UMBRA

EXHIBIT B – ORBITAL DEBRIS ASSESSMENT REPORT

Page

1. SUMMARY OF REPORT FINDINGS

Umbra Lab Inc. ("Umbra") provides an orbital debris assessment report ("ODAR") for the Umbra-11 to Umbra-14 satellites.[1] The analysis uses the Debris Assessment Software, DAS v3.2.6, provided by the NASA Orbital Debris Program Office.

In the event of launch delays, Umbra may substitute the satellite described herein as "Umbra-14" with the satellite described herein as "Umbra-14 ALTERNATE," which has a different configuration from Umbra-11, Umbra-12, Umbra-13, and Umbra-14.

An orbital debris assessment of the Block 3 Tranche 1 shows the mission complies with the applicable requirements for spacecraft end-of-life disposal and risk to human casualty as specified in NASA's Process for Limiting Orbital Debris, NASA-STD-8719.14C.[2]

Satellites utilize near circular orbits at a nominal altitude of 565 km as defined in the Technical Attachment[3]. The satellite deployment from the launch vehicle, as specified by the provider, will be approximately 590 ± 20 km in altitude. In the worst-case scenario that an Umbra satellite is deployed dead-on-arrival ("DOA") at 610 km while fully stowed, it will re-enter the atmosphere in at most 35.9 years. The orbital deployment range will likely be refined as the various launch dates approach.

Spacecraft disposal is accomplished through atmospheric reentry. In the nominal case, each spacecraft is expected to reenter no more than 0.2 years after mission completion with a planned post-mission disposal ("PMD") described herein. Umbra will budget sufficient reserves to ensure the capability to conduct any expected, necessary collision avoidance maneuvers during the lifetime of the mission.

1.1. Self-Assessment of the ODAR

A self-assessment is provided in Table 1 in accordance with the assessment format provided in

[1] In the associated application, Umbra seeks authority to launch and operate four satellites (the "Block 3 Tranche 1").
[2] Process for Limiting Orbital Debris, NASA Technical Standard, NASA-STD-8719.14C (approved Nov. 5, 2021) ("*Process for Limiting Orbital Debris*"). NASA-STD-8719.14C is sufficient to satisfy 47 CFR 25 as updated by FCC 22-74 (approved Aug. 9, 2024) ("*The Orbital Debris Second Report and Order*").
[3] *See* Exh A Technical Attachment Block 3 Tranche1, Table 1

Appendix A.2 of NASA-STD-8719.14C.

Table 1. Orbital Debris Assessment Report Evaluation: Block 3 Tranche 1

Reqmt #	Spacecraft			Comments
	Compliant or N/A	Not Compliant	Incomplete	
4.3-1.a	X			No debris released in LEO
4.3-1.b	X			No debris released in LEO
4.3-2	X			No debris released in GEO
4.4-1	X			Limit risk of explosion
4.4-2	X			Design for passivation
4.4-3	X			No planned breakups
4.4-4	X			No planned breakups
4.5-1	X			Limit debris by collision
4.5-2	X			Complies with Streamlined requirements.
4.6-1(a)	X			Atmospheric reentry option
4.6-1(b)	X			NA - storage orbit option
4.6-1(c)	X			NA - direct retrieval option
4.6-2	X			Not Applicable (GEO)
4.6-3	X			Not applicable (MEO)
4.6-4	X			Complies with FCC five-year de-orbit requirement
4.7-1	X			Reliability of disposal option
4.8-1	X			No tethers used

1. This ODAR is for the Umbra-11, Umbra-12, Umbra-13, Umbra-14, and Umbra-14 ALTERNATE only. No launch vehicle was assessed.
2. This assessment was performed using DAS with DAS solar flux files dated 09/26/2024.

1.2. Assessment Report Format

ODAR Technical Sections Format Requirements:

This ODAR follows the format recommended in NASA-STD-8719.14C, Appendix A.1 and includes the content indicated at a minimum in each Section 2 through 8 below for the Umbra satellites. Sections 9 through 14 apply to the launch platform and are not addressed herein.

2. PROGRAM MANAGEMENT AND MISSION OVERVIEW

2.1. Project Manager

Michael Francis
Operating Manager, Space Systems
Umbra Lab, Inc.

2.2. Foreign Government or Space Agency Participation

None

2.3. Nominal Mission Design and Development Milestones

Nominal mission design and development milestones are described in Table 2 below.

Table 2. Nominal Mission Design and Development Milestones

Phase	Duration	Altitude Range	Description
Launch	See Table **3** "Launch Date"	590 ± 20 km	Launch and orbit insertion
Phase 1: Checkout	< 2 months	Launch Altitude to ≤ 595 km	Checkout and orbit transfer
Phase 2: Higher Operational Orbit	≤ 28 months	565 km ± 30 km	Radar remote sensing
Phase 3: End-of-life Operations	≤ 36 months	535 km to 325 km	Continued remote sensing to 325 km, including maneuvers to elliptical orbit for disposal
Phase 4: End of Mission	≤ 3 months	≤ 325 km	End of mission maneuvering
	≤ 69 months		

2.4. Mission Overview

The Block 3 Tranche 1 is a space-based commercial remote sensing system. It features a synthetic aperture radar ("SAR") that can produce high resolution SAR imagery. The space segment will be inserted via rideshare on a third-party launch vehicle. The ground segment will include a mission operations center and commercial ground stations.

The Block 3 Tranche 1 will contain four satellites, including Umbra-11, Umbra-12, and Umbra-13. The fourth satellite in Block 3 Tranche 1 will be either Umbra-14 or Umbra-14 ALTERNATE, dependent on launch window timelines. Umbra-14 has the same mechanical parameters as Umbra-11, Umbra-12, and Umbra-13, while Umbra-14 ALTERNATE has increased mass associated with an update to the radar payload hardware configuration. .

2.5. Launch Vehicle Description

Table 3 lists the current best estimates for launch parameters associated with the Block 3 Tranche 1 that may apply to the Block 3 Tranche 1 ODAR analysis.

Table 3. Launch Parameters for Block 3 Tranche 1

Orbital Vehicle	Launch Vehicle	Launch Site	Launch Date	Orbit Type
UMBRA-11	SpaceX Transporter-9	Vandenberg AFB or Cape Canaveral	June 2025	Polar
UMBRA-12	SpaceX TSIS-2	Vandenberg AFB or Cape Canaveral	Q4 2025	Polar
UMBRA-13	SpaceX Transporter-15	Vandenberg AFB or Cape Canaveral	Q4 2025	Polar
UMBRA-14	SpaceX Mid-Mission	Vandenberg AFB or Cape Canaveral	Q4 2025	Non-Polar
UMBRA-14 ALTERNATE	SpaceX Mid-Mission	Vandenberg AFB or Cape Canaveral	Q4 2025	Non-Polar

2.6. Launch and Deployment Profile

The launch insertion altitude is estimated to be approximately 590 km, which is reflected in Table 4 below.

The nominal operational orbit for the space vehicle is near circular with a nominal altitude of 565 km. The space vehicle may maneuver from the orbital altitude of separation to the desired nominal operational orbit via a series of Hohmann transfers and minor inclination change maneuvers (if required).

Table 4. Orbital Envelopes

	Orbital Altitude	Polar Inclination	Non-Polar Inclination
Target Insertion Altitude	590 ± 20 km	SSO ± 0.1 deg	SSO ± 0.1 deg
Phase 2 Nominal Operational Orbit Range	535 – 595 km (nominal 565 km)[4]	97.4 ± 1 deg	53.5 ± 1 deg
Phase 3 End of Life Operational Orbit Range	325 – 535 km	97.4 ± 1 deg	53.5 ± 1 deg
Nominal Disposal Orbit	535 x 415 km	97.4 ± 1 deg	53.5 ± 1 deg
Applicable Satellites	Umbra-11, Umbra-12, Umbra-13, Umbra-14, and Umbra-14 ALTERNATE	Umbra-11, Umbra-12, and Umbra-13	Umbra-14 and Umbra-14 ALTERNATE

2.7. Interaction with Other Operational Spacecraft

Interaction and potential physical interference with other operational spacecraft are not planned nor anticipated as part of the Block 3 Tranche 1 mission. Umbra is aware that other operators operate in the 325–595 km orbital range and intends to coordinate physical operations of its

[4] The Block 3 Tranche 1 satellites are capable of performing station-keeping activities to maintain better than ± 30 km variance in the planned operational orbital altitude over the life of the mission. *See infra* Section 3.4.

satellites with all such operators, as necessary. Umbra satellite operators routinely coordinate maneuvers of currently deployed satellites to avoid conjunction events as needed.

3. SPACECRAFT DESCRIPTION

3.1. Physical Description of the Spacecraft

All Umbra Block 3 Tranche 1 satellites have the same physical dimensions in each configuration.

The bus structure consists of an aluminum frame with machined aluminum panels and has dimensions of approximately 0.6 m x 0.6 m x 0.4 m, not including the solar arrays which reside in a stowed condition on either side of the bus. The payload is approximately 0.6 m in diameter and 0.9 m in length in the stowed position. When deployed into the operational configuration, the maximum physical dimensions of the space vehicle are approximately 4 m x 4 m x 2 m.

