1HOPSat Formal Orbital Debris Assessment Report (ODAR) and End of Mission Plan (EOMP)

In accordance with NASA's NPR 8715.6A, this report is presented as compliance with the required reporting format per NASA-STD-8719.14.

Note: This analysis only covers the satellite bus and payload orbital debris issues.

No analysis is implied for the launch vehicle or other systems.

Report Version: 5/2/19

Document Data is Not Restricted.

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

DAS Software Version Used In Analysis: v2.0.2

This document is a part of the 1HOPSat-TD Satellite Project Documentation, which is controlled by the Hera Systems, Inc. Project Configuration Manager, San Jose, California.

PREPARED BY:

David D. Squires
VP, Space Systems
Hera Systems, Inc.

Record of Revisions				
Rev	**Date**	**Affected Pages**	**Description of Change**	**Author (s)**
A	2/15/16	All	Initial Release; Preliminary ODAR	David D. Squires
B	3/3/16	2,3,5,6,8,9,19-37	Included Titanium parts in reentry analysis. Corrected typo's and clarified use of propulsion during disposal.	David D. Squires
C	8/28/18	All	Updated to reflect technology demonstration (TD) spacecraft descriptions.	David D. Squires
D	5/2/19	All	Updated to reflect launch and orbit change of the technology demonstration (TD) spacecraft only. Removed future constellation launches and spacecraft.	David D. Squires

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Self-Assessment and OSMA Review of ODARs (per Appendix A.2 of NASA-STD-8719.14):

Per NASA-STD-8719.14 and NPR-8716.5, paragraph 2.2:

> Each delivered ODAR will be reviewed by the OSMA and by the Space Operations Mission Directorate with technical assistance from the NASA ODPO. After the OSMA review, the check sheet … will be returned to the Headquarters Sponsoring Mission Directorate Program Executive for distribution back to the program. OSMA will also provide a copy to the orbital debris lead at the Center supporting the program for assisting with corrective actions.
>
> Each EOMP is reviewed by OSMA with technical assistance from the NASA Orbital Debris Program Office (ODPO) with final approval and all associated risks accepted by the Associate Administrator of the Mission Directorate sponsoring the mission.

A self-assessment of ODAR and EOMP compliance is provided below (next page) in accordance with the assessment formats provided in Appendix sections A.2, and B.2 of NASA-STD-8719.14 The matrices in the NPR are identical and therefore only a single matrix is provided in this combined ODAR-EOMP report. A copy of the assessment may be included in Appendix C for use in if provided by OSMA.

The 1HOPSat project notes that the ODAR is initially due prior to PDR, and the EOMP is initially due, much later, at the Safety and Mission Success review (SMSR). Accordingly, content in the initial release of this document should be viewed as ODAR-driven content, and any modified version of this document released for SMSR will reflect changes to EOM planning. The final ODAR and EOMP document will reflect any inputs or change requirements received from OSMA.

ODAR and EOMP Self-Assessment Report Evaluation: 1HOPSat Mission

Requirement #	Launch Vehicle (NA, see note 1)				Spacecraft			Comments
	Compliant	Not Compliant	Incomplete	Standard Non-Compliant	Compliant	Not Compliant	Incomplete	
4.3-1.a	☐	☐	☒	☐	☒	☐	☐	No Intentional release of debris
4.3-1.b	☐	☐	☒	☐	☒	☐	☐	No Intentional release of debris
4.3-2	☐	☐	☒	☐	☒	☐	☐	N/A - LEO
4.4-1	☐	☐	☒	☐	☒	☐	☐	Explosion probability is estimated at 0.000096 (Requirement: <0.001).
4.4-2	☐	☐	☒	☐	☒	☐	☐	
4.4-3	☐	☐	☒	☐	☒	☐	☐	No intentional break-up planned
4.4-4	☐	☐	☒	☐	☒	☐	☐	No intentional break-up planned
4.5-1	☐	☐	☒	☐	☒	☐	☐	Prob of large object collision using DAS 2.0.2 = 0.000001 (< 0.001)
4.5-2	☐	☐	☒	☐	☒	☐	☐	Prob of small object collision using DAS 2.0.2 = 0.000000 (< 0.01)
4.6-1(a)	☐	☐	☒	☐	☒	☐	☐	Natural reentry within 14 years of EOM.
4.6-1(b)	☐	☐	☒	☐	☒	☐	☐	N/A
4.6-1(c)	☐	☐	☒	☐	☒	☐	☐	N/A
4.6-2	☐	☐	☒	☐	☒	☐	☐	N/A - LEO
4.6-3	☐	☐	☒	☐	☒	☐	☐	N/A - LEO
4.6-4	☐	☐	☒	☐	☒	☐	☐	
4.6-5	☐	☐	☒	☐	☒	☐	☐	
4.7-1	☐	☐	☒	☐	☒	☐	☐	DAS 2.02 reports “return status” as “Passed”. RHC 1:33,100. Total CDA: 1.26 m^2. Objects reaching Earth with > 15 Joules: Bulkhead and Bench. Note: These items are required for fine thermal and mechanical stability of our imaging payload.
4.8-1					☒	☐	☐	N/A – No Tethers

Note 1: The 1HOPSat-TD satellite is a secondary payload, and the launch vehicle is not managed by Hera Systems. Hera Systems will therefore not analyze LV debris.

Assessment Report Format:

ODAR and EOMP Section Format Requirements:

The ODAR and EOMP follow the formats prescribed in NASA-STD-8719.14, Appendix A.1 and B.1 respectively. Required content is provided for each “ODAR section…” 2 through 8 below for the 1HOPSat. ODAR sections 9 through 14 of the NASA Standard are not covered here as they apply to the launch vehicle. The EOMP section uses the ODAR as a primary basis of compliance information.

Sections provided below are labeled according to ODAR and EOMP Section Numbering.

Mission Description:

Hera Systems Inc. will launch one 1HOPSat-TD spacecraft in a launch window opening on July 31, 2019, and closing in September 2019. This spacecraft will launch to an altitude of 555 km and inclination of 37 degrees.

During launch, the satellite will be contained in a 12U CubeSat payload dispenser attached to the upper stage of the launch vehicle. The 12U dispensers provide full enclosure of the satellite until deployment in orbit. After deployment and prescribed time delays, a hatch panel will open to allow light into the imager aperture, deploy antennas, and reorient a small solar panel. Imaging and communications will begin after this hatch is opened. There will be no propulsion on the 1HOPSat-TD spacecraft. Pointing control is provided by precise attitude determination and control systems. A GPS unit is included for accurate orbit location. The 1HOPSat-TD spacecraft orbit will decay naturally from 555 km.

