Flock Orbital Debris Assessment Report (ODAR)
Special Edition for Propulsion System Tech Demo

This report is presented in compliance with NASA-STD-8719.14, APPENDIX A.

Report Version: 1.0, October 25, 2017

Document Data is Not Restricted.

This document contains no proprietary, ITAR, or export controlled information.

DAS Software Version Used In Analysis: v2.0.2

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.

Orbital Debris Self-Assessment Report Evaluation: Dove Satellite Demonstration with a Propulsion System

Rqmnt #	Spacecraft			Comments
	Compliant	Not Compliant	Incomplete	
4.3-1.a	☒	☐	☐	No Debris Released in LEO.
4.3-1.b	☒	☐	☐	No Debris Released in LEO.
4.3-2	☒	☐	☐	No Debris Released in GEO.
4.4-1	☒	☐	☐	Compliant.
4.4-2	☒	☐	☐	Compliant.
4.4-3	☒	☐	☐	No Planned Breakups.
4.4-4	☒	☐	☐	No Planned Breakups.
4.5-1	☒	☐	☐	Compliant
4.5-2	☒	☐	☐	No Critical Subsystems Needed for EOM Disposal
4.6-1	☒	☐	☐	Atmospheric Entry.
4.6-2	☒	☐	☐	No Disposal Near GEO
4.6-3	☒	☐	☐	No Disposal Between LEO and GEO
4.6-4	☒	☐	☐	Compliant.
4.7-1	☒	☐	☐	Compliant.
4.8-1	☒	☐	☐	No Tethers Used.

Assessment Report Format:

ODAR Technical Sections Format Requirements:

As Planet Labs Inc. is a US company, this ODAR follows the format recommended in NASA-STD-8719.14, Appendix A.1 and includes the content indicated at a minimum in each section 2 through 8 below for the propulsion technology demonstration satellite, aka “Dove Turbo”, in the Flock constellation. Sections 9 through 14 apply to the launch platform and are not covered here.

ODAR Section 1: Program Management and Mission Overview

Project Manager: Planet Labs

Foreign government or space agency participation: None

Launch: Q4 2017 and onwards

Mission Overview:

Dove Turbo is physically identical (dimensions and mass) to regular Dove satellites except the imaging camera is replaced with a propulsion system.

A Dove with a nano propulsion system will be launched on a PSLV rocket into a 505 km SSO orbit, although alternate launch opportunities not exceeding 550 km in altitude may also be available. This analysis covers the specifics of that satellite and orbital parameters and presents the worst-case scenarios for each relevant requirement.

After the launch, the Dove satellite will be commissioned for approximately two weeks. During this period, 3-axis control is enabled and solar arrays are deployed. After commissioning, the satellite will have an operational life not exceeding 3 years. During the operational period, the satellite will conduct a number of tests of the efficacy and performance of the propulsion system validating its potential use for orbit maintenance of future spacecraft. During this period, it is predominantly in a controlled attitude state, with a low drag area (discussed in detail in Section 2). At the end of life and if technically feasible, the propulsion system may undergo a final test to demonstrate its ability to accelerate the period toward re-entry into the atmosphere. If accelerated re-entry is unsuccessful, the satellite will assume a tumbling attitude state, like a regular Dove without propulsion, which greatly increases its average drag area, and results in an accelerated orbital decay.

ODAR Summary:

No debris released in normal operations; no credible scenario for breakups; the collision probability with other objects is compliant with NASA standards; and the estimated nominal decay lifetime due to atmospheric drag is well under 25 years following operations (as calculated by DAS 2.0.2 and STK10) under all cases. None of the components of Dove Turbo will survive re-entry and, accordingly, pose no risk for human casualty.

Mission duration:

Nominal mission duration is 2 years, with a maximum of 3 years.

Launch and deployment profile, including all parking, transfer, and operational orbits with apogee, perigee, and inclination:

The Dove Turbo satellite will be launched into an orbit with a semi-major axis of 6,876km (505 km above sea level), with a 97.4 deg (SSO) inclination, although launch opportunities not exceeding 550 km altitude may also be available. There is no parking or transfer orbits.

ODAR Section 2: Spacecraft Description

Physical description of the spacecraft:

The Dove satellite is a variant of the 3U CubeSat specification, with a launch mass of 5.0 kg. Basic physical dimensions are 120mm x 120mm x 380mm, with two 260mm x 300mm deployable solar arrays. The solar arrays are spring-loaded and deployed after ejection into orbit.

Power storage is provided by 6 18650 Lithium-Ion cells. The batteries will be recharged by solar cells mounted on the body of the satellite and on the two deployable solar panels.

Total satellite mass at launch, including all propellants and fluids: 5.0 kg.

