

Standard

Electrical Power Systems for Unmanned Spacecraft

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Standard

Electrical Power Systems for Unmanned Spacecraft

Sponsored by

American Institute of Aeronautics and Astronautics

Approved

5 January 2007

Abstract

This document, when followed in its entirety, will yield a robust EPS design suitable for very high-reliability space missions. This document specifies general design practices and sets minimum verification and validation requirements for power systems of unmanned spacecraft. The focus of the document is on earth orbiting satellites using traditional photovoltaic/battery power, but does not exclude other primary power generation and storage methods. This document does not address specific launch vehicle requirements however much of the design philosophy used here is applicable to launch vehicle power systems.

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Foreword

This standard has been developed under the auspices of the Chief Engineer's office of the United States Air Force Space and Missile Systems Center (SMC). It is one in a series of standards planned to codify industry best practices to ensure the very highest level of performance and reliability for the next generation of high-reliability military satellites. It follows in the footsteps of MIL-STD-1539, "Electrical Power, Direct Current, Space Vehicle Design Requirements," but has been thoroughly updated to reflect current industry design practices. It also contains new power quality requirements to replace those that were formerly called out in MIL-STD-1541A.

The starting point for this document was "Electrical Power Systems Direct Current Space Vehicle Requirements," TOR 2005(8583)-2, developed by Brian Lenertz of the Power and Analog Engineering Section of the Engineering and Technology Group at The Aerospace Corporation. Substantial improvements and additions were made to the TOR content based on the experience of the committee.

For missions that do not require all the provisions of this standard, it is understood that tailoring may be performed to balance performance, risk, and cost to suit the needs of a particular program.

The final product reflects the best design and verification practices for high-reliability satellite electric power systems. For almost all requirements, consensus was readily achieved. Three requirements, found in sections 5.2.11.1 (b), 5.2.11.3, and 5.5.4.1.7, were prescribed to meet specific needs for some government missions.

At the time of approval, the members of the AIAA Power Systems Committee on Standards were:

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The above consensus body approved this document in December 2006.

The AIAA Standards Executive Council (Mr. Amr ElSawy, Chairman) accepted the document for publication in January 2007.

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In formulating, revising, and approving standards publications, the committees on standards will not consider patents that may apply to the subject matter. Prospective users of the publications are responsible for protecting themselves against liability for infringement of patents or copyright or both.

1 Scope

This document, when followed in its entirety, will yield a robust EPS design suitable for very high-reliability space missions. This document specifies general design practices and sets minimum verification and validation requirements for power systems of unmanned spacecraft. The focus of the document is on earth orbiting satellites using traditional photovoltaic/battery power, but does not exclude other primary power generation and storage methods. This document does not address specific launch vehicle requirements however much of the design philosophy used here is applicable to launch vehicle power systems.

2 Tailoring

When viewed from the perspective of a specific program or project context, the requirements defined in this Standard may be tailored to match the actual requirements of the particular program or project. Tailoring of requirements shall be undertaken in consultation with the procuring authority where applicable.

NOTE Tailoring is a process by which individual requirements or specifications, standards, and related documents are evaluated and made applicable to a specific program or project by selection, and in some exceptional cases, modification and addition of requirements in the standards.

3 Applicable Documents

The following documents contain provisions, which, through reference in this text, constitute provisions of this standard. For dated references, subsequent amendments to, or revisions of, any of these publications do not apply. However, parties to agreements based on this standard are encouraged to investigate the possibility of applying the most recent editions of the normative documents indicated below. For undated references, the latest edition of the normative document referred to applies.

