PREPARED BY Ricky Prasad Senior Systems Engineer	**CAGE CODE 6J8J8**	**NUMBER**
APPROVALS		**TYPE** Analysis Procedure
Approved By: Ehson Mosleh VP Space Payloads		**DATE** **Final release date**
Concurrence: Austin Williams VP Space Vehicles		**SUPERSEDES SPEC DATED** **NA**
		REV **01**

TITLE: CICERO Orbital Debris Assessment Report (ODAR) / End of Mission Plan (EOMP)

Warnings and Disclaimers:

All future revisions to this document shall be approved by the controlling organization prior to release.

REVISION SUMMARY

REV NO.	RELEASE DATE	BRIEF DESCRIPTION/REASON FOR CHANGE	EFFECTIVE PAGES
01	2/23/15	Initial draft for review	All

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ORBITAL DEBRIS SELF-ASSESSMENT: CICERO MISSION

Requirement	Launch Vehicle				Spacecraft			Comments
	Compliant	Not Compliant	Incomplete	Standard Non Compliant	Compliant	Not Compliant	Incomplete	
4.3-1.a			X		X			No debris released in LEO
4.3-1.b			X		X			No debris released in LEO
4.3-2			X		X			No debris released in GEO
4.4-1			X		X			Less than 0.001 probability
4.4-2			X		X			Design to passivate propulsion, electrical power system, and reaction wheels
4.4-3			X		X			No planned breakups
4.4-4			X		X			No planned breakups
4.5-1			X		X			Probability 0.00000 (requirement < 0.001)
4.5-2			X		X			Probability 0.00000 (requirement < 0.01)
4.6-1(a)			X		X			Predicted orbital lifetime 15.18 years
4.6-1(b)			X		X			N/A – using atmospheric entry
4.6-1(c)			X		X			N/A – using atmospheric entry
4.6-2			X		X			N/A – Not GEO
4.6-3			X		X			N/A – Not between LEO and GEO
4.6-4			X		X			Expected probability < 0.001
4.7-1			X		X			No pieces survive reentry
4.8-1					X			No tethers used

CICERO is currently manifested to fly as a secondary “rideshare” payload. Compliance with requirements levied by NASA-STD 9719.14A on the launch vehicle are not applicable to this document and the responsibility of the launch provider.

1.0 PROGRAM MANAGEMENT AND MISSION OVERVIEW

1.1 Program Management

Parameter	Value
Mission Directorate	N/A
Program Executive	Tom Yunck (GeoOptics) / Marco Villa (Tyvak)
Program/project Manager	Marco Villa (Tyvak)
Senior Scientist	Tom Yunck (GeoOptics)
Senior Management	N/A
Foreign government or space agency participation	N/A
Summary of NASA's responsibility under the governing agreement(s)	N/A

Table 1-1: Summary of Program Management Personnel

1.2 Mission Overview

1.2.1 Mission Design and Development Milestones

The schedule of mission design and development milestones is provided in Table 1.2.

Project Kickoff	March 2015
PDR	May 2015
CDR	September 2015
System Integration Completed	April 2016
Environmental Testing	May 2016
Launch	September 2016

Table 1.2 – Summary of Mission Design and Development Milestones

1.2.2 Mission Overview

The goal of the CICERO Mission is to perform GPS Radio Occultation (RO) measurement demonstration utilizing a single 6U CubeSat. The collection of RO data will be used to validate the CICERO system and quality of data collected.

Parameter	Value
Launch vehicle and launch site	Soyuz, Baikonur Cosmodrome, Kazakhstan
Proposed launch date	Q2/Q3 2016
Mission duration	1+ year
Launch and deployment profile	The Soyuz launch vehicle will launch the primary mission satellite. After which, it will deploy the CICERO satellites into their final mission orbit

	(~600km[1], circular, sun-synchronous orbit ~ 97.8° inclination). There is no parking or transfer orbit. The CICERO satellite will decay naturally for debris mitigation and will re-enter within 25 years after completion of mission.

Table 1-2: Summary of Mission Parameters

[1] Worst case expected orbit for the Soyuz vehicle for orbit insertion is 610 km. This will be used for all de-orbit calculations to bound the mission by the worst case expected.

2.0 SPACECRAFT DESCRIPTION

2.1 Physical Description of Spacecraft

The CICERO vehicles have been designed to support a 1+ year mission in LEO, and it is compatible with the P-POD launch environments and designed to the requirements in the CubeSat Design Specification (CDS). The CICERO vehicle is a 6U CubeSat with the vehicle being 30cm x 20cm x 10cm with a mass of roughly 9.2 kg.