The satellite contains an imaging telescope payload for recording images and video of customer-specified regions of the Earth at one (1) meter ground sample distance (GSD). The collected images will be transmitted to Earth through multiple ground stations over a single carrier, OQPSK, X-band radio link. Commanding, telemetry, and supplemental image downlink will be implemented with an experimental C-band radio using 802.11n (OFDM) technology. Commanding and telemetry are supplemented with an Iridium™ short burst data (SBD) radio providing low rate data.

Launch vehicle and launch sites:

Satish Dhawan Space Centre (SDSC)

Proposed launch dates:

1HOPSat-TD Launch: August, 15, 2019

Mission duration: The 1HOPSat-TD spacecraft mission is intended to last 6 months. After the end of its mission, the spacecraft will remain in orbit until reentry occurs through natural decay of the orbit.

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

Figure 1 is representative of launch operations for the 1HOPSat-TD spacecraft launch on a proposed PSLV launch vehicle.

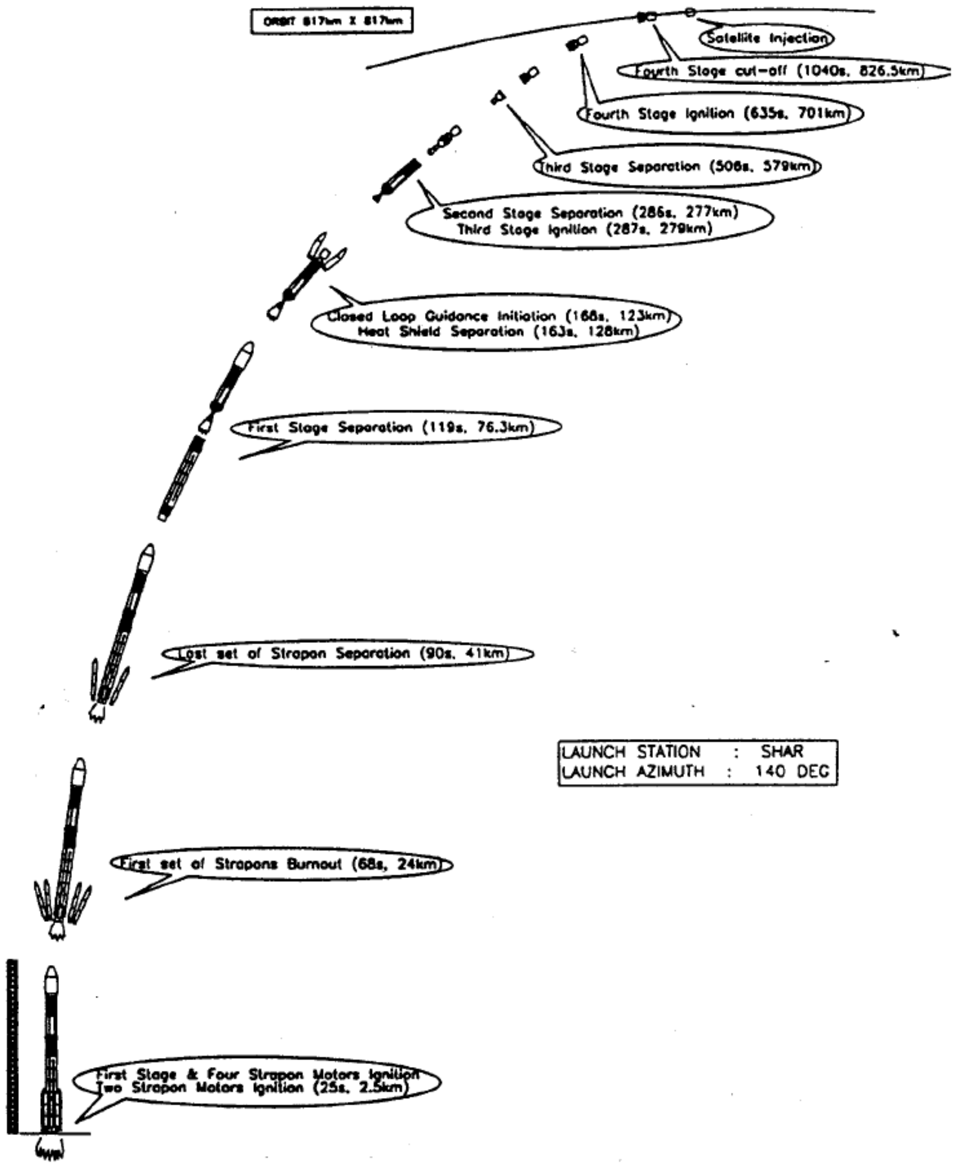

Figure 1, Representative PSLV Launch Sequence (not specific to the 1HOPSat-TD launch)

Secondary payloads including the 1HOPSat-TD will be deployed as coordinated with the primary payload owner.

The upper stage initiates separation events for secondary payloads including 1HOPSat-TD.

Any collision avoidance maneuvers and related separation timing are controlled by the launch operator.

Subsequent to deployment, 1HOPSat-TD will be in a natural orbit without propulsion, and no attempt will be made to modify the orbit by use of drag.

Initial Orbit after Launch:
1HOPSat-TD satellite is deployed to the circular orbit defined below. Imaging and maneuvering operations will take place at this altitude and inclination:
Apogee: 555 km
Perigee: 555 km
Inclination: 37 +/- 0.2 degree

Interaction or potential physical interference with other operational Spacecraft: No intentional interactions are planned. Interferences will be the subject of collision avoidance analysis provided by the launch provider and/or payload dispenser provider.

ODAR/EOMP Section 1: Program Management and Mission Overview

1HOPSats components and main assemblies will be built at Hera Systems Inc. facilities. Th epayload will be built at a contractor facility. Final integration and test of systems will be performed at a contractor facility.

Mission Directorate: Not applicable. 1HOPSat-TD is a commercial satellite.

Program Executive: Roger Roberts, PhD; CEO

Program/project manager: David. D. Squires, VP of Space Systems

Senior scientist: Not applicable. 1HOPSat-TD is a commercial satellite.

Foreign government or space agency participation: India is the provider of the PSLV launch vehicle.

Summary of NASA's responsibility under the governing agreement(s): Not applicable. There is no NASA involvement in these commercial launches.

