Common Instrument Interface (CII) Power Guidelines

DRAFT

John Carey
CII Team

21 April 2011

Design Goals

- The CII guidelines are provided to increase instrument compatibility with spacecraft so that the maximum number of Missions of Opportunity (MoO) can be realized
- The CII guidelines are designed to allow both the instrument and the spacecraft providers to work independently through the early phases of the applicable design processes
- Final implementation details will still require some resolution between the instrument and the spacecraft once paired in an MoO via the Spacecraft to Instrument ICDs

Key Power Guidelines

- ## 3.1.2.1 Survival Heater Power Bus

  There should be three redundant power buses feeding each Instrument. They are designated as the Power Feed #1, Power Feed #2 and the Survival Heater Power Feed

- ## 3.1.2.6 Survival Heater Power Bus

  The Survival Heater Power Bus is a redundant power bus. Survival mode is a Spacecraft initiated mode in which operational power to the Instruments is interrupted. Survival power is used only for heaters and associated passive control circuitry which maintain the Instrument at minimum turn-on temperature. Sides A and B are designated as Survival Heater Bus A and Survival Heater Bus B

- ## 3.2.2 Average Power Consumptions

  Average power is the total power averaged over 2 orbits

- ## 3.2.2 Peak Power Consumptions

  Peak power is the maximum power averaged over any 10 msec period

Key Power Guidelines

- ## 3.2.4 Allocation of Instrument Power

  Each Instrument mode should be allocated peak NTE and average NTE power shown in Table 3.2-1

MODE	PEAK (Watts)	Average (Watts)
OFF	0	0
SURVIVAL	60	30
ACTIVATION	400	200
SAFE	200	100
OPERATION	400	200

Table 3.2-1 Instrument Power Allocation

- ## 3.2.5.1 Primary Instrument Voltage

  The Spacecraft power bus should provide DC voltage within the range of 28 +/- 6 (Vdc), including ripple and normal transients as defined below, and power distribution losses due to switching, fusing, harness and connectors

- ## 3.2.5.5 Bus Undervoltage & Overvoltage Transients

  The Instrument should be able to survive without damage a power bus undervoltage or overvoltage condition occurring

Key Power Guidelines

- ## 3.2.5.18 Abnormal Operation Voltage Limits

  Under abnormal conditions the Instrument should survive, without permanent degradation, steady-state voltages (V) in the range of 0 to 50 Vdc

- ## 3.2.5.19 Power Source Impedance

  The Spacecraft power source impedance should be as indicated in Table 3.2-2

Maximum Source Impedance (Ohms)	Frequency
0.1	1 Hz to 1 kHz
1.0	1 KHz to 20 KHz
2.0	20 KHz to 100 KHz
20.0	100 KHz to 10 MHz

Table 3.2-2 Power Source Impedance

- ## 3.2.6.1 Instrument Turn-on Transient

  For turn-on, the transient current on any Power Feed bus should not exceed 100 percent (that is, two times the steady state current) of the maximum steady-state current and should not be greater than 50-msec surge duration. There is no turn-on transient restriction on the Survival Heater Bus

Key Power Guidelines

- ## 3.2.6.7 Instrument Turn-on Transient

  The peak voltage of transients generated on the Instrument side of the power relay caused by inductive effects of the load should not fall outside the -2 Vdc to +40 Vdc range

- ## 3.2.7 Power-Up & Power-Down

  When the spacecraft is powered via an external supply, the bus voltage will change from 0 to +28 volts within 2 seconds. The maximum rate of change should be less than 2 V/msec. When the battery is used to power the spacecraft, the bus voltage change from 0 to +28 volts will be a step function

- ## 3.2.8 Power-Up & Power-Down

  The Instrument should be designed for nominal and anomalous power-down sequences where the bus voltage change from +28 to 0 volts will be a step function

- ## 3.2.10.1 Instrument Operational Transients

  Operational transients that occur after initial turn-on should not exceed 125 percent of the peak operational current drawn during normal operation

Key Power Guidelines

- 3.2.10.2 Instrument Operational Transients

  The maximum duration of the transients should not exceed 50 msec

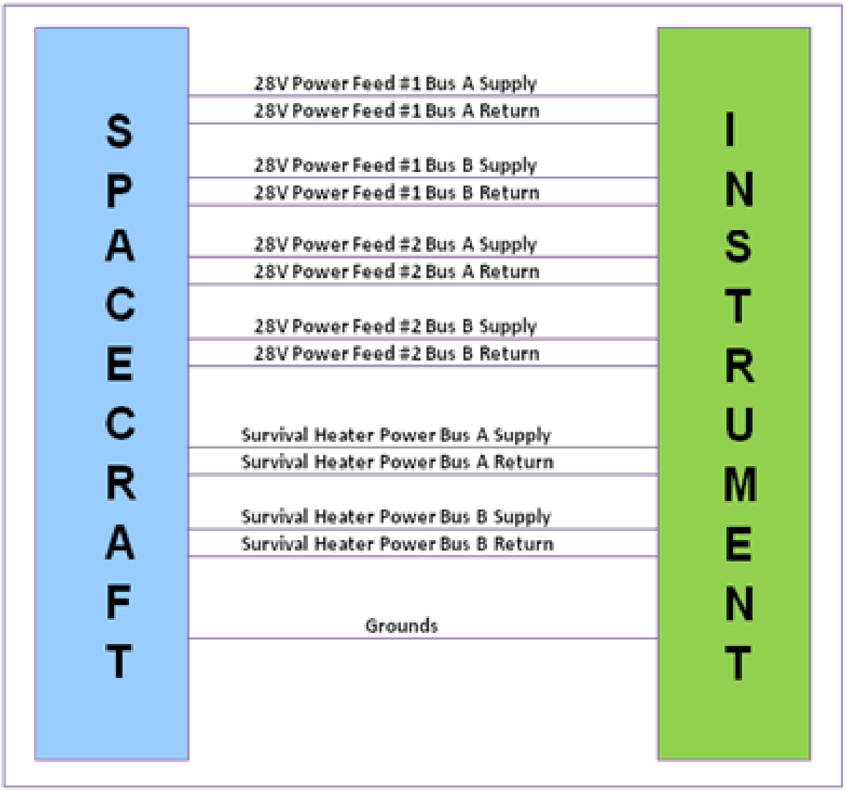
- 3.2.11.8 Instrument Fault Propagation

  The Instrument and S/C should not propagate a single fault occurring on either the “A” or “B” power interface circuit, on either side of the interface, to the redundant interface or Instrument

