TAFITI PROJECT

1U CUBESAT ELECTRICAL POWER SYSTEM

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List of Abbreviations

BMS - Battery Management Systems

COTS - Commercial Off-The-Shelf

DET - Direct Energy Transfer

DoD - Depth of Discharge

EOL - End of Life

EPS - Electrical Power System

GEO – Geostationary Orbit

ISS – International Space Station

LDO – Low Dropout Regulator

LEO - Low Earth Orbit

LEO – Low Earth Orbit

MPPT - Maximum Power Point Tracking Controllers

OAP – Orbit Average Power

OBC - On Board Computer

PCDU- Power Conditioning and Distribution Unit

PCU – Power Conditioning Unit

PDU - Power Distribution Unit

PMAD - Power Management and Distribution

PPT – Peak Power Tracking

SoC - State of Charge

SoH - State of Health

SSR - Solid State Relays

STK – System Tool Kit

1 INTRODUCTION

The Electrical Power System (EPS) is a fundamental subsystem for any satellite, serving as its primary energy lifeline. Its core responsibility involves harvesting energy, converting it into a usable electrical form, and then distributing this power reliably to all other onboard subsystems and payloads. This continuous provision of power is essential for the satellite's effective and uninterrupted operation throughout its mission duration.

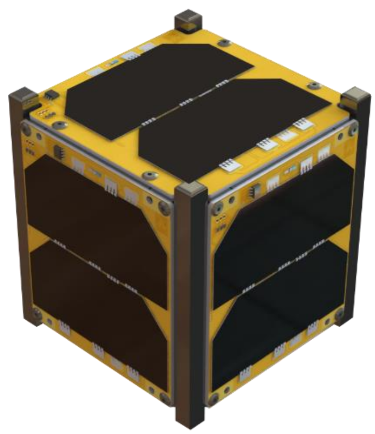

Figure 1.1: A 1U CubeSat with Body-Mounted Solar Arrays

This criticality is amplified in the context of a 1U CubeSat as shown in figure 1.1 above. The CubeSat standard defines a "1U" unit as a cube measuring 10x10x10 cm with a maximum mass of 1.33 kg. These extreme constraints on volume, mass, and available surface area present significant challenges for the EPS designer. The limited surface area restricts the size of solar arrays, thus limiting power generation capacity. The mass and volume budget curtails the size and capacity of the batteries.

Satellites primarily derive their power from a combination of photovoltaic (PV) solar panels and rechargeable batteries. Solar panels are engineered to convert incident sunlight directly into electrical current, acting as the satellite's primary power generators. The batteries function as crucial energy reservoirs, storing excess power and providing continuous supply during periods when direct solar energy is unavailable. Such periods include orbital eclipses, the initial launch phase before solar panel deployment, or unforeseen emergency scenarios

The operational of the EPS is linked to the satellite's orbital mechanics. During the "sun phase" of an orbit, when the satellite is illuminated by the sun, the solar panels are actively generating power. This generated power simultaneously fulfils the satellite's immediate operational loads and charges the onboard energy storage system. As the satellite transitions into an "eclipse phase" and enters the Earth's shadow, solar power generation ceases. At this point, the stored energy from the batteries takes over, ensuring the satellite's continued functionality. This cyclical demand necessitates that the batteries undergo constant discharge and recharge cycles, a particularly pronounced characteristic for satellites in

Low Earth Orbit (LEO), which can experience eclipse durations of up to 36 minutes within a typical 92-minute orbital period.

The EPS is structured around three main units: the power generation unit (solar panels/arrays) and the energy storage unit (batteries), both managed by a power control and distribution system as shown in Figure 1.2 below.

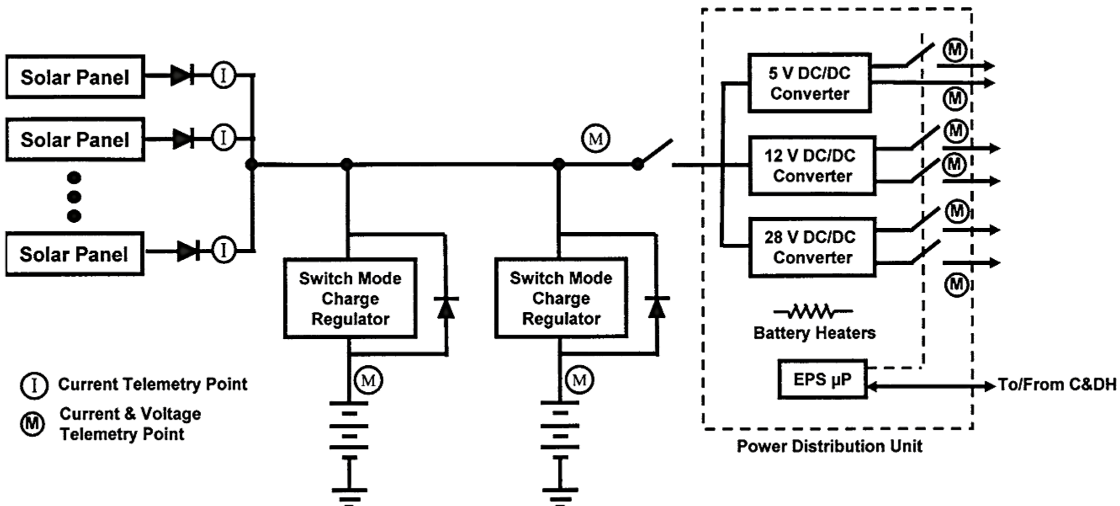

Figure 1.2: Typical Electrical Power Subsystem Block Diagram

2 SUBSYTEMS OF A SATELITTE ELECTRICAL POWER SYSTEM

2.1 Power Generation Subsystem (Solar Arrays)

Spacecraft primarily generate electrical power through photovoltaic (PV) solar panels, which convert sunlight into electricity. PV cells are semiconductor devices that operate on the photovoltaic effect. When photons from sunlight strike the cell, they excite electrons, creating an electron-hole pair. An internal electric field within the cell separates these pairs, driving a current through an external circuit.

Figure 2.1-1 shows the current–voltage and power–voltage characteristics of a solar cell. It is evident from the figure that the solar cell generates its maximum power at a certain voltage.

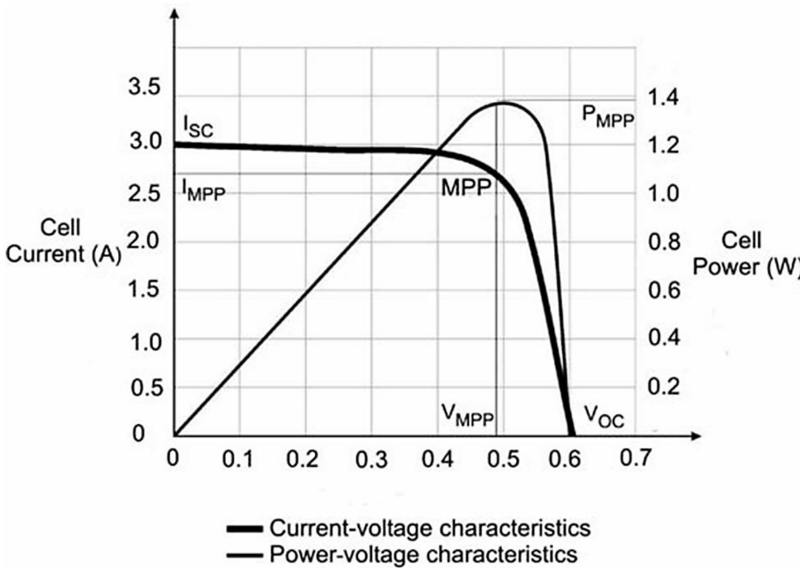

Figure 2.1: Current–voltage and power–voltage characteristics of a solar cell

The Open-Circuit Voltage, V_{OC} is the maximum voltage a cell can produce (when no current is drawn). The Short-Circuit Current, I_{SC} is the maximum current a cell can produce (when the voltage is zero).

