Flock 1 Orbital Debris Assessment Report (ODAR)

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

Report Version: 2.3, 06/20/2013

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: Flock 1 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 is a deployment from the ISS and there is no launch directly associated with the deployment of these satellites. The satellites are deployed from the Multi-Purpose Experiment Platform aboard the Japanese Experimental Module. No Mission Related debris is expected.

Assessment Report Format:

ODAR Technical Sections Format Requirements:

As Planet Labs Inc.[1] 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 satellites in the Flock 1 constellation. Sections 9 through 14 apply to the launch platform, in this case the International Space Station, and are not covered here.

ODAR Section 1: Program Management and Mission Overview

Project Manager: Planet Labs

Foreign government or space agency participation: The satellites will deploy from the Japanese Experimental Module aboard the International Space Station and will therefore involve representatives of the participating space agencies (NASA, the Russian Federal Space Agency, JAXA, ESA, and CSA). Transport to the space station will be aboard a Commercial Resupply Services (CRS) flight provided by Orbital Sciences on the Antares launch vehicle.

Schedule of upcoming mission milestones:

Launch: No Earlier Than December 2013

Mission Overview:
The 28 satellites composing the Planet Labs Flock 1 constellation will be delivered to the ISS aboard an resupply flight provided by Orbital Sciences on the Antares launch vehicle, and will gradually be deployed from the Multi-Purpose Experiment Platform and commissioned. The first 16 satellites will be deployed in pairs, once per orbit in the ISS's "45 degree nadir-aft" direction. The other 12 will be deployed in a similar fashion after a re-loading procedure. The constellation will then begin payload operations that will continue for 11-18 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 well under 25 years following operations (< 2 years, as calculated by DAS 2.0.2 and STK10).

Launch vehicle and launch site: Antares, Wallops Flight Facility

Proposed launch date: No Earlier Than December 2013

Mission duration: Nominal orbit lifetime: 11 months. Maximal orbit lifetime: 18 months

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

The 28 Flock 1 satellites will deploy nadir-aft from the ISS into an inclined orbit from which they will naturally decay due to atmospheric drag. The deployment altitude

[1] Formerly known as Cosmogia Inc.

depends on the ISS station-keeping boost schedule, so the entire altitude range is considered.

***High Insertion Case*:** **Apogee:** 410 km **Perigee:** 410 km

Low Insertion Case: **Apogee:** 380 km **Perigee:** 380 km

Inclination: 51.6 degrees

The Flock 1 satellites have no propulsion and therefore do not actively change orbits. There is no parking or transfer orbit.

ODAR Section 2: Spacecraft Description

Physical description of the spacecraft:

The Flock 1 satellites are variants of the 3U CubeSat specification, with a launch mass of 4 kg. Basic physical dimensions are 100mm x 100mm x 340mm, with two 260mm x 300mm deployable solar arrays.

The 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 deployed by burn-wires.

Power storage is provided by 12 AA 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: 4.3 kg.

Dry mass of satellites at launch, excluding solid rocket motor propellants: 4.3 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. The satellites use unpressurized standard COTS Lithium-Ion battery cells. Each battery has a height of 49mm, a diameter of 14mmm and a weight of 21 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.

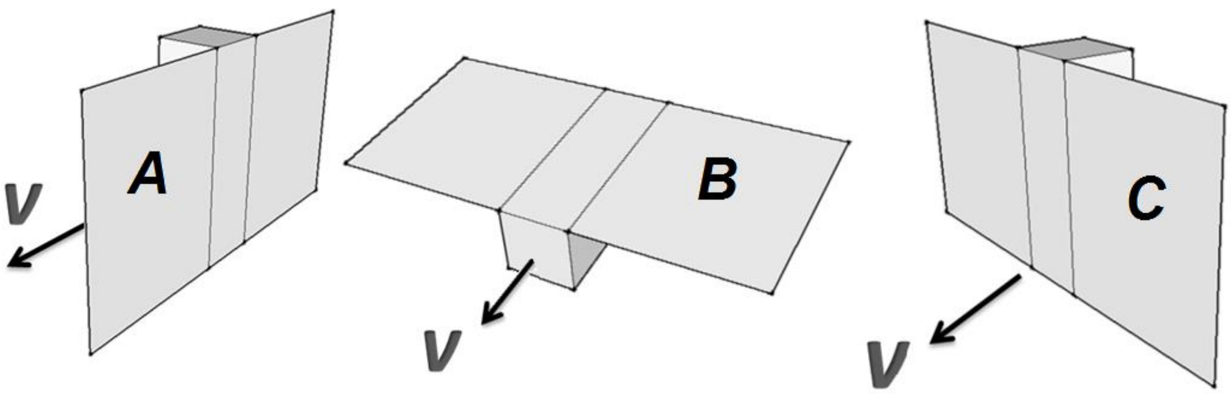

Figure 1: Attitude modes used in Flock 1. A –nadir pointing, B – low drag, C – high drag.

