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Robotic Lunar Exploration Program

Lunar Reconnaissance Orbiter

Electrical Power Subsystem Specification

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Goddard Space Flight Center
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CM FOREWORD

This document is a Lunar Reconnaissance Orbiter (LRO) Project Configuration Management (CM)-controlled document. Changes to this document require prior approval of the applicable Configuration Control Board (CCB) Chairperson or designee. Proposed changes shall be submitted to the LRO CM Office (CMO), along with supportive material justifying the proposed change. Changes to this document will be made by complete revision.

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LUNAR RECONNAISSANCE ORBITER PROJECT

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1.0 INTRODUCTION

1.1 SCOPE

This specification establishes the design, performance and acceptance requirements for the Lunar Reconnaissance Orbiter (LRO) Electrical Power Subsystem (EPS).

1.2 POWER SUBSYSTEM CONCEPT

The EPS configuration includes a single 2-axis tracking Solar Array (SA). SA power directly supplies the Observatory loads through an unregulated bus. The bus voltage shall be maintained between 21-35 Volts Direct Current (VDC) measured at the loads during all mission phases through use of the LRO Power Subsystem Electronics (PSE).

Energy storage is achieved through the use of one Lithium Ion battery.

2.0 DOCUMENTS

2.1 APPLICABLE DOCUMENTS

431-ICD-000018	LRO Power Subsystem Electronics Electrical Interface Control Document
431-PLAN-000033	LRO Battery Handling Plan
431-PROC-000021	LRO Power Subsystem Electronics Box Functional Test Procedure
431-PROC-000022	LRO Power Subsystem Electronics Box Performance Test Procedure
431-PROC-000023	LRO Power Subsystem Electronics Safe-to-Mate Procedure
431-PROC-000024	LRO Solar Array Acceptance Test Procedure
431-PROC-000025	LRO Solar Array Aliveness Test Procedure
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431-PROC-000040	LRO Solar Array Mechanical Integration Procedure
431-PROC-000041	LRO Solar Array Electrical Integration Procedure
431-PROC-000156	LRO Power Subsystem Electronics to Ground Support Equipment Electrical Integration Procedure
431-SPEC-000016	LRO Power Subsystem Electronics Specification
431-SPEC-000027	LRO Power Subsystem Electronics Monitor Card Specification
431-SPEC-000032	LRO Battery Specification
431-SPEC-000037	LRO Solar Array Specification
431-SOW-000038	LRO Solar Array Statement of Work

2.2 REFERENCE DOCUMENTS

430-RQMT-000006	Robotic Lunar Exploration Program Mission Assurance Requirements
431-HDBK-000052	Telemetry and Command Handbook (Vol. 1of 2 – Data Formats)
431-HDBK-000053	Telemetry and Command Handbook (Vol. 2 of 2 – Database)
431-ICD-000049	Ground System Interface Control Document
431-OPS-000042	Mission Concept of Operations
431-PLAN-000005	Systems Engineering Management Plan
431-PLAN-000101	Observatory Verification Plan
431-PLAN-000110	Contamination Control Plan
431-PROC-000006	Risk Management Implementation Procedure
431-RQMT-000004	Lunar Reconnaissance Orbiter Mission Requirements Document
431-RQMT-000045	Radiation Requirements for the Lunar Reconnaissance Orbiter

431-RQMT-000048	Detailed Mission Requirements for the Lunar Reconnaissance Orbiter Ground System
431-SOW-000038	Solar Array Statement of Work
431-SOW-000125	Battery Statement of Work for the Lunar Reconnaissance Orbiter
431-SPEC-000008	Electrical System Specification
431-SPEC-000012	Mechanical Systems Specification
431-SPEC-000057	ITOS Specification
431-SPEC-000063	Flight Dynamics Specification
431-SPEC-000091	General Thermal Subsystem Specification
431-SPEC-000112	Technical Resource Allocations Specification
AFSCM 91-710	Range Safety User Requirements Manual and the National Electric Code (NEC)
ASTM E-595	Standard Test Method for Total Mass Loss and Collected Volatile Condensable Materials from Out-gassing in a Vacuum Environment
EEE-INST-002	Instructions for EEE Parts Selection, Screening, Qualification, and Derating
MIL-HDBK-217	Reliability Predictions of Electronic Equipment
MIL-STD-338B	Electronic Reliability Design
MIL-STD-756B	Reliability Modeling and Prediction
MIL-STD-975	NASA Standard Electrical, Electronic, and Electromechanical (EEE) Parts List
NPD 8720.1	NASA Reliability and Maintainability (R&M) Program Policy
RADC-TR-85-229	Reliability Predictions for Spacecraft

3.0 LRO ELECTRICAL POWER SUBSYSTEM

3.1 SYSTEM REQUIREMENTS

3.1.1 Observatory

3.1.1.1 Mission Lifetime

The Orbiter shall be designed for a one-year mission life (after reaching lunar orbit and post commissioning activities which are expected to be 2 months).

The primary mission shall be followed by an extended mission with a total mission life of up to 60 months as resources allow. The extended mission is not to drive design requirements or resources.

3.1.1.2 Environmental

3.1.1.2.1 Space

The orbiter shall survive the dynamic loads of the mission as specified in LRO Mechanical Systems Specification (431-SPEC-000012).

The orbiter shall survive the thermal environments of the mission as specified in LRO General Thermal Subsystem Specification (431-SPEC-000091).

The spacecraft shall survive the radiation environment of the mission as specified in Radiation Requirements for the Lunar Reconnaissance Orbiter (431-RQMT-000045).

Contamination of sensitive portions of the space segment shall not prevent the mission from meeting requirements. Acceptable levels of contamination shall be maintained on the Orbiter per the LRO Contamination Control Plan (431-PLAN-000110).

Orbiter residual and induced magnetic fields and the natural orbital environmental magnetic field shall not disrupt Orbiter operations or corrupt mission science.

3.1.1.3 Redundancy, Fault Tolerance, Safety Design

3.1.1.3.1 Redundancy and Fault Tolerance

The LRO shall be designed, fabricated, integrated, tested and operated using a single string approach with selective redundancy.

