EXHIBIT D

Pelican Orbital Debris Assessment Report (ODAR)

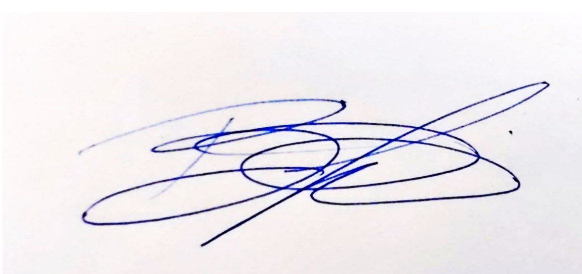
This report is presented in compliance with NASA-STD-8719.14C, APPENDIX A

Report Version: 5.0

April 21, 2022

DAS Software Version Used in Analysis: v3.2.1

SCARAB Software Version Used in Analysis; V3.1L



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Brian Lewis
Senior Director, Systems Engineering
Planet Labs, PBC

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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: Pelican

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:

This ODAR follows the format recommended in NASA-STD-8719.14C, Appendix A.1 and includes the content indicated at a minimum in each section 2 through 8 below for the Pelican spacecraft. 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 PBC (Planet)

Foreign government or space agency participation: None

Initial Launch: Expected early 2023

Mission Overview:
The first Pelican satellite will be launched into a maximum 550 km (525 +/- 25 km) circular sun-synchronous orbit as a secondary payload. Subsequent Pelican launches are expected to have injection orbits either with similar or lower insertion orbits. Accordingly, for purposes of this ODAR, Planet is assuming a worst-case injection altitude of 550 km in assessing collision probability, orbital lifetime, and casualty probability.

After launch, the Pelican spacecraft will be commissioned for a brief period. During this period, 3-axis control and transmitters will be enabled and the solar arrays will be deployed.

Subsequently, the satellite will use onboard propulsion to maneuver to a target, operational 325 km circular altitude. At the end of life, the satellite will enter a coarsely controlled attitude state and re-enter the atmosphere under the influence of natural orbital decay.

Planet plans to launch a fleet of 32 simultaneously operating Pelican spacecraft, the first seven in sun-synchronous orbits and the remaining 25 in inclined Walker constellations. The planned distribution of orbital inclinations is shown in Table 1. As each satellite reaches its end of life, Planet will replace the satellite. Over a fifteen-year license term, Planet would expect to launch up to 96 Pelican satellites in total.

Table 1 - Planned orbital inclinations for Pelican deployments

Range of insertion altitude (km)	Nominal operational altitude (km)	Inclination (°)	Number of Spacecraft
250-550	325[1]	97	7
250-550	325	53	25

ODAR Summary: No debris is released in normal operations; no credible scenario for breakups; the collision probability with other objects is compliant with NASA standards;

[1] The first satellite will be commissioned at 525 km and then enter a low thrust spiral from its injection orbit to its operational altitude of 325 km +/- 25 km, which will take approximately six to twelve months.

and the estimated nominal decay lifetime due to atmospheric drag is under 25 years following operations (as calculated by DAS 3.2.1) under all cases.

Mission duration:

Nominal mission duration per spacecraft is 5 years.

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

Due to a variety of launch options currently under consideration, a range of initial insertion orbits is presented in this analysis. A maximum semi-major axis of 6928 km (550 km above sea level) in either a Sun-synchronous orbit (97°) or medium inclination orbit (53°) is considered for all analyses. The satellites will transfer from the insertion orbit to the nominal 325 km circular operational attitude using onboard electric propulsion. There are no parking orbits.

ODAR Section 2: Spacecraft Description

Physical description of the spacecraft:

The Pelican satellite is an optical Earth imaging smallsat with a launch mass of no greater than 160 kg. Basic physical dimensions are 2300 mm x 1000 mm x 600 mm in the deployed configuration. There are two deployable solar arrays, each with dimensions 1000 mm x 850 mm.

Total satellite mass at launch, including all propellants and fluids: 160 kg

Dry mass of satellite at launch, excluding solid rocket motor propellants: 146 kg

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

Electric propulsion, particularly, Hall Effect Thruster with xenon propellant.

Identification, including mass and pressure, of all fluids (liquids and gasses) planned to be on board and a description of the fluid loading plan or strategies, excluding fluids in sealed heat pipes:

Compressed xenon propellant maintained in a supercritical gaseous state using heaters. 14.5 kg of xenon propellant stored in two tanks that are pressurized to 205 atmospheres (absolute). On-base loading or unloading operations are planned. Discussions with range safety for major US and foreign launch sites are underway with no major blockers anticipated at time of filing.

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 mass of 49 grams.

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

The satellite's attitude control system consists of magnetorquers, reaction wheels, star cameras, horizon sensors, magnetometers, photodiodes, and rate gyros. The nominal imaging attitude will be nadir +/- 30 degrees about the X and Y axes.

Description of any range safety or other pyrotechnic devices:

No pyrotechnic devices are used.

Description of the electrical generation and storage system:

Lithium-Ion battery cells are charged before payload integration and provide electrical energy during the mission. The cells are recharged by InGaAs solar cells mounted on the deployable and body mounted 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:

Two deployable solar arrays with small mechanical springs. Actively restrained at launch using a mechanical hold-down actuator.

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 v3.2.1)

4.3-1, Mission Related Debris Passing Through LEO:

COMPLIANT

4.3-2, Mission Related Debris Passing Near GEO:

COMPLIANT

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

Potential causes of spacecraft breakup during deployment and mission operations:

There is 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:

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 solar arrays will feature a simple spring and burn wire system with deployment commanded after launch during the commissioning period. The probability of detachment 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:

The batteries will be passivated at End of Mission by sending a command to disable battery charging.

At the start of the mission, there will be 14.5 kg of xenon onboard which will deplete over time as the propulsion system is used to maintain altitude. Any remaining xenon has minimal stored energy.

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 gasses, 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 gasses, 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, performed by both the vendor and Planet.
Combined faults required for realized failure: Environmental testing **AND** functional charge/discharge tests must both be ineffective to realize this failure mode.

Failure Mode 2: Internal thermal rise due to high load discharge rate.
Mitigation 2: Cells are tested in the lab for high-load discharge rates in a variety of flight-like configurations to determine the likelihood and impact of an out of control thermal rise in the cell. Cells are also tested in a hot environment to test the upper limit of the cells' capability. Cell selection and pack thermal design avoids potential failure from thermal rise. Cells are similarly tested by the vendor.
Combined faults required for realized failure: Battery pack thermal design must be incorrect **AND** spacecraft thermal design must be incorrect **AND** external over-current detection and disconnect function must all fail to realize 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) 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 realize this failure mode.

Failure Mode 4: Inoperable vents.
Mitigation 4: Internal battery vents are not blocked or inhibited by the battery pack design or the spacecraft.
Combined effects required for realized failure: Vendor vents fail to burst **AND** battery assembler fails to manufacture battery per the design in such a way that the vents are blocked.

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 failure modes in environmental tests must occur to realize 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 will be analyzed in combination with space environment temperatures to show that batteries do not exceed normal allowable operating temperatures and 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 to realize this failure mode.

