ELECTRICAL POWER SYSTEMS FOR CUBESATS

A Seminar Report

Submitted to APJ Abdul Kalam Technological University in partial.

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in

Electrical and Electronics Engineering

By

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CERTIFICATE

This is to certify that the report entitled "**ELECTRICAL POWER SYSTEMS FOR CUBESATS**" submitted by **HARIKRISHNA PRAKASH (LKGR22EE019)** to the APJ Abdul Kalam Technological University in partial fulfilment of the B.Tech. degree in Electrical and Electronics Engineering is a Bonafide record of the seminar carried out by him under our guidance and supervision. This report in any form has not been submitted to any other University or Institute for any purpose.

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DECLARATION

I ***HARIKRISHNA PRAKASH*** hereby declare that the seminar report "***ELECTRICAL POWER SYSTEMS FOR CUBESATS***", submitted for the partial fulfilment of the requirements for the award of degree of Bachelor of Technology of APJ Abdul Kalam Technological University, Kerala is a Bonafide work done by me under the supervision of Prof. ***DARSANA BABU***.

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ABSTRACT

ELECTRICAL POWER SYSTEMS FOR CUBESATS

The growing demand for miniaturized satellite missions has led to a surge in the development and deployment of CubeSats compact, cost-effective spacecraft used for a variety of space applications such as remote sensing, scientific exploration, and communication. A crucial subsystem in these satellites is the Electrical Power System (EPS), responsible for efficient generation, storage, conditioning, and delivery of power under the harsh conditions of space. This seminar explores the design and architecture of EPS tailored for CubeSats, emphasizing fault tolerance, efficiency, and low size, weight, and power requirements. Key topics include solar energy generation using advanced triple-junction photovoltaic cells, lithium-ion battery storage solutions, thermal management strategies, and protective schemes like Latch-Up Current Limiter (LCL) and Undervoltage Protection (UVP). Additionally, techniques such as Maximum Power Point Tracking (MPPT) and optimal sizing of power components are discussed to ensure reliable CubeSat operations. The presentation concludes by highlighting emerging trends, including AI-based health monitoring and novel energy storage technologies, positioning EPS design at the forefront of next generation space microgrids.

Keywords: CubeSats, EPS, Triple-junction cells, Li-ion batteries, thermal control, LCL, UVP, MPPT.

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- **ADCS** — Attitude Determination and Control System
- **AoI** — Angle of Incidence
- **BMS** — Battery Management System
- **CC-CV** — Constant Current — Constant Voltage (charge method)
- **DoD** — Depth of Discharge
- **EPS** — Electrical Power System
- **EOL** — End of Life
- **GaN** — Gallium Nitride
- **I-V** — Current–Voltage
- **LEO** — Low Earth Orbit
- **LCL** — Latch-up Current Limiter
- **LED** — Light Emitting Diode (if used in tests/telemetry)
- **MLI** — Multi-Layer Insulation
- **MPPT** — Maximum Power Point Tracking
- **MSFC** — Marshall Space Flight Centre (NASA)
- **MTBF** — Mean Time Between Failures (general reliability term)
- **MTTR** — Mean Time to Recover (general operations term)
- **P-POD** — Poly-Picosatellite Orbital Deployer
- **PV** — Photovoltaic
- **SoC** — State of Charge
- **SoH** — State of Health
- **TVAC** — Thermal Vacuum (testing)
- **UVP** — Undervoltage Protection
- **Vmp** — Voltage at maximum power point
- **Voc** — Open-circuit voltage

CHAPTER 1

INTRODUCTION

Satellites are sophisticated systems that rely heavily on a continuous supply of electrical power to perform their functions. Every subsystem inside a satellite, whether it is related to communication, navigation, payload operation, data handling, or attitude control, requires electrical energy to operate. The Electrical Power System (EPS) is therefore one of the most critical subsystems, and its failure inevitably results in the complete loss of the mission. While large satellites have the advantage of sufficient space and resources to accommodate redundant and robust power systems, CubeSats are bound by strict constraints of size, weight, and cost. Designing an efficient and reliable EPS for CubeSats is thus a unique challenge that requires innovative solutions.

CubeSats are nanosatellites that are built according to a standardized unit structure. A single CubeSat unit, also known as 1U, measures ten centimetres on each side and weighs approximately 1.33 kilograms. Larger configurations are possible by combining these units into 2U, 3U, 6U, or even 12U CubeSats depending on the needs of the mission. Because CubeSats are small, inexpensive, and relatively quick to build, they have enabled universities, research institutions, and small companies to enter the space sector. Their applications have expanded beyond educational purposes and now include Earth observation, scientific research, interplanetary exploration, communication, and defence.

The EPS of CubeSats generally consists of solar panels for power generation, batteries for storage, converters and regulators for conditioning, and protection circuits for safety. The limited surface area available for solar panels and the restricted mass budget for batteries create a situation in which every watt of power is valuable. EPS designers must optimize energy generation and storage while ensuring reliability in the harsh environment of space. This report systematically analyses all aspects of CubeSat EPS, beginning with the classification of satellites, moving through CubeSat basics and history, and covering power generation, battery technologies, thermal management, protection, sizing, and finally the challenges and future trends in this field.

CHAPTER 2

CLASSIFICATION OF SATELLITES

Satellites are categorized by mass, mission complexity, and development cadence, and this classification determines subsystem sizing, launch options, and acceptable mission risk. Large satellites exceed 1000 kg and support high-capability payloads with multi-year development schedules, medium satellites range from 500 to 1000 kg for sophisticated missions, mini satellites fall between 100 and 500 kg, micro satellites are under 100 kg and often serve universities and small commercial teams, nanosatellites weigh 1–10 kg and include CubeSats built from standardized 1U units, pico and femto satellites are smaller than 1 kg and carry highly constrained payloads. CubeSats are modular units of 10 × 10 × 10 cm and roughly 1.33 kg per 1U, with common configurations such as 1U, 3U, 6U, and 12U that enable trade-offs between payload capability and cost. The CubeSat standard prescribes mechanical rails, electrical connectors, and deployment constraints that designers must follow to ensure compatibility with deployers and launch providers. Limited surface area and strict mass budgets force CubeSat designers to prioritize high-efficiency solar cells, compact power electronics, and careful thermal and structural integration. Typical bus voltages in CubeSat EPS designs include 3.3 V and higher options selected by balancing converter efficiency, I^2R losses, and component availability. Electrical interfaces such as I2C, SPI, UART, and CAN are commonly used for telemetry and command between subsystems, and power distribution relies on switched outputs, fuses, or per-load protection to isolate faults. Representative CubeSat missions demonstrate how EPS sizing varies with mission type: Earth observation payloads require sustained average power, communications systems have bursty high-power demands, and technology-demonstration satellites often present variable duty cycles that stress control and protection algorithms.

1.2 Classification by mass (macro, mini, micro, nano, Pico)

Satellites are commonly categorized by mass because weight strongly influences launch options, design margins, subsystem choices and mission cost. Macro or large satellites (greater than about 1000 kg) are full-featured platforms that can carry extensive payloads, large solar arrays, heavy shielding and multiple redundant subsystems; their Electrical Power Systems (EPS) are correspondingly robust, with large-capacity batteries, multi-kW arrays and active thermal control, allowing extensive redundancy and long operational lifetimes. Minisatellites (≈100–500 kg) and microsatellites (≈10–100 kg) occupy the middle ground: they permit more capable instruments than nanosatellites while still demanding careful trade-offs between power, mass and volume; EPS design for these classes often balances moderate array area, more sophisticated BMS and some redundancy without the mass penalty of large platforms. Nanosatellites (≈1–10 kg), including CubeSats built from 1U, 2U, 3U modules, impose strict size, weight and power (SWaP) limits that force EPS designers to maximize energy per unit area and mass—using high-efficiency cells, compact Li-ion packs, low-loss converters, conservative depth-of-discharge policies and minimal but telemetry-enabled protection. Picosatellites (<1 kg) represent extreme miniaturization where available energy is tiny and mission functionality is severely constrained; EPS solutions here are often single-string, ultra-low-power designs using small cells or supercapacitors with minimal telemetry and virtually no redundancy. Across these classes the core engineering trade-off remains the same: larger mass allows more redundancy, higher stored energy and more aggressive thermal and radiation mitigation, while smaller mass forces prioritization of essential loads, tighter operational scheduling, aggressive efficiency choices and system-level co-design (ADCS, structure and EPS) to extract the maximum capability from minimal resources.

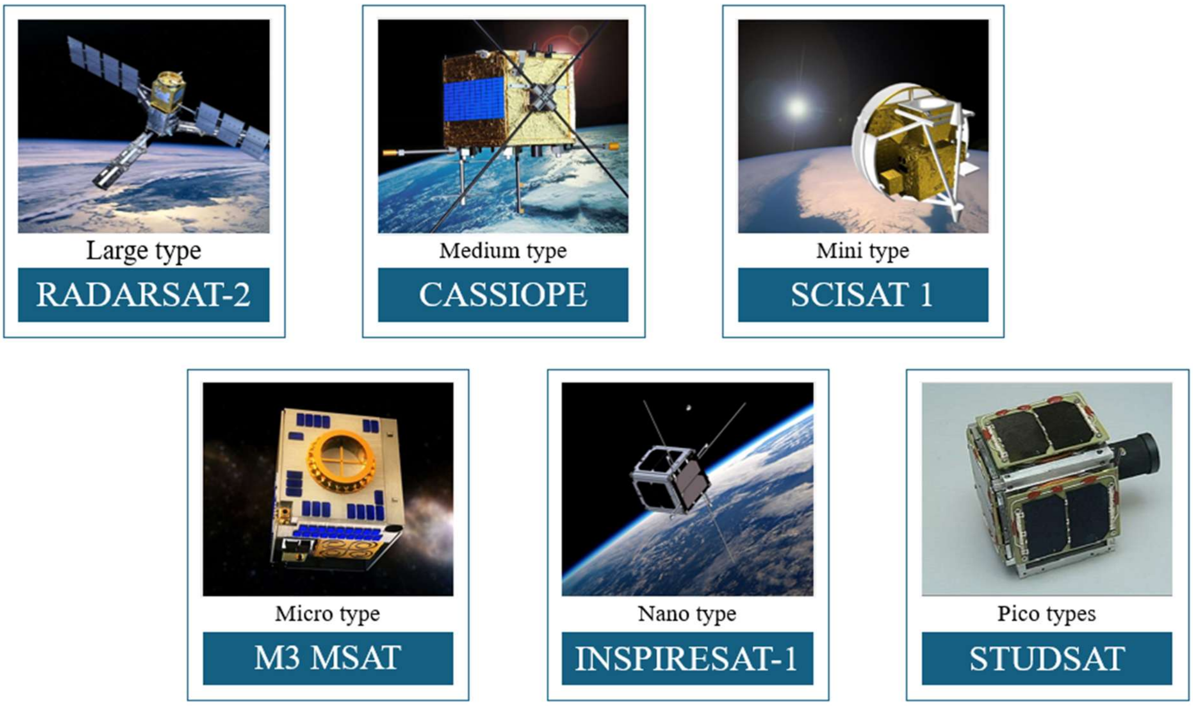

Figure 1: Types of satellites

CHAPTER 3

CUBESAT BASICS AND APPLICATIONS

CubeSats are nanosatellites developed according to a standardized cube-shaped design. The concept was introduced in the late 1990s to provide universities with affordable access to space. Each CubeSat unit, or 1U, is a ten-centimetre cube with a mass of about 1.33 kilograms. This modular structure allows researchers to combine multiple units to form larger satellites, such as 2U or 3U CubeSats. The standardization of size has enabled the development of deployers, such as the Poly-Picosatellite Orbital Deployer (P-POD), which makes launching multiple CubeSats on a single rocket possible.

