Gravitational Reference Advanced Technology Test in Space

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Data Restrictions

The data in this ODM is considered Sensitive/Proprietary

Section 1: Mission Overview

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 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.

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.

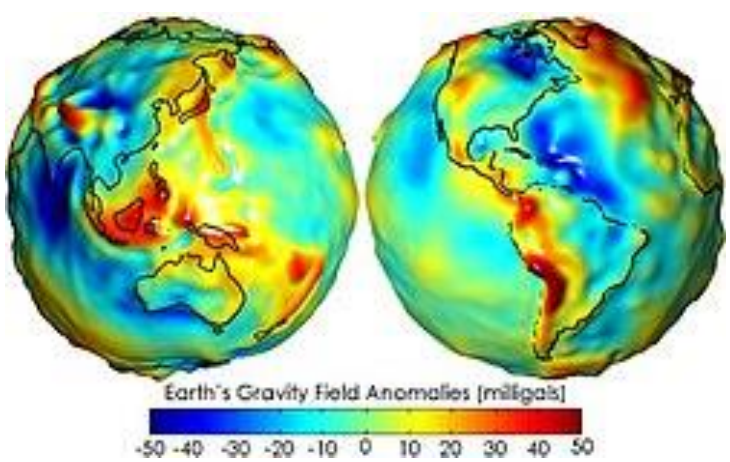

Figure 1-1: Earth Gravity Field Anomalies (credit: NASA Earth Observatory)

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 minisatellite bus (~230 kg) from Apex Space. GRATTIS is slated fly on the SpaceX Transporter-19 mission to 510 km – 590 km (circular) Sun Synchronous Orbit (SSO) in Q1 2027 (notionally Feb 1, 2027). 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 SGR-S and ODIN instruments.

GRATTIS has a propulsion system sized to: 1) raise the orbit to a nominal operating orbit if the launch vehicle deploys the spacecraft lower than 570 km at the start of the mission, and 2) lower the orbit after the technology demonstration is complete to meet FCC <5-year mission lifetime requirement.

Table 1: Mission Overview

1.1: Company Name	University of Florida
1.2: Mission Name	Gravitational Reference Advanced Technology Test in Space (GRATTIS)
1.3: Mission Description	The GRATTIS technology demonstration will precisely measure the Earth's gravity to track water and ice movement
1.4: Number of Satellites	1
1.5: Launch Date	Feb 1, 2027 (Notional)
1.6: Launch Vehicle and Launch Site	SpaceX Falcon 9 Transporter-19 Mission Vandenberg Air Force Base
1.7: Deployment Apogee (km)	510 - 590 km
1.8: Deployment Perigee (km)	510 - 590 km
1.9: Deployment Inclination (deg.)	Sun Synchronous Orbit (SSO) 97.7 deg for a 590 km SSO
1.10: Operational Apogee (km)	590 km
1.11: Operational Perigee (km)	590 km
1.12: Operational Inclination (deg.)	Sun Synchronous Orbit (SSO) 97.7 deg for a 590 km SSO

1.13: Reason for Selecting Orbit	SSO provides near global coverage and consistent orbital geometry, allowing for precise, repeated measurements of the Earth's gravitational field.
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

Section 2: Spacecraft Information

2.1 Spacecraft Description

2.1.1 Spacecraft Bus

GRATTIS utilizes an Aries commercial spacecraft bus from Apex. Aries is a standardized, ESPA-class minisatellite (~230 kg) bus. Table 2 outlines GRATTIS generic attributes. Refer to Figure 2-1 and 2-3 for the spacecraft stowed and deployed configurations with key dimensions.

Table 2: 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)	235 kg (Initial – Wet) 216 kg (Final – Dry)

