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: CPOD 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: CPOD 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

CPOD 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	NASA / Ames Research Center (ARC) / Center Chief Technologist (CCT)
Program Executive	Andy Petro (NASA HQ)
Program/project Manager	Roger Hunter (NASA ARC) / Marco Villa (Tyvak)
Senior Scientist	Elwood Agasid (NASA ARC)
Senior Management	Roger Hunter (NASA ARC)
Foreign government or space agency participation	N/A
Summary of NASA's responsibility under the governing agreement(s)	N/A
Mission Directorate	NASA / Ames Research Center (ARC)

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.

ATP	01 Oct 2012
Kick-off	15 Nov 2012
IDR#1 (SRR)	19 Feb 2013
IDR#2	09 May 2013
PDR	09 Jul 2013
IDR#3	Dec 2013
CDR	Mar 2014
Launch	March 2017

Table 1-2: Summary of Mission Design and Development Milestones

1.2.2 Mission Overview

The goal of the CubeSat Proximity Operations Demonstration (CPOD) is to validate the technologies needed to support rendezvous, proximity operations, docking (RPOD), servicing, and formation flight by utilizing a pair of identical 3U CubeSats, and leveraging CubeSats inherently lower spacecraft and launch costs. The demonstration concept uses a synthesis of GPS for rendezvous, visible & IR cameras for proximity operations bearing and range determination, image recognition for near-field object determination for relative positioning, and universal docking system. A basic overview of the vehicle and RPOD components is shown below:

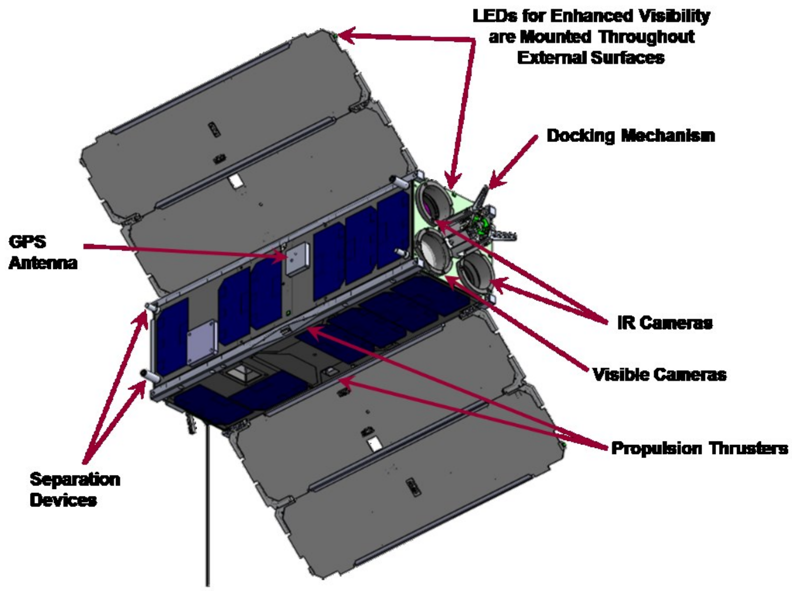


Figure 1-1: CPOD Vehicle with RPOD Components

The CPOD vehicles begin operations attached to one another through Spectra line, which is brought into tension following the release of the vehicles by the deployer. The vehicles are separated from one another following de-tumble to allow for a controlled release via burn circuit and separation springs. Each CPOD vehicle then completes general commissioning before beginning proximity operations. Following successful commissioning, one vehicle is designated the chaser and the other the target vehicle, the chaser then utilizes a systematic and controlled approach to validate the sensor data products for bearing, range, and relative positioning. Following full validation, the chaser makes a final approach to the target for final docking.

Parameter	Value
Launch vehicle and launch site	Minotaur-C, Vandenberg AFB, CA[1]
Proposed launch date	Q4 2016
Mission duration	1+ year
Launch and deployment profile	The Minotaur-C launch vehicle will launch the primary mission satellite. After which, it will deploy the CPOD satellites into their final mission orbit (~500km[2], circular, sun-synchronous orbit ~ 97.8° inclination). There is no parking or transfer orbit. The CPOD satellite will decay naturally for debris mitigation and will re-enter within 25 years after completion of mission.

