Gravitational Reference Advanced Technology Test in Space

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Data Restrictions

The data in this ODAR is considered Sensitive/Proprietary

Document Revisions

Rev	Changes	Pages
A	Initial Release (CDR)	All

NASA DAS Version

NASA DAS Version: 3.2.7
Solar flux table: *solarflux_table_03262026 (updated 2026-04-10)*

ODAR Self-Assessment

The following ODAR results are shown in the table below per **NASA-STD-8719.14C Appendix A.2.**

Requirement	Spacecraft (Compliant, N/A, Not Compliant, or Incomplete)	Comments For all incompletes, include risk assessment (low, medium, or high risk) of non-compliance & Project Risk Tracking #
4.3-1.a *MRD 25-year limit*	N/A	No planned debris release
4.3-1.b *MRD <100 object x year limit*	N/A	No planned debris release
4.3-2 *GEO MRD*	N/A	No planned debris release
4.4-1 *<0.001 Explosion Risk*	Compliant	On board energy source incapable of debris-producing failure
4.4-2 *Passivate Energy Sources*	Compliant	
4.4-3 *Limit Intentional BU*	N/A	No planned breakups
4.4-4 *Limit Intentional BU*	N/A	No planned breakups
4.5-1 *<0.001 10 cm Impact Risk*	Compliant	
4.5-2 *<0.01 Small MMOD Impacts*	Compliant	

Requirement	Spacecraft (Compliant, N/A, Not Compliant, or Incomplete)	Comments For all incompletes, include risk assessment (low, medium, or high risk) of non-compliance & Project Risk Tracking #
4.6-1a-c *LEO Disposal*	Compliant	
4.6-2 *Storage or Earth-escape*	N/A	
4.6-3 *Long-term Reentry*	N/A	
4.6-4 *Disposal Reliability*	Compliant	
4.7-1 *Reentry Risk*	Compliant	
4.8-1 *Special Classes*	N/A	No tether

Section 1: Program Management and Mission Overview

1.1 Program Management

The Gravitational Reference Advanced Technology Test in Space (GRATTIS) is a NASA-funded technology demonstration that is PI-led by the University of Florida (UF) as part of NASA Earth Science Division (ESD) In-Space Validation of Earth Science Technologies (InVEST) program under Grant No. 80NSSC24K1408.

The Sponsoring NASA Program Executive is Dr. Sachidananda "Sachi" Babu. The NASA HQ Technology Validation Program Manager is Dr. Sachidandana Babu.

The responsible Principal Investigator / Project Manager is Dr. John Conklin from the University of Florida. The spacecraft is owned and operated by the University of Florida.

1.2 Mission Schedule

The following table details the GRATTIS technology demonstration design and development milestones.

Milestone	Date
ATP	8/1/1024
SRR	10/24/2024
PDR	4/3/2025
CDR	9/3/2025
IRR or TRR	4/23/2026
FRR	1/11/2027
Launch (Notional)	NET 2/1/2027
Commissioning	2/1/2027

Milestone	Date
Science Operations	3/1/2027
Decommissioning	7/31/2027
Project Closeout	7/31/2027

1.3 Mission Description

The primary objective of GRATTIS is to test the Simplified Gravitational Reference Sensor (S-GRS). This next generation inertial sensor is significantly more sensitive than those used on missions like GRACE-FO (Gravity Recovery and Climate Experiment Follow-On). By measuring nanometer-scale changes in gravity, this demonstration aims to provide unprecedented data on:

- Hydrology: Monitoring groundwater reserves and droughts.
- Cryosphere: Tracking the melting of ice sheets and glaciers.
- Oceanography: Observing changes in sea levels and ocean currents.

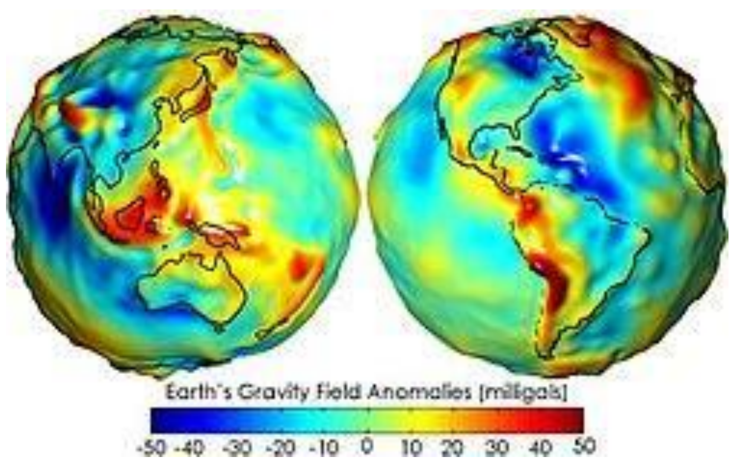

Figure 1.3-1: Earth Gravity Field Anomalies (credit: NASA Earth Observatory)

The secondary objective is to test the Optomechanical-Distributed instrument for Inertial sensing and Navigation (ODIN) instrument for advanced satellite motion tracking. ODIN is a gyro-free inertial navigation unit based on an array of linear optomechanical resonators.

Both SGR-S and ODIN instruments will fly on-board a single commercial ESPA-class mini satellite bus (235 kg).

1.4 Launch

The following table shows a high-level summary of the GRATTIS launch, which is detailed further in this section.

Table 1.4-1: GRATTIS Summary

Owner/Operator	University of Florida
Mission Name	Gravitational Reference Advanced Technology Test in Space (GRATTIS)
Mission Description	The GRATTIS technology demonstration will precisely measure the Earth's gravity to track water and ice movement
Number of Satellites	1
Launch Date	Q1 2027 (Notionally Feb 1, 2027)
Launch Vehicle and Launch Site	SpaceX Falcon 9 Transporter-19 Mission Vandenberg Air Force Base
Deployment Apogee (km)	510 - 590 km
Deployment Perigee (km)	510 - 590 km
Deployment Inclination (deg.)	Sun Synchronous Orbit (SSO) 97.7 deg for a 590 km SSO
Operational Apogee (km)	590 km
Operational Perigee (km)	590 km
Operational Inclination (deg.)	Sun Synchronous Orbit (SSO) 97.7 deg for a 590 km SSO
Disposal Apogee (km)	590 km
Disposal Perigee (km)	234 km
Disposal Inclination (deg.)	Sun Synchronous Orbit (SSO) 97.7 deg for a 590 km SSO

GRATTIS is slated fly on the SpaceX Transporter-19 mission to a 510 km – 590 km (circular) Sun Synchronous Orbit (SSO) in Q1 2027 (notionally Feb 1, 2027) from Vandenberg Air Force Base, CA. GRATTIS has communicated to SpaceX that the preferred drop off is 590 km. This orbit was selected for GRATTIS to provide near global coverage and consistent orbital geometry, allowing for precise, repeated measurements of the Earth's gravitational field utilizing the GRATTIS science instruments.

The total nominal operating lifetime of the spacecraft is 12 months if the spacecraft is dropped-off at the operational altitude of 590 km, and up to 13 months if the spacecraft is dropped off at 510 km (1 additional month of orbit raising from 510 km to 590 km).

The GRATTIS spacecraft (~230 kg wet) is designed to remain in orbit for the entirety of the technology demonstration and utilizes the APEX Space Aries commercial spacecraft bus.

Table 1.4-2: GRATTIS Mission Duration by Phase

Phase	Duration
Spacecraft Bus Commissioning	1 mo.
Orbit Raise (if required, 510 km to 590 km)	Up to 1 mo.
Payload Operations	5 mo.
Orbit Lowering (590 km to 234 km, 100 m/s dV)	Up to 6 mo.
Total	12 mo. (nominal) Up to 13 mo. if an orbit raise is required.

After the orbit lowering maneuver when spacecraft reaches 590 km x 234 km, it will take approximately 2 months for the spacecraft to re-enter atmospherically. Hence, the total Mission Lifetime to spacecraft demise is nominally 14 months, and up to 15 months if an orbit raise is required at the beginning of the mission.

GRATTIS is a stand-alone free-flyer and does not intend to interact or physically interfere with other operational spacecraft.

The GRATTIS spacecraft has a 3-axis attitude determination and control system (ADCS), and a 1-axis electric propulsion system. The Aries GNC subsystem utilizes a suite of sensors and actuators that allow for effective ADCS and momentum management. Sensors include star trackers, sun sensors, magnetometer, GNSS receiver, and IMU. The actuators include the reaction wheels and magnetic torque rods. The magnetic torque rods are used to desaturate the reaction wheels. Refer to Section 2.5 for detailed ADCS specifications. The ADCS will be activated after the spacecraft is deployed and a safe distance away from the launch vehicle, nulls any tip-off rates, and is utilized throughout the technology demonstration to point the spacecraft for communication activities, science operations, momentum management, and orbit raising/lowering maneuvers.

The GRATTIS spacecraft also has a Exoterra Iris 250 Hall-Effect Thruster (HET) with Xenon propellant. The prop system is not throttle-able, and is sized, with a safety factor of 2, to perform: orbit raise maneuvers (if required if orbit insertion is <590 km), and orbit lowering for atmospheric reentry at end of mission (EOM) to meet FCC <5-year lifetime requirements. See section 2.7 for detailed prop specifications.

Section 2: Spacecraft Description

2.1 Spacecraft Overview

GRATTIS utilizes an Aries commercial spacecraft bus from Apex Space. Aries is a standardized, ESPA-class minisatellite bus. Table 2.1-1 outlines GRATTIS generic attributes. Refer to Figure 2.1-1 for the spacecraft stowed and deployed configurations with key dimensions.

Table 2.1-1: GRATTIS Generic Attributes

SC Name	SC Qty	SC size (mm)	SC Mass (kg)
GRATTIS	1	0.861 m x 1.140 m x 0.947 m (stowed) 0.861 m x 2.493 m x 0.947 m (deployed)	230 kg (Initial – Wet) 210 kg (Final – Dry)

