the free molecular flow in the ram direction is made of 2-cm thick aluminum plate. Assuming conservatively that the heat flux is steady at the worst-case value, and ignoring any convective effects, which are not a factor in free-molecular flow, how much time is required to raise the temperature by 5 K at a depth of 1 cm into the plate?

9.3 What fraction of solar energy lies in the visible range, which we will define as being 0.75–0.35 μm?

9.4 What is the average spacecraft temperature for the situation in Example 9.2, if the spacecraft is in a noon-midnight orbit? Make reasonable assumptions as required.

9.5 A solar panel on a GEO satellite tracks the sun; the back of the panel faces dark space. The cells have a 90% packing factor and an energy conversion efficiency of 12%. The effective front-surface solar absorptivity is $\alpha = 0.90$, and the infrared emissivity is $\varepsilon = 0.94$. The anodized aluminum back surface panel has IR emissivity $\varepsilon = 0.80$. What is the steady-state operating temperature of the array?

9.6 For the spacecraft in Example 9.2, assume a total mass of 100 kg and an average heat capacity equal to that of aluminum, $C = 961$ J/kg·K. A piece of onboard equipment fails, causing the internal power generation to drop to 35 W. Treating the spacecraft as isothermal, as in the example, what is the new steady-state temperature, and approximately how long does it take to reach it?

9.7 A radiator on a LEO spacecraft will be oriented toward dark space while in use, and must dissipate 200 W on average. The radiator uses a pumped fluid loop containing water-glycol and operates at a nominal temperature of 310 K. The blackbody efficiency is $\eta = 0.85$ and the emissivity is $\varepsilon = 0.94$. What is the required radiator area?

9.8 For the radiator of problem 9.7, the 50/50 water-glycol mixture freezes at about 230 K. To allow an appropriate safety margin, the radiator must be maintained at or above 250 K. What is the minimum power that must be dissipated by the radiator to maintain safe operation? If for any reason this level of power usage in the spacecraft cannot be maintained, what operational strategy might be used to avoid freezing the radiator?

9.9 Using the parameters of Example 9.2, with the noon-midnight orbit of problem 9.4, and the spacecraft mass and heat capacity of problem 9.6, what temperature is reached by the spacecraft immediately prior to exiting its eclipse period? What temperature is reached after the equipment failure of problem 9.6?

10 Power Systems

10.1 Introduction

Constraints on available spacecraft power have imposed major limitations on space vehicle design since the beginning of the space age. The earliest orbiting vehicles flown by both the United States and Russia depended on batteries. The limited energy storage capabilities of the batteries then available prevented operations of more than a few days. This was not satisfactory for missions of the duration required for detailed scientific observations or military reconnaissance, and solar power arrays quickly appeared on the scene. Although not highly efficient in turning sunlight into electricity, solar arrays (or solar panels) were in many ways admirably suited to powering spacecraft. Because no consumables were used in generating electrical power, the life expectancy of the power system was limited only by degradation of the components of which it was composed. Spacecraft operating lifetimes of several years became feasible with the development of these photoelectric arrays, with batteries used to handle peak load requirements and to provide energy storage for those periods when the spacecraft was in eclipse.

Solar panels and batteries in combination have powered the majority of unmanned spacecraft so far launched. Exceptions include a few short-lived battery-powered systems, some outer-planet missions using radioisotope thermoelectric generators (RTGs), and some spacecraft (mostly Russian radar imaging satellites) powered by nuclear reactors. Early manned spacecraft, including Mercury, some Gemini spacecraft, and the Russian Vostok/Voshkod vehicles (which were essentially the same design) used batteries. The later Gemini spacecraft and the Apollo command and service module (CSM) and Lunar Module (LM) used hydrogen/oxygen fuel cells, as does the space shuttle, while the Russian Soyuz employs solar cells and batteries in a fashion similar to a typical unmanned spacecraft. The space stations so far built, including Salyut, Skylab, Mir, and the International Space Station, have all used solar arrays for prime power generation, with batteries for loadleveling and eclipse periods.

Solar power systems are unsatisfactory for missions beyond the asteroid belt, where the sun's energy becomes unacceptably diffuse. As interest developed in outer-planet missions, a new power source was required. At the same time, certain military spacecraft missions required a sturdy compact power source. Both requirements were met by the development of RTGs. These devices convert the heat energy produced by radioisotope decay into electricity via the thermoelectric effect. Power output is independent of the sun, and lifetime is limited only by component degradation and the half-life of the radioisotope. RTGs are also useful for operations on planetary surfaces where extended dark periods may be encountered. Thus, outer solar system spacecraft such as Pioneer, Voyager, Galileo, and Cassini, as well as the Viking Mars Landers and the Apollo Lunar Surface Experiment Packages have all been RTG powered, as have some Earth orbiting spacecraft.

Nuclear reactor systems offer very high power in a compact package for quite a long duration and tend to be highly independent of the external environment. After an extensive development program in the 1960s, all U.S. space reactor work, for both power and propulsion, was terminated as a result of space program funding reductions in the early 1970s. Only recently has there been a revival of interest in power plants of this type. Although Russia continues to fly relatively short-lived reactor power systems on an operational basis, the United States has flown only a single reactor test mission, the SNAP-10A in 1972. The joint DoD/NASA/DoE SP-100 project of the mid-to-late 1980s was intended to remedy this matter; however, the program was delayed and eventually canceled because of its high cost and limited mission applicability. As this is written, interest in nuclear-powered systems has again arisen, because they are the only practical means of generating relatively high power for long periods in the absence of adequate sunlight.

As can be seen, the power system is a major driver in any spacecraft design and is in turn strongly driven by a variety of mission, system, and subsystem considerations. It interfaces directly with almost every other subsystem and, as a result, requires considerable attention from the systems engineer. Power system technology continues to evolve, especially in the application of automation to routine functions (e.g., battery reconditioning, to be discussed), and the development of more efficient power conditioning and control circuitry.

10.2 Power System Functions

The obvious functions of a spacecraft power system are to generate and store electric power for use by the other spacecraft subsystems. Other subsystems may have various specific requirements for voltage, frequency, stability, noise limits, or other characteristics, and the power system may be called upon to supply them. A significant system-level tradeoff underlies the decision as to whether to require the power system to meet these various individual requirements, or to supply all subsystems with the same basic power and let each subsystem meet its specific power conditioning requirements. For example, a requirement for a very high voltage in a particular scientific instrument might be supplied by the spacecraft power system, or by a dedicated high-voltage supply within the instrument that operates off the basic power bus. Similar tradeoffs exist for special cleanliness requirements (e.g., absence of ripple on a dc line, ac harmonic suppression, etc.).

Regardless of the conclusion of these tradeoffs, the power system must control, condition, and process the raw power received from the primary source to comply with the needs of the spacecraft system. The system must supply stable, uninterrupted power for the design life of the system. Failures in many other subsystems can be tolerated, with solutions often found through operational compromises. However, if the power system does not work essentially as planned, the mission is lost.

To maintain the long-term reliability of the system, the power system must provide protection to other subsystems against reasonably likely failures either external to or within the power system itself. For example, no short circuit in another subsystem should be allowed to drag the main bus voltage down to the point of inducing failure elsewhere in the spacecraft. Similarly, failure protection should be implemented in the power system itself to allow for continued functioning of the system (perhaps in a degraded mode) following some degree of malfunction.

In the course of normal operation, the power system must accept commands from onboard and external sources and provide telemetry data to allow monitoring of its operation and general health.

Finally, it may be necessary to meet highly specialized power requirements for particular functions such as firing ordnance.

10.3 Power System Evolution

The evolution of spacecraft power systems has been characterized by growth from subsystems delivering a few watts to those delivering tens of kilowatts or more. The International Space Station required about 75 kWe initially with growth currently planned to 220 kWe or more. Line losses and other efficiency factors, including the desire to minimize the mass of spacecraft wire harnesses, have resulted in a trend toward higher voltages as power demands have increased. Figure 10.1 illustrates this trend and projects broadly what may be anticipated in the near future.

The design lifetime of space systems tends to increase along with required power levels as spacecraft become more complex and expensive. As the power level and lifetime change, the choice of a primary power source may change as well. Figure 10.2 illustrates the general operating regimes of various types of power sources. There is a substantial overlap between the regimes, and various other considerations may dictate use of some power source at a location in the power vs endurance space that may not otherwise appear to be optimum. Figure 10.2 provides a basis for preliminary concept design in regard to power source choices.

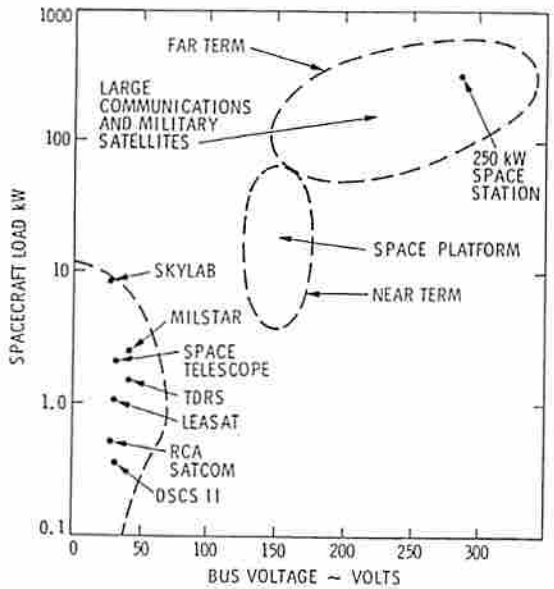

Fig. 10.1 Trends in spacecraft power.

10.4 Power System Design Drivers

A variety of considerations may affect the design of the power system. Table 10.1 presents a number of these considerations. Not all will be applicable to each system design, and, conversely, some designs may involve considerations not listed here. However, most cases of common interested will be treated. The designer should view Table 10.1 as a checklist, to be used as a reminder to cover all points in the initial design and, as the design matures, to assess the impact of changes.

Discussing the checklist items briefly, the customer or user may have specific requirements such as size, observability, or operational constraints that will limit the choices in regard to the primary power source or other subsystem elements. The target planet, whether Earth or another planet, and the resultant distance from the sun will in some cases limit design flexibility because of restrictions on available solar energy per unit area or, conversely, the requirement to control the temperature of exposed surfaces.

Lifetime requirements in a given operating environment may also drive the power system design. Solar array degradation due to radiation exposure may prevent use of these devices on long-lived spacecraft operating in the Van Allen

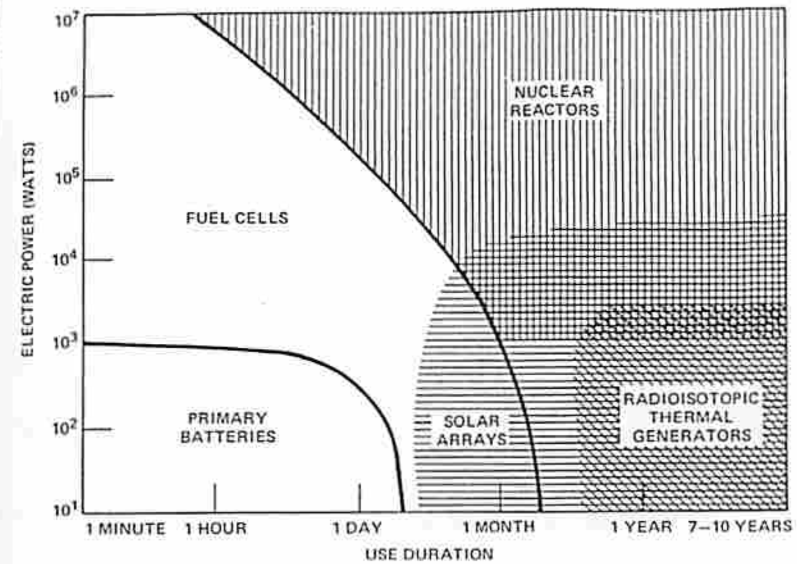

Fig. 10.2 Operating regimes of spacecraft power sources.

belts, for example. As will be seen, gallium arsenide solar cells offer improved radiation tolerance compared to silicon-based cells. Even so, the high radiation flux, particularly from inner-belt protons, place serious limits on array lifetime. Many spacecraft have a variety of operating modes requiring different power levels. The percentage of time in each mode is of great significance and may indicate a hybrid system using more than one power source or type of energy storage.

