Vigoride-1 Spacecraft

Orbital Debris Assessment Report (ODAR)

v1

11/05/2019

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Document contains no ITAR or otherwise restricted data.

DAS Software Version used in Analysis: v2.1.1

Revision History

Revision Number	Updates	Page #	Author	Date
1	Initial Release	All	Sam Avery	11/05/19

Table of Contents

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Momentus

Orbital Debris Self-Assessment Evaluation

Reqm't #	Launch Vehicle: Compliant	Launch Vehicle: N/A	Launch Vehicle: Not Compliant	Launch Vehicle: Std. Non-Compliant	Launch Vehicle: Incomplete	Spacecraft: Compliant	Spacecraft: N/A	Spacecraft: Not Compliant	Spacecraft: Incomplete	Comments *For all incompletes, include risk assessment (low, medium, or high risk) of non-compliance & Project Risk Tracking #*
4.3-1.a *25 year limit*	☐	☒	☐	☐	☐	☒	☐	☐	☐	See comment 1. No debris released.
4.3-1.b *<100 object x year limit*	☐	☒	☐	☐	☐	☒	☐	☐	☐	See comment 1. No debris released.
4.3-2 *GEO +/- 200km*	☐	☒	☐	☐	☐	☒	☐	☐	☐	See comment 1. No debris released.
4.4-1 *<0.001 Explosion Risk*	☐	☒	☐	☐	☐	☒	☐	☐	☐	See comment 1.
4.4-2 *Passivate Energy Sources*	☐	☒	☐	☐	☐	☒	☐	☐	☐	See comment 1.
4.4-3 *Limit BU Long term Risk*	☐	☒	☐	☐	☐	☒	☐	☐	☐	See comment 1. No intentional breakups.
4.4-4 *Limit BU Short term Risk*	☐	☒	☐	☐	☐	☒	☐	☐	☐	See comment 1. No intentional breakups.
4.5-1 *<.001 10cm Impact Risk*	☐	☒	☐	☐	☐	☒	☐	☐	☐	See comment 1.
4.5-2 *Postmission Disposal Risk*	☐	☐	☐	☐	☐	☒	☐	☐	☐	
4.6-1a-c *Disposal Method*	☐	☒	☐	☐	☐	☒	☐	☐	☐	See comment 1.
4.6-2 *GEO Disposal*	☐	☒	☐	☐	☐	☒	☐	☐	☐	See comment 1.
4.6-3 *MEO Disposal*	☐	☒	☐	☐	☐	☒	☐	☐	☐	See comment 1.
4.6-4 *Disposal Reliability*	☐	☒	☐	☐	☐	☒	☐	☐	☐	See comment 1.
4.7-1 *Ground Population Risk*	☐	☒	☐	☐	☐	☒	☐	☐	☐	See comment 1.
4.8-1 *Tether Risk*	☐	☐	☐	☐	☐	☒	☐	☐	☐	No tethers used.

Comment 1. This ODAR considers only VR-1 and mass dummy payloads. The launch vehicle and other launch vehicle payloads are not considered in this ODAR.

Assessment Report Format

ODAR Technical Sections Format Requirements: This ODAR follows the format recommended in NASA-STD-8719.14, Appendix A.1 and includes the content indicated at a minimum in sections 2 through 8 for the Vigoride-1 (“VR-1”) satellite. Sections 9 through 14 of the NASA standard apply to the launch vehicle ODAR and are not covered here.

I. Program Management and Mission Overview

Project Manager:

- Sam Avery

Foreign Government or space agency participation:

- None.

Schedule of Upcoming Mission Milestones:

- Launch - May 2020

Mission Overview:

Momentus Inc. (“Momentus”) provides efficient and inexpensive in-space transportation services for small satellites. We deliver small satellites to exact orbits cheaper and faster by providing connecting flights in space. Our orbit transfer vehicles are built to carry customer payloads on either dedicated or rideshare rockets. Once a launch vehicle reaches initial orbit following launch, a Vigoride spacecraft is able to deliver customer satellites to one or multiple custom final orbits. Vigoride is designed around a Microwave Electrothermal Thruster (MET), which runs using non-toxic and low-pressurized water propellant to provide low-thrust orbit transfers. The Momentus-designed Vigoride service will provide operators a cost-effective means of achieving custom orbits for their spacecrafts. The ability to customize orbits within and adjacent to high density orbits empowers a greater user population to make more efficient and safer use of common orbits.

This ODAR evaluates the Momentus initial demonstration mission, Vigoride-1 (“VR-1”), which has a planned launch on a SpaceX Falcon 9 rocket in May 2020. VR-1 will have the capacity to transport and deploy multiple payloads (individually, “Payload 1,” “Payload 2,” and “Payload 3,” and together, the “Payloads”). Payload 1 is expected to be a standard 6U cubesat and Payloads 2 and 3 are expected to be standard 3U cubesats. The de-orbit analysis for customer satellite payloads will be addressed through the licensing process for the relevant payload. Nonetheless, in an abundance of caution for purposes of this ODAR, Momentus assumes that the Payloads will be mass dummies with parameters representative of the expected customer payloads.[1]

Launch Vehicle:

- SpaceX Falcon 9

[1] In the event the customers are unable to secure the requisite licensing for their respective payloads, Momentus will launch VR-1 with the applicable mass dummy.

Expected Launch Site:

- Cape Canaveral, FL, USA

Operational Mission Duration:

- Planned for approximately 6 months.

The VR-1 concept of operations is as follows:

1. Launch vehicle arrives at initial orbit (220x380 km nominal)[2]
2. VR-1 separates from launch vehicle
3. VR-1 undergoes commissioning and testing
4. VR-1 conducts orbit raising maneuvers to second orbit (max. 380 km circular)
5. VR-1 deploys Payloads 1 and 2
6. VR-1 conducts orbit raising maneuvers to third orbit (max. 500 km circular)
7. VR-1 deploys Payload 3
8. VR-1 conducts de-orbit maneuvers (max. 500 x 300 km)

[2] The launch vehicle operator has indicated it may select an alternative insertion orbit of 289 km circular. In that event, Momentus will notify the FCC by letter. For completeness, this ODAR includes a decay analysis for that alternative insertion orbit. *See* Section VI *infra*.

Table 1: Orbital Parameters

	Insertion Orbit	Payloads 1 and 2 Orbit	Payload 3 Orbit	VR-1 End-of-Life Orbit
Apogee Altitude	380 km	380 km (max)	500 km (max)	500 km (max)
Perigee Altitude	220 km	380 km (max)	500 km (max)	300 km
Inclination	~53°	~53°	~53°	~53°
Period	90 mins	90-96 mins	90-96 mins	90-96 mins
Argument of Perigee	N/A	N/A	N/A	N/A
Right Ascension of the Ascending Node (RAAN)	Initial Launch RAAN (Falcon 9)	All	All	All
Maximum De-Orbit Life	VR-1[3] 3 months	VR-1[4] 1 year Payloads 1,2 (dummy masses)[5] 15 months	VR-1[6] 3 years Payload 3 (dummy mass) 4 years	VR-1 1 year

[3] This is the de-orbit duration if VR-1 fails after deployment from the launch vehicle. In the event the insertion orbit is 289 km circular, the maximum de-orbit duration for VR-1 at this orbit is expected to be 3 months.
[4] This is the de-orbit duration if VR-1 fails after deployment of Payloads 1 and 2.
[5] In the event that any of the Payloads are customer satellites, the de-orbit analysis will be addressed through the licensing process for the relevant payload.
[6] This is the de-orbit duration if VR-1 reaches a 500 km circular orbit and then fails. At this altitude, VR-1 experiences the worst-case de-orbit duration possible for the mission.