[1] Minotaur-C launch is the current scheduled launch, but final launch is still being finalized with NASA HQ

[2] 500 km is associated with the scheduled Minotaur-C launch, for the purposes of a worst case analysis, 620 km is used for de-orbit analysis completed as part of ODAR, all other analysis will use 500 km

Table 1-3: Summary of Mission Parameters

1.2.3 Identification of any interaction or potential physical interference with other operational spacecraft

The CPOD vehicles will only interact with one another during primary operations. There are no plans to interact or interfere with any other spacecraft on orbit. The CPOD vehicles are separated from one another in a controlled manner following de-tumble. The known spring force and systematic and controlled method in which they approach one another prevents any uncontrolled interaction, and is explained further in Section 5.0 of this ODAR.

2.0 SPACECRAFT DESCRIPTION

2.1 Physical Description of Spacecraft

The CPOD 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 CPOD vehicle is a 3U CubeSat with the core of the vehicle being 30cm x 10cm x 10cm with a mass of roughly 5.2 kg. The vehicle uses a total of four deployable solar panels and each is roughly 30cm x 10cm in size.

The CPOD 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 CPOD payload utilizes multiple sensors to support range determination at different operating distances. The CPOD payload houses two IR and two visible cameras, as well as the docking mechanism, and electronics for image processing.

Parameter	Value
Total satellite mass at launch, including all propellants and fluids	~5.2 kg
Dry Mass of satellite at launch, excluding solid rocket motor propellants	~4.7 kg
Identification, including mass and pressure, of all fluids	R236fa (common refrigerant), 420 grams, 67 psig at room temp.
Fluids in Pressurized batteries	NONE. CPOD uses unpressurized standard COTS Li-ion battery cells

Table 2-1: Summary of Spacecraft Parameters

2.1.1 Description of Propulsion Systems

The CPOD cold gas propulsion system utilizes a mature design that was developed by VACCO Industries and tested extensively (70,000+ firings) in a vacuum by the US Air Force Research Lab and traces heritage to DARPA and Aerospace Corp programs. The highly integrated unit utilizes R236fa as a propellant that is stored as liquid for volume efficiency. All sensor and control electronics are contained inside the unit and only requires power and serial data connections. Extensive materials compatibility testing and analyses have demonstrated that the propellant is compatible being immersed around the electronics. The CPOD propulsion module is made out of aluminum and has eight thrusters located at the corners of the unit. The unit can hold roughly 420 grams of propellant.

2.1.2 Description of attitude control system

The CPOD 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 CPOD vehicles has the long axis (z-axis) of the vehicle pointed along the velocity vector. The vehicle is rotated about the long axis to point the deployable panels in a zenith direction for energy collection. The CPOD vehicle will spend a majority of their time in this attitude.

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 four deployable solar panels and additional solar cells that are mounted on the core of the vehicle. Energy storage is accomplished using standard COTS Li-ion battery cells in a 3S2P (3 in series, 2 parallel) configuration. 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.

2.1.6 Identification of any other sources of stored energy

None.

2.1.7 Identification of any radioactive materials on board

None.

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.

The propulsion tank will be depressurized by opening the valves at EOM.

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).

<u>Compliance statement:</u>

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).

<u>Compliance statement:</u>

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.

The propulsion tank will be depressurized by opening the valves at EOM.

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: 0.00134 COMPLIANT

5.2 Proximity Operations

The CPOD primary mission is to validate the technologies needed to support rendezvous, proximity operations, docking (RPOD), servicing, and formation flight by utilizing a pair of identical 3U CubeSats. NASA has sponsored the project to perform this in a systematic and controlled manner with vehicle safety as the focus of all Mission Assurance. The approach method and safety gates integrated into the RPOD CONOPS has taken all precautions into account in order to mitigate any potential risk to uncontrolled collisions. A description of the planned RPOD CONOPS is attached to this ODAR as an appendix and describes the process for data product validation and method for conducting controlled approach.