Key Power Guidelines

3.2.12 Power Connections

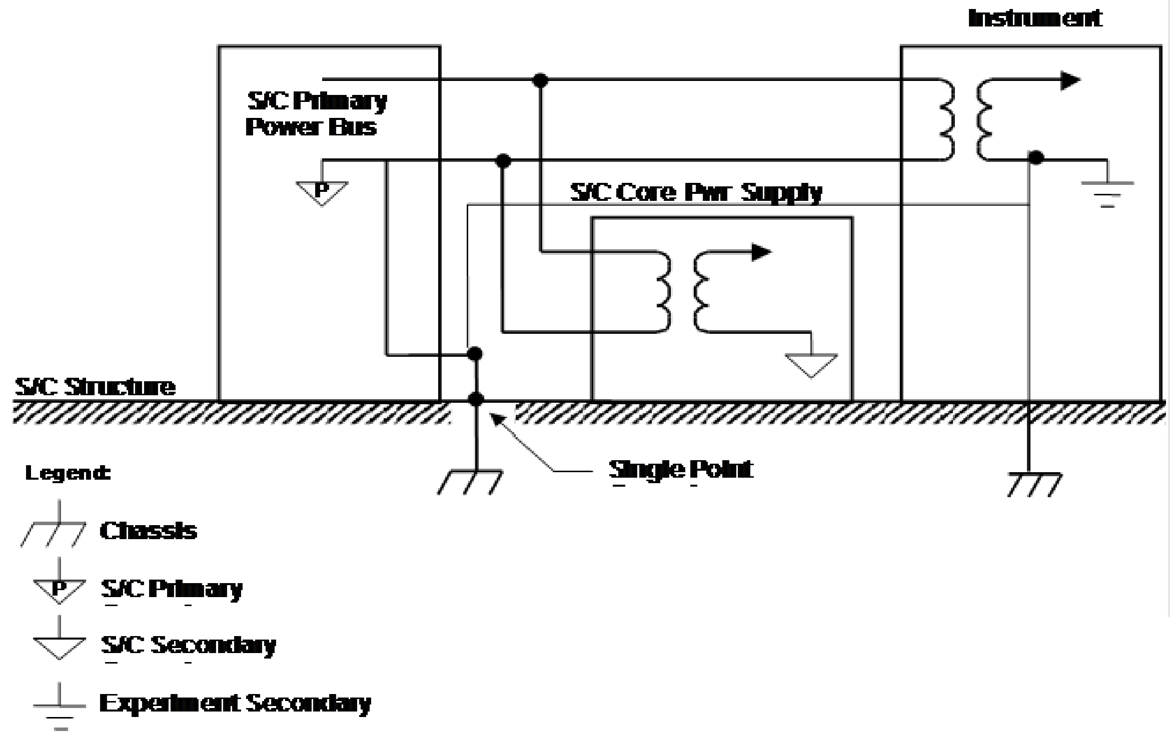
The Spacecraft should provide separate connections to redundant power sources to each Instrument as illustrated in Figure 3.2-2. These connections are designated as Power Feed Bus #1, Power Feed Bus #2 and the Survival Heater Power Bus. All buses have prime and redundant sides designated as power bus A and power bus B



Key Power Guidelines

- ## 3.3.1 Grounding Responsibility

To ensure that the Instrument grounding configuration will be compatible with the spacecraft, the Instrument grounds are required to be wired as described in the following sections. Instrument grounding should appear in the Electrical Interface drawing. Figure 3.3-1 shows the grounding scheme to be used between Instrument and spacecraft



Key Power Guidelines

- ## 3.3.3.4 Primary Power Isolation

  The 28 V primary power leads and returns should be isolated from signal and signal return, and chassis ground by > 1 MΩ (dc) when measured at the Instrument input

- ## 3.3.3.5 Secondary Power Isolation

  Secondary power should be isolated from 28 Vdc primary power by >1 MΩ (dc)

- ## 3.3.26 Component Grounding

  For electrical components that generate or are susceptible to electromagnetic energy, the chassis-to-structure bond should comply with MIL-STD-464A, with a maximum DC resistance of 10 mΩ across the mated surface

- ## 3.3.28 Component Grounding

  For non-EMI sensitive hardware, the chassis-to-structure bond should also comply with MIL-STD-464A

Key Power Guidelines

- ## 3.4.1 Harness Provider

  All Spacecraft and Spacecraft to Instrument harnessing should be provided by the Spacecraft Contractor

- ## 3.4.2 Harness Provider

  All Instrument harnessing should be provided by the Instrument

- ## 3.4.3 Harness Hardware Documentation

  Harnesses, connectors, ground straps, and associated service loops should be documented in the appropriate electrical Interface Control Drawing

- ## 3.4.4 Harness Wiring Requirements

  All requirements for harness construction, pin-to-pin wiring, cable type, etc. should be documented in the appropriate electrical Interface Control Drawing

Backup Materials

3.0 ELECTRICAL INTERFACE GUIDELINES

3.1 Electrical Interface Requirements

All requirements in Section 3 should be met at the electrical interface.

3.1.1 Electrical Interfaces

The electrical interfaces (see Figure 3.2-2) should include the following:
Operational Power Interface
Survival Heater Power Interface
Grounding Interface

3.1.2 Electrical Interface Definitions

3.1.2.1 Power Interface

There should be three redundant power buses feeding each Instrument. They are designated as the Power Feed #1, Power Feed #2 and the Survival Heater Power Feed.

3.1.2.2 Power Interface

Each power bus should have separate ground returns.

3.1.2.3 Power Bus

Each power bus should be redundant with independent sides designated as side Power Bus A and Power Bus B.

3.1.2.4 Power Bus

For power bus loads with current change greater than 2 amps, the rate of change of current should not exceed 500 mA/μsec.

3.1.2.5 Power Bus

Each power bus is redundant with primary and redundant power designated as Power Bus A and Power Bus B.

