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NASA Contractor Report 195029

# Small Spacecraft Power and Thermal Subsystems


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Contract NAS1-19244, Task 15

December 1994

National Aeronautics and
Space Administration
Langley Research Center
Hampton, Virginia 23681-0001



(NASA-CR-195029) SMALL SPACECRAFT POWER AND THERMAL SUBSYSTEMS (McDonnell-Douglas Aerospace) 125 p

N95-24428

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## ACRONYMS

| | |
|---|---|
| ACS | Attitude Control System |
| ADR | Adiabatic Demagnetization Refrigeration |
| Ah | ampere-hour |
| AHI | ampere-hour integrator |
| AMPTE/CCE | Active Magnetospheric Particle Tracer Explorer |
| BOL | Beginning Of Life |
| CAD | Computer-Aided Drawing |
| CCHP | Constant-Conductance Heat Pipes |
| COBE | COSMIC Background Explorer |
| CPL | Capillary Pumped Loop |
| CPV | Common Pressure Vessel |
| DET | Direct Energy Transfer |
| DOD | Depth of Discharge |
| DoD | Department of Defense |
| EPS | Electrical Power Subsystem |
| EMI | Electromagnetic Interference |
| EOS-AM | Earth Observing System - AM |
| EUVE | Extreme Ultraviolet Explorer |
| FOV | Field of View |
| FUSE | Far Ultraviolet Spectroscopic Explorer |
| GaAs | Gallium Arsenide |
| GMM | Geometric Math Model |
| GRO | Gamma Ray Observatory |
| GSFC | Goddard Space Flight Center |
| HST | Hubble Space Telescope |
| IR | Infrared |
| LaRC | Langley Research Center |
| LEO | Low Earth Orbit |
| MLI | Multi-layer Insulation |
| MMS | Multimission Modular Spacecraft |
| MOCVD | Metal Oxide Chemical Vapor Deposition |
| NASA | National Aeronautics and Space Administration |
| NiCd | Nickel-Cadmium |
| NiH2 | Nickel-Hydrogen |

| | |
|---|---|
| PDU | Power Distribution Unit |
| PPT | Peak Power Tracker |
| PWM | Pulse-Width Modulated |
| RFI | Radio Frequency Interface |
| SAMPEX | Solar Anomalous and Magnetospheric Particle Explorer |
| SAS | Small Astronomy Satellite |
| Si | Silicon |
| SMEX | Small Explorer |
| SMM | Solar Maximum Mission |
| SOC | State of Charge |
| SPRU | Standard Power Regulator Unit |
| SPV | Single Pressure Vessel |
| STEP | Space Test Program |
| TEC | Thermal Electric Control |
| TIMED | Thermosphere, Ionosphere, Mesosphere Energetics and Dynamics |
| TIROS | Television Infrared Observing Satellite |
| TMM | Thermal Math Model |
| TOMS-EP | Total Ozone Mapping Spectrometer - Earth Probe |
| TOPEX | (Ocean) Topography Experiment |
| TRMM | Tropical Rainfall Measuring Mission |
| UARS | Upper Atmosphere Research Satellite |
| UV | Ultraviolet |
| V/T | Voltage-limited Temperature-compensated |
| VCHP | Variable-Conductance Heat Pipes |
| XTE | X-ray Timing Explorer |

# 1. INTRODUCTION

This white paper was developed at the request of the Space Systems and Concepts Division, Spacecraft and Sensors Branch of the NASA Langley Research Center, under contract NAS 1-19244, task 15. The purpose is to provide a general guide to the conceptual design of satellite power and thermal control subsystems with special emphasis on the unique design aspects associated with small satellites (smallsats). For the purposes of this white paper, small satellites are defined to be those that can be launched on a Pegasus class of launch vehicle. The focus of this white paper is further restricted to low Earth orbits (1000 km and below) where the Pegasus is capable of providing adequate launch performance and where the majority of the current small satellites operate.

The scope of this paper was limited to the Pegasus vehicle first, because it is projected to be capable of launching a majority of smallsat spacecraft, second, because the paper would have become far too complex if the many other developing smallsat launch vehicles, such as Taurus, Conestoga, LLV1, LLV2, etc. were to be included, and third, because it is the smallest and likely the least costly launch vehicle suitable for most smallsat launches. In general, the conclusions presented herein will not change if other launch vehicles are used. However, the specific impacts of increased launch capabilities (higher altitude, longer mission life, etc.) are addressed.

The emphasis of this white paper is also placed on the NASA small satellites as opposed to DoD and commercial small satellites. Special topics have been incorporated into appropriate sections at the request of LaRC. For this reason, some topics such as cyro-coolers are discussed at greater length than other comparable items. Also, the impact on satellite cost and complexity is discussed, as this is perhaps the number one constraint in the current program development environment. Issues and conclusions having special relevance to smallsat design are highlighted in this report using shadowed boxes. These items have also been collected to form Appendix B.

At several places in this document, numeric or graphical examples are provided that refer to the TIMED (Thermosphere, Ionosphere, Mesosphere Energetics and Dynamics) satellite study. This GSFC study included an Earth-observing, nadir-oriented lightweight satellite in a 400-km circular orbit. Examples from this program are used because they provide a good, recent example of the technologies and design approaches appropriate to modern smallsats.

This white paper is organized into a general design discussion for both power and thermal control subsystems in Sections 2 and 3, including a review of current design requirements and future predicted design requirements, and our evaluation of current and predicted subsystem performance. Section 4 contains results and recommendations. A list of references is provided in Section 5.

# 2. POWER SUBSYSTEM DESIGN CONSIDERATIONS

## 2.1 OVERVIEW

A spacecraft's electrical power subsystem (EPS) requires two basic components: an energy source and an energy converter. The latter converts the source energy to electricity. A variety of additional components are also used to implement the system. The specific architecture depends on the application. Nevertheless, the nature and operation of power systems can be generally described using a limited set of subsystems which incorporate the key components.

The photovoltaic system is the only system discussed here because of extensive experience with it and its almost exclusive use in low Earth orbit (LEO) for moderate lifetime spacecraft. In photovoltaic power subsystems, the Sun is the energy source, and the energy converter is a solar array (an array of photovoltaic solar cells), which converts solar energy into electricity.

For full-time operation, an alternative energy source is required during eclipsed portions of the flight or when the solar cells are not facing the Sun. Typically, rechargeable (secondary) batteries are this energy source. They also provide power during circumstances when payload power consumption exceeds the capability of the solar array, such as happens when the array is oriented poorly or during deployment of the array, during emergencies, or with peak loads.

The energy used during an eclipse or during periods of peak demands in excess of array capability (plus losses) must be fully restored. Therefore, the solar array must be designed to achieve energy balance; i.e., have the capability to collect energy for powering the spacecraft during the sunlit portion and during the eclipse. For typical LEO missions, solar array output must be approximately 2.2 times greater than the total load requirement.

In designing an EPS, a sequence of tasks is generally followed (Figure 2–1). First the mission requirements and constraints, such as orbit parameters, load requirements, and operational constraints that drive the EPS architecture are defined. The EPS topology must be selected next, based on such factors as total load, orbit type, and reliability and cost constraints. This topology determines whether direct energy transfer or peak power tracking will be used, and the type of power distribution subsystem. Then, the solar array is designed and sized, and the battery is selected and sized. Before the design is completed, various nonflight operating phases are considered. The interaction with other subsystems is then considered. Finally, an analysis is done of the relative performance of available hardware, from which predictions of EPS cost, mass, and size can be made. The detailed design effort will then proceed using this analysis as a basis. Three excellent general source books on spacecraft power systems are References 1, 2, and 3.

## 2.2 MISSION REQUIREMENTS

### 2.2.1 General

The science, operations, or technology objectives of the mission define the orbit, lifetime, and duty cycle requirements. Selection of these basic mission requirements is determined primarily, but not entirely, by the payload requirements. The biggest impact on the EPS design is the magnitude of the power requirements. The power requirements and the orbit, lifetime and duty cycle requirements drive the EPS design. These requirements should be recognized as negotiable to reduce the cost, complexity, and risk at the subsystem level. Table 2–1 lists the hierarchy of design factors that should be followed in making such tradeoffs. This negotiation should be done as early in the project development as possible.

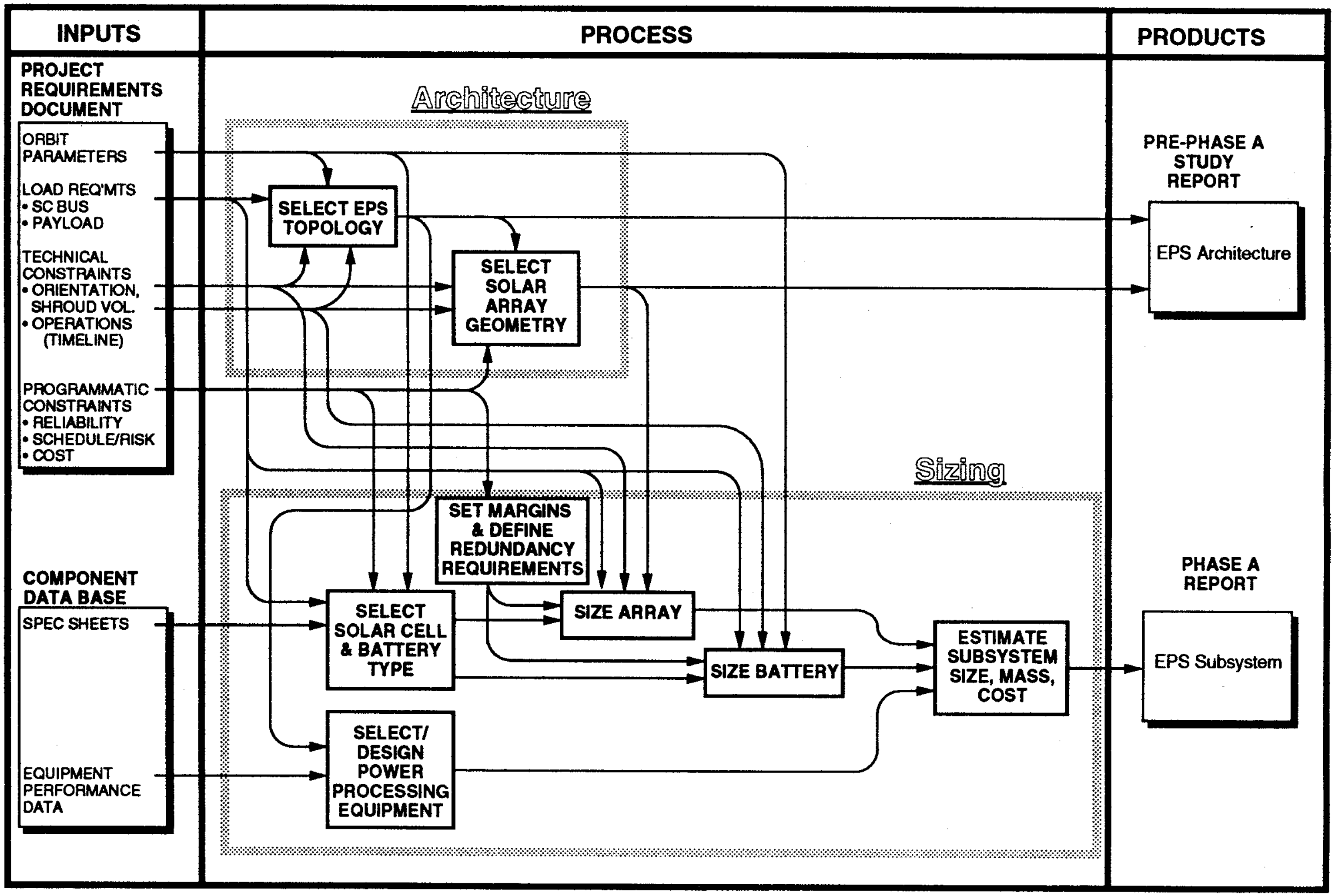


Figure 2-1. Electrical Power Subsystem Design Process

Table 2–1
Hierarchy of Design Factors

| Element | Primary Factors | Significant Parameter | Related Feature |
|---|---|---|---|
| *Mission Definition* | *Launch epoch* | *Year* | • Solar Flare Rad. (Max.--Min.) |
| | | *Day of Year* | • Solar Intensity VS Time |
| | *Orbit (alt., incl., ecc.)* | *Period* | • Thermal Cycling Rate |
| | | *Eclipse Fractions* | • Battery Temperature and DOD<br>• Temperature Extremes |
| | | *Ambient Orbital Environments* | • Atmospheric Drag<br>• Particulate Radiation<br>• Ultra-Violet Radiation<br>• Atomic Oxygen<br>• Micrometeoroids and Debris |
| | *Lifetime* | *Accumulated Effects* | • Array Thermal Cycles<br>• Radiation Fluence<br>• UV Exposure<br>• AO Exposure<br>• Battery Charge/Disch. Cycles |
| *Missions Requirements* | *Payload Requirements* | *Power* | • Amount and Duty Cycle<br>• Quality |
| | | *Field of View* | • Array Size, Location, Etc. |
| | *Spacecraft Bus Requirements* | *Essential Power* | • Amount<br>• Quality |
| | | *Non-Essential Power* | • Amount and Duty Cycle<br>• Quality |
| | | *Pulse Power (e.g., pyro)* | • Amount and Duty Cycle |
| | | *Subsystem Impacts* | • Thruster Exhaust<br>• Temperature/DOD |
| *Impacts* | *Power Subsystem Impacts* | *Array Design* | • Field- of-View Restrictions<br>• Reflections and Shadowing<br>• Temperature and Reradiation<br>• Outgassing<br>• Magnetic Fields |
| | | *Battery Design* | • Magnetic Fields |

Orbital factors pertinent to EPS design include: epoch, altitude, inclination, and eccentricity. These factors, in turn, determine period, eclipse duration, and most of the environmental factors. The environmental factors include: solar intensity (including infra-red, visible, and ultra-violet), particulate radiation (including trapped particles, solar flare, and cosmic ray components), micrometeoroids and debris, and atomic oxygen. The amount and intensity of these factors depend on the altitude. For example, the Earth's environment varies with the altitude (such as the amount of atomic oxygen that exists). In addition, the Earth's environment influences external factors that may impact the spacecraft (such as the Earth's magnetic field and its shielding against solar flares). Also the orbit parameters change as the altitude changes (such as eclipse ratios change, affecting time of exposure to UV). For certain regions, inclination must be considered. For example, spacecraft passing through the South Atlantic region are exposed to higher flux radiation. This environment is known as the South Atlantic Anomaly. Lifetime requirements determine the duration of exposure of the components to the environments to which they may be sensitive. Design interactions are secondary factors that are influenced by the environment or the spacecraft design or both. They include: temperature and thermal cycling, magnetic fields, thruster exhaust, and outgassing.

Environmental factors determine the degradation of the solar array. The array must be sized considering the predicted end-of-life degraded output. Period and eclipse duration relate directly

to battery cycling. The mission lifetime determines the extent of the degradation or the amount of the cycling.

Another factor that affects array sizing and configuration is atmospheric drag. For low-altitude orbits, large arrays can seriously reduce the lifetime of satellites with no on-board propulsion subsystems, or can require excessive fuel to maintain the orbit. Minimizing the frontal area by constraining solar arrays to edge-forward geometries may be necessary, but this adjustment usually requires increased array area. These opposing factors must be traded off in the early mission design phase.

### 2.2.2 Impacts of Orbit Parameters

The orbit parameters are the second most significant EPS design drivers as summarized in Table 2-2. These parameters, which include altitude, inclination, and eccentricity, and spacecraft attitude (relative to the Earth and the Sun) affect spacecraft power requirements and the design of the solar array and the battery. In addition, the variation of solar angle, Beta (the angle between the sun-line and the plane of the spacecraft orbit), has a major effect on the eclipse duration (battery discharge time), the sunlight duration (solar panel operation plus recharge time), and the solar angle of incidence on the array panels (solar array output, depending on design). The duty cycle or power profile (times when the power is used during the orbit) is also a significant driver.

Table 2-2
Impact of Orbital Parameters on Power Subsystem Design

| Parameter Orbital Factor | Solar Array Geometry | Solar Array Sizing | Battery Sizing | PSE Component Selection | External Material Selection |
|---|---|---|---|---|---|
| Eclipse Duration<br>• Altitude(s) (period)<br>• Inclination<br>• Eccentricity | | Yes, solar array power to recharge batteries. | Yes, eclipse period energy requirement. | | Thermal cycling of interconnects and bonded materials (usually not a concern). |
| Beta Angle History<br>• Epoch<br>• Nodal Precession<br>- Altitude<br>- Inclination | Yes, design to assure adequate power generation capability throughout mission. | Yes, design to designated worst case conditions. | | | |
| Radiation<br>• Epoch<br>• Altitude<br>• Inclination<br>• Eccentricity | | Yes, EOL power generation capability. | | Yes, radiation hardening to appropriate levels. | Solar array cover glass thickness and type of solar cell. |
| Solar Intensity<br>• Epoch | | Yes, energy input. | | | |

The duration of each eclipse and the number of eclipses over the life of the mission in large part determine the size of the batteries. Eclipse duration and load power profile determine the maximum depth of discharge (DOD) of the battery. Over many orbits, the eclipse duration is not constant, but careful analysis can determine just how often the worst case DOD will occur. This knowledge can be used to produce a histogram of predicted battery DODs, which provides a more realistic basis for battery sizing. This process is described in Paragraph 2.7.2.

## 2.3 LIFE-CYCLE PHASES (OPERATING MODES)

The EPS must also accommodate all the mission operating modes. These requirements can be a critical driver in how the subsystem is designed.

### 2.3.1 Ground Operations

The preflight testing, storage, and handling of spacecraft EPS components can be critical to mission success. The hardware must be properly checked out and prepared for flight, yet must be protected from adverse environments and rough handling (overtest, for example).

2.3.1.1 Batteries—The batteries are the most sensitive components of the EPS. They are electrochemical devices, have a limited life, and are susceptible to damage through mishandling. Flight batteries are almost never used for ground tests or operations. Instead a substitute set of test batteries is used. Although test batteries will not be flown, reasonable care must be exercised in handling them to preclude damage to the spacecraft under test from overheating, excessive internal pressure buildup, and explosion. A good guideline is to handle, operate, and monitor the test battery as if it were a flight battery.

In general, the test batteries will undergo environmental testing with the spacecraft as integrated components. Flight batteries do not undergo this environmental testing. However, both test and flight batteries are subjected separately to qualification and acceptance test sequences to verify that they will be suitable for the intended mission. A typical sequence consists of: physical measurements, physical inspection, functional tests, vibration, peak load, thermal vacuum, capacity tests, and a final physical inspection. Functional tests include two types: (A)–electrical insulation resistance; electrical continuity; connector bonding resistance; temperature sensor, thermostatic switch, and heater circuit operation; seal leakage test; capacity verification test; charge-retention test; and peak load test; and (B)–conditioning; seal leakage; charge-retention; electrical continuity; and connector bonding resistance.

The flight batteries, meanwhile, must be properly stored before launch. The type of storage depends on the type of battery. Batteries are generally stored in a discharged state at low temperatures. Certain batteries can be stored with a shorting resistor across the terminals, but others require an open circuit storage or even a light trickle charge. A short time before launch, the flight batteries replace the test batteries on the spacecraft. They are reconditioned and maintained on trickle charge until launch.

A recently issued handbook, Reference 4, provides detailed guidelines for handling and storage of conventional Nickel-Cadmium batteries. These guidelines can be used as a basis for handling and storing advanced NiCd batteries. However, additional guidelines should be obtained from the cell and battery manufacturers. Guidelines for handling and storage of $NiH_2$ batteries are provided in another recently published handbook, Reference 5.

2.3.1.2 Solar Array—The solar array presents a unique set of ground handling problems since it is difficult to test in a truly space-like environment. Additionally, the large number of exposed small, fragile parts and interconnections adds risk to dynamic testing of the flight hardware. Because of this risk, most mechanical tests of solar arrays are performed on an engineering test panel populated primarily with dummy cells (non-functioning aluminum pieces that represent the mass of the attached solar cells). Often small portions of the test article will have real cells and coverglasses installed in areas of high stress or deflection to monitor the design's stability under test conditions and to verify the assembly process. Use of actual flight hardware is generally deemed too expensive for most mechanical and environmental tests.

In general, the solar arrays are treated separately from the spacecraft for four reasons: (1) The arrays are external components and are subjected to significantly different environments. For example, the attachment of the array to the spacecraft results in significant differences in vibration and acceleration compared to the spacecraft with its hard-mount to the launch vehicle transition ring. Also, the space environment alternately exposes the array to direct sunlight and to cold space resulting in wide temperature swings. (2) The arrays are fragile and their components require special handling and storage. (3) The array's size and shape, especially after deployment, make it relatively unwieldy. (4) The array schedule frequently trails behind the schedule of the spacecraft. The delay results because the array design cannot be firmed until the spacecraft loads are well understood, and the array usually requires a long lead time to build.

Because of the array's make-up and its sensitivity to the various environments, it is usually stored and shipped, shock-mounted in a sealed container with a dry (desiccated) nitrogen atmosphere (at a positive pressure). It generally will not be integrated to the spacecraft, except for mating, fit, and mechanical interference checks, which sometimes include deployment tests. To avoid any possible damage, it is mated to the spacecraft as late in the launch schedule as possible, preferably just before the spacecraft is sealed in the nose cone.

Deployment and articulation tests of some sort must be performed to ensure proper actuation of all mechanisms without interference to other parts of the spacecraft. These tests must be designed to accommodate array sizes and space limitations and to negate effects of gravity.

Environmental testing usually consists of vibration testing of mechanical panels with mock-up cells; full thermal cycle qualification of a small "qual. panel" with live circuits; acoustic vibration tests to simulate the launch environment, and abbreviated thermal cycling of flight panels. This last test is done to identify weaknesses, if any, due to improper workmanship.

2.3.1.3 Power Processing Electronics—The power regulators, converters, and battery-charging electronics do not share the environmental and handling susceptibility of the batteries and solar arrays. Therefore, these components can be functionally tested on the bench, in concert with other spacecraft components and subsystems, and during integrated spacecraft environmental testing.

2.3.1.4 Environments—In designing the EPS, consideration must be given to possible excursions from nominal conditions during ground operations, including fabrication, testing, ground transportation, integration, and launch preparations. Conditions in these environments must be carefully controlled. Control factors include cleanliness, temperature, humidity, and vibration. In addition, special care must be taken when transporting items. Some items may need special packaging to avoid damage during transportation. Other items, such as solar arrays, cannot be transported as one piece and must be assembled at the launch site.

### 2.3.2 Launch and Ascent

The launch environment consists primarily of vibration, acceleration, and mechanical shock, and sometimes temperature and thermal shock. These factors are not controlled but must be adequately known to be accommodated by design and/or verified by test. Vibration and acceleration are launch-vehicle dependent. Components are usually tested to assure satisfactory operation during launch, if required, and after launch. The components, especially a honeycomb-type solar-array substrate, must be rapidly vented to allow for the depressurization that occurs with ascent into space. Heating of external surfaces, (especially the solar array because of its area and proximity) due to reradiation from the aerodynamically heated nose cone must be considered. Finally, protection of high-voltage circuits against corona must be afforded to prevent breakdown and insulation failure.

During this phase, the spacecraft almost always relies on external power via the launch vehicle or an umbilical. Therefore, the spacecraft EPS design must include this capability for power input. In addition, since the quality of this power (range of voltage and noise) is often not the same as provided by the on-orbit components, some input filtering may be required.

The launch vehicle power is removed at satellite separation (or sooner). This factor can drive the battery size. The spacecraft separation mechanism may also include a switch that activates the spacecraft processor and/or arms the EPS. Such a switch is often tied into the power distribution circuits, making it the responsibility of the EPS.

### 2.3.3 Pre-stabilization and Transfer Orbit

During this mission phase, the spacecraft must depend on the battery (sometimes entirely) until the solar arrays are deployed and oriented towards the Sun and the spacecraft achieves a power-positive state. This fact affects the sizing of the battery and control of the electrical load. The spacecraft must be able to operate on a minimal load budget so as not to overdischarge the battery. While primarily a function of the command subsystem, the EPS must be configured to shut down noncritical subsystem loads selectively.

Assuming deployable arrays are used, the spacecraft will be unable to radiate much heat while the array panels are still wrapped around the spacecraft body. The EPS components must be able to withstand possible high operating temperatures during this period.

For small spacecraft, launched on currently available launch vehicles, the initial tip-off (error angle from velocity vector due to unbalanced separation forces) rates pose special concerns for EPS design. The actual rates depend on many factors, such as spacecraft mass properties, target orbit, etc.; however, rates as high as 1.0 deg./sec./axis can be achieved. If the attitude control system (ACS) actuators cannot null this rate quickly, the battery can be depleted before achieving a power-positive state. Also, any deployed solar arrays must be designed to accommodate this rate through the deployment sequence.

### 2.3.4 Normal Operations

The overall mission requirements are based on normal spacecraft operations. The performance requirements and the orbit environment are the primary drivers for the EPS design. Degraded performance and operating conditions (orbit parameters, duty cycles, etc.) at the end of the mission must be determined and used to size the EPS.

2.3.4.1 Temperature—High temperatures result from incident solar energy and Earth albedo (reflection of incident solar energy). These temperatures are highest for solar-oriented systems and lower for non-oriented systems, such as spinning spacecraft. Low temperatures result during eclipse and are lowest at the end of eclipse combined with a minimum of albedo. Performance of EPS components depends on temperature.

2.3.4.2 Thermal Cycling—Thermal cycling results from the sunlight/eclipse cycling of the orbiting spacecraft and depends on the sunlight/eclipse ratio, which, in turn, depends on orbital mechanics (orbit altitude, orbit inclination, time of year, etc.) and on the spacecraft configuration and lifetime. Thermal cycling is extreme for light-weight (low thermal mass) solar arrays extended from the spacecraft. Fatigue and thermal expansion coefficient mismatches in the elements of the solar array result in small interconnect failures and a degradation of power.

2.3.4.3 Radiation—Space radiation (Van Allen Belts, solar flare emissions, cosmic rays) degrades the EPS component elements, especially semiconductors. The solar cells on the solar array are particularly susceptible to degradation because of their exposed location. A harsher

radiation environment requires more shielding (thicker cover glasses) and therefore more mass. The radiation environment may dictate the type of solar cell. For example, Gallium Arsenide (GaAs) cells are significantly more radiation resistant than Silicon (Si) cells. For a high radiation orbit, the use of Si cells imposes a significant mass penalty, up to 40 percent of array mass on a small satellite.

2.3.4.4 Plasma—Buildup of charge on the array or spacecraft due to space plasma can create power leakage paths on the array. The solution is to apply special coatings on exposed conducting surfaces. However, this solution is labor intensive, and adds substantially to the costs in addition to increasing the mass.

2.3.4.5 Ultraviolet—Solar ultraviolet radiation can degrade optical materials, and especially surface coatings. Transmission characteristics of the glass shields used for protection of the solar cells against radiation can degrade, causing a loss in array power.

2.3.4.6 Micrometeorites—Damage from micrometeorites is frequently discounted because designs generally include "more than enough" shielding for other reasons. However, the probability that significant impacts can occur and can cause mechanical damage to sensitive elements is finite though extremely small.

## 2.4 INTERACTIONS

In the design of a spacecraft EPS, the foremost considerations are the quantity of power to be produced and the management of that power. The EPS must produce sufficient power of proper quality to operate the spacecraft throughout its mission lifetime. In addition, each component of the EPS must work together without adversely affecting the other EPS components, the payload, or other spacecraft subsystems.

Frequently, the interactions among components, the payload, and other spacecraft subsystems depend on the size of the EPS. The impacts increase as the size of the subsystem increases. Although common design practice accommodates the interactions in general, each must be considered in the design of the EPS. The primary interactions are summarized below and illustrated in Table 2–3.

### 2.4.1 Payload Interactions

The power level and quality of power required from the EPS are largely determined by the requirements of the payload, whereas the impact of the EPS on the payload and instruments is primarily a result of its distortion of the immediate environment.

2.4.1.1 Impact on EPS of Payload Requirements—Battery and solar array design are affected by the payload mission (orbit and lifetime), which determines eclipse time. Battery design depends on load, eclipse time, and allowable DOD compatible with mission life. Array design depends on load, sun-light/eclipse ratio (i.e., battery recharge requirements), thermal cycling, and radiation degradation parameters that are orbit dependent.

Instrument sensitivity influences the design of the power processing electronics. This sensitivity sets the allowable ripple/noise level in the EPS, limits bus impedance (to prevent cross-talk between instruments), and may dictate switching frequencies within the EPS.

The size and location of instrument apertures, dedicated radiators, baffles, etc., limit the surface area that can be used for solar cells in a body-mounted solar array. For deployed arrays, the size, shape, and orientation of the array must be such that they do not obstruct the field of view (FOV) of the instruments for all possible array orientations. The reverse is also true. The array

geometry can affect the locations and orientations of the instruments. Since some leeway in selecting instrument locations and orientations usually exists, judicious tradeoffs can be made to accommodate the instrument FOV and the array geometry requirements. The limited "real estate" on small satellites can make these accommodations more difficult.

**The limited size of small satellites makes it more difficult to avoid interference between payload fields-of-view and solar array and other appendage geometry.**

2.4.1.2 Impact of EPS on Payload and Instruments—The array configuration and location affect the look angles available for instruments and experiments. Charged particle (plasma) buildup on the array and subsequent rapid discharge can affect instrument measurements and sensitive instrument electronic devices or components. Permanent fields created by magnetic materials and stray fields created by uncompensated current loops (e.g., in the array or battery) can interfere with instrument measurements. Magnetometer measurements and energetic charged-particle and plasma measurements can be affected.

2.4.2 Spacecraft Subsystem Interactions

The power must be conditioned to a quality adequate for operation of the particular subsystem. Simultaneously, radiated electromagnetic interference EMI or RFI must be limited so as not to affect subsystem operations. In addition to these primary interactions between the EPS and the other subsystems, many other interactions occur, which are summarized in Table 2–3.

2.4.2.1 Impact on EPS of Other Subsystems—The heat rejection and absolute temperature control capabilities of the thermal control subsystem have a major effect on the EPS. Additionally, a heat path must be created to transport the heat from the source to a heat rejection surface (i.e., radiator). Solar array efficiency, and power output depend on the operating temperature. Solar array integrity depends on thermal cycling and temperature extremes. Battery life and acceptable DOD depend on temperature. Predictable and reliable operation of converters and regulators depends on maintaining equipment temperature within specified limits for internal devices (transistors, diodes, transformers). Shunt regulators, in particular, require thermal dissipation of excess power for proper operation.

2.4.2.2 Impact of EPS on Other Subsystems—Other subsystems are primarily affected by the heat load the EPS creates and the large array area. These effects generate additional requirements for example, excess power generated by the EPS must be thermally dissipated. In addition, the thermal subsystem must maintain the battery and power electronics at proper operating temperatures.

Other subsystems are also affected by the solar array. Aerodynamic and solar pressure torques on the solar array affect the control load on the attitude control and determination subsystem. Mechanical frequencies of the solar array react with the spacecraft, coupling into the ACS. Arrays that continually track the Sun (orbit rate tracking) impose a step-wise continuous disturbance onto the spacecraft, which may be unacceptable if the satellite is small.

The solar array can obstruct and interfere with Sun- and Earth-sensing instruments. This interference can occur because the FOV is obstructed, or because the array surfaces (solar cell coverglass shields) reflect some sunlight.

Table 2–3
Power Subsystem Interactions

| Other Subsystems | Effect of Power Subsystem on Other Subsystem | Effect of Other Subsystems on Power Subsystems |
|---|---|---|
| Structure | • Heat transport path<br>• Support for solar arrays & other equipment | • Heat transport path<br>• Probable limited support for solar arrays |
| Mechanisms | • May require deployment devices | • Gimbals, slip rings, and/or flex cables may be needed |
| ACDS | • Disturbance torques from arrays<br>• Vibration from arrays requires control<br>• FOV blockage of Sun & Earth sensors | • Pointing errors for solar array<br>• Sun & Earth sensors FOV impact on array placement and motion |
| Propulsion | • Drag make-up | • Possible contamination of array surface |
| Thermal | • Efficiencies affect heat load<br>• Waste heat from shunt resistors and transistor switches<br>• Waste heat from batteries and electronics<br>• Battery temperature control | • Heat rejection capability controls minimum battery temperature, and hence DOD and life<br>• Temperature control capability sets array and electronics operating point and efficiencies. |
| C&DH | • Degree of EPS autonomy determines level of C&DH support required<br>• Battery charge control may be completely dependent on C&DH<br>• Support for safe modes | • Interface limitations for charge control modes<br>• Support for safe modes |
| Payload | • Solar array may restrict look angle and/or FOV<br>• Solar array charge build-up may induce EMI<br>• Stray magnetic fields from battery may induce EMI | • Sets the orbit and hence eclipse duration and radiation environment<br>• Life requirements (affects array sizing and battery DOD)<br>• Load and duty cycle requirement<br>• If deployed, may limit array deployment or create shadows<br>• Bus quality requirements influence charge control architecture and electronics design<br>• Switching and isolation requirements determine power distribution and bussing design<br>• Requirements for failure and safety modes and redundancy) influence architecture and power electronics design |

### 2.4.3 Electromagnetic Interference/Radio Frequency Interference

Radiated EMI or RFI from EPS oscillations or switching transients can interfere with sensitive instrument operations or measurements and with data handling. Shielding of all susceptible wiring is mandatory.

2.4.3.1 Regulators—The operation of regulators and/or inverters can produce noise and ripple on the power bus. This interference can be transient when sequential shunts are turned on or can be relatively continuous with pulse-width modulated (PWM) regulation or inverter oscillations.

2.4.3.2 Power Returns and Grounding—In implementing power return and grounding schemes, interactions among instrument busses and returns must be minimized and detrimental ground loops must be avoided.

2.4.3.3 Outages—Power subsystem outages, such as occur during switching, or overload or anomalous operation, affect payload and instrument equipment and interrupt or distort the measurements.

## 2.5 ARCHITECTURE

### 2.5.1 Power Processing (Bus Regulation) Options

The bus voltage for the photovoltaic system is bounded on the high end by the solar array output voltage during sunlight and on the low side by the battery discharge voltage during eclipse. Both voltages are variable and depend on the characteristics of the array and battery and on the load.

At times, this voltage must be regulated, such as when an unregulated bus voltage exceeds the variability limits acceptable for the equipment. Array voltage is controlled by using a shunt regulator (Figure 2–2) or by inserting a series regulator between the solar array and the battery (Figure 2–3). The battery voltage is controlled by regulating its charge and discharge. If required, special series regulators can be used for individual pieces of equipment.

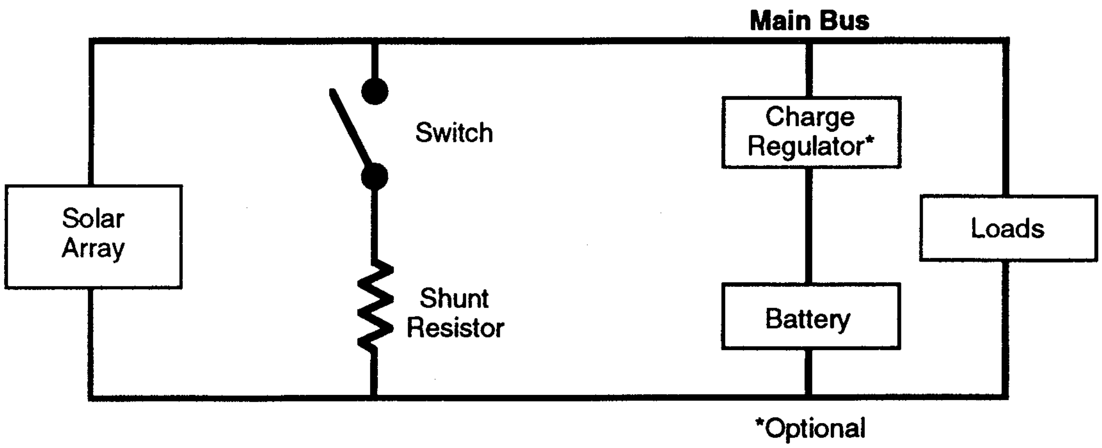


Figure 2–2. Basic Shunt Regulator

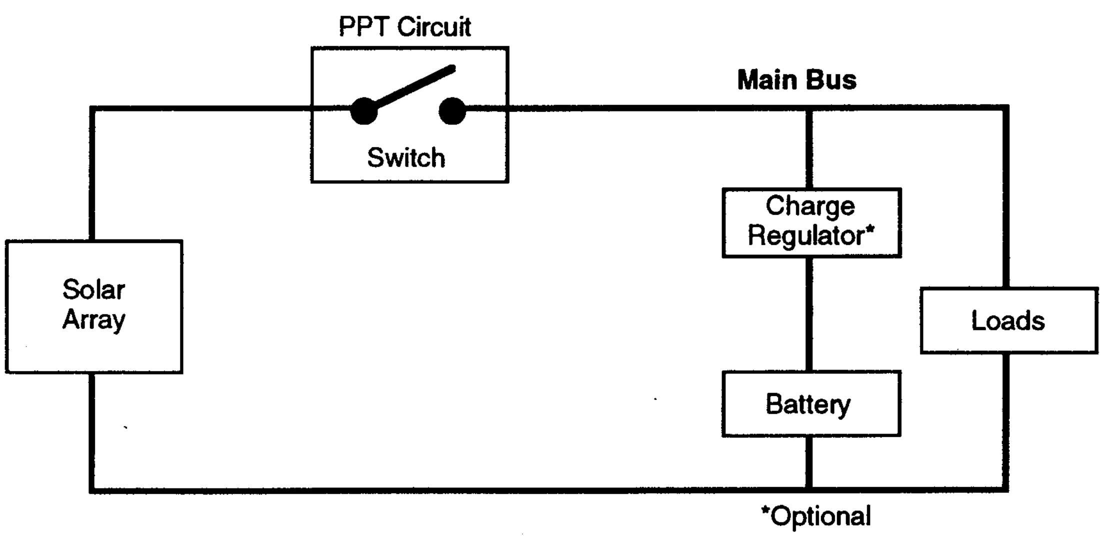


Figure 2-3. Basic Series Regulator

A shunt regulated system is also known as a direct energy transfer (DET) system since the solar array is tied directly to the load bus. It is a relatively simple system. The excess energy is diverted and dissipated. During discharge periods, the battery regulates the bus voltage (unless other charge/discharge regulators are used).

Inserting a regulator in a series between the solar array and the battery is generally implemented as a peak power tracker (PPT). This system adjusts the array current and voltage to draw only the required power, up to the array maximum power, from the array. Any excess energy is not dissipated, but instead is left on the array.

2.5.1.1 Choosing a Power Processing Option—A series regulator using PPT adjusts the array current and voltage to vary the solar array operating point. This enables maximum power to be drawn when operating at the peak power point. When load demand is reduced, the array is operated away from the peak power point. Since this approach allows only the required power to be transferred from the solar array, there is no excess energy to be dissipated, as in a DET design. This results in a more efficient use of the solar array, which will also now generally operate at a higher voltage and a lower current. These features generally permit spacecraft with PPT architectures to have a smaller solar array (Reference 6).

A drawback of PPT designs is that the PPT is usually implemented by a PWM switching regulator which can be noisy and therefore requires filtering. On the other hand, many shunt circuits also use PWM switching drives, and the related filters required in DET designs add weight. In general, the PPT array regulators are advantageous for spacecraft in LEO with loads less than 1000 watts. For larger loads, the balance favors DET designs (Reference 7). Until recently, shunt designs using DET have been simpler and less costly to implement. Lately, PPT control circuits that use microprocessors have become more common. Such designs are planned for use on smallsats, such as the STEP satellites for the Air Force, NASA GSFC's TOMS-EP, Orbital Sciences' SeaStar, and GSFC's TRMM and FUSE spacecraft. A simplified flow diagram of the process used to select an EPS architecture is given in Figure 2-4.

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|---|
| The development of microprocessor-controlled peak power tracking charge control circuits has allowed this approach to be competitive with otherwise simpler direct energy transfer power processing techniques. |

2.5.1.2 DET Design Options—DET regulators are shunt regulators (simple resistors or power transistors) which hold the maximum bus voltage below some predetermined fixed value regardless of the changes in load current and/or solar array output (Reference 8). Shunt regulators may be full (across the whole array) or partial (across part of the array). A full linear shunt design is shown in Figure 2–5. In a partial linear shunt system, the array is broken into segments, and each is fed through a dedicated regulator.

Most DET regulators have multiple, parallel-shunt stages operated sequentially. Most stages are either full on or full off, except the final stage which runs in either an analog/linear mode or in a switching mode with a PWM driver. The latter approach reduces waste heat generated by transistors operating in the linear region. Note that PWM switching regulators can be used in both shunt or series architectures.

Many options and variations exist between the extremes of a simple, single-stage linear shunt and multiphase/multistage PWM regulator. Issues involved in choosing an approach include how the array is split electrically and the effect on switch (power transistor) location and harnessing. These options are summarized in Table 2–4.

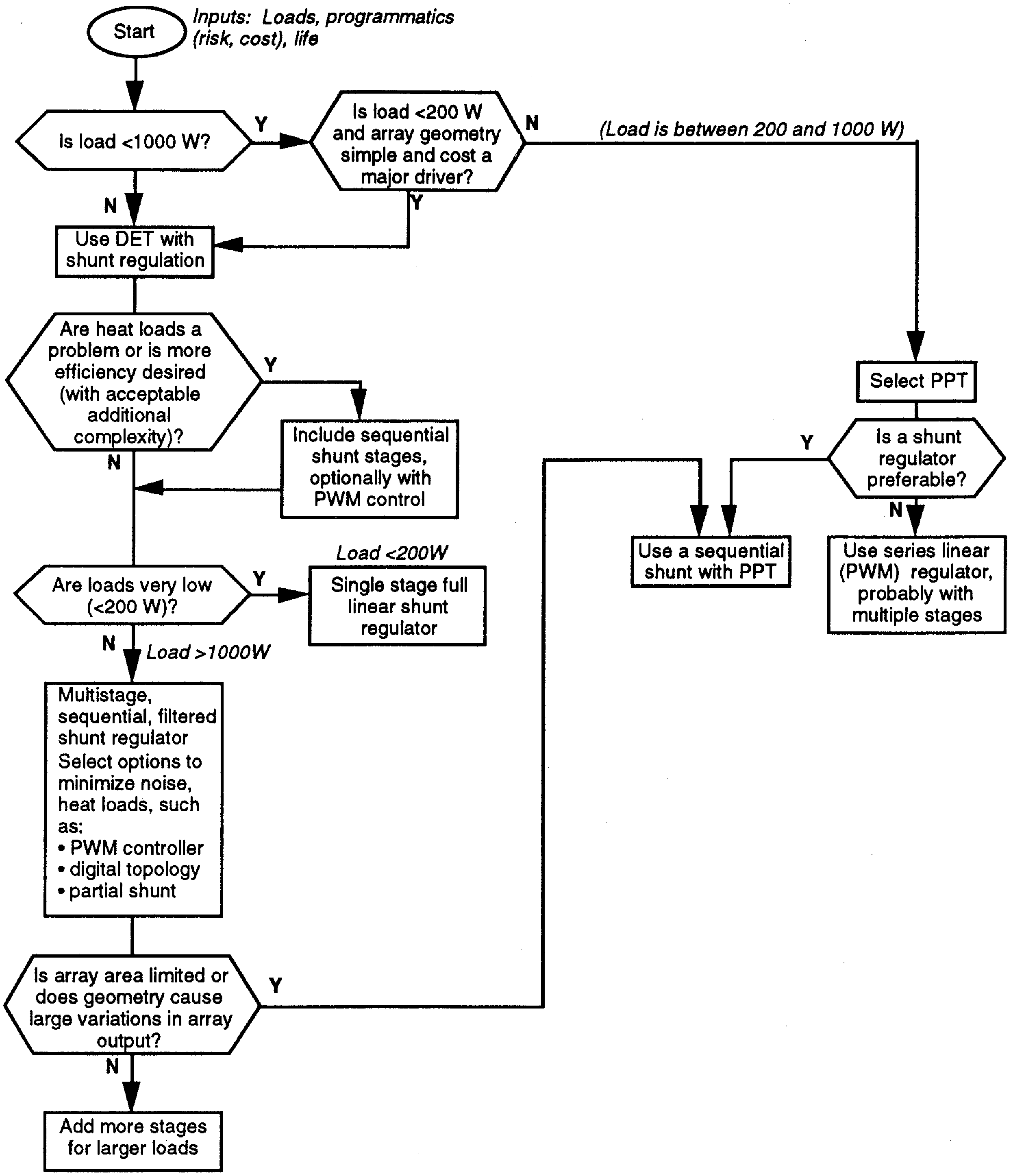


Figure 2–4. EPS Architecture Selection Flowchart

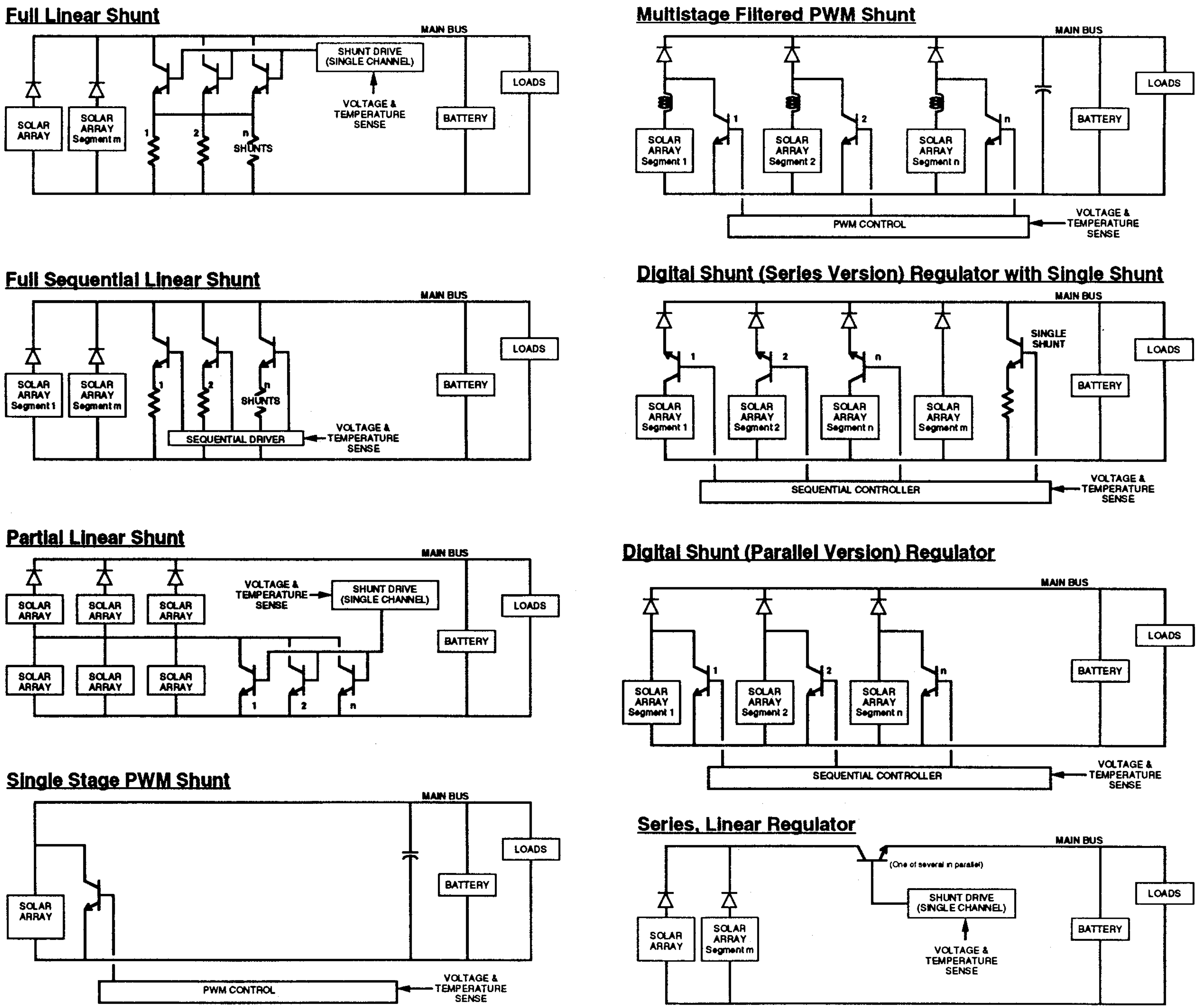


Figure 2-5. Solar Array Regulator Architecture Options

Table 2–4
Comparison of Shunt Regulator Designs

| Characteristic | Full Linear | Full Linear Sequential | Partial Linear | PWM Single Stage | PWM Multi-Stage | PWM Multi-Phase | Digital |
|---|---|---|---|---|---|---|---|
| Program | IMP/RAE | SAMPEX | TIROS | | XTE | | HST |
| Control Bandwidth | Large | Large | Good | | Medium | | Large |
| Transient response | Very fast | Very fast | Good | Good | Good | | Very fast |
| Ripple | Low | Low, with filter | | High | High | | Low |
| EMI problem | Low | Low | Low | Large | Large | Reduced | Low |
| Efficiency | Poor | Good | Moderate | Moderate | Good | Medium | High |
| Volume | Average | Moderate | | | Moderate | | Low |
| Mass | Low | Moderate | | | Moderate | | Low |
| High | High | Highest | Moderate | Mod-high | Low-mod | | Low |
| Complexity | Simple | Simple | Medium | | Medium | | Simple |

*The descriptive terms "High" and Low" are relative for that item across that row.*

2.5.1.2.1 Full Linear Shunt—This design was used on many early satellites; in it, all of the array segments are bussed together. A single shunt stage or multiple, parallel stages may be used, each consisting of a large power transistor operating in linear mode. A resistor or resistor set is often included in series with the power transistors to share some of the load. The design is simple with a large control bandwidth (good response).

