Texstar: The All-Texas Educational Satellite System

In Response to RFP #
ASE274L-S90

Submitted to:

Dr. Wallace Fowler
Department of Aerospace Engineering
and Engineering Mechanics
The University of Texas at Austin
Austin, Texas

by:

The Longhorn Satellite Company
Austin, Texas

May 7, 1990

Abstract

This report documents the research and design which has been completed by the Longhorn Satellite Company (LSC) in response to RFP # ASE274L-S90. LSC has designed Texstar, an educational satellite communications system which will be considered as a means of equalizing the distribution of educational resources throughout the state of Texas. Texstar will be capable of broadcasting live lectures and documentaries in addition to transmitting data from a centralized receiving-transmitting station. Included in the design of Texstar is the system and subsystem design for the satellite and the design of the ground stations. The launch vehicle used will be the Texas-built Conestoga 421-48. The Texstar system incorporates three small satellites in slightly inclined geosynchronous orbits. Due to the configuration and spacing of these satellites, the system will be accessed as if it were one large, geostationary satellite. Texstar has been shown to be a viable option to the educational crisis in the state of Texas.

Table of Contents

Abstract....................i
Table of Contents....................ii
List of Figures....................iv
List of Tables....................v
1. Introduction....................1
  1.1 Purpose....................1
  1.2 Background....................1
  1.3 Project Scope....................2
  1.4 Report Organization....................2
2. System Design....................3
  2.1 Design Criteria....................3
  2.2 Design Procedure....................4
  2.3 Design Overview....................4
    2.3.1 Launch Vehicles....................4
      2.3.1.1 Conestoga 421-48B....................5
      2.3.1.2 Scout II....................6
      2.3.1.3 Launch Sites....................7
    2.3.2 Orbit Determination....................7
      2.3.2.1 Geosynchronous Orbit....................7
      2.3.2.2 Low-Earth Orbits....................9
    2.3.3 Preliminary Satellite Configuration....................11
      2.3.3.1 Mass Approximation....................11
      2.3.3.2 Drum-Stabilization....................12
      2.3.3.3 Body-Stabilization....................13
    2.3.4 Ground Stations....................13
      2.3.4.1 Receiving Stations....................14
      2.3.4.1 Receiving-Transmitting Stations....................15
3. System Design....................16
  3.1 Candidate Systems....................16
  3.2 Final Orbit Design....................16
  3.3 Satellite Design....................17
    3.3.1 Power and Solar Array Sizing....................17
    3.3.2 Structural Configuration....................18
    3.3.3 Thermal Control....................19
    3.3.4 Attitude Control....................22
    3.3.5 Propulsion....................24
    3.3.6 Communications....................24
  3.4 Ground Stations and Satellite Tracking....................26
  3.5 Launch Vehicle and Launch Site....................26
  3.6 Expandability Options....................27
4.0 Conclusions....................30
5.0 References....................31
Figures....................32
Tables....................53

Appendix A: TK! Solver Models 59
Appendix B: Sizing of Solar Arrays 66
Appendix C: Sizing of the Radiator 70
Appendix D: Calculations for Solar Array Temperature 72

List of Figures

Figure 1: Conestoga 421-48 Configuration 33
Figure 2: Conestoga Launch Sequence 34
Figure 3: Launch Capabilities of Conestoga and Scout II 35
Figure 4: Conestoga Shroud 36
Figure 5: Conestoga Launch Sites 37
Figure 6: GEO Satellite Ground Track 38
Figure 7: Number of Satellites for Complete Coverage vs. Circular Orbits at Different Altitudes 39
Figure 8: Drum-Satellite Mass vs. Number of Transponders 40
Figure 9: Body-Satellite Mass vs. Number of Transponders 41
Figure 10: Example of Spin-Stabilized Satellite 42
Figure 11: Example of Body-Stabilized Satellite 43
Figure 12: Schematic of Simple Power Subsystem 44
Figure 13: Schematic of Power Shunt Dissipative Regulation 45
Figure 14: Single View Stowed Configuration 46
Figure 15: Elements of Thermal Control System 47
Figure 16: Schematic of Attitude Control System 48
Figure 17: Schematic of Hydrazine Thruster 49
Figure 18: Transponder Bandwidth 50
Figure 19: Schematic of Communication System 51
Figure 20: Schematic of Expansion System 52

List of Tables

Table 1: Preliminary Candidate Designs 54
Table 2: Sample Power Distribution 55
Table 3: Preliminary Mass Budget 56
Table 4: Spacecraft Temperature Operating Limits 57
Table 5: Attitude Control Weights 58

1. Introduction

This document presents the findings of the Longhorn Satellite Company (LSC) in response to the RFP # ASE274L-S90 [1] made by the University of Texas Department of Aerospace Engineering and Engineering Mechanics (UT). Responding to RFP # ASE274L-S90 has required the research and development of an educational satellite system which will be used to help equalize the quality of education across the state of Texas.

1.1 Purpose

The educational satellite system, Texstar, has been designed by LSC in an effort to equalize the quality of education across the state of Texas. By broadcasting live lectures in addition to pre-recorded documentaries, Texstar will present subject material previously unavailable to remotely located and economically disadvantaged school districts. In addition, Texstar will available to local and state governments for data transmission. The system has been designed to incorporate several secondary goals, such as vehicle tracking for Department of Public Safety (DPS) and Emergency Medical System (EMS), interactive conferencing, and data services for business.

1.2 Background

In an effort to unite the educational resources throughout the state of Texas, the 1989 Texas Senate passed the Senate Conference Resolution No. 23. This resolution directs the Automated Information and Telecommunications Council to study the feasibility of a state-sponsored educational satellite project [2]. In response to this request and RFP # ASE274L-S90, LSC has designed Texstar.

1.3 Project Scope

LSC has designed Texstar, a satellite communication system which will provide continuous and uninterrupted television transmission to the entire state of Texas. This report presents the final design of the Texstar system, which includes both the satellite system and the ground communications system. It also presents the orbit design and the proposed Texas-built launch vehicle to be used. Additionally, a model of the system and subsystems has been constructed to demonstrate the design concept. Finally, a poster has been prepared to further illustrate the final design.

1.4 Report Organization

Section 2 of this report discusses the design criteria, design procedure, and the initial findings of LSC in the areas of orbit determination, available launch vehicles, appropriate satellite configurations, and ground stations. Each candidate system that has been considered is also presented in Section 2. The final system design, which includes the specifics of the subsystems, is presented in detail and justified in Section 3. Section 4 concludes the findings of LSC.

2. System Design

The engineering of a satellite communications system includes the determination of the orbit and launch vehicle, the design of the orbiting satellites, and the design of the ground stations. This section presents the design criteria for the project and the design methodology that has been used for the Texstar project.

2.1 Design Criteria

The design criteria for Texstar were based on the requirements presented in RFP # ASE274L-S90 [1]. Although most requirements were explicit, several were inferred by LSC. The criteria for the system are as follows:

1. The Texstar system must provide twenty-four hours of continuous coverage to the entire state of Texas without interruption.
2. The satellite design must be of a mass which can delivered to the proper orbit by a Texas-built booster. Multiple satellites are permissible.
3. The system shall be capable of handling twenty (20) signals simultaneously. LSC and UT have jointly defined this as meaning twenty one-way color television signals, several of which can be dedicated to other uses such as data transmission.
4. Altitudes from 500 kilometers to geostationary altitude (GEO) in increments of 500 kilometers must be considered.
5. The system must be expandable to meet the ever-changing needs of the Texas educational system.
6. The system must be as inexpensive as possible.

2.2 Design Procedure

The following procedure has been followed in the determination of the Texstar system design.

1. For each Texas-built launch vehicle, LSC determined the maximum payload masses that could be launched into orbits of different altitudes and inclinations.
2. Four initial orbital configurations of various eccentricities, periods, inclinations, and altitudes were established.
3. The number of satellites required to provide continuous coverage at each of these four orbital configurations was determined.
4. The advantages and disadvantages of each configuration were determined, and a final design was chosen.
5. The system and subsystem design was completed for the final configuration.

2.3 Design Overview

In order to determine the most advantageous design for Texstar, research has been conducted in several areas. This section details the findings of the research in these areas: launch vehicles, orbit determination, satellite configuration, and ground stations. The research completed in the area of subsystem design was done for the final configuration as presented in Section 3.

2.3.1 Launch Vehicles

RFP # ASE274L-S90 has required that launch vehicles built in the state of Texas be used in the design of this satellite system. There exist two manufacturers of rocket systems which meet this requirement: Space Services Incorporated (SSI) of Houston has a conceptual design for the Conestoga 421-48B [3], and LTV Missiles and Electronics of Dallas (LTV) has a conceptual design for the Scout II [4]. The findings of the research done by LSC on these boosters is presented in this section.

2.3.1.1 Conestoga 421-48B

The Conestoga 421-48, shown in Figure 1, is a three stage rocket designed by SSI. The first and second stages are composed of four and two parallel Castor IV boosters, respectively. Each of these stages has a burn time of 55 seconds. The third stage is a single Castor IV booster which burns for 59 seconds. After the third stage burn-out, the spacecraft coasts for 500 seconds. During this time, the upper stage is spin-stabilized. After stabilization, the Star 48B upper stage booster is ignited and the satellite is sent into final orbit. This sequence is shown in Figure 2.

Figure 3 shows the launch capabilities of the Conestoga 421-48B and the Scout II as a function of altitude [4]. The TK! model used to calculate these capabilities is given in Appendix A. The Conestoga is capable of placing a 193 kilogram satellite in a geostationary orbit, which is at an altitude of approximately 35,948 kilometers. At lower altitudes the payload mass capabilities increase. The lowest orbital altitude that has been considered for the Texstar satellites is 500 kilometers. The Conestoga is capable of boosting a 711 kilogram satellite into this orbit.

The shroud of the launch vehicle is important in that it limits the size of the satellite. The shroud of the Conestoga 421-48B is shown in Figure 4. It has an interior diameter of 1.45 meters and a total available payload volume of

173.7 cubic meters.. A larger shroud which has an inner diameter of 1.96 meters is presently being designed by SSI. This larger shroud should have a payload bay volume of 317.3 cubic meters.

Although the Conestoga 421-48B has never flown, each of its individual components have been successfully incorporated in other launch systems. The Star 48B has flown 41 times, with a success rate of 94%. The Castor IV is currently being used by the LTV Standard Scout rocket. Lastly, the 1.45 meter diameter shroud has previously been used on the Thor launch vehicles.

The estimated cost of a SSI Conestoga 421-48B launch is between 18 and 20 million U.S. dollars. SSI performs all services, which includes obtaining the necessary communication frequencies used during launch, transportation to the launch site, placing the satellite in its proper orbit, and activating the satellite.

2.3.1.2 Scout II

Designed by LTV, the Scout II is a Standard Scout modified with two strap-on boosters, a design which is similar to that of the Arianne IV. The Standard Scout is stacked and uses an Algol IIIA, Castor IIA, and an Altair IIIA for its first, second and third stages, respectively. Although the Standard Scout is not capable of placing a satellite into geostationary orbit, the additional strap-on boosters will permit launch of a 69 kilogram satellite to geostationary orbit altitude. As with the Conestoga, the payload mass capabilities increase as the orbit altitude decreases. This is shown in Figure 3 with the Conestoga capabilities. The Scout II can place 238 kilograms to an altitude of 925 kilometers.

The cost estimate for the LTV Scout II is 15 million U.S. dollars. LTV also performs all launch services. These services are the same as those provided by SSI.

2.3.1.3 Launch Sites

SSI has begun negotiations with several launch sites for the Conestoga 421-48B. These facilities include Cape York, a peninsula off the northeast coast of Australia at a latitude of 10° South, Hawaii, at a latitude of 22° North, Kennedy Space Center in Florida at a latitude of 28.5° North, French Guinea, in South America, at a latitude of 5° North, and San Marco, Kenya, at a latitude of 3° South. The launch site considered for the Scout II is the San Marco Range. The locations of these sites are shown in Figure 5.

2.3.2 Orbit Determination

The orbit chosen for the Texstar satellites has a profound effect on the amount of power required and the amount of incident radiation on the satellites. The orbit also influences the amount of available mass due to the limited launch capabilities. This, in turn, determines the number of satellites required. Finally, the orbit design determines whether the satellites will require tracking or fixed ground station antennae. Several types of orbits have been considered in the design of the Texstar system. The different types of orbits are described and compared in this section.

2.3.2.1 Geosynchronous Orbit

The orbits considered for Texstar are categorized by their orbital period. A satellite in geosynchronous earth orbit (GEO) has a period equal to the sidereal rotation of the Earth, which is 23 hours 56 minutes and 4.09 seconds. Satellites in GEO have a semi-major axis of 42,164 km, or an average altitude of 35,786 kilometers. GEO orbits offer the advantage of allowing fixed antennae ground stations.

For a GEO satellite system, it would be most desirable to have a single satellite that provides continuous signal coverage. Due to the limitations of Texas-built launch vehicles, it is not possible to lift a single satellite to GEO that is large enough to meet the needs of the state of Texas. Therefore, if a GEO orbit is to be considered, several smaller satellites which each handle a portion of the requirements must be implemented.

It is possible to use multiple satellites in GEO without requiring multiple slots. This is accomplished by clustering the satellites in slightly inclined geosynchronous orbits with properly staged ascending nodes. To an observer on the Earth, an inclined geosynchronous orbit appears to move in a figure-eight pattern. This configuration is shown graphically in Figure 6. With multiple satellites, each would appear to move in the same path but would have different ascending nodes than the other satellites. Since each satellite will be in a different orbital plane, there is no danger of collision, provided that the ascending nodes are not staged with a 180° difference.

Another possible configuration is to place the satellites in a 0° inclination orbit with a slight eccentricity. In this configuration, the satellites will appear to move slightly back and forth along the equator. They will actually be moving in a circular motion that is perpendicular to the Earth, again assuring that collisions will not occur. The difficulty with this configuration that interference between satellites passing above and below one other may occur. This problem could be eliminated by having a slight inclination and slight eccentricity for each.

Once the orbit has been chosen, the number of satellites required to meet the design criteria is then determined by the mass capabilities of the chosen launch vehicle. If the inclinations of the orbits are kept relatively small, fixed antennae ground stations may be used.

2.3.2.2 Low-Earth Orbits

In this project, a low-earth orbit (LEO) is defined as being any orbit with an average altitude lower than that of GEO. Satellites in circular LEO orbits have shorter orbital periods than the Earth's sidereal period. Satellites with periods that are a fraction of the Earth's sidereal period pass over the same locations at the same time each day. These orbits are called periodic orbits.

To design a LEO satellite system, the main consideration is the number of satellites that will be required to obtain uninterrupted coverage of the state of Texas. The number of required satellites depends on the altitude, shape, and inclination of the satellite orbit as well as the ground surface area to be serviced. For ground stations at low latitudes, circular, equatorial orbits provide coverage for the fewest number of LEO satellites. Inclined orbits cause the satellite to pass north and south of the equator. These orbits are most useful if the orbit is periodic and passes over the area being covered. If the orbit is not periodic, additional satellites are required for complete coverage.

When using elliptical orbits, the perigee position moves due to the nonsphericity of the Earth. If the orbit is inclined to the critical inclination of approximately 63°, this problem is eliminated. The Russian Molniya satellites have exploited this phenomena. These orbits, however, are most useful for communications at high latitudes and were deemed unsatisfactory for Texstar.

