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Solar Arrays and Battery Power Sources Conceptual Design for Low Earth Orbit Microsatellites

Mohammed Bekhti, Messaoud Bensaada

Algerian Space Agency, Centre for Satellites Development, Research Department in Space Instrumentation, Oran, Algeria

Email address:
mbekhti@cds.asal.dz (M. Bekhti)

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Abstract: The power system is a vital subsystem in a spacecraft. As long as the spacecraft has power, it can perform its mission. Almost all other failures can be worked out by ground operations from ground stations but a power loss is very fatal for the spacecraft. In the early years of spaceflight, the power system was also the limiting factor in any mission duration. Many studies show that solar cell power (short-circuit current and open-circuit voltage) are degraded by space environment radiation. The power system is designed such that the end of life (EOL) power is adequate for the mission's requirements. Beginning of life (BOL) power is set by the estimate of the radiation damage over the spacecraft's lifetime. It is well known in the literature, the radiation damage to solar cells is caused by high-energy protons from solar flares and from trapped electrons in the Van Allen belt. The purpose of this paper is to investigate the power system design trades involved in the mission analysis of a low earth orbit (LEO) satellite at an altitude of 700 km. Based on the power requirements of the payload and the constant power requirements for the remainder of the spacecraft (platform subsystems), the solar arrays and batteries for the spacecraft will be sized.

Keywords: Eclipse Time, Solar Arrays, Battery, Battery Capacity, Nickel-Cadmium, Lithium Ion, Nickel Metal Hydride, Primary Source, Gallium Arsenide, Solar Cells Efficiency, Fill Factor

1. Introduction

A satellite power system consists of four main units: a power source, a power storage, power conditioning and distribution units. For a satellite orbiting the earth, the most common method to obtain energy is using solar arrays where electricity is generated by the photovoltaic effect using sun rays. The produced power generated by the solar panels can be either used directly or saved in the power storage unit which is basically consisting of space qualified batteries.

The maximum eclipse duration is a key parameter for defining the power system characteristics. It simply gives how long the satellite will not be able to produce power by solar panels and the power system needs to be fed by the batteries. The altitude of the orbit and therefore the orbital period time is also an important parameter showing how long the solar panels can produce energy by subtracting the eclipse time from it.

To enable various missions, low earth orbit (LEO) satellites are required to fulfill their requirements on power consumption to sustain their life in space. As typical ones use photovoltaic cells to generate solar power, the presence of a sunlight phase is important. Apart from the issue of having the sun to generate power, a greater concern shall also be placed on its eclipse duration, when the power requirements for the spacecraft rely only on the batteries. [1-5].

2. Orbital, Eclipse and Sunlight Periods

The orbital period and the fraction of time a spacecraft is in sunlight (and eclipse) are of a fundamental importance to the design of both the thermal and the power systems. These periods determine the number of battery discharge cycles, a major parameter determinant of the battery lifetime. The orbital period T derived from Newton's formulation of Kepler's third law is reproduced here as equation (1). It is independent of the orbit eccentricity and is useful for circular and elliptical orbits. [1-6].

$$T \cong 2.\pi\sqrt{\frac{a^3}{\mu}} = 1.6285x10^{-4}.a^{3/2} \quad (1)$$

Where:

T: orbital period (minutes),

μ: $3.986005x10^{14}$ m^3/s^2

a: orbit semi major axis = R_E+orbit altitude (circular orbits)

R_E: radius of the earth≈6378.137 km.

Figure 1 shows, for circular orbits, the geometry for calculating the fraction of time in sunlight and eclipse for this “minimum sun” case. Consider that the Sun is in the orbit plane, its rays are essentially parallel, and the terminal rays are tangent to the earth. A right triangle is then formed with the earth’s radius RE as one leg and a hypotenuse of length equal to the sum of RE and the orbit altitude A. These observations lead to the equations for the fraction of time in sunlight and eclipse, which are given respectively in equations (2) and (3). [1-5].

