[redacted] The GLAS -Z geodetic pierce point shall be controlled to within 30 m, or 50 μrad (1σ), of a reference groundtrack, The GLAS XZ plane (during low orbit beta angles ) or YZ plane (during high orbit beta angles) shall be controlled to within ±2° (3σ) of the orbit plane.

ICESat requires off-axis pointing. ICESat shall provide capability for unlimited operation in any commanded attitude with the GLAS +-Z axis within 5° of nadir. Commanded attitudes within 30° of nadir shall be accommodated, accepting minor degradations in power and science link availability and possible bus star tracker outages. ICESat shall accomplish 5° repointings about the pitch or roll axes within 30 seconds, including settling time. As a derived requirement, ICESat shall accomplish a 90° yaw within [redacted] including settling time.

As a derived requirement, ICESat shall provide attitude knowledge of [redacted] using only sun sensors and magnetometers, to allow for situations when the bus star tracker is blinded and the GLAS attitude sensors are unavailable. These sensors shall also allow attitude stabilization from a tumble rate of [redacted] about any axis, without use of the GLAS attitude sensors. In general, the Observatory will be oriented to make bus star tracker and instrument attitude data available during thruster burns, but pointing and burn planning shall not require its availability.

ICESat shall use ~~a~~ GFE TurboRogue GPS receiver antennas and cables for real-time navigation and to provide raw data in telemetry for ground post-processing with Gipsy/Oasis or similar algorithms. ~~Ball assumes that these interfaces are electrically and functionally identical to those used on the RS2000's Motorola Viceroy units.~~ Real-time navigation shall use filtered TurboRogue outputs at a 30-sec update rate and an on-board orbit propagator. Real-time position knowledge shall be better than 50 m (1σ), and updated once per second. All TurboRogue navigation outputs will be stored on the science data recorder, using a pair of 19.2 kbps data streams.

### 3.2.5 Command and Data Handling Performance

The Command and Data Handling (C&DH) subsystem provides the command handling, data collection and data storage functions. It consists of the [redacted] The system can input, decode, and process real-time and stored commands. It also can collect and store engineering status information from throughout the spacecraft, format this information into a serial data stream, and output it to the transmitter for transmission to the ground.

The SCC uses a processor card and multiple interface cards. An [redacted] The SCC can be reprogrammed on orbit through software updates during normal operations. Multiple levels of safe modes protect entire spacecraft from errors including software errors. Command and telemetry processing is handled [redacted].

Storage for payload data is provided by the SSR [redacted].

GLAS instrument commands and program uploads shall be sent in real-time or via stored commands over the 1553B interface.

Multiple data streams shall be ingested and stored on the science SSR during all normal operations. The SSR shall accommodate ~~50,000~~ 60,000 seconds of data, ingesting the following data streams:

- GLAS science data (~230 kbps)
- TurboRogue GPS receiver outputs and real-time position estimates (~40 kbps)
- GLAS attitude reference system output and instrument monitoring and control data (~75 kbps)
- Spacecraft bus star tracker [redacted]

Instrument health and welfare data shall be ingested and stored [redacted]. The [redacted] shall accommodate 60,000 seconds of data, including both spacecraft bus and instrument data. Instrument health and welfare data telemetry points will be sampled every 32 seconds, and shall consist of:

- 8 (TBR) eight-bit analog outputs
- 8 (TBR) bi-level outputs
- 12 thermistor outputs
- 1 MIL-STD-1553B interface

This health and welfare data shall also be available in real-time during engineering station contacts, sampled every 4 seconds.

ICESat shall be capable of providing commands in real-time and by stored ground commands. Stored commands shall be capable of execution 24 hours after upload, and shall have a 0.2-second resolution on execution start time. Payload commands shall be correlated with the 1-Hz timing pulse and the time signals received at 30-second intervals by the TurboRogue GPS receiver. Mode commands shall be transferred using the 1553 data bus. 14 (TBR) high-level and 10 (TBR) low-level discrete commands shall be allocated to GLAS functions.

### 3.2.5.1 Commanding

The command modulation ground equipment is capable of transmitting commands to the spacecraft in S-Band using 60 bit command words at rates of 250, 1000, or 2000 bps. The command modulation equipment provides all accommodations to receive command data from the STOC, and format, modulate, and transfer the signal to the antenna subsystem for transmission.

The spacecraft stored command capability is currently [redacted] commands but can easily be expanded if necessary [redacted].

Any automatic on-board function may be overridden and inhibited by the operator. All platform attitude control subsystem commands normally executed autonomously during normal operation can be manually executed by ground command under all normal operations and predictable abnormal operations.

Should an under-voltage condition occur, the operation of the command system is assured, along with other critical housekeeping functions. No permanent damage will result from a recoverable under-voltage condition.

### 3.2.5.2 Telemetry

Sufficient telemetry points are provided to accurately evaluate all subsystems' status, health, and performance.

The ICESat system uses a [redacted] Some telemetry, such as component temperatures, are only telemetered once every 32 s.

There are [redacted] of engineering data telemetry storage available. Impacts to increase the storage size, if required, are minimal.

For telemetry operations during anomalies, the system operates during under-voltage conditions as do other critical housekeeping functions. No permanent degradation in the telemetry subsystem will result from recoverable under-voltage conditions.

The accuracy of analog signals is [redacted] Accuracy of temperatures is [redacted] Higher accuracies and resolutions can be achieved with minimal electronics parts changes, if required.

The telemetry system is fully redundant [redacted] All telemetered data can be provided by either the prime or redundant branch.

Telemetry is transmitted via an omnidirectional antenna on the spacecraft. Reception is possible whenever the spacecraft is more than 5 deg above the horizon for the ground station.

### 3.2.6 Communications Performance

The spacecraft RF uplink/downlink uses a STDN-compatible transponder for narrow band data transmission and reception, ensuring compatibility with National Aeronautics and Space Administration (NASA) ground stations. The uplink command data rate of 2000 bps has a worst-case link margin of 15.3 dB for the baseline ICESat orbit. Stored engineering data is downlinked at [redacted] Real-time data is downlinked [redacted] A patch antenna configuration provides spherical coverage (90% of the spacecraft).

The 6-W ICESat X-band transmitter, high gain antenna, and SSR interface provides the high rate downlink of payload data. The system supports a dual-channel (I&Q) total downlink rate of 40 Mbps.

Ground station performance assumes a G/T value of 23 or 35 dB/k, and an EIRP of 92 dBm. These requirements are consistent with the ICESat ground stations.

Cross-strapping of redundant components provides a system capable of exceeding the 5-year life expectancy with no credible single-point failures.

The ICESat Observatory shall be capable of communicating with the EOSDIS Polar Ground Stations (EPGS) in compliance with the CCSDS format, at least four times per day as defined in the ICESat Mission Operations Concept, given the orbit elements specified in Section 3.1.2. The ground station characteristics are shown in Figure 3-8.6. General communications performance, downlink, and uplink characterizations follow.

| Characteristic | Wallops (WPSA) | EPGS | |
|---|---|---|---|
| | | Platabergat (PLAT) | Poker Flat (PKFT) |
| Latitude | 39.9270 deg | 78.1300 deg | 65.1170 deg |
| Longitude | 284.525 deg | 14.400 deg | 212.5380 deg |
| Altitude | 5 m | 430 m | 420 m |
| No. of Antenna | One | Two | One |
| Antenna Diameter | 11.3 m | 11.3 m | 11.3 m |
| Autotrack through Zenith | Yes | Yes | Yes |
| Station Mask | 5 deg | 5 - 20 deg (TBR) | 5 - 20 deg (TBR) |
| X Band G/T | None | 35 dB/K | 35 dB/K |
| S Band EIRP | 96 dBW | 96 dBW | 96 dBW |
| S Band G/T | 23 dB/K | 23 dB/K | 23 dB/K |
| $P_s$ | 0.99 | 0.99 | 0.99 |

***Figure 3-8.6.** ICESat Ground Station Characteristics*

For general communications performance, the spacecraft shall:

- Incorporate zenith and nadir omni antenna coverage for S-Band communication, to assure command and telemetry integrity with the ground if the Observatory tumbles.
- Be capable of simultaneously transmitting and receiving any combination of S-Band downlink, X-Band downlink, and S-Band uplink signals.