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

Mission Selection:	June, 2018
Mission Preliminary Design Review:	June, 2018
Mission Critical Design Review:	July 2018
FRR:	December 2018
PSRR:	January, 2019
ORR:	July, 2019
Launch:	August, 2019
Primary Mission Complete	1HOPSat-TD: L+ 6months
Extended Mission Complete	1HOPSat-TD: (TBD)

ODAR/EOMP Section 2: Spacecraft Description

Physical description of the spacecraft:
The 1HOPSat-TD satellite is a 12U CubeSat rectangular box conforming to common 12U dispenser payload size and mass specifications. The spacecraft has a hatch-door opening to allow light to enter its nadir-pointing imager. Hatch-mounted antennas provide for main X-band downlink transmission, C-band command and telemetry. An Iridium radio provides two-way short burst data for limited command and telemetry. The satellite dimensions are 22.6 cm x 22.6 cm x 34.0 cm. The satellite is constructed primarily of aluminum with some subsystem components and fasteners made of printed circuits, stainless steel, copper, plastics, optical glass, and titanium. Titanium components provide benefits to thermal management and thermal-mechanical stability of the imager.

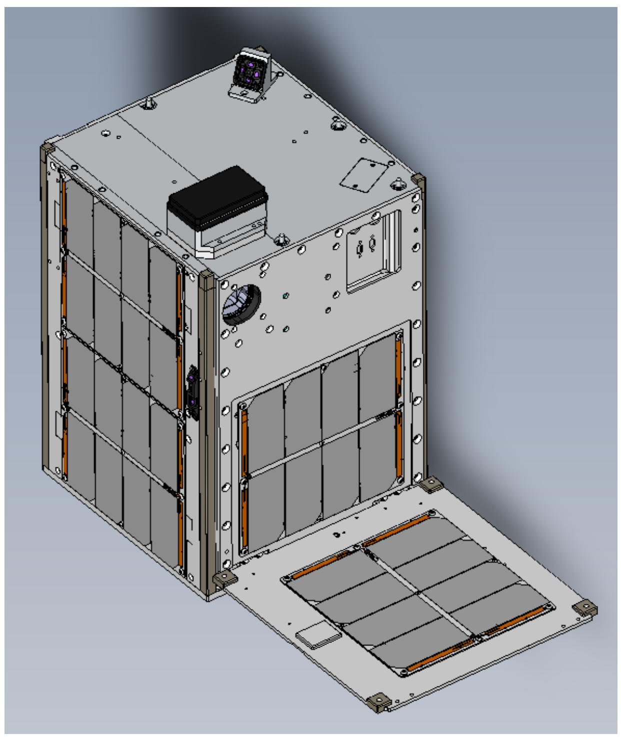

Figure 3, Hera 1HOPSat-TD 12U Satellite Configuration

Total satellite mass at launch: 19 kg.
Dry mass of satellite at launch, excluding solid rocket motor propellants: 19 kg; no propulsion.

Description of all propulsion systems (cold gas, mono-propellant, bi-propellant, electric, nuclear): There is no propulsion on the 1HOPSat-TD spacecraft.

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:
There are no fluids planned to be onboard the spacecraft.

Fluids in Pressurized Batteries: None. The 1HOPSat-TD satellite 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:
The 1HOPSat_TD satellite uses an integrated ADCS system. Normal attitude for 1HOPSat-TD is to present its smallest cross-section to the velocity vector direction. This orientation is used for periods of the orbit when imaging and high-rate radio communications are in not in use.

Description of any range safety or other pyrotechnic devices: No pyrotechnic devices are used.

Description of the electrical generation and storage system: 29.5% efficient triple-junction Solar cells generate power for storage in Lithium-Ion Batteries.

Identification of any other sources of stored energy not noted above: None.

Identification of any radioactive materials on board: None**.**

ODAR/EOMP 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
>
> (Note that Hera Systems, Inc. does not manage the release of staging components, deployment hardware, or other objects).
>
> **4.3-2, Mission Related Debris Passing Near GEO:** COMPLIANT

ODAR/EOMP 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 nine (9) independent, mutually exclusive failure modes to lead to explosion.

Detailed plan for any designed spacecraft breakup, including explosions and intentional collisions:

Not applicable. 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:

Lithium Ion batteries shall be passivated at EOM. This will be done using accelerated cycles of battery charge-discharge. The accelerated charge-discharge cycles are implemented by commanding payload and bus system loads to remain ON, thereby increasing power consumption. A few percent of chargeable capacity (<20 kJ) could remain in the batteries at the end of the passivation cycling.

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

Not applicable.

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: 0.000096

Supporting Rationale and FMEA details:

Propellant Tank Sealed Container Failure:
The nominal propellant tank pressure is 14.7 PSIa. At this pressure, the tanks are considered to be “sealed containers” and not pressure vessels. This contained pressure is considered to be insufficient to cause catastrophic failure of the vessel.

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, most debris from the battery rupture should be contained within the vessel due to the lack of penetration energy.

Probability: Very Low. It is estimated to be much less than 0.001 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: Complete proto-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.

Expected Probability: ~0.000012 based on millions of cells in circulation (we will use 10 million as a baseline). This battery type also has many cell-decades of demonstrated reliability in space, increasing the probability of acceptable performance of this design. However, cell-years in orbit are not considered in the calculation, but should add confidence in the reliability estimate. An overall derating factor of 10 is applied to account for space environment effects.

Hence, given that the spacecraft uses 12 cells: Pf = 0.0000001*10*12 = 0.000012 (per spacecraft)

Failure Mode 2: Internal thermal rise due to high load discharge rate.
Mitigation 2: The cell array is in series with three (3) MOSFET transistors and two (2) current sensing resistors. In the case of over-discharge current, the active protection circuit will drive all three MOSFETs to a high resistance state (OFF).

Considering the case of a failure of the battery protection circuit **AND** failure of a down-stream power client AND failure of the downstream regulator **AND** failure of a downstream current measurement / turn off circuitry. A "race to failure" condition will exist. The candidate components for first to fail producing a steady state are 3 MOSFETs and 2 current sense resistors and the battery array, conservatively ignoring a similar protection configuration downstream (e.g. if perhaps a wire failure causes a short). A working over-current is 20.4 A which is derived from the formula: 2C * 3 = 3.4 A * 2 * 3 = 20.4A. The power dissipations of the candidate components are:

Component	Power	Rating
0.01 OHM	4.16 W	1 W
0.02 OHM	8.3 W	1 W
mosfet1 (on)	0.624 W	2.3 W
mosfet 1 (off)	416 W	2.3 W
mosfet 2 (on)	2.08 W	2.3 W
mosfet 2 off	416 W	2.3 W
mosfet 3 (on)	2.08 W	2.3 W

The above table suggests a cascade of failures. The turned off MOSFETs will fail and then the 0.02 OHM resistor will fail. Again considering the worst case as MOSFETs fail short, the resistor will fail open leading to steady state zero current. Since the 2C current assumption is within the rated short term safe operation range of the batteries, the possibility of battery explosion by over-current discharge is vanishingly small.

