CHAPTER 1

INTRODUCTION

1. The beginnings

The first artificial satellite, the 184-pound Sputnik I (Figure 1.1), was launched on October 4, 1957, and carried a silver-zinc primary battery as its only power source. The battery provided one watt to power the two transmitters which ceased broadcasting three weeks later. The satellite reentered the atmosphere in January, 1958, but not before marking the dawn of the space age (Walls, 1995). The primary battery (i.e., not a rechargeable one) effectively defined the useful life of the spacecraft since the spacecraft itself did not re-enter the Earth's atmosphere until some weeks after the batteries were spent. This initial satellite was followed soon thereafter by the launch of Vanguard I, the first satellite to carry solar cells coupled to secondary (i.e., rechargeable) batteries. The batteries were included to provide electrical power during periods of eclipse. Since then, the sophistication of artificial satellites and the attendant demands for electrical power to make them functional have increased by many orders of magnitude. What was once a scientific curiosity has become an indispensable tool of modern communications, meteorology, observation, navigation, geodesy, national defense, and entertainment, as well as scientific discovery.

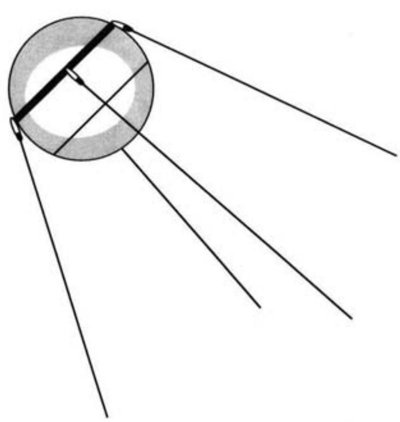


Figure 1.1 Sputnik I, the Earth's First Artificial Satellite

Since those early days, the frequency of satellite launches has made the event commonplace. Figure 1.2 shows this growth in the number of spacecraft launched worldwide over the past 40 years (Curtis, 1994; Thompson, 1994). This growth has occurred not only in the number of satellites launched, but in their size also. While the first Sputnik was only a few kilograms, the size of present-day satellites can be judged by the capabilities of several current launch vehicles shown in Table 1.1.

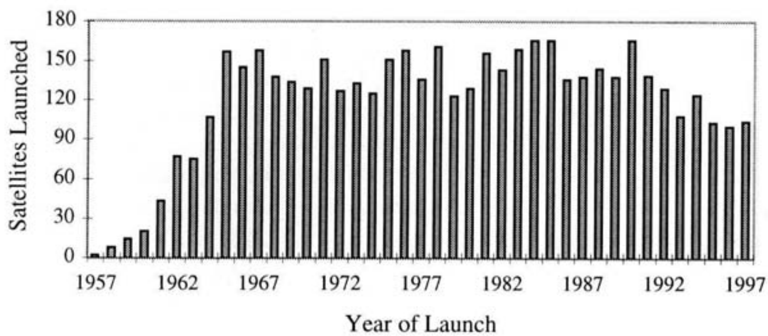


Figure 1.2 The Satellite Launch Rate, Worldwide, Since 1957

Table 1.1 Representative Capabilities of Current Launch Vehicles

Launch Vehicle	Payload to LEO (kg)	Payload to GEO(kg)	Payload to GTO (kg)
Delta II- 7925	5,000	1,800	
Titan IV	17,700	4,450	
Ariane 5			6,800
Proton K	20,100	2,100	4,615
Shuttle	24,400	5,900	

Although enormous payloads can be placed in orbit with relative ease using these and other modern launch systems, the cost of launch still remains very high (typically several thousands of dollars (U.S.) per kilogram into low Earth orbit). This cost places a premium on minimum mass and high system reliability, especially for the bus systems (e.g., guidance and control, telemetry, power, etc.) that are often assumed by the mission planners.

1.1 The increasing demand for spacecraft electrical power

The increases in satellite sophistication, and with it increases in payload size, have been accompanied by an ever-growing requirement for electrical power aboard the spacecraft. Figure 1.3 shows the growth in electrical power needed for specific spacecraft over the past 40 years. In some sense, the communications satellite demands for electrical power have diverged along two tracks based on orbits: geosynchronous communications satellites which often require ten to twenty kilowatts of power versus the lower-orbit, smaller communications spacecraft which typically require only tens to hundreds of watts. For many other applications, the trend has generally been for more power, although not exclusively. The NASA program to develop less-expensive, lighter satellites has also increased demands for less, rather than more power. While the demand for spacecraft electrical power extends across a broad range of values from several hundred watts to many tens of kilowatts, in a real sense, the more challenging task may be at the lower end of the power range.

Regardless of the power levels, in all space applications there is the need to improve the system specific power. This has placed great demands on the engineering skills of spacecraft power designers, and the response has been to develop new technologies and to refine existing technologies especially to enable the missions which require higher power levels. This has been done in the face of a rather limited menu of options available for generating electrical power in space.

