A SunCam Online Continuing Education Course

Spacecraft
Electrical Power

Spacecraft Considerations / Distribution Design Guidance
Battery: Budget/Balancing/Sizing/Protection

by

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Nomenclature

AGND	Analog Ground
AWG	American Wire Gauge
BOL	Beginning of Life
BMS	Battery Management System
C	charge/discharge rate of battery in rated A-h
CAD	Computer Aided Design
COMSATS	Commercial Satellites
DC	Direct Current
DGND	Digital Ground
DoD	Depth of Discharge
EED	Electro Explosive Device
EGSE	Electrical Ground Support Equipment
EoCV	End of Charge Voltage
EoDV	End of Discharge Voltage
EOL	End of Life
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EPS	Electric Power System
FMEA	Failure Modes and Effects Analysis
FMECA	Failure Modes, Effects, and Criticality Analysis
G	forward transfer function (Gain)
GEO	Geosynchronous Earth Orbit
GSE	Ground Support Equipment
GSO	Geostationary Orbit
H	feedback transfer function
HEO	Helical Earth Orbit
ICD	Interface Control Documents
ISS	International Space Station

KPP	Key Performance Parameters
KSA	Key System Attributes
LEO	Low Earth Orbit
LIB	Lithium-Ion Battery
MEO	Medium Earth Orbit
NGO	Non-Governmental Organization
OMB	Office of Management and Budget
ORU	Orbital Replacement Unit
PMAD	Power Management and Distribution
PSIA	Pounds per Square Inch Absolute
p-s	parallel-series
QA	Quality Assurance
s	complex "signal" frequency
s-p	series-parallel
S/C	Space Craft
SOC	State of Charge
SOH	State of Health
SSO	Sun Synchronous Orbit
TAYF	Test as You Fly
TRL	Technology Readiness Level
WCA	Worst Case Analysis

Symbols

A	area
$A\text{-}h$	ampere-hours
C	charge rate
I	current
L	length
N	number
P	power
R	resistance
s	complex frequency
t	time
V	voltage
W	watt
ρ	resistivity
λ	wavelength

Subscripts

0	initial (zero value)
avg	average
BW	bundled wire
C	capacity
d	discharge
f	final / frequency
SW	single wire

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TABLE OF CONTENTS

INTRODUCTION

The spacecraft electrical power subsystem (EPS) provides generation, storage, management, and distribution of electrical energy to the bus and payload user loads. Satellite vehicle power is distributed between bus and payload user loads, which vary depending on the operating mode and orbital environments. Satellite orbit type (such as LEO or GEO), mission duration, and payload duty cycle are key factors in EPS architectural design for energy generation, storage, and power distribution. The electrical power subsystem comprises up to one-third of the total spacecraft mass and volume. [A][1]

The spacecraft EPS provides electrical energy to user loads under all mission operating conditions for the duration of the entire spacecraft life. Spacecraft vehicle power loads are distributed between bus housekeeping, payload, and heaters which are a function of the spacecraft operating mode and orbital environments, such as LEO or GEO.

The spacecraft bus generally defines the infrastructure of a satellite that supports the payload and subsystems. Most unmanned commercial satellite bus systems are comprised of an EPS, communications, telemetry and command, propulsion, attitude control, guidance, and thermal subsystems. As a mission critical subsystem, the EPS provides generation, storage, management, and distribution of electrical energy to the spacecraft power load users (Fig. 1).[2]

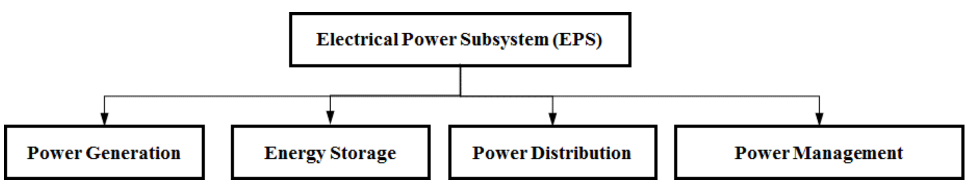

Figure 1: Fundamental Elements of a Typical Spacecraft Electrical Power Subsystem

[1] References will be shown in the "[*]" format. References contain indepth information of use for those directly involved with spacecraft design. Some are noted as export controlled or limited to those with special access. All information in this course is in the public domain.

[2] Some figures and material adapted from interactions and discussions with Thomas Barrera, PhD—at Boeing we were both Technical Fellows. His expertise is in batteries and mine resides primarily in electric power. I'd also like to acknowledge the depth of learning I accumulated by listening to Brian Reed, also a Boeing Technical Fellow, whose spacecraft knowledge is extensive and was generously shared.

A simplified functional block diagram of a spacecraft EPS is shown in Figure 2. Electric power for conventional COMSATS[3] can be generated by using either solar photovoltaic or solar thermal systems depending on spacecraft load power duty cycle, mission orbit, and spacecraft lifetime requirements. The most commonly used power generation device is the solar photovoltaic DC power generation system which employs arrays of photovoltaic solar cells to convert solar energy into electrical power. Earth-orbiting satellites ranging in power from watts (microsatellites) to hundreds of kilowatts (NASA-ISS) commonly employ photovoltaic solar array technologies as a DC current source[4] for meeting satellite user load demands. Energy storage includes secondary batteries—such as the lithium-ion batteries, flywheels, and energy storage devices such as capacitors. This course focuses primarily on lithium-ion battery power though the power distribution guidance is apropos to all different power sources.

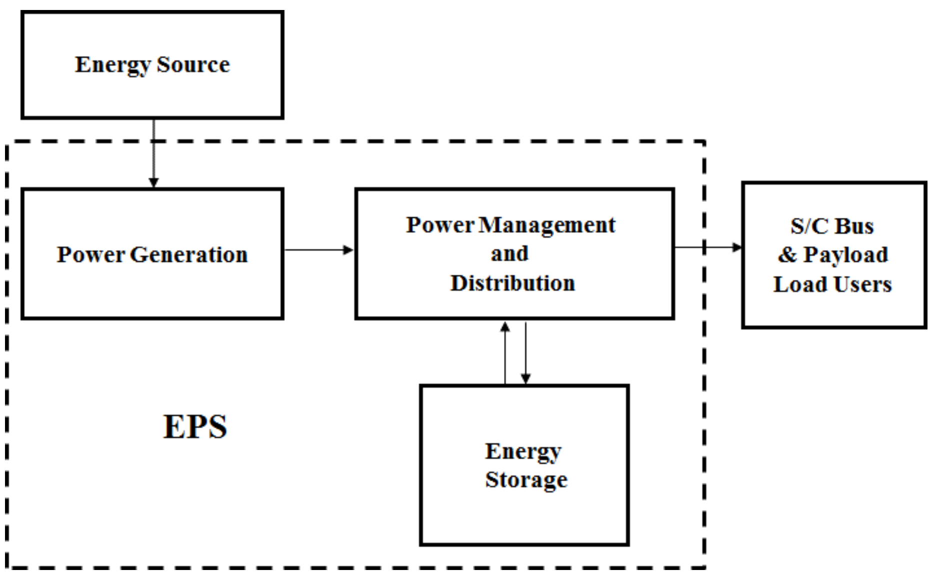

Figure 2: Earth-Orbiting Spacecraft EPS Functional Block Diagram

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[3] The term "COMSAT" is a trademark in the U.S. for the Commercial Satellite Corporation, which was established via the Communications Satellite Act of 1962 and freed NASA from developing such satellites; although, it has changed owners several times since then. COMSATS is also a generic term for any commercial satellite.

[4] Solar arrays are current-limited voltage sources. As the current increases the voltage drops significantly and in short timeframes. Batteries, by contrast, are voltages sources. As the current increases the voltage drops but over a longer time period. Since both are connected to the power bus, different methods are utilized to control each.

Electrical power management and distribution (PMAD) includes the regulation, control, and distribution of electrical power for the entire spacecraft. The PMAD is comprised of avionics for solar array power processing, battery management system (BMS) electronics for charge-and discharge-control and distribution wire harness for power distribution.

SPACECRAFT CONSIDERATIONS

Orbital Types and Requirements

The EE designer of the EPS system should have a fundamental (basic) grasp of orbits and their impacts. It is to that end for which this section is focused.

The primary drivers for satellite systems and subsequently the EPS, from a power perspective, are the total load, the desired lifetime, and the eclipse time. The eclipse time is the time a satellite will be in the Earth's shadow and will need to shift from solar panels to another source, for our purposes—a LIB storage system.[5] The Earth's shadow consists of two zones, the first is the penumbra that is a partial eclipse meaning some solar power generation is possible, though limited. The second is the umbra, which is the fully shaded inner region of the Earth's shadow.

The desired lifetime is limited by the type of orbit, the depth of discharge, the LIB chemistry itself, and other mundane items such as fiscal constraints.[6] The total load is not the simple sum of the spacecraft individual loads, but must be determined using mission operational requirements discussed more fully in Energy Balance and Battery Sizing and the Power System Requirements sections.

The main types of orbits are shown in Fig. 3. The sun-synchronous orbit (SSO), also called the heliosynchronous orbit (not shown) is a nearly polar orbit in which the satellites passes

[5] It may be an actual eclipse that is causing the loss of solar power as opposed to being in the Earth's shadow.

[6] I'm being a bit facetious here. The fiscal constraints are the starting argument for the possibilities available in any given mission.

over the same point on the Earth the same local solar mean time.[7] The differences between the GEO and GSO orbits are shown in Figs. 4 and 5.

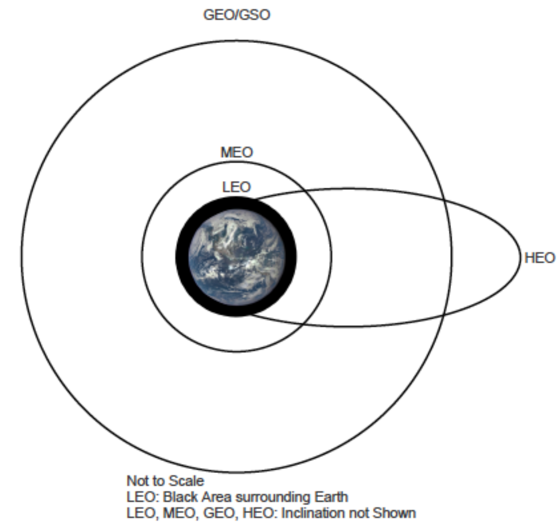

Figure 3: Earth Orbits

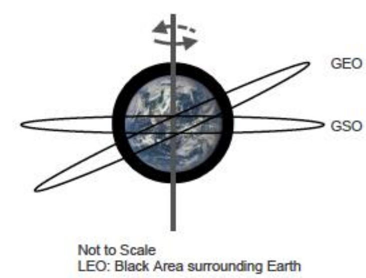

Figure 4: GEO and GSO Orbits

Figure 5: GEO and GSO Movement on Earth's Surface

[7] Such an orbit allows solar panels to provide energy continuously. Batteries are still required for backup & recovery.