II. Spacecraft Description

Physical Description:

VR-1 has the following subsystems: Propulsion System, Structures, Mechanisms, Electric Power System, and Avionics.

VR-1 includes a primary and a secondary structural assembly with: Propellant Tanks, MET, Reaction Control System thrusters, Solar Array Assemblies, a 15” payload adapter system for interfacing with the launch vehicle, one 6U cubesat deployer and one dual 3U cubesat deployer.

The VR-1 spacecraft bus includes two spring-loaded UHF antennas which are deployed after separation from the launch vehicle by a burn wire. In addition, VR-1 includes two 4-panel 150W deployable solar arrays which are deployed using a frangibolt Hold Down and Release Mechanism (HDRM). Both of these deployments are controlled by a software timer via the flight computer.

The Payloads (or mass dummies) are fully stowed in their deployers and their power is inhibited prior to on-orbit deployment. The Payloads are expected to follow the form and mass characteristics of a standard 6U cubesat and standard 3U cubesats. The VR-1 spacecraft platform components all have their power inhibited until launch vehicle separation occurs.

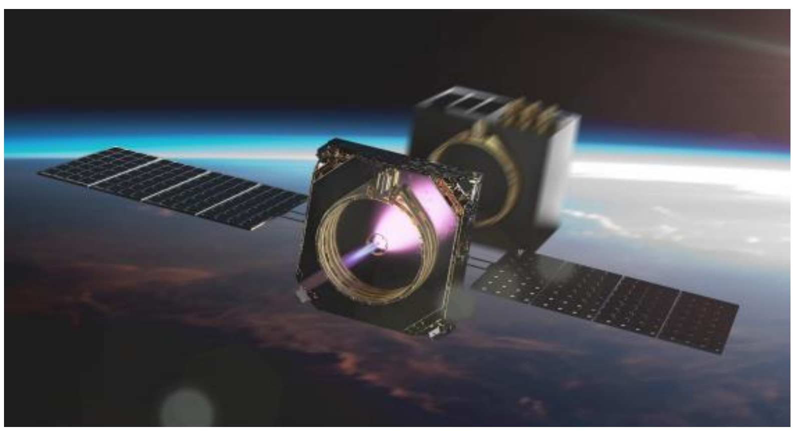


Figure 1. Artist rendering of Vigoride deploying a customer spacecraft.

Table 2. General Spacecraft Description

Criteria	Description	Notes
Spacecraft Total Launch Mass	101 kg	Includes 6 kg of propellant and planned 21 kg of

		deployable payloads.
Spacecraft Launch Dry Mass	95 kg	
Propulsion System	MET and resistojet reaction control thrusters. See *Propulsion System Description*.	Propulsion system operates using non-toxic and low-pressure liquid water propellant.
Body Dimensions	0.75x0.64x0.53 m^3	
Deployed Solar Array Dimensions	1.4x0.64x0.03 m^3	Dimensions per solar array wing.
Identification of all Fluids	• Liquid/vapor water mixture (propellant) • Nitrogen (tank pressurant) See *Fluids Description*.	
Fluids in Pressurized Batteries	None. VR-1 uses unpressurized lithium ion cells.	
Attitude Determination and Control	Attitude Determination • Star Tracker • Sun Sensors • Magnetometers • Gyroscope Attitude Control • Magnetic Torque Rods • Reaction Wheels • Reaction Control Thrusters See *Attitude Determination and Control System Description*.	
Range Safety or Pyrotechnic Devices	The launch vehicle clampband separation system, a RUAG PAS381S, includes a low-shock and sealed (non-debris generating) pyrotechnic device.	

Electrical Generation and Storage System	Two 150W solar array wings for power generation. One Lithium-Ion 7S1P battery (~28V) and one Lithium-Ion 1S4P battery (~4.2V) are included and charged prior to launch vehicle integration. A separate failsafe deployment mechanism includes a Lithium-Ion 1S1P battery (~4.2V).	
Other Stored Energy	• Solar array spring energy stored in hinges. • Clampband separation system spring energy. • CubeSat deployer hinged door spring energy. • Launch vehicle clampband separation system pyrotechnic energy.	
Radioactive Materials	Not applicable.	

Propulsion System Description

The Vigoride propulsion system energizes distilled water into plasma using RF microwave energy. The plasma is expelled out of the thruster using a nozzle to produce thrust at a specific impulse (Isp) exceeding traditional chemical propulsion systems. The expected thrust and Isp will vary with input power levels but will not exceed 15 mN. The VR-1 mission will nominally include 6+/-1 kg of liquid water propellant at launch. The 6 kg of water is roughly split into two parts: 4 kg total in two diaphragm tanks; and 2 kg total in two prototype propellant tanks. The maximum expected total impulse is approximately 25 kNs.

In addition, there are four experimental reaction control thrusters with estimated maximum thrust of 5 mN and specific impulse of 80 seconds. These reaction control thrusters use the same stored liquid water as a propellant.

Fluids Description

The propulsion system includes two diaphragm tanks with liquid water propellant pressurized using inert gaseous nitrogen. Each diaphragm tank is expected to include 2 kg of propellant. In addition, there are two prototype propellant tanks filled with water propellant and also pressurized at launch using inert gaseous nitrogen. These prototype propellant tanks are expected to include 1 kg of propellant. At spacecraft integration the diaphragm tanks are pressurized to 3 atmospheres (~44 psi at 68°F), one prototype propellant tank at 3 atmospheres (~44 psi at 68°F), and another prototype propellant tank at the vapor pressure of water (~0.3 psi at 68°F). The tanks are expected to remain within 20% of these pressures during transportation to the launch site, during launch, and post launch until thruster operations.

At the end of the mission, following a de-orbit maneuver, the propellant flow valve will be fully opened to allow all propellant and pressurant to vent into space and remove thrusting capability from the system. The spacecraft will be oriented such that any resultant thrust from the drain operations will result in a lower orbital altitude.

Attitude Determination and Control System Description

The VR-1 spacecraft includes 3-axis control with magnetic torque rods for coarse pointing and detumbling, reaction wheels for fine control (1° pointing accuracy), and experimental reaction control thrusters. For attitude determination, the spacecraft includes a star tracker, sun sensors, magnetometers, and a gyroscope, providing nominally >3° pointing knowledge.

- A sun tracking mode that is optimized for solar power generation from the satellite. The spacecraft's body will be oriented in two axes, and on-board Solar Array Drive Assemblies (SADAs) will rotate the panels along the third axis.
- A targeted tracking mode will allow the thrust axis to be pointed in any direction in inertial space.

