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Power modeling and budgeting design and validation with in-orbit data of two commercial LEO satellites

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ABSTRACT

A satellite's *Power Subsystem* is in charge of delivering power to the rest of the satellite subsystems during the different phases of the mission. This critical subsystem can be modeled as a three-component system: a primary energy source, a secondary energy source that will be used whenever the primary source is not available or sufficient and third, the loads of the subsystem, which represents the rest of the satellite power requirements. In this paper, the process for modelling, simulating and sizing these three components is presented as well as the results of applying this methodology on two real commercial earth observation satellites called NewSat-1 and NewSat-2. Finally, in-orbit telemetry is presented and used to verify the subsystem functionality.

INTRODUCTION

The power subsystem of a satellite is responsible for delivering conditioned power to all the other subsystems and is made-up of several hardware and software components. A failure or degradation on the subsystem will impact directly on the satellite mission. Therefore, it is extremely important to design a robust, fault tolerant and well rated power subsystem to ensure that it will provide the required power through all the mission lifespan.

The starting point for this project was to design a power subsystem for small earth observation satellites that were being developed at Satellogic. These satellites, called NewSats, are commercial earth observation satellites with different payloads and thus the power requirements during the lifespan are designed to change according to the mission and payloads used.

Traditional engineer methodologies such as the V-model [1] are usually used to define, design, implement and validate engineering projects, but this was not possible when starting this project mostly because the initial starting point of this methodologies are the requirements which are usually hardly defined which in this case, were not. During the development of this power subsystem, all the other subsystems of the satellite were being developed too so the power requirements of the system where not clearly defined. Neither were the available area for solar arrays or the available mass budget.

The only initial soft-requirements at the beginning of the project were:

1. Deliver unregulated power to all the satellite loads for the different mission's modes (attitude profile and power consumption profiles) even if the satellite was at eclipsed phase of the orbit.
2. Do not degrade more than 10% for the first 3 years in orbit in terms of power delivery and energy storage capacity.
3. Be able to work on any LEO ranging from equatorial to polar and SSO (from 10:30 to 14:30 LTAN) below 700 Km.

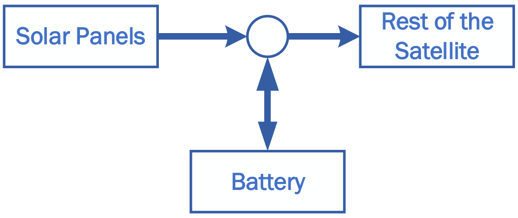
These conditions created the need for a parametrizable power system model that could be used to test different configurations during the satellite design phase to converge with the rest of the subsystem in what was going to be the final satellite design. For example, to test different solar cells models, different configurations of solar arrays, battery sizes or power consumption profiles.

In the following sections, the process used to model, simulate and validate the power subsystem for the is presented. This process had been developed during the last 5 years in Satellogic for designing power subsystems for LEO satellites and has been implemented in 5 different missions from 2U CubeSat's, 25 Kg technology demonstration satellites as well as the new commercial observation satellite, the NewSats.

ARCHITECTURE OF THE POWER SUBSYSTEM

A simplified model of the power subsystem is illustrated in Figure 1. The primary energy source of this subsystem is the Sun, which's energy is transformed from electromagnetic to electric using Solar cells arranged as panels all around the satellite. These solar panels, will deliver power to the rest of the satellite and, if some rest is available, to the secondary energy source, which in this case is rechargeable electrical battery,

Figure 1: Simplified subsystem model



This secondary energy source is used when the primary energy source is not available, like during eclipses, or when it's just not enough to supply the demand of power from the rest of the satellite, like when using all the payloads and radios at the same time and the peak power consumption is more than the power generated by the solar panels.

This configuration will always obey the *energy conservation principle* which states that all the incoming power into the system will be transferred to the rest of the components or stored into it. This is of extremely importance in the subsystem design, because an overload of incoming power to the system will be something that will need to be predicted and managed because upon the impossibility of converting the electromagnetic energy of the cells into electric power, they will heat up and this can possibly damage them.

