Power, Thermal and Environment

Brandon Brown-Webster
Rogan Flowers
Peter Grossman
Richard Park
Stephen Pratt
Joseph Sanderson
Kristopher Velez
Michael Weronski

November 18, 2004

Contents

List of Figures

List of Tables

List of abbreviations

AC	Alternating Current
AD& C (or D& C)	Attitude Dynamics and Control
BOL	Beginning of Life
CC& DH	Communication, Command, and Data Handling
CPU	Central Processing Unit
DC	Direct Current
DET	Direct Energy Transfer
DOD	Depth of Discharge
EOL	End of Life
GEO	Geosynchronous (orbit)
IR	Infrared (radiation)
ISS	International Space Station
LEO	Low-Earth Orbit
PPT	Peak Power Transfer
PT& E	Power, Thermal, and Environment
RAM	Random Access Memory
S& LV	Structures and Launch Vehicles
UV	Ultraviolet (radiation)

List of Symbols

Roman	
A	Area of interaction
A	Area of the spacecraft-mounted radiators
A	Surface area of the spacecraft
A_{sa}	Solar array area
a	Attenuation
a	Device surface length
B	Local magnetic field intensity
B_0	Magnetic field at the equator on the Earth's surface, $B_0 = 0.30$ gauss
b	Device surface width
C	Total battery cell capacity
C_B	Battery capacity
c	Device depth
D	Laser objective diameter
D_γ	Dose
E	Energy
E_B	Energy density
F	Flux in power per cross-sectional area
F	Geometrical factor based on the position of the spacecraft relative to the sun
F_{albedo}	Geometrical factor based on the position of the spacecraft relative to the Earth
F_{EIR}	Geometrical factor based on position of spacecraft relative to the Earth and Sun
F_n	Neutron fluence
F_0	Fluence
F_x	X-ray fluence
f	Frequency of electromagnetic radiation
f_x	Fraction of energy emitted as X-rays
h	Convection heat transfer coefficient
I	System current
I_d	Inherent degradation of the solar cell

I_{EIR}	Infrared intensity
I_{solar}	Solar intensity
J	Angular jitter of the beam
K	Average solar constant 1367 Wm^2
k	Thermal conductivity of a material
L	Radiation mapping constant
$L_{0.25}$	Linear energy transfer at 25% of the limiting cross section
L_d	Degradation of a solar array
m_B	Battery mass
N	Number of battery cells
N_e	Number of electrons per cm^3
n	Index of refraction
P	Average output power
P	Load power
P_{BOL}	BOL power per unit area
P_d	Spacecraft power requirements during daylight
P_e	Spacecraft power requirements during eclipse
P_{EOL}	EOL power per unit area
P_o	Ideal solar cell output performance per unit area
P_{sa}	Power required during daylight
Q	Quality of the laser beam
Q_c	Critical charge
$Q_{conducted}$	Conduction heat transfer
$Q_{convection}$	Convection heat transfer
$Q_{external}$	Heat input from outside the spacecraft
Q_{in}	Heat added to the spacecraft
$Q_{internal}$	Heat generated inside the spacecraft
Q_{out}	Heat removed from the spacecraft
$Q_{radiator}$	Heat radiated out of the spacecraft
q_{albedo}	Heat reflected off the Earth from the Sun onto the spacecraft
$q_{backload}$	Backload heat
$q_{earthIR}$	IR emittance from the Earth on the spacecraft per unit area
$q_{external}$	External environment heat load on the radiators per unit area
q_{solar}	Solar emittance from the Sun on the spacecraft per unit area
R	Distance in Earth radii from the idealized point dipole near the Earth's center
R	Distance in kilometers from a nuclear detonation
R	Dose rate
R	Number of upsets or errors per bit day
R	Radial distance measured in Earth radii
R	Range from the laser to the target
R_o	Reference dose rate
T	Absolute temperature

T	Surface temperature of the body
T	Time period
T_d	Period in daylight per orbit
T_e	Period in eclipse per orbit
T_f	Atmospheric temperature
T_p	Total orbit period
T_s	Surface temperature of the spacecraft
ΔT	Temperature difference
t	Discharge time
t_o	Reference time
V	Satellite velocity
V	System voltage
V_B	Battery average discharge voltage
V_{bus}	Bus voltage
X_e	Efficiency with which the spacecraft can relay power from the array to the battery to the loads in eclipse
X_d	Efficiency with which the spacecraft can relay power from the array to the loads in daylight
Δx	Thickness of a material
Y	Yield value

Greek

α	Absorptivity of a material
ϵ	Emissivity of a material
Θ	Sun incidence angle from the solar panel normal vector
λ	Magnetic latitude
λ	Wavelength of the laser
μ	Gravitation parameter equal, 3.986 x 105 km^3/s^2
ν	Frequency of collision of electrons with ions
ρ_{albedo}	Earth's albedo
ρ_N	Number density of atomic oxygen
σ	Stefan-Boltzmen constant, 5.67 x 10^{-8} W m^{-2} K^{-4}
σ_L	Limiting cross section, or sensitive area
ϕ	Incidence angle

Chapter 1

Introduction

This report is a guide to the basic design of thermal, power, and environmental spacecraft subsystems. Power subsystem design includes the determination of the primary power source, energy storage, and power distribution system component selection. Thermal subsystem design involves insulation, passive heat dissipation, and active heat dissipation. To ensure the power and thermal subsystems are capable of operating in a vacuum with intense electromagnetic fields and radiation, environmental concerns must be taken into account. Thus, the environmental section deals primarily with modeling the space environment and the environment's effect on spacecraft materials.

Before describing the processes used in design and providing sample calculations, a through understanding of the connections and interactions between the power; thermal; attitude dynamics; control; communications, command and data handling subsystems and their interaction with the space environment is necessary. Chapter 1 enumerates the connections between the thermal and power subsystems and the space environment with each other and the remaining spacecraft subsystems. Consequently, Chapter 2 provides technical information, compiled by industry, concerning the design of the power and thermal subsystems and the simulation of space environment to determine the environmental effects of space on the materials, and consequently subsystems, used in spacecraft.

1.1 Power

A spacecraft must be able to supply sufficient power to its subsystems to accomplish its desired task. The power subsystem's primary concern is supplying sufficient energy to the spacecraft during peak and average energy consumption. Another consideration for the power system design is the beginning-of-life and end-of-life requirements. Also, thermal outputs can be an issue when designing the power system due to the amount of thermal energy radiated by both the power subsystem and other spacecraft subsystems. Power subsystem design should only be undertaken after a rough estimate of the power needs of the payload and CC& DH and AD& C subsystems has been accomplished. There are four power source subsystems for a spacecraft: the power source, energy storage system, power distribution system, and the power regulation and control system.

1.1.1 Uses and Implementations

The heart of any power system is the power source. There are four methods of supplying power to a spacecraft: photovoltaic, static sources, dynamic sources, and fuel cells. Photovoltaic power is generated by using photovoltaic cells to convert incident solar radiation directly to electrical energy. Photovoltaic sources are the most common power source used among spacecraft today because of the abundance of solar radiation, energy, given off by the sun.

Static power involves converting thermal energy generated by a power source, such as a nuclear reactor, directly to electrical energy. The typical static power source is a nuclear reactor utilizing either a uranium or plutonium core for fuel. Due to the dangerous nature of radioactive materials, static power sources are used primarily for interplanetary missions where conventional energy collection methods, photovoltaics, are unable to fulfill the spacecraft power requirements.

Dynamic power involves indirectly converting thermal energy generated by one or more power sources, such as nuclear and photovoltaic, chemically to electrical energy by utilizing a Brayton, Stirling, or Rankine cycle. Dynamic power sources are advantageous for missions with high power requirements, long operating lives, eclipse power needs, and remain relatively close to sun.

Typically, missions with short life cycles and manned missions utilize fuel cells as as their power source. Fuel cells are non-renewable, meaning they can not be recharged. Fuel cells generate energy by combining hydrogen and oxygen in a chemical reaction to produce electrical power. The hydrogen-oxygen reaction produces drinkable water, thus reducing the payload for manned missions. Missions with large power needs during their eclipse phase benefit from fuel cell technology, which does not require any solar radiation to operate.

Energy storage is important for spacecraft with long mission lives or eclipse needs. The common means of energy storage is batteries, which are categorized into one of two types: primary or secondary. Primary batteries are batteries that can only be used once because they can not be recharged. Primary batteries are usually used for short missions. Secondary batteries are used for longer missions and can be recharged.

Secondary battery selection is determined using battery charge and discharge rates. When a spacecraft is orbiting a body it is either in view of the Sun or is eclipsed by the body. When a spacecraft is in the sun's view it uses its solar cells to produce electricity and charge the battery. While the spacecraft is eclipsed, the energy stored in the battery is used to power the spacecraft. This eclipsed period is known as the discharging phase.

With the power source and energy storage determined, it is necessary to transmit power throughout the spacecraft using a power distribution system. The components of the power distribution system are chosen to minimize power loss and mass, and to maximize survivability, cost, reliability, and power quality.

The power distribution system is composed of: cables and cabling harnesses, fault protection, switches, power convertors, and the power bus. Power is distributed throughout the spacecraft using cables which are held in place with cabling harnesses. A large percentage of the power subsystem's mass is composed of the cables and cabling harasses.

The fault protection system protects powered spacecraft components by detecting, isolating, and fixing faults. Faults are indicators of a short circuits, which drain excessive power and can lead to total spacecraft failure. Typically, faults are difficult to repair, so the fault protection system attempts to isolate faults to prevent catastrophic chained component failure.

Power switches control the flow of power throughout the spacecraft. Component power requirements and the number of connected components are used to select power switches. Most spacecraft utilize mechanical power switches due to their exceptional reliability and durability.

Systems requiring special power conditioning use power convertors to switch between different operating voltages and currents. Power convertors are used sparingly to reduce the overall power subsystem mass, which indicates that components should use similar power to limit the number and size of power converters.

Finally, the power bus serves as the backbone of the distribution system by connecting the power supply to the subsystems. Typically, a spacecraft bus is run at 28V with power conditioning equipment applied where necessary.

Once the physical components have been selected the designer must determine the power subsystem's operating characteristics such as current type, power regulation system, bus regulation system,and battery arrangement and charging system. A desired operating characteristic may require the designer to reiterate the component selection process.

The major power characteristic of spacecraft is the current type, direct current (DC) or alternating current (AC). Spacecraft typically use dc power, but ac power is used on some larger spacecraft such as the International Space Station. Converting power to ac requires more electronics and adds more to the mass of the spacecraft.

Typically, power conditioning is accomplished with the use of power converters. There are two types of converters,centralized and decentralized. Centralized converters are located at the bus and regulate power output within the bus. Decentralized converters are located at the bus output to each load. The advantage to using centralized converters is simplified integration of the power loads. Decentralized converters are used primarily in high power applications such the International Space Station.

Power regulation ensures the spacecraft's subsystems are not saturated with excess power. An important aspect of the power regulation system is the power supply regulation. There are two methods of regulating the power source, Peak Power Transfer (PPT) and Direct Energy Transfer (DET). PPT extracts the exact amount of power needed from the solar array. DET regulates power by shunting away power not used by the other subsystems or batteries. Shunt regulation is typical due to its efficiency, fewer parts, and lower mass.

Bus voltage is also regulated by the power regulation system. Regulation can be unregulated, quasi-regulated, or fully regulated. Quasi-regulation regulates the bus voltage during charge of the batteries but not during discharge. Quasi-regulation has a lower efficiency than the unregulated method and may cause high electromagnetic interference when used with PPT system. Fully regulated systems regulate power during both charge and discharge, but are inefficient. The fully regulated method can be used on low power spacecraft and makes design integration of subsystems simple. However, fully regulated systems have low efficiency and high electromagnetic interference.

Battery charging systems are part of the power regulation system. Batteries can be charged in parallel or series. Typically, batteries are charged in parallel orientations due to being simple and inexpensive. However, parallel configurations are known to degrade batteries over time. For missions longer than five years, batteries should be charged individually in a series system, because series connections extend the life of the batteries. However, series configurations add impedance to a system, complicate design, and create thermal dissipation problems.

1.1.2 Interactions with other Subsystems

To design a power sub-system capable of providing safe and reliable energy to the spacecraft requires collaboration with other subsystem design groups. Each subsystem has a minimum power budget and requires electrical conditioning. Therefore, information concerning voltage, amperage, connection outlets, protection, and thermal concerns must be made available to ensure mission success and optimal performance.

The dynamics and controls subsystems require power for sensors, propulsion, and controllers. The power subsystem team needs information concerning the bus connection, average power consumption, peak power consumption, power conditioning requirements, and thermal information to ensure proper system integration. Furthermore, information concerning the physical arrangement of the dynamics and controls subsystems will be necessary to ensure proper cabling and proper temperature conditions are met.

The structures and launch vehicle team will work closely with the power team to ensure optimal bus design. Launch vehicle selection must ensure that a proper power subsystem is included to power the spacecraft. The structures subsystem provides information concerning the design and final form of the power bus, the backbone of the power subsystem. Information concerning size, position, and orientation of the power subsystem is necessary to ensure all subsystems are properly powered while limiting effects of electromagnetism and excessive heat. The thermal subsystem team provides information concerning the heating and heat dissipation needs of the spacecraft. Information concerning the heating cycles, average power needed to maintain operating conditions, heat dissipation, and sensory equipment power needs must be addressed.

The environmental subsystem requires power mainly for sensors and monitoring equipment. The thermal and environmental teams are closely related in the sense that the environment will have a large effect on the temperature of the spacecraft. Thus, accurate information concerning the environmental effects on the temperature mapping provided by the thermal subsystem team must be addressed. Furthermore, the environmental team provides information concerning the effects of radiation and gravity on the power subsystem. Information concerning shielding and cable orientation ensures minimal interference from both the environment and the power subsystem itself on the spacecraft.

The communication, command, and data handling is the most power-intensive subsystem on the spacecraft. Extensive information concerning shielding, power transformations, surge protection, power connections, and power conditioning will be necessary. Information such as average and peak power consumption will be necessary to properly size electrical equipment required. The CCDH subsystem will require a backup power system to ensure operation in case the main power system fails. The backup system allows for emergency diagnostics to help prevent total failure of the spacecraft. Finally, some CCDH equipment may require special thermal and electromagnetic considerations. The power team must be made aware of any special requirements concerning CCDH equipment to ensure proper operation.

Essentially, the goal of the power team is to gather information concerning power use, thermal information, and shielding concerns to design a power system capable of supplying reliable power while leaving the connected subsystems unaffected. The information provided by the other subsystem teams allows the power team to size batteries, power source, discharge system, and determine the bus design.

1.1.3 Future Technologies

Currently, most spacecraft in earth orbit rely on solar panels to sustain rechargeable batteries that power the craft. Unfortunately these panels must be large, sometimes to the point of being obstructive, and become less effective the further out into the solar system it travels. New advances in power technology may allow cheaper and more efficient missions to take place.

Possibly the most well known potential power source is nuclear fusion. Hydrogen would fuel the fusion reactor, producing the energy needed to power the craft and sustain the reaction. Since the by-product of hydrogen fusion is helium gas, there is practically zero environmental pollution. This lack of pollution differs from present nuclear power sources by not producing radioactive by-products. However, there are a few obstacles to using a fusion power source. Currently researchers are having difficulty getting more energy out of the reaction than is put in, resulting in a net loss of energy. Also the fusion reaction itself takes place at extremely high temperatures, at least 50 million Kelvin, meaning the size of the reactor must be larger and heavier than conventional nuclear fission reactors. The added size accommodates the additional thermodynamic problems that arise.

The European Space Agency is considering biomass as an alternative power source. Biomass is a reusable source, so depletion in the future is not a pressing issue. Experiments are being made with organic matter, such as food and human waste. Biomass systems could cut down on the amount of fuel needed at launch and decrease costs. A space station orbiting a planet would not need as many refueling trips as Mir or the International Space Station (ISS), if any. Biomass would only be economical on manned vehicles though, as they would have the readily available sources. Also, the amount of matter needed to sufficiently power a craft may exceed what can be produced by the inhabitants. Therefore, it is necessary to regularly re-supply spacecraft in orbit.

While the emphasis of future power systems is on new technologies, there will also be new advances for current systems. For example, advances may be made that allow solar panels to collecting solar energy more efficiently or at greater distances. Until radical new advances in power are made, these adjustments will help lower the cost per mission, at launch and during operation.

1.2 Thermal

When examining a given body in orbit, consideration must be given to the thermal control subsystem. The thermal control subsystem measures, tests and regulates the spacecraft's temperature while in space. Thermal measurements are necessary to keep the spacecraft within safe operating temperature limits. Temperature changes are due to many factors including internal electrical components, radiation influences from both the sun and earth, and orbiting from the dark side of the planetary body back into direct sunlight. Therefore it is important for a spacecraft to communicate and relay information quickly and reliably between its various subsystems. Thought must also be given to examining how different subsystems' needs are competitive with each other in terms of complexity, effectiveness, weight and overall cost. An object hierarchy must then be created to examine the varying subsystems in order of their importance and effectiveness.

The three main thermal concerns that need to be addressed are the spacecraft's power system, detectors, and attitude determination. The thermal control subsystem must examine it power levels and battery life as they pertain to the ships mass and overall mission time-line. As for the spacecraft's multiple cameras and infrared detectors, protection must be given to the colder temperatures present around such sensitive devices to insulate its affect on its neighboring systems. The overall performance of a spacecraft's attitude determination and control system is mainly affected by temperature changes due to various radiation sources.