III. Spacecraft Debris Released during Normal Operations

Table 3. Payload Release Description

	Payload 1[7] (dummy mass)	Payload 2 (dummy mass)	Payload 3 (dummy mass)
Dimensions	~21x11x34 cm^3	~11x11x34 cm^3	~11x11x34 cm^3
Mass	10 kg	5.5 kg	5.5 kg
Materials	Aluminum (frame)	Aluminum (frame)	Aluminum (frame)
Rationale	technology demonstration	technology demonstration	technology demonstration
Time of Release	within 2 months post-launch	within 2 months post-launch	within 5 months post-launch
Release Velocity	<2 m/s (CubeSat Deployer)	<2 m/s (CubeSat Deployer)	<2 m/s (CubeSat Deployer)
Expected Apogee	380 km (max)	380 km (max)	500 km (max)
Expected Perigee	380 km (max)	380 km (max)	500 km (max)
Expected Inclination	53°	53°	53°
Orbital Lifetime	15 months	15 months	4 years

Payload Re-Contact Mitigation

Momentus will plan to support at least three re-contact mitigation strategies:

1. Payload deployments will be spaced apart by at least 90 minutes, or 1 full orbit.

[7] The de-orbit analysis for customer satellite payloads will be addressed through the licensing process for the relevant payloads. Nonetheless, the mass dummy values in the chart for the Payloads are also representative of prospective customer satellite payloads. All analyses are performed to determine the longest expected orbital lifetime based on worst case area-to-mass ratios and deployment orbits.

2. Payload deployments will alternate between along-track deployment with the velocity vector and with the anti-velocity vector.
3. On-board propulsion may also be used for maneuvers to minimize the risk of re-contact.

Persistent Liquids and Propellant-Related Debris

During primary mission operations, any water released into space through the MET or the reaction control thrusters will be vaporized at sufficiently high temperature (>500K) to prevent the formation of debris. In the off-nominal case of a leak or flow of liquid water, there is potential for the creation of small water ice crystals. Any generated water ice crystals are expected to sublimate within minutes of exposure to sunlight.

Assessment of spacecraft compliance with Requirements 4.3-1 and 4.3-2 (per DAS v2.1.1)

- **4.3-1, Mission Related Debris Passing Through LEO:** COMPLIANT
- **4.3-2, Mission Related Debris Passing Near GEO:** COMPLIANT

IV. Intentional Breakups and Potential for Explosions

Potential causes of spacecraft breakup during deployment and mission operations:

There are two potential scenarios that could potentially lead to a breakup of the satellite:

1) Rupture of the propellant tank (H_20, N_2)
2) Lithium-ion battery cell failure

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 failure modes and effects analysis ("FMEA") (see requirement 4.4-1 below) describe the combined faults that must occur for any of seven (7) independent, mutually exclusive failure modes to lead to an explosion.

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

- There are no planned breakups.

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

- Twelve (12) Lithium-ion battery cells.

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

- None.

Assessment of spacecraft compliance with Requirements 4.4-1 through 4.4-4:

Requirement 4.4-1: Limiting the risk to other space systems from accidental explosions during deployment and mission operations while in orbit about Earth or the Moon:

For each spacecraft and launch vehicle orbital stage employed for a mission, the program or project shall demonstrate, via failure mode and effects analyses or equivalent analyses, that the integrated probability of explosion for all credible failure modes of each spacecraft and launch vehicle is less than 0.001 (excluding small particle impacts) (Requirement 56449).

- **Compliance statement:**
  - Required Probability: 0.001.

- Expected probability: 0.000.

Supporting Rationale and FMEA details:

- **Pressure Tank Explosion:**
  - **Effect:** A rupture of one propellant tank would release water and nitrogen. Due to the low pressure (~44 psi), the penetrating energy of any debris would be relatively low. The tank is enclosed in the solid aluminum structural walls of the spacecraft. These aluminum walls would contain any released debris within the body of the spacecraft.
  - **Probability:** Very low. A structural failure of the tank would need to occur, and the mechanisms by which these failures occur are very well understood. The factor of safety for all vessels is 2, and the surrounding structure has a higher factor of safety.
- **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 and multiple enclosures surrounding the batteries.
  - **Probability:** Extremely Low. Estimated to be less than 0.01% 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:* All of the following testing has or will be performed prior to flight: protoflight level sine burst, sine, and random vibration testing in all three axes, thermal vacuum cycling and extensive functional testing, system rate-limited charge and discharge cycles, and subsystem and component level functional testing. The testing helps prove that no internal short circuit sensitivity exists.
    - *Combined Faults Required for Realized Failure:* Environmental testing AND functional charge / discharge tests must both be ineffective in discovery of the failure mode.
  - **Failure Mode 2:** Internal thermal rise due to high load discharge rate.
    - *Mitigation 2:* Battery cells were tested in lab for high load discharge rates in a variety of flight-like configurations to determine the feasibility of an out-of-control thermal rise in the cell. Cells are also tested in a hot,

thermal vacuum environment (5 cycles at 50°C, then to -20°C) in order to test the upper limit of the cells' capability. No failures were observed or identified via satellite telemetry or via external monitoring circuitry.
    - *Combined Faults Required for Realized Failure:* Spacecraft thermal design must be incorrect AND external over-current detection and disconnect function must fail to enable this failure mode.
- **Failure Mode 3:** Excessive discharge rate or short-circuit due to external devices failure of terminal contact with conductors not at battery voltage levels (due to abrasion or inadequate proximity separation).
    - *Mitigation 3:* Qualification testing of short circuit protection on each external circuit, design of battery packs and insulators such that no contact with nearby board traces is possible without being caused by some other mechanical failure, and observation of such other mechanical failures by protoflight level environmental tests (sine burst, random vibration, thermal cycling, and thermal-vacuum tests).
    - *Combined Faults Required for Realized Failure:* An external load must fail/short-circuit AND external over-current detection and disconnect function must all occur to enable this failure mode.
- **Failure Mode 4:** Inoperable vents.
    - *Mitigation 4:* Battery venting is not inhibited by the battery holder design or the spacecraft design. The battery can vent gases to the external environment.
    - *Combined Faults Required for Realized Failure:* The cell manufacturer OR the satellite integrator fails to install proper venting.
- **Failure Mode 5:** Crushing.
    - *Mitigation 5:* This mode is negated by spacecraft design with no moving parts in the proximity of the batteries.
    - *Combined Faults Required for Realized Failure:* A catastrophic failure must occur in an external system AND the failure must cause a collision sufficient to crush the batteries leading to an internal short circuit AND the satellite must be in a naturally sustained orbit at the time the crushing occurs.
- **Failure Mode 6:** Low level current leakage or short-circuit through battery or short-circuit through battery pack case or due to moisture-based degradation of insulators.
    - *Mitigation 6:* The battery holder and case design are made of non-conductive plastic and operation in vacuum ensures that no moisture can affect insulators.