3.1.1.3.2 Safety

The spacecraft, instruments, and associated Ground Support Equipment (GSE) shall meet the safety design requirements contained in AFSCM 91-710, Range Safety User Requirements Manual and the National Electric Code (NEC).

3.1.1.4 Reliability

The LRO Project shall plan and implement a reliability program applicable to the development of all software and hardware products and processes consistent with the requirements delineated in the Lunar Reconnaissance Orbiter Mission Assurance Requirements (431-RQMT-000004) and the guidelines contained in the following standards and references:

- NPD 8720.1 NASA Reliability and Maintainability (R&M) Program Policy
- MIL-HDBK-217 Reliability Predictions of Electronic Equipment
- MIL-STD-338B Electronic Reliability Design
- MIL-STD-756B Reliability Modeling and Prediction
- RADC-TR-85-229 Reliability Predictions for Spacecraft

3.1.2 Spacecraft

3.1.2.1 Electrical Power Subsystem Mass

The EPS mass shall not exceed the allocation as specified LRO Technical Resource Allocations Specification (431-SPEC-000112).

Table 3-1. LRO EPS Mass Estimates

	Subsystem Mass Allocation Kg.
SA Add - On+Subs(7M*M)	20
PSE	16
Battery	40
Total	76

3.1.2.2 Power Requirements

3.1.2.2.1 Observatory Operational Modes

The EPS shall support all observatory operational modes and mission phases as defined in the Lunar Reconnaissance Orbiter Mission Assurance Requirements (431-RQMT-000004) and summarized here:

- Pre-launch/Launch Readiness
- Launch and Lunar Transfer
- Orbiter Activation/Commissioning
- Measurement Operations
- Extended Mission operations
- End-of-Mission/Disposal

3.1.2.2.2 Subsystem Orbit Average Power Capability

The LRO EPS shall support the total power allocation of the LRO spacecraft bus and instrument suite with margins as specified in LRO Technical Resource Allocations Specification (431-SPEC-000112). The system power capability shall not exceed 824 Watts (W) orbital average.

3.1.2.2.3 Subsystem Peak Power Capability

The LRO EPS shall provide an unregulated 22-35Volts (V) bus as measured at the PSE to support a peak load of 1500W for a period of 5 minutes.

3.1.2.2.4 Safe-hold

When in "Sun Pointing" mode the Attitude Control System (ACS) shall put the sun within 15 degrees of the specified position.

3.1.2.2.5 Sizing Parameters

The design load (orbital average) allocations shall be used for the sizing of the power subsystem services (SA, battery, and electric distribution subsystem).

The peak power allocations shall be used for the sizing of harnessing and circuit protection.

3.1.2.2.6 Subsystem Power Consumption

The EPS power consumption shall not exceed the allocation as specified LRO Technical Resource Allocations Specification (431-SPEC-000112).

Table 3-2. LRO EPS Power Consumption Estimate

EPS Power Allocation	Orbit Avg. W	Peak W
PSE	45	45
Battery	N/A	N/A
SA	N/A	N/A
Total	45	45

3.1.2.2.7 Subsystem Power Distribution

The PSE shall provide non-switched primary power to the spacecraft receiver and Command and Data Handling (C&DH) subsystem.

Over current protected primary power shall be provided to the other spacecraft subsystems and instruments per Lunar Reconnaissance Orbiter Mission Assurance Requirements (431-RQMT-000004).

3.1.2.2.8 Subsystem Voltage Tolerance Capability

The EPS, except the battery, shall survive indefinite 0 to 40 VDC exposure without damage.

3.1.2.2.9 Spacecraft Minimum Power

To insure compliance with voltage specifications the spacecraft will present a minimum load to the power bus of 180W whenever the system is powered up.

3.1.3 Ground System Interfaces

3.1.3.1 Subsystems Ground Operations Requirements

The complete LRO Ground System requirements are delineated in the Detailed Mission Requirements for the Lunar Reconnaissance Orbiter Ground System (431-RQMT-000048). In the event of conflicts between the contents of this document and the specification this document shall take precedence.

3.2 SUBSYSTEM FUNCTIONAL REQUIREMENTS

3.2.1 Electrical Design

The power subsystem shall be designed to conform to the Lunar Reconnaissance Orbiter Electrical Systems Specification (431-SPEC-000008).

3.2.2 EPS Components

The power subsystem shall consist of the PSE, SA, and Battery.

3.2.2.1 Component Reference Axis

The mission attitude reference frame shown (Figure 3-1) will be supported by the EPS according to the following axis definition:

- +X axis parallel to orbital velocity vector (nominal launch configuration)
- +Z axis parallel to Nadir vector, and
- +Y axis normal to the orbital velocity vector according to the right hand rule

3.2.3 System Topology

The LRO power system shall be a Direct Energy Transfer (DET) design distributing main bus power only to all instruments and subsystems.

3.2.4 Functional Description

3.2.4.1 Specific Functions

The LRO EPS shall provide the following functions:

