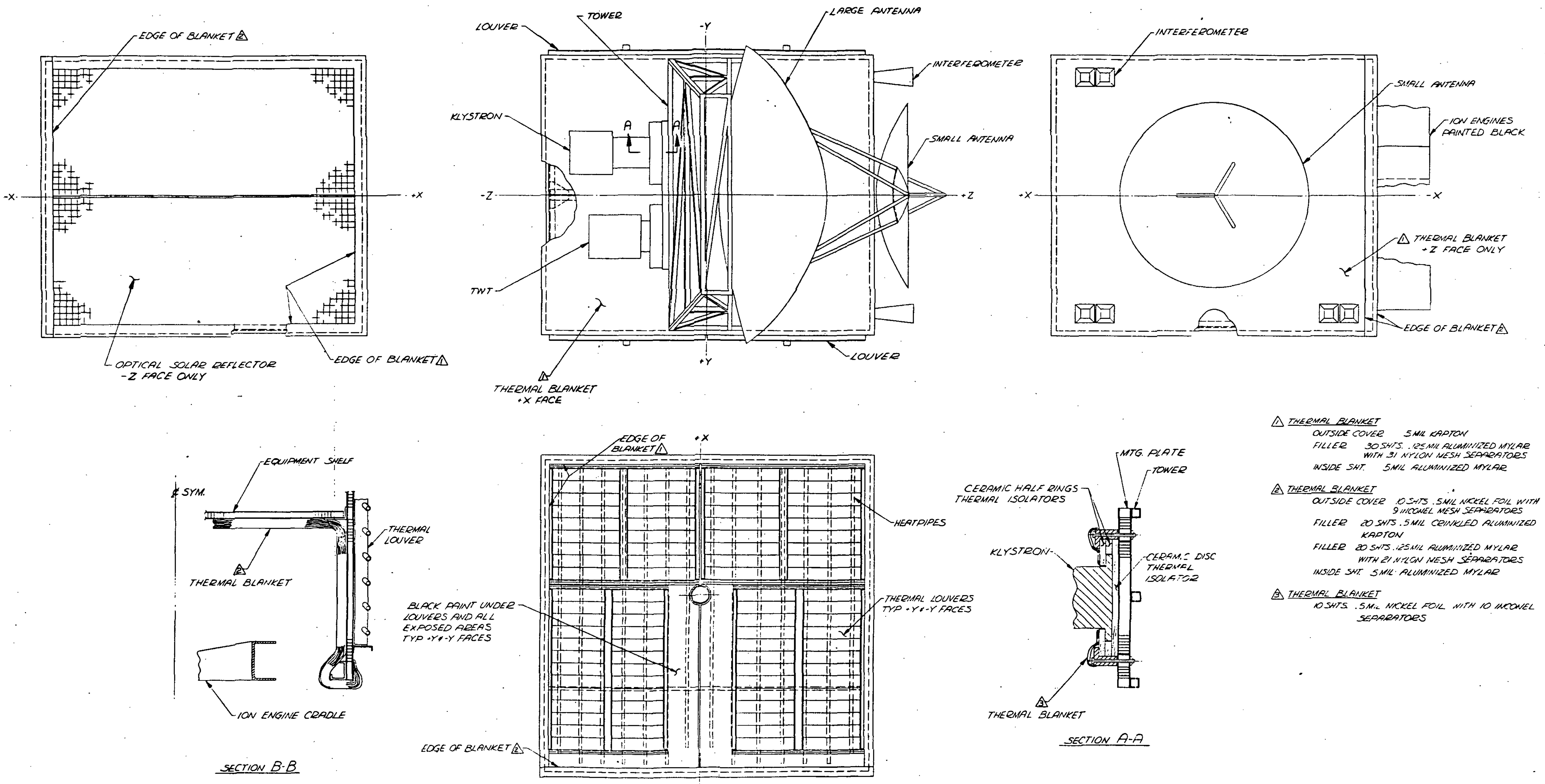


Figure 4.8.2-2. Thermal Control System
Delta Launched Earth
Oriented Spacecraft

4.9 TELEMETRY AND COMMAND SUBSYSTEM

The Telemetry and Command (T&C) subsystem provides the means for controlling the spacecraft from the ground and for reporting to the ground the status of key performance parameters. A block diagram of the T&C subsystem is shown in Figure 4.9-1 and major T&C message characteristics are summarized in Table 4.9-1.

4.9.1 COMMAND SUBSYSTEM

The command signal is transmitted to the satellite at one of two assigned frequencies at two independent antenna assemblies and command receivers for space diversity reception.