The CICERO vehicle design uses subsystem modules built from printed circuit boards (PCB) or miniature enclosures mounted to the open frame primary structure. The open structure permits the vehicle to be built incrementally with open access for securing interconnects. The subsystems are placed within the vehicle to optimize mass properties, radiation protection, thermal heat rejection, power handling, vehicle orientation, and cabling length. The body mounted side panels attach directly to the primary structure and are used for thermal management and can be easily removed to get access to the interior of the vehicle. The vehicle is primarily constructed out of aluminum and PCB materials.

The CICERO payload utilizes a GPS array antenna mounted to the Minus-Y panel to receive GPS RO signals. The signals are then captured by the CION Payload board which outputs the data files to the CICERO CDH for later transmission to the ground via UHF.

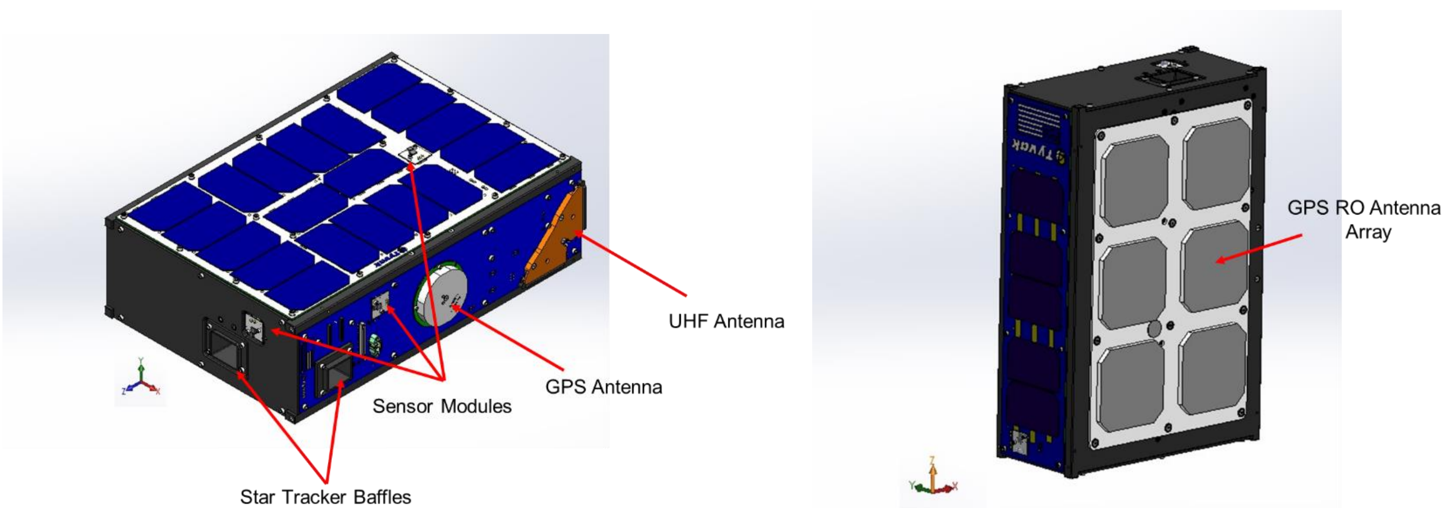


Figure 2-1: CICERO Vehicle Layout

Parameter	Value
Total satellite mass at launch, including all propellants and fluids	~9.2 kg
Dry Mass of satellite at launch, excluding solid rocket motor propellants	~9.2 kg
Identification, including mass and pressure, of all fluids	NONE. CICERO has no propulsion
Fluids in Pressurized batteries	NONE. CICERO uses unpressurized standard COTS Li-ion battery cells
Identification of any other sources of stored energy	NONE
Identification of any radioactive materials on board	NONE

Table 2-1: Summary of Spacecraft Parameters

2.1.1 Description of Propulsion Systems

None.

2.1.2 Description of attitude control system

The CICERO attitude determination and control system consists of a processor, Inertial Reference Module (IRM), nano-Reaction Wheel Array (nRWA), GPS receiver, Sun sensors, magnetometers, and integrated torque coils. Primary attitude knowledge is provided by the IRM which hosts two star sensors and the inertial measurement unit (IMU). Primary attitude control is provided by the nRWA which consists of an orthogonal set of three wheels. Momentum management and vehicle detumble are provided by a set of three torque coils.

