ORBITAL DEBRIS ASSESSMENT REPORT
FOR UMBRA BLOCK TWO CONSTELLATION

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Configuration Control

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Revisions

Revision	Date	Changes
01	3/20/2023	Initial Revision

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1. SUMMARY OF REPORT FINDINGS

As part of the Application, Umbra provides an orbital debris assessment report ("ODAR") for all satellites covered under this Application (the "Umbra Block Two Constellation").[1] The analysis uses the Debris Assessment Software, DAS v3.2.3, provided by the NASA Orbital Debris Program Office ("ODPO").

An orbital debris assessment of the Umbra Block Two Constellation 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]

The Umbra Block Two Constellation will operate at a nominal altitude of 535 km and a nominal inclination of 97.4°. The satellite deployment from the launch vehicle, as specified by the provider, will be 500 ± 20 km in altitude, eccentricity less than 0.004, and SSO ± 0.1 degrees inclination. In the worst-case scenario that an Umbra satellite is deployed dead-on-arrival ("DOA") at 520 km while fully stowed, it will re-enter the atmosphere in at most 10.5 years. The orbital deployment range will likely be refined as the launch date approaches.

Spacecraft disposal is accomplished through atmospheric reentry. In the nominal case, each spacecraft is expected to reenter no more than 0.4 years after mission completion with a planned post-mission disposal ("PMD") maneuver described herein. Umbra will budget sufficient reserves to ensure the capability to conduct the PMD maneuver and to take 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 Appendix A.2 of NASA-STD-8719.14C.

[1] In the associated Application, Umbra seeks authority to launch and operate two satellites (the "Umbra Block Two Constellation"). This ODAR applies to each of those two satellites.
[2] Process for Limiting Orbital Debris, NASA Technical Standard, NASA-STD-8719.14C (approved Nov. 5, 2021) ("*Process for Limiting Orbital Debris*").

Table 1. Orbital Debris Assessment Report Evaluation: Umbra Block Two System

Reqmt #	Launch Vehicle				Spacecraft			Comments
	Compliant	Not Compliant	Incomplete	Standard Non Compliant	Compliant or N/A	Not Compliant	Incomplete	
4.3-1.a			X		X			No debris released in LEO
4.3-1.b			X		X			No debris released in LEO
4.3-2			X		X			No debris released in GEO
4.4-1			X		X			Limit risk of explosion
4.4-2			X		X			Design for passivation
4.4-3			X		X			No planned breakups
4.4-4			X		X			No planned breakups
4.5-1			X		X			Limit debris by collision
4.5-2			X		X			Complies with Streamlined requirements.
4.6-1(a)			X		X			Atmospheric reentry option
4.6-1(b)			X		X			NA - storage orbit option
4.6-1(c)			X		X			NA - direct retrieval option
4.6-2			X		X			Not Applicable (GEO)
4.6-3			X		X			Not applicable (MEO)
4.6-4			X		X			Not required to meet 25 yr.
4.7-1			X		X			Reliability of disposal option
4.8-1					X			No tethers used

1. This ODAR is for the Umbra Block Two Constellation only. No launch vehicle was assessed.
2. This Assessment was performed using DAS v3.2.3.

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
Vice President of Space Systems
Umbra Lab, Inc.

2.2. Foreign Government or Space Agency Participation

None

2.3. Mission Design and Development Milestones

Launch:	See	
Table **2**	Launch and orbit insertion	
Phase 1:	<2 months	Checkout and orbit transfer
Operations:	≤58 months[3]	Radar remote sensing
End of Mission:	3 months	End of mission maneuvering

2.4. Mission Overview

The Umbra Block Two Constellation 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.

2.5. Launch Vehicle Description

Table 2 lists current best estimates for launch parameters associated with the Umbra Block Two

[3] Radar remote sensing timeline depends on spacecraft launch sequence to ensure mission duration complies with the 6-year operational lifetime requirement set forth in the Streamlined Part 25 rules.

Constellation that may apply to the Block Two ODAR analysis.

Table 2. Launch Parameters for Umbra Block Two Constellation

Launch Vehicle	Launch Site	Launch Date
Falcon 9	Cape Canaveral or Vandenberg AFB	October 2023

2.6. Launch and Deployment Profile

The launch insertion window has been indicated to possibly be as low as 480 km and as high as 520 km, which is reflected in Table 3 below.