The VHF antennas are capable of transmitting and receiving circularly polarized waves and provide -3 dB gain over a $\pm$ 17° conical region and - 10 dB over 90% of the 4 π Steradian sphere surrounding the spacecraft.

Upon command verification by transmission of the received command to the ground, an execute tone sent to the spacecraft causes the command to be executed. A 128 Hz sine wave clock signal is extracted by the Spacecraft Command Decoder/Distribution unit (SCDD) for control tuning within the command subsystem. The spacecraft recognizes four different command types: Data word, spacecraft discrete, paddle #1 discrete and paddle #2 discrete. The received signals are down-converted in solid state receivers by a crystal controlled oscillator to 10.2 MHz I.F., filtered and amplified with AGC providing demodulated command signals at reasonably constant audio output levels to the SCDD units. The two redundant SCDD's are digital processors which decode the command frames and distribute them to the desired system. One of the two redundant Paddle Command Decoder Distribution Units is energized at a time to decode the FDM discrete commands from the SCDD.

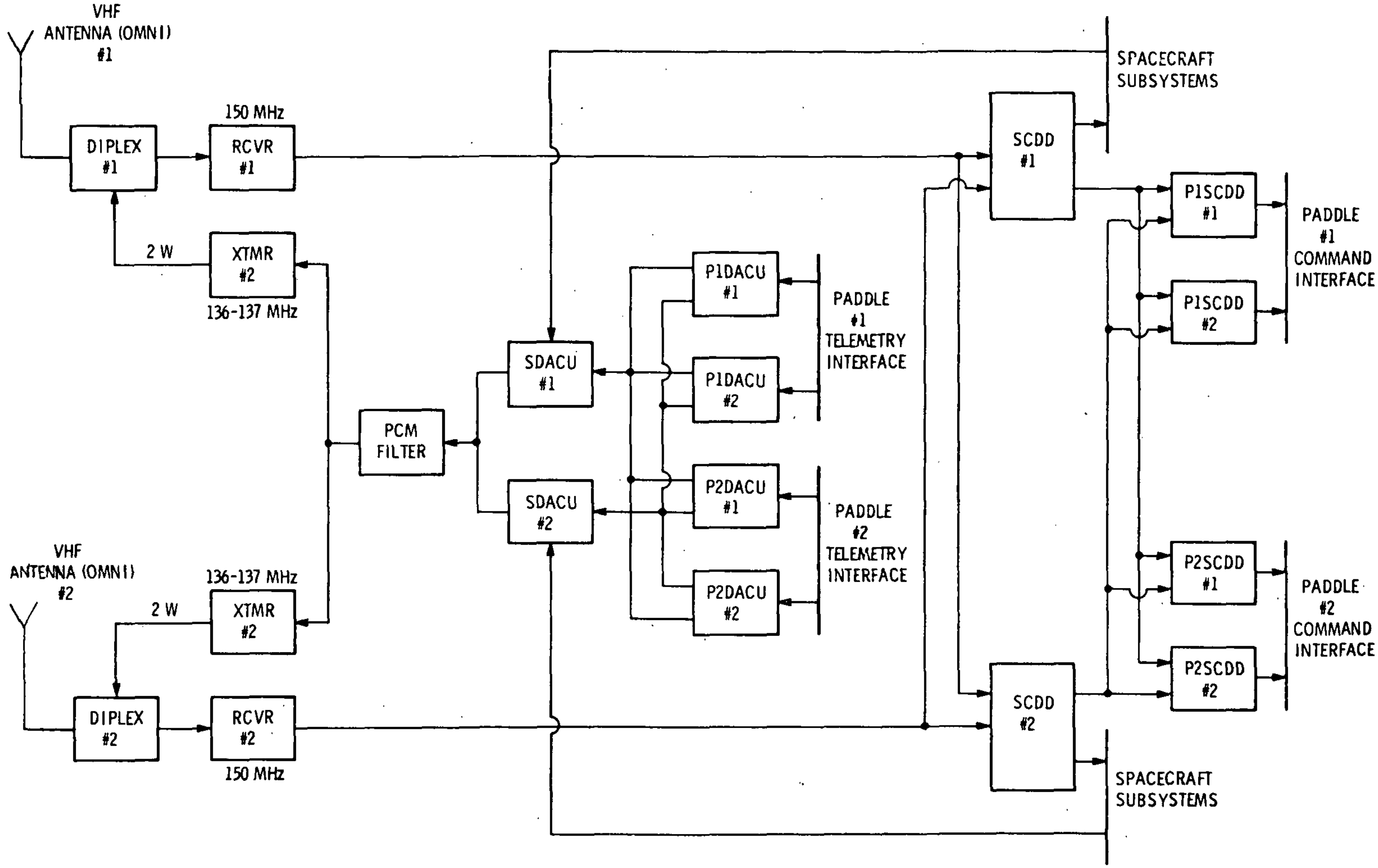


Figure 4.9-1. T&C Subsystem Block Diagram

Table 4.9-1. T&C Message Characteristics

	Command	Telemetry
Word length	64 bits	9 bits
Bit rate	128 bps	400 bps
Spacecraft	512 discrete commands	180 analog words
	70 (9 bit) data words (Attitude control-2, antennas/feeds-10, power-6, experiments/spares-52)	70 digital words
		8-10 calibration words
Paddle # 1	512 discrete commands	70 analog words
		10 digital words
		3 calibration words
Paddle # 2	512 discrete commands	70 analog words
		10 digital words
		3 calibration words
Command Verification	Through telemetry link	16 telemetry words
Command execution	Execute tone	—
Minor frame length		330 words
Major frame length		660 words

4.9.2 TELEMETRY SUBSYSTEM

The telemetry subsystem is in three parts: The Spacecraft Data Acquisition unit (SDACU) and the DACU for paddle #1 and the DACU for paddle #2. Telemetry messages are in fixed format with all data reporting in digital form with analog information translated into digital form by a 9-bit analog-to-digital converter in each DACU. The Spacecraft Data Acquisition and Control Unit (SDACU) 1) monitors points within the spacecraft body and performs the necessary multiplexing, A/D conversion and formatting to make up words in telemetry format and 2) sequences, synchronizes and controls the Paddle Data Acquisition and Control Units (PDACU) in their generation of paddle