Prox-1

Orbital Debris Assessment Report
(ODAR)

FINAL
18 July 2016

Submitted by:

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Record of Revisions

Revision	Date	Affected Pages	Description of Change	Authors
A	11 March 2016	All	Initial Revisions	Kevin Okseniuk
FINAL	18 July 2016	1, 4, 5	ITAR restriction removed, signature added, ODAR self-assessment updated	David Spencer

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Self-Assessment of the ODAR

A self-assessment of the ODAR is provided below.

Orbital Debris Self-Assessment Report Evaluation: Prox-1 Mission

	Spacecraft			
Requirement #	Compliant	Not Compliant	Incomplete	Comments
4.3-1.a	X			No Debris Released in LEO.
4.3-1.b	X			No Debris Released in LEO.
4.3-2	X			No Debris Released in LEO.
4.4-1	X			
4.4-2	X			
4.4-3	X			No Planned Breakups.
4.4-4	X			No Planned Breakups.
4.5-1	X			
4.5-2	X			
4.6-1(a)	X			
4.6-1(b)	X			
4.6-1(c)	X			
4.6-2	X			
4.6-3	X			
4.6-4	X			
4.6-5	X			
4.7-1	X			
4.8-1	X			

Assessment Report Format

ODAR Technical Sections Format Requirements:
This ODAR follows the format 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 Prox-1 satellite. Sections 9 through 14 apply to the launch vehicle ODAR and are omitted.

ODAR Section 1: Program Management and Mission Overview

Mission Description

The Georgia Institute of Technology Prox-1 mission will demonstrate automated trajectory control in low-Earth orbit relative to a deployed three-unit (3U) CubeSat, for an on-orbit inspection application. The Prox-1 flight system will deploy The Planetary Society's LightSail-2 CubeSat using a standard 3U Poly-Picosat Orbital Deployer (P-POD). Using vision-based relative navigation and a guidance strategy that employs relative orbital elements for station-keeping and an artificial potential function for persistent collision avoidance, Prox-1 will demonstrate automated proximity operations with a university-class microsatellite platform. Funded by the Air Force Office of Scientific Research/Air Force Research Laboratory through the University Nanosatellite Program-7, Prox-1 is scheduled to launch in Q1 2017 as a secondary payload on the Space Test Program-2 launch. The student-led Prox-1 mission has involved more than 150 undergraduate and graduate students during design, development, integration and testing.

Launch vehicle and launch site: SpaceX Falcon Heavy from Cape Canaveral, FL.

Proposed launch date: Q1 2017

Mission duration: 6 weeks

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

The Prox-1 orbital elements are defined as follows:

Apogee: 720 km
Perigee: 720 km
Inclination: 24.0 degrees

Prox-1 has a propulsion unit required for post-ejection rendezvous and station-keeping with respect to LightSail 2 (LS2). The proximity operations phase of the mission is designed to operate within a radius of 250m of LS2, therefore the propulsion unit is not intended for significant orbit changes. The net propulsive delta-V capacity of Prox-1 is 16.5 m/s.

