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: 8/28/18

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 Satellite Project Documentation, which is controlled by the Hera Systems, Inc. Project Configuration Manager, San Jose, California.

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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

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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 = 0.00000 (< 0.001)
4.5-2	☐	☐	☒	☐	☒	☐	☐	Prob of small object collision using DAS 2.0.2 = 0.000000 (< 0.01)
4.6-1(a)	☐	☐	☒	☐	☒	☐	☐	Technology demonstration spacecraft: Natural reentry within 8 years of EOM. SSO spacecraft: Natural reentry within 17 years of EOM, in worst case failure mode.
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	☐	☐	☒	☐	☒	☐	☐	DCA of 0.67 m^2
4.8-1					☒	☐	☐	N/A – No Tethers

<u>Note 1</u>: 1HOPSat satellites are secondary payloads, 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 first launch a single technology demonstration (TD) spacecraft in a launch window opening on February 28, 2019, and closing in June 2019. This spacecraft will launch to an altitude of 500 km and inclination between 41 and 45 degrees. Subsequently, a constellation of eight (8) 22 kg satellites is planned for launch from November 2020 through the first quarter of 2021. These satellites will launch to 600 km orbits as secondary payloads on launch vehicles not yet contracted.

During launch, the satellites will be contained in 12U CubeSat payload dispensers 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, solar panels and antennas will deploy, imaging and communications will begin. There will be no propulsion on the TD spacecraft, but electric propulsion will be included on the constellation spacecraft. Accordingly, the constellation spacecraft will begin thrusting within hours to days after there are released from the launch vehicle. For all spacecraft, pointing control is provided by precise attitude determination and control systems. A GPS unit is included for accurate orbit location. The TD spacecraft orbit will decay naturally from 500 km. The constellation spacecraft orbit altitude will be lowered propulsively to 342 km, and inclination will be adjusted and maintained, by use of non-combustible electric propulsion.

The satellites each contain 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 and telemetry 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:

TD Spacecraft: Cape Canaveral, Florida.

Constellation Spacecraft: TBD

Proposed launch dates:

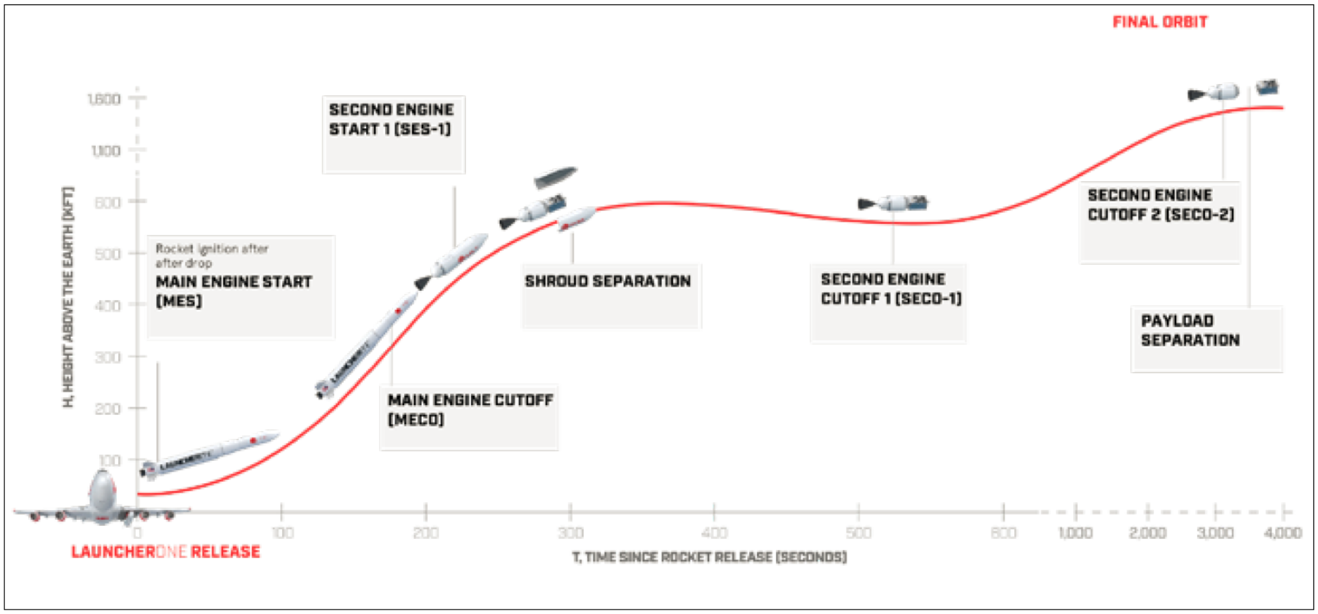
TD Launch: February 28, 2019

Constellation spacecraft: TBD, possibly in 2020-2021.

