7 Spacecraft Power Generation

1 Introduction

The power subsystem is the most critical system on any spacecraft because nearly every other subsystem requires power. This makes the choice of power systems the most important task facing spacecraft designers. The designers must be aware of the various power systems available for use on spacecraft. This report is a collection of information on typical spacecraft power systems.

There are five major types of space power generation: solar power, fuel cells, batteries, nuclear, and microwave. Solar power systems can be subdivided into solar photovoltaic and solar dynamic power generation systems. Fuel cell power systems include non-regenerative fuel cells, such as those used on the U.S. Space Shuttle, and regenerative fuel cells currently under development. Battery systems included in this paper are both rechargeable and non-rechargeable types. Nuclear systems can be divided into small-scale power supplies (radioisotope thermoelectric generators) and large-scale power supplies (multi-megawatt reactors).

The choice of an appropriate power system depends on the amount of power required, the duration of the mission, constraints on mass and volume, and the impact of the system's hardware on the spacecraft design. These considerations are especially important when manned systems are considered. For example, nuclear power systems will require additional shielding to protect the crew. Manned spacecraft also require more reduncancies then unmanned spacecraft.

7.2 Power Subsystem Interface

The major spacecraft subsystems with which the power subsystem must interface are: attitude control, communication, data handling (computer), environmental control/life support system (ECLSS), guidance-navigation-control (GNC), propulsion, sensors, structures, and thermal management. The power system design process can be defined by the information needed from the other subsystems to design the power system (inputs) and the resulting power system operating characteristics (output). The only information needed from the attitude control, communication, data handling, GNC, propulsion, and sensor subsystems is the amount of power required by each subsystem. Thus, this is a one-way transfer with information entering the power system from the subsystems mentioned. No information needs to be returned to these subsystems. The interface between the power system and the structures subsystem is also one-way. In this case, the power subsystem must pass information on possible nuclear radiation shielding requirements to this subsystem. The environmental control and life support system (ECLSS) must pass information on power requirements to the interface with the power system, and in return, receives information on shielding requirements and operating temperature of the power system. The thermal management subsystem must pass any information on possible power requirements to the power system interface. The power system, in turn, must pass information on the amount of waste heat that needs to be rejected, as well as operating temperature data.

7.3 Power Subsystems Design Considerations

There are a number of items to be taken into account in the power subsystem design. A list of considerations is presented in Table 1. Beginning with the first item, the customer may have specific requirements such as size or operational constraints which could limit the choice of power source. Structural constraints have an influence on power designs. For example, a spacecraft which must perform rapid maneuvers would not be powered by large, flexible solar arrays. The solar distance will also drive the power design.

The lifetime, or operational duration, requirement in a given environment also plays an important role in choosing a power source. For example, radioisotope thermoelectric generators (RTGs) are best suited for long duration and long distances from the Sun. Solar arrays are best suited for missions nearer the Sun.

Attitude control plays a part in power system design in terms of configuration constraints such as solar arrays and in responding to the power required to run the attitude control devices. Orbital parameters are a major consideration in the choice of power source and in the energy storage requirements in eclipse cycles.

The lifetime consideration is depicted in Figure 7.3.1. The various points must be weighed and applied to power systems in order to eliminate the performance overlaps and make the best choice.

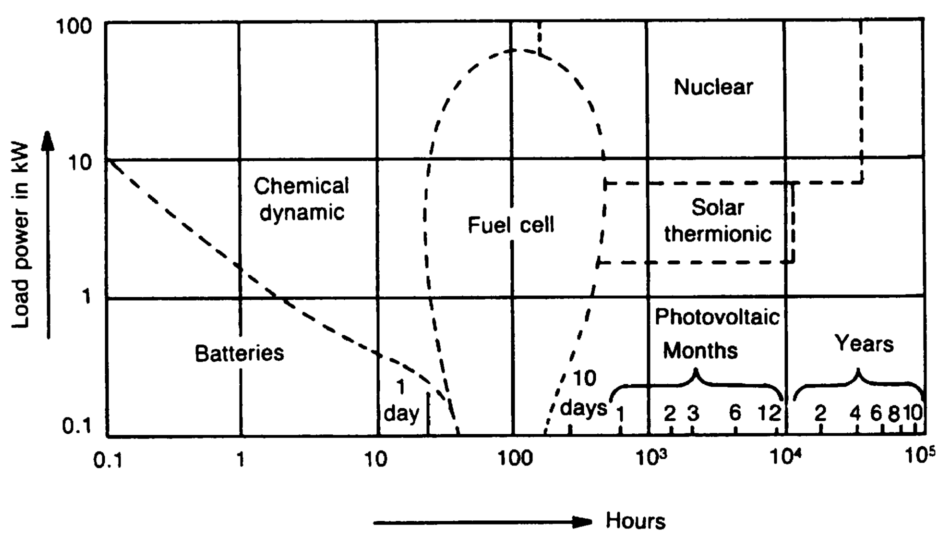


Figure 7.3.1 Most suitable electrical power sources as a function of mission duration [Wertz].

Table 7.3.1 Power Subsystem Design Considerations.

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

7.4 Power Systems

This section includes discussion of seven power systems: solar photovoltaic arrays, solar dynamic systems, fuel cell power systems, batteries, radioisotope power systems, large-scale nuclear power systems, and microwave power beaming.

7.4.1 Solar Arrays

Solar arrays consist of a large number of individual solar cells arranged on a substrate which convert solar energy into electric power by photovoltaic conversion. The solar cells are made in various shapes and sizes which put out relatively low current and voltage. The first use of solar power dates back to March 17, 1958 when Vanguard 1 was launched utilizing six solar panels which provided less than one watt of power for over six years with a 10% conversion efficiency. Since then solar array technology has advanced considerably in the types of deployment, cell materials and efficiency improvement. An example is the first use of roll-out solar arrays on the Hubble Telescope.

Solar array deployment began with the drum-type spin stabilized vehicle where 40% of the array was exposed to the sun at any one time. Deployable paddle-like arrays evolved from the need for increased power outputs. With the technical evolution of thinner solar cells, a variety of roll-out and fold-out solar arrays have been designed and demonstrated. With the advent of flexible solar arrays, a much larger array area can be packaged for the same mass of a paddle-like deployable array.

Solar cells are connected in series to maximize voltage and in parallel for current. To minimize power losses with a single cell failure, the solar array cells are connected in a series parallel ladder network. The basic elements of a solar array are shown in Figure 7.4.2.

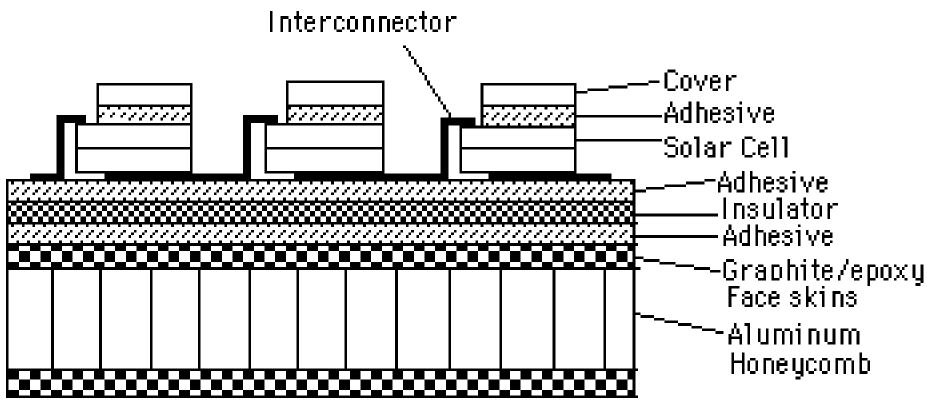


Figure 7.4.1 Solar Array Construction.

7.4.1.1 Solar Cells

Solar cell design is rated by its ability to convert a certain percentage of the solar energy into electric power which is known as the solar cell efficiency which is defined as

$$h = P_{out}/P_{in}$$

where P_{out} is the electrical power output and P_{in} is the solar energy input.

The typical efficiency for silicon cells is 11.5% at 1 AU. The cell design is affected by various factors which must be considered such as I-V characteristics, temperature dependence, distance from sun, incidence angle, and radiation degradation.

7.4.1.2 I-V Characteristics

The current-voltage (I-V) characteristics of solar cells are of importance in the design of solar arrays. From the I-V plot (Figure 7.4.2), an array can be designed for minimum mass and maximum efficiency at the maximum power point (MPP). The MPP is calculated where the product of I and V is at a maximum which is defined by the maximum area rectangle within the plot. This point falls at the knee of the curve as shown in Figure 7.4.2.