Prior to 1957, most design experience related to the engineering of electrical power systems was rooted in terrestrial systems or power systems designed for use on aircraft. Most of the terrestrial power design guidelines were not relevant to operations in space

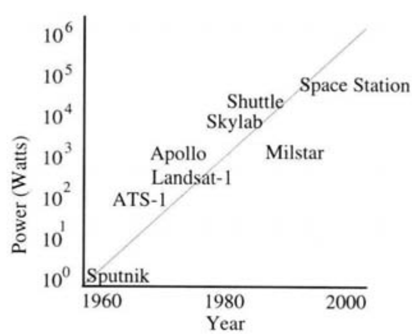


Figure 1.3 The Growth in Requirements for Spacecraft Electrical Power

because their design assumes certain elements that simply are not available in space. These have been identified, with some humor, as Earth, air, fire, and water. The Earth provides a convenient grounding mechanism (grounding to the space plasma is problematic), air offers efficient convective cooling (radiative cooling from the spacecraft is the only realistic option), fire is an inexpensive heat source for conversion to electricity (not a space option), and water is used universally in thermal management (the control of waste heat, especially in conjunction with power generation, is a serious design constraint). On a less whimsical basis, Walls (1995) points to several differences between space-based and Earth-based power systems that seriously limit technology interchange. These differences are contrasted in Table 1.2.

It is not surprising, therefore, that early-on aircraft engineering practices were adapted for spacecraft. But even that was insufficient to accommodate the more stringent constraints imposed on operations in the space environment. While mass, reliability, and cost are considerations in aircraft power systems, those systems have the advantages of very large prime power sources, the aircraft engines that develop many times more power than any electrical power that might be drawn as an incidental load. Further, flight times are measured in hours rather than years.

No constraint is more demanding on space operations, however, than reliability.

Table 1.2 Comparison of Terrestrial and Space-Based Power Systems

Attribute	Terrestrial System	Space-Based System
Scale	Tens to hundreds of megawatts	One to ten kilowatts, typically
Sources	Many options: hydro, coal, nuclear, large rotating machines, etc. or chemical	Few options with premium on mass: solar, nuclear,
Transmission	High voltage operations, AC is standard, switching and voltage conversion are simplified	High voltage not compatible with space plasmas, DC is standard, DC-DC inverters are needed
Costs	Addressed through scale	Includes cost of delivery to orbit, premium paid for high reliability and mass and volume reduction
Energy management	Providers adjust to customers' need	Energy budget is fixed by the power system and all power management is load management
Operations	Large, interconnected grids to provide redundancy	Autonomous operation, no interconnectedness for redundancy

While reliability is taken seriously in the design of terrestrial and airborne power systems, the appearance of a problem in either of these applications can be addressed with relative ease. In the case of space-based systems, maintainability is a prohibitively expensive option, if it is available at all. It takes only ten minutes to get into space, but at $50 million a minute, everything must be totally reliable (Kinesix, 1998). Reliability must be engineered from the beginning since failure in a non-redundant electrical power system can mean the end of the mission.

1.2 The architecture of a spacecraft

The complexity and cost of building and launching satellites have also increased during this same period. In this evolution, spacecraft operators have worked with innovation and great engineering skill to improve the lifetime, efficiency, reliability, and compactness of each of the subsystems aboard the spacecraft. What began as a simple design centered on a power source and a transmitter has become a complicated interrelationship among a number of subsystems, each requiring electrical power. The general architecture of a spacecraft is shown in Figure 1.4. The satellite can be viewed as being comprised of two major parts: the mission payloads and the support subsystems. The payloads, specific to each satellite, are the reasons for the mission and to a large extent will define the overall satellite design. As Stark *et al.* (1995) point out, however, the

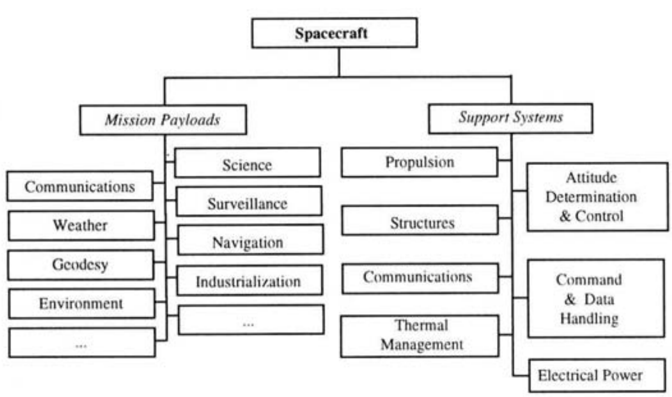


Figure 1.4 Spacecraft Systems

final design goes beyond the nature of the payloads and is often a reflection of the design philosophy of the contractor or the nation responsible for the satellite.

Support systems, sometimes referred to as bus systems, are generic and tend to be the same, functionally, from satellite-to-satellite. In an effort to reduce costs, much effort has been devoted in recent years to the modular design of these bus systems. Virtually every system onboard a satellite, payload or bus, will require electrical power, usually at differing peak and average power levels, voltages, and duty cycles. To illustrate the complexity of this load, consider the attitude control system (ACS) which is designed to maintain the satellite pointing in the proper direction. It was the failure of the ACS on Galaxy IV in May, 1998, which caused the loss of that communications satellite carrying 90% of the electronic-pager traffic in the United States. Figure 1.5, adapted from Barter (1992), shows the many parts of a modular attitude-control system (ACS), and it is clear that each component will require electrical power. Several of the ACS subsystems such as the accelerometers, sensors, and computers for data manipulation require low voltages and currents, while the drives and electromagnetic actuators for solar arrays require high peak powers. This situation is repeated in each of the mission and bus systems shown in Figure 1.4. In the overall satellite design, a power budget for each of these systems is an important part of the process of sizing the power system. A more complete discussion of the architecture of spacecraft and the functions and power requirements of the bus systems can be found in several recent books in the