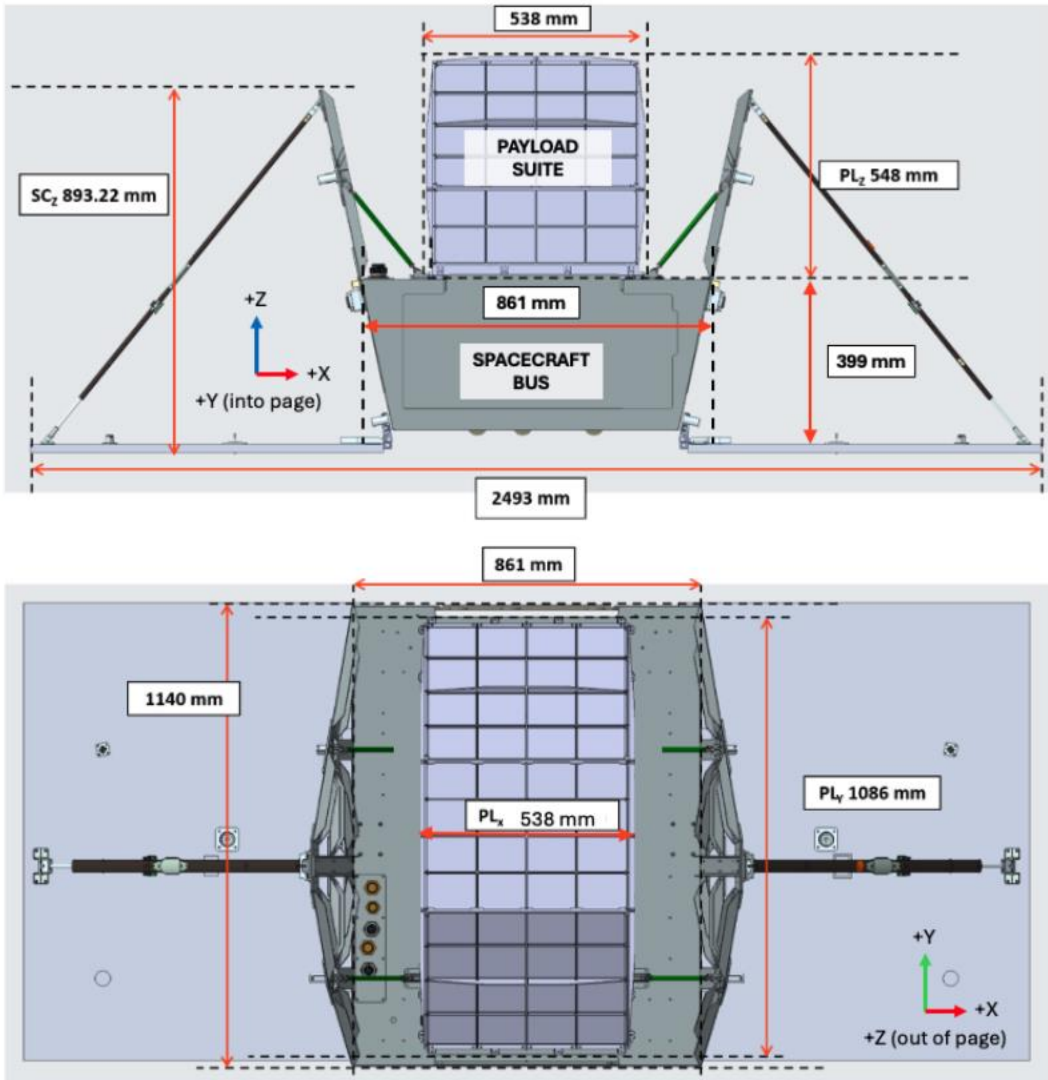

Figure 2-1.1: GRATTIS Key Bus Dimensions. Top: Top view (deployed).
Bottom: Side view (deployed)

