11.3.3 C&DH Basics

This section is a list of details of great concern to command and data handling system design and operation. Many of these concerns are of absolute necessity when determining C&DH requirements and generating procurement specifications. Emphasis is placed on the command system because of the severity of the effects if these guidelines are not followed. Data handling basics such as data rates and the number of bits per sample are covered in Sec. 13.2.

Interfaces to other equipment must be protected so that their faults do not propagate into the command decoder.

It is paramount that no commands or any transient signals appear on command outputs during application or removal of prime power, or during under/over prime power voltage conditions.

It is a basic philosophy of command decoder designs that if the integrity of a command message is in doubt, the command is not issued. It is rejected! This is especially true when firing an ordnance device or the spacecraft is launched from a manned vehicle. It is for this reason that received command messages are not corrected, although the capability exists, using error check bits.

For safety concerns, operations such as firing ordnance, an engine, or thruster, require multiple commands configured in series forming a logical AND function. No single command causes the operation to occur. In a typical ordnance application, three commands are required: safe, arm, and fire. In this case, safe and arm are relays that enable a high level discrete command, fire. The commands must (shall) be isolated within the command decoder such that no single component or physical failure results in inadvertent function execution. To achieve this, the Hamming distance of controlling command messages must be two or greater (for isolation in the decoding scheme), and command outputs must be physically isolated to the greatest extent possible using different decoding circuits and interface connectors.

It is advised not to have any commands that turn a command decoder off during flight. In addition, there should be no commands that interrupt the uplink source to the command decoder.

In redundant applications, where command outputs are cross strapped, the interface circuits and interconnection have to be designed such that no single component or physical failure prevents the active output from functioning. Along the same lines, where telemetry inputs and serial interface outputs are cross strapped, the interface circuits and interconnections have to be designed such that no single component or physical failure prevents the interface from functioning.

The rising and falling edges of discrete command and serial telemetry outputs are often limited in frequency content so that they are not a source of noise emissions on the spacecraft.

11.3.4 A Final Note

The C&DH subsystem is often one of the last on the spacecraft to be defined. It is a tool, used to configure, control, or program the payload and other spacecraft subsystems. It is the spacecraft's senses reporting internal environment, health, and status information. C&DH equipment cannot be completely defined until the requirements of other systems have been established. The mission designer's main task is that of listing the command, telemetry and other data needs for each spacecraft system. The list must also include the rate at which commands are issued and telemetry is gathered for determination of composite data rates. Issues such as data format, encoding, and

security must then be addressed. At this point it may be advantageous to stop and take an overall view of the spacecraft for other functions, which if included in the C&DH, would simplify overall design. Remember that the C&DH interfaces to nearly all spacecraft functions. Next the impact of the mission environments, duration and required reliability on the C&DH hardware is assessed. When these tasks are complete the C&DH subsystem can be fully characterized.

11.4 Power

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As illustrated in Fig. 11-8, the *electrical power subsystem (EPS)* provides, stores, distributes, and controls spacecraft electrical power. Table 11-30 lists typical functions performed by the EPS. The most important sizing requirements are the demands for average and peak electrical power and the orbital profile (inclination and altitude). We must identify the electrical power loads for mission operations at *beginning-of-life, BOL*, and *end-of-life, EOL*.

For many missions, the end-of-life power demands must be reduced to compensate for solar array performance degradation. The average electrical power needed at EOL determines the size of the power source. Section 10.3 shows a sample power budget that we may use to begin the sizing process. We usually multiply average power by 2 or 3 to obtain peak power requirements for attitude control, payload, thermal, and EPS (when charging the batteries). Fortunately, all the systems do not require peak power at the same time during the mission.

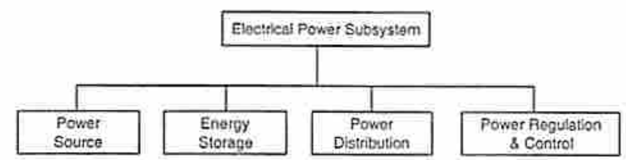

Fig. 11-8. Functional Breakdown for the Spacecraft's Power Subsystem. We start with these four functions and must determine requirements for the hardware, software, and interfaces for each.

TABLE 11-30. Typical Top-Level Power Subsystem Functions. Each of these functions consists of subfunctions with a myriad design characteristics which we must develop to meet mission requirements.

- Supply a continuous source of electrical power to spacecraft loads during the mission life.
- Control and distribute electrical power to the spacecraft.
- Support power requirements for average and peak electrical load.
- Provide converters for ac and regulated dc power buses, if required.
- Provide command and telemetry capability for EPS health and status, as well as control by ground station or an autonomous system.
- Protect the spacecraft payload against failures within the EPS.
- Suppress transient bus voltages and protect against bus faults.
- Provide ability to fire ordnance, if required.

Table 11-31 summarizes the power subsystem design process, which we discuss further in the following subsections, and Table 11-32 shows the principal effects of mission requirements on the power system design. We will work through the design process, beginning with the selection of a power source.

TABLE 11-31. The Preliminary Design Process for the Power Subsystem. All of these design steps must link back to mission requirements to satisfy the owner and users. Note that derived requirements may impact previous design decisions and force designers to iterate the design process.

Step	Information Required	Derived Requirements	References
1. Identify Requirements	Top-level requirements, mission type (LEO, GEO), spacecraft configuration, mission life, payload definition	Design requirements, spacecraft electrical power profile (average and peak)	Secs. 10.1, 10.2
2. Select and Size Power Source	Mission type, spacecraft configuration, average load requirements for electrical power	EOL power requirement, type of solar cell, mass and area of solar array, solar array configuration (2-axis tracking panel, body-mounted)	Secs. 10.1, 10.2 Table 10-9 Sec. 11.4.1 Table 11-34
3. Select and Size Energy Storage	Mission orbital parameters, average and peak load requirements for electrical power	Eclipse and load-leveling energy storage requirement (battery capacity requirement), battery mass and volume, battery type	Sec. 11.4.2 Tables 11-3, 11-4, 11-38, 11-39, 11-40 Fig. 11-11
4. Identify Power Regulation and Control	Power-source selection, mission life, requirements for regulating mission load, and thermal-control requirements	Peak-power tracker or direct-energy-transfer system, thermal-control requirements, bus-voltage quality, power control algorithms	Sec. 11.4.4

TABLE 11-32. Effects of System-Level Parameters on the Power Subsystem. Most aspects of the mission affect the power subsystem because so many other subsystems require specific power attributes.

Parameter	Effects on Design
Average Electrical Power Requirement	Sizes the power-generation system (e.g., number of solar cells, primary battery size) and possibly the energy-storage system given the eclipse period and depth of discharge
Peak Electrical Power Required	Sizes the energy-storage system (e.g., number of batteries, capacitor bank size) and the power-processing and distribution equipment
Mission Life	Longer mission life (> 7 yr) implies extra redundancy design, independent battery charging, larger capacity batteries, and larger arrays
Orbital Parameters	Defines incident solar energy, eclipse/Sun periods, and radiation environment
Spacecraft Configuration	Spinner typically implies body-mounted solar cells; 3-axis stabilized typically implies body-fixed and deployable solar panels

11.4.1 Power Sources

The power source generates electrical power within the spacecraft. Launch vehicles such as Titan IV or Delta use *primary batteries* (discussed in Sec. 11.4.2) as the power source for electrical loads because the batteries usually need to last less than an hour. But batteries alone are too massive for missions that last from weeks to years. These missions need a source that can generate power over many orbital cycles to support electrical loads and recharge the batteries.

