AEROSPACE REPORT NO.
TOR-2005(8583)-2

Electrical Power Systems, Direct Current, Space Vehicle Design Requirements

11 May 2005

Prepared by

B. A. LENERTZ
Electrical and Electronics Systems Department
Electronics Engineering Subdivision

Prepared for

SPACE AND MISSILE SYSTEMS CENTER
AIR FORCE SPACE COMMAND
2430 E. El Segundo Blvd.
El Segundo, CA 90245

Contract No. FA8802-04-C-0001

Systems Planning and Engineering Group

THE AEROSPACE CORPORATION
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B. A. LENERTZ
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Electronics Engineering Subdivision

Approved by:

A. F. RIVERA, Director
Electrical and Electronics Systems Department
Electronics Engineering Subdivision

V. I. LANG, Principal Director
Office of the Corporate Chief Architect/Engineer
Systems Planning and Engineering Group

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Abstract

This Technical Operating Report baselines an updated set of requirements for spacecraft electrical power and distribution systems. It is intended to be used as a starting point for upgrading of previous military specifications in this area, or for development of a new specification dedicated solely to power system requirements. An ancillary use of the document is to edify those in the acquisition process such that they may more thoroughly understand the basic considerations of power system design, as well as subtler and sometimes unaddressed issues that can adversely affect mission success if not addressed.

Foreword

This Technical Operating Report baselines an updated set of requirements for spacecraft electrical power and distribution systems. It is intended to be used as a starting point for upgrading of previous military specifications in this area, or for development of a new specification dedicated solely to power system requirements. An ancillary use of the document is to edify those in the acquisition process such that they may more thoroughly understand the basic considerations of power system design, as well as subtler and sometimes unaddressed issues that can adversely affect mission success if not addressed.

The motivation behind this work is the policy directive letter of Lt. Gen. Brian A. Arnold, Commander of the USAF Space and Missile Systems Center (SMC), in 2002. In it, General Arnold calls for a return to “high-priority critical specifications and standards” that contribute to mission success and successful program implementation through heightened insight into program status, risk elements, and critical process definitions. However, General Arnold states that these specifications and standards are to be used in “a less prescriptive manner than in the past.” The contractor “may propose the listed specification/standard or another government, industry, technical society, international or company version provided it is comparable in vigor and effectiveness. Proof of this comparability must be provided.”

Unfortunately, in the case of electrical power systems (EPS) for space systems, there is no one military specification or standard that previously governed all aspects of design. MIL-STD-704E (Aircraft Electric Power Characteristics), MIL-STD-1541A (Electromagnetic Compatibility Requirements for Space Systems), and MIL-STD-1539 (Electrical Power, Direct Current, Space Vehicle Design Requirements) together comprise a fairly complete set of requirements, but they were written so long ago that they are difficult to apply to many of today’s design practices.

Of these three standards, MIL-STD-1539 is the most general collection of overall EPS requirements. Therefore, it is recommended as a good candidate for update and expansion. This TOR is written essentially as a draft rewrite of 1539 (which could be called 1539A, but could also be given a new MIL-STD number or be taken up by another standards group such as AIAA), with additional material included (in *italics*) to elaborate on the basic content. This is done not only to provide background material to non-specialists, but also to spur discussion and deliberation concerning the final form of the updated standard.

It is difficult to formulate a set of requirements that are universally applicable to the many different types of EPS, the various mission and payload types, and the wide range of power levels that different spacecraft types might use. The requirements could be reduced to two basics: 1) the power system shall reliably provide power under all normal and some abnormal conditions, and 2) the power system shall be compatible with all the loads. These are obvious requirements, but not particularly useful, because they do not give any guidance as to how these requirements are to be satisfied, nor do they provide a basis for verification by the procurement activity.

On the other hand, levying too many hard requirements can unnecessarily restrict the contractor’s design space. For example, the existing MIL-STD-1539 calls out a bus voltage of 28.0 ± 6.0V. A requirement like this – although it may have seemed like a good idea to follow the aircraft-derived requirements back in 1973 – is clearly anachronistic. Many large, present-day spacecraft operate at voltages above 50V, while nanosats and picosats opt for much lower voltages such as 5.0V. The military specification should not dictate what specific bus voltage to use, but should identify *de facto* standard voltages for the sake of compatibility with existing third-party hardware. The specific levels would be called out in the system specification for a particular space vehicle.

The key to successful specification of EPS requirements, then, lies not so much in specifying absolute quantities or design techniques, but in laying out in a more general sense all the technical concepts that must be addressed in an EPS design, coupled with firm requirements for the contractor to show (via test, analysis or simulation) how the design solutions that are chosen will meet overalls goals of mission success and longevity.

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1.0 SCOPE

1.1 General

This standard establishes requirements for direct current (DC) electrical power systems (EPS) for space vehicles .

1.2 Purpose

This standard ensures compatibility between the space vehicle DC EPS and all its interfaces, including the utilization equipment. It ensures this compatibility in all intended states, modes, and conditions. It also ensures that the space vehicle will not be damaged or degraded by certain unintended or anomalous conditions, as described herein.

1.3 Basis for Requirements

The requirements, characteristics, and limits specified in this standard build upon those of MIL-STD-1539 (1 August, 1973) and MIL-STD-1541A (30 December, 1987).
The focus of this standard is establishment of requirements for the fundamental performance of the EPS. While this standard allows much freedom in design choices, it imposes stricter quantitative measures of EPS performance than in the past, along with requirements to provide specific types of analysis and simulations to demonstrate compliance to the performance specifications.

2.0 REFERENCED DOCUMENTS

The following documents form a part of this standard to the extent specified herein:

DOD-W-83575A	Wiring Harness, Space Vehicle, Design and Testing, General Information For
TOR-2005(8583)-1	Electromagnetic Compatibility Requirements for Space Equipment and Systems
AIAA-G-020-1992	Guide for Estimating and Budgeting Weight and Power Contingencies

3.0 DEFINITIONS

3.1 Electrical Power Subsystem (EPS) The EPS of a space vehicle is the 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 utilization equipment.

3.1.1 Power Generating Subsystem The power generating subsystem consists of all equipment involved in the generation of DC power for use by the utilization equipment and for charging the energy storage devices. Solar arrays are the most common technology for spacecraft power generation, but other technologies, such as thermoelectric devices, fall into the category.

3.1.2 Energy Storage Subsystem The energy storage subsystem is comprised of devices that store some of the energy generated by the power generation subsystem, for use in powering the utilization equipment during periods – such as eclipse – when the output of the power generation subsystem is insufficient to meet the overall load demand. Secondary (rechargeable) batteries are the prevalent means of energy storage, but ultracapacitors, flywheels, fuel cells, and primary (nonrechargeable) batteries also fall into this category.

3.1.3 Power Control Subsystem The power control subsystem consists of all hardware and software, including analog and digital circuits, command interfaces, switches, relays, interconnects, sensors, chargers, dischargers, and other related devices, used to control and steer electrical power from the power generating subsystem, to and from the energy storage subsystem, and to the power distribution subsystem

3.1.3.1 Main Bus The main bus, or simply "bus," is the designation for a single distribution point for the electrical power flowing from the power generation subsystem, to or from the energy storage subsystem, and to the power distribution subsystem. The term includes both the positive and return connections for this distribution point, and the voltage differential between positive and return is referred to as the "main bus voltage," or simply "bus voltage" (for single-bus systems).

3.1.4 Power Distribution Subsystem The set of all equipment, software and interconnects, whose function is to steer electrical power from the power control subsystem to the utilization equipment, is called the power distribution subsystem. It includes mechanical and electrical switching devices, fuses or circuit breakers, current monitors, and secondary DC-DC converters for selected load groups.

3.1.4.1 Distribution Point Any location in the power distribution subsystem where the power wiring branches to power two or more pieces of utilization equipment, not counting heaters, is called a distribution point. 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.

3.1.4.2 Voltage Reference Subsystem The VRS consists of all wires, structures, and connections that determine the return current paths in the EPDS. The layout of the VRS is designed to avoid interference between users of electrical power and to assure meeting conducted electromagnetic compatibility (EMC) requirements.

3.1.4.2.1 Ground Point Reference The GPR is a single point in the EPDS, often a location on vehicle structure, that serves as a reference point for measurement of potential differences within the VRS.

3.1.4.2.2 Single-Point Ground (SPG) This is a commonly-used type of VRS in which DC return currents from the utilization equipment or subsystem ground planes are carried via low-impedance conductors back to a single grounding point. Use of structure for DC return currents is prohibited for an SPG-type VRS.

3.1.4.2.3 Multipoint Ground. This type of VRS configuration allows use of structure as a low-impedance return path for currents. 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.

3.1.5 Regulated and Unregulated Buses An unregulated bus is one whose voltage is approximately the same as the battery voltage, minus harness and switching losses. A regulated bus is one whose voltage is controlled by means of a closed-loop negative feedback control scheme. A sunlight-regulated bus maintains a regulated bus during insolation and is unregulated during eclipse.

3.2 Utilization Equipment Any device or unit that uses electrical power provided by the EPS is considered to be part of the utilization equipment. Commonly called "loads" or "payload units." Units and devices comprising the EPS components are themselves considered part of the utilization equipment 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.

3.2.1 Essential Loads These are loads that are essential for minimum controllability and commandability of the spacecraft.

3.2.2 Nonessential Loads These are loads that can be powered off without adversely affecting the minimum controllability and commandability of the spacecraft.

3.2.3 Thermal Loads These are dissipative heaters used for temperature control of the spacecraft components.

3.2.4 Payloads A payload is a self-contained instrument, sensor, or device that fulfills some mission objective.

3.2.5 Load Groups A load group is a physical or logical partitioning of one or more loads. For example, 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.

3.3 Operational States This term covers all foreseeable and intentional combinations of states, modes, or conditions within the EPS hardware and software.

3.3.1 Mission Phases Mission phases the EPS are can be divided into factory test, launch-processing test, pre-launch, launch, transfer orbit or ascent, deployment, on-orbit or on-station, safe-hold or survival mode, and disposal or de-orbit.

3.3.2 Normal Operation Normal operation refers to operational states of the space vehicle that exist or occur by design, according to the expectations of the mission designers and planners. Safe-hold or survival mode is considered part of normal operation, as it is an anticipated reaction to vehicle anomalies.

3.3.3 Abnormal Operation Abnormal operation of the EPS encompasses unforeseen circumstances that are not handled via established contingency plans and operational states such as safe-hold mode.

3.3.4 Single-point Failure
A single component, wiring, or connector failure, software glitch or computer failure that results in the permanent loss of the space vehicle's ability to perform its primary mission for the intended design life-span, is termed a single-point failure (SPF).

3.4 EPS Design Terminology

3.4.1 Class

3.4.1.1 Class One From AIAA-G-020-1992, "A new design which is one-of-a-kind or a first generation device."

3.4.1.2 Class Two From AIAA-G-020-1992, "A generational design that follows a previously developed concept and expands complexity or capability within an established design envelope, including new hardware applications to meet new requirements."