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

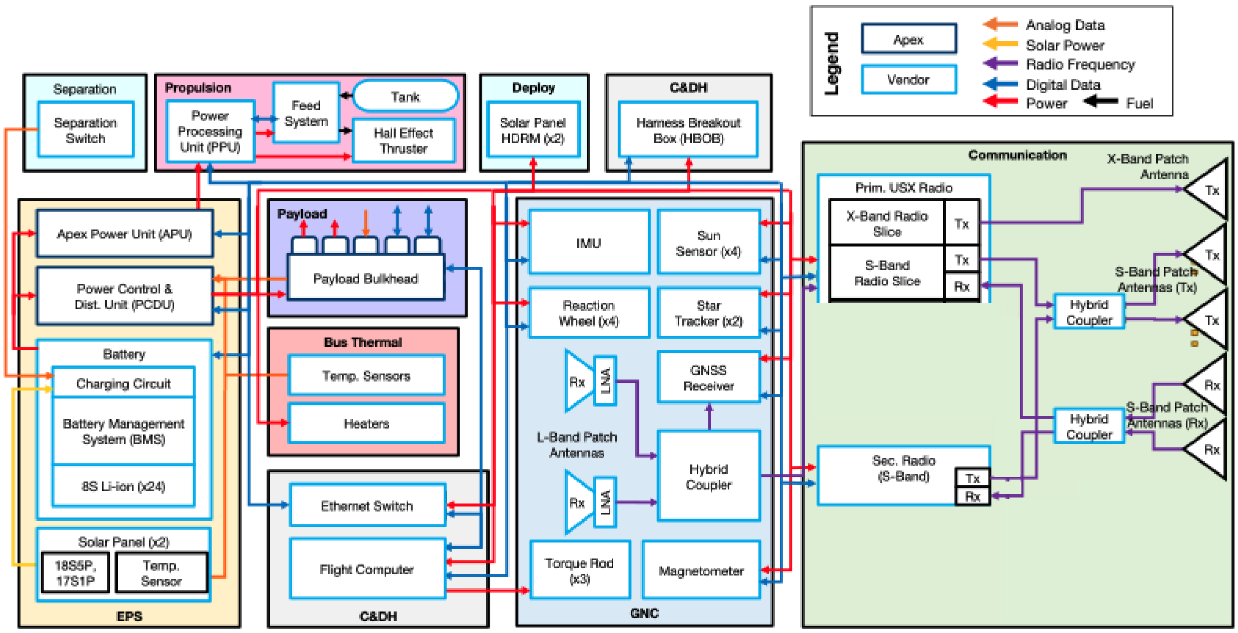

Figure 2.1-2 GRATTIS spacecraft bus block diagram

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

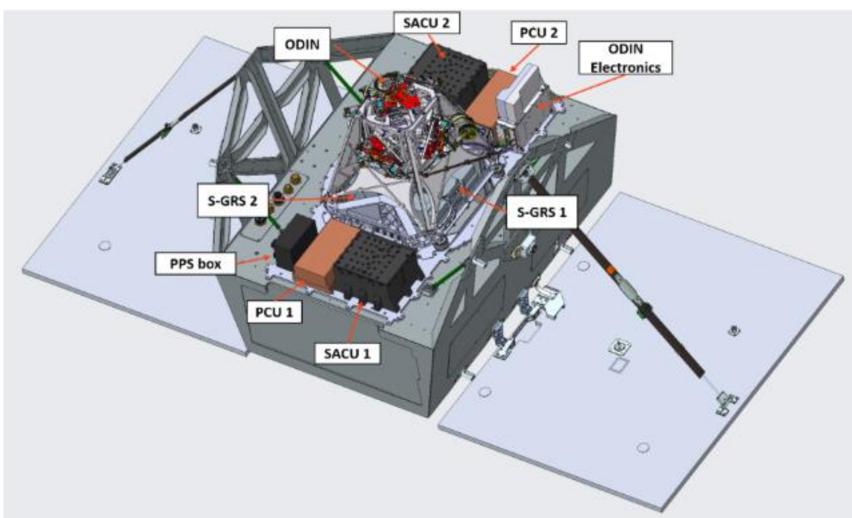

Figure 2.1-3 GRATTIS spacecraft payload configuration

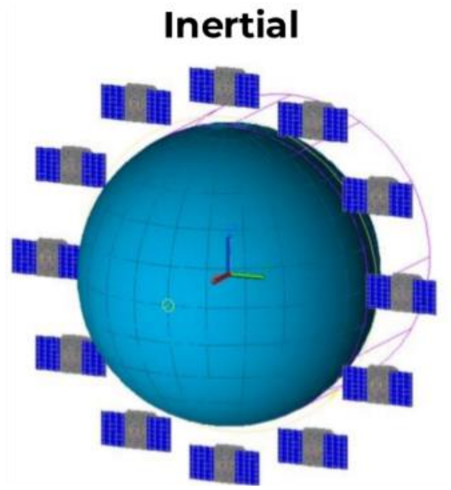

Figure 2.1-4 GRATTIS mission operation configurations

The bus structure is made of three Al 6061-T6 core components, shown in below. The middle area between the Top and Base deck will contain the primary spacecraft subsystems (e.g.: avionics, guidance & navigation system, attitude determination and control system, communication system, and electric power system). For additional detail on each of the specific spacecraft subsystems, refer to the *GRATTIS-PL-003 GRATTIS Mission Description* document.

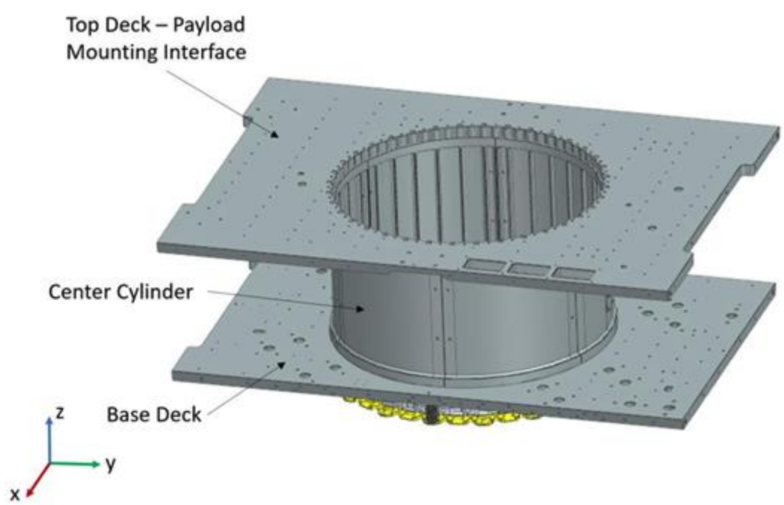

Figure 2.1-4: SC Primary Structure

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 2.3 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, as shown below.

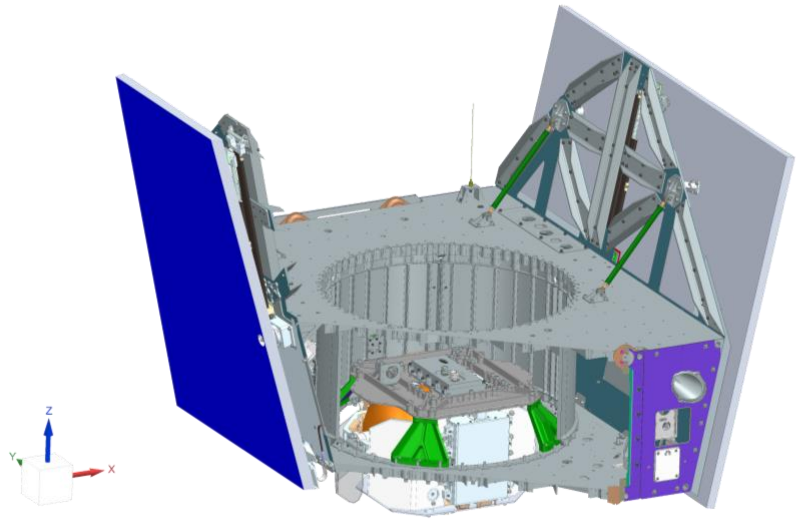

Figure 2.1-5: Cross-sectional view of SC showing the propulsion system

The Top deck primarily consists of the GRATTIS payload instruments, shown below. See Section 2.2 and 2.3 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.

