PRELIMINARY DESIGN OF A COMMUNICATIONS SPACECRAFT--GEOCOMM

by

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1.0 GEOCOMM Overview

Companies and their employees enjoy more of a global presence now than ever before. As a result, it is not always feasible or cost effective to bring everyone together for meetings. Communications throughout an organization being one of the most important indicators of a company's success, what can be done to facilitate effective communications without face-to-face meetings. Effective meetings are difficult enough to organize and conduct when the participants share the same room, but even more difficulties arise when they are held at geographically separate locations. Strict telephone conferencing has worked to an extent, but ask anyone who has used the system and he or she will tell you that the system is almost useless in facilitating effective decision-making meetings. So what can be done to satisfy this pending need. The next best thing to face-to-face meetings would incorporate all of the important aspects of a face-to-face meeting into a viable conferencing system. A system as such saves much company time and money when compared to flying all of the participants to the same location for a one or two hour meeting.

The important aspects of a face-to-face meeting that are requirements for the conferencing system include the ability to speak freely to another person, the ability to see the other person, and the ability to give the other person data in the form of graphics or text. The delay times must be very insignificant for the system to operate effectively and foster a pseudo-live meeting environment. In addition, the system must operate over long distances and for a nominal fee.

At present, the demand for satellite telecommunications services is dwindling. Fiber optic networks are gradually becoming the medium of choice for telecommunications. However, due to the limited number of fiber optic cables presently laid and the carrier limitations of fiber optic networks, they are not yet economically viable as a medium for voice/video/data communications. As such, a satellite communications system can provide voice/video/data communications to the United States for video conferencing at a reasonable rate. The optimal system would join any two points in the United States via satellite communications at a rate which is both reasonable to the consumer and profitable for the communications company. Any number of participants may video conference simultaneously, but each must establish its own two-way link between itself and each of the other participants. A feasibility study/preliminary analysis is desired prior to beginning an expensive full-scale development of the system.

1.1 Functional Analysis

The basic functions that must be provided by the system have been identified as the receipt and transmission of voice, video, and data. Figure 1.1 depicts the top level functional flow of the system.

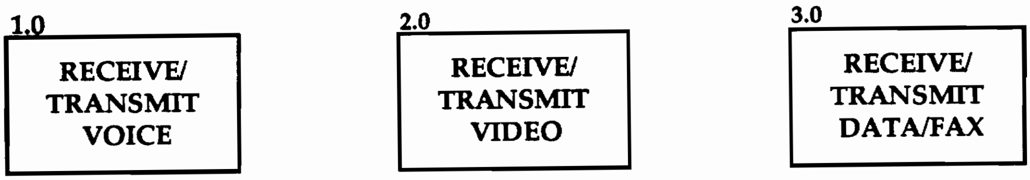


Figure 1.1 First Level Functional Flows

These functions operate independent of each other; therefore, no flows from one to the another are shown at this high level. As it turns out, their interactions occur at the third level in which the signals are combined and multiplexed into one signal and sent to the other locations as part of a packet. The second level functional descriptions of the top level functions are identical except in the type of data that comprises the signal being sent. If the link is digital, which it will be, then the representation of all three types of data is converted into a consistent signal type for transmission. Figure 1.2

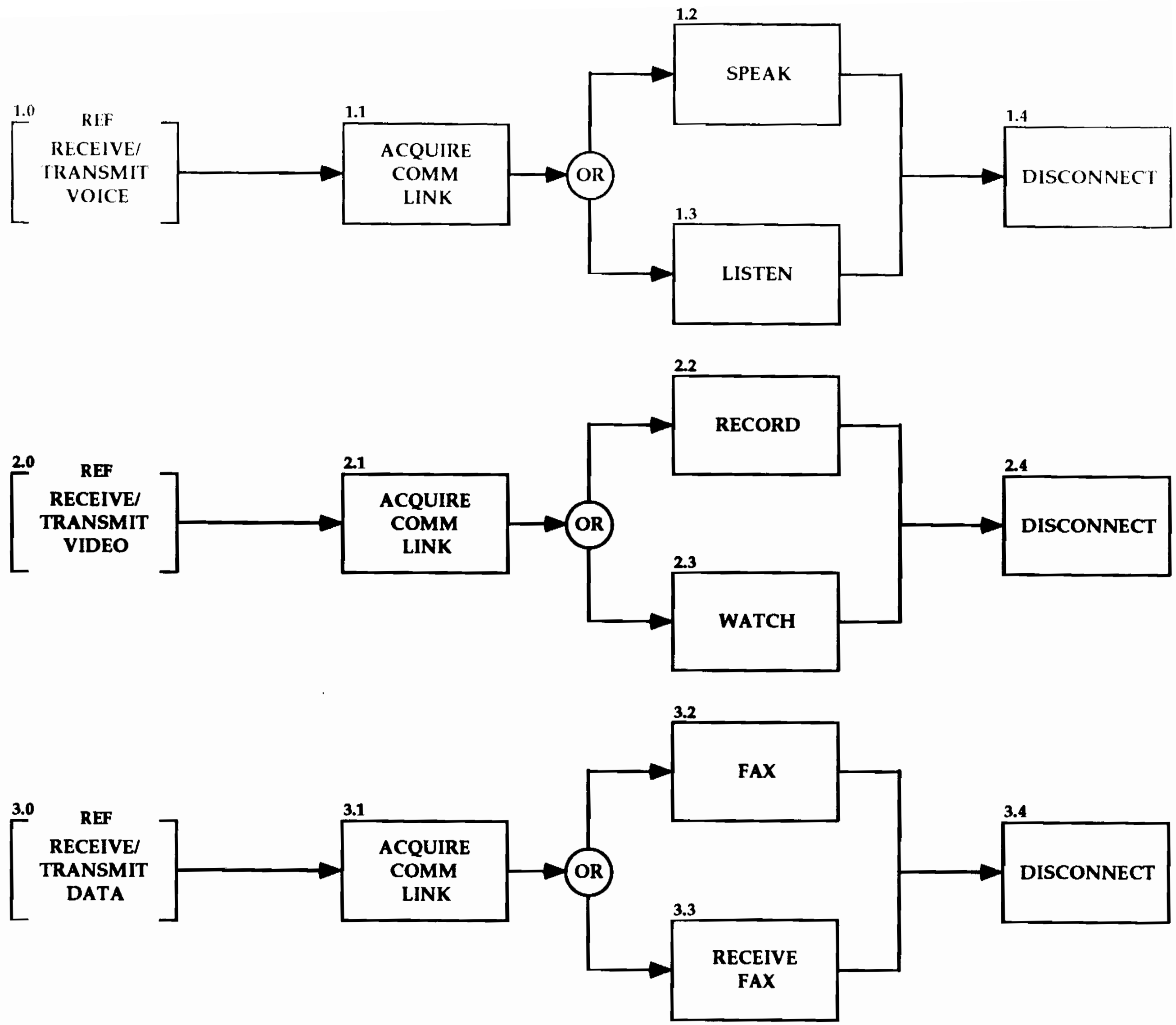


Figure 1.2 Second Level Functional Flows

depicts the three second level flows, one for voice, video, and data. Only one of the third level flows is shown because this is where the individual functions are joined through multiplexing. Figure 1.3 shows this important third level flow. It is with the X.1.2 and X.1.3 blocks that this paper is concerned. The path that the signal takes upon transmission until reception at the desired location is satellite communications.

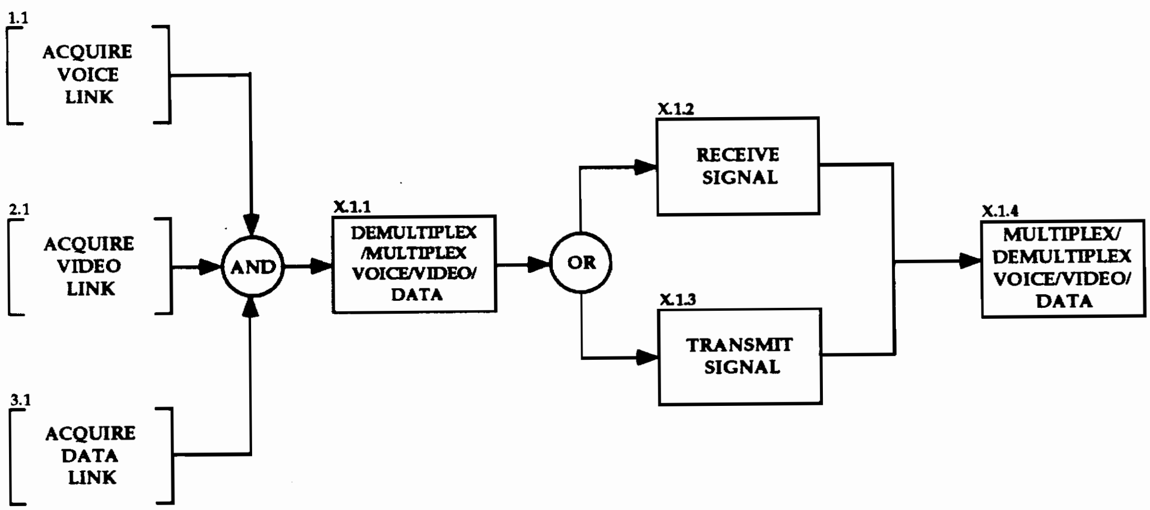


Figure 1.3 Third Level Functional Flow

The requirements upon which this paper is based are derived from the basic requirements for video conferencing. The main thrust of the paper is the design of the spacecraft that routes the signals to the correct parties and provides a clean link at a reasonable cost. Financial information related to

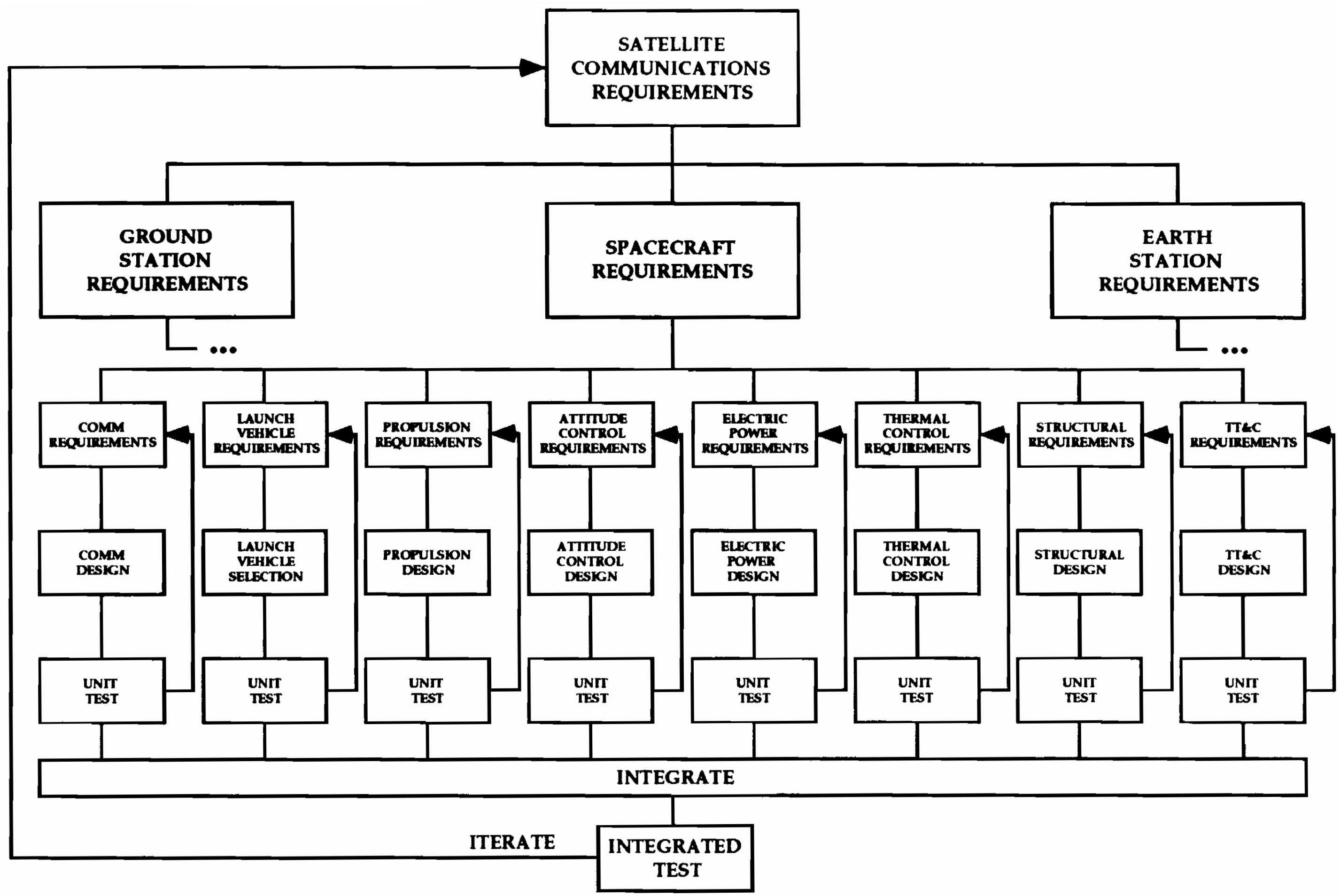


FIGURE 1.4 Iterative Design Flow of a Communications Satellite

the design and operation of a spacecraft is not included here due to its difficult nature of acquisition.

The basic flow followed in the design of a spacecraft is shown in Figure 1.4. High level system requirements are levied for a communications link and allocated to the lower level subsystems where preliminary designs are developed and tested and iterated if necessary. Each subsequent iteration results in a more exact design and employs progressively more powerful and expensive evaluation methods and tools that the previous iteration. Once acceptable designs have been achieved, the subsystems are integrated and tested wherein more requirements are uncovered and fed back into the system requirements for the next iteration. The scope of this paper addresses only the first of such iterations. All requirements are high level and the calculations are considered rough at best. Results from this level iteration help engineers and financiers to estimate the overall cost, time, and effort associated with such an undertaking, and provide the information necessary to justify the decision to commence with a satellite communications system. Cost, time, and effort information is not included in this paper due to difficulty in acquiring such information.

1.2 High Level Communications Requirements

In this case, a satellite communications link is being proposed and analyzed, although the assumed system could just as easily have been a terrestrial system satisfying the same requirements. A satellite communications system is comprised of a ground station, a spacecraft, and a number of earth stations. The video conferencing communications link requirements are allocated downward to each of these three major subsystems and are in turn allocated downward to their respective sub-subsystems. Design of the ground station and earth stations is out of the scope of this paper. The main requirements being considered here are those that comprise the spacecraft in a satellite communications system.

The basic high-level requirements for a video conferencing system are as follows:

- The channel must be capable of supporting voice, video, and data in at least full simplex and at a high enough data rate so as not to interrupt the normal flow of a meeting.
- The channel must provide service with a bit error rate of less than 2 errors/10000 bits.
- The channel must provide uninterruptable service 24 hours per day, 365 days per year.

• The system must be economically feasible.

• Another preliminary analysis determined that economic feasibility is dependent upon supplying at least 7500 simultaneous video conferencing channels.

• The FCC will only allocate Ku frequency bands at this time.

Space itself levies the majority of the requirements upon the design of the spacecraft. The communications subsystem is driven by economic and technical requirements. The remaining subsystems comprising a spacecraft are driven by the communications subsystem design, the harsh realities of space, and the other subsystems.

The only viable method for designing a spacecraft is iterative. The remainder of the document represents a generic method for producing a preliminary design for a communications spacecraft. Examples are provided to illustrate in more detail the calculations involved in some of the steps of the preliminary design. GEOCOMM (Geosynchronous Communications) is a spacecraft being designed as part of the illustrative nature of the paper. Each section provides subsystem requirements identification, commonly used equipment, and a rough GEOCOMM design. A simple depiction of the envisioned system is provided in Figure 1.5. The spacecraft antennas constantly point downward towards the earth, but the solar panels must rotate since the sun appears to move about the spacecraft once per day. Panels that move with the sun's position provide more electric power than cells that do not track the sun's apparent movement.

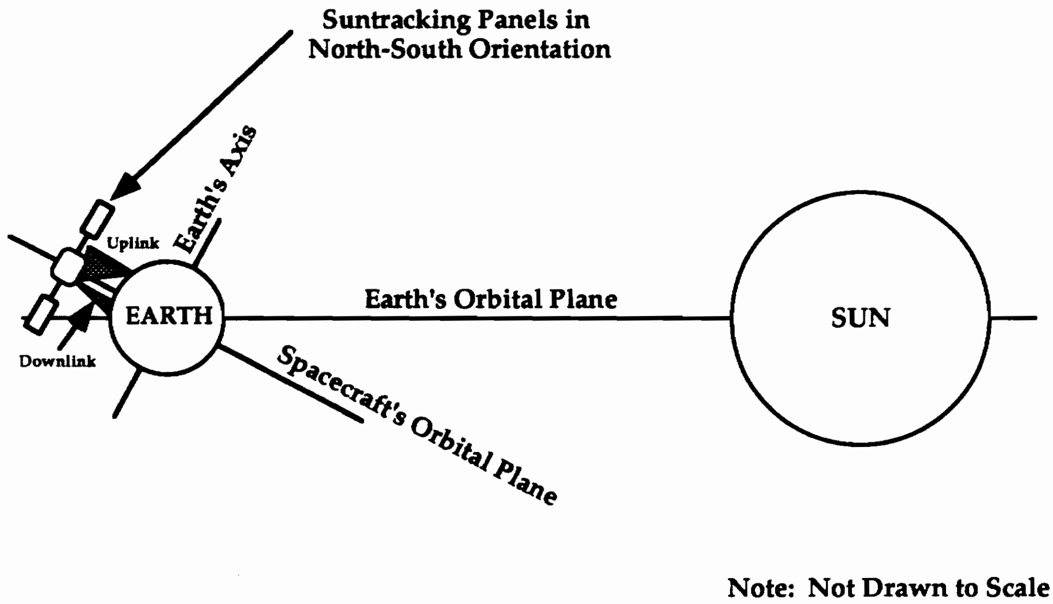


Figure 1.5 Depiction of Envisioned Communications System

2.0 Communications Subsystem

2.1 Communications Requirements

The first step in the design of a communications spacecraft is to identify the bounds of the system. Identify what the system is required to do, how well, how long, in what conditions, etc. An up-front analysis must identify potential customers of the system and analyze their needs from a communications and cost standpoint. After an initial feasibility study, if it is determined that the customers are willing to spend at the level commensurate with their needs, then a decision can be made to commence with the design of a communications system. Several alternatives should be analyzed such as satellite communications, microwave, landlines, and other forms of terrestrial communications systems. Once a trade-off analysis of these communications systems has identified a spacecraft communications system as the most cost-effective system, work on design of the satellite can begin.

Choice of spacecraft communications equipment is based on many parameters. For instance, the communications engineer must identify the number of expected users of the system, the expected utilization of the system, the overall cost of providing the service, the expected lifetime of the system, the required reliability of the system, the expected availability of the system (during eclipse when there is a loss of power from the solar arrays, for instance), and the complexity of the receive/transmit equipment relative to the design of the spacecraft. In addition to just thinking about the spacecraft, the communications engineer must also consider the associated cost of the ground station used to control the spacecraft throughout its lifetime. GEOCOMM is to be a spacecraft very similar to Intelsat V, but will provide telecommunications coverage to the continental United States only.

The area of satellite communications is a dynamic field in which new methods are constantly sought to provide increased communications capabilities at decreased costs to the consumer. The basic elements of a communications subsystem are relatively simple, but the way in which those relatively simple elements are implemented is not simple. Antenna(s) and transponder(s) comprise the communications payload on-board a communications spacecraft. From a spacecraft designer's standpoint, the size, mass, number, power, pointing accuracy, and thermal requirements of the antennas and transponders is all that is necessary to aid in the preliminary design of a communications spacecraft. The requirements levied by the communications subsystem are the most important ones which must be satisfied by the spacecraft designer. From the preliminary design, associated costs can be calculated, and an overall decision can be made confidently to proceed with a spacecraft. The requirements presented above do not represent all of the intricacies of payload design, but are representative of the types of decisions that a communications engineer must make to provide useful preliminary information to the spacecraft designer.