DoD-W-83575A	Wiring Harness, Space Vehicle, Design and Testing, General Information For
AIAA S-111	Qualification and Quality Requirements for Space Solar Cells
AIAA S-112	Qualification and Quality Requirements for Space Solar Panels
MIL-STD-1547B	Electronic Parts, Materials and Processes
MIL-STD-1543B	Reliability Program Requirements For Space & Launch Vehicles
MIL-STD-1629A	Procedures for Performing FMECA
AFSPC Man 91-710	Range Safety User Requirements
MIL-STD-975	NASA Standard Electrical, Electronic, and Electromechanical (EEE) Parts List
MIL-STD-1528A	Manufacturing Management Program
MIL-STD-1522	Standard General Requirements for Safe Design
MIL-STD-461C	Electromagnetic Emission and Susceptibility Requirements for the Control of Electromagnetic Interference

4 Vocabulary

4.1 Acronyms and Abbreviated Terms

AFSPC	Air Force Space Command
AIAA	American Institute of Aeronautics and Astronautics
ASTM	American Society for Testing and Materials
BOL	Beginning of Life
C&DH	Command and Data Handling
CDR	Critical Design Review
CIC	Coverslide Interconnect Cell
CoDR	Conceptual Design Review
CS	Conducted Susceptibility
DC	Direct Current
DoD	Department of Defense
DOD	Depth of Discharge
DRC	Design Reference Cases
EBA	Energy Balance Analysis
EDTL	Equipment Design Transient Level
EEE	Electrical, Electro-mechanical, and Electronic
EGSE	Electrical Ground Support Equipment
EIDP	End Item Data Package
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EOL	End of Life
EPS	Electrical Power Systems
ESD	Electrostatic Discharge
EWR	Eastern and Western Range
FAA	Federal Aviation Administration
FMECA	Failure Modes Effects and Criticality Analysis
GEO	Geosynchronous Earth Orbit
GSE	Ground Support Equipment
GTO	Geosynchronous Transfer Orbit
HEO	Heliocentric Earth Orbit

I/V	Current vs Voltage
LEO	Low Earth Orbit
Li-Ion	Lithium Ion
MEO	Medium Earth Orbit
MPG	Multipoint Ground
MSE	Minimum Stored Energy
NASA	National Aeronautical & Space Administration
NiH ₂	Nickel Hydrogen
P/L	Payload
PARD	Periodic and Random Deviation
PDR	Preliminary Design Review
PMAD	Power Management and Distribution
PMP	Parts, Material, and Processes
PMPCB	Parts, Material, and Processes Control Board
PRR	Preshipment Readiness Review
PWFSW	Power Management Flight Software
RE	Radiated Emissions
SAS	Solar Array Simulator
SDR	System Design Review
SEE	Single Event Effects
SOC	State of Charge
SPG	Single Point Ground
SV	Space Vehicle
TAYF	Test As You Fly
TBT	Thermal Balance Test
TMM	Thermal Math Model
TOR	Technical Operating Report
TRL	Technology Readiness Level
UV	Ultraviolet
VCRM	Verification Cross Reference Matrix
VRP	Voltage Reference Point
VRS	Voltage Reference Scheme

4.2 Terms and Definitions

For the purposes of this document, the following terms and definitions apply.

Abnormal Operation

encompasses unforeseen circumstances that are not handled via established contingency plans and operational states

Actual Battery Capacity

beginning of life (BOL) capacity that is available in ground test under the expected on-orbit operating conditions

NOTE Actual battery capacity may be either higher or lower than rated capacity. Actual capacity is determined at specific operating conditions of temperature, age and load. It may be determined by analysis where sufficiently proven models exist and have been accepted. On orbit measurement of actual capacity generally involves complete discharge of the battery.

Branch Circuit

conductors which run between a load switching center and the load itself

NOTE The impedance of this bus is only common to any parallel loads in the same load.

Bus Voltage

nominal DC voltage at the main bus or at any distribution point

Common Impedance

any impedance shared by two or more loads

Contingent Load Power

for the entire spacecraft, the sum of the individual load estimates (each including maturity factors) plus corresponding losses plus the highest expected battery recharge power

NOTE Also called load contingency.

Contingent Source Power

nominal predicted power generation capability times an uncertainty factor that is based on the maturity of the solar arrays and of the analysis estimate

NOTE also called source contingency.

Credible Failure

mode of failure that has a probability of occurrence within a single year of greater than one in a million

NOTE Spacecraft structure is not considered a credible failure. Generally, non-moving, non-electrical parts are not considered credible failure modes. Insulation layers or insulating devices can be considered credible failure modes in the presence of sufficiently high breakdown voltages, mechanical stresses, or contamination.