Combined faults required for realized failure: The spacecraft thermal design must be incorrect **AND** external over current detection and disconnect function must fail **AND** the down-stream power client must fail **AND** the downstream regulator **AND** downstream current measurement / turn off circuitry must fail **AND** 3 MOSFETs must fail short **AND** two (2) current sense resistors must NOT fail **AND** the batteries must fail within their rated capacity to enable this failure mode.

Expected Probability: ~0.000012 based on millions of cells in circulation (we will use 10 million as a baseline) and discharge rate limit protection. This battery type also has many cell-decades of demonstrated reliability in space, increasing the probability of acceptable performance of this design. However, cell-years in orbit are not considered in the calculation, but should add confidence in the reliability estimate. An overall derating factor of 10 is applied to account for space environment effects.

Hence, given that the spacecraft uses 12 cells: Pf = 0.0000001*10*12 = 0.000012 (per spacecraft)

Failure Mode 3: Overcharging and excessive charge rate.
Mitigation 3: The satellite bus battery charging circuit design and Battery Protection Circuit design and program eliminates the possibility of the batteries being overcharged if circuits function nominally. For the charging circuit failure to be realized each of the 3 protection circuits of the charger must fail. Which are 1) output current feedback; 2) battery current feedback 3) thermal feedback. In addition the battery protection module must fail as described in **Failure Mode 2.**

This circuit is 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 at 4.2V per series cell. If this circuit fails to operate, continuing charge can cause gas generation. The batteries include overpressure release vents that allow gas to escape, virtually eliminating any explosion hazard.

Combined faults required for realized failure:

1) For overcharging: The charge control circuit must fail to function **AND** the battery protection circuit must fail **AND** the overpressure relief device must be inadequate to vent generated gasses at acceptable rates to avoid explosion.

2) For excessive charge rate: Based on dynamic analysis of sun pointing behavior, the solar arrays are capable of generating a maximum of 5.4 Amps. This is equivalent to 0.53C battery charge rate for the three (3) parallel strings of battery cells. If all of this current went into charging batteries, the resultant charge rate would be well below the recommended 0.7C nominal charging rate for the Panasonic NCR18650B type batteries used. For this failure mode to become active, it is therefore likely that two strings of batteries would have to fail to accept a charge **AND** the all spacecraft and payload loads must be off **AND** the charge control circuit on the remaining string must fail such that it allows charging below 11.6 volts (4-cell series voltage) **AND** the battery protection module must fail **AND** the overpressure relief vent must be inadequate to relive generated gas.

Estimated Probability: ~0.000012 based on millions of cells in circulation (we will use 10 million as a baseline), quadruple fault protection of proven devices for overcharge protection, and zero probability of exceeding charge rate limit due to absence of power generation. This battery type also has many cell-decades of demonstrated reliability in space, increasing the probability of acceptable performance of this design. However, cell-years in orbit are not considered in the calculation, but should add confidence in the reliability estimate. An overall derating factor of 10 is applied to account for space environment effects.

Hence, given that the spacecraft uses 12 cells: Pf = 0.0000001*10*12 = 0.000012 (per spacecraft)

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) proto-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: The battery protection module must fail as described in **Failure Mode 2** **AND** an external load must fail/short-circuit **AND** over-current detection and disconnect function must all fail in order to enable this failure mode.

Estimated Probability: ~0.000012 based on millions of cells in circulation (we will use 10 million as a baseline to account for standard protection built into each cell), and triple fault of proven devices for excessive discharge protection. This battery type also has many cell-decades of demonstrated reliability in space, increasing the probability of acceptable performance of this design. However, cell-years in orbit are not considered in the calculation, but should add confidence in the reliability estimate. An overall derating factor of 10 is applied to account for space environment effects.

Hence, given that the spacecraft uses 12 cells: Pf = 0.0000001*10*12 = 0.000012 (per spacecraft)

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.

Expected Probability: ~0.000012 based on millions of cells in circulation (we will use 10 million as a baseline). This battery type also has many cell-decades of demonstrated reliability in space, increasing the probability of acceptable performance of this design. However, cell-years in orbit are not considered in the calculation, but should add confidence in the reliability estimate. An overall derating factor of 10 is applied to account for space environment effects.

Hence, given that the spacecraft uses 12 cells: Pf = 0.0000001*10*12 = 0.000012 (per spacecraft)

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.

Expected Probability: 0.000001 as calculated by DAS 2.0.2 in requirement 4.5-1.

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.

Expected Probability: ~0.000012 based on millions of units in circulation (we will use 10 million as a baseline). This battery type also has many cell-decades of demonstrated reliability in space, increasing the probability of acceptable performance of this design. However, cell-years in orbit are not considered in the calculation, but should add confidence in the reliability estimate. An overall derating factor of 10 is applied to account for space environment effects.

Hence, given that the spacecraft uses 12 cells: Pf = 0.0000001*10*12 = 0.000012 (per spacecraft)

Failure Mode 8: Excess temperatures due to orbital environment and high discharge combined for the hottest orbit.

Mitigation 8: The spacecraft thermal design negates this possibility as demonstrated in the NASA O/OREOS mission which used the same cell types and similar current loading during full sun orbits totaling roughly 13 weeks in 3.5 years of operations without failure. 1HOPSat will not experience this extreme condition for its propulsively maintained sun-synchronous orbit, nor for its lower inclination orbit(s).

Thermal rise has also been analyzed in the context of the mission space environment temperatures. Battery temperatures are expected to be well below temperatures of concern for explosions. The maximum battery temperature is estimated to be just below

30 ^{o}C, allowing an operational temperature margin of 15 ^{o}C relative to the datasheet recommended maximum of 45 ^{o}C during charging. The margin during discharge is 30 ^{o}C relative to a datasheet recommended maximum of 60 ^{o}C.

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.

Expected Probability: ~0.000012 based on millions of units in circulation (we will use 10 million as a baseline) and discharge rate limit protection. This battery type also has many cell-decades of demonstrated reliability in space, increasing the probability of acceptable performance of this design. However, cell-years in orbit are not considered in the calculation, but should add confidence in the reliability estimate. An overall derating factor of 10 is applied to account for space environment effects.

Hence, given that the spacecraft uses 12 cells: Pf = 0.0000001*10*12 = 0.000012 (per spacecraft)

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 EPS processor will stop when voltage drops too low. This disables ALL connected loads, creating a guaranteed power-positive charging scenario. In addition the battery protection module senses battery voltage and disables discharge. The spacecraft will not restart or connect any loads until battery voltage is above the acceptable threshold. At this point, only the main OBCS board, EPS board, CC&T radios, and ADCS in low-power Safe Mode are enabled, maintaining a power positive mode until ground commands are received 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 battery protection module must fail **AND** the charge control circuit must fail for this failure mode to occur.
Expected Probability: ~0.000012 based on millions of units in circulation (we will use 10 million as a baseline. This battery type also has many cell-decades of demonstrated reliability in space, increasing the probability of acceptable performance of this design. Cell-years are not considered in that calculation, but should add confidence in the reliability estimate. An overall derating factor of 10 is applied to account space environment effects.

