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THERMAL CONTROL OF SPACECRAFT

1.1 INTRODUCTION

In the design of spacecraft, thermal control is needed in order to maintain structural and equipment integrity over long periods of time. It has been recognized since the conception and design of the first space vehicles that a prime engineering requirement is a system for temparature control that permits optimum performance of many components. In fact, if it were possible to operate equipment at any temperature, there would be no need for thermal control. The thermal design of a spacecraft proceeds by identifying the sources of internal and external heat and designing the path for transporting and rejecting heat such that all components remain within required temperature limits.

1.2 THEORY OF HEAT TRANSFER

The first step in determining that thermal controls will be needed is to determine the equilibrium temperatures that the spacecraft will be exposed. This thermal analysis is based on the first law of the thermodynamics.

$$Q-W=dU/dT$$

(Heat transfer into system – work done by the system = time rate of change of the internal energy). The internal energy is proportional to the temperature. However, heat (Q) is not equal to temperature (T). Case in point: the ambient neutral temperature at LEO is about 2000K. However, since the neutral density is so low, it would take a long time to bring an external body into thermal equilibrium due to neutral collisions.

Forms of Energy Exchange or Heat Transfer:

There are three forms of heat transfer:
1. Conduction: Heat transfer through matter in absence of fluid motion.
2. Convection: Thermal energy transfers between a flowing fluid and an interface.
3. Radiation Transfer of energy via electromagnetic radiation.
Convection is not important for unmanned spacecraft since there is no fluid (gas/atmosphere) to convect heat. So it is ignored here.

The equation for heat conduction was developed by Fourier (1822) and is of the form.

$$Q=kA(dT/dx)$$

where A is the cross sectional area of the material and k is the constant of proportionally called the thermal conductivity. This equation states that the heat flow in a solid proportional to the temperature gradient and that it flows from the higher to the lower temperature potential. Thermal conductivity is analogus the electrical conductivity. High thermal conductivity means heat flows easily while small k means the opposite. (Aside: the insulation constant R, given on insulation building material is given by

$$R=L/k$$

where L is the thickness of the insulation. The higher R value is better the insullation).

Radiation is important since solar energy is the primary external sources of energy for a satellite. The solar constant is the rate of energy flux per second at the top of the Earth's atmosphere from the Sun. It has a value of 1353 W/m^2.

All bodies above absolute zero radiate electromagnetic energy. The frequency and amount of that emission depends on temperature. The amount is giving by the Stephan Boltzman equation.

$$Q=\epsilon\sigma T^4$$

where Q is the total radiated energy per unit time per unit area, ε is the emissitivity, σ is equal to 5.67 (exp –8) W/m^2K

No real surface is a black body at all wavalengths. Black matter paint comes close at visible wavelengths but has the emissitivity fall off at higher and lower wavelengths.

Kirchoff's law states

$$Q_{absorbed} + Q_{reflected} + Q_{transmitted} = Q_{incident}$$

The ratio of three quantities on LHS with the RHS are called α=absorptivity, ρ=reflectivity, and τ= transmittence. Kirchoff 's law also states that

$$\varepsilon(\lambda)=\alpha(\lambda)$$

which means that good emitters of a certain wavelength are also good absorbes and vice-versa. Most materials have selective surface meaning emissitivity and absorptivity are not constant with wavelength.It can be calculated the equilibrium temperature of a surface using.

$$Q_{absorbed} + Q_{dissipated} - Q_{emitted}=0$$

If you assume that the surface is completely insulated from the underlying structure no heat is dissipated in the material and above equation reduces to

$$G_sA_i\alpha=\varepsilon\sigma T^4A_t.$$

This temperature is called the equilibrium.

1.3 THE THERMAL ENVIRONMENT

The termal conductivity of the Earth's atmosphere is a function of atmospheric temperature gradients and is dependent on variations in pressure or density at altitudes below 90 km. However, above 90 km the molecular mean free path becomes comparable to the distance in which the temperature gradient varies appreciably, and the thermal conductivity ceases to be pressure independent. By 300 km. altitude, convective heat transfer is negligible. Another way of expressing this is to say that aerodynamic heating can be considered to be negligible above 300 km.altitude.

Hence, heat transfer is by radiation only and the actual physical temperature of a spacecraft is determined by the exchange of energy by means of following:

1. Direct solar radiation
2. Solar radiation reflected from nearby planets (albedo radiations)
3. Thermal energy radiated from the nearby planets
4. Radiation to deep space from the spacecraft.

1.4 THERMAL DESIGN

There are two main methods of regulating temperature on a spacecraft: passive and active control. Passive control incorporates such things as selecting the proper surface material and insulation, the structural design, and the modes of operations and orientations specified for the satellite. Active control consists of putting mechanic devices to regulate the temperature (refrigerators, heaters, and pumps) or movable structure to regulate the flow of radiated heat onto the spacecraft (mechanical louvers or covers). Most spacecraft incorporate a combination of the two methods and which one is primariliy utilized depends on the design criteria of the mission and its instrument (i.e, infra-red detector carrying satellites need cryogenic temperatures to operate while GPS satellite need a temperature range more conductive for battery life and operation). In general active controls cost more money (in terms of mass, power, and volume).

One way to eliminate excess heat is to exchange the heat from the outer surface of the spacecraft to deep space. This is done through radiation and hence surfaces that accomplish this are called space radiators. Space radiators are part of both active and passive heat control systems. Other types of system include phase change devices (such as heat pump) to move heat from one part of the spacecraft using the material's latent heat content properties.

Passive Thermal Components :

Multi Layer Insulation (MLI): MLI closely spaced layers of aluminized Mylar and Kapton alternated with a course net material. Insulation reduces the rate of heat flow per unit area between two boundary surfaces and prevents a large heat influx.

Phase Change Devices: Phase change devices absorb thermal energy by changing from a solid to a liquid. As the temperature decreases, the material re-solidifies. It's especially useful for electric equipment that experiences short power spikes.

Thermal Doublers: Thermal doublers are heat sink made of a highly conductive material placed in thermal contact with a component. Heat is conducted to the sink during an increase in temperature and then dispersed by radiation or conduction.

Thermal Control Coatings: Thermal control coatings surfaces such as black and white paints, and gold, silver, and aluminium foils, that have special radiation properties. Coatings may be combined to obtain more desirable average value for surface absorptivity and emittivity.

Active Thermal Control Components

Heat Pipe: Heat pipe closed system in which heat is dissipated by evaporation and condensation. Thermal energy is absorbed by a liquid within the pipe. The liquid is turned into a gas which transports the heat to the other end of pipe. The gas then condenses, cooling back into a liquid and releasing the heat to radiator.

Louvers: Louvers are surfaces that are like venetian blinds mounted on the front of a radiator. The blades and the baseplate are covered with different thermal coatings.

Second Surface Mirror: A second surface mirror is a mirror which reflects incident energy and can radiate out internal energy. Second-surface mirrors are highly efficient and have all but replace louvers and Maltese crosses.

Maltese Cross: A Maltese Cross surface is similar to a louver, but has a pinwheel- like shape.

Thermal Switch: A thermal switch provides a direct conduction path between the heat sources and the equipment mounting plate when the contracts are closed.

Cold Plates: Cold plates are used for mounting heat dissipating equipment. In an active system, there are fluid passeges within the plate itself. The fluid is then pumped to a radiator. For passive system , the cold plate is usually combined with the radiator.

Electrical Heater: An electrical heater is a device that is controlled by a thermostat and used to heat cold regions of the spacecraft. They generate heat by running electrical current through a resistor.

Water Evaropator: A water evaporator is a device that is part of the water coolant loop in which water is sprayed directly on to a hot plate. The water evaporates and carries the heat away with the steam.

Pumped Loop System: A pumped loop system is a system which uses a fluid to collect heat at a cold plate and transport it to a space radiator for dissipation. There are three types of coolant loops: aircoolant, water coolant, and freon coolant.

Radiators: A radiator is a device with a large surface area used to radiate heat. For active an system, radiator consist of a lattice work of fluid loops (usually freon)

Space Radiator: Heat dissipater located on the outer surface of a spacecraft used to radiate thermal energy to deep space. Second surface mirrors are considered surface radiators.

Heat Exchanger:A heat exchanger is used to transfer thermal energy between two ore more fluids at different temperatures. Heat exchangers can use direct or indirect contact.

