Electric power systems

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1 Introduction

Because of the CubeSats' small size, one of the main challenges during the mission design process is to design (or select from a provider) an electric power system (EPS) that assures enough electrical power for the satellite bus and payload during the entire mission duration. The EPS functions are to generate, store, regulate, and distribute the electrical power that a satellite requires to fulfill the mission requirements. Fig. 1 shows the basic EPS block components. The mission team needs to understand the behavior behind electric power generation in orbit, to size and select the proper components on the electric power generation block. Although there are several technologies that can be used for generating electrical power on a satellite, only solar cell generation will be addressed in this chapter (Section 2), because approximately 85% of nanosatellites (as of 2010) are using solar cells for power generation according to a NASA report on state of the art of small spacecraft technology [1].

Section 3 discusses some of the most used electric power storage devices and the main parameters required for the design. In addition the power supply control that is part of the electric power storage block will be discussed on this section, and methods like peak power tracking (PPT) to optimize the power capture from solar panels will be showed. The power distribution architectures for small satellites will be reviewed in Section 4 and the types of voltage regulators, to allow mission team to have decision elements essential to select the proper devices based on the subsystem requirements.

The creation of an electric power budget will help the mission teams in the design of the EPS. The electric power budget shows a relationship between in-orbit satellite subsystem electric power consumption, in-orbit satellite electric power generation, and in-orbit satellite electric power storage. This tool will help the mission teams to determine the size and performance of several components required by the EPS design, and also, it can help the teams to make the final decision about what kind of EPS will be used on the mission (i.e., their own development or one obtained from a commercial provider). Section 5 will explain how to create an electrical power budget for a particular mission.

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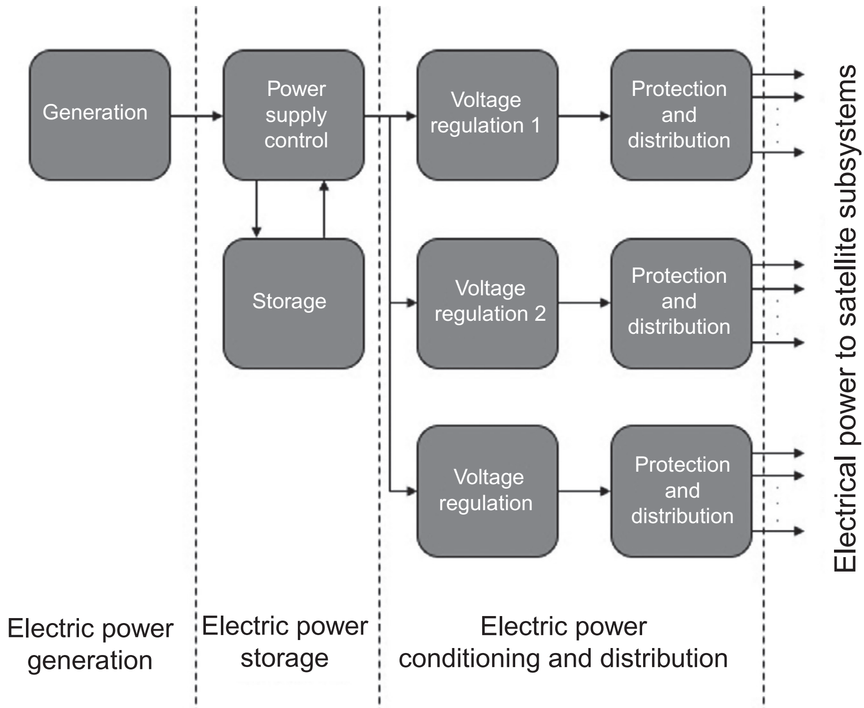


Fig. 1 Electric power system block components.

2 Electric power generation

A CubeSat requires electrical energy to operate. There are several sources of energy that can be transformed to electricity. The different electric power sources that can be used on a satellite can be classified, based on the type of energy, as chemical energy, solar energy, and nuclear energy.