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: ~4.7kg (dry mass)

Cross-sectional Area: 0.08 m^2 (average of min and max areas).

Area to mass ratio: (0.08 m^2)/(4.7 kg) = 0.017 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.*

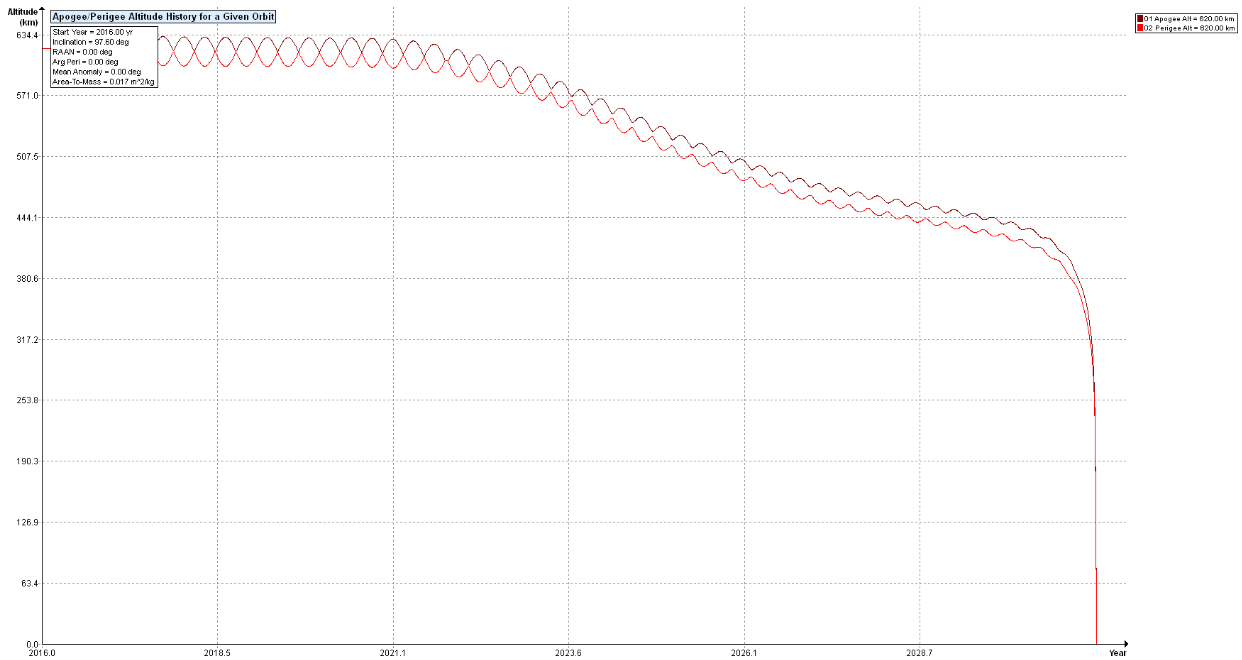
c. Direct retrieval:

- *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 620 km. All orbits below this will have a lower orbital lifetime. The orbital lifetime is predicted to be 15.184 years; COMPLIANT

Figure 6-1: CPOD Deorbit Lifetime



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

<u>Compliance statement:</u>

Not applicable. CPOD mission orbit is a LEO.

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

<u>Compliance statement:</u>

Not applicable. CPOD mission orbit is a LEO.

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

<u>Compliance statement:</u>

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 propulsion tank will be depressurized by opening the plenum and thruster valve(s) at EOM.