1.5 A SAMPLE: IMAGER AND SOUNDER THERMAL CONTROL

Optical and radiometric performance of the Imager and Sounder are maintained throughout the 24 hour orbit by a combination of louver cooling and electrical heating. Thermal control is divided into two primary areas. First is thermal control for sensor module as defined by the scan mirror and telescope assembly along with the optical bench or telescope baseplate and all structural sidewalls. Second is thermal design of the detector radiant cooler assembly; this is treated seperately from the first inasmuch as these two components are intended to be adiabatic (thermally isolated) from each other; the thermal performance of one has little or no effect on the other.

Optical performance is maintained by restricting the total temperature range. Radiometric performance is maintained by limiting the temperature change between views of cold space (rate of change in temperature). Thermal control also contributes to channel registration and focus stability.

The basic thermal design concepts include:

- Maintaining the instruments as adiabatic as possible from the rest of the spacecraft structure.
- Controlling the temperature during the hot part of the synchronous orbit diurnal cycle with a north facing radiator whose net energy rejection capatibility is controlled by a louver system
- Providing makeup heaters within the instruments to replace the infrared energy less to space through the scanner aperture during the cold portion of the diurnal cycle.

Additionally, a sun shield is provided around the scan aperture to block incident solar radiation into the instruments, thus limitting the time in a synchronous orbit day when the scanner can recieve direct solar energy. Uncontrolled temperature variations are reduced by the sun shield arond the scan cavitiy opening, a passive automatic louıver-controlled cooling surface, and electrical heating. Electrical heat decreases temperature excursions during the cold part of the daily cycle, but increases the average temperature. To obtain lower temperature ranges, louver-controlled cooling is provided during the direct sunlight portion of the orbit. A sun shield is installed on the Earth end of the louver system to reduce incident radiaton.

MLI blankets are applied on the outside of all but the north side of the instruments. The cover over the radiation cooler is designed to provide thermal protection of the radiation cooler patch during transfer orbit. This cover has MLI blankets on both sides and is deployed onto the Earth face after reaching synchonous orbit.

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2

SPACECRAFT POWER SYSTEMS

2.1 THE BEGINNINGS

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

2.2 THE INCREASING DEMAND FOR SPACECRAFT ELECTRICAL POWER

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

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

2.3 THE ELECTRICAL POWER SYSTEM

2.3.1 AN OVERVIEW OF ELECTRICAL POWER SYSTEMS

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

The conversion of one form of energy into another involves technologies that are both old and new. While the technologies listed do not form an exhaustive list of options, their great number reflects the richness of possibilities. For example, nuclear sources, primarily viewed as sources of heat, are not included but are certainly important sources of energy for space operations.

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

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

2.4 ELECTRICAL POWER SYSTEM DESIGNS

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

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

2.5 SOLAR ENERGY CONVERSION

2.5.1 SPACE PHOTOVOLTAIC POWER SYSTEMS

A space power system is comprised of a number of subsystem elements, one of which is the solar array. A block diagram of a typical photovoltaic space power system is shown in Figure 3.2.1. The result is that designing and building even just the solar array requires an interdisciplinary, well-coordinated effort similar to that required building the complete power system.

The performance of a space solar array, as well as the entire power system: power per unit mass, expressed in watts/kilogram (W/kg), and power per unit area, expressed in watts/square meter (W/m2). These are referred to simply as specific power and area power density, respectively. The inverses of these quantities are also often used, and are known as specific mass (kg/kW) and specific area (m^2/kW). Typical values for state-of-the-art (SOA) space solar arrays, using silicon solar cells mounted on rigid panels, are 30 to 40 W /kg and 90 to 110 Wlm2 at the start of the mission, or beginning-of-life (BOL). The end-of-life (EOL) values for any given an-ay are dependent on mission time and location. Environmental factors affecting the ratio of EOL to BOL array output include electron and proton radiation induced damage to the cells, along with mechanical and electrical degradation of the cell interconnections and other array Figure

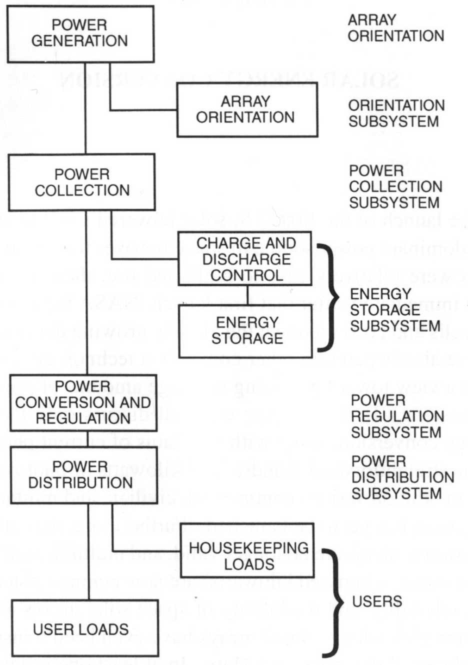


Figure 2.1 Schematic diagram of a space satellite power systems

components from thermal cycling and interactions with the ambient orbital environment. Elimination, or at least substantial mitigation, of such effects is at the heart of all space photovoltaic device and system research and development efforts and will be described in more detail in later sections of this chapter.

2.5.2 SPACE POWER SYSTEM APPLICATIONS AND REQUIREMENTS

Table 3.2.1 lists the broad mission categories into which space missions can be roughly divided, qualitative estimates of the power levels required for each, and the primary attributes any sort of power system must have if it is to be considered for use on such missions. The desired attributes are listed in relative priority order for each mission class with the understanding that detailed trade studies are required to establish the actual order of the priorities for any given mission. It is clear, however, that low mass and long lifetime are important power system drivers in virtually all-potential space missions. Power system cost and size have greater or lesser importance depending on the mission objectives and operational environment (orbital, planetary surface, interplanetary, deep space, etc.).

Table 2.1 Qualitative categorization of space mission

Mission Subset	Power Level	System Attributes
Unmanned Near Earth (Leo, Heo, Geo) and Unmanned Planetary	Low to Intermediate	Low Mass, Long Life
Space Station	High	Minimum Area, Low Mass, Low Cost, Long Life
GEO Platform	Intermediate	Long Life, Low Mass
Lunar Base, Manned Planetary	Intermediate to High	Low Mass, Portability, Long Life
Electric Propulsion Orbit Transfer	High	Reusability, Minimum Area, Low Mass

For example, low total cross-sectional area is a critical attribute for the space station because of the drag produced by the residual atmosphere in the low altitude orbits in which it will fly (Nored and Bernatowicz, 1986). In this case, the array has an important effect on the total life cycle cost of the mission because it directly affects the amount, and hence the cost, of constantly providing fuel to maintain the space station at its proper altitude. A mission to the lunar surface, on the other hand, would not be subject to such a phenomenon; although there would be a life cycle cost associated with standard array maintenance, it could reasonably be expected to be much less than in the International Space Station case. In the lunar mission, although total area may be important because of other factors such as ease of construction and deployment, it is not a primary driver in selecting a particular technology for the mission. Specific power and resistance to proton radiation damage from solar flares are certainly among the more important factors for selecting a lunar surface solar array technology.

2.6 SPACE SOLAR CELL ARRAYS

Space solar cell array designs and deployed configurations have undergone a steady evolution from the first array launched on Vanguard I. That array consisted of 6 small panels that were distributed over the outside of the nearly spherical spacecraft body and provided about 1 watt of power for more than six years (Rauschenbach, 1980). Typical early satellites continued to be approximately spherical with small panels distributed evenly over their external surface to assure continuous power generation as the spacecraft slowly spun about its axis. The growth in power demands soon required the entire spacecraft body to be covered with solar array panels, and finally, in order to provide even more power, the satellites were outfitted with small paddles mounted on hinged arms that swung out from the body of the spacecraft. Explorer 6 was the first satellite to use a paddle array system and carried four 51 cm2 hinged paddles aloft in August 1959 (Rauschenbach, 1980). The paddles were oriented to provide continuous power as the spacecraft rotated. Folded and hinged rigid panel arrays quickly became the standard configuration for all spacecraft in the decades that followed. Space solar cell arrays have grown in both size and complexity from the first exploratory launches, and output powers now range from a few kilowatts to tens of kilowatts. The International Space Station (ISS) array, when its assembly is complete, will be the largest such system yet deployed with a planned 150 kilowatts at EOL. The large size of this array, which will consist of separate wings producing 37.5 kilowatts each at EOL, requires use of a stretched membrane panel structure to minimize its launch volume.