Typically, we use four types of power sources for spacecraft. *Photovoltaic* solar cells, the most common power source for Earth-orbiting spacecraft, convert incident solar radiation directly to electrical energy. *Static power* sources use a heat source—typically plutonium-238 or uranium-235 (nuclear reactor), for direct thermal-to-electric conversion. *Dynamic* power sources also use a heat source—typically concentrated solar radiation, plutonium-238, or enriched uranium—to produce electrical power using the Brayton, Stirling, or Rankine cycles. The fourth power source is fuel cells, used on manned space missions such as Gemini, Apollo, SkyLab, and the Space Shuttle. Table 11-33 provides a comparison of various power sources.

Static power conversion uses either a thermoelectric or a thermionic concept. The most common static power source for spacecraft is the *thermoelectric couple*. This basic converter uses the temperature gradient between the p-n junction of individual thermoelectric cells connected in a series-parallel arrangement to provide the desired dc electrical output from each converter. This temperature gradient comes from slow decay of the radioactive source. The thermal-to-electric conversion efficiency for a thermoelectric source is typically 5–8%.

Thermionic energy conversion produces electricity through a hot electrode (*emitter*) facing a cooler electrode (*collector*) inside a sealed enclosure that typically contains an ionized gas. Electrons emitted from the hot emitter flow across the interelectrode gap to the cooler collector. There they condense and return to the emitter through the electrical load connected externally between the collector and the emitter. We choose the collector and emitter temperatures for best overall system performance. In choosing the collector temperature, we try to decrease the weight and size of thermal radiators, and we choose materials based on mission life requirements. Thermionic power sources usually rely on a reactor heat source because of the high temperature required for efficient thermionic conversion. Power efficiencies for a thermionic power conversion are typically 10–20%.

In contrast to static sources, *dynamic power sources* use a heat source and a heat exchanger to drive an engine in a thermodynamic power cycle. The heat source can be concentrated solar energy, radioisotopes, or a controlled nuclear-fission reaction. Heat from the source transfers to a working fluid, which drives an energy-conversion heat engine. For a dynamic solar-power source, *the balance of energy* remains as latent and sensible heat in a heat exchanger (molten eutectic salt), which provides continuous energy to the thermodynamic cycle during eclipse periods. A dynamic power source using a nuclear reactor or plutonium-238 decay does not require thermal-energy storage because the source provides continuous heat.

Dynamic power sources use one of three methods to generate electrical power: Stirling cycle, Rankine cycle, or Brayton cycle. *Stirling-cycle engines* use a single-phase working fluid as the working medium. The thermodynamic cycle consists of two isothermal processes (compression and expansion) and two constant-volume processes (heating and cooling). Power-conversion efficiencies for Stirling engines are 25–30%. *Rankine-cycle engines* are dynamic devices that use a two-phase fluid system

TABLE 11-33. Matrix for Comparing Most Common Spacecraft Power Sources. We may use different factors to select the correct power source but specific power and specific cost are used extensively.

EPS Design Parameters	Solar Photovoltaic	Solar Thermal Dynamic	Radio-isotope	Nuclear Reactor	Fuel Cell
Power Range (kW)	0.2–300	5–300	0.2–10	5–300	0.2–50
Specific Power (W/kg)	25–200	9–15	5–20	2–40	275
Specific Cost ($/W)	800–3,000	1,000–2,000	16K–200K	400K–700K	Insufficient Data
Hardness					
– Natural Radiation	Low–Medium	High	Very high	Very high	High
– Nuclear Threat	Medium	High	Very high	Very high	High
– Laser Threat	Medium	High	Very high	Very high	High
– Pellets	Low	Medium	Very high	Very high	Medium
Stability and Maneuverability	Low	Medium	High	High	High
Low-orbit Drag	High	High	Low	Medium (due to radiator)	Low
Degradation Over Life	Medium	Medium	Low	Low	Low
Storage Required for Solar Eclipse	Yes	Yes	No	No	No
Sensitivity to Sun Angle	Medium	High	None	None	None
Sensitivity to Spacecraft Shadowing	Low (with bypass diodes)	High	None	None	None
Obstruction of Spacecraft Viewing	High	High	Low	Medium (due to radiator)	None
Fuel Availability	Unlimited	Unlimited	Very low	Very low	Medium
Safety Analysis Reporting	Minimal	Minimal	Routine	Extensive	Routine
IR Signature	Low	Medium	Medium	High	Medium
Principal Applications	Earth-orbiting spacecraft	Interplanetary, Earth-orbiting spacecraft	Inter-planetary	Inter-planetary	Inter-planetary

employing a boiler, turbine, alternator, condenser, and pump. This power-conversion cycle is essentially the same as that used to generate electricity from fossil and nuclear energy on Earth. Power-conversion efficiencies for Rankine-cycle engines are 15–20%. *Brayton-cycle engines* are dynamic devices that use a single, compressible working fluid as the working medium. The thermodynamic cycle consists of adiabatic compression and expansion stages separated and coupled by stages that add or reject heat at constant pressure. Placed after the turbine, a recuperator-heat exchanger improves the cycle's efficiency. Power conversion efficiencies for the Brayton cycle are 20–35%.

Fuel cells convert the chemical energy of an oxidation reaction to electricity. They are self-contained generators that operate continuously without sunlight, but must carry their own reactant supply, usually. The longer the mission, the larger the reactant tanks. The most popular version for space applications is the hydrogen-oxygen (referred to as "alkaline" because of the KOH electrolyte) fuel cell because of its relatively high specific power (275 W/kg on the Space Shuttle), low reactant mass (hydrogen and oxygen), and useful by-product (water).

A typical single cell produces a voltage of 0.8 Vdc. In combination, a fuel cell unit can create many kilowatts of power (each Shuttle fuel cell produces 16 kW peak or 12 kW continuous). The energy conversion efficiency can run as high as 80% for low current draws, but as current increases, the efficiency drops to 50–60%, due to activation overpotential and electrical resistance in the electrolyte solution between electrodes. However, compared with other power sources, fuel cell efficiencies are high.

The three Space Shuttle fuel cells are state-of-the-art power generators that produce all of the Shuttle electricity for the 28 Vdc bus. Their high efficiency (70%), low weight (118 kg), and excellent reliability (> 99% available) attest to their quality. Other important factors are their 15-min start-up time, instantaneous shutdown, and long lifetime (2,400 hours before refurbishment). Besides electricity, these fuel cells produce crew drinking water, at a rate of 0.36 kg/kWh, or about 104 kg a day.*

Research is underway to solve the short-mission limit with fuel cells, caused by carrying large reactant masses. Because the fuel-cell reaction is reversible, we can use electrolysis to create more reactants from the water by-product. To optimize each process, however, we have to use separate units for generating electricity and separating the water. Any long-duration mission could use this *regenerative* system if it had some input electricity from solar cells, nuclear generators, or other power system during periods of low electrical load.

Earth-orbiting spacecraft at low-Earth to geosynchronous orbits have usually employed photovoltaics as their power source. Often, photovoltaics were the only real candidate for these low-power missions (less than 15 kW) because solar cells were well-known and reliable. Photovoltaic sources are not attractive for interplanetary missions to the outer planets because solar radiation decreases, thus reducing the available energy from a solar array. To configure and size a solar array, we must understand cell types and characteristics; solar-array design issues, types, sizing calculations, configurations, regulation; and radiation and thermal environments. Key design issues for solar arrays include spacecraft configuration, required power level (peak and average), operating temperatures, shadowing, radiation environment, illumination or orientation, mission life, mass and area, cost, and risk. Table 11-34 shows the solar array design process.

