ELECTRICAL MODELING AND SIMULATION OF A MEDIUM-SIZE SATELLITE

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ABSTRACT

Aiming on the requirement of generating an accurate electrical model of a medium size satellite, the software created for this paper provides a complete simulation of a medium to bigger size satellites. This paper analyses the disadvantage in the vulnerability in the existing satellite to harsh environment condition like shadowing, radiation and extreme temperature variations in space. Solution is thereby offered through modeling and simulation of a medium-size satellite in a more realistic fashion. The methodology of the modeling includes system design and solidification of a generic solar array, distribution system, backup battery system, load analysis system and a complete thermal model for the battery and solar array. The software includes solar cell to solar array relations, bypass systems, imbalance in array wings, array power generation and shunting model, battery charge and discharge model and a load profile. The modeled satellite shows during tests that during daylight, the system runs effectively, charging the battery. Thence, excess power is calculated and shunted. And during darkness, the panels were able to provide power through the battery. All cell/array/battery features are therefore highly realistic and database modeled.

Keywords:- Satellite, Solar Array, Power Regulation, Back-up Battery, Electrical Model.

INTRODUCTION

As space missions are getting more involved in space technology, satellites systems are getting more complex in parallel. Even the size of satellites are getting smaller due to budgetary constraints, the amount of power required to run the complete system is getting bigger resulting into larger solar arrays, higher capacity batteries and a much more sophisticated power management

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systems (Ngoc et al, 2016). The primary function of satellite power system is to supply and manage uninterrupted power to its subsystems and payloads. These subsystems include power generation subsystems such as solar arrays, power storage subsystems which are batteries with different chemical structures, power control and distribution subsystems like power converters, power distribution units, power conditioning units, and battery charging/discharging units (Yousef, 2015).

In the present space power domain, most of the satellite power systems use solar cell arrays as their power core. This is mainly related with their high performance to cost ratio (Shah et al, 2015). However, a disadvantage is the vulnerability to harsh environmental conditions like shadowing, radiation and extreme temperature variations in space. Same situation applies to the batteries and related systems. The maturity of the whole satellite power system directly affects the satellite’s performance and working life (Kui et al, 2015). One way to improve this vital situation is to simulate and test the complete electrical system in a more realistic fashion. The complete system put forward here includes distribution system, back-up battery system, load analysis system and a thermal model as well as battery and solar array models (Loo et al, 2014).

Studies involving satellite power systems are available extensively in literature. Every new designed satellite is expected to accomplish a broader range of requirements with more reliability and in a smaller and lighter package. Patel (2014) has covered an extensive amount of material in his book “Spacecraft Power Systems”. Also many studies dedicated to solar arrays (Sharma et al, 2016) and batteries are available in the literature. However, in this work the aim was to cover the whole power system with addition of load profiles rather than focusing on a single unit. This is considered as a unique feature among papers with similar subject (Chang et al, 2016). The paper is divided into four sections. In the second section the solar array is emphasized. The third section deals with the subsystem models which are Load Profile, Battery, Solar Array Power and Thermal Conditions. Finally a conclusion section is added for a short summary of the whole idea.

SOLAR ARRAY DESIGN

The major power source for a spacecraft is the solar array. The array must make up for the utilities, the housekeeping power and charge the batteries during daylight. Therefore the array must produce enough current and voltage. Incorrect design may cause loss of the satellite. Therefore the first step of the model is to generate a solar array of desired attributes. In order to avoid complex trigonometric functions, an assumption was made before the calculation process began. The solar array system is accepted to have a 2-axis motor that always tracks the maximum power, so the calculations will be mainly based on maximum power current (I_{mp}) and maximum power voltage (V_{mp}). In the actual world, most of the developed spacecraft systems are provided with maximum power trackers. Figure 1 shows the obtained I-V curve as power current is plotted against power voltage.

