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Orbital Debris Assessment Report

For UmbraSAR

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Revision	Date	Changes
01	5/6/2020	Initial Revision
02	5/12/2020	Revised to account for input from CSSMA
03	6/15/2020	Revised to account for final orbital data from launch provider.
04	6/25/2020	Typo on High Insertion Case perigee altitude p.9; clarifications to Table.

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1.0 Summary of Report Findings

Umbra Lab Inc. ("Umbra Lab") provides an orbital debris assessment of its experimental satellite. The analysis uses the Debris Assessment Software, DAS 3.0.1, provided by the NASA Orbital Debris Program Office (ODPO).

An orbital debris assessment of the Umbra SAR spacecraft indicates the mission to be in compliance with the applicable requirements for spacecraft end-of-life disposal and risk to human casualty as specified in the NASA approved Process for Limiting Orbital Debris, NASA STD 8719.14B (Superseding NASA-STD-8719.14A).

The Umbra SAR satellite will operate between 500-600 km in altitude. The launch of UMBRA-2001 is scheduled for Dec-Jan of 2020/2021 with a separation altitude of 525 ± 25km. A nominal altitude of 583 km is assumed herein. The spacecraft will operate at 97-98 degree inclination. Analysis of worst-case scenarios including spacecraft failure at the nominal altitude of 583-km in a deployed configuration and Dead-On-Arrival (DOA) at 550-km while fully stowed is also provided. However, every scenario results in re-entry in less than 25 years.

Spacecraft disposal is accomplished through atmospheric reentry. The spacecraft is expected to reenter in 0.48 years after mission completion with a planned Post Mission Disposal (PMD) maneuver. This is compliant with the requirement to reenter within 25 years after mission completion or 30 years following launch.

In the worst case a PMD maneuver is not performed the reentry timeline is 8.2 years which is compliant with the requirement to reenter within 25 years after mission completion or 30 years following launch.

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-8719.14.

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Table 1. Orbital Debris Assessment Report Evaluation: UMBRA SAR 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			No critical subsystems needed for EOM disposal
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 SAR satellite only. No launch vehicle was assessed.
2. This Assessment was performed using DAS v3.0.1

1.2 Assessment Report Format

ODAR Technical Sections Format Requirements:

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

2.0 Program Management and Mission Overview

1.1 Project Manager

Michael Francis
Senior Spacecraft Systems Engineer
Umbra Lab, Inc.

1.2 Foreign Government or Space Agency Participation

None

1.3 Mission Design and Development Milestones

Launch:	Q4 2020	Launch and orbit insertion (Dec – Jan window)
Phase 1:	1 month	Checkout and orbit transfer
Operations:	58 months	Radar remote sensing
End of Mission:	1 month	End of mission maneuvering

1.3 Mission Overview

The UMBRA SAR system is a space based commercial remote sensing system. It features an experimental synthetic aperture radar that can produce highly resolved synthetic aperture radar imagery (<0.25-m). The space segment will be inserted via a ride share on a third-party launch vehicle. The ground segment will include a mission operations center and one or more remote ground terminals. The objective of the first UMBRA SAR mission is to demonstrate payload capability and end-to-end mission operations.

1.4 Launch Vehicle Description

Launch Vehicle:	Falcon 9
Launch Site:	Vandenberg AFB or Cape Canaveral
Launch Date:	Q4 2020 (Current launch window Dec – Jan)
Mission Duration:	5 years

1.5 Launch and Deployment Profile

Our orbital altitude of separation ranges between 500 km and 550 km with a target separation altitude of 525 km.

The nominal orbit for the space vehicle is circular sun-synchronous with an altitude of 583 km. The space vehicle will maneuver to the desired orbit via a series of Hohmann transfers and minor inclination change maneuvers (if required).

Table 2. Operational Orbital Envelope

	Apogee	Perigee	Inclination
High Insertion Case	550 km	550 km	97-98 deg
Low Insertion Case	500 km	500 km	97-98 deg
Target Operational Altitude	583 km	583 km	97-98 deg
Post Mission Disposal Case	583 km	380 km	97-98 deg

1.6 Orbit Selection Rationale

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

The range of altitudes for orbit insertion reflects the planned rideshare service.