- Be capable of communicating with the ground when the spacecraft is maneuvering and when it is not.

The spacecraft shall support the following link characteristics:

**X Band (Science) Downlink:**

Transmit rate: 40 Mbps

Modulation: ~~Segmented~~ SQPSK

Format: CCSDS

Frequency: ~~8.03~~ <u>8.185</u> GHz (TBR)

EIRP: 36.8 dBm

Content:

GLAS science data

TurboRogue outputs and real-time navigation solutions

Instrument star camera and gyro outputs

Bus star camera outputs

GLAS instrument monitoring and control data

Link Quality: $1 \times 10^{-7}$ BER or better

Planned contacts:

Time: 3-4 times per day (TBR); 12.5 minutes per day

Duration: ten minutes, maximum

Maximum time between downlink contacts: ~~8~~ <u>10</u> orbits

Antenna Mask: 5 - 20° (TBR)

**S-Band (Real-Time Housekeeping) Downlink:**

Transmit rate: 16 kbps

Modulation: BPSK

Format: CCSDS

Frequency: 2.228 GHz (TBR)

EIRP: 25.5 dBm

Content:

Real-time ICESat housekeeping data

Real-time GLAS instrument monitoring and control data

Real-time GLAS health and welfare data

Real-time navigation solutions

Link Quality: $1 \times 10^{-7}$ BER or better

Planned contacts

Time: four times per day (minimum), nominally simultaneous with science playback

Duration: ten minutes, average

Maximum time between downlink contacts: 10 orbits

**S-Band (Stored Housekeeping) Downlink:**

Transmit rate: 256 kbps

Modulation: BPSK

Format: CCSDS

Frequency band: 2.228 GHz (TBR)

EIRP: 25.5 dBm

Content:

Stored spacecraft housekeeping data

Stored GLAS instrument monitoring and control data

Stored GLAS health and welfare data

Stored navigation solutions

Link Quality: $1 \times 10^{-7}$ BER or better

Planned contacts

Time: four times per day (minimum), nominally simultaneous with playback science

Duration: 500 seconds, maximum

Maximum time between downlink contacts: 10 orbits

**Uplink:**

Transmit rate: 2 Kbps

Modulation: BPSK

Format: CCSDS

Frequency band: transponder-compatible with 2.228 GHz (TBR) downlink

Content:

GLAS commands

ICESat commands

Flight software uploads including off-nadir programmed track tables and algorithms

Weekly star catalogue updates (TBR)

GLAS processor uploads

Link Quality: $1x10^{-7}$ BER or better

Planned contacts:

Simultaneous with Science and Housekeeping downlinks

### 3.2.7 Thermal Control

ICESat is an all-beta vehicle capable of maintaining any orientation relative to the sun for any length of time. The thermal control subsystem is primarily passive with the exception of active heater control for the battery and propulsion modules. The system maintains the bus operating temperature to within 10 °C of component qualification temperature limits. Nonoperating temperature limits are maintained from -30 °C to 70 °C. Blankets are designed to remove risk of electrostatic discharge events and to provide adequate venting during launch.

The system maintains thermal isolation between the bus and the payload to less than [redacted]. Detailed thermal models exist and can be modified for specific payload and missions in a relatively short time.

The GLAS radiator shall see a clear hemisphere, free of any obstructions from the spacecraft bus. During periods of low orbit beta angle, the spacecraft will be oriented so that the GLAS +X direction is in the orbit plane. During periods of high orbit beta angle, the GLAS +X direction will be oriented along the orbit normal or anti-normal, away from the Sun. The transition from one yaw orientation to the other will occur when beta angle is ~32°; several months will elapse between yaw transitions. ~~These restrictions shall not apply during periods of moderate orbit beta angle (25 - 40°) after nominal mission completion.~~

ICESat shall continuously monitor the GLAS temperatures using 12 thermistors, with outputs described in Section 3.2.5 and characteristics described in Section 5.2.6.3. These devices shall be supplied by Ball and mounted to mutually-agreeable locations on GLAS. Thermistor readings shall be included in the housekeeping telemetry stream, available in real-time and stored on the engineering DSU. The GLAS instrument will be thermally isolated from ICESat.

~~The RS2000 shall protect the GLAS radiator as follows:~~

- ~~Incident Earth infrared energy shall be <66 W/m2~~
- ~~Reflected solar energy, Earth infrared, and albedo energy shall be <5 W~~
- ~~No direct thermal radiation or reflected environmental energy shall be provided by the spacecraft into the radiator~~

### 3.2.8 Electrical

ICESat components are designed to meet the following environmental requirements.

#### *3.2.8.1 EMI/EMC Environment*

ICESat components meet the following requirements of MIL-STD-461C, Part 3, for Category A2a: CE01/02/03; CS01/02/03/04/05/06; RE01; RE02; and RSO1/02/03.

ICESat subsystems are designed and qualified to the applicable sections of MIL-STD-461C, Class A2. Electromagnetic compatibility verification will be accomplished with a thorough EMC test. This test will include, but not be limited to, verification of:

- Electrical bonding
- Receiver noise margins
- Power quality and transients
- Simultaneous subsystem functional performance

An electromagnetic environmental effects analysis shall be performed to assess effects of GLAS operations on ICESat, as integrated, upon receipt of relevant payload characteristics.

Results of this analysis shall be compared with the environmental requirements stipulated by MIL-STD-461D, governing GLAS design, and reconciled.

ICESat component signal interfaces are grounded in the following manner:

[redacted]

[redacted]

[redacted]

[redacted]

[redacted]

Secondary power grounds, inherently signal grounds, are returned to respective sources. Differential, bilevel, relay, and passive analog telemetry sensors shall be isolated from chassis. RF signals shall be referenced directly to chassis. High speed digital grounds shall have the capability to be connected directly to chassis to control common mode paths.

All MLI blankets shall be grounded to chassis, typically with a minimum of two points. All exposed nonconductive materials shall be treated, coated, or covered with a conductive layer to limit electrostatic charge buildup and discharge.

All metallic structure shall be electrically bonded with a Class R bond not to exceed 2.5 milliohms. All electrical or electronic components shall be electrically bonded to structure with a Class R bond not to exceed 2.5 milliohms. All electrical connectors carrying shielded wiring shall be electrically bonded to the receptive components not to exceed 10 milliohms. All other connectors shall be electrically bonded to structure with a Class S bond not to exceed 1 ohm.

All shields shall be grounded at each connector, preferably using EMI backshells.

#### 3.2.8.2 Magnetic Performance

ICESat components are designed to produce a magnetic dipole less than 0.1 A-m$^2$ (10 gauss-cm$^3$). Components are designed so they are not damaged by a 25-gauss demagnetization field.

#### 3.2.8.3 ESD Sensitivities

The GLAS instrument is ESD sensitive and shall be handled with due regard to avoiding electrostatic buildup by properly grounding connecting cables, work surfaces, equipment, and personnel.

MIL-STD-1686 is the guiding document to determine component and part ESD sensitivities, and all paperwork and packaging is annotated to reflect when the component or part is an ESDS item. Sensitivities vary, depending upon the payload and potential component upgrades required to meet particular mission requirements. Components are designed using best practices to minimize susceptibility to ESD damage in ground handling and on-orbit operations.

### 3.2.9 Radiation Tolerance

ICESat components are designed to operate during and after exposure to the radiation environment (trapped protons/electrons, solar flare events, and cosmic ray background) commensurate with low earth orbits. A reference orbit of 600 km at 98°, for a 5-year duration, was used. Higher orbit altitudes and other inclinations can be accommodated with the use of spot shielding and varying the thickness of the component enclosure.

ICESat's radiation tolerance shall provide a five-year lifetime in a 600-km, 94° orbit. The effects of SEUs shall be temporary and correctable, with no credible possibility of latchup.