2.1.3 Description of normal attitude of the spacecraft with respect to the velocity vector

The nominal attitude of the CICERO vehicles is an LVLH orientation with the long axis (z-axis) facing Nadir towards the Earth and the Y-axis aligned with the velocity vector. The vehicle will rotate about the x-axis for pointing the boresight of the vehicle GPS array at the Earth limb during RO collections and will slew for sun-pointing periodically.

2.1.4 Description of any range safety or other pyrotechnic devices

None.

2.1.5 Description of the electrical generation and storage system

Energy generation is accomplished using the primary Plus-Y solar panels and additional solar cells that are mounted on the Minus-X panel. Energy storage is accomplished using standard COTS Li-ion battery cells in a 3S2P (3 in series, 2 parallel) configuration. CICERO utilizes two parallel 3S2P packs to increase the overall capacity of the vehicle. The cells are recharged by the solar cells mounted on the deployable and body panels. The power management and distribution is provided by the electrical power system and battery protection circuitry.

3.0 ASSESSMENT OF SPACECRAFT DEBRIS RELEASED DURING NORMAL OPERATIONS

No intentional release of any object > 1mm is expected.

Parameter	**Value**
Identification of any object (>1mm) expected to be released from the spacecraft at any time after launch	**None**
Rationale/necessity for release of 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 of each object after release	N/A
Calculated orbital lifetime of each object	N/A
Compliance 4.3-1 Mission related debris passing through GEO	COMPLIANT
Compliance 4.3-2 Mission related debris passing through LEO	COMPLIANT

Table 3-1: Summary of Spacecraft Debris Released During Normal Operations

4.0 ASESSMENT OF SPACECRAFT POTENTIAL FOR EXPLOSIONS AND INTENTIONAL BREAKUPS

4.1 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.

4.2 Summary of failure modes and effects analysis of all credible failure modes

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 (Appendix A, see requirement 4.4-1) describe the combined faults that must occur for any of seven (7) independent, mutually exclusive failure modes to lead to explosion.

Over-pressure due to temperature control failure or crushing of the propulsion tank could lead to vent or burst of the propulsion tank. The propulsion safety systems discussed in the FMEA (Appendix A, see requirement 4.4-1) describe the combined faults that must occurs for any of the three (3) independent, mutually exclusive failure modes to lead to tank failure.

4.3 Detailed plan for any designed spacecraft breakup

There are no planned breakups.

4.4 List of components which shall be passivated at End of Mission (EOM)

The nRWA will be passivated at EOM through a series of commands to reduce wheel momentum to a minimum level and then to transition the vehicle to free drift mode.

The batteries will be passivated by discharging the cells to a minimum state and then disconnecting them from the solar panels and charging circuitry.

4.5 Rational for all items which are required to be passivated, but cannot be due to their design

None.

4.6 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 COMPLIANT

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

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

Compliance statement:

The batteries will be passivated by discharging the cells to a minimum state and then disconnecting them from the solar panels and charging circuit. In the unlikely event that a battery cell does explosively rupture, the small size, mass, and potential energy of these batteries is such that while the spacecraft could be expected to vent gases, most debris from the battery rupture would be contained within the vehicle due to lack of penetration energy and also because the cells are housed in a substantial aluminum bracket.

The nRWA will be passivated at EOM through a series of commands to reduce wheel momentum to a minimum level and then to transition the vehicle to free drift mode.

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.

5.0 ASSESSMENT OF SPACECRAFT POTENTIAL FOR ON-ORBIT COLLISIONS

5.1 Assessment of spacecraft compliance with Requirements 4.5-1 and 4.5-2:

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).*

Compliance statement: (Large Object Impact and Debris Generation Probability)

Required Probability: 0.001

Expected probability: 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).*

Compliance statement: (Small Object Impact and Debris Generation Probability)

Required Probability: 0.01

Expected probability: 0.00000 COMPLIANT

6.0 ASSESSMENT OF SPACECRAFT POSTMISSION DISPOSAL PLANS AND PROCEDURES

6.1 Description of spacecraft disposal option selected

The satellite will de-orbit naturally by atmospheric re-entry. The propulsion system is not used for re-entry.

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

None.