The fundamental idea behind CubeSats is to provide a low-cost, rapid development platform that lowers the entry barrier to space research. Instead of requiring years of design and billions of dollars, CubeSats can be built in months with a fraction of the cost. This democratization of space access has significantly expanded the number of missions and participants in the space sector. However, the small form factor of CubeSats imposes strict constraints. Every subsystem, including EPS, communication, thermal management, and payload, must be designed to fit into the limited volume and mass budget.

Despite these constraints, CubeSats are increasingly used for a variety of applications. In scientific research, CubeSats have been employed to study the Earth's atmosphere, track space weather, and even conduct interplanetary exploration, as seen in missions like Mars Cube One. In remote sensing, CubeSats equipped with cameras and sensors provide valuable data for agriculture, environmental monitoring, and disaster response. In communications, CubeSat constellations support internet access, ship and aircraft tracking, and emergency communication systems. They also serve as technology demonstrators, testing innovative solar cells, battery designs, and propulsion systems before these are deployed on larger satellites. Additionally, CubeSats have become a critical tool in education, allowing students to gain hands-on experience in satellite design and operations. defence and security agencies have also begun to recognize their potential for surveillance and secure communication.

CubeSats represent a revolution in space exploration and utilization. Their widespread adoption is made possible only by the reliability of their EPS, which determines whether these tiny satellites can successfully achieve their mission objectives.

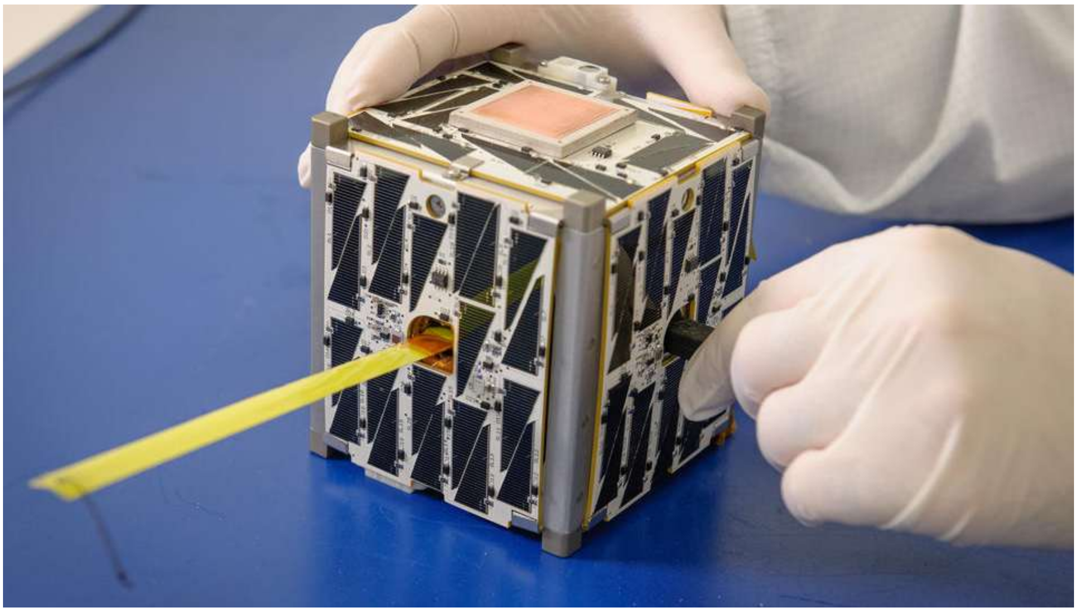

Figure 1: overview of CubeSat

CHAPTER 4

CUBESAT LAUNCH HISTORY

The history of CubeSat launches began in the year 2000, when the first pair of 1U CubeSats was launched as part of a student initiative. Initially, CubeSats were primarily academic projects designed for educational purposes. Between 2003 and 2010, the number of launches grew slowly, with most missions limited to simple demonstrations and experiments. During this early period, CubeSats often faced high failure rates, particularly due to underdeveloped EPS subsystems.

4.2 Early CubeSat launches (2000–2010)

The first decade of CubeSat activity was driven by university teams converting lab prototypes into flight hardware, which produced rapid growth in launches but a high failure rate. Common causes were inadequate subsystem integration and testing—especially immature power systems, deployment mechanism failures, wiring and connector faults, lack of thermal-vacuum and vibration verification, software boot/safe-mode issues, and poor attitude control that prevented correct pointing. These lessons led the community to adopt flight-proven EPS modules, per-panel protection and richer telemetry, fault-injection and TVAC/vibration testing, conservative DoD and degradation margins, and simple robust boot and safe-mode software, improving CubeSat reliability.

4.3 Commercialization and scaling (2010–2020)

Between 2010 and 2020 CubeSats moved from mostly academic experiments to commercial, institutional and government operational use, driven by falling launch costs, improved miniaturized electronics, and standardized form factors. Companies began building large constellations for Earth imaging, weather and IoT services, creating steady demand for reliable, repeatable small-satellite hardware and operations. Commercial EPS suppliers appeared with flight-proven modules, reducing custom development and schedule risk for small teams. Standardized deployers, mass production of high-efficiency cells and COTS avionics lowered unit cost and enabled economies of scale, while launch brokers and rideshare programs made frequent, affordable access to orbit possible.

Operational needs pushed EPS designs toward higher maturity: emphasis on long-term degradation modelling, radiation derating, better BMS and telemetry, and more rigorous qualification testing. At the same time, software and operations matured—automated health monitoring, predictive maintenance, and coordinated constellation scheduling increased service availability. The net effect was rapid scaling of mission volume, improved reliability compared with the 2000s, and a commercial ecosystem where many teams buy validated EPS and avionics instead of building them from scratch.

4.4 Notable CubeSat missions and constellations – examples

- ✓ Planet Labs (Dove constellation, 2013–present): A large fleet of ~3U CubeSats providing high-cadence Earth imagery for agriculture, mapping and disaster response; demonstrated rapid production, constellation operations and commercial imaging at low cost.
- ✓ Spire Global (LEMUR satellites, 2013–present): Constellations of CubeSats carrying GPS-RO, Automatic Identification System (AIS) and meteorological payloads for weather forecasting, ship tracking and analytics; notable for operational data services and vertical market monetization.
- ✓ MarCO (Mars Cube One, 2018): Twin 6U CubeSats that flew to Mars as relays during the Insight landing, marking the first successful

interplanetary mission using CubeSat platforms and proving deep-space communications and navigation capability.

- ✓ QB50 (2017–2018, distributed science network): An international multi-CubeSat campaign of university satellites designed to perform coordinated in-situ measurements of the lower thermosphere, showcasing cooperative science across many small teams.
- ✓ PhoneSat / CubeSat-X (early NASA/university demos, 2012–2014): Low-cost technology demonstration missions using smartphone electronics to validate rapid prototyping approaches and highlight limitations and lessons for CubeSat avionics.
- ✓ IceCube / RadCube / other mission-specific CubeSats: Numerous 3U–6U missions (polar ORBITAL ICE sensing, radiation/space-weather experiments, technology demonstrations) that illustrate how CubeSats are now used for focused scientific objectives once reserved for much larger platforms.

4.5 Current launch vehicles and deployment trends

Large rockets such as Falcon 9 routinely carry many CubeSats as low-cost rideshare secondary payloads, while medium launchers like PSLV, Soyuz, Atlas and Long March continue to accommodate mixed manifests and small-sat slots. Dedicated small launchers such as Rocket Lab Electron offer more flexible, faster access and precise orbit delivery, and an expanding set of regional small vehicles provides short-lead local options. Standard deployers (P-POD, ISIPOD) remain the common separation hardware for 1U–3U units, with ESPA rings and custom adapters used for larger or mixed payloads, and emerging services such as space tugs enable transfer to nonstandard or higher orbits. Together, rideshare economics support large constellations while dedicated launches suit missions needing specific orbits or schedules; regulators deorbit requirements and the push for flight-proven EPS and standardized interfaces are driving teams toward proven hardware and earlier test/mission-assurance steps.

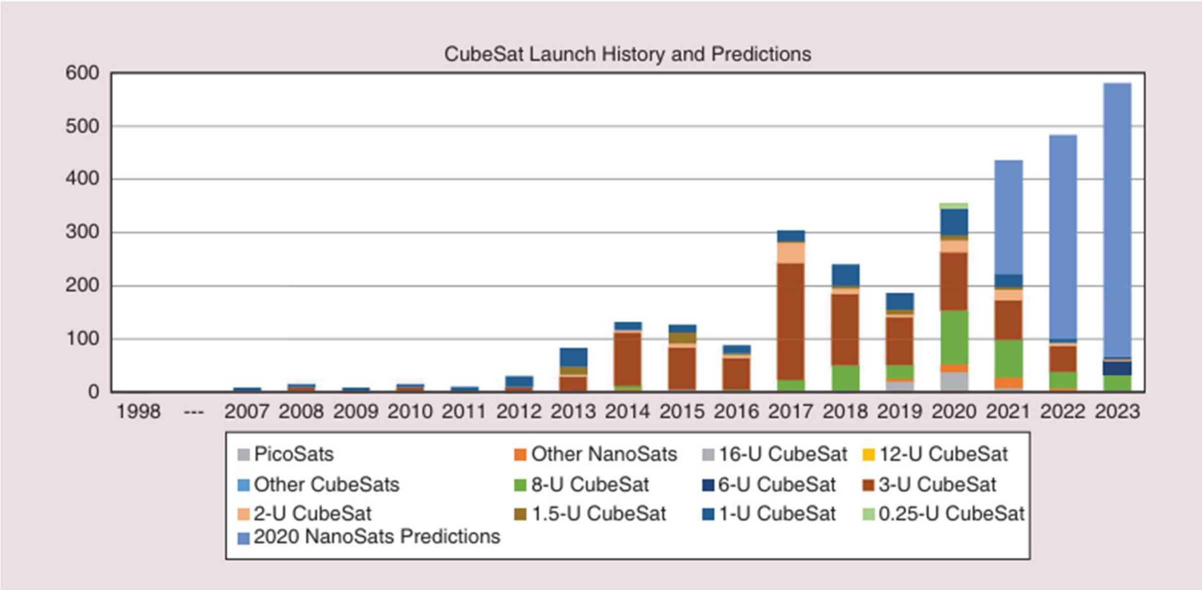

Figure 2: CubeSat Launch History

CHAPTER 5

EPS OVERVIEW & ARCHITECTURE

The Electrical Power System is responsible for power generation, storage, conditioning, distribution, and protection, and EPS anomalies are a frequent cause of satellite mission degradation or loss. Generation uses solar arrays with MPPT front ends, storage relies on battery packs and a BMS to manage charge and health, conditioning employs converters and regulators to produce stable bus voltages, distribution uses switched load lines and fuses to route power, and protection elements such as latch-up current limiters and undervoltage protection prevent damage from overloads and deep discharge. EPS architectures can be centralized, providing a single main bus with simplified telemetry at the cost of single-point failure risk, or modular/distributed with per-panel MPPT and per-load protection that improves redundancy and fault isolation. Commercial CubeSat EPS platforms exemplify typical trade-offs: integrated products like the GOMspace P31U provide multiple solar inputs with built-in MPPT and load switching, simplifying integration for student and small-team missions. Bus voltage selection must balance the number of series battery cells, converter efficiency, and I^2R losses, and higher voltages reduce current but complicate component selection and safety.

