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A Comprehensive Review on Small Satellite Microgrids

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***Abstract* — The Small Satellite (SmallSat) industry has recorded incredible growth recently. Within this class, among Mini-, Micro-, and Nanosatellites, the Cube Satellite (CubeSat) is primed for an explosion of growth. These satellites are fascinating for remote sensing, earth observation, and scientific applications. Remarkable attention from the space operators makes it valuable because of its low cost, cubic shape, less manufacturing time, lightweight, and modular structure. Among the various subsystems comprising the SmallSat, the Electrical Power System (EPS) is the most crucial one because unreliable power supply to the rest is most of the time detrimental to the mission. The EPS is formed by electrical sources, storage units, and loads, all interconnected via different power converters, the operation of which must be closely orchestrated to accomplish efficient use of photovoltaic power, optimal battery management, and resilient power delivery. At the same time, the EPS design must address a series of challenges such as size restrictions, high power density, harsh space environments (e.g., atomic oxygen, radiation, and extreme temperatures) which significantly impact the EPS electrical and electronic equipment. In terms of power systems, a SmallSat EPS can be considered a space microgrid owing coordination and control of distributed generation (DG), storage and loads in a small-scale electrical network. From this point of view, this paper reviews and explores SmallSat microgrid's research developments, energy transfer and architectures, converter topologies, latest technologies, main challenges, and some potential solutions which will enable building a more robust, resilient, and efficient EPS. The research gaps and future developments are underlined before the paper is concluded.**

***Index Terms*—battery technologies, CubeSat, converter topologies, EPS, NanoSat, microgrid, PV technologies, SmallSat, switches.**

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I. INTRODUCTION

The Small Satellite (SmallSat), i.e., mini, micro, nano, pico, and femto satellites industry in recent years has geared up and is in the state of continuous growth. Because of the recent progress and developments in SmallSat subsystem technologies i.e. integrated circuits (ICs), digital signal processing (DSP), microelectromechanical systems (MEMS), additive built-up, accessibility of affordable and innovative Commercial-Off-The-Shelf (COTS) technologies, smaller mass, volume, least development time, and cheaper cost have been achieved [1], [2]. Future roadmaps for NanoSat applications include the creation of constellations of such satellites to form space-based telecommunication networking exploited for mobile communications and global internet coverage [3]-[6]. Therefore, an ignition to the NanoSat market is expected in the upcoming years. The increased growth of interest for the NanoSats, has also been boosted by the incremental rise of a particular kind of NanoSats, the CubeSats. CubeSat has gained the attention of diverse vendors and consumers like governments, educational scientists, and commercial organizations, since 2003 when the first satellite was launched [7]. The CubeSat took its name due to the cubic shape of its main building block, which is a unit (U) weighing 1.33 Kg, one liter of volume, and 10 x 10 x 10 cm dimensions [8]. CubeSats are extendable for higher payload demands, via the addition of multiple cubic units, as shown in Table I. It is noteworthy that, though Nanosats can weigh from 1 kg up to 10 kg of mass, as shown in Table II, CubeSats, though classified within the NanoSat class, are not restricted within this range (e.g. the heaviest reported CubeSat is comprised of 27U and weights 40kg) [3]. To address this high interest, a specific standard for CubeSats a project was started in 1999 by California Polytechnic State University and Stanford University in the USA as a collaborative effort [9]. Though CubeSats are mainly considered an educational tool for the students [10], their applications are not limited to this, since CubeSats have been

TABLE I
CUBESAT CLASSIFICATION ACCORDING TO MASS AND VOLUME [11].

CubeSat Specifications	Max Mass (kg)	Max Volume (cm)
1U	1.33	10×10×10
2U	2.66	10×10×20
3U	4.00	10×10×30
6U	8.00	10×20×30
12U	16.00	20×20×30

TABLE II
SPACECRAFT CLASSIFICATION ACCORDING TO MASS AND MANUFACTURING COST [12].

Type	Mass (kg)	Manufacturing Cost (US $)
Large-satellite	>1000	0.1-2 B
Medium-satellite	500 - 1000	50-100 M
Mini-satellite	100 - 500	10-50 M
Micro-satellite	10 - 100	02-10 M
Nano-satellite	1 - 10	0.2-02 M
Pico-satellite	0.1 - 1	20-200 K
Femto-satellite	<0.1	0.1-20 K

deployed to serve several Earth observations, astronomical, and communications applications where short revisit times or even continuous monitoring is required [13], [14].

Nanosats are mostly designed to be placed in Low Earth Orbit (LEO) for several types of missions [15]. These satellites are required to perform several complex tasks in education, scientific research, space exploration, Earth observation, space weather, high-resolution imagery, ship tracking, airplane tracking, etc. [16]-[18], while maintaining the cost low. However, the harsh radiation environment, size limitations, high power density requirement, and low-cost components (especially COTS) make their design challenging.

The Electrical Power System (EPS) is one of the critical components in all satellites generally. This claim is backed up by the reliability analysis performed in [19], and [20], which reveals that in earlier stages (in prior 4-weeks of launch) the EPS leads to the second most fatal failures, while after the preliminary 4-weeks period, it leads to the most fatal failures. While EPS failures result 25% of all spacecraft failures. The SmallSat EPS on the other hand is a small-scale remote electrical network which operate in islanded mode and is regarded as a space microgrid. Microgrids are assembly of distributed generation (DG), autonomous distribution networks, storage units, and loads that work in a controlled and coordinated way [21] [22]. However, the small satellites are mostly custom-built with commercial-off-the-shelf components. Where, the EPS may be the most scalable of the subsystems yet demand high reliability, efficiency, and resiliency to sustain and supply full power during the mission. Compared to the terrestrial, space microgrid are compact, redundant and demand high design and testing requirements. Therefore, significant research is placed on the robust design of the EPS. Though the SmallSat research fields have attracted a significant portion of the already published research, it is noteworthy that only a handful of review papers exist regarding a satellite's EPS in general. This fact is contradicting the criticality and importance of the EPS. It should be noted however that some general research review publications handle the topic of SmallSats subsystems holistically. Kopacz *et al.* [23] presented an overview of SmallSats, where, a comprehensive review of the latest missions is presented, especially focusing on NanoSat launch history, classification, origin, and utilization. Davoli *et al.* [24] surveyed different aspects of CubeSats, including mission goals, structure, and hardware components, focusing mostly on network and communication aspects. Sweeting *et al.* [25] have presented a broad overview of SmallSats evaluation, applications, capabilities, and future technology trends.

In terms of EPS, the available recent research is underlined in the following works. Timothy *et al.* [26] have presented a modular EPS for SmallSats, discussed operational aspects, control issues, and encouraging future work regarding the EPS design. Johnston-Lemke *et al.* [27] have given the concept of a modular, scalable, and highly efficient EPS, which is deployable to the satellites for the power demand spans of 1W to 1kW requirements. Khan *et al.* [28] have proposed a comprehensive design and control methodology for SmallSats EPS where, a method is specified for sizing key elements of the EPS i.e., photovoltaic (PV) array and battery. The proposed design takes into account the irradiance forecasting, the PV array geometry, the PV cell characteristics, round trip efficiency, and state-of-charge (SOC) of the battery. Edpuganti *et al.* [29] have presented a review article on conventional and emerging CubeSats EPS architectures. The EPS architectures are identified and grouped into 17 categories based on the PV panels interfacing, conversion stages, and DC-bus regulation. In addition, based on a qualitative comparison the merits and demerits of the different EPS architectures are illustrated. On the other hand, Yost *et al.* [30] have presented a report on state-of-the-art SmallSat EPS technologies that offer valuable insight into the major EPS components available from different commercial manufacturers. To address the EPS subsystem holistically, Bintoudi *et al.* [31], has proposed its characterization as a microgrid. More specifically, an introduction to the space microgrids has been addressed by Lashab *et al.* [32], where the authors presented an overview on EPS for satellite-based microgrids, the energy generation, Energy Storage System (ESS), and an insight on protection schemes, followed by the sizing guidelines. However, this work is limited and does not cover EPS energy transfer systems, EPS architectures, recent technological advancements, design, and operation challenges of SmallSat EPS. Therefore, even though there are individual pieces of work regarding various topics of the EPS of a SmallSat (architectures, topology, converters, controllers, etc.), to the best knowledge of the authors, no review papers are offering a system level and a comprehensive literature review regarding the latest developments on the most critical aspects of SmallSat EPS.

To that end, this paper treats the SmallSat EPS as a whole, from the perspective of an isolated microgrid, to provide a comprehensive review of several different perspectives of the EPS. More specifically, this paper reviews the EPS topologies, the state-of-the-art regarding converter topologies, PV array, and electrochemical battery technologies used in SmallSats. The inclusive structure of the SmallSat which has been reviewed is described in Fig. 1. Moreover, the most unique feature of this review paper is the detailed analysis of realistic technologies and converter topologies, which are applied in the latest and modern actual SmallSat projects. This paper is organized as follows. Section II is an overview of the SmallSat EPS structure, architectures, and important energy transfer methods. Regarding power generation, distribution, and

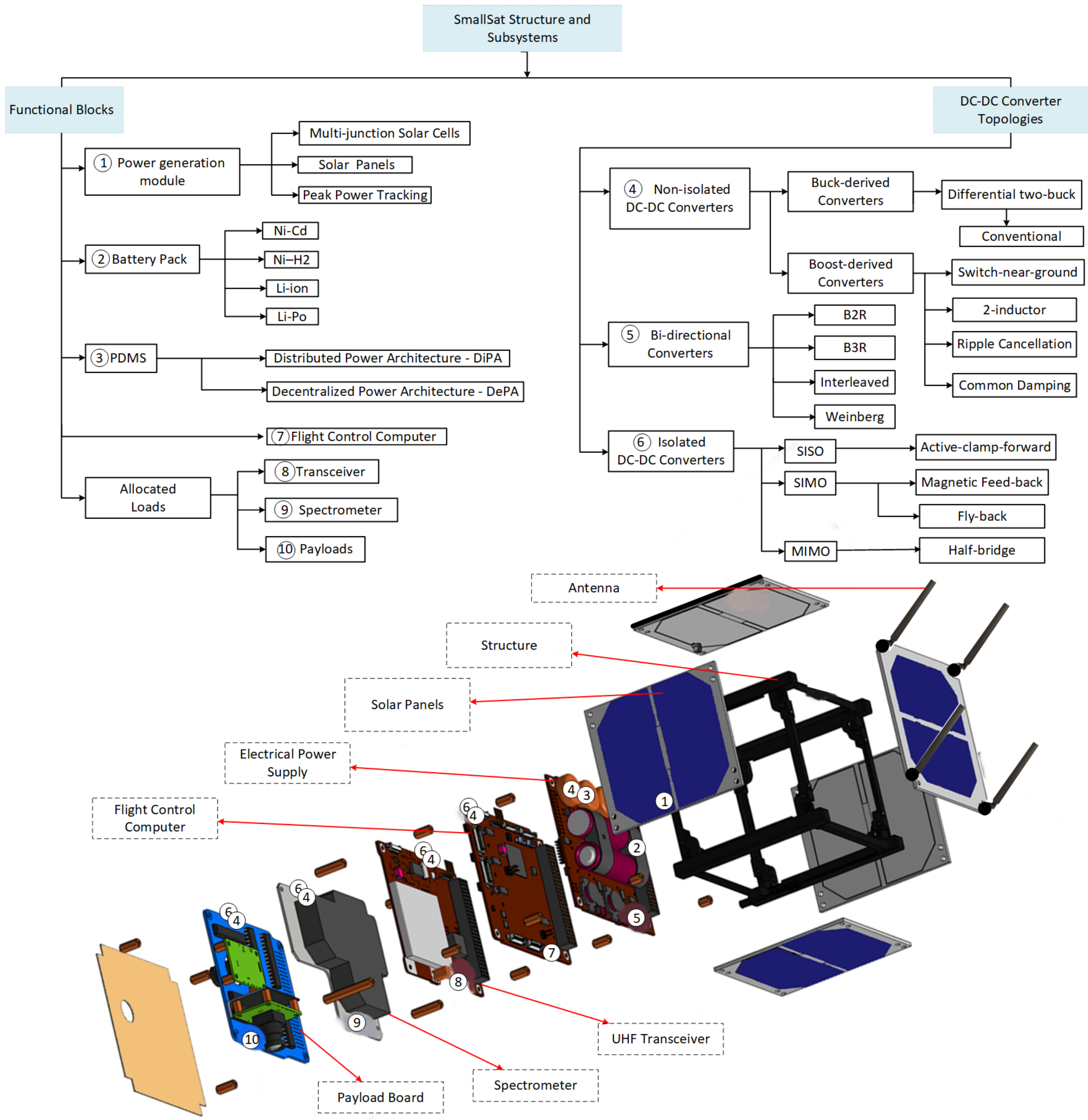


Fig. 1. An inclusive structure of the SmallSat, important subsystem including energy generation module, storage module and converter topologies.

management the latest non-isolated and isolated converter architectures, switches, and suitable protocols are described in Section III. Section IV includes an overview of the latest developments in space-qualified solar cells, panels along with the PV architecture for SamallSat applications. The state-of-the-art power storage and different battery technologies available for SmallSats are discussed, followed by the battery charge regulation in Section V. Section VI is a discussion of the challenges and some potential solutions in EPS design and operations. The paper is concluded with a brief discussion regarding future directions in Section VII.