In approximating the number of satellites required for a LEO satellite system, a circular, equatorial orbit with a ground station at 0° latitude was assumed. The number of satellites was thus dependant only on the altitude of the orbit. It was necessary to first determine the visibility angle at each altitude. This angle is the angle swept by the satellite from the time it rises above the horizon until it sets below the horizon. The number of satellites required was then determined by dividing this angle by 360° and rounding up to the next integer. Since the visibility angle can never be equal or greater than 180°, the minimum number of satellites required will always be three. These equations can be found in the TK! Solver model found in Appendix A. Figure 7 shows the number of satellites required versus altitude. For ground stations at higher latitudes, the number of required satellites increases as the minimum visible altitude increases.

As can be seen in Figure 7, altitudes above 6400 kilometers require three satellites. This includes the number required for GEO. Four satellites are required for altitudes between 2600 kilometers and 6400 kilometers. The number of required satellites rises quickly for altitudes less than 1500 kilometers. To represent each of these distinct areas, it was decided that orbits in each be considered. In the upper region, the lowest altitude, 6400 kilometers was chosen because this would allow for the largest total satellite mass for a three satellite system. The same criteria was used in the central region for four satellites at an altitude of 2600 kilometers. However, in the lower region, the highest altitude was chosen because it allowed the fewest number of satellites -- five at an altitude of 1500 kilometers.

2.3.3 Preliminary Satellite Configuration

Different satellite configurations were considered for Texstar; these are body and drum configurations. Each configuration has inherent advantages and disadvantages as described in the following sections.

2.3.3.1 Mass Approximation

To establish the approximate mass and number of transponders for a satellite, a historical survey of existing communication satellites in GEO was done. Many geosynchronous satellites were found to weigh more than the payload capacity of the launch vehicles being considered for Texstar [5]. Thus, it became necessary to determine the approximate mass of a satellite per the number of transponders on board. Figure 8 shows the satellite mass for existing drum-stabilized satellites versus the number of transponders. More recent drum-configured satellites fall into the range of 620 to 640 kilograms with 24 transponders. Figure 9 is a plot of body-configured masses versus the number of transponders. It was determined that body-configured satellites weigh slightly more than drum-configured satellites for the same number of transponders. However, because of the limited fairing size of the Conestoga, a body-configuration was chosen for Texstar. Launch data showed that the Conestoga 421-48 was capable of putting 193 kilograms into GEO. Based on the large number of body-stabilized satellites being produced with 24 transponders in the 700 - 800 kilograms range, it was determined that the satellite mass per transponder is approximately 35 kilograms. This would permit Texstar to utilize a 5 or 6 transponder satellite. Using a 4 transponder satellite weighing approximately 140 kilograms gives well over the 5% mass margin used in designing the satellite.

2.3.3.2 Drum-Stabilization

Figure 10 is an example of a drum-stabilized satellite. There are several advantages in using a drum-stabilized configuration: thermal control is more simple and attitude control is generally more simple. The solar panels are arrayed on the outside of the drum and absorb solar radiation on only a third of the surface while emitting heat from the complete surface area. The spinning of the satellite also ensures that no large temperature differences result from uneven solar heating. A disadvantage is that three times the number of solar panels are required to provide the same amount of power. The drum, however, requires fewer thrusters and relies on centrifugal force rather than pressurization for propellant feed. The decreased amount of hardware means that the drum configuration will, in general, weigh less than the body configuration. However, due to the limited size of the Conestoga fairing, the satellite drum would have to be long to accommodate all the necessary solar panels. This significantly complicates the spin-stabilization process. Also, the spinning drum configuration makes any refueling techniques difficult.

2.3.3.3 Body-Stabilization

Figure 11 is an example of a body-stabilized satellite which features deployable solar panels and antennae. The extended solar panels make spin-stabilization infeasible, thus it is stabilized on three axes. Attitude control is maintained through single or multiple flywheel momentum exchange systems. Adding more wheels to both systems increase reliability by avoiding single-point failures.

A major advantage of the body-stabilized satellite is that the communications configuration is extremely flexible. The antennae are steerable, as are the solar panels; this allows for greater adaptability in both communications and primary power supply. The steerable antennae make the body configuration desirable for LEO, where the satellite requires pointing capabilities to remain in communication with the ground stations. A major disadvantage is that a larger number of thrusters is required to keep the momentum wheels within reasonable speeds. Also, separate pressurized propellant systems are necessary to feed propellant to the thrusters. However, due to the expansion capabilities and limited fairing size of the Conestoga, Texstar will be a body-stabilized satellite.

2.3.4 Ground Stations

The earth segment of the Texstar system consists of receiving stations and receiving-transmitting stations. The majority of school districts will require only a receiving stations; several receiving-transmitting stations can be located throughout the state to serve educational, governmental and business needs. The possibility also exists for mobile transmitting units that can be shared between users as the need arises. The details of the receiving and receiving-transmitting stations are detailed.

2.3.4.1 Receiving Stations

Most schools will require a master antenna TV system, which provides reception of TV signals to a small group of users. This concept is generally used for hotels or apartment complexes, where one satellite dish services a moderate number of users. Each user has access to all of the channels independently of the other users. This master antenna TV system is easily adaptable to the classroom scenario, where a number of different broadcasts will be accessed at the same time in different classrooms. This also applies well to governmental and business transmissions, where many transmissions can be received simultaneously by the same antenna.

The master antenna TV system consists of an outdoor and an indoor unit. The outdoor unit involves a receiving antenna which feeds directly into a low-noise amplifier/converter combination (LNA/C). One LNA/C is required for each channel in the master antenna TV system. The indoor unit consists of an amplifier which passes the signal into a tracking filter, down converter and demodulator. For satellites in LEO orbits a tracking antenna is required at the ground station. The ground stations for geostationary satellites utilize fixed antennas.

The installation of fixed satellite antenna systems costs between $1500 and $1800 per antenna under state contract. Tracking antennas cost between $2500 and $3000 per antenna [6]. In that there are over 6000 middle school and high school buildings in the state of Texas, and over 3000 state government agency buildings, the difference in cost between the fixed and tracking receiving antenna dishes becomes significant. These figures do not include local governmental agencies and higher education facilities which may also wish to access the system.

2.3.4.1 Receiving-Transmitting Stations

Receiving-transmitting stations have several purposes. The first is to uplink a signal to the satellite, which in turn converts and re-transmits the signal to the earth receiving stations. Additionally, the receiving-transmitting stations are responsible for receiving telemetry, tracking and command (TT&C) information from the satellite Telemetry refers to information obtained from the satellite such as attitude information, environmental information (such as magnetic field intensity and satellite temperature), power supply voltages, and stored-fuel pressure. Commands are sent from the TT&C station in response to the satellite information obtained; attitude corrections may be made, communication transponders circuits may be modified, and station-keeping maneuvers may be performed. The TT&C functions of the earth receiving-transmitting station are of extreme importance. These stations are designed with many redundancies to insure proper functioning.

Initially, only one receiving-transmitting station will be required. This is easily expandable for educational and governmental agencies which choose to broadcast in addition to receiving information. For those agencies that may need to transmit infrequently, mobile transmitter-receiver stations may be shared. These units will allow special interactive sessions with students, or transmit data from remote governmental agencies.

3. System Design

This section details the system design of Texstar. First, the candidate systems are reviewed. The final chosen orbital configuration is presented, and the system and subsystem design is given in detail.

3.1 Candidate Systems

Table 1 summarizes the four preliminary satellite system designs. Having determined different orbital configurations for each of the four designs, the number of satellites required to fulfill the continuous coverage requirement was determined for each. This was done by making a mass approximation for the satellite per transponder, and comparing this with the previously determined launch capabilities. For each of the orbit designs, the launch data and mass approximations were integrated to determine the number and masses of satellites that would be required to provide continuous coverage. The results of this analysis is also presented in Table 1.

3.2 Final Orbit Design

The orbit chosen for the satellites in the system is a 3° inclined geosynchronous orbit. Each satellite will be placed in circular orbits at an altitude of 35,786 km. To allow for system expansion the nodes will be spaced such that five satellites could be placed in operation with the nodes equally spaced at approximately 36°. This arrangement will cause the satellites to appear to move in a figure-eight pattern above a point on the equator. The small inclination will allow fixed ground station antennae aimed at the central point.

3.3 Satellite Design

The specifics of the satellite design are presented in this section. Included is information on the power systems and solar array sizing, structural configuration, thermal control, attitude control, propulsion, and communications subsystems.

3.3.1 Power and Solar Array Sizing

The electric power subsystem is responsible for providing all electrical needs of the satellite components from time of launch until the satellite is spent. The three major components of the subsystem are the solar arrays, the batteries, and the power regulators [7,612-613]. A sample of a simple power sub-system is shown in Figure 12.

Table 2 shows the power distribution of a Texstar satellite. The total power budget is 434.1 watts. As can be seen, the communications subsystem requires the majority of the power.

Silicon solar arrays, with an efficiency of 15%, were selected to provide the power for the satellite. Silicon arrays have proven to be highly reliable and readily available. Gallium Arsenide cells were considered but the higher cost did not compensate for the increase in efficiency [8,632-642]. The calculations of the sizing of the solar arrays are shown in Appendix B. It has been determined that a total solar array size of 3.62 m^2 will provide the necessary power for all house-keeping and communications requirements. As mentioned earlier, silicon solar arrays were determined to be the best choice for the Texstar satellite system. The arrays will be deployable to minimize launch size. They will have a light honey-combed, tubular structure with a mass of approximately 12.81 kilograms, including the solar cells. This design is very weight efficient and has been used by previous satellites [8,344].

The batteries will provide the power for the satellite during the ecliptic periods of the orbit. Approximately 0.76% of the power provided by the solar arrays must be provided by the batteries during eclipses. Nickel-Cadmium batteries have been extensively used in communication satellites with a life-span of 7 years and will power the Texstar satellite. They have excellent electrical characteristics which include a low terminal voltage drop as a result of discharge, and a high charge-discharge cycle endurance [8,350-354]. The batteries will provide the Texstar with 330 watts of power and will have a mass of approximately 15 kilograms.

When the solar panels are not providing the satellite with power, the batteries must take their place. This occurs during eclipses, and as a result of switching from power source to power source, there must be a power regulator that controls the voltage along the bus each cycle. Power regulation of Texstar will be conducted through a partial shunt dissipative regulator with a single bus network, as can be seen in Figure 13. The single bus was selected because it weighs less than the dual bus, and the power shunt dissipative regulator was selected because of its low power dissipation, its simplicity, and its greater efficiency [8,363].

3.3.2 Structural Configuration

The structural configuration for the Texstar satellites will be based on existing body-stabilized satellites such as Intelsat V, Arabsat, and Spacenet. Originally, a drum-stabilized configuration was chosen, but the limitations of the Conestoga fairing size and the difficulty in refueling and expanding spinning satellites made a body-stabilized satellite more reasonable for this project. As discussed previously, the satellite dry mass will be approximately 171 kilograms and will carry 4 transponders. Table 3 shows the preliminary mass budget used to design the basic subsystems of Texstar. The solar panels on Texstar will be deployable arrays with an area of 3.62 m^2. The main body of the satellite will be approximately 1.0 m on each side. As indicated in Figure 14, the solar panels will be connected to the north-south faces with the apogee kick motor on the aft side and the antenna assembly on the forward side.

3.3.3 Thermal Control

As with all thermal control systems, the Texstar system must protect the satellite from large temperature variations as well as from overheating or freezing. All the spacecraft components must be kept within the operating temperatures given in Table 4. Active or passive systems may be used to control spacecraft temperatures. Active control systems focus on changing the spacecraft radiation characteristics after launch while passive thermal control involves the selection of spacecraft components with certain radiation characteristics and form factors. Active techniques include moving thermal shields, varying the absorptivity vs. emissivity ratio with devices such as shutters, and using resistance heating strips to change the heat conduction path in the spacecraft [9]. Passive methods involve the application of finishes, coatings and plating to exposed surfaces and insulation of vital components with thermal blankets and reflective foils.[9]

In determining the specific thermal control design for Texstar, the satellite configuration and operating environment at GEO were determined. This environment depends on the amount of incident solar flux striking the spacecraft. The radiation effects on the spacecraft vary seasonally and diurnally. At the aphelion, the flux is at a minimum of 1309 W/m^2. The incident radiation will reach a maximum of 1399 W/m^2 during perihelion with an average solar flux of 1353 W/m^2 [8]. Because the satellite makes on revolution with respect to the sun in one day, the solar flux varies. This change, however, affects only the east and west faces of the satellite and not the north and south faces. The albedo flux from earth, which is roughly one-tenth that of the sun, has been neglected in this situation because of its relatively small contribution to the overall flux on the satellite.

The configuration chosen for Texstar will have a great effect on the thermal control of the spacecraft. The choice of a body-stabilized satellite makes the thermal control more difficult than if the satellite were spin-stabilized. With spin-stabilization, the rotation of the spacecraft heats the exterior evenly, thus avoiding large temperature variations due to flux. Because the three-axis satellite maintains the same attitude with respect to the sun and the earth, temperature variations are more likely. Despite this, simple passive techniques may still be used for the thermal control of Texstar. The techniques used have been based on those used for the thermal control of Intelsat V [8].

Three basic modules have been examined in designing the thermal system for Texstar: the body, antenna, and solar array modules. Two views of the body module with several of the thermal control techniques to be used are shown in Figure 15. The main body thermal control deals with three basic concepts: heat dissipation from components in the communications and supports subsystems, absorption of solar energy and re-emission to space of infrared energy. Because the north and south face of the satellite do not experience diurnal variation in flux, high energy equipment will dissipate heat efficiently in these areas. The solid state amplifiers (SSAs) with heat sinks will be placed near these faces and will use optical solar reflectors (OSRs) as radiators. Most heat producing equipment within the satellite will be thermally linked to the radiators via direct conduction paths. Assuming a thermal dissipation of 100 W on each of the north/south panels, and a thermal efficiency of 90% during a non-eclipse period, the radiators will be approximately 0.4 m^2 on both sides. Appendix C provides more specific details.

The east/west panels of the satellite will be covered with multi-layer insulation blanket (MLI) to minimize heat fluctuations during diurnal cycles. Sensitive communications equipment such as receivers will be placed near the east/west panels away from the OSRs in stable temperature regions. The hydrazine tanks will also be placed near the east/west panels to minimize the length of fuel lines. The fuel lines and tanks will be insulated and warmed with small heater elements to keep the temperature above the freezing temperature of hydrazine. The batteries will also be insulated and heated to keep them within their operating range.

The main body must be protected from plume heating from the apogee kick motor and large temperature fluctuations from the antenna module. The entire main body will be isolated thermally from the antenna module using MLI, graphite/epoxy coating for the connecting legs and low conductance thermal spacers. The use of high temperature blankets on the aft side and low emittance surfacing on the thrust tube will be used to protect against heating from the apogee kick motor.

Due to simultaneous sun and full shadowing on different portions of the antenna structure, the antenna module is subjected to great thermal fluctuations. To protect against solar heating, most of the structural portion of the antenna module will be covered with a thermal shield. The reflectors will be coated on the concave surfaces with white paint to enhance reflection and MLI will be added on convex surfaces to inhibit great fluctuations in temperature.