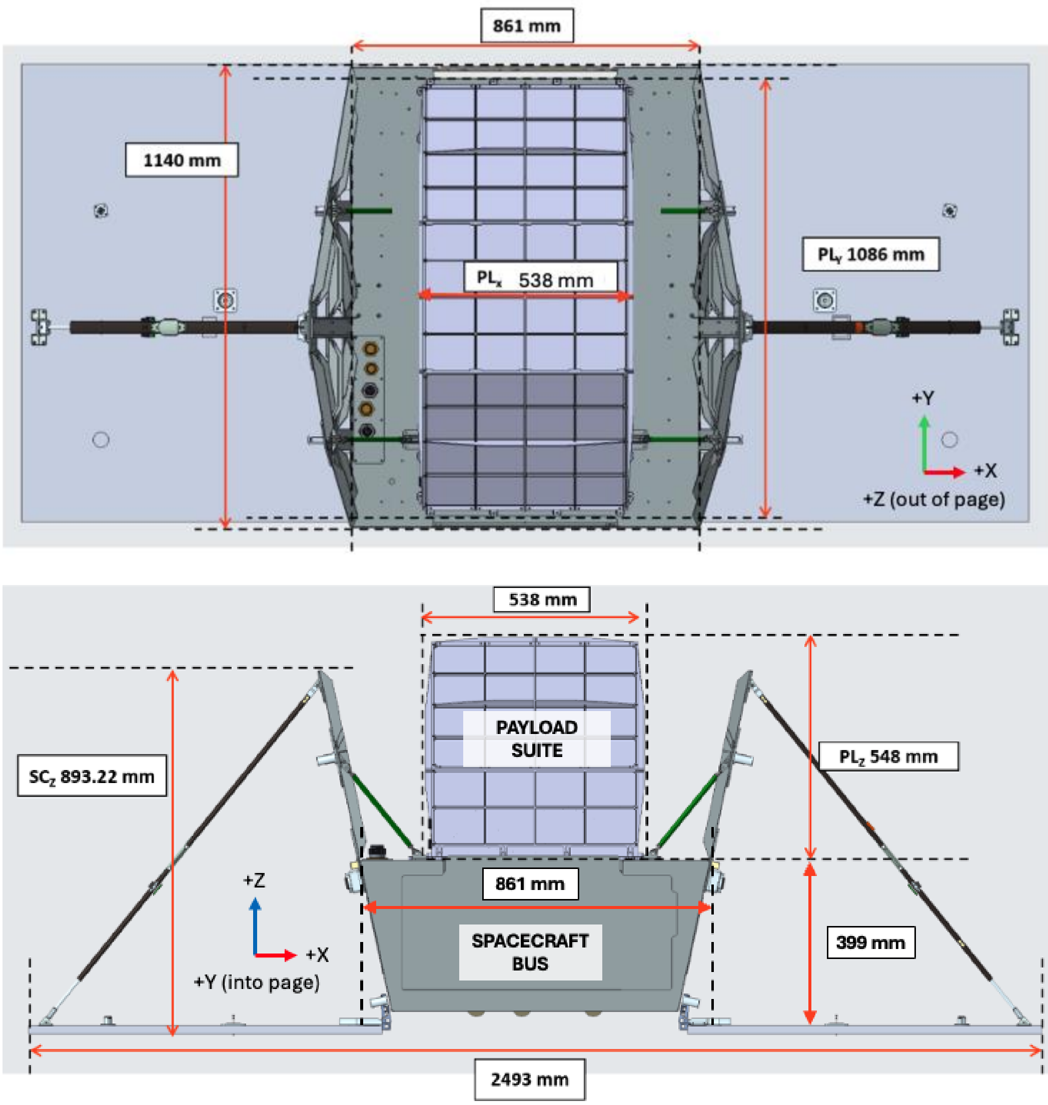

*Figure 2-1.1: GRATTIS Key Bus Dimensions. Top: Top view (deployed).
Bottom: Side view (deployed)*

A block diagram of the spacecraft bus is shown below, which details the individual subsystems.

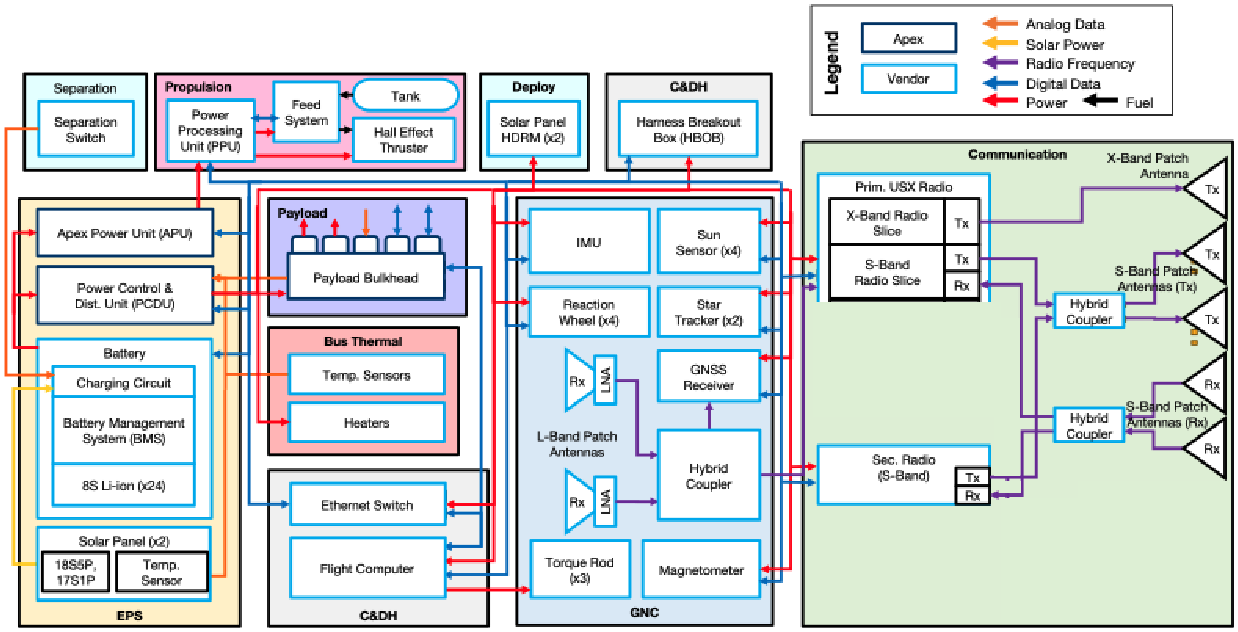

Figure 2.1-2 GRATTIS spacecraft bus block diagram

There are two primary science modes, inertial pointing and nadir pointing. In both cases the solar arrays are oriented to be coplanar with the orbit plane to minimize drag, while oriented to be pointed in the direction relative to the Sun that supports power generation.

The figures below illustrate the GRATTIS spacecraft w/ payload deck (thermal shield not shown) in the mission operation configurations.

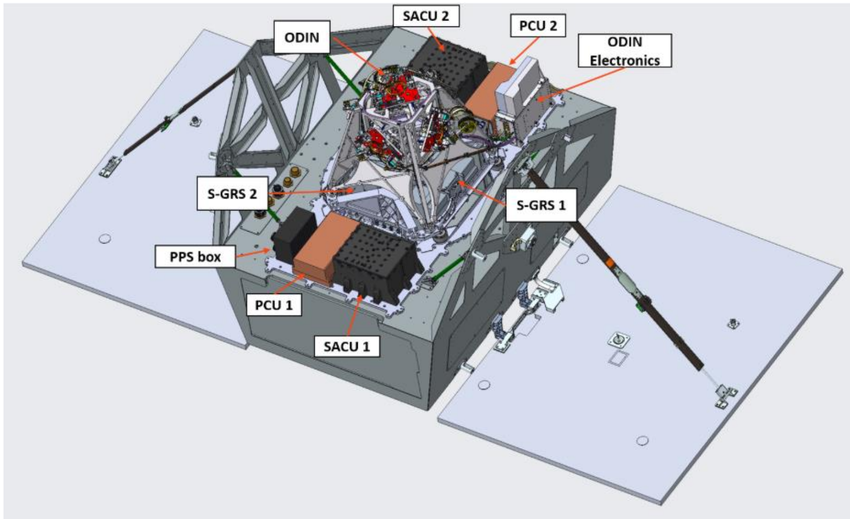

Figure 2.1-2 GRATTIS spacecraft payload configuration

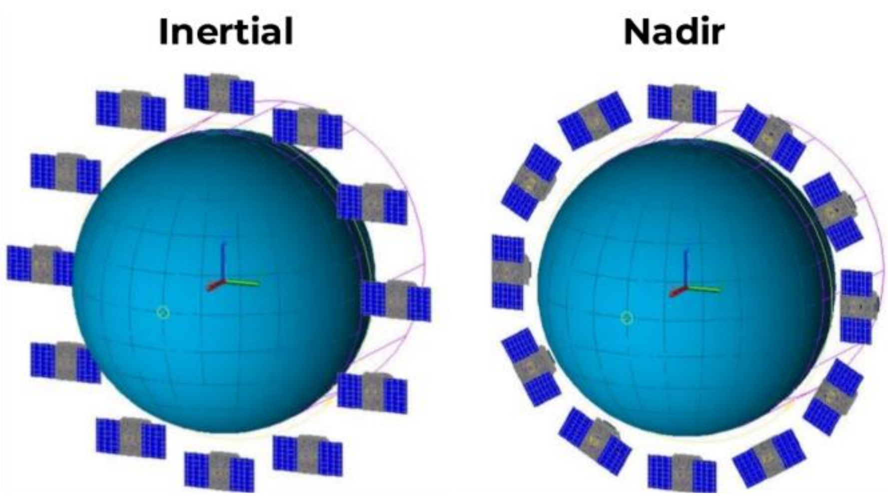

Figure 2.1-3 GRATTIS mission operation configurations

2.2 Structures

The bus structure is made of three Al 6061-T6 core components, shown below. The area between the Top and Base deck contains the primary spacecraft subsystems (e.g.: avionics, guidance & navigation system, attitude determination and control system, communication system, and electric power system).

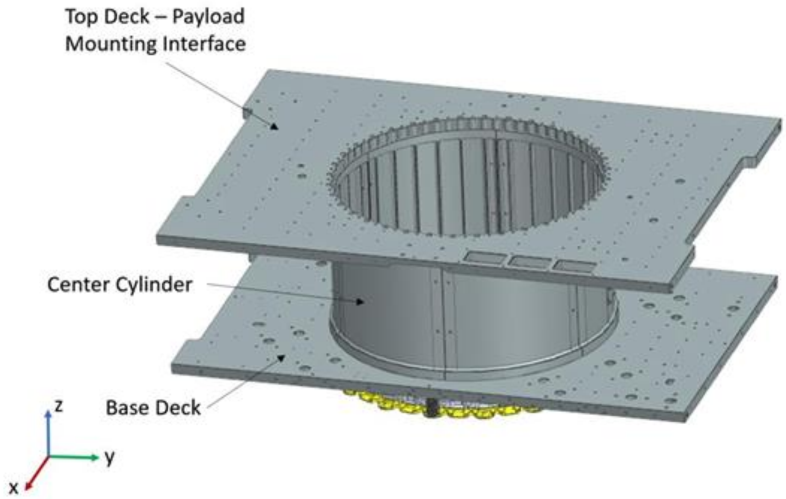

Figure 2.2-1: SC Primary Structure

The Center Cylinder acts as the primary structural bridge between the base and top-deck. Its 24-inch bolt pattern aligns directly with the LV separation mechanism. The interior of the Center Cylinder contains the propulsion system. See Section 7 for additional information regarding the propulsion system. The area between the plates contain the remainder of the spacecraft subsystems. The area in the middle of the Center Cylinder, biased on the -Z face, is the propulsion system, shown below.

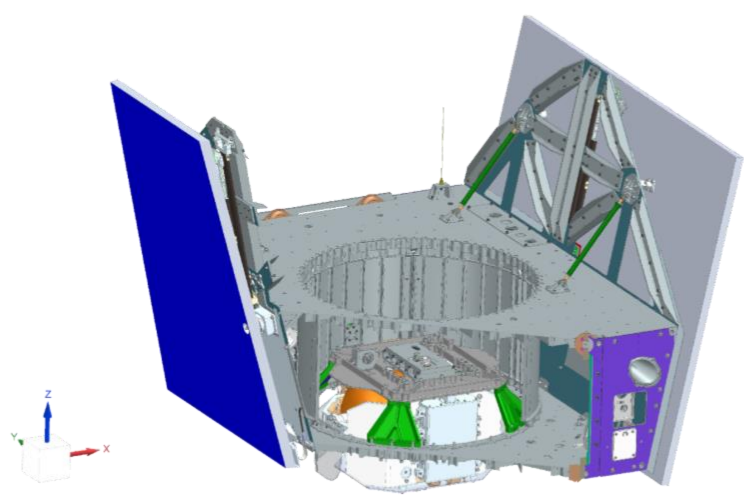

Figure 2.2-2: Cross-sectional view of SC detailing the propulsion system (stowed)

The Top deck primarily consists of the GRATTIS payload instruments, shown below. See Section 2.10 for additional detail for these payloads. The top deck also serves as the interface to thermally couple the payloads to rest of the spacecraft bus.