1.7 Interaction with Other Operational Spacecraft

No interaction or potential physical interference with other operational spacecraft is planned or anticipated as part of the UMBRA SAR mission.

2.0 Spacecraft Description

2.1 Physical Description of the Spacecraft

The UMBRA SAR satellite fits within the standard ESPA envelope when stowed for launch. The bus structure consists of an aluminum frame with machined aluminum panels and has dimensions of approximately 58 cm x 58 cm x 19 cm, not including the solar arrays which reside in a stowed condition on either side of the bus. The payload is approximately 74 cm x 53 cm diameter in the stowed position. When deployed into the operational configuration, the maximum physical dimensions of the space vehicle are approximately 3.8 m x 3.8 m x 2.0 m.

2.2 Spacecraft Illustration

The figure below shows both the stowed and operational configurations of the UMBRA SAR space vehicle. The details of the payload are not shown, but approximate relative dimensions are captured.

Figure 1. UMBRA SAR Space Vehicle External Views

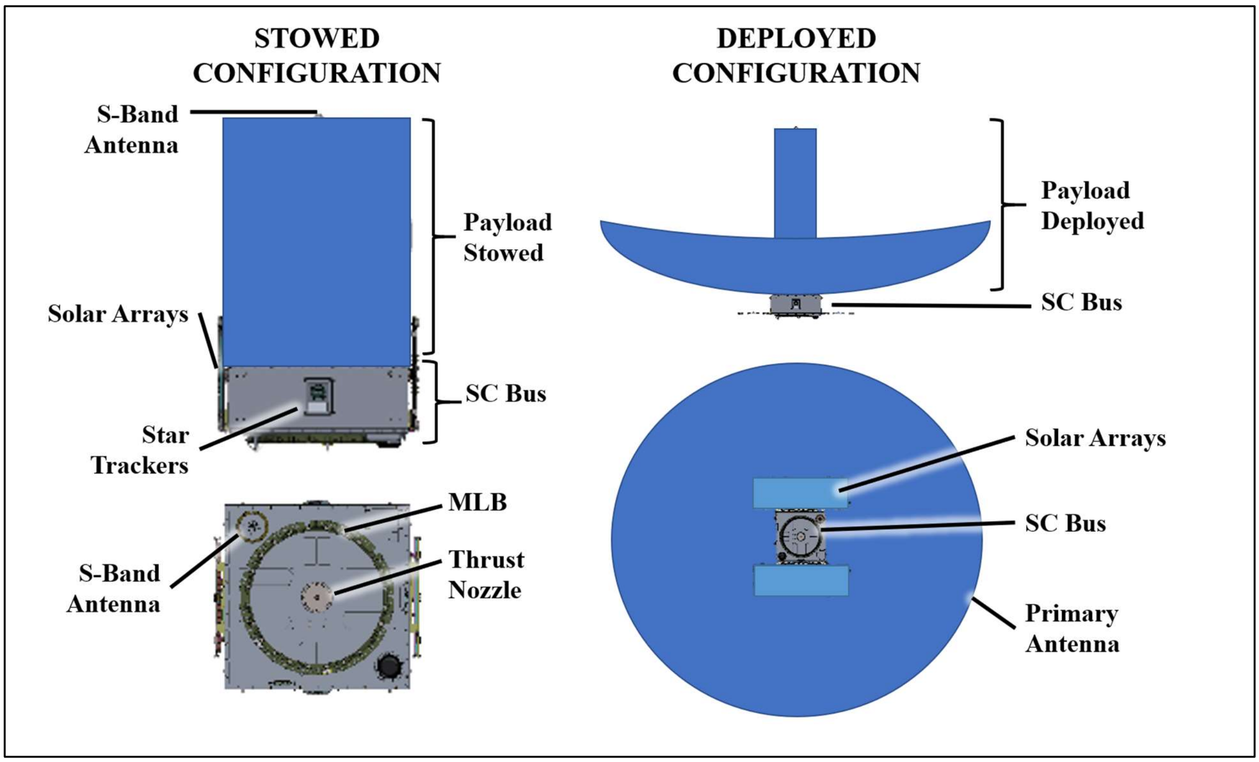


Table 3. Area-to-Mass Ratios Used in Analysis

Stowed Area-to-Mass Ratio (wet)	6.03×10^{-3}	m^2/kg
Deployed Area-to-Mass Ratio (dry)	17.15×10^{-3}	m^2/kg
PMD Area-to-Mass Ratio (dry)	45.4×10^{-3}	m^2/kg

2.3 Space Vehicle Mass

Wet Mass: 65 kg
Dry Mass: 60 kg

2.4 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 primarily for station keeping and can be used for collision avoidance if necessary. Below are some technical details of the capabilities of the prop system. Further, Umbra can submit a more detailed Collision Avoidance Process if required. When notified of a collision potential we will work with the relevant parties to formulate a collision avoidance plan.

Table 4. On-Board Propulsion Metrics

ΔV	130	m/s
Nominal Acceleration	2.6×10^{-4}	m/s^2
ISP	180	sec

2.5 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.