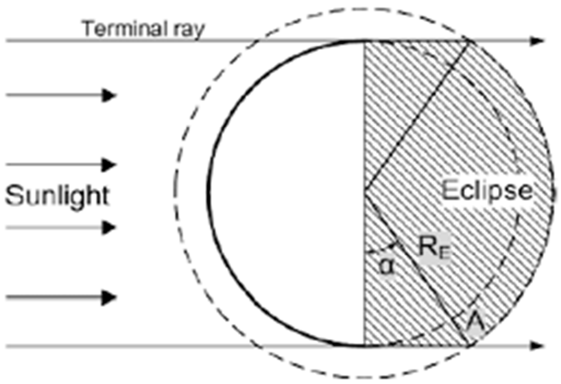

Figure 1. *Geometry for the minimum fraction of time in sunlight (maximum eclipse).*

A= orbit altitude

R_E = radius of the Earth = 6378.137 km.

$$\alpha = \cos^{-1}\left(\frac{R_E}{R_E + A}\right)$$

$$\text{fraction of time in sunlight (FTS)} = \frac{180+2\alpha}{360} \quad (2)$$

$$\text{fraction of time in eclipse (FTE)} = \frac{180-2\alpha}{360} \quad (3)$$

Assuming an orbit altitude A=700 km, equation (1) gives the orbital period equal to T = 98.7629 minutes. Equations (2) and (3) give respectively the fraction of time the spacecraft is in sunlight and eclipse:

1. Fraction of time the spacecraft is in sunlight = 64.28%
2. Fraction of time the spacecraft is in eclipse = 35.72%

Now that the orbital period T and the fraction of time the spacecraft is in eclipse are known, the eclipse period of time is FTE = 35.28 minutes. [1-5].

3. Power Requirements Estimation

Table 1 summarizes the satellite’s payload and bus power consumptions with margins. These margins are taken from the published literature and must be revised each time a subsystem’s power consumption is precisely tuned. Duty cycle values are expressed as a ratio to full orbit duration, ie, 98 minutes. [6-7]

Table 1. *Preliminary power requirements for the satellite's mission.*

Payload power requirement	Duty cycle %	Average power value W	Peak power value W
Camera	(2 minutes) 2%	1	3
GPS	100%	3	6
Communications	(5 minutes) 5%	5	5
Bus power requirement			
Thermal	0	0	0
Structure	0	0	0
Propulsion	0	0	0
ADCS	100%	3	5
OBC	100%	2.5	2.5
Mass memory	10%	4	4
TM/TC	100%	0.5	0.5
Power	100%	5	5
Margins 5 - 15%		1.2	4.7
Total power requirement		25.2	35.7

4. Solar Arrays and Battery Design Considerations

4.1. Selection of Solar Cells

The three major types of solar cells available in the market are made of Silicon (Si), Gallium Arsenide (GaAs) and multi-junction (triple junction). Table 2 summarizes the characteristics of selected solar cell technologies under standard measurement conditions of 28°C and AM0 sunlight. The three cell types range in efficiency from 15% for silicon to 28% for triple junction (multi-junction) cells. All of these are currently available and have been used on different missions. Note that cell efficiencies are traditionally referenced to 1358 W/m^2. [8-11]

Table 2. *Si, GaAs and multi-junction solar cells characteristics at 28°C and AMO.*

Solar cell parameter	Si (10Ω-cm)	GaAs (single junction)	Triple junction (multijunction)
Jsc (mA/cm^2)	42.5	30	17.0
Jmp (mA/cm^2)	39.6	28.5	16.2
Voc (V)	0.605	1.02	2.66
Vmp (V)	0.5	0.9	2.345
Pmp (mW/cm^2)	19.8	25.5	37.9
FF (fill factor)	0.77	0.82	0.83
Efficiency (%)	15	19.8	28.0
Degradation/year (%)	3.75	2.75	< 2
Radiation Degradation ratio @ 10^{15} MeV	0.71	0.77	0.86

