Astro Digital Landmapper-Demo Orbital Debris Assessment Report (ODAR)

ODAR-1.1

This report is presented as compliance with NASA-STD-8719.14, APPENDIX A.
Report Version: 1.1, 5/21/2020

Astro Digital US, Inc.
3171 Jay Street
Santa Clara, California 95054

DAS Software Version Used In Analysis: v2.1.1

Astro Digital Landmapper-Demo Orbital Debris Assessment Report
ODAR-1.1

APPROVAL:

Chris Biddy
CEO, Astro Digital

/s/ Chris Biddy

Jan A. King
CTO, Astro Digital

/s/ Jan. A. King

Patrick Shannon
Program Manager
Landmapper-Demo Program

/s/ Patrick Shannon

Revision Record				
Revision:	Date:	Affected Pages:	Changes:	Author(s):
1.0	5/8/2020	All –Initial	Initial	B. Cooper
1.1	5/21/2020	Various	Orbit, reentry time	B. Cooper

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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				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.

<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 Landmapper-Demo satellites, specifically Landmapper-Demo6 and Demo7. Sections 9 through 14 apply to the launch vehicle ODAR and are not covered here.

<u>Astro Digital Landmapper-Demo Space Mission Program</u>:

<u>ODAR Section 1</u>: Program Management and Mission Overview

Program Manager: Patrick Shannon
Mission Manager: Brian Cooper
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: Q4 2020
- First Launch: Q4 2020

Mission Overview: Landmapper-Demo is a technology demonstration mission consisting of two separate, identical spacecraft: Landmapper-Demo6 (Demo6) and Landmapper-Demo7 (Demo7). Each spacecraft includes two payloads: an optical communication system; and a high-performance payload data processor.

The spacecraft bus is the Corvus-XL design. The common satellite bus uses reaction wheels, magnetic torque coils, star trackers, magnetometers, sun sensors, and gyroscopes to enable precision 3-axis pointing. The satellites have an electric propulsion system and a mono-propellant chemical thruster to provide functional redundancy at the mission level for orbital maneuvers.

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

Proposed Initial Launch Date: Q4 2020

Mission Duration: The design lifetime of the spacecraft hardware is a minimum of 5 years in LEO. Given the spacecraft's maximum expected deployment orbital altitude of 600 km, the spacecraft will passively deorbit within a maximum of 15.8 years. However, the electric propulsion system will be used at the end of the mission to accelerate the orbital decay to a total of approximately five years after completion of the deorbit maneuver.

Launch and deployment profile, including all parking, transfer, and operational orbits with apogee, perigee, and inclination: The selected launch vehicle will deliver the two satellites directly to their operational circular polar orbit at an altitude of 450-600 km. During operations, the satellites will vary their relative separation. They will remain within a separation range of 100 km to 5000 km. Astro Digital will use the propulsion systems primarily to maintain distancing between the satellites, facilitate the testing of the optical payloads, and support the end-of-mission de-orbiting. The spacecraft will operate for an expected 5 years from an orbit with the following parameters:

Deployment Orbital Altitude Range: 450 km to 600 km

Eccentricity: 0.0000 to 0.0033

Inclination: 97.4° to 97.8°

After the spacecraft has demonstrated all relevant technologies and completed payload operations, the electric propulsion system will be used to execute a deorbit maneuver, targeting an orbital altitude of less than 450 km. This orbit will result in a passive orbital decay within 5 years after the completion of the deorbit maneuver.

Astro Digital will reserve propellant to conduct this post-mission orbit-lowering maneuver.

