Effect of the bus voltage level on the power system design for microsatellites

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ABSTRACT

Each time there is a need for a satellite mission analysis, engineers need to coordinate together the choice of one technology with regard to the other (i.e. solar cells technology, transistors, microcontrollers, and batteries). The process of selecting one technology to another or a component to another is called 'trade off'. This concept is a hard and situational decision that involves diminishing one quality, of a system or a design in return for gains in other aspects such as weight, performance, speed and so on. The present paper will focus essentially on the use of the 28 V unregulated bus voltage because early satellites, all used, a 14 V unregulated bus voltage. The paper will also show trade-offs made when using the 28 V unregulated bus voltage with different topologies.

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1. INTRODUCTION

The design of the power system reflects the need for autonomous operation independent of all other systems. It therefore should require no intervention from the ground station in the event of an anomaly on the spacecraft. Autonomy is obtained by having redundant systems that automatically switch between each other either when a fault is detected or multiple systems to provide graceful degradation in the event of a failure. Basically, there are four identical battery charge regulators (BCRs), each of them must be capable of transferring power from the solar panels to the battery and the rest of the spacecraft [1]–[6]. A small satellite power system is composed of the following components as shown in Figure 1: i) solar arrays, ii) battery charge regulator (BCR), iii) battery pack, iv) power conditioning module (PCM), and v) power distribution module (PDM). Today, more sophisticated and more capable spacecrafts have eight identical BCRs. Each ½ of a solar panel is connected to a BCR capable of transferring power to the battery and the rest of the spacecraft [1]–[6].

There are also two identical PCMs to ensure that regulated power from the main unregulated bus is available to the subsystems. The PCMs are each equipped with a logic control circuitry capable of detecting faults and dictating the switching over to the redundant PCM [5]. In addition, the hardware isdesigned to adapt to the environmental changes. The most obvious example of this situation is the ability of the BCR to monitor temperature changes of the battery and the solar panels. The BCR predicts the maximum power point MPP of the solar arrays using the array temperature. Another main function of the BCR is its ability to predict the end of charge (EoC) voltage of the battery pack.

The paper presents different topologies related to the design of the power system on board microsatellites. In addition, the article gives an overview on the choice of the 28 V bus as well as the advantages linked to this choice. It is clear that the choice of the 28 V unregulated bus makes it possible to use lower load currents and therefore reduce losses in wiring and power switches [1]–[5].

It is important to add that the following article clearly shows different topologies that exist in the field of power systems design but the choice of a topology with regard to another remains the work of the engineers designing the power system in accordance with the requirements of the satellite mission as well as the constraints of the mission.

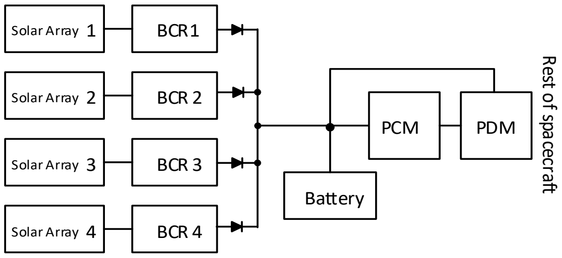

Figure 1. Standard microsatellite power system bloc diagram

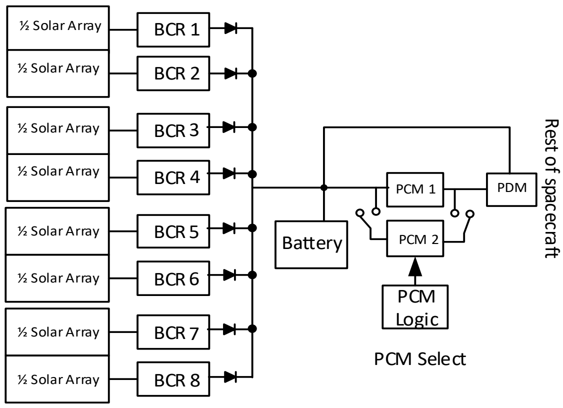

Figure 2. New configuration for a microsatellite power system

2. IMPACTS OF CHANGE FROM 14 V TO 28 V UNREGULATED BUS

Most spacecrafts subsystems in their current design have evolved to be compatible with the unregulated 28 V power bus.By doubling the bus voltage, the loads currents for systems using the bus voltage are halved. Due to path losses being equal to R.I2, a reduction in the load current can significantly reduce losses in the harness and the power switches. In addition, the forward voltage drop of a diode reduces with current. Hence, less power will be dissipated in the blocking diodes.

