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Rationale for the Power Standard

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ABSTRACT

The **Power standard** defines the requirements on electrical power systems (EPS) to be applied by contractors and subcontractors to ESA spacecraft and associated equipment or payloads.

The **Power Standard, ESA PSS-02-10,** consists of two volumes, which support two guideline specifications.

This document, **Volume 2**, a reference document is called the power standard rationale **PR**; it serves as a guide for volume 1 by explaining the **rationale** of the requirements specified in volume 1.

Volume 1, the standard itself, is called power standard **PS**.

ESA PSS-02-101 gives the guidelines for the establishment of a power-supply subsystem specification, while **ESA PSS-02-102** gives the guidelines for the establishment of an electrical unit specification.

DOCUMENT CHANGE RECORD

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TABLE OF CONTENTS

	PAGE
1.INTRODUCTION	1
2. APPLICABLE AND REFERENCE DOCUMENTS	3
2.1 APPLICABLE DOCUMENTS	3
2.2 REFERENCE DOCUMENTS	7
3.MISSION DESCRIPTION AND REQUIREMENTS	9
3.1 MISSION	9
3.2 MISSION POWER REQUIREMENTS	12
4. ELECTRICAL REQUIREMENTS	13
4.1 GENERAL REQUIREMENTS	13
4.2 SOURCE REQUIREMENTS	21
SOLAR ARRAY	28
BATTERY	30
CONVERTER OR REGULATOR	43
5. DISTRIBUTION AND PROTECTION REQUIREMENTS	47
5.1 GENERAL REQUIREMENTS	47
5.2 RELAYS AND FUSES	48
5.3 SSPCs	50
6. REQUIREMENTS ON EQUIPMENT AND PAYLOADS	51
6.1 GENERAL REQUIREMENTS	51
6.2 CONVERTER TYPE LOADS	54
7. THERMAL AND MECHANICAL REQUIREMENTS	55
8. RELIABILITY REQUIREMENTS	57
9. SAFETY REQUIREMENTS	59
10. PARTS, MATERIALS AND PROCESSES	61
11. MAINTAINABILITY AND AVAILABILITY REQUIREMENTS	63

12. OPERABILITY REQUIREMENTS 65

13. QUALITY ASSURANCE REQUIREMENTS 67

14. QUALIFICATION AND TESTING REQUIREMENTS 69

15. TERMS, DEFINITIONS, ABBREVIATIONS, SYMBOLS AND UNITS ... 71

KEYWORD/REQUIREMENT CROSS REFERENCE 73

1.INTRODUCTION

This **Power standard rationale (PR), ESA PSS-02-10 VOL 2,** serves as a **guide** to the **Power standard (PS), ESA PSS-02-10 VOL 1**.

It explains the rationale of the parameters specified in the Power Standard ESA PSS-02-10 VOL 1. It is also a reference to some key sources of information needed, when analysing and designing power systems, equipment and payloads.

After Chapter 3 it has the same chapter, section and subsection numbering as the Power Standard ESA PSS-02-10 VOL 1.

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2. APPLICABLE AND REFERENCE DOCUMENTS

2.1 APPLICABLE DOCUMENTS

The latest issue of the following documents is applicable to the extent specified in Sections 1.2 and 1.3 (documents marked with a "*" are in preparation or still TBD):

ESA PSS-01-0	Basic requirements for product assurance of ESA spacecraft and associated equipment.
ESA PSS-01-10	Product assurance management and audit systems for ESA spacecraft and associated equipment.
ESA PSS-01-11	Configuration management and control for ESA space systems.
ESA PSS-01-20	Quality assurance requirements for ESA space systems.
ESA PSS-01-201	Contamination and cleanliness control.
ESA PSS-01-202	Preservation, storage, handling and transportation of ESA spacecraft hardware.
ESA PSS-01-203	Quality assurance of test houses for ESA spacecraft and associated equipment.
ESA PSS-01-30	Reliability assurance of ESA space systems.
ESA PSS-01-301	Derating requirements and application rules for electronic components.
ESA PSS-01-302*	Failure rates for ESA space systems.
ESA PSS-01-303*	Requirements for failure mode, effects and criticality analysis and associated activities on ESA space systems.
ESA PSS-01-40	System safety requirements for ESA space systems.

ESA PSS-01-401	ESA fracture control requirements.
ESA PSS-01-402*	Design safety requirements for ESA space systems.
ESA PSS-01-403*	Hazard analysis requirements and methods for ESA space systems.
ESA PSS-01-404*	Risk assessment requirements and methods for ESA space systems.
ESA PSS-01-50	Maintainability requirements for ESA space systems.
ESA PSS-01-60	Component selection, procurement and control for ESA space systems.
ESA PSS-01-603	ESA preferred parts list.
ESA PSS-01-604	Generic specification for silicon solar cells.
ESA PSS-01-608	Generic specification for hybrid microcircuits.
ESA PSS-01-609*	Radiation design handbook.
ESA PSS-01-610	Design guidelines for capability approval of film hybrid microcircuits and microwave hybrid integrated circuits (MHIC's).
ESA PSS-01-70	Material and process selection and quality control for ESA space systems and associated equipment.
ESA PSS-01-701	Data for selection of space materials.
ESA PSS-01-702	A thermal vacuum test for the screening of space materials.
ESA PSS-01-704	A thermal cycling test for the screening of space materials and processes.
ESA PSS-01-707	The evaluation and approval of automatic machine wave soldering for ESA spacecraft hardware.

ESA PSS-01-708	The manual soldering of high reliability electrical connections.
ESA PSS-01-710	The qualification and procurement of two-sided printed circuit boards (fused tin-lead or gold-plated finish).
ESA PSS-01-721	Flammability testing for the screening of space materials.
ESA PSS-01-722	The control of limited-life materials.
ESA PSS-01-726	The crimping of high-reliability electrical connections.
ESA PSS-01-728	The repair and modification of printed - circuit boards and solder joints for space use.
ESA PSS-01-738	High-reliability soldering for surface - mount and mixed - technology printed circuit - boards.
ESA PSS-01-802	Environment requirements specification for space equipment: unit-level environmental test specification.
ESA PSS-02-101*	Guidelines for the establishment of a power supply subsystem specification.
ESA PSS-02-102*	Guidelines for the establishment of an electrical unit specification.
ESA PSS-02-11*	Test methods for the ESA power standard.
ESA PSS-02-12*	Power and signal cabling standard.
ESA PSS-02-13*	Ni-H_2 cell, battery standard.
ESA PSS-02-14*	Ni-Cd cell, battery standard.
ESA PSS-02-20*	ESA standard on EMC requirements for space systems.
ESA PSS-02-303*	Requirements for HV transformers and components used in electronic power conditioners.

ESA PSS-03-10*	Thermal control design and testing - standard for general requirements.
ESA PSS-03-108	Spacecraft thermal-control design data handbook.
ESA PSS-04-106	Packet telemetry standard.
ESA PSS-04-107	Packet telecommand standard.
ESA PSS-08-10*	Natural environment.
MIL-HDBK 217	Failure rates.

2.2 REFERENCE DOCUMENTS

The following documents are mentioned as a reference only. It is, however, strongly advisable to take their recommendations into account.

1. Satellite power system topologies (ESA Journal 1989, vol. 13, pp 77-88).

2. Pulse-width-modulation (PWM) conductance control (ESA Journal 1989, vol. 13, pp 33-46).

3. Simple pole/zero modelling of conductance controllers (ESA SP-294, pp 415-423).

4. PWM converter topologies (ESA SP-294, pp 297-305).

5. The sequential switching shunt regulator S^3R (ESA SP-126, pp 123-136).

6. Temperature derivative battery charge control (ESA SP-230, pp 107-122).

7. Nickel Hydrogen cell specification for GEO spacecraft (ESA-NH-GEO).

8. Nickel Hydrogen cell performance, design and qualification test requirements for Columbus (Issue 8).

9. Solar Array Design Handbook
   Rauschenbach, Hans S
   Van Nostrand Reinhold Company, 1980

10. Solar Cell Radiation Handbook
    Third edition
    JPL Publication 82-69, November 1982

11. Procedure for the measurement of spacecraft solar arrays electrical performance (ESTEC working paper nr 1700, January 1993)
    ESTEC XP division

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3.MISSION DESCRIPTION AND REQUIREMENTS

This chapter shall list the mission objectives to such an extent that the power system can be defined.

