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Comparative Study Of Energy Balance For A 3U CubeSat For Different Orientation Flight Scenarios

Comparative Study Of Energy Balance For A 3U CubeSat For Different Orientation Flight Scenarios

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Abstract. The CubeSat standard makes development, launch, and operation possible for those with no experience in space technology to participate. The power demand coverage by the power supply system during all modes of operation is a critical issue that needs to focus on to avoid satellite mission failure. Therefore, the solar array (SA) and battery pack should be sized precisely based on the satellite architecture and flight orientation scenarios. A 3U Cube-Satellite has been used as a case study to evaluate the effect of satellite flight orientation (Nadir and sun tracking) on the energy balance between the supply system and the load demand and find out the appropriate flight orientation for achieving the mission successfully. The orbit simulation has been done via STK environment for both flight orientation scenarios with the same load profile where the electrical power system (EPS) has confronted the worst-case scenario of operation. Furthermore, the unregulated bus configuration has been selected to simulate EPS response during the orbit period using SIMULINK for different orientation flight scenarios.

1. Introduction

A new class of highly capable smaller, faster, and less expensive satellites has been developed to complement the traditional large satellite systems [1],[2]. This development has been facilitated by the combination of commercially available microelectronic technologies created for terrestrial use and adapted to the space environment. As a result, it has been noted that there is growing interest in the design of small and micro satellites, and more specifically in the use of simulation analysis for the deployment and control of satellite subsystems [3], [4]. A tiny satellite lifespan is greatly influenced by its electrical power system (EPS). In order to efficiently meet a number of objectives, the EPS must be able to generate, manage, store, control, safeguard, and distribute power to the platform equipment and payloads of spacecraft [5].

The EPS must manage peak, pulse, and transient power requirements as well as the battery charge/discharge cycle while preventing spacecraft instability and performance degradation from a regulatory perspective [6].

A primary power source, an energy storage, a power management unit that deals with power conditioning and charge/discharge control, and a power distribution unit are the four main building blocks that, at a certain level of abstraction, can be separated to form a fairly general architecture of an

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EPS [4]. [7], The sizing of the subsystems and the design of an efficient power management strategy are complex and critical tasks usually undertaken in the satellite design phase [8].

A spacecraft power system conceptual design involves the best possible choice of technology for various components, including solar cells, solar arrays and batteries. The electrical architecture of spacecraft, however, is not conventional and must typically be customized almost case by case. [9].

The main goal of this paper, first is to develop an energy balance budget model for a small earth observation 3U CubeSat, second is to ensure the energy sufficiency of the satellite during the whole designed modes of operations with the different flight modes (nadir flight – sun tracking flight), third a comparative Trade off synthesis of the energy budget of the CubeSat during the different flight modes to reach the optimum flight mode which is compatible with the mission flight scenarios.

Also, the PSS model with the power budget calculation will be simulated by using the MATLAB software program. The state of charge of the battery (SOC) will be calculated during worst-case scenario to prove that the battery is operating in an optimum condition as the power budget calculation

2. Space mission specifications and power supply system description

2.1. Mission analysis

The proposed satellite architecture is shown in Figure 1, where it consists of two deployable solar panels placed in the +Y direction perpendicular to the orbital plane and one body-mounted solar panel. This satellite is employed for earth observation and remote sensing for two years in a sun-synchronous orbit which keeps the satellite seeing the sun at a fixed time every day with an inclination angle 97.99^{o}. The orbit altitude is 600 km where the maximum eclipse time is 35.45 min during 96.66 min orbital period.

2.2. Spacecraft bus configuration

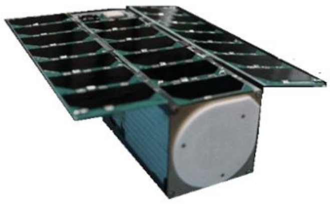


Figure 1. The proposed 3U CubeSat architecture.

An unregulated unified bus configuration has been chosen as in fig. with 16.8V voltage level to decrease the number of control domains where the battery pack has been latched directly to the bus and just a regulator for each solar panel as shown in Figure 2. The battery pack shares the solar panels supplying power to the load through the daylight based on the modes of operation besides delivering power as a unified power source at the eclipse time. Therefore, there is a stress on the battery pack during mission's lifetime.