Mission duration: The TD spacecraft mission is intended to last 6 months. Constellation spacecraft primary operations are to last 3 to 5 years after launch. From launch, each spacecraft is designed to remain in LEO for 5 years until reentry effected by either natural decay of the orbit or low-thrust propulsive deorbit maneuvers.

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 TD spacecraft launch on a proposed Virgin Orbit launch vehicle.



Secondary payloads including the 1HOPSat-TD will be deployed as coordinated with the primary payload owner. The upper stage will deploy the 1HOPSat-TD into a 500 km orbit at inclination between 41 and 45 degrees.

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

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: 500km
Perigee: 500km
Inclination: 41-45 degrees

Figure 2 is representative of the launch operations profile for each planned launch. Secondary payloads including 1HOPSat(s) will be deployed as coordinated with the primary payload owner. The Soyuz upper stage will deploy 1HOPSat(s) into either 600 km Sun-synchronous orbits at 97.79 degrees inclination and 10:30 LTAN, or a 402 km altitude at 51.6 degrees inclination.

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

ColA maneuvers after dispensing are performed by the Soyuz launch operator.

Subsequent to deployment, 1HOPSat(s) will begin a series of coordinated orbit lowering and, where desired, inclination plane change thrust events using electrospray thrusters. A final operational orbit will be established and maintained below 350 km. This orbit will be maintained by use of regular re-boost thrusting over the course of more than three 3 years.

Initial Orbit after Launch:
1HOPSat satellites are deployed to initial circular orbits defined below. Preliminary imaging and maneuvering operations may take place at this altitude and inclination:
Apogee: 600km (8 satellites); 402 km (1 satellite)
Perigee: 600km (8 satellites); 402 km (1 satellite)
Inclination: 97.79 (8 satellites)