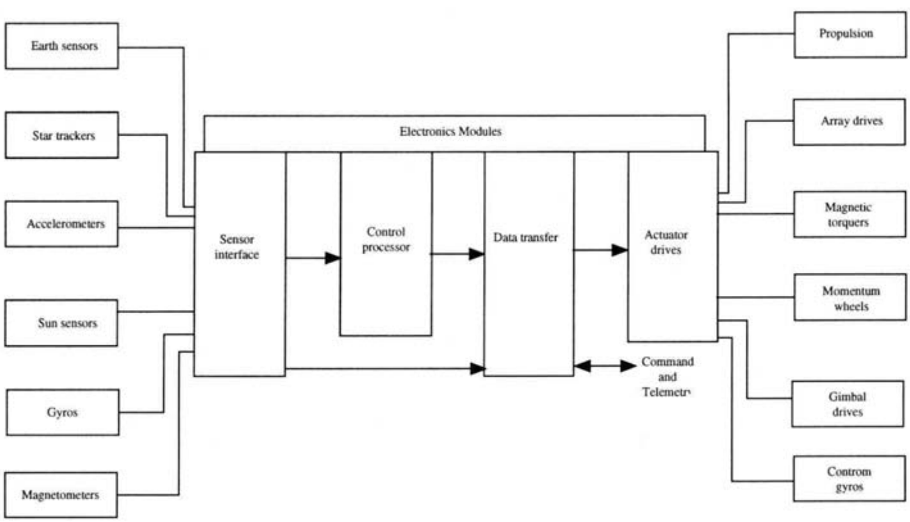


Figure 1.5 A Modular Attitude-Control System

field, among them Fortescue and Stark (1995), Griffin and French (1991), Larson and Wertz (1992), Pisacane and Moore (1994), and DeWitt, Duston, and Hyder (1993).

2. The electrical power system

2.1 An overview of electrical power systems

The enabling system aboard any satellite is the electrical power system (EPS). In its simplest form, a satellite electrical power system consists of four major components as shown in Figure 1.6. The prime power source will provide energy for conversion into electricity. As an intermediate step, in certain cases (e.g., thermophotovoltaic), all or part of that prime energy may be stored before conversion takes place. Conversion into electricity then occurs through a variety of methods, depending on the nature of the prime source and the spacecraft electrical loads. The electricity that is generated will need to be managed, regulated, monitored, and conditioned to match the electrical needs of the spacecraft systems.

The conversion of one form of energy into another involves technologies that are both old and new. Table 1.3 lists several forms of energy that can be considered as potential input energy for conversion to another form, electricity in the case of spacecraft power. While the technologies listed do not form an exhaustive list of options, their great number reflects the richness of possibilities. For example, nuclear sources, primarily viewed as sources of heat, are not included but are certainly important sources of energy for space operations. Entries on the diagonal of Table 1.3 may be viewed as storage options rather than conversion mechanisms.

The choices available as prime power sources in space are limited to three: nuclear, chemical, or solar. As shown in Figure 1.7, the duration of the mission is a key factor in the selection of the prime power source. For short-duration missions, or to supply the

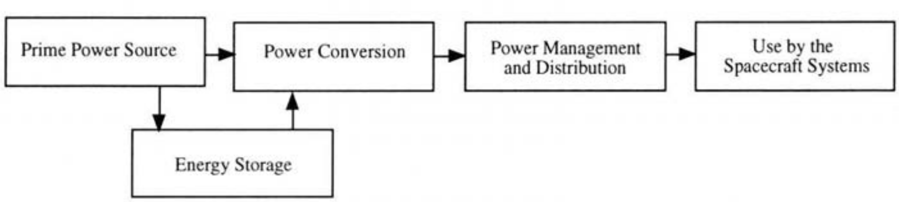


Figure 1.6 Elements of the Electrical Power System

Table 1.3 Energy Conversion and Storage Options

Output Energy / Input Energy	**Electricity**	**Heat**	**Chemical**	**Photons**	**Kinetic**
Electricity	Batteries Fuel cells Superconducting magnets Inductors	Ohmic heaters Heat pumps	Electrolysis Ionization and recombination	LEDs Discharges Light bulbs Lasers & transistors Microwaves	Flywheels JxB thrusters Motors
Heat	Thermoelectrics Thermionics Generators Fuel cells	Phase-change materials Chemical reactions High C_p materials	Thermochemical electrolysis	Radiators	All thermodynamic cycles
Chemical	Fuel cells Capacitors Batteries	Combustors	Propellants Explosives	Chemical lasers	Rocket exhaust Gas turbines
Photons	Photovoltaic cells	Thermal concentrators Thermal absorbers	Photolysis electrolysis	Resonant cavities Metastable atoms	Radiometers
Kinetic	MHD Generators Homopolar devices Compulsators	Friction	Impact ionization	Triboluminescence	Flywheels