In the following sections, the developed model for each block of this high-level model will be introduced and the interaction between them will be analyzed to define and size them accordingly.

MODELLING AND SIMULATION

The Loads

The first analysis that must be done is called the "power budgeting". Starting with the loads, it was gathered information on how much power each one will require and how is this power going to be spent during the operating time. This was done with two different perspectives: an average power consumption of each component on every mode which leaded to an energy requirement for each load and an instant peak power consumption. The first one is summarized in Table 1.

The third column, indicates the amount of power that each component or subsystems uses in full power mode. Then for each mode, a duty cycle is estimated according the operation profile and the orbital average power (OAP) is calculated.

Table 1: Simplified power budget

	Component	Power [w]	Consumption per mode [%] & [OAP]					
			Nominal		Payload 1		Payload 1	
BUS	Power modules	2.74	100	2.74	100	2.74	100	2.74
	On board computers	5.38	13	1.19	1.19		26	8.98
	low bitrate radios	5.85	100	5.85	100	5.85	100	5.85
	high bitrate radios	25.11			8	2.28	8	2.28
	AOCS	9.63	13	1.27	80	7.65	80	7.65
Payload	Multispectral	27.4					100	27.4
	Hyperspectral	13.0			100	13.0		
	CloudCam	3.01			100	3.01		
	Total OAP		**11.05 WO**		**35.72 WO**		**54,9 WO**	

For example, power modules and low-bitrate radios are turned on all the time and high bitrate radios or payloads are only used on specific modes and for short periods of time.

Finally, the average power requirement for each mode was calculated and it was concluded that 10, 25 and 40 Watts OAP were required for safe, nominal and mission modes for the NewSat respectively. Also, from another analysis not included here, it was determined that up to 80 Watts instantaneously peaks were necessary on some modes.

These inputs not only present requirements for the primary and energy source but also helps defining the hardware rating of different component such as power transistors, cable gauges, connectors amperage, etc.

Primary Energy Source: Solar Panels

Solar cell Generation Model

The power that a solar cell can generate can be calculated using equation 1.

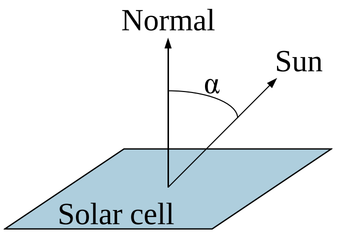
$$\boldsymbol{Power}[W] = \boldsymbol{P_i}\left[\frac{W}{m^2}\right] * \boldsymbol{area}[m^2] * \boldsymbol{\eta} * \boldsymbol{\cos(\alpha)}$$

(1)

where: P_i is the incident power from the Sun (which is about 1367 W/m^2 in LEO according to the AM0 model [2]). *area* represents the solar cell effective generating area and η is the efficiency of the cell and are usually specified in the cells datasheet, which states how much of the received power the cell can convert to electric energy under certain circumstances like temperature and aging. The variable α is the angle between the normal of the cell and the incident light as illustrated in Figure 2.

The cosine in equation REF gives a representation of the effective view area that the sun is seeing. For example, if α is equal to 90deg then, no power is generated cause no sunlight is hitting the solar cell surface.

Figure 2: Sun incident angle with respect to cell normal



It also a common practice to use an actual measurement of the angle response of the cell taken in a laboratory instead of the pure cosine function to obtain better results but this was not considered in this work.

Equation REF is just a first order approximation to the actual power generated by the solar cells. Other parameters can be added to Equation 1 to account for solar cell degradation like degradation due to UV exposure, radiation damage and temperature. These second order parameters will become important when analyzing the begging of life (BOL) vs. end of life (EOL) performance.

The result for the NewSats cells degradation is summarized in Table 2 where each cell of the table represents the degradation compared to the original datasheet values for each temperature and radiation value.

Table 2: Solar cell degradation and temperature effects.