2.2 Communications Equipment and Implementation

A communications engineer must be familiar with the communications frequencies available for allocation. There are several (6/4 GHz and 14/11 GHz) commonly used up- and down-link frequencies, but the remaining allocations of these are relatively limited. The likelihood of acquiring 11/14 GHz is moderate and much less for 6/4 GHz since they are most popular. Also contained within this decision is a decision to utilize frequency reuse. Frequency reuse allows multiple times as much traffic to be communicated on the same frequency by polarizing channels to each other. Identifying the bandwidth permitted with a particular frequency and making a decision to polarize the signal to decrease transmission errors also enters into the equation. Polarization must be orthogonal for frequency reuse. Each beam may have one or two polarizations. Once these decisions have been made, design of the spacecraft antenna(s) and transponder(s) can begin. Antennas come in many different forms such as: wire antennas, which are used in very constrained circumstances and not very good for wide coverage; horn antennas, which are good for global or wide-area coverage; and reflector antennas, which can be used to provide a very wide-area coverage or can be shaped to provide intense spot/area coverage where necessary. Spacecraft antenna design is directly connected to the expected earth-bound antenna design. Although earth antennas are not power-limited, they may be cost-prohibitive from a customer standpoint. If the intent is to provide cheap communications service to remote areas where $750 might be considered too much for an antenna, the satellite antenna and transponder would have to be designed to pick up functionality where the cheaper (≈$250 - $500) earth antennas fall short.

Once the signals that the spacecraft must be able to discern and transmit are characterized by power and frequency, the transponders may be designed. Basically transponders do nothing more than convert the incoming signal to the down-link frequency and amplify the signal so that the earth antennas will be able to detect it. It sounds simple, but in practice it is not. Much undesired noise is added to the signal on its trip through the transponder. Transponders are often interconnected to provide redundancy in the case of failure. The amplifiers used in transponders are of two forms, travelling wave tubes (TWT) and solid state amplifiers. Solid state amplifiers are less linear than their TWT counterparts, but they require less power and non-linearity is less of a problem in digital circuits. The transponder also requires an oscillator which converts the received signal to the down-link frequency. Sometimes this is accomplished in more than one conversion, converting down to a nominal value and then back up to the desired value. Usually multiple conversions are done on spacecraft supporting more than one up- and down-link frequency.

2.3 GEOCOMM Communications Subsystem Design

The communications subsystem of GEOCOMM is an assumed design so that the other subsystems of GEOCOMM may be designed more accurately as a result of requirements levied here. The design based on requirements, but not an in-depth analysis of the communications needs of the customers. A generic flow depicting the design of a communications subsystem is shown in Figure 2.1.

For the sake of economic feasibility, this iteration GEOCOMM will deploy about 6 large array-fed antennas and 24 transponders to provide video conferencing channels to the United States. The preliminary design iteration

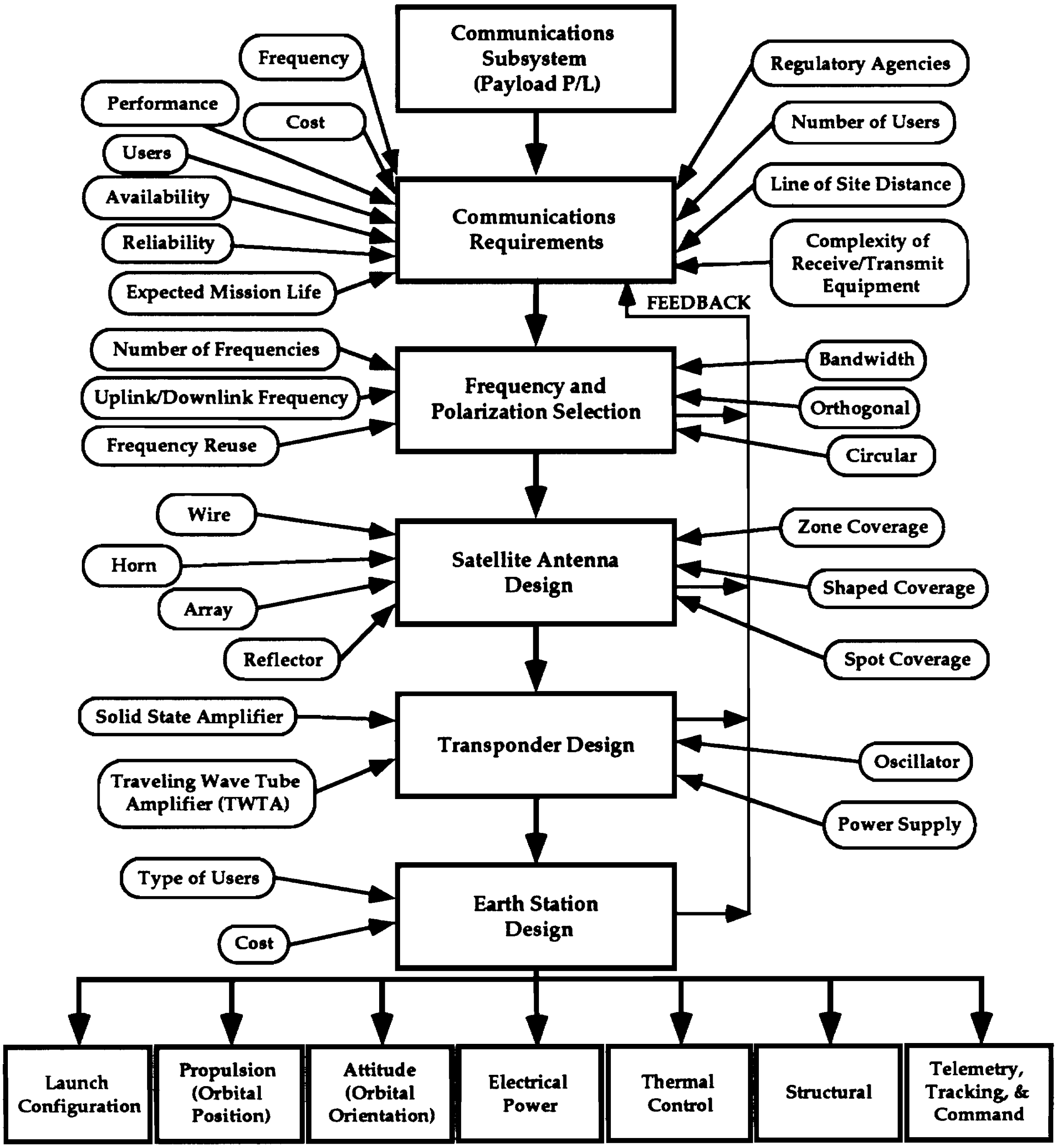


Figure 2.1 Communications Subsystem Design Flow

allows designers freedom in experimenting with different configurations and determining the optimal econimic and technical solution to the communictions problem.

At the bottom of Figure 2.1 there are seven blocks which represent requirements levied on the spacecraft by the communications subsystem. The other spacecraft subsystems must support these requirements in order for the communications system to function. The launch vehicle must be able to achieve geosynchronous orbit and not harm the subsystems significantly. The propulsion subsystem must ensure that the proper orbit is maintained. The attitude subsystem must ensure that the antennas are pointing within 0.1° of their desired directions. For the communications subsystem to satisfy mission requirements, the electrical power subsystem must provide 2000 W constant to the communications subsystem until spacecraft end-of-life (EOL). The thermal control subsystem must ensure that the communications subsystem maintains proper operating temperatures -10 to +30℃ for the electrical equipment and -170 to +90℃ for the antennas. The structural subsystem must protect the communications subsystem from the launch environment, keep the antennas in place, and maintain relative positions between subsystems. Finally the telemetry, tracking, and command subsystem must provide payload (communications) operations and health data to the ground station for analysis and action. Together these subsystems comprise a spacecraft whose sole responsibility is to keep communications antennas in place and transponders operating.

3.0 Launch Vehicle

3.1 Launch Vehicle Requirements

An orbit is initiated by placing the spacecraft at a distance (radius) from the center of mass of the object being orbited (earth), and imparting a velocity in the desired direction of the orbit. It is the launch vehicle that imparts this velocity, and does so in a manner which protects the spacecraft from the harsh conditions experienced while escaping the earth's atmosphere. Launch vehicles are used to place spacecraft outside of the earth's atmosphere and into initial orbit. Not all launch vehicles deliver spacecraft to the same initial orbit. The Space Shuttle delivers spacecraft to a circular parking orbit, while most rockets deliver spacecraft to an elliptical transfer orbit with the apogee of the transfer orbit coincident with the apogee of the desired geosynchronous orbit. While the launch vehicle itself is not part of the overall operational satellite communications system, it is required by the spacecraft for deployment, and it in turn levies requirements on the spacecraft. This section identifies the requirements levied on the launch vehicle by the mission.

The place to start in the selection of a launch vehicle is with an analysis of the desired final orbit of the spacecraft. The orbit of GEOCOMM is geosynchronous, which means that the satellite appears to remain in the same position in the sky at all times. In order for an orbit to be geosynchronous it must be circular (eccentricity = e = 0.00), its inclination must be zero (i = 0), and its radius and velocity must be kept constant. A variety of perturbations exist to pull a spacecraft out of its orbit, but in a preliminary analysis and design they can be neglected for orbital analysis. However, these perturbations are not neglected later in the preliminary design of the propulsion subsystem which keeps the spacecraft in its desired orbit.

GEOSYNCHRONOUS ORBIT

The period of the orbit is exactly one day. A sidereal day, the time from the rising of a star to the next rising of the same star, is used to determine the period of the earth's rotation since it is a more accurate representation of a true "natural" day. A sidereal day is 23 hours 56 minutes 4.09 seconds, or 86,164.09 s. The radius of the circular geosynchronous orbit is determined by

$$r_{geo} = [\mu_e P_{geo}{}^2/(4\pi^2)]^{1/3} \tag{3-1}$$

$$r_{geo} = \text{Radius of geosynchronous orbit from earth's center} \tag{3-2}$$

$$\mu_e = GM = 398{,}601.2\ \text{km}^3/\text{s}^2 \tag{3-3}$$

$$G = \text{Gravitational constant } [km^3/(kg * s^2)] \quad (3\text{-}4)$$

$$M = \text{Mass of earth [kg]} \quad (3\text{-}5)$$

$$P_{geo} = \text{Period of geosynchronous orbit} = \text{Period of earth's orbit} \quad (3\text{-}6)$$

$$P_{geo} = 86{,}164.09 \text{ s} \quad (3\text{-}7)$$

$$r_{geo} = [398{,}601.2 * 86{,}164.09^2/(4\pi^2)]^{1/3} = 42{,}164.2 \text{ km.} \quad (3\text{-}8)$$

The earth's radius is 6,378.2 km; therefore, the altitude (measured from sea-level) at which a spacecraft must be located to be in a geosynchronous orbit is 35,786.0 km. In addition to having an altitude of 35,786.0 km, the spacecraft must also have the proper velocity or the spacecraft will not stay in that orbit for any period of time. The velocity for geosynchronous orbit is calculated by

$$V_{geo}^2 = \mu_e(2/r_{geo} - 1/a_{geo}) \quad (3\text{-}9)$$

$$V_{geo} = \text{Velocity at radius } r \text{ in orbit with semimajor axis } a \text{ [km/s]} \quad (3\text{-}10)$$

$$a_{geo} = \text{semimajor axis of geo orbit} = r_{geo} = 42{,}164.2 \text{ km.} \quad (3\text{-}11)$$

Since a circular orbit requires that the semimajor axis be equal to the radius $a = r$, the equation for the velocity of a geosynchronous orbit is simplified to

$$V_{geo}^2 = \mu_e/r_{geo} \quad (3\text{-}12)$$

$$V_{geo} = (\mu_e / r_{geo})^{1/2} \tag{3-13}$$

$$V_{geo} = (398{,}601.2 / 42{,}164.2)^{1/2} = 3.075 \text{ km/s}. \tag{3-14}$$

Geosynchronous orbit is defined then as $r_{geo} = 42{,}164.2$ km, $V_{geo} = 3.075$ km/s, and $i = 0$.

GETTING TO GEOSYNCHRONOUS ORBIT

Unfortunately, there are no launch vehicles currently available that launch spacecraft directly into geosynchronous orbit. A series of orbital maneuvers utilizing a variety of launch systems is used to attain the desired orbit. The most commonly used propulsive systems include the Space Shuttle, rockets, apogee boost/kick motors (ABM/AKM), and perigee motors. The Space Shuttle delivers spacecraft into a low-altitude (alt = 185.2 km) circular parking orbit, while rockets deliver spacecraft into transfer orbit ($r_{apogee} = 42{,}164.2$ km). If a spacecraft is lifted to initial orbit with the Space Shuttle, it will need a perigee motor to go from parking orbit to transfer orbit, where it will require an apogee motor to go from transfer orbit to geosynchronous orbit. If a spacecraft is lifted to initial orbit by a rocket, then the spacecraft only needs an apogee motor to go from transfer orbit to geosynchronous orbit.

Obviously there are two other orbits, parking and transfer, which must be analyzed and understood in the planning of a geosynchronous spacecraft launch. Figure 3.1 shows the three main orbits which are important to a geosynchronous spacecraft mission.

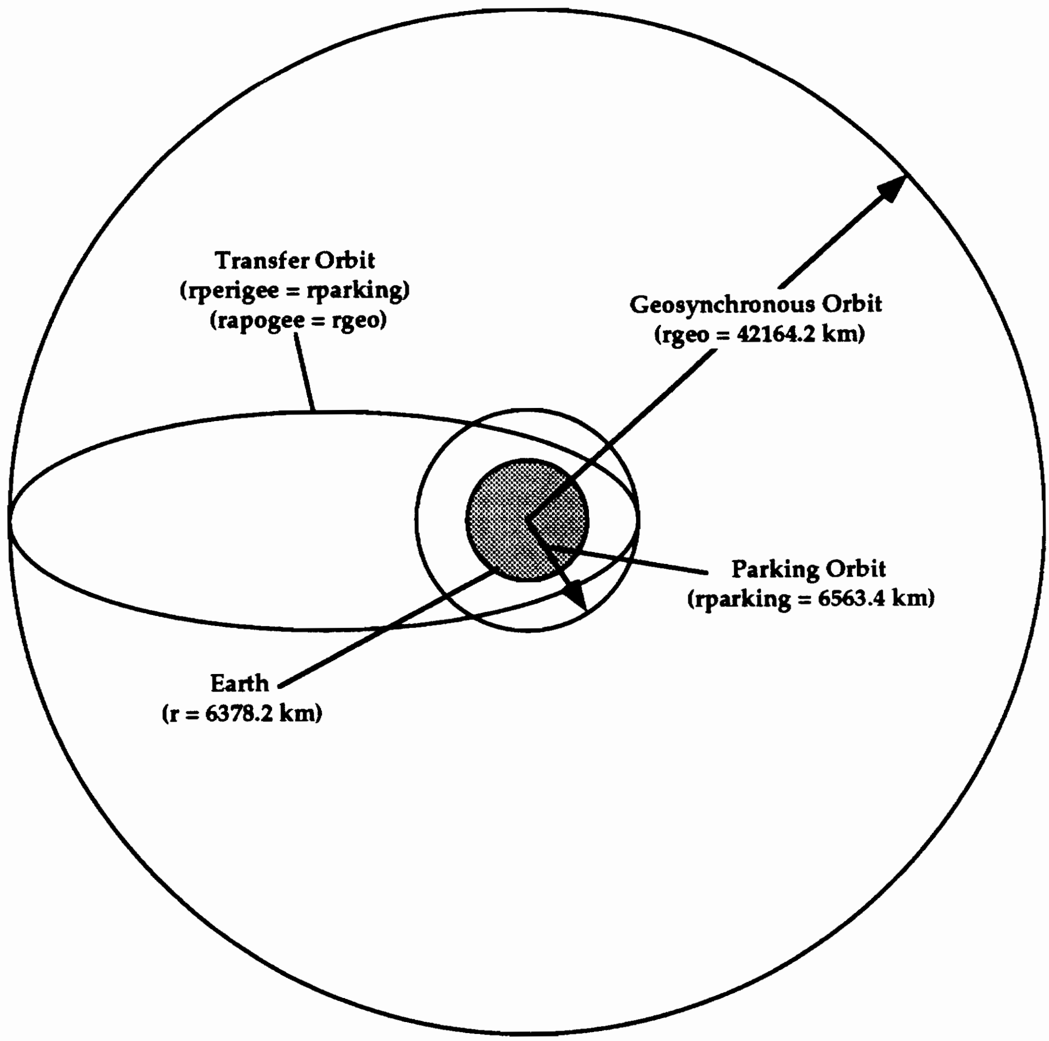


Figure 3.1 Three Main Orbits for Launch of Geosynchronous Spacecraft

The parking orbit is a circular orbit with radius = $r_{parking}$ = 185.2 + 6378.2 = 6563.4 km and is the orbit into which the Space Shuttle delivers spacecraft. The velocity of a spacecraft in parking orbit is

$$V_{parking} = (\mu_e / a_{parking})^{1/2} \tag{3-15}$$

$$a_{parking} = \text{Semimajor axis of parking orbit} = r_{parking} = 6{,}563.4 \text{ km} \tag{3-16}$$

$$V_{parking} = (398{,}601.2 \ / \ 6{,}563.4)^{1/2} = 7.79 \text{ km/s.} \tag{3-17}$$

The period of an orbit is important since orbital transfers are usually performed at either the apogee or perigee, and knowledge of the period of an orbit helps to predict apogee/perigee crossings. The period of the parking orbit is

$$\tau_{parking} = 2\pi a_{parking}{}^{3/2} / \mu_e{}^{1/2} \tag{3-18}$$

$$\tau_{parking} = \text{Period of the parking orbit} \tag{3-19}$$

$$a_{parking} = \text{Semimajor axis of parking orbit} = r_{parking} = 6{,}563.4 \text{ km} \tag{3-20}$$

$$\tau_{parking} = 2\pi(6{,}563.4)^{3/2} / 398{,}601.2^{1/2} = 5{,}291.81 \text{ s} = 1.47 \text{ hr.} \tag{3-21}$$

The transfer orbit intersects the parking orbit at its perigee ($r_{transfer@perigee}$ = $r_{parking}$), and the geosynchronous orbit at its apogee ($r_{transfer@apogee}$ = r_{geo}). Since the transfer orbit is elliptical, its velocity is not constant along the orbit; therefore, the velocity is needed at two points in the orbit, perigee (fastest) and apogee (slowest). The velocity of a spacecraft in transfer orbit at perigee is

$$V_{transfer@perigee} = [\mu_e(2/r_{transfer@perigee} - 1/a_{transfer})]^{1/2} \tag{3-22}$$

$$V_{transfer@perigee} = \text{Velocity of spacecraft in transfer orbit perigee} \tag{3-23}$$

$$r_{transfer@perigee} = \text{Radius of transfer orbit perigee} = 6{,}563.4 \text{ km} \tag{3-24}$$

$$a_{transfer} = \text{Semimajor axis of the transfer orbit} \tag{3-25}$$

$$a_{transfer} = (r_{parking} + r_{geo})/2 = (6{,}563.4 + 42{,}164.2)/2 \tag{3-26}$$

$$a_{transfer} = 24{,}363.8 \text{ km} \tag{3-27}$$

$$V_{transfer@perigee} = [398{,}601.2(2/6563.4 - 1/24{,}363.8)]^{1/2} = 10.25 \text{ km/s.} \tag{3-28}$$

The velocity of the spacecraft in transfer orbit at apogee is

$$V_{transfer@apogee} = [\mu_e(2/r_{transfer@apogee} - 1/a_{transfer})]^{1/2} \tag{3-29}$$

$$V_{transfer@apogee} = \text{Velocity of spacecraft in transfer orbit apogee} \tag{3-30}$$

$$r_{transfer@apogee} = \text{Radius of transfer orbit at apogee} = r_{geo} \tag{3-31}$$

$$r_{transfer@apogee} = 42{,}164.2 \text{ km} \tag{3-32}$$

$$V_{transfer@apogee} = [398{,}601.2(2/42{,}164.2 - 1/24{,}363.8)]^{1/2} = 1.60 \text{ km/s.} \tag{3-33}$$

The period of the transfer orbit is

$$\tau_{transfer} = 2\pi a_{transfer}{}^{3/2}/\mu_e{}^{1/2} \quad (3\text{-}34)$$

$$\tau_{transfer} = \text{Period of the transfer orbit} \quad (3\text{-}35)$$

$$a_{transfer} = \text{Semimajor axis of the transfer orbit} = 24{,}363.8 \text{ km} \quad (3\text{-}36)$$

$$\tau_{transfer} = 2\pi(24{,}363.8)^{3/2}/398{,}601.2^{1/2} = 37{,}846.7 \text{ s} = 10.5 \text{ hr.} \quad (3\text{-}37)$$

Table 3.1 summarizes the orbital velocities and periods for the various orbits which are used to achieve geosynchronous orbit insertion. The second and third columns denote whether or not the particular orbit in question is used in a typical Space Shuttle or rocket launch to achieve geosynchronous orbit (GEO). An *X* in the column denotes an affirmative response.