Dry mass of satellites at launch, excluding solid rocket motor propellants: 5.0 kg

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

Electric propulsion, particularly Indium Field Emission Electric Propulsion. This propulsion system consist of a 0.25 kg cylinder of indium, associated heaters, ion generator mechanism, and plume neutralizer. Thrust is generated by the acceleration of indium ions via an applied electric field between and emitter crown and an extractor electrode. The magnitude to thrust generated is up to 350 micronewtons (uN).

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:

None. The propellant is solid indium which is melted and ionized electrically by energy provided by the solar arrays and batteries.

Fluids in Pressurized Batteries:

None. The satellites use unpressurized standard COTS Lithium-Ion battery cells. Each battery has a height of 65mm, a diameter of 19mm and a weight of 48.5 grams.

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

Satellite attitude is controlled by magnetorquers and reaction wheels. The nominal attitude varies between two states depending on mission mode: long axis nadir-aligned with the solar panels constrained to the orbit-plane-normal (known as “Nadir Pointing”, see Figure 1-A), and long axis velocity-aligned and the solar panels zenith constrained (known as “Low Drag”, see Figure 1-B). A third attitude state, long axis nadir-aligned and the solar panels constrained to the orbit-plane-perpendicular may also be used for collision avoidance and orbital spacing (known as “High Drag”, see Figure 1-C). The High Drag configuration is the dynamically stable orientation of the satellite. When the satellite is in an uncontrolled state, like at EOL, it begins to tumble and will eventually assume the High Drag orientation.

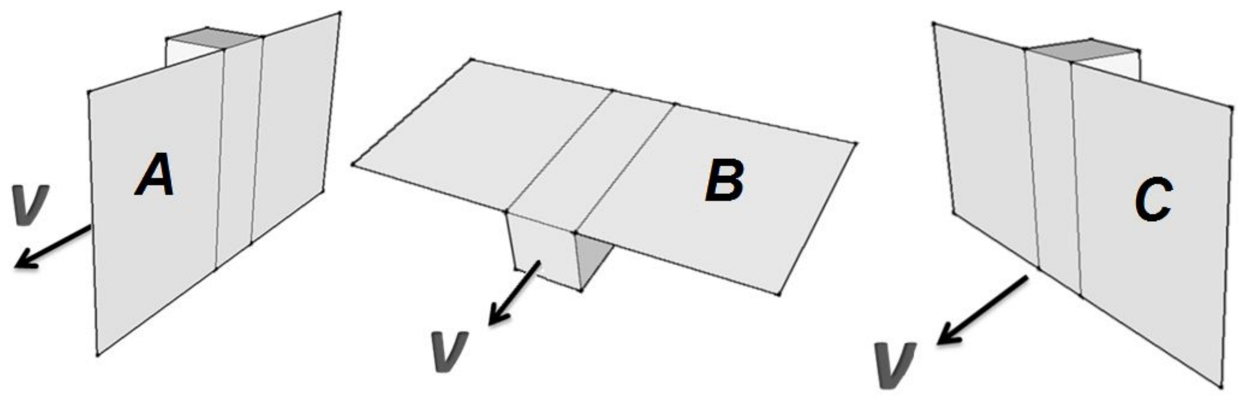


Figure 1: Attitude modes used in the Flock satellites. A –nadir pointing, B – low drag, C – high drag.

In addition, the electric propulsion system will be tested for performance and efficacy of orbit maneuvers. It is not part of the regular attitude control of the satellite but will regularly be used for the testing of orbit maintenance capability.

Description of any range safety or other pyrotechnic devices:

No pyrotechnic devices are used.

Description of the electrical generation and storage system:

Standard COTS Lithium-Ion battery cells are charged before payload integration and provide electrical energy during the mission. The cells are recharged by GaAs solar cells mounted on the deployable arrays. A battery cell protection circuit manages the charging cycle, performs battery balancing, and protects against over and undercharge conditions.

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:

There are no intentional releases.

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 v2.0.2)

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 no credible scenario that would result in spacecraft breakup during normal deployment and operations.

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

The propellant is solid indium and cannot ignite or explode from an impact or otherwise accidentally. Energy release is solely from the electrical system (batteries and solar arrays) which can only melt and ionize the indium material generating a propulsive force when passed through an appropriate electric field.

In-mission 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 explosion. The deployment of the three solar arrays will feature a simple spring and stopper system, released by a simple burn-wire. The probability of a detachment during deployment is negligible.

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:

None. The batteries will not be passivated at End of Mission due to the low risk and low impact of explosive rupturing, and the extremely short lifetime at mission conclusion. The maximum total chemical energy stored in each battery is ~41kJ.

At the start of the mission, there will be 250 grams of indium onboard which will deplete over time as the propulsion system is used. Any remaining indium material (even if all of it is remaining) has no stored energy and will burn up completely in the atmosphere during re-entry.

Rationale for all items which are required to be passivated, but cannot be due to their design:

The battery charge circuits include overcharge protection and a parallel design to limit the risk of battery failure. 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 vessel due to the lack of penetration energy.

