Noname manuscript No.
(will be inserted by the editor)

Modelling and Simulation of the Power Subsystem of a LEO satellite

Varsha Parthasarathy
Philip Ferguson

Received: date / Accepted: date

Abstract Modelling and simulation are important tools in research and satellite design. To enhance the reliability of the satellite, a power subsystem simulator has been furnished within the MATLAB/SIMULINK environment as a useful tool. Statistical process control (SPC) is used to perform post-simulation analysis to detect faults in the system which is discussed briefly. This paper presents an architectural template for a simple power subsystem of small satellite missions. The simulator has a modular structure with mathematical models of system components like batteries, solar arrays, and the power control unit (PCU). This simulation helps with preliminary sizing and the verification of rating co-ordination of selected electrical components in an unregulated bus voltage scenario. Battery thermal analysis is not considered. The solar arrays, batteries and PCU are modelled using square waves, a voltage-capacity lookup table and repeating sequences respectively. The design is adapted to simulate the power subsystem of ManitobaSat, a student satellite built as part of the Canadian CubeSat project. Batteries and solar cells are well sized to supply the necessary power during daylight and eclipse period using lookup tables, relays and conditional switches. Outputs from this model will be used by the SPC module to detect out of control conditions as the simulation will describe the baseline or 'normal' behaviour.

Keywords Power Subsystem · Simulation · Modelling · Statistical Process Control · Satellite

Varsha Parthasarathy
Department of Mechanical Engineering
University of Manitoba
E-mail: parthasv@myumanitoba.ca

Philip Ferguson
NSERC/Magellan Aerospace Industrial Research Chair in Satellite Engineering
Associate Professor, Department of Mechanical Engineering
University of Manitoba
E-mail: philip.ferguson@umanitoba.ca

1 Introduction

During the past few years, the number of CubeSats launched has been at an all time high. Assembling real satellite prototypes for validation and testing can be extremely expensive and laborious. With advances in modelling, simulation, computation and engineering for research and development purposes, it is now possible to reduce cost and time in satellite design. The need for easier and cheaper testing and verification of design requirements have led to the widespread use of simulators in the aerospace industry. That being said, simulators are yet to be used extensively for small satellites[1].

Specifically, a power subsystem simulator for a small satellite can be used to assess system operations, behaviour of the power source and subsystems, and analyse the power consumption and generation cycles throughout the mission. A dynamic simulation helps validate our static power analysis and establish whether the choice and sizing of electrical components meets the requirements. They work as ready to use software tools in cases where verification of operation of control algorithms and strategies isn't possible[2].

A simple and accurate power subsystem simulator is developed in Matlab/Simulink for ManitobaSat-1, a student-built CubeSat mission that is part of the Canadian CubeSat project sponsored by the Canadian Space Agency. Traditional languages like C and FORTRAN typically result in large code making it difficult to modify and understand. Unlike them, the modular approach with block diagrams facilitates easier handling and flexibility on Matlab/Simulink. The design makes it easy to make modifications as needed. As such, this simulator can be applied to any satellite by simply adapting mission parameters in each of the blocks.

This simulator also has a warning system that helps develop and test the spacecraft overall is discussed briefly. Identifying faults and anomalies in this power subsystem as quickly as possible is paramount to a mission's success. This is done with a SPC interface which raises alarms and warnings to perform health monitoring of the satellite's power subsystem. It is used to estimate the control limits and trigger warnings when close to failure or when the system isn't functioning as expected[3]. SPC is executed in real-time to identify anomalies in the system that might otherwise not be detected by operators before they lead to catastrophic failures.