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 CPOD 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 CPOD 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

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01 21 2016; 13:01:42PM Processing Requirement 4.5-1: Return Status : Passed
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Run Data
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**INPUT**
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    Space Structure Type = Payload
    Perigee Altitude = 500.000000 (km)
    Apogee Altitude = 500.000000 (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 = 0.017000 (m^2/kg)
    Start Year = 2016.000000 (yr)
    Initial Mass = 5.200000 (kg)
    Final Mass = 4.700000 (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 ===============
```

APPENDIX C - REQUIREMENT 4.6 DAS 2.0.1 LOG

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01 21 2016; 13:49:29PM Processing Requirement 4.6 Return Status : Passed
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Project Data
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**INPUT**
    Space Structure Name = CPOD_Satellite
    Space Structure Type = Payload
    Perigee Altitude = 620.000000 (km)
    Apogee Altitude = 620.000000 (km)
    Inclination = 97.800000 (deg)
    RAAN = 0.000000 (deg)
    Argument of Perigee = 0.000000 (deg)
    Mean Anomaly = 0.000000 (deg)
    Area-To-Mass Ratio = 0.017000 (m^2/kg)
    Start Year = 2016.000000 (yr)
    Initial Mass = 5.200000 (kg)
    Final Mass = 4.700000 (kg)
    Duration = 1.000000 (yr)
    Station Kept = False
    Abandoned = True
    PMD Perigee Altitude = 618.186912 (km)
    PMD Apogee Altitude = 618.186912 (km)
    PMD Inclination = 97.816768 (deg)
    PMD RAAN = 356.261802 (deg)
    PMD Argument of Perigee = 17.746233 (deg)
    PMD Mean Anomaly = 0.000000 (deg)
**OUTPUT**
    Suggested Perigee Altitude = 618.186912 (km)
    Suggested Apogee Altitude = 618.186912 (km)
    Returned Error Message = Passes LEO reentry orbit criteria.
    Released Year = 2031 (yr)
    Requirement = 61
    Compliance Status = Pass
==============
=============== End of Requirement 4.6 ===============
```