Chemical energy sources for satellites refer mainly to various types of chemical batteries. Nuclear energy sources are primarily radioisotope thermoelectric generators (RTGs) that generate electrical power when the thermocouples are heated by the nuclear reaction. These kinds of energy sources are mentioned as a reference of energy sources on satellites, but as previously indicated, the most common used source of power for CubeSats is solar power by using the photovoltaic effect.

The photovoltaic effect is the direct transformation from light to electricity. The photovoltaic effect was discovered in 1839 by the French physicist Becquerel while he experimented with metal electrodes and electrolytes. Since then, solar cells (devices created to use the photovoltaic effect) have been created using different types of semiconductor materials and techniques to improve the energy transformation efficiency.

The most common materials used in commercial solar cells are silicon (Si) (medium efficiency) and gallium arsenide (GaAs) (high efficiency). Based on the

Table 1 Important solar cell parameters.

Parameter	Commonly referred as	Units
Open circuit voltage	V_{cc}	mV
Open circuit current	I_{cc}	mA
Voltage at max. power	V_{mp}	mV
Current at max. power	I_{mp}	mA
Average efficiency	η	%
Dimensions	Length × width	mm
Cell area	A	cm^2

manufacturing process, solar cells can be monocrystalline (higher efficiency and high cost), polycrystalline (less efficient than monocrystalline cells and less expensive), or amorphous (thin film, flexible solar cells with low efficiency). The most common solar cells for CubeSats are based on monocrystalline GaAs technology and can reach up to 30% efficiency or slightly above. The most important solar cell parameters that the mission team requires for the design are shown in Table 1.

To obtain a proper voltage and current for the satellite power bus, arrays of solar cells are commonly used. If high voltage is needed by the power supply control, then parallel solar cells connections are required; if high current is needed, then serial solar cell connections are required. When solar arrays are used, it is important to consider a solar cell bypass diode for reverse bias protection. Some commercially available solar cell providers include a protection diode on the solar cell.

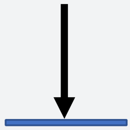
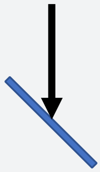
A solar cell or solar array will produce the maximum electrical power when the light source is perpendicular to its surface, that is, with an incidence angle equal to 0 degrees. If the Sun is considered as the light source, the energy received close to the Earth (specifically the solar flux) can be considered as 1358 W/m^2, and the solar cell or solar array generated power can be calculated by the equation shown below in Table 2.

Table 3 shows the power generated based on different incidence angles, the arrow representing the light source and the bar representing the surface of the solar array or solar cell.

Table 2 Solar power generated calculation.

Parameter	Units	Equation	Comments
Solar cell or solar array power generated	W	$P_{SA} = P_{in}\eta A_{eff}\cos\theta$	P_{in}: solar input density = 1358 (W/m^2) A_{eff}: solar cell or solar array's effective area (m^2) η: solar cell efficiency (%) θ: incidence angle (degree)

Table 3 Solar power generated by different incidence angles.

			
$\theta = 0$ degree P_{SA}max	$\theta = 45$ degrees $0.707 \times P_{SA}$max	$\theta = 70$ degrees $0.342 \times P_{SA}$max	$\theta = 90$ degrees 0 (W)

As previously mentioned, the maximum power generated for the solar cell (or solar array) is when the incidence angle is equal to zero degrees. For that reason, some satellites require the use of Sun-tracking arrays or particular attitude control systems, to guarantee maximum power generation. Other factors that need to be considered in a solar cell's on-orbit performance evaluation are temperature, chemical degradation, and eclipses. Solar cells' performance can be affected by high temperatures (above 28°C) [2], solar cell degradation, and eclipses. Typically, solar cells' efficiency is reduced between 0.025% and 0.075% per degree Celsius when the temperature increases above 28°C. Also, solar cells are degraded around 3% by year because of radiation and charged particles. In addition, solar cells cannot generate electrical power during eclipses. In Section 4 a method to calculate the eclipse time is presented within the power budget.