In the following sections a more in depth introduction of both the thermal interaction and implementation of the varying thermal control subsystems will be spoken to.

1.2.1 Uses and Implementation

There are multiple thermal considerations that need to be examined when designing a spacecraft that needs to first ascend through the Earth's atmosphere, and then cope with various forms of radiation once it reaches orbit in space. While leaving the Earth's atmosphere, temperature changes can occur rapidly due to the booster rockets' high temperatures. After the boosters are jettisoned, molecular friction occurs with the atmosphere, heating the payload. These problems must be addressed in order to get the payload into its proper orbit. When the payload has entered outer space the problem of radiation is encountered.

The different forms of radiation encountered are direct sunlight, blackbody radiation from planets, and sunlight reflecting off planets (albedo). These harmful effects can cause great fluctuations in the temperature of the spacecraft's vital subsystems if not controlled properly. There are several different radiation-reflecting thermal coatings with characteristic properties, which are tailored for the mission as dependant upon blackbody emissivity and solar absorptivity.

Direct sunlight accounts for most of the thermal energy supplied to a spacecraft. Thus, when a spacecraft goes into the shadow of a planet, or comes back out, the greatest thermal changes will occur. Many devices are used to control temperature including heaters, louvers, and heat pipes. Insulation is also used to prevent heat from escaping certain vital components of the spacecraft while radiators release any excess heat. All of these technologies will be discussed in greater detail during design considerations.

1.2.2 Interactions with other Subsystems

The thermal subdivision of the PT & E functional group's interactions with the rest of the spacecraft can be broken into two aspects: interactions with other subsystems in the functional group; and interactions with other functional groups. The interactions with the power subsystem are simple. Most power systems besides solar arrays generate heat. Whether the power systems are fuel cells, static power or dynamic power, it is likely that the heat generated will need to be dissipated: a problem that falls into the realm of the thermal subsystem.

The Environment subsystem is where most of the work of the thermal control system lies. The changes in temperature that occur when the spacecraft goes from the dark side to the light side of a planet while in orbit are the largest fluctuation that might be expected while in operations so it is this phenomenon that demands the most attention. The thermal coatings used outside the ship are important decisions that must be made for this aspect, which brings us to our first Functional division interaction.

The thermal design of the outside of the ship must be done concurrently with the structural design (in S & LV functional group) to ensure that neither design inhibits the effectiveness of the other. Also within this subsystem lies the issue of the launch vehicle. Depending on which launch vehicle is chosen, any thermal control flaws in the launch vehicle must be overcome by the thermal design of the spacecraft it is carrying.

The interactions with the AD & C functional group are mutual. If the attitude control system involves anything where temperature needs to be kept constant, then it must be designed by the thermal subsystem. Also the propulsion system must be heat shielded to avoid heating other spacecraft components. If the control system uses aero braking as part of its control system, then the aero brake must be heat shielded. Next, it is important for those determining the amount of propellant needed for the attitude control system to know the mass of the various thermal control systems.

For the CC & DH functional group, the main task will be to control the temperature of the telemetry and data handling systems to keep them within their operational temperatures.

1.3 Environment

Atmospheric and near-Earth space environments, as well as numerous other space environments, strongly shape the lifetime, performance, and cost of operational space systems. Environmental influences may also cause costly malfunctions, such as loss of components and subsystems which can lead to the complete loss of a spacecraft. Harmful environmental influences include atmospheric heating and erosion, atomic displacement through radiation, surface charging from intermittent plasma fields, orbital debris, and ground-based hostile threats such as anti-satellite missiles.

1.3.1 Effect of Operating Environments on Spacecraft

Once the craft reaches orbit it is subjected to a wide array of hazards. The vacuum-like conditions of space cause most organic materials to outgas, thus necessitating outgas modeling. Outgassing is the generation of spurious molecules which may act as contaminants to other surfaces. Moreover, a spacecraft moving through the Earth's upper atmosphere is under the influence of aerodynamic lift, drag, and heating. These effects can damage subsystems as well as violate pointing requirements. If a spacecraft is to remain in lower orbits around the Earth, the neutral atmosphere can erode satellite surfaces, affect thermal and electrical properties of the craft's surface, and possibly decay the spacecraft structure. Structural decay can be attributed, at least partially, to the predominance of the reactive form of oxygen, atomic oxygen, in altitudes between roughly 200 and 600 kilometers. This form of oxygen reacts with organic films, advanced composites, and metallic surfaces. Additional environmental influences effecting spacecraft design include spacecraft charging due to electron or ion flow from energized plasma to the craft.

Plasma fields are created by magnetic substorms which dissipate the magnetic energy stored in the Earth's magnetic field. This magnetic energy is the product of interaction between the Earth's magnetic field and solar wind fluctuations. As a spacecraft passes through these plasma fields, a differential charge forms on its surface. On charged surfaces, an electrostatic arc has the potential of forming. Electromagnetic interference from this arc can cause spacecraft to operate erratically. The transient generated by discharge can propagate into the spacecraft electronics and cause upsets ranging from logic switching to complete system failure. Discharges can also cause long term degradation of exterior surface coatings and enhance contamination of surfaces, degrading thermal properties. Vehicle torquing or wobble can also be produced when multiple discharges occur. It also compromises scientific missions seeking to measure properties of the space environment. A spacecraft must either be designed such that it is able to keep the differential charging below levels which cause arcs, or can tolerate the electrostatic discharges.

Radiation hazards have three main origins: the Earth, the sun, and sources outside the solar system. Trapped radiation in the Earth's Van Allen belts is a permanent danger to orbiting spacecraft. Solar particle events are associated with solar flares and have important consequences with man made systems. Solar particle events degrade solar arrays, add to background noise, and cause illness in astronauts. Galactic cosmic rays are particles which reach the vicinity of Earth from outside the solar system. Cosmic rays pose a serious hazard because a single particle can cause malfunctions in common electrical components such as RAM and microprocessors. Cosmic rays also generate background noise in various satellite subsystems such as: star sensors, infrared detectors, and components employing charge-coupled devices creating events which impersonate real signals. These noisy signals can affect subsystems depending on the nature of the genuine signal. When a single particle causes a malfunction, it is termed a single event phenomenon. To determine the proper amount of radiation shielding the dosage rate for the desired orbit, as a function of shield thickness, must be found.

1.3.2 Orbital and Man Made Hazards

Space vehicles in orbit around the Earth are exposed to hazardous debris represented by both natural meteoroids and man-made space debris particles. All meteroids originate from either comets or asteroids. Nearly all meteoroids are generally distributed isotropicly and are termed 'sporadic' meteors. The meteoroids that retain their parent body orbit are termed 'meteor streams'. Thus, the total meteoroid environment is comprised of the average of the sporadic meteors together with the yearly average of meteor streams. Space debris particles also co-exist with the natural particle flux. Most of these man-produced debris particles are in high-inclination orbits and consist of spent rocket stages, inactive payloads, and debris from exploded satellites.

Additional man made hostile threats to a spacecraft include ground and space based laser weapons, high velocity fragments, weapons, homing kinetic energy weapons, neutral atomic particle beam weapons, and nuclear weapons. Several methods exist for avoiding anti-satellite threats ranging from passive decoys to active maneuvering and self defense capabilities. A nuclear blast poses the most serious threat to a spacecraft or ground station. These threats must be addressed from the outset of mission design when considering satellite survivability. Barring a direct hit, the most damage from a nuclear blast stems from radiation effects. Therefore by increasing the possible dosage rate a satellite might experience, the required amount of radiation shielding increases as well.

Free electrons created in a nuclear blast, or electrons from radiation in general, can absorb and reradiate radio frequency energy and refract the electromagnetic waves of the radio communications links between the ground and spacecraft, distorting the signal. These distortions can reduce the strength of the signal, thus interrupting communications.

1.3.3 Future Technologies

Various environmental effects must be considered when designing future space systems. Of course, to adequately estimate these effects, the environment must be well defined. In order to properly model, analyze, document environmental effects and communicate results, mission information such as orbit selection and communication frequency must be known. The most critical piece of information may be the description of the mission's orbit. Once flight altitude and path are known, the presence and frequency of occurrence of aerodynamic and radiation effects, for example, can be determined.

Chapter 2

Equations and Modeling

When designing a spacecraft the designer must make sure that an adequate amount of power can be supplied to all of the subsections of the spacecraft. Also, the designer must also ensure that the payload and structure of the spacecraft are properly protected against thermal effects while in space and during re-entry. Moreover, the designer must also ensure the spacecraft is capable of operating properly in its environment. The following section will show how to properly model a spacecraft's power system, thermal system and model against environment effects. This section will present the equations necessary to design and model these subsystems.

2.1 Power

The power system is divided into 3 components: the power source, the energy storage device and the power distribution system. each of these subsystems must be optimized and designed to work together for the power system to perform optimally.

2.1.1 Power Sources

There are four methods for supplying power to a spacecraft: photovoltaics, static power, dynamic power, and fuel cells. The designer must decide which power source is best for the spacecraft power requirements; then, certain formulas must be used to calculate the power required and whether the chosen method can accommodate these requirements.

Cost, power range, stability and maneuverability, and degradation over life must be considered when selecting a power source. The designer must compare the advantages and disadvantages of each option in order to choose the best method of powering the spacecraft. Table 11-33[8] compares these characteristics of solar photovoltaics, solar thermal dynamic, radioisotope, nuclear reactor and fuel cell power.

After the power source has been initially chosen, calculations must be performed to determine whether it provides the proper amount of power required by the spacecraft. The power source must be appropriate for the mission type; for example, interplanetary missions require that the spacecraft travel throughout the solar system; photovoltaics are a poor choice for some interplanetary missions because the effectiveness of solar arrays decreases with the distance from the sun due to decreasing solar radiation intensity.

If photovoltaics are used for a mission, the designer must first determine the mission life and average power requirements of the spacecraft. The array must be designed to meet the power requirements of the spacecraft at End of Life (EOL). Thus, the power output will be oversized for the Beginning of Life (BOL) requirements. The designer must first calculate the illumination intensity of the spacecraft's orbit. Illumination intensity depends on the sun incident angle, eclipse period, solar distance, and concentration of solar energy. Solar array design requires optimization of conflicting parameters such as cost, array area, mass, and risk. The next step is to determine the power required during daylight, P_{sa}.

$$P_{sa} = \frac{\frac{P_e T_e}{X_e} + \frac{P_d T_d}{X_d}}{T_d} \tag{2.1}$$

where P_e and P_d are the spacecraft power requirements during eclipse and daylight respectively; Te and Td are the periods in eclipse and daylight per orbit respectively; Xe is the efficiency with which the spacecraft can relay power from the array to the battery to the loads in eclipse; and Xd is the efficiency with which the spacecraft can relay power from the array to the loads in daylight.[8] The efficiencies depend mainly on the power regulation system as discussed in section 2.1.3.

In order to determine the periods in eclipse and in daylight, the designer must first determine the total orbit period, Tp. If the designer is given either the eclipse period or the daylight period, the other can be found by subtracting the given period length from the total period time. Equation 2-2 can be used to calculate total orbit period.

$$T_p = 2\pi(\frac{a^3}{\mu})^{1/2} \tag{2.2}$$

Where μ is the gravitation parameter equal to 3.986 x 10^5 km^3/s^2 for Earth.

Next, the solar cell material must be selected. The designer must determine the performance degradation over time for different types of solar cells. The main performance parameter, energy conversion efficiency, is defined as the power output divided by the power input. This parameter is used to calculate the ideal solar cell output performance per unit area, Po, equal to the illumination intensity times the efficiency. The designer can then choose a suitable solar cell material by considering the efficiency and degradation.

Next, the designer must determine the BOL power per unit area: defined as

$$P_{BOL} = P_o I_d \cos\Theta \tag{2.3}$$

where I_d is the inherent degradation of the solar cell and Θ is the sun incidence angle from the solar panel normal vector.[8] The inherent degradation depends on the design efficiencies, shadowing, and temperature variations.

I-V curves are often used to characterize BOL and EOL performances to determine the spacecraft power requirements; I is the system current and V is the system voltage. An I-V curve (as shown in Figure 2.1) illustrates the performance of a solar cell or solar array.

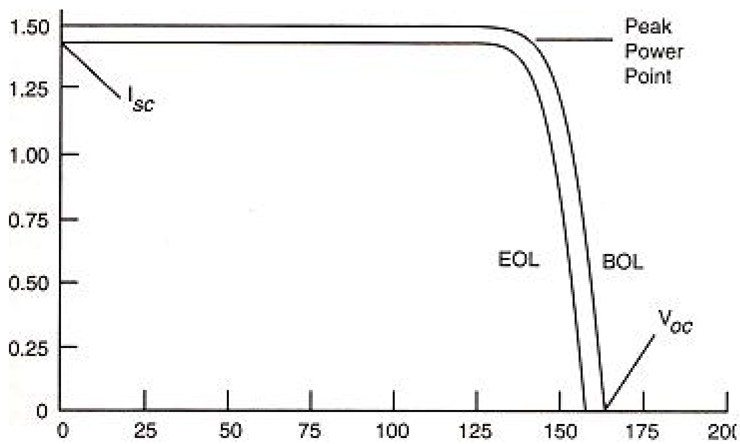

Figure 2.1: I-V Plot for a Planar Array[8]

The power available is equal to the area under the I-V curve.

Temperature is another major factor in the performance of the solar array. The solar array's temperature coefficient is defined as the percent degradation of the array's performance with the increase in temperature. This coefficient depends on the type of cell, the power output characteristics, the operating temperature, and the radiation environment.

The number of solar cells connected in series determines the bus voltage required at EOL. The current output required is dependent on the number of solar cells in parallel orientation. The designer must also decide whether the solar array will be a planar or a concentrator array. A planar array consists of solar cells mounted on the surface of the spacecraft or the surface of an extension arm. A concentrator array utilizes mirrors or lenses to intensify solar radiation and reflect radiation that has passed through the array unused to be used again.

Shadowing from antennas and other appendages must be minimized; if the solar cells are oriented in series, the shadowing of one cell can result in the loss of the entire cell string.

The orientation of the array depends mostly on the dynamics of the spacecraft. Body-mounted solar arrays are usually used on spinning spacecraft. Panel-mounted solar arrays are used on 3-axis-stabilized spacecraft to enable the array to be pointed toward the sun at all times using sun sensors.

The designer must also decide whether or not coatings, coverslides, and back surface reflectors should be used to increase the performance of the solar array. A material should be chosen with a suitable index of refraction to maximize the amount of incoming solar light. Coverslides, coatings, and back surface reflectors are modeled by Snell's Law, given in equation 2-4.

$$n_1 \sin\phi_1 = n_2 \sin\phi_2 \tag{2.4}$$

where n_1 and n_2 are the indices of refraction of the media the light is passing between and ϕ_1 and ϕ_2 are the angles of incidence of the light on each surface.

The last step in designing a solar array is to consider degradation effects. Radiation can reduce a solar array's output voltage and current. Life degradation (L_d) of a solar array can occur because of thermal cycling in and out of eclipses, micrometeoroid strikes, plume implement from thrusters, material outgasing, and electrons or protons trapped in the earth's magnetic field. Life Degradation is defined as

$$L_d = (1 - \frac{\text{degradation}}{\text{year}})^{\text{satelite life}} \tag{2.5}$$

After the life degradation is calculated the array's performance at EOL can be calculated.[8]

$$P_{EOL} = P_{BOL} L_d \tag{2.6}$$

The solar array area (A_{sa}) can then be calculated as.[8]

$$A_{sa} = \frac{P_{sa}}{P_{EOL}} \tag{2.7}$$

The designer should examine different geometries of solar arrays to determine the best arrangement,[8] worst case sun incident angle to develop an estimated P_{EOL} to test if the solar array can accomplish power needs at EOL.

Static power, dynamic power, and fuel cell power design are less complicated than solar array design. There are two types of static power sources, thermoelectric couples and thermionic energy conversion. Thermoelectric couples use a temperature gradient between the positive and negative junctions of individual cells in a series or parallel to provide the desired dc electric output; the temperature gradient is provided by the slow decay of radioactive sources. Thermoelectric system design consists mainly of deciding which radioactive material to use by comparing specific power values. Typical thermoelectric system efficiencies are 5-8%.[8]

Thermionic energy conversion uses a hot emitter facing a cold collector inside a sealed container of ionized gas. The designer must choose the emitter and collector temperatures to optimize performance. The designer must minimize the weight and size of the thermal radiators while maximizing the efficiency. The designer must also choose the radioactive material based on the mission life requirements. Typical thermionic system efficiencies are 10-20%.[8]

Dynamic power uses a heat source combined with a heat exchanger. The designer must decide what type of heat source to use: concentrated solar energy, radioisotopes, or controlled nuclear fission. The designer also select the engine to transfer heat to the working fluid. The Stirling cycle uses a single phase working fluid through four processes: isentropic compression, isentropic expansion, constant volume heating, and constant volume cooling.