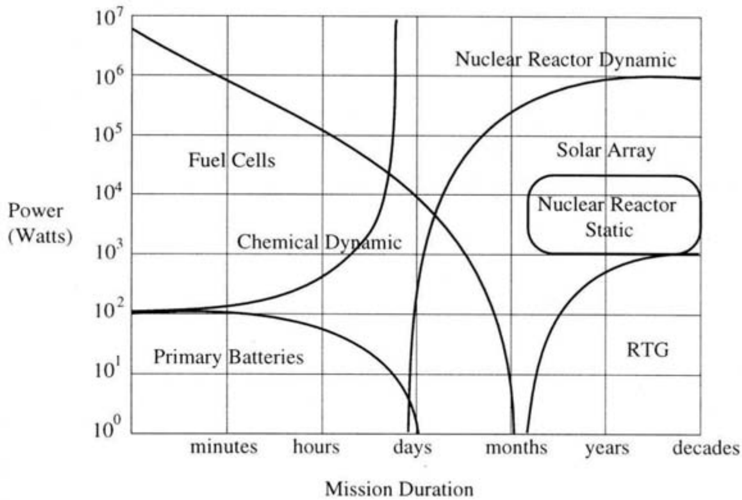


Figure 1.7 Options For Various Mission Power Needs and Durations

power for activities that will be completed relatively quickly within the framework of a longer mission, chemical systems such as primary batteries, fuel cells, or chemical dynamic conversion, may be the appropriate choice, depending on the total power required. Often, primary batteries are used in meeting the high power and high energy demands of the launch vehicle itself as well as in the activation of pyrotechnic devices related to explosive stage separation. For longer duration missions, the choices are restricted to solar arrays in conjunction with secondary batteries or regenerative fuel cells, or to nuclear systems, either reactors or radioisotope thermoelectric generators. Other operational issues may certainly influence the choice of prime power sources. For example, the survivability of solar arrays in certain orbits could exclude their choice in spite of their ability to provide the necessary power within limitations of mass, cost, etc. The restricted maneuverability of large solar arrays, an unacceptable level of the infrared signature of nuclear systems, or compatibility with mission-related sensors can also eliminate certain prime power options which otherwise would have been logical choices.

In some applications, the demand can be for a very large peak power, but for a short enough duration that the total energy can be surprisingly small. In other cases, such as housekeeping power requirements aboard an operational satellite, the average power requirements can again be modest but the extended time over which the power is needed can create the need for large amounts of total energy.

As an example, a military mission that requires 100 MW for ten minutes will demand about the same total energy as ten kW missions that must remain on orbit for 10 years. In this example, while solar cells can be sized to provide 100 MW, since the duration is so short, an expendable fuel option makes more sense from a total mass argument. A space-based radar which requires one MW of power in a one ms burst needs only 10^3 J per pulse, an application that could be met with the use of capacitors and a few kW baseload power system. Figure 1.7, therefore, presents options based on total energy that is required for the mission as well as the rate at which that energy can be delivered.

2.2 Electrical power system designs

The electrical power system (EPS) is designed and configured to perform several key functions: it must be a continuous and reliable source of peak and average electrical power for the life of the mission; it must control, distribute, regulate, and condition the power provided to the various loads; it must be capable of providing data regarding the health and status of its operation; and it must protect itself and its loads from electrical faults anywhere within the spacecraft (McDermott, 1992). Many factors contribute to the final design and the choice of technologies that must be integrated. This process (shown schematically in Figure 1.8) starts with the mission and its requirements. The mission payloads will define the peak and average power needed, together with the lifetime of the satellite, the orbit, and the overall configuration of the spacecraft. Each of these constraints will carry implications for the design of the EPS, such as the end-of-life power needs, the degree of redundancy needed for an acceptable level of reliability, the environmental factors against which the system must be protected, and options for the thermal management (TM) system.

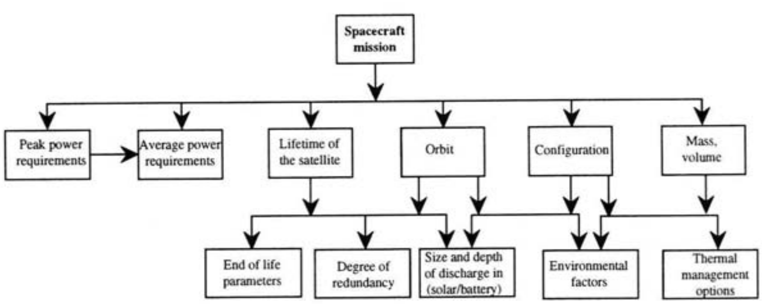


Figure 1.8 The EPS Selection Process

The selection process will focus not only on the prime power source but each of the other subsystems in the EPS (Figure 1.9) as well. In many cases, the prime power source serves only as a source of heat to be converted into electrical power. All three sources– nuclear, solar, and chemical– are capable of producing heat for conversion into electricity through either static or dynamic processes. As the name suggests, the static processes– thermoelectric, thermionic, AMTEC, and others– do the conversion without benefit of moving parts. This is often demanded by the pointing-and-tracking requirements of the payload. Dynamic conversions involve the thermodynamic-cycle processes such as Rankine, Stirling, and Brayton. The most common EPS in use is the photovoltaic array, involving solar energy and a static conversion process, the photovoltaic cell. In these cases, the energy must be stored, usually through a chemical process (mostly batteries, but sometimes regenerative fuel cells), so that the spacecraft can be powered during the eclipse periods or when load demands exceed solar array output. Regardless of the prime power source, energy storage is an option using thermal, chemical, or mechanical means. Mechanical storage mechanisms (e.g., high-rpm flywheels) are not in use but offer very large storage potential, again addressing the possible need for large peak powers simultaneous with modest average power. Following the conversion process, the unregulated electrical power is delivered to the Power Management and Distribution (PMAD) subsystem. The PMAD links the generation process to the storage elements and the spacecraft loads. Although PMAD is indicated as a subsystem interfacing with the spacecraft loads, in reality it is distributed throughout the EPS, and functional elements can be found virtually everywhere in the electrical system.