It provides the most continuous control and the lowest noise of all of the shunt options. Unfortunately, it is not very efficient and has a heat sinking problem. This is because it usually has only a single stage operating in linear mode which is not very efficient, and therefore generates a lot of waste heat. This limits its applications to spacecraft with low total load power. While not used on recent designs, its simple implementation might have value on smallsats.

> A simple full linear shunt regulator, while somewhat inefficient, is a viable power processing candidate for small satellites with very low load power requirements.

2.5.1.2.2 Sequential Linear Shunt—In this shunt design, all of the array segments are bussed together as in the full linear shunt, but several parallel shunt stages are used instead of a single power device. The elements operate in sequence, and all but the final stage are either full on or full off. Little waste heat is dissipated in the spacecraft itself. Instead, it is transferred to shunt resistors located on the array panels. Figure 2–6 shows the increase in heat as each additional stage goes through a peak heat dissipation in its linear mode. The final stage is a large power transistor operating in linear mode. This approach has been used on recent spacecraft such as COBE and SAMPEX, and is baselined for others in the Small Explorer (SMEX) line.

> Several recent small NASA GSFC spacecraft use a sequential, staged linear shunt regulator for power processing.

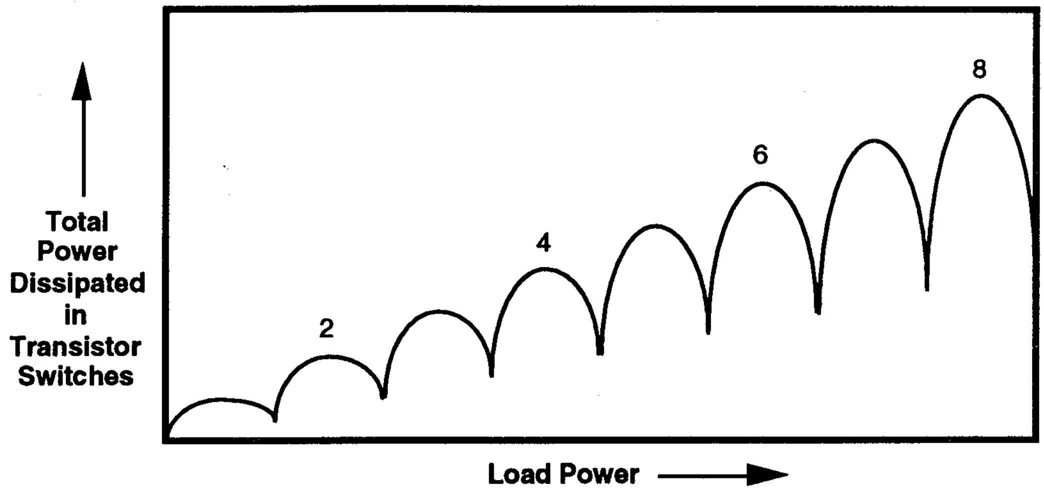


Figure 2-6. Full Linear Shunt Sequential Transistor Power Dissipation

2.5.1.2.3 Partial Linear Shunt—Instead of shunting the full solar array (the entire series string of cells), a partial shunt taps into the array at an intermediate point in the string. The portions of the array will be mismatched, so the operating point drops down to match the load requirements at the bus voltage. At the end of life of a satellite with a degraded array, this type of shunt matches of capability to requirements. However, it results in more waste heat (shunt loads) early in the mission. Because the array strings are split, locating the switches on the array panels is usually easier from a cabling point of view. This arrangement requires the shunt transistors to withstand frequent large temperature cycles.

The shunt circuit for this approach, although used on less than the full array string, is implemented either as a linear shunt or in a PWM mode. Also note that, because much of the unused energy is left out on the array and not dumped into loads, this is an example of a non-dissipative DET system. It has been used for many years on the NIMBUS and TIROS series of meteorological satellites.

2.5.1.2.4 Pulse-Width Modulated (PWM) Single Stage—This design is similar to the Full Linear Shunt except that a PWM circuit is used as the regulator instead of using a transistor operating in linear mode. All of the array segments are bussed together, and only a single switching stage is used. As the power transistor rapidly switches between full on or full off states, less power is dissipated, and the regulator runs relatively cooler. When the transistor is in the on state, the array is shorted. Therefore, power is not transferred to the load bus, but the bus voltage is maintained by a filter capacitance. The disadvantage of using this design is that a large filter is required to maintain the regulated bus voltage.

2.5.1.2.5 PWM Multistage—In this design, the solar array is divided into several sections. Each section is shunted by a transistor switch driven by a central PWM control circuit. This architecture provides for a very high power handling capability. However, all of the switches operate synchronously and generate large current pulses and significant EMI. Therefore, filtering is required. The option to have a PPT incorporated into this approach is among the factors that led to its use on XTE.

2.5.1.2.6 PWM Multiphase—This approach attempts to alleviate the EMI problems of the PWM Multistage Regulator. All stages duty cycle simultaneously but the controlling PWM signal is phase shifted for each stage, and the noise is reduced.

2.5.1.2.7 PWM Multistage and Multiphased—The features of both types of PWM shunt circuits can be combined to reduce noise since the pulses are out of phase. Despite this, the combined PWM design is not generally used in a primary array power processor. This design is more often found in voltage regulators, such as the boost regulators (battery discharge regulators) on TIROS and EOS-AM.

2.5.1.2.8 Digital Shunt—This regulator is also referred to as an array shedding regulator and is used on the Hubble Space Telescope. Versions of this architecture have also been used on GEOSAT, AMPTE/CCE, and the Small Astronomy Satellite (SAS). This approach divides the solar array into parallel sections which are connected to the load bus only when required. The array segments are switched full on or full off as required by the total load demand.

> A digital shunt regulator improves its efficiency by disconnecting solar array sections that are not required to satisfy the load requirement.

Two versions have been used. In one version, each array segment can be open circuited, except for the last segment which has a single linear shunt regulator permanently tied across it. In the other version, the array segments are selectively shorted out by a transistor switch, and a single shunt circuit is used for fine adjustments. This final shunt stage is sized to handle the power of just one solar array section, thus the bus voltage is maintained at a fixed level from no load to full load. This approach is simple and fairly efficient. It has good transient response. As array sections are brought on line, however, there can be large current and voltage spikes, resulting in noise and discrete jumps. This effect can be minimized by carefully designing the array segment switches and by using filtering. This approach has been shown to work well for larger power systems. Also, the bus voltage regulators usually used in higher power spacecraft filter out such noise.

2.5.1.3 Series Regulators—A non-dissipative series regulator uses a buck (switching) converter with PPT to control the bus power by controlling the output power from the array. The array voltage is controlled (reduced) by adjusting the desired array operating point via the PWM switching converter. Series regulation thus permits the array to be operated at any power level up to maximum power output, making optimum use of the solar array power-producing capability as required by the load demand. A charge/discharge controller is used to control the battery functions. In this system, the control unit senses the voltage and current of the array, the battery, and the bus and the battery temperature. It controls the series regulator and the battery charge/discharge controller to maintain the desired logical operation of the system. However, shunt regulator designs can also be used with PPT.

2.5.1.3.1 Series Regulator—The Standard Power Regulator Unit (SPRU or sometimes just PRU) is a PPT series regulator that is tied to the total solar array output. The entire solar array is bussed together and the SPRU processes the power for delivery to the load bus and battery(s). The SPRU actually consists of six parallel power stages, each featuring a large PWM-operated switching transistor.

The SPRU is a component of the Multimission Modular Spacecraft (MMS) which has been used on the Solar Max Mission, Landsats 4 and 5, GRO, UARS, TOPEX, and Explorer Platform (EUVE).

2.5.1.3.2 Parallel Series Regulator—This series regulator design splits the solar array into parallel segments, with each segment having a filtered PPT/PWM regulator in series with the main bus. The PWM control is centralized, similar to the PWM multistage shunt design of Paragraph 2.5.1.2.5. Here, however, the regulators are a true buck converter design with their outputs tied together. This design is used on TOMS.

2.5.1.3.3 Battery Charge and Discharge Regulation and Control—In most DET and PPT designs, the control circuits that drive the regulators also provide the battery charge and discharge control functions. The battery charge circuitry must feed into the shunt regulator or series regulator. An option is to have additional regulators placed in series with the battery charge and/or discharge paths. Some shunt systems have very coarse current control and do not provide fine enough control for proper battery charging, so a series regulator is added. The discharge regulator (a boost regulator) is used during eclipse to maintain the bus voltage at the required level, essentially providing a regulated bus. Introduction of either of these additional regulators adds inefficiencies, mass, cost, and complexity.

Separate charge and discharge regulators can simplify the main array power processor by separating the function of controlling the battery current. Inclusion of such regulators can also be used to maintain more stringent limits on the bus voltage. This combination permits very tight regulation, typically ±2 percent of the bus voltage. A control unit is incorporated in the system to sense the bus voltage and provide the desired logical control of both the array and the charge regulators (and discharge regulator, if used). This control unit is usually autonomous with some commandable settings. Having three separate regulators can lead to duplication of components, however, and is not commonly used.

2.5.1.3.4 Battery Charging—The current level used for battery charging is described by a ratio of the battery rated capacity (in ampere-hours) to the time it would take to provide that amount of charge at a fixed current. For example, a 25 ampere-hour (Ah) battery charged at 2.5 amperes is a C/10 rate (25 Ah/10 hours = 2.5 A). The same current applied to a 50 Ah battery results in only a C/20 rate (50 Ah/20 hours = 2.5 A). The C/time ratio provides a useful measure of the proper charge rate because it is independent of battery size.

Generally, batteries are charged at a high rate (C/2 to C/5) followed by a low rate (C/10 to C/100) trickle charge to "top off" the battery, often with an intermediate "taper" charge. This process is shown in Figure 2–7. The battery near full state of charge must not be overcharged. Overcharging results in excessive oxygen generation and degrades the life of the battery. Two common methods of accomplishing this charging exist: the temperature compensated voltage regulated (V/T) method and the constant current method.

2.5.1.3.5 Temperature-Compensated, Voltage Regulated (V/T) Charging—Battery charging based on a voltage level that varies as a function of temperature is referred to as "V/T" charging. V/T battery charging has become the preferred method of battery charge control for LEO satellites. It applies to both DET and PPT designs. Batteries are very sensitive to temperature and overcharge, and the V/T method is a proven approach to maximizing battery life. This technique is usually combined with a trickle charge control circuit and an ampere-hour integrator (AHI). The AHI is a back-up to the V/T controller to ensure that the battery is not overcharged.

The V/T method uses all the available current until a preset voltage limit is reached. Then the current is exponentially tapered such that the battery voltage is maintained at the appropriate fixed level. These levels are set according to battery temperature. In a pure V/T controlled charger, the constant voltage taper mode is maintained until discharge begins. In systems that include an AHI, taper continues only until a high return factor (usually adjustable via ground command) is detected, then the trickle charge circuit is used to complete the charge process without excessively overcharging the battery. This trickle charge controller is usually a more sensitive means of control than the V/T controller.

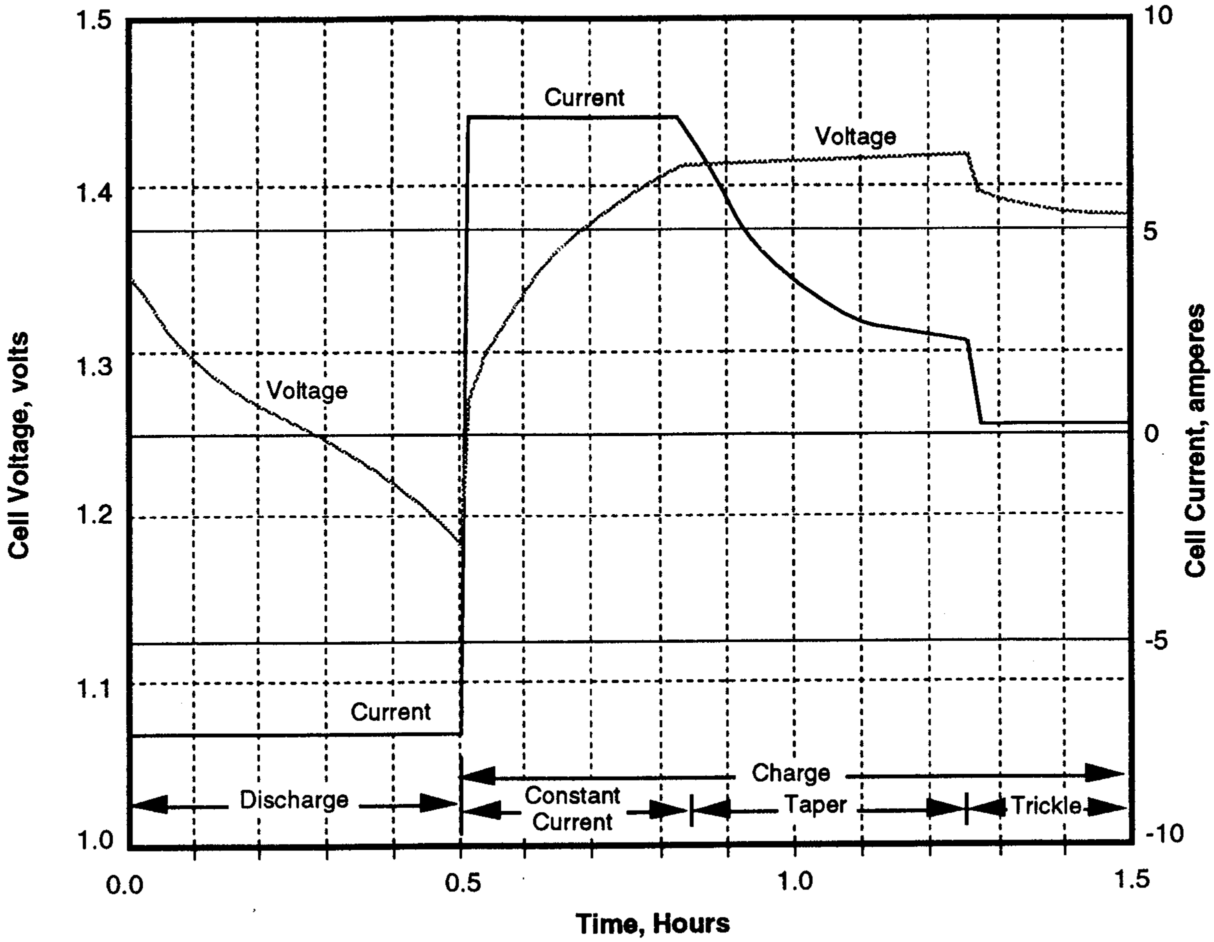


Figure 2–7. Current and Voltage Charge-Discharge Profile for Advanced NiCd Cell

2.5.1.3.6 Constant Current Charging—With the constant current technique, the battery is charged at a fixed current rate until a preset return fraction (Ah out/Ah in) is achieved, usually around 1.05. The actual return fraction is determined depending on the type of battery and its operating conditions (temperature and end-of-discharge voltage, for example). The charging then switches to a constant current at a much lower rate, typically C/50. This method is usually not adjusted for temperature.

This method is most commonly used on geostationary spacecraft, although there are some methods based on state of charge (SOC) that use constant current modes as an adjunct to V/T control in LEO applications.

2.5.2 Voltage Regulation for Payloads

The power processing schemes described in the previous sections set limits to the main bus voltage based on the solar array and battery operating points. When additional regulation is required, a central regulator can be used to provide a tighter regulation (usually 28±2 volts) rather than the typical 22-34 volts that an unregulated system provides. However, this procedure is not often used because most equipment will also need other (+5V, -5V, +15V, etc.) levels that are inefficient to distribute at low current levels over the spacecraft. This fact may not be as true for very small satellites, since line losses will be low due to the short run lengths, but the use of a

centrally regulated bus is still uncommon. Most instrumenters prefer to provide their own regulators.

A central regulator, however, can provide a clean 28±2 V that could increase the efficiency and reduce the size of a user's own regulator. On large satellites, the noise and voltage drops picked up in cabling throughout the spacecraft negates much of this advantage. On a small satellite with few payloads and short wire lengths, a central regulator may be worth considering. It could reduce the size of instrument electronics. However, the interface with such a spacecraft-provided regulator would have to be tightly coordinated with the instrument designers to assure against cross-talk.

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| A centralized voltage regulator could be a viable option on smallsats if power distribution run lengths are short enough to minimize voltage drops and noise. |

2.5.2.1 Load Distribution, Switching, and Bus Protection—In many satellites, the power is distributed to the various loads by one or more power busses. Some loads may require special ground isolation, and often a separate "dirty" bus for pyro or motor loads is provided. Pyro busses are also isolated for EMC and safety purposes in order to ensure no stray signals prematurely initiate events. The main load bus is usually divided into "essential" and "nonessential" busses, with instruments, payloads and any noncritical subsystems on the nonessential branch. This allows for load shedding in autonomous safety modes.

The distribution system includes load switching networks for transferring power to the various loads and fusing protection against faults in the loads. The device is to maintain the integrity of operation of the EPS by isolating loads.

2.5.3 Power Processing Electronics Performance

2.5.3.1 Regulation and Control (Electronics)—DET systems are used on most current satellites, but series-regulated designs using PPT can offer very efficient power processing options for small satellites. For LEO, a regulated bus is seldom used. All of the various combinations perform well, but specific comparisons depend upon the actual configuration used on a given application.

Series power regulation with an unregulated bus allows maximum use of the solar array capability. The voltage on the bus varies approximately ±25 percent. Because of this variation, control of the series regulator (control of power to the payload) and control of battery charge are required. This requirement adds to the overhead of control functions and thus increases the mass charged to the EPS.

For the shunt-regulated system, control of up to three separate regulators (the shunt regulator, and the battery charge and discharge regulators) is required to regulate the bus voltage. Without the discharge regulator, the bus voltage is tied to the battery and again varies by 25 percent. If tighter regulation is desired, usually to about ±2 percent, a discharge regulator may be necessary. In addition, control of operation of the protection system is required for fault handling in both systems.

The type of architecture selected will affect the solar array sizing, based on the factors used for each approach in the energy balance equation. The different factors used account for losses in the power processing electronics. The PPT approach has two factors. The first is the accuracy of the peak power tracking circuit, usually 0.95. The second is the efficiency of the PWM regulator, typically 0.92. The DET system has only one factor, known either as the voltage variation or the solar array utilization factor. This factor accounts for the operation of the DET

system off of the peak power point of the solar array, while the array sizing process is based on peak power output ratings. Depending on the solar array design and the particular implementation of the DET circuit, this value can range from 0.82 to 0.93.

2.5.3.2 Distribution—The power bus is typically a low voltage dc bus operating at a nominal 28 volts. Higher voltages of about 35 volts have been used, and the European Space Agency (ESA) typically uses a 50 volt bus. However, these busses are still considered low-voltage busses. High voltage is considered to be at least 100 V and is advantageous for applications with very large loads or involving large physical distribution distance. The design for the EOS-AM (Earth Observing System) spacecraft uses a 120 volt dc bus. The primary problem for higher voltage busses is that an increased number of battery cells in series is needed, leading to reliability concerns.

Both positive and negative busses have been used, but the negative bus is used only when dictated by specialized experiment requirements. Transfer of bus power across rotating joints is accomplished by multiple redundant slip rings and brushes, or by a flexible cable. In some designs, the motion is intermittent, and/or the cable is rewindable. Transfer could also be done using rotary transformers, but only for AC distribution systems. For these distribution systems, EMI/RFI (electromagnetic compatibility), grounding, (avoidance of ground loops), and system protection (against faults in user equipment), are serious considerations.

2.5.3.3 Future Electronics Performance—The approach that will provide the most gains for power processing in smallsats is to use more efficient electronics with robust designs. Completely redundant units are not generally affordable on small, "inexpensive" satellites, so the electronics must have selective redundancy and be able to accommodate faults without losing the entire power processing capability. The electronics must be smarter and more autonomous since these satellites won't have 100 percent ground contact (i.e., few smallsats have TDRSS capability). The power electronics will either have their own intelligence or will be tightly coupled to the spacecraft's core processor for control functions.

> Power processing electronics using high efficiency circuits and robust designs will provide more performance and reliability benefits for smallsats than the use of completely redundant units.

Worthy of future study is the conversion to high voltage (150-300 volt) systems and to AC systems in order to reduce distribution mass or losses and accommodate high power requirements. Several previous studies have examined the potential advantages of higher voltages, but they only considered larger platforms. The current supposition is that small satellites are not likely to benefit from such measures as high voltage busses because the load currents and distribution run lengths are too small. Additionally, so much equipment is designed to operate at 28 volts that conversion to any other level would probably result in more money expended than could be saved. The recommendation is to conduct a trade study to determine if high voltage is worthwhile on a smallsat.

> Although their relatively low total load power use suggests that high voltage power distribution offers few benefits to smallsats, a thorough trade study on this topic could identify any potential mass or cost savings.

## 2.6 SOLAR ARRAY DESIGN

The selection of the array geometry is driven by many factors. Generally, the designer's options are bounded by the need for a large solar array area to satisfy the load requirements and by the small volume and mass allocation available, especially for small launch vehicles.

The designer must work within the bounds of the solar input energy that is available to the spacecraft. The amount of this energy depends on the orbit characteristics, spacecraft, and array geometry. In addition, wide variations in Beta angle, the angle between the Sun line and the orbit plane, can lead to wide variations in solar array output. If a constant load capability is to be maintained for all conditions throughout the mission, then the solar array must be large to accommodate the poor Beta angle or else array drives must be used to increase array output. Because of their importance to array sizing, eclipse time, sunlight time and Beta angle will be integral to the discussions of the circular orbit, the elliptical orbit, and the sun-synchronous orbit that follow.

### 2.6.1 Impacts of Orbit Parameters

2.6.1.1 Circular Orbits—For circular orbits, the orbital period and maximum eclipse time vary with altitude as shown in Figure 2–8. The equations driving the chart are also provided. The eclipse duration will vary with time of year. Its rate of change depends on a combination of solar declinations (seasonal) and orbital precession. The variation in eclipse time depends on inclination, launch date, and time. When the orbit plane is in the plane of the Earth's orbit, the eclipse time will remain constant and at the maximum value throughout the year. For other orbits inclined to the Earth's orbit plane, the eclipse time varies through the year. As the inclination to the Earth's orbit plane increases toward 90°, the eclipse time can reduce significantly and disappear. At a high enough inclination and with precession, the orbit can theoretically be maintained in full sunlight throughout the year.

$$\text{Period, hours} = G \times (Re + \text{Altitude})^{1.5}$$

$$G = 2.7722\text{E-}06\text{, gravitational constant}$$

$$Re = 6378\text{ km., Earth radius}$$

$$\text{Maximum eclipse duration} = \frac{\text{Period}}{180} \times \left(90 - \frac{\text{acos}\,[Re/(Re + \text{Altitude}])}{57.14^\circ/\text{radian}}\right)$$

$$\text{Minimum Sun Time} = \text{Period} - \text{Maximum Eclipse Duration}$$

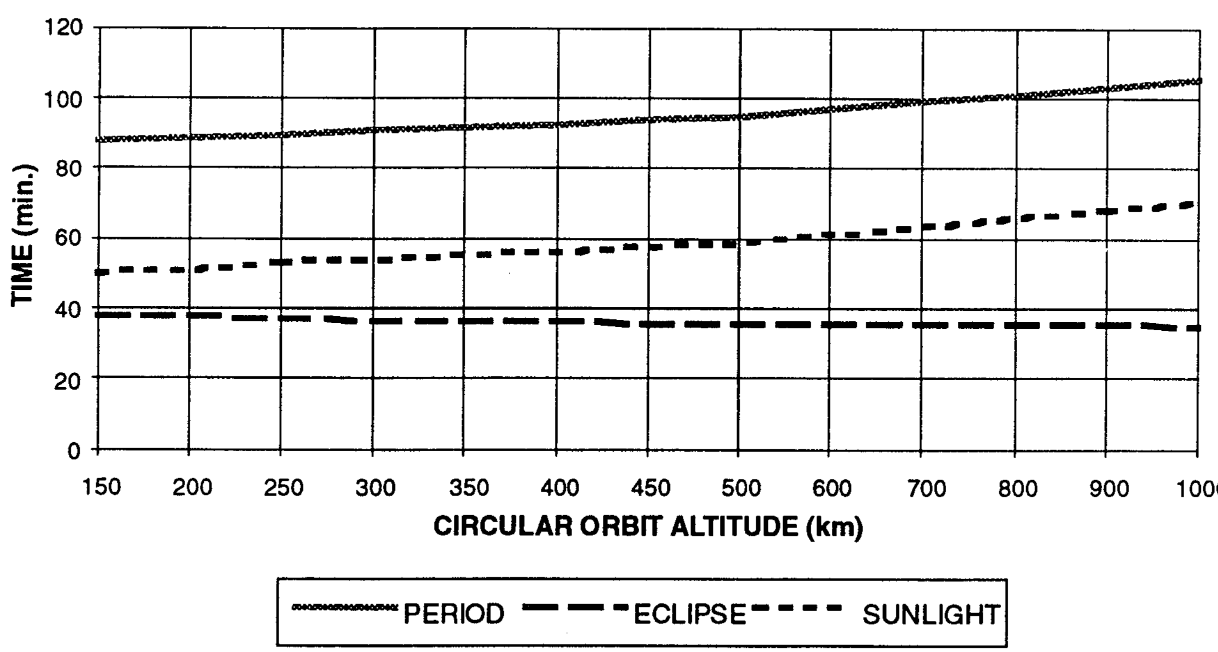



Figure 2–8. Period, Minimum Sun, and Maximum Eclipse Time

These factors are significantly affected by orbit altitude, with the effects being more pronounced at the higher altitudes. This situation results from the geometry and the physics. At the higher altitude, the Earth's shadow intersects a smaller orbital angle. In addition, the period increases with altitude while the maximum eclipse time decreases. Since the sum of the sunlight time and the eclipse time is equal to the period, the result is a larger sunlight-to-eclipse ratio.

The Beta angle also varies throughout the year, except when the spacecraft orbital plane is coincident with or perpendicular to the Earth's orbital plane or the perpendicular orbit is precessing at 360/365.24 degrees/day. This precessing orbit is called sun synchronous.

2.6.1.2 Elliptical Orbits—The same effects must be considered for elliptical orbits as for circular orbits. However, there are significant differences in the results. Since the spacecraft orbits in an ellipse with the Earth at one of the foci and since the speed of the spacecraft varies with distance from the Earth time-of-year effects are modified. For example, at one time of year, the spacecraft may be in eclipse at perigee where it is moving at a higher speed than at apogee. Later it is eclipsed at apogee, where, because it is moving slower, the eclipse will be longer, with associated effects on battery discharge. Simultaneously the sunlight portion of the orbit is reduced, providing less recharge time.

The variation of Beta angle will be similar to that for circular orbits. However, the possibility now exists to have an unfavorable Beta angle occur simultaneously with unfavorable eclipse conditions. The frequency of such worst case situations must be evaluated before the Beta angle is used to drive EPS sizing.

Also with elliptical orbits the spacecraft is subjected to a variety of altitude-dependent environments, e.g., radiation, atomic oxygen, etc. While traversing the path from perigee to apogee. For orbits with apogee altitudes high enough to traverse the radiation belts, this effect must be incorporated into the calculation of end-of-life (EOL) power generation capability.

Cumulative radiation values on a per orbit basis are typically calculated by determining the dose between altitude bands (slices) and adding these values. This laborious slice technique is required because the radiation data only exists for circular orbits and computer codes for performing these calculations are not in common use.

2.6.1.3 Sun-Synchronous Orbits—As noted above, sun-synchronous orbits can be attained with orbit precession at high inclinations. The required inclination depends on altitude as shown in Figure 2–9. For these orbits, the line of nodes (the intersection of the orbit plane with the Earth's equatorial plane) rotates eastward at precisely the same rate as the mean Sun (360 degrees per year). However, other Beta angles, depending on payload requirements, could be selected by varying the launch time. With fully synchronous precession, the Beta angle would stay fixed. Drift in Beta angle occurs because of non-idealities such as orbit-altitude decay and the nonspherical shape of the Earth. This drift which, if not adequately compensated by on-board orbit (altitude and inclination) maintenance, results in Beta angle variation and associated changes in eclipse conditions that affect both the array and battery designs.

2.6.2 Configuration Options

Figure 2–10 is a simplified flow chart of the array geometry design process. The initial factor used to determine the appropriate array configuration is the load requirement. If the loads are very small, body-mounted arrays may be adequate. Once the power requirement reaches a value where body-mounted arrays will no longer suffice, deployed panels become the only option. As power requirements increase further, the array area must get larger or orientation devices (gimbals) must be added to maximize the solar energy collected.

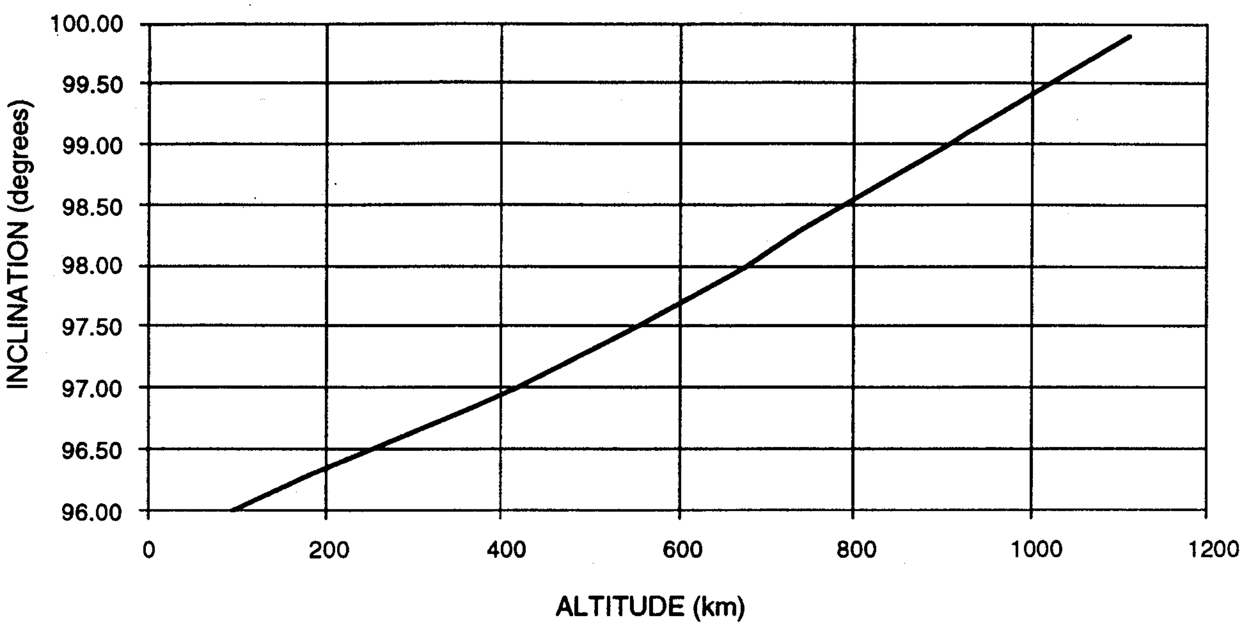



Figure 2–9. Sun-Synchronous Orbit Inclination

Orbit characteristics and spacecraft orientation are the next major factors to consider. If the spacecraft is not pointing to the Sun, the sun angle will continuously vary for fixed solar arrays (body-mounted or deployed). In this case, the deployed arrays will need to be large enough to accommodate the worst sun angles and may require one or two axes-of-orientation mechanisms. For many satellites, especially those in sun-synchronous orbits, the design has one side designated as the "sun side" always facing the Sun. This side can be used for mounting solar arrays. Since the opposite side sees dark space most of the time, it is used for thermal radiators. These factors affect the placement and orientation of body-mounted and deployed arrays.

If the satellite is in a very low altitude orbit, atmospheric drag may be enough of a problem that the arrays must be always oriented edge-forward to minimize the drag. This configuration generally limits the average energy output and drives up the total area requirement. This factor is especially a concern for small satellites with low ballistic coefficients, which are often in low orbits.

| |
|---|
| The impact of atmospheric drag on solar array configurations is a significant issue for spacecraft in low orbits. |

An option is to design movable panels. The "down side" is that drive and packaging of movable panels adds cost and increases reliability risks. In addition, the drive motors must be packaged carefully and software and testing for the drive and gimbal systems adds additional costs. These negatives must be balanced by the gains in performance offered by gimbaled solar arrays. Providing constant orientation to the Sun can greatly reduce the overall size, mass, and cost of the array. This advantage can balance the cost and mass of the drive motors. It also makes some missions possible when power capabilities of fixed arrays are exceeded.

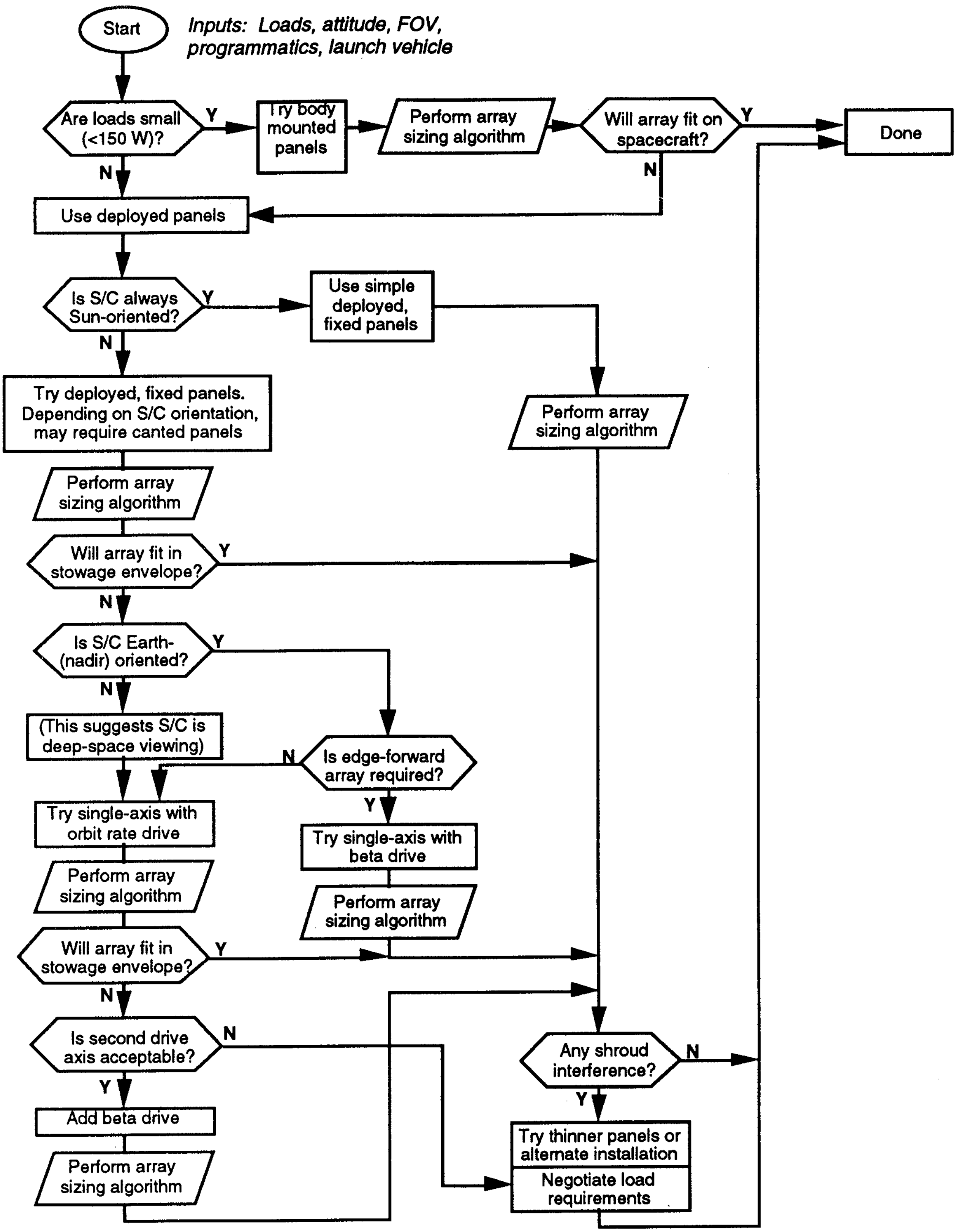



Figure 2–10. Solar Array Geometry Selection Flowchart

2.6.2.1 Body-mounted Arrays—Body-mounted arrays do not require extra structure, deployment pyros, hinges, or gimbals. However, body-mounted arrays may not be an acceptable choice for small satellites. Because of their limited body area and because only one side of the body can face the Sun at a time, they can support only about 150 watts of load. In addition to the solar cells, room must be made for thermal radiators, structural attachments, and instrument apertures. Body-mounted arrays also complicate the spacecraft thermal design because while they are efficient collectors of solar energy, they are poor radiators of heat.

2.6.2.2 Deployed Fixed Arrays—When the power requirements grow beyond the capabilities of body-mounted solar arrays, fixed deployable arrays are an option. Many configurations and geometries are possible, but the arrays are always fixed relative to the main spacecraft body. Thus area is added without introducing drive motors and slip rings or cabling for power transfer. The deployment mechanisms are usually simple pre-tensioned, spring-driven devices released by pyrotechnic or similar one-shot actuators. Recently heat-actuated Nitinol devices have came into common use.

If atmospheric drag is not an issue, a large fixed deployed array can accommodate many small satellite mission requirements. For higher power requirements deployed arrays, oriented using either one or two axis gimbals may be necessary. The substrates are rigid, lightweight structures, usually aluminum honeycomb with bonded skins.

| |
|---|
| As long as atmospheric drag is not an issue, a simple deployed, fixed, solar array panel is often the simplest design solution for smallsats. |

2.6.2.3 Deployed Oriented Arrays—Solar array drives are generally used in two ways, both of which assume a nadir-oriented spacecraft. The first way is to use an orbit-rate drive that makes up for the orbital motion of the spacecraft as it proceeds around the Earth. This drive motor must be in constant use as it drives the array one revolution each orbit (about 90-100 minutes). Slip rings or a complete rewind during eclipse are necessary to transfer power and signal lines across this mechanism. A rotary transformer could also be used.

The other way to use a drive is for Beta angle adjustment. The Beta angle changes seasonally, usually a degree or two every few days. The exact amounts depend on orbital inclination and altitude. This approach can also reduce the array size substantially, depending on the particular spacecraft orientation and array geometry. Two advantages of using this type of drive are that the motion is occasional and is not a full 360 degrees. The need for a slip ring is eliminated. A flexible cable is all that is needed. Jitter and vibration transferred to sensitive instruments are reduced.

If atmospheric drag is a concern, the arrays may require an edge-forward configuration to minimize frontal area. A Beta-tracking drive can still be used to provide significantly more orbit average power than a fixed array. If drag is not a problem, use of the orbit-rate drive provides greater and more uniform power over a single orbit, although it may not be best for orbits that have a wide range of Beta angles.

Trade studies have been performed for various deployed array configurations for nadir-oriented satellites. Results for one such study (Reference 9) of a case where atmosphere drag was a significant issue are summarized in Figure 2–11. The study evaluated five array configurations as shown in the figure. Two (the Dart and the Twin Panels with Beta tracker) had edge forward arrays, which resulted in rather large areas. The others all had orbit rate drives, which reduced the total area but increased the drag area to an unacceptable level. Therefore, the drag problem led to an edge forward array design (Twin Panels with Beta tracker) as a selected feature. From

the results, it was also concluded that the selected approach should feature two deployed panels using gimbals to make occasional adjustments for the seasonal Beta angle changes.

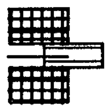
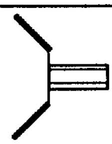
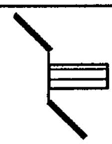
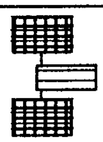
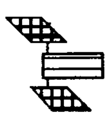
| ARRAY STYLE | | ORBIT RATE DRIVE | BETA DRIVE | CELLS ON 2 SIDES | RELATIVE ARRAY AREA | RELATIVE DRAG AREA | COMMENTS |
|---|---|---|---|---|---|---|---|
|  | Dart | No | No | Yes | 3.59 | ~0 | Slightly larger battery required |
|  | Twin Panels w/Orbit Drive | Yes | No | No | 2.83 | 1.27 | |
|  | Reverse Twin Panels w/Orbit Drive | Yes | No | Yes | 1.41 | 0.63 | |
|  | Twin Panels w/ß Tracker | No | Yes | No | 2.14 | ~0 | Slightly larger battery required |
|  | Twin Panels w/ß & Orbit Drives | Yes | Yes | No | 1.0 | Worst: 0.63 Else: 0.63xcosß | |

Figure 2–11. Relative Merits of Candidate Solar Array Geometries

2.6.3 Packaging Issues

Limited launch vehicle shroud volume constrains the solar array to be as small as possible. This constraint usually conflicts with the payload requirement to have as much power available as possible. Deployment mechanisms (hinges, lock-downs, release devices, etc.) and orientation mechanisms (drive motors, gimbals, etc.) require additional mass and volume. These mechanisms also increase costs and add some mission risk. To keep costs down the overall load power requirement must be strictly managed. For smallsats it is critical that project management prevent requirements from growing beyond what a "simple" power subsystem can provide.

> **Project managers most take an active role in maintaining and controlling realistic performance requirements in order to minimize cost and complexity.**

2.6.3.1 Deployable Panel Packaging Options—Deployed panels may be packaged accordion style against two opposing sides of the spacecraft or wrapped around the entire vehicle. The first option is most appropriate for satellites with rectangular cross sections, while the second option is appropriate for many-sided satellites. Considerations include available mounting and tie-down points, instrument accommodations (location and view factor), shroud volume, and access to the satellite.

Deployable arrays should be designed as opposing pairs to maintain symmetry. This is especially important on smallsats where the attitude control capability may be limited. Small simple arrays consisting of single rigid panel may be readily folded against the sides of the spacecraft. If the area of one wing is larger than the face of the spacecraft body, the wing must be divided into several hinged panels and folded. Simple, spring-loaded hinges that are compact and reliable are used. A single motor can perform double duty as the deployment drive and the articulation drive for gimbaled arrays if the hinge line for the deployment motion is the same as the gimbal axis. If the deployment axis is different, the drive motor must be placed closer to the spacecraft than to the deployment axis to avoid having the power and signal transfer cables flex across both the deployment axis and the gimbal axis. Keeping the gimbal drive closer to the spacecraft also permits it to be partially buried in the spacecraft structure, thus simplifying the packaging.

