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Technical Support for Defining
Advanced Satellite Systems Concepts

Final Report for Task Order 7:
MASS and POWER MODELING of COMMUNICATION SATELLITES

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Summary

Analytical models are developed to estimate subsystem mass and power requirements of communication satellites. The models are useful for making rapid evaluations of system scenarios in situations where a detailed analysis would be too resource consuming. A set of analytical or tabular relations are given which link the mass and power requirements of subsystems to their respective key driving parameters. These models are based on sound engineering principles, and are calibrated or verified with available data.

Modeling tasks are performed for seven satellite subsystems or areas: (1) power; (2) attitude control; (3) structure; (4) payload (multiple beam antennas and baseband electronics); (5) telemetry, command, and control; (6) miscellaneous factors (such as integration and contingencies); and (7) launch and on-orbit propulsion subsystems.

The estimating relationships are developed by identifying the major subsystem components. For each of these identified components, the performance drivers and system requirements that influence their selection are used to develop estimating relationships for mass and power. Once each of the equations is developed, a systematic implementation is recommended to arrive at which equation or technology to utilize. Figures are given as implementation guides which flow down system requirements, subsystem component options, and selection criteria.

The conclusions of this work are summarized in Section 1.2 of Chapter 1, Executive Summary.

- For the power subsystem, the model estimates parameters for the solar array, positioning mechanism and support structure, batteries, power processing unit, and thermal control.
- For the attitude control subsystem, 3-axis, single-axis spin, and dual spin pointing and orientation are considered with respect to stabilization reference, sensors, reaction control systems, and fuel requirements.
- Structural sizing and selection guidelines are given.
- For the payload, multiple beam antenna and on-board baseband electronics mass and power consumption are estimated.
- TT&C selection guidelines are given and sizing is estimated.
- Electrical integration, mechanical integration, and mass contingency sizing assumptions and estimates are given.
- Launch and on-orbit propulsion sizing and selection guidelines are given. Various propulsion technologies are compared and system sizing charts are presented.

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Chapter 1

Executive Summary

This chapter is organized as follows:

1.1 Background
1.2 Summary of Work
1.3 Potential Enhancements
1.4 Organization of Report

1.1 Background

NASA, Lewis Research Center, (NASA/LeRC) makes use of analytical models to estimate subsystem mass and power requirements of communication satellites. These are useful for making rapid evaluations of system scenarios as detailed analysis would be too resource consuming.

NASA/LeRC's current models are in need of update. What is needed is a set of analytical or tabular relations which link the mass and power requirements of subsystems to their respective key driving parameters. For example, solar array area (exclusive of orientation and support structure) varies directly with needed power, but depends on technology (thin cell Si, GaAs, etc.) in a discrete way. Similarly, key driving parameters can be identified for other subsystems and models constructed. To the extent possible, these models should be based on sound engineering principles, and should be calibrated or verified with available data.

1.2 Summary of Work

How the Estimating Relationships Were Developed

Each of the satellite subsystems were identified in the work breakdown structure along with major subsystem components. For each of these identified components, the performance drivers and system requirements that influence their selection have been used to develop estimating relationships for mass and power.

Figure 1-1 lists the three main mission assumptions and the system sizing impacts.

Table 1-1 lists the derived system requirements and subsystem performance drivers as a function of subsystem. These drivers are then used as independent variables in the development of the mass and power equations.

How The Model is Laid Out

Once each of the equations have been developed, a systematic implementation has been recommended to arrive at which equation or technology to utilize. Figure 1-2 represents an implementation guide which flows down system requirements, subsystem component options, and selection criteria. There is always a risk when making top-level subsystem component selection based on mass that many other tradeoff issues may drive the selection.

Figure 1-3 indicates other significant variables that influence component selections, such as cost, risk, schedule, operations, heritage, and life requirements. In using a performance-based mass model for first cut system sizing, one should assess these other issues at least qualitatively.

Modeling Tasks

Modeling tasks are performed for seven satellite subsystems or areas. The work is described in the next seven subsections.

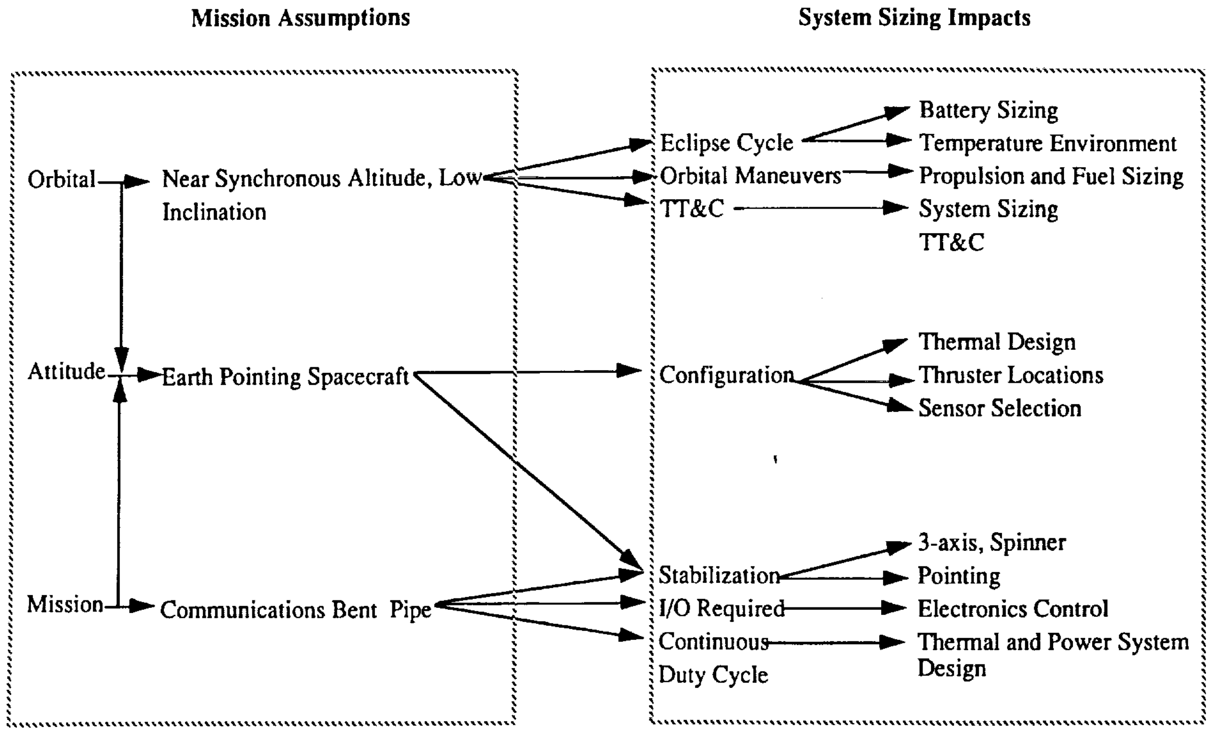


Figure 1–1: Mission Assumptions and System Sizing Impacts

Table 1–1: Performance Drivers as a Function of Subsystem

Subsystem	System Level Drivers	Subsystem Level Drivers
Power	End-of-life power Mission life Attitude stabilization	Array power capacity Eclipse power
Attitude Control (ACS)	Life Array capacity	Nominal boresight pointing accuracy Maximum cyclic disturbance torque
Propulsion	Life Dry mass Launch site Attitude stabilization	Reposition maneuvers
Structure	Dry mass	
Payload	Operating frequency Communications capacity	Aperture diameter Number of feed elements
TT&C	Spacecraft command, control Complexity Operating frequency	
Thermal, Integration, Contingency.	Dry mass End-of-life power	

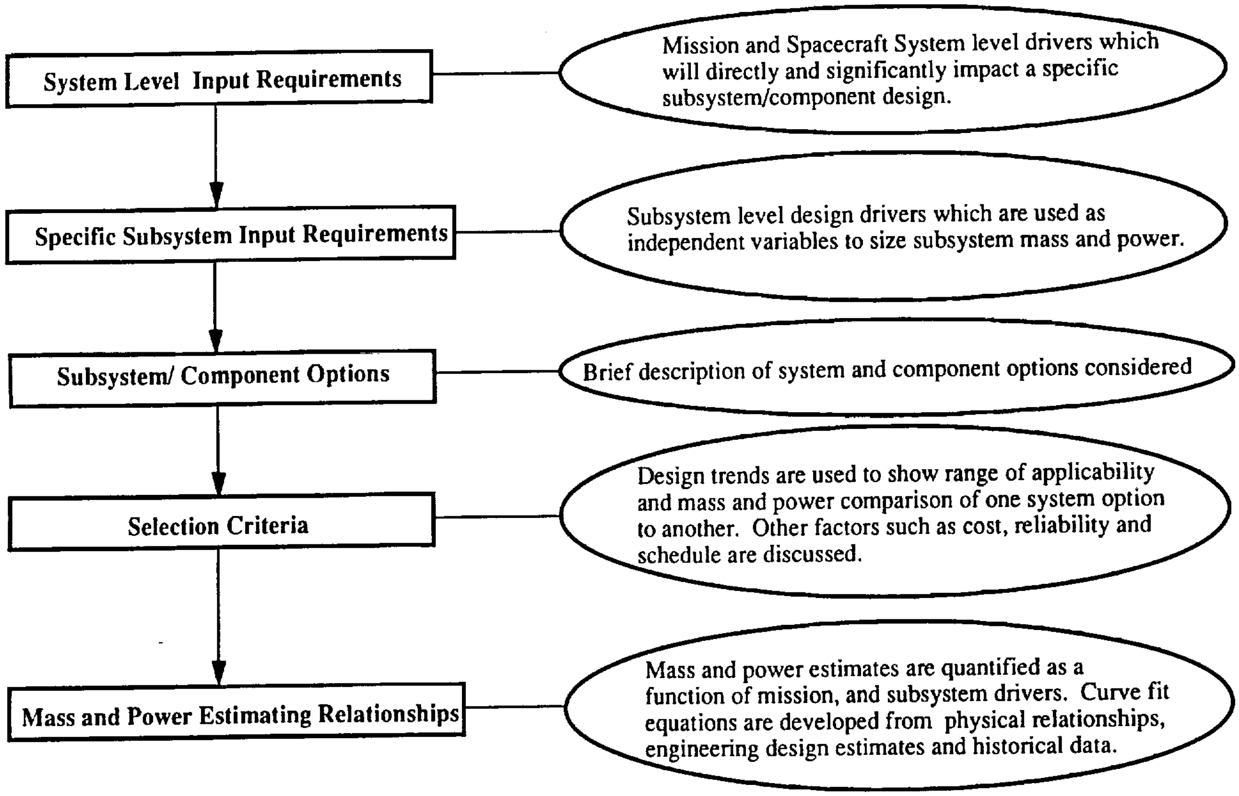


Figure 1–2: Implementation Guide to Model

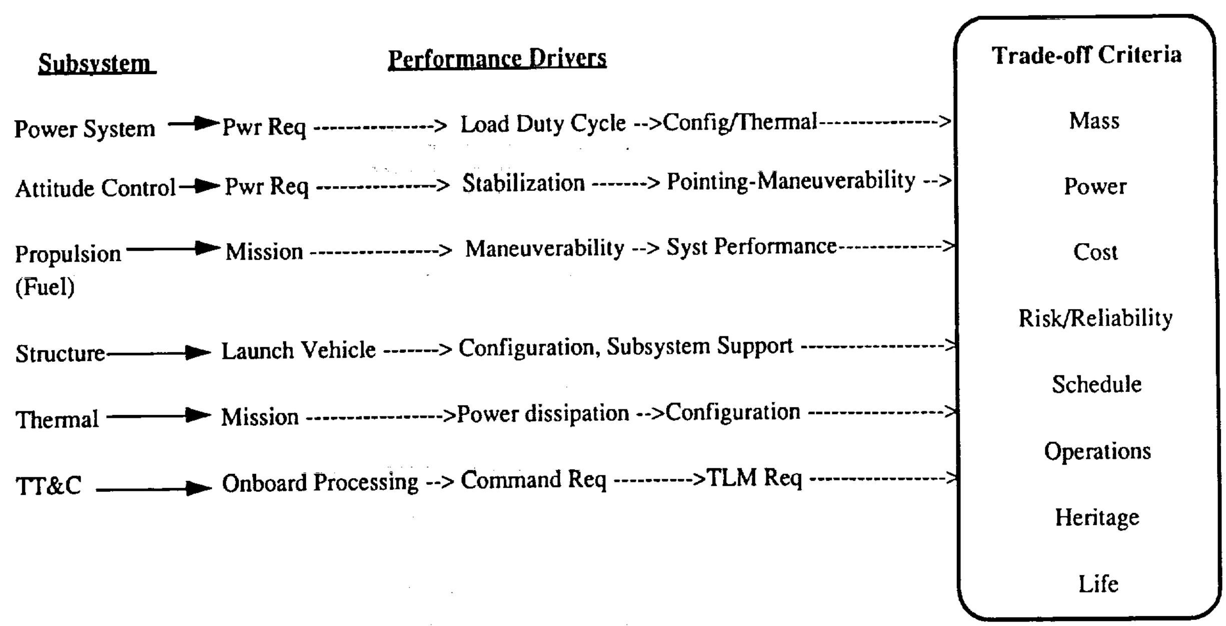


Figure 1–3: Subsystem Drivers and Tradeoff Criteria

1.2.1 Power Subsystem
1.2.2 Satellite Positioning and Orientation
1.2.3 Structure
1.2.4 Payload
1.2.5 Tracking, Telemetry, and Control
1.2.6 Miscellaneous
1.2.7 Launch

1.2.1 Power Subsystem

An analytical model is developed which includes an accounting for the following:

1. **Array** — The model estimates array mass and area as a function of power delivered. It includes the effects of design life and accounts for various technologies including panels, blankets, cylinders (spin satellites) for both Si and GaAs.

   Figure 1-4 is a summary of the solar array sizing and selection guidelines. Various solar cell and solar panel technologies are compared, and system sizing charts are presented.

2. **Positioning Mechanism, Power Transfer, and Support Structure** — The model estimates mass and power as a function of power delivered.

3. **Batteries** — The model estimates mass, volume, and charging requirements as a function of energy or power delivered. It includes the effects of design life; and accounts for various technologies including NiCad, NiH, NaS.

4. **Power Processing Unit** — The model estimates mass power, and efficiency as a function of power delivered and includes the effects of design life, if significant as well as battery technology, if significant.

5. **Thermal Control** — The model estimates mass requirements for power subsystem thermal control as a function of delivered power.

1.2.2 Satellite Positioning and Orientation

Analytical models are developed for 3-axis, single-axis spin, and dual spin pointing and orientation subsystems. Figure 1-5 is a summary of the attitude control sizing and selection guidelines. Various sensor and actuator technologies are compared, and system sizing charts are presented. The models include an accounting for the following:

1. **Stabilization Reference** — Satellites have made use of mass references in momentum wheels; earth's magnetic field in magnetic torquers; and gravity gradient stabilization. Analytical methods are provided for each of these methods with appropriate guides as to when each should be applied.

2. **Sensors** — Depending on required pointing accuracy, a variety of reference (earth, sun, star, etc.) sensors could be used. Analytical models are provided for key sensors used today, and guides provided as to when each should be applied.

3. **Reaction Control System** (RCS) — A variety of thrusting systems (monopropellant, pressurized gas, bipropellant, electric, etc.) could be used. Analytical models are provided for key modern systems, and guides provided as to when each should be applied, if possible (the tradeoff for electric propulsion versus conventional is complex and simple guides might not be possible). Tankage, lines, pressurant, and power processing (when necessary) are taken into account.

4. **Fuel Requirements** — Models are provided to estimate the fuel requirements for all of the above RCS subsystems. The models account for technology, life, attitude accuracy, and position keeping accuracy.

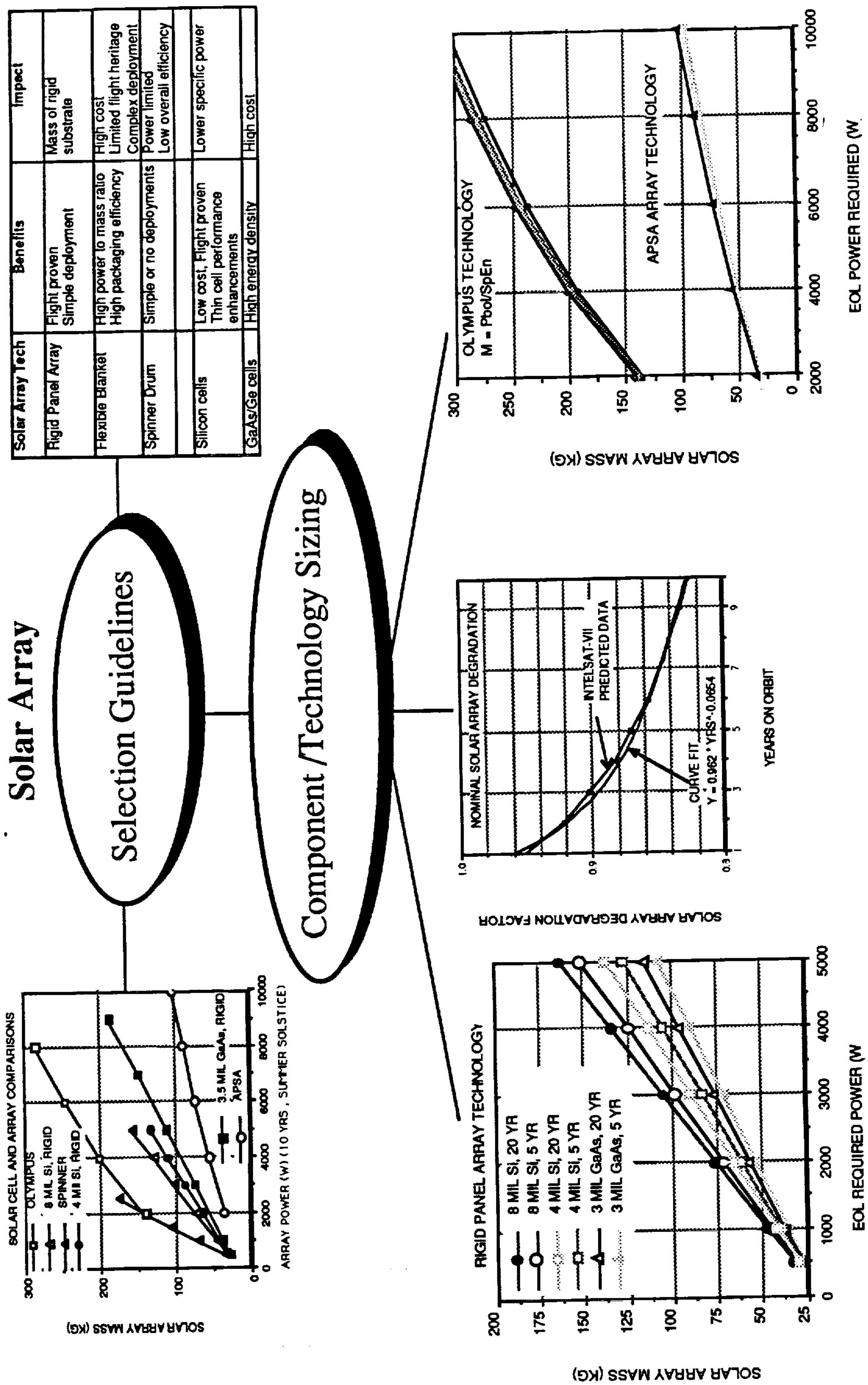
1.2.3 Structure

The structure includes support surfaces and internal framework necessary for attachment of all subsystems and the payload. It excludes external support structures for the power system and the antenna subsystems. The model accounts for conventional metallic honeycomb materials as well as modern composites.

Figure 1-6 is a summary of the structural sizing and selection guidelines. A percentage of the level of advanced composites utilized as a function of spacecraft dry mass is presented.

1.2.4 Payload

The payload includes all items needed to accomplish the basic communications mission. However, for this subtask, only antennas and on-board baseband electronics are required to be modeled. The antenna models accounts for multibeam designs, including all support structures and positioning equipment, and account for



Solar Array Tech	Benefits	Impact
Rigid Panel Array	Flight proven Simple deployment	Mass of rigid substrate
Flexible Blanket	High power to mass ratio High packaging efficiency	High cost Limited flight heritage Complex deployment
Spinner Drum	Simple or no deployments	Power limited Low overall efficiency
Silicon cells	Low cost, Flight proven Thin cell performance enhancements	Lower specific power
GaAs/Ge cells	High energy density	High cost

Figure 1–4: Solar Array Sizing and Selection Guidelines

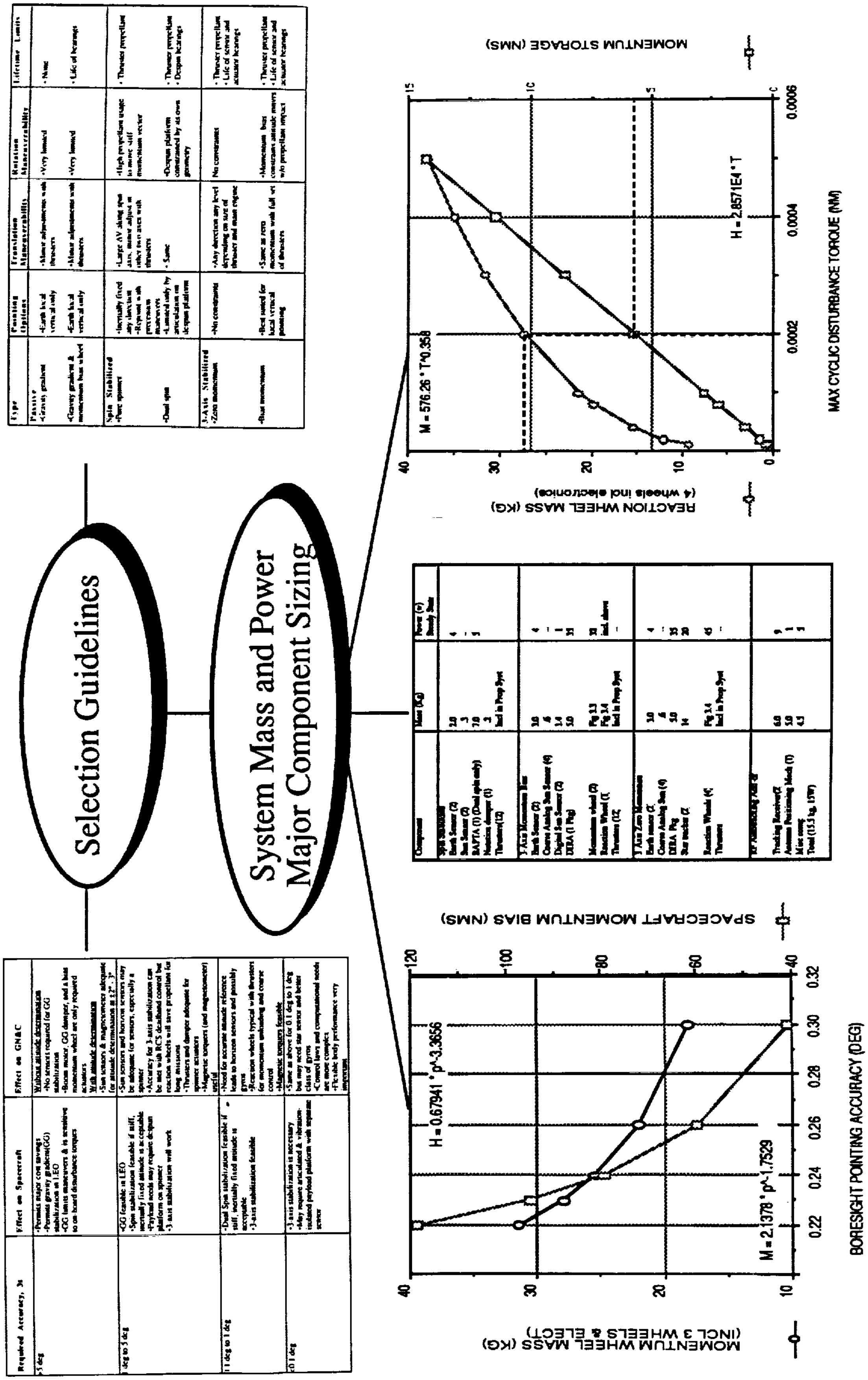


Figure 1–5: Attitude Control Sizing and Selection Guidelines

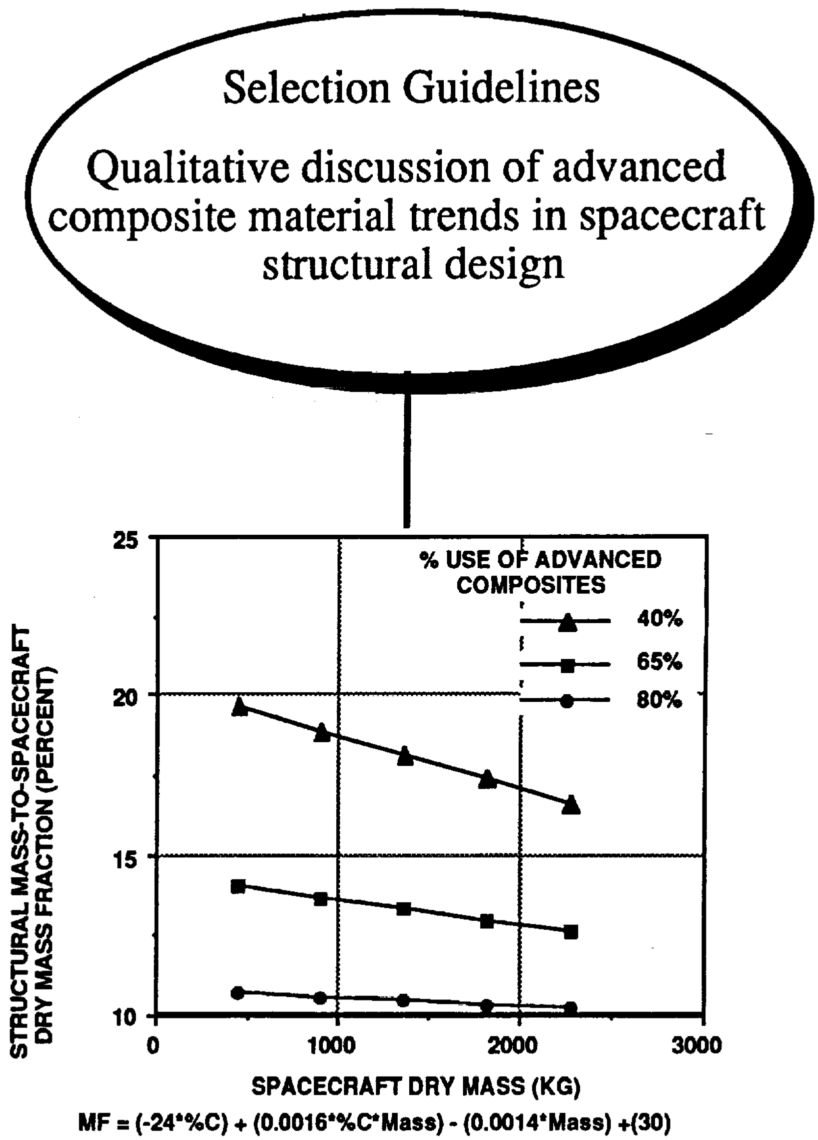


Figure 1–6: Structural Sizing and Selection Guidelines

differences in design (rigid, segmented, mesh, etc.). On-board baseband electronics models include baseband processing and switching.