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 Si 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:

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 12 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 ~10kJ.

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. This electrical power system has already been flight qualified on the Dove 1 and Dove 2 missions.

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. This design has been verified through the Dove 1 and Dove 2 missions.
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.

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: < 0.00000; COMPLIANT.

Analysis (per DAS v2.0.2):

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06 20 2013; 16:49:09PM      Processing Requirement 4.5-1:        Return Status :  Passed

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

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         Initial Mass = 4.300000 (kg)
         Final Mass = 4.300000 (kg)
         Duration = 1.000000 (yr)
         Station-Kept = False
         Abandoned = True
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         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

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```

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 the entire 28 satellite constellation and the given probability is the sum of the individual collision probabilities of each of the 28 satellites. In addition, STK's Conjunction Analysis Toolkit (STK/CAT) was used to perform a close approach/conjunction analysis for the Flock 1 deployment and orbit. This analysis compares the Flock 1 members' 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). This analysis is deterministic rather than statistical, but can be used as a point reference to validate the DAS results. We also specifically investigate ISS re-contact risks.

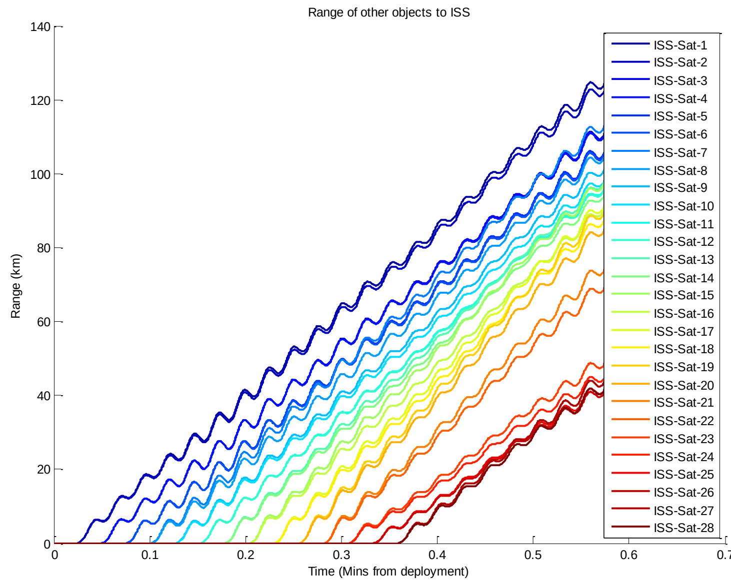

Figure 2: Scalar range from ISS to each Flock 1 satellite

Part 1: Flock 1 vs ISS

The first 16 Flock 1 satellites will be deployed in pairs, once per orbit in the ISS's "45 degree nadir-aft" direction. The other 12 will be deployed in a similar fashion after re-loading procedure. This deployment direction has been selected to minimize risk of collisions and at least 5 CubeSats have already been safely deployed from the ISS in this fashion. To simulate the collision risk to the ISS, we modeled this deployment scenario in STK using the Astrogator force model propagation tool. Following deployment, each satellite will assume a 3U random tumbling drag profile for 2 days before assuming the nominal mission attitude and deploying solar arrays.

This analysis shows no risk of collisions with the ISS. The pairs of deployed satellites, having had their semi-major axes reduced during the deployment, rapidly drift away from the ISS at a rate of about 55 m per minute (or about 5 km per orbit).

The Flock 1 satellites comply with the ISS requirements of a ballistic number greater than 100 kg/m^2 which means that they will always decay faster than (and be below) the ISS. The initially deployed satellite eventually drifts all the way around the orbit to pass the ISS approximately 75 days after deployment. However, at this time the closest approach to the ISS is about 9km, mostly in the radial direction (where orbit uncertainty is the least).

An STK/CAT simulation of the 28 Flock 1 satellites against the ISS two line element (TLE), over the entire mission duration, confirmed zero risk of collision.

