Dove 3 Orbital Debris Assessment Report (ODAR)

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

Report Version: 2, 10/9/2012

Document Data is Not Restricted.

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

DAS Software Version Used In Analysis: v2.0.1

VERSION APPROVAL and/or FINAL APPROVAL*:

Chris Boshuizen
CTO

*Approval signatures indicate acceptance of the ODAR-defined risk.

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 3 Mission

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

Notes:

1. This launch has several spacecraft manifested and the Cosmogia spacecraft are not the primary mission.

Assessment Report Format:

ODAR Technical Sections Format Requirements:

As Cosmogia 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 Dove 3 satellite. Sections 9 through 14 apply to the launch vehicle ODAR and are not covered here.

ODAR Section 1: Program Management and Mission Overview

Project Manager: Chris Boshuizen

Foreign government or space agency participation: Russia and Ukraine share ownership of ISC Kosmotras, who operate the Dnepr space launch vehicle.

Schedule of upcoming mission milestones:

FRR:	November 2012
Launch:	No Earlier Than February 2013

Mission Overview:
Dove 3 will be ejected from an ISIPOD 3U CubeSat dispenser into a planned elliptical orbit of 800 x 597 km at 97.8 degrees. The experiment will operate for a maximum duration of 24 months.

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 under 25 years following operations (12.2 years after 2 years of nominal operations, as calculated by DAS 2.0.1).

Launch vehicle and launch site: Dnepr, Yasny/Dombarovskiy, Russia

Proposed launch date: No Earlier Than February 2013

Mission duration: Maximum Nominal Operations: 24 months, Post-Operations Orbit lifetime: 12.2 years until reentry via atmospheric orbital decay (14.2 years in total).

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

The Dnepr rocket, being able to restart to perform orbit changing maneuvers, will be dispensing more than 20 payloads to various orbits close to a circular 600km sun-synchronous orbit.

The Dove 3 satellite will deploy to, and decay naturally from, an elliptical orbit defined as follows:

Apogee: 800 km

Perigee: 597 km

Inclination: 97.79 degrees

Dove 3 has no propulsion and therefore does not actively change orbits. There is no parking or transfer orbit.

ODAR Section 2: Spacecraft Description

Physical description of the spacecraft:

Dove 3 conforms to the 3U CubeSat specification, with a launch mass of 5.2 kg. Basic physical dimensions are 100mm x 100mm x 340mm, with two 260mm x 300mm deployable solar arrays.

The Dove 3 load bearing structure is comprised of three 100mm x 100mm skeleton plates, with L rails along each 300mm corner edge. The solar arrays are spring-loaded and burn-wire deployed.

Power storage is provided by 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.

Dove 3 attitude is approximately determined using the magnetic field vector, measured by onboard magnetometers. Dove 3 attitude will be controlled by a 3-axis magnetorquer controller.

Total satellite mass at launch, including all propellants and fluids: ~5.2 kg.

Dry mass of satellites at launch, excluding solid rocket motor propellants: ~5.2 kg

Description of all propulsion systems (cold gas, mono-propellant, bi-propellant, electric, nuclear): None.

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

Fluids in Pressurized Batteries: None. Dove 3 uses unpressurized standard COTS Lithium-Ion battery cells. Each battery has a height of 65mm, a diameter of 14mmm and a weight of 26 grams.

Description of attitude control system and indication of the normal attitude of the spacecraft with respect to the velocity vector:
Dove 3 utilizes 3 coil magnetorquers, which allow the satellite to despin after initial satellite deployment, 'lock' to the magnetic field, and perform 3-axis control. The nominal attitude will be flying 'edge on', with the bus' long (3U) axis nadir-pointing and two solar arrays laying in the orbital plane, facing towards the sun vector (**A**). At the end of operations, the 3-axis controller will be used to rotate the satellite into maximum drag configuration, which is also the dynamically stable orientation (**B**).

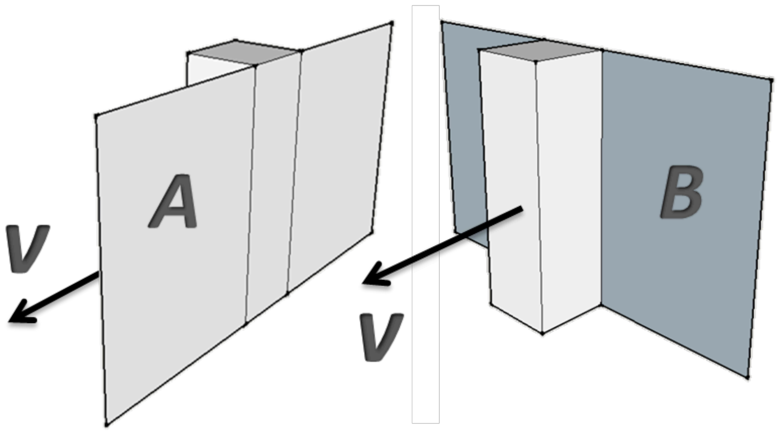

Figure 1: A - 'edge on' attitude during operations
B - 'max drag' attitude at End of Mission (v = velocity)

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 solar cells mounted on the deployable arrays. The battery cell protection circuit manages the charging cycle.