Also, the range of high reliability DC-DC converters available off the shelf operate within the range of 16 V to 40 V. However, there are very few of these modules available for voltages below 16 V. The use of isolated DC-DC converters for radio frequency (RF) subsystems remove the problem of ground loops being created in the structure [7]–[10].

Due to a reduction in the difference between the solar array voltage and the battery voltage, the efficiency of the BCR system (through which all the power required by the spacecraft passes through) is noticeably increased. Finally, if the power system were to use a 14 V bus voltage, two battery packs must be used to achieve the required capacity. This, in turn, would result in a significant decrease in the power system efficiency due to losses in the diode [1]–[5].

3. REVIEW OF ALTERNATIVE POWER SYSTEM TOPOLOGIES OPTIONS

3.1. The baseline 14 V unregulated bus

Primary power to the satellite is supplied via four body mounted solar panels. The power generated from the solar panels feed into dual BCRs. The BCRs are selected by means of a relay on the input. Logic monitors the BCR operation and switches to the other BCR should a failure occur. The BCR selection can be overridden by a command from the TTC.

The BCRs estimate the maximum power point (MPP) of the solar arrays using a temperature compensation method, and tracks the end of charge (EoC) of the single battery also by using a temperature compensation method that reduces the charge current when the EoC is reached. This temperature compensation, both for the battery and solar arrays, is based on thermistors potted with adhesive in the battery pack and the solar panel substrates. Both the EoC and the MPP tracking can be overridden using the on-board computer (OBC) control [1]–[6].

The circuit in Figure 3 shows a power system configuration based on a 14 V unregulated bus. Figure 3 depicts that a failure of a BCR results in the automatic switch over to the redundant system. Each BCR design can sustain a continuous power of 80 W with sufficient de-rating. In general, for a 60×60 cm solar panel using single junction gallium arsenide (GaAs) solar cells, assuming a fill factor of 80% and an effective area of 80%, the power from a single panel can reach 60 W. This would result in an instantaneous array power of over 80 W when 2 panels are illuminated at an angle of 45° to the sun direction. Therefore, the BCR would need to be redesigned to support this increase of power [11]–[13].

Nowadays for the new enhanced microsatellites, the standard 14 V battery is no longer sufficient because an increase of the depth of discharge (DoD) would certainly result in the decrease of the life cycle of the battery. A solution would be the use of a single high capacity battery (15–20 Ah). However, this would mean either the qualification of new battery cells or the use of space qualified battery packs from aerospace battery companies resulting in an increase in the cost [14]–[20].

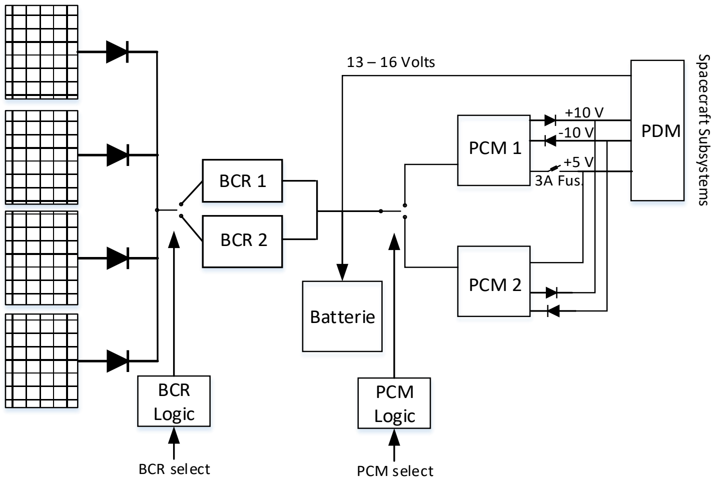

Figure 3. 14 V unregulated power system bus

3.2. The four BCR and a two batteriespower system topology

Primary power to the satellite is supplied via four body mounted solar panels. The solar cell strings on each panel are equally split between the BCR pairs and the batteries. Note that the BCR outputs are isolated from each other using blocking diodes (as shown in Figure 4) [11]–[13]. The BCRs have the role of estimating the MPP of the solar arrays, using thermilinear components YSI 44203. These thermilinear networks consist of resistors and precise thermistors which produce an output voltage linear with the temperature being sensed.