2.6.1 RIGID PANEL PLANAR SOLAR ARRAYS

The most commonly used rigid panel construction has been the so-called honeycomb panel, which usually consists of two thin-aluminum face sheets glued to a honeycomb like core. The core consists of a hexagonal cell structure, the cell walls of which are made from thin (approximately 0.02 mm thick) aluminum ribbon. The total thickness of the panel structure can vary from about 6 mm to 25 mm, depending on the mechanical load requirements established for the array. An individual panel is very stiff and strong, but relatively lightweight. In recent years honeycomb panels have been made from materials other than aluminum, most notably from graphite/epoxy sheets and ribbons. Hybrid panels, which have an aluminum core covered by epoxy/glass face sheets, have also been made and flown. The automatic deployment of rigid panels arrays is accomplished by using springs to actuate motion around a series of hinges between the panels. Once deployed, the panels are locked in place by a set of latches to become a stiffened solar array, and after being locked, such an array cannot be refolded or restowed.

2.7 CHEMICAL STORAGE AND GENERATION SYSTEMS

Spacecraft power can be viewed as the ultimate requirement of portable or remote energy storage devices, Energy storage must be accommodated in the spacecraft power system to provide power for the various mission requirements, The mission requirements determine the various types and levels of energy storage, Both primary (one discharge) and secondary (rechargeable) batteries have been used in space applications, The latter is generally recharged using the photovoltaic array on the spacecraft, Fuel cells provide power for primary Shuttle operations and life support as well as power for other equipment, instruments, and spacecraft in the storage bay, Fuel cells are similar to primary cells in that the stored energy is limited to the fuel and oxidant.

The electrochemical cells in the battery are the basic source of the stored energy, The electrically, mechanically, and thermally connected cells form the battery, Each electrochemical cell is a self-contained device that releases stored chemical energy as electrical energy on demand from an electrical load, When the load is connected across its terminals, one electrode in the cell will spontaneously release electrons while the other spontaneously and simultaneously accepts them, The circuit is closed within the cell by the flow between both electrodes of charged species (ions) in the electrolyte, The number and capacity of the connected cells in the battery determine the energy and power capability.

In a primary cell the reactions are irreversible and therefore the chemical energy can be converted to electrical energy only once, In a rechargeable cell the reactions are reversible, and thus, by reversing the flow of electrons (e.g. from a solar array during the sunlight period), the reactions are reversed, restoring the potential energy difference of the electrodes as chemical energy, The ability to reverse the discharge-charge process thousands of times is a function of the cell chemistry.

The fuel cell system (often referred to simply as the 'fuel cell') has been used for manned missions and is the primary power source for the Shuttle, The fuel cell system includes a number of fuel cells electrically assembled like the cells in a battery to form the fuel cell stack, The remainder of the system includes the external fuel and oxidant tanks, water collection apparatus, and the associated electrical, valving, and plumbing hardware.

The difference between the individual battery cell and the individual fuel cell is that in the former the chemical energy is stored and converted to electrical energy within each cell case. In the fuel cell, the chemical energy is stored in the form of hydrogen gas or more recently methanol (the fuel) and oxygen gas (oxidant) in tanks external to the cells. The energy output of each fuel cell is the result of hydrogen gas or methanol reacting at one electrode releasing electrons on demand from the load and the spontaneous and simultaneous reaction of oxygen gas and the electrons at the other electrode. The circuit is closed within the cell by the flow between both electrodes of charged species (ions) in the electrolyte. The product is pure water, which can be used for consumption. Like the battery, the voltage of the fuel cell stack is the sum of the individual fuel cell voltages required for the spacecraft power. However, the fuel cell system energy storage capability is limited in life by the quantity of Hz and Oz gasses in the external storage tanks.

2.7.1 CELL DESIGN

An electrochemical cell consists of two electrodes (anode and cathode), electrolyte, separator, insulator, insulated seal(s), terminals, and case. The anode and cathode electrodes, each comprised of one or more plates, contain active material of opposing potential energy and polarities. Plates of different polarities are alternated and separated from each other with a separator, which provides electrical insulation and in some cases serves as a container for the electrolyte. Each plate contains electrochemically-producing active material in contact with a current collector from which the electrons either enter or exit from the cell. A porous conductive substrate is used in some cases to house the active material. Cylindrical cells are either spirally wound to provide higher rate capability or of bobbin construction which utilizes a center anode or cathode with the alternate electrode surrounding the center rod. A diagram of the two types of cylindrical cells is given in Figure 2.2.

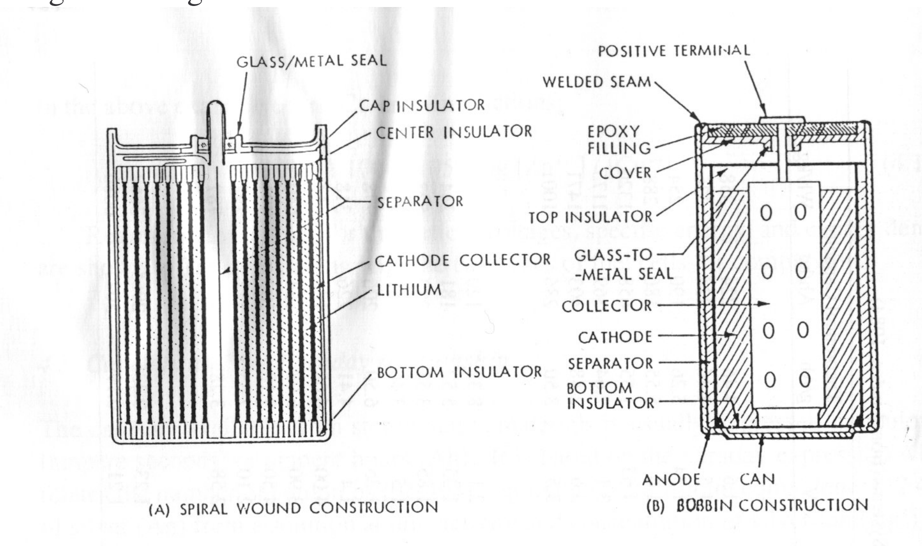


Figure 2.2 Diagrams of cylindrical cell construction

A cell pack consists of the electrodes and separator in close contact to minimize ohmic losses. Exiting from the plates is an electrically conducting metallic tab for making electrical contact to the terminal inside the cell. The tabs of the same polarity are connected together directly or via a busbar and can easily be welded to their designated terminal within the cell during the cell assembly process.

The non-woven nylon or polypropylene separator, which is chemically inactive in the cell environment, is used to maintain separation of plates thus avoiding shorting. It also has the function of providing a path for ions in the electrolyte to diffuse between plates and thus maintain optimum conductivity. In the case of a 'starved' (semi-dry) cell design wherein the electrolyte volume is limited in the cell, the separator can also serve as a sponge to maintain contact between the plates. The separator material must be chemically and thermally stable as well as have a minimal effect on the resistance to flow of ions within the cell.

Activation of the cell is accomplished by addition of the electrolyte, an ionically conducting solution that provides the ion movement between plates. The electrolyte provides the means for completion of the electrochemical oxidation/reduction reactions and thus completes the electrochemical circuit. The electrolyte consists of ionic species (salt, acid, or alkali) specific to the electrochemical nature of the electrodes dissolved in a solvent. The combination must provide maximum conductivity and be chemically, electrochemically, and thermally stable. The solvent can be aqueous, inorganic or organic, and liquid or solid. In some cases, specifically in the lithium primary cells, the solvent also serves as the cathode active material, referred to as 'catholyte' and is consumed during the discharge.

2.7.2 BATTERY DESIGN

A battery consists of a number of cells electrically connected in a series or parallel arrangement, a housing and base plate, connectors, and sensors. The battery structure contains the intercell wiring, thermal fins, voltage and temperature sensors, and connectors for power, signals from the sensors, and voltage monitoring. Although surprising to some, the battery is only a structure that contains the electrochemical cells in the desired voltage arrangement and does not store energy in and of it.