1.1 Literature review

There have been a few simulators for a CubeSat's electrical power subsystem in the past. Many of these sophisticated models employ nodal analyses where the circuit is analysed in a complex network using the nodes as inter-connective reference points. Voltage and current values at each node are calculated and the entire circuit is solved using simulatenous equations [4]. *Melone* [5] developed an electrical power subsystem simulator for preliminary design and test of the TINYSCOPE nanosatellite, a three-axis stabilized, low earth orbiting, electro-optical imager. It had high power requirements which posed challenges in terms of power collection, energy storage and power management and distribution. This was overcome with an analytical-numeric approach. *Bauer* [4] describes a method to perform an electrical analysis and a transient thermal analysis of a satellite electric power subsystem. The program he developed runs the power subsystem through one or more complete orbits and plots curves to investigate voltages, currents and temperature changes. But, this program doesn't consider load variation for various subsystems during the mission. *Kim et al.* [6] performed statistical analysis of the satellite electrical power subsystem's on-orbit failures and anomalies with a focus on comparison between LEO and GEO satellites. Partial failures are classified into different classes depending on their severity. Non-parametric estimation is used to study the failure and degradation behaviour[6].

Though there have been simulators developed, none of them have a reliability warning system which can detect anomalies in the system. SPC hasn't been used in space system simulators yet. This is the first simulation model that has an embedded real-time statistical process component. This allows for early fault detection and save expensive missions.

2 ManitobaSat-1 Power Subsystem

ManitobaSat-1 is a 3U CubeSat weighing approximately 3.8 kg with deployable solar panels. This satellite aims to study space weathering effects on geological samples by exposing them to direct solar radiation [7]. The satellite consists of an attitude control and determination system (ADCS), communication subsystem including radio RX (uplink receiver) and radio TX (downlink receiver), command and data handling subsystem (C&DH), power subsystem and payload. The payload camera takes images of the samples at regular intervals. Solar energy generated by solar arrays is the only energy source for the satellite. For storage, Lithium Iron Phosphate battery cells (LFP-18650HT) are connected to the bus providing an unregulated voltage to all subsystems[8]. Concept of operations helps prepare a power budget with power requirements of different subsystems. A static power analysis provides two arrays, one on the front and one on the back, of Spectrolab XTJ Prime solar cells[9]. The front array has five strings of three cells and the back array consists of two strings with three cells each. Only the front array is sun-facing during normal operations.

2.1 Static Power Analysis

The CubeSat has an estimated mission life of 2 years with peak power consumption of 12.65 W. The estimated orbit average power (different from the peak power) budget is given in Table 1. With static power budgeting, it is found that the orbit average solar array power generation is 9.50 W (see Table 2) and the orbit average power consumption is 7 W. As the generation is greater than the consumption, the solar array sizing is suitable.

The worst-case power time line is formulated on the basis of the power budget and concept of operations. It is observed that the power consumption repeats itself after every three orbits. Important factors influencing the power consumption time line are:
a) The satellite remains in sun for a minimum of 61.15(%) of the orbit time. This is obtained from Satellite Tool Kit (STK) simulations. The worst case eclipse occurs on 2nd Jan. No payload images are taken during eclipse and most components function at minimum power to reduce power consumption. Each orbit is 5560 seconds.

Table 1 Power Budget

Component	Orbit Avg. Power Consump. (Watts)
Thermal	1.125
C&DH	1.054
Radio RX	0.594
Radio TX	0.631
Power Control Unit	0.525
Payload Camera	0.014
Payload Controller	0.020
ADCS	3
Total	6.96

Table 2 Power Generation Calculations

Parameter	Value	Unit
Maximum power per string	3.42	W
Worst-case pointing accuracy	24	deg
Total power from the main array	15.54	W
Orbit Avg. Power Generation	9.50	W

In three orbits or 16680 seconds, worst-case eclipse occurs between 3400-5560 seconds, 8960-11120 seconds and 14520-16680 seconds. Satellite Tool Kit (STK) is used to obtain the eclipse times assuming ISS orbit for the satellite.
b) Payload images are taken once every two hours.
c) Communication between the satellite and the University of Manitoba ground station takes place five times per day on an average. Worst contact duration is 400 seconds. Only worst-case communication times are used for simulation purposes.