The Stirling cycle efficiency is 25-30%.[8] The Rankine cycle uses a two-phase fluid system. The fluid is transferred to energy using a boiler, a turbine, an alternator, a condenser, and a pump. The Rankine cycle efficiency is 15-20%.[8] The Brayton cycle uses a single compressible working fluid and consists of adiabatic compression and expansion stages separated and coupled by constant pressure heat addition and rejection stages. The Brayton cycle efficiency is 20-35%.[8]

Fuel cells are generators that convert chemical energy to electrical energy. The main reactants are usually hydrogen and oxygen because the by-product is water, which can be used in manned spaceflight. A typical fuel cell produces 0.8 V of dc electricity. Fuel cells have higher efficiencies than other power sources; typical fuel cell efficiencies are 80% in normal conditions and 50-60% at low current draws.[8]

2.1.2 Energy Storage

Energy storage is an integral part of the power subsystem. Storage devices provide power for entire missions of duration less than one week or back-up power for longer mission. Spacecrafts using photovoltaic cells or solar thermal dynamics for power require energy storage for peak-power demands and eclipse periods. Batteries typically store energy, though alternate systems such as fuel cells and flywheels can be used for energy storage.

A battery consists of two or more cells connected in series or parallel; the number of cells required is determined by the bus-voltage. The energy capacity of the battery is characterized by the ampere-hour capacity or, when multiplied by the operating voltage, the watt-hour capacity. Batteries connected in series increase the voltage output, whereas batteries in parallel increase the current output. Both configurations increase the watt-hour capacity with an increase in batteries.

Table 11-37[8] shows physical, electrical, and programmatic factors that must be considered in the conceptual phase of mission designs. Providing a stable voltage for all operating conditions during mission life is the highest priority factor because load users favor semi-regulated bus voltages.

Batteries come in two varieties: primary or secondary. Primary batteries are one time use because the cells can not reverse the process of converting chemical energy into electrical energy. Since these cells are not rechargeable, primary batteries are typically used for short missions or long-term tasks which use little power, such as memory backup. The most commonly used primary battery types and their ranges of specific energy density and applications are shown in Table 11-38.[8]

The chemical reaction that occurs within the battery determines the specific energy density. The reactants have theoretical electrochemical capacities measured in Amp-hours/kg, and the reaction produces a theoretical cell voltage measured in volts. Table 1-10[9] shows the theoretical voltages and electrochemical capacities for selected primary and secondary battery systems. The specific energy density for the reaction is defined as

$$\frac{W\,h}{kg} = \frac{A\,h}{kg}V \tag{2.8}$$

measured in Watt-hours/kg.

Secondary battery cells can convert electrical energy into chemical energy as well as the reverse, and can repeat the process for thousands of cycles. Secondary batteries provide power during eclipse periods and recharge during sunlight periods. Orbital elements such as altitude and inclination angle determine the number of charge/discharge cycles the batteries must support, and indicate the number and length of eclipses during the mission life. The batteries of a geosynchronous satellite need to store energy for two 45-day eclipse periods per year, lasting no more than 72 minutes a day, while a LEO spacecraft has about 15 eclipses per day lasting no more than 36 minutes each. The percent of total battery capacity used during a discharge period is defined as depth of discharge (DOD). Because the batteries on the LEO spacecraft charge and discharge more often, the DOD is lower for the LEO satellite (15-25%) than the geosynchronous satellite (50%).[8] The DOD is defined as

$$DOD = \frac{(\text{Load Power})\,(\text{Discharge Time})}{(\text{Capacity})\,(\text{Battery avg discharge voltage})} \tag{2.9}$$

or

$$DOD = \frac{P\,t}{C\,V_b} \tag{2.10}$$

Lower DOD values imply longer battery life, as shown in Figure 2.2. This Figure uses nickel-cadmium (NiCd) and nickel-hydrogen (NiH2) as examples. The total capacity of the batteries can be determined from the number of cycles and the average depth of discharge. Table 2.1 show examples of secondary battery types.

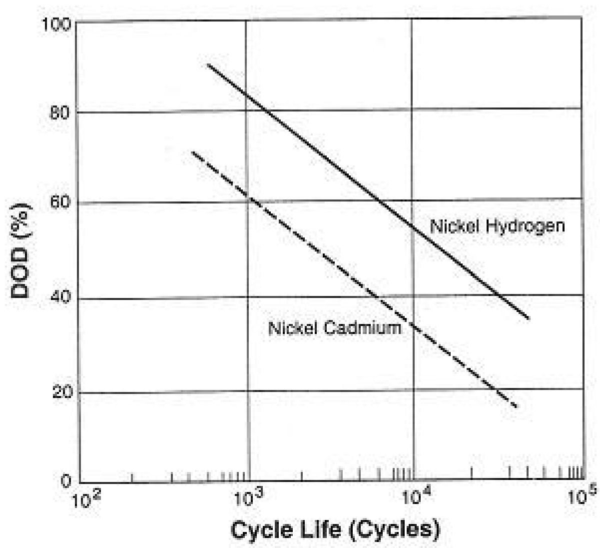

Figure 2.2: Depth-of-Discharge vs. Cycle Life for Secondary Batteries.[8]

NiCd batteries are still common secondary storage devices for many applications. They have been space-qualified and there are extensive databases for most missions. NiCd batteries

Table 2.1: Characteristics of Selected Secondary Batteries.[8]

Seconday Battery Couple	Specific Energy Density (Whr/kg)	Status
Nickel-Cadmium	25-30	Space-qualified, extensive database
Nickel-Hydrogen (individual pressure vessel design)	35-43	Space-qualified, good database
Nickel-Hydrogen (common pressure vessel design)	40-56	Space-qualified for GEO and planetary
Nickel-Hydrogen (single pressure vessel design)	43-57	Space-qualified
Lithium-Ion (LiSO2, LiCF, LiSOCl2)	70-135	Space-qualified
Sodium-Sulfur	140-210	Under development

for aerospace missions typically have 22-23 cells and energy capacities of 5 to 100 Amp-hr.[8]

For applications where longer life and higher specific energies are important, NiH2 has recently become the qualified energy storage device of choice. There are currently three space-qualified NiH2 design configurations: individual pressure vessels, common pressure vessels, and single pressure vessels. The individual pressure vessels were the first technology introduced for aerospace applications; in these batteries, a single electrochemical cell is contained inside the pressure vessel. typically designs consists of multiple cells connected in series to obtain the desired voltage. A working terminal voltage of 1.22 to 1.25 Vdc and cell diameters of 9 to 12 cm with capacity ranges from 20 to 300 Amp-hr are typical of individual pressure vessels.[8] The electrode stacks are connected in parallel.

The main difference between individual and common pressure vessels is in the wiring of the electrode stacks; common pressure vessel have two sets of electrode stacks connected in series rather than parallel. Connecting the stacks in series yields working terminal voltages of 2.44 to 2.50 Vdc, double that of individual pressure vessels. Space-qualified cell diameters are 6 to 9 cm for capacities of 12-120 Amp-hr; The specific energy of the battery is also higher because there are half as many pressure vessels required for common pressure vessels.[8]

The single pressure vessel design for a NiH2 battery contains a common hydrogen supply used by three or more cells connected in series. Each cell has its own isolated electrolyte supply in individual cell stack containers. The major design characteristic is allowing the free movement of the hydrogen while maintaining cell stack isolation. Single pressure vessel battery designs are currently obtainable in either 12.5 or 25 cm diameters.[8]

Lithium ion batteries provide considerable energy density advantages and wider operating temperature ranges than either NiCd or NiH2 batteries. The operating voltage of a lithium ion cell is three times that of a NiCd cell, allowing for a one-third reduction in cell numbers, providing a 60% volume reduction and 50% mass reduction for most aerospace applications.[8] Lithium ion batteries have recently been used on NASA's Mars exploration rovers Spirit and Opportunity; in early 2004, Eutelsat's W3A became the first GEO satellite to use Lithium ion batteries.

Table 11-40[8] shows the steps used to select a battery for a spacecraft. The ideal battery capacity, defined as the average eclipse load times eclipse duration, must be large enough to account for the battery-to-load transmission efficiency and the DOD constraints. Spacecraft typically use two to five batteries, though the number of batteries, N, may be equal to one when determining the average DOD over a 24 hour period with fully charged batteries. At least two batteries are used on-board for redundant operation, unless the battery uses redundant cells; more than five batteries require complex recharging components. For some missions, the secondary batteries may operate during peak power loads during full sun conditions in which case peak power load would drive the required battery capacity instead of eclipse load.

The number of cells in a battery and the voltage per cell to power a spacecraft with a load power, P, and a bus voltage, V_{bus}, determine the battery capacity and mass. The number of cells in the battery, assuming only one battery is used, is

$$N = \frac{V_{bus}}{V/cell} \tag{2.11}$$

where V/cell is the voltage per cell of the battery. The number of battery cells is rounded down to avoid overloading the bus. The battery voltage is then given as

$$V_B = N\,V/cell \tag{2.12}$$

Using Equation 2-10 to find the total cell capacity, the battery capacity is

$$C_B = C\,V_B \tag{2.13}$$

The mass is calculated by dividing the battery capacity by the specific energy density, as in Equation 2-14.

$$m_B = \frac{C_b}{E_B} \tag{2.14}$$

Figure 2.3 shows the charge/discharge characteristics of the energy storage device of a spacecraft. A flat discharge curve that extends through most of the capacity and little overcharge is preferred since overcharging degrades most batteries. Charge imbalances may also degrade and stress batteries, resulting in shortened life, so the electrical characteristics of the battery cells also are matched.

2.1.3 Power distribution and regulation

The primary components of the power distribution system are cabling, fault protection, and switching gear. These three characteristics are subject to the spacecraft's power loads and switching requirements. Low power loss, low mass, durability, cost efficiency, reliability, and good power quality are desirable power distribution system characteristics.

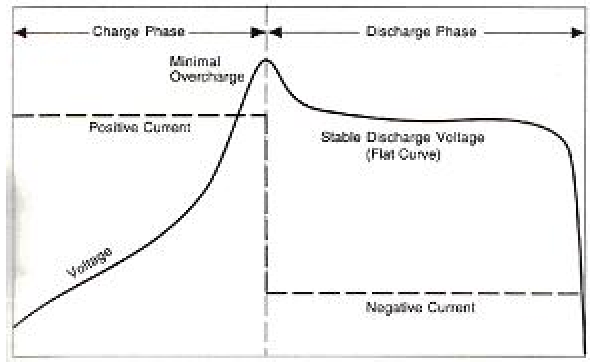

Figure 2.3: Profile of Charge/Discharge Voltages for Batteries.[8]

Most spacecrafts utilize mechanical power due to their reliability, durability, and simplicity. Electrical switches should be used sparingly to prevent faults in the system.

The design of the power distribution system is determined by the power load profile of a spacecraft. Most satellites use DC power systems typically in the range of (5-270 Vdc). Larger spacecrafts such as the International Space Station may use a single-phase (115 Vrms, 60Hz) AC profile or a three-phase (120/440 Vrms, 400 Hz) profile.[8] Alternating current power systems are used only when necessary due to the inefficiency of power converters.

The power distribution system is divided into two categories of power conversion. Centralized systems regulate the power through the bus; for loads requiring further conditioning, controllers are set up at the bus exit. De-centralized systems have an unregulated bus. To condition the power, converters are set up at every exit of the bus. Smaller power systems typically use centralized system to simplify the design and reduce the mass. Larger power systems as found in the International Space Station typically use a de-centralized system due to special power requirements.

The cabling component of the power system is important for minimizing the mass. The cabling and harness system accounts for 10-25% of the total power system mass.[8] Therefore, properly sizing the cable both physically (lengths) and electrically (amperage) is important. the length of cable should be kept to a minimum and the cable must be sized to safely carry

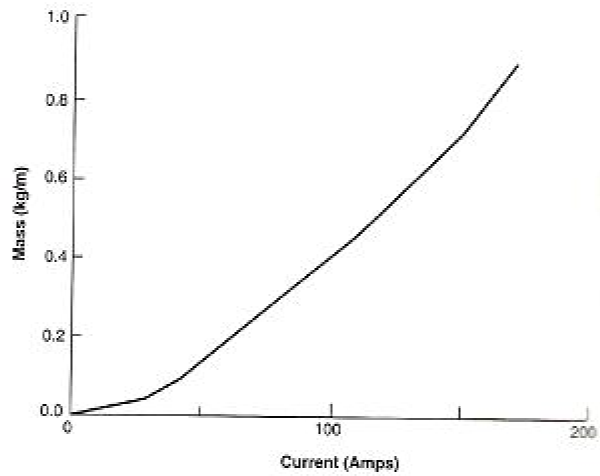

Figure 2.4: Comparing the effect of increased amperage requirements on cable mass.[8]

the specified power load. Figure 2.4 shows the effect of high amperage loads on the mass of the cabling system. Clearly, amperage must be accounted for during load designs. It is recommended that spacecraft utilize low-amperage systems under 30 amperes.[8]

The distribution system also requires fault protection. Faults can be devastating to missions by draining power reserves and stressing both cabling and switching systems. Therefore, fault detection and circumvention systems are necessary in all power systems. Fault correction is usually accomplished using fuses strategically placed in the power bus. Knowing where faults have occurred is important to the continuation of a mission, thus the designer may want to add fault detection circuits to the power distribution system.

The power regulation system falls into three categories: controlling the power supply, regulating the bus voltage, and charging the batteries. The power flow from the power supply to the spacecraft systems and batteries can be achieved through two distinct methods. The first, peak power transfer (PPT), extracts the exact amount of energy needed by the spacecraft. The second, direct-energy-transfer (DET) draws a constant amount of energy from the power source. Any excess energy is sent to a bank of resistors which dissipate the energy, thus regulating the power supply.

Peak power transfer systems are designed to dynamically adjust the power supply voltage to the proper level. Peak power transfer systems excel at gathering and storing large amounts of energy in short periods of time. However, since PPT systems use dc-dc converters they will use 4-7% of the total system power.[8] Typically, PPT systems are used for missions requiring large power at BOL and limited to 5 year life-cycle.

Direct energy transfer systems remove excess power generated by the power supply by placing the shunt resistors in parallel with the supply. When power is no longer needed, current is drawn from the array and shunted to the resistors. DET requires no power, and therefore is extremely efficient. DET systems also have the advantage of simple design, low mass, and high efficiency even at EOL.

Bus regulation is split into three categories unregulated, quasi-regulated, and fully regulated. The Voltage of an unregulated bus varies by approximately 20% from charge to discharg.[8] The unregulated bus voltage is equal to the battery voltage. The quasi-regulated system regulates the bus only when the batteries charge. When the batteries have completed charging the system becomes unregulated. Quasi-regulated systems tend to be inefficient and noisy (high electromagnetic interference) when used with PPT systems. The fully regulated system regulates power when the batteries are charging and discharging. Although fully regulated systems are inefficient, they are commonly used in systems with low power and a need for a highly regulated bus. Fully regulated systems simplify design integration, but are inefficient, and produce electromagnetic interference when combined with a PPT system.

The final component of the power regulation system is the battery configuration. Batteries can be arranged in either parallel or series configurations. Parallel configurations are simple to design and cost effective. However when charging in parallel, one battery receives the bulk of the charge over time causing battery degradation, which shortens the lifecycle of a mission. Parallel configurations are recommended for missions with lifecycles less than 5 years. Series configurations provide flexible battery placement and, since batteries are charged independently (to their own capacity limits), battery degradation is minimized. Series systems inherently cost more and require a more complex design and therefore should only be used when the mission lifecycle exceeds 5 years.

2.2 Thermal

The thermal control system is divided into three groups of subsystems: heat dissipation, insulation and shielding and active components. Each of these subsystems must be optimized to keep the spacecraft within operational temperature ranges. The two main principles in determining thermal control of the spacecraft are the first law of thermodynamics

$$Q_{in} = Q_{out} \tag{2.15}$$

where Q_{in} is the heat added to the spacecraft and Q_{out} is the heat removed from the spacecraft; and the Stefan-Boltzman equation of black-body radiation

$$Q = \epsilon\sigma AT^4 \tag{2.16}$$

where σ is the Stefan-Boltzmen constant, T is the surface temperature, a is the absorptivity, ϵ is the emissivity of the material. Equation 2-15 applies for all radiating bodies including planets or moons, the sun, and radiators on the spacecraft

2.2.1 Insulation and Shielding

To reduce the load on the radiators and to protect the exterior of the spacecraft from the environmental temperature factors. Insulation and heat shielding are necessary. Insulation typically covers most of the exterior of the spacecraft and serves to reduce the external heat load. A low absorptivity, α, is necessary for effective insulation.

Reflective coatings are also important for keeping the spacecraft within survival temperatures. They will also serve to reduce the load on both the insulators and the radiators. The reflective coatings are films or paints that are either white or shiny such that they have high reflectivity, ρ, so that less radiation reaches the insulation and the surface of the spacecraft. the relationship between these radiation properties and the transmitivity, τ is

$$\rho + a + \tau = 1 \tag{2.17}$$

Equation 2.14 can be modified to determine the external heat load on the spacecraft with insulation

$$Q_{external} = (Q_{solar} + Q_{albedo} + Q_{earthIR})\epsilon_{insulation} \tag{2.18}$$

If the mission requires atmospheric interactions such as re-entry or aero-braking, heat shielding must be provided by the thermal control system to ensure that the hull temperature does not exceed the operational range of the structural elements; there will be heat convection with the atmosphere in this case. Newton's law of cooling is used to determine convection heat transfer

$$Q_{convection} = hA(T_s - T_f) \tag{2.19}$$

where h is the convection heat transfer coefficient, A is the surface area of the spacecraft, T_s is the surface temperature of the spacecraft and T_f is the atmospheric temperature.[2] The temperature of the atmosphere in the case of atmospheric re-entry is elevated due to free molecule heating by friction of the air with the surface of the spacecraft.