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: 0.000.

Supporting Rationale and FMEA details:

Battery explosion:

Effect: All failure modes below might theoretically 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 the selected COTS batteries is such that while the spacecraft could be expected to vent gases, most debris from the battery rupture should be contained within the vessel due to the lack of penetration energy.
Probability: Extremely Low. It is believed to be a much less than 0.1% probability that multiple independent (not common mode) faults must occur for each failure mode to cause the ultimate effect (explosion).

Failure mode 1: Internal short circuit.
Mitigation 1: Qualification and acceptance shock, vibration, thermal cycling, and vacuum tests followed by maximum system rate-limited charge and discharge to prove that no internal short circuit sensitivity exists.
Combined faults required for realized failure: Environmental testing **AND** functional charge/discharge tests must both be ineffective in discovery of the failure mode.

Failure Mode 2: Internal thermal rise due to high load discharge rate.
Mitigation 2: Cells were tested in lab for high load discharge rates in a variety of flight-like configurations to determine like likelihood and impact of an out of control thermal rise in the cell. Cells were also tested in a hot environment to test the upper limit of the cells capability. No failures were seen.
*Combined faults required for realized failure: S*pacecraft 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 4: 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) obviation of such other mechanical failures by proto-qualification and acceptance environmental tests (shock, 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 failure must all occur to enable this failure mode.

Failure Mode 4: Inoperable vents.
Mitigation 5: The batteries used in this spacecraft are non-venting, and have already been approved for flight to the International Space Station.
Combined effects required for realized failure: The final assembler fails to install proper venting.

Failure Mode 5: Crushing.
Mitigation 6: 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 7: 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 failure modes 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 8: 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 which are 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 postmission 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 battery charge circuits include overcharge protection and a parallel design to limit the risk of battery failure. 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 vessel due to the lack of penetration energy.

The propulsion system has no stored energy and no potential to ignite. In the unlikely event a ruptured battery causes enough heat to melt the indium propellant, it will result only in liquefying the indium.

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.

ODAR Section 5: Assessment of Spacecraft Potential for On-Orbit Collisions

Assessment of spacecraft compliance with Requirements 4.5-1 and 4.5-2 (per DAS v2.0.2, 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:

Collision Probability (505 km SSO): < 1.0E-6;   COMPLIANT.

Collision Probability (550 km SSO): <1.0E-6;   COMPLIANT.

Analysis (per DAS v2.0.2):

Note this is a previous run of DAS at 500 km (substantially similar to 505 km) for a regular Dove satellite with identical physical property inputs as Dove Turbo.

07 11 2017; 08:45:40AM   **Processing Requirement 4.5-1:**   **Return Status : Passed**

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

Run Data

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

INPUT

Space Structure Name = satellite

Space Structure Type = Payload

Perigee Altitude = 500.000000 (km)

Apogee Altitude = 500.000000 (km)

Inclination = 97.300000 (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 = 2018.000000 (yr)

Initial Mass = 5.000000 (kg)

Final Mass = 5.000000 (kg)

Duration = 3.000000 (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 = 0.000000

Returned Error Message: Normal Processing

Date Range Error Message: Normal Date Range

Status = Pass

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

============== End of Requirement 4.5-1 ===============

Note this is re-run of DAS at 550 km for the Dove Turbo satellite but with identical physical property inputs and results as a regular Dove.

08 08 2017; 07:26:54AM **Processing Requirement 4.5-1: Return Status : Passed**

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

INPUT

Space Structure Name = satellite
Space Structure Type = Payload
Perigee Altitude = 550.000000 (km)
Apogee Altitude = 550.000000 (km)
Inclination = 97.300000 (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 = 2017.000000 (yr)
Initial Mass = 5.000000 (kg)
Final Mass = 5.000000 (kg)
Duration = 2.000000 (yr)
Station-Kept = True
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 = 0.000001
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range
Status = Pass

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

============== End of Requirement 4.5-1 ===============

Supporting Deployment and Collision Risk Analysis:

The above collision probability is a product of NASA's DAS 2.0.2 software. This analysis was for a single satellite at a target altitude of 500 km and the highest potential altitude of 550 km SSO orbit.

In addition, as previously analyzed and submitted in previous ODAR reports to the Commission, STK's Conjunction Analysis Toolkit (STK/CAT) was used to perform a close approach/conjunction analysis for the typical Dove. The Tech Demo satellite shares the same orbital characteristics as a typical Dove and therefore the previous results apply and reiterated here.

This analysis compares the satellite's orbit against the orbits of all of the objects in the US Space Catalog (debris, satellites and human space missions, including ISS), reporting all close approaches (within 5 km). While individual collision probabilities are calculated statistically (using Patera's method), this analysis is deterministic rather than statistical and can be used as a point reference to validate the DAS results. We also specifically discuss our strategy for avoiding collisions.