3.4.1.3 Class Three From AIAA-G-020-1992, "A production level development based on an existing design for which multiple units are planned, and a significant amount of standardization exists."

3.4.2 Design Stages / Maturity From AIAA-G-020-1992, the six reference levels are as follows:

Bid	Proposal or Bid Stage
CoDR	Conceptual Design Review
PDR	Preliminary Design Review
CDR	Critical Design Review
PRR	Preshipment Design Review
FRR	Flight Readiness Review

3.4.3 Power Category From AIAA-G-020-1992, the spacecraft power categories are defined as follows:

Category AP	0 to 500 Watts
Category BP	500 to 1,500 Watts
Category CP	1,500 to 5,000 Watts
Category DP	5,000 Watts and up

3.4.4 Design Life The design life of the spacecraft is the minimum period of time called out in the system requirements specification, during which the spacecraft must be capable of performing all mission operational goals and objective as delineated in the system requirements specification.

3.4.5 Design Verification Design verification refers to all activities, including test, analysis, simulation, and inspection, that are performed to verify that a design meets its specified requirements.

3.4.6 Design Reference Cases (DRCs) In EPS parlance, a DRC is an example mission or set of operational conditions that is used in an analytical or simulation setting to show that a design meets or exceeds its performance requirements. A finite (hopefully small) set of DRCs may be formulated to cover the worst-case behavior of the system under all operating modes and conditions.

3.5 EPS Behavioral Terminology

3.5.1 Energy-Related Terms

3.5.1.1 Depth of Discharge (DOD) Depth of Discharge is the ratio of the number of Ampere-hours removed from a fully charged battery to the nameplate rated capacity of the battery, times 100.

3.5.1.2 State of Charge (SOC) State of Charge is the ratio of the number of Ah present in a battery to the rated capacity C(Ah) of the battery, times 100.

3.5.1.3 Energy Balance In EPS parlance, energy balance refers to the balance between solar array available power and the and the electrical power flow to the utilization equipment and the battery, over a defined orbital period. When positive energy balance exists, the spacecraft has enough power to perform the mission and recharge the batteries during the defined orbital period. When negative energy balance exists, the batteries will eventually discharge completely. The EPS is designed typically to have zero energy balance at EOL; i.e., there is exactly enough array power to power the loads and just barely recharge the batteries.

3.5.1.3.1 Power Margin Power margin is the amount of extra loading (in Watts) that could be added to the maximum anticipated load level that would result in the storage devices reaching their Minimum Stored Energy (MSE) level during a defined orbital period.

3.5.1.3.2 Minimum Stored Energy (MSE) Level The MSE level is the minimum allowable level of stored energy in a device, as agreed upon for a particular technology under particular operating conditions. For a battery, the MSE level is often stated as a maximum allowable DOD or a minimum allowable SOC.

3.5.2 Electrical Terms

3.5.2.1 Bus Types

3.5.2.1.1 Unregulated Bus An unregulated bus is one whose voltage is not controlled to a DC level by any feedback scheme. The voltage is approximately the same as the battery voltage, minus rectifier, harness, and switching voltage drops.

3.5.2.1.2 Regulated Bus A regulated bus is one whose voltage is controlled to a particular DC level by employing one or more feedback loops.

3.5.2.1.3 Sunlight Regulated Bus

A sunlight regulated bus behaves as a regulated bus during insolation when the available solar array power exceeds the bus load power. During eclipse, or other intervals when the load power exceeds the available solar array power, the bus behaves as an unregulated bus.

3.5.2.2 Bus Voltage The term bus voltage refers to the average DC voltage at the main bus or at any distribution point, as defined in 3.1.4.1.

3.5.2.3 Power Quality Power quality refers to the acceptability of the time-domain variation in bus voltage induced by the periodic and aperiodic currents flowing to and from the utilization equipment and to self-generated currents and voltages from the EPS equipment itself and by the GSE during ground testing.

3.5.2.3.1 Transients A transient is the bus voltage time-domain response due to an aperiodic event, or due to a periodic low-frequency (50 Hz or less) train of events.

3.5.2.3.2 Ripple Ripple is the cyclic variation of voltage about the mean level of the DC voltage during steady-state operation of the EPS. The ripple voltage 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.

3.5.2.3.3 Spikes Spikes are narrow impulse-like voltage waveforms that are produced by switching or fault-clearing events. Spikes are generally are measured in terms of their volt-second impulse strength and peak voltage amplitude.

3.6 Miscellaneous Terms

3.6.1 Fault Management Fault management in EPS is the process of detecting and reacting to the occurrence of a fault or anomaly, whether in hardware or software.

3.6.2 EPS Software EPS software is all software 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.

3.6.3 Ground support equipment (GSE) GSE for the EPS is all support equipment that is used in the ground testing of the EPS as integrated on the spacecraft, either at the contractor facility or at the launch site.

4.0 GENERAL REQUIREMENTS

4.1 Purpose of EPS

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

4.2 Power Quality

The power quality of the delivered electrical power shall conform to the detailed requirements of this standard, Section 6.2, as tailored for the specific mission.

Previously, power quality at the system level was part of MIL-STD-1541A. It makes more sense to move it to this TOR in order to have a complete set of design requirements for the EPS as a whole. "Power quality" refers to the time-domain behavior of the voltage at the main bus, that is, the ripple and transients that the utilization equipment has to operate despite. The unit-level frequency-domain and time-domain requirements of 1541A (and its new successor, a TOR ostensibly called 1541B) are universal for all equipment, whether components in the EPS or part of the utilization equipment. The unit-level EMC requirements therefore stay with 1541B. But system power quality is unique to the EPS and should therefore be specified in this EPS specification.

4.3 Voltage

4.3.1 DC Voltage Range

The EPS shall provide DC power to the utilization equipment at a voltage or voltages compatible with the utilization equipment, including all payloads, and shall not deviate from the nominal voltage or voltages chosen by more than ± 3% for regulated buses or +/- 20% for unregulated buses. If a payload interface is undefined, a range of 28VDC +8, -6V shall be assumed.

In an acquisition environment where the host spacecraft bus is procured separately from one or more payloads, it is common for the required bus voltage to be at first undefined. Much heritage payload equipment is built to the old 28V standard, so this can still be considered a default value. However, many bus contractors have gone to much higher voltages for cost, weight, and efficiency reasons. Normally, payload contractors will redesign their equipment *to conform to the higher voltage ranges or due to parts obsolescence, but this can be costly and time-consuming. One compromise solution can be to have a high bus voltage but also have a secondary downconverter to 28V for heritage equipment that needs it. This issue is one of the first that the acquisition authority should address prior to source selection. Potential payload providers should be polled regarding the voltage requirements (and power quality, as well) of their equipment in advance of the acquisition activity for the bus.*

The +/-20% range given here is approximately the expected voltage swing of a battery-backed unregulated bus using Nickel Hydrogen batteries, going from 0 to 80% Depth-of-Discharge. The 3% variation for regulated buses encompasses the worst-case tolerances one would expect to see in such a system (2% for distribution drops & 1% for source regulation).

4.3.2 Undervoltage

No utilization equipment shall be damaged by the application of a bus voltage between zero volts DC and the minimum bus voltage allowed per Section 4.3.1.

4.4 Stability

4.4.1 Feedback Stability

The EPS of a space vehicle shall be proven to remain stable over the entire range of expected variations in power generation, energy storage, and load conditions in all operating modes, temperatures, orbital phases or conditions, over the mission design life. The EPS shall also remain stable and meet its power quality requirements through the largest anticipated step load increase or decrease in the utilization equipment. Computer analysis or simulations shall be performed that assure a beginning-of-life phase margin of at least 60 degrees, and a BOL gain margin of at least 10dB, in all feedback control modes under worst-case conditions. Unit- and system-level testing shall be performed to validate the predicted BOL analytical results.

4.4.2 Interface Stability

The power interfaces at the main bus and at other distribution points, between the EPS and the utilization equipment, shall be verified by analysis to have a minimum BOL gain margin of 6dB and a phase margin greater than 45 degrees. System-level step-load response tests shall be performed to validate the predicted BOL performance.

Interface stability means that the voltage provided to the utilization equipment remains within the power quality requirements at all times for all combinations of loads between minimum load and maximum load. This can be difficult to prove, as there may be many loads. The number of combinations of loads is at least 2^n, where n is the number of load configurations. The number of load configurations may be low in a spacecraft that has its loads hardwired to the bus and usually on, or high in a system with individual loads or load groups switched onto and off of bus power. Complicating the analysis is the need to *consider the input impedances of each load as a function of frequency. Further discussions on this topic are contained in Appendix A.*

Generally, a bus contractor will have a good idea whether or not his EPS is stable for a typical sets of loads, but there can be instabilities introduced by a number of means – adding new types of loads, high peak-power loads, solar-array shadowing, or addition of filters or additional bus capacitance. There are modeling and simulation techniques that can predict stability margins for all load combinations. Of course, it can be very difficult in the early stages of a program to get all the information needed to perform such an analysis. Certainly, such an analysis should be possible by CDR, or whenever final load-configuration information is available.

By specifying large BOL phase and gain margins, we assure the system will be stable at EOL – worst-case component drifts are encompassed.

4.5 Energy Balance

The EPS of a space vehicle shall provide positive power contingency (relative to the defined Minimum Stored Energy (MSE) level of the given storage technology) in all anticipated operating modes and in all expected orbital phases or conditions, taking into account worst-case conditions as defined in Section 6.3.4.1 and the most-stressing load timeline, as determined by analysis. Where load levels are undefined or incompletely defined, worst-case assumptions shall be made until such time in the design cycle that they can be more accurately defined.

Power margin (also called power contingency), as defined in Section 3.5.1.3.1, shall be the basis of evaluation for the energy balance analysis.

To cover uncertainties in load levels in the early stages of an EPS design, the EPS design shall comply with Section 4.2 (reproduced here), "Schedule of Power Contingencies," per AIAA-G-020-1992, "Guide for Estimating and Budgeting Weight and Power Contingencies," using the Class 1, 2, and 3 definitions of that document (see Section 3.4 of this document).

4.2 Schedule of Power Contingencies

Description/ Categories	Minimum Standard Power Contingencies (Percents)														
	Proposal Stage			Design Development Stage											
	Bid Class			CoDR Class			PDR Class			CDR Class			PRR Class		
	1	2	3	1	2	3	1	2	3	1	2	3	1	2	3
Category AP (0 to 500 watts)	90	40	13	75	25	12	45	20	9	20	15	7	5	5	5
Category BP (500 to 1,500 watts)	80	35	13	65	22	12	40	15	9	15	10	7	5	5	5
Category CP (1,500 to 5,000 watts)	70	30	13	60	20	12	30	15	9	15	10	7	5	5	5
Category DP (5,000 watts & up)	40	25	13	35	20	11	20	15	9	10	7	7	5	5	5

Notes:
1. Bidder/Designer should state the class and category applied to the article which is the subject of the bid or design review.
2. Power categories are for the ranges shown.
3. Any deviation of above values should be clearly documented in applications document.
4. If this table is reproduced, please cite "Guide for Estimating and Budgeting Weight and Power Contingencies for Spacecraft Systems" (AIAA-G-020-1992).