The EPS of a CubeSat is designed to manage all aspects of power flow within the satellite. It consists of four major functional blocks: **power generation, power storage, protection, and loads**.

- **Generation** is typically accomplished by solar arrays, which serve as the primary energy source in orbit.

- **Storage** is achieved using rechargeable batteries, which provide power during eclipse periods when the satellite is not exposed to sunlight.
- **Protection** circuits are implemented to safeguard against electrical faults such as short circuits, overvoltage, or undervoltage conditions.
- **Loads** refer to the various subsystems of the satellite, such as the communication system, payload instruments, sensors, and attitude control systems.

The Electrical Power System of a CubeSat is often described as the central nervous system of the spacecraft because it integrates generation, storage, regulation, and distribution of power across all subsystems. The functional architecture of a CubeSat EPS is built around four main components: solar arrays, batteries, power conditioning units, and distribution and protection modules.

Solar arrays, which are typically mounted on the CubeSat's outer surfaces, capture sunlight and convert it into electrical energy. This energy is then processed by maximum power point tracking converters to maximize efficiency before being sent to the onboard battery for storage or directly to the power bus. The batteries act as reservoirs that provide energy during eclipse periods when the satellite is in the Earth's shadow. Power conditioning and control units, often realized through DC–DC converters and voltage regulators, maintain the stability of the bus voltage and ensure that the power supplied to different loads remains within required limits. Protection and distribution modules are responsible for isolating faults, preventing damage, and prioritizing power delivery to critical subsystems.

The architecture is usually managed by a microcontroller or onboard computer, which monitors battery health, load conditions, and fault status. In CubeSats, the limited space and mass constraints prevent extensive redundancy, but fault-tolerant design and modular protection are incorporated to ensure reliability. This makes the EPS not only a supplier of energy but also an intelligent manager of the satellite's overall power resources.

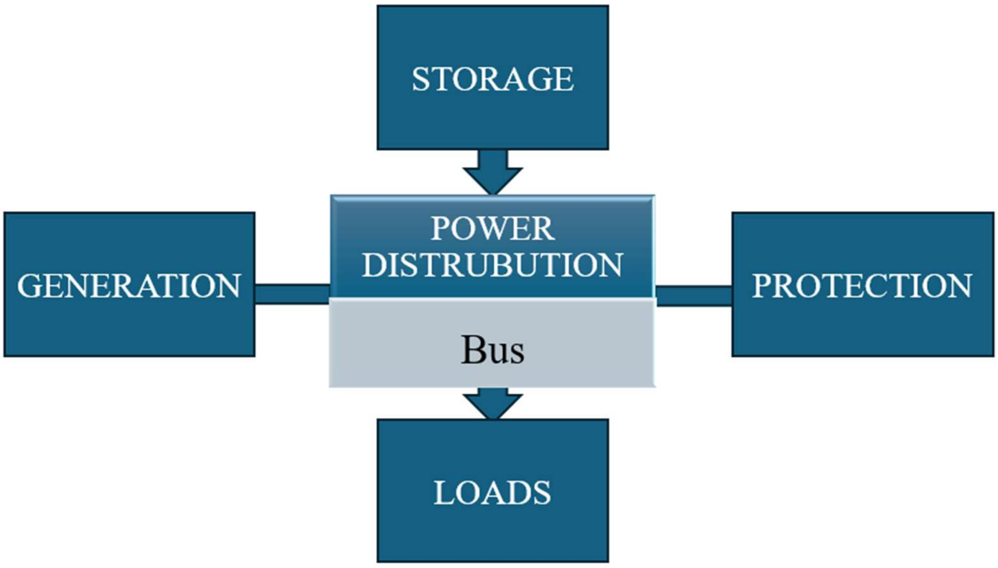

Figure 3:EPS Functional Blocks

5.2 Functional blocks overview

The EPS is built from four tightly integrated functional blocks: generation, storage, conditioning, and distribution. Generation captures sunlight with solar panels and delivers raw power to the bus, often through per-panel interfaces and bypass diodes to handle shadowing and degradation. Storage uses rechargeable batteries and a Battery Management System (BMS) to provide energy through eclipses, manage cell balancing, temperature limits and state-of-charge reporting. Conditioning converts and regulates voltages with MPPT controllers and DC–DC converters to maximize harvested power and supply clean, stable voltages to subsystems. Distribution routes power to loads via switches, current sensors and protective elements (LCLs, UVP, fuses or solid-state switches), implements load prioritization and fault isolation, and provides telemetry to the onboard computer so control software can perform scheduling, safe-modes and health management.

5.3 Typical components and their roles

An EPS combines several key components that work together to harvest, store, condition and deliver power: MPPTs continuously adjust converter operating points to extract maximum power from solar panels under changing illumination and temperature; the BMS protects and manages the battery by monitoring cell voltages and temperatures, enforcing CC-CV charge profiles, balancing cells and logging faults to maximize life and safety; DC–DC converters regulate and transform voltages between the solar array, battery and payloads, providing stable rails with high efficiency and low EMI; and switches and protection devices (solid-state switches, LCLs, fuses and UVP logic) route power, isolate faults and implement staged load shedding to preserve mission-critical functions. Integrated telemetry and an onboard controller coordinate these elements so power policies, safe-modes and diagnostics can run autonomously while hardware fallbacks ensure survival when software is unavailable.

5.4 Design constraints for redundancy and fault tolerance

Redundancy and fault tolerance in CubeSat EPS design are limited by strict size, weight and power (SWaP) budgets, so designers must prioritize where redundancy yields the most mission value; full duplication of major subsystems is usually impossible, therefore selective redundancy (critical loads, communications path, and telemetry) and distributed protection (per-panel diodes, per-branch switches, and local current sensing) are preferred. Hardware redundancy choices must account for quiescent power cost and added mass, so passive one-time devices (fuses, PTCs) are used where low standby loss is essential while telemetry-enabled solid-state switches are chosen where recoverable isolation and diagnostic visibility matter. Fault-tolerance policies must balance automatic hardware actions (UVP/LCL) that operate independent of the OBC against software-driven recovery sequences that provide flexible staged shedding and retries; thresholds and timeouts require conservative margins and hysteresis to avoid oscillation in noisy conditions. Thermal, mechanical and radiation hardening constraints further limit redundant options, forcing teams to combine operational mitigations (conservative DoD, schedule high-power tasks in sunlight, predictive health monitoring) with compact hardware protections and detailed fault logging so ground teams can diagnose and recover remotely.

CHAPTER 6

POWER GENERATION USING PV CELLS

Space-grade power generation for CubeSats Favors multi-junction photovoltaic cells because their higher efficiency and superior radiation resistance provide more energy per unit area than silicon cells, a vital advantage when surface area is limited. Triple-junction GaInP/GaAs/Ge cells convert a larger fraction of the solar spectrum into electrical energy and maintain useful output longer under proton and electron irradiation typical of low Earth orbit. Designers must also consider sub-system constraints: cell selection determines open-circuit voltages and current densities that influence MPPT converter topology and series/parallel stringing of cells to meet bus voltage targets. When budgets permit, purchase of flight-qualified cell assemblies or pre-qualified panels reduces development risk compared with assembling cells in-house, but COTS panels sized for CubeSats are a common compromise for university and early-stage commercial missions.

Panel deployment and mounting strategies strongly affect energy yield. Fixed body-mounted panels are simplest and minimize mechanical complexity but produce less energy for missions requiring high duty-cycle payloads or unfavourable attitude profiles. Deployable panels or multi-faceted arrays increase collecting area yet demand reliable deployment mechanisms and must be validated across vibration and thermal-vacuum tests. Optical and thermal factors—incidence angle, seasonal beta-angle variations, eclipse fraction, and surface degradation from contamination or radiation—must be included in energy budgets and in margin calculations. Practical PV design for CubeSats therefore blends high-efficiency cells, careful mechanical placement, and conservative margins for degradation and off-nominal pointing to ensure the satellite meets its power needs across the mission lifetime.

Solar cell technology choices drive power density and voltage available to CubeSat EPS; silicon cells are mature with efficiencies around 15–18 percent, whereas triple-junction GaInP/GaAs/Ge cells achieve roughly 28–30 percent efficiency and provide superior voltage per area for LEO missions. The electrical behaviour of PV cells can be modelled with the single-diode one-sun model to derive I–V curves, MPP, and fill factor, and cell parameters such as Voc and Isc shift with temperature and irradiance. Environmental factors including temperature, displacement damage from radiation, and angle of incidence cause performance degradation and require margining for end-of-life power. CubeSat panels may be body-mounted or deployable; deployable increase generating area and mission capability at the cost of mechanical complexity and additional qualification requirements. Space-qualified commercial panels such as the GOMspace P110 are designed to operate across typical LEO temperature ranges and provide compact, rugged mounting and electrical interfaces that integrate with MPPT front-ends. Designers must compute I–V curves at representative operating points and include derating for temperature, reflection losses, and expected degradation when sizing the PV array and selecting MPPT hardware.

6.2 PV cell types

Photovoltaic cells for space are mainly **silicon** and **multi junction** technologies. Silicon cells are single junction devices that are mature, lower cost, mechanically robust and tolerant to handling. They offer moderate conversion efficiency and simpler manufacturing and testing workflows. Multi junction cells stack multiple semiconductor layers each tuned to a different portion of the solar spectrum, which increases total conversion efficiency at the cost of complexity and price.

6.3 Silicon cells — characteristics and uses

Silicon cells deliver reliable performance for low-to-medium power CubeSats and missions where cost, ease of integration and mechanical durability matter. They work well for planar arrays or flexible panels where high areal efficiency is not the primary driver. Silicon cells degrade predictably in orbit and are easier to procure and qualify for university and small commercial missions.