3.2. Spacecraft Illustration

The figures below show both the stowed and operational configurations of each Block 3 Tranche 1 space vehicle. The details of the payload are not shown, but approximate relative dimensions are captured.

Figure 1. Umbra Space Vehicle Deployed Side View

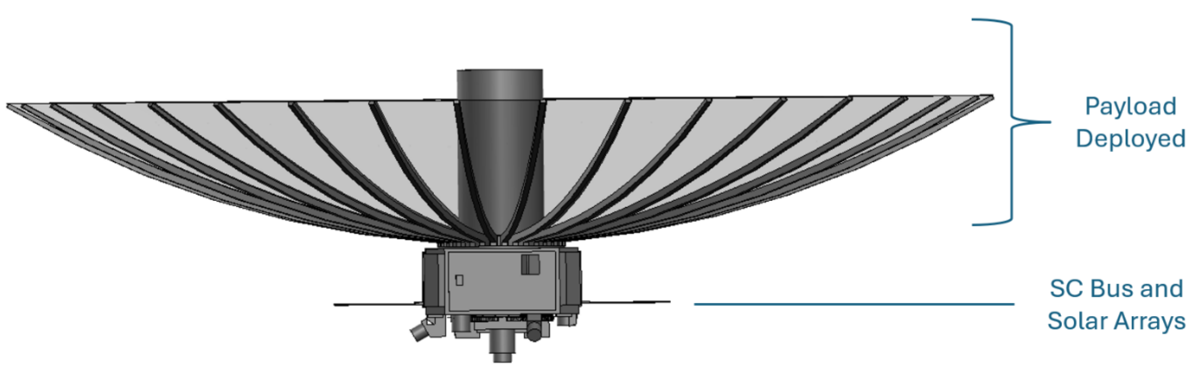


Figure 2. Umbra Space Vehicle Deployed Bottom View

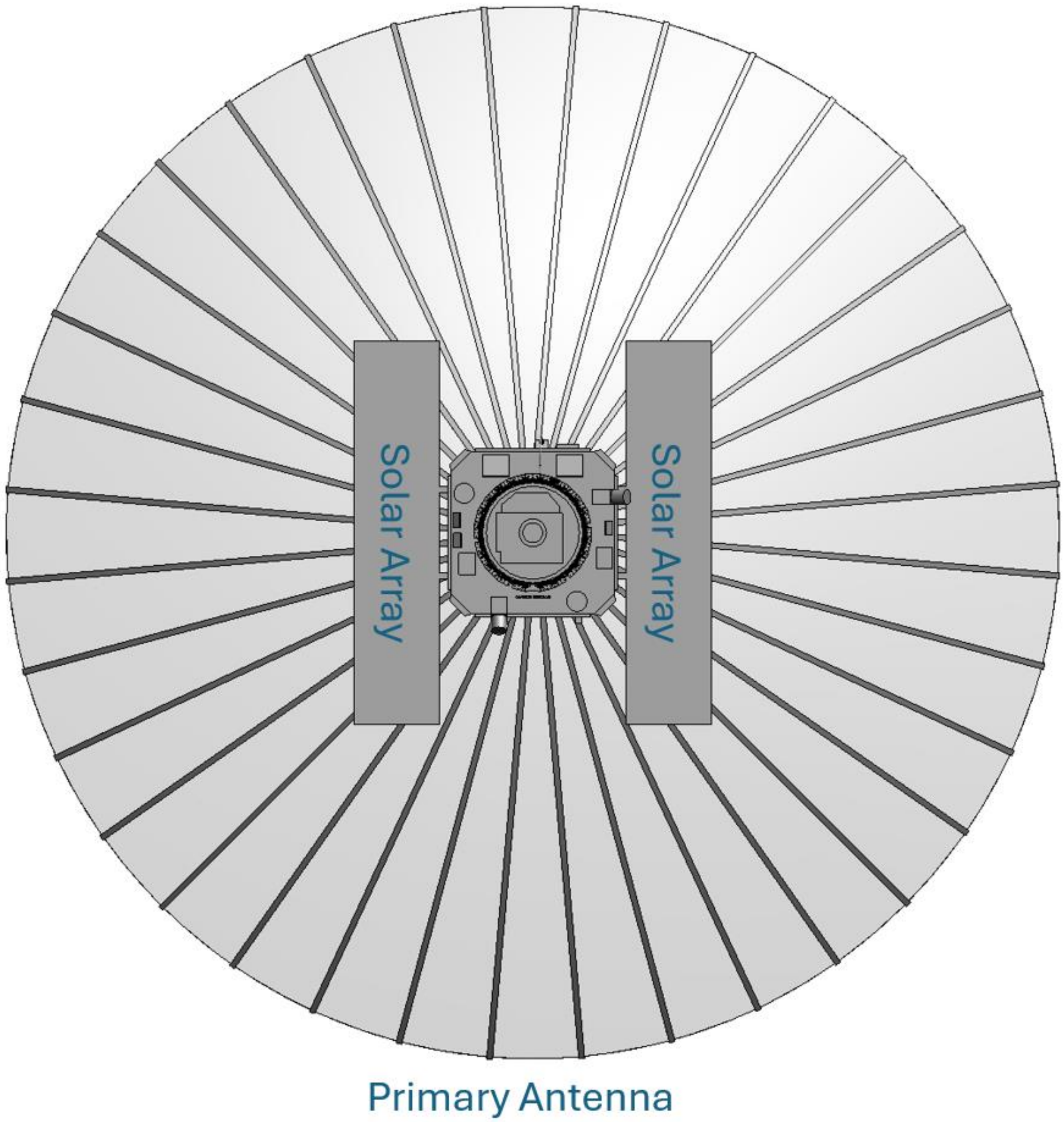


Figure 3. Umbra Space Vehicle Stowed Side View

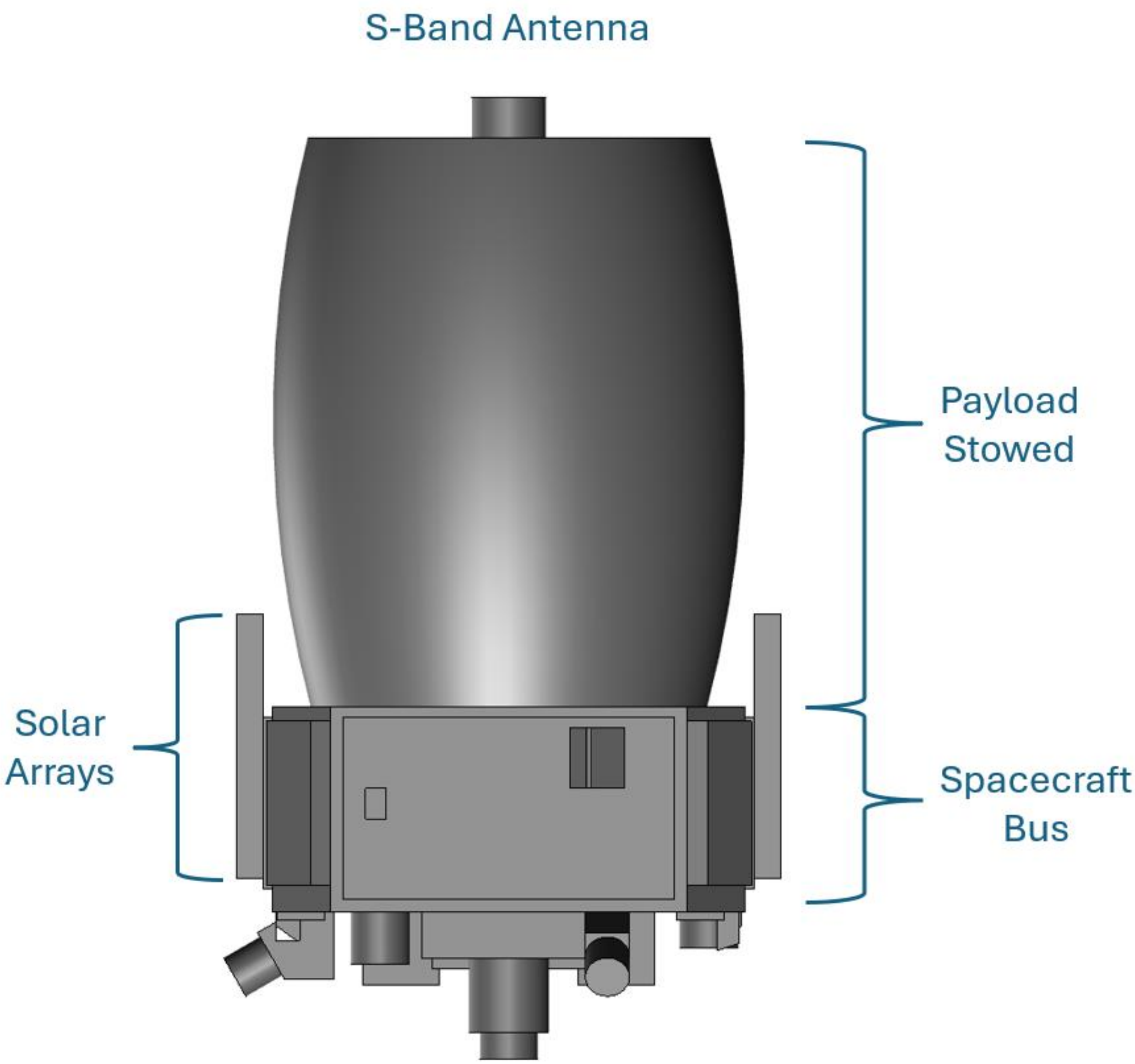
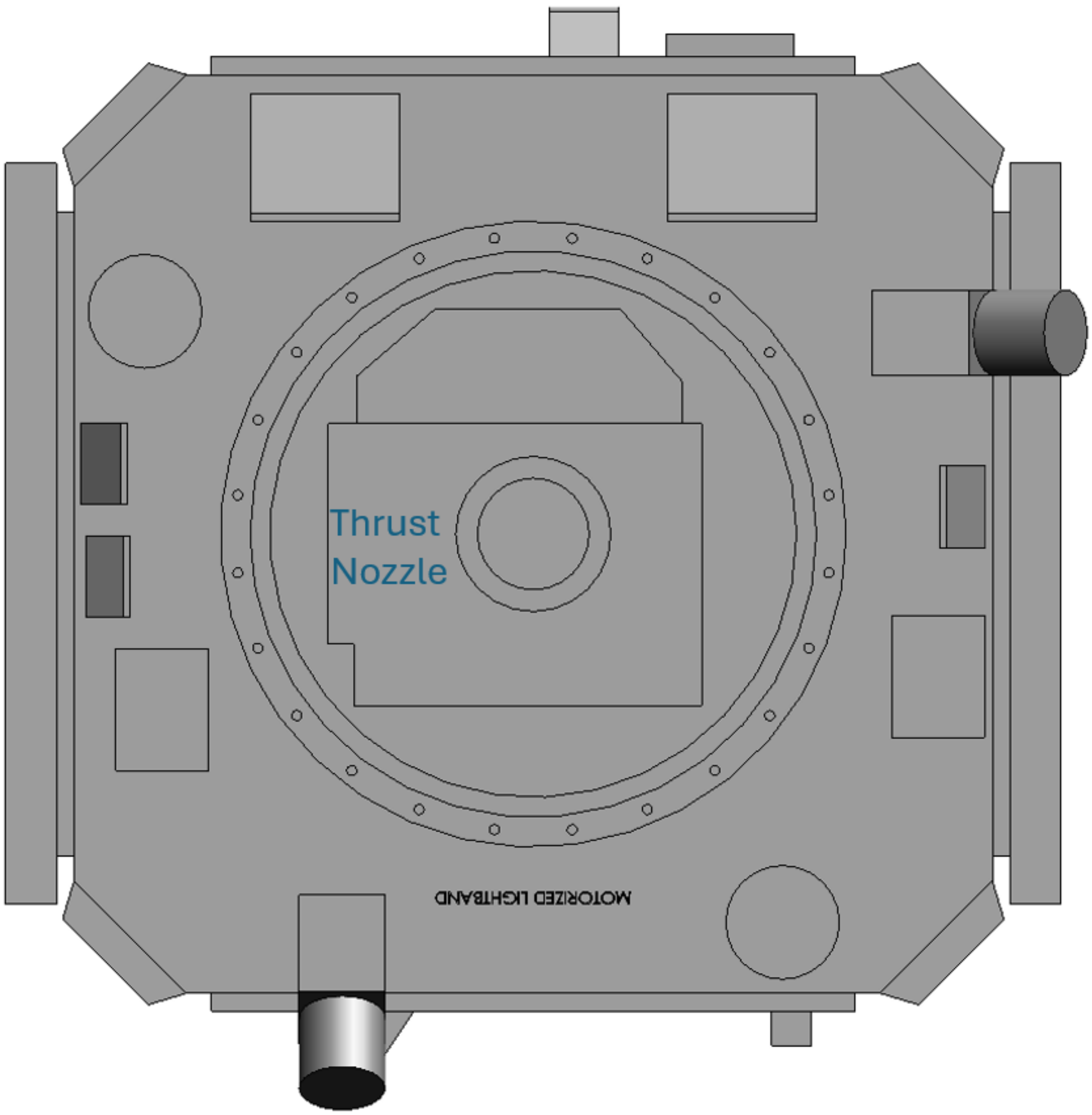


Figure 4. Umbra Space Vehicle Stowed Bottom View



3.3. Space Vehicle Mass

The satellites Umbra-11, Umbra-12, Umbra-13, and Umbra-14 have the same mass information as described in Table 5 in the “GENERAL” column. Umbra-14 ALTERNATE has unique mass information as described in the “ALTERNATE” column of Table 5.

The mass information in Table 5 is used to calculate all DAS values for compliance assessment.

Table 5. Mass Information Used in Analysis

	GENERAL	ALTERNATE
Wet Mass (kg)	104.3	111.7
Dry Mass (kg)	101.9	109.3
Stowed Area-to-Mass Ratio (wet, m^2/kg)	5.2 $x10^{-3}$	4.9 $x10^{-3}$
Deployed Area-to-Mass Ratio (dry, m^2/kg)	16.3 $x10^{-3}$	15.2 $x10^{-3}$
PMD Area-to-Mass Ratio (dry, m^2/kg)	26.8 $x10^{-3}$	25.6 $x10^{-3}$

3.4. Propulsion System

All Block 3 Tranche 1 satellites utilize the same propulsion system as described in this section.