II. Electrical Power System and Architectures

The EPS is essentially the lifeline of the other subsystems in any satellite which incorporates the power generation, energy storage, power distribution, and management system (PDMS), including the loads. The EPS comprises approximately one-third of the total satellite mass [30]. The prime function of the

EPS is to maintain continuous power supply to the satellite bus during its mission life, including adequate power even during the eclipse when no power is generated from solar panels and protection under fault conditions. The EPS manages the power input from the solar panels, charges/discharge onboard batteries, and distributes the electrical power to the subsystem elements of the satellite at their required voltage levels. Additionally, the monitoring, operation status, and health of the EPS subcomponents are communicated to the onboard computer [33], [34]. Indicatively, the power budgets for some SmallSats subsystems are given in Table III. As mentioned, the spacecraft EPS can be regarded as a microgrid, since it is the composition of modular dc-dc micro-converters, distributed energy resources, and several loads [31], [32], [35]. The block diagram of a comprehensive EPS with the two basic architectures is shown in Fig. 2 (a) and (b).

Depending on the exploitation mode of the solar power, two well-known topologies are derived: the Direct Energy Transfer (DET) and the Maximum Power Point Tracking (MPPT) architectures, shown in Fig. 2 (a) and (b) respectively. The DET architecture operates at a fixed voltage point on the I-V characteristics, distributes the necessary power to the loads in a regulated or unregulated form, and shunts unnecessary power. This conversion method is simple yet. DET architectures are mostly encountered in applications with power budgets less than 100W [36]. However, the PV I-V curve is a direct function of the solar cell temperature, irradiation, and degradation therefore, DET architectures are not exploiting the full potential of the harvested solar power by the solar arrays. To overcome this limitation, the MPPT-based architecture is proposed [37], which essentially forces the solar array output voltage to be always set at the value which results in the maximum power transfer from the array to the aggregated load, regardless of the solar cell temperature and degradation degree. MPPT architectures demand at a minimum one dedicated dc-dc converter in series with the PV array to drive the PV cell operating voltage and it demands at a minimum 4-7% of the solar array nominal output power to operate [36].

The main DC power bus can either be regulated or unregulated to dispatch power to the loads. In the case of an unregulated topology, the main bus follows the battery voltage, while regulated bus architectures demand the usage of dc-dc converters to achieve full regulation of the main bus voltage. In Fig. 2 (a) and (b), both DET and MPPT topologies are shown with unregulated bus voltages which can be regulated with the addition of dc-dc regulators illustrated by dashed lines. A comparison of the peak power tracking EPS architectures in [38], demonstrate that EPS architecture with series connected MPPT converters and unregulated dc-bus has greater reliability, lower component count, good battery life, and the highest efficiency for all operating modes.

In terms of EPS sizing, the SmallSat power budget is defined according to the worst-case scenario which is the simultaneous operation of the loads and not according to the installed load capacity. Therefore, task scheduling strategies in SmallSats can be strongly related to energy harvesting from PV, optimal power management, efficiency, and quality-of-service

TABLE III
POWER CONSUMPTION OF DIFFERENT SUBSYSTEMS AND PAYLOADS FOR SOME NANOSATS.

Nano Satellites	Reference	Subsystem Power Sizing (W) ADCS	OBDH	COM	CAM	Total Power
NanoSat	[39]	0.375	0.2	1.9	0.24	2.71
ISRASAT1	[40]	1.22	1.7	1.87	0.5	4.66
SuryaSat	[41]	3	0.04	0.48	0.50	4.02
ESTCube1	[42]	1.29	0.3	6.2	4.48	12.2
NanoSat	[43]	0.375	0.412	3.13	0.5	4.17
CubeSat	[44]	1.23	0.38	1.26	0.3	3.17
3Cat-3/MOTS	[45]	2.5	3.2	15.1	10.5	31.0

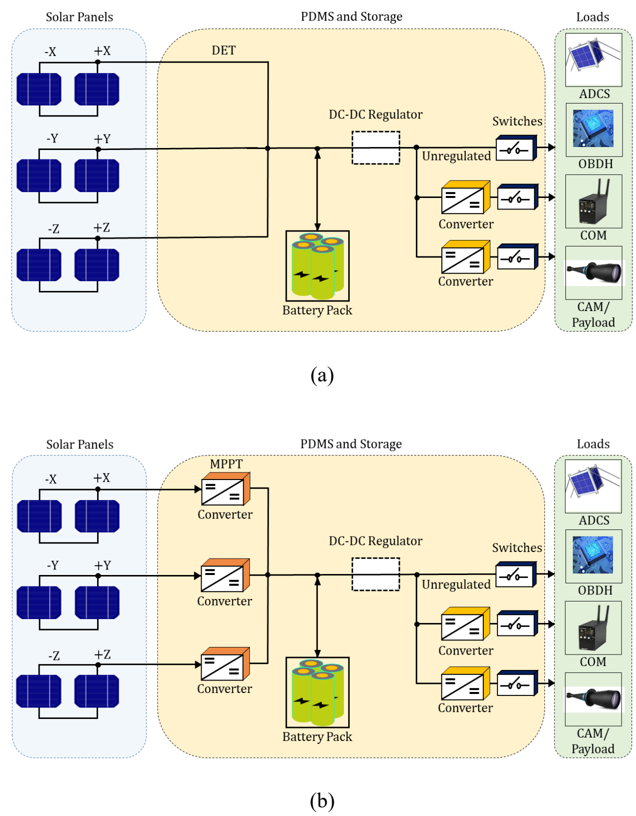


Fig. 2. The two basic architectures of EPS (a) DET solar power conversion and (b) MPPT solar power conversion topologies with unregulated buses.

assurance [46]. Moreover, tasks are formulated based on the number of duties, task priority, the maximum and minimum duration of the task, and execution time [47].

III. Power Distribution and Management

The Power Distribution and Management Subsystem (PDMS) of the EPS distributes the flow of the power pertaining dc-dc converters at different distribution levels to regulate, control the generated power, and supply power to various analog and digital loads [48], [49]. The PDMS component blocks are bus regulators, battery charge/discharge dc-dc converters, various switches for EPS branches, loads and, batteries, the respective converter digital controllers, sensors circuitry, and point-of-load (POL) dc-dc converters. However, the embedded designers face challenges in design that operate in space is the limited pool of components that were designed to operate in this environment. Resulting, EPS failure at the early stage of the deployment or during the mission. To withstand the effects of radiation semiconductors that operate in space need to be 'hardened'. The available semiconductors are not exactly leading edge or 'state-of-the-art' and there is not a huge portfolio of components to choose from. The space version components tend to be very expensive, on the other hand. Therefore, the cost has been limited by using properly selected COTS devices, while reliability has been achieved by design diversity and through redundancy.

A. DC-DC Converters

SmallSat dc-dc converters are core building blocks of the EPS that ensure reliable and efficient power delivery. The converter design must be able to maintain constant output voltage regardless of input disturbances, while power quality must be high so that sensitive onboard equipment operates seamlessly. At the converter power stage, the switching devices are mostly high-efficiency field-effect transistors (FETs) due to small gate charge, on-state resistance, and less complicated drive circuit, which can more efficiently switch at higher frequencies [50]-[52]. Most of the typical voltages required for the CubeSat onboard equipment are in the range of 3 to 6 volts. The EPS is highly segmented, each segment of which is usually supplied through a dedicated buck or boost dc-dc converter and necessary switching gear [53]. These subsections are well fragmented to overcome the noises and ripple creeping since the nature of the loads can be RF, analog, or digital [54]. Depending on system architecture, isolation and voltage regulation are required at different stages of power conversion. Moreover, for interfacing low voltage renewable sources, galvanically isolated dc-dc converters are the best suitable solutions [55]. Regarding distribution in modern satellites, two basic design approaches prevail: distributed power architecture (DiPA) and decentralized power architecture (DePA) [56]. The DiPA approach is very popular for ensuring dynamic, efficient, and reliable system performance. This approach uses an isolated DC-DC converter and multiple POL converters, as shown in Fig. 3. An isolated converter called an intermediate bus converter (IBC), supplies an intermediate fully regulated, semi-regulated, or simply unregulated voltage to the subsequent non-isolated voltage regulators, namely the POL converters. The IBC is usually physically distanced from the main digital devices board, due to mechanical considerations and cooling. However, every POL converter is mounted on a board near the corresponding loads to minimize the parasitic impedances. Overcoming the disadvantages of the greater number of conversion stages and losses in DiPA, the DePA has been proposed [56]. As shown in Fig. 4 (a) and (b), there are two main variations of DePA: in the first one (Fig. 4a), a single isolated converter with multiple outputs followed by several load switches, is responsible for the power distribution to the

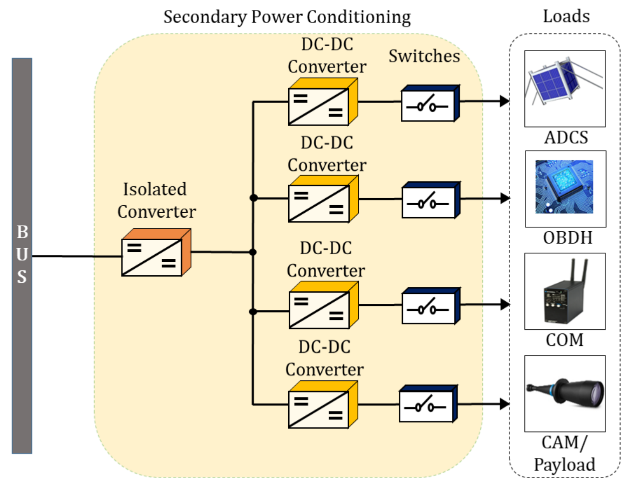
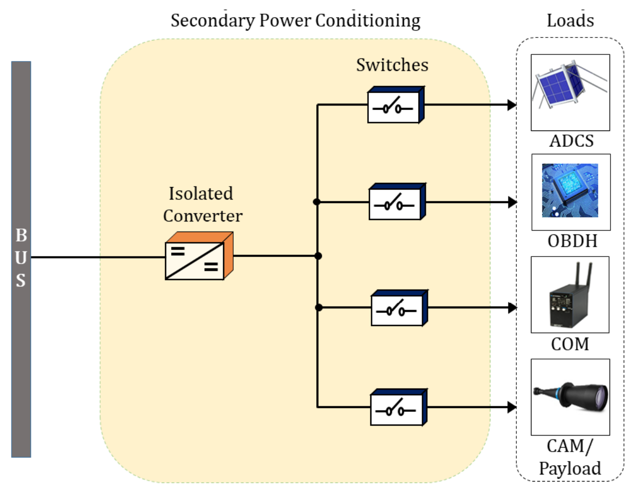
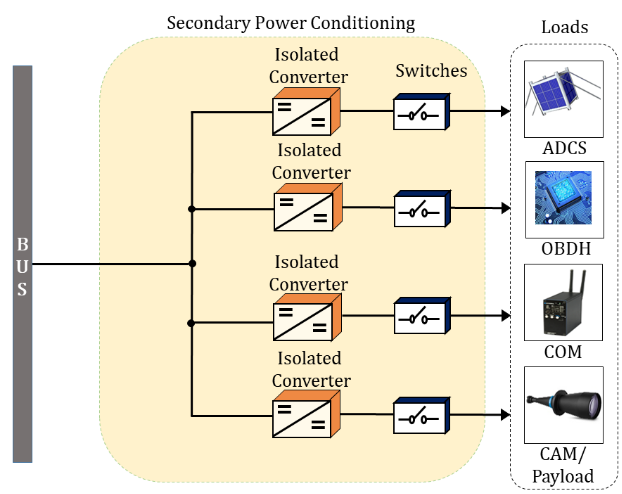


Fig. 3. The Distributed Power Architecture - DiPA.