APPENDIX D - REQUIREMENT 4.7-1 DAS 2.0.1 LOG

```
01 21 2016; 13:04:09PM *********Processing Requirement 4.7-1
Return Status : Passed
***********INPUT****
Item Number = 1
name = CPOD_Satellite
quantity = 1
parent = 0
materialID = 3
type = Box
Aero Mass = 4.700000
Thermal Mass = 4.700000
Diameter/Width = 0.100000
Length = 0.300000
Height = 0.100000

name = RPO_camera_nfow
quantity = 1
parent = 1
materialID = 9
type = Cylinder
Aero Mass = 0.133000
Thermal Mass = 0.133000
Diameter/Width = 0.042000
Length = 0.093000

name = RPO_camera_wfov
quantity = 1
parent = 1
materialID = 9
type = Cylinder
Aero Mass = 0.103000
Thermal Mass = 0.103000
Diameter/Width = 0.031000
Length = 0.052000

name = RPO_camera_ir
quantity = 1
parent = 1
```

```
materialID = 9
type = Cylinder
Aero Mass = 0.336000
Thermal Mass = 0.336000
Diameter/Width = 0.045000
Length = 0.093000

name = RPO_subframe
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 0.085000
Thermal Mass = 0.085000
Diameter/Width = 0.100000
Length = 0.100000
Height = 0.020000

name = RPO_lens_baffle
quantity = 3
parent = 1
materialID = 9
type = Cylinder
Aero Mass = 0.033000
Thermal Mass = 0.033000
Diameter/Width = 0.045000
Length = 0.050000

name = RPO_docking_mech
quantity = 1
parent = 1
materialID = 9
type = Cylinder
Aero Mass = 0.238000
Thermal Mass = 0.238000
Diameter/Width = 0.050000
Length = 0.050000

name = RPO_pcb
quantity = 1
```

```
parent = 1
materialID = 23
type = Flat Plate
Aero Mass = 0.069000
Thermal Mass = 0.069000
Diameter/Width = 0.100000
Length = 0.100000

name = RPO_end_panel
quantity = 1
parent = 1
materialID = 23
type = Flat Plate
Aero Mass = 0.187000
Thermal Mass = 0.025000
Diameter/Width = 0.100000
Length = 0.100000

name = battery
quantity = 6
parent = 9
materialID = 54
type = Cylinder
Aero Mass = 0.027000
Thermal Mass = 0.027000
Diameter/Width = 0.018000
Length = 0.065000

name = battery_assembly
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 0.218000
Thermal Mass = 0.218000
Diameter/Width = 0.100000
Length = 0.100000
Height = 0.050000

name = GPS_receiver
```

```
quantity = 1
parent = 1
materialID = 23
type = Box
Aero Mass = 0.024000
Thermal Mass = 0.024000
Diameter/Width = 0.100000
Length = 0.100000
Height = 0.016000

name = GPS_antenna
quantity = 2
parent = 1
materialID = 9
type = Flat Plate
Aero Mass = 0.055000
Thermal Mass = 0.055000
Diameter/Width = 0.080000
Length = 0.080000

name = sysproc_assembly
quantity = 1
parent = 1
materialID = 23
type = Flat Plate
Aero Mass = 0.066000
Thermal Mass = 0.066000
Diameter/Width = 0.100000
Length = 0.100000

name = rwa_mount_assembly
quantity = 1
parent = 1
materialID = 23
type = Flat Plate
Aero Mass = 0.170000
Thermal Mass = 0.050000
Diameter/Width = 0.100000
Length = 0.100000
```

```
name = rwa_motor_flywheel
quantity = 3
parent = 15
materialID = 16
type = Cylinder
Aero Mass = 0.020000
Thermal Mass = 0.020000
Diameter/Width = 0.040000
Length = 0.013000

name = rwa_motor
quantity = 3
parent = 15
materialID = 16
type = Cylinder
Aero Mass = 0.010000
Thermal Mass = 0.010000
Diameter/Width = 0.040000
Length = 0.013000

name = rwa_brackets
quantity = 3
parent = 15
materialID = 9
type = Box
Aero Mass = 0.010000
Thermal Mass = 0.010000
Diameter/Width = 0.040000
Length = 0.040000
Height = 0.013000

name = irm_assembly
Page 25
DAS Activity Log
quantity = 1
parent = 1
materialID = 23
type = Box
Aero Mass = 0.125000
Thermal Mass = 0.025000
```

```
Diameter/Width = 0.100000
Length = 0.100000
Height = 0.050000

name = imu
quantity = 1
parent = 19
materialID = 9
type = Box
Aero Mass = 0.018000
Thermal Mass = 0.018000
Diameter/Width = 0.024530
Length = 0.038000
Height = 0.011100

name = star_camera
quantity = 2
parent = 19
materialID = 9
type = Cylinder
Aero Mass = 0.041000
Thermal Mass = 0.041000
Diameter/Width = 0.030000
Length = 0.047000

name = sband_assembly
quantity = 1
parent = 1
materialID = 23
type = Box
Aero Mass = 0.040000
Thermal Mass = 0.040000
Diameter/Width = 0.100000
Length = 0.500000
Height = 0.100000

name = sband_transmitter
quantity = 1
parent = 1
materialID = 9
```

```
type = Box
Aero Mass = 0.044000
Thermal Mass = 0.044000
Diameter/Width = 0.032000
Length = 0.086000
Height = 0.010000

name = sband_antenna
quantity = 2
parent = 1
materialID = 9
type = Flat Plate
Aero Mass = 0.104000
Thermal Mass = 0.104000
Diameter/Width = 0.100000
Length = 0.100000

name = uhf_assembly
quantity = 1
parent = 1
materialID = 23
type = Box
Aero Mass = 0.040000
Thermal Mass = 0.040000
Diameter/Width = 0.100000
Length = 0.100000
Height = 0.050000

name = uhf_radio
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 0.030000
Thermal Mass = 0.030000
Diameter/Width = 0.036000
Length = 0.083000
Height = 0.004000

name = uhf_antenna
```

```
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 0.052000
Thermal Mass = 0.052000
Diameter/Width = 0.098000
Length = 0.098000
Height = 0.007000

name = isl_assembly
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 0.026000
Thermal Mass = 0.026000
Diameter/Width = 0.020000
Length = 0.029000