The thermal control of the solar arrays is fairly simple. The solar cells will absorb solar radiation and the structure backing the cells will be coated with graphite/epoxy to enhance re-emission. For solar panels of 3.62 m^2, Appendix D shows that the solar array temperature varies between 309.66 K and 319.49 K. The structural yokes will be coated with white paint to inhibit absorption and structural deformation.

3.3.4 Attitude Control

The attitude control system for Texstar will keep the antenna pointed at the correct earth location and the solar cells oriented towards the sun, as well as correcting for attitude changes resulting from orbital disturbances [10,112]. Because the final Texstar design is a three axis-stabilized satellite in GEO, this problem is slightly more complex than for the drum-stabilized spacecraft. With the satellite in GEO, gravity gradient stabilization is not adequate because the gradient diminishes as the cube of the distance to the center of earth. The spin-stabilization requires that the antenna platform be despun to maintain proper orientation.and active nutation control to counteract the effects of fuel sloshing in partially full fuel tanks, and unwanted moments of inertia about the orthogonal axis. A body-stabilized configuration, however, requires at lease six thrusters while a spin-stabilized configuration would only require three thrusters to produce the required torques and moments [10, 119].

Once the body-stabilized configuration was chosen, it was necessary to choose between multiple or single flywheel systems. In the case of multiple flywheels, the wheels are used as reaction wheels along each axis. The speed of the wheels changes according to the disturbing torques from solar radiation. The dynamics along each axis are thus uncoupled from the other two axes. Thrusters are necessary in both single and multiple flywheel systems to keep the wheel speeds in realistic ranges. However, because six wheels are necessary for complete redundancy in the multiple flywheel system as opposed to two wheels in the single flywheel system, the single fixed momentum wheel has been determined to be more efficient for Texstar.

The single flywheel is set along the pitch axis as a momentum wheel. The spinning wheel gives gyroscopic stiffness to the system; changes in wheel speed cause moments around the pitch axis. The roll axis is controlled directly by thrusters. The yaw axis is controlled indirectly through dynamic coupling to the equations of motion about the roll axis as well as being offset from the roll axis [8, 142]. Figure 16 is a schematic of the attitude control system for Texstar. The coupling of the roll and yaw axes make it possible to eliminate the yaw sensor from the design of the attitude control system. Without a yaw sensor, only two sensors are necessary: an earth horizon sensor to maintain proper antenna direction and a sun sensor to maintain solar array orientation. Gyro assemblies are used to sense angular rates. Table 5 summarizes the components and weights of the attitude control system.

3.3.5 Propulsion

The major purposes of the propulsion system are station-keeping, major attitude control and orbit corrections. The method used in all cases are thrusters, but several types of thrusters are available: electrothermal, ion, monopropellant and bipropellant thrusters. The most commonly used systems are monopropellant or bipropellant thrusters with hydrazine as the major propellant. A schematic of a hydrazine thruster is shown in Figure 17. At least six thrusters are required to maintain attitude control of a body-configured satellite. Four additional thrusters are necessary for station-keeping, with some of these thrusters used for orbit transfer and major orbit corrections. Thus, at least ten hydrazine thrusters are required, with twenty for full redundancy. These thrusters are listed with the attitude control components in Table 5.

3.3.6 Communications

The communications subsystem includes the communications hardware on the satellite in addition to the receivers and transmitters on the ground. These are intricately linked and are specifically designed for the Texstar system.

Communication satellites are allocated a 500 MHz bandwidth in which the communications subsystem must operate. The satellite transponders, which receive, modify, amplify, and re-transmit the signal, each process different portions of this bandwidth. Standard travelling-wave tube technology transponders process a 36 MHz bandwidth, which allows for a 4 MHz bandwidth buffer zone between signals. This bandwidth is the bandwidth required to send a quality one-way color television signal, and is refered to as a channel. Newer, solid-state technology transponders are capable of processing a 72 MHz bandwidth with an 8 MHz buffer zone between signals with little distortional effect. In essence, this technology which will be used by Texstar halves the number of transponders required to cover the entire 500 MHz bandwidth. Twenty-four travelling-wave tube transponders were previously required to accommodate the twenty-four channels, whereas twelve solid-state transponders are currently needed to service the same number. This is shown graphically in Figure 18.

Frequency reuse is now currently being utilized as a method to economically utilize the frequency bandwidth. Frequency reuse sends two signals that are polarized horizontally and vertically in the same bandwidth. Essentially, the number of channels available in the same 500 MHz bandwidth doubles. Separate transponders are required for horizontal and vertical signals.

The Texstar design meets the twenty-four channel requirement by using the solid-state transponder technology coupled with the concept of frequency reuse. Each Texstar satellite carries four transponders which are capable of processing two channels each. With three satellites, this provides a total of twenty-four channels, which adequately meets the requirements.

In designing the system, it was necessary to model the specifics of the ground stations as well as the satellite communications system. To do so, an existing TK! Solver model was modified to fit the requirements of Texstar. This model is found in Appendix A. The receiving-transmitting station utilized a standard 30 meter antenna. The receiving facilities used a 3 meter antenna. From this and the previously established uplink frequency of 14.5 GHz and downlink frequency of 12 GHz, the specifics of the communications network were obtained. These are shown graphically in Figure 19 and are given in greater detail in the TK! Solver model found in Appendix A.

3.4 Ground Stations and Satellite Tracking

Initially only one receiving-transmitting station will be required. This station will be responsible for all broadcast transmissions for the satellite system in addition to all TT&C functions. For schools and governmental agencies that wish to expand capabilities temporarily , a mobile receiving-transmitting unit may be shared.

Each school or government agency that participates in the Texstar program will require a antenna receiver dish and master TV system. This will allow access to all functions of the Texstar system except for transmission.

3.5 Launch Vehicle and Launch Site

The Conestoga 421-48B is the only Texas-built booster which meets the requirements of Texstar. The only disadvantage to the Conestoga is that, while all the individual parts of the booster have successfully flown, the entire configuration has not. The Conestoga 421-48B is capable of launching 193 kilograms to GEO.

The estimated cost of a SSI Conestoga 421-48B launch is between 18 and 20 million U.S. dollars. SSI performs all services, which includes obtaining the necessary communication frequencies used during launch, transportation to the launch site, placing the satellite in its proper orbit, and activating the satellite.

San Marco, Kenya, has been selected as the prime launch site for Texstar. San Marco, Kenya has a latitude of 3° South. Two alternate sites have been chosen. The first is Kourou, French Guinea at a latitude of 5° North, and second is Hawaii, U.S.A at a latitude of 22° North. Both San Marco and Kourou have final agreements pending with SSI. An agreement has already been reached with the Hawaii site.

3.6 Expandability Options

Because the individual Texstar satellites are small, they have a limited capacity for fuel. Unless some method is devised for refueling the satellites, their lifetimes will be short. Several possibilities are currently being considered for extending the system lifetime. The first is to eliminate spent satellites by deorbiting them or putting them on an escape trajectory and replacing them with new ones. Another option is to refuel the existing satellites with a mission by the currently planned orbital maneuvering vehicle (OMV). Lastly, a bus system could be used.

If the bus system were chosen, the Texstar program would have a seven year cycle period. In the first three years of the program, a new satellite would be launched each year. These satellites would still be clustered in the manner mentioned previously. In the fourth year, a structural bus such as the one shown in Figure 20 would be launched. The mechanical arm launched with the bus would be used to capture the satellites and attach them to the bus. The bus and the satellites would be fitted with plumbing and electrical connections which would allow the satellites to share fuel and power through the extendable boom of the bus.

The only satellite design modifications required by this expansion option is the addition of plumbing and electrical ports. These ports would be placed on the satellites as shown in Figure 21. The ports were placed such that the antennae and solar panels of the separate panels would not interfere with each other. The connection points on the main bus will be placed over the hydrazine tanks shown in Figure 14.

There are several advantages to this expansion program. Launching a new satellite every year allows new technology to be incorporated in the system design. Also, since the design lifetime of each satellite is seven years, the launch of the bus in the fourth year would extend the lifetime and usefulness of the original satellites. Since it is proposed that all the satellites share fuel and power through the main booms, this will not necessarily shorten the lifetime of the most recent satellite. In fact, the extra fuel sent up in the bus may extend that satellite's lifetime as well. Another factor, less concerned with reliability than with responsibility, is the concentration of Texstar's refuse in one place. Once all the satellites are defunct, any remaining fuel may be used to launch the system out of the geosynchronous slot. The refuse may be launched into a higher orbit for eventual retrieval, or sent back into a lower orbit for retrieval or reentry.

Another system cycle may then be started, or an overlapping cycle may be completed with the new bus system replacing the old one. The cycle of the program will also allow the refinement and testing of the bus design. This is a conceptual design, but the simplicity of the concept should allow significant advances in the first years of the project.

4.0 Conclusions

This report has documented the work completed by LSC in designing Texstar, an educational communications satellite system for the state of Texas. Texstar has been designed to be launched by the Texas-built Conestoga 421-48 booster. The limitations of this booster influenced the system design. Because only small payloads could be lifted to GEO with the Conestoga, the system utilizes three small communications satellites in slightly inclined geosynchronous orbits. Because of the spacing between them, they essentially function as a single large satellite. The satellite subsystem designs as well as the ground communications systems are included in this report .

5.0 References

1. Fowler, Dr. Wallace, *Spacecraft and Mission Design,* The University of Texas at Austin, Packet #107, Spring 1990.

2. Brown, Senator Buster, *Senate Concurrent Resolution, 69th Legislature of the State of Texas,* 1989.

3. Daniels, Mark, Space Services Incorporated, Telephone Conversation, March 19, 1990.

4. Seekam, Ken, LTV Missiles and Electronics, Telephone Conversation, March 26, 1990.

5. Caprara, Giovanni. *The Complete Encyclopedia of Space Satellites*, Portland House, New York, N.Y., 1986.

6. Gessner, Donna, State Purchasing Agency, Telephone Conversation, April 7, 1990.

7. Morgan, Walter L. and Gordon, Gary D. *Communications Satellite Handbook*, Wiley, N.Y., 1989.

8. Agrawal, Brij N. *Design of Geosynchronous Spacecraft*, Prentice-Hall, Englewood Cliffs, N.J., 1986.

9. Adams, James L., *Space Technology, Volume II: Spacecraft Mechanical Engineering*, NASA, Washington, D.C., p. 84.

10. Pritchard, Wilbur L. and Sciulli, Joseph A., *Satellite Communications Systems Engineering,* Prentice-Hall, Inc., Englewood Cliffs, N.J.

Figures

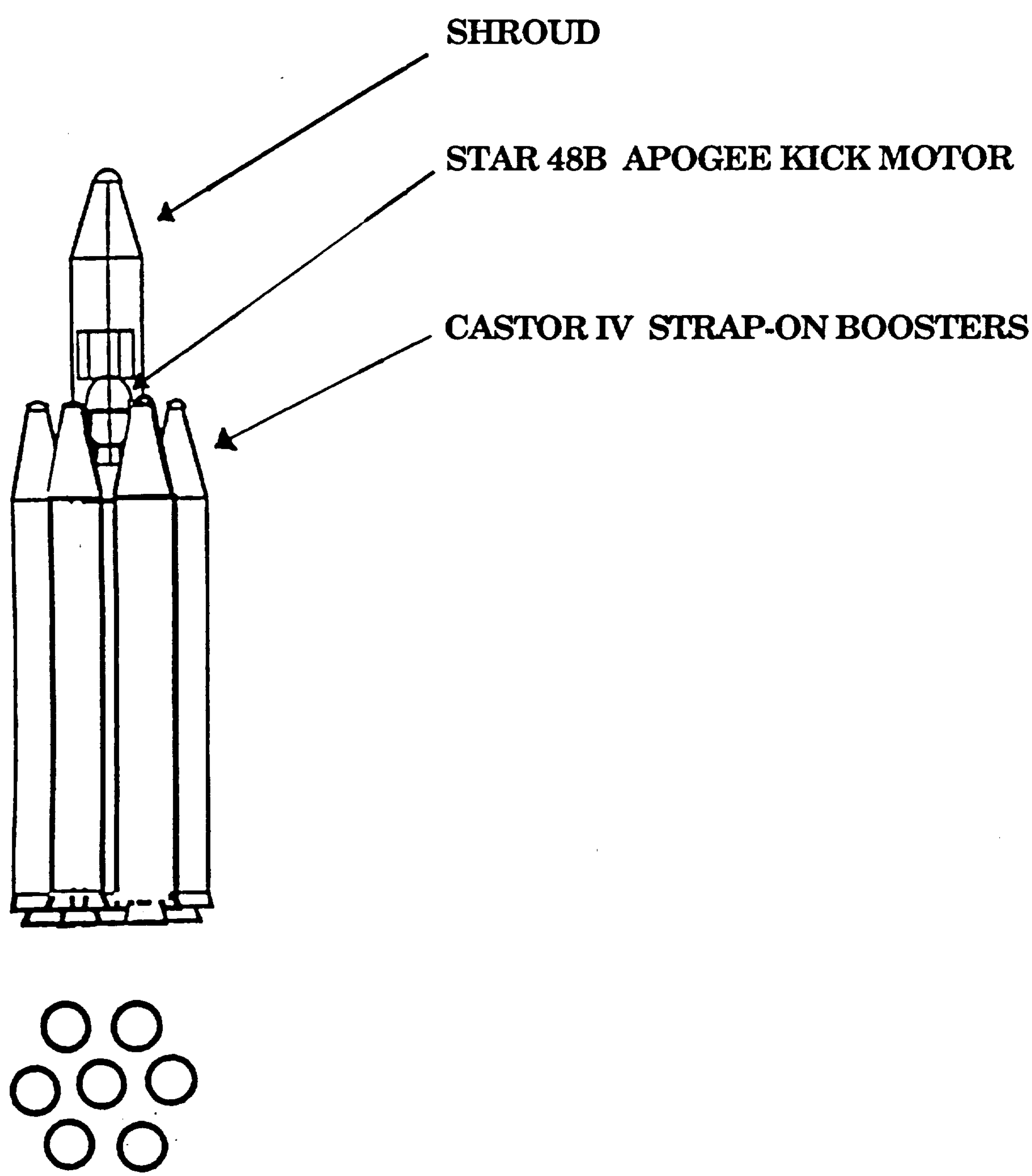
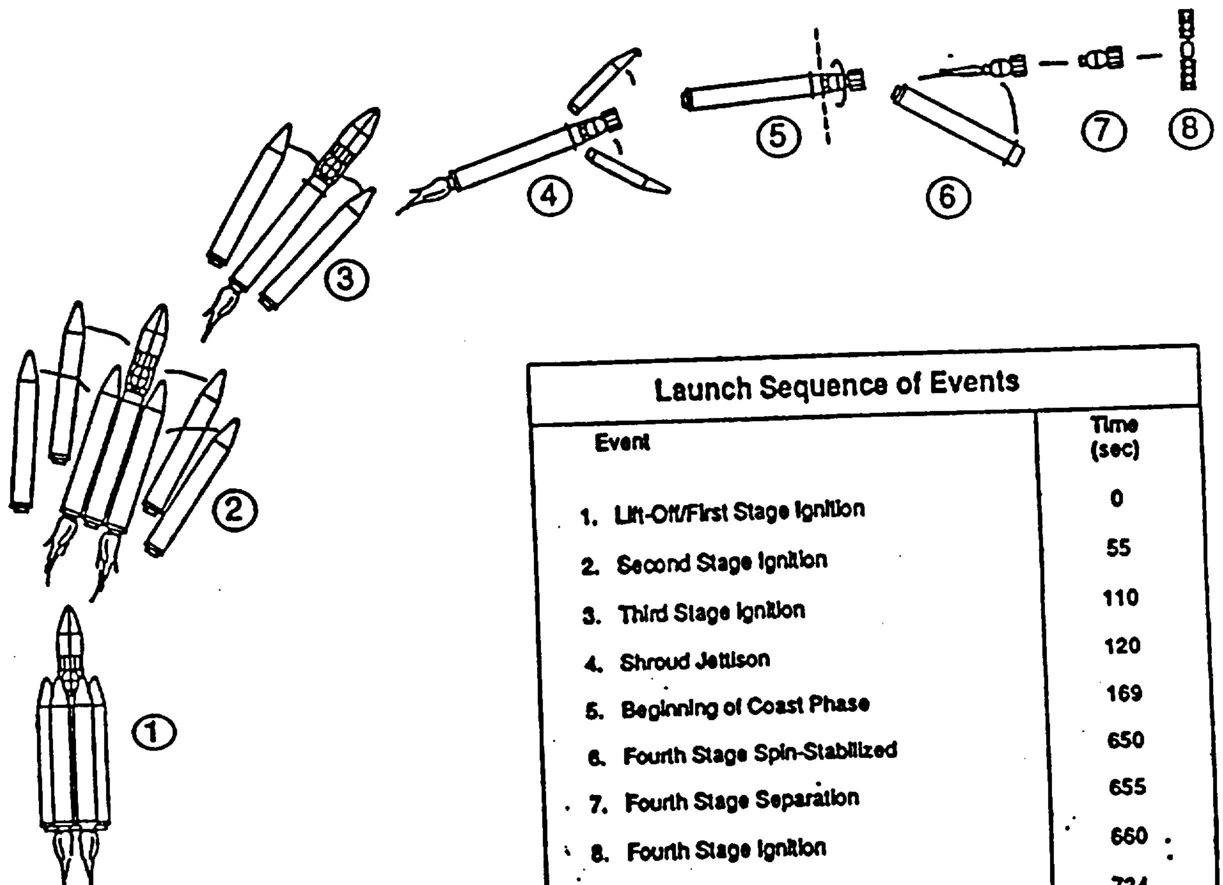