(a)



(b)

Fig. 4. The Decentralized Power Architecture - DePA, (a) single input with multiple outputs, (b) multiple isolated converters.

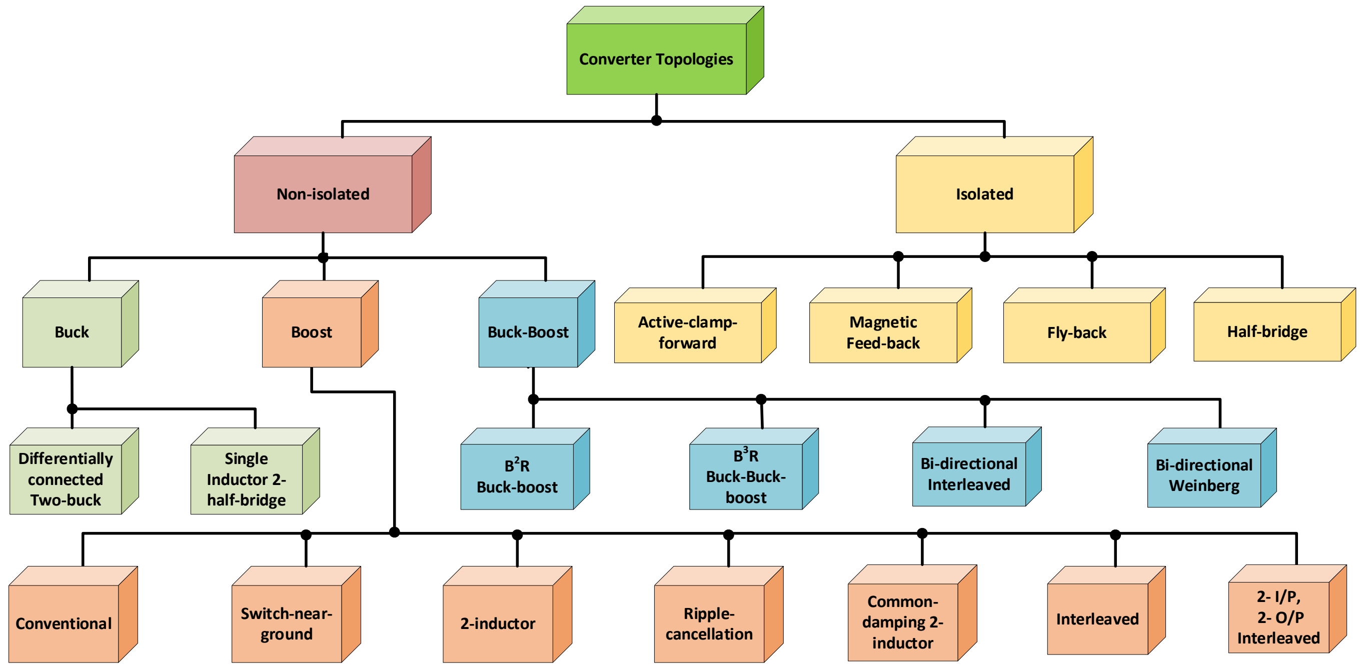


Fig. 5. State-of-the-art converter topologies reviewed and categories for space mission applications in this paper.

loads, providing a fully regulated voltage level 1V, 1.5V, 2.5V, 3.3V, 5V, etc., while in the second one (Fig. 4b) several independent isolated converters are operated for the provision of different outputs of fully regulated voltages required for the operation of the satellite loads. This architecture is advantageous due to its lower losses and lighter weight, compared to DiPA where power is lost in each conversion step.

The high reliability requirements in the design of EPS for space applications is hard to quantify relatively applying COTS components. Space converters deal with many constraints suck as, electromagnetic interface (EMI) compatibility, vacuum environment, radiation affects, shock and vibrations. Additionally, the reduced mass, high efficiency, and provision of high-power quality are strictly required. These stringent requirements have narrowed down the choices to conventional converter topologies due to their inherent simplicity and the minimum number of components [57], [58]. To overcome these barriers selection of radiation hardened components, their mechanical dimensions and thermal analysis to withstand in vacuum environment must be considered. Selection of the converter topologies to withstand and support radiation effects is significantly considered.

Converters can be classified into different categories based on the configurations and types of components used. Various factors are considered to form these categories, such as single-stage and multi-stage power processing, inductor or transformer-based configurations, and types of switches. The converter clusters are shown in Fig. 5.

B. Overview of the Non-isolated DC-DC Converters.

This section focuses on the performance analysis of different non-isolated dc- converters dc for space applications. The main benefits of the converters covered in this subsection are design

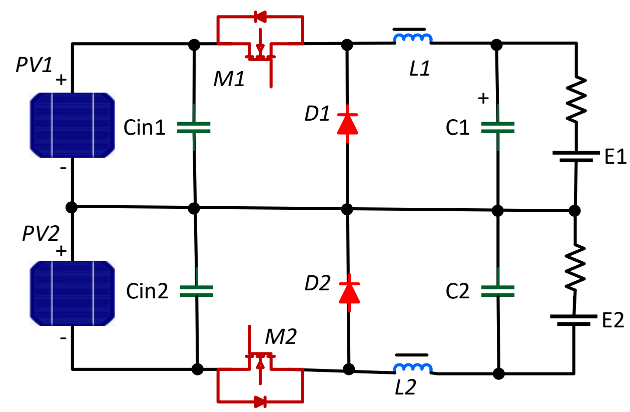


Fig. 6. Differentially connected two-buck converters topology.

simplicity and therefore greater reliability, and fewer number of components, and consequently dry mass [59]. Nevertheless, the extreme duty cycle operation of conventional converters leads to high voltage and current stress, hence, it suffers from reduced controllability and extreme losses [60]. These converters are usually used in Nano and CubeSat applications at different stages for conversion and regulations of power.

1) Buck-derived converter topologies: A new converter for common-mode noise reduction is introduced in [61] as illustrated in Fig. 6. The common-mode noise reduction architecture interfaces the PV panels to the power bus of the satellite with minimum leakage currents avoiding unpredicted ground bouncing conditions. While ground bounce in high density digital circuits is a delay in reaching ground in a transistor after a signal transition. The ground bounce can produce transients with amplitudes of volts; most often changing magnetic flux is the cause. The proposed converter

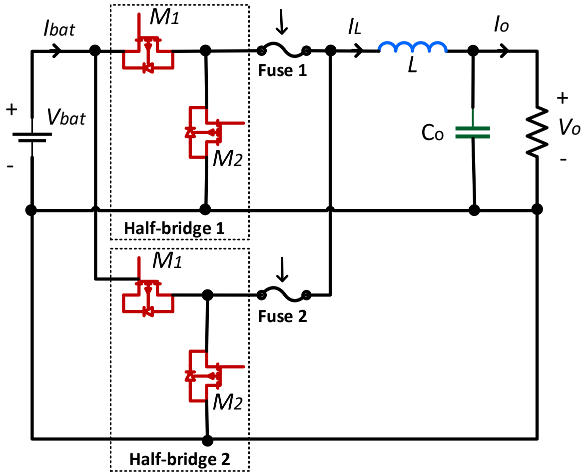
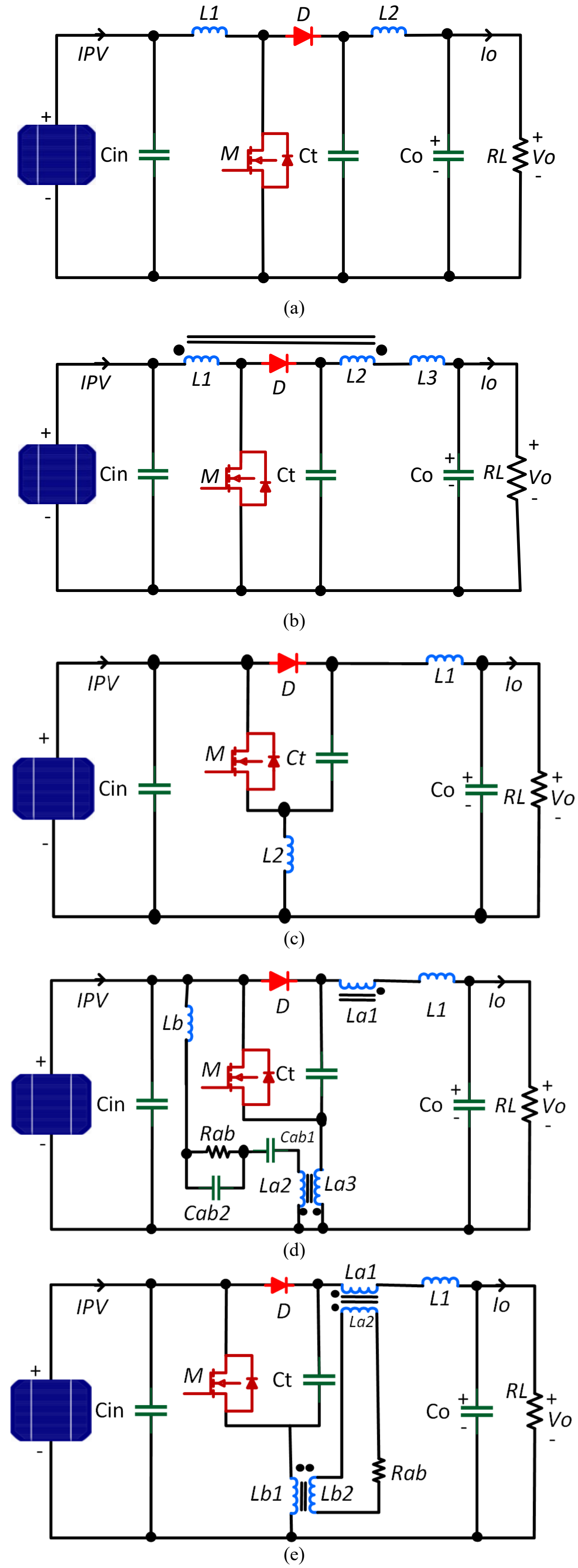


Fig.7. The load-side redundant buck converters topology.

connects two buck-derived converters differentially ensuring zero common mode current in the ground which makes the converter safe at slight ground bouncing, improves the converter electromagnetic compatibility (EMC), and negates the parasitic effects at normal operation. In this converter, two inductors are cumulatively coupled at the output stage for reduction of phase current ripple while increasing the efficiency by reducing the component count of the EPS. The coupled inductors offer a moderate transient handling capability and more power density [62]. For the CubeSat application, the load-side redundant buck converter topology has been proposed in [63], as shown in Fig. 7. This converter consists of two independent half-bridge switching modules and a common inductor. The half-bridge module and inductor are connected through a fuse for over-current protection. In the fault conditions, the redundant module activates based on the designed fault diagnostic system. However, the feasibility of redundant components in smaller satellites needs some special efforts due to the restricted satellite weight and volume.