The depth of discharge of the battery, battery capacity and power draw during eclipse are calculated. The design is based on the concept of Direct Energy Transfer (DET) systems which allows the bus voltage to fluctuate with the state of charge of the battery as the bus voltage is maintained equal to the voltage across the battery [10]. DET systems have been successfully implemented for several University satellite missions [11]. In the DET solar array interface, the solar arrays are directly connected to the battery and the voltage through the battery and solar arrays is maintained at the same level [11]. A conceptual model of ManitobaSat-1's power subsystem is given in Figure 2.

3 Simulation

3.1 Simulator Model Structure

The design process of developing the simulator largely depends on parameters like pointing accuracy of the attitude control system, data acquisition cycles, imaging intervals, worst-case eclipse period and when it occurs, orbital time period and concept of operations. The simulator only considers the normal operations, that is, the spacecraft is assumed to have completed the commissioning process, the mission objectives are being fulfilled and the payload is functioning as intended. From the analysis, factors which influence the size of the battery like charging cycles, discharging cycles and depth of the discharge will be determined. Since, the power subsystem should be able to support worst-case and peak power requirements at the worst-case eclipse, the simulator uses peak power calculations with positive margins.

3.2 Matlab/Simulink Implementation

The simulator model structure is formulated and can be seen in Figure 2.

Power Consumption Input The values for the power consumption with corresponding orbit time for first three orbits is fed as a repeating table for the entire simulation time (Figure 3).

Battery Module From the conceptual model, it can be seen that the integrator gives the capacity of the battery. The corresponding bus voltage is deduced from it by using a Simulink lookup table which uses values from the voltage vs capacity curve for the battery given in its data sheet [8].

Solar Array Module The solar cell configuration is three cells per string with five strings facing Sun. Using the bus voltage, the solar array lookup table gives the solar array current when no eclipse occurs. The Simulink lookup table contains values from the V vs I curve for the solar cells from their data sheet [9]. Again, linear interpolation methods are applied to get the corresponding solar array current. The voltage across the solar arrays is equal to the bus voltage owing to DET.

Eclipse Flag With the solar array current, the condition of eclipse is introduced. A vector is created which stores zero when an eclipse occurs and one during non-eclipse period for corresponding orbit time for the first three orbits. This is entered into a repeating table with the solar array current. A switch is implemented to pass the current when the flag value is one and zero current when the flag is zero, that is, during eclipse time. It is important to note that the power time line is synchronized with the eclipse flag such that the worst-case power consumption occurs right after the eclipse time. The worst case pointing accuracy of the ADCS is $\pm 24°$ and its cosine is multiplied to the current to obtain the actual solar array current.

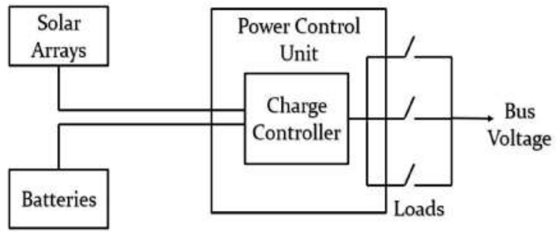

Fig. 1 Power Subsystem System Block Diagram

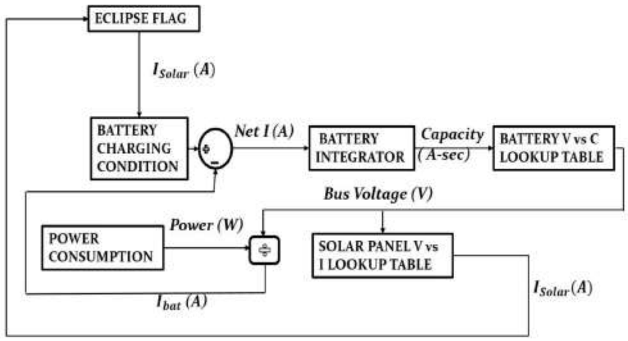

Fig. 2 Power Simulator Functional Model

Battery Charging Condition The battery data sheet specifies a cut-off voltage of 3.54 V (each battery cell)[8]. For the 2S3P arrangement of the battery, the cut-off voltage is 7.08 V. When this voltage is reached, the battery stops charging until the voltage again drops to a value lesser than 7.08 V.