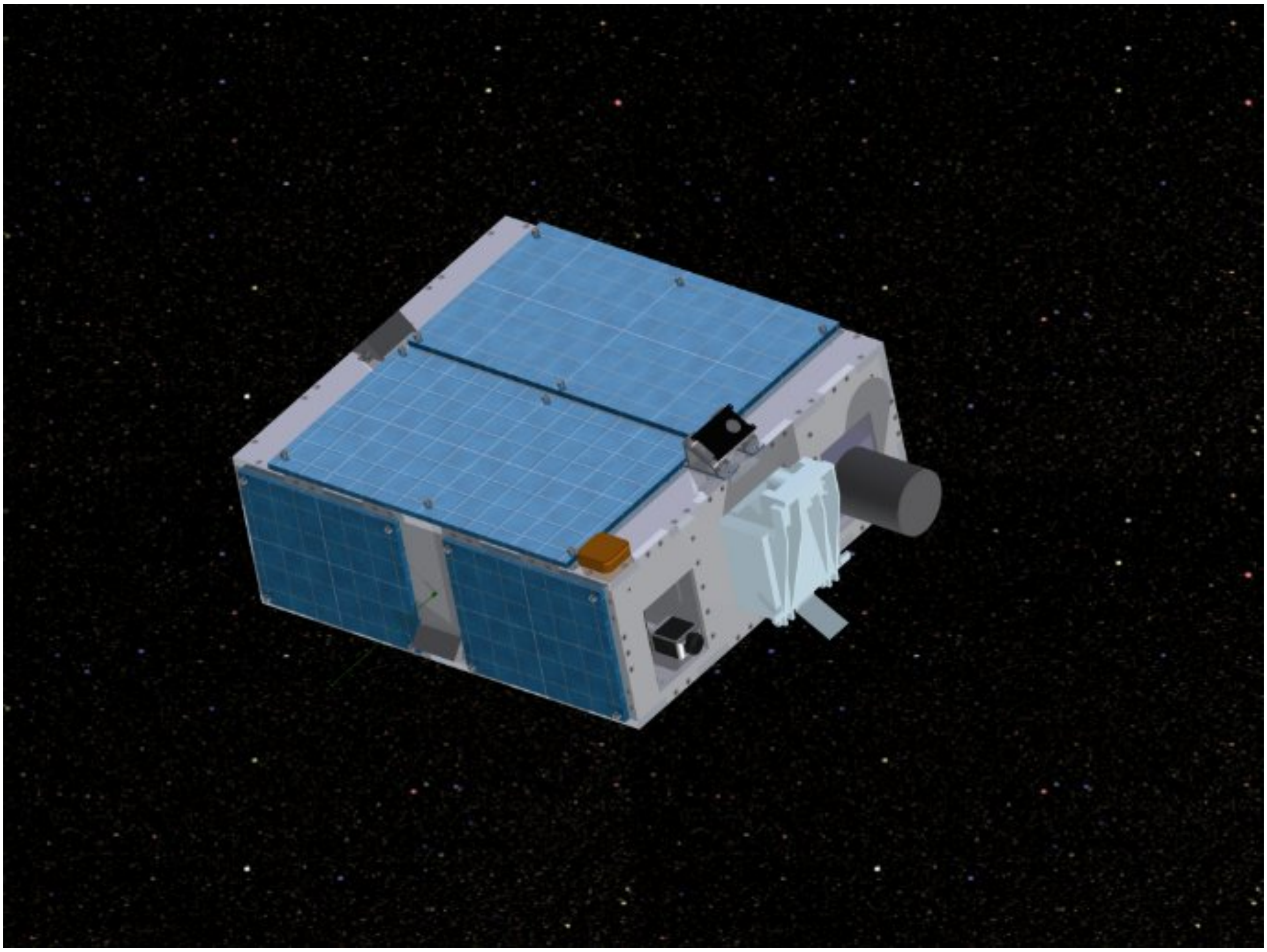
Figure 1: Prox-1 nominal flight configuration.

Interaction or potential physical interference with other operational spacecraft:

- At this time, we know of no potential interaction or physical interference between Prox-1 and any other operational spacecraft. The Prox-1 guidance algorithms have collision-avoidance capability if the distance between Prox-1 and LS2 becomes less than 25 m. The planned operating range during nominal proximity operations is 50 – 200 m.