Step 1. Mission life and the average power requirement are the two key design considerations in sizing the solar array for most spacecraft. We size a photovoltaic system to meet power requirements at EOL, with the resulting solar array often oversized for power requirements at BOL. This excess power at BOL requires coordinated systems engineering to avoid thermal problems. The longer the mission life, the larger the difference between power requirements at EOL and BOL. We usually consider photovoltaics a poor power source for missions lasting more than 10 years because of natural degradation in the solar array. Section 11.4.4 discusses how we manage excess power from the solar array. The average power requirement can be obtained from Secs. 10.1 and 10.2.

* Telephone conversation with Jay Garrows, International Fuel Cells, Inc., Oct. 98.

TABLE 11-34. Solar Array Design Process. In the FireSat example column, I_d represents inherent degradation, θ is the Sun incidence angle, L_d is life degradation, and X_e and X_d represent the efficiencies of the power distribution paths. The material following the table further explains these quantities.

Step	Reference	FireSat Example
1. Determine requirements and constraints for power subsystem solar array design		
• Average power required during daylight and eclipse	Input parameter, Secs. 10.1, 10.2	110 W during daylight and eclipse
• Orbit altitude and eclipse duration	Input parameter, end papers	700 km 35.3 min
• Design lifetime	Chaps. 2, 3	5 yr
2. Calculate amount of power that must be produced by the solar arrays, P_{sa}	Step 1 Eq. 5-5, end papers (Orbit period – T_e) Eq. 11-5	$P_e = P_d = 110$ W $T_e = 35.3$ min $T_d = 63.5$ min Assume a peak power tracking regulation scheme with $X_e = 0.6$ and $X_d = 0.8$ $P_{sa} = 239.4$ W
3. Select type of solar cell and estimate power output, P_o, with the Sun normal to the surface of the cells	*Si: $P_o = 0.148 \times 1{,}367$ W/m^2 = 202 W/m^2 *GaAs: $P_o = 0.185 \times 1{,}367$ W/m^2 = 253 W/m^2 *Multijunction: $P_o = 0.22 \times 1{,}367$ W/m^2 = 301 W/m^2	Si solar cells $P_o = 202$ W/m^2
4. Determine the beginning-of-life (BOL) power production capability, P_{BOL}, per unit area of the array	Table 11-35 Eq. 5-7 Eq. 11-6	$I_d = 0.77$ $\theta = 23.5$ deg (worst case) $P_{BOL} = 143$ W/m^2
5. Determine the end-of-life (EOL) power production capability, P_{EOL}, for the solar array	Performance degradation Si: 3.75% per yr, GaAs: 2.75% per yr, Multijunction: 0.5% per yr Eq. 11-7 Eq. 11-8	Performance degradation is 3.75% per year $L_d = 0.826$ for 5 yr mission $P_{EOL} = 118.1$ W/m^2
6. Estimate the solar array area, A_{sa}, required to produce the necessary power, P_{sa}, based on P_{EOL} an alternate approach	Eq. 11-9 Eq. 10-12†	$A_{sa} = 2.0$ m^2 $A_{sa} = 2.5$ m^2
7. Estimate the mass of the solar array	Eq. 10-13†	$M_a = 9.6$ kg
8. Document assumptions		

* Typical demonstrated efficiencies for Si, GaAs, and multijunction solar cells are 14.8%, 18.5%, and 22%, respectively.
† Use P_{sa} in these equations.

In designing a solar array, we trade off mass, area, cost, and risk. Silicon presently costs the least for most photovoltaic power applications, but it often requires larger area arrays and more mass than the more costly gallium-arsenide cells. Programs for which mass and volume (solar array area) are critical issues may allow higher costs or technical risks. They could select a system based on gallium arsenide or some other advanced type of solar cell. Risk develops from the unproven reliability and fabrication of the photovoltaic source.

A solar array's illumination intensity depends on orbital parameters such as the Sun incidence angles, eclipse periods, solar distance, and concentration of solar energy. Tracking and pointing mechanisms on the solar array often adjust for these influences. If we mount the cells on the body of the spacecraft, we must orient them so they will generate adequate power throughout the mission.

Step 2. To estimate the solar-array area required for a spacecraft, we first determine how much power, P_{sa}, the solar array must provide **during daylight** to power the spacecraft for the entire orbit

$$P_{sa} = \frac{\left(\frac{P_e T_e}{X_e} + \frac{P_d T_d}{X_d}\right)}{T_d} \tag{11-5}$$

where P_e and P_d are the spacecraft's power requirements (excluding regulation and battery charging losses) during eclipse and daylight, respectively, and T_e and T_d are the lengths of these periods per orbit. The terms X_e and X_d represent the efficiency of the paths from the solar arrays through the batteries to the individual loads and the path directly from the arrays to the loads, respectively. The efficiency values for eclipse and daylight depend on the type of power regulation: direct energy transfer or peak-power tracking. (A description of these methods follows in Sec. 11.4.4.) For direct energy transfer, the efficiencies are about $X_e = 0.65$ and $X_d = 0.85$; for peak-power tracking they are $X_e = 0.60$ and $X_d = 0.80$. The efficiencies of the former are about 5% to 7% greater than the latter because peak-power tracking requires a power converter between the arrays and the loads.

Step 3. Table 11-35 shows the efficiencies and radiation-degradation sensitivities of three main types of cells. Gallium arsenide has the advantage of higher efficiencies, whereas indium phosphide reduces the degrading effects of radiation. Silicon solar cell technology is mature and has the advantage of lower cost per watt for most applications. Gallium arsenide and indium phosphide cost about 3 times more than silicon.

The *energy-conversion efficiency* of a solar cell is defined as the power output divided by the power input. The power input value for a planar solar array is the solar-illumination intensity (1,367 W/m^2). Thus, a solar panel with a BOL efficiency of 18% will provide 246 W/m^2. We must be aware that reported efficiency values for solar panels often apply only to single cells. We need to identify losses inherent to panel assembly (diodes, interconnect cabling, transmission losses) to size the array adequately. We also need to note that these efficiency values often refer to **laboratory** cells and not **production** cells, which have lower average efficiencies.

To complete this step, we identify the type of solar cells and how their performance will degrade during the mission. Ideally, silicon and gallium arsenide solar cells have efficiencies of about 14.8% and 18.5%. These solar cell efficiencies give us **ideal** solar cell output performance per unit area, P_o, of 202 W/m^2 and 253 W/m^2, respectively, if the incident solar radiation (1,367 W/m^2) is normal to the surface.

TABLE 11-35. **Performance Comparison for Photovoltaic Solar Cells.** Note that the stated efficiencies are for single solar cells, not solar arrays.

Cell Type	Silicon	Thin Sheet Amorphous Si	Gallium Arsenide	Indium Phosphide	Multijunction GaInP/GaAs
Planar cell theoretical efficiency	20.8%	12.0%	23.5%	22.8%	25.8%
Achieved efficiency: Production Best laboratory	14.8% 20.8%	5.0% 10%	18.5% 21.8%	18% 19.9%	22.0% 25.7%
Equivalent time in geosynchronous orbit for 15% degradation – 1 MeV electrons – 10 MeV protons	10 yr 4 yr	10 yr 4 yr	33 yr 6 yr	155 yr 89 yr	33 yr 6 yr

Step 4. Next, we must determine the **realistic** power production capability of the manufactured solar array. As shown in Table 11-36, an assembled solar array is less efficient than single cells due to design inefficiencies, shadowing and temperature variations, collectively referred to as *inherent degradation*, I_d. Solar cells are applied to a substrate, usually honeycomb aluminum, and interconnected, resulting in losses of 10% of the solar array's substrate area. This accounts for the design and assembly losses. If we configure the spacecraft well, its appendages will shadow few cells, and shadowing losses should be slight. The temperature of a typical flat solar panel receiving normal incident radiation ranges from about 67 °C in LEO to 53 °C in GEO. The reference temperature for silicon solar cells is 28 °C, with performance falling off 0.5% per degree above 28 °C. Body-mounted arrays on nonspinning spacecraft are typically about 5 °C warmer than deployed solar arrays because they can't radiate heat into deep space as efficiently.