The nominal operational orbit for the space vehicle is circular sun-synchronous with an altitude of 535 km. The space vehicle will 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 3. Orbital Envelope

	Orbital Altitude	Inclination
Nominal Insertion Orbit Range	480 – 520 km	SSO ± 0.1 deg
Operational Orbit Range	505 – 565 km (nominal 535 km)[4]	97.4 ± 1 deg
Nominal Post Mission Disposal Orbit	505 x 380 km	97.4 ± 1 deg

2.7. Orbit Selection Rationale

The nominal operational orbit is the result of an optimization between the remote sensing payload resolution, the desire to achieve a 3–5 year mission duration, and the availability of launch services.

2.8. Interaction with Other Operational Spacecraft

Interaction and potential physical interference with other operational spacecraft are not planned nor anticipated as part of the Umbra Block Two Constellation mission. Umbra is aware that other operators

[4] The Block Two 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.

operate in the 505–565 km orbital range and intends to coordinate physical operations of its satellites with all such operators, as necessary. Umbra satellite operators have successfully coordinated maneuvers of currently deployed satellites to avoid conjunction events.

3. SPACECRAFT DESCRIPTION

3.1. Physical Description of the Spacecraft

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 0.3 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 0.8 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 the Umbra Block Two space vehicle. The details of the payload are not shown, but approximate relative dimensions are captured.