Project Management:

- Principal Investigator: Dr. David Spencer, Professor of the Practice, Georgia Institute of Technology
- Project Manager: Christine Gebara, Undergraduate Student, Georgia Institute of Technology
- System Engineer: Swapnil Pujari, Undergraduate Student, Georgia Institute of Technology

Key engineering personnel:

- ADCS Lead: Phillip Reeder, Undergraduate Student, Georgia Institute of Technology
- COMM Lead: Nicholas Zerbonia, Undergraduate Student, Georgia Institute of Technology
- GNC Lead: Peter Schulte, Graduate Student, Georgia Institute of Technology

- Mission Operations Lead: Teresa Spinelli, Undergraduate Student, Georgia Institute of Technology
- Structures Lead: Swapnil Pujari, Undergraduate Student, Georgia Institute of Technology
- TCS Lead: Dylan Radford, Undergraduate Student, Georgia Institute of Technology
- EPS Lead: Joseph Boettcher, Undergraduate Student, Georgia Institute of Technology

Foreign government or space agency participation:

- No foreign agency is participating in this mission. All personnel are United States citizens.

Summary of NASA's responsibility under the governing agreement(s):

- Not applicable.

Schedule of mission design and development milestones from NASA mission selection through proposed launch date, including spacecraft PDR and CDR (or equivalent) dates*:

Date	Milestone
Q1-Q2 2013	Detailed Design
Q3 2013 – Q3 2015	Development
Q4 2015 – Q2 2016	Flight Unit Integration & Testing
Q3-Q4 2016	Delivery, Environmental Testing, LV Integration
Q1 2017	Launch
Q1-Q2 2017	Deployment and Operations

Section 2: Spacecraft Description

Physical description of the spacecraft:
The Prox-1 satellite has a net launch mass of 71.3 kg (69.4kg dry launch mass). Its physical dimensions are 60.98 cm x 55.88 cm x 30.48cm (24 x 22 x 12 inches), and all of the flight structure is contained within the envelope of this prism. There are no deployable structures attached to the spacecraft. Prox-1 will eject LS2 (net mass 4.9 kg) via the P-POD mechanism. Therefore, assuming no fuel was consumed in orbit, its de-orbit mass after ejection will be 66.4 kg. The LS2 mission is submitting its own ODAR.

The Prox-1 satellite will contain the following systems:

- Analog Devices AD16365 inertial measurement unit
- Four ELMOS e910.86 coarse sun sensors (MEMS)
- SpaceQuest GPS-ANT antenna
- SpaceQuest GPS-12 GPS receiver
- SpaceQuest MAG-3 Magnetometer
- 3 In-house built magnetic torquers (Aluminum 6061 casing, Hiperco-50A core)
- Honeybee Microsatellite Control Moment Gyroscope
- Poly-Picosatellite Orbital Deployer (P-POD) Mark II
- The Planetary Society's LightSail 2 3U CubeSat
- In-house built solar cells

- ClydeSpace Ltd. 3U Flex EPS and Power Distribution Module
- 3x 30 W-Hr ClydeSpace Lithium Polymer
- University of Texas-Austin Cold Gas Thruster (plus R-236f propellant)
- Aluminum 6061 mechanical structure
- Printed circuit boards (made out of FR-4, standard-grade)
- Texas Instruments' BeagleBoard-XM flight computer
- 3x Arduino MEGA2560 subsystem micro controllers
- SpaceQuest RX-445 UHF receiver (plus monopole antenna)
- SpaceQuest TX-2400 S-band transmitter (plus monopole antenna)
- Kantronics KPC-9612 terminal node controller

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

- 71.3 kg

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

- 69.4 kg

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

- The propulsion system is composed of a cold gas thruster developed at University of Texas-Austin. The propellant is a non-toxic R-236f refrigerant. A smaller version of this unit has flight heritage, and a filled Prox-1 unit has been extensively (160 hours) and successfully (no mass loss) tested in a laboratory vacuum environment representative of LEO.

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

- The only fluid on-board is the R-236f refrigerant used in the propulsion unit. The mass of the propellant will be 1.84kg. The maximum pressure of the propellant will be 70.6 psia (at 316 K). The non-toxic propellant will be loaded at the environmental testing facility (Kirtland Air Force Base) prior to delivery and integration to the launch vehicle.