TABLE 11-36. **Elements of Inherent Solar Array Degradation.** Although individual solar cells may have adequate efficiency, after we manufacture the solar array, these elements cause some degradation in the cumulative efficiency by the amounts indicated.

Elements of Inherent Degradation	Nominal	Range
Design and Assembly	0.85	0.77–0.90
Temperature of Array	0.85	0.80–0.98
Shadowing of Cells	1.00	0.80–1.00
Inherent Degradation, I_d	0.77	0.49–0.88

As mentioned earlier, we commonly refer to the current-voltage characteristics of a solar cell as the I-V curves. Figure 11-9 depicts a first-quadrant I-V curve for a planar array in LEO. This curve characterizes BOL and EOL performance. As the figure illustrates, the three significant points for solar-array design are:

- *Short-circuit current*, I_{sc}, where voltage = 0
- *Peak-power point*, where voltage times current is maximized
- *Open-circuit voltage*, V_{oc}, where current = 0

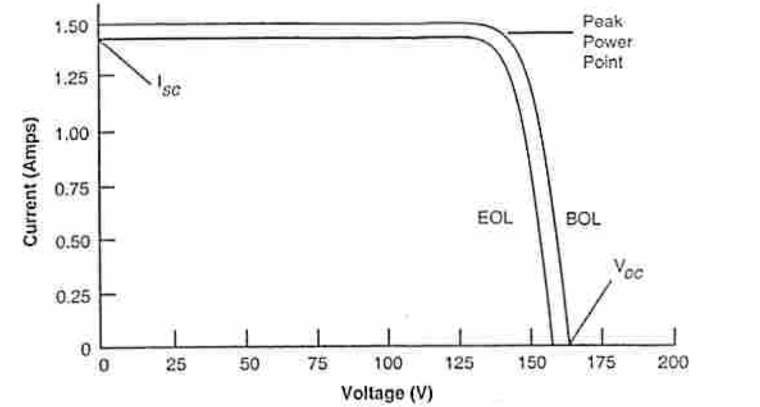

Fig. 11-9. **I-V Plot for a Planar Array.** The power available is simply the area under the curve.

We must also consider how temperature affects the I-V characteristics. While the spacecraft is in eclipse, the solar-array temperature can get as low as –80 °C. The highest operating temperature for an LEO spacecraft solar array is 100 °C, occurring near the end of a full Sun period during an orbit.

The operating temperature of the array is a key issue because the solar cell's performance depends on temperature. A *current-voltage*, or *I-V plot*, illustrates the performance of a solar-array cell, or the array (see Fig. 11-9). A change in the operating temperature of the solar cell or array causes three changes in the I-V curve:

- A scaling of the I-V curve along the current axis
- A translation or shifting of the I-V curve along the voltage axis
- A change in the I-V curve shape affecting the roundness of the knee region

The *temperature coefficient*, or percent degradation of performance with increasing temperature, for any solar cell depends on factors such as the type of cell and its output-power characteristics, actual operating temperature, and radiation environment. Gallium arsenide and indium phosphide have lower temperature coefficients, but higher temperature still means reduced performance. Solar arrays using gallium arsenide and indium phosphide also resist radiation better than silicon and provide greater EOL power for a given area. We must establish a profile for operating temperatures during a mission, so the photovoltaic system can generate adequate power throughout.

The peak-power point depends on the array's operating temperature at BOL and EOL. Thus, an array often provides maximum power coming out of an eclipse period because it is at its coldest operating temperature. Transient voltage excursions often occur when leaving eclipse, so we may need to clamp voltages to protect spacecraft loads. By understanding how the array's performance relates to these variables, we can get the highest output power from the array.

Usually, solar cells in series-parallel combinations make up a *solar array*. The number of series-connected solar cells in one string establishes the bus voltage required at EOL at the operating temperature; the number of parallel strings depends on the required current output. Isolation diodes mounted within the solar array typically minimize the effects of shadowing and reversed-biased solar cells.

Solar-array configurations are either planar or concentrator, and either type can be body- or panel-mounted. Most photovoltaic applications to date have employed a *planar array* in which solar cells are mounted onto a surface (typically insulated aluminum honeycomb) with an adhesive. A Kapton, Kevlar, or fiberglass sheet usually insulates the solar cell from the aluminum honeycomb support structure. *Concentrator solar arrays* increase the solar cell's output by using mirrors or lenses to focus more solar radiation on the cells.

Panel-mounted solar arrays usually apply only to 3-axis stabilized spacecraft. The panel-mounted approach tracks and points the solar array to get the best Sun incidence angle. The body-mounted approach reduces the requirements for tracking and pointing on any spacecraft (spinning or stabilized). But the less effective Sun incidence angle and increased array temperature of body-mounted cells produce a lower efficiency in orbit. Panel-mounted solar arrays are usually mounted on a boom. Deployable panel arrays are either flexible or rigid, according to the type of substrate material employed for mounting. For most spacecraft, we try to place the solar array away from the payload and other spacecraft subsystems because of the variable and often high temperature of the solar cells.

Body-mounted planar cells are typical on spinning spacecraft, which provide thermal control by radiating excess heat to space as the spacecraft spins. Body-mounted solar arrays use cells inefficiently because of higher temperature and reduced voltage. Thus, they generate lower power per unit area than a deployed, oriented panel. When solar cells are body-mounted to a spinning spacecraft, the array's total output power decreases because the cells are not always oriented toward the Sun. This decrease depends on the spacecraft's configuration and the drive mechanisms of the solar array (if any). For example, a stabilized array using Sun-tracking and pointing on two axes would fully use the solar array's surface area. But the array's reduction in output power per total surface area would be approximately π for body-mounted cells on a cylindrical, spinning spacecraft and 4 for body-mounted cells on a cubic-shaped spacecraft that does not employ active tracking. The output power decreases because not all cells are illuminated. We must trade the cost and design for the solar array's total surface area against the cost and complexity of stabilizing the spacecraft and using a drive system for the solar array.

Shadowing considerations are important because a solar cell will go into open circuit (become high resistance) when not illuminated. In a series-connected string of solar cells, the shadowing of one cell results in the loss of the entire string. Shadowing may be caused by spacecraft components such as transmitting or receiving antennas, deployment mechanisms, or structures such as the solar-array. We can reduce shadowing effects by actively pointing and tracking solar arrays on 3-axis stabilized spacecraft, using diodes, or designing series-parallel arrays. On spinning spacecraft, we must lay out solar cells so all solar cells within a string are illuminated. Diodes, which bypass groups of solar cells in a string, help prevent damage to reduce the advance effects of shadowed solar cells.

We can improve solar cell performance with coverslides, coatings, and back-surface reflectors. Coverslides provide a hermetic seal yet allow the cell to receive sunlight and reject heat. They are textured or smooth. A textured coverslide is used for body-mounted solar cells that do not actively point toward the Sun. It reflects incident

solar energy back onto the solar cell, improving the overall efficiency. Smooth coverslides are used for spacecraft whose arrays actively track and point. By decreasing reflective losses on solar cells, coatings allow cells to use more of the incident energy. Back-surface reflectors direct incident solar radiation that passes through the solar cell back through the cell again to improve overall efficiency. By reducing solar absorptance, they help the solar array manage thermal energy. Solar-cell vendors are continually improving the mechanical and thermal characteristics of coverslides, coatings, and back-surface reflectors. Thus, we must coordinate mechanical and thermal characteristics of these cells with the vendors.