### 3.2.9.1 Total Dose

Components are designed to meet the total ionizing dose environment provided in Figure 3-9. These curves do not contain margin; therefore, components are designed using a factor of 3 (when existing data is available on the same part number) and a factor of 5 (when parts are produced on the same fabrication [process] line as the candidate part). When data is not available for a part or inadequate design margins exist, testing is performed using MIL-STD-883 as a guide; part testing will be at 150% of the expected incident total dose.

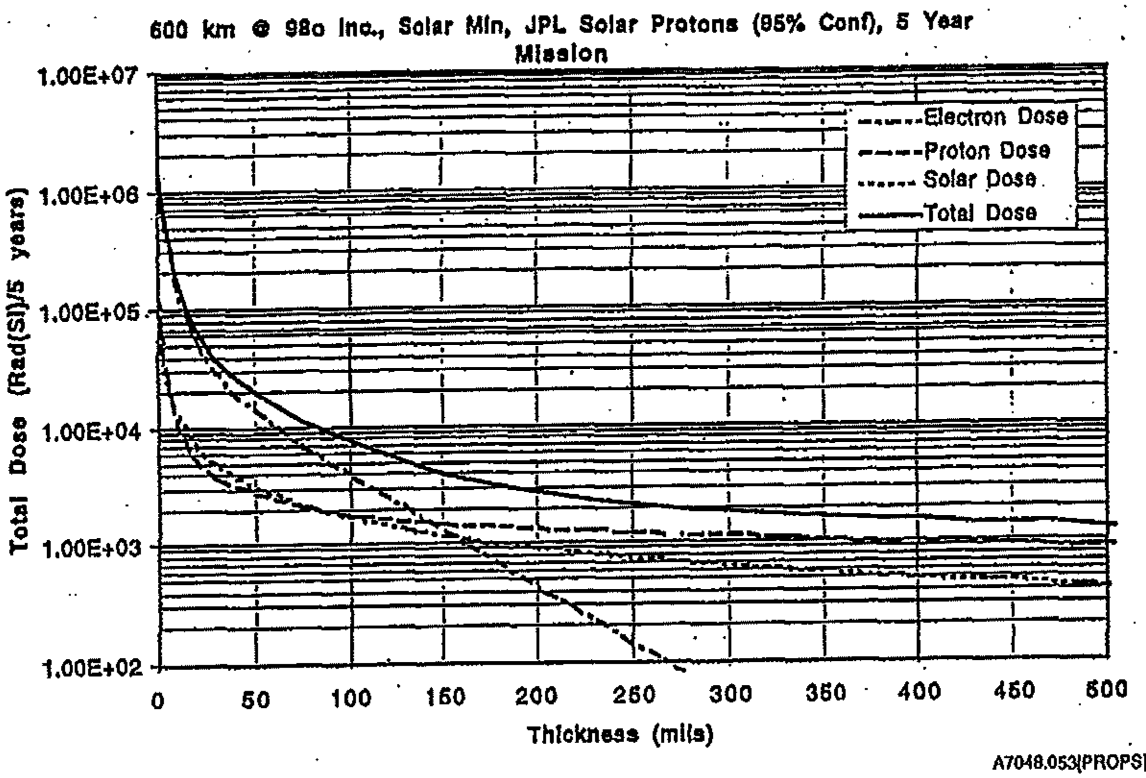


*Figure 3-9. Dose Depth Curve for 98 deg Inclination at 600 km*

### 3.2.9.2 Single-Event Effects

ICESat Systems and Subsystems are designed so no single-event effect can cause permanent damage. Excluding radiation degradation, SEU effects on components are temporary and correctable by automatic reset or ground command. Part selection to prevent latch-up is based on the parts capability to have no credible probability of latch-up when exposed to less than 100 MeV-cm$^2$/mg.

**Figure 3-10** provides the integral particle flux as a function of particle LET; CREME M=4, 7, and 9 curves include the trapped proton environment. SEU-system effects are determined by applying a safety factor of 2 to the evaluation of the upset rate. SEE testing, if required, will be performed using ASTM F1192-88 as a guide.

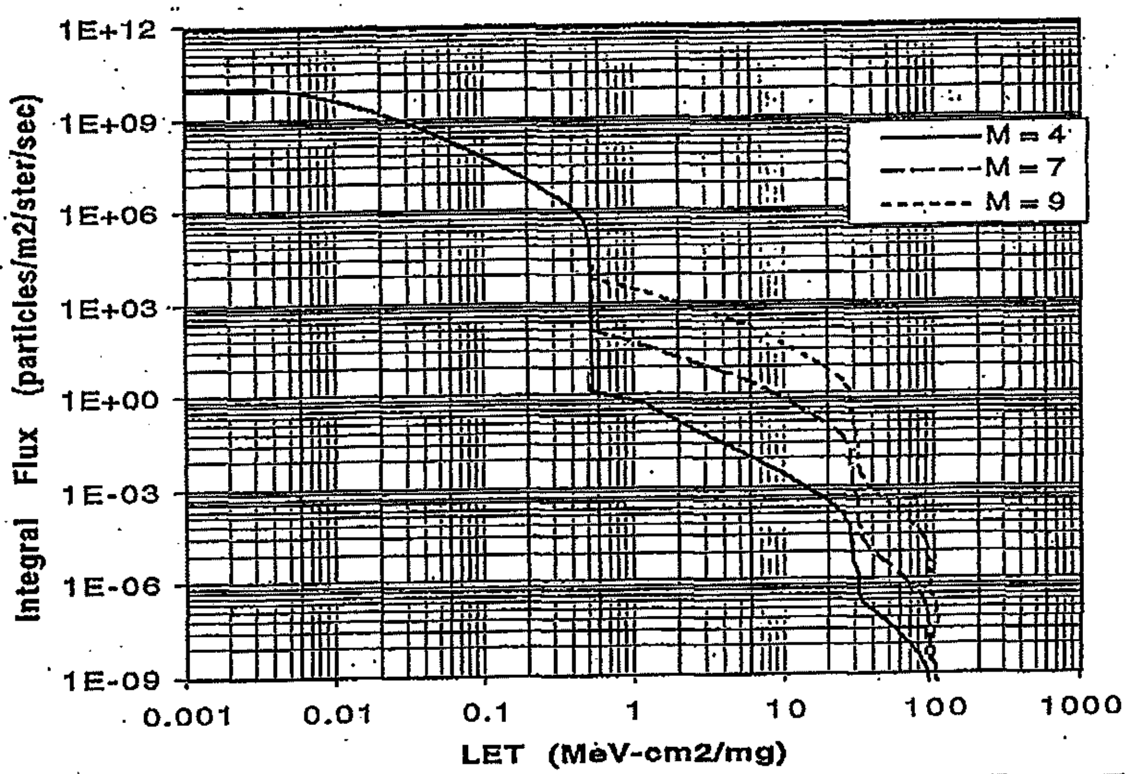


*Figure 3-10. Integral Particle Flux as a Function of LET*

### 3.2.10 Safe Modes

Multiple levels of safe mode protect the spacecraft under all conditions. [redacted]

Following LV separation, the platform is capable of autonomously stabilizing itself and then acquiring a safe attitude after separation. There are no unstable modes of operation.

ICESat is designed for automatic reacquisition of spacecraft attitude after loss of control. All attitude commands may be controlled by the ground. All attitude control modes may also be determined by the ground.

Operational attitude control is primarily autonomous with ground commanded tasking instruction sets uplinked into the command buffer memory. The satellite may be placed in an orbit frame attitude, which will be maintained indefinitely without ground contact.

All platform attitude-control subsystem commands normally executed autonomously during normal operation can be manually executed by ground command under all normal operations and predictable abnormal operations.