        - *Combined Faults Required for Realized Failure:* Abrasion or piercing failure of circuit board coating or wire insulators AND dislocation of battery packs AND failure of battery terminal insulators AND failure to detect such failures in environmental tests must occur to result in this failure mode.
    - **Failure Mode 7:** Excess temperatures due to orbital environment and high discharge combined.
        - *Mitigation 7:* The spacecraft thermal design will negate this possibility. Thermal rise has been analyzed in combination with space environment temperatures showing that the batteries do not exceed normal allowable operating temperatures under a variety of modeled cases, including worst case orbital scenarios. Analysis shows these temperatures to be well below temperatures of concern for explosions.
        - *Combined Faults Required for Realized Failure:* Thermal analysis AND thermal design AND mission simulations in thermal-vacuum chamber testing AND over-current monitoring and control must all fail for this failure mode to occur.

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

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

- Compliance statement:
    - VR-1 includes the ability to fully disconnect the Lithium Ion cells from the charging current of the solar arrays. At End-of-Life, this feature can be used to completely passivate the batteries by removing all energy from them. 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, the debris from the battery rupture should be contained within the spacecraft due to the lack of penetration energy to the multiple enclosures surrounding the batteries.
    - As discussed above in the propulsion system section, all energy will be released from the propulsion system prior to spacecraft deactivation. The spacecraft will be oriented such that any thrust generated from propellant release results in an orbit lowering maneuver. All thruster valves will be opened until all

propellant and pressurant are completely released. No attempt will be made to activate the RF microwave element, which will result in a “cold gas” thruster firing.

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

- Compliance statement:
    - 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:
    - Not applicable. There are no planned breakups.

V. Spacecraft Potential for On-Orbit Collisions

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:**
    - VR-1 = 0.00000; COMPLIANT.
    - Payload 1 (mass dummy) = 0.00000; COMPLIANT.
    - Payload 2 (mass dummy) = 0.00000; COMPLIANT.
    - Payload 3 (mass dummy) = 0.00000; COMPLIANT.

Requirement 4.5-2. Limiting debris generated by collisions with small objects when operating in Earth or lunar orbit:

- For each spacecraft, the program or project shall demonstrate that, during the mission of the spacecraft, the probability of accidental collision with orbital debris and meteoroids sufficient to prevent compliance with the applicable postmission disposal requirements is less than 0.01 (Requirement 56507).
- **Small Object Impact and Debris Generation Probability:**
    - 0.00000; COMPLIANT

Identification of all systems or components required to accomplish any postmission disposal operation, including passivation and maneuvering:

- None are specifically required, but the propulsion system is expected to be used for a postmission de-orbit maneuver to decrease the time until atmospheric re-entry to less than 1 year. The flight computer, radio hardware, battery system, and control boards will be used to vent all propellant and to disconnect the battery from the solar array current. The mass dummy payloads will not conduct any postmission maneuvers.

VI. Spacecraft Postmission Disposal Plans and Procedures

Description of spacecraft disposal option selected:

- VR-1 and the Payloads will de-orbit naturally by atmospheric re-entry without any intervention. However, VR-1 will attempt a de-orbit maneuver to reduce the time to atmospheric re-entry to less than a year.

Plan for any spacecraft maneuvers required to accomplish postmission disposal:

- No maneuvers are required for postmission disposal. However, VR-1 will attempt a de-orbit maneuver as a proof of concept for future missions. The maneuver will be performed using the on-board propulsion system to lower the orbital perigee to 300 km altitude. Note that there is no planned controlled re-entry.

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

- Spacecraft Final Mass: 74 kg (worst case mass)
- Cross-sectional Area: 0.59 m^2 (estimated average area in tumbling)
- Area to mass ratio: 0.008 m^2/kg

Requirement 4.6-1. Disposal for space structures passing through LEO:

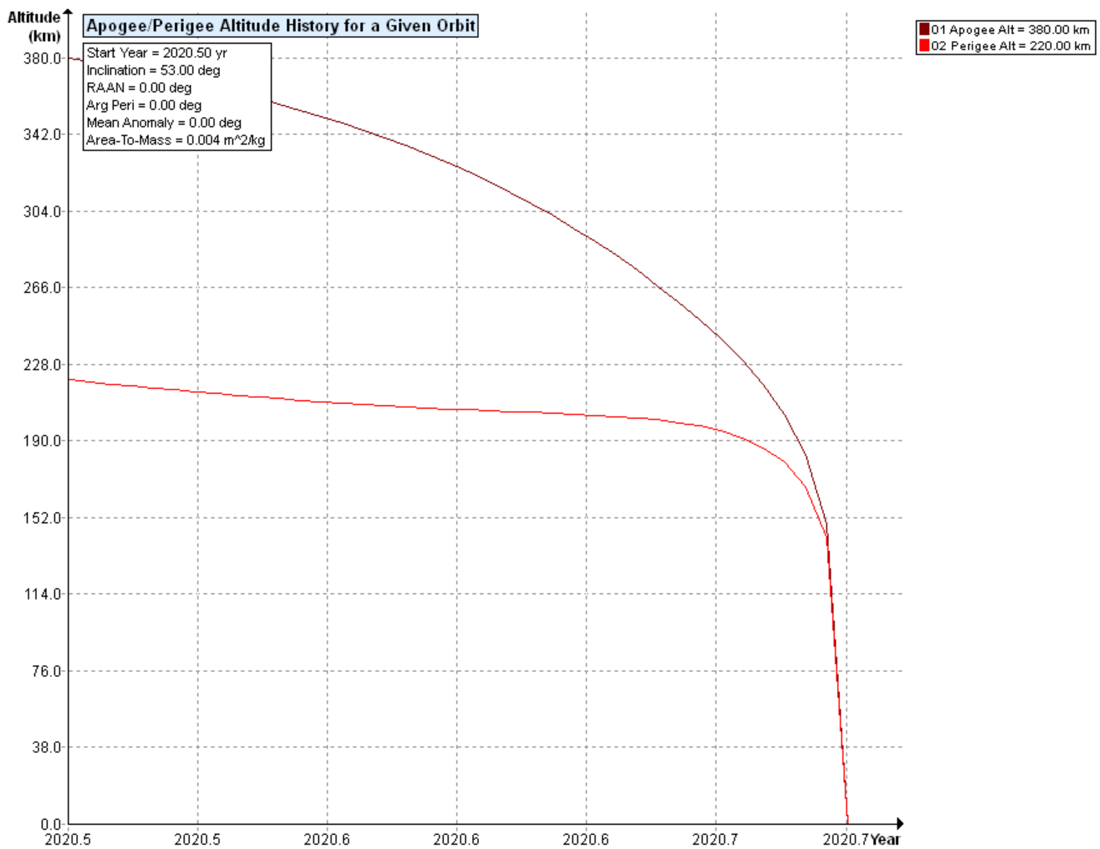
A spacecraft or orbital stage with a perigee altitude below 2000 km shall be disposed of by one of three methods: (Requirement 56557)

1. Atmospheric reentry option:
    a. 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
    b. Maneuver the space structure into a controlled de-orbit trajectory as soon as practical after completion of mission.
2. Storage orbit option: Maneuver the space structure into an orbit with perigee altitude greater than 2000 km and apogee less than GEO - 500 km.
3. Direct retrieval: Retrieve the space structure and remove it from orbit within 10 years after completion of mission.