2.2.2 Heat Dissipation

Heat dissipation is typically accomplished by radiators. Radiators are panels mounted either on the exterior surface of the spacecraft or on extension arms like solar cells. To optimize the spacecraft's radiators, sources of heat on the spacecraft first be determined.

The most significant environmental heat source on an orbit is the sun. The sun radiates with constant power at all times so it is easy to predict the sun's heat. Near the Earth (average orbital distance), the solar constant is 1367 W/m^2 and this value decreases as r^{-2} as the distance from the sun increases. Because of this variance on distance, heat from the sun only changes ± 3.5% over the course of the earth's orbit due to the eccentricity of the orbit. For an Earth orbiting satellite, this heat source remains fairly constant; however, if the satellite leaves the earth's orbit, it will encounter varying radiation from the sun which can be predicted by the path of the spacecraft throughout its life. The sun radiates light predominantly in three ranges: 7% ultra violet (UV), 46% visible light, and 47% short wavelength infrared (IR). The sun's IR radiation is at a shorter wavelength than that of room temperature black-body radiation; thus it is possible to create spacecraft surfaces that simultaneously reflect sun light and emit radiation from the spacecraft. The sun's radiation heat is defined as

$$q_{solar} = KF \tag{2.20}$$

where K is the solar constant F is the geometrical factor based on the position of the spacecraft relative to the sun;

Besides solar radiation, two other significant heat sources in outer space exist: albedo reflection and IR emission by planets, moons, and other large orbiting bodies. Albedo reflection is the sunlight incident on a planet or planetoid which is reflected back into space. Albedo intensity from earth is dependent on several factors: it is greater over land than water, it is greater with smaller local solar elevation angles, and it is greater with increased cloud cover. All of these factors generate an increase in albedo with increasing latitude. The Earth's albedo radiation heat is defined as

$$q_{albedo} = aI_{solar}\rho_{albedo}F_{albedo} \tag{2.21}$$

where I_{solar} is the solar intensity, ρ_{albedo} is the Earth's albedo, and F_{albedo} is the geometrical factor based on the position of the spacecraft relative to the Earth.

Infrared emissions arise from non-reflected sunlight absorption by planets and later reemission. The IR emission is dependent on the local surface temperature and cloud cover. Thus IR emission decreases with increasing latitude. The Earth's IR emission heat is defined as

$$q_{earthIR} = \epsilon I_{EIR}F_{EIR} \tag{2.22}$$

where ϵ is the emicivity of the earth's surface, I_{EIR} is the infrared intensity and F_{EIR} is a geometrical factor based on position of spacecraft relative to the Earth and Sun

Tables 11-45 a and b[8] show albedo and IR emission values for various inclination angle of earth orbits and average values for other planets. The local variation of IR emission is less than that of albedo reflection. However, emission by planets has a similar wavelength to that of spacecrafts; therefore, spacecraft radiators can be back loaded by IR emissions of the central body.

In addition to environmental factors, waste heat produced by the spacecraft systems must also be dissipated by the radiators. Values for waste heat must be provided by the designers of the other subsystems.

Once all the heat sources are determined, Equation 2-14 is adapted to solve for the heat dissipated by the radiators

$$Q_{radiator} = Q_{external} + Q_{internal} \tag{2.23}$$

where $Q_{external}$ is the heat input from the environment equal to the sum of the solar, albedo and Earth IR heat incident upon the ship; and $Q_{internal}$ is the waste heat generated inside the spacecraft.[8] The required radiator area, $A_{Radiator}$, and thus, the radiator mass, $m_{Radiator}$, can be determined by adapting Equation 2.15

$$Q_{radiator} = \epsilon \sigma A_{Radiator} T^4 \tag{2.24}$$

$$m_{radiator} = A_{Radiator} x mass/vol \tag{2.25}$$

where mass/vol is the mass per unit volume of the radiator material.

2.2.3 Internal Systems and Active Components

The internal subsystems of the thermal control system are primarily responsible for keeping all of the spacecraft systems in every functional divison within their operational temperature ranges during active and within survival temperatures otherwise.

Batteries typically require an operational temperature range of 0 to 15 C and a survival range of -10 to 25 C; power box baseplates typically require an operational temperature range of -10 to 50 C and a survival range of -20 to 60 C. If Nickel-Hydrogen batteries are used on the spacecraft, then an additional requirement is the temperature range from one cell to another must be minimized as well as temperature gradients within a single cell. Solar panel operational temperature ranges are -150 to 110 C and survival temperature ranges are -200 to 130 C; Antennas typically require an operational temperature range of -100 to 100^o C and a survival range of -120 to 120^o C; antenna gimbals typically require an operational temperature range of -40 to 80^o C and a survival range of -50 to 90^o C; CC & D H box baseplates typically require an operational temperature range of -20 to 60^o C and a survival range of -40 to 75^o C; Reaction wheels typically require an operational temperature range of -10 to 40^o C and a survival range of -20 to 50^o C; gyros typically require an operational temperature range of 0 to 40 C and a survival range of -10 to 50^o C; star trackers typically require an operational temperature range of 0 to 30^o C and a survival range of -10 to 40^o C; and hydrazine tanks and lines typically require an operational temperature range of 15 to 40^o C and a survival range of 5 to 50^o C. Special consideration must be given to solar panels, antennas and antenna gimbals, as they are external to the spacecraft and thus encounter more extreme temperature variations than other systems. If hydrazine or a similar propulsion system is used, the rest of the spacecraft must be heat shielded from this system due to its high output temperature.[8]

To keep all of these systems within operational temperature ranges it is necessary to transfer heat from these systems to the radiators or to shield these systems from the rest of the spacecraft. Conduction becomes an important phenomenon when determining heat exchanges present within the spacecraft. Conduction heat transfer behaves as follows

$$Q_{conducted} = kA\Delta T/\Delta x \tag{2.26}$$

where k is the thermal conductivity of the material, A is the area of interaction , ΔT is the temperature difference across the material and Δx is the thickness of the material.[2] Equation 2-25 is useful for determining heat transfer across different sections of the spacecraft. A high $Q_{conducted}$ is desirable to transfer heat from spacecraft components to radiators whereas a low Qconducted is desirable to isolate different sections of the spacecraft.

In special cases where the communication system requires IR detectors or optical elements, other temperature control measures are necessary. Infrared detectors require operational temperatures below 125 K and thus require special cooling systems. Super shielded passive radiators are sometimes used, and only work at the upper end of the cryogenic range. An alternative passive system is stored cryogens. Cryogens are expendable fluids which absorb waste heat reliably; however, stored cryogens are only useful for short term missions due to weight limitations. If necessary, active refrigeration systems can be used.

Optical elements require smaller temperature changes due to differing coefficients of expansion in the glass optics and the metal substrates. If the optical element includes several optics mounted to a single bench that work together, then bulk temperature control of the whole system is necessary to ensure all components are kept at the same temperature.

2.3 Environment

Environmental factors of interest are divided into four categories: Atmospheric, plasma and magnetic, radiation, and man-made concerns. These factors must be identified and modelled so that the spacecraft can be designed to deal with any concerns that arise.

2.3.1 Atmospheric concerns and modeling

The upper atmosphere effect on spacecrafts is to generate aerodynamic drag, heat, and chemical corrosion effects by highly reactive elements such as atomic oxygen.

Drag is dependent on atmospheric density, airspeed velocity, ballistic coefficient, atmospheric composition and temperature. Ballistic coefficient is typically estimated by spacecraft configuration. Spacecrafts with perigees below roughly 120 km have such short lifetimes due to drag that their orbits have no practical importance; whereas, above 600 km spacecrafts maintain their orbits longer than their operational lifetime. Altitudes between 120 and 600 km are within the Earth's thermosphere where the absorption of extreme ultraviolet radiation from the sun causes rapid increase in temperature with altitude. Heating from this radiation and its solar cycle variation has the greatest effect on satellite lifetime. Orbital decay is slow during solar minimum and fast during solar maximum. As the spacecraft's altitude decreases, radiation effects increase. Ballistic coefficient also influences the effect of solar maxima. Spacecrafts with low ballistic coefficients will respond quickly to the atmosphere and tend to decay swiftly; whereas those with higher ballistic coefficients endure more solar cycles and will decay much more slowly.

Atomic oxygen is the predominant atmospheric component at altitudes between 200 and 600 km.[8] This is a reactive form of oxygen that degrades system performance. For example, Kapton, a material commonly used for seals and insulation, erodes at approximately 2.8 micrometers per 10^{24} atoms/m^2 of atomic oxygen fluence. Fluence, F_0, over a time period T, is given by

$$F_0 = \rho_N V T \tag{2.27}$$

where V is the satellite velocity and ρ_N is the number density of atomic oxygen.[8] The number density of atomic oxygen is seen to vary with solar cycles as well. At altitudes higher than 300 km the number density of atomic oxygen at solar maximum is an order of magnitude greater than at solar minimum. Figure 2.5 shows atomic oxygen number density variation with altitude for solar maximum (F10.7 =225) and solar minimum (F10.7=75). This large solar cycle variation in atomic oxygen means spacecraft materials can be chosen based on phasing the mission to the solar cycle.

2.3.2 Plasma and magnetic modeling

The Earth's magnetic field is roughly dipolar, and given by

$$B(R, \lambda) =_f rac(1 + \sin^2 \lambda)^{1/2} B_0 R^3 \tag{2.28}$$

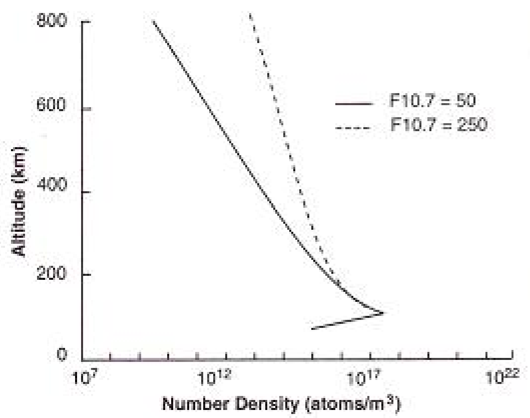

Figure 2.5: Atomic Oxygen Number Density Variation with Altitude for Solar Minimum and Maximum.[8]

Where B is the local magnetic field intensity, R is the radial distance measured in Earth radii, λ is the magnetic latitude, and B_0 is the magnetic field at the equator on the Earth's surface ($B_0 = 0.30$ gauss).[8]

The interaction between the solar wind and the Earth's magnetic field causes the magnetic field on the night side of the Earth to stretch into an elongated structure called the magnetotail, as shown in Figure 2.6. The magnetotail extends over 1,000 Earth radii parallel to the flow velocity of the solar wind and is split by a thin plasma sheet.

Ionic particles, termed the plasma environment, can cause differential charging of spacecraft components on the surface and interior. Under certain conditions, the charged components will release a damaging electrostatic arc. At altitudes near 300 km, roughly 1% of the atmosphere is ionized. In the geosynchronous environment it is essentially 100% ionized.[8]

Surface charging detrimental to spacecraft operation occurs mainly in orbits where electrons with energies of 10 to 20 keV dominate the electron current from the plasma to the spacecraft.[8] At low altitudes this charging occurs only at high latitudes where auroral electrons collide with the spacecraft as it passes through an otherwise cold, low-density plasma. For other low altitude locations, low energy electrons usually develop enough current to keep

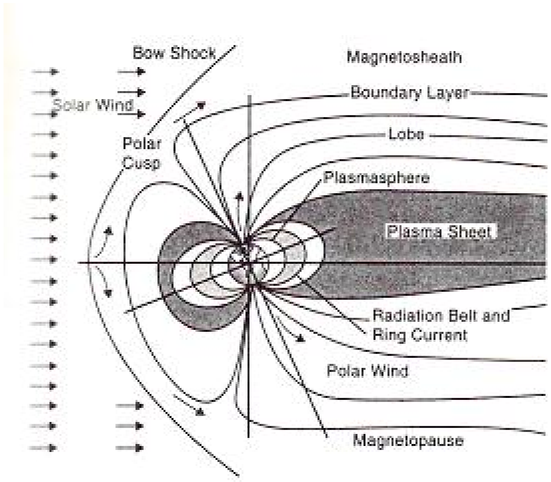

Figure 2.6: Cross section of Earth's Magnetosphere.[8]

electric fields below breakdown levels, preventing an electrostatic arc.[8]

In higher orbits, surface charging occurs during magnetospheric substorms between the longitudes corresponding to midnight and dawn. The electron flux distribution during a substorm can be approximated by summing a cold and hot Maxwellian distribution. The hot component has a density of about 2.3 cm^{-3} and a temperature of 25 keV; the cold component has a density of 0.2 cm^{-3} and a temperature of 0.4 keV.[8]

2.3.3 Radiation Modeling

Energetic charged particles can be found in the trapped radiation belts, solar flare protons, and galactic cosmic rays. Microelectronic devices, solar arrays, and sensors can be degraded by the total dose effects of this high energy radiation. A single energetic particle can also cause single event phenomena within microelectronic devices, temporarily disrupting or permanently damaging components.

The Van Allen radiation belts are a permanent hazard to orbiting spacecrafts. These belts consist of electrons and ions having energies greater than 30 keV and are distributed nonuniformly within the magnetosphere. A magnetic L-shell is the surface generated by

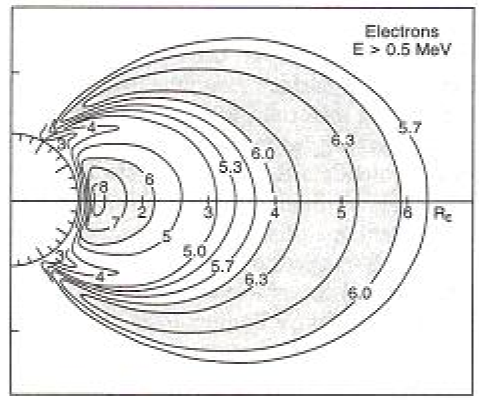

Figure 2.7: Electron Belts of the Inner and Outer Zones.[8]

rotating a magnetic field line around the dipole axis which approximately satisfies the equation

$$R = L\cos^2\lambda \tag{2.29}$$

where, R is the distance in Earth radii from the idealized point dipole near the Earth's center, and λ is the magnetic latitude. A generalized concept of L takes account of the way higher harmonics of the main geomagnetic field perturb the motion of charged particles from their dipolar trajectories.[8] This concept is used for mapping the trapped radiation environment. Figure 2.7 illustrates Earth's electron belts and their preferential toroidal population. The pair of toroidal regions are centered on the magnetic shells L 1.3 (inner zone) and L 5 (outer zone).

The radiation dose rate for the desired orbit must be computed to determine the appropriate amount of shielding for spacecrafts. This is done by applying a radiation transport computation to the time average space environment. The total radiation dose consists of three components: proton dose, electron dose and the bremsstrahlung X-ray dose produced by the interaction of electrons with the shielding material. In low Earth orbits, energetic protons in the inner radiation belt contribute the most to the total radiation dose. Radiation dose is strongly dependant on altitude; below 1,000 km the dose increases by approximately the 5th power of the altitude, whereas at geosynchronous altitude, the greater than 5 MeV proton dose is negligible and the bremsstrahlung dose dominates the electron dose for shield thicknesses greater than 1 cm.[8]

Solar particle events are associated with solar flares and refer to rapid increases in the flux of energetic particles lasting from several hours to several days. A typical time evolution of a solar particle event depends on the time evolution of the original solar flare, how long the energetic particles take to diffuse within the solar corona, and how the particles propagate within the interplanetary medium. Importance of a single solar particle event depends on its maximum intensity, its duration, and the abundance of the highest-energy components and heavy nuclei. The intensities of typical solar proton events closely follow a log-normal distribution. Hence, a few individual events can dominate the total proton fluence observed over a complete solar cycle. Solar array outputs, for example, typically degrade by a few percent following exposure to fluences above roughly 109 cm-2 at energies over about 1 MeV.[8] However, actual degradation rates depend on cover glass thickness, solar cell thickness and cell age.

Galactic cosmic rays are energetic particles originating from outside the solar system. Past studies of galactic cosmic rays suggest they experience solar cycle modulation. Cosmic rays are hazardous because a single particle can cause malfunctions in common electronic components such as random access memory, microprocessors and hexfet power transistors. This is termed a single event phenomenon when a single passing particle causes the malfunction. Cosmic ray energy loss depends primarily on the square of the particles charge, Z, and can be increased if the particle undergoes nuclear interactions within an electronic component.[8] This means that lower-charge ions, the most abundant, deposit as much energy in a device as less abundant, higher-charge ions. When the galactic cosmic ray leaves

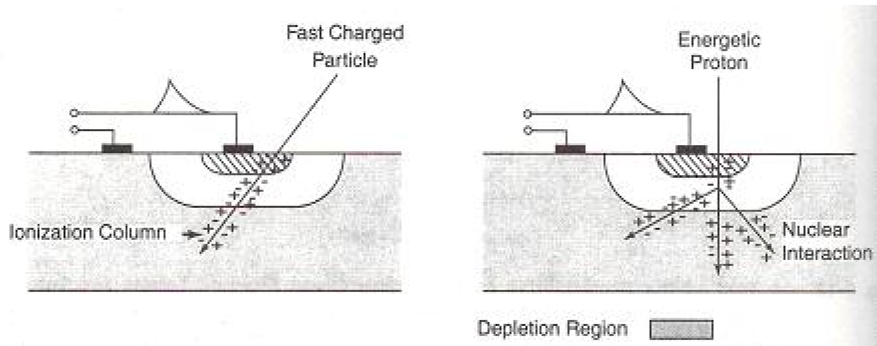

Figure 2.8: Schematic of Galactic Cosmic Ray Energy Deposition in an Electronic Device.[8]

electron-hole pairs in a depletion region of an electronic device, the electric field in that region sweeps up the pairs. This process is shown schematically in Figure 2.8.