Hence, given that the spacecraft uses 12 cells: Pf = 0.0000001*10*12 = 0.000012

Requirement 4.4-2: Design for passivation after completion of mission operations while in orbit about Earth or the Moon:

Design of all spacecraft and launch vehicle orbital stages shall include the ability to deplete all onboard sources of stored energy and disconnect all energy generation sources when they are no longer required for mission operations or post mission disposal or control to a level which cannot cause an explosion or deflagration large enough to release orbital debris or break up the spacecraft (Requirement 56450).

Compliance statement: The only significant stored energy is in the battery packs. If desired prior to reentry at EOM, energy storage capacity in the Lithium Ion batteries can be degraded more rapidly than normal through application of repeated deep depth-of-discharge cycles (cycling between 60% and 90% depth of discharge). This function is enabled when a command is sent to increase power consumption in the bus and payload. This results in an accelerated number of charge-discharge cycles per day. A few percent of chargeable capacity (<20 kJ) could remain in the batteries at the end of the passivation cycling. It is predicted that the chargeable capacity can be dropped to this level in less than 2 years after the command is issued.

Requirement 4.4-3. Limiting the long-term risk to other space systems from planned breakups:

Compliance statement:
There are no planned breakups.

Requirement 4.4-4: Limiting the short-term risk to other space systems from planned breakups:

Compliance statement:
There are no planned breakups.

<u>ODAR/EOMP Section 5: Assessment of Spacecraft Potential for On-Orbit Collisions</u>
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:** 0.000001; 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 post mission disposal requirements is less than 0.01 (Requirement 56507).

- **Small Object Impact and Debris Generation Probability:** 0.000000; COMPLIANT
- **Identification of all systems or components required to accomplish any post mission disposal operation, including passivation and maneuvering:**

Critical surface1: Iridium Radio used to Control Passivation
1HOPSat can passivate its battery pack at end of mission through use of a command that caused repeated deep discharge cycles, resulting in loss of useable charge capacity. The spacecraft bus and payload contain circuits that must execute or support (as loads) the

battery passivation functions. However, the Iridium radio is the most vulnerable component supporting this function. (Note that redundancy of this function can be provided by the C-band radio as well.) For the Iridium radio, the integrated circuits that control the passivation functions are on printed circuit cards within the spacecraft bus frame. These integrated circuits have negligible areal density associated mainly with the plastic encapsulant, circuit card material, and conformal coating surrounding the semiconductor chips. To be highly conservative, this analysis considers the protective benefit of only the exposed areal density of the plastic encapsulant behind the outer aluminum walls of the Iridium radio housing. The plastic penetrability is estimated using polycarbonate plastic with 1250 kg/m^3 density. Assuming 0.5 mm thickness and a total of 2 cm^2 surface area for the devices of concern, mass of 0.125g, and areal density of 0.0625 g/cm^2 are estimated. The closest distance of this surface to the spacecraft outer wall panels is approximately 3cm.

Critical Surface 2: Battery Cells/Battery Pack outer layers
If one of the cells in a battery pack became disabled due to meteoroid impact, then passivating one of the series-connected cells would be prevented. Each battery cell has attributes as provided in figure 4. There are twelve (12) cells in all. The cells are contained behind the external panels of the spacecraft (described above). Surface area per cell is 43.5 cm^2. Mass per cell is 44.5 grams. Hence the per-cell areal density may be seen as 1.02 g/ cm^2. But, estimating that failure might be induced at meteoroid penetration depth of roughly one tenth the cell diameter, the effective areal density used will be (1/100)*1.02 g/cm^2, or 0.0102 g/cm^2. The closest distance of this surface to the spacecraft outer wall panels is approximately 1cm.

Note that additional surfaces were evaluated in DAS 2.0.2 to investigate the probability of losing loads that might be used for passivation, or propellant that can be used for reentry management. Critical surfaces for these systems are defined similarly to Critical Surface 1 and 2, but are not directly tied to the failure of passivation function.

Specifications

Rated capacity[1]	Min. 3200mAh
Capacity[2]	Min. 3250mAh Typ. 3350mAh
Nominal voltage	3.6V
Charging	CC-CV, Std. 1625mA, 4.20V, 4.0 hrs
Weight (max.)	48.5 g
Temperature	Charge*: 0 to +45°C Discharge: -20 to +60°C Storage: -20 to +50°C
Energy density[3]	Volumetric: 676 Wh/l Gravimetric: 243 Wh/kg

[1] At 20°C [2] At 25°C [3] Energy density based on bare cell dimensions

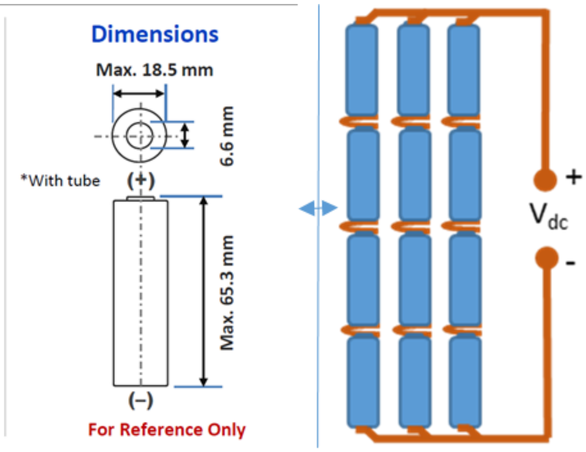

Figure 4: (left) 1HOPSat-TD Battery Cell Specifications (1 of 12); (right) battery pack wiring.

Outer walls:
At a minimum, critical surfaces are surrounded on all sides by aluminum housing or panels made of 7075-T6 Aluminum. The thinnest aluminum areas are provide effective areal density of at least 1.3 g/cm^2 (ignoring solar cell contributions) as seen from the location of critical surfaces. In some cases an effective density of many times this value may be seen for surfaces that are intermediated by payload walls and/or other structures using 7075-T6, 6061-T6 aluminum, optical glass, and grade-5 titanium. Values selected for this analysis appear in the DAS 2.0.2 log file provided in Appendix A.

ODAR/EOMP Section 6: Assessment of Spacecraft Post Mission Disposal Plans and Procedures

6.1 Description of spacecraft disposal option selected: The satellite will de-orbit by natural orbit decay due to drag.

6.2 Plan for any spacecraft maneuvers required to accomplish post mission disposal: No maneuvers are planned to facilitate post-mission disposal.