Figure 1-7 is a summary of the payload sizing and selection guidelines. Various antenna and baseband electronics technologies are compared, and system sizing charts are presented.

1.2.5 TT&C

The model includes an accounting for the following Tracking, Telemetry, and Control (TT&C) functions:

1. Command processing and subsystem monitoring equipment.

2. Accessory RF sensors needed for tracking purposes.

3. Communications equipment and antennas.

Figure 1-8 is a summary of the Telemetry, Tracking and Command sizing and selection guidelines. Three levels of spacecraft complexities are qualitatively discussed and mass and power estimates are generated.

1.2.6 Miscellaneous

A few miscellaneous items have small impact, but are necessarily accounted for. Often these items are estimated by simply adding a percentage of the total satellite active systems. These models account for:

1. **Electrical Integration** — Wiring or other means used to connect power and signals to and from subsystems.

2. **Mechanical Integration** — Mechanical fittings and other means for attaching subsystems to the basic structure.

3. **Contingency** — Guides on mass contingency to be included to account for uncertainties in the above mass estimates.

Table 1-2 lists the sizing assumptions and estimates for other miscellaneous spacecraft subsystems such as thermal control, mechanical and electrical integration, and typical mass and power contingency values.

1.2.7 Launch

The models account for technology, tankage, fuel and support equipment including solid, integrated bipropellant, and hybrid launch subsystems.

Figure 1-9 is a summary of the launch and onboard propulsion sizing and selection guidelines. Various propulsion technologies are compared, and system sizing charts are presented.

1.3 Potential Enhancements

Potential Enhancements There are many areas of improvement that could make this model more accurate and more robust. Listed below is a list of recommendations for future work.

Attitude Control
- Magnetic torquer sizing algorithm.
- Gyro mass and power sizing as a function of technology and performance.
- Spin stabilized spacecraft: single spin, dual spin, fixed and deployable skirt solar arrays.

Propulsion
- Simple monopropellant system with solid apogee motor.
- Electric propulsion sizing for orbit transfer and for GEO insertion.

Electric Power
- Power processing electronics.
- Sodium Sulfur (NaS) battery mass and thermal control system impacts.
- Concentrator solar array technology.

Thermal Control
- Thermal control system sizing as a function of heat dissipation and component technology.

1.4 Organization of Report

Table 1-3 gives the organization of this Final Report by chapter. Chapters 2 through 8 discuss in turn the seven modeling subtasks. The Subtask 2 work on attitude control is discussed in Chapter 3, Attitude Control, but the Subtask 2 work on the reaction control system is discussed in Chapter 8, Launch and On-Orbit Propulsion.

Appendix 1 gives propulsion system selection guidance charts. Appendix 2 gives an example calculation of propulsion system mass for the GOES-I spacecraft.

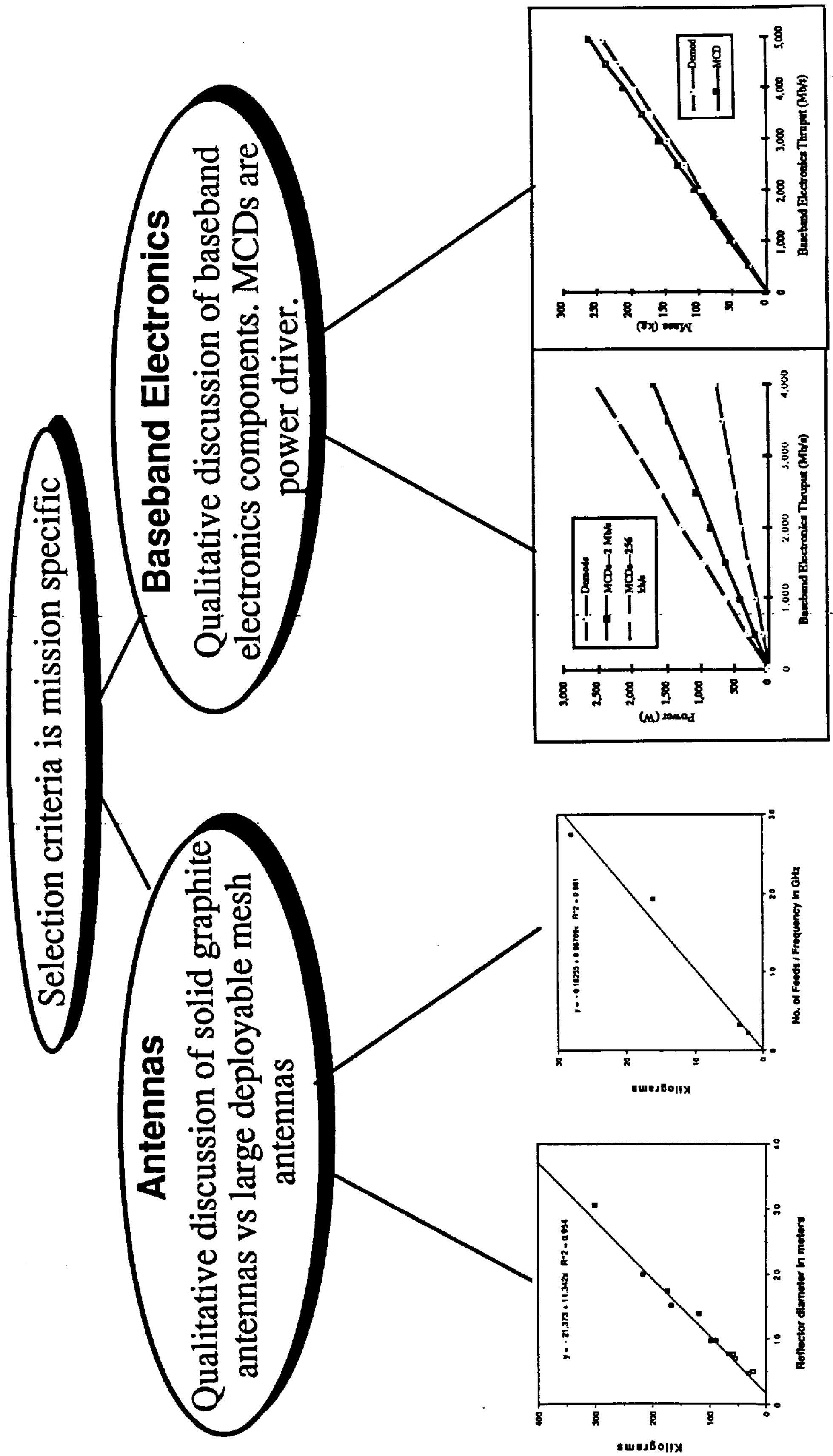


Figure 1–7: Payload Sizing and Selection Guidelines

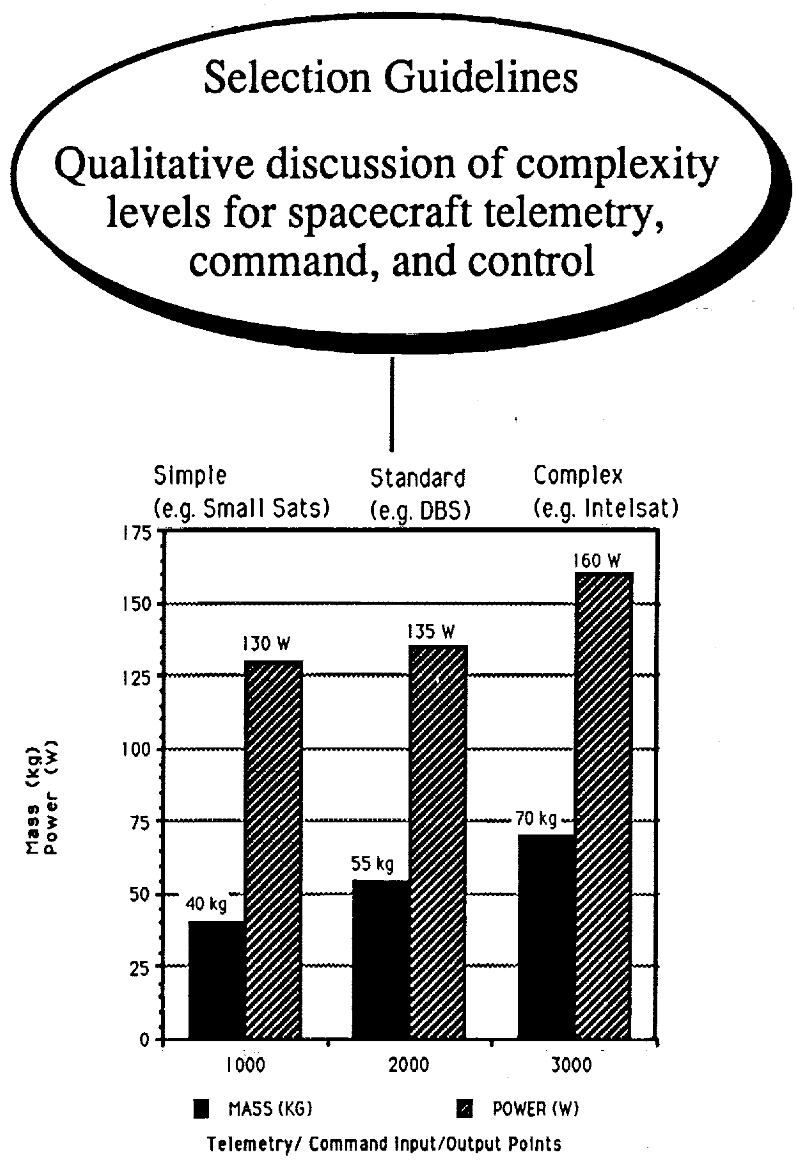


Figure 1–8: Telemetry, Tracking and Command (TT&C) Sizing and Selection Guidelines

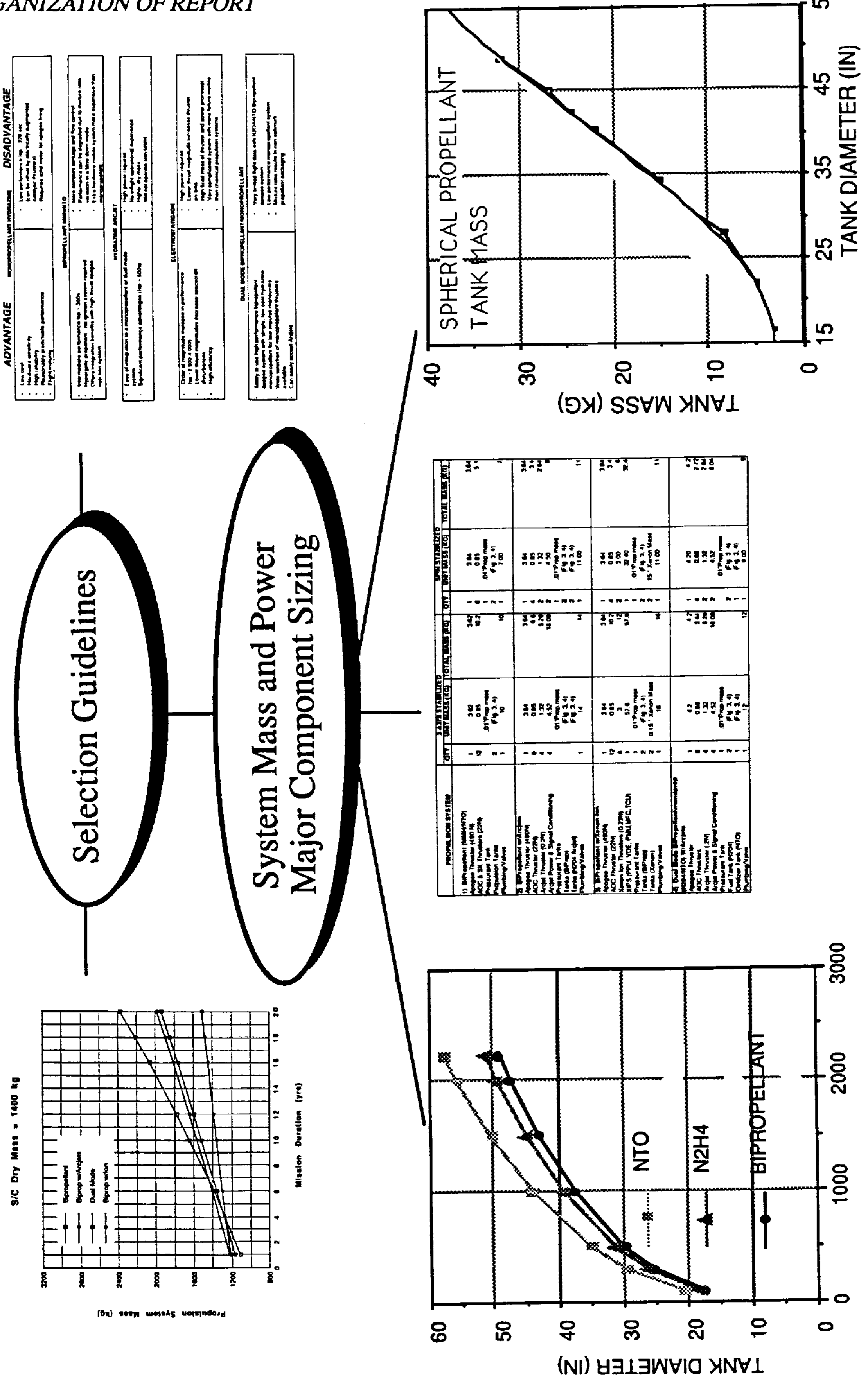


Figure 1–9: Launch and Onboard Propulsion Sizing and Selection Guidelines

Table 1–2: Sizing Assumptions and Estimates for Other Miscellaneous Spacecraft Subsystem

Category	Basis of Estimate
Mechanical Integration Mass	3% of the spacecraft dry mass
Electrical Integration Mass	4.5% of the spacecraft dry mass
Thermal Control System Mass	0.0295 times end-of-life power
Thermal Control System Power	0.0546 times end-of-life power
Mass Contingency	8% of the spacecraft dry mass
Power Contingency	5% of the end-of-life power

Table 1–3: Organization of Report

Chapter	Contents
1.	Executive Summary
2.	Power Subsystem Modeling
3.	Attitude Control
4.	Structure Modeling
5.	Payload Modeling
6.	TT&C Modeling
7.	Miscellaneous Item Modeling
8.	Launch & On-Orbit Propulsion
A1.	Propulsion System Selection Charts
A2.	Example – GOES-I Spacecraft

Chapter 2

Power Subsystem

2.1 Introduction

Power subsystem sizing task describes the power system selection and sizing considerations for a communications satellite designed for a geosynchronous orbit. Analytical models are developed to size component mass and power. Also, a methodology has been developed to assist the user in choosing an appropriate system based on top level mission parameters.

Modelling architecture layout. Figure 2-1 is a logic chart which describes the process including input data, performance requirements, decision points, system selection guides, and mass sizing tables and charts necessary to generate a system selection and estimates of mass and power.

Communications satellite assumptions used to generate system architectures: The most significant mass drivers in a power system are the solar array and battery. The harness mass is estimated in the electrical integration section. At a conceptual design stage it is reasonable to neglect detailed design parameters when those variables do not significantly affect the system level mass and power. However, it is a risk when sizing for only mass and power that important issues may also be overlooked which can significantly affect performance, reliability, and operations. Therefore, the flexibility has been constrained to minimize the chance for such an occurrence.

Data was gathered from existing flight programs and from technology development programs, with the constraint that the mission is typical of an Earth pointing, near-equatorial, geosynchronous satellite. The results become less accurate as either inclination or off-nadir attitude maneuvering is increased. Also, since the battery depth of discharge was not treated as a variable, the battery sizing is accurate for typical geosynchronous eclipse duty cycles. The battery design is typically robust so that greater duty cycles can be accommodated. However, the battery designs would not be acceptable for low Earth orbit eclipse duty cycles.

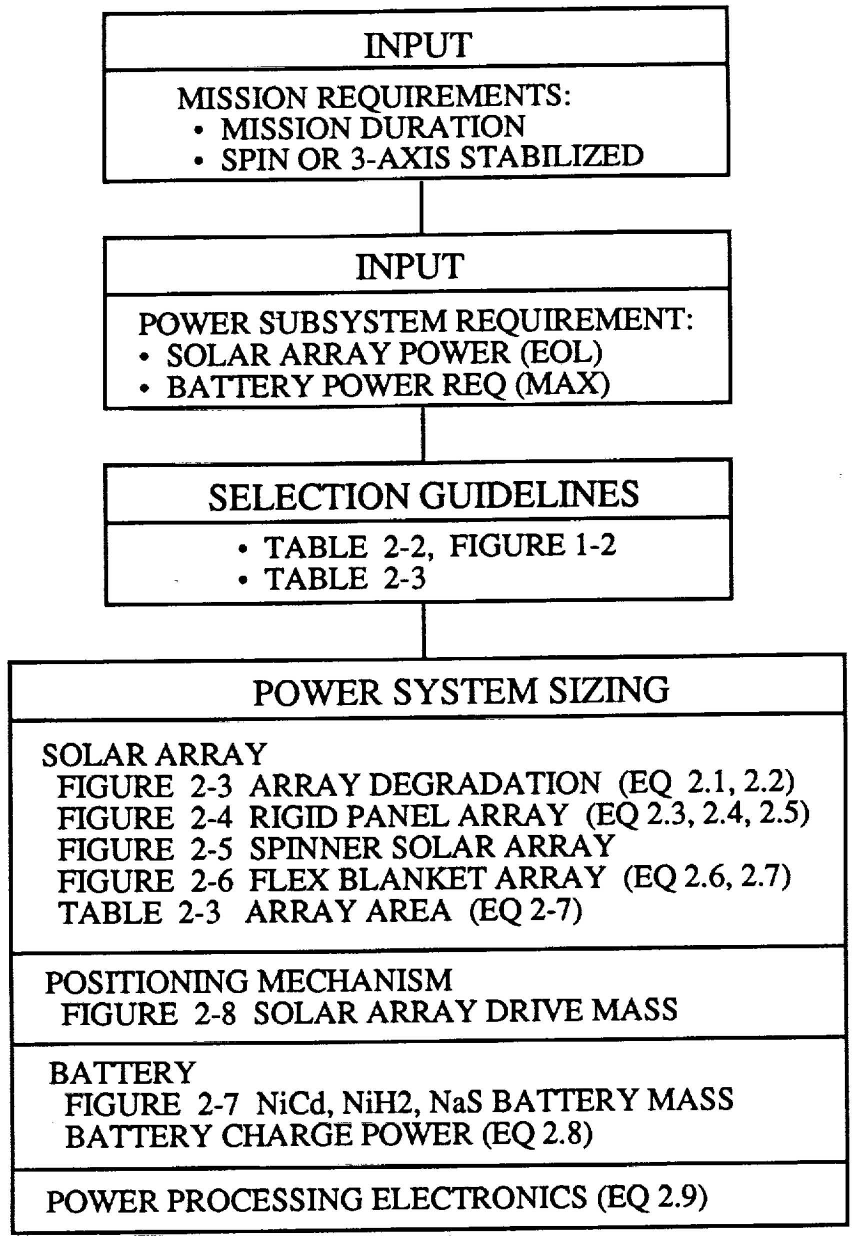


Figure 2-1: Power System Sizing Diagram

Regulated bus and voltage assumptions. The selection of a power management and distribution topology requires very specific trade-offs with very specific requirements. System level trade-offs include fully and partially regulated power bus, charge arrays vs battery charge converters, voltage regulation (28 V, 42 V, 50 V, 120 V), etc.. These architectural issues are critical to an optimal power system design which takes into account mass, overall efficiency, redundancy, reliability, other subsystem impacts, and packaging integration. However, the emphasis of this model is on conceptual level mass/power modelling. Therefore, the analytical expressions generated are based on power system mass drivers which may not fully characterize and optimize the power system.

The power system is assumed to be fully regulated during both sunlight and eclipse periods. By regulating the bus during eclipse, rather than operating directly from the degrading battery voltage, a more efficient power system can be designed. For example, the dc-dc converters that regulate the voltage for the user equipment and power amplifiers can be designed more efficiently when the input voltage from the bus is regulated within a smaller range. The improved efficiency has a total systems benefit by decreasing solar array requirements and thermal rejection requirements.

Another system benefit of the fully regulated bus is that the battery design can be based on a system optimization rather than the series combination of battery cells required to match the solar array voltage at the start of eclipse. The mass and cost can be minimized and the efficiency maximized by designing the battery to provide the required watt-hours at a lower voltage and higher capacity, and then regulate to the design bus voltage.

2.2 Mission Requirements

The following mission level requirements are required:

Mission duration: The lifetime at which a certain probability of success is achieved while delivering the required minimum level of power to the user equipment.

Attitude stabilization: The method of attitude stabilization, spin stabilized or 3-axis stabilized must be specified to determine the solar array configuration.

2.3 Power System Requirements

The following subsystem level input data must be specified:

Solar Array Power : The average steady state power required to satisfy the mission. This is considered the end-of-life power requirement.

Battery Power: The maximum power required during eclipse periods, or the maximum watt-hours required by the battery.

2.4 Major Power System Component Descriptions

2.4.1 Power Generation

This study has restricted power generation technologies to Si and GaAs solar cells mounted on rigid panels and flexible blanket solar arrays. The following solar cell/array technologies have been considered:

Options	Cell Technology	Array Technology
1	Si, 8 mil	Deployable panel
2	Si, 4 mil	Deployable panel
3	Si, 4 mil	Spinner
4	GaAs/Ge, 4 mil	Deployable panel
5	Si, 2 mil	Flexible blanket, APSA program technology
6	Si, 2 mil	Flexible blanket, Olympus satellite program

The Si cell technology represents a state-of-the-art cell, but is continually being improved to improve the specific power (specific power = Watts per kilogram). The 8 mil cell has typically been used, but, the thinner cells are seeing a wider application. The 2 mil cell is being used for blanket-type solar arrays with specific power levels approaching 150 W/kg. The latest advancements in GaAs solar cell technology is the increased strength and mass reduction by using germanium substrate. However, even with this new benefit, the cost per watt for a given mass is still many times higher than silicon.

The three different array technologies included are the rigid panel and flexible blanket technology for a deployable solar array, and the body mounted cylinder for a spin stabilized spacecraft. The rigid panel assumes a composite substrate layup with an aluminum honeycomb core. Two technologies of foldout flexible blanket arrays are considered; the Advanced Photovoltaic Solar Array Program (APSA), and the Olympus Spacecraft Program. The Olympus spacecraft program has flown a foldout flexible solar array in GEO, and the APSA is a NASA technology program to attain a BOL solar array mass goal of 300 W/kg.

The body mounted solar array used for a spin stabilized spacecraft consists of a cylindrical shell shaped substrate which makes up the exterior of the satellite. Solar cells are bonded to the outer surface and spin with the rotating section. Specific performance of a spinner drum array is much lower than a deployed array because a flat panel array always is oriented towards the sun. With the body mounted drum array, many more solar cells are required to generate a constant power source while the spacecraft spins.

2.4.2 Power Storage

Power storage for satellite applications is typically a rechargeable battery technology, and is used to augment the solar array during eclipse and for peak power loads. NiCd and NiH2 batteries have been used almost exclusively. The NiCd technology has the most flight heritage, and it operates by the conversion of chemical to electrical energy. The cadmium negative electrode is oxidized during discharge which gives up electrons to the positive nickel electrode through an aqueous electrolyte solution. The NiH2 technology utilizes hydrogen gas and a catalyst material, platinum, as the negative electrode to achieve electrochemical reaction.

The NiH2 battery is not affected by the same time and temperature degradation factors which limit the life of a NiCd battery, and therefore, the NiH2 battery has considerably longer operational life at higher depths of discharge than NiCd batteries. The inconel pressure vessel that contains the hydrogen gas of a NiH2 battery is a significant mass driver. A technology initiative to consolidate the NiH2 cells into a common pressure vessel has the potential to improve the specific mass of a NiH2 battery, named a common pressure vessel (CPV) NiH2 battery. A mass estimating relationship for this technology has been included as a comparison to demonstrate the potential advantages of this near term technology

Another developing battery technology is Sodium Sulfur (NaS). The high energy density of this technology has pushed development towards a Space Shuttle flight experiment. However, the operating temperature of this battery (approx. 300° C) has created major spacecraft integration issues to isolate the battery from the rest of the spacecraft equipment.

2.4.3 Positioning Mechanism

The solar array drive positioning mechanism provides mechanical support for the deployed solar array, positioning of the wing about a single axis for sun orientation, and transfer of power and control signals between the wing and the main body. The mechanism is usually of the geared stepper motor type with two independent motor windings and redundant electronics to provide the step positioning function.

2.4.4 Power Processing Unit

The primary voltage regulation and distribution system consists of a power control unit and solar array shunt units. The power electronics estimating expressions also makes provisions for pyro actuation control and various dc/dc converters.

The power control unit is the principal element for management and control of spacecraft primary power. The primary function is to integrate power from the solar array and battery into a consolidated main bus, regulate the bus voltage during both sunlight and eclipse operations, and distribute this power to the spacecraft. The key features of the power control unit include; charge/discharge converters to regulate the voltage during battery operation and to regulate the battery charge current during sunlight operation, combine solar array and battery power sources, compare power demand loads with available power and send error amplifier signals to control the shunt.