Early in the design phase, assumptions are made regarding load levels. The solar array, battery, and all other components of the EPS are sized to meet these levels. Frequently, though, as the loads become better defined, it is found that the EPS sizing is inadequate to stay below a certain battery Depth-of-Discharge (DOD). Unless the load power consumptions are expressed in terms of their worst-case maxima, there can be painful surprises later as the weight of the EPS rises due to increasing the battery size or expanding the solar array size. It is especially important for heater estimates to be worst-case until a realistic thermal analysis is done.

The referenced document, AIAA-G-020-1992, seems to be the only available standard that addresses the need to formalize the concept of power contingency as a function of design phase and maturity. Although the document was written in the early nineties, it still provides realistic guidelines for the amount of power contingency that ought to be included in EPS designs.

4.6 Power Distribution

The DC power shall be distributed to the utilization equipment in such a manner as to meet the wiring requirements of DOD-W-83575A, the electromagnetic compatibility requirements of TOR-2005(8583)-1, including applicable magnetic field strength requirements.

4.7 Grounding and Bonding

The EPS shall interface with the spacecraft's Voltage Reference Subsystem (VRS), consistent with the requirements of MIL-STD-1541B, Section 4.3. A single Ground Point Reference (GPR) shall be designated at some point in the EPS where primary power return is bonded to spacecraft structure. Although paired power and return lines (with the return line connected to the GPR) is the preferred method of power distribution, it is acceptable to use structure as a return as long as the following requirement is met: no point in the VRS shall develop, due to structure currents, a voltage with respect to the GPR whose DC level or frequency components from DC to 100MHz result in the failure of any spacecraft subsystem, payload, or of any piece of utilization equipment to meet its performance specification, including development of an unacceptable magnetic dipole moment, or cause the system to fail any of its electromagnetic compatibility requirements.

The original 1539 callout for grounding was that positive and return power lines had to be paired, and a Single Ground Point had to be used. This is a proven design practice, but not the only viable approach. There are contractors who get acceptable results from using a conductive structure, or a composite structure with conductive elements, as a return for all power lines on the spacecraft. As long as the design meets overall EMC requirements and manages its magnetic moment effectively, there is no need to specify the exact implementation of power distribution.

Note that loads using two wires under a SPG approach often have worse radiated emissions due to common-mode emissions on the power lines. Loads that use a structure return for power, by contrast, are easier to filter for all frequencies and have lower radiated emissions. Effective common-mode filters for two-wire systems are more difficult to design than for structure-return systems, especially if there is a restriction on common-mode capacitance.

4.7.1 Bonding of EPS Components to Structure

Electrical bonding of EPS components to spacecraft structure shall be less than 2.5 milliohms per bond. The number of parallel bonds at a given interface shall be sufficient to ensure that the power loss through the equivalent bond resistance is not greater than 1% of the total power associated with the distribution path associated with the bonds.

4.8 Fault Management

4.8.1 Mission Single-Point Failures

The EPS shall be free of credible mission single-point failures.

This is a requirement that is usually levied at the space vehicle system level. It bears repeating at the EPS level, since EPS operation is vital to the operation of all other systems on the space vehicle*. For some satellites, such as experiments or Class C missions, it can be tailored out. The word "credible" should probably be defined at some point to have an actual probability of failure over some period of time, such as the mission design life.*

4.8.2 Mitigation of Unfused Power Bus Short-Circuit Susceptibility

4.8.2.1 Protection Against Insulation Failure

Bus bars and wiring or other connections to the main power bus that are not protected by fuses or other protective current-limiting devices shall employ isolation techniques to ensure that the failure or degradation of any insulating layer will not result in a permanent short circuit on the bus. The minimum thickness of each insulating layer shall provide 2X margin against the amount of insulation degradation that would result from any mechanical damage due to any foreseeable wear mechanism**.**

Protection against SPFs on unfused parts of the power bus is generally obtained by double-insulating wires and using adequate spacers to keep bus bars away from chassis or structure. A minimum thickness for an insulating layer is called out to establish safe spacing practices.

4.8.2.2 Protection against plasma arcs

To protect unfused primary power from failures induced by metal plasma arcs, all metallic conductors, including but not limited to wires, bus bars, and printed wiring board traces, that have voltages exceeding fifteen volts, shall be completely insulated such that no bare conductors are exposed to a vacuum environment.

The best protection against plasma arcs is not to let them start. The most worrisome types of conductors are those containing tin. Although it is well known by now that pure tin produces tin whiskers, variations and errors in the plating process have allowed escapements, producing parts such as lugs that had pure tin plating, despite the certifications that said they were not pure tin. Wherever tin-containing parts are used, it is essential to use a thick-enough conformal coating to prevent growth of tin whiskers from puncturing through the encapsulated area.

4.9 Design Verification

4.9.1 Test

4.9.1.1 Test-As-You-Fly (TAYF)

TAYF principles shall be incorporated into the testability of the EPS design to the greatest extent practicable. Specifically, for the EPS in system test, this includes (but is not limited to) the following:

- use of a solar array simulator with dynamic I/V characteristics that can be adjusted to match those of the actual solar array over life and temperature
- use of test batteries with the same or nearly the same characteristics of impedance, dynamic behavior, capacity, and thermal response as the flight batteries

- exercise of all redundancy features and paths
- exercise of all commands and all telemetry measurements over the full range
- exercise of all foreseeable modes of operation for each mission phase with minimum and maximum load levels, and for the worst step-load changes, anticipated for each mode and phase.

4.9.2 Analysis and Simulation

The EPS design shall be verified via the analytical and simulation techniques described in Appendix A. A stability simulation model (as described in Appendix A) shall be used to prove stability of the EPS at BOL per the requirement of Section 4.4.

5.0 INTERFACE REQUIREMENTS

5.1 Utilization Equipment

5.1.1 Load Groups

The utilization equipment shall be partitioned into identifiable physical load groups. Redundant loads shall be placed in separate physical load groups. Separate sets of wires or cables from the main bus shall be used for separate load groups.

5.1.2 Essential Loads

Essential loads shall be partitioned into a single physical load group. If redundancy is used in the essential loads, separate load groups shall be used for each set of redundant essential loads.

5.1.3 Fault Protection

No failure in a piece of utilization equipment shall result in permanent degradation of the ability of the EPS to provide nominal power to the remaining utilization equipment. No failure in a piece of utilization equipment shall cause a failure in any other piece of utilization equipment or in any component of the EPS or power distribution system.

This is primarily a requirement about the fusing scheme. The contractor must show that a fault in any load branch that causes a fuse to blow will not cause any other load to become disabled. This can happen in four ways: 1) fault causes upstream fuse to blow instead of downstream fuse, causing loss of all loads served by the upstream fuse; 2) inductive energy stored in the faulted branch due to high fault current damages other equipment served by that branch after the fuse for the faulted load clears; 3) sudden outrush of current from loads when the bus voltage collapses during a fault causes fuse to blow (especially of concern in loads with inrush limiting, which generally does not restrict the flow of reverse current); and 4) sudden inrush of current (due to bus voltage recovery) into unfaulted loads after fuse *clearing in the faulted branch causes one or more fuses to blow in the unfaulted loads. This situation is covered by 6.4.3.1. Outrush limiting is covered by 6.4.3.2.*

5.2 Space Vehicle Interfaces

5.2.1 EPS Telemetry

The set of EPS telemetry measurements shall include, at a minimum, the following:

- the voltage at the main power bus
- the individual voltages and currents of each energy-storage device
- the currents of each major load group
- the total load current
- the total solar array current
- individual battery cell voltages
- For NiH2 batteries, the internal pressures of each cell
- baseplate temperatures of each separate box or unit
- at least two temperature measurements per battery pack
- status of every EPS functional element and DC switching device
- the trend data outlined in 6.1.1

5.2.2 Command Interfaces

No single ground command to the EPS shall be capable of causing permanent damage to the EPS or any of its components or cause the EPS to enter an unrecoverable state.

5.3 Launch Vehicle Interfaces

5.3.1 Protection Devices

All power, command, and critical telemetry lines to or from the launch vehicle shall employ protection devices to preclude damage from lightning strikes, launch site radars, or anomalous voltages from launch vehicle or ground equipment. These protection devices may be either on the launch vehicle side or the spacecraft side of the interface, as long as their efficacy can be shown by test and/or analysis.

5.3.2 Telemetry Lines

Telemetry lines need not be protected unless deemed critical to mission success; however, damage to telemetry lines or circuits shall not result in damage to any other spacecraft equipment.

5.3.3 Testing of Redundant Paths

Where redundancy exists in power or command lines, it shall be possible to test the redundant paths separately to ensure the continued viability of each path prior to launch.

5.3.4 Loss of Launch Vehicle Power During Ascent

For space vehicles that receive main power from the launch vehicle during launch and ascent, the EPS shall be capable of tolerating a loss of power prior to space vehicle separation, such that the space vehicle will be capable of autonomously entering a self-powered state after separation.

Some past designs have had the unpleasant feature of being unable to command their own power systems on after even a momentary loss of launch vehicle power due to shock-induced chatter of a switching relay. Wherever possible, it is desirable to have the spacecraft self-powered through launch and ascent, rather than relying on the launch vehicle upper stage to provide power.

5.3.5 Space Vehicle Battery Protection

The power interface between the space vehicle and the launch vehicle, or between the space vehicle and the launch support GSE, shall be protected from faults, such that the space vehicle batteries cannot be unintentionally discharged.

5.4 Ground Support Equipment Interfaces

5.4.1 Protection Devices

Protection devices shall be employed to protect GSE interfaces with the spacecraft from damage due to malfunction or misapplication of the GSE.

5.4.2 Stability with GSE

The EPS shall be stable in all test configurations where power is provided by GSE. This shall be determined by stability analysis using the characteristics of the GSE and associated cabling.

5.5 Connector Keying

The design of all EPS equipment shall preclude the inadvertent misconnection of cables by using unique connector keyings or an equivalent technique.

5.6 GSE Isolation All GSE power sources that will be used to provide power to the spacecraft EPS shall be electrically isolated from the AC power mains.

5.7 Facility Ground

The spacecraft VRS shall include an electrical terminal useable for electrical connection to the facility ground network during spacecraft integration and prelaunch activities.

6.0 DETAILED REQUIREMENTS

6.1 <u>EPS Operation</u>

6.1.1 <u>EPS Trend Data Collection</u>

6.1.1.1 <u>Solar Array</u>

The design of the EPS shall include provision for the measurement, for trending purposes, at any point in time and anywhere in the orbit, of the I/V curves of at least 3% of the solar array strings for the spacecraft. Trend data acquired when the spacecraft is out of view of a ground station shall be stored until it can be transmitted to a ground facility.