Geosynchronous Earth Orbit

A geosynchronous orbit (GEO) is a prograde, low inclination orbit about Earth having a period of 23 hours 56 minutes 4 seconds. A spacecraft in geosynchronous orbit appears to remain above Earth at a constant longitude, although it may seem to wander north and south. The spacecraft returns to the same point in the sky at the same time each day. [B]

Geostationary Orbits

To achieve a geostationary orbit (GSO), a geosynchronous orbit is chosen with an eccentricity of zero, and an inclination of either zero, right on the equator, or else low enough that the spacecraft can use propulsive means to constrain the spacecraft's apparent position so it hangs seemingly motionless above a point on Earth. (Any such maneuvering on orbit, or making other adjustments to maintain its orbit, is a process called station keeping.) The orbit can then be called geostationary. This orbit is ideal for certain kinds of communication satellites and meteorological satellites. The idea of a geosynchronous orbit for communications spacecraft was first popularized by science fiction author Sir Arthur C. Clarke in 1945, so it is sometimes called the Clarke orbit. [C]

Orbital types and the important eclipse time are summarized in Table 1. Pay special attention to the eclipse time and the relative number of orbits as compared to the GEO/GSO. For example, for every one GSO orbit and LEO satellites goes around the Earth some 12 times.[8] This impacts the eclipse time, the depth of discharge, and the time available to recharge.

[8] This is an approximate value, based on averages, and should only be used as a thumb rule.

Table 1: Orbital Information [Varies by Source]

Orbit Type	Height [Reference: Earth's Surface]	Orbits & Eclipse
LEO	0 km – 2000 km[1] (0 mi – 1243 mi)	Circular Orbit 12 Orbits[2] Earth Eclipse Time[3] 35 min – 120 min
MEO[4]	>2000 km (1243 mi) <35,786 km (22 ,236 mi) 20,000 km Nominal	Circular Orbit 2 Orbits Earth Eclipse Time Up to 67 min 7 Weeks per Year
GEO[5] GSO[6]	35,786 km (22,236 mi)	Circular Orbit 1 Orbit 23 hr 56 min 4.09 sec Earth Eclipse Time 2 Periods 45/Days per Year Up to 72 Minutes
HEO	1000 km perigee 47,000 km apogee	Elliptical 6 Orbits Earth Eclipse Time 3-4 Weeks per Year Up to 67 Minutes

Table Notes:

1. LEO is defined by NASA as anything less than 2000 km. So, in some sources LEO is listed as 0 km – 2000 km.
2. Orbits listed are the number of orbits a satellite will do compared to one in GSO. Values are approximate and depend upon satellite altitude and inclination. Use these as approximate values or memory aides.
3. The Earth eclipse time (technically, the time in the Earth's shadow) depends upon the altitude, size of the Earth, orbital beta angle, which is the angle between the Earth & Sun, and the orbital plane (angle between the sun and the plane of the Satellite's orbit).
4. Many MEO satellites are at an orbit of 20,000 km where the orbital period is 12 hr. For example, this is the orbital height of the GPS system.
5. GEO satellites have some eccentricity and this move north and south along a single longitudinal line.
6. GSO satellites have an eccentricity of zero and this maintain a single position (point) on a longitudinal line.

The EPS designer needs to have a general understanding of the orbits and the typical constraints each brings. To that end, a summary of orbits follows.

LEO

Advantages/Features

(1) Propagation Delay less (than MEO and GEO satellites) (5 msec – 10 msec)
(2) Less power Required for communication
(3) Relatively small satellites (<500 kg)
(4) Link diversity better

Disadvantages

(1) Total Deployment Cost higher (than MEO and GEO Satellites)
(2) Doppler offset is higher

MEO

Advantages/Features

(1) Period of revolution 12 hours
(2) 24 satellite configurations in 6 orbits at 55º inclination allows a minimum of 4 satellites to always be visible from the Earth's surface
(3) Uniform global coverage
(4) Link diversity can be employed
(5) Propagation delay less than GEO/GSO satellites

Disadvantages

(1) Doppler offset more than GEO/GSO
(2) Network to control satellites more complex than GEO/GSO

GSO

Advantages/Features

(1) Small number of satellites required for global coverage (as few as 3)
(2) Simpler ground station tracking

Disadvantages

(1) Poor service in high latitudes
(2) Expensive to launch and maintain
(3) Large propagation delay (~300 ms)
(4) Large power required to communicate

HEO

Advantages/Features

(1) Service at high altitudes
(2) Selected/Specific area of coverage

Disadvantages

(1) Passes through Van Allen radiation belt
(2) High propagation delay
(3) Complex to operate

Example 1

Which of the following orbits is a spacecraft able to remain over the same spot on the Earth: GSO, HEO, LEO, MEO?

Solution

The two orbits that allow a spacecraft to remain the same spot on the Earth are GEO and GSO. The answer for this question is GSO.

Spacecraft Configuration and Architectures

Satellite EPS architectures vary depending on mission technical design requirements. Spacecraft power and mass requirements typically drive EPS energy storage requirements for the battery power system. EPS systems may be regulated or unregulated where system voltages vary with the battery or solar arrays. A functional block diagram of a typical electrical power system is shown in Fig. 6.

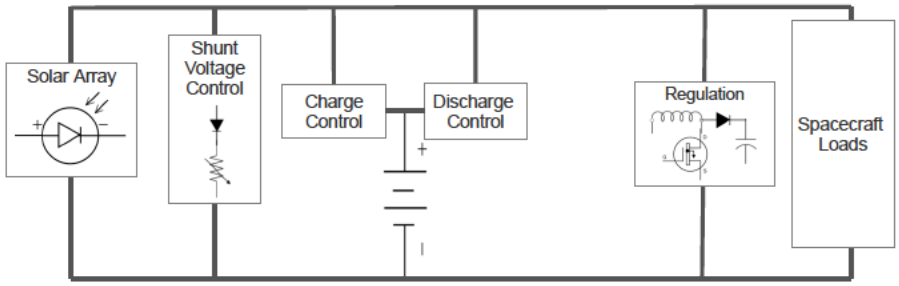

Figure 6: Functional Block Diagram of Typical EPS

Small Scale Architecture

Small scale architecture is the design and configuration used at the circuit board, or load level.

Unregulated System

An unregulated bus is one whose voltage is approximately that of the battery (or source) minus harness and switching losses [D].[9] An unregulated system can be modeled as shown in Fig.7. The dynamic unit *G* represents any circuitry used to alter or filter the input voltage but without a feedback mechanism to adjust the output. The term *G* (for *gain*) is also known as the forward transfer function or direct transfer function. [E]

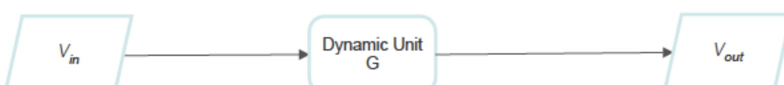

Figure 7: Unregulated System

The unregulated system is subject to battery voltages changes due to depth-of-discharge and end of life values. An example of the battery voltage changes for a LIB system depending upon the depth-of-discharge is shown in Fig. 8. The important point provided from the voltage

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[9] AIAA S-122, *Electrical Power Systems for Unmanned Spacecraft* is an extensive and comprehensive reference for power systems' design. *Space Vehicle Design* is also highly recommended as a thorough review of the myriad aspects mentioned in this course. Specific guidance on the associated electrical calculations can be found in the author's book *PE Power Reference Manual*.

vs. depth-of-discharge curve is that the voltage level of a LIB remains relatively constant over a significant depth-of-discharge (DoD). Only at a DoD of 80% or greater does the voltage level become a potential for concern. The curve varies for different chemistries but generally retains the shape shown.

As the number of cycles increases, a LIB will reach the DoD percentages shown earlier—essentially shifting the curve down and to the left.

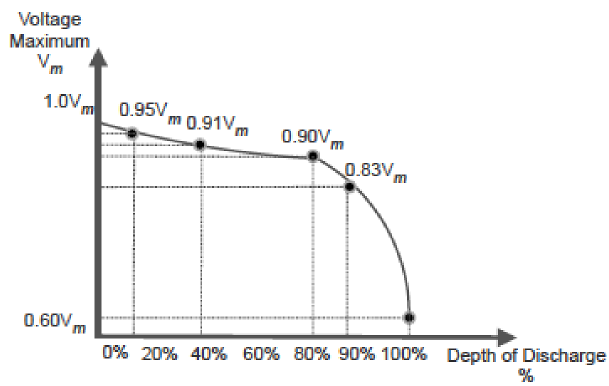

Figure 8: Lithium-Ion Battery Voltage vs Depth-of-Discharge

Regulated System

A regulated system can be modeled as shown in Fig. 9. The term *H* represents the feedback transfer function also known as the reverse transfer function.[10] The feedback is designed to keep the desired parameter, often voltage, within the design range. The open-loop transfer function represents the value of the output after going around the feedback loop one time and is often shown as $\pm G(s)H(s)$.

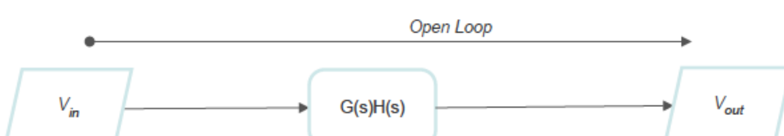

Figure 9: Regulated System—Open Loop Transfer Function

[10] The term *H* represents the next letter in the alphabet following *G*.