2) Boost-derived converter topologies: The low output voltage of the PV with a parallel-connected structure makes it very high demanding to achieve high efficiency, low cost, and high-step-up dc-dc conversion [64]. In satellite applications where the bus voltage is greater than the solar array one, for the implementation of MPPT a boost power converter is applied. Garcia *et al.* [65] have compared and analyzed some boost-based topologies for space applications that use a 100V voltage bus, as shown in Fig. 8 (a) to (f). These boost dc-dc converters are: conventional, boost converter with a switch near-ground topology, two inductor boost converter, boost converter with ripple cancellation, common damping two-inductor, and interleaved boost. Depending on the satellite's requirements in terms of operational voltage range, the capability of power handling, operational voltage, conduction emission, and solar array output impedance the topologies are designed and optimized. The analysis has been carried out from the viewpoint of the control loop bandwidth, mass, and power losses. From these topologies, the conventional boost topology requires fewer components and therefore it is characterized by high



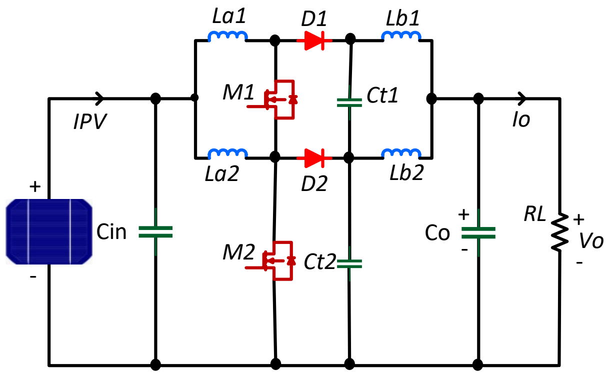


Fig. 8. The boost dc-dc converter topologies, (a) conventional, (b) boost converter with a switch near ground, (c) two inductor boost converter, (d) boost converter with ripple cancellation, (e) common damping two-inductor, and (f) interleaved.

TABLE IV
COMPARISON OF DIFFERENT BOOST CONVERTER TOPOLOGIES

Converter topology	Reliability	Bandwidth (Hz)	Mass (gr)
(a)	High	300	207.8
(b)	High	300	287.1
(c)	Medium	300	336.1
(d)	High	400	161.6
(e)	High	30 e^3	336.1
(f)	High	600	161.3
Fig.11	High	100e^3	Unkn.

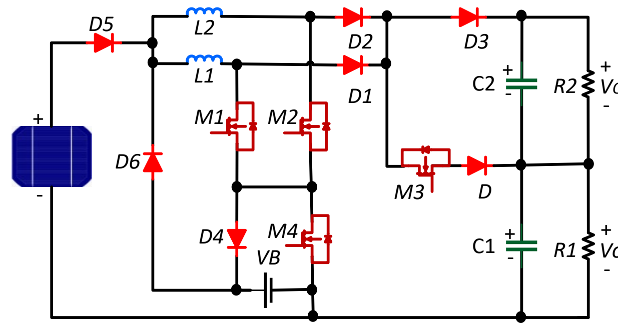


Fig. 9. Two input, multi-output Interleaved dc-dc boost converter.

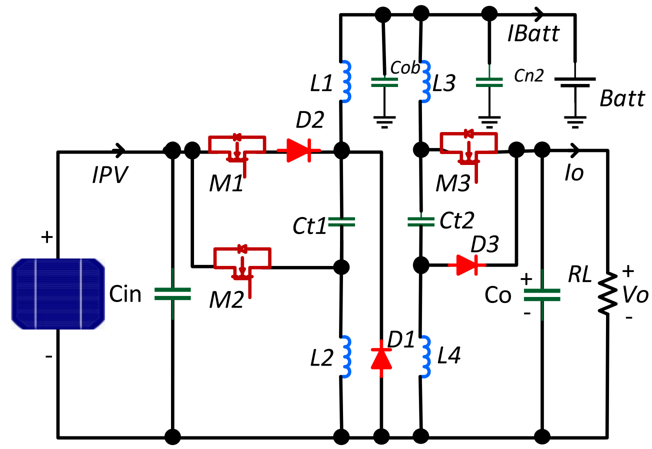


Fig. 10. Circuit diagram of B^{2}R topology.

reliability. However, it leads to high power losses due to its reverse recovery diode. Table IV gives a comparison based on reliability, mass, bandwidth frequency, and input current ripple. Moreover, the bandwidth efficiency of (d) and (e) is better compared to other given topologies. In SmallSat applications (a), (b), and (c) looks feasible with a good trade-off between reliability, efficiency, and the component count of the topologies. For the switch near-ground topology shown in Fig. 8 (b), the implementation of the driving circuit is easier compared to the rest of the same category given the fact that the power transistor is grounded. In addition, this converter topology combined with a DET EPS architecture offers good efficiency [65]. Furthermore, the two-inductor boost converter in Fig. 8 (c) has the merit of operating at continuous conduction mode (CCM) and therefore, this converter requires a small output capacitor to achieve reduced output current ripple.

Another effective solution to step-up the voltage level is the interleaved structure, which improves transient response, decreases output current ripple, and can reduce the passive component size [66]. Gorji *et al.* [67] presented an interleaved dc-dc boost converter with two input/multi-output capabilities for spacecraft applications represented in Fig. 9. The main task of this converter is MPPT power supply, bus voltage regulation, and battery charge control. The proposed converter replaces three separated converters through the usage of one compact circuit and operates in three different operation modes. In the first operation mode, the loads are supplied directly by the PV without requiring any battery operation, in the second mode, the loads are supplied directly from the battery power, while in the third operation mode, the converter supplies the loads through the combined operation of the PV and the battery. One of the prominent features is the converter battery base is grounded avoiding the noise of battery damage, hence, increasing the battery's useful life. Therefore, this topology is characterized by reduced size, low losses, and high efficiency.

3) Buck-boost based converter topologies: The converters described here can step up or down the input voltage to a regulated output voltage level and they support bi-directional power flow, enabling the fully controlled charge and discharge of the onboard batteries, based on the monitored state of the charge (SOC) [68]. A high-efficiency step-up and step down-conversion with the buck-boost regulator (B^{2}R) is presented in [69], which is shown in Fig. 10. The use of the converter has been applied in DET architectures of unregulated or regulated bus voltages with the battery directly connected to the main dc bus. In case the MPP architecture is selected, the usage of B^{2}R assists towards maintaining the main bus voltage close to the MPP solar array voltage level, a fact that ensures better performance, as shown in the practical application of such configurations in SmallSat [70]. This is an advantage that is not offered by a single buck or a single boost converter alone. B^{2}R topology has been improved in [70]. A multi-port dc-dc converter is drawn by merging two inductors, a topology known as buck-buck-boost regulator (B^{3}R). All the discussed

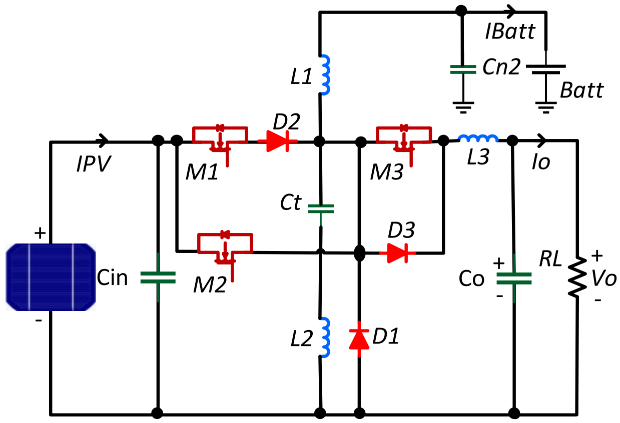


Fig. 11. Circuit diagram of B^{3}R topology.

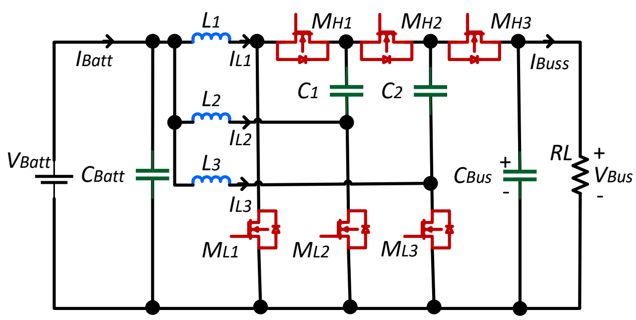


Fig. 12. A new proposed interleaved PWM bi-directional converter.

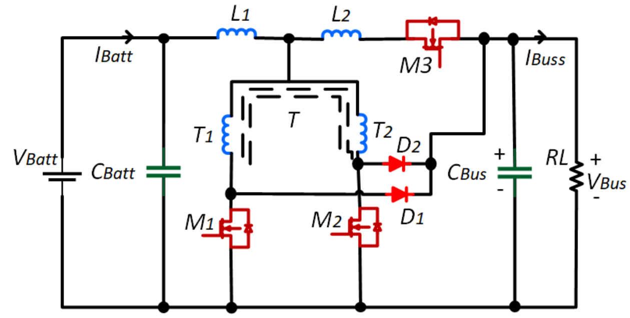


Fig. 13. Improved Weinberg topology for bi-directional power flow.

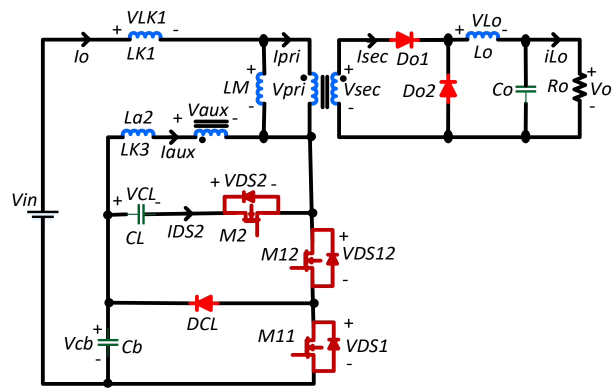


Fig. 14. A new ACF topology with a continuous current waveform.

operations of B^{2}R are performed by B^{3}R topology, including step-up, step-down of the solar array power to the battery level, and step-down battery power to the regulated bus level. Thus, this topology gives mass and cost savings as compared to B^{2}R, since the passive, and protection components are reused for PV array and battery regulations as appears in Fig. 11.

Conventional bi-directional converters are very challenging at extreme duty cycle operation, which is required to bridge the high voltage gap between battery cells and the main dc bus. Additionally, the battery's low voltage side increases current stress on the converter which is associated with more losses. In SmallSat applications, a bi-directional interleaved PWM converter with high voltage-conversion and current balancing ability is proposed in [71]. This converter is derived from the conventional interleaved PWM three-phase converter by adding two additional capacitors C_1 and C_2 as given in Fig. 12. Thus, by adding the capacitors the voltage conversion ratio is tripled at a given duty cycle, which relieves the stresses on the switches, and inductor current is balanced by the added capacitors. This procedure contributes to efficient power conversion for a single-cell battery SmallSat EPS. This converter operates at step-up and step-down at charge and discharge of the battery, respectively. However, analysis is performed based on ideal circuit elements with large capacitors, and parasitic components are not considered. Finally, in [72] a new bidirectional converter is proposed, being an improved Weinberg topology. A conventional buck circuit is placed on one side and a Weinberg boost topology circuit on the other side. It has a simple structure, high power density, and efficiency. This topology adds the diode and switches to realize the function of the bidirectional power flow, as shown in Fig. 13. This topology works on two modes, buck, and boost modes, achieving thus bi-directional power flow and consequently higher energy density within a smaller mass.