ODAR Section 2: Spacecraft Description:

Physical description of the spacecraft:

Landmapper-Demo6 and Demo7 are based on the standard Corvus-XL bus and each has a total mass of ~80 kg. The main spacecraft body has dimensions of 52.4 cm x 52.4 cm x 62 cm. The main solar panel is fixed to the body and overhangs the spacecraft body in the -X axis, making it 120 cm long in total. There are two Optical Intersatellite Link units (OISLs), one attached to the +Z face of the spacecraft and one on the -Z face. 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 15-inch Planetary Systems Lightband on the +X face of the spacecraft that is used to deploy the spacecraft from the launch vehicle. Two independent TT&C antennas protrude from the +X face of the spacecraft and are canted by 45 degrees in the +Y and -Y direction respectively. 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 both of their antennas are placed on the -Z face. A star tracker points out of the -X face. The spacecraft includes two independent propulsion systems. The first is a mono-propulsion chemical system with approximately 1000 Newton-seconds of total impulse and a maximum thrust of 1 Newton. The second is a field emission electric propulsion system with a total impulse of 6000 Newton-seconds and a maximum thrust of 350 micronewtons.

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 2017 on various missions including Corvus-BC1-4, Sirion Pathfinder-2, and Momentus X1. The attitude determination and control system has been operating on orbit since 2014 on Perseus-M1 and Perseus-M2. The high-voltage electrical power system has been qualified for flight. The spacecraft is depicted in Figure 1.

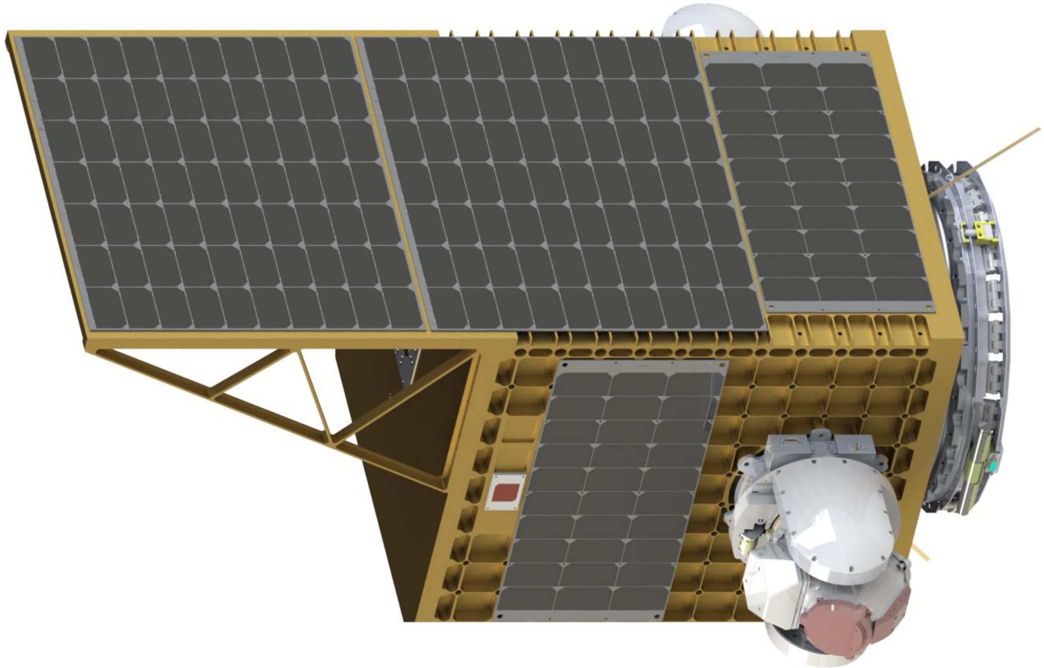

Figure 1: Landmapper-Demo *Spacecraft*

Total satellite mass at launch, including all propellants and fluids:
80.7 kg +/- 5.0 kg

Dry mass of satellites at launch:
79.5 kg +/- 5.0 kg

Description of all propulsion systems (cold gas, mono-propellant, bi-propellant, electric, nuclear): The Enpulsion field-emission electric propulsion system ionizes solid indium propellant and accelerates it using a strong electric field along a micro-structure emitter tip. The resulting plasma is expelled to produce thrust at an Isp in excess of 2000 seconds. The expected thrust and Isp can be varied with power input levels. Thrust will not exceed 350 μN maximum. The Enpulsion propulsion system includes only 200 grams of solid indium propellant. The thruster is capable of providing ~7000 N-s of impulse with this propellant load, and ~6500 N-s is needed for this mission. This system does not contain any stored energy.