Height = 0.020000

name = deploy_panel_assembly
quantity = 4
parent = 1
materialID = 23
type = Flat Plate
Aero Mass = 0.119000
Thermal Mass = 0.119000
Diameter/Width = 0.100000
Length = 0.300000
Page 27
DAS Activity Log

name = deploy_panel_edge_stiffener
quantity = 8
parent = 1
materialID = 9
type = Box
Aero Mass = 0.008000
Thermal Mass = 0.008000
```

```
Diameter/Width = 0.005000
Length = 0.230000
Height = 0.003000

name = solar_cells
quantity = 48
parent = 1
materialID = 23
type = Flat Plate
Aero Mass = 0.010000
Thermal Mass = 0.010000
Diameter/Width = 0.050000
Length = 0.100000

name = body_panel_assembly
quantity = 4
parent = 1
materialID = 23
type = Flat Plate
Aero Mass = 0.119000
Thermal Mass = 0.119000
Diameter/Width = 0.100000
Length = 0.300000

name = propulsion_module
quantity = 1
parent = 1
materialID = 9
type = Box
Aero Mass = 0.282000
Thermal Mass = 0.282000
Diameter/Width = 0.100000
Length = 0.100000
Height = 0.100000

name = structure_assembly
quantity = 1
parent = 1
materialID = 9
type = Box
```

```
Aero Mass = 0.102000
Thermal Mass = 0.102000
Diameter/Width = 0.010000
Length = 1.200000
Height = 0.010000

*************OUTPUT****
Item Number = 1
name = CPOD_Satellite
Demise Altitude = 77.996418
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = RPO_camera_nfow
Demise Altitude = 75.086316
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = RPO_camera_wfov
Demise Altitude = 73.625449
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = RPO_camera_ir
Demise Altitude = 71.583972
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = RPO_subframe
Demise Altitude = 76.665894
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = RPO_lens_baffle
Demise Altitude = 76.881121
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = RPO_docking_mech
Demise Altitude = 71.464347
```

```
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
************************************
name = RPO_pcb
Demise Altitude = 77.072980
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
************************************
name = RPO_end_panel
Demise Altitude = 77.686871
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
************************************
name = battery
Demise Altitude = 73.215582
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
************************************
name = battery_assembly
Demise Altitude = 75.453707
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
************************************
name = GPS_receiver
Demise Altitude = 77.729472
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
************************************
name = GPS_antenna
Demise Altitude = 76.471761
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
************************************
name = sysproc_assembly
Demise Altitude = 77.115230
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
************************************
name = rwa_mount_assembly
Demise Altitude = 77.367871
```

```
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = rwa_motor_flywheel
Demise Altitude = 69.454660
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = rwa_motor
Demise Altitude = 75.220715
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = rwa_brackets
Demise Altitude = 76.749676
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = irm_assembly
Demise Altitude = 77.811644
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = imu
Demise Altitude = 76.312066
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = star_camera
Demise Altitude = 75.728418
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = sband_assembly
Demise Altitude = 77.929652
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = sband_transmitter
Demise Altitude = 76.404191
```

```
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = sband_antenna
Demise Altitude = 75.961488
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = uhf_assembly
Demise Altitude = 77.674762
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = uhf_radio
Demise Altitude = 76.764793
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = uhf_antenna
Demise Altitude = 77.023097
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = isl_assembly
Demise Altitude = 75.798582
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = deploy_panel_assembly
Demise Altitude = 77.389840
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = deploy_panel_edge_stiffener
Page 31
DAS Activity Log
Demise Altitude = 77.590316
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
```

```
name = solar_cells
Demise Altitude = 77.788730
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = body_panel_assembly
Demise Altitude = 77.389840
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = propulsion_module
Demise Altitude = 75.668199
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
name = structure_assembly
Demise Altitude = 77.540707
Debris Casualty Area = 0.000000
Impact Kinetic Energy = 0.000000
*************************************
=============== End of Requirement 4.7-1 ===============
```