At beginning-of-life, the array's power per unit area is

$$P_{BOL} = P_o I_d \cos\theta \tag{11-6}$$

where $\cos\theta$ is referred to as the *cosine loss*. We measure the *Sun incidence angle, θ,* between the vector normal to the surface of the array and the Sun line. So if the Sun's rays are perpendicular to the solar array's surface, we get maximum power. Obviously, the geometry between the array and the Sun changes throughout the mission and different solar array panels will have different geometry. We configure the solar array to minimize this cosine loss. For example, in GEO with a flat, silicon solar array and a worst-case Sun angle of $\theta = 23.5$ deg angle between equatorial and ecliptic planes and the nominal value of I_d, the power output at beginning-of-life is 143 W/m^2.

Step 5. Radiation damage severely reduces a solar array's output voltage and current. At geosynchronous altitude, we must guard against solar-flare protons on-station, trapped electrons on-station, and trapped electrons and protons during transfer orbits. (Chapter 8 explains these terms.) Electrons and protons trapped in the Earth's magnetic field cause most degradation of solar cells. Silicon solar cells protected by coverslides lose 15% of their voltage and current (shielding assumed) when exposed to a total fluence of 10^{15} MeV equivalent electrons (4 to 5 years for a LEO spacecraft). As mission planners, we should coordinate degradation characteristics with the solar-cell manufacturer, based on the radiation environment the spacecraft will encounter. Degradation of a solar cell also depends on its design. Advanced technologies, such as indium phosphide cells, are more radiation hardened.

Next, we must consider the factors that degrade the solar array's performance during the mission. *Life degradation*, L_d, occurs because of thermal cycling in and out of eclipses, micrometeoroid strikes, plume impingement from thrusters, and material outgassing for the duration of the mission. In general, for a silicon solar array in LEO, power production can decrease by as much as 3.75% per year, of which up to 2.5% per year is due to radiation. For gallium-arsenide cells in LEO, the degradation is about 2.75% per year, of which radiation causes 1.5% per year. The actual lifetime degradation can be estimated using

$$L_d = (1 - degradation/yr)^{satellite\ life} \tag{11-7}$$

The array's performance per unit area at end-of-life is

$$P_{EOL} = P_{BOL} L_d \tag{11-8}$$

Using the FireSat example array in Table 11-34 for a 5-year mission, L_d is 82.6%, resulting in a P_{EOL} of 118.1 W/m^2. The solar-array area, A_{sa}, required to support the spacecraft's power requirement, P_{sa}, is

$$A_{sa} = P_{sa} / P_{EOL} \tag{11-9}$$

The resulting solar-array area for the example spacecraft is about 2.0 m^2. If we had used a perfectly pointed array, the BOL power would have been 155 W/m^2, resulting in an EOL power of 128 W/m^2 and an array area of 1.9 m^2. So, having to account for the cosine loss costs us 0.1 m^2 in array size and the equivalent mass.

Solar-array sizing is more difficult than it appears from the above discussion. Typically, we must consider several arrays with varying geometry. Also, the angle of incidence on the array surface is constantly changing. We must predict that angle continuously or at least determine the worst-case angle to develop an estimate of P_{EOL}.

11.4.2 Energy Storage

Energy storage is an integral part of the spacecraft's electrical-power subsystem providing all the power for short missions (< 1 week) or back-up power for longer missions (> 1 week). Any spacecraft that uses photovoltaics or solar thermal dynamics as a power source requires a system to store energy for peak-power demands and eclipse periods. Energy storage typically occurs in a battery, although systems such as flywheels and fuel cells have been considered for various spacecraft.

A *battery* consists of individual cells connected in series. The number of cells required is determined by the bus-voltage. The amount of energy stored within the battery is the *ampere-hour capacity* or *watt-hour* (ampere-hour times operating voltage) *capacity*. The design or nameplate capacity of the battery derives from the energy-storage requirements. Batteries can be connected in series to increase the voltage or in parallel to increase this current output—the net result being an increase in watt-hour capacity.

Table 11-37 lists issues to consider early in the conceptual phase of any program. Most of all, we try to provide a stable voltage for all operating conditions during the mission life because load users prefer a semi-regulated bus voltage. The difference in energy-storage voltage between end of charge and end of discharge often determines the range of this bus voltage.

TABLE 11-37. Issues in Designing the Energy Storage Capability. Energy storage usually means large batteries and we must consider all their characteristics when designing this subsystem.

Physical	Size, weight, configuration, operating position, static and dynamic environments
Electrical	Voltage, current loading, duty cycles, number of duty cycles, activation time and storage time, and limits on depth-of-discharge
Programmatic	Cost, shelf and cycle life, mission, reliability, maintainability, and produceability

Figure 11-10 highlights the charge-discharge characteristics of a spacecraft's energy-storage system. We want a flat discharge curve that extends through most of the capacity and little overcharge. Overcharging quickly degrades most batteries. We also need to match the electrical characteristics of the battery cells. Otherwise, charge imbalances may stress and degrade the batteries, resulting in a shorter life for the electrical-power subsystem.

All battery cells are either primary or secondary. *Primary battery* cells convert chemical energy into electrical energy but cannot reverse this conversion, so they cannot be recharged. Primary batteries typically apply to short missions (less than one day) or to long-term tasks such as memory backup, which use very little power. The

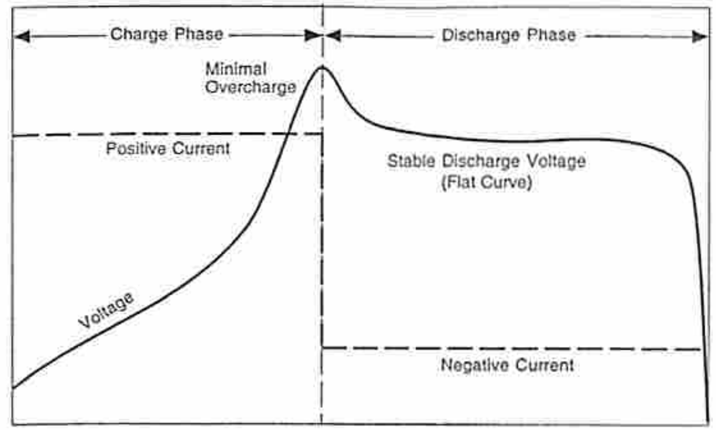

Fig. 11-10. Profile of Charge/Discharge Voltages for Batteries. Secondary batteries may cycle through this type of profile hundreds or thousands of times during their mission life. At the left edge, the voltage is low because the spacecraft just came out of eclipse where it used battery power. During the charge phase, there is positive current from the power regulator, so the battery voltage rises. In the discharge phase (in eclipse again), there is a negative current, so the battery voltage decreases.

most common batteries use silver zinc, lithium thionyl chloride, lithium sulfur dioxide, lithium monoflouride, and thermal cells. Table 11-38 highlights the applications and relative merits. It also depicts the wide ranges in each couple's specific-energy density. We cannot specify a value for specific-energy density because cells vary in design and depend on mission requirements. We must coordinate mission requirements with the battery manufacturer to specify battery performance.

TABLE 11-38. Characteristics of Selected Primary Batteries. Primary battery manufacturers can meet power requirements within these ranges of specific energy density. We must tradeoff cost and mass with capacity while ensuring mission accomplishment.