3.1 MISSION

3.1.1 Objectives

The mission dictates the orbit and launch window and puts one set of requirements on the power system, e.g. GEO or LEO orbit. The payload power requirement is, however, the real design driver; platform power is important too, but secondary.

The payload is the reason for the spacecraft mission and the power system is ideally tailored to its needs. Financial and schedule constraints do, however, often influence the power concept, and the concept thus evolves around already developed hardware units. Hence the desire to standardise bus voltages.

The spacecraft launch and transition into orbit put other constraints on the power system. Battery energy has to be sufficient to carry the essential payload and platform functions through the launch and injection and transfer orbit until the solar array can be partly or fully operational.

Key power system drivers for ESA spacecraft have been low electromagnetic emissions (GEOS, ISEE-B, Giotto) and highly dynamic loads in telecommunications and earth resources payloads. In each case the payload characteristic has forced a particular type of power system.

Mission duration is another parameter that can influence the power system concept, since it may affect the level of redundancy employed and the design margins allowed in order to cope with degradation caused by radiation and ageing, not to mention natural faults.

The spacecraft platform imposes its own requirements on the power system. There are a number of essential functions: telecommand and telemetry system and AOCS system functions, which must have ensured power regardless of the state of the payload. There is also a question of pyro functions which are mission critical, but typically early in the mission profile.

3.1.2 Launch date and launch conditions

The launch date and conditions will help in understanding the need for battery energy during launch and will help in defining solar array input power in that particular season and transfer orbit.

3.1.3 Mission profile and duration

The mission profile and duration are a chronological listing of spacecraft and payload operations as they affect the concept, sizing and operation of the power system.

Key parameters are:

Launch and early orbit operation:

- Time on autonomous power until solar array power takes over.
- Essential S/C power.
- Solar array configuration and spin rate during injection and transfer orbit.
- Solar aspect angle.
- Sunlight/eclipse period.
- Ground station coverage and operator's means of reacting to S/C events during launch and transfer orbits.
- Time in transfer orbit.

- Pyrotechnic operations.
- Injection into mission orbit.

Mission operations:

- Mission operators' reaction time to S/C events. Need for autonomous control.
- Mission orbit.
- Solar aspect angle.
- Solar intensity, variation.
- Sunlight/eclipse periods.
- S/C emergency modes.

3.1.4 Environment in all phases and duration

Key parameters for the solar array design:

- Solar intensity and variation.
- Solar aspect angle.
- Sunlight and eclipse duration.
- Radiation (accumulated dose).
- Particle density.
- Temperature.

Parameters for the battery design:

- Sunlight/eclipse duration.

- Battery temperature.
- Life duration.

Parameters for the electronics design:

- Radiation dose (accumulated).
- Temperature.

3.2 MISSION POWER REQUIREMENTS

The power requirement is the real design driver. Power is needed by the payload and the platform functions. The things to look out for are the highly dynamic (modulated) loads or loads (experiments) with stringent EMC demands. A load is rarely a load with a constant power demand.

3.2.1 Power profile and duration

The designer needs to have the load profiles summed up for each mission phase in order to be able to derive the peak power and peak energy demand on the power system.

Profiles are needed from the time the S/C goes on power prior to launch.

3.2.2 Load requirements in all phases

Average power levels, peak power levels and load modulation are needed in order to select the power system concept.

3.2.3 Power budget (average and peak)

The power budget should be in a numerical format from which the solar array EOL power and battery DOD are compared against.

4. ELECTRICAL REQUIREMENTS

In this chapter all system requirements, specifying a parameter are applicable under worst-case conditions.

4.1 GENERAL REQUIREMENTS

4.1.1 Power bus type

On board a spacecraft,power shall be generated and distributed via one or more DC voltage-regulated buses in order to avoid redeveloping already known concepts and in order to use existing design experience and hardware. As the concept of a fully regulated bus utilises a modular structure of power regulating elements and power is distributed and protected at the regulation source, no decentralised function should cause a single point-failure to occur. The only area susceptible to single-point failure is the central node of the power control-unit, where double isolation and utilisation of components which are guaranteed to be single point failure free, ensure the security of the power bus. In other words, the modularity of the source and distribution elements effectively turns the single power bus into a multiple power bus concept.

4.1.2 Bus voltages

The recommended nominal bus voltage shall be 28 V, 50 V or 120 VDC. These three regulated voltages have been selected to allow a coverage of various power levels and applications.

For the 28 V and 50 V, there considerable experience in their use has been gained and satisfactory hardware is available. The 120 V is the harmonised NASA/ESA voltage to be use for the manned space station.

The bus voltage selection is based on a 20 mΩ minimum bus output impedance, a 50% load modulation and an allowance of a factor 2 between LEO and GEO.

For a LEO mission, the same solar array power gives roughly half the power that it would give in a GEO mission. In both cases the power handled by the power system is roughly the same.

LEO case:

A 50% load modulation (i.e. 0.5 P/U) acting on a 20 mΩ bus impedance should not produce a bus disturbance greater than 1% of the nominal bus voltage (i.e. 0.01 U).

Hence

$$0.5\ P/U \times 0.02 < 0.01\ U$$

resulting in

$$U > \sqrt{P}\ (V) \text{ for LEO.}$$

GEO case:

For a given output power P, the required solar array power for GEO is about half the power required for LEO. This explains why

$$U > \sqrt{0.5\ P} \text{ in } (V) \text{ for GEO}$$

EXAMPLE:

BUS VOLTAGE (V)	P_{MAX} (W) IN LEO	P_{MAX} (W) IN GEO
28	784	1568
50	2500	5000
120	14400	28800

Table 4.1

4.1.3 Location of the common ground point (CGP)

The return of a voltage regulated bus will be grounded to the spacecraft structure at the negative side of the main bus capacitor (CGP).

This is a convention which applies to most S/C except where for plasma reference reasons a negative ground is preferred (e.g. US Space Station Freedom).

4.1.4 Bonding to structure

The bonding requirements are such that each piece of equipment can be considered as a Faraday cage where its case and connector shells are part of the structure.

4.1.5 Power/energy budget

During operational life, power bus shall be able to satisfy each average power/energy and maximum peak power demand with a margin of not less than 10%, in order to avoid overload because of the inherent uncertainty in predicting the actual power consumption. Experience shows that during development, the margins, have a tendency to narrow, and that the less mature a design, the wider margins may have to be used.

4.1.6 Power budget parameters

A power budget analysis will increase confidence in the power budget by taking the following EOL parameters into account:

Maximum battery cell voltage (charge)
Average battery cell voltage (charge)
Average battery cell voltage (discharge)
Minimum battery cell voltage (discharge)
Battery K-factor
Percentage power distribution loss
Solar array loss factors
Percentage solar array harness loss
Percentage power conditioning efficiency
Eclipse duration GEO/LEO
Sunlight duration GEO/LEO

Solar array power
Battery DOD
Battery temperature
Number of battery cell failures allowed

Guidelines to the most appropriate values for these parameters are given in Table 4.2 in order to permit rational comparison between proposals, to assist quick-look system-level computations and to highlight key system characteristics.