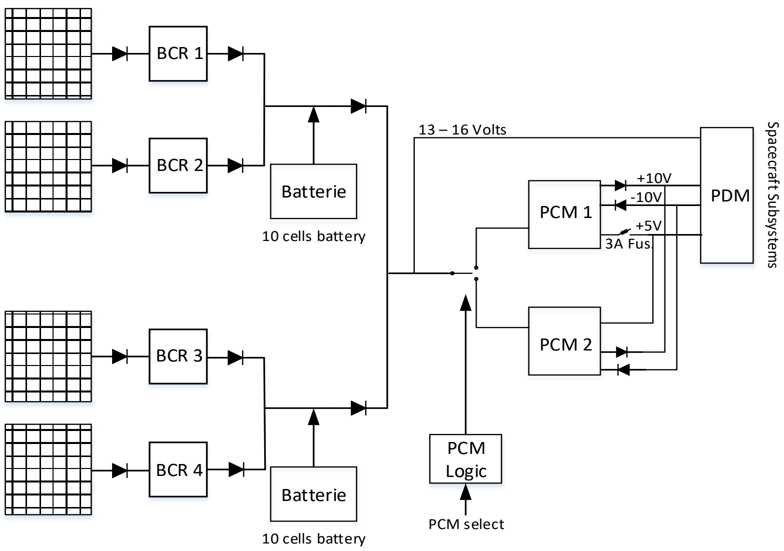

Figure 4. Block diagram of the four BCR and two batteries power system

This temperature compensation, both for the battery and solar arrays, is based on thermistors potted with adhesive in the battery pack and the solar panel substrates. The MPP tracking can be overridden using the on-board computer (OBC) control. The redundancy in the BCR design in this power system topology takes the form of a 'graceful degradation'. Failure of one of the BCRs will result in the loss of up to half of the orbital average power, depending on the attitude of the spacecraft (i.e., worst case having a single panel in sunlight, half of which is connected to the failed BCR, best case no loss of power as a panel with no connection to the failed BCR is used) [11]–[13]. As an advantage, the multiple BCR configuration provides a simple interface to a two batteries power system. Due to splitting the panel strings between each battery, the charge to each battery is continuous during the sunlight periods, irrespective of whether the spacecraft is spinning or 3-axis stabilised.

3.3. One BCR for half panel and a two batteries power system topology

As shown on Figure 5, in total there are eight BCRs, four per battery. The BCR outputs are isolated from each other using blocking diodes [14]–[17]. The power-conditioning module (PCM) being unchanged from the standard microsatellite design. The PDM may use field effect transistor (FET) power switches throughout because they are more power efficient.

There is no need for blocking diodes on the solar array output as there is a dedicated BCR per half solar array. Due to having a dedicated BCR per half solar array, the BCR MPP set point will be more accurate over the panel temperature range, resulting in a higher orbital average power. The loss of a BCR results in a loss of only one half of a panel. Eight BCRs are required for this configuration. This will result in significantly more mass and more assembly and test time for the module.

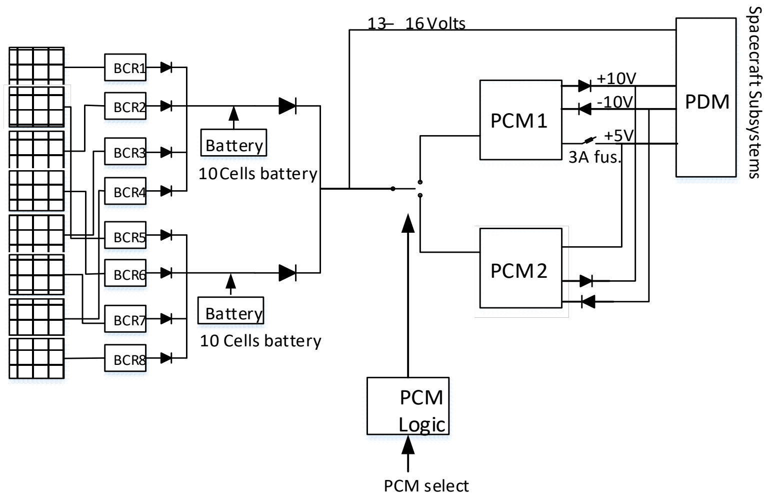

Figure 5. Block diagram of the one BCR for half a panel and two batteries power system

3.4. Dual redundant four BCR with a two batteries power system topology

Primary power to the satellite is supplied through four body mounted solar panels. The solar cell strings, on each panel, are equally split between two BCRs and two batteries (as shown in Figure 6). There are four BCRs in total in a dual redundant configuration to give two per battery. Each redundant pair of BCRs are fed by half of each of the solar arrays. The BCRs estimate MPP of the solar arrays using a temperature compensation circuitry. The EoC of its associated battery is tracked also using a temperature compensation method reducing the charge current when the EoC voltage is reached.