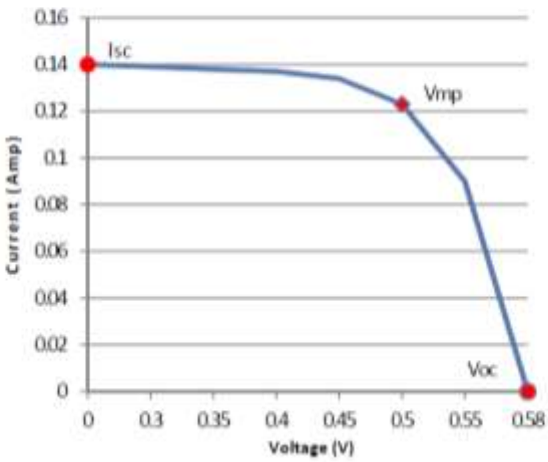

Figure 1: Power current Vs Power voltage curve

The computer model retrieves and uses the solar cell parameters V_{mp}, I_{mp} and short circuit current (I_{sc}) and bus voltage. Given the required average power and peak voltage, the model computes the number of parallel circuits, the number of strings per circuit and the number of cells per string. By-pass diode feature is added to increase reliability. The number of series cells combined with a by-pass diode is called a block. For ease of calculation and display, the model uses this block as the basic element of the array where applicable. There are other efficiency factors that are accounted for. While calculating corrected bus voltage, regulator efficiency, maximum power tracker efficiency and blocking diode loss are all used in computations to generate more realistic results. Solar Array Drive slip ring limit is 5A, therefore, circuits are designed to prevent exceeding that value.

SUBMODELS

The simulation consists of many parts called submodels. These submodels include a solar array wing power calculator, a battery model, a load profile model and a thermal model that interact and retrieve data. All models are highly accurate and fast. In this part each submodel will be handled thoroughly. The initial step is to calculate the solar array requirements as in section II. After the computer outputs are accepted by the user, the computer proceeds on to the next step.

A. Load Profile

Load profile is a file that consists of two different arrays as shown in Figure 2. One of the arrays provides data for instantaneous power demand. The other array provides day or night data. The spacecraft orbit is Low Earth Orbit, and orbit period is 90 minutes, 30 minutes of which is under eclipse (darkness). When the data is fed to the simulator, it first processes the day/night data. If the spacecraft is illuminated, then power required will be the sum of demanded power provided in the load profile plus the power required for charging the batteries of applicable. Therefore the solar array must generate enough power for both. On the other hand, if the spacecraft is under eclipse, then power demanded will be retrieved from the load profile. This demand will be met by batteries only.

The generic load profile used during this work is not limited to 90 minutes only. The whole 90 minute profile can be used repeatedly, or a portion of it can be used. Also profile allows using charging / discharging functions of the battery independently.

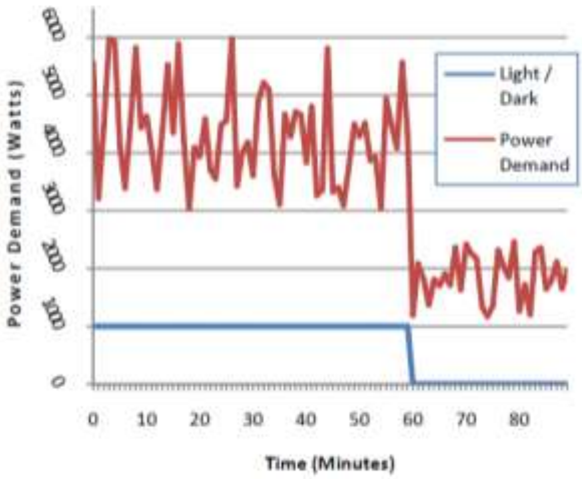

Figure 2: The load profile of the Submodels

B. Battery

Despite the presence of maximum power trackers, the satellite dwells in a dark region where it receives no sunlight. A low earth orbit satellite, depending on its height, has orbit time around 90 minutes. While the solar arrays generate electrical power during around 60 minutes of this orbit time, for roughly 30 minutes no sunlight can be received. Thus, the spacecraft depends on battery cells during darkness. If the batteries cannot provide enough power for basic functions such as maintaining the orbit, the satellite could be lost. There are two basic functions of the battery block: To charge under sunlight by the solar array, and to provide power for the spacecraft while in the eclipse period. Both functions are accurately embedded into the simulator.