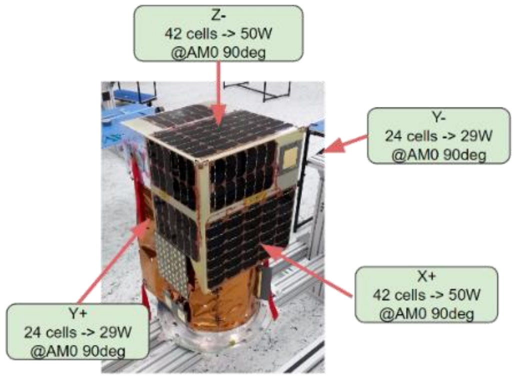
		Temperature [C]						
		30	40	50	60	70	80	90
1MeV Fluence [e/cm^2]	BOL = 0	100.0%	98.0%	96.0%	93.9%	91.8%	89.7%	87.6%
	2.50E+14	94.5%	92.3%	90.2%	88.0%	85.8%	83.5%	81.2%
	5.00E+14	92.4%	90.2%	87.9%	85.6%	83.2%	80.9%	78.5%
	1.00E+15	90.9%	88.7%	86.5%	84.3%	82.0%	79.8%	77.4%

Finally, to complete the real generation model this power must be extracted from the solar cells with some electronic circuitry. There are different methods of doing this. One popular method is Maximum Peak Point Tracking (MPPT) [3] and will not be covered in this paper. This methodology can be considered into the modelling of solar cells power in the calculations as another efficiency term in the equation. Current MPPT technology in the NewSats has an efficiency of more than 90% and was included in the calculations accordingly.

Geometric Generation Model

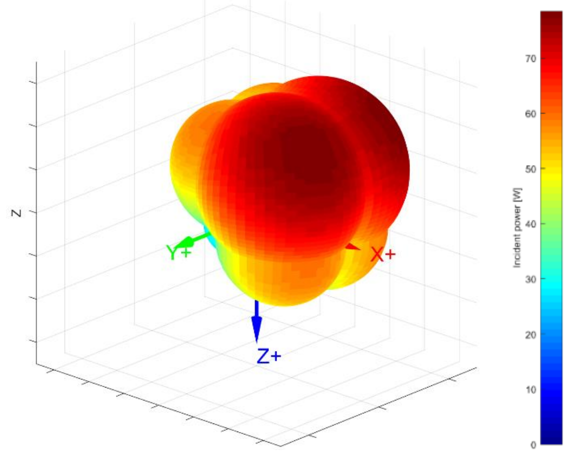
The solar cells described in the previous sections must be mounted in the satellite body or in deployable solar panels and the precise location and orientation of this cells will impact directly on the power generation profile. Depending on the topology of the MPPT and battery charger circuits, as well as the required bus voltage, cells are arranged in series strings and connected in parallel to form different arrays or panels. NewSats satellite has 4 fixed, body-mounted solar panels corresponding to the X+, Y+, Y- and Z- body axes as illustrated in Figure 3.

Figure 3: Solar cells distribution



Similar to a typical radiation diagram of an antenna, using **Error! Reference source not found.** for all the cells at the same time with the correct α for each face, it is possible to calculate the total power generated for each possible sun incidence angle. This is illustrated in Figure 4. Using this calculation, it is possible to calculate the amount of power that each face will generate for any solar incidence vector in body axes.