[redacted]

ICESat shall autonomously enter safe-hold operation upon detecting a mission-threatening fault (see Section 4.1.4). [redacted]

[redacted]

[redacted]

[redacted] Ball will work with the GLAS team on how to best protect the mission, considering GLAS and ICESat heater requirements, power availability, and battery resilience. [redacted]

# 4. Subsystem Characteristics

## 4.1 Overall ICESat Bus/Systems Architecture

[redacted]

[redacted]

[redacted]

Figure 4-1a. ICESat Bus On-Orbit Configuration [redacted] Ball will work with the GLAS team on how to best protect the mission, considering GLAS and ICESat heater requirements, power availability, and battery resilience. [redacted]

[redacted]

*Figure 4-1b. ICESat Bus On-Orbit Configuration*

[redacted]

[redacted]

[redacted]

[redacted]

*Figure 4-2a.*
*ICESat Stowed Configuration in a Taurus 2.33-m Diameter Fairing*

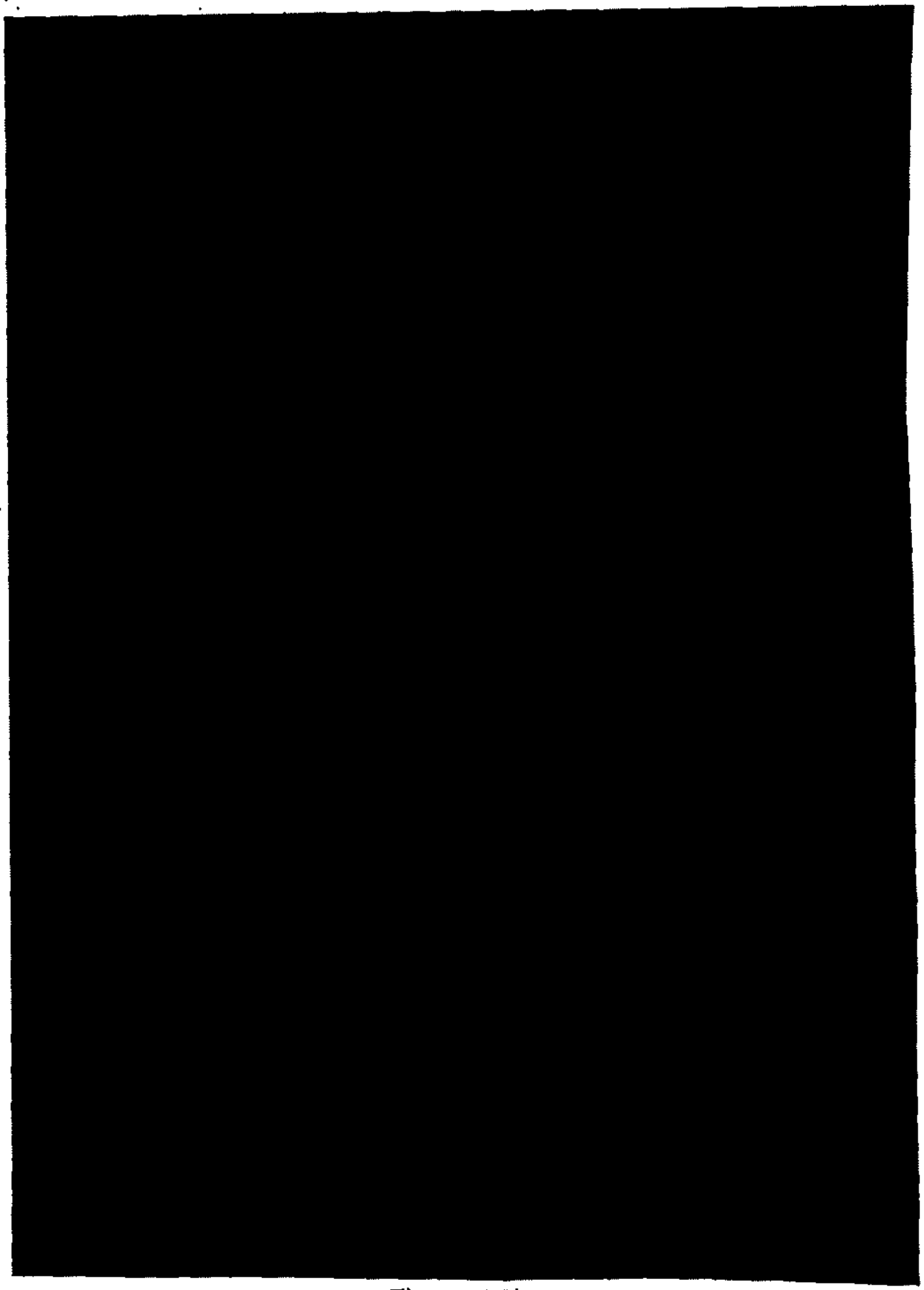

*Figure 4-2b.*
*ICESat Stowed Configuration in the Athena-2 Fairing*

[redacted]

Figure 4-3. ICESat Functional Block Diagram

| Unit | Manufacturer/Model No. | Heritage | Mass | Orbit Avg Power |
|---|---|---|---|---|
| [redacted] | [redacted] | [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] | [redacted] | |
| [redacted] | [redacted] | | [redacted] | [redacted] |
| | | | | |
| [redacted] | [redacted] | [redacted] | [redacted] | [redacted] |
| | | | | |
| [redacted] | [redacted] | [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] | [redacted] | |
| [redacted] | [redacted] | [redacted] | [redacted] | |
| [redacted] | [redacted] | [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] | [redacted] | |
| [redacted] | [redacted] | [redacted] | [redacted] | |
| [redacted] | [redacted] | [redacted] | [redacted] | |
| [redacted] | [redacted] | [redacted] | [redacted] | |
| [redacted] | [redacted] | [redacted] | [redacted] | |
| [redacted] | [redacted] | [redacted] | [redacted] | |
| [redacted] | [redacted] | [redacted] | [redacted] | |

*Figure 4-4. Typical ICESat Components*

### 4.1.1 Structural/Mechanical Subsystem

[redacted]

[redacted]

*Figure 4-5. ICESat Structure Configuration*

[redacted]

[redacted]

[redacted]

[redacted]

[redacted]

[redacted]

| Item | Mass (kg) |
|---|---|
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |

*Figure 4-6. ICESat Bus Mass Summary*

### 4.1.2 Power System

[redacted]

[redacted]

| Item | Units | Value | Notes |
|---|---|---|---|
| [redacted] | [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] | |

*Figure 4-7. Power System Key Features*

[redacted]

[redacted]

| Item | Power (W) |
|---|---|
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |

*Figure 4-8. Power System Average Loads*

[redacted]

### 4.1.3 Propulsion Subsystem

[redacted]

[redacted]

*Figure 4-9. ICESat Propulsion Module*

[redacted]

[redacted]

[redacted]

[redacted]

- [redacted]
- [redacted]

### 4.1.4 Attitude Determination and Control Subsystem (ADCS)

[redacted]

[redacted]

*Figure 4-10. ADCS Subsystem Functional Block Diagram*

[redacted]

| Mode | Description |
|---|---|
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |

*Figure 4-11. Attitude Control Modes. Simple attitude control modes are used to reduce risk and ground control workload. Two survival modes–safe and emergency–prevent mission loss in contingency situations and allow for rapid recovery in the event of critical system faults.*

### 4.1.5 Command & Data Handling Subsystem (C&DH)

[redacted]

[redacted]

- [redacted]
- [redacted]

[redacted]

[redacted]

[redacted]

| [redacted] | [redacted] |
|---|---|
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |
| [redacted] | [redacted] |

*Figure 4-12. Flight Processor Features*

[redacted]

[redacted]

[redacted]

[redacted]

[redacted] [redacted]

[redacted]

| Parameter | Units | Value |
|---|---|---|
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | | |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | | |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |

*Figure 4-13. Uplink Performance*

| Parameter | Units | Value |
|---|---|---|
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | | |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | | |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | | |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] |

*Figure 4-14. Downlink Performance*

[redacted]

[redacted]

[redacted]

| Parameter | Units | Value (5° mask) | Value (20° mask) |
|---|---|---|---|
| [redacted] | [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] | [redacted] |
| [redacted] | [redacted] | [redacted] | [redacted] |
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*Figure 4-15. Science downlink performance*

### 4.1.7 Thermal Control Subsystem

[redacted]

[redacted]

[redacted]

[redacted]

[redacted]

## 4.1.8 ICESat Flight Software

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| Table | Function |
|---|---|
| [redacted] | [redacted] |

Figure 4-15.5. Flight Software Tables

[redacted]

[redacted]

[redacted]

*Figure 4-16. Platform Simulator*

### 4.1.9 ICESat Ground Support Equipment

[redacted]