As a starting point some parameters are required. The computer starts all calculations based on power required during eclipse time (load demand). Given the demand, the battery cell capacity, the battery bus voltage, eclipse time and initial temperature, the systems generate the series and parallel combination of the battery cells. A battery controller was added to supervise battery functions. The controller checks the battery State-of-Charge (SOC), and the battery temperature. Under sunlight, if the battery is full (over 99.9% SOC) or overheated, the handler will not charge the batteries. During eclipse, the controller will check the battery temperature and desired maximum depth of discharge, and if the batteries are hot and/or excessively depleted below allowable depth of discharge, it will not allow further discharge.

If necessary conditions are met, the controller will charge the battery under sunlight. The charge handler must choose between four charge rates according to the SOC. An unnecessary quick charge would overheat the system, while a slow charge would not be efficient enough to fully charge the batteries. The preset rates are $\frac{C}{2}$, $\frac{C}{10}$, $\frac{C}{20}$ and $\frac{C}{40}$ in Amp/h. Figure 3 shows the curves of the charge efficiency versus state of charge for the specified temperatures for a preset rate of $\frac{C}{2}$. At the other extreme, for a preset rate of $\frac{C}{40}$, the resulting curves are shown in Figure 5, for equivalent temperatures. After the charge rate is chosen by the computer, it looks up to charge rate curve files for the specific charge rate and interpolates the charging efficiency (ηc) according to instantaneous battery temperature and SOC. Charge gained (ΔC), SOC and V_{cell} is calculated as follows (Shah et al, 2015):

$$\Delta C = \frac{ch\arg eerate}{60} * \eta c$$ (charge gained per minute)

(1)

$$SOC_{t+1} = \frac{SOC_t * capacity + \Delta C}{capacity} \qquad (2)$$

$$V_{cell} = V_{cell\max} T^0 C - k * DOD \qquad (3)$$

where *k* is the charging constant and

$$DOD = 1 - SOC \qquad (4)$$

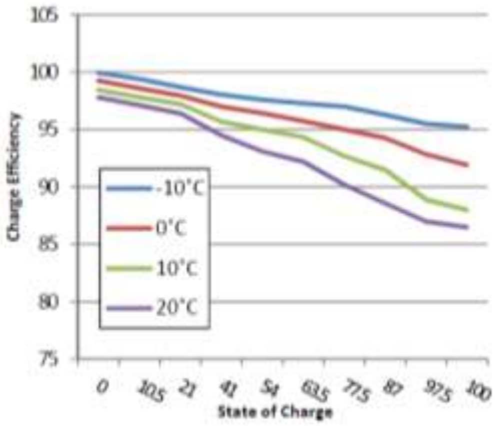

Figure 3: Curves of the charge efficiency versus state of charge for the specified temperatures for a preset rate of $\frac{C}{2}$

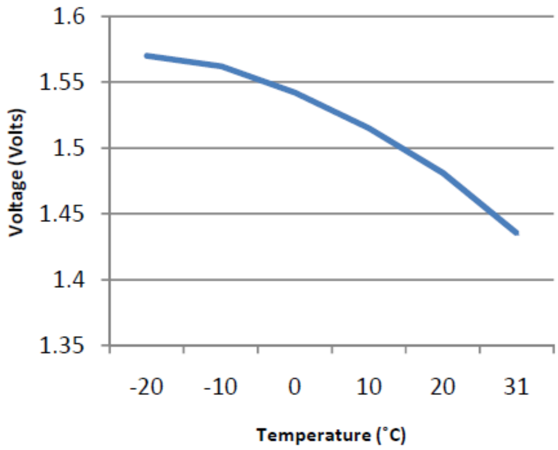

Figure 4: Maximum battery cell voltage capacity

The next step is to calculate resultant current, and update the battery temperature. Maximum battery cell voltage capacity will then be obtained for new battery temperature as shown in Figure 4.