The sequential shunt units connect the solar array to the main bus through a series of diodes and shunt transistors. When a transistor is in the shunt state, the diode is reversed biased, effectively removing that solar array circuit from the bus. The number of solar array circuits is a function of power level, control granularity, and mass. A typical solar array design is sized assuming one circuit failed.

2.4.5 Thermal Control

Thermal control modelling for the power subsystem is an integral part of each component design. For example, thermal control of the battery is accomplished by conducting the waste heat through the battery support sleeves or support plate directly to the radiator panel. Heaters are mounted directly to the battery cells to maintain minimum battery temperature. This aspect of the battery thermal design is included in the mass estimating relationships. Additional thermal control elements such as multi-layer insulation, shields, and optical solar reflectors are considered part of the spacecraft thermal design. This approach is similar for other power system components, eg., individual power conditioning units mass is included in the estimating equations and the spacecraft thermal control system transports and radiates the waste heat to space.

2.5 Guidelines for the Selection of Power System Technologies

There are many factors that influence the selection of power system technologies. Some of these include:

- System development and qualification costs,
- Recurring costs,
- Spacecraft integration
- Launch vehicle constraints
- Launch, Transfer and on-orbit operations
- Overall system reliability and risk,
- Structure, power and thermal subsystem implications,

These general considerations are summarized in the **Table 2-1** which lists the advantages and disadvantages of each of the candidate system. Even though these considerations weigh heavily in the selection process, the bottom line is usually cost and risk considerations. If a satellite can be launched on a smaller launch vehicle because of higher performance and lower weight satellite subsystems, then a cost savings generally results. But, if there is a significant risk to development and/or schedule, the cost of risk may far exceed the launch related savings.

Table 2-1: Solar Array Technology Benefits and Impacts

Solar Array Technol.	Benefits	Impact
Rigid Panel Array	Flight proven Simple deployment	Mass of rigid substrate
Flexible Blanket	High power to mass ratio High packaging efficiency	High cost Limited flight heritage Complex deployment
Spinner Drum	Simple or no deployments	Power limited Low overall efficiency
Silicon cells	Low cost, Flight proven Thin cell performance enhancements	Lower specific power
GaAs/Ge cells	High energy density	High cost

Figure 2-2 plots a mass comparison of solar array mass as a function of EOL power required. The range of power levels plotted for each technology is consistent with existing configurations and current projections. The projections and existing data for the APSA program indicate that this technology will always be lighter over all power ranges. However, the APSA mass benefits decrease over the rigid panel arrays at lower power levels. And when a system level cost and impact study is performed, the cost effectiveness of the the APSA technology may not be optimal. The Olympus flexible blanket technology has been developed and has flown. It uses existing mechanical and electrical technology and is much heavier compared with fixed and APSA technology.

The comparison between the rigid panel array technologies show a mass competitive edge for the GaAs/Ge over the Si arrays. But, unless configuration constraints drive the decision to the GaAs/Ge technology, the Si is typically found to be the most cost effective solution.

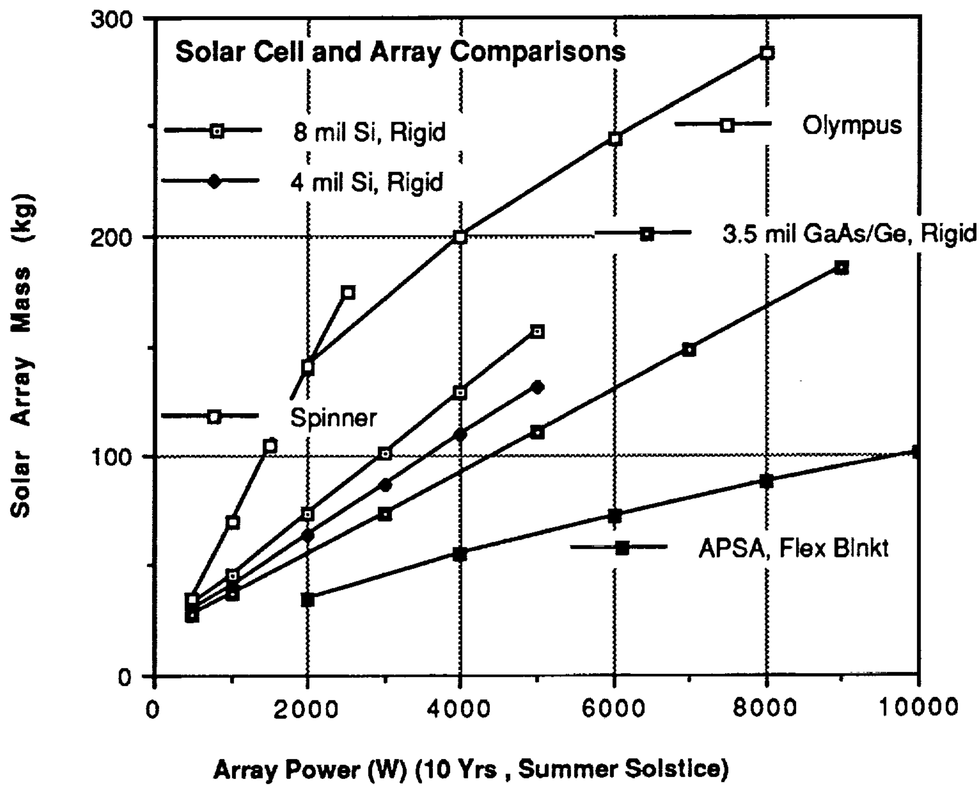


Figure 2-2: Solar Array Mass Comparison as a Function of Required Power

The spinner solar array is usually heavier than a deployable array because it must be sized for the 1/Pi illumination effect (i.e. the sun does not simultaneously illuminate the entire array because of the cylindrical shape of the array) . Also, typical launch vehicle fairings restrict the size of the spinner spacecraft drum, a deployable skirt solar panel design at higher power levels (>1,200 W) .

Spinner versus 3-Axis Stabilized Spacecraft
The solar array trade-offs between a spinner and 3-axis stabilized spacecraft is a function of EOL power required, and other mission considerations such as attitude pointing, spacecraft maneuvering, and orbital requirements. The spinner drum diameter and length is constrained by launch vehicle fairing dimensions, and by design requirements for a stable spin axis. The unique aspects of the Hughes deployable skirt solar array is an option to increase drum area, but mass and reliability become significant concerns.

Fixed Panel versus Flexible Solar Array
Almost all deployable solar arrays that have been used to date have been of the rigid panel type with power levels ranging from 0.5 to 5 kW. Their design and use is well understood and flight proven. Flexible arrays are lighter for high power levels, but their design and use is not well proven. With each new flight application, the flexible array will attain the understanding and confidence necessary for consistent use for high power applications.

The panel substrate of the rigid array are heavier than the thin blanket used to support cells on the flexible array. Thus, as array size increases, the flexible array mass increases more slowly than the rigid array. On the other hand, the deployment hardware (ie., masts motors, canisters) for the flexible array is heavier than the deployment hardware for the rigid array.

Si versus GaAs Solar Cell Technology
Si solar cells are the standard cell used in the majority of solar arrays. It is relatively inexpensive to produce, and is in the $10 price range per 2 x 4 cm cell, with an overall energy conversion efficiency between 10% - 14%. GaAs on the other hand is currently in the $100 price range per cell but can achieve energy conversion efficiencies of approximately 18%. The ability to develop thin, light weight GaAs cells on a repeatable basis is the primary reason for the high cost. The current approach toward a thin GaAs cell is the germanium substrate which can be made thinner on a repeatable basis. This GaAs/Ge cell is now available in a 3 mil thick cell, and it is approximately 40% higher in specific energy and comparable in mass to a typical 8 mil Si cell.

NiCd versus NiH2 Battery Technologies

The NiH2 battery has become the standard battery application for most GEO communications satellites. The high energy density over a 10 to 15 year mission life has been the primary reason for its wide spread application. Higher DOD, up to 80%, are being baselined for near-term spacecraft. This further improves the benefits of a NiH2 Battery over a NiCd. The major draw backs of the NiH2 battery is the increased recurring cost and lower volume efficiency compared to the NiCd battery. But, when launch related costs are factored in, the NiH2 battery is generally found to be favorable. **Table 2-2** summarizes the benefits and impacts of each battery technology.

The NaS battery offers a substantial W-hr per kilogram performance factor over the NiH2. However, the technology requires significant development and testing prior to flight application. Ongoing development programs within NASA and DOD is progressing towards flight test experiments aimed at the mid 1990s. In addition to the NaS technology development milestones, thermal control techniques are being developed and refined to maintain adequate battery temperature, and to isolate the battery thermal environment from the spacecraft.

Table 2-2: Battery Technology Benefits and Impacts

Battery Tech	Benefits	Impact
Nickel Hydrogen (NiH2)	High specific energy Higher allow DOD	Higher cost Lower volume efficiency
Nickel Cadmium (NiCd)	Low cost High volume efficiency Very mature design	Low Specific energy
Sodium sulfur (NaS)	Very high specific energy	Immature Technology High temperature operation

2.6 Mass Estimating Relationships and System Sizing

2.6.1 Solar Array

The estimate of the solar array size and mass is a function of lifetime, or years in orbit exposed to the natural radiation environment. The predicted data from the Intelsat-VII indicate the following relationship as shown in **Figure 2-3.** Although the exact nature of the degradation is a function of many variables, the following is representative of typical solar array degradation factors. The data was curve fit for typical Si and GaAs cell technology, and parameterized as follows.

Life factor = 0.9622 * x^-0.06542 (Si) **Equation 2.1**

where Life factor is solar array degradation factor, and x is the mission duration in years.

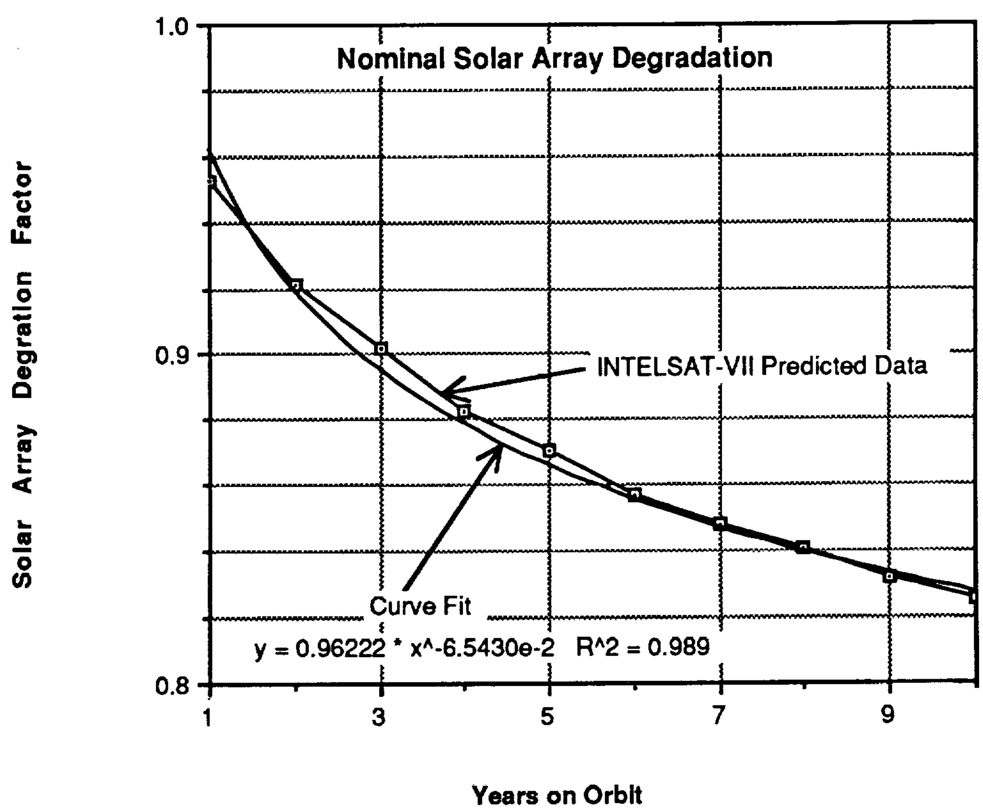


Figure 2-3: Nominal Solar Array degradation Curve Fit

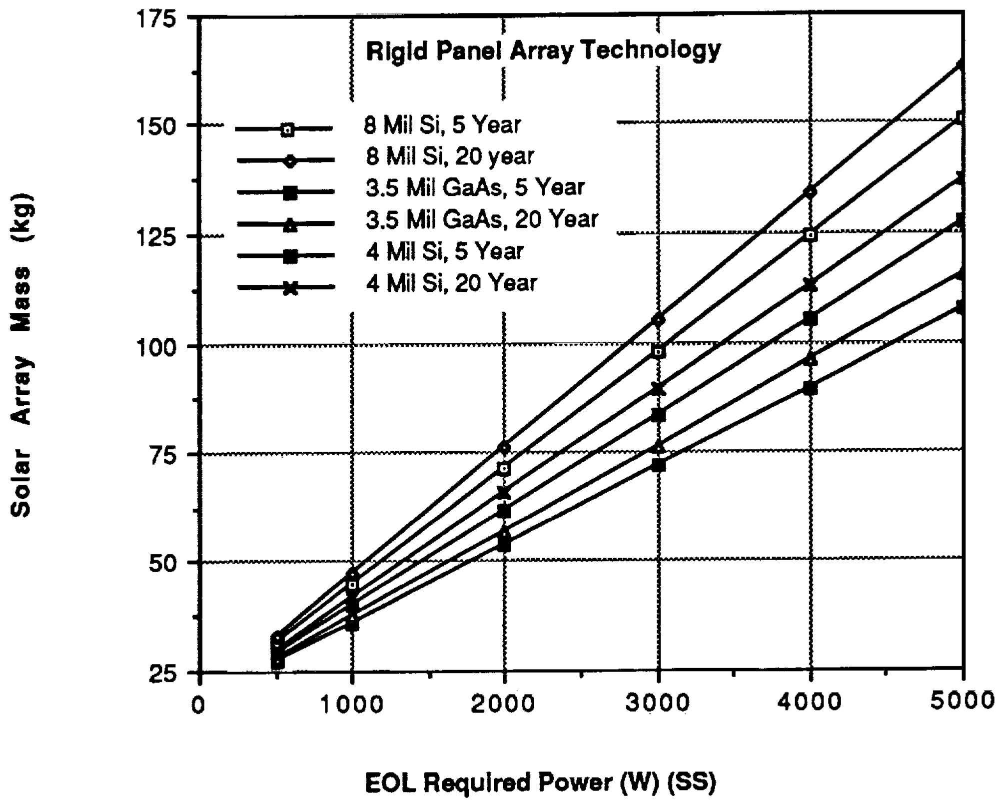


Figure 2-4: Mass Estimating Trends for Rigid Panel Array Technology

Figure 2-4 plots the mass for three rigid panel array/cell technologies as a function of end of life required power and mission duration. The equations shown below include a fixed budget for solar array yokes, hinges, and holddown mechanisms. The mass budget for cells, interconnects, coverglass, adhesive, and rigid panel substrate is a function of power level and mission duration. For each cell technology, the fixed substrate assumes a one inch graphite epoxy face skin substrate with aluminum honeycomb core. There is a small variable component to approximate additional mass for hinges, holddown, and damper system as the number of panels required is increased.

The following relationships have been developed for rigid panel solar array technology.

4 mil Si cell	M	=	18.3 + 0.0230 * Peol/Life	**Equation 2.2**
8 mil Si cell	M	=	18.3 + 0.0194 * Peol/Life	**Equation 2.3**

3.5 mil GaAs/Ge M = 18.3 + 0.0156 * Peol/Life **Equation 2.4**

where Peol is the end of life power requirement, and life is the solar array degradation factor calculated by Equation 2.1.

Figure 2-5 plots the mass of a spinner solar array as a function of end-of-life power and mission duration. The data was derived from a Hughes HS-376 series spacecraft with the deployable drum extension to provide greater power capability[1].

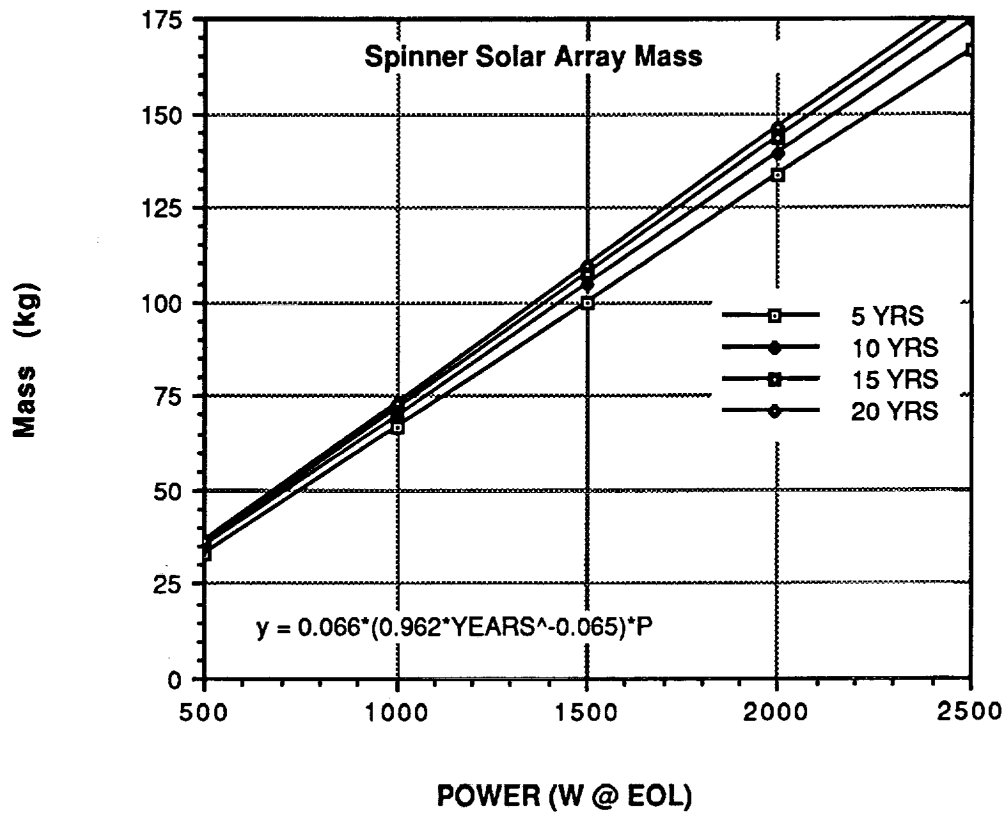


Figure 2-5: Mass Estimating Trends for Spinner Array Technology

1 Foder, J. S. etal., "SBS-1A Solar Arrays With Ultrathin, High Efficiency Solar Array Cells", Proceedings of the 23rd Intersociety Energy Conversion Engineering Conference, Denver Co., 1988.

Figure 2-6 plots the mass of two flexible blanket solar array technologies as a function of EOL power and mission duration. The Olympus array technology is based on actual solar array designs for the Olympus geosynchronous satellite program[2]. The data was curve fit given four existing data points, and parameterized as follows.

Marray = 2.824*(Peol/life)^0.501 (kg) (Olympus technology) **Equation 2.5**

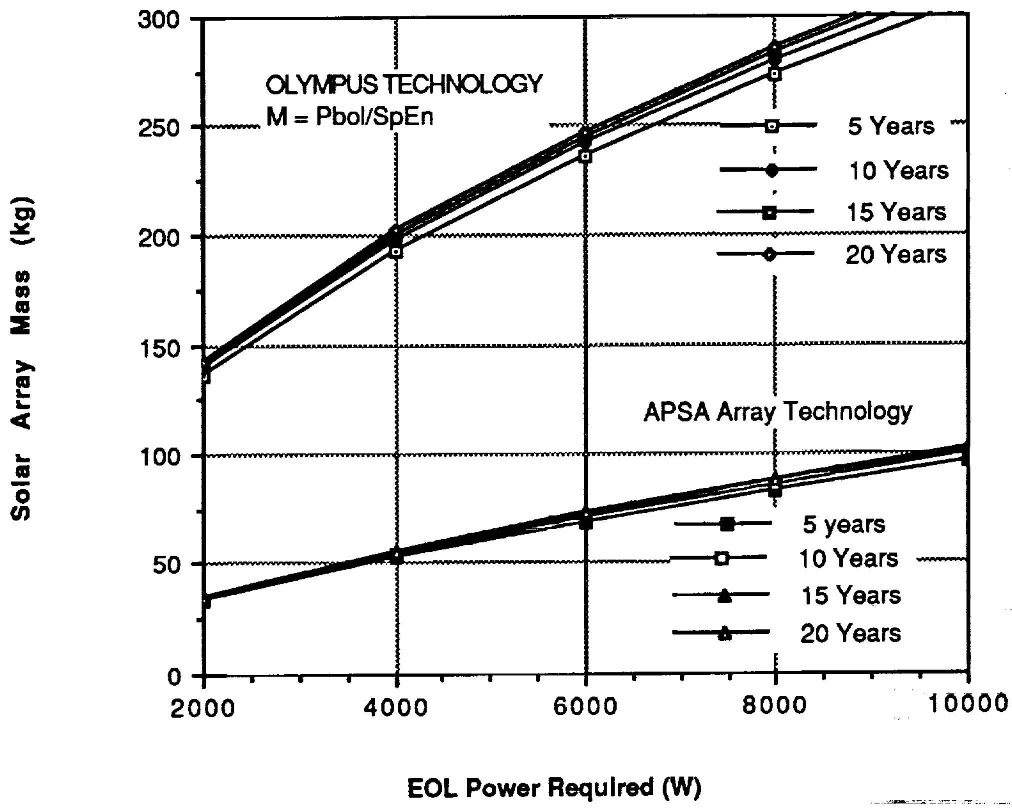


Figure 2-6: Mass Estimating Trends for Flex Blanket Array Technology

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2. Briefing Package PR275, Spar Aerospace Limited Advanced Technology Systems Division, "Olympus solar array technology"

The Advanced Photovoltaic Solar Array Program[3] is a long term technology program to increase the state-of-the-art for flexible blanket solar arrays. The specific objectives of this technology is to demonstrate that a solar array can provide >130 W/kg at beginning of life for a 10 year geosynchronous mission. The data included represents intermediate results based on prototype hardware and subassemblies. The data was curve fit using three data points from the aforementioned reference, and parameterized as follows.

Marray = 0.196*(Peol/life)^0.663 (kg) (APSA technology) **Equation 2.6**

Solar Array Area

Table 2-3 lists the specific power for each of the solar array technologies at beginning of life summer solstice. Total solar array area as a function of end of life required power is given by the following,

Solar Array Area = Peol/life/Sp.Power **Equation 2.7**

where Sp. Power is the watts of power generated from the solar array per square meter of area.

Table 2-3: Specific Power for Different Solar Array Technologies

Cell Technology	Array Technology	Sp. Power (W/sqm) (SS)
Si, 8 mil	Deployable panel, Grep/Al core	117
Si, 4 mil	Deployable panel, Grep/Al core	129
Si, 4 mil	Spinner, Grep/Al core	47
GaAs/Ge, 4 mil	Deployable panel, Grep/Al core	164
Si, 2 mil	Flex blanket, APSA program technology	122
Si, 2 mil	Flex blanket, Olympus satellite program	68.2

3 Kurland, R., Stella, P., Status of Advanced Photovoltaic Solar Array Program", Proceedings of the 23rd Intersociety Energy Conversion Engineering Conference, Denver Co., 1988.

2.6.2 Battery

Figure 2-7 plots the battery mass for a NiCd, NiH2, and NaS battery technology as a function of the maximum eclipse power required and battery voltage. To first order, the mass of the NiCd battery is not a function of battery voltage, whereas the NiH2 battery is a function of voltage. This factor is observed primarily due to the mass of the NiH2 cell pressure vessel. Since the NiCd does not operate under pressure, most of the cell mass is the electrodes and not the pressure vessel.

Upon close examination of the NiH2 battery curves, a nonlinear aspect occurs due to a jump in pressure vessel diameter size from 3.5" to 4.5" diameter cell. Cell data from Gates Aerospace was used to develop the cell mass as a function of capacity relationship. The NiH2 mass estimating relationship shown in Figure 1.8 is linear with respect to power and voltage, but this expression does not include the cell diameter discrete jump when cell capacity increases beyond 100 Ah.

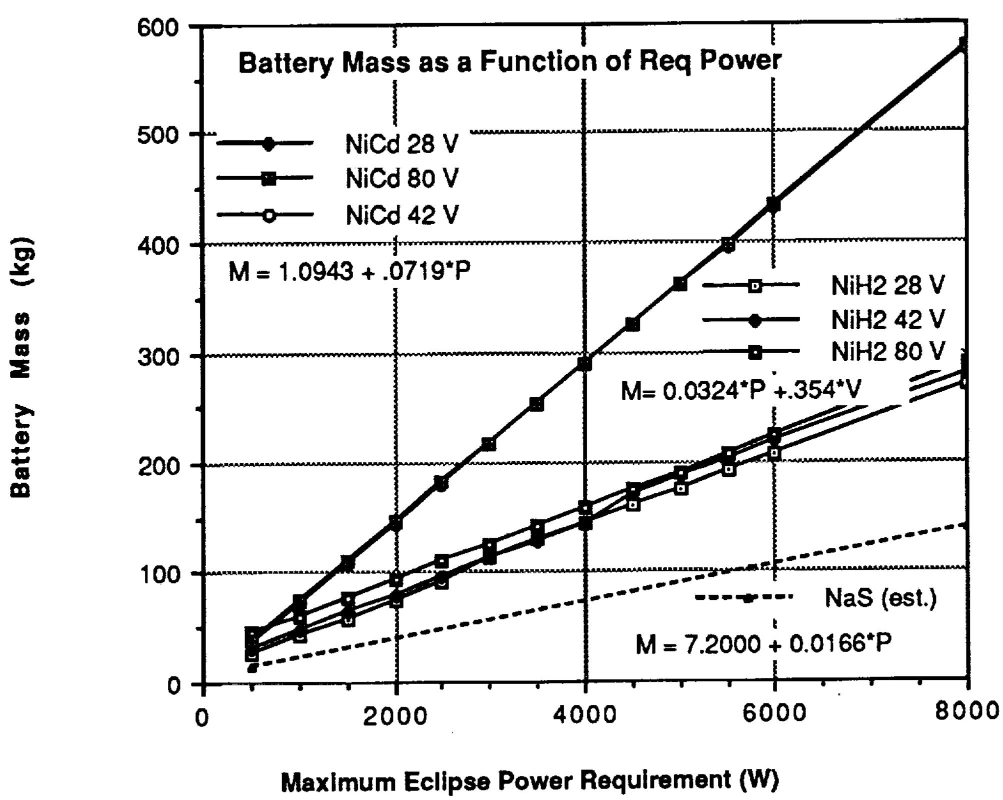

Figure 2-7: Battery Mass for Various Battery Technologies

Battery Charge Power
The power that must be allocated in a power budget to charge the battery is a function of the worst case discharge of the battery, the time requirement to recharge the system, and the efficiency of the charging regulator. The following equation is used to estimate battery charge power.