Analysis:

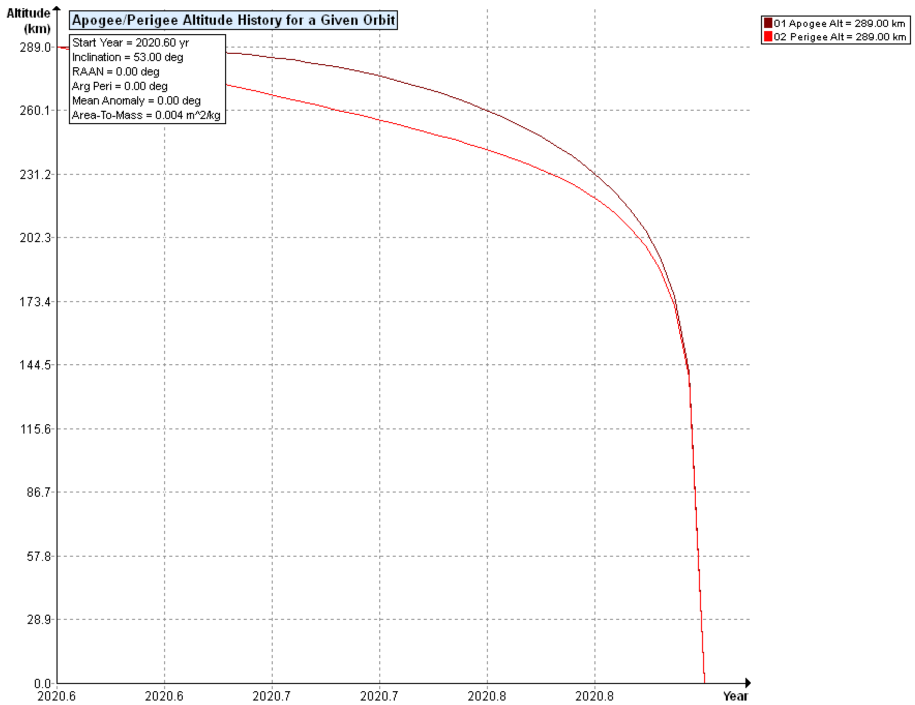
VR-1 will follow a concept of operations to ensure a safe disposal well within 25 years of the end of the mission. To demonstrate the thruster, the orbit apogee will be initially raised. Following mission completion, the thruster will be used to lower the perigee and reduce orbit lifetime to approximately 1 year. In the event of a propulsion system failure or inability to perform postmission maneuvers, the spacecraft is expected to undergo atmospheric re-entry within 3 years. Additionally, the worst case release of the payloads (Payload 3 at 500x500 km) will be within 4 years. *See* Table 1 *supra*.

VR-1 Propulsion Failure at Launch Insertion Orbital Decay (220x380 km)[8]

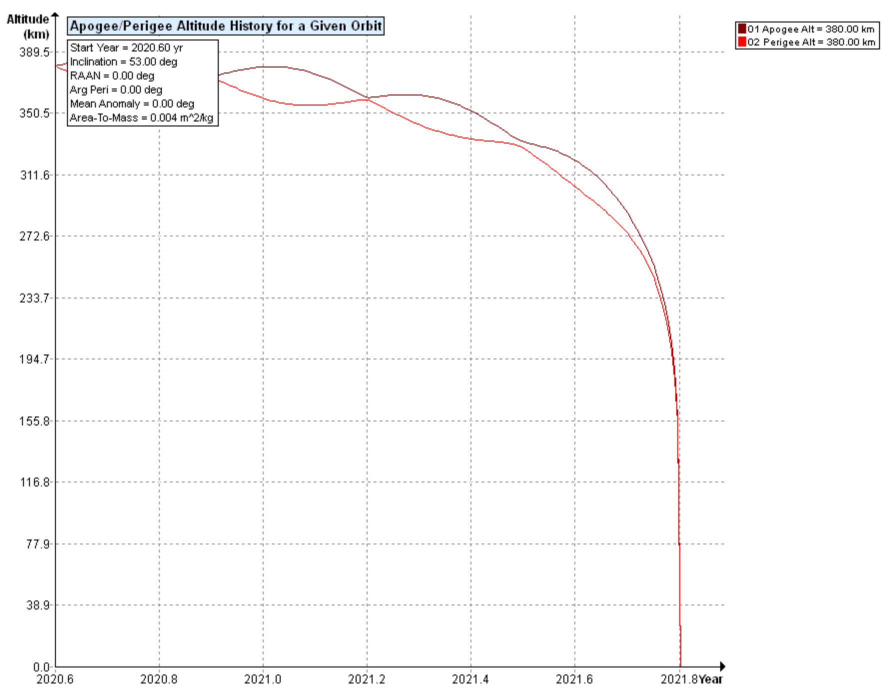


[8] VR-1 has an initial Area-to-Mass ratio of 0.004 m^2/kg prior to deployment of its solar arrays, and a final Area-to-Mass ratio of 0.008 m^2/kg post deployment of its solar arrays.

VR-1 Alternative Launch Insertion Orbital Decay (289x289 km)[9]

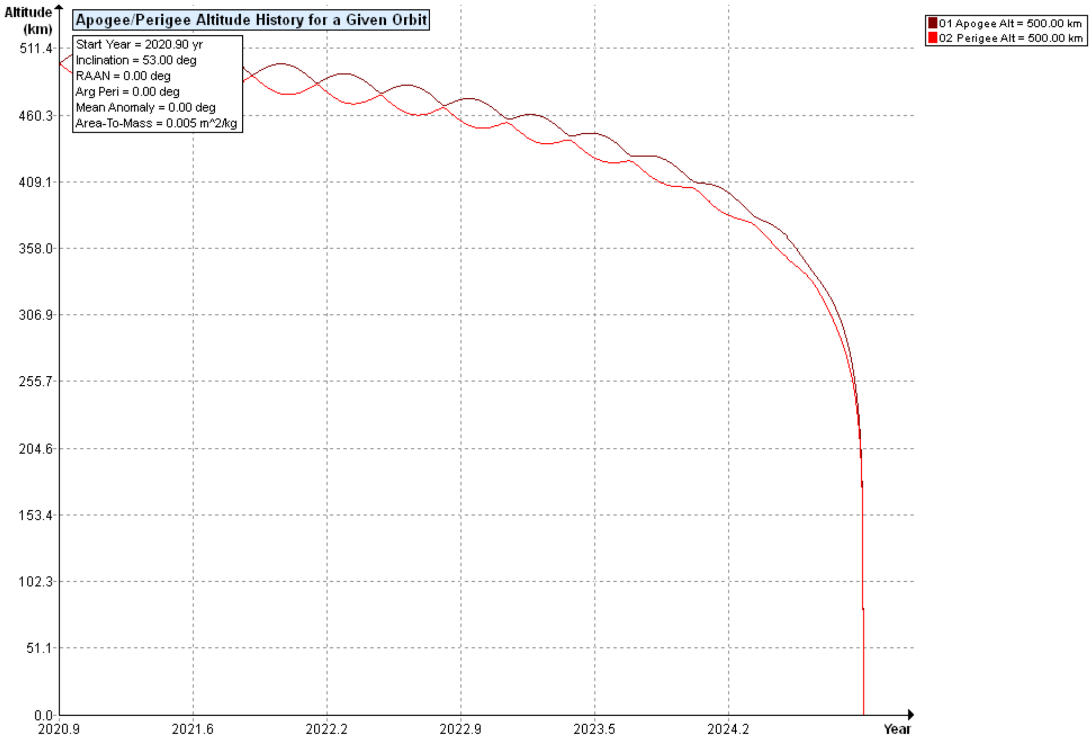


Payloads 1 and 2 Worst-Case Orbital Decay (380x380 km)