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

Atmospheric reentry by natural decay of orbit is a fallback if propulsive reentry fails
Spacecraft Mass: 19 kg
Cross-sectional Area: 0.12 m^2 (Calculated by DAS 2.0.2).
Area to mass ratio:
0.12/19 = 0.00632 m^2/kg

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.

Analysis: The 1HOPSat-TD spacecraft is COMPLIANT using method “a.” above. The TD spacecraft has no propulsion, but will reenter the Earth’s atmosphere by natural orbit decay in less than 14 years based on DAS 2.0.2 engineering utility calculations and as illustrated in the figure below.

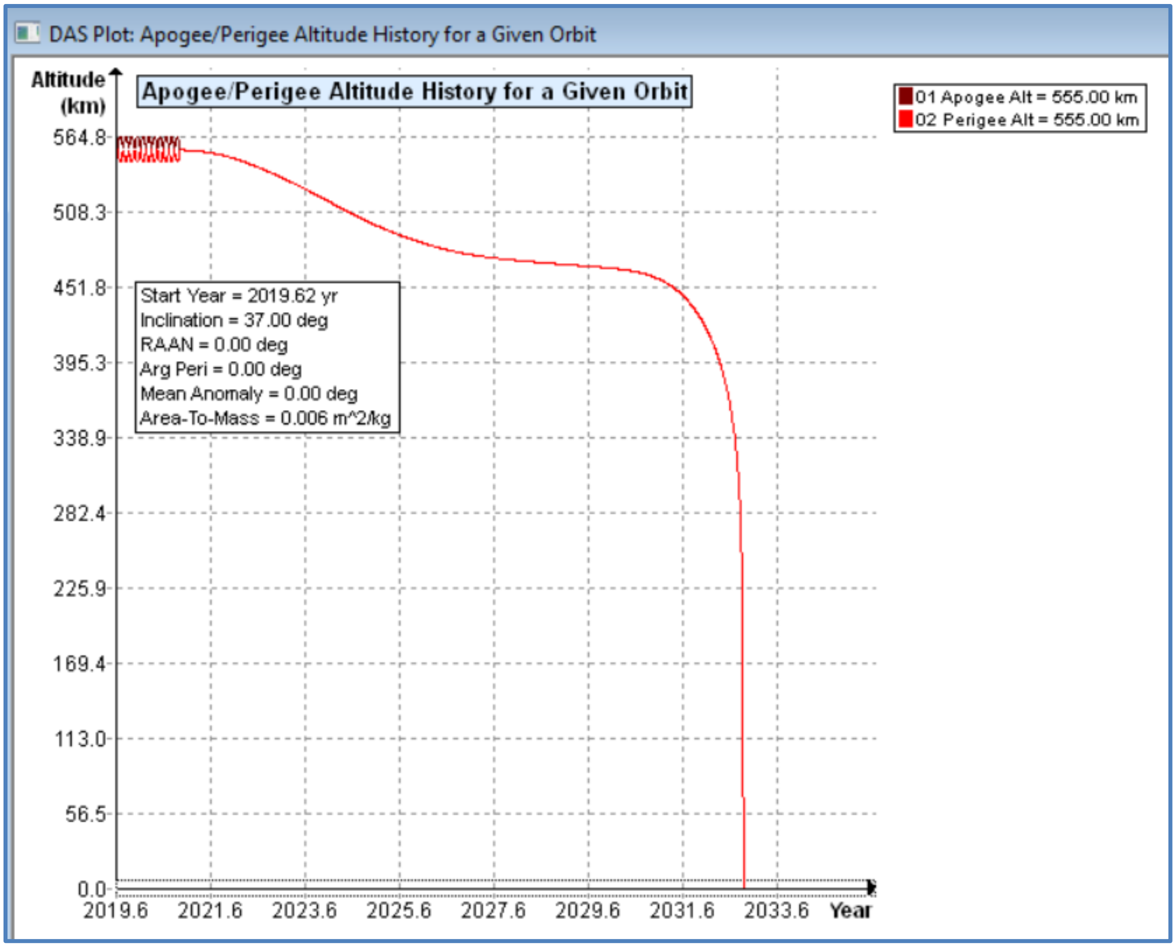

Figure 5: Orbit decay in less than 14 years

Requirement 4.6-2. Disposal for space structures near GEO.
Analysis: Not applicable. 1HOPSat uses LEO.

***Requirement 4.6-3. Disposal for space structures between LEO and GEO*.**
Analysis: Not applicable. 1HOPSat orbit is LEO.

Requirement 4.6-4. Reliability of Post mission Disposal Operations
Analysis: There are no required 1HOPSat-TD post mission disposal operations. The spacecraft can reenter by natural decay of orbit (see Requirement 4.6.1, above).

ODAR/EOMP 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 v2.0.2 reports that 1HOPSat is **COMPLIANT** with the requirement.

Total human casualty probability is reported by the DAS software as 0.0000302 (1:33,100).

> Note: The objects arriving at the Earth's surface with greater than 15 Joules of energy are the titanium bulkhead and bench structural components. These are required for the thermal-mechanical stability of our imager. Other materials evaluated lacked the thermal stability to maintain imager optical alignments throughout the expected temperature range and gradients expected in orbit.

Requirements 4.7-1b, and 4.7-1c below are non-applicable requirements because 1HOPSat-TD does not implement precise and predictable 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/EOMP Section 7A: Assessment of Spacecraft Hazardous Materials

There are no materials on the spacecraft that are designated as hazardous.

ODAR/EOMP Section 8: Assessment for Tether Missions

Not applicable. There are no tethers in the 1HOPSat mission.

Appendix A: DAS v2.0.2 Analysis Results

05 02 2019; 12:18:54PM DAS Application Started
05 02 2019; 12:18:54PM Opened Project C:\Users\Dave\AppData\Local\NASA\DAS 2.0\project\12U-Tech_demo-2019\
05 02 2019; 12:19:02PM Processing Requirement 4.3-1: Return Status : Not Run

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

=============== End of Requirement 4.3-1 ===============
05 02 2019; 12:19:05PM Processing Requirement 4.3-2: Return Status : Passed

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

=============== End of Requirement 4.3-2 ===============
05 02 2019; 12:19:06PM Requirement 4.4-3: Compliant

=============== End of Requirement 4.4-3 ===============
05 02 2019; 12:19:11PM Processing Requirement 4.5-1: Return Status : Passed