Fluids in Pressurized Batteries:

- None. Prox-1 uses unpressurized standard COTS Lithium-Ion battery cells.

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

- Attitude determination is accomplished with four types of sensors. The Analog Devices ADIS16365 Inertial Measurement Unit (IMU) provides 3-axis acceleration and angular rate measurements. The SpaceQuest MAG-3 magnetometer provides precise measurement of the Earth's magnetic field in the spacecraft body-frame. Four ELMOS 986.10 sun sensors provide course sun angles. High-rate orbital state and timing information for precise orbit determination is given by a SpaceQuest Ltd. GPS-12-V1 receiver processing unit and antenna suite. Position accuracy of the GPS unit is specified as 10 m, and velocity accuracy is 0.03 m/s RMS. The subsystem interfaces with the flight computer via two microcontrollers with an ATMEGA2560 processor, four custom hardware circuit boards, and a specialized set of harnessing. Software libraries for these components were developed at Georgia Tech.

- Attitude control is achieved by a set of three orthogonally-oriented torque rods and CMGs. Three torque rods are mounted along the spacecraft body axes, and their design specification of producing a dipole moment of 10 A-m2 have been verified experimentally. The torque rods will be used for detumbling the spacecraft, and for CMG desaturation. A microsatellite CMG unit developed by Honeybee Robotics for the Air Force Research Laboratory (AFRL) will be utilized for precise attitude control for Prox-1. The unit houses four gyroscopes in an integrated system that provides robust three-axis control. The microsatellite CMG unit will provide the fine attitude control necessary to accomplish precise pointing and agile maneuvering during proximity operations.
- The normal attitude of the spacecraft will be with the P-POD axis (the spacecraft's major axis) aligned with the inertial velocity vector, and the top panel's normal aligned with the nadir vector.

Description of any range safety or other pyrotechnic devices:
- Prox-1 will use a burn wire for deployment of the LS2 satellite via the P-POD mechanism. The wire will burn at a resistor temperature of 200 C and a nominal voltage of 12.0V. The duration of the burn is 20 milliseconds and this has been repeatedly verified in laboratory experiments.

Description of the electrical generation and storage system:
- Power is generated by seven body-mounted dual junction solar panels on four faces of the spacecraft for an average power generation during sunlit operations of 45 Watts. These solar panels were designed and fabricated by student personnel. Power generation and storage is regulated by a custom ClydeSpace Ltd. 3U Flex EPS and Power Distribution Module. These units have been modified to accommodate a total of 18 battery charge regulators for power generation. A sun-regulated bus distribute powers from the solar cells and batteries to the loads, with a nominal bus voltage of 28 +/- 1.5 VDC. During sunlight, shunt control dissipates excess power. Voltage is regulated to 5 V, 7 V, 12 V, and 28 V for instrument and subsystem operation. Energy is stored via the use of three 30 W-Hr ClydeSpace Lithium Polymer battery packs.

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:
- LightSail-2
- 3U CubeSat, 10 x 10 x 30 cm
- Net mass: 4.9kg

Rationale/necessity for release of each object:

- LightSail 2 will be used as the target for Prox-1 to perform proximity operations. Prox-1 will provide on-orbit inspection of the LS2 sail deployment event.

Time of release of each object, relative to launch time:
- LS2 ejection is scheduled to occur no later 14 days after launch in the nominal mission timeline. In case of a communication link failure, an on-board watchdog circuit and algorithm will automatically eject LS2 after 28 days since the flight computer was first initialized.

Release velocity of each object with respect to spacecraft:
- LS2 will be deployed with delta-V of 1.6 m/s relative to Prox-1, in the cross-track direction.

Expected orbital parameters (apogee, perigee, and inclination) of each object after release:
- The orbital parameters of LS2 will be negligibly different from the Prox-1 orbit parameters. Following deployment in the cross-track direction, LS2 will be on a 720km circular orbit at 24.0 degrees of inclination.