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 cannot 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 gasses, most debris from the battery rupture should be contained within the vessel due to the lack of penetration energy. The batteries will be passivated at the end of operations by transmission of commands that disable battery charging.

The propulsion is an electric ion system with a gaseous propellant that ionizes only upon application of high voltage electrical power. In an inert state, there is no plausible mechanism for an uncontrolled release of energy and no passivation method is required. Planet plans, however, to fully deplete the propellant tanks before end of mission.

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 v3.2.1, 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, 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. For the purpose of this assessment, 100 years is used as the maximum orbital lifetime for the storage disposal option. (Requirement 56506).

Large Object Impact and Debris Generation Probability:
Individual Collision Probability
The collision probability (P_c) for the first satellite in its injection orbit is calculated to be (550 km SSO): 5.7667E-05; COMPLIANT. This collision probability is calculated using NASA's DAS 3.2.1 software.

For the projected launches shown in Table 1, Table 2 shows the collision probabilities in the relevant transfer and operational orbits. Each Pelican satellite will fly in its operational orbit of 325 km x 325 km for around 4.75 years of its 5 year lifetime. However, it will spend around 3 months in a low thrust spiral from its injection orbit to its operational orbit.[2] For compliance with most of the FCC rules, this orbit is inconsequential. However, in assessing the probability of on-orbit collision (Requirement 4.5-1), this transfer orbit contributes a greater probability of collision than that of the operational orbit. Planet has evaluated the collision probability risk for the transfer orbit and operational orbits separately and provides a joint collision probability as shown in Table 2. The DAS logs in the Appendix show a 325 km x 550 km transfer orbit for analysis as a result of the limitations of the DAS software. Planet calculates the probability of collision in its combined transfer and operational orbits as shown below. Values for these probabilities are calculated using DAS and are shown in Table 2 and in the logs in Appendices.

$$P_{collision} = 1 - (1 - P_{transfer}) \cdot (1 - P_{operations})$$

[2] As discussed in the Technical Appendix, the first Pelican satellite will have an extended commissioning and transfer period of approximately six to twelve months. *See* Technical Appendix at 1-2.

Table 2 - Calculated collision probability for Pelican spacecraft in their transfer and operational orbits

Orbit Inclination (°)	P_c	$P_{transfer}$	$P_{operations}$
97	7.01E-06	3.83E-06	3.18E-06
53	3.58E-06	1.88E-06	1.70E-06

Note that these collision probabilities assume no collision avoidance maneuvers. However, collision avoidance will be a nominal part of operations for the 5 year mission duration. The probability of collision for each satellite should be treated as zero when accounting for the propulsion system, as determined by the FCC.[3] Planet expects to maintain the apogee/perigee of the Pelican satellites within +/- 25 km and the inclination within +/- 1°. The right ascension of the ascending node(s) (RAAN) for the SSO satellites will depend on the launch and will precess at a constant rate of approximately 1°/day to maintain sun synchronous orbit. RAAN for the inclined Pelican spacecraft will drift as the orbits precess.

Cumulative Collision Probability

Although not required under the FCC rules, Planet has calculated a cumulative probability of collision from this proposed constellation using the equation shown below.

$$P_{cum} = 1 - (1 - P_{col\,SSO})^{N_{SSO}} * (1 - P_{col\,53})^{N_{53}}$$

Using the distribution of spacecraft from Table 1 with the collision projections from Table 2, Planet calculates a cumulative collision probability of 1.39E-04 or 1:7,212 for 32 operational satellites. Planet also ran the analysis for the theoretical 96 satellites needed to replenish the constellation over 15 years, resulting in a calculated collision risk of 4.09E-04 or 1:2,445. Thus, the 96 satellites over the lifetime of the license meets the 1:1,000 threshold.

See DAS Analysis log at the end of this ODAR

Supporting Deployment and Collision Risk Analysis

The above collision probability is calculated using NASA's DAS 3.2.1 software.

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

[3] *See Mitigation of Orbital Debris in the New Space Age*, Report and Order and Further Notice of Proposed Rulemaking, 35 FCC Rcd. 4156 ¶ 35 (2020); *see also* 85 Fed. Reg. 52,455 (Aug. 25, 2020).

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 majority of the Pelican satellites will be deployed into very low Earth orbits, and all Pelican satellites will be operated at very low altitudes where the density of resident space objects is low, and therefore the probability of collisions, is low. Our expected insertion altitude range is 250-550 km, and the nominal operational orbit is 325 km (well below the International Space Station orbit).

Small debris penetrations will not cause the satellite to break up but may cause the loss of function of the satellite. From an initial injection altitude of 550 km, the Pelican spacecraft will re-enter in ~12 years from an inert, uncontrolled state. From the Pelican operational orbit at 325 km, the Pelican spacecraft will re-enter in less than 3 months from an inert, uncontrolled state. Figures 1 through 4 demonstrate the evolution of the spacecraft altitude in such a state. No propulsive maneuvers, passivation activities, or other active commanding or function is required to meet re-entry and disposal requirements. Therefore, the probability that an individual space station will become a source of debris by collision with small debris or meteoroids that would cause loss of control and prevent disposal is zero, which is less than the 0.01 requirement.

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

None.

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.

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

6.1 Description of spacecraft disposal option selected:

The satellite will deorbit naturally by atmospheric re-entry.

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

No maneuvers are required for post-mission disposal.

At satellite end of life, the satellite may remain in a stable attitude given its aerodynamic features designed for VLEO operations and therefore, no assumption for increased drag or tumbling through de-orbit is made. 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 (worst-case):	160 kg
Average Cross-Sectional Area:	1.126 m^2
Area to mass ratio:	0.00987 m^2/kg

6.4 Assessment of spacecraft compliance with Requirements 4.6-1 through 4.6-5 (per DAS v 3.2.1 and NASA-STD-8719.14 section):

Requirement 4.6-1: Disposal for space structures passing through LEO:
Requirement 4.6-1. Natural reentry, direct reentry, or direct retrieval shall comply with the following:
a. Natural reentry: Leave the space structure in an orbit in which, using conservative projections for solar activity, atmospheric drag will limit the orbital lifetime to as short as practicable but no more than 25 years after the completion of mission.
b. Direct reentry: Maneuver the space structure into a controlled de-orbit trajectory as soon as practical after completion of mission.
c. Direct retrieval: Retrieve the space structure and remove it from orbit preferably at completion of mission, but no more than 5 years after completion of mission

The following plots show apogee and perigee altitude histories for the proposed injection and operational orbits.