Depth of Discharge (DOD)

ratio of the number of Ampere-hours removed from a battery (under specific operating conditions) to the nameplate or rated capacity of the battery, times 100

NOTE: For batteries that are subcharged, that is, not recharged to full capacity, DOD is the percentage of capacity expended in a discharge from the subcharged point. For example, a battery that is subcharged to 70% state of charge (SOC) and then cycled down to 40% SOC is considered to have cycled over 30% of its capacity and the DOD is 30%.

Design Reference Cases (DRCs)

example mission or set of operational conditions that are used in an analytical or simulation setting to show that a design meets or exceeds its performance requirements

NOTE A finite set of DRCs may be formulated to cover the worst-case behavior of the system under all operating modes and conditions.

Design Verification

all activities, including test, analysis, simulation, and inspection, that are performed to verify that a design meets its specified requirements

Distribution Point

location in the power distribution element where the power wiring branches to power two or more pieces of loads, not counting heaters

NOTE The source impedance of a distribution point is that measured “looking back” toward the main bus with the distribution point unloaded. It is the impedance at the main bus (including other loads) plus the wiring impedance to the distribution point and any capacitive loading added at the distribution point.

Double Insulation

two independent insulating devices or layers of insulating material of sufficient thickness and durability to avoid damage due to electrical or mechanical stresses

NOTE 1 Insulating layers are used in such a way that it would take failure of both devices or layers to result in a short circuit of the unfused power to ground or any other conductor

NOTE 2 This technique is most often used to protect unfused power against shorts to ground or other conductors

Electrical Power Subsystem (EPS)

set of all equipment, wiring, and EPS-controlling software whose task is the generation, storage, control, and distribution of electrical energy to the input power terminals of the loads

Energy Balance

balance between energy generated or available and energy consumed or wasted over a defined orbital period

Energy Reserve

total amount of usable energy (in Wh), remaining in a battery that has been discharged to the maximum allowed DOD under normal operating conditions

NOTE Energy reserve provides enough energy to ensure positive energy balance during the maximum sun-outage time when a loss of attitude control occurs coincident with the end of the longest eclipse. Energy reserve may also be used for other rare, deep discharges such as relocation with electric propulsion, or those that may occur in transfer orbit.

Energy Storage

devices that store energy for use in powering the loads during periods (such as eclipse) when the output of the power generation element is insufficient to meet the overall load demand

NOTE Secondary (rechargeable) batteries are the prevalent means of energy storage, but ultracapacitors, flywheels, fuel cells, and primary (non-rechargeable) batteries also fall into this category.

EPS Software

software (or firmware) that performs control functions for any aspect of EPS operation, whether it is contained within EPS equipment or in some other piece of spacecraft equipment, and whether or not it is stand-alone or part of some other piece of software

Essential Loads

loads that are essential for minimum controllability and commandability of the spacecraft

EXAMPLE Examples of essential loads are command receivers, on-board computers, and critical heaters.

Fault Management

process of detecting and reacting to the occurrence of a fault or anomaly, whether in hardware or software

Feeder Bus

conductors which run between the main bus and a load-switching center

NOTE The impedance of this bus is common to all loads powered by the load-switching center into which it terminates.

Ground Support Equipment (GSE)

non-flight items used in the ground testing of the EPS as integrated on the spacecraft, either at the contractor facility or at the launch site

Grounding System

all wires, structures, and connections that determine the return current paths in the EPDS

NOTE The layout of the VRS is designed to avoid interference between users of electrical power and to assure meeting conducted electromagnetic compatibility (EMC) requirements.

Impulse Strength

integrated volt-second product of a transient surge or spike measured relative to the nominal voltage before and after the transient

Loads

device or unit that uses electrical power provided by the EPS

NOTE Commonly called "loads" or "payload units." Units and devices comprising the EPS components are themselves considered part of the loads in that they also consume power and are subject to EMC requirements in addition to their main purpose of steering electrical energy throughout the space vehicle. Battery recharge power is also considered part of the loads.

Load Groups

physical or logical partitioning of one or more loads

NOTE A physical load group may share a common power harness, location, or distribution hardware; a logical load group may be a set of loads to be turned off in safe-hold or survival mode.