GEO (70% DOD)	Ni-Cd	Ni-H_2
Maximum battery cell voltage (charge)	-	-
Average battery cell voltage (charge)	1.39	1.44
Average battery cell voltage (discharge)	1.15	1.20
Minimum battery cell voltage (discharge)	-	-
Battery recharge ratio (k-factor)	1.05	1.10

LEO (20% DOD)	Ni-Cd
Maximum battery cell voltage (charge)	-
Average battery cell voltage (charge)	1.43
Average battery cell voltage (discharge)	1.15
Minimum battery cell voltage (discharge)	-
Battery recharge ratio (k-factor)	1.06

LEO (40% DOD)	Ni-H_2
Maximum battery cell voltage (charge)	-
Average battery cell voltage (charge)	1.48
Average battery cell voltage (discharge)	1.10
Minimum battery cell voltage (discharge)	-
Battery recharge ratio (k-factor)	1.06

Table 4.2

SYSTEM PARAMETERS	VALUE
Power distribution loss (%)	2
Solar array harness loss (%)	2
Power conditioning efficiency (%)	90
Eclipse duration for LEO (hrs)	0.6
Sunlight duration for LEO (hrs)	0.95
Eclipse duration for GEO (hrs)	1.2
Sunlight duration for GEO (hrs)	22.8

Table 4.2 (continued)

4.1.7 Safety assurance

As the concept of a fully regulated bus utilises a modular structure of power regulating elements and since power is distributed and protected at the regulation source, no decentralised function should cause a single- point failure to occur. The only area susceptible to single-point failure is the central node of the power-control unit, where double isolation and utilisation of components which are guaranteed to be single point failure free, ensure the security of the power bus. In other words, the modularity of the source and distribution elements effectively turns the single power bus into a multiple power bus concept. No single-point failure of the electrical system or equipment shall cause the loss of the spacecraft. For manned missions a minimum of two-failure tolerance is required. Any deviation from this requirement must be submitted to the Agency with supporting justification for approval. This requirement is dealt with in the following way:

On the conceptual level users are divided into essential users and non-essential users. The first category is TC/TLM function and, if applicable, AOCS functions, which if lost will prevent access to the S/C. Everything else can, depending on the mission, be considered non-essential.

Essential users can, by definition, not be disconnected from the bus and must therefore be so designed that they cannot short the bus.

Non-essential users can on the other hand be disconnected automatically if they overload the bus. The single-point-failure criteria need not be applied for this type of load.

Redundancy can be used to overcome the effect of single-point failures.

4.1.8 Redundant functions: general

Redundant functions shall be looked at in their full perspective. The use of redundancy shall take into account that any shared or common component device, cable or connector is likely to be a single-point-failure item. Redundant harness can be a redundant set of wires in the same bundle.

4.1.9 Redundant functions: physical location

Redundancy requires also that common causes do not affect two or more redundant branches.

4.1.10 Essential functions

Essential functions are those functions without which:

1. the satellite operator cannot recover the spacecraft, following any conceivable on-board or ground-based failure;

2. the spacecraft cannot be commanded;

3. the satellite permanently loses attitude and orbit control;

4. the satellite consumables (fuel, energy, etc.) are depleted to such an extend that more than 10% of the satellite lifetime is affected;

5. the safety of crew is threatened.

Essential functions (e.g. synchronisation or auxiliary power supply) are difficult to prove failure free if they rely on **centrally generated** auxiliary functions. The simpler way is to make them autonomous and not depend on external synchronisation and power.

4.1.11 Single point failure free

This requirement means that minimum mission objectives are defined. Usually it implies some kind of graceful degradation of the power system which in turn leads to a modular concept.

4.1.12 Temperature monitoring

Items of equipment that can dissipate more than 20 W in normal or failure mode are a potential fire hazard if they lose their coupling to a cold source. The temperature monitor is to be seen in this light.

4.1.13 Component derating

Established design practice, which should ensure a long operating life, is to be followed.

4.1.14 Telecommand/Telemetry: General

4.1.15 Telecommand: noise discrimination

The requirement is based on experience with undesired noise triggering equipment on board an earlier S/C.

4.1.16 Telecommand/Telemetry: voltage transient protection

Established design practice.

4.1.17 Telecommand and telemetry: unit protection

Established design practice.

4.1.18 Telecommand and telemetry: ESD protection

Established design practice.

4.2 SOURCE REQUIREMENTS

The expression domain is used to describe the operation of a servo loop in, say a shunt regulator which controls the main bus voltage level. Multiple domain control is used when the main bus voltage is controlled by, for instance a shunt regulator, a battery charge-regulator or a battery-discharge regulator.

Within the domains, the bus is controlled in a proportional manner. On the boundary between two domains, the main bus is controlled in a non-linear way since it will switch between two linear modes with a gap.

4.2.1 Bus voltage regulation

The bus voltage (at the regulation point), with steady-state tolerances within the ±1% range (with respect to the nominal voltage), can be achieved if one applies integral control and chooses stable components. A reference Zener diode having a temperature coefficient of 10 ppm/K and a voltage divider made from 25 ppm/K metal film resistors will be sufficient to meet the requirement of temperature, load, and life.

4.2.2 Location of the regulation point

The regulation point is chosen to be within the power system at the main bus voltage regulator output capacitor (e.g. in the PCU/PCDU).

4.2.3 Bus impedance at the regulation point: Operation in one domain

RATIONALE:

0.1 to 10 kHz range: 1% AC bus voltage (i.e. 0.01 U) due to 50% current modulation (i.e. I = 0.5 P/U) leading to:

$$Z_{max} = (0.01\ U) / (0.5\ P/U) = 0.02\ U^2/P$$

0.01 kHz to 0.1 kHz: Effect of integral feedback.

10 kHz to 100 kHz: Experience and the effect of the output capacitor.

< 0.01 kHz and > 100 kHz:

0.002 U^2 /P due to residual resistance.

Hence:

C_{bus} = P/400 πU^2 (minimum required value)

EXAMPLE:

For U = 50 V and P = 1 KW

Z_{max} = 50 mΩ
C_{bus} = 318 μF (minimum value)

4.2.4 Domain bus transient

The calculations based on single domain operation cannot be applied to the bus as a whole. Bus transients at the regulation point due to multiple domain operation are mainly caused by the non-linear control between domains.

During operation between domains, the bus error signal has to pass the guard band between domains, causing an uncontrolled bus at that time. The means of control in this mode is that of a bang-bang servo system which switches between two limits.

4.2.5 Bus stability

The bus stability can be verified by one of the following methods:
The maximum output impedance shall be lower than the impedance of the regulator output capacitor at all frequencies.

The regulator current response (i.e. the current before the output capacitor) to a step load shall not present any overshoot. The step load applied shall be sufficient to permit discrimination between the response and the bus ripple and spikes.

The voltage response to a current step shall have a wave shape where the ratio of

$$\frac{\text{initial slope of leading edge}}{\text{maximum slope of trailing edge}}$$

is greater than 5.3.

RATIONALE:

The equivalent circuit diagram of a regulated bus operating within one domain is shown in Figure 4.1.

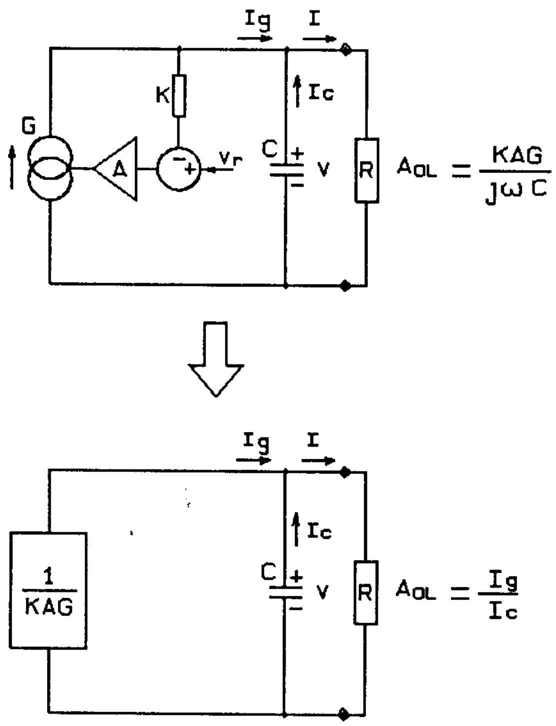

Figure 4.1

This is valid for any regulator topology (buck, boost, etc.) if the principle of conductance control is used. The open loop gain:

A_{ol} = (KAGR) / (1+jωRC)

For frequencies close to the cross-over frequency:

$A_{ol} = (KAG) / (j\omega C)$

Also from the circuit diagram:

$A_{ol} = \mathbf{i_g} / \mathbf{i_c}$

At the cross-over frequency ($\omega = \omega_C$),

$|\mathbf{i_g}| = |\mathbf{i_c}|$

Hence we can draw the vector diagram shown in Figure 4.2.