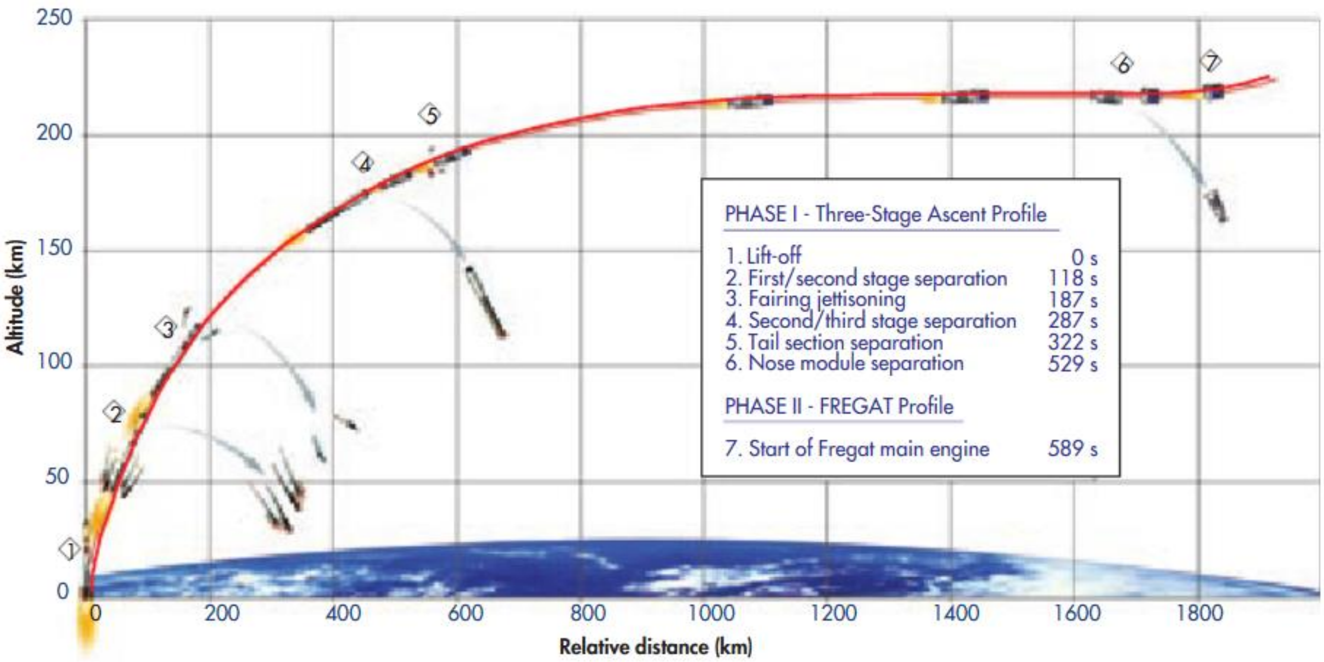


Figure 2, Generic Soyuz launch sequence

Orbit Lowering, Inclination Change, and Final Orbit:
Proper orbit inclination for 1HOPSat 10:30 LTAN sun-synchronous orbits will be maintained as the orbit is slowly lowered over a period of roughly 280 days.

Orbit inclination for the 1HOPSat satellite at 51.6 degrees may be raised by a few degrees depending on the commercial need. This inclination raising decision will be made after launch.

Operational Orbit:
Apogee: <350 km
Perigee: <350 km
Inclination: 97.79 degrees for eight (8) satellites in SSO Orbits. 51.6 to 55 degrees for one satellite.

Extended Operations:
Using the operational orbits defined above, 1HOPSats may continue mission operations until their electric propulsion propellant reserves are exhausted, or until end of mission is commanded.

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. 1HOPSats are commercial satellites.

Program Executive: Roger Roberts, PhD

Program/project manager: David. D. Squires

Senior scientist: Not applicable. 1HOPSats are commercial satellites.

Foreign government or space agency participation: Soyuz launch vehicle provided by Russia (not applicable to launch of the single TD spacecraft in 2019).

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:	February, 2019
Launch:	February/March, 2019

Primary Mission Complete	TD: L+ 6months; Constellation: L+3.5 years
Extended Mission Complete	TD: (TBD); Constellation: L+5 years (planned)

ODAR/EOMP Section 2: Spacecraft Description

Physical description of the spacecraft:
1HOPSat satellites are 12U CubeSat rectangular boxes conforming to common 12U dispenser payload size and mass specifications. Each spacecraft has a hatch-door opening to allow light to enter its nadir-pointing imager, deployable solar panels, and body-mounted antennas for its main downlink transmitter and two transceivers. 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, and titanium. Titanium components provide benefits to thermal management, but will be kept as small as possible. The payload also contains various types of glass for optical components. Note: TD the TD spacecraft will not have large deployable panels shown in Figure 3.

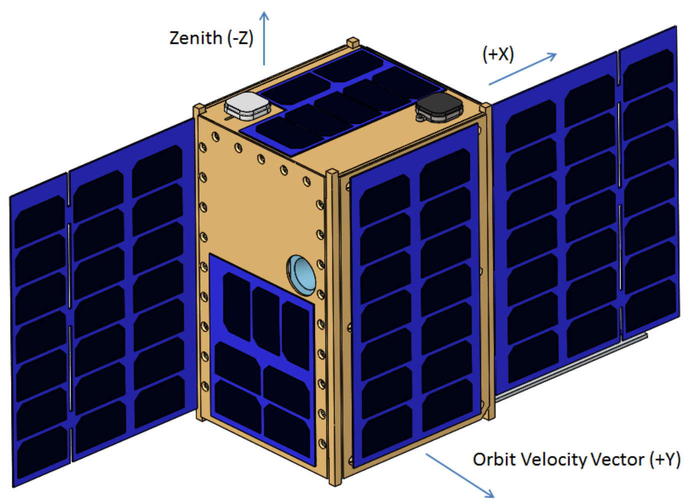


Figure 3, Hera 1HOPSat 12U Satellite Configuration (partial)

Total satellite mass at launch, including all propellants and fluids: 22 kg.
Dry mass of satellite at launch, excluding solid rocket motor propellants: TD Spacecraft: 22 kg; Constellation Spacecraft: ~21.1 kg. No solid rocket motors are used.

Description of all propulsion systems (cold gas, mono-propellant, bi-propellant, electric, nuclear): There is no propulsion on the TD spacecraft. Each constellation spacecraft uses up to four (4) electric thrusters that use non-combusting propellant. Operational average thrust ranges from 250 to 600 micro-Newtons per satellite. Peak thrust per satellite will not exceed 1 milli-newton.