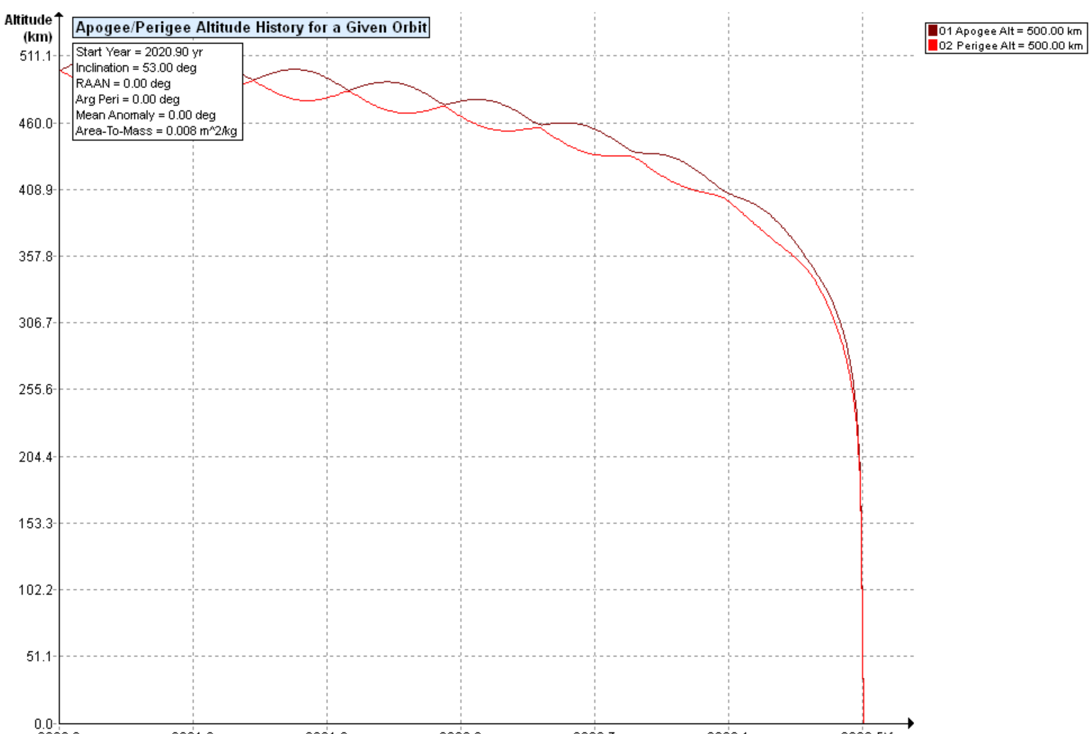


[9] In the event of a propulsive failure at the alternative launch insertion orbit of 289 km circular, orbital decay is expected within 3 months. *See* Table 1 *supra.*

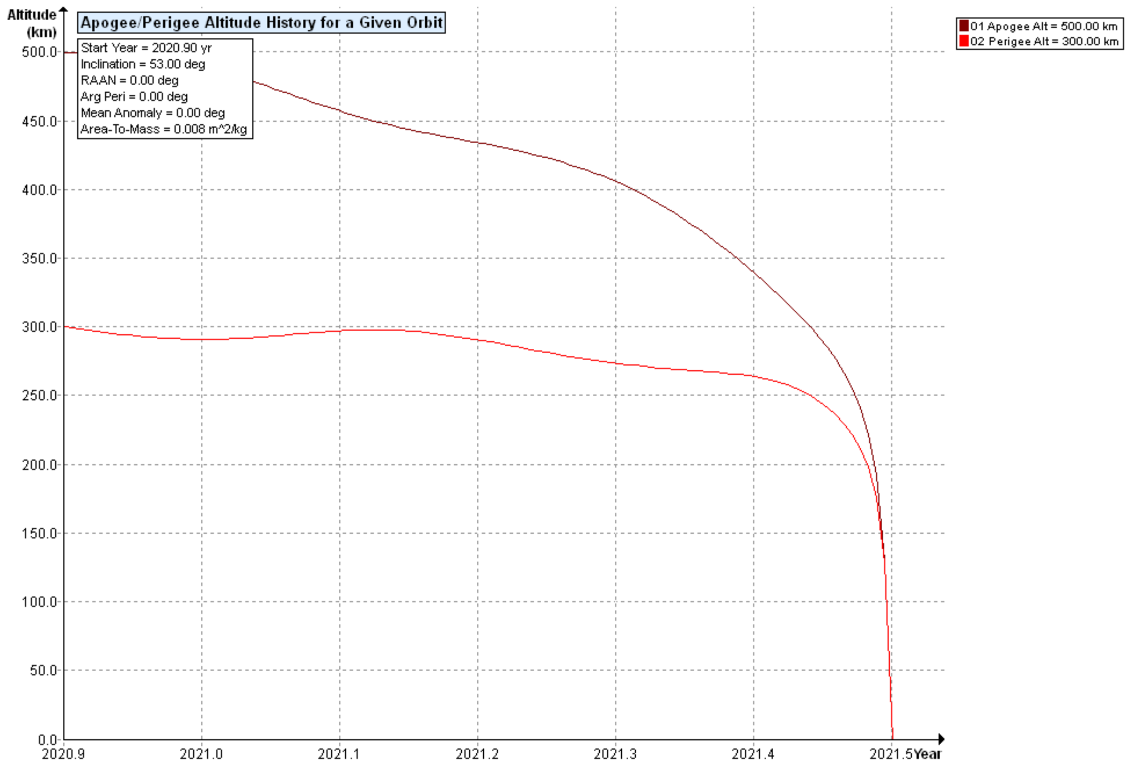
Payload 3 Worst-Case Orbital Decay (500x500 km)



VR-1 Worst-Case Postmission Orbital Decay (500x500 km)



VR-1 Planned Postmission Orbital Decay (300x500 km)



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

- Not applicable.

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

- Not applicable.

Requirement 4.6-4. Reliability of Postmission Disposal Operations

- Not applicable. The satellite will reenter passively without post mission disposal operations within allowable timeframe.

VII. Spacecraft Reentry Hazards

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:

1. 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 v2.1.1 reports that VR-1 is compliant with the requirement. As shown below, three VR-1 components may survive re-entry. However, in each case the impact energy is less than 15 joules.