The prime requirement for a battery is to be capable of providing the required power and energy at the desired voltage and over the required period of time. Among the overriding requirements are those of minimum size, volume, and cost to meet spacecraft requirements. Caution: On occasion the spacecraft designers will dictate the allowed volume, mass, and cost prematurely and without due consideration for the real long-term energy storage capability and the usual growth of the power budget.

From a configuration standpoint, for space use a battery must be mechanically configured to withstand a wide range of shock and vibration, be capable of dissipating heat and maintaining a uniform temperature between cells and across the battery, utilize wire size and intercell connectors to minimize voltage drop, be equipped with sensors to provide the power system with the data with which to make decisions, and include connectors to interface power, sensing, and controls to the spacecraft.

The size of a battery will depend on the voltage and capacity required, These in turn determine the number and size of cells, The design of a battery can be initiated after selecting (described later) the cell capacity and configuration (prismatic, cylindrical) required to meet mass, volume, voltage life, and energy storage requirements with adequate margin, It is essential that the cells in a battery and between batteries in parallel be of the same type and capacity to avoid discontinuous performance leading to early battery failure. In addition, it is most important that the cells be from the same manufacturing lot so as to assure consistent performance across and between batteries and to assure mission reliability.

2.7.3 FUEL CELL SYSTEMS

A fuel cell system consists of a group of cells connected in series (fuel cell stack), the fuel and oxidant stored external to the stack, and ancillaries including pumps, plumbing, sensors, and controls to process the products and reactant product. The fuel cell system differs from a battery in that the reactants are stored outside the cell in cylinders. This arrangement infers that the more fuel and oxidant available the larger the energy storage capacity and the longer the fuel cell system will operate. Battery storage capacity, however, is limited by the quantity of active material contained within the case. Fuel cell stack power is the sum of the voltages of each cell times the current. The fuel cell system uses some of the stack power for operation of the ancillaries and thus will be less than the stack power. The capacity of a fuel cell system is determined by the quantity of fuel and oxidant stored. This is a convenient system for manned flight because the fuel cell system can be supplied with enough fuel and oxidant to meet the relatively short mission length and provide water and heat for life support. For longer missions, such as LEO and GEO, rechargeable batteries are used.

The applicability of a fuel cell power plant for space use has, besides the attractive features of being a pollution-free power source based on direct conversion with immovable parts, the primary advantage of its ability to be incorporated into the ecological cycle of the space crew. Further, liquid oxygen and hydrogen are also available for propulsion on board space vehicles because of their high specific impulse. These characteristics have extended the utility of fuel cell power sources for space use in the Gemini, Apollo, Shuttle, and Spacelab programs. The Hz-Oz alkaline fuel cell has emerged as the most attractive candidate for space use.

REFERANCES

3

CONFIGURATION SYSTEM

Under this topic we will look at available information of configuration for Eurasiasat-1 produced by Alcatel Space Company.

3.1 SENSORS

For the geosynchronous satellites, the magnetic field is too weak to be useful as an attitude reference. Therefore, one is left with only Earth, Sun and star references. For communication satellites, accuracies of 1 degree are generally sufficient [2], and precise sensors are not required. Thus, the general sensor configuration here is a static horizon sensor and a digital Sun sensor. During the part of the orbit, the Earth will occult the Sun. To ensure continuous knowledge of the attitude, one generally includes a gyro. Another solution is to include a sensor that measures the direction of the polar star.

3.2 EARTH SENSOR

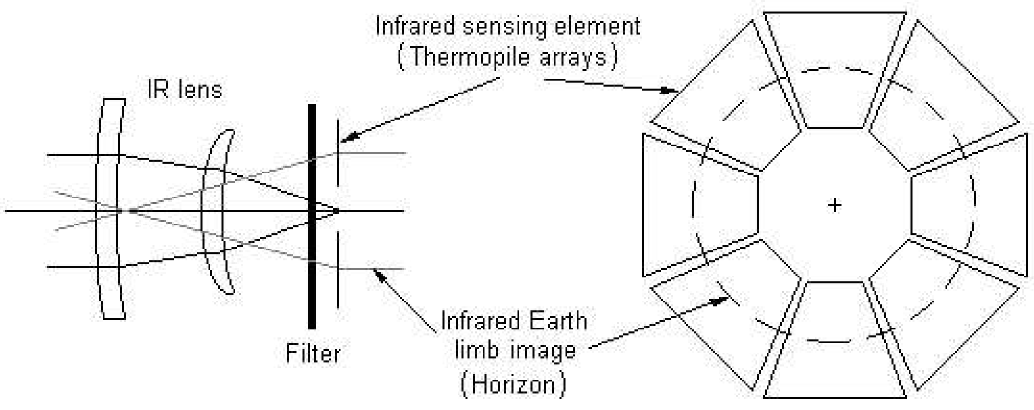


Figure 3.1 Typical static Earth horizon sensor’s schematic view.

Static horizon sensors have a field of view slightly larger than the Earth and contain a number of sensors that sense infrared radiation from Earth’s surface as seen in Figure 3.1. The signal from each of the individual sensing elements is proportional to the fraction of its field of view on which the Earth intrudes. From these signals, the direction of the geocenter, that is, the nadir, can be determined with an accuracy of from 0.1 degrees in near-Earth orbit to 0.01 degrees in geosynchronous attitudes. Their use is generally restricted to spacecraft with circular orbits. For technical information about installed Static Earth Sensor see section 3.8.1 (It has 32-bit digital output).

3.3 SUN SENSOR

Purely digital Sun sensors (also called as digital sun sensors, or DSSs) determine the angles of the Sun by determining, which of the light sensitive cells in the sensor is most strongly illuminated. Thus, the accuracy of this sensor is limited by the angular diameter of the Sun, which is approximately 0.5 degrees, as seen from the Earth [2]. Fine Sun sensors use analog information in addition to digital information. By knowing the intensity of Sunlight striking neighboring pixels, the direction of the centroid of the Sun can be computed to within a few arcseconds. For technical information about installed Digital Sun Sensor see section 3.8.2.

3.4 STAR SENSOR

Form the perspective of data processing; the two-axis star sensor is very similar to that of the fine Star sensors. The new line of star sensors is charged-coupled device (CCDs), similar to the optical element in a video camera, to measure the point of impact. These solid-state devices have much smaller power requirements and hence smaller currents. Two-axis star-sensors are used typically only on three-axis stabilized spacecraft, because the sensors do not function accurately at angular velocities above a few hundred arcseconds per second. Two-axis star sensors are called star trackers if they are able to lock onto follow a star. Star Trackers are mounted on a three-axis-stabilized base. Their field of view is sufficient to include several stars, and their detector/control enables them to select, locate and track one or more of these accurately. Each star has a different vector direction, and so the spacecraft's orientation about the sensor's axis can be determined. For technical information about installed Star Tracker Sensor see section 3.8.3.

3.5 INERTIAL SENSOR: GYROSCOPE

Gyroscopes form the basis of the inertial sensing system for attitude. Fiber optic gyros (FOG), using fibre optic coils to guide the beams round the sensitive axis, with a bias stability of order 0.1 to 10 degree/h [3]. In this mission a FOG was assumed to use from Silicone Sensing Systems. For technical information about installed fibre optic gyroscope see section 3.8.4.

3.6 TORQUERS

As a torquer in a communication satellite solid booster, electrical thrusters, reaction wheels and other mechanisms can be used. In the mission Eurasiasat-1, solid apogee kick motor, reorentation electrical thrusters and reaction wheels was used as a torquer.

3.6.1 ARCJET THRUSTERS

This torquer is installed to complete attitude error correction during on-orbit mission with thermal gas propulsion. Arcjets properties are as below [3].

Thrust range	: 200 – 1000 *mN*
Specific impulse	: 400 – 1000 *sec*

Thrust duration : Months
Typical propellants : H_2, N_2, N_2H_4, NH_3
Kinetic power per unit thrust : 2 – 3 *W/mN*
Thruster efficiency : 30 – 50 %

Advantages of arcjet electrothermal and electromagnetic thrusters are direct heating of gas, low voltage, relatively simple devices, relatively high thrust, ability to use catalytic hydrazine augmentation, and inert propellant. Disadvantages of it are low efficiency, erosion at high power, low I_s, high current, heavy wiring, heat loss, more complex power conditioning. As a result, high-thrust units need P_e of 100 *kW* or more, and it is operational for any kind of geosynchronous satellite.