The Tesseract Lyra 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 2100 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 store 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 propulsion systems results in the release of any persistent liquids. The hydrogen peroxide in the Tesseract system will completely vaporize when exposed to vacuum. The indium in the Enpulsion system is a solid ingot of metal. The only way for the indium to flow out of the thruster is for the heater to melt the ingot and allow it to flow out of the thrust head. The path from the tank to space requires the indium to pass through extremely small pores in the tungsten emitter tips, and the only way for the indium to flow through these tips is for it to be vaporized into extremely small particles (*i.e.*, only a few atoms wide at a time).

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 indium ingot in the Enpulsion system is launched as a solid and only melted once on-orbit. The unit is delivered from the manufacturer already loaded with propellant.

The Tesseract mono-propulsion system includes 739 grams of hydrogen peroxide and less than 2 grams of helium pressurant. The tank is pressurized to 2100 kPa prior to launch. The propellant and pressurant will be loaded at the launch site through fill/drain valves located on the -X face of the spacecraft.

Fluids in Pressurized Batteries: None

The Corvus-XL satellite design uses four unpressurized standard COTS Lithium-Ion battery cells in parallel for the low voltage system. The energy capacity of each cell is 17 W-hrs, and the total capacity of the low voltage system is 68 W-hrs. The spacecraft also includes a high voltage electrical system which consists of four batteries made up of six cells each in series. The peak voltage of the battery packs is 24 volts. This high voltage battery system includes 408 W-hrs of total storage capacity.

Description of attitude control system and indication of the normal attitude of the spacecraft with respect to the velocity vector: The Landmapper-Demo spacecraft will activate its attitude determination system within 2 minutes after deployment. Fifteen minutes 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.

- A *<u>sun pointing safe mode</u>* 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. This mode will make use of magnetometers, sun sensors, gyroscope, reaction wheels, and magnetic torquers to orient the spacecraft correctly.

- A *sun pointing link mode* that is optimized for solar power generation and allows the satellite to maintain an intersatellite link with the +Z OISL. The spacecraft's large fixed panels will be oriented towards the sun and the star tracker will be clocked to point along the velocity vector. This mode will make use of magnetometers, sun sensors, gyroscope, reaction wheels, and magnetic torquers to orient the spacecraft correctly.
- A *target tracking mode*, which will allow the spacecraft to point +X axis towards an arbitrary inertial axis to allow a dual laser communication link. This mode will make use of reaction wheels and a star tracker to orient the spacecraft.
- A *velocity tracking mode*, which will be used to point the thrust head face along the velocity or anti-velocity vector to allow for phasing maneuvers between the two spacecraft. This mode will also be used to lower the spacecraft's orbit at End-Of-Life. This mode will make use of the reaction wheels and a star tracker to orient the spacecraft.

Description of any range safety or other pyrotechnic devices: None. The solar panels are fixed to the body of the spacecraft and the UHF antennas are already extended prior to launch.

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 capacity of this battery is 68 W-hrs.

The all-parallel bus battery is charged through the solar panels and also through a higher voltage "payload battery" that consists of 4 batteries with 6 battery cells in series each. This results in a robust architecture where the bus electronics are effectively always being charged as if in sunlight, even in eclipse or intensive operations modes. The capacity of the payload battery is 408 W-hrs.

The main solar panels are equipped with 14 strings of 12 cells in series (168 cells total). The spacecraft also includes 3 "backup" solar panels on non-sun pointing faces to provide power in the case of a safe mode tumble.

Typical bus operations consume 12 watts of power on average. The thruster can consume up to 40 watts while firing. Each OISL consumes up to 40 watts. The payload data processor consumes up to 250 watts. 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 v2.1.1)
4.3-1, Mission Related Debris Passing Through LEO: COMPLIANT
4.3-2, Mission Related Debris Passing Near GEO: N/A

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
2) Hydrogen peroxide combustion or pressurization 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.