6.3 Calculation of area-to-mass ratio after postmission disposal:

Spacecraft Mass: ~9.2 kg (dry mass)

Cross-sectional Area: 0.07684 m^2

Area to mass ratio: (0.07684 m^2)/(9.2kg) = 0.0083521739 m^2/kg

6.4 Assessment of spacecraft compliance with Requirements 4.6-1 through 4.6-5:

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.*

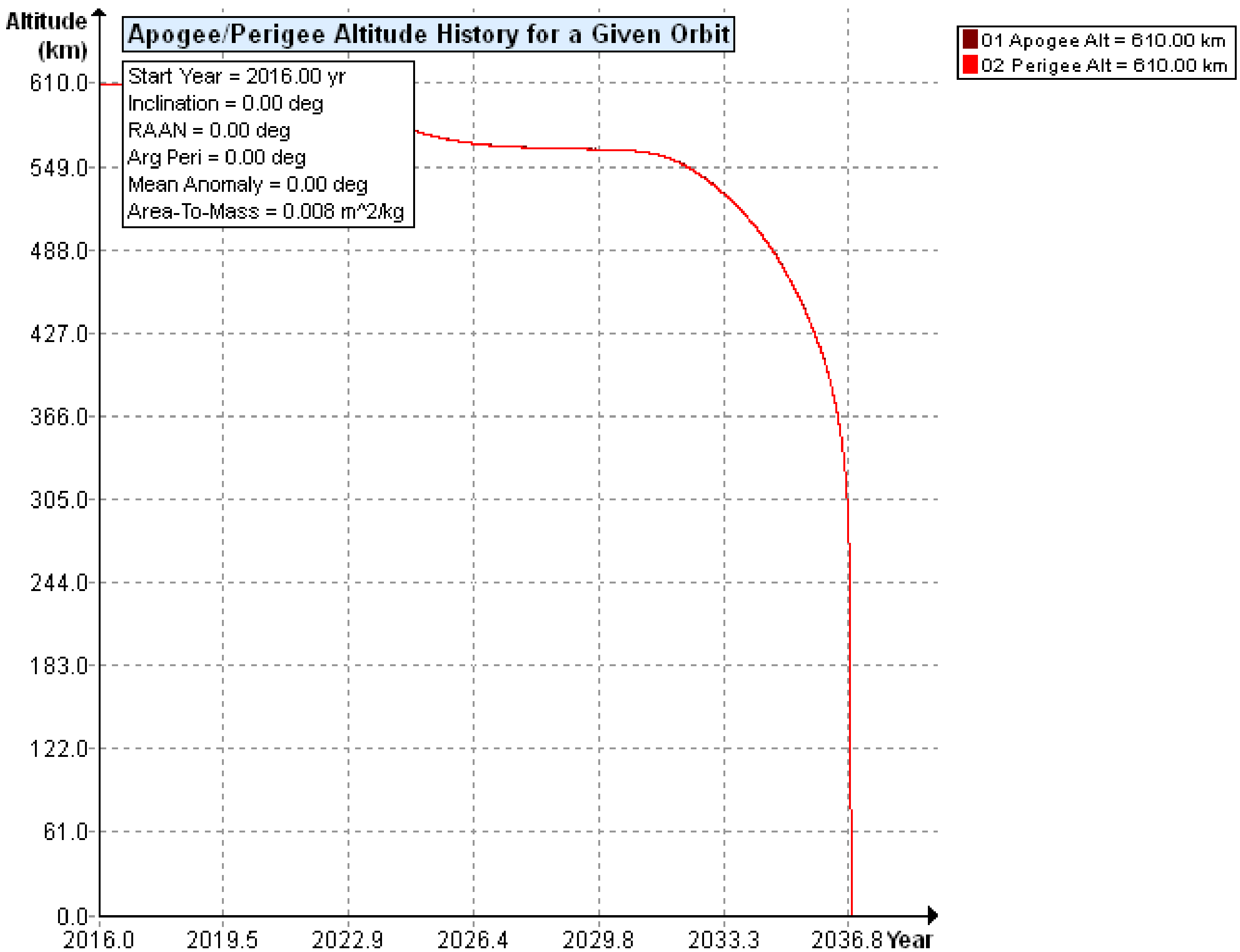
c. Direct retrieval:

- *Retrieve the space structure and remove it from orbit within 10 years after completion of mission*

Compliance statement:

The orbit used for disposal of structure analysis is a worst case orbit of 610 km. All orbits below this will have a lower orbital lifetime. The orbital lifetime is predicted to be 20.753 years; COMPLIANT

Figure 6-1: CICERO Deorbit Lifetime



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

<u>Compliance statement:</u>

Not applicable. CICERO mission orbit is a LEO.