Figure 1. Umbra Space Vehicle Deployed Side View

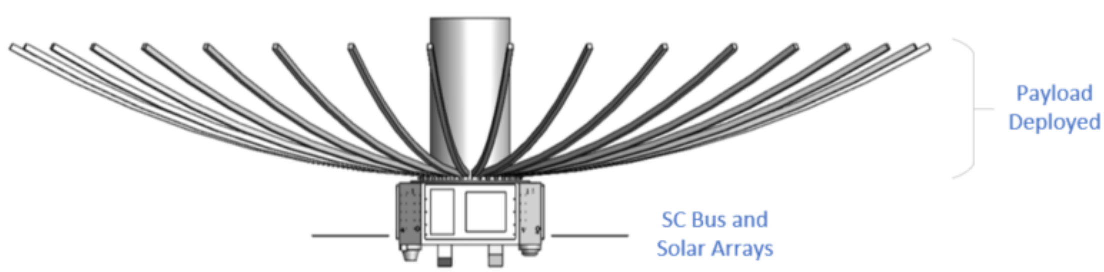


Figure 2. Umbra Space Vehicle Deployed Bottom View

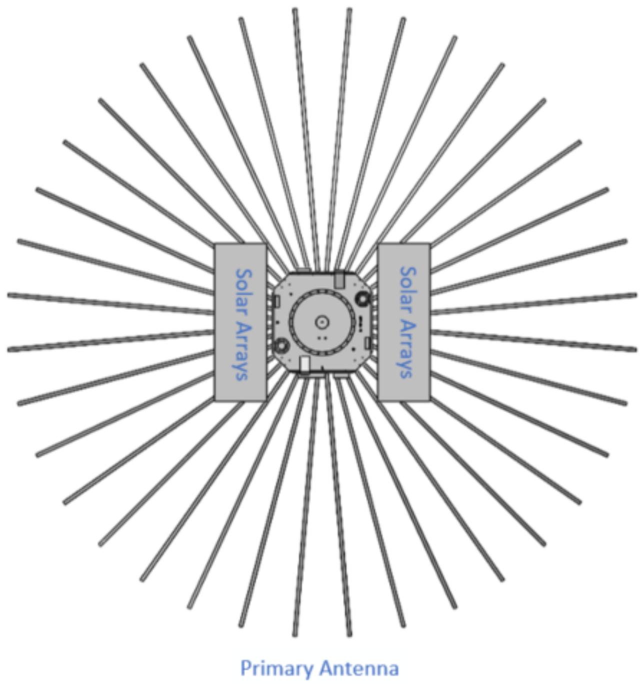


Figure 3. Umbra Space Vehicle Stowed Side View

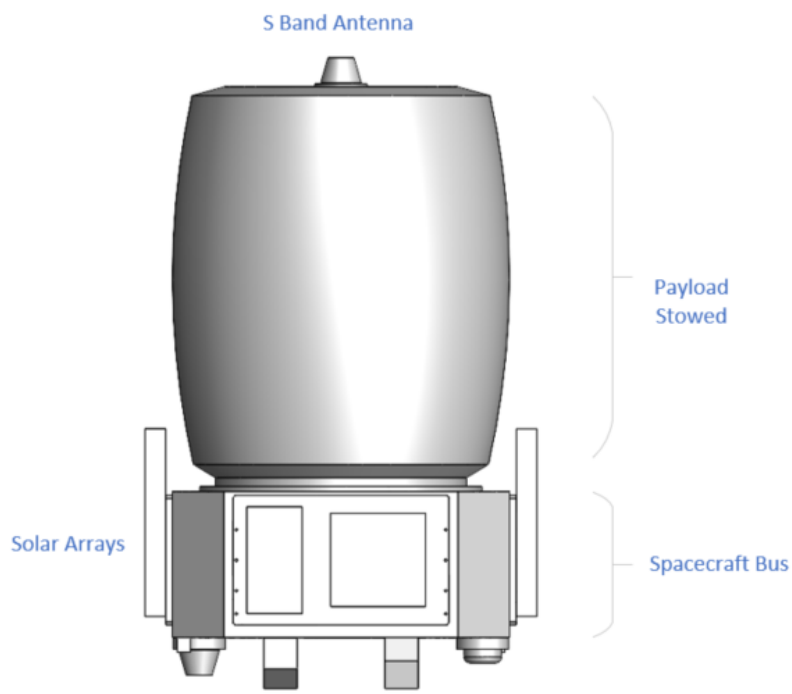


Figure 4. Umbra Space Vehicle Stowed Bottom View

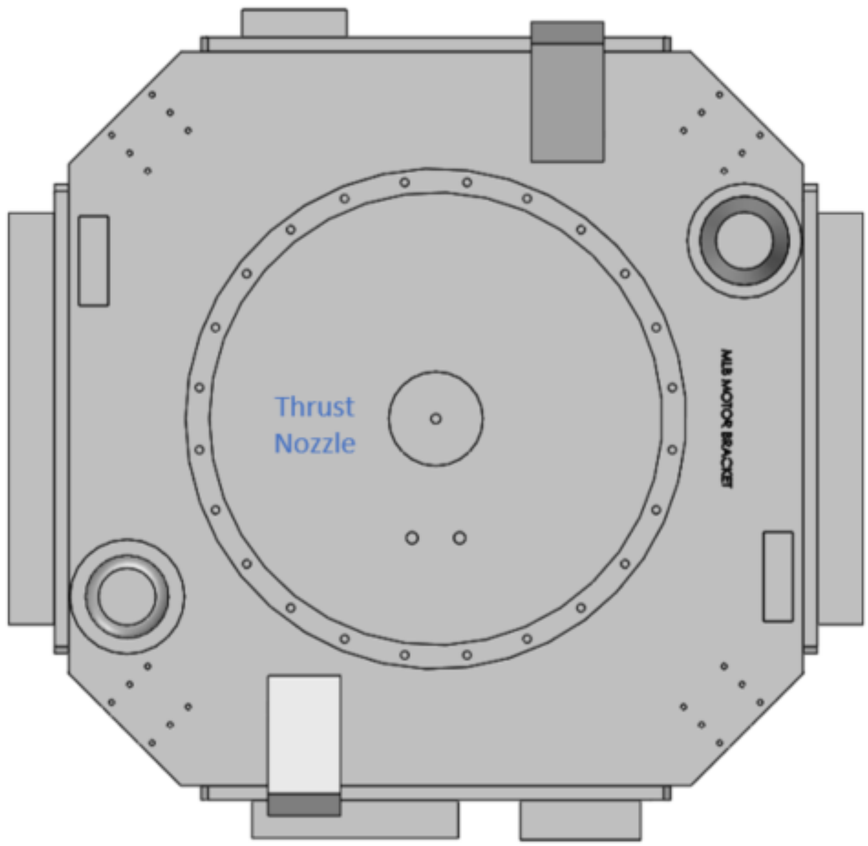


Table 4. Area-to-Mass Ratios Used in Analysis

Stowed Area-to-Mass Ratio (wet)	6.0×10^{-3}	m^2/kg
Deployed Area-to-Mass Ratio (dry)	15.3×10^{-3}	m^2/kg
PMD Area-to-Mass Ratio (dry)	33.7×10^{-3}	m^2/kg