A failure of the hydrogen peroxide tank has the potential to result in an explosion. Increasing the temperature of the tank results in 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, two redundant pressure release valves, and fully passivating the system at 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: Twenty-eight (28) Lithium Ion Battery Cells. Solar array charging will be disabled, which will fully discharge all cells within two days.

Hydrogen peroxide tank and pressurant. The spacecraft will execute a final orbit lowering maneuver. The maneuver will open all valves on the thruster permanently, allowing all fuel and pressurant 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:

Required Probability: 0.001

Expected probability: 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° C, 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° C.

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

Failure mode 3: Both pressure release valves fail.

Mitigation 3: This is fundamentally mitigated by the fact that both valves must fail, which has an 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° C, 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: The Landmapper-Demo satellites have 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 Enpulsion propulsion system, so no passivation is required.

All flow valves on the Tesseract 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 are fully exhausted. This will remove all stored energy from the system while also reducing the expected orbital lifetime of the spacecraft.

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 v2.1.1 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: 0.000039; COMPLIANT.

Additionally, Astro Digital will use the propulsion systems on the Landmapper-Demo satellites, as appropriate, to mitigate the probability of collision.

The collision probability of a single Landmapper-Demo satellite is 0.000039. The collision probability of a single Landmapper-BC satellite is 0.000007. The probability of collision of a Landmapper-HD satellite is 0.000014. The overall collision probability for the Landmapper constellation comprised of 2 Landmapper-BC satellites, 2 Landmapper-Demo satellites, and 26 Landmapper-HD satellites is 0.000456, which is less than 0.001.

Moreover, because the Landmapper-Demo satellites are maneuverable and the Landmapper-HD satellites will also be maneuverable, the effective system collision probability will be even lower. The satellites will be able to maneuver to avoid in-orbit collisions. And at end-of-life, the satellites will lower their altitude to facilitate deorbit.

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: 0.003007; 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 Tesseract Mono-Propulsion System are needed to complete passivation operations. At the maximum deployment altitude of 600 km, the spacecraft will passively reenter within 15.8 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 naturally by atmospheric re-entry. Astro Digital will reserve propellant to conduct a post-mission orbit-lowering maneuver to speed the process, but it is not needed to de-orbit within NASA guidelines.

6.2 Plan for any spacecraft maneuvers required to accomplish post-mission disposal: No maneuvers are required to accomplish post-mission disposal within NASA guidelines. Nonetheless, Astro Digital intends to conduct a post-mission de-orbit maneuver to expedite de-orbit lifetime for the satellites. See discussion above.

6.3 Calculation of area-to-mass ratio after post-mission disposal, if the controlled reentry option is not selected:

Spacecraft Mass: 80.7 kg (selected as worst-case mass)
Cross-sectional Area: 0.623 m^2 (average tumbling)
(Calculated by DAS 2.1.1). Area to mass ratio: 0.0077 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:
The Landmapper-Demo satellites will passively reenter within 15.8 years post-mission for the "worst-case" initial orbital altitude of 600 km. Additionally, the on-board propulsion systems will be used to attempt to lower the orbit and accelerate the post-mission reentry. By lowering the spacecraft to below a 450 km altitude, the spacecraft will reenter the Earth's atmosphere within approximately 5 years after the end of mission.

This analysis was performed with the NASA Debris Assessment Software 2.1.1. Figure 2 and Figure 3 show the output data from this analysis.