3.6.2 HALL EFFECT THRUSTER

This torquer is installed to complete transfer orbit mission from LEO to GTO. Hall effect thruster properties are as below [3].

Thrust range : 0.01 – 2000 *mN*
Specific impulse : 1500 – 2000 *sec*
Thrust duration : Months
Typical propellants : *Xe*, *Ar*
Kinetic power per unit thrust : 100 *W/mN*
Thruster efficiency : 30 – 50 %

Advantages of Hall effect thruster are desirable I_s range, compact and relatively simple power conditioning, and inert propellant (Xe). Disadvantages of it are single propellant, high beam divergence, and erosion. As a result, this torquer can be operational for any geosynchronous satellite.

3.6.3 REACTION WHEELS

The disturbance torques, in high Earth orbit, are very small. It is possible to use small reaction wheels to absorb them with an active control system to maintain three–axis stability. In such a system, gyroscopes are generally used to sense and feedback any body motion to the wheel torque motors on each axis. While wheels near saturation, the angular momentum is adjusted using electrical propulsion. The advantages of a three–axis stabilized reaction wheel system are as below.

1. Capability of continuous high–accuracy pointing control,
2. Large–angle slewing maneuvers without fuel consumption,
3. Compensation without fuel consumption.

3.7 SOLAR PANELS DRIVING MECHANISMS

In the communication satellite Eurasiasat-1 the Solar Electric Power Transfer Assembly-Electronics (SEPTA-E) is used to drive solar panels. The function overview supplied by the Alcatel Space Company for the solar panel control mechanism is as in Figure 4.2 as below. For more information see section 3.8.5.

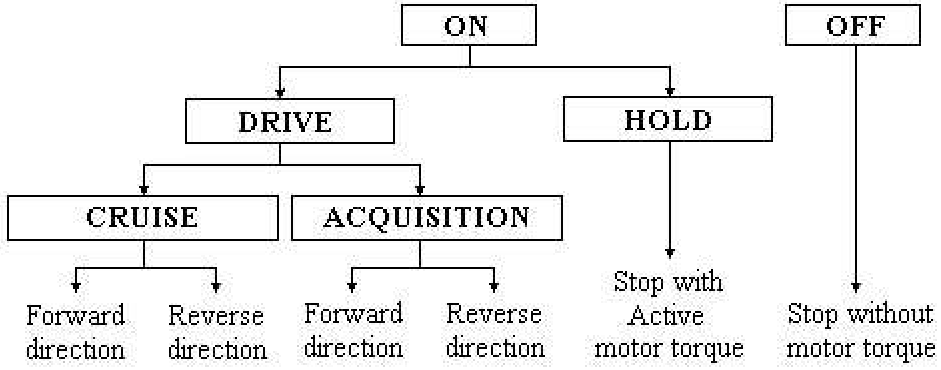


Figure 3.2 Solar panels driving mechanism function overview block diagram.

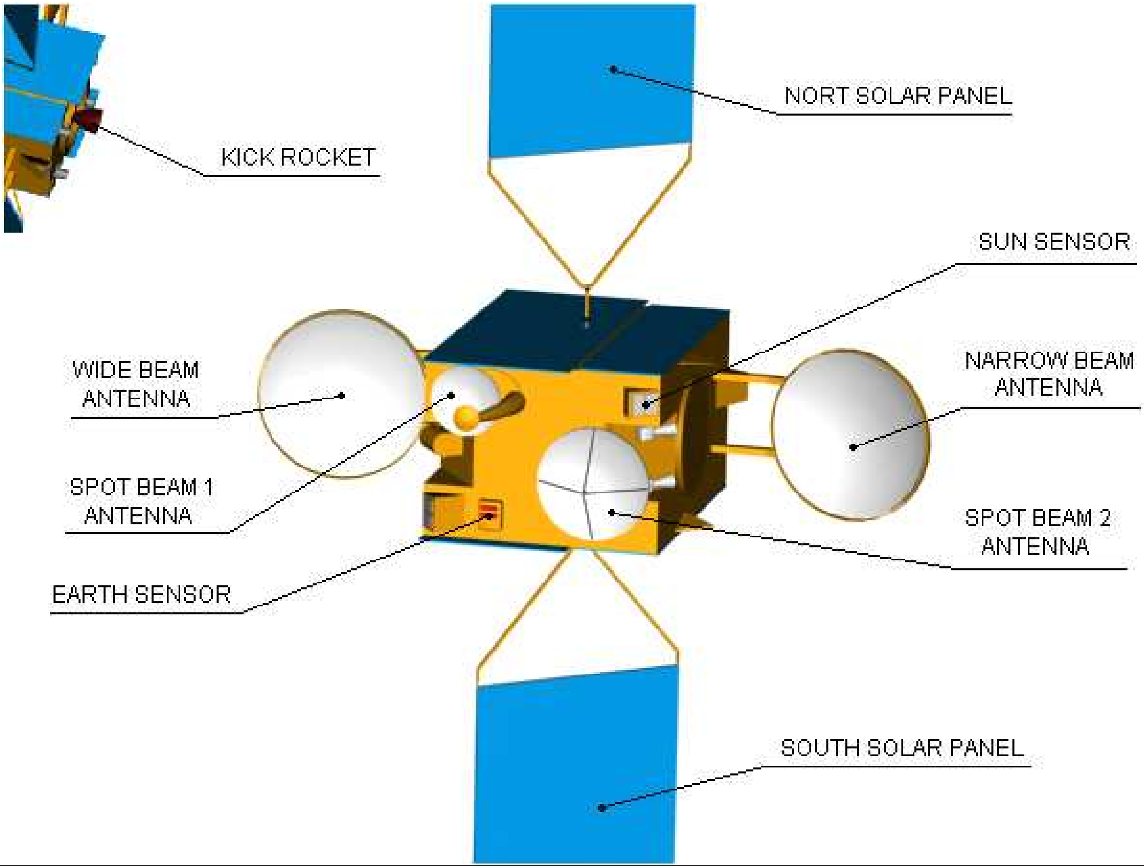


Figure 3.4 3D model for installed payload and sensing electronic and mechanic devices.