As solar cell technology progresses, it is vital to obtain orbital trend data to avoid the kind of guesswork that plagues investigation of premature degradation phenomena. It is reasonable that, for existing spacecraft designs or incremental changes to existing designs, this requirement might be tailored out. But for the design of new power processing equipment, it should be a straightforward matter to include I/V curve measurements. This data would also be invaluable in verifying the proper operation of peak-power tracking or pseudo-peak-power tracking EPS designs.

6.1.1.2 <u>Battery</u>

Provision shall be made for battery voltage, current, temperature, and (for NiH2 batteries) pressure data to be retained as trend data. The update rate for this data shall be not less than once every second for voltage and current, and not less than once every 20 seconds for temperature and pressure. Trend data acquired when the spacecraft is out of view of a ground station shall be stored on the space vehicle until it can be transmitted to a ground facility for use by operations personnel.

6.1.2 <u>Dead Bus Recovery</u>

In the event of fully depleted batteries due to loss of insolation on the solar arrays, provision shall be made for recharge of the batteries if insolation is restored.

6.2 <u>Power Quality</u>

Power quality requirements for the spacecraft power bus, which includes the combined effects of the EPS and the utilization equipment as a whole, shall be per the following subparagraphs for time-domain effects.

This TOR imposes power quality requirements only in the time domain, and only at power distribution points, such as the main bus. Both the ripple and transients are determined by characteristics and interactions of and between the EPS and the utilization equipment.

6.2.1 Ripple

The EPS, in concert with the utilization equipment as a whole, shall not generate a ripple voltage at the main bus or at other distribution points with a peak-to-peak magnitude greater than 7% of the nominal bus voltage.

See Appendix A, Section V, for discussion on the derivation of this requirement.

6.2.2 Transient Voltages

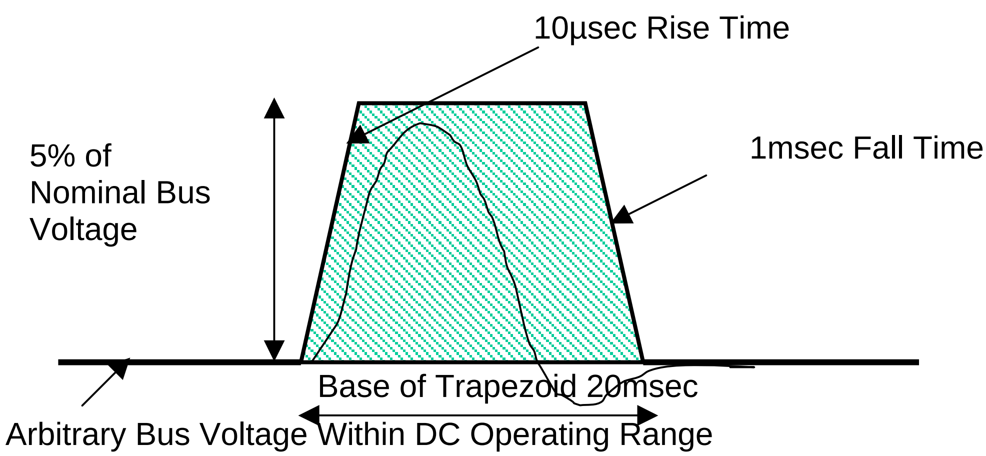
Note: The use of "should" instead of "shall" in some cases in this section occurs when discussing whether load equipment must remain operational or not during transients. As this is the EPS specification and not a system-level document, it would be inappropriate to specify how the loads must respond. The EPS specification merely describes the range of behaviors of the bus itself. The "should" statements are included only as recommendations.

6.2.2.1 Step Load Transients

According to the overshoot and undershoot requirements in the next two paragraphs, the bus voltage may go 5% above and 5% below the allowable DC voltage range, as specified in Section 4.3.1. It is imperative that in specifying the DC voltage range, these expected surge voltages are taken into account. Designers of utilization equipment must be aware that their equipment must operate through these surges outside the DC range. This effectively widens the required operating range for equipment, although steady-state operation over this extended range is not required.

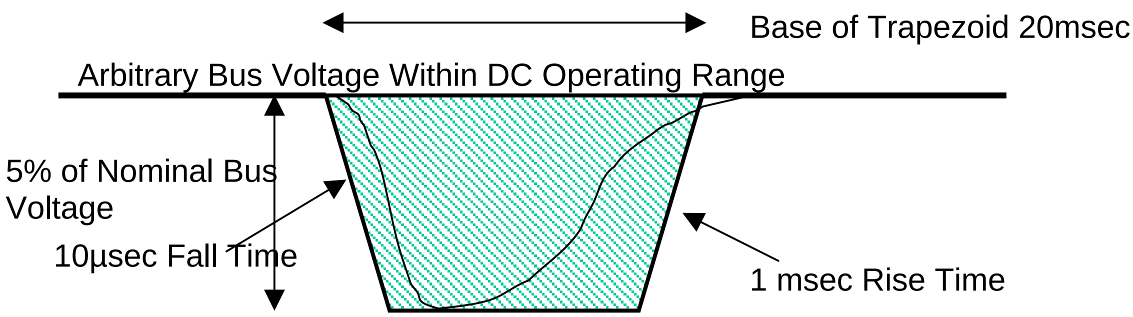
6.2.2.1.1 Overshoot Surges

The voltage overshoot of the power bus in response to the worst expected step load change shall not be greater than 5% of the nominal bus voltage, per the following diagram. The positive portion of the voltage overshoot shall remain within the shaded trapezoidal area. All utilization equipment should operate normally through this transient.



6.2.2.1.2 Undershoot Surges

The voltage overshoot of the power bus in response to the worst expected step load change shall not be greater than 5% of the nominal bus voltage, per the following diagram. The negative portion of the voltage undershoot shall remain within the shaded trapezoidal area. All utilization equipment should operate normally through this transient.



6.2.2.2 Fault-Clearing Transients

6.2.2.2.1 Overvoltage Surge

The peak voltage reached at any distribution point during a surge due to the clearing of a fault shall be less than 150% of the highest allowable value of DC bus voltage with a rise time greater than 10µsec and an impulse strength less than half the allowable value of DC bus voltage times 5 msec. Non-essential utilization equipment shall survive this transient, but they should not be required to operate normally through it. Essential equipment should operate through the transient or, at a minimum, recover to a safe, defined state at the end of the transient.

The peak overshoot on fault recovery stems partly from the stored energy in the source impedances leading to the load branch with the fault, interacting with the response of the loads as a whole. There can be other phenomena as well, depending on the EPS topology. For instance, in a switched solar-array system, the overshoot due to overcharging of the bus capacitance while the error amplifier sequentially switches off solar arrays strings after the fault clears may be substantial. The level specified, 150% of the highest voltage (i.e., the surge itself is 50% of and superimposed on the high end of the allowable DC voltage range) for 5msec, is a level which brackets unusually high, but still plausible, levels of overshoot. It is a level that should be readily accommodated by the load designers.

6.2.2.2.2 Undervoltage Surges

The voltage at any distribution point during a short-circuit event in the utilization equipment or in the power distribution equipment shall be assumed to fall as low as zero volts DC, with a fall time of 10µsec, returning to within 5% of the nominal bus voltage within 100msec. Non-essential utilization equipment shall survive this transient, but they should not be required to operate normally through it. Essential equipment should operate through the transient or, at a minimum, recover to a safe, defined state at the end of the transient.

The actual distribution-point voltage observed during a short-circuit event is determined by voltage division between the source impedance leading up to the fault location and the fault resistance to ground or structure. In most cases, the voltage would not drop by more than 50%, but it is possible for the bus voltage, in the case of a severe short in an EPS with high source resistance or source-current limiting, to be arbitrarily low, so a lower limit of zero volts is specified here.

The total transient due to a fault includes the undervoltage surge from onset of the fault up to the moment of clearing the fault (i.e., fuse opening or circuit breaker tripping), as well as the subsequent overvoltage spike and surge after the fault clears. The requirement here for the undervoltage surge is intended to set a time limit on how long the bus voltage is suppressed, that is, it could be as bad as zero volts for 100msec. But the 100msec would also include the fault clearing and the subsequent overvoltage events. As long as the bus returns to within 5% of its nominal voltage within 100msec, the requirement is met.

6.2.2.3 Short-Duration Aperiodic Spikes

6.2.2.3.1 Positive-Going Spikes

Short-duration aperiodic positive transients at any power distribution point, including those arising due to fault-clearing events, shall be limited to a peak value less than 50 volts above the DC level with a voltage-time area or impulse strength less than 0.5 volt-milliseconds.

The phenomenon of the microsecond spike is due to the rapid discharging of the energy in the wiring inductance in the faulted branch. In a battery-backed bus, fault currents in loads can reach several hundred Amperes. The wiring inductance to the faulted load can hold considerable energy. When the fault clears, the fuse absorbs some of the energy, but much of it dumps into adjacent loads. The values given here are based on testing using typical fuse values and wire lengths.

It is essential for the system designer to understand the characteristics of the fuses used, particularly the actual opening time (not the total clearing time, but the very fast time going from conduction to no conduction), as this determines how much inductive energy is dissipated in the fuse. Typical fuses used currently have very rapid opening times and no internal arcing, which makes inductive spikes higher. The only way to understand what kind of spikes will occur is to conduct tests using flight-like cabling, fuses, and, if possible, emulated loads. Computer modeling is possible, but it is very difficult to model the cable impedance during the fuse clearing, as the skin effect provides damping in excess of that provided by the DC cable resistance. Fuse modeling is similarly tricky.

Note also that longer cables do not necessarily give the worst-case spike magnitude. Although the inductance is higher for longer cables, the fault current is less, due to the higher cable resistance. Since the inductive energy is $\frac{1}{2} LI^2$*, the resistance is a more* *important factor than the inductance. There is actually a specific length that gives the worst-case energy, which can be found by simple calculus.*

Once representative spike behavior is understood, measures can be taken to limit the spikes seen by adjacent loads. Capacitive loading or transient suppression devices will probably be necessary to limit spikes to 50V at many distribution points.

Finally, it should be pointed out here that the spike levels we are talking about for fault-clearing events have a very large impulse strength. There is no separate requirement for "normal" aperiodic spikes. Typical spikes due to load switching or other fast rise time events are much smaller (~ 1 volt-μsec) than fault-clearing spikes. Since loads must pass their CS06 requirement, which is four times the level of the fault-clearing spikes, there is no need to specify a magnitude for the smaller variety of spikes.

6.2.2.3.2 <u>Negative-Going Spikes</u>

Short-duration aperiodic negative transients at any power distribution point, including those arising due to fault-clearing events, shall be limited to a peak value not lower than -25V, with a voltage-time area or impulse strength less than 0.25 volt-milliseconds.

The negative-going spike follows the positive spike. It is part of a damped ringing response. Equipment designers must ensure that it causes no harm.