Satellite propulsion is provided by a hall effect thruster propulsion system with extensive flight heritage that uses xenon as its propellant. The xenon propellant will be loaded through a fill/drain valve prior to satellite integration on the launch vehicle. The system consists of a single thruster, a single propellant tank, a plenum, fill & drain ports, and an electronics enclosure. All pressurized components will be acceptance tested to standards required by SpaceX for Rideshare vehicles.

The xenon-based propulsion system will be used for station-keeping and collision avoidance as necessary. Table 6 contains some technical details of the propulsion system capabilities. A more detailed collision avoidance process is provided in Section 6.

Table 6. On-Board Propulsion Metrics

ΔV	140	m/s
Nominal Acceleration	2.3×10^{-4}	m/s^2
ISP	600	sec

The propulsion system is sufficiently capable of performing station-keeping activities to maintain better than ± 30 km within the planned orbital altitude over the life of the mission. It can do this while retaining the ability to perform any necessary collision avoidance maneuvers.

3.5. Fluids, Fluid Management, Fluid Systems

All Block 3 Tranche 1 satellites utilize the same fluids, fluid management, and fluid systems as described in this section.

All fluids are contained within the propulsion system. The system includes a thruster, fill-drain valves for the propellant, carbon composite propellant tank with aluminum liners, steel plenum, and avionics. The qualified propulsion system module will be subject to random vibration, shock, and thermal cycling tests.

Table 7. Spacecraft Fluids

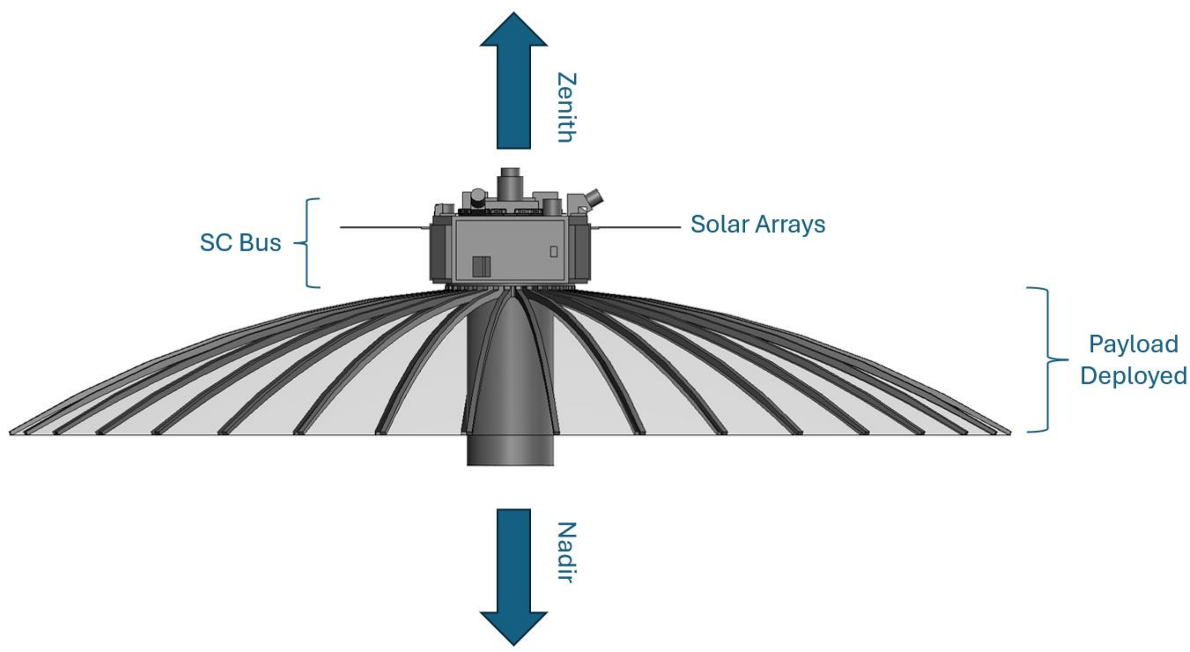
Description	Fluid	Mass (kg)	Max Pressure (psi)
Propellant	Xe	< 2.9	2000

3.6. Attitude Control Systems

All Block 3 Tranche 1 satellites utilize the same attitude control systems as described in this section.