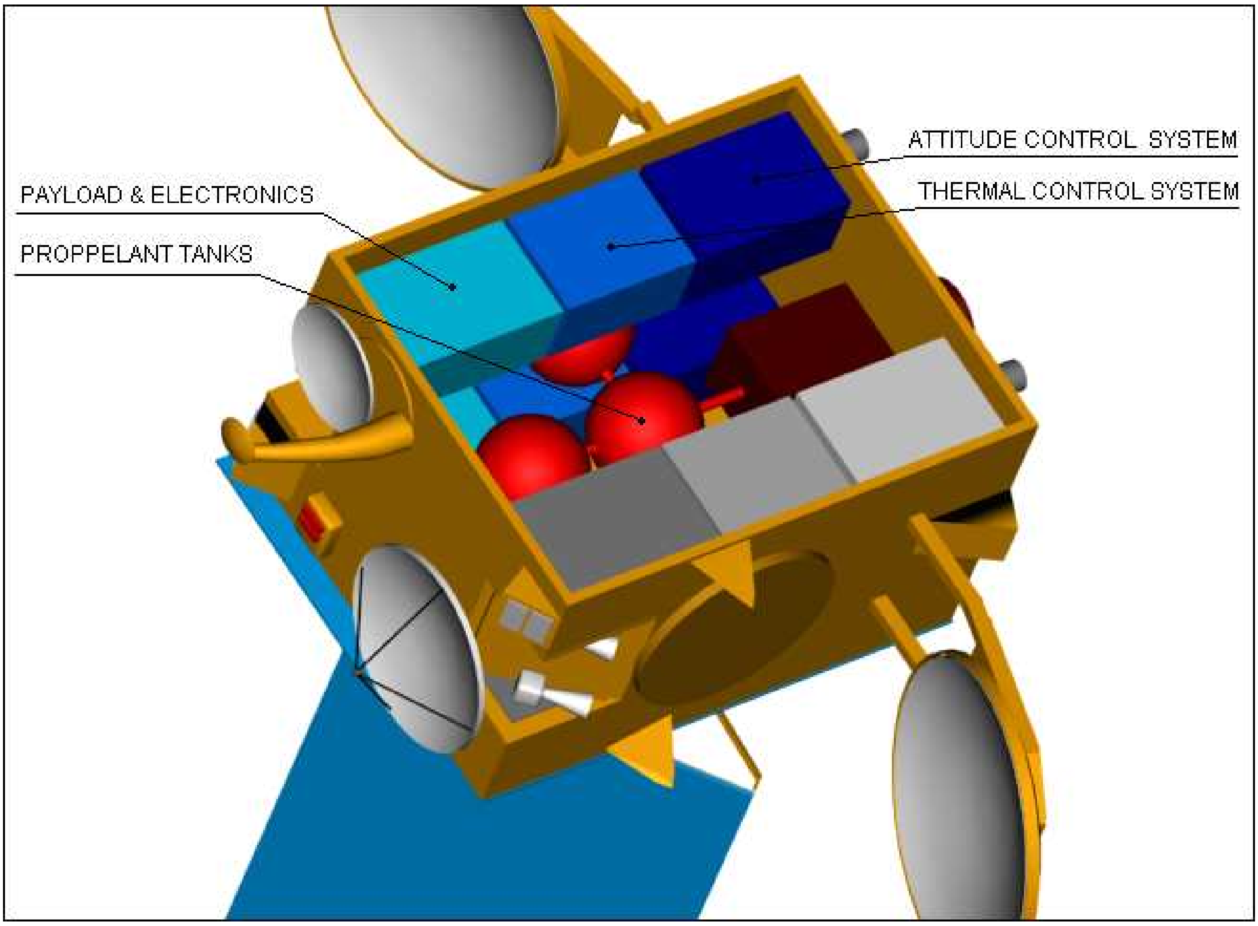


Figure 3.5 3D model illustrates communication satellite Eurasisat–1's inside configuration.

3.8 SPECIFIC INFORMATIONS FOR ELECTRONIC DEVICES

3.8.1 IRES-NE INFRARED EARTH SENSOR [OFFICINE GALILEO]

DESCRIPTION

IRES-NE is a two-axes InfraRed Earth Sensor for the attitude control of three-axes stabilized spacecraft operating in GEO (Geostationary Earth Orbit). IRES-NE allows meeting transfer orbit operations of different launchers thanks to its extended acquisition and operational capabilities.

Operating principle is based on opto-mechanical modulation of the radiation coming from the Earth horizon in the 14-16.25mm band. Four IR (Infra Red) beams in a single telescope are deflected by a mirror along a scan path at 45° Earth latitude North and South. Earth/Space and Space/Earth pulses are compared in phase with the internal encoder reference.

HERITAGE

More than 180 IRES units have been manufactured. The IRES-NE is fully in flight qualified and 20 units have been already produced.

CHARACTERISTICS

Operating Modes
- Earth Acquisition mode (Wide Scan)
- Earth Pointing mode (Narrow Scan)
- Chord Mode (single beam crossing)

Performances
- 14-16.25 mm wavelength operating band
- Earth acquisition mode between 15,300 and 53,000 km altitude
- operational capability up to 140,000 km altitude
- Operational range at GEO

- Pointing mode linear range:
- 5.5° pitch; ±2.5° roll
- Acquisition mode linear range:
- 11° pitch; ±2.5° roll
- Chord mode linear range:
- 23° pitch; ±14° roll
- Acquisition mode sign range:
- 22° pitch; ±13° roll

• 10 Hz output data rate
• Accuracy

- <0.09 deg random error (3σ)
- <0.03 deg bias error

Data Interfaces
- Digital Serial Interface 32 bit or MIL-STD
1553B or MACS-Bus available

Mechanical Interfaces
- 150.8x165.7x178 mm size
- <2.3 kg mass

Electrical Interfaces
- 24 to 50 V unregulated Power Bus
- < 4¸ 5W power consumption depending on data interface

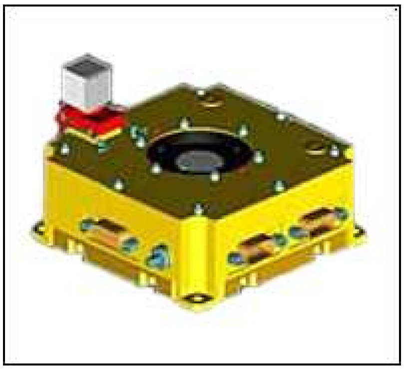
Environmental Conditions
- -30°C to +55°C operational temperature
- -40°C to +60°C storage temperature
- ambient and space vacuum pressure
- Vibration levels

- Sine: 20 g peak
- Random: 18 g rms
- Shock: 2000g from 3 to 10 kHz

• > 15 years lifetime in GEO

3.8.2 DSS DIGITAL SUN SENSOR [OFFICINE GALILEO]

DESCRIPTION



DSS is a new medium/high precision, wide FOV (Field of View) Sun sensor, which use no optics to sense the sun angle in two axes. It has a large dynamic range, and thus combines the function traditionally performed by separate fine and coarse sensors.

DSS applications cover commercial, Earth observation and scientific satellites during transfer orbit maneuvers as well as for on station operations.

High reliability, radiation tolerance, small size, and low mass make DSS suitable also for use in attitude control systems of small satellites.

CHARACTERISTICS

Performances
- FOV: 128x128°
- Accuracy: <0.02°
- Resolution <0.005°

Interfaces
- RS-422 data interface
- 5V pre-regulated or +50V bus power
- <1W power consumption
- 110x110x40 mm size
- <390 grams mass
- -40 to +70°C operating temperature

3.8.3 A-STR AUTONOMOUS STAR TRACKER [OFFICINE GALILEO]

DESCRIPTION

A-STR is a medium field of view (FOV), low mass and low power consumption autonomous star tracker for the attitude control of three-axes stabilized spacecraft operating in GEO (Geostationary Earth Orbit), but its flexible design allows an easy tailoring to other missions as LEO (Low Earth Orbit) platforms for Earth observation, small scientific satellites and interplanetary missions.

Initial attitude acquisition is performed without any apriori information about the S/C orientation (lost in space solution). All operations are executed under microprocessor control by means of SW modules with in-flight re-programming capability.

A-STR is a single package sensor including processing electronics, data/command interface and power supply to spacecraft's main bus.

HERITAGE

A-STR derives directly from the autonomous star trackers of the Rosetta Programme. That, in turns, it is based on more than 10 years of heritage in successfully flown high-resolution star trackers for scientific missions.

CHARACTERISTICS

Operating Principle
• Embedded star catalogues and algorithms for pattern recognition and attitude estimations
• Up to 10 tracked stars

Operating Modes
• Acquisition mode
• Tracking mode
- automatic after acquisition mode
- by attitude control computer to resume attitude without passing through acquisition phase

Performances
• 16.4 x 16.4° FOV
• 1.4 to 5.5 Mi dynamic range
• Tracking rate
- 0.5 °/sec full accuracy
- 2 °/sec reduced accuracy
• <6 sec acquisition time
• 10 Hz update rate
• Accuracy (3σ, EOL, full temperature range)
- <13 arcsec bias error about three axis
- low frequency error: <7 arcsec pitch & yaw; <30 arcsec roll
- random error @ 0.5°/sec tracking rate: <13 arcsec pitch & yaw; <132 arcsec roll
- random error @ 2°/sec tracking rate: <26 arcsec pitch & yaw; <200 arcsec roll

Data Interfaces
• MIL-STD-1553B
• RS 422 (optional)
• Serial interface for EGSE

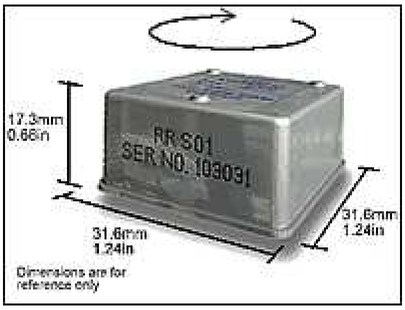
Mechanical Interfaces
• 158(L) x 146(W) x 160(H)mm size
• 2.5 kg mass without baffle

Electrical Interfaces
• 22 to 50 V main bus: 50 and 100 V regulated main bus as option
• power consumption
- 9.5 W @ 20°C;
- 14.5 W @ 50°C