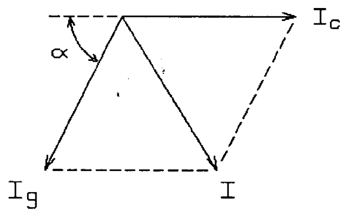

Figure 4.2

$$|\mathbf{i_g}| = |\mathbf{i_c}|$$
$\boldsymbol{\alpha}$ = phase margin
$$\mathbf{i} = \sqrt{2-2\cos\boldsymbol{\alpha}} \cdot \mathbf{i_g}$$

From this we calculate:

$$i = i_g\sqrt{2-2\cos\alpha} = i_g\ 2\sin\alpha/2$$

where

α = phase margin

Hence we can derive to the following stability criteria:

1.
$$V = \frac{i_c}{\omega_c \cdot C} = \frac{i_g}{\omega_c \cdot C} = \frac{i}{\sqrt{2-2\cos\alpha}} \cdot \frac{1}{\omega_c \cdot C}$$

Hence:

$$\frac{V}{i} = \frac{1}{\sqrt{2-2\cos\alpha}} \cdot \frac{1}{\omega_c \cdot C}$$

2.
$$\frac{i_g}{i} = \frac{1}{\sqrt{2-2\cos\alpha}}$$

3. The third stability criterion (time domain) was derived from computer models in the frequency and time domain. It resulted in the following formula:

$$\alpha = 36 \cdot Ln \frac{\text{initial slope of the leading edge}}{\text{maximum slope of the trailing edge}}$$

where

α = phase margin in degrees

EXAMPLE:

For a phase margin of 60°:

1.
$$Z_{max} = \frac{1}{\omega_c \cdot C}$$

2.
$$\frac{i_g}{i} = 1$$

3.
$$\frac{initial\ slope\ of\ leading\ edge}{maximum\ slope\ of\ trailing\ edge} = 5.3$$

COMMENT:

The stability in the linear mode is a necessary condition for bus stability. It is, however, advisable to verify bus stability using large steps in load power in order to see how the bus responds when driven into linear as well as non-linear modes.

4.2.6 Steady state bus ripple voltage at the regulation point

The peak-to-peak voltage ripple should be a fraction ±1% mainbus tolerance. A 0.5% pp ripple is believed to be feasible.

DEFINITION: Bus ripple voltage is defined as peak-to-peak amplitude of coherent signals at multiples of the switching frequency

4.2.7 Steady state bus commutation spike voltage (at the regulation point)

The bus spike voltage should be related to the ±1% bus tolerance. A 2% pp level is believed to be feasible.

DEFINITION: Bus spike voltage is defined as peak-to-peak amplitude of non-coherent signals (with respect to the switching frequency) with a frequency greater than the switching frequency.

SOLAR ARRAY

4.2.8 Solar array capability

On average there must be surplus power from the solar array, otherwise the bus will be uncontrollable.

4.2.9 Solar cell qualification

4.2.10 Solar array section: general

The solar array is made up of individual cells that are series connected into cell strings in order to meet the voltage requirements. The number of parallel strings per section is dictated by the required section current. Fault tolerance considerations lead to sectioning of the array. This means that the output of a section has a known short circuit current. In attaching a shunt dump per array section one gets a modular power element, and a convenient way of building up power by paralleling elements.

A series array regulator should be avoided for the following reasons:

1. There will be a start-up problem, because the series MOSFET cannot be switched on if no alternative power / energy source is available. If Olympus had used 100% series switching then recovery from a flat battery situation would not have been possible.

2. Owing to the MOSFET internal diode, the bus will be short- circuited in the case of a slip ring or array section/wiring short to ground (e.g. SADM failure on Olympus and solar array section short circuits on ECS and Marecs).

3. The voltage stress on the slip rings and the series MOSFETs is higher than the corresponding stress when shunt sections are used. For instance, lower Rds-on MOSFETs (e.g. IRF 150) can be used for the shunt sections. However, the series section requires something like an IRF 250.

4. An interrupted current snubber is required for the series MOSFET.

5. The series MOSFET requires a floating drive (more complex than the drive for a shunt dump section).

6. Future power increases could well result in higher bus voltages and shunt switching might then become the only option.

4.2.11 Solar array section: slip ring

Established practice of defining and limiting current over a contact.

4.2.12 Solar array section: parameters

The design of the solar array sections is a job for a specialist, but some of the performance parameters have an effect beyond the solar array itself.

Minimising the BOL/EOL current ratio reduces the need for dumping the excess power at BOL. This is usually a problem when a linear dissipative shunt is used.

The type of capacitance which is required is defined by the interface requirements on the power conditioning side: the control unit has to handle a solar array section of which it sees an equivalent capacitance (and inductance). This should be the specified equivalent capacitance, which is therefore defined by its test method.

The solar array section capacitance is not a directly controlled parameter. However, the depletion capacitance depends mainly on base resistivity (which is controlled within ±30%) and geometry (which is well controlled). The cell-to-substrate capacitance is also controlled by geometry control. The diffusion capacitance, which can be dominating at higher forward voltage, depends exponentially on the applied voltage.

BATTERY

4.2.13 Battery electrical design: cell matching

The cell capacity matching criterion comes from experience in GEO applications. Although it might be expected that a wider capacity range would be permissible for the lower depths of discharge generally used in LEO applications, no test data are available which would allow a value to be specified.

4.2.14 Battery electrical design: electrical isolation

Established isolation requirement.

4.2.15 Battery electrical design: bonding

Established bonding requirement.

4.2.16 Battery electrical design: monitoring connector

Established test requirement; is applicable to any piece of electronic equipment.

4.2.17 Battery electrical design: monitoring

The more cells in series per monitor point, the more difficult it is to diagnoses a bad cell among the good ones. However, monitoring of each individual cell increases the hardware complexity. Therefore, as a compromise it is recommended that 2-4 cells per monitor point be selected.

4.2.18 Cell by-pass protection (e.g. using series redundant diodes) for Ni-H_2 batteries

Experience with Ni-Cd battery cells allows open-circuit failures to be considered as non-credible. In the case of Ni-H_2 cells, open-circuit failures must be considered as possible. For example, leakage of the pressure vessel will eventually lead to an open-circuit failure.

4.2.19 Battery mechanical design: Ni-Cd cell pressure constraint

Established design practice.

4.2.20 Battery mechanical design: battery to cell mass ratio

The battery structure has three main functions:

- Mechanical (to protect of cells against mechanical damage during launch, and to support (Ni-Cd) cells against internal pressure);

- Thermal (to maintain cells within acceptable temperature range and temperature gradients within and between cells);

- Electrical (to provide interconnection of cells, isolation of cell cases from the battery structure, connectors and where applicable, cell bypass and battery heater circuitry).

A battery design should aim to meet all these requirements whilst keeping the mass of the battery as low as practical. The appropriate figure of merit, usually referred to as the "battery packaging factor", is given by the mass of the whole battery ("box level, i.e. including all internal battery connections and connectors) divided by the total mass of the cells alone.

The whole battery includes the base plate and all the components within the battery structure (cell interconnection harness, heaters, etc.) but excludes components outside the structure (radiator, external harness, thermal blanket, etc.).

For applications involving a large DOD, the thermal design begins to drive the battery mass. The battery packaging factors listed in Table 4.3 are considered to be reasonable targets for batteries with a total cell capacity in excess of 200 AH (cell capacity x number of cells).