Pbattery charge = Peclipse * tmax eclipse duration)/ (0.9 * trecharge) **Eq. 2.8**

where Peclipse is the steady state eclipse power requirement, tmax eclipse duration) is the maximum eclipse duration, trecharge is the allowable time to recharge the battery, and 0.9 is a charge system efficiency factor. The maximum eclipse duration for a GEO satellite is approximately 1.2 hours, and a typical recharge time is 12 - 15 hours. As the allowable recharge time increases, the recharge power requirement from the solar array decreases.

2.6.3 Positioning Mechanism: Solar Array Drive Assembly

Figure 2-8 plots the mass of two solar array drive assemblies (SADA) as a function of maximum load through each SADA. The spacecraft configuration assumes a 3-axis stabilized satellite with a two solar array symmetric configuration. The SADA design assumes a one axis of rotation mechanism which steps through a revolution at the orbital rate of once per day. The SADA slip rings are sized to handle the end of life power required by the spacecraft and additional small slip rings for telemetry signals. The discontinuous jumps in the 28V and 42 V curves represent discrete slip ring current ratings. As current requirements increase, bigger and more massive slip rings are required.

2.6.4 Power Processing Unit

The mass of the power processing electronics has been developed from a historical survey of itemized power conditioning electronics as a function of power level and the mass of the battery and solar array[4].

Mppu = (0.173 + 0.01856*P)(Mbattery + Msolar array) **Equation 2.9**

where P is the nominal end of life power requirement in kW, and Mbattery and Msolar array are the mass of the battery and solar array, respectively.

4 Capel, A., et al. , "High Voltage Conditioning for Space Applications", Proceedings of the IEEE, Vol.76, No. 4, April, 1988, pp. 391-408.

Chetty, P., R., K., Satellite Technology and its Applications, Tab Books Inc., B;ue Ridge Summit, PA, 1988.

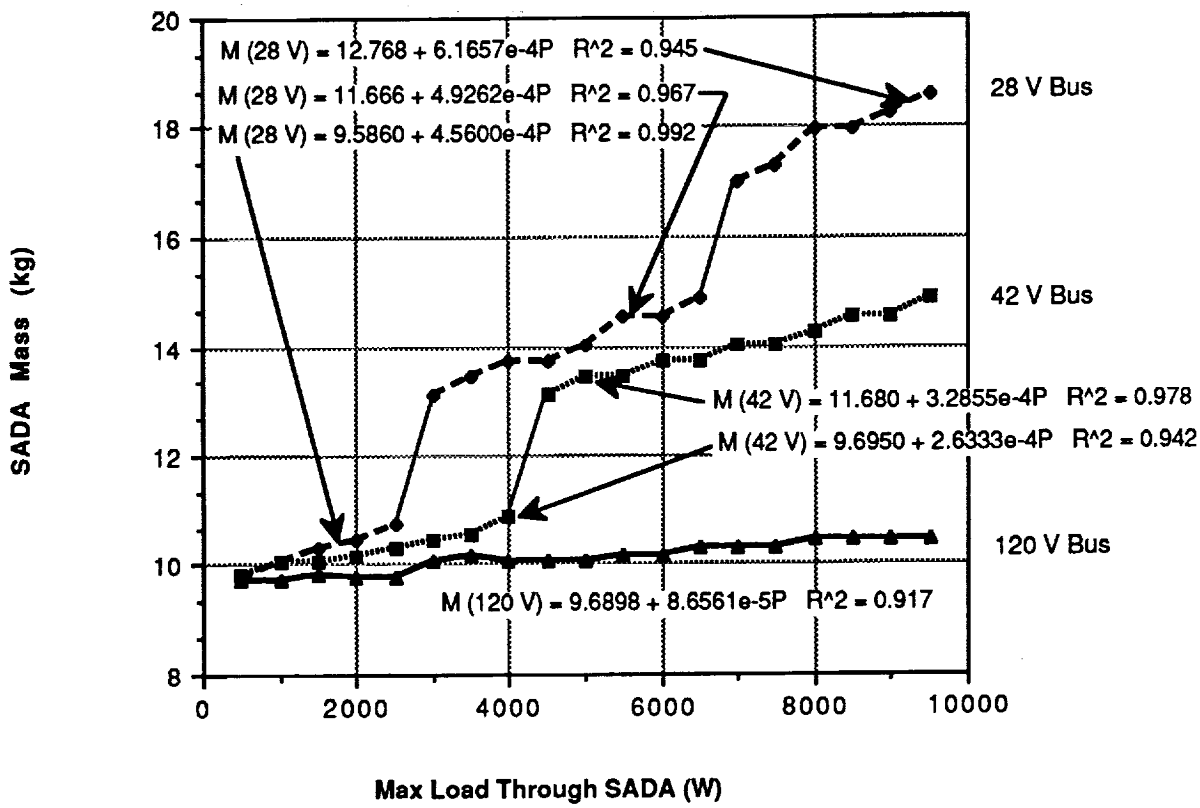

Figure 2-8: SADA Mass versus Power for a Two Solar Array Configuration

Chapter 3

Attitude Control

3.1 Introduction

Attitude Control System (ACS) sizing task describes the selection and sizing considerations for a communications satellite designed for a geosynchronous orbit. Representative attitude control systems have been defined for three levels of spacecraft pointing accuracy with their associated mass and power properties for actuators and sensors. A discussion is also presented to assist the user in choosing the appropriate system.

Modelling architecture layout is described by the logic chart in **Figure 3-1** which shows the process including input data, performance requirements, decision points, system selection guides, and mass sizing tables and charts necessary to generate a system selection and estimates of mass and power.

Communications satellite assumptions are used to generate system architectures. At a conceptual design stage it is reasonable to neglect detailed design parameters when those variables do not significantly effect the system level mass and power. The most significant mass drivers in the attitude control system are the actuators and electronics. The electronics mass has been treated in two ways. Electronics that interface between the spacecraft control system and the user equipment, such as wheel drive electronics and sensor electronics, are included in the sensor and actuator mass and power estimates. The spacecraft level control electronics are included in the TT&C Spacecraft Processor and Distributed Electronics section.

The architecture of the attitude control system is a mission specific and detailed design task. There is some correlation between mission type, performance requirements, and mass and power. However, a significant variation in pointing performance may not drastically influence the system mass and power. In addition, the definition of pointing is very specific, and when used loosely is quite ambiguous. Pointing accuracy usually refer to end-to-end boresight spatial alignment of a specific antenna . However, the size, shape, and spacecraft configuration affect the thermal behavior and alignment of the antenna pattern. So there can be a significant difference between the spacecraft body and antenna pointing performance. Also, there is many different spacecraft operational modes that influence the pointing accuracy.

INPUT

MISSION REQUIREMENTS:
- MISSION DURATION
- EOL POWER REQUIREMENT

INPUT

ATTITUDE CONTROL SUBSYSTEM REQUIREMENT:
- SPACECRAFT POINTING ACCURACY
- MAXIMUM CYCLIC DISTURBANCE TORQUE

ATTITUDE CONTROL SELECTION GUIDELINES

- TABLE 3-1: ATTITUDE CONTROL METHODS AND THEIR CAPABILITIES
- TABLE 3-2: EFFECTS OF CONTROL ACCURACY ON SENSOR SELECTION AND AOC DESIGN

ATTITUDE CONTROL SYSTEM SIZING

TABLE 3-3: SPIN STABILIZED SYSTEM
TABLE 3-3: 3-AXIS MOMENTUM BIAS SYSTEM
TABLE 3-3: 3-AXIS ZERO MOMENTUM SYSTEM

FIGURE 3-2: MOMENTUM BIAS AND WHEEL MASS VS. SPACECRAFT POINTING
FIGURE 3-3: REACTION WHEEL SYSTEM MASS AND MOMENTUM STORAGE VS. WORST CASE CYCLIC TORQUE

Figure 3-1: Attitude Control System Sizing Flow Diagram

Therefore, the following approach has been used to approximate ACS mass and power as a function of pointing performance and stabilization technique.

1. Three attitude control systems architectures have been defined corresponding to stabilization method: dual spin, 3-axis momentum bias, and 3-axis zero momentum bias.
2. Each system will quote a nominal pointing accuracy, the values being based on typical GEO spacecraft. An RF autotracking system, which can reduce structural alignment errors and thermal distortion errors, has been included as an add-on system in order to improve antenna boresight pointing accuracy to approximately 0.05°.

3.2 Mission Requirements

The following mission level requirements are required:

Mission duration is the lifetime at which a certain probability of success is achieved while delivering the required minimum level of power to the user equipment.

Solar array power is the average steady state power required to satisfy the mission. This is considered the end-of-life power requirement.

3.3 Attitude Control System Requirements

The following subsystem level input data must be specified:

Pointing accuracy is the angular pointing tolerance during the normal on-orbit operating mode for a simple deployed antenna from either the top or side of the spacecraft body.

Maximum cyclic disturbance torque. Typical values for a symmetric geosynchronous spacecraft are provided if this parameter is not available.

3.4 Major Attitude Control System Descriptions

3.4.1 Attitude Stabilization

The methods described below are generally accepted methods of attitude stabilization. The distinction between attitude control and orbit is that orbit control is the point mass motion of the spacecraft in orbit and attitude control is angular motion of the spacecraft about its center of mass.

Spin stabilized spacecraft uses the inertia of the entire vehicle to generate gyroscopic stiffness which is used to maintain a near inertial reference for the spacecraft spin axis. The gyroscopic stiffness passively controls the spacecraft from external disturbances due to solar torques, magnetic field interactions, gravity gradient torques, and from internal disturbances do to thruster transients and equipment motion. The mono spin spacecraft spins the entire spacecraft at a rate to provide the necessary gyroscopic stability. The payload instruments must either be designed as scanning or in the case of Earth pointing antennas, electronically or mechanically despun. The dual spin spacecraft also spins the spacecraft to achieve gyroscopic stability, but one section of the spacecraft is spinning at the orbital rate which allows fixed antennas to continually point to the Earth.

Gravity gradient stabilization system interacts with the gravitational field to maintain the spacecraft attitude. Since the gravitational field is proportional to the inverse of the radius cubed, gravity gradient stabilization is a candidate only for low altitude satellites. Gravity gradients require no moving parts other than deployed booms so that one of the spacecraft principal moments of inertia is smaller than the other two in order to align the minor axis along the nadir vector.

3-axis momentum bias stabilized spacecraft controls all three axes. The momentum bias generated with a momentum wheel utilizes the benefits associated with a gyroscopic stiffness and reference. The momentum bias is oriented about the orbit-normal like the spin or dual spin stabilized spacecraft. However, the roll axis is directly controlled, and the yaw axis is indirectly controlled through the kinematic coupling due to orbital motion. The typical momentum bias system functions without a direct yaw sensor by estimating the relationship between roll which is measured and yaw which is not, by the reference momentum vector of the spacecraft momentum wheel.

3-axis zero momentum stabilized system differs from the momentum bias system by the simple fact that all three spacecraft axes have the same momentum, typically zero. This system is typically implemented in a highly maneuverable spacecraft by a combination of reaction wheels. This control system will require direct yaw measurement through either a star sensor or a long life continuously running gyro.

3.4.2 Attitude Control Actuators

Momentum wheel assembly is spinning flywheel operating at a nominal wheel speed. The angular momentum used to create a gyroscopic stiffness against disturbance torques is the product of the wheel inertia and the angular rate. The momentum wheel provides a variable momentum storage device about its rotation axis, which is usually fixed the vehicle.

Reaction wheel is a flywheel fixed in the vehicle, but designed to at zero momentum bias. It operates in a control system by responding to external disturbances by temporarily absorbing the torque through a wheel speed increase. The nominal wheel speed is zero, but spins up and down in both direction in response to spacecraft disturbances.

Control moment gyro (CMG) consists of a single or double gimballed momentum wheel. The gimbals allow for control of the flywheel momentum vector, and is typically used to provide high torque response for spacecraft attitude maneuverability.

Magnetic torquers are magnetic coils or electromagnets that generate an electric field to interact with the Earth's magnetic field to generate control torques on the spacecraft. The control torques generated is a function of the spacecraft altitude and inclination, and the size and power of the magnetic torquer.

Thrusters are used to control spacecraft attitude and unload the momentum accumulation of spacecraft control actuators by expelling propellant in the opposite direction. Thrusters can operate using cold gas, chemical, or electric propulsion technologies which are discussed in the propulsion section. Although thrusters are an integral part of an attitude and orbit control system, their use is limited in attitude control to minimize expendable propellant.

3.4.3 Attitude Determination Sensors

Earth sensors measure the Earth's infrared radiation typically through a bolometer, which is a sensitive resistance thermister. Typical Earth sensors are either fixed sensors, and mounted on a spinner satellite, or incorporate a scan mirror. The are commonly used as the primary reference for Earth pointing spacecraft. Typical knowledge errors using high fidelity Earth sensors is approximately 0.01° - 0.02°.

Sun sensors have been used on almost every orbiting spacecraft. This is because the solar brightness and angular diameter are relatively constant for all possible values and orbital elements. These sensors can be of the coarse analog to fine digital sensors with knowledge accuracies that range from 0.1° to 0.02°.

Star sensors lock on and track a star rather than simply monitor the star field as does a star mapper. Star trackers are high accuracy devices that determine inertial attitude for pointing and for navigation in space. Their pointing accuracies are a function of their field of view, which in tern is a function of optics. Historically, star sensors are heavy and expensive, and require more power than other attitude sensors. However, charge coupled diode (CCD) technology will reduce both mass and cost of future star sensors.

Gyros measure rotational motion of the spacecraft and provide short term accurate attitude information. They are commonly used on 3-axis spacecraft to provide attitude information during orbit acquisition and thruster maneuvers. Two major types of gyros are the spun mass and those that use the change in path length due to the rotation of a light beam as it transverses a closed path. The latter may be either a ring laser or fiber optic gyro. Spun mass gyros have been used most often in space applications but are life limited due to bearing wearout. The laser and fiber optic gyros are newer technologies with long life potential, but require development and refinement.

3.5 Guidelines for the Selection of Attitude Control System

There are many factors that influence the selection of power system technologies such as:

- System development and qualification costs,
- Recurring costs,
- Spacecraft integration
- Launch vehicle constraints
- Launch, Transfer and on-orbit operations
- Overall system reliability and risk,
- Structure, power and thermal subsystem implications.

These general considerations for the selection of different attitude control architectures and their capabilities are summarized in the **Table 3-1**. **Table 3-2** provides general recommendations for the selection based on pointing performance.

The spin stabilized class of spacecraft can achieve moderate absolute accuracy, but is limited by jitter due to the spin dynamics. It has minimum control system and propulsion weight, but is power limited and configuration sensitive. The 3-axis momentum bias has good pointing stability with moderate accuracy, and provides yaw estimation without the need for star sensors or continuously running gyros. The passive roll/yaw stability due to the momentum wheel bias incorporates the benefits of the spin stabilized concept without the configuration or power system constraints. However, the momentum bias is heavier and requires more power.

The 3-axis zero momentum system with active sensing and control about all three axes can provide the best pointing performance and mission flexibility. This option does require either a star sensor or a continuously running gyro. The star sensor has a mass and cost impact but it is also computationally intensive. Limited spacecraft field of view due to the solar array rotation can require additional star sensors or blockage periods. When a mission requires high slew rates or frequent attitude reorientation, the zero momentum system is often the only choice. And as star sensor and long life gyro technologies are enhanced, the reliability and cost will allow the zero momentum system to be implemented more often. However, in the early to mid 1990s, the momentum bias architecture is the best choice for larger, higher power communications spacecraft.

Table 3-1[1]: Attitude Control Methods and Capabilities

Type	Pointing Options	Translation Maneuverability	Rotation Maneuverability	Lifetime Limits
Passive				
•Gravity gradient	•Earth local vertical only	•Minor adjustments with thrusters	•Very limited	• None
•Gravity gradient & momentum bias wheel	•Earth local vertical only	•Minor adjustments with thrusters	•Very limited	• Life of bearings
Spin Stabilized				
•Pure spinner	•Inertially fixed any direction •Repoint with precession maneuvers	•Large ΔV along spin axis, minor adjust in other two axes with thrusters	•High propellant usage to move stiff momentum vector	• Thruster propellant
•Dual spin	•Limited only by articulation on despun platform	• Same	•Despun platform constrained by its own geometry	• Thruster propellant • Despin bearings
3-Axis Stabilized				
•Zero momentum	•No constraints	•Any direction any level depending on size of thruster and main engine	No constraints	• Thruster propellant • Life of sensor and actuator bearings
•Bias momentum	•Best suited for local vertical pointing	•Same as zero momentum with full set of thrusters	•Momentum bias constrains attitude mnvrs w/o propellant impact	• Thruster propellant • Life of sensor and actuator bearings

1 Wertz, J. R., Larson, W. J., Space Mission Analysis and Design, Kluwer Academic, Boston, MA, 1991.

Table 3-2[1]: Effects of Control Accuracy on Sensor Selection and AOC Design

Required Accuracy, 3s	Effect on Spacecraft	Effect on GN&C
>5 deg	•Permits major cost savings •Permits gravity gradient(GG) stabilization in LEO •GG limits maneuvers & is sensitive to on-board disturbance torques	Without attitude determination •No sensors required for GG stabilization •Boom motor, GG damper, and a bias momentum wheel are only required actuators With attitude determination •Sun sensors & magnetometer adequate for attitude determination at ±2° - 3°
1 deg to 5 deg	•GG feasible in LEO •Spin stabilization feasible if stiff, inertially fixed attitude is acceptable •Payload needs may require despun platform on spinner •3-axis stabilization will work	•Sun sensors and horizon sensors may be adequate for sensors, especially a spinner •Accuracy for 3-axis stabilization can be met with RCS deadband control but reaction wheels will save propellant for long missions •Thrusters and damper adequate for spinner actuators •Magnetic torquers (and magnetometer) useful
0.1 deg to 1 deg	•Dual Spin stabilization feasible if stiff, inertially fixed attitude is acceptable •3-axis stabilization feasible	•Need for accurate attitude reference leads to horizon sensors and possibly gyros •Reaction wheels typical with thrusters for momentum unloading and coarse control •Magnetic torquers feasible
<0.1 deg	•3-axis stabilization is necessary •May require articulated & vibration-isolated payload platform with separate sensor	•Same as above for 0.1 deg to 1 deg but may need star sensor and better class of gyros •Control laws and computational needs are more complex •Flexible body performance very important

3.6 Mass Estimating Relationships and System Sizing

Table 3-3 lists the representative mass and power for the three attitude control system options. In all cases,the attitude control thrusters are listed for completeness but are included in the propulsion system mass estimates. Propellant required for attitude control is a very small percentage compared to the propellant required for orbit control. The spin stabilized spacecraft is very simple compared to the the 3-axis stabilized spacecraft. The only required sensors are Earth sensors and sun sensors. The only required actuators are the thrusters, however, for the dual spin stabilized configuration, a despin motor assembly also known as a bearing and power transfer assembly (BAPTA) is required.

The 3-axis stabilized spacecraft require additional sensors and actuators to provide control about the three axes. Additional sun sensors are required for initial acquisition and contingency operations to ensure an inertial reference is available. And gyros are needed to provide 3-axis attitude knowledge during orbit control maneuvers. Attitude control actuators are required to impart reaction torques necessary to counteract disturbance torques. A system made up of only thrusters could be used, however, since many of the disturbance torques acting on the spacecraft are cyclic, a momentum storage device is more mass efficient over the life of the spacecraft.

The components of each 3-axis stabilized system is, to first order, a constant except the momentum and reaction wheels. The component mass and power is summarized in **Table 3-3**. In the momentum bias system two momentum wheels are nominally aligned with the pitch axis but offset by an angle to form a V configuration. Simultaneous speed change provides control about the pitch axis, and differential speed change provides control about the roll axis. In case of a wheel failure, the reaction wheel, which is aligned along the yaw axis is turned on to provide the roll and pitch control. As stated earlier, the yaw axis can be controlled because orbital motion transfers a yaw error into a roll error at quarter-orbit intervals.

The momentum wheel in a momentum bias control system must be sufficient to stabilize the yaw axis. As the momentum bias increases, the yaw error decreases. The momentum wheel for a typical geosynchronous communications spacecraft can be sized as a function of the allowable yaw error component of the total attitude pointing budget. **Figure 3-2** shows the relationship between the momentum wheel system mass and momentum bias as a function of end-to-end antenna pointing accuracy. The following assumptions have been made to generate this relationship:

- Maximum external torque is solar torque of magnitude 5E-5 Nm. This is based on a symmetric spacecraft configuration with deployable solar arrays on the north and south sides, and deployable antenna reflectors on the east and west sides.

Table 3-3: Attitude Control System Mass and Power Estimates

Component	Mass (kg)	Power (W) Steady State
Spin Stabilized		
Earth Sensor (2)	2.0	4
Sun Sensor (2)	.3	--
BAPTA (1) (Dual spin only)	7.0	5
Nutation damper (1)	.2	
Thrusters(12)	Incl in Prop Syst	
3-Axis Momentum Bias		
Earth Sensor (2)	3.0	4
Coarse Analog Sun Sensor (4)	.6	–
Digital Sun Sensor (2)	1.4	1
DIRA (1 Pkg)	5.0	35
Momentum wheel (2)	Figure 3-2	32
Reaction Wheel (1)	Figure 3-3	incl. above
Thrusters (12)	Incl in Prop Syst	–
3-Axis Zero Momentum		
Earth sensor (2)	3.0	4
Coarse Analog Sun (4)	.6	–
DIRA Pkg	5.0	35
Star tracker (2)	14.0	20
Reaction Wheels (4)	Figure 3-3	45
Thrusters	Incl in Prop Syst	–
RF Autotracking Add-on		
Tracking Receiver(2)	6.0	9
Antenna Positioning Mech (1)	5.0	1
Misc comp	4.5	5
Totals	15.5 kg	15 W

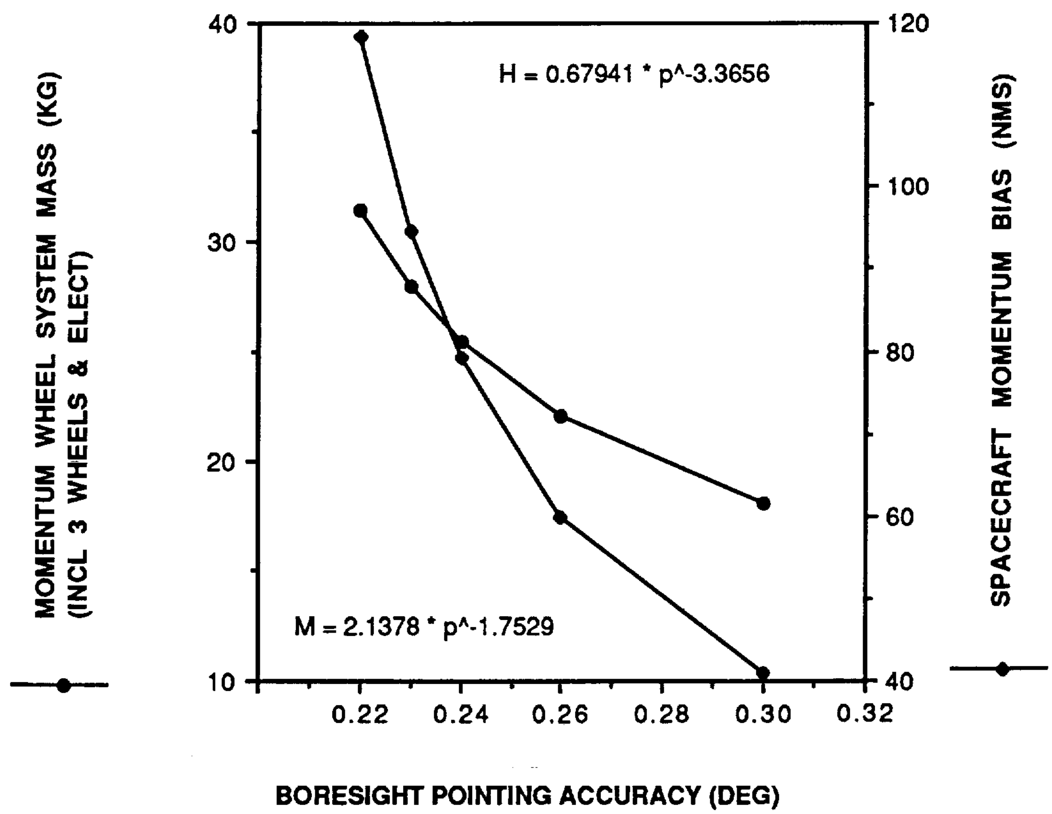

Figure 3-2: Momentum Wheel Sizing

- Maximum roll and pitch pointing errors do not exceed 0.13° and 0.15°, respectively. These values assume representative pointing errors associated with random long and short term, diurnal, mean seasonal, orbital, and modelling errors.

- Yaw error was calculated based on a 35° latitude. The pointing error for target Latitudes greater than ±35° would be slightly greater. For example, a change in latitude from 35° to 60° would increase the pointing error from 0.22° to 0.23° at a momentum bias of 128 Nms.

- The wheel system mass is based on a two momentum wheel system and the mass vs momentum relationship was derived from a selection of Honeywell momentum wheels.

The reaction wheels in a zero momentum system must be sized for torque and momentum storage to absorb cyclic torques and temporarily store momentum produced by slew or reorientation maneuvers. Since the reaction wheels operates with nominally a zero momentum all three axes are controlled directly by reaction wheel speed change. The reaction wheel torque is sized by the worst case anticipated disturbance torque. Unless attitude slew requirements are significant, worst case disturbance torques are well below almost all candidate reaction wheels. Therefore, reaction wheel sizing shall be based on momentum storage requirements. The worst case disturbance torque during normal mode operations for a GEO communications satellite is solar torque. And for a symmetric spacecraft design, which is always an important design consideration, a conservative value is 1E-4 Nm can be used.