3.1.2.6 Survival Heater Power Bus

The Survival Heater Power Bus is a redundant power bus. Survival mode is a Spacecraft initiated mode in which operational power to the Instruments is
interrupted. Survival power is used only for heaters and associated passive control circuitry which maintain the Instrument at minimum turn-on temperature. Sides A and B are designated as Survival Heater Bus A and Survival Heater Bus B.

3.1.2.7 Survival Heater Power Bus

The survival heaters should be redundant

3.1.2.8 Survival Heater Power Bus

The survival heater power bus primary and redundant power should be electrically isolated from each other and from chassis

3.1.2.9 Survival Heater Power Bus

The survival heater power bus should have independent power returns. The survival power buses can be switched off by the spacecraft, but will normally be continuously powered during flight.

3.1.2.10 Survival Heater Power Bus

Instrument survival heater power should not exceed 60W EOL at worst-case low bus voltage. This power covers all of the heater power requirements required to maintain the Instrument at or above the non-operational survival temperature when the spacecraft is either in the LEOP portion of the mission, in Safe Mode, or performing an orbit correction maneuver. The spacecraft provides the switched power to the Instrument when required by the mission.

3.1.2.11 Survival Heater Power Bus

The Instrument design should prevent a stuck on condition of the survival heaters due to internal failures.

3.1.2.12 Survival Heater Power Bus

Survival power should be used within the Instrument only for resistive heaters (and associated thermal control device) which maintain the Instrument at minimum turn-on temperature when the main power bus is disconnected from the Instrument.

3.1.2.13 Survival Heater Power Bus

The system design should be such that having both primary and redundant survival heater circuits enabled does not violate any thermal or power requirement.

3.1.2.14 Survival Heater Power Bus

Survival heater power buses should be electrically isolated from each other, from other Instrument thermal control, from chassis, and have independent power returns.

3.1.2.15 Survival Heater Power Bus

The spacecraft should ensure that both the primary and redundant survival heater circuits are normally enabled on-orbit when an Instrument is off. However, even the survival heaters may be turned off in the event of an emergency where the survival of the spacecraft is in jeopardy.

3.1.2.16 Power Switching

The S/C should provide switched Primary and Redundant Operational Power to the Instrument.

3.1.2.17 Survival Heater Power Bus

The S/C should sequentially apply power to each survival heater power feed.

3.1.2.18 Survival Heater Power Bus

The S/C should provide switched Primary and Redundant Survival Power to the Instrument.

3.2 POWER SPECIFICATIONS

This section specifies the characteristics, connections, and control of the Spacecraft power provided to each Instrument as well as the requirements that each Instrument must meet at this interface. This section applies equally to the Power Buses and the Survival Heater Power Buses.

3.2.1 Instrument Power Harness

Instrument power harnesses should be appropriately sized to support the peak allocated power levels and both spacecraft and Instrument fusing.

3.2.2 Average Power Consumption

Average power is the total power averaged over 2 orbits.

3.2.3 Peak Power Consumption

Peak power is the maximum power averaged over any 10 msec period.

3.2.4 Allocation of Instrument Power

Each Instrument mode should be allocated peak NTE and average NTE power shown in Table 3.2-1.

MODE	PEAK (Watts)	Average (Watts)
OFF	0	0
SURVIVAL	60	30
ACTIVATION	400	200
SAFE	200	100
OPERATION	400	200

Table 3.2-1 Instrument Power Allocation

3.2.5 Voltage

3.2.5.1 Primary Instrument Voltage

The Spacecraft power bus should provide DC voltage within the range of 28 +/- 6 (Vdc), including ripple and normal transients as defined below, and power distribution losses due to switching, fusing, hamess and connectors.

3.2.5.2 Unannounced Removal of Power

Unannounced removal of power should not cause damage or degraded instrument performance

3.2.5.3 Energy Recovery Mode

In the event that the spacecraft battery state-of-charge falls below 50%, the spacecraft will power off the Instrument after appropriately placing the Instruments in a safe state and Instrument operation will not resume until the ground operators have determined it is safe to retum to normal operations.

3.2.5.4 Low Voltage Detection

A voltage excursion that causes the S/C Primary Power Bus to drop below 22V for a duration > four seconds, will constitute an under-voltage condition. In the event of an under-voltage condition, the S/C will shed various loads without delay, including the Instrument. A ground command should be required to re-power the load.

3.2.5.5 Bus Undervoltage and Overvoltage Transients

The Instrument should be able to survive without damage a power bus undervoltage or overvoltage condition occurring.

3.2.5.6 Bus Undervoltage and Overvoltage Transients

Derating factors should take into account the stresses that components are subjected to during periods of undervoltage or overvoltage, including conditions which arise during ground testing, while the bus voltage is slowly brought up to its nominal value.

3.2.5.7 Bus Undervoltage and Overvoltage Transients

The Instrument should not generate a spurious response that can cause equipment damage or otherwise be detrimental to the spacecraft operation during bus voltage variation, either up or down, at ramp rates below the limits specified in the sections below, and over the full range from zero to maximum bus voltage.

3.2.5.8 Normal Transients

The bus voltage should not vary by more than 1.0V during a 5 A in 1 msec step load change (+1V during a -5A step, -1V during a +5A step).

3.2.5.9 Normal Transients

At entry into eclipse, the bus voltage should not vary by more than –0.2 volts per second, with a maximum voltage change of –6 volts, not to go below 22V at the Instrument input.

3.2.5.10 Normal Transients

At exit from eclipse, the bus voltage should not change by more than 0.2 volts per second, with a maximum change in sunlight of 6 volts, not to exceed 34V at the Instrument input.

3.2.5.11 Abnormal Transients

An undervoltage event should be defined as a transient decrease in voltage on the +28V bus to no less than +10V, maintaining the decreased voltage for no more than 10 msec, and returning to its previous voltage in less than 200 msec.

3.2.5.12 Abnormal Transients

All spacecraft components should be designed such that overstress does not occur to the unit during the undervoltage.

3.2.5.13 Abnormal Transients

Additionally, units which shut-off during an undervoltage should return to a nominal power-up state at the end of the transient.