Table 3.1 Orbital Parameters for Various Orbits

Orbit	Shuttle	Rocket	$V_{perigee}$	V_{apogee}	τ	a
Parking	X		7.79 km/s	7.79 km/s	1.47 hr	6,563.4 km
Transfer	X	X	10.25 km/s	1.60 km/s	10.51 hr	24,363.8 km
GEO	X	X	3.08 km/s	3.08 km/s	23.93 hr	42,164.2 km

Now that the velocities and radii for the various geosynchronous launch orbits are calculated, there is just the matter of removing inclination from the orbit. As stated earlier, a geosynchronous orbit must lie in the equatorial plane. If the launch is made in the eastward direction to take advantage of the velocity due to the spin of the earth, then the initial orbit will have the same inclination as the latitude of the launch site. It is necessary to remove this inclination. This maneuver is usually done in conjunction with the apogee motor burn performed when going from transfer orbit to geosynchronous orbit. Vectorally, the physical situation is shown in Figure 3.2.

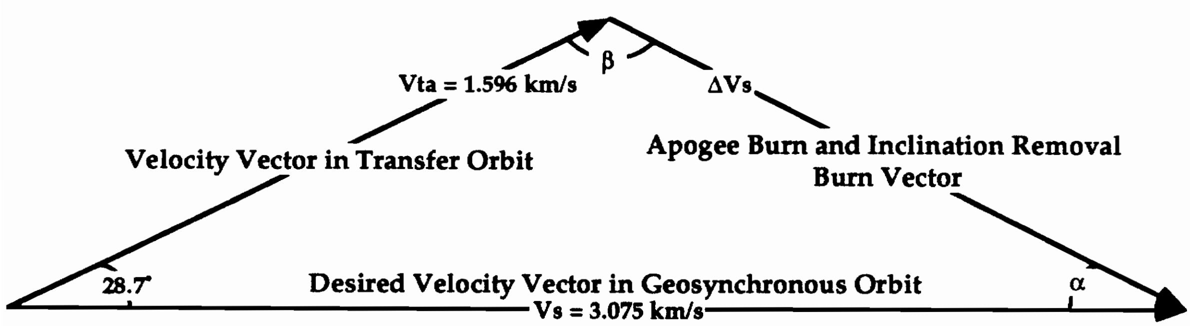


Figure 3.2 Geometry of Apogee Burn and Inclination Removal

The long vector on the bottom of the triangle represents the desired geosynchronous velocity. The vector pointing upward at an angle γ is the velocity of the spacecraft in transfer orbit at apogee. The angle γ is equal to the latitude of the launch site provided the launch is made in the eastward direction. The resolvent vector provides the magnitude and direction of a burn that, once accomplished, will leave the spacecraft in a geosynchronous orbit.

The resolvent vector is calculated with the use of the Law of Cosines

$$\Delta V_s^2 = V_{transfer@apogee}^2 + V_{geo}^2 - 2V_{transfer@apogee}V_{geo}\cos\gamma \quad (3\text{-}38)$$

ΔV_s = Velocity change required for geosynchronous orbit insertion and inclination removal from transfer orbit (3-39)

γ = Latitude of launch site = 28.5° for Kennedy Space Center (3-40)

$$\Delta V_s = [1.60^2 + 3.08^2 - 2(1.60)(3.08)\cos 28.5°]^{1/2} = 1.84 \text{ km/s.} \quad (3\text{-}41)$$

Likewise, the direction of the burn is calculated easily using the Law of Sines

$$\sin\alpha / V_{transfer@apogee} = \sin\gamma / \Delta V_s \quad (3\text{-}42)$$

α = Angle between the equatorial plane and the burn direction (3-43)

$$\alpha = \sin^{-1}[\sin\gamma * V_{transfer@apogee} / \Delta V_s] \quad (3\text{-}44)$$

$$\alpha = \text{sin-1}[\sin 28.5° * 1.60/1.84] = 24.51° \quad (3\text{-}45)$$

$$\beta = 180° - (24.51° + 28.5°) = 126.99°. \quad (3\text{-}46)$$

The above velocities and altitudes (radii) constitute requirements which must be satisfied by the launch vehicle. The launch vehicle must be able to provide the proper amount of thrust at the desired altitude for the spacecraft to achieve its desired orbit. In addition to these requirements, the launch vehicle must also protect the spacecraft during its escape from the atmosphere. In order for launch vehicles to achieve their desired injection points, they must be launched at particular times in the day. These launch times are usually referred to as launch windows and fall twice per day. Launching during an open launch window ensures that the vehicle is injected into the proper initial orbit, the spacecraft has enough sunlight to generate some power during transfer orbit, and is not overheated due to sunlight shining on an undesirable surface for an extended period of time.

The launch vehicle requirements presented above represent some of the major requirements which must be satisfied, along with cost and reliability, by the launch vehicle in order to ensure a successful mission.

Launch vehicle selection is heavily dependent upon spacecraft mass, size, and shape. In fact, launch vehicles usually dictate allowable sizes and shapes. An estimate must be made regarding the expected size, shape, and mass of the spacecraft. Launch vehicles do not come in all sizes and shapes to satisfy any particular spacecraft configuration. On the contrary, as much as spacecraft requirements drive launch vehicle performance, launch vehicles place constraints on spacecraft dimensions and mass. One of the reasons that launch vehicle selection is performed so early in the engineering process is because of the constraints it places a spacecraft.

In the case of communications satellites, mass estimation is performed on the basis of other spacecraft that have been previously built and extrapolations are performed where identical capabilities do not exist. Assumptions are made about desired launch vehicle and predicted beginning-of-life (BOL) mass, but since spacecraft design is essentially an iterative process and this is the first iteration, any error made here is not likely to result in a catastrophe. For the sake of preliminary design, assume that the launch vehicle is the Space Shuttle and the BOL mass of the spacecraft lies between Intelsat V ($\approx$ 1000 kg) and Intelsat VI ($\approx$1600 kg) at 1533 kg. The Space Shuttle is chosen as the launch vehicle for political reasons and to keep the economy in the U.S. viable. A perigee motor will be required. Also assume that the spacecraft is three-axis stabilized and utilizes a bi-propellant (N_2O_4-MMH) unified propulsion system for apogee injection, station keeping, and attitude control.

The following few equations estimate spacecraft masses empirically, and do not result in high accuracy estimates as a result, but the results are good enough for launch vehicle selection. The first order of business is to predict propellant masses based on BOL mass and the specific impulse (I, ≈efficiency) of the propellant. Propellant mass is calculated with the following equation

$$M_p = M_i(1 - e^{-\Delta V/Ig}) \quad (3\text{-}47)$$

M_p = Mass of the propellant [kg] (3-48)

M_i = Initial mass of the spacecraft before firing motor [kg] (3-49)

ΔV = Velocity change as result of firing motor [m/s] (3-50)

I = Specific impulse of the propellant [s] (3-51)

g = Gravitational acceleration [m/s^2]. (3-51)

Assuming the BOL (after apogee injection) mass to be 1,533 kg and a required velocity change of 1.84 km/s (as calculated above for apogee injection and inclination removal), the required propellant mass for apogee injection (bi-propellant I = 300 s for apogee injection) is

$$M_{p(\text{apogee injection})} = (M_{p(\text{apogee injection})} + M_i)(1 - e^{-\Delta V/Ig} \quad (3\text{-}52)$$

$$M_{p(\text{apogee injection})} = M_i(e^{\Delta V/Ig}\text{-}1) \quad (3\text{-}53)$$

$$M_{p(\text{apogee injection})} = 1{,}533(e^{1{,}840/(300*9.81)}-1) = 1{,}331 \text{ kg}. \qquad (3\text{-}54)$$

Inclination, or north-south station keeping propellant mass can be calculated based on an allowable inclination drift of 0.1°. Roughly, this results in a 10.7 m/s maneuver every 86.14 days or so. Over a ten-year period the total velocity change required is 10 years * 365 days/year * 1 maneuver/86.14 days * 10.7 m/(s maneuver) = 453.4 m/s. The propellant mass (bi-propellant I = 285 s for station keeping) necessary to effect this change in velocity is

$$M_{p(\text{N-S station keeping})} = 1{,}533(1 - e^{-453.4/(285 * 9.81)}) = 229 \text{ kg}. \qquad (3\text{-}55)$$

Longitudinal, or east-west station keeping propellant mass can be roughly calculated based on an allowable longitude drift of ±0.1°. This results in a 0.15 m/s maneuver every 31 days or so. Over a ten-year period the total velocity change required is 10 years * 365 days/year * 1 maneuver/31 days * 0.15 m/(s maneuver) = 17.7 m/s. The propellant mass necessary to effect this change in velocity is

$$M_{p(\text{E-W station keeping})} = 1{,}533(1 - e^{-17.7/(285*9.81)}) = 10 \text{ kg}. \qquad (3\text{-}56)$$

Additional propellant is assumed for orbital maneuvering and attitude control to the quantity of 30 kg.

A rough estimate of the dry mass of the unified bi-propellant propulsion system is

$$M_{UB} = C_{UB}M_{PR} \tag{3-57}$$

M_{UB} = Dry mass of the unified propulsion system [kg] (3-58)

C_{UB} = 0.084 for three-axis stabilization (3-59)

M_{PR} = Total mass of propellant = 1,331 + 229 + 10 + 30 = 1,600 kg (3-60)

$$M_{UB} = 0.084 * 1{,}600 = 134 \text{ kg.} \tag{3-61}$$

The mass of the electric power subsystem can be estimated by referring to previously built spacecraft and extrapolating specific masses for an assumed power load of 2,300 W. Since communications spacecraft usually employ Ni-H_2 batteries, they are assumed in the estimation. For a three-axis stabilized system the following quantities are estimated

$$M_{solar\ array} = 43 \text{ g/W} * \text{Power} = 43 * 2{,}300 = 99 \text{ kg,} \tag{3-62}$$

$$M_{charge\ array} = 7.2 \text{ g/W} * \text{Power} = 7.2 * 2{,}300 = 17 \text{ kg,} \tag{3-63}$$

$$M_{shunt} = 7.5\ g/W * Power = 7.5 * 2{,}300 = 17\ kg, \tag{3-64}$$

$$M_{charge\ control} = 1.5\ g/W * Power = 1.5 * 2{,}300 = 3\ kg \tag{3-65}$$

$$M_{battery} = 52\ g/W * Power = 52 * 2{,}300 = 120\ kg \tag{3-66}$$

$$M_{discharge\ control} = 12.5\ g/W * Power = 12.5 * 2{,}300 = 29\ kg \tag{3-67}$$

$$M_{electric\ power} = 99 + 17 + 17 + 3 + 120 + 29 = 285\ kg. \tag{3-68}$$

The mass of the structure of the spacecraft is very difficult to accurately estimate, especially because designers themselves disagree on what is considered to be part of the structure and what is part of the other subsystems. A rough estimate can be calculated with

$$M_{ST} = C_{ST} M_{SP} \tag{3-69}$$

$$M_{ST} = \text{Structure mass [kg]} \tag{3-70}$$

$$C_{ST} = 0.087 \text{ for three-axis stabilization} \tag{3-71}$$

$$M_{SP} = \text{Spacecraft separation mass} = 1{,}533 + 1{,}331 = 2{,}864\ kg \tag{3-72}$$

$$M_{ST} = 0.087 * 2{,}662 = 249\ kg. \tag{3-73}$$

The mass of the thermal control subsystem is also difficult to estimate accurately without prior knowledge of the temperature limits required on-board. A rough estimate is obtained with the use of

$$M_T = C_{MT}M_{SC} \quad (3\text{-}74)$$

$$M_T = \text{Mass of the thermal control subsystem [kg]} \quad (3\text{-}75)$$

$$C_{MT} = 0.032 \text{ for three-axis stabilization} \quad (3\text{-}76)$$

$$M_{SC} = \text{Mass of spacecraft} = 1{,}362 - 1{,}129 + 1{,}300 = 1{,}533 \text{ kg} \quad (3\text{-}77)$$

$$M_T = 0.032 * 1{,}533 = 49 \text{ kg.} \quad (3\text{-}78)$$

The mass of the attitude control subsystem is a function of the size and mass of the spacecraft, the types of attitude control units, the types of attitude sensors, and the redundancy of all of the units. For a three-axis stabilized spacecraft, the attitude control system mass is given by

$$M_{AC} = 65 + 0.022(M_{SC} - 700) = 65 + 0.022(1{,}533 - 700) = 83 \text{ kg.} \quad (3\text{-}79)$$

The mass of the telemetry, tracking, and command subsystem is assumed to be similar to, but more complex than the Intelsat V system. The Intelsat V system has a mass of 28 kg, the GEOCOMM is estimated to have a mass of 1.3 * 28 = 36 kg.

Also the electrical and mechanical integration masses are extrapolated from the Intelsat V. The integration equipment has an estimated mass of

$$M_E = 0.039 M_{SC} = 0.039 * 1{,}533 = 60 \text{ kg} \qquad (3\text{-}80)$$

$$M_M = 0.014 M_{SC} = 0.014 * 1{,}533 = 21 \text{ kg}. \qquad (3\text{-}81)$$

The preliminary spacecraft masses are tabulated in Table 3.2. It is important to remember that the design of a spacecraft is an iterative process. These numbers are mere estimations at this stage in the process, but as the final design nears, these estimates are replaced by highly accurate values. The estimation of initial spacecraft mass allows for selection of an appropriate launch vehicle. This does not necessarily mean that once the launch vehicle is chosen it cannot be changed. On the contrary, upon selection of a launch vehicle, more accurate designs are forwarded and analyzed alongside the selected launch vehicle. If the launch vehicle is not adequate for the design, a new launch vehicle is sought.

Table 3.2 Preliminary Spacecraft Mass Summary

Subsystem	**Mass [kg]**
Communications/Antennas	370
Electric Power	285
Structure	249
Thermal	49
Propulsion	134
Telemetry, Tracking, and Command	36
Attitude Control	83
Electric Integration	60
Mechanical Integration	21
Mass Margin	130
Dry Spacecraft Mass	1,264
BOL Propellant/Pressurant	269
BOL Spacecraft Mass	1,533
Apogee Motor Propellant/Pressurant	1,331
Spacecraft Mass at Separation	2,864

3.2 Launch Vehicle Equipment

At this early stage in the systems engineering of a spacecraft it is necessary to choose an appropriate launch vehicle. So much of the final design depends on the launch vehicle that without choosing one now would be like trying to design the spacecraft without knowing 25% of its requirements. This section describes the various launch vehicles available on the commercial market.

The United States relies very heavily on the Space Shuttle, while other countries rely heavily on the European Space Agency's (ESA) Ariane launch vehicle. Table 3.3 shows some of the more common launch vehicles with spacecraft mass and size restrictions. The latitude of the launch site is given since this is the inclination of the initial orbit the launch vehicle gives to the spacecraft. A smaller latitude results in a fuel savings when performing inclination removal at apogee injection.

Table 3.3 Launch Vehicles and Parameters

Name	Latitude [°]	Length [m]	Diameter [m]	Mass [kg]
Atlas-Centaur	27.1		3.0	2,100
Delta	28.7		2.4	1,312
Titan	28.5	≈10	≈3.65	
Space Shuttle	28.5	18.3	4.6	29,484
Ariane I - V	8.5	≈5	≈4	>4,200
Japanese N-Vehicles	30			
Russian	>48			

ESCAPING THE ATMOSPHERE

Rockets are unmanned, expendable, vehicles which place spacecraft into transfer orbit. A rocket's job is done once it has delivered the spacecraft to the transfer orbit. Rocket launches are reasonable in price, and prices should continue to decrease as more countries develop their own launch vehicles and vie for position on the world market.

The Space Shuttle is the only manned launch vehicle (with the exception of the untested Soviet shuttle) and it has distinct advantages over unmanned vehicles since it can be used to deploy, retrieve, and repair spacecraft. If a spacecraft is not deployed properly, the crew from the Shuttle can bring it back for troubleshooting--a luxury not afforded with rocket launches. The Space Shuttle deploys spacecraft into a circular parking orbit which means that the spacecraft must undergo two more orbital maneuvers before it is in a geosynchronous orbit. Spacecraft deployed by the Shuttle must be designed with both a perigee motor and an apogee motor.

PERIGEE MOTORS

Perigee motors provide the force to a spacecraft to change its orbit from a circular parking orbit to an elliptical transfer orbit. Fortunately these motors are available commercially and can be easily integrated into spacecraft designs.

Some of the Shuttle payload capacity is unfortunately taken up by the need for a perigee motor.

APOGEE MOTORS

Apogee motors provide the thrust necessary to progress from transfer orbit into the geosynchronous orbit and eliminate inclination. These too are available commercially and may be designed around. Apogee motors that are integrated into the spacecraft propulsion system afford much flexibility when it comes to orbital maneuvering. Maneuvers may be done slowly and accurately, and in the end any propellant saved from the apogee injection maneuver can be used to extend mission life. All spacecraft whether they are rocket- or Shuttle-launched need apogee motors.