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

Compliance statement:

Not applicable. CICERO mission orbit is a LEO.

Requirement 4.6-4. *Reliability of Postmission Disposal Operations*

Compliance statement:

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

6.5 Detailed plan for passivating (depleting all energy sources) of the spacecraft:

The nRWA will be passivated at EOM through a series of commands to reduce wheel momentum to a minimum level and then to transition the vehicle to free drift mode. The free drift mode does not utilize any attitude control actuators, specifically the nRWA. The power service to the nRWA will also be deactivated so that no inadvertent switch to another attitude control mode can actuate the nRWA.

The batteries will be passivated by permanently disconnecting solar array power from the battery module and discharging the cells to a minimum state under load of the spacecraft bus.

7.0 ASSESSMENT OF SPACECRAFT REENTERY HAZARDS

7.1 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).

Compliance statement:

DAS v2.0.2 reports that CICERO is COMPLIANT with the requirement. The vehicle is primarily composed of Aluminum and PCB (Fiberglass) material and none of the components is expected to survive re-entry. The predicted Total Debris Casualty Area is 0.00. Appendix D located in the back of this report contains the DAS 2.0.2 modeling input and results.

Requirement 4.7-1., b) *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).*

Compliance statement:

Not applicable. No controlled reentry planned.

Requirement 4.7-1., c) 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).

Compliance statement:

Not applicable. No controlled reentry planned.

8.0 ASSESSMENT FOR TETHER MISSIONS

Not applicable. There are no tethers in the CICERO mission.

APPENDIX A – FMEA DETAILS AND SUPPORTING RATIONALE

Propulsion Module Failure:

Effect: All failure modes below might result in Propulsion explosion with the possibility of orbital debris generation. The pressure vessel burst pressure has been designed with 2.5 times operating pressure at maximum storage temperature.

Probability: Very Low. It is believed to be less than 0.1% given that multiple independent (not common mode) faults must occur for each failure mode to cause the ultimate effect (explosion).

Failure mode 1: Overpressure due to the Propellant Tank Heater (H2) failed powered. The propellant temperature is normally controlled by a thermostat circuit using a thermistor (T2) to sense temperature. Should H2 fail powered, propellant temperature could exceed the maximum qualification temperature of 70°C causing the propellant pressure to exceed the corresponding design pressure of 307 psia.

Mitigation 1: The Propulsion Module pressure boundary is designed and verified by analysis to withstand a Burst Pressure of 768 psia without external leakage. The system will be tested to a Proof Pressure of 461 psia.

Mitigation 2: The Propulsion Module Controller has a separate fail-safe circuit that will cut power to H2 should the pressure as measured by P2 rise above 315 psia.

Combined faults required for realized failure: Both the thermostat using T2 **AND** the fail-safe circuit using P2 must fail with H2 powered to realize the failure.

Failure mode 2: Overpressure due to the Gas Volume Heater (H1) failed powered. The propellant temperature is normally controlled by a thermostat circuit using a thermistor (T1) to sense temperature. Should H1 fail powered, propellant temperature could exceed the maximum qualification temperature of 70°C causing the propellant pressure to eventually exceed the corresponding design pressure of 307 psia.

Mitigation 1: The Propulsion Module Gas Volume pressure boundary is designed and verified by analysis to withstand a Burst Pressure of 768 psia without external leakage. The system will be tested to a Proof Pressure of 461 psia.

Mitigation 2: The Propulsion Module Controller has a separate fail-safe circuit that will cut power to H1 should the pressure as measured by P1 rise above 315 psia.

Combined faults required for realized failure: Both the thermostat using T1 **AND** the fail-safe circuit using P1 must fail with H1 powered to realize the failure.

Failure Mode 3: Crushing.

Mitigation 2: This mode is negated by spacecraft design. There are no moving parts in the proximity of the propulsion module with sufficient kinetic energy to damage the unit.

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 tank leading to a pressure vessel or electronics failure.

Battery Explosion Failure:

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. The battery is housed within a substantial aluminum bracket.

Probability: Very Low. It is believed to be less than 0.1% given that multiple independent (not common mode) faults must occur for each failure mode to cause the ultimate effect (explosion).

Failure mode 1: Battery Internal short circuit.

Mitigation 1: Qualification and acceptance tests include vibration, thermal cycling, and vacuum tests followed by maximum system rate-limited charge and discharge to prove that no internal short circuit sensitivity exists.

Mitigation 2: Over/under voltage cell protection circuitry guards against stress conditions that can cause the development of internal shorts.