Environmental Conditions
• -25 to +60°C operational temperature
• -30 to +70°C storage temperature
• ambient or space vacuum pressure
• Vibrations levels
- 23.5 g rms in plane
- 26 g rms out of plane

Lifetime
• 18 years in GEO orbit

3.8.4 SIRRS01 FIBRE OPTIC GYROSCOPE [SILICONE SENSING]



Rate range………………………. 110 deg/s
Scale factor……………….……. 18.2mV/deg/s
Scale factor variation over
temperature…….………………. 3%
Bias (setting tolerance)………… ±0.3 deg/s
Bias (over temperature)…..……. ±2 deg/s
Ready time…………….………. 0.3 sec
Bandwidth……………………... 50Hz
Quiescent noise (typical)……… 0.25 deg/s (rms)
Operating temperature………… -40 to +75 deg C
Operating shock………..………. 60G 30ms 1/2 sine
Operating voltage……………... +4.85 to +5.15 V
Operating current……………… <50mA
Mass……………………………. <35g
Size……………………….……. h17.3, w31.6, l 31.6 mm

3.8.5 RSI 01-5/28 REACTION WHEEL [TELDIX]

Angular momentum at nominal speed	: 0.12 Nms
Operational speed range	: ±2800 rpm
Speed limiter (EMF)	: <3000 rpm
Reaction Torque	: 5 mNm (at 2800 rpm)

Dimensions
- Diameter : 95 mm
- Height : 102 mm
- Mass : <0.7 kg

Telemetry Data:
- Speed : Available
- Torque : Available
- Motor Current : Available
- Inner Temperature : Available

Power Consumption:
- Steady State at Nom. Speed : < 2W
- Max. Torque at Nom Speed : < 4W

Electrical Interface:
- Full Duplex (9600, n, 8, 1) : RS – 485

3.8.6 SEPTA-E SOLAR ELECTRIC POWER TRANSFER ASSEMBLY-ELECTRONICS [ALCATEL]

Technical description

SADE supplies for both North and South mechanisms:
- Motors in controlled current mode; sinus and cosinus waveform
- Rough position sensors (accuracy: 10°)
- Panel position reference detectors (0.1°)

Mean features

1. SEPTA-E receives and manages from each mechanism:
   - Rough position signal
   - Reference position pulse
2. Two full cold redundant channels
3. Each channel drives North and South mechanisms
4. Operating mode of each mechanism programmable by TC/TM interface: (graphs chart)

Typical specification

Boost Mode... activation of the two channels (main and redundant) to increase the motor torque
Full autonomous operation.................... better than 10-5
Speed accuracy...................................... 25 VDC to 55 VDC
Input voltage:
PIN…………………………………….. < 25 W in Acquisition
PIN…………………………………….. < 20 W in Cruise Mode
Simplified TC/TM interface:
Mass... 3.7 kg
Size.. 246 x 236 x 100 mm

REFERANCES

1. Alcatel Space Company, web: www.alcatel.com/space
2. Pisacane, V. L., and Moore, R. C. Editors. *Fundamentals of Space Systems*. Oxford University Press, New York, 1994.
3. Sutton, George P., Biblarz, Oscar. *Rocket Propulsion Elements*. John Wiley & Sons Inc., 2000.

4

STUCTURAL SYSTEMS

4.1 DESIGN PHILOSOPHY

Spacecraft design is an iterative process which starts with the initial concept and nominally ends at a predetermined freeze date at the start of the construction phase. With such multi-disciplianary projects it is conventient to specify a number of formal design factor and subsysytems. This activity must proceed rapidly over very complex technical interfaces and benefits considerably from the use of sophiticated computer aided design methods.

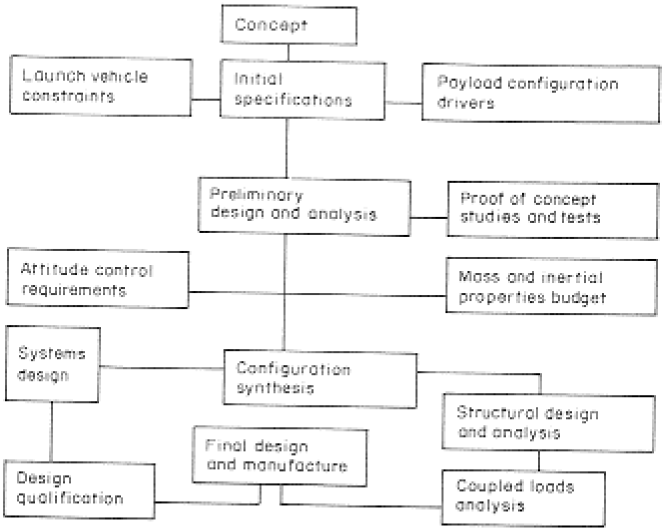


Figure 4.1 Design methodology.

The first step in structural design is to establish a specification based on the mission requrements with as much detail as can be expected at the concept stage. This may be quite mission specific, contrasting the thermal dissipation of a communications satellite with the precision requirements of large antennas or telescopes. A generalized list of requirements is

- Accommodation for the payload and spacecraft systems
- Ability to withstand launch loads
- Stiffness
- Provision of environmental protection
- Alignment
- Thermal and electrical part
- Accessibility

At this stage the configuration will dominate as the distrubution of masses becomes established. The concept of a load path from the interface with the launch vehicles through the spacecraft structure to the mounting points of individual systems or units is followed. At each mounting point individual sets of interface requirements will be generated, thus setting some of the constraints to be applied to the design of the structure.

In the case of large structures such as a space station, reconfigurations well beyond the initial concept will occur on orbit during its 25-year lifetime. Addition of modules and long beams will change the very low frequency dynamic characteristics during attitude manoeuvres and orbit boosting, presenting an additional design criterion.

4.2 MATERIALS SELECTION

The selection of an appropriate material *for* an application requires a knowledge of the way each property can best be used and where each limitation must be recognized.
Selection criteria can encompass the following:

- Specific strength
- Stiffness (the relation between load and deformation)
- Stress corrosion resistance
- Fracture and fatigue resisitance
- Thermal parameters
- Sublimation / erosion
- Ease of manufacture / modification
- Advanced materials
- polymer composite
- advanced metal matrix and ceramics materials

All datas are shown in Table 4.1 and 4.2

Table 4.1 Sample materials properties

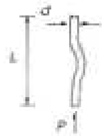
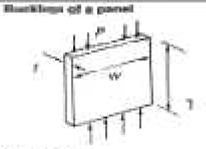
C. R. corrosion resistance
S. C. C. stress corrosion cracking

Caution: There is considerable variation of the properties of materials according to conditions (ageing, temper, form and structure orientation). Consult manufacturers' data

	Density (kg/m³)	Young's modulus E (GPa)	Yield strength f (MPa)	Selection criteria E/ρ	$E^{1/2}/\rho$	$E^{1/3}/\rho$	f_y/ρ	Thermal expansion (μm/m K⁻¹)	Fracture toughness (MPa√m)	Fatigue strength (MPa)	Comment
Aluminium alloy											
6061.T6	2700	68	270	24	2.9	1.5	98.6	23.6	166	97	Good C.R.
7075.T6	2800	71	503	26	3.1	1.5	186.3	23.4	24	159	Prone to SSC in T6 Form
Magnesium alloy											
AZ 31B	1700	45	220	26	3.9	2.1	129.4	26			Prone to SCC
ZK 60 A.T5 extr	1700	45	234	26	3.9	2.1	137.6	26		124	
Titanium alloys											
Ti-6A1-4V (annealed)	4400	110	825	25	2.4	1.1	187.5	9	75	500	
(solution treated and aged)			1005						42	890	
Beryllium alloys											
S 65 A	2000	304	207	151	8.7	3.4	103.5	11.5			Hot pressed sheet — Low fracture toughness
SR 200 E			345								
Ferrous alloys											
INVAR		150	276-415					1.66			Low expansion Ferromagnetic
Stainless steel											
AM 350 (SCT850)	7700	200	1034	26	1.84	0.8	134.3	11.0	40-60	550	
304L Ann	7900	193	170	25	1.8	0.7	21.5	17.2			Austenitic
Composites											
KEVLAR 49 0°[a]	1380	76[a]	1379[b]	55	6.3	3.1	999.3	−4			Structure members
(Aramid fibre) 90°[a]	1380	5.5	29.6	4	1.7	1.3	21.4	57			Pressure vessels Rocket casings
Graphite epoxy sheets (unidirectional) GY70/934	1620	282	586	174	10.4	4.0	361.7	−11.7 (Longitudinal) 29.7 (Transverse)			Sheet
Column ref. (see text)	A	B	C	D	E	F	G	H	I	J	