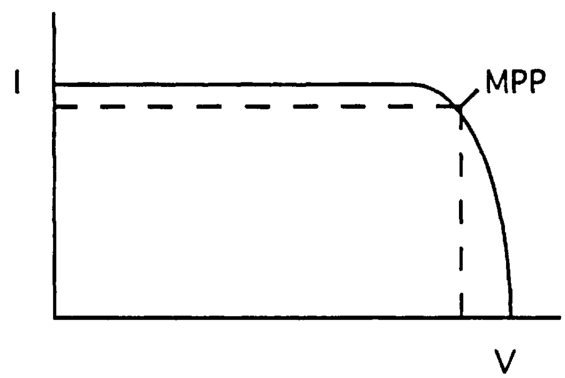
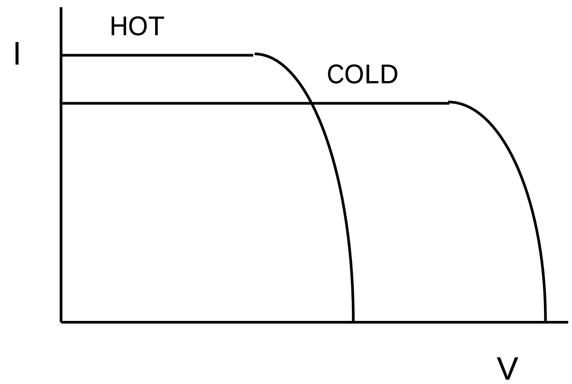


Figure 7.4.3 Typical Solar Cell Power Characteristic.

7.4.1.3 Temperature

Solar cell efficiencies are usually obtained at 25°C to 28°C. A decrease in temperature results in an increase of voltage which can be estimated at 2 mV/°C as seen in Figure 7.4.3. It should be noted that as voltage increases, the current drops. This may have an adverse effect during the exit of an eclipse, where the panels are initially very cold. The initial power surge must be given serious design consideration.



.Figure 7.4.2 Effect of Temperature on Solar Cells.

7.4.1.4 Solar Distance

As the distance from the Sun increases, the available current drops with the voltage remaining constant or may increase due to decreasing temperature. This is illustrated in Figure 7.4.3. In the design of a planetary vehicle, the solar array should be sized at the MPP corresponding to the distance from the sun at which it will be exploring.

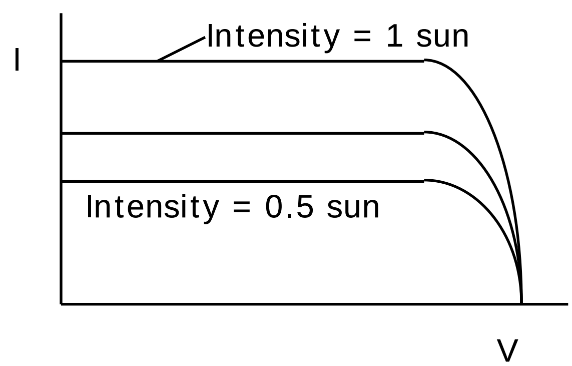


Figure 7.4.3 Effect of Solar Distance.

7.4.1.5 Incidence Angle

At an angle normal to the Sun, the solar array produces peak power. As that angle decreases from the normal, the current decreases with a cosine relationship. Past 60^{o}, the cosine relationship breaks down and the effects of cell thickness and reflection from the cover glass must be taken into account. Figure 6 shows an approximate curve shape for current vs. Sun angle.

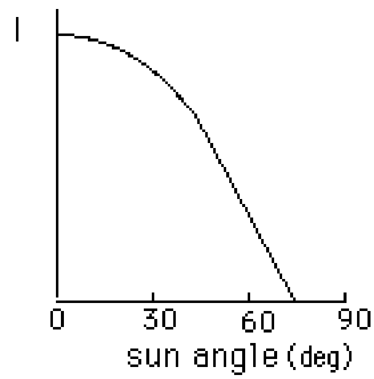


Figure 7.4.4 Effect of Sun Angle on Solar Array Power.

7.4.1.6 Radiation Degradation

The major types of radiation damage in solar cells occur through ionization and atomic displacement. Depending on the space environment and duration in which the solar array will be operating, the damage due to radiation can vary. For geosynchronous satellites, the major source of radiation degradation is due to electrons trapped in the earth's magnetic field and some additional damage is due to solar flare protons. If the vehicle is operating in the Van Allen belts for any long duration , degradation will be more severe. The effect of this degradation is to reduce the short circuit current and open circuit voltage of the cells, thus reducing the maximum power point for the cells. Over a five year span these losses can be approximately a six to twelve percent short circuit current loss and about a two to five percent open circuit voltage loss. Figure 7.4.5 gives the general degradation effect by comparing the beginning-of-life (BOL) and end-of-life (EOL) of the vehicle. Therefore, the spacecraft's power system should be designed based on the EOL rather than on the BOL.

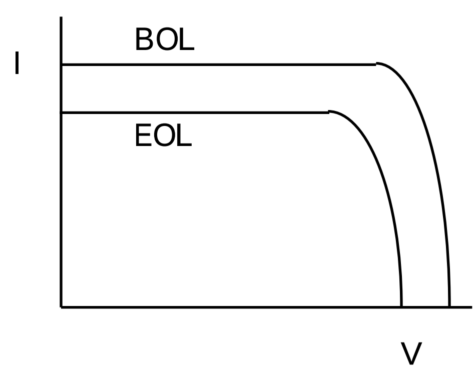


Figure 7.4.5 Effect of Radiation on Solar Cells.

7.4.1.7 Comparison of Silicon and Gallium-Arsenide Solar Cells

With the need for more power, silicon (Si) as the primary semiconductor material in solar cells will need to be replaced by a higher-efficiency material. Gallium-arsenide (GaAs) is that proposed material, which is 40% more efficient than silicon (Si cell efficiency is 18.5%). GaAs is also more resistant to radiation than Si. The only drawback to GaAs cells is their cost, which is nine to ten times that of silicon cells. The big savings come from the launch costs, where a smaller and lighter GaAs array can satisfy the same power needs as a larger and heavier Si array.

Silicon's efficiency degrades approximately 0.5% per ^{o}C rise in temperature. GaAs degradation is approximately half that of silicon, making GaAs especially advantageous in a solar collector array. Another type of solar cell material under research is indium phosphide (InP) which is similar to GaAs in efficiency but is more resistant to radiation degradation. InP could be a good power supply for long duration applications such as a Moon base.

7.4.1.8 Preliminary Sizing of Solar Arrays

The following problem illustrates the calculations to be carried out to arrive at a preliminary solar array size. The array size depends on the power required at the end of the satellite's lifetime (EOL). The beginning-of-life (BOL) power accounts for lifetime degradations.

Problem: Size an array to support a 500W load with battery charge for a Martian orbiter.

Given:

Solar cell material:	Silicon
Cell size:	8×10^{-4} m^{-2} (2cm x 4 cm)
Packing factor:	90%
Solar cell efficiency:	11.5%
Temperature coefficient:	-0.5% / ^{o}C
Operating temperature:	50°C
Solar intensity (1.5 AU):	600 W / m^2
Sun angle:	5°
Lifetime degradation:	15%
Battery capacity:	90 A-h
Battery voltage:	27.5 V

Calculations:

1) Array voltage should exceed battery voltage for battery to charge. 20% above battery voltage is the recommended value.

Array voltage =

2) Total EOL power = load + battery charge
= 500W + battery charge
= 700W

3) Temperature effect = (50 - 28) x 0.005 = 0.11

4) Array BOL capacity =
=
= 929 W

5) Total cell area =
=
= 13.5 m^2

6) Number of cells =
=
= 16875 silicon cells

7) Array size =
=
= 15 m^2 .

For a detailed account of solar array sizing, the reader is referred to the book, *Solar Cell Array Design Handbook* by H.S.Rauschenbach. This is a 500 page book which gives an excellent description of sizing solar arrays.

4.2 Solar Dynamic Systems

To provide higher efficiencies for solar power production, the development of space solar dynamic power systems has been proposed. The difference between solar photovoltaic and solar dynamic power is the power conversion technique. Instead of direct conversion of solar power into electricity as with solar photovoltaics, solar dynamic systems use solar power to heat a working fluid to drive a heat engine which is used to generate electricity. The advantage of solar dynamic systems over solar photovoltaic systems is that dynamic systems in general have a higher thermal efficiency and can be used for higher power levels.

A solar dynamic system consists of four basic components, the collector/concentrator, receiver, radiator, thermal storage material, and the heat engine. The power conversion cycle can be any of the common thermodynamic cycles: Rankine, Brayton, or Stirling. Table 2 lists some of the important design parameters of current and projected solar dynamic systems.

Table 2. Solar Dynamic Systems Performance Data.

Cycle	Availability	W/kg	Thermal efficiency	Temp.	Radiator area (W/m^2)	Collector area (W/m^2)
Rankine	Current	4.14	27.5%	675 K	148	131
Brayton	Current	4.74	33.3%	1012 K	189	158
Stirling	Current	3.91	34.0%	1080 K	217	147
Stirling	Projected	6.25	34.0%	1080 K	217	147

Based on development costs, solar dynamic generation is generally most attractive for power requirements between 20 kW and 100 kW. Larger needs are best met by large scale nuclear systems, and smaller needs are best met by solar photovoltaics, RTGs, or fuel cells.