3.3 GEOCOMM Launch Vehicle Selection

A generic flow describing the selection process of a launch vehicle is shown in Figure 3.3. For GEOCOMM, the Space Shuttle is the launch vehicle of choice. The Space Shuttle launches from the United States, it has the largest cargo bay of any launch vehicle, it has the greatest mass capacity of any launch vehicle, it is also a good deal financially, and if something goes wrong during deployment, the crew can bring GEOCOMM back to earth (reducing insurance costs). On the other hand, the Space Shuttle launches at a latitude of 28.5°

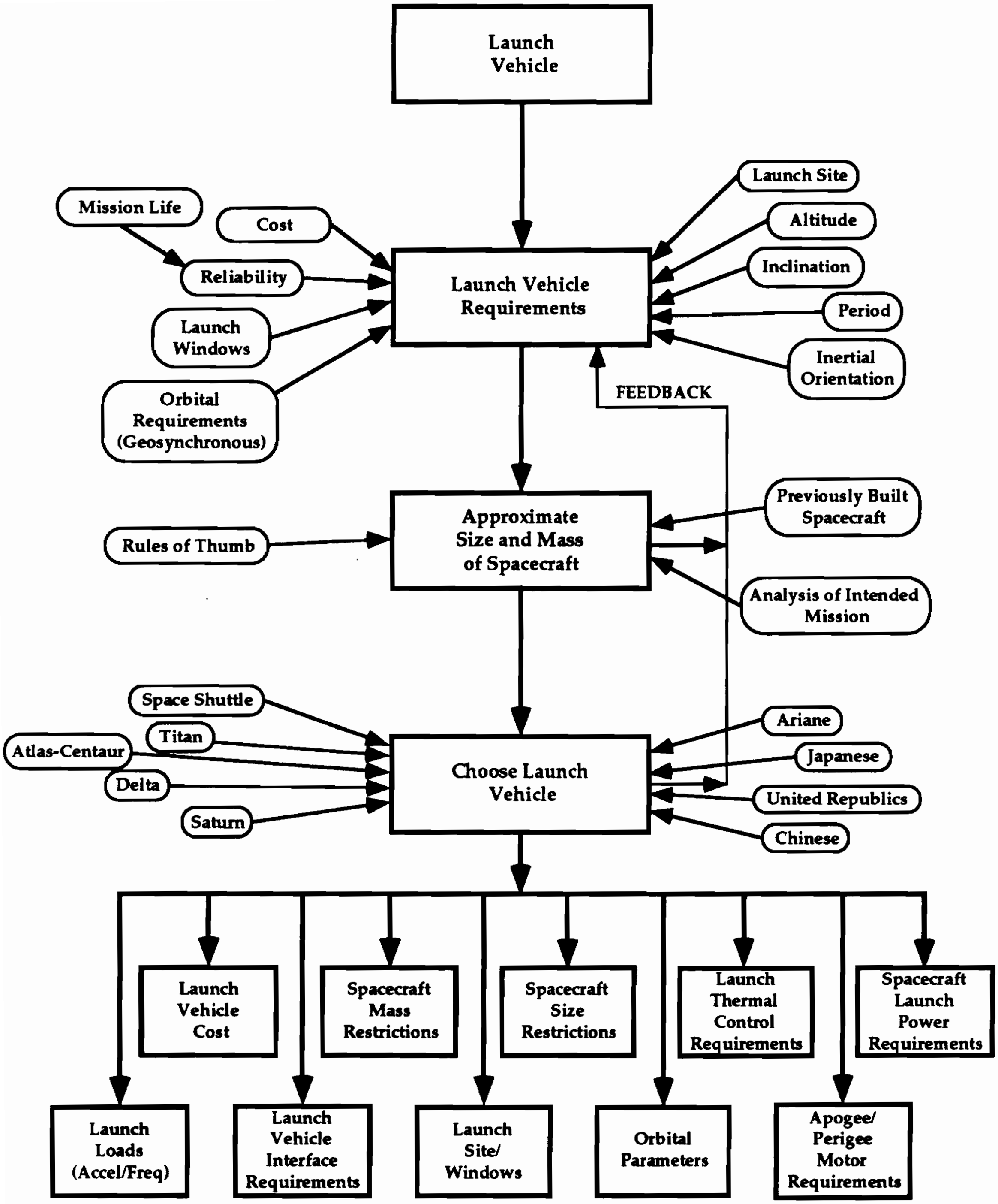


Figure 3.3 Launch Vehicle Evaluation and Selection Flow

requiring more propellant for inclination removal, it requires that GEOCOMM have a perigee motor, and it does not launch spacecraft frequently. In most communications satellites, flexibility is usually designed into the spacecraft so that it may be launched from a variety of launch vehicles. For ease of structural design, this is not opted for with GEOCOMM. The perigee motor chosen is the McDonnell-Douglas Payload Assist Module Delta version (PAM-D) which is an off-the-shelf motor that acts just like the upper stages of the Delta rocket.

4.0 Propulsion Subsystem

4.1 Propulsion Requirements

The parameters which must be satisfied by a spacecraft to ensure the proper orbit are very uncompromising. If any parameter strays even slightly, the spacecraft does not maintain its orbit. In addition, there are many forces that contribute to perturbing the spacecraft from its desired orbit such as solar pressure, solar wind, the presence of other masses, the oblateness of the earth, space-borne particles, etc. It is obvious that the spacecraft must have a system with which it can correct its orbit in the absence of a launch vehicle. Periodically, orbital corrections are performed by the propulsion subsystem to correct such things as latitude errors, longitude errors, attitude errors, and to perform station repositioning. The correction of latitude and longitude errors is referred to as station keeping. Much of the same rocket-technology used in launch vehicles is employed for the thrusters, but on a much smaller scale. Requirements levied on the propulsion subsystem include such quantities as: how often orbital maneuvers must be performed to maintain desired orbital position accuracy, the total number of maneuvers expected over the lifetime of the spacecraft, and the total ΔV or propellant requirements.

NORTH-SOUTH STATION KEEPING

North-south station keeping is performed when the inclination (or latitude) of an spacecraft's orbit is not zero. Table 4.1 provides inclination drift rates which can be used to determine the average inclination drift rate per year and the average time interval between north-south station keeping corrective maneuvers.

Table 4.1 Inclination Drift Rates (See Reference #1)

Date *January 1*	Ω *moon* (*Deg.*)	i_t (*Deg.*)	Ω_t (*Deg.*)	$\left.\frac{di}{dt}\right\vert$ moon (*Deg./Yr.*)	$\left.\frac{di}{dt}\right\vert$ *Total* (*Deg./Yr.*)
1979	171.332	18.375	2.459	0.480	0.749
1980	152.004	19.049	7.417	0.491	0.760
1981	132.623	20.300	10.975	0.513	0.782
1982	113.295	21.902	12.769	0.542	0.811
1983	93.966	23.627	12.910	0.574	0.843
1984	74.638	25.279	11.694	0.607	0.876
1985	55.257	26.703	9.447	0.636	0.905
1986	35.929	27.776	6.489	0.657	0.926
1987	16.601	28.422	3.089	0.671	0.940
1988	357.273	28.595	−0.511	0.674	0.943
1989	337.892	28.284	−4.088	0.668	0.937
1990	318.563	27.511	−7.388	0.652	0.921
1991	299.235	26.330	−10.171	0.628	0.897
1992	279.907	24.830	−12.155	0.598	0.867
1993	260.526	23.135	−13.023	0.565	0.834
1994	241.198	21.423	−12.436	0.533	0.802
1995	221.870	19.898	−10.138	0.506	0.775
1996	202.542	18.793	−6.132	0.487	0.756
1997	183.161	18.311	−0.903	0.479	0.748
1998	163.833	18.557	4.504	0.483	0.752
1999	144.505	19.476	8.993	0.498	0.767
2000	125.177	20.888	11.875	0.523	0.792
2001	105.796	22.568	13.006	0.554	0.823
2002	86.468	24.287	12.581	0.587	0.856
2003	67.139	25.865	10.929	0.619	0.888

Assuming that GEOCOMM is launched January 1, 1993 and is expected to survive for at least ten years, the average inclination drift rate per year from January 1, 1993 to January 1, 2003 is (from Table 4.1)

$$i_{\text{drift rate}} = (\Sigma i_{\text{yearly drift rate}}) / \text{ Lifetime of Spacecraft} \tag{4-1}$$

$$i_{\text{drift rate}} = (0.834 + 0.802 + 0.775 + 0.756 + 0.748 + 0.752 + 0.767 + 0.792 + 0.823 + 0.856)/10 = 0.7905^\circ/\text{year}. \tag{4-2}$$

The calculated number of orbital maneuvers has not been optimized by any means. In practice orbital maneuvers are usually performed when the spacecraft has drifted only half way to its extreme tolerance. It is then adjusted back to half way to the other extreme and allowed to drift through its optimum point. It is also important to know how often station keeping maneuvers are to be performed when purchasing thrusters capable of supporting many on-off cycles. The frequency with which north-south station keeping maneuvers are performed can be calculated only when the inclination tolerances are known. Communications spacecraft have low tolerances and their inclinations are usually kept to within $\pm 0.1^\circ$. At this tolerance, the average time interval between north-south station keeping maneuvers is

$$T_{n\text{-}s\ station\ keeping} = 2i * 365.25 / i_{drift\ rate} \quad (4\text{-}3)$$

$$i = \text{Inclination accuracy desired of spacecraft} = 0.1^\circ \quad (4\text{-}4)$$

$$T_{n\text{-}s\ station\ keeping} = 2 * 0.1 * 365.25 / 0.7905 = 92.41 \text{ days.} \quad (4\text{-}5)$$

Every 92.41 days GEOCOMM must perform a north-south station keeping maneuver to ensure that its requirements for inclination are satisfied. The total number of maneuvers required for a ten year lifetime is

$$N_{n\text{-}s\ station\ keeping} = \text{Lifetime of spacecraft} / T_{n\text{-}s\ station\ keeping} \quad (4\text{-}6)$$

$$N_{n\text{-}s\ station\ keeping} = 10*365.5/92.41 = 39.5 \approx 40 \text{ maneuvers.} \quad (4\text{-}7)$$

Additionally, the required change in velocity (ΔV) necessary to ensure orbital inclination correctness throughout the ten year lifetime of the spacecraft is (equation is empirically derived--see reference #1)

$$\Delta V_{n\text{-}s\ station\ keeping} = N_{n\text{-}s\ station\ keeping} * 6.148 \sin i \quad (4\text{-}8)$$

$$\Delta V_{n\text{-}s\ station\ keeping} = 40 * 6.148 \sin(0.1) = 0.429 \text{ km/s.} \quad (4\text{-}9)$$

The requirements uncovered by this simple analysis are that the spacecraft needs thrusters capable of imparting a velocity of 0.0107 km/s in the north or south direction approximately every 92.41 days for a total of ≈40 maneuvers and a total change in velocity of 0.429 km/s.

EAST-WEST STATION KEEPING

East-west station keeping is performed when the longitude of the spacecraft is no longer the desired longitude. The expected longitude of GEOCOMM is 263°E. Analogous to the case for north-south station keeping, orbital perturbations tend to move the spacecraft from its desired longitude. Once again, these figures have not been optimized in any way. Several iterations into the design cycle a relatively exact and optimal number of East-West station keeping maneuvers will be calculated, but for a preliminary design the figures presented below represent a good estimate. The longitudinal drift acceleration is given by

$$\lambda_{accel} = -0.00168 \sin 2(\lambda - \lambda_s) \tag{4-10}$$

$$\lambda_{accel} = \text{Longitudinal drift acceleration } [^\circ/\text{day}^2] \tag{4-11}$$

$$\lambda = \text{Desired longitude of the spacecraft} = 263^\circ\text{E for GEOCOMM} \tag{4-12}$$

$$\lambda_s = \text{Stable longitude} = 75^\circ\text{E or } 255^\circ\text{E} \tag{4-13}$$

$$\lambda_{accel} = -0.00168 \sin 2(263 - 255) = -0.000463\ ^\circ/\text{day}^2. \tag{4-14}$$

The tolerance for longitude station keeping is to be the same as for north-south station keeping (for GEOCOMM ±0.1°). Based on this requirement for station keeping, the time between east-west station keeping maneuvers is

$$T_{\text{e-w station keeping}} = 4 * (\Delta l / | l_{accel} |)^{1/2} \tag{4-15}$$

$$T_{\text{e-w station keeping}} = 4 * (0.1/0.000463)^{1/2} = 58.79 \text{ days.} \tag{4-16}$$

East-west station keeping maneuvers must be performed approximately every 58.79 days for a grand total in a ten year period of

$$N_{\text{e-w station keeping}} = \text{Lifetime of spacecraft} \; / \; T_{\text{e-w station keeping}} \tag{4-17}$$

$$N_{\text{e-w station keeping}} = 10 * 365.25 \; / \; 58.79 = 62.13 \approx 62 \text{ maneuvers.} \tag{4-18}$$

The total velocity change which must be imparted upon the spacecraft per year to insure east-west station keeping tolerances are satisfied is

$$\Delta V_{\text{e-w station keeping}} = 1.74 \sin 2(l - l_s) \text{ [m/s yr]} \tag{4-19}$$

$$\Delta V_{\text{e-w station keeping}} = 1.74 \sin 2(263 - 255) = 0.480 \text{ m/s yr.} \tag{4-20}$$

For the ten year lifetime of the satellite, the total velocity that must be imparted by the east-west station keeping thrusters is

$$\Delta V_{\text{e-w total}} = \text{Lifetime of spacecraft} * \Delta V_{\text{e-w station keeping}} \tag{4-21}$$

$$\Delta V_{\text{e-w total}} = 10 * 0.480 = 4.8 \text{ m/s.} \tag{4-22}$$

ATTITUDE CONTROL

The propulsion system is sometimes used for attitude control, but since attitude is so tightly constrained by requirements, harsh movements as those experienced when a thruster is fired usually excludes many thruster attitude control systems. Attitude control does employ thrusters and is covered in greater detail in the attitude control section. Thrusters are sometimes used to spin up the momentum devices that are used for attitude control and therefore propellant is necessary, however minimal, to ensure that the momentum devices remain spinning as necessary.

PERIGEE MOTOR

Perigee motors are needed only by spacecraft launched by the Space Shuttle. Perigee motors are used to inject spacecraft from parking orbit into transfer orbit. From Table 3.1 it can be calculated that a velocity change of $V_{\text{transfer@perigee}} - V_{\text{parking@perigee}} = 10.25 - 7.79 = 2.46$ km/s is necessary from the perigee motor for successful injection into transfer orbit. The perigee motor chosen for GEOCOMM in Section 2.1 is not an integral part of the spacecraft, but rather an attached device that is jettisoned once its duty is performed. Therefore, there are no perigee motor requirements for the propulsion system.

APOGEE MOTOR

Apogee motors are needed by all spacecraft, whether launched by rocket or Space Shuttle. Apogee motors are used to inject spacecraft from transfer orbit into geosynchronous orbit and remove any inclination. From Section 2.2.1 a velocity change of $\Delta V = 1.84$ km/s is necessary from the apogee motor for successful injection into geosynchronous orbit and removal of inclination from a Shuttle Launch. GEOCOMM is Shuttle launched, and therefore does require an apogee motor. GEOCOMM's apogee motor is integrated into its propulsion subsystem.

4.2 Propulsion Subsystem Equipment

THRUSTERS

On-orbit spacecraft propulsion systems can be divided into two classes: chemical and electric. Propulsion systems provide the force to perform orbital maneuvers once the launch vehicle has been separated. The quantity and type of thrusters necessary for a successful mission are determined from the requirements analyzed in Section 4.1. The thrusters must be reliable for at least the duration of the mission, they must be able to support the frequency of firing dictated by the station keeping requirements, they must deliver the proper magnitude of force to the spacecraft in the time specified, and they must be cost effective. Commonly used thrusters include: cold gas systems, monopropellant, bi-propellant, solid propellant, and electric.

CHEMICAL THRUSTERS

Almost exclusively, chemical thrusters are used on-board spacecraft. Chemical thrusters include: cold gas systems, monopropellant, bi-propellant, and solid propellant. The advantages of chemical thrusters lies in the fact that they provide a lot of power in a short time span, but as a result they use their propellant very inefficiently. Chemical thrusters are capable of providing high, medium, and low thrust levels. It is the ability to provide variable thrust that makes chemical thrusters so versatile when designing station keeping motors, attitude control motors, apogee injection motors, and sometimes perigee motors. Unfortunately, the lifetime of a spacecraft with chemical thrusters is often limited by the amount of on-board propellant carried into orbit at BOL. Chemical thrusters are reliable for use in high cycle ranges, but are usually paired in redundancy to ensure mission objectives are satisfied.

Cold gas systems utilize a gas, usually inert, under high pressure that is allowed to escape from the tank under controlled conditions through a nozzle. Thrust levels of 20 mN are achieved with such systems, but even these low levels are important when extremely high pointing accuracies are desired which are unachievable with other thrusters.

Monopropellant systems utilize the decomposition of hydrazine N_2H_4 for propellant. The propellant is sent through a nozzle to yield the desired power output. Specific impulses lie between 200-250 s which generate forces greater than or equal to 10 N. These systems are used extensively on communications spacecraft and have shown a high degree of reliability. Monopropellant systems may be used for high thrust applications such as apogee injection (I = 235 s), medium thrust applications such as station keeping (I = 220 s), and/or low thrust applications such as attitude control (I = 135 s).

Bi-propellant systems are similar to monopropellant systems except that they utilize a mixture of N_2O_4 and MMH resulting in higher specific impulses (I =

300, 285, and 175) and therefore higher propellant efficiency, but at the cost of more mass for the propulsion system. It turns out that the increased mass is more than offset by the increased propellant savings and additional mission life afforded as a result of using a bi-propellant system.

Solid propellant thrusters are usually used for perigee and apogee motors since they are only useful for one burn. Solid propellant is easy to deal with because the details of its use are usually straight forward. Propulsion subsystem design considerations are relaxed when using solid propellant, but at the expense of a loss of control. Once a solid propellant burn is initiated, it cannot be arrested--even if the spacecraft is pointing in an incorrect direction. Liquid propellant thrusters can be modulated correctly in short bursts to achieve desired results. Clearly liquid propellant systems are desirable when accuracy is a premium, but solid thrusters are still used and economical in today's market.

ELECTRIC THRUSTERS

Electric thrusters are new to the spacecraft industry and hold the promise of longer mission life. Electric thrusters eject very small atomic particles at very high rates of speed. The net effect is a large specific impulse, but a very low power level. Electric thrusters impact the electric power subsystem significantly in that they need electrical power for their propellant. For now, electric thrusters are only used experimentally.

4.3 GEOCOMM Propulsion Subsystem Design

A generic flow depicting the design of a propulsion subsystem is shown in Figure 4.1. GEOCOMM is to employ bi-propellant thrusters for increased efficiency, reliability, and controllability, and so that the apogee motor may be incorporated into the system. The thrusters are positioned just as Intelsat V's thrusters are configured. Thruster purposes are given in Table 4.2. Blank positions are reserved for attitude control thrusters. The quantity of propellant that must be carried into space for all requirements is of the utmost importance for mass and lifetime considerations. The quantity of propellant that must be carried on-board to satisfy all mission criteria can be calculated. Propellant for the propulsion subsystem falls into three categories: final orbit acquisition (apogee motor), station keeping, and attitude control. The propellant mass for each requirement is calculated below using the assumption that the beginning of life mass of the spacecraft is 1533 kg.

APOGEE MOTOR

Based on the velocity requirement for the apogee motor of 1.84 km/s and a beginning of life mass of 1,533 kg, the bi-propellant fuel requirement for

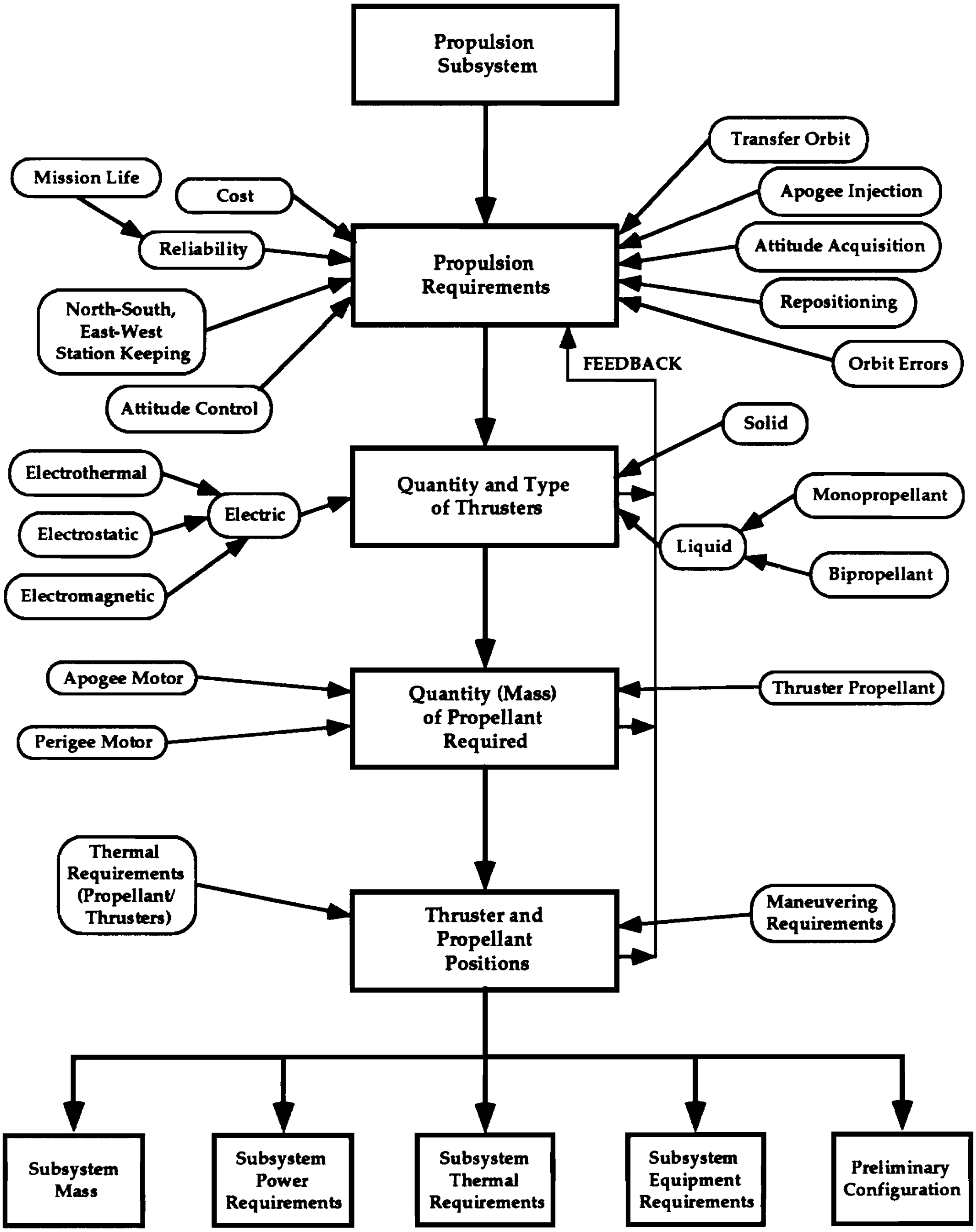


Figure 4.1 Propulsion Subsystem Design Flow

apogee injection is 1,331 kg as calculated in Section 3. The numbers in the original estimate were for a bi-propellant system for a spacecraft of the same mass, and are applicable here.