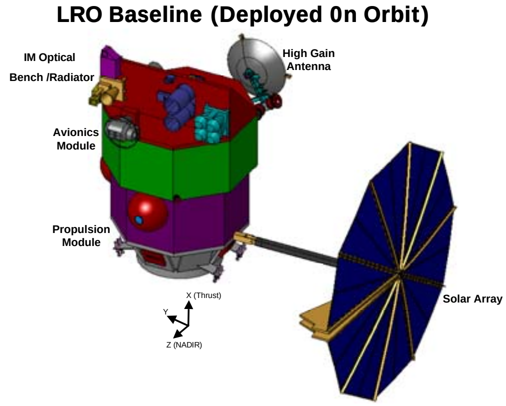
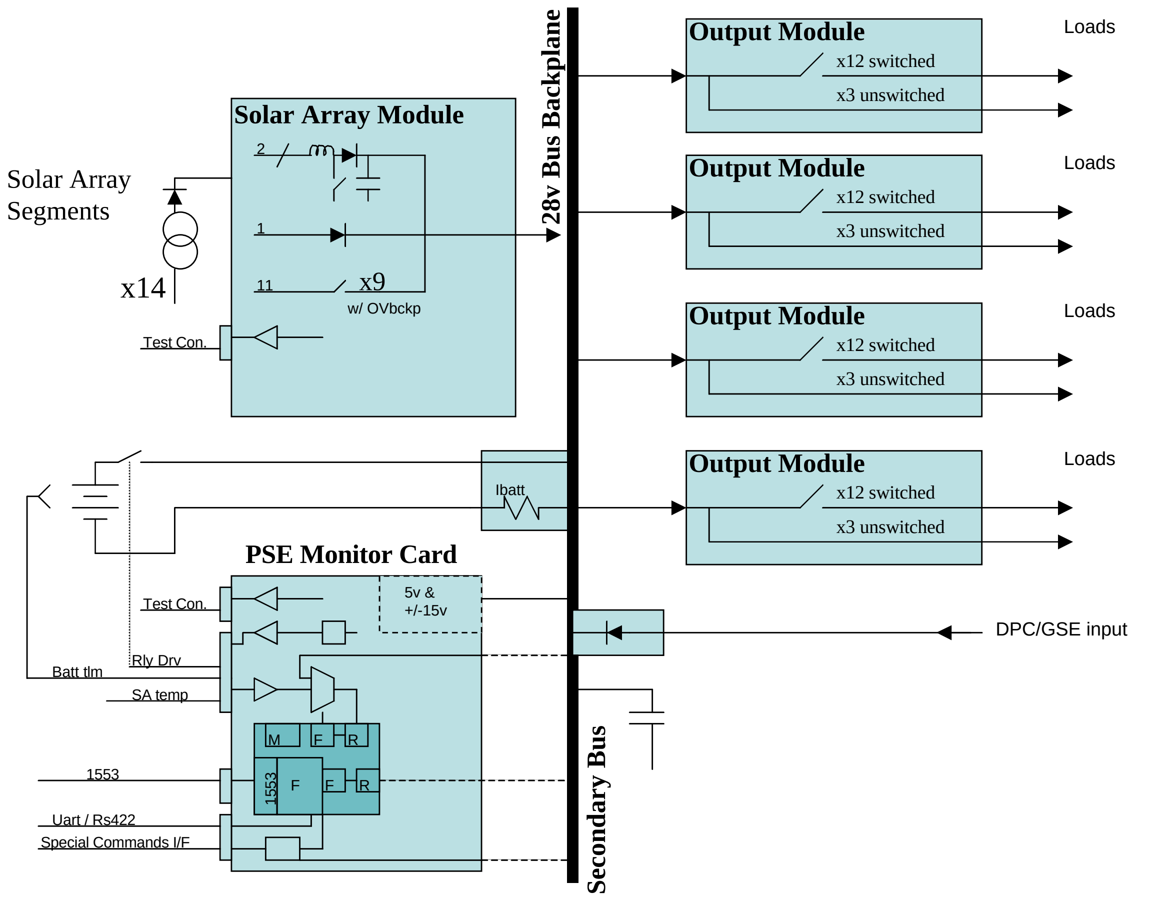


Figure 3-1. LRO Reference Axis

- Satisfy load power requirements for all nominal mission modes
- Provide battery charging and control
- Provide interface to the C&DH subsystem for power system performance, monitoring, configuration and control
- Provide power interface to the electrical subsystem
- Provide capability to respond to 4 (#) Special Commands
- Provide power distribution and load switching capability
- Provide external power interfaces for Integration and Test (I&T), ground, and pre-launch operations
- Provide bus protection by automatically shedding loads if a defined fault condition occurs
- Restoration of power to shed loads shall be remotely controllable via ground command.

3.2.4.2 Battery Enable Relay

A battery enable relay will be incorporated into the power system. This relay will be disabled, enabled through the GSE for use during I&T. A relay enable command only will also be provided through the PSE. No in flight disable command capability will be allowed.



*All double sided cards with heatsinks

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Figure 3-2. Block Diagram of LRO Power System

3.3 CONFIGURATION ITEM REQUIREMENTS

The following list the primary detailed design documentation for the EPS.

3.3.1 LRO Power Subsystem Electronics

431-ICD-000018	LRO Power Subsystem Electronics Electrical Interface Control Document
431-PROC-000021	LRO Power Subsystem Electronics Box Functional Test Procedure
431-PROC-000022	LRO Power Subsystem Electronics Box Performance Test Procedure
431-PROC-000023	LRO Power Subsystem Electronics Safe-to-Mate Procedure
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431-PROC-000156	LRO Power Subsystem Electronics to Ground Support Equipment Electrical Integration Procedure
431-SPEC-000016	LRO Power Subsystem Electronics Specification
431-SPEC-000027	LRO Power Subsystem Electronics Monitor Card Specification

3.3.2 LRO Battery

431-PLAN-000033	LRO Battery Handling Plan
431-PROC-000034	LRO Battery Mechanical Integration Procedure
431-PROC-000035	LRO Battery Electrical Integration Procedure
431-SOW-000125	LRO Battery Statement of Work
431-SPEC-000032	LRO Battery Specification

3.3.3 LRO Solar Array

431-PROC-000024	LRO Solar Array Acceptance Test Procedure
431-PROC-000025	LRO Solar Array Aliveness Test Procedure
431-PROC-000039	LRO Solar Array Handling Procedure
431-PROC-000040	LRO Solar Array Mechanical Integration Procedure
431-PROC-000041	LRO Solar Array Electrical Integration Procedure
431-SOW-000038	LRO Solar Array Statement of Work
431-SPEC-000037	LRO Solar Array Specification

3.3.4 Solar Array

3.3.4.1 Mission Orbit Altitude

The SA shall survive the thermal and radiation environment of a primary circular polar mapping orbit with a nominal mean altitude of 50 +/- 20 kilometers (km). Altitude is measured to mean lunar surface.

3.3.4.2 Mission Orbit Inclination

The EPS shall support a lunar polar orbit of 90 degrees +/- 1 degree longitude. This orbit creates eclipse periods that vary seasonally from 0 to 48 minutes. The EPS shall recharge the battery after each eclipse assuming a DOD of no more than 30%.