BATTERY PACKAGING FACTORS		
APPLICATION	Ni-Cd	Ni-H_2
GEO	1.15	1.25
LEO: < 22% DOD	1.15	1.25
LEO: 22 - 32% DOD	1.19	1.35
LEO: 32 - 42% DOD	1.22	1.42

Table 4.3

4.2.21 Battery thermal design: general

It is the cell charge process which largely determines the operational temperature range of Ni-Cd and Ni-H_2 batteries. Apart from the useful charge reaction, there is a parasitic reaction in which water of the electrolyte is electrolysed, giving rise to oxygen evolution at the Ni-electrode. This oxygen then recombines at the negative electrode at a rate which accelerates rapidly with rise in temperature.

4.2.21.1 Lower operational temperature limit

At low temperatures slow oxygen recombination can lead to a significant build-up of oxygen pressure towards the end of charge. Oxygen recombination is much easier in Ni-H_2 cells than Ni-Cd. In addition, in the case of Ni-Cd, if the cell voltage is allowed to rise above 1.55 V, hydrogen will be evolved from the Cd-electrode. This can only recombine very slowly and can lead to a serious pressure increase within the cells.

For the above reasons, the lower operational temperature limit is dependent on cell type and charge rate. Maximum recommended charge rates (as C-values) when **the cell is close to full charge** are given in Table 4.4.

LOWER OPERATIONAL TEMPERATURE (°C)	MAXIMUM CHARGE RATE (C-RATE) NI-CD	MAXIMUM CHARGE RATE (N-RATE) $NI\text{-}H_2$
0	0.1	0.4
10	0.25	0.7
20	0.5	1.3

Table 4.4

Higher charge C-rates than these can be used when the cell is far from fully charged. Consequently the lower temperature limit, estimated from the above for an application with a particular DOD, depends also upon the charge current profile.

Cells can be discharged at lower temperatures, at least down to -10°C. However, the thermal inertia of a typical battery together with the fact that most heat is generated during discharge means that this lower limit during discharge cannot usually be exploited during LEO cycling.

4.2.21.2 Upper operational temperature limit

The upper temperature limit of operation is determined by the associated reduction in useful capacity and lifetime. The useful capacity reduces because of the lower charge efficiency associated with increased oxygen generation at the Ni-electrode. Attempts to compensate by increasing charge current or time only serve to increase the temperature further. The lifetime at higher temperatures also decreases rapidly above 20°C. The various degradation mechanisms all tend to be accelerated by temperature increase and in some cases (e.g. oxidation of separator, Cd-migration), by the higher oxygen levels in the electrolyte. Maximum recommended operational temperatures, measured at the hottest point on the external surface of the cell, are those shown in Table 4.5.

Battery maximum operational temperatures		
Application	Ni-Cd	Ni-H_2
LEO nominal	15°C	15°C
LEO excursions totalling < 24 hrs. during life	25°C	25°C
GEO (average during cycling)	10°C	10°C
GEO peak discharge at max. DOD	30°C	30°C
GEO solstice trickle charge	5°C	5°C

Table 4.5

4.2.22 Battery thermal design: thermal gradient

The temperature gradient limit is necessary because battery parameters (charge efficiency, charge voltage, etc.) depend strongly on temperature and it is necessary to ensure that cells within the same battery maintain a well-matched state of charge. The same applies between different batteries if a spacecraft contains a group of batteries which are **not** charged independently, as is the case if they are wired electrically in parallel.

4.2.23 Battery DC magnetic field design

Ag-based batteries have a much lower magnetic moment than the Ni-based batteries.

4.2.24 Battery safety: Ni-H_2 battery

Established safety requirement.

4.2.25 Battery management: GEO charging technique

Overcharge is detrimental to battery life and leads to an increased rate of fall in the end of discharge voltage. Battery charge efficiency is limited at the lower level by decreasing charge efficiency and at the upper level by degradation effects of very high charge rates.

4.2.26 Battery management: LEO charging technique

The minimum recharge ratio (K_{min}) required to return the charge removed during the previous discharge is a function of temperature and depth of discharge. It is higher than unity because, towards the end of charge, the charge current is used less and less efficiently, an increasing proportion leading to the generation of oxygen on the positive electrode. This side reaction stores no useful energy but simply results in heat dissipation as the oxygen reacts at the negative electrode. High local temperatures and the simultaneous presence of oxygen lead to accelerated chemical breakdown of cell internal components. To limit these effects, the amount of overcharge and the charge current near the end of charge should be reduced as much as possible.

4.2.27 Battery management: reconditioning capability for GEO

For GEO missions of long duration it has been shown that reconditioning has had a very beneficial effect on the battery. The OTS battery to which reconditioning was applied experience virtually no degradation.

4.2.28 Battery management: reconditioning capability for LEO

In Section 4.2.30 below, a description is given of the memory effect and the circumstances in which reconditioning may be needed.

4.2.29 Number of parallel batteries

Parallel operation of batteries, within a given power bus, require active current sharing to ensure proportional discharge rates for each battery.

4.2.30 Battery DOD versus life: Ni-Cd application

Whilst battery cells must eventually fail irreversibly, experience to date shows that the most likely mode of failure for a spacecraft application is a drop in the end of discharge voltage to a value insufficient to maintain the power bus. The useful lifetime of a battery may therefore depend on the design of the power conditioning circuitry. End-of-charge voltage only increases slightly with ageing. It should be emphasised that the reasons for the drop in discharge voltage are complex and not yet sufficiently well understood to be reliably predicted.

4.2.30.1 LEO applications

What follows is the result of an analysis of limited data from ongoing life tests on batteries of VO24S and VO40S cells.

Typically, the end of discharge voltage (V_{eod}) during a life test with a constant DOD passes through three phases:

1. A fairly rapid fall from the beginning-of-life value lasting typically 1 to 3 years.

2. A much slower steady decline which is irreversible.

3. Final phase leading to cell failure. The V_{eod} becomes unstable, sometimes rising temporarily and eventually dropping dramatically.

All three phases are accelerated by higher temperatures of operation and greater DODs.

Phases 1 and 2 actually occur in parallel. The effects of phase 1 are reversible by reconditioning, after which the fall starts anew and with the same time profile. Consequently regular reconditioning allows a higher end-of-discharge voltage to be maintained until failure. It is not known (yet) whether regular reconditioning has any influence on the time of eventual failure.

When the DOD varies from orbit to orbit, but the maximum DOD occurs reasonably frequently (say every 50 orbits or less), the end-of-discharge voltage for the orbit with the greatest DOD is the same or slightly less than if that DOD were reached every orbit.

Table 4.6 indicates for Ni-Cd batteries end-of-discharge voltages expected for different temperatures and maximum DODs after 5 years continuous cycling with and without reconditioning and gives some indication of the expected life time. $V_{eod(corr)}$ is end-of-discharge voltage per cell corrected for cell internal resistance. In order to calculate the actual worst-case end-of-discharge voltage to be expected in a particular case, it is necessary to identify the worst case combination of DOD and end-of-discharge current for the mission discharge profile. Normally this will be the orbit with the highest DOD. The V_{eod} value is obtained from the $V_{eod(corr)}$ value with or without reconditioning as appropriate, minus the end-of-discharge current multiplied by the cell's internal resistance.

To estimate V_{eod} for orbits with lower DODs, the value of $V_{eod(corr)}$ with regular reconditioning should be used, correcting for internal resistance as above whether or not regular reconditioning is used (this is because of the reconditioning effect of the higher DOD orbits).

Values in brackets are extrapolated from test data and hence are less certain.

MAXIMUM % DOD	TEMP. RANGE (°C)	VEOD-CORRECTED AFTER 5 YR NO RECOND.	VEOD-CORRECTED AFTER 5 YR 6M RECOND.	LIFE TO FAILURE (YEARS)
10	-5 to +25	1.23	1.25	> 7
20	-5 to +25	1.16	1.21	> 7
30	-5 to +15	1.12	(1.20)	> 5
30	25	1.12	(1.15)	(5)
40	-5 to +15	1.12	(1.20)	> 4
40	25	1.08		3.2

Table 4.6

Here $V_{eod\text{-}corrected}$ = end of discharge voltage per cell corrected for the cell internal resistance.