The Maximum Power Point (MPP) is the "knee" of the curve where the product of voltage and current ($Power = V \times I$) is at its absolute maximum. This is the optimal operating point for the cell. The goal of an efficient EPS is to force the solar array to always operate at this point. An MPPT tracker is employed to ensure that the solar array's operating point remains at its MPP, which can shift due to varying environmental conditions such as solar irradiance, temperature, and shadowing. MPPT systems typically utilize boost/buck converters controlled by an algorithm to optimize the power delivered to the load, thereby preventing "brown-outs" where the load attempts to draw more power than the PV array can supply. MPPT circuits are generally connected in series between the solar cells and the batteries.

Solar Cell Technologies:

While various technologies exist, the choice for a CubeSat is driven by efficiency (power per unit area).

i. Silicon (Si): Lower efficiency (14-20%), but cost-effective and robust.
ii. Gallium Arsenide (GaAs) - Triple Junction (TJ): The industry standard for CubeSats. These cells have efficiencies of 28-30%+, generating nearly twice the power of Silicon for the same area. This is a critical advantage for a 1U CubeSat with only ~400 cm² of available surface area.

Feature	**Triple-junction GaAs**	**Monocrystalline Silicon**
Efficiency	28–30%	18–22%
Cell Voltage (per cell)	~2.3 V	~0.5–0.6 V
Current (per cell @ AM0)	~120 mA/cm²	~40 mA/cm²
Area per face	~85 cm² usable (per 10×10 cm side)	
Power per face (sun-facing)	~2.5–3.0 W	~1.5–2.0 W

Power (5 sides total in orbit)	~8–12 W	~5–7 W
Radiation tolerance	Very High	Moderate
Degradation	<1%/year	5-10%/year

Table 1: Comparison of Various Solar Cell Technologies

Solar Array Configurations

Body-mounted panels are suitable for low-power CubeSats. Deployable arrays are needed for higher power missions, offering sun-tracking capabilities for optimal energy absorption but requiring complex mechanisms. Modern deployable arrays integrate additional functions (antennas, sensors) and can fold into ultra-thin profiles while delivering high power. For a 1U CubeSat, panels are almost exclusively mounted directly to the 5 or 6 faces of the aluminium chassis. Deployable panels, while offering more power, introduce mechanical complexity and points of failure that are generally unacceptable for this form factor.

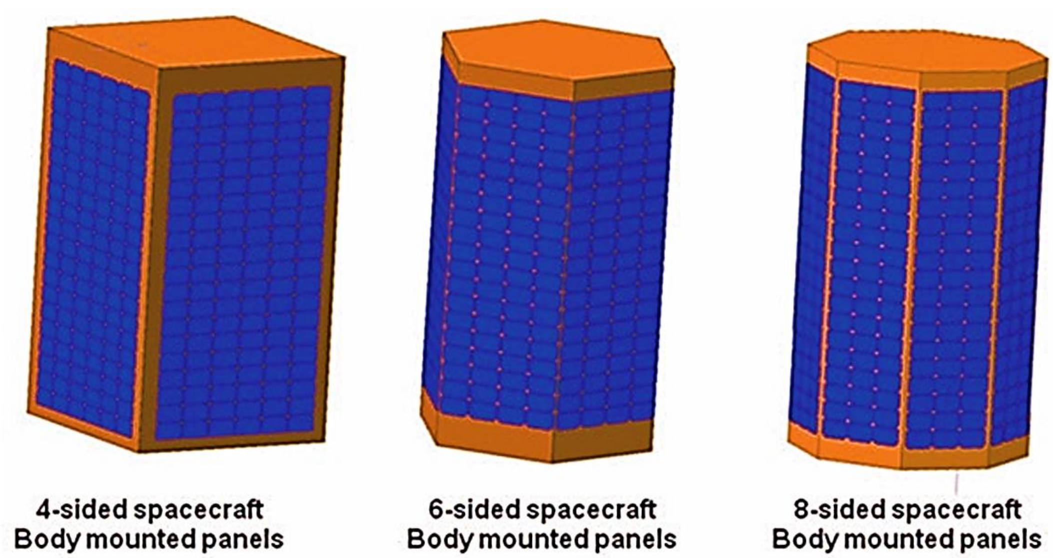

Figure 2.2: Several Body-mounted solar array configurations

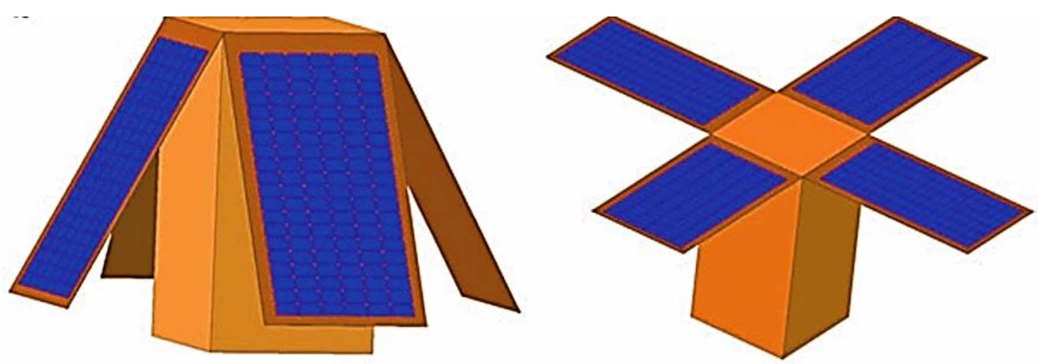

Figure 2.3: Deployable solar array configuration

Each string of solar cells on a panel is connected in series with a **blocking diode**. This is a critical reliability feature. If one panel is in shadow, it stops generating power and can actually become a resistive load, draining power from the illuminated panels. The blocking diode is a one-way gate that prevents this reverse current, isolating the dark panel and preserving the power from the rest of the array.

There are various calculations used in analysis and design of the power generation unit, these are

Available /Incident Solar Power: To estimate the power available from the solar panels, we need to know the solar irradiance at the satellite's orbit and the efficiency of the solar panels.

$$P_{solar} = A_{solar} \times G_{orbit} \times \eta_{solar} \qquad \cdots (2.1)$$

Where:
$P_{solar} = Power\ generated\ by\ the\ solar\ panels\ (W)$
$A_{solar} = Area\ of\ solar\ panels\ (m^2)$
$G_{orbit} = Solar\ irradiance\ at\ orbit\ \left(\frac{W}{m^2}\right)$
$\eta_{solar} = Efficiency\ of\ \ the\ solar\ panels$

Panel Area Calculation: To find the required area of solar panels to meet the power requirements:

$$A_{solar} = \frac{P_{required}}{G_{orbit}\ x\eta_{solar}} \qquad \cdots (2.2)$$

Where:
$P_{required} = required\ power\ for\ the\ satellite\ during\ sunlight\ (W)$
$A_{solar} = Area\ of\ solar\ panels\ (m^2)$
$G_{orbit} = Solar\ irradiance\ at\ orbit\ \left(\frac{W}{m^2}\right)$
$\eta_{solar} = Efficiency\ of\ \ the\ solar\ panels$

Solar Panel Orientation and Exposure Time: To calculate the average solar power depending on the exposure time to the sun.

$$P_{avg} = P_{solar} \times Sunlight\ Fraction \qquad \cdots (2.3)$$

Where:
Sunlight fraction depends on the orbit and orientation. It normally ranges from 0.2 to 1 in high exposure orbits. $P_{avg} = average\ power\ consumption$

Our satellite architecture features three main microcontrollers — the Raspberry Pi 4, ESP32, and MSP430 — each with different processing capabilities and power profiles. These MCUs are supported by sensors, communication modules, and other peripheral components, all contributing to the total power draw.