6.3 <u>Energy Management</u>

Energy management design evolves during the design phases of a spacecraft from initial sizing estimates to final, measured performance in vehicle test, with orbital performance as the final indicator of success.

An energy balance analysis shall be performed for the EPS. The results of the analysis shall be expressed in terms of a power margin (see 3.5.1.3.1).

6.3.1 <u>Minimum Stored Energy</u>

For the particular type of energy storage technology chosen, a minimum stored energy level shall be defined. For battery technology, this level may be expressed as the maximum-allowable percentage Depth-of-Discharge (DOD) (relative to a fully-charged battery) or minimum-allowable State-of-Charge (SOC) (from a depleted battery). The EPS design shall ensure that the stored energy does not fall below the defined minimum level at any time during the mission design life of the spacecraft, while in any anticipated operating mode, including abnormal modes.

An exception to this requirement is that for events or combination of events shown to be very rare or highly improbable, the stored energy may fall to 67% of the defined minimum energy storage level if the calculations showing this comply with worst-case analysis requirements defined in 6.3.5.1.

This section is worded generally to allow for flywheels, ultracapacitors, or other energy storage technologies, even though batteries are the mainstay of spacecraft energy storage at this time.

For a given battery type in a given orbit, the maximum DOD is set according to the maturity of the technology and the number and depth of charge/discharge cycles expected over life. This is a target level that should be observed for the vast majority of cycles in order to maximize battery life. However, there are events and confluences of events that could result in exceeding the maximum DOD under rare circumstances. For example, a loss of sun pointing (and thus battery charging) followed by a sun-seek operation just prior to eclipse is a rare, but not unreasonable, scenario. To demand strict adherence to the maximum DOD requirement in this case is not reasonable, since the battery life should not be adversely affected by one or a few deeper discharges over the mission.

6.3.2 Solar Array Analysis

An analysis shall be performed to determine Beginning-of-Life and End-of-Life power availability of the solar arrays, using the degradation factors listed in Appendix A. The results shall be used in the energy balance analysis.

6.3.3 Battery Analysis

An analysis shall be performed to determine the Beginning-of-Life and End-of-Life performance of the batteries. Life-test data shall be used to determine the worst-case expected capacity over the expected operating temperature range, the worst-case terminal voltage as a function of current in charge and discharge, and the charge efficiency as a function of SOC, current, and temperature. The results of the analysis shall be used in the energy balance analysis.

6.3.4 Power Consumption

6.3.4.1 Utilization Equipment

All spacecraft utilization equipment shall be included in a power budget spreadsheet. The consumed electrical power for each piece of utilization equipment shall be included at nominal, maximum, and minimum bus voltage. Alternatively, the consumed electrical power for each piece of utilization equipment may be stated as the parallel combination of constant power, constant resistance, and constant current components.

6.3.4.2 Losses

Power losses are due to wiring and contact IR drops and to conversion efficiencies in power converters. Losses internal to an individual piece of utilization equipment shall be expressed as part of the overall power consumption of that piece of equipment, per Section 6.3.3.1.

Power losses due to wiring, fusing, and contact IR drops in the vehicle wire harnesses shall be expressed independently of the utilization equipment, using estimations and/or measurements commensurate with the design phase of the spacecraft, per the guidelines of Appendix A.

6.3.5 Energy Analysis Methodology

Energy analysis shall be performed for a set of Design Reference Cases encompassing the worst-case power margin in every anticipated normal and abnormal operating mode, to determine whether the stored energy meets the minimum stored energy level, as defined in Sec. 6.3.1. This energy analysis shall use worst-case estimates for all relevant mathematical quantities, as defined below.

6.3.5.1 Worst-Case Methodology

Mathematical quantities contributing to the stored energy analysis shall be assigned values (over the range of possible values) that result in the worst-case calculated value of stored energy. For any given orbital, environmental, or operational condition, it is permissible to assign only values consistent with that condition. It is not necessary to use contradicting worst-case values that could not realistically occur simultaneously within that condition, even if they do result in a lower calculated value of stored energy.

This paragraph is basically saying that it is all right to use a realistic worst-case approach, rather than a worst-worst-case approach. For example, it is not required to use aphelion sun along with the longest eclipse length, if it is true that the longest eclipse length occurs at equinox. It is appropriate to match the eclipse length with the seasonal solar intensity corresponding to the longest eclipse.

6.3.5.2 Timelines

Time-varying mathematical quantities contributing to the stored energy analysis (such as the power consumption of the utilization equipment) shall be expressed in an electronic tabular form that can readily be parsed by computer software, showing changes in the value of the quantity versus time, over the period of interest. The use of averaged quantities rather than a timeline is permissible only if it can be shown that averaging results in a value of stored energy lower than one would achieve using a detailed timeline.

6.3.5.3 Application Guidelines

The stored energy analysis shall be performed per the guidelines of Appendix A of this specification.

6.4 Power Distribution

6.4.1 Wiring Requirements

Wiring in the EPS shall conform to the requirements of DOD-W-83575A.

6.4.2 Wiring Thermal Analysis

A thermal analysis shall be performed for all harnesses containing power wiring showing the worst-case temperature rise in orbital operation due to self-heating and all other sources of heat.

6.4.3 Fusing

A fusing analysis shall be performed to ensure that the downstream fuse in any power wire will always blow under fault conditions before harm is done to the upstream fuse, or, in the case of unfused wires, to the upstream power wire.

6.4.3.1 Inrush Current

An inrush current analysis shall be performed to demonstrate that the expected inrush current into any fuse-protected piece of utilization equipment, due to the sudden application of a voltage step from zero volts to the nominal DC voltage of the equipment, with a rise time of 1msec, shall not be capable of operating or stressing the fuse or result in the unintended activation of or damage to any switch, either mechanical or solid-state, or protection circuit.

Some spacecraft use solid-state switches to control power to the loads. These devices can contain current-limiting circuitry as well. Neither the switch nor any of the protection circuits should inadvertently trip due to an inrush current to any load.

6.4.3.2 Outrush Current

An outrush current analysis shall be performed to demonstrate that the expected outrush current out of any fuse-protected piece of utilization equipment, due to the suddent drop of its input voltage from the nominal DC level to zero volts, with a fall time of 10μsec, shall not be capable of operating or stressing the fuse or result in the unintended activation of or damage to any switch, either mechanical or solid-state, or protection circuit.

Note that solid-state switches may be prone to damage by outrush current due to MOSFET body diode conduction.

6.4.3.3 Wires as Fuses

Power distribution wires , including bond wires in electronic devices, shall not intentionally be used as fuses.

6.4.3.4 Unsealed Fuses

The use of unsealed fuses in EPS applications is prohibited.

Appendix A

Application Guidelines

I. Preface

The following paragraphs provide additional background material for the key requirements in this TOR. Eventually, when the TOR is revised and molded into the new MIL-STD-1539A, this appendix will be modified and expanded by moving much of the italicized material into it, as well as by adding additional technical detail and other rationale for requirements. A good model for the eventual form of this appendix is the appendix in MIL-STD-462D, which was also incorporated into MIL-STD-461E. That appendix provided a necessary adjunct to the requirements for the measurement of electromagnetic interference by thoroughly explaining the rationale for each requirement and test method.

II. Energy Balance Analysis

A. Discussion

Every spacecraft must achieve positive energy balance, that is, it must produce more energy than it consumes. If negative energy balance existed, the batteries would be completely drained, and the spacecraft would cease to function. Energy balance may also be defined in terms of "power margin" or "load power margin," which is the amount of extra loading (in Watts) that can be added to the worst-case expected load of the spacecraft in order to produce a battery maximum DoD (as defined by some battery or mission requirement). For example, consider a spacecraft with a worst-case average orbital power of 2000W that results in a battery DoD of 50%. If the maximum allowed DoD for the mission is 60%, this means we could hypothetically add more loads. A rough, first order calculation tells us that each 400W of load contributes 10% DoD, so we could add 400W to the 2000W load, and the resultant total DoD would be 60%. We then say that our power margin is 400W. Note that a more exact analysis could produce a somewhat different result due to the nonlinear relationship between load power and battery voltage (which drops during discharge).

As it is absolutely critical to maintain positive energy balance over the life of the mission, there are many factors that must be precisely known both at BOL and EOL. Of course, if a spacecraft were designed with a great deal of extra solar array power, the energy balance analysis would not be so critical, but this is almost never the case. Even if it were, a precise energy balance analysis could be useful in determining the life expectancy of such a system, as eventually, inevitably, the arrays and batteries will degrade.

Since the energy balance analysis is so critical, it is imperative to know the right and wrong ways to do it. Basically, as long as the analysis is kept on the conservative or worst-case side, one would think a simple hand calculation might suffice. For example, if we take our most pessimistically-low EOL solar array power at our lowest expected battery voltage with our highest expected loads (taken as a constant over the orbital period) and the worst-case battery capacity with one cell failed, and we add double the expected harness losses, and we still show positive power margin, then we might not necessarily have to go any further.

But there are excellent reasons to always do a rigorous energy balance spreadsheet model, or even better, a precise simulation using either a custom computer code or commercial simulation software:

- all information used for deriving the solar array BOL and EOL I/V curves is in one place, so that all assumptions are plainly verifiable
- using timelines of load powers, sun angles, and mission events instead of just orbital averages gives a more accurate result than a hand calculation can provide
- using an accurate battery model that gives voltage as a function of current, DoD, and temperature yields more accurate results
- a good model can easily and quickly be rerun for parametric studies, contingency analysis and other what-if scenarios
- models are adaptable to situations not amenable to hand analysis – e.g. transfer orbit and safe-hold anomaly recovery scenarios
- thermal load effects are more easily tied in. Energy balance model can be coupled to thermal model
- simply building an accurate model contributes to an enhanced understanding of EPDS operation by subsystem and component engineers alike
- a model can be adapted for use as a launch training tool, and as a tool to evaluate EPDS performance and battery health during the actual launch

B. Energy Balance Analysis Outputs

The results of an energy balance or power margin analysis can be expressed simply in terms of the maximum battery DoD or minimum SOC, or in terms of the overall power margin of the EPDS. A preferred approach is to provide a plot showing battery DoD or SOC -- alone or with other key parameters such as battery voltage and temperature, load levels, relative sun intensity, etc. – versus time, over some meaningful interval such as one orbital rev.

Along with the results of the analysis must be included the mission conditions, such as the time of year, BOL or EOL, the beta angle, the load profile, failed cells, etc. Failure to provide these assumptions with the results renders the analysis useless.

Some contractors use specialized tools for power system analysis. Some of these are impressive tool sets containing numerous pre-built models for components and equation sets for certain types of calculations. In the hands of a capable analyst, these tools simplify the analysis task and can give very precise results. However, in the hands of an inexperienced user, or if the input dataset contains errors or if false assumptions are used, the results can be wrong and very misleading. These results may be presented at design reviews and give the impression that all is well in the power margin department, when in fact, the design does not close and the mission will be lost if the spacecraft is ever launched.