STATION KEEPING

The total station keeping velocity changes are calculated above to be $\Delta V_{n\text{-}s\ station\ keeping} + \Delta V_{e\text{-}w\ station\ keeping} = 0.429 + 0.0048 = 433.8$ km/s. The total mass of bi-propellant fuel necessary for station keeping is

$$M_{total\ station\ keeping} = M_i(1 - e^{-\Delta V/(I * 9.81)}) \quad (4\text{-}23)$$

$$M_i = \text{Initial mass of spacecraft} = 1{,}533 \text{ kg} \quad (4\text{-}24)$$

$$I = \text{Specific impulse of propellant} = 285 \text{ s for station keeping} \quad (4\text{-}25)$$

$$M_{total\ station\ keeping} = 1{,}533(1 - e^{-433.8/(285*9.81)}) = 220.3 \text{ kg.} \quad (4\text{-}26)$$

ATTITUDE CONTROL

Attitude control and margin propellant constitute an important part of the overall mass of propellant required. Attitude corrections are not preformed as a routine procedure like station keeping, but maneuvers are performed as necessary to maintain attitude requirements. Another function lumped under this category includes the propellant needed to spin the spacecraft up for orbit transfer thrusting and spin down for operations. Spinning bodies tend to wobble (nutation) so there are thrusters to control the nutation during transfer orbit. Propellant for attitude control and margin is assumed to be 40 kg. Note that this quantity is larger than the initially estimated value of 30 kg.

The total mass of bi-propellant required to ensure mission requirements are satisfied for ten years is

$$\text{Total Propellant} = \text{Apogee motor} + \text{Station Keeping} + \text{Attitude Control} \tag{4-27}$$

$$\text{Total Propellant} \approx 1331 + 220 + 40 \approx 1591 \text{ kg.} \tag{4-28}$$

The entire propulsion system is computer-controlled. In fact, the computer is shared among all subsystems, but is utilized mainly by the attitude control, propulsion, and TT&C subsystems. Additional information regarding the computer is given in the TT&C section. Propellant tanks and a transport system are also necessary. One very important configuration requirement is that the propellant takes must be placed in a position such that use of propellant does not appreciably change the center of mass of the spacecraft. This would be devastating to the attitude control system which relies heavily on accurate knowledge of the center of mass of the spacecraft.

Thruster positions are shown in Figure 4.2 for GEOCOMM. Table 4.2 lists the functions of each of the thrusters shown.

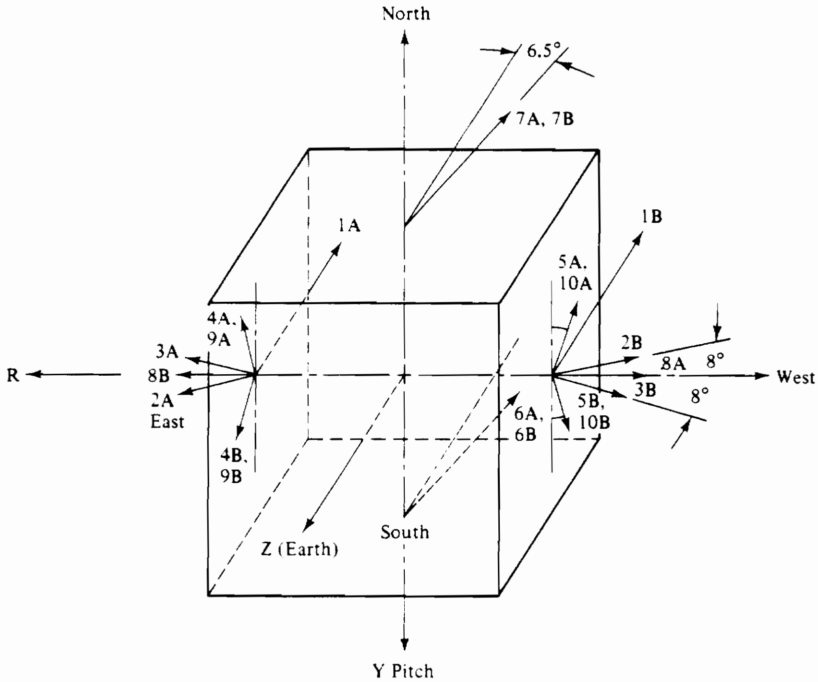


Figure 4.2 Thruster Positions for GEOCOMM (See Reference #1)

Table 4.2 GEOCOMM Propulsion Subsystem Thrusters (bi-propellant)

Thruster	Function
1A	
1B	
2A	West Station Keeping
2B	East Station Keeping
3A	West Station Keeping
3B	East Station Keeping
4A	Spin Up for Transfer Orbit, Redundant South Station Keeping
4B	Spin Down from Transfer Orbit, Redundant North Station Keeping
5A	Spin Down from Transfer Orbit, Redundant South Station Keeping
5B	Spin Up for Transfer Orbit, Redundant North Station Keeping
6A	
6B	
7A	
7B	
8A	Redundant East Station Keeping
8B	Redundant West Station Keeping
9A	South Station Keeping
9B	North Station Keeping
10A	South Station Keeping
10B	North Station Keeping

5.0 Attitude Control Subsystem

5.1 Attitude Control Requirements

Orbital mechanics deals with the motion of the center of gravity of a spacecraft about the baricenter of the spacecraft and a larger body, while attitude determination and control deals with the motion of the spacecraft about its own center or gravity. In more general terms, attitude determination and control is concerned with the direction in which a spacecraft is pointing with respect to a frame of reference (usually earth). The following three basic requirements must be satisfied by the attitude control and structural subsystems: the communications antennas must point towards earth (payload requirement), the solar arrays must point towards the sun (electric power requirement), and the blackbody radiators must point towards deep space (thermal control requirement). Communications service will be interrupted or degraded if antennas do not point properly. Modern satellite communications require pointing accuracies of about $\pm 0.1^\circ$ in all directions (N, S, E, W). Power generation is reduced which may impact on operations if solar arrays are not pointing the correct direction. Serious damage may be done to the spacecraft if its blackbody radiators do not face deep space, but rather point towards a "warmer" source. Blackbodies not only radiate efficiently, they also absorb heat efficiently. Improper pointing of a blackbody might overheat the entire spacecraft or just one integral part which could shut down satellite operations for good.

During orbit acquisition, the spacecraft is spun to provide stability to the thrusting maneuver. While spinning a spacecraft tends to nutate in a cone-like shape. Thrusters are used to eliminate nutation and facilitate proper thrusting direction. Maintaining attitude is difficult since many perturbations (same as orbital perturbations) exist which work to constantly change the attitude of the spacecraft.

Momentum is conserved in natural systems and therefore it is a preferred choice for spacecraft stabilization. Giving controlled momentum to a spacecraft provides a buffer against undesirable momentum imparted on the spacecraft due to perturbations. Small perturbations that would otherwise slightly disturb a spacecraft possessing no momentum are absorbed by the momentum of a spacecraft containing controlled momentum. Controlled momentum is usually given to a spacecraft in several of four manners: spin stabilization, momentum wheels, reaction wheels, and/or control moment gyros (CMGs). Interestingly enough, the decision to use spin stabilization or a specific system of momentum control is usually made as a result of company experience. For example, Hughes Aerospace is very adept at producing spin-stabilized spacecraft and often does so. Other companies are less inclined to produce these type of spacecraft because of their expertise in other areas. GEOCOMM is a three-axis stabilized spacecraft because of designer preference.

5.2 Attitude Control Subsystem Equipment

THRUSTERS

Thrusters are used to control the attitude of a spacecraft through short bursts offering small torques. They are convenient because they can be integrated into the station keeping thrusters and share a common fuel supply. The magnitude of the thrust for attitude control is much smaller than that of station keeping. Other attitude control devices do not offer the variable torques provided by thrusters. On the down-side, thrusters used for attitude control contribute to the on-off cycles of the thrusters, further limiting the useful lifetime of the thrusters. In addition, use of the thrusters also contributes to use of the fuel.

MAGNETIC TORQUERS

Magnetic torquers use the earth's magnetic field with a magnetic field created on-board to create a controlling torque. These torquers are excellent for use on spacecraft since they require no fuel and possess essentially limitless lifetimes. An added power requirement is levied by the use of magnetic torquers since the magnetic field created on-board is created through the use of electricity passing through coiled wires, but it is not very great.

MOMENTUM WHEELS/REACTION WHEELS/CMGs

Internal momentum devices utilize heavy spinning wheels that are spun up with the use of thrusters or spin motors. These momentum devices act as buffers to unwanted perturbations in that the momentum in the wheel(s) is decreased while the external motion of the spacecraft does not change. Control moment gyros are gimballed so that the wheels can be moved effecting a reaction of equal magnitude but in the opposite direction to the gimbal resulting in an external movement of the attitude. These devices are relatively reliable and used extensively, but they are too heavy to provide redundancy which often leads to the spacecraft operating on less than a full compliment.

ATTITUDE SENSING DEVICES

The attitude of the spacecraft must be constantly rectified when necessary; therefore, it is necessary to have on-board devices that "sense" the attitude of the vehicle. The attitude is known if three pieces of independent (uncoupled) information are known about it. This information is gathered through on-board sensors. Attitude sensors come in two distinct types: reference sensors, and inertial sensors. Reference sensors provide information about the attitude of the spacecraft relative to an external phenomena (e.g., relative position of the moon to the spacecraft). Inertial sensors provide information about the attitude of a spacecraft relative to what the attitude was at a previous time period.

REFERENCE SENSORS

Reference sensors include: sun, earth, star, radio frequency, and magnetometers. Each sensor provides information to the spacecraft about the spacecraft's relative orientation to these phenomena. The radio frequency sensor is useful for geosynchronous communications spacecraft since their pointing direction is usually fixed on the ground station. A radio frequency is sent up from the ground station and the spacecraft fixes on the source of the frequency as a method of attitude sensing. Magnetometers measure magnetic fields and changes in those fields to determine attitude. The star sensor is the most accurate sensor measuring attitude to ≈ 0.01 arcsecond, but as a consequence has a high cost and mass impact on the spacecraft.

INERTIAL SENSORS

Gyroscopes and accelerometers are inertial sensors. They essentially measure changes in the attitude of a spacecraft from its last fixed point. Therefore, if the spacecraft is known to be pointing correctly at time t_i, and at time t_f the attitude is sensed to have changed 3° to the north by a gyroscope or accelerometer, then the new attitude is known and can be corrected properly as a result of this information.

ATTITUDE CONTROL SUBSYSTEM COMPUTER

The entire attitude control system is controlled by a computer that compares the current attitude as measured by the sensors to the desired attitude and initiates changes by using thrusters and momentum devices. The computer used for attitude control is the same one used for Telemetry, Tracking, and Command processing.

5.3 GEOCOMM Attitude Control Subsystem Design

A generic flow describing the design of an attitude control subsystem is shown in Figure 5.1. GEOCOMM is designed to be a three-axis stabilized spacecraft which means that three momentum wheels are necessary to maintain each axis (pitch, roll, and yaw). The sensors used to measure attitude are chosen on the basis of accuracy, mass, and cost. Pointing requirements are satisfied by employing one sun sensor, one radio frequency (RF) sensor, and one accelerometer. The ground station provides the uplink radio frequency which is tracked by the RF sensor.

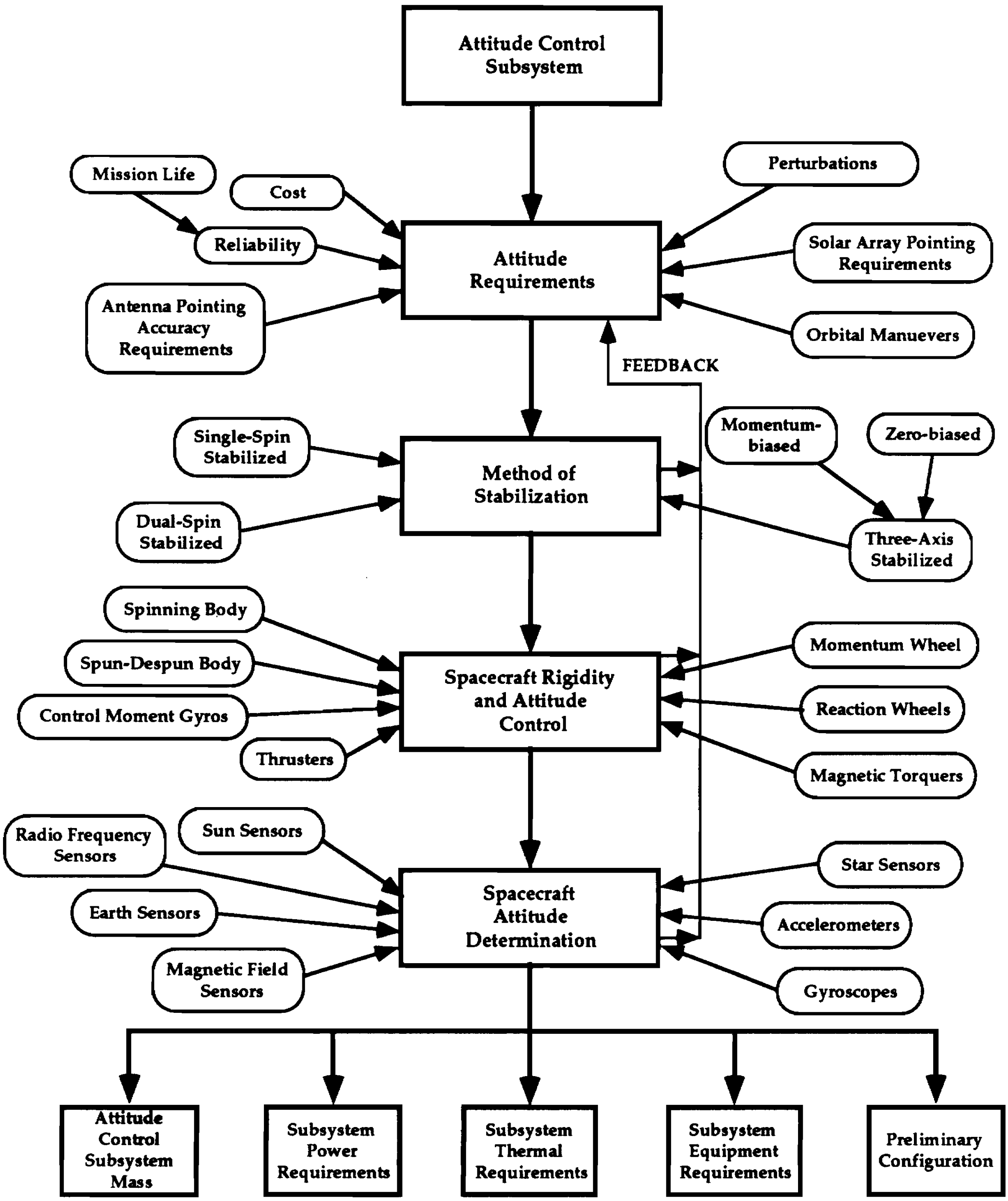


Figure 5.1 Attitude Control Subsystem Design Flow

The thrusters used for attitude control are listed in Table 5.1, and are identical to the ones used on Intelsat V. Blank positions denote thrusters used solely by the propulsion subsystem. The placement of the thrusters is shown in Figure 4.2. The thrusters are integrated into the bi-propellant propulsion subsystem. A well made decision made earlier about using a bi-propellant system provided design flexibility in this subsystem. An on-board computer is used to compare desired attitude to sensed attitude and initiate actions to reconcile the two. This computer is shared by the TT&C subsystem and is covered in more detail in the TT&C section. The sensors, momentum wheels, thrusters, and computers are combined to form the system shown in Figure 5.2.

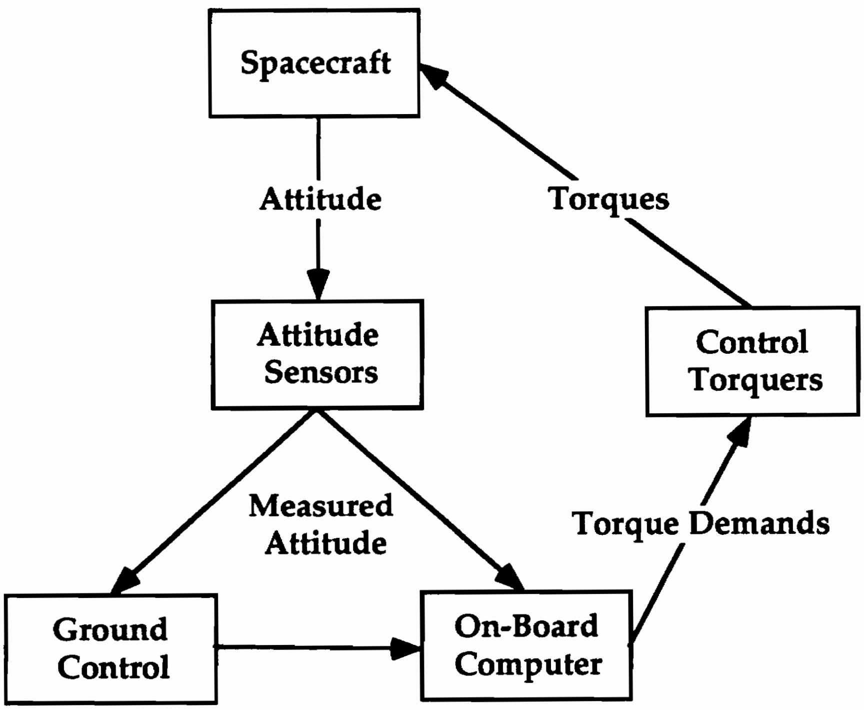


Figure 5.2 Generic Attitude Control System (See Reference #4)

The attitude control subsystem is a hybrid composed of parts of various other subsystems. The computer and sensors may be categorized as TT&C components, and the thrusters could be listed as propulsion subsystem elements. Only the momentum wheels are specific to the attitude control subsystem.