Subsystem	Controller	Operating Voltage	Peak Current	Peak Power	Average Power	Operational Notes
Onboard Computer (OBC) / Payload	Raspberry Pi 4	5V	2.5 A	12.5 W	3–6 W	High usage during image processing or uplinks
Telemetry	ESP32	3.3V	0.25 A	0.83 W	0.3–0.5 W	Transmits housekeeping and system data
EPS Control	MSP430	3.3V	0.0005 A	0.0016 W	<0.001 W	Always on; very low power watchdog role
Other Peripherals	Sensors, LEDs, etc.	3.3–5V	~0.1–0.2 A	~0.5–1.0 W	~0.3 W	Depends on mission payload and onboard sensors

Table 2: Subsystem Microcontrollers Power and Current draws

Total Peak Power is approximately 13.8 W while the typical operational power is 4.5–7.0 W. The Raspberry Pi 4 is the dominant power consumer, and will only be powered during critical tasks such as payload operations, image processing, or data uplink/downlink. The ESP32 operates semi-continuously, especially for telemetry and simple command/control tasks. The MSP430 is always on, performing real-time EPS supervision and acting as a watchdog or fallback controller. Power cycling the RPi 4 and other subsystems intelligently is critical to conserving energy during eclipse or low battery conditions.

2.2 Energy Storage Subsystem

Batteries are indispensable components within a satellite's EPS, serving to store excess power generated by solar panels during daylight hours and to supply power to the satellite during eclipse periods or when solar panel generation is insufficient to meet peak loads.

Batteries for space applications can be broadly categorized into primary (single-use) and secondary (rechargeable) types. Primary batteries are typically employed for short-duration missions. Silver-Zinc (AgZn) batteries, known for their high energy density and ability to deliver strong current pulses, notably powered critical systems in the Apollo program. Lithium-based primary batteries generally offer even higher energy density.

Secondary, rechargeable batteries are the workhorses for longer-duration missions. Key types include:

- **Lithium-ion (Li-ion):** Currently ubiquitous and considered state-of-the-art for small spacecraft due to their high energy density (150-250 Wh/kg), low weight, and rechargeability.[4] They offer flexibility in power management, exhibit a low self-discharge rate (5-15% per month), and are free from the "memory effect". Their optimal operating temperature range is typically between

20-40°C. Li-ion batteries are widely used in Mars rovers, LEO communication constellations, and Earth observation satellites.

- **Lithium Polymer (Li-Po):** A variation of Li-ion technology that utilizes a polymer electrolyte, offering a flexible form factor that is highly advantageous for small satellites and CubeSats with strict volume constraints. They share many benefits of Li-ion, including high energy density (150-200 Wh/kg) and low self-discharge.
- **Nickel-Hydrogen (NiH2):** Distinguished by exceptional cycle life (2000+ cycles, often exceeding 20,000 with minimal degradation) and high specific power (300-500 W/kg). They were historically used in the ISS and Hubble Telescope, making them ideal for long-design-life GEO satellites. However, they have limitations such as higher self-discharge, lower volumetric energy density, and the requirement for high-pressure hydrogen storage.
- **Nickel-Cadmium (NiCd):** Known for their robustness, reliability, and tolerance to harsh radiation environments and repeated deep discharge cycles.[11] They possess a moderate energy density (50-80 Wh/kg). Their use in new designs has declined due to lower energy density compared to newer technologies and environmental concerns related to cadmium toxicity.

Several key parameters dictate the suitability of a battery for space applications:

i. Energy Density (Wh/kg): The amount of energy stored per unit mass, a critical factor for minimizing launch costs.

ii. Specific Power (W/kg): The rate at which energy can be delivered or absorbed per unit mass.[11]

iii. Cycle Life: The number of full discharge-recharge cycles a battery can undergo before its capacity significantly degrades.

iv. Depth of Discharge (DOD): The percentage of the battery's total capacity that has been depleted. Higher DOD values (e.g., 80-90%) are beneficial in space applications if they do not lead to performance degradation.

v. Operating Temperature Range: The temperature range within which the battery can operate effectively and safely.

vi. Self-Discharge Rate: The rate at which a battery loses its stored charge when not in use.

Cells are combined in series (S) and parallel (P) to create a pack with the desired voltage and capacity. For example, a 2S2P configuration uses two cells in series. This provides a higher bus voltage (~7.4V nominal), which is more efficient because it reduces resistive (I^2R) losses in the wiring for the same amount of power delivered. (More voltage results in less current).

The required energy of the satellite's subsystems E_{tot} can be written as follows:

$$E_{tot} = \frac{E}{\eta} \qquad \cdots (2.4)$$

where E represents the energy obtained from the solar cells, and η is the conversion stage efficiency. Therefore, we can determine the number of batteries to provide the needed amount of energy for the CubeSat's subsystems.

The solar cells harvested energy is given as:

$$E_{sun} = \int_0^{T_{sunlight}} P_{in}(t).\, dt \qquad \cdots (2.5)$$

where $T_{sunlight}$ is the interval in which the CubeSat receives the solar irradiance, and $P_{in}(t)$is the instantaneous received power. Thus, the batteries' energy is the difference between the total subsystems' requirements and the energy generated by the solar cells.

The batteries' energy can be expressed as:

$$E_{batt}^{r} \geq \eta.\left(E_{sun} - \frac{T_{sunlight}}{T_{orbit}} E_{tot}\right) \qquad \cdots (2.6)$$

Where E_{batt}^{r} is the energy required from satellite's batteries and $T_{orbit}$is the rotation period of the CubeSat around the Earth.

As a result, one nominal battery energy can be described as follows:

$$E_{batt} = \frac{U_{max} + U_{min}}{2}.\, C \qquad \cdots (2.7)$$

Where the Li-Ion battery voltage limits are U_{max} and U_{min}, whereas C represents the battery capacity in Ah.

Hence, the needed number of batteries would be:

$$\eta = \frac{E_{batt}^{r}}{E_{batt}} \qquad \cdots (2.8)$$

Other Battery Sizing calculations include the following:

Required battery capacity

The battery capacity should be able to supply the satellite during the eclipse periods.

$$E_{battery} = P_{avg} \times t_{eclipse} \qquad \cdots (2.9)$$

Where:

$E_{battery} = required\ battery\ capacity\ (Wh)$
$P_{avg} = average\ power\ consumption (W)$
$t_{eclipse} = time\ spent\ in\ eclipse$

SoC calculations

This is essential to ensure the battery forever operates in its safe operating conditions and does not discharge.

$$E_{usable} = E_{battery}\ x\ SoC\ factor \quad \cdots (2.10)$$

Where:

$E_{usable} = usable\ battery\ capacity\ after\ accounting\ for\ the\ depth\ of\ discharge\ (DoD)$

Battery Voltage and Size

To find the total number of battery cells is a matter of consideration of the capacity of a cell vs the number of cells that can be housed in the structure. This is a matter of optimization.

$$Number\ of\ Cells = \frac{E_{usable}}{V_{cell} \times C_{cell}} \quad \cdots (2.11)$$

Where:

$V_{cell} = voltage\ per\ battery\ cell\ (V)$

$C_{cell} = capacity\ of\ each\ cell\ (Ah)$

Power Consumption Calculations

This involves calculations of the power consumption of all satellite subsystems. Care should be taken to incorporate all sensors and peripherals power needs to avoid shortage during operations.

$$P_{total} = \sum P_{susbystem} \times\ Duty\ Cycle \quad \cdots (2.12)$$

Where:

$P_{susbystem} = Power\ consumption\ of\ each\ subsytem$

$Duty\ Cycle =\ \ time\ period\ during\ which\ an\ individual\ susbytem\ is\ active$

The total power drawn from the battery should also account for the losses due to factors such as converter efficiency and heat loss. This means that the total power drawn from the battery or solar cells should be slightly higher than the one calculated above if all the subsystems should receive adequate power.