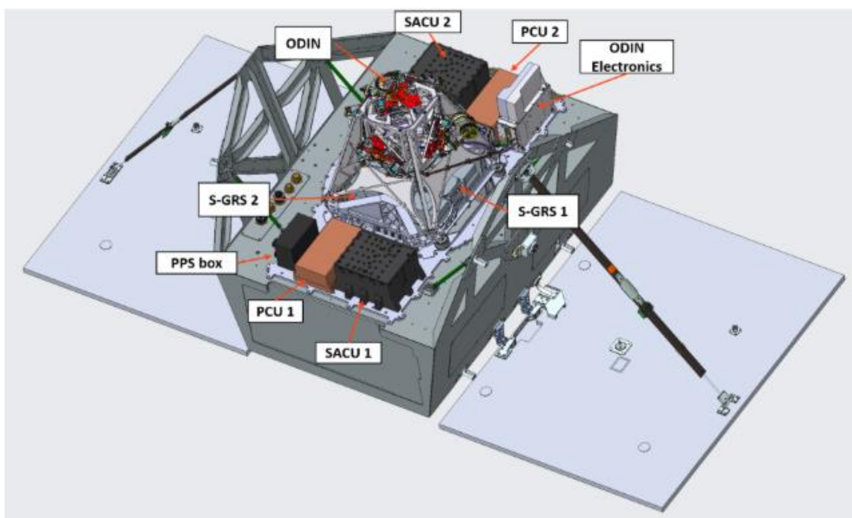

Figure 2.2-3: S-GRS & ODIN Payload Suite on SC Top Deck (deployed)

The figure below illustrates the external spacecraft bus component locations, and Figure 2.1-8 illustrates the external spacecraft bus component locations. Note these illustrations do not show the propulsion system or payloads.