If the DOD suddenly increases following a long period at lower DOD (say longer than three months) then the end-of-discharge voltage to be expected for this orbit will be significantly lower than the value one would have obtained if the battery had been cycled at this large DOD every orbit.

If there is a risk that this voltage will be insufficient to maintain the power bus then one of the following steps must be taken.

EITHER

A realistic battery life test will be performed that demonstrates that there will be a sufficient voltage margin. (It is not possible at present to predict the magnitude of the depression in the end of discharge voltage.)

OR

Reconditioning will be performed:

1. Once, shortly before the large DOD event, if the event is planned/foreseen.

2. Regularly (e.g. every six months) if the event is unpredictable (e.g. contingencies).

Since batteries will normally have been deep-cycled shortly before launch, deep discharges during the first three months of a mission are relatively safe.

The preferred temperature range of operation (measured at the battery base plate) for longest lifetime is -5°C to +15°C. Operation at 25°C reduces lifetime considerably and is not recommended for DODs above 20%. It should be noted that the above data are for batteries subjected to taper charging mostly with rather high recharge ratios (1.08 to 1.16). Lower recharge ratios are expected to lead to longer lifetime but this has yet to be experimentally demonstrated. Use of other charge techniques, especially those in which battery cells experience higher rates of charge when they are near full charge (e.g. charge cut-off), may have a negative impact on lifetime and alter cell discharge behaviour. No comparative test data are yet available.

4.2.30.2 Geosynchronous applications

Whilst similar arguments to those for LEO applications concerning end-of-discharge voltage behaviour are applicable to geosynchronous applications, the situation is much simpler because of the greatly reduced number of cycles required for the GEO mission and the practicality of reconditioning the batteries before each eclipse season.

In-orbit experience of up to 13 years has been achieved on OTS without failure on Ni-Cd-technology. This performance has been achieved for a maximum DOD value up to 60%. In addition, test data gathered from test indicate that lifetime of Ni-Cd battery for GEO mission is not influenced by the maximum DOD.

Within the temperature range and the management scheme detailed in Section 4.2.22, a nominal GEO mission with maximum DOD up to 80% (to 90%) of the **NAME PLATE CAPACITY** is acceptable.

The measured relation between battery life, $V_{eod(EOL)}$ and DOD (referred to name plate capacity) is shown in Table 4.7.

DOD (%)	NI-CD (YEARS)	NI-CD VEOD(EOL)
70%	13	1.13
80%	13	1.08
90%	10	1.03

Table 4.7

It is predicted however that lifetimes in excess of these figures (up to 15%) are possible. This shall be dealt with on a case by case basis.

4.2.31 Battery DOD versus life: Ni-H_2 application

4.2.31.1 LEO applications

LEO life test data are rather limited. From an analysis of available US and European data rough indications as to the maximum average DOD that can be used are listed in Table 4.8.

MISSION REQUIREMENT (YEARS)	MAXIMUM % DOD
1	70
2	50
3	40
5	35
> 5	no data yet available

Table 4.8

"Memory" effects similar to those observed for Ni-Cd batteries may apply also to Ni-H_2. It has not yet been clearly established whether this is in fact the case. Current evidence suggests that such effects will be less important.

Oxygen recombination is more rapid in Ni-H_2 cells than Ni-Cd, so Ni-H_2 is expected to be more tolerant to high charge rates. Nevertheless, it is expected that the lifetime will be enhanced if the amount of overcharge at high currents is minimised.

4.2.31.2 Geosynchronous applications

The measured relation between battery life, $V_{eod(EOL)}$ and DOD (referred to name plate capacity) is shown in Table 4.9.

DOD (%)	Ni-H_2 (YEARS)	Ni-H_2 VEOD(EOL)
70	13	1.10
80	10	1.05

Table 4.9

4.2.32 STORAGE AND DESTORAGE OF BATTERIES

Various irreversible chemical-corrosion-type reactions take place steadily in a cell whether it is in use or not. The rate of these reactions is accelerated with increases in temperature, so the cumulative degradation is minimised by storage at low temperature.

When a cell is not undergoing cycling, chemical changes (also accelerated by increase in temperature) occur in the electrodes, which result in reduced efficiency on charge and low voltage on discharge, when the cell is again brought into use. These effects can be reversed by performing conditioning cycling.

In order to limit the generation of high local temperatures and oxygen pressures on charge, the first conditioning charge after storage must be performed at relatively low rate (typically about C/20) and the following discharge must be slow and deep in order to produce a reconditioning effect on the electrodes. Several charge / discharge cycles may be needed before the specified capacity is obtained.

Detailed storage and conditioning needs depend on the particular design of a cell and the materials used within it. Equally important is that storage recommendations have been validated by previous experience and tests. Manufacturer's detailed recommendations must therefore be followed.

CONVERTER OR REGULATOR

The following requirements apply if the source for a regulated bus is a converter or regulator:

4.2.33 External synchronisation

Multiphase synchronisation can be applied to modular converters or regulators in order to reduce their combined ripple. One must, however, consider the effect on the main bus of losing the synchronisation. The difficulty with designing a single-point-failure-free synchronisation is often that it is easier to lower the ripple per module. Synchronisation is mostly used on S/C carrying sensitive electromagnetic experiment packages that can pick up the switching noise from the converters.

4.2.34 Free running frequency

The free running frequency shall be lower than the synchronisation frequency by at least 10% in order to permit direct synchronisation of the oscillator.

4.2.35 Essential function protection

Any function supporting essential functions in converters or regulators is a potential single-point failure. Care must therefore be taken to ensure that there is no common cause that can affect more than one essential function.

4.2.36 Conducted common-mode current

PWM converters/regulators have their active (e.g. MOSFET) and passive (e.g. diode) switching elements mounted, galvanically isolated, on a heat sink, which is usually in direct contact with the unit case. This unit case will be normally bonded to the S/C structure. The unit electrical reference is usually isolated from the unit case and only connected externally to the S/C central ground point (one particular point on the S/C structure).

Hence there will be parasitic capacitance between the switching elements and the unit case, and consequently, there will be current injected into the unit structure.
($I = C \cdot dV/dt$), which can only return to its origin via the unit ground connection (connection between unit electrical reference and the S/C central ground point). This will cause unit common mode noise, which can only be reduced by capacitive decoupling inside the unit (i.e. a capacitance between unit electrical reference and the unit case). See also Section 4.2.39.

It is believed that 5% of the maximum unit current (i.e. $0.05 \cdot P/U$) is an acceptable common mode current.

4.2.37 Conducted common-mode voltage

The peak-to-peak common mode voltage should not exceed the 1% bus voltage tolerance.

DEFINITION: The peak-to-peak common-mode voltage is defined as the peak-to-peak voltage amplitude between the unit electrical reference and the S/C central ground point.

4.2.38 Electrical isolation: DC (see rationale in Section 4.2.39)

The electrical zero-volt reference shall be isolated from unit case by more than 1 kΩ. However, the isolation resistance shall be more than 20 kΩ for satellites with stringent magnetic-field requirements.

4.2.39 Electrical isolation: AC

The specified values for the conducted common-mode current, DC isolation and AC isolation are based on the following compromise:

The impedance between the electrical reference and the unit case compared with the impedance of the ground wire[1] should be:

---

[1] The ground wire is a wire that connects the unit electrical reference to the system central ground reference or to a secondary distributed ground reference.

1. Very large for DC and low frequencies in order to minimise currents in ground loops, which minimise at the same time the generation of magnetic fields.

2. Very small for frequencies greater than, say, 5 MHz in order to minimise the voltage drop between references due to common mode currents.

Figure 4.3 shows the comparison between the two above-mentioned impedances for the specified values.