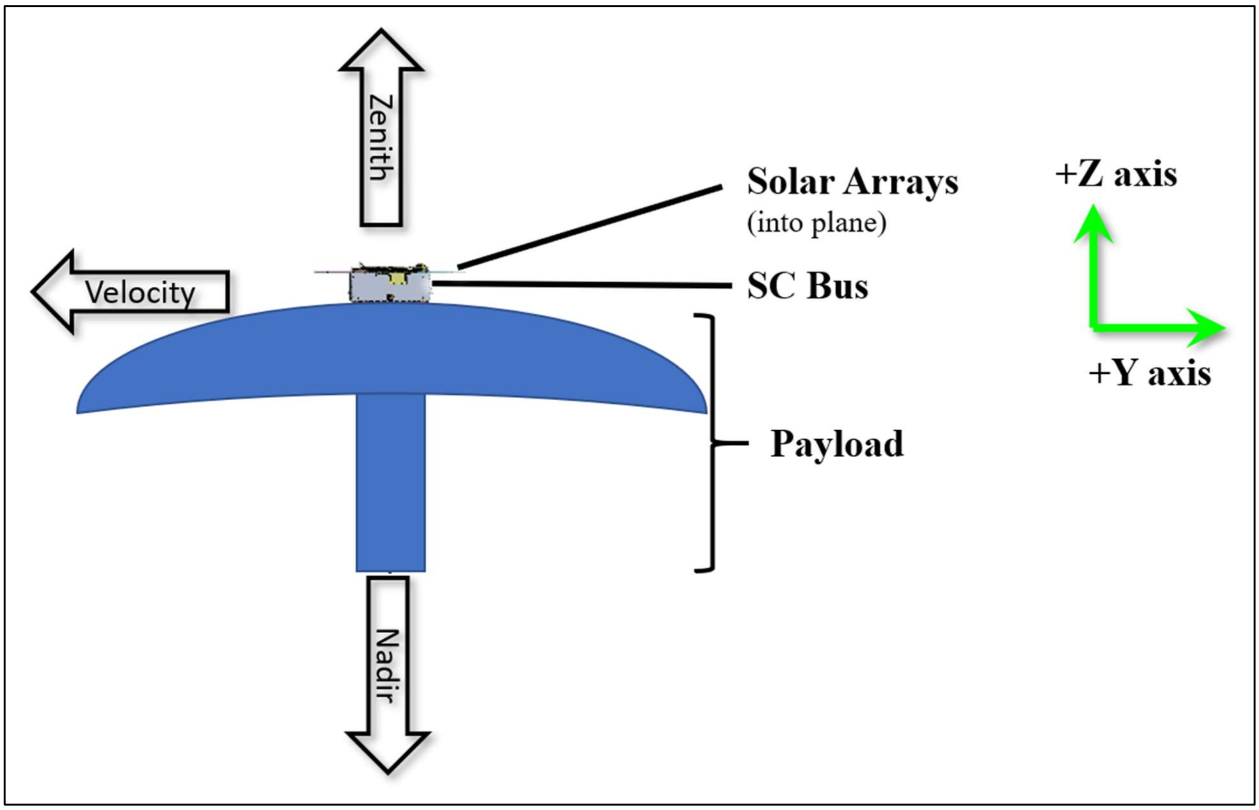
Table 5. Spacecraft Fluids

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

2.6 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 2. Satellite attitude will be varied among other pointing control modes to orient solar arrays towards the sun, to orient the payload for imaging, and to orient antennas for communication.

Figure 2. Nominal Umbra SAR Attitude



2.7 Range Safety and Pyrotechnic Devices

None.

2.8 Electrical Generation and Storage System

Power storage is provided by a battery consisting of Lithium-Ion cells arranged in an 8S4P 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.

2.9 Other Sources of Stored Energy

None.

2.10 Radioactive Materials

None.

3.0 Assessment of Spacecraft Debris Released during Normal Operations

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

3.1 Identification of any Objected Expected to be Released

There are no intentional releases of objects.

3.2 Rationale for Release of Each Object

Not Applicable.

3.3 Time of Release for Each Object Relative to Launch Time

Not Applicable.

3.4 Release Velocity of Each Object with Respect to Spacecraft

Not Applicable.

3.5 Expected Orbital Parameters of Each Object After Release

Not Applicable.

3.6 Calculated Orbital Lifetime of Each Object

Not Applicable.

3.7 Compliance Assessment for Requirements 4.3-1 and 4.3-2

Requirement 4.3-1: Mission Related Debris Passing Through LEO

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

Requirement 4.3-2: Mission Related Debris Passing Near GEO

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

4.0 Assessment of Spacecraft Intentional Breakups and Potential for Explosions

4.1 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.

4.2 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.

4.3 Plan for Any Designed Spacecraft Breakup

There are no planned breakups.

4.4 Components Which are Passivated at EOM

4.4.1 Propulsion System:

Residual propellant will be depleted via 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 is an inert gas and its release does not pose any credible hazard. Per the manufacturer it is inert, non-corrosive, electrically non-conductive and has zero ozone depleting potential. As the propellant used in this case is water there is no risk from persistent liquids as any release of propellant evaporates and dissipates. This propellant is unable to persist in droplet form in the space environment.

4.4.2 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.

4.4.3 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.

4.5 Compliance Assessment for Requirements 4.4-1 through 4.4-4

Umbra Lab has completed Failure Mode Verification Analysis (FMEA; see Appendix) 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 Umbra SAR satellite or do not otherwise generate orbital debris. As described above, energy sources are both safely contained during the mission and/or depleted at the time of post mission disposal.

5.0 Assessment of Spacecraft Potential for On-Orbit Collisions

5.1 Calculation of Spacecraft Probability of Collision

Assessment of spacecraft compliance with Requirements 4.5-1 and 4.5-2 per NASA-STD-8719.14b was performed using DAS v3.0.1. See Appendix A.1.

5.2 Compliance Assessment for Requirement 4.5-1 and 4.5-2

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. The computed probability for Large Object Impact and Debris Generation is: 2.3386E-05.

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):**
> This requirement is not applicable. Although a PMD maneuver is planned after 5 years of operation UMBRA SAR does not require post-mission disposal activities.

6.0 Assessment of Spacecraft Post-mission Disposal Plans and Procedures

6.1 Description of spacecraft disposal option selected

The satellite will de-orbit naturally by atmospheric re-entry. The combination of the chosen operational orbit and a high area-to-mass ratio result in rapid orbital decay after station keeping has ceased.