3 Power storage

The previous section mentioned that during eclipses there is no electric power generation; for that reason a device that stores power during sunlight is need. Batteries are devices than can be used to store energy and then release it as electricity.

Batteries can be classified in two main groups: primary batteries that are not rechargeable and secondary batteries that can be recharged. Considering that a CubeSat in Low Earth Orbit (LEO) can undergo an eclipse 15 times a day, a 2-year mission might face an eclipse more than 1000 times. For this reason the use of rechargeable batteries is required.

There are several types of secondary batteries used for CubeSats, but the most commonly used for short mission durations are lithium ion (Li-ion) and lithium polymer (LiPo). The selection of the type of battery mainly depends on the parameters of energy capacity (Wh), specific energy (Wh/kg), and voltage (V). Table 4 shows the most import battery parameters to consider in the EPS design. In Section 4 a method to calculate the battery capacity is presented within the power budget.

Typically, Li-ion batteries have a 0–45°C charge temperature range and −20°C to 60°C discharge temperature range. Batteries can be discharged over a large temperature range, but the charge temperature is limited. Extreme cold and high heat reduce

Table 4 Important batteries parameters.

Parameter	Units
Nominal capacity	mAh
Nominal voltage	V
Constant current charge (*CC*)	A to V
Constant voltage charge (*CV*)	V to mA
Max. discharge current	A
Charge operating temperature	°C range
Discharge operating temperature	°C range
Cycle life at $\times$% *DOD*	Cycles
Gravimetric energy density	Wh/kg
Volumetric energy density	Wh/L
Nominal weight	g
Dimensions (length $\times$ width $\times$ height)	mm $\times$ mm $\times$ mm

charge acceptance, so the battery must be brought to a moderate temperature before charging. Due to this limitation, electrical heaters are installed close to the batteries to avoid low temperatures, and batteries are usually mounted on heat dissipaters to keep batteries on the proper temperature range to avoid battery life reduction.

The depth of discharge (*DOD*) parameter indicates the percentage of the battery that has been discharged relative to the overall capacity of the battery. The higher the *DOD*, the lower the cycle life. Cycle life is defined as the number of cycles (with a 100% *DOD*) a cell can perform before its capacity drops to 80% of its initial specified capacity and then starts to reduce visibly its performance. It is an important parameter to consider based on the number of eclipses expected for the mission as was discussed on the previous section. Commonly, Li-ion has 500–1000 cycles and LiPo 300–500 cycles. How can it be possible to support a LEO mission of 2 years with more than 1000 eclipses with those numbers then? The answer is to change the percentage of *DOD*. There is a logarithmic relation between cycle life and *DOD*: the number of life cycles for a battery goes up exponentially when the DOD is reduced. This means that if a battery has 500 cycles at 100% *DOD*, it will have 2050 cycles at 30% *DOD* and 15,000 cycles at 5% *DOD*, approximately. For that reason, *DOD* lower than 30% is considered during the initial design.

According to Fig. 1, the stage of the EPS indicated as “electric power storage” consists of the storage device and the power supply control. The power supply control is in charge of receiving the generated electricity, sending part of it to the storage device, and sending other parts to the power bus for regulation and distribution to the next stage. There are many techniques to control the electrical power, but the two primary strategies used in CubeSat power supply controls are direct energy transfer (DET) and peak power tracker (PPT) [3].

All solar cells have a unique current versus voltage curve (*i*/*V*) that describes the power output based on voltage and current variations. There is a point on the curve that represents the maximum power output point (MPP). The DET technique is the easiest way to control the generated electricity. Given that the output voltage from the solar cells is constant and the total current varies depending on the operating location on the *i*/*V* curve, this technique can waste power. The PPT technique constantly adjusts the current and voltage from the solar cells to stay on the MPP; this maximizes the generated power. Both strategies are viable, and the one to be used depends on the mission profile.