Maximum Insertion Orbit - 550 km, SSO

BOL Orbit: 550 × 550 km, 97° (SSO)
Mission duration: 5 years (*with propulsion*)
Orbit Lifetime: 12.12 years *(worst case propulsion failure)*

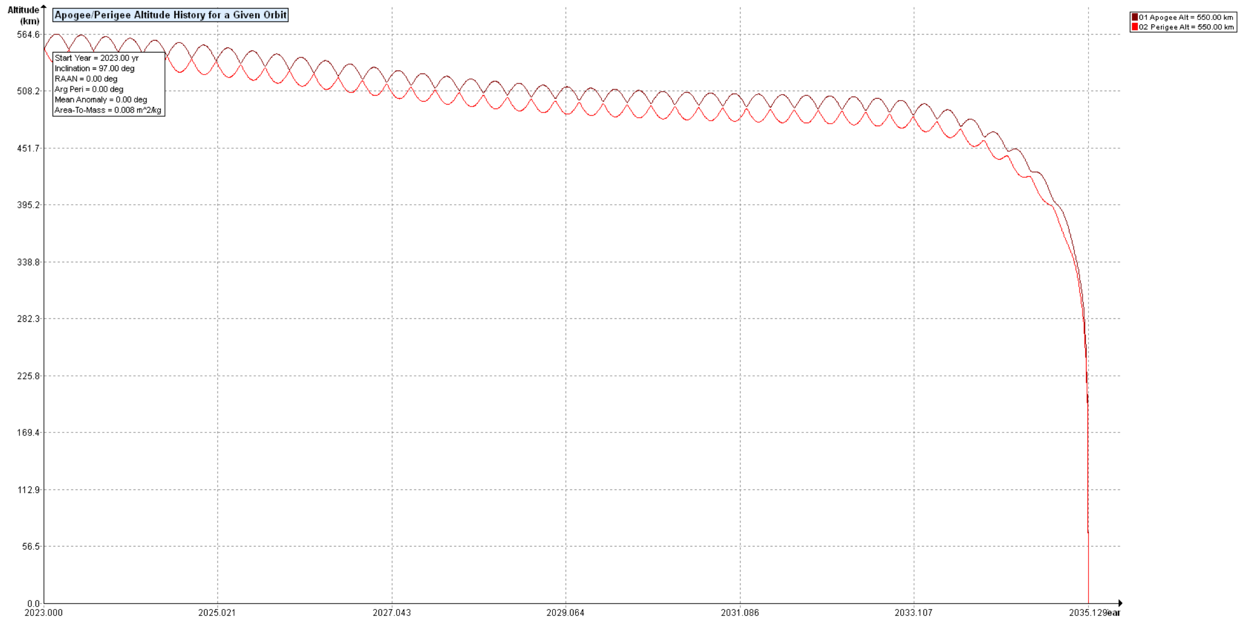


Figure 1 - Apogee and Perigee Lifetime History from Maximum Insertion Orbit, SSO

Nominal Operational Orbit - 325 km, SSO

EOL Orbit: 325 km x 325 km x 97° (SSO)
Mission Duration: 5 years (with propulsion)
Orbit Lifetime: 0.17 years (Following propellant depletion)

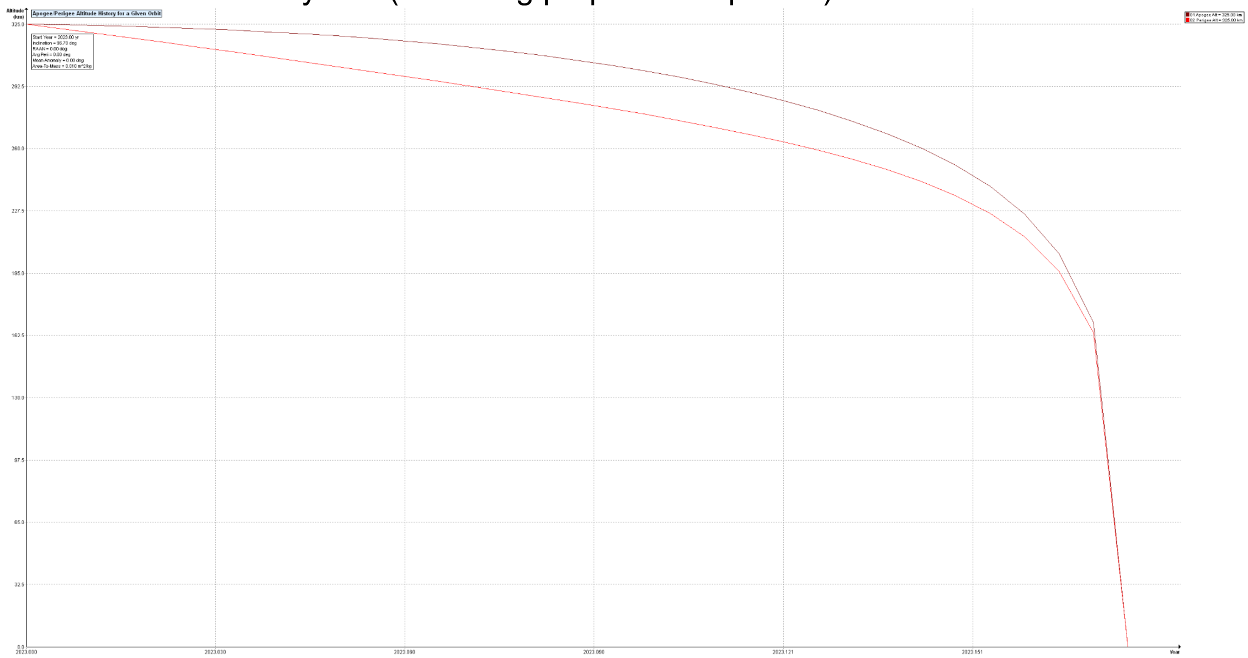


Figure 2 - Apogee and Perigee Lifetime History from Operational Orbit, SSO

<u>Maximum Insertion Orbit - 550 km, 53°</u>

BOL Orbit: 550 km x 550 km x 53°
Mission Duration: 5 years (*with propulsion*)
Orbit Lifetime: 12.33 years *(worst case propulsion failure)*

