Astro Digital Demo9 (“Shasta”) Orbital Debris Assessment Report (ODAR)

ASTRO-DIGITAL-DEMO9-ODAR-1.1

This report is presented as compliance with NASA-STD-8719.14, APPENDIX A.
Report Version: 1.1, 3/18/2021

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DAS Software Version Used In Analysis: v3.1.1

Astro Digital Demo9 Orbital Debris Assessment Report
ASTRO-DIGITAL-DEMO9-ODAR-1.1

APPROVAL:

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DocuSigned by:

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Revision Record				
Revision:	Date:	Affected Pages:	Changes:	Author(s):
1.0	3/7/2021	All –Initial	Initial	B. Cooper
1.1	3/18/2021	Updated DAS3.1.1		D. Thorne

Table of Contents

Self-assessment of the ODAR using the format in Appendix A.2 of NASA-STD-8719.14:

A self assessment is provided below in accordance with the assessment format provided in Appendix A.2 of NASA-STD-8719.14.

Requirement #	Launch Vehicle				Spacecraft			Comments
	Compliant	Not Compliant	Incomplete	Standard Non Compliant	Compliant	Not Compliant	Incomplete	
4.3-1.a	☐	☐	☒	☐	☒	☐	☐	No Debris Released in LEO. See note 1.
4.3-1.b	☐	☐	☒	☐	☒	☐	☐	No Debris Released in LEO. See note 1.
4.3-2	☐	☐	☒	☐	☒	☐	☐	No Debris Released in GEO. See note 1.
4.4-1	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.4-2	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.4-3	☐	☐	☒	☐	☒	☐	☐	No planned breakups. See note 1.
4.4-4	☐	☐	☒	☐	☒	☐	☐	No planned breakups. See note 1.
4.5-1	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.5-2					☒	☐	☐	No critical subsystems needed for EOM disposal
4.6-1(a)	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.6-1(b)	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.6-1(c)	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.6-2	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.6-3	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.6-4	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.6-5	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.7-1	☐	☐	☒	☐	☒	☐	☐	See note 1.
4.8-1					☒	☐	☐	No tethers used.

Assessment Report Format:

ODAR Technical Sections Format Requirements:

Astro Digital US, Inc. is a US company. This ODAR follows the format in NASA-STD-8719.14, Appendix A.1 and includes the content indicated as a minimum, in each of sections 2 through 8 below for the Demo9 satellite. Sections 9 through 14 apply to the launch vehicle ODAR and are not covered here.

Astro Digital Demo9 Space Mission Program:

ODAR Section 1: Program Management and Mission Overview

System Engineer: David Thorne
Senior Management: Chris Biddy

Foreign government or space agency participation: None.

Summary of NASA's responsibility under the governing agreement(s): N/A

Schedule of upcoming mission milestones:

- Shipment of spacecraft: Q2 2021
- Launch: June 2021

Mission Overview: Demo9 is a technology demonstration of a hyperspectral imaging payload.

The spacecraft bus is the Corvus-Micro design. The satellite bus uses reaction wheels, magnetic torque coils/rods, star tracker(s), magnetometers, sun sensors, and gyroscopes to enable precision 3-axis pointing without the use of propellant.

Launch Vehicles and Launch Sites: SpaceX Falcon 9, Kennedy Space Center, USA

Proposed Initial Launch Date: June 2021

Mission Duration: The design lifetime of the spacecraft hardware is a minimum of 5 years in LEO. Given the spacecraft's maximum expected orbital altitude of ~550 km, it will passively deorbit within 14.1 years after launch.

Launch and deployment profile, including all parking, transfer, and operational orbits with apogee, perigee, and inclination: The selected launch vehicle will deliver Demo9 directly to its operational circular polar orbit at an altitude of ~550 km (expected). The spacecraft will operate from an orbit with the following parameters:

Maximum Injection Orbital Altitude: ~550 km

Eccentricity: 0.0000

Inclination: 97.7°

After the spacecraft has demonstrated all relevant technologies and completed payload operations, the spacecraft will passively decay within 14.1 years after launch.