To avoid this possibility, it is vital that the analyst provide a complete listing of all models and assumptions used by the analysis tool. It is a good idea to have the analyst perform several “calibration” runs to verify the correct behavior of the various model elements. The items to verify are the battery voltage behavior for a standard charge/discharge profile (such as the acceptance test profile), the solar array I/V curves at different temperatures, and the appropriate PMAD losses and conversion efficiencies.

A typical plot showing battery DoD and voltage over a single rev is shown in Fig. II.B-1. It is for a GEO mission, and it begins at the entrance into a 72-minute eclipse. The battery voltage drops during eclipse, and rises at eclipse exit. The DoD drops and rises during the same interval. We can observe the peaking of the battery voltage near the end of charge, as well as the slight curvature of the DoD curve due to the decreased charge acceptance efficiency at high DoD. Finally, we see the battery voltage step downward at the switchover from full charge to trickle charge. The maximum DoD reached in this example is 48%, and the battery is easily recharged well in advance of the next eclipse.

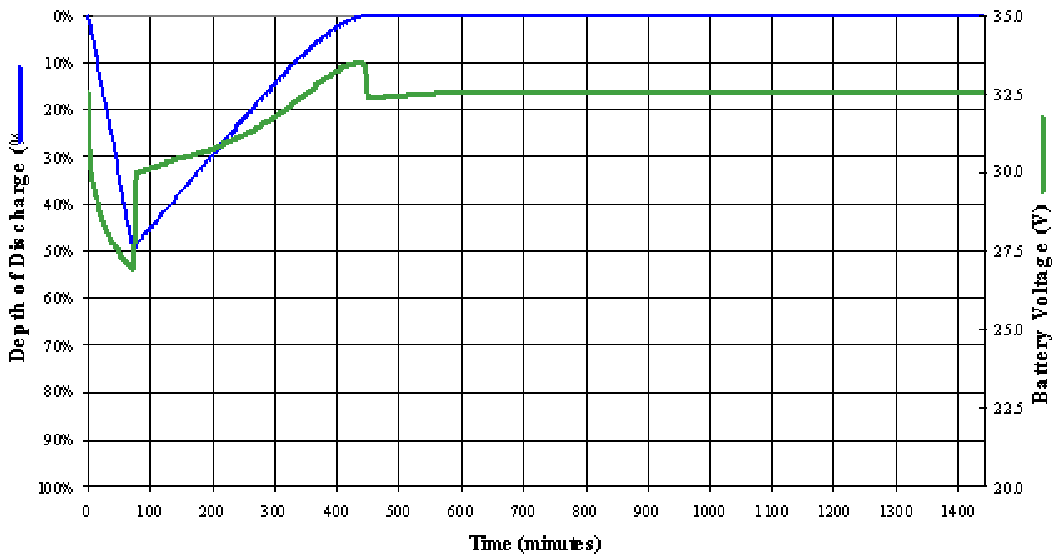


Figure II.B-1 Typical DoD Plot for an Energy Balance Analysis

C. <u>Checklist of Energy Balance Parameters</u>

Solar Arrays:

- BOL and EOL I/V curves of the Cell-Interconnect-Coverglass assemblies (i.e., the cell I/V characteristics with the chosen coverglass), along with temperature coefficients for Isc, Voc, Imp, and Vmp.
- The number of series cells per string and strings per group. This allows construction of an equivalent BOL and EOL I/V curve for a group.

- The array operating temperature profile for BOL and EOL. If this is a constant during sunlight, use it. If it varies much, the model must take that into account, as temperature drastically changes the power available from the array.
- Array degradation factors – cell mismatch, UV darkening, cracked cells, failed strings, deviation from cosine law (for off-angle sun), cover glass edge trapping
- Shadowing effects, if any (this is very difficult to factor into the energy balance analysis – it requires 3D modeling of some sort. It is not generally acceptable just to reduce the array's "available power" figure by the percentage of the array that is shaded).
- Wiring resistances in the total path of each string or group
- Voltage drop due to blocking diodes in the string or group path

Batteries:

- Battery voltage – the recommended approach is to use a model (either a physical model or a table lookup model) that gives the battery voltage as a function of current, temperature, and SOC for both charge and discharge. This data should be derived from cell acceptance or life data and adjusted by multiplying by the number of series cells and including wiring drops. In some cases, it may be acceptable to use an average midpoint voltage for charge and discharge, but it must be demonstrated that this gives worst-case performance for the EPDS in question
- Include the one-cell failed scenario
- Wiring harness resistance

Loads:

- If loads are relatively constant, an orbital average may sometimes be used
- If loads fluctuate a lot, or if there are high peaks, it is better to use a detailed timeline
- If the mission payload operations are highly variable, it is advised to establish a "Day in the Life" design reference case which can include a detailed timeline corresponding to the highest anticipated payload duty cycle
- Worst-case (cold) heater power predictions should always be used for the energy balance analysis, unless significant flight history exists for a very similar vehicle. In that case, flight averages may be used as long as some margin is added

Power Management and Distribution (PMAD):

- Where wiring resistances, diode voltage drops, and power converter conversion efficiencies are known, the total power losses may be calculated or simulated. The positive temperature coefficient of wire resistance means that if power wires are run at 100C (50% of their 200C rating, which is the maximum allowed), the resistances – and thus the losses – will be almost double those at room temperature. It is important to factor this into the analysis.
- If nothing is known about the wiring, assume 5% of the total maximum load power.
- If the wiring details are not known, but the spacecraft is of a heritage design with flight data that allows wiring losses to be estimated, then the total wiring loss may be estimated

at 2 to 3% of the total maximum load. Whatever values are used, a solid justification should be provided.

- Diode drops should be a realistic worst-case maximum. This usually may mean using the cold-temperature value, even if the actual operating temperature is higher.
- Charge or discharge converters in the PMAD have conversion efficiencies which are generally a function of the charge or discharge current. Furthermore, sometimes this efficiency includes the standby or housekeeping power consumed by the converter. If this is factored out and booked as a separate load, it decreases the effective variation in the true conversion efficiency of the converter over the range of minimum to maximum current. If the energy balance model is sophisticated enough to use a variable efficiency as a function of current, that is fine, but a simpler approach is to use the lowest efficiency over the expected range of currents

Orbital:

- Day of year affects both eclipse length and seasonal sun intensity. For energy balance, it is not necessary to make each of these worst-case if that combination is not possible. For example, a GEO analysis worst case is the longest eclipse (72 minutes) that occurs during eclipse season. The sun intensity used should correspond to the sun on the day of the longest eclipse, which is always around the time of the equinoxes, so the intensity relative to AM0 is about 1.0. There is no need to use the aphelion sun intensity of .965, since this occurs in summer, just past the summer solstice.
- Finding the worst day of the year sun-wise can be complicated when solar arrays are used that are not sun-tracking, or are only partially sun-tracking. When the array normal vector does not align with the sun vector, the angle between them must be determined. If this angle changes not only with beta angle but over an orbital revolution as well (as would be the case with an agile spacecraft, for instance), it can be very difficult to find the worst case. Orbital-analysis software can be employed to assist in the task.
- Transfer orbit can be especially difficult to analyze. The arrays may be folded, the vehicle will be spinning, and the sun angle to the arrays may shift many times. Only essential electronic loads will be powered, so the heater loads – particularly the battery heater – represent a larger proportion of total load. The battery heater in particular steals available power from the arrays, making it unavailable for the other. loads or for charging the battery. To know how much power the battery heater requires takes a detailed power/thermal model.
- Solar array shadowing requires first of all a 3D model or equivalent that can locate shadows cast from appendages, such as booms or antennae. A behavioral, dynamic model of the EPS may be used to gauge the stability of the EPS with some array strings shaded. The energy balance analysis may be performed in the worst-case by assuming zero power from shaded strings; a more sophisticated model could estimate a degraded power level from shaded strings and use that in the energy balance analysis.

III. Bus Stability

A. Discussion

Negative feedback techniques are used throughout the EPDS and in the loads. Most regulated loads are of the constant-power variety, which gives them a negative input impedance. Unless thorough worst-case analysis is done, instabilities may occur, either at the spacecraft main bus, or locally in a load. It is even possible for instability to occur at the solar array itself. Some instabilities might be apparent at BOL; others could possibly manifest only after some time on orbit. Some instabilities may be mere nuisances, not harmful in any way, resulting in at most a few tens or hundreds of millivolts of low-frequency voltage ripple that does not cause any malfunction or degraded operation. Other instabilities, though, may be catastrophic.

One such case was the discovery that a particular EPDS design, which in one mode regulated the solar array voltage rather than the bus voltage, contained a resonance between the solar array and the input filter that would have caused massive oscillation on orbit that would have destroyed the power processing hardware. The resonance was not discovered in test because a low-impedance voltage source had been used instead of a high-fidelity solar array simulator (SAS). Previous analysis had not detected the problem because a critical component – a feedthrough capacitor -- had been left out of the analysis. Fortunately, an independent analysis showed the critical instability, and a damping network was added to the hardware prior to launch. The contractor was persuaded to thenceforth employ a SAS in their testing, in line with "Test-Like-You-Fly" principles.

B. Stability Analysis

As mentioned in Section 4.4, stability analysis can be very difficult if there are many loads that may be switched on or off. Further complicating the situation is that the EPDS may have many different operating modes or regimes, each of which must be analyzed separately. It is also sometimes the case that one must consider intermode stability. For example, consider a regulated bus coming out of eclipse. At some given effective sun intensity during penumbra, the discharger current drops to zero. The discharger control loop drops out, and the array regulation loop takes over. The battery charge loop may also try to kick in, but if the commanded charge rate is too high, this sags the bus and the discharger loop may come back into play and the array and charger loops will cut out. Then the whole process repeats. The result is a cycle of mode-hopping which persists for some time until the sun intensity is high enough to ensure operation with the array and charger loops only. Depending on the specific design details, this could be a harmless situation or a harmful one.

To analyze a system for stability for one set of loads and in one operating mode or regime, a commercial circuit simulator is usually employed. Circuit models must be made for all of the loads, and a model for the source impedance of the bus or distribution point must be made as well. Large signal models may be used, which requires that the simulator perform the small-signal linearization for the AC analysis, or the small-signal linearization may be done manually. To model a load, the schematic for the input filter is used, and it is terminated by a constant power load. For a large-signal model, a constant-power behavioral block must be available; if the small-signal linearization is done by hand, the constant-power load is represented by a negative resistance

$$R = -\left(\frac{V}{I}\right)$$

or,

$$R = \frac{-P}{I^2} \qquad \text{since} \qquad P = V \cdot I$$

where V is the load DC input voltage, I is the current at that voltage, and P is the power consumed by the load. A typical model for a load is shown in Fig. III.B-1.