Figure 1. Launch configuration of the Conestoga 421-48B [4].



Launch Sequence of Events

Event	Time (sec)
1. Lift-Off/First Stage Ignition	0
2. Second Stage Ignition	55
3. Third Stage Ignition	110
4. Shroud Jettison	120
5. Beginning of Coast Phase	169
6. Fourth Stage Spin-Stabilized	650
7. Fourth Stage Separation	655
8. Fourth Stage Ignition	660
9. Fourth Stage Burnout	724
10. Payload Separation	729

Figure 2. Launch sequence of the Conestoga 421-48B [4].

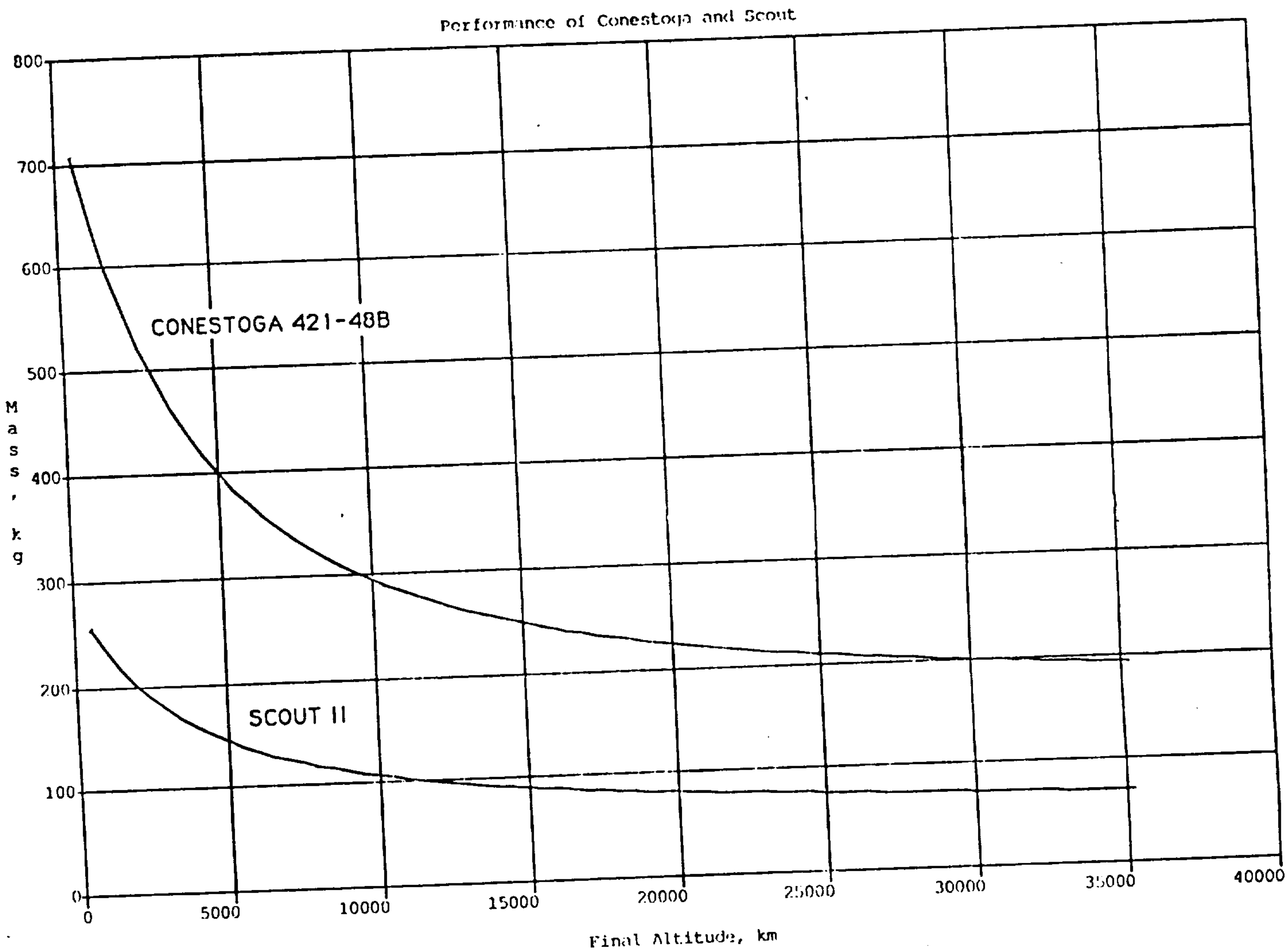


Figure 3. Launch capabilities of the Conestoga 421-48B and Scout II.

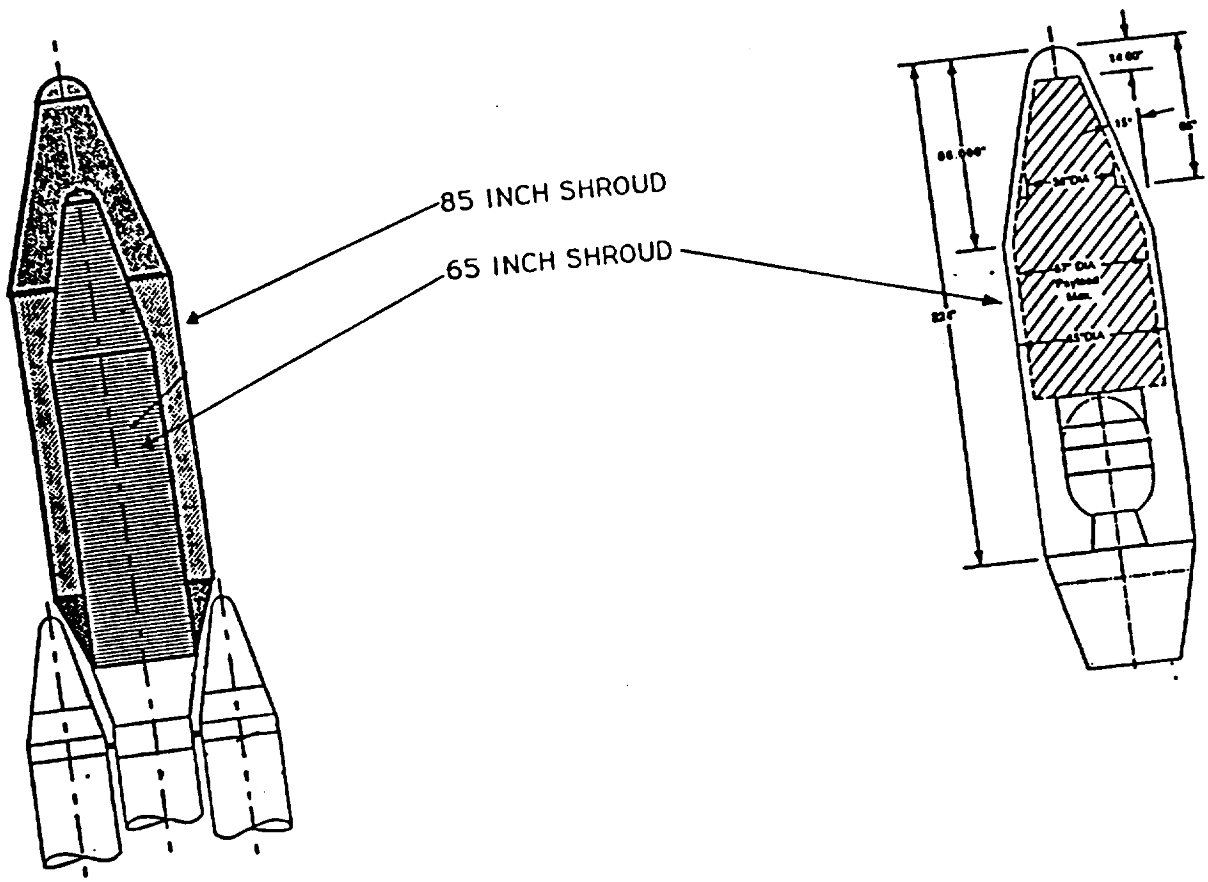


Figure 4. Payload shroud of the Conestoga 421-48B [4].

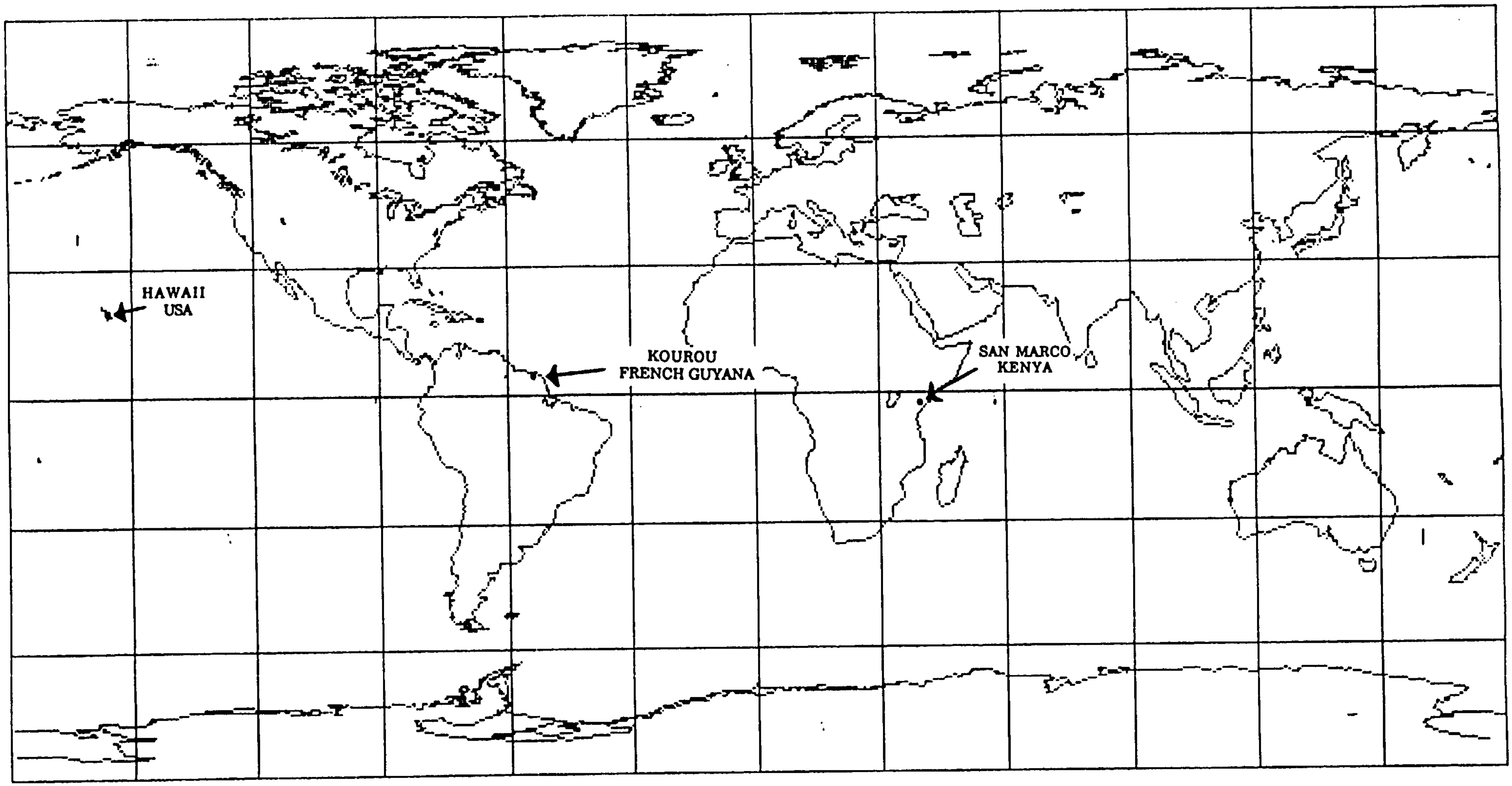


Figure 5. Launch sites of the Conestoga 421-48B [4].