Combined faults required for realized failure: Environmental testing **AND** functional charge/discharge tests must both be ineffective in discovery of infant mortality failure rate (IMFR) related faults **OR** protection circuitry malfunctions and fails to protect cells from stress conditions.

Failure Mode 2: Internal thermal rise due to high load discharge rate.

Mitigation 3: Each cell includes an internal positive temperature coefficient (PTC) variable resistance device that reduces discharge current as cell temperature increases to prevent thermal runaway.

Mitigation 4: External under-voltage lockout circuitry disconnects battery when battery discharge voltage droop crosses a predefined threshold.

Combined faults required for realized failure: Spacecraft thermal design must be incorrect **AND** internal **AND** external over current detection and protection must fail for this failure mode to occur.

Failure Mode 3: Overcharging and excessive charge rate.

Mitigation 5: The satellite bus battery charging circuit design eliminates the possibility of the batteries being overcharged if circuits function nominally. This circuit will be extensively bench-tested and be proto-qualified for survival in vibration, and thermal-vacuum environments. The charge circuit disconnects the incoming current when cell voltage indicates normal full charge at 4.2V and limits charge current within battery specification. If this circuit fails to operate, continuing or excessive charge current 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 limit charge voltage **AND** the PTC device must fail (or temperatures generated must be insufficient to cause the PTC device to modulate) **AND** the overpressure relief device must be inadequate to vent generated gasses at acceptable rates to avoid explosion.

2) For excessive charge rate: The charge control circuitry must fail to limit charge current **AND** the PTC device must fail (or temperatures generated must be insufficient to cause the PTC device to modulate) **AND** the overpressure relief device must be inadequate to vent generated gasses at acceptable rates to avoid explosion.

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 6: 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 or structure 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 PTC must fail AND an external load must fail/short-circuit **AND** external over-current detection and disconnect function must fail to enable this failure mode.

Failure Mode 5: Inoperable vents.

Mitigation 7: Battery vents are not inhibited by the battery holder design or the spacecraft.

Combined effects required for realized failure: The spacecraft design inhibits cell venting, or cell venting clearance is sensitive to environmental stress.

Failure Mode 6: Crushing.

Mitigation 8: This mode is negated by spacecraft design. There are no moving parts in the proximity of the batteries. Qualification and acceptance tests including vibration, thermal cycling, and vacuum tests will demonstrate cell venting clearance insensitivity to environmental stress.

Combined faults required for realized failure: A catastrophic failure must occur in an external system **AND** the failure must cause a collision sufficient to crush the batteries leading to an internal short circuit **AND** the satellite must be in a naturally sustained orbit at the time the crushing occurs.

Failure Mode 7: Excess temperatures due to orbital environment and high discharge combined.

Mitigation 9: The spacecraft thermal design will negate this possibility. Thermal rise will be analyzed in combination with space environment temperatures showing that batteries do not exceed normal allowable operating temperatures which are well below temperatures of concern for explosions.

Combined faults required for realized failure: Thermal analysis **AND** thermal design **AND** mission simulations in thermal-vacuum chamber testing **AND** the PTC device must fail **AND** over-current monitoring and control must all fail for this failure mode to occur.

Failure Mode 8: Polarity Reversal Due to Over-Discharge

Mitigation 10: The spacecraft battery chemistry (Li-ion) is not susceptible to polarity reversal due to over-discharge.

Combined faults required for realized failure: Spacecraft battery module assembled with incorrect cell chemistry AND failure of cell protection circuitry

APPENDIX B - REQUIREMENT 4.5-1 DAS 2.0.1 LOG

```
01 26 2016; 12:55:27PM Processing Requirement 4.5-1: Return Status : Passed
==============
Run Data
==============
**INPUT**
    Space Structure Name = CICERO
    Space Structure Type = Payload
    Perigee Altitude = 610.000000 (km)
    Apogee Altitude = 610.000000 (km)
    Inclination = 97.800000 (deg)
    RAAN = 0.000000 (deg)
    Argument of Perigee = 0.000000 (deg)
    Mean Anomaly = 0.000000 (deg)
    Final Area-To-Mass Ratio = 0.008352 (m^2/kg)
    Start Year = 2016.000000 (yr)
    Initial Mass = 9.200000 (kg)
    Final Mass = 9.200000 (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 = 0.000005
    Returned Error Message: Normal Processing
    Date Range Error Message: Normal Date Range
    Status = Pass
==============

=============== End of Requirement 4.5-1 ===============
```