[redacted]

#### ~~4.1.10 TT&C Ground System Equipment~~

[redacted]

#### ~~4.1.10.1 Spacecraft Test and Operations Center (STOC)~~

[redacted]

[redacted]

### ~~4.1.10.2 Command Modulation Equipment~~

[redacted]

### ~~4.1.10.3 Telemetry and Data Acquisition~~

[redacted]

[redacted]

### ~~4.1.10.4 Mission Operations Software~~

[redacted]

- [redacted]
- [redacted]
- [redacted]
- [redacted]
- [redacted]

### ~~4.1.10.5 Spacecraft Event Prediction and Command Management~~

[redacted]

[redacted]

~~use by operations personnel. Pass plans integrate all relevant information needed to conduct real-time activities, monitor ongoing stored commanding, and coordinate station activities.~~

~~Flight table and software load inputs are generated by custom software packages. These packages format and assemble the data into files that are usable by the STOC function for uplink to the spacecraft.~~

# 5. Payload Accommodation

The payload accommodation section describes how ICESat can accommodate payloads. Its purpose is twofold: to describe the baseline ICESat bus interfaces in a generic manner and to serve as a template for a payload specific ICD. Areas where generic information is not available but where mission specific information needs to be addressed has been included and noted as "payload specific" or "TBD."

## 5.1 Physical Requirements

### 5.1.1 Spacecraft Coordinate System

**Figure 5-1** show the ICESat coordinate system. The coordinate system frame is centered on the spacecraft centerline and is 3.6-in. above the separation plane. The bottom surface of the zenith deck is considered "station 0.0"

[redacted]

The GLAS +X axis is nominally parallel to the ICESat +Z axis. The GLAS +Y axis is nominally parallel to the ICESat +Y axis. The GLAS +Z axis is nominally parallel to the ICESat -X axis.

In the ICESat coordinate frame, the origin of the GLAS coordinate system is nominally at:

X = ~~28.8 cm (Taurus XL configuration),~~ 21.2 cm ~~(Athena-2 configuration)~~

Y = 0.0 cm

Z = 220.3 cm

### 5.1.2 Available Payload Envelope

**Figure 5-2** shows the payload mounting surface for ICESat. **Figure 5-3** shows more detail of the mounting interface. The payload nominally mounts to the areas indicated with squares using a 3 point thermally isolated kinematic mount. Details of the mechanical interface is given in Section 5.1.6. Space is also available within the bus cavity below the nadir deck. The available volume above the nadir deck is limited by the payload fairing and stowed edges of the solar array. Space also needs to be allocated for the antenna masts. The available volume is shown in **Figure 5-4**. For compatibility with the Athena-2 fairing envelope, minor reshaping of the GLAS radiator is assumed~~will be required~~. No reduction in radiator area is needed.

