Chapter 22

Spacecraft Design, Structure, and Operations

Maj Brian W. Garino, USAF; and Maj Jeffrey D. Lanphear, USAF

Spacecraft are fairly complex vehicles by nature. Thousands of parts and pieces are combined and packed into the nose cone of a rocket and blasted into the cold vacuum of space. Once in orbit, the spacecraft must supply the payload with electrical power, keeping it not-too-hot and not-too-cold, pointing its sensors in the right direction, and processing its data. This chapter will attempt to explain how these functions are accomplished.

A typical spacecraft consists of a mission payload and the bus (or platform). The bus is made up of five supporting subsystems: structures, thermal control, electrical power, attitude control, and telemetry, tracking, and commanding (TT&C).

Structures Subsystem

The functions of the structures subsystem are to enclose, protect, and support the other spacecraft subsystems and to provide a mechanical interface with the launch vehicle. Structural members provide the mating and attachment points for subsystem components such as batteries, propellant tanks, electronics modules, and so on. The structure must also sustain the stresses and loads experienced during environmental testing, launch, perigee and apogee firings, and deployment of booms, solar arrays, and antennas.

Noises, high g-forces, and vibrations can be especially severe on the spacecraft during launch. Acoustic noise is at its highest in the early stages of the launch and is transmitted from the rocket motors by the air through the fairings or housing and into the spacecraft. Steady loads are transmitted through the structure as the rockets accelerate the spacecraft to the velocities required for injection into orbit.[1] A wide range of vibration frequencies is transmitted through the spacecraft supports from the rocket motors. Pyrotechnic devices and springs send sudden shocks through the structure as the spacecraft separates from the booster and various components are deployed into their operational configurations.

When the spacecraft reaches its final orbital position, the loads on the spacecraft are greatly reduced in the zero-gravity environment, but the alignment requirements of sensitive instruments can be very rigorous. In addition, there are many environmental protection factors that exist in space that must also be considered. The designer must satisfy all these requirements while minimizing the structure's mass and cost.[2]

Structure Types

There are two main types of satellite structure: open truss and body mounted. An open truss structure has a specific shape to it (fig. 22-1), usually a box or a cylinder.

Figure 22-1. Space-based infrared telescope facility. (NASA image)

Inside the body of the spacecraft is a honeycomb structure where the equipment boxes are attached. In a body-mounted structure, equipment is attached directly to the structural elements. These satellites do not have a specific shape to them. There are also combinations of these two structure types in which part of the satellite has a shape such as a box, with some equipment attached to the exterior.[3]

Inflatable structures are the latest trend in spacecraft structures. Inflatable structures have the advantage of low mass and low volume during launch, but following deployment, they can expand to volumes not achievable in rigid structures (fig. 22-2).

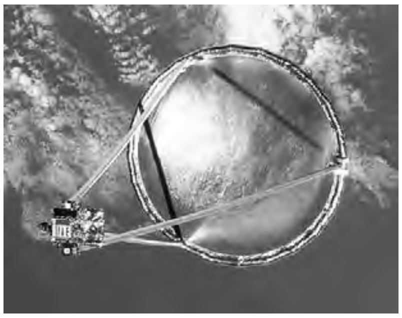

Figure 22-2. Spartan 207 inflatable antenna. (NASA photo)

Materials

When designing a component for structural use in a spacecraft, the engineer must at some point in the analysis decide what materials to use (fig. 22-3). Thousands of different materials are used in making a spacecraft. Many of them serve dual or triple roles to save weight and avoid complexity. For example, the frame of a spacecraft could be a heat sink and electrical ground as well as the main structure.

During its lifetime, the spacecraft will be subjected to severe conditions. These may include various mechanical loads, vibrations, thermal shocks, electrical charges, radiation, or a chemical and particulate environment. The material selected must meet various standards for strength, stiffness, weight, thermal expansion, and melting point. Other properties must also be examined since structural materials often serve multiple roles.

Figure 22-3. Various materials evident in a spacecraft. (NASA photo)

Finally, the availability, formability, and ease with which parts can be machined out of a particular material will influence the selection process. Some materials are scarce and expensive. Others are extremely brittle or soft. Some are hard to cast, forge, or machine. Almost every material presents some type of fabrication problem.

Aluminum, magnesium, titanium, and beryllium are the elements that make up the major lightweight alloys used in space vehicles. They are all much lighter than steel and are nonmagnetic. Aluminum alloys are the most widely used structural materials. Their strength-to-weight ratio exceeds steel, which, combined with their availability and ease of manufacture, makes them very desirable.

Magnesium is lighter than aluminum, but not as strong. It is useful for lower-strength, lightweight applications at temperatures up to 400° Fahrenheit (F). Fabrication is similar to that of aluminum in that parts can be made and joined together in much the same way. Corrosion in the presence of moisture is a problem with magnesium and its alloys; coatings and finishes are needed for protection.

Titanium can replace aluminum in higher-temperature environments, as it has the ability to remain strong at temperatures up to 1,200° F. In situations where a structure must be lightweight and strong when subjected to 400– 1,200° F, aluminum cannot be used. Unfortunately, titanium is not as light or durable as aluminum. It has a tendency to become brittle at low temperatures and when placed under repeated loads. It is also more difficult to weld.