<u>ODAR Section 2</u>: Spacecraft Description:

Physical description of the spacecraft:

The Demo9 satellite is based on the standard Corvus-Micro bus and has a total mass of ~22.5 kg. The main spacecraft body has dimensions of ~34 cm x ~34 cm x ~49 cm. The satellite has multiple body-mounted solar panels (no deployable). The superstructure is comprised of 6 aluminum iso-grid outer panels. All the internal components are attached to the inner faces of these 6 structural panels. There is a Planetary Systems Lightband on the +X face of the spacecraft that is used to deploy the spacecraft from the launch vehicle. Two independent UHF TT&C antennas protrude from opposite sides of the spacecraft body. Two S-band TT&C antennas are placed on the +Z face and the -Z face and feed into the same TT&C radio to allow for full hemispherical coverage. The spacecraft includes two independent GPS receivers and associated antennas as well as a single star tracker.

Power is locked away from all spacecraft platform and payload components by means of redundant series separation switches. These switches cannot be activated until the spacecraft is deployed from the launch vehicle. The bus electronics are largely identical to many past Astro Digital missions. The Flight Computer, low-level Electrical Power System, TT&C transceiver, and GPS receiver have all been flying as a system on orbit since 2018. The spacecraft is depicted in Figure 1.

Figure 1: Demo9 Spacecraft

Total satellite mass at launch, including all propellants and fluids:

Demo9: 22.5 kg

Dry mass of satellites at launch:
Demo9: 22.5 kg

Description of all propulsion systems (cold gas, mono-propellant, bi-propellant, electric, nuclear):
The Demo9 satellite does not have a propulsion system.

Identification, including mass and pressure, of all fluids (liquids and gases) planned to be on board and a description of the fluid loading plan or strategies, excluding fluids in sealed heat pipes:
The Demo9 satellites does not have any pressure vessels and/or stored fluids.

Fluids in Pressurized Batteries: None

The Corvus-Micro satellite design uses eight unpressurized standard COTS Lithium-Ion battery cells in parallel for the low voltage system.

Description of attitude control system and indication of the normal attitude of the spacecraft with respect to the velocity vector:

The Demo9 spacecraft will activate its attitude determination system following deployment from the launch vehicle. Shortly after deployment, the reaction wheels will be used to detumble the spacecraft from any initial deployment rates and the spacecraft will enter a sun pointing safe mode with the star tracker pointed anti-nadir. All the following attitude modes use a combination of the following sensors and actuators to perform maneuvers. A magnetometer, sun sensors, gyroscope, reaction wheels, torque rods and star trackers are used to orientate the spacecraft correctly.

- A *sun pointing safe mode* that is optimized for solar power generation from the satellite. The spacecraft's body-mounted solar panels will be oriented towards the sun and the star tracker will be clocked anti-nadir.
- A *target tracking mode*, which will allow the spacecraft to point high speed communication link and track the ground receive station.
- A nadir tracking mode will be used for payload operation when the satellite enters the sun or exits eclipse. Nadir-pointing is the action of pointing directly below the satellite perpendicular to Earth.
- A sun clocking mode will be used as a complementary mode to any of the fourth mentioned modes in which the spacecraft is commanded to orientate itself around a fixed axis in such way that maximum power generation can be achieved.

Description of any range safety or other pyrotechnic devices: None.

Description of the electrical generation and storage system:

Standard COTS Lithium-Ion battery cells are charged before payload integration and provide electrical energy during eclipse and during high power consumption modes. All power required for the operation of the bus electronics is supplied through an "all-parallel" battery arrangement that results in increased safety thanks to natural voltage balancing between cells.

The all-parallel bus battery is charged through the solar panels.

The spacecraft is equipped with 3 main solar panels equipped with Spectrolab UTJ and XTJ Prime cells. An additional 2U panel is mounter in the direction of the payload with Spectrolab XTE-SF cells.

Typical bus operations consume 12 watts of power on average. The charge/discharge cycle is managed by a power management system overseen by the Flight Computer and Electrical Power Subsystem.

Identification of any other sources of stored energy not noted above: None

Identification of any radioactive materials on board: None

<u>ODAR Section 3</u>: Assessment of Spacecraft Debris Released during Normal Operations:

Identification of any object (>1 mm) expected to be released from the spacecraft any time after launch, including object dimensions, mass, and material: None.

Rationale/necessity for release of each object: N/A.

Time of release of each object, relative to launch time: N/A.