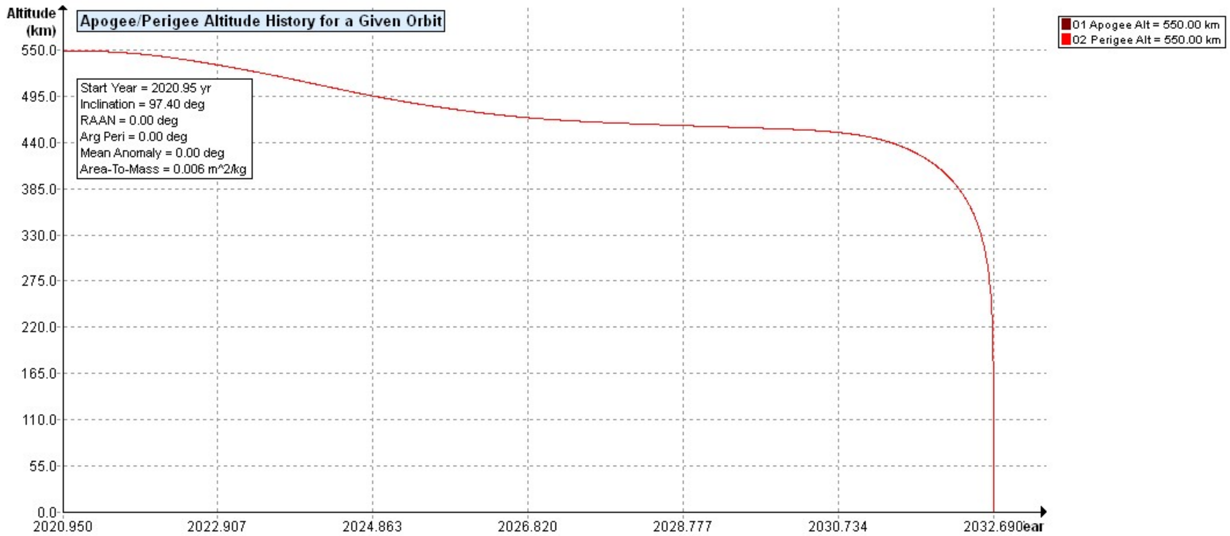
6.2 Systems or Components Required to Accomplish Post-mission Disposal Operations

None.

While a post-mission maneuver is accounted for in the propellant budget, one is not required to meet the 25-year reentry requirement for objects abandoned in LEO. Figure 3 shows that UMBRA SAR will naturally reenter after station keeping maneuvers have ceased, well within the required timeline.

In a worst-case scenario where the satellite is delivered to a 550 x 550 km orbit, and remains in its stowed configuration (DOA) it will reenter in 11.7 years as shown by Figure 3.

Figure 3. Worst Case Orbit Decay Profile for UMBRA SAR (DAS v3.0.1)



6.3 Post-mission Disposal Maneuver Plan

Nominally, to avoid interaction with LEO assets as well as to accelerate reentry, a Post-mission Disposal (PMD) maneuver to lower the orbit to 583 x 380 km will be performed in conjunction an End-of-mission (EOM) maneuver orienting the Z-Axis with the velocity vector. This orientation is also the most stable equilibrium orientation that the spacecraft would naturally assume thereby accelerating the deorbit of a non-functional satellite without any external input.

The DAS prediction for orbit lifetime following the described EOM maneuver is 0.48 years as shown in Figure 4. A post-mission disposal maneuver is not required to meet the 25 year reentry requirement as shown in Figure 4 as even it is assumed that the lower drag flight attitude is maintained the Spacecraft re-enters in 8.2 years after the end of operational mission.