The multiple BCR configuration provides a simple interface to a two-battery power system. Due to splitting the panel strings between each battery pack, the charge to each battery is continuous during the sunlight periods irrespective of whether the spacecraft is spinning or 3-axis stabilised. In addition, a loss of one BCR results in no loss of power due to the redundant BCR configuration [17]–[20].

However, there is a need for blocking diodes on the solar array output. This is due to connecting strings together from different panels in parallel and the connection between the two batteries [17]–[20]. A potential failure mechanism to lose half of the array power is through the failure of a relay in series with the BCR. Although, this failure has never been reported in the literature on previous microsatellites but it still remains as a single failure point.

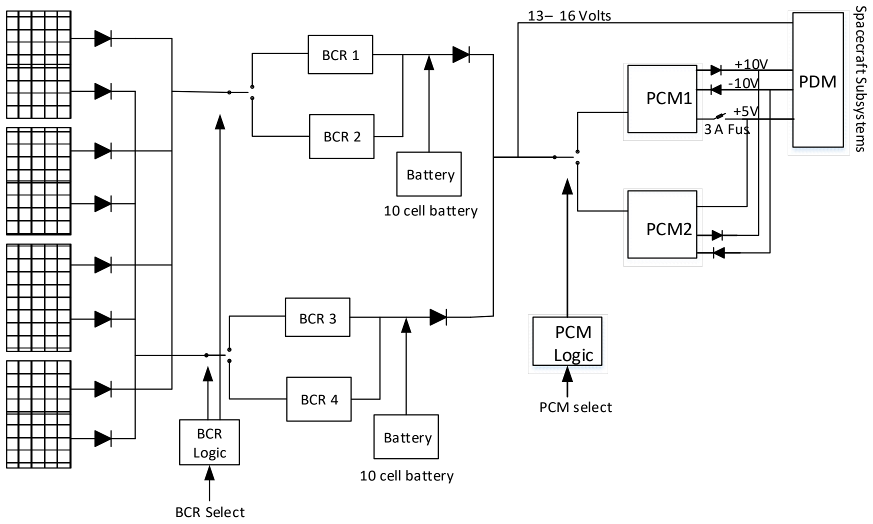

Figure 6. Block diagram of the dual redundant four BCRs with two batteries power system

Figure 7 represents a single battery, multiple BCRs configuration. It also provides a very efficient power transfer from the solar arrays to the battery and the rest of the spacecraft. There is no need for blocking diodes on the solar array output, as there is a dedicated BCR per solar array. Due to having a dedicated BCR per solar array, the BCR MPP set point will be more accurate over the panel temperature range resulting in a higher orbital average power.

The redundancy in the BCR design design in this power system topology takes the form of 'graceful degradation'. Failure of one BCR results in a loss of one panel. There is a need for blocking diodes on BCR. This can result in an instantaneous loss of approximately 1.5W per diode between BCR and PDM maily due to the forward voltage drop in the diodes [14]–[21].