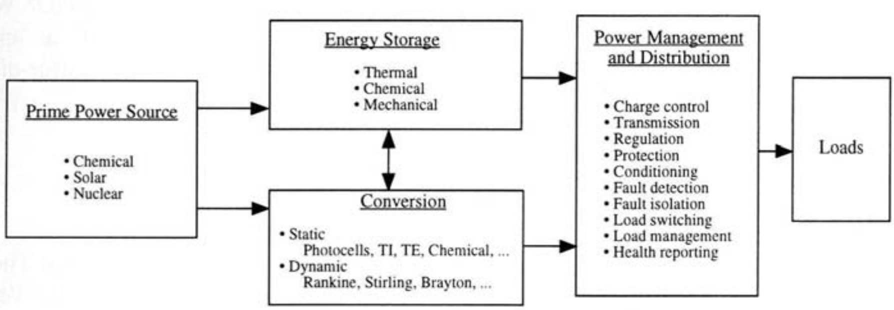


Figure 1.9 Functional Breakdown of the EPS

2.3 Examples of missions and their electrical power systems

To provide examples of the impact of mission requirements on EPS design, we conclude this introduction with a brief discussion of several satellites recently launched or in the late planning stages.

Spartan

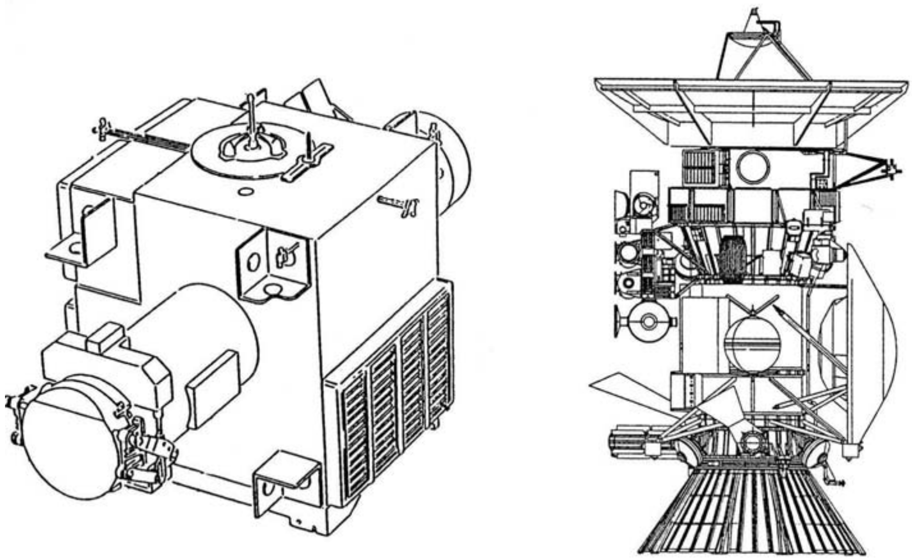
The Spartan (Figure 1.10a), a free-flying platform for scientific experiments, is released and recovered from the Shuttle. It uses a common service module containing the ACS, electronics, batteries, the TM system, data handling electronics, and a cold plate. In a throwback to Sputnik I, since the unit has an operational lifetime of only 40 to 50 hours, silver-zinc primary batteries supply the electrical power. The batteries have a capacity of 30 kWh and deliver power at 28 VDC. Since it will remain in a relatively benign low-Earth orbit, there are no special constraints imposed by the orbital environment.

Cassini

At the other extreme, the mission of spacecraft Cassini (Figure 1.10b) is to explore the Saturnian system. It is designed to carry 12 instruments on the 2,100-kg orbiter (remaining in orbit around Saturn for four years) and six on the 350-kg probe (which will explore the moon Titan in situ). The satellite was launched in October, 1997, and is due to arrive at Saturn in June, 2004. Several requirements distinguish Cassini from other interplanetary missions: the distances to the Earth and the Sun, the extended length of the mission, the number and complexity of the scientific experiments, and the four gravity-assists enroute to Saturn. It has a design life of 13 years, much of which will be at such great distances from the Sun that solar arrays are impractical. These factors, even with relatively low power requirements (about 750 W at beginning–of–life and 628 W at end-of-mission), make a nuclear prime power system mandatory. In the case of Cassini, three radioisotope thermoelectric generators (RTGs) are employed. Lithium sulfur-dioxide (LiSO2) primary batteries provide power for the Huygens probe.

Magellan

Magellan (Figure 1.11) was designed to study the geological structure of Venus. The primary payloads included a synthetic aperture radar that imaged 98 percent of the Venetian surface with a resolution of 100 m and an S-band radio-tracking package to measure the planet's gravitational field. The satellite was lost in October, 1994. Because of



(a) (b)

Figure 1.10 Schematic Representations of the (a) Spartan and (b) Cassini Spacecraft

the relatively short mission duration and the proximity to the Sun, power for the four-year mission was provided by a 12.5 m^2 solar array used in conjunction with two 30-Ah nickel-cadmium batteries. The system provided 1029 W of power at the end-of-life.

The International Space Station

The basic element of the ISS power system is the photovoltaic power module PVPM (Baraona, 1990). It consists of five major components: (1) two solar array assemblies and associated sequential shunt units; (2) the beta gimbals; (3) the integrated equipment assembly (IEA); (4) the thermal control system and radiator; and (5) the truss structural elements that enable the PVPM to be attached to the main ISS structure. The IEA holds several different types of boxes called orbital replacement units, or ORUs. Each ORU is dedicated to a specific subsystem of the full power system. There is an ORU for the batteries, battery charge/discharge units, direct current switching units, dc-dc converters, power distribution and control units, junction boxes for fluid and electrical services, and thermal control system pump units.

