Astro Digital Demo8 (“Tenzing”) Orbital Debris Assessment Report (ODAR)

ASTRO-DIGITAL-DEMO8-ODAR-1.1

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

ASTRO DIGITAL

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DAS Software Version Used In Analysis: 3.1.1

Astro Digital Demo8 Orbital Debris Assessment Report
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Revision:	Date:	Affected Pages:	Changes:	Author(s):
1.0	3/7/2021	All –Initial	Initial	B. Cooper
1.1	3/18/2021		Update DAS3.1.1.	D. Thorne

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

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: Compliant	Launch Vehicle: Not Compliant	Launch Vehicle: Incomplete	Launch Vehicle: Standard Non Compliant	Spacecraft: Compliant	Spacecraft: Not Compliant	Spacecraft: Incomplete	Comments
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.

<u>Assessment Report Format</u>:

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 Demo8 satellite. Sections 9 through 14 apply to the launch vehicle ODAR and are not covered here.

<u>Astro Digital Demo8 Space Mission Program</u>:

<u>ODAR Section 1</u>: 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: Demo8 is a technology demonstration of: a Rapidly Attachable Fluid Transfer Interface ("RAFTI") – an auxiliary propellant tank, fill/drain valve, and docket adapter, on-board propulsion systems – a Benchmark hydrogen-peroxide thruster and Tiled Ionic Liquid Electrospray (TILE) electric propulsion, which will each perform thruster firings to assess maneuvering capabilities for future collision-avoidance, spacecraft-disposal activities, and future rendezvous, proximity operations, and docking ("RPOD") activities; and two stereoscopic cameras, which will conduct non-Earth imaging to assess imaging capabilities for future RPOD activities.

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.

The system concept of operations is to perform multiple low delta-V maneuvers to demonstrate the functionality of the propulsion systems and on-board control software as well as manage the orbital phasing. The on board propulsion systems will not be used to maintain any specific orbital parameters but rather to demonstrate the capabilities to maneuver. A sufficient amount of propellant will be reserved to support an end of mission set of de-orbit maneuvers to accelerate the passive de-orbit of the satellite as well as for collision avoidance maneuvers.

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, as a worst-case scenario, it will passively deorbit within 15.6 years after launch, or 10.6 years after the end of the mission.

Launch and deployment profile, including all parking, transfer, and operational orbits with apogee, perigee, and inclination: The selected launch vehicle will deliver Demo8 directly to its operational circular polar orbit with the following parameters:

Injection Orbital Altitude: 550 km

Eccentricity: 0.0000

Inclination: 97.7°

After the spacecraft has demonstrated all relevant technologies and completed payload operations, the on board propulsion system(s) may be used to execute a deorbit maneuver accelerating the time to de-orbit the satellite. Astro Digital will budget sufficient fuel reserves to target deorbiting the satellite to an orbital altitude of approximately 400 km and engage in anticipated collision avoidance maneuvers.

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

Physical description of the spacecraft:

The Demo8 satellite is based on the standard Corvus-Micro bus and has a total mass of ~35 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.