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 v2.0.1)

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 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 8 batteries will not be passivated at End of Mission due to the low risk and low impact of explosive rupturing. The maximum total energy stored in each battery is 12kJ.

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

Dove 3 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: Battery vents are not inhibited by the battery holder design or the spacecraft.
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:
Dove 3 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.

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 v2.0.1, and calculation methods provided in NASA-STD-8719.14, section 4.5.4):

Requirement 4.5-1: Limiting debris generated by collisions with large objects when operating in Earth orbit:
For each spacecraft and launch vehicle orbital stage in or passing through LEO, the program or project shall demonstrate that, during the orbital lifetime of each spacecraft and orbital stage, the probability of accidental collision with space objects larger than 10 cm in diameter is less than 0.001 (Requirement 56506).

Large Object Impact and Debris Generation Probability:

Dove 3; Collision Probability: 0.000011; COMPLIANT.

STK Close Approach Analysis

The above analysis results are a product of NASA's DAS 2.0.1 software. Members of the Cosmogia team have previously worked at NASA researching debris and the risks thereof, and we therefore choose to do some further analysis of the debris threat posed by this mission. We used STK's Conjunction Analysis Toolkit (STK/CAT) to perform a close approach analysis for the Dove 3 orbit. This analysis compares the Dove orbit against the orbits of all of the objects in the US Space Catalog (debris, satellites and human space missions e.g. ISS), reporting all close approaches (within 5 km). This analysis is deterministic rather than statistical, but can be used as a point reference to validate the DAS results.

To approximate the collision probability from the above STK analysis we assume 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 1km normal (radial) to the trajectory. We then assume hard spheres of diameter 1m for Dove 3 and 2m for all other objects in the catalog (these are very conservative). This allows estimation of the probability of collisions with any object in the catalog. This is done for the two year analysis period and then extrapolated to 14.2 years to give an approximation of the absolute upper bound of collision probability (since it assumes no atmospheric decay and the collision probability reduces with altitude below 800km). These collision risk estimates are higher than those calculated by DAS, but are still compliant to Requirement 56506:

Dove 3, P = 2.1577e-006, or approx 0.00046 over the 14.2 year lifetime.

STK Threat to Active Payloads Analysis

Since Dove 3's insertion apogee is 800 km, it passes through a relatively dense orbital regime, including through the Iridium constellation (see the figure below, taken from NASA's Debris Quarterly newsletter). Although both STK and DAS show that the collision probability to be compliant with NASA standards, we strive for a further level of fidelity by performing analysis on the threat to active payloads posed by Dove 3 in this orbit.

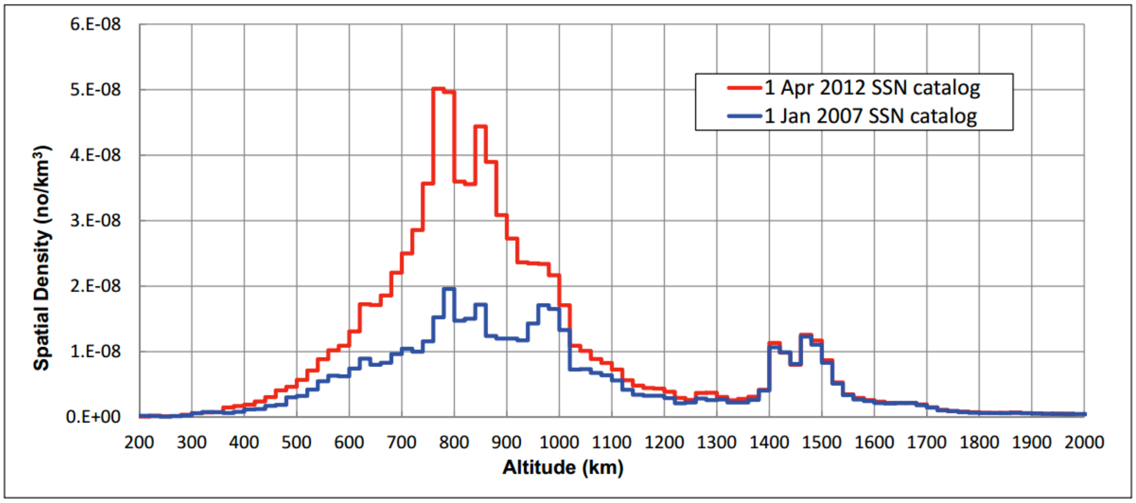

Figure 2: Spatial distribution of catalogued objects for two dates (courtesy NASA).