2.6.3.2 Rigid Substrates Versus Flexible Blankets—Usually, the primary design driver for the array structure is achieving low mass. However, the panels also must maintain orientation, survive the launch environment, and not pass disturbances to the spacecraft via thermal "twang" or reinforcement of frequencies of the basic structure. These factors drive the array's minimum stiffness level. The standard approach is to use a honeycomb structure made out of aluminum, composites, or a combination. All composite panels have a rigid carbon fiber frame and solar cells made of a very thin carbon fiber substrate (Reference 10). The Solar Maximum Mission (SMM) satellite, which was launched in February 1990, used an aluminum core with graphite epoxy skins.

Very large arrays can minimize mass by avoiding rigid substrates. In these designs, the solar cells (including coverglasses, interconnects, etc.) are attached directly to a lightweight flexible blanket, such as Kapton Figure 2-12. Stiffeners and deployment structures, such as bi-stems, are still required. This type of design minimizes mass and can be used for both large and small satellites.

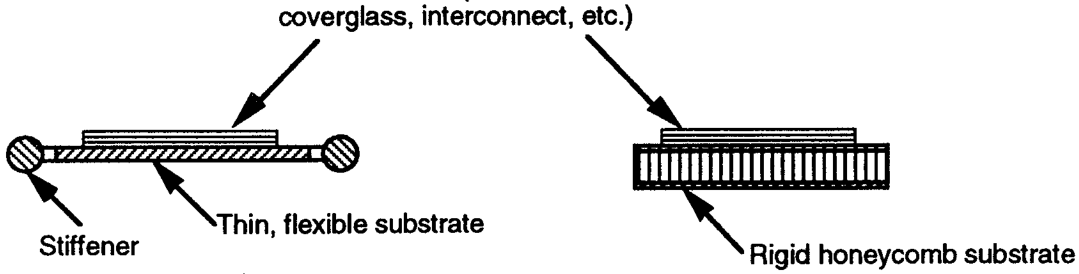



Figure 2–12. Cross-Section of Flexible and Rigid Array Panel Substrates

| The use of flexible substrates offers the possibility of significant solar array panel mass reduction. |
|---|

An array can be stowed most efficiently if the array blanket is made of a flexible substrate. Then all that is required is a hinged structural frame. The Ranger telerobotics technology satellite, under development by NASA and the University of Maryland, may be designed with a parasol-fan type of deployment mechanism (Figure 2–13) (Reference 11).

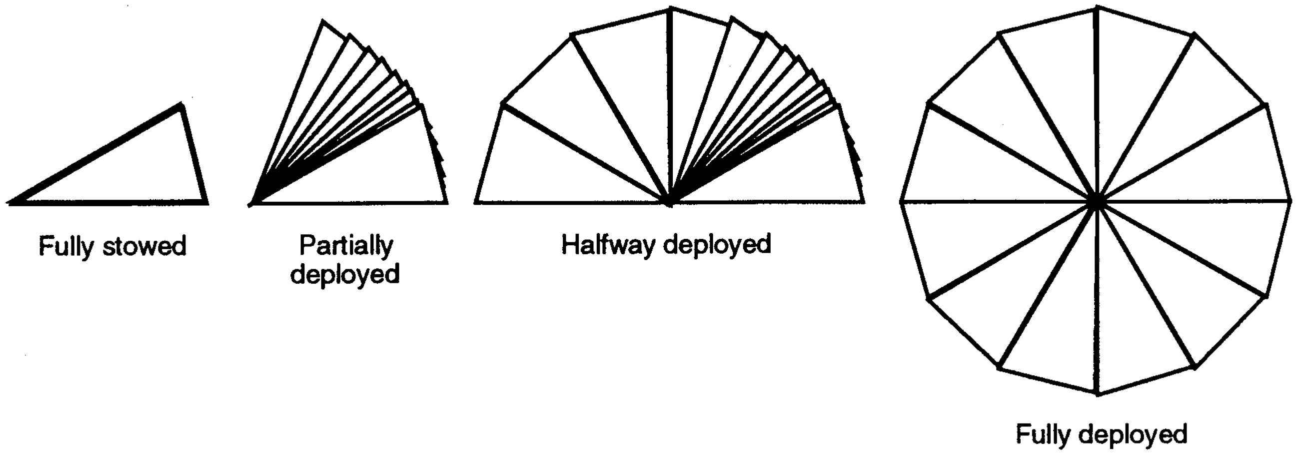



Figure 2–13. Flexible Parasol Deployable Solar Array Concept

2.6.4 Sizing the Array

2.6.4.1 General—Driving the size of the array are the spacecraft's power requirements. For this determination, spacecraft power can be defined as the spacecraft bus power, the instrument or payload power, the power required for recharging the batteries, the power lost due to subsystem inefficiencies, the power required to meet energy balance conditions, and the power margins that must be added to cover power growth during project development and required margin of safety to limit the risk.

The power generation capability of the solar array also influences the size of the array. This capability is determined first by determining the production efficiency of a solar cell, increasing the area of the array to compensate for the degradation of the cells that may occur during the panel fabrication process, and subtracting the effects of operating in orbit over the mission lifetime on the power generation capability of the cells.

Basic solar array design results from conducting iterative trade-off studies that compare solar cell and coverglass properties against end-of-life power generation, mass, size, and cost. These trade-off studies include pertinent factors in the array system, such as substrates, drives, etc. From these studies, the cell type is selected and the array is sized. The overall sizing and cell selection process is illustrated in Figure 2–14.

Among the earliest trade studies in the design of a power subsystem is the trade to select the type of solar cell to be used in the array. The factors used in selection are: flightworthiness (stability and qualification status), size, mass, cost, and availability (including availability of a performance data base). At the present time only Silicon and GaAs solar cells meet the first and last of these requirement factors. As a result the trade studies involve only these two cell types. The array size relates directly to the on-orbit EOL efficiency. This is determined by the BOL efficiency, the temperature and temperature coefficient of power, and the radiation and radiation degradation. Because radiation at the low altitudes of smallsats is relatively benign, differences due to radiation are relatively insignificant. The combined effects of BOL efficiency and temperature can be significant. On TRMM, for example, the Si/GaAs array area ratio is approximately 1.4. This results in similar impact on the array mass, because the entire substrate, hinges, and other mechanical system components are involved. These must be traded against array cost. GaAs cell costs are significantly (as much as a factor of 10) higher than Si cell costs, but because of the reduced area (number of cells required, substrate size, complexity, and other factors) the cost ratio at the array level is reduced to about 1.5. Final cell selection is based on

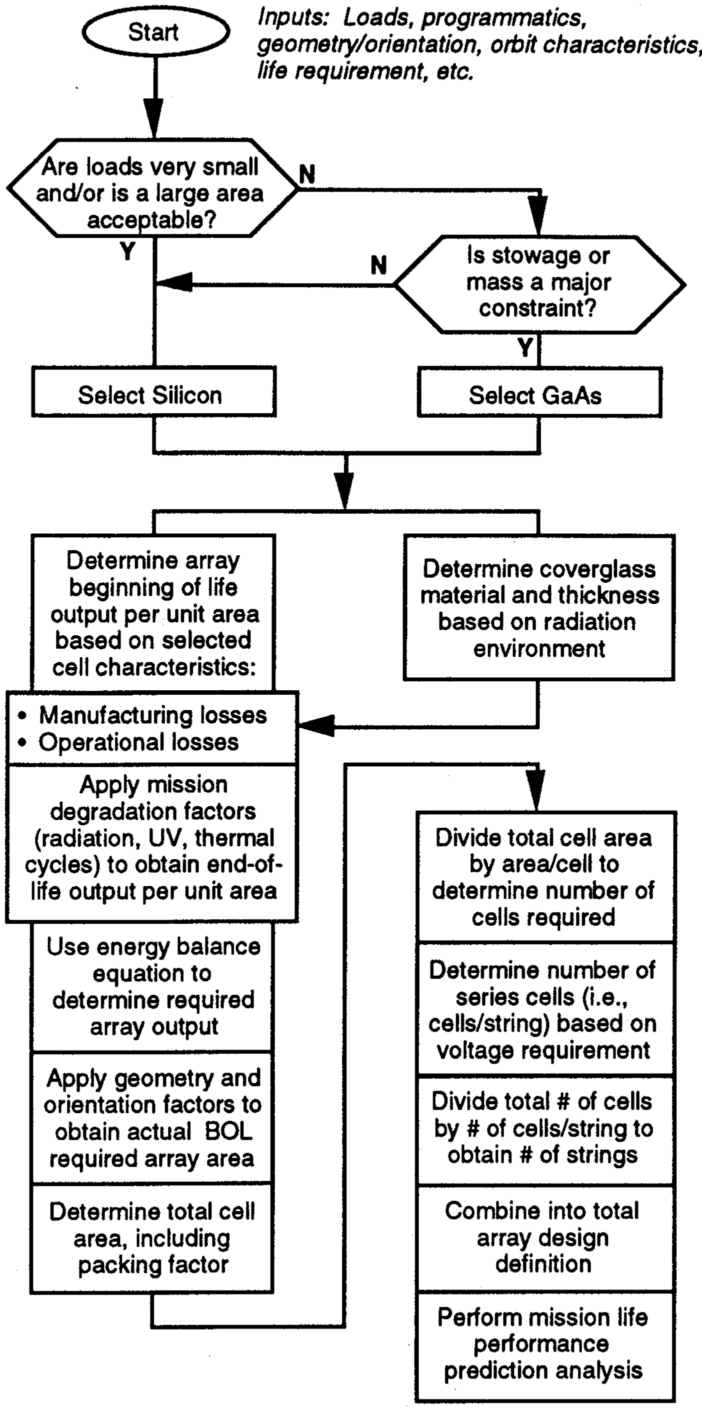



Figure 2–14. Solar Cell Selection and Array Sizing Flowchart

detailed trades among these factors. It should be noted that any one could be an over-riding factor. For example, on SAMPEX and FAST, Si arrays, because of size limitations, could not meet the power requirements, while on XTE GaAs pricing uncertainties led to the selection of Si.

| Gallium arsenide solar cells offer a significant reduction in array area compared to silicon, but careful cost trade-offs must be made before a final cell selection is made. |
|---|

2.6.4.2 Solar Array Performance

2.6.4.2.1 Current Technology—Silicon solar cells are usually used on current satellites, although GaAs is becoming increasingly common.

Typical solar cell performance characteristics are shown in Table 2–5. Data for Si and GaAs/Ge cells are obtained from manufacturer's literature and proposals. Tandem cells and InP cells are currently considered experimental. Their performance estimates are based on discussions with the cell developers (References 12, 13, and 14). Note that radiation degradation is given for a 1000 km orbit. For a 400 km orbit, which is more typical for smallsats, the radiation degradation is less than 1 percent for all types. For the cells, the primary factor of significance is efficiency. Low temperature coefficients and radiation resistance are significant because they help maintain the efficiency through the lifetime of the array. For the array, the cell efficiency is even more important because high efficiency allows for proportionate reduction in array size and overall (including blanket, substrate, and mechanisms) mass. A cross section of typical array mechanical design features is shown in Figure 2-15. In combination, these features lead to higher EOL efficiency, which increases both specific power (W/kg) and power density (W/m$^2$), as shown in Tables 2-6 (Reference 15), and 2–7.

Table 2–5
Solar Cells

| Parameter | Silicon | GaAs/Ge | Tandem ($GaInP_2$/ GaAs/Ge) | InP |
|---|---|---|---|---|
| Efficiency (%) | 14.5 | 18.0 | 22.0 | 19.0 |
| PWR. Temp. Coeff. (%/°C) | 0.46 | 0.22 | 0.22 | 0.22 |
| Rad. Deg.* (%) | 5.1 | 4.7 | 4.7 | 2.5 |

**2-Yr. at 1000 km*

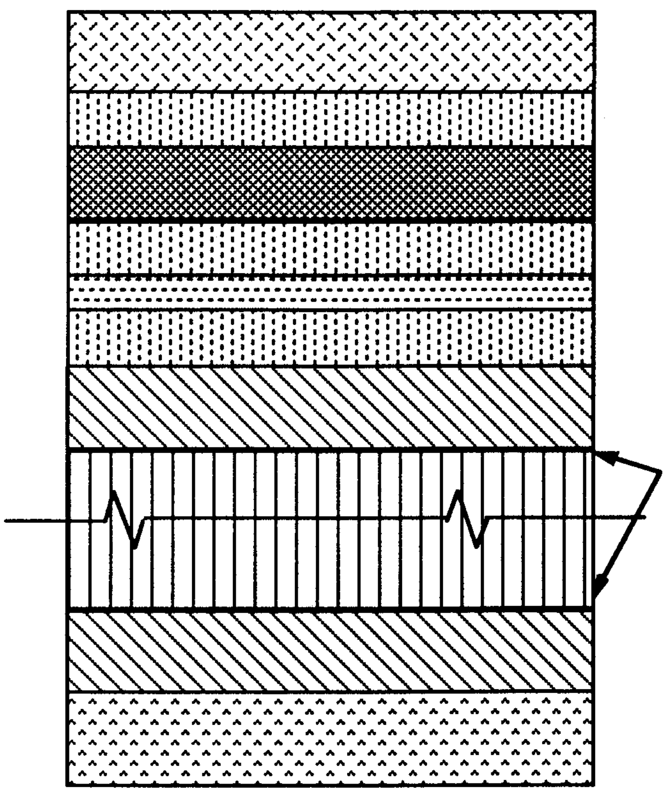
Table 2–6
Solar Array Specific Power (W/kg) at 2 Year EOL

| Cell Type | Flat Plate | Rigid Fold-Out | Flexible Roll-Out | Flexible Fold-Out |
|---|---|---|---|---|
| Silicon | 20-25 | 20 (TRMM) | 16 (HST)-24 | 66 (SAFE)-130 (APSA) |
| GaAs/Ge | 50 | 27 (TRMM) | 32 (EOS) | NA |

Table 2–7
Nominal Calculated Cell and Array Power Output

| Production Stage | Silicon | | GaAs | |
|---|---|---|---|---|
| | Degradation factor | Power density, W/m2 | Degradation factor | Power density, W/m2 |
| Bare production solar cell | 1.000 | 196.2 | 1.000 | 243.5 |
| After array fab. (p.f. = 0.9) | 0.809 | 158.7 | 0.809 | 197.0 |
| Array BOL (T = 70°C) | 0.725 | 115.1 | 0.816 | 160.8 |
| Array EOL (2-yr. @ 400 km) | 0.946 | 108.8 | 0.946 | 152.3 |
| Array EOL (2-yr. @ 1000 km) | 0.897 | 103.2 | 0.901 | 144.9 |

**Assumes a circular orbit at altitude shown, inclinations from 30° to 90°, 70°C cell operating temperature (typical for deployed arrays on honeycomb substrates), and 6 mil coverglasses. Degradation will be slightly lower for lower inclinations.*



| Item | Thickness, cm | Areal Density (gm/sq. cm) |
|---|---|---|
| Coverglass (6 mil) | 0.0127 | 0.0331 |
| Adhesive | 0.0102 | 0.0110 |
| Solar cell | 0.0140 | 0.0842 |
| Adhesive | 0.0102 | 0.0110 |
| Insulator | 0.0064 | 0.0065 |
| Adhesive | 0.0102 | 0.0110 |
| Aluminum facesheet | 0.0152 | 0.0411 |
| Core adhesive (within core) | 0.0127 | 0.0178 each |
| Aluminum core | 1.27 | 0.0107 |
| Aluminum facesheet | 0.0152 | 0.0411 |
| Thermal paint | 0.0178 | 0.0267 |

Figure 2–15. TIMED Solar Array Cross Section
(Outermost 2 of 3 Panels)

The differences in specific power going from Si to GaAs/Ge solar cells shown in Table 2–6 result from power increases due to the three factors shown in Table 2–5, and from mass reductions due to the reduced panel area made possible by the higher power capability. The BOL efficiency of GaAs/Ge cells is 24 percent greater than for Si. The temperature coefficient of power is different by a factor of 2.09, resulting in a difference of 8 percent at 60°C. The radiation degration, however, is not significant at the low, 1000 km altitude, and only amounts to

about 0.5 percent. This total of 32.5 percent translates to a 32.5 percent reduction in array area. In addition to mass reductions because of the reduced area there are mass reductions due to the elimination of hinges, wire lengths, etc., resulting in an overall gain of as much as a factor of 2 depending on details of the array design.

2.6.4.2.2 Future Performance—The use of GaAs cells is now common. Although more expensive and heavier than silicon cells, the increased performance (higher efficiency, low temperature coefficient and radiation insensitivity) makes up for the differences in all but the simplest applications. The use of new coverglasses with bandpass (as opposed to low wavelength cut off) filters can add 1-3 percent to the array's performance.

Multi-bandgap solar cells have the potential to achieve 30 percent energy conversion efficiency compared to 18 percent. Small experimental cells with efficiencies in excess of 25 percent have been made. Large cells suitable for flight solar arrays with efficiencies of 22 percent have been built. Because of the MOCVD production techniques used, the efficiency gain (20 percent compared to GaAs) for these cells is expected to exceed the increased cost (10 percent compared to GaAs). In addition, higher performance cells will reduce array size and therefore the total manufacturing costs. This gain in performance could benefit missions that need smaller arrays to meet size and mass limitations.

New multi-bandgap solar cells could be very beneficial to small satellites with severe constraints on solar array size and mass.

The use of lighter weight array substrates made of composite materials is also becoming more common and is especially applicable to smallsats. The advantage is the lower mass. As manufacturing techniques become more mature and as standardized designs and components become available, costs will decrease and approach the cost of aluminum honeycomb designs.

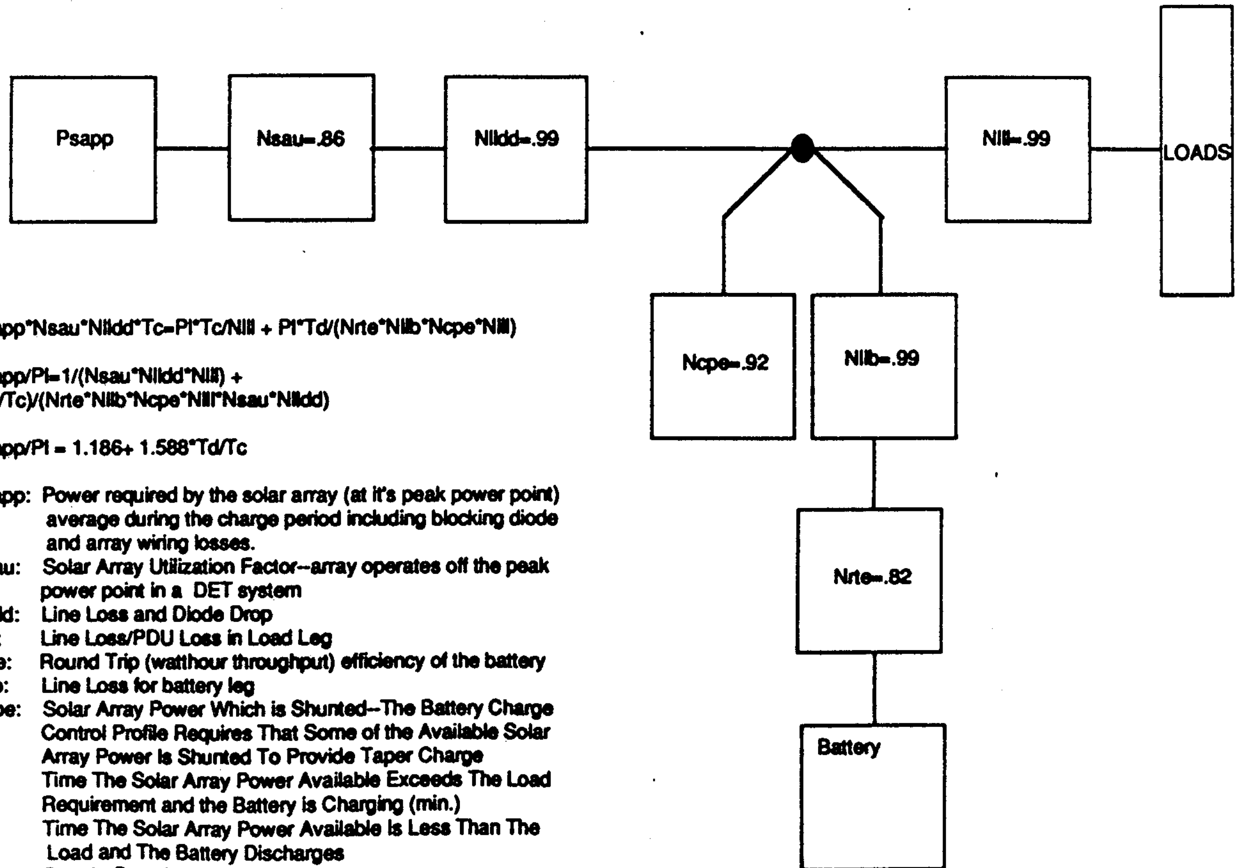
Solar array panel substrates made of lightweight composite materials offer mass savings for smallsats.

2.6.4.3 Loads—Spacecraft power requirements (the load) are determined by adding the requirements for each of the subsystems and each of the instruments. Each element's power-time profile is calculated. The power requirement is prorated according to each element's proposed duty cycle.

Since the energy balance calculations are based on the combination of a full orbit's eclipse and sun periods, orbit average loads can be used if they are roughly the same during eclipse as they are in sunlight. If the loads are appreciably different, and the power-time profile includes significant load level changes, more complex calculations are required.

2.6.4.4 Energy Balance—The power necessary to recharge the batteries is the energy (power x time) used from the batteries to operate the spacecraft during eclipse or at peak load conditions when solar array power alone is inadequate. The "Energy Balance Equation" is then applied to determine the solar array power generation requirement by integrating the power used, the inefficiencies of the subsystem, and the energy balance requirements imposed.

The basic energy balance equation for PPT systems is shown below and in Figure 2–16:




Psapp*Nsau*Nlldd*Tc=Pl*Tc/Nlll + Pl*Td/(Nrte*Nllb*Ncpe*Nlll)

Psapp/Pl=1/(Nsau*Nlldd*Nlll) +
(Td/Tc)/(Nrte*Nllb*Ncpe*Nlll*Nsau*Nlldd)

Psapp/Pl = 1.186+ 1.588*Td/Tc

Psapp: Power required by the solar array (at it's peak power point) average during the charge period including blocking diode and array wiring losses.
Nsau: Solar Array Utilization Factor--array operates off the peak power point in a DET system
Nlldd: Line Loss and Diode Drop
Nlll: Line Loss/PDU Loss in Load Leg
Nrte: Round Trip (watthour throughput) efficiency of the battery
Nllb: Line Loss for battery leg
Ncpe: Solar Array Power Which is Shunted--The Battery Charge Control Profile Requires That Some of the Available Solar Array Power Is Shunted To Provide Taper Charge
Tc: Time The Solar Array Power Available Exceeds The Load Requirement and the Battery is Charging (min.)
Td: Time The Solar Array Power Available Is Less Than The Load and The Battery Discharges
Pl: Steady-State Load Requirement

Figure 2-16. DET Energy Balance Equation

$$P_{sa} = \frac{P_l}{N_{sll} \times N_{ppt} \times N_{pdu} \times N_{reg}} \left[ 1 + \frac{T_e/T_s}{N_b \times N_{bll}} \right]$$

In these expressions, $P_{sa}$ is the required array power on an orbit average basis, $P_l$ is the required load power (total of spacecraft plus instruments), $T_e/T_s$ is the discharge-to-charge time ratio, and the various Ns are the power subsystem efficiency factors. Typical values (which will vary depending on details of the design) and definition of these factors are as follows:

| | | | | |
|---|---|---|---|---|
| $N_{sll}$ | = | 0.99 | = | line loss between solar array and main bus |
| $N_{ppt}$ | = | 0.95 | = | accuracy of peak power tracker |
| $N_{pdu}$ | = | 0.99 | = | line loss between main bus and power distribution unit (PDU), including loss in PDU |
| $N_{reg}$ | = | 0.92 | = | loss in array series regulator |
| $N_b$ | = | 0.82 | = | battery charge-discharge inefficiency |
| $N_{bll}$ | = | 0.99 | = | line loss between main bus and battery |
| $N_{vv}$ | = | 0.82 to 0.93 | = | voltage variation factor for DET |

Sometimes a diode drop factor is included in energy balance equations. This factor is related to the isolation diode at the solar array output. In this discussion, that factor will be assumed to be included in the factors used to size the solar array. Another potential factor that applies to PPT and DET systems is the solar array charge profile. If, for example, the array geometry creates a charge profile shaped like a sine function as seen in Figure 2-17 (initially low, highest at orbit noon, then low at sunset), the battery will not recharge optimally. Battery recharge is most

efficient when the initial rate is high. Therefore, a penalty must be assessed in the sizing process to account for a poor charge profile.

Another factor in the energy balance equation, the effective eclipse ratio ($T'_e/T'_s$ or discharge-to-charge ratio), is also based on array geometry. This must be considered after the "raw" array area requirement is calculated using the "true" eclipse ratio, $T_e/T_s$, based solely on time in eclipse. The energy balance equation is used to calculate the energy required from the array each sun period. To determine the array area that will deliver this energy, corrections must be made for the time-integrated actual area projected towards the Sun each orbit.

This determines the effective eclipse period, because the true eclipse period, which is determined by orbital parameters only, is often not the same as the duration that the batteries are discharging. Effectively, the spacecraft/solar panel combination must rotate into a sun orientation after coming into orbital day. The difference between these ratios (time in eclipse/time in sun versus time discharging/time charging) can change the solar array size that was determined from the energy balance equation. With sun synchronous or orbit-rate gimbaled arrays, this process is simplified. For fixed or Beta tracking arrays in Earth-oriented satellites, the geometry calculations are more complex.

Figure 2-17 shows three time intervals that must be calculated: 1) the time the spacecraft is in sunlight, $T_s$; 2) the time the Sun illuminates the solar cells, $T_{sa}$; and 3) the time the array current is greater than the load demand thus allowing the battery to be charged, $T_s''$. This particular plot is for the array configuration used on the two TIMED spacecraft, a fixed edge-forward array at a Beta angle of 0° (Sun in the orbit plane). Thus the peak output is at noon and is at minimum just after sunrise and just before sunset. Any fixed array will have a similar sine-shaped curve, although the peaks and valleys might be at different times. As the Beta angle increases, the Sun will illuminate the array for a longer period of time (eclipse duration decreases as Beta increases) and the incidence angle becomes closer to normal at sunrise and sunset. This assumes a drive mechanism is used to adjust the array about the Beta tracking axis. Note that beginning around Beta = 30°, there are portions of the orbit when the array is facing the Sun but the spacecraft is still in the eclipsed portion of the orbit. At Beta = 90°, the Sun is normal to the orbit plane, and hence the solar array (for a edge-forward array) throughout the orbit. Figure 2-18 shows the array output over the course of the orbit as a function of the Beta angle for a Beta-tracking, edge-forward array (Reference 16).

The energy balance calculations determine how large the array must be to meet the load requirements, and this is generally done for worst case conditions. As can be seen in Figure 2–18, if the array output is orbit averaged (over the illuminated portion of the orbit), the 0° Beta case is the worst in terms of energy production. At higher Beta angles the required array area would be smaller, since the eclipses are shorter and the average incidence angle is more favorable. For fixed arrays, especially edge-forward ones, this results in a large divergence in array output versus array power required, as charted in Figure 2–19.

For elliptical orbits, the calculations are more challenging. The projected area geometry calculations are now time-dependent, as the orbital motion around the Earth (and hence, the angle of the solar array relative to the Sun) is not constant. Also, seasonally dependent factors, such as the location of the spacecraft during eclipse or sunlight relative to perigee or apogee, become very significant because of this time-dependency and add to the complexity.

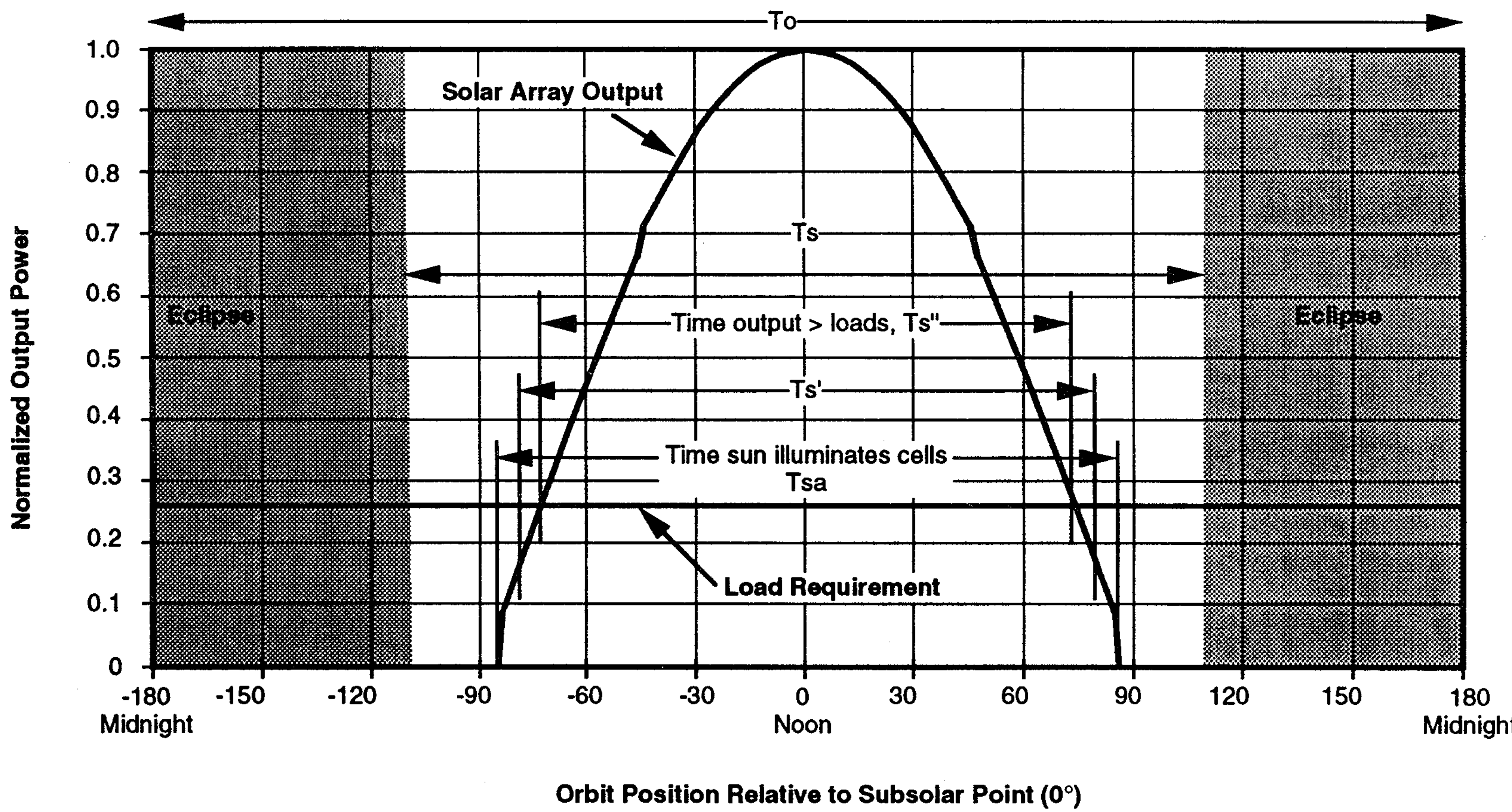



Figure 2-17. Solar Array Output Profile

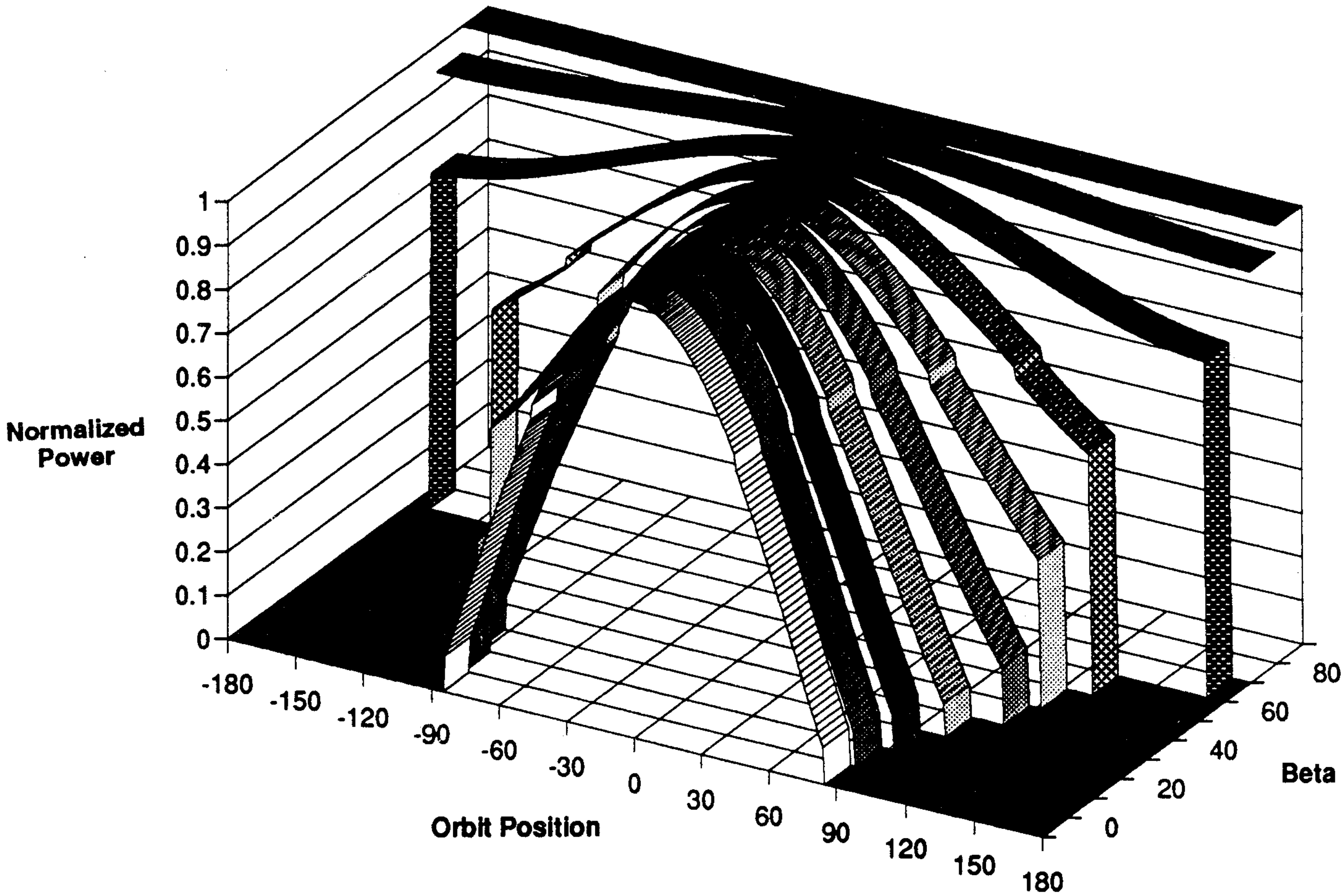



Figure 2–18. TIMED Solar Array Output During a Single Orbit Versus Beta Angle

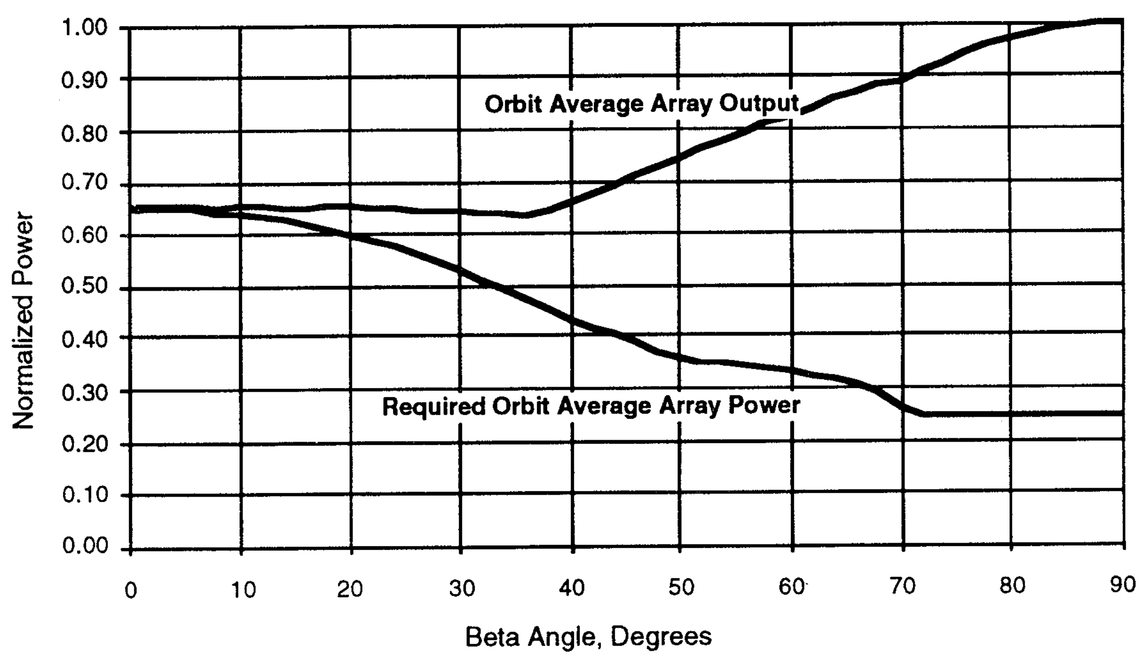



Figure 2–19. Array Power Requirement and Output Versus Beta Angle

2.6.4.5 Array Degradation—Once the "raw" EOL array output requirement is determined by the energy balance and geometry calculations, degradation factors are applied to obtain the beginning of life (BOL) values.

The solar panel power is calculated based on power density (power per unit area) starting with the electrical characteristics of the selected cell type. A typical set of values for GaAs cells is shown in Table 2–8. This output is reduced by anticipated manufacturing losses, such as shown in Table 2–9. These losses include those associated with assembly of the individual cells into an array. The packing factor can range from 0.85 for irregular surfaces with cut-outs to 0.92 for clear unobstructed surfaces. Cell performance data and manufacturing loss estimates are obtained from current American cell manufacturer's literature and proposals (References 17 through 20).

The BOL power is further reduced by the operational effects due to seasonal solar intensity variation, due to operating in space at a temperature different from laboratory standard tests and the cell's temperature coefficient of power, and due to operating at a voltage point, not at the peak power point, for DET systems (Table 2–10). Finally, EOL power is determined using degradation factors for UV, radiation, and thermal cycle losses for the particular cell assembly type, orbit and lifetime. Typical values for a 2-year mission in a 400 km circular orbit are shown in Table 2–11.

Table 2–8
Initial GaAs Cell Characteristics

| Characteristic | Product. Spec. |
|---|---|
| Efficiency (%) | 18.00 |
| Pmp (mw/cm$^2$) | 24.35 |
| Vmp (volts) | 0.87 |
| Jmp (ma/cm$^2$) | 28.00 |
| Jsc (ma/cm$^2$) | 30.40 |
| Voc (volts) | 1.02 |
| Pmp (w/m$^2$) | 243.54 |

Table 2–9
Manufacturing Loss Factors

| Element | Factor |
|---|---|
| Glassing | 0.985 |
| Mismatch | 0.980 |
| Measurement | 0.980 |
| Cleanliness | 0.990 |
| Packing Factor | 0.900 |
| Harness & Diodes | 0.960 |
| COMBINED | 0.809 |

Table 2–10
GaAs Operational Loss Factors

| Element | Value | Factor |
|---|---|---|
| Solar Int. (w/m$^2$) | 1353 | 0.967 |
| Temperature (°C) | 70 | 0.908 |
| Temp. Coeff. (%/°C) | -0.22 | |
| Voltage Offset | | 0.930 |
| COMBINED | | 0.817 |

Table 2–11
2-Yr Flight Loss Factors

| Element | Si Factor | GaAs Factor |
|---|---|---|
| UV Degradation | 0.965 | 0.965 |
| Rad. Degr. (1000km)* | 0.949 | 0.953 |
| Thermal/Random | 0.980 | 0.980 |
| COMBINED | 0.897 | 0.901 |

**At 800km and below, radiation degradation is negligible.*

As part of this process, trade studies must be made to select such component parameters as the coverglass thickness. This value is highly dependent on the radiation environment. For satellites in 400 km circular orbits, radiation degradation is not severe and generally a 6 mil. coverglass will suffice. However, an extreme example would be the radiation degradation for a spacecraft in a 4350 x 146 km orbit. Figure 2–20 shows the effect of the coverglass thickness on the overall radiation degradation over the first year in orbit. This analysis is one of several trade studies that must be performed as part of the array design process.

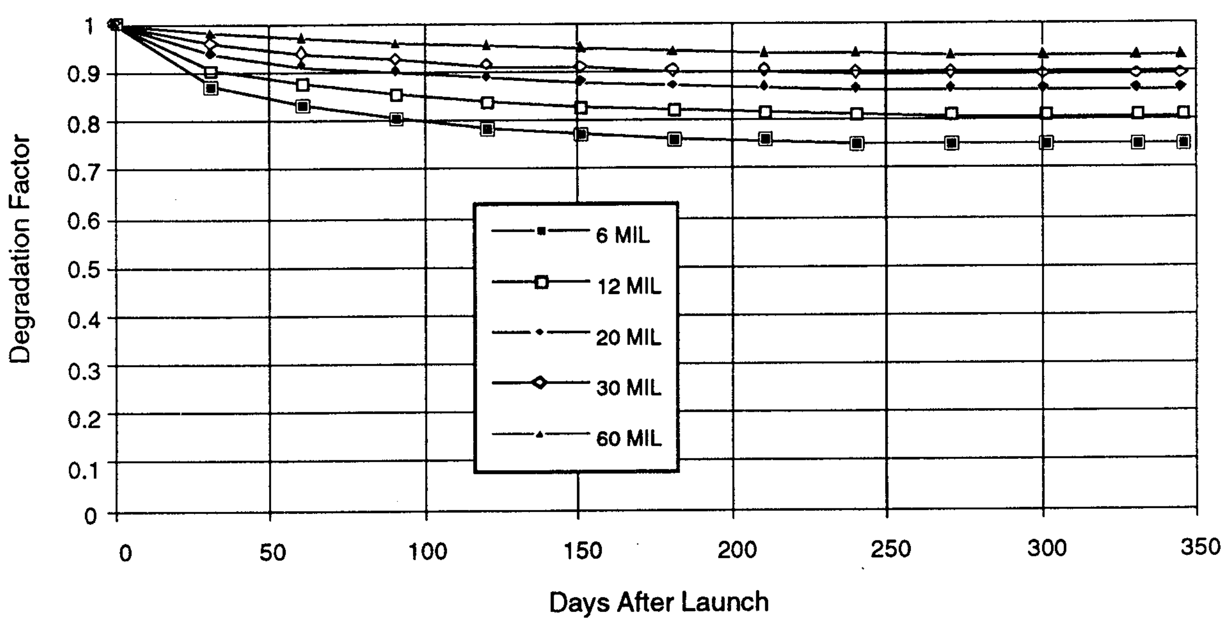



Figure 2–20. Solar Array (4350 x 146km) Degradation vs Glass Thickness for an Elliptical Orbit

> Solar array degradation is negligible for circular Earth orbits of 500 km or less, regardless of cell type.

The array sizing analysis proceeds by applying the degradation factors for the two-year, 400 km orbit example to the initial BOL cell output value of 243.5 $W/m^2$, as given in Table 2–8. Applying the loss factors from the next three tables results in an EOL array output of:

$$243.5\ W/m^2 \times 0.809 \times 0.817 \times 0.946 = 152.3\ W/m^2$$

Note that for a low 400 km orbit the radiation degradation in Table 2-11 would be 1.000, resulting in a flight loss factor of 0.946. To determine the cell circuit configuration, the number of cells required must be calculated. The EOL output of 216.5 $W/m^2$ was obtained assuming a packing factor of 0.9. To calculate the number of cells required, one must get back to the cell level, and this packing factor must be taken out.

$$152.3\ W/m^2 / 0.9 = 169.2\ W/m^2$$

The output for a 2 x 6 cm cell is found via:

$$\frac{169.2\ W}{m^2} \times \frac{m^2}{10000\ cm^2} \times \frac{2\ cm \times 6\ cm}{cell} = \frac{0.203\ W}{cell}$$

The number of cells required is based on the EOL orbit average array power requirement as determined by the energy balance calculations (including geometry effects). The energy balance equation is discussed in another section of this paper. The cell quantity calculation is a straightforward division of the load requirement by the individual cell output. For example, if the array power required is 500 W, then:

$$\frac{500\ W}{0.203\ W/cell} = 2463\ cells$$

The number of cells per string depends on the EOL voltage output at maximum power (Vmp). This voltage is reduced at EOL according to the following factors:

| | |
|---|---|
| Mismatch | 0.98 |
| Measurement | 0.99 |
| Radiation | 0.99 |
| Combined | 0.96 |

$$\begin{aligned} Vmp\ (EOL) &= Vmp\ (BOL) \times \text{combined factor} \\ &= 0.87\ V/cell \times 0.96 \\ &= 0.836\ V/cell \end{aligned}$$

This value is based on cell data taken at 28°C, and must be corrected for the predicted operating temperature of the array in space. Assuming a typical temperature of 70°C, the cell voltage is found via:

$$0.836\text{ V} - (70°\text{C} - 28°\text{C}) \times 2.1\text{ mV/°C}$$
$$= 0.836\text{ V} - 0.088\text{ V}$$
$$= 0.748\text{ V per cell}$$

The EOL array terminal voltage required is found by increasing the required load bus voltage of 32 volts according to:

32.0 V required at the output
+1.5 V array harness and diode drop
<u>+1.0 V margin</u>
34.5 V required at the cell string terminals

Therefore the number of series cells required is:

$$34.5\text{ V} / 0.748\text{ V/cell} = 46.1\text{ cells}$$

This must be rounded up to an integral 47 series cells per string. To determine the number of parallel strings needed, divide the number of cells in series into the total number of cells required:

$$2463 / 47 = 52.4\text{ strings}$$

Round this up to a whole number, 53. If the spacecraft has two solar panels, is would be desirable to have the number of strings divisible by two. Adding one additional string provides some margin and gives the actual number of cells used:

$$53\text{ strings} \times 47\text{ series cells} = 2491\text{ cells}$$

The total cell area is then:

$$2491\text{ cells} \times 2\text{ cm} \times 6\text{ cm} = 2.99\text{ m}^2$$

The total required array area reintroduces the packing factor of 0.90:

$$2.99\text{ m}^2 / 0.9 = 3.32\text{ m}^2$$

The ratio between the EOL array output and the load requirement is a good measure of the efficiency of the overall solar array design. This ratio is a strong function of geometry and orbit parameters as previously discussed. One of the fundamental drivers of this value is the actual eclipse ratio (time in the Earth's shadow versus time in sunlight), which is a function of altitude. Because of altitude effects on radiation degradation, a comparison can be made based on solar cell types, which have different radiation susceptibility.