There are to be 4 PVPMs with a total of eight solar array wings, each of which will produce 32 kilowatts at the beginning of full ISS operation. The net power delivered to

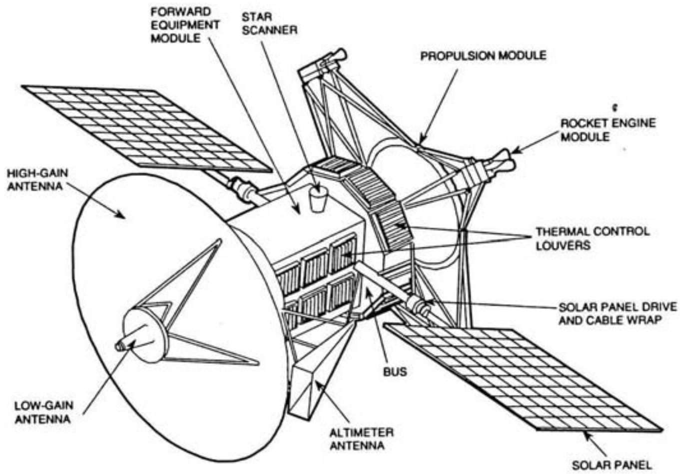


Figure 1.11 The Magellan Spacecraft

the ISS will be about 75 kilowatts of the 250 kilowatts available at the arrays. The remaining power is used to charge the batteries and to account for various system inefficiencies. The array operating voltage is 160V dc, and the distribution system voltage is 120V dc. The solar array power is regulated to the primary distribution voltage by the sequential shunt units in the PVPM and transmitted through the beta gimbals by roll rings to dc switching units. The power management and distribution (PMAD) system is designed so that any combination of two power system failures will not cause a loss of all electrical power to the ISS. This is accomplished by the use of redundant switching and controlling units and multiple independent cables to each critical user, such as the pressurized modules and the experimental pallets on the station trusses. Power management and distribution control will be through the use of semiautonomous local controllers linked to a central controller. The control system is designed to monitor and detect faults, isolate malfunctioning circuits, and reconfigure and recover system performance. The same semiautonomous controllers will schedule power use to a certain extent to help prevent overloads and to assure full battery charge at the start of each eclipse period. The batteries are 81 amp-hr nickel-hydrogen cells arranged to provide 120V dc output over a 35% depth of discharge. Power system thermal control is accomplished via a pumped loop cooling system which regulates the temperature of the batteries and other critical electrical control system hardware. A radiator assembly completes the system. Battery design life is five years, and array design life is 15 years.

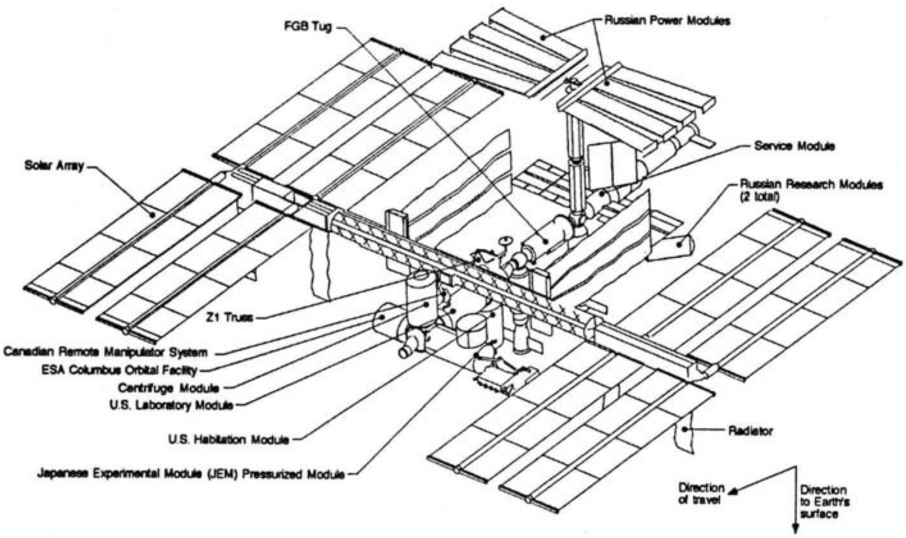


Figure 1.12 The International Space Station

Galileo

Galileo (Figure 1.13) was designed to study Jupiter's atmosphere, moons, and surrounding magnetosphere. The spacecraft also deployed a probe into Jupiter's atmosphere in December, 1995. In spite of a failure in the spacecraft's high-gain antenna, most of the scientific objectives were accomplished. Its eight-year design life and it distance from the Sun favored the choice of ^{238}Pu RTGs as the power source. Two were used, and each provided 570 W at the beginning-of-life decreasing to 485 W at the end-of-life. Had solar arrays been used for the power source, over 150 m^2 of solar panels would have been needed. Eighteen-Ahr primary lithium-sulfur batteries powered the probe, which was released into the planet's atmosphere. Galileo was deployed in October, 1989, and entered orbit around Jupiter in July, 1995. The long shelf life of the primary batteries was a factor in their selection.