C. Isolated Converter Topologies

Specific satellite digital loads operate at higher currents (5 to 10A or more) while demanding a low voltage supply, which should be characterized by minimized ripple and fluctuations. Although, before the final load voltage, multiple stages of power conversion are accomplished. Regulation and isolation are required to be implemented [56]. In the past decade, isolated converters are widely employed in various formats in SmallSat applications. In addition, instead of using multiple converters working independently, one multi-port converter can be used instead, offering several advantages, such as fewer conversion stages and a smaller number of components, however, at the cost of reliability given that it can be proven a Single Point of Failure (SPoF).

1) Single-input single-output (SISO) converter topologies: The conventional active-clamp-forward (ACF) converter topology is comparatively due to its less component count, simple structure, and regarding its switch voltage stress, the good clamping competence, is a good candidate for SmallSat applications. However, the high switching stress, large *di/dt,* and pulsating input current are the drawbacks in this converter [73]. To overcome the disadvantages, a new ACF, shown in Fig. 14, with a continuous current waveform is proposed in

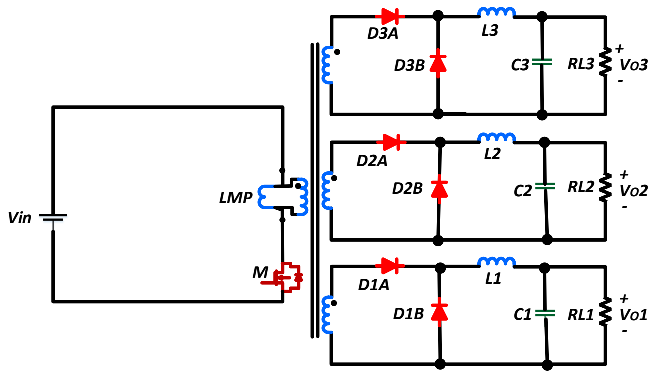


Fig. 15. Multi-output magnetic feedback forward converter.

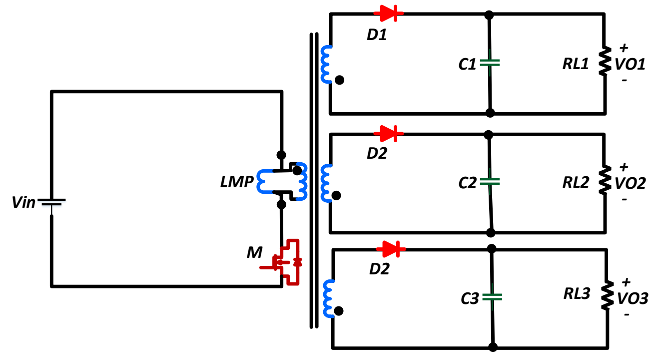


Fig. 16. Multi-output low drop-out fly back converter.

[58]. In the presented topology, two series-connected switches are utilized for relieving the stress on the main switch of the converter. For the turn-off time, a clamping circuit and a turn-off delay are applied to prevent an imbalance switch voltage. The voltage stress of both the switches are always clamped to V_{CL} and V_{in} respectively. Thus, the low voltage stress on the switches and low drain to source resistance results in a converter with high power density and reliability.

2) Single-input, multi-output (SIMO) converter topologies: A decentralized multi-output magnetic feedback forward converter for space applications is demonstrated in [74]. This converter is a galvanically isolated topology, with a magnetic feedback system implemented instead of the classic optocoupler feedback, which is sensitive to radiation and temperature fluctuations. The magnetic feedback and PWM controller are insensitive to the radiation effects and are in general insensitive to temperature deviations, increasing thus reliability [75]. This converter has simpler circuitry compared to other multi-output converters and is more efficient because of the self-resonant reset mechanism. Compared to some popular converter topologies such as half-bridge, full-bridge, and push-pull, the flyback multi-output converter is usually selected in SmallSat given the fact that it's less complex. This transformer-based converter enhances the topology with multiple output channels, with some minor additional circuitries [74]. The decentralized multi-output magnetic feedback forward converter is shown in Fig. 15.

Another like the previous SIMO converter is presented in [76], as given in Fig. 16. This converter is a low drop-out (LDO) fly-back converter for the spacecraft power subsystem. This converter topology is implemented for the high-power auxiliary output of the converter based on strict voltage regulations. Multiple outputs can be generated with some additional circuits since energy is stored in the transformer before transferring to the converter output. Through the transformer's turn ratio, the output voltages can be selected. Both converters are compared in Table V, in terms of electrical input and output, switching frequency, and efficiency.

3) Multi-input multi-output (MIMO) converter topologies: Integrated multi-port converters have fewer conversion stages, fewer component count, and switching devices. As a result, a multi-port converter is compact, more reliable, and has a lower mass compared to the independent port converters. Besides, there are no communication requirements, and the centralized control system enhances the dynamic performance of the converter. Qian *et al.* [77], [78] present the power and control strategy of an integrated three-port converter for satellite applications as shown in Fig. 17. Table VI gives a comparative overview between conventional and three-port converters. The three ports correspond to the solar input, the battery, and an isolated output port. During the illumination period, the

TABLE V
ELECTRICAL SPECIFICATION OF MULTI-OUTPUT MAGNETIC FEED-BACK AND FLY-BACK CONVERTER.

Converter Topology	I/P Voltage Range	O/P Load Current (Io) 1	2	3	O/P Voltage (Vo) 1	2	3
Magnetic feedback	18 to 50V	4A	0.67A	0.67A	5V	15V	15V
Fly back converter	24 to 42V	5A	0.1A	0.3A	5 V	12.5V	15V

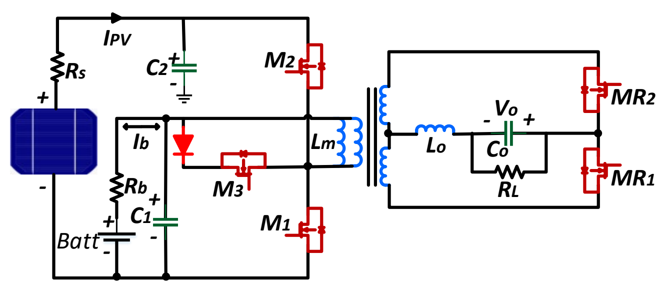


Fig. 17. Half-bridge three-port modified converter topology.

TABLE VI
A COMPARISON OF INTEGRATED MULTI-PORT AND CONVENTIONAL MULTI-CONVERTER.

Differences	Integrated Multi-port	Conventional Multi-converter
Control design	complex	simple
Control input	multi	single
Number of components	less	more
Mass	low	high
Conversion stages	one	more
Control loop decoupling	required	not necessary
Communication	no	yes

TABLE VII
A SUMMARY OF THE REVIEWED CONVERTER STRUCTURES FOR SPACECRAFT APPLICATIONS.

DC-DC Converter Topologies	Reference	Figure No.	No. Semiconductors		No. Passive Elements		SW frequency kHz	Efficiency (%)	Main Features
			Switch	Diode	Inductors	Caps			
Non-isolated	[61]	Fig. 6	2	2	2	4	400	97	• Physical electrical connections. • Normally, Simple structures with low build-up cost and light weight. • Suitable for medium and low power levels.
	[63]	Fig. 7	2	0	1	1	Unkn.	Unkn.	
	[65]	Fig.8 (a-f)	4	7	2	2	100	>96	
	[67]	Fig. 9	3	3	4	6	100	96	
	[69]	Fig. 10	3	3	3	3	100	96	
	[70]	Fig. 11							
	[71], [72]	Fig. 12, 13							
Isolated	[58]	Fig. 14	3	3	7	3	300	<95	• Précised magnetically coupled design. • Liberty of multiple output design with positive or negative voltages. • Reduced EMI and noise hitches.
	[74]	Fig. 15	1	6	8	3	500	>75	
	[76]	Fig. 16	1	3	5	3	200	86	
	[77], [78]	Fig. 17	5	1	5	4	100	>90	
Bi-directional	[69]	Fig. 10							• Two way power flow. • Required a complex control and switch drive circuit. • Capable for regenerative applications
	[70]	Fig. 11							
	[71]	Fig. 12	6	0	3	4	100	90	
	[72]	Fig. 13	3	2	4	2	50	>95	
Uni-directional	[61]	Fig. 6							• Single way power flow • Control and modulation is simple • Lower cost and simple structure compared to bidirectional
	[63]	Fig. 7							
	[65]	Fig.8 (a-f)	1	1	2	2	130	97	
		Fig. 8 (a)	1	1	2	2	130	<97	
		Fig. 8 (b)	1	1	2	2	130	<97	
		Fig. 8 (c)							
	[67], [58]	Fig. 9, 14							
	[74], [76]	Fig. 15, 16							
	[77], [78]	Fig. 17							
Current-fed	[65]	Fig.8 (a-b)							• Constant input current with low ripples • In-built boost characteristics • Low dynamic response
		Fig. 8 (f)	2	2	4	3	130	<97	
	[71], [72]	Fig. 12, 13							
	[76], [67]	Fig.16, 9							
Voltage-fed	[61]	Fig. 6							• High input current ripples • In-built buck characteristics • Fast dynamic response
	[63]	Fig. 7							
	[65]	Fig. 8 (c)	1	1	3	4	130	97	
		Fig. 8 (d)	1	1	2	2	130	<97	
		Fig. 8 (e)							
	[69], [70]	Fig. 10, 11							
	[74], [76]	Fig. 15, 16							
	[77], [78]	Fig. 17							

converter operates at MPPT, while the battery is being charged and the loads receive fully regulated power. The three circuit stages are based on the half-bridge converter control which operates on modified PWM with a constant switching cycle and results in two independent control variables d_1, and d_2 duty cycles for operating switches M1 and sM2 respectively, which tightly regulates two ports of the converter, whereas the third port offers balance to the circuit. However, the control design is more complicated and demands more modeling efforts as compared to classic two-port converters. Further analysis and operation of the circuit are reported in [79], [80]. For more insights, the main features of the reviewed dc-dc converter topologies for spacecraft applications are summarized in Table VII.

D. Converter Switches

While previously silicon-based switching technologies (Si) dominated the space industry, by mid 2010s gallium nitride (GaN) switches have dominated over the Si ones. GaN devices are radiation tolerant, and sustainable for the high-temperature space environment. Additionally, these devices offer higher efficiencies with lower cost, mass, and volume, therefore achieving higher power densities. The radiation tolerance of this switching device is due to the high bandgap energy of 3.4

eV for GaN [50], [81], [82]. Additionally, enhancement-mode GaN (eGaN) transistors are more suitable for high-efficiency converter designs due to on-state resistance, output capacitance, and small gate charge [50]. The soft-switching capability of GaN transistors along with their improved output power density under resonant operation has been evaluated in [83], [84]. In these papers, the benefit of replacing Si-MOSFET with eGaN-FET has been experimentally demonstrated for a POL printed circuit board (PCB). The GaN FETs with low parasitic packaging need a low parasitic PCB layout for the device capability utilization. Based on this assessment, an optimal layout with eGaN FETs is suggested to attain the best performance and this design has offered a 10% decrease in total power loss, 40% reduction in high-frequency loop inductance, and 35% voltage overshoot minimization, compared to the classical PCB scheme [84]. In comparison to the traditional radiation-hardened switches, GaN technology offers significant cost reduction, high switching frequency with smaller component packages, and introduces fewer losses [85].