3.3.4.3 SA Tracking Accuracy

The SA shall track the sun from Beta angles 0 to 90 degrees with an accuracy of 5 degrees to support power generation over the one-year mission.

3.3.4.4 Physical

The SA is expected to be a circular flexible array of approximately 9 square meters with exact dimensions as specified in the Lunar Reconnaissance Orbiter Solar Array Specification (431-SPEC-000037).

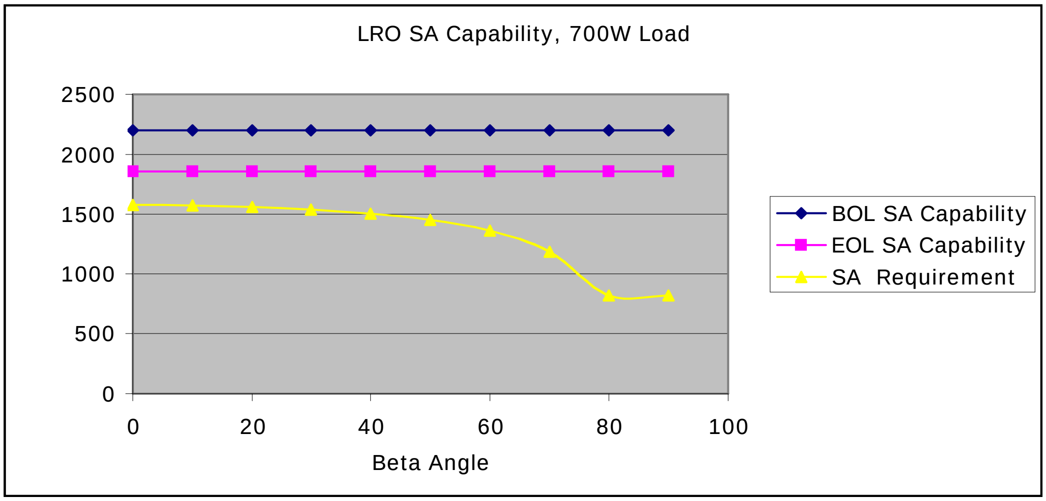


Figure 3-3. SA Output vs. Beta Angle

3.3.4.5 Power Capability

3.3.4.5.1 Beginning of Life

With the wings in the flight configuration at air-mass-zero (AM0) illumination, no shadowing, and an array design voltage of 35V the maximum peak power output of the total array shall not exceed 2120W at any Beta angle.

3.3.4.5.2 End of Life

At Beta 0, in flight configuration, after exposure to the space environment in a lunar orbit for 14 months, the minimum total array output including all degradation factors and shadowing shall exceed 1849W at a design voltage of 35V.

3.3.4.6 Array Segmentation

The SA shall be divided into 14 segments, each of which shall provide no greater than 4.7A under max. peak Beginning of Life (BOL) conditions.

The power from each segment will be provided to the PSE through its own set of power and return wires.

3.3.4.7 Temperature Measurement

Four temperature sensors will be incorporated into the array design.

3.3.5 Power Supply Electronics

3.3.5.1 Requirements

3.3.5.1.1 PSE Components

The PSE shall consist of a single box, 1 bus selectively redundant design, with sub-components listed below:

- 1 Solar Array Module (SAM)
  - Up to 14 segments
- 1 Power Monitor Card (PMC)
- Output Modules (OM)
  - Up to 4 cards and 48 switched and 12 non-switched services total
- 1 Backplane
- 1 Chassis

3.3.5.1.2 Hardware Responsibility

Code 563 will be responsible for providing the following PSE components. Spares will consist of components as specified, not completed cards.

Table 3-3. PSE Card Deliverables

Flight Cards:				
	PSE			
Card Type		**Spares**	**TOTALS**	**Card Descriptions:**
PMC	1	1		Power Monitor Card
OM	4	1		Output Module card
SAM	1	1		Solar Array Module
Backplane	1			
Chassis	1	Flight spares include pcb, heatsink, and all parts		
ETU Cards:				
Card Type		**Spares**	**TOTALS**	**Card Descriptions:**
PMC	1	0		Power Monitor Card
OM	4	0		Output Module card
SAM	1	0		Solar Array Module
Backplane	1			
Chassis	1	ETU spares include pcb, heatsink except where noted. * These ETU spares include pcb, heatsink, and parts		

3.3.5.1.3 PSE Dimensions and Volume

The PSE including connectors will fit in a volume described by the following dimensions:

Table 3-4. PSE Physical Parameters (Estimate)

Battery Estimate			
	Height	Width	Length
Dimensions (Cm)	24.89	27.94	26.5
Mass (Kg.)		14	

3.3.5.1.4 Functional description

The PSE will provide power control as described below:

- The PSE PMC shall monitor battery voltage, current, and temperature and will determine the power flow within the system by providing battery charge current and shunt drive signals to the Solar Array Module (SAM).
- Excess SA power shall be used to charge the battery up to a limit of 30A.
- Power in excess of load and charge requirements shall be shunted back to the array.
- The PSE shall charge the battery based on a programmable voltage clamp.

- The PSE shall maintain the programmed voltage clamp limit by allowing the battery current to taper naturally as required.
- The SAM shall be designed with hardware back-ups in case the control signals from the PMC are lost.

3.3.5.1.5 Battery Recharge Requirement

The power subsystem shall restore full battery capacity during each orbit after a nominal battery discharge not exceeding 30%, and shall be capable of supporting a DOD of 60% for launch phase and 80% for lunar eclipse periods and other mission modes for a total of not to exceed 10 cycles.

In the case of lunar eclipses or special operating modes where battery Depth of Discharge (DOD) does not exceed 80% multiple orbits may be required to obtain battery recharge.

3.3.5.1.6 Output Impedance

The bus source impedance, as measured at the output connector of the PSE, shall be less than 80 milliohms for frequencies between 0.1 hertz (Hz) and 100 kilohertz (kHz).