Dove vs US Space Catalog

To fully understand the risk of collisions and debris generation we need to assess the additional risk to the whole NORAD catalog of space objects due to a Dove satellite. We assumed that the orbits in the US space catalog have a covariance that results in a fixed threat volume ellipsoid defined as 10km tangential (along-track), 2km cross-track and 2km normal (radial) to the trajectory. We then assume hard spheres of diameter 1m for the Flock satellites and 2m for all other objects in the catalog. This allows estimation of the probability of collisions between any Flock satellite and any existing object in the catalog. This was done for an analysis period that covered the entire orbital lifetime:

Compliance Summary:

Collision Probability (500 km, SSO): 7.44E-05; COMPLIANT.

Collision Probability (550 km, SSO): 2.88E-4; COMPLIANT.

Part 2: Flock vs Flock

To additionally reduce the risk of collisions, Planet Labs plans to perform differential drag station-keeping to maintain an even along-track spacing across the constellation.

Nominally this would result in zero probability of a collision between any Flock members. This technique has been demonstrated in space between the 11 Flock 1c satellites launched in 2014 and is operational today with Flock 2P, 3P and 2K currently operating in similar orbits.

Part 3: Effects of the Propulsion system

It is expected that the propulsion system will enable a reduction in collision risk with other satellites and objects by its use for collision avoidance maneuvers. However this is a factor of the testing and demonstration and a reliable estimate for the reduction of collision risk is not yet possible. Therefore we present the collision probability above as the worse case scenario as if the propulsion system becomes non-functional. As testing is completed, a better understanding will be achieved of its usefulness for collision avoidance maneuvers, which will inform Planet for the development of a collision avoidance process using the propulsion system.

Requirement 4.5-2: Limiting debris generated by collisions with small objects when operating in Earth or lunar orbit**:**

For each spacecraft, the program or project shall demonstrate that, during the mission of the spacecraft, the probability of accidental collision with orbital debris and meteoroids sufficient to prevent compliance with the applicable post-mission disposal requirements is less than 0.01 (Requirement 56507).

The Dove satellite is to be deployed into a very low Earth orbit. The density of resident space objects, and therefore the probability of collisions, reduces quickly with altitude below 800 km. The maximum altitude could be 550 km and therefore we perform a representative DAS analysis for this case.

Small Object Impact and Debris Generation Probability:
Probability of Post-Mission Disposal Failure (550 km): 0.0001;   COMPLIANT.

Analysis (per DAS v2.0.2):

Note this is a previous run of DAS at 500 km (substantially similar to 505 km) for a regular Dove satellite with identical physical property inputs as Dove Turbo.

07 11 2017; 08:49:02AM   **Requirement 4.5-2: Compliant**

==

Spacecraft = satellite
Critical Surface = Wall X
==

INPUT

Apogee Altitude = 500.000000 (km)
Perigee Altitude = 500.000000 (km)
Orbital Inclination = 97.300000 (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 = 5.000000 (kg)
Final Mass = 5.000000 (kg)
Station Kept = No
Start Year = 2018.000000 (yr)
Duration = 3.000000 (yr)
Orientation = Fixed Oriented
CS Areal Density = 2.700000 (g/cm^2)
CS Surface Area = 0.030000 (m^2)
Vector = (0.000000 (u), 1.000000 (v), 0.000000 (w))
CS Pressurized = No

OUTPUT

Probabilty of Penitration = 0.000846
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range

==
Spacecraft = satellite
Critical Surface = Wall Y
==

INPUT

Apogee Altitude = 500.000000 (km)
Perigee Altitude = 500.000000 (km)
Orbital Inclination = 97.300000 (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 = 5.000000 (kg)
Final Mass = 5.000000 (kg)
Station Kept = No
Start Year = 2018.000000 (yr)
Duration = 3.000000 (yr)
Orientation = Fixed Oriented
CS Areal Density = 2.700000 (g/cm^2)

CS Surface Area = 0.190000 (m^2)
Vector = (0.000000 (u), 0.000000 (v), 1.000000 (w))
CS Pressurized = No

OUTPUT

Probabilty of Penitration = 0.000853
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range

===
Spacecraft = satellite
Critical Surface = Wall Z
===

INPUT

Apogee Altitude = 500.000000 (km)
Perigee Altitude = 500.000000 (km)
Orbital Inclination = 97.300000 (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 = 5.000000 (kg)
Final Mass = 5.000000 (kg)
Station Kept = No
Start Year = 2018.000000 (yr)
Duration = 3.000000 (yr)
Orientation = Fixed Oriented
CS Areal Density = 2.700000 (g/cm^2)
CS Surface Area = 0.010000 (m^2)
Vector = (1.000000 (u), 0.000000 (v), 0.000000 (w))
CS Pressurized = No

OUTPUT

Probabilty of Penitration = 0.000000
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range

Note this is re-run of DAS at 550 km for the Dove Turbo satellite but with identical physical property inputs and results as a regular Dove.