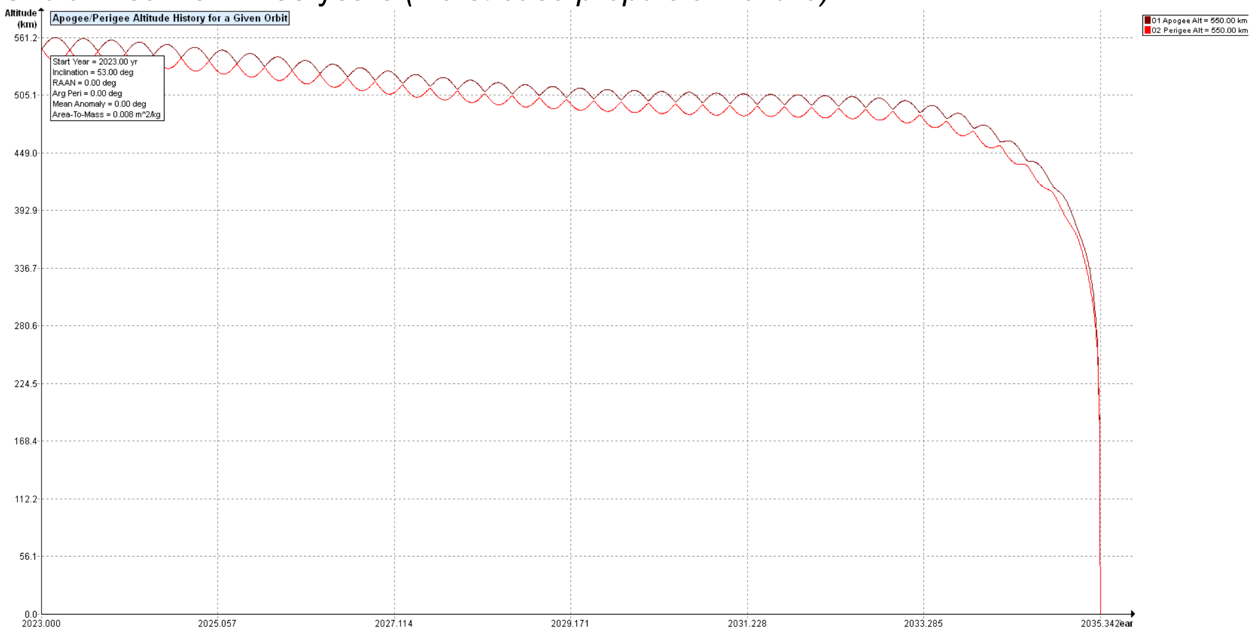


Figure 3 - Apogee and Perigee Lifetime History from Maximum Insertion Orbit, 53° Inclined

Nominal Mission Orbit - 325 km, 53°

EOL Orbit: 325 km x 325 km x 53°
Mission Duration: 5 years (with propulsion)
Orbit Lifetime: 0.18 years (Following propellant depletion)

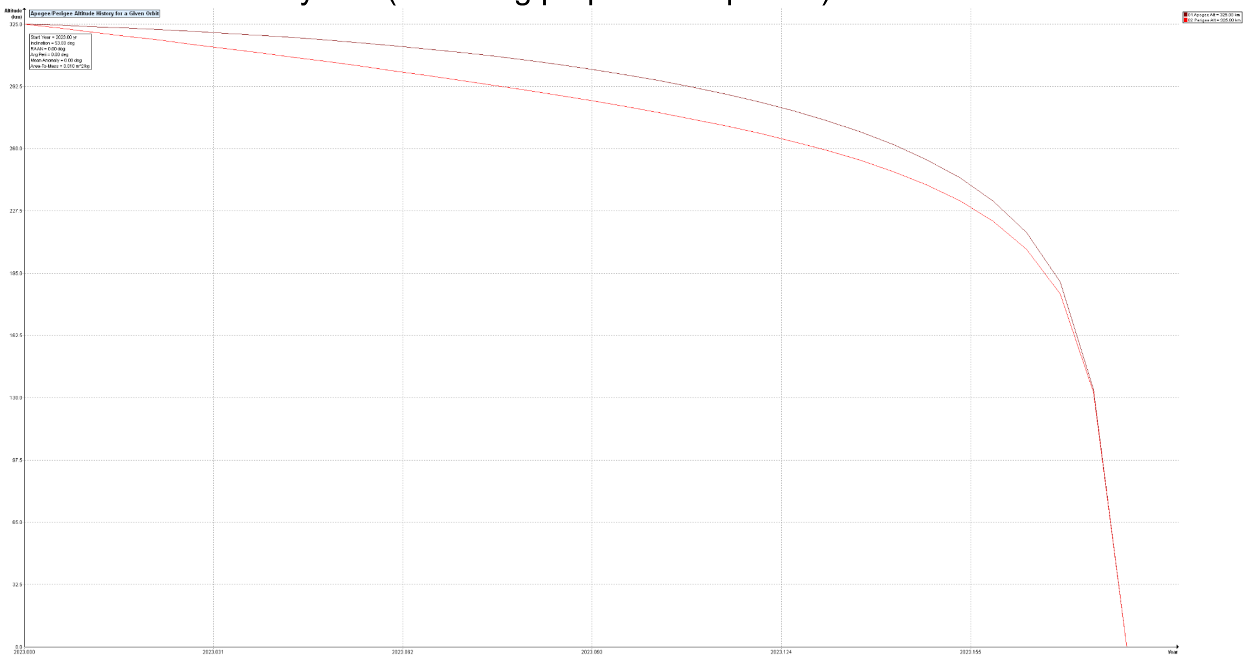


Figure 4 - Apogee and Perigee Lifetime History from Operational Orbit, 53° Inclined

DAS Analysis: The Pelican satellite re-entry is COMPLIANT for the proposed injection orbits as well as the proposed operational orbits. From Pelican's operational orbit, re-entry lifetime is projected to be less than 3 months as shown in Figures 2 and 4, meeting the intent of the NASA guideline to be as short as practicable. From an insertion orbit, the re-entry timeline is still compliant with re-entry times of ~12 years. Spacecraft failure is the only envisioned reason that a Pelican spacecraft would not be lowered to its operational orbit.

Requirement 4.6-2. Storage and Earth escape.

Analysis: Not applicable.

Requirement 4.6-3. Long-term re-entry for space structures in MEO, Tundra Orbits, highly inclined GEO, and other orbits.

Analysis: Not applicable.

Requirement 4.6-4. Reliability of Post-mission 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).*

Probability of Risk to Human Casualty:

Planet contracted with Hyperschall Technologie GMBH (HTG), the German developer and maintainer of the high fidelity Spacecraft Atmospheric Re-Entry and Aerothermal Break-Up (SCARAB) codebase used by the European Space Agency to prepare the following re-entry and casualty calculations. All results referred to in this section were generated using SCARAB version 3.1L. This analysis is a higher fidelity analysis than that generated using the NASA DAS software, as discussed below. SCARAB results are generally considered to be equivalent to results from NASA's high fidelity ORSAT modeling software.

Description of SCARAB and modeling principles

SCARAB is an ESA software tool allowing the analysis of mechanical and thermal destruction of spacecraft and other objects during re-entry (controlled or uncontrolled). It is an integrated software package (flight dynamics, aerodynamics, aerothermodynamics, thermal and structural analysis) used to perform re-entry risk assessments (quantify, characterize and monitor surviving fragments during re-entry). The software application has been validated with in-flight measurements and re-entry observations, and it has been compared to other re-entry prediction tools of the international community.

SCARAB has been developed under ESA/ESOC contracts since 1995 under the lead of HTG and with support from other European and international partners. SCARAB is considered to be operational software. The software development has evolved over time, based on lessons learned from preceding software versions, upgrades and specific requests on re-entry analyses performed for numerous satellites, and for the ATV and the Ariane 5 launcher programs.

The SCARAB model of a re-entry object primarily consists of a detailed 3D geometric model of the spacecraft with all external and internal subsystem compounds. Materials are assigned to each geometric element (linking the geometry to a temperature dependent material database), thus the geometric model provides also a mass model (i.e. total mass, center of gravity, moments of inertia) and the physical basis for all destruction processes to be computed during re-entry (e.g. melting, breaking, oxidation, bursting, explosion, etc.).