3.3.5.1.7 Voltage Transients

The power subsystem unregulated output voltage shall remain between 22-35 VDC as measured at the PSE terminals except for load turn-on transients and failures. For an initial load turn-on with a 10A peak and 10 microsecond duration followed by a 20 milli-Ampere/ microsecond maximum rate of rise of the load current to a peak load of 300% of the maximum steady state current, the voltage transients shall not exceed +/- 3V.

3.3.5.1.8 Output Ripple

The bus ripple contributed by PSE shall be less than 0.3V peak-to-peak (p-p).

3.3.5.1.9 Software Fault Detection

Software fault detection is done by monitoring the following signals:

- Bus voltage
- Battery voltage
- Battery cell voltages
- Battery temperature
- Battery current
- Switch Status

3.3.2.1.9.1 Commanding

All software driven fault detection circuits shall be capable of being individually disabled by ground command.

3.3.5.1.10 Battery Command and Telemetry Parameters

Battery and cell parameters shall be programmable within the ranges specified in the following table.

Table 3-5. Battery/Cell Program Parameters

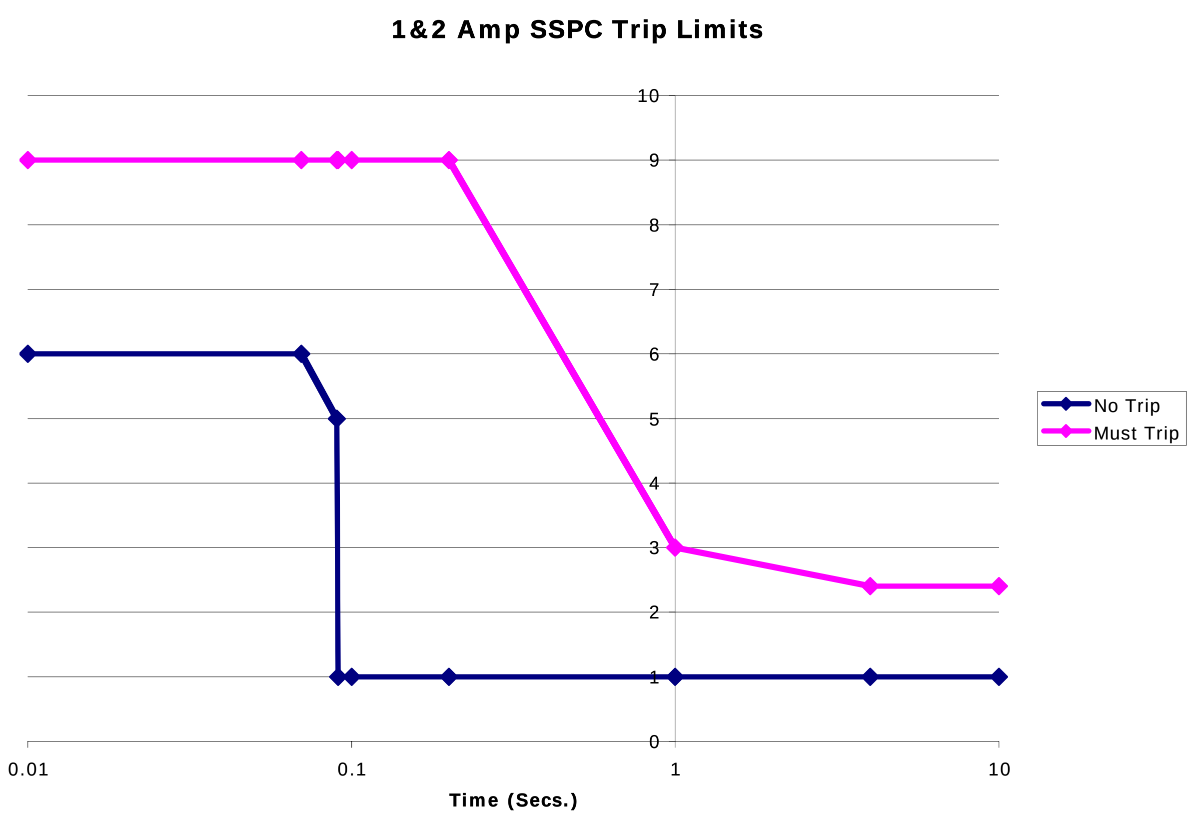
Hardware Overvoltage Minimum	34	V
Hardware Overvoltage Maximum	35	V
Battery Charge Voltage Minimum	23	V
Battery Charge Voltage Maximum	33.8	V
Battery Charge Voltage Increment	200	Mv
Charge Accuracy	+/- 130	Mv
Battery Voltage Telemetry Minimum	0	V
Battery Voltage Telemetry Maximum	35	V
Battery Voltage Telemetry Accuracy	+/- 130	Mv
Battery Cell Voltage Telemetry Minimum	0	V
Battery Cell Voltage Telemetry Maximum	5	V
Battery Cell Voltage Telemetry Accuracy	+/- 95	Mv
FDC Battery Voltage Undervoltage Minimum	0	V
FDC Battery Voltage Undervoltage Maximum	35	V
FDC Battery Voltage Overvoltage Minimum	0	V
FDC Battery Voltage Overvoltage Maximum	35	V
FDC Battery Voltage Voltage Incremement	200	Mv
FDC Battery Voltage Voltage Accuracy	+/- 130	Mv
FDC Cell Voltage Undervoltage Minimum	0	V
FDC Cell Voltage Undervoltage Maximum	5	V
FDC Cell Voltage Overvoltage Minimum	0	V
FDC Cell Voltage Overvoltage Maximum	5	V
FDC Cell Voltage Voltage Incremement	20	Mv
FDC Cell Voltage Voltage Accuracy	+/- 95	Mv
Battery Charge Current Command Minimum	0	A
Battery Charge Current Command Maximum	30	A
Battery Charge Current Command Increment	200	Ma
Battery Charge Current Command Accuracy	50	Ma

Battery Current Telemetry (Charging maximum)	-60	A
Battery Current Telemetry (Discharging maximum)	30	A
Battery Current Telemetry Accuracy	.224 to .645	A
Battery Temperature Telemetry Minimum	-10	C
Battery Temperature Telemetry Maximum	50	C

3.3.5.1.11 Hardware Fault Detection

Hardware over current protection of the main power bus shall be done for a maximum of 48 switched services. Protection shall be accomplished with the use of Solid State Power Controllers (SSPC) with re-settable circuit breaker capability. These devices respond with a circuit break according to an I^2t curve. By monitoring the status bits of these SSPCs, the fault can be identified in the part or the load, and action can be taken accordingly.