Beryllium is used to make phenomenally light alloys. Its strength is close to that of steel, and its density is comparable to aluminum. This makes for extremely stiff, lightweight structures. An added plus is beryllium's ability to retain its properties at temperatures up to 1,000° F. However, beryllium is more difficult to fabricate than aluminum and is susceptible to surface damage while it is being machined due to its brittle properties. An additional consideration is beryllium's toxicity. It presents a serious health hazard to unprotected workers. Finally, beryllium is more costly than many other metals.

Graphite, plastics, nylon, and ceramics comprise the nonmetallic materials used in spacecraft. Graphite is not usually thought of as a structural material. It is weak and brittle at room temperature, but it is widely used as a thermal protection material. Since the strength of graphite improves with higher temperature up to about 4,500° F, it is very possible that vehicles which must enter Jupiter's atmosphere or orbit very close to the sun may have some structural parts made of graphite.

Plastic has many desirable qualities as a spacecraft component material. It is very inexpensive, readily available, and easy to fabricate into intricate shapes. It is also durable and is a good electrical and thermal insulator. For spacecraft interiors, where temperatures are relatively low, plastic may be a good replacement for light alloys.

Nylon has a unique advantage in that mechanisms made of it may not need lubrication. Nylon may be the optimal material for low-power gear trains in space.

The general property of ceramics is that they are extremely weak in tension and very brittle. They can, however, withstand very high temperatures, protecting themselves by gradual erosion. Hence, ceramics are useful in some radomes, jet vanes, leading edges, and solid-rocket nozzles.

In the future, aerospace fabrication will make greater use of composites. Composites are two or more materials manufactured together to form a single piece that can have almost any property an engineer specifies. Uni- or omni-directional strength, resistance to high temperatures, and resistance to corrosives are a few of these properties. Examples of composites are fiberglass and carbon epoxy, both structural materials, and carbon composite, a thermal protection material used on leading edges of the space shuttle.[4]

Thermal Control Subsystem

The sources of thermal energy in a spacecraft include people (in manned missions), electronic equipment, frictional heat generated as the vehicle leaves or reenters the atmosphere, the sun, heat reflected from the earth (altitude dependent), and Earth thermal radiation (altitude dependent).

The purpose of the spacecraft thermal-control subsystem is to control the temperature of individual components, which ensures proper operation through the life of the mission. Some components must be maintained below a critical temperature. For example, high temperature limits the reliability and lifetime of transistors due to increased electromigration effects. Optical sensors require that the temperature stay within a critical range to minimize lens distortion, and hydrazine propellant must be maintained above a critical temperature (10° Celsius [C]), or it will freeze.[5] The thermal control process has to meet the requirements of all subsystems. Balance between structural and thermal requirements is necessary to achieve the best spacecraft configuration to permit proper thermal balance.

The thermal control subsystem uses every practical means available to regulate the temperature on board a satellite. Selection of the proper thermal control system requires knowledge of mission requirements as well as the operational environment. Temperatures within space vehicles are affected by both internal and external heat sources.[6] Thermal control techniques can be divided into two classes: passive thermal control and active thermal control.[7]

Passive Thermal Control

A passive thermal-control system maintains temperatures within the desired temperature range by control of the conductive and radiative heat paths. This is accomplished through the selection of the geometrical configuration and thermo-optical properties of the surfaces. Such a system does not have moving parts or moving fluids and does not require electrical power. Passive systems offer the advantages of high reliability due to the absence of moving parts or fluid, effectiveness over wide temperature ranges, and light weight. A disadvantage is low thermal capacity. Passive thermal-control techniques include thermal coatings, thermal insulations, heat sinks, and phase-change materials.

Spacecraft external surfaces radiate energy to space. Because these surfaces are also exposed to external sources of energy, their radiative properties must be selected to achieve a balance between internally dissipated energy, external sources of energy, and the heat rejected into space. The two properties of primary importance are the emittance of the surface and solar absorbency. Paints and coatings can be used to reduce reflection and to increase or decrease absorption of heat or light energy. Two or more coatings can be combined in an appropriate pattern to obtain a desired average value of solar absorbance and emittance (i.e., a checkerboard pattern of white paint and polished metal).[8]

On radiators, low absorbance and high emittance are desirable to minimize solar input and maximize heat rejection to space. The initial values of a radiator coating are important because of degradation over the lifetime of the mission. Degradation can be significant for all white paints. For this reason, the use of a second surface mirror-coating system is preferred. An example of such a coating is vapor-deposited silver on 0.2 millimeter (mm)-thick fused silica, creating an optical solar reflector. Degradation of thermal coatings in the space environment results from the combined effects of high-vacuum, charged particles and ultraviolet radiation from the sun.[9]

Thermal insulation is designed to reduce the rate of heat flow per unit of area between two boundary surfaces at specified temperatures. Insulation may be a single, homogeneous material such as low-thermal-conductivity foam or an evacuated multilayer insulation in which each layer acts as a low-emittance radiation shield and is separated by low-conductance spacers.

Multilayer insulation consists of several layers of closely spaced radiation-reflecting shields, which are placed perpendicular to the heat-flow direction. The aim of the radiation shields is to reflect a large percentage of the radiation the layer receives from warmer surfaces.[10] Multilayer insulations are widely used in the thermal control of spacecraft and components in order to accomplish the following:

- Minimize heat flow to or from the component.
- Reduce the amplitude of temperature fluctuations in components due to time-varying external radiative heat flux.
- Minimize the temperature gradients in components caused by varying directions of incoming external radiative heat.