Figure 3-3 shows the relationship between the maximum cyclic disturbance torque and the reaction wheel momentum and system mass. The dotted line on the chart indicates a typical design point for a 1E-4 solar maximum solar torque with a design margin of two.

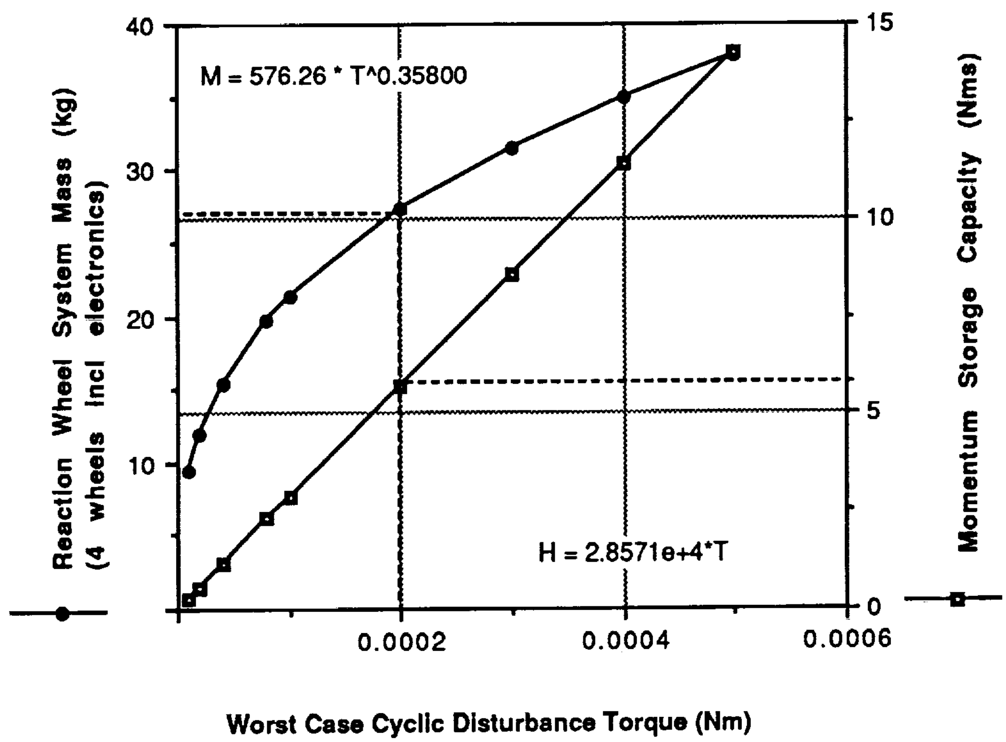

Figure 3-3: Reaction Wheel Sizing

Chapter 4

Structure

4.1 Introduction

The structure consists of all spacecraft elements providing mechanical support for all subsystems in a configuration that meets system requirements and launch vehicle constraints. The structure must be capable of sustaining all direct and cumulative load combinations occurring during satellite testing, ground handling, launch, and orbital maneuvers without exceeding elastic deformation limits. It also satisfies subsystem requirements such as alignment of sensors, actuators, and antennas.

A typical spacecraft structural design is based on a system of trade-offs to determine the spacecraft configuration which will meet subsystem and launch vehicle constraints. Analysis such as static and dynamic loads are performed and the structure is redesigned if stress margins are not adequate or if the natural frequencies and thermal distortion requirements are not meet. The configuration is mission dependant but the structure consists of tubes, struts, cylinders, and panels made of either aluminum, graphite epoxy, or honeycomb panels. In some cases other materials are used but the above mentioned materials are predominantly used. A central cylinder with panel and strut configuration is common for GEO 3-axis stabilized spacecraft as is strut or cruciform designs.

Communications satellite assumptions used to generate system architectures.
The structural design and configuration is driven by many mission specific variables. It is not practical to estimate the influence of all these design aspects nor is it feasible to compare one manufactures base configuration to another. However, there is a correlation of historical trends that relate the structural spacecraft mass to the total dry spacecraft mass. This is used quite often at the conceptual stages of design to gain an approximation of structure mass without the need to evaluate structural and configuration issues. This methodology has been assumed in this section to estimate the structural subsystem mass.

4.2 Mission Requirements

The following mission level requirements are required:

Overall spacecraft dry mass: This spacecraft mass is the sum of the payload and spacecraft bus masses and excluding propellant. This is an iterative estimate process since the structure mass is estimated based on total mass, but the value quickly converges on the structural mass value after 2 to 3 iterations.

4.3 Mass Estimating Relationships and System Sizing

Figure 4-1 shows the relationship between percentage of spacecraft structural mass to total spacecraft dry mass with the percentage of advanced composites used as a variable. The structural mass is determined by estimating the overall spacecraft dry mass and calculating the mass fraction as a function of the percentage of advanced composites used in the structure. The structural mass is then simply the product of the mass fraction and the spacecraft dry mass. Spacecraft dry mass is the mass of the payload and support subsystems excluding pressurant and propellant, and launch vehicle adapter. This is an iterative process because the structural mass is a component of the overall spacecraft dry mass. But, it is a linear approximation and converges within two to three iterations.

Typical geosynchronous communications satellites utilize between 40% to 65% graphite epoxy or fiberglass in the structural design. Advanced composites save mass and have very favorable low coefficients of thermal expansion. However, advanced composites have not in the past exhibited the thermal conductivity characteristics that are required for thermal control. But with proper fiber orientation, graphite epoxy and carbon-carbon structures have been exhibiting superb thermal transport characteristics. Therefore, the percentage of advanced composites is expected to increase toward 80% as this technology is demonstrated and as fabrication processes become more efficient.

In an effort to estimate the structural mass fraction and the percentage of composites in the structure, it was necessary to make many assumptions. Examining many typical geosynchronous spacecrafts, the variation in the structural mass fraction ranged from 12% to 18%. But in many cases it was not obvious whether this variation was affected by the material, some very influential design or schedule requirements, or a mass properties bookkeeping difference from program to program. The spacecraft design is always a compromise between cost, schedule commitments, and mass efficiency to satisfy program goals. So there can be examples of a spacecraft structural design that is not represented by the trends described in this section.

The specific assumptions that were made in generating **Figure 4-1** was that the structure is representative of a central cylinder, panel and strut design with data from primarily Intelsat VII series spacecraft being used. The central cylinder is the main load bearing component. Equipment panels are aluminum honeycomb with either aluminum or graphite epoxy faceskins. Struts and longerons are used to provide equipment and panel attachments from the central cylinder. The variation and impact of advanced composites were generated from modifications and mass reduction exercises throughout the Intelsat VII program, and by extrapolating these values to lower and higher ranges.

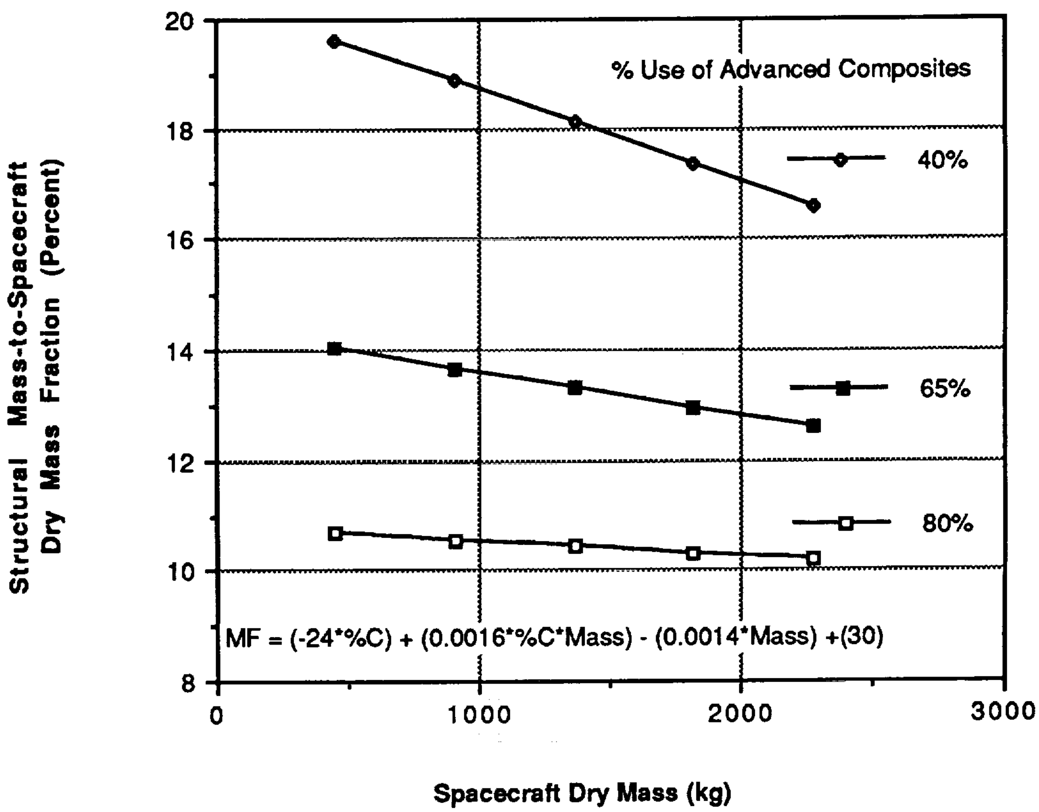

Figure 4-1: Structural Mass Fraction as a Function of Spacecraft Dry Mass and Percentage of Advanced Composites

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Chapter 5

Payload

The communications payload modeling is restricted to the multiple beam antenna and the on-board baseband electronics. The chapter is divided into two parts:

5.1 Multiple Beam Antennas

5.2 Baseband Electronics

5.1 Multiple Beam Antennas

5.1.1 Introduction

The modern communication satellite is experiencing growing demands on the capabilities of its antenna systems in the following areas:

- Providing high antenna gain to multiple shaped coverage areas,
- Facilitating polarization and/or spatial frequency reuse,
- Reconfiguring antenna coverage beam shapes, and
- Forming nulls in the coverage pattern to reject jamming interference.

The multiple beam reflector antenna (MBRA) has been the choice in most communication satellite antenna system designs on account of its capability and flexibility in meeting these increasingly complex requirements, and is likely to remain the prevalent technology for years to come.

There are stringent requirements on the mass, mechanical strength and thermal properties of spacecraft antenna systems. Carbon fiber (graphite) has been the preferred material for fabricating the reflecting surfaces of MBRA's with 3-m or smaller aperture diameters. Carbon fiber has also been applied in the manufacturing of feed array components such as horn radiating elements and waveguides for weight minimization.

For large reflectors such as those used in the mobile communication satellites, mesh-type surfaces are usually employed to construct the reflector. Various types of mechanical supporting structures have been used to support the mesh surfaces:

- Radial rib
- Wrap rib
- Geo-truss
- Hoop column
- Prismatic truss
- Cable catenary

This study examines the mass properties of MBRA's. The two primary parts affecting the mass of an MBRA are the reflector and the feed array. A graphite type reflector is assumed for antennas with diameters of approximately 3 m or less and a mesh type reflector is assumed for larger antennas.

5.1.2 Mass Properties of Reflector

The mass of the reflector is a function of antenna aperture size and type of the material. Figure 5-1 shows the reflector mass versus diameter for rigid, graphite-type reflectors with 3 m or less diameter. The data are compiled from various communication satellite MBRA systems launched from the early 1980's up to the present. The mass includes the reflecting surface and all the necessary mechanical backup structure. A near linear relationship between the mass and the diameter can be deduced from the data:

$$M = 6.674D - 3.802$$

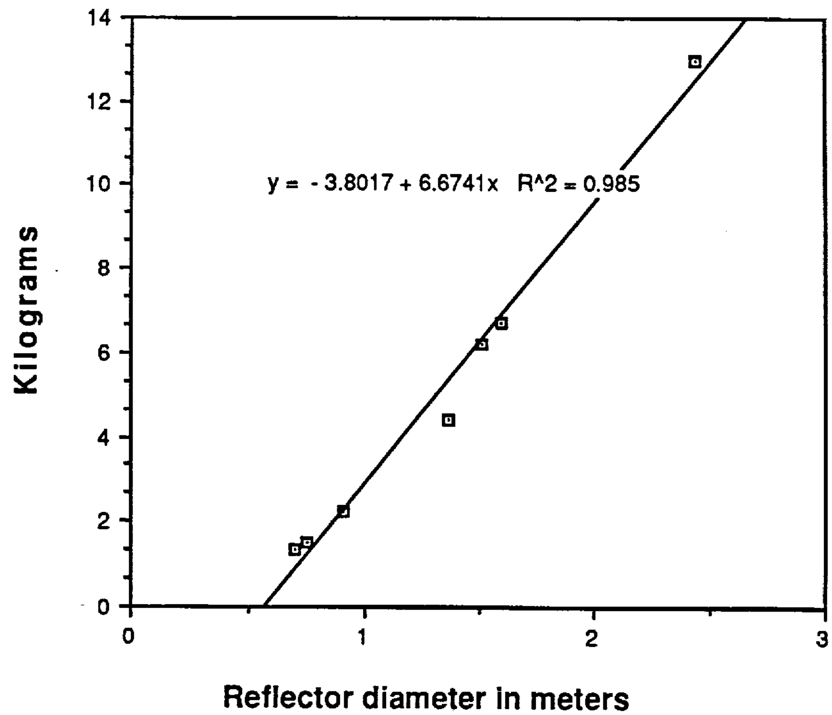

Figure 5–1: Mass Versus Diameter for Solid Carbon Fiber Reflectors

where M is the reflector mass in kilograms and D is the reflector diameter in meters. The range of applicability for this formula is solid carbon-fiber reflectors from 0.5 m to 3 m in diameter.

Figure 5-2 shows the reflector mass versus diameter for mesh type reflectors with diameters of 5 m and larger. The data, which is summarized in Table 5-1, has been compiled from both flown spacecraft antenna systems and prototype developments in various companies in the aerospace industry. The mass includes the reflecting surface and all the necessary mechanical backup structure. A linear relationship is deduced from the data:

$$M = 11.342D - 21.373$$

where M is the reflector mass in kilograms and D is the reflector diameter in meters. The range of applicability for this formula is mesh reflectors from 5 m to 30 m in diameter.

It is interesting to note that for a 4-m diameter reflector which is outside the nominal range of the two equations, the results are in relatively close agreement. The graphite-fiber 4-m MBRA mass prediction is 23 kg and the mesh 4-m MBRA mass prediction is 24 kg.

5.1.3 Mass Properties of Feed Array

The mass of the feed array depends on both the antenna operating frequency and the number of elements in the feed array. The size and the mass of the feed array is, to first order, linearly proportional to the inverse of the frequency. On the other hand it is proportional directly to the number of radiating elements in the array. In addition, it is also a function of the material used in the construction and the technology employed in the beam forming network.

Some representative data were compiled based on feed arrays employing graphite horn radiators and barline-type beam forming networks. The mass of the feed array is plotted versus the number of feed radiating elements normalized by frequency as shown in Figure 5-3.

The relation between the feed array mass and the normalized number of feeds is given by

$$M = 0.9671N - 0.1826$$

where M is the feed array mass in kilograms and N is the normalized number of elements in the feed array, or the number of feed elements divided by the operating frequency in GHz.

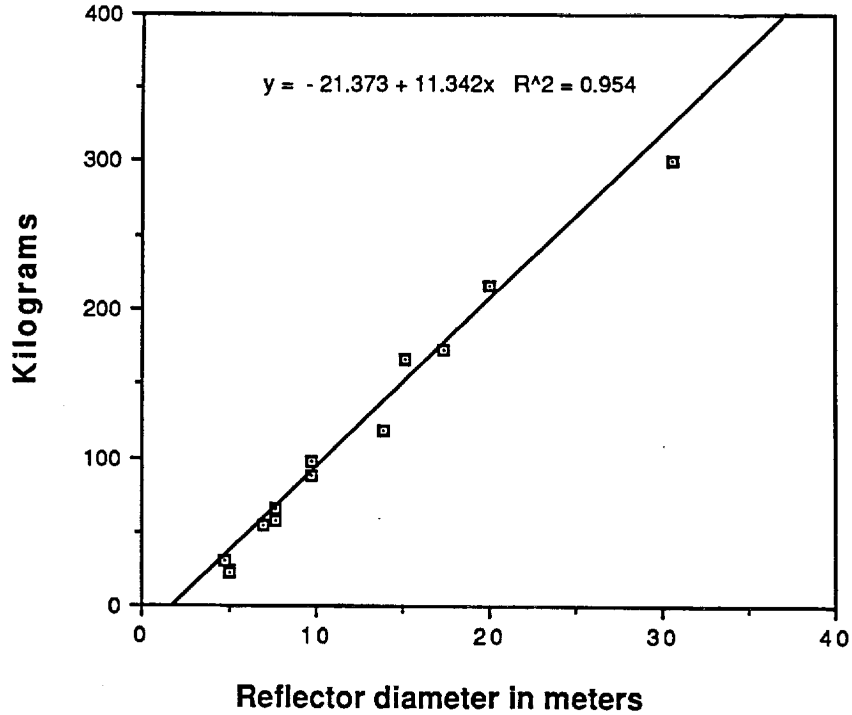

Figure 5–2: Mass Versus Diameter for Mesh Reflectors

Table 5–1: Data Base for Mesh Reflectors — Diameter, Mass, Frequency Band, and Technology

Reflector Diameter (m)	Reflector Mass (kg)	Frequency Band	Technology
4.80	31.0	Ka-band	Radial rib
5.00	24.3	C/K-bands	Geo-truss
5.00	22.5	L-band	Radial rib
7.00	54.0	2 GHz	Prismatic truss
7.62	58.0	15 GHz	Wrap rib
7.62	66.0	20 GHz	Wrap rib
9.75	88.0	15 GHz	Wrap rib
9.75	97.0	20 GHz	Wrap rib
14.00	118.0	18 GHz	Wrap rib
15.20	167.0	18 GHz	Wrap rib
17.40	173.0	18 GHz	Wrap rib
20.00	216.0	0.85 GHz	Wrap rib
30.50	300.0		Cable catenary

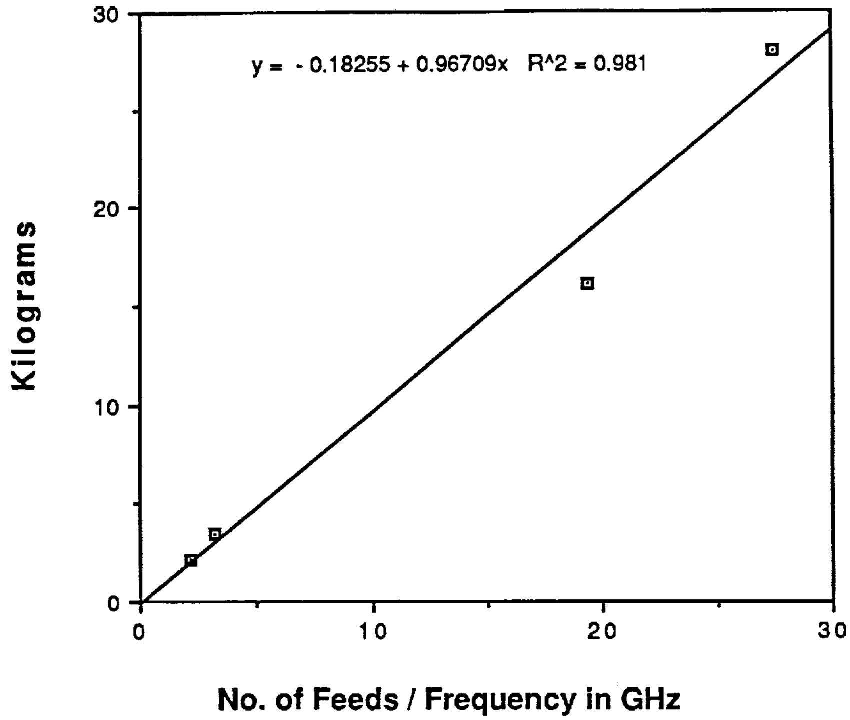

Figure 5–3: Mass vs. Normalized Feed Array Size for Graphite Feeds and Barline Beamforming Network

5.1.4 Power Consumption of Antennas

The multiple beam reflector antenna (MBRA) does not consume power unless it is steerable. The feed array, consisting of feeds, polarizers, diplexers, power dividers and combiners, does not normally consume power. However, an active feed array with ferrite or MMIC devices to reconfigure or point the beam will have power consumption for active components as well as thermal control. These designs tend to be application specific and are not amenable to general modeling.

Thus we consider that the MBRA has no power consumption. (An active element phased array approach to forming multiple beams would have a substantial power consumption, particularly for a transmit phased array with high power SSPA's.)

5.2 Baseband Electronics

5.2.1 Introduction

The discussion of on-board baseband electronics includes equipment for baseband processing and switching as shown in the payload schematic of Figure 5-4. This potentially includes the following items:

- Demodulators
- Input processors
- Switch
- Output processors
- Modulators
- Switch controller

As shown in Figure 5-4, the baseband switch includes input and output processors, switch electronics, and switch controller. The baseband electronics is divided into four parts for purposes of mass and power estimates: (1) demodulators, (2) baseband switch, (3) modulators, (4) structure, and (5) power supply.

All-Digital Technology Assumed

An all-digital approach is assumed for the baseband electronics. Clearly there are alternate technologies such as SAW or optical processors which may be preferred for a particular application, and which may become the technology of choice in the future.

Other Assumptions

Digital systems technology is developing at a rapid pace. Over the next ten years (by the year 2002), mass is

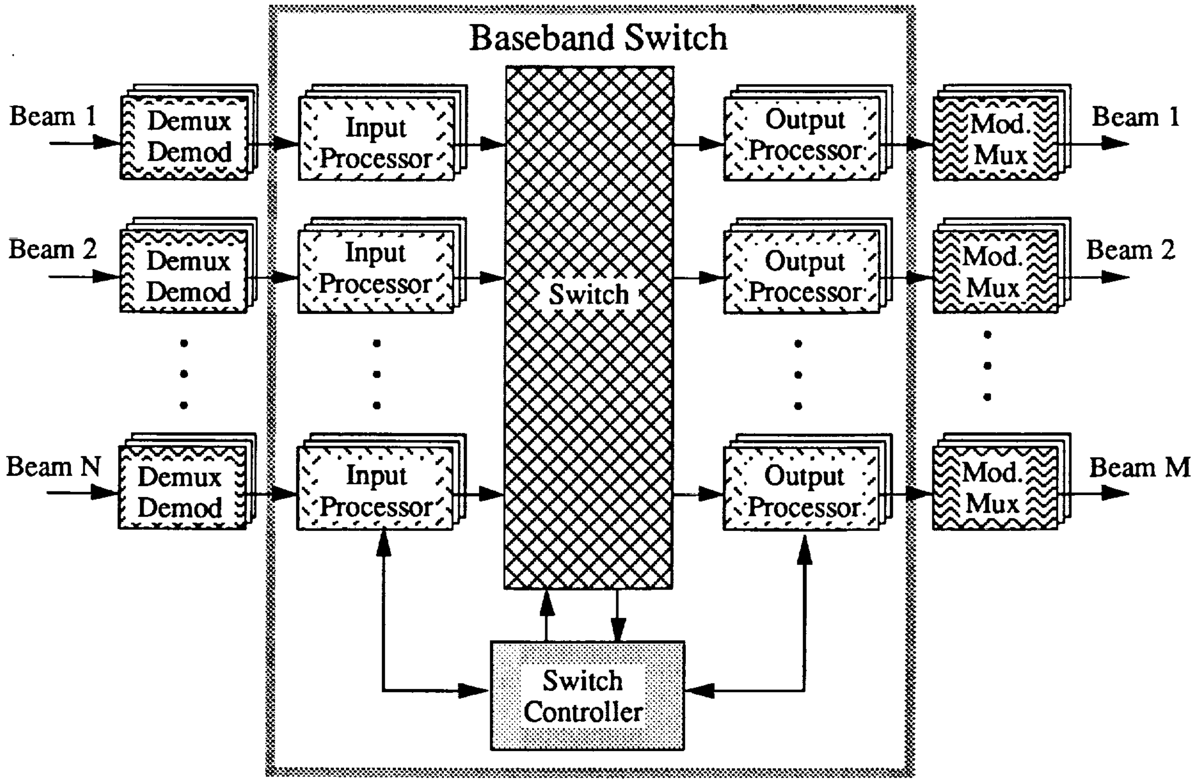

Figure 5–4: Schematic of On-Board Baseband Electronics Equipment

expected to drop by a factor of 30 and power by a factor of 100. This makes it exceedingly difficult to reliably predict baseband electronics mass and power.

The following assumptions are made as a basis of the mass and power estimates:

- 1994 technology time frame. This implies that the satellite launch date would be after 1997.
- 50% reduction from 1991 electronics mass and power figures is assumed. This is a 50% reduction from the present levels of power per gate.

  (Present levels for GaAs are 0.1 mW power/gate, 50,000 usable gate density, and 5 Gb/s speed. Present levels for radiation-hard HCMOS are 12 μW/MHz power/gate, 50,000 usable gate density, and 400 Mb/s speed.)
- Availability of 16 K x 32 SRAMs. (Radiation-hard memory technology has current sizes of 16 K x 4.)
- A regulated power supply is included as a separate part of the baseband electronics. A dc-to-dc conversion efficiency of 85% is assumed.

Key Parameters

The data rate (Mb/s) is the key parameter according to which mass and power of the baseband electronics scales directly. This is expected since the baseband electronics is sorting and routing bits, and thus scales directly according to the number of bits to be processed.

Another key parameter is the number (m) of individual channels which are demodulated. More channels within the same overall data rate results in higher mass and power consumption. Mass and power scale directly according to log m.

5.2.2 Demodulators

Single channel and multichannel demodulators will be considered separately. A given satellite design may have a mixture of demodulator types and/or sizes. The total mass of the demodulators is the sum of the masses of the different types.

Single Channel Demodulators

The mass and power of the single channel demodulators are given by:

$$M = 0.008\ n\ R_{demod}$$

$$P = 0.06\ n\ R_{demod}$$

where M is mass (kg), n is the number of demodulators, R_{demod} is the individual unit data rate (Mb/s), and P is the power consumption (W). $(n \times R_{demod})$ can be replaced by R_{total}, the total baseband electronics throughput data rate.