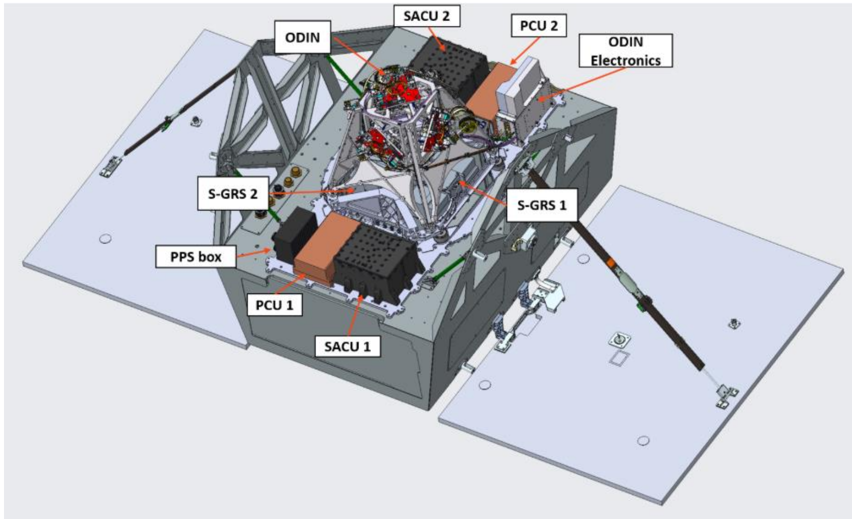

Figure 2.2-3: S-GRS & ODIN Payload Suite on SC Top Deck (deployed)

The Figure below illustrates the external spacecraft bus component locations, and Figure 2-8 illustrates the external spacecraft bus component locations. Note the payloads and propulsion system are not shown in these figures

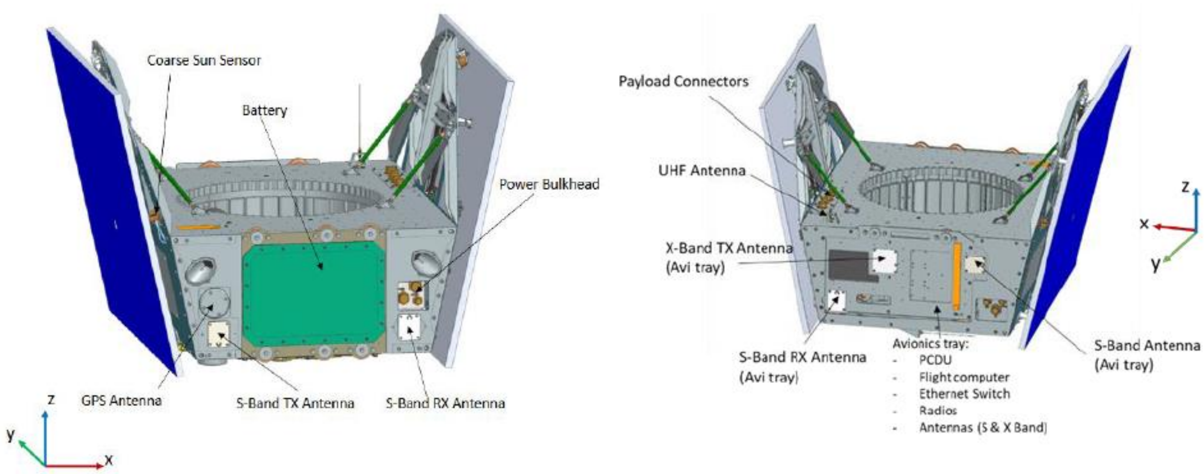

Figure 2.2-4: GRATTIS external illustration

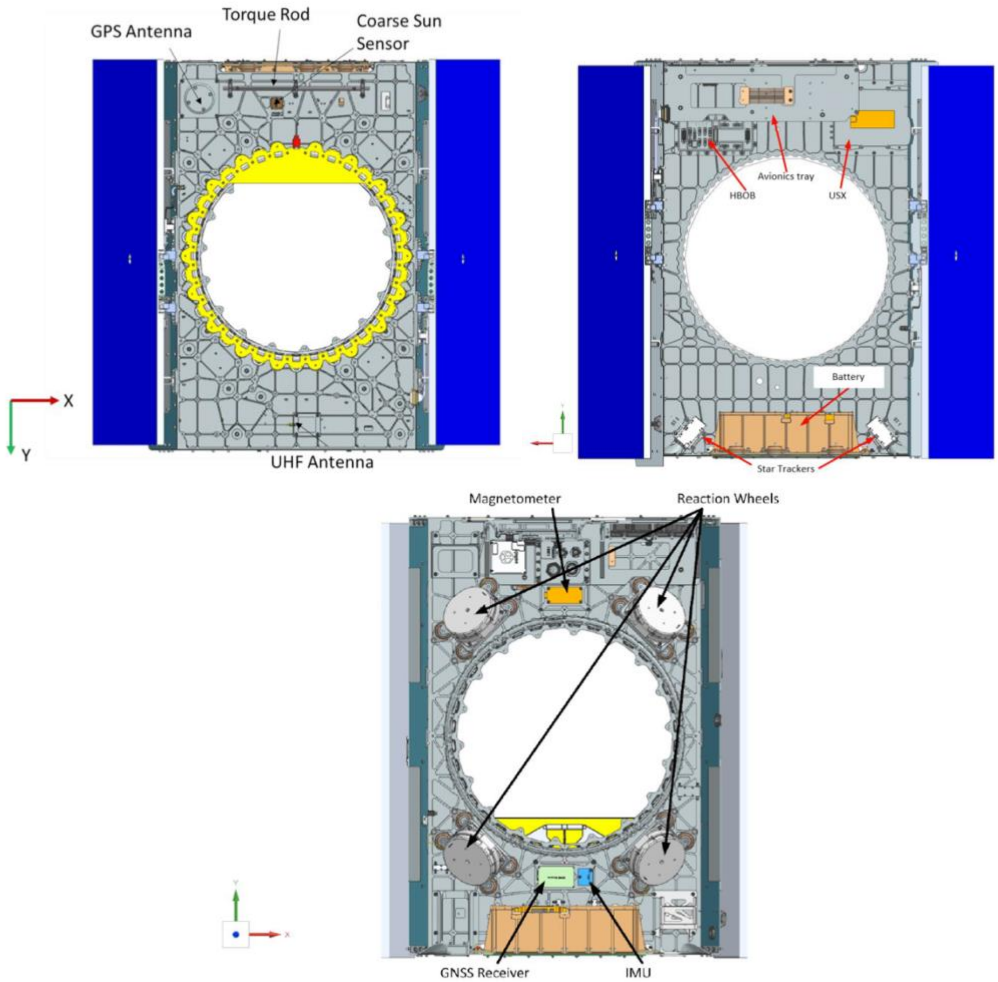

Figure 2.2-5: GRATTIS labeled external illustrations

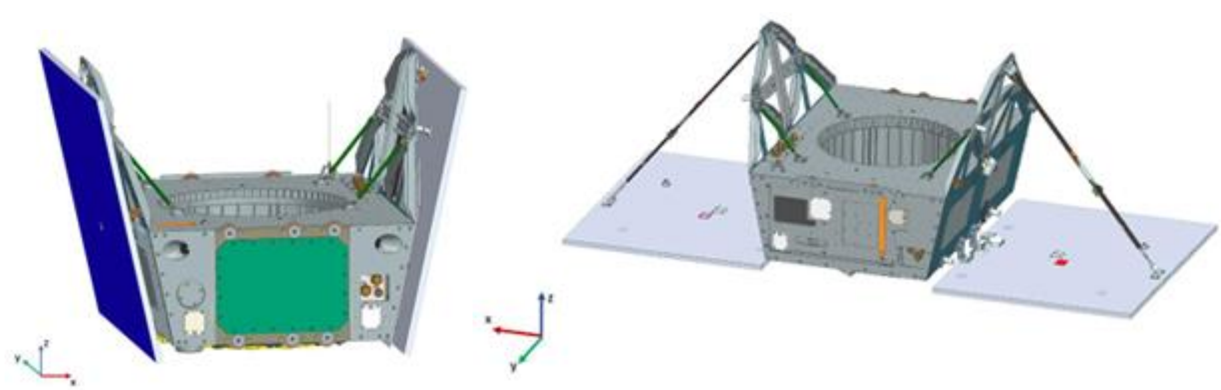

Figure 2.2-6: SC solar array stowed (left) and deployed (right) configurations

2.3 Avionics

The Aries Flight Computer is based off a multi-processor System on a Chip (SoC) that is constantly monitored via triplicated logic residing on a radiation qualified FPGA. Both the SoC and FPGA are housed on a single board computer which contains 4GB of volatile program memory, 128GB of non-volatile mass data storage NAND Flash, and 2x 256MB of NOR boot memory. The Flight Computer consists of multiple circuit cards that interface with the SBC, provide power conditioning and spacecraft physical interfaces. The Flight Computer provides 64GB non-volatile data storage capability for payload playback data.

The Flight Computer provides the following system level functions:

- Spacecraft command and data handling
- Payload data interfaces, data processing, storage and payload data downlink.
- Receives and processes spacecraft commands from the tracking, telemetry and command (TTC) transceivers.
- Gathers spacecraft telemetry from equipment, packages and provides the data to the TTC transceivers.
- Runs the GNC control software.
- Runs spacecraft Failure Detection, Identification, and Response (FDIR) software.
- Equipment power control
- Controls and transitions the spacecraft modes.
- Tracks and distributes spacecraft relative and absolute timing.

The Aries Software is a configurable and modular system that runs on an embedded Linux platform. The main flight software binary and other software components are started and stopped using the 'systemd' Linux program. The software runs on a System on Module that also has an FPGA component (ProASIC3)

Major software components include:

- Core FSW - based on MAX FSW, an existing framework with heritage across multiple Buses and mission types.
- Router - main interface between the radios and ground. Handles encryption and endpoint selection(which radio channel to use)
- Boot Selector - Flight software is broken into 4 partitions between 2 separate memory chips, where two partitions are golden images. The partition to boot into is selected by command sent during the previous boot cycle
- Watchdogs – to monitor Pro-asic 3, Battery, and Software for faulty behaviour
- Other: Timing, Sequencing, Fault Detection, Isolation, and Recovery (FDIR), Firecodes.

2.4 Electric Power System

The power system of the Apex Aries bus consists of 2 solar arrays, a battery and a Power Control and Distribution Unit (PCDU). The grounding architecture for the Apex Aries bus is a single point scheme, with the return being the PCDU. The PCDU also performs thermal control by reading and managing temperature sensors and heater circuits. These heater circuits are unconditioned power. Figure 2.4-1 details the power system block diagram.

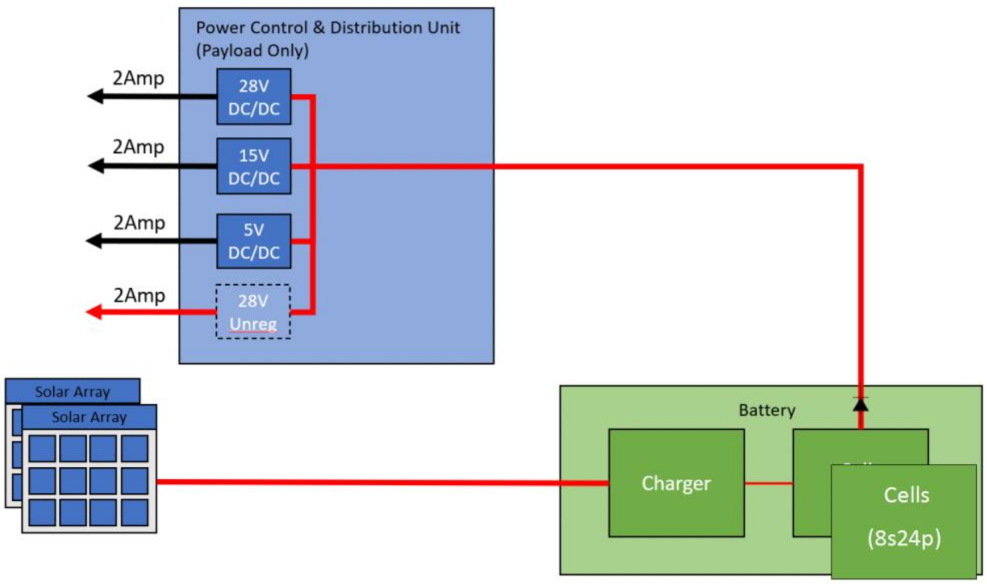

Figure 2.4-1: GRATTIS EPS Block Diagram

2.4.1 Battery

GRATTIS utilizes a Stellar Exploration Inc. 'High Current, High Discharge with Rev C Module' model battery system. The battery module consists of 160 Li-ion LG INR18650 MJ1 lithium salt batteries in a 20 parallel – 8 series configuration. The battery capacity is 1680 Whrs, and designed not to exceed a 20% depth of discharge (DOD) during nominal operations to maximize lifetime performance. Each 8 cell module is charged and sensed independently with a dedicated charger and Battery Management System (BMS). The battery has a 24 hr watchdog timer capable of power cycling the bus triggered by loss of communication between the Flight Computer and the battery controller. The also battery has integrated charging and MPPT. The battery has inhibits, and a FDIR (Fault Detection, Isolation, and Recovery), UVLO (Undervoltage Lockout), OVLO (Overvoltage Lockout), and Thermal Runaway Protections.