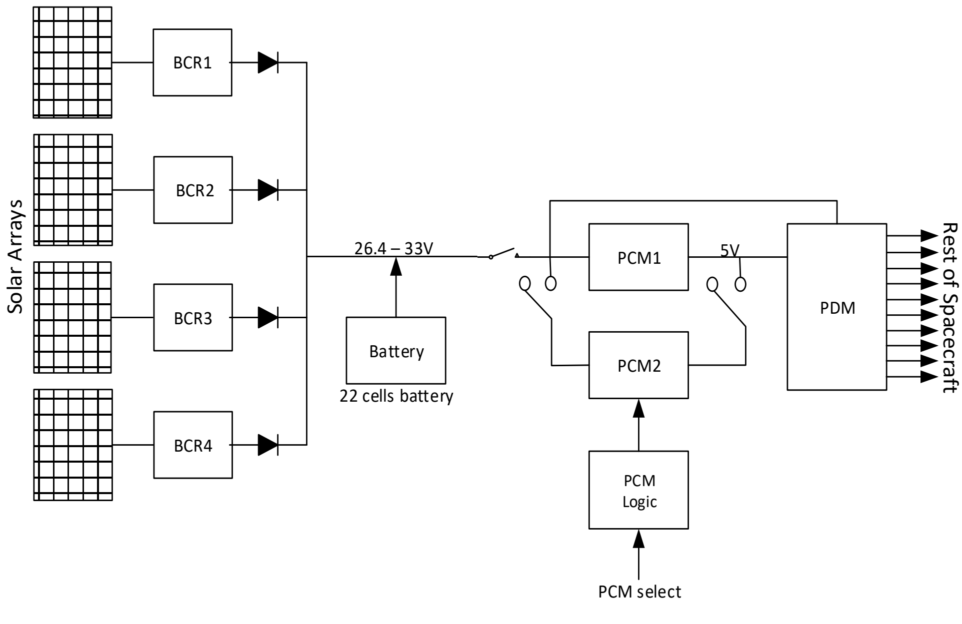

Figure 7. Proposed 28 V unregulated bus power system

Table 1 gives an overview of the 28 V unregulated bus. The table shows clearly the characteristics of the topology used these days on all small satellites. The topology seems more efficient, no blocking diodes are required on solar arrays and most importantly, we cannot rack the MPP more accurately [14]–[21].

Table 1. Comparison between 14 V and 28 V power system buses

	14 V unregulated bus	28 V unregulated bus
Description	Used on early microsatellites	Used on all small satellites (micros and minis)
Harness	High RI^2	Lower RI^2 losses
Blocking diodes	High loss	Lower loss due to reduced current
DC/DC converters availability	None or a few	Yes
Compatibility with COTS spacecraft systems	None or a few	Yes
Battery capacity	6Ah batteries are used. Battery size should be modified to improve DoD	4Ah batteries are used with a reasonable DoD
BCR	Low efficiency due to the high voltage difference between solar array and battery	Higher efficiency due to the reduction of voltage difference between solar array and battery
PCM and PDM	Higher losses on power switches	Lower losses on power switches

4. SUMMARY OF TOPOLOGIES

Table 2 is a look up table for the designer to give a deep insight about the existing different topologies used for satellites power systems [17]–[21]. Table 2 shows a comparison between different topologies. Moreover, the engineers themselves will make the choice of a topology regarding mission requirements and constraints [14]–[21].

Table 2. Summary of different topologies

	Baseline 14 V power system	28 V power system	4 BCR with two battery packs	8 BCRs with two battery packs	Dual redundant 4 BCRs with two battery packs
Battery cells	11	22	22 (2*11)	22 (2*11)	22 (2*11)
Bus voltage (V)	13 – 16	26.4 – 33	13 – 16	13 – 16	13 – 16
Battery capacity (Ah)	6	4	4	4	4
Battery capacity (Wh)	86	112	112	112	112
Battery redundancy	No	No	Yes	Yes	Yes
Depth of discharge (%)	25	19.2	19.2	19.2	19.2
BCR relay	Yes	No	No	No	Yes
Number of BCRs	2	4	4	8	4
One (01) BCR loss	Fully operational	Loss of one BCR	Loss of up to ½ OAP	Loss of up to ½ OAP	Fully operational
One (01) array loss	-	Loss of one panel	Loss of up to ½ orbit average power	Loss of up to ½ orbit average power	Loss of up to ½ orbit average power
Max. panel power (W)	60	80	160	160	96
Forward diode loss (W)	~ 3	~ 1.5	~ 6	< 6	< 6
MPP temperature compensation	Average of 4 panels	Exact	Average of 4 panels	Exact	Average of 4 panels

5. RESULTS AND DISCUSSION

This paper focuses on the effect of the bus voltage level on the power system design for microsatellites. In addition, the paper compares the uses of a 14 V unregulated bus with the use of a 28 V unregulated bus. Different topologies were investigated and the final choice belongs to the power system engineer taking into consideration the spacecraft's mission requirements and mission constraints.

Table 2 gives a summary of the different topologies being used in the field of satellite power system design. One has to be careful when deciding to choose a configuration rather than the other. Cost and weight of the whole system are important parameters to be looked at when making trade-offs.

6. CONCLUSION

The 14 V unregulated voltage bus has proved to be less efficient than the 28 V voltage bus. The higher the voltage you use the lower the current requirements and therefore the losses. The 28 V power system is becoming more and more an 'industry standard' for small satellites bus voltages. The 28 V system has also no battery redundancy.