3.2.5.14 Abnormal Transients

An undervoltage event should be defined as a transient increase in voltage on the +28V bus to no greater than +40V, maintaining the increased voltage for no more than 10 msec, and returning to its previous voltage in less than 200 msec.

3.2.5.15 Abnormal Transients

All spacecraft components should be designed to operate through the overvoltage transient, with no degradation in performance, and no overstress of electrical components.

3.2.5.16 Input Ripple

The ripple voltage on the power bus will be less than 5% peak-to-peak of line voltage (28 V) over all frequency ranges from 1 Hz to 10 MHz. The output ripple of the power subsystem alone shouldl be less than 250 mV peak-to-peak over the frequency range of 1 Hz to 10 MHz.

3.2.5.17 Input Ripple

The output ripple of the power subsystem including the effect of all loads should not exceed 500 mV peak to peak over the frequency range of 1 Hz to 10 MHz.

3.2.5.18 Abnormal Operation Steady-State Voltage Limits

Under abnormal conditions the Instrument should survive, without permanent degradation, steady-state voltages (V) in the range of 0 to 50 Vdc.

3.2.5.19 Power Source Impedance

The Spacecraft power source impedance should be as indicated in Table 3.2-2.

Maximum Source Impedance (Ohms)	Frequency
0.1	1 Hz to 1 kHz
1.0	1 KHz to 20 KHz
2.0	20 KHz to 100 KHz
20.0	100 KHz to 10 MHz

Table 3.2-2 Power Source Impedance

3.2.6 Current

3.2.6.1 Instrument Turn-on Transients

For turn-on, the transient current on any Power Feed bus should not exceed 100 percent (that is, two times the steady state current) of the maximum steady-state current and should not be greater than 50-msec surge duration. There is no turn-on transient restriction on the Survival Heater Bus.

3.2.6.2 Instrument Turn-on Transients

During initial turn-on of the Instrument, or as a result of a change in operating mode, the operational inrush current drawn by the DC power input should meet the following requirements (Figure 4.2-1).

3.2.6.3 Instrument Turn-on Transients

After application of +28V power at t_0, the initial inrush (charging) current due to distributed capacitance, EMI filters, etc., should be completed in 10 μsec with its peak no greater than 10 Amps.

3.2.6.4 Instrument Turn-on Transients

The rate of change of inrush current after the initial application of +28V power should not exceed 20 mA/μsec.

3.2.6.5 Instrument Turn-on Transients

After 10μsec, the transient current peak should not exceed three times the maximum steady state current.

3.2.6.6 Instrument Turn-on Transients

Steady state operation should be attained within 50 msec from turn-on or the start operating mode change, except for motors.

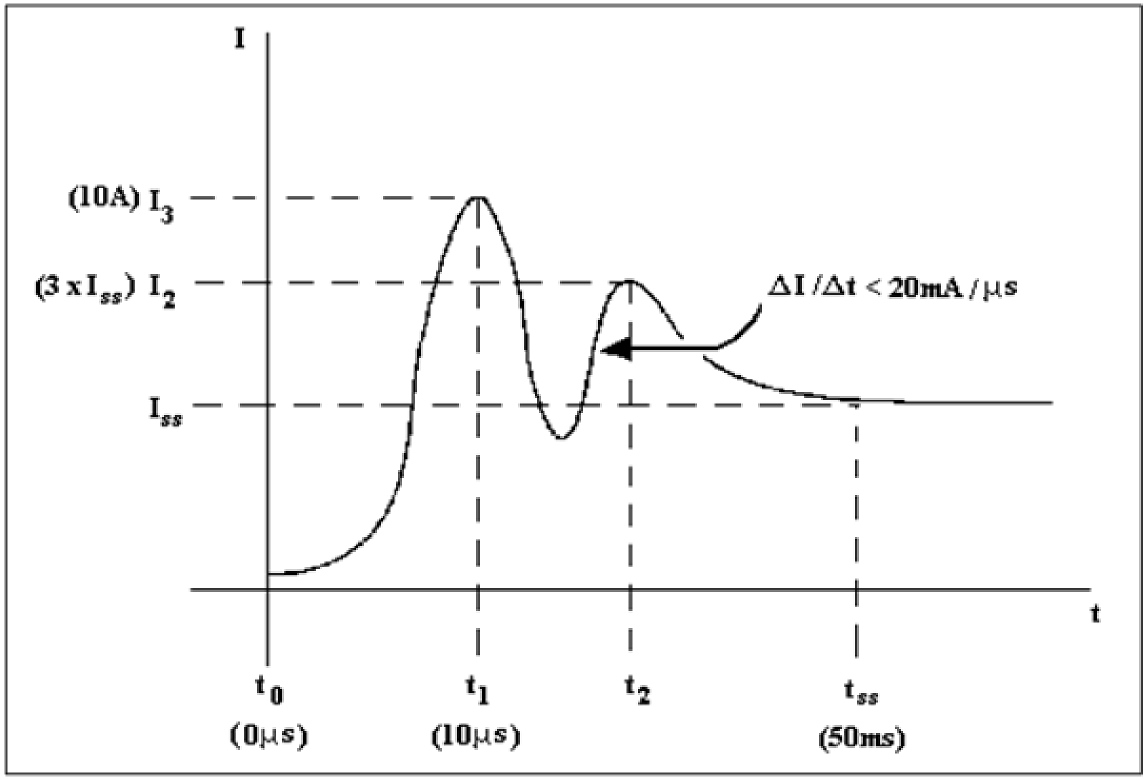


Figure 3.2-1 Maximum Inrush Current

3.2.6.7 Instrument Turn-off Transients

The peak voltage of transients generated on the Instrument side of the power relay caused by inductive effects of the load should not fall outside the -2 Vdc to +40 Vdc range.

3.2.6.8 Instrument Turn-off Transients

The Instruments should use suppression devices such as diodes
across all filter inductors, relay coils, or other energy sources, which could induce transients on the power lines during turn-off.

3.2.6.9 Instrument Turn-off Transients

The suppression devices should be located at the source of the inductive transients.

3.2.7 Power-Up and Power-Down

When the spacecraft is powered via an external supply, the bus voltage will change from 0 to +28 volts within 2 seconds. The maximum rate of change should be less than 2 V/msec. When the battery is used to power the spacecraft, the bus voltage change from 0 to +28 volts will be a step function.