Battery Capacity

$$Q = C \times V \quad \cdots (2.13)$$

Where:

$Q = charge\ capacity\ (Ah)$

$C = charge\ (C)$

$V = voltage\ (V)$

Charging time

$$t_{charging} = \frac{Q}{I} \quad \cdots (2.14)$$

Where: $Q = charge\ capacity\ (Ah)$, $t_{charging} = charging\ time\ (h)$, $I = charging\ current\ (A)$

Discharging time

$$t_{discharging} = \frac{Q}{I} \quad \cdots (2.15)$$

Where:
$Q = battery\ capacity\ (Ah)$
$t_{discharging} = discharging\ time\ (h)$
$I = discharging\ current\ (A)$

2.3 The Power Conditioning and Distribution Unit

The Power Conditioning and Distribution Unit (PCDU) serves as the central intelligence of the satellite's Electrical Power System (EPS), controlling the flow of power from solar arrays and batteries to all onboard subsystems. Its primary role is to ensure that every component receives clean, stable, and well-regulated power, regardless of fluctuations in generation or demand. Given that the entire spacecraft depends on the PCDU for reliable energy distribution, its design directly influences the satellite's operational efficiency, thermal management, and overall mission success.

At its core, the PCDU consists of two key subsystems: the **Power Conditioning Unit (PCU)** and the **Power Distribution Unit (PDU)**. The PCU is responsible for stabilizing raw power from solar panels and batteries, compensating for voltage variations caused by changing sunlight conditions or battery discharge cycles. Meanwhile, the PDU acts as a power router, directing conditioned electricity to different subsystems while monitoring for faults and managing load priorities.

The PCDU Consists of the following key components:

2.3.1 Maximum Power Point Tracking (MPPT) Charge Controller

Solar arrays do not always operate at peak efficiency due to varying illumination and temperature conditions. The MPPT controller addresses this by continuously adjusting the array's operating voltage to extract the maximum available power. Using algorithms like *Perturb and Observe (P&O)*, it makes small voltage adjustments, measures the resulting power output, and fine-tunes its approach to stay at the optimal point. This dynamic adjustment can improve energy harvesting by 20-40% compared to a simple direct connection, making it indispensable for power-constrained missions.

2.3.2 DC/DC Voltage Regulators

Different satellite subsystems require specific voltage levels—microprocessors might need 5V, while sensors and actuators could require 3.3V or higher. DC/DC converters, particularly buck (step-down) regulators, efficiently transform the unregulated battery bus voltage into these precise levels. Efficiency is critical here; a converter operating at 95% efficiency wastes only 5% of input power as heat, whereas an 85% efficient unit dissipates 15%, increasing thermal load. Designers must balance switching frequency—higher frequencies allow for smaller components but may reduce efficiency due to switching losses.

2.3.3 Intelligent Power Distribution Switches

Rather than relying on traditional fuses, modern PCDUs use solid-state e-fuses or load switches, which offer several advantages:

i. Resettable Protection: If a subsystem draws excessive current, the switch can be reset via command instead of requiring physical replacement.

ii. Current Limiting: Prevents damage by capping current at safe levels.

iii. Real-Time Telemetry: Provides feedback on power consumption, aiding in system health monitoring.

iv. Fault Isolation: A short circuit in one payload won't disrupt power to others, enhancing overall reliability.

2.3.4 Design Trade-offs and Considerations

Centralized vs. Distributed Architecture
Small satellites, such as 1U CubeSats, often use a centralized PCDU, where all power conversion and switching occur on a single board. This approach simplifies design and saves space. Larger spacecraft may adopt a distributed architecture, where power is distributed at a high voltage and local point-of-load (POL) converters near each subsystem fine-tune the voltage. This improves electrical performance but adds complexity and mass.

Bus Voltage Selection
Higher bus voltages (e.g. 5V and 3.3V) reduce current for the same power level, minimizing resistive losses and allowing thinner, lighter wiring. However, they require more robust insulation and safety measures. Smaller satellites typically use 28V buses, while high-power systems—such as those with electric propulsion—may employ specialized Power Processing Units (PPUs) to step up voltage for thrusters.

Redundancy and Fault Protection
To ensure uninterrupted operation, critical PCDU components often feature hot redundancy, meaning backup units operate in parallel and take over seamlessly if a failure occurs. Latching Current Limiters (LCLs) provide robust overcurrent protection, isolating faults before they cascade. Galvanic isolation is also used in sensitive applications to prevent noise and electromagnetic interference from affecting payloads.

Operational Functions

Beyond basic power routing, the PCDU performs several vital tasks:

i. Voltage Regulation: Adjusts power levels using DC/DC converters, with buck, boost, or linear regulators selected based on efficiency and noise requirements.

ii. Dynamic Power Allocation: During periods of low power availability, non-essential systems may be temporarily shut down to prioritize critical functions.

iii. Current Monitoring: Continuously tracks power usage to prevent overloads and optimize energy budgets.

iv. Noise Filtering: Ensures clean power delivery to sensitive electronics by suppressing electrical noise and harmonics.

Power Bus Design

The physical power distribution network must be carefully engineered to minimize losses and ensure reliability:

- **Current Handling:** Wiring and PCB traces must support peak current demands with sufficient margin.
- **Voltage Drop Mitigation:** Keeping power paths short and using thicker conductors helps maintain stable voltage levels.
- **Redundancy:** Backup routes ensure power delivery even if a primary connection fails.

2.4 Battery Management System (BMS)

A Battery Management System (BMS) is an intelligent electronic device crucial for monitoring and managing the performance, use, and safety of a battery pack, particularly for lithium-ion batteries.Its primary role is to ensure the safe and optimal operation of the battery throughout its lifecycle. The BMS continuously monitors critical parameters such as the temperature, voltage, State of Health (SOH), and State of Charge (SOC) of each cell within the battery pack.

Modern BMS are capable of performing complex tasks beyond basic protection and monitoring. Key features often include:

i. Battery Monitoring: Continuous tracking of voltage, current, temperature, SOC, and SOH for each cell.

ii. Battery Estimations: Algorithms to accurately estimate SOC and SOH, crucial for mission planning and power budgeting.

iii. Battery Protection: Safeguards against overcharging, over-discharging, overcurrent, short circuits, and thermal runaway. This often includes hardware and software overcurrent protection for each power output channel.

iv. Battery Balancing Techniques: Active or passive cell balancing to ensure uniform voltage and charge across all cells in a pack, preventing unbalanced cell conditions and maximizing usable capacity.

v. Diagnostics and Prognostics: Advanced capabilities to diagnose faults and predict remaining useful life, enabling proactive maintenance or operational adjustments.

vi. Temperature Management: Integrated battery heaters and control functions to maintain the battery within its optimal operating temperature range, especially critical for Li-ion batteries.

vii. Configurable Slew Rate: Support for configurable slew rates on power outputs to handle inrush currents of various loads.

viii. Communication Interfaces: Support for standard communication protocols (e.g., CAN, UART/RS422) for interfacing with the On-Board Computer (OBC) and ground control.

ix. Integrated Charger Circuit: Simplifies integration operations by providing a built-in charging capability.

x. Undervoltage Lockout: Protection by a main switch to prevent battery discharge below a safe minimum voltage.

3 THE EPS DESIGN PROCESS

A systematic process is the best approach for designing the EPS for a 1U CubeSat majorly driven by the mission requirements. The goal is to create a system that is perfectly balanced, where power generation, storage, and consumption are in harmony over the entire mission lifetime. The design process starts by estimating payload power requirements, sizing the solar panels to meet average power needs (with margin for eclipse), and sizing the batteries to supply worst-case eclipse loads. The following steps outline this systematic process.

3.1 Mission Analysis and Requirements Definition

Analyse the Orbit

The mission's orbital parameters are the single most important input. Using tools like Systems Tool

Kit (STK), the designer determines:

i. Orbit Period: The total time for one revolution (e.g., ~95 minutes for a 500 km orbit).
ii. Sunlight Duration (T_d): The portion of the orbit where the solar panels can generate power.
iii. Eclipse Duration (T_e): The portion of the orbit where the satellite is in Earth's shadow and must run on battery power.