For example, the demodulators for a 2 Gb/s satellite (assuming 40 each 50-Mb/s demods) would have a mass of 16 kg and power of 120 W.

Multicarrier Demodulators

Multicarrier demodulators (MCDs) also contain the demultiplexers for the multiple carriers. As stated in the assumptions, a digital approach is assumed for the mass and power estimates.

The mass and power of the MCD are given by:

$$M = 0.008\ n\ R_{MCD}\ log(36\ m/R_{MCD})$$

$$P = 0.20\ n\ R_{MCD}\ log(36\ m/R_{MCD})$$

where M is mass (kg), n is the number of MCD's, m is the number of carriers per MCD, R_{MCD} is the total data rate (Mb/s) through an individual MCD, and P is the power consumption (W). Note that $n\ R_{MCD}$ can be replaced by R_{total}, the total baseband electronics throughput data rate.

For example, the MCD's for a 2 Gb/s satellite (assuming 56 each 36-Mb/s MCD's with 18 each 2 Mb/s carriers) would have a mass of 20 kg and power of 502 W.

Figure 5-5 plots total power required by the MCD's as a function of total throughput. It is assumed that a number of MCD's of the same carrier size are used to accommodate the total throughput. Figure 5-5 gives plots for different carrier size MCD's — 64 kb/s, 256 kb/s, 512 kb/s, 2 Mb/s, and 6 Mb/s. The power consumption increases by a factor of 3.55 times as carrier size changes from 6 Mb/s to 64 kb/s.

5.2.3 Baseband Switch

As shown in Figure 5-4, the baseband switch includes input and output processors, switch electronics, and switch controller. Also included are codecs and necessary redundancy. The assumption is made that the inputs and outputs to the switch are in the range of 30 Mb/s to 120 Mb/s (i. e. not lots of little carriers).

The mass and power of the baseband switch are given by the following equations:

$$M = 0.015\, R_{total}$$

$$P = 0.10\, R_{total}$$

where M is mass (kg), R_{total} is the total throughput rate (Mb/s), and P is the power consumption (W).

For example, a 2,000 Mb/s baseband switch would have a mass of 30 kg and power of 200 W.

5.2.4 Modulators

The mass and power of the modulators are given by:

$$M = 0.003\ p\ R_{mod}$$

$$P = 0.003\ p\ R_{mod}$$

where M is mass (kg), p is the number of modulators, R_{mod} is the individual modulator data rate (Mb/s), and P is the power consumption (W). $(p \times R_{mod})$ can be replaced by R_{total}, the total baseband electronics throughput data rate.

For example, the modulators for a 2 Gb/s satellite would have a mass of 6 kg and power consumption of 6 W.

5.2.5 Structure

The structure mass provision is as follows:

$$M = .02\ R$$

where M is structure mass (kg) and R is the total throughput data rate (Mb/s),

For example, the structure for a 2 Gb/s satellite (2,000 Mb/s throughput) would have a mass of 40 kg.

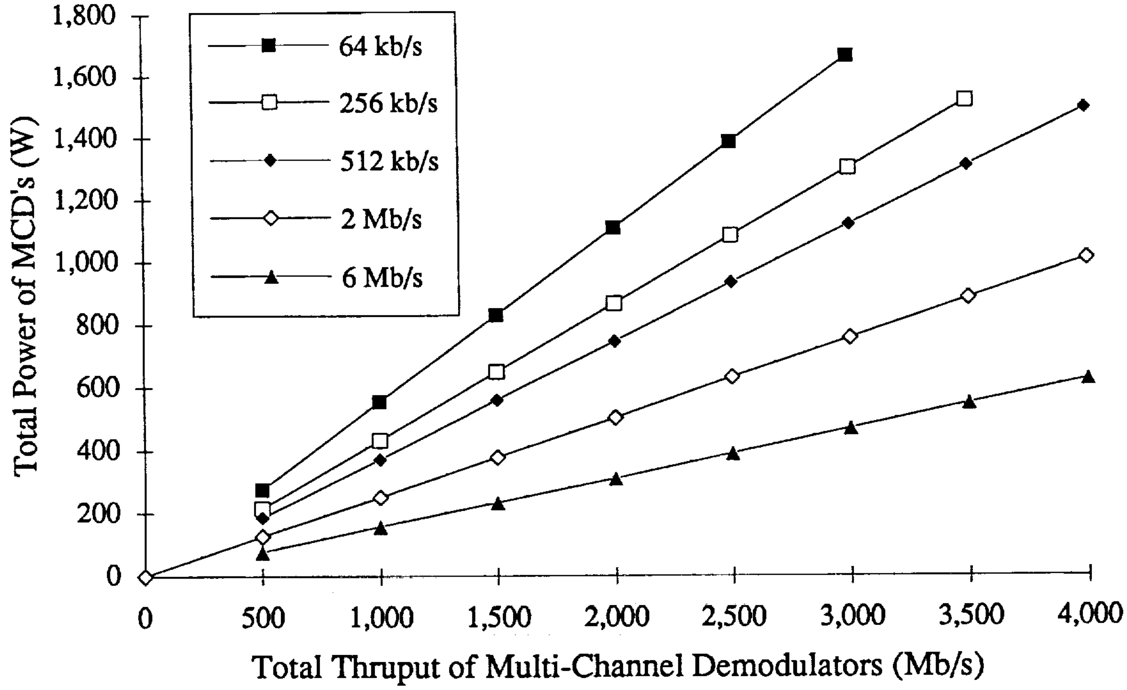

Figure 5–5: Multi-Carrier Demodulator Power vs. Total Data Throughput and Carrier Size

5.2.6 Power Supply

The mass and power of the regulated dc-to-dc power supply are given by:

$$P_{powersupply} = 0.18 \sum P_i$$

$$M = 0.07\ P_{powersupply}$$

where $P_{powersupply}$ is the power (W) consumed by the power supply, P_i is the power consumed by other individual parts of the baseband electronics, R is the individual unit data rate (Mb/s), and M is mass (kg) of the power supply.

For example, if the rest of the baseband electronics uses 700 W, the power supply consumes 126 W and has a mass of 8.8 kg.

5.2.7 Totals for Baseband Electronics

The total baseband electronics mass and power is given by the sum of the values for the demodulators, baseband switch, modulators, structure, and power supply.

Baseband Electronics with Single Carrier Demods

For baseband electronics with single carrier demodulators, the totals are as follows:

$$M_{total} = 0.048\, R_{total}$$

$$P_{total} = 0.192\, R_{total}$$

where M_{total} is total baseband electronics mass (kg), R_{total} is the total throughput data rate (Mb/s), and P_{total} is the baseband electronics power (W). For example, a 2 Gb/s (2,000 Mb/s) total throughput baseband electronics would have a mass of 96 kg and power consumption of 384 W.

Baseband Electronics with MCD's

For baseband electronics with MCD's, the expressions are more complex due to the log(number of MCD carriers) factor. For 2 Mb/s MCD carriers, the expressions become

$$M_{total} = 0.052\, R_{total}$$

$$P_{total} = 0.417\, R_{total}$$

where M_{total} is total baseband electronics mass (kg), R_{total} is the total throughput data rate (Mb/s), and P_{total} is the baseband electronics power (W). For example, a 2 Gb/s total throughput baseband electronics would have a mass of 104 kg and power consumption of 834 W.

For 256 kb/s MCD carriers, the expressions become

$$M_{total} = 0.062\, R_{total}$$

$$P_{total} = 0.629\, R_{total}$$

For example, a 2 Gb/s total throughput baseband electronics would have a mass of 124 kg and power consumption of 1,258 W.

The MCD is currently the most important item in determining the baseband electronics power consumption. As noted earlier under the assumptions, mass is expected to drop by a factor of 30 and power by a factor of 100 over the next ten years. This makes it exceedingly difficult to predict baseband electronics mass and power.

Graphs of Results

Figure 5-6 plots baseband electronics mass as a function of data rate throughput for systems with single carrier demodulators (Demod) and multiple carrier demodulators (MCD). The MCD carrier size is 2 Mb/s for the case plotted. The mass of the baseband electronics with MCD's is 8% higher than that with the single-carrier demodulators (cases plotted). Baseband electronics mass with 256 kb/s carrier MCD's would be 29% higher.

Figure 5-7 plots baseband electronics power as a function of data rate throughput for systems with single carrier demodulators (Demods) and multiple carrier demodulators (MCDs) of 256 kb/s and 2 Mb/s carriers. Baseband electronics with 2 Mb/s carrier MCD's require 2.2 times the power of a single-carrier demodulator system; and the system with 256 kb/s carrier MCD's requires 3.3 times the power.

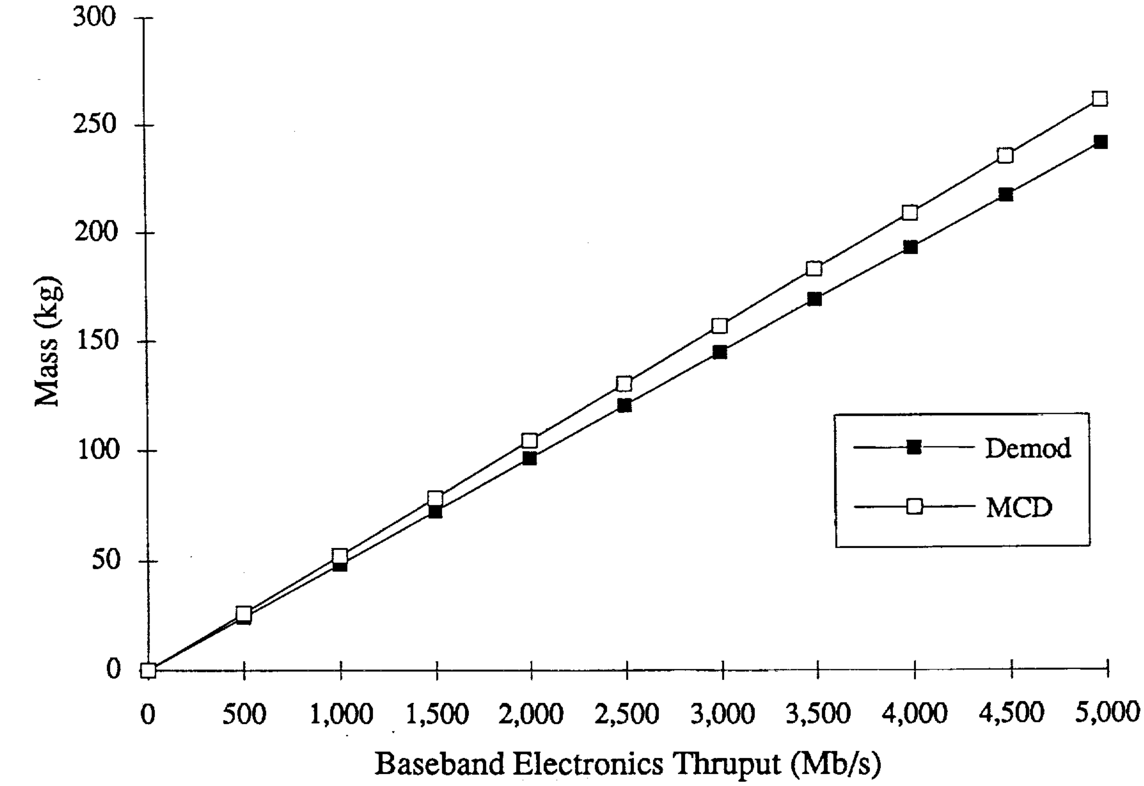

Figure 5–6: Baseband Electronics Mass Versus Total Data Throughput

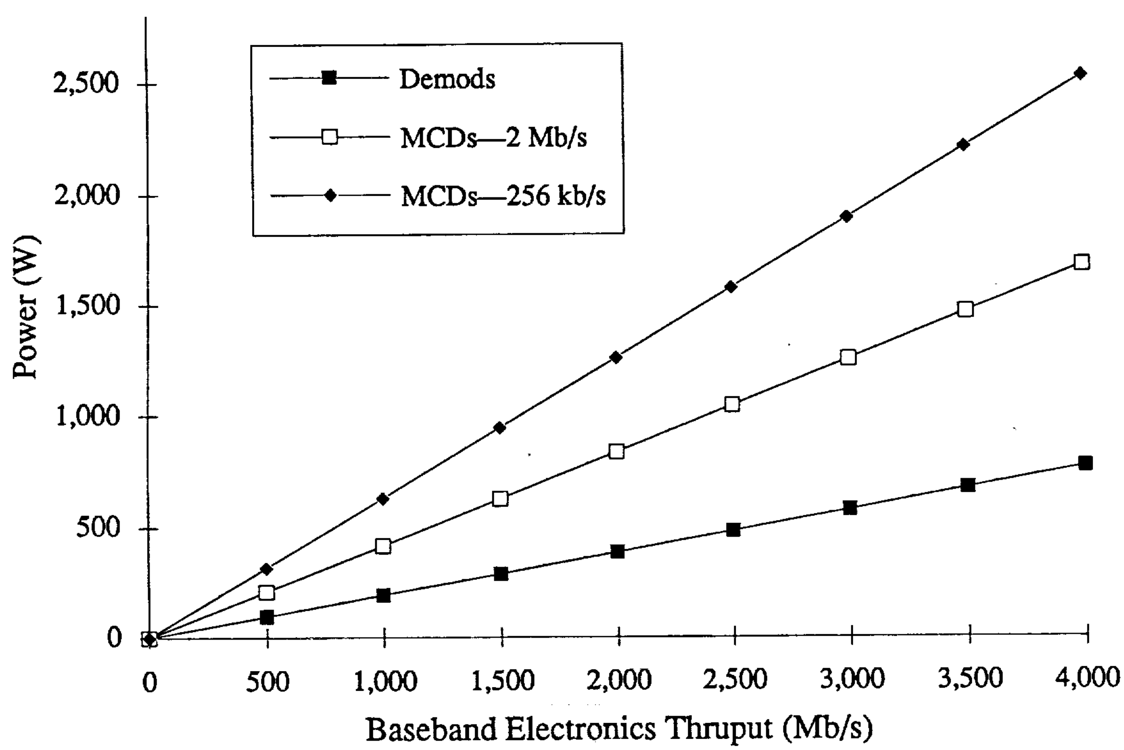

Figure 5–7: Baseband Electronics Power Versus Total Data Throughput

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Chapter 6

Telemetry, Command, and Control

6.1 Introduction

The onboard electronics for state of the art spacecraft perform the telemetry, command, and control (TT&C) functions. As spacecraft become more complex and incorporate more failure modes and contingency action modes, it is essential to provide accurate monitoring and a fast response capability to minimize mission interruption and potentially loss of spacecraft. The following functions are performed by the spacecraft control system: telemetry, tracking and command, (digital), attitude determination and control, thermal control, battery power subsystem management, propulsion management, deployments and antenna positioning, and spacecraft control safety management.

The RF communications equipment necessary to receive and transmit telemetry, command and ranging signals consists of Telemetry, Command and Ranging (TC&R) transponder, antennas, and associated waveguide, coax, and miscellaneous channel components.

This chapter addresses the conceptual mass and power modelling for the spacecraft control electronics and TC&R equipment as a function of a qualitative estimate of telemetry, command and control requirement complexity.

How the modelling effort architecture is laid out.
The logic chart shown in **Figure 6-1** describes the process including input data, performance requirements, decision points, system selection guides, and mass sizing tables and charts necessary to generate a system selection and estimates of mass and power.

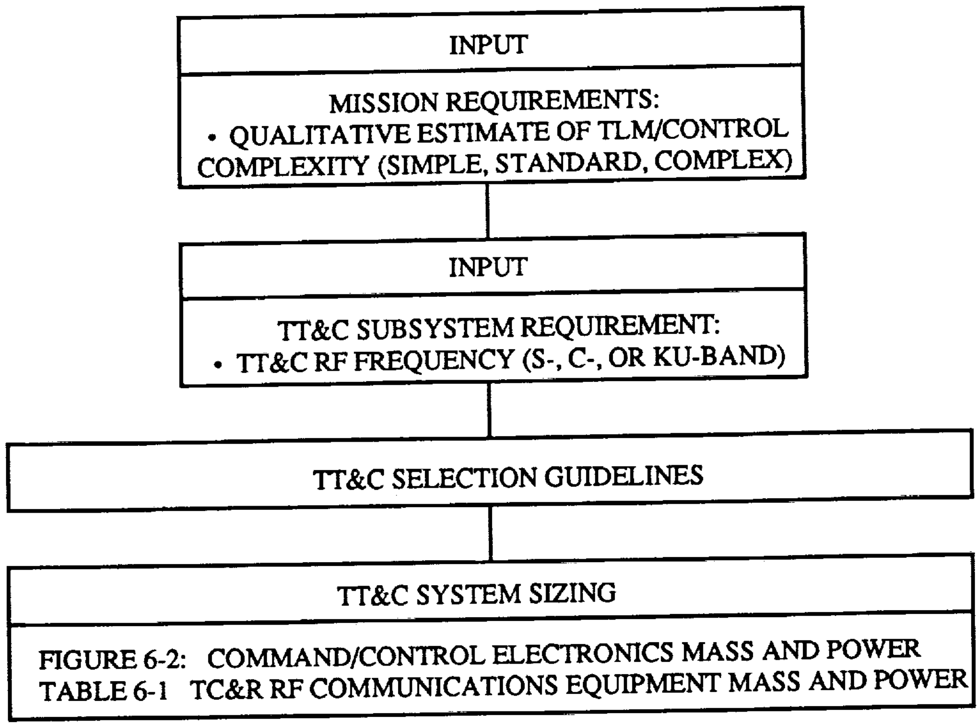

Figure 6-1: Attitude Control System Sizing Flow Diagram

6.2 Mission Requirements

The following mission level requirement is required:

Spacecraft command and control complexity. Qualitative assessment of either a simple, a standard, or a complex spacecraft control and monitoring requirement.

6.3 TT&C System Requirements

The following subsystem level requirement is required:

Operational frequency: S-Band, C-Band, or Ku-Band telemetry and command RF frequency.

6.4 Major Subsystem Descriptions

6.4.1 Spacecraft Control Architecture

The control electronics is an integrated distributed set of electronics implementing control, command, and telemetry functions. The particular architecture being assumed uses a CMOS-SOS microprocessor that implements the MIL-STD-1750A chip set, and a two-wire MIL-STD-1553B data bus to communicate with the distributed control units. The CPU controls the system by implementing firmware programs contained in memory. The remote units are located near the unit equipment to minimize wire harness mass. These distributed units perform the interfacing function between the CPU and the user equipment to implement the telemetry and command functions.

The Attitude Determination and control scheme performs the following:

- Attitude sensor data processing and calibration
- Attitude control mode control
- Momentum management
- Redundancy management
- Attitude determination and control

6.4.2 Telemetry

The Telemetry subsystem collects, formats, and transmits data on the status, performance, environment and health of all units, subsystems and systems in quantities, accuracies, and interval consistent with operation of the spacecraft over its design life. The telemetry subsystem also supports ranging operations. Typically the following information is telemetered:

Status of relays, switches, and valves that determine operational state

- On/Off status of all units connected to the spacecraft bus
- Load currents for all critical units
- Dynamic content of all registers or memories that determine an operational state
- Diagnostic telemetry of microprocessors and memory
- Individual on-off status of thermostatically-controlled heaters
- Communication system gains and frequency states
- Critical performance parameters of TWTAs

6.4.3 Command

The command system will receive, interpret and direct commands so that the proper operation of the spacecraft can be maintained. And the system will facilitate control and recovery of the spacecraft during periods of attitude instability. The command I/O is either a pulse command, relay command, or serial command, and is determined by the nature of the command action.

6.5 Guidelines for the Selection of TT&C System

The digital electronics that controls and monitors spacecraft systems is highly dependant on the complexity of the spacecraft and customer requirements. The primary hardware mass and power driver is the telemetry and command input/output (I/O) requirement. The primary harness mass driver is the configuration of spacecraft equipment and the location and number of control units. As both the I/O requirement and spacecraft configuration increase, the optimal hardware and harness distribution will change to minimize overall system mass. The balance between the mass of the hardware and harness is typically treated as an integrated system level trade-off. Because as the spacecraft size grows, the wire harness could increase much more rapidly than the hardware if the optimal number and location of control boxes is not considered.

A qualitative approach has been implemented to select and size the digital control system mass and power since spacecraft I/O requirements and configuration are not always known at the conceptual stage of design. Three representative programs have been developed which provide an estimate of system mass and power over a range of telemetry/command I/O requirements. These categories range from a simple, a standard, and a complex spacecraft control and monitoring architecture. The simple system would be representative of a medium size, low cost spacecraft with a total requirement of about 1000 I/O points. A standard communications spacecraft would require approximately 2000 I/O points, and a complex spacecraft about 3000 or more I/O points. Therefore, making a qualitative estimate of the spacecraft control and monitoring system complexity will provide sufficient input to estimate system mass and power.

The selection of a RF TT&C system is a mission specific requirement. An S-Band system may be appropriate to utilize the NASA standard SGLS (Space Ground Link System). However, if the communication payload is designed for Ku-Band then it may be more efficient to multiplex the TT&C data with the communication signals. The mass driver in the RF equipment is the transponder, and depending on the location of the antennas relative to the receiver, the signal transmission (waveguide or coax) could be significant. Although the waveguide is smaller and lighter as frequency increases, often the real driver between one system and another is the imposed requirements specific to the mission redundancy and performance.

6.6 Mass Estimating Relationships and System Sizing

6.6.1 Spacecraft Control Electronics

Figure 6-2 gives the mass and power results for three representative spacecraft control electronic systems with differing I/O capabilities. A typical complex spacecraft from a telemetry, command and control point-of-view is the Intelsat series spacecraft. This class of system could require 1500 to 1800 telemetry points (digital status, analog), and 800 to 1000 command points (pulse, relay, serial). Typically, the spacecraft bus will account for nearly 70% of the I/O requirements in a bend-pipe communications spacecraft. A typical standard class telemetry, command and control spacecraft is direct broadcast. This class of system could require 1000 to 1300 telemetry points, and 600 to 800 command points. A typical small class telemetry, command and control spacecraft is a small-sat. The mission emphasis is towards a small architecture, and therefore, would minimize the I/O capacity. This class of system could require 400 to 700 telemetry points, and 300 to 500 command points.

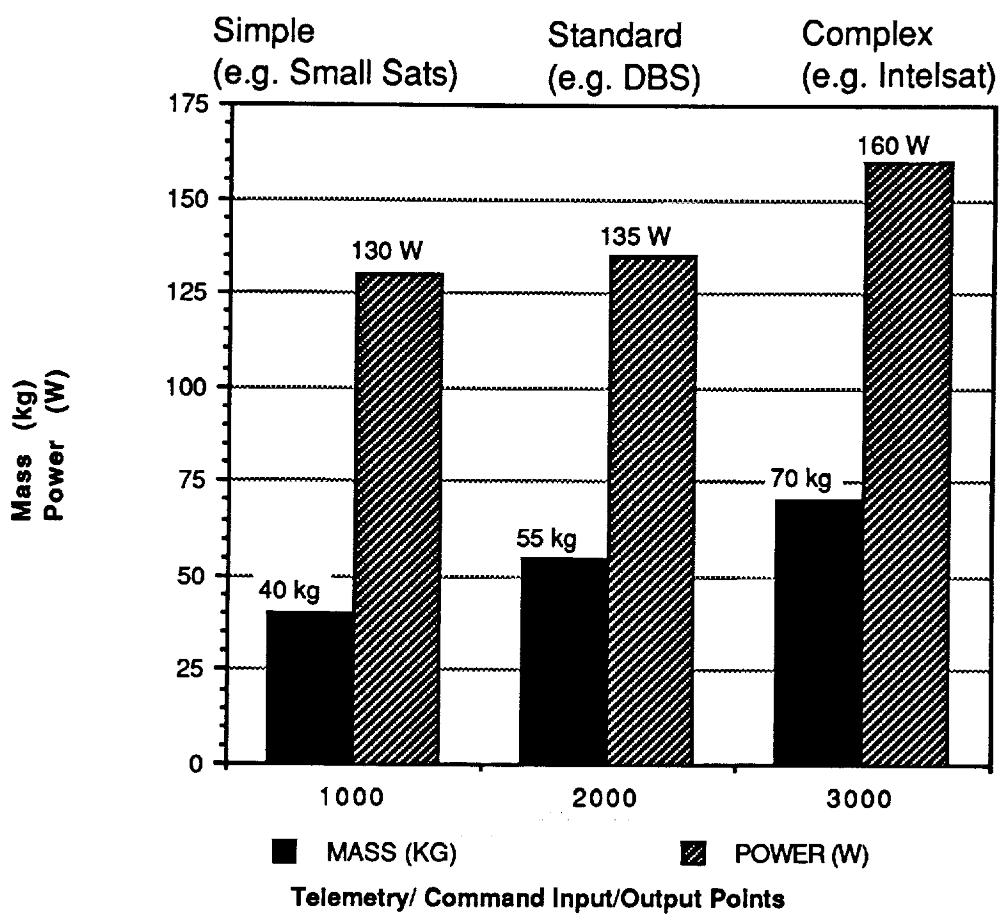

Figure 6-2: Telemetry, Command, and Control Mass and Power

6.6.2 RF Communications Equipment and Antennas

Table 6-1 provides a breakdown of mass and power of three frequency regimes. There variation between is not significantly affected by frequency. The mass of the receivers and transmitters are approximately 2 kg each. The power requirement is dependant on data rate and the antenna gain in the system design. The data rate for the considered is 250 bits per second (bps) for the command receivers, and between 1000 bps and 5000 bps for the telemetry transmitters.

Table 6-1: TT&C RF Communications Mass and Power

Frequency Band	Component	Mass Kg	Power W
S-Band (SGLS)	Omni Antennas (2) Telemetry Transmitter (2) Command Receiver (2) Waveguide/Coax Misc	18	28
C-Band	Al Horn Antennas (4) Telemetry Transmitter (2) Command Receiver (4) Waveguide/Coax Misc	17	28
Ku-Band	Al Horn Antennas (3) Telemetry Transmitter (2) Command Receiver (2) Waveguide/Coax Misc	16	18

Chapter 7

Miscellaneous

7.1 Introduction

Mechanical and electrical integration represent a line item to account for the physical inter-connectivity between various subsystem and payload components. When wires, nuts, bolts, and small brackets are not part of a specific subsystem, they are accounted for in these categories. There impact on the spacecraft mass budget is typically overlooked at the conceptual definition design stages. However, they do represent a significant percentage of the spacecraft mass by the shear numbers of little parts in these miscellaneous sections. The best method of approximating their contribution is by the relative mass contribution compared with representative past programs. The following mass estimates have been derived from geosynchronous spacecraft programs such as Intelsat VII and Superbird (SCS).