4.3 Fuel Cell Power Systems

A fuel cell is a device that directly converts the chemical energy of reactants (a fuel and an oxidant) into low-voltage electricity, via electrochemical reactions. A fuel cell is thus similar to a conventional chemical battery. The main difference is that in the ordinary battery, the "fuel" is the built-in expendable electrode. When this electrode is depleted, the battery is either "dead" or requires recharching in order to restore the chemical state of the electrode. A fuel cell is a converter only, using an external fuel supply. Since ideally no part of a fuel cell should undergo any irreversible chemical change, it can continue to operate as long as it is fed a suitable fuel and oxidant and the reaction products are removed .

Because of its high energy density when stored as a cryogenic liquid, hydrogen has become the fuel of choice for aerospace applications. The corresponding oxidant is liquid oxygen. For such fuel cells, the fuel and oxidant typically will be stored as stoechiometric amounts of liquid hydrogen and liquid oxygen. Energy is released when two hydrogen molecules ($2H_2$) exothermally combine with an oxygen molecule (O_2). The reaction releases two free electrons, and the waste product is water. Figure 8 is a diagram of the processes within a fuel cell.

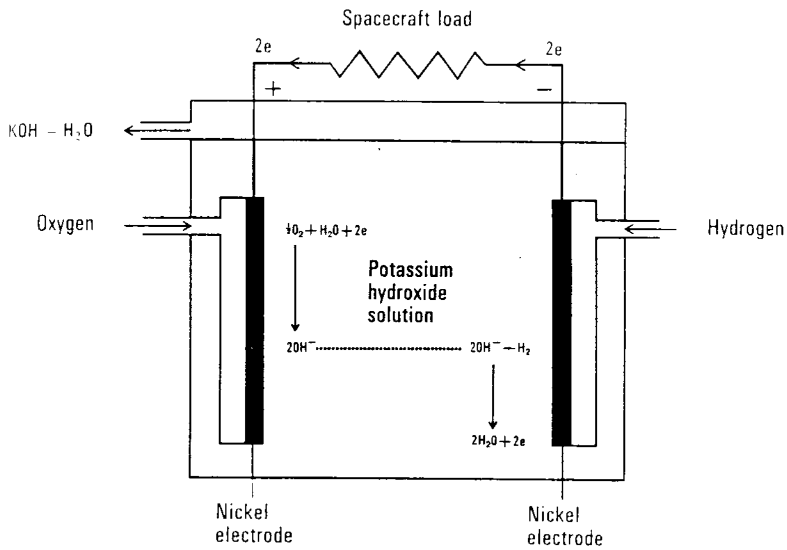


Figure 8: Diagram of a fuel cell.

4.3.1 History of Fuel Cells in the US Space Program

Fuel cells have played a major role in the programs of NASA since the start of manned space flight. They were included as power sources in the Gemini and the Apollo programs because of their high energy density and long lifetime as compared to conventional batteries. The major NASA use of fuel cells at the present time is in providing in-flight electrical power for the Space Shuttle. All the power for the Orbiter comes from its three fuel cell stacks. Each PC17C Stack can provide up to 7kW of continuous power. Table 3 summarizes the main characteristics of the fuel cells used in the Gemini, Apollo, and Shuttle programs.

Table 3. Fuel Cell Performance Data.

Mission	Cell Type & Number	Mission Duration (h)	Misssion Energy (kWh/unit)	0_2/H_2 consump-tion (W/kg)	Current density (mA/cm^2)	Average power (W/unit)	Specific Power (W/kg)	Total Mass (kg)	Lifetime (h)
Gemini	acid 3 units	360	65	0.6944	36	1000	33.33	30	1000
Apollo	alkaline 2 units	192	115	0.5050	68	1420	12.99	110	400
Shuttle	alkaline 3 units	168 (nominal)	2600	0.6521	172	7000	90.91	91	2000

4.3.2 Regenerative Fuel Cells

In non-regenerative fuel cell systems, the water by-product is used for life support in the environmental control and life support system (ECLSS) or as a coolant in the thermal management system. In a regenerative fuel cell (RFC) system, the water by-product is broken down into hydrogen and oxygen through electrolysis and reused in the fuel cell. Electrolysis in turn requires power which is usually obtained from a secondary power source such as a solar photovoltaic system. A truly regenerative fuel cell would be one capable of operating both in charge and discharge modes. The word "regenerative," however, is mainly used to describe a H_2/O_2 fuel cell system with an external electrolyser. The electrolyser decomposes the fuel cell waste into 0_2 and H_2. Figure 9 is a diagram of a regenerative fuel cell system.

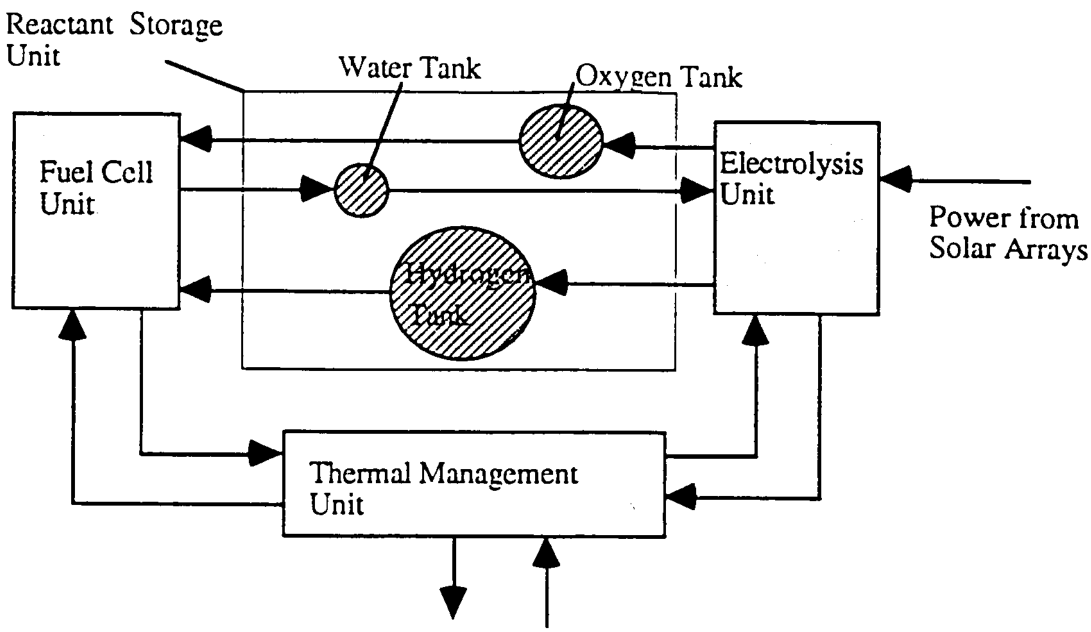


Figure 9: Diagram of a Regenerative Fuel Cell Concept.

The regenerative fuel cell concept is viable only if the power system weight is small compared with the stored reactant weight. RFCs should be chosen over rechargeable batteries when a large amount of energy must be stored to produce a given output power, i.e. when storage is for hours rather than for minutes. Moreover, the concept is attractive because electrolysers are already required for life-support systems, and may in future be used to manufacture cryogenic oxygen-hydrogen fuels from extra-terrestrial resources .

4.3.3 Different Types of Fuel Cells

There are three different kinds of fuel cells with possible applications in the aerospace field:

Alkaline electrolyte fuel cells: The fuel cells used so far in the US space program have all used aqueous alkaline electrolytes, which have excellent electrochemical properties and behave well with hydrogen as a fuel. Aqueous alkaline electrolyte systems have a low activation energy for the cell reactions. They therefore have high power output even at below-ambient temperatures. This type of fuel cell, however, is not suited for use as a truly regenerative cell (i.e. without an electrolyser) due to material problems. The alkaline fuel cell system with a separate electrolyser appears to be best for shorter missions (<5000 hours).

Solid polymer electrolyte (SPE): These fuel cells are fully regenerative (i.e. can operate in both charge and discharge modes). They are less massive than RFCs using an alkaline electrolyte, but also have lower efficiency. Studies have indicated that for longer missions (>1 year), the acid solid polymer electrolyte RFC may be most appropriate.

Solid oxide electrolytes: These materials are semi-permiable ceramics that have the ability to conduct an electric current by the passage of oxygen ions through the crystal lattice at sufficiently high temperatures. The electrolyte material used is mostly Zirconia. This type of fuel cell is fully reversible and will reach high specific energies (≥l kW/kg). The major problem is the design of an efficient thermal system to maintain the 1000C operating temperature.

4.3.4 Design Considerations

Designing a fuel cell power system involves determining the amount of power required (and thus the number of fuel cells needed) and the duration for which it is needed. This information is then used to determine the amount of fuel and oxidizer needed to run the fuel cells and the mass and volume of the storage tanks. In the case of a RFC system, the characteristics of the electrolysis system are determined by the amount of water that will be electrolyzed and the time allowed for regeneration. As a rule of thumb, the mass of the electrolysis system is usually assumed to be equal to 60-70% of the fuel cell mass. The characteristics of the primary power source needed to operate the electrolyser are determined by its input power and the allowable regeneration time.