APPENDIX C - REQUIREMENT 4.6 DAS 2.0.1 LOG

```
01 26 2016; 13:26:20PM Processing Requirement 4.6 Return Status : Passed
==============
Project Data
==============
**INPUT**
    Space Structure Name = CICERO
    Space Structure Type = Payload
    Perigee Altitude = 610.000000 (km)
    Apogee Altitude = 610.000000 (km)
    Inclination = 97.800000 (deg)
    RAAN = 0.000000 (deg)
    Argument of Perigee = 0.000000 (deg)
    Mean Anomaly = 0.000000 (deg)
    Area-To-Mass Ratio = 0.008352 (m^2/kg)
    Start Year = 2016.000000 (yr)
    Initial Mass = 9.200000 (kg)
    Final Mass = 9.200000 (kg)
    Duration = 1.000000 (yr)
    Station Kept = False
    Abandoned = True
    PMD Perigee Altitude = 608.878544 (km)
    PMD Apogee Altitude = 608.878544 (km)
    PMD Inclination = 97.815726 (deg)
    PMD RAAN = 357.965049 (deg)
    PMD Argument of Perigee = 8.848460 (deg)
    PMD Mean Anomaly = 0.000000 (deg)
**OUTPUT**
    Suggested Perigee Altitude = 608.878544 (km)
    Suggested Apogee Altitude = 608.878544 (km)
    Returned Error Message = Passes LEO reentry orbit criteria.
    Released Year = 2036 (yr)
    Requirement = 61
    Compliance Status = Pass
==============
=============== End of Requirement 4.6 ===============
```

APPENDIX D - REQUIREMENT 4.7-1 DAS 2.0.1 LOG

```
01 26 2016; 16:50:37PM *********Processing Requirement 4.7-1
Return Status : Passed
***********INPUT****
Item Number = 1
name = CICERO
quantity = 1
parent = 0
materialID = 9
type = Box
Aero Mass = 9.200000
Thermal Mass = 9.200000
Diameter/Width = 0.226000
Length = 0.340000
Height = 0.100000

name = CICERO PX Panel
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 0.362000
Thermal Mass = 0.362000
Diameter/Width = 0.082000
Length = 0.335000
Height = 0.010000

name = CCICERO MX Panel
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 0.700000
Thermal Mass = 0.700000
Diameter/Width = 0.082000
Length = 0.335000
Height = 0.011500
```

```
name = CIERO PZ Panel
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 0.272000
Thermal Mass = 0.272000
Diameter/Width = 0.100000
Length = 0.209000
Height = 0.020000

name = CICERO MZ Panel
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 0.248000
Thermal Mass = 0.248000
Diameter/Width = 0.082000
Length = 0.205000
Height = 0.020000

name = CICERO PY Panel
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 1.800000
Thermal Mass = 1.800000
Diameter/Width = 0.226000
Length = 0.330000
Height = 0.011500

name = CICERO MY Panel
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 2.100000
```

```
Thermal Mass = 2.100000
Diameter/Width = 0.226000
Length = 0.330000
Height = 0.011500

name = IRM
quantity = 1
parent = 1
materialID = 23
type = Box
Aero Mass = 0.345000
Thermal Mass = 0.125000
Diameter/Width = 0.100000
Length = 0.100000
Height = 0.050000

name = IMU
quantity = 1
parent = 8
materialID = 9
type = Box
Aero Mass = 0.018000
Thermal Mass = 0.018000
Diameter/Width = 0.024500
Length = 0.038000
Height = 0.011100

name = Star Camera
quantity = 2
parent = 8
materialID = 9
type = Cylinder
Aero Mass = 0.041000
Thermal Mass = 0.041000
Diameter/Width = 0.030000
Length = 0.047000

name = RWA_Motor_Flywheel
quantity = 3
parent = 8
```

```
materialID = 9
type = Cylinder
Aero Mass = 0.020000
Thermal Mass = 0.020000
Diameter/Width = 0.040000
Length = 0.013000

name = RWA_Motor
quantity = 3
parent = 8
materialID = 9
type = Cylinder
Aero Mass = 0.010000
Thermal Mass = 0.010000
Diameter/Width = 0.040000
Length = 0.013000

name = RWA_Brackets
quantity = 3
parent = 8
materialID = 9
type = Box
Aero Mass = 0.010000
Thermal Mass = 0.010000
Diameter/Width = 0.040000
Length = 0.040000
Height = 0.013000

name = Battery Module_1
quantity = 2
parent = 1
materialID = 9
type = Box
Aero Mass = 0.493000
Thermal Mass = 0.352000
Diameter/Width = 0.042000
Length = 0.086000
Height = 0.042000

name = Batteries
```