 PRECEDING PAGE BLANK NOT FILMED

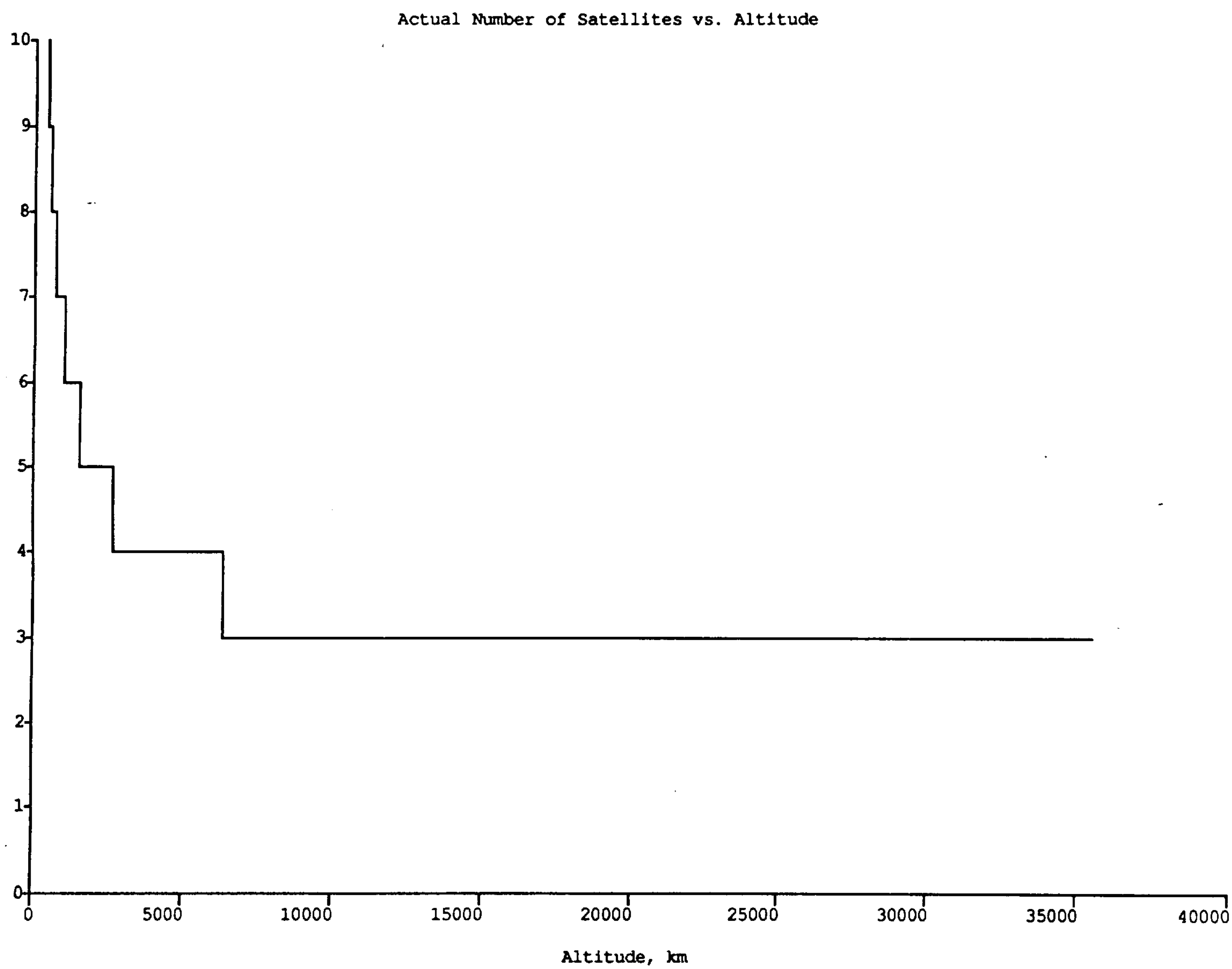


Figure 7. Number of satellites for complete coverage.

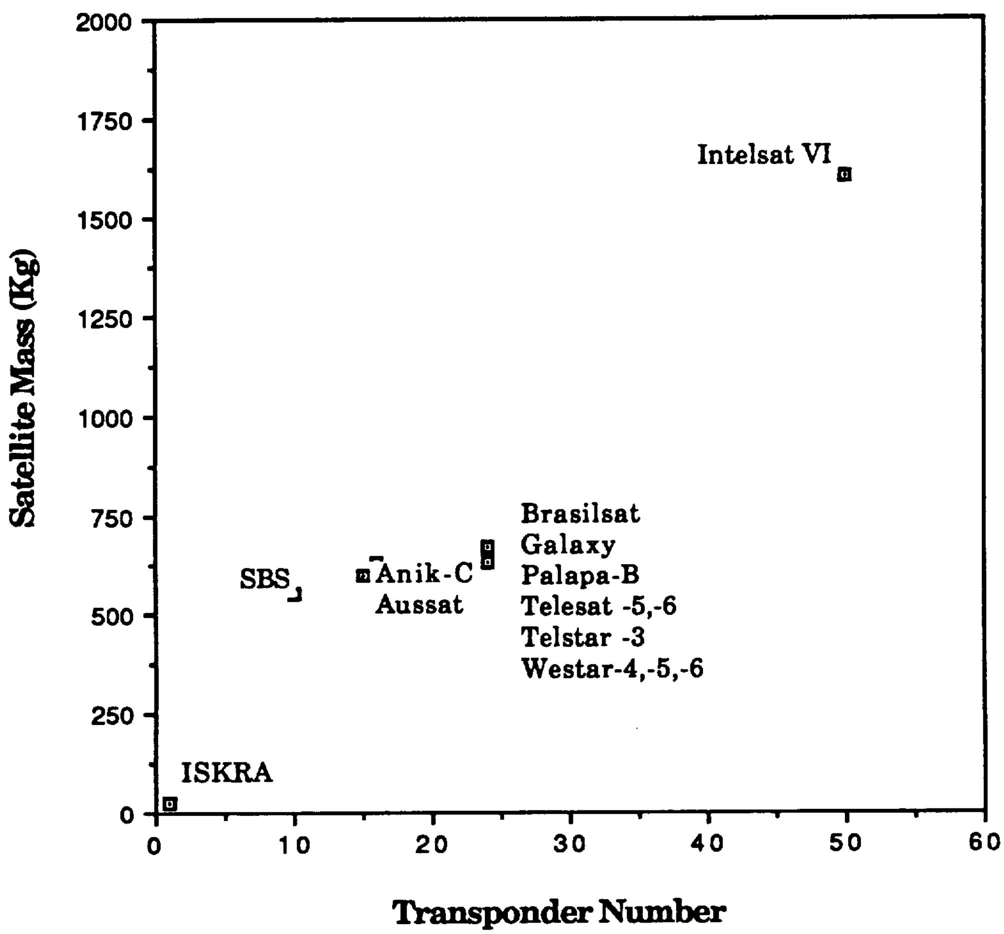


Figure 8. Drum satellite mass vs. number of transponders [5].

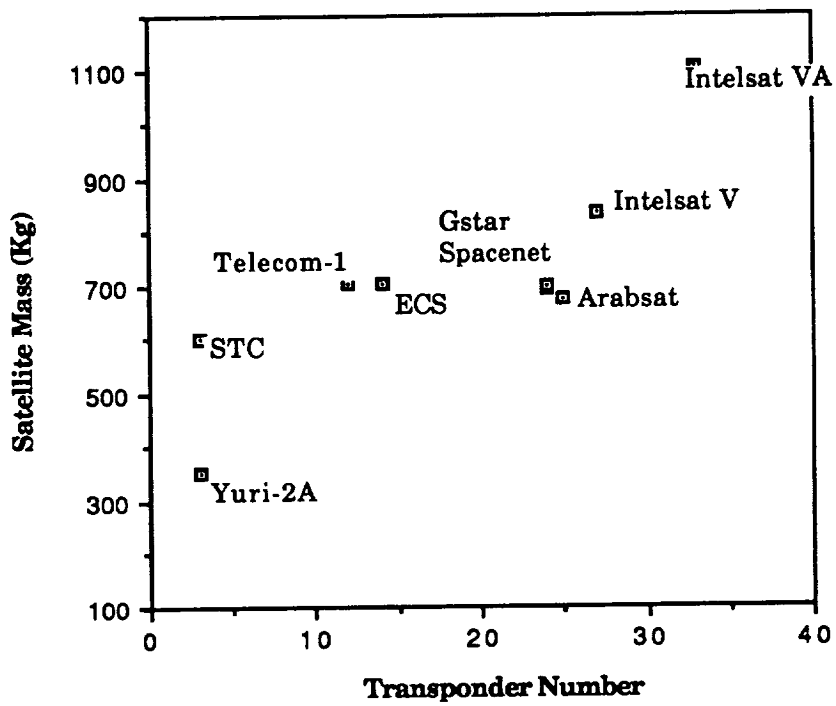
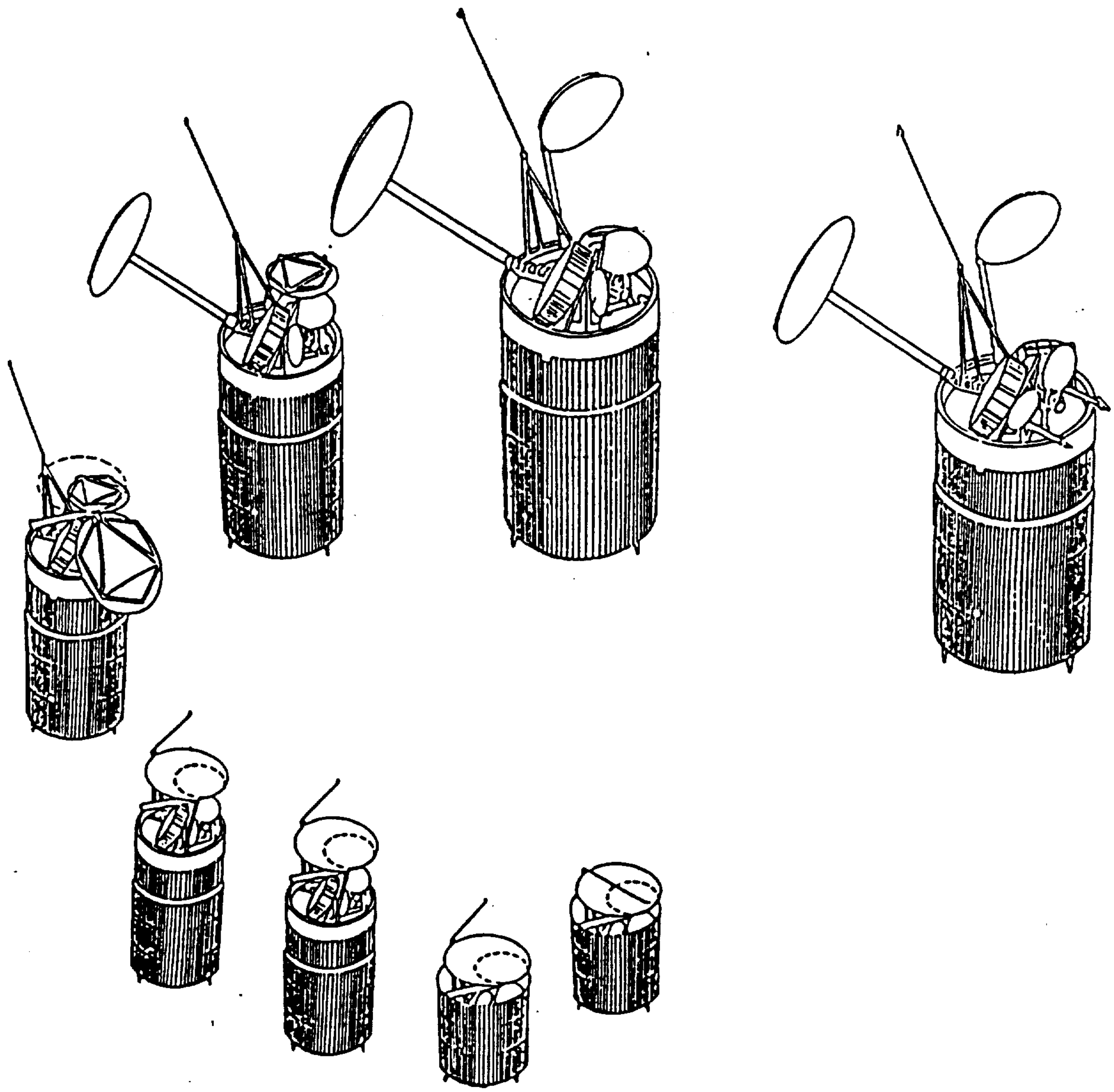
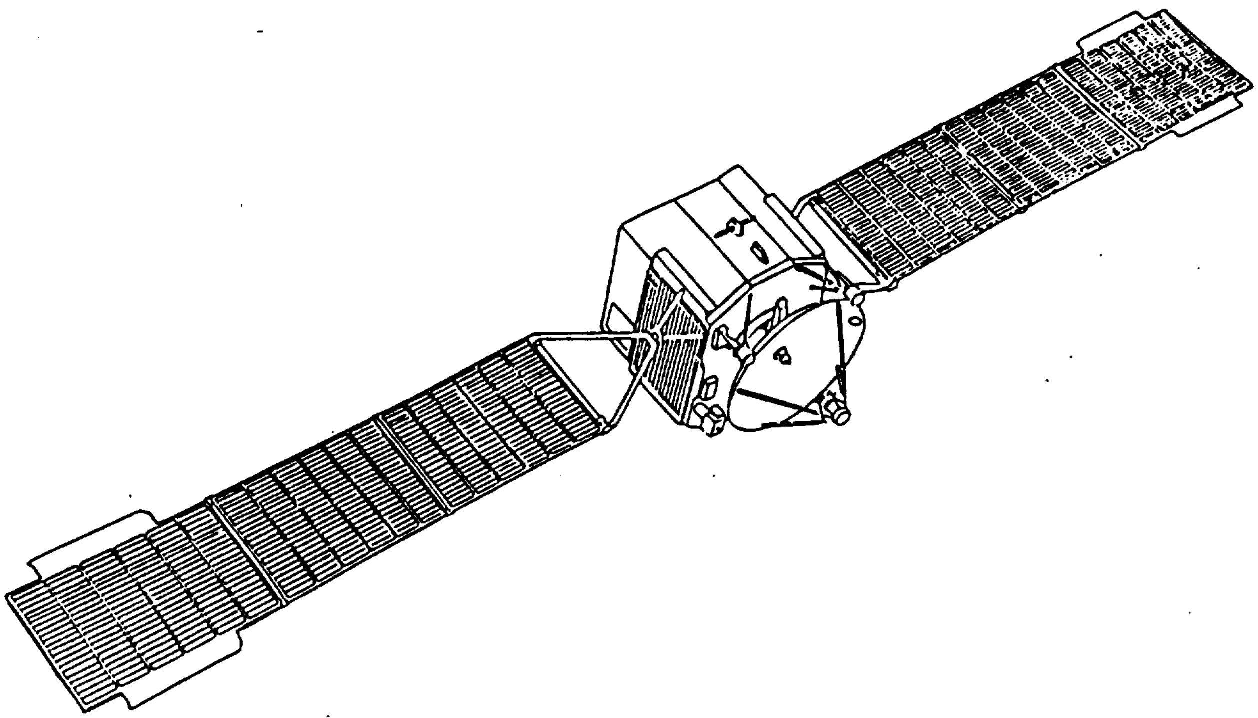


Figure 9. Body satellite mass vs. number of transponders [5].



INTELSAT VI Deployment
Taken from "The Complete Encyclopedia of Space Satellites"

Figure 10. Example of spin-stabilized satellite [5].



MARECS-A
Taken from "The Complete Encyclopedia of Space Satellites"

Figure 11. Example of body stabilized satellite [5].