Primary Battery Couple	Specific Energy Density (W·hr/kg)	Typical Application
Silver Zinc	60 – 130	High rate, short life (minutes)
Lithium Thionyl Chloride	175 – 440	Medium rate, moderate life (< 4 hours)
Lithium Sulfur Dioxide	130 – 350	Low/medium rate, long life (days)
Lithium Monoflouride	130 – 350	Low rate, long life (months)
Thermal	90 – 200	High rate, very short life (minutes)

A *secondary battery* for energy storage can convert chemical energy into electrical energy during discharge and electrical energy into chemical energy during charge. It can repeat this process for thousands of cycles. Table 11-39 shows ranges of specific-energy density for common secondary batteries. A secondary battery provides power during eclipse periods on spacecraft that employ photovoltaics and can also level loads. Secondary batteries recharge in sunlight and discharge during eclipse. The spacecraft's orbital parameters, especially altitude, determine the number of charge/discharge cycles the batteries has to support during the mission life. A geosynchronous satellite needs to store energy for two 45-day eclipse periods per year with eclipses lasting no more than 72 min each day. The geosynchronous orbit demands few charge/discharge cycles during eclipse periods, thus allowing a fairly high (50%) depth-of-discharge. On the other hand, LEO spacecraft encounter at most one eclipse period each orbit or about 15 eclipse periods per day, with maximum shadowing of approximately 36 min. Therefore, the batteries must charge and discharge about 5,000 times each year, and the average depth-of-discharge is only 15–25%—much lower than for geosynchronous spacecraft.

TABLE 11-39. Characteristics of Selected Secondary Batteries. Though secondary batteries have much lower specific energy densities than primary batteries, their ability to be recharged makes them ideal for backup power on spacecraft powered by solar cells.

Secondary Battery Couple	Specific Energy Density (W·hr/kg)	Status
Nickel-Cadmium	25 – 30	Space-qualified, extensive database
Nickel-Hydrogen (individual pressure vessel design)	35 – 43	Space-qualified, good database
Nickel-Hydrogen (common pressure vessel design)	40 – 56	Space-qualified for GEO and planetary
Nickel-Hydrogen (single pressure vessel design)	43 – 57	Space-qualified
Lithium-Ion ($LiSO_2$, $LiCF$, $LiSOCl_2$)	70 – 110	Under development
Sodium-Sulfur	140 – 210	Under development

Depth-of-discharge (DOD) is simply the percent of total battery capacity removed during a discharge period. Higher percentages imply shorter cycle life as shown in Fig. 11-11. Once we know the number of cycles and the average depth of discharge, we can determine the total capacity of the batteries.

Figure 11-11 illustrates the relationship between average depth-of-discharge (DOD) and cycle life for secondary batteries using nickel cadmium (NiCd) and nickel hydrogen (NiH_2). Extensive data supports the predictions for both NiCd and NiH_2.

The NiCd battery is still a common secondary energy storage system for many aerospace applications. NiCd technology has been space qualified, and we have extensive databases for nearly any mission. A 28 Vdc aerospace NiCd battery usually consists of 22–23 series-connected cells. NiCd battery cells for aerospace missions have typical capacities of 5 to 100 Amp-hr.

NiH_2 technology has been the recently qualified energy storage system of choice for aerospace applications where higher specific energies and longer life are important. The three space-qualified design configurations for NiH_2 are individual pressure vessel, common pressure vessel, and single pressure vessel. The individual pressure vessel was the first NiH_2 technology used for aerospace application. Here, only a single electrochemical cell is contained within the pressure vessel. It has a working

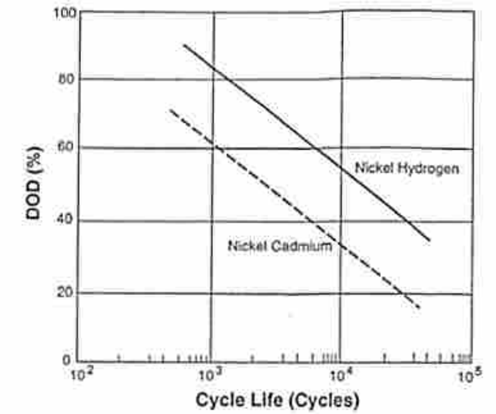

Fig. 11-11. Depth-of-Discharge vs. Cycle Life for Secondary Batteries. Increased cycle life reduces the amount of energy available from the batteries during each cycle—DOD decreases with cycle life.

terminal voltage of 1.22 to 1.25 Vdc depending upon discharge loads. The typical individual pressure vessel battery design consists of multiple cells connected in series to obtain the desired battery voltage. Cell diameters are typically 9 to 12 cm, with capacity ranges from 20 to over 300 Amp-hr. The common pressure vessel NiH_2 technology is very similar to individual pressure vessel, with the primary difference in the wiring connection of the internal electrode stacks. In the individual pressure vessel, the electrode stacks are all connected in parallel. In a common pressure vessel, there are two sets of electrode stacks within the pressure vessel that are series connected, yielding a working terminal voltage of 2.44 to 2.50 Vdc. This design has a higher specific-energy at the battery level since there are half as many pressure vessels and a significant reduction in cell piece-parts. Common pressure vessel NiH_2 technology has been space qualified in the 6 cm and 9 cm cell diameter configuration for capacities in the 12 to 20 Amp-hr range. Batteries with larger Amp-hr capacities should be qualified for aerospace application in the near future. The single pressure vessel NiH_2 battery is designed such that a common hydrogen supply is used by three or more series connected cells with a single pressure vessel. Each cell stack contains its own electrolyte supply which is isolated within individual cell stack containers. The key operating characteristic of this design is to allow the free movement of hydrogen within the cell stacks while maintaining cell stack electrolyte isolation. These batteries are presently available in a 12.5 cm or 25 cm diameter design.

Lithium Ion battery technology offers a significant energy density advantage and a much wider operating temperature range over NiCd and NiH_2 battery types. Typical cell constituents are lithium thionyl chloride, lithium sulfur dioxide, and lithium carbon monofluoride. The nominal operating voltage for a lithium ion cell is 3.6 to 3.9 Vdc, which allows us to reduce the number of cells by approximately one-third when compared to NiCd or NiH_2 cells. The lithium ion secondary battery system offers a 65% volume advantage and a 50% mass advantage for most present day aerospace battery applications. Lithium ion battery technology should be qualified for a planetary mission by the year 2000, with space qualification for GEO and LEO applications by the years 2005–2010.

To size a secondary battery, we must identify the parameters and apply the equation in Table 11-40. The parameter values used in the equation can vary significantly with battery type. The ideal battery capacity is the average eclipse load, P_e, times the eclipse duration, T_e. This ideal capacity must be increased to include the battery-to-load transmission efficiency, n, and the depth-of-discharge constraints. For LEO, we expect the battery's DOD to be 40–60% for NiH_2 technology, compared to 10–20% for NiCd technology. We base these expectations on the average DOD over 24 hours and assume the batteries are fully recharged at least once during this period. The number of batteries, N, may be equal to one for this calculation if you simply require a battery capacity. Two to five batteries are typical. We must have at least two (unless the battery uses redundant cells) because the spacecraft needs redundant operation with one unit failed. But more than five batteries require complex components for recharging. The secondary batteries may be required to help meet peak power loads during full Sun conditions. For some missions, the peak power loads may drive the required battery capacity rather than the eclipse load. To design the Energy-Storage subsystem, follow the steps in Table 11-40

TABLE 11-40. Steps in the Energy Storage Subsystem Design. To obtain the required battery capacity in Amp-hr, divide by the required satellite bus voltage.