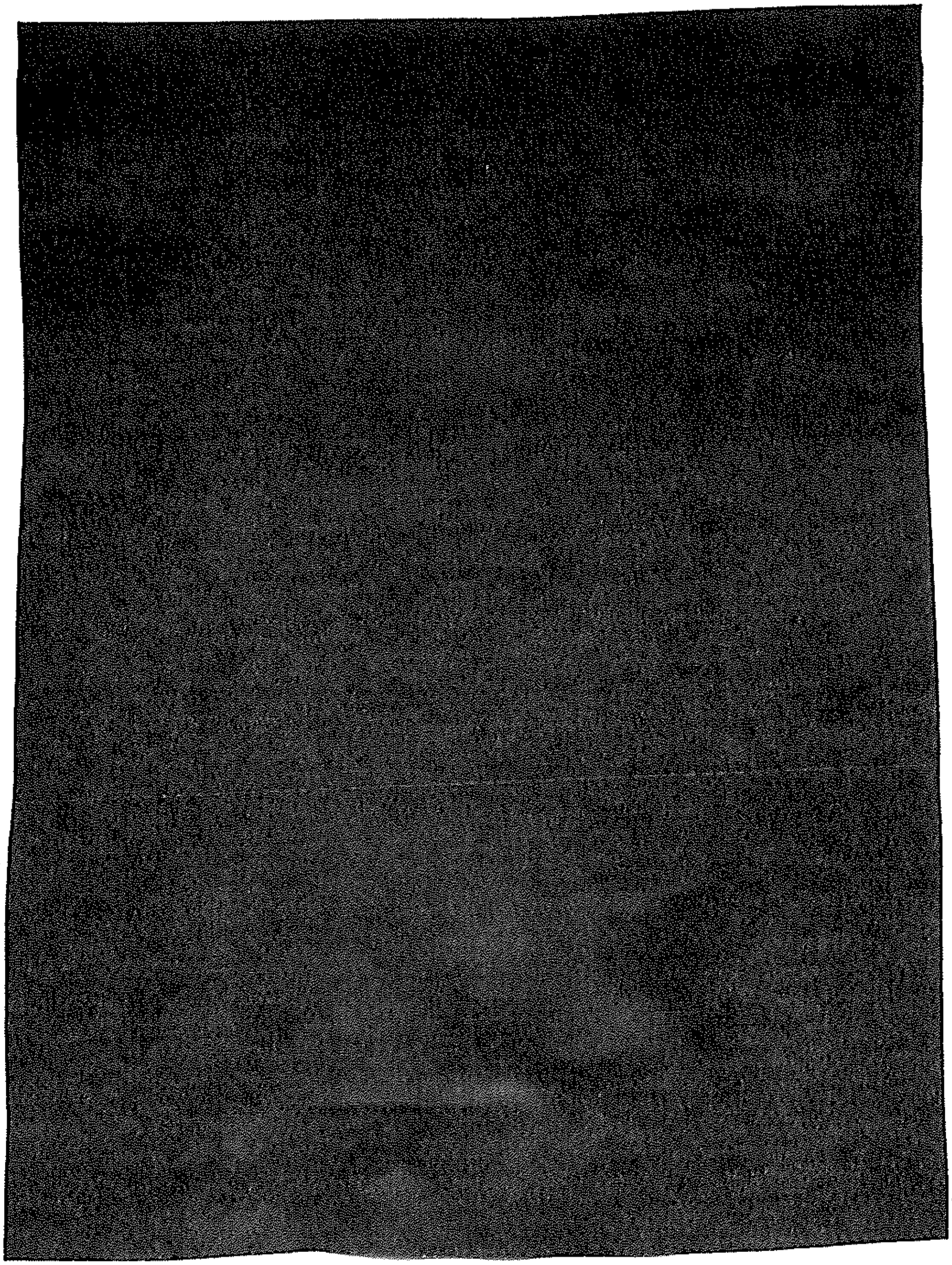

*Figure 5-3. Payload Mounting Interface*

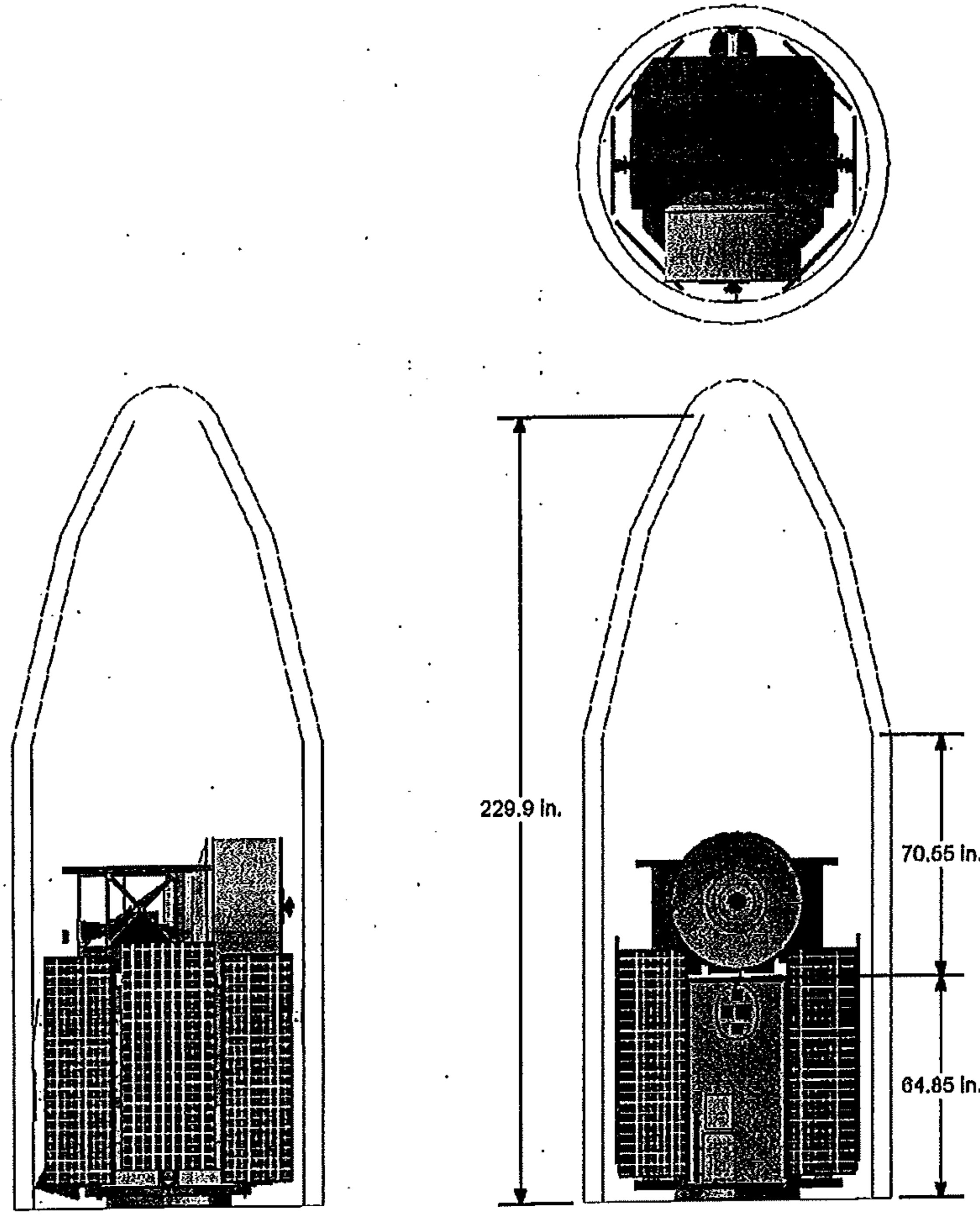


*Figure 5-4. ICESat stowed configuration in a Taurus 2.33-m diameter fairing*

### 5.1.3 Mass

ICESat shall support a GLAS mass of 300 kg, including the mass of the GLAS mounting structure. ICESat shall also incorporate GFE GPS receivers and antennas (2 each), weighing a total of 7.5 kg (TBR) exclusive of GLAS allocation.

### 5.1.4 Payload Center of Mass

ICESat shall accommodate a GLAS cg location anywhere within the following envelope with respect to the ICESat coordinate frame:

X = ~~0.0 ± 7 cm (Taurus XL configuration),~~ -7.6 ± 7 cm ~~(Athena 2 configuration)~~

Y = 8.2 ± 7 cm

Z = 207.3 ± 7 cm

### 5.1.5 Moments of Inertia

ICESat shall accommodate the mass moments of inertia of the GLAS instrument. These will not exceed the following maximum values, measured about the GLAS center of mass in the GLAS coordinate frame:

Ixx = 38.3 kg-m2 ± 10%

Iyy = 30.0 kg-m2 ± 10%

Izz = 28.1 kg-m2 ± 10%

Ixy = -0.9 kg-m2 ± 10%

Iyz = 2.9 kg-m2 ± 10%

Ixz = -8.4 kg-m2 ± 10%

### 5.1.6 Mechanical Interfaces

#### *5.1.6.1 Mounting*

GLAS will be mounted to the ICESat bus using ~~semi~~-kinematic mounts on the instrument side of the interface. ICESat shall provide hard points for this interface.

**Mounting Method.** The mounting method shall accommodate manufacturing tolerance, structural, and thermal distortions. The method by which each payload component is mounted to the spacecraft shall be defined in detail.

**Mounting Interface Documentation.** The spacecraft mounting interface for each payload component shall be documented in this ICD.

**Mounting Location and Documentation.** The integrating contractor working with the sensor contractor shall determine the location of the payload on the spacecraft. This location shall be documented in the ICD.

### *5.1.6.2 Alignment*

**Alignment References.** The payload contractor shall provide a payload alignment reference. The spacecraft shall provide a spacecraft alignment reference. Both the payload alignment reference and the spacecraft alignment reference shall be viewable from two orthogonal directions.

**Alignment Responsibilities.** The payload contractors shall be responsible for measuring the alignment angles between the payload boresight (line-of-sight), if applicable, and the payload alignment reference. The spacecraft contractor shall be responsible for aligning the sensor alignment reference to the spacecraft attitude reference..

**Alignment Control.** The spacecraft contractor shall control the alignment of the payload alignment reference with respect to the spacecraft attitude reference to within values specified by the payload contractor.

**Spacecraft Attitude Reference.** For spacecraft pointing an attitude reference frame shall be defined~~ in accordance with Section 3.1.5.1~~.

Alignment Accuracy. Alignment accuracy TBD.

~~*Attitude Reference Knowledge.* The spacecraft will supply a three-axis attitude of the spacecraft attitude reference for ground processing. The supplied attitude will be time-tagged and possess an angular rms accuracy as required by the payload provider.~~

~~*Attitude Reference Control.* The rms of the attitude reference control error over a bandwidth of dc to 10 Hz shall be less than 0.01 deg per axis.~~

~~*Attitude Reference Rate Error.* The rate error of the attitude reference frame shall be less than 0.03 deg/s during all mission data collection periods.~~

~~**Ephemeris Knowledge.** The spacecraft will provide a spacecraft ephemeris estimate with an rms uncertainty of 50 meters for radial/in-track/cross-track components.~~

### *5.1.6.3 General Structural Design Requirements*

A-basis material allowables shall be used for design. An A-basis allowable is defined as a value where 99% of a population of values is expected to equal or exceed the allowable, with a confidence of 95%.

**Structural Support.** The spacecraft shall provide structural support for the payload so the loads transmitted across the interface into the payload do not exceed interface limit loads to be determined by the spacecraft contractor. The payload and interface equipment shall be designed to design load factors determined by launch vehicle acceleration levels. A survey of typical launch vehicle environments (accelerations, frequencies, temperatures) is included in Section 3.9.

**Payload Structural Dynamics.** When the payload is in its launch-locked configuration, the fundamental natural frequency of the payload shall be 50 Hz or greater, axial and lateral. For a deployable, the spacecraft contractor shall specify a deployed frequency so the payload will not saturate the satellite's control capability. The lowest natural frequency for a deployed payload shall be greater than 6 Hz. The payload contractor shall ensure that the sensor dynamic characteristics and control capability (for example, a gimbaled sensor) will meet the requirements specified for the deployed frequency.

ICESat shall accommodate GLAS structural modes and design loads. The GLAS instrument has a minimum first fundamental frequency of 75 Hz when mounted to a rigid interface via flexures. The net cg quasi-static flight level loads are ±12 g in each of three orthogonal axes acting independently. ICESat shall not couple loads that exceed the acceleration levels~~net cg quasi-static flight level loads~~.

During science operations, the maximum jitter spectrum at the ICESat interface with GLAS shall not exceed 20% of the acceleration levels as shown on the upper line in Figures 5-6a and 5-6b, showing both thrust direction (X) and lateral (Y and Z) acceleration requirements. Estimated jitter is overlaid on these requirements during quiescent operations (solid curves) and when the solar array is articulating (dotted).