4 Power conditioning and distribution

The last EPS stage is the electric power conditioning and distribution stage (see Fig. 1). At the end of the stage previously considered (electric power storage), a power bus is provided with a specific voltage and a maximum current capability. Because the CubeSat subsystems may need different voltages, voltage regulators may be required to increase or reduce the voltage bus to proper levels. In addition, some kind of protection is needed to avoid a collapse of the EPS because of a circuit shortcut or due to an overload.

Conditioning the electric power can be accomplished by using two different EPS architectures, a centralized EPS or a decentralized EPS. The difference between these is that a centralized system conditions the electric power within the EPS and a decentralized system conditions the electric power at each satellite subsystem. Considering the integration of the CubeSat subsystems from different providers, the centralized EPS is often the best option.

The device in charge of conditioning the voltage bus to match the required subsystem voltage level is the voltage regulator. A voltage regulator is used to regulate (maintain) a voltage level. It generates a fixed output voltage that remains constant even in the event of changes in an input voltage or load conditions. In general, there are two types of voltage regulators, linear voltage regulators (LVR) and switching voltage regulators (SVR).

LVRs are compact and easy to use; they act like a voltage divider; for that reason, they have low efficiency. Other advantages are their low output ripple voltage, fast response time for load or voltage input changes, low electromagnetic interference, and low electronic noise. The main disadvantages are low efficiency and the fact that the output voltage cannot be increased above the input voltage. LVRs are commonly used as voltage reference only.

SVRs use elements to rapidly switch devices like capacitors and inductors to provide the proper current and voltage level at the SVR output. Switching is controlled by a feedback mechanism. SVRs are high efficiency because they dissipate almost no power. SVRs are able to generate output voltages that are higher than the input voltage or output voltages with opposite polarity than the input voltage. The main disadvantages are higher output ripple voltage, slower transient recovery time, and very noisy output because of the generation of electromagnetic interference (EMI). Different SVR configurations are summarized in Table 5.

Table 5 Switching voltage regulator configurations.

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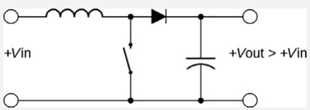
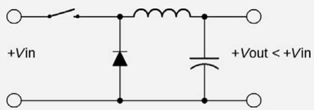
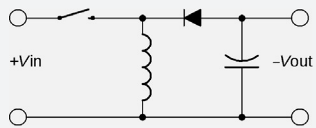
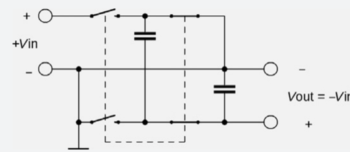
<tr>
<td>

Step-up
Boost
Raises input voltage</td>
<td>

Step-down
Buck
Lowers input voltage</td>
<td>

Step-up/step-down
Boost/buck
Raises or lowers or inverts input voltage</td>
<td>

Charge pump
DC-DC converter without inductors
Raises or lowers or inverts input voltage (low power applications)</td>
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5 Power budget

The most common method of generating power on a CubeSat is using solar arrays, as previously mentioned. Solar arrays are made up of solar cells that, owing to several factors, will reduce the power generated over time, which has to be considered in a power budget (PB). There are several ways to create a power budget; some of them are very complex and require the use of additional software tools [like Systems Tool Kit (STK) by Analytical Graphics, Inc.] to simulate several conditions. In this section, a method to estimate a power budget with a sufficient precision for typical CubeSats missions in LEO is shown. By using the requirements in Table 6, a power budget will be created with the goal to determine the solar array power generation margin ($P_{SA}M$) and the battery capacity (*Batt*) required by the specific mission.