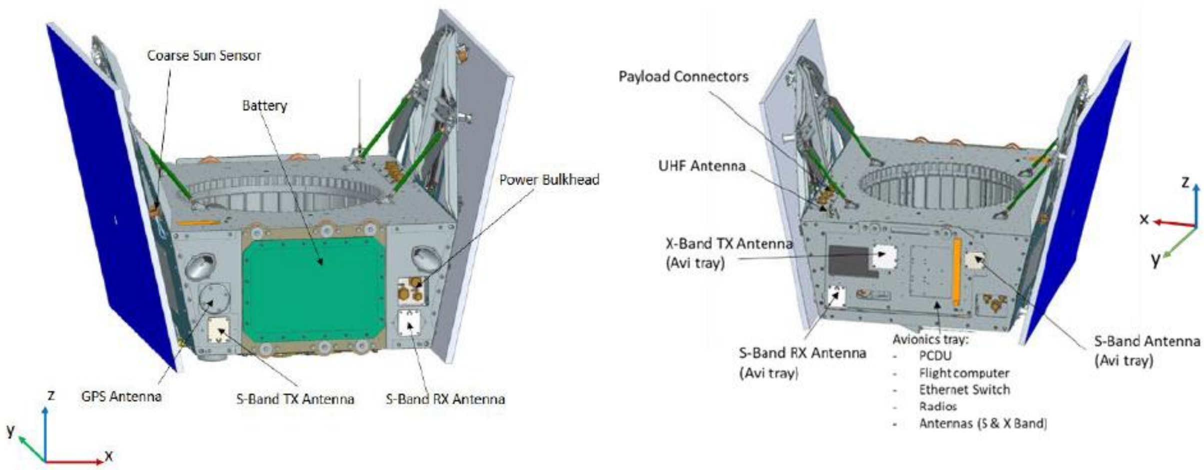

Figure 2.1-7: GRATTIS external illustration

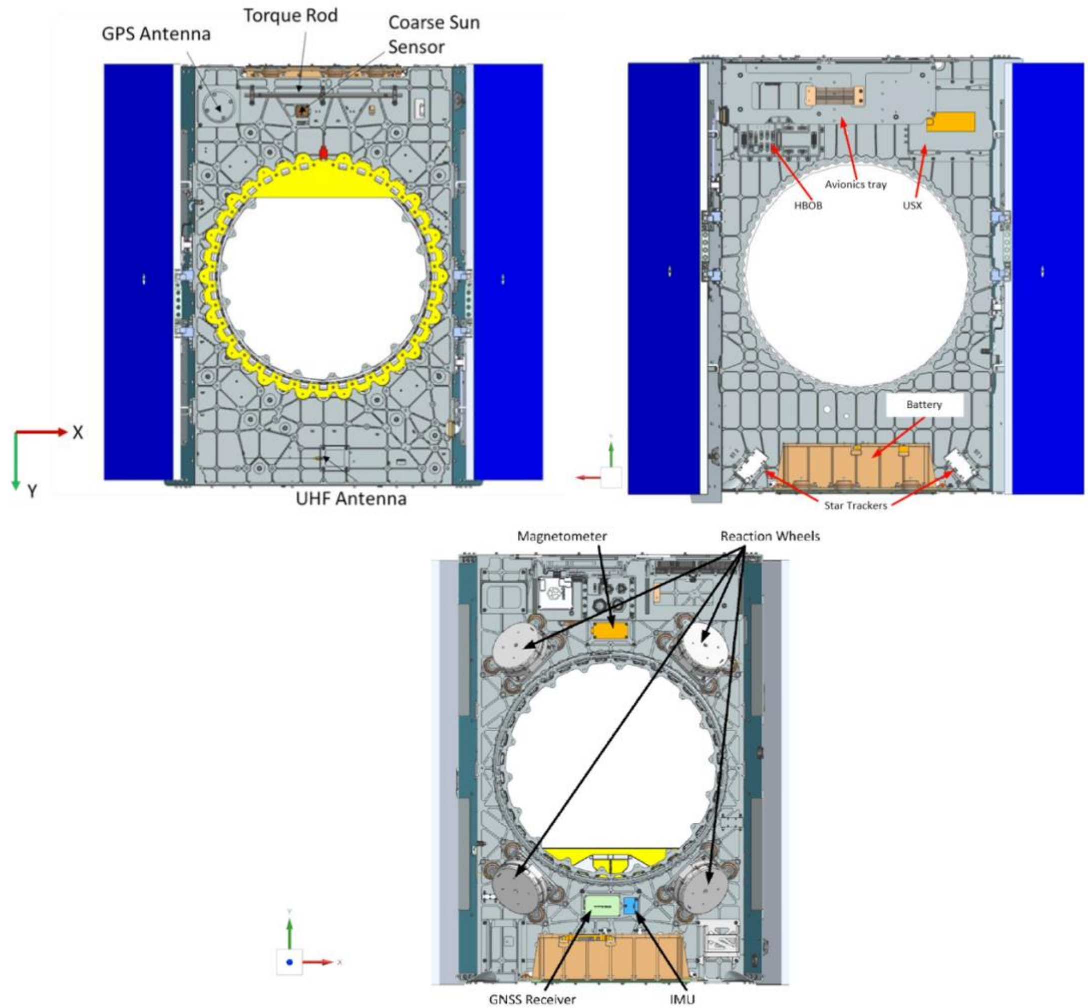

Figure 2.1-8: GRATTIS labeled external illustrations

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 2.1-9. 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.

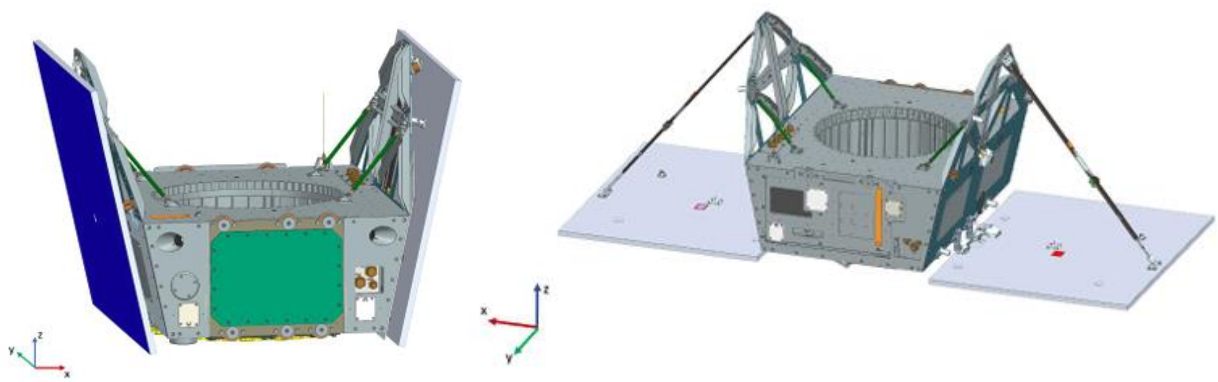

Figure 2.1-9: SC solar array stowed (left) and deployed (right) configurations

2.1.2 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 2.1-10 shows an exploded view of a S-GRS.

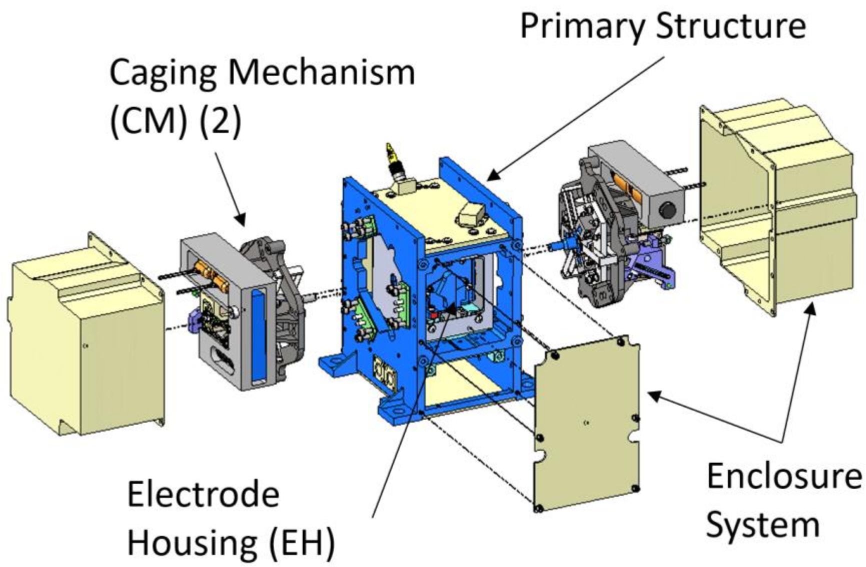

Figure 2.1-10: S-GRS Sensor Head containing the test mass, electrode housing, Caging and Release Mechanism, and Enclosure

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.