From table 2, it can clearly be seen that the performances of GaAs and triple junction solar cells are far better than those for silicon solar cells. [13-17]

4.2. Solar Arrays Sizing for Spacecraft Power Requirements

The first step of a solar array sizing consists in making an estimation of the total power requirement for the solar arrays, Psa. This can be achieved using equation (4): [8-10]

$$P_{sa} = \frac{\frac{P_e \cdot T_e}{X_e} + \frac{P_d \cdot T_d}{X_d}}{T_d} \qquad (4)$$

Where: P_e and P_d are the spacecraft power requirements during eclipse and daylight, respectively. In the current study, P_e and P_d are taken equal to 30W (orbit average power) and T_e and T_d are the length of eclipse and daylight periods. P_e and P_d are assumed to be equal to the peak load (P_l) since there is no big difference between mission requirements between day and night. X_e and X_d represent path efficiency from solar panels to batteries which is around 0.65 and 0.85 for a peak power tracking system. [8-10]

There is power decay in time due to solar cells degradations. This means at the end of the mission, the total power availability, P_{EOL}, is going to be lower than the initial power availability, beginning of life (P_{BOL}) which can be estimated by equation (5): [13-17]

$$P_{BOL} = P_o . I_d . \cos(\theta) \qquad (5)$$

Where θ represents the sun incidence angle between the vector normal to the surface of the array and the sun line, I_d represents inherent degradation and P_0 represents power density of a cell.

The angle θ is taken equal to 20° (worst case) sun angle. P_0 is calculated by multiplying the solar cell efficiency with the solar illumination intensity, 1358 W/m^2. [8-10]

Typical degradation rates for solar cells are between 2-4% per year. Life degradation (L_d) occurs because of thermal cycling in and out of eclipses, micro meteoroid strikes, remnants from thrusters, and material out gassing along the duration of the mission. L_d of solar cells can be estimated as in equation (6):

$$L_d = (1 - \text{degradation/year})^{\text{mission lifetime}} \qquad (6)$$

And power that has to be available at the end of the mission can be estimated as shown by equation (7):

$$P_{EOL} = P_{BOL} \cdot L_d \qquad (7)$$

Finally, the total size of solar array (s) required for this mission can be estimated as below, equation (8):

$$A_{sa} = P_{sa}/P_{EOL} \qquad (8)$$

The baseline solar cells used in this power system design were those made by CESI/Italy for its client ENE S. A under order number SFR-A1/036357. The delivery consisted of 300 solar cells GaAs/Ge (size 40*20 mm^2) submitted to reverse breakdown screening. The solar cells were manufactured using the MOCVD process on Ge substrates. The average electrical performances @ 28°C and AM0 are as below:

Table 3. *Average electrical characteristics of GaAs solar cells.*

Isc (mA)	Voc (V)	Pm (mW)	Im (mA)	Vm (mA)	FF	Efficiency%	Weight (gr)
250	1.023	211	234	0.90	0.83	19.80	0.806

The electrical measurements, table 3, have been performed using a solar cells tester equipped with a high pressure Xe lamp. [13-17]

A beginning of life (BOL) average efficiency of the GaAs solar arrays is assumed to be 19.80%. The solar illumination intensity on the planar solar arrays is assumed to be 1358 W/m^2. Also, the inherent degradation of the solar cell power transfer is assumed to be a factor of 0.77.