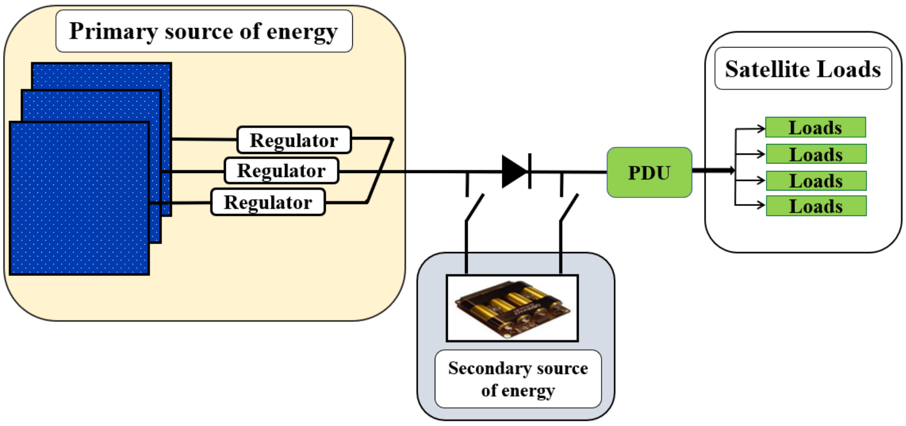


Figure 2. Simple Configuration of EPS.

3. Power Budgeting calculations

3.1. Subsystems power consumption

ADCS, EPS, OBC, and communication receiver are necessary subsystems that are called critical flight systems. These subsystems must work all the time as a continuous load while payload and data-link subsystems are mission loads that can be turned off as short-time loads. This concept of operation is elaborated in Table 1.

Table 1. Satellite Loads.

Subsystem	Power (W)
Attitude Determination and Control system (ADCS)	4
Electrical Power System (EPS)	0.7
Payload (camera)	7
Communication Transmitter (Tx)	12
Communication Receiver (Rx)	0.5
Onboard Computer (OBC)	1
GPS receiver	0.42

Table 2. Subsystems average power consumption during each mode of operation (Wh).

	DTI[1]	DnTI[2]	DTnI[3]	DnTnI[4]	ET[5]	EnT[6]
ADCS	4	4	4	4	4	0
EPS	0.7	0.7	0.7	0.7	0.7	0.7
Camera	1.4	0.7	0	0	0	0
Tx	2.4	0	2.4	0	2.4	0
Rx	0.5	0.5	0.5	0.5	0.5	0.5
OBC	1	1	1	1	1	1
GPS	0.42	0.42	0.42	0.42	0.42	0.42
Total	10.42	7.32	9.02	6.62	9.02	2.62

[1] Daylight with Transmission and Imaging.
[2] Daylight without Transmission but with Imaging.
[3] Daylight with Transmission but without Imaging.
[4] Daylight without Transmission and Imaging.
[5] Eclipse with Transmission.
[6] Eclipse without Transmission.

3.2. Solar array sizing

According to limited surface area and high-power demand, a special solar cell with high efficiency is required for building up the solar panels. A quadruple-junction GaAs Solar Cell has been chosen with mechanical and electrical parameters shown in Table 3 and Table 4 respectively. Firstly, the required power from the solar panels should be determined during the daylight to recharge the battery pack besides covering the load power demand which is calculated from the following equation at the worst-case scenario as [8]:

$$P_{SA}=\left(\frac{P_d T_d}{X_d}+\frac{P_e T_e}{X_e}\right)\frac{1}{T_d} \quad (1)$$

Where:

- P_d and T_d are daylight power demand and the daylight time,
- P_e and T_e are the consumed power from battery during eclipse and the eclipse time,
- X_d and X_e efficiency of charging and discharging paths, their values according to [8] are 0.8, 0.6 respectively.

Table 3. Solar cells mechanical parameters.

Parameter	Value	Unit
Length	8	cm
Width	4	cm
Area	32	cm^2
Weight	84	mg/ cm^2

Table 4. Solar cells electrical parameters.