Calculated orbital lifetime of each object, including time spent in Low Earth Orbit (LEO):
- LS2 is expected to deorbit from LEO in 700 days. The LightSail 2 project will provide its own ODAR separate from Prox-1.

Assessment of spacecraft compliance with Requirements 4.3-1 and 4.3-2 (per DAS v2.0):
- **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

There are no intentional breakups scheduled during on orbit operation. We are aware of no known potential causes of spacecraft breakup during deployment and mission operations.

Potential causes of spacecraft breakup during deployment and mission operations:
- There is no credible scenario that would result in spacecraft breakup during normal deployment and operations.

Summary of failure modes and effects analyses of all credible failure modes which may lead to an accidental explosion:
- The battery safety systems discussed in the FMEA (see requirement 4.4-1 below) describe the combined faults that must occur for any of nine independent, mutually exclusive failure modes that could lead to a battery venting. If the batteries fail, they are expected to vent gas rather than explode.

Detailed plan for any designed spacecraft breakup, including explosions and intentional collisions:
- There are no planned intentional breakups by explosion, collision, nor by any other means.

List of components which shall be passivated at End of Mission (EOM) including method of passivation and amount which cannot be passivated:

- There are two components that will be passivated at EOM. One is the propulsion unit: all propellant will be exhausted, in a direction to minimize the likelihood of close approach to other satellites. Next, the ground station will send a command to initiate EOM. All on-board components with the exception of the TNC (terminal node controller) will be disconnected from their respective power bus (i.e. the bus will be shut down through a command from the flight computer to the EPS computer). After confirmation is received in the ground, the EPS will disconnect the bus that carries power to the TNC, effectively and permanently disabling any communications to and from Prox-1. Without active attitude control, Prox-1 is in a power-negative state, and the nominal mission power draw from an idle flight computer will drain the batteries in approximately 6 days.

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

- It was deemed unnecessary to passivate the lithium ion batteries for EOM, as the satellite will break up on re-entry at the end of the mission.

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: 0.0001.

Supporting Rationale and FMEA details:

Battery explosion:

Effect:

- All failure modes below might result in battery explosion with the possibility of orbital debris generation. However, in the unlikely event that a battery cell does explosively rupture, the small size, mass, and potential energy, of these small batteries is such that while the spacecraft could be expected to vent gases, debris from the battery rupture will be contained within the vessel due to the lack of penetration energy.

Probability:

- Extremely Low. It is believed to be less than .0001 given 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 if the feasibility 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: Overcharging and excessive charge rate.

Mitigation 3: The satellite bus battery charging circuit design eliminates the possibility of the batteries being overcharged if circuits function nominally. This circuit has been proto- qualification tested for survival in shock, vibration, and thermal-vacuum environments. The charge circuit disconnects the incoming current when battery voltage indicates normal full charge.

Failure Mode 4: 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 must all occur to enable this failure mode.

Failure Mode 5: Inoperable vents. *Mitigation 5:* Battery vents are not inhibited by the battery holder design or the spacecraft. *Combined effects required for realized failure:* The manufacturer fails to install proper venting.

Failure Mode 6: 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 7: 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 failures in environmental tests must occur to result in this failure mode.

Failure Mode 8: 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** the PTC device must fail **AND** over- current monitoring and control must all fail for this failure mode to occur.

Failure Mode 9: Polarity reversal due to over-discharge caused by continuous load during periods of negative power generation vs. consumption.

Mitigation 9: In nominal operations, the spacecraft EPS design negates this mode because the processor will stop when voltage drops below limits. This disables ALL connected loads, creating a guaranteed power-positive charging scenario. The spacecraft will not restart or connect any loads until battery voltage is above the acceptable threshold. At this point, only the safemode processor and radio receiver are enabled and charging the battery. Once the battery reaches 90% of the peak voltage, it will switch to nominal mode and will be able to receive ground commands for continuing mission functions.