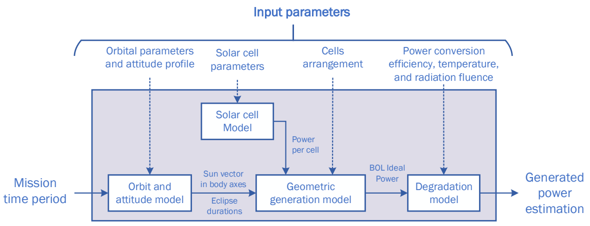
Figure 4: Solar incidence plot



The Complete Generation Model

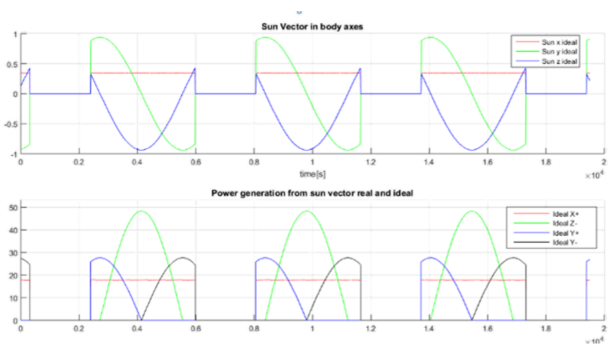
Finally, with the addition of a model of the orbit, the earth and the sun relatives' positions with respect to the satellite can be calculated. Adding this to the different attitude profiles that the satellite will have, the sun vector in body axes can be calculated and inputted to the geometrical solar model. Then all the previously described parts of the model together are connected as illustrated in the total generated power was calculated for the input parameters of the model. The parameters can then be adjusted to obtain the required power and the size the solar panels according to the requirements.

Figure 5: Complete solar generation model



In Figure 6, the result of running the complete model is presented. Top subplot represents the result of the orbit simulator and the attitude mode which is the sun vector in body axes. These was then inputted to the solar incidence model giving the amount of power that each panel will generate. In this example, the NewSat was simulated in a SSO orbit of 10:30 LTAN with the Y+ satellite's axis aligned to the velocity vector and the Z+ axis aligned to Nadir. Eclipses are represented in this simulation as a [0,0,0] sun vector which generates not power at all. Because of the geometrical arrangement of the problem, sun will appear from side Y+ of the satellite, which can be seen as blue lines, then go around to the Z- axis and finally end the cycle through the Y-. In all this cycle, the relative "sun elevation angle" for the X+ face will stay almost constant and so is the generated power.

Figure 6: Generation model output



The final configuration of the NewSats gives a total of 40W orbital average power (OAP) in the nominal nadir pointing attitude mode and 20W in the safe spinning attitude mode for BOL. A less than 10% degradation factor was determined at EOL.

Secondary Energy Source: Batteries

Li-Po batteries has been proved in many LEO missions in the last few years REF. They were a good candidate based on the high market availability, low cost, good specific energy and very low maintaining efforts like low self-discharge current, ambient temperature storage, etc.

The satellite batteries were made up from several cells connected in series and parallel that sum up to obtain the total capacity, the desired voltage and charge/discharge maximum currents available. The total capacity, which is the sum of the capacities of each individual cell, no matter if they are in series or parallel, will be the most important parameter to determinate.

Meanwhile parallel and series configurations must be defined according to other system constraints at the same time. In the following sections the procedure used for selecting this values for the NewSats will be described.

Battery voltage definition: series configuration

To select the battery voltage, the quantity of cells in series must be defined. For this, the two main drivers are the battery charger's voltage limits that are connected to the solar panels and the bus voltage requirements. Also, having a voltage as high as possible is desirable because this will impact on the size and weight of the wiring. For the NewSat a 3 cell in series was selected and was limited by the maximum possible voltage compatible with the solar cells battery chargers.

Battery charge/discharge current: parallel configuration

The second parameter that was determined was the charge and discharge current that was necessary. This is more a limitation than a feature in any battery and to prevent damage of the cell, it must be taken care that the maximum charge/discharge current of each cell is not exceeded. Otherwise the battery capacity will be seriously affected during the mission's lifetime. To do this, many parallel series strings is hardwired in the battery circuit to distribute the currents evenly into these strings.

The value of the maximum and steady state current needed was calculated from the loads analysis that was made on Section **Error! Reference source not found.** and because of the requirement of delivering power also in the eclipsed phase of the orbit, the battery itself was defined to be able to deliver the 100% of the power needed. These analyses gave a premature requirement of

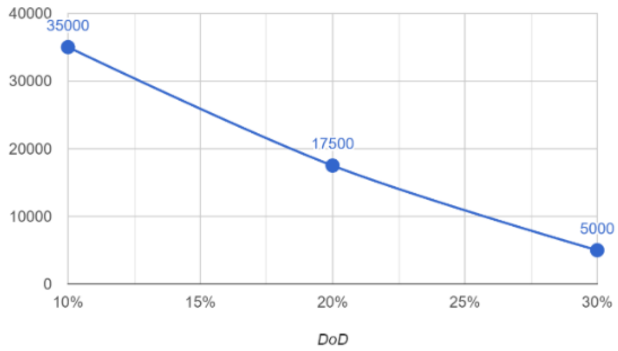
2 parallel packs of 3-cell strings to fulfill the requirements.