In this study, there is a need to determine the elements which have a direct effect on the solar arrays degradation. Table 4 reports all elements contributing in the inherent degradation of space solar arrays. [13-17]

Table 4. *Elements of inherent degradation of space solar cells.*

Elements of inherent degradation	Nominal values	Range values
Design and assembly	0.85	0.77 – 0.90
Temperature of solar cells	0.90	0.80 – 0.98
Shadowing of cells	1.00	0.80 – 1.00
Inherent degradation Id	0.77	0.49 – 0.88

Therefore, the estimated BOL power output of the GaAs solar cells, P_0, can be calculated as: [8-10]

$$P_o = \text{efficiency} * \text{solar illumination intensity}$$

$$P_o = 19.80\% * 1358 = 268.884 \ {}^{W}/_{m^2}$$

Table 5. *Comparison between solar cells in terms of Po.*

Solar cells considered	Efficiency %	Power provided from solar cell Po
Silicon cells	15	203.70 W/m^2
GaAs cells	19.8	268.88 W/m^2
Tecstar cells	28	380.24 W/m^2

Due to the fact that the angle of the sun onto the solar array is not exactly normal, a correction must be applied to find the exact power at BOL. This can be calculated using equation (5): [8-10]

$$P_{BOL} = P_o . I_d . \cos(\theta)$$

$$P_{BOL} = 268.88 * 0.77 * \cos(20) = 194.55 \ {}^{W}/_{m^2}$$

Where P_0 = 268.88 W/m^2 for GaAs solar cells (see table 5), I_d = 0.77, and θ = 20° (worst case), which gives us P_{0BOL} = 194.55 W/m^2. Table 6 compares BOL power capability for different solar cells under 1358 W/m^2 and AM0. [8-10]

Table 6. *BOL power capability per unit area for different solar cells.*

Solar cells considered	BOL power capability per unit area
Silicon cells	147.39 W/m^2
GaAs cells	194.55 W/m^2
Tecstar cells	275.13 W/m^2

Now that the estimated BOL output power of the solar cells has been determined, the required power the solar array must provide during daylight to power the spacecraft for an entire orbit can be found, Psa. From equation (4) and using the values for Xe and Xd respectively 0.65 and 0.85, the required solar array power is equal to:

$$Psa = 60.90 \text{ W}$$

The solar cell lifetime degradation is required to estimate the solar array area needed to provide the required power at end of life EOL. This degradation factor can be found using equation (6). [8-10]

$$L_d = (1 - \text{degradation/year})^{\text{mission lifetime}}$$

$$Ld = (1\text{-}0.0275)^5 = 0.86985 \approx 0.87$$

The degradation term in equation (6) was taken as 2.75% (see table 2) for a nominal lifetime of 5 years. Using the degradation calculated above, the EOL power output of the solar cells for the satellite's mission is estimated, equation (7): [8-10]

$$P_{EOL} = P_{BOL} \cdot L_d$$

$$P_{EOL} = 194.55 * 0.87 = 169.26 \, {}^{W}\!/_{m^2}$$

Table 7. *BOL power capability per unit area for different solar cells.*

Solar cells considered	Degradation per year	EOL power capability per unit area
Silicon cells	3.75% per year	121.75W/m^2
GaAs cells	2.75% per year	169.26W/m^2
Tecstar cells	2.00% per year	247.62W/m^2

The solar array area required for the spacecraft average power can now be found using equation (8):

$$A_{sa} = P_{sa}/P_{EOL}$$

$$A_{sa} = 60.90/169.26 = 0.3598 \text{ m}^2$$

For a satellite with square solar panels, we can calculate the satellite's side as being: [13-17]

$$SAT_{side} = \sqrt{0.3598} = 59.98 \text{ cm.}$$

Table 8. *Solar array area for different types of solar cells.*

Solar cells considered	EOL power capability per unit area	Solar array area in m^2
Silicon cells	121.75 W/m^2	0.5002 m^2
GaAs cells	169.26 W/m^2	0.3598 m^2
Tecstar cells	247.62 W/m^2	0.2459 m^2

The solar cells used in this project are from CESI/Italy for its client ENE S. A under order number SFR-A1/036357. The solar cells are made of GaAs/Ge (size 40*20 mm^2). An estimate of the number of solar cells used for each panel can be calculated as in equation (9).

$$\text{number of solar cells} = \frac{A_{sa}}{\text{solar cell size}} \tag{9}$$

$$\text{number of solar cells} = \frac{0.3598}{8.10^{-4}} = 450$$

The number of solar cells for each type of solar cells can now be estimated. Knowing the area as shown in table 8, it becomes easy to calculate the lot of solar cells required for each type. Table 9 gives the number of solar cells required for Si, GaAs and Tecstar types.