A Fuel Cell System usually includes several parallel stacks, each stack being composed of several individual cells. Each cell is characterized by its

Electrode area, S (cm^2),
Current Density, I (mA/cm^2) (Space Shuttle=470mA/cm^2),
Output Voltage, V (usually between 0.7V and 1.0V),
Mass, m, and dimensions.

Given the number of stacks, N, the number of cells within a stack, n, the output power, P_{fc}, is given by

and the Mass, M_{fc}, by

Note that the mass of the subsystems (cooling, pumps, etc.) and controls must be added to determine the total system mass. It is usually estimated that these subsystems account for half the total mass.

An easier way to estimate the fuel cell mass is to use existing power density data. For the Space Shuttle system, the power density for 7kW continuous output power is 130W/kg (total fuel cell weight). Advanced alkaline systems are expected to provide up to 500W/kg. Using this data, the fuel cell mass is

.

4.3.5 Sample Regenerative Fuel Cell Sizing

As a rule-of-thumb, the mass of the electrolyser system is generally estimated to be equal to 60% of the mass of the fuel cell system, including all the connections between the two systems.

The mass of hydrogen consumed, $M(H_2)$ (kg/hour), for given power, P_{fc} (W), and time, T_e (hours), is given by this equation used by NASA for the Orbiter:

$$M_{H_2} = \frac{T_e}{2.2} \times \left[0.0833 \times P_{fc} + \left(0.338 \times 10 \times P_{fc}^2\right)\right],$$

and the mass of oxygen by

.

Considering cryogenic H_2 and O_2 contained in spherical tanks made of stainless steel type 420, 1 mm thick (7.765 kg/m^2), we can compute the volume and mass of the tanks:

$$V_{H_2}\left(m^3\right) = \frac{M_{H_2}}{7.0 \times 10^{-5}}$$

$$V_{O_2}\left(m^3\right) = \frac{M_{O_2}}{1.149 \times 10^{-3}}$$

$$V_{H_2O}\left(m^3\right) = 10^{-3} \times \left(V_{O_2} + V_{H_2}\right)$$

$$M_{tank}\left(kg\right) = 7.765 \times 4\pi \left(\frac{3}{4\pi} V_{tank}\right)^{2/3} .$$

We thus can compute the total mass:

Estimation of the power needed by the electrolyser:

where η_{fc}= efficiency of fuel cell,

η_{el}= efficiency of electrolyser,

T_{fc}= period of fuel cell operation, and

T_{el}= period of electrolyser operation.

The efficiencies of both the fuel cell system and the electrolyser are usually between 55% and 65%.

Estimation of the waste heat production rate of the fuel cell:

$$\dot{Q}_{fc} = P_{fc} \times \frac{1}{\eta_{fc}} \times \left(1 - \eta_{fc}\right) .$$

4.4 Battery Power Systems

Batteries have been in use for spaceflight applications since the flight of Sputnik. Since that time, batteries have matured from non-rechargeable one-use power systems to rechargeable multi-use backup power systems. In the early years of spaceflight, relatively short flight times encouraged the use of batteries as a primary source of power. As the mission durations grew longer, solar and nuclear energy took as the primary sources. This did not, however, close the door on battery use. In fact, this development sparked the need for batteries as secondary power sources. These batteries provided power when the primary source could not. For example, when a satellite with solar panels enters a period of eclipse, batteries provide power until the satellite emerges from the occultation. Even with nuclear power sources, there are circumstances under which the peak electrical load is greater than the normal operational load. Instead of designing a nuclear reactor that operates at the peak power, a battery subsystem can provide the excess power and the nuclear reactor can be downsized for normal operation. Batteries, therefore, remain an essential component of aerospace hardware. Much research is being done to further enhance battery characteristics. To best understand these characteristics, it is best to review the basic design of batteries.

4.4.1 Basic Battery Design

A battery consists of several subunits called cells. Each cell is an identical unit which can be considered a "black box" with a positive terminal and a negative terminal. Within the black box, the terminals connect to electrodes which reside within a bath of electrolyte. Energy is stored in the cell using an oxidation-reduction reaction. This reaction uses the interaction between the electrodes and the electrolyte in which electrons are transferred from the oxidation reaction at one terminal to the reduction reaction at the other terminal. This stored energy can then be tapped by reversing the reaction and letting the current flow through the load. The diagram in Figure 10 gives a schematic view of a battery cell.

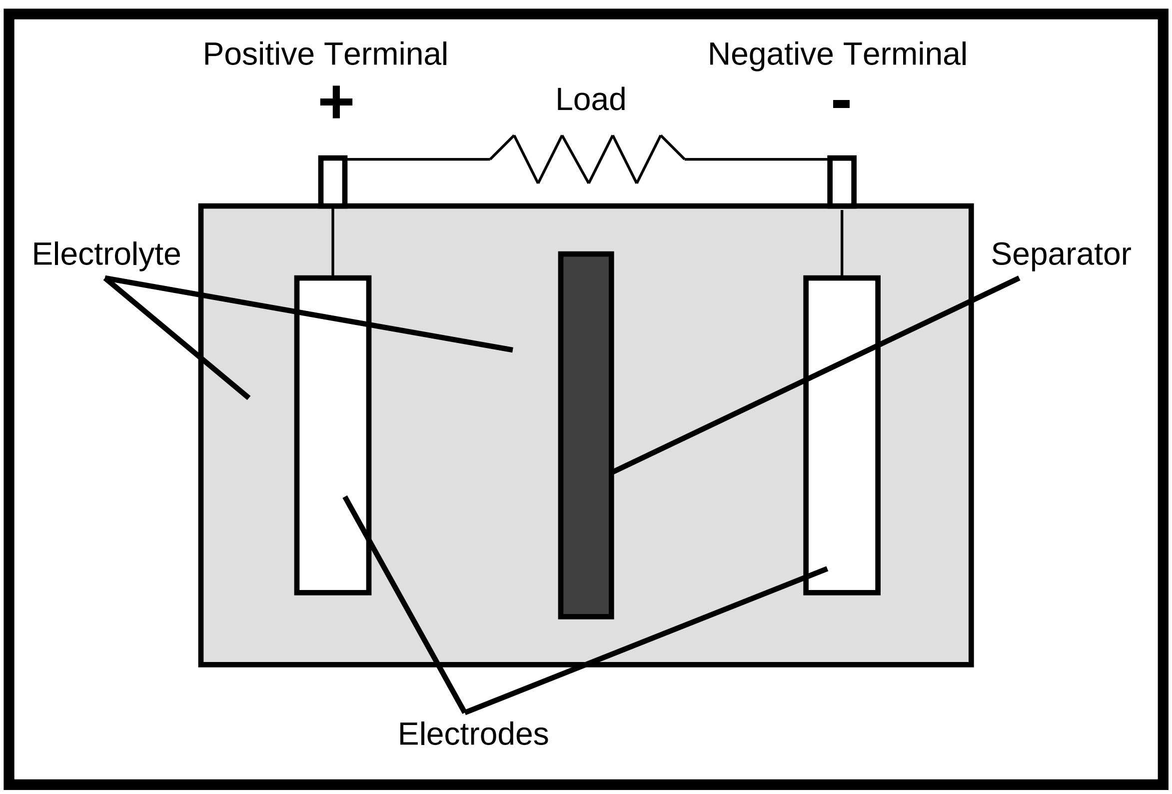


Figure 10. Diagram of a Typical Electrolytic Cell.

Batteries differ due to the materials used in the design. Different electrode-electrolyte combinations produce different oxidation-reduction reactions, and, therefore have different cell characteristics. Although the basic theory behind oxidation-reduction reactions has not changed for decades, the choice of materials has diversified the performance range of batteries on the market, and new designs have also made the old combinations more efficient.

4.4.2 Rechargeable and Non-rechargeable Cells

The first major development in the design of batteries for space applications was the development of rechargeable cells. This development actually came long before the flight of the first satellite. Their use, however, has changed the role of batteries, forever. Early batteries were silver-zinc (AgZn) non-rechargeable (primary) batteries. These provided the best compromise among several characteristics such as lifetime, energy density, and mass. Later, in the early 1960s, rechargeable AgZn batteries were used for the first time on the Ranger missions. Rechargeable AgZn batteries flew on the majority of spacecraft until the mid-1970s, when nickel-cadmium (NiCd) batteries superceded Ag-Zn. Today, NiCd still ranks as the most-used rechargeable battery for space applications, but another changing of the guard is close at hand. Nickel-hydrogen (NiH_2) batteries are proving to be the state of the art system and will likely become the chief rechargeable battery over the next decade or so.

The main reason rechargeable batteries have become dominant is obvious when we consider battery performance requirements. During most missions, batteries no longer perform as primary sources of power. Solar cells and radioactive thermoelectric generators are much more efficient primary power sources. Batteries act as secondary sources during periods when the primary source cannot generate sufficient power.

The following sections discuss the different characteristics of rechargeable batteries, present information on current systems, and detail a typical design problem for a satellite in low inclination, low Earth orbit.