3.3. Space Vehicle Mass

Wet Mass: 83.5 kg

Dry Mass: 78.8 kg

3.4. Propulsion System

Satellite propulsion is provided by a thermo-electric propulsion system that uses water as its propellant. The system consists of a single thruster, 2 propellant tanks, fill & drain ports, and an electronics enclosure. The water-based propulsion system will be used for station-keeping, PMD maneuvers, and collision avoidance as necessary. Below are some technical details of the propulsion system capabilities. A more detailed Collision Avoidance Process is provided in Section 6.

Table 5. On-Board Propulsion Metrics

ΔV	130	m/s
Nominal Acceleration	$2.6\text{x}10^{-4}$	m/s^2
ISP	180	sec

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

3.5. Fluids, Fluid Management, Fluid Systems

All fluids are contained within the propulsion system. The system includes a thruster, fill-drain valves for the pressurant and propellant, propellant tanks with elastomeric bladders, and avionics. The qualified propulsion system module will be subject to random vibration, shock, and thermal cycling tests.

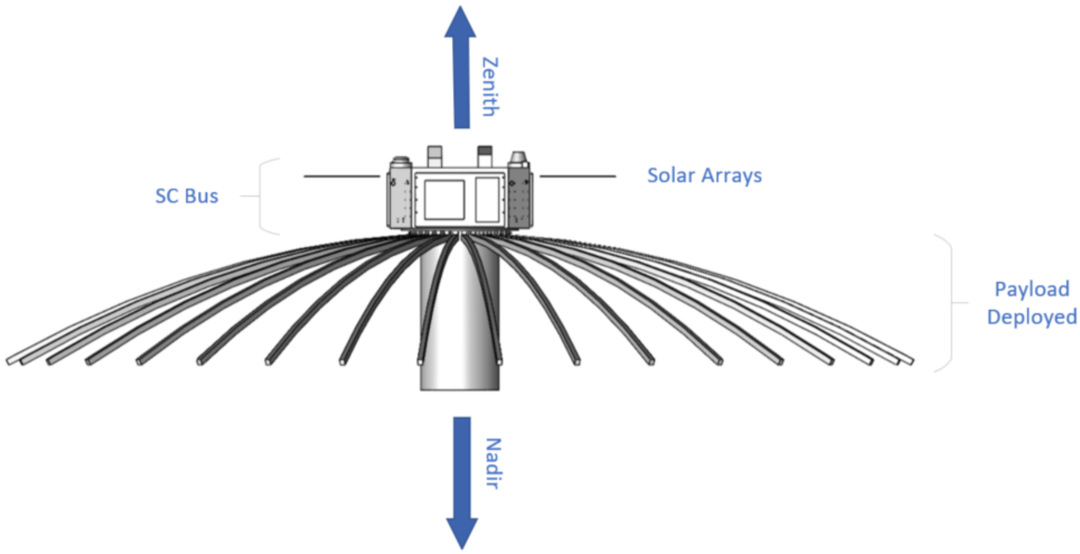
Table 6. Spacecraft Fluids

Description	Fluid	Mass (kg)	Max Pressure (psi)
Propellant	H_20	< 5	190
Pressurant	HFC-236	<< 1	190

3.6. Attitude Control Systems

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 Umbra Block Two Attitude



3.7. Range Safety and Pyrotechnic Devices

None.

3.8. Electrical Generation and Storage System

Power storage is provided by a battery consisting of lithium-ion cells arranged in an 8S2P configuration in four (4) battery modules. The 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. In-mission failure of the propulsion system leading to explosion or breakup of the space vehicle is not a credible scenario. An electrothermal propulsion system employing a liquid water propellant was selected in part to eliminate this hazard.