Release velocity of each object with respect to spacecraft: N/A.
Expected orbital parameters (apogee, perigee, and inclination) of each object after release: N/A.

Calculated orbital lifetime of each object, including time spent in Low Earth Orbit (LEO): N/A.

Assessment of spacecraft compliance with Requirements 4.3-1 and 4.3-2 (per DAS 3.1.1)
4.3-1, Mission Related Debris Passing Through LEO:

COMPLIANT

4.3-2, Mission Related Debris Passing Near GEO:

COMPLIANT

ODAR Section 4: Assessment of Spacecraft Intentional Breakups and Potential for Explosions.

Potential causes of spacecraft breakup during deployment and mission operations: There is only one potential scenario that could potentially lead to a breakup of the satellite.

1) Lithium-ion battery cell failure

Summary of failure modes and effects analyses of all credible failure modes which may lead to an accidental explosion:

The in-orbit failure of a battery cell protection circuit could lead to a short circuit resulting in overheating and a very remote possibility of battery cell explosion. The battery safety systems discussed in the FMEA (see requirement 4.4-1 below) describe the combined faults that must occur for any of seven (7) independent, mutually exclusive failure modes to lead to such an explosion.

Detailed plan for any designed spacecraft breakup, including explosions and intentional collisions: There are no planned breakups.

List of components which shall be passivated at End of Mission (EOM) including method of passivation and amount which cannot be passivated:

Eight (8) Lithium Ion Battery Cells. Solar array charging will be disabled, which will fully discharge all cells within two days.

Rationale for all items which are required to be passivated, but cannot be due to their design: None

Assessment of spacecraft compliance with Requirements 4.4-1 through 4.4-4:

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: See detailed information below

Required Probability: 0.001

Expected probability, Demo9: 0.0000

Supporting Rationale and FMEA details:

Battery explosion:

Effect: All failure modes below might result in battery explosion with the possibility of orbital debris generation. However, in the unlikely event that a battery cell does explosively rupture, the small size, mass, and potential energy, of these small batteries is such that while the spacecraft could be expected to vent gases, most debris from the battery rupture should be contained within the spacecraft due to the lack of penetration energy to the multiple enclosures surrounding the batteries.

Probability: Extremely Low. It is believed to be less than 0.01% given that multiple independent (not common mode) faults must occur for each failure mode to cause the ultimate effect (explosion). Each battery cell is UL/UN certified with individual over-voltage and over-current protection. Identical batteries have been flown on all Astro Digital spacecraft. Even in extreme cases (such as a launch vehicle hydrazine explosion in proximity to the spacecraft), the batteries showed no signs of damage or degradation.

Failure mode 1: Internal short circuit.

Mitigation 1: Protoflight level sine burst, sine and random vibration in three axes of the spacecraft, thermal cycling of the spacecraft and extensive functional testing followed by maximum system rate-limited charge and discharge cycles were performed to prove that no internal short circuit sensitivity exists. Additional environmental and functional testing of the batteries at the power subsystem vendor facilities were also conducted on the batteries at the component level.

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 2: Internal thermal rise due to high load discharge rate.

Mitigation 2: Battery cells were tested in lab for high load discharge rates in a variety of flight-like configurations to determine if the feasibility of an out-of-control thermal rise in the cell. Cells were also tested in a hot, thermal vacuum environment (5 cycles at 50° C, then to -20°C) in order to test the upper limit of the cells capability. No failures were observed or identified via satellite telemetry or via external monitoring circuitry.

Combined faults required for realized failure: Spacecraft thermal design must be incorrect **AND** external over-current detection and disconnect function must fail to enable this failure mode.

Failure Mode 3**:** Excessive discharge rate or short-circuit due to external device failure or terminal contact with conductors not at battery voltage levels (due to abrasion or inadequate proximity separation).

Mitigation 3: This failure mode is negated by:

a) qualification tested short circuit protection on each external circuit,

b) design of battery packs and insulators such that no contact with nearby board traces is possible without being caused by some other mechanical failure,

c) observation of such other mechanical failures by protoflight level environmental tests (sine burst, random vibration, thermal cycling, and thermal-vacuum tests).

Combined faults required for realized failure: An external load must fail/short-circuit AND external over-current detection and disconnect function must all occur to enable this failure mode.

Failure Mode 4: Inoperable vents.