Heat sinks are materials of large thermal capacity, placed in thermal contact with the components whose temperature is to be controlled. When heat is generated by the components, the temperature rise is restricted because the heat is conducted into the sink. The sink will then dispose of this heat to adjacent locations through conduction or radiation. Heat sinks are commonly used to control the temperature of those items of electronic equipment that have high dissipation or a cyclical variation in power dissipation.

Solid liquid phase-change materials (PCM) present an attractive approach to spacecraft passive thermal control when the incident orbital heat fluxes, or onboard equipment dissipation, change widely for short periods. The PCM thermal control system consists primarily of a container filled with a material capable of undergoing a chemical phase change. When the temperature of spacecraft surfaces increases, the PCM will absorb excess heat through melting. When the temperature decreases, the PCM gives heat back and solidifies. Phase-change materials used for temperature control are those with melting points close to the desired temperature of the equipment. Then the latent heat associated with the phase change provides a large thermal inertia as the temperature of the equipment passes through the melting point. However, the phase-change material cannot prevent a further temperature rise when all the material is melted.

One of the more common methods of rejecting heat generated from onboard electronics is to mount the electronics just inside the spacecraft bus structure. Thus, the energy is conducted over a short path to an external spacecraft thermal-control surface (frequently referred to as a radiator and sometimes as a shearplate). This surface is usually coated with a low-solar-absorbance/high-infrared-emittance coating (usually a white paint). Such surfaces are usually positioned by spacecraft orientation to point to deep space. Thus, the natural environment is minimized or eliminated, and maximum heat rejection occurs.[11]

Active Thermal Control

Passive thermal control may not be adequate and efficient for the applications where the equipment has a narrowly specified temperature range or where there is great variation in equipment power dissipation and solar flux during the mission. In such cases, temperature sensors may be placed at critical equipment locations. When critical temperatures are reached, mechanical devices are actuated to modify the thermo-optical properties of surfaces, or electrical power heaters turn on or off to compensate for variations in the equipment power dissipation. Active thermal-control techniques include louvers, electrical heaters, and cooling systems.

For a spacecraft in which the changes in internal power dissipation or external heat fluxes are severe, it is not possible to maintain the spacecraft equipment temperatures within the allowable design temperature limits unless the ratio of absorbance to emissivity can be varied. A very popular and reliable method that effectively gives a variable ratio is through the use of louvers. When the louver blades are open, the effective ratio is low (low absorbitivity, high emissivity); when the blades are closed, the effective ratio is high (high absorbitivity, low emissivity). The louvers also reduce the dependence of spacecraft temperatures on the variation of the thermo-optical properties of the radiator.[12]

Electrical heaters (resistance elements) are used to maintain temperatures above minimum allowable levels. Electrical heaters can be turned on or off from the ground, thermostatically controlled, or continuously on. In most cases, satellites have redundant sets of heaters and thermostats to increase reliability.

Some sensors, especially infrared sensors, require constant cold temperatures. These types of sensors must be isolated from heat-producing system components and may need a further cooling system to function properly. Depending on mission length, the cooling system can be either an open- or closed-loop system. On shorter missions, an open-loop system using an expendable coolant may be selected for its simplicity and higher reliability. Expendable systems commonly depend on the cooling effects of materials undergoing phase change from a solid or liquid to a gaseous form. The gas is then vented out into space after use. For longer missions, closed-loop systems are needed. These systems normally depend on a cryogenic cooler using a liquid such as nitrogen, which is recirculated between the sensor and the cooler.

Radiators are another type of closed-loop system used in cooling. They are active due to the circulation of fluid through the system. Radiator systems require large surface areas to dissipate heat into space, a major disadvantage of this type of system.[13]

Electrical Power Subsystem

A successful mission is dependent on the reliable functioning of the power subsystem. The stringent demands on performance, weight, volume, reliability, and cost make the design of the spacecraft power subsystem a challenging endeavor. Significant advances have been made in this area, resulting in the development of more reliable and lightweight power systems. At the same time, research continues to develop new and novel designs that will maximize reliability while further lowering weight.

Elements of a Spacecraft Power Subsystem

The amount of electrical power a spacecraft requires is dictated by the mission. Uninterrupted power must often be supplied for up to 10 years or more. The generation of electrical power on board a spacecraft generally involves four basic elements:

1. A source of energy, such as direct solar radiation, nuclear power, or chemical reactions.
2. A device for converting the energy into electricity.
3. A device for storing the electrical energy to meet peak and/or eclipse demands.
4. A system for conditioning, charging, discharging, regulating, and distributing the generated electrical energy at specified voltage levels.