Part 2: Flock 1 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 the Flock 1 constellation. 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 Flock 1 satellites and 2m for all other objects in the catalog. This allows estimation of the probability of collisions between any Flock 1 satellite and any existing object in the catalog. This was done for an analysis period that covered the entire orbital lifetime:

Collision Probability: 2.6838E-05; COMPLIANT.

Part 3: Flock 1 vs Flock 1

Individual pairs of satellites will initially start with a separation of about 5km from the previous pair. Pairs of Flock 1 satellites will initially drift apart due to the slight difference in deployment velocity out of the deployer and then due to differing atmospheric drag forces. These conditions were simulated in STK and an STK/CAT conjunction analysis, specifically focused on collisions between members of the full Flock 1 constellation, and was performed over the entire orbit lifetime:

Collision Probability: 4.7485E-05; COMPLIANT.

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 1 members.

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).

Flock 1 is to be deployed into a very low Earth orbit. The density of resident space objects, and therefore the probability of collisions, reduces with altitude below about 800km. Therefore the "high insertion" scenario (where satellites are deployed at 410km) represents the highest collision probability insertion scenario and we perform the DAS analysis for this case.

Small Object Impact and Debris Generation Probability:
Collision Probability (single satellite): 0.000004; COMPLIANT.

Collision Probability (complete system): 0.000112; COMPLIANT.

Analysis (per DAS v2.0.2):

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06 20 2013; 17:27:34PM      Requirement 4.5-2:  Compliant

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Critical Surface = WallX
=================================================

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         Argument of Perigee = 0.000000 (deg)
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         Initial Mass = 4.300000 (kg)
         Final Mass = 4.300000 (kg)
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         CS Pressurized = No
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**OUTPUT**

         Probabilty of Penitration = 0.000001
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         Date Range Error Message: Normal Date Range

=================================================
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Critical Surface = WallY
=================================================

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         Argument of Perigee = 0.000000 (deg)
         Mean Anomaly = 0.000000 (deg)
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         Initial Mass = 4.300000 (kg)
         Final Mass = 4.300000 (kg)
         Station Kept = No
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         CS Surface Area = 0.190000 (m^2)
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         CS Pressurized = No
         Outer Wall 1   Density: 2.700000 (g/cm^2)  Separation: 0.200000 (cm)

**OUTPUT**

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         Date Range Error Message: Normal Date Range

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Spacecraft = F1
```

```
Critical Surface = WallZ
=================================================

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        Mean Anomaly = 0.000000 (deg)
        Final Area-To-Mass = 0.007665 (m^2/kg)
        Initial Mass = 4.300000 (kg)
        Final Mass = 4.300000 (kg)
        Station Kept = No
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        Duration = 1.000000 (yr)
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        CS Areal Density = 2.700000 (g/cm^2)
        CS Surface Area = 0.010000 (m^2)
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**OUTPUT**

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        Returned Error Message: Normal Processing
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Date Range Error Message: Normal Date Range

Identification of all systems or components required to accomplish any postmission disposal operation, including passivation and maneuvering:
None.

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 within two years of deployment.

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

Rapid atmospheric decay is ensured since the mission extends through most of the orbital lifetime. The nadir pointing or velocity vector alignment requirements determine the ballistic coefficient up until the perigee altitude is approximately 200km. After this point, the satellites may be allowed to tumble, and assuming minimum drag area reentry will occur within one week from this altitude.

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

Spacecraft Mass:	4.3 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:	0.0077 m^2/kg
	Low Drag configuration:	0.0030 m^2/kg
	High Drag configuration:	0.0441 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 naturally assume the High Drag configuration and begin to de-orbit at the fastest rate possible.

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

High Insertion Case

Satellite Name	Flock 1
BOL "High" Orbit	410 × 410 km
EOM Orbit	200 × 200 km
Min Lifetime*	10 months
Max Lifetime*	18 months
Post-ops Life	< 1 week

Low Insertion Case

Satellite Name	Flock 1
BOL "Low" Orbit	380 × 380 km
EOM Orbit	200 × 200 km
Min Lifetime*	6 months
Max Lifetime*	12 months
Post-ops Life	< 1 week

* Min and Max lifetimes take into account variation of operational modes and space weather uncertainty to bound the orbit lifetime