Table 7 shows the equations that will be used to calculate the primary parameters for a specific orbit. The time of eclipse (*TE*) and the time in sunlight (*TS*) are required to calculate the power budget; it is important to determine how much power can be generated during *TS* and how much power the batteries must provide during the *TE*.

Table 6 EPS requirements.

Requirements	Comments
1U CubeSat using body-mounted solar cells	Solar cells covering the complete surface of each side
Mission duration 3 years	
Orbit height 500 km	Circular orbit
Solar cells 30% efficiency	Gallium arsenide (GaAs)
Solar cells efficiency degradation 3% per year	
Solar array packing factor (*PF*) 0.75	
Batteries depth of discharge (*DOD*) 30%	Lithium ion (Li-ion)
Onboard data handling (OBDH) system $V = 3.3\,\mathrm{V}$ $P = 0.5\,\mathrm{W}$	Duty cycle 100%
Telemetry, tracking, and command system (TT&C) $V = 12\,\mathrm{V}$ RX mode: $P = 0.2\,\mathrm{W}$ TX mode: $P = 2.5\,\mathrm{W}$	RX mode: duty cycle 100% TX mode: duty cycle 15%
Electric power system (EPS) $P = 0.1\,\mathrm{W}$	Duty cycle 100%
Attitude determination and control system (ADCS) $V = 3.3\,\mathrm{V}$ $P = 0.3\,\mathrm{W}$	Duty cycle 100%
Payload (PLD) $V = 5\,\mathrm{V}$ $P = 0.8\,\mathrm{W}$	During sunlit: duty cycle 30% During eclipse: duty cycle 0% (off)

Table 7 Equations for calculating orbital parameters required by the power budget.

Parameter	Units	Equation	Comments
Orbital period (T)	s	$T=\sqrt{\frac{4\pi^2R^3}{\mu_{\text{Earth}}}}=2\pi\sqrt{\frac{R^3}{\mu_{\text{Earth}}}}$	LEO and circular orbit μ_{Earth}: 3.986×10^5 (km^3/s^2) R: Earth's radius 6378 (km)
Earth's angular radius (ρ)	degrees	$\rho=\sin^{-1}\left(\frac{R}{h+R}\right)$	h: orbital altitude (km) R: Earth's radius 6378 (km)
Time of eclipse (TE)	s	$TE=\frac{2\rho}{360°}T$	ρ: Earth's angular radius (degrees) T: orbital period (s)
Time in sunlight (TS)	s	$TS=T-TE$	T: orbital period (s) TE: time of eclipse (s)

Table 8 Orbital parameters obtained based on the EPS requirements.

Parameter	Units	Value
Orbital period (T)	s	5676.81
	min	94.61
	h	1.58
Earth's angular radius (ρ)	degrees	68.02
Time of eclipse (TE)	s	2145.15
	min	35.75
	h	0.60
Time in sunlight (TS)	s	3531.66
	min	58.86
	h	0.98

To calculate TE and TS, the orbital period (T) and the Earth's angular radius (ρ) need to be known or calculated. Using the equations in Table 7, the results of Table 8 are obtained.

The next step is to calculate the power generated and the power required by the CubeSat to determine the power generation margin. Table 9 shows the equations that will be required to calculate the power generated considering the requirement parameters for this particular mission.

The first calculation is to determine the solar array power generated at the beginning of operational life (BOL). To do this, several factors need to be considered. As was shown in Section 2, the solar array area, solar cell efficiency, and the incidence angle are required to determine the generated power. The solar cell efficiency will be affected by degradation and high temperatures. Degradation will change solar cell efficiency over the years until the end of life time (EOL), and high temperatures will change it during sunlight periods on every orbit. On the other hand, if the CubeSat is

Table 9 Equations for calculating primary power parameters required by the PB.