3.2.8 Power-Up and Power-Down

The Instrument should be designed for nominal and anomalous power-down sequences where the bus voltage change from +28 to 0 volts will be a step function.

3.2.9 Load Current Ripple

The peak-to-peak amplitude of steady state load current generated by the Instrument should not exceed 2% of the maximum average steady state current drawn by the Instrument.

3.2.10 Load Current Ripple

The load current ripple due to rpm mode changes should not exceed 2 times the steady state current during the period of the motor spin-up or spin-down.

3.2.10.1 Instrument Operational Transients

Operational transients that occur after initial turn-on should not exceed 125 percent of the peak operational current drawn during normal operation.

3.2.10.2 Instrument Operational Transients

The maximum duration of the transients should not exceed 50 msec.

3.2.10.3 Instrument Reflected Ripple Current

The peak-to-peak load current ripple generated by the Instrument should not exceed 25 percent of the average current on any Power Feed bus.

3.2.11 Overcurrent Protection

The Spacecraft should provide protection of the Spacecraft power system by providing overcurrent protection devices on each Instrument power connection.

3.2.11.1 Overcurrent Protection Device Size

The sizes of the overcurrent protection devices for each Instrument should be based on the values presented in TBR

3.2.11.2 Overcurrent Protection

Turn-on, operational, and turn-off transients should be considered in the analysis.

3.2.11.3 Overcurrent Protection

Harness wire sizes should be consistent with overcurrent protection device sizes.

3.2.11.4 Overcurrent Protection

Power supply output capability should be adequate to clear a fuse or other overcurrent protection device.

3.2.11.5 Overcurrent Protection Device Size Documentation

The agreed-upon sizes and characteristics of the overcurrent protection devices should be documented in the Spacecraft to Instrument ICD.

3.2.11.6 Instrument Internal Overcurrent Protection

All Instrument internal overcurrent protection devices should be accessible at the Spacecraft integration level without the disassembly of the Instrument.

3.2.11.7 Instrument Internal Overcurrent Protection

The Instrument should not have internal fuses

3.2.11.8 Instrument Fault Propagation Protection

The Instrument and S/C should not propagate a single fault occurring on either the “A” or “B” power interface circuit, on either side of the interface, to the redundant interface or Instrument.

3.2.12 Power Connections

The Spacecraft should provide separate connections to redundant power sources to each Instrument as illustrated in Figure 3.2-2. These connections are designated as Power Feed Bus #1, Power Feed Bus #2 and the Survival Heater Power Bus. All buses have prime and redundant sides designated as power bus A and power bus B.

3.2.12.1 *Instrument High-Voltage Restriction*

To allow ambient testing, Instrument high-voltage supplies should be capable of being operated at atmospheric pressure.

3.2.12.2 *Instrument High-Voltage Restriction*

The output of each Instrument's high-voltage supply should be current limited to prevent the spacecraft and other Instruments from being damaged by the supply's discharge.

3.2.12.3 *Instrument High-Voltage Restriction*

If an Instrument high-voltage supply cannot be operated at atmospheric pressure, it should be appropriately disabled by manual means to allow ambient testing.

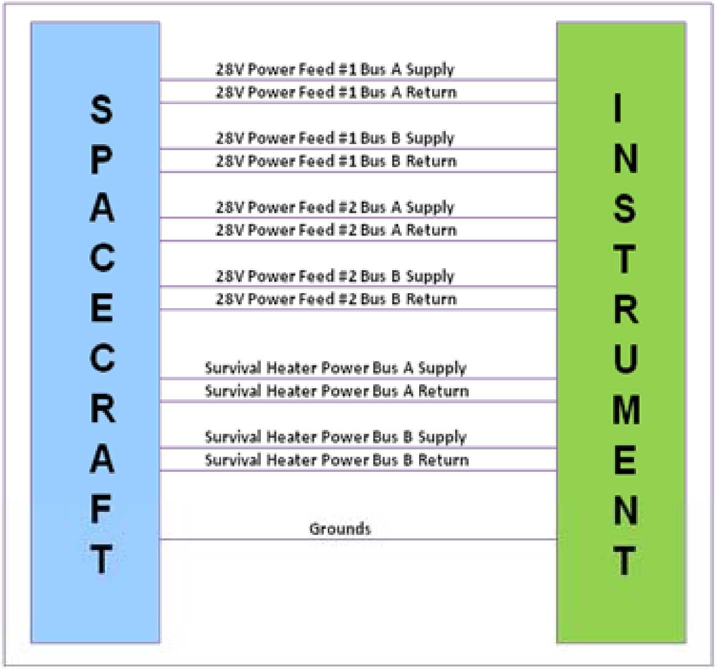


Figure 3.2-2 Spacecraft-Instrument Electrical Interface

3.3 Grounds, Returns and References

3.3.1 Grounding Responsibility

To ensure that the Instrument grounding configuration will be compatible with the spacecraft, the Instrument grounds are required to be wired as described in the following sections. Instrument grounding should appear in the Electrical Interface drawing. Figure 3.3-1 shows the grounding scheme to be used between Instrument and spacecraft.

3.3.2 Power Harnessing

3.3.2.1 Power Routing and Shielding

The delivery of Instrument power by the Spacecraft Contractor should be thru twisted conductor (pair, quad, etc.) cables with both power and return leads enclosed by an electrical shield.

3.3.3 Power Leads and Returns

Each Instrument primary power service should have a distinct and isolated return.

3.3.3.1 Power Shield Bonding

The Instrument should provide the capability to form a low impedance electrical path (a bond) between the power conductor shield and the Instrument chassis via the connector shell at the Instrument primary power input.

3.3.3.2 Isolation

Isolation requirements from primary power to chassis and primary power to secondary power should be adhered to at each primary power service.

3.3.3.3 Power Input Isolation

At the Instrument interface, isolation should exist between the six power buses shown in Figure 3.2-2.

3.3.3.4 Primary Power Isolation

The 28 V primary power leads and returns should be isolated from signal and signal return, and chassis ground by > 1 MΩ (dc) when measured at the Instrument input.