The *overall transfer function*, also known as the *closed-loop transfer function*, *control ratio*, *system function*, and *closed-loop gain* is the function that exists when going around the loop multiple times (see Fig. 10). The characteristic equation that, when evaluated, contains the response of the output to a forcing function input is given by 1 + $G(s)\mathrm{H}(s) = 0$. [E]

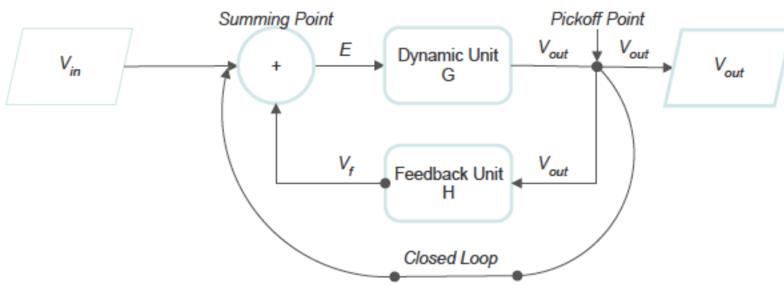

Fig. 10: Regulated System—Closed Loop

When one knows the characteristic equation, the output as a function of the input can be determined and the stability determined as well (see Stability). [E] The equations are explained in [D]. The derivations of the equations that follow are found in [F].

$$
\begin{aligned}
G_{\text{loop}}(s) &= \frac{V_{\text{out}}(s)}{V_{\text{in}}(s)} \\
&= \frac{G(s)}{1+G(s)H(s)} \quad \text{[negative feedback]} \\
&= \frac{G(s)}{1-G(s)H(s)} \quad \text{[positive feedback]}
\end{aligned}
$$

The takeaway for the EPS designer, especially when out-sourcing a given subsystem, is that the transfer function must be known. If the circuitry is proprietary, the contract should specify either (1) a black box model of the circuit with the appropriate inputs and outputs or (2) the actual transfer function of the entire subsystem. Additionally, the items mentioned should be certified by the supplier as being accurate.

Example 2

The ratio of the output voltage to the input voltage for a given circuit −100. The open loop transfer function value is −19. An amplifier used in a given circuit has a sensitivity that results in positive feedback.[11] What is the gain, G, of the circuit that makes it functional?

Solution

The ratio of the output voltage to the input voltage is given by the following.

$$\begin{aligned} G_{\text{loop}}(s) &= \frac{V_{\text{out}}(s)}{V_{\text{in}}(s)} \\ &= -100 \end{aligned}$$

The open loop transfer function is given by $\pm G(s)H(s)$, which is given as −19.

Substituting the given values in a positive feedback circuit gives the following.

$$\begin{aligned} G_{\text{loop}}(s) &= \frac{V_{\text{out}}(s)}{V_{\text{in}}(s)} \\ &= \frac{G(s)}{1 - G(s)H(s)} \quad \text{[positive feedback]} \\ -100 &= \frac{G}{1 - (-19)} \\ G &= (-100)(1 - (-19)) \\ &= (-100)(20) \\ &= -2000 \end{aligned}$$

[11] Sensitivity is not discussed in this course. See Ref. E, Chap. 41 “Analysis of Control Systems” for more information is desired.

Large Scale Architecture

All prime COMSAT manufacturers offer highly-optimized modular satellite bus platforms as part of their medium-to-large high-power satellite product lines. Table 2 lists various commercial satellite providers bus subsystem architectures and energy storage system design details. LIB capacity and voltage architecture is configured by changing the number of basic cells in parallel (and/or series) within the regulated bus voltage range. In other words, LIB capacity is scalable depending on spacecraft end-of-life power requirements.

Table 2: COMSAT LIB-Based Bus Platform Types[12]

COMSAT Provider	Platform Type	Primary Bus Voltage (V)	Payload Power Range (kW)
Airbus Defense and Space	Eurostar 3000	100	7-25
Chinese Academy of Space Technology	DFH	100	Up to 11
Information Satellite Systems Reshetnev	EXPRESS 2000	100	Up to 15
Lockheed Martin	A2100	70	Up to 20
Maxar	SSL 1300	100	5-25
Mitsubishi Electric Corp.	DS2000	100	Up to 15
Northrup Grumman Corp.	GEOStar	36	Up to 8
OHB System AG	Small GEO	50	Up to 10
Thales Alenia Space	Space Bus 4000	50	5-20
The Boeing Company	702	100	8-25

[12] Data taken from a variety of open source material including advertisements by the manufacturers.

Energy Balance and Battery Budget

Throughout the lifetime of the spacecraft, DC electrical power is generated and distributed to user loads. In Earth-orbiting spacecraft systems, the most common means to store energy is using electrochemical batteries. Batteries are characterized by the ability to store and deliver energy at a characteristic DC voltage to the spacecraft user loads. Most Earth-orbiting spacecraft utilize solar arrays to generate energy coupled to rechargeable batteries for energy storage. To maintain a positive energy balance during all mission phases, energy must be stored to meet user power demands during eclipse (no or partial sunlight) when the solar arrays are not capable of converting sunlight into electrical energy. During non-eclipse (full sunlight) orbital time periods the spacecraft user loads may exceed the solar array power generation capability. In these operational modes, the spacecraft batteries may be required to support help meet user load demands. A robust battery is also sized to support mission operations during transfer-orbit, station-keeping, peak power periods, and spacecraft contingencies. Rechargeable battery cycling is a primary operating characteristic governing battery lifetime and performance capability. Battery cycling characteristics such as end of charge (EoCV) and discharge voltages (EoDV), Depth of Discharge (DoD) and charge/discharge rates are typically controlled by the EPS PMAD on-board flight software hardware default protections. Critical control parameters can be updated throughout the life of the battery spacecraft ground control.

The battery budget, which determines the battery size, is not a topic directly covered in this course, per se. Nevertheless, the importance for the overall design of knowing the power budget for various modes of operations is essential to selecting the correct battery size—and the supporting distribution system wiring and its size. Examples of ISS power budgets over time can be found in [G]. A notional example is shown in Fig. 11. Note the “array power” is the “available array power” meaning not all of it may be utilzed. the unused portion is shunted.

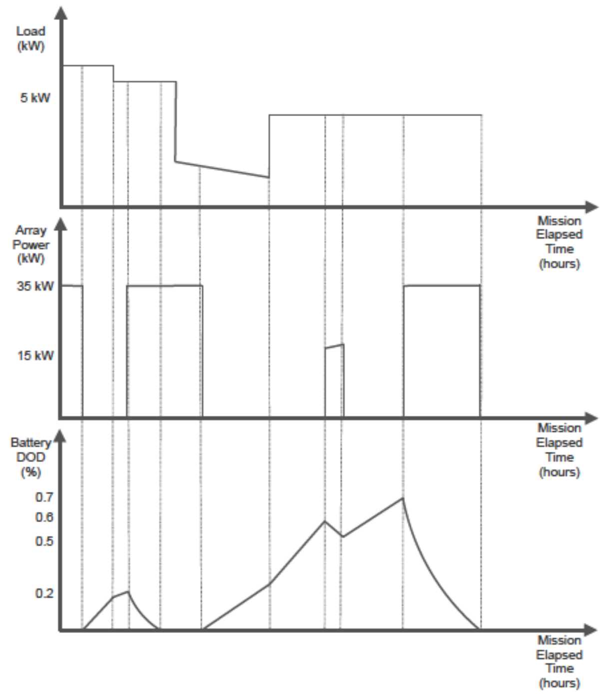

Figure 11: Operational Load Profile--Notional

The EPS system likely has multiple different operational requirements, which set the power budget for any given satellite. A fictional satellite's loads, corresponding to operational missions, is shown in Fig. 12. Such loads include on board remotely programmable computers, digital processors, maneuvering control circuits, attitude control loads, any experimental instruments, Earth observation equipment, emergency shutdown circuits[13] and recovery (safe start and dead bus) circuitry.

[13] Solar flares, as a single example, may require the temporary shutdown of onboard electronic equipment to prevent damage.

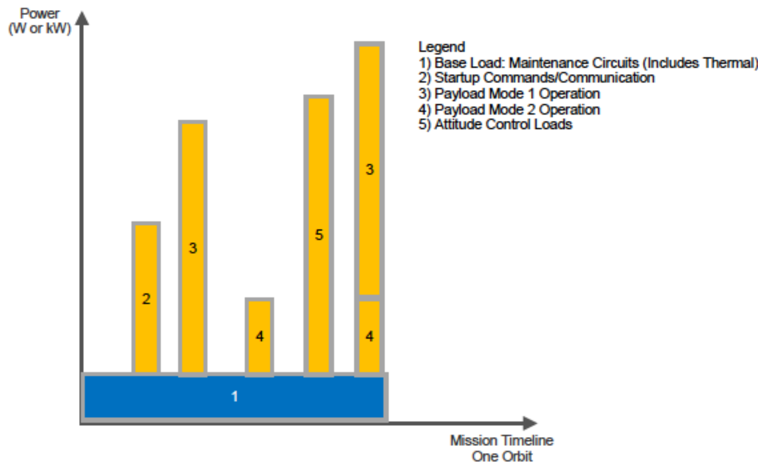

Figure 12: Notional Satellite Loads

Battery Balancing

Battery balancing also known as *battery redistribution* refer to the techniques used to improve the availability and lifetime of batteries with multiple cells. [H] Due to factors such as manufacturing or assembly differences individual cells may have different states of charge (SOC) at any given depth-of-discharge (DoD). This occurs in cells wired in series. (If in parallel, the cells naturally balance one another.)

Balancing can be passive or active and is performed by DC-DC converters. Passive balancing is accomplished by removing energy from the highest voltage cells and dissipating said energy through resistors. This method is wasteful and generates undesirable heat. Active balancing draws energy from the highest voltage cells and transfers it via capacitor- or inductor-based DC-DC converters to the lower energy cells. Active balancing is more complex and expensive. [H]

Other methods are possible, including a simple resistor and diode setup in series at each node. The charger can monitor the voltage at the node and adjusts its output to obtain the desired, and uniform, terminal voltage.

Battery Sizing

Once the power budget (consumption) is known, along with the allowable depth-of-discharge, the battery can be properly sized using the following equations. The power budget determines the amount of energy removed from the battery at any given time. The energy removed by a load is the power of the load (P_{load}) multiplied by the time the load is discharging the battery (t_d). (Power is energy per unit time.) The time constraints for recharging an equivalent amount depend on (a) the time available to recharge, which must be greater than the discharge time, (b) the charging system capacity in A-h (C_{charge}), and (c) the average voltage the system is to be charged to (V_{avg}). [B]

$$DOD = \frac{\text{Eclipse Energy Used}}{\text{Battery Stored Energy}}$$

$$DOD = \frac{P_{\text{load}} t_d}{C_{\text{charge}} V_{avg}}$$

The units in the numerator and denominator are W-h (or A-h), which makes the DOD unitless as required. The charge rate must be high enough to ensure complete charging without resulting in adverse effects such as overheating. Protocols for charging LIBs can be found in [I]. Generally, LIBs are charged at a constant current until 3.9 V – 4.2 V, then charging continues at a constant voltage until the manufacturer's recommended rate, typically C/50 or C/100.