Step	Consider	FireSat Example
1. Determine the energy storage requirements	• Mission length • Primary or secondary power storage • Orbital parameters – Eclipse frequency – Eclipse length • Power use profile – Voltage and current – Depth of discharge – Duty cycles • Battery charge/discharge cycle limits	• 5 yrs • Secondary power storage • 16 eclipses per day • 35.3 min per eclipse (T_e) • Eclipse load 110 W (P_e) – 26.4 V, 4.2 A (max) • 20% (upper limit) • TBD—depends on observations taken and downlinked during eclipses
2. Select the type of secondary batteries	• NiCd (space qualified) • NiH_2 (space qualified) • Li-Ion (under development) • NaS (under development)	• NiCd or NiH_2—both are space-qualified and have adequate characteristics
3. Determine the size of the batteries (battery capacity)	• Number of batteries • Transmission efficiency between the battery and the load	• N = 3 batteries (nonredundant) • n = 0.90 • C_r = 119 W-hr • C_r = 4.5 Amp-hr (26.4 V bus)

Battery Capacity: $C_r = \dfrac{P_e T_e}{(DOD)Nn}$ W-hr (for battery capacity in Amp-hr, divide by bus voltage)

11.4.3 Power Distribution

A spacecraft's power distribution system consists of cabling, fault protection, and switching gear to turn power on and off to the spacecraft loads. It also includes command decoders to command specific load relays on or off. The power distribution system is a unique feature of the electrical-power subsystem and often reflects individual spacecraft loads and power-switching requirements. Power distribution designs for various power systems depend on source characteristics, load requirements, and subsystem functions. In selecting a type of power distribution, we focus on keeping power losses and mass at a minimum while attending to survivability, cost, reliability, and power quality.

Power switches are usually mechanical relays because of their proven flight history, reliability, and low power dissipation. Solid-state relays, based on power technology, which uses metal-oxide semiconductor field-effect transistors are available.

The load profile of a spacecraft is a key determining factor in the design specifications of a power distribution subsystem. Predominant spacecraft loads (radar, communications, motors, computers) may require low- to high-voltage dc (5–270 Vdc), high-voltage single-phase ac (115 Vrms, 60 Hz), or high-voltage three-phase ac (120/440 Vrms, 400 Hz)—all converted from the 28-Vdc power bus. Because the regulation requirements for these loads vary, the bus voltage may need further regulating, leveling up or down, and, possibly, inverting through dc-dc converters. Spacecraft power loads often turn on or off or otherwise vary their power consumption. Transient behavior within a load may produce noise that the distribution system translates to other loads, potentially harming working components. In addition, certain spacecraft loads require a voltage different from the bus voltage. Power converters often connect loads susceptible to noise or requiring voltage conversion to the distribution system. These converters typically isolate the load from the noise on the bus and regulate the power provided to the load against disturbances from the load and the bus. They also keep load failures from damaging the power-distribution system and provide on-off control to desired loads. Any dc-dc converter connected to the bus must dampen its electromagnetic-interference filter to keep step loads from causing excessive ringing.

We need to know the boundaries of the load profile to evaluate its effects on required bus voltage and frequency. Most spacecraft have demanded low power (< 2,000 W), so power distribution has relied on a standard, 28 V bus. This standard, with electronic parts built to match, has limited study of the best bus voltage. As power systems expand to many kilowatts, the 28 V bus may not work for power distribution because of losses in cabling and limits on mass. The harness or cabling that interconnects the spacecraft's subsystems is a large part (10–25%) of the electrical-power system's mass. We must keep harnesses as short as possible to reduce voltage drops and to regulate the bus voltage. Figure 11-12 depicts the relationship between current and cable mass.

Systems for distributing power on spacecraft have been predominantly dc because spacecraft generate direct current power. Direct-current systems will dominated throughout the 1990s. Conversion to ac would require more electronics, which would add mass to the EPS. Alternating-current power distribution applies only for high-power spacecraft, such as the International Space Station, which have many electrical loads with varying duty cycles. Even on the space station, however, recent decisions have taken planners back to dc for the entire distribution system.

Power distribution systems are either centralized or decentralized, depending on the location of the converters. The decentralized approach places the converters at each

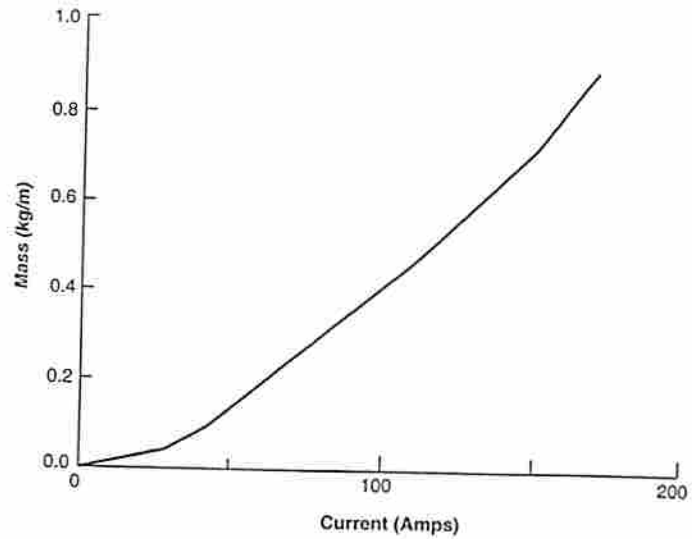

Fig. 11-12. Cable Mass vs. Current. We must account for the cable and harness mass when designing the Power Subsystem. Operating low current (less than 30 amps) devices helps keep this mass low.

load separately, whereas the centralized approach regulates power to all spacecraft loads within the main bus. The decentralized approach implies an unregulated bus because distributed converters regulate power. A regulated power bus typically has some power converters at the load interface because electronics may require different voltages (+5, ±12 Vdc). An advantage of the centralized system is that we do not have to tailor-design the EPS for different applications. Larger spacecraft with high power levels use the decentralized distribution systems, with an unregulated bus, usually.

Fault protection within the EPS focuses on detection, isolation, and correction of faults. Its main purpose is to isolate a failed load that could eventually cause loss of the mission or the spacecraft. A failed load typically implies a short circuit, which will draw excessive power. If this condition continues, the failed load may stress cables and drain the energy-storage reserve. Typically, we would isolate these faults from the EPS bus with fuses (sometimes resettable). Most spacecraft power loads have some sort of fuse in series with the power bus to isolate faults. Of course, if the mission requires us to know where load faults occur, we can add fault-detection circuits. To design the Power Distribution subsystem, follow the steps in Table 11-41.

11.4.4 Power Regulation and Control

The energy source determines how we regulate a spacecraft's power. For example, we regulate a static or dynamic power source through the direct energy transfer method discussed below. But because most aerospace applications use solar photovol-

TABLE 11-41. Steps in the Power Distribution Subsystem Design.

Step	Consider	Possibilities
1. Determine the electrical load profile	• All spacecraft loads, their duty cycles, and special operating modes • Inverters for ac requirements • Transient behavior within each load • Load-failure isolation	• Low-voltage dc: 5 V • High-voltage dc: 270 V • High-voltage 1-phase ac: 115 V_{rms}, 60 Hz • High-voltage, 3-phase ac: 120/440 V_{rms}, 400 Hz
2. Decide on centralized or decentralized control	• Individual load requirements • Total system mass	• Converters at each load—for a few special loads • Centralized converters control voltage from the main bus (no specialized power requirements)
3. Determine the fault protection subsystem	• Detection (active or passive) • Isolation • Correction (change devices, reset fuses, work around lost subsystem)	• Cable size (length and diameter) and excess current-carrying ability • Size of power storage in case of a short circuit • Location of fuses and their type

taics, we will examine power regulation emphasizing that viewpoint. Power regulation divides into three main categories: controlling the solar array, regulating bus voltage, and charging the battery.