Mitigation 4: Battery venting is not inhibited by the battery holder design or the spacecraft design. The battery can vent gases to the external environment.

Combined effects required for realized failure: The cell manufacturer OR the satellite integrator fails to install proper venting.

Failure Mode 5: Crushing

Mitigation 5: This mode is negated by spacecraft design. There are no moving parts in the proximity of the batteries.

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.

Failure Mode 6: Low level current leakage or short-circuit through battery pack case or due to moisture-based degradation of insulators.

Mitigation 6: These modes are negated by:

a) battery holder/case design made of non-conductive material, and

b) operation in vacuum such that no moisture can affect insulators.

Combined faults required for realized failure: Abrasion or piercing failure of circuit board coating or wire insulators **AND** dislocation of battery packs **AND** failure of battery terminal insulators **AND** failure to detect such failures in environmental tests must occur to result in this failure mode.

Failure Mode 7: Excess temperatures due to orbital environment and high discharge combined.

Mitigation 7: The spacecraft thermal design will negate this possibility. Thermal rise has been analyzed in combination with space environment temperatures showing that batteries do not exceed normal allowable operating temperatures under a variety of modeled cases, including worst case orbital scenarios. Analysis shows these temperatures to be well below temperatures of concern for explosions.

Combined faults required for realized failure: Thermal analysis **AND** thermal design **AND** mission simulations in thermal-vacuum chamber testing **AND** over-current monitoring and control must all fail for this failure mode to occur.

Requirement 4.4-2: **Design for passivation after completion of mission operations while in orbit about Earth or the Moon:**

'Design of all spacecraft and launch vehicle orbital stages shall include the ability to deplete all onboard sources of stored energy and disconnect all energy generation sources when they are no longer required for mission operations or post-mission disposal or control to a level which can not cause an explosion or deflagration large enough to release orbital debris or break up the spacecraft (Requirement 56450)."

Compliance statement: Demo9 includes the ability to fully disconnect the Lithium Ion cells from the charging current of the solar arrays. At End-Of-Life, this feature will be used to completely passivate the batteries by removing all energy from them. In the unlikely event that a battery cell does explosively rupture, the small size, mass, and potential energy, of these small batteries is such that while the spacecraft could be expected to vent gases, the debris from the battery rupture should be contained within the spacecraft due to the lack of penetration energy to the multiple enclosures surrounding the batteries.

Requirement 4.4-3. Limiting the long-term risk to other space systems from planned breakups: Compliance statement:

This requirement is not applicable. There are no planned breakups.

Requirement 4.4-4: Limiting the short-term risk to other space systems from planned breakups: Compliance statement:
This requirement is not applicable. There are no planned breakups.

<u>ODAR Section 5</u>: Assessment of Spacecraft Potential for On-Orbit Collisions

Assessment of spacecraft compliance with Requirements 4.5-1 and 4.5-2 (per DAS 3.1.1 and calculation methods provided in NASA-STD-8719.14, section 4.5.4):

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)."

Status: COMPLIANT
Probability: 5.96E-6

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 postmission disposal requirements is less than 0.01 (Requirement 56507)."

Status: COMPLIANT

Identification of all systems or components required to accomplish any post-mission disposal operation, including passivation and maneuvering:

The Flight Computer, Telemetry Transceiver, and Electrical Power Subsystem are needed to complete passivation operations. The spacecraft will passively reenter within 14.1 years of launch.

<u>ODAR Section 6</u>: 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.

6.2 Plan for any spacecraft maneuvers required to accomplish post-mission disposal:

No maneuvers are required to accomplish post-mission disposal.

6.3 Calculation of area-to-mass ratio after post-mission disposal, if the controlled reentry option is not selected (Calculated by DAS 3.1.1):

Spacecraft Mass: 22.5 kg
Cross-sectional Area: 0.1666 m^2 (average tumbling)
Area to mass ratio: 0.0074 m^2/kg

6.4 Assessment of spacecraft compliance with Requirements 4.6-1 through 4.6-5 (per DAS v 2.1.1 and NASA-STD-8719.14 section): Requirement 4.6-1. Disposal for space structures passing through LEO:

"A spacecraft or orbital stage with a perigee altitude below 2000 km shall be disposed of by one of three methods: (Requirement 56557)

a. Atmospheric reentry option: Leave the space structure in an orbit in which natural forces will lead to atmospheric reentry within 25 years after the completion of mission but no more than 30 years after launch; or Maneuver the space structure into a controlled de-orbit trajectory as soon as practical after completion of mission.

b. Storage orbit option: Maneuver the space structure into an orbit with perigee altitude greater than 2000 km and apogee less than GEO - 500 km.

c. Direct retrieval: Retrieve the space structure and remove it from orbit within 10 years after completion of mission."