Example 3

A Boeing 702 Platform uses a series-parallel topology. Assume LIB average cell voltage of 3.8 V for the chemistry utilized. How many cells in series are required?

Solution

Voltage of cells in series are additive. The number of cells required to generate the minimum 100 V is as follows.

$$
\begin{aligned}
N_{\text{Cells}} &= \frac{V_{Batt}}{V_{\text{cell}}} \\
&= \frac{100\ \text{V}}{3.8\ \dfrac{\text{V}}{\text{cell}}} \\
&= 26.32 \quad (27)
\end{aligned}
$$

Battery sizing must be correlated with the expected conditions at end-of-life (EOL). Battery capacity is affected by storage temperatures, time between charges, operational temperatures, discharge conditions, number of cycles, vibration and shock, as well as numerous other conditions. [I] Such numerous other conditions (environments) may not be obvious: fog, fungus, fine sand, and radiation for example.

System Requirements Overview

Customer spacecraft systems are typically defined in general terms by specifying spacecraft performance and capability mission requirements. Spacecraft performance requirements such as mission duration, orbit type, launch phase interfaces, payload power, and mass all impact the architecture of the payload and bus subsystems. Typically, the spacecraft supplier is responsible for decomposing and allocating customer-defined spacecraft performance requirements into bus, EPS and component-level specification requirements. Bus and EPS electrical, mechanical, thermal, and software interface control documents (ICD)'s are also derived from customer-defined specifications to ensure spacecraft-to-subsystem compatibility. ICD requirements also aid in establishing the spacecraft-to-launch vehicle design solution.

Figure 13 shows a standard requirements flow diagram which depicts how performance and capability design requirements are allocated to a spacecraft battery unit. Note the battery specifications are the child of the EPS requirements. Early in the preliminary design phase of a LIB power system there may be an iterative engineering requirements exchange between the EPS and battery unit engineering teams. This iteration process allows for specific derived battery requirements to be captured by the EPS to optimize performance and compatibility at the bus and spacecraft level of integration. Requirements iteration between the battery and cell subassembly design progression is less commonplace due to the COTS nature of most space qualified cell designs. Cells are selected from available COTS products to meet the higher-level assembly requirements.

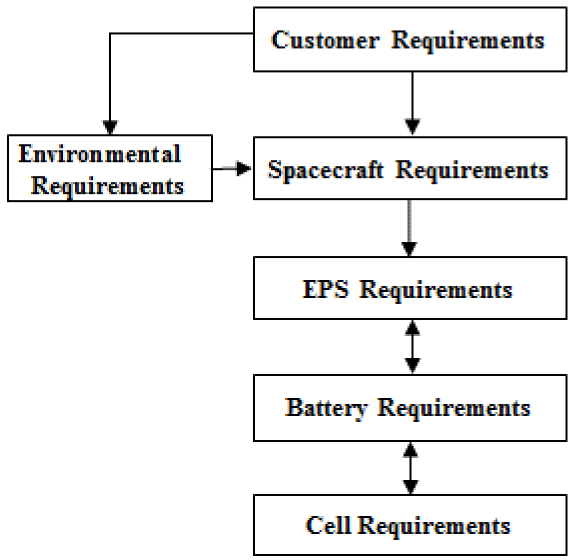

Figure 13: Spacecraft Requirements Flow Diagram

Setting requirements flow into steps gives the following.

1. Identify Top Level Requirements
    a. Customer Requirements
    b. Payload Requirements
    c. Mission Lifetime
    d. Mission Profile
    e. Environments

    Results: Spacecraft Requirements—Electrical Power Profile

2. EPS Selection and Sizing
    a. Load Power Requirements
    b. Spacecraft Configuration
    c. End-of-Life Power

    Results: Electrical Power System Requirements / Solar Array & Battery

3. Energy Source Selection and Sizing
    a. Orbital Parameters
    b. Average Power
    c. Eclipse Time
    d. Peak Power

    Results: Battery Requirements

4. Power Regulation and Control
    a. Thermal Control Requirements
    b. Mission Load Requirements
    c. Voltage Regulation Control
    d. Power Source Selection

    Results: Electrical Power System; Battery; and Cell Requirements

Iteration among the steps occurs as any given requirement is updated. A robust change process is required at the system engineering level to ensure compliance with all requirements.

Spacecraft Requirements Summary

Spacecraft-level performance requirements serve as parent requirements for the next lower-level of system integration. Requirements such as spacecraft payload power, orbit type, mission design lifetime (ground plus mission), reliability, and mass are critical to systems trade studies which baseline energy storage and generation component design. Key spacecraft requirements such as the payload power profile, mission lifetime, and mass are allocated to the EPS specification. Further requirements decomposition to derive battery voltage, capacity, and reliability enable power system energy storage design trade studies. As a result, choice of battery cell tec00hnology (secondary, primary, or reserve) and battery architecture (s-p or p-s topology)[14] result from early program trade studies designed to demonstrate requirements compliance at the EPS-level. Keep in mind that the development of the EPS system is done in parallel with the design of the payload components (sensors, transmitters, receivers, navigation and guidance system, etc.), and it is not uncommon for the payload power requirements to change over the course of the spacecraft development.

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[14] Batteries in series increase system voltage. Batteries in parallel add system capacity.

EPS Requirements Summary

The scope of an EPS requirements specification is to establish the design, performance, development, and verification requirements for the spacecraft power system. The majority of EPS-level requirements are derived from the applicable and relevant spacecraft payload and bus parent requirements. Relevant spacecraft level and environmental requirements are decomposed and allocated to the EPS specification for the purposes of defining the energy generation, storage, management, and distribution architecture.

Typical EPS level requirements which have a significant impact the battery design follow and are in the general order of the requirements flow figure.

Mission Lifetime

Spacecraft mission life can vary from hours to years depending on the given application. Battery design life is composed of both calendar (ground storage, test, and standby) and cycle life at the specified DoD during both eclipse and non-eclipse periods. Battery sizing, cell chemistry, operating temperature, reliability, fault management, and redundancy also impacts the ability of the ESS to meet mission life requirements.

EPS User Load Profile

Peak and non-peak user load profiles impact the battery sizing parameters such as capacity, impedance, and thermal control. BMS charge-discharge rates are also derived from the user load profile and spacecraft orbital period. The battery maximum DOD is derived from the user load profile, environmental operating conditions, and mission lifetime requirements.

Battery Charge Management

The spacecraft EPS provides the capability to autonomously maintain the battery and receive spacecraft commands to update battery maintenance parameters throughout life. LIB based EPS may also require battery cell balancing capability based on mission lifetime, orbit type, and battery architecture.

Bus Voltage

EPS bus voltage (regulated or un-regulated) determines the choice of energy storage technology and battery sizing. Bus voltage requirements also dictate the type of BMS employed to support battery state-of-health monitoring, which occurs via the communication system telemetry.

Battery Telemetry

Battery SOH is determined by trending battery voltage and temperature characteristics throughout the operating and non-operating lifetime of the spacecraft EPS. As such, EPS requirements for voltage sense monitoring and number of temperature sensors may be specified. LIB specifications also specify requirements for cell-to-cell balancing capability to meet mission performance and lifetime requirements. Depending on the complexity of the system, the EPS may provide telemetry capability to receive uplinks from ground stations to issue commands to change charge voltage limits, initiate charge balancing, or engage bypass circuitry for EPS systems designed with redundant cells, battery modules, or batteries.

Requirements Precedence

Requirements impacting the design engineer directly are most of system specifications and interface control documents (ICDs). Additionally, key performance parameters (KPP) and key system attributes (KSA) provided in the contract and flowed down into the specifications directly or a derived requirements.

The priority of these requirements depends upon the type of contract guidance provided, but in general, the lowest level items are dealt with most often but they cannot violate the requirements above. An example of the priorities is shown in Table 3.

Table 3: Requirement Precedence Example

Law Mandates
Regulations
Performance Oriented Documents [Performance Statement of Work]
Detailed Design Documents
Government Standards/Specifications
[Outside Defense of Federal Standards for Non-Repetitive Acquisitions]
NGO Standards
Industry Standards
[ISO, IEEE, ANSI, SAE, AIAA]
Federal Specifications
Schedule
Contract Requirements
Specifications & ICDs [Includes KPPs and KSAs]
Drawings

In the US, Federal specifications do not necessarily override industry standards, hence their location in the table. This comes from guidance in the OMB Circular A-119, Development that states, “Use Voluntary Consensus Standards and in Conformity Assessment Activities, agencies must use voluntary consensus standards (e.g., ISO, IEEE, ANSI, SAE), when they exist, in lieu of Government-unique , standards except where inconsistent with law or other impractical.” [J]

Regardless of the priority, requirements should have the following characteristics. [K]

- Achievable
- Verifiable: Avoid “sufficient” or “excessive” in Terminology, be Specific
- Unambiguous: One Meaning to ensure Testability/Verifiability

- Complete: Include Environments or Operational Conditions
- Performance Based: Express a "need" not the "solution"
- Avoid Conflicts in the Precedence Chain
- Appropriate for the Hierarchy Level: Avoids Constraints on Lower Level Designs

Regardless of the requirements selected, all requirements are validated. Validation is a process by which requirements and specifications are checked against the customer's needs—as reflected in the contract.

ELECTRICAL POWER DISTRIBUTION DESIGN GUIDANCE

The purpose of the EPS is to deliver reliable power to all loads in all operational states, for all foreseeable environments, over the lifetime of the spacecraft. [L]

The EPS is developed in an iterative style, as is the case in most design work. Nevertheless, certain steps can be followed in a logical sequence to ensure all requirements are satisfied. This list is in the approximate order in which items would be considered. Any change to an item lower in the list will result in changes higher in the list. Multiple teams will work any given portion of the list, so again, robust change control and coordination is requisite to success.