[redacted]

*Figure 5-6a. PSD of Jitter Motion for a Sample Operating Case—Thrust Direction.*

[redacted]

*Figure 5-6b. PSD of Jitter Motion for a Sample Operating Case—Lateral Direction.*

**Interface Design Limit Loads Requirements.** The flight hardware shall be able to withstand all worst-case load conditions to which it may be exposed during ground (handling and transportation), prelaunch, launch, and on-orbit operations. Positive structural margins of safety must be maintained so that the sensor can meet all of its design requirements after being subjected to the worst case loads combination. In cases involving maintenance of payload critical components for on-orbit operations, the precision elastic limit shall be used for structural materials. The design factors of safety shown in **Figure 5-7** shall be applied to all loading conditions.

| Design/Test Options | Factors of Safety (Yield) | Factors of Safety (Ultimate) |
|---|---|---|
| 1. Dedicated test article | 1.00 | 1.25 |
| 2. Test on flight article | 1.25 | 1.40 |
| 3. Proof test each flight article | 1.10 | 1.25 |
| 4. No-static test | 1.60 | 2.00 |

*Note: The level of required analysis increases significantly with increased option number. For the no-static test option, a detailed and comprehensive structural analysis is required and must be available for review by the space vehicle customer.*

**Figure 5-7.** Factors of Safety

The dedicated test article is a qualification test article that will be subjected to the maximum expected loads times 1.25. The test on flight article option refers to a protoqualification on the structure. All composite structures and structural bonded joints shall be proof tested regardless of safety factor, but a metallic structure is usually qualified such that each unit will not have to be tested, or it is protoqualed. The no-static test option allows the capability of the structure to be determined via purely analytical methods, with the analytical models not being verified by test but verified by the integrator or government for accuracy.

*Combined Structural Dynamics Analysis Responsibility.* The spacecraft contractor shall be responsible for the combined structural dynamics analysis of the spacecraft bus and the payloads.

*Combined Structural/Dynamic Analysis.* All models shall be exchanged in NASA Structural Analysis (NASTRAN) bulk data format. A test-verified model is preferred when available and is required if the payload lowest frequency is less than 50 Hz as shown by analysis.

*Combined Structural Dynamics Analysis Results.* The spacecraft contractor shall provide the combined structural dynamics analysis results to both the customer's program office and the payload contractors.

*Coupled Loads Analysis Results.* The launch vehicle-spacecraft coupled loads analysis will be performed by the launch vehicle contractor. The spacecraft contractor shall be responsible for providing the results of the launch vehicle-spacecraft coupled loads analysis in a standard format (TBR) to the payload contractors.

*Structural Analyses.* A structural analysis using maximum equivalent loads shall be conducted by the payload developer on all sensors. In addition, those sensors with modes under 50 Hz (as shown by the model) must have a full modal survey test completed in a base fixed configuration to obtain all mode shapes and frequencies to correlate the dynamics model.

An analysis using static loads shall be performed if those loads exceed the maximum equivalent values. The spacecraft contractor shall provide mission-specific information for maximum equivalent loads to the payload developer for his static load analyses.

**Pressurized System Design.** Payloads with pressurized systems shall follow the agreed-upon requirements ~~of MIL-STD-1522~~.

**Sensor Mass Model.** The requirement for mass models shall be determined by the payload provider and the spacecraft contractor.

**Mechanisms and Deployables.** Payload developers shall use the design and test guidelines provided in Goddard standards~~MIL-A-83577~~ to increase reliability of Moving Mechanical Assemblies (MMAs) and facilitate integration and test activities.

~~Actuating Devices. Actuating circuitry shall be two-fault-tolerant to unanticipated deployment or release.~~

**Sensor Disturbance Allocations.** The payload developer shall ensure that the "swept" or deployed volume is verified to ease integration and operation, accounting for all distortions and misalignments. In addition, the spacecraft contractor shall provide estimates of allowable disturbance torques, vibration, and end-of-travel or latch-up loads to the payload developer.

**Uncompensated Momentum.** Each payload having movable components shall not exceed an uncompensated momentum contribution to be defined and agreed to in an ICD between the payload contractor and the spacecraft contractor.

**Constant and Periodic Disturbance Torque Limits.** The magnitude spectrum of the disturbance torque that the payload imparts to the spacecraft shall be defined in the ICD.

**Torque Profile Documentation.** The actual payload torque time profile shall be documented in the ICD.

**Thrust Direction Definition.** The magnitude and direction of net thrust resulting from the expulsion of expendables by the payload shall be documented in the ICD.

**Magnetics.** ~~Avoid using large quantities of magnetic materials where possible.~~ If magnets are inherent to the payload design, early estimates of magnetic fields and residual magnetic dipole moments shall be provided to the integrating contractor . A full magnetic survey shall be conducted by the corresponding payload contractor if the sensor has a total residual uncompensated magnetic moment greater than 0.1 ampere-turn-$m^2$.

**Access Identification.** Access requirements shall be documented in the ICD.

*General Access.* All items to be installed, removed, or replaced at the ~~satellite~~ observatory level shall be accessible without disassembling of the unit.

**Handling Fixtures.** The payload contractor shall provide proof tested handling fixtures for each component. Handling fixtures shall be designed to 5 times limit load for ultimate and 3 times limit load for yield. Handling fixtures shall be tested to 2 times working load.

**Mounting Orientation.** Payloads shall be able to be mounted to the spacecraft with the spacecraft in the horizontal or vertical position.

**Payload to Spacecraft Integration and Test Mounting.** Payloads shall be able to be mounted or removed without removal of other components.

**Nonflight Equipment.** All nonflight items to be installed or removed prior to flight shall be identified in the ICD.

### *5.1.6.4 ~~Star Tracker Interface~~ GPS Cable Length*

~~A star tracker assembly and its shade are nominally mounted near the zenith deck.~~ The length of the cable between the GPS receiver and its antenna shall be no longer than 1 meter.

### 5.1.7 Thermal Interface

GLAS thermal control is the responsibility of the payload developer. The payload must be thermally isolated from the bus structure. This is typically accomplished by using low conductance kinematic mounts and Multi-Layer Insulation (MLI) blankets.

Heat rejection from the payload can be achieved by radiation directly from any exposed surfaces of the payload or to a dedicated payload radiator that is mounted to the -X bus panel. This radiator is typically used for tight temperature control of sensitive instruments or electronics and is thermally isolated from the bus.

A significant amount of payload support electronics can be mounted to the spacecraft middeck and external panels. External panel mounted boxes are normally covered by MLI and dissipate their heat to the underlying panel. If required, additional cooling can be achieved on high dissipation components by removing MLI to provide a direct view to space.

[redacted]

A total of 12 thermistors are available for monitoring payload interface temperatures in the telemetry stream.

**Figure 5-8** summarizes the thermal interface parameters for payload mounted in the central bus cavity. **Figure 5-9** lists the thermal requirements for support electronics mounted to the middeck or the external panels.

| Parameter | Value |
|---|---|
| Bus Panel Operating Temperature Range | -10 to 40 °C |
| Bus Panel Nonoperating Temperature Range | -20 to 50 °C |
| Bus Panel Emissivity | <0.10 |
| Maximum Interface Conductance. | 1.5 W °C |
| Maximum Allowable Payload to Bus Heat Transfer Rate (includes radiation and conduction ) | ±38 W |
| Nominal Heat Load to Dedicated Payload Radiator (@-2 °C) | 14 W |

*Figure 5-8. Interface Parameters—Payload Cavity Mounted Instrument*

| Parameter | Value |
|---|---|
| Component Operating Temperature Range | -10 to +50 °C |
| Component Nonoperating Temperature Range | -20 to 60 °C |
| Allowable Orbit Average Heat Dissipation (total for all boxes ) | 77 W |

*Figure 5-9. Thermal Requirements—Bus Mounted Support Electronics*

### 5.1.7.1 Thermal Control Hardware

External payload thermal control hardware shall be documented in the ICD. The responsibility for providing thermal control hardware is defined in **Figure 5-10.**

| Hardware | Responsibility |
|---|---|
| Interface Heaters and Thermistors | Spacecraft Provider |
| External Payload Radiator | Payload Provider |
| Internal/External Payload Thermal Control Hardware | Payload Provider |
| External Radiator Interface Hardware (such as heat pipes, straps, brackets, etc.) | Payload Provider |

*Figure 5-10. Thermal Control Hardware Responsibility*

### 5.1.7.2 Multilayer Insulation

The MLI used in thermal control design shall have the following provisions: venting, interfacing with spacecraft thermal control surfaces, and electrical grounding to prevent Electro Static Discharge (ESD). The ~~integrating~~ contractor shall approve the MLI selection in the Parts, Materials, and Processes Control Board (PMPCB) review process.