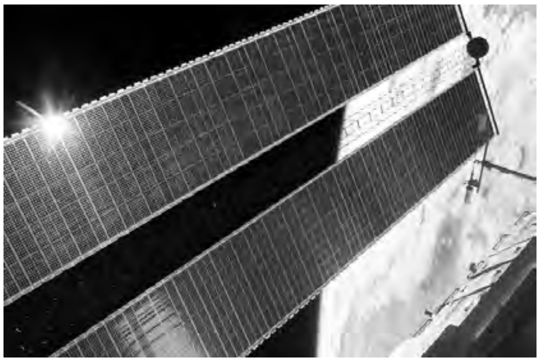

Figure 22-4. Solar panels provide energy for Earth-orbiting satellites. (NASA photo)

The most favorable energy source for Earth-orbiting satellites is solar radiation (fig. 22-4). Because Earth-orbiting satellites pass into and out of Earth's shadow, solar radiation must normally be augmented by another source. Chemical sources such as rechargeable storage batteries serve this purpose. These batteries employ electrochemical processes and have typical efficiencies of 75 percent.[14] When a satellite is in Earth's shadow, it often switches over to battery power, and when it is in the sunlight, the solar arrays power the spacecraft (as well as recharge the batteries).

As an alternative to solar energy, nuclear-powered radioactive isotope generators have also been used. This power source is especially practical for exploration missions to the outer planets, where solar radiation levels are low. For example, the solar radiation reduces from about 54 watts per square foot in the vicinity of Mars to about 4.6 watts per square foot near Jupiter. It therefore becomes necessary to use other primary sources of energy for spacecraft missions to Jupiter and beyond.[15]

Photovoltaic and solar thermionic devices both harness energy from the sun. The photovoltaic energy source uses potential differences created by electromagnetic radiation illuminating semiconductors to provide power. The solar thermionic system uses a temperature gradient set up across different types of semiconductors to create a flow of current. This method is seldom used.

Choosing a spacecraft power source for a particular mission may be difficult. Continuous power requirements, solar eclipse conditions, and power subsystem weight are all major factors in the final decision. Sometimes a combination of energy sources may be required.[16]

Solar Arrays

Solar arrays are mounted on the satellite in various forms. They may be body mounted, stationary, or on directional, steerable wings. A solar array consists of solar cells that convert solar energy into electric power by the photovoltaic effect (fig. 22-5).

The power output of a single cell is quite low, so the individual cells are arranged in series to provide the desired voltage and in parallel to achieve the desired current requirements. In addition, solar array modules are constructed to minimize power loss

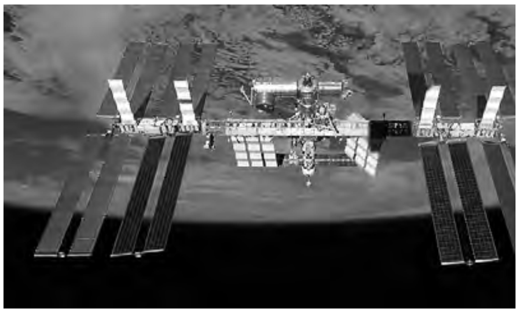

Figure 22-5. International Space Station with large deployable solar arrays. (NASA photo)

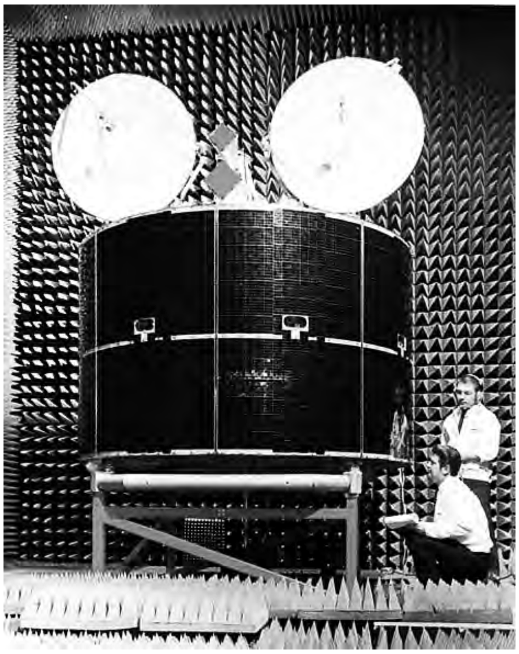

Figure 22-6. Spacecraft with body-mounted solar array. (Air Force photo)

resulting from individual solar cell failures. Without this precaution, the loss of a single cell would create an open circuit for that entire string, and the output from that string would be totally lost.[17]

Some satellites cannot use deployable solar arrays because of the type of attitude control system they employ. Spin-stabilized satellites cannot support large deployable solar arrays because of the stresses placed on the panels while the satellite rotates. For this reason, spin-stabilized satellites require body-mounted solar arrays (fig. 22-6). Body mounting is a very simple approach that utilizes available space on the satellite surface.

Some solar arrays are directional. A solar array drive is employed to control the angle of the arrays so they are always perpendicular to the sun's rays. In contrast, stationary arrays are deployable arrays locked into position relative to the spacecraft body once deployed.

The power of a solar array varies with time due to:

- The variation in solar intensity.
- Variation in the angle between the solar array surface and solar rays.
- Radiation degradation in solar-cell power characteristics.
- Array contamination by thruster propellants and so forth.
- Temperature of the solar array.

When designers select the proper size of the solar arrays, it is important that they consider these factors so that the satellite will have enough power to remain mission effective to the end of its life.[18]

Solar array size is driven by a combination of satellite power requirements and the efficiency of the solar cells to convert solar energy to electrical energy. Thermal control of the solar array panels is achieved by the absorption of solar radiation by the solar cells on the front surfaces of the panels and reemission of infrared energy from the front and back of the panels.