Design estimates for the ratio of EOL solar array power to load requirement for various orbit altitudes are shown in Table 2–12. The table also shows that the required BOL ratios are essentially the same for Silicon arrays as for GaAs arrays. This is because the degradation is primarily due to UV and thermal cycling, with only a small amount of radiation exposure at these low altitudes. As a result, the radiation resistance capabilities of GaAs compared to Si arrays is of little advantage. However, the efficiency advantage of GaAs is significant because it results in an array of proportionately lower mass and size, factors of importance to smallsats.

While the radiation tolerance of GaAs cells offers little advantage for most smallsats, its high efficiency results in smaller, lighter, and easier-to-package solar arrays.

Table 2–12
Array Design Estimates

| Altitude (km) | Max. Eclipse (min.) | Min. Sun (min.) | Ratio Sun/ Eclipse | EOL Ratio* | BOL Ratio* Si (2-Yr.) | BOL Ratio* GaAs (2-Yr.) |
|---|---|---|---|---|---|---|
| 400 | 36.1 | 56.5 | 1.57 | 2.04 | 2.16 | 2.16 |
| 600 | 35.5 | 61.2 | 1.72 | 1.97 | 2.08 | 2.08 |
| 800 | 35.1 | 65.8 | 1.87 | 1.91 | 2.03 | 2.03 |
| 1000 | 34.9 | 71.2 | 2.04 | 1.86 | 2.08 | 2.07 |

**Solar array power/load requirement (assumes oriented array).*

2.6.4.6 Performance Analyses—Once the array size is determined, its predicted performance can be analyzed over the operational conditions of the mission. The effects of changes in Beta angle, load profiles, and array output can be modeled with relatively simple computer programs. These studies can verify the occurrence of worst case conditions and identify periods of high and low performance (as influenced by Beta angle and array degradation). "Campaigns," in which the spacecraft payload operates at a high-duty cycle for one orbit out of several, can also be evaluated using this technique.

Figure 2–21 shows the energy budget simulation for a small spacecraft (TIMED-L) in a 400 km, 49° orbit over a 2-year mission. A 6.57m$^2$ Beta-tracking solar array was used. For this mission, the solar array size was fixed by project-imposed limitations, rather than by load requirements, and maximum load capability was not always possible. Therefore, a performance analysis was done to determine the available power versus time, based on a particular array size. Because of the variation in Beta angle, periods when the array output is adequate to meet energy balance, and other periods exist when the array output is not sufficient. During these deficit periods, the instrument payload can be operated at less than 100 percent duty cycle. Figure 2–22 shows the available duty cycle during the mission. Two curves are plotted, with one allotting all available power to instrument operation and the other withholding 25 percent growth margin (resulting in lower duty cycle capability).

Another type of analysis calculates the heat rejection capabilities of the thermal subsystem as a function of Beta angle. This Beta dependence makes it convenient to combine the thermal analysis with the computer program that predicts the power subsystem performance. Like the power analysis, the thermal analysis compares capabilities versus requirements. The heat rejection limitations of the spacecraft radiators are compared to the heat generated by the spacecraft equipment and payload.

Heat rejection capability is calculated based on an assumed bulk spacecraft temperature and set of curves provided by the thermal engineer. The basic heat rejection requirements are based on rejecting all of the electrical power used on the spacecraft as heat. In addition to the basic electrical loads, two sources of waste heat exist. The first is the shunt transistors, which generate heat when they switch excess array power to the shunt resistors located on the solar panels. The second source is the batteries, which generate heat when they are discharging. These waste heat values are determined by the computer model. These results are plotted in Figure 2–23 for the same 400 km, 49° inclination spacecraft (TIMED) as the previous examples. The plot shows the requirements versus capabilities, with two curves for the heat rejection requirement. One curve

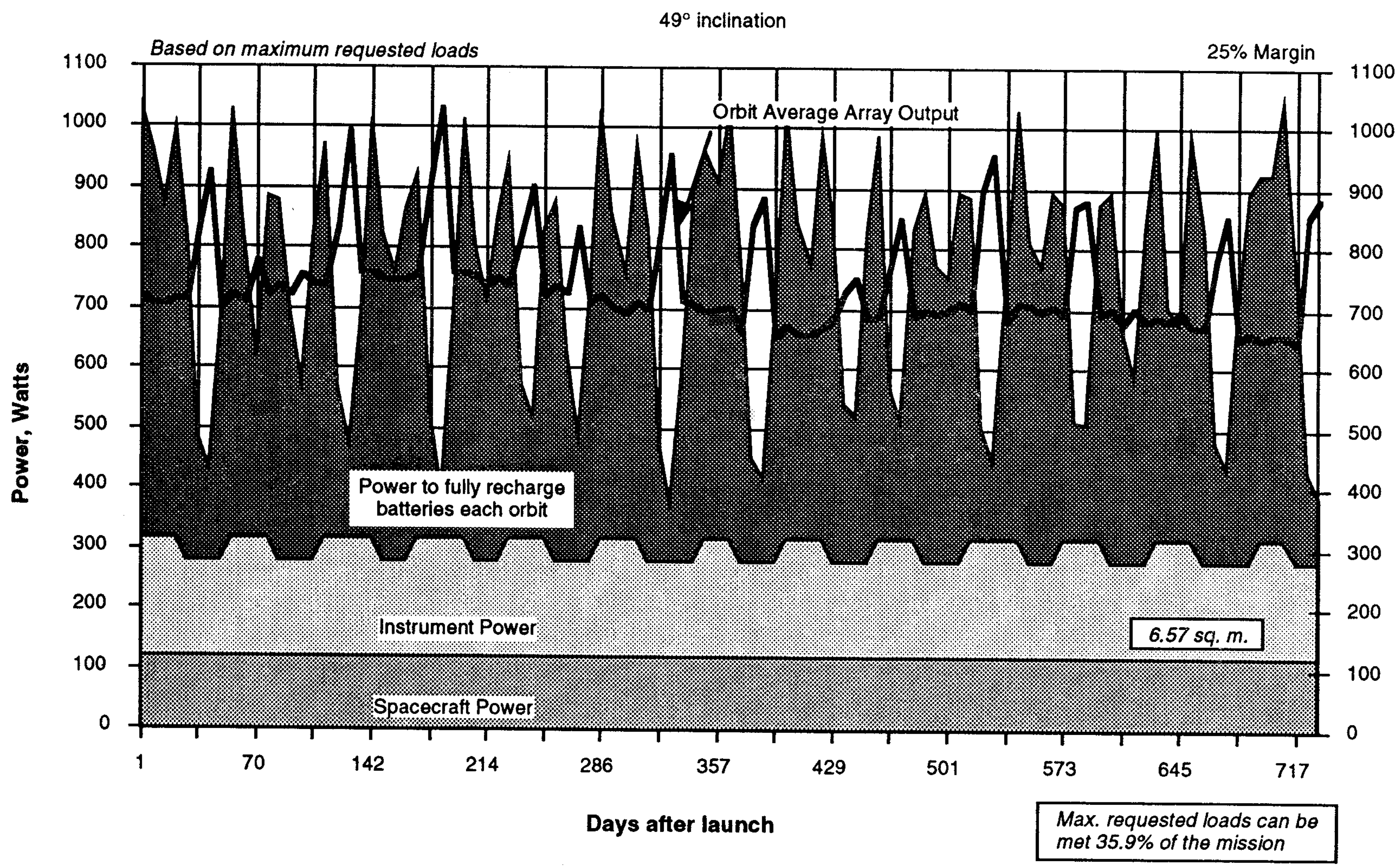



Figure 2-21. TIMED L Power Requiremetns Versus Capability

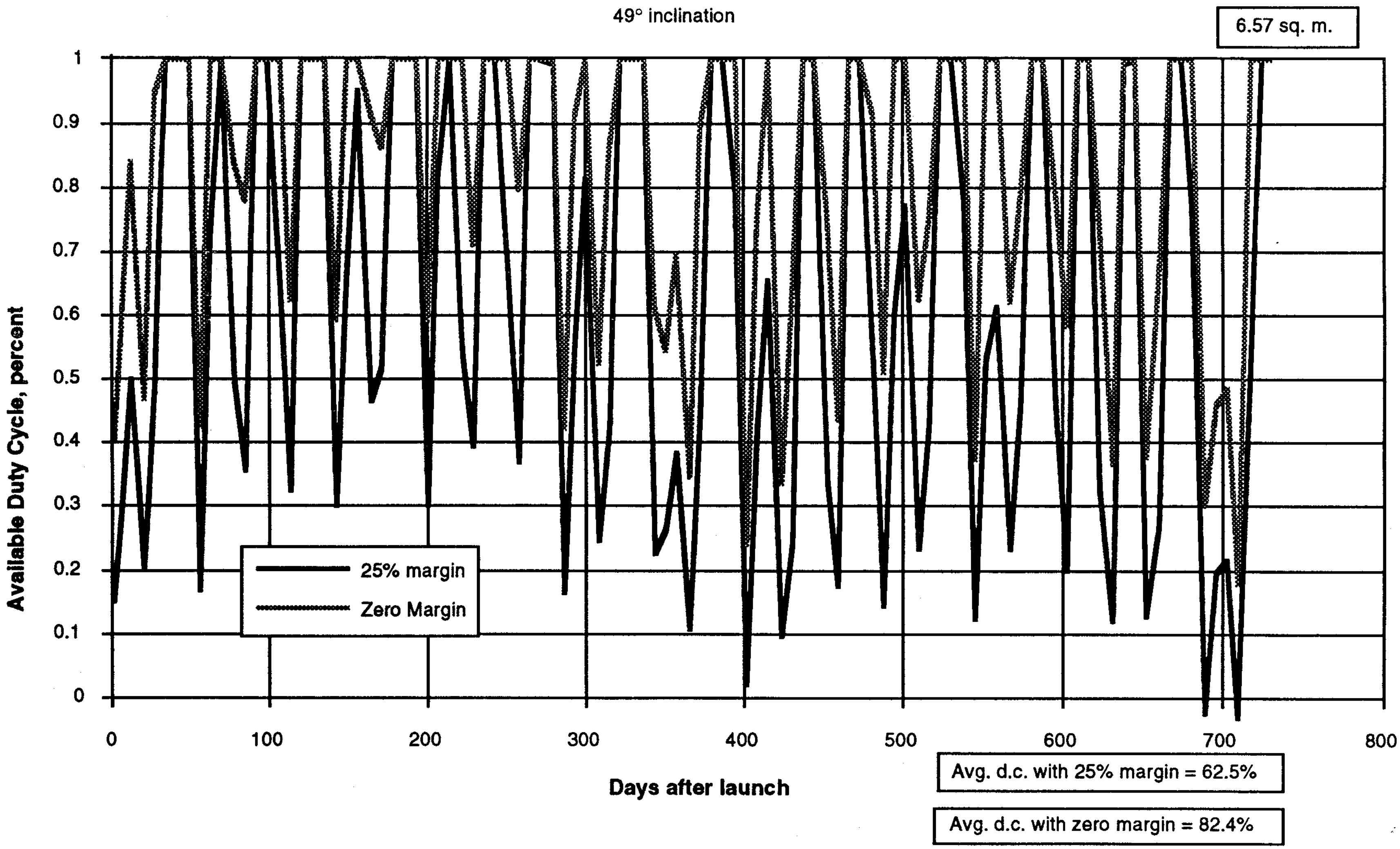



Figure 2-22. TIMED L Instrument Operating Capability

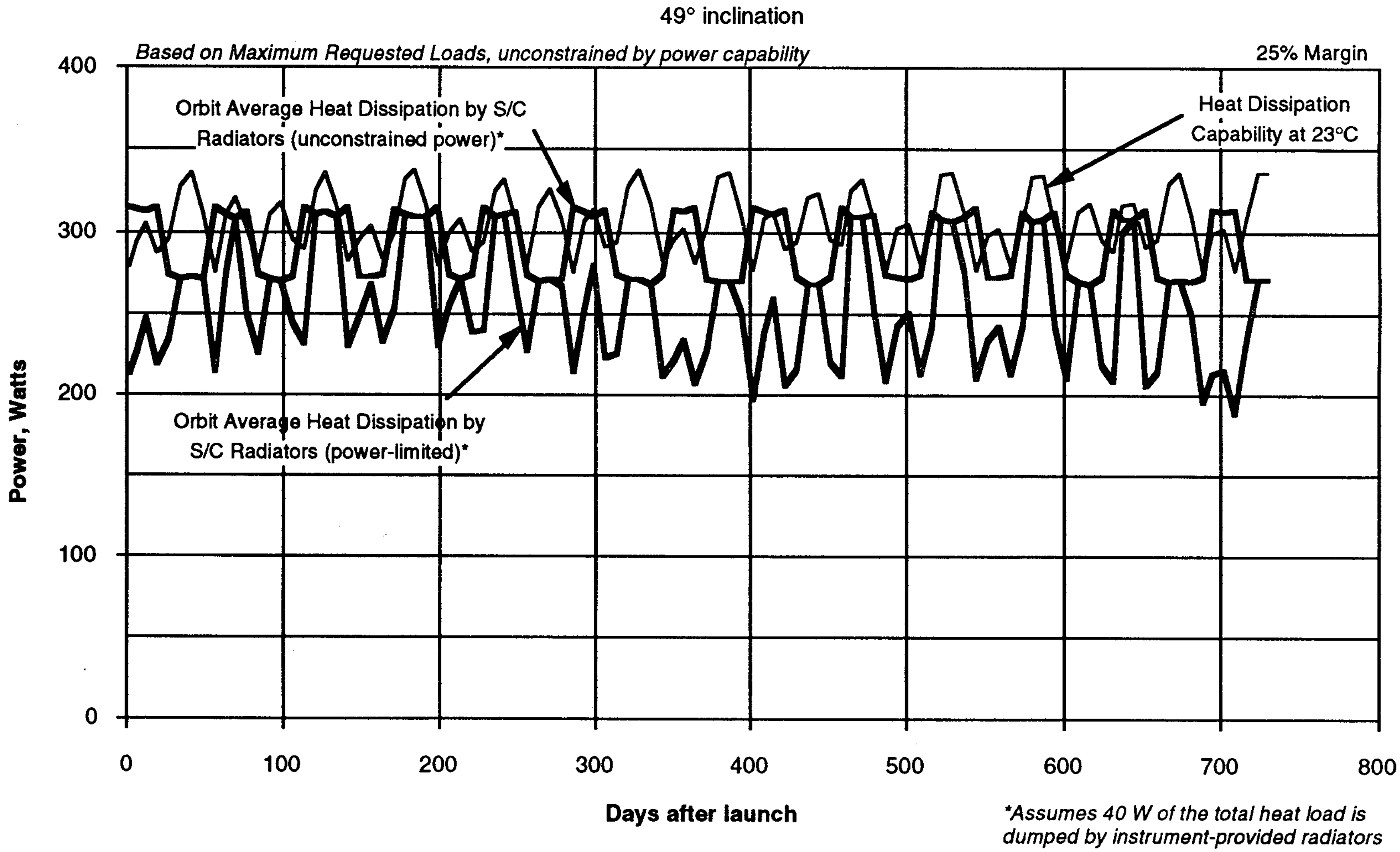



Figure 2-23. TIMED L Heat Dissipation Analysis

is for the maximum requested loads regardless of whether enough array power is available to satisfy those loads. The other, more meaningful curve, is the heat dissipation requirement using loads based on actual capabilities. The heat dissipation requirements are represented by the two heavy lines, while the spacecraft's heat rejection capability at 23°C is shown by the lighter line.

These types of analyses enable trade studies to be made for various array designs, including the overall configuration and deployment geometries. This is particularly useful in smallsat power subsystem design, as array size and mass are highly constrained.

**Solar array configuration and area, precious commodities on smallsats, can be optimized by evaluating the power subsystem's predicted performance over the duration of the mission, thus identifying more realistic worst-case conditions on which to set sizing requirements.**

## 2.7 BATTERIES

### 2.7.1 General

Battery performance, reliability, and length of life are all important factors in selecting batteries. In analyzing batteries for small satellites, the objective is to find light-weight, low-cost batteries that have an acceptable depth of discharge (DOD). The number of cycles, battery operating temperature, and type and capacity of battery are analyzed to determine the DOD for that battery. Then battery lifetime can be matched to mission lifetime. A simplified flow chart for the battery sizing process is shown in Figure 2–24. A typical nickel-cadmium (NiCd) battery configuration is shown in Figure 2-25.

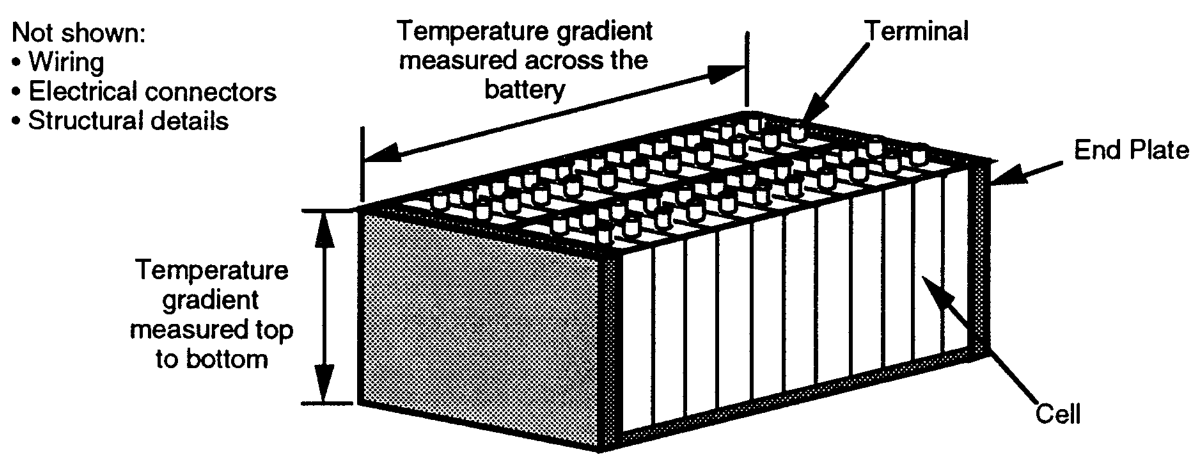



Figure 2–25. Features of a Typical 22-Cell Nickel Cadmium Battery

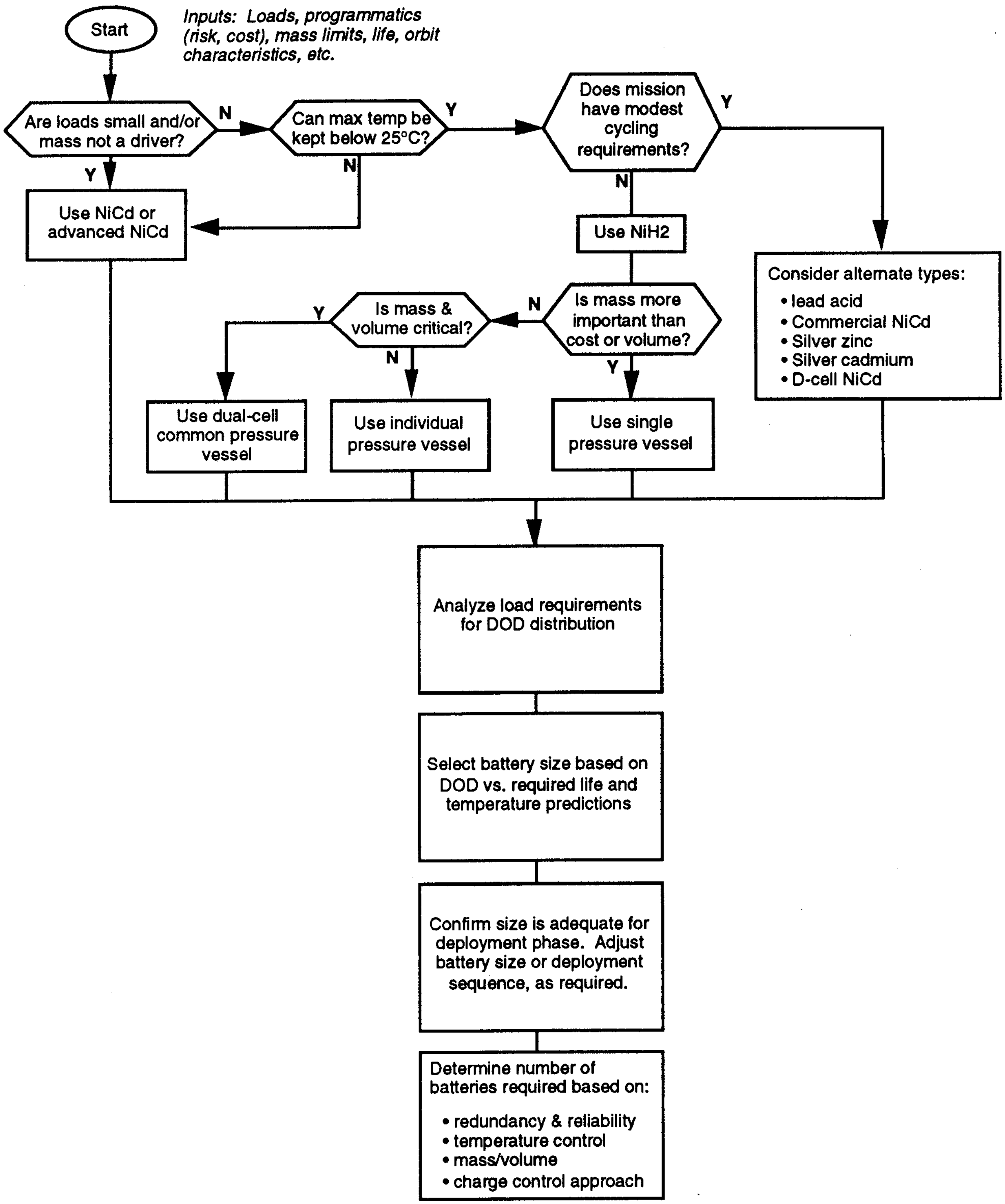



Figure 2–24. Battery Selection and Sizing Flowchart

2.7.2 Battery Sizing

2.7.2.1 Normal Operations—Battery performance and reliability depend on the expected number of charge/discharge cycles, the battery operating temperature, and the DOD. These factors are depicted for NiCd batteries in Figure 2–26. Low cycle number, low temperature, and shallow DOD are the conditions to be desired. There is presently insufficient statistical data for advanced NiCd or $N_iH_2$ batteries to estimate lifetime curves, but they would show similar trends. Other factors, such as pre-launch environments, wet life, pre-launch cycling, and reconditioning also affect battery operations. Because of these factors, the variability of cell manufacturing and especially the problems of battery failures in flight spacecraft, battery designs are usually very conservative compared to those shown in the figure.

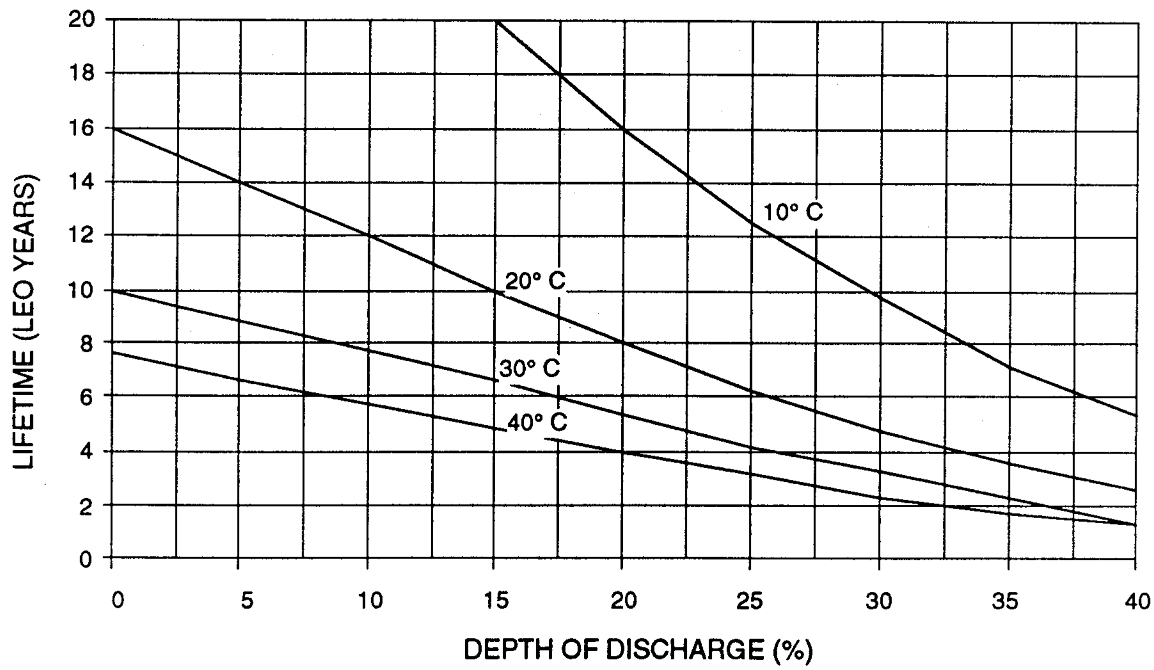



Figure 2–26. NiCd Battery Lifetime

The number of cycles is determined from the mission lifetime and the orbit. Nominally, 6000, 35-minute discharge cycles per year can be anticipated for a spacecraft in a low inclination, circular LEO. However, these numbers will vary considerably with inclination and orbit synchronization with the Sun. For example, in the higher inclination orbits, there are significant periods of eclipse-free (full sunlight) operation. Similarly for elliptical orbits, eclipse time and numbers vary significantly with inclination and time of year. For example, an apogee eclipse would be considerably longer than a perigee eclipse in the same orbit and would result in a significantly higher percent of battery discharge. However, the number of cycles can be calculated. DOD trend curves and histograms of DOD are useful ways to display this data. Examples of such plots from the TIMED study are provided in Figures 2–28 and 2–28.

The battery operating temperature is determined by the design of the spacecraft. Such factors as location, heat sinking, use of heat pipes or other thermal control, etc., affect the battery temperature. It is desirable to keep battery temperatures between 0°C and 10°C. This often is difficult to accomplish. Since control methods usually do not bring batteries to as low a temperature as desired, the requirement then becomes to keep the battery at as low a temperature as possible. Since the number of cycles is determined by mission requirements, the DOD must

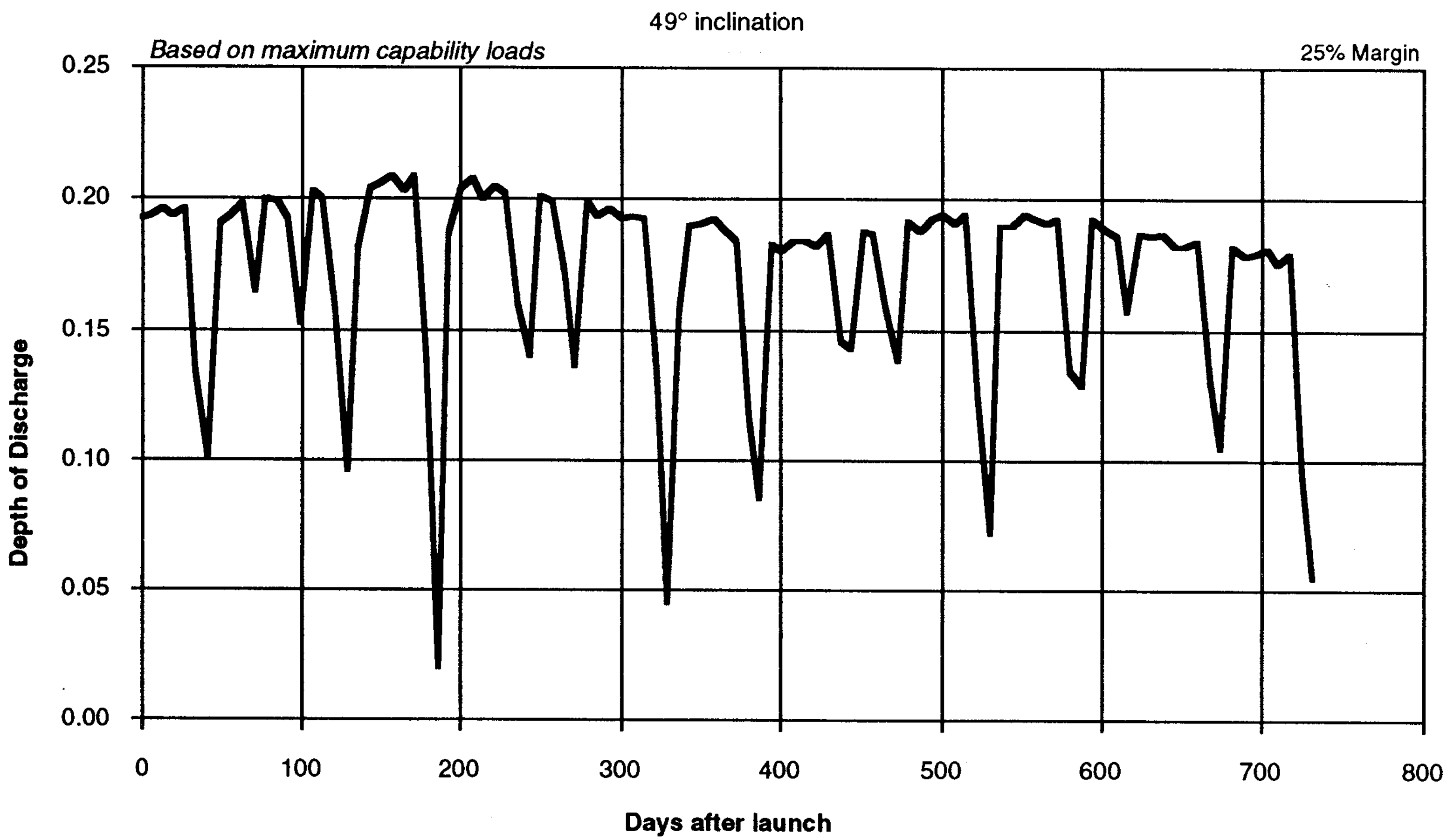



Figure 2-27. TIMED L Battery DOD

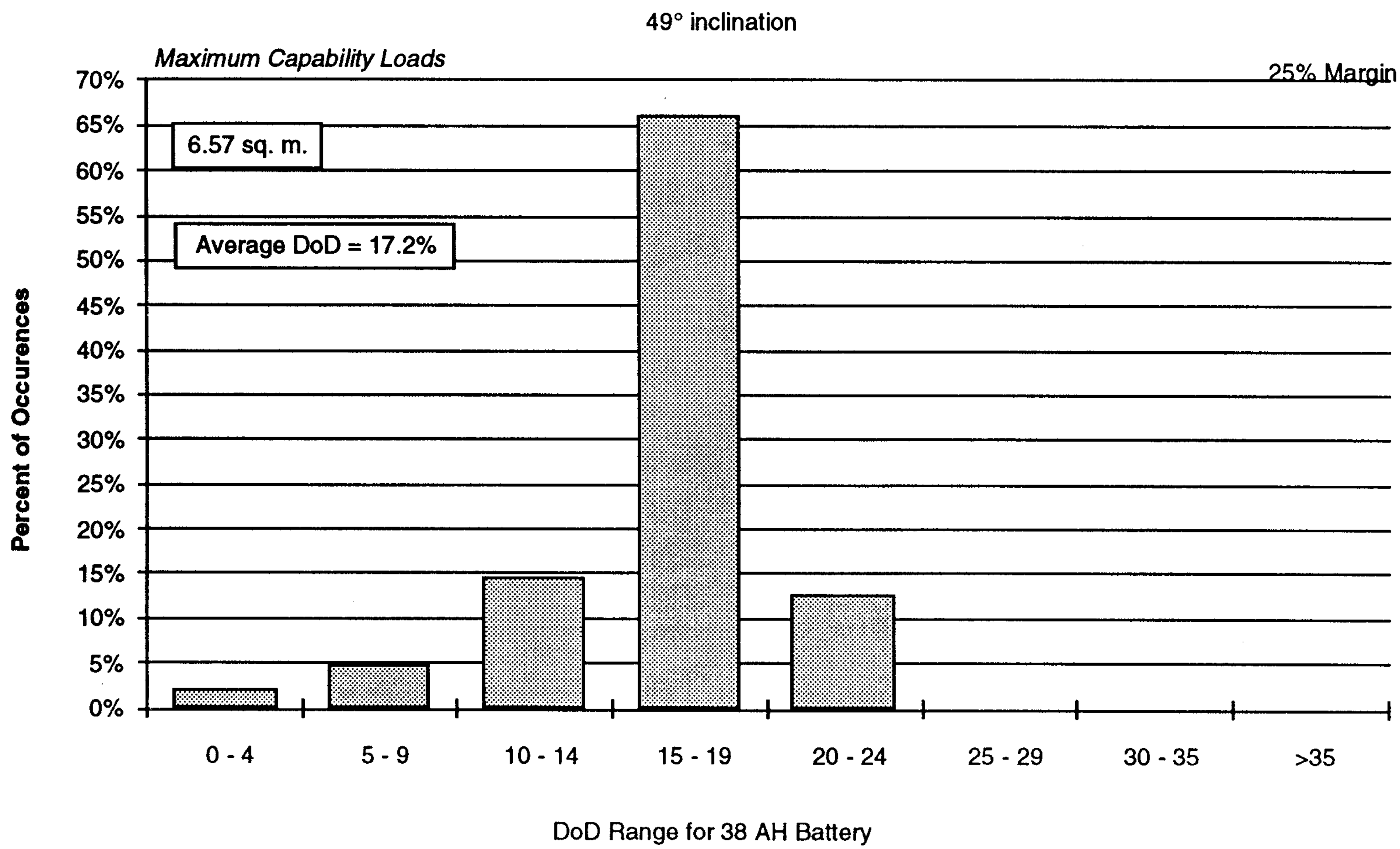



Figure 2-28. TIMED L Frequency distribution of Battery DOD

be reduced to compensate for battery temperatures that are higher than optimum and to maintain reliability. However, a lower DOD may result in the need for a larger capacity battery, which increases the bus mass.

Multiple batteries must be maintained within 5°C, or less, of each other as determined by the batteries' voltage response at different temperatures. The acceptable maximum thermal gradient from the bottom of a battery to the top is 5°C (Reference 21). The end-to-end gradient across the top of a battery cannot exceed 3°C (see Figure 2-26) (Reference 22). Ideally, heat pipes can be used to equalize battery temperatures and ensure uniformity of battery response and equal current sharing. They are often necessary on small satellites that have more than one battery since there isn't room to mount two heavy batteries close together.

> Heat pipes can be used to equalize the temperature of two batteries when they cannot be located in thermally identical environments, as is usually the case with small satellites.

Battery discharge is determined by the load profile on the battery during the orbit. From lifetime performance curves (for the type of battery used), such as shown in Figure 2–28 for standard NiCd batteries, and the predicted battery temperature; the maximum DOD is determined as a percent of the battery nameplate capacity. Once this figure is known, the required capacity can be calculated. However, it might not be a standard, off-the-shelf capacity. This discrepancy may require a trade off between a new design (requiring qualification) and a design with a different capacity. A higher capacity battery would weigh more and take up more space. On the other hand a lower capacity battery would have increased DOD and risk. An alternative option is to use two or more smaller batteries, which grants some degree of redundancy but increases the complexity of the system.

Maximum DOD design requirements for various loads during eclipse are shown for various battery capacities in Table 2-13. The primary purpose of the table is to illustrate the dependence of these requirements on the load and the battery capacity requirements for various allowable DODs. It is readily apparent that increasing the allowable DOD permits the use of lower capacity (and hence cheaper, lighter, smaller) batteries. In addition, it can be seen that, although eclipse durations are decreased as altitude increases, the decrease is not sufficient to make a significant difference in DOD.

Table 2–13
Battery Maximum DOD Designs

| Altitude (km) | Total Load (W) | *Battery Disch. (Ah) | % DOD (10 Ah) | % DOD (20 Ah) | % DOD (30 Ah) | % DOD (40 Ah) | % DOD (50 Ah) |
|---|---|---|---|---|---|---|---|
| 400 | 200 | 4.30 | 43.0 | 21.5 | 14.3 | 10.8 | 8.6 |
| | 400 | 8.60 | 86.0 | 43.0 | 28.7 | 21.5 | 17.2 |
| | 600 | 12.89 | XXX | 64.5 | 43.0 | 32.2 | 25.8 |
| | 800 | 17.19 | XXX | 85.6 | 57.3 | 43.0 | 34.4 |
| | 1000 | 21.49 | XXX | XXX | 71.6 | 53.7 | 43.0 |
| | | | | | | | |
| 1000 | 200 | 4.15 | 41.5 | 20.8 | 13.8 | 10.4 | 8.3 |
| | 600 | 12.46 | XXX | 62.3 | 41.5 | 31.2 | 24.9 |
| | 1000 | 20.77 | XXX | XXX | 69.2 | 51.9 | 41.5 |

**Assumes 28 volts.*

2.7.2.2 Deployment Mode Operations—In addition to normal operations, the battery must also be sized to provide energy for the initial orbits before the solar arrays are deployed and oriented

toward the Sun. If the battery is completely discharged before the arrays are out and the spacecraft properly oriented, the spacecraft will be lost.

This problem is a significant one for small satellites that do not have large batteries and must accommodate large initial attitude rates from the launch vehicle. While small satellites generally have proportionally lower bus loads than larger spacecraft, the battery capacity is typically reduced by a larger ratio. For example, a large satellite might have 100 Ah of storage with a 200 W bus load. The battery can maintain this load for 10.5 hours at which time it reaches 75 percent DOD. The typical smallsat has a battery of 9 Ah and a bus load of 100 W. The battery can only maintain this load for 1.9 hours to a 75 percent DOD. Figure 2–29 shows a predicted load profile, including battery state-of-charge, of the first two orbits of the Total Ozone Mapping Spectrometer-Earth Probe (TOMS-EP) spacecraft. This is an example of a pre-flight computer simulation done to verify that the battery has sufficient capacity to survive various early orbit contingencies.

> The lower battery capacity typical of smallsats cannot accommodate extended deployment mode operations prior to solar array deployment and attitude stabilization.

Typically, the batteries are not sized for the one-time deployment phase. Operations planning minimizes the loads until the arrays are deployed allowing only a single occurrence of high DOD. The solar arrays can also be stowed with cells exposed before deployment, so some energy will be generated for support the load.

2.7.2.3 Choosing to Use One or Two Batteries—Because batteries are large and heavy, using two batteries on a small satellite is difficult to justify. Occasionally, two batteries are necessary because the required battery size is not available as an off-the-shelf item (see Table 2–15). The failure of the battery in single battery design will cause loss of the mission, although complete battery failures are rare. In a two-battery system, loss of one battery will reduce the capability, but some capability can be maintained by the remaining battery operating at a higher DOD for the remainder of the mission. Mission lifetime and payload operations will generally have to be reduced.

Due to recent problems with spacecraft batteries, the trend is to overdesign using excessive capacity and complex control systems when two batteries are used. Using multiple batteries can lead to the development of complex charging circuitry for individually charging each battery. This requires adding a capability for additional cell monitoring and a capability for reconditioning. These measures minimize program risk but increase the cost, complexity, and mass.

Another factor to consider in selecting a two-battery design is the requirement that multiple batteries must be kept at nearly the same temperature. The spacecraft's thermal control subsystem design must accommodate this requirement.

A common approach is to use an extra cell in each battery. A nominal 28 volt, NiCd battery requires 21 cells. Most failures are short-circuits on the cell level, so this is a simple option to improve reliability.

### 2.7.3 Battery Performance

2.7.3.1 Battery Selection—Currently, only two battery chemistries are available for use on moderate life (1-3 years) small satellites in LEO, on which 5000 to 20,000 charge/discharge cycles are required. They are NiCd and $NiH_2$. Each of these has several design variations to

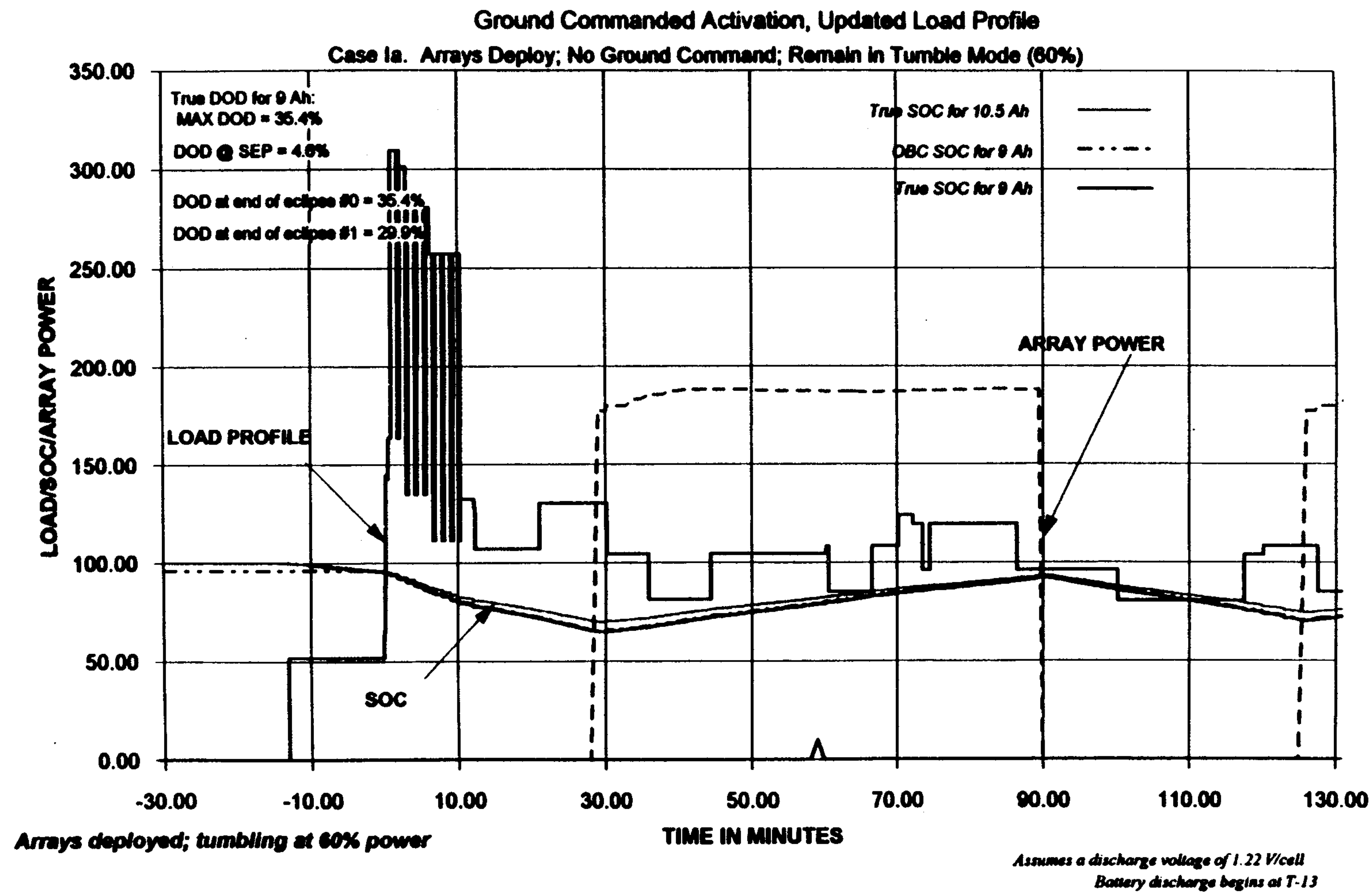



Figure 2–29. Deployment Phase Simulation

consider. For small satellites in LEO, cycling capability is the overriding factor in battery selection. Although NiCd batteries are the most common type on LEO spacecraft, $NiH_2$ batteries are starting to be used in LEO applications.

Because the battery is critical to the survival of the spacecraft, battery reliability is of prime importance. Secondary trade studies that address performance, mass, volume, and cost must also be considered. Specific performance parameters that compare different battery types are presented in Paragraph 2.7.3.4 of this report.

The choice between NiCd and $NiH_2$ batteries becomes one primarily of the importance of mass versus volume. If mass is significant, then $NiH_2$ is selected to take advantage of the higher (by a factor of 1.7) specific energy (Wh/kg). If volume is more important, then NiCd is selected, taking advantage of the higher (by a factor of 1.4) energy density (Wh/l). Other factors of significance, such as operating temperature and allowable DOD for the particular cycle life under consideration must also be considered in the overall trade.

Unusual requirements, such as stringent non-magnetic requirements, or unusual cases, such as minimal cycling conditions, could lead to the selection of a battery with an alternative cell chemistry, for example a AgCd battery to eliminate nickel which is paramagnetic.