1) Harness Guidance
    a. Select Harness Guidance/Requirement
        i. Output: Proper Separation; Number of Harnesses/Connectors
    b. Determine Maximum Length
        i. Output: Maximum Voltage Drop
    c. Coordination: Structural/Manufacturing
2) Ampacity
    a. Determine Load Power Requirement/Specification
        i. Output: Required Amperage
    b. Select EEE Parts Derating Guidance/Requirement
        i. Output Wire Size to Meet Amperage Capacity
    c. Coordination: Harness/Connector/Thermal Analysis

3) Routing
    a. Determine Routing
        i. Output: Mass/Structural Support
    b. Coordination: Voltage Drop/Ampacity/Harness Separation/Connector Size/Manufacturing Engineering
4) Voltage Drop
    a. Determine Load Voltage Requirement/Specification
        i. Wire Size to Meet Voltage Drop
    b. Coordination: Ampacity/Connector/Routing/Thermal Analysis
5) Mass
    a. Determine Mass Allowance
        i. Allotted by Teams outside EPS Team
    b. Coordination: Required to Remain within, or extend, Allowance
6) Grounding
    a. Determine Required Grounding
        i. Output: AGND, DGND, Safety Ground, Distributed Ground
    b. Coordination: Harness/Routing

Harness

Harness requirements are found in numerous documents [D, L, M]. The one selected depends upon contract and specification requirements. Regardless, such standards allow for tailoring for particular projects. Therefore, a given team may pull elements from each of the standards and create a tailored in-house specification.

Normally, a team is assigned to tracking the accuracy of the start- and end-points of the harness wiring as well as the wire sizing and separation. The EPS designer should be aware of these parameters and set up margins, usually monitored by a Systems Engineering team, to ensure that changes in load power, amperage, or voltage requirements can be immediately assessed as to its impact to the overall system.

Harness design can be considered an essential part of PMAD (Power Management and Distribution). Harnesses should not be used as mechanical support. Power lines within, and sometimes the harnesses themselves, are twisted to minimize inductance. The routing is kept as short as possible and close to structure to minimize loop area, again, to minimize inductance. Multiple harnesses are designed to minimize failure propagation. However, harnesses can be "open" meaning they have no protective outer covering, or "protected". [M] Spacecraft normally use protected harness types, with shielding. To accomplish this, separation and shielding for and between types of electricity is requisite. [L]

Reference [L], which is normally tailored for specific space projects, lists five categories of wiring within the harness types. Category I is for Power and Control. Category II is for High Level Signals. Category III is for Low-Level Signals. Category IV is for Electro Explosive Devices. Category V is for High-Frequency Signals.

Each category has unique twisting, shielding, and separation requirements. The stated requirements are applicable to each sub-category which are split by circuit character: Direct Current; Alternating Current below 0.1 MHz; Alternating Current between 0.1 MHz and 1 MHz; Alternating Current above 1 MHz; Pulse Currents with rise or fall times greater than and less than 1 µsec; and one for Electro Explosive Device (EED). Each of the subcategories is further split by voltage and current limitations. An excerpt from Table 1 from [L] is showing the more important separations for most satellites is given as Table 4.

In addition to the separation mentioned in Table 4, electrically unprotected wiring from a primary electrical power system should not be bundled or grouped with distribution wiring. Further, two or more sources (think solar and battery) shall not be in the same bundle or group to prevent damage from one source causing damage to the other. Flight control equipment also uses separate harnesses and connectors due to their essential nature. Redundant equipment is routed in separate bundles to ensure one fault does not remove all redundancy. [M]

Table 4: Harness Category & Shielding Requirements

Circuit Character	**Level Volts (V) Amperes (A)**	**Category**	**Shielding**
Direct Current	<10V & <5A	IIIa	Shield as a group from other categories
Direct Current	<10V & >5A	Ib	None
Direct Current	>10V	Ia	None
Pulse Rise Time >1µsec	<5V Vpeak	IIIe	Each pair shielded

Circuit Character	Level Volts (V) Amperes (A)	Category	Shielding
Pulse Rise Time >1μsec	5V < Vpeak <25V	IIa	Each pair shielded
Pulse Rise Time >1μsec	Vpeak > 25V	Id	None

A thumb rule that the author has seen used but, is not documented as far as is known by the same, is to allow for expansion within each harness (or connector) during the design process.[15] Often loads are added and additional wiring is required. A *minimum* of 10% and up to 20% of additional spare contact/pin connections is suggested at the beginning of the design process. Even though such connections are "spares", they should still have pins and sockets installed. Indeed, in firewall applications, they are also wired with pigtails, which are dead-ended. [M]

Example 4

Harness design concerns include all of the following. But, which of the items is of the least concern to the electrical engineer, at least initially?

a) length/routing
b) mass
c) separation
d) voltage drop

Solution

Electrical requirements must be met regardless of other concerns. Given such, the mass—though tracked and limited where possible—is not an overriding concern.

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[15] Local procedures are used to document such items where they would be considered "derived" requirements. If a design is stable there may not be a need for such spares.

Another part of harness design is what part of the wire to connect to the pin and to the socket. Connectors contain male (pin) connections and female (socket) connections. When a connector is pulled apart, the design is normally such that the socket connection will be the one attached to the power source (hot side). [M] This is because it is more difficult to touch, or short to, a socket rather than a pin.

Harness wires can also be specified by their color as opposed to wire markings making it easier to differentiate the sizes, and their correct connection points during fabrication. Connectors have to be installed such that they can be disconnected easily, if required, yet positioned so they do not become hand holds or footrests. [M] Additionally, connectors of the same type, mounted closely together, must be keyed to prevent misconnection.

In summary, harness and wiring design shall follow this precedence: Safety in Flight; Ease of Maintenance; and Cost-Effectiveness. Wiring fabrication and installation shall follow this precedence: Maximum Reliability; Minimum Interference & Coupling between Systems; Accessibility; and Prevention of Damage. [M]

Ampacity

NASA has numerous research centers, flight centers, institutes, space centers, and flight facilities. Each of these specialize in certain core competencies.[16] MSFC focus on systems engineering, systems design and analysis, avionics and electrical systems led them to release Ref. N, which specifies the allowed ampacity for spacecraft wiring. This is also provided in NASA's EEE Parts Management Program for the ISS, but is export controlled. [O]

Reference N is a Parts Management Standard. In it are derating requirements for a variety of equipment, for wires see App. A labeled "Wire and Cable". The values are closely

[16] Ames Research Center in Moffett Field, CA. Armstrong Flight Research Center in Edwards, CA. Glenn Research Center in Cleveland, OH. Goddard Space Flight Center in Greenbelt, MD. Goddard Institute of Space Studies in New York, NY with Facilities IV and V in Fairmont, WV. The Jet Propulsion Laboratory in Pasadena, CA. Johnson Space Center in Houston, TX. Kennedy Space Center in Florida. Langley Research Center in Hampton, VA. Marshall Space Flight Center in Huntsville, AL. NASA Headquarters in Washington D.C. Stennis Space Center in Mississippi. Wallops Flight Facility on Wallops Island, VA. White Sands Test Facility in Cruces, NM.

aligned. Factors to consider when using Ref. N follow. An excerpt from Ref. N, App. A "Derating Requirements" for "Wire and Cable" is shown in Table 5.

1) Use the vacuum column for space limitation
2) Use the non-vacuum column for ground operations
3) Realize that when a wire is bundled, the wire single current, I_{SW}, is derated based on the number of wires, N, in the bundle to a new value, I_{BW}, for the bundle as follows.

$$N < 15$$
$$I_{BW} = I_{SW}\left(\frac{29 - N}{28}\right)$$

$$N \geq 15$$
$$I_{BW} = 0.5 I_{SW}$$

4) The wire currents assume an ambient in vacuum of 94ºC (72ºF). To that end, an interface with the thermal analysis team exist here. The EPS designer must know the maximum ambient conditions in order to properly size the wiring.
5) The rating of the wire is accounted for through the use of factors. For example, a 150ºC wire uses only 65% of the vacuum column current level whereas a 135ºC wire uses only 45% of the same value.
6) The maximum wire temperature for the maximum single wire current for vacuum use is 147ºC (295ºF).

Table 5: MSFC-STD-3012 Wire & Cable Derating

Wire Size (AWG)	**Vacuum <4.3 PSIA** I_{SW} **(A)**	**Non Vacuum >4.3 PSIA** I_{SW} **(A)**
20	8.8	10.0
14	18.0	20.0
12	25.0	29.0
10	34.8	40.0
6	80.0	92.0
2	150.5	170.5

Portions of other references that can be used follow. For example, in Ref. D, harnesses are sized to keep wire temperature below the maximum temperature of the wire while passing 200% of the fuse current rating, or maximum current rating of the appropriate current limiting device. Ampacity in Ref. M is based on a 60,000 ft altitude, an ambient of 70ºC, and no more than 20% of the wire harness capacity being utilized. Additionally, charts are provided for differing ambient conditions and for the prevention of corona discharge.

The derating of fuses and all other electrical and electronic devices are also covered within the appendix to Ref. N.

Example 5

A wire is rated for 30 A. The wire is located in a bundle of 8 wires. What is the rated current for the wire considering only the bundle derating?

Solution

The derating for a bundle of wire is given by the following.

$$N < 15$$
$$I_{BW} = I_{SW}\left(\frac{29 - N}{28}\right)$$

$$N \geq 15$$
$$I_{BW} = 0.5 I_{SW}$$

Since less than 15 wires are bundles, the first equation is applicable.

$$N < 15$$
$$\begin{aligned} I_{BW} &= I_{SW}\left(\frac{29 - N}{28}\right) \\ &= (30\text{ A})\left(\frac{29 - 8}{28}\right) \\ &= (30\text{ A})\left(\frac{21}{28}\right) \\ &= 22.50\text{ A} \quad (22\text{ A}) \end{aligned}$$

The rounding down is deliberate to avoid exceeding the requirement. One can, however, carry the decimal through all the derating calculations and round appropriately when complete.

Routing

Routing goals are to minimize inductive coupling, or EMI/EMC, which is accomplished through twisting of wiring and harness, short routing, shielding, and separation. [L] In fact, Ref. L goes so far as to require sensitive circuits below 5 V and high impedance circuits above 1000 Ω be isolated by routing or shielding *even if the circuits are within the same category*.

Mockups of wiring harnesses, even with CAD techniques are used to ensure complex harnesses will fit as expected. Routing through small structural openings shall be avoided to minimize difficulties during installation. (This is one of the many areas where an interface with Manufacturing Engineering occurs.) Obviously, routing shall be such as to minimize damage.