Figure 4. Orbital Decay Profile with EOM Maneuver (DAS v3.0.1)

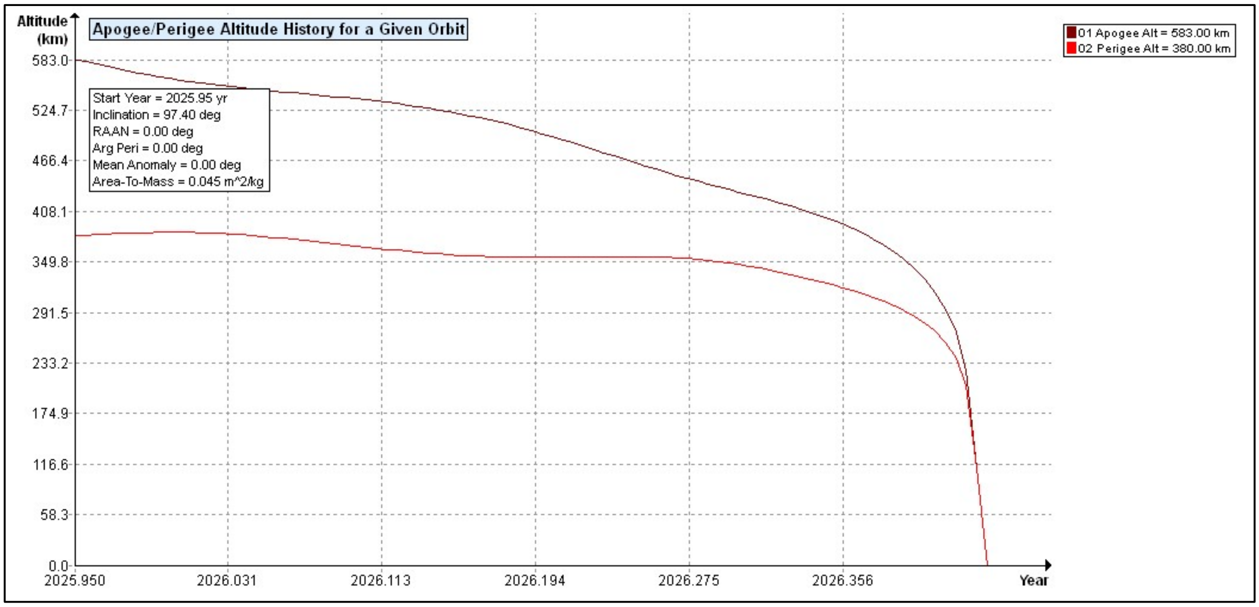
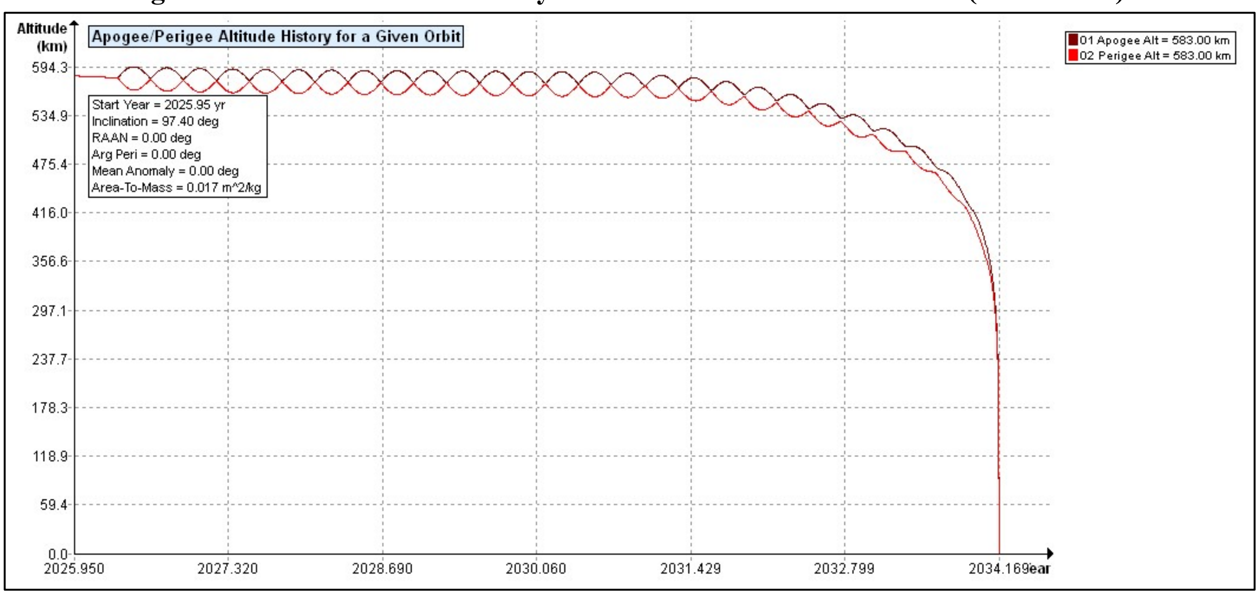