Satellite attitude is controlled by torque rods and reaction wheels integrated into a 3-axis control system that also includes star trackers and sun sensors. The nominal attitude mode places the satellite in a "Nadir Pointing" orientation as shown in Figure 5 below. Satellite attitude will be varied among other pointing control modes to orient solar arrays towards the sun, the payload for imaging, and antennas for communication.

Figure 5. Nominal Block 3 Tranche 1 Attitude



3.7. Range Safety and Pyrotechnic Devices

None.

3.8. Electrical Generation and Storage System

The satellites Umbra-11, Umbra-12, Umbra-13, and Umbra-14 have power storage provided by a battery subsystem consisting of lithium-ion cells arranged in an 8S2P configuration in four (4) battery modules.

Umbra-14 ALTERNATE has power storage provided by a battery subsystem consisting of lithium-ion cells arranged in an 8S2P configuration in six (6) battery modules.

All Block 3 Tranche 1 satellite batteries will be recharged by solar cells mounted on the two (2) deployable solar array wings extending from the bus structure.

3.9. Other Sources of Stored Energy

None.

3.10. Radioactive Materials

None.

4. ASSESSMENT OF SPACECRAFT DEBRIS RELEASED DURING NORMAL OPERATIONS

Assessment of spacecraft compliance with Requirements 4.3-1 and 4.3-2 of NASA-STD-8719.14C.

4.1. Identification of any Object Expected to be Released

There are no intentional releases of objects.

4.2. Rationale for Release of Each Object

Not Applicable.

4.3. Time of Release for Each Object Relative to Launch Time

Not Applicable.

4.4. Release Velocity of Each Object with Respect to Spacecraft

Not Applicable.

4.5. Expected Orbital Parameters of Each Object After Release

Not Applicable.

4.6. Calculated Orbital Lifetime of Each Object

Not Applicable.

4.7. Compliance Assessment for Requirements 4.3-1 and 4.3-2

4.7.1. Requirement 4.3-1: Mission Related Debris Passing Through LEO

Compliance Statement (4.3-1): Compliant. Requirement is not applicable to the mission profile.

4.7.2. Requirement 4.3-2: Mission Related Debris Passing Near GEO

Compliance Statement (4.3-2): Compliant. Requirement is not applicable to the mission profile.

5. ASSESSMENT OF SPACECRAFT INTENTIONAL BREAKUPS AND POTENTIAL FOR EXPLOSIONS

5.1. Potential Causes of Spacecraft Breakup During Deployment and Mission Operations

There is no credible scenario that would result in spacecraft breakup during normal deployment and operations.

5.2. Summary of Failure Modes and Effects Analyses Which May Lead to an Accidental Explosion

Rupture of a lithium-ion cell leading to explosion or breakup of the space vehicle is not a credible scenario. The battery sub-system would have to survive the harsh launch vehicle environment then fail in a way that releases enough energy to puncture the solid aluminum of the spacecraft body.

In-mission failure of the propulsion system leading to explosion or breakup of the space vehicle is not a credible scenario. The propulsion system would have to survive the harsh launch vehicle environment then fail in a way that releases enough energy to puncture the solid aluminum of the spacecraft body. The propulsion system tank in use has been qualified for pressure cycling with safety factor exceeding planned testing and on-orbit usage.

5.3. Plan for Any Designed Spacecraft Breakup

There are no planned breakups.

5.4. Components Which are Passivated at EOM

5.4.1. Propulsion System

Residual propellant will be depleted via end-of-mission ("EOM") burns or venting upon demise at the end of mission. The propellant (xenon) is inert and is not toxic, thus its release does not pose any credible hazard. Per the manufacturer, it is non-corrosive, electrically non-conductive, free of residue, and has zero ozone depleting potential. As the propellant used in this case is xenon, there is no risk from persistent liquids because any release of propellant evaporates and dissipates. This propellant is unable to persist in droplet form in the space environment.

5.4.2. Batteries

Batteries will not be passivated at EOM due to the low risk and low impact of a cell or cells rupturing and the extremely short lifetime at mission conclusion. The maximum total chemical energy stored in each lithium-ion cell is 15 kJ. If a single cell were to rupture, the debris would be contained within the rugged battery housing, which itself is contained within an aluminum bus structure. These structures would retain any debris that could be ejected by a ruptured cell.

5.4.3. Rationale for Non-Passivation

The battery and solar array configurations were designed in concert to minimize the possibility of overcharging the battery. However, in the unlikely event that a battery cell does rupture, the small size, mass, and potential energy of these batteries is such that, while the spacecraft could be expected to vent gases, debris from the battery rupture would be contained within the vessel due to the lack of penetration energy.

5.4.4. Compliance Assessment for Requirements 4.4-1 to 4.4-4

Umbra has completed Failure Mode and Effects Analysis ("FMEA") and concluded that the appropriate steps have been taken to assure that any failure of energetic components (limited to batteries and propulsion system) do not result in fragmentation of the satellites or do not otherwise generate orbital debris.[5] As described above, energy sources are both safely contained during the mission and/or depleted at the time of post mission disposal.

6. ASSESSMENT OF SPACECRAFT POTENTIAL FOR ON-ORBIT COLLISIONS

Umbra has developed a standard course of action for receipt of a conjunction data message ("CDM"). Immediately following receipt of the CDM, Umbra will evaluate, using the information provided in the message, whether the associated risk falls above or below the predetermined threshold. The preliminary set point for this threshold is 1×10^{-4}, however this is software configurable and may be subject to change as necessary. If the risk stated by the United States Space Force's ("USSF") Space Defense Squadrons is higher than this threshold, Umbra will contact the other entity sharing the potential collision risk using information provided in the CDM, if any. Umbra will then collaborate to avoid a collision. This process has been demonstrated through our current on-orbit operations. However, Umbra plans to actively work towards a more optimized process that minimizes the response time to perform any necessary coordination and/or maneuvers.

6.1. Calculation of Spacecraft Probability of Collision

Assessment of spacecraft compliance with Requirements 4.5-1 and 4.5-2 per NASA-STD-8719.14c was performed using DAS.

6.2. Compliance Assessment for Requirement 4.5-1 and 4.5-2

6.2.1. **Requirement 4.5-1:** Limiting debris generated by collisions with large objects when operating in Earth orbit:

[5] *See infra* Section 9.A.2.

For each spacecraft and launch vehicle orbital stage in or passing through LEO, the program or project shall demonstrate that, during the orbital lifetime of each spacecraft and orbital stage, the probability of accidental collision with space objects larger than 10 cm in diameter is less than 0.001 (Requirement 56506).

Compliance Statement (4.5-1):

Compliant. As shown in Table 8 below, the computed probability for Large Object Impact and Debris Generation for each satellite (and the system as a whole) is less than 0.001, not considering propulsive maneuvers.

Table 8. Probability of Collision with Large Objects

Satellite Name	Year of Launch	Orbit Type	Probability Per Satellite
Umbra-11	2025.5	Polar	7.6103E-05
Umbra-12	2025.8	Polar	7.165E-05
Umbra-13	2025.8	Polar	7.165E-05
Umbra-14	2025.8	Non-Polar	3.7404E-05
Umbra-14 ALTERNATE	2025.8	Non-Polar	3.8944E-05

6.2.2. Requirement 4.5-2: Limiting debris generated by collisions with small objects when operating in Earth or lunar orbit:

For each spacecraft, the program or project shall demonstrate that, during the mission of the spacecraft, the probability of accidental collision with orbital debris and meteoroids sufficient to prevent compliance with the applicable post-mission disposal requirements is less than 0.01 (Requirement 56507).

Compliance Statement (4.5-2):

Compliant. As shown below in Table 9, the probability of collision with small objects resulting in PMD failure for each satellite is below the 0.01 requirement.

Table 10 lists the critical surfaces and their redundancies, which would allow the spacecraft to accomplish PMD in the event of a single critical surface-related failure. Taking this into account, we believe the probability of PMD failure due to collision with small objects to be less than 4 x 10^{-4} for each satellite.

The components of the system that pose the greatest risk are listed in Table 10.