08 08 2017; 07:28:15AM **Requirement 4.5-2: Compliant**

===
Spacecraft = satellite
Critical Surface = Wall X

===

INPUT

Apogee Altitude = 550.000000 (km)
Perigee Altitude = 550.000000 (km)
Orbital Inclination = 97.300000 (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 = 5.000000 (kg)
Final Mass = 5.000000 (kg)
Station Kept = Yes
Start Year = 2017.000000 (yr)
Duration = 2.000000 (yr)
Orientation = Fixed Oriented
CS Areal Density = 2.700000 (g/cm^2)
CS Surface Area = 0.030000 (m^2)
Vector = (0.000000 (u), 1.000000 (v), 0.000000 (w))
CS Pressurized = No
Outer Wall 1 Density: 2.700000 (g/cm^2) Separation: 0.200000 (cm)

OUTPUT

Probabilty of Penitration = 0.000040
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range

===
Spacecraft = satellite
Critical Surface = Wall Y
===

INPUT

Apogee Altitude = 550.000000 (km)
Perigee Altitude = 550.000000 (km)
Orbital Inclination = 97.300000 (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 = 5.000000 (kg)
Final Mass = 5.000000 (kg)
Station Kept = Yes
Start Year = 2017.000000 (yr)
Duration = 2.000000 (yr)
Orientation = Fixed Oriented
CS Areal Density = 2.700000 (g/cm^2)
CS Surface Area = 0.190000 (m^2)
Vector = (0.000000 (u), 0.000000 (v), 1.000000 (w))
CS Pressurized = No
Outer Wall 1 Density: 2.700000 (g/cm^2) Separation: 0.200000 (cm)