Analysis using DAS v2.1.1:

11 04 2019; 15:22:11PM Processing Requirement 4.3-2: Return Status : Passed

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

No Project Data Available

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

=============== End of Requirement 4.3-2 ===============

11 04 2019; 15:22:13PM Requirement 4.4-3: Compliant

=============== End of Requirement 4.4-3 ===============

11 04 2019; 15:34:44PM Processing Requirement 4.5-1: Return Status : Passed

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

Run Data

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

INPUT

Space Structure Name = Vigoride-1

Space Structure Type = Payload

Perigee Altitude = 500.000000 (km)

Apogee Altitude = 500.000000 (km)

Inclination = 53.000000 (deg)

RAAN = 0.000000 (deg)

Argument of Perigee = 0.000000 (deg)

Mean Anomaly = 0.000000 (deg)

Final Area-To-Mass Ratio = 0.008000 (m^2/kg)

Start Year = 2020.500000 (yr)

Initial Mass = 101.000000 (kg)

Final Mass = 74.000000 (kg)

Duration = 1.000000 (yr)

Station-Kept = False

Abandoned = False

PMD Perigee Altitude = 300.000000 (km)

PMD Apogee Altitude = 500.000000 (km)

PMD Inclination = 53.000000 (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

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

INPUT

Space Structure Name = Payload 1

Space Structure Type = Payload

Perigee Altitude = 380.000000 (km)

Apogee Altitude = 380.000000 (km)

Inclination = 53.000000 (deg)

RAAN = 0.000000 (deg)

Argument of Perigee = 0.000000 (deg)

Mean Anomaly = 0.000000 (deg)

Final Area-To-Mass Ratio = 0.004000 (m^2/kg)

Start Year = 2020.500000 (yr)

Initial Mass = 10.000000 (kg)

Final Mass = 10.000000 (kg)

Duration = 1.000000 (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

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

INPUT

Space Structure Name = Payload 2

Space Structure Type = Payload

Perigee Altitude = 380.000000 (km)

Apogee Altitude = 380.000000 (km)

Inclination = 53.000000 (deg)

RAAN = 0.000000 (deg)

Argument of Perigee = 0.000000 (deg)

Mean Anomaly = 0.000000 (deg)

Final Area-To-Mass Ratio = 0.005000 (m^2/kg)

Start Year = 2020.500000 (yr)

Initial Mass = 5.500000 (kg)

Final Mass = 5.500000 (kg)

Duration = 1.000000 (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

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

INPUT

Space Structure Name = Payload 3

Space Structure Type = Payload

Perigee Altitude = 500.000000 (km)

Apogee Altitude = 500.000000 (km)

Inclination = 53.000000 (deg)

RAAN = 0.000000 (deg)

Argument of Perigee = 0.000000 (deg)

Mean Anomaly = 0.000000 (deg)

Final Area-To-Mass Ratio = 0.005000 (m^2/kg)

Start Year = 2020.500000 (yr)

Initial Mass = 5.500000 (kg)

Final Mass = 5.500000 (kg)

Duration = 1.000000 (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 ===============

11 04 2019; 15:38:28PM   Processing Requirement 4.6   Return Status : Passed

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

Project Data

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

INPUT

Space Structure Name = Vigoride-1

Space Structure Type = Payload

Perigee Altitude = 500.000000 (km)

Apogee Altitude = 500.000000 (km)

Inclination = 53.000000 (deg)

RAAN = 0.000000 (deg)

Argument of Perigee = 0.000000 (deg)

Mean Anomaly = 0.000000 (deg)

Area-To-Mass Ratio = 0.008000 (m^2/kg)

Start Year = 2020.500000 (yr)

Initial Mass = 101.000000 (kg)

Final Mass = 74.000000 (kg)

Duration = 1.000000 (yr)

Station Kept = False

Abandoned = False

PMD Perigee Altitude = 300.000000 (km)

PMD Apogee Altitude = 500.000000 (km)

PMD Inclination = 53.000000 (deg)

PMD RAAN = 0.000000 (deg)

PMD Argument of Perigee = 0.000000 (deg)

PMD Mean Anomaly = 0.000000 (deg)

OUTPUT

Suggested Perigee Altitude = 300.000000 (km)

Suggested Apogee Altitude = 500.000000 (km)

Returned Error Message = Passes LEO reentry orbit criteria.

Released Year = 2021 (yr)

Requirement = 61

Compliance Status = Pass

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

INPUT

Space Structure Name = Payload 1

Space Structure Type = Payload

Perigee Altitude = 380.000000 (km)

Apogee Altitude = 380.000000 (km)

Inclination = 53.000000 (deg)

RAAN = 0.000000 (deg)

Argument of Perigee = 0.000000 (deg)

Mean Anomaly = 0.000000 (deg)

Area-To-Mass Ratio = 0.004000 (m^2/kg)

Start Year = 2020.500000 (yr)

Initial Mass = 10.000000 (kg)

Final Mass = 10.000000 (kg)

Duration = 1.000000 (yr)

Station Kept = False

Abandoned = True

PMD Perigee Altitude = 315.929928 (km)

PMD Apogee Altitude = 333.766017 (km)

PMD Inclination = 52.993534 (deg)

PMD RAAN = 354.818234 (deg)

PMD Argument of Perigee = 74.165751 (deg)

PMD Mean Anomaly = 0.000000 (deg)

OUTPUT

Suggested Perigee Altitude = 315.929928 (km)

Suggested Apogee Altitude = 333.766017 (km)

Returned Error Message = Passes LEO reentry orbit criteria.

Released Year = 2021 (yr)

Requirement = 61

Compliance Status = Pass

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

INPUT

Space Structure Name = Payload 2

Space Structure Type = Payload

Perigee Altitude = 380.000000 (km)

Apogee Altitude = 380.000000 (km)

Inclination = 53.000000 (deg)

RAAN = 0.000000 (deg)

Argument of Perigee = 0.000000 (deg)

Mean Anomaly = 0.000000 (deg)

Area-To-Mass Ratio = 0.005000 (m^2/kg)

Start Year = 2020.500000 (yr)

Initial Mass = 5.500000 (kg)

Final Mass = 5.500000 (kg)

Duration = 1.000000 (yr)

Station Kept = False

Abandoned = True

PMD Perigee Altitude = 283.230355 (km)

PMD Apogee Altitude = 299.001506 (km)

PMD Inclination = 52.989437 (deg)

PMD RAAN = 348.450187 (deg)

PMD Argument of Perigee = 75.916159 (deg)

PMD Mean Anomaly = 0.000000 (deg)

OUTPUT

Suggested Perigee Altitude = 283.230355 (km)

Suggested Apogee Altitude = 299.001506 (km)

Returned Error Message = Passes LEO reentry orbit criteria.

Released Year = 2021 (yr)

Requirement = 61

Compliance Status = Pass

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

INPUT

Space Structure Name = Payload 3

Space Structure Type = Payload

Perigee Altitude = 500.000000 (km)

Apogee Altitude = 500.000000 (km)

Inclination = 53.000000 (deg)

RAAN = 0.000000 (deg)

Argument of Perigee = 0.000000 (deg)

Mean Anomaly = 0.000000 (deg)

Area-To-Mass Ratio = 0.005000 (m^2/kg)

Start Year = 2020.500000 (yr)

Initial Mass = 5.500000 (kg)

Final Mass = 5.500000 (kg)

Duration = 1.000000 (yr)

Station Kept = False

Abandoned = True

PMD Perigee Altitude = 489.802276 (km)

PMD Apogee Altitude = 500.007504 (km)

PMD Inclination = 52.997927 (deg)

PMD RAAN = 116.044941 (deg)

PMD Argument of Perigee = 29.117352 (deg)

PMD Mean Anomaly = 0.000000 (deg)

OUTPUT

Suggested Perigee Altitude = 489.802276 (km)

Suggested Apogee Altitude = 500.007504 (km)

Returned Error Message = Passes LEO reentry orbit criteria.