Define the Mission Lifetime

This involves defining how long the mission should take. This dictates the total number of charge-discharge cycles the battery must endure and the total radiation exposure the solar cells will receive. This is crucial for calculating component degradation.

Characterize the Loads

Every component that consumes power is a "load." The designer must create a comprehensive list detailing:

i. Voltage Requirement: Does it need 3.3V, 5V, or can it run on the unregulated battery bus?
ii. Power Consumption: How many watts does it draw when active?
iii. Duty Cycle: What percentage of an orbit will this component be active? A radio transmitter might have a very high-power draw but only a 5% duty cycle, while the OBC has a low power draw but a 100% duty cycle.

3.2 Construct the Power and Energy Budgets

With the requirements defined, the next step is to perform the core analysis. This is typically done in a detailed spreadsheet.

Create Operational Modes

A satellite doesn't do the same thing all the time. The designer must define different operational modes and calculate the total power consumption for each. Examples include:

i. Safe Mode: The bare minimum power to keep the satellite alive (OBC on, receiver listening).
ii. Nominal Mode: Standard operations (e.g., ADCS active, sensors collecting data).
iii. High-Power Mode: A specific, power-intensive task (e.g., transmitting data to a ground
iv. station).

Calculate Orbit Average Power (OAP)

For the most demanding but sustainable operational mode, the OAP is calculated. This represents the average power the EPS must provide over a full orbit.

$$OAP_{load} = \sum P_{component} \times Duty\ Cycle_{component} \qquad \cdots (3.1)$$

This sum is taken over all components.

Apply a Design Margin

No model is perfect. To account for uncertainties, component variations, and unforeseen operational needs, a healthy design margin of 20-30% is added to the calculated OAP load. This becomes the target generation requirement for the solar array.

3.3 Size The Solar Arrays

The solar array must be sized to provide the target OAP while also recharging the energy used from the battery during the eclipse. Its performance is calculated at its End-of-Life (EOL).

Calculate Required Energy Generation

The total energy needed per orbit is the sum of the energy used by loads during sunlight and the energy needed to replenish the battery for what was used during eclipse. A simplified but effective equation is provided below.

$$Solar\ Array\ Power, P_{SA} = \frac{P_d}{\eta_d} + \left(P_e \times \frac{T_e}{\eta_e}\right) \qquad \cdots (3.2)$$

Where:

P_d, P_e is the average power consumed in daylight and eclipse.

T_d, T_e= Duration of daylight and eclipse.

η_d= Efficiency of the power path from array to load (~95% for MPPT).

η_e= Efficiency of the charging and discharging path ($\eta_{charge} \times \eta_{discharge}$, ~85-90%).

Account for Degradation

Solar cell efficiency degrades over time. The design must be based on the EOL power.

$$End\ of\ Life\ Power, P_{EOL} = P_{BOL} \times L_d \qquad \cdots (3.3)$$

Where L_d is the life degradation factor (e.g., 0.97 for 3% degradation per year). The P_{SA} calculated above must be what the array can produce at EOL.

Determine Physical Layout

The designer then checks if the required number of high-efficiency cells can physically fit on the available faces of the 1U CubeSat. If not, the process must be iterated.

3.4 Battery Sizing

The battery must be large enough to survive the longest eclipse without being discharged too deeply, which would shorten its life. Calculate Energy Required for Eclipse

$$Required\ Eclipse\ Energy, C_{req} = \frac{P_e \cdot T_e}{\eta_{discharge}} \qquad \cdots (3.4)$$

Where $\eta_{discharge}$ is the efficiency of the DC/DC converters drawing from the battery (~90-95%). This gives the energy in Watt-hours (Wh) that the battery must deliver during eclipse.

Apply Depth of Discharge (DoD) for Lifetime: To ensure the battery survives thousands of LEO cycles, the C_{req} should only represent a small fraction of its total capacity.

$$Total\ Required\ Battery\ Capacity, C_{total} = \frac{C_{req}}{DoD} \qquad \cdots (3.5)$$

For a long-life LEO mission, a DoD of 20-25% is a standard design goal.

Select a Physical Battery: The designer then finds a commercially available battery pack that meets or exceeds the calculated C total and fits within the allocated mass and volume.

3.5 Iteration and Finalisation

If the required solar array is too large, or the battery is too heavy or voluminous, the design is not viable. The EPS designer must go back to the team and negotiate—"Can the radio transmit for 4 minutes instead of 5? Can we reduce the payload duty cycle?" This iterative loop continues until a balanced and physically achievable design is reached.

4 EPS ARCHITECTURE AND OTHER DESIGN CONSIDERATIONS

The "architecture" refers to the high-level schematic of how the main EPS subsystems are interconnected. The choice of architecture is a fundamental design decision with significant implications for efficiency, complexity, and reliability.

The classification of the state-of-the-art CubeSat EPS architectures is shown in the figure below, which is done based on the following aspects:

a. dc-bus voltage regulation
b. Interface of PV panels
c. Location of power converters
d. Number of conversion stages

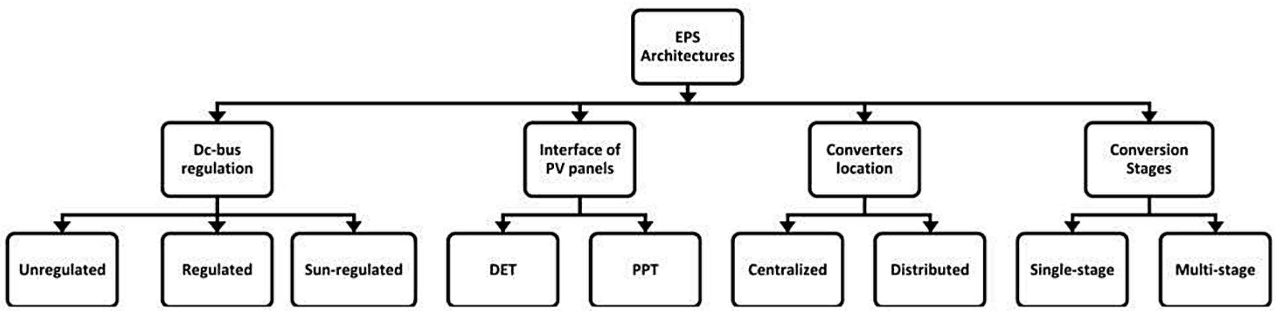

Figure 4.1: Classification of CubeSat EPS Architectures

The following are the most relevant for 1U CubeSat mission:

Key Decision Points for 1U:

4.1 PV Interface (DET Vs PPT)

The PV panels are the main source of energy for the CubeSat and based on their interface, the EPS architectures are categorized into DET and PPT

4.1.1 Direct Energy Transfer (DET)

In this architecture, the PV panels with series diodes are directly connected to storage system and/or loads. It usually has a shunt regulator in parallel to the PV panel to divert the excess PV power when the battery is fully charged or when the load demand is less. The excess power is dissipated as heat inside CubeSat if resistor is used in shunt regulator otherwise it is dissipated on the PV panel. The reliability of shunt regulator is very important otherwise it results in loss of mission. It is simpler, more reliable, fewer components. The PV panel is connected almost directly to the battery bus. However, it is inefficient as it does not operate the panel at its maximum power point.

4.1.2 Peak Power Tracking (PPT)

The architecture utilizes PV panels interfaced with dc-dc converters as shown in the figure below to achieve MPPT over wide range of operating conditions such as solar irradiation, PV panel temperature, and sun inclination angle. On the generation-side, the PV panels on the opposite faces of CubeSat are connected in parallel and they are interfaced with dc-dc converter for maximum power point tracking (MPPT) under wide range of irradiation conditions and battery voltage. The type of dc-dc converter depends on the maximum power point (MPP) voltage of the PV panel and the battery voltage.