Table 9. Probability of PMD failure due to Collision with Small Objects

Satellite Name	Year of Launch	Orbit Type	Probability Per Satellite
Umbra-11	2025.5	Polar	1.0210E-03
Umbra-12	2025.8	Polar	1.0548E-03
Umbra-13	2025.8	Polar	1.0548E-03
Umbra-14	2025.8	Non-Polar	5.943E-04
Umbra-14 ALTERNATE	2025.8	Non-Polar	5.943E-04

Table 10. Small Object Damage Analysis

Critical Surface	Umbra-11 Redundancy	Umbra-12 Redundancy	Umbra-13 Redundancy	Umbra-14 Redundancy	Umbra-14 ALTERNATE Redundancy
Propulsion Electronics	None	None	None	None	None
S-Band Radio	Single	Single	Single	Single	None
GNC	None	None	None	None	None
Battery Module	Triple	Triple	Triple	Triple	Quintuple

7. ASSESSMENT OF SPACECRAFT POST-MISSION DISPOSAL PLANS AND PROCEDURES

7.1. Description of Spacecraft Disposal Option Selected

The satellites will deorbit by atmospheric re-entry as described in Section 7.3 Post Mission Disposal Plan.

7.2. Systems or Components Required to Accomplish Post-Mission Disposal Operations

In a worst-case scenario for any individual satellite that is deployed DOA to a 610 x 610 km orbit and remains in its stowed configuration due to a hardware malfunction, it will deorbit by atmospheric reentry in the timeframe shown in Table 11.

Table 11 Worst-Case DOA Reentry Timelines

Satellite Name	Year of Launch	Worst-Case DOA Time to Reentry
Umbra-11	2025.5	33.4 years
Umbra-12	2025.8	33.4 years
Umbra-13	2025.8	33.4 years
Umbra-14	2025.8	34.6 years
Umbra-14 ALTERNATE	2025.8	35.9 years

7.3. Disposal Plan

Nominally, the maneuvers to enter Phase 3 performed by Umbra will achieve an operational orbit of approximately 535 x 415 km. While conducting maneuvers to achieve this orbital configuration Umbra intends to continue imaging operations when not affecting required thrust maneuvers. With respect to the operational strategies to minimize risk to the ISS or any other inhabitable spacecraft in low Earth orbit, Umbra intends to monitor the location of its satellites and work closely with organizations, such as the USSF Space Defense Squadrons, to maximize predictability of Umbra spacecraft while crossing those systems' orbital altitude, and Umbra will revise our maneuver plan as necessary. From the achieved elliptical Phase 3 orbital configuration Umbra plans to continue imaging operations while the system naturally decays to an approximately circular orbit at 325-km. At this altitude an EOM maneuver is performed to orient the spacecraft into a configuration with the Z-Axis aligned with the velocity vector. This orientation maximizes drag and accelerates the deorbit of a non-functional satellite with minimal external input.

Umbra plans to begin the maneuver to Phase 3 operations with sufficient time to ensure that the total in-orbit lifetime of each satellite is six years or less as required.[6] Umbra expects this process to follow the mission milestones in Table 2 such that the start of Phase 3 circular orbit with 535 km altitude is achieved ≤ 28 months after the launch of each satellite. Each satellite will then maneuver to a 535 x 415 km orbital configuration as shown below in Figure 6 and described in Table 12. At this point the EOM maneuver results in deorbit as described in Table 12 in ≤ 3 months concluding operations ≤ 69 months after launch.

[6] 47 C.F.R. § 25.122(c)(2) ("The total in-orbit lifetime for any individual space station will be six years or less").

Figure 6. Orbital Decay Profile with PMD Plan and EOM Maneuver

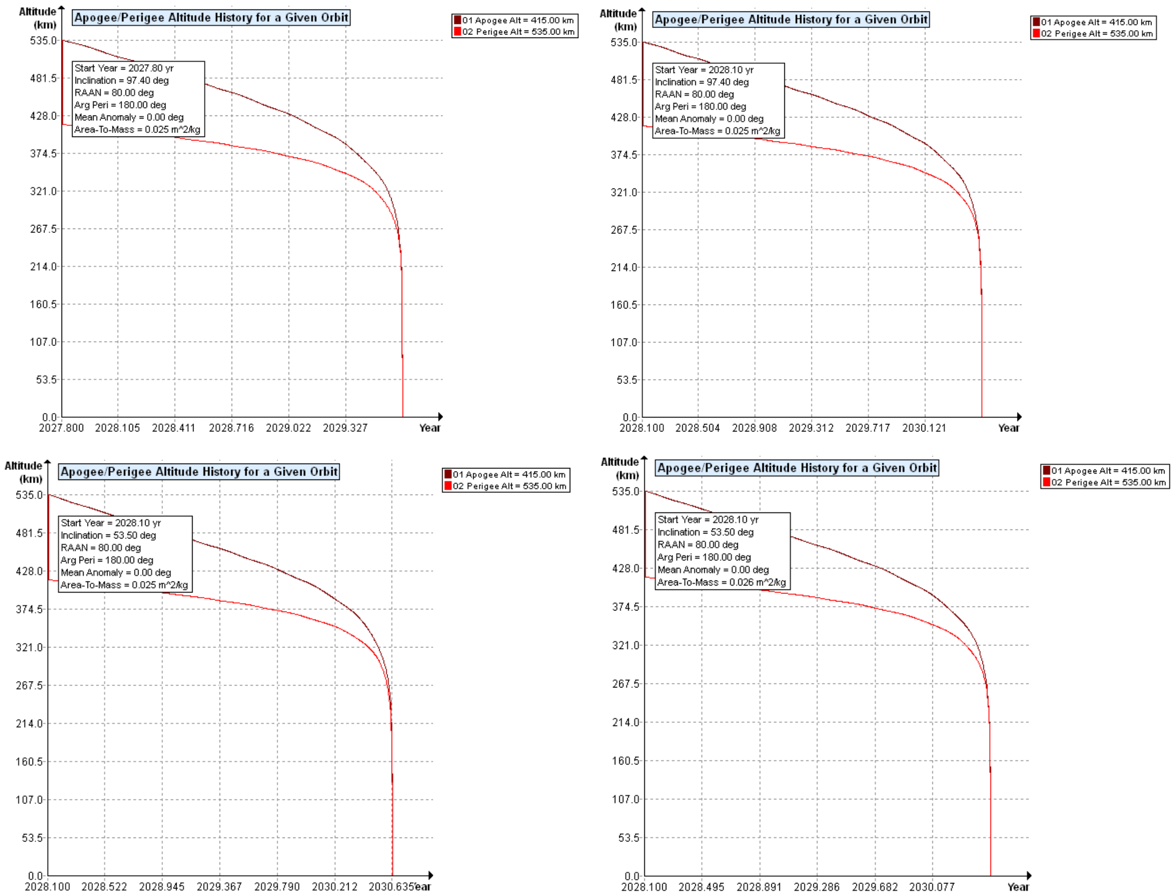


Table 12. PMD Timings with EOM Maneuver

Satellite Name	Year of Launch	Orbit Type	Approx Date Begin Maneuver to Disposal Orbit	Approx Date Disposal Orbit Reached	Approx Date 325 km Alt Reached	Approx Date of Reentry
Umbra-11	2025.5	Polar	2027.5	2028.0	2030.1	2030.2
Umbra-12	2025.8	Polar	2027.6	2028.1	2030.4	2030.5
Umbra-13	2025.8	Polar	2027.6	2028.1	2030.4	2030.5
Umbra-14	2025.8	Non-Polar	2027.6	2028.1	2030.5	2030.6
Umbra-14 ALTERNATE	2025.8	Non-Polar	2027.6	2028.1	2030.3	2030.4

The described timings in Table 12 are approximate and subject to change based on updated simulation of solar cycle conditions. Umbra will ensure that the Date of Reentry with any modification to these timings is compliant with the FCC orbital debris requirements[7] as described in Section 7.5.4

7.4. Preliminary Plan for Spacecraft Controlled Reentry

Not Applicable.

7.5. Compliance Assessment for Requirement 4.6-1 to 4.6-4

7.5.1. Requirement 4.6-1: Disposal for space structures passing through LEO.

Compliance Statement (4.6-1):

Umbra-11, Umbra-12, Umbra-13, Umbra-14 and Umbra-14 ALTERNATE reentry are COMPLIANT using 4.6.2.1 described within NASA-STD 8719.14c. Each of these satellites, after Phase 3 maneuvers and Phase 4 EOM maneuver, will re-enter the Earth's atmosphere < 3 years after the completion of mission and will be commanded to re-enter within the required timeline under streamlined Part 25 license rules.

7.5.2. Requirement 4.6-2: Disposal for space structures near GEO.

Compliance Statement (4.6-2):

The requirement is not applicable. Umbra-11, Umbra-12, Umbra-13, Umbra-14 and Umbra-14 ALTERNATE will not be located or disposed of near GEO.

7.5.3. Requirement 4.6-3: Disposal for space structures between LEO and GEO.

Compliance Statement (4.6-3):

The requirement is not applicable. Umbra-11, Umbra-12, Umbra-13, Umbra-14 and Umbra-14 ALTERNATE will not be located or disposed of between LEO and GEO.