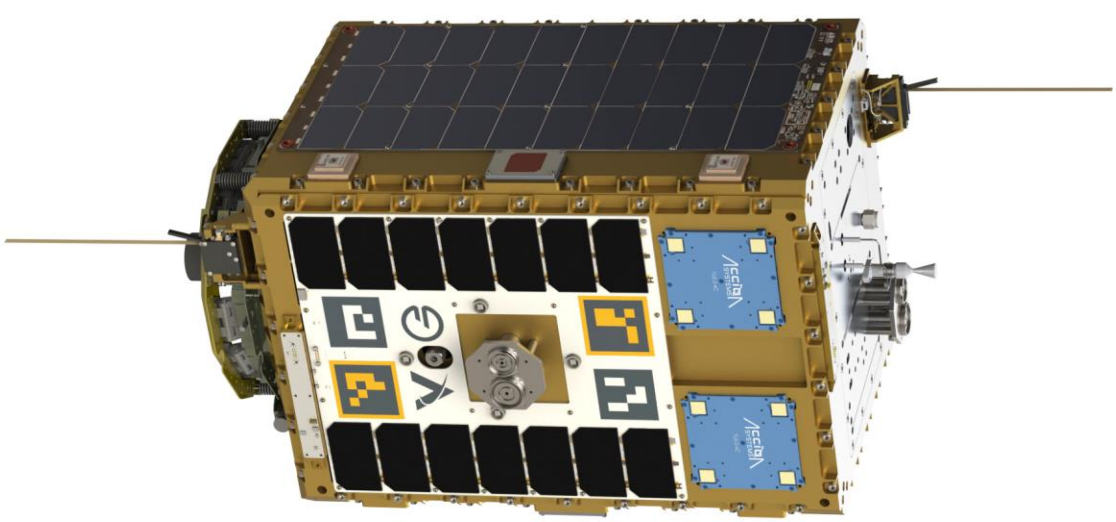

Figure 1: Demo8 Spacecraft

Total satellite mass at launch, including all propellants and fluids:
Demo8: 35 kg

Dry mass of satellites at launch:
Demo8: 29.6 kg

Description of all propulsion systems (cold gas, mono-propellant, bi-propellant, electric, nuclear):

1) Accion Systems Tiled Ionic Liquid Electrospray (TILE) electric propulsion system

2) Benchmark Halcyon mono-propulsion chemical system flows hydrogen peroxide fuel over a platinum on ceramic catalyst bed. The hydrogen peroxide decomposes into water and oxygen, providing up to 1 Newton of thrust. The system is pressurized to 3400 kPa and blows down over the course of the mission to 525 kPa. The fluid flow system has two independent flow check valves to prevent fluid leaks. This thruster has 1014 Newton-seconds of total impulse, and the thrust is reduced to approximately 0.4 Newtons as the tank pressure drops. This system contains stored chemical energy in the hydrogen peroxide and stores pressure in the pressurant tank. The thruster design has already been tested for a NASA flight project and the tank and fluid system has been qualified through an Air Force project.

Neither the Accion TILE nor the Halycon propulsion systems release any persistent liquids. Both propellants will completely vaporize when exposed to vacuum.

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:

Tiled Ionic Liquid Electrospray (TILE) electric propulsion uses Ionic liquid as propellant and is stored at low pressure.

The Benchmark mono-propellant propulsion system uses hydrogen peroxide propellant which is pressurized to 3400 kPa prior to operation. The total propellant mass is 5.4 kg. The propellant will be loaded at the launch site through fill/drain valves located on the spacecraft external structure.

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 Demo8 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 large fixed panels will be oriented towards the sun and the star tracker will be clocked anti-nadir.
- A target tracking link mode will be used to point the spacecraft towards a fixed reference point while passing through its line of sight.
- A *velocity tracking mode,* which will be used to point the thrust head face along the velocity vector to lower the spacecraft's orbit at End-Of-Life.
- 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 panel is mounter in the direction of the payload interfaces 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

ODAR Section 3: 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

<u>ODAR Section 4</u>: Assessment of Spacecraft Intentional Breakups and Potential for Explosions.

Potential causes of spacecraft breakup during deployment and mission operations: There are two potential scenarios that could potentially lead to a breakup of the satellite.

1) Lithium-ion battery cell failure
2) Hydrogen peroxide combustion or pressurization failure (in either the Halcyon propulsion system or auxiliary Hydrogen Peroxide propellant tank)

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.

A failure of either hydrogen peroxide tank (Halycon propulsion system and/or auxiliary propellant tank with RAFTI) has the potential to result in an explosion. Increasing the temperature of the tank results in an increased internal pressure due to both an increased rate of decomposition of the hydrogen peroxide and increased pressure of the already existing pressurant gas. These failure modes will be mitigated through extensive qualification testing, thermally isolating the propellant tank, and pressure release valves. In addition to the pressure relief value the Halycon propulsion system will be fully passivated at the End-Of-Life by releasing all stored propellants and pressurants.

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.

Halycon propulsion system Hydrogen peroxide tank and pressurant. At end of life the spacecraft will execute a set of orbit lowering maneuvers. Following the orbit altitude lowering maneuvers the thruster valves on the thruster will be opened allowing the fuel and pressurant contained in the Halcyon propulsion system to flow out the nozzle until the tank pressure is zero.

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, Demo8: 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 both spacecraft, thermal vacuum cycling of both 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 plastic, 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.