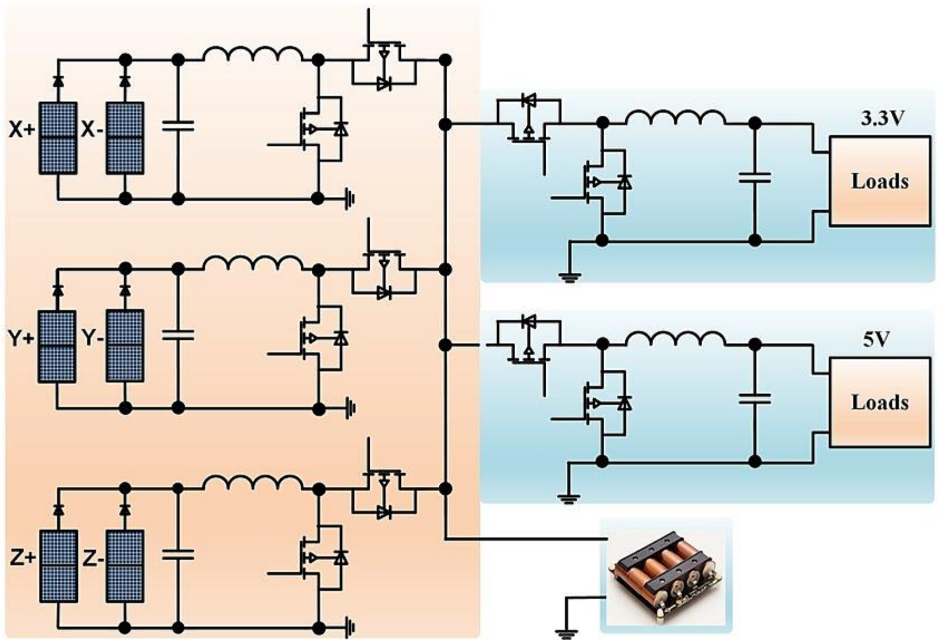

Figure 4.2: Peak Power Tracking (PPT) Architecture

It is widely used in CubeSat designs as they have limited power generation due to shorter sunlit periods and space constraints for using larger PV panels. The MPPT can be achieved by either digital micro-controller (MCU) or analogue controllers. A MCU has advantages of simplicity and flexibility in tuning but it is more susceptible to failure due to radiation damage. The analogue controller with discrete components is considered more robust although not efficient as MCU. The CubeSat may implement analogue controllers as main control or as back-up control to be used in case of MCU failure.

4.2 DC-Bus Regulation (Unregulated Vs Regulated)

The dc-bus acts as intermediate stage between PV panels, energy storage system, and loads. Based on the dc-bus voltage regulation, the EPS architectures can be classified as unregulated dc-bus EPS, regulated dc-bus EPS, and sun-regulated dc bus.

4.2.1 Unregulated Bus

The main power bus that feeds the satellite is the battery terminal. The bus voltage therefore varies with the battery's state of charge (e.g., from 8.4V down to 6.0V for a 2s Li-ion packs). There is no central regulator wasting power. Power flows directly from the battery to the loads. It is also more reliable as there is no single-point-of-failure main regulator. However, all subsystems that require a stable voltage (like the 3.3V OBC) must have their own small, local DC/DC regulators. The efficiency gains are critical, and the requirement for local regulators is a widely accepted and manageable trade-off all which makes it a dominant architecture.

4.2.2 Regulated Bus

A central, main DC/DC converter is placed after the battery to provide a constant bus voltage (e.g., a stable 5V) to the entire satellite. It simplifies the design of other subsystems, as they all receive a clean, stable input voltage. However, it introduces a constant efficiency loss (typically 5-15%) from this main regulator, which is always on. It also represents a critical single point of failure—if the main regulator dies, the entire satellite dies. The efficiency penalty and reliability risk are generally considered too high for the benefit of a stable bus.

4.2.3 Sun-regulated Bus

In sun-regulated dc-bus architecture, the dc-bus voltage is regulated to reference value only during the sunlit period and during the eclipse period. The battery connects to the dc-bus via diode. It is also referred as partial regulated dc-bus or quasi regulated dc-bus.

4.3 Converter Location (Centralised Vs Distributed)

Based on the location of power converters, the EPS architectures are categorized as centralized/concentrated and decentralized/distributed architectures.

4.3.1 Centralized

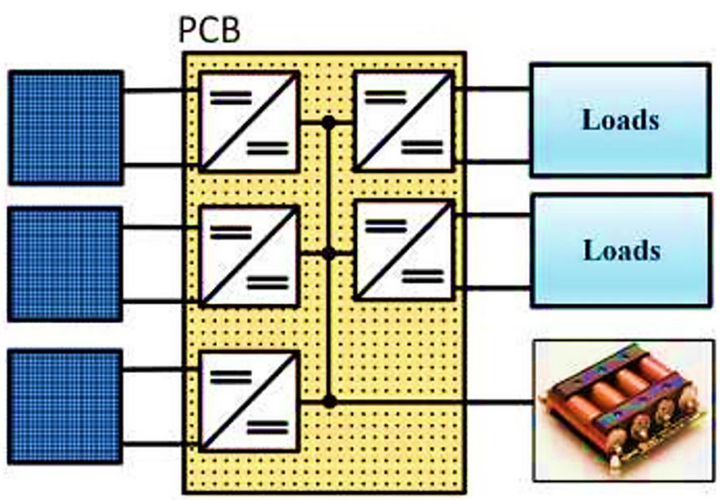

Figure 4.3: Centralised

In the centralized architecture as shown in figure 4.3.1, all the power converters along with the controllers are placed on a single printed circuit board (PCB) which connects to the PV modules, storage system, payloads, and subsystems through specific voltage rails. It has been widely used in CubeSats due to simplicity, physical space efficiency, and several COTS EPS designs. In this architecture, fewer voltage regulators are required because multiple payloads and subsystems use same voltage rail. One main disadvantage is that the voltage regulators must be designed for peak load demand and hence, the converter operates at lower efficiency for most of the time. Another disadvantage is lower reliability as the failure of one converter affects multiple subsystems.

4.3.2 Distributed

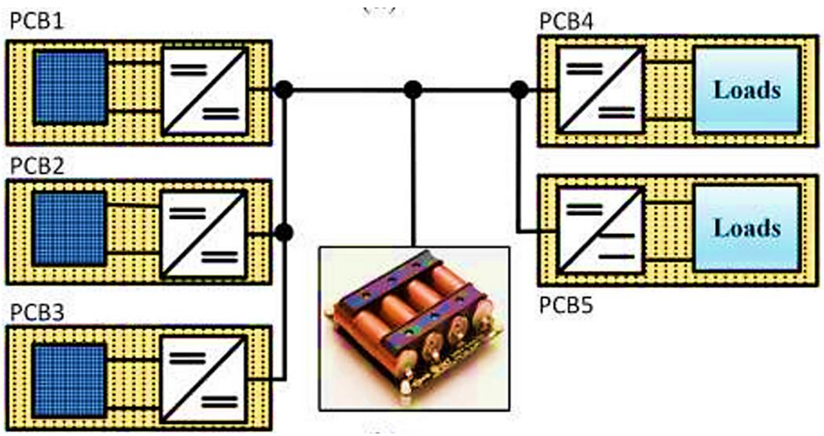

Figure 4.4: Distributed

The distributed architecture has dc-bus supplied throughout the system and the power converters are placed close to individual subsystems and in some designs, MPPT converters are placed close to PV panels. It utilizes several PCBs in the entire design and is commonly used in bigger satellites but has not become popular in CubeSats due to higher number of dedicated power converters.

4.4 EPS–OBC Interface: Communication Protocols and Operational Interactions

The interface between the Electrical Power System (EPS) and the Onboard Computer (OBC) is critical for spacecraft operation, enabling power monitoring, command execution, and safety management. This interaction relies on standardized communication protocols, each with distinct advantages depending on mission requirements.