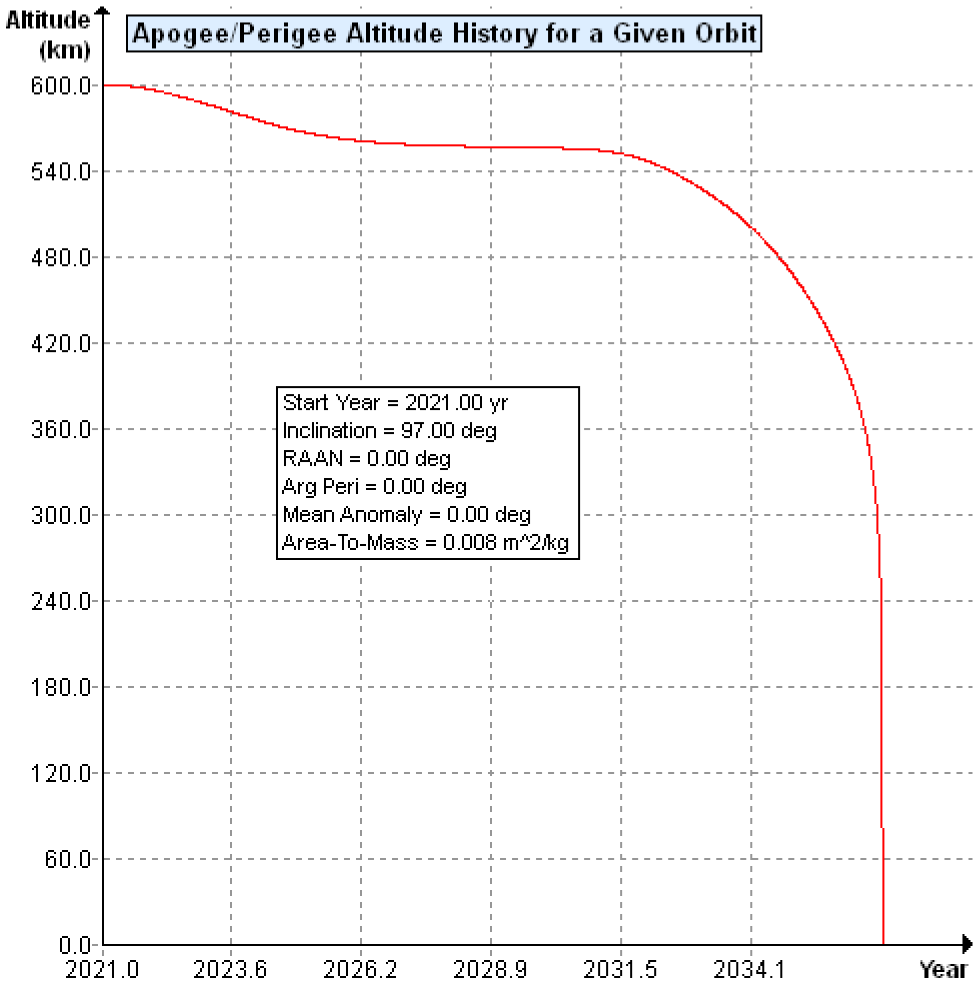

Figure 2: Landmapper-Demo Orbital Altitude History (Failure Upon Deployment). Satellite decays naturally after 15.8 years.