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. 1HOPSat satellites use 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:
1HOPSat satellites use an integrated ADCS system. Normal attitude for 1HOPSat satellites is to present their smallest cross-section to the velocity vector direction. That is, deployed solar panel faces will be parallel to the direction of the velocity vector.

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

Description of the electrical generation and storage system: 30.2% 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 each 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 each 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 each 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 each 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 each 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: 0.000000 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 each 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 each 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 each spacecraft uses 12 cells: Pf = 0.0000001*10*12 = 0.000012 (per spacecraft)

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, most likely faster since the batteries will have been in orbit for many years prior to initiating this command.

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/EOMP 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:** COMPLIANT. Below required probability for all orbits; calculated result is less than the round off value of the DAS 2.0.2 software.

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

Critical surface1: Battery Passivation Circuits
1HOPSat can passivate its battery pack at end of mission through use of a command or by lowering altitude to a point where explosive failure of the batteries does not pose a risk of generating orbiting debris. The spacecraft bus and payload contain circuits that must execute or support (as loads) the battery passivation functions. 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. This is estimated using polycarbonate plastic at 1250 kg/m^3. 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.

Outer walls:
The critical surfaces are surrounded on all sides by aluminum-backed solar panels made of 6061-T6 Aluminum. The thinnest aluminum areas are 1.5 mm thick. Therefore, the effective areal density of these panels is at least 0.406 g/cm^2 (ignoring solar cell contributions) as seen from the location of critical surfaces. In some cases an effective density of twice or three times this value may be seen for surfaces that are intermediated by payload walls and/or other structures using predominantly 7075-T6 and 6061-T6 aluminum. Values

selected for this analysis appear in the DAS 2.0.2 log file provided in “ODAR Section 7” content in this document.

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

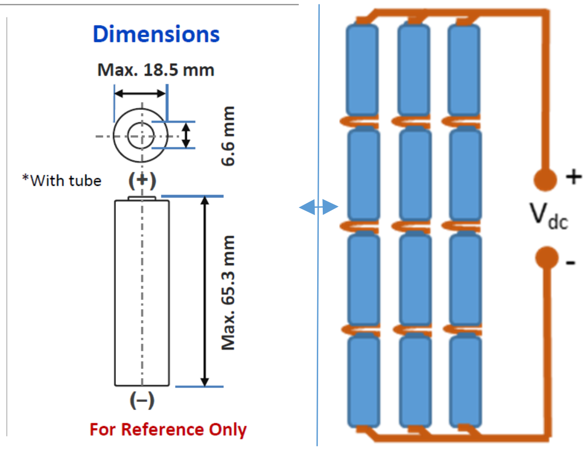


Figure 4: (left) 1HOPSat Battery Cell Specifications (1 of 12); (right) battery pack wiring.

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 use of low thrust propulsion.

6.2 Plan for any spacecraft maneuvers required to accomplish post mission disposal: The post mission disposal plan is to use low thrust to shorten the time required to achieve reentry. Time of reentry will be roughly estimated and might increase the likelihood of reentry over an ocean although this is not a requirement.

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: ~22 kg
Cross-sectional Area: Due to the dynamic motion of the spacecraft, cross sectional area varies from 0.08 to 0.24 m^2 (Calculated by DAS 2.0.2).
Area to mass ratio:
Minimum: 0.08/22 = 0.00364 m^2/kg
Maximum: 0.24/22 = 0.01091 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: 1HOPSat TD and Constellation reentries are 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 8 years. The 1HOPSat constellation spacecraft have propulsion that will reduce orbit altitude to shorten their disposal period, but this propulsion does not enable significant control over the de-orbit trajectory. For the Constellation spacecraft, nominal de-orbit after end of mission can be implemented within seven (7) days or so by use of propulsion. However, if the propulsion fails to operate, a constellation spacecraft with its solar panels deployed, left in a 600 km by 600 km circular orbit, would reenter by natural decay within ~16.4 years after launch.

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 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 1:89,800. As seen in the analysis outputs below (see Requirement 4.7-1), the impact kinetic energy for our titanium bulkhead (the only component with impact energy above the threshold of concern for human safety) is 84 Joules and the impact casualty area is 0.67 square meters.