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 (water) is not energetic and is not toxic, thus its release does not pose any credible hazard. Likewise, the pressurant, Hexafluoropropane (FE-36), 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 water, 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 Block Two 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^{-5}, however this is software configurable and may be subject to change as necessary. If the risk stated by the 19th Space Defense Squadron ("19SDS") is higher than this threshold, Umbra will contact the other entity sharing the potential collision risk using information provided by 19SDS 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.

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[5] *See infra* Appendix A.2.

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 v3.2.3. See Appendix A.1.

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:

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 7 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. The probability of collision for each satellite is equal to zero when accounting for the utilization of our propulsion system for collision avoidance.

Table 7. Probability of Collision with Large Objects

Number of Satellites	Year of Launch	Probability Per Satellite
2	2023.8	3.5445e-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 8, the probability of collision with small objects resulting in PMD failure for each satellite is less than $6x10^{-5}$, well below the 0.01 requirement.

The components of the system that pose the greatest risk are listed in Table 9.

Table 8. Probability of PMD failure due to Collision with Small Objects

Number of Satellites	Year of Launch	Probability Per Satellite
2	2023.8	7.03E-05

Table 9. Small Object Damage Analysis

Critical Surface
Propulsion Electronics
Radio
GNC
Battery

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. The combination of the chosen disposal orbit and a high area-to-mass ratio results in rapid orbital decay after station-keeping has ceased.

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

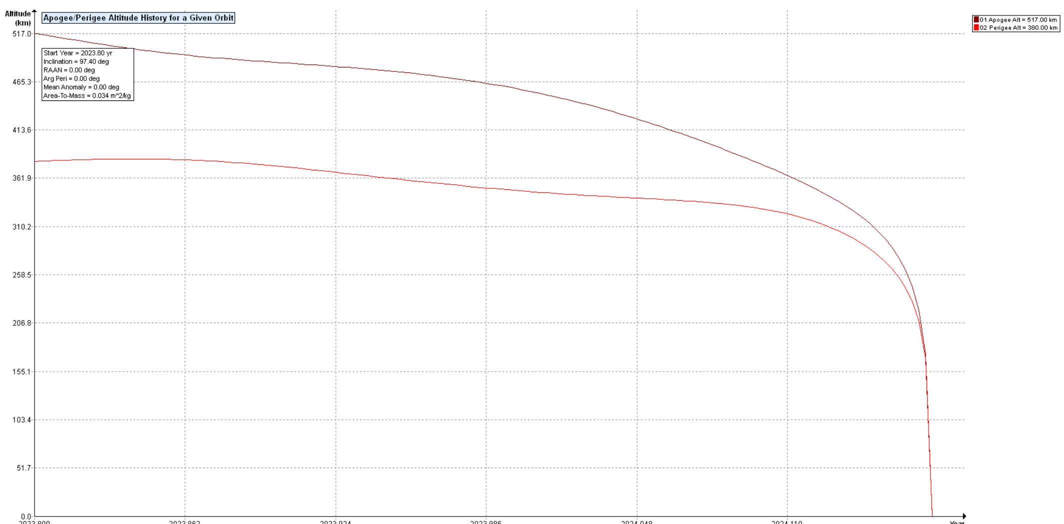
In a worst-case scenario for any individual satellite that is deployed DOA to a 520 x 520 km orbit and remains in its stowed configuration due to a hardware malfunction, it will reenter in less than 10.5 years.

7.3. Post-Mission Disposal Maneuver Plan

Nominally, to avoid interaction with LEO objects, as well as to accelerate reentry, a PMD maneuver to lower the satellite to a disposal orbit of approximately 505 x 380 km will be performed in conjunction with an EOM maneuver orienting the Z-Axis with the velocity vector. This orientation is the most stable equilibrium orientation that the spacecraft would naturally assume, thereby accelerating the deorbit of a non-functional satellite without any external input. Umbra plans to begin the PMD operations once each satellite passes five years of on-orbit operations, or earlier as needed, to ensure that the total in-orbit lifetime of each satellite is six years or less as required.[6]

As shown in Figure 6, the DAS prediction for orbit lifetime following the described PMD/EOM maneuver is less than 0.4 years. In the event of a hardware failure or other anomaly during operations at the nominal 535-km circular altitude, an Umbra satellite would naturally deorbit within 2 years. In the event of a hardware failure or other anomaly during operations at the highest requested nominal operating orbit of 565 km, an Umbra satellite would naturally deorbit within 2.5 years.