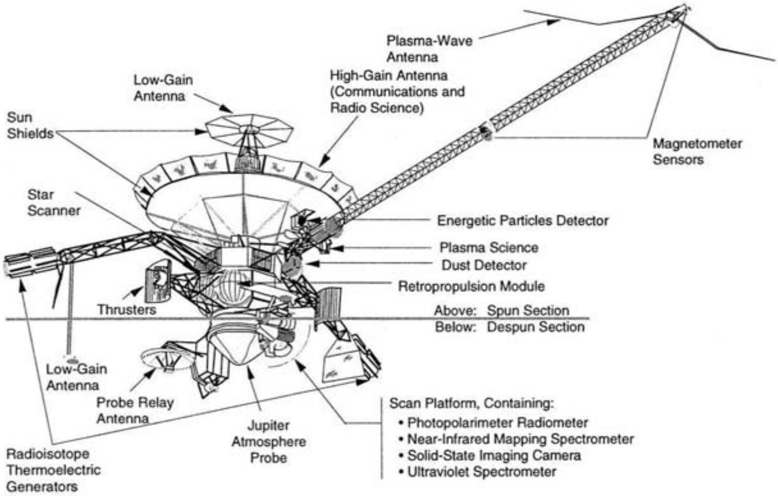


Figure 1.13 The Galileo Spacecraft

TOPEX/Poseidon

The TOPEX/Poseidon mission, launched in August, 1992, is a remote-sensing scientific program undertaken jointly by the Centre National d'Estudes Spatiales (CNES) and the National Aeronautics and Space Administration (NASA). The TOPEX (Ocean Topography Experiment) satellite monitors the Earth's oceans from an altitude of about 1300 km to better understand the ocean climate, weather, and surface features, as well as to enhance coastal storm warnings and safety at sea. TOPEX is one of a series of satellites to use the multimission modular spacecraft bus (MMS) to support the essential subsystems such as attitude control, power, command and data handling, and propulsion. The payloads for TOPEX include radar altimeters, a Doppler-tracking receiver, a microwave radiometer, and a laser retro-reflector array. During its first three years in orbit, the spacecraft has measured sea heights to within 4 cm.

Figure 1.14 shows the fully deployed TOPEX/Poseidon satellite. The spacecraft electrical system is powered by a solar array, providing about 3400 W at the beginning-of-life and designed for maximum energy transfer using a non-dissipative, unregulated main power bus. The modest power requirement and the relatively benign LEO orbit naturally lead to a photovoltaic power option. The MMS and its Modular Power Subsystem (MPS) was also used on the Solar Max Mission, Landsat 4 and 5, the Upper Atmosphere Research Satellite (UARS), and the Gamma-Ray Observatory, among others. (See Chapter 8 for an extended discussion of the TOPEX power system.)

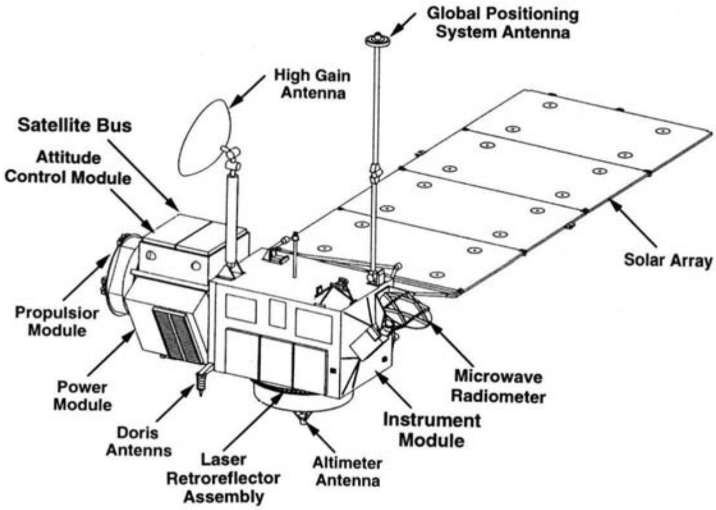


Figure 1.14 The Fully-Deployed TOPEX Satellite

Envisat

The Envisat system, scheduled to deploy in late 1999, is in some ways similar to the TOPEX/Poseidon program. The purpose is to observe the Earth's oceans and ice fields and their interactions with the atmosphere and specific aspects of atmospheric chemistry. The spacecraft will operate in a sun-synchronous orbit (see Chapter 2) at an altitude of about 800 km. The complex mission payloads include radars, laser reflectors, spectrometers, radiometers, altimeters, interferometers, and occultation optics. The low-Earth orbit, four-year lifetime, and large electrical power budget led to a design using a 6.5 kW (end-of-life) solar array with eight 40Ahr NiCd batteries. The power will be distributed on a voltage bus at 23 to 37 volts. Much of the EPS design is driven by the demands of the synthetic aperture radar which will draw a peak power of 1.2 kW and an average power of 750 W.

2.4 Spacecraft electrical power technologies

The improvements that have taken place in the technologies which enable spacecraft power systems have been achieved in spite of two serious constraints on power system development. The first is the issue of ownership of power technology development. Often the payload mission office is reluctant to endorse a spacecraft design that demands more power than is readily available from the existing technology. This is understandable since to do otherwise would impose a development burden on the mission that might be considered tangential to the primary purpose of the flight. This recurring theme limits the resources to promote dramatic technological breakthroughs in power system design. As a result, the improvements in spacecraft electrical power systems that have come about over the past three decades are more properly characterized as evolutionary rather than revolutionary. The second constraint, as we have seen, is related to the limited options available for generating electrical power in space. In the case of chemical or nuclear systems, the source of the power must be transported into orbit at great expense in cost and mass. In the case of solar-battery systems, the total power available is limited by both nature (the solar constant at one A.U. is 1.4 kW/m^2) and technology (still relatively low conversion efficiencies). There have been improvements, however, and they will continue. Table 1.4 offers a comparison of the state–of–the–art in several key technology areas from the mid–1980s to the end of this decade. Progress will continue and with it expanded options for powering satellites and their payloads.