Single event phenomena can cause three different effects in electronic components: single event upset, single event latchup and single event burnout. Single event upset, or bitflip, neither damages the part nor interferes with its operation. Single event latchup causes the part to hang up and draw excessive power until the device is turned off then back on. This extra current draw can destroy the device if the power supply cannot handle the current. Single event burnout causes the device to fail permanently. On-orbit failure rates due to single event upsets in memory devices with well defined sensitive volumes can be reasonably predicted by

$$R = \frac{200\sigma_L}{L_{0.25}^2} \tag{2.30}$$

where R is the number of upsets or errors per bit day, sL is the limiting cross section, or sensitive area, of the device in cm^2, and $L_{0.25}$ is the linear energy transfer at 25% of the limiting cross section in units of MeV/mg/cm^2.[8] If experimental cross section data is not known, but device modeling data is known, then geometric data can be used with the predicted critical charge to predict the failure rate:

$$R = \frac{2x10^{-10}abc^2}{Q_c^2} \tag{2.31}$$

where a and b are device surface dimensions in μ m, c is the device depth in μ m, and Q_c is the critical charge in pC.[8] These two equations predict upset rates in the geosynchronous orbit for solar minimum conditions with reasonable accuracy. Single event upset rates in complex devices such as microprocessors, or single event latchups or burnouts in any device can not be reasonably predicted. For these cases, simulated accelerator test observations and flight performance information from similar devices must be relied upon.

2.3.4 Man Made Space Environment

The ability of a space system to perform its intended function after being exposed to a stressing environment created by an enemy or hostile agent is termed survivability. An attribute defining the environmental stress level that a space system can survive is hardness. In the aerospace industry the purely natural environments are not considered in the definition of hardness and survivability because technology developed over the last thirty years has made it possible for all spacecrafts to withstand natural environments. A military space system or commercial satellite must be survivable in a man made environment if its services are needed in times of high stress such as a nuclear war.

The greatest threat to spacecraft or space systems is nuclear weapons. If a nuclear weapon directly attacks a spacecraft or any of its systems, the object of attack will be completely destroyed. If a nuclear bomb explodes in space, approximately eighty percent of the energy from that weapon appears in the form of X-rays. The remaining energy includes gamma rays, neutrons, radioactivity, and kinetic energy.

Just after detonation, nuclear bomb material is at 10-100 million Kelvin causing X-radiation to occur. As a first approximation, the hot bomb material acting like a black body will quickly radiate energy. According to the Stefan-Boltzmann law:

$$E = \sigma T^4 \tag{2.32}$$

Where E is the energy in W/m^2, σ is the Stefan-Boltzmann constant equaling 5.67 10-8 W m^{-2} K^{-4}, and T is the absolute temperature in Kelvin.[8] The X-ray fluence, F_x, is given by

$$F_x = \frac{f_x}{4\pi R^2} = 6.4Y/R^2 \tag{2.33}$$

and is in the numerical form of cal/cm^2. R is distance in kilometers from a nuclear detonation of yield Y measured in kilotons. The fraction of the energy emitted as X-rays is f_x, which is approximately equal to 0.8.[8]

Neutron radiation depends on the design of the nuclear weapon. From the fission of approximately 1.45 x 10^{23} nuclei, one kiloton of equivalent nuclear energy arises producing 2 or 3 neutrons.[8] During the few tens of nanoseconds of energy generation, approximately half of those neutrons escape. Accordingly, the neutron fluence is approximated by

$$F_n = \frac{1.4x10^{12}Y}{R^2} n/cm^2 \tag{2.34}$$

Again, R is the distance in kilometers from a nuclear detonation of yield Y measured in kilotons.[8]

Prompt radiation is gamma radiation emitted during the actual nuclear burn time. Fission reactions, neutron captures, and inelastic neutron scattering events occur during intense generation of nuclear energy resulting in prompt gammas. The prompt gamma rays' total energy and energy distribution depend on the nuclear weapon's specific design. The dose, D?, known as energy deposited per unit mass in silicon semiconductor material from prompt gamma radiation is used to calculate preliminary survivability and is expressed as

$$Dy = \frac{7x10^4 Y^{2/3}}{R^2} rads(Si) \tag{2.35}$$

The range R from a nuclear burst in space is measured in kilometers, and Y is in kilotons.[8]

Delayed radiation is gamma rays emitted after the nuclear burn time. Residual radiation occurs when radioactive fission products decay. Products include gammas, neutrons, positrons, and electrons. The decay rate from residual nuclear radiation is nearly constant for about the first second after a nuclear explosion. Following the approximate law, the dose rate one second after detonation is

$$R = R_0(t/t_0)^{-1/2} \tag{2.36}$$

When the dose rate at the reference time t_0 is R_0, the dose rate R is usually in rads/hr at any elapsed time, t, after to. The -1.2 power of time is an approximation because the fission products causing the dose rate contain more than 300 different isotopes of 36 elements from the periodic table. Six months after the nuclear explosion, the dose rate approximation is accurate to within 25 percent.[8]

Electromagnetic pulse, a secondary effect of nuclear weapon detonations, happens when x-rays and gamma rays impinge upon the upper atmosphere. An electron flux is created which radiates in the rf region of the spectrum with most of its spectral energy in the 1 MHz to 100 MHz range.[8] If a satellite is not protected from electromagnetic pulse, the rf energy arriving at the satellite will induce currents and voltages that may damage or kill the satellite.

Electrons caused by a nuclear weapon detonated at a high altitude join the naturally occurring Van Allen radiation belts. As a satellite repeatedly traverses the Van Allen belts, the absorbed dose in unshielded materials may increase as a result of the electron flux increasing by many orders of magnitude. Solid-state electronic circuits normally enclosed in aluminum are protected with an aluminum wall thickness ranging from 0.01 in to 0.3 in.[8]

X-ray energy is absorbed almost instantaneously in solid material through the photo-electric effect and Compton scattering. The photo-electric effect causes bound electrons in the material to be ejected from their orbitals with kinetic energy equal to the difference between the energy of the incident photon and the atom's ionization energy. The incident photon's absorption per atom is proportional to the 5th power of the atomic number of the absorbing material and is inversely proportional to the 7/3 power of the incident photon's energy. Therefore, materials with a high atomic number shield against X-rays more effectively. For incident-photon energy ranging from 1 to 20 keV, the absorption cross section decreases dramatically.[8]

An elastic scattering event in which an electron receives some of the energy of the incident photon and the incident photon changes direction is termed Compton scattering. When Compton scattering occurs, the incident photon's energy decreases and its wavelength increases, causing electronic circuits to malfunction. The energy of the free electrons appears as heat in the material.

Neutrons colliding with atomic nuclei impart energy to the atoms of the material and displace the atoms from their normal position in the lattice. Changing the lattice structure can seriously harm solid-state electronic devices due to their dependence on the characteristics of their lattices. Neutrons can cause solid-state devices to stop working at fluences greater than about 1011 n/cm2.[8]

A nuclear weapon detonated in space near the Earth radiates electromagnetic energy creating large-scale ionization in the bomb material and in the atmosphere due to interaction with the Earth's magnetic field. Radioactive debris contributes beta particles, which are positrons and electrons that move Along the lines of force of the geomagnetic field. These energetic particles interact with the atmosphere as the magnetic field lines enter the atmosphere creating more electrons. These free electrons absorb and reradiate rf energy and create phase and amplitude changes as they refract the electromagnetic waves of the radio communications links between ground and satellite.

The attenuation, a, is determined using the theory of electromagnetic propagation:

$$a = \frac{4.4x10^4 N_e \nu}{(2\pi f)^2 + \nu^2} dB/km \tag{2.37}$$

Ne is the number of electrons per cm^3, ν is the frequency in Hz of collision of electrons with ions, atoms or molecules and f is the frequency of the electromagnetic radiation in Hz.[8] The form of the equation and the fact that the density distribution of the atmosphere is approximately exponential are used to infer the absorptive behavior of re signals as functions of frequency because the values of the parameters are nearly impossible to obtain. For any given radiation frequency, the absorption is at a maximum between 60 and 80 km where the absorption varies with the inverse square of the radiation frequency [Larson ed.2, 220]. Thus, the highest communication frequency must be chosen to minimize the absorption in the satellite-to-ground links due to nuclear weapons.

Developed as potential ground based or space-based weapons, high-power lasers' flux in power per cross-sectional area is given by

$$F = \frac{PD^2}{\pi R^2((1.22\lambda Q)^2 + (JD)^2)} \tag{2.38}$$

where D is the laser objective diameter, P is the average output power, Q is the quality of the laser beam, is the wavelength of the laser, R is the range from the laser to the target, and J is the angular jitter of the beam measured in radians. Laser weapons can have a beam quality, Q, of 1.5 to 3.0. Equation 2-36 is valid for pulsed and continuous-wave lasers and is correct for the average flux from a pulsed laser.[8]

An antisatellite weapon using fragmentation pellets can attack spacecrafts in low-Earth orbit. Launched from the ground, the weapon achieves an orbit with nearly the same orbital elements as the target spacecraft. The weapon is brought close to the target spacecraft through the use of radar or optical guidance; the target spacecraft is then damaged or killed by impact of many small fragments created from a high explosive.

High-power microwave weapons generate a beam of rf energy intense enough to damage or interfere with a spacecraft's electronic systems. Frequencies of operations of the high-power microwave weapons range from 1 to 90 GHz;[8] within this range are the commonly used frequencies for command, communication, telemetry, and control of most modern spacecrafts. A spacecraft's antenna will amplify the received radiation from the weapon possibly damaging rf amplifiers, downconverters, or other devices in the front end of a receiver.

Weapons using particle accelerator technology must be based in space because particles cannot penetrate the atmosphere. Particles would be accelerated as negative hydrogen or deuterium ions. By stripping an electron as they emerge from the accelerator, the ions would then be neutralized. To damage a satellite, the particles must be electrically neutral to avoid being deflected by the Earth's magnetic field.

Hardening is the most effective survivability option. Presently, hardening is used to prevent damage or electronics upset from nuclear-weapon effects. Other strategies for achieving spacecraft survivability include mobile ground stations, satellite decoys, and self-defense or escort defense. If a satellite has more powerful thrusters than the ones used for attitude control, then the satellite can maneuver or dodge an antisatellite attack. An on-board attack reporting system is useful for total system survivability.

Chapter 3

Examples and Recommendations

3.1 Power

To design an effective power system, it is necessary to design each component effectively: power source, energy storage device and power regulation system. Designers must compute values for important characteristics of each of these systems to ensure proper selection.

3.1.1 Power Sources

The design of the power source for the spacecraft is a fundamental concern of the spacecraft design team. There are presently three major power supplies available for spacecrafts: nuclear power, photovoltaic power, and batteries. Both nuclear and photovoltaic power supplies are primary power sources meaning they provide power to the spacecraft and charge the batteries. Batteries can be used as either a primary or secondary power source. Many spacecrafts with short mission cycles such as the space shuttle use batteries as their primary power source. Spacecrafts with extended life cycles use batteries as a secondary power sources to store power generated by the primary source until it is needed (i.e. photovoltaic satellites eclipsed by the earth).

Nuclear power is a plentiful source of power for an extended interplanetary mission. Dynamic and static power sources both use nuclear power as their primary source of energy. When considering nuclear power, the length of the mission is the primary consideration. Nuclear power systems provide ample power for earth orbiting space missions and interplanetary missions, with power levels ranging from twenty-five kilowatts to hundreds of kilowatts. Furthermore, it can maintain these power outputs for lifecycles greater than five years. Certain spacecraft characteristics must be examined to ensure system compatibility to design a spacecraft with a nuclear power source. These design factors are compatibility with spacecraft command and data handling subsystems, spacecraft attitude and control, vehicle survivability, thermal interactions, and power quality.

The following is an example of the steps a designer follows when developing a nuclear power system. The designer must consider multiple choices for the reactor type, the power conversion method, and the thermal management system. The selection of these components is based on the requirements of the mission. The designer should also consider the future availability of these technologies. If the selected technology will not be available by the launch date, then the technology is not suitable for use.

For the following example, a 100 kW power requirement will be examined; the power source will have a lifetime of 7 yrs, a size of 6.1 m, a mass of 3000 kg, a power distribution of 100 to 400 V DC, a reliability factor of 0.95, a lifetime limit for neutron fluence to payload of 1013 n/cm^2, and a lifetime limit for gamma dose to payload of 5 x 105 rads. The power system must be subcritical when immersed in water and on ground impact for safety purposes. This example was carried out by Los Alamos National Laboratory.[5]

The temperature range of the reactor must be considered to select a suitable material to insulate the nuclear material. Table 3.1 shows the materials used as insulators for different temperature ranges.

Level 0 materials are used at temperatures below 1000 K; Level I materials are used at

Table 3.1: Temperature requirements and suitability of materials for space nuclear power systems

Material Level	Applicable Temperature Range	Heat Rejection Applications	Reactor Applications	Dynamic Converter	Thermoelectric Converter
0	< 1000 K	Be, Al, Ti, S.S.			
I	$<$ 1200-1300 K		Steels	Superalloys	SiGe
II	< 1400 K		Nb and Ta Alloys	Nb and Ta Alloys	SiGe GaP
III	$<$	W-Mo-Re Alloys	W-Mo-Re Alloys	SiGe GaP with Improved Coating	
IV	> 1500 K		Ceramics	Ceramics	Carbides/Sulfates

temperatures below 1200-1300 K; Level II materials are used at temperatures below 1400 K; Level III materials are used at temperatures below 1500 K; and level IV materials are used at temperatures above 1500 K.[5]

One of the most important parts of nuclear power source design is selecting a reactor type; the designer must consider whether the reactor type is a pressure vessel system because pressure vessel systems may violate safety requirements during reentry. The available reactor types are gas cooled reactor, liquid metal cooled reactors, heat pipe reactors, and thermoionic reactors.

Table 3.2 shows examples of reactors that have been used in space programs, the dates of operation, the power outputs, the converter types, the reactor fuel, the reactor mass, and the core temperature. The Topaz program is a fission reactor system using thermoionics. The SAFE-400 program is an example of a heat pipe reactor system.[4] The SNAP-50 program of the early 60's implemented a liquid metal cooled reactor and was capable of producing 300-1000 kW of electricity. The 710-Program of the early 60's used gas cooled reactor and produced 200 kW of electricity for up to 10,000 hrs.

Table 3.2: Space Reactor Power Systems.[5]

	SNAP-10	SP-100	Romashka	Bouk	Topaz-1	Topaz-2	SAFE-400
	US	US	Russia	Russia	Russia	Russia-US	US
dates	1965	1992	1967	1977	1987	1992	2007?
kW thermal	45.5	2000	40	< 100	150	135	400
kW electrical	0.65	100	0.8	< 5	5-10	6	100
converter	t'electric	t'electric	t'electric	t'electric	t'ionic	t'ionic	t'electric
fuel	U-ZrHx	UN	UC2	U-Mo	UO2	UO2	UN
reactor mass, kg	435	5422	455	< 390	320	1061	512
neutron spectrum	thermal	fast	fast	fast	thermal	thermal/epithermal	fast
control	Be	Be	Be	Be	Be	Be	Be
coolant	NaK	Li	none	NaK	NaK	NaK	Na
core temp. C, max	585	1377	1900	?	1600	1900?	1020

For the 100 kW example above, the designer would choose a reactor type in the acceptable power range. For this example a heat pipe system is appropriate because it is in the acceptable range and it is relatively simple. The operating temperature of a heat pipe system is 1500 K and the reactor will use 93% enriched uranium using molybdenum and rhenium as the refractory alloys.[5]

Next, the designer must choose a method of heat rejection, which is closely related to the selection of the reactor type. Methods of heat rejection are pump fed loop with fin systems, heat pipe systems, liquid droplet systems, and belt systems. For the example above, the heat pipe reactor provides multiple heat removal paths.

The designer must next decide which power conversion system to use. Either static systems such as thermoelectric, thermoionics, and thermovoltaics, or dynamic systems such as Brayton, Rankine, and Stirling engines may be used. Since the reactor operates at higher temperatures than the converter, the converter must be one level below the reactor. For this example, the designer would choose a Level-II converter because the reactor operates at 1500 K and is thus classified as Level-III reactor. Direct conversion via thermoelectrics was selected for the above example.