Analysis:
Demo9 will passively reenter within 14.1 years post-launch from the initial maximum orbital altitude of ~550 km.

This analysis was performed with the NASA Debris Assessment Software 3.1.1. Figure 2 and Figure 3 show the output data from this analysis.

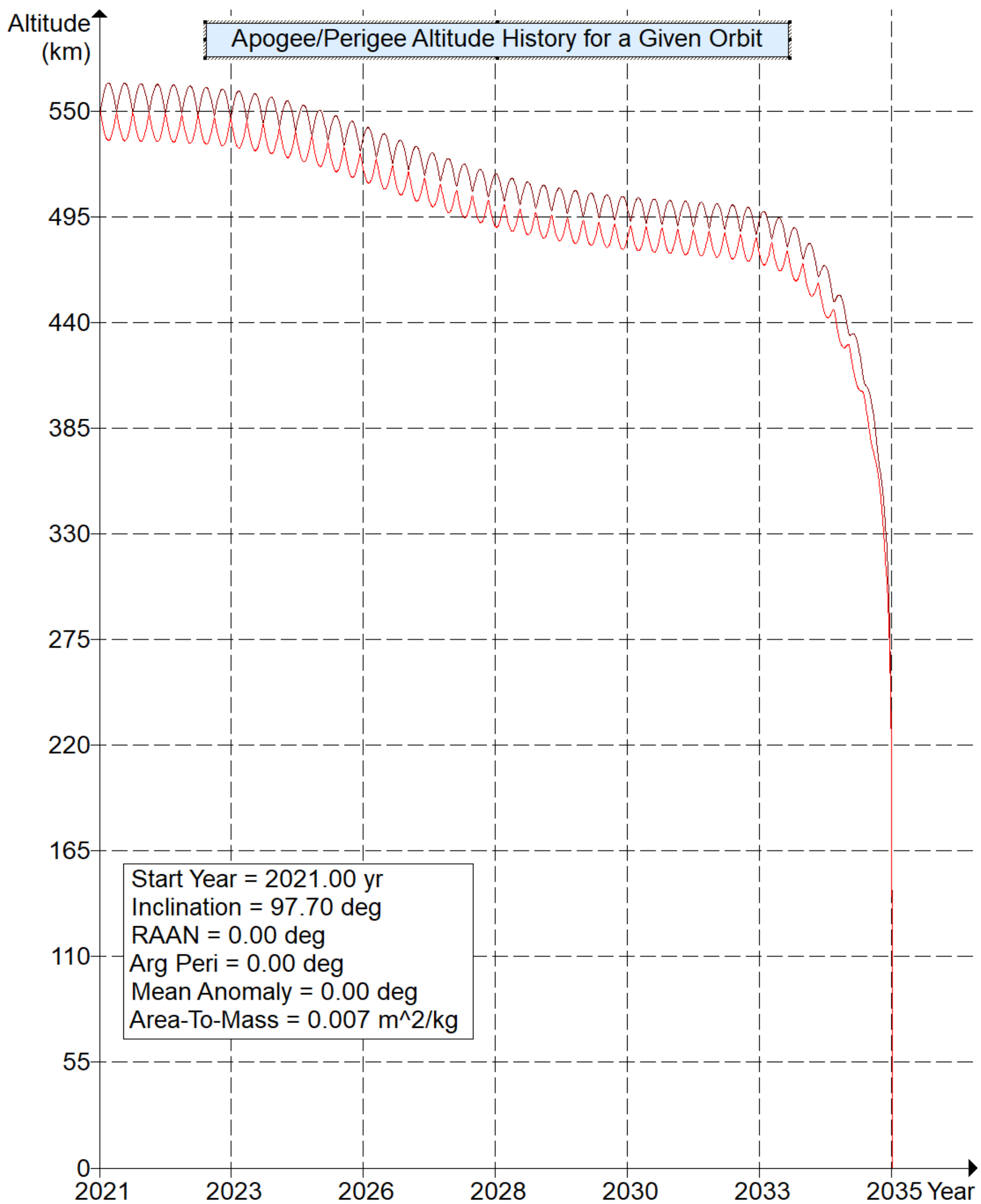

Figure 2: Demo9 Orbital History

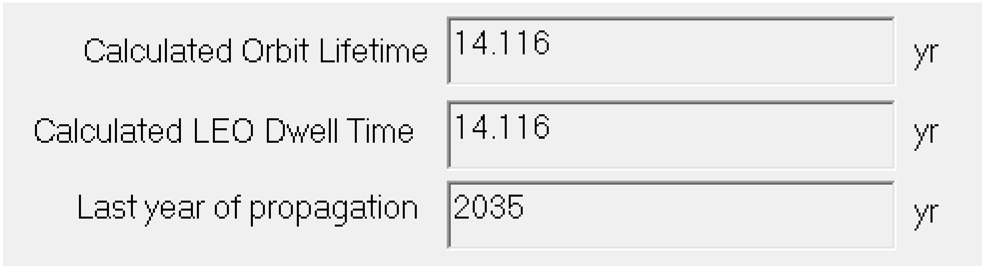

Figure 3: Demo9 Orbit Lifetime/Dwell Time

Requirement 4.6-2. Disposal for space structures near GEO:
Analysis is not applicable.

Requirement 4.6-3. Disposal for space structures between LEO and GEO:
Analysis is not applicable.

Requirement 4.6-4. Reliability of Post-mission Disposal Operations:

The Demo9 spacecraft will satisfy the requirement to deorbit within 25 years after the conclusion of the mission without the functioning of any subsystem.

<u>ODAR Section 7</u>: Assessment of Spacecraft Reentry Hazards:

Assessment of spacecraft compliance with Requirement 4.7-1: Requirement 4.7-1. Limit the risk of human casualty:

"The potential for human casualty is assumed for any object with an impacting kinetic energy in excess of 15 joules:
a) For uncontrolled reentry, the risk of human casualty from surviving debris shall not exceed 0.0001 (1:10,000) (Requirement 56626)."
Summary Analysis Results: DAS v2.1.1 reports that Demo9 is COMPLIANT with the requirement. The critical values reported by the DAS software for the worst-case subcomponent are:

- Demise Altitude = >63 km (worst case)
- Debris Casualty Area = 0.00 m^2
- Impact Kinetic Energy = 0.0 Joules
- Risk of Human Casualty = 1:100000000