Load Margin

power margin divided by the contingent load power, often stated as a percentage

NOTE Load margin represents the additional percentage of load power the source could provide and still recharge the batteries.

Lock-up (also called latch-up)

condition wherein the solar array is operated at a point well below its maximum power point such that the solar array output power is insufficient to power the load and recharge the battery

NOTE Lock-up arises in EPS architectures that have multiple stable operating points determined by combined solar array, battery charge controls, battery, and load characteristics. When lock-up occurs, system operation is at an operating point below the solar array maximum power point, resulting in insufficient battery recharging and, possibly, some battery discharge.

Losses

distribution losses and energy storage system inefficiency losses associated with supporting the spacecraft electrical loads

Main Bus

main conductors to which the sources are attached prior to any branching

NOTE The impedance of this bus is common to all loads.

Mission Life

the contractually required period of time over which the spacecraft must meet all of its performance requirements

NOTE Mission life includes transfer orbit, orbit-raising, on-orbit, and de-orbit modes. EPS required life or an EPS component required life can be longer than the spacecraft mission life requirement. A design life can be longer than a required life.

Multipoint Ground

allows use of structure as a low-impedance return path for currents

NOTE Care must be taken to avoid large DC currents that can interfere with low-level circuitry. Multipoint grounding offers some advantages for high-frequency subsystems.

Nonessential Loads

loads that can be powered off without adversely affecting the minimum controllability and commandability of the spacecraft

Normal Operation

range of operational states of the space vehicle that exist or occur by design and in which the vehicle spends the vast majority of its time, according to the expectations of the mission designers and planners

Operational States

all foreseeable and intentional combinations of states, modes, or conditions within the EPS hardware and software

Payloads

self-contained instrument, sensor, or device that fulfills some mission objective

Periodic and Random Deviation (PARD)

instantaneous time-domain deviation of a voltage about a nominal value as observed on an oscilloscope with a specified bandwidth and stated as a peak-to-peak or root-mean-square value

Power Budget

method of accounting for the spacecraft's electrical loads and losses associated with these loads

NOTE Often, loads are expressed in terms of orbital average power. It may also contain, as required by the energy balance analysis, information on the time behavior of loads. This could be expressed in terms of an orbital duty cycle or a detailed, moment-by-moment power prediction. A power budget usually contains multiple load predictions corresponding to different mission modes or DRCs.

Power Control

all hardware and software used to control and steer electrical power from the power generating element, to and from the energy storage subsystem, and to the power distribution subsystem

Power Generation

all equipment involved in the generation of DC power for use by the loads and for charging the energy storage devices

NOTE Solar arrays are the most common technology for spacecraft power generation; other technologies, such as thermoelectric devices, fall into the category.

Power Interface Loop Gain

ratio of the source impedance to the load impedance

NOTE The loop gain is a complex function of frequency and includes both passive elements and active contributors such as the negative impedances of DC-converters.

Power Margin

difference between contingent source power and contingent load power

NOTE it is the margin between the lowest possible array output power and the highest possible load power.

Power Quality

acceptability of the time-domain variation in bus voltage induced by the periodic and aperiodic currents flowing to and from the loads and to self-generated currents and voltages from the EPS equipment itself and by the GSE during ground testing

Rated (or Nameplate) Battery Capacity

minimum capacity that is based on charging the battery at a predefined rate of charge and temperature and discharging it at the same temperature; a predefined rate of discharge and end of discharge voltage

Reconditioning

act of restoring lost capacity to a battery through a process of deep discharging and recharging per some prescribed charging regimen

Regulated Buses

one whose voltage is controlled by means of a closed-loop negative feedback control scheme

NOTE A sunlight-regulated bus is regulated during insolation and unregulated during eclipse.

Unregulated Buses

one whose voltage is approximately the same as the battery voltage, minus harness and switching losses

Safe-hold or Survival Mode

state entered into by the spacecraft (typically the result of an on-board fault detection response) that maximizes the probability that the spacecraft will survive until mission operators can restore normal operations

NOTE Transition to this state from normal operations typically includes the shedding of non-essential loads, and the use of redundant and/or back-up essential loads.