A SCARAB model is built from a limited set of geometric primitives, e.g. spheres, cylinders, cones, circular tori, boxes, flat plates (triangles, rectangles, polygons). Therefore, not every freeform surface can be modeled. In addition, not all components are modeled down to the smallest level (for example, the model does not include screws, bolts, nuts, etc.). In summary, a SCARAB model is a simplified model, similar to those numerical models which are used for structural or thermal analyses.[4]

The Fragmentation model used in the SCARAB runs is considered either state of the art with respect to the current development status of SCARAB, or as a reasonable compromise between computational effort and minimum fragment size resolution.[5]

Analysis Results

All fragments with an impact energy below 15 J are not considered to be harmful and thus are filtered. The casualty area A_c has been calculated based on the cross sections of the fragments surviving the re-entry using the following formula:

$$A_c = \sum_i \left(0.6 + \sqrt{A_i}\right)^2$$

A_i represents the mean cross-sections of the surviving fragments (arithmetic mean from three projected areas). The constant 0.6 is equal to the square root of the cross-section of an average human body (0.36 m²). This definition is equivalent to the one given in the ESA Space Debris Mitigation Handbook.

[4] The aerothermodynamics model, aerodynamics model, and structural and mechanics models used in SCARAB are described in “Re-entry Risk Assessment for Launchers Development of the new SCARAB 3.1L” by T. Lips, B. Fritsche, and M Homeister from the proceedings of the 2nd IAASS conference, ESTEC, Noordwijk, The Netherlands, 2007.

[5] Specifically, the following fragmentation settings have been used: (1) f*ragmentation at melting temperature* (two or more fragments are generated and further analyzed separately if all connecting elements have reached their material dependent melting temperature); and (2) m*inimum* f*ragment* m*ass = 30 g* (all fragments with a mass less than 30 g at the time of fragment generation are not considered because such fragments are very likely either to demise completely, or to be harmless if a fraction of such fragments survive until ground, and to limit computational effort).

Using this methodology, the SCARAB results show the calculated casualty areas of the Pelican spacecraft in the three mission orbits as shown in Table 3.

Table 3 - Calculated Casualty Areas of Pelican Spacecraft

Orbit Inclination (°)	Calculated Casualty Area (m^2)
53	1.474
97	0.683

The European Space Agency's ORIUNDO tool projects the population density for the orbit inclinations as shown in Table 3 as a function of year.

Table 4 - Projected Population Density by Year ($1/km^2$)

Year	53°	97°
2025	17.54	11.89
2030	18.17	12.37
2035	18.73	12.80
2040	19.24	13.20

The probability of a casualty can be calculated as the product of the average casualty area (Table 3) and the population density (Table 4). The resultant casualty risks are shown by year and by the Pelican inclinations as shown in Table 5.

Table 5 - Projected Casualty Risk by Year (E-05)

Year	53°	97°
2025	2.58E-05	0.81E-05
2030	2.68E-05	0.84E-05
2035	2.76E-05	0.87E-05
2040	2.84E-05	0.90E-05

As Table 5 shows, the casualty probability for each individual Pelican spacecraft launched between 2025 and 2040 meets the FCC’s 1:10,000 requirement.

Although it is not required under the FCC’s rules, Planet has calculated a cumulative probability of casualty from this proposed fleet. Planet calculates the cumulative probability using the equation shown, with the probability of casualty calculated using the year of re-entry.

$$P_{cum} = 1 - (1 - P_{cas\,SSO})^{N_{SSO}} * (1 - P_{cas\,53})^{N_{53}}$$

For the 32 spacecraft considered combining the spacecraft distribution from Table 1 and the casualty predictions from Table 5, Planet calculates a cumulative casualty probability of 7.28E-04 or 1:1,373.

Although also not required under the FCC's rules, Planet also ran the analysis for the theoretical 96 satellites needed to replenish the fleet over 15 years, which generated a casualty risk of 2.29E-03 or 1:437.

Non-demising Fragments

The SCARAB analysis showed that up to four components could survive re-entry and impact with an energy greater than 15J. Some components were consistent across all cases, while others depended upon the orbit inclination and initial orientation of the vehicle. The surviving components shown in Table 6 were used by HTG to generate the casualty areas predicted in Table 3. Survival is shown as a percentage and mean impact energy is shown as these results are generated from the multiple re-entry cases that HTG modeled.

Table 6 - Non-demising Fragments and their kinetic energy

Component	53 deg		97 deg	
	Survival %	Mean Impact Energy (J)	Survival %	Mean Impact Energy (J)
Reaction Wheel	50%	722.8	17%	238.8
Tank Port	17%	967.3	0%	N/A
Telescope Core	100%	36125.1	100%	37177.5
Telescope M2 Bracket	17%	28.9	0%	N/A

Design for Demise and Containment Considerations

Planet used some strategies in the design of the Pelican spacecraft based on published frameworks for enhancing demisability. We limited the amounts of titanium and invar that could be used in the telescope. We eliminated the use of titanium propellant tanks in favor of composite overwrapped pressure vessels with lightweight aluminum liners. Ballast and balance masses have been designed from low melt point materials.

Based on DAS and SCARAB analysis, we have identified the four specific spacecraft components shown in Table 5 that are candidates for a targeted design for demise or design for containment campaign. While the design is compliant as is, even with conservative models, this will allow the casualty risk to be further reduced in future design revisions.

- Component 1 - Reaction Wheels
    - The reaction wheels are currently placed in a low heat flow area of the bus. We will examine the use of demisable joints allowing the bus the break apart faster on re-entry to increase heat flow to the wheels at higher altitudes.
    - Planet’s vendor is providing hundreds of parts to commercial and government missions and we are supporting them to assess design for demise options including changing the flywheel material and shape along with demisable joints to open the case more rapidly.
- Component 2 - Stainless Steel Plumbing Part
    - Part is modeled to re-enter from inclined orbits, but not from sun synchronous orbits and only for certain re-entry attitudes.
    - The use of demisable joints to allow the bus to break apart faster will similarly expose this component to more regular heat flow to cover the re-entry attitudes of concern.

- Component 3 - Telescope Optical Core
  - This component has received attention for a design for containment strategy. Invar and titanium components have had bonding methods and materials selected to reduce probability of demise to allow the core to remain in one large fragment
- Component 4 - M2 and Bracket
  - Part consists of glass with invar bonded to it.
  - Design for demise will involve changing the optical material, lightweighting it or changing the geometry to provide regular heating.
  - Planet will also examine changing the material and bonding to component #3 to keep both parts together through re-entry in a design for containment strategy.

Requirements 4.7-1b, and 4.7-1c are non-applicable requirements as the Pelican satellite does not use controlled reentry.

ODAR Section 8: Assessment for Tether Missions

Not applicable. There are no tethers in the Pelican mission.