Minimum distance of 50 mm[17] is required between a harness and any line containing flammable liquids or gasses (as may be present for maneuvering). [L]

As with many a reference one will call out several others. Reference L calls out Ref. M for "protection and support", that is, physical support of wiring harnesses and cable assembles. While the EPS designer may not be directly responsible such details, they should ensure the requirements are called out for the installers and QA inspectors to know of the correct supports to confirm.

Routing ensure reliability by avoiding chafing; use of wiring as handholds/foot steps; damage from fluids or moving parts; providing adequate support as well as necessary slack and most importantly from an electrical standpoint, avoid EMI/EMC.

Voltage Drop

When determining the ampacity, although there were many variables, no mention was made of the voltage required for the load itself. Once the wire size is determined, based on the required amperage of the load, an iteration must occur to ensure both the margin to the ampacity and the voltage at the load under worse case conditions.

Use the data, specifically the resistivity, from the wire standard (or the manufacturer's data) for the chosen wire combined with the routing length and the maximum current to determine the voltage drop as shown below.

$$R_{\text{one-way}} = \frac{\rho \cdot L}{A}$$

$$\Delta V = V_F - V_0 = -I_{\max}\left(2R_{\text{one-way}}\right)$$

$$V_F = V_0 - I_{\max}\left(2R_{\text{one-way}}\right)$$

Important points, and common mistakes made during such calculations include:

[17] Reference M by contrast requires a 2 inch separation but allows for 0.5 inch if certain conditions are met. See Sec. 3.11.11 of [M].

- Not including the two way or total resistance of the wire. And, not recalling that the return path length may differ from the power path.
- Not accounting for twisting effects on length.
- Using something other than the worst case initial voltage for the source.
- Using the rated current of the wire when the maximum current of the load is all that is necessary.[18]

The voltage drop calculations mentioned here are for the EPS when the battery is the source. Refer to Fig. 6. When the solar array, which is deliberately oversized, is on line, a shunt regulator is used to lower the voltage output of the array to the desired value. This value should be higher than the initial value, V_0. Meaning a check should be done to ensure that when the solar array is on line, the voltage to the load does not exceed the voltage allowable.

Final point, though only the wiring was mentioned, the voltage drop is often done via a spreadsheet so that the drops due to the wire pin/socket connection, the pin and socket themselves, and the contact drop expected due to the force of the pin to sock connection may all be included.

Mass

Mass contributors to the EPS system are extensive and should be carefully tracked and allocated during the design process to ensure requirements are met and propulsion/attitude control systems are able to operate as expected. A general list of contributors to the overall EPS mass follow. [D]

- Power generation sources
  - solar array assembly including all deployment and control devices
  - energy storage systems including the battery packs, installed heaters, and embedded electronics
- Power Management & Distribution elements
  - charge and discharge controllers
  - bus and payload switching units
  - bus voltage converters

[18] Careful with this. If the load changes, or is tested and draws more than expected, one can underestimate the voltage drop. The conservative approach is to use the rating of the wire.

- health monitory systems
- voltage / power regulators
- harness components
    - monitoring
    - power
    - load

The masses mentioned are BOL masses. Some mass loss may occur over the mission life time due to outgassing in a vacuum environment—though this is designed against. High mass components, such as some capacitors, are to be avoided if possible and are specifically mentioned in Ref. N.

Grounding

The grounding system is all wires, structures, and connections that determine the return current paths and the zero voltage reference plane of a spacecraft. While that is the definition, the structure should NOT be used as a power return path to avoid ground loops and EMI. [K]

Ideally, a spacecraft would use a single-point ground system (SPG). That is, all DC return currents from loads or ground planes are carried by low impedance conductors back to a single point. Nevertheless, as frequency increases the impedance of low impedance path to ground is degraded and the conductors can become antennas. This generally occurs when the conductor length is approximately 1/20th of the wavelength of the highest operational frequency utilized in the system. [D]

$$L_{\text{conductor}} = \frac{\lambda_{\text{highest}\,f}}{20}$$

Where the length of the conductors are greater than that calculated, a multi-point ground system is preferred.[19] The multi-point ground system is characterized by using the hull structure

[19] On a multi-stage rocket, rather than a spacecraft, each stage using a SPG is sometimes called a "distributed grounding system".

as the return path.[20] See Fig. 14 for an example of both. A hard ground is shown—a direct connection to the spacecraft structure. A soft-ground is a resistor-capacitor parallel circuit that is used for safety purposes. A soft-ground is defined as one with resistance high enough to limit DC current to < 5 mA. Such a ground may not draw enough current to trip fault protection circuitry.

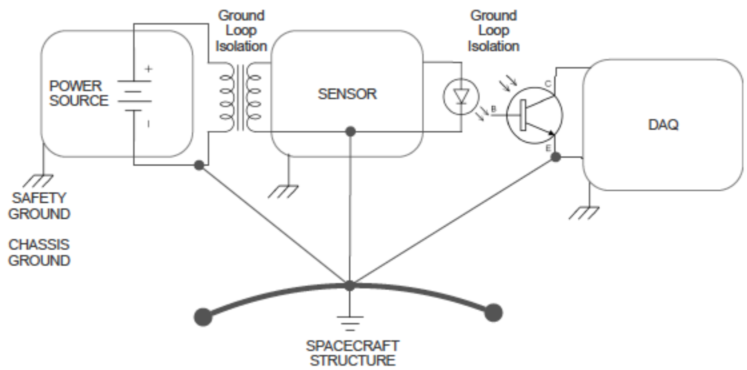

Figure 14(a): Single-Point Ground Scheme

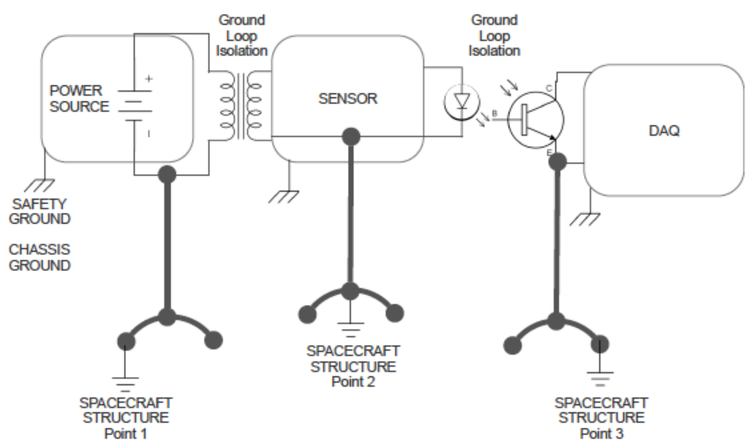

Figure 14(b): Multi-Point Ground Scheme

[20] In the figure the return lines connect to the source. They don't necessarily have to be so connected. The structure could be used as the return path saving the weight of the return wires. Such a setup though may result in EMI issues.

Bonding is the process of connecting parts to ensure adequate connectivity. Requirements vary, but for non-current carrying parts the maximum resistance to ground allowed is 2.5 mΩ. Various values and the reason for each are explain in Ref. P. The different classes are as follows.

Grounding & Bonding Classes

Class A: Antenna (No Bonding Resistance Specified)
Class C: Current Return Path [Fault Current vs. Resistance Table Provided]
Class H: Shock Hazard (0.1 Ω) ["H"—think "Human"]
Class L: Lightning Protection [Minimizes Internal Voltages to <500 V]
Class R: RF Potentials (2.5 mΩ)[Electronic Units to Structure]
Class S: Static Charge (1.0 Ω)

Of note, Class R has no scientific basis [as a limit]. It represents "good metal-to-metal contact" and is therefore used. The need for Class R bond in modern balanced electronics is less obvious. [P]

Protection and Safety

Protection comes in many guises. From an electrical perspective, this can be electronic charging/discharging controls; over- and under-voltage detection; short-circuit cutouts; overload devices; fuses; circuit breakers; load-shedding controls; bus filtering for transient limitation and EMI/EMC performance; state of health (SoH) monitoring; as well as telemetry to track overall performance and long term trends for voltage, current, power output and consumption, and temperatures.

The goal is overall mission accomplishment with minimal downtime, introduction of redundant circuits before failure, if necessary. Reliability and fault tolerance requirements are documented in the EPS specification and flow down to the appropriate sub-systems. A designer must determine if a failure is even credible before expending effort to protect against such an occurrence. The definition of a credible failure is a model of failure with a probability of occurrence within a single year of one in a million. [D]

The focus of failures for the EPS here will be electrical, and those most common, in nature. But, failures occur in a variety of forms: insulation, plasma arcs, corona discharge, outgassing, radiation, thermal, vibration, shock, as well as material (tin whiskers; red plague)[21] generated. To that end, a general review of the environments in which a spacecraft may find itself is useful.

Battery structure must comply with dynamic and mechanical static environments induced by launch and other spacecraft mission events. Launch induced random vibration environments have the most significant impact to battery structural components. Pyro-shock events induced by solar array, radiator, or payload deployments must also be analyzed and tested if the battery is located in a spacecraft shock zone. Spacecraft batteries are also designed to meet all performance requirements when subjected to the depressurization environment that occurs during spacecraft launch and ascent.

Other spacecraft natural environments such as radiation, electromagnetic interference (EMI) and compatibility (EMC), and electrostatic discharge require specialized testing and analysis to ensure survivability of active battery electrical components and materials of construction. LEO spacecraft batteries directly exposed to the space environment (such as the NASA-ISS battery ORU's) may also be required to meet stringent micro-meteoroid and orbital debris (MMOD) vulnerability and survivability requirements.

Ground Operations

During ground operations the system is tested as close to actual flight configuration as possible—called "testing as you fly". This includes placing equipment in the proper locations, having harnesses of lengths identical to that on the spacecraft, testing the parts as a whole and individually if parts testing did not adequately cover contractual or mission requirements.

Protection devices must be included on the ground support equipment (GSE), also called electrical ground support equipment (EGSE), to prevent damage to the spacecraft. Such protection includes both over- and under-voltage circuits that monitor voltage and react in a pre-programmed manner to avoid spacecraft damage. The same must occur for current. Surge suppression also exists to prevent AC transients from interfering with or damaging electronic circuitry. Reverse polarity protection protects against the misapplication of power. There is

[21] Tin whiskers form when tin is used as solder and it contains less than 3% lead. The red plague is a phenomenon whereby silver coated copper conductors are galvanically corroding forming Cu_2O (red) and then CuO (black).

always an overriding emergency shutdown button located in the testing and control spaces for any unforeseen circumstances to remove power to the spacecraft. And, finally, there is some type of battery safing mechanism to prevent over-discharging the battery in the event of a spacecraft malfunction.