The attitude control concept employed will affect the power system both in terms of configuration constraints from solar arrays, waste heat radiators, and other elements, and in responding to specific power needs of the attitude control devices. The flexibility and frequency response of large arrays can, in turn, dictate the choice of attitude control effectors. Space mission history offers several notable examples of undesired control-structure interactions due to poorly modeled solar array flexibility effects. Finally, the attitude control system engineer will usually be involved in the design of whatever scheme is used to orient the solar arrays toward the sun.

Orbital parameters will strongly affect the choice of the primary power source and its configuration, as well as onboard energy storage requirements. However, despite the difficulties posed by some unique orbits and space mission requirements, system operation on a planetary surface will often be the most

Table 10.1 Power system design considerations

Customer/user
Target planet, solar distance
Spacecraft configuration
Mass constraints
Size
Launch vehicle constraints
Thermal dissipation capability
Lifetime
Total
Percentage in various modes, power levels
Attitude control
Spinner
Three-axis stabilized
Nadir pointing
Thrusters
Momentum wheel
Gravity gradient
Pointing requirements
Orbital parameters
Altitude
Inclination
Eclipse cycle
Payload requirements
Power type, voltage, current
Duty cycle, peak loads
Fault protection
Mission constraints and requirements
Maneuver rates
g loads

environmentally demanding, the most difficult in terms of deploying large solar arrays, and the most challenging in regard to meeting energy storage demands.

Specific mission demands may also impact the power system design. For example, a spacecraft that must maneuver rapidly may not be able to tolerate large, flexible solar arrays. A low-observable spacecraft may preclude use of concepts requiring high-temperature operation, such as imposed by RTGs.

10.5 Power System Elements

Figure 10.3 presents a typical spacecraft functional block diagram that identifies the major elements in the power system. A substantial variety of options exist within each of these elements. Table 10.2 identifies the options most likely to be encountered in normal spacecraft design practice.

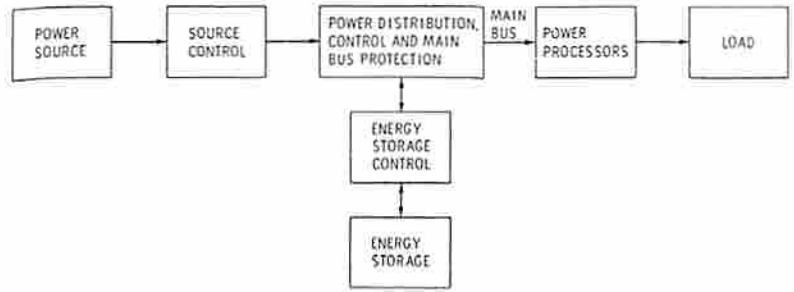

Fig. 10.3 Power subsystem functional block diagram.

10.6 Design Practice

Although details of design practice will vary from one organization to another, some broadly applicable rules can be articulated. These are discussed in this section.

10.6.1 Direct Current Switching

As a general rule, switches or relays should be in the positive line to a given element, with a direct connection to "ground" on the negative side. The purpose

Table 10.2 Power system elements

Power source
Solar photovoltaic
Radioisotope thermoelectric generator (RTG)
Nuclear reactor, static or dynamic energy conversion
Radioisotope dynamic
Solar dynamic
Fuel cells
Primary batteries
Source control
Shunt regulator
Series regulator
Shorting switch array
Energy storage control
Battery charge control
Voltage regulation
Power conditioning
DC-DC converters
DC-AC inverters
Voltage regulation

of this design practice is of course to allow power to be shut off in the event of a short circuit or other high-current flow failure within the element.

In a famous manned spaceflight emergency, the Gemini 8 mission was aborted less than a day into a planned three-day mission, following loss of flight control caused by a roll thruster stuck in the "on" position. Post-flight inspection of the spacecraft revealed a small solder ball shorting the thruster (which was switched in the negative, or return, line and thus was electrically "hot") to spacecraft ground, rendering the astronauts' hand controller inoperative with respect to that thruster.

Not surprisingly, astronauts Neil Armstrong and David Scott were unable to diagnose the problem while trying to cope with roll rates of up to 300 deg/s. Control could only be regained by shutting down the entire system and reverting to the backup reentry flight control system, after which individual reactivation of the primary thrusters, one by one, revealed the culprit. Disabling this thruster effectively solved the problem. However, according to the mission rules then in force, activation of the reentry flight control system required a mandatory abort, resulting in the loss of numerous mission goals.[1] Subsequent vehicles were rewired to have the thrusters switched in the positive control line, so that the thruster body was electrically inert when not firing.

10.6.2 Arc Suppression

To maximize effectiveness, arc suppression devices should be located as close to the source of the arc as possible. As discussed in Chapter 3 in connection with spacecraft charging, conductive cables, connectors, solar array edges, and other current-carrying elements on LEO spacecraft should not be exposed to the ambient plasma, particularly if the spacecraft operates at a bus voltage comparable to the arcing threshold of the conductor materials. High altitude spacecraft should be designed to ensure conductive paths between all spacecraft elements to preclude differential charging of isolated sections and subsequent arcing between them.

10.6.3 Modularity

Modular construction is desirable to simplify testing and to expedite replacement of failed or suspect units during system test or launch preparation. Spacecraft designers learned early, and at their peril, that the necessity to disassemble or remove several non-offending units to achieve access to a suspect system often resulted in "collateral damage" to the innocent parties, not to mention an excessive workload for numerous technicians. It was realized that the savings in mass and volume were more than offset by the delays and reductions in reliability caused by excessive "stacking" of components and subsystems. Thus, modern space systems design favors considerations of access and maintainability as well as conservation of mass and volume.

10.6.4 Grounding

Spacecraft grounding practices are often the subject of considerable debate among practitioners. This text presumes no single approach to spacecraft grounding practice. Many successful satellite and spacecraft programs have used a variety of grounding techniques of varying levels of complexity. However, some design principles are widely accepted,[2] and we advocate those here.

Use of a common ground cable is generally preferable to individually grounding various components and circuit elements to the structure, because it is difficult to maintain high continuity between isolated structural elements. When such difficulty occurs, electrical resistance results between various parts of the spacecraft ground structure that are intended to be at a common potential i.e., to have no electrical resistance between them. Thus, different portions of the electrical system, intended to be at the same electrical potential above the common ground, will not be so, and may therefore not function as intended.

As a simple example, a semiconductor switch may be designed to "trip" upon application of a 5-V potential, but not at a 3-V potential difference to avoid spurious switching due to electrical noise on the line. If there is enough electrical resistance in the ground loop to generate a 2-V drop on the return line, the switch can never be activated. While this may seem to be an extreme example, a voltage drop of this magnitude can easily occur across a long ground loop such as might be found on a large spacecraft. Numerous examples, some of direct experience to the present authors, occurred during early shuttle operations, when experience with large payloads remotely mounted in a cargo bay aboard an even larger vehicle was then minimal.

For these reasons, such "ground loops" are obviously to be avoided whenever possible. This gives rise to the practice of providing a common, low-resistance grounding strap or cable to all subsystems throughout the spacecraft, which is then carefully grounded at a single point to the spacecraft structure. Any ground circuit current flow is less likely to disturb sensitive components if confined to a properly isolated and connected ground cable. When electrical noise is thought to be a problem for certain circuits or instruments, as is often the case, noisy power-switching ground lines will often be kept separate from "signal" ground lines, prior to structural grounding. For similar reasons, radio-frequency (RF) grounding requirements may be incompatible with the needs of other systems, except again for the general requirement to be ultimately tied back to the primary structure.

On some occasions it may be necessary to isolate completely a given instrument or subsystem from other sources of spacecraft electrical noise. When this is the case, some portions of the vehicle will be electrically isolated, or "floated," with respect to others. Provided that it is intentional, this practice is acceptable. However, it is then necessary to take care to ensure that the separately floated ground does not inadvertently contact the "common" ground, because current would then flow between them.

It is sometimes very awkward, if not impossible, to provide a single-point common structural ground to subsystems in all portions of a large or complex space vehicle. When a single-point ground is difficult to achieve, but a common ground plane remains necessary, a multipoint grounding architecture will be employed. Several different grounding points to the structure are provided, with every effort made to maintain very low resistance paths between them. Nonetheless, the single-point ground generally remains the design objective to be achieved.

10.6.5 Continuity

Good continuity should be maintained between structural elements, thermal blankets, etc., to minimize the probability of buildup of static electrical potential or other voltage differences.

10.6.6 Shield Continuity

Shield continuity must be maintained across all connections. A single-point shield ground is desirable to minimize the possibility of shield current flow. Most circuits, especially noise-sensitive or noise-generating circuits, will be shielded, with the shields sharing a common ground if possible, as previously discussed.

10.6.7 Complexity

In keeping with good general engineering practice, the spacecraft power system should be no more complex than is necessary to do the job. Excessive, unnecessary complication will increase design, fabrication, and test costs and increase the probability of failure. It is for exactly this reason that the choice is often made to offer a relatively simple menu of power supply bus voltages to the various spacecraft subsystems, and to allow each of them to deal separately with any special requirements. Such a solution is rarely the least massive, and never the most electrically efficient, but it usually offers gains in simplicity that should be ignored only when no reasonable alternative exists.

Particular circumstances may force a violation of any of the previously mentioned rules to meet some overriding requirement. In the absence of such a requirement, however, adherence to these rules is very much recommended and will generally have a desirable impact on the overall operation.

10.7 Batteries

Batteries have been and will continue to be for the foreseeable future the primary means of electrical energy storage onboard spacecraft. In the following

discussion, a variety of terms relating to batteries will be used. These are defined here to enhance understanding of the material to follow:

Charge capacity, C_{chg}	Total electric charge stored in the battery, measured in ampere hours (e.g., 40 A for 1 h = 40 Ah).
Energy capacity, E_{bat}	Total energy stored in the battery, equal to charge capacity (Ah) times the average discharge voltage, typically measured in units of Joules or watt hours.
Average discharge voltage, V_{avg}	Number of cells in series times cell discharge voltage (1.25 V for many commonly used cells).
Depth of discharge, DOD	Percent of battery capacity used in the discharge cycle (75% DOD means 25% capacity remaining, the DOD is usually limited to promote long cycle life).
Charge rate, R_{chg}	Rate at which the battery can accept charge (measured in amperes per unit time).
Energy density, e_{bat}	Energy per unit mass [J/kg or (W · h)/kg] stored in the battery.

A battery (strictly speaking, an individual cell of a battery) is a device that converts chemical energy directly to electrical energy. A single cell has a negative electrode, a conductive electrolyte, and a positive electrode. The electrolyte may be in liquid, paste, or solid form; potassium hydroxide (KOH) is a common choice. If the cell is connected to an external electrical load, electrons flow from the negative electrode, through the load, and back to the positive electrode. The chemical reaction essentially ceases when the load is removed; however, it should be noted that the battery will slowly degrade chemically over time, whether used or not. Thus, most batteries have a "shelf life" within which they must be used.

Batteries are divided into two major categories: primary and secondary. The former offer higher energy and power densities for a given battery chemistry but are by definition not rechargeable. This definition is sometimes stretched to include as primary batteries those which are rechargeable for only a few cycles. Primary batteries are especially well adapted to one-time events requiring substantial power and minimal mass, as with missiles and expendable launch vehicle stages.