Single-point Failure

single component, wiring, or connector failure, software glitch, or computer failure that results in the permanent loss of the spacecraft's ability to perform its primary mission for the intended design life span

Single-Point Ground (SPG)

a commonly-used type of voltage reference scheme (VRS) in which DC return currents from the loads or ground planes are carried via low-impedance conductors back to a single grounding point

NOTE Use of structure for DC return currents is prohibited for an SPG-type VRS.

Source Power Margin

power margin divided by contingent source power

Spikes

narrow impulse-like voltage waveforms that are produced by switching or fault-clearing events

NOTE Spikes are generally measured in terms of their volt-second impulse strength and peak voltage amplitude.

State of Charge (SOC)

ratio of the number of Ah present in a battery (under specific operating conditions) to the rated capacity C(Ah) of the battery, times 100

Technology Readiness Level (TRL)

numerical ranking system for assessing the maturity of and relative risk of using a particular technology

NOTE For solar arrays employing crystalline solar cells, for the purposes of this document, TRL levels have been defined as follows:

- | | |
|---|---|
| 9 | Solar array has flown successfully on its intended mission |
| 8 | Solar array flight qualified on the ground or is flying successfully as primary power |
| 7 | Solar array prototype is flying successfully |
| 6 | Solar array has passed life-cycle and qualification test at the panel/coupon level for mission conditions |
| 5 | Solar cell/Coverglass-Interconnect-Cell (CIC) is space qualified |
| 4 | Solar cell/CIC is characterized at prototype level |
| 3 | Solar cell prototype has been fabricated |
| 2 | Solar cell design and modeling completed. Subcell components demonstrated |
| 1 | Solar cell conceptual design formulated |

The final determination of TRL may vary somewhat from the levels defined above, depending on mission duration and criticality, orbit, and other factors. The final determination of TRL is up to the procuring authority for a particular program.

Test String (I/V Measurement)

at least three solar cell assemblies, electrically connected in series, with end termination tabs

NOTE The test strings will be manufactured at the same time as the power generating strings. They will incorporate the same parts, materials and processes used to manufacture the power generating strings, and will be bonded to the flight panels using the same materials and processes used to bond the power generating strings.

Thermal Loads

dissipative heaters used for temperature control of the spacecraft components

Transients

bus voltage time-domain response due to an aperiodic event or due to a periodic low-frequency train of events

Virtual Cell

group of cells wired in parallel

Voltage Reference Point

a point, usually on structure, used as the reference from which other system voltage levels are calculated

NOTE On an unregulated bus it is at the output of the box, which establishes the main power bus. On a regulated bus it is the output of the main bus regulator.

Voltage Reference Scheme

all wiring, structures, and connections that determine the return current paths in the EPS, designed to avoid interference between loads and to meet EMC requirements

EXAMPLE Single Point Ground and structure-return are two examples of VRS

Voltage Ripple

cyclic variation of voltage about the mean level of the DC voltage during steady-state operation of the EPS

NOTE The ripple generally contains multiple frequency components as well as small spikes outside the average envelope of the ripple. Overall ripple is measured in RMS or peak-to-peak volts, while the spikes are generally measured in terms of their volt-second impulse strength and peak voltage amplitude.

5 Purpose of EPS

The electrical power system (EPS) of a space vehicle shall be designed to ensure the reliable delivery of electrical power compatible with all loads under all foreseeable operational states and environments, during all mission phases, and over the intended design life of the loads and of the space vehicle.

5.1 Functional Description

Electrical power is used by all active spacecraft systems and equipment for their operation. Direct Current (DC) electrical power at different bus voltages is normally utilized for this purpose. All DC EPS include power generation, energy storage, power management, and distribution as well as protection devices and load shedding controls.

The EPS of a spacecraft is comprised of the hardware and software used to generate, store, condition, and distribute power as required by the loads; it performs these functions throughout all mission phases in the presence of all environments encountered.

The sizing and design of all power system components take into account the spacecraft orbit and location, mission phases and durations, modes of orbit transfer, natural and man-made environmental effects, and other mission-specific requirements such as orbital relocation and solar shadowing events.

5.1.1 Power Generation/Energy Conversion

Power generation involves all equipment used in the generation of DC power for use by the loads and for charging the energy storage devices.