3.3.3.5 Secondary Power Isolation

Secondary power should be isolated from 28 Vdc primary power by >1 MΩ (dc).

3.3.4 Power Reference

The Signal Reference Plane is the Spacecraft conducting plate or other structure to which all ground planes are connected. The Primary Power Reference is the reference point for Spacecraft voltage control. The Spacecraft should connect all power returns to the Signal Reference Plane only at the Spacecraft Primary Power Reference Point. (See Figure 3.3-1).

3.3.4.1 Secondary Power Return

Secondary power circuits should provide current return leads from each Instrument component which utilizes secondary power.

3.3.4.2 Secondary Power Return

These secondary power returns should be connected at a single point referred to as the Secondary Power Reference.

3.3.4.3 Secondary Power Reference

Secondary Power Reference should be bonded to the chassis ground.

3.3.4.4 Isolated Secondary Referencing

If isolated secondary power is used within the Instrument, the isolated returns should not be left floating.

3.3.4.5 Isolated Secondary Referencing

In this instance isolated returns should be referenced to chassis ground through a resistive impedance whose value is selected by the Instrument provider.

3.3.4.6 Signal Reference

The Instrument should be designed such that the Secondary Power Reference and the signal reference for Spacecraft interface circuits are electrically the same point.

3.3.4.7 Signal Reference Connectivity

All Instrument signal references for the Spacecraft interface circuits should be electrically connected within the Instrument.

3.3.4.8 Signal Reference Constraints

Neither signal nor chassis ground reference points should be used as power conductors.

3.3.5 Chassis Ground

3.3.5.1 Instrument Ground Plane

The Spacecraft Contractor should provide a common electrically conductive ground plane to which all Instrument chassis should be electrically connected.

3.3.5.2 Component Grounding

3.3.5.3 Component Ground Location

The Instrument should provide a designated chassis ground terminal on each component, as close to all electrical connectors as possible, for connection to the ground plane.

3.3.5.4 Component Ground Connection

The Spacecraft Contractor should electrically bond the chassis ground terminal of each Instrument component to the Instrument ground plane.

3.3.5.5 Component Bonding Straps

Where direct bonding is not possible and/or movable metal-to-metal joints are present, bonding straps should be used.

3.3.5.6 Connector Grounding

The Instrument should electrically bond all Instrument interface connector shells to the Instrument chassis.

3.3.5.7 Chassis Ground Current

Instruments should not use chassis ground to conduct power and signal currents under normal conditions. Only fault and leakage currents should be conducted through chassis grounds.

3.3.5.8 External Ground Tie Point

Each Instrument should identify an external chassis ground tie point to be used for external connections while the Instrument is being moved.

3.3.5.9 External Ground Tie Point

This should be documented in the Electrical Interface Control Drawing.

3.3.6 Signal Reference Plane

3.3.6.1 Instrument Ground Plane Connection

The Instrument Ground Plane is the electrically conductive surface to which all Instrument components are electrically connected (bonded). The Spacecraft Contractor should electrically connect the Instrument ground plane to the Signal Reference Plane.

3.3.7 Thermal Blanket Grounding

3.3.7.1 Thermal Blanket Layer Interconnection

All thermal insulation blankets should be designed with metallized and conductive layers electrically interconnected such that the resistance between layers is $< 10\ \Omega$ (dc).

3.3.7.2 Thermal Blanket Chassis Grounding

Thermal insulation blankets should be connected to chassis ground with a resistance of less than 10 Ω (dc).

3.3.8 Grounding of External Surfaces

To prevent electrostatic charging, external insulating materials and surface finishes should have a surface resistivity of less than or equal to 1×10^9 ohms per Square.

3.3.9 Grounding of External Surfaces

The following requirements in Table 3.3-1 should apply to surface coatings (e.g., conductive paints, thermal protection):

Description	Maximum Surface Resistivity (ohms per Square)
Surface Coating over Dielectric	1×10^9
Surface Coating over Metal or Conductive Composite	$\frac{1 \times 10^9}{\text{coating thickness (cm)}}$
Thin Film Material	1×10^{12}

Table 3.3-1 Surface Resistivity

3.3.10 Grounding of External Surfaces

The resistance of the connection between the conductive paint and the basic structure should be less than 1×10^5 Ω when dry.