Parameter	Value	Unit
Open circuit voltage (V_{oc})	3.292	V
Short circuit current (I_{sc})	0.4533	A
Voltage at maximum power (V_{mp})	2.866	V
Current at maximum power (I_{mp})	0.428	A
max power (P_{max})	1.227	W
Efficiency (η) at EOL	29.7%	

The space environment has a bad effect on the solar panel performance which shortens its life. Therefore, the degradation factors are considered to estimate the solar panel output power at the end of life (EoL) as in the following equation:

$$P_{EoL}=S\ \eta\ L_d\ N\ A_{cell}\cos\alpha \quad (2)$$

Where:

- S is the solar constant 135.3 mW/cm^2,
- η is the solar cell efficiency,
- N is the total number of cells,
- A_{cell} is the cell area,
- α is the angle between sun vector and the normal of the solar array,
- L_d is the degradation factor which can be evaluated from

$$L_d=(1\text{-degradiation per year })^{\text{life time}} \quad (3)$$

According to [8], GaAs solar cells degrade by 2.75% per year.

Finally, the total number of cells for each panel can be determined. Subsequently, the area of the panel can be calculated as:

$$A_{panel}=\frac{NA_{cell}}{FF} \tag{4}$$

Where: FF is the filling factor which is about 80%.

3.3. The battery pack sizing

Many LEO missions in recent years have emphasized that Li-Ion batteries are viable because of their broad market accessibility, low cost and high specific energy besides low self-discharge current and storage at ambient temperature.

There is a limitation in the battery pack depth of discharge (DoD) which is 30% for LEO and 70% for GEO. Therefore, DoD should be considered when the battery pack capacity is calculated to ensure mission accomplishment without failure. The needed energy from the battery is estimated in the worst-case scenario (the battery delivers power in eclipse and during peak load demand during daylight which related with the misorientation away from sun for imaging and/or transmitting data) as in equation(5).

$$E_{bat} \geq \frac{P_e\ T_e+P_p\ T_p}{DoD} \tag{5}$$

Where: P_p ,T_p are the power required from battery at day light and the time period of peak demand.

It is recommended to use a battery that has a space heritage, for this reason the suitable battery pack for the mission has the data listed in Table 5. the battery pack composed of four cells in series taking into consideration the regulations from ESA power standard (ESA-PSS-02-10).

Table 5. Battery cell parameters from GOMSPACE battery datasheet.

Parameter	Value
Cell Capacity (mAh)	2600
Cell voltage (V)	4.2
DoD	30%

4. Results and Discussion

Orbit propagation is performed in STK where the orbit path footprint on the ground is shown in Figure 3. From the figure, it is obvious that the proposed cube-sat. visits the same area on the ground at the same time of the day and the suggested deployable architecture reduces the dynamics in the power generation with a proper attitude control to stabilize the satellite.

The simulation is performed using the mission parameters as input to get the output power profile from solar panel for both flight orientations sun-tracking and Nadir as shown in Figure 4 and Figure 5 respectively.

In Figure 4, the sun-tracking flight orientation output power almost constant at maximum power point during the daylight time which permits charging battery and covering the load power demand without battery intervention. However, the flight orientation changing in the first two orbit periods for taking photos makes the solar panels working at low power point where the battery should share the load power demand with the solar panel.

Figure 5 depicts the output power profile of the solar panel takes a bell shape due to the change in the angle between the sun vector and the normal of the solar panel which allows the battery to work in daylight and eclipse times. This makes stress on the battery most of the time and decreases its lifetime.

Figure 3. Sun synchronous orbit path on ground.