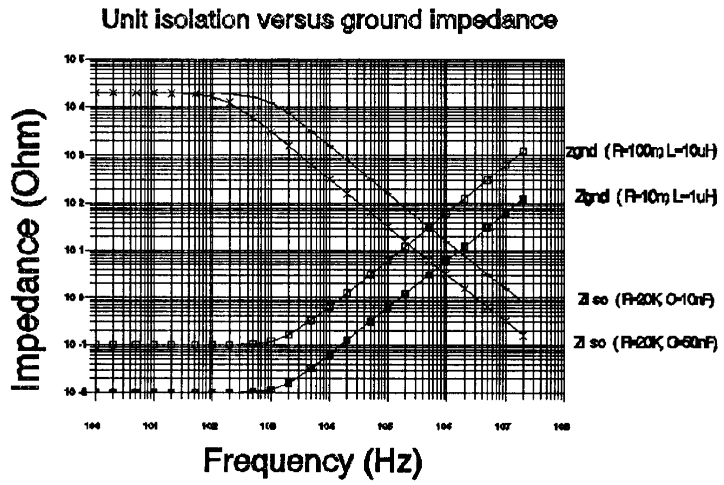

Figure 4.3

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5. DISTRIBUTION AND PROTECTION REQUIREMENTS

All requirements in this chapter specifying a parameter are applicable under worst-case conditions.

5.1 GENERAL REQUIREMENTS

5.1.1 Protection: general

The main bus at the regulation point is able to deliver power up to its maximum capacity. There is at this point no protection against short circuits. Hence protection is implemented in the distribution system which is under power system control or within user (load) control. The latter is not desirable since it is outside the power designer's control. All interfaces with a power bus shall be protected as close as possible to the regulation point, since any piece of harness or connector can be the cause of a short circuit.

5.1.2 Protection: non-protected sections

See Section 5.1.1.

5.1.3 Current limitation: bus capability

The current shall not increase above the total bus capability since it will overload the bus and bring down the voltage causing severe disturbances through the disconnection of non-essential loads and the possible triggering of S/C emergency modes.

5.1.4 Current limitation: general

5.1.5 Current limitation: safety

Established design practice.

5.1.6 Single-point-failure free

It is again the requirement that a common cause shall not generate a failure that can affect redundant functions, in this case buses.

5.1.7 Disconnection of non-essential loads

This is part of the strategy of ensuring S/C survival in the case of power bus overload.

5.1.8 Protection: harness over current

The effect of a failure must not give rise to other failures.

5.1.9 Protection: slip ring

Established design practice.

5.1.10 Harness resistance

The rationale is based on a harness voltage drop (line and return) of 0.5% of the nominal bus voltage.

EXAMPLE:

For U = 50 V and P = 1 kW: harness resistance = 12.5 mΩ.

5.1.11 Harness inductance up to 100 kHz

The rationale is based on a harness break frequency (see Section 4.3.39) of 5 kHz. This harness break frequency should be in line with the closed- loop bandwidth of the regulated bus (usually around 5 kHz = 100 kHz/20 where the switching frequency is 100 kHz).

EXAMPLE

For $R_{harness}$ = 12.5 mΩ: harness inductance = 400 nH.

Parallel twisted pairs are required for the harness in order to achieve this, because each twisted pair has an inductance of about 0.8 µH/m.

5.2 RELAYS AND FUSES

The use of fuse protection must be justified. This justification must be submitted to ESA for approval.

The following requirements apply to a relay / fuse type power distribution:

5.2.1 Bus recovery from a fuse clearance

During a fuse clearance, the bus will be short-circuited, causing maximum bus current delivery (i.e. maximum solar array and battery current). After the fuse clearance, the bus error signal has to travel through several of the bus domains and guard band separations before it arrives in its normal operating domain. Hence there is a tendency for the bus voltage to peak and it is judged that up to 4% peaking with a time constant less than 2 ms can be accepted.

Note that the impedance mask allows 1% bus voltage variation for 50% load modulation and 2% for 100% modulation in one domain.

5.2.2 Fuse clearance time

5.2.3 Current inrush

The rationale is based on a 2% bus voltage drop due to inrush (applied to the actual bus capacitor).

The inrush current is the initial transient current drawn by equipment or payload when the bus voltage is applied to it or when it switches on.

EXAMPLE:

For $U = 50$ V and $C_{bus} = 1000$ µF:

integrated inrush current = 1 mC

5.2.4 Peak relay switch current

Established design practice.

5.2.5 Peak relay switch voltage

5.2.6 Load switching induced transients

See rationale on Section 5.2.3.

5.2.7 DC distribution resistance excluding harness

The rationale is based on a voltage drop of 0.5% of the nominal bus voltage.

5.3 SSPCs

The following requirements apply to a solid-state power-controller (SSPC)-type power distribution:

5.3.1 SSPC DC series resistance

The rationale is based on test results on existing designs including a margin.

5.3.2 SSPC equivalent series inductance

The rationale is based on test results on existing designs including a margin.

5.3.3 SSPC inrush capability

The rationale is based on test results on existing designs including a margin.

5.3.4 Current slew rate

The rationale is based on test results on existing designs including a margin.

6. REQUIREMENTS ON EQUIPMENT AND PAYLOADS

All requirements in this chapter specifying a parameter are applicable under worst-case conditions.

6.1 GENERAL REQUIREMENTS

6.1.1 Secondary voltage requirement

The main power distributed in a spacecraft is derived from the regulated DC bus or DC buses. Any power required by a load at a voltage different from the regulated bus voltage has to be **generated locally** and distributed at equipment or payload level.

On earlier projects (e.g. GEOS) the power system delivered experimenter power at the requested voltages, but it turned out to be very difficult to manage since experimenter demands changed during the development phase.

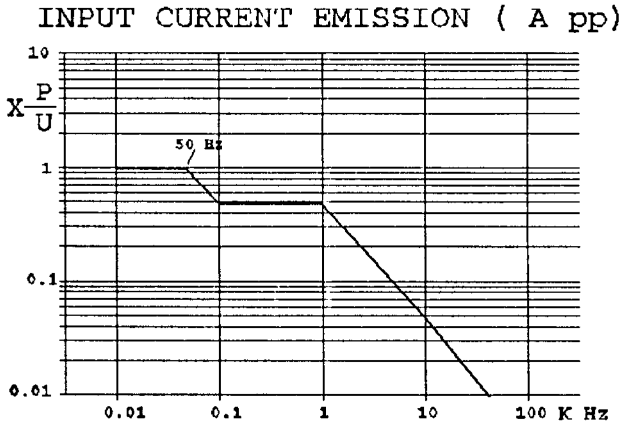
6.1.2 Failure propagation

The requirement means that the user is free to fail as long as he does not interfere with the rest of the S/C.

6.1.3 Power conditioning component current

The peak current requirement (excluding initial surge) of 15 A per power cell is based on available qualified power transistors and diodes and their derating. It also reflects design experience with respect to how large currents can be switched without too many problems in the circuit layout due to high dI/dt.

6.1.4 Conducted input current modulation



U = Nominal regulated voltage (volt)

P = Maximum load power (watt)

Figure 6.1

The rationale is based on an 1% AC bus voltage due to the application of conducted input current modulation on the sum of bus impedance (see Section 4.2.3) and the power distribution harness inductance.

6.1.5 Essential function protection

Any function supporting essential functions in converters or regulators is a potential single-point failure. Care must therefore be taken to ensure that there is no common cause that can affect more than one essential function.

6.1.6 Safe operation: General

Established design practice.

6.1.7 Safe operation during switch on and survival

Each equipment can be exposed to any level of voltage between zero and full bus voltage. Oscillations between these levels can also occur, at least during integration and test where many abnormal situations just happen.

6.1.8 Protection: test points

See also Section 6.1.7. Test points can also be an unintentional way of coupling interference into the system when used. The designer may consider this point and not place test point in sensitive parts of the equipment without using appropriate buffers.

6.1.9 Protection: stimulus points

See also Section 6.1.7.

6.1.10 Protection: TC/TM polarity inversion

Established design practice. Problems like these occur typically during integration and test.

6.1.11 Maximum inrush current

Any inrush current can cause instantaneous overload of the bus and cause regulator saturation. The result will be a large transient on the bus.The peak currents may damage contacts.

6.1.12 Integrated inrush current

See Section 5.2.3.

6.1.13 Hybrids: Requirements

6.1.14 Hybrids: Design

6.1.15 Hybrids: Design analysis

6.1.16 DC isolation

6.1.17 AC isolation

See Section 4.2.39.