7.2 Mission Requirements

The following mission level requirement is required:

Overall spacecraft dry mass.
This spacecraft mass is the sum of the payload and spacecraft bus masses excluding propellant. This is an iterative process because the miscellaneous mass is a component of the overall spacecraft dry mass. But, it is a linear approximation and converges within two to three iterations.

7.3 Electrical Integration

An allocation of 4.5% of the spacecraft mass will account for electrical integration hardware.

The electrical integration typically includes electrical harness for various panels and control functions. These include spacecraft bus main harness, payload harness, peripheral harness, and pyrotechnic and electroexplosive device harness.

7.4 Mechanical Integration

An allocation of 3.0% of the spacecraft mass will account for mechanical integration hardware.

The electrical integration typically includes brackets for various subsystems and hardware for the integration of boxes to panels, and panels to panels. These include propulsion brackets and supports, payload brackets and spacers, attitude control brackets/shims and covers, antenna brackets and shields, and various inserts, rivnuts and integration hardware.

7.5 Thermal Control

The spacecraft thermal control system was not included in the task statement. However, it does represent a significant contribution to the overall spacecraft mass and power budgets. At the conceptual definition stage of design and system definition, a simple relationship is adequate to approximate the thermal control system mass and power relative to the thermal dissipated power. This assumption is reasonable for a typical 3-axis stabilized geosynchronous communications spacecraft. The following relations are accurate with 20% given the above assumptions, and have been provided as an estimate for completeness.

$$M_{TCS} = 0.0295 * P$$

$$P_{TCS} = 0.0546 * P$$

where, M_{TCS} is the estimated mass of the spacecraft thermal control system, P_{TCS} is the estimated power of the spacecraft thermal control system, and P is the total end of life power requirement.

7.6 Mass Contingency

An allocation of 8.0% of the spacecraft mass will account for overall weighted average of mass growth for new programs.

In practice, mass allocations are determined by considering all three elements of design maturity: estimated, calculated, and actual. It is typical that items based on actual measurements vary less than 1% from flight to flight and program to program. When component estimates are supported by detailed calculations of released drawings and parts lists, an uncertainty of approximately 5% is realized. And items in which the mass is based on engineering estimates vary approximately 10%.

Chapter 8

Launch and On-Orbit Propulsion

8.1 Introduction

Reaction control system and fuel sizing task describes the propulsion system sizing considerations for a communications satellite designed for a geosynchronous orbit. Analytical models are developed to size component mass, power (power processing units only), and propellant budgets. Mass estimates are presented for tankage, thrusters, valves, plumbing, and fuel. Also, a methodology has been developed to assist the user in choosing an appropriate system selection based on the top level mission parameters. Four propulsion systems architectures have been identified as the top candidates for GEO communication satellites. These system combinations include: a bipropellant (MMH/NTO), a bipropellant augmented with either arcjets, or xenon ion thrusters, and a dual mode bipropellant/monopropellant system (N2H4/NTO) augmented by arcjet. In each case the on-board propulsion system will provide for apogee injection from GTO to GEO, stationkeeping, attitude control, dispersion correction, repositioning, and deorbit.

Modelling architecture layout. A logical flow chart shown in **Figure 8-1** describes the process necessary to generate a system selection and estimate of mass and power. This process includes input data, performance requirements, decision points, system selection guides, and mass sizing tables and charts. Since spacecraft dry mass is used as a key parameter for system sizing, and since the model output modifies the satellite dry mass, an iterative approach is required when sizing the spacecraft.

Communications satellite assumptions are used to generate system architectures. At a conceptual design stage it is reasonable to neglect second-order design parameters. However, when sizing only for mass and power important issues may be overlooked which can significantly affect performance, reliability, and operations. To minimize the chance for such an occurrence, two safeguards have been provided; A set of integrated propulsion systems have been defined to ensure that each propulsive maneuver is adequately accounted for, and component mass estimates account for redundancy sufficient to remove any single-point failures.

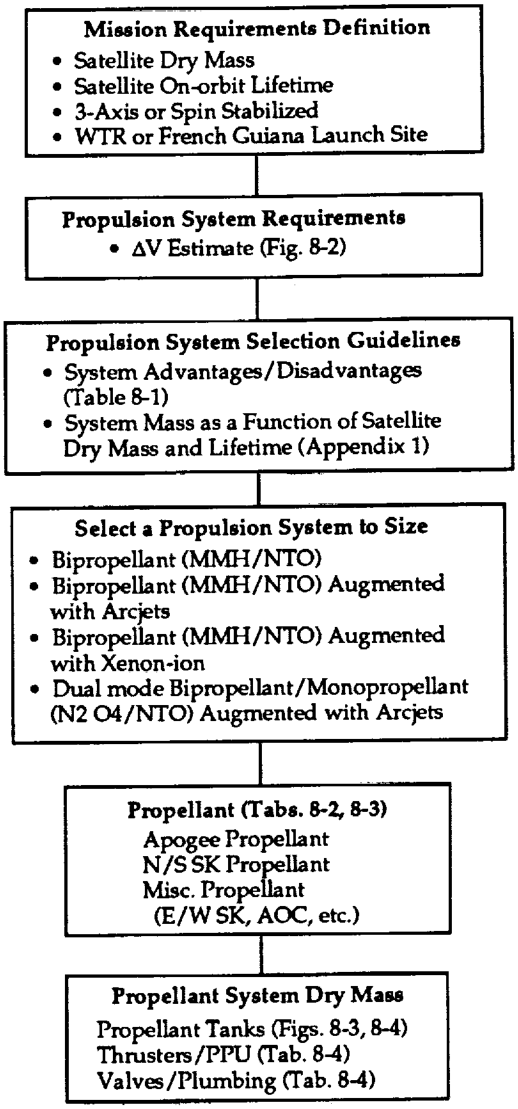

Figure 8-1: Propulsion System Flow Diagram

Why the cold gas and monopropellant systems were not modelled. The trend of GEO communications satellites is toward higher power, and increased spacecraft mass. This increase is a result of high launch costs per pound of spacecraft, increasing demand for communications capacity, and a limited number of orbital geosynchronous slots. A large spacecraft will maximize the payload mass fraction (payload mass to satellite dry mass ratio), and lower the specific cost of the space system. For these same reasons, spacecraft lifetimes have increased from an average of 5-7 years to 10-15 years.

Propulsion system performance improvements offer a large potential mass savings for a GEO communications satellite. Therefore, the cold-gas and conventional monopropellant systems have not been modelled. It has been shown that there is a mass benefit using the bipropellant propulsion system over a monopropellant for small-to-medium sized GEO spacecraft.

8.2 Mission Requirements

Many of the detailed mission requirements that influence the design of a GEO communications satellite propulsion system are not addressed. The only design drivers that are treated as independent variables are: spacecraft dry mass, ton-orbit lifetime, method of attitude stabilization, and launch site. The dry mass, lifetime, and launch site drive the propellant mass, and the system selection and propellant mass drive the tankage. The spacecraft attitude stabilization, either 3-axis or spin stabilized, drive the quantity and mass of the thrusters, valves and plumbing.

Spacecraft life is defined as the maneuver life needed to size stationkeeping and attitude control propulsion requirements.

Estimate of spacecraft dry mass is defined as the spacecraft weight without propellant. It includes all payload and spacecraft bus subsystems.

Orbit design. The on-board spacecraft propulsion system requirements are very sensitive to launch vehicle performance and the launch site location. The launch site latitude usually determines the inclination of the transfer orbit, and therefore, the plane change requirements for the spacecraft. The launch vehicle performance for this study assumes the spacecraft is inserted into a GTO with apogee altitude equal to GEO. Therefore, no perigee velocity augmentation (PVA) is incorporated.

Spacecraft stabilization. The propulsion system layout differs for 3-axis and spin stabilized spacecraft. Twelve thrusters are usually required for a 3-axis stabilized spacecraft, and 6 thrusters for a spin stabilized spacecraft.

8.3 Propulsion System Requirements

The propulsion system requirements generally consist of providing for orbit control forces in and out of the orbital plane, attitude control torques about all spacecraft axes, minimizing plume impingements, and in general operating without disturbing the pointing performance of the spacecraft. These generic requirements have been incorporated into the four propulsion system options. Thruster quantity and configuration have been accounted for to satisfy the redundant maneuver requirements necessary to satisfy a 10-15 year mission. **Figure 8-2** summarizes the Delta-V required for the maneuvers discussed below.

Apogee maneuver. The apogee maneuver is the largest single maneuver that is required by the spacecraft. The maneuver is usually accomplished within the first week after launch, and it circularizes the orbit from the highly eccentric GTO. It requires a propellant supply approximately equal to 67% to 83% of the spacecraft dry mass using a state-of-the-art bipropellant system. Electric propulsion systems were not considered for this maneuver because of the excessive time delay that would result before the spacecraft would be operational.

Lifetime Dependant Maneuvers

• North/south Stationkeeping	47 m/s/year
• East/west SK & Attitude Control	5 m/s/ year

Lifetime Independant Maneuvers

• Apogee Injection (assumes due-east launch, and no PVA)	
Eastern Test Range Launch	1,800 m/s
French Guiana Launch	1,500 m/s
• End-of-Life Maneuver	10 m/s
• Reposition Maneuvers	(see chart)

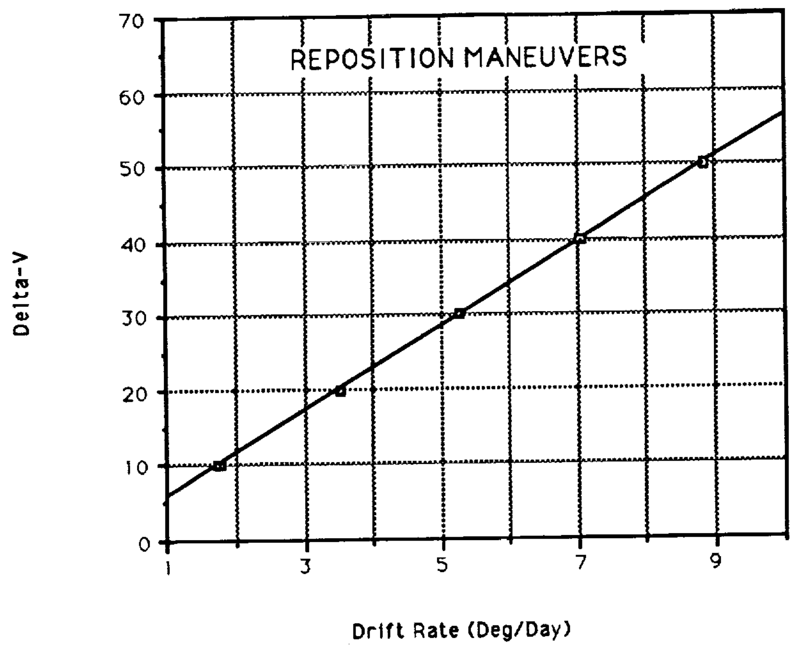

Figure 8-2: Delta-V Requirements for a GEO Communications Satellite

Stationkeeping requirements (N/S, E/W). North/south stationkeeping maneuvers represent the greatest on-orbit maneuver requirement. The natural tendency of the spacecraft is to drift in inclination approximately 0.865° per year as a result of solar and lunar gravitational forces. A maneuver requirement of 47 m/s per year is a reasonable approximation to control the more dominant terms over a typical solar cycle.

East/west stationkeeping maneuvers are required to counteract an Earth oblateness perturbation which, if uncorrected, would cause the satellite to drift in longitude relative to an Earth fixed target. The magnitude of the east/west correction is a function of both satellite longitude, and satellite area to mass ratio. Although the location and configuration of a GEO satellite are not the same, the correction maneuvers are generally small and on the order of 3 to 5 m/s per year.

Reposition maneuvers. Reposition maneuvers are mission specific. The requirement changes depending on the application. The propellant requirement is a function of the rate at which a spacecraft must be repositioned (degrees per day). Therefore, a separate chart, shown in **Figure 8-2**, plots the Delta-V requirement for a single reposition as a function of reposition rate requirement. The smaller the reposition rate (longer reposition time), the lower the maneuver requirement. This Delta-V requirement should be added to the total maneuver requirement for the spacecraft.

EOL maneuver is assumed:. An end-of-life (EOL) maneuver of 10 m/s is included for a one-time maneuver to deorbit the spacecraft from the geosynchronous orbit.

AOC requirements. It is assumed that the attitude control system controls disturbance torques by actuators such as momentum wheels, reaction wheels, magnetic torquers, etc.. The use of thrusters are assumed for momentum management and for attitude control during the stationkeeping maneuvers.

8.4 Propulsion System Options and Descriptions

The Propulsion systems are sized for both 3-axis and spin stabilized geosynchronous communications satellites. In all of the systems, a bipropellant was assumed for satellite injection into geosynchronous orbit. The electric propulsion options are only considered for the north/south stationkeeping maneuvers which represent more than 80% of the on-orbit requirement. It has been assumed that the propulsion system operates in a pressure regulated mode during the apogee injection maneuver phase, and that all other maneuvers operate in a pressure blowdown mode.

Bipropellant: Monomethylhydrazine (MMH) / Nitrogen Tetroxide (NTO)
This bipropellant propulsion system is a flight qualified system that has flown since the early 1980s. The combination of the fuel (MMH) and oxidizer (NTO) is hypergolic; thus it requires no ignition system. The fuel and oxidizer are fed under pressure to the desired thrusters. Two titanium propellant tanks of equal volume are assumed to store the fuel and oxidizer, and one or two pressurant tanks are sized to provide the necessary Helium pressurant. Twenty-two Newton (N) thrusters provide redundant force and torque control, and one 490 N apogee thruster provides orbit circularization.

Bipropellant (MMH/NTO) with electrostatic ion augmentation. The MMH/NTO system is augmented with a Xenon-ion propulsion system to perform the north/south stationkeeping maneuvers. The ion system consists of four 23 mN ion thrusters, two Xenon propellant tanks, power and signal processing units, and the necessary valves and plumbing.

Experimental spacecraft including the SERT and ATS-VI spacecraft have flown ion propulsion. The first Xenon ion system to be operated in geosynchronous orbit will be the Japanese experimental spacecraft, ETS-VI, scheduled for launch in 1993.

Bipropellant (MMH/NTO) with arcjet augmentation. The MMH/NTO system is augmented with a arcjet propulsion system to perform the north/south stationkeeping maneuvers. The arcjet system consists of four 0.2 N thrusters, two hydrazine (N2H4) propellant tanks, power processing units, and the necessary valves and plumbing. Arcjet propulsion is the baseline on the Telstar-IV spacecraft scheduled for first launch in 1992/93.

Dual-Mode Bipropellant: N2H4 / N2O4 with Arcjet Augmentation
The dual-mode bipropellant system uses hydrazine instead of monomethylhydrazine (MMH). The system incorporates the performance of a bipropellant for the high impulse apogee injection, and then relies on monopropellant for on-orbit operations. The addition of the hydrazine arcjets for north/south stationkeeping provides a high performance system that minimizes the tankage complexity by using the existing fuel tanks of the bipropellant system.

8.5 Guidelines for the Selection of a Specific Propulsion System

There are many factors that influence the selection of a propulsion system technology. Probably the most visible is the mass savings that can be realized with improved propulsion system performance. However, increasing performance usually increases the complexity and propulsion system dry mass. Other significant design factors include:

- System development and qualification costs,
- Recurring costs,
- Overall system reliability and risk,
- Structure, power and thermal subsystem implications,
- Logistics impact for on-orbit operations.

These general considerations are summarized in **Table 8-1**, which lists the advantages and disadvantages of each of the candidate propulsion systems. Even though these considerations weigh heavily in the selection process, the bottom line is usually launch vehicle related mass requirements. If a satellite can be launched on a smaller launch vehicle because of higher performance and lower weight satellite subsystems, then a cost savings generally results. This becomes a more obvious trade-off when more than one satellite is being procured.

Table 8-1: Propulsion System Comparisons

ADVANTAGES	MONOPROPELLANT HYDRAZINE / *DISADVANTAGES*
• Low cost • Hardware simplicity • High reliability • Reasonably predictable performance • Flight maturity	• Low performance Isp ~ 220 sec (can be offset by electrically augmented catalytic thrusters) • Requires solid motor for apogee firing
BIPROPELLANT MMH/NTO	
• Intermediate performance Isp ~ 300s • Hypergolic propellant - no ignition system • Offers integration benefits with high thrust apogee injection system	• More complex tankage and flow control • Performance can be degraded due to mixture ratio variation in a blow down mode • Extra hardware makes system more expensive than monopropellant
HYDRAZINE ARCJET	
• Ease of integration to a monopropellant or dual mode system • Significant performance advantages (Isp ~ 500s)	• High power required • No inflight operational experience • Higher dry mass • Will not operate with MMH
ELECTROSTATIC-ION	
• Order of magnitude increase in performance Isp ~2,500-4,000) • Lower thrust magnitudes decrease spacecraft disturbances • High efficiency	• High power required • Lower thrust magnitude increases thruster on-time • High fixed mass of thruster and power processor Very complicated system with more failure modes • than chemical propulsion systems
DUAL MODE BIPROPELLANT/MONOPROPELLANT	
• Ability to use high performance bipropellant apogee system with simple, low cost hydrazine monopropellant for low impulse maneuvers. • Wide selection of monopropellant thrusters available. • Can easily accept Arcjets	• Very limited flight data with N2O4/NTO Bipropellant apogee system • Low performance monopropellant system • Mixture ratio results in non-optimum propellant packaging

The propellant mass savings of a higher performance propulsion system must overcome the higher system dry mass. This is primarily a function of satellite dry mass and the design life of the satellite. As these variables increase, the more likely that the higher performance system will become advantageous due to the increased maneuver requirements.

Appendix 1, Propulsion System Selection Guideline Charts, contains five figures (numbered 1A through 1E) that show the mass trends for each of the propulsion systems, as a function of satellite dry mass and lifetime. At low values of mass and lifetime, the electric propulsion system dry mass is not competitive. But as mass and lifetime increase, the electric propulsion systems are clearly superior. These charts should be used as a guide in selecting a specific propulsion system. To generate these curves, simplifying assumptions were made which compromise the absolute value of the charts. However, the relative trends between the four systems are valid.

These charts show the lowest propulsion system choices are the bipropellant (MMH/NTO) at low values of dry mass and lifetime, and the bipropellant system w/ion at higher dry mass and lifetime. Although this is a reasonable result from a mass point of view, other considerations are important in the system selection process. These issues were described above, and in **Table 8-1**.

8.6 Mass Estimating Relationships and System Sizing

8.6.1 Fuel Computation

The propellant mass calculation is derived in many spacecraft design and rocket propulsion text books (e.g. Agrawal, B. N., "Design of Geosynchronous Spacecraft", Prentice Hall, 1987). A series of plots could be generated as a reference, but it is easier to use the equation directly.

$$M_p = M_{final}(e^{(DeltaV/Isp/g)} - 1) \quad \textbf{Equation 8-1}$$

where

DeltaV	=	Velocity increment (m/s)
M_p	=	Propellant mass (kg)
Mfinal	=	Final spacecraft mass or dry mass (kg)
I_{sp}	=	Propulsion system specific impulse (lbf/lbm/sec)
g	=	9.81 m/s/s

Table 8-2 is a list of the propulsion system specific impulse (I_{sp}) performance data for each class of maneuver, and **Table 8-3** shows a typical propellant budget format. The apogee maneuver uses the large apogee thruster for all the systems. The north/south stationkeeping maneuver utilizes the hydrazine arcjets for systems (2) and (4), and Xenon ion for system (3). The miscellaneous category accounts for the smaller impulse maneuvers such as east/west stationkeeping, attitude control, dispersion corrections, EOL maneuver, and residuals.

Table 8-2:
Propulsion System Specific Impulse Performance Data (lbf/lbm/s)

System	Apogee Maneuver	NSSK	Misc.
1) Bipropellant (NTO/MMH)	311	290	280
2) NTO/MMH w/ Arcjets	311	500	280
3) NTO/MMH w/ ion	311	2800	280
4) Dual Mode (NTO/N2O4) with Arcjets	314	500	220

Table 8-3: Typical Propellant Budget

Event	Thruster	Delta V (m/s)	ISP (sec)	Mass Change (kg)	Final Mass (kg)
Spacecraft Total Weight					2,100.0
Less Adapter (B-1)				69.0	2,031.1
Apogee Maneuver	Main	1,800.0	310.8	905.8	1,125.2
Pre On Station AOC				1.0	1,119.7
NSSK First 5 Years	AOC	272.6	289.0	102.7	1,017.1
EWSK/AOC	AOC	29.0	287.7	10.4	1,006.7
Station Reposition (3°/day)	AOC	17.0	287.7	6.0	1,000.6
NSSK Remainder	AOC	0.0	289.0	0.0	1,000.6
End-of-life De-orbit	AOC	10.0	287.7	3.5	997.1
On Orbit Dispersions				4.4	992.7
Residuals/Holdups				8.4	984.3
Total Spacecraft Propellant				**1,046.8**	
Pressurant				3.9	980.4
Dry Spacecraft (end-of-life)					**980.4**

0.1° inclination deadband; on-orbit operational lifetime = 5.8 yr

8.6.2 Propulsion System Dry Mass

The propulsion dry mass includes thrusters, tanks, power processing units, and plumbing and valves. **Table 8-4** is a spread sheet for each of the propulsion system dry mass components. Some line items are constant values, and some require reference to indicated tables.

Tankage. **Figures 8-3 and 8-4** are used to size the propellant tank mass. **Figure 8-3** is a plot of propellant tank spherical diameter versus propellant mass for three different tank combinations. The proper use of **Figure 8-3** is described below to calculate the required tank diameter. Once the diameters for each of the tanks are calculated, **Figure 8-4** is used to determine tank mass from the diameter. These curves are valid for all liquid propellant tanks. The gaseous xenon tanks simply use a 15% mass fraction to size the xenon tank mass.

<u>Bipropellant (MMH/NTO).</u> The bipropellant curve in **Figure 8-3** sizes the diameter for two identical tanks to store the MMH and the NTO. The total calculated propellant mass from **Equation 8-1** is used.

<u>MMH/NTO with arcjets.</u> The same bipropellant curve is used to size the tank diameters for the bipropellant maneuvers. However, the propellant mass should not include the NSSK propellant. The hydrazine arcjet tanks are sized using the NSSK propellant calculated above, and using the N2H4 curve in **Figure 8-3**.

<u>MMH/NTO with ion.</u> The same bipropellant curve in **Figure 8-3** is used to size the tank diameters. However, the propellant mass should not include the NSSK propellant. The Xenon ion tanks are sized using a propellant mass fraction estimate. Xenon tank mass is equal to 0.15* Xenon propellant mass.

Dual Mode with arcjets. These tanks are more complicated to size because the fuel tanks (N_2H_4) are used for all maneuvers, whereas the oxidizer tank is used only for the apogee maneuver. Therefore, the apogee maneuver propellant must be split up into oxidizer mass and fuel mass using the mixture ratio (0.8) of the apogee thruster. The fuel mass used to size the N2H4 tanks using **Figure 8-3** is the following:

$$M_{fuel(N2H4)} = M_{apogee\ prop}/1.8 + M_{all\ other\ fuel}$$

where	$M_{fuel(N2H4)}$	=	total system fuel mass used in **Figure 8-3**
	$M_{apogee\ prop}$	=	total apogee propellant calculated in **Eq. 8-1**
	$M_{all\ other\ fuel}$	=	total propellant for NSSK and misc. mnvrs

The oxidizer tank is a single tank which contains sufficient oxidizer to perform the apogee maneuver. The mass of the oxidizer used in **Figure 8-3** to calculate the tank diameter requirement is the following:

$$M_{oxidizer} = M_{apogee\ propellant} - M_{fuel(N2H4)}$$

where, $M_{oxidizer}$ = total oxidizer mass used in **Figure 8-3**

Thrusters: Thruster quantity and fixed mass is listed in **Table 8-4** for each propulsion system.

Valves & Plumbing: Propulsion system valves and plumbing were estimated using typical data for the bipropellant system. A value of 2 kg per propellant tank is used for valve mass, and 0.5 kg per thruster is used for plumbing mass.

Table 8-4: Propulsion Dry Mass Summary

Propulsion System	3-Axis Stabilized Qty.	3-Axis Stabilized Unit Mass (kg)	3-Axis Stabilized Total Mass (kg)	Spin Stabilized Qty.	Spin Stabilized Unit Mass (kg)	Spin Stabilized Total Mass (kg)
1. Bipropellant (MMH/NTO)						
Apogee Thruster (490 N)	1	3.62	3.62	1	3.64	3.64
AOC & SK Thrusters (22N)	12	0.85	10.20	6	0.85	5.10
Pressurant Tank		.01*prop mass		1	.01*prop mass	
Propulsion Tanks	2	(Figs. 8-3, 8-4)		2	(Figs. 8-3, 8-4)	
Plumbing/Valves	1	10.00	10.00	1	7.00	7.00
2. Bipropellant w/arcjets						
Apogee Thruster (490N)	1	3.64	3.64	1	3.64	3.64
AOC Thruster (22N)	8	0.85	6.80	4	0.85	3.40
Arcjet Thruster (0.2N)	4	1.32	5.28	2	1.32	2.64
Arcjet power & signal conditioning	4	4.52	18.08	2	4.50	9.00
Pressurant Tanks		.01*prop mass		1	.01*prop mass	
Tanks (BiProp)		(Figs. 8-3, 8-4)		2	(Figs. 8-3, 8-4)	
Tanks (N204 Arcjet)		(Figs. 8-3, 8-4)		2	(Figs. 8-3, 8-4)	
Plumbling/Valves	1	14.00	14.00	1	11.00	11.00
3. Bipropellant w/xenon-ion						
Apogee Thruster (490N)	1	3.64	3.64	1	3.64	3.64
AOC Thruster (22N)	12	0.85	10.20	4	0.85	3.40
Xenon-Ion Thrusters (0.23N)	4	3.00	12.00	2	3.00	6.00
XIPS (PPU, VDE, PMU,MFC,TCU)	1	57.60	57.60	1	32.40	32.40
Pressurant Tanks	1	.01*prop mass		1	.01*prop mass	
Tanks (BiProp)	2	(Fig. 3, 4)		2	(Figs. 8-3, 8-4)	
Tanks (Xenon)	2	.15 * xenon mass		2	.15 * xenon mass	
Plumbing/Valves	1	16.00	16.00	1	11.00	11.00
4. Dual mode bipropellant/monoprop (N204/NTO) w/arcjets						
Apogee Thruster	1	4.20	4.20	1	4.20	4.20
AOC Thrusters	8	0.68	5.44	4	0.68	2.72
Arcjet Thruster (.2N)	4	3.00	12.00	2	8.00	16.00
Arcjet power & signal conditioning	4	4.52	18.08	2	4.52	9.04
Pressurant Tank	1	.01*prop mass			.01*prop mass	
Fuel Tank (N204)	2	(Figs. 8-3, 8-4)		2	(Fig. 3, 4)	
Oxidizer Tank (NTO)	1	(Figs. 8-3, 8-4)		1	(Fig. 3, 4)	
Plumbing/Valve	1	12.00	12.00	1	9.00	9.00

Note: The propulsion dry mass does not include structure, thermal, or power subsystem modifications necessary to implement electric propulsion.