Parameter	Units	Equation	Comments
Solar array power generated at *BOL* ($P_{SA}BOL$)	W	$P_{SA}BOL = P_{in}\eta A_{eff}\cos\theta$	P_{in}: solar input density = 1358 (W/m^2) A_{eff}: solar array's effective area = long × wide × *PF* = 10 cm × 10 cm × 0.75 = 0.0075 (m^2) η: solar cell efficiency = 30% θ: incidence angle = 0 (degrees)
Solar cell efficiency at *EOL* (ηEOL)	%	$\eta EOL = \eta(1 - YD)^{YM}$	η: solar cell efficiency = 30% *YD*: solar cell eff. Degradation per year = 3 (%) *YM*: years of mission duration = 3
Solar array power generated at *EOL* ($P_{SA}EOL$)	W	$P_{SA}EOL = \eta EOL(P_{SA}BOL)$	ηEOL: solar cell efficiency at *EOL* (%) $P_{SA}BOL$: solar array power generated at *BOL* (W)

using body-mounted solar cells, it is expected that the CubeSat will have some rotation or tumbling (depending on the attitude motion), and because of this, the incidence angle will be constantly changing; therefore the power generated will also change. As can be expected to calculate the power generated, a simulation including the satellite dynamics in orbit and the solar array temperature variations in orbit is required. To create a realistic model is nontrivial and requires simulation tools. Based on previous experience with body-mounted solar cells on CubeSats, a practical estimation for keeping the incidence angle constant and equal to zero is to consider that one side of the 1U CubeSat will be generating energy during sunlight. Table 10 shows the practical values for different types of CubeSats.

Note that for this power budget and based on the EPS requirements shown in Table 10, the effective area is the area of one of the CubeSat sides times the packing factor *PF* ($0.1\,\text{m} \times 0.1\,\text{m} \times 0.75 = 0.0075\,\text{m}^2$). The packing factor indicates the portion of a CubeSat face covered by solar cells. A $PF = 1$ means that the whole face is

Table 10 Practical solar array effective area consideration.

CubeSat	No. of CubeSat sides with solar cells	Solar array's effective area practical consideration (m^2)
1U	6	$1 \times A_{eff}$ from 1 side ($0.1\,\text{m} \times 0.1\,\text{m} \times PF$)
2U	4 (X/Y facets)	$0.80 \times A_{eff}$ from 1 side ($0.1\,\text{m} \times 0.2\,\text{m} \times PF$)
3U	4 (X/Y facets)	$0.86 \times A_{eff}$ from 1 side ($0.1\,\text{m} \times 0.3\,\text{m} \times PF$)

Table 11 Power parameters obtained based on the EPS requirements.

Parameter	Units	Value
Solar array power generated at *BOL* ($P_{SA}BOL$)	W	3.06
Solar cell efficiency at *EOL* (ηEOL)	%	27.38
Solar array power generated at *EOL* ($P_{SA}EOL$)	W	2.79

covered by solar cells. A $PF = 0.75$ has been considered in this example, meaning that the cells cover 75% of the face. Considering another mission in which the solar cells will have a different size, the effective area will be adjusted to the proper packing factor (*PF*).

Table 11 shows the results for the power parameters calculated considering the power generated at the beginning of operational life, the solar cell efficiency after 3 years, and the power generated 3 years after operating in orbit.

Once the generated power is calculated, it is time to calculate the power required by the CubeSat and the average orbital power required (*AOPR*) for each subsystem and for eclipse and sunlight events. This can be calculated by multiplying the power requirements by the duty cycle. Considering the requirements of Table 11, Table 12 is obtained.

The results in Table 12 show the *AOPR* during sunlit and during eclipse; the next step is to calculate the energy required per orbit to calculate the power required by the solar array during one orbit. Table 13 shows how to calculate these parameters for one orbit.

After calculating the required energy during sunlit and eclipse, the power required by the solar array can be calculated. Table 14 shows these results.