Requirements 4.7-1b, and 4.7-1c below are non-applicable requirements because 1HOPSat can 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: Representative DAS v2.0.2 Analysis Results (SSO orbit)

Note: This analyis represents worst case as it includes tanks and deployable PVAs that will not be included in the Technology Demonstration Flight, but might be included in the Constellation Spacecraft.

03 03 2016; 15:22:01PM DAS Application Started
03 03 2016; 15:22:01PM Opened Project C:\Users\Dave\AppData\Local\NASA\DAS 2.0\project\12U-22kg-LargePVA-52deg\
03 03 2016; 15:22:11PM Processing Requirement 4.3-1: Return Status : Not Run

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

=============== End of Requirement 4.3-1 ===============
03 03 2016; 15:22:13PM Processing Requirement 4.3-2: Return Status : Passed

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

=============== End of Requirement 4.3-2 ===============
03 03 2016; 15:22:15PM Requirement 4.4-3: Compliant

=============== End of Requirement 4.4-3 ===============
03 03 2016; 15:22:18PM Processing Requirement 4.5-1: Return Status : Passed

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

INPUT

Space Structure Name = 1HOPSat-52deg
Space Structure Type = Payload
Perigee Altitude = 342.000000 (km)
Apogee Altitude = 342.000000 (km)
Inclination = 51.650000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass Ratio = 0.011370 (m^2/kg)
Start Year = 2016.836000 (yr)
Initial Mass = 22.000000 (kg)

Final Mass = 21.100000 (kg)
Duration = 3.500000 (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 = 0.000000
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range
Status = Pass

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

=============== End of Requirement 4.5-1 ===============
03 03 2016; 15:23:34PM Requirement 4.5-2: Compliant

==
Spacecraft = 1HOPSat-52deg
Critical Surface = Propulsion_Tank_1
==

INPUT

Apogee Altitude = 342.000000 (km)
Perigee Altitude = 342.000000 (km)
Orbital Inclination = 51.650000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass = 0.011370 (m^2/kg)
Initial Mass = 21.100000 (kg)
Final Mass = 21.100000 (kg)
Station Kept = No
Start Year = 2016.836000 (yr)
Duration = 3.500000 (yr)
Orientation = Random Tumbling
CS Areal Density = 0.697000 (g/cm^2)
CS Surface Area = 0.020000 (m^2)

Vector = (0.000000 (u), 0.000000 (v), 0.000000 (w))
CS Pressurized = No
Outer Wall 1 Density: 1.400000 (g/cm^2) Separation: 5.000000 (cm)
Outer Wall 2 Density: 1.400000 (g/cm^2) Separation: 8.000000 (cm)
Outer Wall 3 Density: 1.400000 (g/cm^2) Separation: 2.000000 (cm)
Outer Wall 4 Density: 1.400000 (g/cm^2) Separation: 15.000000 (cm)
Outer Wall 5 Density: 7.590000 (g/cm^2) Separation: 4.000000 (cm)

OUTPUT

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

===
Spacecraft = 1HOPSat-52deg
Critical Surface = Propulsion_Tank_2
===

INPUT

Apogee Altitude = 342.000000 (km)
Perigee Altitude = 342.000000 (km)
Orbital Inclination = 51.650000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass = 0.011370 (m^2/kg)
Initial Mass = 21.100000 (kg)
Final Mass = 21.100000 (kg)
Station Kept = No
Start Year = 2016.836000 (yr)
Duration = 3.500000 (yr)
Orientation = Random Tumbling
CS Areal Density = 0.697000 (g/cm^2)
CS Surface Area = 0.020000 (m^2)
Vector = (0.000000 (u), 0.000000 (v), 0.000000 (w))
CS Pressurized = No
Outer Wall 1 Density: 1.400000 (g/cm^2) Separation: 5.000000 (cm)
Outer Wall 2 Density: 1.400000 (g/cm^2) Separation: 8.000000 (cm)
Outer Wall 3 Density: 1.400000 (g/cm^2) Separation: 2.000000 (cm)
Outer Wall 4 Density: 1.400000 (g/cm^2) Separation: 15.000000 (cm)
Outer Wall 5 Density: 7.590000 (g/cm^2) Separation: 4.000000 (cm)