Battery Depth of discharge vs Lifetime

While the charge/discharge current limits are not voided, the battery lifetime will depend mostly on the depth of discharge (DoD) of the cells which represents how much of the total capacity of the battery was used at any certain moment. The complement of DoD is the state of charge (SoC) and it's simply the calculated as 100% - DoD.

The degradation of the cells with respect to DoD is measured in how many cycles (charge/discharge) can the battery sustain before degrading to a certain level. For NewSats battery cells, this limit is shown in Figure 7.

Figure 7: Battery lifetime vs. DoD



Because of the requirements of a 3-year lifetime and the LEO orbit, almost 17000 orbits/cycles of charge-discharge will be needed. To obtain this, **Error! Reference source not found.** states that the maximum DoD allowable was 20%. This implies directly in the battery capacity sizing because now, the required 2 packs in parallel turned to be just the 20% of the required capacity to survive the mission lifespan.

The immediate conclusion after this analysis is that the total capacity of the system calculated in the previous sections must increase five times to make it suitable for the mission. Finally, a 3S10P Li-Po battery was determined for the NewSat satellites.

THE COMPLETE SUBSYSTEM

The complete power subsystem module (without the distribution module) is illustrated in Figure 8. This was implemented using customized battery packs and other components that were originally developed for CubeSat's missions. The final configuration for this module was a 3S10P Li-Po battery with approximately 140Wh of capacity and up to 80W constant power source capability. In the following sections, the validation of this systems is presented using in orbit telemetry.

Figure 8: NewSat power subsystem batteries and power supply management unit

SYSTEM VALIDATION WITH IN-ORBIT RESULTS

Each NewSat satellite records during the entire orbit different types of telemetry, more than 50 different power measurements are recorded at configurable rates to allow ground analysis and downloaded on every ground station contact. Also, about 100 temperature sensors are placed on different places of the satellite. Correlating this data with attitude and orbit information valuable analysis can be made. At the moment of writing this paper, 1 year of telemetry was recorded by each satellite and is available for analysis. In the following subsection, different types of analysis used for validating the power subsystem design are presented.

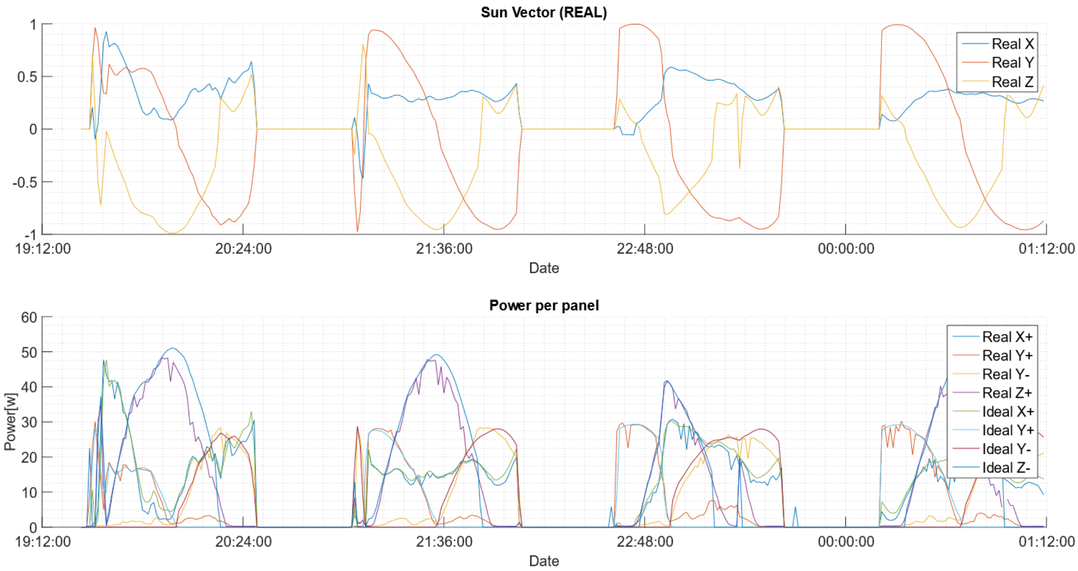
Solar panel Generation

Each satellite has a coarse sun sensor set that gives a reading of the sun vector with respect to the body axes. These values are plotted in the upper subplot of Figure 9 for 4 consecutive orbits. This can be compared to REF they will be found very like the ideal one generated by the mathematical model of the orbit and the attitude determining that the attitude and orbital model and the real attitude an orbit of the satellite are comparable.