Internal Resistance of the battery From the data sheet, the average internal resistance of the battery is 20 mOhms [8]. Hence, for a 2S3P configuration, the equivalent resistance is approximately 13 mOhms.

We know,

$$V_{Electromotive\ Force} = V_{Bus} - I_{Battery}R \tag{1}$$

where R is the internal resistance of the battery. This equation is used to calculate the electromotive force (EMF) [12]. EMF is the battery's internal driving force used to provide energy to a load [13], but differs from the bus voltage due to internal resistance of the battery. The current going into the battery is considered positive and current going out of the battery is negative.

Initial conditions of the Integrator The initial conditions for the battery are 4500 mAhr. This is chosen as a reasonable initial SOC(97%) [8]. The saturation limits are 4650 mAhr and 1450 mAhr as calculated from the data sheet for the selected configuration. 1550 mAhr is the upper limit of the capacity for the battery and with a 2S3P configuration, it comes to 4650 mAhr. 1450 mAhr is the lower limit of the capacity [8].

The functional model of the simulator is given in Figure 2. All calculations are done in SI units. The power consumption time line and eclipse flag are represented in Figure 3. Since this is the worst-case power simulation, battery heaters are on all the time.

4 Results and Discussion

The results of the simulator are provided below.

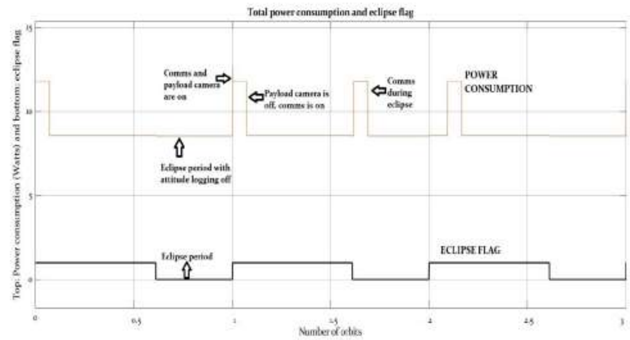

Fig. 3 Output power consumption and eclipse flag waveforms of the satellite over three orbits

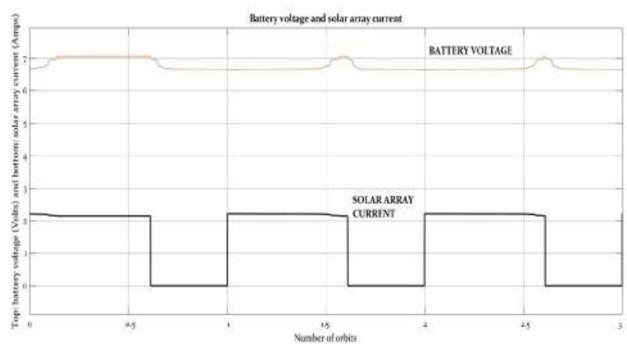

Fig. 4 Output battery voltage and solar array current waveforms for three orbits

The following trends can be observed from the results which help us check if the the simulator is functioning appropriately:

a) It is observed from Figure 4 that the solar array current isn't completely constant throughout the sun-facing time of an orbit. This can be explained by looking at the corresponding changes in the battery voltage. The current starts to decrease with an increase in the voltage midway through an orbit. At the corresponding times in Figure 4 and 5, it is clear that the solar array current is least or 0 during eclipse as expected.