Common Communication Interfaces

1. **I²C (Inter-Integrated Circuit)**

Advantages: Low pin count, simple implementation, widely supported.

Limitations: Limited speed and range; susceptible to noise in long-distance connections.

Best for: Small satellites (e.g., CubeSats) where simplicity and minimal wiring are priorities.

2. **CAN Bus (Controller Area Network)**

Advantages: Robust error detection and fault tolerance, suitable for noisy environments.

Limitations: Higher complexity than I²C or UART.

Best for: Larger spacecraft or missions requiring high reliability.

3. **UART/RS-485**

Advantages: Simple, easy to implement, supports moderate distances.

Limitations: No built-in error checking; requires software-level protocol management.

Best for: Basic telemetry and command exchange where speed isn't critical.

4. **SPI (Serial Peripheral Interface)**

Advantages: High-speed, full-duplex communication.

Limitations: Not ideal for multi-device buses due to lack of built-in addressing.

Best for: High-bandwidth data transfers between EPS and OBC when minimal devices are involved.

Function	How It Works	When It's Used
Power Monitoring	The EPS transmits real-time telemetry (voltage, current, battery state of charge) to the OBC via I²C or UART.	Periodic updates (e.g., every 5–10 seconds) for health monitoring and power budgeting.
Command Execution	The OBC sends control commands (e.g., enable/disable power rails, reset subsystems) to the EPS.	During fault recovery, mode transitions (e.g., switching from safe mode to science mode), or scheduled operations.
Safety Mechanisms	The EPS autonomously disconnects power from a subsystem if an overcurrent or short circuit is detected, often triggering an interrupt to alert the OBC.	Instantaneous response to hardware faults to prevent damage.
Wake-Up Events	The EPS wakes the OBC from low-power mode based on predefined triggers (e.g., eclipse exit, scheduled transmission window).	Used in power-saving modes to conserve energy during eclipses or standby periods.

Table 3: EPS-OBC Functional Interactions

Key Design Considerations

Protocol Selection:

I²C or UART are sufficient for small satellites with basic telemetry needs. CAN Bus is preferred in larger missions where noise immunity and redundancy are critical. SPI may be used for high-speed sensor data but is less common in EPS-OBC communication.

Error Handling & Redundancy:

CAN Bus inherently supports error detection and retransmission. I²C/UART implementations may require software-based checksums or acknowledgment protocols.

Power Efficiency: Low-power communication modes (e.g., partial sleep with UART wake-up) help conserve energy.

Interrupt-Driven vs. Polling: Critical events (e.g., overcurrent faults) should trigger hardware interrupts for immediate response. Routine telemetry can be polled at fixed intervals.

4.5 Thermal Considerations

The solar panels, batteries and even other components inevitably produce heat. Thermal management systems are essential in this case. Due to the limited size of the 1u satellite, passive thermal control systems are necessary to maintain battery and component temperatures within safe operating limits. Such methods include thermal coatings and heat pipes.

It is also important to note that during eclipse periods, as the satellite is not exposed to the sun as much, rapid cooling will be experienced. The cooling mechanisms should therefore be sensitive to such changes to avoid exposing the satellite to extreme temperatures past its range.

4.6 Monitoring, Redundancy, Protection and Radiation Effects

It is important to constantly monitor the state of the EPS especially in terms of the battery's charge level, temperature and health. This is to prevent it from over discharging or overheating. This means that sensors have to be integrated to provide real time telemetry data for monitoring.

Redundancy

To minimize the risk of a failure causing the whole system to shut down, there should be redundant sources of power like backup power converters, backup solar arrays and backup batteries as the space allows. Mechanisms could also be employed to power down non-critical systems during low sunlight or eclipse modes.

Protection

Systems should be put in place to effectively detect faults in real time and prevent cascading failure in the face of power faults like over voltages, over currents and short circuits. Such entail inclusion of (resettable) fuses and circuit breakers.

Radiation effects

Space radiation can degrade the performance of solar panels and batteries, which needs to be accounted for in the EPS design.

5 RECOMMENDATION

Assuming a 1U CubeSat (typically 6 solar panels, one per face), we calculate:

During Sunlight (Peak + Battery Charging)

- Peak Power Needed: 13.8W (worst-case scenario).
- Average Power Used: 7W (typical operation).

- Battery Charging Requirement: If the satellite operates at 7W but must also recharge the battery for eclipse, we need extra power.

During Eclipse (Battery-Powered)

- Typical Low Earth Orbit (LEO) Eclipse Duration: ~35% of orbit (~30 mins eclipse per 90-min orbit).
- Energy Needed for Eclipse:

$$Eclipse\ Energy = 7W \times 0.5h = 3.5Wh$$

- Solar Panels Must Recharge This in Sunlight (60 mins):

$$Recharge\ Power = \frac{3.5Wh}{1h} = 3.5W$$

- Total Power Needed in Sunlight:

$$7W(runtime) + 3.5W(recharge) = 10.5W$$

Solar Panel Sizing per Face

Assumptions:

i. Solar Cell Efficiency: ~28% (typical for high-efficiency triple-junction GaAs cells).
ii. Orbital Beta Angle & Sunlight Angle: ~70% effective illumination due to non-ideal sun angles.
iii. Degradation & Losses: ~20% loss over mission lifetime (radiation, thermal cycling).

Required Power Before Losses:

$$Required\ Power\ (after\ losses)\ = 13.8W\ (peak)\ or\ 10.5W\ (average)$$

$$Actual\ Needed\ Power = \frac{10.5W}{0.70(sun\ angle) \times 0.8(degrdation)} \approx 18.75W$$

Power per Solar Panel (6 faces):

$$\frac{18.75W}{6} \approx 3.1Wperpanel$$

Physical Size of Solar Panels:

- Typical GaAs Solar Cell Power Density: ~300W/m² (AM0 space conditions).
- Required Area per Panel:

$$\frac{3.1W}{300W/m^2} = 0.0103m^2$$

- For a 1U CubeSat (10cm × 10cm faces):
  - If using body-mounted panels, each face can generate 3.1W.
  - If using deployable panels, power can be much higher (e.g., 5–10W per panel).

5.1 Power Generation: Solar Arrays

For optimal power generation, deployable triple-junction Gallium Arsenide (GaAs) solar arrays are recommended. A fused silica cover glass should be applied to minimize radiation-induced degradation (<1% per year).

GaAs has:

i. High efficiency (30–39%), outperforming traditional silicon cells.
ii. Superior radiation resistance, crucial for long-term missions.
iii. Deployable configuration maximizes power output (~10–12W for a 1U CubeSat).

Configuration:

- 3–6 panels
- Electrical arrangement: 3 cells in series per string (3S1P) to boost voltage.
- MPPT (Maximum Power Point Tracking) required to compensate for shadowing and thermal variations.

5.2 DC-DC Converters (Power Conditioning)

A synchronous buck/boost DC-DC converter is ideal for efficient power regulation.

Key Benefits:

- Converts variable solar input to stable bus voltages (3.3V, 5V, or 12V).
- Convertors with Gallium Nitride (GaN) technology improves efficiency, reduces size, and minimizes switching losses.
- Supports MPPT algorithms (Perturb & Observe or Incremental Conductance).

In this design, two main voltage rails are derived from the battery:

- 5V for the Raspberry Pi 4 and other potential USB devices
- 3.3V for the ESP32, MSP430, sensors, and control logic

Use of a buck converter for 5V to power the Raspberry Pi 4 and an LDO or low-current buck for the 3.3V line (ESP32, MSP430) were considered.

Some of the suggested ICs for regulation are;

Function	**Recommended ICs**	**Notes**
5V Buck	MP2307, TPS5430, LM2596	Supports up to 3 A, high efficiency
3.3V LDO	AMS1117, MCP1700	Simple, cost-effective for low power

3.3V Buck	RT8059, AP63203	Use if LDO heat is an issue

Table 4: Suggested ICs for voltage regulation

To monitor rail health, voltage dividers are connected to MSP430 or ESP32 ADCs as well as enabling early shutdown of non-critical rails if voltage drops (e.g., battery <6.8 V)

5.3 Energy Storage

Lithium-ion (Li-ion) or Lithium Polymer (Li-Po) batteries in a 2S2P configuration (7.4V nominal, 4 cells total) provide the best balance of energy density and reliability.