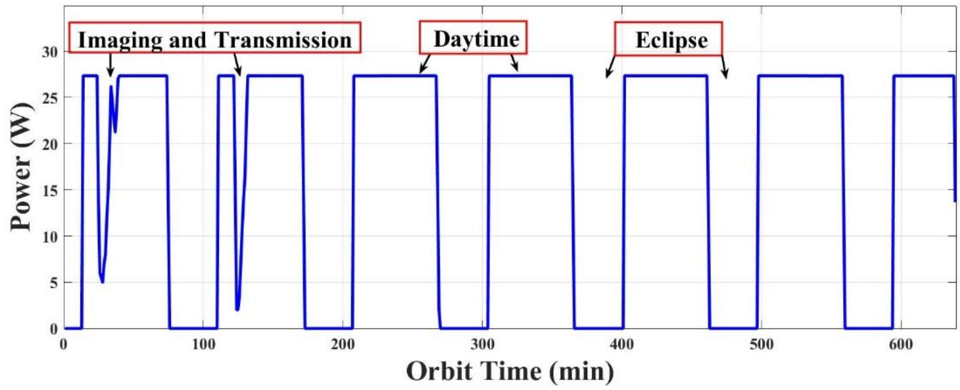
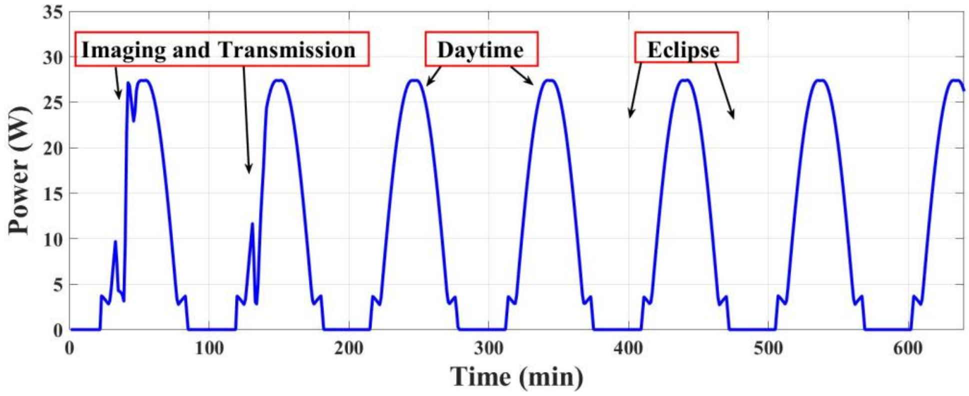


Figure 4. Power Generation profile during sun tracking mode.



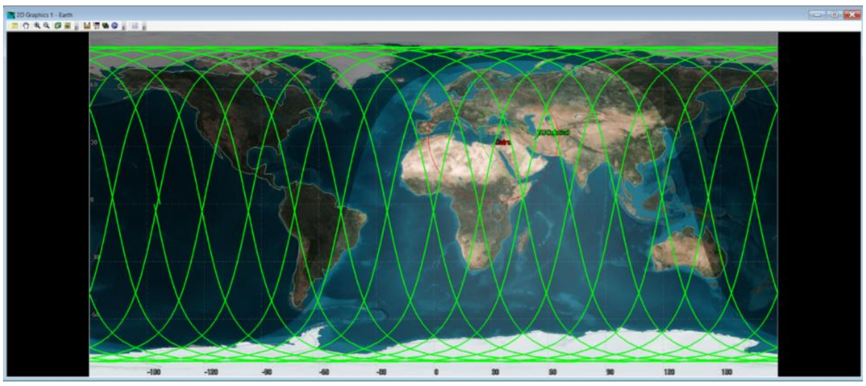


Figure 5. Power generation profile during nadir operation.

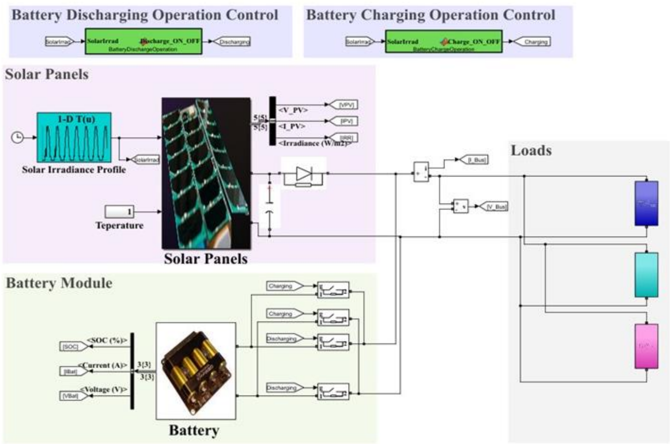


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Figure 6. SIMULINK model for EPS with unregulated bus.

A SIMULINK model as shown in Figure 6 has been built to figure out the energy balance during various modes of operation. The model is composed of three main parts solar panels, Battery pack module and satellite loads. For solar panel, a 3-parallel and 7-series cells arrangement is used. the solar irradiance profile based on satellite orientation is the governing input that forms the power profile. The battery module consists of 4-series cells each of a 4.2 voltage. Battery operation is run by two functions that created to control the switching of the module. Finally, the load profile is operated by a function that imitates the operation sequence of each load. The three blocks in load area emulates the types of loads within satellite. The first one is the housekeeping loads that is responsible for keeping the satellite alive. The other two blocks are payload and transmitter. Each simulation time step represents one minute in orbit. Simulation is run for 640 minutes of orbit time and the results are figured out as follows.