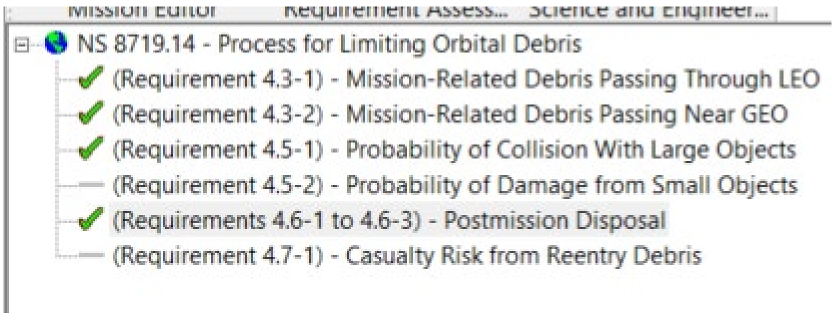
END of ODAR

(DAS Logs start next page)

APPENDIX A

<u>Sun-Synchronous Operational and Transfer Orbits</u>

325 km x 325 km x 97° (SSO) DAS GUI Results



Requirement 4.5-2 is not applicable as Pelican relies on passive atmospheric re-entry and damage from small objects will not prevent re-entry within required time durations.

Requirement 4.7-1 was evaluated using SCARAB 3.1L as described in ODAR Section 7 and no DAS logs are available for this option.

325 km x 550 km x 97° (SSO) DAS Logs (Transfer Orbit):

03 24 2022; 15:45:32PM Closed Project C:\Users\blewis\AppData\Local\NASA\DAS3.2.1\
03 24 2022; 15:46:02PM Activity Log Started
03 24 2022; 15:46:02PM Opened Project C:\Users\blewis\AppData\Local\NASA\DAS3.2.1\
03 24 2022; 15:46:16PM Processing Requirement 4.3-1: Return Status : Not Run

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

=============== End of Requirement 4.3-1 ===============
03 24 2022; 15:46:18PM Processing Requirement 4.3-2: Return Status : Passed

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

=============== End of Requirement 4.3-2 ===============
03 24 2022; 15:46:19PM Processing Requirement 4.6 Return Status : Passed

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

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

INPUT

Space Structure Name = Test
Space Structure Type = Payload

Perigee Altitude = 325.000000 (km)
Apogee Altitude = 550.000000 (km)
Inclination = 97.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Area-To-Mass Ratio = 0.009877 (m^2/kg)
Start Year = 2023.000000 (yr)
Initial Mass = 160.000000 (kg)
Final Mass = 146.000000 (kg)
Duration = 0.250000 (yr)
Station Kept = True
Abandoned = True
PMD Perigee Altitude = 325.000000 (km)
PMD Apogee Altitude = 550.000000 (km)
PMD Inclination = 97.000000 (deg)
PMD RAAN = 0.000000 (deg)
PMD Argument of Perigee = 0.000000 (deg)
PMD Mean Anomaly = 0.000000 (deg)
Long-Term Reentry = False

OUTPUT

Suggested Perigee Altitude = 325.000000 (km)
Suggested Apogee Altitude = 550.000000 (km)
Returned Error Message = Passes LEO reentry orbit criteria.

Released Year = 2024 (yr)
Requirement = 61
Compliance Status = Pass

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

============== End of Requirement 4.6 ==============
03 24 2022; 15:48:01PM Processing Requirement 4.5-1: Return Status : Passed

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

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

INPUT

Space Structure Name = Test
Space Structure Type = Payload
Perigee Altitude = 325.000 (km)
Apogee Altitude = 550.000 (km)
Inclination = 97.000 (deg)
RAAN = 0.000 (deg)
Argument of Perigee = 0.000 (deg)
Mean Anomaly = 0.000 (deg)
Final Area-To-Mass Ratio = 0.0099 (m^2/kg)
Start Year = 2023.000 (yr)
Initial Mass = 160.000 (kg)
Final Mass = 146.000 (kg)
Duration = 0.250 (yr)
Station-Kept = True
Abandoned = True
Long-Term Reentry = False

OUTPUT

Collision Probability = 3.8329E-06
Returned Message: Normal Processing
Date Range Message: Normal Date Range
Status = Pass

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

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

03 24 2022; 15:48:48PM Activity Log Started
03 24 2022; 15:48:48PM Opened Project C:\Users\blewis\AppData\Local\NASA\DAS3.2.1\
03 24 2022; 15:48:59PM Project Data Saved To File

325 km x 325 km x 97° (SSO) DAS Logs (Operational Orbit):

03 24 2022; 15:49:03PM Project Data Saved To File
03 24 2022; 15:49:04PM Project Data Saved To File

03 24 2022; 15:49:08PM   Processing Requirement 4.3-1:   Return Status : Not Run

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

=============== End of Requirement 4.3-1 ===============
03 24 2022; 15:49:10PM   Processing Requirement 4.3-2: Return Status : Passed

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

=============== End of Requirement 4.3-2 ===============
03 24 2022; 15:49:11PM   Processing Requirement 4.6 Return Status : Passed

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

INPUT

Space Structure Name = Test
Space Structure Type = Payload

Perigee Altitude = 325.000000 (km)
Apogee Altitude = 325.000000 (km)
Inclination = 97.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Area-To-Mass Ratio = 0.009877 (m^2/kg)
Start Year = 2023.000000 (yr)
Initial Mass = 160.000000 (kg)
Final Mass = 146.000000 (kg)
Duration = 4.750000 (yr)
Station Kept = True
Abandoned = True
PMD Perigee Altitude = 325.000000 (km)
PMD Apogee Altitude = 325.000000 (km)
PMD Inclination = 97.000000 (deg)
PMD RAAN = 0.000000 (deg)
PMD Argument of Perigee = 0.000000 (deg)
PMD Mean Anomaly = 0.000000 (deg)

Long-Term Reentry = False

OUTPUT

Suggested Perigee Altitude = 325.000000 (km)
Suggested Apogee Altitude = 325.000000 (km)
Returned Error Message = Passes LEO reentry orbit criteria.

Released Year = 2027 (yr)
Requirement = 61
Compliance Status = Pass

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

============== End of Requirement 4.6 ===============
03 24 2022; 15:55:22PM Processing Requirement 4.5-1: Return Status : Passed

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

INPUT

Space Structure Name = Test
Space Structure Type = Payload
Perigee Altitude = 325.000 (km)
Apogee Altitude = 325.000 (km)
Inclination = 97.000 (deg)
RAAN = 0.000 (deg)
Argument of Perigee = 0.000 (deg)
Mean Anomaly = 0.000 (deg)
Final Area-To-Mass Ratio = 0.0099 (m^2/kg)
Start Year = 2023.000 (yr)
Initial Mass = 160.000 (kg)
Final Mass = 146.000 (kg)
Duration = 4.750 (yr)
Station-Kept = True
Abandoned = True
Long-Term Reentry = False

OUTPUT

Collision Probability = 3.1761E-06
Returned Message: Normal Processing

Date Range Message: Normal Date Range
Status = Pass

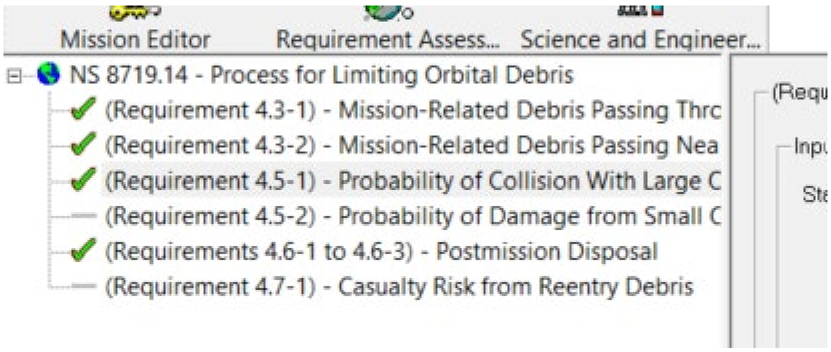
==============

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

03 24 2022; 15:56:29PM Activity Log Started
03 24 2022; 15:56:29PM Opened Project C:\Users\blewis\AppData\Local\NASA\DAS3.2.1\

Sun-synchronous Injection Orbit

550 km x 550 km x SSO DAS GUI Results



Requirement 4.5-2 is not applicable as Pelican relies on passive atmospheric re-entry and damage from small objects will not prevent re-entry within required time durations.