Electric power can be generated by using solar photovoltaic, solar thermal, or nuclear energy conversion devices, depending upon the load power requirements and mission orbit and duration. The DC power systems specified in this standard assume arrays of photovoltaic solar cells which convert sunlight into useful electric power. The solar arrays are sized to satisfy average power demand (including battery recharge power) during operational life in sunlight. Further, the solar arrays are sized to satisfy the power contingency requirements from beginning to end of life, including losses of power due to statistically expected performance degradation and solar cell failures.

5.1.2 Energy Storage

Energy storage devices store some of the energy generated by the power generation element for use in powering the loads during eclipse periods when the output of the power generation element is insufficient to meet the overall load demand.

For spacecraft orbiting earth the energy is generally stored in secondary/rechargeable batteries. Flywheels and other energy devices such as capacitors can also be used for this purpose depending upon the spacecraft load power profile and any peak load requirements. This standard only covers the subsystem level performance requirements for secondary batteries. For non-battery systems the

principles inherent in this standard for battery systems shall be applicable to the greatest extent possible, whenever a design or verification requirement can meaningfully be translated across technologies.

Batteries should be sized based on predicted worst-case mission requirements and expected capacity degradation over mission life. Batteries are designed to support the spacecraft through the launch sequence, including all anticipated contingencies and through all foreseen losses of solar energy during the mission, including those resulting from failures. Where system requirements dictate a battery tolerate a single fault, the battery is designed to operate with one cell either failed shorted or open. Provisions for monitoring the critical battery performance parameters and any reconditioning requirements are also included in the design. Battery thermal design should take into account maximum and minimum temperature of operation under all intended operating conditions and maximum allowed temperature gradients between different parts of the same cell, between cells or group of cells within a battery, between batteries, and as a result of any failure modes. The charging technique is designed to ensure that the depth of discharge and the recharge ratio applied are appropriate for the particular cell technology, temperature of operation, and mission life requirements.

5.1.3 Power Management and Distribution (PMAD)

PMAD generally involves power conditioning, regulation, control, and distribution and protection functions. It includes all essential solar array power and battery charge and discharge regulation electronics, bus under- and over- voltage detection and protection electronics, bus filtering to manage power-bus frequency and transient response requirements, and all EMI/EMC-related performance requirements such as bus voltage and current ripples at both power input and output ports. The solar array power regulators could either be shunt or series or peak power trackers. PMAD also includes the essential command input and telemetry output functions to enable functional/performance verification of the EPS during spacecraft assembly, integration, and test phase and to continue to monitor the EPS performance throughout the mission life. Commands are generally used to reconfigure the regulation and distribution functions to optimize the subsystem performance and reset any fault detection and protection electronics/devices. Telemetry generally includes bus voltage, power source and load currents, battery charge and discharge currents, power electronics operating temperatures, and status of fault detection and protection electronics. PMAD encompasses the spacecraft grounding requirements and implementation of the same. It includes sufficient functional or block level redundancy in order to be completely free of credible single point failures.

5.1.4 Flight Software

PMAD may also contain software, generally called the Power Management Flight Software (PMFSW) for battery charge and discharge management, load shedding, and command verification to ensure uninterrupted flow of power to and from energy storage and power conversion devices. It is also used for EPS telemetry data conversion, manipulation, and interpretation. PMFSW generally resides in the spacecraft on-board computer or located in a processor dedicated to the spacecraft electrical power subsystem only.

5.1.5 Harness

Power collection and distribution harnessing is an essential component of PMAD. No piece of harness can be utilized as a mechanical support. For transmission of power, each power line is twisted with its return and/or routed close to structure to minimize current loop area and harness inductance. The power distribution is protected in such a way that it eliminates failure propagation. The design of PMAD also takes into account the safety aspects as documented elsewhere in this standard and other pertinent documents.

5.2 Functional and Performance Requirements

5.2.1 Orbital Profile

The EPS architecture and design implementation thereof shall be selected to meet all mission requirements within the environments and conditions of the mission orbit type (LEO, MEO, GEO, or HEO), and within the ascent or transfer orbit and deployment environments and conditions.