Combined faults required for realized failure: The microcontroller must stop executing code **AND** significant loads must be commanded/stuck "on" **AND** power margin analysis must be wrong **AND** the charge control circuit must 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:

- Prox-1 battery charge circuits include overcharge protection and 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 v2.02, and calculation methods provided in NASA-STD-8719.14, section 4.5.4):

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

  Large Object Impact and Debris Generation Probability: 0.0001; COMPLIANT.

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

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

ODAR Section 6: Assessment of Spacecraft Postmission Disposal Plans and Procedures

6.1 Description of spacecraft disposal option selected:

- The satellite will de-orbit with the aid of an electromagnetic tether (Tethers Unlimited Ltd. Nanosat Terminator Tape™), installed in the bottom plate of the satellite. The unit has an autonomous clock that will deploy the tether after 6 months of operation. The satellite will then finalize deorbit by atmospheric re-entry within a 25-year period. There is no propulsive maneuver planned for de-orbit.

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

- None.

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

- **Spacecraft Mass:** 66.4 kg
- **Cross-sectional Area:** 0.17 m^2 (Calculated for frontal area, nominal attitude with respect to velocity vector).
- **Area to mass ratio:** 0.17/66.4 = 0.002565 m^2/kg

6.4 Assessment of spacecraft compliance with Requirements 4.6-1 through 4.6-5 (per DAS v 2.0 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 Prox-1 satellite reentry is COMPLIANT using Method “a”. Prox-1 will re-enter within 25 years, as shown by analysis provided by the electrodynamic tether’s vendor (see AIAA 00-0329 by Hoyt and Forward). Additionally, an equivalent aerodynamic “drag” force was created to match that of the average force exerted by the electromagnetic tether. Given a new aerodynamic drag coefficient derived from such equivalent force, DAS v2.0 showed that Prox-1 would deorbit in 6.4 years after the tether is deployed. A graphical result is shown below.