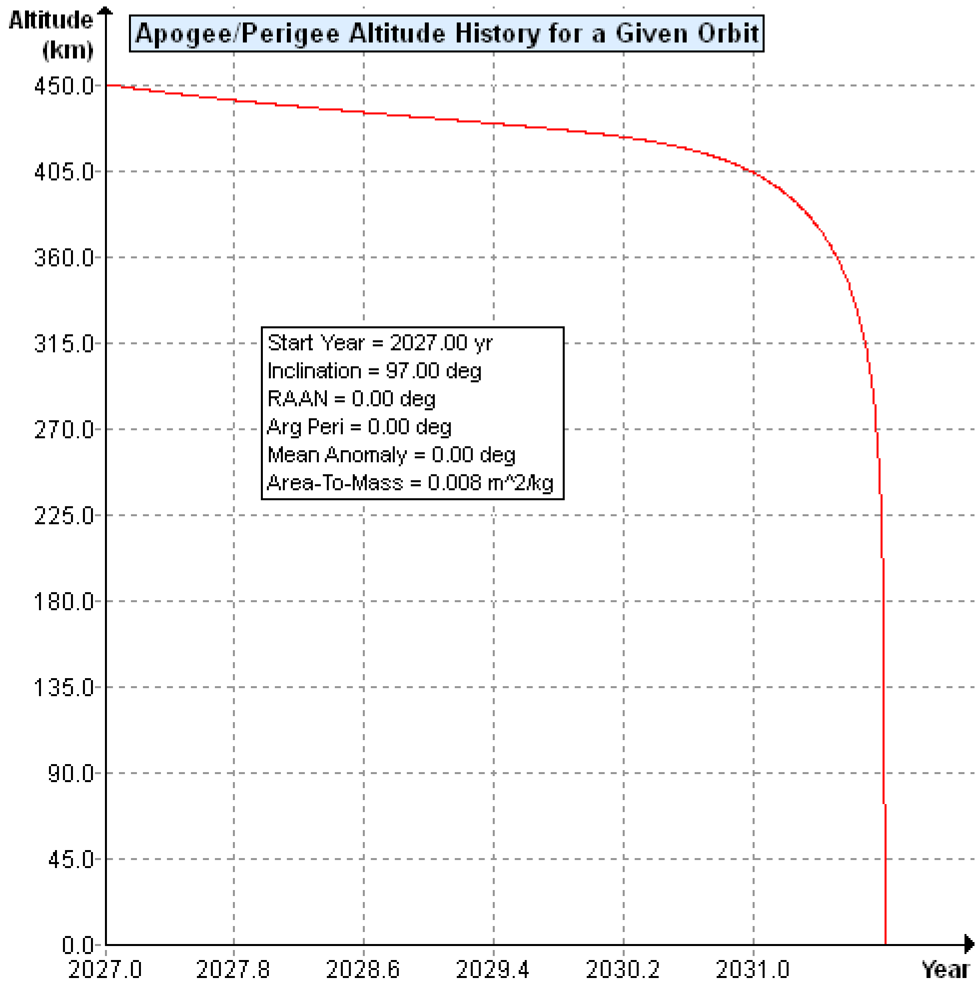

Figure 3: Landmapper-Demo Post-Mission Orbit History with Disposal Maneuver to 450 km

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 Landmapper-Demo 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 Enpulsion propulsion system 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. The magnetic torquers provide momentum desaturation for the reaction wheels. The spacecraft requires the ability to fire magnetic torquers along a minimum of two independent axes to maintain attitude control. A total of six torque coils are included in the spacecraft in two groups with different reliability chains to prevent a systematic failure. In the unlikely case of a reaction wheel failure, the magnetic torquers can be used for primary attitude control to continue the deorbit maneuver.

The Enpulsion propulsion system has no moving parts and has been flight proven on orbit. Subsystem-level redundancy is not provided in this case, but the reliability of the propulsion system is judged to be high due to its proven flight heritage both as a spacecraft propulsion system and a spacecraft charge neutralizer. The thruster itself includes internal redundancy in areas that are determined to be relatively higher risk, including the neutralizer and electron emitter portions of the thruster. The total impulse provided by the thruster is much greater than that required to accomplish the necessary deorbit maneuver.

In the unlikely case of an Enpulsion thruster failure, the Tesseract thruster can be used to execute a modified deorbit acceleration burn. The total available impulse of the thruster is less than the Enpulsion thruster, so the perigee will be preferentially lowered in this case to increase the drag on the spacecraft as much as possible.

ODAR Section 7: 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 Landmapper-Demo6 and Demo7 satellites are COMPLIANT with the requirement. The critical values reported by the DAS software are:

- Demise Altitude = 0.0 km
- Debris Casualty Area = 0.00 m^2
- Impact Kinetic Energy = 14 Joules and 5 Joules for the largest objects
- Risk of Human Casualty = 1:100000000

This is expected to represent the absolute maximum casualty risk, as calculated with DAS's modeling capability. The two surviving components reach the ground with energy less than 15 Joules, meaning no casualties are expected.

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 to Landmapper-Demo. 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 Landmapper-Demo.