10.7.1 Primary Batteries

In cases where extremely long installed storage is required, e.g., a missile in its silo or a planetary atmosphere probe that is inert during interplanetary transfer, the battery is often dry (i.e., without electrolyte) prior to activation. Upon activation, a pyrotechnic valve fires to allow the electrolyte to enter the battery from a separate reservoir. This approach provides a highly reliable quick-reaction power source that is nevertheless protected from degradation and requires no maintenance during extended storage. Another quick-reaction, dry storage battery is the thermal battery. In this case, the electrolyte is solid at normal temperature. Ignition of a chemical heater, which melts the electrolyte and results in a fully charged battery, activates the battery. The battery stays active as long as the electrolyte is molten or until it is fully discharged.

A major application of these types of batteries in long-life space systems is to supply power to activate pyrotechnic charges and other deployment devices. Such devices typically are operated at the beginning of the mission or for relatively brief periods during a longer mission. Another application is to short-duration, high power-drain devices such as electromechanical actuators. For a variety of reasons, such as minimizing power drain or isolating noisy circuits from the main power bus, it may be desirable to operate these circuits from a primary battery that is completely isolated from the main power system.

During the early years of space systems development, the most common type of primary battery was the silver-zinc battery, usually abbreviated Ag-Zn. This battery has excellent energy density and is still the battery of choice in many cases. In recent years a variety of batteries based on lithium in combination with various other materials have come on the scene. Some of these batteries offer the highest energy density currently available. Certain types of lithium batteries experienced significant "teething problems" in early applications, showing a distressing tendency to explode in some situations. Leakage and corrosion problems have also been encountered. However, these problems have largely yielded to better understanding of battery characteristics and ensuing engineering development, and lithium batteries can today be reliably employed in many space vehicle applications.

10.7.2 Secondary Batteries

The rechargeable or secondary battery generally has a much lower energy density, which is further aggravated by limitations on the depth of discharge. Again, silver-zinc batteries were the most commonly used for a number of years and have demonstrated good energy density (which is nonetheless reduced as compared to the primary form of these batteries, due to the extra wrapping material used to isolate each cell in the battery when it is intended to be recharged). However, these batteries suffer from life limitations, especially in applications involving a large number of charge/discharge cycles. As a result, nickel-cadmium (Ni-Cd, or nicad) batteries have been for many years very nearly the standard for spacecraft applications. Certainly they have been the most common in LEO spacecraft designs.

A more recent development in battery technology is the nickel-hydrogen (Ni-H_2) design. This battery differs from other types in that a large amount of free hydrogen is generated as part of a charge/discharge cycle. As a result, quite high pressures are generated and the battery case is, in fact, a pressure vessel. (Actually, other battery types, such as Ni-Cd, do generate some pressure and require a reasonably strong case to contain it. However, Ni-H_2 battery pressure exceeds that of nicads by a factor of 10.) Ni-H_2 batteries are capable of greater depth of discharge than nicads and, even with the penalty of the high pressure case, offer better energy density. Ni-H_2 batteries do not require reconditioning. In large part because of this advantage, Ni-H_2 batteries have been very competitive in recent years with nicads, particularly for GEO spacecraft, and as this is written may even be used in the majority of new spacecraft.

The pressure vessel cases of Ni-H_2 batteries are generally cylindrical with hemispherical ends. This makes close packing difficult. In an effort to avoid the pressure containment problem while retaining the other advantages of nickel-hydrogen systems, the nickel-metal hydride battery (Ni-MH) was developed. This battery depends on the ability of some metallic hydrides to contain large amounts of hydrogen in the structure at low pressure. This allows the battery cell case to be rectangular like most other batteries, allowing for more efficient packing. Ni-MH batteries are in very common commercial use in cell phones, laptop computers, etc. Unfortunately, space applications of Ni-MH batteries have been few, primarily because of the limited cycle life so far demonstrated for this technology.

Lithium-based secondary batteries are also available and offer excellent energy density. Some of the chemistries available do require reconditioning. Table 10.3 provides a list of various battery types and the characteristics of each.

As noted earlier, the battery average discharge voltage V_{avg} is the product of the individual cell average discharge voltage and the number of cells in series. As seen in Table 10.3, the cell voltage of most of the battery chemistries discussed here is approximately 1.25–1.50 V. All batteries will have a higher discharge voltage when fully charged than when nearly depleted. Indeed, the drop in output voltage below a specified threshold is the indication that it is time to recharge the battery. A nicad cell might have an average discharge voltage of 1.25 V, with a charging cycle mandated should the voltage drop below 1.1 V.

Most spacecraft systems flown to date by the United States have used 28 VDC as the nominal bus voltage; thus, most associated battery hardware has also been designed for 28 VDC. This practice reflects the heritage of early spacecraft avionics from aircraft systems in terms of electronic component design and usage. Generally speaking, this was satisfactory for the relatively small, low-power spacecraft flown in earlier decades. However, as larger and more powerful systems have become common, higher voltage systems have become more attractive. The future will undoubtedly see a continuing trend toward the use of higher voltage spacecraft power buses (see Fig. 10.1). This practice reduces the current-handling requirements of the spacecraft wire harness and thus the attendant weight of that harness. Also, because resistive losses (and heating) are proportional to the square of the current being carried, such inefficiencies are also minimized by the use of a higher bus voltage.

Table 10.3 Battery chemical types

Silver-zinc, (AgZn)
- Commonly used in early space systems; still popular
- Good energy density [175 (W · h)/kg primary, 120–130 (W · h)/kg secondary]
- Limited cycle life (2000, 400, 75 at 25, 50, 75% DOD)
- 1.50 V/cell

Silver-cadmium, (Ag-Cd)
- Better cycle life than Ag-Zn, better energy density than Ni-Cd
- Fair energy density [60–70 (W · h)/kg secondary]
- Fair cycle life (3500, 800, 100 at 25, 50, 75% DOD)
- 1.10 V/cell

Nickel-cadmium, (Ni-Cd)
- Most common secondary battery presently in use
- Low energy density [20–30 (W · h)/kg]
- Long cycle life (20,000, 3000, 800 at 25, 50, 75% DOD)
- Good deep discharge tolerance
- Can be reconditioned to extend life
- 1.25 V/cell

Nickel-hydrogen, (Ni-H_2)
- High internal pressure requires bulky pressure vessel configuration
- Good energy density [60–70 (W · h)/kg]
- Good cycle life (15,000, 10,000, 5000 at 25, 50, 75% DOD)
- No reconditioning required
- 1.30 V/cell

Nickel-metal hydride, (Ni-MH)
- Same chemistry as nickel-hydrogen
- Hydrogen adsorbed in metal hydride to reduce pressure
- Improved packaging relative to nickel-hydrogen
- Good energy density
- Limited cycle life
- 1.30 V/cell

Lithium batteries
- Several types (Li-$SOCl_2$, Li-V_2O_5, Li-SO_2)
- Both primary and secondary designs available
- Very high energy density [650 (W · h)/kg, 250 (W · h)/kg, 50–80 (W · h)/kg secondary]
- Higher cell voltage (2.5–3.4 V)

The use of higher bus voltages is not an unmitigated good. As discussed in Chapter 3, LEO spacecraft tend to accumulate negative charge from the ambient plasma, to a level about 90% of the maximum negative exposed-conductor voltage, relative to the plasma reference potential. If this level exceeds the arcing threshold of common conductors, problems will occur. Earlier, lower voltage bus levels were not vulnerable to this effect.

Depth of discharge limitations usually require a tradeoff between battery mass due to the unused capacity and battery degradation and lifetime reduction due to repeated deep discharge. Spacecraft in low-altitude, low-inclination orbits around the Earth or another planet typically experience the most severe usage in terms of charge/discharge cycles, because they experience eclipse on each orbit. In LEO a spacecraft battery will be discharged and charged some 12–16 times per day. This results in some 10,000 or more cycles in only a few years, yet modern spacecraft are normally expected to function for substantially longer periods of time.

Most battery chemistries so far developed cannot accept so many charge-discharge cycles; thus, for such applications, Ni-Cd batteries have been the system of choice, despite their low energy density. However, even with nicads, it is necessary to limit the depth of discharge to a relatively small amount, 15–25%, and to recondition the batteries periodically if the desired total lifetime is to be obtained. As experience has grown with Ni-H_2 batteries, their advantages in this regard have made them the system of choice in many cases, especially for large spacecraft.

Eclipse time in low orbit can be as high as 40% of the orbital period, or on the order of 35 min for Earth orbits. Spacecraft in synchronous equatorial (geostationary) orbits go for extended periods without encountering eclipse. However, GEO spacecraft encounter two eclipse seasons each year, with each period being 45 days long. During these periods the spacecraft encounters one eclipse each day ranging from momentary duration at the beginning and end of the period up to 72 min at the midpoint.

Some spacecraft in near-polar sun-synchronous orbits with the orbit plane aligned essentially along the terminator may never be in eclipse; these are often called dawn-dusk orbits. The same may be true for deep space vehicles. This does not usually mean batteries are not needed, however. It may be necessary to maneuver the spacecraft off the sun line to obtain proper thruster pointing for course correction. Even if this is not necessary, it may be more efficient to use a battery to handle intermittent peak loads rather than to oversize the solar arrays to cope with the peak load. For similar reasons batteries may be required even on spacecraft using power sources (such as RTGs) that do not depend on the sun.

Given the power usage of the spacecraft and the maximum allowable depth of discharge (DOD) for the design lifetime of the battery, the battery can be sized by the following equation:

$$\mathrm{DOD} = \frac{\text{Energy required during eclipse}}{\text{Stored battery energy}} \tag{10.1a}$$

or

$$\mathrm{DOD} = \frac{P_L t_d}{C_{\mathrm{chg}} V_{\mathrm{avg}}} = \frac{P_L t_d}{E_{\mathrm{bat}}} \tag{10.1b}$$

where

P_L = load power in watts
t_d = discharge time in hours
C_{chg} = charge capacity in ampere hours
V_{avg} = battery average discharge voltage in volts
E_{bat} = total battery energy capacity

The charge rate also drives battery size; a power input level that is too high can result in overheating of the battery and, if carried to extremes, explosive destruction. Although strict mathematical guidelines do not exist, a good rule of thumb for the allowable charge rate is

$$R_{chg} = \frac{C_{chg}}{15\,\text{h}} = I_{chg} \tag{10.2}$$

where the charge capacity is given in ampere hours. A "trickle charge," used when it is desired to store the maximum amount of charge in a battery, might use a charge rate of $C_{chg}/45$ h. Note that the rate of charge has dimensions of current, in this case a charging current.

According to the empirical rule of Eq. (10.2), a battery can accept a charge equal to 1/15 its total capacity per hour. This can prove to be a significant constraint in many cases. In a typical LEO spacecraft, where 40% of the orbit is spent discharging and 60% charging, the depth of discharge during eclipse is limited by the rate at which the charge can be restored during the illuminated phase, with the allowable rate given roughly by Eq. (10.2). All of the expended energy must be restored during the charge cycle, or there will be a net drain on the battery, which will ultimately result in the need to reduce the operations load to avert failure of the power system. Thus, the size of the battery in this case is driven not by the maximum allowable DOD as governed by the battery chemistry, but by the charge rate, with the depth of discharge per orbit limited to 7–8%. On the other hand, a GEO spacecraft will encounter only a few hundred discharge cycles in a 10-year life, and will have more than ample recharge time. Much deeper discharge can be tolerated in this case.

Equation (10.2) is quite conservative. Substantially higher recharge rates may be acceptable for a given battery; the manufacturer's specifications should always be the ultimate guide. Also, Eq. (10.2) is rather simplistic, because a variety of environmental factors can influence the allowable rate of charge, most importantly the battery temperature. Additionally, it should be noted that a battery generally must be charged at a slightly higher voltage than V_{avg}, or a full charge cannot be restored. Typically, the charging voltage will be of order 20% higher than the average discharge voltage. This will have implications for solar array design, as we will see in the following.

We offer here a simple example to demonstrate the process of preliminary battery size definition.