quantity = 6

parent = 14

materialID = 54

type = Cylinder

Aero Mass = 0.047000

Thermal Mass = 0.047000

Diameter/Width = 0.018000

Length = 0.065000

name = POD Antenna

quantity = 1

parent = 1

materialID = 9

type = Cylinder

Aero Mass = 0.075000

Thermal Mass = 0.075000

Diameter/Width = 0.060000

Length = 0.030000

name = CION Payload

quantity = 1

parent = 1

materialID = 23

type = Box

Aero Mass = 0.700000

Thermal Mass = 0.100000

Diameter/Width = 0.200000

Length = 0.300000

Height = 0.005000

name = CION Cover

quantity = 1

parent = 17

materialID = 9

type = Box

Aero Mass = 0.600000

Thermal Mass = 0.600000

Diameter/Width = 0.200000

Length = 0.300000

Height = 0.005000

```
name = KL_BPF
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 0.010000
Thermal Mass = 0.010000
Diameter/Width = 0.015000
Length = 0.030000
Height = 0.010000

name = Miteq_LNA
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 0.010000
Thermal Mass = 0.010000
Diameter/Width = 0.015000
Length = 0.030000
Height = 0.010000

name = Backplane
quantity = 1
parent = 1
materialID = 23
type = Box
Aero Mass = 0.200000
Thermal Mass = 0.200000
Diameter/Width = 0.200000
Length = 0.300000
Height = 0.004000

name = UHF_assembly
quantity = 1
parent = 1
materialID = 23
type = Box
Aero Mass = 0.122000
```

```
Thermal Mass = 0.040000
Diameter/Width = 0.050000
Length = 0.100000
Height = 0.050000

name = UHF_radio
quantity = 1
parent = 22
materialID = 9
type = Box
Aero Mass = 0.030000
Thermal Mass = 0.030000
Diameter/Width = 0.036000
Length = 0.083000
Height = 0.007000

name = UHF_antenna
quantity = 1
parent = 22
materialID = 19
type = Box
Aero Mass = 0.052000
Thermal Mass = 0.052000
Diameter/Width = 0.098000
Length = 0.098000
Height = 0.007000

name = OEM628
quantity = 1
parent = 1
materialID = 23
type = Box
Aero Mass = 0.537000
Thermal Mass = 0.037000
Diameter/Width = 0.060000
Length = 0.100000
Height = 0.009000

name = GPS_housing
quantity = 1
```

```
parent = 25
materialID = 9
type = Box
Aero Mass = 0.500000
Thermal Mass = 0.500000
Diameter/Width = 0.060000
Length = 0.150000
Height = 0.030000

*************OUTPUT****
Item Number = 1
name = CICERO
Demise Altitude = 77.999472
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = CICERO PX Panel
Demise Altitude = 75.487902
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = CCICERO MX Panel
Demise Altitude = 73.325293
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = CIERO PZ Panel
Demise Altitude = 75.944988
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = CICERO MZ Panel
Demise Altitude = 75.895691
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = CICERO PY Panel
Demise Altitude = 71.422621
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
```

****

name = CICERO MY Panel

Demise Altitude = 70.371027

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = IRM

Demise Altitude = 77.040246

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = IMU

Demise Altitude = 75.619340

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Star Camera

Demise Altitude = 74.938879

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = RWA_Motor_Flywheel

Demise Altitude = 75.409761

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = RWA_Motor

Demise Altitude = 76.223316

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = RWA_Brackets

Demise Altitude = 76.452472

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Battery Module_1

Demise Altitude = 72.679707

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Batteries

Demise Altitude = 68.542160

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = POD Antenna

Demise Altitude = 75.468566

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = CION Payload

Demise Altitude = 77.719051

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = CION Cover

Demise Altitude = 74.904175

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = KL_BPF

Demise Altitude = 76.736418

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Miteq_LNA

Demise Altitude = 76.736418

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Backplane

Demise Altitude = 77.346519

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = UHF_assembly

Demise Altitude = 77.622363

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = UHF_radio

Demise Altitude = 76.515996

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = UHF_antenna

Demise Altitude = 76.760738

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = OEM628

Demise Altitude = 77.459793

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = GPS_housing

Demise Altitude = 72.251371

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

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