As long as the controller will command charge, the computer will repeat same procedure. Each step represents a minute. According to the updated battery temperature and SOC, the controller can change charge rates, or stop the charging process. When the system cannot get demanded power from the solar array due to eclipse, the batteries will discharge and provide power for the spacecraft. According to Ngoc et al, 2016, discharge logic can be represented as:

$$\Delta C = \frac{I_{bat}}{60} \quad (5)$$

$$SOC_{t+1} = \frac{SOC_t * capacity - \Delta C}{capacity}$$ and

(6)

$$V_{cell} = V_{cell\max} T^0 C - k * DOD \quad (7)$$

While calculating the V_{cell}, SOC and battery power there are important factors that need to be taken into consideration. Battery bus voltage regulator efficiency, charger and discharger efficiency, blocking/by-pass diode losses and discharge converter efficiency must be assessed.

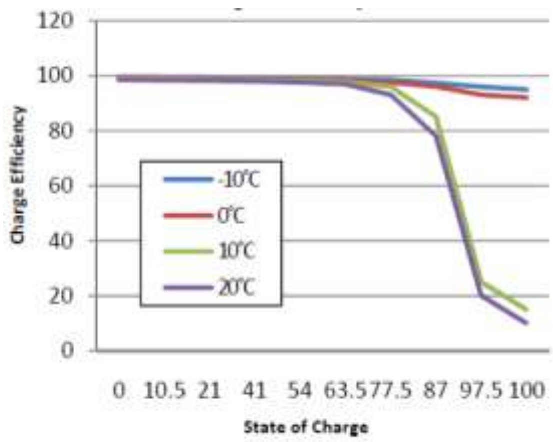

Figure 5: Curves of the charge efficiency versus state of charge for the specified temperatures for a preset rate of $\frac{C}{40}$

C. Solar Array Power

The array consists of series solar cell blocks called the strings, and parallel strings form the circuits. Therefore, the solar array is a combination of series and parallel circuits. The simulation has an algorithm that calculates the prevailing voltage among the parallel strings in all circuits. The prevailing voltage (V_{pr}) will be unregulated circuit voltage (Yousef, 2015).

$$V_{pr}=V_{hi}=(Blocks*cellsperblock*V_{op}) - V_{bl} \quad (8)$$

where V_{bl} is blocking diode loss for each string and V_{op} is the operating voltage. The overall current of a circuit in the wing can be computed according to Loo et al, 2014 as follows:

$$I_{cir} = \sum_{1}^{n} I_{str}(n) \tag{9}$$

It must be kept in mind that when voltage asymmetries occur in parallel circuits, the operating voltages in the I-V curve shifts to mimic the voltage of the string with the highest voltage. Therefore the new string current value must be found using the I-V curve. The overall current of the array wing can be computed as follows (Patel, 2014):

$$I_{wing} = \sum_{1}^{m} I_{cir}(m) \tag{10}$$

Bus power can simply be depicted as:

$$P_{bus} = \eta cr \sum_{1}^{m} I_{cir}(m) * V_{pr} \tag{11}$$

where ηcr is the converter efficiency.
Therefore I_{bus} will be:

$$I_{bus} = \frac{P_{bus}}{V_{bus}} \tag{12}$$

We now have the preset bus voltage, the bus current and the power generated by the wing. Almost all of the time, power generated is greater than power required. This brings out the necessity to shunt, which is also provided in the simulator. The shunt controller gradually leaves out the required strings of the wing until power generated is equal to power required.

D. Thermal Model

The solar array is a thin array of solar cells. The more power required, the more the array has to expand in the design process. Since the maximum power trackers will always motor the array for maximum insolation, it can be assumed that the solar energy will illuminate the surface of the array with an acute angle. The primary heat source will be insolation on the front side of the array. While some of this energy is absorbed and converted to electricity, most of it causes the panel to heat excessively because of the low efficiency of the panels.