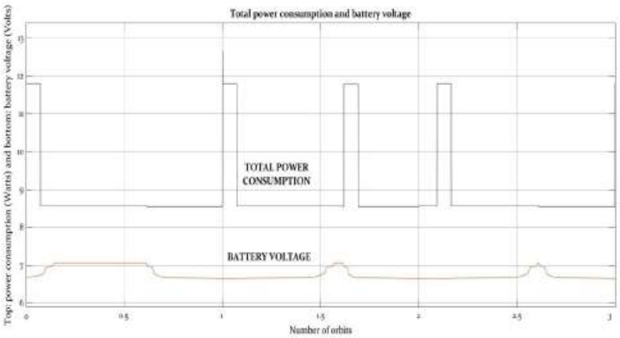

Fig. 5 Output waveforms of the power consumption and the battery voltage for three orbits

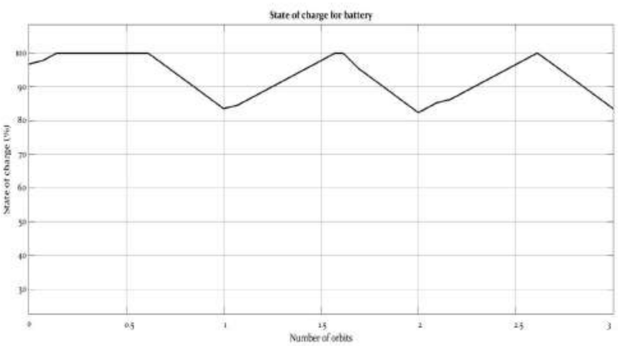

Fig. 6 State of charge of battery over three orbits

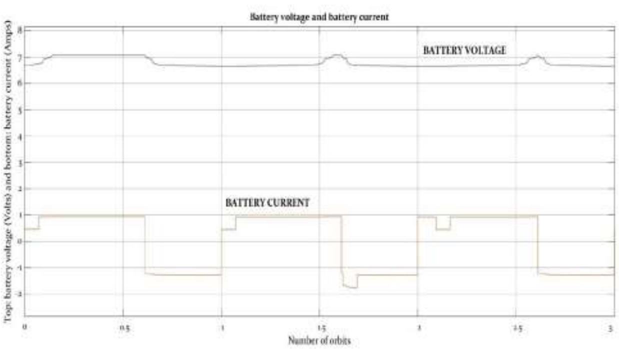

Fig. 7 Battery voltage and current into the battery for three orbits

b) In Figure 6, the initial state of charge of the battery is about 97%. The battery is charged during non-eclipse time to a 100% SOC and discharges during eclipse leading to a lower SOC. This is understandable as solar current is not available during eclipse.

c) Looking at Figure 7, the current into the battery is positive during sun-facing time and negative during eclipse time.

d) Power consumption is highest when there is communication and payload camera is on and lowest when eclipse takes place without communication and with attitude logging being off during eclipse. Worst-case communication occurs between 1- 400 seconds, 5560-5960 seconds, 9000-9400 seconds and 11641-12041 seconds. These have been obtained using Satellite Tool Kit (STK). Between 9000-9400 seconds, eclipse and communication take place simultaneously and between 5560-5562 seconds, the payload camera takes an image and communication takes place. These help analyse worst-case power consumption scenarios.

e) The eclipse flag toggles between zero and one. Zero denotes eclipse period and one denotes no eclipse. The plot when compared with the power consumption trend shows that eclipse causes a very minor decrease in power consumption as attitude logging doesn't take place.

The state of charge drops during eclipse as mentioned earlier. We get the depth of discharge by dividing the power draw in eclipse by the battery capacity for the designed configuration. The battery capacity is 1728 Wmin [8]. The depth of discharge during eclipse comes to 17.8% which is reasonable (see Figure 6). The peak current going into the battery is 0.9286 A. This is lesser than the maximum charge current (3 A for 2S3P configuration) specified for the battery in its datasheet [8]. The results confirm that the selected battery sizing is suitable. Together, the static and dynamic analyses prove that the power subsystem design is suitable and sufficient.