2.7.3.2 Current NiCd Battery Technology—Of any battery type used in space, NiCd batteries have the most extensive use. Through the years slow, but steady, progress has been made in reducing mass and volume, i.e., increasing specific energy (Wh/kg) and energy density (Wh/l). However, in spite of a tremendous amount of cycle-life testing under a wide variety of ambient conditions, uncertainty in reliability continues as an enduring problem. One reason is that the designs have been subjected to many changes. Some of these are major changes, such as the forced changes in separator materials (due to discontinued production, etc.).

The extensive testing has defined expected performance and requirements, including design, manufacture, testing, and handling for standard cells and batteries. However, cells and batteries cannot be consistently manufactured to perform reliably. As a result, extensive testing and matching is necessary to build confidence in the cells, and extensive controls are needed in space to bound the external influences on battery performance. It is precisely this lack of certainty that leads to design requirements for in-flight monitoring, even at the cell level, for voltage and/or temperature divergence; for state of charge (ampere-hour integration) monitoring in addition to partially automated (V/T) charging; and for on-orbit reconditioning of the batteries. It is also this lack of confidence that drives the continuous search for "better" alternatives, either in battery design or battery type.

Alternative NiCd designs that have been flown in recent years are called "advanced" cells because they are designed to perform more reliably under more adverse conditions of DOD, temperature, or cycle life. One such cell is the Hughes Super NiCd™, which incorporates a zirconium oxide separator and a 31 percent KOH electrolyte with a proprietary additive. Its electrode impregnation is done by electrochemical deposition. Laboratory DOD/temperature/cycle-life data for these types of cell are sparse. As a result, the operating levels acceptable for conventional NiCd cells are used as the guideline. The advantage becomes one of a small gain in energy density and specific energy, but at higher cost. Another advanced cell type is the Eagle-Picher Magnum™ cell, which is nearly identical to the Hughes Super-NiCd, but uses a polypropylene separator. It is proposed that this cell has the performance characteristics of the Hughes cell, but costs less because of the change in separator. Testing of this cell is yet to be accomplished, and since the cell is yet to be flown, there are no flight data.

In summary, it is the special care in the preparation, handling, and operation of these cells that is of prime importance, rather than the small differences in design.

2.7.3.3 Current $NiH_2$ Battery Technology—The advantage of $NiH_2$ batteries is a high specific energy compared to NiCd batteries. It is about 30 percent higher in the 20 to 25 Ah capacity range and it increases as capacity increases. This increase in energy results in significant mass reduction. The drawback is that the energy density of $NiH_2$ batteries is much less than that (as low as 50 percent) of NiCd, resulting in a much larger volume requirement. These characteristics are inherent to the cells because of the electrodes and the external configuration of the $NiH_2$ cells which must, of necessity, be pressure vessels (domed cylinders). This reduced energy density results in a lighter cell because the hydrogen electrode replaces the cadmium electrode of the NiCd cells. A larger volume requirement results because of the lost space in the domes and in the packaging of cylinders compared to the prismatic NiCd cells. $NiH_2$ batteries also require lower operating temperatures than NiCd batteries, placing more stringent requirements on the thermal subsystem.

The volume problem can be partially overcome by packaging more than one cell in the same pressure vessel. Thus, there are three major categories of $NiH_2$ cells/batteries to be considered: The individual pressure vessel (IPV) cells, common pressure vessel (CPV) cells, and single pressure vessel (SPV) batteries. The CPV most frequently contains two cells, while the SPV contains the full number of cells necessary for the battery; i.e., 22 cells for a 28 volt battery.

The negative aspects for the CPV and SPV batteries are threefold. First, the energy density problem is not completely solved. Second, monitoring, isolation, thermal control, and other desirable cell level features are difficult. Third, a negligible amount of LEO orbit test and flight performance information is available.

The applications of $NiH_2$ batteries have been increasing continually since they were first flown on Intelsat V in 1980. Most applications have been in geosynchronous orbit where eclipse durations are long (up to 79 minutes), but infrequent (two seasons per year of only 44 cycles each). Thus, the batteries undergo deep discharge cycling, but cycling is limited to less than 1000 cycles, even for a spacecraft with a 10-year lifetime. These flight results do not apply to LEO spacecraft. Some $NiH_2$ batteries have been flown in LEO, but these batteries are generally large, high capacity batteries (80 Ah on HST, for example) not appropriate for a smallsat. However, a few smaller $NiH_2$ batteries have flown in LEO, and an SPV design was used successfully on Clementine, and the number of proposed applications is increasing.

> Small nickel hydrogen batteries have been developed specifically for small satellite applications and have been used successfully.

As the number of applications increase, and test and flight data become more extensive, more confidence will be gained in the reliability of the designs. At the present time, there is a justified reluctance to use these cells to their fullest apparent capability. Again the reliability under the necessary conditions of performance is the driving factor in the selection.

2.7.3.4 Relative Performance—Typically, NiCd battery capacity has been in the 20 to 50 Ah range, but smaller sizes (5 and 9Ah, for example) are becoming available to meet the demand of smallsat designers (see Table 2–14). Operational temperature limits are set to 10°C ±10°C, and DOD is limited to accommodate lifetime requirements.

Table 2–14
Selected Battery Sizes Based on Current Production Cells

*Advanced NiCd:*

| Capacity (AH) | Size (LxWxH) (cm) | Mass (kg) | Cell Manufacturer | Comments |
|---|---|---|---|---|
| 5 | 33x16x15 | 6.2 | Hughes | |
| 9 | 30x18x11 | 12 | Hughes | Used on SAMPEX |
| 19 | 41x22x28 | 20.6 | Hughes/EP | Estimate for TIMED |
| 21 | 47x24x19 | 23.5 | Hughes | |

Allowable DOD is based on the results of life testing performed by various Government and industry facilities. The relative status of battery cell testing is illustrated in Table 2–15. There is a wealth of cycle-life testing information available for conventional NiCd battery cells. As a result it is possible to define general cycle-life/DOD/temperature limits for NiCd battery designs, but not for others. However, because factors such as manufacturer, vintage, and proprietary design information cause considerable variability, it is generally necessary to use quite conservative designs, even for NiCd cells. This approach can lead to excessive battery cost, mass, and volume. The need for a broad database to reduce conservatism becomes obvious when it is noted that there is only limited elevated temperature testing of new cell types, which are hopefully capable of higher DOD.

Table 2–15
Battery Cell Cycle-Life Test Results

| Cell Type | Size (Ah) | Voltage (Volts/Cell) | Test Results LEO Life/DOD/T (Cucles/ Percent/°C) |
|---|---|---|---|
| NiCd | 20 | 1.22 | (1) |
| A-NiCd | 21 | 1.22 | 15,000/40/20 |
| $NiH_2$-IPV | 50 | 1.25 | 29,000/40/20 |
| $NiH_2$-CPV-(2 CELLS) | 10 | 2.50 | 9,100/40/10 |
| $NiH_2$-SPV-(26 CELLS)* | 24 | 1.23 | 7,000/44/? |
| NiMH | 22 | 1.22 | 4,236/50/25 |

**32 Volt Battery*
*(1) Extensive cycling test data are available.*

Smaller sized $NiH_2$ batteries are being produced by cell manufacturers to meet smallsat demands. These new offerings include several CPV and SPV designs. SeaStar uses a 6 Ah design with two cells per pressure vessel. Clementine used a 28 volt battery in a single pressure vessel design. A sampling of currently manufactured sizes is shown in Table 2–16.

Table 2–16
Sample of Currently Manufactured Battery Sizes

*$NiH_2$ (Typically 1.25 V/cell)*

| Capacity (Ah) | Size (L x W x H) (cm) | Mass (kg) | Cell Mfr. | Type | Comments |
|---|---|---|---|---|---|
| 6 | | 12.5 | E. P. | CPV | 10 cells, for SeaStar |
| 10 | | 2.8 | J. C. | SPV | 10 cells |
| 15 | | 7.86 | J. C. | SPV | 22 cells |
| 19 | 54x30x32 | 15.8 | J. C. | SPV | Estimate for TIMED |
| 25 | | 18.6 | E. P. | SPV | 22 cells |
| 35 | 16x12x12 | 22 | E. P. | IPV | GeoSat |
| 40 | | 23.7 | J. C. | SPV | 22 cells |

*J.C. = Johnson Controls*
*E.P. = Eagle Picher*

The performance characteristics of specific energy and energy density for various battery cells are shown in Table 2–17. Comparison of the usable portions of these factors shows the battery cell potential, which depends on the possible DOD in the battery design. The table illustrates improvements that are now beginning to be used in space. It also shows the potential of future developments. Significant gains in both usable specific energy and in usable energy density can come with higher allowable DODs. This factor is significant for smallsats.

More efficient battery use can be achieved by simply increasing the maximum allowable depth-of-discharge.

Table 2–17
Battery Cell Energy Data

| Cell Type | Size (Ah) | Total Specific Energy (Wh/kg) | Total Energy Density (Wh/l) | Conservative DOD LIMITS* (% at 20°C) | Usable Spec. Energy (Wh/kg) | Usable Energy Density (Wh/l) |
|---|---|---|---|---|---|---|
| NiCd | 20 | 25 | 86 | 20 | 5.0 | 17.2 |
| S-NiCd | 19 | 29 | 87 | 25 | 5.8 | 21.8 |
| $NiH_2$-IPV | 20 | 42 | 61 | 40 | 16.8 | 24.4 |
| $NiH_2$-CPV-(2 ) | 10 | 47 | 61 | 40 | 18.8 | 24.4 |
| $NiH_2$-SPV-(26) | 20 | 43 | N/A | 40 | 17.2 | N/A |
| NiMH | 24 | 44 | 151 | 50 | 22.0 | 75.5 |

**Based on 18,000 cycles (3-year lifetime).*

2.7.3.5 Future Performance—Smallsats have been on the vanguard in the use of new battery technology. One of the early users of an advanced NiCd battery was the SAMPEX Project. Clementine used a single pressure vessel $NiH_2$ design. Such designs will continue to find applications in smallsats because of their excellent performance and low mass. On the near horizon, is the use of a low-capacity (around 10 Ah) common pressure vessel with two cells specifically designed for the lightsat market. As with solar cell and panel technology, battery costs should decrease as more designers use this technology and these new batteries become part of standard designs.

Higher battery DOD should also be achievable when more reliable separators are incorporated and electrolyte formulations are improved. Once NiMH cells mature, they should provide improved energy density and battery life. This could be of significant value to smallsats.

Improvements to current cell technologies and the introduction of new battery types, such as nickel metal hydride, should offer significant improvement in battery energy density.

## 2.8 EPS REQUIREMENTS

### 2.8.1 Current Requirements

The primary design driver for the EPS is the total electrical load that must be satisfied. This discussion is limited to electrical load requirements measured as an orbit average value.

Considering recent vintage spacecraft designed for LEO operations, the power requirements range up to about 300 W. Approximately 50 percent of the power is for the payload instruments and 50 percent for the spacecraft subsystems (bus). This ratio is improving (from the payload point of view) as subsystems become more efficient and payloads demand more resources.

For small satellites, the payload-to-bus ratio is not as good. On small satellites, the total spacecraft bus has minimum load requirement of about 100 watts. This requirement is still rather large perhaps because of an attempt to use components originally designed for large satellites on a smaller platform. But as newer satellites take advantage of subsystems designed specifically for smallsats, the subsystem loads should go down, and the ratio should improve. A more enlightened design philosophy (driven by spacecraft capabilities rather than payload requirements) also improves this ratio.

The relatively fixed load requirement of the spacecraft bus leaves proportionally less power available for payloads on small satellites with a low total load power capability.

Table 2-18 lists the load requirements for recent small satellites, with the bus and payload requirements listed separately if that data was available. The mass fraction is defined as the ratio of the EPS mass to the total spacecraft bus (no payload or instrument) mass. This data is not available for the list of newer programs.

For the older programs listed in the top table, the average power requirements for payloads runs about 60 W, while the spacecraft bus typically requires 120 W. For the newer programs, the load capability is a larger fraction of the total power output.

### 2.8.2 Future Requirements

Despite the trend towards smaller spacecraft, performance requirements are expected to continue to increase. While actual load requirements will go up, the amount of usable power that can be generated is limited for a small spacecraft. The trend will be to get more from less, both in performance per unit mass and performance per unit cost.

It is difficult to translate this into specific component performance requirements. It is obvious that the overall efficacy of power subsystem components must increase in order to get the "more from less" as just noted. Solar cell efficiencies must improve, batteries that can be operated at greater depths of discharge will be required, and more efficient power electronics must be developed. Because these components work together as a complete system, it is not possible to assign specific numeric performance requirements to individual subsystem items.

Table 2–18
Load Requirements of Recent Satellites

*Current or Near-Term Projects:*

| Spacecraft | Payload (W) | Bus (W) | Total Capability | EPS Mass Fraction | Type of Payload |
|---|---|---|---|---|---|
| Alexis | 50 | 50 | 100 | | Astronomy |
| HCMM | 25 | 155 | 180 | .26 | Earth observing |
| SAGE | 21 | 159 | 180 | .47 | Earth observing |
| Viking | 56 | 66 | 122 | .26 | In situ science |
| AMPTE | 40 | 60 | 100 | | Earth observing |
| SAMPEX | | | 79 | .45 | Astronomy |
| HETE | 67 | 43 | 110 | | Astronomy |
| MSTI-1 | ? | ? | 90 | .29 | Earth observing |
| STEP 0 | 100 | 95 | 195 | .33 | Autonomous Ops. |
| STEP 1 | 70 | 35 | 105 | .29 | In situ science |
| STEP 2 | 40 | 55 | 95 | .37 | Comm technology |
| STEP 3 | 80 | 75 | 155 | .38 | Technology Demo. |
| TOMS-EP | 25 | 103 | 128 | .18 | Earth observing |
| SeaStar | 58 | 142 | 200 | | Earth observing |
| APEX | 117 | 100 | 177-280 | .25 | High Rad. Technology |
| MicroLab | 65 | 30 | 95-228 | .25 | Varies |
| Clementine | 68 | 292 | 360 (1Au) | | Remote sensing |
| FAST | 17 | 38 | 55 | .40 | Earth observing |
| SWAS | 85 | 145 | 230 | .29 | Astronomy |
| NEAR | 41 | 360 | 400 | | Remote sensing |
| ACE | 96 | 276 | 374 | | Astronomy |
| Average | 59 | 120 | 170 | | |

*Short-Term Future:*

| Spacecraft | Payload (W) | Bus (W) | Total Capability | Type of Payload |
|---|---|---|---|---|
| SA-3 | | | 310 | |
| COMET | | | 181 | Material processing |
| GEOSAT follow-on | 120 | 200 | 320 | Oceanography |
| PRoFILE | 11 | 10 | 21 | |
| AV/SAT (T-24) | 95 | 18 | 113 | |
| SALT (T-35) | 97 | 34 | 131 | |
| TECHSAT A | 40 | 70 | 110 | Technology testbed |
| Average | 73 | 66 | 169 | |

A big growth area for small satellites will be constellations of small, LEO, and mid-altitude communications satellites. Loads will be higher and steadier than for science satellites. Loads will not be driven by seasons or the day-night cycle, rather the load profiles will be based on whether the ground track is over populated areas or not. The assembly line fabrication planned for these constellations will hopefully drive down costs and perhaps make many subsystem components more affordable for other users and spacecraft developers.

Smallsat-class direct broadcast satellites are being developed (by CTA for a South American country). These satellites will have relatively high loads. Specifically, a payload with maximum load requirements ranging from 200 to 300 watts, and bus loads leveling out at about 100 watts, might become a reality. This bus load cannot be reduced much further, especially since stringent attitude control is required for the direct broadcast payload. Many smaller applications with modest load requirements will exist, but they will demand high performance as the total spacecraft bus mass will be strictly limited.

Future smallsats could achieve a total load capability of around 400 W with reasonable sized solar arrays.

### 2.8.3 General EPS Requirements

2.8.3.1 Payload Requirements—Payload requirements can either directly or indirectly affect the EPS. A direct effect would be a payload that requires unusual duty cycles. An indirect effect would be a requirement that does not directly affect the EPS, but imposes a requirement on another subsystem, which, in turn, affects the EPS.

2.8.3.1.1 Direct Impact Requirements—Instruments, such as radar systems and lasers, that operate in a pulsed mode directly affect the EPS design. They require alternately large amounts of power and then much less power. Normal design practices use an orbit average load as the design point, but with such a unique duty cycle, this point must be modified. One approach would be to design for the basic orbit average loads, ignoring the pulse requirement, and then augment the design for the pulse requirements. The specifics of the system would depend on the frequency and relative magnitude of the pulse loads.

A less extreme case might be a sensor that requires a high load but only by a factor or two or three times higher than its standby load. In this case, the system would be designed for whatever the orbit average worked out for the worst case load profile. The result might be that a system is oversized for 75 percent of the conditions. For these situations, special operating modes could be considered. These modes include "campaigns" that are temporary operating profiles. The payload has the power it needs for a limited amount of time. Then, the payload power is reduced while the EPS "catches up" or while it goes through a period of less favorable orbital conditions (i.e., poor Beta angles). Types of payload that are candidates for "campaign" modes include infrared sensors that require a stirling cycle or some other power-hungry cooler.

A trade off may be made between the payload duty cycle and power delivery capability. If payload requirements are high but the power subsystem cannot provide the required power in the worst-case, end-of-life conditions, it may be desirable to limit payload operation to less than 100 percent duty cycle in the periods of low energy availability. This situation is also a candidate for "campaign" mode operations, where the payload operates for a full orbit but the power subsystem does not reach energy balance within the same orbit (i.e., the batteries are not fully recharged at the end of the sun period). During the following orbit, the payload is either shut down or operates at a reduced duty cycle in order to allow the batteries to come to full charge. A variation would be to operate the instruments at a reduced duty cycle during any orbit that is energy poor. The specific design depends in the orbital conditions, the instrument requirements, and the power subsystem capability.

In all of the special cases, restricting payload operations to sunlight periods only could have a significant effect on the design of the EPS. This approach can be termed a capabilities-driven design approach, rather than the more traditional requirements-driven design approach, and is likely to be necessary for many smallsats.

Smallsat costs and complexity are contained by designing according to the subsystems' available capabilities rather than by unconstrained payload requirements.

Other direct impacts, not related to loads, are the FOV requirements. For deployed, and especially gimbaled, arrays, interference between the arrays and the instrument FOV can be a major design driver.

2.8.3.1.2 Indirect Impact Requirements—An instrument that requires very accurate pointing may need large reaction wheels that draw a large amount of power, which indirectly affects the design of the EPS. The EPS could also be indirectly affected by a instrument that requires an extremely stable platform. This requirement could influence the design of the solar array, which would require some minimum stiffness for reducing vibration and jitter. Certain types of payloads with motors may create motor loads or generate noise on the power bus. These effects could result in a design that includes filtering and isolation.

In almost all cases, the spacecraft bus is designed to be at a positive potential, with the return being grounded or negative. Because this is the common design, spacecraft components and devices for use with positive busses are readily available. There is, however, a special class of instruments that make in situ measurements of low energy plasmas in low, near-Earth orbits. These instruments include Langmuir probes, retarding potential analyzers, ion drift meters, etc. Since positive solar array interconnect tabs attract large electron currents, the spacecraft will achieve a large negative potential. These potentials preclude ionospheric electron measurements and severely compromise ion measurements. As a result, the solar array must either be a negative array or, if the array is positive, all conductors must be insulated from the ambient plasma. Both solutions are possible, but the latter is the usual solution. Insulation coatings were used early on Canadian Alouette and ISIS I and II, and on the Pioneer Venus Orbiter. Negative busses have also been used, for example on the Dynamics Explorers, DE-1 and DE-2.

Since small satellites provide a likely opportunity for this type of instrument, trade offs among the availability and cost of equipment and devices for a negative bus versus instrument sensitivity, and cost of insulating conductors will be required to determine the "best" solution for any given set of instruments.

**Instrument payloads requiring a negative bus are most likely to be flown on smallsats, and will require careful trade studies to meet this unique requirement in a cost-effective manner.**

2.8.3.2 Spacecraft Bus Requirements—As with the payload requirements, the spacecraft bus and its subsystems can have special requirements that affect the EPS.

2.8.3.2.1 Direct Impact Requirements—Primary among the items that directly affect the EPS are the inefficiencies in any subsystem's use of power, which result in increasing the power required from the array-battery combination.

The requirement that the bus operate during launch and orbit acquisition on battery power can result in deep discharge of the battery. Therefore, design limits for discharge must be set to preclude loss of the mission. Similarly, safe mode requirements for minimal power, for spacecraft orientation, for array sun-pointing, etc., place a variety of power requirements on the design of the subsystem. Most of the power required by the thermal subsystem is for operation of either heaters or refrigerators, which directly affects the EPS. Because of the orbital cycling of the spacecraft between sunlight and eclipse, the thermal subsystem places cyclic power requirements on the EPS. In addition, thermal subsystem limitations result directly in higher battery temperature and hence, limiting the allowable battery DOD. The effectiveness of thermal paints in maintaining a low array temperature directly affect the array efficiencies.

Structural elements also directly affect the EPS design. Since they are thermal conductors, maintenance of the low temperatures required for arrays, batteries, and power conditioning equipment becomes a necessary design element. Structural requirements also affect the size and weight of the arrays and batteries. The instrument structure can affect the output of the array if a part of it, such as a boom, comes between the Sun and the solar cells. This can be significant

because the net effect of shadowing a single cell is to create an open circuit for the entire cell string, although bypass diodes are occasionally used to overcome this effect. Structures also affect deployment possibilities for the solar arrays. Deployment of structures with pyrotechnics require pulse power.

The attitude control subsystem (ACS) is a major power user. The more stringent the requirements for attitude control, the more power will be required. This statement applies to the requirements for attitude control of both the bus and the array. The size, stowage and deployment of the array can all be affected by ACS considerations with respect to drag moments.

The exhaust from the propulsion subsystem directly affects the location of the array. The array must be protected from contamination. In addition, the impingement of the exhaust must be prevented from placing unwanted torques on the spacecraft.

2.8.3.2.2 Indirect Impact Requirements—Requirements for thermal surfaces and associated paths for heat rejection can result in restrictions on possible array locations. Design of the structure, including hinges, reenforcements, available attach point locations, etc., can affect cell layout and packing factor. In addition, the location of structural elements can cause reflections and thermal emissions resulting in undesirable heating of the array. Field of view requirements for ACS sensors can restrict array location and articulation. Conversely, the array design must prevent reflections from interfering with the sensors.

# 3. THERMAL SUBSYSTEM DESIGN CONSIDERATIONS

## 3.1 OVERVIEW

The thermal design of a large or small spacecraft is basically the same. The components, analyses, and technologies used are identical. The difference comes from the requirements of the spacecraft and payload components, their operational scenarios, and the spacecraft orbits.

Due to NASA's current emphasis on flying "cheaper, smaller, and faster" missions, small spacecraft with a standard bus architecture and expandable capabilities are being developed. This emphasis also discards the design approach of using unique spacecraft components and embraces the use of existing, modular, and expandable spacecraft components.

In this type of environment, the imperative becomes to constantly challenge the flight worthiness of the existing designs for a particular mission. This injunction becomes particularly important when designing thermal subsystems since they are extremely sensitive to the orbital environment. A bus that is designed and optimized for a sun-synchronous orbit will not provide stable thermal control for non-sun-synchronous orbits during which the Sun radiates all spacecraft faces. Through eclipse periods and during spacecraft operation, noncritical loads are switched off to provide for the survival of the components.

This section describes the function of the thermal design, the four life cycle conditions/boundaries that must be accommodated, their unique characteristics, a brief overview of the spacecraft subsystem characteristics with "problem components" identified, and a brief discussion the environment (References 8 and 23 through 26).

## 3.2 THERMAL SUBSYSTEM FUNCTIONS

The function of a satellite thermal control subsystem is to maintain the temperature of all spacecraft components within defined limits over the mission lifetime, subject to a given range of environmental conditions and operating modes. Thermal control subsystem design is unusual in that, given good technique and reasonable circumstances, the thermal subsystem may involve little or no actual spacecraft hardware, whereas more demanding subsystems require on-board hardware. Even so, the thermal control engineer must participate in the design of all on-board subsystems. Specifically, the thermal design must maintain component temperature limits throughout the mission under changes in attitude, power dissipation, sun angles, and operating modes (power levels). In addition, the power, weight, and volume allowed for the thermal control subsystem are constrained. In the case of small satellites, these resources are further limited by the reduced spacecraft size and the limited launch vehicle performance.

### 3.2.1 Impacts of Orbit Parameters

An orbiting spacecraft attempts to reach thermal equilibrium with its surrounding environment. The environmental inputs of solar radiation, Earth albedo, and Earth infrared (IR) radiation are balanced against the radiated spacecraft thermal energy, as depicted in Figure 3–1. Orbit parameters, such as altitude and inclination, influence these three environmental inputs and the view factors governing the spacecraft radiated output. Normalized views of the solar and Earth inputs are integrated to define the total input flux radiation environment that the spacecraft must accommodate. The normal sun-incident radiation dominates the thermal design at all altitudes. Because of this fact, the thermal design is evaluated against the "hot" and "cold" thermal cases, which are driven by the sun-incident angle on the spacecraft. The variation in incident angle defining the "hot" to "cold" cases is primarily a function of orbital altitude and inclination. For slowly precessing orbits, the time-of-day of launch determines where the mission starts on the periodic variation of incident angle with time. This starting time could affect the thermal design

for short duration missions. Also, variations in load power due to operating scenarios or contingency situations can contribute to the definition of the hot and cold cases.

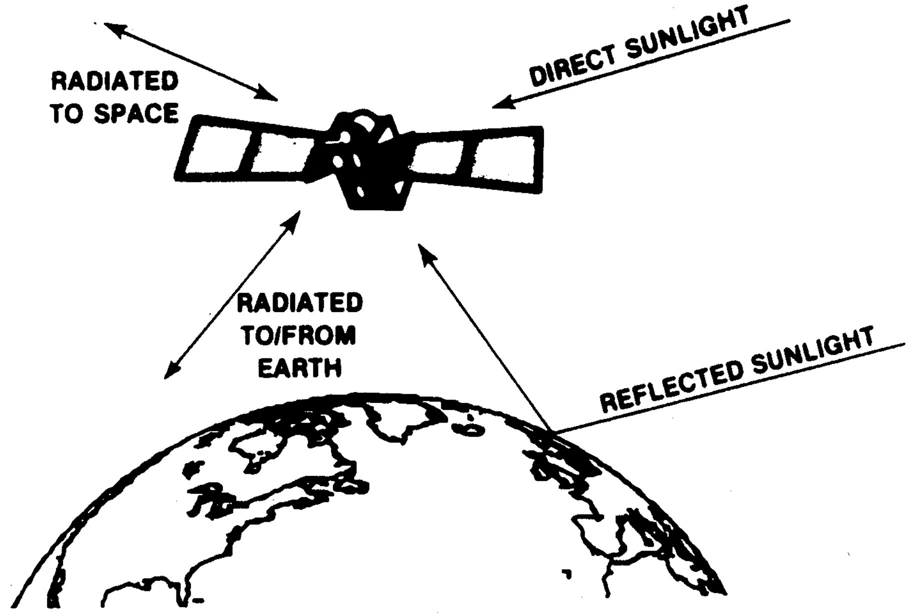



Figure 3–1. Energy Balance for an Earth Orbiting Spacecraft

The Sun provides a secondary radiative input as reflected energy off the Earth. This input is called Earth albedo and is usually accounted for as a percentage of the direct sunlight. Figure 3-2 presents spacecraft incident energy as a function of altitude and sun-incident angle.

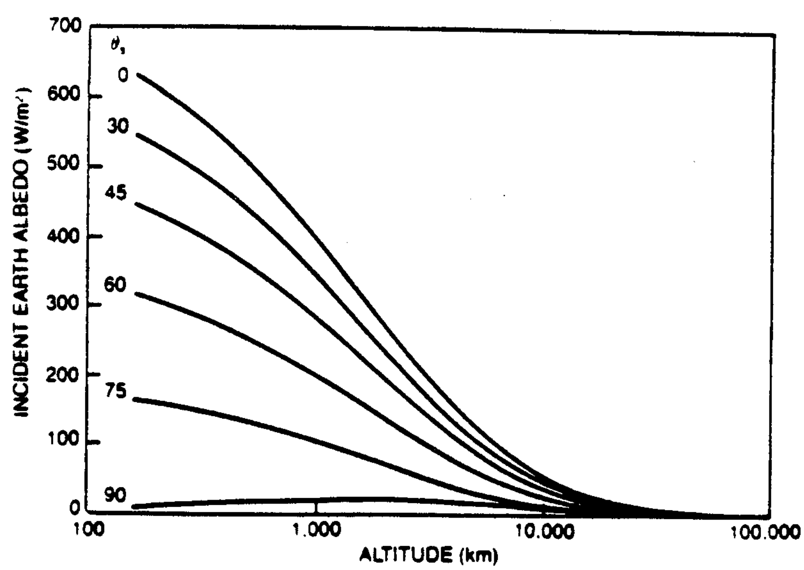



Figure 3–2. Incident Earth Albedo Per Unit Cross-Sectional Area of a Spherical Satellite as a Function of Altitude (Courtesy of NASA)

The Earth can be represented as a blackbody which emits long wave IR radiation and has an average temperature of approximately 254K. It provides an additional radiative input to the orbiting spacecraft as illustrated in Figure 3–3.

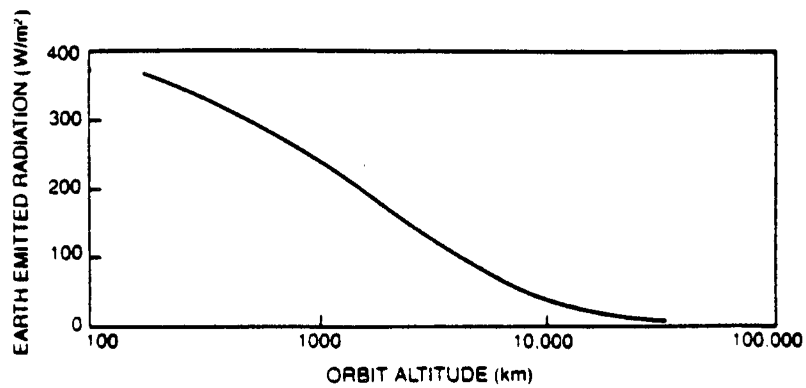



Figure 3–3. Earth Emitted Radiation to Sphere of Unit Projected Area (The Earth is Considered a Black Body at 254 K) (Courtesy of NASA)

3.2.1.1 Sun-Synchronous Orbits—This circular orbit offers the thermal design an advantage because direct Sun radiation impinges on one defined side of the spacecraft. The direct Sun radiation on only one side means that the spacecraft has a true anti-sun surface from which to radiate heat. Sun-synchronous orbits are defined by inclination and altitude, as was shown earlier in Figure 2-9.

## 3.3 LIFE-CYCLE PHASES (OPERATING MODES)

Thermal control work is dominated by analysis and coordination with other subsystems until the spacecraft is fully integrated and ready for system test. From the thermal standpoint the important phases are: Ground Operations; Launch and Ascent; Pre-stabilization and Transfer Orbit; and Final Orbit and Stabilization. Although it is useful to consider them separately, the problem solved in one phase must not create a problem in another phase.

### 3.3.1 Ground Operations

Ground Operations, except for thermal vacuum testing, are done in air, and thus the advantages of cooling by convection can be employed. Since most space electronics work well in the normal laboratory environment, 20°C and 40 percent relative humidity, little additional work is usually needed to achieve acceptable temperatures. Exceptions include local cooling ducts to control the high dissipation component and items that operate at temperatures below the local dew point and may have to be bagged, shrouded, and flooded with dry nitrogen at the proper temperature. If cryogenic temperatures are necessary for the checkout of a component, then a system with a permanent vacuum may be required.

Shipping the completed satellite to the launch site also presents thermal problems and winter/summer variations must be accommodated.

When the spacecraft is integrated to the launcher a new set of problems arise. Generally, the spacecraft is less accessible and operations, such as spin balance or mating to a solid rocket motor, are carried out at different sites.

### 3.3.2 Launch and Ascent

During launch, the thermal environment is very transient and consists mainly of radiative heating before the fairing is ejected and aerodynamic heating after it is ejected. In a daylight launch, the solar flux will also have to be included along with the aerodynamic heating. Fairing temperatures are normally provided by the launch vehicle supplier and by the radiative properties of the fairing walls. After the fairing is jettisoned, the heating rates must be calculated from the velocity and altitude profiles. Frequently, the heat rate normal to a flat plate is provided and can be used as a worst case estimate. Thermal blankets are particularly vulnerable because of their very low thermal capacity (small mass), but, any item with low thermal capacity is at risk since the fluxes are greater than several equivalent suns.

For small satellites, the aerodynamic heating is more of a concern because thermal mass of the spacecraft is lower and unique launch vehicle considerations exist. The Pegasus launch vehicle attempts to maximize performance by jettisoning the payload shroud as early as possible. This action may require additional thermal analysis to ensure that the spacecraft is not harmed. Negotiation for a later separation time may be necessary.

> The limited thermal mass and unique launch vehicle characteristics associated with smallsats makes aerodynamic heating during boost an area of concern for small satellites.

### 3.3.3 Pre-stabilization and Transfer Orbit

After launch vehicle separation, the satellite may be in an uncontrolled attitude and in an interim orbit. In the interim orbit, the percent sun may be different, the operation mode is different, and an interim attitude is used. The thermal design must work over an extended period of time, and the satellite cannot coast through on thermal capacity as it can during the ascent phase. During this period, many payload components are turned off, and the thermal control requirement is expanded to wider survival temperature limits. Some items may not be deployed, such as solar panels, sunshades, and antennas. In some cases, these items may be used to cover radiators, which will maintain temperatures with lower power. Conversely, if the spacecraft subsystems are creating any significant load power failure to quickly deploy solar arrays may create a thermal "cocoon" effect. Locating the solar cells on the outside of the pre-deployment solar panels increases this effect.

Small satellite designs must accommodate greater values and dispersions for initial attitude and attitude rates. If low power actuators, such as magnetic torque rods, are used for initial attitude capture, the time required to achieve a stable attitude can extend to six to eight orbits. Since the attitude rates create a thermal averaging over the spacecraft surfaces, this time extension usually does not present thermal problems.

### 3.3.4 Operational Orbit and Stabilization

Most of the spacecraft life will be spent in this configuration. Therefore, the thermal control subsystem design is optimized for this phase. This environment is well defined, but may be constantly changing because of the precession of the orbit and the changing operating modes. The control of individual packages is critical. Operating limits are most stringent in this phase. Control requirements are tighter and must be maintained for long periods of time. Life factors of thermal control materials and devices must be considered, and sufficient design margin must be included to account for these changes.

## 3.4 SYSTEM INTERACTIONS

The thermal control subsystem interacts strongly with nearly all other satellite subsystems. Table 3-1 presents the interaction and dependence of the various spacecraft subsystems with the thermal control subsystem. Thermal control subsystem interactions with the power, attitude control, and propulsion subsystems are especially important. In addition, the spacecraft operational modes must be considered in the overall design.

Table 3–1
Thermal Subsystem Interactions

| Other Subsystems | Effect of Thermal Subsystem on Other Subsystem | Effect of Other Subsystems on Thermal Subsystems |
|---|---|---|
| ACS | Fluid flow in loops | Attitude affects heat balance power of wheels. |
| Power | Use of heater power to compensate for losses | Efficiency of power equipment |
| Propulsion | No effect | Temperature limits of propellants; contamination of coatings by effluents |
| Command and Data Handling | Data and command for housekeeping and control | Low temperature requirements of Josephson Junction Devices |
| Structure | Addition of heat pipes changes stiffness | Low-conductive materials preclude good bi-directional heat transfer (composites for small satellites) |
| Communication | No effect | Antennas changing orientation varies thermal loads |
| Laser Communication | Optics and laser require thermal control and stability | |
| Cryogenic | Heat extraction from sensors could affect alignment | Large refrigerator with high heat load requires large radiators |
| Electromechanical | Deployable radiators | No effect |
| Optics | May require special coatings and devices for isothermality | High heat loads to be dissipated |
| Payloads | Must be integrated with instruments and sensors | Require narrow temperature limits |
| Environment | No effect | Contamination of coatings |

### 3.4.1 Electrical Power Subsystem

The thermal control subsystem must reject the heat generated by the electrical power subsystem (EPS). The EPS sometimes generates more power than is needed because of a requirement for a power margin and because all components may not operate simultaneously. This excess power can vary over large extremes from a few watts (for a non-dissipative subsystem) to nearly several hundred watts (for a dissipative subsystem). In either case, the excess must be rejected by the thermal control subsystem in a controlled way. The thermal control subsystem must also accommodate unusual changes in load requirements, such as happens during a low voltage shut down in which all of the non-critical loads are removed from the bus. This situation may last from a few minutes to several hours.

The battery presents unique design problems for the thermal control subsystem. Batteries discharge and charge and sometimes are stored off line. In all of these cases, closer temperature control is required than is required with most other components. Ten degrees centigrade, plus or minus five, is typical. This tight control, coupled with the large variation in power dissipation in

the battery from zero to tens of watts, makes battery temperature control one of the most difficult aspects of the thermal design.

Another effect of the EPS on the thermal control subsystem is the location of the solar array. For body-mounted solar arrays, the thermal control subsystem design is severely constrained because most of the satellite surface area is covered with solar cells. Recently, the needs of scientific instruments for more power have nearly eliminated body-mounted solar array in favor of separately mounted solar panels.

### 3.4.2 Attitude

Choosing the type of thermal control subsystem to use depends on satellite attitude. The attitude along with the orbit determines the direction of solar flux impingement on the satellite. This factor is one of the primary factors in choosing a radiator location. The absence of solar flux on a surface permits more choice in thermal control materials because the solar absorbing properties of the materials do not need to be considered.

During the life of a satellite, the attitudes required by the mission may result in many different solar flux impingement angles, which must be accommodated.

The type of attitude control can also affect the thermal control subsystem design. If the control is Gravity Gradient with a long boom, a thermal consideration is boom bending. The attitude control has little interaction with thermal once the final orbit and stabilization are established. Using magnetic coils for control can require large amounts of power for substantial lengths of time and then no power for longer time periods. Momentum wheels present different problems because the power consumption is a function of temperature.

### 3.4.3 Propulsion

The thermal control subsystem design must keep the fuel and associated plumbing at temperatures above the fuel's freezing point. Typically mono-propellant and bi-propellant systems employ hydrazine as the fuel. Hydrazine must be kept above 5°C. Before a jet is used, the catalyst bed must be warmed to the proper temperature to prevent degradation of the hydrazine. This requirement may require significant heater power.

### 3.4.4 Operational Modes

Spacecraft rarely remain in a single operating mode throughout their life even during the operational phase. As a result, the power level changes, and the thermal distribution changes. Large changes are seen between transfer orbits and final orbit. Frequently power availability is less in the transfer orbit. In other cases, the EPS cannot support the loads full time so they are duty cycled. This procedure means that active control may be necessary if thermal capacity alone will not smooth out the transients. Limiting modes should be identified early in the design so that limitations are not later imposed on operations.

Small satellites are especially susceptible to variations in the load power because the mass of the launch vehicles for small satellites is limited, which time often limits available power and instrument operation. For campaign modes, significant time may be spent in a high power mode, followed by a significant time in an off-mode for battery recharging. This type of operation results in extreme hot and cold thermal cases.

## 3.5 DESIGN DETAILS

Throughout the phases of spacecraft missions, the heat dissipated internally by components and in the external heating fluxes varies significantly. During the orbit transfer phase, the power dissipation is usually very low. In the on-orbit phase, the spacecraft is either three-axis stabilized, spin stabilized, or gravity-gradient stabilized. Depending on the method of stabilization and the mission modes of operation, power dissipation varies over a wide range. Through eclipse periods and during spacecraft operation, non-critical loads (usually instruments) are switched off to provide for the survival of the components. Table 3–2 depicts typical spacecraft subsystem and component temperature requirements. The thermal control subsystem design becomes challenging when the spacecraft carries a dimensionally stable platform that is needed to accurately point antennas, attitude sensors, and/or supports high-power, low-temperature payloads.

Table 3–2
Subsystem Design Temperature Levels

| Subsystem/Component | **Operating Design Temperature (degrees centigrade)** Maximum | Minimum |
|---|---|---|
| Communications Subsystem | 65 | -5 |
| Transmitter | 60 | 10 |
| Antenna | 70 | -5 |
| Command & Data Handling Subsystem | 50 | -5 |
| Power Subsystem | 60 | 0 |
| Batteries | 20 | 0 |
| Solar Array Mechanical Assembly | 90 | -80 |
| Attitude Control System | 60 | 0 |
| Propulsion Subsystem | 50 | 5 |

Note: – The design temperatures shown represent the typical range within the subsystem. Individual components may have other design temperatures within the design range shown. Acceptance temperatures are defined as design maximum plus 5 degrees centigrade and design minimum minus 5 degrees centigrade. Qualification temperatures are defined as design maximum plus 10 degrees centigrade and design minimum minus 10 degrees centigrade.

### 3.5.1 Thermal Designer Road Map

Because of payload requirements, the spacecraft orbital parameters and type of attitude control subsystem are typically fixed before design of the thermal control subsystem begins. The spacecraft size to support the payload is fixed and the electrical power budget requirements are also defined. The orbits that represent the spacecraft "hot" case and cold case must be determined. Typically the cold case occurs when the Sun is in the spacecraft orbit plane (Beta=0°) because the Earth maximally occults the Sun during part of the orbit and lowers the orbit average solar radiance flux input. The hot case occurs when the eclipse time is smallest. For a high inclination orbit, this happens when Beta = 90°, resulting in an eclipse time of zero. The spacecraft receives full solar radiation for the total orbit time. Additional conditions, such as maximum and minimum solar flux due to variations in the distance from the Earth to the Sun and maximum and minimum power dissipation conditions, are also combined to produce a worst case hot and cold case. The Beta angle variation is repetitive, and the time between cycles is a function of orbit inclination and altitude. Figure 3–4 depicts a 95°, high inclination orbit Beta angle history that has a repeat period of about 2-1/2 years, while Figure 3–5 depicts a low 49° inclination orbit in which the solar cycle repeats about every 30 days (orbital altitude of 400 km

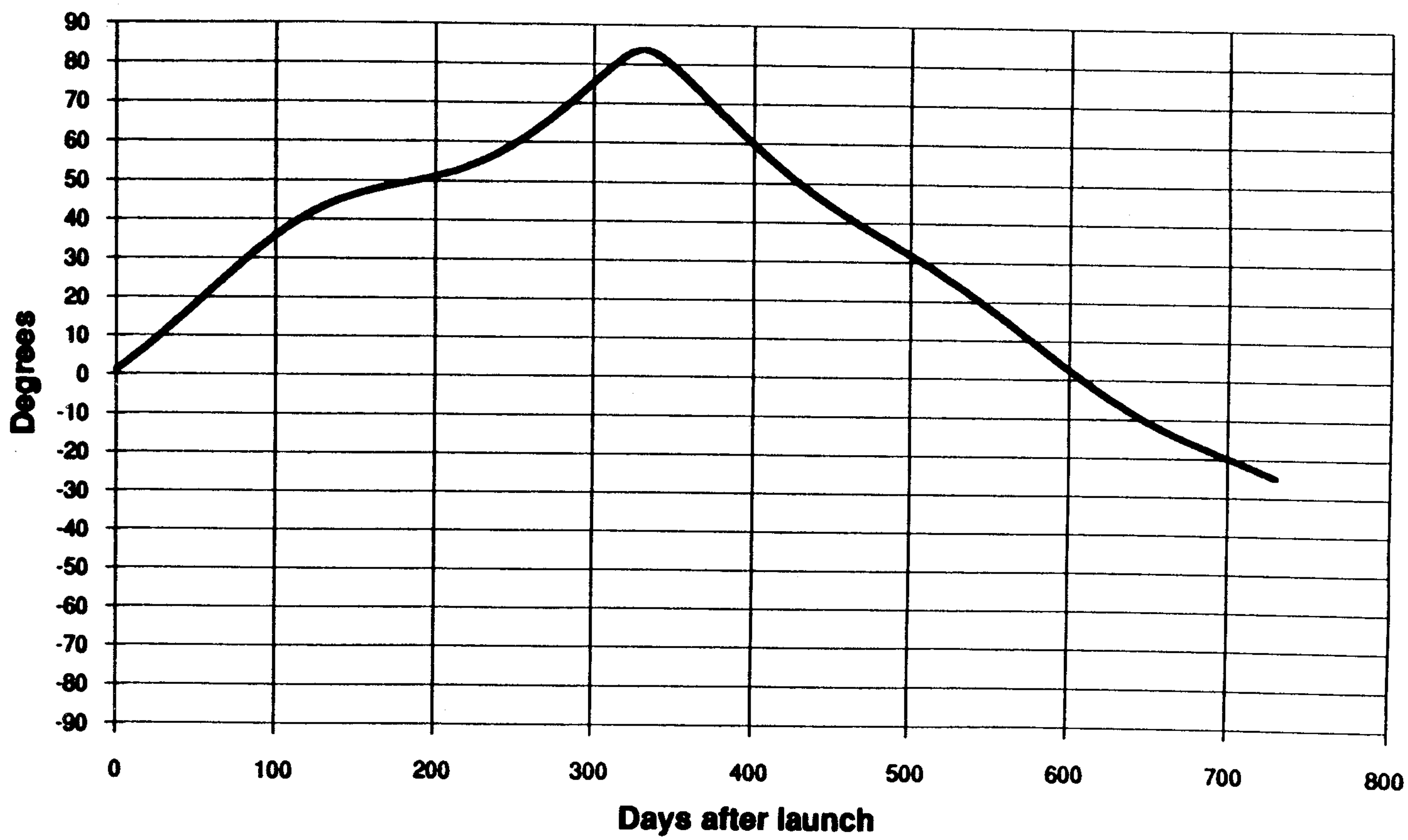



Figure 3–4. High Inclination (95˚) Solar Beta Angle

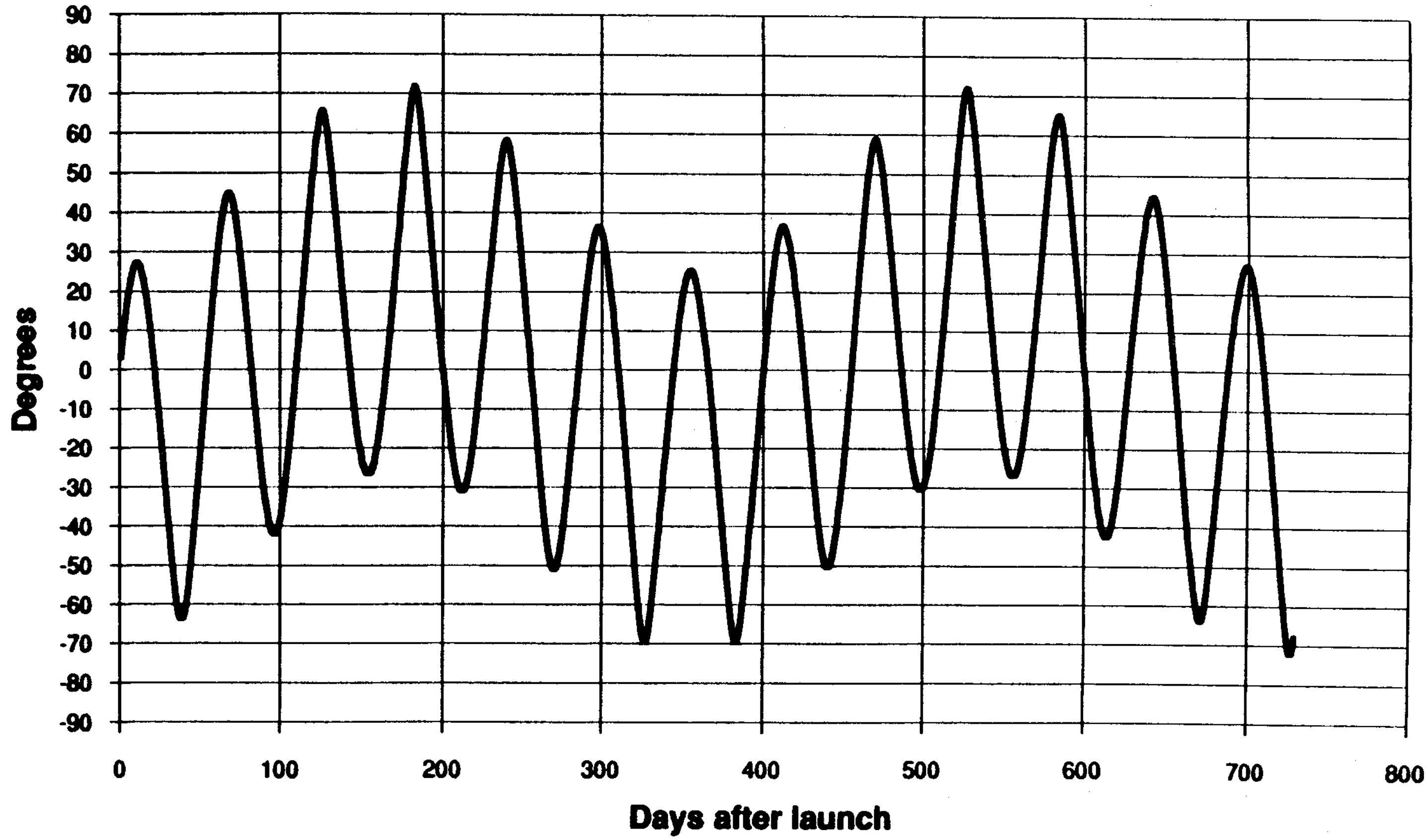



Figure 3–5. Low Inclination (49°) Solar Beta Angle

for both cases). The cyclic nature of the Beta angle is driven by a combination of the Earth's annual motion about the Sun and the precession of the satellite's orbit plane. For both cases, the solar flux impinges on all sides of the spacecraft during each cycle.