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

INPUT

Space Structure Name = 1HOPSat-TD
Space Structure Type = Payload
Perigee Altitude = 555.000000 (km)
Apogee Altitude = 555.000000 (km)
Inclination = 37.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass Ratio = 0.006320 (m^2/kg)
Start Year = 2019.622000 (yr)
Initial Mass = 19.000000 (kg)
Final Mass = 19.000000 (kg)
Duration = 13.229000 (yr)
Station-Kept = False
Abandoned = True
PMD Perigee Altitude = -1.000000 (km)
PMD Apogee Altitude = -1.000000 (km)

PMD Inclination = 0.000000 (deg)
PMD RAAN = 0.000000 (deg)
PMD Argument of Perigee = 0.000000 (deg)
PMD Mean Anomaly = 0.000000 (deg)

OUTPUT

Collision Probability = 0.000001
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range
Status = Pass

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

=============== End of Requirement 4.5-1 ===============
05 02 2019; 12:25:16PM Requirement 4.5-2: Compliant

==
Spacecraft = 1HOPSat-TD
Critical Surface = Battery_Shells
==

INPUT

Apogee Altitude = 555.000000 (km)
Perigee Altitude = 555.000000 (km)
Orbital Inclination = 37.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass = 0.006320 (m^2/kg)
Initial Mass = 19.000000 (kg)
Final Mass = 19.000000 (kg)
Station Kept = No
Start Year = 2019.622000 (yr)
Duration = 13.229000 (yr)
Orientation = Random Tumbling
CS Areal Density = 0.010200 (g/cm^2)
CS Surface Area = 0.043000 (m^2)
Vector = (0.000000 (u), 0.000000 (v), 0.000000 (w))
CS Pressurized = No
Outer Wall 1 Density: 1.393500 (g/cm^2) Separation: 1.000000 (cm)
Outer Wall 2 Density: 41.000000 (g/cm^2) Separation: 12.000000 (cm)
Outer Wall 3 Density: 14.000000 (g/cm^2) Separation: 3.000000 (cm)
Outer Wall 4 Density: 3.500000 (g/cm^2) Separation: 5.000000 (cm)
Outer Wall 5 Density: 3.000000 (g/cm^2) Separation: 5.000000 (cm)
Outer Wall 6 Density: 3.000000 (g/cm^2) Separation: 5.000000 (cm)

OUTPUT

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

===
Spacecraft = 1HOPSat-TD
Critical Surface = Iridium_Radio
===

INPUT

Apogee Altitude = 555.000000 (km)
Perigee Altitude = 555.000000 (km)
Orbital Inclination = 37.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass = 0.006320 (m^2/kg)
Initial Mass = 19.000000 (kg)
Final Mass = 19.000000 (kg)
Station Kept = No
Start Year = 2019.622000 (yr)
Duration = 13.229000 (yr)
Orientation = Random Tumbling
CS Areal Density = 1.393500 (g/cm^2)
CS Surface Area = 0.012800 (m^2)
Vector = (0.000000 (u), 0.000000 (v), 0.000000 (w))
CS Pressurized = No
Outer Wall 1 Density: 1.393500 (g/cm^2) Separation: 1.000000 (cm)
Outer Wall 2 Density: 50.000000 (g/cm^2) Separation: 15.000000 (cm)

OUTPUT

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

05 02 2019; 12:45:51PM Processing Requirement 4.6 Return Status : Passed

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

INPUT

Space Structure Name = 1HOPSat-TD
Space Structure Type = Payload

Perigee Altitude = 555.000000 (km)
Apogee Altitude = 555.000000 (km)
Inclination = 37.000000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Area-To-Mass Ratio = 0.006320 (m^2/kg)
Start Year = 2019.622000 (yr)
Initial Mass = 19.000000 (kg)
Final Mass = 19.000000 (kg)
Duration = 13.229000 (yr)
Station Kept = False
Abandoned = True
PMD Perigee Altitude = 128.829919 (km)
PMD Apogee Altitude = 128.829919 (km)
PMD Inclination = 36.960266 (deg)
PMD RAAN = 102.375131 (deg)
PMD Argument of Perigee = 204.851773 (deg)
PMD Mean Anomaly = 0.000000 (deg)

OUTPUT

Suggested Perigee Altitude = 128.829919 (km)
Suggested Apogee Altitude = 128.829919 (km)
Returned Error Message = Passes LEO reentry orbit criteria.

Released Year = 2032 (yr)
Requirement = 61
Compliance Status = Pass

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

=============== End of Requirement 4.6 ===============
05 02 2019; 12:46:10PM *********Processing Requirement 4.7-1
Return Status : Passed

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

name = 1HOPSat-TD
quantity = 1
parent = 0
materialID = 9
type = Box
Aero Mass = 19.000000
Thermal Mass = 19.000000
Diameter/Width = 0.260000
Length = 0.340000

Height = 0.260000

name = PVA_Body_Mt_XandY
quantity = 4
parent = 1
materialID = 9
type = Flat Plate
Aero Mass = 0.400000
Thermal Mass = 0.400000
Diameter/Width = 0.260000
Length = 0.340000

name = PVA_Hatch
quantity = 1
parent = 1
materialID = 9
type = Flat Plate
Aero Mass = 0.400000
Thermal Mass = 0.400000
Diameter/Width = 0.260000
Length = 0.260000

name = Nadir_ANT_Panel
quantity = 1
parent = 1
materialID = 9
type = Flat Plate
Aero Mass = 0.400000
Thermal Mass = 0.400000
Diameter/Width = 0.260000
Length = 0.260000

name = ADCS_Box
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 0.250000
Thermal Mass = 0.250000
Diameter/Width = 0.100000
Length = 0.100000
Height = 0.050000

name = Reaction_Wheels
quantity = 4
parent = 1
materialID = 54
type = Cylinder
Aero Mass = 0.100000

Thermal Mass = 0.100000
Diameter/Width = 0.030000
Length = 0.020000

name = OBCS_and_other_PCBS
quantity = 8
parent = 1
materialID = 23
type = Flat Plate
Aero Mass = 0.100000
Thermal Mass = 0.100000
Diameter/Width = 0.100000
Length = 0.100000

name = Payload_Structures
quantity = 4
parent = 1
materialID = 8
type = Box
Aero Mass = 0.150000
Thermal Mass = 0.150000
Diameter/Width = 0.070000
Length = 0.090000
Height = 0.070000

name = Primary_Mirror
quantity = 1
parent = 1
materialID = 71
type = Cylinder
Aero Mass = 2.000000
Thermal Mass = 2.000000
Diameter/Width = 0.200000
Length = 0.060000

name = Secondary_Mirror
quantity = 1
parent = 1
materialID = 71
type = Cylinder
Aero Mass = 0.100000
Thermal Mass = 0.100000
Diameter/Width = 0.060000
Length = 0.020000