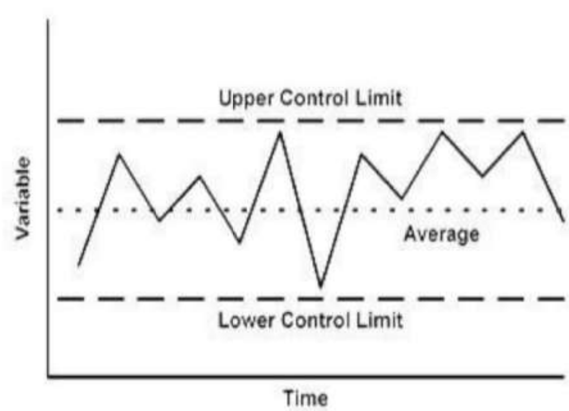

Fig. 8 A Typical Control Chart[3]

5 Statistical Process Control Module

The power subsystem simulator is modified to introduce different kinds of failures such as an increase in the internal resistance of the battery, solar array string failure and excessive power consumption. Also, measurement noise is introduced into battery voltage and current with a mean of 0 V and standard deviation of 50 mV and a mean of 0 A and standard deviation of 20 mA respectively. The results are processed using the SPC module to perform post-simulation analysis. Though SPC has been part of the space industry in the form of manufacturing quality control, it has not been used in satellite operations and maintenance. We hypothesize that SPC tools will be beneficial to this research in many ways. It will enable monitoring a constellation of satellites with ease. The process variables will be analysed through automated control charts to detect issues that require attention to monitor the power subsystem's health. Patterns, trends and out of control conditions will aid in continually verifying if the "process" (i.e., the power subsystem's operations) is in equilibrium. It will act as a visualization tool reporting on system's performance. This also reduces dependence on human resources. A detailed description of the SPC module is beyond the scope of this paper. To demonstrate how this module enhances the reliability of the satellite by detecting failures, excessive battery resistance leading to battery degradation is considered. The failure is introduced at a particular time in the system and the SPC module should be able to detect it as soon as possible before hindering the satellite functions.

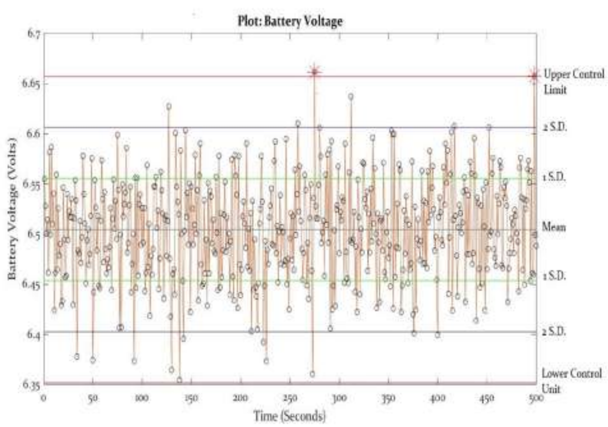

Fig. 9 Battery Voltage Control Chart

Process behaviour charts called control charts are used to determine if a process is in equilibrium or in statistical control[14]. A typical control chart as seen in Figure 8 has a centre line, upper and lower control limit lines[3]. These are used along with SPC control rules to detect anomalies.

The maximum resistance value for each battery cell is 45 mOhms according to its data sheet [8]. With a 2S3P configuration, that gives an equivalent resistance of 30 mOhms. An anomaly with a mean of 70 mOhms is introduced in the simulation to test the SPC module. The resistance increases gradually from normal to ab-

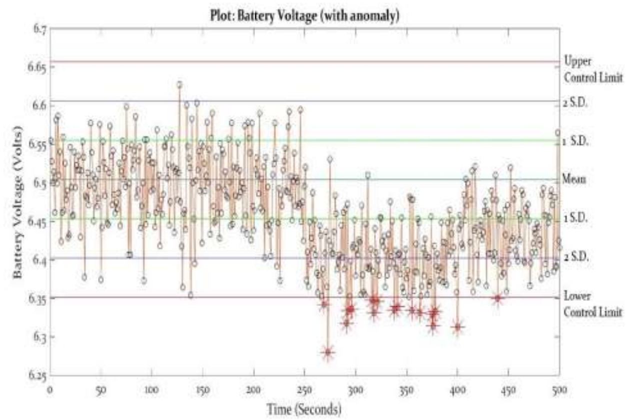

Fig. 10 Battery Voltage (with anomaly) Control Chart

normally high values after half the simulation time. The control chart for the battery voltage with and without the resistance anomaly can be seen in Figures 9 and 10.