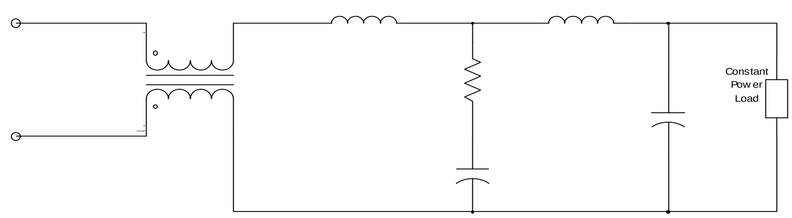
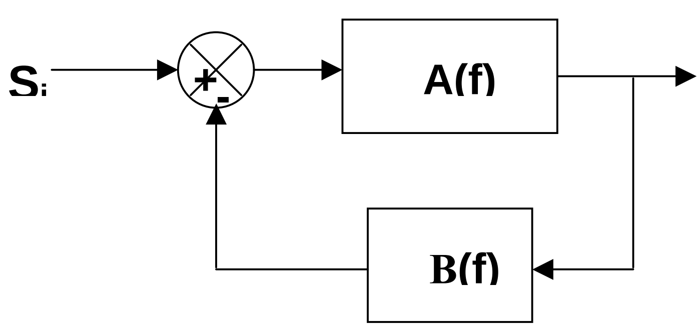


Figure III.B-1 Typical Input Filter Model

There are two types of stability to be considered in power systems. The first is feedback stability. In any feedback system, the stability at any given operating point may be determined by either Bode plot analysis or root-locus analysis of the loop gain.



$$S_o = S_i \cdot \frac{A(f)}{1 + A(f) \cdot \beta(f)}$$

Figure III.B-2 Canonical Negative Feedback System

In the canonical feedback system shown in Figure III.B-2, the gain G of the system is

$$G = \frac{S_0}{S_i} = \frac{A(f)}{1 + A(f)\cdot\beta(f)}.$$

The term in the denominator is called the loop gain T.

$$T = A(f)\cdot\beta(f)$$

The gain and phase of T may be plotted on a Bode plot, as in Figure III.B-3, or the locus of 1+T may be plotted on a Nyquist graph, looking for encirclements of the (-1,j0) point.

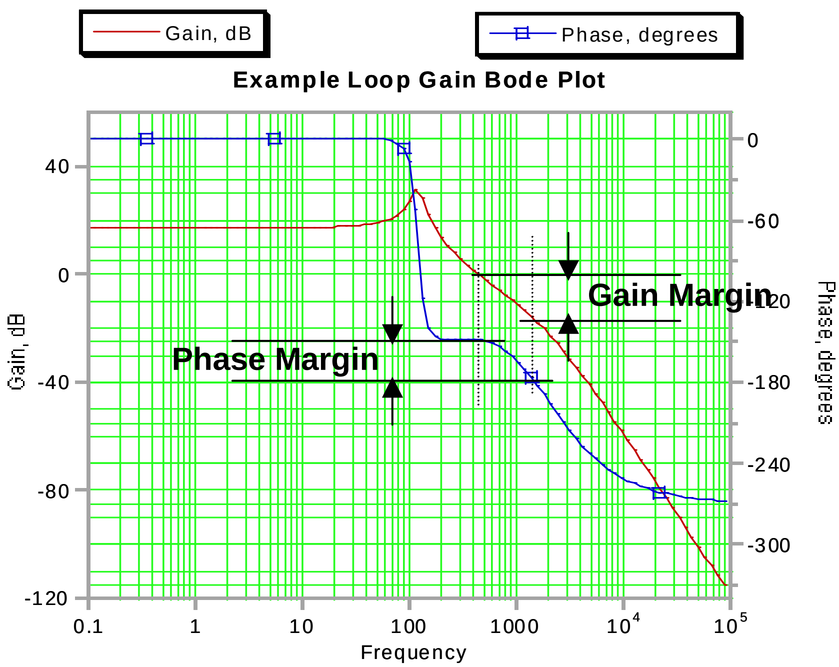


Figure III.B-3 Bode Plot

The *phase margin* is the number of degrees above -180 at the frequency where the magnitude crosses 0 decibels (~30 degrees in this example). The *gain margin* is the number of dB below zero dB of the gain at the frequency where the phase crosses -180 degrees (~18 dB in this example).

The requirement for stability in feedback systems, per Section 4.4 of this TOR, is 60 degrees phase margin and 10dB gain margin at beginning-of-life. Verification that a design meets the requirement is demonstrated via detailed circuit modeling and simulation of the EPDS components along with the loads. For systems that contain multiple feedback loops that are not active concurrently, each loop may be analyzed separately for stability. For systems with multiple loops that *are* active concurrently – such as nested loops – the analytical expression for the loop gain is much more complex. Nested loops may be approached by first analyzing the inner loop by itself, then analyzing the outer loop with the inner loop treated as a fixed transfer function. For more complex loops, other methods of classical or modern control theory may be utilized.

The second type of stability to be considered in power systems is that at the main bus or at any other power distribution point. Whenever constant power loads are used and there is an interface between a power source with output impedance Z_S and a load with input impedance Z_L, as shown in Figure III.B-4, we must check for source/load interactions that could cause instability. We say there is source/load interaction if

$$\left|Z_S\right| \cdot \left|Z_L\right| \geq 1.$$

Graphically, if we plot the magnitudes of Z_S and Z_L as functions of frequency f or ω, we will see the two curves cross when this relation holds true, as shown in Figures III.B-5a. The case where they do not cross is shown in III.B-5b.

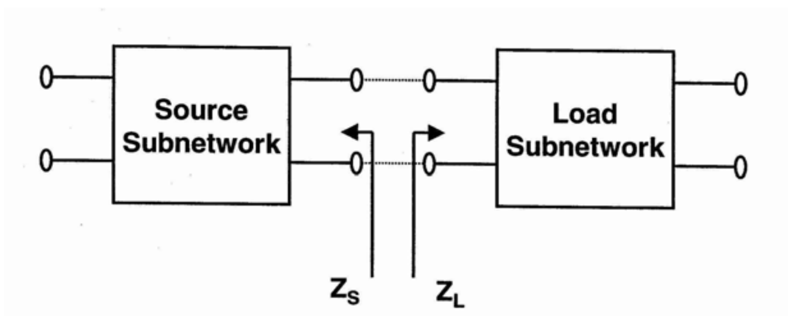
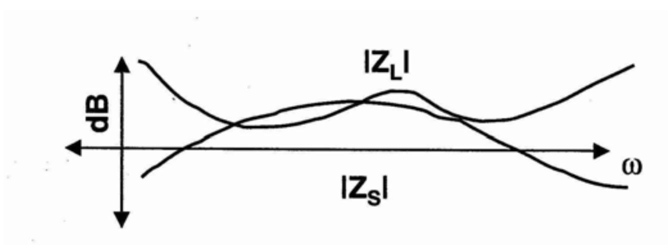
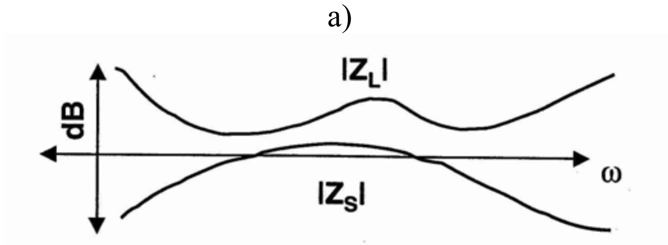


Figure III.B-4 Interface Between Source and Load Subnetworks



a)



b)

Figure III.B-5 a) Interacting Source and Load b) Non-interacting

The stability of the interface is evaluated by plotting the function Z_S / Z_L on a Nyquist chart, as shown in Figure III.B-6. Gain and phase margins for the interface are as shown in the plot.

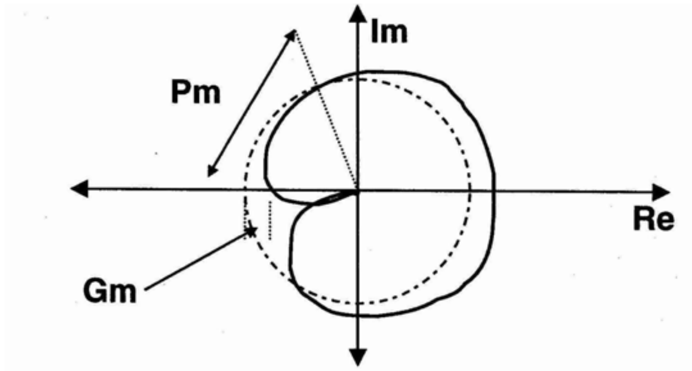


Figure III.B-6 Nyquist Plot of Z_S / Z_L

The source impedance Z_S is the complex impedance looking towards the source. It can include the effects of source capacitance (including equivalent series resistance effects), harness resistance and inductance, battery impedance, and dynamic output impedance of regulator stages. The load impedance Z_L includes harness resistance and inductance, load filter impedances, and the dynamic impedances of constant power, constant current, or resistive loads beyond the filter.

For either feedback stability or source/load stability, when the stability margin is low, the system may be stable, but it may be highly underdamped. It is critical to perform step-load testing at the system level, and to check equipment input-filter responses to bus voltage step loads to ensure proper operations.

To evaluate the stability of the bus with multiple loads connected can be a difficult task. First, it may be difficult to obtain schematic diagrams for load front ends from all the subcontractors involved in the design of bus and payload equipment. Second, loads may be switched onto or off of the bus, or may have multiple operating modes, each of which would require a different model. Third, when multiple loads are powered from the same bus, there will be interactions between their poles and zeros, creating new poles and zeroes.

The first issue is solved by writing the contract for the spacecraft such as to require release of a simplified load model from each vendor for use in the stability analysis. The second and third issues may be addressed as follows:

1) if the number of combinations of loads and load states is not excessive, a circuit simulator may be used to combine all the loads together and determine the frequency response of the dynamic impedance of the whole network (treated as a two-port). It is essential that the AC analysis be carried out at the proper DC operating point, if the computer software is performing the small-signal linearization automatically.

Once the frequency response of a group of loads is determined, it may be desirable to employ network synthesis techniques to build an equivalent 2^{nd}, 3^{rd}, or 4^{th} order model.

The process is repeated for all combinations of load and load states. Each resulting Z_L may then be plotted (as Z_S / Z_L) as shown above.

2) if the number of combinations of loads and load states is excessive, it is not feasible to simulate each configuration as a whole. In this case, the only practical recourse is to measure the impedance of each load independently, and then use a program such as MATLAB to systematically sum the admittances of the loads that are switched onto the bus, excluding those that are switched off. For N loads with two states each (ON and OFF), this results in 2^N combinations. This ignores the possibility of pole-zero interactions between loads, but it is highly likely that the worst-case stability margin lies somewhere within the 2^N load combinations. The actual tests of Z_S / Z_L for each load combination can also be automated within MATLAB, making the whole process fairly quick.