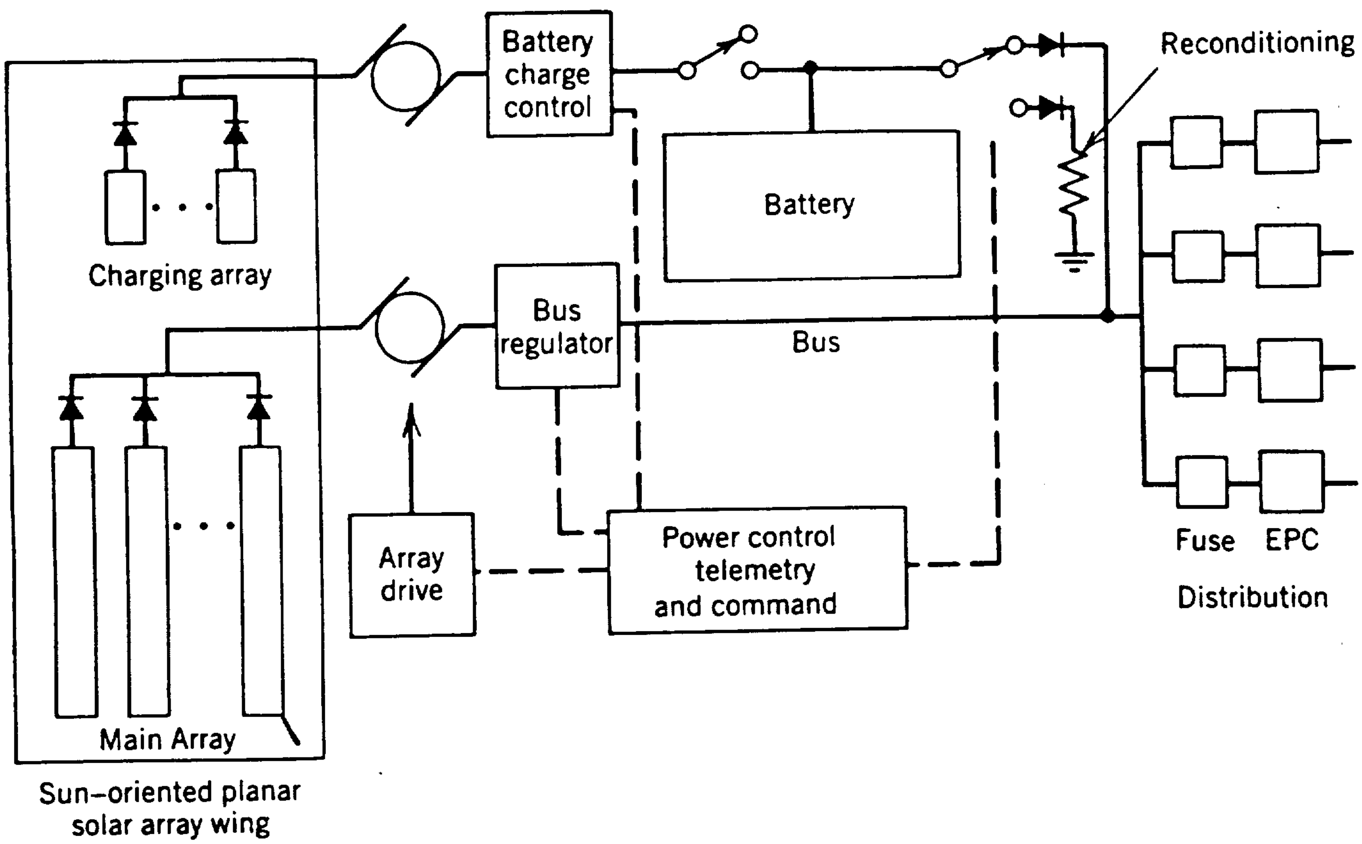


Figure 12. Schematic of simple power system [7].

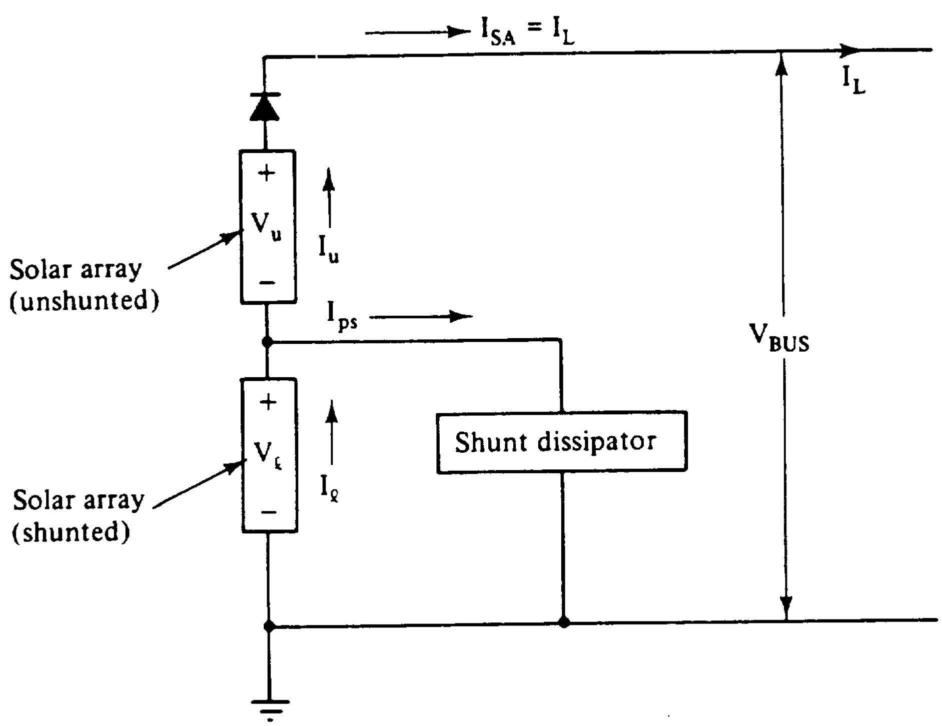


Figure 13. Schematic of power shunt dissipative regulator [8]

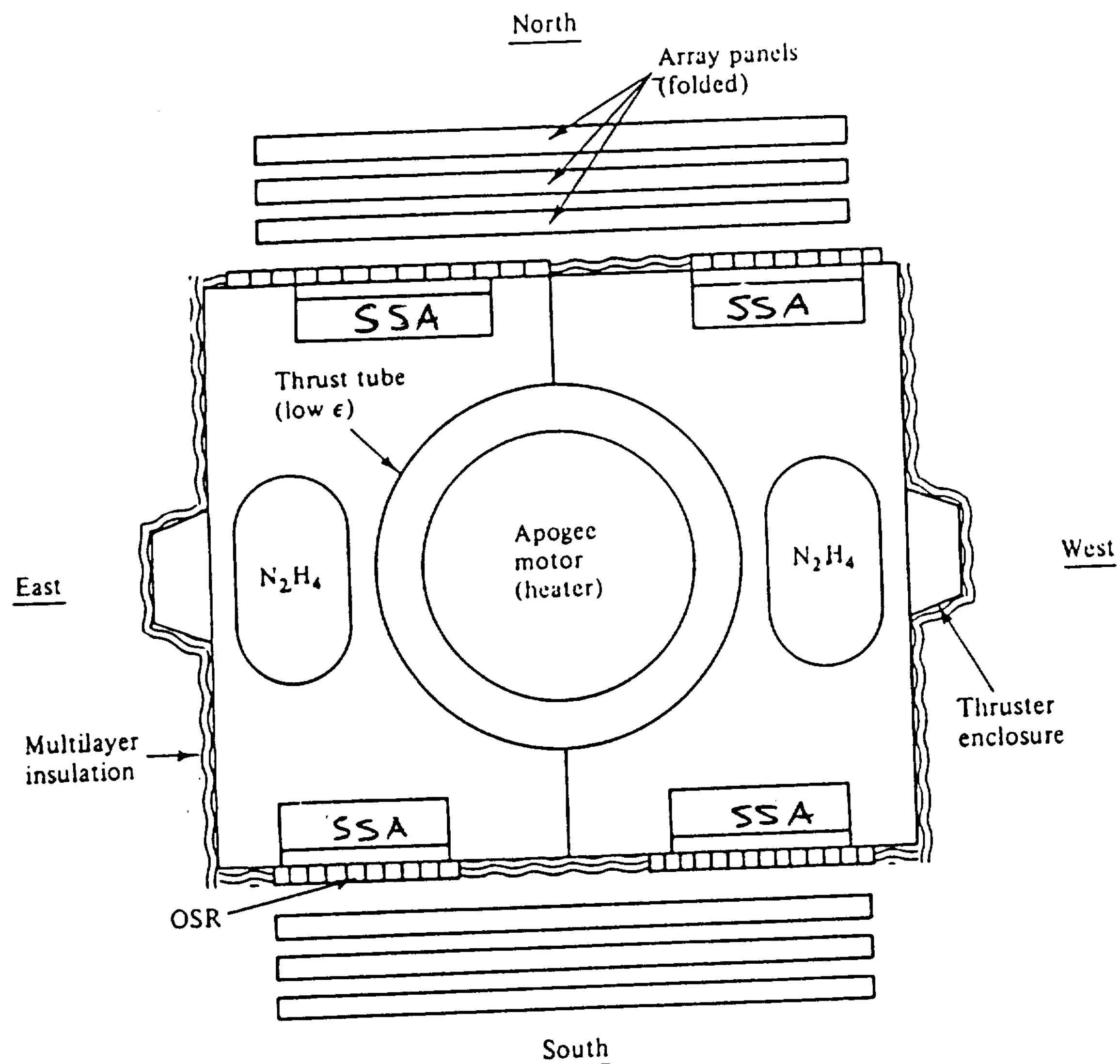
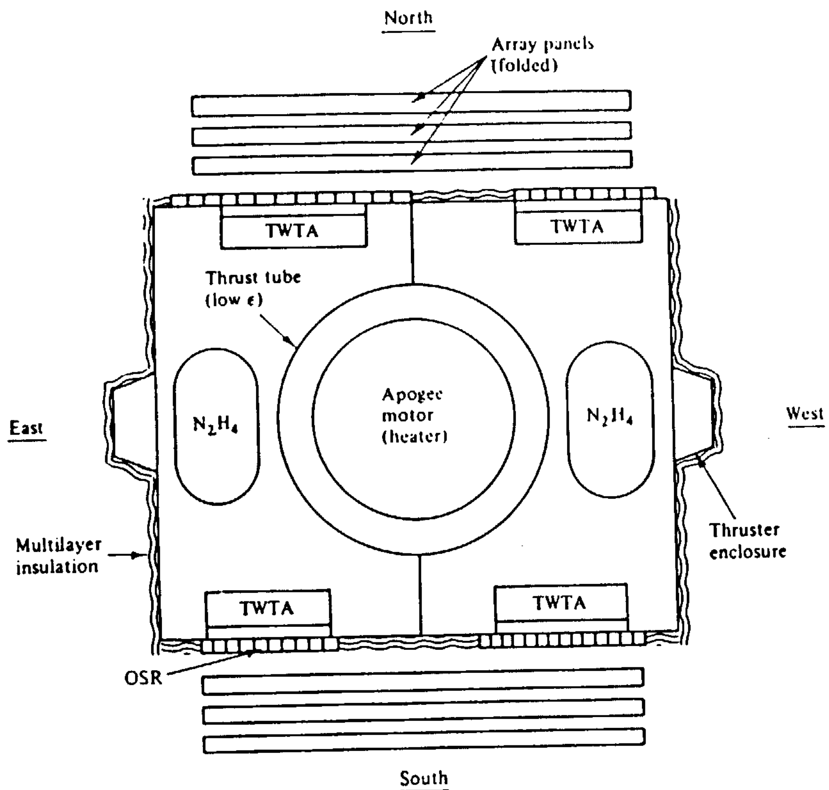


Figure 14. Single view of stowed satellite configuration [8].



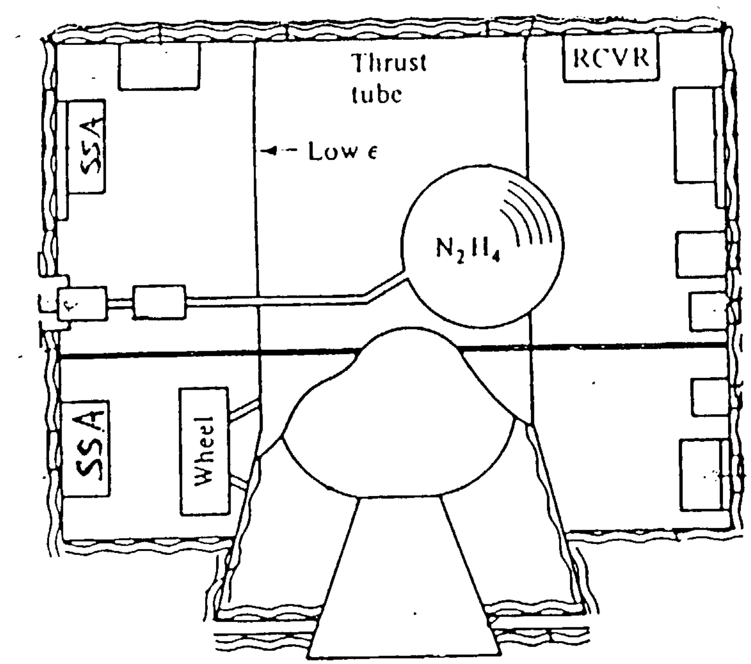


Figure 15. Elements of thermal control subsystem [8].