4.4.3 Design Constraints

When beginning the design process, four major constraints tend to limit the range of design possibilities: required power, mission lifetime, system mass, and cost. The first two constraints, required power and mission lifetime, give an expected performance outline for our battery system. Many different types of systems can meet these requirements, so we must look to the last two constraints to choose two or three viable solutions. Many times, system mass and cost are related. If we look at the spacecraft as a whole, a more massive battery system can cause the launching cost to rise. However, less-massive battery types tend to cost more to develop and produce. Here, a design and cost tradeoff must take place. Once this compromise has been achieved, a detailed design can be made. The next step is to gain an understanding of the important characteristics of batteries.

4.4.4 Battery Characteristics

Batteries have many characteristics which further influence our design. We can group these characteristics into four main categories: electrical, charge, discharge, and cell lifetime characteristics.

The electrical characteristics pertain to those parameters determined completely by the oxidation-reduction reaction which takes place within the cell. These parameters include the nominal voltage, the capacity, the operating temperature, and the energy density. The nominal voltage is the average voltage which the cell produces. This quantity varies with each electrode-electrolyte combination, but most cells are designed to produce a voltage in the range of 1.1 to 2.0 volts.

The capacity is the amount of current, in amperes, that the battery can supply in one hour for a total discharge. For example, a 20 A-hr battery can supply 20 amperes for one hour or 5 amperes for four hours. This quantity is directly related to the size of the battery and the number of cells.

The operating temperature is the range of temperatures at which the cell performance varies the least. This temperature range relates directly to the temperature range at which the oxidation-reduction reaction is quickest and most efficient. Should the temperature drop below this range during use, the voltage, current, and energy density will drop off. Should the temperature rise above this temperature range, electrolyte and electrode decomposition can occur irreversibly. This can hinder later battery performance. In fact, should decomposition occur in great amounts, the battery may explode.

The final and most important electrical characteristic is the battery's energy density. This quantity represents the amount of energy (in watt-hours) that can be stored in the electrolyte-electrode system, per unit mass or volume. This value has a theoretical limit based on the energy component of the oxidation-reduction reaction. The limit is unattainable due to the non-ideal system components.

The charge characteristics of the batteries become important on missions with quick discharge and recharge times. An example of this scenario is a satellite in a low inclination, low Earth orbit (LEO). The satellite experiences an eclipse nearly sixteen times a day. The total eclipsed time is on the order of 720 minutes, with the recharge period being equivalent in length. The critical parameter is the charge current rate, C, which recharges the battery to 100% capacity in one hour. Fully recharging in an hour would be desirable in this case, but for some battery types this proves to be impossible. It has been found that charge rates of 1C or greater can limit the lifetime of a battery. More efficient cell designs have made it possible to charge batteries more quickly, but charge rates of several times C are rarely used.

The discharge characteristics pertain to the amount and rate at which the battery's stored energy is released. The discharge rate is similar to the charge rate. This rate is the speed at which a charge can be used and is also based on the parameter C. Most batteries can discharge at a rate of 10C. In fact, some batteries can discharge at a rate of 50C. This means that a battery can discharge its entire capacity in one-fiftieth of an hour. Another important discharge characteristic is the depth of discharge (DOD). This parameter is a measure of the percentage of the capacity that was used between charges. DOD can be decreased to increase cell lifetime, since high DOD shortens cell lifetime.

The final characteristic is cell lifetime, which is dependant on the electrical, charge, and discharge characteristics mentioned above. Cell lifetime is the number of discharge-recharge cycles that the cell can withstand before failure. It is therefore very important to have a good model of this parameter during the design process to ensure acceptable battery performance throughout the lifetime of the spacecraft.

4.4.5 Conventional and Exotic Systems

There are dozens of electrolyte-electrode combinations. The following list of systems is but a small fraction of the total, but it contains most of the systems currently in use in the space industry. Additionally, this list includes some next generation systems that are now under development.

1) Nickel-Cadmium: (NiCd)
Nickel-cadmium batteries have been used in the majority of space applications since the mid 1970s. This system possesses good energy density, operating temperatures, and discharge voltage characteristics. However, this system does not handle overcharging well, and it has a relatively short lifetime.

2) Nickel-Hydrogen: (NiH_2)
Nickel-hydrogen batteries have the strong points of NiCd batteries, with improved energy density and cycle lifetime. These two improvements have led to a dramatic increase in NiH_2 battery use. NiH_2 batteries, however, use hydrogen gas as a reactant in the oxidation reaction. During use, high pressures develop in the cell. Therefore, the cell case is a rigid pressure vessel. Research is underway to develop lightweight, inexpensive, reliable materials for use in NiH_2 cell cases.

3) Silver-Zinc: (AgZn)
The silver-zinc system was the first rechargeable system used in the United States space program. It is still in use today. The AgZn cell is practical, reliable, and inexpensive compared to other cells of similar performance. The cell possesses good energy density, but has a low cycle lifetime. This system is ideal, however, for short to medium length missions which require few discharge-recharge cycles.

4) Nickel-Zinc: (NiZn)
The nickel-zinc cell is a derivative of the silver-zinc system, using a more efficient and reliable nickel electrode instead of a silver electrode. Silver tends to migrate from electrode to electrode, whereas nickel does this to a much lesser extent. The performance for the NiZn cell is moderate but the initial cost is high.

5) Silver-Cadmium: (AgCd)
The silver-cadmium system is quite similar to the NiCd system mentioned earlier. The AgCd system does, however, provide higher

energy densities than the nickel counterpart. AgCd cells have a lower cycle lifetime than NiCd cells.

6) Lead-Acid: (Pb-Acid)

This system is best known for its use in automobiles. It is the least expensive system in this list, providing good performance for the dollar. Additionally, the nominal voltage per cell is high. The drawbacks, however, are significant. Pb-Acid batteries are quite heavy and have bad low-temperature characteristics.

7) Sodium-Sulphur: (NaS - exotic)

The NaS system is a member of a new cell type: solid electrolyte cells. This system uses the low conductivity of ß-aluminium and the high ionic conductivity of metallic sodium ions to produce a very high theoretical energy density. In this configuration, the electrolyte, ß-aluminium, is solid and the two electrodes are liquid sodium and liquid sulphur. However, this system has an operating temperature between 300°C and 400°C.

8) Lithium-Titanium Sulphide: ($LiTiS_2$ - exotic)

Lithium-titanium sulphide cells also belong to a new breed of battery cells: fast ion conducting solid systems. In this system, the solid titanium-disulphide electrode conducts lithium ions found in an organic-based electrolyte solution. This cell operates well at room temperature and offers high energy density for a very low mass.

9) Lithium-Iron Sulphide: ($LiFeS_2$ - exotic)

This, too, is a new battery type which uses a fused-salt electrolyte to provide higher energy densities than solid electrolyte types. A $LiFeS_2$ battery operates at a high temperature of 400° C, and has similar performance to sodium-sulphur batteries.

4.4.6 Typical Design Problem

The most common use of batteries is on satellites in low Earth orbit. As an example, let's design a battery system for a low inclination, low Earth orbit satellite using the method found in *The Design of Geosynchronous Spacecraft*, by Brij N. Agrawal. For the example, assume the following parameters:

1) Two independent buses
2) Constant bus voltage of 28 volts
3) Constant load of 1500 watts
4) 70% depth of discharge
5) 90-minute eclipse
6) Nickel-hydrogen batteries.

First, calculate the minimum battery discharge voltage and the number of cells. At the end of our mission, assume that the minimum cell voltage is 1.1 volts, and that at least one cell has failed. We want the minimum battery discharge voltage, V_{DB}, to be 28 volts. So,

$$V_{DB} = (N-1) \times V_D - V_{DD}$$

where V_D is the minimum cell discharge voltage and V_{DD} is the bypass diode voltage over the failed cell.
Using: V_{DB} = 28V, V_{DD} = 1.1V, and V_D = 1.1V, we find,

$$28\text{V} = (N-1) \times (1.1\text{V}) - (1.1\text{V})$$
$$N = 27.45 = 28 \text{ cells.}$$

Thus, V_{DB} = (28 -1) × (1.1 V) - (1.1 V) = 28.6 V.

The power per bus is equal to half that of the total power required:

$$P_B = (1500 \text{ watts})/2 = 750 \text{ Watts.}$$

Now calculate the capacity,C, in ampere-hours, of the battery that is needed to deliver the 750 W during the 90-minute eclipse.

$$C = (P_B \times t)/(V_{DB} \times DOD)$$

where P_B = bus power (750 W),
t = length of discharge (90 minutes),
V_{DB} = minimum battery discharge voltage (28.6V), and
DOD = maximum depth of discharge (70%).