Table 5.1 GEOCOMM Attitude Control Subsystem Thrusters (bi-propellant)

Thruster	Function
1A	Active Nutation Control During Transfer Orbit
1B	Active Nutation Control During Transfer Orbit
2A	Positive Pitch Attitude Control
2B	Positive Pitch Attitude Control, Positive Yaw Attitude Control
3A	Negative Pitch Attitude Control
3B	Negative Pitch Attitude Control
4A	
4B	Negative Yaw Attitude
5A	Negative Yaw Attitude
5B	Positive Yaw Attitude Control
6A	Positive Roll and Negative Yaw
6B	Positive Roll and Negative Yaw
7A	Negative Roll and Positive Yaw
7B	Negative Roll and Positive Yaw
8A	
8B	
9A	
9B	
10A	
10B	

6.0 Electric Power Subsystem

6.1 Electric Power Requirements

Electrical power must be provided to the communications payload in order for it to operate properly. In addition, proper operation of other spacecraft subsystems is reliant upon electrical power. A relatively constant power supply is required by the spacecraft during its lifetime. Power must be either brought into orbit with the spacecraft or generated while in orbit. Communications spacecraft typically allocate approximately eighty-seven percent [See Reference #1] of the total power required to the communications payload. Of the eighty-seven percent, eighty-two percent is consumed by high voltage TWTAs or solid-state RF amplifiers. Power requirements are set by each subsystem at the end of the design of each subsystem. Assume that GEOCOMM requires 2000 W of communications power and 300 W of housekeeping power, for a total of 2300 W of power required.

Once the power levels are set, a method of providing the required power is chosen. As a result of the power generation system chosen, additional requirements are levied on the spacecraft. For example, if solar cells are chosen as the power generating devices, the spacecraft must point its arrays towards the sun (attitude control subsystem), and eclipse conditions become much more important to the engineers. In fact, nearly all communications satellites employ solar arrays for power generation purposes. Solar arrays are employed by GEOCOMM. Eclipse conditions are therefore important in the design of GEOCOMM, its batteries, and its solar arrays. In geosynchronous orbit, spacecraft experience forty-five days of eclipse season centered around the equinoxes with maximum eclipse lasting around seventy-two minutes. Solar intensity is greatest during the autumnal equinox, but least during the summer solstice. For design of the batteries it is assumed that approximately 900 charge/discharge cycles will be performed in a ten year period.

6.2 Electric Power Subsystem Equipment

An electric power subsystem consists of three distinct parts: power generation devices, power storage devices, and power control electronics. The most common power generation devices used are solar cells, but others exist that take advantage of chemical and/or nuclear reactions to produce power. Chemical power generation devices are usually limited in their effective lifetime and therefore not desirable for missions longer than about six months. Nuclear power generation devices, on the other hand, are very efficient and effective, but fear of launch failure and the possibility of radiation spread over populated areas has limited their uses to very few applications. Power storage devices are used in conjunction with solar cells to provide power during eclipse phases. Power control electronics account for the fact that power generation devices degrade over time and the power that they produce at the beginning of life is much higher than the power produced at the end of life. In addition, each device on-board may have different current and voltage requirements. The electric power subsystem must provide power to each of the subsystems for the lifetime of the spacecraft (GEOCOMM = 10 years).

SOLAR CELLS

Solar cells convert solar radiation into useful electrical energy. Solar cells are currently the only devices that efficiently convert enough solar energy into electrical energy to be useful on spacecraft. Unfortunately, solar cell efficiency degrades due to radiation damage over time. There are a number of different types of solar cells, conventional silicon solar cells of which there are two types, violet and black, and gallium arsenide solar cells. Violet cells take advantage of the fact that solar radiance is abundant in the blue and ultraviolet regions of the spectrum and an increase in efficiency can be achieved by increasing the spectral response of the solar cells to these specific regions of the spectrum. Their efficiency is about 14 to 14.5 percent which delivers 76 mW from a 2x2 cm cell. Black cells are violet cells which have chemically etched surfaces to reduce their reflectivities and thus increase their efficiencies. The increase in efficiency is about 1 percent. Black cell efficiency is about 15.5 percent which delivers 84 mW from a 2x2 cm cell. Gallium arsenide cells have not been proven extensively in the space environment, but much increased efficiency (≈17 percent) and less degradation of power over time due to solar radiation damage have contributed to the high expectations of the engineers.

A solar array is a combination of solar cells arranged to produce voltage and current to satisfy the requirements of the spacecraft during its lifetime. Summer solstice corresponds to the lowest radiation levels received from the sun and satellite end-of-life represents the largest degradations in performance. Cells are arrayed in series to satisfy voltage requirements, and in parallel to satisfy current requirements. Arrays are wired so that the loss of one solar cell does not severely impact the power generation performance of the entire solar array. Since the launch vehicle fairing (the shroud that contains the spacecraft during launch) or Shuttle cargo bay is of limited size, solar arrays are usually launched in a folded or rolled position and then deployed outward once the spacecraft is deployed. Solar arrays for three-axis stabilized spacecraft can be deployed in any manner that ensures that power requirements are satisfied. The most common deployment methods include flexible rollout blankets, foldout blankets, and rigid foldout honeycomb panels. The power produced by a solar array is proportional to the solar intensity, F, and the cosine of the angle, q, between the solar array surface normal and the solar rays. Maximum solar intensity is at its maximum during vernal equinox and at its minimum during summer solstice.

BATTERIES

Batteries provide power to the spacecraft at the times that the solar array cannot. For example, during eclipse, launch, and deployment, solar cells are either not illuminated, or not deployed. In fact, pyrotechnic devices used in the aid of deployment and deployment of the solar arrays are powered by the batteries. Battery performance is usually measured in terms of energy density, charge/discharge efficiency, high temperature performance, depth of discharge DOD, and number of charge/discharge cycles performed during a mission.

Batteries are made of many different types of materials. Each has its advantages and disadvantages for specific mission requirements. Six of the most popular battery types are: Ni-Cd, Ag-Zn, Ni-H_2, Ag-H_2, Li-FeS, and Na-S. Ni-Cd batteries are used extensively for spacecraft. They exhibit good overall performance, but are limited in DOD and recharging must be carefully implemented to reach desired lifetimes. Ag-Zn batteries have low cycle lifetimes and are therefore limited for use in long-term missions. $Ni\text{-}H_2$ batteries were developed to replace the Ni-Cd battery and have done so admirably. $Ni\text{-}H_2$ batteries exhibit more predictable and reliable behavior, and have longer cycle lifetimes and deeper DODs than Ni-Cd batteries. $Ag\text{-}H_2$ batteries have high energy density and long cycle lifetimes with deep DODs, but are relatively untested as of yet. Also, Li-FeS batteries are as of yet untested in space applications, but in time could replace $Ni\text{-}H_2$ batteries for geosynchronous communications applications. Na-S batteries are good for large power requirements over short periods of time. These are mainly useful for scientific missions. The two batteries that are used extensively for long-term geosynchronous missions are the Ni-Cd and $Ni\text{-}H_2$ batteries. Occasionally, batteries must be reconditioned, discharged fully, during the lifetime of the mission to ensure proper operating voltage. Without reconditioning, batteries becomes effectively weaker through disuse of portions of the battery.

POWER CONTROL DEVICES

Power control electronics ensure proper voltage and current is delivered to each subsystem. The solar array does not always produce the desired quantity of power. During autumnal equinox, more power than is required for the spacecraft is produced, while during summer solstice at end-of-life less power is produced than is necessary. Power control electronics ensure that the battery is charged during times of excess power generation, and that the battery is discharged into the load during the times of diminished power generation.

6.3 GEOCOMM Electric Power Subsystem Design

A preliminary design flow for an electric power subsystem is shown in Figure 6.1. This section presents preliminary design calculations for the electric power subsystem components (e.g., solar array and batteries) of GEOCOMM.

GEOCOMM utilizes sun-tracking flat panels that are rolled out once deployed in geosynchronous orbit. The load is constant at 2300 W, and the bus voltage is maintained between 60 V during sunlight and 35 V during eclipse. These voltages correspond to equipment requirements on-board the spacecraft. High voltages and voltage spikes are detrimental to any piece of equipment, and low voltage does not ensure that the equipment functions properly or at all. To ensure lifetime expectancy and minimal mass, a dual-bus unregulated system is chosen for GEOCOMM.

A dual-bus design provides reliability in that half of the power required by the spacecraft is handled by one bus and the other half by the other bus. If one bus

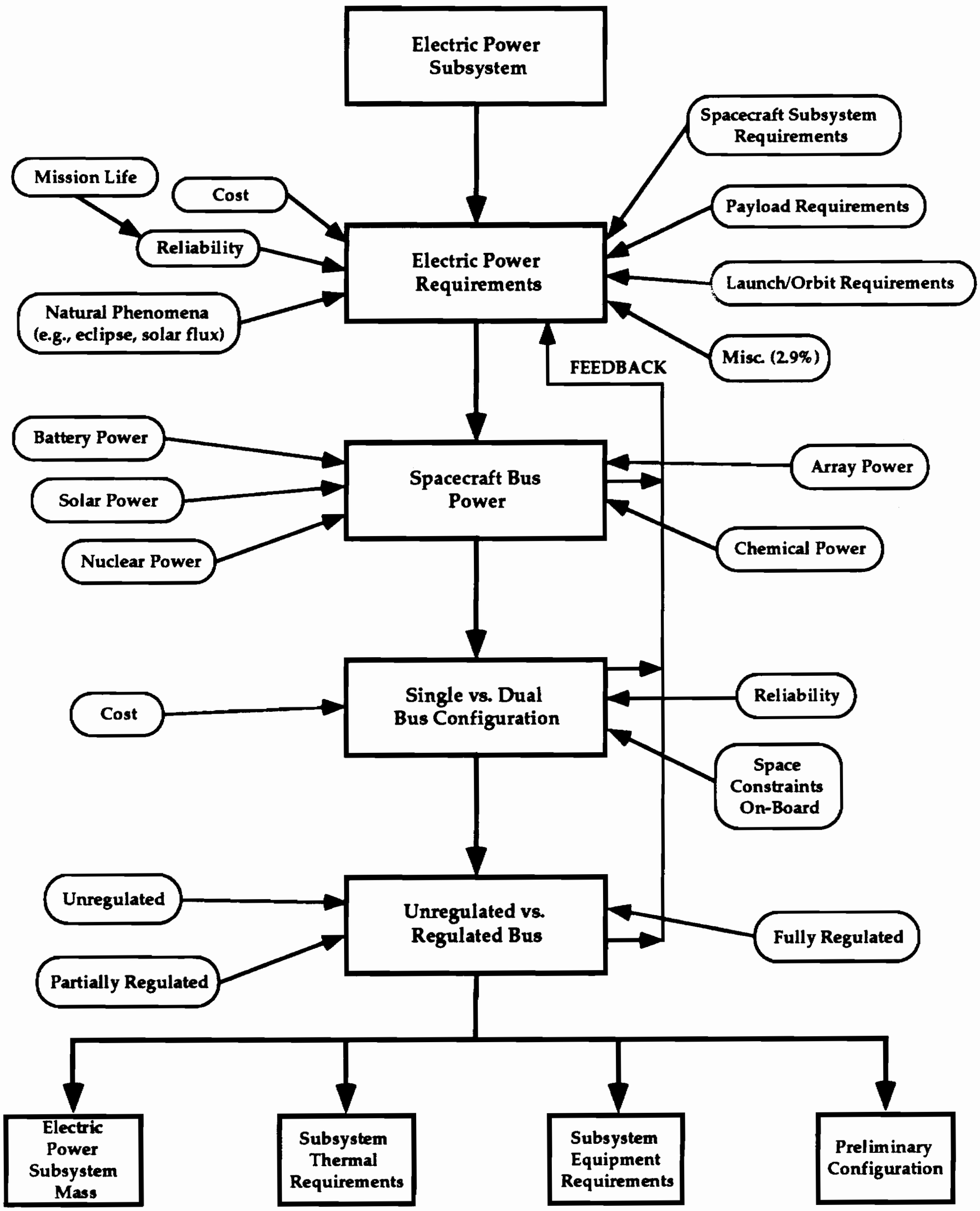


Figure 6.1 Electric Power Subsystem Design Flow

is defective, all of the systems are not shut down immediately. Additionally, power requirements differ for the various devices on-board. Power can be conditioned centrally or at each device independently. This is referred to a regulated or unregulated bus. Fully regulated busses are expensive and add significantly to the mass of the spacecraft, while unregulated busses require that device has its own power conditioning equipment.

BATTERIES

Ni-H_2 batteries are chosen since they can handle a 10 year mission and support high energy requirements. They are also mission tested and proven to be useful for long-term geosynchronous missions. A battery consists of multiple cells combined to satisfy power requirements. The minimum discharge voltage of a cell at the end of 10 years is assumed to be 1.25 V. The number of cells in series (to provide the minimum voltage necessary) is calculated during eclipse when all 2300 W are to be provided by the battery. Assuming that one battery cell is not operating properly at the end of life, there is an open circuit condition resolved through a bypass diode which results in a voltage drop, V_{DD}, of 1.25 V. The number of battery cells required in series is

$$V_{DB} = (N - 1)V_D - V_{DD} \quad (6\text{-}1)$$

V_{DB} = Minimum Discharge Voltage [V] = 35 V (6-2)

N = Number of Battery Cells in Series [ ] (6-3)

V_D = Voltage Provided by Each Cell in Series = 1.25 V / Cell (6-4)

V_{DD} = Voltage Drop Due to Faulty Cell = 1.25 V (6-5)

$N = (V_{DB} + V_{DD}) / V_D + 1 = (35 + 1.25) / 1.25 + 1 = 30$ Cells. (6-6)

In a dual bus configuration, each bus must provide half of the required power. In this case, each bus must provide 2300 W / 2 = 1150 W of power. During a 72 minute solar eclipse the battery will receive its largest requirement for power. The required capacity of each battery cell is

$C = P * t / (V_{DB} * DOD)$ (6-7)

C = Cell capacity [Ampere hours, Ah] (6-8)

P = Power expected from the battery = 1150 W (6-9)

t = Maximum time battery must maintain power = 1.2 hr (6-10)

V_{DB} = Minimum discharge voltage = 35 V (6-11)

DOD = Depth of Discharge = 70% for $Ni\text{-}H_2$ batteries (6-12)

$C = 1150 * 1.2/(35 * 0.70) = 56.33 \approx 56$ Ah. (6-13)

During charging, the maximum allowable battery charge voltage is 1.5 V. Open-circuit failures of battery cells during charge is accommodated by three series-connected silicon diodes connected in parallel with the cell. The voltage drop in each diode is 0.8 V. The maximum charge voltage is

$$V_{BC} = V_{BCM} * N + N_d * V_d \tag{6-14}$$

$$V_{BC} = \text{Maximum charge voltage [V]} \tag{6-15}$$

$$V_{BCM} = \text{Maximum charge voltage per cell} = 1.5\text{ V} \tag{6-16}$$

$$N = \text{Number of battery cells} = 30\text{ Cells} \tag{6-17}$$

$$N_d = \text{Number of diodes protecting open circuits} = 3 \tag{6-18}$$

$$V_d = \text{Voltage drop attributed to each diode} = 0.8\text{ V} \tag{6-19}$$

$$V_{BC} = 1.5 * 30 + 3 * 0.8 = 47.4\text{ V.} \tag{6-20}$$

Charging voltages must be higher than load voltages, so sometimes a separate charge array is added to boost voltage. If the main bus voltage is limited to 60 ± 2 V, no boost in voltage is required by a separate charge array on GEOCOMM to ensure that the battery charges fully. The batteries are charged at different rates during specific times of the year. During the equinox when there are solar eclipses each day, the batteries are charged quickly (one fifteenth of the capacity per hour = Capacity/15 = C/15 = 56/15 = 3.73 A). During the solstice periods when there are no eclipses, the batteries are slowly charged (one forty-fifth of the capacity per hour = Capacity/45 = C/45 = 56/45 = 1.24 A) to prevent overcharging and maintain viability of the battery capacity. Charging efficiency is about 90%. The power required for charging the battery quickly during equinox is

$$P_{charge@equinox} = I_{equinox} * V_{charging} \tag{6-21}$$

$$P_{charge} = \text{Power required for charging the battery at equinox [W]} \tag{6-22}$$

$$I_{equinox} = \text{Current required for quick charging} = 3.73 \text{ A} \tag{6-23}$$

$$V_{charging} = \text{Voltage necessary for battery charging} = 47.4 \text{ V} \tag{6-24}$$

$$P_{charge@equinox} = 3.73 * 47.4 = 176.8 \approx 177 \text{ W} \tag{6-25}$$

The power required for charging the battery slowly during solstice is

$$P_{charge@solstice} = I_{solstice} * V_{charging} \tag{6-26}$$

$$P_{charge@solstice} = \text{Power required to charge battery at solstice [W]} \tag{6-27}$$

$$I_{solstice} = \text{Current required for slow charging} = 1.24 \text{ A} \tag{6-28}$$

$$P_{charge@solstice} = 1.24 * 47.4 = 58.8 \approx 59 \text{ W.} \tag{6-29}$$

The time to recharge the battery after use during equinox is given by

$$t_{recharge} = P_{discharge} * t_{discharge} / (P_{charge} * h) \tag{6-30}$$

$$t_{recharge} = \text{Time to recharge battery after use [hr]} \tag{6-31}$$

$$P_{discharge} = \text{Power discharged by spacecraft} = 1150 \text{ W} \tag{6-32}$$

$t_{discharge}$ = Time spacecraft relies on battery = 1.2 hr max eclipse (6-33)

P_{charge} = Power required to charge battery = 177 W at equinox (6-34)

h = Efficiency of charging the battery = 0.90 (6-35)

$t_{recharge}$ = 1150 * 1.2 hr / (177 * 0.90) = 8.7 hrs.

The time to recharge the battery at other times is not calculated because use of the battery is minimal all times other than eclipse during equinox.

SOLAR ARRAY DESIGN

The sun-tracking panels are oriented in the north-south axis of spacecraft orbit so that they may receive the maximum amount of incident radiation. The power load placed on the electric power subsystem consists of the requirements levied by all of the other subsystems, and the power required to charge the batteries. A 10% margin is figured into the power requirements to account for degradation of equipment due to radiation, and other uncertainties. The solar array design load at equinox is given by

$$L_{equinox} = 1.1(P_{spacecraft} + P_{charge@equinox}) \quad (6\text{-}36)$$

$L_{equinox}$ = Load on the Solar Array at Equinox [W] (6-37)

$P_{spacecraft}$ = Power Requirements of the Spacecraft = 2300 W (6-38)

$P_{charge@equinox}$ = Power Reqts for Battery Charging = 177 W (6-39)

$L_{equinox} = 1.1(2300 + 177) = 2724.7$ W. (6-40)

The solar array design load at solstice is given by

$L_{solstice} = 1.1(P_{spacecraft} + P_{charge@solstice})$ (6-41)

$L_{solstice}$ = Load on Solar Array at Solstice [W] (6-42)

$P_{charge@solstice}$ = Power Reqts for Battery Charging = 59 W (6-43)

$L_{solstice} = 1.1(2300 + 59) = 2594.9$ W. (6-44)

Each bus will provide 1362 W during equinox and 1297 W during solstice.