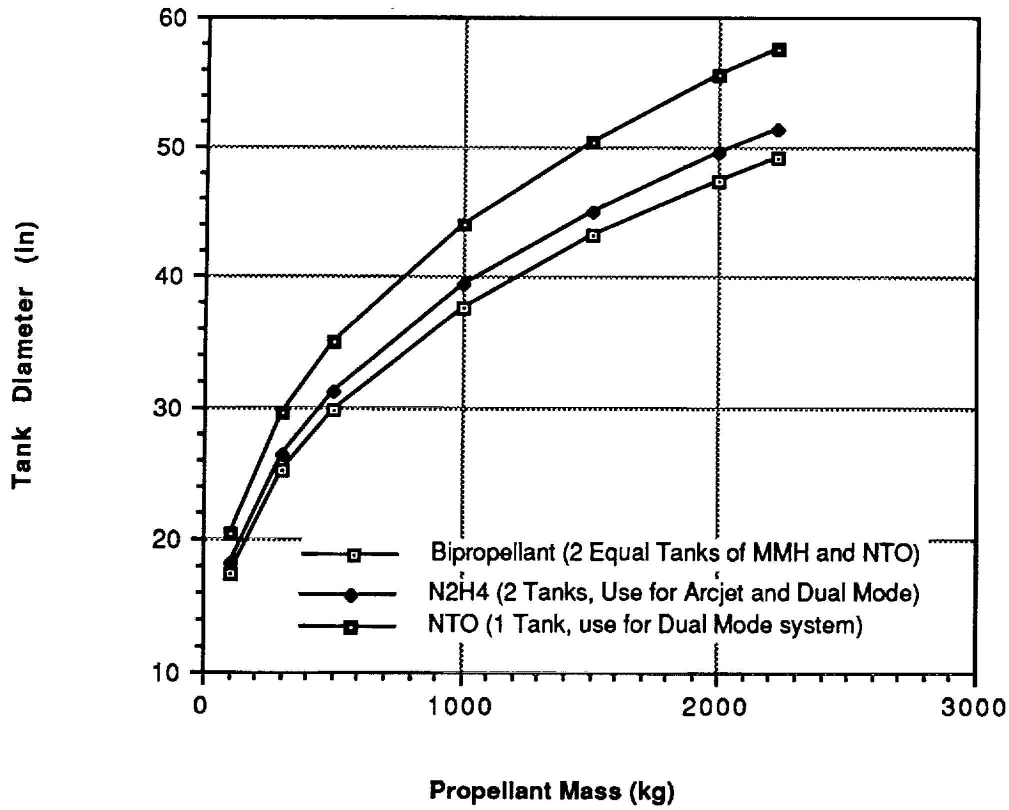

Figure 8-3: Propellant Tank Diameter versus Propellant Mass

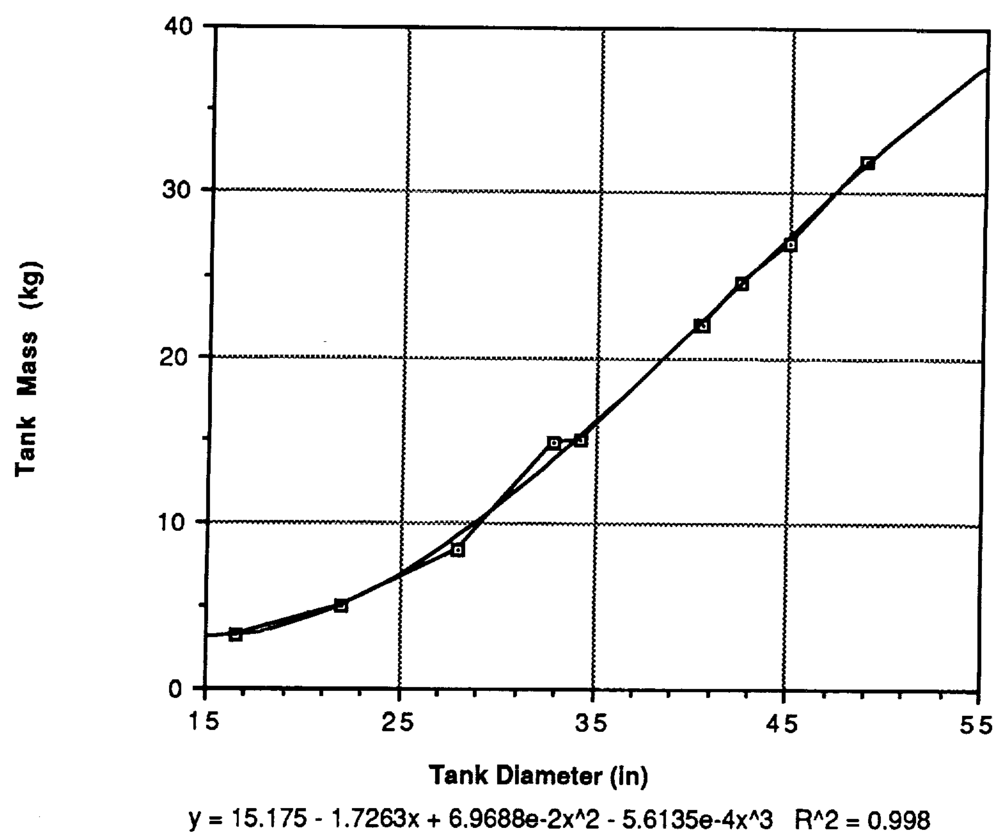

Figure 8-4: Spherical Propellant Tank Mass versus Diameter

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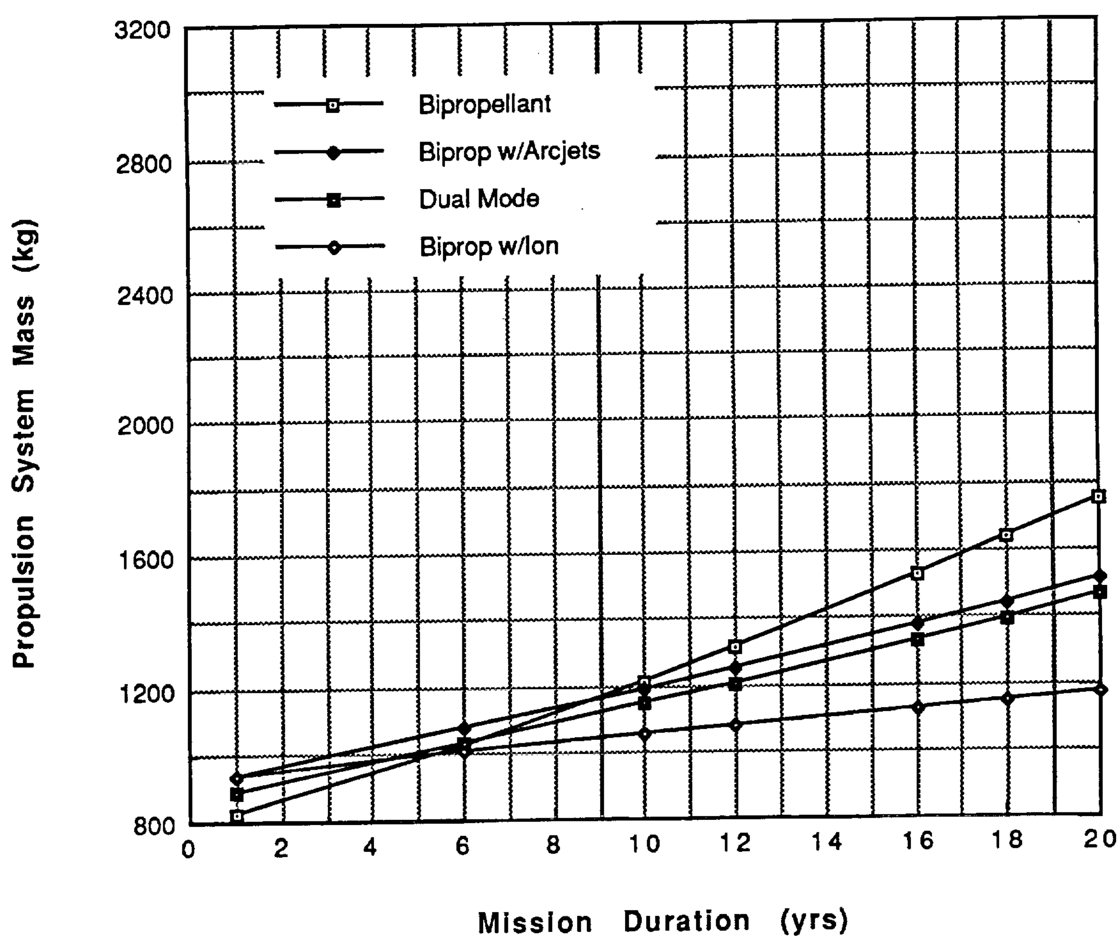
Appendix 1

Propulsion System Selection Charts

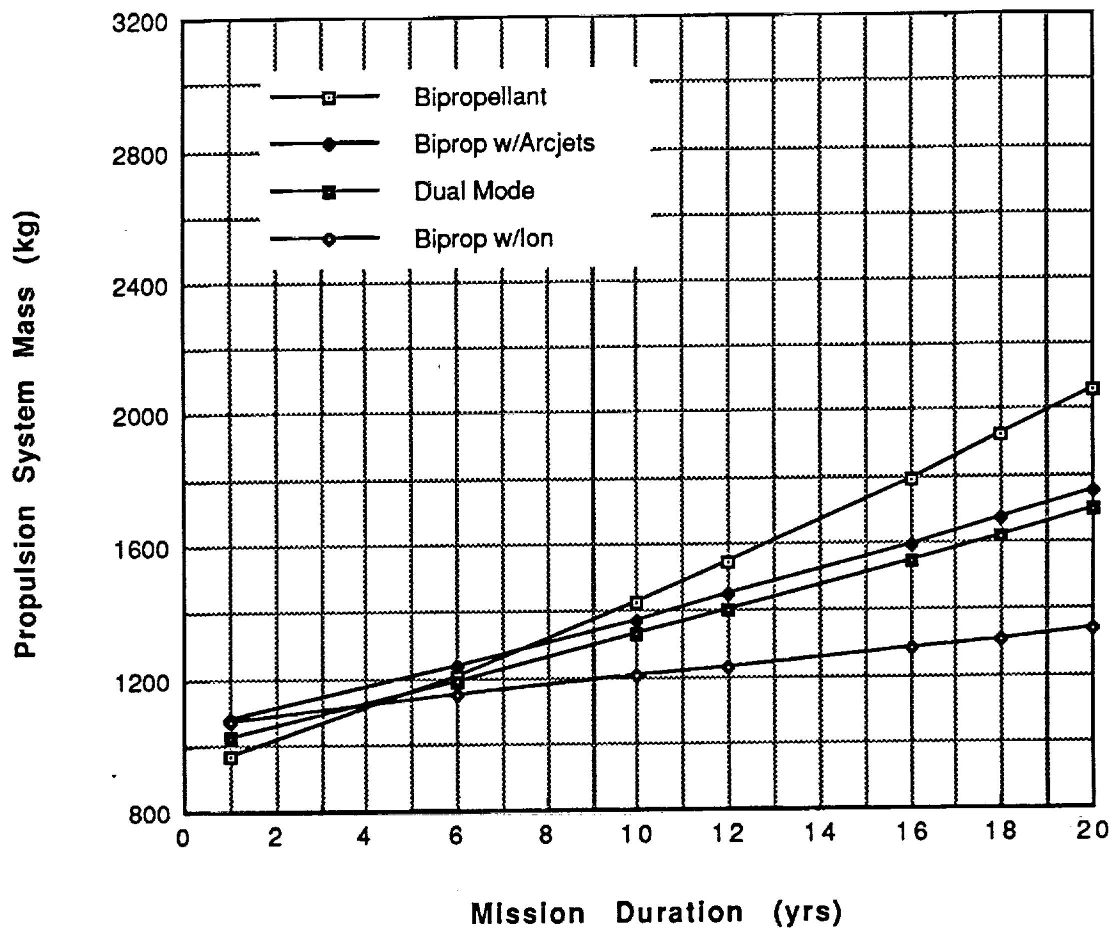
This appendix contains five figures (numbered 1A through 1E) that show the mass trends for each of the propulsion systems, as a function of satellite dry mass and lifetime. At low values of mass and lifetime, the electric propulsion system dry mass is not competitive. But as mass and lifetime increase, the electric propulsion systems are clearly superior. These charts should be used as a guide in selecting a specific propulsion system. To generate these curves, simplifying assumptions were made which compromise the absolute value of the charts. However, the relative trends between the four systems are valid.

These charts show the lowest propulsion system choices are the bipropellant (MMH/NTO) at low values of dry mass and lifetime, and the bipropellant system w/ion at higher dry mass and lifetime. Although this is a reasonable result from a mass point of view, other considerations are important in the system selection process. These issues are discussed in Chapter 8 and shown in **Table 8-1**.

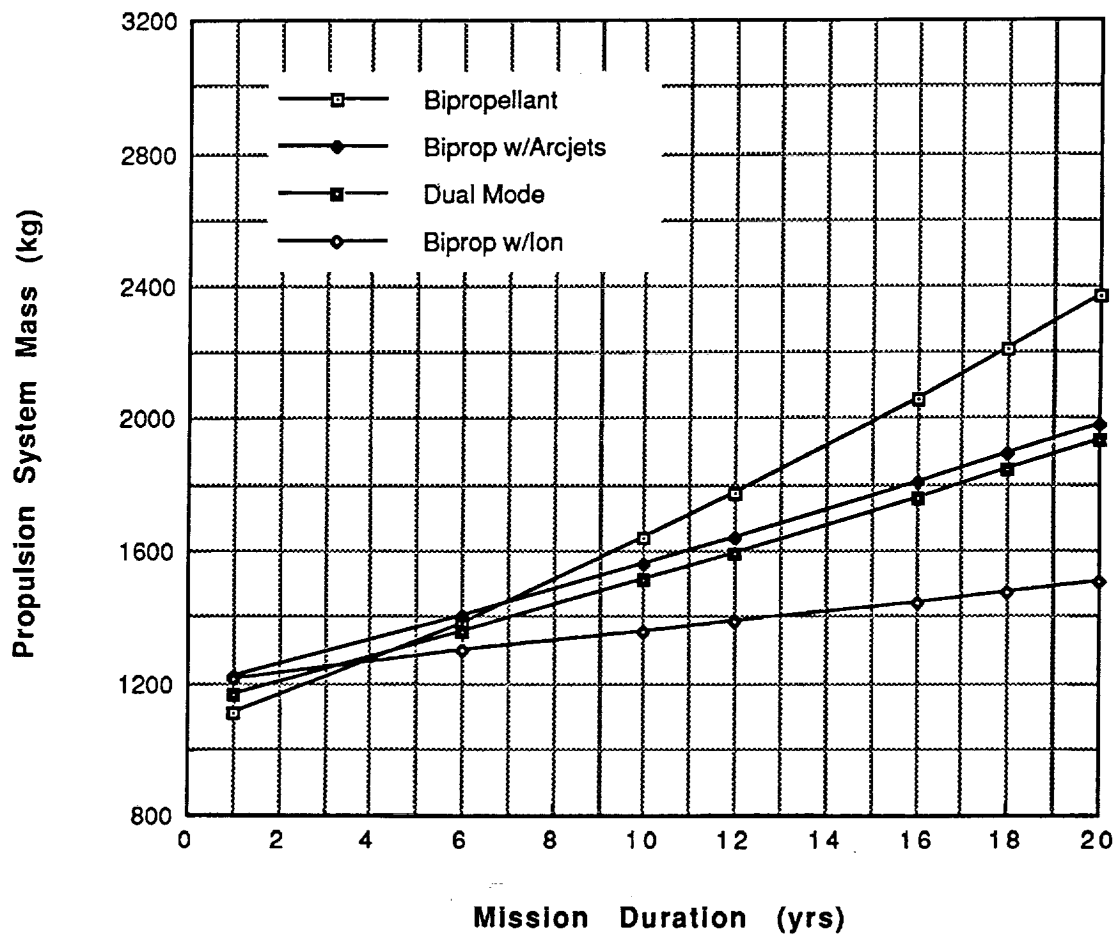
1A: Propulsion System Mass versus Life (1,000 kg dry mass satellite)



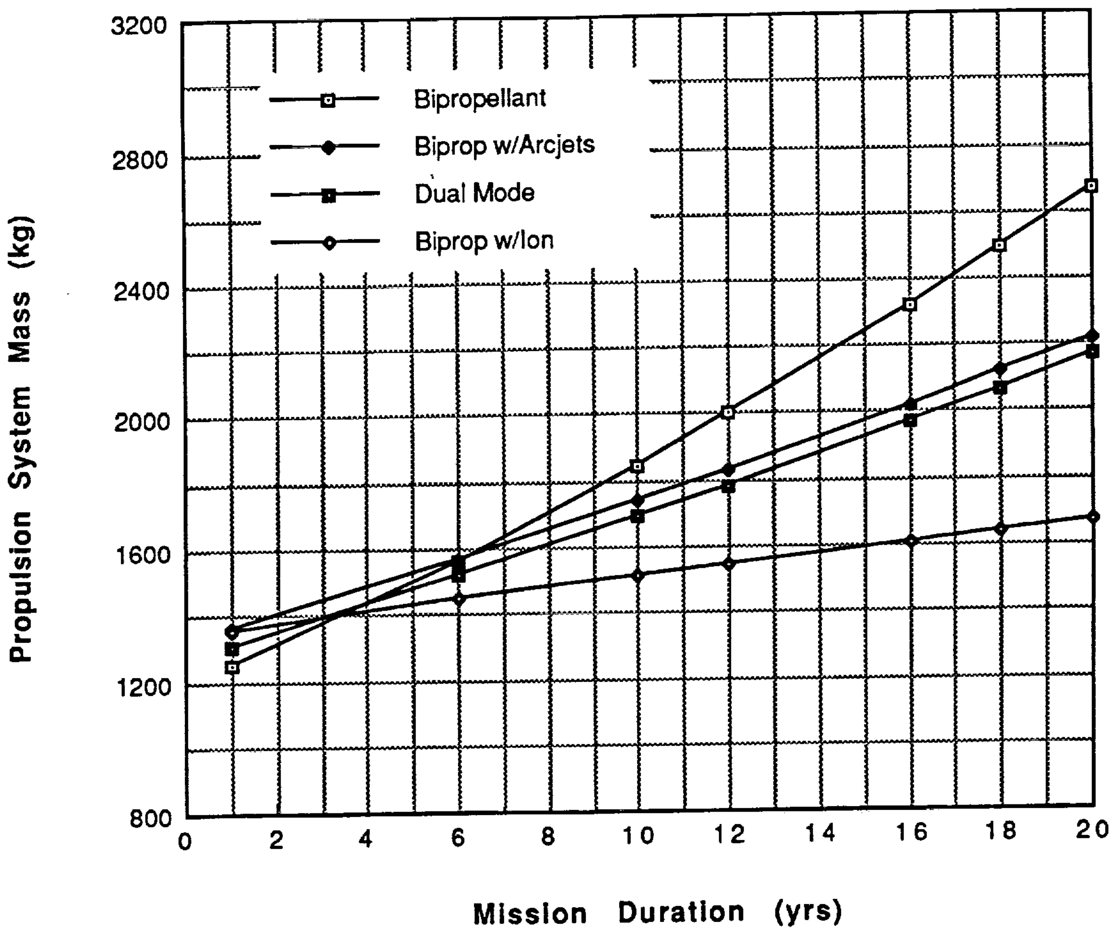
1B: Propulsion System Mass versus Life (1,200 kg dry mass satellite)



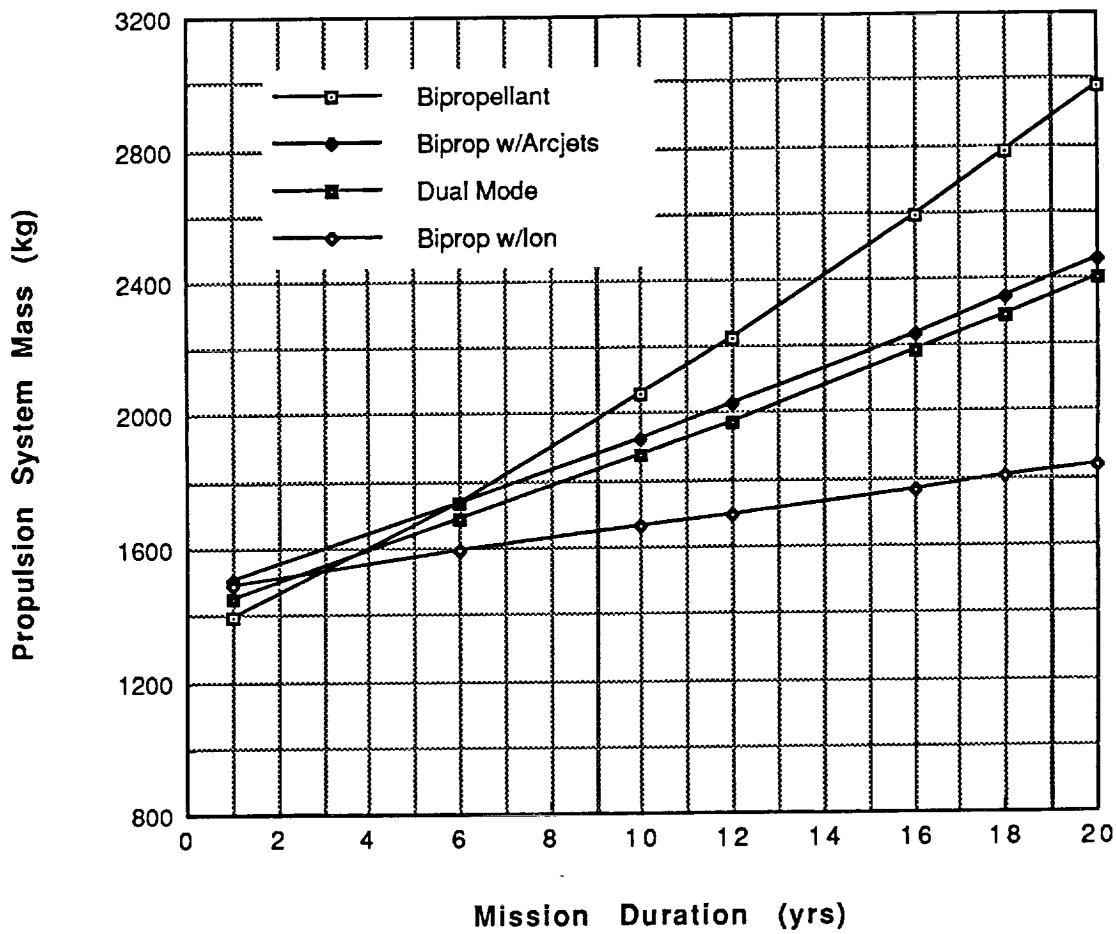
1C: Propulsion System Mass versus Life (1,400 kg dry mass satellite)



1D: Propulsion System Mass versus Life (1,600 kg dry mass satellite)



1E: Propulsion System Mass versus Life (1,800 kg dry mass satellite)



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Appendix 2

Example GOES-I Spacecraft

Mission Requirements

Spacecraft Dry Mass	980 kg
On-Orbit Propellant Lifetime	5.8 yr
Attitude Stabilization	3-axis
Launch Site	ETR

Propulsion System Requirements

Apogee Maneuver	1,800 m/s
NSSK	273 m/s
EWSK/AOC	29 m/s
EOL maneuver	10 m/s
Reposition maneuver (1 mnvr at 3°/day)	17 m/s

Baseline System Selection Selection

The Bipropellant (MMH/NTO) was selected for based on design heritage.

Propellant Mass Budget

Using Equation 8-1 and Table 8-3, the total propellant mass is 1,047 kg.

Propulsion System Dry Mass

Propellant tank diameter (Figure 8-3)	15 cm
Propellant tank mass, each (Figure 8-4)	19.4 kg
Total propulsion system dry mass (Table 8-4)	75 kg

Actual GOES Propulsion System Data

Propellant budget	1,098 kg
Propulsion system dry mass	88 kg

The error in the propellant mass calculated by the model is less than 5%. The error in the propulsion system dry mass calculated by the model is 14.7%. This error is a result of the actual size of the propellant tanks. The GOES propellant tanks are approx. 15 cm (38 in) in diameter, and have a total propellant capacity of 1,260 kg. The model sizes the propellant tanks by actual propellant calculated. If 15 cm tanks were sized, the error in dry mass from the GOES actuals would reduce to under 10%.

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13. ABSTRACT *(Maximum 200 words)*

Analytic estimating relationships for the mass and power requirements for major satellite subsystems are described. The model for each subsystem is keyed to the performance drivers and system requirements that influence their selection and use. Guidelines are also given for choosing among alternative technologies which accounts for other significant variables such as cost, risk, schedule, operations, heritage and life requirements. These models are intended for application to first order systems analyses, where resources do not warrent detailed development of a communication system scenario. Given this ground rule, the models are simplified to "smoothed" representations of reality. Therefore, the user is cautioned that cost, schedule, and risk may be significantly impacted where interpolations are sufficiently different from existing hardware as to warrant development of new devices.

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