This is expected to represent the absolute maximum casualty risk, as calculated with DAS's modeling capability. Zero components are expected to survive reentry and reach the ground. For detailed information on each subcomponent values please refer to appendix.

Requirements 4.7-1b, and 4.7-1c:
These requirements are non-applicable requirements because the spacecraft does not use controlled reentry.

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)."*

Not applicable. The satellite does not use controlled reentry.

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)."*

Not applicable. The satellite does not use controlled reentry.

<u>ODAR Section 8</u>: Assessment for Tether Missions
Not applicable. There are no tethers used in Demo9.

END of ODAR for Demo9

APPENDIX A

The raw DAS report as follows for Demo9:

Processing Requirement 4.3-1: Return Status : Not Run
=====================
No Project Data Available
=====================

Processing Requirement 4.3-2: Return Status : Passed
=====================
No Project Data Available
=====================

Processing Requirement 4.5-1: Return Status : Passed
==============
Run Data
==============
INPUT
Space Structure Name = Shasta
Space Structure Type = Payload
Perigee Altitude = 550.000 (km)
Apogee Altitude = 550.000 (km)
Inclination = 97.700 (deg)
RAAN = 0.000 (deg)
Argument of Perigee = 0.000 (deg)
Mean Anomaly = 0.000 (deg)
Final Area-To-Mass Ratio = 0.0074 (m^2/kg)
Start Year = 2021.000 (yr)
Initial Mass = 22.500 (kg)
Final Mass = 22.500 (kg)
Duration = 5.000 (yr)
Station-Kept = False
Abandoned = True
OUTPUT
Collision Probability = 5.9558E-06
Returned Message: Normal Processing
Date Range Message: Normal Date Range
Status = Pass
==============
=============== End of Requirement 4.5-1 ===============