Substituting the given values into the equation,

$$C = (750\text{W} \times 1.5\text{hours})/(28.6\text{V} \times 0.70)$$
$$C = 56.19\text{A-hr.}$$

Let's find an approximate mass for our two batteries. Assuming an energy density, E_D, of 60W-hr/kg, we find that each battery has a mass, M_B, of

$$M_B = (C \times V_{DB})/(E_D)$$
$$M_B = (56.19\text{A-hr} \times 28.6\text{V})/(60\text{W-hr/kg})$$
$$M_B = 26.78\text{kg.}$$

So our battery system has an approximate mass, M, of

$$M = 2 \times M_B$$
$$M = 53.56\text{kg.}$$

4.4.7 Summary

Battery design has come quite a long way since the flight of Sputnik. Today, designers have a plethora of battery systems to choose from. As power needs and lifetime requirements increase, battery technology will have to advance even further. Surprisingly enough, most of the electrical needs of the next decade's spacecraft can be met with existing systems. Minor improvements will maximize the efficiency of current designs. However, as the space program pushes forward into the next century with its exploration and settlement of the solar system, large capacity high energy batteries must be developed.

This small discussion on the state of battery design is meant as but an introduction to the subject. Since the field is advancing rapidly, much of this information may become outdated quickly. Concepts behind the development of batteries will not change, but the implementation of the concepts will vary as new discoveries are made with new electrode-electrolyte combinations. Therefore, the best source of information concerning batteries would be from battery manufacturers, power related journals, and summaries of battery symposiums.

4.5 Radioisotope Power Systems

Radioisotope power systems convert the heat energy of decaying radioactive material into electricity. Use of these systems has constraints not associated with other power sources, but radioisotope power is compact and removes dependence on the Sun.

4.5.1 Radioisotope Power Generator Systems

Up to the time of the writing of this paper the United States has launched 41 Radioisotope Thermoelectric Generators (RTGs), on 23 spacecraft (see Table 4). The power production ranges from 2 watts to 300 watts. The RTGs use Plutonium-238 with an 87.7-year half life. Table 5 lists some characteristics of the radioisotope power systems.

Table 4. United States RTG History.

RTG	Spacecraft	Mission	Launch Date	Status
SNAP-3A	Transit 4A	Navigational	06/29/61	Earth orbit
SNAP-3A	Transit 4B	Navigational	11/15/61	Earth orbit
SNAP-9A	Transit 5BN1	Navigational	09/28/63	Earth orbit
SNAP-9A	Transit 5BN2	Navigational	12/05/63	Earth orbit
SNAP-9A	Transit 5BN3	Navigational	04/21/64	Aborted
SNAP-19B2	Nimbus B-1	Weather	05/18/68	Retrieved
SNAP-19B3	Nimbus III	Weather	04/14/69	Earth orbit
SNAP-27	Apollo 12	Lunar	11/14/69	Lunar surface
SNAP-27	Apollo 13	Lunar	04/11/70	Retrieved
SNAP-27	Apollo 14	Lunar	01/31/71	Lunar surface
SNAP-27	Apollo 15	Lunar	07/26/71	Lunar surface
SNAP-19	Pioneer 10	Planetary	03/02/72	Extra-solar
SNAP-27	Apollo 16	Lunar	04/16/72	Lunar surface
Transit RTG	Triad-01-1X	Navigational	09/02/72	Earth orbit
SNAP-27	Apollo 17	Lunar	12/07/72	Lunar surface
SNAP-19	Pioneer 11	Planetary	04/05/73	Extra-solar
SNAP-19	Viking 1	Planetary	08/20/75	Mars
SNAP-19	Viking 2	Planetary	09/09/75	Mars
MHW	Les 8/9	Communication	03/14/76	Earth orbit
MHW	Voyager 1	Planetary	08/20/77	Extra-solar
MHW	Voyager 2	Planetary	09/05/77	Extra-solar
GPHS	Galileo	Planetary	10/18/89	Jupiter (in transit)
GPHS	Ulysses	Planetary/Solar	10/06/90	Solar orbit

Table 5. RTG Specifications.

RTG	Power (W)	Mass (kg)	Specific Power (W/kg)	Lifetime (yrs)	Thermocouple efficiency	Voltage (V)
SNAP-11	25					
SNAP-19	35			1		
SNAP-27	73	19.7	3.7	1		
Module	20.5	2.2	9.4	(8)	7.6% (SiGe)	28
GPHS	285	56.2	5.1	(8)	7.6% (SiGe)	(30)
18 GPHS	340	44.2	7.7	(8)	7.6% (SiGe)	30.8

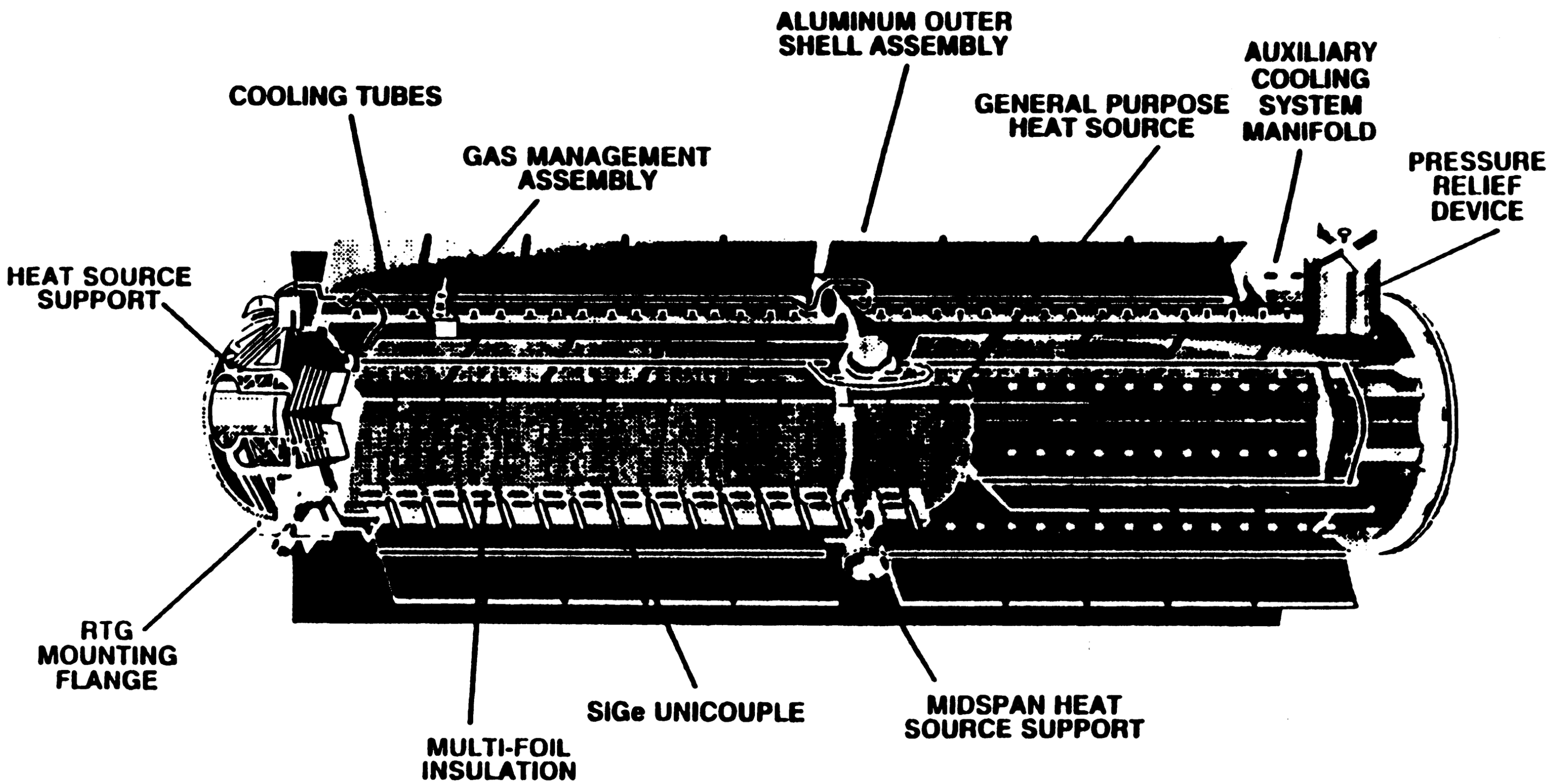
Parenthases indicate estimated values.

RTGs were used on Apollo missions to power and heat systems and sensors on the Lunar surface, and were deactivated after the astronauts departed the Moon. RTGs powered systems on the LEM until just before ascent to Lunar orbit. The RTG's were removed and placed in seismic sensors left on the Moon. Apollo 13 kept their RTG until reentry due to emergency conditions. The fuel cask survived reentry and was abandoned in a deep part of the Pacific Ocean.

Thirty-five watt RTGs were installed on the Viking landers. The waste heat was tapped to maintain instrument temperature. Due to the ferocity of Martian dist storms, the RTGs had to be shielded from excess convection and shield erosion by using wind screens.

4.5.2 The Genral Purpose Heat Source

The Genral Purpose Heat Source (GPHS; see Figure 11) is currently being used on the Galileo and Ulysses missions. The 18 GPHS (Figure 12) is presently untested. They take advantage of modular construction techniques, allowing each RTG to be sized according to the mission's requirements. Power requirements are met by installing the necessary number of modules.