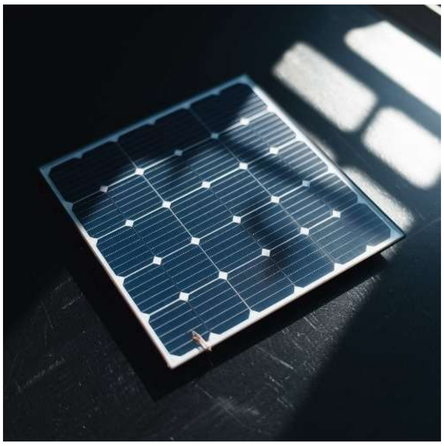

Figure 4: silicon cell

6.4 Multi junction cells — characteristics and uses

Multi junction cells achieve significantly higher efficiencies by using two or more junctions stacked to capture different wavelength bands of sunlight. They are especially valuable where power per unit area is critical such as high-power density panels, deployable arrays and missions with tight aperture constraints. Multi junction cells command higher upfront cost, require careful thermal and mechanical handling and may need specific qualification for radiation sensitivity.

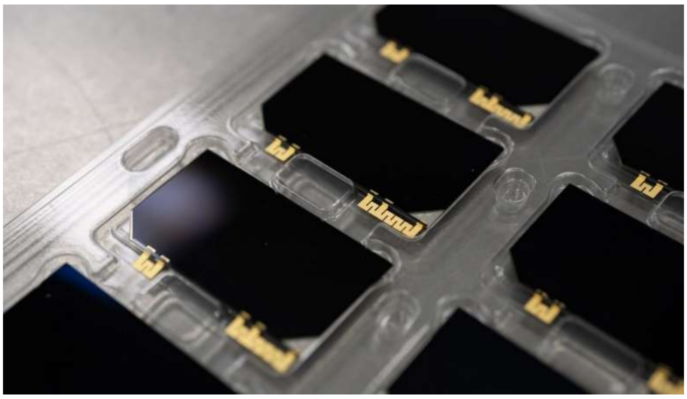

Figure 5 : Triple-junction GaInP/GaAs/Ge cells

6.5 Triple junction benefits

Triple junction cells combine three semiconductor layers to capture short, mid and long wavelengths of sunlight, producing much higher conversion efficiency than single junction silicon. The main benefits are higher power output per unit area, better performance at elevated temperatures and improved end-of-life power when radiation damage is considered because the layered design spreads spectral energy capture across junctions. These benefits enable smaller arrays or higher payload power budgets, which is critical for small satellites with limited surface area.

6.6 Trade-offs and selection guidance

Choose silicon when budget, mechanical robustness and supply chain simplicity are priorities and when array area is ample. Choose multi junction, and specifically triple junction, when maximizing power per area or minimizing stowed volume matters and when mission budget and qualification timelines can absorb higher cost and handling complexity. Account for thermal management, radiation environment and qualification testing when selecting cell type.

CHAPTER 7

MPPT AND POWER CONTROL

Maximum Power Point Tracking maximizes harvested energy by adjusting converter duty cycles to operate PV panels at their instantaneous MPP under changing irradiance and temperature, and common algorithms include Perturb-and-Observe, Incremental Conductance, and fractional VOC/ISC methods with trade-offs in convergence speed and steady-state oscillation. Converter topologies used in CubeSat EPS typically include boost converters when panel Voc is below battery voltage, buck converters for stepping down, and SEPIC or quasi-Z topologies when wide input ranges are required; synchronous rectification and radiation-tolerant switching devices improve efficiency and reduce thermal load. Implementation constraints require careful selection of ADC resolution and sampling rates for accurate MPPT decisions, evaluation of SEU and radiation effects on control electronics, and coordination between MPPT and battery-charging algorithms to respect battery health constraints. Multi-input MPPT architectures, where each panel has an independent MPPT channel, provide resilience against partial shadowing and attitude-induced variability and are widely adopted in commercial CubeSat EPS modules.

7.2 MPPT algorithms - P&O, incremental conductance, adaptive

Perturb-and-observe (P&O) perturbs the operating voltage and measures power change to climb toward the maximum power point; it is simple, low-cost and effective under slowly varying irradiance but can oscillate around MPP and respond poorly to rapid light changes. Incremental conductance (IncCond) compares instantaneous conductance to its incremental change to find the MPP exactly, offering better steady-state accuracy and faster response to changing illumination at the cost of slightly higher computational complexity. Adaptive

MPPT techniques change step sizes, switch between algorithms, or use predictive models (irradiance estimates or learning-based predictors) to combine stability and speed: they reduce oscillation in steady state while tracking quickly during fast transients, making them attractive for per-panel MPPT where illumination varies rapidly.

7.3 Converter topologies — boost, buck, buck-boost, multi-input

Boost converters increase voltage from low-voltage arrays to the battery or bus charging voltage, useful when solar Vmp is below battery voltage; they must handle start-up and operate efficiently over wide input. Buck converters step down battery or bus voltage to supply regulated payload rails with high efficiency, suitable for stable output rails. Buck-boost topologies maintain regulation when input can be above or below the desired output, enabling continuous service across the battery state-of-charge swing but adding complexity and potential efficiency trade-offs. Multi-input converters aggregate multiple sources (several panels, solar plus auxiliary) into a single regulated output with internal source arbitration or blending logic, reducing wiring and MPPT overhead while enabling seamless transition between sources.

7.4 Multi-MPPT and per-panel strategies

Multi-MPPT assigns an MPPT controller to each panel or panel group so each face can operate at its own optimum, preventing a shaded or off-angle panel from dragging down the whole array; this is especially beneficial on multi-faced CubeSats and deployable. Per-panel MPPT increases harvested energy in partial shading, low-sun or attitude-varying regimes, but increases hardware count, quiescent power and complexity. A pragmatic approach uses hybrid grouping per-panel MPPT for faces that see independent illumination and grouped MPPT where faces are co-planar, balancing power gain against mass, cost and standby consumption.

7.5 Battery charge control — CC-CV and temperature compensation

CC-CV charging uses a constant current phase to quickly bring the battery toward target voltage followed by a constant voltage phase that limits voltage while current tapers, protecting cells and finishing charge safely. Proper implementation enforces controlled cutoffs, timeouts and end-of-charge detection to prevent overcharge and excessive tapering that shortens life. Temperature compensation adjusts charge voltage and current limits based on cell temperature to avoid overcharging at high temperature and under-charging when cold; it also triggers thermal cutoffs or derating if temperatures exceed safe bounds. Cell balancing and hardware safety cutoffs complement CC-CV to preserve cell-to-cell uniformity and provide fail-safe protection.

7.6 Load management, prioritization and software policies

Load management policies schedule high-power activities during sunlight and implement staged shedding to protect battery and critical functions during low power events. Prioritization ranks loads (critical comms, attitude control, payload, housekeeping) and enforces rules for interlocks, timed retries and graceful degradation; noncritical services are disabled first while essential telemetry and command paths are preserved. Software implements watchdogs, health-based decisions using telemetry and timeouts, and coordinated safe-modes that work with hardware protections to recover after faults; conservative hysteresis, logging and remote command capability are included so ground teams can diagnose, override and reconfigure priorities when needed.

CHAPTER 8

MODELING OF PV SYSTEMS, CHARACTERISTICS OF CUBESAT SOLAR PANELS

Accurate PV models are the foundation of correct EPS sizing. Start with orbital mechanics to derive insolation time series: compute orbital period, sunlight fraction, eclipse durations and beta-angle behaviour for seasons. Overlay expected attitude profiles or worst-case pointing errors to obtain projected angle-of-incidence on each panel facet. For each time step apply the panel's I-V characteristics adjusted for temperature (using temperature coefficients) and angle-of-incidence losses (using empirically derived or manufacturer-provided curves). Include reflection and wiring losses, MPPT inefficiency, connector losses and margin for manufacturing variability. End-of-life degradation must be applied — typical LEO missions use 10–30% degradation allowance depending on proton/electron flux and mission duration.

Run time-domain simulations across many orbits and apply statistical analyses: worst-case daily energy, minimum orbit-average power, and distribution of recharge times. Sensitivity studies (pointing error, colder/warmer temperatures, greater degradation) reveal which design changes (adding deployable area, adopting higher-efficiency cells, improving pointing) deliver the most benefit. Use hardware-in-the-loop tests to validate model assumptions: illuminate panels at representative intensities and incidence angles, run MPPT and converter hardware with representative battery loads, and record power flow and efficiency. Thermal-vacuum tests validate temperature-dependent performance; combined tests with harness and connectors assess real harness losses.

Well-validated PV models permit confident sizing decisions and enable operations teams to plan payload schedules that meet energy constraints while avoiding battery over-use. Always document model assumptions, input orbit parameters and margin choices so future teams can update calculations for alternate orbits or different component options.

8.2 Orbital insolation and eclipse modelling

Orbital insolation and eclipse modelling Computes when and how much sunlight the CubeSat receives along its orbit using inputs such as altitude, inclination, orbital period and beta angle, producing a time series of sunlight/eclipse intervals and the Sun vector in spacecraft coordinates.

This time series determines the sunlight fraction per orbit and the exact start and stop times of eclipses which directly control when solar panels can generate power and when the battery must supply loads.

Generating high resolution Sun visibility data for many orbits captures seasonal and beta angle variations and provides the driver signals for PV, thermal and battery simulations.

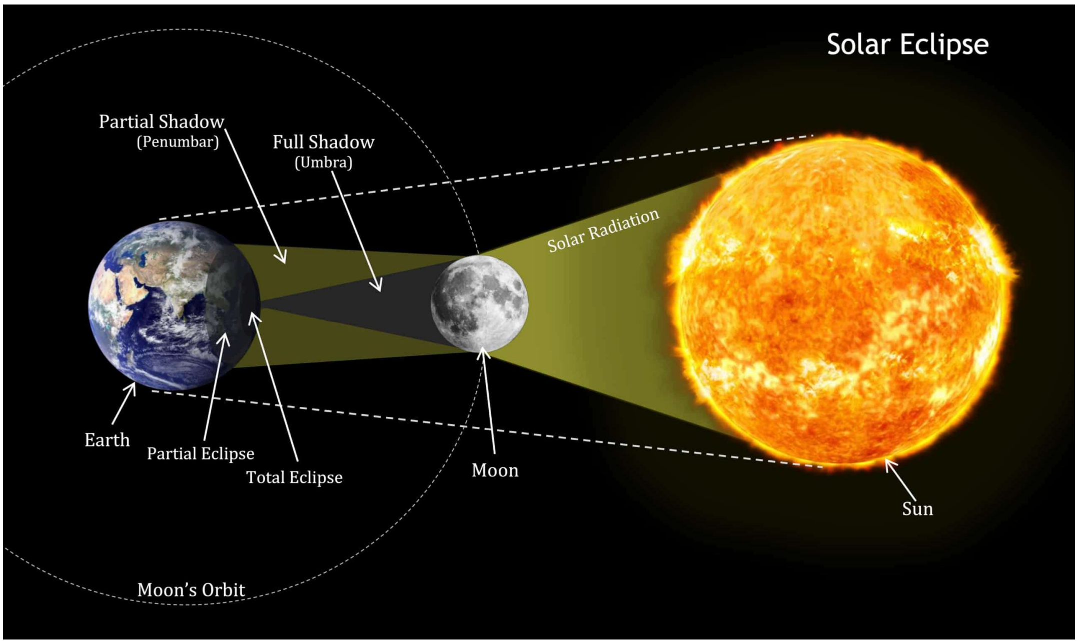

Figure 6:- Orbital geometry / Sun incidence diagram

8.3 Angle-of-incidence and temperature corrections

Angle of incidence and temperature corrections compute the angle between the Sun vector, and each panel face normal for every time step and convert that angle into an angular response factor that modifies the ideal cos(θ) projection; manufacturer angular response curves are used when available.