2.1.3 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 10 shows the ODIN assembly.

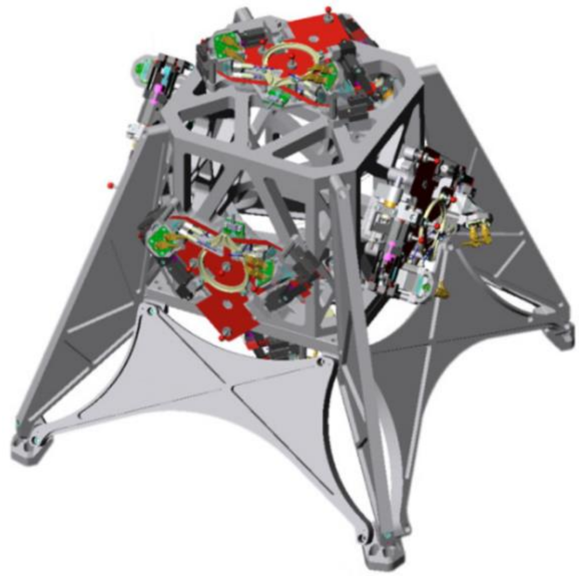

Figure 2.1-11: ODIN Optomechanical Sensor Package

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.

2.2 Effective Area-to-Mass Ratio

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.

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

Equation 1: Mean Cross Sectional Area for Convex Objects

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

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. The initial mass at the beginning of the mission with fully loaded propellant is 229.3 kg. The final mass at the end of the mission after the propellant is fully expelled is 209.8 kg.

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

The next section details the SC Lifetime with propulsive orbit lowering to meet the FCC <5 year lifetime requirement.

2.3 Satellite Maneuverability & Lifetime

2.3.1 Spacecraft 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.

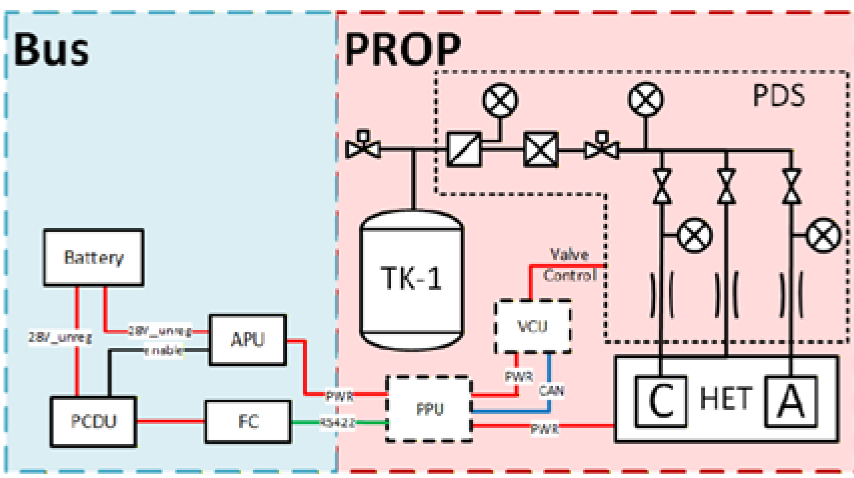

Figure 2.3-1: Propulsion Block Diagram

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.

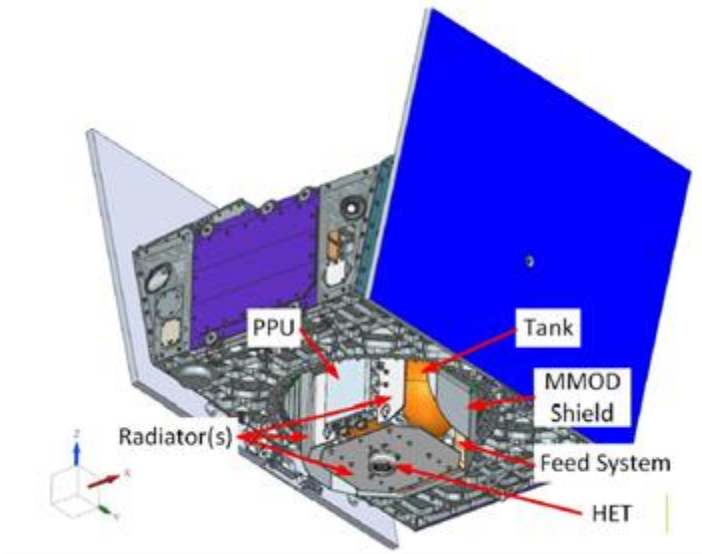

Figure 2.3-2: SC view detailing propulsion system

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.

2.3.2 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 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 without 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 the section prior (Section 2.3.1). 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 to demise with an orbit lowering maneuver is 16.4 months, meeting the FCC <5 year requirement. Details on the mission timeline are shown in the table below.

Table 2.3-1: GRATTIS Mission Lifetime Summary

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.