The Dove 3 orbit was propagated with STK's Astrogator tool until the apogee altitude decayed to 740km (this was approximately 4 years). Dove 3 and the objects in the catalog were assigned a threat volume ellipsoid, which allowed STK/CAT to determine close approaches. This threat volume was defined as a fixed size ellipsoid, 10km tangential (along track), 2km cross-track and 1km normal (radial) to the trajectory. STK/CAT then looked for ellipsoid intersections with all satellite payloads in the US Space Catalog. By assuming hard spheres of diameter 1m for Dove 3 and 4m for objects in the catalog (these are conservative), STK/CAT also calculated collision probabilities.

Over the four years, there was a total collision probability of 1.12e-4 or 1:8921 with other active payloads. On average, any single conjunction had a maximum collision probability of 5.62e-8. These statistics imply a maximum annual collision probability of 0.00003 with other active payloads, translating to less than 0.0004 over the whole mission lifetime.

These statistics imply a larger collision probability than that shown by the close approach analysis above and that shown by DAS, but the conservative sizing of the hard satellite spheres and the choice of ellipsoid size play a role in calculating probabilities.

Ultimately, by all metrics, we are confident that Dove 3 is compliant with NASA debris risk mitigation standards, poses a very low risk of debris generation due to collisions, and poses a very low threat to other active satellites.

Requirement 4.5-2: Limiting debris generated by collisions with small objects when operating in Earth or lunar orbit**:**
For each spacecraft, the program or project shall demonstrate that, during the mission of the spacecraft, the probability of accidental collision with orbital debris and meteoroids sufficient to prevent compliance with the applicable postmission disposal requirements is less than 0.01 (Requirement 56507).

Small Object Impact and Debris Generation Probability:
Dove 3; Collision Probability: 0.00001; COMPLIANT.

Identification of all systems or components required to accomplish any postmission disposal operation, including passivation and maneuvering:
To actively place the satellite in the final "maximum drag" configuration requires the flight computer and ADCS subsystems to be working. However, this configuration is the dynamically stable state for satellite, so even in the event of system failure this attitude will eventually be achieved.

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

6.1 Description of spacecraft disposal option selected: The satellite will de-orbit naturally by atmospheric re-entry. At the end of Dove 3's operational life (i.e. at EOM) the attitude control system will stop counteracting the aerodynamic disturbance torques and will rotate the satellite into the maximum drag configuration. This will result in Dove 3 gradually assuming a dynamically stable configuration. To determine this stable orientation, an in-house developed aerodynamic simulation based on free-molecular flow with a simplified particle/surface interaction model (NASA SP-8058 eq 2-2) was used to compute force and moment coefficients for the spacecraft in all attitudes. One stable and several unstable equilibrium attitudes were found. The spacecraft is stable aerodynamically when the velocity vector is aligned with the +Y (panel normal) body axis (the "high drag" configuration). This is suitable for a rapid de-orbit at end of mission. Gravity gradient torques act to align the Z body axis (3U long) axis with zenith/nadir; this does not act to oppose the aerodynamic stability torque. The combination of gravity gradient and aerodynamic torques results in a fully-constrained 3-axis attitude with +Y along the velocity vector, + or - Z at nadir, and X perpendicular to the orbit plane. This is the maximum drag configuration, and to accelerate the orbital decay we plan to orient the satellite in this configuration at the end of operations.

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

The stable drag/gravity gradient configuration enables aerodynamic reentry. To accelerate the orbital decay we plan to orient the satellite in this maximum drag configuration at the end of operations.

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

Spacecraft Mass: ~5.2kg

Cross-sectional Area: 0.19 m^2 (dynamically stable)

Area to mass ratio: 0.037 m^2/kg (dynamically stable)

6.4 Assessment of spacecraft compliance with Requirements 4.6-1 through 4.6-5 (per DAS v 2.0.1 and NASA-STD-8719.14 section):

Requirement 4.6-1: Disposal for space structures passing through LEO:
A spacecraft or orbital stage with a perigee altitude below 2000 km shall be disposed of by one of three methods:
(Requirement 56557)
a. Atmospheric reentry option:

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

b. Storage orbit option: Maneuver the space structure into an orbit with perigee altitude greater than 2000 km and apogee less than GEO - 500 km.
c. Direct retrieval: Retrieve the space structure and remove it from orbit within 10 years after completion of mission.