6.2 CONVERTER TYPE LOADS

The following requirements apply to loads containing one or more regulated converters:

6.2.1 Converter stability

See Section 4.2.5.

6.2.2 Converter input ripple current

This ripple current is the differential mode current caused by converter commutation and reduced by input filters. The limit is based on a practical balance between converter filter mass, bus impedance and equipment susceptibility to bus noise. However, more stringent EMC requirements may lead to a reduction in the value indicated in this standard.

6.2.3 Conducted common-mode current

See Section 4.2.36.

6.2.4 Free running frequency

See Section 4.2.34.

7. THERMAL AND MECHANICAL REQUIREMENTS

The listed requirements reflects established design practices.

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8. RELIABILITY REQUIREMENTS

The listed requirements reflects the QA aspects of the design process.

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9. SAFETY REQUIREMENTS

The listed requirements are ESA safety requirements and reflect established design practices.

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10. PARTS, MATERIALS AND PROCESSES

The listed requirements are established ESA standards.

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11. MAINTAINABILITY AND AVAILABILITY REQUIREMENTS

The listed requirements are established ESA standards.

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12. OPERABILITY REQUIREMENTS

12.1 DIAGNOSTIC REQUIREMENTS

Telemetry monitoring points shall enable the operator to monitor the status and the allow diagnostic analysis of the power subsystem during all mission phases and during ground check-out. This will normally imply monitoring of the following:

solar array section current (one or more)
solar array temperature
shunt section current (one or more)
battery voltage
battery current
battery temperature
main bus voltage
main bus load current
status, disconnect non-essential loads
status, redundant BCR, BDR, shunts
temperature of equipment with > 20 W dissipation

12.2 TELEMETRY PRECISION

The precision required of any monitoring electronics is typically such that the electronics can be designed using standard components such as 25 ppm/K metal film resistors and where applicable, 10 ppm/K reference diodes.

12.3 TELEMETRY SAMPLING RATE

The telemetry sampling rate permits an assessment of the nominal or out- of-tolerance status of the monitored parameters. Various modes can be provided by the TTC system. Temperature, voltage and current monitor points are normally sampled approximately every 19-20 s. When pyro initiators are fired, the sampling rate can be reset to measure the form of a 1-5 ms current pulse.

In the resent past it has proved difficult to find the cause of problems in orbit, owing to the poor sampling capability on telemetry.

12.4 TELEMETRY CHANNELS

Established standard.

12.5 TELECOMMAND STANDARD

Established standard.

12.6 POWER SYSTEM TELECOMMAND CAPABILITY

Requires a definition of tolerable failures or degraded mode of operation. Should be listed under mission objectives in Chapter 3.

12.7 TELECOMMAND ACCESS TO PROTECTION CIRCUITS

Any automatic switch-off function must be monitored so that effect and cause can be traced. Inhibiting or enabling an automatic function gives the operator the option of continuing a mission with a failed protection circuit.

13. QUALITY ASSURANCE REQUIREMENTS

Established ESA standards.

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14. QUALIFICATION AND TESTING REQUIREMENTS

Established ESA standards.

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15. TERMS, DEFINITIONS, ABBREVIATIONS, SYMBOLS AND UNITS

AC	alternating current
ADC	DC current
Ag-Cd	silver-cadmium
Aol	open loop gain
App	ampere peak to peak
BCR	battery charge regulator
BDR	battery discharge regulator
BOL	begin of life
BW	band width
C_{bus}	main bus capacitor
CGP	common ground point
dB	decibel
DC	direct current
disch	discharge
DOD	depth of discharge
EGSE	electrical ground support equipment
EMC	electromagnetic compatibility
EMI	electromagnetic interference
EOL	end of life
EPS	electrical power system
ESA	european space agency
ESD	electrostatic discharge
FMECA	failure mode effects and criticality analysis
GEO	geostationary orbit
HDBK	handbook
Hrs	hours
IC	integrated circuit
I_{nom}	nominal current
LEO	low earth orbit
LEV	level
max	maximum
mC	milli coulomb
MGSE	mechanical ground support equipment
min	minimum
Ni-Cd	nickel cadmium
$Ni\text{-}H_2$	nickel hydrogen
PCB	printed circuit board
PCDU	power control and distribution unit

PCU	power control unit
PDU	power distribution unit
PF	power specification format
PR	power standard rationale
PS	power standard
PSS	procedure, standard and specification document
PWM	pulse width modulation
S/C	spacecraft
S^3R	sequential switching shunt regulator
SMT	surface mounting technique
SSPC	solid state power controller
TBD	to be defined
TC/TM	telecommand and telemetry
VDC	DC volt
Veod	end of discharge voltage
Yr(s)	year(s)

KEY WORD/REQUIREMENT CROSS REFERENCE

ac isolation . . . 44, 54
analysis . . . 3, 4, 15, 36, 40, 53, 65, 71
applicable documents . . . 3
battery . . . 5, 7, 9-12, 15-17, 21, 28, 30-36, 38-41, 49, 65, 71
battery dc magnetic field design . . . 34
battery dod versus life . . . 36, 40
battery electrical design . . . 30
battery management . . . 35
battery mechanical design . . . 31
battery safety . . . 34
battery thermal design . . . 32, 34
bonding . . . 15, 30
bus impedance . . . 14, 22, 52, 54
bus recovery from a fuse clearance . . . 49
bus ripple voltage . . . 27
bus spike voltage . . . 27
bus stability . . . 23, 27
bus voltage regulation . . . 21
bus voltages . . . 9, 13, 29
cell by-pass protection . . . 30
common ground point . . . 14, 71
component current . . . 51
component derating . . . 20
conducted common-mode current . . . 43, 44, 54
conducted common-mode voltage . . . 44
connector . . . 15, 19, 30, 47
connectors . . . 31
converter input ripple current . . . 54
converter or regulator . . . 43
converter stability . . . 54
converter type loads . . . 54
current inrush . . . 49
current limitation . . . 47
current slew rate . . . 50
dc isolation . . . 44, 54
diagnostic requirements . . . 65
distribution and protection . . . 47

electrical isolation 30, 44
emc requirements 5, 54
energy budget 15
eps 3, 71
equipment and payloads 1, 51
essential function protection 43, 52
essential functions 10, 19, 20, 43, 52
external synchronisation 20, 43
failure propagation 51
free running frequency 43, 54
fuse clearance time 49
harness 15, 18, 19, 31, 47, 48, 50, 52
harness inductance 48, 52
harness resistance 48
hybrids 53
input current modulation 52
integrated inrush current 49, 53
life tests 36
load switching induced transients 50
maintainability and availability 63
maximum inrush current 53
non-essential loads 47, 48, 65
operability requirements 65
peak relay switch current 49
peak relay switch voltage 49
power budget 12, 15
power bus type 13
printed circuit boards 5
protection 20, 21, 30, 43, 47, 48, 52, 53, 66
qualification and testing requirements 69
quality assurance requirements 3, 67
reconditioning 35-37, 39, 42
redundant functions 19, 47
reference documents 3, 7
regulation point 21, 22, 27, 47
relays and fuses 48
reliability requirements 57
safe operation 52, 53
safety assurance 18
safety requirements 3, 4, 59

secondary voltage requirement 51
single point failure free 13, 18, 20
slip ring 28, 29, 48
solar array 7, 9-16, 18, 28, 29, 49, 65
solar array capability 28
solar array section 28, 29, 65
solar cell qualification 28
soldering 4, 5
source requirements 21
sspc 50, 72
storage and destorage of batteries 42
telecommand 6, 10, 20, 21, 66, 72
telecommand access 66
telecommand capability 66
telecommand standard 6, 66
telemetry 6, 10, 20, 21, 65, 66, 72
telemetry channels 66
telemetry precision 65
telemetry sampling rate 65
temperature monitoring 20
terms, definitions, ... 71
test 3-5, 7, 29, 30, 36-40, 50, 53
thermal and mechanical requirements 55