7.5.4. Requirement 4.6-4: Reliability of post-mission disposal operations in Earth Orbit.

[7] Federal Communications Commission, Space Innovation; Mitigation of Orbital Debris in the New Space Age**,** Final Rule, 89 Fed. Reg. 54764 (Aug. 9, 2024) ("FCC-22-74"). Additionally, 47 C.F.R. § 25.122(c)(2) ("The total in-orbit lifetime for any individual space station will be six years or less").

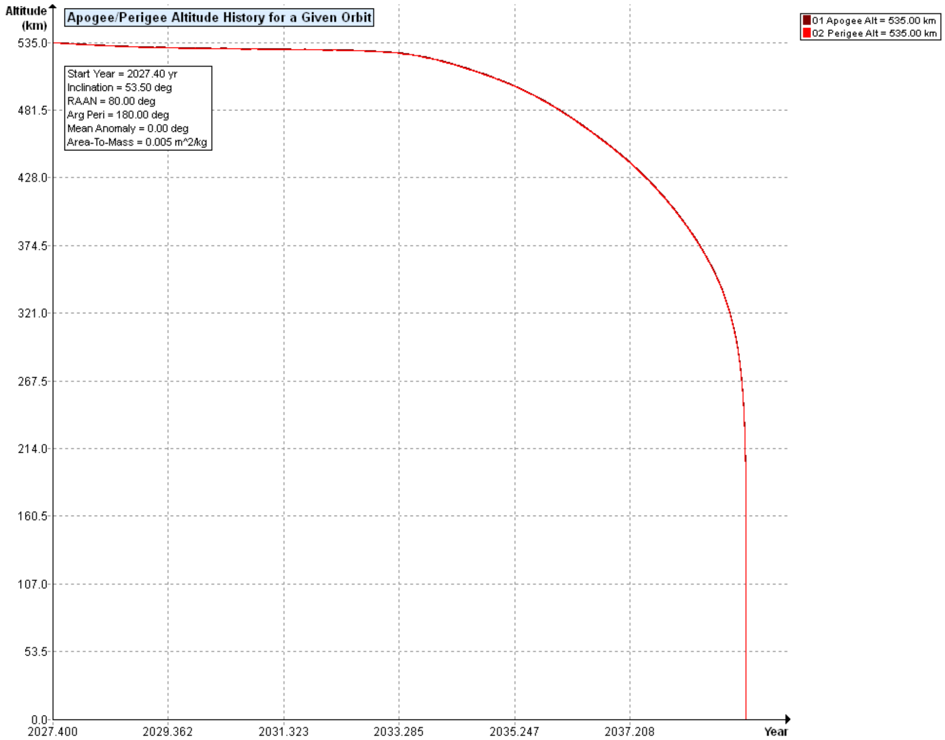
Compliance Statement (4.6-4):

Compliant. Umbra-11, Umbra-12, Umbra-13, Umbra-14 and Umbra-14 ALTERNATE, after executing Phase 3 maneuvers and Phase 4 EOM maneuver, will re-enter the Earth's atmosphere 3 years after the completion of mission. An EOM maneuver is not required to limit the probability of human casualty to 1:10,000 per Requirement 4.7-1.[8]

Umbra expects to begin PMD maneuvers from an altitude of 535 km at the bottom of the Phase 2 operational range and has had a 100% success rate for the hardware GNC controller, Flight Computer, and radios required to orient the spacecraft in the EOM configuration. Additionally, the propulsion system used in all Block 3 Tranche 1 vehicles has extensive flight heritage. Our on orbit fuel budget allows a composite margin of 30% including deobit maneuvers and the total impulse has a margin of over 950% against the propulsion vendor's tested hardware lifetime. Further analysis for Requirement 4.5-2 confirms there is 0.004% chance that orbital debris would prevent Umbra from conducting a PMD maneuver. Because fewer components are necessary to achieve the EOM spacecraft configuration, there is a less than 0.04% chance that small object debris could affect the satellites' ability to re-enter within 2.5 years as described and as such the Block 3 Tranche 1 satellites readily meet the 90% reliability requirement.

[8] *See infra* Section 8.4.

Figure 7. Orbital Decay Profile In the Event of Hardware Failure (DAS v3.2.6)



8. ASSESSMENT OF SPACECRAFT REENTRY HAZARDS

8.1. Detailed Description of Spacecraft Components

Table 13 provides the Spacecraft Model used in DAS for Umbra-11, Umbra-12, Umbra-13, and Umbra-14. Table 14 provides the Spacecraft Model used in DAS for Umbra-14 ALTERNATE.

Table 13. Spacecraft Model – Umbra-11, Umbra-12, Umbra-13, and Umbra-14

Parent Component	Level 1 Sub-Component	Level 2 Sub-Component	Material	Qty.	Mass (kg)
Bus Structure			Aluminum 7075-T6	1	36.5
	Battery Chassis		Aluminum (generic)	4	1.4
		Battery	Aluminum (generic)	16	0.05
	Torq Rods		Aluminum (generic)	3	0.29
	Reaction Wheels		A356	4	1.38
	S-Band Radio Assembly		Aluminum (generic)	1	1.06
	Flight Avionics		Aluminum (generic)	1	1.6
	GNC Avionics		Aluminum (generic)	1	1.78

	XCVR Electronics		Aluminum (generic)	1	1.62
	Propulsion System		Aluminum 7075-T37	1	3.095
		Prop Tank	Aluminum (generic)	1	0.86
		Prop Feed	Aluminum (generic)	1	1.36
	Largest Fastener		Stainless Steel (generic)	62	0.01
	Avionics Harness		Copper Alloy	24	0.6
	Largest Connector		Steel AISI 304	2	0.01
	Switch		Aluminum (generic)	1	0.52
Prop Thruster			Aluminum (generic)	1	1.13
MLB			Aluminum (generic)	1	0.7
Solar Array Hinge			Ti-6Al-4V	6	0.003
Solar Array			Graphite Epoxy 1	2	4.08
SAR rib			Graphite Epoxy 1	108	0.07
Base Ring			Aluminum 7075-T6	1	0.14
Antenna Element			Aluminum 6061-T6	1	0.28
Canister			Aluminum 6061-T6	1	0.95
	Amplifier Assembly		Aluminum (generic)	1	1.9
	Strut Rod		Aluminum (generic)	6	0.094
	Electronics Chassis		Aluminum 7075-T6	3	0.14
Ti Hinge			Titanium (6 Al-4 V)	8	0.003
Ti Hinge 2			Titanium (6 Al-4 V)	108	0.0182
Root Hinge			Aluminum 7075-T6	2	0.08

Table 14. Spacecraft Model – Umbra-14 ALTERNATE

Parent Component	Level 1 Sub-Component	Level 2 Sub-Component	Material	Qty.	Mass (kg)
Bus Structure			Aluminum 7075-T6	1	37.1
	Battery Chassis		Aluminum (generic)	4	1.4
		Battery	Aluminum (generic)	16	0.05
	Torq Rods		Aluminum (generic)	3	0.29
	Reaction Wheels		A356	4	1.98
	S-Band Radio Assembly		Aluminum (generic)	1	1.06
	Flight Avionics		Aluminum (generic)	1	1.6

	GNC Avionics		Aluminum (generic)	1	1.78
	XCVR Electronics		Aluminum (generic)	1	1.62
	Propulsion System		Aluminum 7075-T37	1	3.095
		Prop Tank	Aluminum (generic)	1	0.86
		Prop Feed	Aluminum (generic)	1	1.36
	Largest Fastener		Stainless Steel (generic)	62	0.01
	Avionics Harness		Copper Alloy	24	0.6
	Largest Connector		Steel AISI 304	2	0.01
	Switch		Aluminum (generic)	1	0.52
Prop Thruster			Aluminum (generic)	1	1.13
Q-Sep			Aluminum (generic)	1	1.07
Solar Array Hinge			Ti-6Al-4V	6	0.003
Solar Array			Graphite Epoxy 1	2	4.08
SAR rib			Graphite Epoxy 1	108	0.07
Base Ring			Aluminum 7075-T6	1	0.14
Antenna Element			Aluminum 6061-T6	1	0.28
Canister			Aluminum 6061-T6	1	0.95
	Antenna Battery Chassis		Aluminum (generic)	2	1.4
	Antenna Battery		Aluminum (generic)	16	0.05
	Amplifier Assembly		Aluminum (generic)	1	2.25
	Strut Rod		Aluminum (generic)	6	0.094
	Electronics Chassis		Aluminum 7075-T6	3	0.14
Ti Hinge			Titanium (6 Al-4 V)	8	0.003
Ti Hinge 2			Titanium (6 Al-4 V)	108	0.0182
Root Hinge			Aluminum 7075-T6	2	0.08

Table 15 provides the details showing modeling constraints replicated for Umbra-11, Umbra-12, Umbra-13, and Umbra-14. Table 16 provides the details showing modeling constraints replicated for Umbra-14 ALTERNATE.

These tables also show the DAS-calculated demise altitude for components of each Umbra satellite. The DAS-calculated debris casualty risk was shown to be compliant.