For Earth orbiting spacecrafts, photovoltaics are the dominant power source because the sun emits large amounts of useful solar energy. A solar array must be designed so that the spacecraft's power needs during daylight and eclipse phases are met. The following is an example of a solar array design process. The solar array is designed for a 5 year mission cycle with an orbit altitude of 700 km, an eclipse duration of 35.3 min, and an average power requirement of 110 W during daylight and eclipse.[8]

The first step in designing this solar array is to determine the total amount of power the array must provide during its operational periods by summing the power requirements of the spacecraft during daylight and the charge necessary in secondary batteries so that the eclipse power needs can be met. Using equation 2-1, P_e and P_d are equal to 110 W, Te is equal to 35.3 min; T_d can be found using equation 2-2.

$$T_p = 2\pi(\frac{7078.136^3}{3.986\,\mathrm{x}\,10^5})^{1/2} = 5926.38 \text{ s} = 89.77 \text{ min.}$$

T_d is equal to T_p minus T_e; T_d is equal to 63.5 min. The efficiencies during daylight and eclipse are approximately $X_d = 0.80$ and $X_e = 0.60$. Using equation 2-1 we obtain,

$$P_{sa} = \frac{\frac{110\,\mathrm{x}\,35.5}{0.6} + \frac{110\,x\,63.5}{0.6}}{63.5} = 239.416 \text{ W}$$

Next, the designer must select the solar cell material. Usually, solar cells are composed of either silicon, gallium arsenide, or indium phosphide. Silicon is used for this example because it is approximately one third the cost per watt of the other two materials. Silicon cell efficiencies are 14.8%, Gallium arsenide cell efficiencies are 18.5%, and indium phosphide cell efficiencies of 18%.[8] The efficiency is multiplied by the solar constant to determine the power output per unit area (Po). The solar constant is 1367 W/m^2.

$$P_o = (0.148)(1367) = 202.3 \text{ W/m}^2$$

Using the power output, the designer can determine the power production capacity at BOL. Ideally, the power output would be P_o, but the manufacturing of solar panels leads to inherent degradation (I_d) which is an efficiency factor describing the quality of the solar cell. A typical value for $I_d = 0.77$.[8] We use a sun incidence angle, $\Theta = 23.5^o$, as the worst case scenario value to achieve data at less than desirable conditions. Using equation 2-2,

$$P_{BOL} = (202.3)(0.77)\cos 23.5^o = 142.86 \text{ W/m}^2$$

The amount of incident light absorbed or reflected can be improved by using: solar cell coatings, coverslides, or back surface reflectors. Table 3-3 shows different materials that are commonly used as coating materials, coverslides, and reflectors. Materials with high indices of refraction and high reflection percentages should be used as back surface reflectors.

The designer's next concern is the power production capacity at EOL. Before calculating the power at EOL, the lifetime degradation must first be calculated. Silicon degrades at a rate of 15% every 10 years [Larson, 414]. This averages to a degradation per yr of 3.75%. using equation 2-3 with a mission duration of five years,

$$L_d = (1 - 0.0375)^5 = 0.8260.$$

Using this life degradation L_d value, the power production at EOL is calculated using equation 2-4,

$$P_{EOL} = (142.86)(0.826) = 118.01 \text{ W/m}^2$$

Now, the solar array area is calculated using equation 2-5,

Table 3.3: Indices of Refraction and First Surface Reflection Losses at Normal Incidence from Several Solar Cell Cover Materails.[7]

Material	Wavelength Λ (μm)	Index of Refraction n	Reflectance r (%)
Empty Space	0 to 8	1	0
Fused silica (Corning 7940)	0.5 to 0.7	1.46	3.5
	1.0	1.45	3.4
Microsheet (Corning 0211)		1.531	4.4
Ceria-doped microsheet		1.537	4.5
FEP-Teflon		1.341 to 1.347	2.2
R63-489		1.43	3.1
adhesive		1.41	
SiO	0.4 to 1.1	1.8 to 2.0	9.6
TiOx		2.2	14.0
Silicon	0.5	4.1	37.0
	1.0	3.5	31.0
Sapphire	0.4 to 1.1	1.71	6.7
Natural Quartz	0.4 to 1.1	1.53 to 1.57	4.7
MgF2	0.4 to 1.1	1.37 to 1.39	2.6
Ta2O5	0.4 to 1.1	2.15	13.3

$$\mathrm{A}_{sa} = \frac{239.416}{118.01} = 2.029 \text{ m}^2.$$

If the specific performance of the planar array is 25 W/kg, the mass of the solar array can be estimated using Ma = (1/25) x Psa. For this example, the estimated mass is

$$\mathrm{M}_a = (0.04)(239.416) = 9.576 \text{ kg}.$$

Finally, the designer must geometrically configure the solar array to optimize energy collection and efficiency. With the completion of the power generation process the designer must look to methods of storing energy, primarily battery storage.

3.1.2 Energy Storage

Battery selection is an essential design requirement that affects all spacecraft systems. Batteries store energy for use during eclipse periods, and during peak operating periods. Without batteries, a spacecraft orbiting the earth using photovoltaics as its primary power source would be without power for 40% of its orbital period. Factors such as maximum DOD, battery mass, and charge rate must be examined to select the appropriate batteries for a mission.

For the following example, we study a satellite in Low Earth Orbit with a 1500 W power load. A spacecraft in this orbit encounters an eclipse period lasting 36 minutes. This spacecraft uses a nickel-cadmium battery with a bus voltage of 28 V dc.

The number of cells needed, N, in the battery is found using equation 2-11.

$$\mathrm{N} = \frac{28\,V}{1.25\,V/cell} = 22.4$$

The number of cells is rounded down to 22 to avoid overloading the bus voltage, giving a battery voltage V_B from equation 2-12

$$V_B = 22 \text{ x } 1.25 \text{ V} = 27.5 \text{ V dc.}$$

Assuming a maximum DOD of 70%, the total capacity, C, of the battery cells is found by rearanging equation 2-10.

$$\mathrm{C} = \frac{1500\,\mathrm{W\,x}\,36\,\mathrm{min}(\frac{1\,\mathrm{h}}{60\,\mathrm{min}})}{0.70\,\mathrm{x}\,27.5\,\mathrm{V}} = 46.8 \text{ Ah.}$$

yielding a battery capacity, C_B, from equation 2-13

$$C_B = 46.8 \text{ Ah x } 27.5 \text{ V} = 1286 \text{ Wh}$$

A NiCd battery with this capacity has a mass m from equation 2-14

$$\mathrm{m} = \frac{1286\,\mathrm{Wh}}{25\,\frac{\mathrm{Wh}}{\mathrm{kg}}} = 51.4 \text{ kg.}$$

using a conservative energy density estimate taken from Table 2.5. By comparison, a NiH2 battery using an individual pressure vessel design has a mass m of

$$\mathrm{m} = \frac{1286\,\mathrm{Wh}}{35\,\frac{\mathrm{Wh}}{\mathrm{kg}}} = 36.7\ \mathrm{kg}.$$

The NiH2 battery utilizes the same number of cells as the NiCd and the voltage per cell is the same. However, the higher energy density of the NiH2 decreases the mass by 29%.

For a Li-ion battery, the voltage per cell is three times higher for this scenario the number of cells needed is

$$\mathrm{N} = \frac{28\,\mathrm{V}}{3.6\,\mathrm{V/cell}} = 7.78,$$

or 7 cells, approximately one-third the cells needed for the previous two cases. Completing the previous analysis for Li-ion yields a battery voltage of

$$\mathrm{V}_B = 7 \text{ x } 3.6\ \mathrm{V} = 25.2\ \mathrm{V\ dc}.$$

a total cell capacity of

$$\mathrm{C} = \frac{1500\,\mathrm{W} \,\mathrm{x}\, 36\,\mathrm{min}(\frac{1\,\mathrm{h}}{60\,\mathrm{min}})}{0.70 \,\mathrm{x}\, 25.2\,\mathrm{V}} = 51.0\ \mathrm{Ah}.$$

and a battery capacity of

$$\mathrm{C}_B = 47.1\ \mathrm{Ah} \text{ x } 27.3\ \mathrm{V} = 1286\ \mathrm{Wh}$$

The mass of the Li-ion battery is found to be

$$\mathrm{m} = \frac{1286\,\mathrm{Wh}}{110\,\frac{\mathrm{Wh}}{\mathrm{kg}}} = 11.7\ \mathrm{kg}.$$

This is 77% less mass than the NiCd battery and 68% less than the NiH2 battery.

It should be noted that this battery capacity is the same value as those obtained from the NiCd and NiH_2 analysis; this is the case for all batteries chosen for this example. The difference between the batteries is the number of cells needed to achieve this capacity or the mass of the cells.

The analysis of the battery types for the previous example shows the incentive for using Li-ion batteries on spacecrafts, where extra mass increases launch and power costs. Mass is not, however, the only factor in selecting the battery type. One drawbacks of Li-ion batteries is that they are relatively new to space applications; whereas extensive data for most space applications exists for NiCd batteries. Li-ion batteries are also more expensive than established batteries, so cost is another factor that may make Li-ion batteries less desirable. NiCd batteries are much cheaper than both other types of batteries and have long lives, making them a suitable choice for spacecrafts that discharge and recharge more often, such as LEO spacecrafts, which may encounter nearly 12,000 cycles in two years.[6]

The charge rate of the battery must also be considered. The charge rate is given as

$$\text{charge rate} = \frac{\text{Capacity}}{15\,\text{h}} \tag{3.1}$$

where capacity is the total capacity of the battery cells.[6] The 1/15 factor is a conservative estimate and higher recharge rates may be used, but may damage certain types of batteries. For NiCd batteries, the charge rate is

$$\text{charge rate} = \tfrac{46.8\,\text{Ah}}{15\,\text{h}} = 3.12\text{ A}$$

while Li-ion batteries have a charge rate of

$$\text{charge rate} = \tfrac{51.0\,\text{Ah}}{15\,\text{h}} = 3.4\text{ A}$$

For a LEO spacecraft, 40% of the time is spent discharging, while 60% is spent charging; whereas GEO spacecrafts only spend 8% of the time discharging.[6] This would cause the charge rate to be a more important factor than the maximum DOD.

3.1.3 Power Distribution and Regulation

The design of the power regulation and control system iincludes three aspects; power supply control, bus voltage regulation, and battery charging. First, the electrical control subsystem must be designed; The power source, battery charging scheme, and spacecraft heating must be considered. Only two electrical control subsystem types are available: PPT and DET. PPT control systems extract the exact amount of energy required by the system whereas DET control systems collect the maximum amount of energy dissipating excess energy to shunt resistors. The electrical arrangement of PPT and DET control systems components are determined by the method of bus regulation. Figure 11-13[8] shows the electrical configuration of PPT and DET control systems for unregulated bus with parallel batteries, unregulated bus with linear charge current control, quasi-regulated bus with constant current chargers, and fully regulated bus systems. PPT regulation systems are placed in series with the power source and load; DET regulation systems are placed in parallel with the power source and load. PPT control systems can collect large amounts of energy in short periods of time by dynamically changing the peak voltage to optimize system performance; however, PPT systems consume only 4-7% of the total power. PPT systems are typically used for missions under 5 years with high BOL power requirements. The DET system is unable to dynamically change the peak voltage, but is extremely efficient. DET control systems require no power because excess power is shunted by a bank of resistors. Consequently, DET control systems are simple, lower mass, and have a higher efficiency at EOL. For missions with limited lifecycles ($<$ 5years) and high BOL power requirements PPT control system are used. Missions requiring high efficiency at the EOL well beyond five years use DET power control systems.

The designer must also determine the bus regulation system. There are three bus regulation methods in use: unregulated, quasi-regulated, and fully regulated. Typically, unregulated busses are used on large spacecraft, such as the International Space Station, which has high power consumption and highly specialized payloads. High power and special power conditions would require power converters at each load, thus making a regulated bus voltage unnecessary. Quasi-regulated systems are only regulated during battery charging. The advantage of quasi-regulated system is control of the battery recharging process. Many batteries require a highly controlled recharging process to prevent physical and electrical damage to the batteries. When the batteries have completed charging, the bus becomes unregulated. Fully regulated power buses allow for simplified design integration, but are also the most complex bus systems. Fully regulated and quasi-regulated buses both have low efficiency and high electromagnetic interference if used with PPT. The final decision concerns parallel or series charging of the batteries. For missions less than five years parallel charging is recommended and for longer missions independent charging is required to avoid over charging the batteries.

Power distribution components are composed of cabling, fault protectors, switcher gears, power converters, and power buses. Selection for power distribution components relies heavily on the power load of the spacecraft's payloads. An often overlooked, but important component of the power distribution system is cabling. Cabling accounts for 10-25% of the electrical subsystems mass. Poor cabling systems have either poor placement of components requiring unnecessary extensions of cables or power intensive payloads larger than 30amps requiring large gauge wire with a larger mass per unit length. For example, for a 1000W heating component with a resistance of 10O we examine the current necessary to power the heater. Power and current are related by Ohm's power equation

$$P = I^2 R \tag{3.2}$$

where P is the power in watts, I is the current in amps, and R is the resistance in ohms.

Using the values from the example heater above and solving for current yields

$$\mathrm{I} = \sqrt{\frac{P}{R}} = \sqrt{\frac{1000\,\mathrm{W}}{10\,\Omega}} = 10\ \mathrm{A}$$

From Ohm's Power equation, it is evident the current necessary to power a device is a balance between the power load and the total device resistance. Figure 11-12[8] shows the significant increase in mass per unit length of cabling with respect to current.

Fault protection selection is more qualitative than quantitative. Fault protection devices must be capable of detecting, isolating, and correcting power faults such as short circuits. Primarily, fault devices isolate permanent power failures. Faults in the power system can draw excessive power causing critical damage to other systems or the spacecraft.

Power switches are used to turn on and off power to systems as needed. Typically, mechanical power switches are used because of their reliability and low power dissipation. Mechanical power switches have several configuration options: armature activation voltage and current, maximum armature voltage and current, number of armatures, number of contacts per armature, and whether the contacts are normally open or closed. Solid State relays on their most simplistic level behave like large transistor switches. Although solid state relays are very reliable, they still do not have the system independence and reliability of mechanical relays; Therefore, solid state relays should be used sparingly.

Power converters play an important part in the distribution subsystem. Power converters transform low voltage high current energy to high voltage low current energy or vice versa. All that is necessary to select a power converter is the bus energy conditions and the required load energy conditions (i.e. current and voltage for dc power applications).

The most fundamental component of the power distribution system is the power bus. The power bus distributes power collected from the power supply to the rest of the spacecraft. Typically, spacecrafts require less than 2000 Watts. Due to similarity in spacecraft power requirements, the space industry has developed a standard 28V bus with payloads built to match a power bus. For spacecraft requiring more than 2000W, custom bus solutions may be necessary. For all bus designs it is necessary to know the total power the bus will carry, the bus voltage, the number of payload connections, and the type of payload connections.

3.2 Thermal

Insulation, reflection, shielding, heat dissipation and internal thermal control are all necessary aspects of the thermal control system. Each must be optimized to keep the spacecraft and all subsystems within operational temperature. Thus the designers must calculate important characteristics for each component to maximize its effectiveness.

3.2.1 Insulation and Shielding

Insulation is one of the most important aspects of the thermal control system on a spacecraft. Insulators protect the inside of the spacecraft from the harsh elements of the space environment and thermally separate the different components within the spacecraft. The objective of every insulation system is to reduce the effective emissivity, $\epsilon*$ of the insulation material.

The most common type of insulation currently on spacecrafts in operation is Multi-Layer Insulation (MLI). Generally, MLI will cover the entirety of the exterior of the spacecraft with cutouts for external sensors and radiators. MLI can also be used on the inside of the spacecraft to protect propellant tanks and lines and any other temperature sensitive chambers. Most MLI consists of many layers of embossed 1/4 millimeter Mylar sheets. The sheets are coated on one side with vacuum-deposited aluminum; thus, the Mylar acts as a low-conductivity spacer between the aluminum layers. For MLI of this sort, theoretical effective emissivity, $\epsilon*$ is proportional to 1/(1+n) where n is the number of layers of insulation; there is a point of diminishing returns where the radiative losses through the insulation are overcome by the losses due to shorts in the layers of the MLI, usually at approximately 25 layers of insulation. In practice, for medium sized operations, MLI typically has an emisivity in the range of 0.015 to 0.030. For larger scale applications, emisivities approaching 0.005 are possible due to the lower density of seams, edges and holes for exterior instrumentation. Figure 3-1 shows the theoretical effective emissivity, $\epsilon*$ vs. number of layers as well as actual experimental values for various MLIs.

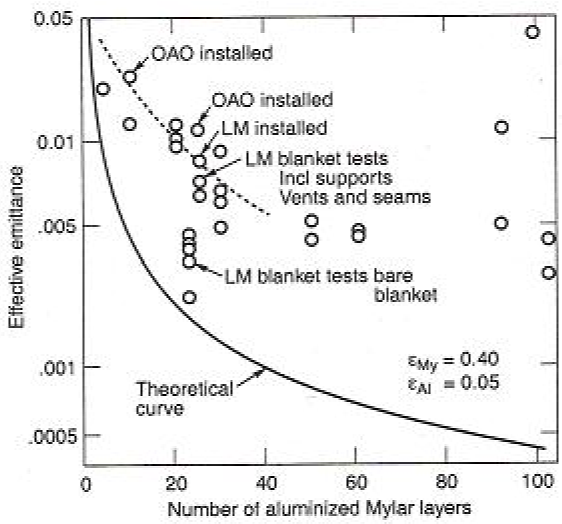

Figure 3.1: Effective emissivity, $\epsilon*$ vs. Number of aluminized Mylar layers for various types of MLI.[8]

Typically, the mass of MLI is approximately 0.73 kg/m^2; therefore the total mass of MLI for a spacecraft is this value multiplied by the surface area of the spacecraft[8] (assuming that the cut outs for exterior radiators and instrumentation are approximately equal to the extra MLI used on the interior of the spacecraft).