Section 3: Launch and Deployment

3.1 18th Space Defense Squadron Registration

Prior to deployment, GRATTIS will be registered with the 18th Space Defense Squadron (or successor entity).

3.2 18th Space Defense Squadron Coordination

The GRATTIS mission operations team intends to share initial deployment ephemeris received from the launch vehicle provider, Space X, to the 18th Space Defense Squadron (18 SDS).

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 includes, but not limited to owner/operator, ephemeris, start/end maneuver, delta-V, maneuver type, and 24/7 point of contact. The GRATTIS team is committed to ensuring the safety an accuracy of the maneuver.

3.3 Trackability

GRATTIS is an ESPA-class minisatellite (~230 kg) operating in LEO and will be actively tracked via 18 SDS. GRATTIS also features a GPS system capable of providing ephemeris refinement.

3.4 Identification

Following deployment, the GRATTIS Mission Operations team will use the initial orbital ephemeris data based on launch vehicle deployment parameters from Space X, to assist in identifying GRATTIS after separation and establish communication. Early extended ground station passes will be conducted to account for spacecraft identification uncertainty. Once ground communication is established, GRATTIS GPS information will be downlinked. The GRATTIS Mission Operations team intends to share all initial ephemeris information from SpaceX and GPS information from the spacecraft in coordination with 18 SDS to estimate and establish GRATTIS' NORAD ID.

3.5 Ephemeris Accuracy

The spacecraft will be equipped with a GPS receiver to determine its orbital parameters, including apogee, perigee, inclination, and right ascension of the ascending node(s), with accuracy consistent with commercial GPS positioning capabilities in low Earth orbit.

3.6 Orbital Evolution

GRATTIS in a nominal 550 km sun-synchronous orbit (SSO) causes the orbital plane to precess at the same rate as Earth's orbit around the Sun, maintaining a consistent local time of ascending node (LTAN). Over time, perturbations from Earth's oblateness, lunar and solar gravitational influences, and atmospheric drag will cause variations in orbital elements including semi-major axis, eccentricity, and argument of perigee. GRATTIS propulsion system carries additional propulsion mass (4.75 kg), resulting in a total of 250 m/s dV allowing for up to five (5) 50 m/s dV orbit maintenance maneuvers to maintain the science orbit, as-needed.

Refer to Section 2.2 on spacecraft lifetime and demise. A summary of the spacecraft lifetime and demise is shown below.

In total, including mission operations, the expected spacecraft *maximum* lifetime from launch to atmospheric re-entry demise, including an orbit lowering maneuver, is 16.4 months, which meets the FCC <5 year requirement. Details on the calculation are shown in Section 2.3.2, and a copy of the Table 2.3-1 is shown below.

Table 2.3-1: GRATTIS Mission Lifetime Summary

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.

Section 4: Satellite Operations and Collision Risk

4.1 Collision with Large Objects

4.1.1 Space stations / Debris in planned orbit

There are numerous active satellites and known debris in the 550 km +/- 20 km SSO region. GRATTIS will rely on 18 SDS to track these items and notify the GRATTIS UF MOC via Space-Track.org of any conjunction risks. The GRATTIS team will coordinate with the 18th Space Defense Squadron (18 SDS) on any collision avoidance maneuvers, if required.

4.1.2 Potential Risk of Collision

The GRATTIS team committed to maintaining 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.

Efforts of reducing overall risk of collision is taken by the GRATTIS team by applying appropriate design practices and operational procedures are in place to prevent the creation of orbital debris, including passivation of all energy sources at end-of-life, safely venting residual propellant, adherence to FCC's <5-year deorbit guidelines, and no intentional breakups or releases that could generate debris. See Section 4.1.5 for the potential probability of collision.

4.1.3 Conjunction Coordination

The GRATTIS team will ensure enrollment and email notifications from Space-Track.org to receive conjunction warnings, and coordinate with the 18 SDS, and any required associated entities, for conjunction assessments and avoidance maneuvers (if-required) throughout the mission to avoid in orbit collisions.

4.1.4 Geostationary Assessment

GRATTIS will be launched and operated in LEO, therefore, the assessment of any known geostationary satellites located within the GRATTIS spacecraft is not applicable.

4.1.5 Collision Probability Assessment

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 570 km orbit altitude was utilized to capture the worst-case scenario altitude for a SpaceX notional 550 km +/-20 km launch. Although the GRATTIS orbital mission lifetime is 1 year, 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 wet mass; and the second was the nominal scenario at end-of-mission with spacecraft solar panels deployed, and dry mass. Utilizing guidance from NASA NASA-STD-8719.14B, the Mean cross-sectional area (CSA) was calculated for each configuration. Details regarding the Mean CSAs 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, meet the requirement of less than 0.001 (1 in 1,000).

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)	235	kg
	Area-to-Mass	0.0058	m^2/kg
	Probability of collision	**1.6236E-05**	**10^x**
	Pc < 0.001?	*COMPLIANT*	
	Mean CSA non-Convex	2.265	m^2

End-of-mission Configuration	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*	

Collision avoidance maneuvers will be planned when the collision probability exceeds 1×10^{-4} (1 in 10,000). The goal of any collision avoidance maneuver is to reduce the collision probability (Pc) to below NASA CARA's recommendation of 3.16×10^{-6}, representing a risk reduction of at least 1.5 orders of magnitude below the initial threshold. Additional screening may use miss distance thresholds (typically coordinated with 18 SDS conjunction data messages) alongside Pc calculations. GRATTIS features 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.