Hydrogen Peroxide Tank Explosion:

Effect: The pressurant tank is stored at an initial pressure of 2100 kPa and is qualified to a burst pressure twice this value. If the burst pressure were to be exceeded, the tank would rupture and potentially breach the aluminum walls of the spacecraft body, releasing debris.

Probability: Extremely Low. Multiple failures would need to occur simultaneously for the pressure to exceed the burst pressure. The tank would need to exceed 100℃ and both independent parallel pressure release valves would need to fail. The tank will be insulated from the rest of the spacecraft, and there are no internal heat sources that could heat the tank to that temperature in any case.

Failure mode 1: Poorly built tank ruptures below burst pressure.

Mitigation 1: Acceptance testing will be performed on the tank, including vibe, thermal cycling, and a proof pressure test at 1.5x the operating pressure.

Failure mode 2: Temperature of the tank exceeds 100℃.

Mitigation 2: The tank will be thermally insulated from external heat sources. A heater is included in the tank to keep it above 0℃. This heater is sized such that even if it is accidentally left on continuously, it cannot cause the tank to exceed 100℃.

Failure mode 3: Both pressure release valves fail.

Mitigation 3: This is fundamentally mitigated by the fact that both valves must fail, which is extremely low probability. Even if both valves fail though, the system should still never experience an over-pressure event once on orbit. As the thruster is fired, the pressure only decreases in the tank.

Failure mode 4: The tank drops below 0℃, hydrogen peroxide freezes, and the tank or fluid system is damaged to the point of leaking.

Mitigation 4: Any leak would only result in hydrogen peroxide and pressurant being released on orbit. Both substances are a gas when exposed to vacuum, so would pose no collision risk once released.

Failure mode 5: The heat of decomposition causes an exothermic chain reaction with hydrogen peroxide igniting backwards through the fluid system and reaching the tank, thus causing an explosion.

Mitigation 5: Extensive testing has been conducted on this system to prevent this exact scenario. Multiple check valves are present in the fluid flow system to prevent this from happening, even in off-nominal scenarios. Mono-propulsion with hydrogen peroxide is a very well understood reaction, having been used since the early days of the space age. This particular thruster design has been fired for a cumulative time much greater than will be needed to empty the entire propellant tank on this mission. Thus, this failure mode is judged to be extremely improbable.

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: Demo8 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.

As discussed above in the propulsion system section, there is no stored energy on the system, so no passivation is required.

All flow valves on the Benchmark mono-propulsion system will be opened at End-Of-Life to execute a final orbit-lowering maneuver. The valves will be left open until all propellant and pressurant is fully exhausted. This will remove all stored energy from the system while also reducing the expected orbital lifetime of the spacecraft.

The RAFTI system utilizes redundant pressure relief values to mitigate over pressurization of the auxiliary propellant tank. A passive pressure relief value for the tank will actuate at a pressure that is less than 50% of the burst pressure of the tank. As part of the End of Life passivation, the system will be configured through the activation of solenoid values to vent the propellant from the tank.

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)."

Large Object Impact and Debris Generation Probability:
Status: COMPLIANT
Probability: 6.9E-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)."

Small Object Impact and Debris Generation Probability:
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, Electrical Power Subsystem, and Benchmark Mono-Propulsion System are needed for post mission disposal operations. The spacecraft is estimated to passively reenter within 15.6 years regardless of any orbit lowering maneuver.

<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 passively atmospheric re-entry. A post-mission orbit-lowering maneuver will be attempted to speed the process, but it is not required.

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 Initial Mass: 34 kg
Initial area to mass ratio: 0.00476 m^2/kg
Spacecraft Final Mass: 29.5 Kg
Final area to mass ratio: 0.0056 m^2/kg
Cross-sectional Area: 0.1666m^2 (average tumbling)

6.4 Assessment of spacecraft compliance with Requirements 4.6-1 through 4.6-5 (per DAS 3.1.1 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:
Demo8 will passively reenter within 15.6 year post-launch from an initial maximum orbital altitude of ~550 km. Additionally, the on-board propulsion systems may be used to attempt to lower the orbit and accelerate the post-mission reentry.
This analysis was performed with the NASA Debris Assessment Software 3.1.1.
Figures 2 and Figure 3 shows the output data from this analysis.