Advantages:

- High energy density (150–250 Wh/kg).
- Low self-discharge and long cycle life (~500–2000 cycles).
- Compact form factor, fitting CubeSat volume constraints.

Battery Management System (BMS) Requirements:

- State of Charge (SOC) & State of Health (SOH) monitoring.
- Cell balancing & overvoltage/current protection.
- **Thermal monitoring + heater** for extreme conditions.

5.4 Current & Voltage Sensing

Hall-effect or shunt-based sensors (e.g., Allegro ACS7xx, INA219/INA260) with I²C/SPI/analogue outputs ensure accurate power monitoring. This is essential for MPPT, power budgeting, and fault detection. Hall-effect sensors provide isolation and low-power operation.

5.5 Architecture: EPS-6 (Centralised, Unregulated PPT)

This is the de-facto standard for commercial off-the-shelf (COTS) 1U EPS boards and has extensive flight heritage (GOMspace, Nano-avionics, etc.). The EPS-6 architecture is as shown below:

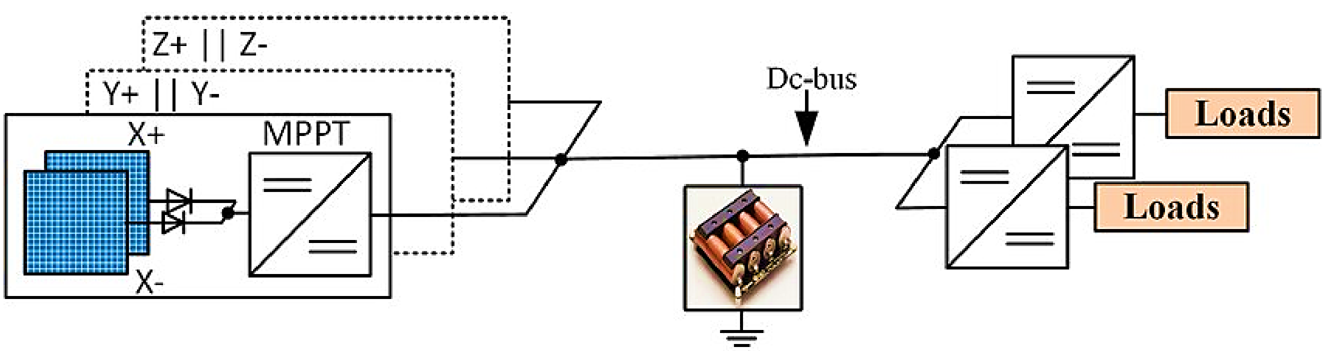

Figure 5.1: EPS-6 (Centralised, Unregulated PPT)

PV Interface: PV panels are connected to dedicated MPPT converters. Often, panels on opposite faces (e.g., +X and -X) are connected to a single MPPT converter.

Bus: The outputs of the MPPT converters are connected directly to the battery, creating an unregulated bus.

Operation: During sunlit periods, the MPPT converters charge the battery and supply power to the loads. If the battery is full, the MPPT converter throttles back. During eclipse, the battery directly powers the bus.

Distribution: Point-of-Load (POL) converters tap off this unregulated bus to provide stable 3.3V and 5V rails to other subsystems.

Why it's ideal for 1U:

i. Efficiency: PPT maximizes power from the small solar panels. The unregulated bus
ii. minimizes primary power conversion losses.
iii. Simplicity & Volume: The centralized design is compact and fits onto a single PC/104-form
iv. factor board.
v. Reliability: It's a proven design with significant flight heritage.

ALTERNATIVE ARCHITECTURE: EPS-1 (CENTRALIZED, UNREGULATED DET)

In case of EPS-1 shown in the figure below, the output of PV panels is connected to battery in parallel with shunt regulator and the load-side converters for further conversion.

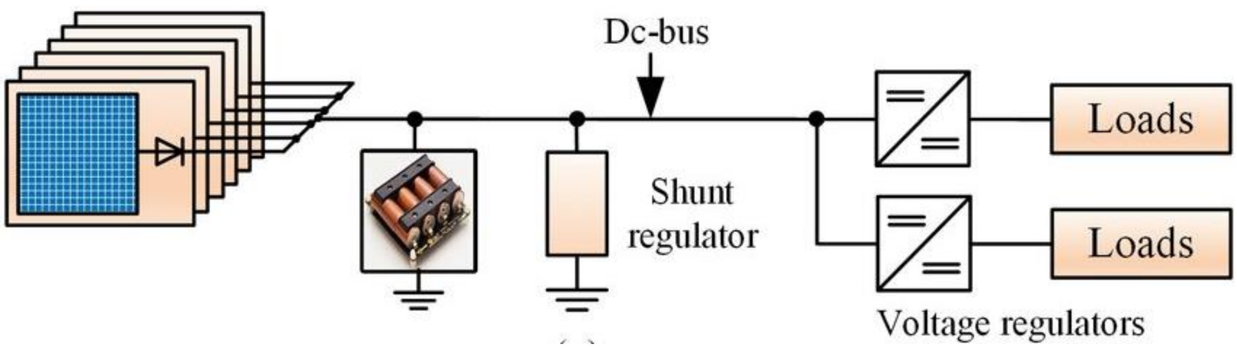

Figure 5.2: EPS-1 Architecture (Centralised, Unregulated DET

The PV panel's output voltage is clamped to floating battery voltage and its output current depends on the I-V characteristics curve. The battery feeds to loads directly during the eclipse period and it has protection system to avoid over-current, over-voltage, and under-voltage conditions. It has higher conversion efficiency due to power conversion by just one dc-dc converter before feeding the load demand.

In the use Case for a 1U CubeSat this architecture is only suitable for missions with very low power requirements where simplicity and reliability are valued far more than power efficiency. Its lower component count is an advantage, but the power loss from not using MPPT is a significant drawback for most 1U missions.

6 CONCLUSION

The design of the Electrical Power System (EPS) for a 1U CubeSat is a complex and critical task that demands careful consideration of power generation, storage, conditioning, and distribution. The EPS serves as the lifeline of the satellite, ensuring uninterrupted operation throughout its mission duration. This document has systematically outlined the key subsystems of the EPS, including solar arrays for power generation, batteries for energy storage, and the Power Conditioning and Distribution Unit (PCDU) for efficient power management.

The recommended EPS-6 architecture, featuring a centralized, unregulated bus with Peak Power Tracking (PPT), emerges as the optimal choice for a 1U CubeSat. This architecture maximizes power extraction from the limited solar array area, ensures high efficiency, and maintains reliability—a proven design with significant flight heritage. Triple-junction Gallium Arsenide (GaAs) solar cells are preferred for their high efficiency and radiation resistance, while Lithium-ion (Li-ion) batteries in a 2S2P configuration offer the best balance of energy density and cycle life. The inclusion of a robust Battery Management System (BMS) and intelligent power distribution switches further enhances the system's reliability and fault tolerance.

Key design considerations such as thermal management, radiation effects, redundancy, and real-time monitoring are essential to address the harsh space environment. Passive thermal control methods and radiation-hardened components help mitigate these challenges. The EPS-OBC interface, leveraging communication protocols like I^2C or CAN Bus, ensures seamless coordination between power management and onboard computing.

In summary, the proposed EPS design for the 1U CubeSat is a balanced, efficient, and reliable system tailored to meet the mission's power demands while adhering to strict size, mass, and volume constraints. By leveraging advanced technologies like GaAs solar cells, MPPT controllers, and Li-ion batteries, the EPS ensures the satellite's operational success throughout its mission lifetime.