We must control electrical power generated at the array to prevent battery overcharging and undesired spacecraft heating. The two main power control techniques, illustrated in Fig. 11-13, are a *peak-power tracker* (*PPT*) and a *direct-energy-transfer* (*DET*) subsystem. A PPT is a nondissipative subsystem because it extracts the exact power a spacecraft requires up to the array's peak power. The DET subsystem is a dissipative subsystem because it dissipates power not used by the loads. However, a DET subsystem can dissipate this power at the array or through external banks of shunt resistors to avoid internal power dissipation. DET subsystems commonly use shunt regulation to maintain the bus voltage at a predetermined level. Figure 11-13 depicts the main functional differences between varying PPT and shunt-regulated DET subsystems.

A PPT is a dc-dc converter operating in series with the solar array. Thus, it dynamically changes the operating point of the solar-array source to the voltage side of the array (Fig. 11-13) and tracks the peak-power point when energy demand exceeds the peak power. It allows the array voltage to swing up to its maximum power point; then the converter transforms the input power to an equivalent output power, but at a different voltage and current. Solar-source characteristics permit us to extract large amounts of power when the array is cold (post eclipse) and at the beginning of life. A peak-power tracker replaces the shunt-regulation function by backing off the peak-power point of the arrays toward the end of the battery's charging period. Because the PPT is in series with the array, it uses 4–7% of the total power. A PPT has advantages for missions under 5 years that require more power at BOL than at EOL.

For direct energy transfer systems a *shunt regulator* operates in parallel to the array and shunts the array current (typically at the array) away from the subsystem when the

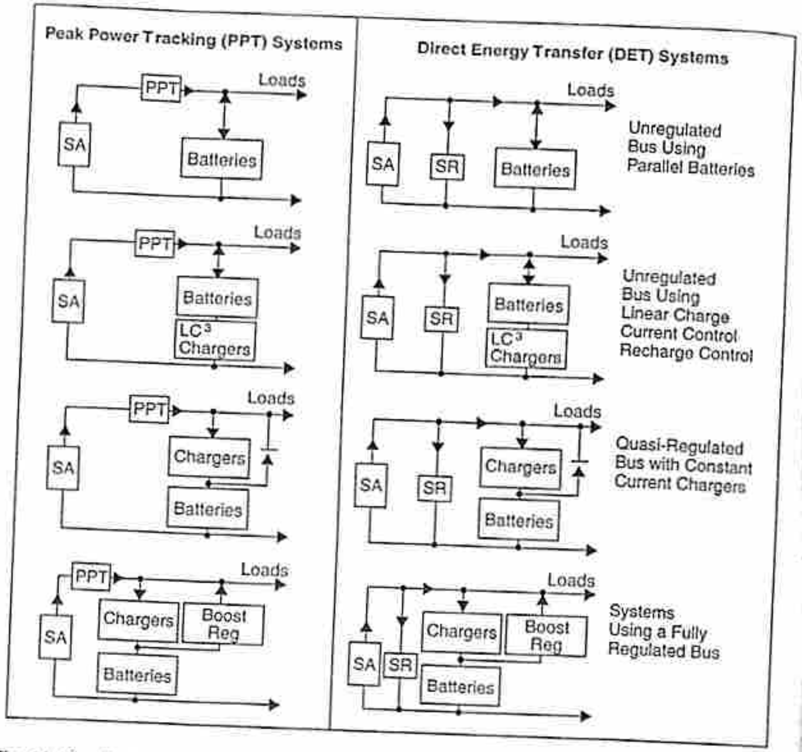

Fig. 11-13. Techniques for Power Regulation. The basic approaches are *Peak Power Tracking (PPT)*, which places a regulator in series with the solar arrays and the load, and *Direct Energy Transfer (DET)*, which uses a regulator in parallel with the solar arrays and load.

loads or battery charging do not need power. Power subsystems with shunt regulation are extremely efficient. They dissipate little energy by simply shunting excess power at the array or through shunt resistor banks. A shunt-regulated subsystem has advantages: fewer parts, lower mass, and higher total efficiency at EOL.

Techniques for controlling bus voltage on electrical-power subsystems fall into three categories: unregulated, quasi-regulated, or fully regulated. Figure 11-13 illustrates the main differences between these techniques. An unregulated subsystem has a load bus voltage that varies significantly. The bus-voltage regulation derives from battery regulation, which varies about 20% from charge to discharge. In an *unregulated* subsystem, the load bus voltage is the voltage of the batteries.

Quasi-regulated subsystems regulate the bus voltage during battery charge but not during battery discharge. A battery charger is in series with each battery or group of parallel batteries. During charge the bus voltage fixes at a potential several volts above the batteries. As the batteries reach full charge, the drop across the chargers decreases, but the bus voltage is still constantly regulated. The bus becomes unregulated during discharge when the voltage is about a diode drop lower than the batteries and decreases as the batteries further discharge. A quasi-regulated power subsystem has low efficiency and high electromagnetic interference if used with a peak-power tracker.

The *fully regulated* power subsystem is inefficient, but it will work on a spacecraft that requires low power and a highly regulated bus. This subsystem employs charge and discharge regulators. We can design the regulators so the charge regulator uses linear technology and the discharge regulator is a switching converter, but for best efficiency both should be converters. The advantage of this type of power subsystem is that, when we connect it to the loads, the system behaves like a low-impedance power supply, making design integration a simple task. But it is the most complex type of power subsystem, with an inherent low efficiency and high electromagnetic interference when used with a PPT or boost converter.

We can charge batteries individually or in parallel. A parallel charging system is simpler and has the lower cost, but does not allow flexibility in vehicle integration. It can also stress batteries so they degrade faster. When batteries are charged in parallel, the voltage is the same but the current and temperature are not. Because current is not rigidly controlled, one battery could receive all the available charge current, and a thermal runaway condition could result if we do not control the bus voltage from the hottest battery. Parallel batteries eventually end up balancing out, so we could use them for missions under five years. To ensure a battery life greater than five years, we should seriously consider independent chargers, such as the *linear, charge-current-control* (LC^3) design in Fig. 11-13.

Batteries usually limit the life of a spacecraft. To support a seven-year life, we must charge the batteries independently to degrade the battery as little as possible. Individual charging optimizes the battery use by charging all the batteries to their own unique limits. It also forgives battery deviations in systems with several batteries. Unfortunately, individual chargers add impedance, electronic piece parts, and thermal dissipation not present in a parallel system. To design the Power Regulation and Control subsystem, follow the steps in Table 11-42.

TABLE 11-42. Steps in the Power Regulation and Control Subsystem Design.

Step	Consider	Possibilities
1. Determine the power source	• All spacecraft loads, their duty cycles, and special operating modes	• Primary batteries • Photovoltaic • Static power • Dynamic power
2. Design the electrical control subsystem	• Power source • Battery charging • Spacecraft heating	• Peak-power tracker • Direct-energy transfer
3. Develop the electrical bus voltage control	• How much control does each load require? • Battery voltage variation from charge to discharge • Battery recharge subsystem • Battery cycle life • Total system mass	• Unregulated • Quasi-regulated • Fully regulated • Parallel or individual charging – < 5 yrs—parallel charge – > 5 yrs—independent charge