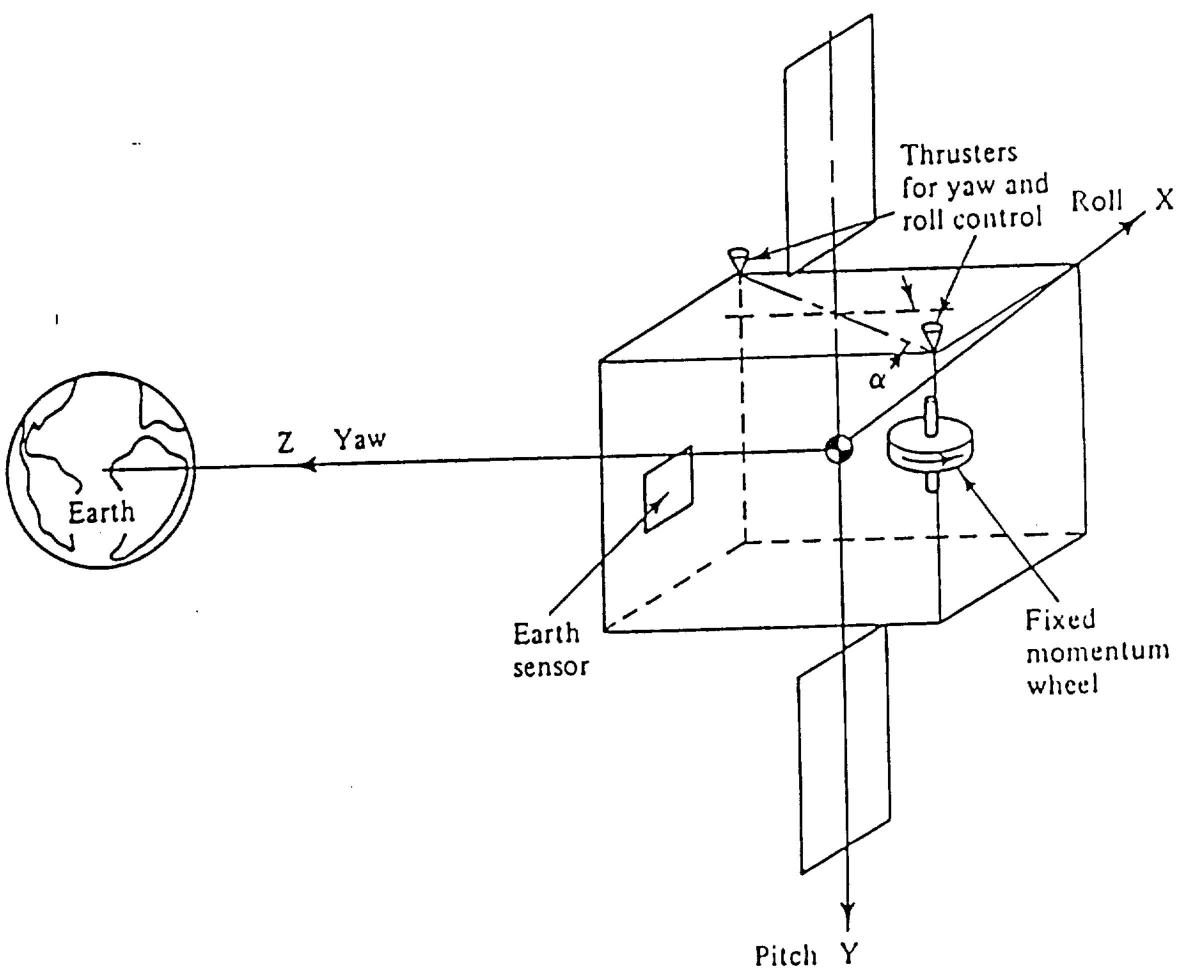
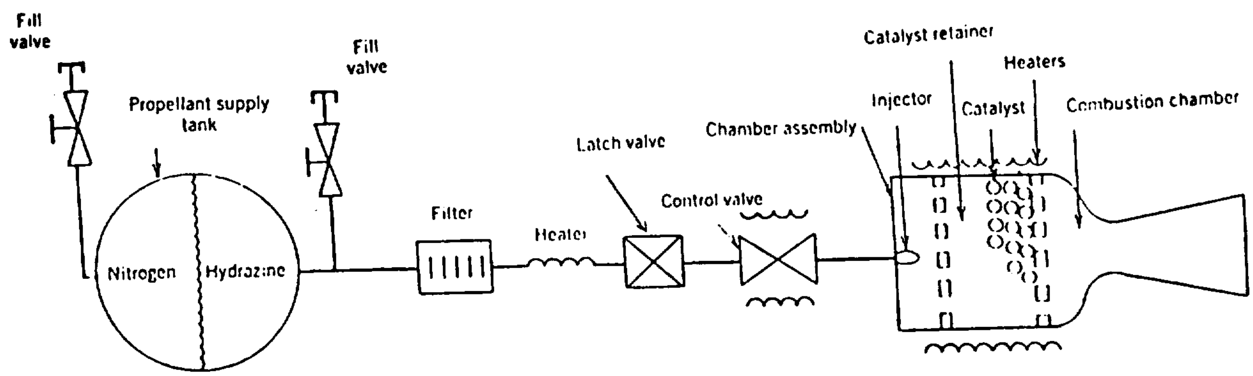


Figure 16. Schematic of attitude control subsystem [8].



FROM COMMUNICATIONS SATELLITES HANDBOOK

Figure 17. Schematic of hydrazine thruster [7].

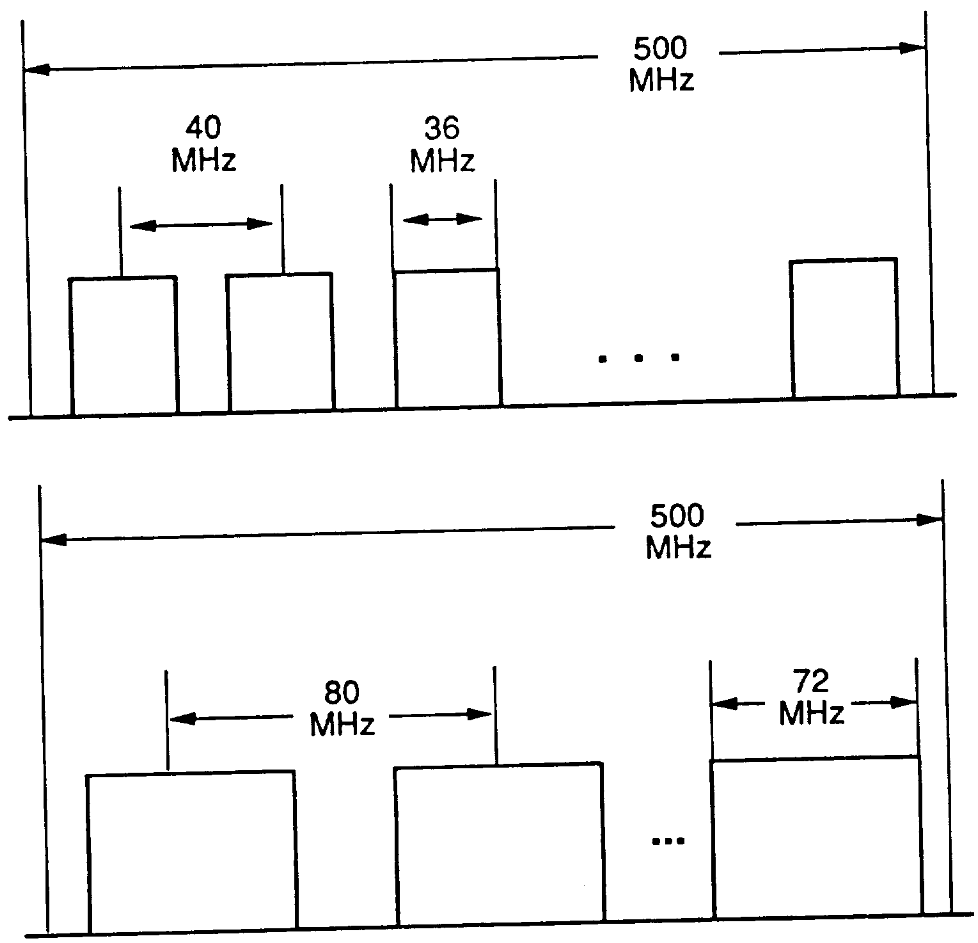


Figure 18. Bandwidth of transponders.

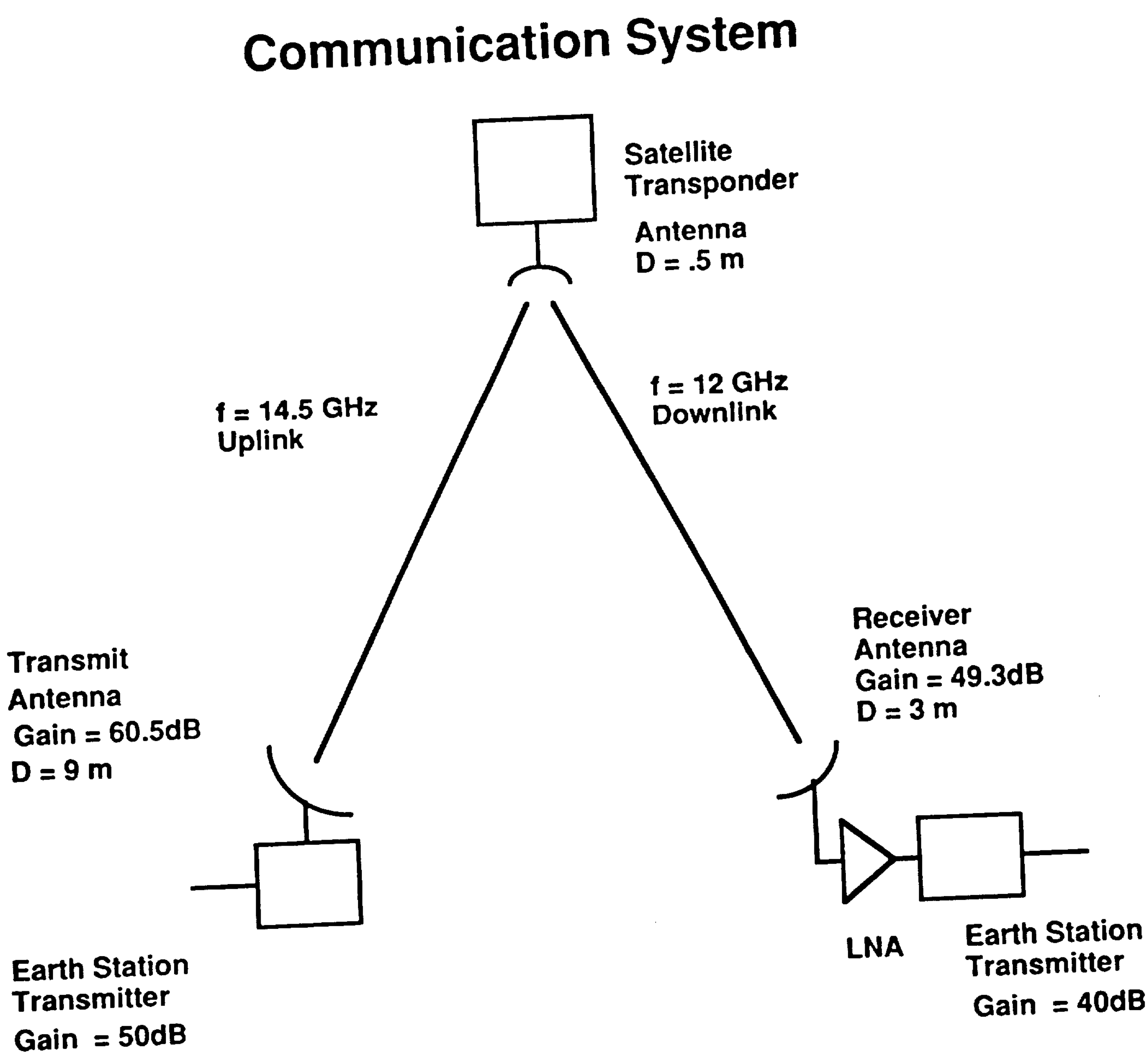


Figure 19. Schematic of communication system.

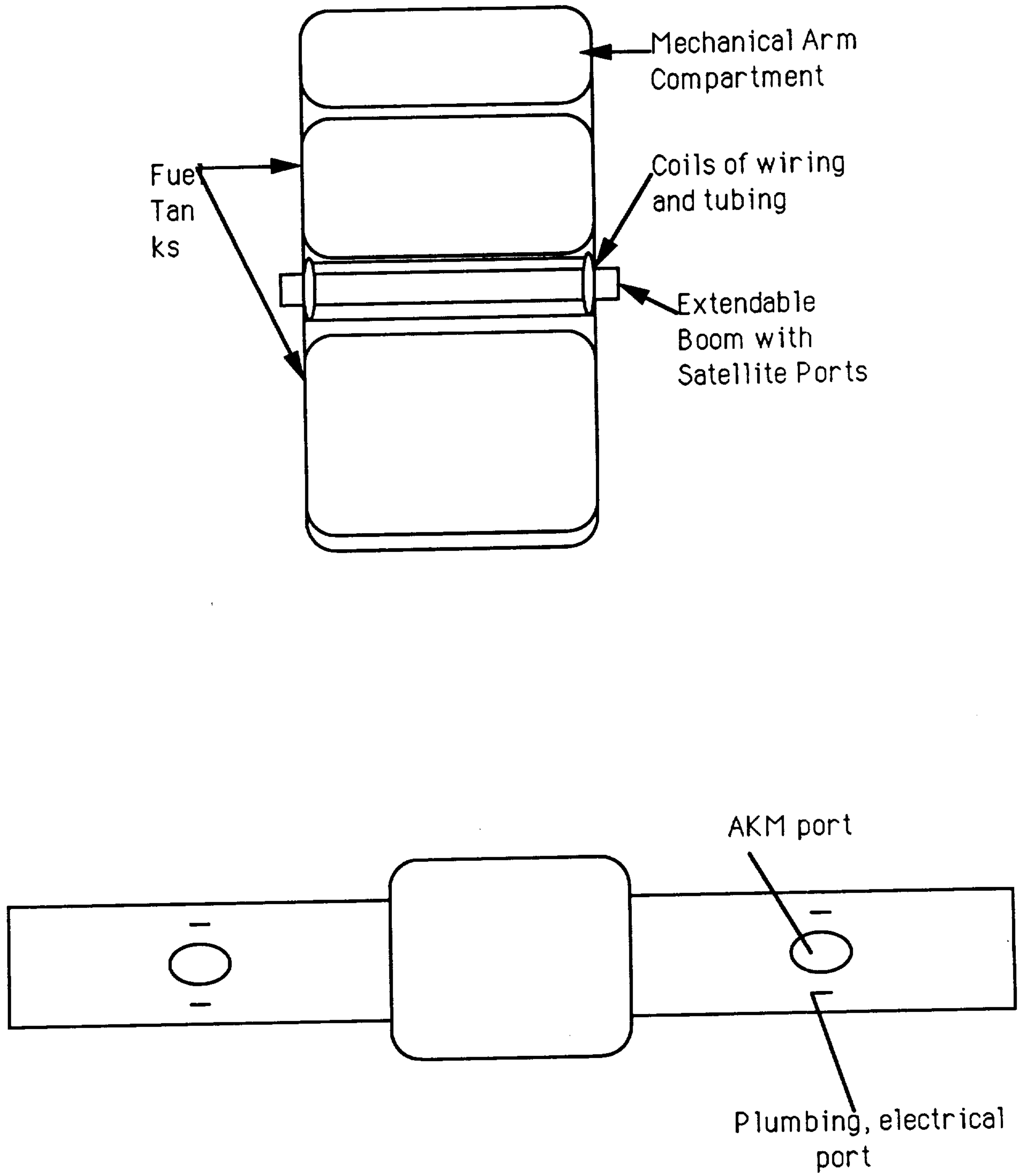


Figure 20. Schematic of expansion bus.