Requirement 4.7-1 was evaluated using SCARAB 3.1L as described in ODAR Section 7 and no DAS logs are available for this option.

550 km x 550 km x SSO DAS Logs

03 24 2022; 14:54:49PM Opened Project C:\Users\blewis\AppData\Local\NASA\DAS3.2.1\
03 24 2022; 14:55:09PM Project Data Saved To File
03 24 2022; 14:55:14PM Project Data Saved To File
03 24 2022; 14:55:14PM Mission Editor Changes Applied
03 24 2022; 14:55:14PM Project Data Saved To File
03 24 2022; 14:55:14PM Project Data Saved To File
03 24 2022; 14:55:16PM Project Data Saved To File
03 24 2022; 14:55:23PM Processing Requirement 4.3-1: Return Status : Not Run

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

=============== End of Requirement 4.3-1 ===============
03 24 2022; 14:55:26PM Processing Requirement 4.3-2: Return Status : Passed

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

=============== End of Requirement 4.3-2 ===============
03 24 2022; 14:55:27PM Processing Requirement 4.6 Return Status : Passed

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

INPUT

Space Structure Name = Test
Space Structure Type = Payload

Perigee Altitude = 550.000000 (km)
Apogee Altitude = 550.000000 (km)
Inclination = 97.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Area-To-Mass Ratio = 0.009877 (m^2/kg)
Start Year = 2023.000000 (yr)
Initial Mass = 160.000000 (kg)
Final Mass = 146.000000 (kg)
Duration = 5.000000 (yr)
Station Kept = True
Abandoned = True
PMD Perigee Altitude = 550.000000 (km)
PMD Apogee Altitude = 550.000000 (km)
PMD Inclination = 97.000000 (deg)
PMD RAAN = 0.000000 (deg)
PMD Argument of Perigee = 0.000000 (deg)
PMD Mean Anomaly = 0.000000 (deg)
Long-Term Reentry = False

OUTPUT

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

Released Year = 2036 (yr)
Requirement = 61
Compliance Status = Pass

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

============== End of Requirement 4.6 ===============
03 24 2022; 15:21:33PM Processing Requirement 4.5-1: Return Status : Passed

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

INPUT

Space Structure Name = Test
Space Structure Type = Payload
Perigee Altitude = 550.000 (km)
Apogee Altitude = 550.000 (km)
Inclination = 97.000 (deg)
RAAN = 0.000 (deg)
Argument of Perigee = 0.000 (deg)
Mean Anomaly = 0.000 (deg)
Final Area-To-Mass Ratio = 0.0099 (m^2/kg)
Start Year = 2023.000 (yr)
Initial Mass = 160.000 (kg)
Final Mass = 146.000 (kg)
Duration = 5.000 (yr)
Station-Kept = True
Abandoned = True
Long-Term Reentry = False

OUTPUT

Collision Probability = 5.7667E-05
Returned Message: Normal Processing
Date Range Message: Normal Date Range
Status = Pass

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

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

03 24 2022; 15:21:43PM   Project Data Saved To File

53° Inclined Operational and Transfer Orbits

325 km x 325 km x 53° DAS GUI Results (Operational Orbit):

- NS 8719.14 - Process for Limiting Orbital Debris
  - ✓ (Requirement 4.3-1) - Mission-Related Debris Passing Through LEO
  - ✓ (Requirement 4.3-2) - Mission-Related Debris Passing Near GEO
  - ✓ (Requirement 4.5-1) - Probability of Collision With Large Objects
  - — (Requirement 4.5-2) - Probability of Damage from Small Objects
  - ✓ (Requirements 4.6-1 to 4.6-3) - Postmission Disposal
  - — (Requirement 4.7-1) - Casualty Risk from Reentry Debris

Requirement 4.5-2 is not applicable as Pelican relies on passive atmospheric re-entry and damage from small objects will not prevent re-entry within required time durations.

Requirement 4.7-1 was evaluated using SCARAB 3.1L as described in ODAR Section 7 and no DAS logs are available for this option.

325 km x 550 km x 53° DAS Logs (Transfer Orbit):

```
03 24 2022; 16:18:41PM    Project Data Saved To File
03 24 2022; 16:18:42PM    Project Data Saved To File
03 24 2022; 16:18:45PM    Processing Requirement 4.3-1:     Return Status :  Not Run

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

=============== End of Requirement 4.3-1 ===============
03 24 2022; 16:18:46PM    Processing Requirement 4.3-2: Return Status : Passed

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

=============== End of Requirement 4.3-2 ===============
03 24 2022; 16:18:48PM    Processing Requirement 4.6  Return Status :  Passed

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

**INPUT**
```

Space Structure Name = Test
Space Structure Type = Payload

Perigee Altitude = 325.000000 (km)
Apogee Altitude = 550.000000 (km)
Inclination = 53.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Area-To-Mass Ratio = 0.009877 (m^2/kg)
Start Year = 2023.000000 (yr)
Initial Mass = 160.000000 (kg)
Final Mass = 146.000000 (kg)
Duration = 0.250000 (yr)
Station Kept = True
Abandoned = True
PMD Perigee Altitude = 325.000000 (km)
PMD Apogee Altitude = 550.000000 (km)
PMD Inclination = 53.000000 (deg)
PMD RAAN = 0.000000 (deg)
PMD Argument of Perigee = 0.000000 (deg)
PMD Mean Anomaly = 0.000000 (deg)
Long-Term Reentry = False

OUTPUT

Suggested Perigee Altitude = 325.000000 (km)
Suggested Apogee Altitude = 550.000000 (km)
Returned Error Message = Passes LEO reentry orbit criteria.