Finally, after knowing the amount of power generated and the amount of power required per orbit, a power margin can be obtained. If the power margin is high, the power system is likely to work properly and can handle possible power anomalies.

Table 12 Average orbit power required.

Subsystem	Power requirement (W)	Sunlit duty cycle (%)	Average orbit power required during sunlit $AOPR_S$ (W)	Eclipse duty cycle (%)	Average orbit power required during eclipse $AOPR_E$ (W)
OBDH	0.5	100	**0.5**	100	**0.5**
TT&C (RX)	0.2	100	**0.2**	100	**0.2**
TT&C (TX)	2.5	15	**0.375**	15	**0.375**
ADCS	0.3	100	**0.3**	100	**0.3**
EPS	0.1	100	**0.1**	100	**0.1**
PLD	0.8	30	**0.24**	0	**0**
Total power (W)			**1.715**		**1.475**

Table 13 Equations to calculate the energy and solar array power required per orbit.

Parameter	Units	Equation	Comments
Energy required during eclipse (E_E)	Wh	$E_E = AOPR_E\ TE$	$AOPR_E$: average orbit power required during eclipse = 1.475 W TE: time of eclipse = 0.60 (h)
Energy required during sunlit (E_S)	Wh	$E_S = AOPR_S\ TS$	$AOPR_S$: average orbit power required during sunlit = 1.715 W TS: time in sunlight = 0.98 (h)
Energy required to produce (EP)	Wh	$EP = E_E + E_S$	E_E: energy required during eclipse (Wh) E_S: energy required during sunlit (Wh)
Solar array power required ($P_{SA}R$)	W	$P_{SA}R = \frac{EP}{TS}$	EP: required to produce (Wh) TS: time in sunlight = 0.98 (h)

Table 14 Energy and solar array power required per orbit obtained based on the EPS requirements.

Parameter	Units	Value
Energy required during eclipse (E_E)	Wh	0.88
Energy required during sunlit (E_S)	Wh	1.68
Energy required to produce (EP)	Wh	2.56
Solar array power required ($P_{SA}R$)	W	2.61

If the margin is negative, it means that the satellite requires more power than the generated and needs to be modified in some way, reducing the power consumption of its subsystems, modifying satellite system's duty cycle, increasing the power generation, or by other methods.

In addition, the required battery capacity can be calculated based on the requirements. Tables 15 and 16 show how to calculate the solar array power margin and battery capacity as well as the results based on the EPS requirements.

Table 15 Equations to calculate the solar array power margin and battery capacity.

Parameter	Units	Equation	Comments
Solar array power margin ($P_{SA}M$)	W	$P_{SA}M = P_{SA}EOL - P_{SA}R$	$P_{SA}EOL$: solar array power generated at EOL (W) $P_{SA}R$: solar array power required (W)
Battery capacity ($Batt$)	Wh	$Batt = \frac{E_E}{DOD}$	E_E: energy required during eclipse (Wh) DOD: battery's depth of discharge (%)

Table 16 Solar array power margin and battery capacity obtained based on the EPS requirements.

Parameter	Units	Value
Solar array power margin ($P_{SA}M$)	W	0.18
	%	6.81
Battery capacity (*Batt*)	Wh	2.93

Results for this specific example show that there is a low positive power margin at the end of the operational life equal to 6.81%. The battery capacity is 2.93 Wh; therefore the mission team can create battery arrays (in series and/or in parallel) to meet the required battery capacity. Using the generated power budget, the mission team can modify some of the parameters to adjust the final results.

6 Conclusions

In summary, it is important to understand the technology behind the generation, storage, regulation, and electrical distribution in the CubeSat design to select the appropriate components to meet the requirements of the specifications for the power system. In addition, the creation of a power budget can be as complex as required by the mission. In Section 5 a method to create a relatively simple power budget was shown, which proved to be a good approximation. These elements and factors can be very useful for modeling the first numbers during the CubeSat mission design process.

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