OUTPUT

Probabilty of Penitration = 0.000023
Returned Error Message: Normal Processing

Date Range Error Message: Normal Date Range

```
==================================================
Spacecraft = satellite
Critical Surface = Wall Z
==================================================

**INPUT**

        Apogee Altitude = 550.000000 (km)
        Perigee Altitude = 550.000000 (km)
        Orbital Inclination = 97.300000 (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 = 5.000000 (kg)
        Final Mass = 5.000000 (kg)
        Station Kept = Yes
        Start Year = 2017.000000 (yr)
        Duration = 2.000000 (yr)
        Orientation = Fixed Oriented
        CS Areal Density = 2.700000 (g/cm^2)
        CS Surface Area = 0.010000 (m^2)
        Vector = (1.000000 (u), 0.000000 (v), 0.000000 (w))
        CS Pressurized = No
        Outer Wall 1   Density: 2.700000 (g/cm^2)  Separation: 0.200000 (cm)

**OUTPUT**

        Probabilty of Penitration = 0.000000
        Returned Error Message: Normal Processing
        Date Range Error Message: Normal Date Range

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

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

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

If possible, we will attempt to de-orbit the satellite at a faster rate using any remaining propellant and capability of the electric propulsion system, however de-orbit within the estimated lifetime presented in this report is not dependent on a functional propulsion system. If successful, it is estimated an accelerated de-orbit could be completed in much less than one year from its nominal orbit altitude.

ODAR Section 6: Assessment of Spacecraft Post-mission Disposal Plans and Procedures

6.1 Description of spacecraft disposal option selected:

The satellite will, at minimum, deorbit naturally by atmospheric re-entry.

6.2 Plan for any spacecraft maneuvers required to accomplish postmission disposal:

Once the satellites end their life, they begin to tumble and assume a drag area greater than during operations and thus ensuring rapid orbital decay. If possible, any remaining operational capability of the propulsion system will be used to accelerate the rate of decay, but de-orbit within the estimated lifetime presented in this report is not dependent on a functional propulsion system.

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

Spacecraft Mass: 5.0 kg

Cross-sectional Area: Nadir pointing configuration: 0.033 m^2 (drag area)

Low Drag configuration: 0.013 m^2 (drag area)

High Drag configuration: 0.19 m^2 (drag area)

Area to mass ratio: Nadir pointing configuration: 0.0066 m^2/kg

Low Drag configuration: 0.0026 m^2/kg

High Drag configuration: 0.038 m^2/kg

The High Drag configuration is the dynamically stable orientation of the satellite. In the case of loss of control, the satellite will tumble until naturally assuming the High Drag configuration and begin to de-orbit close to the fastest rate possible. On-orbit results on previous missions have confirmed that the average drag area for a tumbling Flock satellite is in the range 0.15 to 0.20 m^2.

6.4 Assessment of spacecraft compliance with Requirements 4.6-1 through 4.6-5 (per DAS v 2.0.2 and NASA-STD-8719.14 section):

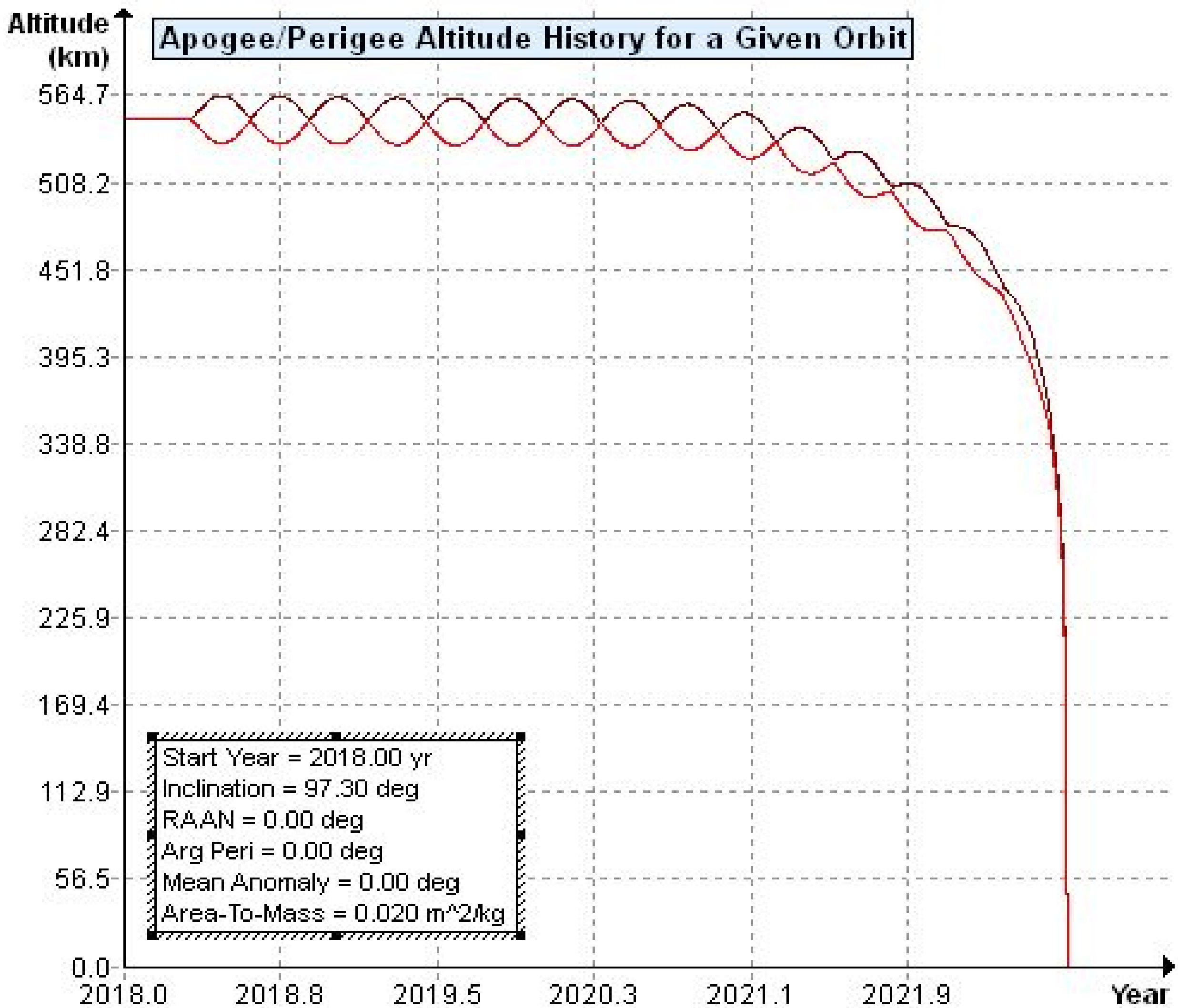
Requirement 4.6-1: Disposal for space structures passing through LEO:

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