Released Year = 2024 (yr)

Requirement = 61

Compliance Status = Pass

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

============== End of Requirement 4.6 ===============

11 04 2019; 15:38:31PM Requirement 4.5-2: Compliant

11 04 2019; 15:39:13PM *********Processing Requirement 4.7-1

Return Status : Passed

***********INPUT****

Item Number = 1

name = Vigoride-1

quantity = 1

parent = 0

materialID = 8

type = Box

Aero Mass = 74.000000

Thermal Mass = 74.000000

Diameter/Width = 0.600000

Length = 0.600000

Height = 0.250000

name = Structure

quantity = 2

parent = 1

materialID = 8

type = Box

Aero Mass = 7.700000

Thermal Mass = 7.700000

Diameter/Width = 0.250000

Length = 0.600000

Height = 0.250000

name = Diaphragm Tank

quantity = 2

parent = 1

materialID = 27

type = Sphere

Aero Mass = 1.800000

Thermal Mass = 1.800000

Diameter/Width = 0.200000

name = Prototype Tank

quantity = 2

parent = 1

materialID = 8

type = Box

Aero Mass = 1.800000

Thermal Mass = 1.800000

Diameter/Width = 0.150000

Length = 0.200000

Height = 0.150000

name = Chamber

quantity = 1

parent = 1

materialID = 54

type = Cylinder

Aero Mass = 0.030000

Thermal Mass = 0.030000

Diameter/Width = 0.030000

Length = 0.032000

name = Window

quantity = 1

parent = 1

materialID = 1

type = Cylinder

Aero Mass = 0.008000

Thermal Mass = 0.008000

Diameter/Width = 0.030000

Length = 0.015000

name = Reaction Wheels

quantity = 3

parent = 1

materialID = 54

type = Cylinder

Aero Mass = 0.080000

Thermal Mass = 0.080000

Diameter/Width = 0.080000

Length = 0.060000

name = Reaction Control Thruster

quantity = 4

parent = 1

materialID = 19

type = Box

Aero Mass = 0.085000

Thermal Mass = 0.085000

Diameter/Width = 0.050000

Length = 0.050000

Height = 0.040000

name = Shields

quantity = 4

parent = 1

materialID = 66

type = Flat Plate

Aero Mass = 0.025000

Thermal Mass = 0.025000

Diameter/Width = 0.060000

Length = 0.060000

name = 6U CubeSat Deployer

quantity = 1

parent = 1

materialID = 8

type = Box

Aero Mass = 4.000000

Thermal Mass = 4.000000

Diameter/Width = 0.350000

Length = 0.600000

Height = 0.200000

name = Radiator Plate

quantity = 1

parent = 1

materialID = 8

type = Box

Aero Mass = 3.500000

Thermal Mass = 3.500000

Diameter/Width = 0.600000

Length = 0.600000

Height = 0.020000

name = Solar Array

quantity = 2

parent = 1

materialID = 8

type = Flat Plate

Aero Mass = 2.600000

Thermal Mass = 2.600000

Diameter/Width = 0.600000

Length = 1.400000

name = Dual 3U Deployer

quantity = 1

parent = 1

materialID = 8

type = Box

Aero Mass = 4.500000

Thermal Mass = 4.500000

Diameter/Width = 0.350000

Length = 0.600000

Height = 0.200000

name = Payload Interface Adapter

quantity = 2

parent = 1

materialID = 8

type = Flat Plate

Aero Mass = 2.000000

Thermal Mass = 2.000000

Diameter/Width = 0.200000

Length = 0.300000

name = Payload Interface Plate

quantity = 1

parent = 1

materialID = 8

type = Flat Plate

Aero Mass = 4.000000

Thermal Mass = 4.000000

Diameter/Width = 0.400000

Length = 0.400000

**************OUTPUT****

Item Number = 1

name = Vigoride-1

Demise Altitude = 77.997406

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Structure

Demise Altitude = 70.753174

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Diaphragm Tank

Demise Altitude = 75.568542

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Prototype Tank

Demise Altitude = 72.426384

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Chamber

Demise Altitude = 72.840271

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Window

Demise Altitude = 0.000000

Debris Casualty Area = 0.385906

Impact Kinetic Energy = 1.099591

****

name = Reaction Wheels

Demise Altitude = 0.000000

Debris Casualty Area = 1.343816

Impact Kinetic Energy = 11.485363

****

name = Reaction Control Thruster

Demise Altitude = 76.222977

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Shields

Demise Altitude = 0.000000

Debris Casualty Area = 1.742400

Impact Kinetic Energy = 2.833810

****

name = 6U CubeSat Deployer

Demise Altitude = 73.721046

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Radiator Plate

Demise Altitude = 69.989639

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Solar Array

Demise Altitude = 74.414131

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Dual 3U Deployer

Demise Altitude = 73.177330

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Payload Interface Adapter

Demise Altitude = 69.054794

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Payload Interface Plate

Demise Altitude = 65.286514

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

***********INPUT****

Item Number = 2

name = Payload 1

quantity = 1

parent = 0

materialID = 8

type = Box

Aero Mass = 10.000000

Thermal Mass = 10.000000

Diameter/Width = 0.210000

Length = 0.340000

Height = 0.110000

name = P

quantity = 1

parent = 1

materialID = 8

type = Box

Aero Mass = 10.000000

Thermal Mass = 10.000000

Diameter/Width = 0.210000

Length = 0.340000

Height = 0.110000

**************OUTPUT****

Item Number = 2

name = Payload 1

Demise Altitude = 77.999893

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = P

Demise Altitude = 55.182610

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

***********INPUT****

Item Number = 3

name = Payload 2

quantity = 1

parent = 0

materialID = 8

type = Box

Aero Mass = 5.500000

Thermal Mass = 5.500000

Diameter/Width = 0.110000

Length = 0.340000

Height = 0.110000

name = P

quantity = 1

parent = 1

materialID = 8

type = Box

Aero Mass = 5.500000

Thermal Mass = 5.500000

Diameter/Width = 0.110000

Length = 0.340000

Height = 0.110000

**************OUTPUT****

Item Number = 3

name = Payload 2

Demise Altitude = 77.995438

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = P

Demise Altitude = 63.544582

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

***********INPUT****

Item Number = 4

name = Payload 3

quantity = 1

parent = 0

materialID = 8

type = Box

Aero Mass = 5.500000

Thermal Mass = 5.500000

Diameter/Width = 0.110000

Length = 0.340000

Height = 0.110000

name = P

quantity = 1

parent = 1

materialID = 8

type = Box

Aero Mass = 5.500000

Thermal Mass = 5.500000

Diameter/Width = 0.110000

Length = 0.340000

Height = 0.110000

**************OUTPUT****

Item Number = 4

name = Payload 3

Demise Altitude = 77.995438

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = P

Demise Altitude = 63.544582

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

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

VIII. Tether Missions

Not applicable.