Requirement 4.5-2: Compliant
=============== End of Requirement 4.5-2 ===============

Processing Requirement 4.6 Return Status : Passed

==============
Project Data
==============
INPUT
Space Structure Name = Shasta
Space Structure Type = Payload
Perigee Altitude = 550.000000 (km)
Apogee Altitude = 550.000000 (km)
Inclination = 97.700000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Area-To-Mass Ratio = 0.007400 (m^2/kg)
Start Year = 2021.000000 (yr)
Initial Mass = 22.500000 (kg)
Final Mass = 22.500000 (kg)
Duration = 5.000000 (yr)
Station Kept = False
Abandoned = True
PMD Perigee Altitude = 515.333489 (km)
PMD Apogee Altitude = 534.189182 (km)
PMD Inclination = 97.685933 (deg)
PMD RAAN = 28.095051 (deg)
PMD Argument of Perigee = 133.896510 (deg)
PMD Mean Anomaly = 0.000000 (deg)
OUTPUT
Suggested Perigee Altitude = 515.333489 (km)
Suggested Apogee Altitude = 534.189182 (km)
Returned Error Message = Passes LEO reentry orbit criteria.
Released Year = 2035 (yr)
Requirement = 61
Compliance Status = Pass
==============
============== End of Requirement 4.6 ===============

Processing Requirement 4.7-1
Return Status : Passed
***********INPUT****
Item Number = 1
name = Shasta
quantity = 1
parent = 0
materialID = 8
type = Box
Aero Mass = 22.500000
Thermal Mass = 22.500000
Diameter/Width = 0.340000
Length = 0.490000
Height = 0.340000

name = Avionics
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 3.640000
Thermal Mass = 3.640000
Diameter/Width = 0.120000
Length = 0.200000
Height = 0.088000

name = Telescope
quantity = 1
parent = 1
materialID = 8
type = Box
Aero Mass = 4.000000
Thermal Mass = 4.000000
Diameter/Width = 0.100000
Length = 0.200000
Height = 0.100000

name = Cooler
quantity = 1
parent = 1
materialID = 54
type = Cylinder
Aero Mass = 0.265000
Thermal Mass = 0.265000
Diameter/Width = 0.066000
Length = 0.090000

name = Panel_PX
quantity = 1
parent = 1
materialID = 8
type = Flat Plate
Aero Mass = 2.075000
Thermal Mass = 2.075000
Diameter/Width = 0.340000
Length = 0.340000

name = Panel_NX
quantity = 1
parent = 1
materialID = 8
type = Flat Plate
Aero Mass = 0.512000
Thermal Mass = 0.512000
Diameter/Width = 0.340000

Length = 0.340000

name = Panel_PY
quantity = 1
parent = 1
materialID = 8
type = Flat Plate
Aero Mass = 0.927200
Thermal Mass = 0.927200
Diameter/Width = 0.340000
Length = 0.490000

name = Panel_NY
quantity = 1
parent = 1
materialID = 8
type = Flat Plate
Aero Mass = 1.070900
Thermal Mass = 1.070900
Diameter/Width = 0.340000
Length = 0.490000

name = Panel_PZ
quantity = 1
parent = 1
materialID = 8
type = Flat Plate
Aero Mass = 1.016900
Thermal Mass = 1.016900
Diameter/Width = 0.340000
Length = 0.490000

name = Panel_NZ
quantity = 1
parent = 1
materialID = 8
type = Flat Plate
Aero Mass = 1.019400
Thermal Mass = 1.019400
Diameter/Width = 0.340000
Length = 0.490000

**************OUTPUT****
Item Number = 1
name = Shasta
Demise Altitude = 77.999107
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Avionics

Demise Altitude = 63.240761
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Telescope
Demise Altitude = 62.812889
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Cooler
Demise Altitude = 65.266609
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Panel_PX
Demise Altitude = 69.939079
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Panel_NX
Demise Altitude = 76.070679
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Panel_PY
Demise Altitude = 75.555702
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Panel_NY
Demise Altitude = 75.153069
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Panel_PZ
Demise Altitude = 75.305656
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Panel_NZ
Demise Altitude = 75.300835
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
============== End of Requirement 4.7-1 ==============