IV. ENERGY GENERATION

The available energy sources in space are naturally of solar or nuclear origin. In recent years, a massive surge in the use of PV cells has been seen in satellites regardless of their size, with a total ~85% of all SmallSat spacecraft using solemnly solar panels as the primary power-generating source [86]. The solar cells used in space applications, compared to the terrestrial ones, harvest energy from a broader spectrum of solar irradiation [87]. More specifically, space-qualified solar cells are designed for the spectrum AM0 (ASTM E-490), with an integrated power density of 1366.1 W/m^2 [88], while terrestrial cells correspond to the direct and global spectrums AM1.5D and AM1.5G (ASTM G-173) respectively, with the corresponding integrated power densities of 900 W/m^2 and 1000 W/m^2. For reference, the terrestrial and LEO spectra are given in Fig. 18 (a) and (b). It is therefore evident, that there is increased potential for elevated solar energy harvesting and, that different technique is necessary for space-qualified solar cells. The power generating components are selected from the available COT components available from different producers. However, power generation-related failure is the major among the other EPS elements, in which cell, solar array, cell interconnection, array mechanical failures, and darkening of solar reflectors or glass has been reported [89]. Some cell degradation and failure are due to unexpected radiation-induced degradation caused by energetic solar flare and others are due to manufacturing defects. But, since the solar array is sized for the end-of-life power, a large enough margin is added in the design to mitigate the impact of degradation. Conventionally, two solar cells are serially wired in the CubeSat solar panel with no redundancy in case of solar string failure which can cause a serious threat due to loss of electric energy from one solar panel. To address the reliability in the case of solar string failure or following switched power regulator failure a redundant architecture is proposed in [90]. In the proposed architecture a greater number of the cells are applied for advanced interconnection in a single panel to form independent power generation branches to escape from the overall failure of the panel.

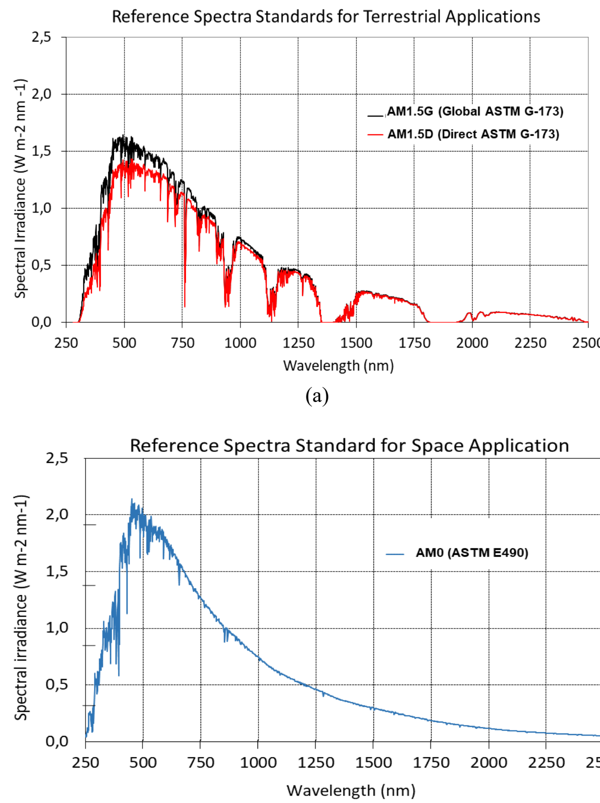


Fig. 18. Standard sunlight spectrum for (a) terrestrial and (b) space solar cells.

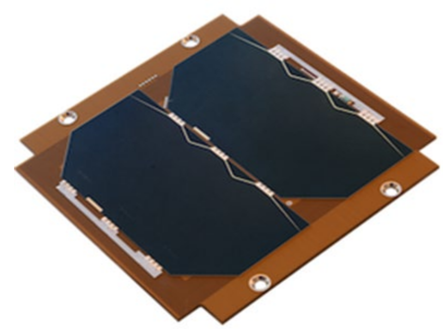


Fig. 19. NanoSat solar panel “NanoPower P110” [Courtesy GOMSpace].

A. Space-qualified Solar cells

The more all-electric satellites are expanding, the need for electric propulsions will keep significantly increasing the power/energy budgets, a fact that will impact significantly and directly the solar panel design in terms of sizing and power density. Therefore, the increase in the number of solar panels significantly raises the overall mission cost in terms of mass and volume, thus high efficiency, small size, and lightweight solar panels are desired. An example of which is shown in Fig. 19. There are various types of solar cells with different efficiencies.

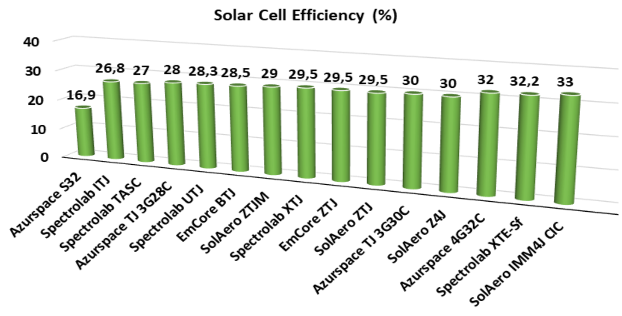


Fig. 20. Space solar cell technologies and their efficiencies [91], [92].

TABLE VIII.
SPACE VERSION SOLAR PANELS WITH THEIR EFFICIENCIES AND PRAGMATIC SOLAR CELLS.

Manufacturer	Name of Panels	Applied Solar Cells	Efficiency
SoleAero	COBRA COBRA-1U	SolAero ZTJ	29.5%
Clyde Space	Solar Panel (0.5-12U) Deployable Solar Panel (1U, 3U)	AzureSpace 3G30C Spectrolab XTJ Spectrolab UTJ	29.6% 29.5% 28.3%
MMA	HAWK eHAWK	SolAero XTJ & Prime	29.5-30.7%
GomSpace	NanoPower	AzurSpace 3G30A	29.6%
Endurosat	Solar Panel	CESI Solar cells CTJ30	29.5%
DHV	Solar Panel (5×5 cm, 1U, 3U, custom)	AzureSpace 3G30C Advance	29.6%
SpectroLab	Space Solar Panel	SolAero XTJ Prime SolAero XTJ SolAero UTJ SolAero ITJ	30.7% 29.5% 28.3% 26.8%

In terrestrial applications, the most common solar cells are Si-based, which can reach up to ~20% conversion rates. However, such cells perform even worse when in orbit, not only due to their inherent low conversion rates but most importantly because they experience severe efficiency degradation over time deriving from the radiation environment. Thus, Si solar cells are not preferred for modern satellite applications. To overcome the limitations of Si-cells, multi-junction solar cells (MJSC) have been developed for space applications. MJSCs are well over 10% more efficient compared to Si-based ones [93].

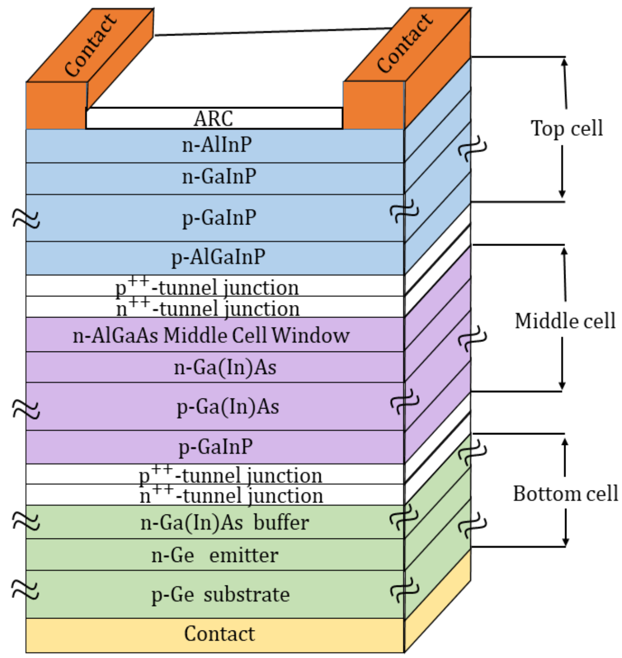


Fig. 21. Design layers of 3-J Ge/GaAs/InGaP solar cells.

Some available space solar cell technologies for SmallSat applications are shown in Fig. 20. More specifically, recent studies have demonstrated that MJSCs under concentrated sunlight reach energy conversion efficiencies of 44.4% for 3-Junctions (3-J) and 46.1% for 4-junction (4-J) [94], [95].

Regarding SmallSat missions, mostly 3-j solar cells are used, however, in the past 5 years, 4-j cells are growing fast in applications [92]. The prevalent 3-j commercial COTS space-qualified PV cell technologies designated for SmallSats are summarized and listed in Table VIII, as to their reported efficiencies and solar cell types [96]-[104].

B. Design and Construction of Multi-junction Solar Cells

The MJSC are manufactured by the combination of several layers of gallium arsenide (GaAs), indium gallium phosphate (InGaP), and germanium (Ge) or Si to capture the largest possible spectrum of sunlight. The architecture of the 3-J solar cell is shown in Fig. 21, which highlights the design layers of the 3-J Ge/GaAs/InGaP solar cell. Such cells are easier to manufacture compared to other higher-order MJSC which achieve better performance [105]. Moreover, this solar cell is super radiation tolerant and owes a higher cell MPP voltage (VMPP). The 3-J solar cell consists of three p-n junctions arranged one on top of another, connected via tunnel junctions for the addition of sub-voltages and maintaining the overall polarity of the device. For a single-junction GaAs, the nominal cell voltage is 0.89V and the temperature coefficient is 2 mV/°C at VMPP [106]. In addition, the radiation response of the cell is controlled by the most radiation-sensitive sub-cell photocurrent [107]. The conversion efficiency of the 3-J solar cell has been steadily improved to approximately more than 30% at the beginning of life (BOL) [108]. The base layers of

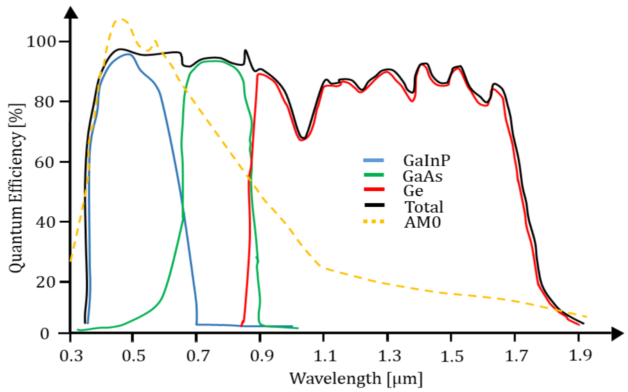


Fig. 22. Wavelength spectrum covered by the base layers of the structure.

the PV cell cover a wavelength spectrum which is shown in Fig. 22, at an Air-Mass Zero (AM0) illumination and a temperature of 28°C.

Recent research has suggested a new cell architecture called Inverted Metamorphic Multi-Junction (IMM). IMM cells are lighter in terms of mass and are more efficient compared to 3-J cells [109], [110]. Yamaguchi H. *et al.* [109], [111] proposed space solar sheets with inverted 3-J cells. The authors carried out detailed reliability tests, which eventually lead to the I-V characteristic improvement of IMM-3J space solar sheets. Another 3-j film type solar sheet has been proposed in [109], which poses 10% higher efficiency than previous Ge/GaAs/InGaP 3-J solar cells and is lighter in weight. A comprehensive reliability test for the space environment has been conducted including a successful thermal cycling test, which has shown sensible performance in outdoor field testing.

C. Electrical Circuit Model of Multi-junction Solar Cell

Theoretically, an ideal solar cell could be modeled as a current source in anti-parallel with a diode. A direct current is generated with solar radiation, and it accordingly varies when the cell is subjected to light. The improvement in the model embraces the effects of shunt and series resistors [112]. The Ge/GaAs/InGaP 3-J solar cells can be modeled as the circuit in Fig. 23 where each sub-cell is representing a single independent solar cell. The three equivalent solar cells are arranged in a way that enables them to be shrinking the gaps and be connected in series from top to bottom. The electrical performance (I-V diagram) of 3-J solar cells can be derived from the three sub-cells and the sum of total cells, which is shown in Fig. 24. While each sub-cell possesses the same current because all sub-cells are connected in series [113], [114]. Similarly, 4-J cells can be modeled as a four-level equivalent circuit.