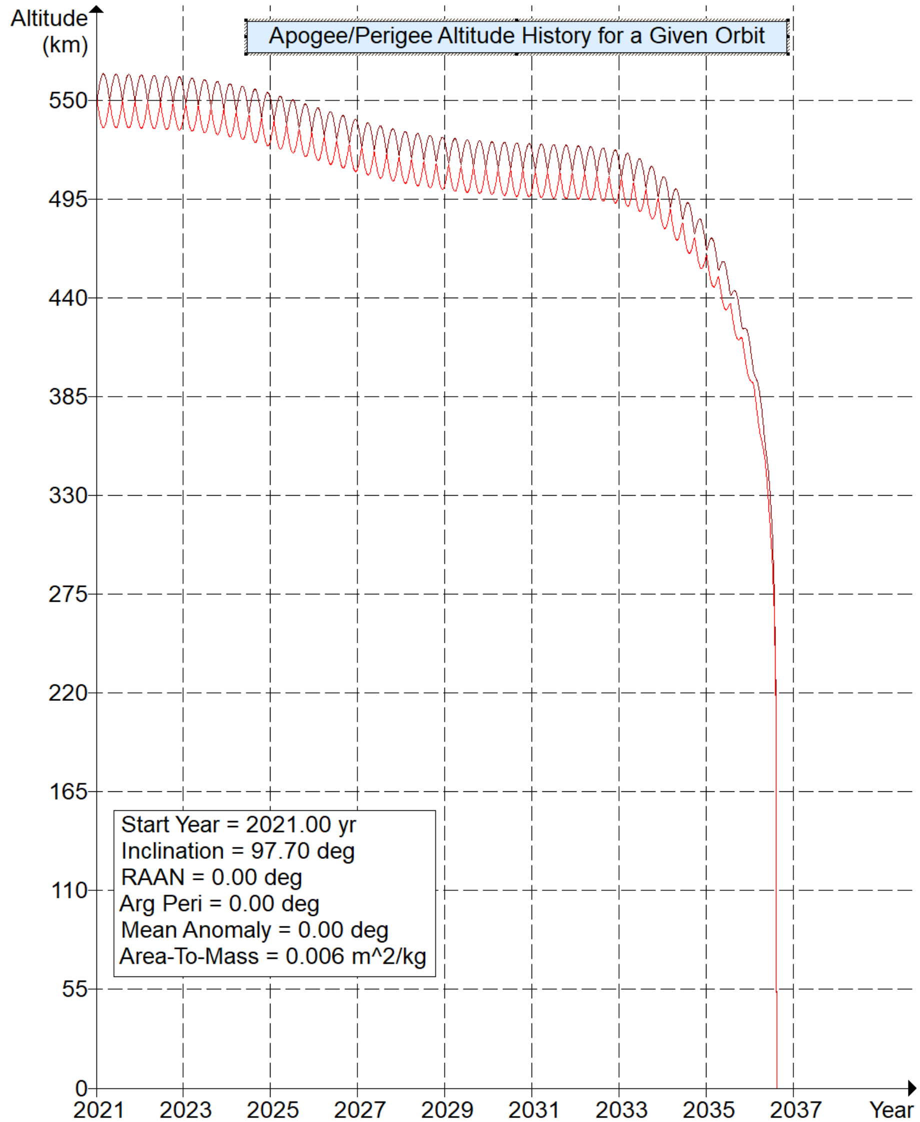

Figure 2: Demo8 Orbital History

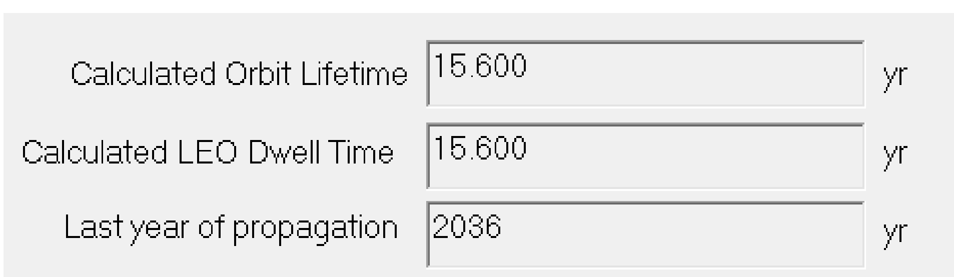

Figure 3: Demo8 Orbit Lifetime/Dwell Time

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 Demo8 spacecraft will satisfy the requirement to deorbit within 25 years after the conclusion of the mission without the functioning of any subsystem. In order to perform the disposal acceleration burn, the spacecraft requires the proper functioning of its attitude determination and control subsystem (ADCS) as well as its propulsion system(s) in order to successfully execute the planned deorbit maneuver. Accordingly, redundancy and reliability have been carefully considered in these disposal-critical areas.