Example 10.1

What is the required size of a nicad battery to support a 1500-W payload in geostationary orbit, given the following design data:

Bus voltage	28 VDC
Peak load	1500 W
Maximum load duration	1.2 h
Battery energy density	15 (W · h)/lb at 100% DOD
Average cell voltage	1.25 V
Maximum DOD	70%

Solution. The number of cells is

$$N_{cell} = \frac{V_{bus}}{V_{cell}} = 22.4$$

We can choose either 22 or 23 cells; selecting 22 cells saves mass and results in a perfectly acceptable bus voltage of 27.5 VDC. From Eq. (10.1b) the total charge capacity and battery energy capacity are

$$C_{chg} = \frac{P_L t_d}{(V_{avg}\,\text{DOD})} = \frac{(1500\,\text{W})(1.2\,\text{h})}{(0.7)(27.5\,\text{V})} = 93.5\,\text{Ah}$$

and

$$E_{bat} = C_{chg} V_{avg} = (93.5\,\text{Ah})(27.5\,\text{V}) = 2571\,\text{W} \cdot \text{h}$$

The battery mass is

$$m_{bat} = \frac{E_{bat}}{e_{bat}} = \frac{2571\,\text{W} \cdot \text{h}}{15\,(\text{W} \cdot \text{h})/\text{lb}} = 171\,\text{lb}$$

It may be desirable to split the battery into two or three individual battery packs for ease in packaging, placement, and balance. Each battery pack must contain 22 series-connected cells to maintain the proper voltage. Finally, redundancy management issues have been ignored in this example.

10.7.3 Nicad Reconditioning

As mentioned earlier, to obtain maximum life from a nicad battery subject to repeated discharge, a reconditioning process is required. Reconditioning consists of a very deep discharge to the point of voltage reversal, followed by recharge under carefully controlled conditions. Figure 10.4 shows the effect of this operation. In the absence of reconditioning, the battery voltage begins to decline and, after four or five eclipse seasons, declines fairly rapidly. Usable depth of discharge is also greatly diminished. On the other hand, with periodic reconditioning the voltage declines only slightly from the "new" level and reaches a steady-state level. (Note that "eclipse season" refers to the two periods

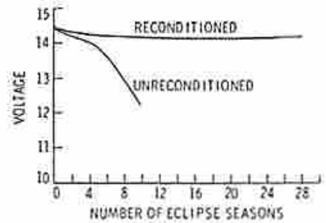

Fig. 10.4 Reconditioning of NiCd batteries.

per year when a geosynchronous spacecraft is eclipsed once per orbit. Thus, the scale on the abscissa converts to years when divided by two.) Because each eclipse season lasts for a few weeks at six-month intervals, it is easy to perform a reconditioning cycle during the several months of full solar exposure between eclipses. Other battery chemistries show a similar characteristic but are less widely used in space applications.

10.8 Primary Power Source

Once beyond the relatively small range of mission requirements for which batteries alone are suitable, choice of a prime power source includes several possibilities. The choice is governed by a variety of factors including the required power level, operating location, life expectancy, orientation requirements, radiation tolerance, and cost. Figure 10.2 depicts operating ranges that are deemed generally suitable for various prime power sources, based on power level and lifetime. Such curves are intended to be broadly indicative rather than specifically definitive. There is in any case substantial overlap between the various regions, indicating areas where more than one choice may be feasible. Also, other factors may bias the choice in a direction that would not be optimal from the viewpoint of simply meeting requirements on mass, power, and lifetime.

As previously observed, power requirements for spacecraft have tended to grow with time, and the related main bus voltage has risen accordingly in an effort to reduce conductor and component masses and resistive losses. Figure 10.1 presents data on past spacecraft as well as predictions concerning near- and far-term applications, both civil and military. The lifetime required of space assets has increased as well, a consequence of the large investment to build and operate a modern spacecraft and its associated ground equipment. In the case of scientific spacecraft, the required operational duration has increased because the targets are more distant, or the missions more complex, or both. In any case, life expectancy has become an increasingly important factor, especially because most spacecraft cannot readily be serviced or refurbished.

Environmental factors to be considered include the obvious issue of access to adequate solar illumination. If the spacecraft is too far from the sun (and Mars at 1.5 AU is at roughly the useful outer limit), solar arrays are not a viable choice. Concentrators may extend their capability to a limited degree, but eventually the inverse-square law renders solar energy simply too diffuse to be useful. Radiation resistance is another significant consideration; solar cells are seriously degraded by extensive exposure to radiation. This can be a major consideration for a spacecraft that must operate extensively in the Van Allen belts or other high-radiation environments.

In subsequent sections, various prime power sources are discussed, particularly in terms of the capabilities and limitations of each. Detailed technical descriptions of each are beyond the scope of this book, and the interested reader is referred to more specialized literature for such information.

10.9 Solar Arrays

Regardless of the size of the total array, each array is made up of a very large number of individual cells arranged on a substrate of some type. Although each cell puts out a relatively small current and voltage, proper series and parallel connection can provide any desired current and voltage within reasonable physical limitations. Individual cells are made in a variety of shapes and sizes. Probably the most common as this is written is the rectangular cell with dimensions on the order of 2×4 cm; however, cells in common use range from 2×2 cm to 2.5×6.2 cm. The rectangular shape allows for reasonably efficient packing, enabling array size and mass to be minimized. A well-designed array might have a cell packing density of 90%. Because some minimum spacing and allowance for connections must be provided, it is difficult to improve significantly on this, although innovative techniques may allow some gains to be made.

Because solar arrays can be quite large for higher power spacecraft, it quickly becomes impossible to find adequate area on the fixed spacecraft structure. Early low-powered spacecraft did in fact restrict the array area to the spacecraft skin. Most designs were drum-shaped spinning spacecraft, where only about 40% of the array was illuminated by the sun at any time. As power requirements grew, fixed arrays that were not specifically part of the spacecraft structural shell or skin were tried; however, launch vehicle nose fairing dimensions limit the utility of such an approach, and deployable solar arrays made an early appearance. Figure 10.5 depicts a variety of solar array designs.

Deployable solar arrays have typically been semirigid paddle-like structures that are deployed from the main structure after the spacecraft is injected into orbit. Keeping the array firmly locked to the spacecraft structure during launch allows the use of extremely lightweight structures. Such designs are constrained more by the need for rigidity than for strength; thus, structures having very thin

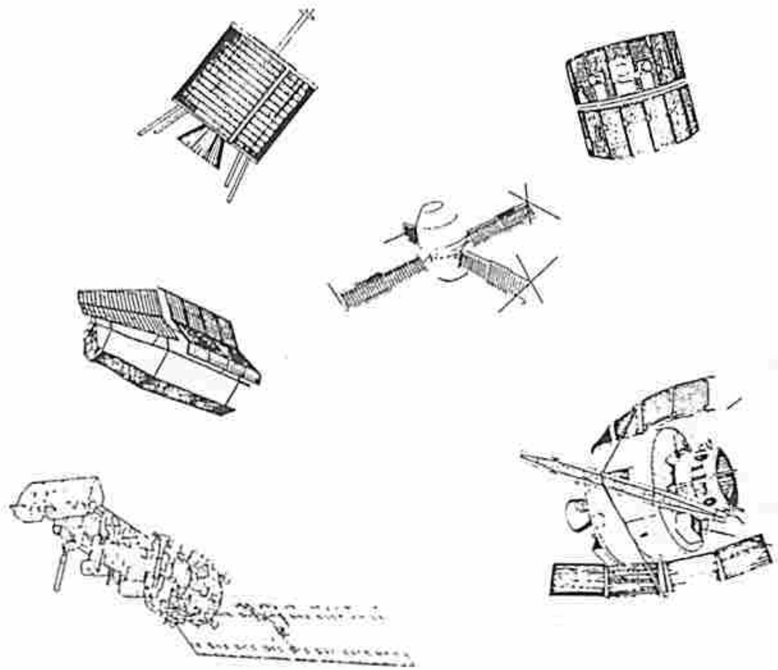

Fig. 10.5 Spacecraft solar array concepts.

cross sections are possible. Such structures are highly susceptible to handling damage, and often the primary criterion defining material thickness is the need to handle the assembly during installation. The development of highly rigid composite materials in recent years has greatly enhanced the possibilities for lightweight solar array design.

Because all photons incident upon a solar cell are absorbed within (at most) the first 10 μm, the active semiconductor material need not be at all thick. (Indeed, very thin cells allow red and infrared radiation to pass through the material without being absorbed, enhancing the conversion of shorter wavelength and hence higher energy photons, and increasing overall cell efficiency.) Similarly, the antireflective coating, cover glass, adhesive, and substrate that comprise the complete cell do not enhance the design by being thicker. Thus, the possibility and potential convenience of roll-up solar arrays was recognized early in the history of spacecraft design.

However, the early technology did not lend itself to such an approach; solar cells were too thick, connections were too stiff, and suitable substrates did not exist. Subsequently, however, advances in technology caught up with the

concept, and a variety of flexible roll-up and fold-up solar arrays are now in routine use. Perhaps the most spectacularly visible example was the 12.5-kW array demonstrated on space shuttle mission STS-10. This early demonstration led to the deployable arrays presently used on the International Space Station. This type of design, along with a large variety of deployable rigid array concepts, allows convenient packaging of very large arrays for launch.

Flexible solar arrays can introduce problems simply because of their flexibility. The primary resonant frequency of the roll-up arrays originally used on the Hubble Space Telescope (HST) allowed an undesirable interaction with the bandwidth of the attitude control system, causing difficulty in achieving the accurate pointing required. On-orbit replacement with stiffer arrays has largely corrected the problem. Another problem with very large arrays and the attendant high voltage and power levels they produce is the conductor mass and the required insulation between circuit elements. This can be particularly trying in flexible arrays, and represents one of the practical limits that solar array technology may impose on the spacecraft designer. Still, roll-up arrays offer the lowest mass approach currently available for providing large array areas.

To extend the capabilities of solar arrays to regions farther from the sun, reflective concentrators of various types have been proposed. Figure 10.6 shows two concentrator concepts. The flat concentrator array for use with silicon (Si) cells is basically a trough that increases the collection area relative to the cell area. This concept is useful at solar distances beyond 1.5 AU by increasing the energy available for conversion and keeping the cells from becoming excessively cold. Such concentrators can probably extend the useful range of Si cells out to 3–4 AU and possibly farther.

The other concentrator concept is particularly directed toward gallium arsenide cells. These cells, while coming into more common use, are nonetheless quite expensive relative to silicon arrays, and it is therefore desirable to minimize cell area. Also, these cells function best at a higher temperature than their silicon counterparts. The concept shown concentrates sunlight from a large collection area onto a small cell area. This reduces cell cost and brings the cells to a higher operating temperature than would otherwise be the case, thus providing a double benefit. An obvious disadvantage is that such concentrators are complex and expensive to manufacture. Any concentrator, even the relatively simple one shown for the conventional silicon array, clearly complicates stowage and deployment.

The highly successful Deep Space 1 technology demonstrator spacecraft used an array of Fresnel lenses to concentrate sunlight on the solar cells.

The HS-702 geosynchronous communications spacecraft used a trough-type concentrator for many years. However, problems began to develop, resulting in unacceptable power output degradation, and the concentrators were eventually eliminated. It appeared that the power loss was due to contamination, and it is possible that a change of material could have solved the problem while retaining the advantage of the concentrators.

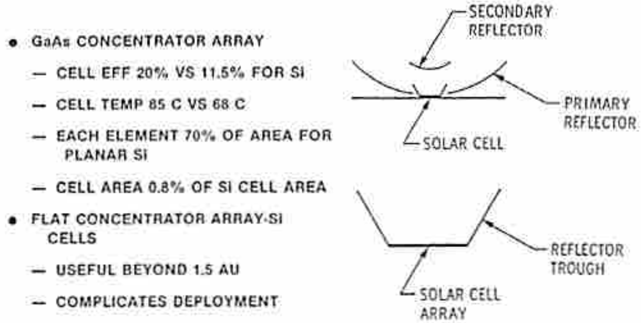

Fig. 10.6 Solar concentrators.