Between periods of testing, batteries will be charged. The charge management circuits should reflect the type to be utilized in space and be monitored for proper operation. Lithium-ion batteries are lighter and have higher energy density, hence their widespread use; but they are sensitive to overcharging effects, therefore the performance of the charge/discharge controllers and associated software is of critical importance.

Protection unrelated to flight occur during ground operations such as battery handling and movement, battery storage, battery conditioning prior to initial use, tagout procedures during maintenance, access control, transportation (and packaging) of equipment

Space Operations

Fault protection design of an EPS system shall be analyzed to ensure the following. [D]

- fuses or breakers operate before wires are damaged
- inrush and outrush currents can be accommodated without tripping
- fault clearing surges and spikes do not cause issues at other distribution points

All parts must function as advertised with no sneak paths for failure and the appropriate fault tolerance—none[22], one-, or two-fault tolerance—is achieved. To achieve this end, a FMECA—Failure Modes, Effects, and Criticality Analysis is often done using guidance from [Q] or a tailored version of it. Though Ref. Q was cancelled by the military in 1988 it remains in widespread use for guidance. It differs from FMEA, or Failure Modes and Effects Analysis, which is also is use. FEMCA differs in the focus on criticality analysis. That is, analyzing the failures along with the severity of their consequences.

For the protection and safety of the entire system, it must be electrically stable under all operating modes and conditions. The analysis of this stability, as well as any FMECA or FMEA

[22] No fault tolerance may be required for smaller satellites, such as CubeSats, or where redundant satellites are available.

should be accomplished using worst case analysis (WCA). That is, proper functionality should be shown during WCA for the entire range (high to low) for voltages, currents, radiation, tolerances, temperatures, environments, and mission life.

During spacecraft operations, the EPS must be able to autonomously detect, respond, and isolate faults to maintain proper operation. And, it must do so without ground commands.

Testing

Testing is one part of verification, which also includes analysis, simulation, and inspection.[23] Verification is necessary to ensure specified requirements are satisfied. Testing includes individual parts, components, line replaceable units (LRUs), or whole systems or subsystem as occurs in functional performance testing.

Verification methods definitions follow. Test is making observations on equipment and comparing with specified requirements. Test as you fly (TAYF) is a level of testing that mimics, as closely as possible the conditions under which the spacecraft will operate. Inspection is a visual test for compliance. Demonstration is a type of testing that exhibits performance without the use of test devices. Demonstrations are normally monitored by qualified personnel and may be tests that are impractical or unnecessary to verify more than once. Analysis takes many forms, but always requires adequate documentation.[24]

The amount and type of testing is driven by four main items.

- The testing essential to mission success.
- The testing compulsory by contract.
- The Grade of parts used in the EPS.
- The TRL (Technology Readiness Level) of the EPS.

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[23] Recall that "validation" is ensuring the requirements meet the customer's expectation and contractual obligations. On the other hand, "verification" is ensuring that the system / subsystem / part as built meets the requirements.

[24] Verification may also occur by similarity. The requirements for this verify widely and should be researched thoroughly before invoking such an approach.

Testing is of course focused on what it takes to ensure mission success and to meet all contractual requirements. The main guides to the amount of testing required are the Grade of parts used and the TRL of the various pieces, parts, subsystems, and systems that make up a given unit under test.

The grade of parts is defined in Ref. N with the following quality breakdown.

- Grade 1: EEE parts of the highest practicable quality standards
- Grade 2: EEE parts of high, but not the highest, quality standards
- Grade 3: EEE parts that meet some formal industry quality standards, but usually the lowest available
- Grade 4: EEE parts with no predefined quality standards

See Fig. 15 for an overview of the NASA Technology Readiness Levels, taken from Ref. R. The Department of Defense has adopted the TRL system in Ref. S. Additionally, it has been adopted internationally as ISO 16290. [T]

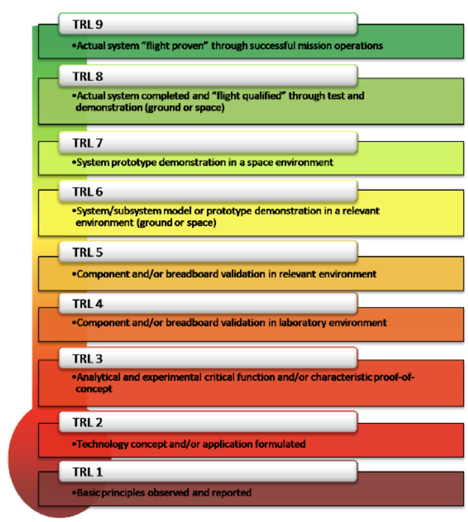

Figure 15: NASA TRLs

The higher the Grade and the higher the TRL level the less comprehensive testing is required to qualify a part or a system.

Minimum sets of verification and validation requirements for EPS of Earth-orbiting spacecraft include provisions for test and analysis of compliant EPS. LIB power system requirements such as cell charge-discharge control, thermal management, and fault protection are common to LEO and GEO LIB battery power system designs.

The primary pass-fail criteria, specifically for a LIB battery system will be overall capacity, as expected depth-of-discharge responds (see Fig. 8), and cell voltage uniformity following life-cycle testing testing—including accelerated testing designed to ensure the battery can meet end-of-life mission requirements when operated over all operational modes.

Accelerated tested is accomplished using established techniques. Such testing is vital given that LIB systems are operated in parallel with solar arrays (see Fig. 6) whose output is extremely variable (i.e., "dirty") consisting of a great deal of harmonics. Nevertheless, recent testing documented in Ref. U indicated minimal impart. An excerpt from that report is provided here.

> With the vehicle industry poised to take the step into the era of electric vehicles, concerns have been raised that AC harmonics arising from switching of power electronics and harmonics in electric machinery may damage the battery. In light of this, we have studied the effect of several different frequencies on the aging of 28 Ah commercial NMC/graphite prismatic **lithium-ion battery** cells. The tested frequencies are 1 Hz, 100 Hz, and 1 kHz, all with a **peak amplitude** of 21 A. Both the effect on cycled cells and calendar aged cells is tested. The cycled cells are cycled at a rate of 1C:1C, i.e., 28 A during both charging and discharging, with the exception of a period of constant voltage at the end of every charge. After running for one year, the cycled cells have completed approximately 2000 cycles. The cells are characterized periodically to follow how their capacities and power capabilities evolve. After completion of the test about 80% of the initial capacity remained and no increase in resistance was observed. No negative effect on either capacity fade or power fade is observed in this study, and no difference in aging mechanism is detected when using non-invasive **electrochemical methods** of post mortem investigation.

Stability

Stability studies are generally separated into Steady State, Transient, and Dynamic Studies. Steady State Stability involve small or gradual changes in the system operating conditions. Transient Stability involve large, major, changes in the system operating conditions. Dynamic Stability is the study of a power system's ability to maintain nominal parameters under continuous small disturbances. Dynamic stability is also known as small-signal stability. At times both dynamic and steady-state stability are considered as small-signal stability.

Steady State stability can be determined from electrical models of the system using the principles below as well as testing a system before deployment. Transient studies can be determined from electrical models once the operational conditions (various loading configurations) are known and very commonly by testing before delivery and deployment. Software programs allow for modeling of systems and multiple types of loading studies.[25] Testing, as always, should be the final arbiter of a systems true ability.

In stability studies and analyses, the relationship between the source impedance and the load impedance is the critical factor. Consider Fig. 16, Source and Load Stability. This figure shows the magnitude of the source and load impedances as frequency increases. The condition for unconditional stability is when the following condition is met.

$$\frac{|\mathbf{Z}_{\text{Load}}|}{|\mathbf{Z}_{\text{Source}}|} > 1$$

When this condition is not met, the system must be analyzed by one of the methods to be discussed. In straightforward terms, when the load impedance is less than the source impedance, the load may appear to be a short-circuit to the source.[26]

During stability studies at the design stage, impedances of the source and loads are estimated and analyzed over the expected frequency range. Even in DC systems, those with batteries, instabilities can occur as more and more loads are added in parallel. Especially when such loads have power supply inverters changing the DC into AC for use by various circuits (NASA Technical Fellow, personal communication, 2017). Also, solar panels inject harmonics into the power system that may result in instabilities.

[25] One such program is ETAP®, Electrical Transient Analyzer Program for large power systems. Another useful tool, especially for electronic projects, is LTspice®. OrCAD® PSpice®, OrCAD being *Or*egon + Computer Aided Design, and Pspice being Personal Simulation Program with Integrated Circuit Emphasis, is another.

[26] While not specifically correct, such memory aides or thumb rules are useful.

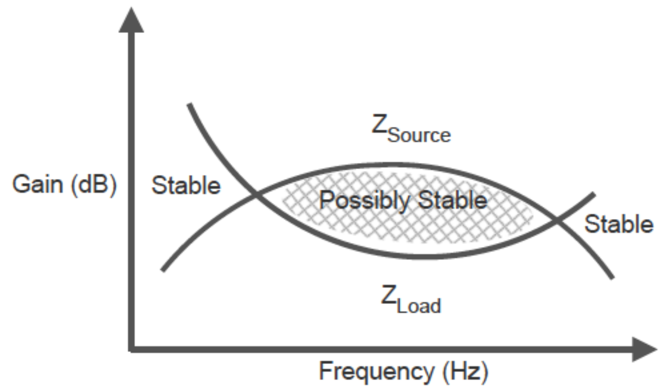

Figure 16: Source and Load Stability

Unregulated power systems provide power and make no attempt to control the output voltage as the load changes. A regulated power system is designed to ensure the output voltage remains at rated value while the load (current) changes. A regulated system will have a feedback system of some type. A general model is shown in Fig. 17.