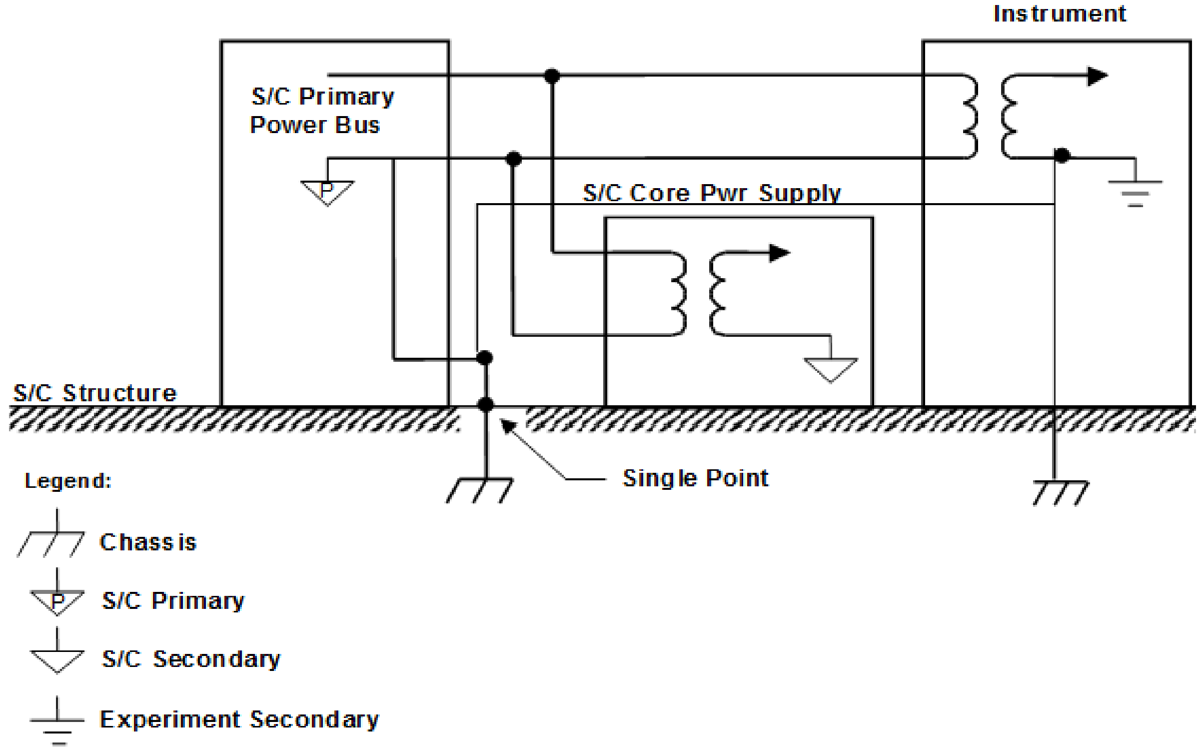


Figure 3.3-1 Spacecraft Power Grounding Scheme

3.3.11 Grounding

In each unit before connection to the harness, a DC isolation of at least 1 MΩ should exist from each line of primary power to each line of secondary power, and between all grounds which are not common.

3.3.12 Grounding

AC isolation should be at least 0.5 µF.

3.3.13 Grounding

Grounds which are common should have a measured DC resistance of less than 10 mΩ between connector pins.

3.3.14 Power Ground

Power return conductors should be provided for all current drawn from the +28 V Power Bus.

3.3.15 Power Ground

Power returns should not be grounded to chassis internal to any loads.

3.3.16 Power Ground

Power return-to-chassis isolation should be at least 1 MΩ.

3.3.17 Signal Ground

Signal ground should be the zero reference voltage for the secondary side of all DC/DC converters

3.3.18 Signal Ground

Signal ground should be the ground reference for all telemetry circuits.

3.3.19 Signal Ground

A signal ground wire should be connected from each component to the spacecraft Single Point Ground.

3.3.20 Signal Ground

Signal ground-to-chassis continuity should be less than 10 mΩ.

3.3.21 Signal Ground

Each connector should have at least two separate pins for analog and digital signal return.

3.3.22 Shielding Requirements

Multi-point grounding should be provided if necessary for shielding connections. The reference for shields should be chassis ground.

3.3.23 Component Grounding

The Instrument component housings supporting or containing electrical/electronic assemblies should be electrically bonded to preclude electrostatic discharge from causing inadvertent initiation of deployments or damage to sensitive electronics.

3.3.24 Component Grounding

In general, Instrument housings should be grounded to the satellite structure via the component mounting feet or via the baseplate together with an electrically and thermally conductive adhesive.

3.3.25 Component Grounding

Any electrically nonconductive anticorrosion finish should be removed from the joint faces before grounding. Instrument may be grounded to the satellite structure via an external ground wire for the following reasons:
Component was not designed for grounding via the mounting feet.
Component requires thermal isolation from its mounting surface.
Component requires special, non-conductive, shock absorbing mounts.
Component employs electrically non-conductive adhesive between baseplate and panel.

3.3.26 Component Grounding

For electrical components that generate or are susceptible to electromagnetic energy, the chassis-to-structure bond should comply with MIL-STD-464A, with a maximum DC resistance of 10 mΩ across the mated surface.

3.3.27 Component Grounding

The total DC resistance between Instrument housings and single point ground should not exceed 10 mΩ multiplied by the number of mechanical interfaces between the housing and S/C single-point ground.

3.3.28 Component Grounding

For non-EMI sensitive hardware, the chassis-to-structure bond should also comply with MIL-STD-464A.

3.3.29 Component Grounding

Interfaces between all isolated conducting items (except antenna) which are external to the vehicle and have any linear dimension greater than 3 inches, or otherwise are subject to frictional charging, should have secure connections to the spacecraft structure.

3.3.30 Component Grounding

The resistance of the connection should be less than 1 Ω.

3.3.31 Component Grounding

All metallic subchassis, chassis, and enclosures of the Instrument, including all connector shells and other fittings, should be considered as electrical extensions of the ground plane.

3.4 Harness

3.4.1 Harness Provider

All Spacecraft and Spacecraft to Instrument harnessing should be provided by the Spacecraft Contractor.

3.4.2 Harness Provider

All Instrument harnessing should be provided by the Instrument.

3.4.3 Harness Hardware Documentation

Harnesses, connectors, ground straps, and associated service loops should be documented in the appropriate electrical Interface Control Drawing.

3.4.4 Harness Wiring Requirements

All requirements for harness construction, pin-to-pin wiring, cable type, etc. should be documented in the appropriate electrical Interface Control Drawing.

3.4.5 Tie Points

3.4.5.1 Tie Point Locations and Provider

The locations of harness tie points should be based on an agreement between the instrument provider and the spacecraft bus provider.

3.4.5.2 Tie Point Documentation

The locations of the tie points should be documented in the appropriate electrical Interface Control Drawing.

3.4.6 Connectors

The following mechanical requirements should be followed for all interface connectors.

3.4.7 Connectors

Connector selection should consider materials outgassing, EMI and reliability.

3.4.8 Connectors

Connectors should be low magnetic content material.

3.4.9 Connectors

Circular connectors should feature scoop proof design and include grounding fingers.

3.4.10 Connectors

Throughout all hardware development, test, and integration phases, connector savers should be used to preserve the mating life of component flight connectors.

3.4.11 Connectors

The instrument provider should deliver connector saver adapters with high reliability connectors to minimize mating and demating operations with the flight connectors during integration and test.

3.4.11.1 Connector Usage and Pin Assignments

Separate harness interface connectors should be provided on all components for each of the following functions:
+28 volt bus power and return
telemetry and command signals with returns
deployment actuation power and return (where applicable)

3.4.11.2 Electrical Connector Constraints

The connector half that has the primary power source should use socket contacts.

3.4.11.3 Connector Usage and Pin Assignments

Telemetry return and relay driver return pins should be assigned on the same connector(s) as the command and telemetry signals.

3.4.11.4 Connector Usage and Pin Assignments

Harness side power connectors and all box/bracket mounted connectors supplying power to other components should have female contacts (connector does not have exposed contacts).