SSPC service over current trip point shall be as defined in Figure 3-4.



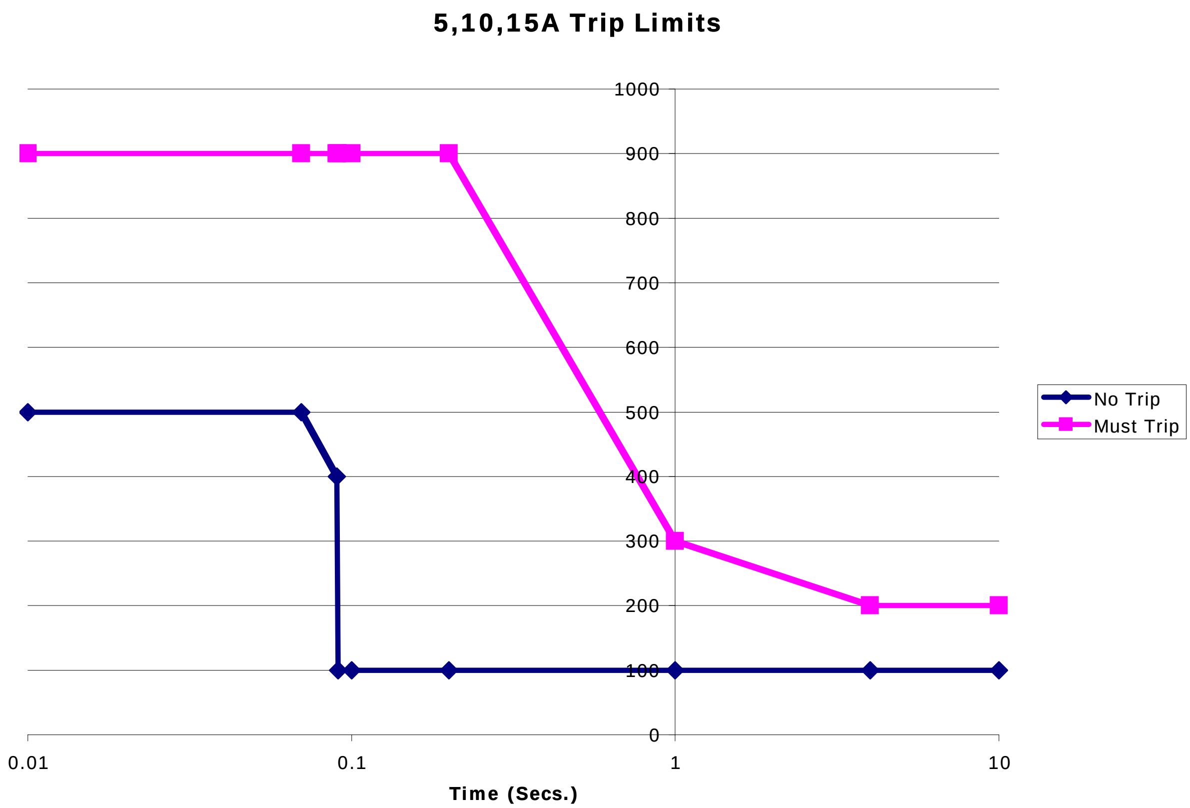


Figure 3-4. SSPC Trip Characteristics

3.3.5.1.12 Un-Switched Services

Un-switched power shall be un-fused since the loss of any critical component on one of these services means the loss of mission. Un-switched services should be fused as a safety precaution during I&T.

3.3.5.1.13 PSE Test Connector

PSE test connectors will be located on the SAM and PMC. The signals available on these connectors are:

SAM current	Chassis current
SAM Pulse Width Modulator (PWM) status	+5V monitor
+15V monitor	-15V monitor
Battery current	Battery voltage
Bus voltage	Battery cell voltage

3.3.5.1.14 Telemetry

The PSE will provide current telemetry for each individual OM service. Other specific telemetry points are detailed in the LRO Power Subsystem Electronics Specification (431-SPEC-000016).

3.3.6 Battery

3.3.6.1 Requirements

3.3.6.1.1 Ampere Hour Capacity

The LRO battery energy storage capacity shall be > =80 Ampere Hour (AH)

3.3.6.1.2 Operational Voltage range

The LRO battery operational voltage range shall be between 22 and 35V including the situation of a 1 out of 8 battery cell failure.

3.3.6.1.3 Battery Physical Parameters

The battery including connectors will fit in a volume described by the following dimensions:

Table 3-6. Battery Estimates

Battery Estimates			
	Height	Width	Length
Dimensions (Cm)	27.94	27.94	55.88
Mass (Kg.)		40	

3.3.6.1.4 Battery Life

After storage for a maximum of 3 years the battery shall support the LRO for a minimum of 18 months with a 1cell failure.

3.3.6.1.5 Depth of Discharge

The nominal Depth of Discharge (DOD) during a typical lunar orbit shall be limited to 30% with 1 cell failure.

In addition the battery shall be capable of supporting a DOD of 60% for launch phase and 80% for lunar eclipse periods and other mission modes for a total of not to exceed 10 cycles.

3.3.6.1.6 Usage

The battery shall support full mission loads during eclipse portions of any orbit as well as periods when SA power is unavailable or is insufficient to support the observatory loads (i.e. pre-acquisition or maneuver mode operations).

3.3.6.1.7 Connectors

The battery shall provide three interface connectors, power, telemetry, and test.

3.3.6.1.7.1 Power Connector

All power shall be routed through a single connector for interface with the spacecraft.
The connector and all wiring shall be appropriately sized to provide current at a C rate.