The power from the solar cell is maximized when the angle of incidence of illuminating light is zero (i.e., it is perpendicular to the solar cell surface). The power decreases as the angle of incidence deviates from zero. The primary reason for the increased loss of power at greater angles of incidence is the change in reflection coefficients at large angles.[19]

Storage Batteries

In most spacecraft power systems that use solar radiation, the storage battery is the main source of continuous power. Batteries must provide continuous power to the spacecraft during peak power cycles and eclipse periods. The frequency and duration of eclipse periods depend on the spacecraft orbit.

The eclipse seasons in geostationary orbits occur twice per year, during spring and autumn. These eclipse seasons are 45 days long and are centered on the vernal and autumnal equinoxes. There is one eclipse period per 24 hours with the maximum period being 72 minutes. The batteries discharge during an eclipse and are charged during the sunlight period. So the charge-discharge cycles for any storage battery on board a spacecraft in geosynchronous orbit will be about 90 per year.

In the case of low-orbiting satellites, the number of eclipses increases as the altitude of the satellite decreases. For a 550 km circular orbit, there will be about 15 eclipses per day. The maximum shadow duration is about 36 minutes during each 96-minute orbit. There will be about 5,500 charge-discharge cycles per year in this orbit. Depending on the orbit inclination, the spacecraft may be in continuous sunlight for long periods several times a year.

As mentioned above, batteries are necessary to maintain steady, reliable spacecraft power. A battery is an electrochemical device that stores energy in the chemical form and then converts it into electrical energy during discharge. Chemical reactions taking place inside the battery produce electrical energy whose magnitude is dependent upon various cell characteristics (i.e., individual cell voltage, efficiency of the electro-chemical reaction, size of the cell, etc.).[20]

Batteries are classified as either primary or secondary. Primary batteries are used on spacecraft in which the battery is the only source of electrical power and cannot be recharged. Thus, primary batteries are used for short-duration missions usually of less than a week. Primary batteries have the advantages of being cheap, reliable, and able to deliver relatively large amounts of energy per pound of battery (20–100 watt-hours/lb.).

Secondary batteries are rechargeable. They convert chemical energy into electrical energy during discharge and convert electrical back to chemical during recharge. This process can be repeated many times. Secondary batteries are used for longer-duration missions, such as those of the Defense Meteorological Satellite Program (DMSP), Defense Satellite Communications System (DSCS), and many others, in which solar arrays are the primary source of power. The advantages of secondary batteries are:

- Capability of accepting and delivering power at high rates (eclipse operations and peak power demands).
- Large number of charge-discharge cycles or long charge-discharge cycle life under a wide range of conditions.
- Long operational lifespan.
- Low volume.
- Low cost.
- High, proven reliability.

The disadvantages of secondary batteries are:

- The memory-effect process.
- The complexity and expense of charge-discharge monitoring equipment.
- Low energy-storage capability per pound of battery (6–45 watt-hours/lb.).

There are many types of secondary batteries available. However, only some are considered suitable for space applications. The nickel-cadmium (Ni-Cad) battery is probably one of the most common batteries used in spacecraft today. The primary factors affecting the useful life of a Ni-Cad battery are temperature, depth of discharge, and overcharging. Prolonged exposure of a Ni-Cad battery to high temperature will hasten the breakdown of the battery's internal components, while repeated overcharging at low temperatures can result in pressure buildup within the battery. Therefore, battery temperature is an extremely critical parameter. It is common practice to use radiators and heaters to keep battery temperature between 4° and 24° C (39–75° F).[21]

Repeated deep battery discharges tend to damage the internal structure, causing cracks. These cracks absorb electrolyte and gradually dry out the battery. For a synchronous orbit application of seven to 10 years, a battery will encounter approximately 1,000 charge-discharge cycles over its lifetime. For this number of cycles, Ni-Cad battery depth of discharge is generally limited to between 50 and 60 percent of maximum capacity.

The batteries exhibit a gradual decay of terminal voltage during successive discharge periods. This effect is most pronounced when the charge-discharge cycle is repetitive and is referred to as the memory effect. When the battery is cycled to a fixed depth of discharge, the active material that is not being used gradually becomes unavailable, resulting in an effective increase in depth of discharge. In addition to the gradual decay of discharge voltage, the batteries also exhibit a tendency toward the divergence of the individual cell voltages during charge and discharge. Battery performance can be restored to a certain extent by reconditioning. A typical reconditioning process for a rechargeable battery consists of effecting a deep discharge and then recharging at a high rate. Reconditioning is a process regularly begun before eclipse season on many spacecraft. Procedures to enhance battery life include maintaining batteries within a small temperature range, proper reconditioning, and trickle charging between eclipse seasons to prevent cadmium migration from negative electrodes to positive electrodes.

Another type of secondary battery is the nickel-hydrogen battery (Ni-H_2). This battery is actually a hybrid battery–fuel cell device. It has a positive electrode, much like a conventional battery, and a fuel-cell negative electrode. Hydrogen gas is diffused onto a catalyst, usually platinum, at the negative electrode where the reaction occurs. High-pressure vessels (500 pounds per square inch [psi]) are required to store the hydrogen gas.