telemetry words. This digital information is transferred serially along with spacecraft words through the SDACU for filtering and for phase modulating a VHF carrier in two redundant transmitters each individually commanded on-and-off by ground command. Each VHF transmitter is solid state with a 2.0 watt power output. The final output of each transmitter will go through a low-pass filter to suppress spurious output signals. From the telemetry link analysis adequate margin exists for coherent detection over both the -3dB gain ± 17° cone and the -10 dB gain spherical coverage.

4.9.3 T & C PHYSICAL AND POWER CHARACTERISTICS

Table 4.9.3-1 lists for all subassemblies of the T & C subsystem the weight and required power of each unit with estimate shown for both spiral and direct ascent.

Table 4.9.3-1. T&C Subsystem Equipment List

Item	ATS - AMS III and II		ATS - AMS I	
	Weight KG (lb)	Power (W)	Weight KG (lb)	Power (W)
ANTENNAS (2)	0.6	0	0.6	0
DIPLEXER (2)	0.9	0	0.9	0
TRANSMITTER (2)	0.4	5.3	0.4	5.3
LOW PASS FILTER	0.1	-	0.1	-
SPACECRAFT DACU (2)	15.4	17	8.2	7.4
PADDLE 1 DACU (2)	-	-	4.5	6.1
PADDLE 2 DACU (2)	-	-	4.5	6.1
RECEIVER (2)	1.2	1.6	1.2	1.6
SPACECRAFT CDD (2)	16.0	17	11.0	13.5
PADDLE 1 CDD (2)	-	-	3.7	2.1
PADDLE 2 CDD (2)	-	-	3.7	2.1
TOTAL	34.6 (76.34)	40.9 W	38.8 (85.5 lb)	44.2 W

4.10 STRUCTURES

The ATS-AMS III basic structure is very similar to the ATS-F spacecraft. The equipment module and launch vehicle interface structure was scaled from the ATS-F counterparts. Conventional structure is used.

The equipment/antenna sandwich platform is supported on a cylinder which interfaces with the equipment module with two large diameter bearings. This allows freedom of rotation and adequate load paths for supporting the platform. The truss supported antenna reflector and the feed system are track-mounted on the rotating platform. Positive launch tie-downs for these components are provided. Deployment serves to locate the feed at the antenna focal point and to balance the spacecraft. Additionally, the honeycomb sandwich-type, rigid reflector is deployed by a hinge system.

The solar arrays are of the flexible, roll-up type and are hoisted and pretensioned from a graphite fiber reinforced plastic, or similar highly efficient material, mast deployed from the side of the spacecraft. The pretensioning is provided to give the system adequate stiffness and therefore a frequency greater than the control system output frequency. This "stiffening" effect may be observed in a violin string, where the higher the tension, the higher the pitch (frequency).