3.3.6.1.7.2 Telemetry Connector

The battery will provide the following interface to the spacecraft for telemetry and control:

- Battery Voltage
- Cell Voltage (except for designs using multiple small cells)
- Battery temperature
- Relay status
- Relay disable command

3.3.6.1.7.3 Test Connector

The battery will provide the following interface to the battery GSE for monitoring and control:

- Battery Voltage
- Cell Voltage (except for designs using multiple small cells)
- Battery temperature
- Relay status
- Relay enable/disable command
- Plus and minus lines for battery charge at C/10

3.3.7 Ground Support Equipment

GSE will be provided to allow full qualification level subsystem testing at the bench level as well as operational level testing during the I&T process.

3.3.7.1 Box Level Qualification Testing

The following units will be provided:

- Computer interface unit (ITOS)
- SA Simulator (SAS)
- Battery Simulator
- Direct Power conditioner (DPC)
- Load Simulator

3.3.7.2 Integration and Test Power

I&T power shall be made available through the following units:

- DPC
- SAS
- Battery Simulator

3.3.7.3 Test Battery

One battery shall be supplied by GSFC for flight qualification and I&T testing.

3.3.7.4 Battery Conditioning Console

A battery-conditioning console shall be used to condition and check out the flight battery before installation into the observatory in accordance with the Lunar Reconnaissance Orbiter Battery Handling Plan (431-PLAN-000033).

3.3.7.5 Battery Cooling Cart/AC Unit

A high-capacity air condition unit shall be used for battery cooling during I&T and at the launch site (with fairing on) to cool battery during reconditioning.

3.3.7.6 Solar Array

The array panel illumination facility Large Area Pulsed Solar Simulator (LAPSS) shall be used to illuminate the LRO solar panels after reception from the vendor and after completion of selected spacecraft environmental testing with the integrated SA.

A portable solar illuminator "sun gun" shall be used to illuminate individual array panels to verify electrical integrity of each panel during subsystem integration.

3.4 INTEGRATION AND TESTING/VERIFICATION

3.4.1 Responsibility for Verification

Unless otherwise specified, GSFC shall be responsible for providing all facilities, ancillary equipment, tools, test sets, etc.; conducting all tests; and, recording and processing all data necessary to fulfill the test requirements.

3.4.2 Observatory

The power subsystem shall be verified to withstand or shall be protected against the worst probable combination of environments as specified in the Lunar Reconnaissance Orbiter Observatory Verification Plan (431-PLAN-000101). This shall be accomplished by a combination of test and analysis. The verification program shall consist of activities whose result shall demonstrate design qualification, allow acceptance of hardware, and demonstrate that performance, operational safety, and interface requirements of this specification are met.

3.4.3 Subsystem Derived

The power subsystem shall be verified to insure mission requirements can be met. Verification shall consist of test, analysis, and inspection. Detailed test requirements for each subsystem component will be part of each component specification. The requirements for recording test data and witnessing test results shall be as specified in the applicable test procedure(s).

Table 3-7. Verification Method

LRO Power System Verification Matrix

Paragraph Number	Description	Component Level	Subsystem Level	Spacecraft Level
	Mission Life	A	A	A
	Max. Min Observatory Loads	A,T	A,T	A,T
	Launch/deployment	N/A	N/A	T
	Mission Modes Testing	N/A	N/A	T
	Power System Telemetry	A,T	A,T	A, T
	Power System Commanding	A,T	A,T	A,T
	EPS to S/C interface verification	A,T,I	A,T	A,T
	Solar Array- BOL, EOL performance	A,T	A,T	T
	PSE voltage control tests	A,T	A,T	A,T
	Battery- 80AH, 30%DOD	A,T	A,T	A,T
	Single Point Failures - Critical Redundancy	A	A	A
	Test Connector Signals - PSE/Battery	I,T	T	T
	PSE over-voltage control test	A,T	T	T
	Output Impedance	A,T	A,T	N/A
	Voltage transients	A,T	A,T	A
	Output ripple	A,T	A.T	N/A
	Battery recharge functions	A,T	A,T	A,T
	Bus & battery fault detection software	A,T	A,T	A,T

Paragraph Number	Description	Component Level	Subsystem Level	Spacecraft Level
	Bus & battery fault detection hardware	A,T	A,T	A,T

A=analysis, I=inspection, T=test, N/A = not applicable

3.4.4 Component Level

3.4.4.1 Solar Array

SA panel component level testing shall be conducted at the SA vendor's facility prior to delivery. Each SA panel shall undergo a series of functional, electrical, and environmental tests as defined in the Lunar Reconnaissance Orbiter Solar Array Specification (431-SPEC-000037) and Lunar Reconnaissance Orbiter Solar Array Statement of Work (431-SOW-000038).

3.4.4.2 Battery

The flight battery shall be activated, capacity and leak checked, and conformally coated. Testing shall be as defined in the Lunar Reconnaissance Orbiter Battery Specification (431-SPEC-000032) and Lunar Reconnaissance Orbiter Battery Statement of Work (431-SOW-000125).

3.4.4.3 Power Subsystem Electronics

The PSE shall undergo functional and qualification environmental tests. The following tests shall be conducted on the flight unit:

- Command-ability
- Output ripple
- Regulation
- Fault detection
- Charge rates
- Efficiency and losses
- Power consumption
- Battery charge control
- Electrical interfaces
- Transient response
- Telemetry calibration
- Electromagnetic Compatibility (EMC)/Electromagnetic Interference (EMI) test
- Vibration test
- Thermal vacuum test
- Physical measurements

Details are described in the Lunar Reconnaissance Orbiter Power Subsystem Electronics Specification (431-SPEC-000016).