Cell temperature is estimated from a thermal model or empirical relation, and the temperature coefficients for Voc and Vmp are applied so the cell I–V parameters reflect real operating temperature.

The combined AoI and temperature corrections give the adjusted per facet irradiance and updated I–V parameters used by the MPPT and power calculations. I–V curve adjustments and loss factors

I–V curve adjustments scale the reference I–V curve with instantaneous irradiance to update Isc and use temperature corrections to update Voc and Vmp, while partial shading or mismatch is handled by string-level modelling or per-panel inputs.

Usable electrical power is then reduced by realistic loss factors including MPPT and DC–DC converter efficiency, wiring and connector resistance, mismatch and shading losses, reflection and cover glass losses, and an end-of-life degradation allowance.

These adjustments produce time step Pmp values that, after applying system efficiencies, become the actual usable electrical power feeding the bus and battery.

8.4 Time-domain simulation and statistical outputs

A time domain simulation integrates the per facet usable power over each time step to produce orbit by orbit power and energy time series, battery state of charge profiles, charge/discharge cycles and peak power windows.

From many simulated orbits you extract statistical outputs such as minimum orbit energy, worst day energy, average usable Wh per orbit, maximum continuous recharge time and SoC envelopes across mission life.

Sensitivity scenarios (nominal, best, worst) vary pointing, panel efficiency, temperature and degradation to identify which assumptions most affect battery capacity and panel area. Hardware-in-the-loop and thermal-vacuum validation

Hardware in the loop tests illuminate flight or engineering panels with calibrated sources at representative irradiances and incidence angles while running the actual MPPT, converters and BMS to record real dynamic response and steady state power flow.

Thermal vacuum tests validate temperature dependence of PV output, battery behaviour and electronics, confirm heater strategies, and verify that thermal approximations used in the model match measured temperatures and efficiencies.

Fault and integration tests (partial shading, connector faults, vibration and fault injection) validate protection logic, telemetry richness and recovery sequences so that on orbit behaviour matches the model and operations procedures.

CHAPTER 9

BATTERY TECHNOLOGIES, PACK DESIGN AND MANAGEMENT

Battery selection centres on energy density, cycle life, thermal performance and safety. Lithium-ion chemistries dominate for their energy-per-mass and mature supply chains. Key pack-level design choices include series-count for bus voltage, parallel strings for capacity and redundancy, cell selection for cycle life and operating temperature window, and choice of passive versus active balancing. Active balancing prolongs pack life but adds complexity and quiescent draw; passive balancing is simpler but wastes energy and can be less effective at equalization during long missions.

Lithium-ion batteries are the dominant energy storage choice for CubeSats because they combine high energy density with demonstrable flight heritage, while alternatives like $LiFePO_4$ offer improved thermal stability at the price of lower energy density and supercapacitor hybrids provide high-power buffering but insufficient energy for long eclipses. Battery electrical characteristics include capacity in ampere-hours, nominal voltage that determines pack configuration, internal resistance that affects heat generation and voltage sag at high C-rates, and energy calculated as $E = C \times V_{nom}$. Depth-of-Discharge determines usable energy per cycle and strongly influences cycle life; conservative DoD policies around 30–50 percent extend battery longevity for multi-month to multi-year missions. Charging typically follows a CC–CV profile where constant current charging continues until an end-of-charge voltage is reached and then voltage regulation tapers charging current, and BMS functions such as cell balancing, temperature monitoring, over/under-voltage protection, and telemetry are essential for safe operation. Temperature affects usable capacity and internal resistance; low temperatures reduce available capacity and high temperatures accelerate chemical degradation, so thermal management must maintain batteries in their recommended operating window. Battery sizing proceeds by calculating eclipse energy needs, dividing by allowable DoD to obtain required stored energy, converting to Ah at the chosen bus voltage, and adding margins for end-of-life degradation and thermal derating. Safety and transport regulations require rigorous pre-launch testing and on-board protection to prevent thermal runaway and ensure safe handling.

The Battery Management System (BMS) is critical: it enforces CC-CV charge profiles, monitors per-cell voltages and temperatures, performs balancing, logs events, and asserts hardware-level protections on overvoltage, undervoltage and overcurrent. BMS telemetry must be exposed to flight software and ground operations for trend analysis and anomaly detection. Thermal coupling of the battery pack to structure and heaters is crucial: low temperatures increase internal resistance and reduce usable capacity; high temperatures accelerate capacity fade and shorten lifetime. Design battery placement to allow conductive paths for waste heat from onboard electronics when convenient and to enable heaters when necessary.

Depth-of-Discharge (DoD) is a major lifetime lever: a conservative DoD (30–40%) can dramatically increase usable cycles versus deep cycling, at the cost of greater initial capacity and mass. For long or demanding missions plan for end-of-life capacity margins and include accelerated cycle testing on representative cells to inform aging models. Radiation testing or derating is recommended for long-duration missions or high-radiation orbits; radiation increases internal resistance and leakage and can create safety concerns that must be accounted for in safety analyses.

Finally, battery fault modes deserve careful attention: internal cell failures, thermal runaway propagation, wiring faults and BMS failures are mission-threatening. Include hardware-level isolation switches, thermal fuses, cell-level vent consideration and non-volatile fault logs so that ground teams can diagnose and take recovery actions.

9.2 Battery chemistries and selection criteria

Lithium-ion variants remain the default for CubeSats because they strike the best balance of energy density, cycle life and mature BMS support; LiFePO4 is chosen when safety and thermal stability matter more than energy per mass, and experimental solid-state or advanced chemistries require extensive qualification before flight. When selecting cells, prioritise proven flight heritage and match datasheet specs to the mission envelope: energy density (Wh/kg), usable capacity at expected temperatures, C-rate for charge acceptance, self-discharge, expected cycle life, radiation sensitivity and vendor qualification notes. Accept modest energy penalties to gain better low-temperature charge acceptance or safety if thermal or operational margins are tight and always verify cell behaviour across the orbital temperature and charge/discharge profile expected for the mission.

9.3 Pack configuration: series/parallel, balancing choices

Pack topology is chosen to meet bus voltage and capacity needs: series strings raise voltage (reducing current for a given power) while parallel strings increase capacity and can provide some redundancy but complicate fault modes. Balancing strategy depends on mission duration and DoD policy: passive resistive balancing is simple and low-mass but wastes charge and is less effective for long missions, while active balancing transfers energy between cells to maximise usable capacity and lifetime at the cost of added complexity and quiescent draw. Design the series count to match bus and converter constraints, include per-string monitoring, and favour active or high-accuracy balancing for missions with long life, tight DoD constraints, or many parallel strings.

9.4 BMS functions and telemetry requirements

A robust BMS enforces CC-CV charging, monitors per-cell voltages and temperatures, performs balancing, detects overcurrent/overvoltage/undervoltage events and executes hardware isolation when safety thresholds are crossed, while logging events to non-volatile memory for post-event diagnosis. The BMS should also estimate SoC and SoH and expose key telemetry: pack SoC/SoH, individual cell voltages, cell temperatures, charge/discharge currents, active balancing state, and time-stamped fault flags. Rich, timely telemetry allows ground teams to detect trends such as rising internal resistance or capacity fade and to adapt operational schedules to preserve battery life and mission availability.

9.5 Thermal coupling and heater strategies

Thermal design should keep cells inside their safe and charge-friendly window primarily with passive measures: conductive mounting, thermal straps, selective surface finishes and MLI to minimise heater energy needs. Where active heating is required, heaters must be sized for worst-case cold, controlled with hysteresis to avoid short cycling, and scheduled preferentially during sunlight or surplus-power windows to conserve battery energy. Place temperature sensors on every cell group and at expected hot spots so the BMS can inhibit charging or enable heaters as needed; include heater duty cycles in the energy budget and validate them during thermal-vacuum testing.

9.6 Safety, failure modes and mitigation

Anticipate cell overcharge/overdischarge, internal shorts, thermal runaway propagation, wiring/connector faults and reduced charge acceptance from aging or radiation. Mitigate with layered protections: hardware isolation switches or fuses, thermal fuses and vent paths, LCLs for overcurrent, UVP to prevent deep discharge and redundant sensing for critical parameters, combined with conservative software policies (DoD limits, adaptive task scheduling and autonomous safe modes). Validate mitigations with accelerated cycling, fault-injection, thermal-vacuum, vibration and connector stress tests, and ensure all protection actions produce clear, time-stamped telemetry for root-cause analysis.

CHAPTER 10

THERMAL MANAGEMENT OF EPS AND BATTERIES

Thermal management is tightly coupled to EPS performance. Batteries, converters and MPPT electronics all exhibit temperature-dependent efficiency and aging characteristics, and CubeSats' low thermal mass creates rapid temperature swings between sunlight and eclipse. Begin thermal design with orbit-based thermal simulations that include solar heating, Earth IR, albedo, internal dissipation from electronics and duty-cycled payload heating. Passive thermal control (MLI, surface finishes, thermal straps, conductive couplings to structural radiators) should satisfy as much of the thermal requirement as possible because active heaters consume scarce energy.

The LEO thermal environment imposes cyclic heating and cooling as satellites transition between sunlight and eclipse, and this drives the need for thermal design that keeps EPS components, notably batteries, within functional temperature limits. Passive thermal control strategies such as multi-layer insulation, surface coatings, and conductive paths to the structure are preferred for CubeSats because of their simplicity and low mass, while active methods like resistive heaters are used selectively to maintain minimum battery temperature during extended eclipses. Battery thermal control typically combines insulation with thermostatically controlled heaters that activate under low-temperature conditions, and heater control logic uses temperature sensors and hysteresis to avoid excessive heater duty cycles. Thermal design must include margining for worst-case hot and cold scenarios, and verification requires thermal-vacuum cycling and thermal balance testing to demonstrate survivability and functional performance across the mission temperature range.

When heaters are required, size them and program control hysteresis carefully to minimize duty cycle; coordinate heater operation with MPPT and load management so heaters run preferentially during surplus generation. Place temperature sensors on or adjacent to cells and on hot spots of converters; use the BMS to abort charging if cell temperatures exceed safe thresholds. Thermal interactions with operations are important: long high-power communications bursts heat local electronics but also may reduce battery heating needs; conversely, prolonged eclipses increase heater energy consumption. Model heater duty cycles across worst-case seasons and add the expected heater energy to the PV and battery sizing analysis.

Thermal-vacuum testing of the fully integrated flight-like assembly under representative duty cycles is essential to verify models and confirm heater strategies. Include worst-case cold and hot bakeouts, and measure sensor placement effectiveness and heater latency. Design mechanical structures and harnessing to tolerate repeated thermal cycling and to avoid thermo-mechanical shifts that might affect connectors and deployment mechanisms.