Tables

Table 1. Preliminary Candidate Designs Considered

Design	Orbital Altitude (km)	Number of Satellites	XPDR's per Satellite	Satellite Type.	Satellite Mass (kg)
1	GEO 35786	3	4	Drum/ Body	183
2	6400	3	6	Body	281
3	2600	4	10	Body	475
4	1500	5	11	Body	518

Table 2. Sample Power Distribution of Satellite

	Autumnal Equinox	Summer Solstice	Eclipse
Communications	769	769	768
Telemetry, command, and ranging	39	39	39
Attitude control	48	73	48
Electric power	9	9	9
Thermal control	136	86	30
I^2R harness losses	10	10	9
Battery charging	100	30	—
Total load	1111	1016	903
Power margin	243	272	76
Total power	1354	1288	979

Table 3. Preliminary Mass Budget

Conestoga to Orbit, kg		193.00
Mass Margin, %/kg	5.00	9.65
Mass to Orbit, kg		183.35
Delta-V, m/s (ACS&RCS)	200.00	
Isp,s (ACS&RCS)	295.00	
Mass of Propellant, kg		12.24
Dry Mass, kg		171.11

	% Dry Mass	Mass, kg
Support Subsystems:		
Structure	18.00	30.80
Attitude control (ACS)	7.00	11.98
Propulsion (RCS)	5.00	8.56
Thermal	4.00	6.84
TT&C	4.00	6.84
Cable-harness	4.00	6.84
AKM case	7.00	11.98
Support Subsystems Total	49.00	83.84
Totals:		
Support Subsystems	49.00	83.84
Communications	28.00	47.91
Primary Power	23.00	39.36
Total	100.00	171.11

Table 4. Temperature Operating Limits

Subsystem Component	Nonoperating (C)	Operating (C)
Communications		
Receiver	-30/+55	+10/+45
Input multiplexer	-30/+55	-10/+30
Output multiplex	-30/+55	-10/+40
Amplifiers	-30/+55	-10/+55
Antenna	-170/+90	-170/+90
Electric Power		
Solar array wing	-160/+80	-160/+80
Battery	-10/+25	+0/+25
Shunt Assembly	-45/+65	-45/+65
Attitude Control		
Earth/sun, sensor	-30/+55	-30/+50
Angular rate sensor	-30/+55	+1/+55
Momentum Wheel	-15/+55	+1/+45
Propulsion		
Solid apogee, motor	+5/+35	---
Propellant Tank	+10/+50	+10/+50
Thruster catalyst bed	+10/+120	+10/+120

Table 5. Attitude Control Masses

Subsystem Component(s)	Component Weight
Momentum Bias Wheels (2)	3.6 Kg
Earth Sensors (2)	1.0
Sun Sensors (2)	1.5
Gyro Assemblies (2)	3.0
Control Electronics	3.0
Hydrazine Thrusters (20)	6.64
Total	18.74 Kg

Appendix A: TK! Solver Models

Launch Vehicle Model

St	Input	Name	Output	Unit	Comment
	9.812	g		m/s^2	gravitational acceleration
	3.986E14	mu		m^3/s^2	gravitational constant
	6378.145	Re		km	radius of Earth
	300	alt1		km	altitude of parking orbit
L	35786	alt2		km	altitude of final orbit
		r1	6678.145	km	radius of parking orbit
L		r2	42164.145	km	radius of final orbit
L		at	24421.145	km	semimajor axis of transfer orbit
		vc1	7.7257513	km/s	circular velocity of parking orbit
L		vc2	3.0746593	km/s	circular velocity of final orbit
L		vt1	10.15148	km/s	initial velocity of transfer orbit
L		vt2	1.6078366	km/s	final velocity of transfer orbit
	295	Isp		sec	specific impulse of engine
L		dv1	2.4257287	km/s	initial delta-v
L		dv2	1.4668227	km/s	final delta-v
L		DV	3.8925514	km/s	total delta-v
					Conestoga Payload Masses
	962.32854	Wc0		kg	Mass in LEO
L		Wc1	416.26411	kg	Mass after 1st burn
L		Wc2	365.17715	kg	Mass after 1st stage sep.
L		Wc3	220	kg	Mass after 2nd burn
L		Wc4	193	kg	Mass after 2nd stage sep.
		Rc	.87727273		Conestoga mass ratio, W2/W1=W4/W3
					Scout Payload Masses
	515.79929	Ws0		kg	Mass in LEO
L		Ws1	223.11375	kg	Mass after 1st burn
L		Ws2	161.00992	kg	Mass after 1st stage sep.
L		Ws3	97	kg	Mass after 2nd burn
L		Ws4	70	kg	Mass after 2nd stage sep.
		Rs	.72164948		Scout mass ratio, W2/W1=W4/W3

S Rule

```
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
" Compute delta-v's required for transfer
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
* alt1 = r1 - Re
* alt2 = r2 - Re
* vc1 = sqrt(mu/r1)
* vc2 = sqrt(mu/r2)
* at = (r1+r2)/2
* vt1 = sqrt(mu*(2/r1 - 1/at))
* vt2 = sqrt(mu*(2/r2 - 1/at))
* dv1 = vt1 - vc1
* dv2 = vc2 - vt2
* DV = dv1 + dv2
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
" Compute Conestoga Masses
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
* dv1 = Isp*g*ln(Wc0/Wc1)
* dv2 = Isp*g*ln(Wc2/Wc3)
* Rc = 193/220
* Wc2/Wc1 = Rc
* Wc4/Wc3 = Rc
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
" Compute Scout Masses
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
* dv1 = Isp*g*ln(Ws0/Ws1)
* dv2 = Isp*g*ln(Ws2/Ws3)
* Rs = 70/97
* Ws2/Ws1 = Rs
* Ws4/Ws3 = Rs
```

Orbit Model

St	Input	Name	Output	Unit	Comment
	3.142	pi			pi
	398600	emu		km^3/s^2	grav. param. of earth
	6378	erad		km	radius of earth
L	328.2	alt		km	satellite alt. above earth
L		rad	6706	km	radius of satellite orbit
L		period	1.518	hr	period of satellite orbit
L		theta	36	deg	maximum angle between satellites
L		nsat	10		theoretical number of satellites
L		isat	10		actual number of satellites
L		ne	.00007272	rad/s	rotation rate of earth
L		ns	.00115	rad/s	rotation rate of satellite
L		nrel	.001077	rad/s	relative rotation rate of sat.
L		prel	1.621	hr	synodic period of sat.
L		vis	.1621	hr	time satellite is above horizon
L		overlap	1.234E-19	hr	overlap time between satellites
L		n	15.81		orbits of satellite before repeat
	1	d			rotations of earth before repeat
L		repeat	24	hr	repeat time

S Rule

```
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
  " Compute View Angle
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
* alt = rad - erad
* theta = 2*acos(erad/rad)
* period = 2*pi*sqrt(rad^3/emu)
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
  " Compute number of satelites required
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
* nsat = 2*pi/theta
* isat = int(nsat+1)
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
  " Compute visibility time
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
* ne = 2*pi/(24*3600)
* ns = 2*pi/period
* nrel = ns - ne
* prel = 2*pi/nrel
* vis = prel/nsat
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
  " Compute satellite overlap
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
* overlap = (isat*vis-prel)/isat
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
  " Compute repeat time
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
* repeat = (24*3600)*d
* period = (24*3600)*(d/n)
```

Communications Model

St Input	Name	Output	Unit	Comment
35786	d		km	Distance from Earth surface to sat.
.001	PI		W	Signal power from source
50	GETdB		dB	Gain of the Earth xmitter
.6	NUAET		-	Eff. of Earth xmitter antenna
14.5	fu		GHz	Uplink Frequency
9	det		m	Dia. of the Earth xmitter antenna
	PIdB	-30	dBW	Signal power from source
	GET	100000	-	
	GAETdB	60.491828	dB	Gain of the Earth xmitter antenna
	GAET	1119909.2	-	
	PETdB	80.491828	dBW	EIRP transmited from Earth station
	PET	111990924	W	
	SLUdB	206.74934	dB	Uplink space loss
	SLU	4.7308E20	-	
	PSRdB	-126.2575	dBW	EIRP of signal received by sat.
	PSR	2.367E-13	W	
.001	PO		W	Output power to device
40	GERdB		dB	Gain of the Earth receiver
50	GLNAdB		dB	Gain of the Low Noise Amplifier
.6	NUAER		-	Eff. of Earth reciever antenna
12	fd		GHz	Downlink Frequency
3	der		m	Dia. of the Earth receiver antenna
	POdB	-30	dBW	Output power to device
	GER	10000	-	
	GAERdB	49.305668	dB	Gain of the Earth receiver antenna
	GAER	85224.96	-	
	PERdB	-169.3057	dBW	EIRP Recieved by Earth station
	PER	1.173E-17	W	
	SLDdB	205.10561	dB	Downlink Space Loss
	SLD	3.2401E20	-	
	PSTdB	35.799938	dBW	EIRP of satellite xmitted signal
	PST	3801.8394	W	
1100	foot		km	Radius of the Earth fFootprint
	BW	3.5212384	deg	Beamwidth
	DSATT	.49698424	m	Diameter of the sat. xmitter antenna
.6	NUAST		-	Eff. of satellite xmitter antenna
	GASTdB	33.690095	dB	Gain of satellite xmitter antenna
	DSATR	.49698424	m	Dia. of satellite receiver
.6	NUASR		-	Eff. of satellite reciever antenna
	GASRdB	35.33383	dB	Gain of satellite receiver antenna
	GSTRdB	93.033524	dB	Gain of the satellite xpdr
3	ELdB		dB	Edge Loss
1.5	TAdB		dB	Transmitter aging factor
1	TRLdB		dB	Transmitter to reciever loss factor
	GEdB	30.690095	dB	Edge gain
	PC	5.7674552	W	Power per channel
	PCdB	7.6098423	dBW	
8	NC		-	Number of channels
.35	DCTRFE		-	DC to radio frequency power efficiency
	PT	131.82755	W	Total power for all channels
	PTdB	21.200062	dBW	

S Rule

```
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
"  Calculate the EIRP of the signal received by the satellite
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
* PIdB=10*log(PI)
* GET=10^(GETdB/10)
* GAETdB=10*log(1.096e-16*fu^2*det^2*NUAET)
* GAET=10^(GAETdB/10)
* PETdB=PIdB+GETdB+GAETdB
* PET=10^(PETdB/10)
* SLUdB=10*log(1.757e-15*fu^2*d^2)
* SLU=10^(SLUdB/10)
* PSRdB=PETdB-SLUdB
* PSR=10^(PSRdB/10)
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
"  Calculate the EIRP required of the satellite
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
* POdB=10*log(PO)
* GER=10^(GERdB/10)
* GAERdB=10*log(1.096e-16*fd^2*der^2*NUAER)
* GAER=10^(GAERdB/10)
* PERdB=POdB-GERdB-GLNAdB-GAERdB
* PER=10^(PERdB/10)
* SLDdB=10*log(1.757e-15*fd^2*d^2)
* SLD=10^(SLDdB/10)
* PSTdB=PERdB+SLDdB
* PST=10^(PSTdB/10)
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
"  Calculate satellite antenna sizes and gains
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
* BW=2*atan(foot/d)*180/pi()
* DSATR=DSATT
* DSATT=21e9/(fd*BW)
* GASTdB=10*log(1.096e-16*fd^2*DSATT^2*NUAST)
* GASRdB=10*log(1.096e-16*fu^2*DSATR^2*NUASR)
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
"  Calculate the gain of the satellite transponder
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
* GSTRdB=(PSTdB-PSRdB)-GASTdB-GASRdB
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
"  Calculate the BOL DC power required by communication subsystem
""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""""
* GEdB=GASTdB-ELdB
* PC=10^((PSTdB+TAdB+TRLdB-GEdB)/10)
* PCdB=10*log(PC)
* PT=NC*PC/DCTRFE
* PTdB=10*log(PT)
```

Appendix B: Sizing of Solar Arrays

Appendix

The following section describes the equations used to determine the size of the solar array. Also included in this section is a list of variables with their description and their source.

$$P_{EQ} = a(1+h)P_t + P_{h_o} + \frac{t_e}{d\eta_c\eta_d t_c}\left[a(a+h)P_t + P_{h_o} + P_{h_c}\right]$$

$$P_{SOL} = a(1+h)P_t + P_{h_o}$$

$$P_A = m_A e^{kN}\frac{P_{EQ}}{F_{EQ}}$$

$$P_A = m_A e^{kN}\frac{P_{SOL}}{F_{SOL}}$$

$$A = \frac{P_A}{G\eta\eta_A S}$$

EQUATION VARIABLES AND SOURCE

VARIABLE	VALUE	DESCRIPTION	SOURCE OF VALUE
$P_{A_{SOL}} =$	569.33 W	REQUIRED ARRAY POWER BOL	COMPUTED
$G =$	1370 W/m^2	SOLAR CONSTANT	SCSE
$\eta =$	.15	SOLAR CELL EFFICIENCY	AGW
$\eta_o =$	.85	FACTOR DUE TO LOSSES	SCSE
$S =$	.90	SHADOWING	SCSE
$m_A =$	1.05	SAFETY MARGIN	SCSE
$K =$	.025	DEGRADATION	SCSE
$N =$	7 YRS	SATELLITE LIFETIME	COMPUTED
$P_{SOL} =$	431.5 W.	POWER AT SOLSTICE	SCSE
$F_{SOL} =$	.948	SOLAR FLUX	COMPUTED
$P_{EQ} =$	438.04 W.	POWER AT EQUINOX	SCSE
$F_{EQ} =$	1.008	SOLAR FLUX	SCSE
$a =$	1.11	RECEIVER FACTOR	MAX VALUE ASSUMED.
$h =$	1.00	TRANSPONDER HOUSKEEPING FACTOR	COMPUTED
$P_t =$	130. W.	TRANSMITTER POWER	DETERMINED
$P_{ho} =$	98.5 W	HOUSEKEEPING POWER	ACR
$t_e =$	630 mins.	TOTAL ECLIPTIC TIME	ACR
$d =$	.55	DEPTH OF DISCHARGE	ASSUMED
$\eta_c =$	.90	CHARGING EFFICIENCY	ACR
$t_c =$	10,190. mins	TIME OF CHARGEING	

ORIGINAL PAGE IS OF POOR QUALITY

P_{he} = 30W ECLIPTIC HEATER POWER COMPUTED

η_d = .90 DISCHARGE EFFICIENCY ASSUMED

e = 1 ECLIPSE TRANSPONDER OPERATION ASSUMED

Appendix C: Sizing of the Radiator

APPENDIX : RADIATOR SIZE

$P = 100\ W$ = thermal dissipation

$\eta = 0.90$ = efficiency

$T = 310K$ = maximum allowable temperature

$\varepsilon = 0.8$ = emittance

$\alpha = 0.21$ = absorptance

$S = 1399\ W/m^2$ @ winter solstice radiation

$\theta = 23.5°$ = incidence angle

$\sigma = 5.67 \times 10^{-8}\ W/m^2/K^4$

A = radiator size

$$\varepsilon \sigma T^4 \eta A = \alpha_s A S \sin\theta + P$$

$$A = \frac{100\ W}{0.8(5.67 \times 10^{-8})(310)^4 \times 0.9 - 0.21 \sin 23.5 \times 1399}$$

$$A = 0.385\ m^2$$

At equinox during noneclipse:

$$\varepsilon \sigma T^4 \eta A = P$$

$$T = \left(\frac{P}{\varepsilon \sigma \eta A}\right)^{1/4} = \left(\frac{100\ W}{0.8 \times 5.67 \times 10^{-8} \times 0.9 \times 0.385}\right)^{0.25}$$

$$T = 282.42\ K = 9.28°C$$

Appendix D: Calculations for Solar Array Temperature

APPENDIX : SOLAR ARRAY TEMPERATURE

o2 m² A - solar array area = 3.62 m^2

α_{SE} - effective solar absorptance

82 = α_S - average solar cell array absorptance

).90 = F_P - ratio of active solar cell area to inactive solar cell area

o.14 = η - solar cell operating efficiency

$$\alpha_{SE} = \alpha_S - F_P \eta = 0.82 - (0.9)(0.14)$$

$$\alpha_{SE} = 0.694$$

$\sigma = 5.67 \times 10^{-8}$ W/m²K⁴ - Stefan Boltzman constant

A_F = array front = 3.62 m^2

A_B = 3.62 - Array back (worst case)

ε_F = 0.75 - solar cell emittance

ε_B = 0.85 - graphite/epoxy cell backing emittance

$S \cos\alpha$ = 1362 W/m² at vernal equinox

$S \cos\alpha$ = 1202 W/m² incident radiation at summer solstice

$$T_{op} = \left[\frac{\alpha_{SE} A_F S \cos\alpha}{(\varepsilon_F A_F + \varepsilon_B A_B)\sigma} \right]^{1/4}$$

T_{op} ($S\cos\alpha$ = 1362 W/m²) = 319.49 K

T_{op} ($S\cos\alpha$ = 1202 W/m²) = 309.66 K