END of ODAR for Landmapper-Demo

The raw DAS report as follows for Landmapper-Demo:

=====================
No Project Data Available
=====================

=============== End of Requirement 4.3-1 ===============
04 08 2020; 12:18:39PM Processing Requirement 4.3-2: Return Status : Passed

=====================
No Project Data Available
=====================

=============== End of Requirement 4.3-2 ===============
04 08 2020; 12:18:42PM Requirement 4.4-3: Compliant

=============== End of Requirement 4.4-3 ===============
04 08 2020; 12:44:49PM Processing Requirement 4.5-1: Return Status : Passed

==============
Run Data
==============

INPUT

Space Structure Name = Landmapper-Demo
Space Structure Type = Payload
Perigee Altitude = 600.000000 (km)
Apogee Altitude = 600.000000 (km)
Inclination = 97.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass Ratio = 0.007000 (m^2/kg)
Start Year = 2021.000000 (yr)
Initial Mass = 81.000000 (kg)
Final Mass = 80.000000 (kg)
Duration = 5.000000 (yr)
Station-Kept = False
Abandoned = True
PMD Perigee Altitude = -1.000000 (km)
PMD Apogee Altitude = -1.000000 (km)
PMD Inclination = 0.000000 (deg)
PMD RAAN = 0.000000 (deg)

PMD Argument of Perigee = 0.000000 (deg)
PMD Mean Anomaly = 0.000000 (deg)

OUTPUT

Collision Probability = 0.000039
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range
Status = Pass

==============

=============== End of Requirement 4.5-1 ===============
04 08 2020; 13:30:34PM Processing Requirement 4.6Return Status : Passed
04 21 2020; 16:40:51PM Requirement 4.5-2: Compliant

==
Spacecraft = Landmapper-Demo
Critical Surface = Solar panel
==

INPUT

Apogee Altitude = 600.000000 (km)
Perigee Altitude = 600.000000 (km)
Orbital Inclination = 97.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass = 0.007000 (m^2/kg)
Initial Mass = 80.000000 (kg)
Final Mass = 80.000000 (kg)
Station Kept = No
Start Year = 2021.000000 (yr)
Duration = 5.000000 (yr)
Orientation = Random Tumbling
CS Areal Density = 1.000000 (g/cm^2)
CS Surface Area = 0.600000 (m^2)
Vector = (0.000000 (u), 0.000000 (v), 0.000000 (w))
CS Pressurized = No
Outer Wall 1 Density: 1.000000 (g/cm^2) Separation: 3.000000 (cm)

OUTPUT

Probabilty of Penetration = 0.003007 (0.003011)
Returned Error Message: Normal Processing

Date Range Error Message: Normal Date Range

=============== End of Requirement 4.5-2 ===============

==============
Project Data
==============

INPUT

Space Structure Name = Landmapper-Demo
Space Structure Type = Payload

Perigee Altitude = 600.000000 (km)
Apogee Altitude = 600.000000 (km)
Inclination = 97.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Area-To-Mass Ratio = 0.007000 (m^2/kg)
Start Year = 2021.000000 (yr)
Initial Mass = 81.000000 (kg)
Final Mass = 80.000000 (kg)
Duration = 5.000000 (yr)
Station Kept = False
Abandoned = True
PMD Perigee Altitude = 566.326069 (km)
PMD Apogee Altitude = 566.326069 (km)
PMD Inclination = 96.984482 (deg)
PMD RAAN = 194.000067 (deg)
PMD Argument of Perigee = 357.381097 (deg)
PMD Mean Anomaly = 0.000000 (deg)

OUTPUT

Suggested Perigee Altitude = 566.326069 (km)
Suggested Apogee Altitude = 566.326069 (km)
Returned Error Message = Passes LEO reentry orbit criteria.