10.2 Thermal environment and modelling inputs

The thermal environment defines the heat sources and sinks the CubeSat will see in orbit and sets the boundary conditions for every thermal decision. Key modelling inputs are orbital parameters (altitude, inclination, beta angle) that determine sunlight/eclipse timing; spacecraft attitude and operational modes that set solar incidence histories; external fluxes (direct solar irradiance, Earth albedo, Earth infrared) and expected seasonal/ephemeral variations; internal dissipation maps from electronics, payloads and heaters; and material/geometry properties (emissivity, absorptivity, thermal capacitance, conductive paths). Use these inputs to build a time-resolved heat balance for each component so you can predict worst-case hot and cold soak temperatures, heater duty cycles, and the interaction between thermal state and EPS performance.

10.3 Passive thermal control techniques (MLI, finishes, straps)

Passive measures change how the spacecraft exchanges heat without consuming power. Multi-Layer Insulation (MLI) reduces radiative heat loss to space and is used on external surfaces to control net heat flux; surface finishes (high/low emissivity or absorptivity paints, anodizing, thermal coatings) tune radiative exchange with space and Earth; thermal straps and conductive pads redistribute heat between hot and cold zones, coupling batteries to warm electronics or radiators to bleed heat away. These techniques are preferred because they are passive, reliable and consume no electrical power; design them early and co-optimise with structure and harness routing to minimise heater burden and avoid thermal gradients that stress components.

10.4 Active thermal control and heater management

Active control uses electrical heaters, thermostatic controllers and control logic to keep temperature-sensitive components within safe or charge-friendly windows. Heater strategies include sizing heaters for worst-case cold, implementing hysteresis to avoid short cycling, and layering heater control so coarse hardware thresholds protect the battery while finer software schedules heaters during surplus-power windows. Coordinate heater operation with MPPT and task scheduling so heaters run when solar generation is available and implement fail-safe hardware cutoffs to prevent runaway heater power. Plan energy budgets for heater duty cycles and design heater control paths that work even in degraded modes (BMS or MCU down).

10.5 Sensor placement and thermal telemetry

Accurate sensing is essential for both control and post-flight diagnosis. Place temperature sensors at critical thermal nodes: on or immediately adjacent to battery cells or cell groups, on MPPT/converter hot spots, near payload heaters, and at representative structural locations to track gradients. Use redundant sensors where safety or charging decisions depend on a reading. Expose thermal telemetry in both compact health packets for regular monitoring and higher-resolution traces stored in non-volatile memory for fault analysis. Timestamped temperature logs combined with power and current telemetry enable correlation of thermal events with operational actions and are vital for tuning heater policies and understanding degradation.

10.5 Thermal test plan and verification

Verification proves the model and validates control strategies under representative conditions. A practical test plan includes thermal-vacuum cycling covering worst-case hot and cold extremes, thermal balance tests to validate steady-state and transient models, and integrated tests with MPPT, BMS and heaters running representative duty cycles. Include combined environment tests (vibration + thermal cycling) to expose thermo-mechanical issues and perform heater latency and sensor-placement checks. Run margin and fault cases (e.g., heater stuck-off, sensor failure, prolonged eclipse) to verify safe behaviour and logging. Use test data to update the thermal model, refine heater schedules and freeze requirements for flight hardware.

CHAPTER 11

PROTECTION SYSTEMS: LCL, UVP, FAULT ISOLATION AND RECOVERY

Protection systems protect the EPS and payloads from overcurrent, latch-up, and deep discharge events; common elements include latch-up current limiters that limit fault currents and either autonomously re-try or trip loads after a timed interval, undervoltage protection that disconnects non-essential loads to prevent battery deep discharge and uses hysteresis to avoid rapid re-cycling, and distribution-level switches and fuses to isolate faults. LCL implementations range from active current limiting with MOSFETs and sense resistors to crowbar mechanisms and polymer fuses for catastrophic failures, and their logic must include trip timers, retry policies, and clear telemetry of fault events for ground analysis. UVP thresholds should be chosen to protect battery chemistry while preserving critical housekeeping functions, and designers must weigh centralized UVP controllers against distributed per-load UVP circuits to balance simplicity and single-point failure risk. Robust fault detection, logging, and built-in test telemetry enable post-event diagnosis and improve recovery strategies, and fault-injection and qualification testing are necessary to demonstrate predictable protection behaviour under launch and on-orbit conditions.

A robust protection architecture prevents a single electrical fault from ending a mission by both limiting immediate damage and enabling predictable recovery so the satellite can continue essential operations. Protection must preserve battery health, protect loads and provide sufficient functionality for telemetry and safe-mode actions; because CubeSats are constrained in volume, mass and power, protection solutions must be compact, energy-efficient and observable through telemetry. The fundamental design principle is layered protection that progresses from limiting to isolating to logging and then recovering, and thresholds should be conservative with hysteresis to prevent chattering. Hardware and software actions must be coordinated so that protection events are logged and reported in sufficient detail for ground diagnosis, and quiescent overhead from protection electronics should be minimized.

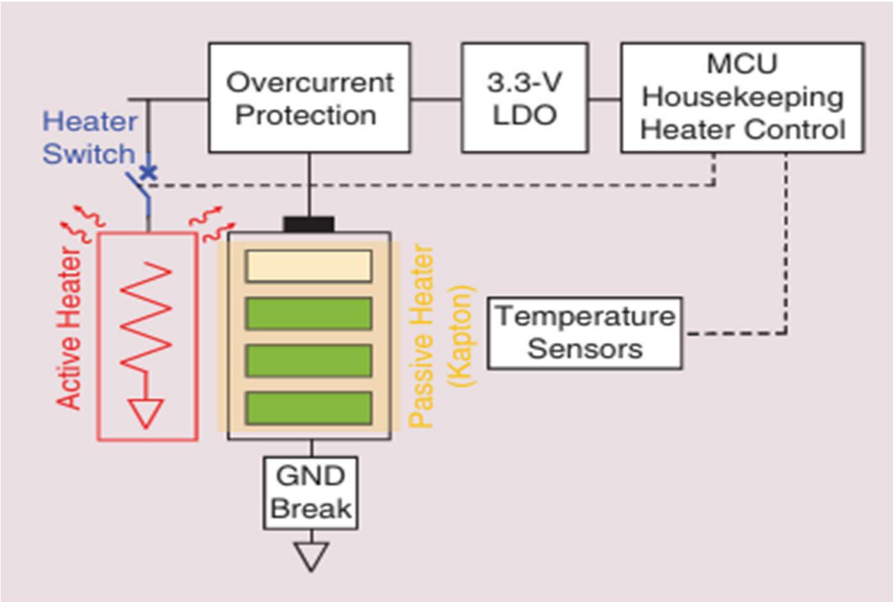

Figure 7:Protection Systems: LCL, UVP, Fault Isolation and Recovery

11.2 Latch-up current limiters (LCL) detect, and control sudden overcurrent conditions caused by latch-up, shorts or component failures in order to avoid battery collapse and thermal damage. In normal operation an LCL passes nominal current with minimal voltage drop; when current exceeds a designed threshold the device enters a controlled limiting mode that preserves the bus as much as possible. If the overcurrent persists beyond a configured timeout the limiter opens the branch to isolate the fault and stores an event for telemetry; after a safe cooldown interval a soft-restart or auto-retry may be attempted to verify whether the fault was transient. Implementation choices range from simple passive polymer fuses, which are slow and do not provide telemetry, to active analogue limiters built from sense resistors, comparators and MOSFETs that offer fast, tuneable response, timed trips and controlled soft-starts, and finally to smart solid-state switches that include programmable limits, readable fault registers and controlled turn-on sequencing. Practical LCL design sets the limit threshold above expected peak operational currents but below destructive levels, selects trip timeouts to distinguish transients from persistent faults, and ensures the sense and switching elements impose minimal series resistance during normal operation so conversion losses remain small.

11.3 Undervoltage protection (UVP) prevents deep discharge of battery cells and protects the system when bus voltage falls below safe operating points, which preserves battery longevity and reserves energy for critical telemetry and beacon functions. A UVP system monitors bus voltage and implements hysteresis and delays so that momentary dips do not cause repeated connect/disconnect cycles. When thresholds are crossed, UVP initiates staged shedding: nonessential loads are disconnected first, and higher-priority loads are removed progressively if the voltage continues to fall; if the situation continues to worsen the system enters a hard safe-mode that preserves only the minimum energy path required for survival and reporting. UVP implementations can be hardware-based comparators, BMS-managed relays or MCU-supervised switches, but hardware-level UVP is important because it guarantees action even when the main flight computer is unresponsive. UVP thresholds must be aligned with cell-level protections, account for converter dropout voltages and measurement tolerances, and record events to non-volatile memory so ground teams can reconstruct the fault sequence.

Fault isolation strategy determines the granularity at which protection acts and therefore strongly influences survivability. A distributed protection approach places protection on each panel input and on each load branch so that a single short cannot disable the whole vehicle, while a centralized approach simplifies design but creates a single point of failure. A hybrid approach is often optimal for CubeSats: per-panel input protection and bypass diodes protect generation sources, per-branch switches isolate loads such as payloads, communications and ADCS, and a supervisory UVP watches the main bus. Isolation devices range from solid-state MOSFET switches, which allow controlled re-enable and telemetry, to latching relays or one-time polymeric fuses where extremely low quiescent loss or guaranteed single-action protection is required. Whatever the topology, every isolation action should generate a concise telemetry flag and a time-stamped non-volatile log entry containing measured current, voltage and temperature to support root-cause analysis on the ground.

Effective fault detection and diagnostics require adequate sensing and thoughtfully designed telemetry. Per-branch current and voltage sensing, battery cell voltages and temperatures, switch states and event timestamps must be sampled at rates sufficient to capture transients and to reconstruct trip waveforms. Telemetry packets should include compact health summaries for real-time monitoring and larger, detailed traces stored in non-volatile memory for later download. Diagnostic data should combine pre-event rolling statistics, trip peak values and environmental context such as battery state-of-charge and temperature so ground analysts can identify root causes and tune thresholds for future operations. Instrumenting protection with telemetry-enabled devices wherever possible is far preferable to relying on passive devices that provide no readback.

Verification of protection strategies requires an extensive test program that includes functional tests, fault-injection, thermal-vacuum cycles, and vibration and shock testing with integrated harnesses. Fault-injection tests must simulate shorts, latch-up events and undervoltage scenarios across the full range of battery state-of-charge and temperature extremes to confirm that LCLs trip as intended, that UVP shedding follows the staged policy, and that soft-restart and logging work reliably. Timing parameters such as trip timeouts, re-enable delays and hysteresis margins must be validated to avoid oscillation in noisy environments. Environmental testing should demonstrate consistent protection behaviour after vibration, thermal cycling and connector stress.

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11.4 Protection principles and layered approach

A robust protection architecture uses multiple overlapping layers so a single fault cannot immediately end the mission; the lowest layer is fast, hardware-based limiting that prevents catastrophic currents or voltages, the middle layer isolates faulty branches or sources to preserve the remainder of the vehicle, and the top layer is supervisory software that logs events, attempts controlled recovery and enforces policy. Layers should be designed with conservative thresholds and hysteresis to avoid chattering, coordinated so hardware actions always protect the battery even if the flight computer is unresponsive, and instrumented so every protective action produces concise telemetry and a non-volatile timestamped log for ground diagnosis.