Functional redundancy is provided in the attitude determination subsystem. The spacecraft uses a star tracker as its primary method of attitude determination, and a blend of the high-accuracy gyro, sun sensors, and magnetometers as a secondary method.

Attitude control is accomplished with the reaction wheels. Three wheels, one oriented along each axis, are used for precision pointing. These reaction wheels are manufactured by Sinclair Interplanetary and have extensive on-orbit heritage. The magnetic torquers provide momentum desaturation for the reaction wheels. The spacecraft requires the ability to pulse magnetic torquers along a minimum of two independent axes to maintain attitude control. In the unlikely case of a reaction wheel failure, the magnetic torquers can be used for primary attitude control to continue the deorbit maneuver.

<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 3.1.1 reports that Demo8 is COMPLIANT with the requirement. The critical values reported by the DAS software for the worst-case subcomponent are:

- Demise Altitude = 0.0 km
- Debris Casualty Area = 0.4167 m^2

- Impact Kinetic Energy = 10.7708 Joules
- Risk of Human Casualty = 1:100000000

This is expected to represent the absolute maximum casualty risk, as calculated with DAS's modeling capability. For detailed information on each subcomponent values please refer to the 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 Demo8.

END of ODAR for Demo8

APPENDIX A

The raw DAS report as follows for Demo8:

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

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

Processing Requirement 4.5-1: Return Status : Passed
==============
Run Data
==============
INPUT
Space Structure Name = Tenzing
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 = 0.0056 (m^2/kg)
Start Year = 2021.000 (yr)
Initial Mass = 35.000 (kg)
Final Mass = 29.600 (kg)
Duration = 5.000 (yr)
Station-Kept = False
Abandoned = True
OUTPUT
Collision Probability = 6.8782E-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 = Tenzing
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 = 0.005600 (m^2/kg)
Start Year = 2021.000000 (yr)
Initial Mass = 35.000000 (kg)
Final Mass = 29.600000 (kg)
Duration = 5.000000 (yr)
Station Kept = False
Abandoned = True
PMD Perigee Altitude = 526.769006 (km)
PMD Apogee Altitude = 543.242184 (km)
PMD Inclination = 97.690883 (deg)
PMD RAAN = 25.747143 (deg)
PMD Argument of Perigee = 142.458936 (deg)
PMD Mean Anomaly = 0.000000 (deg)
OUTPUT
Suggested Perigee Altitude = 526.769006 (km)
Suggested Apogee Altitude = 543.242184 (km)
Returned Error Message = Passes LEO reentry orbit criteria.
Released Year = 2036 (yr)
Requirement = 61
Compliance Status = Pass
==============
=============== End of Requirement 4.6 ===============

Processing Requirement 4.7-1 Return Status : Passed
***********INPUT****
name = Tenzing

quantity = 1
parent = 0
materialID = 8
type = Box
Aero Mass = 29.600000
Thermal Mass = 29.600000
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 = Accion
quantity = 2
parent = 1
materialID = 9
type = Box
Aero Mass = 0.523000
Thermal Mass = 0.523000
Diameter/Width = 0.095000
Length = 0.095000
Height = 0.052000

name = Panel_PX
quantity = 1
parent = 1
materialID = 8
type = Flat Plate
Aero Mass = 2.090000
Thermal Mass = 2.090000
Diameter/Width = 0.340000
Length = 0.340000

name = Panel_NX
quantity = 1
parent = 1

materialID = 8
type = Flat Plate
Aero Mass = 0.540000
Thermal Mass = 0.540000
Diameter/Width = 0.340000
Length = 0.340000

name = Panel_PY
quantity = 1
parent = 1
materialID = 8
type = Flat Plate
Aero Mass = 1.130000
Thermal Mass = 1.130000
Diameter/Width = 0.490000
Length = 0.490000