The major tasks in developing an initial thermal control subsystem design are itemized below:

1. Determine the orbit environment parameters and establish resulting input flux levels.
2. Determine the internal electrical power requirements and quantify the temperature limits for each component.
3. Identify the worst hot case and cold case conditions.
4. Locate and size the radiator(s) to accommodate the worst hot case.
5. Analyze the cold case to ensure temperature limits are maintained and provide active control such as heaters, if needed.
6. Iterate between tasks 4 and 5 to obtain optimum design balance.

Additional tools, techniques, and materials to accomplish these tasks are depicted in Figure 3–6.

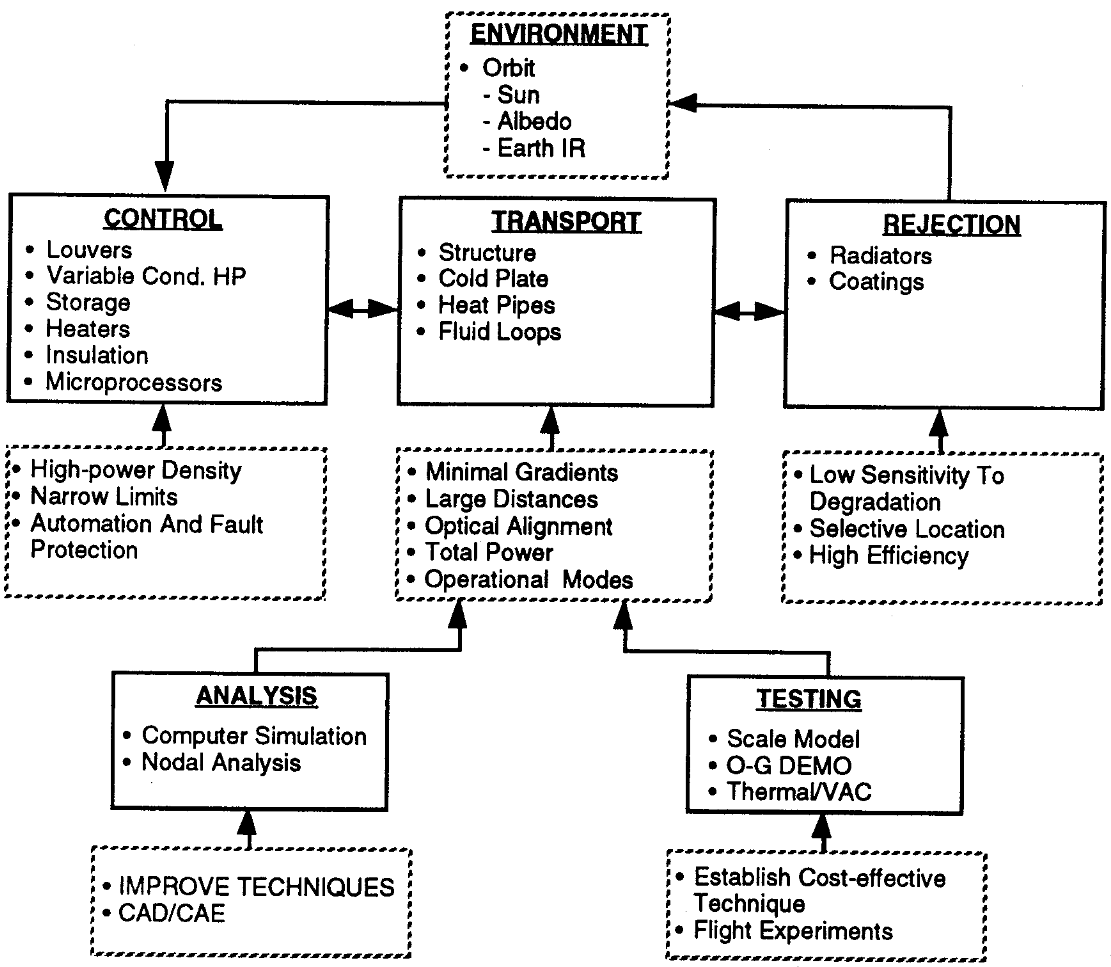



Figure 3–6. Thermal Control Subsystem Design Tasks

### 3.5.2 Internal Heating

The spacecraft can be viewed as a cocoon with a defined exit path for heat to radiate to the environment. By understanding the orbit external environment and controlling the heat flux escape rate, the internal temperature of the cocoon is controlled. Therefore, the internal design task is to account for all heat sources as a function of time and operational modes. Each heat source must be dissipated either by radiation or conduction to a radiating surface. The spacecraft internal environment is usually separated from the external environment by wrapping the spacecraft in a thermal insulating blanket called multi-layer insulation (MLI). Interface boundaries are then designed to control heat flux movement.

Small spacecraft typically require 150-200 watts for the spacecraft subsystems plus the payload power, which typically is less than 100 watts. Thus, the total power to be rejected from the internal cocoon is approximately 300 watts.

### 3.5.3 External Heating

The emitted radiation from the Sun is constant within a fraction of 1 percent at all times. However, due to the Earth's elliptical orbit, the intensity of sunlight reaching the Earth varies approximately $\pm 4.9$ percent, depending on the Earth's distance from the Sun. At summer solstice, the intensity is at a minimum (1287 $W/m^2$). It is at a maximum (1419 $W/m^2$) at winter solstice.

Solar intensity also varies as a function of wavelength, as shown in Figure 3–7. The sun-equivalent temperature is 5800K. The energy distribution is approximately 7 percent ultraviolet, 46 percent visible, and 47 percent near (short wavelength) infrared. The radiation energy emitted by the Sun is at a much shorter wavelength than the radiation energy emitted by a body near room temperature. This distinction allows the selection of thermal control finishes which are very reflective in the solar spectrum, but are highly emissive to room temperature (long wavelength) IR, as shown in Figure 3–7. These finishes minimize solar energy input and maximize waste heat rejection from the spacecraft. White paint absorbs weakly in the short solar wavelength band and emits strongly in the long IR wavelength band. The result is that a white-painted surface runs cool in the sunlight.

The value of the Earth's albedo is highly variable. Reflectivity is greater over continental (as compared to oceanic) regions and increases with decreasing local solar elevation angles and increasing cloud coverage. Albedo increases with latitude because of increasing snow and ice coverage, decreasing solar elevation angle, and increasing cloud coverage. These variations make selection of albedo constant values for a thermal analysis rather uncertain. It is not unusual to find variations throughout the industry. A good approximation is to assume the albedo to be equal to 30 percent of the incident sunlight. During final design analysis, specific input values similar to those in Table 3–3 are used for design concept verification.

The Earth achieves thermal equilibrium by balancing the energy absorbed from the Sun with the energy re-emitted as long wave-length IR radiation. This balance is maintained fairly well on a global annual average basis. The intensity of IR energy emitted at any given time from a particular point on the Earth, however, can vary considerably. Factors determining this variation include surface and air temperatures, atmospheric moisture content, and cloud coverage. The highest intensities are over clear tropical regions. The intensity decreases with increasing latitude and increasing cloud cover.

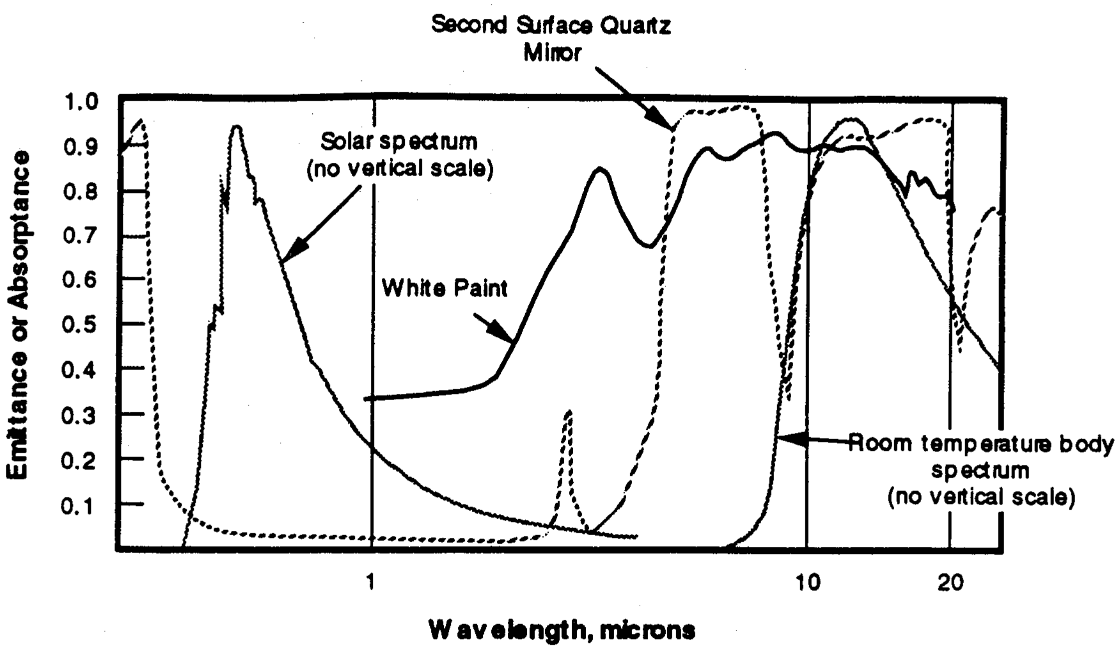



Figure 3–7. Thermal Performance of White Paint and Second Surface Mirror

Table 3–3
Zonal Mean Albedo (Percent)

| Latitude (deg) | | Month | | | | | | | | | | | | Annual | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | J | F | M | A | M | J | J | A | S | O | N | D | Min | Avg | Max |
| 90 | 80 | 80* | 78* | 69 | 58 | 69 | 75 | 69 | 60 | 44 | 70* | 75* | 78* | 44 | 69 | 80 |
| 80 | 70 | 81* | 83 | 69 | 63 | 62 | 54 | 50 | 49 | 66 | 77 | 77* | 80* | 49 | 68 | 83 |
| 70 | 60 | 78 | 56 | 55 | 54 | 49 | 42 | 39 | 40 | 42 | 51 | 64 | 69 | 39 | 53 | 78 |
| 60 | 50 | 56 | 50 | 49 | 42 | 40 | 39 | 38 | 38 | 37 | 43 | 49 | 55 | 37 | 44 | 56 |
| 50 | 40 | 46 | 43 | 40 | 36 | 34 | 35 | 33 | 32 | 33 | 34 | 39 | 46 | 32 | 37 | 46 |
| 40 | 30 | 37 | 36 | 34 | 32 | 30 | 28 | 27 | 26 | 29 | 29 | 33 | 37 | 26 | 31 | 37 |
| 30 | 20 | 30 | 29 | 27 | 25 | 25 | 25 | 26 | 26 | 26 | 26 | 28 | 30 | 20 | 26 | 30 |
| 20 | 10 | 24 | 23 | 22 | 20 | 22 | 24 | 26 | 27 | 25 | 25 | 26 | 26 | 25 | 24 | 27 |
| 10 | 0 | 24 | 24 | 24 | 24 | 26 | 26 | 26 | 26 | 26 | 25 | 26 | 26 | 24 | 25 | 26 |
| 0 | -10 | 25 | 24 | 24 | 24 | 21 | 22 | 23 | 23 | 22 | 23 | 24 | 25 | 21 | 23 | 25 |
| -10 | -20 | 24 | 23 | 23 | 22 | 21 | 21 | 22 | 22 | 22 | 23 | 24 | 24 | 21 | 22 | 24 |
| -20 | -30 | 23 | 24 | 24 | 23 | 24 | 24 | 25 | 25 | 25 | 25 | 24 | 23 | 23 | 24 | 25 |
| -30 | -40 | 27 | 28 | 29 | 29 | 30 | 30 | 30 | 30 | 29 | 29 | 28 | 27 | 27 | 28 | 30 |
| -40 | -50 | 33 | 34 | 34 | 37 | 38 | 39 | 39 | 36 | 35 | 34 | 34 | 33 | 33 | 35 | 39 |
| -50 | -60 | 41 | 41 | 40 | 42 | 44 | 47 | 48 | 45 | 43 | 44 | 43 | 42 | 40 | 43 | 48 |
| -60 | -70 | 46 | 47 | 46 | 54 | 62 | 72 | 77 | 65 | 56 | 56 | 52 | 49 | 46 | 56 | 77 |
| -70 | -80 | 61 | 62 | 61 | 86 | 86* | 87* | 87* | 88 | 79 | 65 | 66 | 64 | 61 | 74 | 83 |
| -80 | -90 | 70 | 72 | 40 | 80* | 81* | 82* | 82* | 83* | 80 | 67 | 75 | 75 | 40 | 74 | 83 |
| Min | | 23 | 23 | 22 | 20 | 21 | 21 | 22 | 22 | 22 | 23 | 24 | 23 | | | |
| Avg | | 45 | 43 | 39 | 41 | 42 | 42 | 42 | 41 | 39 | 41 | 43 | 44 | | | |
| Max | | 81 | 83 | 69 | 86 | 86 | 87 | 87 | 88 | 80 | 77 | 77 | 78 | | | |

* Estimated value

The IR energy emitted by the Earth, which is around room temperature, is of approximately the same wave length as that emitted by satellites. It is of much longer wave length than the IR energy emitted by the Sun at 9500°F. Unlike short wave-length solar radiation, the Earth IR loads can not be reflected away with special thermal control coatings since the same coating would block the radiation of waste heat away from the spacecraft. Because of this fact, Earth-emitted IR energy presents a particularly heavy backload on spacecraft radiators in low-altitude orbits, which must emit energy at the same wave length.

Orbit average values of Earth IR can be used in most analyses, with recommended values for IR heating being 250 W/m$^2$ for summer solstice and 225 W/m$^2$ for winter solstice. Table 3–4 presents specific values as a function of latitude and month.

Another form of environmental heating is aero-heating, which usually occurs in the altitude region where free molecular heating predominates. This heating is usually only encountered during launch just after the booster's payload fairing is ejected. It is desirable to drop the fairing as soon as possible after launch to minimize the amount of dead weight which the booster must deliver to orbit. The point at which the fairing is separated is often determined by a trade-off between the desire to maximize launch vehicle performance and the need to protect the spacecraft from excessive atmospheric heating. On-orbit free molecular heating should only be assessed for spacecraft perigee altitude of less than 200 km.

Table 3–4
Monthly Average IR and Albedo
(Stevens, Campbell, and Von der Haar)

Zonal Mean IR (W/M$^2$)

| Latitude (deg) | | Month | | | | | | | | | | | | Annual | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | J | F | M | A | M | J | J | A | S | O | N | D | Min | Avg | Max |
| 90 | 80 | 165 | 155 | 146 | 173 | 189 | 207 | 207 | 199 | 178 | 170 | 163 | 177 | 146 | 177 | 207 |
| 80 | 70 | 157 | 149 | 154 | 183 | 197 | 211 | 212 | 207 | 186 | 175 | 160 | 156 | 149 | 178 | 212 |
| 70 | 60 | 165 | 164 | 170 | 196 | 208 | 221 | 224 | 217 | 198 | 186 | 173 | 167 | 164 | 190 | 224 |
| 60 | 50 | 175 | 177 | 188 | 204 | 213 | 222 | 228 | 224 | 211 | 200 | 188 | 182 | 175 | 201 | 228 |
| 50 | 40 | 191 | 194 | 203 | 216 | 226 | 235 | 244 | 243 | 232 | 220 | 205 | 198 | 191 | 217 | 244 |
| 40 | 30 | 217 | 218 | 224 | 235 | 241 | 254 | 259 | 263 | 253 | 247 | 231 | 222 | 217 | 238 | 263 |
| 30 | 20 | 250 | 248 | 251 | 265 | 266 | 268 | 262 | 261 | 261 | 263 | 253 | 251 | 248 | 258 | 268 |
| 20 | 10 | 266 | 264 | 261 | 270 | 260 | 251 | 241 | 236 | 242 | 250 | 251 | 259 | 236 | 254 | 270 |
| 10 | 0 | 251 | 251 | 248 | 240 | 232 | 233 | 233 | 235 | 235 | 244 | 242 | 250 | 232 | 241 | 251 |
| 0 | -10 | 240 | 240 | 240 | 243 | 257 | 261 | 261 | 261 | 260 | 257 | 248 | 246 | 240 | 251 | 261 |
| -10 | -20 | 248 | 247 | 250 | 264 | 270 | 273 | 272 | 276 | 271 | 266 | 257 | 253 | 247 | 262 | 276 |
| -20 | -30 | 261 | 256 | 254 | 263 | 258 | 260 | 260 | 264 | 259 | 258 | 256 | 262 | 254 | 259 | 264 |
| -30 | -40 | 253 | 251 | 244 | 239 | 233 | 229 | 231 | 232 | 233 | 238 | 239 | 249 | 229 | 239 | 253 |
| -40 | -50 | 232 | 232 | 225 | 217 | 213 | 209 | 205 | 207 | 211 | 219 | 220 | 229 | 205 | 218 | 232 |
| -50 | -60 | 217 | 217 | 208 | 204 | 199 | 195 | 188 | 187 | 192 | 201 | 208 | 216 | 187 | 202 | 217 |
| -60 | -70 | 209 | 204 | 193 | 186 | 177 | 172 | 164 | 161 | 165 | 180 | 197 | 209 | 161 | 184 | 209 |
| -70 | -80 | 196 | 184 | 165 | 153 | 146 | 146 | 131 | 124 | 128 | 151 | 183 | 220 | 124 | 160 | 220 |
| -80 | -90 | 187 | 171 | 148 | 121 | 105 | 110 | 104 | 94 | 94 | 126 | 170 | 190 | 94 | 135 | 190 |
| | Min | 157 | 149 | 146 | 121 | 105 | 110 | 104 | 94 | 94 | 126 | 160 | 156 | | | |
| | Avg | 215 | 212 | 209 | 215 | 216 | 219 | 218 | 216 | 211 | 213 | 213 | 218 | | | |
| | Max | 266 | 264 | 261 | 270 | 270 | 273 | 272 | 276 | 271 | 266 | 257 | 262 | | | |

3.5.4 Conceptual Design Steps

The following list delineates the thermal control subsystem conceptual design steps:

- Determine the internal power generated by subsystems and as a function of operational modes and time.
- Determine the orbit profile characteristics, identify hot and cold cases, and size the total external heat fluxes.
- Assume the spacecraft will be 100 percent covered with MLI except for locations of the designed radiator. No external heat load is assumed for this covered area.
- Collect all heat and conduct it toward the radiator. Assume 85 percent through conduction and 15 percent through radiation.
- Determine temperature drops in the pathway conducting to the radiator, especially for key subsystem components.
- Position the radiator(s) on a spacecraft location that is optimized; i.e., least view angle of the Sun, Earth IR, and albedo.
- Determine appropriate radiator surface material/coating; i.e., if anti-sun, use white paint; if Sun facing, use OSRs or silver Teflon. Use EOL emittance and absorption values. Typically white paint EOL emittance is 0.2 and absorption is 0.8.
- Do a preliminary energy balance. Use orbit average fluxes for Sun, Earth IR, and albedo input fluxes. Use incident values and eclipse times to derive orbit average value at the radiator proposed surface location on the orbiting spacecraft.

  Q electrical + Q Sun + Q Earth IR + Q albedo =
  radiator area x emittance($\underline{e}$) x Boltzmann constant (s) temperature $(T)^4$.

- Size radiator to maintain the desired temperatures of key subsystem components.
- Add thermal control elements as required to meet the mission requirements (heaters, heat pipes, louvers, etc.). Maintain as simple a system as possible. For small satellites, additional concerns are minimal power consumption, weight, and shroud volume requirements.

An approach to thermal design that has been widely used recently is the cold bias design. In this approach, the spacecraft is designed to be just barely in thermal equilibrium under the worst hot case conditions by using standard passive techniques. This generally results in larger than usual radiators. During colder conditions, heaters are used to maintain the temperatures within the desired range. The term "cold bias" comes from the fact that the spacecraft radiator is oversized, thereby causing all components to run towards the cold end of their operating range. Then the make-up heaters can be set to bring the components' temperatures to their normal operating range. This type of design is very forgiving because the heaters dynamically control and compensate for errors and degradation over time. This also allows for simpler pre-flight thermal tests.

### 3.5.5 Detailed Design Analysis

Once a design concept is identified, a detailed analysis fine tunes the design and predicts spacecraft component temperatures under the range of flight conditions. Two generalized computer programs are typically used. The first is a radiation program that defines the absorbed energy on the spacecraft's external surfaces. This same program also defines the radiation-exchange and view factors, which determine the radiation interaction between the spacecraft's surfaces. Next the designer identifies analysis cases to be run and constructs a geometric math model (GMM) and a thermal math model (TMM) of the spacecraft.

The GMM and TMM serve different purposes. The GMM is a mathematical representation of the physical surfaces of the spacecraft and is used to calculate blackbody radiation couplings between surfaces and heating rates due to environmental fluxes. The TMM is most often a lumped-parameter network representation of the thermal mass and conduction and radiation couplings of the spacecraft and is used to predict spacecraft temperatures. The radiation interchange couplings and environmental heat fluxes calculated by the GMM are used in constructing the TMM. Both the GMM and TMM use industry-standard codes. The most common TMM code is SINDA. Other commercially available codes do exist, and some large companies have their own internally developed codes.

## 3.6 HEAT-TRANSFER MECHANISMS

The basic heat-transfer mechanisms are radiation, conduction, and convection. Radiation is used to control the overall energy balance between a spacecraft and space. Conduction is used to control the energy flow among different areas of the spacecraft. Convection is unimportant to the design of small spacecraft, except in the control the pre-launch environment, transfer orbits, and very low altitude orbits.

### 3.6.1 Radiation

Thermal radiative transfer is a function of the temperature of the emitting and receiving bodies, the surface materials of the bodies, the intervening medium, and the relative geometry. The heat flux from a surface varies as the fourth power of its temperature. For heat rejection at low temperature, a large area is required. This requirement may cause a geometrical problem in that requirements for field of view, size, and weight must also be met. Also, for low temperature instruments on small spacecraft with a limited amount of surface area, heat rejection may become a problem.

> **Heat rejection, especially for low-temperature sources, can be a significant problem for smallsats because of the limited area available for radiators.**

### 3.6.2 Conduction

Conduction is effective when adequate conduction paths are provided. Meeting this requirement means more than selecting a material with suitable conductivity. It also means using conduction pads, thermal grease, or metal-loaded epoxy for areas, such as unwelded joints, that do not conduct heat very well. Metal-loaded epoxy is used only when high or repeatable conductivity is essential to the design.

## 3.7 GENERIC TYPES OF THERMAL CONTROL SUBSYSTEMS

Thermal control architectures may be broadly grouped into two classes: passive and active. Passive techniques, such as sunshades, fans, coatings, or insulating blankets, do not require

spacecraft resources and ground intervention to implement. They are usually preferred to active controls because of the simplicity, reliability, and low cost of such systems. Active thermal control of the spacecraft requires use of devices, such as heaters or coolers, or change of the spacecraft attitude.

### 3.7.1 Passive Thermal Control

A passive thermal control subsystem uses conduction paths, geometry, coatings, insulation blankets, sun shields, radiating fins, and heat pipes to maintain the temperature of spacecraft components within the required limits.

3.7.1.1 Geometry—"Geometry" refers to manipulating the spacecraft configuration to optimize thermal control. It may mean providing the required thermal radiating area, placing low-temperature components in shadow, and exposing high-temperature components to the Sun or burying them deeply within the structure.

3.7.1.2 Insulation Blankets—Insulation blankets are multilayer insulation consisting of layers of aluminized Mylar or other plastic spaced with nylon or Dacron mesh. External coverings of fiberglass, Dacron, or other materials may be used to protect against atomic oxygen attack, micrometeoroids, etc.

3.7.1.3 Sun Shields—Sun shields may be as simple as a polished aluminum sheet or gold-plated aluminum. More sophisticated shields use silvered Teflon, which acts as a second-surface mirror. It has silver on the back which provides reflectivity and Teflon on the other side which provides high IR emissivity. Glass second-surface mirrors may also be used. They are more efficient, but are heavier and may become brittle.

3.7.1.4 Radiating Fins—Radiating fins are often used to improve the radiation efficiency, especially when a large surface area is required or large amounts of heat must be radiated. Fins have some disadvantages. For example, large numbers of fins in circular configurations tend to obscure the view to space. In addition, the ability to conduct heat through very long fins is limited, thus decreasing the effectiveness of the fins.

3.7.1.5 Heat Pipes—Heat pipes are tubular devices that contain a wick which runs the length of the pipe. The pipe is partially filled with a fluid, such as ammonia, and is connected between a portion of the spacecraft from which the heat is removed to a portion from which the heat is being dumped (Reference 27).

Heat pipes have a much greater thermal conductivity than metals do. Also, heat pipes transport thermal energy at small temperature differentials. Since they transport heat from internal equipment to a radiator surface, they distribute dissipation, minimize hot spots, and provide heat radiation at moderate temperatures.

3.7.1.5.1 Heat Pipe Applications—Heat pipe can be used to create isothermal surfaces and act as a thermal transformer to change the flux density of the heat flow. One way heat pipes have tested and flown is as variable conductance heat pipes, which were used to maintain a constant temperature evaporator surface under varying load conditions. For this applications, the heat pipe must be properly designed and plans for testing must include time for adequate ground testing. Heat pipes are often bonded to a radiating fin to increase the efficiency of the heat transfer. This configuration also optimizes the weight of the system for the energy dissipated.

3.7.1.5.2 Heat Pipe Operation—Consider a simple horizontal heat pipe in equilibrium with an isothermal environment. The liquid in the wick and the vapor in the vapor space are at saturation. If heat is applied to the evaporator, raising its temperature, liquid in the wick

evaporates, removing some of the added heat. The meniscus is depressed in the evaporator since less liquid is present. This process also raises the local vapor pressure since it must be saturation with the heated liquid in the wick.

The difference between the increased curvature of the meniscus in the evaporator wick and the unchanged meniscus in the condenser wick causes a difference in capillary pressure sufficient to pull liquid from the condenser wick toward the evaporator. This action replenishes the liquid in the evaporator wick. At the same time, heated vapor flows from the evaporator to the condenser, which is at a lower pressure. When this vapor comes in contact with cooler condenser surfaces, it condenses. This cycle is shown schematically in Figure 3-8.

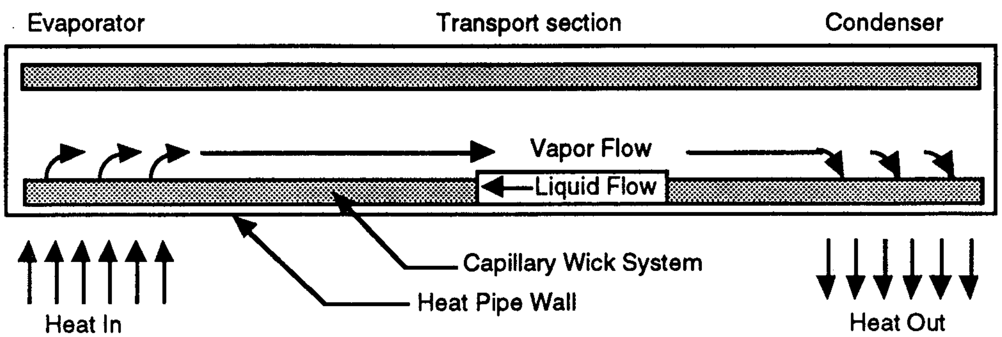



Figure 3–8. Heat-pipe Schematic

Since the latent heat of vaporization of most heat-pipe working fluids is high, only small amounts of fluid must flow in order to transport significant quantities of heat. The driving mechanism, the temperature difference between the evaporator and condenser wall, is also small.

Caution is required in using heat pipes to make sure that the hot end is not so hot as to dry the wick completely, thus rendering capillary action ineffective in transporting new fluid into that end. Similarly, the cold end must not be so cold as to freeze the liquid.

Heat pipes work quite differently in zero-g because of the absence of free convection, making interpretation of ground test results a problem. To ensure adequate ground testing, the heat pipe layout must be oriented so that it may be tested horizontally. The use of heat pipes in small satellites is advantageous in that the thermal conductivity of a structural element can be increased considerably without increasing aluminum face sheet thickness or adding aluminum doublers, thereby avoiding adding excessive material (and mass). Thus, for small satellites that require additional heat transport, heat pipes offer a mass-effective solution at the expense of some added complexity in embedding or surface mounting the heat pipes with the structure and the additional testing.

3.7.1.5.3 Types of Heat Pipes

3.7.1.5.3.1 Constant-Conductance Heat Pipes (CCHPs)—Elements of this most basic heat pipe consist of a working fluid, a wick structure, and a surrounding wall structure. It need not be shaped like a "pipe." Large flat plates several feet across have been built and tested for special applications.

CCHPs are often categorized according to the type of wick structure used. The simplest design consists of axial grooves in the wall of extruded aluminum tube. This class of wick is very susceptible to gravity effects during ground testing, but is relatively inexpensive to produce and very consistent. Heat-transfer capability is moderate, but sufficient for many applications.

Grooves are typically rectangular or trapezoidal in shape, but more complex shapes such as the "teardrop" or "keyhole" have been extruded with difficulty.

3.7.1.5.3.2 Variable-Conductance Heat Pipes—These heat pipes use a gas reservoir, connected to the end of the condenser, that is filled with a noncondensable gas to control the operating area of the condenser as a function of evaporator temperature. In effect, the active-radiator area becomes a function of the electronic box cold-plate temperature, with increasing box temperatures leading to increased radiator areas. Although complicated models of the gas front exist, the gas front may be considered an impermeable floating piston. If the temperature at the cold plate rises, the vapor in the evaporator (at the saturation pressure of the liquid in the evaporator) rises rapidly. The pressure of the mixture of control gas and vapor in the reservoir must rise to compensate, so the "gas-front-as-piston" will retreat further into the condenser, decreasing the volume of control gas. This opens up more of the condenser area to heat-pipe operation. This is shown schematically in Figure 3-9.

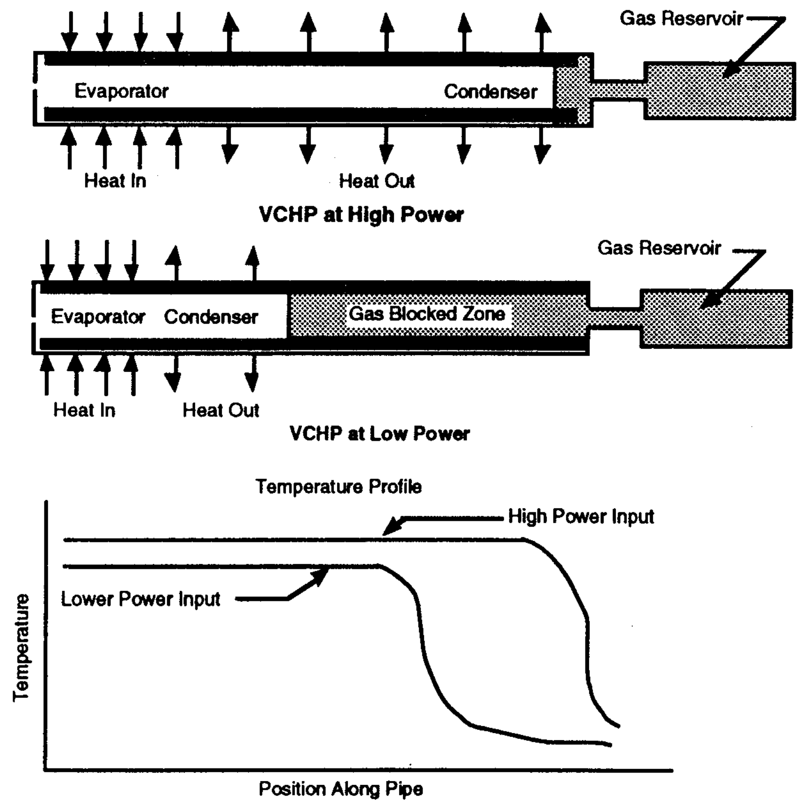



Figure 3–9. VCHP Operation

A number of schemes have been flown, which differ mainly in the treatment of the reservoir (some have wicks, some are kept hot or cold by exposure to different environments, and some become elements of an active thermal control system, a heater is connected via a feedback control to a sensor at the evaporator or payload). Sufficient control gas is usually present in the reservoir to enable these pipes to function as gas diodes if the heat pipe is reversed.

3.7.1.6 Capillary Pumped Loops—The capillary pumped loop (CPL) is a two-phase heat-transfer loop that uses a capillary structure to return liquid to the evaporator. The concept is still under development, although several systems have flown as Shuttle experiments. It shows promise when high heat-transfer rates must be sustained over relatively long distances. A schematic of the loop in operation is shown in Figure 3-10. As heat is applied, vapor is vented

from the evaporator into the condenser duct, where it begins to condense on the wall. Liquid film on the wall is carried along by the vapor flow, with the thermodynamic quality (the ratio of vapor to total mass) of the flow dropping. At a certain point in the loop, surface-tension forces will predominate, and the flow will consist of slugs of liquid mixed with collapsing bubbles of vapor. By the time it reaches the liquid side of the evaporator, pulled by capillary forces within the evaporator, the stream will consist solely of subcooled liquid.

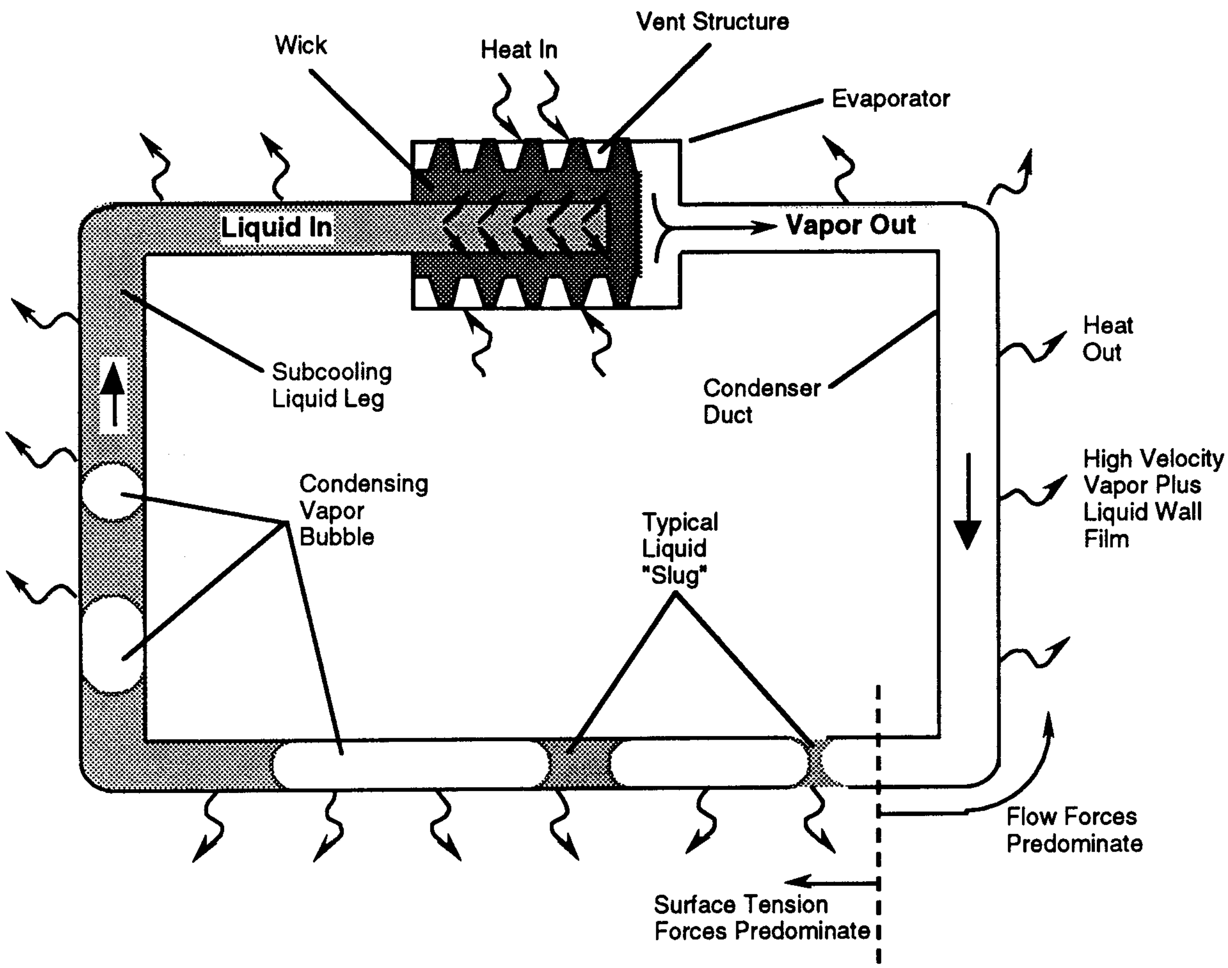



Figure 3–10. Capillary Pumped Loop Schematic

This device is essentially a heat pipe with the return flow of condensate to the evaporator in a separate unwicked tube. The only wick in the CPL is in the evaporator, where it distributes liquid from the supply tube over the actual evaporator surface.

A heat pipe can be used when the heat source and heat sink must be physically separated. If a heat pipe is used, it is not necessary to mount all hardware to be cooled directly on radiator panels, or to use relatively inefficient conductive couplings (This requirement tends to occur when cooling boxes must be kept close to each other for more efficient electrical or microwave design.). By the same token, heaters need not be mounted directly on hardware to be heated if a heat pipe is employed. Heat pipes are also used to reduce temperature gradients in structures to minimize thermal distortion.

The diode heat pipe was first proposed as a means of connecting a device to two radiator panels on opposite sides of a spacecraft, with at least one of the radiators free of direct solar load at all times. The diodes would couple the device to the cold radiator, while preventing heat from leaking back into the system from the radiator in the Sun.

The VCHP can be used to control the amount of active radiator area, providing reasonable good temperature control without the use of heaters. This design is attractive if electrical power is limited. This type of design has been flown on a number of satellite experiments. If the application requires maintaining a box or baseplate at virtually a constant temperature, feedback control (at the expense of some heater power) may be employed. A sensor on the baseplate of the device to be controlled can be routed to an on-board computer, and whenever the temperature drops below the desirable range, heaters on the VCHP reservoirs are activated, causing the control gas to expand and block off more of the radiator area. If the temperature rises above the range desired, power to the reservoir heaters is reduced, increasing the active radiator area. This concept usually requires less power than using heaters directly on the box or system to be controlled.

### 3.7.1.7 Finishes

In spacecraft thermal designs, wavelength-dependent thermal control coatings are used for various purposes. Solar reflectors, such as second-surface mirrors and white paints or silver- or aluminum-backed Teflon are used to minimize absorbed solar energy, and emit energy almost like an ideal black body. To minimize both the absorbed solar energy and IR emission, polished metal, such as aluminum foil or gold plating, is used. If it is desired to exchange energy within the spacecraft compartments or among other components, black paint is commonly used. Development and qualification of a new coating is normally unnecessary since many fully qualified coatings exist.

For external surfaces of a spacecraft, the two important surface properties are the IR emittance and the solar absorption. Figure 3–11 indicates the range of properties available for different types of materials. Two or more coatings are sometimes combined in a checkerboard or striped pattern to obtain the desired characteristics.

3.7.1.7.1 Thermal Surface Degradation—Thermal control finishes are affected in orbit by charged particles, ultraviolet radiation, high vacuum, degradation caused by atomic oxygen, and contaminant films. The result is in an increase in solar absorptivity and almost no change in IR emittance. This result is undesirable because spacecraft radiators are sized to account for the substantial increase in absorbed solar energy that occurs due to degradation over the mission. These radiators, which are oversized to handle the high solar loads at end-of-life, cause the spacecraft to run much cooler in the early years of the mission. The lower operating temperature may mean vent heaters must be added to electronics components at their proper temperature. Stable coating properties of materials minimize these problems.

The four types of thermal control surfaces are: (See Figure 3–12):

a) Solar reflector - Has a very low a/e ratio.

b) Solar absorber - Absorbs solar energy but emits only a small percentage of the IR energy. Polished metal surfaces approximate solar absorbers.

c) Flat reflector - Reflects energy throughout the spectral range; i.e., reflects in the solar and IR regions.

d) Flat absorber - Absorbs throughout the spectral range.