5.2.1.1 Worst Eclipse

The EPS shall be designed to provide power to the loads during the worst eclipse for the mission orbit. The worst eclipse is a combination of eclipse length, seasonal sun intensity, beta angle, SOC at the beginning of the eclipse, and load levels during the eclipse.

5.2.1.2 Maximum Sun-outage Time

The EPS shall provide power to the required safe-hold loads during the maximum sun-outage time following the loss of vehicle attitude, even when this period begins at the end of the worst eclipse or other high-discharge events.

5.2.1.3 GEO-Specific Requirements

5.2.1.3.1 Battery Recharge

In normal operation during the eclipse season, the batteries shall be fully recharged, for the worst-case loading and longest eclipse, before the onset of the next eclipse.

5.2.1.3.2 Relocation (with electric propulsion)

- a) The EPS shall maintain positive energy balance while maintaining battery voltage above the minimum required for proper operation of the EPS during relocation operations with electric propulsion.
- b) The effect of relocation on battery life shall be included in the battery life data and analysis.

5.2.1.4 LEO-Specific Requirements

5.2.1.4.1 Thermal Cycles

- a) The number of thermal cycles for a LEO mission shall be defined as the mission life in days times the average number of Earth eclipses per day.
- b) For solar arrays, the total number of thermal cycles shall be this number plus the number of shadowing or off-sunpointing events over the mission life.

5.2.1.4.2 Energy Balance

- a) For some types of LEO missions, it is acceptable to adopt a strategy wherein the energy balance is either on a per orbit or the batteries are not fully recharged every orbital revolution.
- b) A justification for the strategy chosen based on cell or battery testing shall be provided.

5.2.1.5 Lunar Eclipses

The occurrences and characteristics (effective sun intensity as a function of time) of lunar eclipses shall be calculated over the designated mission life. It shall be shown via energy balance analysis that the EPS can maintain acceptable SOC and battery voltage during the lunar eclipse. Load-shedding strategies, if necessary, shall be included in the analysis. If the lunar eclipse occurs during solar eclipse season for the satellite, the effects of the previous solar eclipse shall be included in the analysis.

5.2.1.6 Re-entry disposal

For satellites that are de-orbited via atmospheric re-entry, the EPS shall be capable of providing power as needed to perform a targeted re-entry.

5.2.1.7 Environments

5.2.1.7.1 Radiation

The EPS design shall take into account all degradation and upset mechanisms due to natural radiation types and levels in the specified orbit.

5.2.1.7.2 Total Dose

The degradation of electronic parts parameters due to total dose radiation shall be taken into account in the EPS circuit-level worst case analysis.

5.2.1.7.3 Single Event Effects (SEE)

The EPS design shall take SEE into account such that no SEE shall cause the damage/destruction of any component or cause upset of critical functions that could result in spacecraft malfunction or loss of mission.

5.2.2 Mission Life

- a) The energy conversion, energy storage, and PMAD functions shall be designed and sized to provide reliable power as specified herein to the loads over the mission life.
- b) Appropriate power generation, energy storage, and load contingencies and margins, as well as fault protection and fault tolerance provisions, shall be included as specified herein in order to meet EPS reliability requirements specified at the system level.

5.2.3 Mission Phases

The EPS shall provide reliable power per the requirements specified herein during all operating modes, states, and mission phases, including ground processing, prelaunch, ascent, transfer orbit, deployment, normal operation, safehold, survival mode, de-orbit mode, and disposal.

5.2.4 EPS Architecture

- a) The EPS architecture shall be selected based on the mission level requirements and constraints.
- b) The EPS architectural trades shall be documented including quantitative performance comparisons over the different mission modes at the EPS System Design Review (SDR).

5.2.4.1 Voltage Choice

- a) The EPS power bus voltage shall be selected based on the mission power profile requirements, the energy conversion technology, and the energy storage technology.
- b) The EPS shall provide the distribution voltage to the electrical loads on the spacecraft per the power quality specified within this document.
- c) The interface with the electrical load shall be defined at the input to the user equipment at the end of the system cable.
- d) The EPS power bus voltage DC regulation shall be defined at the load interfaces including cable losses.
- e) The choice of bus voltage shall take into account the expected heritage bus and payload equipment that may be utilized.