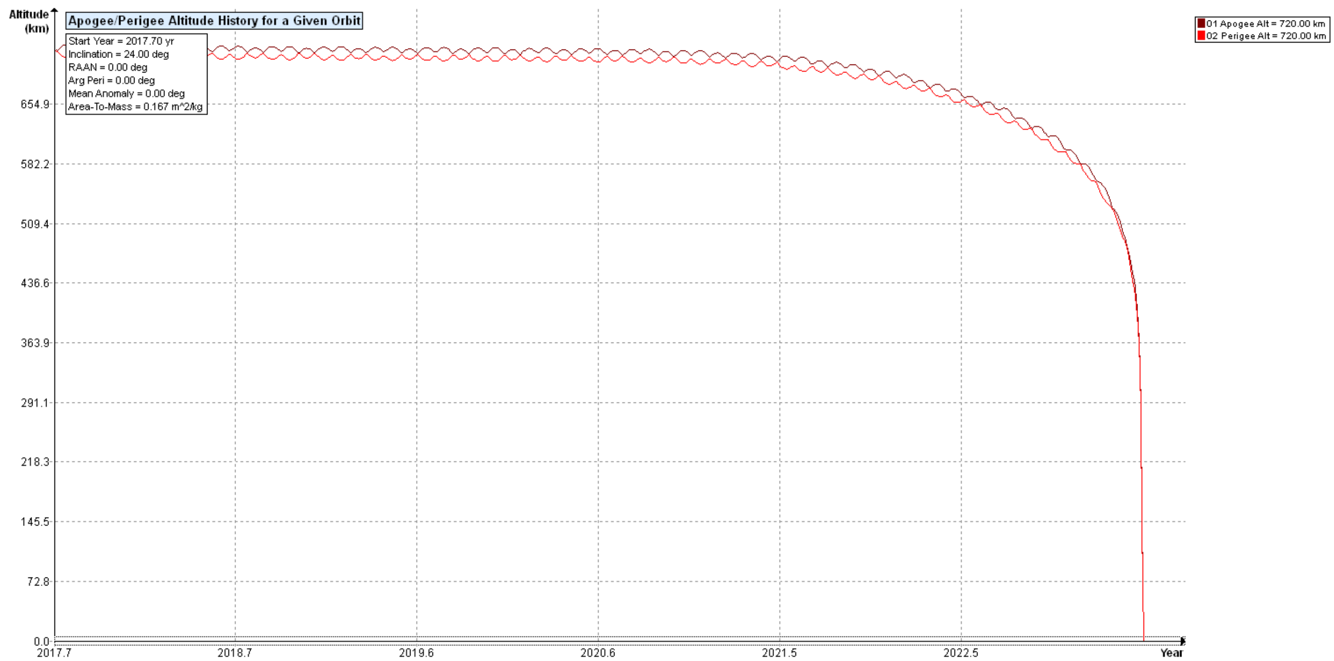

Figure 2: Prox-1 orbit history, assuming an aerodynamic drag force equivalent to that exerted by the electrodynamic tether.

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

Analysis: Not applicable. Prox-1 orbit is LEO.

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

Analysis: Not applicable. Prox-1 orbit is LEO.

Requirement 4.6-4. Reliability of Postmission Disposal Operations

Analysis: Prox-1 relies on a Tethers Unlimited electrodynamic tether to provide post-mission disposal operations. The tether is model "MicroSat Terminator Tape" and is widely used on ESPA-class satellites. The tether element is provided by the Program Sponsor (University Nanosatellite Program) and will be installed at Kirtland Air Force Base prior to environmental testing. The tether mechanism is completely autonomous, and does not rely upon Prox-1 aliveness for deployment.

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 reports that Prox-1 is compliant with the requirement. It predicts that only portions of the mechanical structure will survive reentry and reach the ground. The computed risk of human casualty is 0.00003 (1:36,300), with a total debris casualty area of 1.24 m^2. Note that only the most massive components of the spacecraft were considered for this analysis, as all other lightweight components will burnup/breakup more rapidly due to exposure to same thermal loading.

Object	Compliance Status	Risk of Human Casualty	Sub Component	Demise Altitude (km)	Total Debris Casualty Area (m2)	Kinetic Energy (J)
Prox-1	Compliant	1:42000			1.24	
			Torque Rods	60.5	0.000000	0.000000
			CMG	69.2	0.000000	0.000000
			Board Stacks	73.3	0.000000	0.000000
			TNC	73.0	0.000000	0.000000
			Propulsion Unit	72.9	0.000000	0.000000
			Mechanical Structure	0.0	1.24	47239

Table 1: Summary of results for verification analysis of Requirement 4.7-1.

Requirements 4.7-1b and 4.7-1c below are non-applicable requirements because Prox-1 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 7A: Assessment of Spacecraft Hazardous Materials

Not Applicable. There are no hazardous materials contained on the spacecraft.

ODAR Section 8: Assessment for Tether Missions

(Requirement 4.8-1) - Mitigate the collision hazards of space tethers in Earth or Lunar orbits:

Intact tether systems in Earth and lunar orbit shall meet the requirements limiting the generation of orbital debris from on-orbit collisions (Requirements 4.5-1 and 4.5-2) and the requirements governing postmission disposal (Requirements 4.6-1 through 4.6-4) to the limits specified in those paragraphs. Due to the potential of tether systems being severed by orbital debris or meteoroids, all possible remnants of a severed tether system shall be compliant with the requirements for the collision, debris, and disposal of space structures.

Analysis: DAS v2.0 was used to verify this requirement. The table below represents the output for the DAS simulation given tether specifications from the tether's manufacturer. All pertinent requirements were successfully met.

Tether State	Requirement 4.5-1	Requirement 4.5-2	Requirement 4.6	Orbital Decay (days)
Failed EM	Compliant	Compliant	Compliant	2588.0
Successful EM	Compliant	Compliant	Compliant	2588.0