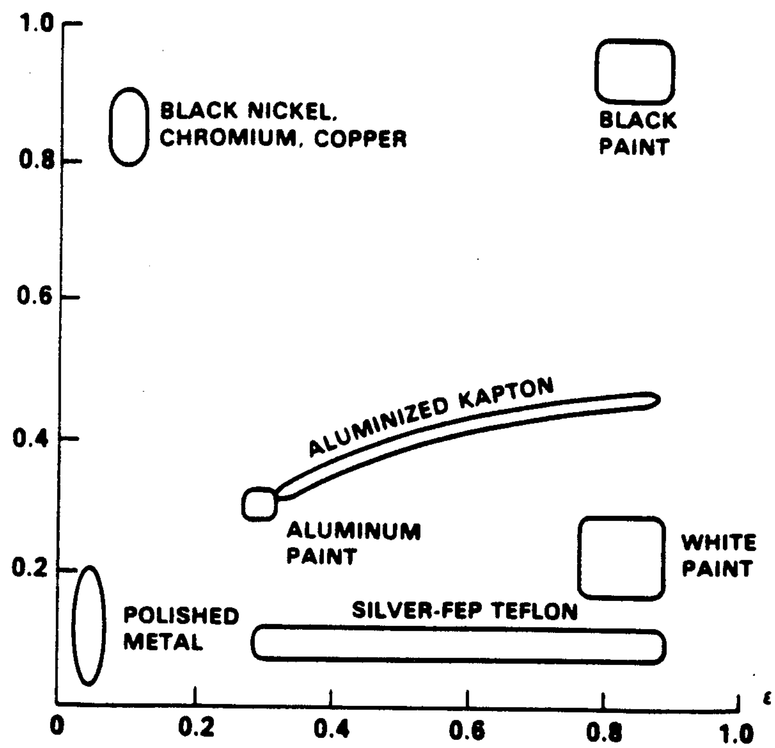



Figure 3–11. Typical Solar Absorptivity and Emissivity of Thermal Control Surfaces

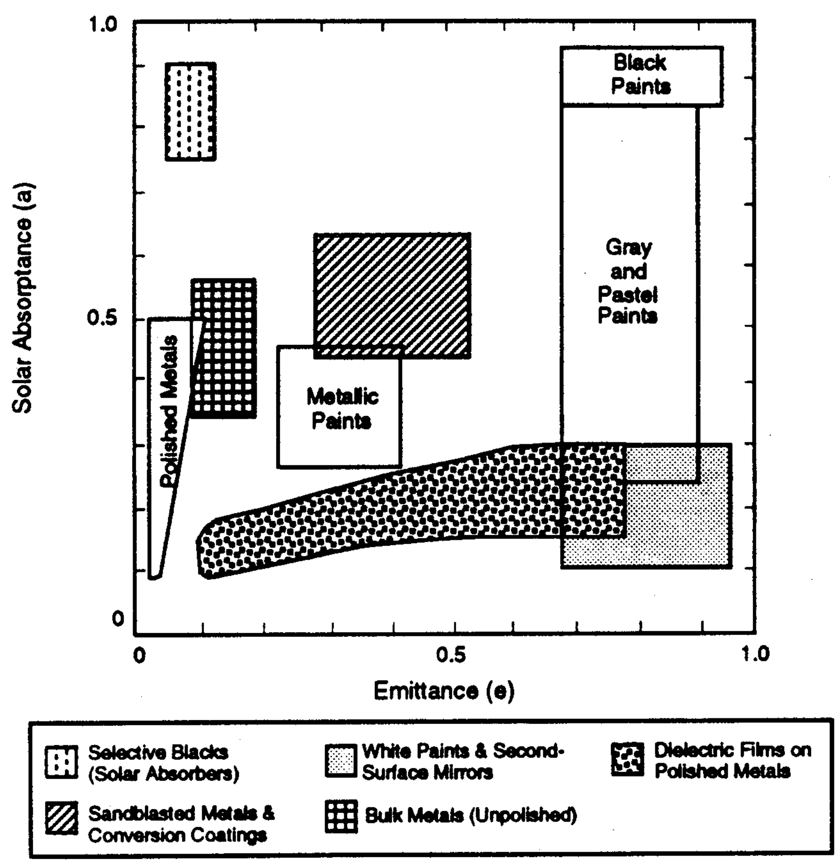



Figure 3–12. Surface Properties by Type of Finish

3.7.1.8 <u>Multilayer Insulation</u>–Multilayer insulation (MLI) and single-layer radiation shields are among the most common thermal control elements on a spacecraft. MLI blankets prevent components from losing heat excessively or from becoming overheated because of environmental fluxes, rocket plumes, etc. Most spacecraft are covered with MLI blankets that have cut-outs. The cut-outs provide surface for internally generated waste heat to be rejected. MLI blankets are also used to protect internal propellant tanks, propellant lines, solid rocket motors, and cryogenic dewars. Single-layer radiation shields are used when a lesser degree of thermal isolation is required. These shields are lighter, and they are cheaper to manufacture.

MLI is composed of multiple layers of low-emittance films. The simplest construction is a layered blanket assembled from embossed, thin Mylar sheets (1/4 mil thick). Each layer has a vacuum-deposited aluminum finish on one side which minimizes radiative heat paths between the layers. The layers are aluminized on one side only so that the Mylar can act somewhat as a low-conductivity spacer thereby reducing conductance. More complex, high-performance construction uses Mylar film metallized on both surfaces (aluminum or gold) with silk or Dacron net, Tissuglas paper, or "Super-Flock" whiskers as the low-conductance spacers.

Heat transfer through MLI is a combination of radiation, solid conduction, and, under atmospheric conditions, gaseous conduction. The gaseous-conduction heat transfer is minimized by allowing the insulation to vent to space or by using the insulation in an evacuated wall (such as in the space between a cryogenic pressure vessel and the external vacuum jacket shell). The performance of an MLI system can be severely degraded by the pressure of even very modest amounts of gas. Solid-conduction heat transfer is minimized by keeping the density of the low-conductance spacers between the reflective surfaces as low as possible and making the blanket "fluffy" to minimize contact between layers. Radiation heat transfer is minimized by interposing as many enclosing reflective surfaces (metallized sheets) as is practical.

3.7.2 <u>Active Thermal Control</u>

Active thermal control of the spacecraft requires devices such as heaters and coolers, shutters or louvers, or cryogenic materials to transport the heat which may be actively implemented by pumped circulation loops.

3.7.2.1 <u>Heaters</u>—Ideally, thermal control of a satellite or component is achieved using only passive techniques, such as surface finishes. Unfortunately, variations in environment and component heat-generation rates, along with degradations of surface finishes over time, can drive temperature variations in a passive design to ranges larger than some components can withstand. Heaters are then required to protect components under cold-case environmental conditions or to make up for heat that is not dissipated when an electronic box is turned off. A third common use for heaters is to warm-up components to their minimum operating temperatures before they are turned on.

3.7.2.1.1 <u>Patch Heaters</u>—The most common type of heater used on spacecraft is the patch heater. It consists of an electrical resistance element sandwiched between two sheets of flexible electrically insulating material, such as Kapton. The patch may have one circuit, or more than one for redundancy. Redundancy is generally required on spacecraft systems since heater circuits can fail. Sometimes the redundancy is provided within the patch and sometimes it is provided by using two separate patches. The patch heater shape usually used is a simple rectangular patch of standard dimension.

3.7.2.1.2 <u>Cartridge Heater</u>—A cartridge heater is often used to heat blocks of material or high-temperature components such as hydrazine-thruster catalyst beds. Such a heater consists of a wound resistor enclosed in a cylindrical metallic case. Attachment techniques include the use of

a clamp or small bracket to hold the heater or insertion into a drilled hole. These heaters are typically a quarter inch in diameter and a few inches long.

3.7.2.1.3 Control Devices–Heaters may be used with thermostats or solid-state controllers to provide precise temperature control of a particular component. Almost all heaters have some sort of control over their operation. This typically involves a relay that is commandable from the ground to enable or disable power to the heater, a fuse to protect the spacecraft from a short circuit, and, usually, a thermostat or solid-state controller to turn the heater on and off at predetermined temperatures. More sophisticated satellites use their on-board computer to monitor temperatures and turn heaters on and off using relays.

3.7.2.1.3.1 Command—The simplest arrangement involves the heater, a fuse, and a ground commandable relay to turn the heater on and off. This arrangement is used for heaters that are activated only for special events or for heaters that can be left on all the time. A typical application requires heating up the catalyst beds on hydrazine thrusters to around 100°C before the thruster is fired.

3.7.2.1.3.2 Bimetallic Switches—These switches are the mechanical thermostat for the heater. Each switch is in a small hermetically sealed can and is driven by a snap-action bimetallic actuator. The temperature at which the thermostat clicks on, known as its set point, is fixed for any given thermostat. The dead band, or the difference between the temperatures at which the thermostat turns on and turns off, is also important. A smaller dead band reduces the temperature swing of the device being heated and reduces power consumption a little (since the average temperature is lower). The smaller dead band also increases the number of cycles on the thermostat itself and decreases its reliability. Dead bands less then 4°C are not recommended, since the thermostat will experience "dithering" problems in which it rapidly cycles on and off. This "dithering" can cause the set point to drift lower and can result in too low a temperature for the component.

3.7.2.1.3.3 Solid-State Switches—Solid-state switches provide precise temperature control. They use a temperature sensor that can be located with the controller or at a remote location. Solid-state switches are often used to control the temperature of optical systems, sensors, and electronics since the switches can maintain extremely tight dead bands (<0.1°C). In addition to providing more accurate control, solid-state switches also are more reliable and have a longer life expectancy.

3.7.2.2 Coolers—Various cooling devices can be used as part of a common or independent thermal control subsystem. Thermoelectric, or Peltier cooling, has been used with some success for cooling small, well-insulated objects. The primary application is to use cooling detector elements in IR observational instruments that are operated for long periods.

A cooling device that is often used is the cryostat. This device achieves cooling by expanding a high-pressure gas through an orifice. Two-stage cryostats, using nitrogen in the first stage and hydrogen in the second, have been used to achieve very low temperature. The nitrogen precools the hydrogen to an initial temperature approaching the liquid temperature upon expansion. The disadvantage of such systems is the use of a consumable gas, which limits the number of operating cycles and operational lifetime.

For long-term cooling to low temperature, cryogenic fluids are used especially for IR measurements. Cooling by expanding liquid helium through a porous plug can reduce a telescope optical temperature to 4K, allowing observations at very long IR wavelengths without interference from the telescope thermal noise. Lifetime is limited to the time it takes to deplete the cryogen, usually 10 to 14 months.

Liquid or solid cryogens are typically not applicable for small satellite cooling applications due to the large mass required for missions of more than a few months duration. The proposed Wide-Field Infrared Explorer (WIRE) satellite, a smallsat, uses a solid hydrogen cooler but only has a life expectancy of four months.

Since the observing life ends when the cryogen is exhausted, even though all other systems are functioning, cryogenic mechanical refrigerators become useful. These refrigerators can achieve temperatures of 50K-60K and support heat loads approximately 100-300 mW. A more detailed discussion of cryogenic coolers is found in Appendix A-1.

3.7.2.3 Louvers—Louvers are among the most common active thermal control devices. They provide about a six-to-one variation in heat radiation from fully closed to fully open. They are used when the internal power dissipation varies widely because of the component's duty cycle.

3.7.2.3.1 Venetian Blind—The most widely used louver assembly is the bimetallic spring-actuated rectangular-blade type, often called a venetian blind. This type of louver reduces radiation throughput by approximately 33 percent.

This arrangement of activators, housing, blades, and structure is shown schematically in Figure 3-13. Each louver blade is independently actuated with a bimetallic clock spring alone or with a bimetallic clock spring used in conjunction with a controller. The second option can decrease the closed-to-open temperature range from 7°C to as little as 1°C. The narrower 1° closed-to-open range is typically used if the assembly is exposed to solar illumination.

This device is reliable. Since each louver blade is independently actuated, a single-point failure exists for only one blade at a time. This device is also simple and less costly than some other heat control devices. However, it cannot be as finely tuned, and it uses surface area less efficiently. It has a 2-inch mechanical profile that is typically independent of blade size, which decreases the maximum spacecraft radius. This 2-inch dimension is significant for small satellites given the constraints placed on the shroud volume.

3.7.2.3.2 Pinwheel Louvers—The pinwheel louver consists of a lobed louver blade, an actuator assembly, a guard ring, and a special radiator pattern, as shown in Figure 3–14. This louver may be selected due to its low mechanical profile (less than 0.5 inch) or its tolerance of solar loads. The louver will open passively through the action of a Bimetallic spring or may be driven open using an electronic controller and a small heater on the spring. This type of louver reduces effective radiator efficiency approximately 55 percent because the blades and supporting structure reduce the clear view to space relative to a plain radiator of the same size.

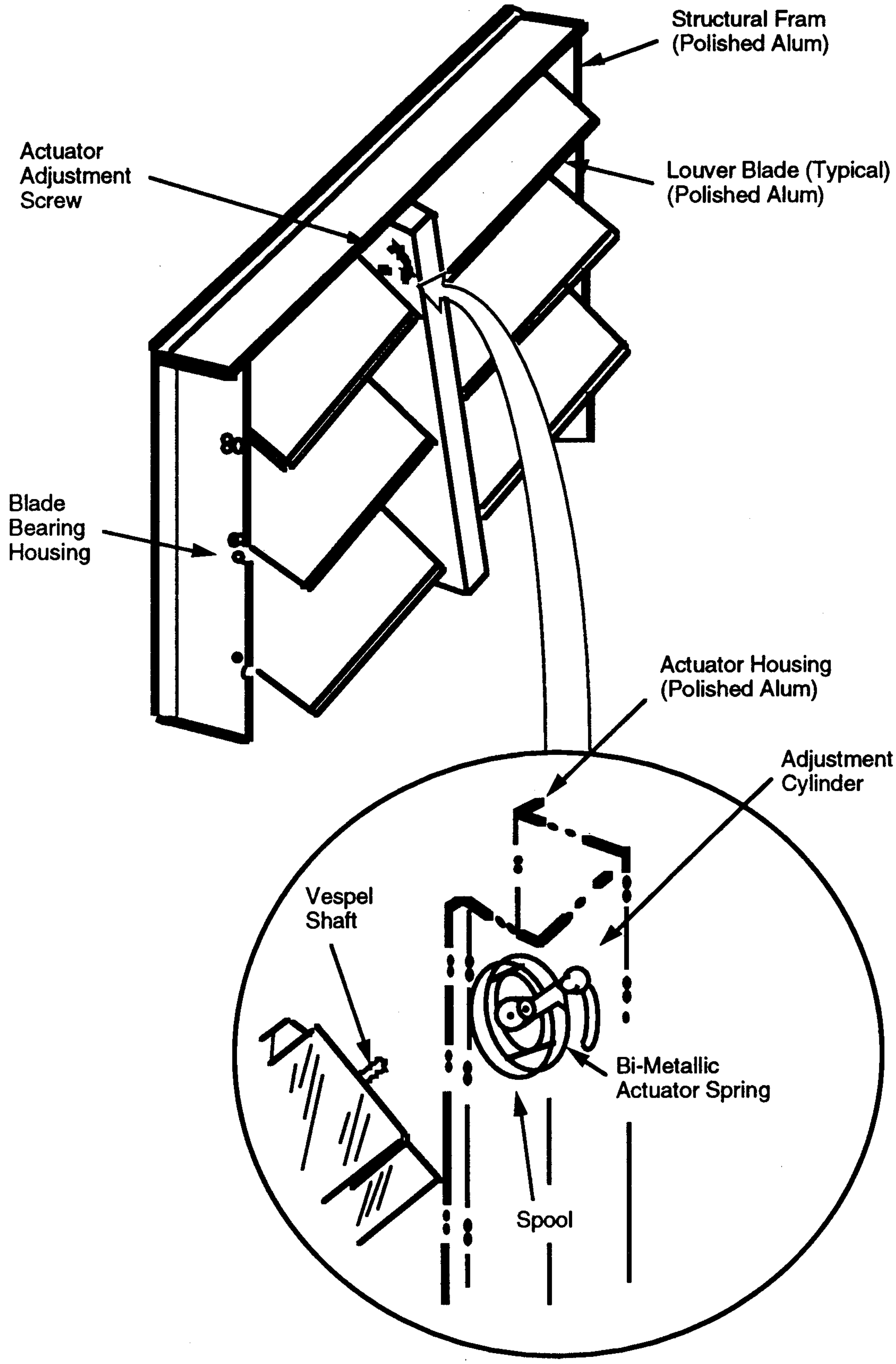



Figure 3–13. Typical Venetian Blind Louver Assembly Schematic



C-2.

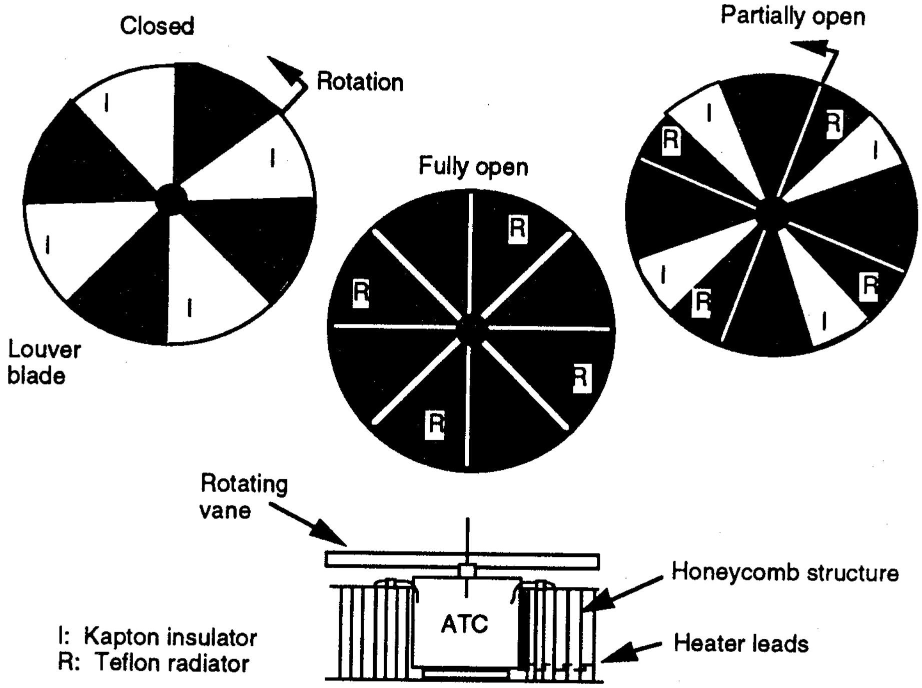



Figure 3–14. Typical Pinwheel Louver

## 3.8 PERFORMANCE REQUIREMENTS

### 3.8.1 General Requirements

3.8.1.1 Payload Requirements—On recently-built small satellites, the load requirement for an individual payload ranges from 3 watts to 50 watts, while the total payload, typically consisting of three to five experiments, may require up to 110 watts orbit average electrical power for normal operation. The temperature limits for the experiments during operation are in accordance with Table 3–2. Frequently, the payload will require special thermal resources to hold unique sub-elements at temperature levels either tighter than the spacecraft nominal variance or at a particular temperature totally outside of the spacecraft orbit average. Special radiators may be provided to accommodate these requirements. Typical special requirements may include sinking thermal electric control (TECs) devices or providing a cold radiator around an instrument's optics. TECs may require up to 15 watts of dissipation at sink temperature ranges between 0° and 15°C. Instrument radiators, which are most typically integral to the payload, may require 10 to 30 watts dissipating at sink temperature ranges of +5° to -50°C. The colder radiators are usually associated with a cyrogenic instrument.

Additionally some experiments place secondary temperature control on the spacecraft structure. This control is necessary to avoid structural distortions that may cause errors in pointing and knowledge. In severe cases, this control could mean holding the structure end-to-end temperature at just a few degrees temperature differential.

Generally, all internal components including the payload are painted black. Black provides the most ideal radiative interface as it obeys the ideal emission and absorptivity laws. It makes the analysis easier and the thermal design more forgiving.

3.8.1.2 Spacecraft Requirements—The spacecraft has several subsystems that require special attention. These are the propulsion, power (batteries) and the structures subsystems.

The propulsion subsystem as presented earlier is only concerned with maintaining the hydrazine ~5° above it freezing point at all times and preheating of catalyst beds.

Within the EPS, battery temperature limits are always more restrictive then the spacecraft orbit average excursions. Additionally, depending on the orbit, the power dissipation associated with discharge can be 20-30 watts and the battery must be held within a ±10° limit. If two or more batteries are used, they must be kept at the same temperature to ±3°. This requirement usually mandates mounting them in close proximity to each other and using a doubler or strap. Because of spacecraft mass balance requirements, multiple batteries are often placed on opposite side of the spacecraft. In this case, heat pipes are almost always used to maintain a minimal temperature difference between them.

> Heat pipes can be used to equalize the temperature of two batteries when they cannot be located in thermally identical environments, as is usually the case with small satellites.

Additionally, the structure is usually responsible for controlling conductive heat flux flow to the spacecraft radiator. The thermal designer must work hand-in-hand with the structural designer to insure thermal conduction requirements are met. Often, trades are made to determine the best solution. The simplest way to provide good conductive paths is to increase the mass of the path. Structurally, spacecraft typically use .008" face sheets bonded to the honeycomb. For thermal reasons, these sheets could increase from .020" to .030" if mass budgets permit in order to promote better conduction. Particular hot spot areas are often minimized through use of "doublers." Because of the limited weight margins associated with small satellites, alternate more weight-efficient approaches include the use of thermally conductive graphite honeycomb and face sheets and/or the use of embedded heat pipes. Typically a large spacecraft thermal subsystem accounts for 8-12 percent of the spacecraft total dry mass, where it only accounts for 4-8 percent on the lower power ~300 watt smallsats.

### 3.8.2 Future Requirements

Future requirements are driven by payload power. Table 2–11 shows that there are planned smallsats with total loads as high as 320 W. In the future, it is reasonable to expect that the average spacecraft will be required to provide a total ~400 watts orbit average. The spacecraft bus itself should remain stable at ~100 watts or decrease slightly based on new component technology implementation, and the remaining ~300 watts dedicated to the payloads.

## 3.9 CURRENT AND FUTURE PERFORMANCE

To understand the thermal capabilities of a smallsat that plans to use a Pegasus launch vehicle the following parametrics are used to size available solar array area for power generation and available radiator area for heat reject to the environment.

### 3.9.1 Packaging Parametrics

The objective of the parametric is to define the maximum solar array area and resulting spacecraft side panel area (that can be used as a thermal radiator) for a spacecraft that is bounded by a Pegasus shroud. The analysis utilizes a computer-aided drawing (CAD) system to configure both single and bifold solar arrays on 4, 6, 8, and 12 sided spacecraft within the Pegasus static launch envelope. The single array option installs one panel between the spacecraft and shroud dynamic envelope, while the bifold allows panels to fold back on themselves, fitting two layers within the allowed envelope. The analysis uses 3/4" thick panels and allows 1" between panels and the spacecraft for deployment mechanisms.

Table 3–5 presents the packaging of the single arrays while Table 3–6 presents the bifold array configurations. Figures 3–15 and 3–16 depict the results of the CAD analysis and calculate the range of solar array areas for the four geometric spacecraft configurations. Note that the cross-sectional area available for the spacecraft is larger for the single panel 8- and 12-sided configurations compared to the bifold versions.

Table 3–5
Maximum Possible Area Single Solar Array Panels

| S/C Sides # of Panels | Total Array Area * ($m^2$) | S/C Flat Side (Radiator) Area ($m^2$) | | S/C Volume ($m^3$) |
|---|---|---|---|---|
| | | Per Side | Total | |
| 4 | 5.44 | 1.2 | 4.87 | 0.90 |
| 6 | 5.55 | .8 | 4.82 | 1.02 |
| 8 | 5.25 | .6 | 5.07 | 1.17 |
| 12 | 5.04 | .42 | 5.12 | 1.23 |

**Note: Solar cell packing factor of 0.9 was used for the rectangular array section and 0.7 for the tapered region.*

Table 3–6
Maximum Possible Area BiFold Solar Array Panels

| S/C Sides # of Panels | Total Array Area * ($m^2$) | S/C Flat Side (Radiator) Area ($m^2$) | | S/C Volume ($m^3$) |
|---|---|---|---|---|
| | | Per Side | Total | |
| 4 | 9.95 | 1.2 | 4.87 | 0.90 |
| 6 | 9.84 | .8 | 4.82 | 1.02 |
| 8 | 9.75 | .6 | 4.85 | 1.01 |
| 12 | 7.94 | .42 | 4.78 | 1.08 |

**Note: Solar cell packing factor of 0.9 was used for the rectangular array section and 0.7 for the tapered region.*

### 3.9.2 Power Parametric

The objective of the analysis is to quantify the maximum power that a spacecraft limited by the Pegasus launch vehicle could provide to a spacecraft on a LEO mission. For this analysis, it is assumed that GaAs solar cells can generate 150 watts of orbit average power for every $m^2$ of array in LEO. This also assumes the use of an orbit rate array tracking mechanism. Volume for these mechanisms was accounted for in the array packing factor and in the mechanical spacing between array panels and the spacecraft. Using the results of Table 3–5 we see the single array packaging the maximum deployable array size would be 5.44 $m^2$ for the four sided spacecraft. This would yield ~815 watts of array output power. By comparison the bifold solar panel configuration would yield ~1492 watts orbit average panel output power. It is additionally understood that the array output is diminished by ~50 percent for useful mission power as a result of running the energy balance equation. Therefore the maximum usable mission power that can be obtained in a Pegasus launch vehicle is 1492/2 or ~750 watts. This would clearly be enough for most smallsat missions. The few exceptions that require more (refer to Table 2–11) are not feasible on a Pegasus launch vehicle.

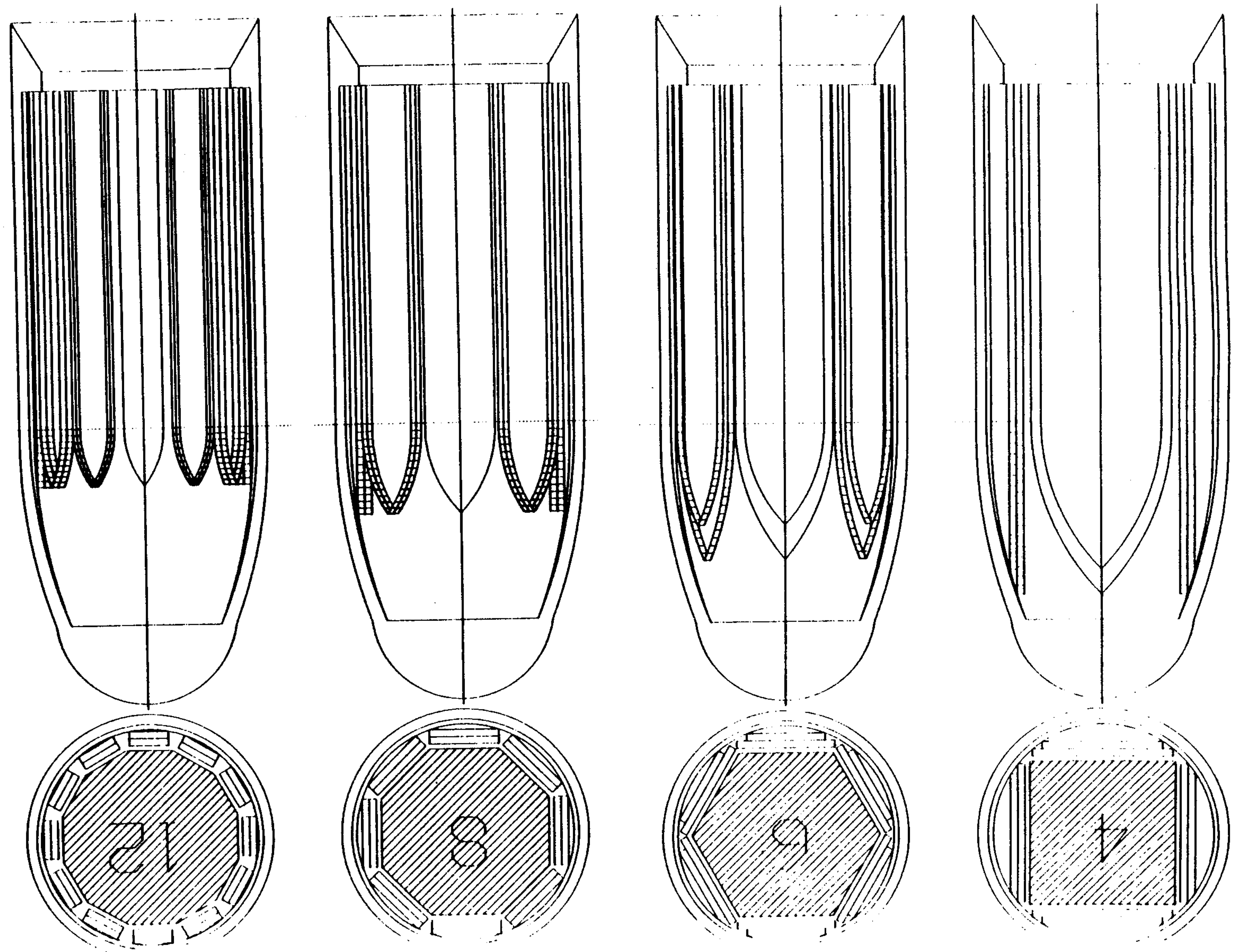



Figure 3–15. Spacecraft Configurations for Single Solar Arrays

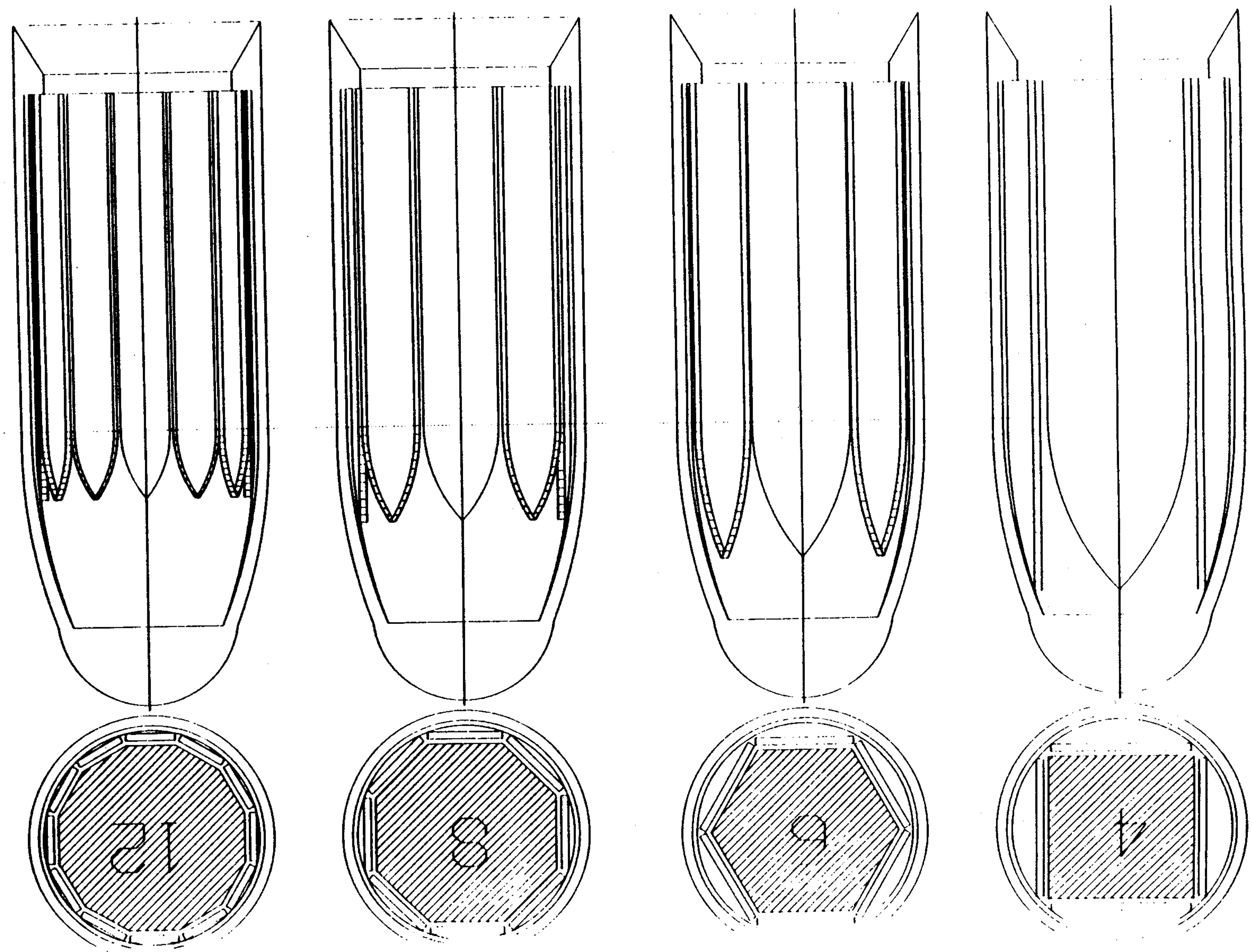


Figure 3–16. Spacecraft Configurations for BiFold Solar Arrays

### 3.9.3 Radiator Sizing Parametric

The purpose of the analysis is to evaluate the maximum power the Pegasus bounded spacecraft can handle thermally. Before using any of the output information presented in Table 3–5 we must understand the LEO thermal environment of the spacecraft. This was done by using simplified calculations using a one-meter cube spacecraft. The orbit average input environmental heat fluxes on each spacecraft face were calculated as a function of 400 km and 800 km spacecraft altitudes and three different Beta angles (0°, 45° and 90°). The resulting input fluxes are presented in Table 3–7 column #6. Next the fluxes of column #6 are converted to equivalent blackbody temperature in both Centigrade and Kelvin. See columns #7 and #8 of Table 3–7. Lastly, using the radiation temperature equation equivalent heat rejection capabilities for each spacecraft $1m^2$ surface as calculated for seven different temperatures between -10°C and +50°C. For this calculation the spacecraft side was assumed to be all radiator and painted white, thus the absorptivity was .2 and the emissivity was .8. Figures 3–17 through 3-22 graph the power rejection capability of each spacecraft surface.

Reviewing Figures 3–17 through 3–22 shows that the anti-sun side of the spacecraft is best for heat rejection to the environment. The next best side would be either the ram, wake or zenith. All three have about the same orbit average capability. The last surface to be used would be the sun side of the spacecraft. This is expected as the Sun directly impinges on this surface. The curves are provided as a standard guide for determining the radiator size during the normal design process. Our initial analysis will assume the spacecraft has a true anti-sun side. This can occur as a function of a sun synchronous orbit or the spacecraft could be maneuvered to avoid sun on one of its sides. The UARS spacecraft presently performs a yaw maneuver every 34 days to avoid sun impingement on it radiators.

Using the anti-sun spacecraft surface the heat rejection to the nominal space environment for a +40°C radiator temperature would be ~380 per $m^2$ of radiator. For the Pegasus sized spacecraft the largest surface area is provided by the four sided configuration. Using the $1.2m^2$ radiator provided by this configuration, the anti-sun surface would provide ~450 of rejection. Provided this area was available, this single surface could handle the thermal needs of most smallsat missions. But clearly if the mission reached the full solar array capability (of ~750 watts) this single surface area would not suffice. Additional radiator area would need to be located between the ram, zenith and wake surfaces. In total, these three additional surfaces all together could supply over ~900 watts of thermal heat rejection. This shows that there is sufficient radiator area to reject the ~750 watts provided to the mission by the the maximum Pegasus sized solar array.

Figures 3–17 through 3–22 also show the minimum power required to keep the radiator at temperature above about -5°C. During the cold case this minimum power would not be available, therefore louvers would probably be used to cover the radiators so as to reduce the radiator space view to 0 $m^2$. VCHPs could also be used in a similar manner to accomplish radiator shut down when it reached a predetermined low temperature. The use of louvers has the disadvantage of reducing the equivalent radiator area by 50 percent to 70 percent. This would not be a problem as the anti-sun, ram, wake and zenith spacecraft surfaces combined can reject over ~1300 watts. In the true sun synchronous orbit, each of these surfaces would have a rejection capability the same as the anti-sun surface in Figure 3-17.

### 3.9.4 Conclusions

In conclusion, it felt that the thermal system could be developed to handle the maximum power the Pegasus configured solar array can develop. It is also true that such a system would most assuredly require the use of either louvers or VCHPs to keep the spacecraft from getting too cold during the cold case orbits.

# Table 3–7
## Sun/Earth Flux Inputs and Power Dissipation Per Square Meter

| Conditions | | | Sun/Earth Inputs W/m^2 | | | Temperture | | Radiator Power Disipation W/m^2 @ °C | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Altitude (km) | Direction | Sun Beta (°) | Earth I R | Sun + Albedo | Total | °C | °K | -10 | -5 | 0 | 10 | 30 | 40 | 50 |
| 400 | RAM | 0 | 55 | 64 | 119 | -59.0 | 214.0 | 121.8 | 138.8 | 156.8 | 195.8 | 287.2 | 340.2 | 398.5 |
| 400 | NADIR | 0 | 168 | 28.6 | 196.6 | -30.3 | 242.7 | 59.7 | 76.7 | 94.7 | 133.7 | 225.1 | 278.1 | 336.5 |
| 400 | ZENITH | 0 | 0 | 86.9 | 86.9 | -75.1 | 197.9 | 147.5 | 164.5 | 182.4 | 221.4 | 312.8 | 365.9 | 424.2 |
| 400 | WAKE | 0 | 55 | 64 | 119 | -59.0 | 214.0 | 121.8 | 138.8 | 156.8 | 195.8 | 287.2 | 340.2 | 398.5 |
| 400 | -NORMAL | 0 | 55 | 7.4 | 62.4 | -90.9 | 182.1 | 167.1 | 184.1 | 202.0 | 241.0 | 332.4 | 385.5 | 443.8 |
| 400 | +NORMAL | 0 | 55 | 7.4 | 62.4 | -90.9 | 182.1 | 167.1 | 184.1 | 202.0 | 241.0 | 332.4 | 385.5 | 443.8 |
| 400 | RAM | 45 | 55 | 48.7 | 103.7 | -66.2 | 206.8 | 134.1 | 151.1 | 169.0 | 208.0 | 299.4 | 352.4 | 410.8 |
| 400 | NADIR | 45 | 168 | 23.2 | 191.2 | -32.0 | 241.0 | 64.1 | 81.1 | 99.0 | 138.0 | 229.4 | 282.4 | 340.8 |
| 400 | ZENITH | 45 | 0 | 61.5 | 61.5 | -91.5 | 181.5 | 167.8 | 184.8 | 202.8 | 241.8 | 333.2 | 386.2 | 444.5 |
| 400 | WAKE | 45 | 55 | 48.7 | 103.7 | -66.2 | 206.8 | 134.1 | 151.1 | 169.0 | 208.0 | 299.4 | 352.4 | 410.8 |
| 400 | -NORMAL | 45 | 55 | 5.2 | 60.2 | -92.5 | 180.5 | 168.9 | 185.9 | 203.8 | 242.8 | 334.2 | 387.2 | 445.6 |
| 400 | +NORMAL | 45 | 55 | 131 | 186 | -33.7 | 239.3 | 68.2 | 85.2 | 103.2 | 142.2 | 233.6 | 286.6 | 344.9 |
| 400 | RAM | 90 | 55 | 0 | 55 | -96.5 | 176.5 | 173.0 | 190.0 | 208.0 | 247.0 | 338.4 | 391.4 | 449.7 |
| 400 | NADIR | 90 | 168 | 0 | 168 | -39.7 | 233.3 | 82.6 | 99.6 | 117.6 | 156.6 | 248.0 | 301.0 | 359.3 |
| 400 | ZENITH | 90 | 0 | 0 | 0 | -273.0 | 0.0 | 217.0 | 234.0 | 252.0 | 291.0 | 382.4 | 435.4 | 493.7 |
| 400 | WAKE | 90 | 55 | 0 | 55 | -96.5 | 176.5 | 173.0 | 190.0 | 208.0 | 247.0 | 338.4 | 391.4 | 449.7 |
| 400 | -NORMAL | 90 | 55 | 0 | 55 | -96.5 | 176.5 | 173.0 | 190.0 | 208.0 | 247.0 | 338.4 | 391.4 | 449.7 |
| 400 | +NORMAL | 90 | 55 | 268 | 323 | 1.7 | 274.7 | -41.4 | -24.4 | -6.4 | 32.6 | 124.0 | 177.0 | 235.3 |
| 800 | RAM | 0 | 41.5 | 67.9 | 109.4 | -63.4 | 209.6 | 129.5 | 146.5 | 164.4 | 203.4 | 294.8 | 347.9 | 406.2 |
| 800 | NADIR | 0 | 150 | 29.3 | 179.3 | -35.9 | 237.1 | 73.6 | 90.6 | 108.5 | 147.5 | 238.9 | 291.9 | 350.3 |
| 800 | ZENITH | 0 | 0 | 85 | 85 | -76.2 | 196.8 | 149.0 | 166.0 | 184.0 | 223.0 | 314.4 | 367.4 | 425.7 |
| 800 | WAKE | 0 | 41.5 | 67.9 | 109.4 | -63.4 | 209.6 | 129.5 | 146.5 | 164.4 | 203.4 | 294.8 | 347.9 | 406.2 |
| 800 | -NORMAL | 0 | 41.5 | 5.6 | 47.1 | -103.2 | 169.8 | 179.3 | 196.3 | 214.3 | 253.3 | 344.7 | 397.7 | 456.1 |
| 800 | +NORMAL | 0 | 41.5 | 5.6 | 47.1 | -103.2 | 169.8 | 179.3 | 196.3 | 214.3 | 253.3 | 344.7 | 397.7 | 456.1 |
| 800 | RAM | 45 | 41.5 | 54.2 | 95.7 | -70.3 | 202.7 | 140.5 | 157.5 | 175.4 | 214.4 | 305.8 | 358.8 | 417.2 |
| 800 | NADIR | 45 | 150 | 28 | 178 | -36.3 | 236.7 | 74.6 | 91.6 | 109.6 | 148.6 | 240.0 | 293.0 | 351.3 |
| 800 | ZENITH | 45 | 0 | 60.3 | 60.3 | -92.4 | 180.6 | 168.8 | 185.8 | 203.7 | 242.7 | 334.1 | 387.1 | 445.5 |
| 800 | WAKE | 45 | 41.5 | 54.2 | 95.7 | -70.3 | 202.7 | 140.5 | 157.5 | 175.4 | 214.4 | 305.8 | 358.8 | 417.2 |
| 800 | -NORMAL | 45 | 41.5 | 4 | 45.5 | -104.7 | 168.3 | 180.6 | 197.6 | 215.6 | 254.6 | 346.0 | 399.0 | 457.3 |
| 800 | +NORMAL | 45 | 41.5 | 141.6 | 183.1 | -34.6 | 238.4 | 70.5 | 87.5 | 105.5 | 144.5 | 235.9 | 288.9 | 347.3 |
| 800 | RAM | 90 | 41.5 | 0 | 41.5 | -108.5 | 164.5 | 183.8 | 200.8 | 218.8 | 257.8 | 349.2 | 402.2 | 460.5 |
| 800 | NADIR | 90 | 150 | 0 | 150 | -46.2 | 226.8 | 97.0 | 114.0 | 132.0 | 171.0 | 262.4 | 315.4 | 373.7 |
| 800 | ZENITH | 90 | 0 | 0 | 0 | -273.0 | 0.0 | 217.0 | 234.0 | 252.0 | 291.0 | 382.4 | 435.4 | 493.7 |
| 800 | WAKE | 90 | 41.5 | 0 | 41.5 | -108.5 | 164.5 | 183.8 | 200.8 | 218.8 | 257.8 | 349.2 | 402.2 | 460.5 |
| 800 | -NORMAL | 90 | 41.5 | 0 | 41.5 | -108.5 | 164.5 | 183.8 | 200.8 | 218.8 | 257.8 | 349.2 | 402.2 | 460.5 |
| 800 | +NORMAL | 90 | 41.5 | 268 | 309.5 | -1.2 | 271.8 | -30.6 | -13.6 | 4.4 | 43.4 | 134.8 | 187.8 | 246.1 |

Front/Rea Absorption = 0.2   Emission = 0.8   I R = 237W/m2   Sun = 1353W/m2   Inclination = 75°

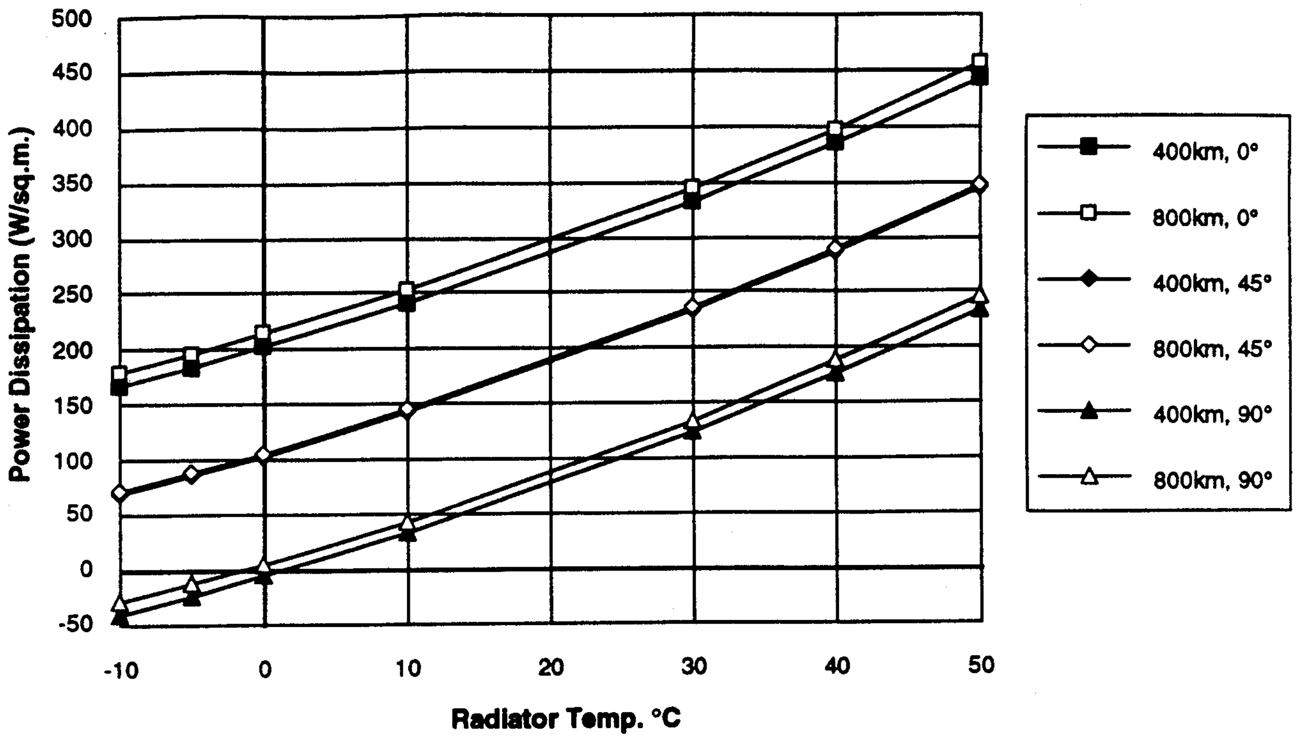



Figure 3–17. Radiator Dissipation (+Normal)