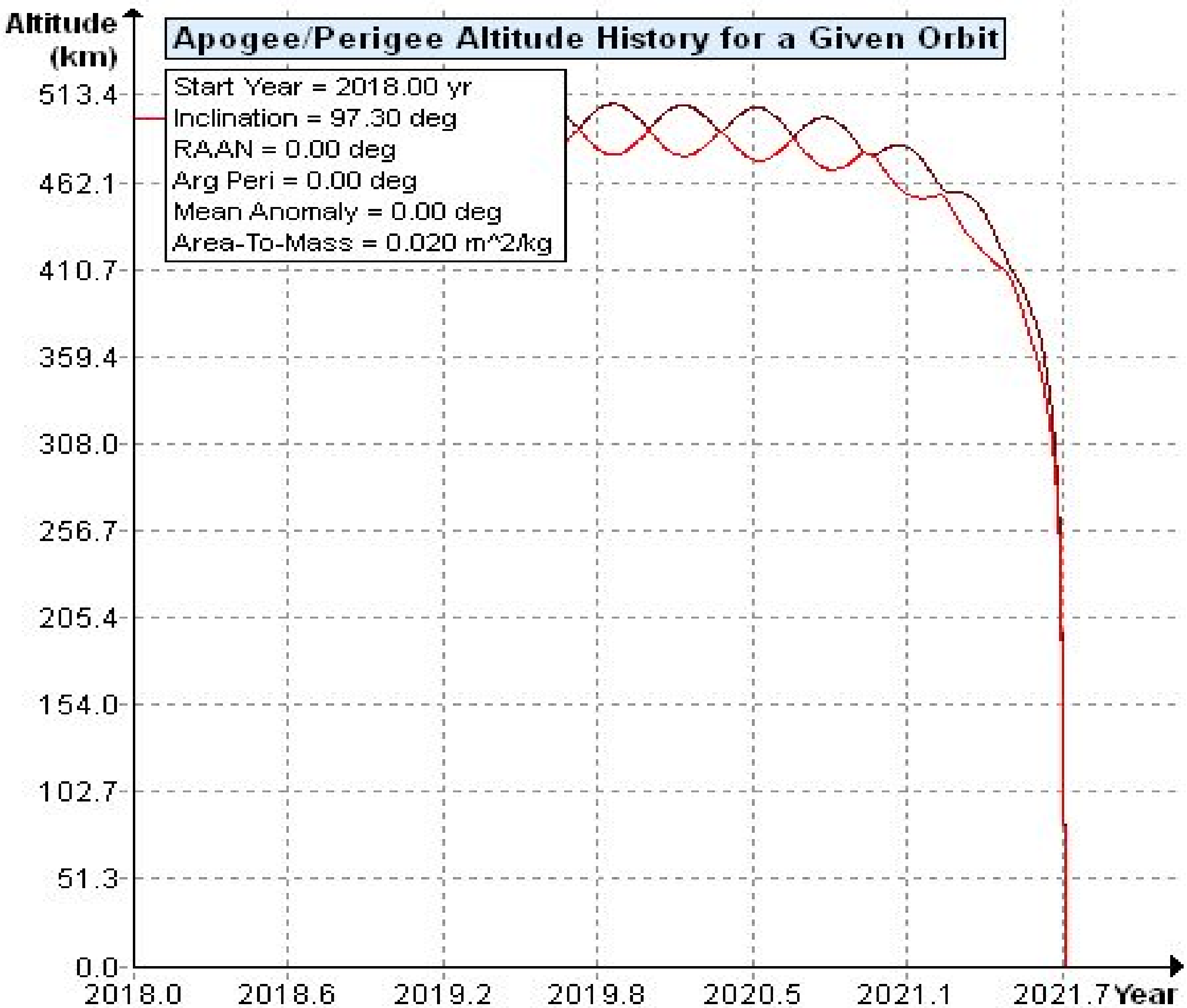
As first presented in previous ODAR reports to the commission, we re-iterate the results for two lifetime cases to represent satisfaction of this requirement. The first represents a worst case altitude of 550 km circular SSO, where the satellite operates for 2 years before assuming an uncontrolled state, and eventually high drag, until atmospheric reentry. The second case gives the lifetime of a satellite from the target 500 km circular SSO orbit which holds the operational attitude for its whole orbital lifetime.



Maximum Orbit

BOL Orbit: 550 × 550 km SSO
Mission duration: 3 years
EOM Orbit: 516 × 516 km SSO
Orbit Lifetime: 4.9 years

Note that an STK analysis provides a more conservative orbit lifetime estimate of 7.8 +/- 0.5 years based on expected solar activity conditions over the period.



Nominal Orbit

BOL Orbit: 500 × 500 km SSO
Mission duration: 3 years
EOM Orbit: 407 × 407 km SSO
Orbit Lifetime: 3.7 years

Note that an STK analysis provides a more conservative orbit lifetime estimate of 5.4 +/- 0.2 years based on expected solar activity conditions over the period.

DAS Analysis: The Flock satellite reentry is COMPLIANT using method “a” for all orbits within the proposed range of this application.

Note this is a previous run of DAS at 500 km (substantially similar to 505 km) for a regular Dove satellite with identical physical property inputs as Dove Turbo.

07 11 2017; 10:49:42AM Science and Engineering - Orbit Lifetime/Dwell Time

INPUT

Start Year = 2018.000000 (yr)
Perigee Altitude = 500.000000 (km)
Apogee Altitude = 500.000000 (km)
Inclination = 93.700000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Area-To-Mass Ratio = 0.020000 (m^2/kg)

OUTPUT

Orbital Lifetime from Startyr = 3.690623 (yr)
Time Spent in LEO during Lifetime = 3.690623 (yr)
Last year of Propagation = 2021 (yr)
Returned Error Message: Object reentered

Note this is re-run of DAS at 550 km for the Dove Turbo satellite but with identical physical property inputs as a regular Dove.

08 08 2017; 07:33:27AM **Processing Requirement 4.6 Return Status : Passed**

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

INPUT

Space Structure Name = satellite
Space Structure Type = Payload

Perigee Altitude = 550.000000 (km)
Apogee Altitude = 550.000000 (km)
Inclination = 97.300000 (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 = 2017.000000 (yr)
Initial Mass = 5.000000 (kg)
Final Mass = 5.000000 (kg)
Duration = 2.000000 (yr)
Station Kept = True

Abandoned = True
PMD Perigee Altitude = 550.000000 (km)
PMD Apogee Altitude = 550.000000 (km)
PMD Inclination = 97.300000 (deg)
PMD RAAN = 0.000000 (deg)
PMD Argument of Perigee = 0.000000 (deg)
PMD Mean Anomaly = 0.000000 (deg)

OUTPUT

Suggested Perigee Altitude = 550.000000 (km)
Suggested Apogee Altitude = 550.000000 (km)
Returned Error Message = Passes LEO reentry orbit criteria.

Released Year = 2026 (yr)
Requirement = 61
Compliance Status = Pass

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

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

Requirement 4.6-2. Disposal for space structures near GEO.

Analysis: Not applicable.

***Requirement 4.6-3. Disposal for space structures between LEO and GEO*.**

Analysis: Not applicable.

Requirement 4.6-4. Reliability of Postmission Disposal Operations

Analysis: Not applicable.

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).*