2.4.2 Solar Panels

The solar array utilizes a flight-proven design from Space Tech GmbH (STI) featuring triple-junction Gallium Arsenide solar cells (CTJLC half-pipe), with layout 18s5p+17s1p. The solar panels are a composite honeycomb structure.

Deployment of each solar array is mechanically and electrically independent but are actuated automatically in a sequential manner during the separation sequence upon boot up. Three components enable the deployment motion of each solar array: hold down release mechanism (HDRM), hinge, strut. The HDRM has burn-wire that when sufficiently powered, will release the solar array to move freely about the hinge-line. As the solar array deploys, the strut will extend until it reaches the end of motion as showed in Figure 2.4-2. The hinge does not have a locking mechanism; instead, the strut will passively lock in place as shown when it reaches the end of motion.

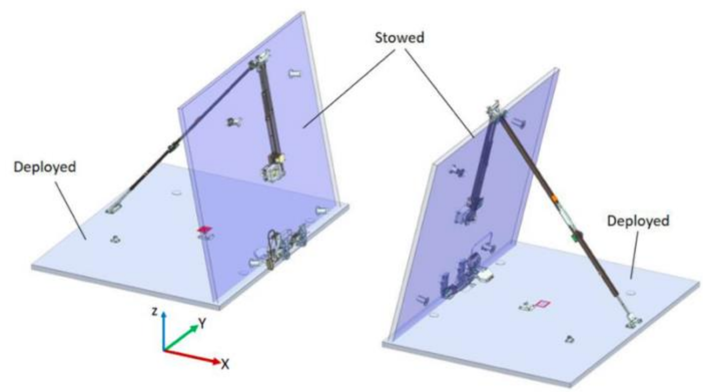

Figure 2.4-2: SC solar array stowed (left) and deployed (right) configurations

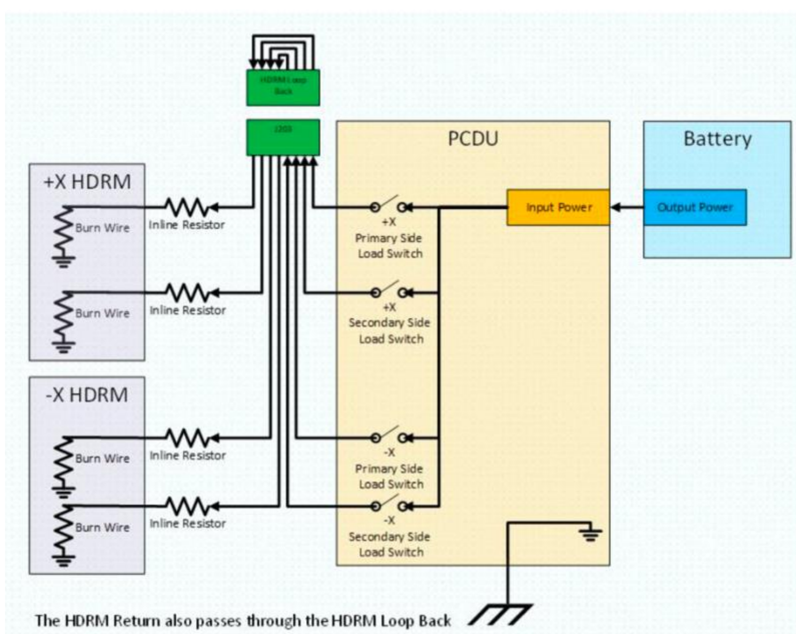

Figure 2.4-3: SC solar array HDRM diagram

2.4.3 Separation Signal

The separation signal is implemented in hardware by means of the Separation Switch acting as a loopback electrically connecting the battery to the PCDU via an array of MOSFETs, that triggers the bus to power on. The separation signal is designed to retain a 78-day encapsulation period.

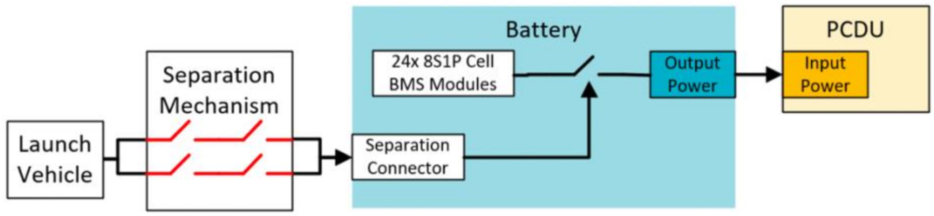

Figure 2.4-3: Separation Signal Power Inhibit Diagram

2.5 ADCS

The Aries GNC subsystem utilizes a suite of sensors and actuators that allow for effective 3-axis ADCS and momentum management, with Pointing knowledge < 0.06 deg (215 arcsec), 3-sigma, and Pointing control < 0.12 deg (430 arcsec), 3-sigma.

Sensors and Actuators include:

- Reaction Wheels (4x) – 1 Nms RW-1.0
- Inertial Measurement Unit (1x) – MEMS-based gyro, resolution: 0.22 deg/h
- Star Tracker (2x) – High accuracy, robust, copact star tracker for attitude determination, lost-in-space algorithm
- Magnetic Torque Rod (3x) – 30.4 Am2 dipole moment, wire wound copper
- Magnetometer (1x) – Tri-axial sensor head, with three AMR sensors, and supporting electronics. Sensitivity better than 9nT and range of +60,000nT to -60,000nT

The Guidance and Navigation Control (GNC) subsystem leverages the integration of a simulation and dynamics system, called ODySSy, within the MAX FSW. The SC consists of a Global Positioning System (GPS) system to provide time, position, velocity, and acceleration.

2.6 Comm System

The communication system contains a primary Anysignal USX radio that covers 2 frequencies (S-Band, and X-Band) and a block-redundant Satlab SRS S-Band radio. A block diagram of the GRATTIS communication system is shown in Figure 2.6-1. S-band is fully redundant with two TX/RX antennas and hybrid couplers that cross-strap these to two sets of S-band TX and RX radio slices.

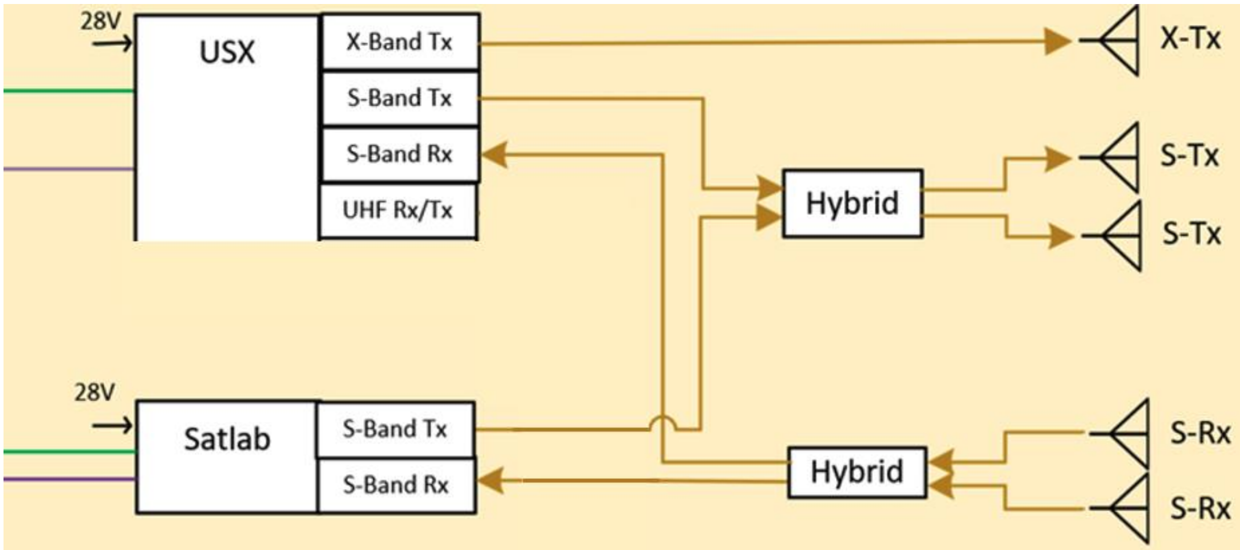

Figure 2.6-1: GRATTIS Comm block diagram

S- and X-Band communications are required to bifurcate critical command functions from high-volume science data transfer. The S-Band link is essential for robust Telemetry, Tracking, and Command (TT&C), providing a reliable, low-rate interface for continuous spacecraft health monitoring and emergency recovery. Simultaneously, the X-Band downlink is essential for the high-resolution gravitational sensing suite, which generates a significant daily data volume of 9,500 Mbit/day. Utilizing the X-Band spectrum allows for the efficient offloading of this large science payload within the constraints of two daily ground passes, ensuring that the S-Band remains clear for vital flight-safety operations. The spacecraft is only capable of storing up to two days of science data to accommodate one missed ground contact.

Two ground station passes per day are planned, leveraging the Leaf Space commercial ground station network (licensed separately from the spacecraft). Mission operations will be conducted by University of Florida (UF) through their Mission Operations Center (MOC).

The S-Band receiver stays on at all times throughout the demonstration. The S- and X-Band transmitters only transmit in response to ground command. All uplink commands are secured utilizing AES-256 encryption standards to prevent unauthorized access and ensures the integrity of the command link between the UF MOC and the spacecraft. Both S-band and X-band transmitters are all capable of ceasing transmissions when commanded, as-needed.