Analysis: The Dove 3 satellite reentry is COMPLIANT using method "a".

Satellite Name	Dove 3
BOL Orbit	800 x 597 km
EOM Orbit*	799 x 596 km
Total Lifetime	14.2 years
Post-ops Life	12.2 years

* EOM orbit was calculated using STK's astrogator, taking into different attitude configurations during the various phases of the mission

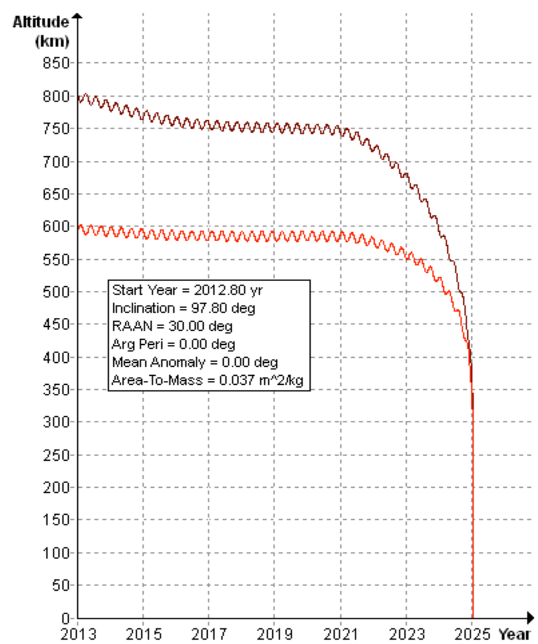

Figure 5: Dove 3 (right) orbit history with apogee (brown) & perigee (red) plotted

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: The maximum drag configuration is the aerodynamically stable state, meaning that even under massive subsystem failure we would eventually assume this orientation.

ODAR Section 7: Assessment of Spacecraft Reentry Hazards

Assessment of spacecraft compliance with Requirement 4.7-1:

Requirement 4.7-1: Limit the risk of human casualty:
The potential for human casualty is assumed for any object with an impacting kinetic energy in excess of 15 joules:

a) *For uncontrolled reentry, the risk of human casualty from surviving debris shall not exceed 0.0001 (1:10,000) (Requirement 56626).*

Summary Analysis Results: DAS v2.0.1 reports that Dove 3 is compliant with the requirement. There is a low probability of the Invar telescope tube reaching the ground (see DAS input data below for input parameters). However, the DAS software does not allow explicit modeling of a thin cylindrical tube wall (inputs are cylinder shape and thermal mass), so these numbers are expected to be larger than anticipated. Total human casualty probability is reported by the DAS software as **1:156200** for Dove 3**.** This is expected to represent the absolute maximum casualty risk, as calculated with DAS's limited modeling capability.