The representation of the mathematical expression of the current generated by each sub-cell is given as follows:

$$I^i = I^i{}_{PV} - I^i{}_{sh}\left(e^{\frac{q\cdot(v^i+I^iR_s^i)}{a^i\cdot K\cdot T}}-1\right)-\frac{(v^i+I^iR_s^i)}{R_{sh}^i} \qquad (1)$$

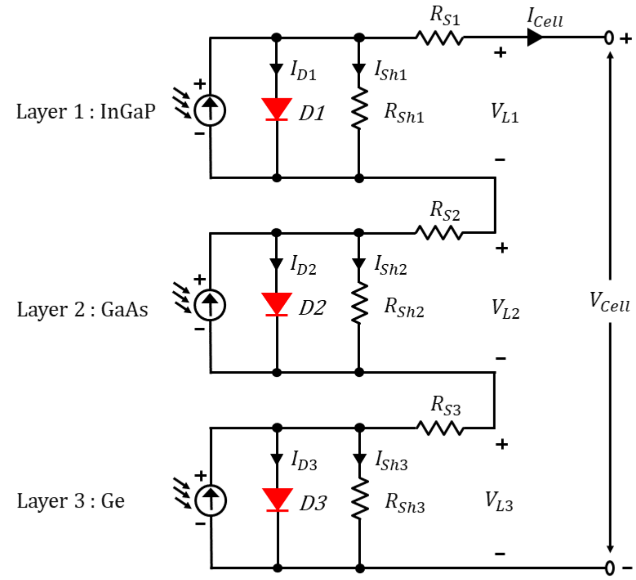


Fig. 23. Equivalent circuit of a 3-J PV cell.

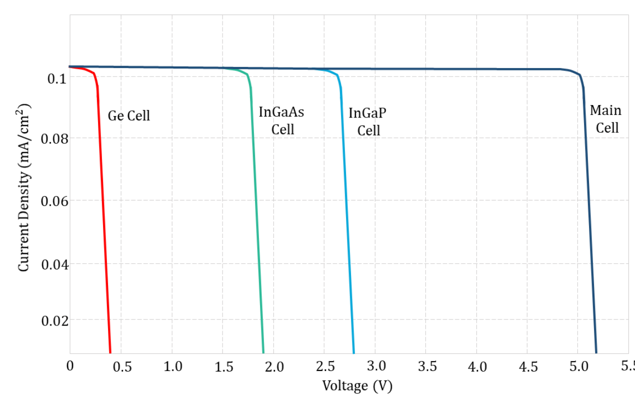


Fig. 24. I-V curve for three sub-cells and their sum.

where I is the current of sub-cells, i is the number of sub-cells, (top: i=1), (medium: i=2), (bottom: i=3). I_{PV} is the sub-cell photocurrent, I_s is the sub-cell inverse saturation current of the diode, a is symbolize Boltzmann constant, T is the temperature of the sub-cell, K is the electron charge, and q is the diode ideality factor, V is the total voltage across the cell, R_s and R_{sh} are series and shunt resistances [115].

The sum of the voltages of all sub-cells is equal to the total voltage of the cell as follows:

$$V = \sum\nolimits_{i=1}^{3} V^i \qquad (2)$$

The total current is limited to the sub-cell that generates the minimum, from the three cells connected in a series configuration which can be expressed as:

$$I = \min(I^i) \qquad (3)$$

For the I-V model of the panels, as the shunt current I_{sh} is minimum, shunt resistance R_{sh} may be neglected [116].

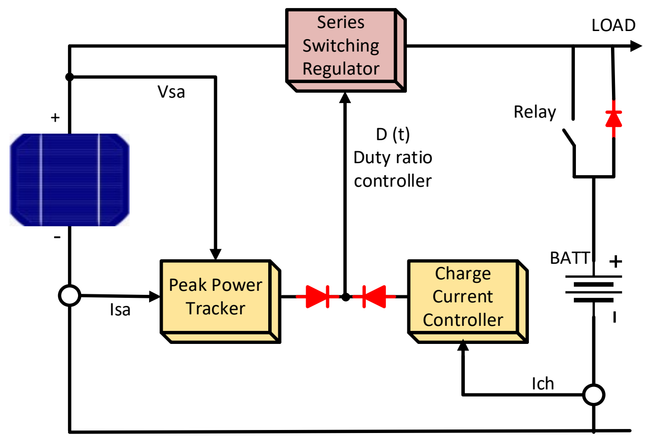


Fig. 25. An architecture of PPT for a mission with wide solar and temperature variations.

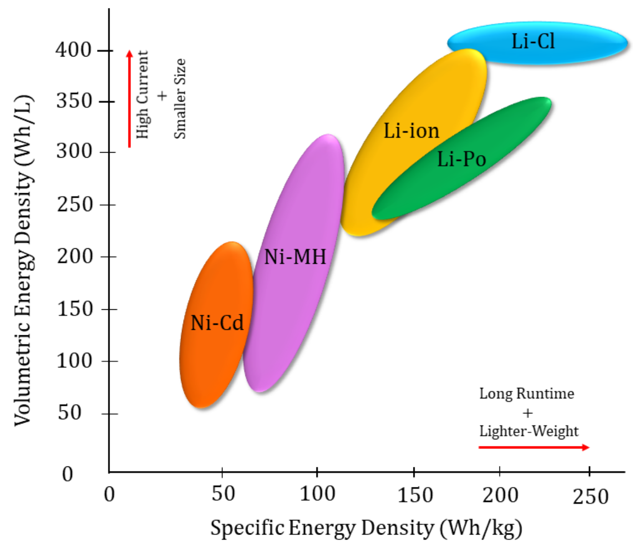


Fig. 26. Energy densities of different batteries for nano satellite applications.

D. The Maximum Power Point Tracking

In MPPT architectures, a switch-mode converter is placed in series with the solar array to dynamically regulate the array output impedance to match the loads. The majority of the SmallSats utilize MPPT architecture [38]. It manipulates either the operating current or voltage of the solar array and drives the operating point of the solar array by controlling the operation of the switching converter between the rest of PDMS and the solar array. The MPP calculation depends on the number of parameters such as relative positioning to the Sun, PV cell type, operating temperature, and total solar irradiation [117], [118]. The block diagram of MPPT architecture is shown in Fig. 25. There are several MPPT calculation techniques, to name a few perturb and observe (P&O), constant voltage, artificial neural networks, and incremental conductance (INC), each characterized by different accuracy degrees and complexity [119]. P&O algorithm is mostly used in LEO SmallSats due to its easy implementation, low complexity, and accurate tracking of MPP [120]. However, it suffers from plenty of drawbacks such as steady-state oscillation around the maximum power [121], therefore there are several efforts i.e., incremental perturbation and observation (IP&O), the optimal P&O towards improving it, without compensating them completely [122].

V. Energy Storage System in SmallSat

To maintain the continuous operation of the satellite under eclipse and peak load periods, all SmallSat require an energy storage system (ESS) that includes batteries. Cell is the elementary unit of the battery and multiple cells are series, parallel, or both combined as a battery pack. The onboard batteries are differentiated based on their usage as the primary and secondary batteries: the primary batteries (e.g., pyro batteries) are not rechargeable; they are used for short mission requirements (approximately from a single day to a week), after usage they are disposed-off [123]. On the other hand, the secondary batteries are rechargeable and are an essential, permanent part of the EPS. In this review, we focus only on the secondary batteries, which are for long period applications (rechargeable) and are the main storage devices required for the mission life. The secondary batteries according to their volumetric and specific energy densities for nano satellite applications are shown in Fig. 26. Another, vital step in any satellite design is the correct sizing and right selection of the battery type, capacity, and technology which are made according to each mission's requirements. The most prominent aspect in sizing and right selection is the mission lifetime, the power/energy budgets, and the operating temperature range along with the available thermal management strategy [124]. Moreover, the space extreme conditions need intense design and component selection strategies because batteries are chemical reaction-based energy releasing devices and operating environmental conditions affect their performance. In some missions the temperatures fluctuate from -20 to -100°C which greatly affect the rate of charge and discharge. Also, thermal runaway can occur if a battery gets too hot [125].Thus it must be made certain that the batteries can function at these temperatures where batteries need to undergo intense testing under several different conditions to be approved for use. Also, to protect batteries from the extreme fluctuations of space, heaters are added throughout the battery's cells to regulate their temperature.

A. State-of-the-art Battery Technologies

A survey made in [126] reveals that for nano-satellite applications batteries used are 4% Lithium-Chloride (Li-Cl), 12% Lithium-polymer (Li-po), 16% Nickel-Cadmium (Ni-Cd), and 2% other chemistries. While the rest of 66% Li-ion batteries have been applied. The Li-Cl is one of the highest energy content carrier batteries however, the high energy capability makes it very sensitive from the designing point of view [127]. Previously, battery technologies including nickel-hydrogen (Ni-H2) have been employed in different space missions [128]. The Ni-Cd up to the 1990s was a good solution for LEO space applications due to their high reliability and long-life cycle,

TABLE IX
THE COMMERCIAL OF-THE-SHELF BATTERY TECHNOLOGIES AVAILABLE FOR SMALL SATELLITE APPLICATIONS.

Manufacturer	Product Name	Cells Used	Energy Density ($Whkg^{-1}$)
ABSL	COTS 18650 Li-ion Battery	Sony, MoliCell, LG, Sanyo, Samsung	90 – 243
EaglePicher	Rechargeable Space Battery (LP-33330)	EaglePicher Li-ion	105
GomSpace	NanoPower BP4 NanoPower BPX	GomSpace NanoPower Li-ion	1432- 154
Vectronic	Li-ion Battery Block VLB-X	SAFT Li-ion	Unkn.
Blue Canyon Technologies	BCT Battery	Li-ion or LiFePo4	Unkn.
Canon	BP-930s	*fo*ur 18650 Li-ion cells	132
AAC Clyde	40Whr CubeSat Battery	Clyde Space Li-Polymer	119

while Ni–H2 battery was the choice for flights requiring high charge rates, varied operating temperatures, and resiliency to disturbances [129]. However, Ni-Cd battery has a relatively low energy density while Ni-H2 holds a reduced cycle. Striving to overcome the drawbacks, Li-ion technology has emerged, and it has several benefits such as a long-life cycle, low self-discharge rate, improved working life, and no memory effect. The li-ion cells are extensively applied in CubeSats because of their right size, tolerance in the space environment (±100^0C). These cells are available Cylindrical 18650 in an improved form of 3500-3600 mAh, an average voltage of 3.6 V, and 50g of weight. The energy density of 18650 cells is 762 Wh/l and 252 Wh/kg [123]. Moreover, Li-ion batteries have low volume and mass and are available in a variety of forms [130]. The available cells are 65mm in length and 18mm diameters bearing efficiency of up to 97% at BOL [131], [132]. Another similar battery technology is polymer Li-ion (Li-Po) which is available in a pouch instead of a cylinder, which gives freedom to develop lighter and thin cell designs for achieving advanced specific energy. Therefore, Li-Po cells are suitable for high power and energy SmallSat applications; however, they are exposed to temperature and external mechanical destruction of the space environment for their thin cell casings [133]. This problem has been solved by wrapping each cell in a thin copper layer and soldering the copper substrate to the battery cells with the power conditioning board [134]. Generally, a Li-ion cell has a larger capacity, but Li-Po has a gravimetric energy capacity of 1.2 to 1.6 times larger and only 17% volume of Li-ion. Some available latest battery technologies for SmallSat applications are presented in Table IX [30]. Supercapacitors (SC) or ultracapacitors can be considered an alternative ESS for small satellites because of their high-power density, long charge/discharge life, and operation in wider temperature ranges [135]. SC could be used as the sole energy storage system for CubeSat that overcomes some of the disadvantages of Li-ion batteries like limited lifetime, high cost, and stable temperature requirements [136]. However, the low energy density of SC makes it limited as main energy storage system. Therefore, a hybrid energy storage system of SC and Li-ion offer the advantages of each technology for the challenging mission requirements. Chin *et al.* [137], have proven an onboard Li-ion battery and SC technologies, a hybrid ESS in 2U CubeSat flight. The primary and secondary phases of the mission have been completed. The results have shown an excellent agreement between the two technologies and the performance characteristics in different conditions. The ground test results have sufficiently met, particularly concerning the percent of capacity contributions between the Li-ion cell and the SC. A feasibility study in [138], demonstrates that the supercapacitors can qualify in a radiation environment, high cycles life (>100K), testing and launching process for small satellite applications. The characterization of different battery technologies suitable for SmallSat applications is examined in [139] which has reported procedures and results of several environmental-related tests of performance degradations for Li-Po cells.