3) depending on the actual load configuration, a combination of methods 1) and 2) is possible. A core group of loads that are always connected may be modeled as a single impedance, while switchable loads may be handled via the combinatoric method.

IV. Single-Point Failure Mode Mitigation

The following are guidelines used in power system design to avoid single-point failure modes. The main bus is the most worrisome area, since it is unfused; a permanent short circuit takes down the mission. This risk is mitigated by employing proper spacing between conductors, and by using insulation thicknesses that can withstand the highest anticipated wear-through of the insulation, due to launch vibration or other factors, with margin. Besides the main bus, SPFM's are avoided through the use of redundancy in all EPS elements. But even where redundancy exists, one has to check for SPFM's at cross-strapping points.

Here is a summary of typical methods for mitigating SPFM's::

- Bus bars -- must have adequate spacing between bus bars and other components. Have thick insulating washers to separate bus bars from the supporting structure, as well as thick insulating grommets to prevent bolts from contacting the bus bars. Use reliable means of ensuring that wires stay connected to the bus bar, i.e. two clamping screws per wire.
- Wiring -- double insulation must be used wherever possible to ensure that manufacturing or workmanship flaws cannot result in an inadvertant short, especially at susceptible points such as at splices and where wires or cables pass over sharp or rough edges.
- Connectors -- unfused power running through connectors is hazardous. Prefer using large fuses between bus bar and connectors, using several wires such that if one wire shorts at the connector, the fuse will blow and only one of the current paths is lost.

- Unfused power on printed wiring boards (PWBs) -- employ adequate spacing between conductors and box structure. If power goes to the case of a power device such as a BJT or MOSFET, double insulation of adequate thickness must be used between the device and the PWB conductive areas under the device, even though this will give a poorer thermal conduction path for heat dissipated in the device.
- Capacitors -- capacitors across unfused power points must be series-connected or fused
- Feedthrough filters -- since these cannot generally be fused, they represent a SPFM. The only way to ensure high reliability is to use Class S parts or upscreened parts. Ensure proper installation via test and visual inspection.
- Transient suppression devices -- where possible, use series fuses and analyze to be sure the fuse will not blow due to a normal transient, only for a the transient following a fault. Again, use Class S or upscreened parts. It is necessary for the rating of each transient suppression device used to be sufficient to handle the total fault energy, since it would be expected that in a system with multiple transient suppression devices there would be one that would have a lower trip point than the rest, and would thus hog all of the energy of the fault-recovery transient.
- Cross-strapping points -- wherever redundant units of any kind are cross-strapped together, there can be a SPFM introduced. It is important to evaluate designs carefully to check for this possibility.
- Power electronics -- A/B redundancy or N+1 redundancy is used for the converters and control circuitry.
- Solar array shunts or switches -- assume one will fail over life
- Solar array -- assume some number of failed strings and cracked cells over life
- Batteries --- use cell bypassing in case of a cell short; use N+1 cells per battery, where N is the minimum number of cells required to meet mission requirements.
- In higher-voltage systems, proper spacing must be maintained to prevent arcing. If arcing is a possibility, one must look at surrounding materials that could feed a sustained arc. It should not be assumed that arcs would be self-extinguishing without careful test and analysis. All arcs must be assumed to be potentially catastrophic.
- Because of the possibility of catastrophic arcing, all exposed conductors carrying greater than 15 volts must be fully insulated:
    - powder-coated bus bars
    - thixotropic conformal coat of adequate thickness to inhibit puncture by dendritic growths
    - other types of encapsulation, taking care to avoid bubbles or voids
- Be absolutely sure that pure tin is not used that would give rise to tin whiskers. It is a good idea to apply encapsulation of sufficient thickness to withstand tin whiskers over all plated parts, as there can be escapements in the plating process that may inadvertently allow pure tin to be used.

V. <u>Power Quality</u>

MIL-STD-1541A, Section 5.2.10, defined power quality requirements for space vehicles. For voltage ripple, the requirement was 500mV peak-to-peak. This is only 1.78% of 28VDC, the implicit bus voltage assumed in 1541A. This level is very low and does not allow for typical bus voltage swings due to inter-mode transitions in the EPS or to reasonable overshoot and undershoot behavior due to normal periodic load or mode switching. Therefore, the requirement is being relaxed in this TOR (Section 6.2.1) to 7% of the DC bus voltage. However, it is important for designers to understand what the new requirement means and how to (and how not to) design to it.

The time domain ripple specification does not contain any information regarding the frequency content of the ripple. The ripple is composed of three parts: 1) self-generated ripple from the EPS, 2) reflected ripple from the loads, and 3) inter-mode or "deadband" ripple. The first component might be due to switching ripple from a bus-regulating DC-DC converter, or perhaps a solar array switching unit dithering between two switch states. The second component is due to the conducted emissions from the loads reflecting against the total bus impedance.

The third component is the largest portion of the ripple, and it is due to low-frequency phenomena. It is due either to the EPS toggling between regulation modes with different voltage setpoints (passing through a regulation deadband), or to overshoot and undershoot transients due to a toggling step load. The other components of the ripple, by contrast, are broadband or high-frequency in nature, and they must not produce high ripple voltages at frequencies that would result in excess power dissipation in capacitors directly connected to the bus. At low frequencies, capacitor currents produced by the ripple voltage are low due to high capacitive reactance, but at higher frequencies the very low capacitor ESR (equivalent series resistance) would be the main limiter of ripple currents. Power dissipations in individual capacitors in excess of 50 to 100mW could cause overheating and possible early failure, not to mention reducing overall power conversion efficiency.

It is tempting to levy an additional requirement on bus ripple voltage to limit its amplitude at higher frequencies, but it is felt that this unnecessarily constrains designers as it would be difficult to specify a "one size fits all" frequency cutoff point or even a consistent way of measuring the frequency content of the ripple. The designer is already required to provide a worst-case analysis; this analysis must show that the bus capacitors are not overstressed. Typical design practices do not result in excessive capacitor dissipations, since load units are required to meet conducted emissions requirements that, on the whole, produce little voltage ripple at higher frequencies (provided that the bus capacitance consists of enough parallel capacitors to ensure a very low effective ESR).

Raising the allowed ripple level and making it a function of bus voltage has implications for EMC requirements. The Conducted Susceptibility (CS) requirements must be raised to ensure that load equipment can withstand the higher levels of ripple. Since loads are generally constant-power, the current varies inversely with voltage, so we have lower load currents at higher bus voltage. Because load currents are lower, this means that conducted emissions are lower as well. This means that the Conducted Emissions (CE) requirements can be changed to accommodate variable bus voltages. TOR-2005(8583)-1, "Electromagnetic Compatibility Requirements for Space Equipment and Systems," which is essentially a revision of MIL-STD-1541A, contains new CS and CE requirements that meet the needs of higher-voltage buses with ripple as specified herein.

The new CS requirement for equipment operating at different bus voltage levels is as follows:

- 10% of the nominal bus voltage, rms, from 30Hz to 5kHz, sloping down from there to the MIL-STD-461E limit at high frequencies.

This is equivalent to 28% peak-to-peak at low frequencies, which is 12dB above the time-domain ripple requirement of 7% specified in this TOR.

The new CE requirement is as follows:

- the limits from 30Hz to 1kHz shall be 10% of the unit's nominal operational current, but no more than 130dBuA and no less than 80dBuA, and from the value at 1kHz, the limit shall fall at 20dB/decade to 20kHz, and from the value at 20kHz, the limit shall be a straight line to 20dBuA at 1MHz, and shall remain at 20dBuA to 400MHz

The new CE requirement not only adjusts for the lower current at higher bus voltage, but it makes the CE a function of the unit's operating power as well. Previous CE requirements had a single low-frequency specification of allowed ripple current that was independent of the power consumed by the unit. Thus, a 10W unit was allowed the same emissions level as a 1000W unit. Under the new requirement, low-power units are allowed less CE than are high-power units, with a maximum level of 3A (130 dBuA) for large loads and a lower maximum level of 10mA (80dBuA) for small loads.

VI. Peak-Power Extraction

The large number of mission failures and degraded missions (chiefly in the commercial space world) due to early solar array degradation mechanisms has rekindled interest in peak-power tracking (PPT) systems. A PPT system allows extraction of the full power available from a solar array at any given time. Conventional systems operate at a specific array voltage, and they cannot use the full array power except at end-of-life, when the peak-power voltage has degraded to match the bus voltage. Some feel that a requirement to use PPT would provide a kind of insurance against array degradation, extending the useful life of the spacecraft by allowing extraction of whatever array power is available under all normal and abnormal conditions. While it is true that a PPT does make this possible, there are many considerations that need to be weighed, as follows:

Pros

- full utilization of BOL power for spacecraft that can use it
    - telecom – can fly more transponders
    - agile s/c with fixed arrays
    - small s/c with fixed solar panels
    - manned missions

- provides max power from a degraded array
  - may save mission, although mission life will still be degraded
- may allow some spacecraft to fly with fixed arrays rather than rotating
- there are good PPT architectures available – not a cutting-edge technique
- advantageous for transfer orbit – allows full utilization of power from folded solar arrays

Rebuttals to Pros

- most government spacecraft cannot use the extra power
- improvements in solar arrays due to new standards will obviate many problems. If arrays don't fail, then PPT is not needed.
- there is a lot that can go wrong with the highly nonlinear control schemes needed for a PPT system
- an easier way to buy down the risk of early array degradation is simply to add more solar cells

Cons

- requires strong expertise and high cost to implement
- lower efficiency compared with standard techniques
- increases EMI
- increased complexity and hence lower reliability of PMAD equipment
- very high voltages on arrays if bus voltage is > 100V

Rebuttals to Cons

- companies would have to hire the right people, but once developed, the PPT could be reused from then on. If the government wants the insurance against early S/A power degradation, they will pay for the development
- since most systems won't be in PPT mode much of the time, the loss of efficiency is diluted. Overall, to achieve the same EOL power, an array would probably only need to be a couple percent larger, but the benefits in terms of access to full BOL power outweigh the cost
- although PPT would lower the reliability of the PMAD itself, use of PPT to offset the risk of premature S/A degradation would improve overall reliability by extending the usable life of a degraded spacecraft
- Any system with a converter between the array and the bus has EMI issues already. PPT would be no worse
- series-connected power stages could be used to allow lower voltage arrays while still giving a high bus voltage. Also, the low Voc temperature coefficient of modern multi-junction solar cells means that the ratio of cold to hot Voc is not as great as it is for silicon arrays

Because there are these advantages and disadvantages to PPT, and because a standard should reflect best practices currently in use (theory follows practice and not the other way around), it is felt that it would not be appropriate at this point to require use of a PPT system. However, developers as well as the procurement authority are strongly encouraged to consider use of PPT for future designs and conduct trade studies to show the costs and benefits for their application. The requirement for a PPT system may be specified for a given spacecraft acquisition effort without making it part of the standard. Finally, future industry standardization efforts based on the use of this TOR should include serious consideration of making some kind of peak-power extraction scheme a requirement.

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