### 5.1.7.3 Other Considerations

Thermal control surfaces shall be cleanable to visibly clean or better. Any sealed or closed system, such as heat pipes, thermal control enclosures, or fluid loops, shall be analyzed to demonstrate that a safety hazard does not exist.

## 5.1.8 Field of View Constraints (RF, Optical, Thermal)

<u>Nadir Fields Of View</u>

ICESat shall provide:

1. An unobstructed optical field of view of 0.4 mr about nadir for the GLAS receiver telescope
2. An unobstructed optical field of view of ± 60 degrees about nadir for the GLAS satellite laser reflector
3. A glint-free FOV of plus or minus 66 degrees about the GLAS -Z axis shall be provided

**Zenith Fields Of View**

ICESat shall provide:

1. An unobstructed glint-free optical field of view of ±8 degrees about zenith for the GLAS stellar reference system
2. An unobstructed hemispherical RF field for the GPS antenna with minimized multipath interference

**Thermal Field Of View**

The spacecraft shall not obstruct any of the GLAS +X hemisphere for the thermal radiator. This view will be normal to nadir. During periods of low orbit beta angle, the spacecraft will be oriented so that the GLAS +X direction is in the orbit plane. During periods of high orbit beta angle, the GLAS +X direction will be oriented along the orbit normal or anti-normal, away from the Sun.

#### 5.1.9 Contamination

There shall be no venting from the bus to the payload.

## 5.2 Electrical Power and Signals/Data Interface

### 5.2.1 Data Bus Interface Protocol(s)

#### *5.2.1.1 MIL-STD-1553B Narrowband Serial Digital Interface*

The payload interfaces with redundant MIL-STD-1553B command/response data buses for the purpose of receiving command and status messages from the Bus Electronics (BE) and sending data to the BE.

The payload will send and receive commands from the BE on the 1553 data bus.

**Payload Terminal Identification.** The BE will function as the bus controller, either via the Primary 1553 Bus A or the Redundant 1553 Bus B. The payload will function as Remote Terminal Units (RTU)

**Signal Characteristics at payload Interface.** The MIL-STD-1553B data bus characteristics at the payload interface will be in accordance with **Figure 5-11.**

| Characteristic | Value |
|---|---|
| Data Bus Requirements | Mil-Std 1553B, Remote Terminal |
| Data Rate | 1 MHz |
| Word Length | 20 bits (3 sync, 16 data/command, 1 parity) |
| Number of Data Words per Transfer | 1 to 32 |
| Transmission Technique | Half-duplex |
| Operation | Asynchronous |
| Encoding | Manchester II Biphase |
| Bus Coupling | 1:2 Stub-to-DPU Electronics Turns Ratio (typical) |
| Bus Control | Single or Multiple (single word or block transfer) |
| Transmission Media | Twisted pair shielded |
| Characteristic Impedance of Cable | 70Ω to 85Ω at 1 MHz |

*Figure 5-11. Payload MIL-STD 1553 Interface Characteristics*

The MIL-STD-1553B data bus characteristics at the BE interface will be in accordance with the following **Figure 5-12.**

| Characteristic | Value |
|---|---|
| Data Bus Requirements | Mil-Std 1553B, Bus Controller |
| Data Rate | 1 MHz |
| Word Length | 20 Bits |
| Number of Data Bits / Word | 16 |
| Transmission Technique | half-duplex |
| Operation | Asynchronous |
| Encoding | Manchester II Bi-phase |
| Bus Coupling | Transformer Coupled, 1.4:1 Bus-to-Stub Turns Ratio |
| Bus Control | Single or Multiple (Single word or block transfer) |
| Transmission Media | Twisted pair Shielded |
| Characteristic Impedance of Cable | 70Ω to 85Ω at 1 MHz |

*Figure 5-12. BE MIL-STD 1553 Interface Characteristics*

**1553 Message and Word Formats.** The message and word formats will be in accordance with MIL-STD-1553B.

### 5.2.2 Maximum Payload Data Ingest Rate

The payload data ingest rate is governed by the capabilities of the ~~SR~~SSR. These capabilities are detailed in Section 5.2.5, Solid State Recorder Interface.

### 5.2.3 Payload/Spacecraft Bus Synchronization

The SCCs interface with the payload via redundant 1553 buses; the active SCC is the 1553 Bus-Master. [redacted]

### 5.2.4 Electrical Power Interface

The ICESat bus will provide electrical power interfaces to the payload.

#### *5.2.4.1 Power Availability*

~~During science operations, ICESat shall provide 300 W orbit average, 350 W (TBR) peak, for primary payload power for the entire five-year mission life.~~

ICESat shall provide GLAS with ~~300~~ 350 W orbit-average, ~~350~~ 400 W (TBR) peak, at 28±6 V for the entire 5-year consumables lifetime. The GLAS instrument will provide its own power conditioning. ICESat shall provide primary payload power on six separate individually switched and unfused feeds, with the following characteristics:

- Three GLAS laser power feeds. Each feed shall provide ~~100~~ 106 W steady-state to a laser and 5 W start-up load to a thermal control circuit. Energizing any of these three feeds shall be indicated via a Laser On flag in the engineering telemetry. Each feed has a turn-on inrush current of 5 A for 5 msec. ICESat will power only one laser at a time, activated during real-time ground contacts.
- One GLAS star camera power feed. 15 W orbit-average shall be provided.

- One GLAS IRU power feed. ~~50~~ 27 W orbit-average shall be provided.
- One GLAS electronics subsystem power feed. ~~135~~ 155 W orbit-average shall be provided.
- One GLAS LRS power feed. 16 W orbit-average shall be provided.

To preserve mission integrity, ICESat shall be able to shed the entire primary GLAS power autonomously, without prior coordination with GLAS nor with the EOSDIS and ICESat flight operations. ICESat shall autonomously enter safe-hold operation upon detecting a mission-threatening fault (see Section 4.1.4). For robustness, ICESat incorporates both a Safe Mode and an Emergency Mode.

A separate power feed is required for GLAS keep-alive power. ~~GLAS keep-alive power is not required prior to Observatory initialization (that is, through launch and orbit-injection). Note that the GLAS laser crystals suffer hard and irrevocable failure when cooled below their minimum non-operating temperature limit. D~~Beginning no more than 5 orbits (TBR) after launch, during Safe Mode and Emergency Mode, ICESat shall provide ~~100~~ ~~150~~ 50 W (TBR). GLAS primary power shall be shed for all ~~six~~ seven feeds at once, not by individual power distributions.

#### *5.2.4.2 Voltage*

[redacted]

#### *5.2.4.3 Ripple*

The ripple voltage shall be less than 0.8-V peak to peak as measured over the bandwidth from dc to 10 MHz when the power system is delivering the maximum rated current into resistive loads.

#### *5.2.4.4 ~~Under-voltage~~ (Deleted)*

~~The payload shall not sustain permanent damage or performance degradation by applying any input voltage in the range of 0 to +25 V for any time duration or any voltage transition (monotonic, or nonmonotonic) within this range for any rise and fall time.~~

### *5.2.4.5 (Deleted)*

### *5.2.4.6 ~~Reverse-Voltage Protection~~ (Deleted)*

~~The payload shall not be damaged or otherwise adversely impacted as a result of applying reverse polarity power to the payload.~~

### *5.2.4.7 Conducted Transient Voltage*

The payload shall not generate transients on the primary dc power lines in excess of the limits shown in Figure 5-13 when power is applied to the equipment through the specified Line Impedance Simulation Network (LISN), as discussed in Section 5.2.4.9. This limit shall be applicable during turn-on, turn-off, and normal operation.

*Figure 5-13. Voltage Transient Limit*

### *5.2.4.8 Raw Input Current*

**Off State.** The maximum power when the payload is in the off-state shall be no greater than 4 W at 28 V with relays closed, not including thermal control.