OUTPUT

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

==
Spacecraft = 1HOPSat-52deg
Critical Surface = Propulsion_Tank_3
==

INPUT

Apogee Altitude = 342.000000 (km)
Perigee Altitude = 342.000000 (km)
Orbital Inclination = 51.650000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass = 0.011370 (m^2/kg)
Initial Mass = 21.100000 (kg)
Final Mass = 21.100000 (kg)
Station Kept = No
Start Year = 2016.836000 (yr)
Duration = 3.500000 (yr)
Orientation = Random Tumbling
CS Areal Density = 0.697000 (g/cm^2)
CS Surface Area = 0.016000 (m^2)
Vector = (0.000000 (u), 0.000000 (v), 0.000000 (w))
CS Pressurized = No
Outer Wall 1 Density: 1.400000 (g/cm^2) Separation: 8.000000 (cm)
Outer Wall 2 Density: 7.590000 (g/cm^2) Separation: 4.000000 (cm)
Outer Wall 3 Density: 1.400000 (g/cm^2) Separation: 2.000000 (cm)
Outer Wall 4 Density: 1.400000 (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

==
Spacecraft = 1HOPSat-52deg
Critical Surface = Propulsion_Tank_4
==

INPUT

Apogee Altitude = 342.000000 (km)
Perigee Altitude = 342.000000 (km)
Orbital Inclination = 51.650000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass = 0.011370 (m^2/kg)
Initial Mass = 21.100000 (kg)
Final Mass = 21.100000 (kg)
Station Kept = No
Start Year = 2016.836000 (yr)
Duration = 3.500000 (yr)
Orientation = Random Tumbling
CS Areal Density = 0.697000 (g/cm^2)
CS Surface Area = 0.016000 (m^2)
Vector = (0.000000 (u), 0.000000 (v), 0.000000 (w))
CS Pressurized = No
Outer Wall 1 Density: 1.400000 (g/cm^2) Separation: 8.000000 (cm)
Outer Wall 2 Density: 7.590000 (g/cm^2) Separation: 4.000000 (cm)
Outer Wall 3 Density: 1.400000 (g/cm^2) Separation: 2.000000 (cm)
Outer Wall 4 Density: 1.400000 (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

===
Spacecraft = 1HOPSat-52deg
Critical Surface = OBCS_PCB_and_Plastics
===

INPUT

Apogee Altitude = 342.000000 (km)
Perigee Altitude = 342.000000 (km)
Orbital Inclination = 51.650000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass = 0.011370 (m^2/kg)
Initial Mass = 21.100000 (kg)

Final Mass = 21.100000 (kg)
Station Kept = No
Start Year = 2016.836000 (yr)
Duration = 3.500000 (yr)
Orientation = Random Tumbling
CS Areal Density = 0.080000 (g/cm^2)
CS Surface Area = 0.010000 (m^2)
Vector = (0.000000 (u), 0.000000 (v), 0.000000 (w))
CS Pressurized = No
Outer Wall 1 Density: 1.400000 (g/cm^2) Separation: 2.000000 (cm)
Outer Wall 2 Density: 1.400000 (g/cm^2) Separation: 20.000000 (cm)
Outer Wall 3 Density: 1.400000 (g/cm^2) Separation: 8.000000 (cm)
Outer Wall 4 Density: 1.400000 (g/cm^2) Separation: 8.000000 (cm)
Outer Wall 5 Density: 1.400000 (g/cm^2) Separation: 8.000000 (cm)
Outer Wall 6 Density: 7.590000 (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-52deg
Critical Surface = ADCS_Control_PCB
===

INPUT

Apogee Altitude = 342.000000 (km)
Perigee Altitude = 342.000000 (km)
Orbital Inclination = 51.650000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass = 0.011370 (m^2/kg)
Initial Mass = 21.100000 (kg)
Final Mass = 21.100000 (kg)
Station Kept = No
Start Year = 2016.836000 (yr)
Duration = 3.500000 (yr)
Orientation = Random Tumbling
CS Areal Density = 0.697000 (g/cm^2)
CS Surface Area = 0.040000 (m^2)
Vector = (0.000000 (u), 0.000000 (v), 0.000000 (w))

CS Pressurized = No
Outer Wall 1 Density: 1.400000 (g/cm^2) Separation: 2.000000 (cm)
Outer Wall 2 Density: 1.400000 (g/cm^2) Separation: 15.000000 (cm)
Outer Wall 3 Density: 1.400000 (g/cm^2) Separation: 2.000000 (cm)
Outer Wall 4 Density: 1.400000 (g/cm^2) Separation: 15.000000 (cm)
Outer Wall 5 Density: 1.400000 (g/cm^2) Separation: 2.000000 (cm)
Outer Wall 6 Density: 7.590000 (g/cm^2) Separation: 8.000000 (cm)