3.5 CONNECTORS

3.5.1 <u>Spacecraft Interface</u>

The spacecraft will provide the following skin connectors for EPS interface:

- 1 - SA Power and Signal
- 1 - SA Arming
- 1- SA Test
- 1 - Battery Power
- 1- Battery Test
- 1 -Battery Arming
- 1 - Umbilical

3.5.1.1 Launch Vehicle Umbilical Interface Requirements

The LRO launch vehicle umbilical shall, at a minimum, support the following operations or functions:

- Support DPC interface for soft power-on of the spacecraft
- Support SAS for spacecraft power and battery charging
- Support a battery relay On/Off pulse for power-off of the spacecraft
- Support a hard-line command and telemetry interface for communications to the spacecraft during integrated operations.
- Provide battery relay on/off status regardless of spacecraft power on/off state
- Support battery charge with spacecraft powered off
- Provide battery voltage, battery current, and battery temperature regardless of spacecraft power on/off state
- Provide spacecraft bus voltage and bus current

3.5.1.1.1 Signal Range

The range for all signals is -10V to +10V. For gain and resolution of these signals as well as other analog signals read by the Control Module, see the Lunar Reconnaissance Orbiter Power Subsystem Electronics Specification (431-SPEC-000016).

3.6 PROCESS/DESIGN

3.6.1 Observatory

LRO design and construction requirements are delineated in the following sections of the Robotics Lunar Exploration Program Mission Assurance Requirements document (430-RQMT-000006):

- Section 6: Parts Requirements
- Section 7: Materials, Processes, and Lubrication Requirements
- Section 9: Design Verification Requirements
- Section 10: Workmanship Requirements
- Section 12: Contamination Control Requirements

The following paragraphs delineate the technical design and construction requirements that are contained in the above sections of the Mission Assurance Requirements (MAR).

4.0 MISSION ASSURANCE/PARTS ASSURANCE REQUIREMENTS

4.1 CALIBRATION

All test equipment used in the development of flight hardware will be calibrated per Robotic Lunar Exploration Program Mission Assurance Requirements (430-RQMT-000006).

4.2 QUALITY

4.2.1 Material Processes and Control

A material and processes program beginning with the design stage of each power subsystem component shall be implemented in accordance with Robotic Lunar Exploration Program Mission Assurance Requirements (430-RQMT-000006).

4.2.2 Component Materials

Materials used in space vacuum areas shall be resistant to excessive out gassing as determined in accordance with ASTM E-595. The following source may be used as a guide in the selection and application of materials: NASA RP 1124. Any material with unknown properties or properties that might jeopardize performance, safety, or reliability of the observatory shall be demonstrated to be acceptable for the application on the basis of similarity, analysis, test, inspection, or existing data.

4.2.3 Level

The LRO Project parts reliability requirement is Quality Level 2 in accordance with GSFC EEE-INST-002, "Instructions for EEE Parts Selection, Screening, Qualification, and Derating". The document provides detailed instructions for the selection and testing of electronic parts to be used in GSFC space flight programs depending on mission requirements. The NASA Parts Selection List (NPSL) may be used as a vehicle for parts selection to the specified quality levels.

4.2.4 Electronic Components

The Project-Approved Parts List (PAPL) shall be the sole source of approved parts. All electronic parts shall be selected and processed in accordance with GSFC EEE-INST-002, Instructions for EEE Parts Selection, Screening and Qualification, for Grade 2 parts. Any part not included in the listing of MIL-STD 975 B/C (NASA), the PPL-21, or the GSFC-approved contractor's Preferred Parts List (PPL) is considered nonstandard.

The use of non-standard parts shall be approved by the project Parts Control Board (PCB). The acceptability of a non-standard part for a particular application shall be established. Similarity, existing data, analysis or test, and inspection results may be used to demonstrate acceptability.

4.2.4.1 Component Radiation Requirements

EPS Electrical, Electronic, and Electromechanical (EEE) parts shall meet the requirements of the Radiation Requirements for the Lunar Reconnaissance Orbiter (431-RQMT-00045).

4.3 STRUCTURAL DESIGN

The structural design of each subsystem component shall satisfy the fracture control requirements of the Lunar Reconnaissance Orbiter Mechanical Systems Specification (431-RQMT-000012).

4.4 SAFETY

All activities required for design, development, fabrication, and testing of the LRO EPS will conform to the Lunar Reconnaissance Orbiter Mission Requirements Document (431-RQMT-000004).

5.0 <u>VERIFICATION</u>

5.1 COMPLIANCE MATRIX

A compliance matrix will be generated to insure all higher level requirements have been met and can be tracked.

Appendix A. Abbreviations and Acronyms

Abbreviation/ Acronym	DEFINITION
A	Analysis
A	Ampere
AC	Air Conditioner
ACS	Attitude Control System
AFSCM	Air Force Space Command Manual
AH	Ampere Hour
ASTM	American Society for Testing of Materials
BOL	Beginning of Life
C&DH	Command and Data Handling
CCB	Configuration Control Board
CCR	Configuration Change Request
cm	Centimeter
CM	Configuration Management
CMO	Configuration Management Officer
DET	Direct Energy Transfer
DPC	Direct Power Conditioner
DOD	Direct Energy Discharge
EEE	Electrical, Electronic, and Electromechanical
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EPS	Electrical Power Subsystem
ETU	Engineering Test Unit
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
HDBK	Handbook
Hz	Hertz
I	Inspection
I&T	Integration and Test
ICD	Interface Control Documents
INST	Instrument
ITOS	Integrated Test and Operating System
Kg	Kilogram
kHz	Kilohertz
Km	Kilometer
LAPSS	Large Are Pulsed Solar Simulator
LRO	Lunar Reconnaissance Orbiter
MIL	Military
mV	milli-Volt

Abbreviation/ Acronym	DEFINITION
N/A	Not applicable
NASA	National Aeronautics and Space Administration
NEC	National Electric Code
NPD	NASA Policy Directive
NPSL	NASA Parts Selection List
OM	Output Module
OV	Over Voltage
p-p	Peak-to-peak
PCB	Parts Control Board
PMC	Power Monitor Card
PPL	Preferred Parts List
PROC	Procedure
PRT	Platinum Resistor Temperature
PSE	Power Subsystem Electronics
PWM	Pulse Width nModulator
R&M	Reliability and Maintainability
RLEP	Robotic Lunar Exploration Program
RQMT	Requirement
SA	Solar Array
SAM	Solar Array Module
SAS	Solar Array Simulator
SOW	Statement of Work
SPEC	Specification
SSPC	Solid State Power Controllers
STD	Standard
T	Test
UV	Under Voltage
V	Volts
VDC	Volts Direct Current
W	Watts