Astronautics (SESA2024)

Section 8: Electrical power – suggested solutions

1. ***Solar cells:*** Large areas of semi-conductor material (solar panels) are presented to the Sun, and the incident power is converted to electricity by the photovoltaic effect. Solar arrays are very commonly used on Earth-orbiting spacecraft, where the incident power level is abundant. See recommended text.
Solar dynamic: A solar concentrator is used to heat a working fluid, which is then used to drive a turbine, which in turn drives an electrical generator to produce power. These systems require mechanisms for operation, and are generally of significant mass, although efficiencies are higher than for solar arrays. This type of system has not been commonly used. See recommended text.
Radioisotope Thermoelectric Generators (RTGs): Passive devices containing a radioactively decaying isotope (such as Plutonium 238) which produces heat. This thermal energy is converted to electricity through the Seebeck effect – in other words by using a thermocouple. These sources are often used for missions that travel long distances from the Sun. RTGs are difficult to handle, and have a detrimental impact on the environment in the launch vehicle, due to heat and harmful radiation emission. Over recent years, the 'Green lobby' has protested about launches carrying RTGs, in case a launch failure spreads the radioactive material in the atmosphere and on the ground. See recommended text.

2. Missions lasting a few hours – batteries. Missions lasting several years – solar arrays; RTGs if spacecraft is distant from Sun; nuclear reactor for high power demand.

3. Solar power available at 1 AU is around 1.4 kW/m^2 (1 AU = 1 Astronomical Unit = the mean Earth-Sun distance = 1.496×10^8 km). This value (1.4 kW/m^2), called the 'solar constant', drops off as the inverse square of the distance from the Sun. The most distant missions using solar arrays to date are of the order of 5 AUs from the Sun, where power flux is approximately $1400\ \mathrm{W/m^2}/(5)^2 \approx 55\ \mathrm{W/m^2}$. Taking account of inefficiencies of arrays, this means about 5 or 6 W of electrical power generated per square metre of array at these distances, which in turn means a requirement for very large arrays. Beyond about 5 AU, RTGs are commonly used.

4. The efficiency of the solar cell drops as the temperature increases – see notes – for example, a 25^o C rise in temperature for a silicon cell gives a decrease in efficiency of around 10%. In eclipse, the array becomes very cold, maybe down to –150^o C depending on the duration of the eclipse and the thermal properties of the array material. When such a cold array exits eclipse its efficiency is enhanced, and power levels are high until the array reaches normal sunlit temperature (typically ~ 50^o C) again.

5. High energy particle radiation (from the Van Allen Belts, solar wind, cosmic radiation, etc.) has the greatest effect in reducing power output over the long

term, by virtue of the damage it causes to the structure of the semi-conductor material at the atomic level. A cover-glass is adhered to the upper cell surface to reduce the effects of radiation damage.

6. Refer to ‘A series – parallel solar cell circuit’ in notes. A typical silicon solar cell under Earth orbit illumination (at around 30^{o} C) will produce an open circuit voltage of around 0.5 V, and a short circuit current of around 35 mA/cm^2. To produce the voltages and currents demanded by a spacecraft’s electrical loads, individual cells are wired in a series/parallel arrangement, called strings. The required voltage is produced by having the appropriate number of cells in series, and the require current is produced by having an appropriate number of cells wired in parallel. In other words, the cell’s output current is multiplied by the number of cells connected in parallel, and its voltage output by the number of cells in series.

7. The MPP is the point on the solar cell’s current I vs. voltage V characteristic, where the power $P = VI$ rectangle under the curve has maximum area. MPP tracking is achieved by adjusting the electrical load across the array (due to the S/C systems, and the shunt regulator – see block diagram in the notes) to ensure the cells operate at the MPP.

8.

The ***total capacity*** C of a battery defines the current it can supply over a particular time period for 100 % discharge. For example, a battery with a capacity of 50 A – h (Ampere – hours) can provide a current of 50 A for one hour, or a current of 1 A for 50 hours.
The ***energy density*** $\bar{\varepsilon}$ of a battery is the total stored energy (in W – h) per kg of its mass.
The ***depth of discharge*** DoD of a battery is the percentage of battery capacity used in the process of discharge. For example, a depth of discharge of 40 % means that 60 % of the capacity remains. The DoD is related to the number of discharge cycles (battery lifetime) for a particular battery technology.

9. Refer to recommended text or other texts on battery technology. The packaging of NiCd battery cells is illustrated in the notes. This has a high volumetric efficiency due to the convenient rectangular shape of each cell. On the other hand, NiH_2 battery cells are cylindrical pressure vessels, with hemispherical end caps. The packaging of these units leads to a low volumetric efficiency.

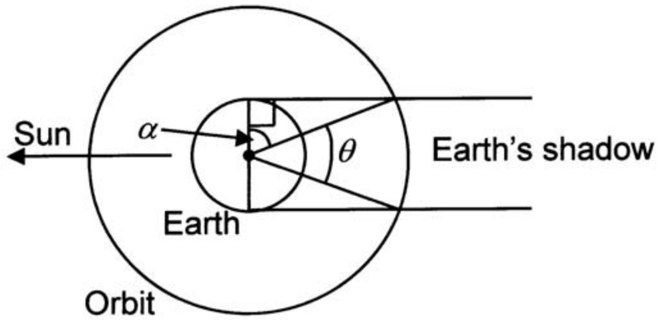
10. The worst-case eclipse condition is with the Earth-Sun vector in the orbit plane – see diagram over page. The angle α is given by

$$\cos\alpha = R_E / R_{GEO} = 1/6.611 \;\Rightarrow\; \alpha = 81.3^{o}$$

Also from geometry in diagram, we have $180^{o} + 2\alpha + \theta = 360^{o}$, so that

$\theta = 180^{o} - 2\alpha$. This gives the eclipse period $t_{ecl} = \left(\dfrac{180^{o} - 2\alpha}{360^{o}}\right)\tau$, where $\tau =$ 23.935 hours is the orbit period. Hence, we have the ***eclipse period***

$t_{ecl} = 1.157$ hours, and the sunlit period $t_{sun} = \tau - t_{ecl} = 22.778$ hours.



Number of charge-discharge cycles over 10 years is $n = 10$ years$/\tau = 3660$

To estimate battery capacity and mass for NiCd system, size the batteries for eclipse period discharge to supply 8000 W for payload and subsystem operation:

calculations for NiCd

NiCd system

DoD = 40%

Capacity $C = \dfrac{(8000\text{ W})(1.157\text{ h})}{0.4(27.5\text{ V})}$

$= 841.45\text{ A - h}$

Stored energy ε

$= (841.45\text{ A - h})(27.5\text{ V})$

$= 23140\text{ W - h}$

Battery mass

$= \dfrac{23140\text{ W - h}}{30\text{ W - h/kg}}$

$= 771.3\text{ kg}$

To estimate the array size, we need to know the power requirement, in addition to the 8000 W, for charging the battery while the S/C is in sunlight. Battery charge rate in sunlight is

$R = (DoD)C / t_{sun} = 0.4(841.45)/22.774 = 14.779$ A for the NiCd system

Required EOL power $P_{EOL} = 8000 + RV_A = 8000 + 14.779(33) = 8487.7$ W.

Therefore ***area of array***,

$A = P_{EOL} / (S \cos\delta\theta \;\; \eta \;\; \eta_p (1 - D_0)) \approx 84\text{ m}^2$.