11.5 Latch-up current limiters and implementation options

Latch-up current limiters (LCLs) detect sudden overcurrent conditions and move quickly from low-loss pass mode to a controlled limiting state that prevents battery collapse and thermal damage; if the condition persists a branch is opened or latched out. Implementation ranges from simple one-time polymer fuses (very low quiescent loss but no telemetry) through timed analogy limiters built from sense resistors, comparators and MOSFETs (tuneable trip and soft-start) to smart solid-state switches that provide programmable limits, readable fault registers and controlled re-enable sequences. Practical LCL design chooses thresholds above expected operational peaks but below destructive levels, selects trip timeouts that distinguish transients from persistent faults, and minimises normal-operation series resistance to avoid wasting harvested energy.

11.6 Undervoltage protection and staged shedding

Undervoltage protection (UVP) prevents deep discharge by monitoring the bus and disconnecting loads in prioritized stages as voltage deteriorates nonessential loads are shed first, progressively higher-priority systems are removed if the bus continues to fall, and a hard safe-mode preserves only the minimum path for telemetry and survival when necessary. UVP must implement hysteresis and delays, so transient dips do not cause oscillation, align thresholds with cell-level protections and converter dropout voltages, and be available as a hardware fallback in case the main computer is offline. Every shedding event should be logged in non-volatile memory with context (SoC, temperature, measured currents) so ground teams can reconstruct the sequence and tune policies.

11.7 Branch isolation strategies and devices

Fault isolation determines how narrowly the protection system can isolate a problem without disabling the whole vehicle; distributed protection (per-panel inputs, per-load branches) maximises survivability while centralized schemes simplify design but create single points of failure. Devices used include low-loss MOSFET load switches for controlled re-enable and telemetry, latching relays or one-time fuses where extremely low quiescent loss or guaranteed single-action protection is required, and bypass diodes on PV inputs to handle panel-level shorts. A hybrid topology—panel input protection plus per-branch switching with a supervisory UVP—often balances complexity and resilience; every isolation device should generate a telemetry flag and a timestamped log entry with measured electrical parameters.

11.8 Fault logging, telemetry and recovery sequences

Effective recovery depends on clear, time-stamped telemetry and non-volatile fault logs that record pre-trip rolling statistics, trip peak values and environmental context such as SoC and temperature; compact health packets are used for routine monitoring while detailed traces are stored for post-event download. Recovery sequences should be deterministic and layered: after a limiter trip or shedding event, attempt controlled soft-restarts with cooldown delays and monitored re-enable, escalate to manual ground intervention if automatic retries fail, and always record each action so analysts can trace cause and effect. Design telemetry to capture enough detail to reconstruct transient waveforms and decision logic without saturating the downlink.

11.9 Test and fault-injection plan

Verification requires planned, repeatable fault-injection across the full operational envelope: simulate shorts, latch-up events and undervoltage scenarios at varied SoC and temperature extremes to confirm LCL trip behaviour, staged UVP shedding, soft-restart timing and logging fidelity. Combine functional fault tests with environmental tests—thermal-vacuum cycles, vibration and connector stress—to ensure consistent behaviour after mechanical and thermal stress. Validate timing parameters (trip timeouts, re-enable delays, hysteresis) to avoid oscillation in noisy conditions, and include negative tests (sensor failure, stuck heater) to confirm safe fail-over. Use test results to refine thresholds, update non-volatile log formats and finalize acceptance criteria for flight hardware.

CHAPTER 12

SIZING: WORKED EXAMPLES, METHODOLOGY AND SENSITIVITY ANALYSES

Sizing begins by creating a complete mission energy budget that enumerates average and peak power requirements for each subsystem during sunlight and eclipse segments, and designers compute required eclipse energy as the sum of loads active during eclipse multiplied by their durations. Battery capacity is obtained by dividing required eclipse energy by the allowed DoD and by the bus voltage to convert energy into ampere-hours, then adding margins for end-of-life degradation and thermal derating. Solar array sizing ensures the daily harvested energy covers mission loads plus energy required to recharge batteries and system losses, and designers must account for sunlit duration, degradation over mission life, pointing errors, and shadowing when selecting panel area and MPPT capabilities. A representative worked calculation for a 3U CubeSat includes specifying a duty cycle, computing selecting series cell count for bus voltage, calculating required panel area using PV efficiency and LEO irradiance, and verifying converter current limits and thermal constraints. Sensitivity analysis that varies DoD, degradation rate, and off-pointing quantifies margins and identifies dominant risk drivers for the EPS design.

step-by-step narrative for converting mission requirements into battery and solar array specifications, shows a compact worked example for a representative 3U CubeSat, and highlights the few sensitivity drivers that most affect final sizing. Start by collecting the inputs that define energy demand and supply: a complete load manifest with each device's power and runtime per orbit, orbital parameters that determine sunlight and eclipse durations, attitude performance that governs average incidence of sunlight on panels, and system policy choices such as bus voltage, allowed depth-of-discharge and an end-of-life capacity margin. Also record realistic efficiency estimates for power electronics and harness losses so that the energy you calculate is net usable energy rather than idealized production.

Begin the sizing process by converting the load manifest into energy per orbit. For every payload, transmitter burst, housekeeping task and heater interval compute energy in watt-hours by multiplying power by hours of operation. Separate these energies into eclipse energy, which must come from the battery, and sunlight energy, which can be supplied directly by the panels and by battery recharge. The eclipse energy becomes the primary starting point for battery sizing: add a contingency margin to account for unexpected events, then divide that required usable energy by your chosen allowable DoD and by the remaining capacity fraction after applying an aging margin. Converting the resulting pack energy into ampere-hours is a simple division by the bus voltage; practical designs round that number up to the nearest commercial pack size and include BMS mass and volume in the payload budget.

Solar array sizing follows from the orbit-level energy balance. Compute how much energy the panels must provide during sunlight to both power immediate loads and to recharge the battery fully for the next eclipse. To turn required energy into area, determine the average in-plane irradiance during sunlight for your chosen panel orientation; multiply that by panel efficiency, then apply net system efficiency factors that capture MPPT and converter losses, harness resistance and connector inefficiency. Multiply usable power per square meter by sunlight time per orbit to obtain usable watt-hours per square meter per orbit and divide the required sunlight energy by that number to get active panel area. Where deployable are considered, repeat the same calculation for deployed geometry and include reliability and stowage mass in trade studies.

A compact worked example helps make the arithmetic concrete. For a typical 3U mission assume a 95-minute low-Earth orbit with 65 percent sunlight fraction, a bus voltage of 7.4 volts, a conservative DoD of 40 percent and a 20 percent end-of-life margin. Suppose the payload produces two short imaging bursts totalling ten minutes, transmissions total five minutes, housekeeping runs continuously at about 1.5 watts, and heaters consume a small amount of energy but run briefly in eclipse. Converting these activities into watt-hours per orbit yields small eclipse energy (under 1 Wh) and a few watt-hours of sunlight load. Add a contingency of roughly 20–25 percent to the eclipse requirement, divide by DoD and the remaining capacity fraction to obtain pack energy, then divide by the bus voltage to estimate ampere-hours. Typical outputs for this scenario produce a practical battery target in the 600–800 mAh range at 7.4 V after rounding and accounting for BMS overhead. For the array, using a realistic projection factor for fixed panels and a triple-junction cell efficiency near 28 percent, the calculation normally returns a required active area on the order of a few hundred square centimetres, which a 3U vehicle can provide by combining multiple faces or a small deployable.

Because many parameters are uncertain early in the program, perform sensitivity checks to find which assumptions most affect mass and area so you can focus engineering effort efficiently. The first major driver is pointing performance: average incidence on the panels directly scales the available irradiance, so improving ADCS or changing panel placement often reduces required panel area more cost-effectively than marginally increasing battery capacity. The second key driver is the thermal heater budget; heater duty cycles during extended cold periods produce large increases in eclipse energy and therefore drive both battery and panel growth. The third major lever is the DoD and aging policy: choosing a conservative DoD for long missions markedly increases battery capacity, while accepting deeper cycling reduces battery mass but shortens usable life. Smaller but still relevant drivers include MPPT and harness efficiency and assumed end-of-life degradation of cells; improving converter efficiency or reducing wiring loss saves a few percent but is worthwhile on tight budgets.

If thermal design can reduce heater energy by passive insulation or conductive coupling, do that before increasing battery capacity. When area is strictly limited, prioritize higher-efficiency cells even at higher cost. Document all assumptions—sunlight fraction, projection factor, DoD, aging percent, MPPT and harness efficiency—so the sizing can be re-run quickly whenever mission parameters change. Following this straightforward methodology and concentrating effort on the few high-impact sensitivities turns an initially uncertain EPS sizing task into a controlled engineering process that yields reliable, testable battery and panel specifications.

12.2 Inputs: load manifest orbit ADCS margins

Collect a complete load manifest listing each subsystem's nominal power, duty cycle, and whether its energy must be served in sunlight or eclipse. Record orbit parameters (altitude, inclination, orbital period, beta angle) to compute sunlight fraction, eclipse duration and the Sun vector time series. Specify the ADCS profile (sun-pointing, nadir-pointing, tumble statistics or worst-case pointing error) to convert Sun vector into incidence on each panel face. Define conservative margins for MPPT and harness losses, manufacturing variability, and end-of-life degradation; document Depth-of-Discharge (DoD) policy and aging margin to be used in battery calculations. These inputs form the single source of truth used by both sizing calculations and the time-domain simulator.

12.3 Battery sizing procedure and worked 3U example

Compute the battery requirement from the eclipse energy: sum all loads that must be powered during eclipse to get Wh per orbit, add contingency margin, then divide by the allowed usable fraction (DoD × (1 − aging margin)) to obtain required pack Wh. Convert Wh to Ah at the chosen bus voltage and round up to a commercial pack size while adding BMS mass/volume. Example 3U: with a 95-min orbit and 65% sunlight, small eclipse energy <1 Wh plus housekeeping yields an eclipse demand; using DoD = 40% and 20% EOL margin typically results in a practical target around 600–800 mAh at 7.4 V after rounding and BMS overhead. Always validate C-rate acceptance to ensure the battery can absorb recharge energy in the available sunlight window.

12.4 Solar array sizing procedure and worked example

Determine required sunlight energy per orbit: add immediate sunlight loads plus energy needed to recharge the battery for the next eclipse. Compute average in-plane irradiance for each facet using the Sun vector and panel geometry, adjust by panel efficiency and incidence/angular response, then apply system efficiencies (MPPT, converter, wiring). Convert required Wh into area by dividing by Wh/m² per orbit (incident flux × panel efficiency × sunlight hours). Worked 3U example: using triple-junction cells (~28% efficiency) and fixed panels with realistic projection factors, the result is typically a few hundred cm² of active area distributed across faces or via a small deployable to meet the energy budget.