Table 15. Spacecraft Component List for Human Casualty Risk Analysis - Umbra-11, Umbra-12, Umbra-13, and Umbra-14

Reentry Data														
Row Num	Name	Parent	Qty	Material	Body Type	Thermal Mass	Diameter/Width	Length	Height	Status	Risk	Demise Alt	Total DCA	KE
1	UMBRABLOCK3	0	1	Aluminum (generic)	Box	101.9	0.62	0.624	0.34	Compliant	1:13400		3.98	
2	Bus	1	1	Aluminum (generic)	Box	36.5	0.62	0.624	0.34			65.7	0	0
3	Battery Chassis	2	4	Aluminum (generic)	Box	1.4	0.12	0.2	0.05			60.3	0	0
4	Battery	3	16	Aluminum (generic)	Cylinder	0.05	0.019	0.065				57.1	0	0
5	Torq Rods	2	3	Aluminum (generic)	Cylinder	0.29	0.033	0.122				61.6	0	0
6	Reaction Wheels	2	4	A356	Box	1.38	0.146	0.146	0.039			57.2	0	0
7	S-Band Radio Assembly	2	2	Aluminum (generic)	Box	1.06	0.103	0.122	0.045			58.6	0	0
8	Flight Avionics	2	1	Aluminum (generic)	Box	1.6	0.107	0.184	0.107			61	0	0
9	GNC Avionics	2	1	Aluminum (generic)	Box	1.78	0.107	0.201	0.107			60.8	0	0
10	XCVR Electronics	2	1	Aluminum (generic)	Box	1.62	0.17	0.209	0.03			58	0	0
11	Propulsion System	2	1	Aluminum (generic)	Cylinder	3.095	0.3	0.36				62.9	0	0
12	Prop Tank	11	1	Aluminum (generic)	Cylinder	0.86	0.109	0.24				59.6	0	0
13	Prop Feed	11	1	Aluminum (generic)	Cylinder	1.36	0.06	0.18				53	0	0
14	Largest Fastener	2	62	Stainless Steel (generic)	Cylinder	0.01	0.006	0.05				64.4	0	0
15	Avionics Harness	2	24	Copper Alloy	Cylinder	0.6	0.035	0.54				64.2	0	0
16	Largest Connector	2	2	Steel AISI 304	Box	0.01	0.01	0.036	0.008			64.8	0	0
17	Switch	2	1	Aluminum (generic)	Box	0.52	0.08	0.133	0.02			61.4	0	0
18	Prop Thruster	1	1	Aluminum (generic)	Cylinder	1.13	0.0775	0.122				69.9	0	0
19	MLB	1	1	Aluminum (generic)	Cylinder	0.7	0.1	0.22				75.2	0	0
20	Solar Array	1	2	Graphite Epoxy 1	Flat Plate	4.08	0.45	1.378				0	3.98	240.69
21	SAR rib	1	108	Graphite Epoxy 1	Flat Plate	0.07	0.13	0.624				78	0	0
22	Base Ring	1	1	Aluminum 7075-T6	Cylinder	0.14	0.025	0.351				77.1	0	0
23	Antenna Element	1	1	Aluminum 6061-T6	Flat Plate	0.28	0.334	0.334				77.8	0	0
24	Canister	1	1	Aluminum 6061-T6	Cylinder	0.95	0.655	1.049				77.9	0	0
25	Amplifier Assembly	24	1	Aluminum (generic)	Box	1.9	0.13	0.179	0.04			70.8	0	0
26	Strut Rod	24	6	Aluminum (generic)	Cylinder	0.094	0.012	0.3				76.7	0	0
27	Electronics Chassis	24	3	Aluminum 7075-T6	Box	0.14	0.119	0.16	0.016			77.2	0	0
28	Ti Hinge	1	8	Aluminum 7075-T6	Cylinder	0.003	0.008	0.025				77.5	0	0
29	Ti rib Hinge	1	108	Aluminum 7075-T6	Flat Plate	0.0182	0.03	0.05				77.2	0	0
30	Root Hinge	1	2	Aluminum 7075-T6	Flat Plate	0.08	0.05	0.304				77.5	0	0
31	Solar Array Hinge	1	6	Titanium (6 Al-4 V)	Cylinder	0.003	0.007	0.025				76.5	0	0

Table 16. Spacecraft Component List for Human Casualty Risk Analysis - Umbra 14 ALTERNATE

Row Num	Name	Parent	Qty	Material	Body Type	Thermal Mass	Diameter/Width	Length	Height	Status	Risk	Demise Alt	Total DCA	KE
1	UMBRABLOCK3.1	0	1	Aluminum (generic)	Box	109.3	0.62	0.624	0.34	Compliant	1:13400		4	
2	Bus	1	1	Aluminum (generic)	Box	37.1	0.62	0.624	0.34			65.8	0	0
3	Battery Chassis	2	4	Aluminum (generic)	Box	1.4	0.12	0.2	0.05			60.5	0	0
4	Battery	3	16	Aluminum (generic)	Cylinder	0.05	0.019	0.065				57.4	0	0
5	Torq Rods	2	3	Aluminum (generic)	Cylinder	0.29	0.033	0.122				61.8	0	0
6	Reaction Wheels	2	4	Aluminum 6061-T6	Box	1.98	0.16	0.16	0.04			62.6	0	0
7	S-Band Radio Assembly	2	1	Aluminum (generic)	Box	1.06	0.103	0.122	0.045			58.9	0	0
8	Flight Avionics	2	1	Aluminum (generic)	Box	1.6	0.107	0.184	0.107			61.2	0	0
9	GNC Avionics	2	1	Aluminum (generic)	Box	1.78	0.107	0.201	0.107			61	0	0
10	XCVR Electronics	2	1	Aluminum (generic)	Box	1.62	0.17	0.209	0.03			58.4	0	0
11	Propulsion System	2	1	Aluminum (generic)	Cylinder	3.095	0.3	0.36				63.1	0	0
12	Prop Tank	11	1	Aluminum (generic)	Cylinder	0.86	0.109	0.24				59.9	0	0
13	Prop Feed	11	1	Aluminum (generic)	Cylinder	1.36	0.06	0.18				53.4	0	0
14	Largest Fastener	2	62	Stainless Steel (generic)	Cylinder	0.01	0.006	0.05				64.6	0	0
15	Avionics Harness	2	24	Copper Alloy	Cylinder	0.6	0.035	0.54				64.3	0	0
16	Largest Connector	2	2	Steel AISI 304	Box	0.01	0.01	0.036	0.008			64.9	0	0
17	Switch	2	1	Aluminum (generic)	Box	0.52	0.08	0.133	0.02			61.7	0	0
18	Prop Thruster	1	1	Aluminum (generic)	Cylinder	1.13	0.0775	0.122				70.1	0	0
19	Q-Sep	1	1	Aluminum (generic)	Flat Plate	1.07	0.67	0.67				77.3	0	0
20	Solar Array	1	2	Graphite Epoxy 1	Flat Plate	4.08	0.45	1.3878				0	4	238.99
21	SAR rib	1	108	Graphite Epoxy 1	Flat Plate	0.07	0.13	0.624				78	0	0
22	Base Ring	1	1	Aluminum 7075-T6	Cylinder	0.14	0.025	0.351				77.1	0	0
23	Antenna Element	1	1	Aluminum 6061-T6	Flat Plate	0.28	0.334	0.334				77.8	0	0
24	Canister	1	1	Aluminum 6061-T6	Cylinder	0.95	0.655	1.049				77.9	0	0
25	Antenna Battery Chassis	24	2	Aluminum (generic)	Box	1.4	0.12	0.2	0.05			73.7	0	0
26	Antenna Battery	25	16	Aluminum (generic)	Cylinder	0.05	0.019	0.065				71.6	0	0
27	Amplifier Assembly	24	1	Aluminum (generic)	Box	2.25	0.18	0.18	0.062			71.4	0	0
28	Strut Rod	24	6	Aluminum (generic)	Cylinder	0.094	0.012	0.3				76.8	0	0
29	Electronics Chassis	24	3	Aluminum 7075-T6	Box	0.14	0.119	0.16	0.016			77.3	0	0
30	Ti Hinge	1	8	Aluminum 7075-T6	Cylinder	0.003	0.008	0.025				77.5	0	0
31	Ti rib Hinge	1	108	Aluminum 7075-T6	Flat Plate	0.0182	0.03	0.05				77.2	0	0
32	Root Hinge	1	2	Aluminum 7075-T6	Flat Plate	0.08	0.05	0.304				77.5	0	0
33	Solar Array Hinge	1	6	Titanium (6 Al-4 V)	Cylinder	0.003	0.007	0.025				76.6	0	0

8.2. Summary of Objects Expected to Survive an Uncontrolled Reentry

Per DAS, the two solar array panels are expected to survive an uncontrolled reentry and reach the Earth's surface for all Block 3 Tranche 1 vehicles. The kinetic energy of each solar array panel is calculated to be approximately 100 joules, for a total of 200 joules of kinetic energy per satellite.