2.5 An overview of the book

This book attempts to present, in a systematic way, a discussion of the evolving technologies that make today's spacecraft power systems the reliable energy systems they have proven to be. The focus is intentionally on the technologies rather than systems because of the large number of candidate systems that can be designed by mixing-and-matching the subsystems as shown in Figure 1.9. The solar array-battery system is the most common and will be emphasized throughout the book. Unfortunately, a full discussion of all possible system configurations is not feasible. To demonstrate this point, consider the system shown in Figure 1.15, a solar-thermal dynamic Brayton-cycle conversion system. Here, solar energy is concentrated and collected into a storage element and is then used to heat a gas in a closed-Brayton-cycle turbine connected to an alternator. This represents one of many combinations that can be considered as candidate systems for large power level electrical systems in space. Hopefully, with the discussions of the underlying technologies presented in this book, the reader will be better prepared to address spacecraft electrical power issues at the systems level.

The book begins with a discussion of the near-Earth environment (Chapter 2) and the limitations that operations in that arena will impose on the EPS. Solar energy is the most common method of generating electricity on satellites, and Chapter 3 presents the theory and practical implications of that form of prime power. Coupled closely with solar power is the use of batteries, and Chapter 4 discusses, from basic chemistry to space-qualified components, the array of chemical storage technologies including both batteries and fuel cells. Although nuclear power, other than RTGs, has been scarcely used in space, it already has a rich heritage, and the evolution of that technology is

Table 1.4 The Evolution of Selected Power Technologies

System or Component	Parameter	Circa 1985	Estimated 2000
Solar-Battery Systems	Power Output	5 kW	100 kW
	Specific Power	10 W/kg	50 W/kg
	Solar Array-Battery Costs	\$3000/W	\$1000/W
Solar Cells and Arrays	Cell Power Output	5 kW	100 kW
	Cell Efficiency (in space)	14%	25%
	Array Specific Power	35 W/kg	150W/kg
	Array Design Life (LEO/GEO)	5yr/7yr	10yr/15yr
	Array Specific Cost	\$1500/W	\$500/W
Batteries			
Primary			
AgZn	Energy Density	150W-hr/kg	
	Design Life	2 yr	
$LiSOCl_2$	Energy Density	200W-hr/kg	700 W-hr/kg
	Design Life	3 yr	5 yr
Secondary			
NiCd (LEO)	Energy Density	10W-hr/kg	
NiCd(GEO)	Energy Density	15 W-hr/kg	
NiCd (LEO/GEO)	Design Life	5yr/10yr	
NiH_2(LEO)	Energy Density	25 W-hr/kg	
NiH_2 (GEO)	Energy Density	30 W-hr/kg	
NiH_2 (LEO/GEO)	Design Life	2yr/3yr	
Primary Fuel Cells			
	Power Load	7 kW	50kW
	Specific Power	100 W/kg	150W/kg
	Specific Cost	\$40/W	\$25/W
	Design Life	~2000 hrs	4000 hrs
Nuclear Power			
Reactors	Power Level	10kW	10kW
	Specific Power	10W/kg	10W/kg
	Efficiency	10%	10%
RTG	Power Level	2 kW	2 kW
	Specific Power	6 W/kg	10W/kg
	Efficiency	8%	12%
Typical Overall System Parameters			
	Power	12 kW	25kW
	Voltage	28 V	50V
	Frequency	DC	DC/AC
	Cost –on–Orbit	~\$1000/kW-hr	
	Radiator Specific Mass	20kg/kW	

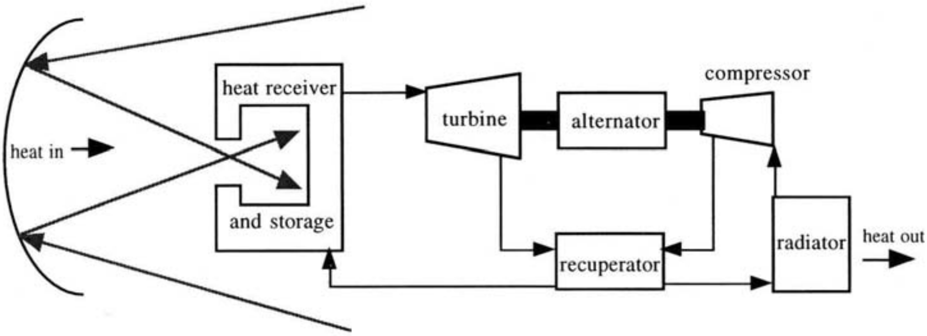


Figure 1.15 A Solar-Thermal Dynamic System

presented in Chapter 5. Chapters 6 and 7 discuss a number of static and dynamic conversion processes that have been used or are being proposed for use in space. While these conversion technologies are presented within the context of a nuclear-powered heat source, any heat source, including solar energy, can be applied. The PMAD subsystem is the subject of Chapter 8, which also presents descriptions of components and techniques for achieving the various PMAD functions and examples of several PMAD configurations. The generation and use of electrical energy on satellites present a difficult design problem in thermal management (TM) and the interactions between power production and thermal management is discussed in the final chapter. Also presented in Chapter 9 are the theory and practical engineering aspects of maintaining the proper thermal environment onboard satellites.

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