OUTPUT

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

===
Spacecraft = 1HOPSat-52deg
Critical Surface = EPS_PCB
===

INPUT

Apogee Altitude = 342.000000 (km)
Perigee Altitude = 342.000000 (km)
Orbital Inclination = 51.650000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass = 0.011370 (m^2/kg)
Initial Mass = 21.100000 (kg)
Final Mass = 21.100000 (kg)
Station Kept = No
Start Year = 2016.836000 (yr)
Duration = 3.500000 (yr)
Orientation = Random Tumbling
CS Areal Density = 0.697000 (g/cm^2)
CS Surface Area = 0.010000 (m^2)
Vector = (0.000000 (u), 0.000000 (v), 0.000000 (w))
CS Pressurized = No
Outer Wall 1 Density: 1.400000 (g/cm^2) Separation: 20.000000 (cm)
Outer Wall 2 Density: 1.400000 (g/cm^2) Separation: 2.000000 (cm)
Outer Wall 3 Density: 1.400000 (g/cm^2) Separation: 8.000000 (cm)
Outer Wall 4 Density: 1.400000 (g/cm^2) Separation: 8.000000 (cm)
Outer Wall 5 Density: 1.400000 (g/cm^2) Separation: 6.000000 (cm)
Outer Wall 6 Density: 7.590000 (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-52deg
Critical Surface = Batteries
==

INPUT

Apogee Altitude = 342.000000 (km)
Perigee Altitude = 342.000000 (km)
Orbital Inclination = 51.650000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass = 0.011370 (m^2/kg)
Initial Mass = 21.100000 (kg)
Final Mass = 21.100000 (kg)
Station Kept = No
Start Year = 2016.836000 (yr)
Duration = 3.500000 (yr)
Orientation = Random Tumbling
CS Areal Density = 0.256000 (g/cm^2)
CS Surface Area = 0.240000 (m^2)
Vector = (0.000000 (u), 0.000000 (v), 0.000000 (w))
CS Pressurized = No
Outer Wall 1 Density: 1.400000 (g/cm^2) Separation: 3.000000 (cm)
Outer Wall 2 Density: 1.400000 (g/cm^2) Separation: 3.000000 (cm)
Outer Wall 3 Density: 1.400000 (g/cm^2) Separation: 3.000000 (cm)
Outer Wall 4 Density: 1.400000 (g/cm^2) Separation: 3.000000 (cm)
Outer Wall 5 Density: 1.400000 (g/cm^2) Separation: 2.000000 (cm)
Outer Wall 6 Density: 7.590000 (g/cm^2) Separation: 14.000000 (cm)

OUTPUT

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

03 03 2016; 15:24:53PM Processing Requirement 4.6 Return Status : Passed

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

INPUT

Space Structure Name = 1HOPSat-52deg
Space Structure Type = Payload

Perigee Altitude = 342.000000 (km)
Apogee Altitude = 342.000000 (km)
Inclination = 51.650000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Area-To-Mass Ratio = 0.011370 (m^2/kg)
Start Year = 2016.836000 (yr)
Initial Mass = 22.000000 (kg)
Final Mass = 21.100000 (kg)
Duration = 3.500000 (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

Suggested Perigee Altitude = 342.000000 (km)
Suggested Apogee Altitude = 342.000000 (km)
Returned Error Message = Reentry during mission (no PMD req.).

Released Year = 2017 (yr)
Requirement = 61
Compliance Status = Pass

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

============== End of Requirement 4.6 ==============
03 03 2016; 15:25:14PM *********Processing Requirement 4.7-1
Return Status : Passed