10.9.1 Solar Cell Characteristics

Certain technical characteristics of solar cells, such as temperature dependence and the current-voltage (I-V) curve, are of interest to the spacecraft designer.

As with other semiconductor devices, solar cell characteristics are temperature dependent. The first-order effect for a typical silicon cell operating within its normal range is a voltage decrease with temperature; the second-order effect is an increase of current flow with temperature. The effect on current is roughly 10% of that on voltage, so that the net result is a decrease in output power with temperature. Figure 10.7 illustrates this behavior; again, as voltage increases, current drops. The temperature coefficient for voltage, γ_V, will be in the range of -2 to -3 mV/K, while γ_I, the temperature coefficient for current, will be approximately 0.2–0.3 mA/K. The temperature corrections (from reference conditions) for voltage and current output are of the form

$$V = V_{\text{ref}} + \gamma_V(T - T_{\text{ref}}) \tag{10.3}$$

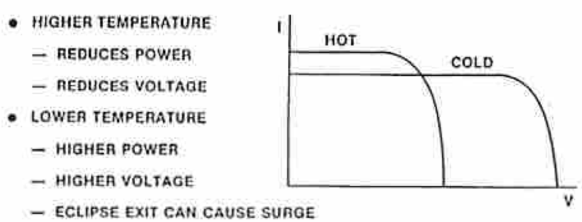

Fig. 10.7 Effect of temperature on solar cells.

and

$$I = I_{\text{ref}} + \gamma_I(T - T_{\text{ref}}) \tag{10.4}$$

Unless it is important to obtain the highest possible accuracy, it is common to ignore the separate variations of voltage and current and use instead a composite temperature coefficient for power in an equation of the same form as Eqs. (10.3) and (10.4). However, the effect of the voltage increase for cold panels, as with a spacecraft exiting an eclipse period, must be considered in the design of the power system, because a major power surge can occur under these circumstances.

The shape of the I-V curve shown in Fig. 10.8 is typical of solar cells and is important in the design of spacecraft power systems. To minimize mass and maximize efficiency, it is obviously desirable to operate the array at its maximum power point. Because power is the product of current and voltage,

$$P = IV \tag{10.5}$$

selection of the operating point to maximize the area under the I-V curve allows the maximum power point to be found. This is the point at which the maximum area rectangle that will fit within the I-V curve intersects the curve. As can be seen in Fig. 10.8, this lies on the knee of the I-V curve. Although some specific applications may dictate operation at some other point, the majority of systems will be designed for maximum power point operation. With the maximum power point for the cells defined, the current and voltage of individual cells is known. This information, in conjunction with the voltage and current requirements of the spacecraft, defines the series-parallel arrangement of the cells in the array.

It is common to specify the maximum power operating point, V_{mp} and I_{mp}, at a given temperature, often room temperature, as reference conditions in Eqs. (10.3) and (10.4), though of course this is not required. For a conventional silicon cell, V_{mp} will be 0.4–0.5 V at moderate temperatures. The maximum power operating point for current, I_{mp}, depends on the area of the cell for a given illumination level. For cell sizes in common use, I_{mp} will be in the range of 30–120 mA.

It occasionally results that a transient load or other problem will drive the operating point off the knee of the curve and down into the lower voltage range. The array may then not be able to return to the normal operating point. Having a battery in the system helps to stabilize it against such eventualities. Even systems

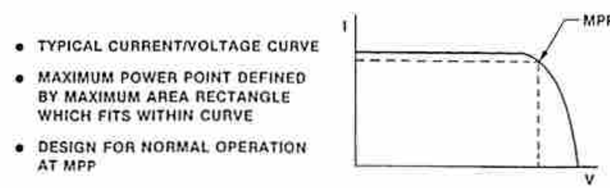

Fig. 10.8 Typical solar cell power characteristic.

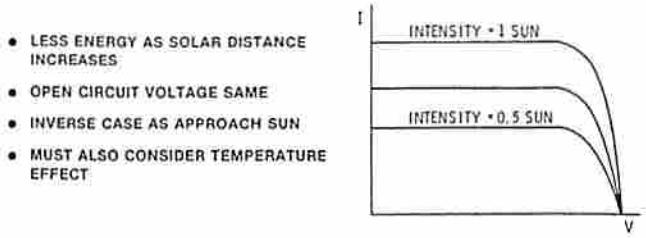

Fig. 10.9 Effect of distance from the sun.

that may not require a battery in normal operation (because they are never in eclipse) may require a battery for this reason. Off-line batteries or other devices that can be switched into the circuit to provide a temporary power boost may also be used. The operational alternative, of course, is controlled load-shedding to allow the bus voltage to recover.

As the solar array moves farther from the sun, the available current drops, while open circuit voltage stays the same, or increases if the temperature is lower. (This behavior is a consequence of the photoelectric effect, one of the first quantum-mechanical physical phenomena to be observed. Not widely realized is the fact that Albert Einstein received his Nobel prize for the explanation of the photoelectric effect, rather than for his development of the theories of special and general relativity, which remained controversial for several decades.) This leads to a family of curves similar to that shown in Fig. 10.9. A similar set could be drawn for an array moving toward the sun, with current increasing and voltage dropping slightly as solar distance decreases. Regarding Fig. 10.9, it will be noted that as a result of these changes the maximum power point moves slightly. This should be considered in spacecraft power system design for planetary missions. For example, in the case of a Mars orbiter, one would normally design for the maximum power point corresponding to Mars distance from the sun, because that is where the demand for power to operate the onboard instruments will be greatest, and because excess power will in any case be available while near the Earth.

10.9.2 Sun Tracking

It will be obvious that maximum power is available when the sun line is normal to the array. As the angle between the sun line and the array normal deviates from 0 deg, one expects that a cosine relationship between incidence angle and array output would obtain. This is indeed the case for angles up to about 60 deg. At larger angles, where the sun angle approaches parallel to the array face, the cosine relation begins to break down due to the finite thickness of the cells and other effects such as specular reflection from the cover glass surface. Figure 10.10 gives an approximate curve for current vs sun angle.

Although early spacecraft were often designed with fixed-orientation panels (if separate panels were used at all), modern spacecraft are almost universally designed to allow sun tracking by the solar array. The required tracking accuracy is not particularly challenging, since even a 10-deg error yields a cosine loss of only 1.5%. However, it is usually better to minimize sudden attitude disturbances, and so typically the solar array will be articulated in small increments more or less continuously, rather than in a few large maneuvers.

To cause a solar panel attached to a spacecraft in a planetary orbit to track the sun requires, in general, two angular degrees of freedom (DOF). The first degree of freedom (often called the α angle) compensates for the apparent rotation of the sun vector in the orbit plane, as seen from the spacecraft in orbit. Clearly, α will range from 0 to 360 deg over the course of a single orbit, a factor that must be considered in making arrangements to transfer power across the rotating interface between the solar panel and the spacecraft body. (The power transfer harness cannot continue to be wound indefinitely around a fixed axle on the spacecraft.) The second degree of freedom, the β angle, is necessary to compensate for the component of the sun vector normal to the orbit plane. Unless the orbit is sun synchronous, the β angle will vary more or less slowly throughout the year, as determined by the particular orbit parameters (see Chapter 4).

It is often possible to use the spacecraft itself to supply one of these degrees of freedom; for example, many spacecraft deal with changes in β angle by means of body rotation, leaving a single solar array drive gear to cope with α angle articulation. However, when numerous other instruments on a spacecraft must also be oriented properly, and when thermal control requirements are taken into account, it will in the end often be simpler to use a full 2-DOF separately

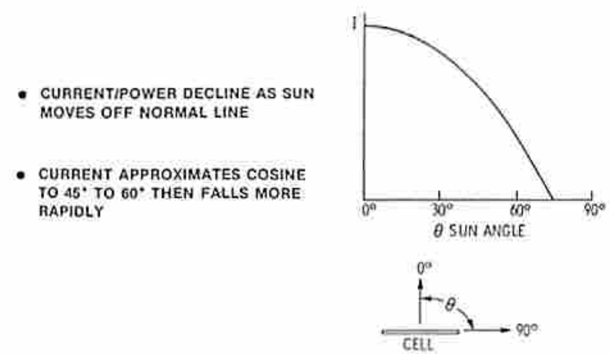

Fig. 10.10 Effect of sun angle on solar array power.

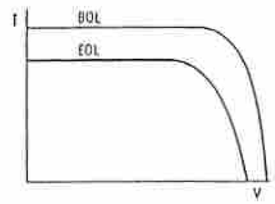

Fig. 10.11 **Radiation effect on solar cells.**

articulated solar array. These problems, and the decisions that arise from them, are usually the joint province of the attitude control engineer and the mechanical design engineer.

10.9.3 Radiation

As mentioned earlier, radiation has a detrimental effect on solar cells. The general effect of this degradation is shown in Fig. 10.11. Some loss of efficiency will take place during any mission of appreciable duration. If the operation takes place in more severe environments, e.g., the Van Allen belts, the rate of degradation will be more severe. Because the spacecraft normally requires as much power late in the mission as at the beginning, the solar array size must be based on end-of-life (EOL) capability rather than on beginning-of-life (BOL) performance characteristics. The radiation environment is discussed in general terms in Chapter 3, but for detailed design the specific environment should be assessed in the context of a particular mission, its orbit characteristics, its intended operational period relative to the 11-year solar cycle, etc.

10.9.4 Solar Cell Efficiency

Considerable effort has been expended on development of gallium arsenide (Ga-As) solar cells, which are more efficient and more radiation tolerant than those made of silicon. Figure 10.12 compares the radiation resistance of the two types of cells. Note that at some point the curves cross, and silicon may well be better again beyond that point. However, this effect occurs at very high radiation fluence and may not be of practical interest.

In terms of efficiency at the cell level, typical crystalline silicon cells at room temperature deliver a solar-to-electric conversion efficiency of 11–16% in production. As this is written, several commercial vendors have recently introduced production silicon cells with efficiencies in the 18–20% range.[3] Efficiency in the mid-20% range can be achieved in limited quantities. Gallium arsenide offers on the order of 18–20% efficiency in production cells, and close to 30% in special cases, but at a cost greater than that of silicon cells. As the cost

- RADIATION EXPOSURE REDUCES AVAILABLE POWER
- RADIATION REDUCES O/C VOLTAGE AND S/C CURRENT

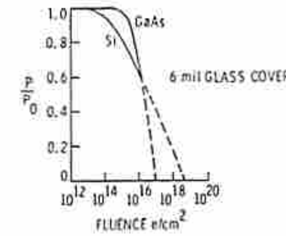

Fig. 10.12 **Solar photovoltaic radiation effect.**

of gallium arsenide cells has dropped, their use has become more common, particularly on long-lived GEO communications spacecraft, where the higher initial cost is easily overcome by the income potential of longer life. Concentrators may still be attractive in some applications to reduce the total expenditure for solar arrays, particularly Ga-As arrays.

A recent development, which offers the promise of very high efficiency, is the multijunction or multilayer solar cell. In this arrangement, the top or outer cell is optimized for conversion of light in the visible regime, where the solar output peaks. Beneath this top cell lie one or two layers of additional cells. These are optimized for energy conversion in the infrared range, thus using some of the energy that would ordinarily be radiated to space as waste heat. The conversion efficiency of the IR cells is much lower than that of the visible wavelength cells, but they nevertheless make a significant contribution. Cells of this type can deliver close to 30% conversion efficiency.

As would be expected, multilayer cells are heavier and more expensive than conventional cells. However, the cost penalty is not as great as might be imagined, because the additional layers of material are all deposited during a single manufacturing sequence in a vacuum chamber.

10.9.5 Preliminary Solar Array Sizing

Although detailed design of solar arrays is beyond the intended scope of this text, it is straightforward to perform preliminary sizing calculations to provide general characteristics of an array required for a given spacecraft. A simple example will illustrate these calculations.