12.5 Sensitivity analysis pointing heaters DoD degradation

Run sensitivity sweeps to find which assumptions most affect mass and area: vary pointing performance (average incidence), heater duty cycles (cold seasons greatly increase battery demand), DoD policy (conservative DoD increases battery mass markedly), and EOL degradation (10–30% shifts panel area and margins). Quantify outcomes as delta-Ah and delta-cm² for each parameter change, rank drivers by impact, and focus design effort on the top drivers (typically pointing, heater budget, then DoD). Use tornado or spider plots to visualise sensitivities and to justify trade decisions to stakeholders.

12.6 Iteration trade studies and practical rules of thumb

Iterate sizing across best, nominal and worst scenarios to produce a compact trade table of battery sizes, panel areas and heater budgets; use hardware test data to replace conservative guesses and tighten margins. Prioritise system-level levers first: improving ADCS or panel placement often reduces required area more cost-effectively than adding battery mass, and passive thermal improvements reduce heater energy before increasing capacity. Practical rules of thumb: prefer DoD 30–40% for multi-year missions; validate battery C-rate for fast recharge; invest in higher-efficiency cells when area is constrained; and always document assumptions so sizing can be quickly re-run as mission parameters evolve.

CHAPTER 13

CHALLENGES, RISK MITIGATION AND FUTURE TRENDS

CubeSat electrical power systems operate at the intersection of hard physical limits and aggressive mission demands. The principal technical challenge is the energy-area trade-off: payloads and communications grow more capable while available surface area for photovoltaic generation is fixed or only modestly expandable, forcing difficult choices between higher peak performance and longer lifetime. Thermal extremes and low thermal inertia amplify this problem; batteries and power electronics face rapid, large temperature swings that reduce charge acceptance, increase internal resistance and accelerate aging. Radiation in low Earth orbit and higher inclinations degrades solar cells and semiconductor performance over time, producing steady loss of generation and sometimes sudden electronic faults. Limited volume and mass budgets constrain shielding and redundancy, so designers must accept higher single-component importance and plan for graceful degradation rather than assuming full hardware redundancy. Finally, operational realities—unpredictable pointing errors, deployment failures, connector intermittency and compressed development schedules on student or small commercial teams—multiply these technical constraints into program risks.

Risk mitigation therefore becomes a systems engineering discipline rather than a set of isolated fixes. The most effective mitigations address the highest-impact failure modes first: improve energy yield per unit area, reduce the energy consumed by thermal control, and protect the battery from lifespan-reducing behaviours. Improving energy yield is often best achieved by better attitude control and panel placement rather than simply adding more area: modest ADCS upgrades that reduce average incidence angle can cut required panel area more effectively than marginally larger deployable. Where area is irreducibly limited, investing in higher-efficiency, flight-qualified multi-junction cells and low-loss harness design yields outsized benefits. Thermal risk is mitigated through early co-design of structure and EPS: use passive insulation, conductive coupling to distribute heat, and place batteries away from heat-producing electronics so heaters run less often. Battery-centric mitigations include conservative Depth-of-Discharge policies, active cell balancing, temperature-aware charge profiles and selecting cells with validated low-temperature charge acceptance and radiation tolerance for the mission duration.

Protection, verification and operations complete the risk-reduction strategy. Layered protection (active LCLs, staged UVP, distributed branch isolation) prevents single faults from cascading while keeping a communications and telemetry path alive for diagnosis. Instrument protections with telemetry and non-volatile fault logs so ground teams can reconstruct events and tune thresholds post-flight. Aggressive ground testing—fault injection, thermal-vacuum testing with representative duty cycles, vibration and connector stress tests—uncovers failure modes that simple functional tests miss. For small teams, rely on flight-proven commercial EPS modules when schedule or expertise is limited, and reserve custom development for mission-critical niches where commercial modules cannot meet constraints. Finally, operational mitigations matter schedule high-power tasks in sunlight, implement onboard autonomous safe modes that shed nonessential loads predictably, and plan regular health-check telemetry so degradation trends are detected early.

Adoption of these advances must be tempered by rigorous qualification: GaN parts require radiation and thermal cycling validation, new battery chemistries demand exhaustive aging and safety testing, and ML-driven autonomy needs robust failure-mode analysis and explainability to be trusted in-flight. The near future for CubeSat EPS is therefore hybrid: incremental hardware improvements (GaN converters, high-efficiency panels, hybrid storage) combined with smarter software and stricter test discipline will together expand mission capability without abandoning the conservative engineering practices that make small satellites reliable. Teams that pursue a balanced program of targeted technology adoption, early cross-disciplinary co-design, and comprehensive testing will achieve the best outcomes—longer missions, higher payload return and resilient operations even in constrained form factors.

13.2 Key technical challenges (energy-area, thermal, radiation)

CubeSat EPS design faces three major technical constraints: limited surface area for solar panels, harsh thermal environments, and radiation exposure in low Earth orbit. The small form factor restricts how much energy can be harvested, especially during eclipse-heavy orbits or off-nominal pointing. Thermal swings between sunlight and eclipse affect battery performance, converter efficiency, and MPPT stability, requiring careful thermal design and heater budgeting. Radiation degrades solar cells, increases battery internal resistance, and can cause latch-up or bit-flips in EPS electronics. These challenges demand conservative margins, robust component selection, and fault-tolerant architecture to ensure mission survival and performance.

13.3 Systems-level risk mitigation strategies

To counter these risks, CubeSat designers adopt layered mitigation strategies across hardware, software, and operations. Hardware-level protections include LCLs, UVP, thermal fuses, and redundant sensing. Software mitigations involve load-shedding policies, adaptive scheduling, and safe-mode recovery logic. Operational strategies include conservative depth-of-discharge limits, seasonal heater planning, and telemetry-based health monitoring. Design margins are applied to panel sizing, battery capacity, and thermal budgets to absorb uncertainties. Cross-subsystem coordination—such as aligning heater duty cycles with MPPT surplus windows—helps optimize energy use and extend mission life without hardware changes.

13.4 Verification, operational and procurement mitigations

Verification ensures that models match reality and that EPS components behave predictably under stress. Thermal-vacuum tests, fault-injection campaigns, and hardware-in-the-loop simulations validate performance across temperature, radiation, and load conditions. Operational mitigations include pre-launch calibration, in-flight telemetry tuning, and contingency plans for degraded modes. Procurement mitigations involve selecting flight-qualified components, reviewing vendor test data, and derating specs for space conditions. Early integration of EPS with payload and ADCS helps avoid late-stage conflicts and ensures that energy budgets are realistic and achievable.

13.5 Software trends (ML health monitoring, adaptive scheduling)

Software is increasingly used to enhance EPS resilience and efficiency. Machine learning algorithms analyse telemetry to detect early signs of degradation, predict failures, and optimize charge/discharge cycles. Adaptive scheduling software aligns high-power tasks with predicted sunlight windows, delays noncritical operations during low-energy periods, and dynamically adjusts load priorities based on battery state and mission goals. These trends shift EPS from static power delivery to intelligent energy management, improving mission reliability and extending operational life without hardware changes.

CHAPTER 14

CONCLUSIONS

A CubeSat's Electrical Power System is the mission enabler: its design determines what the spacecraft can do, how long it lasts, and whether it survives faults and the space environment. Across the report we examined generation, conversion, storage, distribution, protection, thermal control, sizing methodology and verification practices; the recurring theme is that successful EPS design is a systems exercise that balances constrained geometry and mass against mission demands through conservative margins, careful integration, and thorough testing. High-efficiency photovoltaic cells, low-loss MPPT and converter architectures, conservative battery DoD policies with active balancing, layered protection (LCL and UVP), and integrated thermal strategies together form the practical toolkit for reliable CubeSat power systems.

High-efficiency triple-junction solar cells deliver superior power density for volume-constrained CubeSats, and per-panel MPPT enhances energy harvesting in partial-illumination scenarios. Conservative battery management with limited depth-of-discharge and active thermal control extends operational life, and a hybrid protection architecture that combines distributed LCL and UVP with centralized monitoring reduces single-point failure risk. Thorough margining for degradation, off-pointing, and thermal effects and comprehensive qualification testing across electrical, thermal, vibration, and EMC domains are essential to mission success. Emerging technologies such as GaN converters, solid-state storage, and AI-based prognostics present promising improvements provided they undergo rigorous space qualification.

If you want these chapters converted into full multi-page narrative sections suitable for an 80-page report, I will generate Chapter 2 through Chapter 13 as expanded chapter text now.

Design decisions must be driven by a documented mission power budget and realistic orbital parameters so that every trade—higher cell efficiency versus deployable complexity, larger battery mass versus deeper DoD and shorter life, distributed protection versus centralized simplicity—is evaluated in terms of mission risk and operational impact. Early co-design with ADCS and structure pays large dividends: improving average incidence on panels often reduces required area far more cost-effectively than adding hardware, and good mechanical routing and thermal coupling reduces heater energy and extends battery life. Protection and autonomy are equally important; telemetry-rich, active protection devices combined with staged, reversible recovery sequences let a CubeSat survive transient faults and provide the data needed for ground diagnosis.

Verification and validation convert a theoretically sound design into flight readiness. Functional testing, fault injection, thermal-vacuum cycling and vibration testing under representative duty cycles reveal interactions and hidden failure modes that simple component-level tests miss. Documented FMEA, clear acceptance criteria tied to mission survivability, and repeatable test procedures make the difference between a design that works in simulation and one that survives launch and operations. For small teams, leveraging flight-proven EPS modules and focusing custom work where mission needs demand it reduces schedule and technical risk while preserving the opportunity to innovate in control algorithms, thermal design or integration approaches.

Looking forward, incremental adoption of technologies such as GaN power electronics, hybrid energy storage and smarter onboard energy management will expand CubeSat capabilities but must be pursued with disciplined qualification.

The highest returns often come from system-level improvements—better pointing, smarter scheduling, tighter thermal design—rather than from a single component upgrade. Teams that couple cautious technology adoption with strong systems engineering, early integration testing and operations-aware design will deliver the most resilient and scientifically productive CubeSat missions.

CHAPTER 15

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- ✓ Recent conference papers and IEEE TPEL special issues on GaN/SiC converters for space, hybrid battery + supercapacitor systems for pulsed loads, and AI/ML for health monitoring (search IEEE TPEL, IEEE Aerospace conferences, and recent small-sat conferences).
- ✓ Review articles on solid-state and advanced battery chemistries for space applications (battery research journals and conference proceedings).
- ✓ Conference proceedings: AIAA/IEEE Aerospace, IEEE PEDES, IAC small satellite sessions, and IEEE Aerospace Conference often include CubeSat EPS research.

How to find and cite these quickly

- ✓ IEEE Xplore: search terms “CubeSat EPS”, “CubeSat MPPT”, “CubeSat battery reliability”, “per-panel MPPT”, “distributed EPS CubeSat”.
- ✓ NASA NTRS and ESA technical reports: search for “Spacecraft Electrical Power Systems” and CubeSat EPS guidance.
- ✓ Google Scholar queries for the review articles above (Edpuganti 2022 is a high-value starting point; use its references to expand your bibliography).

- ✓ Conference proceedings: AIAA/IEEE Aerospace, IEEE PEDES, IAC small satellite sessions, and IEEE Aerospace Conference often include CubeSat EPS research.

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