[a]Tensile modulus
[b]Tensile strength

Table 4.2 Material selection criteria – stiffness

Case	Deflection of a beam	Buckling of a strut	Buckling of a panel
			 (w fixed) (t and L variable)
Characteristic equations	$\delta = \frac{PL^3}{3EI}$ $I = \frac{wt^3}{12}$ Weight $= L.w.t.\rho$	$P_c = \frac{\pi^2 EI}{L^2}$ $I = \frac{\pi d^4}{64}$ Weight $= \frac{\pi d^2}{4} L\rho$	$P = \text{const.} \frac{E}{1-\nu^2}\left(\frac{t}{w}\right)^2 \cdot tw$ Weight $= w.t.L.\rho$
Structural efficiency $= \frac{\text{load}}{\text{weight}}$	$\frac{P}{WT} = \text{const.}\left(\frac{\delta t^2}{L^4}\right)\cdot\frac{E}{\rho}$	$\frac{P}{WT} = \text{const.}\left(\frac{P_c}{L^2}\right)^{1/2}\frac{E^{1/2}}{\rho}$	$\frac{P}{WT} = \text{const.}\left(\frac{P_c}{L^2}\right)^{2/3}\frac{E^{1/3}}{\rho}$
Structure loading coefficient	$\left(\frac{\delta t^2}{L^4}\right)$	$\left(\frac{P_c}{L^2}\right)^{1/2}$	$\left(\frac{P}{L^2}\right)^{2/3}$
Material efficiency criterion	$\frac{E}{\rho}$	$\frac{E^{1/2}}{\rho}$	$\frac{E^{1/3}}{\rho}$
Nomenclature	P = Load; P_c = Critical buckling load (Euler theory); d = Diameter; ρ = Density; δ = Deflection; L = Length; w = Width; t = thickness; E = Young's modulus; ν = Poisson's ratio (0.3); WT = Weight		

4.3 CHARACTERISTIC OF MATERIALS

4.3.1 METALS

Ferrous alloys have numerous applications in which their properties of high strength, corrosion resistance and toughness are required.

Austenitic stainless steels are used for propulsion and cryogenic systems because of their excellent low-temperature toughness. Other alloy uses are in optical and precision structures where properties can be selected to match expansion criteria in a dynamic thermal environment.

Susceptibility to hydrogen embrittlement is a potential hazard for ferrous alloys, particularly where they have been treated in plating solutions. The result is similar to SCC. The corrective treatment is a severe bake-out within a limited time periodobserving of course that the material properties are not affected.

Some types of stainless steel and Invar cannot be considered as non-magnetic. This can be quite a problem when designing magnetically clean spacecraft for research into electromagnetic fields.

Beryllium alloys are shown in table 1 to have some remarkable properties. However, they can be difficult to machine, and great care must be taken due to their toxicity. Also, they are very expensive and their use is therefore limited to specific specialist applications or to areas where weight control reaches levels of extreme concern.

Composite materials are formed by combining two or more materials that have quite different properties. The different materials work together to give the composite unique properties, but within the composite you can easily tell the different materials apart – they do not dissolve or blend into each other. Composites exist in nature. A piece of wood is a composite, with long fibres of cellulose (a very complex form of starch) held together by a much weaker substance called lignin. Cellulose is also found in cotton and linen, but it is the binding power of the lignin that makes a piece of timber much stronger than a bundle of cotton fibres.

4.3.2 COMPOSITE MATERIALS (FIBRE-REINFORCED)

In the use of fibre-reinforced materials (typically with boron or carbon) advantage can be taken of the high strength offered along the fibre. Many strands of the chosen fibre are aligned parallel and held in an epoxy matrix. The resulting material exhibits high structural properties along the direction of the fibres but is limited to the properties of the matrix (glue) in other directions. However, another sheet of the composite can be aligned at an angle to the first, and in many-layer build-ups it is possible to create a material with structural properties which are tailored to the application. The failure mode of a composite is predominantly delamination-particularly in buckling-but a composite which has suffered this defect may still exhibit its full tensile properties.

4.3.3 COMPOSITE MATERIALS (METAL MATRIX)

The limiting factors of an epoxy matrix (above) can be substantially overcome by employing high-strength fibres in a metallic matrix (diffusion bonding a sandwich lay-up). It is possible to form the resulting panels, and even forging may be possible. Although this technology may represent the ultimate direction for 'designed' materials, current costs are high and NDT techniques need development. However, applications are appearing; one particular example being in lightweight mirrors.

Honeycomb sections may be used to create panels with extremely low weight with very high stiffness. Figure 3-4 shows the general configuration of a honeycomb sandwich. A variety of materials and material combinations can be employed. For use in the space environment the designer must consider vacuum stability (out-gassing), and if thermal stability is a consideration similar materials should be used.

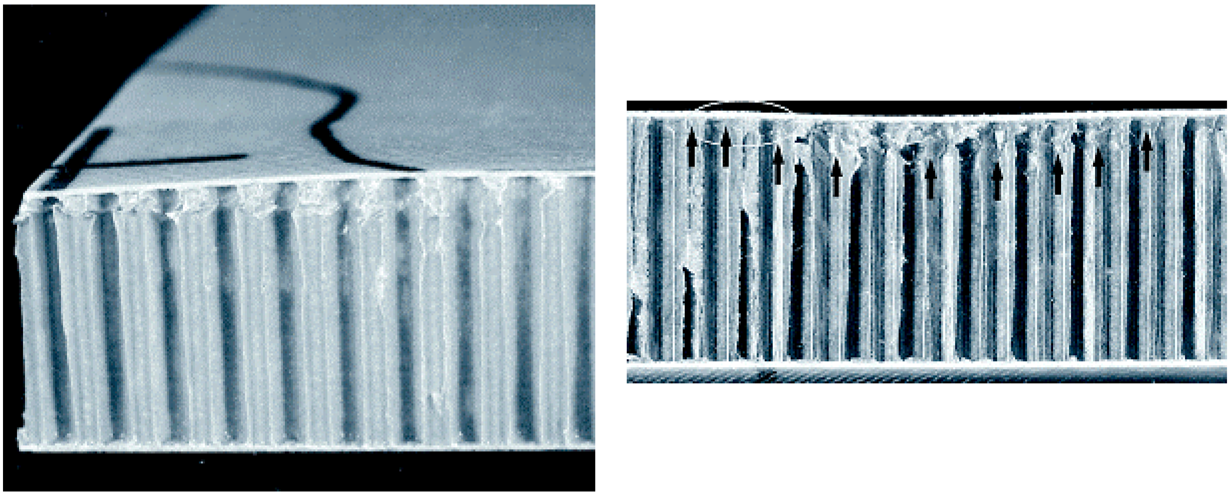
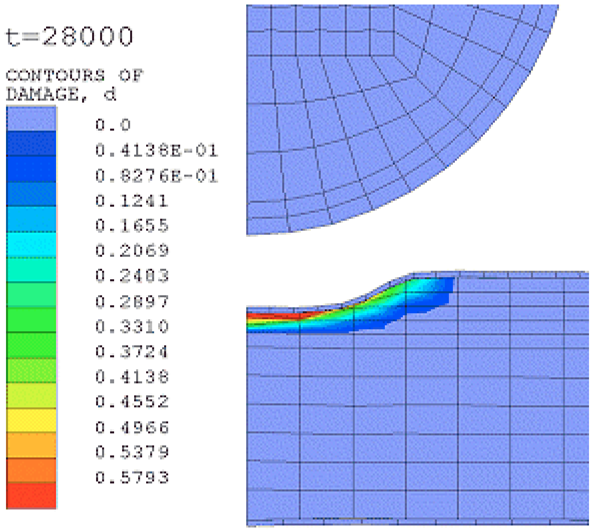


Figure 4.3 Soft body honeycomb at left and hard body honeycomb at right side.

Table 4.3 The result of impact test



REFERANCES