Example 10.2

What is the size of a solar array necessary to support a 1500-W load, plus a suitable level of battery charging? If we assume 2 × 4 cm cells, how many are needed? The follow basic data may be used:

Cell efficiency	11.5% at 301 K
Maximum operating temperature	323 K
EOL degradation (10 years)	30%
Worst-case sun angle	6.5 deg off normal
Solar intensity	1350 W/m^2 at 1 A.U.
Temperature coefficient	-0.5%/K (power)
Packing factor	90% (10% loss for spacing)
Battery capacity	90 Ah

Solution. The array voltage must exceed the battery voltage for the battery to charge. For these voltage levels, a good rule of thumb is that the array must operate at a level 20% above the battery voltage. Assuming a 27.5-V battery as in Example 10.1, we have

$$V_{\text{array}} = (1.2)(27.5\,\text{V}) = 33\,\text{V} = V_{\text{chg}}$$

The EOL power requirement is equal to the 1500-W load plus the required battery charging power, which, from the empirical rule of Eq. (10.2), is found to be

$$P_{\text{chg}} = V_{\text{chg}} I_{\text{chg}} = V_{\text{chg}} R_{\text{chg}} = \frac{V_{\text{chg}} C_{\text{chg}}}{15\,\text{h}} = \frac{(33\,\text{V})(90\,\text{Ah})}{15\,\text{h}} = 198\,\text{W}$$

Thus, the total EOL power required of the array is

$$P_{\text{EOL}} = 1500\,\text{W} + 198\,\text{W} = 1698\,\text{W} \cong 1700\,\text{W}$$

We must now assess the various efficiency factors that cause the BOL power level to degrade to the EOL condition.

The effect of temperature at the hot operating point is to reduce efficiency by an amount proportional to the difference between the specified operating temperature and that at which maximum performance is obtained. Thus,

$$\eta_{\text{temp}} = 1 - \left(\frac{0.005}{\text{K}}\right)(323\,\text{K} - 301\,\text{K}) = 1 - 0.11 = 0.89$$

The degradation from radiation exposure is given as 30%, yielding an EOL efficiency due to radiation of

$$\eta_{\text{rad}} = 1 - 0.3 = 0.7$$

while the cosine loss due to the off-normal sun angle yields

$$\eta_{\text{angle}} = \cos(6.5\,\text{deg}) = 0.9766$$

The end-of-life power is the result of applying these losses to the beginning-of-life array power. Thus,

$$P_{\text{EOL}} = \eta_{\text{rad}}\eta_{\text{temp}}\eta_{\text{angle}} P_{\text{BOL}} = 1700\,\text{W}$$

and we then have

$$P_{\text{BOL}} = \frac{1700\,\text{W}}{0.619} = 2746\,\text{W} \cong 2750\,\text{W}$$

We are given a basic silicon solar array efficiency of $\eta_{\text{Si}} = 0.115$, and a solar illumination intensity of $I_s = 1350\,\text{W/m}^2$, so that in terms of total cell area A_{cell}, we have

$$P_{\text{BOL}} = \eta_{\text{Si}} I_s A_{\text{cell}} = (0.115)(1350\,\text{W/m}^2)A_{\text{cell}} = 2750\,\text{W}$$

hence

$$A_{\text{cell}} = 17.7\,\text{m}^2$$

The packing efficiency was given as $\eta_{\text{pack}} = 0.9$, and so the array area satisfies the relation

$$A_{\text{cell}} = \eta_{\text{pack}} A_{\text{array}} = 17.7\,\text{m}^2$$

and thus

$$A_{\text{array}} = \frac{17.7\,\text{m}^2}{0.9} = 19.7\,\text{m}^2 \cong 20\,\text{m}^2$$

For 2×4 cm cells, the area of a single cell is 8×10^{-4} m^2/cell; hence the number of cells is

$$N_{\text{cell}} = \frac{A_{\text{cell}}}{8 \times 10^{-4}\,\text{m}^2/\text{cell}} = 22{,}142\ \text{cells}$$

10.10 Radioisotope Thermoelectric Generators

The radioisotope thermoelectric generator (RTG) is a power source that renders the spacecraft independent of the sun. Although this is an advantage in many cases, it comes at a price that explains why these units have seen only limited use. The RTG functions by converting the heat energy generated by decay of a radioisotope into direct current electricity by means of the thermoelectric effect. In a typical RTG (Fig. 10.13), a central core of radioisotope material is surrounded by an array of thermocouples connected in series-parallel to obtain the desired voltage and current output. The hot side of the thermocouple junction is in contact with the canister containing the radioisotope, and the cold side is in contact with the external wall of the RTG, from which heat is radiated to space. The efficiency of the RTG is ultimately limited by the conversion efficiency of the thermoelectric elements.

Modern semiconductor thermoelectric devices, such as are used for Galileo and Cassini, can deliver a conversion efficiency of 10–11%, and research continues in an effort to improve this. Other limitations involve the internal thermal conductivity of the assembly. Considerable effort goes into designing a

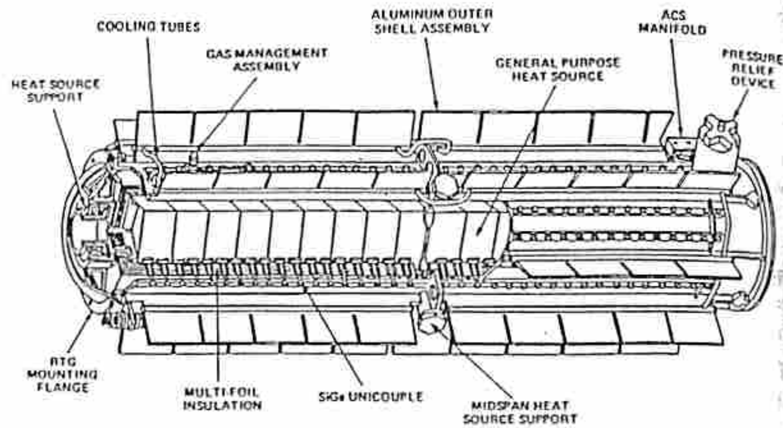

Fig. 10.13 Galileo RTG. (Courtesy of Jet Propulsion Laboratory.)

thermal path with minimum temperature drop from the isotope to the hot junction and from the cold junction to the outer case, while at the same time minimizing any energy leakage between these points that would bypass the thermoelectric conversion elements. Because the conversion efficiency of a given design depends on maximizing the temperature differential across the thermoelectric units, and the upper and lower limits are driven by material limits on the hot side and radiator size (usually) on the cold side, the importance of minimizing conductive drops or thermal leakage is obvious. These factors plus internal resistance and other losses explain why overall RTG efficiency is typically 6–7%, rather than that of the thermoelectric elements alone. All of this means, of course, that a very large amount of waste heat is produced for every unit of electrical energy produced.

The RTG used for the Galileo Jupiter Orbiter delivers 298 W ± 10% at the beginning of life from a mass of about 56 kg. The thermoelectric elements are doped silicon-germanium (Si-Ge). The radioisotope material used is plutonium-238 (^{238}Pu).

Although all radioisotopes exhibit a loss in energy output with time, the 86.7-year half-life of ^{238}Pu is not the major life-limiting mechanism of current RTGs. Degradation of the thermoelectric elements, caused mostly by dopant migration at the relatively high temperatures involved, is a more significant cause of performance loss. Breakdown of insulators because of temperature and radiation is also a factor.

Although ^{238}Pu has been most commonly used in RTGs, there are other candidates offering the combination of reasonably long half-life with high energy output that together qualify an isotope for RTG use. Table 10.4 lists some

Table 10.4 RTG material properties

Property	Po-210	Pu-238	Ce-144	Sr-90	Cm-242
Half-life, years	0.378	86.8	0.781	28.0	0.445
Watts/gram, thermal	141	0.55	25	0.93	120
$/Watt, thermal	570	3000	15	250	495

candidates. Note that, for long space missions, only strontium-90 (^{90}Sr) has a half-life adequate to be a viable candidate in addition to ^{238}Pu.

Each radioisotope has a particular form and energy of radiation that is given off in the decay process. ^{238}Pu gives off an alpha particle and a relatively low-energy beta particle, both of which are relatively easy to shield compared with the high-energy gamma radiation from ^{90}Sr. The long-term effect of RTG radiation upon electronics is definitely a factor in radioisotope selection.

By their nature, RTGs cannot be turned off in the conventional sense. That is, the radioisotope continues to decay and to generate heat regardless of any external action. Similarly, the thermoelectrics will generate electricity whenever a temperature differential exists and a load is placed across the output terminals. Because there is no practical way to control the generation of electricity at an essentially constant rate within the RTG, control must be external. In spacecraft applications control is typically accomplished by use of a shunt regulator to dispose of electrical energy in excess of the operational requirements at any given time. RTGs are usually stored in a shorted condition. This has the desirable characteristic of reducing the temperature of the RTG during storage because of the Peltier thermoelectric cooling effect.

Aside from their high cost, particularly for the radioisotope, RTGs have a variety of problems that have limited their use when other power sources will suffice. The high external temperature and the radiation from the radioisotope are a major problem in ground handling. Special equipment is required because the assembly crew cannot directly handle the units without thermal protection (e.g., heavy gloves). This complicates structural assembly and the making of electrical connections and significantly increases installation time. This in turn is problematic because it increases the exposure of the crew to radiation from the radioisotope. Such radiation can be of sufficient intensity to mandate the use of oversize work crews so that no individual exceeds allowable dose limits. It is also important to allow for contingencies. If the crew is sized to reach the exposure limit in the course of a normal installation, then there will be no one available in the event of a problem requiring the RTGs to be removed and reinstalled.

A further problem is the possibility of Earth contamination by the radioisotope in the event of a launch failure or decay from orbit. The radioisotopes in RTGs launched to date have been extensively protected against both destruction in the event of a launch failure and incineration upon reentry. The fuel itself is normally the oxide of the radioisotope and is therefore reasonably strong and resistant to high temperature by itself. The lumps of oxide are then encased in graphite for atmospheric entry and impact protection. Extensive tests are performed to qualify the fuel elements for this environment, with notable success. The aborted Apollo 13 lunar landing mission eventually resulted in reentry of the RTG fuel element at lunar return velocity, followed by impact in the Pacific Ocean. There has been no subsequent evidence of any release of radioisotope material from this event.

Similarly, most reasonable launch failure scenarios can be accommodated by the internal protection built into the RTGs. It is possible, however, to postulate a launch failure of such severity that the fuel elements will be shattered and the radioactive material scattered into the atmosphere. This may be technically (if not politically) acceptable provided the material is thoroughly dispersed in the upper atmosphere in the form of very fine particles, so that the concentration at any point on the surface will be very low when the material settles out of the atmosphere.

It must be noted that an accident of this magnitude would represent a very improbable launch failure scenario, most of which are rather benign as explosive events are judged (i.e., they are deflagration as opposed to detonation events). For example, analysis indicates that the 1986 Challenger accident would not have created a hazard due to radioisotope dispersal had RTGs been aboard. In another noteworthy case, the launch abort and subsequent destruction of a military payload carrying an RTG was followed by recovery of the intact RTG from the water off Vandenberg Air Force Base, California. The unit was reused on a subsequent mission.

These examples aside, it remains necessary to plan for the worst possible case. Extensive analysis is necessary to determine the possible hazard environment and to devise protection adequate to ensure that the radioactive material comes down in a condition that minimizes dispersion and allows for recovery. This is especially of concern for plutonium, which, besides being radioactive, is extremely toxic.

The radiation from RTGs is detrimental to spacecraft electronics and instruments, making it necessary to mount the units on booms at some distance from the body of the spacecraft, and often to provide shielding as well. It must be noted that the spacecraft configuration as stowed for launch will not allow the RTG boom to be in its deployed configuration. Therefore, the spacecraft must be able to survive whatever radiation exposure will accrue during the stowed period by means of shielding and the inherent radiation tolerance of the onboard electronics.