Nickel-hydrogen batteries are increasingly being used on newer spacecraft such as military strategic and tactical relay (Milstar) and replacement GPS satellites. Compared to Ni-Cad batteries, Ni-H_2 batteries have higher specific energy, can tolerate a higher number of discharge-recharge cycles, and operate at near-optimum output over a wider range of temperatures.[22]

The newest technology is the lithium-ion battery. Lithium-ion batteries offer a 300 to 400 percent increase in specific energy over older Ni-Cad batteries. For the satellite designer, this means reduced weight and volume (fig. 22-7).[23]

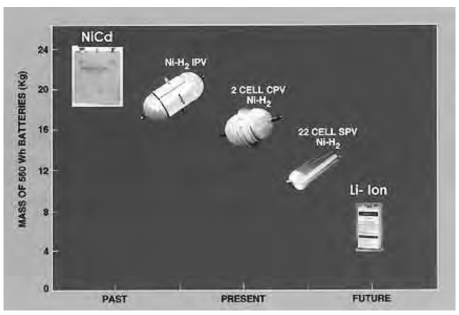

Figure 22-7. Advances in battery technology for space applications. (NASA graphic)

Nuclear Power

Most spacecraft nuclear-power generators are capable of delivering a range of power from a few watts up to several hundred. They have been used very successfully on many deep space missions when solar flux levels were too low for photovoltaic solar cells to be effective.

Political and environmental issues with nuclear-powered satellites were underscored in 1978 after the Soviet Union's *COSMOS 954* plunged to Earth, scattering nuclear material over a large part of northwest Canada. From the beginning of the US space nuclear-power program, great emphasis has been placed on the safety of people and the protection of the environment. The operational philosophy adopted for orbital missions requires that the normal lifetime in space be long enough to permit radioactive decay of the radioisotope fuel to a safe level prior to reentry into the earth's biosphere. Stringent design and operational measures are used to minimize the potential interactions of the radioactive materials with the global populace and to keep any such exposure levels within limits established by international standards.

Like fuel cells, nuclear power generators have a major role in space exploration. There are two basic types of nuclear-powered generators. Radioisotope thermoelectric generators (RTG) rely on the decay of radioisotopes to produce electricity (fig. 22-8). The second type uses the heat from a nuclear fission reactor, much like nuclear generators on Earth, to produce electricity on the spacecraft.[24]

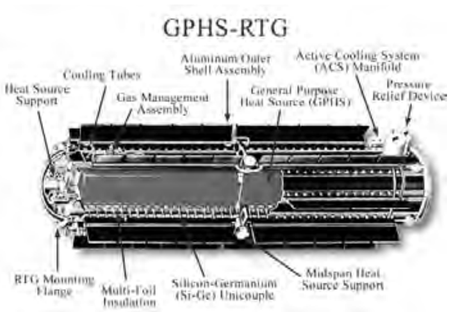

Figure 22-8. Radioisotope thermoelectric generator. (NASA graphic)

With an RTG, the radioactive material is encased in a special container from which the decay particles cannot escape. As the container absorbs energy produced by the alpha and beta particles, it is heated to a high temperature. This heat, in conjunction with a thermoelectric couple, produces electricity.

For the fission reactor, electricity is generated utilizing one of two basic reactor designs, either static or dynamic. The static system uses no moving parts and is usually preferred for this reason. The dynamic system uses the heat to perform mechanical work on a turboalternator assembly, which generates the electricity.

The advantages of nuclear energy include its ability to provide power for long-duration missions without reliance on solar illumination, high system reliability, and high power output versus low mass. Among the primary disadvantages of nuclear power systems are their high cost, shielding requirements (for fission reactors), the need for cooling systems to prevent thermal damage (fission reactors), and relatively low efficiencies (less than 18 percent efficiency). The high level of environmental concern and corresponding political ramifications are also factors that must be addressed with nuclear systems.[25]

Attitude Control Subsystem

Attitude control can be defined as the process of achieving and maintaining a desired orientation in space. An attitude control system is both the process and hardware by which the attitude is controlled. In general, an attitude control system consists of three components: navigation sensors, guidance section, and control section. An attitude maneuver is the process of reorienting the spacecraft from one attitude to another.

When conducting an attitude maneuver, a navigation sensor locates known reference targets such as the earth or sun to determine the spacecraft attitude. The guidance section determines when control is required, what torques are needed, and how to generate them. The control section includes hardware and actuators that supply the control torques.[26]

Active and Passive Control Systems

There are two categories of attitude control systems: active and passive. Active systems use continuous decision making and hardware (closed loop) to maintain the attitude. The most common sources of torque actuators for active control systems are thrusters, electromagnets, and reaction wheels. In contrast, passive attitude control makes use of environmental torques (open loop) to maintain the spacecraft orientation.[27] Gravity gradient and solar sails are common passive attitude-control methods.

Attitude control systems are highly mission dependent. The decision to use a passive or active control system or a combination of the two depends on mission pointing and stability requirements, mission orbital characteristics, and the control system's stability and response time. For example, a near-Earth, spin-stabilized spacecraft could use magnetic coils for attitude maneuvers and for periodic adjustment of the spin rate and attitude. Above synchronous altitudes, thrusters would be required for these functions because the earth's magnetic field is generally too weak at this altitude for effective magnetic maneuvers.