Calculation must be made for the shunted area of the array separately, because this part does not convert the insolation to electricity. Earth's reflection of energy (albedo effect) and earth's infrared energy must also be accounted for. Heat release will be via a radiator panel assembled on the rear side of the array. Total energy absorbed is Q_{in} and Q_{out} is the released energy. Insolation is 1367 W/m^2. *A* represents the panel area and αfr, αr, $\eta cell$, K, α, ε_r represent the front cover transparency factor, rear paint transparency factor, cell efficiency, shunt factor, reflection angle and emissivity of radiators (Sharma et al, 2016). Stefan–Boltzmann constant represented by σ is equal to 5.67×10^{-8} W m^{-2} K^{-4}.

$$Q_{in} = \{[(Insolation * \alpha fr * A * (1-\eta cell)*K] + [Insolation * \alpha fr * A * (1-K)] + [Albedo * \alpha r * A * \sin\alpha] + [Infrared * \sin\alpha * \varepsilon_r * A]\} * 60 \qquad (13)$$

$$Q_{out} = T^4 * \sigma * A * \varepsilon_r * 60 \qquad (14)$$

$$Q_{end} = Q_{in} - Q_{out} \text{ and} \qquad (15)$$

T is the array temperature. Change in temperature is depicted as ΔT while W_{cv}, W_p and W_{cr} are weights of the cover, the panel and the rear cover respectively. C_{cv}, C_p and C_{cr} are the specific heat capacities of the cover, the panel and the rear cover respectively. From Kui et al, 2015, the change in temperature is:

$$\Delta T = \frac{Q_{end}}{\left[\left(W_{cv} * C_{cv}\right) + \left(W_p * C_p\right) + \left(W_{cr} * C_{cr}\right)\right]} \qquad (16)$$

A thermal model designed for the batteries was also included. Due to internal resistance and charging/discharging efficiencies, the batteries thermal status will change. Excessive heat should be dumped; otherwise it could cause explosion and/or fire. Moreover, charge and discharge efficiencies change according to battery temperature. To prevent heat build-up, a thermal radiator was designed and utilized for the batteries, which can be depicted by the equation (Chang et al, 2016):

$$Q_{out} = Tb^4 * \sigma * A * \varepsilon b * 60 \qquad (17)$$

The thermal radiator also provides smoother operation, because unless the battery temperature is within the confined limits, the battery handler will stop battery operations.

DISCUSSION AND CONCLUSION

The modeling and simulation software solves design problems of the electrical system of a satellite. The employed components include system design and solidification of a generic solar array, distribution system, backup battery system, load analysis system and a complete thermal model for the battery and solar array. The software includes solar cell to solar array relations, bypass systems, imbalance in array wings, array power generation and shunting model, battery charge and discharge model and a load profile. Load profile is a data stream fed into the computer including illumination or eclipse data. Initial process begins with identification of peak and average power requirements. Given the bus voltage, single cell open circuit voltage and short circuit current, the software calculates number of cells per block, number of blocks per string, number of parallel strings per circuit and number of circuits for the satellite. According to the load profile which includes the day / night flag the battery is either charged or discharged.

The test result is tabulated as follows:

Table 1: Simulation results of the Submodels

Submodels	State-of-Charge	Charge efficiency
Wing Power Calculator	10.5	98%
Battery model	21	96%
Load profile model	41	93%
Thermal model	54	92%

During daylight, power is used to run the system requirements and charge the battery. Excess power is calculated and shunted. On the other hand, during darkness, when the panels cannot provide power, the battery takes over the supply. Therefore the battery ensured the accuracy of the simulation for both charge and discharge features. Battery function included slow and fast charge rates, and is a complex function of various inputs. Since the whole simulation is heavily dependent on temperature, the system also served as a built-in model for thermal cooling.

The simulation part tests and verifies the results. This greatly enhances the reliability in the design process. The motive of this study, which was to precisely model and simulate the complete electrical function of a satellite, was achieved. Also the fast and reliable simulation allows for proper measures to be taken to adapt to environmental and internal changes.

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