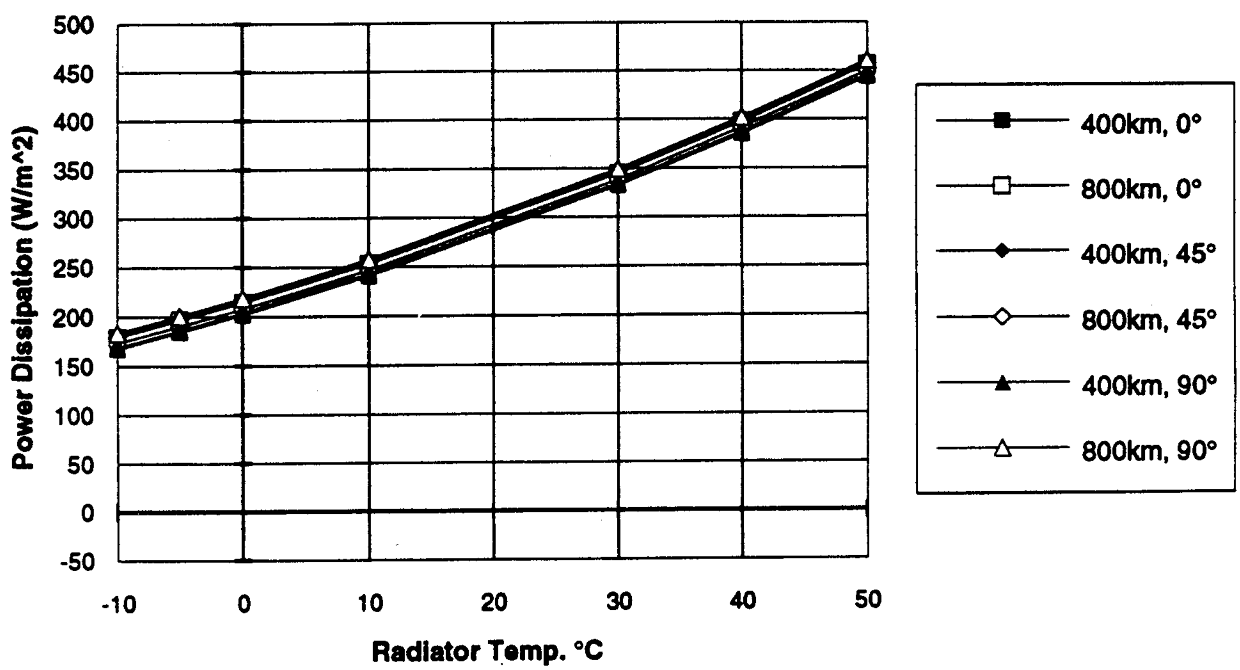



Figure 3–18. Radiator Dissipation (-Normal)

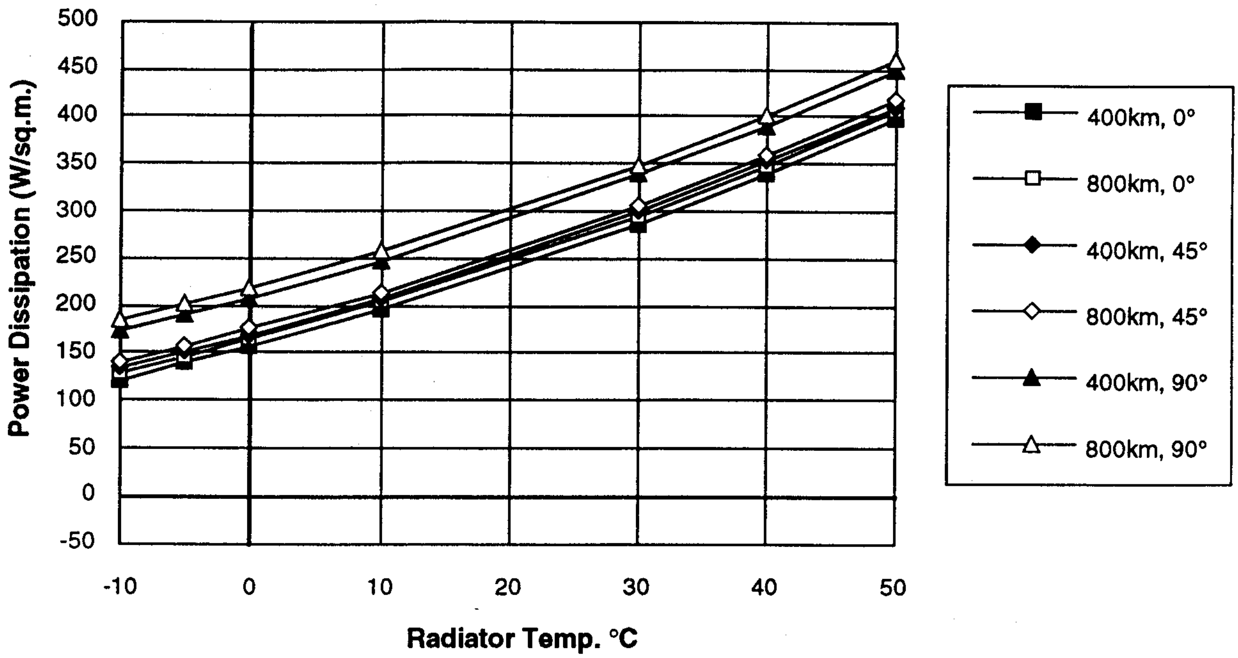



Figure 3–19. Radiator Dissipation (Ram)

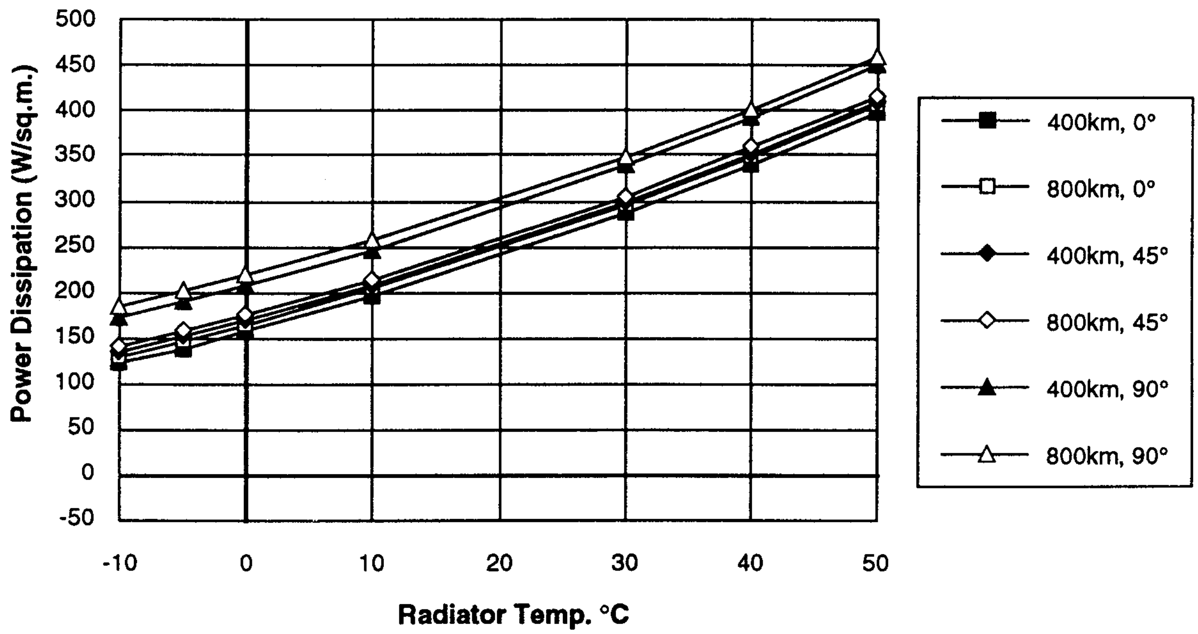



Figure 3–20. Radiator Dissipation (Wake)

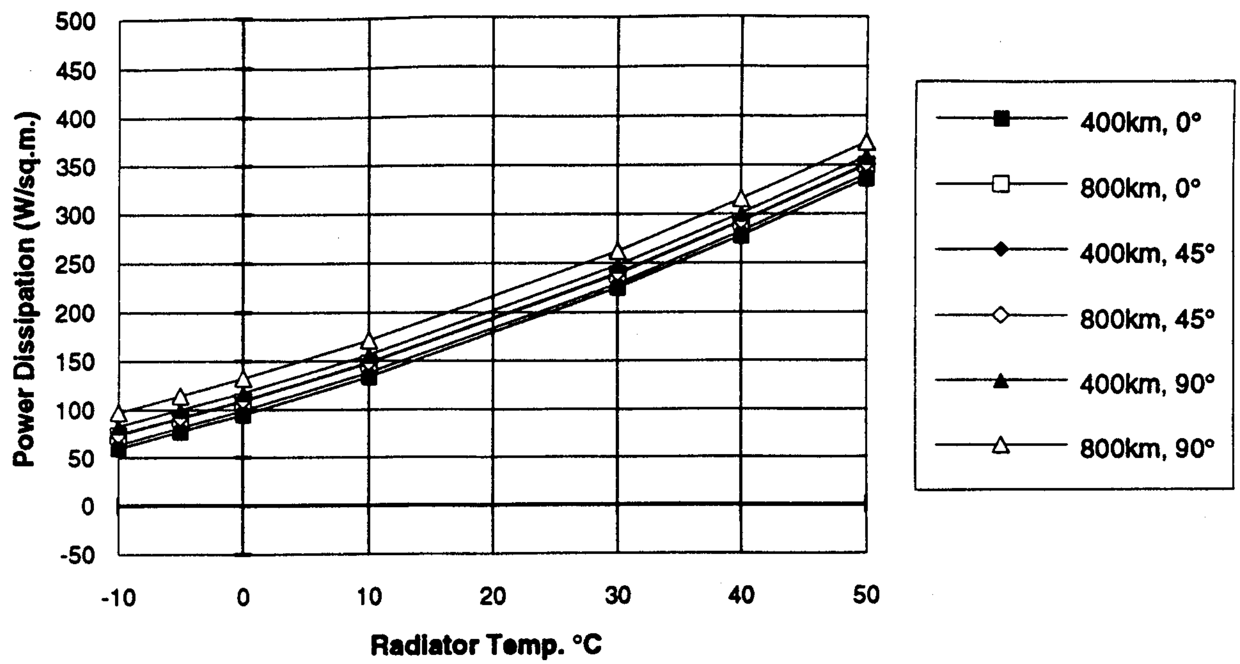



Figure 3–21. Radiator Dissipation (Nadir)

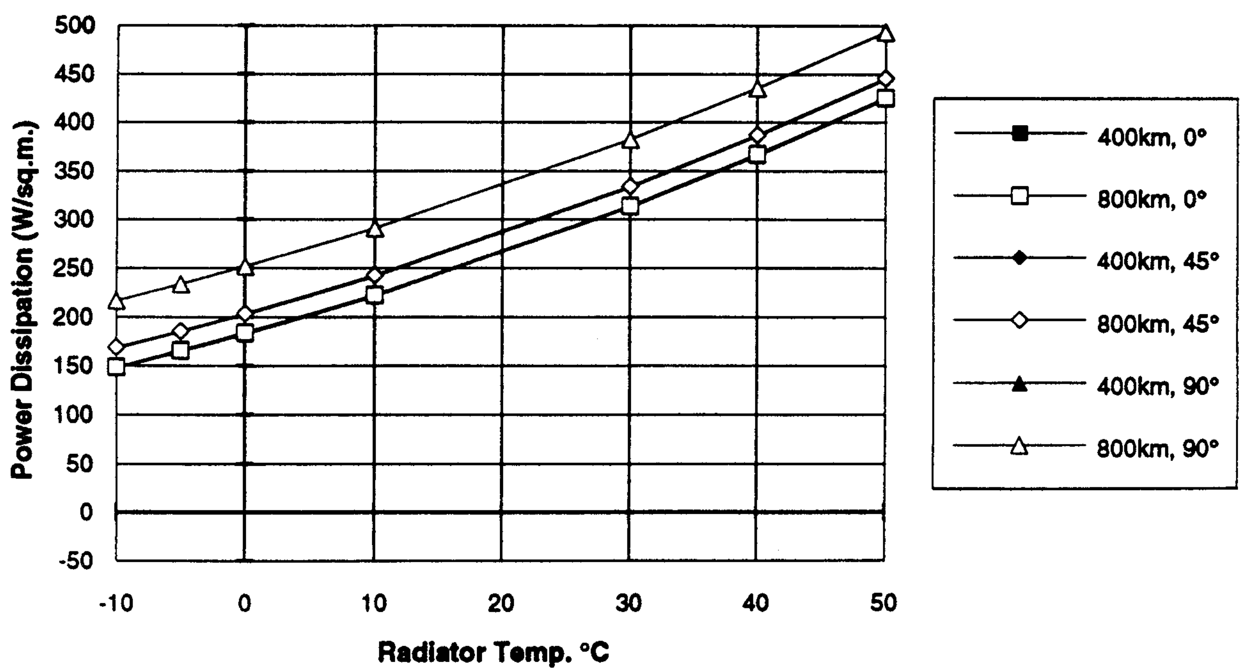



Figure 3–22. Radiator Dissipation (Zenith)

# 4. CONCLUSIONS AND RECOMMENDATIONS

The primary engineering reason used to select a smallsat subsystem technology or design aspect is the performance/mass ratio. But ultimately, the real underlying driver is cost, which is closely related to mass because of launch vehicle constraints. For very small satellites, the launch vehicle costs can be several times the cost of the satellite, so living with the launch vehicle limitations is vital to controlling program costs. Cost will make or break a spacecraft system.

As performance requirements increase while pressure remains to keep the hardware small, the temptation is to add expensive new technology to achieve the desired performance. However, there remains a concurrent need to reduce unit costs so as to make the system affordable. In the long run, the affordability/benefit trade-off underlies the decision for or against the mission. Viewed from the subsystem level, both aspects of the tradeoff can be supported. Affordability can be supported thorough reduction of total subsystem hardware costs, and mission benefits can be supported through improved performance or new (enabling) capabilities that the subsystem provides for the payload.

## 4.1 REQUIREMENTS

The design process of most spacecraft to date has been driven by very specific performance requirements per the science (payload) needs. From this initial design phase, the total cost could be estimated. If costs exceeded budget, some of the requirements might be reduced, an instrument removed, or the life or reliability reduced. Now it is becoming more common, and more practical, for programs to begin with a fixed cost budget and more flexible mission requirements. Instead of designing to a specific set of performance values, the engineer now must design to specific costs goals. The objective of this initial design is now to determine the specific performance *capabilities*, not to determine the total cost.

Changing to a performance-driven design process from the more traditional requirements-driven approach is proving to be a difficult process for the aerospace community. Yet is must be dealt with, as is an unavoidable outcome of the limited budget environment that is now prevalent. Instead of using whatever funds necessary to build a spacecraft to perform a particular mission, the process now becomes one of designing a generic spacecraft (or using an off-the-shelf design, or some similar modular, expandable approach) to determine what sort of mission can be supported on the given budget. "You take what you can get" with the resources that are made available.

This requires sharp engineering judgment to obtain the best performance as a function of mass and cost. Selecting the proper technology and setting an appropriate management style is key. An example of how management influences this is by keeping the requirements modest. Planning a small satellite mission with large numbers of power-hungry payloads will only lead to problems. This drives up the array and battery size and stresses the thermal design. Specifically, this makes the array more expensive, not just from a raw area point of view, but from the stowage, deployment, and orientation devices that go with a large array. A large battery is not expensive in itself, but it drives other factors like thermal control, and can be the major driver in the overall thermal design.

Requirements based on simultaneous combinations of all worst-case conditions should not be imposed. The subsystems should be designed for end-of-life performance, but it is very likely that all of the worst-case conditions never actually occur at the same time. It is also possible that one or more of these conditions could be relaxed, especially in coordination with another. For example, a decision could be made that load requirement (or duty cycle) could be reduced at the worst beta angles. This has enormous impact on the solar array size. Appropriate selection of the launch date can often be made to achieve the most favorable EOL conditions.

Realistic specific temperature requirements should be used. The definition of hot and cold cases should combine reasonable environment input fluxes with electrical load conditions. Use of combined worst case limits burdened with additional margins can lead to requirements creep and over-design. The thermal subsystem can be very forgiving when running hot if the payloads can be duty cycled. If running too cold, heaters are an easy design solution to correct this condition.

## 4.2 PROGRAMMATIC ISSUES

Before the engineer can even begin the design job, the program manager must make several decisions that influence the design process.

### 4.2.1 Risk

The battery performance values described in this report are based on conservative life predictions accepted by NASA GSFC engineers based on available test data and prior experience. Performance predictions from battery manufacturers based on their analysis and the design features they have included will almost always be considerably better. However, their information is not readily confirmed because of proprietary restrictions. The same is true of many of the subsystem components. Management could accept the risk by using a higher battery DOD. Too conservative a design leads to an oversized battery, which increase the mass and cost. One could accept a shorter life, or a short life at a reasonable probability of success with an extended mission at a lower probability, rather than a long mission that has built-in guarantees of long life.

### 4.2.2 Redundancy

Redundancy is closely associated to risk. This topic has already been discussed in terms of selective redundancy in electronics and in the discussion of multiple versus single batteries. Note that multiple does not necessarily imply redundant. Just because there are two batteries or two solar array panels, does not mean they have to be treated as redundant devices and be completely cross-strapped. Some cross-strapping may be desirable and can be done with very little cost impact. All components should not be designed as functionally redundant. The second wing or battery provides a fail-safe or fail-operational capability.

### 4.2.3 Timely Decision Making

An example of decisiveness is to determine early in the spacecraft design phase if the thermal subsystem is to be active or passive. Then the design process should be submitted to a systems analysis. The system engineer and the thermal subsystem engineer can often intuitively define the best design approach. Mandate this approach, thus eliminating time, manpower and dollars to perform trade-offs when the answer is already known. Clear direction as early as possible eliminates the need for unnecessary trades, vacillation and confusion.

### 4.2.4 Paperwork

Strive to have the paperwork weigh less than the spacecraft.

## 4.3 DESIGN AND TECHNOLOGY ISSUES

Deciding what level of complexity or which off the shelf subsystem and components to use is not an easy task. For smallsats, the seemingly obvious route is to select the simplest approach. Generally, one expects that this results in the lowest cost and mass.

However, a simple component or design approach may not always be the best. For example, if a totally passive thermal design is baselined, it would appear to have low hardware costs. But such a system requires considerable engineering and testing including a costly thermal balance test. If a cold bias system is used, however, the analysis can probably be reduced, the system will be forgiving and robust, and the thermal testing could be greatly reduced. While the flight hardware might be more expensive, this approach reduces the overall life-cycle (program) costs.

Starting in the 1980s, the thermal-design philosophy became one of biasing the design to the cold end of the permissible range and using heaters to adjust the temperature upward. This change became possible because heater power was now a small percentage of total power requirement on a large spacecraft. It permits economically significant shortcuts in the amount of expensive testing required as the system is more robust. Heat-balance tests have also become simpler, with solar simulation and vacuum environment usually not necessary. For small satellites, this approach may not be possible due to mass and power budgets. However, it is usually the most cost effective approach and should be used to minimize the small satellite cost wherever possible.

Another basic design decision involves the use of standardized components. The EPS design might call for a 15 ampere hour battery. But suppose no battery presently exists at that exact size, but a space-qualified 19 ampere-hour design is available. It would make little sense to pay for a costly qualification program just to save three or four kilograms in battery mass. This consideration applies even for smallsats where mass is critical.

In the thermal area, the ultimate system would be one that handled from 150 to 1500 watts and was able to hold the entire spacecraft at ±5°C regardless of orbit or environment. Such a system exists. It is called a capillary pumped loop (CPL). Th CPL thermal bus accomplishes this because it operates as a saturated two phase system and contains multiple evaporators and condensers. While initial use of this bus might seem costly, consider a standard modular system which could simply have evaporators and condensers added as needed, and remain self regulating under all conditions of load and environment. Such a modular system could be built by menu, and because it is self regulating, much of the engineering design and ground testing could be eliminated.

### 4.3.2 Impact of Improved Technology

The trend in spacecraft from the beginning of the space program to the late 1980s has been towards larger and more complex spacecraft. This trend has been largely driven by the desire to fly scientific instruments of ever-increasing size, complexity, and sensitivity and to fly them in combination to make simultaneous measurements. Since total program costs are driven up by these large and complex satellites, the emphasis is now to develop space missions with more tightly focused objectives that can be achieved quickly and cheaply on smaller satellites and launch vehicles.

These trends are logically reflected in corollary trends in the spacecraft subsystems, including the EPS. The trend toward larger spacecraft has led to the development of large power subsystems capable of accommodating multi-kilowatt loads. In addition to total load capability, power quality (better regulation, less noise, less EMI/RFI), and total system design have also become required characteristics. Now that smallsats have become a recognized option, power designers are looking for ways to squeeze every last bit of performance out of each kilogram of hardware.

Up to the present time, improved performance in the power system has been accomplished primarily by using more or larger components. Recently, actual improvements in the designs, components, and the devices used have been implemented leading to increased system performance. These improvements include developments such as GaAs solar cells and single

pressure vessel nickel hydrogen batteries. To achieve continued improvements, increased use of new technology and alternative concepts must be explored.

4.3.3 Recommended Efforts

Because smallsats generally use the same technology as larger spacecraft, there are few specific technologies that will prove exclusively valuable to smallsats. Certainly, any effort to reduce the cost of existing technologies is laudable, even if the resulting system has the same performance. The same performance for fewer dollars is a very worthwhile goal. High cost items in the power and thermal area include solar arrays (which involve a considerable amount of touch labor) and heat pipes. Reducing the size of components while maintaining their current performance will also help smallsats. As has been noted earlier, mass very directly drives cost.

There are no enabling technologies on the near horizon that will generate any major breakthroughs in the smallsat market. Today's thermal subsystem elements have been shown to be able to handle the maximum heat dissipation that can be generated by the largest array that will fit in a small launch vehicle envelope. The power subsystem components (batteries, solar cells) are not likely to see huge performance increases. Small battery performance improvements can be squeezed from current NiCd and $NiH_2$ technologies, but NiMH batteries, once qualified for space use, promise nearly a factor of two improvement in energy density. Therefore, development of NiMH batteries for space use is recommended as a priority for both large and small satellites.

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# APPENDIX A

# CRYOGENICS

## A. APPENDIX A – CRYOGENICS

This section is provided to show the state-of-the-art in cryogenic coolers. Each type of cryo cooler will be identified, and its operation and some of its salient characteristics will be explained. The information is based on general knowledge from the GSFC cryogenics group.

### A.1 CRYOGENIC REFRIGERATION

Cryogenic refrigeration is usually classified within two categories: open cycle and closed cycle. Figure A-1 shows the numerous cryogenic systems that can be used to develop cryogenic sinking capabilities. For space applications, these systems use only the change of pressure of the coolant to create cooling cycles and do not use any phase change of the coolant. The section will mainly address dynamic regenerative systems as they offer the best hope of long life in space applications.

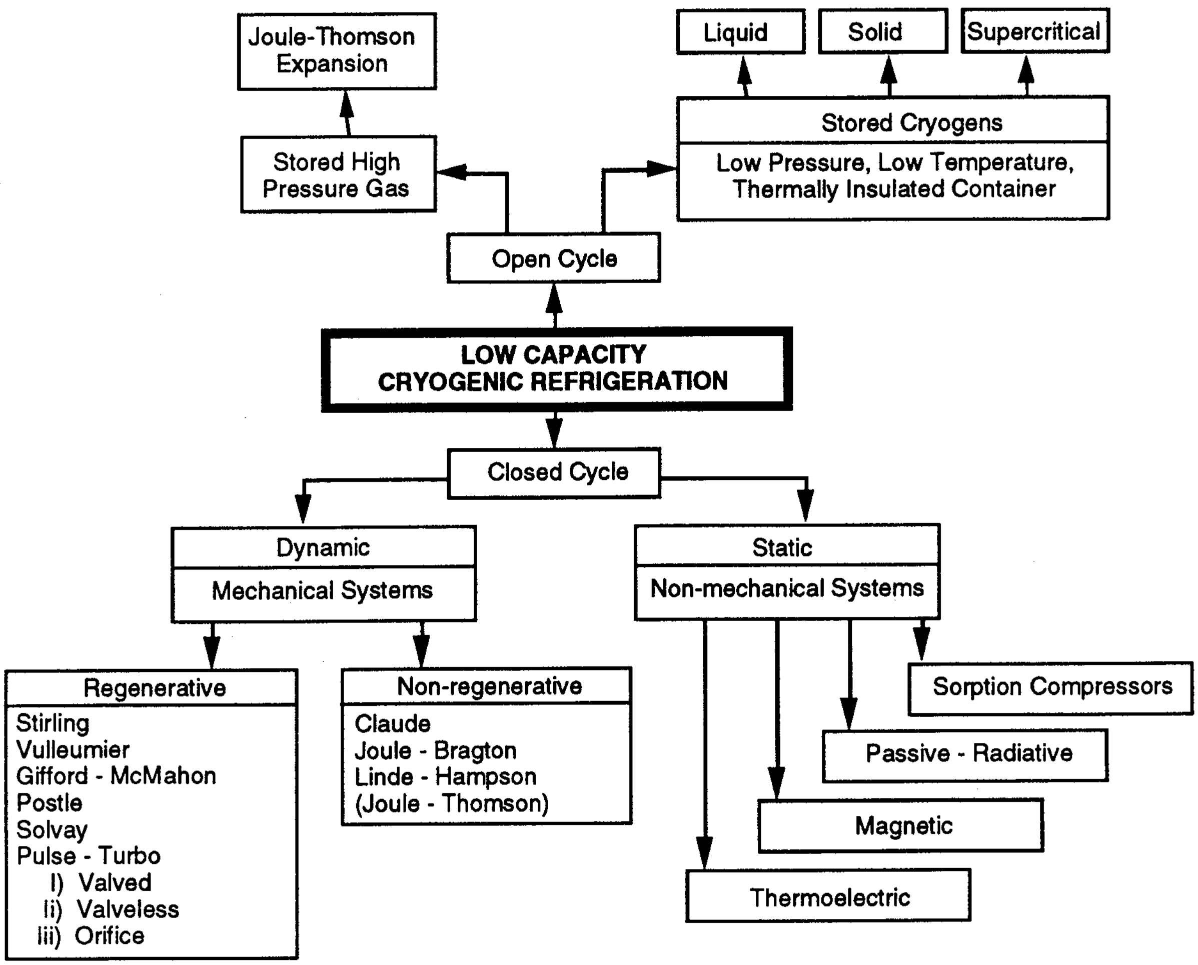


Figure A–1. Classification of Low Capacity Cryogenic Refrigerators

#### A.1.1 Stirling cycle

This system uses two pistons and is shown in Figure A-2.

The pistons are operated 90° out of phase. There are four cycles for one stirling phase. The first piston, located at the heat sink, is called the compressor piston, and the piston located at the heat

source is called the displacer. The system also has a regenerator matrix section. This section acts to impede the instantaneous pressure change between the compressor and displacer piston volumes, and in doing so, absorbs heat. When the high pressure is generated by the compressor piston, heat is removed from the system at its heat sink. Maximum system heat is removed during this compressor compression cycle. However, testing has shown that all the heat will be dissipated at displacer if it is allowed to run cooler than the compressor—so proper heat sinking of the compressor is imperative. The cold finger may have its payload directly attached or a cold strap could be used. The cold strap has the advantage of reducing the mechanical vibrations caused by the two system compressors. Often the regenerator matrix is combined with the displacer as shown in Figure A–3.

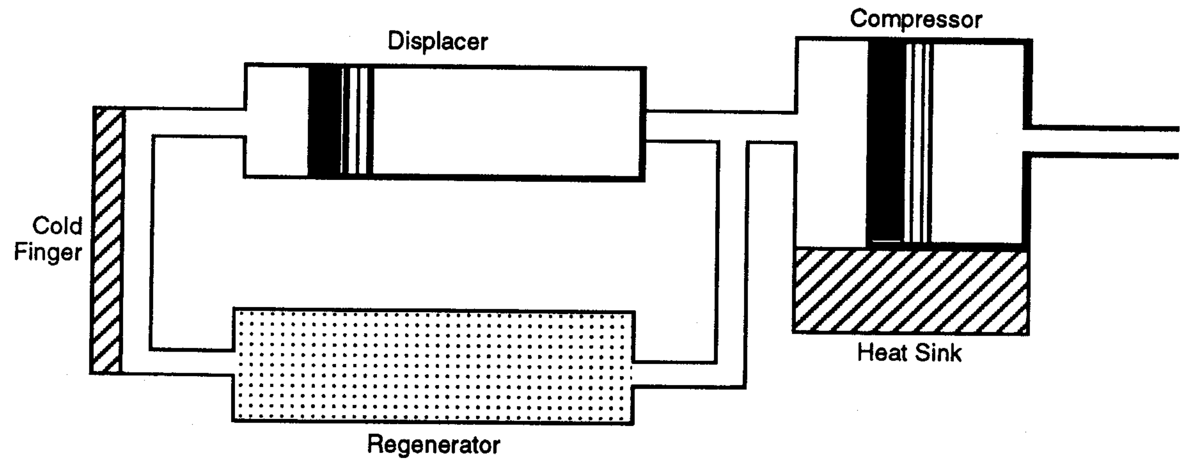


Figure A–2. Stirling Cycle

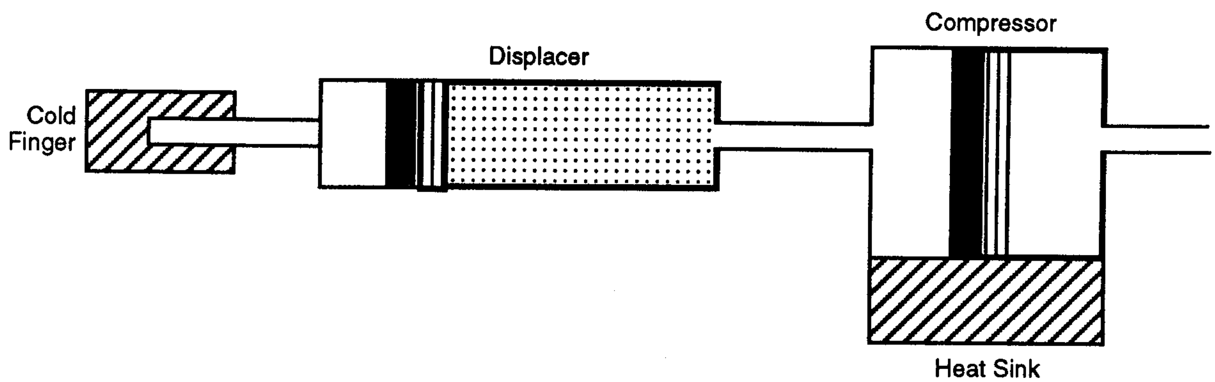


Figure A–3. Displacer

Presently 80K Stirling cycle systems require ~30 watts per watt of cooling, and 65K Stirling Cycle systems require ~40 watts per watt of cooling. Stirling cycle one-stage coolers typically do not go below 65K.

NASA's goal for the life of a satellite's cryogenic refrigeration system is 50,000 hours. Present day technology permits 5,000 hrs. System disturbances due to compressor vibration can be reduced by using counterbalances which operate 180° out of phase with the two system compressor pistons. Several systems exist that use piston counter balancers to minimize vibration.

A.1.2 Pulse Tube

This system, shown in Figure A-4, unlike the Stirling cycle system, has no moving parts operating in the cold part of the system. This system uses one compressor to remove the bulk of the heat. The displacer is replaced by a tube (that is analogous to a traveling wave tube) that is connected to a coolant reservoir. A cryogenic cold finger is located at the regenerator and pulse tube interface. This system has the same regenerative cooler operations as a Stirling cycle system. The pulse tube efficiency is also comparable.

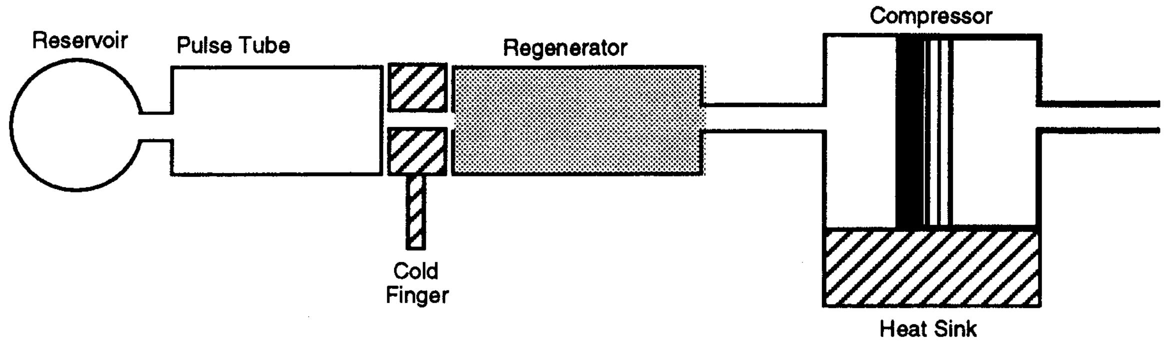


Figure A–4. Pulse Tube

This system may offer the best hope of a long life because it does not have moving parts at cryogenic temperatures and is definitely compatible with smallsat applications.

| |
|---|
| The simple, reliable design of a pulse tube compressor makes it a good candidate for a smallsat cryogenic refrigerator. |

A.1.3 Reverse Brayton

This system is shown in Figure A–5.

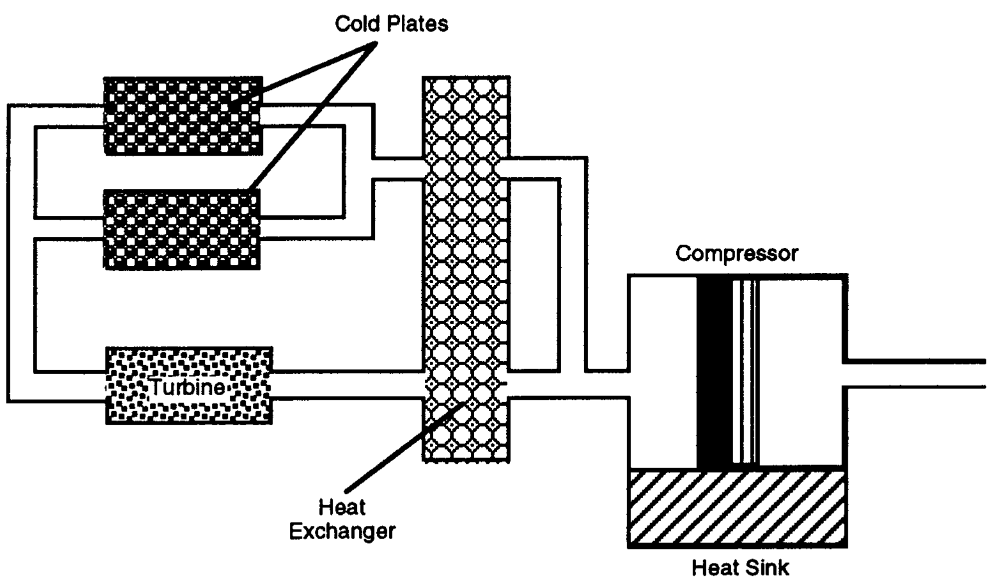


Figure A–5. Reverse Brayton

The Reverse Brayton cycle cryogenic system consists of a compressor, a heat exchanger and a gas bearing turbine. Presently these systems can provide 5 watts of cooling at 65K. This type of system forces the high pressure gas through a turbine before it expands. During the compression cycle, heat is given up at the compressor sink. This system is comparable to the Stirling cycle system in power and weight efficiency. Potential problems for long life are recognized because the turbine runs on a gas bearing. Bearing touch down and start up are the concern. This type of system can be configured to cool multiple cold plates because the coolant flow path is a continuous loop.

Miniature Reverse Brayton systems also exist. These systems typically are a total of 60-90 thousands of an inch in diameter, including the turbine. Such systems produce only a few milliwatts of cooling at cryogenic temperatures.

A.1.4 Joule Thomson

1. The Joule Thomson closed loop system consists of a compressor and an expansion chamber. Single-stage Joule Thomson closed loop systems can attain 30-40K at 3-5 watts rejection while double-stage Joule Thomson can attain 4K at ~10 milliwatts. The first stage of such a system is typically a Stirling cycle system which is used to heat sink the second Joule Thomson at ~50-65K. This first stage would require ~100 watts to operate. Closed loop Joule Thomson systems do not apply to small satellite systems because they are very heavy, and require over 150 watts to operate.

2. The Open-looped Joule Thomson does not have a compressor. It allows the coolant gas to expand at a controlled rate and to escape to the ambient environment. This system can provide 5K to 10K temperature at milliwatts rejection for the limited life of the coolant gas. This limited life is directly proportional to the coolant gas storage container size. A 20- to 30-liter dewar supply of helium could provide up to several hours of cryogenic operation. This mass of the dewar and short lifetime makes this type of system impractical for use on small satellites.

A.1.5 Magnetic refrigeration

Magnetic refrigeration uses the magnetic properties of superconductors to reduce temperature. The process better known as "Adiabatic Demagnetization Refrigeration" (ADR) alternately magnetizes and demagnetizes paramagnetic salts materials to attain cryogenic capabilities of ~1K at .5 milliwatts rejection. The system can hold the cryogenic temperature for 19 hours before it needs a refresh, which takes 1 hour. This system requires ~1 to 10 watts for the refresh. The ADR second stage must be heat sinked to a low temperature cryogenic first-stage system, such as an expendable helium dewar. Such a system would require a 30 to 40 liters of helium for a reasonable 12-18 months lifetime. Its mass is more than 200 kg, making it impractical for use on small satellites.

## A.2 SUPPLIERS OF CRYOGENIC COOLERS

Suppliers of cryogenic coolers are listed in Table A–1. The table presents each company's cooler, its type and capabilities, and the approximate cost per unit. Only systems that require less than 65 watts are applicable for smallsats.

Table A–1
Suppliers of Cryogenic Coolers

| Supplier | Space | Life | Input (W) | Output (W) | Mass (Kg) | Price | Type of Cooler |
|---|---|---|---|---|---|---|---|
| Creare | X | 10 yrs | 200 | 5 | 15 | 1-2M | 65° R. Brayton |
| TRW | X | 5 yrs | 200 | 1 | 15 | 1.1M | 35° Pulse - 2 comp |
| Sunpower | COM | 5 yrs | 120 | 4 | 4-6 | 10K | 80° Stirling |
| Creare | X | 10 yrs | 100 | 2 | 10 | 1-2M | 60° R. Brayrston |
| Mitsubishi, Fu | X | 5 yrs | 90 | 1.2 | 10 | 1-1.5M | 70° Stirling |
| Ball, BAE | X | 5 yrs | 80 | .35 | 13-26 | 1-1.5M | 40° Stirling- 2 comp, 2 stage |
| Ball, BAE | X | 5 yrs | 80 | 1.5 | 13-15 | 1-1.5M | 55° Stirling- 1 or 2 stage, 2 comp |
| TRW | X | 5 yrs | 80 | 1.5 | 10 | 1M | 55° Pulse - 2 comp |
| Hughes (flew 6/94) | X | 5 yrs | 65 | 1 | 18 | 1.5M | 65° Stirling |
| Lockheed | X | 5 yrs | 45 | 1.5 | 12 | 1.5M | 80° Stirling |
| BAE | X | 5 yrs | 40 | 1.2 | 10-15 | 1.5M | 80° Stirling |
| Hughes TI, Magnavox | | 3 mths | 35 | 1 | 3 | 15K | 80° Stirling- Tactical |
| TRW | X | 5 yrs | 30 | .4 | 5-6 | 500K | 80° Pulse |

# APPENDIX B

# CONCLUSIONS AND OBSERVATIONS

## B. APPENDIX B – CONCLUSIONS AND OBSERVATIONS

This appendix lists the conclusions and observations related to design issues particularly relevant to small satellites. These were highlighted by boxed text in the main body of this report. The information has been categorized by topic.

### B.1 GENERAL

- Project managers must take an active role in maintaining and controlling realistic performance requirements in order to minimize cost and complexity.
- Smallsat costs and complexity are contained by designing according to the subsystems' available capabilities rather than by unconstrained payload requirements.
- The relatively fixed load requirement of the spacecraft bus leaves proportionally less power available for payloads on small satellites with a low total load power capability.

### B.2 POWER PROCESSING

- A simple full linear shunt regulator, while somewhat inefficient, is a viable power processing candidate for small satellites with very low load power requirements.
- Several recent small NASA GSFC spacecraft use a sequential, staged linear shunt regulator for power processing.
- A digital shunt regulator improves its efficiency by disconnecting solar array sections that are not required to satisfy the load requirement.
- The development of microprocessor-controlled peak power tracking charge control circuits has allowed this approach to be competitive with otherwise simpler direct energy transfer power processing techniques.
- Power processing electronics using high efficiency circuits and robust designs will provide more performance and reliability benefits for smallsats than the use of completely redundant units.
- A centralized voltage regulator could be a viable option on smallsats if power distribution run lengths are short enough to minimize voltage drops and noise.
- Instrument payloads requiring a negative bus are most likely to be flown on smallsats, and will require careful trade studies to meet this unique requirement in a cost-effective manner.
- Although their relatively low total load power use suggests that high voltage power distribution offers few benefits to smallsats, a thorough trade study on this topic could identify any potential mass or cost savings.

### B.3 SOLAR ARRAYS

- The impact of atmospheric drag on solar array configurations is a significant issue for spacecraft in low orbits.
- As long as atmospheric drag is not an issue, a simple deployed, fixed, solar array panel is often the simplest design solution for smallsats.

- The limited size of small satellites makes it more difficult to avoid interference between payload fields-of-view and solar array and other appendage geometry.
- Solar array degradation is negligible for circular Earth orbits of 500 km or less, regardless of cell type.
- While the radiation tolerance of GaAs cells offers little advantage for most smallsats, its high efficiency results in smaller, lighter, and easier-to-package solar arrays.
- Gallium arsenide solar cells offer a significant reduction in array area compared to silicon, but careful cost trade-offs must be made before a final cell selection is made.
- New multi-bandgap solar cells could be very beneficial to small satellites with severe constraints on solar array size and mass.
- Solar array panel substrates made of lightweight composite materials offer mass savings for smallsats.
- The use of flexible substrates offers the possibility of significant solar array panel mass reduction.
- Solar array configuration and area, precious commodities on smallsats, can be optimized by evaluating the power subsystem's predicted performance over the duration of the mission, thus identifying more realistic worst-case conditions on which to set sizing requirements.
- Future smallsats could achieve a total load capability of around 400 W with reasonable sized solar arrays.

## B.4 BATTERIES

- The lower battery capacity typical of smallsats cannot accommodate extended deployment mode operations prior to solar array deployment and attitude stabilization.
- More efficient battery use can be achieved by simply increasing the maximum allowable depth-of-discharge.
- Small nickel hydrogen batteries have been developed specifically for small satellite applications and have been used successfully.
- Heat pipes can be used to equalize the temperature of two batteries when they cannot be located in thermally identical environments, as is usually the case with small satellites.
- Improvements to current cell technologies and the introduction of new battery types, such as nickel metal hydride, should offer significant improvement in battery energy density.

## B.5 THERMAL

- The limited thermal mass and unique launch vehicle characteristics associated with smallsats makes aerodynamic heating during boost an area of concern for small satellites.
- Heat rejection, especially for low-temperature sources, can be a significant problem for smallsats because of the limited area available for radiators.

- Existing power and thermal subsystem technologies are more than capable of meeting the performance requirements of the largest solar array-based power subsystem that can be packaged into a Pegasus-class launch vehicle.

# REPORT DOCUMENTATION PAGE

*Form Approved*
*OMB No. 0704-0188*

Public reporting burden for this collection of information is estimated to average 1 hour per response, including the time for reviewing instructions, searching existing data sources, gathering and maintaining the data needed, and completing and reviewing the collection of information. Send comments regarding this burden estimate or any other aspect of this collection of information, including suggestions for reducing this burden, to Washington Headquarters Services, Directorate for Information Operations and Reports, 1215 Jefferson Davis Highway, Suite 1204, Arlington, VA 22202-4302, and to the Office of Management and Budget, Paperwork Reduction Project (0704-0188), Washington, DC 20503.

| 1. AGENCY USE ONLY (*Leave blank*) | 2. REPORT DATE | 3. REPORT TYPE AND DATES COVERED |
|---|---|---|
| | December 1994 | Contractor Report |

**4. TITLE AND SUBTITLE**
Small Spacecraft Power and Thermal Subsystems

**5. FUNDING NUMBERS**
C NAS1-19244
TA 15
WU 477-50-00

**6. AUTHOR(S)**
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**8. PERFORMING ORGANIZATION REPORT NUMBER**

**9. SPONSORING / MONITORING AGENCY NAME(S) AND ADDRESS(ES)**
NASA Langley Research Center
Hampton, VA 23681-0001

**10. SPONSORING / MONITORING AGENCY REPORT NUMBER**
NASA CR-195029

**11. SUPPLEMENTARY NOTES**
Langley Technical Monitor: Jeffery T. Farmer
Final Report

**12a. DISTRIBUTION / AVAILABILITY STATEMENT**
Unclassified - Unlimited
Subject Category 20

**12b. DISTRIBUTION CODE**

**13. ABSTRACT** *(Maximum 200 words)*

This white paper provides a general guide to the conceptual design of satellite power and thermal control subsystems with special emphasis on the unique design aspects associated with small satellites. The operating principles of these technologies are explained and performance characteristics of current and projected components are provided. A tutorial is presented on the design process for both power and thermal subsystems, with emphasis on unique issues relevant to small satellites. The ability of existing technology to meet future performance requirements is discussed. Conclusions and observations are presented that stress cost-effective, high-performance design solutions.

**14. SUBJECT TERMS**
Power subsystem; thermal subsystem; small satellites; lightsats

**15. NUMBER OF PAGES**
125

**16. PRICE CODE**
A06

| 17. SECURITY CLASSIFICATION OF REPORT | 18. SECURITY CLASSIFICATION OF THIS PAGE | 19. SECURITY CLASSIFICATION OF ABSTRACT | 20. LIMITATION OF ABSTRACT |
|---|---|---|---|
| Unclassified | Unclassified | Unclassified | UL |