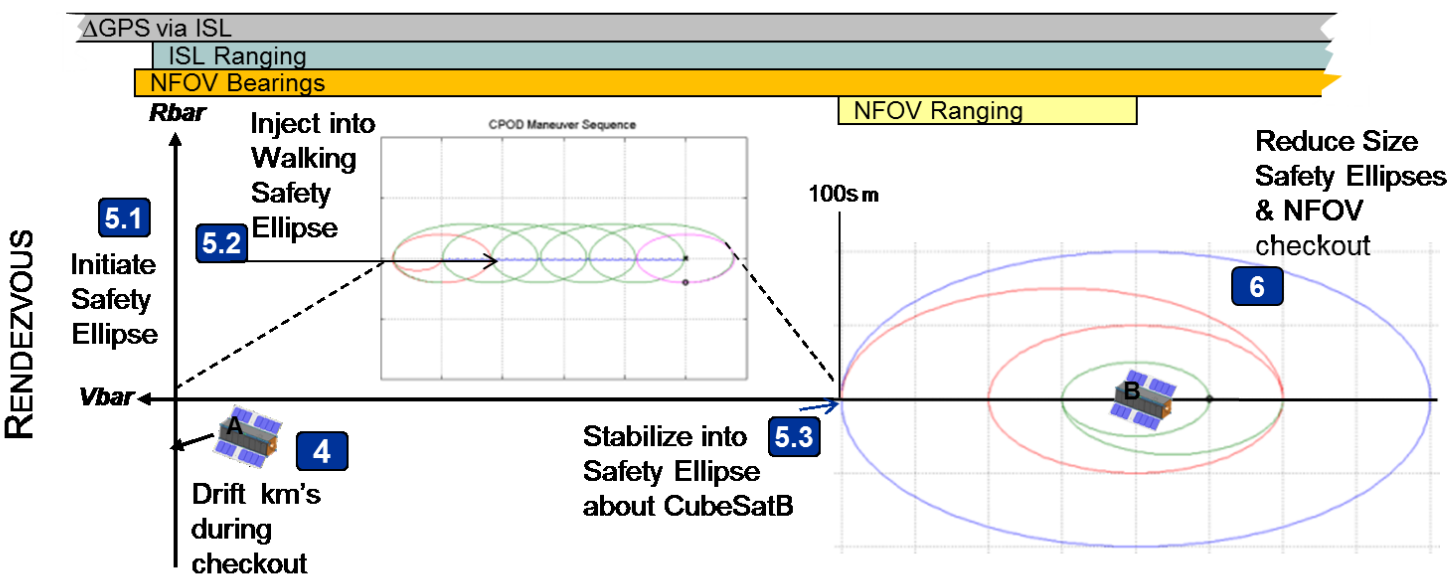
APPENDIX E – CPOD VEHICLE RPOD CONOPS

The CPOD primary mission is to validate the technologies needed to support rendezvous, proximity operations, docking (RPOD), servicing, and formation flight by utilizing a pair of identical 3U CubeSats. NASA has sponsored the project to perform this in a systematic and controlled manner with vehicle safety as the focus of all Mission Assurance.

A key aspect of the risk reduction is the automated Fault Detection, Isolation, and Recovery (FDIR) logic that is integrated into the Flight Software (FSW) which is used to detect anomalies. The trajectories used for all aspects of the approach have associated fault flags with each step, if these are violated the vehicles halt the approach and return to a safe range and hold until commanded out of the mode by ground command, allowing time to understand the faults and make informed decisions before proceeding.

The overall CONOPS can be broken up into two primary phases: Rendezvous and Docking.

The Rendezvous Phase is initiated following individual vehicle commissioning, which consists of component and subsystem telemetry and command verification. The ground and CPOD vehicles will have positive confirmation of both vehicle locations prior to entering Rendezvous from GPS ephemeris data and ISL range data. The figure below shows the general approach for the Rendezvous phase. The top portion of the figure shows the data products available during each phase and the bottom portion shows the general approach, with Vbar referring to the velocity vector of the "target" vehicles and Rbar referring to the "target" vehicles radial vector.