Figure 5. Worst Case Orbit Decay Profile without EOM Maneuver (DAS v3.0.1)



6.6 Preliminary Plan for Spacecraft Controlled Reentry

Not Applicable.

6.7 Compliance Assessment for Requirement 4.6-1 to 4.6-4

Requirement 4.6-1:

Disposal for space structures passing through LEO.

Compliance Statement (4.6-1):

The UMBRA SAR satellite reentry is COMPLIANT using 4.6.2.1.a(1) described within NASA-STD 8719.14b.

UMBRA SAR with no EOM maneuver will re-enter approximately 8 years after mission start assuming Orbit Maintenance ceases at 5-years of operation as planned assuming a low-drag nominal flight attitude is maintained. However, the spacecraft should naturally assume a high drag orientation reducing this time to re-entry to just 6.1 years post mission.

UMBRA SAR with an EOM maneuver will re-enter approximately 0.41 years after the completion of mission and 5.41 years after launch.

Requirement 4.6-2:

Disposal for space structures near GEO.

Compliance Statement (4.6-2):

Compliant. The requirement is not applicable. UMBRA SAR space structures are not located or disposed of near GEO.

Requirement 4.6-3:

Disposal for space structures between LEO and GEO.

Compliance Statement (4.6-3):

Compliant. The requirement is not applicable. UMBRA SAR space structures are not located or disposed of between LEO and GEO.

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.

An EOM maneuver is not required to limit the probability of human casualty to 1:10,000 per Requirement 4.7-1 (A).

7.0 Assessment of Spacecraft Reentry Hazards

7.1 Detailed Description of Spacecraft Components

Table 6: Spacecraft Model

Component	Subcomponent	Qty.	Mass (kg)
Bus Structure		1	14
	Torque Rods	3	0.4
	RWs	4	0.84
	Propulsion	1	6.9
	MLB	1	0.7
	Avionics	1	2
	Battery	4	0.58
Solar Arrays		2	1.5
SAR Antenna		1	25
Payload Electronics		1	6