B. Battery Charge Regulator

The battery charge regulator is responsible for harnessing solar power to sufficiently charge the battery cells. The battery charging system interfaces the battery and solar panels. It is a programmable buck-boost converter and can operate in the constant voltage or current mode depending on the battery state. Though, modern battery technologies are characterized by improved Wh efficiency and extraordinary power density. However, batteries including Li-ion or Li-Po are not tolerant to

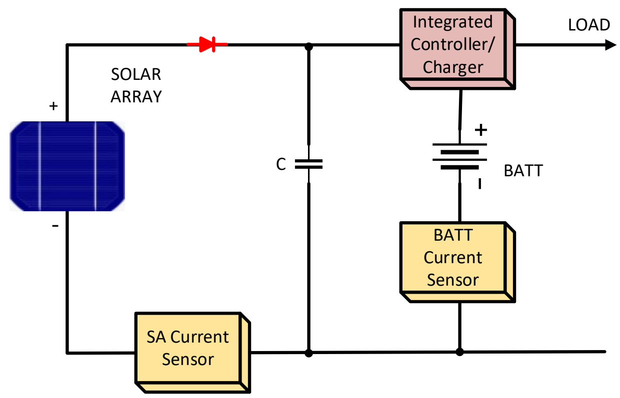


Fig. 27. Integrated battery charger and array controller.

overcharging. An integrated array controller battery charger is proposed for SmallSat in [140], which can be seen in Fig. 27. The circuit is placed between array bus and battery/load, ducking a direct contact of bus voltage and the battery voltage to clamp. With this approach, the battery charging with available power or fully powering the bus is possible, which on the other hand results in more efficiency in battery charging [141]. The control function is carried out by a single PWM system, where a current control boost converter is used to feed the solar array power to the load and after processing, to the battery. The control functionalities are accomplished by varying the duty cycle with the current of the battery as a controlled variable. Moreover, the array voltage is linearly varied across different power generating points to support the variations in power requirements. The integrated controller efficiently charges the battery with constant current to the setpoint and protects the battery by turning off to zero thereafter. This is performed by controlling the converter duty cycle. For implementation, a PWM controller in conjunction with an analog-based two-loop control scheme is applied.

In the first loop known as, Battery Charge Controller (BCR) loop, the charge reference signal is compared with battery current by the integrated controller and generates a control signal to control the duty cycle of the converter. The current controlled boost converter with an additional outer voltage loop is involved in this scheme to cut the overcharging current of the battery. In the second loop, the detected output battery voltage is compared with the set battery charge reference voltage. The voltage loop senses the set voltage, and the battery charge current is minimized to zero indicating the required SOC. Similarly, the battery current reference point is pulled down to zero by the battery voltage loop active pull-down.

VI. THE CHALLENGES AND POTENTIAL SOLUTIONS IN EPS DESIGN AND OPERATIONS

Recently, the SmallSat missions have been transformed to commercial ventures instead of academic and experimental projects changing the prospect of specifications and reliability of the CubeSat. The specific orbits have been extended to larger and mission lifetimes have been extended from months to years. The main challenge in SmallSat design is the requirement of highly reliable and capable components, efficient and simple EPS architecture, low power, low mass, and size constraints. Whereas the space-capable and reliable systems require greater resources of power, volume, and mass [142]. While COTS components are used in the design of the SmallSats, which are primarily not dedicated to the space weather. Typically, the COTS components better perform than space rated parts but lacking the harsh environments survivability [143]. On the other hand, military version aerospace and avionics electronic components are crafted to sustain space radiation, vacuum, vibration, and extreme temperatures encountered in the upper atmosphere and at the lift-off process. The extreme temperatures at the surface of the spacecraft fluctuate up to $\pm 100^0$C at LEO [13], which shortens the life span of the casual electronic components but also mangles the usual operational functionality of the satellite. According to the standard in [144], for SmallSat extreme temperatures (i.e., hot, and cold cases) and rate of change of temperatures, testing in a vacuum is mandatory. Eight thermal cycles shall be performed between maximum and minimum temperature limits. However, the temperature is not the only factor, but cosmic rays also pose an additional challenge which is a constant bombardment of the highly energetic particles carrying energy ions with the capability to energies electronic components of the spacecraft, resulting in damages and malfunctioning. The satellites orbiting low altitudes are more vulnerable to these radiations [145]. There are two types of effects to the satellite electronics from the radiations: single event effects (SEEs) and total ionizing dose (TID). SEE, is an instantaneous failure mechanism expressed in terms of a random failure rate and TID is a long-term failure mechanism that is rated by mean time to failure. Also, there is less structural mass shielding the electronics, but the TID radiation effect can be reduced and the tolerance to this radiation is improved with thinner oxides and finer IC geometries [146]. The vacuum of outer space produces whiskers. Whiskers are spider-web-thin conductive filaments that grow on cadmium, zinc, and metal surfaces of the tin. Tin seems to be more likely to grow over time whiskers in a vacuum, providing a short circuit path between metal-plated surfaces [147]. However, many dedicated SmallSats of this category are offering an incredible job to explore space environments, which will enable building more robust and reliable systems [148]. In addition, at the time of left-off vibration is the sudden application of 9.2 million pounds of thrust to the satellite [146]. There have been many reasons cited for the high failure rate of SmallSat, such as ambitious technology infusion and the lack of testing, possibly related to low budgets in the hobbyist and academic sectors. The electronic components used must be designed and tested to withstand the shocks, vacuum, required temperature, and radiation hardened. The Small Spacecraft Systems Virtual Institute NASA in [143], presents state-of-the-art small spacecraft technologies for EPS applications. Although, potential and less expensive fabrication tools, and testing facilities for vacuum and high-thermal gradient are reported in [149]. Moreover, the analysis of thermal, degradation and management for the EPS equipment are very important in the design process.

A. Thermal and Degradation Analysis of Solar Cell

When exposed to this radiating environment, the semiconductors and mainly, the solar cell electrical performance degrades. This effect is very dangerous and can lead to mission failures. Hence, the radiation response of the solar cells employed in the spacecraft is extremely important for mission life prediction. For the prediction of the degradation level of the solar cells particularly some electrical parameters, e.g., open-circuit voltage, short circuit current, and the maximum power reaction in the radiation environment are necessary to be identified. Xin *et al.* [150] and Sato *et al.* [151] have presented performance evaluation and prediction of InGaP/GaAs/Ge 3-J solar cells under the irradiation environment. By calculating the open-circuit voltage and the short-circuit current, the degradation curves show an accuracy of 5%, which is a good agreement with experimental data. For satellite applications, Meng *et al.* [152] have presented a degradation model of the orbiting current for GaInP/GaAs/Ge 3-J solar cells. The performance parameter, which is crucial for describing the degradation of a solar cell model, is the output current. To model the variation tendency of the output current, a mathematical model is established. The results of the applied degradation model for lifetime prediction of 3-J solar cells contribute to the life expectancy of the cells for space applications. On the other hand, high temperatures extremely degrade the performance of the solar cell. To obtain better performance, passive treatment method such as back surface coatings and paint is applied [153]. Furthermore, the solar array backside thermal surface treatment showed temperature reduction to a great extent, increasing the efficiency of the solar cells broadly. Some thermal analysis, control, and the design of small satellites are reported respectively in [154] [155].

B. Electronic Equipment and Battery Management

The design of the satellite must be validated with all contemplations of the environmental conditions from launch to operation in orbit. The satellite is designed to sustain in the permissible temperature range and space vacuum atmosphere. To ensure the satellite functionality and survivability in space harsh weather for the mission lifetime span, suitable thermal management and design are mandatory to keep all the onboard equipment to their acceptable scaled temperatures. There are two thermal control techniques excessively discussed, active and passive but the latter one is referred to as the best solutions for NanoSats and CubeSats [156]. Passive thermal control generally relies on multi-layer insulations to shield the spacecraft from the incident heat fluxes of the space environment. These multi-layers are coatings and surface finishes, heat sinks, and thermal insulations. Surface finishes are applied on both the exterior and interior surfaces of the NanoSats and CubeSat. To obtain acquired emissivity and absorptivity two or more layers are combined and applied [157]. Excessive cooling can damage and degrade the battery module. For active control, a major technique is the application of a thin film heater. The heaters are resistors and are part of the closed-loop system of the controller and sensors.

VII. Conclusions and Future Perspectives

This paper has reviewed in a holistic manner, the state-of-the-art developments from origin to classification and utilization of SmallSat EPS, an overview of recent research and advances in SmallSat is presented aiming to highlight the current attention from academia and industry. For the LEO satellite bus regulation, battery rapid charge-discharge regulations, and subsystem distributions different converter topologies are available, some of the important topologies have been comprehensively reviewed i.e., non-isolated, isolated, uni-directional, bi-directional, current-fed, and voltage-fed topologies are analyzed in this paper. For bus regulation, some important converter topologies are provided to give an insightful understanding of SmallSat power bus regulation. Additionally, for the switching applications of SmallSat converters and regulators and load ON/OFF switches some up to date GaN converter switches are investigated and compared with classical Si-based switches, which shows high efficiency for SmallSat applications. Moreover, state-of-the-art solar power generation technologies have been discussed with the focus on the solar cells and panels available for SmallSat applications. For power generation various types of PV technologies are surveyed, standard design layer of the 3-J solar cell and PV circuit model is presented in this paper, including commercially available space solar cells, panels, and their characteristics. Although, state-of-the-art space version solar panels with pragmatic solar cells are illustrated which offer excellent candidates for future missions with high conversion efficiencies. The state-of-the-art battery technologies are reviewed and analyzed, with recent developments for the SmallSat application, where characterization of different battery technologies is carried out, and various commercially available battery chemistries are examined. Additionally, the main SmallSat design challenges and some potential solutions are addressed. This paper presents a clear picture for the selection of state-of-the-art architectures, converter topologies, Solar cells, and battery technologies that result in building a more reliable, efficient, and robust EPS for SmallSat applications. Future challenges and prospective are summarized as follows:

- Application of state-of-the-art technologies in the design, modeling, and architectures of electrical power systems for CubeSats will result in a sufficiently reliable and robust EPS.
- Most of the small satellite consists of subsystem COTS components that are modular. These components are to be planned depending on configuration and application requirements. For individual constraints, optimized modular components are needed to meet the strict space requirements in terms of generated power, volume, and mass.
- In future assessments for small spacecraft applications, the 4-junction GaAs solar cells have emerged with more efficiency and reliability. Another advanced solar cell is IMM, which is lighter in weight and is a more efficient solar cell. Thus, 4-junction GaAs and IMM solar cells are recommended for CubeSat analysis.

- In recent literature Li-ion battery technologies are mainly focused on the design, for the future perspective, Li-po, LI-Cl and SC technologies can be more critically analyzed for SmallSat applications due to their flexibility, less volume, and the high energy density compared to Li-ion.

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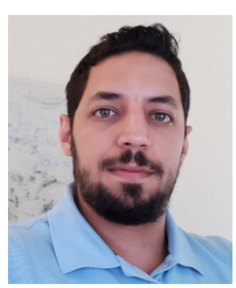

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