3.4.11.5 Connector Usage and Pin Assignments

The connector contact assignment should consider effects of electromagnetic coupling.

3.4.11.6 Connector Usage and Pin Assignments

Incompatible functions should be physically separated.

3.4.11.7 Connector Usage and Pin Assignments

If triaxial cables are used for signals, the inner shield signal ground should be assigned to the pin adjacent to the signal.

3.4.11.8 Connector Usage and Pin Assignments

Instrument and Spacecraft should derate electrical connectors using MIL-HDBK-1547A as a guide.

3.4.11.9 Connector Clearance

At least 50 mm of clearance should be provided around the outside of mated connectors.

3.4.11.10 Connector Clearance

The mated connectors should be accessible on the spacecraft without removal of adjacent Instruments.

3.4.11.11 Connector Location and Types

The spacecraft bus provider should define any areas in which connectors are not allowed as a result of satellite configuration constraints.

3.4.11.12 Connector Location and Types

Connectors should be mounted to assure straight and free engagement of the contacts.

3.4.11.13 Connector Location and Types

Connectors should be spaced far enough apart so that the coupling device can be held firmly either by hand or plug removal tool for connecting and disconnecting.

3.4.11.14 Connector Location and Types

Connector locations, orientations, and keyway locations should be identified in the appropriate Electrical Interface Control Drawing.

3.4.11.15 Connector Type

Connectors should be space-flight qualified.

3.4.11.16 Connector Size and Conductor Gauge

Connector size and conductor gauge should be identified in the appropriate Electrical Interface Control Drawing.

3.4.11.17 Connector Pin Out

Connectors should utilize the conductor supply and return pin outs defined in the appropriate Electrical Interface Control Drawing.

3.4.11.18 Connector Provision

The instrument provider should furnish all instrument mating connectors (Socket Side) to the spacecraft bus provider.

3.4.11.19 Connector Conductor Size and Type

Connector conductor size and should be identified in the appropriate Electrical Interface Control Drawing.

3.4.12 Prevention of Mismating

Alternately, where adjacent signal connectors are the same shell sizes, alternate gender insert/contacts should be utilized.

3.4.13 Prevention of Mismating

Adjacent rectangular "D" shell connectors of the same size and gender should be rotated 180° from each other.

3.4.14 Prevention of Mismating

When receptacles of similar configuration are in close proximity, the mating plugs and receptacles should be suitably marked or coded to indicate clearly the mating connections.

3.4.15 Prevention of Mismating

All other rectangular and circular connectors should be positively keyed to prevent incorrect connections with other accessible connector plugs or receptacles.

3.4.16 Prevention of Mismating

Alternative insert position or shell keyway polarizing should be specified to prevent incorrect connections with other adjacent connector plugs or receptacles of the same shell size. In most cases, normal (N) insert or shell key polarized items are to be chosen for ease of procurement and logistic support.

3.4.17 Prevention of Mismating

Mismating of connectors should be avoided by ensuring that physically adjacent shells are different sizes.

3.4.18 Keying

Connectors should be different sizes, different types, or uniquely keyed to prevent improper connection.

3.4.18.1 Power Connector Keying

The instrument power connectors should be keyed as defined in the appropriate Electrical Interface Control Drawing.

3.4.19 Interface Connector Provider

3.4.19.1 Harness Connectors

All connectors attached to Spacecraft harnesses should be provided by the Spacecraft Contractor.

3.4.19.2 Harness Connectors

The spacecraft bus provider should be responsible for verifying that the instrument connectors and spacecraft connectors are compatible.

3.4.19.3 Connector Types

All connectors to be used by the Instrument should be selected from the GSFC Preferred Parts List (PPL).

3.4.19.4 Connector Type Documentation

The connector types used by the Instrument should be documented in the appropriate Electrical Interface Control Drawing.

3.4.20 Flight Plugs

3.4.20.1 Flight Plug Installation

Flight plugs requiring installation prior to launch should be capable of being installed at the Spacecraft level.

3.4.20.2 Flight Plug Responsibility

Flight plugs, if required, should be provided by the instrument provider.

3.4.20.3 Flight Plug Documentation

Flight plugs and their locations should be documented in the appropriate Electrical Interface Control Drawing.

3.4.20.4 Connector Protective Covers

Captive covers should be provided by the instrument provider for all instrument connectors which are not mated to harnesses or flight plugs.

3.4.20.5 Connector Protective Covers

The instrument provider should supply the protective covers for all instrument provided connectors.

3.4.20.6 Connector Protective Covers

Non-flight covers should be marked in red.

3.4.20.7 Connector Protective Covers

Covers remaining on the satellite for flight should be flight quality.

3.4.20.8 Connector Protective Covers

Connectors requiring servicing at the launch site should have captive covers and hardware.

3.4.20.9 Connector Protective Covers

Circular connector captive covers should be connected to the component with a flexible coupling to prevent separation of the cover from the component during cover removal.

3.4.20.10 Connector Protective Covers

All screws used to secure connector covers to "D" type connectors should be designed to be held captive to the cover.

3.4.21 Test Connectors

3.4.21.1 Test Connector Location and Types

The location of test connectors and test coupler ports should maximize unrestricted access so that connections can be mated and demated without affecting the qualification status of the spacecraft and with minimum risk to the surrounding flight hardware.

3.4.21.2 Test Connector Location and Types

Test connector and coupler port test access should be maintained throughout the buildup of the spacecraft and instrument, up to and including the final launch configuration.

3.4.21.3 Test Connector Accessibility

Test connectors should be accessible at the integrated Spacecraft level without disassembly.

3.4.21.4 Test Connector Documentation

Test connectors and their locations should be documented in the the appropriate Electrical Interface Control Drawing.

3.4.21.5 Breakout Boxes

Instrument test tees and breakout boxes should be provided by the Instrument Provider.

3.4.22 Buffer Connectors and Connector Savers

3.4.22.1 Connector Saver Utilization

Instrument buffer connectors and connector savers should be utilized prior to Spacecraft-level system tests.

3.4.22.2 Connector Saver Provider

Instrument buffer connectors and connector savers should be provided by the Instrument Provider.