Table 7. Spacecraft Component List for Human Casualty Risk Analysis

Name	Qty	Material	Body Type	Mass (Kg)	Demise Alt (km)	Total DCA (m^2)	Kinetic Energy (J)
Umbra SAR-2001	1	Aluminum 6061-T6	Box	60.7		0	
Zenith Panel	1	Aluminum 7075-T6	Flat Plate	2.7	75.2	0	0
Nadir Panel	1	Aluminum 7075-T6	Flat Plate	1.29	76.6	0	0
Side Panel	4	Aluminum 7075-T6	Flat Plate	0.47	77.1	0	0
Canister	1	Aluminum 6061-T6	Cylinder	0.95	77.8	0	0
Canister Base Ring	1	Polycarbonate (aka Lexan)	Cylinder	0.13	77.8	0	0
Canister Upper Restraint	1	Polycarbonate (aka Lexan)	Cylinder	0.18	77.7	0	0
Sub Reflector	1	Aluminum 6061-T6	Cylinder	0.28	77.4	0	0
Electronics Chassis	1	Aluminum 6061-T6	Box	1.03	73.3	0	0
Transceiver PCB	2	Fiberglass	Flat Plate	0.07	73.3	0	0
Root Rib	36	Graphite Epoxy 1	Flat Plate	0.07	0	21.71	2.57
Body Actuator	36	Aluminum 7075-T6	Cylinder	0.01	77.7	0	0
Actuator Guide Rod	36	Steel AISI 304	Cylinder	0.01	77.3	0	0
Root Hub Fitting	36	Aluminum 7075-T6	Box	0.02	77.4	0	0
Root Rib Fitting	36	Titanium (6 Al-4 V)	Box	0.02	0	14.23	4.1
Rib Fitting 2	72	Titanium (6 Al-4 V)	Box	0.008	0	28.02	1.14
Rib Fitting 3	72	Titanium (6 Al-4 V)	Box	0.007	0	27.8	0.94
Torque Rods	3	Aluminum 6061-T6	Cylinder	0.3	71.6	0	0
RWs	4	Aluminum 6061-T6	Box	0.84	66.9	0	0
Tanks	2	Aluminum 7075-T6	Cylinder	1	72.1	0	0
MLB	1	Aluminum 7075-T6	Cylinder	0.7	72.8	0	0
Avionics Harness	1	Copper Alloy	Cylinder	0.19	76.4	0	0
Carrier Card Assembly	1	Aluminum 7075-T6	Box	0.7	71.6	0	0
51 pin Connector	2	Steel AISI 304	Box	0.01	67.5	0	0
Largest Fastener	62	Stainless Steel (generic)	Cylinder	0.01	73.7	0	0
Battery Chassis	4	Aluminum 7075-T6	Box	0.17	76.9	0	0
Battery	32	Stainless Steel (generic)	Cylinder	0.05	73.9	0	0
Root Panel Assembly	6	Graphite Epoxy 1	Flat Plate	0.22	0	5.08	7.69
Root Panel Hinge	8	Titanium (6 Al-4 V)	Cylinder	0.003	0	3.02	0.47
Root Hinge	2	Aluminum 7075-T6	Flat Plate	0.08	77.3	0	0

7.2 Summary of Objects Expected to Survive an Uncontrolled Reentry

Per DAS v3.0.1 the objects in Table 8 are expected to survive an uncontrolled reentry. Items with <15 J of kinetic energy do not contribute to the total debris casualty area.

Table 8. Objects Expected To Survive Uncontrolled Reentry

Component	Debris Casualty Area (m^2)	Kinetic Energy (J)
Root Rib	21.71	2.57
Root Rib Fitting	14.23	4.1
Rib Fitting 2	28.02	1.14
Rib Fitting 3	27.8	0.94
Solar Panel Hinge	3.02	.47
Solar Panels	5.08	7.69

7.3 Calculation of Probability of Human Casualty

DAS v3.0.1 calculated the risk of human casualty to be 1:100,000,000 with a total debris casualty area of 0.0 m^2.

7.4 Compliance Assessment for Requirement 4.7-1

Limit the risk of human casualty.

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 1:100,000,000.

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)):

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

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)):

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

7.A Assessment of Spacecraft Hazardous Materials

7.A.1 Hazardous Materials Summary

The UMBRA SAR satellite does not contain any hazardous materials.

8.0 Assessment for Tether Missions

Not applicable. There are no tethers in the UMBRA SAR system.

Appendix

A.1 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 explosion 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 while the spacecraft could be expected to vent gases without breakup and most 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 COTS battery cells is such that the spacecraft can be expected to vent gases without breakup. Debris from the cell rupture will be contained within the aluminum battery housing, which itself is contained within an aluminum bus structure, due to the lack of penetration energy.

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 to continue to charge the battery beyond 100% SOC.
Mitigation: Battery charge controller monitors string voltage and temperature and engages shunts as required OR can be commanded to a non-sun pointing attitude until nominal operations resume.
Combined faults required for realized failure: Spacecraft EPS sizing must be inadequate to limit maximum battery cell voltage **AND** battery charge controller must fail allowing battery state of charge to exceed nominal maximum **AND** the C&DH subsystem must allow the battery state of charge 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: Not 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** short circuit protection failures must all occur 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.