However, DAS does not calculate partial ablation,[9] so the kinetic energy displayed is an absolute maximum. Further, the solar array hinges (the only mechanisms attaching the solar arrays to the bus) have a relatively high demise altitude of approximately 76.5 km. As the solar arrays are made of carbon fiber, they have a surface area to mass ratio greater than that of any total Block 3 Tranche 1 vehicle and will be further ablated at a faster rate once separated from the bus. Therefore, Umbra believes that the DAS estimate of kinetic energy per solar array to be an overestimate due to modelling limitations.

8.3. Calculation of Probability of Human Casualty

Compliant. DAS calculates the risk of human casualty to be 1:13,400, below the maximum risk of 1:10,000.

[9] NASA Debris Assessment Software User's Guide, Orbital Debris Program Office, NASA/TP-20210025946.

8.4. Compliance Assessment for Requirement 4.7-1: Limit the Risk of Human Casualty

8.4.1. Requirement 4.7-1 (A):

The potential for human casualty is assumed for any object with an impacting kinetic energy in excess of 15 joules. For uncontrolled reentry, the risk of human casualty from surviving debris shall not exceed 0.0001 (1:10,000) (Requirement 56626).

Compliance Statement (4.7-1 (A)):

Compliant. The calculated risk of human casualty is 1:13,400.

8.4.2. Requirement 4.7-1 (B):

For controlled reentry, the selected trajectory shall ensure that no surviving debris impact with a kinetic energy greater than 15 joules is closer than 370 km from foreign landmasses, or is within 50 km from the continental U.S., territories of the U.S., and the permanent ice pack of Antarctica (Requirement 56627).

Compliance Statement (4.7-1 (B)):

This requirement is not applicable since controlled reentry is not an element of the end of mission disposal plan.

8.4.3. Requirement 4.7-1 (C):

For controlled reentries, the product of the probability of failure of the reentry burn (from Requirement 4.6-4.b) and the risk of human casualty assuming uncontrolled reentry shall not exceed 0.0001 (1:10,000) (Requirement 56628).

Compliance Statement (4.7-1 (C)):

This requirement is not applicable since controlled reentry is not an element of the end of mission disposal plan.

8.5. Hazardous Materials Summary

Block Tranche 1 satellites do not contain any hazardous materials.

9. ASSESSMENT FOR TETHER MISSIONS

Not applicable. There are no tethers in Block 3 Tranche 1 satellites.

A.2 Failure Modes and Effects Analysis

Requirement 4.4-1: Limiting the risk to other space systems from accidental explosions during deployment and mission operations while in orbit about Earth or the Moon:

For each spacecraft and launch vehicle orbital stage employed for a mission, the program or project shall demonstrate, via failure mode and effects analyses or equivalent analyses, that the integrated probability of explosion for all credible failure modes of each spacecraft and launch vehicle is less than 0.001 (excluding small particle impacts) (Requirement 56449).

Compliance statement (4.4-1):

Required Probability: 0.001.

Expected probability: 0.000.

Supporting Rationale and Details:

Propulsion tank explosion:

Effect: All failure modes below might theoretically result in propulsion tank rupture with the possibility of orbital debris generation. However, in the unlikely event that a propellant tank does rupture due to internal pressure, the small size, mass, and potential energy of the tank is such that the spacecraft could be expected to vent gases without breakup, and debris from the cell tank should be contained within the closed aluminum bus structure due to lack of penetration energy.

Probability: Extremely low. It is believed to be a much less than 0.1% probability. Tank rupture resulting in the generation of orbital debris is not believed to be credible.

Failure Mode 1: Tank heaters fail closed and the temperature of xenon in propellant tank rises above the boiling point of xenon, generating gas and ultimately exceeding the burst pressure of the tank.

Combined faults required for realized failure: Spacecraft thermal design must be incorrect **AND** temperature control circuits must fail to the power on state.

Mitigation 1: Redundant temperature sensors on tank to indicate excessive temperature. Switch off loads to propulsion system heaters if propulsion system avionics fail to limit the maximum temperature.

Mitigation 2: Size tank heaters to preclude maximum tank temperature that is above the boiling point of xenon.

Failure Mode 2: Vibration / shock to propulsion tank results in rupture.

Combined faults required for realized failure: Spacecraft must survive harsh launch vibration / shock environment **AND** have a defect leading to failure in the comparatively benign on-orbit environment **AND** spontaneously fail in a way that causes an explosive decompression **AND** release enough energy during explosion to rupture the solid aluminum spacecraft walls.

Mitigation 1: Tank selection for large vibration and pressurization safety factor compared to mission. The tank has been designed and tested to 6527 psi. There is a 350% margin on the tank's working pressure, ensuring that there is no risk of accidental explosion of the propulsion system.

Mitigation 2: Vibration, shock, and thermal cycling tests before launch.

Battery explosion:

Effect: All failure modes below might theoretically result in a battery cell rupture. However, in the unlikely event that a battery cell does rupture due to internal pressure, the small size, mass, and potential energy of the selected commercially available off the shelf ("COTS") battery cells are such that the spacecraft can be expected to vent gases without breakup. Due to the lack of penetration energy, debris from the cell rupture will be contained within the aluminum battery housing, which itself is contained within an aluminum bus structure.

Probability: Extremely low. It is believed to be a much less than 0.1% probability. Battery cell rupture resulting in the generation of orbital debris is not believed to be credible.

Failure Mode 3: Internal short circuit.

Mitigation: Qualification and acceptance shock, vibration, thermal cycling, and vacuum tests followed by maximum system rate-limited charge and discharge to prove that no internal short circuit sensitivity exists.

Combined faults required for realized failure: Environmental testing **AND** functional charge/discharge tests must both be ineffective in discovery of the failure mode.

Failure Mode 4: Excessive cell temperature due to high load discharge rate and high initial temperature.

Mitigation: Test cells for high load discharge rates in a variety of flight-like configurations, with a maximum initial temperature, to determine the likelihood and impact of an out-of-control thermal rise in the cell.

Mitigation: Discharge current limiting to include fusing and simulations show discharge to never exceed 25% of cell capability. Screening of cells to assure minimal capacity and internal resistance mismatch between cells.

Combined faults required for realized failure: Spacecraft thermal design must be incorrect **AND** a fault resulting in excessive discharge current must occur simultaneously **AND** discharge current limiting must fail.

Failure Mode 5: Exceed maximum rated cell voltage.

Mitigation: Size solar array strings to limit maximum voltage across battery cell string. Charging circuit and Con-Ops makes it extremely unlikely that solar cells continue to charge the battery beyond 100% state of charge ("SOC").

Mitigation: Battery charge controller monitors string voltage and temperature and engages shunts as required, or it can be commanded to a non-sun pointing attitude until nominal operations resume.

Combined faults required for realized failure: Spacecraft Electrical Power System ("EPS") sizing must be inadequate to limit maximum battery cell voltage **AND** battery charge controller must fail allowing battery SOC to exceed nominal maximum **AND** the Command & Data-handling ("C&DH") subsystem must allow the battery SOC to exceed the nominal maximum without mitigation **AND** alternative solar array configuration would be required to sustain charging in over-voltage condition.

Failure Mode 6: Excessive charge rate.

Mitigation: Power system architecture prevents charge rate from exceeding battery specifications.

Combined faults required for realized failure: No credible scenario could produce a battery over-charge rate condition.

Failure Mode 7: Excessive discharge rate.

Mitigation: Short circuit protection on each external circuit.

Mitigation: Battery design to inhibit internal short circuit.

Mitigation: Vibration, shock, and thermal cycling tests to identify short circuits.

Combined faults required for realized failure: An external load must fail in a short circuit state **AND** all short circuit protections must fail to enable this failure mode.

Failure Mode 8: Inoperable vents.

Mitigation: Inspect machined parts to verify vent features are incorporated. Confirm during battery cell and module screening.

Combined effects required for realized failure: The final assembler fails to adhere to build procedure and limits proper venting **AND** one or more battery cells must rupture or vent into the battery housing. No credible scenario could block module vents sufficiently to cause an issue.

Failure Mode 9: Crushing.

Mitigation: This mode is negated by spacecraft design. There are no moving parts in the vicinity of the battery.

Combined faults required for realized failure: A catastrophic failure must occur in an external system **AND** the failure must cause a collision sufficient to crush the batteries leading to an internal short circuit **AND** the satellite must be in a naturally sustained orbit at the time the crushing occurs.

APPENDIX A

A.1 Acronyms

C&DH	Command & Data-handling
CDM	Conjunction Data Message
COTS	Commercially Off the Shelf
DAS	Debris Assessment Software
DOA	Dead on Arrival
EOM	End-of-Mission
EPS	Electrical Power System
ESPA	EELV Secondary Payload Adapter
FMEA	Failure Mode and Effects Analysis
GEO	Geostationary Orbit
LEO	Low Earth Orbit
MEO	Medium Earth Orbit
ODAR	Orbital Debris Assessment Report
OV	Orbital Vehicle
PMD	Post-Mission Disposal
SAR	Synthetic Aperture Radar
SOC	State of Charge
SSA	Space Situational Awareness