Anyway, at the beginning of each illuminated phase is it possible to observe some high rate signals that are caused by ground station-pointing maneuvers that were performed by the satellite during this period.

In the lower subplot of Figure 9 two different sets of signals are plotted. The first one, labelled "Ideal" is the expected power that each panel should generate when inputting the real sun vector (from the upper subplot) to the complete solar generation model described in the previous chapters. While the ones labeled as "Real" are the measured power generated by each panel.

Figure 9: Validation of the generation model using in-orbit telemetry



Both signals sets are almost identically despite some measurement noise and other second order effects.

It is also visible that during the three orbits the shape of the generated power is not the same and this was caused by different requirement by the subsystem loads on each orbit. In other words, the solar panels only generated the power required, a condition that not always lend to taking the maximum power of each panel as described before.

During this analysis, degradation effects were not considered because the total degradation for each panel is calculated to be less than 10% in 3 years and these measurements were taken only with 6 months of mission, so no considerable losses can be addressed.

Battery performance and DoD.

Figure 10 shows the SoC recorded every 60 seconds for about 19 orbits (16 orbits per day) during normal satellite operations. It is possible to see that each orbit corresponds to a charge/discharge cycle that matches with the illuminated an eclipsed phase of each orbit.

Figure 10: Detailed SoC evolution

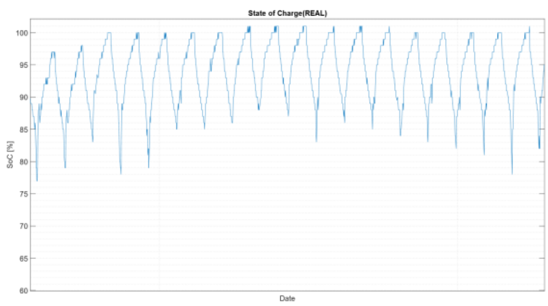
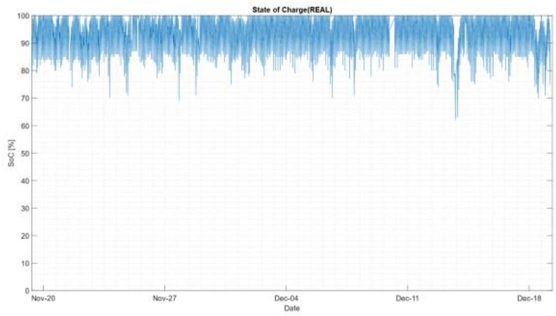


Figure 11 shows the recorded SoC for almost 1 month in orbit. Here it can be seen that the SoC was in average always again above 80% while the satellite was being used for image taking tasks.

Figure 11: one month SoC recording



It can also be seen that there are some peaks where the battery was discharged to values lower than 80%, which corresponds to some special programmed tasks performed by the satellite.

The Loads

During the nominal operation of the satellites it was verified that the average power consumption of each mode was close to the estimated value. Furthermore, periodic software upgrades on different components are reducing the power consumption so the power budget is evolving daily.

CONCLUSIONS

While all the subsystems of a satellite architecture are necessary and important, the power subsystem has always the requirement to be the first to work without problems. A failure on this system will probably cause a massive mission failure. In this paper, the process used to model, simulate and design a power subsystem was presented and in orbit data validation has proved that this methodology is reliable and will be used for future mission design. It's also quite easy to analyze different solar cells configurations and solar panel arrangement as well as different battery topologies.

Also, this soft-requirement approach demonstrated a good wat to achieve optimized results in several subsystems at the same time in comparison to the traditional V-Model approach.

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