Released Year = 2024 (yr)
Requirement = 61
Compliance Status = Pass

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

=============== End of Requirement 4.6 ===============
03 24 2022; 16:20:32PM Processing Requirement 4.5-1: Return Status : Passed

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

INPUT

Space Structure Name = Test
Space Structure Type = Payload
Perigee Altitude = 325.000 (km)
Apogee Altitude = 550.000 (km)
Inclination = 53.000 (deg)
RAAN = 0.000 (deg)
Argument of Perigee = 0.000 (deg)
Mean Anomaly = 0.000 (deg)
Final Area-To-Mass Ratio = 0.0099 (m^2/kg)
Start Year = 2023.000 (yr)
Initial Mass = 160.000 (kg)
Final Mass = 146.000 (kg)
Duration = 0.250 (yr)
Station-Kept = True
Abandoned = True
Long-Term Reentry = False

OUTPUT

Collision Probability = 1.8815E-06
Returned Message: Normal Processing
Date Range Message: Normal Date Range
Status = Pass

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

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

03 24 2022; 16:20:35PM   Project Data Saved To File

325 km x 325 km x 53° DAS Logs (Operational Orbit):

03 24 2022; 16:21:12PM   Project Data Saved To File
03 24 2022; 16:21:13PM   Project Data Saved To File
03 24 2022; 16:21:16PM   Processing Requirement 4.3-1:   Return Status : Not Run

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

=============== End of Requirement 4.3-1 ===============
03 24 2022; 16:21:17PM	Processing Requirement 4.3-2: Return Status : Passed

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

=============== End of Requirement 4.3-2 ===============
03 24 2022; 16:21:18PM	Processing Requirement 4.6 Return Status : Passed

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

INPUT

Space Structure Name = Test
Space Structure Type = Payload

Perigee Altitude = 325.000000 (km)
Apogee Altitude = 325.000000 (km)
Inclination = 53.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Area-To-Mass Ratio = 0.009877 (m^2/kg)
Start Year = 2023.000000 (yr)
Initial Mass = 160.000000 (kg)
Final Mass = 146.000000 (kg)
Duration = 4.750000 (yr)
Station Kept = True
Abandoned = True
PMD Perigee Altitude = 325.000000 (km)
PMD Apogee Altitude = 325.000000 (km)
PMD Inclination = 53.000000 (deg)
PMD RAAN = 0.000000 (deg)
PMD Argument of Perigee = 0.000000 (deg)
PMD Mean Anomaly = 0.000000 (deg)
Long-Term Reentry = False

OUTPUT

Suggested Perigee Altitude = 325.000000 (km)
Suggested Apogee Altitude = 325.000000 (km)

Returned Error Message = Passes LEO reentry orbit criteria.

Released Year = 2027 (yr)
Requirement = 61
Compliance Status = Pass

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

============== End of Requirement 4.6 ==============
03 24 2022; 16:24:43PM Processing Requirement 4.5-1: Return Status : Passed

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

INPUT

Space Structure Name = Test
Space Structure Type = Payload
Perigee Altitude = 325.000 (km)
Apogee Altitude = 325.000 (km)
Inclination = 53.000 (deg)
RAAN = 0.000 (deg)
Argument of Perigee = 0.000 (deg)
Mean Anomaly = 0.000 (deg)
Final Area-To-Mass Ratio = 0.0099 (m^2/kg)
Start Year = 2023.000 (yr)
Initial Mass = 160.000 (kg)
Final Mass = 146.000 (kg)
Duration = 4.750 (yr)
Station-Kept = True
Abandoned = True
Long-Term Reentry = False

OUTPUT

Collision Probability = 1.7027E-06
Returned Message: Normal Processing
Date Range Message: Normal Date Range
Status = Pass

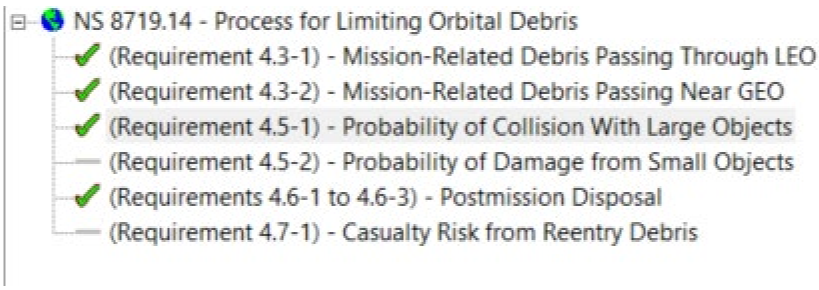
==============

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

03 24 2022; 16:24:48PM	Project Data Saved To File
03 24 2022; 16:25:29PM	Activity Log Started

53° Inclined Injection Orbits

550 km x 550 km x 53° DAS GUI Results:



Requirement 4.5-2 is not applicable as Pelican relies on passive atmospheric re-entry and damage from small objects will not prevent re-entry within required time durations.

Requirement 4.7-1 was evaluated using SCARAB 3.1L as described in ODAR Section 7 and no DAS logs are available for this option.

550 km x 550 km x 53 DAS Logs:

03 24 2022; 15:58:18PM Mission Editor Changes Applied
03 24 2022; 15:58:18PM Project Data Saved To File
03 24 2022; 15:58:18PM Project Data Saved To File
03 24 2022; 15:58:20PM Project Data Saved To File
03 24 2022; 15:58:25PM Processing Requirement 4.3-1: Return Status : Not Run

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

=============== End of Requirement 4.3-1 ===============
03 24 2022; 15:58:27PM Processing Requirement 4.3-2: Return Status : Passed

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

=============== End of Requirement 4.3-2 ===============
03 24 2022; 15:58:30PM Processing Requirement 4.6 Return Status : Passed

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

INPUT

Space Structure Name = Test
Space Structure Type = Payload

Perigee Altitude = 550.000000 (km)
Apogee Altitude = 550.000000 (km)
Inclination = 53.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Area-To-Mass Ratio = 0.009877 (m^2/kg)
Start Year = 2023.000000 (yr)
Initial Mass = 160.000000 (kg)
Final Mass = 146.000000 (kg)
Duration = 5.000000 (yr)
Station Kept = True
Abandoned = True
PMD Perigee Altitude = 550.000000 (km)
PMD Apogee Altitude = 550.000000 (km)
PMD Inclination = 53.000000 (deg)
PMD RAAN = 0.000000 (deg)
PMD Argument of Perigee = 0.000000 (deg)
PMD Mean Anomaly = 0.000000 (deg)
Long-Term Reentry = False

OUTPUT

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

Released Year = 2036 (yr)
Requirement = 61
Compliance Status = Pass

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

============== End of Requirement 4.6 ===============
03 24 2022; 16:18:02PM   Processing Requirement 4.5-1:   Return Status : Passed

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

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

INPUT

Space Structure Name = Test
Space Structure Type = Payload
Perigee Altitude = 550.000 (km)
Apogee Altitude = 550.000 (km)
Inclination = 53.000 (deg)
RAAN = 0.000 (deg)
Argument of Perigee = 0.000 (deg)
Mean Anomaly = 0.000 (deg)
Final Area-To-Mass Ratio = 0.0099 (m^2/kg)
Start Year = 2023.000 (yr)
Initial Mass = 160.000 (kg)
Final Mass = 146.000 (kg)
Duration = 5.000 (yr)
Station-Kept = True
Abandoned = True
Long-Term Reentry = False

OUTPUT

Collision Probability = 3.8953E-05
Returned Message: Normal Processing
Date Range Message: Normal Date Range
Status = Pass

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

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

03 24 2022; 16:18:29PM Activity Log Started