Any satellite orbit requires stabilization to increase its usefulness and effectiveness. For instance, when a satellite is not stabilized, it must use omni-directional antennas so that ground stations can receive its downlink information regardless of the satellite's orientation. This necessitates a high-power transmitter, and only a small portion of the total power is radiated to Earth. On the other hand, if there are means to stabilize the satellite so its directional antennas can be pointed at the earth, then lower power may be used to transmit information to the ground.

Spacecraft attitude-control systems incorporate four functions: satellite pointing, orbital transfer maneuvers, stabilization against torques, and satellite despin.[28] Solar arrays generate maximum power when they are perpendicular to the sun. In addition, some satellites carry scientific payloads which must observe a celestial body. In order to observe it, the spacecraft must be able to accurately find the object, track it, and point applicable sensors at it. Sensors must be accurately pointed at Earth to detect intercontinental ballistic missile (ICBM) launches as well as movement of troops, ships, aircraft, and so forth.

During orbital transfer maneuvers, it is necessary to be as precise as possible. Therefore, before firing, the attitude control system must meet stringent requirements on the accuracy of the spacecraft orientation. Aligning the spacecraft for perigee and apogee motor firing requires knowledge of the orbital characteristics at the time at which the motors are fired. This knowledge optimizes the transfer maneuver by ensuring the thruster firings are aligned with the desired orbital plane, minimizing both time and propellant requirements. If the spacecraft relies on solar energy for electrical power generation during the transfer maneuver, then the spacecraft must be optimized for maximum solar-cell illumination during the transfer. The spacecraft must be reoriented again after the completion of the transfer maneuver.[29]

Disturbance torques are environmental torques (i.e., drag, solar wind, magnetic field, gravity, and micrometeoroid impacts) or unintentional internal torques (i.e., liquid propellant slosh and center of gravity changes). Because these can never be totally eliminated, some form of attitude control system is required. Control torques, such as those produced by thrusters, are generated intentionally to control spacecraft attitude.[30]

Traditionally, spacecraft employing solid-propellant apogee motors have adopted spin stabilization during the parking and transfer phases. Even spacecraft that have active attitude control systems in their operational orbits are frequently spin stabilized in an initial (transfer orbit) phase of their mission. Spin stabilization during transfer orbit allows thermal control to be distributed evenly throughout the spacecraft. If the spacecraft is required to be three-axis stabilized, it must be despun before being injected into the appropriate attitude. If the spacecraft is to be spin stabilized, then the spin rate must be increased or decreased, depending on the final spin rate required.

Navigation Sensors

As mentioned before, sensors are required to determine the orientation of the spacecraft and its current state. The types of sensors used on a particular vehicle depend on several factors, including the type of spacecraft stabilization, orbital parameters, operational procedures, and required accuracy.[31]

Sun sensors are the most widely used sensor type. The sun is sufficiently bright to permit the use of simple, reliable sensors without discriminating among sources and with minimal power requirements. Many missions have solar experiments, most with sun-related thermal constraints, and nearly all require the sun for power. Consequently, many missions are concerned with the orientation of the spacecraft with respect to the sun. Attitude control systems are frequently based on the use of a sun reference pulse for thruster firings. Sun sensors are also used to protect sensitive equipment, such as star trackers, from harmful particle bombardment as well as to position solar arrays to achieve maximum power-conversion efficiency.

The orientation of the spacecraft to the earth is of obvious importance to navigation, communications, weather, and Earth-resources satellites. To a near-Earth satellite, the earth is the second brightest object and covers up to 40 percent of the sky. The earth presents an extended target to a sensor, compared with point source approximations used for sun and star detectors. Consequently, detecting only the presence of the earth is normally insufficient for even crude attitude determination, and nearly all sensors are designed to locate the earth's horizon.[32]

Unfortunately, the location of the earth's horizon is difficult to define because its atmosphere causes a gradual decrease in radiated intensity away from the true or hard horizon of the solid surface. Earth-resources satellites, such as LANDSAT (fig. 22-9), communications, and weather satellites, typically require a pointing accuracy of 0.05 degrees to less than a minute of arc, which is typically beyond the state of the art for horizon sensors.

Earth emanates infrared radiation. The infrared intensity in the 15-micron spectral band is relatively constant. Most horizon sensors now use the narrow 14- to 16-micron band. Use of the infrared spectral band avoids large attitude errors caused by visible light off high-altitude clouds. In addition, the operation of an infrared horizon sensor is unaffected when looking at the shadowed side of the earth. Infrared detectors are less susceptible to sunlight reflected by the spacecraft than are visible-light detectors and, therefore, avoid reflective problems. Sun interference problems are also reduced in the infrared band where the solar intensity is only 400 times that of the earth, compared with 30,000 in the visible spectrum.[33]

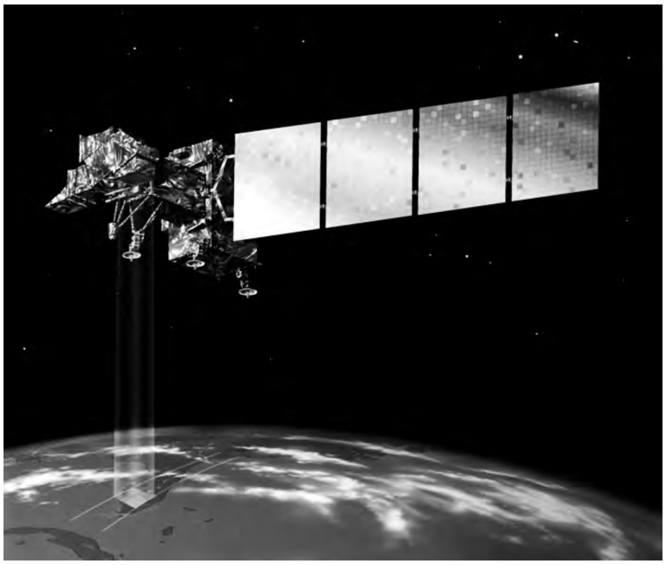

Figure 22-9. LANDSAT with Earth and star sensors. (NASA image)