Antenna locations are shown below:

- (1) X-Band antenna on the +Y face
- (1) S-Band antenna (TX) on the +Y face
- (1) S-Band antenna (RX) on the +Y face
- (1) S-Band antenna (TX) on the -Y face
- (1) S-Band antenna (RX) on the -Y face
- (1) Tri-Band GNSS Antenna (Rx) on the +Y face
- (1) Tri-Band GNSS Antennas (Rx) on the -Y face

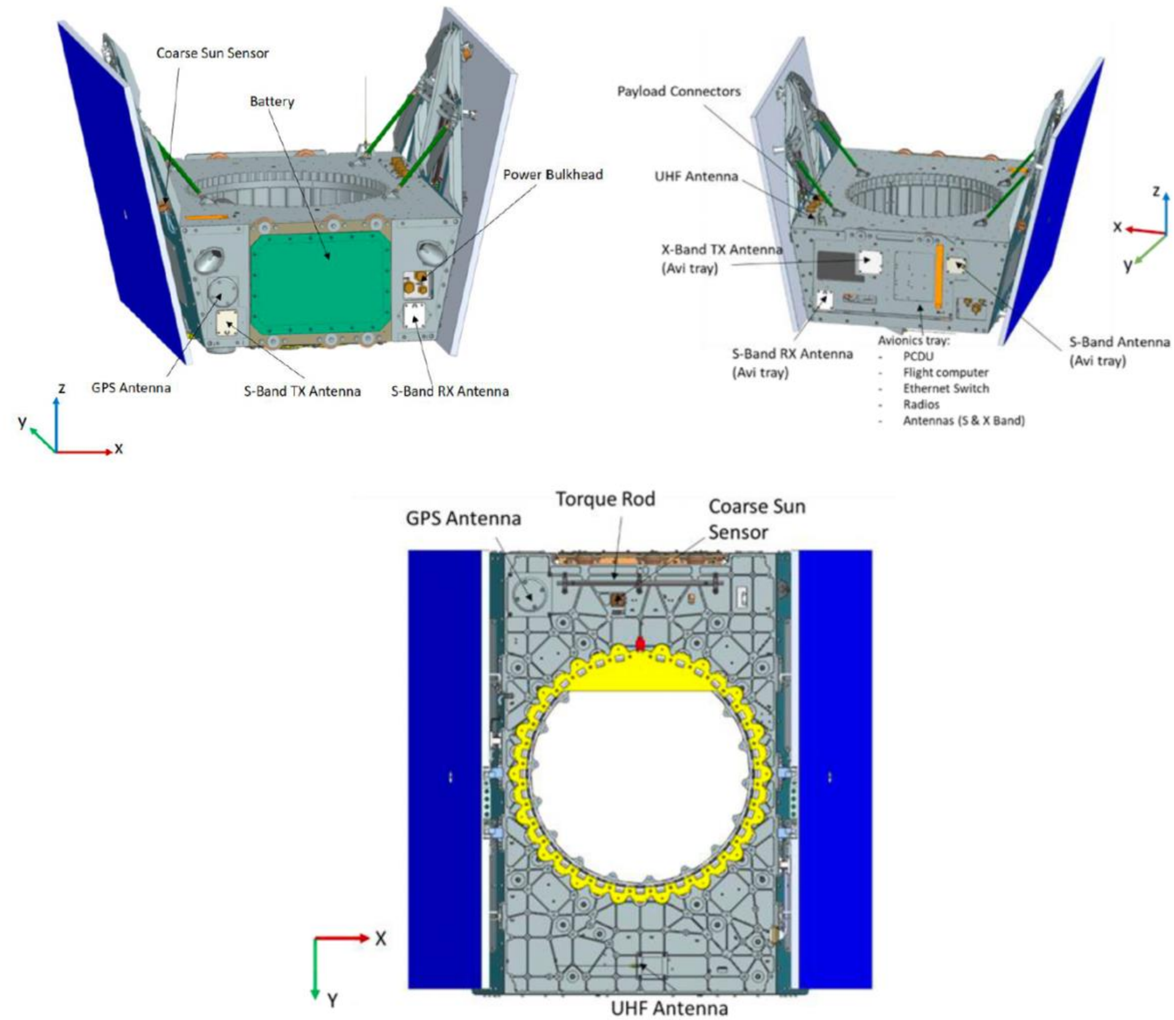

Figure 2.6-2: GRATTIS Comm Antenna locations

2.7 Propulsion System

GRATTIS will utilize a propulsion system sized to reduce the SC altitude and re-enter per FCC guidelines (<5 years). The SC electric propulsion system consists of a Exoterra Iris 250 Hall-Effect Thruster (HET), Power Processing Unit (PPU), Valve Control Unit (VCU), Propellant Distribution System (PDS), propellant tank, and Xenon as Fuel. A block Diagram of the Prop system is shown below. GRATTIS will allocate 200 m/s of delta-v (3.80 kg of xenon propellant) for the end-of-life disposal maneuver, providing a safety factor of 2 relative to the nominal 100 m/s maneuver requirement.

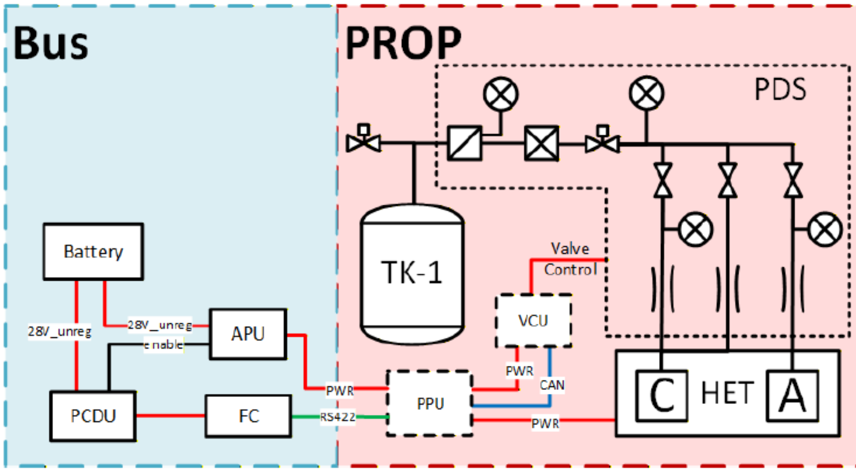

Figure 2.7-1: Propulsion Block Diagram

The thruster is designed to operate in a power range of 350-400W at the thruster input but is limited to 380W due to PPU setpoints. The thruster magnetic circuit has been specifically optimized for extended lifetime operating on Xenon, and the truster is not throttle-able. The system uses instant start, center mounted cathode. The Thruster has been qualified to 400 kN-s (~5000 hours). The propulsion is located in the center-body -Z face of the SC, shown below.

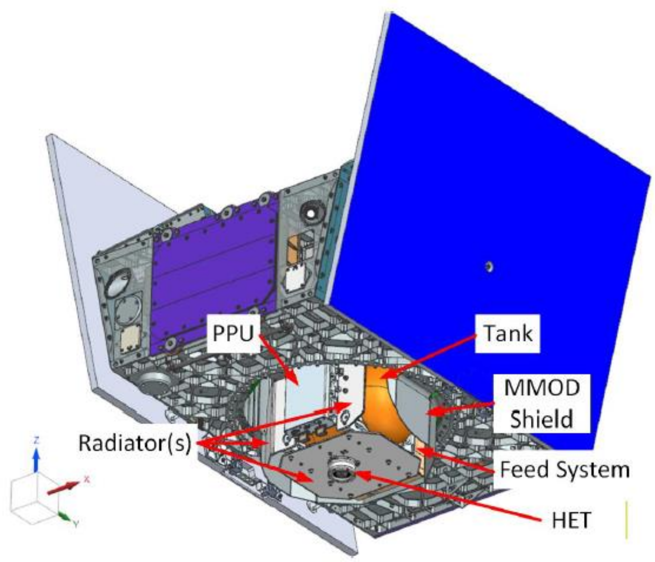

Figure 2.7-2: SC view detailing propulsion system

The propellant tank is composite overwrapped tank construction with Al6061-T6 liner that operates at a nominal propellant load corresponding to 1021.5 psi at 40°C, which provides a substantial 66.0% margin below the tank's Maximum Expected Operating Pressure (MEOP) design limit of 3000 psi and a burst factor of 3.73 (burst pressure of 11,190 psi). The propellant tank is DOT-certified (E-11194, special permit). Approximately 19 kg of xenon on-board for the GRATTIS mission. The propulsion system includes: active temperature monitoring and control, dual-inhibit latching valves preventing unintended propellant release, high-pressure transducers for continuous pressure monitoring, radiation-tolerant fault monitoring systems (TID 30 krad, SEE >41 MeV·cm²/mg), and all-welded interfaces between pressurized volumes eliminating leak paths, components rated to the full system MEOP with minimal internal leakage ($<2.4\times10^{-4}$ sccs GHe at 3000 psia). The electric propulsion system's extensive flight heritage on multiple LEO programs and comprehensive qualification campaign further validate the design's reliability.

2.8 Thermal System

The Apex Aries Bus Thermal Control Subsystem is designed to keep all bus and payload components within acceptance temperature ranges. Components are temperature controlled passively using Multi-Layer Insulation (MLI) and radiators. The bus PCDU controls active heater circuits. A thermal block diagram showing the locations of bus temperature sensors and heaters is shown below. The payload can reject up to 80W (steady state) into the bus without causing any bus thermal issues.

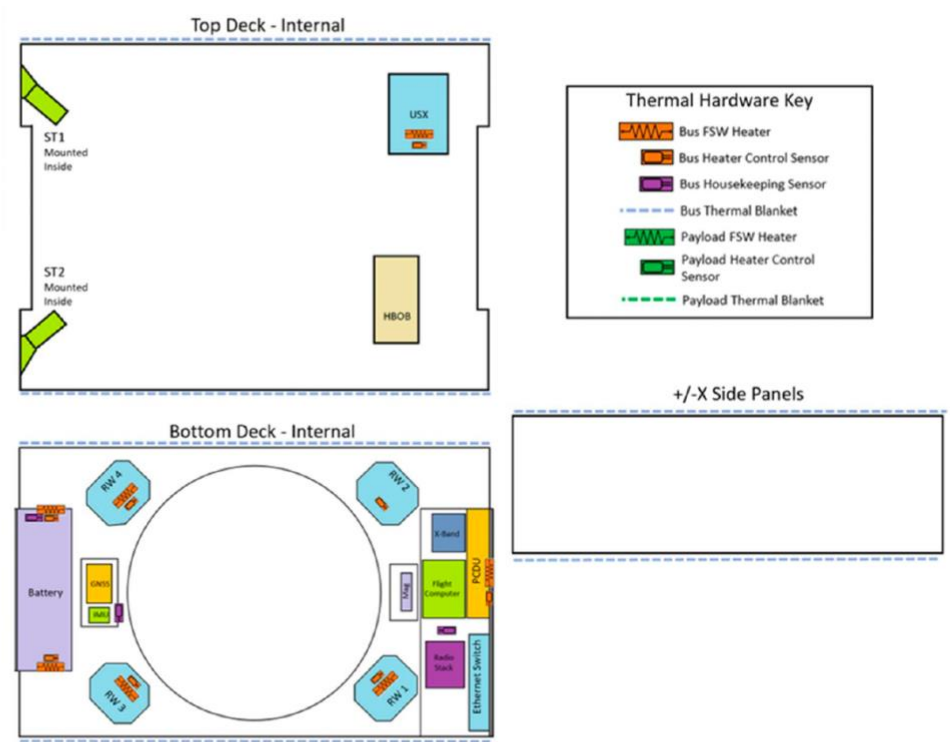

Figure 2.8-1: GRATTIS Bus Thermal Block Diagram

2.9 Other

Aside from the Xenon Propellant, Propulsion tank, Batteries, solar panel release mechanisms as outlined in this section, there are no other liquids, gases, pressure vessels planned to be on board GRATTIS. The spacecraft is not capable of in-space transfers. There are no other range safety or other pyrotechnic devices, electrical generation and storage systems, or other sources of stored energy. There are no radioactive materials on board.

2.10 Payloads

The primary objective of GRATTIS is to test the Simplified Gravitational Reference Sensor (S-GRS). This next generation inertial sensor is significantly more sensitive than those used on missions like GRACE-FO (Gravity Recovery and Climate Experiment Follow-On).

The secondary objective is to test the Optomechanical-Distributed instrument for Inertial sensing and Navigation (ODIN) instrument for advanced satellite motion tracking. ODIN is a gyro-free inertial navigation unit based on an array of linear optomechanical resonators.