Though MLI is effective and easy to obtain, there are some applications where it is not practical. For example, if atmospheric interaction is necessary such as reentry or aero braking, MLI will be ineffective because MLI is not designed for convective heat transfer insulation. In this case, aerogel or foam insulation is more effective as these insulators rely on the presence of a gas as the non-conductive space. Also, if tanks or chambers inside the spacecraft are very small, MLI would be very inefficient ($\epsilon*$ as high as 0.3 due to the small scale). In this situation, thermal coatings are more efficient

Reflective coatings are another form of passive thermal control composed of hybrid paints, metallic alloys and laminates. The purpose of these coatings are to provide shielding and protection for the spacecraft by maximizing emissivity and reflectivity, thus minimizing absorption. Special reflectors consisting of silverized plastics and high-quality mirrors are used due to their ability to maximize the ratio of emissivity to absorptivity. Optical Solar Reflectors (OSR) and Silver-Coated Teflon are two examples of special types of surface radiation reflectors; their are shown in table 11-44.[8]

The following example calculates the surface equilibrium temperature of a spacecraft with a surface coating of white paint with reflectivity (ρ) equal to 0.85 and transmissivity is considered negligible. For this example we assume q_{solar} is equal to 1420 W/m^2, q_{albedo} is equal to 426 W/m^2, $q_{earthIR}$ is equal to 244 W/m^2, $\epsilon*_{MLI}$ is equal to 0.03, and the surface area of the spacecraft is 150 m^2 (all environmental factors are for a worst case scenario for a space craft orbiting earth). First, we use Equation 2-18 to determine the heat incident on the insulated spacecraft from the environment.

$$Q_{external} = A_{spacecraft}\ \epsilon*_{MLI}\ (q_{solar} + q_{albedo} + q_{earthIR})$$
$$Q_{external} = (150 \text{ m}^2)(0.03)(1420 \text{ W/m}^2 + 426 \text{ W/m}^2 + 244 \text{ W/m}^2) = 9405 \text{ W}$$

we then use Equation 2-17 to find the absorptivity, α of the reflective coating.

$$0.85 + \alpha + 0 = 1$$
$$\alpha = 0.15$$

From figure 11-47[8] we get a complimentary emissivity (ϵ) value of 0.85, because of the white selective surface used. The temperature can now be determined using Equation 2-16.

$$Q_{external} = \epsilon \text{ A } \sigma \text{ T}^4$$

$$T = \left(\frac{9405\,\text{W}}{(0.85)(150\,\text{m}^2)(5.670\,\text{x}\,10^{-8}\,\text{W/m}^2\text{K}^4)}\right)^{1/4} = 189.9 \text{ K}$$

The equilibrium temperature is low, due to the white surface's low absorption of solar energy, and high reflection of the heat incident upon its surface. Reflective coatings provide protection against extreme environmental heating and, due to their low mass, are very cost effective. The main drawback with reflective paints and laminates are their relatively short life span. Therefore, due to the amount of surface abuse prevalent throughout every orbit cycle, additional heat shielding is necessary.

Heat shielding is mainly a product of the spacecrafts material structure, density and thickness. These factors affect the amount of radiation and the range of temperatures a spacecraft can survive. Protective shielding can cause an increase in the mission cost because increasing the thickness of the spacecraft structure increases the mass requiring more propellant to achieve orbit and for station keeping once in space. These cost estimates are displayed in Figures 8-16 and 8-17.[8]

3.2.2 Heat Dissipation

Passive radiators outside the spacecraft are necessary to keep the entire spacecraft within survival limits. Radiators are typically body mounted plates or extendible panels exterior to the spacecraft coated with quartz mirrors, silvered Teflon, or white paint. These coatings effect high IR emissivity (> 0.8) and low solar absorptivity (< 0.2) to ensure that waste heat radiation is maximized while solar heat absorption is minimized. The size of the radiator used is determined by the sum of the heat generated within the spacecraft and the heat absorbed from the environment. We continue the example from the previous section with here with $Q_{Internal}$ equal to 800 W, $\epsilon_{RadiatorIR}$ equal to 0.8, $a_{Radiatorsolar}$ equal to 0.2, and radiator operational temperature, T equal to 0^o C = 273 K. To determine the size of the radiator we use Equation 2-23.

$$Q_{Internal} + Q_{External} = Q_{Radiator}$$

Subsituting Equation 2-24 for $Q_{Radiator}$

$$Q_{Internal}\ Q_{External} = (\epsilon_{RadiatorIR} \text{ - } a_{Radiatorsolar})\ \sigma\ A_{Radiator}\ T^4_{Radiator}$$

Solving for $A_{Radiator}$

$$A_{Radiator} = \frac{800\,\mathrm{W} + 9405\,\mathrm{W}}{(273\,\mathrm{K})(5.670\,\mathrm{x}\,10^{-8}\,\mathrm{W/m^2K^4})(0.8{-}0.2)} = 54.00\ \mathrm{m}^2$$

Radiators typically have a mass of approximately 3.3 kg/m^2; applying this value to the previously calculated area equation 2-25, the radiator mass is calculated

$$m_{Radiator} = (54.00\ \mathrm{m}^2)(3.3\ \mathrm{kg/m}^2) = 178.2\ \mathrm{kg}.$$

If the temperature of the radiators is increased to T = 50^o = 323 K, then the area and mass of the radiators is reduced accordingly.

$$A_{Radiator} = \frac{800\,\mathrm{W} + 9405\,\mathrm{W}}{(323\,\mathrm{K})(5.670\,\mathrm{x}\,10^{-8}\,\mathrm{W/m^2K^4})(0.8{-}0.2)} = 27.56.00\ \mathrm{m}^2$$
$$m_{Radiator} = (27.56\ \mathrm{m}^2)(3.3\ \mathrm{kg/m}^2) = 90.95\ \mathrm{kg}.$$

Note that the radiator temperature is a significant factor in radiator efficiency. In the previous example, raising the radiator temperature 50^o C reduced the radiator size by nearly 50%; because $_{Radiator}$ is proportional to T^4, the size of the radiators is reduced drastically as the temperature is increased. The temperature of the radiators is also the easiest property to change; if, instead we were to increase the IR emissivity, the cost of the radiator would increase drastically.

3.2.3 Internal Systems and Active Components

Active components are more expensive and less reliable than insulation, radiators, and surface coating; however, it is often necessary to use active components to control temperature to a limited degree inside a spacecraft. Examples of active component are heaters, louvers, and heat pipes.

Heaters are typically made of Kapton, a material used for its resiliency and space readiness. The power supplied by these heaters can be as much as 7.75 W/cm^2. There are many different types of heaters in current use. Patch heaters use electrical resistance between insulating materials to heat critical equipment on the spacecraft. Cartridge heaters heat components through a wound resistor enclosed in a metal cylinder. This cartridge put inside the component through a drilled hole. Heaters also need some type of thermal sensor to determine when heating is necessary. Thermal controls turn the heater on when the temperature falls to a minimum operational value and off when the temperature reaches a maximum operational value. A mechanical thermostat is most commonly used to determine the switch temperature. Thermostats come in a variety of different configurations such as bimetallic thermostats where two pieces of metal with different thermal expansion coefficients are stuck together so that they bend at a certain temperature threshold and activate the heater; and the electronic temperature sensor which is a digital sensor with a computer to tell the heater when to turn on.

To determine a heater's required size, knowledge of the components' maximum and minimum operating power is needed. For this example, we size a heater for an electronics boxes with a maximum power of 500 Watts and a minimum power of 400 Watts. When not operating, there is no power. Calculating the amount of power required during the non-operating stage yields the size of the heater. Consider a temperature range of -10 to 50^o C. We assume $q_{External}$ equal to 182 W/m^2 and the box area, A_{box}, equal to 2 m^2

At the minimum power, we must determine if there is a necessity for heater power. This is done using equation 2-16 where Q is the sum of $Q_{external}$ and the power into the box (use the minimum power). Solving for T determines the operating temperature of the electronics box.

$$(182 \text{ W/m}^2)(2 \text{ m}^2) + 400 \text{ W} = (5.670 \text{ x } 10^{-8} \text{ W/m}^2\text{K}^4)(2 \text{ m}^2) \text{ T}^4$$

$$\text{T} = 297 \text{ K}$$

The operating temperature of 297 K = 24^o C is above the minimum operational temperature, -10^o C, thus. no heating is required while the electronics are in operation; however, it is necessary to determine how much heat is required while the electronic box is not in use. using the same equation, but replacing power input with heat required; temperature is set to the minimum value of T equal to -10^o C = 263 K and the equation is solved for heat required.

$$Q_{required} = (5.670 \text{ x } 10^{-8} \text{ W/m}^2\text{K}^4)(2)(263)^4 - (182 \text{ W/m}^2)(2 \text{ m}^2) = 131 \text{ W}$$

Using the value of 7.75 W/cm^2 stated above for heater output, the total area of the heater must be at 16.9 cm^2.

Another method of actively maintaining temperature through the spacecraft is the use of louvers. Louvers maintain heat on internal surfaces by controlling the amount of heat transferred to other internal surfaces or to the radiators. The most common type of louvers currently in use is the 'venetian-blind,' which opens or closes to allow more or less heat through.[8] The blades on the louver are opened and closed using actuators, which determine the radiator temperature and adjust the louver accordingly. Louvers typically allow six times the heat transfer while open as they do closed; They are typically used to control the flow of radiant heat from the spacecraft to the environment.

Heat pipes modulate the amount of heat transfer, in instances controlling the active radiator area and limiting the need for heaters. These heat pipes are approximately 0.15kg/m in mass and can supply 10 Watts using variable conductance; however, an additional 1 to 3 kg is added to the mass of this heat pump for each heat reservoir.[8] Heat pipes work through a fluid cycle where large quantities of heat are transferred without using power. Heat enters on one end of the pipe. The outside of the pipe, or wick, is filled with liquid and when it heats, the liquid evaporates into the pipe. From the heat exchange, a pressure build-up occurs which pushes the gas to the other end of the pipe, where the heat is then released. This heat pipe allows for the separation of heat source from the sink where heat is needed. Reducing temperature gradients across the ship to avoid thermal deformation at high contrast points is beneficial side effect of heat pipes.

3.3 Environment

During the early days of America's space program, man ambitiously envisioned missions of space exploration and discovery. But, without important developments by scientists and engineers, man could never have developed systems capable of meeting the challenges of the harsh and dynamic space environment. Typically, these systems were developed using two main approaches: material testing in space and testing on Earth in a simulated space environment.

3.3.1 Material Testing in Space Environment

Material testing in space is critical in determining space-worthiness. Examples of such missions include Spacecraft Charging at High Altitude (SCATHA), Long Duration Exposure Facility (LDEF), and the Materials on the International Space Station Experiment (MISSE). The primary mission of SCATHA was to obtain information about the processes and effects of spacecraft charging, a phenomenon known to have contributed to several on-orbit satellite failures. The mission's specific objectives were to (1) obtain environmental and engineering data to allow the creation of design criteria, materials, techniques, tests and analytical methods to control charging of spacecraft surfaces and (2) collect scientific data about plasma wave interactions, substorms, and the energetic ring. The spacecraft was also known as P78-2.

The SCATHA spacecraft was spin stabilized at about 1 rpm with 5 deg pointing accuracy. A hydrazine propulsion system with 8 thrusters and 2 tanks was included. Body mounted solar cells generated 290 watts of power and the craft used three 8Ahr Nickel Cadmium (NiCd) batteries. The SCATHA craft was capable of downlink at 8.2 kbps at S-Band from redundant 10 W transmitters. Aluminum, titanium, magnesium, glass fiber composed its structure. Seven payload booms were deployed for the experiment. To log the experiment, two tape recorders with about 350 Mb storage were included.

The payload on SCATHA included several experimental modules. The SC1 module held engineering experiments plus VLF and HF receivers which measured surface potentials of various spacecraft materials, measured RF waves between 0-300 kHz, 2-30 Mhz. The SC2 module housed the Spacecraft Sheath Fields plus Energetic Ions experiment which measured low energy electrons and ions, energetic protons, and electrons. The SC3 module held the High Energy Particle Spectrometer which measured high energy electrons and protons. The SC4 module contained the Satellite Electron and Positive Ion Beam System which employed ion and electron beam guns to control spacecraft surface potential. The Rapid Scan Particle Detector was located in the SC5 module. It measured electrons and ions. The SC6 module housed the Thermal Plasma Analyzer, which failed soon after initial turn on, but was designed to measure thermal electrons and ions. The Light Ion Mass Spectrometer, which also failed soon after initial turn on, was designed to measure light ion density, temperature and composition was held in module SC7. The SC8 module held the Energetic Ion

Compositions Experiment which measured low energy electrons and the ion composition of energetic plasma. Module SC9 housed the UCSD Charged Particle Experiment which measured electrons and ions. Module SC10 held the Electric Field Detector. It measured DC and ELF electric fields and satellite potential. The SC11 module contained the Magnetic Field Monitor - measured DC and ELF magnetic fields. Additional experiments, including the ML12 Spacecraft Contamination Plus Thermal Control Materials Monitoring measured contamination rates and property changes of several thermal control material samples. The TPM Transient Pulse Monitor supported the other experiments by providing supporting data about the electromagnetic pulse environment. The twelve experiments had a total mass of 87 kg and consumed 110 W of power.[10]

The LDEF facility exposed materials to the hazardous space environments for extended periods of time. Extremely thin, patchy films of silicon-based contamination were distributed all around the circumference of LDEF. On leading edge surfaces only, it was found that silicones had been oxidized into silicates by the atomic oxygen. However, analysis of LDEF tray clamp bolt heads from a variety of exterior LDEF locations, including the leading edge, revealed a molecular film that contained silicone. Since this silicone film had not reacted with atomic oxygen to form silicates, it would have had to have been deposited after the surfaces were shielded from atomic oxygen, such as would have occurred when LDEF was re-berthed in the Shuttle. In addition, the silicone found on the leading edge locations had to originate from non-LDEF surfaces as all LDEF surfaces were thoroughly outgassed by the time LDEF was retrieved. Any small amounts outgassed toward the end of the mission would have been quickly oxidized by the high flux of atomic oxygen at the end of the mission. Post-flight analysis revealed several forms of contaminants. These contaminants include particulates, molecular films, and silicon based molecular contamination.

Particulate contaminants were identified from preflight exposure sources, from Space Shuttle sources, from on-orbit material degradation, and from post-flight exposures. Except for introduction of sources of stray light scattering on optical materials, small particulates had no effect on materials performance. Where large-scale failure of blanket materials occurred, with subsequent distribution of small particulates, effects on nearby surfaces were often significant. The part of the black radiator panel on tray F09 which was covered by the failed aluminum backing from an adjacent thermal blanket underwent large color changes. Lexan exposed by failure of the adhesive tape on Experiment M0001 and subsequent flexing of the blanket material was severely discolored. It is hard to quantify the effects of these occurrences because the time of failure is not known, and the particle distribution probably changed with time.

The consideration of molecular contaminant films should be separated into what happens to carbon-based, or organic, material and what happens to silicon-containing material. Organics exposed to atomic oxygen are removed rather rapidly. Exposure to atomic oxygen will cause silicone surfaces to oxidize to silicates. The carbon-based functional groups of the silicones are easily oxidized and removed by abstraction processes, leaving the Si-O portion of the polymer chain. Subsequent oxygen atoms add to the Si-O chains, producing a glassy, non-volatile surface. Silicone remaining trapped beneath the surface will darken under UV exposure.

Collectively, there were many sources of silicon contamination on LDEF. Some areas were contaminated prior to flight, silicone-based materials outgassed during flight, and the Space Shuttle was a potential source during both deployment and retrieval. Post-flight contamination must also be considered. Each previous source is in addition to the processes which occurred during the 69 months in orbit. In addition to silicone-based coatings and adhesives used on several experiments, a number of other materials used have silicon present as a basic constituent, or left from the manufacturing process. The S13 type and A276 paints contain silicon, the stainless steel bolts have 1 to 2% silicon by weight in the alloy, and the copper grounding straps have a silicone release film. These facts mask the attempt to evaluate silicon contamination on these particular surfaces. The silverized Teflon (Ag/FEP) and chromic acid anodized surfaces, which together covered more than 78% of the spacecraft exposed surfaces, are materials which contain no silicon. These materials are excellent witness plates when attempting to determine the quantity of silicon deposited.

The gasket seals used on the ground handling tray covers were preflight sources of silicones and compromised certain areas of the anodized tray surfaces. This material was not space qualified and outgassed extensively, with a total mass loss (TML) of over 3%.