- **POWER OUTPUT** - 285 WATTS
- **FUEL LOADING** - 4400 WT; 132,500 Ci
- **WEIGHT** - 124 LBS
- **SIZE** - 16.6 IN × 44.5 IN

Figure 11. General Purpose Heat Source Radioisotope Thermoelectric Generator.

ILLUSTRATIVE GENERATOR

14 Slices, 288 Watts, 68 Lbs. 4.25 W/Lb

MODULAR SLICE

(20.5 Watts at 28 Volts)

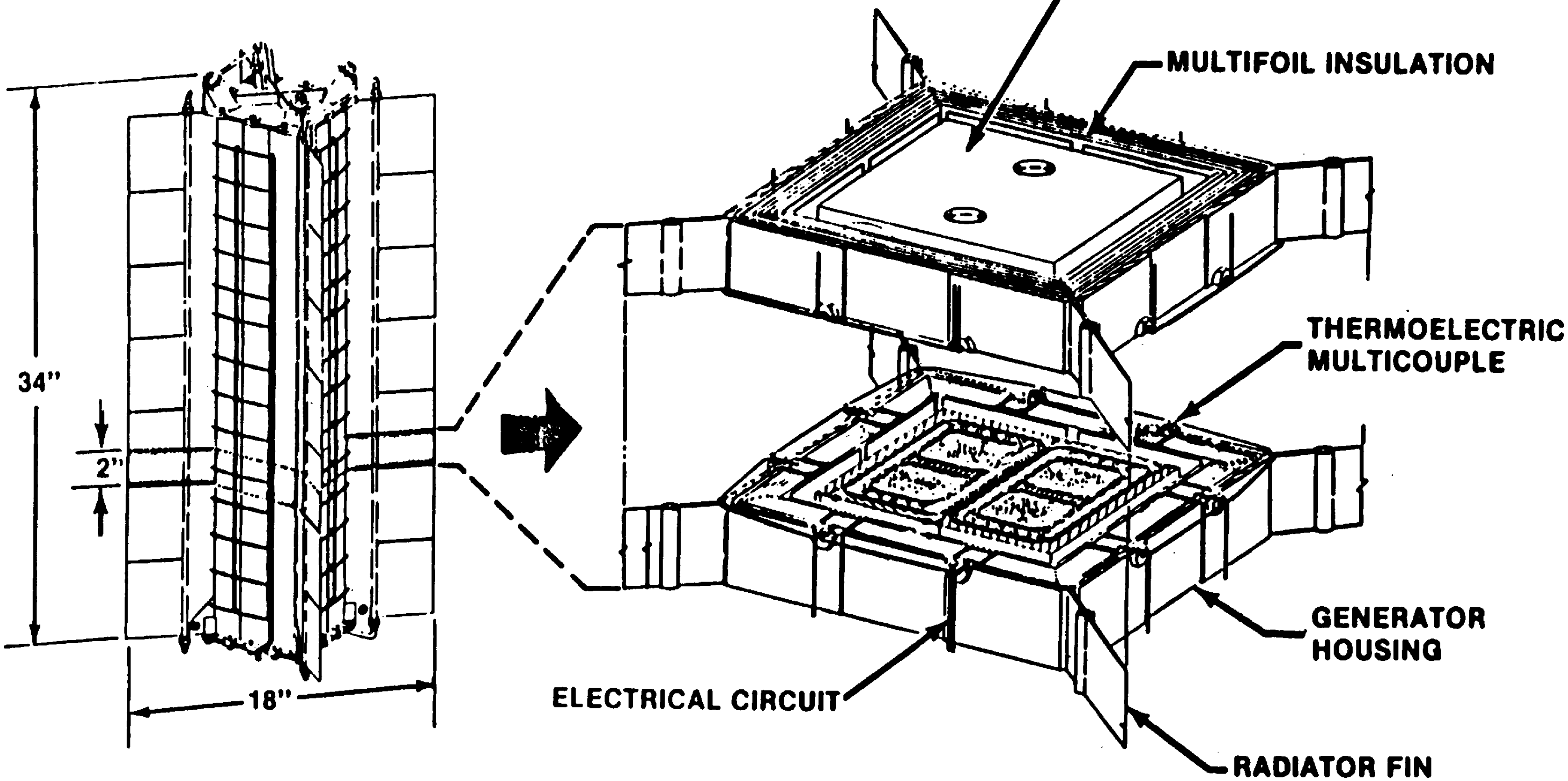


Figure 12. Modular Isotopic Thermoelectric Generator.

The later RTGs built by the United States were designed to survive reentry into Earth's atmosphere. RTGs produce abundant waste heat which can be used to maintain proper temperatures for spacecraft electronics. They produce approximately nine times more heat than electrical power. RTG power curves are unaffected by the spacecraft's environment. The spacecraft must be designed specifically to fully utilize an RTG. For a given spacecraft, RTGs have a higher power-to-weight ratio than a solar array/battery system. RTGs have a specific power of 7.7W/kg, excluding the power savings incurred from the displaced electric heaters for thermal control. General RTG specifications are given in Table 6. RTGs also have a high reliability record. The cost of an RTG can be in excess of $20000 per watt. Procurement of an RTG requires a minimum of 5 years from budget approval to delivery. Intraagency agreements between the Department of Defense, Department of Energy, and NASA, and Presidential approval are required before production.

Table 6. Specifications for Department of Energy RTG Design (1992).

RTG Module	
Fuel mass	250WT
Voltage	28V
Power	20.5W
Mass	2.2kg
Specific power	9.4W/kg
Thermocouple type	Silicon-Germanium
Thermocouple efficiency	7.6%
Number of thermocouples	8
Fuel pile	PuO_2/Iridium/Graphite
18 GPHS	
Number of modules	18
Voltage	30.8V
Power	340W
Mass	44.2kg
Specific power	7.7W/kg
Length	1.08m
Diameter	0.33m
Thermocouple type	Silicon-Germanium
Thermocouple efficiency	7.6%
Number of thermocouples	144

Typically the user pays for:

RTG design and analysis

Converter production line qualification and production
Fabrication and assembly
RTG fueling
RTG acceptance and testing
RTG ground support equipment
RTG models and simulators
Qualification unit design, hardware and testing
Interface support
Ground safety analyses
Plutonium-238 production and processing.

The Department of Energy provides:

Technology development and conceptual design
Engineering unit design, hardware and testing
Heat source development and qualification
Heat source production line qualification
Safety analysis reports
Quality assurance
Facility maintenance and general purpose equipment
RTG shipping containers, safety analyses report for packaging transportation
Emergency Response
Program Management.

4.5.3 Fuel Availability

The Plutonium-238 is produced by irradiating Neptunium-237, which is itself a synthetic element. The Pu-238 isotope can only be produced at the Savannah River Reactor. At one point, the entire plutonium stockpile had been committed to the CRAF/Cassini missions. Fuel production alone requires a time lead of at least 30 months excluding start up time for the reactor. The reactors are active only when required for fuel production. The Savannah River Plant can produce Pu-238 at 15kg per year, which is roughly three 300-watt RTGs per year. Nuclear weapons-grade plutonium cannot be used in RTGs.

4.5.4 Designing a Spacecraft for RTGs

For political and environmental reasons, the United States avoids the use of radioactive power and heat sources in Earth orbit. RTGs are ideal for deep space missions where the probe's trajectory carries it too far from the Sun for a useful solar array / battery system. RTG construction requires a lead time which depends on a variety of bureaucratic and logistical constraints. Users should assume a lead time of at least six years from budget and intraagency approval to RTG delivery.

GPHS RTGs employ a gaseous cooling system in conjunction with radiator fins to provide the temperature gradient for the thermocouples. The heated gas must be pumped away from the RTG assembly. This gas may be piped to other systems requiring temperature control. The excess heat must be dissipated before the gas returns to the RTG. Mounting flanges and other structural supports can also be used to conduct heat away from the RTG. In addition, interference from electromagnetic radiation emitted by the RTG can be avoided by simply installing the unit on an extendable boom. Interference is least along the longitudinal axis of the GPHS.

The GPHS RTG systems are modular in construction. The power requirements determine the number of modules needed per generator unit. Each module produces 20.5 watts at 28 volts. The number of modules multiplied by the power per module gives the total power production of the RTG. Be cautioned that the voltage remains relatively constant, independent of the number of modules. The RTG voltage is the module voltage. In addition, the voltage is a function of the temperature gradient existing at the thermocouples. Hence, the greater the temperature gradient the higher the output voltage. An increase in the temperature gradient does not necessarily produce a power increase. Higher operating temperatures decrease thermocouple efficiency due to difficulties in maintaining proper spacing between the cathodes and anodes of the thermocouples. Higher operating temperatures do, however, allow for dramatically lighter radiators for a given temperature gradient. Therefore, a compromise must be made between a lightweight radiator system and a higher thermocouple efficiency.

4.6 Large-Scale Space Nuclear Generators

We will classify large-scale nuclear power systems as those systems that can generate at least 100 kilowatts of power. This requirement is established to differentiate these systems from RTGs, which are usually constrained to power output in the 30 to 500 watt range. These systems may be *quite* large, not differing much in mass and complexity from their terrestrial counterparts. Three areas of interest are the components of the nuclear power plant, the three major thermodynamic cycles used in space nuclear systems, and aspects of software development.