Table 9. *Number of solar cells required.*

Solar cells considered	Solar array area in m^2	EOL power capability per unit area
Silicon (2cm*2cm)	0.5002 m^2	1250 Si solar cells
GaAs (4cm*2cm)	0.3598 m^2	450 GaAs solar cells
Tecstar (6cm*4cm)	0.2459 m^2	102 Tecstar solar cells

Finally, the solar array mass (mass of the aluminum honeycomb structure not included) can be estimated knowing different specific masses of different types of solar cells, table 10.

Table 10. *Solar arrays mass for different solar cells types.*

Solar cells considered	Specific mass mg/cm^2	Solar array mass
Silicon	40 mg/cm^2	0.200 kg
GaAs	100 mg/cm^2	0.359 kg
Tecstar	86 mg/cm^2	0.212 kg

4.3. Selection of a Battery Type

The battery module is responsible for energy storage and energy supply when the solar arrays are not in sunlight (eclipse). The power storage unit which is proposed in this section is based on Lithium Ion batteries. Li Ion batteries have a better performance compared to Nickel-Cadmium and Nickel-Metal Hybrid batteries with higher energy density rates. The number of battery cells, N, that are needed to satisfy the bus voltage, Vbus, can be calculated from equation (10). [12]

$$N_{cell} = \frac{V_{bus}}{V_{cell}} \tag{10}$$

Where:

1. V_{bus}: bus voltage
2. V_{cell}: average cell voltage.

Equation (10) expresses the number of battery cells in series needed. The bus voltage is taken equal to 26.4V and the average battery cell voltage was taken to 3.4V. [12]

$$N_{cell} = \frac{26.4}{3.4} = 8$$

A major assumption made for the battery sizing was the end of life EOL depth of discharge DoD of the battery pack. A correct value for the DoD, for low earth orbit LEO satellites, can be taken equal to 15%. Also, an average specific energy density of 100 W. hr/kg for the Li Ion batteries is assumed for use in the battery sizing calculations. The battery cell capacity is given by equation (11). [12]

$$C_{bat} = \frac{P_e T_e}{(DoD).N.n} \quad (11)$$

Where:
Te: eclipse time period (hr)
Pe: orbit average power W
N: number of batteries (non-redundant)
n: transmission efficiency (battery → load): 90%

$$C_{bat} = \frac{P_e T_e}{(DoD).N.n} = 130.666 \text{ W.hr}$$

With a nominal bus voltage equal to 26.4 V, the battery capacity in A. hr can be calculated as: [12]

$$C_{bat} = 4.9494 \text{ A.hr}$$

The battery capacity used for the course of this study is equal to 5 A. hr. Next, the total mass of battery cells required to power the spacecraft during eclipse can be determined using equation (12). [12]

$$\text{mass} = \left(\frac{C_{bat}}{100^{W.hr}/_{kg}}\right) * N \quad (12)$$

$$\text{Mass} = 1.30 \text{ kg}$$

Table 11. *Number of battery cells and battery mass for different battery types.*

Battery type	Specific energy density W. hr/kg	Battery cell voltage V	Number of battery cells in series	Mass Kg
Nickel Cadmiun	50	1.2	22 (in series)	2.61
Nickel metal hydride	70	1.2	22 (in series)	1.86
Lithium Ion	100	3.4	8 (in series)	1.30

5. Conclusions

An introduction to the engineering design of a LEO satellite power system was approached. The paper describes one by one the different steps an engineer has to follow precisely when sizing solar panels and batteries for a satellite mission. Fixed data, such as orbit altitude, orbit average power, solar cells size, were used to express values for solar panels area and mass, battery capacity and mass.

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