The large stars on each plot depict the points outside of the control limits indicating an anomaly in the system. The abnormality in resistance was introduced at half the time of the simulation period. As expected, Figure 10 has a lot of beyond limit points right after 250 seconds indicating an abrupt change in the system cautioning the ground station operator with a triggered warning. It is important to note that the plot without anomaly might have a few beyond limit points but that doesn't indicate any anomaly. Regular variation is common in a system leading to false alarms [3].

6 Conclusion

The power subsystem simulator described in this paper provides important parameters like output voltages, state of charge of the battery and solar array current and is realistic and easy to understand. It provides an interface that can be applied to any satellite with trivial edits. The results obtained show that the power consumed by all subsystems is lesser than the power generated by solar arrays which substantiates confidence in the design. The SPC module can be further developed by introducing more failures and studying the behavior of the system using the simulator. The presented model can be applied with basic electrical knowledge and its modular design allows to make modifications effortlessly through future research.

Acknowledgements The authors would like to thank Natural Sciences and Engineering Research Council (NSERC) for their financial support.

References

1. A. Berres, M. Berlin, A. Kotz , T. Terzibaschian and A. Gerndt , "A generic Simulink model template for simulation of small satellites", 7th Symposium on Small Satellites for Earth Observation, pp.247-250 (2009)
2. Y. Merkuryev, R. Zobel and E. Kerckhoffs, "Simulation of spacecraft attitude and orbit dynamics", 19th European Conference on Modelling and Simulation, vol. 4, pp. 1-6 (2005)
3. R. Biswas, M. Masud and E. Kabir, "Shewhart control chart for individual measurement: An application in a Weaving Mill", Australasian Journal of business, social science and information technology, vol. 2, no. 2 (2016)
4. P. Bauer, "Computer Simulation of Satellite Electrical Power Systems", 1969 Intersociety Energy Conversion Engineering Conference, vol. AES-5, no. 6, pp. 934-942 (1969)
5. C.W. Melone, "Preliminary Design, Simulation, and Test of the Electrical Power Subsystem of the TINYSCOPE Nanosatellite", Research Thesis, Naval Postgraduate School (2009)
6. S. Kim, J.F. Castet and J.H. Saleh, "Satellite electrical power subsystem: Statistical analysis of on-orbit anomalies and failures", IEEE Aerospace Conference Proceedings, pp.1-12 (2011)
7. A. Yahyaabadi, M. Driedger, V. Parthasarathy, R. Sahani, A. Carvey, T. Rahman, V. Platero, J. Campos, P. Ferguson, "ManitobaSat-1: Making Space for Innovation", 2019 IEEE Canadian Conference on Electrical and Computer Engineering (CCECE) (2019)
8. AA Portable Power Corp., "Battery Product Specification", Data Sheet, www.batteryspace.com
9. Spectrolab, "29.5 % Next Triple Junction (XTJ)", Data Sheet, www.spectrolab.com
10. C. Clark and A.L. Mazarias, "Power System Challenges for small satellite missions", Proceedings of the 4S Symposium : small satellites, systems and services, vol. 625 (2006)
11. D. Erb, S. Rawashdeh and J.E. Lumpp, "Evaluation of Solar Array Peak Power Tracking Technologies for CubeSats", 25th Annual Proceedings of the AIAA/USU Conference on Small Satellites, vol. 6, no. 11 (2011)
12. A.K. Singal, "The Internal Resistance of a Battery", Technical Report, Indian Space Research Organization, pp. 1-3 (2013)
13. V. Pop, H.J. Bergveld, D. Danilov, P. Regtien and P. Notten, "Battery Management Systems: Accurate State of Charge Indication for Battery-Powered Applications", Philips Research Book Series Volume 9, Springer, The Netherlands (2008)
14. C.D. Montgomery, "Introduction to Statistical Quality Control 6th Edition", John Wiley & Sons, Inc., U.S.A (2009)