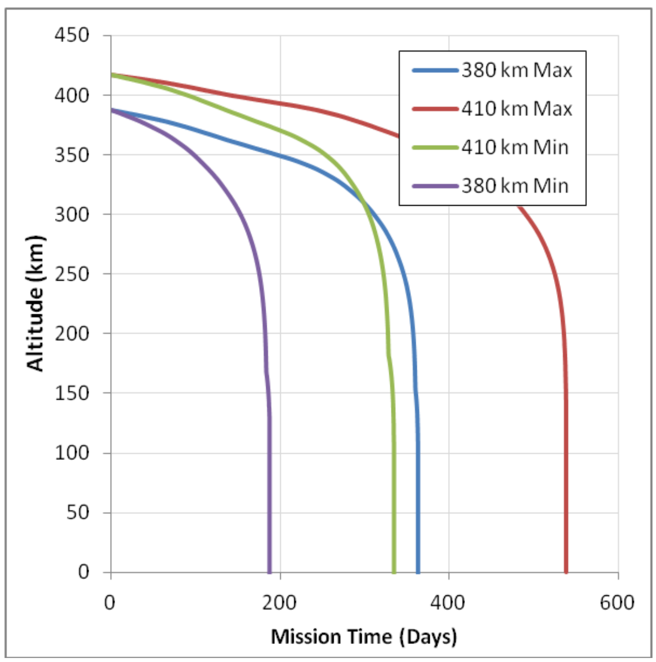

Figure 3: Flock 1 orbit history for the four cases investigated.

DAS Analysis: The Flock 1 satellites' satellite reentry is COMPLIANT using method "a".

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06 20 2013; 17:38:34PM      Science and Engineering - Orbit Lifetime/Dwell Time

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Project Data
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**INPUT**

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        Perigee Altitude = 410.000000 (km)
        Apogee Altitude = 410.000000 (km)
        Inclination = 51.600000 (deg)
        RAAN = 0.000000 (deg)
        Argument of Perigee = 0.000000 (deg)
        Area-To-Mass Ratio = 0.007400 (m^2/kg)

**OUTPUT**

        Orbital Lifetime from Startyr = 0.821355 (yr)
        Time Spent in LEO during Lifetime  = 0.821355 (yr)
        Last year of Propagation = 2014 (yr)
```

Returned Error Message: Object reentered

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).*

Analysis performed using DAS v2.0.2 shows that no part of the satellite is expected to survive reentry, and therefore that the risk of human casualty is ~ 0.

Analysis (per DAS v2.0.2):

```
06 20 2013; 17:52:34PM      *********Processing Requirement 4.7-1
         Return Status :  Passed

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

name = F1
quantity = 1
parent = 0
materialID = 5
type = Box
Aero Mass = 4.300000
Thermal Mass = 4.300000
Diameter/Width = 0.100000
Length = 0.340000
Height = 0.100000

name = Payload1
quantity = 1
parent = 1
materialID = 5
type = Box
Aero Mass = 0.670000
Thermal Mass = 0.670000
Diameter/Width = 0.060000
Length = 0.080000
Height = 0.060000

name = Batteries1
quantity = 12
parent = 1
materialID = 46
type = Cylinder
Aero Mass = 0.021000
Thermal Mass = 0.021000
Diameter/Width = 0.014000
Length = 0.049000

name = Structure1
quantity = 1
parent = 1
materialID = 5
type = Box
Aero Mass = 0.700000
Thermal Mass = 0.700000
Diameter/Width = 0.100000
Length = 0.340000
Height = 0.100000

name = Rails1
quantity = 4
parent = 1
materialID = 5
type = Flat Plate
Aero Mass = 0.100000
Thermal Mass = 0.100000
Diameter/Width = 0.040000
Length = 0.300000

name = Solar Arrays1
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 = Avionics1
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 = Radiators1
quantity = 8
parent = 1
materialID = 5
type = Flat Plate
Aero Mass = 0.150000
Thermal Mass = 0.150000
Diameter/Width = 0.100000
Length = 0.300000

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

name = F1
Demise Altitude = 77.995269
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Payload1
Demise Altitude = 71.611761
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Batteries1
Demise Altitude = 72.196503
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Structure1
Demise Altitude = 75.142261
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Rails1
Demise Altitude = 75.975566
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Solar Arrays1
Demise Altitude = 77.808894
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Avionics1
Demise Altitude = 76.575332
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

*************************************
name = Radiators1
Demise Altitude = 76.466480
Debris Casualty Area = 0.000000
```

```
Impact Kinetic Energy = 0.000000
*************************************
```

Requirements 4.7-1b, and 4.7-1c below are non-applicable requirements because the Flock 1 satellites do 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 Flock 1 mission.

END of ODAR for Flock 1