Figure 6. Orbital Decay Profile with PMD/EOM Maneuver (DAS v3.2.3)



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):

The Umbra Block Two Constellation satellite reentry is COMPLIANT using 4.6.2.1 described within NASA-STD 8719.14c. Each Block Two satellite, after executing a

[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").

PMD/EOM maneuver, will re-enter the Earth's atmosphere 0.4 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 Block Two satellites 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 Block Two satellites 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.

Compliance Statement (4.6-4):

Compliant. An EOM maneuver is not required to ensure deorbit within 25 years per Requirement 4.6-1. However, each Umbra Block Two satellite, after executing a PMD/EOM maneuver, will re-enter the Earth's atmosphere 0.4 years after the completion of mission and will be commanded to re-enter within the required timeline under streamlined Part 25 license rules. An EOM maneuver is not required to limit the probability of human casualty to 1:10,000 per Requirement 4.7-1.

8. ASSESSMENT OF SPACECRAFT REENTRY HAZARDS

8.1. Detailed Description of Spacecraft Components

Table 9. Spacecraft Model

Component	Subcomponent	Material	Qty.	Mass (kg)
Bus Structure		Aluminum 7075-T6	1	9
	Battery	Aluminum (generic)	64	0.05
	Torq Rods	Aluminum (generic)	3	0.3
	Reaction Wheels	A356	4	0.84
	Prop Tanks	Aluminum (generic)	2	1
	Largest Fastener	Stainless Steel (generic)	62	0.01
	Avionics Harness	Copper Alloy	1	0.19
	Largest Connector	Steel AISI 304	2	0.01
	Interior EE Chassis	Aluminum 7075-T6	2	0.75
MLB		Aluminum (generic)	1	0.7
Solar Array		Graphite Epoxy 1	6	0.3
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 chassis	Aluminum (generic)	1	1.71
	Strut Rod	Aluminum (generic)	6	0.094
	Electronics Chassis	Aluminum 7075-T6	3	0.89
	Canister Base Ring	Polycarbonate (aka Lexan)	1	0.13
	Canister Upper Restraint	Polycarbonate (aka Lexan)	1	0.18
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

This analysis was done using DAS v3.2.3 to ensure compliance with 4.7-1. Table 10 below provides the details showing modeling constraints replicated for all spacecraft. The table also shows the DAS-calculated demise altitude for components of an Umbra satellite. The DAS-calculated debris casualty risk was shown to be zero.