Both payloads are located on the spacecraft top-deck, shown below

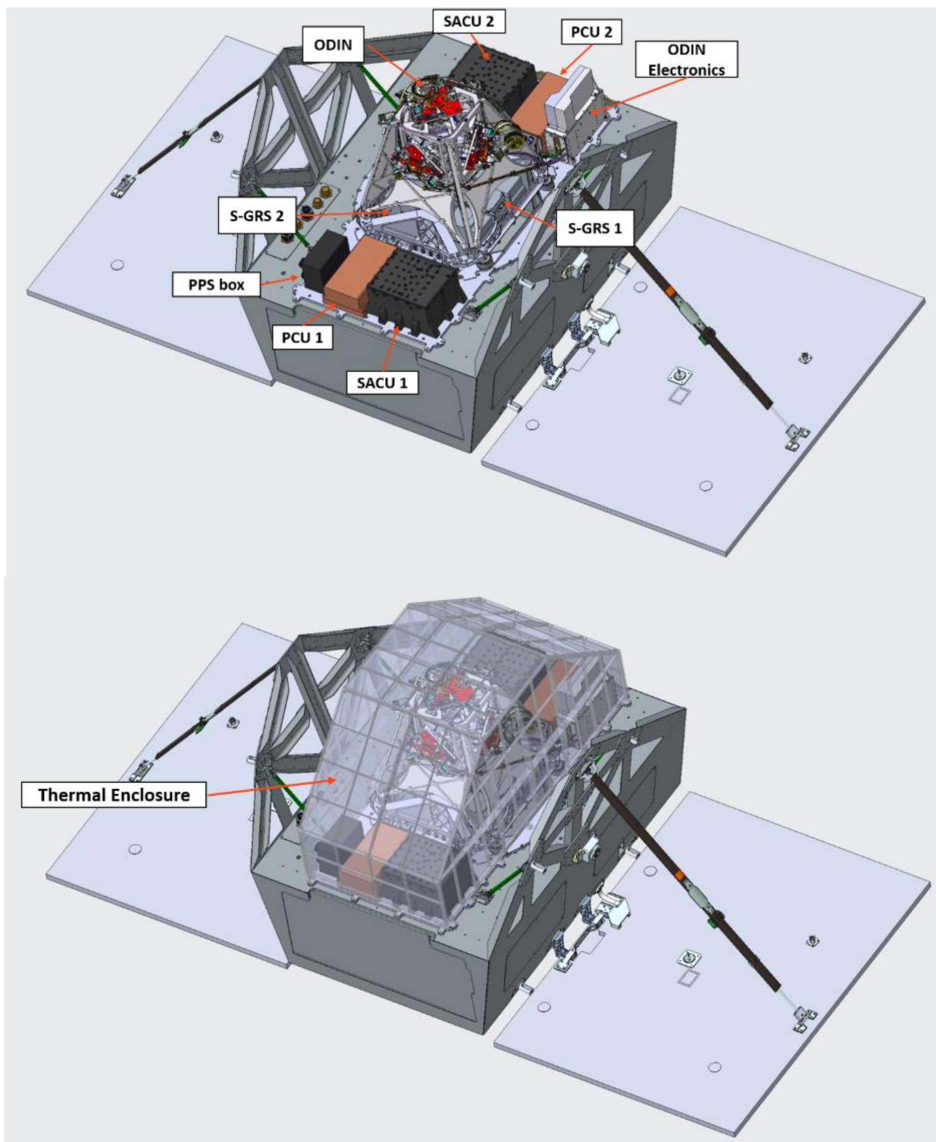

Figure 2.10-1: Payload locations on spacecraft

2.10.1 SGRS Instrument

The Simplified Gravitational Reference Sensor (S-GRS) is an ultra-precise inertial sensor designed to isolate a spacecraft's motion caused purely by Earth's gravity from "noise" like atmospheric drag or solar radiation pressure. GRATTIS will fly two identical S-GRS (two separate test masses) each housed within its own sensor head. Since the S-GRS is highly electrostatically sensitive, the design requires specific materials to minimize the electrostatic noise. Figure 2.10-2 shows an exploded view of a S-GRS.

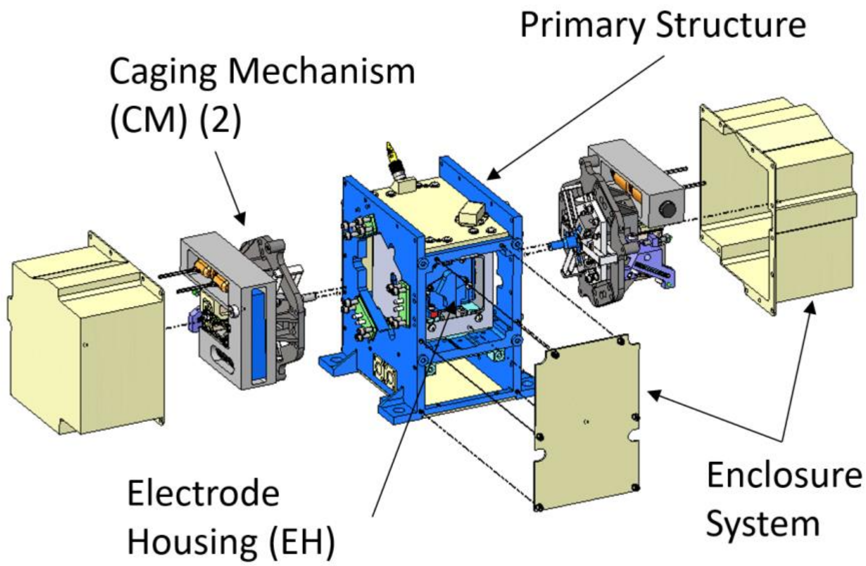

Figure 2.10-2: S-GRS Sensor Head containing the test mass, electrode housing, Caging and Release Mechanism, and Enclosure

The S-GRS frame and EMI shields are Titanium (6 Al-4 V), Test Mass is Tungsten, the electrode housing and cover plates are molybdenum, the optics spacers are sapphire, and the frangibolt to uncage the test masses are nickel. The S-GRS assembly measures 256 mm x 170 mm 155mm, and the Tungsten Test Mass housed within the S-GRS measures 30 mm x 30 mm x 30 mm.

The SGRS accelerometer test masses (2) are made from tungsten. Tungsten is required for its high density and low magnetic susceptibility and is central to maintaining low sensor noise. This is required to meet the scientific goals of the mission.

The GRATTIS payload mounting structure is made from titanium. This structure must maintain the relative orientations of the SGRS and ODIN sensors over the operating temperature range. Titanium is required for its low coefficient of thermal expansion to meet the science goals of the mission.

The SGRS electrode mounting plates are made from molybdenum. Maintenance of electrode plate alignment and low thermal gradients across the sensor are required to meet the science goals of the mission. Molybdenum is used for its high thermal conductivity and low coefficient of thermal expansion.

All other elements of the S-GRS are primarily Aluminum (housing), Steel (caging mechanism) Cu alloy (heat sink and DC-to-DC converters), and FR4 (PCBs). Refer to the detailed ODAR Inputs and GRATTIS Mission Description document for additional details regarding the material and functionality/necessity of each of these components.

2.10.2 ODIN Instrument

The Optomechanical-Distributed instrument for Inertial sensing and Navigation (ODIN) is a high-precision, low-cost secondary payload designed to complement the S-GRS by providing redundant, high-bandwidth measurements of a satellite's linear and angular motion. Unlike traditional heavy accelerometers, ODIN utilizes an array of six compact sensor subassemblies distributed across the spacecraft to capture motion in all three dimensions simultaneously. Figure 2.10-3 shows the ODIN assembly.

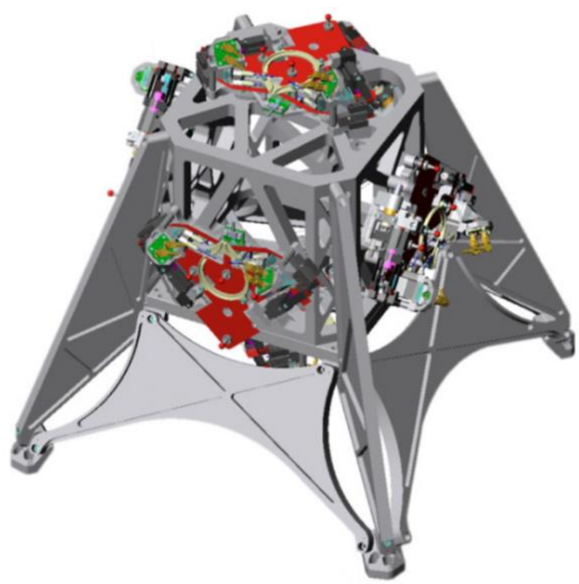

Figure 2.10-3: ODIN Optomechanical Sensor Package

The ODIN resonator mounts and optic mounts are Titanium, and the optics are Fused Silica, Sapphire, H-LAK54, and n-BK7. The ODIN optic mounts do not exceed 8 mm x 20 mm 28 mm. Similarly to SGRS, the mounts must maintain the relative orientations of the SGRS and ODIN sensors over the operating temperature range. Titanium is required for its low coefficient of thermal expansion to meet the science goals of the mission.

All other elements of the ODIN are primarily Steel or Aluminum 7075-T6 (housing), Cu alloy (connectors), polycarbonate (housing and fiber tracks), and FR4 (PCBs). Refer to the detailed ODAR Inputs and GRATTIS Mission Description document for additional details regarding the material and functionality/necessity of each of these components.

Section 3: Assessment of Spacecraft Debris Released during Normal Operations

3.1 Spacecraft Debris Released

There is no intended release of objects from GRATTIS.

3.2 Assessment of compliance with Requirements 4.3-1 and 4.3-2

The table below outlines the GRATTTIS compliance with Requirements 4.3-1 and 4.3-2.

Requirement	Description	GRATTIS Compliance
4.3-1a	Planned debris release in LEO shall be limited to a maximum orbital lifetime of 25 years	N/A No planned debris release
4.3-1b	Planned debris release in LEO total object-time product shall be less than 100 object-years	N/A No planned debris release
4.3-2	Planned debris in GEO shall be left in orbits within 25 years after release exceeding GEO	N/A No planned debris release

Section 4: Assessment of Intentional Breakups and Potential for Explosions

A comprehensive assessment of the spacecraft design and operational profile was conducted to identify potential failure modes that could lead to accidental breakup during deployment and mission operations. The potential causes are categorized and addressed in this section.

4.1 Conjunction with LV or Other Spacecraft

4.1.1 Conjunction upon Deployment

Given there are multiple spacecraft that will be deployed from the SpaceX Transporter launch vehicle, there is a low risk that the GRATTIS spacecraft conjuncts with another spacecraft being deployed from the same vehicle.

Mitigations: SpaceX will deploy the spacecraft at timed intervals depending on the size and estimation. The GRATTIS spacecraft will wait 40 min, as required by the LV, before turning on the spacecraft enabling active attitude control and deploying solar panels, to ensure a safe distance has been established from the LV and other spacecraft.

4.1.2 Conjunction during Mission Operations

Given there are many active satellites and known debris in the 510-590 km SSO region, there is low-medium risk that the GRATTIS spacecraft conjuncts with other active satellites or known debris during on-orbit operations.

Mitigations: GRATTIS is an ESPA-class minisatellite (230 kg) operating in LEO and will be actively tracked. Prior to deployment, GRATTIS will be registered with the 18th Space Defense Squadron (or successor entity). The GRATTIS mission operations team will share initial deployment ephemeris received from the launch vehicle provider, SpaceX, to the 18th Space Defense Squadron (18 SDS to assist in identifying GRATTIS after separation and establish communication. Early extended ground station passes will be conducted to account for any uncertainties. Once ground communication is established, GRATTIS GPS information will be downlinked, and the spacecraft is commanded to activate its UHF beacon, which will then transmit GPS data for regular updates. The GRATTIS Mission Operations team will share all updated GPS information from the spacecraft in coordination with 18 SDS to confirm GRATTIS' NORAD ID.

GRATTIS will maintain responsible space operations by coordinating with the 18 SDS for conjunction assessments and operational debris avoidance maneuvers throughout the mission to avoid in orbit collisions.

Additionally, prior to propulsive maneuvers, the GRATTIS mission operations team will coordinate with 18 SDS, sharing the Maneuver Plans through the Space-Track.org portal. The Maneuver Plan includes, but not limited to owner/operator, ephemeris, start/end maneuver, delta-V, maneuver type, and 24/7 point of contact, to ensure the safety an accuracy of the maneuver.