At sun-tracking flight orientation, the State of Charge (SoC) of the battery has not decreased less than 70% as shown in

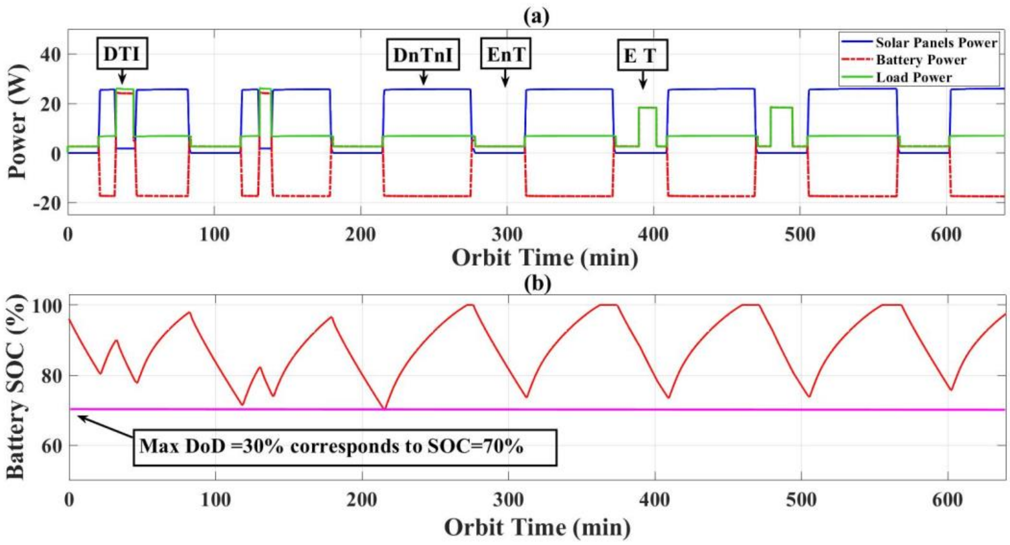

Figure 7 and reached 100% SoC at daylight time, which permits the battery finishing the mission without failure. While Nadir flight orientation makes SoC decreasing less than 60% and not reaching 100% SoC which will affect the mission life time as shown in Figure 8.

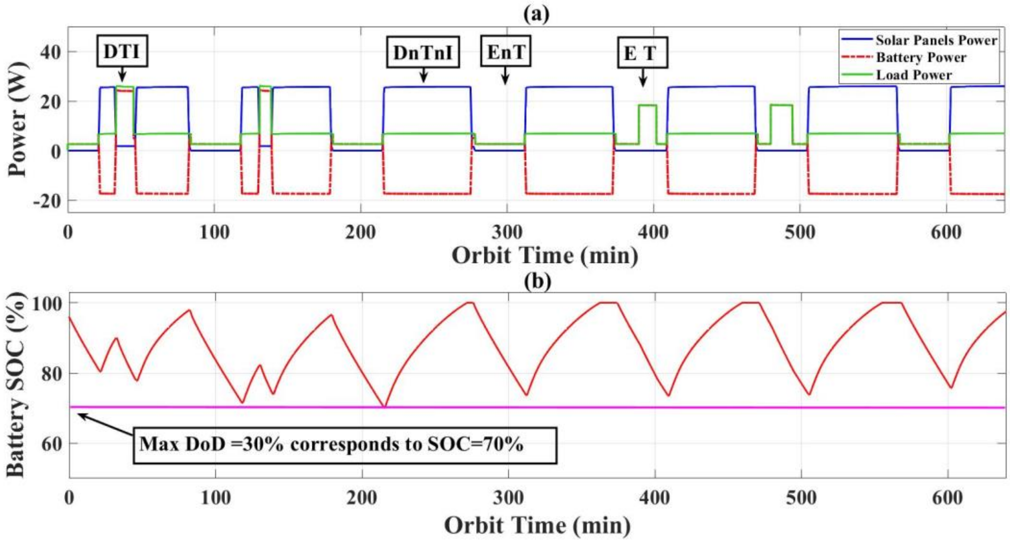

Figure 7. Sun Tracking mode (a) Powers of solar panels, battery and loads (b) Battery SOC with time.