Star sensors measure star coordinates and provide attitude information when these observed coordinates are compared with known star positions and magnitudes obtained from a star catalog. In general, star sensors are the most accurate of navigation sensors, achieving accuracy to the arc-second range. However, this capability is not achieved without considerable cost. Star sensors are heavy, expensive, and require more power than most other navigation sensors. In addition, computer software requirements are extensive because measurements must be preprocessed and identified before attitudes can be calculated. Because of their sensitivity, star sensors are subject to interference from the sun, earth, and other bright objects. In spite of these disadvantages, the accuracy and versatility of star sensors have led to applications in a variety of different spacecraft attitude control systems.

Star sensing and tracking devices can be divided into three major categories: star scanners, which use the spacecraft rotation to provide the searching and sensing function; gimbaled star trackers, which search out and acquire stars using mechanical action; and fixed-head star trackers, which have electronic searching and tracking capabilities over a limited field of view.

Stray light is a major problem for star sensors. Therefore, an effective sun shade is critical to star-sensor performance. Carefully designed light baffles are usually employed to minimize exposure of the optical system to sunlight and light scattering caused by dust particles, clouds, and portions of the spacecraft itself. Even with a well-designed sun shade, star sensors are typically inoperable within 30 to 60 degrees of the sun.

Star scanners used on spinning spacecraft are the simplest of all star scanners because they have no moving parts. Gimbaled star trackers are commonly used when the spacecraft must operate at a variety of attitudes. This type of tracker has a very small optical field of view (usually less than one degree). Gimbaled star trackers normally operate on a relatively small number of target stars. A major disadvantage of gimbaled star trackers is that the mechanical action of the gimbal reduces their longterm reliability. Fixed-head trackers use an electronic scan to search their field of view and acquire stars. They are generally smaller and lighter than gimbaled star trackers and have no moving parts.[34]

Magnetometers can be used to measure both the direction and magnitude of the earth's magnetic field to the milligauss accuracy. They are reliable, lightweight, and have low power requirements. They operate over a wide temperature range and have no moving parts. However, magnetometers are not accurate inertial navigation sensors because the earth's magnetic field is not completely known, and the models used to predict the magnetic field direction and magnitude at the spacecraft's position are subject to substantial errors. Furthermore, because the earth's magnetic field strength decreases with distance from the earth, residual spacecraft magnetic biases eventually dominate the total magnetic field measurement. Magnetometers are generally limited to spacecraft with altitudes below 1,000 km.[35]

A gyroscope is any instrument which uses a rapidly spinning mass to sense and respond to changes in the inertial orientation or its spin axis. There are three basic types of gyroscopes used on spacecraft: rate gyros, rate-integrating gyros, and control-moment gyros. The first two types are attitude sensors used to measure changes in the spacecraft orientation. Rate gyros measure spacecraft angular rates and are frequently part of a feedback system for spin-rate control or attitude stabilization. Rate-integrating gyros measure spacecraft angular displacement directly. Control-moment gyros generate control torques to change and maintain the spacecraft's orientation.[36]

Notes

1. Jerry Jon Sellers, *Understanding Space: An Introduction to Astronautics* (Boston: McGraw Hill, 2004), 501.
2. Ibid., 504.
3. Sellers, *Understanding Space*, 499.
4. Wiley J. Larson and James R. Wertz, *Space Mission Analysis and Design*, 3rd ed. (El Segundo, CA: Microcosm Press, 1999), 459–62.
5. Ibid., 428.
6. Ibid.
7. Sellers, *Understanding Space*, 481.
8. Wiley and Wertz, *Space Mission Analysis*, 431.
9. Ibid., 435–36.
10. Ibid., 4
11. Ibid., 440–46.
12. Ibid., 442.
13. Ibid., 439–42.
14. Sellers, *Understanding Space*, 465.
15. Ibid., 470.
16. Ibid.
17. Larson and Wertz, *Space Mission Analysis*, 411–18.
18. Ibid.
19. Ibid.
20. Ibid., 418–22.
21. Ibid.
22. Ibid.
23. NASA, "Advances in Battery Technology for Space Application," http://sbir.gsfc.nasa.gov/SBIR/successes/ss/7-015text.html (accessed 5 April 2008).
24. Sellers, *Understanding Space*, 470.

25. Ibid., 470–71.
26. Wiley and Wertz, *Space Mission Analysis*, 354–55.
27. Sellers, *Understanding Space*, 422.
28. Ibid., 407.
29. Ibid., 412.
30. Ibid., 413.
31. Wiley and Wertz, *Space Mission Analysis*, 371.
32. Sellers, *Understanding Space*, 418.
33. Ibid.
34. Ibid., 419.
35. Ibid., 421.
36. Ibid., 419.