4.2 Explosion Risk and End of Life Passivation

4.2.1 Explosion Risk & Stored Energy Risks

The spacecraft contains two credible primary sources of accidental explosion: 1) the spacecraft batteries and 2) the propulsion pressure vessel.

Spacecraft Batteries

For spacecraft lithium-ion batteries, the sources of accidental explosions are: 1) thermal runaway, which can occur from thermal runaway propagation, manufacturing defects, overcharging or over-discharging, physical damage, external heating, and internal electrical failures.

GRATTIS mitigations to these explosion risks include: Thermal propagation barriers (aluminum separators between cell packs), active thermal management, manufacturer cell screening for manufacturing defects, proper venting systems for battery effluents, over-charge/over-discharge protection circuits, and end-of-life discharge to zero energy state for passivation. Given the mitigations in place, the probability of battery explosion is very low given heritage and design to meet SpaceX Transporter rideshare safety requirements, and short orbital LEO lifetime.

The spacecraft's lithium-ion batteries represent the highest debris generation risk through thermal runaway events. If a battery cell experiences internal short circuit, overcharging, physical damage from debris impact, or manufacturing defects, it can undergo thermal runaway where stored chemical energy converts to intense heat, producing a "blow torch" effect that propagates to adjacent cells. This exothermic chain reaction generates sufficient pressure and heat to rupture battery containment, potentially fragmenting the spacecraft structure and creating debris.

Propulsion System - Pressure Vessel

For the spacecraft propulsion system, the sources of accidental explosions are: 1) Pressure vessel rupture from external impact (causing rapid depressurization or rupture of the pressurized xenon tank, resulting in gas venting rather than an energetic explosion due to xenon's inert nature), 2) Manufacturing defects in COPV which could propagate over time under pressure cycling, potentially leading to structural failure, 3) Extreme thermal cycling could degrade the composite overwrap adhesion or cause stress concentration at the liner-composite interface, residual pressure hazard tank is not properly vented at end-of-life, and long-term material degradation reducing structural margins and increasing rupture risk if pressure remains.

GRATTIS mitigations to these explosion risks include: the propellant tank operates at a nominal propellant load corresponding to 1021.5 psi at 40°C, which provides a substantial 66.0% margin below the tank's Maximum Expected Operating Pressure (MEOP) design limit of 3000 psi and a burst factor of 3.73 (burst pressure of 11,190 psi); the propellant tank is DOT-certified (E-11194, special permit) composite overwrapped tank construction with Al6061-T6 liner provides robust structural integrity against rupture; and xenon is a chemically inert noble gas that cannot participate in combustion or energetic chemical reactions, eliminating the primary explosion mechanism present in traditional chemical propulsion systems. Multiple design safeguards further mitigate failure risks, including: active temperature monitoring and control, dual-inhibit latching valves preventing unintended propellant release, high-pressure transducers for continuous pressure monitoring, radiation-tolerant fault monitoring systems (TID 30 krad, SEE >41 MeV·cm^2/mg), and all-welded interfaces between pressurized volumes eliminating leak paths, components rated to the full system MEOP with minimal internal leakage ($<2.4\times10^{-4}$ sccs GHe at 3000 psia). The electric propulsion system's extensive flight heritage on multiple LEO programs and comprehensive qualification campaign further validate the design's reliability.

Any pressure vessel failure would manifest as rapid gas release rather than detonation, with the primary concern being debris generation from structural fragmentation rather than explosive energy release. Proper passivation through complete xenon venting at end-of-life eliminates the residual pressure hazard and reduces post-mission explosion risk to negligible levels.

The xenon electric propulsion system stores pressurized xenon gas in a 10.1L Carbon Overwrapped Pressure Vessel (COPV) at a maximum expected operating pressure of 3000 psi, representing stored pressure energy of approximately 2.1 MJ. If the COPV experiences catastrophic structural failure from micrometeoroid impact, manufacturing defect propagation, or material degradation, the rapid conversion of pressure energy to kinetic energy during depressurization could fragment the tank structure and adjacent spacecraft components, generating orbital debris. However, xenon's chemical inertness means no additional energy is released through combustion or chemical reactions. The COPV design incorporates a 66%

pressure margin below design limits and a burst factor of 3.73, reducing rupture probability to <<10^{-6} during normal operations.

4.2.2 Passivation

To prevent debris generation, GRATTIS incorporates end-of-life passivation through complete xenon venting eliminating residual pressure energy and preventing post-mission fragmentation from delayed rupture events.

To prevent debris generation, GRATTIS incorporates thermal propagation barriers (aluminum separators), steel containment tubes, active thermal management, over-charge/over-discharge protection circuits, end-of-life battery discharge to zero energy state (all systems powered off), and solar array cut-off for passivation.

GRATTIS will remove stored energy at the spacecraft's end of life by:

a) depleting residual fuel and leaving all fuel line valves open,
b) by venting any pressurized system,
c) leaving all batteries in a permanent discharge state,
d) and removing any remaining source of stored energy.

4.3 Assessment of compliance with Requirements 4.4-1 through 4.4-4

The table below outlines the GRATTIS compliance with Requirements 4.4-1 through 4.4-4.

Requirement	Description	GRATTIS Compliance
4.4-1	Limiting the risk to other space systems from accidental explosions during deployment and mission operations while in orbit about Earth	Compliant
4.4-2	Design for passivation after completion of mission operations while in orbit about Earth	Compliant
4.4-3	Limiting the long-term risk to other space systems from planned breakups for Earth	N/A No planned breakups
4.4-4	Limiting the short-term risk to other space systems from planned breakups for Earth orbital missions	N/A No planned breakups

Section 5: Assessment of Spacecraft Potential for On-Orbit Collisions

5.1 Limiting Debris Generated by Collisions with Large Objects

NASA DAS 3.2.7 was utilized to calculate the spacecraft probability of collision with space objects larger than 10 cm in diameter during the orbital lifetime of the spacecraft, assuming no collision avoidance capability.

A summary of the DAS inputs and results are shown in the Table below. An upper-bound of 590 km orbit altitude was utilized to capture the worst-case scenario and operational altitude for a SpaceX notional 510 - 590 km launch. Although the GRATTIS orbital mission operations is 12-13 months, a FCC maximum allowable orbital lifetime of 5 years was used as a worst-case scenario for this analysis. Two configurations were analyzed, the first was a worst-case scenario as-deployed from the LV with the spacecraft solar panels stowed, and a spacecraft wet mass; and the second was the nominal scenario at end-of-mission with spacecraft solar panels deployed, and a spacecraft dry mass. Utilizing guidance from NASA NASA-STD-8719.14C, the Mean cross-sectional area (CSA) was calculated for each configuration. Details regarding the Mean CSA calculations can be found in Section 6.2.

Both worst-case and nominal case configurations result in a probability of collision of 3.87E-05 and 6.02E-05, respectively, which both meet the requirement of <0.001 (1 in 1,000).

Table 5-1: GRATTIS Debris Generated by Collsiions with Large Objects Inputs & Results

GRATTIS	Parameter	Value	Unit
Orbit	Apogee x Perigee Inclination	590 x 590 97.7	km deg
Lifetime	Orbital Lifetime	5	years
As-deployed from LV Configuration	Mean CSA Convex	1.335	m^2
	Final Mass (wet)	230	kg
	Area-to-Mass	0.0058	m^2/kg
	Probability of collision	**1.6236E-05**	**10^x**
	Pc < 0.001?	*COMPLIANT*	
End-of-mission Configuration	Mean CSA non-Convex	2.265	m^2
	Final Mass (dry)	210	kg
	Area-to-Mass	0.0108	m^2/kg
	Probability of collision	**2.2207E-05**	**10^x**
	Pc < 0.001?	*COMPLIANT*	

5.2 Limiting Debris Generated by Collisions with Small Objects

DAS 3.2.7 was utilized to calculate the spacecraft probability of collision with space objects, including orbital debris and meteoroids, of sufficient size to prevent postmission disposal.

Requirement 4.5-2 was run utilizing two different configurations for GRATTIS' two different pointing modes: Inertial and Nadir, as shown in Figure 5.2-1. A transformation matrix of the spacecraft coordinates to the DAS UVW definitions are shown in Table 5.1-2.

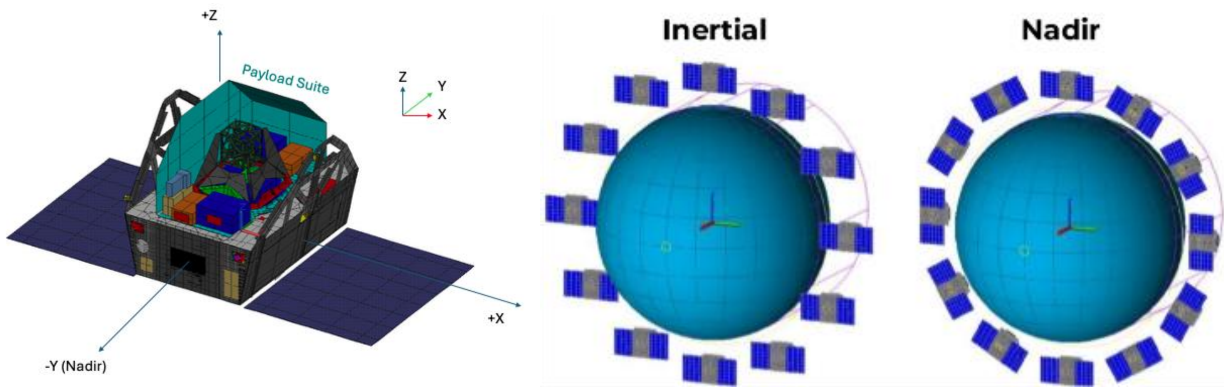

Figure 5.2-1 GRATTIS Spacecraft coordinates with Inertial and Nadir pointing orientations

Table 5.1-2 Transformation Matrix of Spacecraft coordinates and DAS UVW

Spacecraft	U (radial)	V (in-track)	W (Cross-track)
+X	0	1	0
-X	0	-1	0
+Y	-1	0	0
-Y	1	0	0
+Z	0	0	1
-Z	0	0	-1

The evaluation under requirement 4.5-2 was limited to the spacecraft's two critical subsystems: the propulsion system tank and the spacecraft batteries. These two subsystems represent the most critical components for this requirement due to the primary source of potential energy on the spacecraft for a generalized single string spacecraft architecture.

Refer to Attachments GRATTIS*_req52_data_fixed.csv*, and GRATTIS*_req52_data_gravitygradient.csv* for the inputs and outputs for the Fixed Oriented and Gravity Gradient Oriented cases, respectively.

Orientation	Probability of PMD Failure	<0.01?
Gravity Gradient	9.0145E-03	*COMPLIANT*
Fixed Oriented	3.6150E-03	*YES*

5.3 GRATTIS Collision Mitigations

Note that GRATTIS features a GPS, which assists in reducing the covariance of collision estimates, as well as a propulsion system, which assists in performing any collision avoidance maneuvers, if required.

5.4 Assessment of compliance with Requirements 4.5-1 and 4.5-2

The table below outlines the GRATTTIS compliance with Requirements 4.5-1 and 4.5-2.

Requirement	Description	GRATTIS Compliance
4.5-1	Limiting debris generated by collisions with large objects when in Earth orbit	Compliant
4.5-2	Limiting debris generated by collisions with small objects	Compliant

Section 6 Assessment of Spacecraft Post-mission Disposal Plans and Procedures

6.1 Spacecraft Disposal Plan & Procedure

GRATTIS will utilize atmospheric drag for reentry by lowering the initial perigee altitude for the disposal orbit <234 km using the on-board propulsion system, as soon as practical after completion of mission.