The solar cells chosen for GEOCOMM are the same ones used on Intelsat VI K7. Their characteristics include:

Power BOL (28°C) = Power output beginning of life = 307.8 mW (6-45)

Power EOL (28°C) = Power output at end of life = 230.8 mW (6-46)

I_{mp} = Solar cell current at maximum power = 0.644 A (6-47)

V_{mp} = Solar cell voltage at maximum power = 0.478 V (6-48)

Size = 2.5 x 6.2 cm (6-49)

Thickness = 0.02 cm (6-50)

Material = Silicon

The solar cells have a black surface field, surface reflector, and are treated with an anti-reflective coating $T_iO_xAl_2O_3$. The cover is a cmx microsheet with anti-reflective coating, textured, and 0.021 cm thick. The cell current at EOL during summer solstice is [See Reference #1]

$I = [I_{mp} + a_I(T - 25)]K^i{}_A K^i{}_D K_s$ (6-51)

I = Solar cell current at EOL during summer solstice [A] (6-52)

I_{mp} = Solar cell current at maximum power = 0.644 A (6-53)

a_I = Temperature coefficient for current = 0.24×10^{-3} (6-54)

T = Operating temperature = assume 40° C max (6-55)

$K^i{}_A$ = Design factor for assembly losses in current = 0.96 (est) (6-56)

$K^i{}_D$ = Design factor envrn. degradation in current = 0.8154 (est) (6-57)

K_s = Solar intensity factors, with incidence ang = 0.8885 (est) (6-58)

$I = [0.644 + 0.24 \times 10^{-3}(40 - 25)] * 0.96 * 0.8154 * 0.8885 = 0.450$ A. (6-59)

The current that must be produced by each solar array wing at summer solstice is

I_T = Power / Bus Voltage (6-60)

I_T = Current Produced by each Solar Array Wing [A] (6-61)

Power = Power produced by each Array Wing = 1297 W (6-62)

Bus Voltage = Maximum Regulated Bus Voltage = 60 V (6-63)

I_T = 1297 / 60 = 21.62 W. (6-64)

The number of solar cells needed in parallel to produce the desired current is

$N_p = I_T / I$ (6-65)

N_p = Number of Solar Cells in Parallel [ ] (6-66)

I = Solar Cell Current at EOL During Summer Solstice = 0.450 A (6-67)

N_p = 21.62 / 0.450 = 48.04 ≈ 48 solar cells in parallel. (6-68)

Solar cell voltage at end of life during summer solstice is given by [See Reference #1]

$V = [V_{mp} - \Delta V + a_V(T - 25)]K^V{}_E$ (6-69)

V = Solar cell voltage at EOL during summer solstice [V] (6-70)

V_{mp} = Solar cell voltage at maximum power = 0.478 V (6-71)

ΔV = Panel wiring loss per cell = assume 0.005 max (6-72)

a_V = Temperature coefficient for voltage = -0.0022 (6-73)

T = Operating temperature = assume 40° C max (6-74)

$K^V{}_E$ = Radiation degradation factor for voltage = 0.935 (6-75)

$$V = [0.478 - 0.005 + 0.0022(40 - 25)] * 0.935 = 0.411 \text{ V}. \tag{6-76}$$

The number of solar cells needed in series to produce the desired voltage is, assuming two bus voltage drops of 0.9 V each for degradation of equipment,

$$N_s = (\text{bus voltage} + \text{bus voltage drops}) \;/\; \text{cell voltage} \tag{6-77}$$

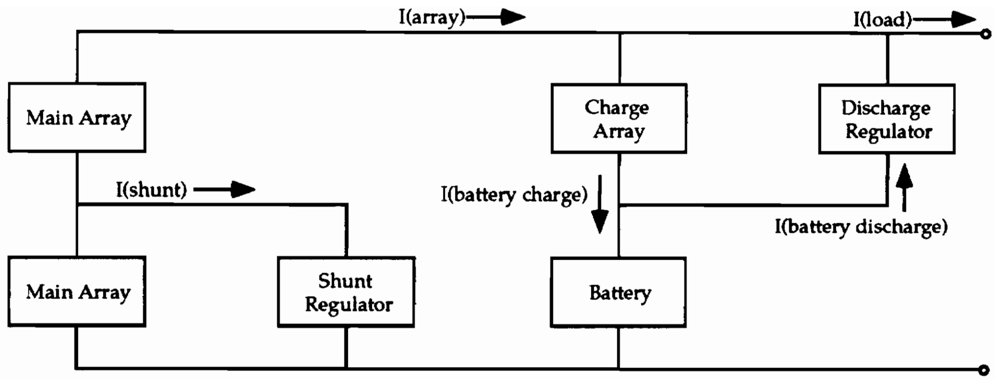
$$N_s = \text{Number of Solar Cells in Series [ ]} \tag{6-78}$$

$$\text{bus voltage drops} = \text{due to degradation of equipment} = 1.8 \text{ V} \tag{6-79}$$

$$N_s = (60 + 1.8) \;/\; 0.411 = 150.4 \approx 150 \text{ solar cells in series.} \tag{6-80}$$

The number of solar cells needed in parallel and in series were rounded down. This was done to conserve mass, solar cells being a large contributor. Since this is only a preliminary evaluation assuming large margins for safety it is likely that the final number of cells in parallel and in series may be even smaller than estimated here.

The batteries and solar arrays are wired to the spacecraft electrical loads with the power control electronics shown in Figure 6.2. These electronics ensure that the load receives proper power and that the batteries are properly charged.



<u>Figure 6.2 Generic Electric Power Subsystem Schematic Diagram (See Reference #1)</u>

As a last comment about the Electric Power subsystem it is pointed out that as the spacecraft travels about the earth the sun appears to travel about the spacecraft with a frequency of one day. In order to maintain the angle between the sun and the solar arrays, the arrays must move in relation to the sun. This introduces a complex mechanical device which moves the solar panels and a complex electric power transfer device. The electric power transfer devices (usually slip rings) allow power to be transferred from the solar array to the power control electronics while the panels are in motion.

7.0 . Thermal Control Subsystem

7.1 Thermal Control Requirements

All equipment has a temperature range in which it operates most effectively, spacecraft are no exception. Since spacecraft are comprised of many different pieces of equipment and structural members, it is not just one temperature that must be maintained on-board. Several temperature range requirements must be satisfied simultaneously. Some equipment must be maintained warm, and other equipment operates more effectively cold. Maintaining specific temperatures in the space environment is not as easy as it is on earth. Between the spacecraft and its environment, radiation is the only method of heat transfer available. Fortunately for the electric power subsystem, the sun shines incessantly and provides much desired energy to the solar cells, but it also provides much undesired heat to the overall spacecraft that must be dissipated. In some cases, the sun's heat energy is desirable to keep to overall spacecraft at about room temperature, but for the most part, the heat energy must be reflected or radiated into to deep space. Internal to the spacecraft, conduction and radiation are the two methods of heat transfer available. High energy dissipating devices such as TWTAs cannot heat up indefinitely. Sooner or later the heat must taken away or the entire unit will fail. Various methods have been devised to ensure that equipment is operated within its proper temperature ranges.

Once the requirements for the other subsystems have been identified and preliminary configurations have been chosen, the thermal control subsystem can be analyzed. Even the spacecraft structure itself has thermal constraints that must be satisfied to prevent warping and bending due to severe temperature gradients. Table 7.1 shows approximate temperature ranges for various equipment on-board a spacecraft.

The temperature ranges shown in Table 7.1 must be maintained by the spacecraft in order for all systems to operate as desired. Undesired heat is introduced into the spacecraft from solar radiation, solar radiation reflected by other planets, thermal radiation from other planets, rocket motors, electronic devices, batteries, and aerodynamic drag during launch. Heat can be eliminated from the spacecraft through black-body radiation. Black-body radiators emit or absorb the maximum amount of thermal energy possible at a given temperature. Black-body radiators are pointed at deep space and undesired thermal energy is routed their way for elimination. During eclipse periods much of the spacecraft's heat energy is eliminated and heaters are needed to ensure the temperature within the spacecraft remains within

Table 7.1 Operating Temperature Ranges for Spacecraft Equipment

Equipment	Operating Temperature Range [°C]
Communications	-10 to +30
Antennas	-120 to +90
Solar Array	-160 to +80
Battery	-5 to +25
Attitude Sensors	-30 to +50
Propellant	+10 to +45
Microprocessors	-5 to +40
Structure	-50 to +50

specified temperature limits. Basically it is this trade-off between eliminating undesirable heat and ensuring that the spacecraft temperature remains within specified limits that the thermal control engineer deals with when designing a spacecraft.

7.2 Thermal Control Equipment

Thermal control devices fall into two distant categories, passive and active. Passive devices require no moving parts, are generally cheaper than active devices, and are more reliable than active devices. Active devices contain moving parts or fluids, are more massive than passive devices, are less reliable than passive devices, but are necessary for the control of large amounts of heat energy.

PASSIVE DEVICES

Passive thermal control devices consist mainly of thermal coatings, paints, blankets, heat sinks, and phase-change materials. Thermal coatings and paints are used on the outside of the spacecraft to avert the harmful effects of external radiation. Thermal coatings and paints either reflect the solar radiation or they absorb it. In addition, these coatings may allow thermal energy to be emitted or not. Often, thermal coatings with differing properties are applied simultaneously to a surface in a pattern (such as a checkerboard) that results in the desired thermal conditions prevailing inside the spacecraft. Thermal blankets are used in areas where isothermal conditions are desired. Thermal blankets are used to totally insulate an area from external conditions. The only heat entering or leaving an isothermal area must do so through designated channels or through devices dissipating heat to the isothermal area from within. Heat sinks and phase-change materials work in a similar manner. Heat sinks receive thermal energy from nearby equipment and either hold it or send it to thermal emitters or radiators for dissipation. Phase-change materials consist of substances with melting points at or near the desired operating temperature of the device they are protecting. As the protected device heats up, excess heat is dumped into the thermal control device which absorbs as much as it can by changing phase from a solid to a liquid. When conditions change and the protected device is having a hard time maintaining the correct temperature, the substance gives its energy back to the protected device causing the reverse phase change back to a solid.

Radiators that are essentially black-bodies are most efficient at eliminating thermal energy. These are considered passive control devices. An approximation of the size radiator necessary can be made by estimating the amount of power dissipated by various equipment on-board. Assume that GEOCOMM dissipates 1000 W of energy, and that this energy must thus be dissipated to space.

The solar array of a spacecraft operates most efficiently if the temperature of the solar cells is kept below 50°C. Paradoxically the solar cells which need the sun to operate, operate more efficiently when the sun is not shining.

ACTIVE DEVICES

Active thermal control devices contain moving parts and are usually used when large amounts of heat must be managed on-board. The most common active devices include the following: heaters, heat pipes, louvers, shutters, bimetallic fins, Peltier heat pumps, and large-scale fluid transport systems. Heaters are used when the temperature of a device falls below its optimal operating temperature and must be warmed up. Heaters are controlled by thermostats which react when the temperature falls below the desired operating temperature and switches the heater on. Heat pipes consist of large capillary tubes with a wick in the middle running the length of the pipe. Liquid, usually an ammonia compound, is transported by the wick to the area to be cooled. Heat energy from the area (or device) to be cooled is transferred into the 'cooler' liquid and the liquid evaporates inside of the capillary tube. Once evaporated, the vapor travels to the other end of the heat pipe where the heat energy is transferred to a radiator for deep space transmission, and the vapor becomes a liquid again. Heat pipes are used for moving large amounts of thermal energy from one location to another. Louvers, shutters, and bimetallic fins are all methods of changing the surface properties of a spacecraft from one that mainly absorbs to one that mainly emits. Louvers and shutters open and close as specified by the on-board computer based on desired temperature. Bimetallic fins operate much as a thermostat. If the fins are too hot they show their reflective coating, if they are too cool they show their absorptive coating. Peltier heat pumps are used for spot cooling and small cooling loads and operate much as a heat pump on earth operates. Fluid transport thermal systems are used only on large-scale manned missions. These are very expensive, massive, and unreliable for long-term use. Often active and passive thermal control devices are used in unison to produce the thermal results desired.

THERMAL MODELLING AND TESTING

Several software packages, ESABASE/THERMAL, LOHARP, and SINDA, are available to aid the designer in properly modeling the space environment and the conditions that may exist on-board. The initial step in any thermal design is proper modelling of the expected thermal situation. Once the model is considered appropriate, thermal control devices can be added to produce the desired conditions on-board. When the model operates as desired, a mock-up of the design is made and rigorously tested in an environment closely simulating the expected on-orbit conditions of the spacecraft. Modifications are made to the model based on the information gained in the mock-up test. The entire procedure is repeated until necessary. Usually only two iterations are necessary, but this is one area in which the designer cannot take chances.

7.3 GEOCOMM Thermal Control Subsystem Design

The generic design flow of a spacecraft thermal control subsystem is shown in Figure 7.1. With the exception of the heaters (thermostatically-controlled),

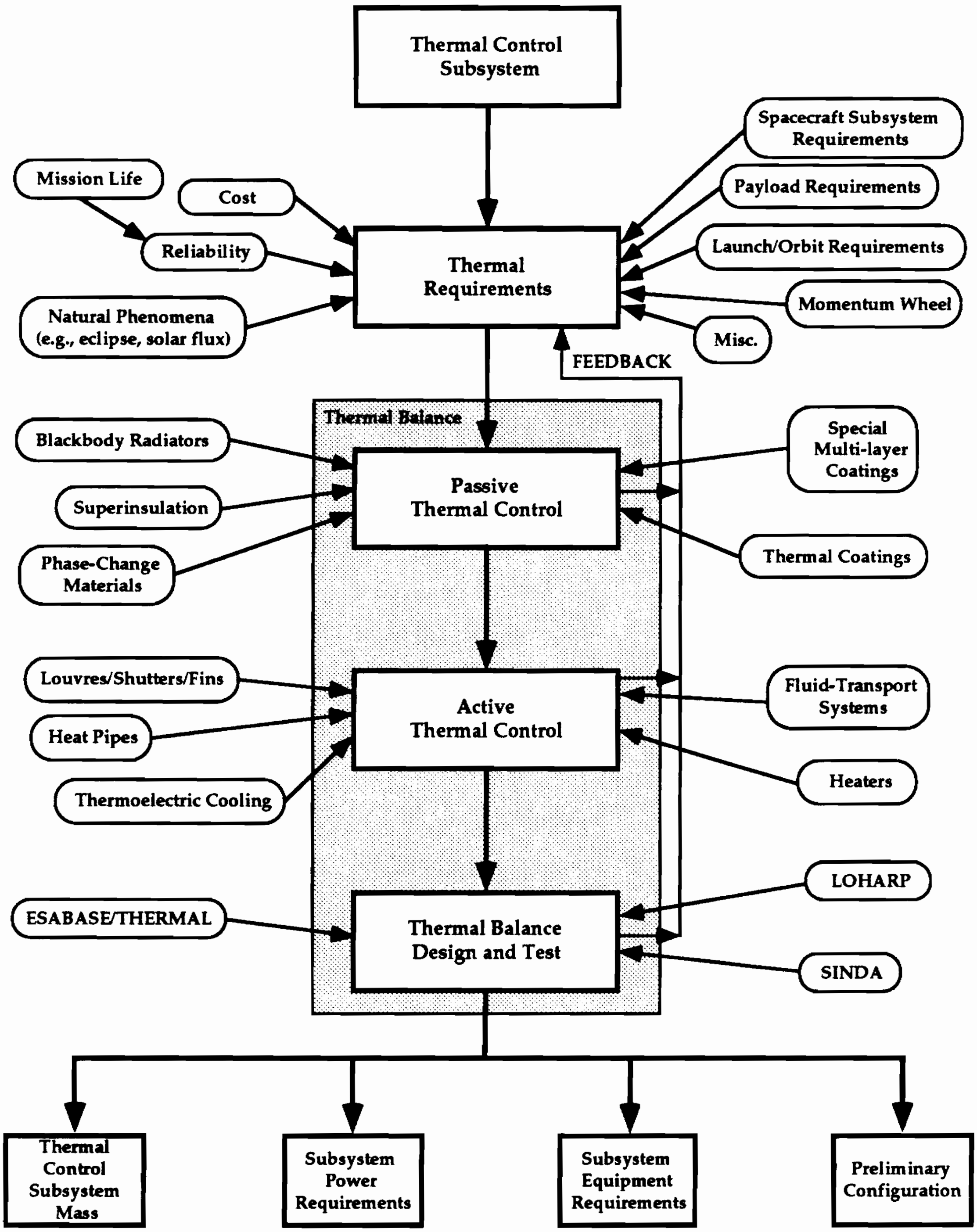


Figure 7.1 Thermal Control Subsystem Design Flow

GEOCOMM employs only passive thermal control devices. Passive devices are much more reliable and cheaper to produce.

Heat dissipation to deep space is achieved through the use of black-body radiators. Assuming that TWTAs are only 20% efficient in performing their operations, then of the 2000 W of total power needed by the communications subsystem 0.8 X 2000 W = 1600 W is dissipated as heat. The total area of black-body radiators needed to dissipate 1600 W of energy is calculated to be

$$\varepsilon\sigma T^4\eta A = \alpha_s AS \sin\theta + P \qquad (7\text{-}1)$$

$$\varepsilon = \text{Emittance of radiator at end-of-life} = 0.8 \qquad (7\text{-}2)$$

$$\sigma = \text{Stefan-Boltzmann constant} = 5.67 \times 10^{-8}\ W{*}K^4/m^2 \qquad (7\text{-}3)$$

$$T = \text{Max allowable temperature of radiator} = 40^\circ C = 313\ K \qquad (7\text{-}4)$$

$$\eta = \text{Radiator efficiency} = 0.85 \qquad (7\text{-}5)$$

$$A = \text{Area of radiator } [m^2] \qquad (7\text{-}6)$$

$$\alpha_s = \text{Absorptance of radiator at end-of-life} = 0.21 \qquad (7\text{-}7)$$

$$S = \text{Solar intensity at winter solstice} = 1397\ W/m^2 \qquad (7\text{-}8)$$

$$\theta = \text{Solar aspect angle at winter solstice} = 23.5^\circ \qquad (7\text{-}9)$$

$$P = \text{Power dissipated by on-board equipment} \approx 1600\ W \qquad (7\text{-}10)$$

$$A = P \,/\, (\varepsilon\sigma T^4\eta - \alpha_s S \sin\theta) \qquad (7\text{-}11)$$

$$A = 1600/(0.8*5.67 \times 10^{-8}*313^4*0.85 - 0.21*1397*\sin 23.5) = 6.95 \text{ m}^2. \quad (7\text{-}12)$$

GEOCOMM requires 6.95 m^2 of radiator area to ensure that thermal requirements for elimination of undesired heat are met. The back of the solar array is a large radiator which transmits the heat taken in on the front of the array and sends it to deep space. High heat dissipating equipment is located alongside the radiators for ease of transport of thermal energy. In addition, the east and west faces of the spacecraft are painted with a highly reflective, high emittance coating to resist overheating. Blankets (superinsulation) is used extensively on-board to protect all equipment from severe thermal conditions. Thermal control equipment used by GEOCOMM is shown in Figure 7.2.

Design of the thermal control portion of GEOCOMM inevitably set requirements for other parts of the spacecraft, even parts which have already been designed. Thermal control devices add mass to the spacecraft, and the heaters levy yet another requirement on the heavily taxed electric power subsystem.

• Thermal Control Subsystem

Subsystems	Equipment Temps [°C]	Control Devices
• Communications	• Transponder (-10 to +30) (including TWTAs)	• Blankets, radiators, heaters
	• Antennas (-170 to +90)	• Surface coatings
• Launch Vehicle	• None	• N/A
• Propulsion	• Thrusters (+5 to +35)	• Surface coatings, blankets
	• Propellant (+10 to +45)	• Blankets, heaters
• Attitude Control	• Attitude sensors (-30 to +50)	• Surface coatings, blankets, heaters
	• Thruster (+5 to +35)	• Surface coatings, blankets
	• Momentum wheels (-15 to +55)	• Blankets, heaters
• Electric Power	• Solar array (-160 to +80)	• Surface coatings, blankets, radiators
	• Batteries (-5 to +25)	• Blankets, heaters
	• Control Electronics (-45 to +65)	• Blankets
• Thermal Control	• None	• N/A
• Structural	• Structure (-50 to +50)	• Surface coatings, blankets, heaters
• TT&C	• Computers (-5 to +50) • Antenna (-170 to 90)	• Blankets

Figure 7.2 GEOCOMM Thermal Control Devices

8.0 Structural Subsystem

8.1 Structural Requirements

The structure of a spacecraft must satisfy two basic requirements: to protect the other subsystems from the harsh conditions experienced during launch, and to maintain the relative positions of spacecraft equipment and antennas throughout the lifetime of the spacecraft. Optimization of the spacecraft structure with respect to mass and strength is key to the successful design of an overall spacecraft. The structure must protect the other spacecraft subsystems from vibration, shock, acoustic coupling, and spin affecting the spacecraft during launch and subsequently during deployment and operations.