Analysis (per DAS v2.0.1):

```
06 19 2012; 16:57:25PM   DAS Application Started
06 19 2012; 16:57:25PM   Opened Project D:\dove3\
06 19 2012; 16:57:35PM   Processing Requirement 4.3-1:       Return Status :  Not Run

=====================
No Project Data Available
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=============== End of Requirement 4.3-1 ===============
06 19 2012; 16:57:52PM   Processing Requirement 4.3-2: Return Status : Passed

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No Project Data Available
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=============== End of Requirement 4.3-2 ===============
06 19 2012; 16:57:57PM   Requirement 4.4-3: Compliant

=============== End of Requirement 4.4-3 ===============
06 19 2012; 16:58:03PM   Processing Requirement 4.5-1:   Return Status : Passed

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

INPUT

Space Structure Name = Dove 3
Space Structure Type = Payload
Perigee Altitude = 597.000000 (km)
Apogee Altitude = 800.000000 (km)
Inclination = 97.790000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass Ratio = 0.036538 (m^2/kg)
Start Year = 2012.880000 (yr)
Initial Mass = 5.200000 (kg)
Final Mass = 5.200000 (kg)
Duration = 2.000000 (yr)
Station-Kept = False
Abandoned = False
PMD Perigee Altitude = 595.700000 (km)
PMD Apogee Altitude = 799.000000 (km)
PMD Inclination = 97.790000 (deg)
PMD RAAN = 0.000000 (deg)
PMD Argument of Perigee = 0.000000 (deg)
PMD Mean Anomaly = 0.000000 (deg)

OUTPUT

Collision Probability = 0.000011
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range
Status = Pass

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

=============== End of Requirement 4.5-1 ===============
06 19 2012; 16:58:10PM   Requirement 4.5-2: Compliant
06 19 2012; 16:58:23PM   Processing Requirement 4.6   Return Status : Passed

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

INPUT

Space Structure Name = Dove 3
Space Structure Type = Payload

Perigee Altitude = 597.000000 (km)
Apogee Altitude = 800.000000 (km)
Inclination = 97.790000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Area-To-Mass Ratio = 0.036538 (m^2/kg)
Start Year = 2012.880000 (yr)
Initial Mass = 5.200000 (kg)
Final Mass = 5.200000 (kg)
Duration = 2.000000 (yr)
Station Kept = False
Abandoned = False
PMD Perigee Altitude = 595.700000 (km)
PMD Apogee Altitude = 799.000000 (km)
PMD Inclination = 97.790000 (deg)
PMD RAAN = 0.000000 (deg)
PMD Argument of Perigee = 0.000000 (deg)
PMD Mean Anomaly = 0.000000 (deg)

OUTPUT

Suggested Perigee Altitude = 595.700000 (km)
Suggested Apogee Altitude = 799.000000 (km)
Returned Error Message = Passes LEO reentry orbit criteria.

Released Year = 2029 (yr)
Requirement = 61
Compliance Status = Pass

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

=============== End of Requirement 4.6 ===============
06 19 2012; 16:59:04PM *********Processing Requirement 4.7-1
Return Status : Passed

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

name = Dove 3
quantity = 1
parent = 0
materialID = 5
type = Box
Aero Mass = 5.200000
Thermal Mass = 5.200000
Diameter/Width = 0.100000
Length = 0.340000
Height = 0.100000

name = Camera
quantity = 1
parent = 1
materialID = 5
type = Box
Aero Mass = 0.370000
Thermal Mass = 0.370000
Diameter/Width = 0.060000
Length = 0.080000
Height = 0.060000

name = Batteries
quantity = 8
parent = 1
materialID = 46
type = Cylinder
Aero Mass = 0.026000
Thermal Mass = 0.026000
Diameter/Width = 0.014000
Length = 0.065000

name = Structure
quantity = 1
parent = 1
materialID = 5
type = Box
Aero Mass = 1.280000
Thermal Mass = 1.280000
Diameter/Width = 0.100000
Length = 0.340000
Height = 0.100000

name = Solar Arrays
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 = Avionics
quantity = 1
parent = 1
materialID = 23
type = Box
Aero Mass = 0.200000
Thermal Mass = 0.200000
Diameter/Width = 0.100000
Length = 0.100000
Height = 0.100000

name = Optical Tube
quantity = 1
parent = 1
materialID = 72
type = Cylinder
Aero Mass = 2.080000
Thermal Mass = 2.080000
Diameter/Width = 0.091000
Length = 0.200000

name = +/- X Structure Walls
quantity = 2
parent = 1
materialID = 5
type = Flat Plate
Aero Mass = 0.200000
Thermal Mass = 0.200000
Diameter/Width = 0.100000
Length = 0.300000

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

name = Dove 3
Demise Altitude = 77.999691
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Camera
Demise Altitude = 72.690324
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Batteries
Demise Altitude = 74.279746
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Structure
Demise Altitude = 73.855996
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Solar Arrays
Demise Altitude = 77.857871
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Avionics
Demise Altitude = 76.855277
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Optical Tube
Demise Altitude = 0.000000
Debris Casualty Area = 0.540089
Impact Kinetic Energy = 2419.714355

****
name = +/- X Structure Walls
Demise Altitude = 76.366027
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****

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

Requirements 4.7-1b, and 4.7-1c below are non-applicable requirements because Dove 3 does not use controlled reentry.

4.7-1, b) **NOT APPLICABLE.** For controlled reentry, the selected trajectory shall ensure that no surviving debris impact with a kinetic energy greater than 15 joules is closer than 370 km from foreign landmasses, or is within 50 km from the continental U.S., territories of the U.S., and the permanent ice pack of Antarctica (Requirement 56627).
4.7-1 c) **NOT APPLICABLE.** For controlled reentries, the product of the probability of failure of the reentry burn (from Requirement 4.6-4.b) and the risk of human casualty assuming uncontrolled reentry shall not exceed 0.0001 (1:10,000) (Requirement 56628).

ODAR Section 8: Assessment for Tether Missions

Not applicable. There are no tethers in the Dove 3 mission.

END of ODAR for Dove 3