After payload operations are complete, the spacecraft will be commanded by the ground to turn off all payloads and begin decommissioning. All propulsive maneuvers will be coordinated with the 18 SDS.

To lower the spacecraft altitude and ensure atmospheric as soon as practical after completion of the mission, GRATTIS will utilize its single-axis propulsion unit to perform a retrograde burn. To do this, GRATTIS will utilize its attitude control system to rotate its body so that the fixed thruster (Z-axis) points directly opposite of the direction of the velocity vector and begin the propulsive maneuver. 6 months have been allocated to perform the orbit lowering maneuver, decreasing the spacecraft perigee to 215 km for atmospheric drag-dominated final re-entry. Prior to re-entry, the spacecraft will be command to vent any remaining spacecraft propulsion, and deplete the battery by cutting off solar panel charging.

6.2 Spacecraft Cross-sectional Area & Area-to-Mass Calculation

The effective Area-to-Mass ratio was calculated GRATTIS from the key dimensions detailed in Section 2, utilizing NASA-STD-8919.14C Equations 1 and 2 below.

$$\boldsymbol{Mean\ CSA} = \frac{\sum \boldsymbol{Surface\ Area}}{\boldsymbol{4}}$$

Equation 1: Mean Cross Sectional Area for Convex Objects

$$\boldsymbol{Mean\ CSA} = \frac{(\boldsymbol{A_{max}} + \boldsymbol{A_1} + \boldsymbol{A_1})}{\boldsymbol{2}}$$

Equation 2: Mean Cross Sectional Area for Complex Objects

For the as-deployed/stowed (worst-case) configuration, the mean CSA was calculated using Equation 1. The stowed CSA represents a failure to deploy the solar panels (there are no other deployable/appendages).

For the nominal/deployed configuration, the mean CSA was calculated using Equation 2. The deployed CSA represents a nominal deployment of the solar panels.

Bus Dimensions	
Bus length (B_x)	0.861 m
Bus width (B_y)	1.140 m
Bus height (B_z)	0.399 m

Solar Panel Dimensions	
SP Length X (SP_X)	2.842 m
SP Width Y (SP_Y)	1.140 m
SP Area (SP_Z) = A_{max}	2.892 m^2

Payload Dimensions	
PL length (PL_X)	0.538 m
PL width (PL_y)	1.086m
PL height (PL_z)	0.548 m

SC Surface Area (Stowed)	
$SC_X = (PL_Y * PL_Z) + (B_Y * B_Z) = A_1$	1.050 m^2
$SC_Y = (PL_X * PL_Z) + (B_X * B_Z) = A_2$	0.638 m^2
$SC_Z = (B_X * B_Y)$	0.982 m^2
$\sum_{SC} = 2*(SC_X + SC_Y + SC_Z)$	3.560 m^2

Cross Sectional Area (CSA)	
CSA_{Stowed} = Convex CSA = $\sum_{SC} / 4$	1.335 m^2
$CSA_{Deployed}$ = Complex CSA = $(A_{max} + A_1 + A_2) / 2$	2.265 m^2

The following masses were considered for the Area-to-Mass ratio.

SC Mass	
Initial (wet)	229.3 kg
Final (dry)	209.8 kg

The maximum Area-to-Mass ratio (AM_{MAX}) was calculated, using the stowed CSA and initial mass. This represents a failure of all systems, which is considered a worst-case highest on-orbit lifetime scenario.

The minimum Area-to-Mass ratio (AM_{MIN}) was also calculated, using the deployed CSA and final mass. This represents a nominal operation of all systems, which is considered the best-case lowest on-orbit lifetime scenario.

SC Area-to-Mass Ratio	
$AM_{MAX} = CSA_{Stowed} / Mass_{Initial}$	0.0058 kg/m^2
$AM_{MIN} = CSA_{Deployed} / Mass_{Final}$	0.0108 kg/m^2

6.3 Spacecraft Lifetime

The spacecraft lifetime was calculated for both Area-to-Mass ratios (AM_{MAX} and AM_{MIN}) with and without propulsive orbit lowering maneuvers. The worst-case lifetime scenario altitude of 590 km was utilized.

The assumptions for the spacecraft lifetime calculation and results <u>without</u> propulsive orbit lowering are shown in the tables below.

Start Year	2027
Perigee Altitude	590 km
Apogee Altitude	590 km
Inclination	97.7
RA of Ascending Node	0 deg (Estimate)
Argument of Perigee	0 deg (Estimate)

SC Lifetime <u>without</u> Propulsive Orbit Lowering	
$AM_{MAX} = 0.0058$ kg/m^2	22.5 years
$AM_{MIN} = 0.0108$ kg/m^2	12.4 years

In order to meet the FCC <5-year lifetime requirement, the GRATTIS spacecraft features a propulsion system, detailed in Section 2.7. The assumptions for the spacecraft lifetime calculation and results with a 100 m/s propulsive maneuver at apogee are shown in the tables below. Note GRATTIS carries 200 m/s of propulsive capability for a factor of safety of 2 for the 100 m/s required propulsive orbit lowering maneuver.

Start Year	2027
Perigee Altitude	**234 km**
Apogee Altitude	590 km
Inclination	97.7
RA of Ascending Node	0 deg (Estimate)
Argument of Perigee	0 deg (Estimate)

SC Lifetime with Propulsive Orbit Lowering	
$AM_{MAX} = 0.0058$ kg/m^2	0.307 years (4.44 months)
$AM_{MIN} = 0.0108$ kg/m^2	0.1369 years (2.03 months)

In total, including mission operations, the expected spacecraft maximum lifetime with an orbit lowering maneuver is 16.4 months, detailed below.

Phase	Duration
Spacecraft Bus Commissioning	1 mo.
Orbit Raise (if required, 510 km to 590 km)	Up to 1 mo.
Payload Operations	5 mo.
Orbit Lowering (590 km to 234 km, 100 m/s dV)	Up to 6 mo.
Re-entry	2 - 4.4 mo.
Total	14 mo. (nominal) Up to 17.4 mo.

6.4 Post Mission Disposal Maneuver Probability of Success

Given GRATTIS will be utilizing a post mission disposal (PMD) maneuver as soon as practical after completion of mission and to meet the FCC <5-year lifetime, the probability of success of this chosen disposal method is >0.9 for GRATTIS.

The PMD success probability is calculated as the product of individual subsystem reliabilities required to execute the disposal maneuver.

$$P(\text{success}) = P(\text{Prop}) \times P(\text{ADCS}) \times P(\text{C\&DH}) \times P(\text{Comm}) \times P(\text{EPS})$$

$$P(\text{success}) = 0.98 \times 0.98 \times 0.98 \times 0.98 \times 0.98 = 0.903$$

Component Reliabilities based on APEX Aries spacecraft heritage, redundant systems, design with margin, component selections for relevant environment, and relatively short mission duration:

- Propulsion (Prop): The Exoterra Halo Hall Effect Thruster system has demonstrated extensive flight heritage on multiple LEO programs with high reliability (>0.98). The system includes a safety factor of 2 providing substantial margin against thruster performance degradation or execution inefficiencies.
- Attitude determination and control system (ADCS): Required to maintain proper spacecraft orientation during the burn, >0.98 reliability for GRATTIS spacecraft.
- Command and data handling (C&DH): Required to execute stored commands or ground commands to initiate the maneuver sequence, >0.98 reliability for GRATTIS spacecraft.
- Communications (Comm): Required for maneuver confirmation and telemetry, >0.98 reliability for GRATTIS spacecraft.
- Electric Power System (EPS): Required to provide sufficient energy for thruster operation and spacecraft housekeeping during the disposal sequence, >0.98 reliability for GRATTIS spacecraft.

6.5 Assessment of compliance with Requirements 4.6-1 through 4.6-4

The table below outlines the GRATTIS compliance with Requirements 4.6-1 through 4.6-4.

Requirement	Description	GRATTIS Compliance
4.61a-c	Lifetime < 5 years	Compliant
4.6-2	Storage and Earth escape compliance	Not Applicable
4.6-3	Long-term re-entry in MEO	Not Applicable
4.6-4	Reliability of PMD maneuver operations	Compliant

Section 7: Assessment of Spacecraft Reentry Hazards

7.1 Spacecraft Components and re-entry

The following table summarizes the top 10 GRATTIS components by size, mass, material, shape that survive atmospheric uncontrolled reentry after orbit lowering using NASA DAS 3.2.7. Refer to Section 2 for component locations on vehicle, and Attachment *GRATTIS_reentry.csv* for the complete list of spacecraft components.

Table 7-1: GRATTIS Top 10 KE Components

Row Num	Name	Parent	Qty	Material	Body Type	hermal Ma	meter/Wi	Length	Height	Status	Risk	Demise Alt	Total DCA	KE
1	GRATTIS	0	1	Aluminum 6061-T6	Box	210	1.039	1.14	0.947	Compliant	'1:11300'		7.26	
168	Payload support plate	1	1	Titanium (6 Al-4 V)	Box	2.51	0.13	0.235	0.04			0	0.49	3561.67
197	Test mass	1	2	Tungsten	Box	0.52	0.03	0.03	0.03			0	0.64	2314.42
203	SGRS frame +y	1	2	Titanium (6 Al-4 V)	Flat Plate	0.727	0.127	0.152				0	0.98	246.92
204	SGRS frame -y	1	2	Titanium (6 Al-4 V)	Flat Plate	0.727	0.127	0.152				0	0.98	246.92
52	tank - Carbon	1	1	Reinforced Carbon-Carbon	Cylinder	1.2	0.229511	0.42				0	0.81	149.2
96	RESONATOR PLATE	1	6	Fused Silica (7980)	Box	0.124363	0.137	0.137	0.007			0	2.55	16.9
51	tank - AL	1	1	E-Glass	Cylinder	0.4	0.229511	0.42				0	0.81	16.57
98	Interferometer	1	12	Sapphire	Box	0.027539	0.033	0.034	0.018			0	3.83	7.46
199	Electrode housing (incl electrodes)	1	12	Molybdenum	Flat Plate	0.015	0.0414	0.0414				0	4.04	1.18
200	Cover plates	1	12	Molybdenum	Flat Plate	0.015	0.0419	0.0419				0	4.05	1.15

7.2 Human Casualty

The risk of human casualty for the expected year of atmospheric drag for reentry and spacecraft orbital inclination is <1:10,000.

The following table details the assumptions in the NASA DAS calculation.

Parameter	Value
Orbit	590 km x 590 km 97.7 deg inc.
Launch	2027
Mission Lifetime (Nominal)	1 year
Mission Lifetime (Max.)	5 years (FFC Req.)

The following table details the results from the NASA DAS human casualty at 2032 which assumes a worst-case 5 year maximum mission lifetime per FCC requirements.

Yr of uncontrolled reentry	Casualty	<1:10,000?
2032 (max 5 yrs)	1:11,300	Compliant

7.3 Assessment of compliance with Requirements 4.7-1

The table below outlines the GRATTIS compliance with Requirements 4.7-1

Requirement	Description	GRATTIS Compliance
4.7-1	Limit the risk of human casualty	Compliant

Section 8: Assessment for Special Classes of Space Missions

GRATTIS does not include a tether and is larger than 10 cm x 10 cm x 10 cm, therefore the assessment of large object collision risks with a tether are not applicable.

8.1 Assessment of compliance with Requirement 4.8-1

Requirement	Description	GRATTIS Compliance
4.8-1	Special classes of space mission requirements.	Not Applicable

Section 9 - 14: Launch Vehicle Assessments

ODAR Sections 9-14 are not applicable for the GRATTIS Spacecraft. Refer to the SpaceX Transporter 19 ODAR for these assessments.

Summary

This document has provided a comprehensive Orbital Debris Assessment Report for the GRATTIS technology demonstration and provides how GRATTIS meets the technical requirements for limiting orbital debris and the methods to comply with the NASA requirements for limiting orbital debris generation, demonstrating that the spacecraft meet acceptable standards for limiting orbital debris generation.