10.11 Fuel Cells

Fuel cells are devices that allow direct conversion of chemical energy into electricity. In this they are like batteries, with the difference that fuel cells operate much more efficiently. An oxidizer and a fuel are fed into the cell, which is roughly similar to a battery in its internal arrangement. Electricity is generated directly from the oxidation reaction within the cell, aided by the presence of a catalytic material, but without the high temperature and other complications associated with combustion. Space applications of fuel cells have been primarily to manned spaceflight, and were first used to power the later Gemini spacecraft as well as the Apollo CSM, the lunar module, and the space shuttle.

Although numerous fuel-oxidizer combinations are possible and have been used experimentally, only hydrogen and oxygen have so far been used as reactants in operational fuel cells for space applications. The output of the cells is essentially pure water, which is used for crew consumption with little or no treatment. Laboratory demonstrations have shown conversion efficiencies approaching 35%.

The overall mass of a fuel cell system is a function of the desired operating time, since the mass of the reactant must be included in the assessment. For a system of fixed mass, however, an energy density of 500 (W · h)/kg at a power level of 2.6 kW is a reasonable figure of merit. Lacking a substantial industrial production base, fuel cells remain a costly source of power, in the range of $3000/kW for commercial systems (essentially the same as for rechargeable batteries) and much higher for space-qualified systems.

Fuel cell development is in many ways still in its infancy. Fuel cells run most efficiently on pure hydrogen and oxygen, but because of the difficulty of storing and handling liquid hydrogen relative to most other fluids, there has been considerable interest in the use of other sources of hydrogen for commercial devices. Commercial fuel cell development has focused on the use of methane, methanol, ethanol, natural gas, and other sources rich in hydrogen. Catalytic conversion is necessary to render the hydrogen free for use in the fuel cell, and carbon-based impurities interfere with the desired operation of the cell. These factors have hindered the development of commercially viable production fuel cell technology.

A variant concept, the so-called direct methanol fuel cell (DMFC), offers the possibility of very small fuel cell power packs for portable applications and could be of great interest as a source of power for space suits and other applications where high energy density and small size are required. Although not as efficient as its larger brethren, the DMFC offers energy density of at least twice that of its lithium-battery competitors and can be "recharged" simply by adding methanol. The first use of DMFC power packs will undoubtedly be in commercial laptop computers, cellphones, and other consumer electronics devices. However, there is nothing in their nature that precludes use in space, and such applications can be expected to follow.

A tantalizing possibility for energy storage in large systems, such as the International Space Station, is to use regenerative fuel cells in lieu of batteries. In this scenario, fuel cells would use stored hydrogen and oxygen to generate electricity during eclipse periods. During the illuminated portion of the orbit, solar arrays would generate electricity to power the spacecraft and to recharge the fuel cells by electrolyzing the water generated during operation. The resulting hydrogen and oxygen would then be stored to provide reactant to the cells for the next eclipse period. Regenerative fuel cells have been demonstrated in the laboratory but have not so far been reduced to engineering practice.

10.12 Power Conditioning and Control

The power conditioning or processing portion of the space power system carries the responsibility for many of the functions listed earlier in this chapter. Power conditioning is necessary because the voltage from the power source may vary substantially, especially with solar arrays, which are of course the most common source of primary spacecraft power, for a variety of reasons including load variability, array temperature, and other external environmental factors.

Broadly considered, the power conditioning subsystem must fulfill three functions on a spacecraft utilizing solar arrays. First, it must control the solar array output in response to changes in load requirements and to changes in array temperature and sun angle, which as we have seen significantly alter the source properties. Second, it must control the battery charge-discharge cycle, supplying the proper charging voltage and current and regulating the average discharge voltage. Finally, the power system must regulate the voltage supplied to the remainder of the spacecraft system to the specified level (within some tolerance), thus protecting the other subsystems from the fluctuations already cited. This last requirement is obviously present even on spacecraft using RTGs, fuel cells, or any other power source. The second requirement may be relevant as well, if the peak loads exceed the steady-state source capability.

In some cases, a variety of voltage levels for different functions may be required, but at the very least, main bus regulation will be needed. This concerns the point, discussed earlier, as to whether the power conditioning system should supply most or all specific subsystem requirements, or whether it should merely be a source of stable bus voltage, allowing individual subsystem designers to deal with their own requirements.

Any electrical noise generated by the power source or the control electronics must be isolated from the main bus. The main bus in turn must be isolated from any power source faults, such as loss of part of a solar array or voltage transients due to entry into and exit from an eclipse period.

Finally, the power system and other spacecraft subsystems must be protected from faults in any other subsystem. We ignore here the issues and design trades surrounding the use, or not, of dual-bus power and spacecraft "housekeeping" systems, and other aspects of redundancy architecture and management. Some of the issues are discussed further in Chapter 12.

Although it is beyond the scope of this text to explore the details of power control circuitry, it will be useful to discuss the basic concepts at the block-diagram level. Those needing more detail are referred to other texts.[4–6]

Power control systems for spacecraft using solar arrays are broadly categorized as dissipative and nondissipative systems. In dissipative systems, as the name implies, excess power is shed resistively, while in nondissipative systems, the solar array itself is regulated through a DC-DC converter to operate at its peak-power point, according to the load demanded from it. As loads decrease, the array output is shifted toward its open-circuit high-voltage operating point, which yields the lower current that is required. Conversely, as loads increase, operation is shifted to a lower-voltage, higher-current operating point, up to the maximum power that can be delivered by the array.

Dissipative systems are also called direct energy transfer (DET) systems, because they are not in series with the array output. For this reason, they offer excellent overall efficiency and have the additional advantage of inherently simpler design, and thus lower parts count. Nondissipative systems are commonly called peak power tracking (PPT) systems, a name aptly descriptive of their operation. PPT systems are more complex and introduce some inherent inefficiency due to the requirement for a DC-DC power converter in series between the solar array and the load. However, for LEO spacecraft encountering a wide range of operating requirements, and for spacecraft needing maximum EOL array operating efficiency, PPT systems can be appropriate.

In most DET systems, a shunt regulator will be connected across the solar array, in parallel with the battery and its charge controller, and with the spacecraft loads, as shown in Fig. 10.14. As the name implies, the shunt regulator controls spacecraft power by dissipating current in excess of that required by the instantaneous load, which consists of battery charging requirements plus spacecraft operational needs. Shunt regulators are common because they are efficient and because they are simply and reliably implemented.

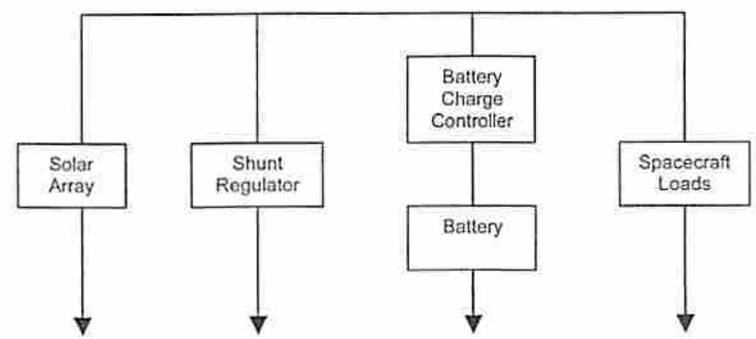

Fig. 10.14 Basic shunt regulator concept.

In the basic shunt-regulator concept, the bus voltage is unregulated and varies between the post-eclipse cold-array voltage on the charge cycle and the battery discharge voltage on the discharge cycle. As indicated earlier, this will result in considerable bus voltage variation, which may be acceptable. If unacceptable to a particular instrument or subsystem, the raw bus voltage must be further regulated within the subsystem.

Alternatively, the spacecraft bus may itself be more carefully regulated, either on the charge cycle, the discharge cycle, or both. Figure 10.15 shows a block diagram example of a shunt-regulated array with a fully regulated bus.

The shunt regulator itself can be either a simple linear controller or a switching shunt. If a switching shunt is used, the output is pulse-width modulated to produce the desired average level, a process resulting in a higher level of self-generated electromagnetic interference (EMI) than for a linear shunt. This will generally result in the requirement for additional shielding of the system and smoothing of the output power to avoid interference with other spacecraft systems.

Series regulation of solar array power to the bus is also possible. Figure 10.16 provides an example of the concept. In this case, the bus is controlled by dissipating excess power through a voltage drop in series with the load. Series regulation tends to be more complex than shunt regulation and is therefore less common, though there are advantages to providing better control of bus voltage as delivered by the solar array.

As will be obvious, more elaborate electronic control circuitry, operating at less overall efficiency, is required if the bus voltage is to be closely regulated. The engineering assessment as to whether overall mass and complexity are minimized by carefully regulating the main bus, as compared with performing the power conditioning at the subsystem level, must be made in each case. In general, larger, higher power, more complex satellites will benefit at the system level by having a central bus controller, and conversely for smaller, simpler spacecraft.

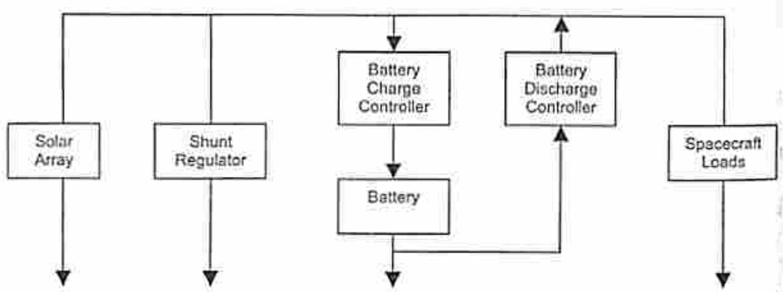

Fig. 10.15 Shunt regulator with battery charge-discharge regulation.

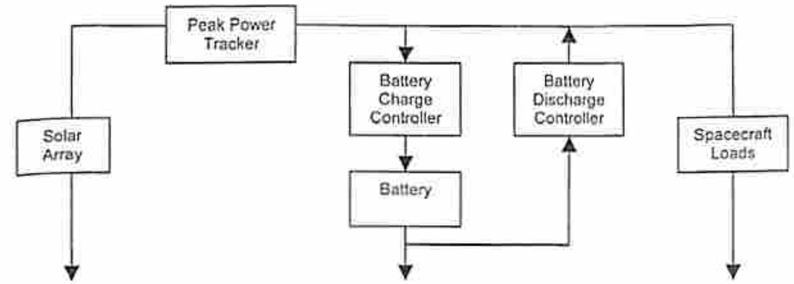

Fig. 10.16 Series regulation with peak-power tracking.

10.13 Future Concepts

10.13.1 Nuclear Reactors

Nuclear reactors offer considerable promise for the future. For very large power levels, hundreds of kilowatts to megawatts, reactors may be the only viable source in the next several decades. The nuclear reaction supplies heat, which is converted to electricity by a variety of techniques. Candidate energy conversion techniques include thermionics, thermoelectrics, and Stirling, Brayton, or Rankine cycle engines driving an alternator.

Thermoelectric energy conversion was discussed briefly in connection with RTGs. An allied concept, in the sense that it requires no moving parts, is that of thermionic energy conversion, which, however, uses a completely different concept. In thermionic conversion, heat is converted to electricity by boiling electrons from a hot emitter, or cathode, and collecting them at a cooler anode. This is exactly the mechanism at work in old-fashioned vacuum tubes, except that the heat is supplied by a nuclear reactor rather than by resistive heating of a wire filament.

Practical thermionic systems require the cathode temperature to be very hot, 1600–2000 K, while the anode must be cooler, 800–1000 K, to avoid significant back-emission of electrons. The spacing between cathode and anode must be relatively small, e.g., <1 mm. Power densities in the 100–1000 W/m^2 range can be achieved at a conversion efficiency of 10–15%. The astute reader will note that although this is higher than for thermoelectric conversion, it is only about half the intrinsic Carnot efficiency at the given cathode/anode temperature difference. A more subtle disadvantage is the tendency of the anode and cathode to expand differentially, threatening to eliminate the gap between them, which must be small but not zero. Manufacturing tolerances for thermionic converters are obviously critical. However, these disadvantages are compensated by their very high tolerance to heat and radiation, high reliability, and compactness.