Requirements for design of the structural subsystem are derived mainly from the launch vehicle relative to protecting the other subsystems, but each subsystem levies its own unique structural requirements in addition. Choice of a launch vehicle defines the direction and magnitude of the launch loads subjected on the spacecraft. However, if the selection of a launch vehicle is not made until late in the design process, over-design is necessary to ensure that the structure can accommodate launch vehicles possessing different loading characteristics. Launch vehicle loads are explained in the handbook provided by the launch vehicle manufacturer. The thermal subsystem is intertwined with the structure in that bending due to thermal heating and cooling is to be avoided as much as possible, especially on spacecraft in which position accuracy is at a premium. The propulsion coupled with the attitude control subsystems require that the structure maintain the positions of the sensors, thrusters, and momentum wheels to ensure attitude accuracy, but they too are coupled with the structure in that the liquid fuel storage must be positioned such that as it is used the center of mass of the spacecraft remains relatively constant.

Basically the design process is iterative with many different configurations being suggested, analyzed, and tested. Design of the spacecraft structure is done continuously from the inception of project to just before it is delivered to the launch pad. As a result, the design must remain flexible so that new sections may be added and others deleted late in the process.

8.2 Structural Equipment

A common, almost necessary, procedure used in the design of a spacecraft involves the use of a Finite Element Analysis (FEA) software package to model and analyze the proposed spacecraft structure. FEA applies the Theory of Elasticity to certain basic design elements such as columns, shells, panels, and rings, which allows stresses and strains at particular discrete points to be calculated based on applied loading. The discrete points are chosen, based on experience, to be points of maximum stress or strain. The design is then iterated to ensure that the desired configuration satisfies the expected loading profile. Selection of materials exhibiting the required properties of strength and mass is made in the model by entering the specific properties into the software. FEA does not optimize designs, but it does allow the designer to see the results of his/her work immediately, and experienced users develop a feel for the system which allows them to quickly hone in on an optimized design. The two basic elements, shapes and materials, that a designer deals with are presented below.

DESIGN SHAPES

Four of the basic design elements available to a designer for quick evaluation are thrust cones, struts and tubes, panels, and rings. The thrust cone comprises the main structure of the spacecraft about which everything else is situated. Struts and tubes provide strength for resistance of compressive forces. Panels are used much like shelves for supporting subsystem equipment. Rings exhibit high strength due to their continuous shape and are used around the thrust cone and as supporting structure for panels braced with tubes. Designers determine how the required masses and shapes of the subsystem components can be supported properly using the available structural shapes available to them. Experience plays a big part in the use of FEA. Once a preliminary configuration is determined, it is analyzed for failure points. If the failure points still meet the required strength conditions then the design is ready to be built and tested. Based on test data the design is reiterated, analyzed, built, tested, and the process is repeated. In cases in which costs prohibit many or any models to be built, recommendation may be made that limited or no testing be performed on the actual spacecraft.

STRUCTURAL MATERIALS

In addition to utilizing proven shapes to gain strength, designers also rely on special materials to provide the characteristics necessary to support a space mission. All materials exhibit characteristics such as ductility, brittleness, creep, fatigue, crack propagation, etc. which make them either desirable or undesirable in particular situations. Ductility is a property of a material which allows it to be drawn out, but not to break. Materials with good ductility are not good for tensile strength requirements since they may elongate easily. Brittleness is a property of a material which causes it to snap easily when applied loads exceed material strength. Creep is the phenomena of a material, especially a metal or alloy, to stretch over time when exposed to long-term loading situations. Materials fatigue when exposed to loads which may be small when compared to material strength, but whose frequency of loading is high and continuous. Crack propagation is the property of a material to either to quickly propagate a small crack or to exhibit no appreciable change in properties as a result of small micro-cracks on the surface. Frequently used materials include metals, special alloys, composites, laminates, and honeycomb materials. Composite and laminates are excellent due to their strength to mass ratio and the fact that they can be assembled such that the strength of the member is in the exact orientation that the designer needs it to be, but they may be difficult to manufacture. Honeycomb is used mainly for panels and offers extremely high rigidity with one of the best strength to mass ratios.

8.3 GEOCOMM Structural Subsystem Design

Figure 8.1 depicts a generic flow for the design of a spacecraft structure. The level of analysis performed in this preliminary design is only a first iteration. No structural analysis software is used and a rough sketch of the basic shape of the spacecraft will suffice. The basic shape of GEOCOMM is provided in Figure 8.2 for visualization purposes.

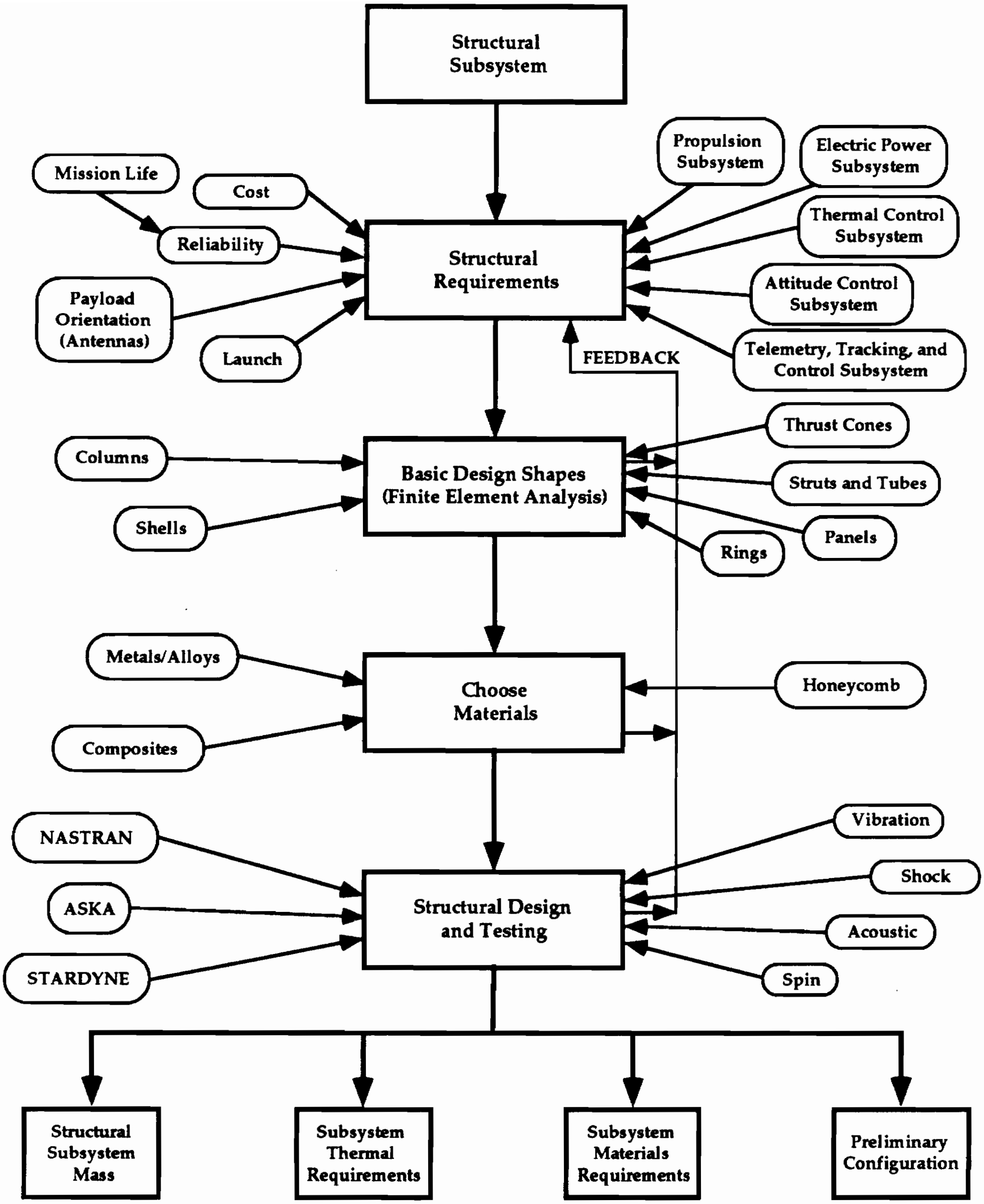


Figure 8.1 Structural Subsystem Design Flow

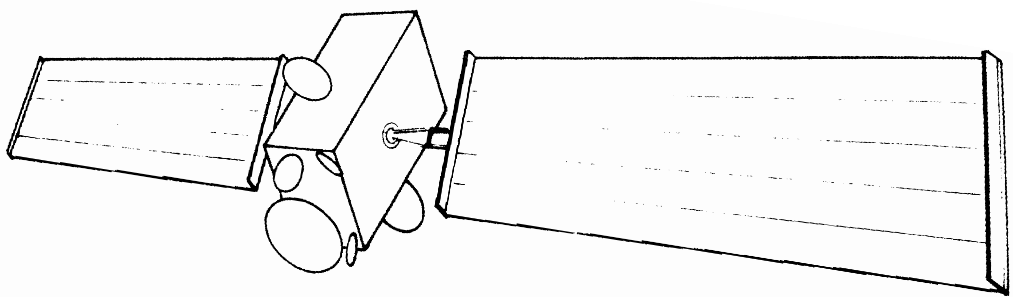


Figure 8.2 Depiction of GEOCOMM Basic Shape (See Reference #4)

9.0 Telemetry, Tracking, and Command Subsystem

9.1 TT&C Requirements

Once a spacecraft is spaceborne direct human interaction is no longer possible. If humans are to remain in contact with, monitor the health of, and effect changes in the spacecraft, advance consideration for these activities must be made. Typically, spacecraft employ a Telemetry, Tracking, and Command (TT&C) subsystem to ensure continued contact with and control of the spacecraft after launch. The TT&C subsystem combines the functions of monitoring spacecraft and payload health and conditions, providing a means for identifying the spacecraft's location in space, and ground-controlling various spacecraft devices. In addition, the TT&C subsystem usually provides all on-board computer processing unrelated to payload. The attitude control subsystem makes use of this processing extensively in its efforts to maintain pointing accuracies for the spacecraft. Figure 9.1 shows the considerations which must be made in the design of a TT&C subsystem, and the requirements that are levied on the rest of the spacecraft as a result of the TT&C design. Initial assessment of the requirements levied on the TT&C subsystem reveals that this subsystem is comprised of various sensors, control devices, orbit tracking transponder(s), computer processing equipment, and receive/transmit equipment. The various sensors associated with the TT&C

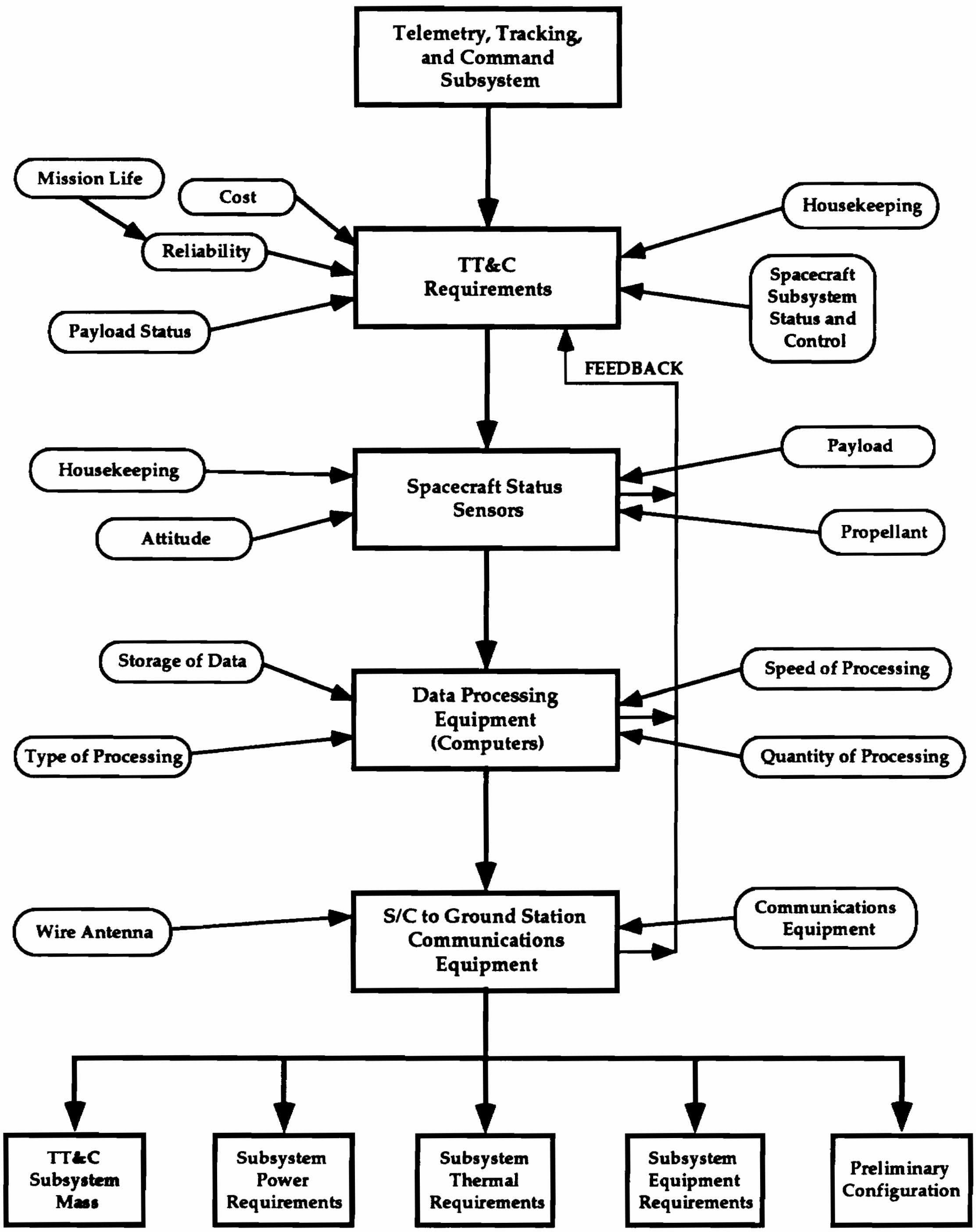


Figure 9.1 Telemetry, Tracking, and Command Subsystem Design Flow

subsystem are used to monitor such quantities as: temperatures, pressures, voltages, and currents of spacecraft equipment. Operating mode information, which includes information regarding the current operational configuration being used; whether or not a redundancy mode is being utilized; and deployment status of various equipment is also measured/detected through use of sensors. Sensors usually provide information to the TT&C computer in terms of measurable quantities of voltages. Additionally, attitude sensors must be compatible with the TT&C computer processing system for both attitude correction and downlink to the ground station. Frequently, control devices are associated with the TT&C subsystem to allow commands from the ground station to control equipment on-board the spacecraft through the TT&C computer. Basically, the TT&C computer must be capable of controlling any devices or subsystems that designers wish to have actively computer-controlled, and possibly ground-controlled. These systems are not necessarily solely ground-controlled; in fact, for the most part, these systems are spacecraft-controlled, and only ground-controlled in special circumstances.

Accurate orbital position determination is important in satellite systems requiring specific orbits. A tracking transponder is included in the TT&C subsystem to facilitate highly accurate orbital position determination.

Numerous highly accurate numerical orbit determination algorithms exist for use when sub-satellite positions are known to a given accuracy. These orbital positions are gained through use of on-board tracking transponders which downlink a signal used for spacecraft position determination.

The entire TT&C subsystem is computer-controlled. Due to the harsh radiation environment encountered in outer space, computer processing equipment must be highly protected and is very limited in its processing capability. High-order languages (Fortran, C, etc.) are not possible for use as of yet. Computer hardware is not very reliable in the space-environment, and since it is not very heavy, redundancy is a must. The TT&C computers are the computers used by the attitude control subsystem.

If the spacecraft is to communicate with an earth station for direction and control, there must be a data link between the two. Since it is impractical to connect a line between the spacecraft and the ground station, although this has been proposed (Martin Marietta, Denver CO), spacecraft communications are handled through atmospheric means. Spacecraft are equipped with antenna(s) used expressly for command and control information flow between the spacecraft and the ground station. Redundancy in the antennas is facilitated through the communications antennas. Wiring provisions must be designed into both subsystems in order for this redundancy to be realized.

9.2 GEOCOMM TT&C Subsystem Design

The various sensors and control devices required for GEOCOMM are shown in Figure 9.2. The operating status and redundancy status sensors present for each subsystem denote the current equipment configuration of the spacecraft. GEOCOMM's TT&C subsystem also employs a single tracking transponder for ground orbit determination. As mentioned above, computer equipment is not very reliable for space applications. Assuming that each CPU has a life expectancy of 120,000 hours, the number of CPUs that much be employed to satisfy lifetime requirements is

$$\lambda = \text{reliability of CPU} = 1/120000 = 8.33 \times 10^{-6} \tag{9-1}$$

$$R = e^{-\lambda t} = e\text{-}(8.33 \times 10^{-6} * 87600) = 0.482 \tag{9-2}$$

$$R_{system} = 1 - (1 - R)^n \quad \text{Where reliability of system is} > .90 \tag{9-3}$$

$$n \approx 4 \text{ for assurance of about 93\% reliability.} \tag{9-4}$$

Four CPUs are utilized on GEOCOMM in parallel redundancy to ensure 93% reliability for 10 years.

• TT&C Subsystem

Subsystems	Sensors	Control Devices
• Communications	• TWTA temperature • Repeater temperature • Signal levels • Operating status • Redundancy status • Deployment of antennas	• Change Operating configuration/ status • Change antenna pointing • Change signal levels/frequencies • On/off control
• Launch Vehicle	• Proper separation • Operating status • Redundancy status	• Pyrotechnic devices • Deployment motor firing
• Propulsion	• Bi-propellant temperature • Bi-propellant pressure • Thruster temperatures • Operating status • Redundancy status	• Thruster firings • Propellant heaters
• Attitude Control	• Attitude sensors - Sun sensor - RF sensor - Gyroscope • Thruster temperarures • Momentum wheel speed • Operating status • Redundancy status	• Thruster firings • On/off control of attitude sensors • Uplink of new attitude control software/algortihms • Spin-up momemtum devices • Momentum control
• Electric Power	• Voltage • Current • Solar array deployment • Array power • Battery charge • Operating status • Redundancy status	• Solar array pointing • Battery charge/discharge (reconditioning) • Power control electronics
• Thermal Control	• Blackbody temperatures • Spacecraft temperature • Operating status • Redundancy status	• Control of all heaters
• Structural	• Structural temperatures • Structural stresses/strains • Deployment of extremities	• Solar array tracking devices • Antenna pointing devices

Figure 9.2 GEOCOMM Sensors and Control Devices

Finally the communications portion of the TT&C subsystem makes use of a communications transponder and a simple wire antenna. Redundancy is wired through the communications antennas. In fact, on orbit some of the communications channels provided by the spacecraft actually act as the TT&C subsystem link and the designed TT&C link is used as the redundant channel.

10.0 GEOCOMM Summary

Preliminary design of GEOCOMM is complete, this being just the first of a series of iterations. Based on the findings in the preliminary design, a decision may be made to proceed with full-scale development of GEOCOMM. If full-scale development is desired, each of the steps performed in the preliminary design must be reiterated. Each subsequent iteration results in greater accuracy of design and uncovers additional requirements. Referring back to Figure 1.4, this paper addressed only a portion of the overall flow presented in the figure, just once through the spacecraft hierarchy. Subsystem and spacecraft testing is not covered, but an entire text could be devoted to the subject. The overall goal of the project was to present a limited-scope generic design flow for the development of a communications spacecraft. Actual development effort requires a minimum of 25-40 person-years of effort for even the most limited of spacecraft. Therefore, specific details concerning cost, effort, and detailed design characteristics of the system were omitted in the interest of time.

11.0 Recommendations

As a result of completing this project a number of additional topics and recommendations may be made to others attempting a similar activity. The design of an entire spacecraft may be too large an effort to complete in a reasonable (1 year) time period with desired accuracy and detail. Each portion of the systems engineering process could be analyzed in detail, each as a project in itself, for the design of a spacecraft. Requirements analysis, both functional and design, for each of the subsystems is a significant effort worthy of projects in themselves. The testing processes and their traceability to the initial requirements would serve as a good exercise to anyone wishing to learn more about spacecraft unit and integrated testing. The best advice that may be given as a result of completing this project is not to attempt too large a project. The magnitude of this project became apparent quickly and initial plans for detail were scrapped in lieu of finishing the project.

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