4.1.6 Tether Collision Risks

GRATTIS does not include a tether, therefore the assessment of large object collision risks with a tether are not applicable.

4.2 Collision with Small Objects

4.2.1 Small Object Risks

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.

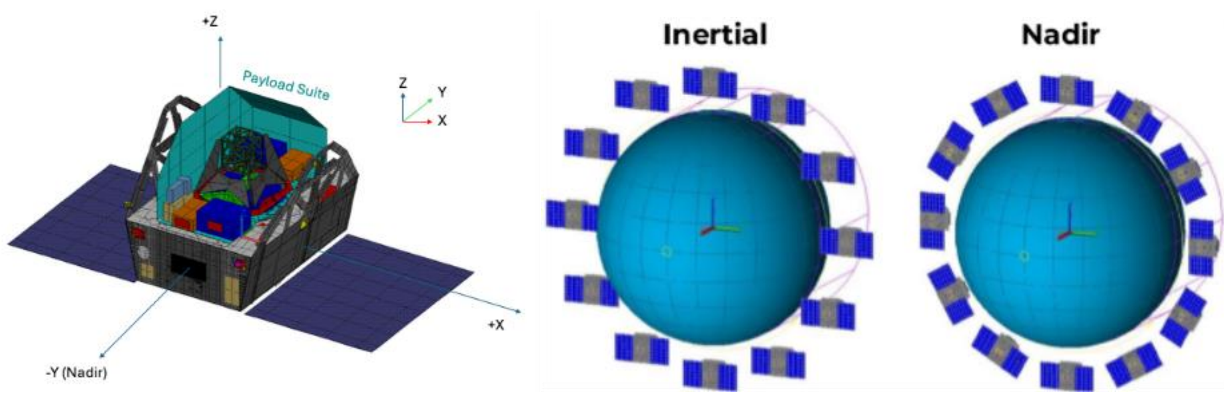

Figure 5.2-1 GRATTIS Spacecraft coordinates with Inertial and Nadir pointing orientations

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	*Compliant*

4.3 Proximity Operation

4.3.1 Proximity Maneuvers

GRATTIS does not intend to perform any close proximity maneuvers with other satellites or objects.

4.3 Debris Released and Deployment Devices

4.4.1 Intentional Debris Release

GRATTIS does not intend to intentionally release debris from the spacecraft.

4.5 Explosion Risk and End of Life Passivation

4.5.1 Explosion Risk

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 (<2.4×10^{-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.

4.5.2 Energy sources that generate debris

The spacecraft contains two primary energy sources that could potentially generate debris through energy conversion:

Lithium-Ion Batteries (Chemical/Electrical Energy)

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.

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.

Xenon Propellant (Pressure Energy)

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.

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.

4.5.3 Stored energy removal at end-of-life

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.

4.5.4 Liquids

There are no liquids on-board GRATTIS that if released would persist in droplet form.

Section 5: Post Mission Disposal

5.1 Post Mission Disposal

GRATTIS will perform an orbit lowering maneuver for atmospheric re-entry/demise.

5.1.a Remaining Fuel

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. All remaining fuel will be vented prior to re-entry. Refer to section 2.3.1 for details on the GRATTIS propulsion system.

5.2 Disposal Method

GRATTIS will perform an orbit lowering maneuver for atmospheric re-entry/demise.

5.2.a.i Lifetime after end-of-mission

From Section 2.3.2 GRATTIS will pass <215 km, after end of mission, and will stay in orbit 1.6 – 3.4 months before atmospheric re-entry. GRATTIS deorbits well within 5 years after end of mission, where the spacecraft is no longer capable of conducting collision avoidance maneuvers.

Below is a summary of the analysis provided in Section 2.3 utilizing NASA DAS.

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)

5.2.a.ii Atmospheric re-entry Survival

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 Appendix A2 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	ameter/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

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

5.2.a.iii Controlled Atmospheric Re-entry

GRATTIS is not performing a controlled atmospheric re-entry (targeting a specific geographic area).

5.3 Post Mission Disposal Maneuver Probability of Success

GRATTIS will be utilizing a post mission disposal (PMD) maneuver to meet the FCC 5-year lifetime. The probability of success of the 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.

5.4 De-orbit Transit

To minimize collision risk during deorbit and avoid operational constraints to any spacecraft in lower altitude regimes, GRATTIS incorporates the following design and operational strategies:

1) an on-board GPS receiver to provide precise orbit determination enabling accurate, prediction of the spacecraft trajectory during disposal maneuvers and supporting conjunction assessment with other spacecraft,
2) reliable critical sub-systems,
3) radiation-hardened fault monitoring ensuring reliable spacecraft control during disposal operations even after extended mission exposure,
4) careful deorbit trajectory planning/timing and coordination with 18 SDS, and
5) Passivation prior to atmospheric entry.

Summary

This document has provided a comprehensive Orbital Debris Management Plan for the GRATTIS technology demonstration and provides how GRATTIS meets the technical requirements for limiting orbital debris, demonstrating that the spacecraft meet acceptable standards for limiting orbital debris generation.