Released Year = 2041 (yr)
Requirement = 61
Compliance Status = Pass

==============

=============== End of Requirement 4.6 ===============

04 08 2020; 13:38:45PM *********Processing Requirement 4.7-1
Return Status : Passed

***********INPUT****
Item Number = 1

name = Landmapper-Demo
quantity = 1
parent = 0
materialID = 5
type = Box
Aero Mass = 80.000000
Thermal Mass = 80.000000
Diameter/Width = 0.500000
Length = 0.600000
Height = 0.500000

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

name = Fluids part
quantity = 1
parent = 1
materialID = 40
type = Cylinder
Aero Mass = 0.002000
Thermal Mass = 0.002000
Diameter/Width = 0.050000
Length = 0.050000

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 = OISL Rod
quantity = 2
parent = 1
materialID = 66
type = Flat Plate
Aero Mass = 0.002000
Thermal Mass = 0.002000
Diameter/Width = 0.010000
Length = 0.050000

name = OISL Glass Lens
quantity = 2
parent = 1
materialID = -2
type = Flat Plate
Aero Mass = 0.282000
Thermal Mass = 0.282000
Diameter/Width = 0.150000
Length = 0.150000

name = OISL Silicon Lens
quantity = 2
parent = 1
materialID = -4
type = Flat Plate
Aero Mass = 0.103000
Thermal Mass = 0.103000
Diameter/Width = 0.110000
Length = 0.110000

name = OISL Ring
quantity = 2
parent = 1
materialID = 54
type = Flat Plate
Aero Mass = 0.191000
Thermal Mass = 0.191000
Diameter/Width = 0.200000
Length = 0.630000

**************OUTPUT****
Item Number = 1

name = Landmapper-Demo

Demise Altitude = 77.996704
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Tank
Demise Altitude = 70.720284
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Fluids part
Demise Altitude = 77.938530
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Valve
Demise Altitude = 68.998596
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = OISL Rod
Demise Altitude = 0.000000
Debris Casualty Area = 0.774666
Impact Kinetic Energy = 0.130532

****
name = OISL Glass Lens
Demise Altitude = 77.183861
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = OISL Silicon Lens
Demise Altitude = 0.000000
Debris Casualty Area = 1.008200
Impact Kinetic Energy = 14.313848

****
name = OISL Ring
Demise Altitude = 0.000000
Debris Casualty Area = 1.823916
Impact Kinetic Energy = 4.722767

****

=============== End of Requirement 4.7-1 ===============
04 08 2020; 14:06:34PM   Requirement 4.5-2: Compliant

==
Spacecraft = Landmapper-Demo
Critical Surface = Solar panel
==

INPUT

Apogee Altitude = 600.000000 (km)
Perigee Altitude = 600.000000 (km)
Orbital Inclination = 97.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass = 0.007000 (m^2/kg)
Initial Mass = 80.000000 (kg)
Final Mass = 80.000000 (kg)
Station Kept = No
Start Year = 2021.000000 (yr)
Duration = 5.000000 (yr)
Orientation = Random Tumbling
CS Areal Density = 1.000000 (g/cm^2)
CS Surface Area = 0.600000 (m^2)
Vector = (0.000000 (u), 0.000000 (v), 0.000000 (w))
CS Pressurized = No
Outer Wall 1   Density: 1.000000 (g/cm^2) Separation: 3.000000 (cm)

OUTPUT

Probabilty of Penetration = 0.003007 (0.003011)
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range

Appendix A: Acronyms

Arg peri	Argument of Perigee
CDR	Critical Design Review
Cm	centimeter
COTS	Commercial Off-The-Shelf (items)

DAS	Debris Assessment Software
ECM	ECM Space Technologies GmbH (also known as ExoLaunch)
EOM	End Of Mission
FRR	Flight Readiness Review
GEO	Geosynchronous Earth Orbit
Isp	Specific Impulse
ITAR	International Traffic In Arms Regulations
Kg	kilogram
Km	kilometer
LEO	Low Earth Orbit
Li-Ion	Lithium Ion
m^2	Meters squared
ml	milliliter
mm	millimeter
N/A	Not Applicable.
NET	Not Earlier Than
ODAR	Orbital Debris Assessment Report
OSMA	Office of Safety and Mission Assurance
PDR	Preliminary Design Review
PL	Payload
PSIa	Pounds Per Square Inch, absolute
RAAN	Right Ascension of the Ascending Node
SMA	Safety and Mission Assurance
Yr	year