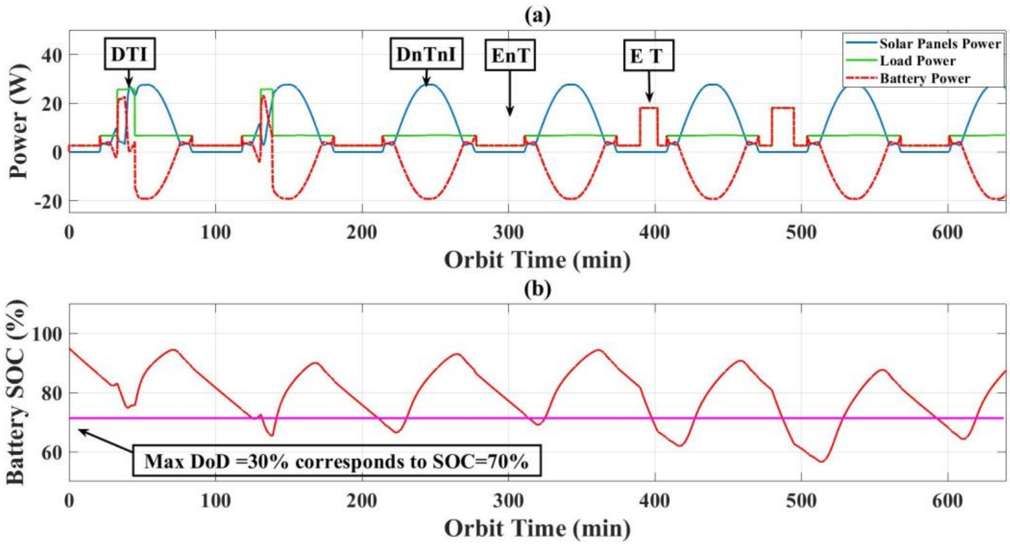

Figure 8. Nadir mode (a) Powers of solar panels, battery and loads with time. (b) Battery SOC with time.

5. Conclusion

The flight orientation of satellite has an important role in shaping the output power profile of the solar panel. The energy balance check should ensure that there is no stress on the battery pack to avoid mission failure and the solar array power supply can cover the load power demand during the daylight time which has been accomplished in sun-tracking flight orientation.

If the satellite has to fly with Nadir orientation, a maximum power point will be sought electronically to minimize the stress on the battery pack during the orbit period. Furthermore, the size of the solar panels may be increased about 30% reflecting on the cost, the weight, and the size. If the size and weight are restricted, a mixed flight orientation should be applied where sun-tracking orientation is taken place to charge battery to 100% SoC and Nadir orientation for taking photos.

References

[1] Sweeting M N 2018 Modern Small Satellites-Changing the Economics of Space *Proceedings of the IEEE* **106** 343–61
[2] Poghosyan A and Golkar A 2017 CubeSat evolution: Analyzing CubeSat capabilities for conducting science missions *Progress in Aerospace Sciences* **88** 59–83
[3] Ali A, Massoud A, Hasna M O, Khattab T, Jabban T and Aref Nema M 2019 Modeling of CubeSat Orientation Scenario and Solar Cells for Internet of Space Provision *2019 9th International Conference on Recent Advances in Space Technologies (RAST)* (IEEE) pp 541–6
[4] Patel M R 2005 *SPACECRAFT POWER SYSTEMS* (Florida: CRC Press)
[5] Dahbi S, Aziz A, Zouggar S, Hafyani M El, Morocco O, Hanafi A, Karim M, Latachi I and Rachidi T 2017 Power budget analysis for a LEO polar orbiting nano-satellite *2017 International Conference on Advanced Technologies for Signal and Image Processing (ATSIP)* (IEEE)
[6] Park H and Cha H 2016 Electrical Design of a Solar Array for LEO Satellites *International Journal of Aeronautical and Space Sciences* **17** 401–8
[7] Gonzalez-Llorente J, Lidtke A A, Hatanaka K, Kawauchi R and Okuyama K-I 2019 Solar Module Integrated Converters as Power Generator in Small Spacecrafts: Design and Verification Approach *Aerospace* **6** 61
[8] J. Richard, Wertz and W. J. Larson 1999 *Space mission analysis and design* (California: Wiley)
[9] Pranay Gupta 2005 *USUSAT II Micro-Satellite Power System Theory, Design, And Testing* (Utah: UTAH STATE UNIVERSITY)