name = PMD component
quantity = 1
parent = 1
materialID = 5
type = Sphere
Aero Mass = 0.100000
Thermal Mass = 0.100000
Diameter/Width = 0.300000

name = RAFTI 1
quantity = 1
parent = 1
materialID = 59
type = Cylinder
Aero Mass = 0.094000
Thermal Mass = 0.094000
Diameter/Width = 0.065000
Length = 0.024000

name = RAFTI 2
quantity = 1
parent = 1
materialID = 59
type = Cylinder
Aero Mass = 0.076000
Thermal Mass = 0.076000
Diameter/Width = 0.025000
Length = 0.032000

name = Halycon tank
quantity = 1
parent = 1
materialID = 5
type = Sphere
Aero Mass = 0.480000
Thermal Mass = 0.480000
Diameter/Width = 0.120000

name = Valve
quantity = 1
parent = 1
materialID = 54
type = Cylinder
Aero Mass = 0.110000
Thermal Mass = 0.110000
Diameter/Width = 0.020000
Length = 0.045000

name = Pressurant trank
quantity = 1
parent = 1
materialID = 5
type = Cylinder
Aero Mass = 0.748000
Thermal Mass = 0.748000
Diameter/Width = 0.129000
Length = 0.080000

name = Check valve
quantity = 1
parent = 1
materialID = 66
type = Box
Aero Mass = 0.022000
Thermal Mass = 0.022000
Diameter/Width = 0.022000
Length = 0.022000
Height = 0.022000

name = Relief manifold
quantity = 1
parent = 1
materialID = 59

type = Box
Aero Mass = 0.120000
Thermal Mass = 0.120000
Diameter/Width = 0.033000
Length = 0.041000
Height = 0.016000

name = Regulator valve
quantity = 1
parent = 1
materialID = 66
type = Box
Aero Mass = 0.046000
Thermal Mass = 0.046000
Diameter/Width = 0.039000
Length = 0.062000
Height = 0.028000

name = Panel_NY
quantity = 1
parent = 1
materialID = 8
type = Flat Plate
Aero Mass = 1.030000
Thermal Mass = 1.030000
Diameter/Width = 0.490000
Length = 0.490000

name = Panel_PZ
quantity = 1
parent = 1
materialID = 8
type = Flat Plate
Aero Mass = 0.990000
Thermal Mass = 0.990000
Diameter/Width = 0.490000
Length = 0.490000

name = Panel_NZ
quantity = 1
parent = 1
materialID = 8
type = Flat Plate
Aero Mass = 0.990000
Thermal Mass = 0.990000

Diameter/Width = 0.490000
Length = 0.490000

name = Depot tank
quantity = 1
parent = 1
materialID = 8
type = Cylinder
Aero Mass = 2.300000
Thermal Mass = 2.300000
Diameter/Width = 0.190000
Length = 0.260000

**************OUTPUT****
Item Number = 1
name = Tenzing
Demise Altitude = 77.994568
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Avionics
Demise Altitude = 64.238113
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Accion
Demise Altitude = 72.559929
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Panel_PX
Demise Altitude = 70.565346
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Panel_NX
Demise Altitude = 76.138092
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Panel_PY
Demise Altitude = 75.382629
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****

name = PMD component
Demise Altitude = 77.492958
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = RAFTI 1
Demise Altitude = 73.821022
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = RAFTI 2
Demise Altitude = 71.038307
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Halycon tank
Demise Altitude = 69.784973
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Valve
Demise Altitude = 67.876602
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Pressurant trank
Demise Altitude = 71.233536
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Check valve
Demise Altitude = 0.000000
Debris Casualty Area = 0.386884
Impact Kinetic Energy = 7.530383
****
name = Relief manifold
Demise Altitude = 69.698166
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Regulator valve
Demise Altitude = 0.000000
Debris Casualty Area = 0.416766
Impact Kinetic Energy = 10.770845
****

name = Panel_NY
Demise Altitude = 75.624199
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Panel_PZ
Demise Altitude = 75.735298
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Panel_NZ
Demise Altitude = 75.735298
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
name = Depot tank
Demise Altitude = 71.842743
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
****
=============== End of Requirement 4.7-1 ===============