4.6.1 Nuclear Power Plant

Every nuclear power plant has a core which consists of fuel and moderator materials. In most cases the fuel is uranium. The moderator is the material that slows down neutrons produced by the fission reactions. "Slow" neutrons are the most effective for continuing the fission process. The core is placed within a pressure vessel (see Figure 13), which contains the coolant and supports the core. The pressure vessel is usually a welded steel vessel or prestressed concrete lined with steel. Shielding is very important in space nuclear power because of radiation's effects on humans and electronics. The shielding surrounds the reactor to prevent neutrons and gamma radiation from interacting with biological and electronic systems. Some shields are made of concrete several feet thick.

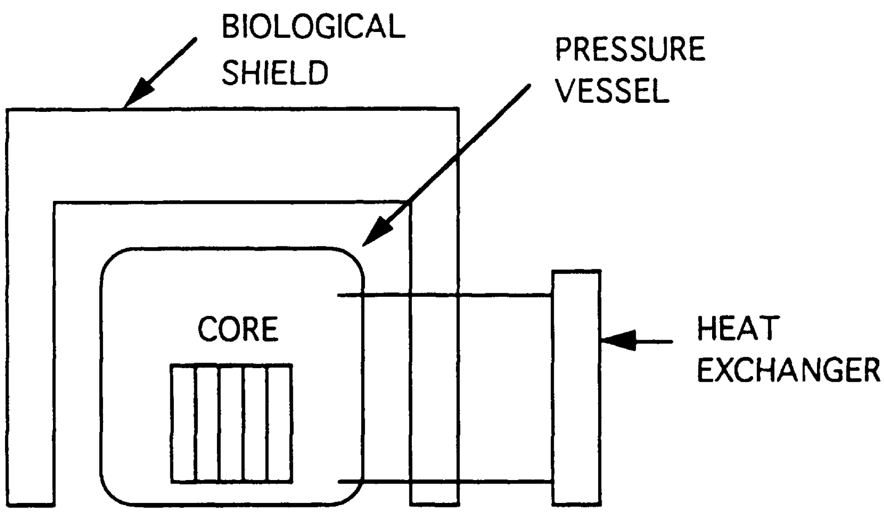


Figure 13. Diagram of a Nuclear Reactor.

The mass and control of a reactor is also important in designing an energy system. The mass of a multimegawatt system using a Rankine cycle is about 35 metric tons. The control system produces changes in reactivity for start-up, power output control, and shut-down of the reactor. Changes in reactivity during operation are small, but the system must also be capable of shutting down rapidly.

4.6.2 Thermodynamic Cycles

There are three major thermodynamic cycles used in space nuclear power: the closed Brayton cycle, the open Brayton cycle, and the Rankine cycle.

Using the closed Brayton cycle (Figure 14), the heat source (reactor heat exchanger) heats a high-pressure gaseous working fluid to the maximum cycle temperature. This gas expands through a turbine, driving the compressor and alternator. The turbine discharge gas is cooled through the recuperator heat exchanger and the radiator. The low-pressure, cooled gas passes from the radiator into the compressor. The compressor raises the gas to highest cycle pressure and passes it to the recuperator for preheating. After the recuperator, the gas goes back through the heat source, closing the cycle. This system produces electrical power at a specific level on an almost continuous duty cycle. The electrical output for the system is around 3.1kW/kg.

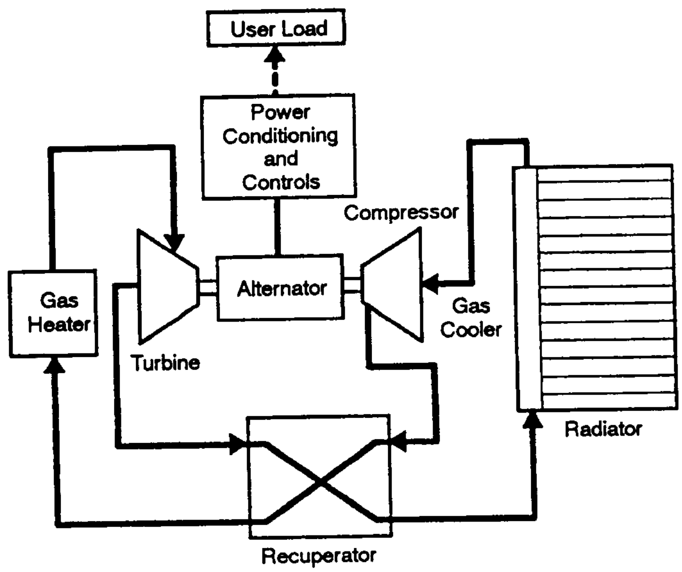


Figure 14. Closed Brayton Cycle [Harty].

The open Brayton cycle is like the closed cycle, but the turbine exhaust is expelled after passing through the recuperator. The main space-nuclear application for this thermodynamic cycle is for cases where large changes in power are needed almost instantly (weapon systems). The output for this system is around 3.8kW/kg.

The Rankine cycle (Figure 15) also expands the working fluid through a turbine, driving the generator. The turbine exhaust passes into a condensor/radiator, and from thence back into the heat exchanger (boiler). The electrical output is about 3.8 kW.

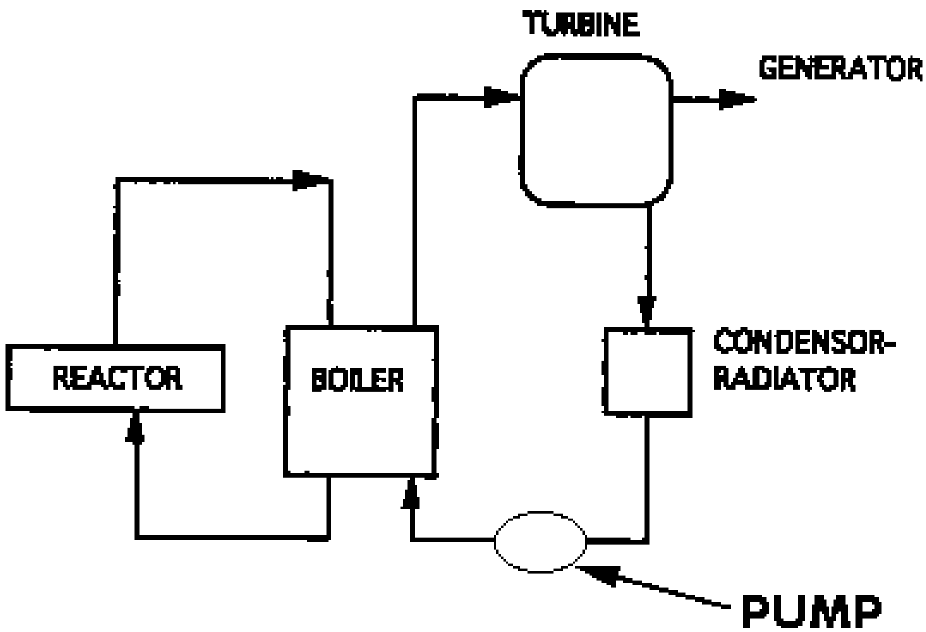


Figure 15. Rankine Cycle.

4.6.3 Computer Models

The generation of accurate design software is quite difficult. Garrett Fluid Systems Division has developed software for space power systems. The Space Power Unit Reactor (SPUR) program considers parameters such as boiler design, condensor design, radiator design, multistage turbine, bearing design, and pumps. Reactor core neutronics may be modelled using the TWODANT code. The models attempt to produce high-fidelity simulations by considering the thermodynamic cycles, heat transfer characteristics, and the nuclear reactor itself.

4.7 Microwave Power Beaming

Electrical energy can be transmitted through space as microwaves. Raytheon demonstrated a microwave-powered helicopter in 1964, with some tests lasting 10 hours. The Goldstone experiment transmitted 34 kilowatts over a distance of 1.54 kilometers. Power beaming technology is still in the research stage. Power beaming is a cheaper and less-massive alternative to solar- and nuclear-dynamic systems. The devices required for power beam reception yield a higher specific power than a liquid metal nuclear reactor. Power beaming may provide spacecraft power in the kilowatt range.

5 Areas for Further Study

Emphasis should be placed on fuel cells. Section 4.3 could be expanded with statistical data on the past history of fuel cells and a description of the electrochemical mechanisms within a fuel cell. Hybrid systems must also be considered. Combinations such as battery/fuel cell, battery/solar cell, or fuel cell/RTG exist in past and current spacecraft. Spacecraft powered by multiple systems require special regulators to maintain proper power distribution.

Microwave power beaming may prove to be a high-power lightweight source for 1kW to 100kW needs. This subject requires additional research into current developments. Since the revival of interest in Mars missions, nuclear reactors have been the focus for propulsion. This caused a revival in research into lightweight reactors employing liquid-metal working fluids. This new data must also be included in a meaningful survey of space power systems.

6 Acknowledgements

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