name = SC_Structures
quantity = 8
parent = 1
materialID = 9

type = Flat Plate
Aero Mass = 0.750000
Thermal Mass = 0.750000
Diameter/Width = 0.230000
Length = 0.340000

name = Batteries
quantity = 14
parent = 1
materialID = 8
type = Cylinder
Aero Mass = 0.046500
Thermal Mass = 0.046500
Diameter/Width = 0.019000
Length = 0.063000

name = EPS_Boards
quantity = 12
parent = 1
materialID = 23
type = Flat Plate
Aero Mass = 0.015000
Thermal Mass = 0.015000
Diameter/Width = 0.100000
Length = 0.100000

name = Cables_and_Connectors
quantity = 15
parent = 1
materialID = 19
type = Cylinder
Aero Mass = 0.020000
Thermal Mass = 0.020000
Diameter/Width = 0.004000
Length = 0.200000

name = Misc_Fasteners
quantity = 150
parent = 1
materialID = 54
type = Cylinder
Aero Mass = 0.000500
Thermal Mass = 0.000500
Diameter/Width = 0.003000
Length = 0.010000

name = Misc_Brackets
quantity = 20
parent = 1

materialID = 9
type = Flat Plate
Aero Mass = 0.025000
Thermal Mass = 0.025000
Diameter/Width = 0.050000
Length = 0.080000

name = Thermal_Straps
quantity = 4
parent = 1
materialID = 19
type = Flat Plate
Aero Mass = 0.300000
Thermal Mass = 0.300000
Diameter/Width = 0.060000
Length = 0.150000

name = Bulkhead
quantity = 1
parent = 1
materialID = 66
type = Flat Plate
Aero Mass = 0.920000
Thermal Mass = 0.920000
Diameter/Width = 0.200000
Length = 0.200000

name = Struts
quantity = 8
parent = 1
materialID = 66
type = Flat Plate
Aero Mass = 0.040000
Thermal Mass = 0.040000
Diameter/Width = 0.030000
Length = 0.120000

name = Secondary_Support
quantity = 8
parent = 1
materialID = 66
type = Flat Plate
Aero Mass = 0.030000
Thermal Mass = 0.030000
Diameter/Width = 0.020000
Length = 0.200000

name = Torque_Rods
quantity = 3

parent = 1
materialID = 38
type = Cylinder
Aero Mass = 0.050000
Thermal Mass = 0.050000
Diameter/Width = 0.080000
Length = 0.060000

name = Bench
quantity = 1
parent = 1
materialID = 66
type = Flat Plate
Aero Mass = 0.850000
Thermal Mass = 0.850000
Diameter/Width = 0.190000
Length = 0.190000

name = Star_tracker
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 0.300000
Thermal Mass = 0.300000
Diameter/Width = 0.050000
Length = 0.100000
Height = 0.050000

name = Baffles
quantity = 4
parent = 1
materialID = 54
type = Flat Plate
Aero Mass = 0.050000
Thermal Mass = 0.050000
Diameter/Width = 0.120000
Length = 0.200000

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

name = 1HOPSat-TD
Demise Altitude = 77.997926
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = PVA_Body_Mt_XandY
Demise Altitude = 76.463699
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = PVA_Hatch
Demise Altitude = 75.979418
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Nadir_ANT_Panel
Demise Altitude = 75.979418
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = ADCS_Box
Demise Altitude = 75.118808
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Reaction_Wheels
Demise Altitude = 66.358730
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = OBCS_and_other_PCBS
Demise Altitude = 76.571824
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Payload_Structures
Demise Altitude = 76.143668
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Primary_Mirror
Demise Altitude = 62.648796
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****

name = Secondary_Mirror
Demise Altitude = 71.116308
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = SC_Structures
Demise Altitude = 74.969925
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Batteries
Demise Altitude = 75.324644
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = EPS_Boards
Demise Altitude = 77.784894
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Cables_and_Connectors
Demise Altitude = 77.124558
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Misc_Fasteners
Demise Altitude = 76.923355
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Misc_Brackets
Demise Altitude = 77.082394
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Thermal_Straps
Demise Altitude = 73.627011
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Bulkhead
Demise Altitude = 0.000000
Debris Casualty Area = 0.640000
Impact Kinetic Energy = 346.029572

****
name = Struts
Demise Altitude = 0.000000
Debris Casualty Area = 3.484800
Impact Kinetic Energy = 7.258166

****
name = Secondary_Support
Demise Altitude = 0.000000
Debris Casualty Area = 3.519158
Impact Kinetic Energy = 3.672606

****
name = Torque_Rods
Demise Altitude = 0.000000
Debris Casualty Area = 1.343815
Impact Kinetic Energy = 4.484085

****
name = Bench
Demise Altitude = 0.000000
Debris Casualty Area = 0.624100
Impact Kinetic Energy = 327.299927

****
name = Star_tracker
Demise Altitude = 74.133527
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Baffles
Demise Altitude = 0.000000
Debris Casualty Area = 2.279613
Impact Kinetic Energy = 1.699261

****

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

Appendix B: Acronyms

ADCS	Attitude Determination and Control System
CC&T	Command, control, and telemetry
CDR	Critical Design Review
cm	Centimeter
CmA	Discharge Rate as a Fraction of Rated Capacity in Milliamperes
cm^2	Centimeter Squared
COTS	Commercial Off-The-Shelf (items)
C&DH	Command and Data Handling
DAS	Debris Assessment Software
DCA	Debris Casualty Area
deg	Degree
1HOPSat-TD	First High Optical Performance nano-Satellite – Technology Demonstator
EPS	Electrical Power Subsystem
EOM/EOMP	End Of Mission/EOM Plan
FRR	Flight Readiness Review
g	Grams
GEO	Geosynchronous Earth Orbit
ITAR	International Traffic In Arms Regulations
J	Joules
kg	kilogram
KE	Kinetic energy
km	kilometer
kJ	Kilo-Joules
LEO	Low Earth Orbit
m^2	Meters squared
N/A	Not Applicable.
OBCS	On-board computer system
ODAR	Orbital Debris Assessment Report
ODPO	Orbital Debris Program Office
ORR	Operations Readiness Review
OSMA	Office of Safety and Mission Assurance
PDR	Preliminary Design Review
Pf	Probability of Failure
PL	Payload
PMD	Post Mission Disposal
PSIa	Pounds Per Square Inch, Absolute
PSRR	Pre-Ship Readiness Review
PTC	Positive Temperature Coefficient
RAAN	Right Ascension of the Ascending Node
SMA/S&MA	Safety and Mission Assurance
TD	Technology demonstrator
Ti	Titanium
u, v, w	Cartesian Coordinate System
yr	year

Appendix C: Independent ODAR and EOMP Evaluation, 1HOPSat-TD Mission

(TBD Pending Independent Review)