Electron Spectroscopy for Chemical Analysis (ESCA) measurements of surface silicon on a variety of materials, which did not originally contain silicon, show silicon present in amounts ranging from 0.1% (lower limit of detection) to over 30%. The amounts present are extremely exposure and location dependent. Interpretation of results are complicated by local conditions, i.e., multiple silicone sources within the same tray, vents, outgassing by adjacent specimens, and changes in exposure conditions over time, such as failed materials moving and covering or partially shielding surfaces. Tray clamp surface studies at Virginia Polytechnic Institute and State University generally showed larger percentages of silicon on leading edge clamps relative to trailing edge clamps. Clamps from space and Earth end locations also tended to be relatively high in silicon surface content. Examination of multiple locations on an individual space end clamp (H6-11) and on an Earth end clamp (G6-5), which were each selected for close proximity to a vent, showed very high surface silicon content.[3]

The purpose of the Materials International Space Station Experiment (MISSE) is to characterize the performance of new prospective spacecraft materials when subjected to the synergistic effects of the space environment. MISSE 1 and MISSE 2 were transported to the International Space Station (ISS) and attached to the exterior of the ISS during the STS 105 mission on August 10, 2001. MISSE 1 and MISSE 2 each have over 400 candidate spacecraft materials which will be exposed to the space environments for approximately two years while on the ISS. The MISSE was retrieved on STS 114 in 2003, and the candidate materials returned to the experiment investigators for analysis.

MISSE 1 and 2 utilize an innovative technical concept that has been identified for the ERT Program at Johnson Space Center (JSC) to expand the utilization of the ISS. It is a cooperative experiment involving Principle Investigators from Boeing Phantom Works, the Materials Laboratory at the Air Force Research Laboratory, and NASA's Langley Research Center and Marshall Space Flight Center and Glenn Research Center.

MISSE 1 utilizes Passive Experiment Containers (PECs) developed by Langley Research Center (LaRC) and first used for ISS Phase I Risk Mitigation Experiments on Mir. MISSE 1 will characterize the performance of candidate new space materials over an approximately two year exposure period on-orbit. Three subsequent missions, MISSE 3, 4, and 5 will replace MISSE 1 and 2 when they are retrieved.[11]

3.3.2 Space Environment Simulation Facilities

Space environment simulation is commonly used to test the space worthiness of materials. NASA has six facilities in the Electro-Physics Branch of the Glenn Research Center including an Atomic Oxygen Beam Facility; a Vacuum Ultraviolet Thermal Cycling Facility; a Wire-Pyrolization, Arc Tracking, and Current-Rating Vacuum Facility; and a Solar Flare X-ray Simulation Exposure Facility.

Low Earth orbital energetic atomic oxygen can threaten a spacecraft's durability by eroding external surfaces. Where openings to the space environment exist, internal component surfaces may be seriously degraded by atomic oxygen. NASA's Atomic Oxygen Beam Facility evaluates vacuum ultraviolet radiation on materials and synergistic vacuum ultraviolet radiation exposure effects of low earth orbit atomic oxygen to determine if spacecraft components meet survivability requirements. Vacuum ultraviolet radiation can be blocked allowing only atomic oxygen exposure or it can irradiate various material samples in the system. The facility can provide accelerated rates of exposure to a directed or a scattered beam of atomic oxygen. The low-energy beam of less than 30ev is generated in a vacuum by operating electron-cyclotron resonance plasma source on pure oxygen. The beam is predominately neutral atomic oxygen with ion content of less than one percent. Uniform atomic oxygen flux arrival, covering an approximate area of 30 by 30 cm, is controllable within a range of 1015 to 1017 atoms/cm^2/s. For larger components, sweeping exposure is possible and materials under mechanical loading is tested. During an exposure test of no more than four, 1-inch diameter samples, the Atomic Oxygen Beam Facility offers in-situ optical characterization by making total hemispherical spectral reflectance measurements. Automatically controlled tests are available for a one month continuous, unattended operation for elevated temperature exposure.[1]

In 1991, NASA Langley researchers developed a new class of polymers based on polyarylene ether benzimidazoles (PAEBI) containing phenyl phosphine oxide groups. The use of the benzimidazole polymer provides high glass transition temperature Tg and high modulus coupled with high optical transmission. PAEBI combines the performance of plastics used for space thermal control applications, with inherent resistance to energetic particles, especially atomic oxygen (AO). When attacked by AO, PAEBI initially reacts similarly to other plastics; however, unique to the PAEBI chemistry is the incorporation of a phosphorus-oxygen linkage in the polymer backbone that forms an in situ protective coating on the surface. This oxide layer renders the plastic 15 times more resistant to the erosion caused by atomic oxygen compared to other thermal control plastics. PAEBI has been developed by Triton Systems Inc. under the trade name TORTM and TOR-LMTM. TORTM films have a high Tg, high tensile strength and modulus, excellent electrical insulating characteristics and a color that is similar to Kapton. TOR-LMTM is a copolymer that has been developed to provide films with a lighter color (lower solar absorption a), lower modulus, tensile strength and Tg while still maintaining the same molar ratio of the phosphorus-oxygen linkage that provides comparable environmental stability as TORTM. Ground simulation tests have been performed on the TORTM and TOR-LMTM films at NASA Marshall Space Flight Center (MSFC), Langley Research Center (LaRC) and Glenn Research Center (GRC), which consistently show significantly lower reaction efficiency (i.e. less material erosion) when compared to other organic polymers used in thermal control products. The ground simulation results are now supported by recent flight data from the POSA-I flight experiment. In prolonged space exposure onboard the MIR space station, Triton's space durable films exhibited as much as 15 times less erosion than other polymer films.[12]

Earth orbiting spacecrafts experience extreme temperature changes when entering and exiting eclipse periods on orbit. This thermal cycling in addition to vacuum ultraviolet radiation can cause serious harm to spacecrafts in low Earth orbits. NASA's Vacuum Ultraviolet Thermal Cycling Facility is used to test materials by exposing them to vacuum ultraviolet radiation in combination with thermal cycling. This simulation is realistic to an orbiting spacecraft environment and provides a vacuum pressure of 10-6 Torr. Two deuterium lamps provide the vacuum ultraviolet radiation in wavelengths of 115-200 nm. A mercury-xenon arc lamp can be used to add ultraviolet radiation in wavelengths of 220-400 nm. For heating, two quartz-halogen lamps provide infrared radiation while an LN2 cooled chamber provides radiant cooling. The facility can be used for thermal cycling alone where at appropriate exposure increments, a manipulator arm automatically cycles test samples between the cold chamber and the heating region. The cycling temperature extremes are adjustable and are typically plus or minus 80 degrees Celsius. The system is automatically controlled for unattended operation. A sample's exposure area can be approximately 8 by 8 cm and the sample can have a thickness of at most 6.4 mm. Some samples tested include solar cells, thermal control coatings, and polymers.[1]

Insulating material may be thermally charred or pyrolized when a momentary short-circuit arcs between a defective polyamide insulated wire and another conductor. The short-circuit arc is sustained by the conductive charred polyimide. Arc tracking occurs when the sustained arc propagates along the wire through continuous pyrolization of the polyimide insulation. The polyimide insulation of other wires within a multiple wire bundle may become thermally charred and start to arc track or flash over if an arcing wire is part of that wire bundle. Consequently failure of an entire wire bundle or harness may occur because of arc tracking. Because polyimide insulated wires are popularly used in aerospace vehicles, NASA has initiated a program to develop space wire derating test procedures to identify candidate wire insulation types that are susceptible to arc tracking. Undoubtedly for space applications, the Arc-Tracking, Wire Insulation-Pyrolization, and Current-Rating Vacuum Facility is used to perform arc tracking and pyrolization tests on samples of wires with candidate insulation materials in various thermal, space and atmospheric simulated environments. The electrical and electronic durability of components used in space applications can be evaluated in a vacuum environment. A cryogenic pump is used in the vacuum system to provide 10-6 Torr operating pressure in a bell jar. For high power research testing with complete automated computer control, a 600 amp electric arc welder is used with three 300 volt, 6-a dc power supplies, and a 240 to 400 kHz ac power supply. For real-time data acquisition, process control, and computation, a 50 MHz EISA computer is used for high-speed data acquisition and report generation. To provide the test area cooling, thermocouples, and high current, sufficient feed through is available while the system optionally provides controlled heating of test articles.[1]

NASA's Solar Flare X-ray Simulation Exposure Facility at the Electro-Physics Branch in the Glenn Research Center provides continuum x-rays of less than 103 keV. Using an absolute vacuum ultraviolet silicon photodiode, in-situ x-ray flux measurements are made. To block energetic electrons from the source, 2m thin film Aluminum barriers are used. The exposure area measures 8 cm by 12 cm and temperature monitoring is available. X-ray targets vary and include Molybdenum and Aluminum. For the Hubble Space Telescope's third servicing mission in the year 2000, exposure and durability tests were done on candidate replacement thermal control material. In addition, X-ray durability tests are conducted on the "Flare X-ray" protective coatings on Teflon fluorinated ethylene propylene substrates.[1]

Chapter 4

Summary and Conclusions

4.1 Power

Energy storage devices, power sources, and power distribution systems are critical to spacecraft and mission design. Multiple energy storage systems exist, primarily batteries, but selection is completed using charge and discharge rates, mission lifetime, mass, and volume. Power sources provide energy to charge the batteries while simultaneously operating the spacecraft. Power source selection depends on payload power requirements, spacecraft size and mass limitations, environmental and safety considerations, and mission lifetime. Finally, after determining a power source, energy storage systems, and payload power requirements, the designer can establish cabling, fault protection, battery arrangement, and power bus needs.

To ensure the best design of the power subsystem the design process must be iterated. Before working on power subsystem design, the designer needs a good approximation of the payload and thermal power requirements. When selecting power subsystem components a trade off exists between performance, cost, durability, and mass. The designers must tailor their component selection to each spacecraft and mission design. For example, designers of a relatively short and inexpensive mission would not select a nuclear reactor with lithium ion batteries. Instead, designers will choose to use fuel cells to reduce mass by eliminating the power supply.

In selecting power components, the bulk of a designer's time will be spent weighing and measuring different power subsystem components with one another. Ultimately, the designer will be forced to look back at the mission objectives and measures of effectiveness, MOEs, to aide him in the selection of power components. Should the designer select a nuclear power source for implementation? The answer is unclear until the designer considers the value system, the MOEs and criterion set by leadership, for a particular mission. Does the design meet the performance, cost, durability, mass and size expectations? Again, the designer must use the mission value system. Ultimately there is no perfect solution; the best solution can only be obtained iteratively with a thorough understanding of the initial value system, constraints, and rewards and consequences of each power subsystem component.

Depending on the mission, the power subsystem will have a direct influence on the thermal control system. Heat generated by the power system impacts the amount of thermal control necessary to maintain survivable temperatures.

4.2 Thermal

Although the thermal control subsystem is rarely involved in completing the primary mission objective, i.e., the payload's objective, it serves an important function on the spacecraft. The thermal control system must keep the entire spacecraft within survivable temperatures, as well as keeping each individual component within its specific operating temperature range. To keep all the spacecraft subsystems within tolerance temperatures, the thermal control subsystem employs a wide range of methods such as radiators, insulation, heat shielding, reflective coatings, and active components.

Radiators radiate heat away from the outside of the ship. Waste heat builds up in the ship as subsystems work and must be removed in this manner. Radiators are the most massive component of the thermal control subsystem so it is here that the total mass of the system must be minimized. The most common type of insulation onboard spacecrafts is multi-layer insulation, MLI. These layers are mounted covering the exterior of the spacecraft insulating the spacecraft from the space environment. One of the best options for external heat insulation are reflective coatings. These special surface coatings are comprised of various mixes of silver-ized alloys, white paints and plastics in order to maximize their innate thermodynamic properties. Active components like heaters and cryogenic coolers are mission specific. They are not always necessary and due to their power draw should not be used unless passive components are ineffective.

More efficient radiators require less surface area and thus, are less massive. Choosing the best radiator for a spacecraft, where to place radiators, whether the radiator should be mounted on the spacecraft or in arrays on booms are all issues that are mission specific. The designers of each specific mission will have to decide what is necessary and choose accordingly. A short mission with low powered systems requiring small radiators and minimal outputs would be able to use small, body mounted radiators. However, a long mission with high powered systems creating large amounts of waste heat may require extensive radiation to maintain operational temperatures. In this case, high temperature radiators are useful to reduce mass while mounting them on booms away from the ship will avoid reheating the spacecraft with the radiators. Multi-layer insulation can also reduce radiator mass because more efficient MLIs allow less environmental heat absorption, therefore, less heat must be radiated. Multi-layer insulation is not the only type of insulation available. Others are more suitable in specific situations such as atmospheric interactions where MLI is non-functional. In an atmospheric environment foam and gel insulation are more effective. Insulation configuration and type are mission specific and must be chosen carefully by the designers. When foam or gel insulations can not be used, reflective coatings are desired. By engineering the best possible reflective coatings, external surface temperatures can be held to a relative minimum. Reflective coatings add a minuscule amount of dry mass to the vehicle. The disadvantage of reflective surfaces is their relatively short life spans incurred by the periodic and drastic temperature ranges experienced by orbiting bodies. Therefore, additional heat shielding is necessary for further external heat insulation. Heat shielding, a function of material density and thickness, greatly enhances the amount of radiation and range of temperatures a spacecraft can survive. However, greater protection requires more weight and cost. Therefore, finding the orbital and dimensional parameters is useful to best tailor the needs of the spacecraft to its mission. Components such as IR detectors requiring low operating temperatures necessitate the use of cryogenic coolers. If the mission will spend a significant amount of time in shadow, then active heaters must be used to keep spacecraft and subsystem temperatures within operational limits.

For a given mission designing a spacecraft's thermal control subsystem is dependant on the environment in which the craft will be flying. Designing an efficient thermal control system reflects the temperature ranges and hazards anticipated.

4.3 Environment

The space environment must be carefully considered when designing a space mission, as it influences a number of spacecraft subsystems. Designers must consider the natural space environment as well as man made hazards.

Currently, the most common and practical method used to limit the detrimental effects of atomic oxygen, surface charging, and single event phenomena is judicious material selection and surface coatings. Surface coatings have been developed to significantly slow atomic oxygen erosion, however, atomic oxygen is highly reactive and its effects can not be eliminated. Conductive coatings are used to keep surface potentials below breakdown levels, although other techniques such as special filtering, cabling, or grounding are available to prevent electrostatic discharge. Radiation, or charged particle, effects are limited only by shield thickness. Since charged particle interaction is on the nuclear level, surface coatings have no effect. Minimizing the probability of single event phenomena with thicker material is the only way to prevent particles from damaging electronic components or sensors. The external torques and dissipative effects created by atmospheric drag are simply unavoidable.

The mission designer must also note solar cycle effects. Over the eleven year cycle, a maximum and minimum in solar activity exist. Atmospheric density and solar radiation vary with the solar cycle. Thus, depending on mission life, thermal and ADCS subsystem requirements and the robustness of onboard electronics will also vary. The importance of shield thickness needs to be weighted against spacecraft mass requirements, often determined by launch vehicle selection. The nature of the mission is the key factor in designing for environmental effects. For example, designers of an interplanetary probe will not be greatly concerned with atomic oxygen erosion, atmospheric drag, and surface charging since the probe will not be operating in the Earth's atmosphere. Designers will need to plan for radiation effects, however.

Besides planning on radiation effects, designers may or may not need to plan for orbital or man made hazards. Orbital hazards are mainly space debris consisting of meteoroids and man made debris such as spent rocket stages. Designers can plan to avoid these obstacles by either knowing debris position or having spacecraft maneuverability. Shield thickness used to protect against radiation also protects a spacecraft from relatively small space debris. A designer may need to plan for man made hostile threats, especially if the designer is working for the military. Spacecraft used for military purposes may have decoys, self defense systems, maneuverability, or additional shielding thickness. Natural environments are not considered in the definition of hardness and survivability because technology developed over the last thirty years has made it possible for satellites to withstand natural environments beyond their mission lifetimes. Hardening is the single most effective survivability option for a spacecraft exposed to a hostile environment.

Bibliography

[1] NASA Glenn Research CenterElectro-Physics Branch. Space environment simulation. Available internet <http://www.grc.nasa.gov/WWW/epbranch/space/>.

[2] Yunus A. Cengel and Michael A. Boles. *Thermodynamics: An Engineering Approach*. McGraw-Hill Inc., NY, fourth edition, 2002.

[3] NASA Langley Research Center. Long duration exposure facility contamination. Available internet <http://setas-www.larc.nasa.gov/LDEF/CONTAMINATION.html>.

[4] Uranium Information Centre. Nuclear reactors for space. Technical Report 82, Uranium Information Centre Ltd, 2004. Available internet <http://www.uic.com.au/nip82.htm>.

[5] El-Genk and Hoover. *Space Nuclear Power Systems, Volume 1*. Orbit Book Company Inc., ?, 1984.

[6] M. D. Griffin and J.R. French. *Space Vehicle Design*. American Institute of Aeronautics and Astronautics Inc., DC, 1991.

[7] California Institute of Technology Jet Propulsion Laboratory. *Solar Cell Array Design Handbook, Volume 1*. NASA, ?, 1976.

[8] Wiley J. Larson and James R. Wertze. *Space Mission Analysis and Design*. Microcosm Inc., CA, third edition, 1999.

[9] D. editor Linden. *Handbook of Batteries and Fuel Cells*. McGraw-Hill Inc, NY, 1984.

[10] Mission and Spacecraft Library. Scatha: Spacecraft charging at high altitude. Available internet `<http://msl.jpl.nasa.gov/QuickLooks/scathaQL.html>`.

[11] NASA. Misse: Material international space station experrment. Available internet `<http://misse1.larc.nasa.gov/>`.

[12] SAGE Publications. Atomic oxygen resistant films for multi-layer insulation schuler et al. high performance polymers. *High Performance Polymers, Vol. 12*, (No. 1):113–123, August 2000.