DAS shows that no component of the satellite is expected to reach the Earth's surface. All components are expected to completely burn up during re-entry. The Probability of risk to Human casualty as reported by DAS is zero (0), which is within the requirements specified above. COMPLIANT.

Analysis (per DAS v2.0.2):

08 14 2017; 14:07:59PM **Processing Requirement 4.7-1 Return Status : Passed**

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

name = satellite
quantity = 1
parent = 0
materialID = 5
type = Box
Aero Mass = 5.000000
Thermal Mass = 5.000000
Diameter/Width = 0.100000
Length = 0.410000
Height = 0.100000

name = Propulsion
quantity = 1
parent = 1
materialID = 5
type = Cylinder
Aero Mass = 0.800000
Thermal Mass = 0.800000
Diameter/Width = 0.090000
Length = 0.080000

name = Batteries
quantity = 4
parent = 1
materialID = 54
type = Cylinder
Aero Mass = 0.046000
Thermal Mass = 0.046000
Diameter/Width = 0.018500
Length = 0.065300

name = Structure1
quantity = 1
parent = 1
materialID = 5
type = Box
Aero Mass = 0.221000
Thermal Mass = 0.221000
Diameter/Width = 0.097000

Length = 0.231000
Height = 0.048500

name = Brackets
quantity = 2
parent = 1
materialID = 5
type = Box
Aero Mass = 0.040000
Thermal Mass = 0.040000
Diameter/Width = 0.096500
Length = 0.121000
Height = 0.019750

name = Solar Panels
quantity = 8
parent = 1
materialID = 24
type = Flat Plate
Aero Mass = 0.050000
Thermal Mass = 0.050000
Diameter/Width = 0.080000
Length = 0.300000

name = Radiator
quantity = 4
parent = 1
materialID = 5
type = Flat Plate
Aero Mass = 0.229000
Thermal Mass = 0.229000
Diameter/Width = 0.099000
Length = 0.358000

name = Avionics
quantity = 1
parent = 1
materialID = 23
type = Box
Aero Mass = 0.400000
Thermal Mass = 0.400000
Diameter/Width = 0.100000
Length = 0.100000
Height = 0.100000

name = Sinclair Wheels
quantity = 4
parent = 1
materialID = 5
type = Box
Aero Mass = 0.131250
Thermal Mass = 0.040000
Diameter/Width = 0.033500
Length = 0.033500
Height = 0.017000

name = Lead Mass
quantity = 1
parent = 9

materialID = 39
type = Box
Aero Mass = 0.365000
Thermal Mass = 0.365000
Diameter/Width = 0.038000
Length = 0.060000
Height = 0.015000

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

name = satellite
Demise Altitude = 77.997887
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Propulsion
Demise Altitude = 67.869925
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Batteries
Demise Altitude = 72.557808
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Structure1
Demise Altitude = 76.544879
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Brackets
Demise Altitude = 77.400824
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Solar Panels
Demise Altitude = 77.853129
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Radiator
Demise Altitude = 76.216652
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Avionics
Demise Altitude = 75.638254
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****

name = Sinclair Wheels
Demise Altitude = 75.212371
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Lead Mass
Demise Altitude = 74.361410
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****

=============== End of Requirement 4.7-1 ===============

Requirements 4.7-1b, and 4.7-1c are non-applicable requirements because the satellites are not baselined to perform controlled reentry.

ODAR Section 8: Assessment for Tether Missions

Not applicable. There are no tethers in the mission.

END of ODAR