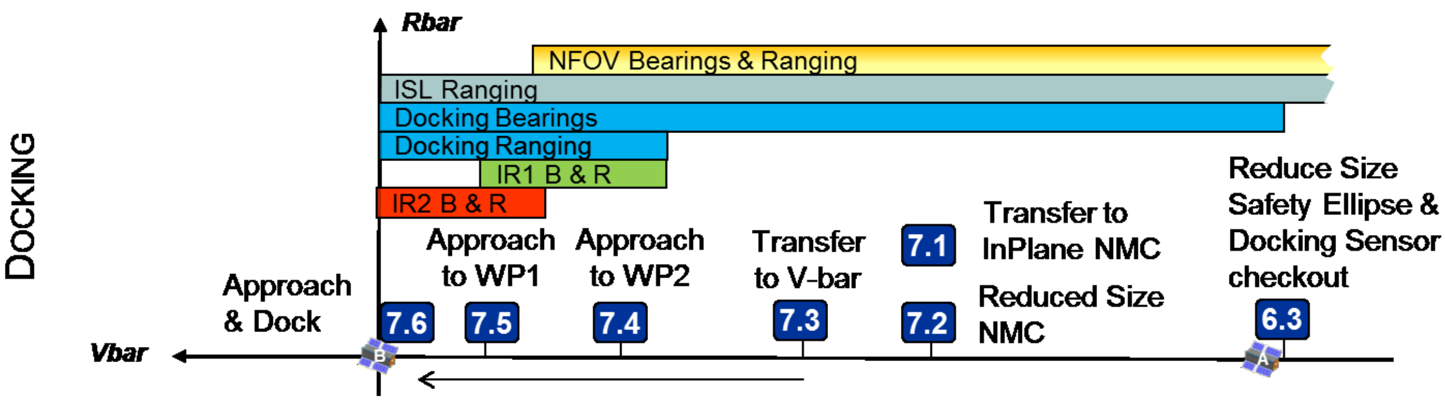


CPOD Vehicle Rendezvous Approach

- **Phase 5.1** initiates the Safety Ellipse for the "Chaser" (Vehicle A) CPOD vehicle. This is referred to as a safety ellipse because it is a passively safe orbit. The safety ellipse is passively safe due to introducing radial and cross track offsets which provide natural motion for the chaser that never crosses the velocity vector of the target. Thus, if either vehicle were to experience a fault, and a relative drift was introduced, the vehicles could safely pass each other without issue.

- **Phase 5.2** injects Vehicle A into a walking safety ellipse. As shown, the vehicle gradually approaches the “target” (Vehicle B) vehicle until the relative distance between the two is approximately 100 m. Similar to the stationary safety ellipse, this trajectory is passively safe due to cross track and radial offsets. Therefore, if a vehicle were to experience a fault, the vehicles would continue past each other without issue.
- **Phase 5.3** shows a safety hold point where Vehicle A stabilizes into a safety ellipse at 100 m around the Vehicle B. At this phase NFOV ranging data is generated.
- **Phase 6** shows the incremental method in which the size of the safety ellipse is reduced as the NFOV ranging continues to be checked out. There are currently 5 different reduced size ellipses with the closest approach in this phase having a relative distance of approximately 25 m.

The Docking Phase is initiated following ranging checkout. At this point the CPOD “Chaser” vehicle has confirmed the required data products docking approach. The figure below shows the general approach for the docking phase.



CPOD Vehicle Docking Approach

- **Phase 6.3** is the final reduction in the Safety Ellipse size that was initiated at the start of Phase 6. Docking sensor checkout is initiated at this phase.
- **Phase 7.1** begins the transfer to In-Plane Natural Motion Circumnavigation (NMC). The vehicles are now in the same orbital plane, but have different orbital characteristics in the orbital plane which induce in-track and radial separation, allowing the natural orbital motion to produce circumnavigation.
- **Phase 7.2** starts the reduction in size for the NMC
- **Phase 7.3** Transitions Vehicle A to the Vehicle B V-Bar
- **Phase 7.4** initiates the approach of Vehicle A to the WP2 (7m) hold point. At this phase the IR1 bearing and range data product is created and validated prior to proceeding to the next phase.
- **Phase 7.5** initiates the approach of Vehicle A to the WP1 (2m) hold point. At this phase the IR2 bearing and range data product is created and validated prior to proceeding to the final phase.

- **Phase 7.6** is the final docking of the two vehicles which is completed with an electro-magnet in the docking module as well as actuated grippers