Table 10. Spacecraft Component List for Human Casualty Risk Analysis

Row Num	Name	Parent	Qty	Material	Body Type	Thermal Mass	Diameter /Width	Length	Height	Status	Risk	Demise Alt	Total DCA	KE
1	UmbraSAR_B2	0	1	Aluminum (generic)	Box	76.4	0.508	0.508	0.19	**Compliant**	'0'		0	
2	Bus Nadir	1	1	Aluminum 7075-T6	Flat Plate	6.97	0.61	0.61				74.2	0	0
3	Largest Fastener	2	62	Stainless Steel (generic	Cylinder	0.01	0.006	0.05				72.5	0	0
4	51 pin Connector	2	2	Steel AISI 304	Box	0.01	0.01	0.036	0.008			73	0	0
5	Interior electronics	2	2	Aluminum 7075-T6	Box	0.75	0.21	15.2	0.037			74.1	0	0
6	Bus Zenith	1	1	Aluminum 7075-T6	Flat Plate	8.7	0.61	0.61				73.5	0	0
7	Torq Rods	6	3	Aluminum (generic)	Cylinder	0.3	0.03	0.2				70.3	0	0
8	Reaction Wheels	6	4	A356	Cylinder	0.84	0.113	0.038				64.9	0	0
9	Prop Tanks	6	2	Aluminum (generic)	Cylinder	1	0.14	0.272				70.7	0	0
10	Avionics Harness	6	1	Copper Alloy	Cylinder	0.19	0.04	2.1				73.4	0	0
11	Bus +X	1	1	Aluminum 7075-T6	Flat Plate	4.32	0.26	0.37				72.7	0	0
12	Battery+x	11	32	Aluminum (generic)	Cylinder	0.065	0.021	0.071				70.8	0	0
13	Solar Array+x	11	3	Graphite Epoxy 1	Flat Plate	0.26	0.32	0.33				0	2.5	5.74
14	Bus -X	1	1	Aluminum 7075-T6	Flat Plate	4.32	0.26	0.37				72.7	0	0
15	Batteries-x	14	32	Aluminum (generic)	Cylinder	0.065	0.021	0.071				70.8	0	0
16	Solar Arrays-x	14	3	Graphite Epoxy 1	Flat Plate	0.26	0.32	0.33				0	2.5	5.74
17	Bus +Y	1	1	Aluminum 7075-T6	Flat Plate	2.41	0.26	0.37				74.8	0	0
18	Bus -Y	1	1	Aluminum 7075-T6	Flat Plate	2.2	0.26	0.37				75.1	0	0
19	Bus +X/+Y	1	1	Aluminum 7075-T6	Flat Plate	1.91	0.17	0.26				73.6	0	0
20	Bus +X/-Y	1	1	Aluminum 7075-T6	Flat Plate	1.91	0.17	0.26				73.6	0	0
21	Bus -X/-Y	1	1	Aluminum 7075-T6	Flat Plate	1.63	0.17	0.26				74.2	0	0
22	Bus -X/+Y	1	1	Aluminum 7075-T6	Flat Plate	1.63	0.17	0.26				74.2	0	0
23	MLB	1	1	Aluminum (generic)	Cylinder	0.7	0.22	0.1				76.1	0	0
24	SAR rib	1	108	Graphite Epoxy 1	Flat Plate	0.07	0.13	0.624				77.9	0	0
25	Base Ring	1	1	Aluminum 7075-T6	Cylinder	0.14	0.025	0.351				77.3	0	0
26	Antenna Element	1	1	Aluminum 6061-T6	Flat Plate	0.28	0.334	0.334				77.9	0	0
27	Canister	1	1	Aluminum 6061-T6	Cylinder	0.95	0.655	1.049				77.9	0	0
28	Assembly	27	1	Aluminum (generic)	Box	1.71	0.14	0.162	0.05			72.8	0	0
29	Strut Rod	27	6	Aluminum (generic)	Cylinder	0.094	0.012	0.3				77	0	0
30	Chassis	27	3	Aluminum 7075-T6	Box	0.89	0.119	0.16	0.016			74.5	0	0
31	Canister Base Ring	27	1	Polycarbonate (aka Lexa	Cylinder	0.13	0.351	0.2				77.9	0	0
32	Canister Upper Restraint	27	1	Polycarbonate (aka Lexa	Cylinder	0.18	0.381	0.2				77.9	0	0
33	Ti Hinge	1	8	Titanium (6 Al-4 V)	Cylinder	0.003	0.008	0.025				77.1	0	0
34	Ti rib Hinge	1	108	Titanium (6 Al-4 V)	Flat Plate	0.0182	0.03	0.05				76.5	0	0
35	Root Hinge	1	2	Aluminum 7075-T6	Flat Plate	0.08	0.05	0.304				77.6	0	0

8.2. Summary of Objects Expected to Survive an Uncontrolled Reentry

Per DAS v3.2.3, two objects specified in Table 10 are expected to survive an uncontrolled reentry and reach the Earth's surface. In each case, the kinetic energy of the surviving object is calculated to be less than 15 joules.

8.3. Calculation of Probability of Human Casualty

DAS v3.2.3 calculated the risk of human casualty to be zero.

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 zero.

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

The Umbra Block Two Constellation satellites do not contain any hazardous materials.

9. ASSESSMENT FOR TETHER MISSIONS

Not applicable. There are no tethers in the Umbra Block Two Constellation.

APPENDIX A

A.1 Acronyms

19 SDS	19th Space Defense Squadron
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
ODPO	Orbital Debris Program Office (NASA)
ODAR	Orbital Debris Assessment Report
OV	Orbital Vehicle
PMD	Post-Mission Disposal
SAR	Synthetic Aperture Radar
SOC	State of Charge
SSA	Space Situational Awareness

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 water in propellant tank rises above the boiling point of water, generating steam 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 water.

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 2: 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 3: 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 4: 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 5: 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 6: 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 7: 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 8: 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.