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

name = 1HOPSat-52deg
quantity = 1
parent = 0
materialID = 9
type = Box
Aero Mass = 21.100000
Thermal Mass = 21.100000
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 = 0.800000
Thermal Mass = 0.800000
Diameter/Width = 0.260000
Length = 0.340000

name = PVA_Zenith
quantity = 1
parent = 1
materialID = 9
type = Flat Plate
Aero Mass = 0.200000
Thermal Mass = 0.200000
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 = Propulsion_Thruster_and_Tanks
quantity = 4
parent = 1
materialID = 9
type = Box
Aero Mass = 0.500000
Thermal Mass = 0.500000
Diameter/Width = 0.100000
Length = 0.150000
Height = 0.030000

name = OBCS_and_other_PCBS
quantity = 8
parent = 1
materialID = 23
type = Flat Plate
Aero Mass = 0.150000
Thermal Mass = 0.150000
Diameter/Width = 0.100000
Length = 0.100000

name = Payload_Structures
quantity = 4
parent = 1

materialID = 8
type = Box
Aero Mass = 0.500000
Thermal Mass = 0.500000
Diameter/Width = 0.070000
Length = 0.090000
Height = 0.070000

name = Primary_Mirror
quantity = 1
parent = 1
materialID = 71
type = Cylinder
Aero Mass = 0.150000
Thermal Mass = 0.150000
Diameter/Width = 0.200000
Length = 0.060000

name = Secondary_Mirror
quantity = 1
parent = 1
materialID = 71
type = Cylinder
Aero Mass = 0.050000
Thermal Mass = 0.050000
Diameter/Width = 0.060000
Length = 0.020000

name = SC_Structures
quantity = 8
parent = 1
materialID = 9
type = Box
Aero Mass = 0.500000
Thermal Mass = 0.500000
Diameter/Width = 0.030000
Length = 0.340000
Height = 0.020000

name = PVA_Deployable
quantity = 2
parent = 1
materialID = 9
type = Flat Plate
Aero Mass = 0.500000

Thermal Mass = 0.500000
Diameter/Width = 0.240000
Length = 0.320000

name = Batteries
quantity = 12
parent = 1
materialID = 8
type = Cylinder
Aero Mass = 0.046500
Thermal Mass = 0.046500
Diameter/Width = 0.019000
Length = 0.063000

name = EPS
quantity = 1
parent = 1
materialID = 23
type = Flat Plate
Aero Mass = 0.150000
Thermal Mass = 0.150000
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 = Ballast
quantity = 4
parent = 1
materialID = 9
type = Box
Aero Mass = 0.850000
Thermal Mass = 0.850000
Diameter/Width = 0.060000
Length = 0.150000
Height = 0.060000

name = Bulkhead
quantity = 1
parent = 1
materialID = 66
type = Flat Plate
Aero Mass = 0.500000
Thermal Mass = 0.500000
Diameter/Width = 0.220000
Length = 0.220000

name = Metering_Structures
quantity = 8
parent = 1
materialID = 66
type = Flat Plate
Aero Mass = 0.005000
Thermal Mass = 0.005000
Diameter/Width = 0.010000
Length = 0.150000

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

name = 1HOPSat-52deg
Demise Altitude = 77.995129

Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

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name = PVA_Body_Mt_XandY
Demise Altitude = 75.164488
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

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name = PVA_Zenith
Demise Altitude = 77.090551
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Nadir_ANT_Panel
Demise Altitude = 76.164222
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = ADCS_Box
Demise Altitude = 75.349738
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Reaction_Wheels
Demise Altitude = 67.103050
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Propulsion_Thruster_and_Tanks
Demise Altitude = 73.763824
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = OBCS_and_other_PCBS
Demise Altitude = 76.049316
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Payload_Structures
Demise Altitude = 72.663886
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Impact Kinetic Energy = 0.000000

****
name = Primary_Mirror
Demise Altitude = 76.085347
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Impact Kinetic Energy = 0.000000

****
name = Secondary_Mirror
Demise Altitude = 74.612597
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = SC_Structures
Demise Altitude = 74.119043
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = PVA_Deployable
Demise Altitude = 76.075933
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name = Batteries
Demise Altitude = 75.520894
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Impact Kinetic Energy = 0.000000

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name = EPS
Demise Altitude = 76.049316
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****
name = Cables_and_Connectors
Demise Altitude = 77.187394

Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

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name = Misc_Fasteners
Demise Altitude = 77.028519
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

****

name = Misc_Brackets
Demise Altitude = 77.145808
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

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name = Ballast
Demise Altitude = 71.957785
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000

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name = Bulkhead
Demise Altitude = 0.000000
Debris Casualty Area = 0.672400
Impact Kinetic Energy = 84.345200

****

name = Metering_Structures
Demise Altitude = 0.000000
Debris Casualty Area = 3.263807
Impact Kinetic Energy = 0.271916

****

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

Appendix B: Acronyms

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	First High Optical Performance Satellite
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.
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 Mission

(TBD Pending Independent Review)