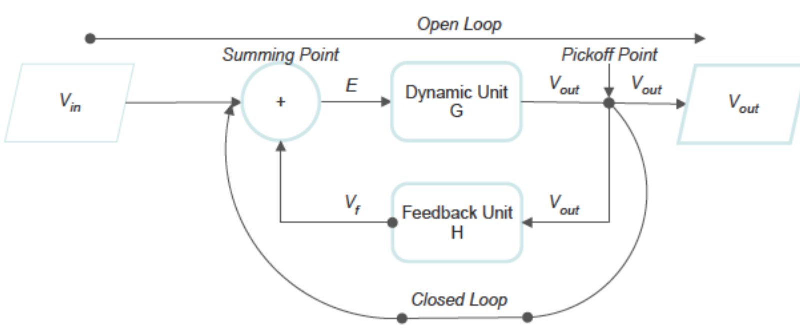

Figure 17: Feedback System

Passive systems (i.e., the homogeneous case) are not acted upon by a forcing function and are always stable. In the absence of an energy source, exponential growth cannot occur. Active systems contain one or more energy sources and may be stable or unstable.

There are several frequency response (domain) analysis techniques for determining the stability of a system, including Bode plots, root-locus diagrams, Routh stability criteria, Hurwitz tests, and Nichols charts. The term frequency response almost always means the steady-state response to a sinusoidal input.

The value of the denominator of T(s) is the primary factor affecting stability. When the denominator approaches zero, the system increases without bound. In the typical feedback loop, the denominator is $1 \pm GH$, which can be zero only if $|GH| = 1$. It is logical, then, that most of the methods for investigating stability (e.g., Bode plots, root-locus diagrams, Nyquist analyses, and Nichols charts) investigate the value of the open-loop transfer function, GH. Because $\log 1 = 0$, the requirement for stability is that log GH must not equal 0 dB.

A negative feedback system will also become unstable if it changes to a positive feedback system, *which can occur when the feedback signal is changed in phase more than 180°. Therefore, another requirement for stability is that the phase angle change must not exceed 180°.*

A stable system will remain at rest unless disturbed by external influence and will return to a rest position once the disturbance is removed. A pole with a value of $-r$ on the real axis corresponds to an exponential response of e^{-rt}. Because e^{-rt} is a decaying signal, the system is stable. Similarly, a pole of $+r$ on the real axis corresponds to an exponential response of e_{rt}. Because e^{rt} increases without limit, the system is unstable.

Because any pole to the right of the imaginary axis corresponds to a positive exponential, a stable system will have poles only in the left half of the s-plane. If there is an isolated pole on the imaginary axis, the response is stable. However, a conjugate pole pair on the imaginary axis corresponds to a sinusoid that does not decay with time. Such a system is considered to be unstable.

Bode Plots

Bode plots are gain and phase characteristics for the open-loop G(s)H(s) transfer function that are used to determine the relative stability of a system. The gain characteristic is a plot of 20 log $|G(s)H(s)|$ versus ω for a sinusoidal input. It is important to recognize that Bode plots, though similar in appearance to gain and phase frequency response charts, are used to evaluate stability and do not describe the closed-loop system response.

The gain margin is the number of decibels that the open-loop transfer function, G(s)H(s), is below 0 dB at the phase crossover frequency (i.e., where the phase angle is −180°). (If the gain happens to be plotted on a linear scale, the gain margin is the reciprocal of the gain at the phase crossover point.) The gain margin must be positive for a stable system, and the larger it is, the more stable the system will be. The phase margin is the number of degrees the phase angle is above −180° at the gain crossover point (i.e., where the logarithmic gain is 0 dB or actual gain is 1). See Fig. 18.

In most cases, large positive gain and phase margins will ensure a stable system—but not in all cases. As a result, these criteria can be used to determine the conditional stability of a system. The phase and gain margins are thus a measure of how far away from oscillation a system is, with oscillation being an indication of instability. When the phase and gain margin of a system or circuit are both zero, oscillations occur. If the margins are maintained, the system is stable, but is only conditionally stable; if the load increases or the voltage decreases, the gain will decrease at the crossover point, potentially causing instability and oscillations in the system.

In simple terms, if a phase shift in the dynamic unit, G (the forward transfer function), is −180°, and the dynamic feedback unit, H (the reverse transfer function), adds another −180°, then the signal feedback will be identical to the input signal. If the gain is 1 or has shifted to 1 because of the conditions mentioned, that is, the gain is not <0 dB (to lower the value of the feedback signal), then the system input and the feedback are identical. The input signal will essentially double and become unstable.

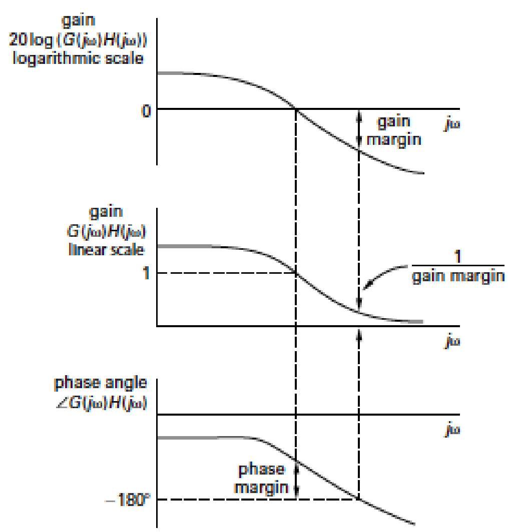

Figure 18: Phase and Gain Margin

In summary, margins can shift, and if the phase is 180° at other frequencies rather than the crossover frequency, instabilities may occur. Though Bode plots are adequate in most instances, to ensure unconditional stability, a Nyquist plot should also be developed. A Nyquist plot shows the ratio of the source to the load impedance over the full frequency band (0 Hz to ∞). Prior to performing a Nyquist plot, a graph of the dynamic load impedance can be used to define the areas of concern. Though used primarily in power systems, electronic loads provide the load impedance and can be restricted by the design of the source. To ensure stability, the load impedance needs to be greater than the source impedance. If the source impedance is greater than the load impedance, a Nyquist plot is required to verify that the phase shifts and gains will remain within acceptable ranges in the potentially unstable regions.

EPS Integrated Testing

EPS integrated and functional testing is guided by the Spacecraft Level Test and Verification Plan that draws guidance from MIL-STD-1540 for environments and MIL-STD-1541 for EMC testing. [V] Successful EMC/EMI testing involves the application of proper shielding and grounding, among other considerations, all of which are covered well in Ref. [W].

In the effort to minimize costs but maintain space qualifications for parts and systems the main EMI/EMC guidance for the International Space Station, SSP 30237 [X], has been widely adopted, adapted, and compared to other requirements to allow COTS equipment to be tested for utilization in space. The conducted emissions (CE), conducted susceptibility (CS), radiated emissions (RE), and radiative susceptibility (RS) of SSP 30237 are compared to FCC requirements in Part 15, the IEC-1000 series, and DO-160 guidance. [V]

While the guidance for a particular spacecraft is a combination of the indicated requirements, the sequence of testing utilized often comes from DO-160 [Y]. While technically a reference for aircraft testing, it has wide application in space. Furthermore, spacecraft are assembled and transported on Earth and thus requirements such as fog, salt spray, and transportation vibration, among others, are taken from DO-160. Verification as well as guidance on test requirements and methods specific to spacecraft can be found in the Space Power Standard AS 5698[Z] An example of a sequence of testing is shown in Fig. 19. Note the functional testing between major tests. This, should a failure occur, assists in isolating the cause.

Acceptance Testing is used to determine the performance of a product to its design specifications. It can be accomplished at the individual part level, line replaceable unit level, or subsystem or system level. It is the basis for a customer's acceptance of the product. Qualification Testing, accomplished prior to. acceptance testing is a formally designed series of tests by which the functional, environmental, and reliability of a part, subsystem, or system is checked. Units that are qualification tested are NOT flown in spacecraft. Only those units whose design was validated with qualification testing, but are subjected to acceptance testing only, are flown on an operational spacecraft.

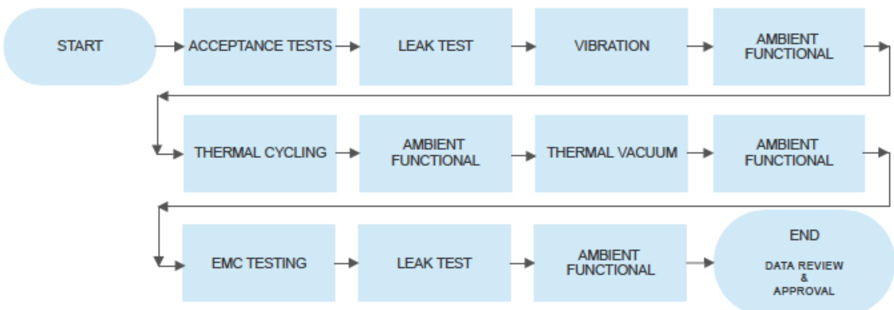

Figure 19: Qualification Testing Sequence Example

BATTERY SYSTEM PROTECTION

An overcurrent is a condition in which the current is in excess of the rated current for the equipment or wired. It consists of an overload, short-circuit, or ground-fault. An overload is any condition beyond the rating of the equipment, which if continued for an extended time will damage the equipment or cause overheating. Overloads are longer term issues than overcurrents, which must be acted upon quickly.

Short-Circuit and Overload Protection

Spacecraft Electric Power Systems are designed to allow the battery to support fault clearing. Each load is individually protected. [D] Fuses, when used, are designed at the lowest rating possible but high enough to allow for in-rush, out-rush, and normal switching transients. It must be noted that fuses cannot be reset or replaced in space. So, they should be used sparingly and a backup circuit should be available for any critical system.[27]

Circuit breakers and other current limiting devices have the same basic requirements of a fuse with the advantage of being able to be reset should the fault clear.

[27] The fuses referred to as thermal fuses. There are resettable that reclose following cooling.

When LIB are subject to overcurrent or overload conditions, temperatures increase and the pressure within the sealed cell will increase. To prevent an overpressurization condition and potential catastrophic failure several devices can be used. One is a circuit interrupt device (CID) that is a mechanical switch that opens the cell current path at a predetermine pressure level. It is difficult to incorporate such switches and thus is not used on high power cells. Another method is the PTC or pressure, temperature, and current switch. This method is more complex but can reset thus does not permanently disable the battery. Tripping a PTC though can irreversibly increase cell resistance resulting in long term impacts. The PTC is typically used in small cylindrical cells.

If the system is subject to large transient conditions due to capacitive or inductive loads, timers can be used to prevent tripping during inrush or outrush conditions. Finally, maintaining the thermal conditioning of the battery in an acceptable range avoids a multitude of problems.

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