All conversion concepts require radiators to reject waste heat to space. These radiators become very large for high-power units and present a major design challenge. Dynamic conversion concepts are generally much more efficient than static designs and therefore require a smaller reactor and, in some cases, a smaller radiator. Mitigating these advantages is the fact that the vibration and other disturbances typical of dynamic systems may be a problem.

Reactors have the advantage that, until they are in operation, they are not highly radioactive and can be handled with relative safety. When in operation, however, the radiation is very intense and much more damaging than that characteristic of RTGs. Heavy shielding is required even in unmanned applications, because electronic components cannot otherwise withstand the radiation from the reactor. For manned applications, the shielding and separation requirements become far more stringent. Figure 10.17 shows a typical reactor-powered spacecraft design using geometric separation to reduce shield mass. In the configuration shown, a shadow shield is used to protect only a relatively small portion of the volume of space surrounding the reactor, thus saving substantial mass. Of course, this design precludes close proximity operations outside the shield shadow. An early concept for the SP-100 nuclear reactor-based space power system was to have a mass of 3000 kg for a 100-kWe system. However, as

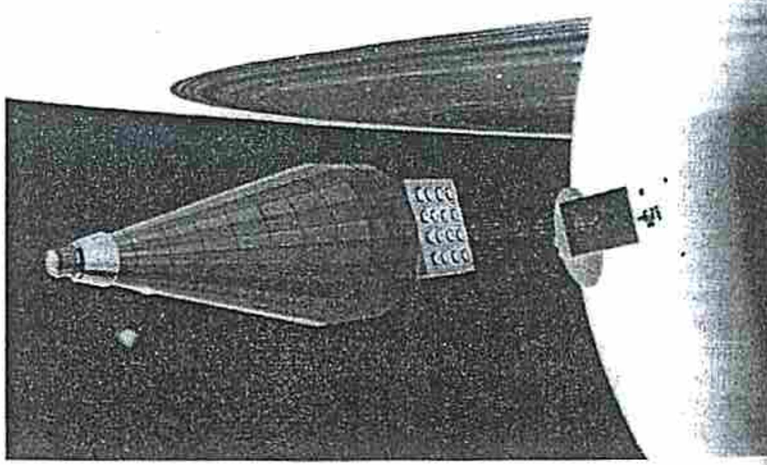

Fig. 10.17 Nuclear-electric spacecraft. (Courtesy of Jet Propulsion Laboratory.)

the SP-100 design matured, the weight essentially doubled, a factor that contributed to the program's eventual cancellation.

The mass-to-power ratio of a reactor design improves somewhat as the size increases. Essentially, reactors perform best where traditional solar power systems do poorly, such as on planetary surfaces, at great distances from the sun, and when large amounts of power are needed. Because of the severe limitations of non-nuclear alternatives, interest has once again arisen in space nuclear power options, especially in the larger power output category, but specific details are unavailable as this is written.

10.13.2 Dynamic Isotope Systems

The dynamic isotope system is a concept for obtaining more electrical power from the same isotope heat source as used for a traditional RTG. In this approach, the heat from the decaying isotope is used to heat the working fluid of a Brayton, Rankine, or Stirling cycle engine, which in turn drives an alternator. Because of the much higher conversion efficiency of these dynamic systems as compared to that of thermoelectric or thermionic systems, 500–700% more power can be obtained from a given quantity of isotope. This has advantages in reducing cost, radiation exposure, and mass. Detrimental factors include a reduction in reliability due to the added moving parts and the vibration that any dynamic system will tend to generate. Another possible disadvantage is the requirement to shed waste heat at a temperature lower than that of a typical RTG, thus requiring larger radiator area. (This comment might appear to conflict with the advantage cited earlier regarding the possibility of having smaller radiators. Indeed, either result may be true. High thermal efficiency requires a low cold-side temperature, and thus a larger radiator. However, higher intrinsic conversion efficiency by itself allows a smaller radiator. The net result depends on the particular system parameters for a given case.) Dynamic isotope conversion systems have been tested extensively but, as this is written, have not been flown.

10.13.3 AMTEC

An interesting energy conversion concept for potential future use is the alkali metal thermal-to-electric conversion (AMTEC). This device has no moving parts (if we may ignore the sodium working fluid being circulated by electromagnetic pumps) but offers potential conversion efficiencies approaching those of dynamic systems. In the AMTEC concept, sodium heated to the point of ionization by the primary energy source is applied to one side of a ceramic membrane that conducts sodium ions but not electrons. Thus, the positive sodium ions pass through, but electrons tend to accumulate. A conductive film on the membrane collects the electrons, which are then conducted through a load to the downstream side of the membrane to neutralize the sodium ions. A number of problems must be solved, including membrane life, sodium condensation management in 0g, and other materials concerns, before this intriguing concept can be considered for operational use.

10.13.4 Solar Dynamic Systems

Solar dynamic systems, as the name implies, feature machines such as Brayton, Rankine, or Stirling cycle engines driving an electrical generator or alternator and using the sun as the primary energy source. These units offer potential conversion efficiency five to seven times that of solar photovoltaic arrays, which becomes very attractive at high power levels, e.g., above 100 kW. At such levels, photovoltaic arrays are expensive and pose attitude control and atmospheric drag problems due to their large size. The reduction in area of collectors for the dynamic system greatly reduces drag and stability concerns and becomes cost-competitive as well at high power. However, this approach does carry the usual dynamic system problems of reduced reliability, possible vibration, and possible attitude control system interactions. Also, the size advantage may be partially offset by the requirement for waste heat radiators associated with these conversion concepts.

10.13.5 Radiators

We have referred on several occasions to the need for radiator surfaces to dispose of waste heat. As systems become larger, the significance of the radiator increases until, for very large systems, it may be the largest single item. Present radiator concepts utilize large thin skins, usually made of metal. The heat to be dissipated may be delivered by conduction, by a pumped fluid loop, or by an array of heat pipes. Conventional radiators are limited by such factors as the allowable material temperature, achievable surface-to-mass ratio, surface emissivity, and thermal conductivity. A variety of innovative concepts have been proposed to provide higher capability radiators, including droplet radiators, membrane radiators, and rotating band radiators.

The droplet radiator offers very high performance because of the large surface-to-volume ratio of the droplets and the possibility of allowing a liquid-to-solid phase change, thus greatly increasing the energy removal. However, several practical problems must be solved before this concept can be implemented, including droplet generation and collection (especially while maneuvering) and materials selection.

The membrane radiator achieves high efficiency by allowing a fluid to flow down the inside of a contoured rotating membrane. The resulting convective heat transfer, itself a high-efficiency heat transfer mechanism (see Chapter 9), may be enhanced by a gas-to-liquid phase change. Small punctures in the membrane can be tolerated, because surface tension in the fluid will prevent leakage. As always, the requirement to rotate may be a problem. Material selection and the

development of credible launch configurations and deployment scenarios may also present difficulties.

The rotating band radiator is simply a broad, thin continuous loop of high-temperature metal moving between heated rollers in the spacecraft, from which it is extruded out into space to reject heat, then back into the spacecraft through other rollers. Effective transfer of heat to the band is crucial to this concept. A similar approach using a rotating disk has also been suggested.

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Problems

10.1 Size a spacecraft power system consisting of a solar array and Ni-Cd batteries to supply 7.5 kW of prime power using 12% efficient 2×4 cm silicon cells with sun-tracking flat panels. The orbit is 500-km circular at 28.5 deg, i.e., approximately that of the Hubble Space Telescope. Assume a five-year life, with a minimum of 28 V required during eclipse. For the battery, assume an average discharge voltage of 1.1 V/cell and a minimum allowed discharge voltage of 1.0 V/cell. Use good design practice to determine:

(a) the number of solar cells in series and parallel.
(b) the size of the array.
(c) the number of battery cells in series and parallel.

10.2 Assume the same situation as problem 10.1, but with 18% efficient 2×4 cm Ga-As solar cells and Ni-H_2 as the battery type. For this case, we will assume a 25% allowed DOD, a minimum allowed discharge voltage of 1.1 V/cell, and an average discharge voltage of 1.3 V/cell. Determine, again using good design practice where parameters are not specified, the following:

(a) the number of solar cells in series and parallel.
(b) the size of the array.
(c) the number of battery cells in series and parallel.

10.3 A geostationary orbital spacecraft requires 10 kW of power for a nominal 10-year lifetime. The bus voltage is to be 42 V in sunlight at EOL. The solar cells to be used are 2 × 6 cm in size, and at 298 K have maximum-power operating characteristics of $V_{mp} = 0.45$ V and $I_{mp} = 0.40$ A. The radiation degradation factors over 10 years for V_{mp} and I_{mp} are 0.95 and 0.97, respectively. Specified solar panel temperature design points are 273, 285, and 340 K, respectively, for summer solstice, autumnal equinox, and post-eclipse. The temperature coefficients for the solar cell at end-of-life are $\gamma_I = 0.25$ mA/K and $\gamma_V = -2.2$ mV/K. Sun-tracking flat panels with a 90% packing factor are assumed.

(a) How many cells are required in series?
(b) How many cells are required in parallel?
(c) What is the required total solar panel area?
(d) What is the end-of-life post-eclipse power output?

10.4 An RTG power system is being designed for a 20-year mission to Pluto. EOL power required is 100 W. ^{238}Pu has been selected for the isotope, and initial thermoelectric conversion efficiency is 7%, degrading to 5% at EOL due to radiation damage to the thermoelectric elements.

(a) What is the approximate mass of the RTG?
(b) What is the required isotope mass if a 30% efficient dynamic energy conversion mechanism is used, assuming this unit does not degrade with radiation exposure?

11 Telecommunications

11.1 Introduction

Telecommunications in space differs from the earthbound version in two major respects: 1) its long range, which may be anything from a few hundred to several billion kilometers, and 2) the potentially large relative velocity between transmitter and receiver, so that Doppler shift becomes significant (± 50 kHz in the *S*-band for low Earth orbit), requiring complex frequency-tracking loops in the receiver. Also, spacecraft in low orbit see very limited communications coverage from any single surface station. A station that can track to within 5° of the horizon will view a spacecraft in a 300-km orbit for only 6.5 min, even for a zenith pass. At the opposite extreme, distant spacecraft move very slowly against the background of the fixed stars, thus, the pass time is essentially governed by the rotation of the Earth. Signals from distant spacecraft, because they are very weak, require tracking by large, specialized equipment, such as NASAs Deep Space Network (DSN).

These factors complicate spacecraft design because of the mismatch between the rates of data acquisition and return. In low Earth orbit (LEO) a spacecraft may collect data throughout the orbit period of perhaps 95 min. Given only one downlink station, the spacecraft can dump data only a few times per day. Clearly, the downlink data rate must be many times that of the acquisition rate even with onboard processing and compression of the data. Power limitations and range restrict the rate at which data can be returned from a spacecraft at another planet. Data may be acquired very rapidly during an encounter and then played back at a relatively low rate over a long period.

Moreover, passage through the Earth's troposphere and ionosphere complicates signal propagation, as a result of energy absorption, rotation of polarized signals, etc. We will examine these effects in more detail later.

Spacecraft telecommunications hardware has power, mass, and volume limitations more extreme than in other applications, even aircraft avionics. Meeting these challenges, in fact, was the original spur that has led to the technology of low-power, low-mass electronics seen in today's consumer electronics market. As discussed in Chapter 3, spacecraft electronics experiences a variety of environmental stresses, such as mechanical shock and the acoustics and vibration of launch and atmospheric flight. Spacecraft are exposed to radiation that can damage electronics over a period of time. Extremes of thermal