With the increased number of battery cells from 10 to 22, problems associated with cells capacity mismatch can be more pronounced resulting in some cells eventually reaching zero capacity. This problem is mitigated by closely matching the capacity and voltage characteristics of the cells in the battery, preventing cell voltage reversal as the cell reaches full discharge.

REFERENCES

[1] J. Z. Slowikowski and M. J. Blewett, "Power supplies, conditioning and distribution on UOSAT," *Radio Electron. Eng.*, vol. 52,

no. 819, pp. 425–430, 1982.
[2] M. N. Sweeting, *UoSAT Spacecraft data Booklet*. Guildford, U.K.: Dept. of Electronic and Electrical Engineering University of Surrey, 1987.
[3] H. S. Wong and M. J. Blewett, "The UoSAT-2 spacecraft power system," *J. Inst. Electron. Radio Eng.*, vol. 57, no. 5S, p. S116, 1987, doi: 10.1049/jiere.1987.0089.
[4] D. O'Sullivan, "Satellite power system topologies," *ESA J.*, vol. 13, no. 2, pp. 77–88, 1989.
[5] V. M. Van der Zel, "Autonomous power system for low earth orbit microsatellite," University of Surrey, Guildford, United Kingdom, 1990.
[6] W. J. Larson and J. R. Wertz, *Space mission analysis and design*, Second Edi. Space Technology Liibrary.
[7] P. Fortescue and J. Stark, *Spacecraft systems engineering*, Second Edi. John Wiley and Sons, 1995.
[8] C. Brown, *Spacecraft Mission Design*, Second Edi. American Institute of Aeronautics and Astronautics, 1998.
[9] M. D'Errico, M. Pastena, and G. P. Dipartimento, "Design and Development of a Power Conditioning Unit for an Earth Observation Microsatellite," 2001.
[10] C. D. Brown, *Elements of Spacecraft Design*. American Institute of Aeronautics and Astronautics, 2002.
[11] N. Mohan, T. M. Undeland, and W. P. Robbins, *Power Electronics: Converters, Applications, and Design*, 3rd editio. 2002.
[12] U. Orlu, G. Yuksel, L. Gomes, and K. Hall, "BILSAT-1: power system sizing and design," in *International Conference on Recent Advances in Space Technologies, 2003. RAST '03. Proceedings of*, 2003, pp. 159–164, doi: 10.1109/RAST.2003.1303900.
[13] M. R. Patel, *Spacecraft Power Systems*. Taylor & Francis, 2004.
[14] R. Birkeland, E. Blom, and E. Narverud, "Design of a small student satellite," NTNU: Norwegian University of Science and Technology, 2006.
[15] K. Shenai, "Spacecraft Power Systems Design to Minimize Electro Magnetic Interference (EMI) Effects," 2007.
[16] C. S. Clark, "An advanced electrical power system for cubesats," 2010.
[17] M. Bensaada *et al.*, "Sliding mode controller for buck DC-DC Converter," in *International Multi-Conference on Systems, Sygnals & Devices*, Mar. 2012, pp. 1–6, doi: 10.1109/SSD.2012.6197933.
[18] R. Anusha and P. Usha, "Fully Regulated 28V Bus Using Bidirectional Battery Charge/Discharge Regulator for Satellite Applications," *Int. J. Adv. Res. Electr. Electron. Instrum. Eng.*, vol. 3, no. 8, pp. 10954–10961, 2014.
[19] M. Bekhti, M. Bensaada, M. Beldjehem, F. Metehri, A. H. Dida, and S. D. Krachai, "Small satellite power distribution module adopting overcurrent protection," in *International Conference on Industrial Application in Engineering Science*, p. 2019.
[20] S. Della Krachai, A. B. Stambouli, M. Della Krachai, and M. Bekhti, "Experimental investigation of artificial intelligence applied in MPPT techniques," *Int. J. Power Electron. Drive Syst.*, vol. 10, no. 4, p. 2138, Dec. 2019, doi: 10.11591/ijpeds.v10.i4.pp2138-2147.
[21] M. Bekhti and M. Bensaada, "Solar Arrays and Battery Power Sources Conceptual Design for Low Earth Orbit Microsatellites," *Eng. Appl. Sci. Spec. Issue Res. Emerg. Technol. Des. Manuf.*, vol. 5, no. 3, pp. 66–70, 2020, doi: 10.11648/j.eas.20200503.13.

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